

TURBINE AEROPLANE AERODYNAMICS, STRUCTURES AND SYSTEMS

Aviation Maintenance Technician Certification Series



- Theory of Flight
- Airframe Structures – General Concepts
- Airframe Structures – Aeroplanes
- Air Conditioning and Cabin Pressurization
- Instruments/Avionic Systems
- Electrical Power
- Equipment and Furnishings
- Fire Protection
- Flight Controls
- Fuel Systems

- Hydraulic Power
- Ice and Rain Protection
- Landing Gear
- Lights
- Oxygen
- Pneumatic/Vacuum
- Water/Waste
- On Board Maintenance Systems
- Integrated Modular Avionics
- Cabin Systems
- Information Systems



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MODULE 11A

FOR B1 CERTIFICATION

TURBINE AEROPLANE AERODYNAMICS, STRUCTURES AND SYSTEMS

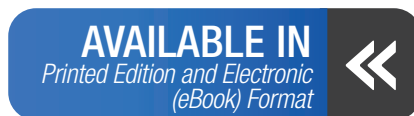
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AVIATION MAINTENANCE TECHNICIAN CERTIFICATION SERIES

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WELCOME

The publishers of this Aviation Maintenance Technician Certification Series welcome you to the world of aviation maintenance. As you move towards EASA certification, you are required to gain suitable knowledge and experience in your chosen area. Qualification on basic subjects for each aircraft maintenance license category or subcategory is accomplished in accordance with the following matrix. Where applicable, subjects are indicated by an "X" in the column below the license heading.

For other educational tools created to prepare candidates for licensure, contact Aircraft Technical Book Company.

We wish you good luck and success in your studies and in your aviation career!

REVISION LOG

VERSION	EFFECTIVE DATE	DESCRIPTION OF CHANGE
001	2015 10	Module Creation and Release
002	2016 01	Minor Revisions
003	2017 09	Format Update
003.1	2019 02	Added section on Pneumatic and Pressure Pumps in Sub-Module 16.
003.2	2019 05	Corrected incorrect answers in Sub-Module 20.
004	2019 12	Typographic format updated; Sequencing of content to Appendix 1 refined.

Version 4 - Enhanced or modified content within the following Sub-Modules:

- Sub-Module 02 Bonding Procedures, NACA Scoops
- Sub-Module 03 Flight Control Surfaces, Image of A330 emergency slide image of turbine nacelle
- Sub-Module 07 Images of interior paneling and cargo container added for clarity
- Sub-Module 19 Common Core Systems
- Sub-Module 20 IFE Systems and Servers
- Sub-Module 21 Airbus Information System Architecture

FORWARD

PART-66 and the Acceptable Means of Compliance (AMC) and Guidance Material (GM) of the European Aviation Safety Agency (EASA) Regulation (EC) No. 1321/2014, Appendix 1 to the Implementing Rules establishes the Basic Knowledge Requirements for those seeking an aircraft maintenance license. The information in this Module of the Aviation Maintenance Technical Certification Series published by the Aircraft Technical Book Company meets or exceeds the breadth and depth of knowledge subject matter referenced in Appendix 1 of the Implementing Rules. However, the order of the material presented is at the discretion of the editor in an effort to convey the required knowledge in the most sequential and comprehensible manner. Knowledge levels required for Category A1, B1, B2, and B3 aircraft maintenance licenses remain unchanged from those listed in Appendix 1 Basic Knowledge Requirements. Tables from Appendix 1 Basic Knowledge Requirements are reproduced at the beginning of each module in the series and again at the beginning of each Sub-Module.

How numbers are written in this book:

This book uses the International Civil Aviation Organization (ICAO) standard of writing numbers. This method displays large numbers by adding a space between each group of 3 digits. This is opposed to the American method which uses commas and the European method which uses periods. For example, the number one million is expressed as so:

ICAO Standard	1 000 000
European Standard	1.000.000
American Standard	1,000,000

SI Units:

The International System of Units (SI) developed and maintained by the General Conference of Weights and Measures (CGPM) shall be used as the standard system of units of measurement for all aspects of international civil aviation air and ground operations.

Prefixes:

The prefixes and symbols listed in the table below shall be used to form names and symbols of the decimal multiples and submultiples of International System of Units (SI) units.

MULTIPLICATION FACTOR	PREFIX	SYMBOL
1 000 000 000 000 000 000 = 10^{18}	exa	E
1 000 000 000 000 000 = 10^{15}	peta	P
1 000 000 000 000 = 10^{12}	tera	T
1 000 000 000 = 10^9	giga	G
1 000 000 = 10^6	mega	M
1 000 = 10^3	kilo	k
100 = 10^2	hecto	h
10 = 10^1	deca	da
0.1 = 10^{-1}	deci	d
0.01 = 10^{-2}	centi	c
0.001 = 10^{-3}	milli	m
0.000 001 = 10^{-6}	micro	μ
0.000 000 001 = 10^{-9}	nano	n
0.000 000 000 001 = 10^{-12}	pico	p
0.000 000 000 000 001 = 10^{-15}	femto	f
0.000 000 000 000 000 001 = 10^{-18}	atto	a

International System of Units (SI) Prefixes

EASA LICENSE CATEGORY CHART

Module Number and Title		A1 Airplane Turbine	B1.1 Airplane Turbine	B1.2 Airplane Piston	B1.3 Helicopter Turbine	B1.4 Helicopter Piston	B2 Avionics
1	Mathematics	X	X	X	X	X	X
2	Physics	X	X	X	X	X	X
3	Electrical Fundamentals	X	X	X	X	X	X
4	Electronic Fundamentals		X	X	X	X	X
5	Digital Techniques / Electronic Instrument Systems	X	X	X	X	X	X
6	Materials and Hardware	X	X	X	X	X	X
7A	Maintenance Practices	X	X	X	X	X	X
8	Basic Aerodynamics	X	X	X	X	X	X
9A	Human Factors	X	X	X	X	X	X
10	Aviation Legislation	X	X	X	X	X	X
11A	Turbine Aeroplane Aerodynamics, Structures and Systems	X	X				
11B	Piston Aeroplane Aerodynamics, Structures and Systems			X			
12	Helicopter Aerodynamics, Structures and Systems				X	X	
13	Aircraft Aerodynamics, Structures and Systems						X
14	Propulsion						X
15	Gas Turbine Engine	X	X		X		
16	Piston Engine			X		X	
17A	Propeller	X	X	X			

GENERAL KNOWLEDGE REQUIREMENTS

MODULE 11A SYLLABUS AS OUTLINED IN PART-66, APPENDIX 1

Level 1

A familiarization with the principal elements of the subject.

Objectives:

- The applicant should be familiar with the basic elements of the subject.
- The applicant should be able to give a simple description of the whole subject, using common words and examples.
- The applicant should be able to use typical terms.

Level 2

A general knowledge of the theoretical and practical aspects of the subject and an ability to apply that knowledge.

Objectives:

- The applicant should be able to understand the theoretical fundamentals of the subject.
- The applicant should be able to give a general description of the subject using, as appropriate, typical examples.
- The applicant should be able to use mathematical formula in conjunction with physical laws describing the subject.
- The applicant should be able to read and understand sketches, drawings and schematics describing the subject.
- The applicant should be able to apply his knowledge in a practical manner using detailed procedures.

Level 3

A detailed knowledge of the theoretical and practical aspects of the subject and a capacity to combine and apply the separate elements of knowledge in a logical and comprehensive manner.

Objectives:

- The applicant should know the theory of the subject and interrelationships with other subjects.
- The applicant should be able to give a detailed description of the subject using theoretical fundamentals and specific examples.
- The applicant should understand and be able to use mathematical formula related to the subject.
- The applicant should be able to read, understand and prepare sketches, simple drawings and schematics describing the subject.
- The applicant should be able to apply his knowledge in a practical manner using manufacturer's instructions.
- The applicant should be able to interpret results from various sources and measurements and apply corrective action where appropriate.

PART-66 - APPENDIX I

BASIC KNOWLEDGE REQUIREMENTS

B1

Sub-Module 01 - Theory of Flight

Sub-Module 01.1 - Aeroplane Aerodynamics and Flight Controls

Operation and effect of:

- roll control: ailerons and spoilers;
- pitch control: elevators, stabilators, variable incidence stabilizers and canards;
- yaw control, rudder limiters;

Control using elevons, ruddervators;

High lift devices, slots, slats, flaps, flaperons;

Drag inducing devices, spoilers, lift dumpers, speed brakes;

Effects of wing fences, saw tooth leading edges;

Boundary layer control using, vortex generators, stall wedges or leading edge devices;

Operation and effect of trim tabs, balance and antibalance (leading) tabs, servo tabs, spring tabs, mass balance, control surface bias, aerodynamic balance panels.

2

Sub-Module 01.2 - High Speed Flight

Speed of sound, subsonic flight, transonic flight, supersonic flight;

Mach number, critical Mach number, compressibility buffet, shock wave, aerodynamic heating, area rule;

Factors affecting airflow in engine intakes of high speed aircraft;

Effects of sweepback on critical Mach number.

2

Sub-Module 02 - Airframe Structures — General Concepts

- (a) Airworthiness requirements for structural strength;

Structural classification, primary, secondary and tertiary;

Fail safe, safe life, damage tolerance concepts;

Zonal and station identification systems;

Stress, strain, bending, compression, shear, torsion, tension, hoop stress, fatigue;

Drains and ventilation provisions;

System installation provisions;

Lightning strike protection provision;

Aircraft bonding.

2

- (b) Construction methods of: stressed skin fuselage, formers, stringers, longerons, bulkheads, frames, doublers, struts, ties, beams, floor structures, reinforcement, methods of skinning, anticorrosive protection, wing, empennage and engine attachments;

Structure assembly techniques: riveting, bolting, bonding;

Methods of surface protection, such as chromating, anodizing, painting;

Surface cleaning;

Airframe symmetry: methods of alignment and symmetry checks.

2

PART-66 - APPENDIX I

BASIC KNOWLEDGE REQUIREMENTS

B1

Sub-Module 03 - Airframe Structures — Aeroplanes

Sub-Module 03.1 - Fuselage (ATA 52/53/56)

Construction and pressurization sealing;
 Wing, stabilizer, pylon and undercarriage attachments;
 Seat installation and cargo loading system;
 Doors and emergency exits: construction, mechanisms, operation and safety devices;
 Windows and windscreen construction and mechanisms.

2

Sub-Module 03.2 - Wings (ATA 57)

Construction;
 Fuel storage;
 Landing gear, pylon, control surface and high lift/drag attachments.

2

Sub-Module 03.3 - Stabilizers (ATA 55)

Construction;
 Control surface attachment.

2

Sub-Module 03.4 - Flight Control Surfaces (ATA 55/57)

Construction and attachment;
 Balancing — mass and aerodynamic.

2

Sub-Module 03.5 - Nacelles/Pylons (ATA 54)

Nacelles/Pylons:
 — Construction,
 — Firewalls,
 — Engine mounts.

2

Sub-Module 04 - Air Conditioning and Cabin Pressurization (ATA 21)

Sub-Module 04.1 - Air Supply

Sources of air supply including engine bleed, APU and ground cart.

2

Sub-Module 04.2 - Air Conditioning

Air conditioning systems;
 Air cycle and vapor cycle machines;
 Distribution systems;
 Flow, temperature and humidity control system.

3

Sub-Module 04.3 - Pressurization

Pressurization systems;
 Control and indication including control and safety valves;
 Cabin pressure controllers.

3

PART-66 - APPENDIX I

BASIC KNOWLEDGE REQUIREMENTS

B1

Sub-Module 04.4 - Safety and warning devices

Protection and warning devices.

3

Sub-Module 05 - Instruments/Avionic Systems

Sub-Module 05.1 - Instrument Systems (ATA 31)

Pitot static: altimeter, air speed indicator, vertical speed indicator;
 Gyroscopic: artificial horizon, attitude director, direction indicator, horizontal situation indicator, turn and slip indicator, turn coordinator;
 Compasses: direct reading, remote reading;
 Angle of attack indication, stall warning systems;
 Glass cockpit;
 Other aircraft system indication.

2

Sub-Module 05.2 - Avionic Systems

Fundamentals of system layouts and operation of:
 — Auto Flight (ATA 22),
 — Communications (ATA 23),
 — Navigation Systems (ATA 34).

1

Sub-Module 06 - Electrical Power (ATA 24)

Batteries Installation and Operation;
 DC power generation;
 AC power generation;
 Emergency power generation;
 Voltage regulation;
 Power distribution;
 Inverters, transformers, rectifiers;
 Circuit protection;
 External/Ground power.

3

Sub-Module 07 - Equipment and Furnishings (ATA 25)

- (a) Emergency equipment requirements;
 Seats, harnesses and belts;
- (b) Cabin layout;
 Equipment layout;
 Cabin Furnishing installation;
 Cabin entertainment equipment;
 Galley installation;
 Cargo handling and retention equipment;
 Airstairs.

2

1

PART-66 - APPENDIX I

BASIC KNOWLEDGE REQUIREMENTS

B1

Sub-Module 08 - Fire Protection (ATA 26)

- (a) Fire and smoke detection and warning systems;
Fire extinguishing systems;
System tests;
- (b) Portable fire extinguisher.

3

1

Sub-Module 09 - Flight Controls (ATA 27)

Primary controls: aileron, elevator, rudder, spoiler;
Trim control;
Active load control;
High lift devices;
Lift dump, speed brakes;
System operation: manual, hydraulic, pneumatic, electrical, fly by wire;
Artificial feel, Yaw damper, Mach trim, rudder limiter, gust lock systems;
Balancing and rigging;
Stall protection/warning system.

3

Sub-Module 10 - Fuel Systems (ATA 28)

System layout;
Fuel tanks;
Supply systems;
Dumping, venting and draining;
Crossfeed and transfer;
Indications and warnings;
Refuelling and defueling;
Longitudinal balance fuel systems.

3

Sub-Module 11 - Hydraulic Power (ATA 29)

System layout;
Hydraulic fluids;
Hydraulic reservoirs and accumulators;
Pressure generation: electric, mechanical, pneumatic;
Emergency pressure generation; Filters;
Pressure Control;
Power distribution;
Indication and warning systems;
Interface with other systems.

3

Sub-Module 12 - Ice and Rain Protection (ATA 30)

Ice formation, classification and detection;
Anti-icing systems: electrical, hot air and chemical;
De-Icing systems: electrical, hot air, pneumatic and chemical;
Rain repellent; Probe and drain heating;
Wiper systems.

3

PART-66 - APPENDIX I

BASIC KNOWLEDGE REQUIREMENTS

B1

Sub-Module 13 - Landing Gear (ATA 32)

Construction, shock absorbing;
 Extension and retraction systems: normal and emergency;
 Indications and warning;
 Wheels, brakes, antiskid and auto braking;
 Tires; Steering; Air ground sensing.

3

Sub-Module 14 - Lights (ATA 33)

External: navigation, anti collision, landing, taxiing, ice;
 Internal: cabin, cockpit, cargo;
 Emergency.

3

Sub-Module 15 - Oxygen (ATA 35)

System layout: cockpit, cabin;
 Sources, storage, charging and distribution;
 Supply regulation;
 Indications and warnings.

3

Sub-Module 16 - Pneumatic/Vacuum (ATA 36)

System layout: cockpit, cabin;
 Sources, storage, charging and distribution;
 Supply regulation;
 Indications and warnings;
 Interfaces with other systems.

3

Sub-Module 17 - Water/Waste (ATA 38)

Water system layout, supply, distribution, servicing and draining;
 Toilet system layout, flushing and servicing;
 Corrosion aspects.

3

Sub-Module 18 - On Board Maintenance Systems (ATA 45)

Central maintenance computers;
 Data loading system;
 Electronic library system;
 Printing;
 Structure monitoring (damage tolerance monitoring).

2

Sub-Module 19 - Integrated Modular Avionics (ATA 42)

Functions that may be typically integrated in the Integrated Modular Avionic (IMA) modules:
 Bleed Management, Air Pressure Control, Air Ventilation and Control, Avionics and Cockpit
 Ventilation Control, Temperature Control, Air Traffic Communication, Avionics Communication
 Router, Electrical Load Management, Circuit Breaker Monitoring, Electrical System BITE, Fuel
 Management, Braking Control, Steering Control, Landing Gear Extension and Retraction, Tire
 Pressure Indication, Oleo Pressure Indication, Brake Temperature Monitoring, etc.
 Core System; Network Components;

2

PART-66 - APPENDIX I BASIC KNOWLEDGE REQUIREMENTS

Sub-Module 20 - Cabin Systems (ATA 44)

The units and components which furnish a means of entertaining the passengers and providing communication within the aircraft (Cabin Intercommunication Data System) and between the aircraft cabin and ground stations (Cabin Network Service). Includes voice, data, music and video transmissions.

The Cabin Intercommunication Data System provides an interface between cockpit/cabin crew and cabin systems. These systems support data exchange of the different related LRU's and they are typically operated via Flight Attendant Panels. The Cabin Network Service typically consists on a server, typically interfacing with, among others, the following systems:

- Data/Radio Communication, Inflight Entertainment System.
- The Cabin Network Service may host functions such as:
 - Access to predeparture/departure reports,
 - E-mail/intranet/internet access,
 - Passenger database,

Cabin Core System;

Inflight Entertainment System;

External Communication System;

Cabin Mass Memory System;

Cabin Monitoring System;

Miscellaneous Cabin System.

Sub-Module 21 - Information Systems (ATA 46)

The units and components which furnish a means of storing, updating and retrieving digital information traditionally provided on paper, microfilm or microfiche. Includes units that are dedicated to the information storage and retrieval function such as the electronic library mass storage and controller. Does not include units or components installed for other uses and shared with other systems, such as flight deck printer or general use display.

Examples include Air Traffic and Information Management Systems and Network Server Systems:

Aircraft General Information System;

Flight Deck Information System;

Maintenance Information System;

Passenger Cabin Information System;

Miscellaneous Information System.

TURBINE AEROPLANE AERODYNAMICS, STRUCTURES AND SYSTEMS

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TURBINE AEROPLANE AERODYNAMICS, STRUCTURES AND SYSTEMS

THEORY OF FLIGHT

THEORY OF FLIGHT

SUB-MODULE 01

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 01

THEORY OF FLIGHT

Knowledge Requirements

11.1 - *Theory of Flight*

11.1.1 - *Aeroplane Aerodynamics and Flight Controls*

Operation and effect of:

- roll control: ailerons and spoilers;
- pitch control: elevators, stabilator's, variable incidence stabilizers and canards;
- yaw control, rudder limiters;

Control using elevons, ruddervators; High lift devices, slots, slats, flaps, flaperons;

Drag inducing devices, spoilers, lift dumpers, speed brakes;

Effects of wing fences, saw tooth leading edges;

Boundary layer control using, vortex generators, stall wedges or leading edge devices;

Operation and effect of trim tabs, balance and antibalance (leading) tabs, servo tabs, spring tabs, mass balance, control surface bias, aerodynamic balance panels.

2

11.1.2 - *High Speed Flight*

Speed of sound, subsonic flight, transonic flight, supersonic flight;

Mach number, critical Mach number, compressibility buffet, shock wave, aerodynamic heating, area rule;

Factors affecting airflow in engine intakes of high speed aircraft;

Effects of sweepback on critical Mach number.

2

11.1 - THEORY OF FLIGHT

AEROPLANE AERODYNAMICS AND FLIGHT CONTROLS

The directional control of a fixed wing aircraft takes place around the lateral, longitudinal, and vertical axes by means of flight control surfaces designed to create movement about these axes. These control devices are hinged or movable surfaces through which the attitude of an aircraft is controlled during takeoff, flight, and landing. They are usually divided into two major groups: 1) primary or main flight control surfaces and 2) secondary or auxiliary control surfaces.

PRIMARY FLIGHT CONTROL SURFACES

The primary flight control surfaces on a fixed wing aircraft include: ailerons, elevators, and the rudder. The ailerons are attached to the trailing edge of both wings and when moved, rotate the aircraft around the longitudinal axis. The elevator is attached to the trailing edge of the horizontal stabilizer. When it is moved, it alters aircraft pitch, which is the attitude about the horizontal or lateral axis. The rudder is hinged to the trailing edge of the vertical stabilizer. When the rudder changes position, the aircraft rotates about the vertical axis (yaw). *Figure 1-1* shows the primary flight controls of a light aircraft and the movement they create relative to the three axes of flight.

Primary control surfaces are usually similar in construction to one another and vary only in size, shape, and methods of attachment. On aluminum light aircraft, their structure is often similar to an all metal wing. This is appropriate because the primary control surfaces are simply smaller aerodynamic devices. They are typically made from an aluminum alloy structure built around a single spar member or torque tube to which ribs are fitted and a skin is attached. The lightweight ribs are, in many cases, stamped out from flat aluminum sheet stock. Holes in the ribs lighten the assembly. An aluminum skin is attached with rivets. *Figure 1-2* illustrates this type of structure, which can be found on the primary control surfaces of light aircraft as well as on medium and heavy aircraft.

Primary control surfaces constructed from composite materials are also commonly used. These are found on many heavy and high performance aircraft, as well as

gliders, homebuilt, and light sport aircraft. The weight and strength advantages over traditional construction can be significant. A wide variety of materials and construction techniques are employed. *Figure 1-3* shows examples of aircraft that use composite technology on primary flight control surfaces. Note that the control surfaces of fabric covered aircraft often have fabric covered surfaces just as aluminum skinned (light) aircraft typically have all aluminum control surfaces. There is a critical need for primary control surfaces to be balanced so they do not vibrate or flutter in the wind.

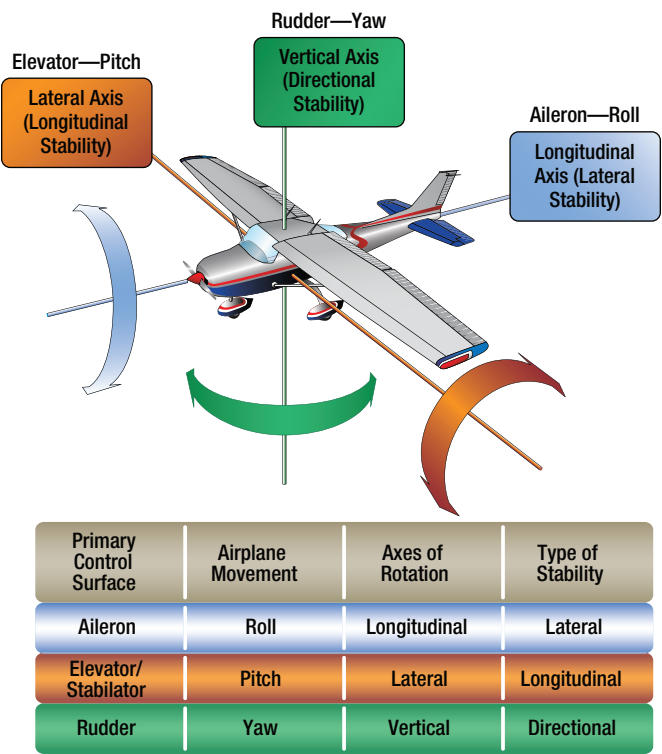


Figure 1-1. Flight control surfaces move the aircraft around the three axes of flight.

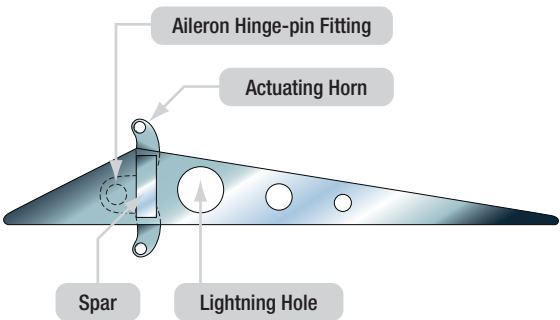


Figure 1-2. Typical structure of an aluminum flight control surface.



Figure 1-3. Composite control surfaces and some of the many aircraft that utilize them.

Performed to manufacturer's instructions, balancing usually consists of assuring that the center of gravity of a particular device is at or forward of the hinge point. Failure to properly balance a control surface could lead to catastrophic failure. **Figure 1-4** illustrates several aileron configurations with their hinge points well aft of the leading edge. This is a common design feature used to prevent flutter.

OPERATION AND EFFECT OF ROLL CONTROL DEVICES

AILERONS

Ailerons are the primary flight control surfaces that move the aircraft about the longitudinal axis. In other words, movement of the ailerons in flight causes the

aircraft to roll. Ailerons are usually located on the outboard trailing edge of each of the wings. They are built into the wing and are calculated as part of the wing's surface area. **Figure 1-5** shows aileron locations on various wing tip designs.

Ailerons are controlled by a side to side motion of the control stick in the cockpit or a rotation of the control yoke. When the aileron on one wing deflects down, the aileron on the opposite wing deflects upward. This amplifies the movement of the aircraft around the longitudinal axis. On the wing on which the aileron trailing edge moves downward, camber is increased and lift is increased. Conversely, on the other wing, the raised aileron decreases lift. (**Figure 1-6**) The result is a sensitive response to the control input to roll the aircraft.

The pilot's request for aileron movement and roll are transmitted from the cockpit to the actual control surface in a variety of ways depending on the aircraft. A system of control cables and pulleys, push pull tubes, hydraulics, electric, or a combination of these can be employed. (**Figure 1-7**)

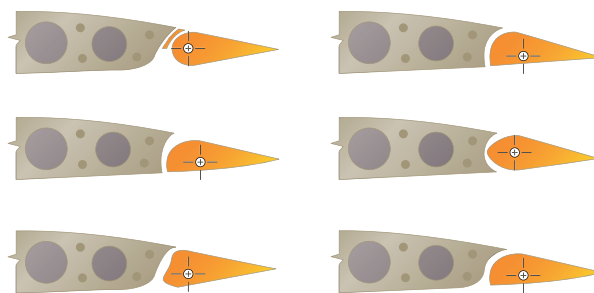


Figure 1-4. Aileron hinge locations are very close to but aft of the center of gravity to prevent flutter.

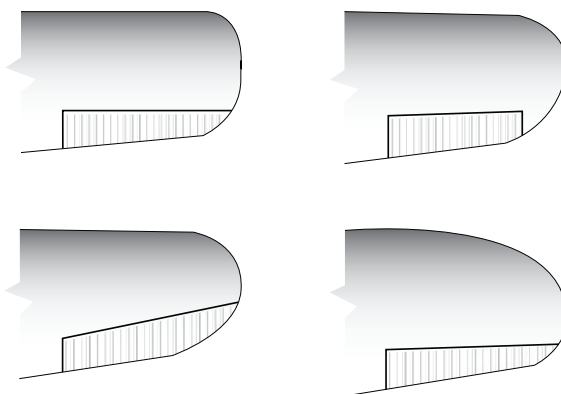


Figure 1-5. Aileron location on various wings.

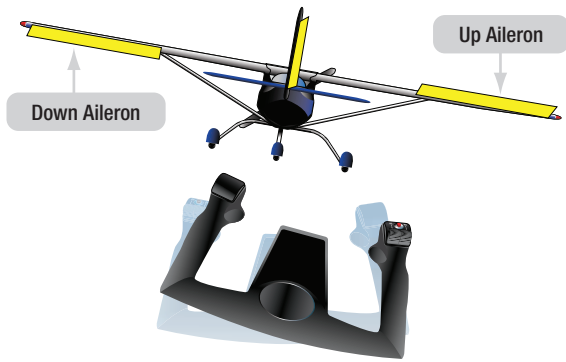


Figure 1-6. Differential aileron control movement. When one aileron is moved down, the aileron on the opposite wing is deflected upward.

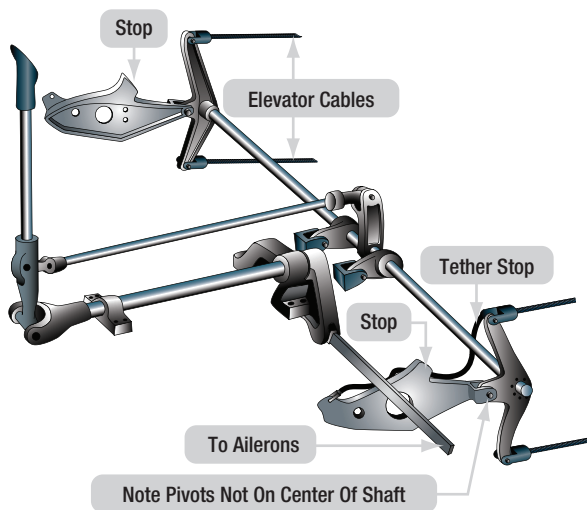


Figure 1-7. Transferring control surface inputs from the cockpit.

Simple, light aircraft usually do not have hydraulic or electric fly by wire aileron control. These are found on heavy and high performance aircraft. Large aircraft and some high performance aircraft may also have a second set of ailerons located inboard on the trailing edge of the wings. These are part of a complex system of primary and secondary control surfaces used to provide lateral control and stability in flight. At low speeds, the ailerons may be augmented by the use of flaps and spoilers. At high speeds, only inboard aileron deflection is required to roll the aircraft while the other control surfaces are locked out or remain stationary. **Figure 1-8** illustrates the location of the typical flight control surfaces found on a transport category aircraft.

SPOILERS

A spoiler is a device found on the upper surface of many heavy and high performance aircraft. It is stowed flush to the wing's upper surface. When deployed, it raises up into the airstream and disrupts the laminar airflow of the wing, thus reducing lift.

Spoilers are made with similar construction materials and techniques as the other flight control surfaces on the aircraft. Often, they are honeycomb core flat panels. At low speeds, spoilers are rigged to operate when the ailerons operate to assist with the lateral movement and

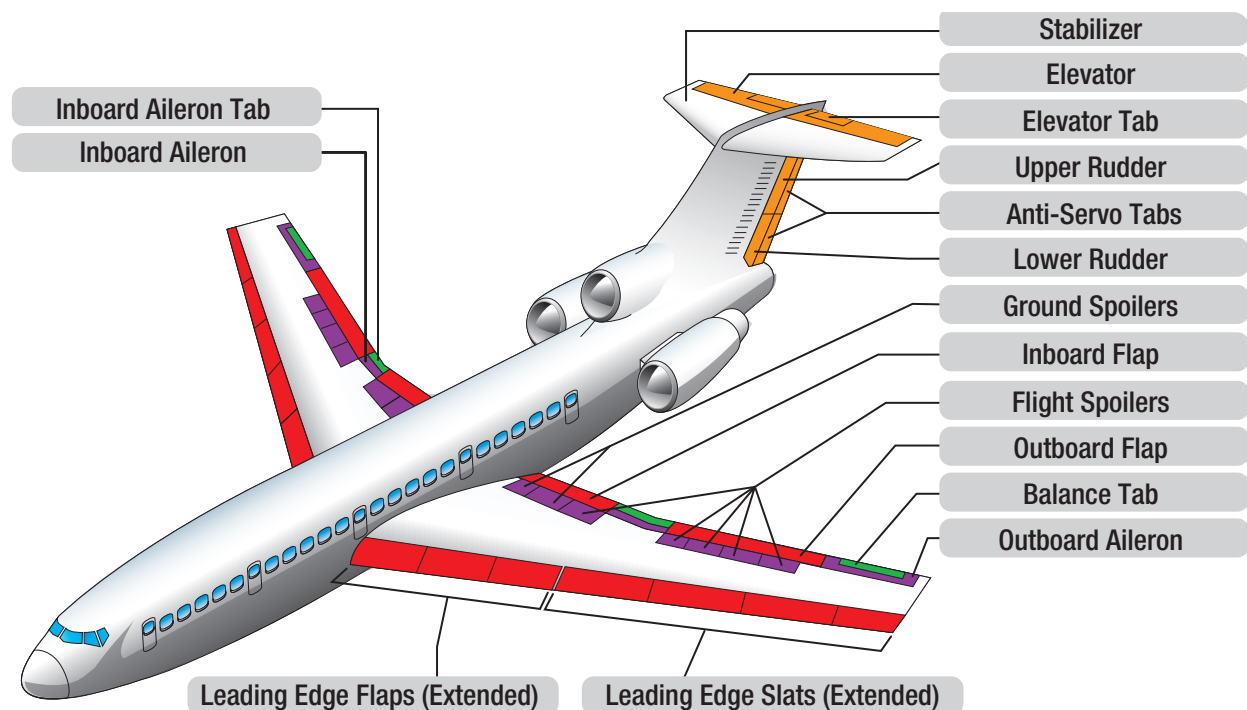


Figure 1-8. Typical flight control surfaces on a transport category aircraft.

stability of the aircraft. On the wing where the aileron is moved up, the spoilers also raise thus amplifying the reduction of lift on that wing. (*Figure 1-9*)

On the wing with downward aileron deflection, the spoilers remain stowed. As the speed of the aircraft increases, the ailerons become more effective and the spoiler interconnect disengages. Note that spoilers are also used in as drag inducing devices.

OPERATION AND EFFECT OF PITCH CONTROL DEVICES

ELEVATORS

The elevator is the primary flight control surface that moves the aircraft around the horizontal or lateral axis. This causes the nose of the aircraft to pitch up or down. The elevator is hinged to the trailing edge of the horizontal stabilizer and typically spans most or all of its width. It is controlled in the cockpit by pushing or pulling the control yoke forward or aft.

Light aircraft use a system of control cables and pulleys or push pull tubes to transfer cockpit inputs to the movement of the elevator. High performance and large aircraft typically employ more complex systems. Hydraulic power is commonly used to move the elevator on these aircraft. On aircraft equipped with fly by wire controls, a combination of electrical and hydraulic power is used.

STABILATORS

A movable horizontal tail section, called a stabilator, is a control surface that combines the action of both the horizontal stabilizer and the elevator. (*Figure 1-10*) Basically, a stabilator is a horizontal stabilizer that can also be rotated about the horizontal axis to affect the pitch of the aircraft.

VARIABLE INCIDENCE STABILIZERS

A variable incidence stabilizer refers to any horizontal stabilizer in which the angle of incidence of the horizontal stabilizer is adjustable. Thus, a stabilator is a variable incidence horizontal stabilizer. Various mechanisms and operating rigging are available. Most large aircraft use a motorized jackscrew to alter the position of the stabilizer often energized by the trim tab switch on the control yoke. The reason for a stabilator or any horizontal stabilizer variable incidence device is

to minimize drag when trimming the aircraft in flight. Deflection of the elevator via the use of a trim tab causes drag and requires a relatively large elevator on large aircraft to achieve all desired trim settings. By varying the angle of the horizontal stabilizer to adjust pitch, less drag is created and elevator size and deflection may be reduced. (*Figure 1-11*)

CANARDS

A canard utilizes the concept of two lifting surfaces. It functions as a horizontal stabilizer located in front of the main wings. In effect, the canard is an airfoil similar to the horizontal surface on a conventional aft tail design. The difference is that the canard actually creates lift and



Figure 1-9. Spoilers deployed upon landing a transport category aircraft.



Figure 1-10. A stabilizer and index marks on a transport category aircraft.

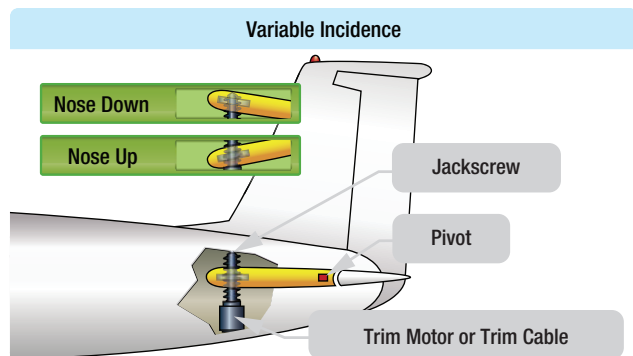


Figure 1-11. Some airplanes, including most jet transports, use an variable stabilizer to provide the required pitch trim forces.

holds the nose up, as opposed to the aft tail design which exerts downward force on the tail to prevent the nose from rotating downward. (*Figure 1-12*)

The canard design dates back to the pioneer days of aviation, most notably used on the Wright Flyer. Recently, the canard configuration has regained popularity and is appearing on newer aircraft. Canard designs include two types, one with a horizontal surface of about the same size as a normal aft tail design, and the other with a surface of the same approximate size and airfoil shape of the aft mounted wing known as a tandem wing configuration. Theoretically, the canard is considered more efficient because using the horizontal surface to help lift the weight of the aircraft should result in less drag for a given amount of lift.

OPERATION AND EFFECT OF YAW CONTROL DEVICES

RUDDERS

The rudder is the primary control surface that causes an aircraft to yaw or move about the vertical axis. This provides directional control and thus points the nose of the aircraft in the direction desired. Most aircraft have a single rudder hinged to the trailing edge of the vertical stabilizer. It is controlled by a pair of foot operated rudder pedals in the cockpit. When the right pedal is pushed forward, it deflects the rudder to the right which moves the nose of the aircraft to the right. The left pedal is rigged to simultaneously move aft. When the left pedal is pushed forward, the nose of the aircraft moves to the left.



Figure 1-12. The Piaggio P180 includes a variable-sweep canard design, which provides longitudinal stability about the lateral axis.

As with the other primary flight controls, the transfer of the movement of the cockpit controls to the rudder varies with the complexity of the aircraft. Many aircraft incorporate the directional movement of the nose or tail wheel into the rudder control system for ground operation. This allows the operator to steer the aircraft with the rudder pedals during taxi when the airspeed is not high enough for the control surfaces to be effective. Some large aircraft have a split rudder arrangement. This is actually two rudders, one above the other. At low speeds, both rudders deflect in the same direction when the pedals are pushed. At higher speeds, one of the rudders becomes inoperative as the deflection of a single rudder is aerodynamically sufficient to maneuver the aircraft.

RUDDER LIMITERS

In flight, most large aircraft oscillate slightly from side to side. Yaw dampener units automatically detect this movement and send signals to the hydraulic power control unit (PCU) that moves the rudder so that it can correct for these yaw oscillations. Similarly, rudders are known to deflect without being commanded to do so by the flight crew. Again, the yaw dampener is designed to correct the fluctuations by signaling the PCU. However, too large of an involuntary deflection to a rudder can cause a loss of control of the aircraft.

A rudder limiter is fitted to many aircraft to prevent any more than a few degrees of involuntary motion of the rudder. Essentially, it limits the movement unless it is commanded from the flight deck.

SECONDARY OR AUXILIARY CONTROL SURFACES

There are several secondary or auxiliary flight control surfaces. Their names, locations, and functions of those for most large aircraft are listed in *Figure 1-13*.

OPERATION AND EFFECT OF TABS

TRIM TABS

The force of the air against a control surface during the high speed of flight can make it difficult to move and hold that control surface in the deflected position. A control surface might also be too sensitive for similar reasons. Several different tabs are used to aid with these types of problems. The table in *Figure 1-14* summarizes the various tabs and their uses. While in flight, it is desirable for the pilot to be able to take his or her hands and feet off of the controls and have the aircraft maintain its flight condition.

Trims tabs are designed to allow this. Most trim tabs are small movable surfaces located on the trailing edge of a primary flight control surface. A small movement of the tab in the direction opposite of the direction the flight control surface is deflected, causing air to strike the tab,

in turn producing a force that aids in maintaining the flight control surface in the desired position. Through linkage set from the cockpit, the tab can be positioned so that it is actually holding the control surface in position rather than the pilot. Therefore, elevator tabs are used to maintain the speed of the aircraft since they assist in maintaining the selected pitch. Rudder tabs can be set to hold yaw in check and maintain heading. Aileron tabs can help keep the wings level.

Occasionally, a simple light aircraft may have a stationary metal plate attached to the trailing edge of a primary flight control, usually the rudder. This is also a trim tab as shown in *Figure 1-15*. It can be bent slightly on the ground to trim the aircraft in flight to a hands off condition when flying straight and level. The correct amount of bend can be determined only by flying the aircraft after an adjustment. Note that a small amount of bending is usually sufficient.

BALANCE TABS

The aerodynamic phenomenon of moving a trim tab in one direction to cause the control surface to experience a force moving in the opposite direction is exactly what occurs with the use of balance tabs. (*Figure 1-16*) Often,

Secondary/Auxiliary Flight Control Surfaces		
Name	Location	Function
Flaps	Inboard trailing edge of wings	Extends the camber of the wing for greater lift and slower flight. Allows control at low speeds for short field takeoffs and landings.
Trim Tabs	Trailing edge of primary flight control surfaces	Eliminates the force needed to move a primary control surface (zero Newtons; hands free).
Balance Tabs	Trailing edge of primary flight control surfaces	Reduces the force needed to move a primary control surface.
Anti-balance Tabs	Trailing edge of primary flight control surfaces	Increases feel and effectiveness of primary control surface.
Servo Tabs	Trailing edge of primary flight control surfaces	Assists or provides the force for moving a primary flight control.
Spoilers	Upper and/or trailing edge of wing	Decreases (spoils) lift. Can augment aileron function.
Slats	Mid to outboard leading edge of wing	Extends the camber of the wing for greater lift and slower flight. Allows control at low speeds for short field takeoffs and landings.
Slots	Outer leading edge of wing forward of ailerons	Directs air over upper surface of wing during high angle of attack. Lowers stall speed and provides control during slow flight.
Leading Edge Flap	Inboard leading edge of wing	Extends the camber of the wing for greater lift and slower flight. Allows control at low speeds for short field takeoffs and landings.

NOTE: An aircraft may possess none, one, or a combination of the above control surfaces.

Figure 1-13. Secondary or auxiliary control surfaces and respective locations for larger aircraft.

Flight Control Tabs			
Type	Direction of Motion (in relation to control surface)	Activation	Effect
Trim	Opposite	Set by pilot from cockpit. Uses independent linkage.	Statically balances the aircraft in flight. Allows “hands off” maintenance of flight condition.
Balance	Opposite	Moves when pilot moves control surface. Coupled to control surface linkage.	Aids pilot in overcoming the force needed to move the control surface.
Servo	Opposite	Directly linked to flight control input device. Can be primary or back-up means of control.	Aerodynamically positions control surfaces that require too much force to move manually.
Anti-balance or Anti-servo	Same	Directly linked to flight control input device.	Increases force needed by pilot to change flight control position. De-sensitizes flight controls.
Spring	Opposite	Located in line of direct linkage to servo tab. Spring assists when control forces become too high in high-speed flight.	Enables moving control surface when forces are high. Inactive during slow flight.

Figure 1-14. Various tabs and their uses.

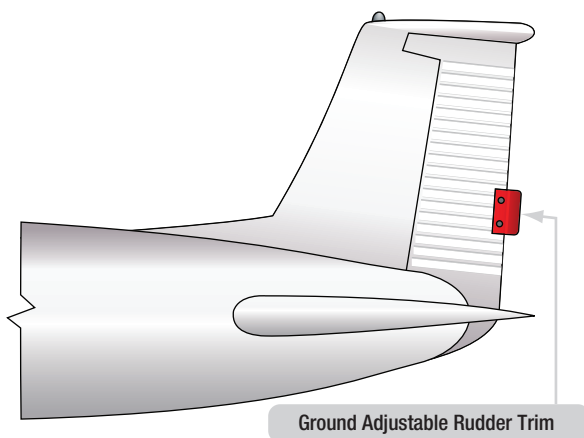


Figure 1-15. Example of a trim tab.

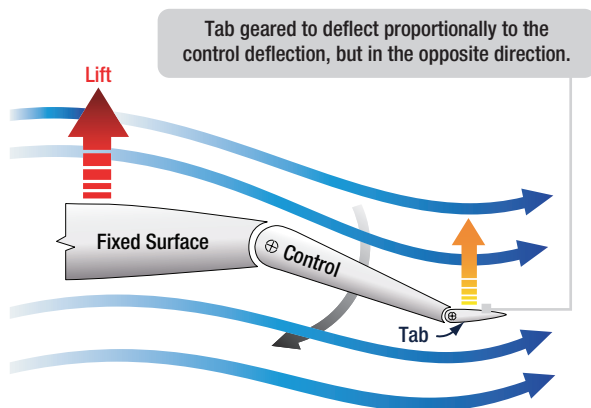


Figure 1-16. Balance tabs assist with forces needed to position control surfaces.

it is difficult to move a primary control surface due to its surface area and the speed of the air rushing over it. Deflecting a balance tab hinged at the trailing edge of the control surface in the opposite direction of the desired control surface movement causes a force to position the surface in the proper direction with reduced force to do so. Balance tabs are usually linked directly to the control surface linkage so that they move automatically when there is an input for control surface movement. They also can double as trim tabs, if adjustable on the flight deck.

SERVO TABS

A servo tab is similar to a balance tab in location and effect, but it is designed to operate the primary flight control surface, not just reduce the force needed to do so. It is usually used as a means to back up the primary control of the flight control surfaces. (*Figure 1-17*)

On heavy aircraft, large control surfaces require too much force to be moved manually and are usually deflected out of the neutral position by hydraulic actuators. These power control units are signaled via a system of hydraulic valves connected to the yoke and rudder pedals. On fly by wire aircraft, the hydraulic actuators that move the flight control surfaces are signaled by electric input. In the case of hydraulic system failure(s), manual linkage to a servo tab can be used to deflect it. This, in turn, provides an aerodynamic force that moves the primary control surface.

ANTI-SERVO/ANTI-BALANCE TABS

Anti-servo tabs, as the name suggests, are like servo tabs but move in the same direction as the primary control surface. On some aircraft, especially those with a movable horizontal stabilizer, the input to the control surface can be too sensitive. An Anti-servo tab tied through the control linkage creates an aerodynamic force that increases the effort needed to move the control surface. This makes flying the aircraft more stable for the pilot. **Figure 1-18** shows an Anti-servo tab in the near neutral position. Deflected in the same direction as the desired stabilator movement, it increases the required control surface input. Anti servo tabs are also known as anti-balance tabs.

SPRING TABS

A control surface may require excessive force to move only in the final stages of travel. When this is the case, a spring tab can be used. This is essentially a servo tab that does not activate until an effort is made to move the control surface beyond a certain point. When reached, a spring in line of the control linkage aids in moving the control surface through the remainder of its travel. (**Figure 1-19**)

AERODYNAMIC BALANCE PANELS

Figure 1-20 shows another way of assisting the movement of an aileron on a large aircraft. It is called an aileron balance panel. Not visible when approaching the aircraft, it is positioned in the linkage that hinges the aileron to the wing. Balance panels have been constructed typically of aluminum skin covered frame assemblies or aluminum honeycomb structures. The trailing edge of the wing just forward of the leading edge of the aileron is sealed to allow controlled airflow in and out of the hinge area where the balance panel is located.

MASS BALANCE

Flutter is an undesirable oscillation of an aircraft control surface which can have catastrophic effect on controllability of the aircraft. The center of lift on a control surface should be aft of the control surface center of gravity to prevent control surface flutter. Often, the addition of weight to the forward surface of an aileron, for example, is sufficient to move the CG of the airfoil forward and prevent flutter. Some aircraft designs, however, place the weight on a lever arm that extends forward of the control surface. This is known as a mass

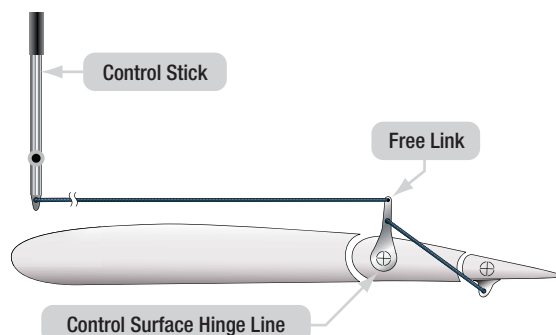


Figure 1-17. Servo tabs can be used to position flight control surfaces in case of hydraulic failure.

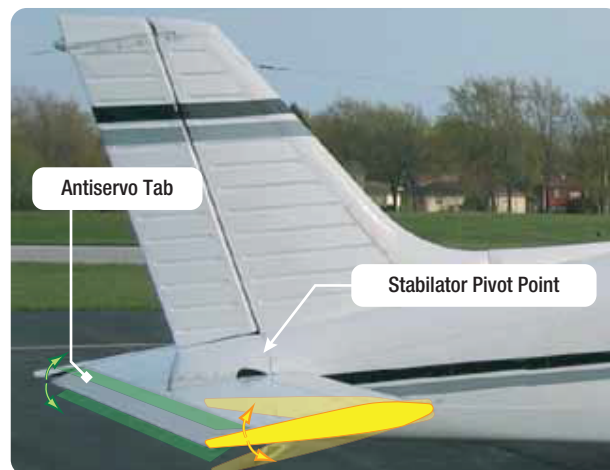


Figure 1-18. An Anti-servo tab moves in the same direction as the control tab. Shown here on a stabilator, it desensitizes the pitch control.

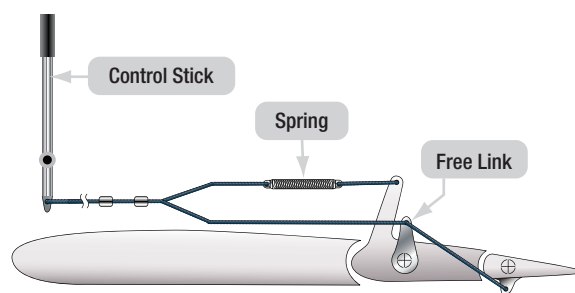


Figure 1-19. Many tab linkages have a spring tab that kicks in as the forces needed to deflect a control increase with speed and the angle of desired deflection.

balance. Mass balances help prevent flutter and also reduce the required control stick pressure used to move a control surface. (**Figure 1-21**)

CONTROL SURFACE BIAS

When a control surface is in the neutral position, is faired with the wing rudder or horizontal stabilizer and no effect on the aircraft's aerodynamic surfaces. Some aircraft are designed with control surface bias.

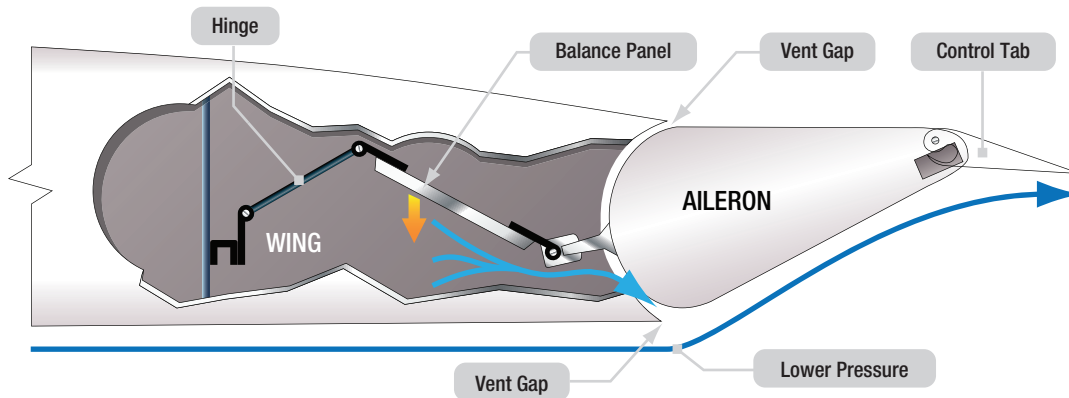


Figure 1-20. An aileron balance panel and linkage uses varying air pressure to assist in control surface positioning.



Figure 1-21. An aileron mass balance.

This means that a control surface is not naturally in the neutral position. It is designed to impart a force on the airfoil at all times. The force is generally used to counter balance a design imbalance and alter the aircraft's aerodynamics for easy hands off flight. This means that

when the aircraft is flying straight and level, the control surface bias has effect but all trim position gauges on the flight deck indicate zero trim.

HIGH LIFT DEVICES

Aircraft wings contain devices that are designed to increase the lift produced by the wing with the devices deployed during certain phases of flight.

FLAPS

Flaps are one such high lift device found on most aircraft. They are usually inboard on the wings' trailing edges adjacent to the fuselage. Leading edge flaps are also common. They extend forward and down from the inboard wing leading edge. The flaps are lowered to increase the camber of the wings and provide greater lift and control at slow speeds. They enable landing at slower speeds and shorten the amount of runway required for takeoff and landing. The amount that the flaps extend and the angle they form with the wing can be selected from the cockpit. Typically, flaps can extend up to 45–50°. *Figure 1-22* shows various aircraft with flaps in the extended position.



Figure 1-22. An aileron balance panel and linkage uses varying air pressure to assist in control surface positioning.

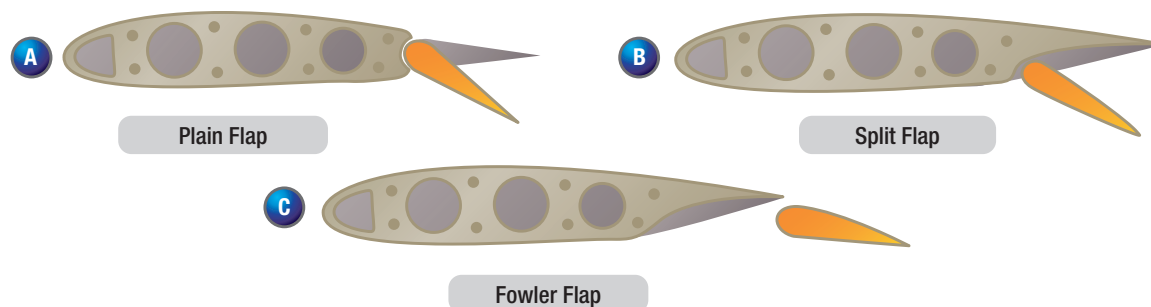


Figure 1-23. Various types of flaps.

Flaps are usually constructed of materials and with techniques used on the other airfoils and control surfaces of a particular aircraft. Aluminum skin and structure flaps are the norm on light aircraft. Heavy and high performance aircraft flaps may also be aluminum, but the use of composite structures is also common.

There are various kinds of flaps. Plain flaps form the trailing edge of the wing when the flap is in the retracted position. (*Figure 1-23A*) The airflow over the wing continues over the upper and lower surfaces of the flap, making the trailing edge of the flap essentially the trailing edge of the wing. The plain flap is hinged so that the trailing edge can be lowered. This increases wing camber and provides greater lift.

A split flap is normally housed under the trailing edge of the wing. (*Figure 1-23B*) It is usually just a braced flat metal plate hinged at several places along its leading edge. The upper surface of the wing extends to the trailing edge of the flap. When deployed, the split flap trailing edge lowers away from the trailing edge of the wing. Airflow over the top of the wing remains the same. Airflow under the wing now follows the camber created by the lowered split flap, increasing lift.

Fowler flaps not only lower the trailing edge of the wing when deployed but also slide aft, effectively increasing the area of the wing. (*Figure 1-23C*) This creates more lift via the increased surface area, as well as the wing camber. When stowed, the fowler flap typically retracts up under the wing trailing edge similar to a split flap. The sliding motion of a fowler flap can be accomplished with a worm drive and flap tracks.

An enhanced version of the fowler flap is a set of flaps that actually contains more than one aerodynamic surface. *Figure 1-24* shows a triple slotted flap. In this

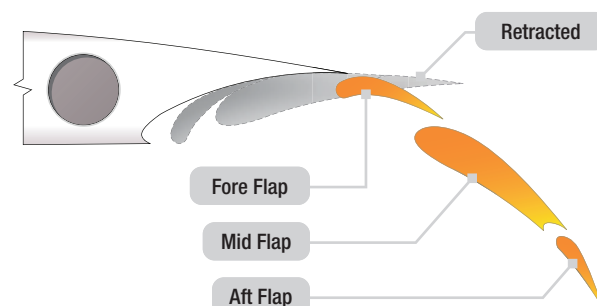


Figure 1-24. Triple slotted flap.

configuration, the flap consists of a fore flap, a mid flap, and an aft flap. When deployed, each flap section slides aft on tracks as it lowers. The flap sections also separate leaving an open slot between the wing and the fore flap, as well as between each of the flap sections. Air from the underside of the wing flows through these slots. The result is that the laminar flow on the upper surfaces is enhanced. The greater camber and effective wing area increase overall lift.

Heavy aircraft often have leading edge flaps that are used in conjunction with the trailing edge flaps. (*Figure 1-25*) They can be made of machined magnesium or can have

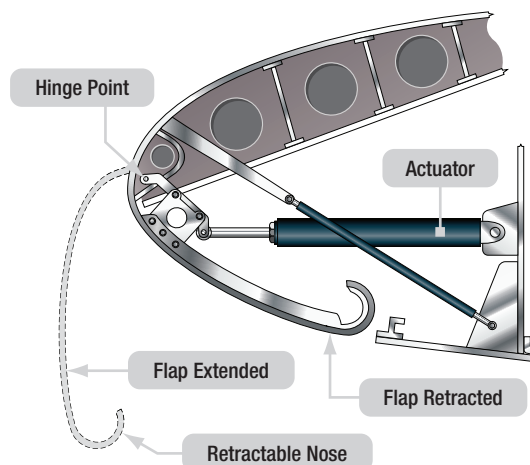


Figure 1-25. Leading edge flaps.



Figure 1-26. Side view (left) and front view (right) of a Krueger flap on a Boeing 737.

an aluminum or composite structure. While they are not installed or operate independently, their use with trailing edge flaps can greatly increase wing camber and lift. When stowed, leading edge flaps retract into the leading edge of the wing.

The differing designs of leading edge flaps essentially provide the same effect. Activation of the trailing edge flaps automatically deploys the leading edge flaps, which are driven out of the leading edge and downward, extending the camber of the wing. **Figure 1-26** shows a Krueger flap, recognizable by its flat midsection.

FLAPERONS

Some aircraft are equipped with flaperons. (**Figure 1-27**) Flaperons are ailerons which can also act as flaps. Flaperons combine both aspects of flaps and ailerons. In addition to controlling the bank angle of an aircraft like conventional ailerons, flaperons can be lowered together to function much the same as a dedicated set of flaps. The pilot retains separate controls for ailerons and flaps. A mixer is used to combine the separate pilot inputs into

this single set of control surfaces called flaperons. Many designs that incorporate flaperons mount the control surfaces away from the wing to provide undisturbed airflow at high angles of attack and/or low airspeeds.

SLATS

Another leading edge device which extends wing camber is a slat. Slats can be operated independently of the flaps with their own switch in the cockpit. Slats not only extend out of the leading edge of the wing increasing camber and lift, but most often, when fully deployed leave a slot between their trailing edges and the leading edge of the wing. (**Figure 1-28**) This increases the angle of attack at which the wing will maintain its laminar airflow, resulting in the ability to fly the aircraft slower and still maintain control.



Figure 1-27. Flaperons on a Skystar Kitfox MK 7.

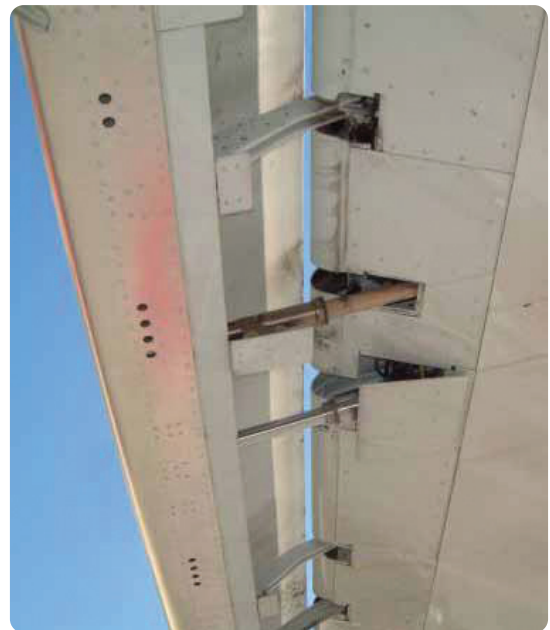


Figure 1-28. Air passing through the slot aft of the slat promotes boundary layer airflow on the upper surface at high angles of attack.

SLOTS

A fixed device mounted to extend the leading edge of the wing forward and downward is known as a slot or cuff. (*Figure 1-29*) It essentially increases the camber of the wing and allows the aircraft to fly at slower speeds and higher angles of attack. Moreover, slots reduce the stall speed of the aircraft by mixing high speed air flow exiting the slot with boundary layer air. The result is a delay in boundary layer separation. However, slots increase drag. The benefits of good low speed handling characteristics when weighed against the increased drag that a slot causes at higher speeds limits the use of slots. Full span slots span the full wing from root to tip. They are commonly used on STOL (short takeoff and landing) aircraft. Partial span slots are positioned on the outboard section of the wing leading edge. This increases the angle of attack at which the outboard wing stalls and ensures that the wing root stalls first. When the wing root stalls first, stall characteristics are docile. Recovery is easier because the partial span slots maintain air flow over the ailerons during the stall.



Figure 1-29. A leading edge slot on a STOL aircraft.

ELEVONS AND RUDDERVATORS

Elevons perform the combined functions of the ailerons and the elevator. (*Figure 1-30*) They are typically used on aircraft that have no true separate empennage such as a delta wing or flying wing aircraft.

They are installed on the trailing edge of the wing. When moved in the same direction, the elevons cause a pitch adjustment. When moved in opposite directions, the aircraft rolls. Elevons may also move differentially in the same direction causing adjustments to roll and pitch. The control yoke or stick activated elevon movement through a mechanical or electronic mixing device. A ruddervator combines the action of the rudder and elevator. (*Figure 1-31*)

This is possible on aircraft with V-tail empennages where the traditional horizontal and vertical stabilizers do not exist. Instead, two stabilizers angle upward and outward from the aft fuselage in a "V" configuration. Each contains a movable ruddervator built into the trailing edge. Movement of the ruddervators can alter the movement of the aircraft around the horizontal and/or vertical axis.



Figure 1-30. Elevons.



Figure 1-31. Ruddervator.

DRAW INDUCING DEVICES

SPOILERS

Spoilers are unique in that they may be fully deployed on both wings to act as speed brakes. The reduced lift and increased drag can quickly reduce the speed of the aircraft in flight. Spoilers are sometimes called lift dumpers.

SPEED BRAKES

Dedicated speed brake panels similar to flight spoilers in construction can be found on the upper surface of the wing trailing edge of heavy and high performance aircraft. They are designed specifically to increase drag and reduce the speed of the aircraft when deployed. These speed brake panels do not operate differentially with the ailerons at low speed like the spoilers.

A speed brake control lever in the cockpit can deploy all spoiler and speed brake surfaces fully when operated. Often, speed brakes surfaces are rigged to deploy on the ground automatically when engine thrust reversers are activated. The location of speed brake panels is visible in *Figure 1-8*.

BOUNDARY LAYER CONTROLS

The boundary layer is a very thin layer of air lying over the surface of the wing and, for that matter, all other surfaces of the aeroplane. Because air has viscosity, this layer of air tends to adhere to the wing. As the wing moves forward through the air, the boundary layer at first flows smoothly over the streamlined shape of the airfoil. This flow is called the laminar layer. As the boundary layer approaches the center of the wing, it begins to lose speed due to skin friction and it becomes thicker and turbulent. Here it is called the turbulent layer.

The point at which the boundary layer changes from laminar to turbulent is called the transition point. Where the boundary layer becomes turbulent, drag due to skin friction is relatively high. As speed increases, the transition point tends to move forward. As the angle of attack increases, the transition point also tends to move forward. With higher angles of attack and further thickening of the boundary layer, the turbulence becomes so great the air breaks away from the surface of the wing. At this point, the lift of the wing is destroyed and a condition known as a stall has occurred.

In *Figure 1-32*, view A shows a normal angle of attack and the airflow staying in contact with the wing. View B shows an extreme angle of attack and the airflow separating and becoming turbulent on the top of the wing. In view B, the wing is in a stall.

VORTEX GENERATORS

Vortex generators are small airfoil sections usually attached to the upper surface of a wing. (*Figure 1-33*) They are designed to promote positive laminar airflow over the wing and control surfaces.

Usually made of aluminum and installed in a spanwise line or lines, the vortices created by these devices swirl downward assisting maintenance of the boundary layer of air flowing over the wing. They can also be found on the fuselage and empennage. *Figure 1-34* shows the unique vortex generators on a Symphony SA-160 wing.

WING FENCE

A chordwise barrier on the upper surface of the wing, called a wing fence or stall fence, is used to halt the spanwise flow of air along the wing. During low speed flight, this can maintain proper chordwise airflow reducing the tendency for the wing to stall. Usually made of aluminum, the fence is a fixed structure most common on swept wings, which have a natural spanwise tending boundary air flow. (*Figure 1-35*)

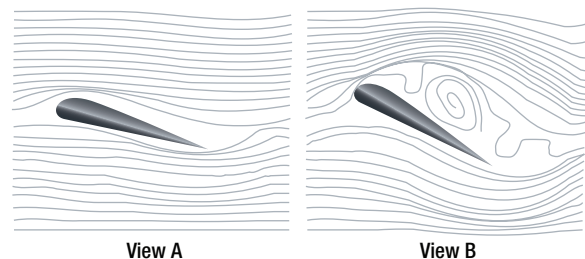


Figure 1-32. Wing boundary layer separation.



Figure 1-33. Vortex generators.



Figure 1-34. The Symphony SA-160 has two unique vortex generators on its wing to ensure aileron effectiveness through the stall.



Figure 1-36. A stall wedge causes the wing root to stall before the outboard wing. This preserves airflow over the ailerons for controlled recovery from the stall.

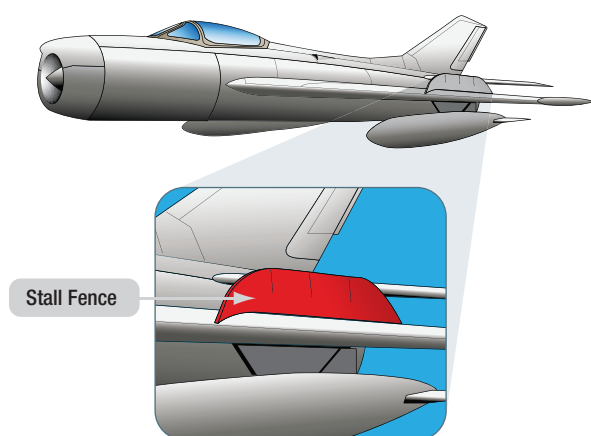


Figure 1-35. A stall fence aids in maintaining chordwise airflow over the wing.

STALL WEDGES

A stall wedge or stall strip is a fixed wedge shaped strip attached spanwise to the wing leading edge. (*Figure 1-36*) It is located on the inboard section of the wing at such a point that it causes the boundary airflow to become turbulent as the angle of attack increases to a certain point. This purposeful destruction of the boundary airflow as the angle of attack increases causes the root of the wing to stall first. Thus, airflow over the outboard wing section and over the ailerons is preserved during the stall making it easier to recover. A wedge can also serve as a stall warning device by invoking turbulence with vibrations warning the pilot.

SAWTOOTH LEADING EDGE

A few aircraft have a sawtooth leading edge where, rather than being a smooth continuous surface, the leading edge juts out slightly at a point(s) determined

to be beneficial by design engineers. The purpose of the sawtooth wing is to utilize the vortex created by an inboard section of the wing to improve boundary layer flow over an outboard section. This increases lift and resistance to stall. Sawtooth wing leading edges are most common on high performance military aircraft.

WINGTIP VORTICES

Wingtip vortices are caused by the air beneath the wing, which is at the higher pressure, flowing over the wingtip and up toward the top of the wing. The end result is a spiral or vortex that trails behind the wingtip anytime lift is being produced. This vortex is also referred to as wake turbulence, and is a significant factor in determining how closely one aeroplane can follow behind another on approach to land. The wake turbulence of a large aeroplane can cause a smaller aeroplane, if it is following too closely, to be thrown out of control. Vortices from the wingtip as well as the inboard edge of the ailerons and from the horizontal stabilizer are visible on the MD-11 shown in *Figure 1-37*.

Upwash and downwash refer to the effect an airfoil has on the free airstream. Upwash is the deflection of the oncoming airstream, causing it to flow up and over the wing.

Downwash is the downward deflection of the airstream after it has passed over the wing and is leaving the trailing edge. This downward deflection is what creates the action and reaction described under lift and Newton's third law.



Figure 1-37. Vortices on an MD-11.



Figure 1-38. A winglet reduces aerodynamic drag caused by air spilling off of the wing tip.

WINGLET

A winglet is an obvious vertical upturn of the wing's tip resembling a vertical stabilizer. (*Figure 1-38*) It is an aerodynamic device designed to reduce the drag created by wing tip vortices in flight. Usually made from aluminum or composite materials, winglets can be designed to optimize performance at a desired speed. They use the flow of air from under the wing to create thrust thereby reducing induced drag. Significant fuel savings are also achieved.

HIGH SPEED FLIGHT

SPEED OF SOUND

Sound, in reference to aeroplanes and their movement through the air, is nothing more than pressure disturbances in the air. It is like dropping a rock in the water and watching the waves flow out from the center. As an aeroplane flies through the air, every point on the aeroplane that causes a disturbance creates sound energy in the form of pressure waves. These pressure waves flow away from the aeroplane at the speed of sound, which at standard day temperature of 15°C, is 340 m/s. The speed of sound in air changes with temperature, increasing as temperature increases. *Figure 1-39* shows how the speed of sound changes with altitude.

MACH NUMBER, SUBSONIC, TRANSONIC AND SUPERSONIC FLIGHT

In high speed flight and/or high altitude flight, the measurement of speed is expressed in terms of a "Mach number"—the ratio of the true airspeed of the aircraft to the speed of sound in the same atmospheric conditions. An aircraft traveling at the speed of sound is traveling at Mach 1.0.

Aircraft speed regimes are defined approximately as follows:

- Subsonic—Mach numbers below 0.75
- Transonic—Mach numbers from 0.75 to 1.20
- Supersonic—Mach numbers from 1.20 to 5.00
- Hypersonic—Mach numbers above 5.00

When an aeroplane is flying at subsonic speed, all of the air flowing around the aeroplane is at a velocity of less than the speed of sound (known as Mach 1). Keep in mind that the air accelerates when it flows over certain parts of the aeroplane, like the top of the wing, so an aeroplane flying at 223 m/s could have air over the top of the wing reach a speed of 268 m/s. How fast an aeroplane can fly and still be considered in subsonic flight varies with the design of the wing, but as a Mach number, it will typically be just over Mach 0.8.

When an aeroplane is flying at transonic speed, part of the aeroplane is experiencing subsonic airflow and part is experiencing supersonic airflow. Over the top of the wing the velocity of the air will reach Mach 1 and a shock wave will form. The shock wave forms 90 degrees to the airflow approximately halfway between the leading and trailing edge of the wing. It is known as a normal shock wave. Stability problems can be encountered during transonic flight, because the shock wave can cause the airflow to separate from the wing. The shock wave also causes the center of lift to shift aft, causing the nose to pitch down.

When an aeroplane is flying at supersonic speed, the entire aeroplane is experiencing supersonic airflow. At this speed, the shock wave which formed on top of the wing during transonic flight has moved all the

way aft and has attached itself to the wing trailing edge. Supersonic speed is from Mach 1.20 to 5.0. If an aeroplane flies faster than Mach 5, it is said to be in hypersonic flight.

SHOCK WAVE

Sound coming from an aeroplane is the result of the air being disturbed as the aeroplane moves through it, and the resulting pressure waves that radiate out from the source of the disturbance. For a slow moving aeroplane, the pressure waves travel out ahead of the aeroplane, traveling at the speed of sound. When the speed of the aeroplane reaches the speed of sound, however, the pressure waves (sound energy) cannot get away from the aeroplane. At this point the sound energy starts to pile up, initially on the top of the wing,

and eventually attaching itself to the wing leading and trailing edges. This piling up of sound energy is called a shock wave. If the shock waves reach the ground, and cross the path of a person, they will be heard as a sonic boom. **Figure 1-40A** shows a wing in slow speed flight, with many disturbances on the wing generating sound pressure waves that are radiating outward. **Figure 1-40B** is the wing of an aeroplane in supersonic flight, with the sound pressure waves piling up toward the wing leading edge.

Normal Shock Wave

When an aeroplane is in transonic flight, the shock wave that forms on top of the wing, and eventually on the bottom of the wing, is called a normal shock wave. If the leading edge of the wing is blunted, instead of being rounded or sharp, a normal shock wave will also form in front of the wing during supersonic flight. Normal shock waves form perpendicular to the airstream. The velocity of the air behind a normal shock wave is subsonic, and the static pressure and density of the air are higher. **Figure 1-41** shows a normal shock wave forming on the top of a wing.

Oblique Shock Wave

An aeroplane that is designed to fly supersonic will have very sharp edged surfaces, in order to have the least amount of drag. When the aeroplane is in supersonic flight, the sharp leading edge and trailing edge of the wing will have shock waves attach to them. These shock

Altitude in Feet	Temperature (°C)	Speed of Sound (m/s)
0	15.00	340
1 000	13.01	399
2 000	11.04	338
3 000	9.06	337
4 000	7.08	335
5 000	5.09	334
6 000	3.11	333
7 000	1.13	332
8 000	-0.85	331
9 000	-2.83	329
10 000	-4.81	328
15 000	-14.72	322
20 000	-24.62	316
25 000	-34.53	309
30 000	-44.43	303
35 000	-54.34	296
*36 089	-56.50	295
40 000	-56.50	295
45 000	-56.50	295
50 000	-56.50	295
55 000	-56.50	295
60 000	-56.50	295
65 000	-56.50	295
70 000	-56.50	295
75 000	-56.50	295
80 000	-56.50	295
85 000	-53.78	297
90 000	-49.21	300
95 000	-44.63	303
100 000	-40.06	306

*Altitude at which temperature stops decreasing

Figure 1-39. Altitude and temperature versus speed of sound.

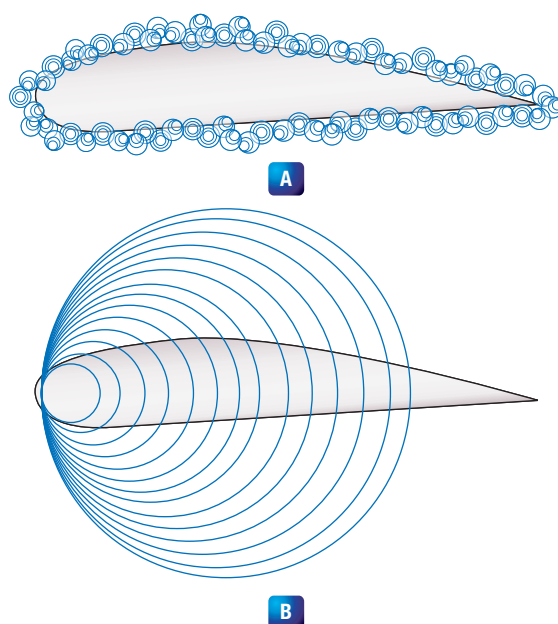


Figure 1-40. Sound energy in subsonic and supersonic flight.

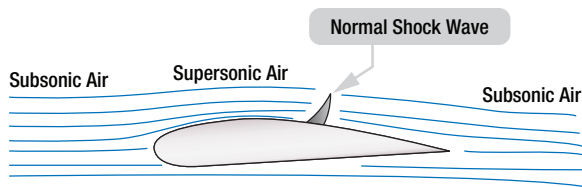


Figure 1-41. Normal shock wave.

waves are known as oblique shock waves. Behind an oblique shock wave the velocity of the air is lower, but still supersonic, and the static pressure and density are higher. **Figure 1-42** shows an oblique shock wave on the leading and trailing edges of a supersonic airfoil.

Expansion Wave

Earlier in the discussion of high speed aerodynamics, it was stated that air at supersonic speed acts like a compressible fluid. For this reason, supersonic air, when given the opportunity, wants to expand outward. When supersonic air is flowing over the top of a wing, and the wing surface turns away from the direction of flow, the air will expand and follow the new direction.

At the point where the direction of flow changes, an expansion wave will occur. Behind the expansion wave the velocity increases, and the static pressure and density decrease. An expansion wave is not a shock wave. **Figure 1-42** shows an expansion wave on a supersonic airfoil.

CRITICAL MACH NUMBER

While flights in the transonic and supersonic ranges are common occurrences for military aircraft, civilian jet aircraft normally operate in a cruise speed range of Mach 0.7 to Mach 0.90.

The speed of an aircraft in which airflow over any part of the aircraft or structure under consideration first reaches (but does not exceed) Mach 1.0 is termed "critical Mach number" or "Mach Crit." Thus, critical Mach number is the boundary between subsonic and transonic flight and is largely dependent on the wing and airfoil design. Critical Mach number is an important point in transonic flight. When shock waves form on the aircraft, airflow separation followed by buffet and aircraft control difficulties can occur. Shock waves, buffet, and airflow separation take place above critical Mach number. A jet aircraft typically is most efficient when cruising at or near its critical Mach number.

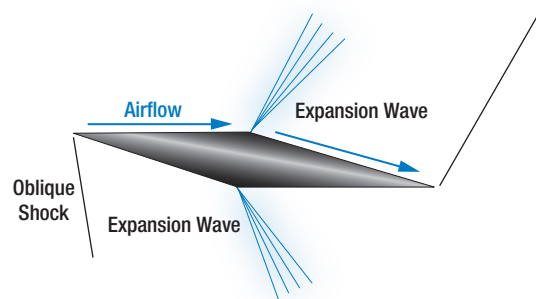


Figure 1-42. Supersonic airfoil with oblique shock waves and expansion waves.

At speeds 5-10 percent above the critical Mach number, compressibility effects begin. Drag begins to rise sharply. Associated with the "drag rise" are buffet, trim and stability changes, and a decrease in control surface effectiveness. This is the point of "drag divergence." (**Figure 1-43**)

AFFECTS OF SWEEPBACK ON CRITICAL MACH NUMBER

Most of the difficulties of transonic flight are associated with shock wave induced flow separation. Therefore, any means of delaying or alleviating the shock induced separation improves aerodynamic performance. One method is wing sweepback. Sweepback theory is based upon the concept that it is only the component of the airflow perpendicular to the leading edge of the wing that affects pressure distribution and formation of shock waves. (**Figure 1-44**)

On a straight wing aircraft, the airflow strikes the wing leading edge at 90°, and its full impact produces pressure and lift. A wing with sweepback is struck by the same airflow at an angle smaller than 90°. This airflow on the

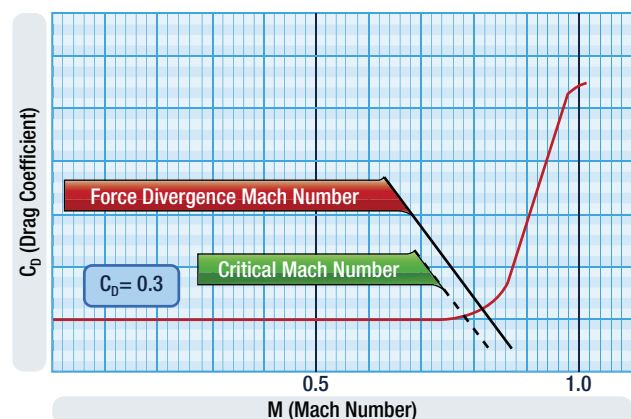


Figure 1-43. Critical Mach.

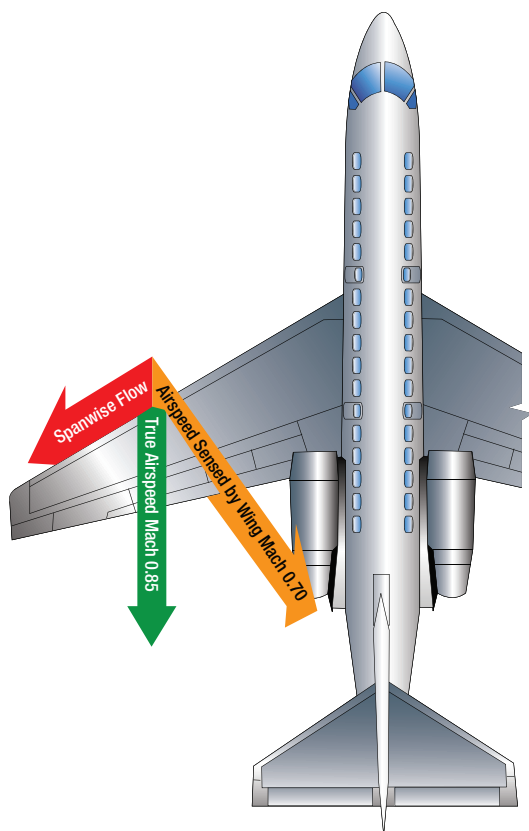


Figure 1-44. Sweepback effect.

swept wing has the effect of persuading the wing into believing that it is flying slower than it really is; thus the formation of shock waves is delayed.

Advantages of wing sweep include an increase in critical Mach number, force divergence Mach number, and the Mach number at which drag rises peaks. In other words, sweep delays the onset of compressibility effects.

The Mach number, which produces a sharp change in drag coefficient, is termed the "force divergence" Mach number and, for most airfoils, usually exceeds the critical Mach number by 5 to 10 percent. At this speed, the airflow separation induced by shock wave formation can create significant variations in the drag, lift, or pitching moment coefficients. In addition to the delay of the onset of compressibility effects, sweepback reduces the magnitude in the changes of drag, lift or moment coefficients. In other words, the use of sweepback "softens" the force divergence.

COMPRESSIBILITY BUFFET

When air is flowing at subsonic speed, it acts like an incompressible fluid. When air at subsonic speed flows through a diverging shaped passage, the velocity

decreases and the static pressure rises, but the density of the air does not change. In a converging shaped passage, subsonic air speeds up and its static pressure decreases. When supersonic air flows through a converging passage, its velocity decreases and its pressure and density both increase. (*Figure 1-45*) At supersonic flow, air acts like a compressible fluid. Because air behaves differently when flowing at supersonic velocity, aeroplanes that fly supersonic must have wings with a different shape.

As stated previously, a pressure wave builds up in front of the aircraft as it approaches Mach 1. However, some localized airflow over the wings reaches Mach 1 before the aircraft reaches this speed. Compressibility buffeting is experienced as the airflow is no longer smooth over these areas. Violent vibration can occur causing possible damage to the aircraft and control surfaces as well as a loss of control of the aircraft.

AERODYNAMIC HEATING

One of the problems with aeroplanes and high speed flight is the heat that builds up on the aeroplane's surface because of air friction. When the SR-71 Blackbird aeroplane is cruising at Mach 3.5, skin temperatures on its surface range from 230°C-540°C. To withstand this high temperature, the aeroplane was constructed of titanium alloy, instead of the traditional aluminum alloy. The supersonic transport Concorde was originally designed to cruise at Mach 2.2, but its cruise speed was reduced to Mach 2.0 because of structural problems that started to occur because of aerodynamic heating. If aeroplanes capable of hypersonic flight are going to be built in the future, one of the obstacles that will have to be overcome is the stress on the aeroplane's structure caused by heat.

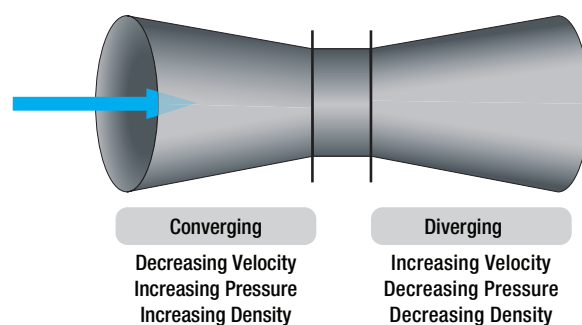


Figure 1-45. Supersonic airflow through a venturi.

AREA RULE

When designing an aircraft for transonic flight, it must be streamlined to keep drag to a minimum. Area rule is a technique for doing so. Using area rule, design engineers consider the area of successive cross section slices of the entire aircraft (not just the fuselage) and shape the total cross sectional area of each slice so that together, they produce a streamlined shape. Use of area rule reduces drag, especially where the wings and fuselage come together.

ENGINE INTAKE AIRFLOW OF HIGH SPEED AIRCRAFT

Engine intake airflow on subsonic aircraft must be kept below the critical Mach number. The shape of the engine intake is designed so that air arrives at the first stage of compression at a designed speed for maximum efficiency. This is typically around .5 Mach. A divergent duct cross section slows airflow to the intake on subsonic aircraft. A convergent duct increases intake airflow speed. On supersonic aircraft, the opposite is true. Regardless, engine intake airflow is controlled through the shape of the intake duct and duct air valves operated at particular speeds to result in the proper intake air speed.

Question: 1-1

Around what three axis do the primary flight controls move an aeroplane?

Question: 1-6

_____ and _____ are lowered to increase the camber of the wings and provide greater lift and control at slow speeds.

Question: 1-2

Movement of the _____ in flight causes the aircraft to roll.

Question: 1-7

_____ are small airfoil sections usually attached to the upper surface of a wing.

Question: 1-3

The _____ is the primary flight control that moves the aircraft around the horizontal or lateral axis.

Question: 1-8

As temperature increases, the speed of sound _____.

Question: 1-4

An _____ tab is used to maintain the speed of an aircraft since it assists in maintaining the selected pitch.

Question: 1-9

Wing sweepback is designed to delay or alleviate shock induced separation of airflow and improves aerodynamic performance.

Question: 1-5

A servo tab is similar to a balance tab in location and effect, but it is designed to operate the primary flight control surface, not just reduce the force needed to do so.

Question: 1-10

A divergent engine intake duct _____ airflow to the engine compressor inlet.

ANSWERS

Answer: 1-1

Lateral or Horizontal.
Longitudinal.
Vertical.

Answer: 1-6

Flaps, slats.

Answer: 1-2

ailerons.

Answer: 1-7

Vortex generators.

Answer: 1-3

elevator.

Answer: 1-8

increases.

Answer: 1-4

elevator.

Answer: 1-9

sweepback.

Answer: 1-5

servo tab.

Answer: 1-10

slows.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → **B1****Sub-Module 02****AIRFRAME STRUCTURES — GENERAL CONCEPTS**

Knowledge Requirements

11.2 - Airframe Structures — General Concepts

- (a) Airworthiness requirements for structural strength;
Structural classification, primary, secondary and tertiary;
Fail safe, safe life, damage tolerance concepts;
Zonal and station identification systems;
Stress, strain, bending, compression, shear, torsion, tension, hoop stress, fatigue;
Drains and ventilation provisions;
System installation provisions;
Lightning strike protection provision;
Aircraft bonding.
- (b) Construction methods of: stressed skin fuselage, formers, stringers, longerons, bulkheads, frames, doublers, struts, ties, beams, floor structures, reinforcement, methods of skinning, anti-corrosive protection, wing, empennage and engine attachments;
Structure assembly techniques: riveting, bolting, bonding;
Methods of surface protection, such as chromating, anodizing, painting;
Surface cleaning;
Airframe symmetry: methods of alignment and symmetry checks.

2

2

11.2 - AIRFRAME STRUCTURES — GENERAL CONCEPTS

AIRWORTHINESS REQUIREMENTS FOR STRUCTURAL STRENGTH

The structure of an aircraft must be strong enough to carry all the loads to which it might be subjected, including the repeated small to medium loads experienced in normal flight, and the large loads experienced in extreme conditions. To fly, the airplane the exterior must have aerodynamic shape. Into this shape must be fitted members having a high strength to weight ratio that are capable of bearing the forces necessary to balance the airplane in flight. The airplane must be capable of withstanding much more force than that imposed by its own weight. When the purpose of a particular design is established, the designers provide structure according to strict standards established by aviation authorities to ensure safety.

Requirements for airworthiness are set by EASA as well as the certifying authority in the country of manufacture if it is a non-EASA country. The goal is to only allow aircraft meeting established minimum standards to fly in an attempt to safeguard aircrews and the general public.

EASA Part 25, also known as Certification Standards 25 (CS25), states the requirement for structural airworthiness for aircraft with maximum total weight above 5 700 kg. Aircraft not meeting these standards cannot obtain a certificate of airworthiness. The standards are far reaching and specific.

Examples of structurally related topics in CS25 are those of strength, control and maintainability. Other requirements for weights, ventilation, factors of safety and even door operation are included. Specific component performance standards such as the effect of tabs and high lift device as well as stability and stall characteristics are also specified. The result is a body of airframe structural requirements to be included on all manufactured aircraft in excess of 5 700 kg.

STRUCTURAL CLASSIFICATION

Aircraft structure is divided into three categories for the purposes of assessing damage and the application of repair protocol that are suitable for the structure under consideration. Manufacturer manuals designate which category a structure falls under and the technician is required to repair and maintain that structure in

accordance with rules specified for the category under which it falls. The three categories for structure are: primary, secondary and tertiary.

PRIMARY STRUCTURE

Primary structure is any portion of the aircraft structure that, if it fails, on the ground or in flight, would likely cause any of the following:

- A loss of control of the aircraft.
- Catastrophic structural collapse.
- Injury to occupants.
- Power unit failure.
- Unintentional operation.
- Inability to operate a service.

Some examples of primary structure are wings spars, engine mounts, fuselage frames, and main floor structural members. Within the primary structure are elements called principle structural elements (PSE's). These elements are those which carry flight, ground and pressurization loads. Primary structure may also be represented as a structurally significant item or SSI. These elements are specified in a supplemental structural inspection document. Due to their structural importance, they may require special inspection and have specific repair limitations.

SECONDARY STRUCTURE

Secondary structure is all non primary structure portions of the aircraft which have integral structural importance and strength exceeding design requirements. These structures weakening without risk of failure such as those described for primary structure. Prominent examples of secondary structure are wing ribs, fuselage stringers and specified sections of the aircraft skin.

TERTIARY STRUCTURE

Tertiary structure is the remaining structure. Tertiary structures are lightly stressed structures that are fitted to the aircraft for various reasons. Fairings, fillets, various support brackets, etc. are examples of tertiary structure.

DAMAGE TOLERANT CONCEPTS

FAIL SAFE

Fail safe means the structure has been evaluated, usually by the manufacturer, to assure that catastrophic failure is

not probable after fatigue failure or obvious partial failure of a single, principal structural element. It is designed so that the aircraft may continue to operate safely until the defect is detected in a scheduled maintenance check. Manufacturer testing and fatigue analysis is used when developing fail safe structural elements. The elements are considered damage tolerant.

SAFE LIFE

Safe life structural elements are those which have a very low risk of unacceptable degradation or failure for a stated amount of time. The fatigue capability of the structure is learned through testing. The stresses applied while in service are designed to be significantly lower.

Also, the calculated time in service before failure is greatly reduced so that failure of the structure before its safe life is highly unlikely. The effects of corrosion, wear and fatigue are considered when operating under the safe life design principle.

DAMAGE TOLERANCE

Designing aircraft with fail safe principles can be somewhat unreliable. Accidents have occurred that prove this. Engineering improvements to a fail safe structure typically come with the extra penalty of adding weight. Thus, the damage tolerant concept of engineering is favored.

By distributing loads over a larger area and designing multiple load paths for carrying loads, a structure can be damage tolerant. The structure retains its integrity and the damage does not worsen in service between inspections when the damage can be detected and repaired. Thus, Damage tolerance means that the structure has been evaluated to ensure that should serious fatigue, corrosion, or accidental damage occur within the operational life of the aeroplane, the

remaining structure can withstand reasonable loads without failure or excessive structural deformation until the damage is detected.

ZONAL AND STATION IDENTIFICATION SYSTEMS

STATION NUMBERING

Even on small, light aircraft, a method of precisely locating each structural component is required. Various numbering systems are used to facilitate the location of specific wing frames, fuselage bulkheads, or any other structural members on an aircraft.

Most manufacturers use some system of station marking. For example, the nose of the aircraft may be designated "zero station," and all other stations are located at measured distances in inches behind the zero station. Thus, when a blueprint reads "fuselage frame station 137," that particular frame station can be located 137 inches behind the nose of the aircraft.

To locate structures to the right or left of the center line of an aircraft, a similar method is employed. Many manufacturers consider the center line of the aircraft to be a zero station from which measurements can be taken to the right or left to locate an airframe member. This is often used on the horizontal stabilizer and wings. The applicable manufacturer's numbering system and abbreviated designations or symbols should always be reviewed before attempting to locate a structural member. They are not always the same. The following list includes location designations typical of those used by many manufacturers.

- Fuselage stations (Fus. Sta. or FS) are numbered in inches from a reference or zero point known as the reference datum. (**Figure 2-1**) The reference datum is an imaginary vertical plane at or near the nose of

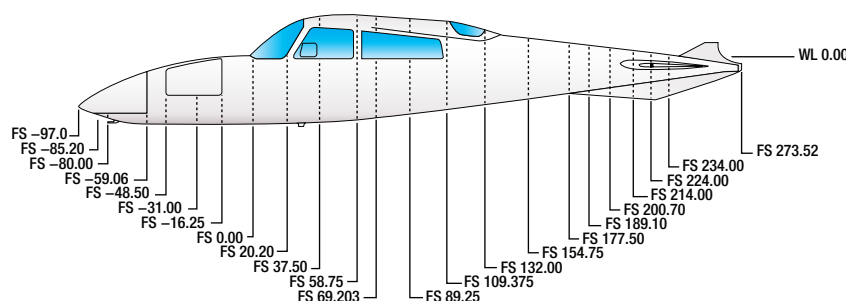


Figure 2-1. The various body stations relative to a single point of origin illustrated in inches or SI equivalent.

the aircraft from which all fore and aft distances are measured. The distance to a given point is measured in inches parallel to a center line extending through the aircraft from the nose through the center of the tail cone. Some manufacturers may call the fuselage station a body station, abbreviated BS.

- Buttock line or butt line (BL) is a vertical reference plane down the center of the aircraft from which measurements left or right can be made. (*Figure 2-2*)
- Water line (WL) is the measurement of height in inches perpendicular from a horizontal plane usually located at the ground, cabin floor, or some other easily referenced location. (*Figure 2-3*)
- Aileron station (AS) is measured outboard from, and parallel to, the inboard edge of the aileron, perpendicular to the rear beam of the wing.
- Flap station (KS) is measured perpendicular to the rear beam of the wing and parallel to, and outboard from, the inboard edge of the flap.
- Nacelle station (NC or Nac. Sta.) is measured either forward of or behind the front spar of the wing and perpendicular to a designated water line.

In addition to the location stations listed above, other measurements are used, especially on large aircraft. Thus, there may be horizontal stabilizer stations (HSS), vertical stabilizer stations (VSS) or powerplant stations (PPS). (*Figure 2-4*) In every case, the manufacturer's terminology and station location system should be consulted before locating a point on a particular aircraft.

ZONAL IDENTIFICATION SYSTEM

Another method is used to facilitate the location of aircraft components on air transport aircraft. This involves dividing the aircraft into zones. Large areas or major zones are further divided into sequentially

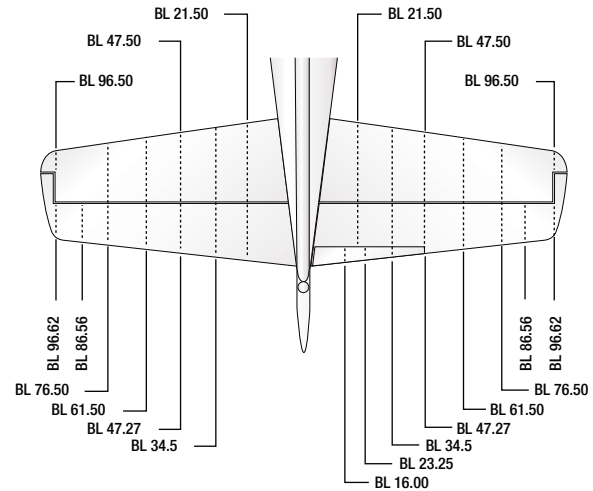


Figure 2-2. Butt line diagram of a horizontal stabilizer.

numbered zones and sub zones. The digits of the zone number are reserved and indexed to indicate the location and type of system of which the component is a part. *Figure 2-5* illustrates these zones and sub zones on a transport category aircraft.

ACCESS AND INSPECTION PANELS

Knowing where a particular structure or component is located on an aircraft needs to be combined with gaining access to that area to perform the required inspections or maintenance. To facilitate this, access and inspection panels are located on most surfaces of the aircraft. Small panels that are hinged or removable allow inspection and servicing. Large panels and doors allow components to be removed and installed, as well as human entry for maintenance purposes.

The underside of a wing, for example, sometimes contains dozens of small panels through which control cable components can be monitored and fittings greased. Various drains and jack points may also be on the underside of the wing. The upper surface of

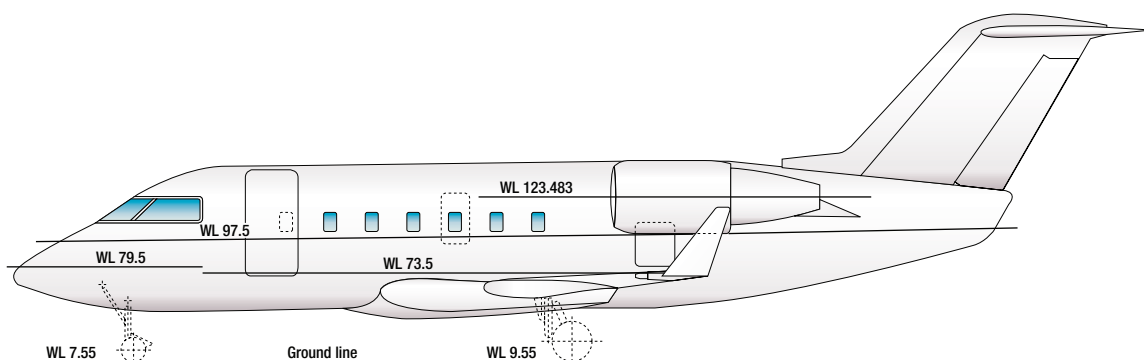


Figure 2-3. Water line diagram.

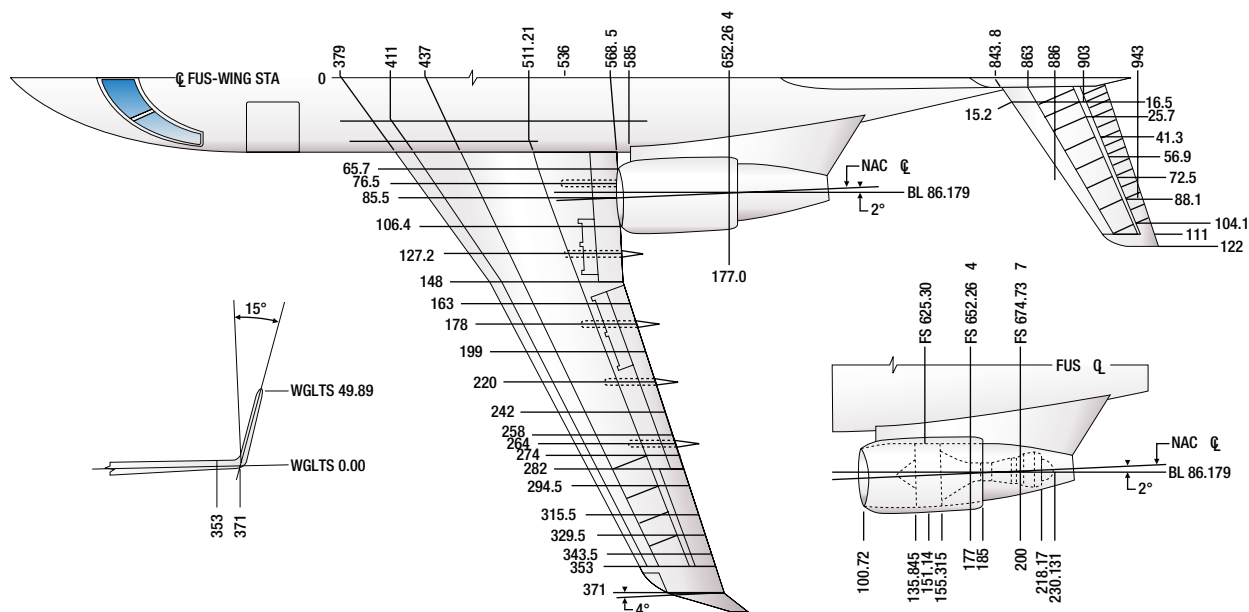


Figure 2-4. Wing stations are often referenced off the butt line, which bisects the center of the fuselage longitudinally. Horizontal stabilizer stations referenced to the butt line and engine nacelle stations are also shown.

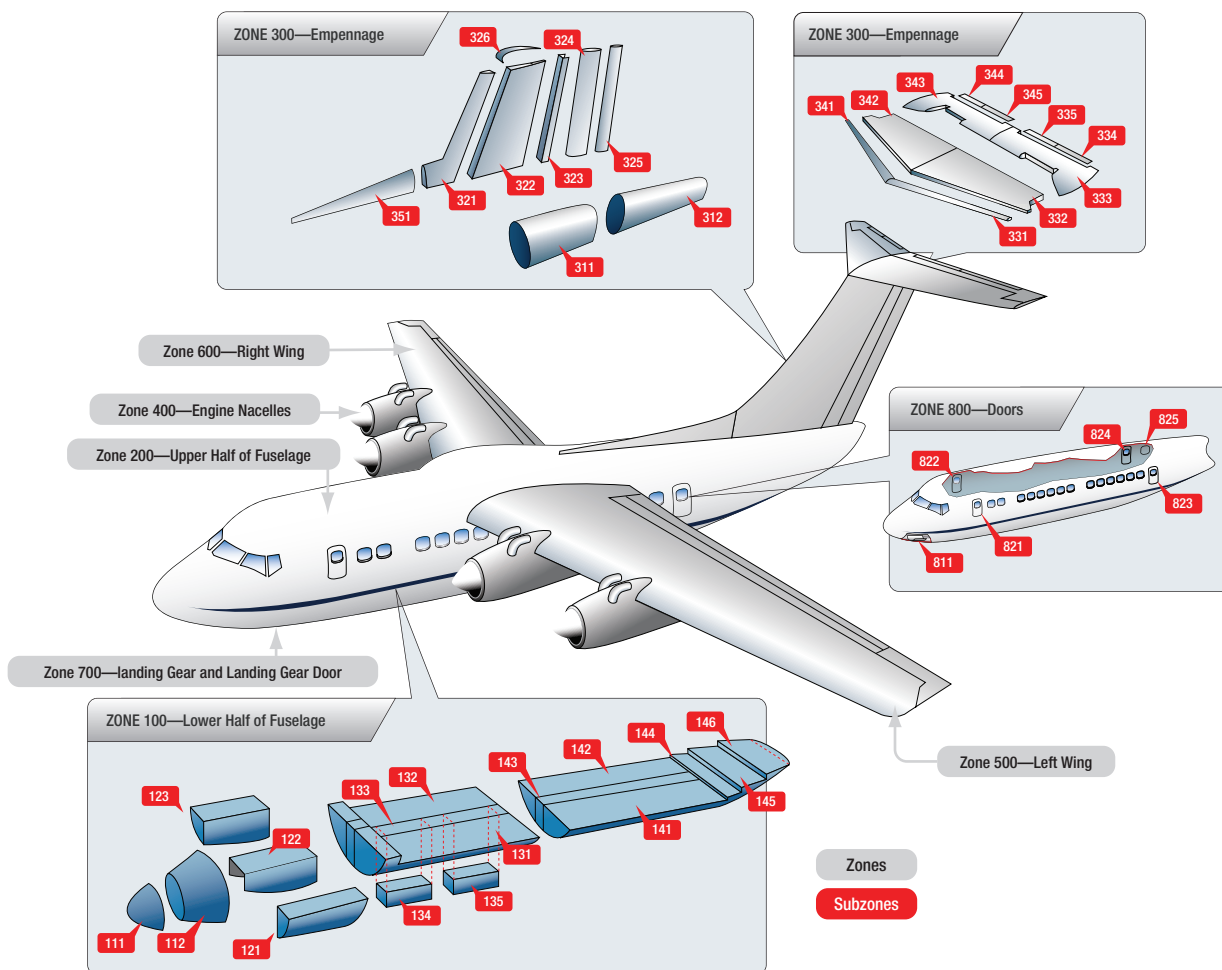


Figure 2-5. Large aircraft are divided into zones and sub-zones for identifying the location of various components.

the wings typically have fewer access panels because a smooth surface promotes better laminar airflow, which causes lift.

On large aircraft, walkways are sometimes designated on the wing upper surface to permit safe navigation by mechanics and inspectors to critical structures and components located along the wing's leading and trailing edges. Wheel wells and special component bays are places where numerous components and accessories are grouped together for easy maintenance access.

Panels and doors on aircraft are numbered for positive identification. On large aircraft, panels are usually numbered sequentially containing zone and sub zone information in the panel number. Designation for a left or right side location on the aircraft is often indicated in the panel number. This could be with an "L" or "R," or panels on one side of the aircraft could be odd numbered and the other side even numbered.

The manufacturer's maintenance manual explains the panel numbering system and often has numerous diagrams and tables showing the location of various components and under which panel they may be found. Each manufacturer is entitled to develop its own panel numbering system.

STRUCTURAL STRESSES

Aircraft structural members are designed to carry a load or to resist stress. In designing an aircraft, every square inch of wing and fuselage, every rib, spar, and even each metal fitting must be considered in relation to the physical characteristics of the material of which it is made. Every part of the aircraft must be planned to carry the load to be imposed upon it.

The determination of such loads is called stress analysis. Although planning the design is not the function of the aircraft technician, it is, nevertheless, important that the technician understand and appreciate the stresses involved in order to avoid changes in the original design through improper repairs.

The term "stress" is often used interchangeably with the word "strain." While related, they are not the same thing. External loads or forces cause stress. Stress is a material's internal resistance, or counterforce, that opposes deformation. The degree of deformation of a

material is strain. When a material is subjected to a load or force, that material is deformed, regardless of how strong the material is or how light the load is. There are five major stresses to which all aircraft are subjected: (**Figure 2-6**)

- Tension
- Compression
- Torsion
- Shear
- Bending

Tension is the stress that resists a force that tends to pull something apart. (**Figure 2-6A**) The engine pulls the aircraft forward, but air resistance tries to hold it back. The result is tension, which stretches the aircraft. The tensile strength of a material is measured in pounds per square inch (psi) and is calculated by dividing the load (in pounds) required to pull the material apart by its cross sectional area (in square inches).

Compression is the stress that resists a crushing force. (**Figure 2-6B**) The compressive strength of a material is also measured in psi. Compression is the stress that tends to shorten or squeeze aircraft parts.

Torsion is the stress that produces twisting. (**Figure 2-6C**) While moving the aircraft forward, the engine also tends to twist it to one side, but other aircraft components hold it on course. Thus, torsion is created. The torsion strength of a material is its resistance to twisting or torque.

Shear is the stress that resists the force tending to cause one layer of a material to slide over an adjacent layer. (**Figure 2-6D**) Two riveted plates in tension subject the rivets to a shearing force. Usually, the shearing strength of a material is either equal to or less than its tensile or compressive strength. Aircraft parts, especially screws, bolts, and rivets, are often subject to a shearing force.

Bending stress is a combination of compression and tension. The rod in **Figure 2-6E** has been shortened (compressed) on the inside of the bend and stretched on the outside of the bend. A single member of the structure may be subjected to a combination of stresses. In most cases, the structural members are designed to carry end loads rather than side loads. They are designed to be subjected to tension or compression rather than bending.

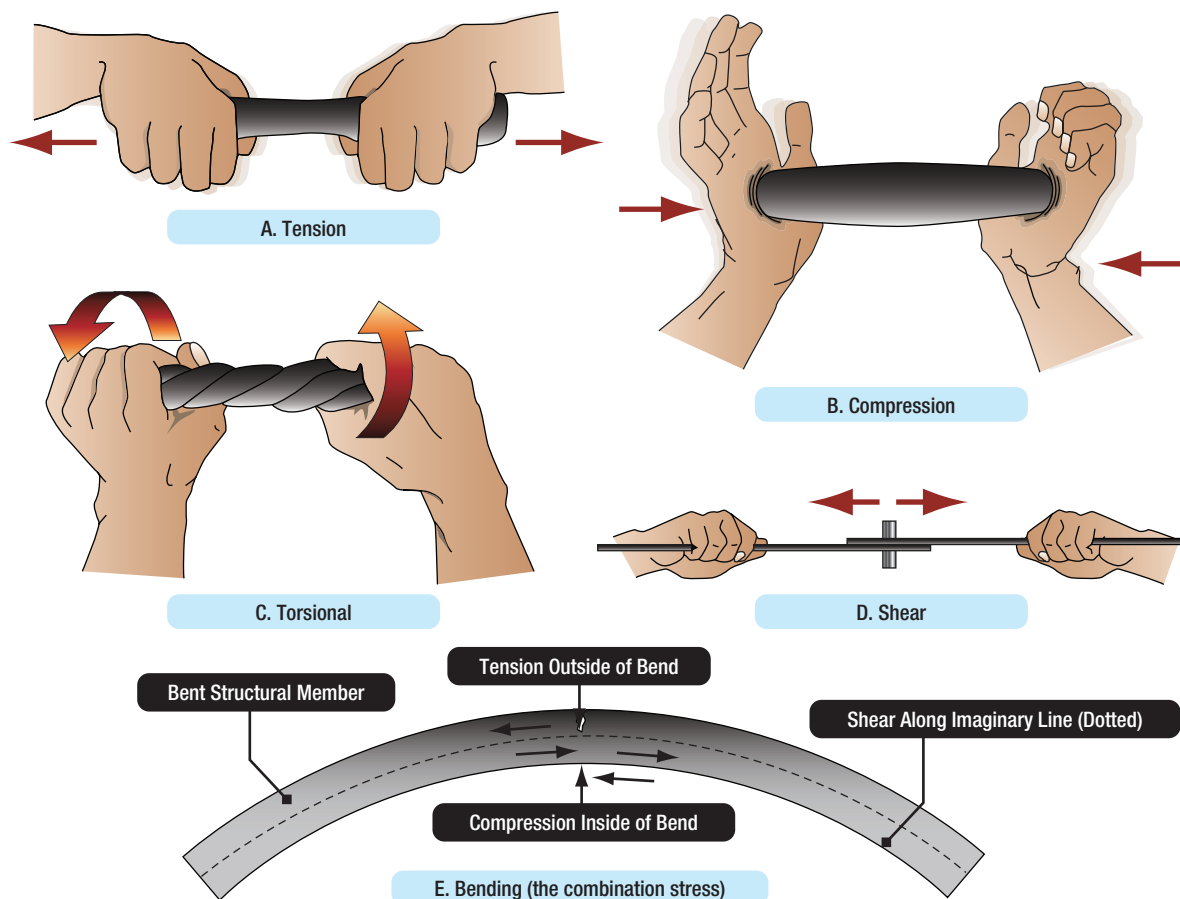


Figure 2-6. The five stresses that may act on an aircraft and its parts.

Strength or resistance to the external loads imposed during operation may be the principal requirement in certain structures. However, there are numerous other characteristics in addition to designing to control the five major stresses that engineers must consider. For example, cowling, fairings, and similar parts may not be subject to significant loads requiring a high degree of strength. However, these parts must have streamlined shapes to meet aerodynamic requirements, such as reducing drag or directing airflow.

HOOP STRESS

Hoop stress is the stress on the airframe structural components caused by pressurization. All transport category aircraft are pressurized. A circumferential load is experienced in hoop stress. The structural fuselage framework resists this load with the aid of the stressed skin.

Note that axial loads in the fuselage are also partially resisted by the stressed skin construction as well as the longitudinal structural members such as longhorns and stringers.

METAL FATIGUE

Metal fatigue is experienced by a component or structural member when a load is repeatedly applied and released or applied and reversed. This cycling weakens the material over time even though the load applied may be well below that which causes damage in a single application.

All materials have an elastic limit. If applied loads do not exceed this limit, the material should be unaffected by the load and returns to its original state when the load is removed. However, an aircraft in flight constantly experiences varying loads. Over time, these small load changes cause fatigue in the form of minute cracks in the metal structure. Each tiny, seemingly inconsequential crack exposes new material to the elements. This may weaken the material through corrosion.

Additionally, when a multitude of tiny fissures combine, larger significant cracks may develop and weaken the metal to the point of failure.

Aircraft structure is tested at the manufacturer to determine a limit not to be exceeded for an aircraft in service. Often, fatigue testing is accomplished on full scale fatigue rigs which subject the elements to cycles of loading and unloading or reversal well beyond that which will be experienced in service by the aircraft. A fatigue index is applied and the aircraft is monitored throughout its service life. If its fatigue life limit is consumed, an aircraft may be reevaluated to perceive its actual condition. If the loading cycles and environmental exposure of the structure was not as harsh as calculated, it is possible to extend the service life of the aircraft. An increase in inspection frequency and/or strengthening modification(s) may be required to do so.

Fatigue characteristics vary with the type of metal and how it is worked. The thickness of the material and type and number of fastener holes can alter the fatigue life. Aging aircraft are monitored and treated by technicians to protect against corrosion which accelerates metal fatigue.

DRAINAGE AND VENTILATION PROVISIONS

DRAINAGE

The collection of water and other fluids in the many cavities found on an aircraft can lead to corrosion and could present a fire hazard. Drainage and ventilation are used to address this issue. There are two types of drains, internal and external.

External drains have openings to the exterior of the aircraft. They are found on the wings, empennage and fuselage as well as engine nacelles. An external drain dumps the fluid overboard. In unpressurized aircraft the drains may remain open at all times. Drain valves are used in pressurized sections of aircraft so that they may remain sealed during pressurization. Typically located along the aircraft keel, some external drains use the pressurizing air to hold the valve closed. A rubber flapper type valve, a plunger type valve or a normally open spring loaded valve are closed by pressurization air. When depressurized, such as when the aircraft is on the ground, the drain valves open. Leveling compound is sometimes used to build up a low area near a drain valve to ensure that no fluid is trapped and it flows out of the drain orifice. This is typically a waterproof rubber like sealant without structural characteristics.

Some fluids accrue during flight and need to be drained. Galley and lavatory drain masts must be heated to prevent ice formation and blockage caused by cold temperatures at high altitude. A drain mast is nothing more than an airfoil shaped projection designed to guide the fluid overboard away from the skin of the aircraft. Most have electric resistance heating elements or use hot air from the pneumatic system to combat icing.

Internal drain paths are required to direct fluid to the external drain sites. Tubes, channels, dams and internal drain holes are all common. The design of structural members often includes considerations that prevent fluids from being trapped.

VENTILATION

Any cavity in the aircraft structure that may experience the presence of a flammable vapor or water must be ventilated to permit the vapor to evaporate. If necessary, vent pipes are used to provide an escape route for the vapor. Some highly susceptible areas, such as an engine nacelle, may even contain ram air inlets and exit points to enable a full flow of fresh air through the cavity. The technician should ensure that all openings designed for ventilation are unobstructed. (*Figure 2-7*)

SYSTEM INSTALLATION PROVISIONS

In addition to designing functioning support systems for operation of the aircraft, design engineers must also make the system components fit into the aircraft. Depending on the system and components, provisions for access and servicing must also be addressed. Items that receive regular maintenance such as filters, fluid level checks, bearing lubrication, etc. must be located so

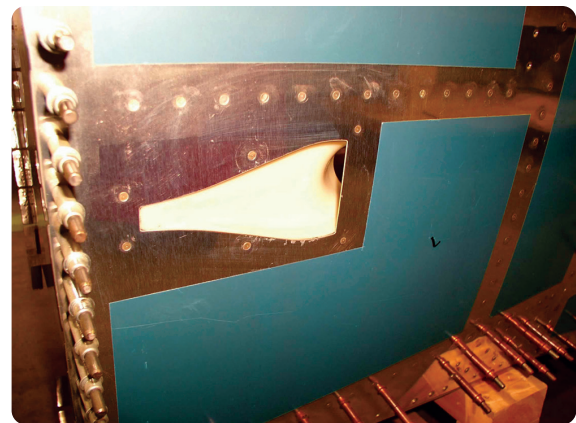


Figure 2-7. A NACA plenum ventilation scoop being built into the fuselage of a small aircraft.

that technicians can easily access them. Line replaceable units (LRU's) must be able to be quickly uninstalled and installed. Aircraft maintenance is a significant expense for the operator. Anything that can be done to locate system components for easy access for maintenance saves time and lowers the cost of operating the aircraft.

Modern airliner designers often group the components of a various systems in a single bay for easy access. Air conditioning, for example, may have its several key components mounted next to each other in an air conditioning bay. The hydraulic reservoir, pumps and filters may all be located in a different bay or in the wheel well area. Avionics and electronics are frequently mounted in an avionics bay. Not only are the "black boxes" easily accessible but environmental conditions can be better controlled than if the units were spread throughout the aircraft.

LIGHTENING STRIKE PROTECTION AND BONDING

Precautions are taken to ensure safe and continuous operation of an aircraft should it happen to be struck by lightning. A single lightning strike may contain 100 000 amperes of current. It must not be allowed to build up or arc from one point on the structure to another.

Aircraft use the predominantly aluminum structure as a ground path for operation of electrical devices. Most components are therefore mounted to structure or attached to the structure with bonding straps. This ensures that all components are at the same potential level electrically and that equal, low resistance paths for current flow exist. (*Figure 2-8*) Not only are electrical components bonded to aircraft structure but different parts of the aircraft structure are bonded together as well. Hinged flight controls, for example, have a bond strap between the movable control surface and the main airframe structure.

As an aircraft flies throughout the air, its surface can become highly charged with static electricity. Static dischargers, or wicks, are installed on aircraft to reduce radio receiver interference. This interference is caused by corona discharge emitted from the aircraft as a result of precipitation static. Corona occurs in short pulses which produce noise at the radio frequency spectrum. Static dischargers are normally mounted

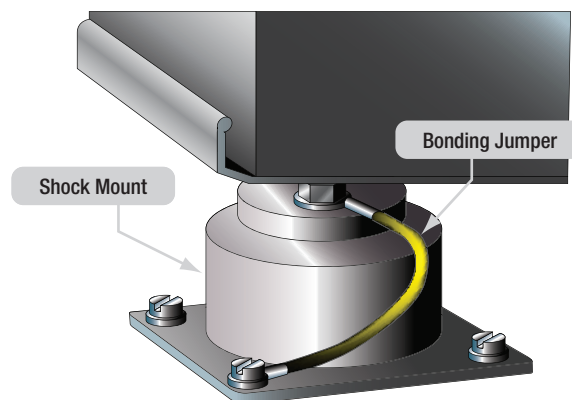


Figure 2-8. A bonding jumper is used to ground an equipment rack and avionics chassis around the non-conductive shock mount material.

on the trailing edges of the control surfaces, wing tips and the vertical stabilizer. They discharge precipitation static at points a critical distance away from avionics antennas where there is little or no coupling of the static to cause interference or noise. Flexible and semi flexible dischargers are attached to the aircraft structure by metal screws, rivets, or epoxy.

The connections should be checked periodically for security. A resistance measurement from the mount to the airframe should not exceed 0.1 ohm. Inspect the condition of all static dischargers in accordance with manufacturer's instructions. *Figure 2-9* illustrates examples of static dischargers.

When lightning strikes an aircraft with all bonding devices in tact and working, there is no difference in potential from one part of the aircraft to another. The electrical energy dissipates over the entire surface of the



Figure 2-9. Static discharger's or wicks dissipate built up static energy in flight at points a safe distance from avionics antennas to prevent radio frequency interference.

aircraft and returns to the atmosphere through the static wicks. Note that all bonding straps should be inspected periodically to ensure that no potential is allow to build so that lightning is dissipated in this manner.

Composite materials used to construct modern aircraft are not naturally conductive. To achieve the same static and lightening protection as an aluminum aircraft, conductive wires or layering of conductive material into composite components during layup ensures even distribution of electrical charges when all bonding procedures are followed.

BONDING PROCEDURES AND PRECAUTIONS

When making bonding or grounding connections in aircraft, observe these general procedures:

- Bond or ground parts to the primary aircraft structure where practical.
- Make bonding or grounding connections in such a way as to not weaken and part of the structure
- Bond parts individually wherever possible.
- Make bonding or grounding connections against smooth clean surfaces.
- Install bonding and grounding connections so that vibration, expansion or contraction, or relative movement will not break or loosen the connection.
- Locate bonding and grounding connections in protected areas when possible. When possible locate connections near hand holes, inspection doors, and other accessible areas for easy inspection and replacement.
- Do not compression fasten bonding or grounding connections through any non-metallic material
- Inspect all grounding and bonding straps to insure they are free of corrosion which will adversely affect performance.
- No more than 4 ground wires should be connected to a common ground stud. Each ground for electric power sources should be connected to separate ground points. Grounds for utilized equipment may be connected to a common point only when supplied from the same power source.

AIRFRAME STRUCTURE METHODS

FUSELAGE

The fuselage is the main structure or body of the fixed wing aircraft. It provides space for cargo, controls, accessories, passengers, and other equipment. In single engine aircraft, the fuselage houses the powerplant. In multi engine aircraft, the engines may be either in the fuselage, attached to the fuselage, or suspended from the wing structure. There are two general types of fuselage construction: truss and monocoque.

TRUSS TYPE

A truss is a rigid framework made up of members, such as beams, struts, and bars to resist deformation by applied loads. The truss framed fuselage is generally covered with fabric. The truss type fuselage frame is usually constructed of steel tubing welded together in such a manner that all members of the truss can carry both tension and compression loads. (*Figure 2-10*)

In some aircraft, principally the light, single engine models, truss fuselage frames may be constructed of aluminum alloy and may be riveted or bolted into one piece, with cross bracing achieved by using solid rods or tubes.

STRESSED SKIN MONOCOQUE TYPE

The monocoque (single shell) fuselage relies largely on the strength of the skin or covering to carry the primary loads. The design is called stressed skin and may be divided into two classes:

1. Monocoque
2. Semimonocoque

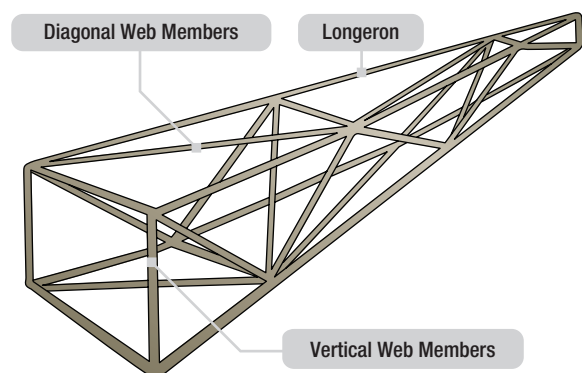


Figure 2-10. A truss-type fuselage. A Warren truss uses mostly diagonal bracing.

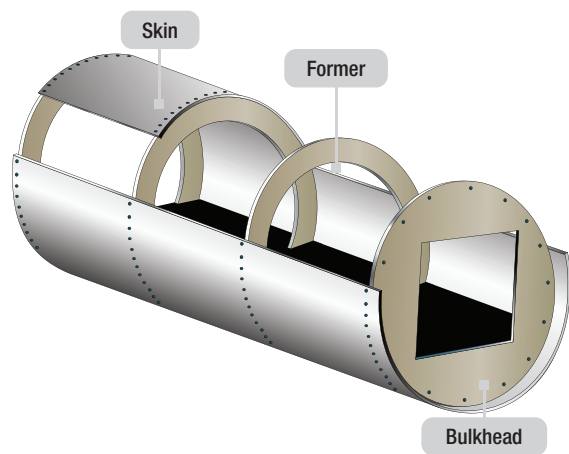


Figure 2-11. An airframe using monocoque construction.

Different portions of the same fuselage may belong to either of the two classes, but most modern aircraft are considered to be of semimonocoque type construction.

The true monocoque construction uses formers, frame assemblies, and bulkheads to give shape to the fuselage. (*Figure 2-11*) The heaviest of these structural members, bulkheads, are partition type walls that typically span the entire fuselage diameter often with an opening for access through the partition. They are located at intervals to carry concentrated loads and at points where fittings are used to attach other units such as wings, powerplants, and stabilizers. Since no other bracing members are present, the skin must carry the primary stresses and keep the fuselage rigid. Thus, the biggest problem involved in monocoque construction is maintaining enough strength while keeping the weight within allowable limits.

SEMIMONOCOQUE TYPE

To overcome the strength/weight problem of monocoque construction, a modification called semimonocoque construction was developed. It also consists of frame assemblies, bulkheads, and formers as used in the monocoque design but, additionally, the skin is reinforced by longitudinal members called longerons. Longerons usually extend across several frame members and help the skin support primary bending loads. They are typically made of aluminum alloy either of a single piece or a built up construction.

Stringers are also used in the semimonocoque fuselage. These longitudinal members are typically more numerous and lighter in weight than the longerons. They come in a variety of shapes and are usually made from single

piece aluminum alloy extrusions or formed aluminum. Stringers have some rigidity but are chiefly used for giving shape and for attachment of the skin. Stringers and longerons together prevent tension and compression from bending the fuselage. (*Figure 2-12*)

Other bracing between the longerons and stringers can also be used. Often referred to as web members, these additional support pieces may be installed vertically or diagonally. It must be noted that manufacturers use different nomenclature to describe structural members. For example, there is often little difference between some rings, frames, and formers.

One manufacturer may call the same type of brace a ring or a frame. Manufacturer instructions and specifications for a specific aircraft are the best guides.

The semimonocoque fuselage is constructed primarily of alloys of aluminum and magnesium, although steel and titanium are sometimes found in areas of high temperatures. Individually, no one of the aforementioned components is strong enough to carry the loads imposed during flight and landing. But, when combined, those components form a strong, rigid framework. This is accomplished with gussets, rivets, nuts and bolts, screws, and even friction stir welding. A gusset is a type of connection bracket that adds strength. (*Figure 2-13*)

To summarize, in semimonocoque fuselages, the strong, heavy longerons hold the bulkheads and formers, and these, in turn, hold the stringers, braces, web

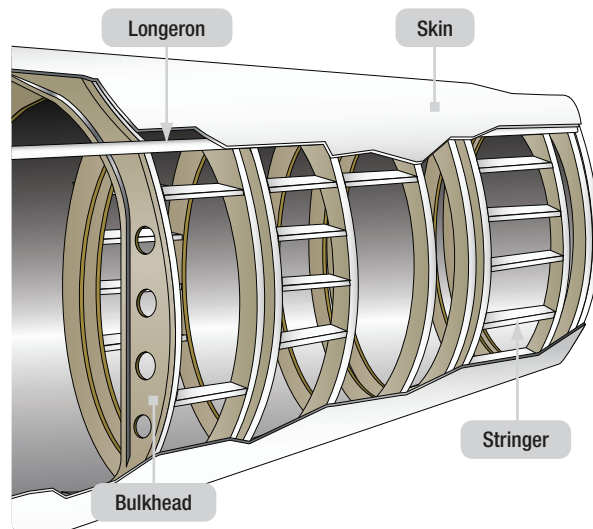


Figure 2-12. The most common airframe construction is semimonocoque.



Figure 2-13. Gussets are used to increase strength.

members, etc. All are designed to be attached together and to the skin to achieve the full strength benefits of semimonocoque design. It is important to recognize that the metal skin or covering carries part of the load. The fuselage skin thickness can vary with the load carried and the stresses sustained at a particular location.

The advantages of the semimonocoque fuselage are many. The bulkheads, frames, stringers, and longerons facilitate the design and construction of a streamlined fuselage that is both rigid and strong. Spreading loads among these structures and the stressed skin means no single piece is failure critical.

This means that a semimonocoque fuselage, because of its stressed skin construction, may withstand considerable damage and still be strong enough to hold together.

BEAMS FLOOR STRUCTURES

In addition to the structural members already mentioned, additional beams, floor structural members and various other reinforcement members are also used to construct an aircraft. A beam may be installed laterally or longitudinally. Beams typically support the floor of the flight deck and the passenger compartment. They are situated to provide secure attachment of the floor panels and also the seats tracks into which the passenger seats are secured. The floor itself is typically made up

of numerous honeycomb constructed panels that are screwed to the floor support structure. Flight deck floor panels may be constructed from sheet metal.

STRUTS AND TIES

Struts and ties are also used in aircraft structure. A strut is a bar or rod shaped reinforcement designed to resist compression loads. A tie is a rod or beam designed to take a tensile load. Both are used as needed to reinforce the aircraft structure throughout the fuselage to carry the loads experienced.

METHODS OF SKINNING

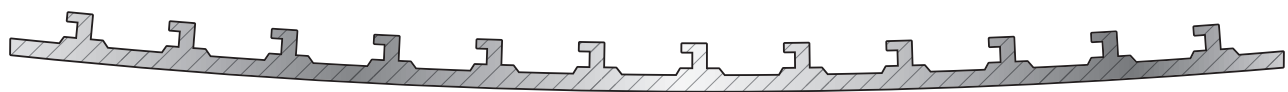
Attached to the outside of the aircraft structure is the aircraft skin, be it stressed or not. Simple, light aircraft generally have skin made from sheet aluminum which is formed to fit, wrapped and riveted to the structural members. Larger, more complex and heavier aircraft used heavier material to form the aircraft skin. This is to transfer and carry the greater loads experience during high performance flight. Some simple sheet metal skin may be found. However, various skin thickness are used to meet the design loads which vary by location around the aircraft.

Since in many areas the skin thickness varies, machining the skin, including integrally formed stringers and risers, from a solid billet of material has become a standard practice. By milling the skin out of a single piece of material, the skin thickness may be varied precisely to meet design requirements. Maximum strength is achieved with minimum weight and no excess. (*Figure 2-14*)

Another process used in skinning a large aircraft is chemical etching. Etching of thicker skin material to form thinner material with supporting raised patterns of material are produced without any stress. Skin with a "waffle plate" pattern is produced this way.

DOUBLERS

A simpler way to reinforce an area of skin on the aircraft which receives greater loads than can easily be carried



Milled Wing Skin

Figure 2-14. A milled wing skin of various thicknesses and integrated stringers is machined into to shape from a solid block of material.

by a single sheet of material is to create a doubler for that area. A doubler is simply a second, reinforcing layer of skin material used to strengthening the load carrying capacity of the skin. It has the advantage of being inexpensive and is able to be shaped for a specific area identified as needing reinforcement. Doublers are also used in sheet metal repair work.

WING, EMPENNAGE AND ENGINE ATTACHMENT

The wings, empennage and engines must be attached to the fuselage. The type of attachment varies with the aircraft design. Typically, special pins or bolts are used. Wings and empennage structure is often constructed with load carrying main members called spars. Attach lugs are securely fitted to these spars mate with lugs that

are fitted to strengthened sections of the fuselage and mounting pins or bolts are passed through both lugs and secured. **Figure 2-15** shows the internal fuselage structure of what is considered the center section of the horizontal stabilizer on a Boeing 737. Its lugs are mated with the lugs on the horizontal stabilizer front spar (each side, top and bottom) and attached with bolts.

Various wing and empennage attach methods exist including a single piece structure that passes through the fuselage making it basically non removable. Configurations where numerous smaller bolts and permanent fasteners are used to attach wings and empennage airfoils are also common. Strength and spreading the load throughout the fuselage attach structure is achieved with any of these methods.

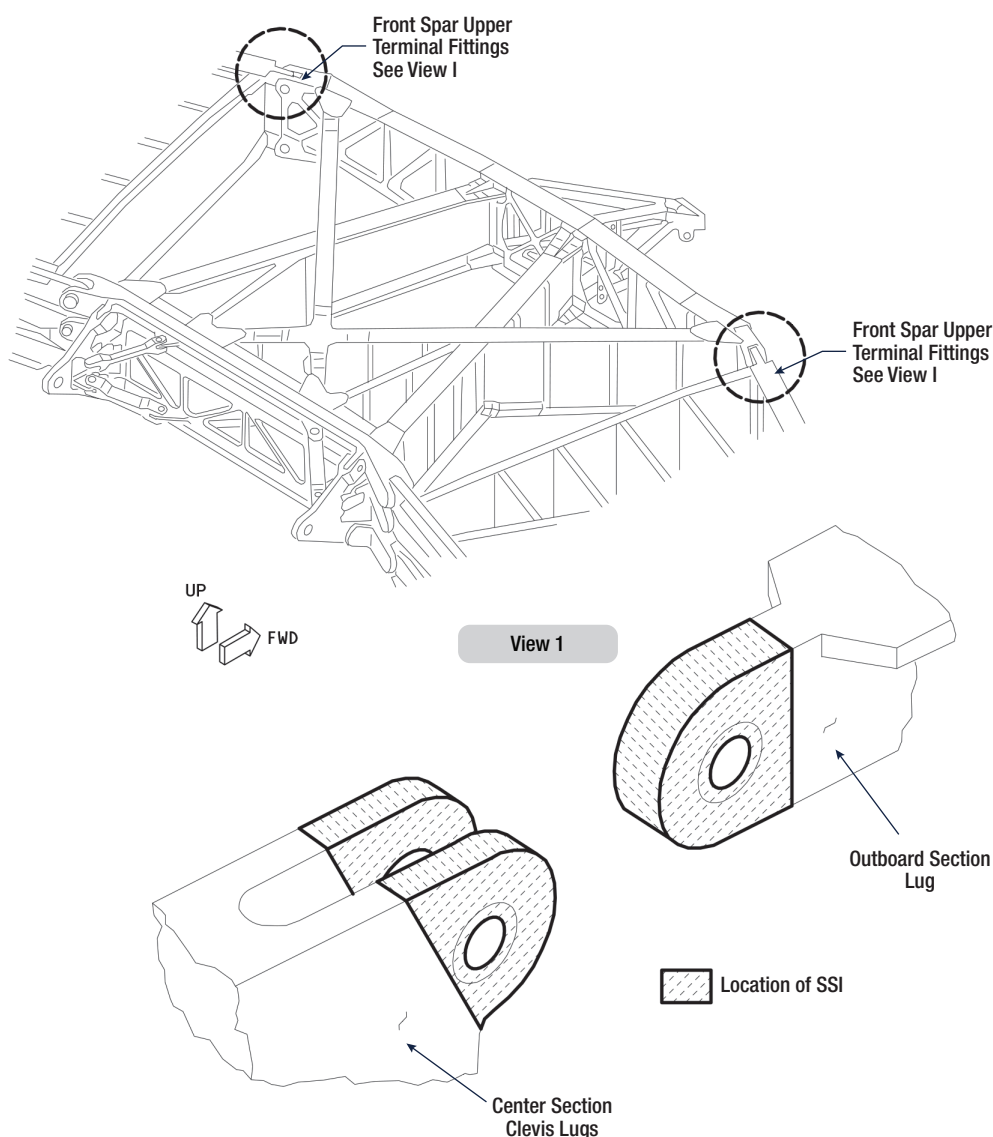


Figure 2-15. Horizontal stabilizer center section outboard front spar terminal fitting lugs.

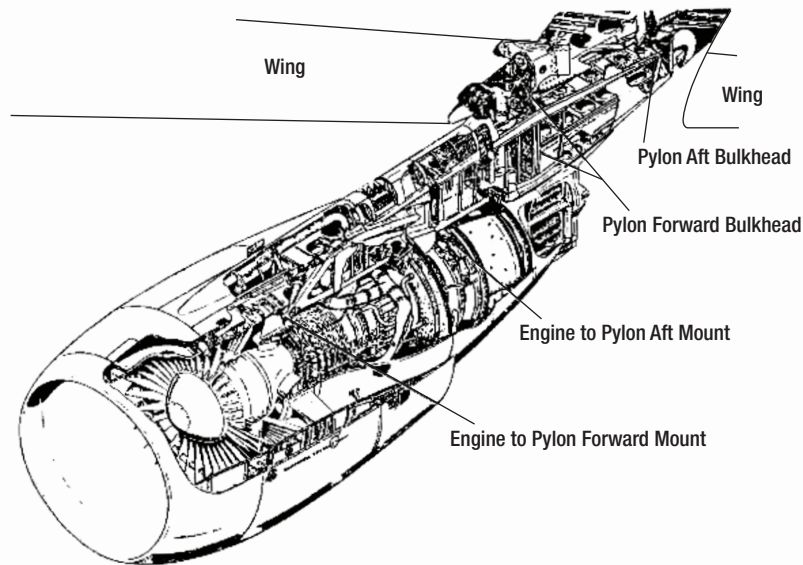


Figure 2-16. Turbofan engine pylon.

Engine attachments vary widely on aircraft depending on where the engines are located and the size and design of the aircraft and engine. A typical arrangement found on transport aircraft is to extend support structure forward and down from the wing spars. The structure is called a pylon. **Figure 2-16** is a rough cutaway drawing of a turbofan engine pylon. It is built to be very strong to support the engine. Attached to the pylon structure are engine mounts to which the engine is bolted.

The engine mounts on most turbofan engines, for example, perform the basic functions of supporting the engine and transmitting the loads imposed by the engine to the pylon and aircraft structure. Most turbine engine mounts are made of stainless steel, and are typically located as illustrated in **Figure 2-17**.

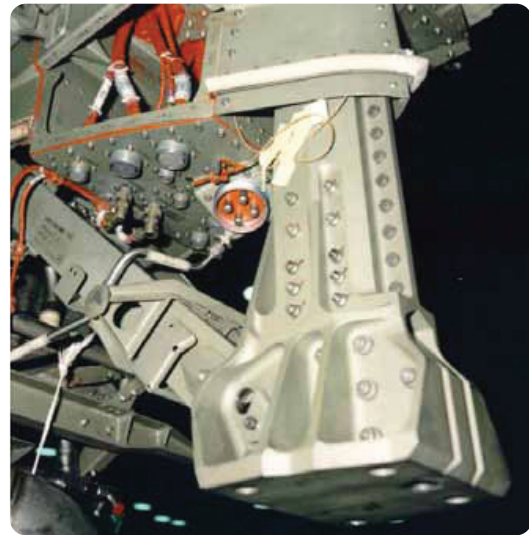


Figure 2-17. Turbine engine front mount.

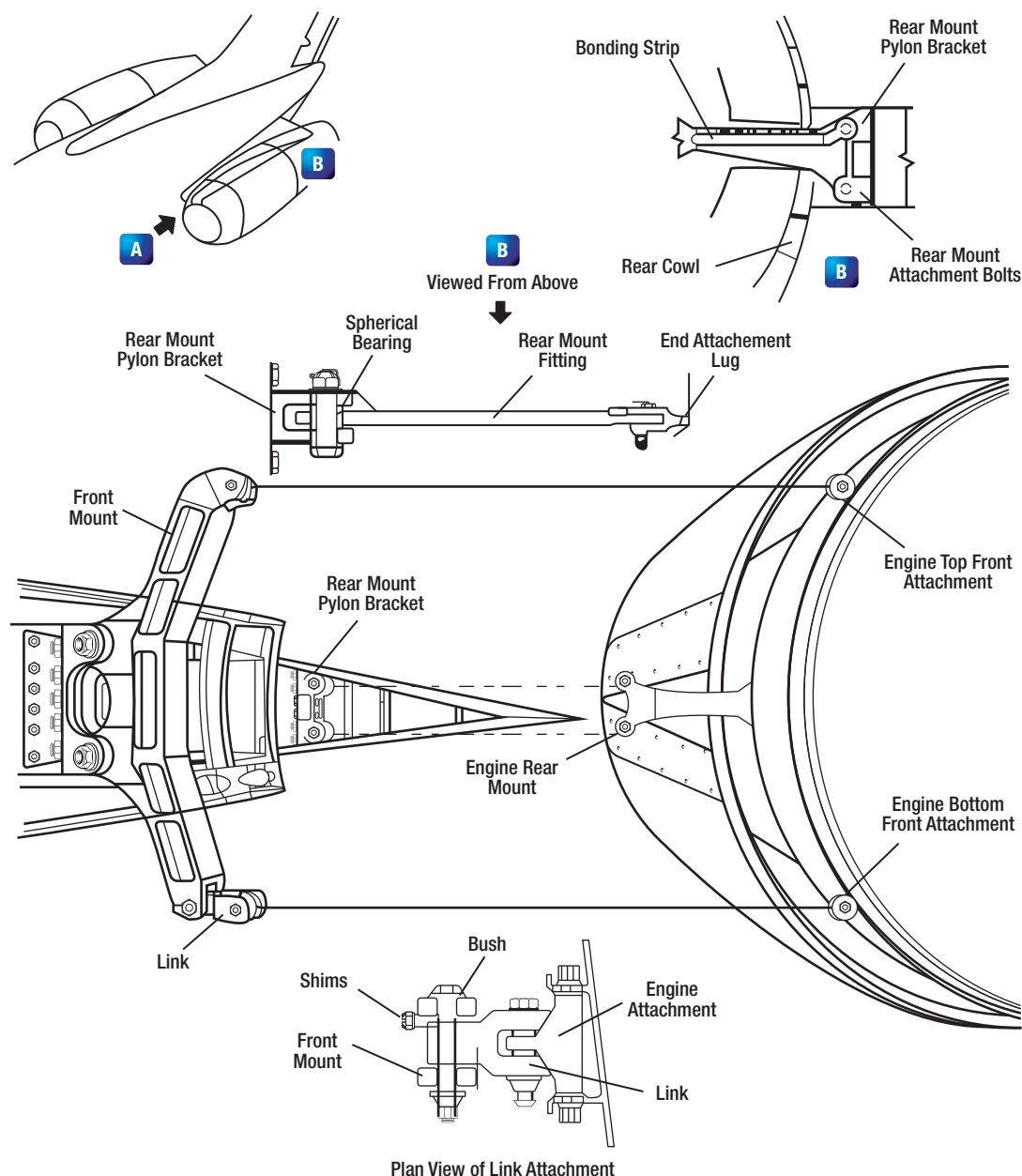
Some engine mounting systems use two mounts to support the forward end of the engine and a single mount at the rear end. The mounting arrangement depends on the position of the engine and the pylon structure. **Figure 2-18** illustrates the pylon and the side engine mount configuration for a rear engine aircraft.

STRUCTURE ASSEMBLY TECHNIQUES

The structures of the majority of today's aircraft are primarily aluminum. However, advances in the use of composite materials such as glass and carbon fiber is steadily increasing. A myriad of fasteners are used to join together aluminum structural elements.

Most common are rivets, bolts and nuts and a wide variety of special application fasteners. A full discussion of aircraft materials and hardware is found in *Module 06 – Materials and Hardware* of this series.

As early "rag and tube" aircraft construction was replaced by aluminum construction, assembly using rivets dominated assembly techniques. Light and heavy aircraft today still use the rivet as a primary fastener on structural and non structural elements. But as aircraft design evolved, larger and heavier aircraft were produced. Structural members increased in size and complexity. Rivets were not always suitable to assemble the new structure. Stronger fasteners, some designed



specifically for use in aircraft assembly, were introduced. Bolts are used in many locations on aluminum aircraft when fastening large structural members and when attaching both fixed and moveable components. Special bolts such as Hi-loks, Jo-bolts and lock-bolts are common as are clevis bolts where hi shear loads are present. Close tolerance bolts are used where a tight drive fit is required.

Special fasteners called blind fasteners are used in areas where access to only one side of an assembly is possible. A variety of blind fasteners are used including several classified as rivets. Structural sections and components

of the aeroplane that are made from composite material may be assembled and attached in a variety of ways. Sleeves and fitting incorporated during construction of a panel, for example, facilitate the use of bolts.

Other fasteners may be specified depending on the design and location of the structure. The panel itself is constructed using methods described in *Module 06 - Materials and Hardware* of this series. It is of the utmost importance to follow manufacturer's instructions when assembling composite structures. Many components are bonded or require special fasteners with specific torque considerations. Note also

that some metal structural members are bonded. Epoxy sheet bonding using autoclave curing is sometimes used to bond metal components resulting in extremely high strength joinery.

Large aircraft maintenance manuals contain specific instruction for the bonding of all materials and sections of the aircraft. ATA section 51 gives a descriptive overview of the aircraft structure and general rules followed in construction of airframe components and sections. The manufacturer's structural repair manual (SRM) details numerous repair procedures and techniques for all aircraft structure repair. A large aircraft fuselage is manufactured in sections that are then mated and fastened together.

The structural sections of a Boeing 737 are shown in *Figure 2-19*. Sections 41, 43, and 48 comprised the pressurized portion of the fuselage. Section 48 is not pressurized but does supply the support structure for the vertical and horizontal stabilizer. It also contains a bay for installation of the auxiliary power unit. A rear pressure bulkhead separates body section 46 from body section 48.

ANTI-CORROSION PROTECTION

Preventing the corrosion of aircraft structures is a consideration when materials are selected for its construction. Suitable anti-corrosion measures are then taken before and during construction. These range from heat treatment of the material to a variety of surface treatments to design and assembly techniques all designed to prevent corrosion.

Heat treatment of a metal can refine its grain structure so that it has the properties required for a specific function while reducing its susceptibility to corrosion. Surface treatments can protect metals from contaminants and moisture which cause corrosion. Plating and cladding of materials are common methods of corrosion protection. When these are designed to degrade rather than having the material they cover degrade, they are known as sacrificial coatings. Common surface treatments such as paints and primers are used as well as metal specific thin surface treatments such as anodizing and chromating. Numerous similar surface treatments have been developed for specific metals in specific applications all of which endeavor to keep the causes of corrosion at bay.

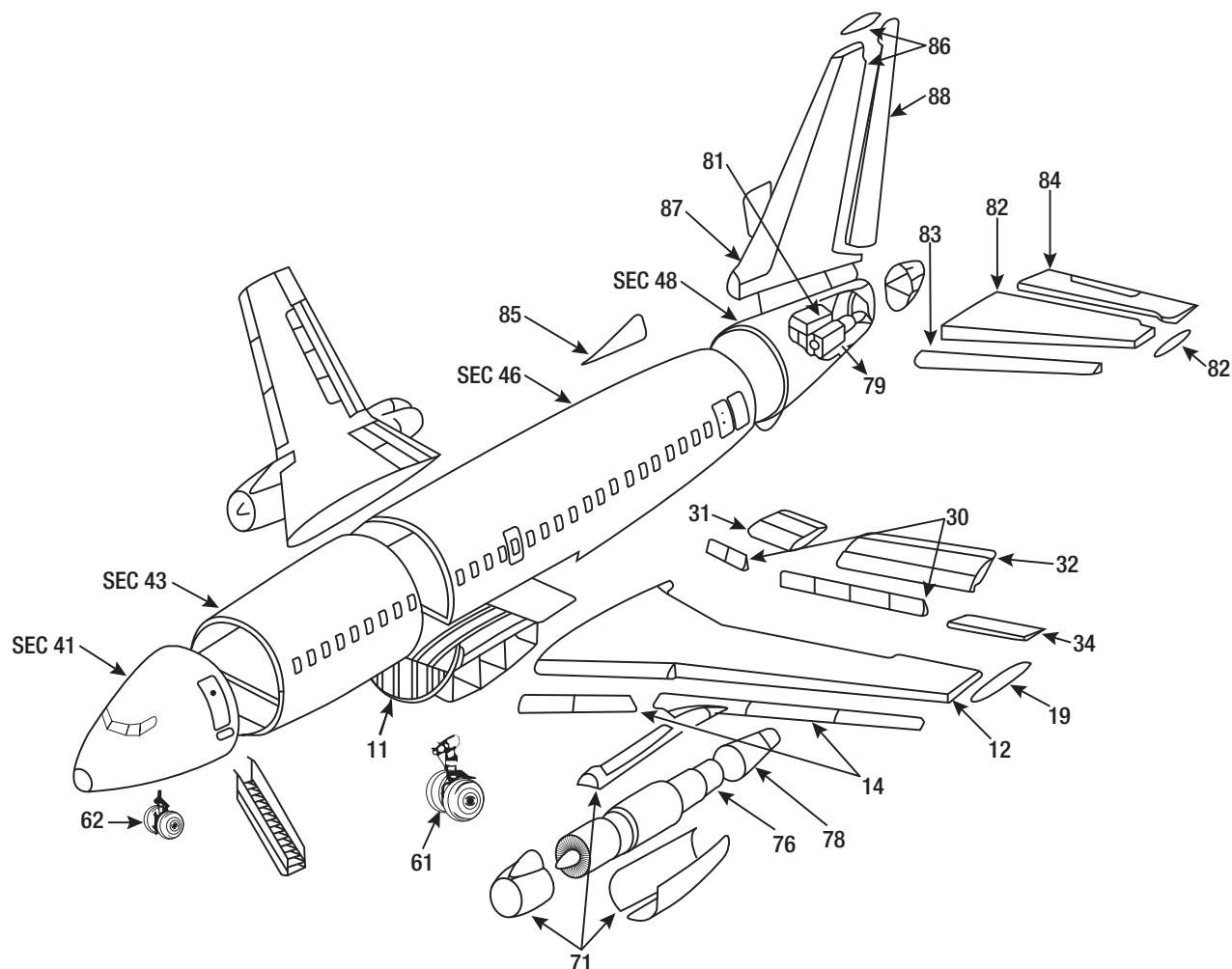
The design of an aircraft part or assembly can be very instrumental in preventing corrosion. Something as simple as a well designed drain path or a drain hole placed in a strategic location can prevent corrosion of material in a vulnerable area. Wet assembly techniques and the use of sealants also provide a barrier to corrosion causing agents.

Manufacturers use all techniques at their disposal to produce a corrosion resistant aircraft. However, varied aircraft operating environments and maintenance practices combine with service loads sustained during operation make corrosion inevitable. Processing of susceptibility data obtained from field operations is used with a wide variety of inspection and testing techniques to find and correct corrosion before it reaches a critical phase. Anti-corrosion treatments and repairs are detailed throughout the manufacturer's maintenance manuals, especially in ATA chapter 51, Structures.

Rarely does corrosion occur on a clean, dry aircraft properly treated by the manufacturer during construction. While in service, it is impossible to avoid exposure of the aircraft to the elements. The agents of corrosion, namely dirt and moisture, are encountered. A program of keeping aircraft clean and diligence to keep the condition of surface treatments in good condition are main combatants for operators when preventing corrosion. Technicians must assist by wiping up spills and removing deposits that contribute to the corrosive environment. Scratches, dents, and scoring should be avoided while performing maintenance. Drain holes must not be plugged so they can function as designed.

METHODS OF SURFACE PROTECTION

The manufacturer's maintenance manual details the surface protection compounds that must be applied by the technician for all of the various areas of the aircraft. Again, ATA Chapter 51 in the maintenance manual and the SRM should be consulted. Different areas on the aircraft may be prone to different contaminants and the recommended treatments are designed accordingly. Do not assume that a product is suitable for treatment of an area of the aircraft structure without consulting the manufacturer's data.



Struct	Title	Struct	Title
10	Wing	60	Landing Gear
11	Wing Stub	61	Main Gear
12	Wing, Outboard	62	Nose Gear
14	Slats and Flaps, L.E.	70	Powerplant
19	Wing Tip	71	Cowling
30	Spoilers	76	Thrust Reverser, Tail Pipe
31	Flap, Inboard	78	Auxiliary Power Unit
32	Flap, Outboard	79	Empennage
34	Aileron	80	Stabilizer Center Section
35	Flap, Center	81	Stabilizer
40	Body	82	Stabilizer L.E.
41	Section 41	83	Stabilizer Elevator
43	Section 43	84	Dorsal Fin
46	Section 46	85	Fin
48	Section 48	86	Fin Tip
		87	Fin L.E.
		88	Rudder

Figure 2-19. Aeroplane section numbers.

Anodizing

Manufacturers use a variety of methods of surface protection on structural metals and hardware. One of the most common for aluminum based alloys is anodizing. Anodizing is an electrolytic treatment that coats the metal with a hard, waterproof and airtight, oxide film. Anodizing usually contains a dye. Various

colors are used. This permits easy identification that a part has been anodized. The oxide film acts as an isolator. When attaching a bonding lead, the film must carefully be removed to ensure electrical conductivity.

(Figure 2-20)

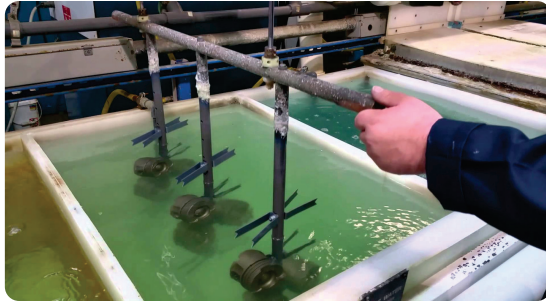


Figure 2-20. An industrial anodizing procedure.

Anodizing provides an excellent base for many finishes as well as for bonding adhesives. Acrylic lacquers, and polyurethane paints adhere well to anodized parts and provide good resistance to chemical attack and wear.

Chromating

An alternative to anodizing used for surface protection on magnesium and zinc alloy parts is chromate. When chromated, parts are generally immersed in a potassium bichromate solution. The chromate coating protects the surface from corrosive elements and has a yellowish appearance on magnesium alloys. Products are available to obtain a chromate coating on a part in the field. Alclad 1200 is one such product.

Cladding

Cladding a material with another, non corrosive material is a popular means of material surface protection. This is done as the raw material is formed into the product material. Sheet aluminum, for example, may be clad to protect the corrosive copper or zinc aluminum alloy from which many aluminum products are made. Alclad is a process of cladding aluminum in which a pure aluminum skin is rolled onto the face of an alloy aluminum sheet. Pure aluminum forms a stable aluminum oxide surface when exposed to air that protects the pure aluminum itself and the material that has been clad.

Painting

Many aircraft structural elements and parts are painted to protect them from corrosion. The paint acts as a barrier so that the agents of corrosion cannot reach the material being protected. To be effective, paint must be applied to a clean dry surface. It must be compatible with the material composition so that a good bond is formed and it adheres when it is applied. Material surface treatments such as paint primer and alodine are used before painting because they bond strongly to the base material as well as to the paint.

SURFACE CLEANING

Nearly all surface treatments to aircraft metals begin with a thorough cleaning of the material. This may include stripping of old paint before new paint or primer is applied. Strippers are specifically recommended by the manufacturer that do not react with the base metal of the structure. Therefore, only use strippers that are recommended. A cleaned surface is often treated with alodine before a primer or painted coating is applied. Clad aluminum parts use a different formula of alodine than non clad alloys. Be sure to use the correct formula.

Personal safety procedures should be followed when cleaning, stripping and applying any surface treatment. Solvents, strippers, cleaners, etchants and conversion coatings can all be hazardous to the health of the technician. Avoid breathing vapors from products of this type and avoid prolonged skin contact. Use protective gloves, goggles, respirators and other protective gear. Know the location of the nearest eyewash fountain when working with these substances. Flush eyes with water if one splashes into the eyes and get medical attention immediately. Generally, specified paint strippers are used on metal surfaces only. Protect all surrounding areas from accidental contact with the stripper. Polyethylene film and suitable adhesive tape is used for masking.

In particular, Teflon lines, self lubricated bearings, electrical terminal plugs, nylon coated wires and nylon bushings should be protected from contact with chemicals used in strippers. Plastics, laminates, composites, fiberglass and bonded structures usually have paint removed by abrasive cleaning. Do not use stripper on composite structures. Use only the methods described by the manufacturer.

EXTERIOR AIRCRAFT CLEANING

Aircraft are cleaned before major inspections. Typically a high pressure water or steam is sprayed in conjunction with cleaning agents to clean the exterior of the aircraft. While a clean aircraft aids in corrosion prevention, the cleaning process may put water and agent where it is not desirable and, thus, it may even cause corrosion. Areas into which the cleaning spray should not enter must be covered or sealed from its entrance. Pitot tubes and static ports are such areas as well as tires and brake assemblies.

The manufacturer's maintenance manual gives detailed instructions on cleaning procedures. Areas to be protected and the proper cleaning agents to use must be noted. A cleaning agent that is suitable for one area of the aircraft may not be for another. Follow all manufacturer instructions when cleaning.

Aircraft are generally washed outside in an area with adequate and environmentally responsible drainage. Washing with cleaning agents should not be performed in high temperatures where the agent may dry before being rinsed off. In certain locations, this may relegate washing to inside of a hangar. Use the ratio of agent to water that is recommended. Use of the wrong agent may cause the agent to attack materials. Hydrogen embrittlement occurs when certain agents soak into an aircraft metal. Minute cracks form and stress corrosion develops. Engine and wheel well areas may require a special washing technique or cleaning agents due to dirt, oil, grease and exhaust debris buildup. Again, follow manufacturer's instructions. Be aware that some cleaning procedures are followed by greasing various locations that may have had grease washed out during the cleaning process.

AIRFRAME SYMMETRY

The position or angle of the main structural components is related to a longitudinal datum line parallel to the aircraft center line and a lateral datum line parallel to a line joining the wing tips. Before checking the position or angle of the main components, the aircraft must be jacked and leveled.

Small aircraft usually have fixed pegs or blocks attached to the fuselage parallel to or coincident with the datum lines. A spirit level and a straight edge are rested across the pegs or blocks to check the level of the aircraft. This method of checking aircraft level also applies to many of the larger types of aircraft. However, the grid method is sometimes used on large aircraft. The grid plate is a permanent fixture installed on the aircraft floor or supporting structure. (*Figure 2-21*)

When the aircraft is to be leveled, a plumb bob is suspended from a predetermined position in the ceiling of the aircraft over the grid plate. The adjustments to the jacks necessary to level the aircraft are indicated on the grid scale. The aircraft is level when the plumb bob is suspended over the center point of the grid.

Certain precautions must be observed in all instances when jacking an aircraft. Normally, rigging and alignment checks should be performed in an enclosed hangar. If this cannot be accomplished, the aircraft should be positioned with the nose into the wind. The weight and loading of the aircraft should be exactly as described in the manufacturer's manual. In all cases, the aircraft should not be jacked until it is determined that the maximum jacking weight (if applicable) specified by the manufacturer is not exceeded.

With a few exceptions, the dihedral and incidence angles of conventional modern aircraft cannot be adjusted. Some manufacturers permit adjusting the wing angle of incidence to correct for a wing heavy condition. The dihedral and incidence angles should be checked after hard landings or after experiencing abnormal flight loads to ensure that the components are not distorted and that the angles are within the specified limits.

There are several methods for checking structural alignment and rigging angles. Special rigging boards that incorporate, or on which can be placed, a special instrument (spirit level or inclinometer) for determining the angle are used on some aircraft. On a number of aircraft, the alignment is checked using a transit and plumb bobs or a theodolite and sighting rods. The particular equipment to use is usually specified in the manufacturer's maintenance manual.

When checking alignment, a suitable sequence should be developed and followed to be certain that the checks are made at all the positions specified.

The alignment checks specified usually include:

- Wing Dihedral Angle
- Wing Incidence Angle
- Verticality of the Fin
- Engine Alignment
- A Symmetry Check
- Horizontal Stabilizer Incidence
- Horizontal Stabilizer Dihedral

CHECKING DIHEDRAL

The dihedral angle should be checked in the specified positions using the special boards provided by the aircraft manufacturer. If no such boards are available, a straight edge and a inclinometer can be used. The methods for checking dihedral are shown in *Figure 2-22*.

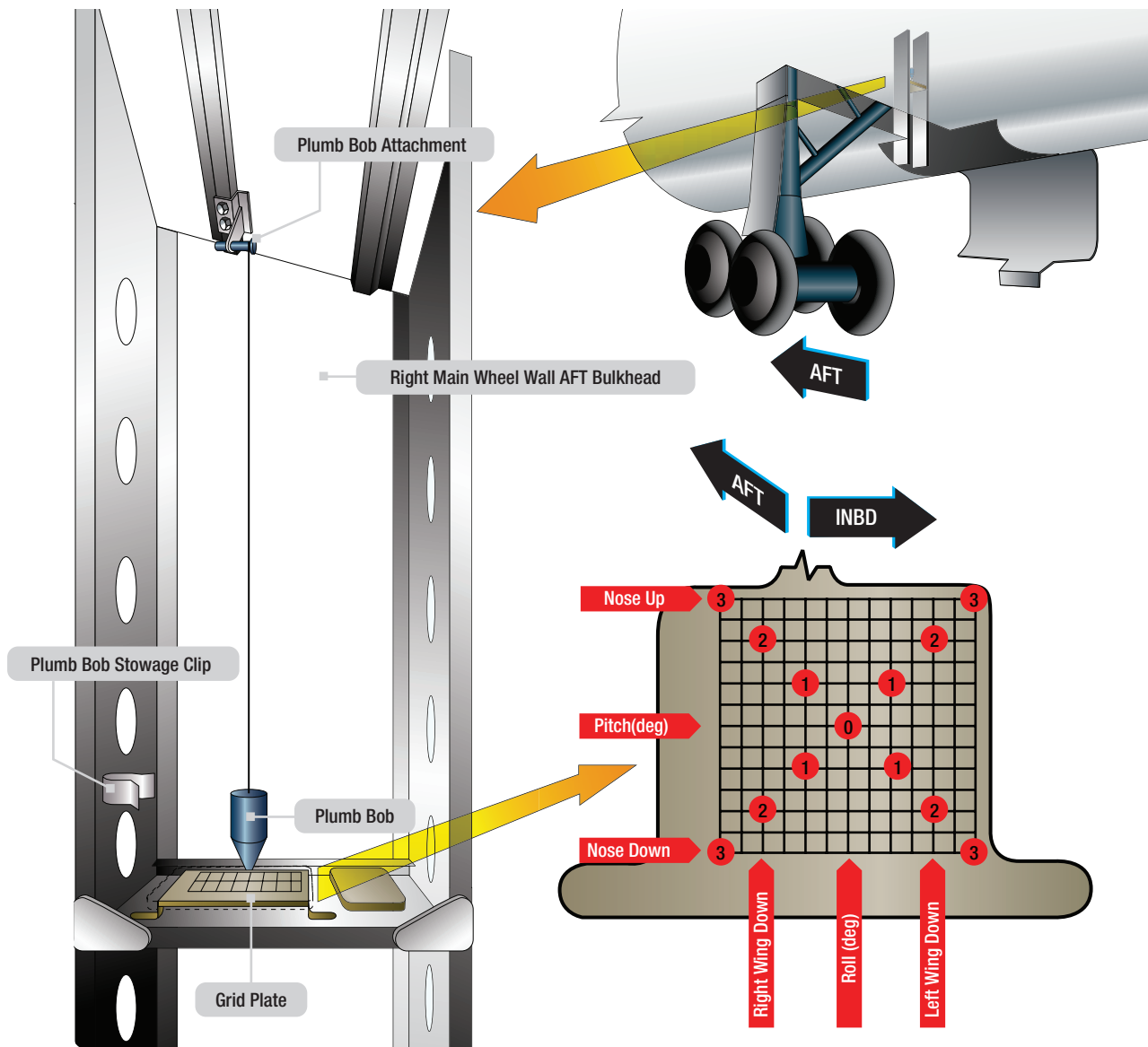


Figure 2-21. Grid plate installed.

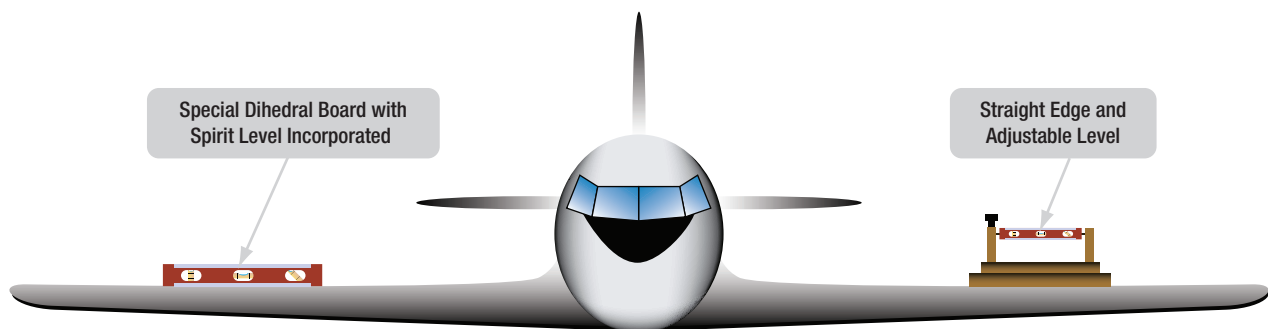


Figure 2-22. Checking dihedral.

It is important that the dihedral be checked at the positions specified by the manufacturer. Certain portions of the wings or horizontal stabilizer may sometimes be horizontal or, on rare occasions, anhedral angles may be present.

CHECKING INCIDENCE

Incidence is usually checked in at least two specified positions on the surface of the wing to ensure that the wing is free from twist. A variety of incidence boards are used to check the incidence angle. Some have stops at

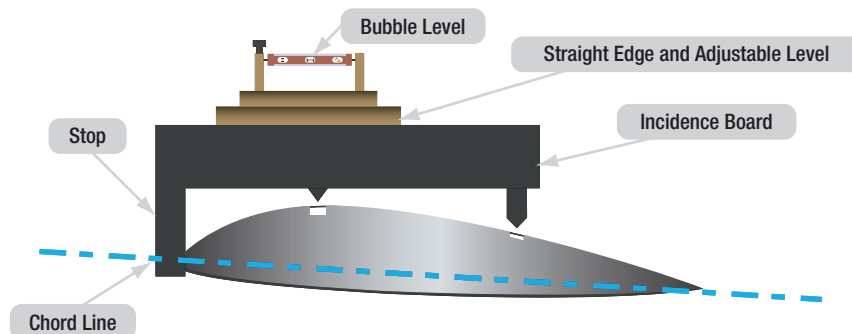


Figure 2-23. A typical incidence board.

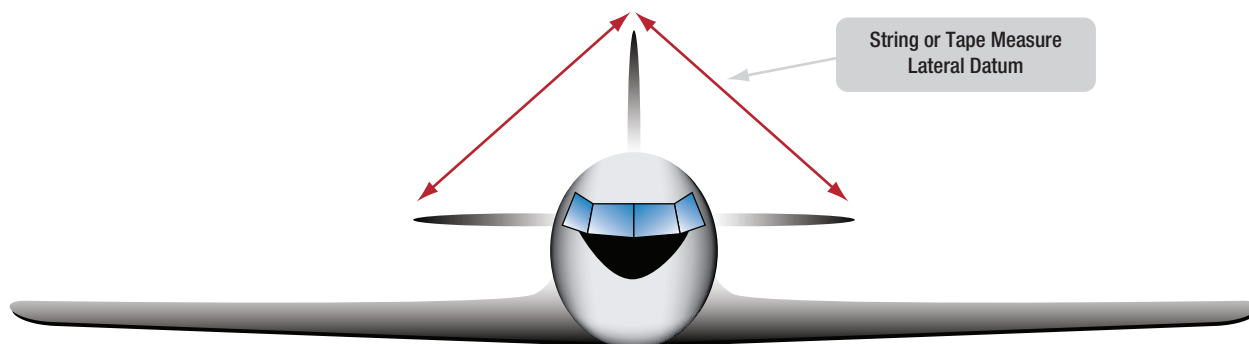


Figure 2-24. Checking fin verticality.

the forward edge, which must be placed in contact with the leading edge of the wing. Others are equipped with location pegs which fit into some specified part of the structure. The purpose in either case is to ensure that the board is fitted in exactly the position intended. In most instances, the boards are kept clear of the wing contour by short extensions attached to the board. A typical incidence board is shown in *Figure 2-23*.

When used, the board is placed at the specified locations on the surface being checked. If the incidence angle is correct, a inclinometer on top of the board reads zero, or within a specified tolerance of zero. Modifications to the areas where incidence boards are located can affect the reading. For example, if leading edge de-icer boots have been installed, the position of a board having a leading edge stop is affected.

CHECKING FIN VERTICALITY

After the rigging of the horizontal stabilizer has been checked, the verticality of the vertical stabilizer relative to the lateral datum can be checked. The measurements are taken from a given point on either side of the top of the fin to a given point on the left and right horizontal stabilizers. (*Figure 2-24*) The measurements should be similar within prescribed limits.

When it is necessary to check the alignment of the rudder hinges, remove the rudder and pass a plumb bob line through the rudder hinge attachment holes. The line should pass centrally through all the holes. It should be noted that some aircraft have the leading edge of the vertical fin offset to the longitudinal center line to counteract engine torque.

CHECKING ENGINE ALIGNMENT

Engines are usually mounted with the thrust line parallel to the horizontal longitudinal plane of symmetry. However, this is not always true when the engines are mounted on the wings. Checking to ensure that the position of the engines, including any degree of offset is correct, depends largely on the type of mounting. Generally, the check entails a measurement from the center line of the mounting to the longitudinal center line of the fuselage at the point specified in the applicable manual. (*Figure 2-25*)

SYMMETRY CHECK

The principle of a typical symmetry check is illustrated in *Figure 2-25*. The precise figures, tolerances, and checkpoints for a particular aircraft are found in the applicable service or maintenance manual.

On small aircraft, the measurements between points are usually taken using a steel tape. When measuring long distances, it is suggested that a spring scale be used with the tape to obtain equal tension. A five pound pull is usually sufficient.

On large aircraft, the positions at which the dimensions are to be taken are usually chalked on the floor. This is done by suspending a plumb bob from the checkpoints and marking the floor immediately under the point of each plumb bob. The measurements are then taken between the centers of each marking.

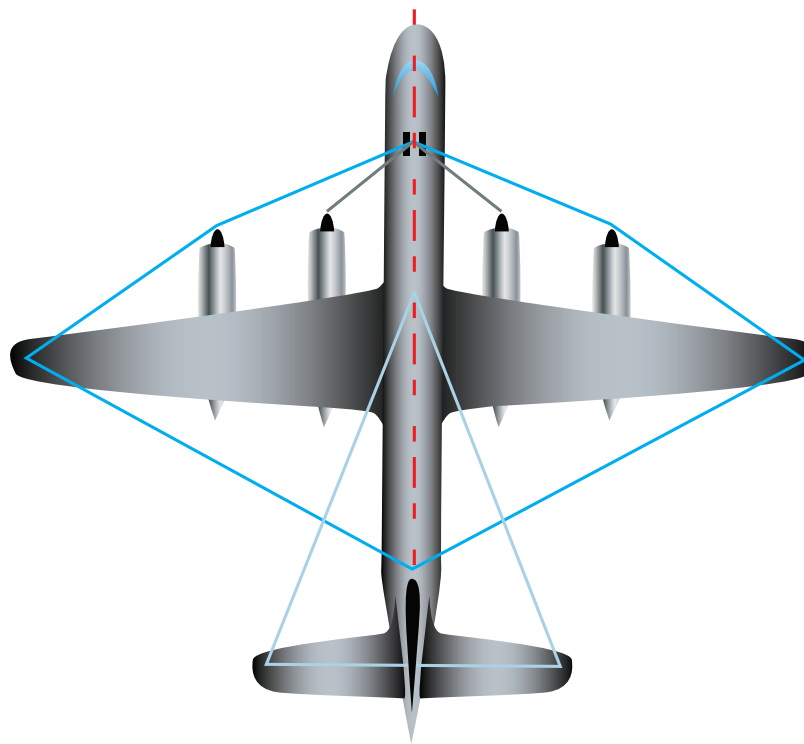


Figure 2-25. Typical measurements used to check aircraft symmetry.

Question: 2-1

If the failure of a structure causes a loss of control of the aircraft or leads to a catastrophic structural collapse, the structure is classified as _____ structure.

Question: 2-5

Wing and empennage structures often contain load carrying members called _____.

Question: 2-2

What are the 5 major stress to which all aircraft are subject?

Question: 2-6

Most turbine engine mounts are constructed of _____.

Question: 2-3

To maintain an airframe wherein all components are at the same potential level electrically, _____ is used.

Question: 2-7

A common widespread electrolytic treatment for aluminum that coats the metal with a hard oxide film is _____.

Question: 2-4

Longerons and stringers are used in _____ fuselage construction.

Question: 2-8

Airframe structural alignment checks performed after abnormal flight loads or upon hard landing measure _____ and _____ angles to insure they are as described by the manufacturer.

ANSWERS

Answer: 2-1
primary.

Answer: 2-5
spars.

Answer: 2-2
Tension.
Compression.
Torsion.
Shear.
Bending.

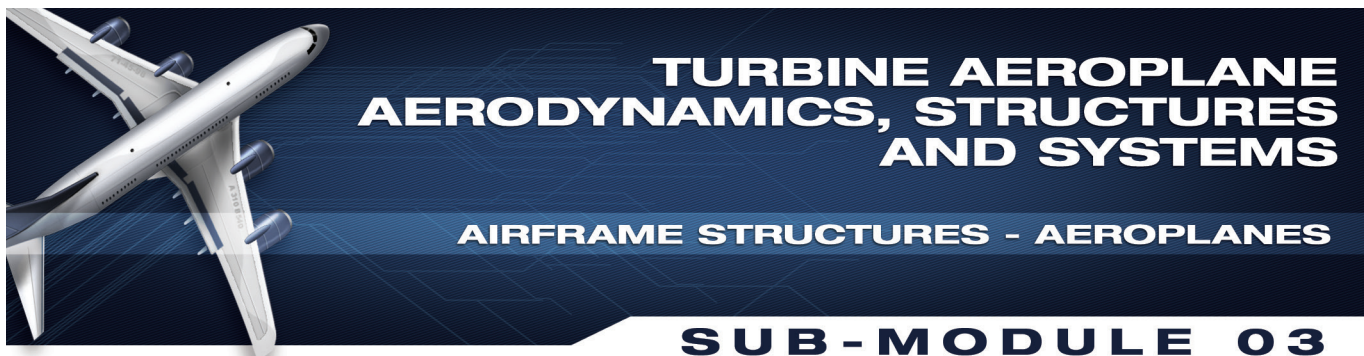
Answer: 2-6
stainless steel.

Answer: 2-3
bonding.

Answer: 2-7
anodizing.

Answer: 2-4
semimonocoque.

Answer: 2-8
dihedral.
incidence.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 03

AIRFRAME STRUCTURES — AEROPLANES

Knowledge Requirements

11.3 - Airframe Structures — Aeroplanes

11.3.1 - Fuselage (ATA 52/53/56)

Construction and pressurization sealing; Wing, stabilizer, pylon and undercarriage attachments; Seat installation and cargo loading system; Doors and emergency exits: construction, mechanisms, operation and safety devices; Windows and windscreen construction and mechanisms.

2

11.3.2 - Wings (ATA 57)

Construction; Fuel storage; Landing gear, pylon, control surface and high lift/drag attachments.

2

11.3.3 - Stabilizers (ATA 55)

Construction; Control surface attachment.

2

11.3.4 - Flight Control Surfaces (ATA 55/57)

Construction and attachment; Balancing—mass and aerodynamic.

2

11.3.5 - Nacelles/Pylons (ATA 54)

Nacelles/Pylons: Construction, Firewalls, Engine mounts.

2

AIRFRAME STRUCTURES
AEROPLANES

11.3 - AIRFRAME STRUCTURES — AEROPLANES

FUSELAGE

CONSTRUCTION

The majority of modern aircraft are constructed using a semi-monocoque construction. With this method, the skin is stressed and is supported on the inside by numerous structural elements such as bulkheads, frames, longerons and stringers. Generally, a fail safe design approach is used so that loads are spread through a variety of paths which allow partial failure without affecting the overall integrity of the aircraft structure.

Airline aircraft tend to have fuselages that are tubular with an oval cross section. The bulkheads and circumferential frames define the shape of the vessel while longerons aid in dispersing longitudinal loads. Transverse beams are used to strengthen the structure and stringers are included as a means for securely attaching the skin. Reinforced cavities are constructed onto the tubular fuselage vessel to incorporate the wings, landing gear and other equipment such as the APU. Cutouts for the doors and windows are reinforced locally to maintain proper load distribution around the openings.

PRESSURIZATION SEALING

The fuselage contains some areas that are pressurized during flight. These include the flight deck and passenger compartments as well as the baggage compartments. Pressure bulkheads are used to enclosed the pressurized areas at the fore and aft ends of the pressurized areas.

The airframe structure in pressurized areas must be sealed to prevent the flow of gases and liquids through the numerous small gaps and cracks that exist. This is done with specified sealing compounds and through the use of weatherstripping type seals. Typically, the area below the floor on a commercial passenger aircraft is unpressurized with the exception of the baggage compartments. This requires that the cabin floor be sealed as well as the wall structure and top of the fuselage circumference.

Sealing of structure is done during assembly by the manufacturer. Many fasteners are installed "wet" which means they are coated with wet sealant that flows to fill the gaps around the fastener and fastener head. Sealant may also be applied between structural members being joined. (**Figure 3-1**) Sealant specification depends upon factors such as location, resistant effects required against certain fluids and temperatures to be experienced. The manufacturer's maintenance manual details the specific sealant to be used in different areas as well as the application methods to be followed.

NOTE: Many areas of the aircraft structure are sealed for other reasons than pressurization such as wing and body sections to contain fuel and surface skin panels to provide aerodynamic efficiency. Sealant is also used in corrosion prevention.

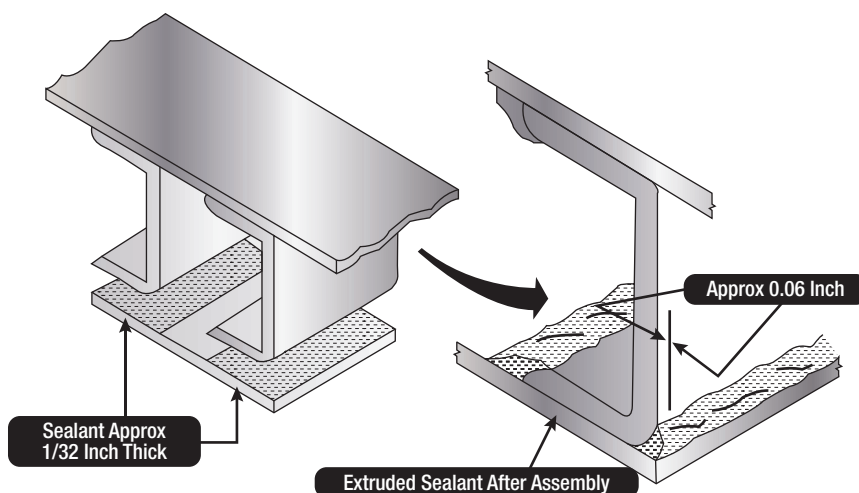


Figure 3-1. Application of sealant on a faying surface.

ATTACHMENTS

Attached to the fuselage are wings, stabilizers and in some cases, engine pylons and landing gear assemblies. Attach points vary widely in location and method. Without exception, the structure in the area of major component attachment must be reinforced to transmit loads from these attached assemblies to the fuselage.

WINGS

Wings are attached either above or below the fuselage structure. The fuselage may be notched or cut out to accept a center wing section. Strong fuselage frames or bulkheads are used in the attach area to carry the loads from the wings and to transfer the loads to the remainder of the fuselage through longhorns and beams. Fittings or lugs mounted on the frame mate with fittings or lugs mounted on the forward and aft wing spar for attachment with bolts or other fasteners.

STABILIZERS

The vertical and horizontal stabilizers are attached similar to the wings. The fuselage main frame structure is built stronger in the attach area. The vertical fin attachment is typically by bolts through fittings on a frame member or bulkhead. The attach fittings on the stabilizer are usually on the front and rear spars.

Some horizontal stabilizers are attached directly to the vertical stabilizer so no additional attach points to the fuselage are needed other than those for the vertical stabilizer. Other horizontal stabilizers are attached to the fuselage frames that are strengthened in the attach areas. Forward and rear spars of the tail plane section usually support the fittings that mates with the fuselage frame fittings.

PYLONS

Fuselage attachments may also include engine pylons. The pylon structure contains the attachments for the engine mounts. Expect heavily built up frame members in the pylon attach area. Most engine pylons attach and extend out away from the fuselage at two points.

UNDERCARRIAGE

The undercarriage of the fuselage includes the wheel wells. The wheel wells are strong, structural, box like enclosures open on the bottom. They are framed by heavy structural members and webs. A bulkhead may form the forward and/or aft wall of the wheel well. A

keel beam may be used in the wheel well to strengthen the fuselage.

In both high and low wing configurations, all or part of the gear retracts into the fuselage wheel well. The landing gear is mounted under the fuselage in many high wing aircraft. On low-wing aircraft the gear typically is attached to the wing spars and retracts into the wheel wells in the fuselage undercarriage.

On Boeing aircraft, it should be noted that the center section of the box wing structure is framed into the fuselage undercarriage along with the wheel wells. Thus, the fuselage undercarriage structure is strong and designed to transmit forces throughout the entire airframe. Longerons and stringers extend from the structure to carry loads fore and aft.

Transverse floor beams typically connect to the fuselage frame on both sides of the fuselage where wheel wells and wing sections do not exist. Upon these transverse members, longitudinal floor beams are attached. They combine with the longerons to transfer loads through the structure while at the same time support the seats and the entire fuselage floor.

Landing gear attachment fittings vary with aircraft design. On the Boeing 737, for example, the nose landing gear attachment fittings are part of the wheel well structure. The main landing gear support beam and side strut attaches to the main fuselage frame member with fittings.

SEAT INSTALLATION

On airliners, seat attachment fittings consist of longitudinal tracks attached to the tops of floor beams and seats that contain strong lightweight frames with quick-release fasteners that fit into the track at any location. (*Figure 3-2*) With this system, operators adjust seat pitch, the fore and aft space between each row of seats, to configure the aircraft with more or fewer seat rows.

Coach and first class seat frames lock into the same tracks. Typically, the seats are ganged together in the seat frame so that two, three or more seats have frame fittings that fit into two adjacent seat tracks.

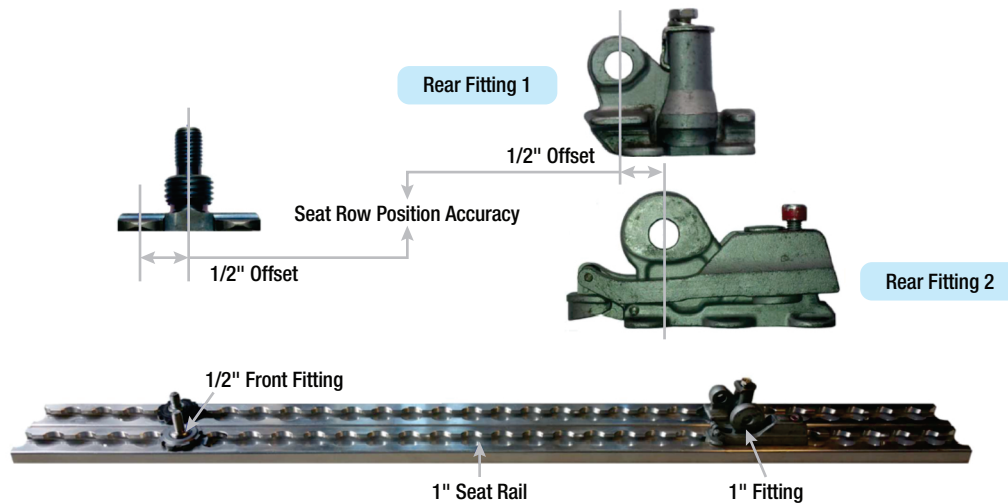


Figure 3-2. Typical aircraft adjustable seat attachment fittings.

Narrow body aircraft may have 4 longitudinal seat tracks into which two set of three seats fit in coach or two sets of two seats fit in first class. Wide body aircraft have additional tracks to accommodate three sections of seats across the width of the fuselage. The seat track fastener method may also be used to install cabin partitions including galley installations. Note that flight attendant seats are often attached this type of partition. Flight crew seats are typically attached directly to the floor structure of the flight deck.

CARGO LOADING SYSTEM

Large turbine powered aircraft are used to carry passengers, cargo, or both. Handling of cargo is often done with special cargo handling equipment such as tracks and rollers to move cargo fore and aft in the aircraft fuselage. A ball-mat used in the entry area allows cargo to be maneuvered during loading and unloading. Many automated cargo handling systems exist.

These are predominately electric powered. Roller and drive systems are manufactured for installation into nearly any type of baggage compartment or main fuselage. Some systems utilize containers that can be pre-packed. Then, the entire container is loaded into the cargo area utilizing power lifting equipment. The aircraft-installed powered cargo system allows the load to be maneuvered once in the aircraft. (*Figure 3-3*)

NOTE: The seat track type fasteners are sometimes used as quick release attachments for webbing nets used to keep cargo from shifting during flight and ground operations.



Figure 3-3. The cargo hold of an Airbus A-300.

DOORS AND EMERGENCY EXITS

There are different types of doors on an aircraft. Operating mechanisms vary depending on the type of door. Cabin entry doors, cargo bay doors, and emergency exits are in the pressurized section of the fuselage. Equipment compartment access doors not in the pressurized portion of the vessel are another type of door.

CONSTRUCTION

Aircraft doors are constructed similarly to the fuselage. Vertical and horizontal aluminum structural members are skinned on the outside and inside of the door for strength. Latching and locking mechanisms are installed within the structure. (*Figure 3-4*) On large aircraft, cabin entry doors open inward. With the help of collapsible top and bottom edges, they are then pushed through the door frame to the outside of the aircraft to permit unobstructed boarding. Newer Boeing doors

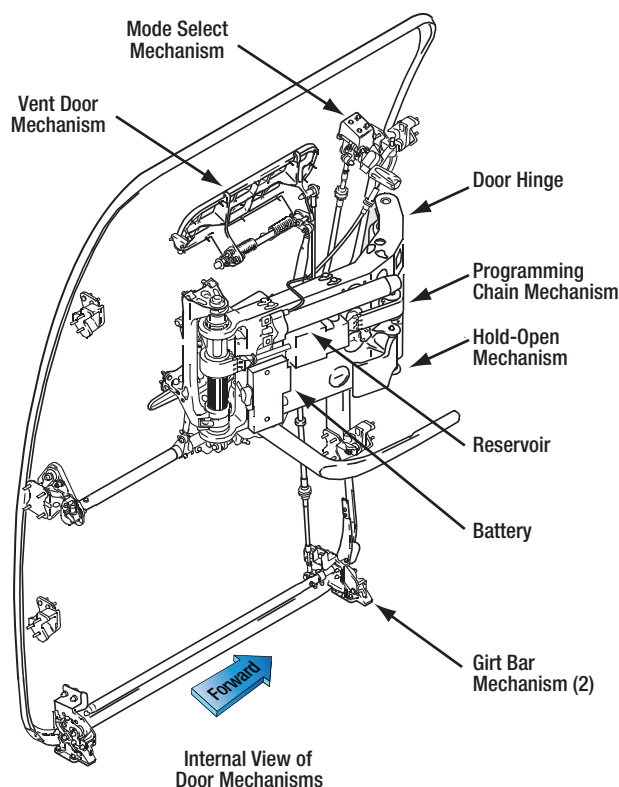


Figure 3-4. Latching mechanism of a main door.

simply move upward and open directly outward. Small cabin emergency exits (plug doors) open inward. Cargo doors typically open outward. Proximity sensors and wiring for flight deck door position status information are installed in doors. An inflatable seal around the door perimeter or other type of sealing mechanism is included on all doors.

MECHANISMS

The design of aircraft doors can be broken down into two main types: plug doors and non-plug doors. Cabin entry doors are usually plug type doors. The size of the door is slightly larger than the door opening where the door "plugs" into the fuselage from the inside; so it can't go through the opening. This type of door contacts the door frame structure and seals around the entire perimeter of the door. The loads from pressurization and flight are evenly distributed and the fit becomes more secure when pressurized. Emergency exit doors are the same.

Plug type, cabin passenger entry doors are hinged to open into the cabin because they are slightly larger than the door opening. When closing, the door moves outward and mates into the door frame. Pressurization inflates the silicon rubber door seal and further pushes

the door outward. Adjustable stops and guide rollers are mounted on the door around the perimeter face just inboard of the inflatable door seal to help align the door and make it seat properly when closed. It is adjusted to fit flush with the fuselage skin. (*Figure 3-5*)

Many cabin doors have gates at the top and/or bottom of the external skin structure. When the door mechanism is moved to open, the gates drop inward and make the door shorter. It can then be swung through the door opening and outside of the fuselage. Over wing emergency exits are small plug type doors or hatches. They are small enough to be unlatched and lifted out of the door opening by a single person. As such, there are no hinges and only a latching mechanism. The emergency exit doors are installed from inside the aircraft and can be opened from inside or outside the aircraft.

Cargo doors are non-plug type doors. They usually open outward to clear the door opening for loading and unloading. Since the cargo area is pressurized, a different means for securing and sealing the door is used than with a plug type door. For strength, large door structural members align with fuselage frame member. When the door mechanism is moved to CLOSED, locking pins inside the door extend outward from the structural members and engage the fuselage frame structure via holes around the door frame. A bell crank with connecting links operates the pins. A mechanism for latching is included. A seal around the door perimeter seals the opening.



Figure 3-5. A plug-type cabin entry door.

OPERATION AND SAFETY DEVICES

Note that the cabin entry doors and the baggage compartment doors, when closed, help strengthen the fuselage. On many aircraft, certain doors must be closed and latched in order to jack the aircraft without damaging the fuselage. Consult the manufacturers maintenance manual before towing or jacking an aircraft.

Cabin and cargo doors are required to open from both sides and have operating mechanisms incorporating an indicator for the user to know when a door is closed and latched. Cargo door are often powered in addition to being hand operated. Electric and hydraulic powered door systems exist.

A typical call warning system uses proximity switches or sensors to illuminate a warning on the flight deck when a door is not closed.

Cabin entry doors have fittings to attach an emergency inflatable slide. The gas-inflated slide stows at the bottom of the inside of the door structure. The door frame has fittings to receive a "girt bar" which activates the slide when the door is opened with the girt bar in the fittings. The girt bar is stowed on the door when operating the door normally. (*Figure 3-6*)



Figure 3-6. An emergency slide stowed on the door of an Airbus A-320.

WINDOWS AND WINDSCREEN ATTACHMENT

CONSTRUCTION

Most small and unpressurized aircraft have cabin windows constructed of a single pane of acrylic plastic or other clear plastic. The windshield is plastic as well. Pressurized, high performance and transport aircraft have two basic sets of windows: passenger cabin windows and flight deck windows. On transport aircraft, passenger cabin windows are made of acrylic plastic or similar.

There are two layers of plastic in most windows as a part of the fail safe design. A third non-structural pane mounted in the interior sidewall panel protects the middle window and reduces noise. *Figure 3-7* illustrates the typical arrangement of the passenger window panes. Cabin door windows are similar.

Flight deck windows on transport aircraft are constructed of laminations of tempered glass and plastic. The order and thickness of the lamination vary from aircraft to aircraft however, typically, the outer laminations are glass. A conductive lamination or embedded conductor is included to electrically heat the window assembly, especially if it is a forward facing window (windshield). The heated window is more resistant to impact breakage. It also anti-ices the windows. The window laminations are set in a sealed frame which is bolted into the fuselage structure. (*Figure 3-8*)

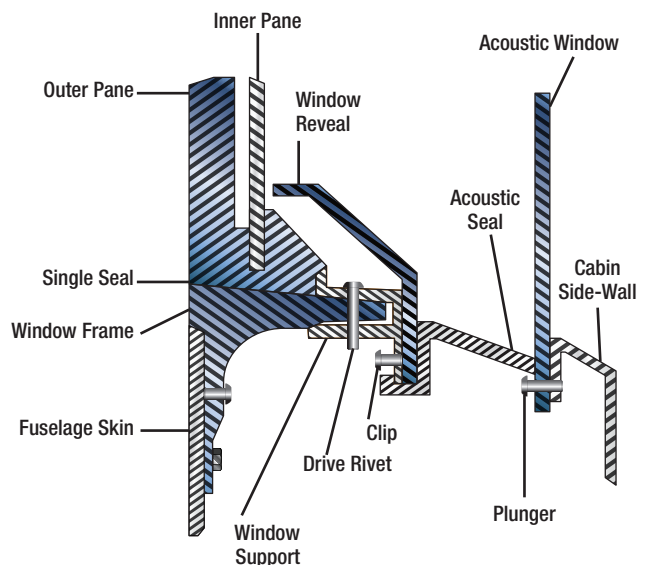


Figure 3-7. Passenger cabin window on an airliner.

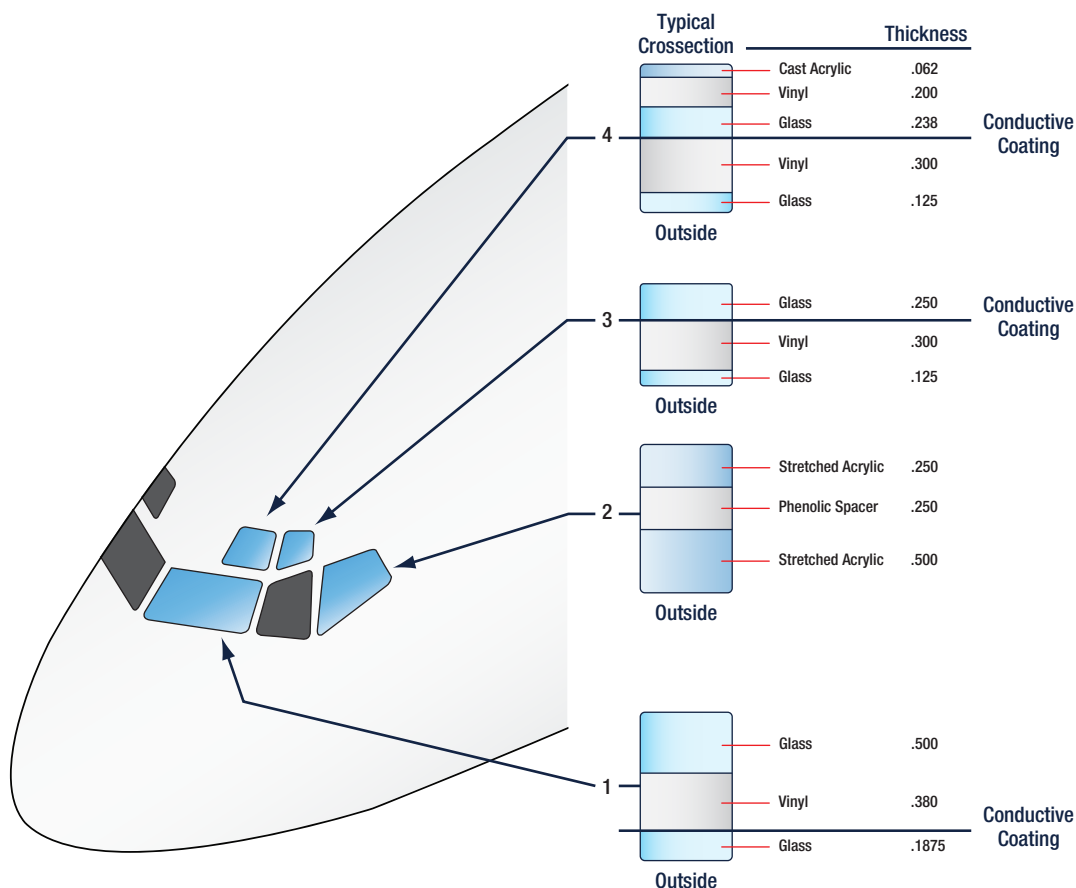


Figure 3-8. Flight deck window laminations.

Both fixed and sliding windows are used on the flight deck. The sliding windows are located on the side of the control cabin and are used as emergency exits in addition to typical window usage during ground and flight operations. A pressurization switch prevents the windows from opening during flight. Windscreens are used in passenger transport aircraft to block weather from entering the cabin when an entry or service door is opened. They are strong, lightweight partitions of honeycomb construction. Some cabin windscreens incorporate storage compartments. (Figure 3-9)

ATTACHMENT

Passenger cabin windows are fixed plug type windows. They are installed from inside the aircraft between fuselage frame members with a single seal that accepts both the middle and outer window panes. (Figure 3-10)

Retaining clips hold the assembly in place against the window frame in the fuselage skin. The middle and outer window panes are each able to withstand the forces of pressurization so if one breaks or is damaged, cabin pressurization is not lost. As stated, flight deck

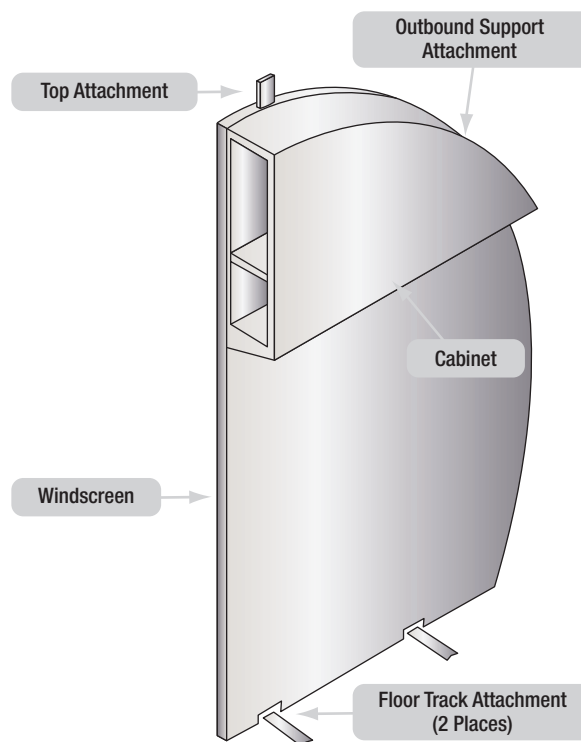


Figure 3-9. A typical windscreen in an airliner cabin.

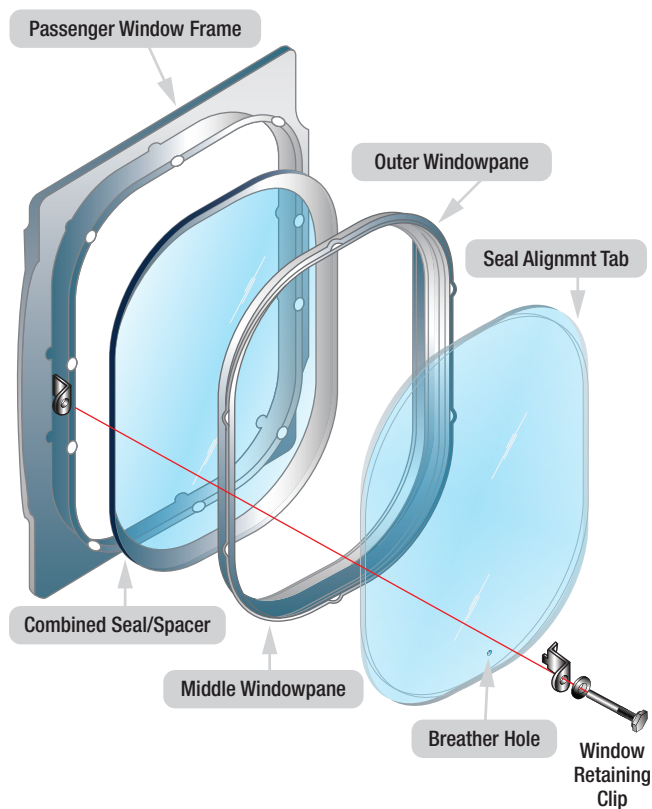


Figure 3-10. Middle and outer passenger cabin windows, seal and window frame.

window assemblies are bolted to the fuselage structure. Windscreens use the longitudinal seat track installed on the cabin floor for attachment. Support is also given by attachment to a ceiling or wall bracket.

WINGS

CONSTRUCTION

The wings of an aircraft are designed to lift it into the air. Their particular design for any given aircraft depends on a number of factors, such as size, weight, use of the aircraft, desired speed in flight and at landing,

and desired rate of climb. The wings of aircraft are designated left and right, corresponding to the left and right sides of the operator when seated in the cockpit. (*Figure 3-11*)

Most often, wings are of full cantilever design. This means they are built so that no external bracing is needed. They are supported internally by structural members assisted by the skin of the aircraft. Some smaller and special purpose aircraft wings use external struts or wires to assist in supporting the wing and carrying the aerodynamic and landing loads.

Wing support cables and struts are generally made from steel. Many struts and their attach fittings have fairings to reduce drag. Short, nearly vertical supports called jury struts are found on struts that attach to the wings a great distance from the fuselage. This serves to subdue strut movement and oscillation caused by the air flowing around the strut in flight. *Figure 3-12* shows samples of wings using external bracing, also known as semi-cantilever wings. Cantilever wings built with no external bracing are also shown.

Aluminum is the common material from which to construct wings and it is the most common material for wing construction on transport aircraft. Modern aircraft are tending toward lighter and stronger materials throughout the airframe and in wing construction. Wings made entirely of carbon fiber or other composite materials exist, as well as wings made of a combination of materials for maximum strength to weight performance.

The internal structures of most wings are made up of spars and stringers running span-wise and ribs and formers or bulkheads running chord-wise (leading edge to trailing edge). The spars are the principle structural

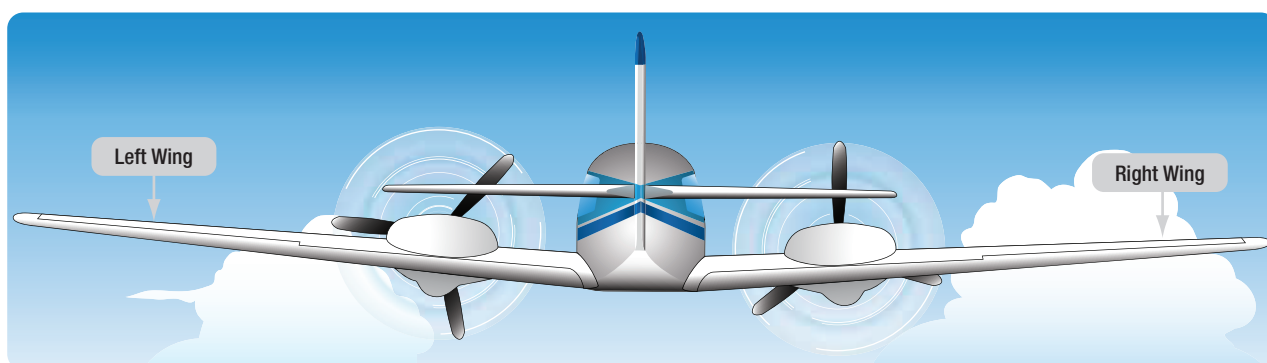


Figure 3-11. “Left” and “right” on an aircraft are oriented to the perspective of a pilot sitting in the cockpit.

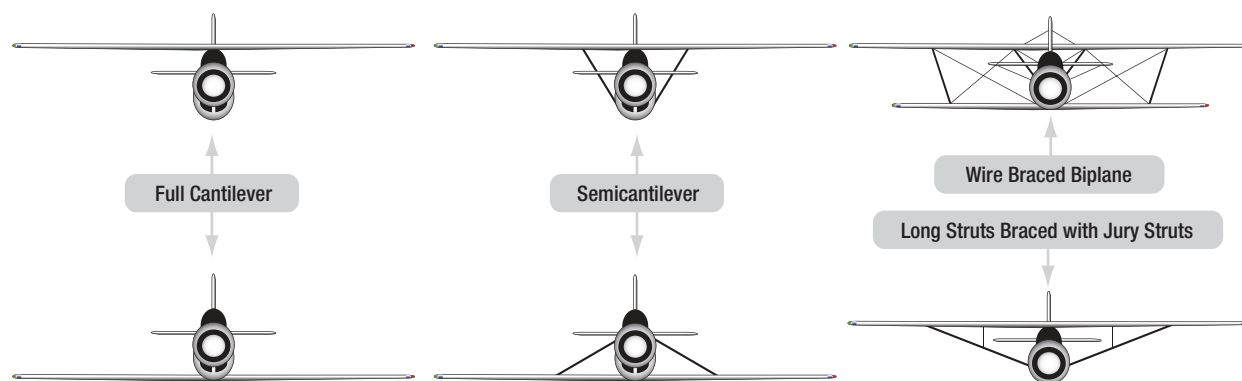


Figure 3-12. Externally braced wings, also called semi-cantilever wings, have wires or struts to support the wing. Full cantilever wings have no external bracing and are supported internally.

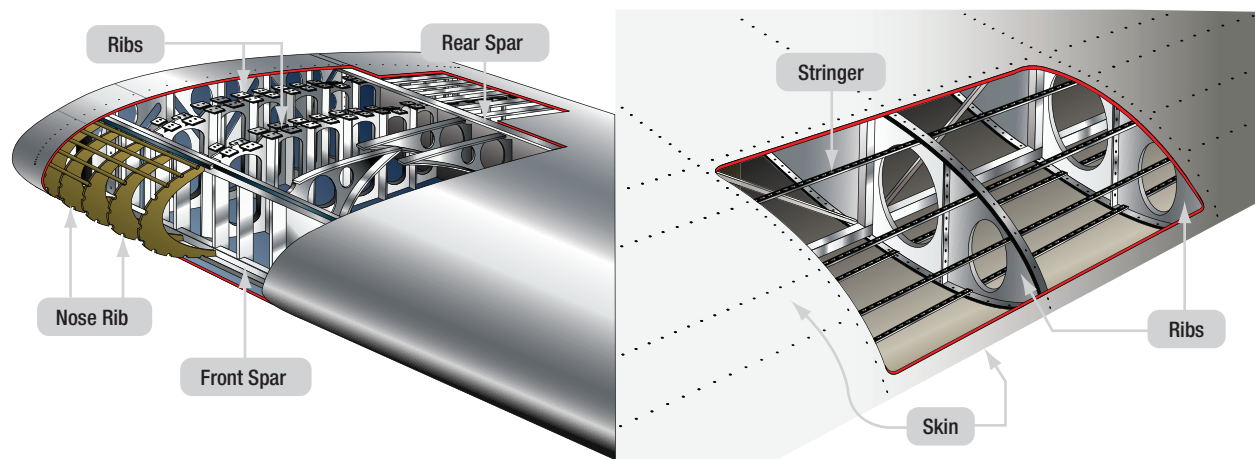


Figure 3-13. Wing structure nomenclature.

members of a wing. They support all distributed loads, as well as concentrated weights such as the fuselage, landing gear, and engines. The skin, which is attached to the wing structure, carries part of the loads imposed during flight. It also transfers the stresses to the wing ribs. The ribs, in turn, transfer the loads to the wing spars. (**Figure 3-13**) In general, wing construction is based on one of three fundamental designs:

1. Monospar
2. Multispar
3. Box beam

Modification of these basic designs may be adopted by various manufacturers. The monospar wing incorporates only one main span-wise or longitudinal member in its construction. Ribs or bulkheads supply the necessary contour or shape to the airfoil.

Although the strict monospar wing is not common, this type of design modified by the addition of false spars or light shear webs along the trailing edge for support of

control surfaces is sometimes used. The multi-spar wing incorporates more than one main longitudinal member in its construction. To give the wing contour, ribs or bulkheads are usually included. Air transport category aircraft often utilize box beam wing construction.

The box beam type of wing construction uses two main longitudinal members with connecting bulkheads to furnish additional strength and to give contour to the wing. (**Figure 3-14**) A corrugated sheet may be placed

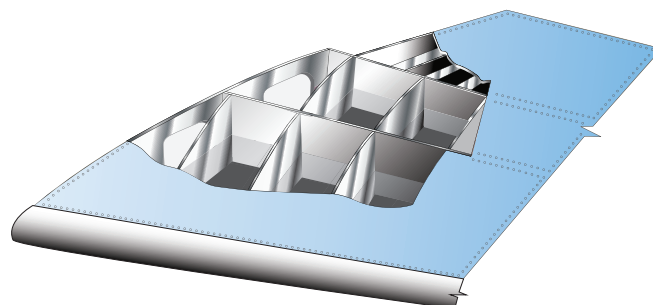


Figure 3-14. Box beam construction.

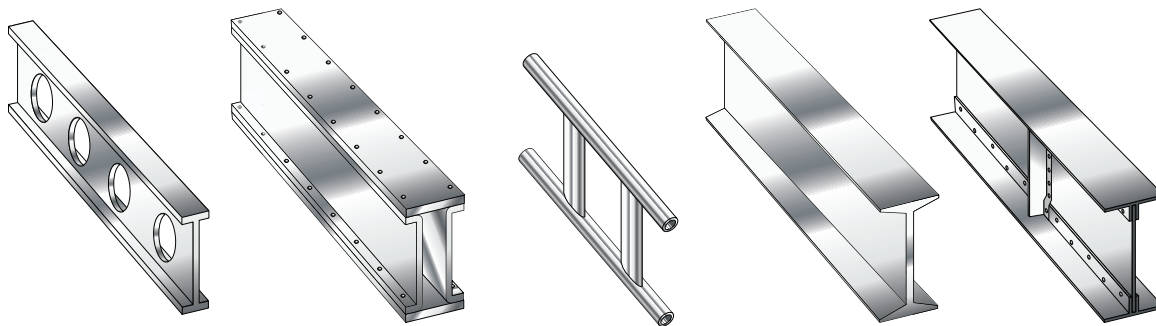


Figure 3-15. Examples of metal wing spar shapes.

between the bulkheads and the smooth outer skin so that the wing can better carry tension and compression loads. In some cases, heavy longitudinal stiffeners are substituted for the corrugated sheets. A combination of corrugated sheets on the upper surface of the wing and stiffeners on the lower surface is also sometimes used.

WING SPARS

Spars are the principal structural members of the wing. They correspond to the longerons of the fuselage. They run parallel to the lateral axis of the aircraft, from the fuselage toward the tip of the wing, and are usually attached to the fuselage by wing fittings, plain beams, or a truss. Spars may be made of metal, wood, or composite materials depending on the design criteria of a specific aircraft.

Most wing spars are basically rectangular in shape with the long dimension of the cross-section oriented up and down in the wing. Currently, most manufactured aircraft have wing spars made of solid extruded aluminum or aluminum extrusions riveted together to form the spar. The increased use of composites and the combining of materials should make airmen vigilant for wings spars made from a variety of materials. A spar may be constructed completely from composite material. **Figure 3-15** shows examples of metal wing spar cross-sections.

In an I-beam spar, the top and bottom of the I-beam are called the caps and the vertical section is called the web. The entire spar can be extruded from one piece of metal but often it is built up from multiple extrusions or formed angles. The web forms the principal depth portion of the spar and the cap strips (extrusions, formed angles, or milled sections) are attached to it. Together, these members carry the loads caused by wing bending, with the caps providing a foundation for attaching

the skin. Although the spar shapes in **Figure 3-15** are typical, actual wing spar configurations assume many forms. For example, the web of a spar may be a plate or a truss as shown in **Figure 3-16**.

It could be built up from light weight materials with vertical stiffeners employed for strength. (**Figure 3-17**) It could also have no stiffeners but might contain flanged holes for reducing weight but maintaining strength. Some metal and composite wing spars retain the I-beam concept but use a sine wave web. (**Figure 3-18**)

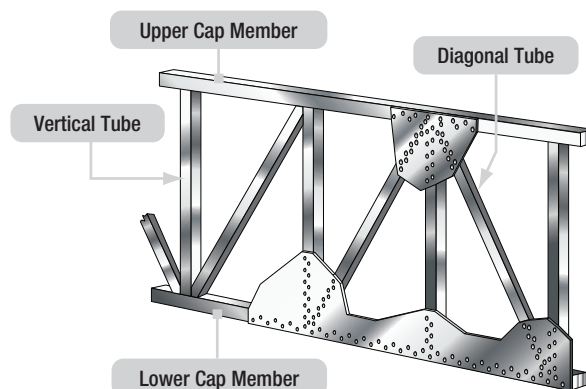


Figure 3-16. A truss wing spar.

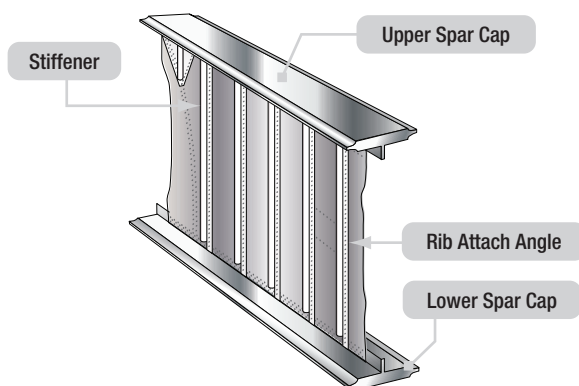


Figure 3-17. A plate web wing spar with vertical stiffeners.

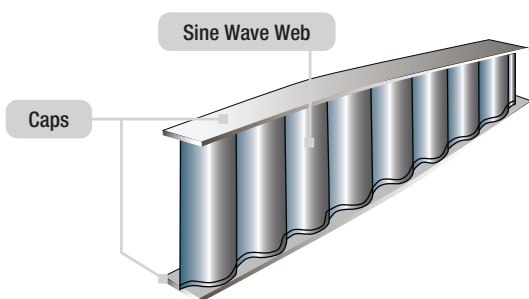


Figure 3-18. A sine wave wing spar can be made from aluminum or composite materials.

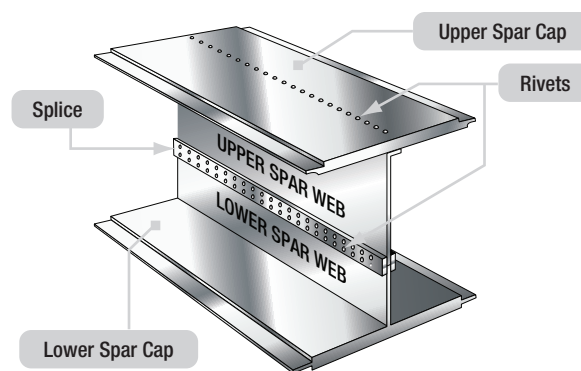


Figure 3-19. A fail-safe spar with a riveted spar web.

Additionally, fail-safe spar web design exists. Fail-safe means that should one member of a complex structure fail, some other part of the structure assumes the load of the failed member and permits continued operation. A spar with failsafe construction is shown in **Figure 3-19**. This spar is made in two sections. The top section consists of a cap riveted to the upper web plate. The lower section is a single extrusion consisting of the lower cap and web plate. These two sections are spliced together to form the spar. If either section of this type of spar breaks, the other section can still carry the load. This is the fail-safe feature. As a rule, a wing has two spars.

One spar is usually located near the front of the wing, and the other about two thirds of the distance toward the wing's trailing edge. Regardless of type, the spar is the most important part of the wing. When other structural members of the wing are placed under load, most of the resulting stress is passed on to the wing spar. False spars are commonly used in wing design. They are longitudinal members like spars but do not extend the entire span-wise length of the wing. Often, they are used as hinge attach points for control surfaces, such as an aileron spar.

WING RIBS

Wing ribs are the structural crosspieces that combine with spars and stringers to make up the framework of the wing. They usually extend from the wing leading edge to the rear spar or to the trailing edge of the wing. The ribs give the wing its cambered shape and transmit the load from the skin and stringers to the spars. Similar ribs are also used in ailerons, elevators, rudders, and stabilizers.

A wing rib may also be referred to as a plain rib or a main rib. Wing ribs with specialized locations or functions are given names that reflect their uniqueness. For example, ribs that are located entirely forward of the front spar that are used to shape and strengthen the wing leading edge are called nose ribs or false ribs. False ribs are ribs that do not span the entire wing chord, which is the distance from the leading edge to the trailing edge of the wing. Wing butt ribs may be found at the inboard edge of the wing where the wing attaches to the fuselage. Depending on its location and method of attachment, a butt rib may also be called a bulkhead rib or a compression rib if it is designed to receive compression loads that tend to force the wing spars together. **Figure 3-20** illustrates the basic structural components of a wing.

WING ROOTS AND TIPS

At the inboard end of the wing spars is some form of wing attach fitting as illustrated in **Figure 3-20**.

These provide a strong and secure method for attaching the wing to the fuselage. The interface between the wing and fuselage is often covered with a fairing to achieve smooth airflow in this area. The fairing(s) can be removed for access to the wing attach fittings. (**Figure 3-21**)

The wing tip is often a removable unit, bolted to the outboard end of the wing panel. One reason for this is the vulnerability of the wing tips to damage, especially during ground handling and taxiing. **Figure 3-22** shows a removable wing tip for a large aircraft wing. Others are different. The wing tip assembly is of aluminum alloy construction. The wing tip cap is secured to the tip with countersunk screws and is secured to the interspar structure at four points with 1/4-inch diameter bolts.

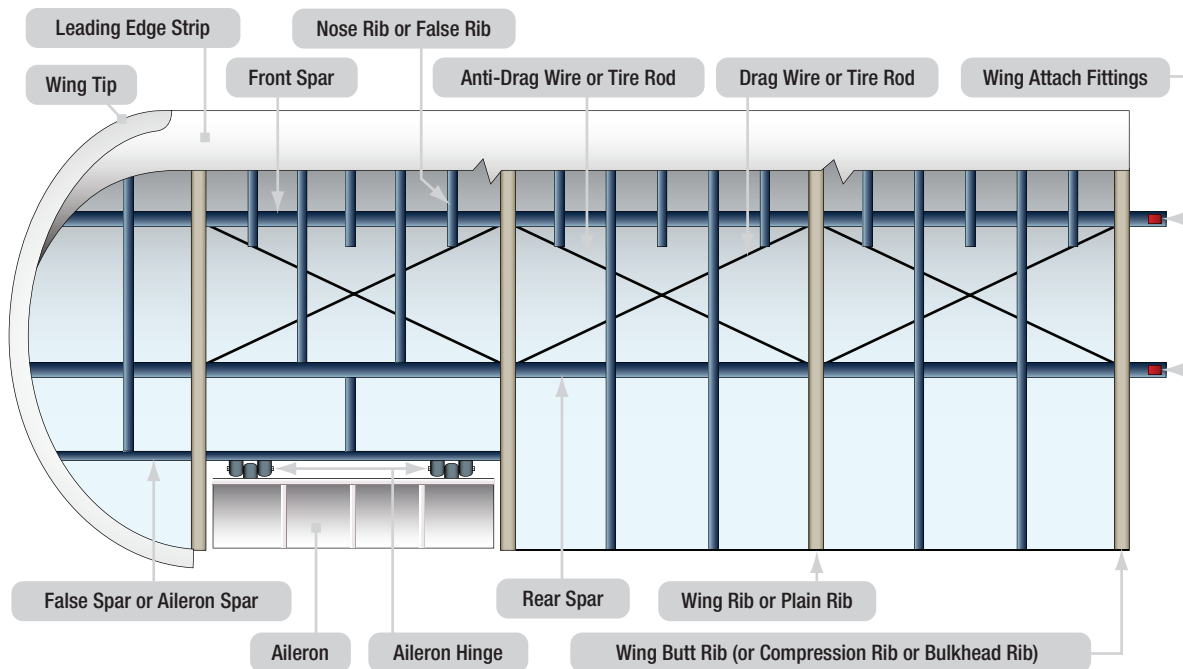


Figure 3-20. Basic wood wing structure and components.



Figure 3-21. Wing root fairings smooth airflow and hide wing attach fittings.

To prevent ice from forming on the leading edge of the wings of large aircraft, hot air from an engine is often channeled through the leading edge from wing root to wing tip. A louver on the top surface of the wingtip allows this warm air to be exhausted overboard. Wing position lights are located at the center of the tip and are not directly visible from the cockpit. As an indication that the wing tip light is operating, some wing tips are equipped with a Lucite rod to transmit the light to the leading edge.

WING SKIN

The skin on many wings is designed to carry part of the flight and ground loads in combination with the spars and ribs. This is known as a stressed-skin design.

The all-metal, full cantilever wing section illustrated in **Figure 3-23** shows the structure of one such design. The lack of extra internal or external bracing requires that the skin share some of the load. Notice the skin is stiffened to aid with this function.

The wing skin on an aircraft may be made from a wide variety of materials such as fabric, wood, or aluminum. Most transport and high performance aircraft use aluminum and composites to skin the wings. When using aluminum, a single thin sheet of material is not always employed. Chemically milled aluminum skin can provide skin of varied thicknesses. The wing skin is stronger and carries more of the loads where it is milled thicker, usually near the wing root.

When milled thin, standard loading takes place and the milled skin may transition to sheet aluminum skin. On aircraft with stressed-skin wing design, honeycomb structured wing panels are often used as skin. A honeycomb structure is built up from a core material resembling a bee hive's honeycomb which is laminated or sandwiched between thin outer skin sheets. **Figure 3-24** illustrates honeycomb panes and their components. Panels formed like this are lightweight and very strong. They have a variety of uses on the aircraft, such as floor panels, bulkheads, and control surfaces, as well as wing skin panels.

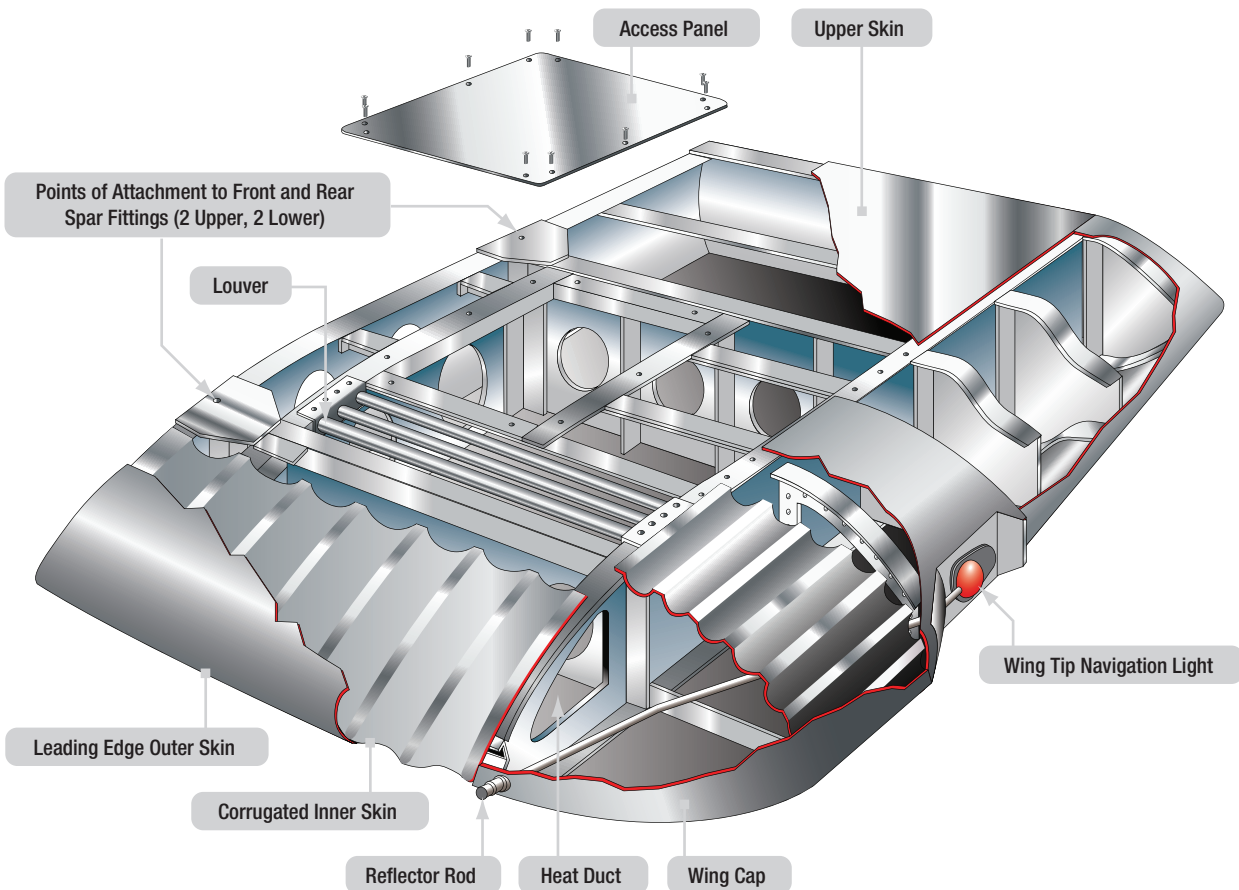


Figure 3-22. A removable metal wing tip.

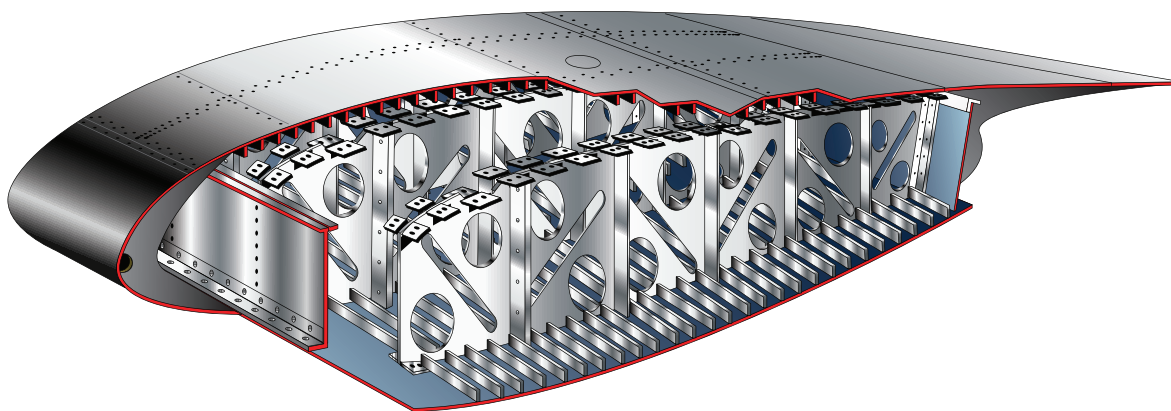


Figure 3-23. The skin is an integral load carrying part of a stressed skin design.

Figure 3-25 shows the locations of honeycomb construction wing panels on a jet transport aircraft.

A honeycomb panel can be made from a wide variety of materials. Aluminum core honeycomb with an outer skin of aluminum is common. But honeycomb in which the core is an Arimid® fiber and the outer sheets are coated Phenolic® is common as well. In fact, a myriad of other material combinations such as those

using fiberglass, plastic, Nomex®, Kevlar®, and carbon fiber all exist. Each honeycomb structure possesses unique characteristics depending upon the materials, dimensions, and manufacturing techniques employed. **Figure 3-26** shows an entire wing leading edge formed from honeycomb structure.

FUEL STORAGE

Fuel is often carried inside the wings of a stressed-skin

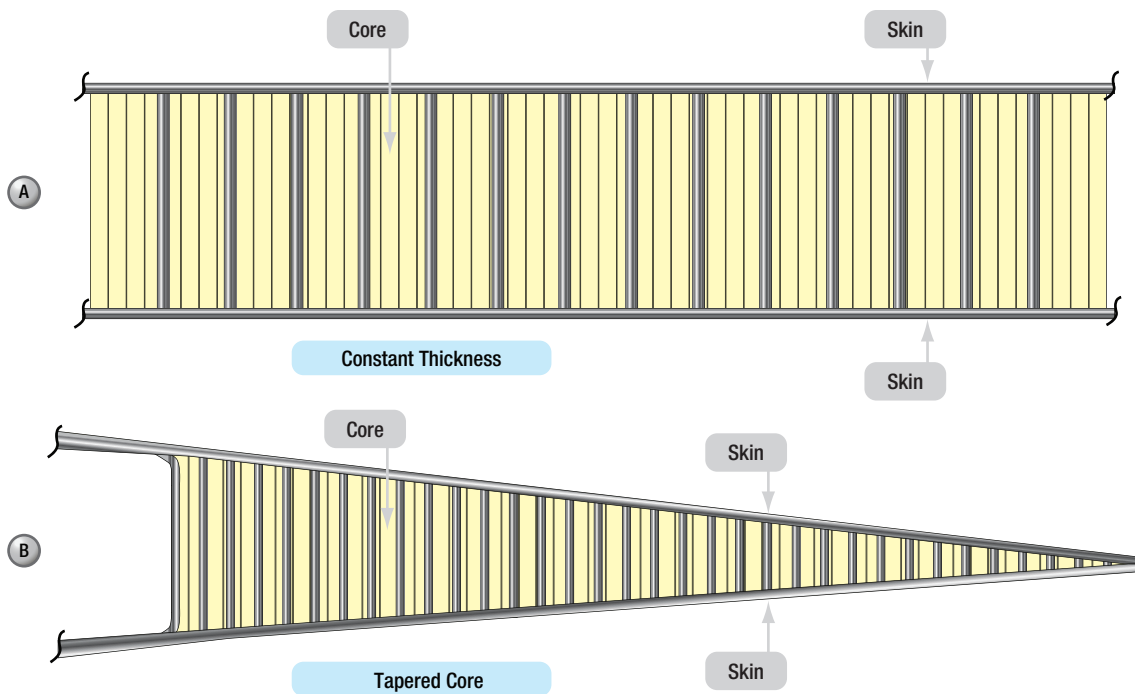


Figure 3-24. The honeycomb panel is a staple in aircraft construction. Cores can be either constant thickness (A) or tapered (B). Tapered core honeycomb panels are frequently used as flight control surfaces and wing trailing edges.

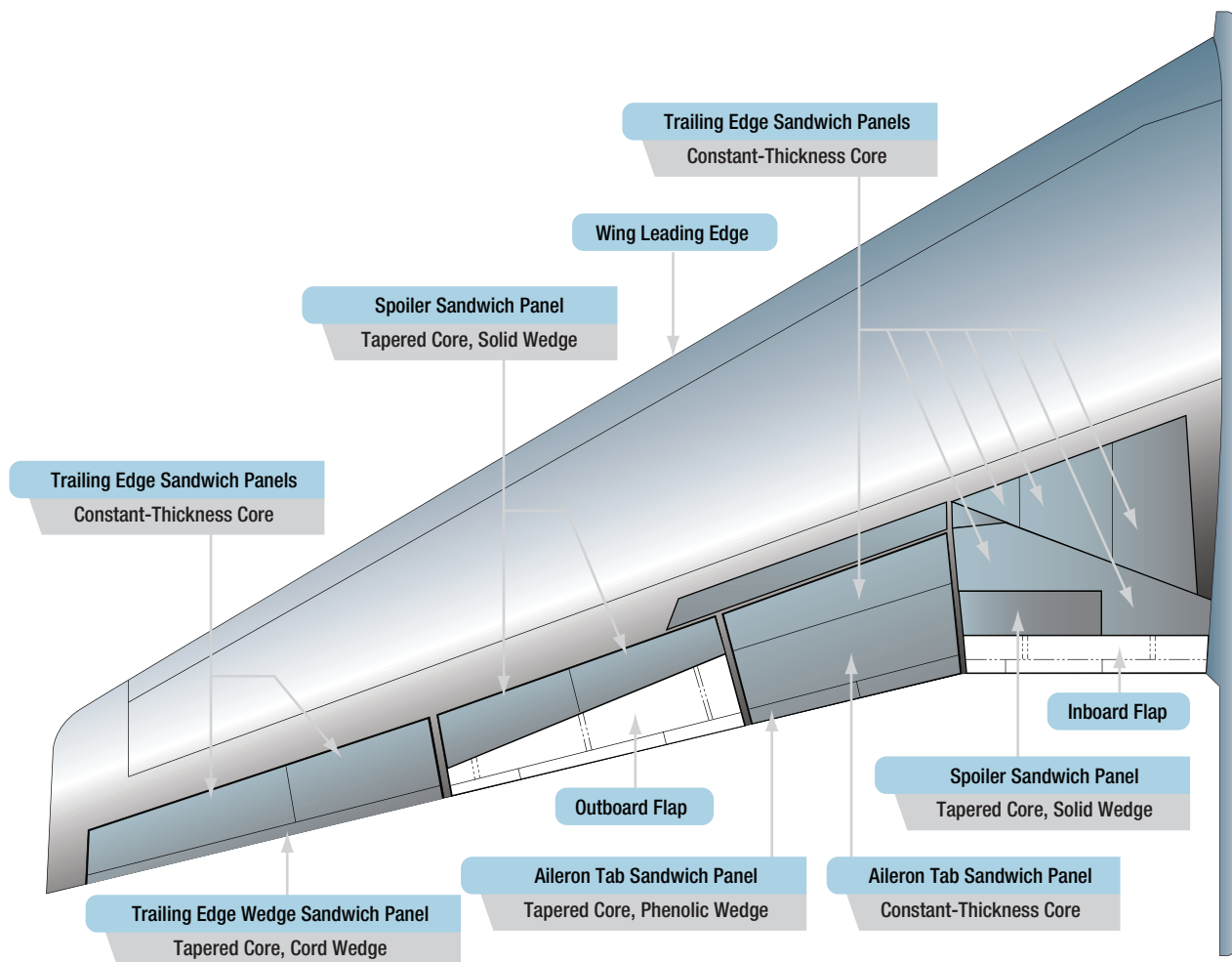


Figure 3-25. Honeycomb wing construction on a large jet transport aircraft.

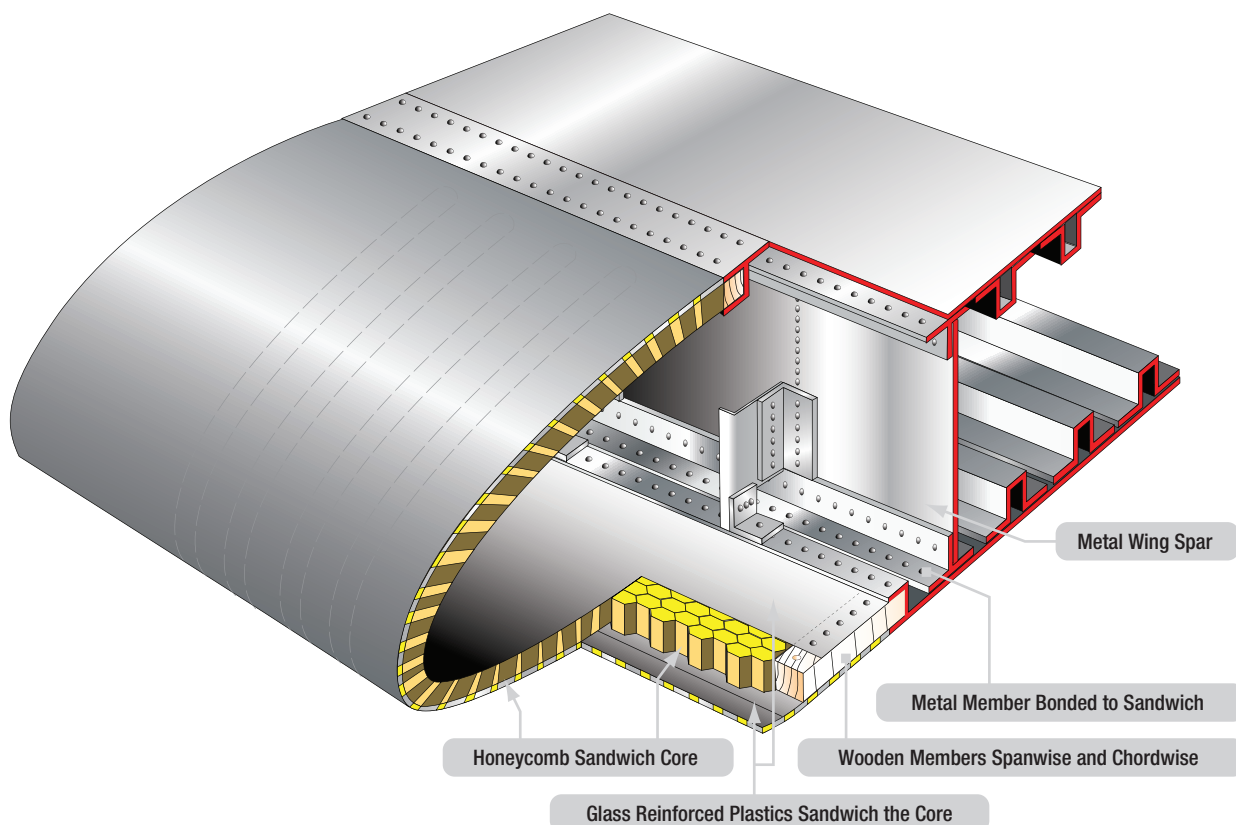


Figure 3-26. A wing leading edge formed from honeycomb material bonded to the aluminum spar structure.

aircraft. The joints in the wings are sealed with a special fuel resistant sealant enabling fuel to be stored directly inside the structure. This is known as wet wing design and the wing fuel tanks are known as integral fuel tanks. Alternately, a fuel-carrying bladder or tank can be fitted inside a wing.

Figure 3-27 shows a wing section with a box beam structural design such as one that might be found in a transport category aircraft. This structure increases

strength while reducing weight. Proper sealing of the structure allows fuel to be stored in the box sections of the wing. An integral fuel tank requires little change in structure of the wing although it is engineered to accept the fuel load. Wing structures that incorporate bladder or rigid fuel tanks installation must be formed into a supported box structure to accept the tank. The advantage of a bladder tank is that a smaller opening in the wing surface is required to insert the tank and thus, much of the wing structure can remain unaltered.

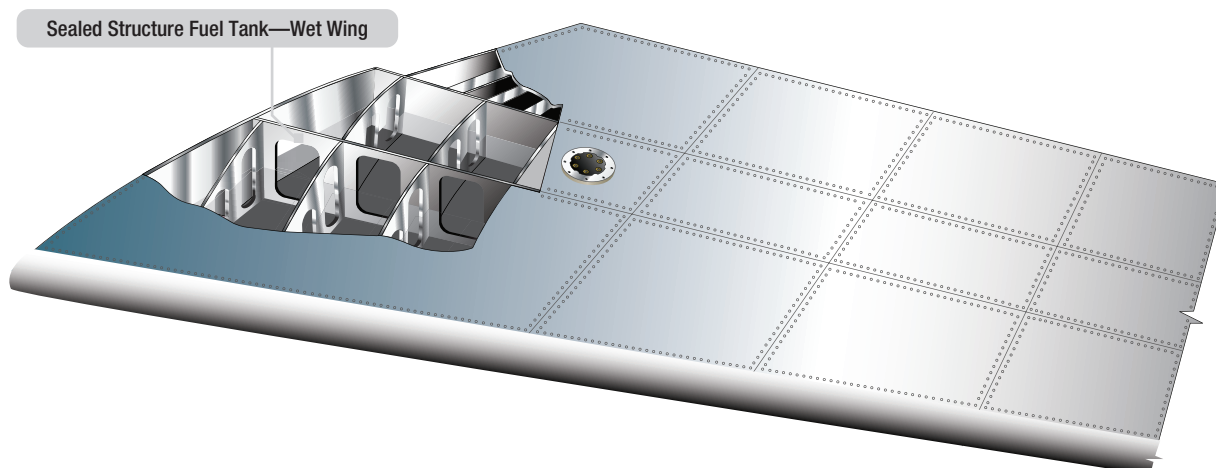


Figure 3-27. Fuel is often carried in the wings.

ATTACHMENTS

Various components and structure are attached to the wings depending on the configuration of the aircraft. The following sections discuss the typical attachments as they relate to wing structure.

LANDING GEAR

On many aircraft, the landing gear is attached to the wings. Specifically, the main gear attach point is the wing spar(s) or a framework that is attached to the spars. Landing gear must be strong enough to withstand the forces of landing when the aircraft is fully loaded. Wings spars are sufficiently strong because their function is to support the entire weight of the aircraft while in flight. Adaptive, built-up structure at the spar attach point may be used for strength and for mounting the gear such that it can be retracted into a recessed wheel well in the wing or fuselage. Wheel wells must be included so that the gear can be retracted to reduce drag. The wheel wells are strong, structural, box like enclosures open on the bottom of the fuselage.

They are framed by heavy structural members and webs. A bulkhead may form the forward and/ or aft wall of a wheel well. Wing wheel well construction includes framing a suitable space between the spars for the gear when retracted. The framework may have provisions for door attachment fittings, hinges and latches.

PYLONS

Engine pylons and nacelles may also attach to the wings. The pylon structure is built out and forward from the wing spars on most aircraft. The pylon structural members are often also called spars with an upper, middle and lower spar possible. (*Figure 3-28*) Additional bracing and frame members are included. The entire pylon structure is constructed strong enough to attach the engine mounts to it. It transfers the thrust developed by the engines to the airframe through the wing spars.

A streamlined enclosure called a nacelle is attached to the pylon to house the engine and its components. The nacelle usually presents a round or elliptical profile to the wind to reduce aerodynamic drag. It contains a firewall between the engine and the pylon and wing structure. Occasionally the pylon also supports the landing gear structure when the gear is attached to it and retracts into the nacelle. (*Figure 3-29*)

CONTROL SURFACES

Control surfaces also attach to the wing, typically at the trailing edge. The wing rear spar contains the attach points for the ailerons. Alternately, a false spar or similar reinforced structure may be built off of the spar for attachment of the ailerons.

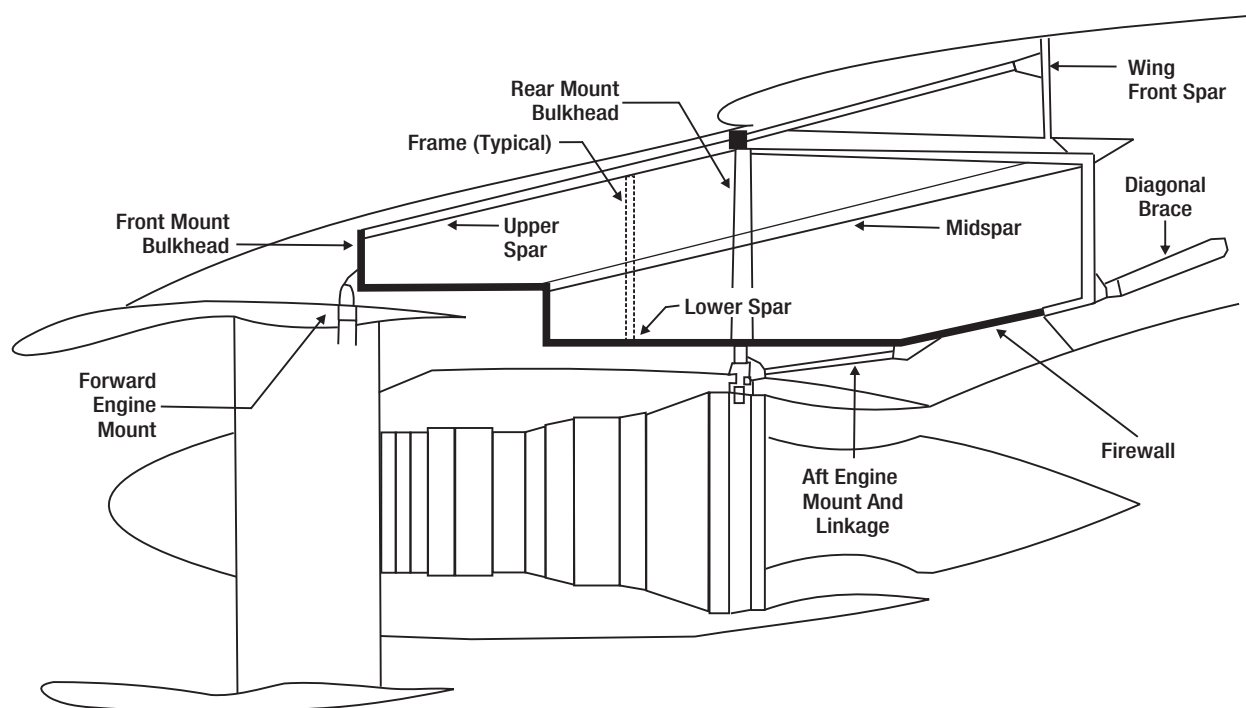


Figure 3-28. Engine pylon structure contains structural spars that extend down and forward of the wing.



Figure 3-29. An engine nacelle firewall.

HIGH LIFT/DRAG DEVICES

High lift and drag devices are attached to the wing structure at the front and rear spars. This including leading and trailing edge flaps, slats, spoilers and speed brakes. The wing spars are used to anchor these devices since the loads created by their use must be transferred to the entire aircraft through the wing structure. *Figure 3-30* illustrates various lift and drag devices in the retracted position that are attached to the wings.

STABILIZERS

CONSTRUCTION

Fixed surfaces that help stabilize the aircraft in flight are known as stabilizers. On most aircraft designs these consist of a horizontal stabilizer and a vertical stabilizer located at the aft end of the fuselage known as the empennage. The structure of the stabilizers is very similar to that which is used in wing construction. *Figure 3-31* shows a typical vertical stabilizer. Notice the use of spars, ribs, stringers, and skin like those found in a wing. They perform the same functions shaping and supporting the stabilizer and transferring stresses.

Bending, torsion, and shear created by air loads in flight pass from one structural member to another. Each member absorbs some of the stress and passes

the remainder on to the others. Ultimately, the spar transmits any overloads to the fuselage. A horizontal stabilizer is built the same way.

CONTROL SURFACE ATTACHMENT

Similar to wing aileron attachment, the rudder and elevator attach to the stabilizer structure at the rear spar. On most aircraft this will be with hinges or fittings that allow the back and forth movement of the surface.

FLIGHT CONTROL SURFACES

CONSTRUCTION AND ATTACHMENT

Aircraft flight control surfaces are aerodynamic devices allowing a pilot to adjust and control the aircraft's flight attitude. Flight control surfaces are grouped as systems and are classified as being either primary and secondary. Primary controls are those that provide control over yaw, pitch, and roll. Secondary controls include the speed brake and flap systems. All systems consist of the control surface, cockpit controls, connecting linkage, and other necessary operating methods.

Flight control surface construction may resemble the wing construction of the aircraft in question. Spars, ribs and skin are all used. Epoxy reinforced composite construction and honeycomb core construction covered with aluminum skin is common as well. Flight control surface construction is guided by the requirement that the flight control be light in weight. They have minimal structure aft of the front spar where honeycomb core sections are common. Aluminum skin bonded with adhesive to the tapered honeycomb forms the trailing edge of the airfoil.

A rear spar, false spar or similar structure may be built into the trailing edge of the flight control in the area where a trim tab is attached. Hinge fittings attached to this structure are used to mount the tab.



Figure 3-30. The position of various lift and drag devices on a wing.

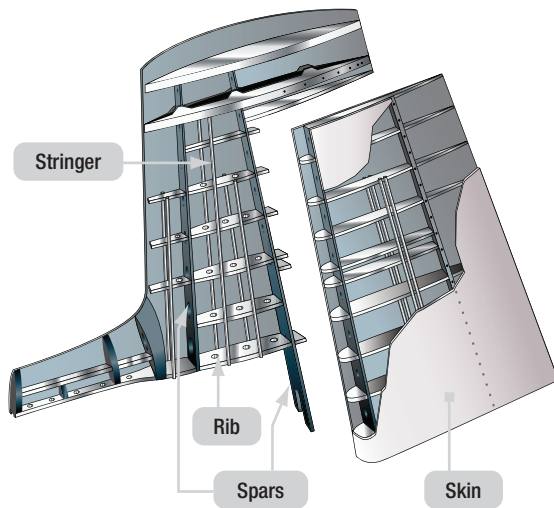


Figure 3-31. Vertical stabilizer.

BALANCING

Flight control "flutter" is of great concern in that it can result in loss of control of the aircraft. Balancing a flight control eliminates flutter. Two common methods of balancing exist; mass balancing and aerodynamic balancing.

MASS BALANCING

Mass balancing is accomplished by adding weights to the leading edge of the control. This moves the center of gravity of the flight control forward and makes the part of the control behind the hinge line as light as possible to eliminate flutter.

AERODYNAMIC BALANCING

Aerodynamic balancing is accomplished by setting the hinge into the control surface so that when deflected from neutral, the part of the surface forward of the hinge line projects into the airstream. The force of the air on the forward portion of the control surface also helps to keep it from fluttering. (*Figure 3-32*)

NACELLES/PYLONS

PYLONS

Nacelles and pylons may be attached to wings or the fuselage. The pylon extends the airframe structure allowing an engine to be mounted far enough away from the airframe so that hot engine and engine exhaust temperatures do not damage the airframe.



Figure 3-32. The bottom of the up aileron pivots into the airstream creating drag and helping to prevent flutter.

The pylon is the structure off of which the nacelle is built to house the engine. Inside the nacelle enclosure, the engine mounts attach to the pylon structure or a nacelle bulkhead. Pylons make access for engine maintenance and engine replacement easier. They also safely distance the engine from the airframe in case of fire or explosion. (*Figure 3-33*)

The framework of a nacelle usually consists of structural members similar to those of the fuselage. Lengthwise members, such as longerons and stringers, combine with horizontal/vertical members, such as rings, formers, and bulkheads, to give the nacelle its shape and structural integrity.

The exterior of a nacelle is covered with a skin or fitted with components inside. Both are usually made of sheet aluminum or magnesium alloy with stainless steel or titanium alloys being used in high-temperature areas, such as around the exhaust exit. Regardless of the material used, the skin is typically attached to the framework with rivets. (*Figure 3-34*)

FIREWALLS

A firewall is incorporated into the nacelle to isolate the engine compartment from the rest of the aircraft. A firewall is basically a stainless steel or titanium bulkhead that contains a fire in the confines of the nacelle rather than letting it spread throughout the airframe. (*Figure 3-35*)

ENGINE MOUNTS

Engine mounts are also found in the nacelle. These are the structural assemblies to which the engine is fastened. They are usually constructed from chrome/molybdenum steel tubing in light aircraft and forged chrome/nickel/molybdenum assemblies in larger aircraft.

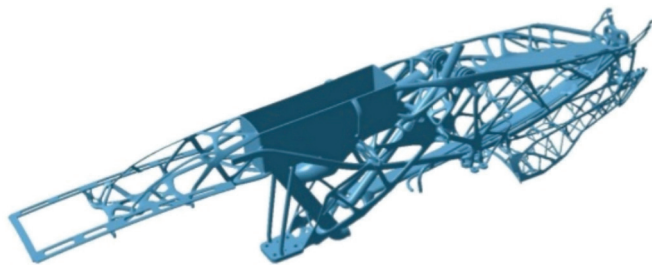


Figure 3-33. An engine pylon for an under wing engine.

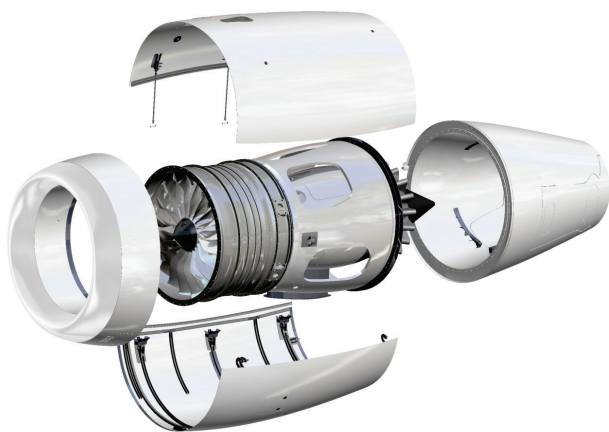


Figure 3-34. The various sections of a turbine engine nacelle.

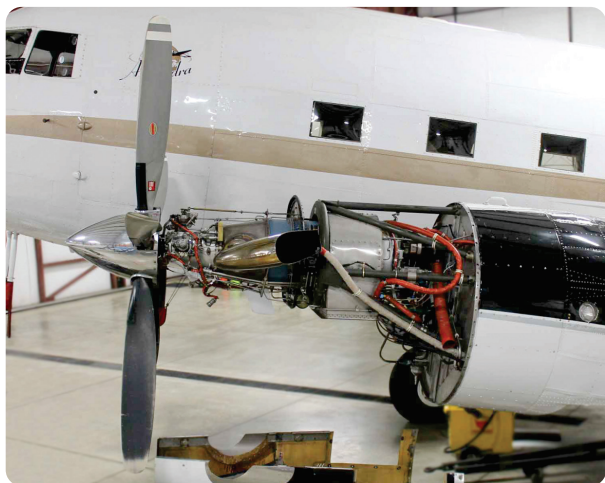


Figure 3-35. A turboprop engine mount assembly and firewall.

Question: 3-1

The majority of modern aircraft are constructed using _____ construction.

Question: 3-5

Flight deck windows on transport aircraft are constructed of _____ of tempered glass and plastic.

Question: 3-2

Name 4 major components that are attached to a fuselage.

Question: 3-6

Most often, wings are of full _____ design, which means they are built so that no external bracing is needed.

Question: 3-3

Seat _____ is the fore and aft space between each row of seats.

Question: 3-7

The _____ are the principle structural members of a wing.

Question: 3-4

Airliner cabin entry doors are usually _____ type doors.

Question: 3-8

A wing rib that is located entirely forward of the front spar that is used to shape and strengthen the wing leading edge is called a _____ rib or false rib.

ANSWERS

Answer: 3-1
semi-monocoque.

Answer: 3-5
laminations.

Answer: 3-2
Engine pylon.
Wings.
Stabilizers.
Landing gear.

Answer: 3-6
cantilever.

Answer: 3-3
pitch.

Answer: 3-7
spars.

Answer: 3-4
plug.

Answer: 3-8
nose.

Question: 3-9

_____ have a variety of uses on the aircraft, such as floor panels, bulkheads, and control surfaces, as well as wing skin panels.

Question: 3-12

Two common methods of balancing a flight control are mass balancing and _____ balancing.

Question: 3-10

The joints in the wings are sealed with a special fuel resistant sealant enabling fuel to be stored directly inside the structure. This is known as _____ design.

Question: 3-13

A _____ is incorporated into the nacelle to isolate the engine compartment from the rest of the aircraft.

Question: 3-11

A streamlined enclosure called a nacelle is attached to the engine _____ to house the engine and its components.

ANSWERS

Answer: 3-9

Honeycomb structured panels.

Answer: 3-12

aerodynamic.

Answer: 3-10

wet wing.

Answer: 3-13

firewall.

Answer: 3-11

pylon.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 04

AIR CONDITIONING AND CABIN PRESSURIZATION (ATA 21)

Knowledge Requirements

11.4 - Air Conditioning and Cabin Pressurization (ATA 21)

11.4.1 - Air Supply

Sources of air supply including engine bleed, APU and ground cart.

2

11.4.2 - Air Conditioning

Air conditioning systems; Air cycle and vapor cycle machines; Distribution systems; Flow, temperature and humidity control system.

3

11.4.3 - Pressurization

Pressurization systems; Control and indication including control and safety valves; Cabin pressure controllers.

3

11.4.4 - Safety and Warning Devices

Protection and warning devices.

3

AIR CONDITIONING, CABIN
PRESSURIZATION (ATA 21)

10.4 - AIR CONDITIONING AND CABIN PRESSURIZATION

AIR SUPPLY

The source of air to pressurize and air condition an aircraft varies mainly with engine type. Reciprocating aircraft have pressurization sources different from those of turbine-powered aircraft. Note that the compression of air raises its temperature. A means for keeping cabin air cool is built into most pressurization systems. It may be in the form of a heat exchanger, using cold ambient air to modify the temperature of the air from the pressurization source. However, a full air cycle air conditioning system with expansion turbine is typically used on commercial transport aircraft. It provides the advantage of cabin temperature control on the ground and at low altitudes.

BLEED AIR

The main principle of operation of a turbine engine involves the compression of large amounts of air to be mixed with fuel and burned. Air bled from the compressor section of the engine is relatively free of contaminants. As such, compressor air is a great source of air for cabin pressurization and air conditioning. However, the volume of air for engine power production is reduced by bleeding air off of the compressor. And, even though the amount of bleed air to the overall amount of air compressed for combustion is relatively small, it should still be minimized. Modern, large cabin turbofan engine aircraft contain recirculation fans to reuse up to 50 percent of the air in the cabin. This reduces bleed air volume and helps maintain high engine output.

There are different ways hot, high-pressure bleed air can be exploited for cabin pressurization and air conditioning. Smaller turbine aircraft, or sections of a large aircraft, may make use of a jet pump flow multiplier. With this device, bleed air is tapped off of the turbine engine's compressor section. It is ejected into a venturi jet pump mounted in air ducting that has one end open to the ambient air and the other end directed into the compartment to be pressurized. Due to the low pressure established in the venturi by the bleed air flow, air is drawn in from outside the aircraft. It mixes with the bleed air and is delivered to the pressure vessel to pressurize it. An advantage of this type of pressurization is the lack of moving parts. (*Figure 4-1*) Disadvantages are that only a relatively small volume of space can be pressurized in this manner and there is no means for cooling the ambient air.

Another method of pressurizing an aircraft using turbine engine compressor bleed air is to have the bleed air drive a separate compressor that has an ambient air intake. A turbine turned by bleed air rotates a compressor impeller mounted on the same shaft. Outside air is drawn in and compressed. It is mixed with the bleed air outflow from the turbine and is sent to the pressure vessel. Turboprop aircraft often use this device, known as a turbo compressor. (*Figure 4-2*)

The most common method of pressurizing and air conditioning turbine-powered aircraft is with an air cycle air conditioning and pressurization system. Bleed air is used, and through an elaborate system that includes heat exchangers, a compressor, and an

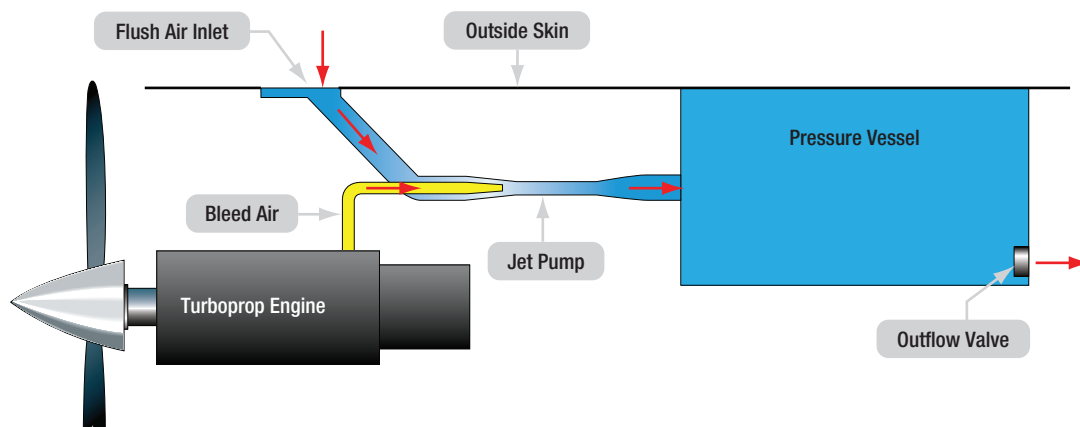


Figure 4-1. A jet pump flow multiplier ejects bleed air into a venturi which draws air for pressurization from outside the aircraft.

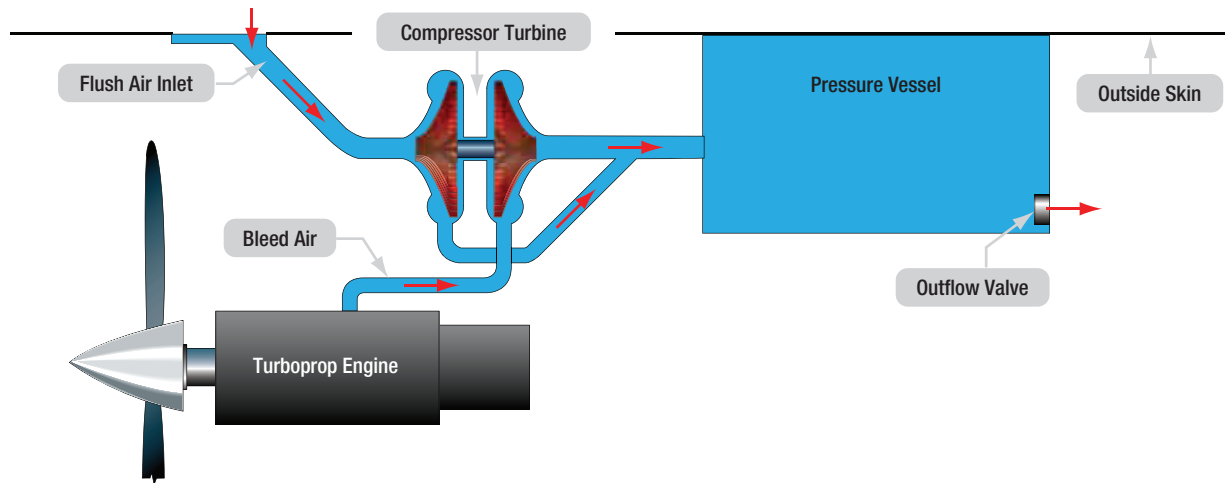


Figure 4-2. A turbo compressor used to pressurize cabins mostly in turboprop aircraft.

expansion turbine, cabin air temperature and pressure are controlled. Air cycle air conditioning is discussed in greater detail below. (*Figure 4-3*)

APU

The source of air for air conditioning does not always have to be bleed air from the compressor section of the main engines. The aircraft's auxiliary power unit (APU) is designed to deliver pressurized bleed air for engine starting and for operation of the aircraft's air conditioning packages. Sufficient pneumatic air from the APU is routed through ducting to the packs so that the cabin can be cooled while the aircraft is on the ground with the engine not running. This is one common method for keeping the cabin at a comfortable temperature which passengers are boarding.

GROUND CART

When an aircraft is on the ground, operating the engines or the APU to provide air for air conditioning is expensive. It increases the time in service of these

expensive components and expedites costly mandatory overhauls that are performed at specified time intervals. A ground cart is available to provide the pressurized source of air for the air conditioning packs. This is typically a portable powerplant that drives a high volume air compressor.

The cart is towed to the aircraft's location on the ramp and is connected into the aircraft's pneumatic system ducting with a 4 inch diameter hose. The connection point is upstream of the air conditioning packs. Cart air is regulated to the normal pneumatic system pressure and can also be used for pneumatic system trouble shooting without the expense of running the APU or main engines.

An even better and more economical solution for cooling the aircraft while it is stationary on the ground exists. Most high-performance, medium-size and larger turbine-powered aircraft are fitted with a receptacle in the air distribution system. The air distribution system basically consists of a series of ducts that carry conditioned air from the packs to wherever it is needed on the aircraft. This is discussed further below. A ground source of conditioned air can be connected directly into the distribution system receptacle. Cool air from a ground-based air conditioner is blown directly into the cabin through the aircraft's distribution system ducting. (*Figure 4-4*) This makes operating the aircraft's air conditioning packs unnecessary. The ground-based air conditioning unit can be a large air conditioner mounted on a truck or it can be a fixed type such as those used in homes and businesses.



Figure 4-3. An air cycle air conditioning system used to pressurize and regulate the temperature of the cabin of a business jet aircraft.



Figure 4-4. A duct hose installed on this airliner distributes hot or cold air from a ground-based source throughout the cabin using the aircraft's own air distribution system ducting.

AIR CONDITIONING SYSTEMS

There are two types of air conditioning systems commonly used on aircraft. Air cycle air conditioning is used on most turbine-powered aircraft. It makes use of engine bleed air or APU pneumatic air during the conditioning process.

Vapor cycle air conditioning systems are often used on reciprocating aircraft. This type system is similar to that found in homes and automobiles. Note that some turbine-powered aircraft also use vapor cycle air conditioning.

AIR CYCLE AIR CONDITIONING

Air cycle air conditioning prepares engine bleed air to pressurize the aircraft cabin. The temperature and quantity of the air must be controlled to maintain a comfortable cabin environment at all altitudes and on the ground. The air cycle system is often called the air conditioning package or pack. It is usually located in the lower half of the fuselage or in the tail section of turbine-powered aircraft. (Figure 4-5)

SYSTEM OPERATION

Even with the frigid temperatures experienced at high altitudes, bleed air is too hot to be used in the cabin without being cooled. It enters the air cycle system and is routed through a heat exchanger where ram air cools the bleed air. This cooled bleed air is directed into an air

cycle machine. There, it is compressed before flowing through a secondary heat exchanger that cools the air again with ram air. The bleed air then flows back into the air cycle machine where it drives an expansion turbine and cools even further. Water is then removed and the air is mixed with bypassed bleed air for final temperature adjustment. It is sent to the cabin through the air distribution system. By examining the operation of each component in the air cycle process, a better understanding can be developed of how bleed air is conditioned for cabin use. Refer to **Figure 4-6**, which diagrams the air cycle air conditioning system of the Boeing 737.

PNEUMATIC SYSTEM SUPPLY

During normal flight, the pneumatic system is supplied by bleed air tap-offs located on each engine compressor section. It consists of a pneumatic manifold, valves, regulators and ducting. Typically, a pneumatic system manifold contains hot air between 30 and 75 psi. The air conditioning packs are supplied by this manifold as are other critical airframe systems, such as the anti-ice and hydraulic pressurization system.

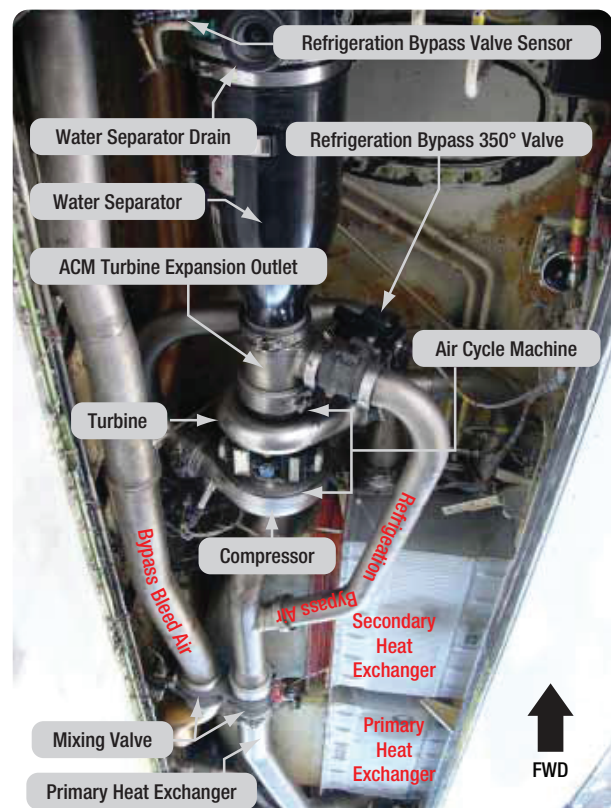


Figure 4-5. Boeing 737 air cycle system. The photo is taken looking up into the air conditioning bay located in the lower fuselage on each side of the aircraft.

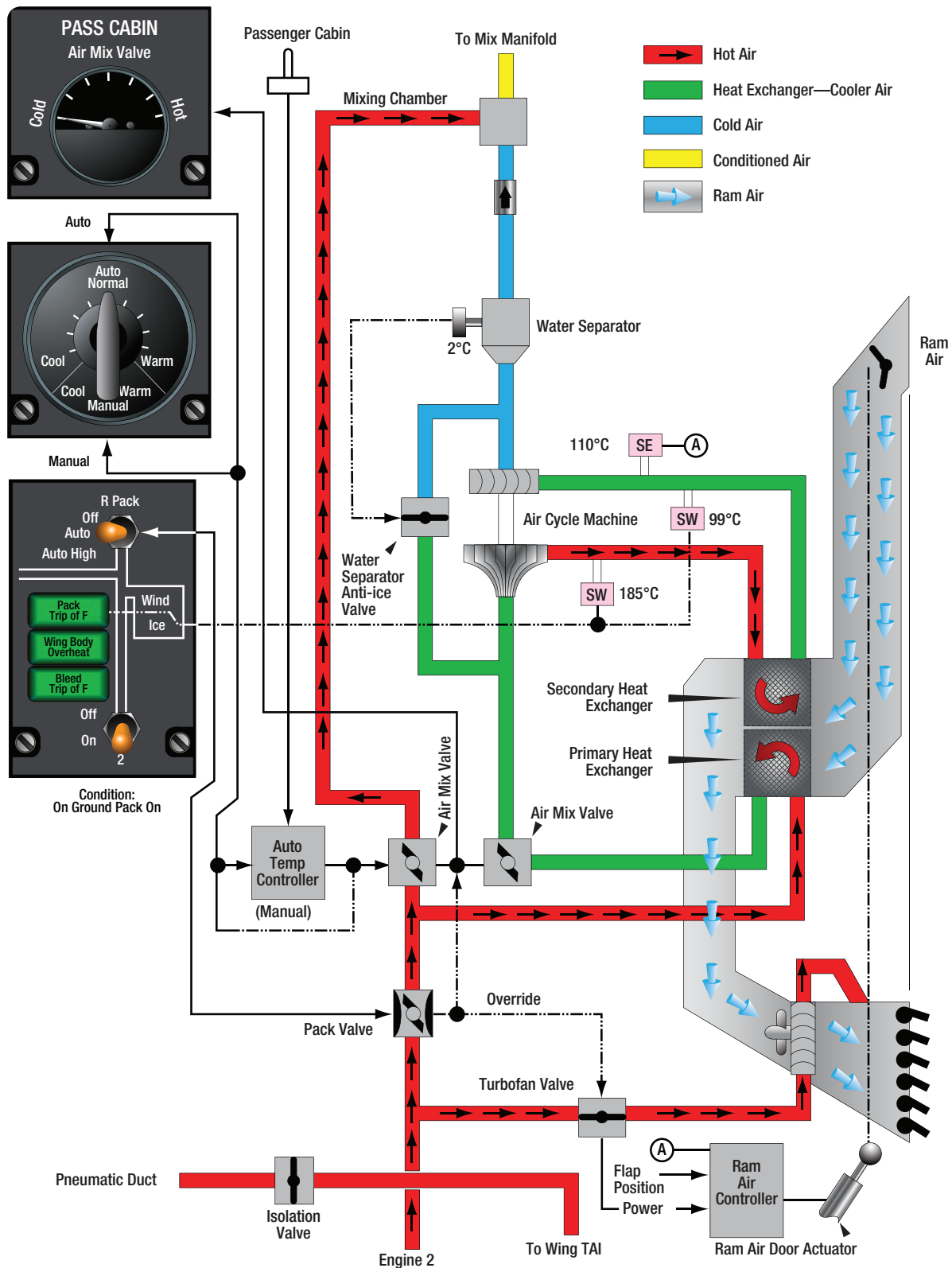


Figure 4-6. The air cycle air conditioning system on a Boeing 737.

COMPONENT OPERATION

Pack Valve

The pack valve is the valve that regulates bleed air from the pneumatic manifold into the air cycle air conditioning system. It is controlled with a switch from the air conditioning panel in the cockpit. Many pack valves are electrically controlled and pneumatically operated. Also known as the supply shutoff valve, the pack valve opens, closes, and modulates to allow the air cycle air conditioning system to be supplied with a designed volume of hot, pressurized air. (Figure 4-7) When an overheat or other abnormal condition requires that the air conditioning package be shut down, a signal is sent to the pack valve to close.

Bleed Air Bypass

A means for bypassing some of the pneumatic air supplied to the air cycle air conditioning system around the system is present on all aircraft. This warm bypassed air must be mixed with the cold air produced by the air cycle system so the air delivered to the cabin is a comfortable temperature. In the system shown in Figure 4-6, this is accomplished by the mixing valve. It simultaneously controls the flow of bypassed air and air to be cooled to meet the requirements of the auto temperature controller. It can also be controlled manually with the cabin temperature selector in manual mode. Other air cycle systems may refer to the valve that

controls the air bypassed around the air cycle cooling system as a temperature control valve, trim air pressure regulating valve, or something similar.

Primary Heat Exchanger

Generally, the warm air dedicated to pass through the air cycle system first passes through a primary heat exchanger. It acts similarly to the radiator in an automobile. A controlled flow of ram air is ducted over and through the exchanger, which reduces the temperature of the air inside the system. (Figure 4-8)

A fan draws air through the ram air duct when the aircraft is on the ground so that the heat exchange is possible when the aircraft is stationary. In flight, ram air doors are modulated to increase or decrease ram air flow to the exchanger according to the position of the wing flaps. During slow flight, when the flaps are extended, the doors are open. At higher speeds, with the flaps retracted, the doors move toward the closed position reducing the amount of ram air to the exchanger. Similar operation is accomplished with a valve on smaller aircraft. (Figure 4-9)

Refrigeration Turbine Unit or Air Cycle Machine and Secondary Heat Exchanger

The heart of the air cycle air conditioning system is the refrigeration turbine unit, also known as the air cycle machine (ACM). It is comprised of a compressor

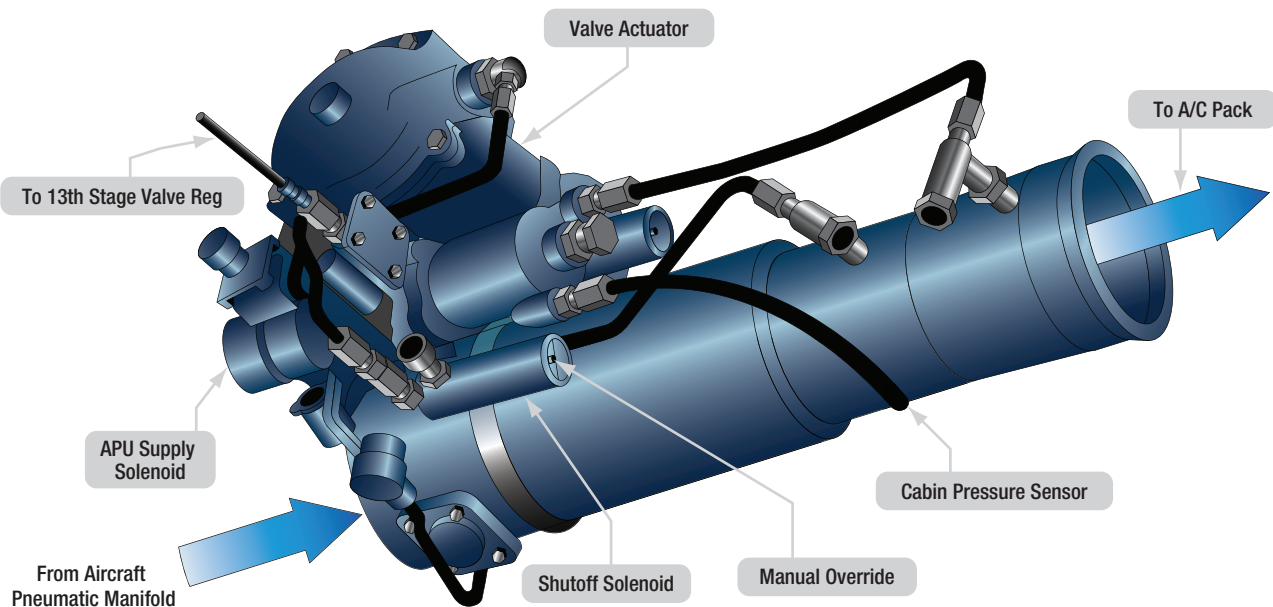


Figure 4-7. This pack valve drawing illustrates the complexity of the valve, which opens, closes, and modulates. It is manually actuated from the cockpit and automatically responds to supply and air cycle system parameter inputs.

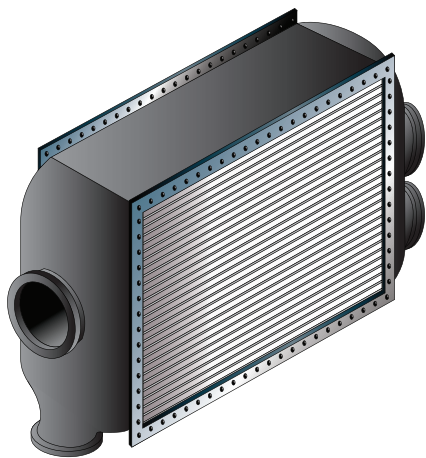


Figure 4-8. The primary and secondary heat exchangers in an air cycle air conditioning system are of similar construction. They both cool bleed air when ram air passes over the exchanger coils and fins.



Figure 4-9. A ram air door controls the flow of air through the primary and secondary heat exchangers.

that is driven by a turbine on a common shaft. System air flows from the primary heat exchanger into the compressor side of the ACM. As the air is compressed, its temperature rises. It is then sent to a secondary heat exchanger, similar to the primary heat exchanger located in the ram air duct. The elevated temperature of the ACM compressed air facilitates an easy exchange of heat energy to the ram air. The cooled system air, still under pressure from the continuous system air flow and the ACM compressor, exits the secondary heat exchanger. It is directed into the turbine side of the ACM.

The steep blade pitch angle of the ACM turbine extracts more energy from the air as it passes through and drives the turbine. Once through, the air is allowed to expand at the ACM outlet, cooling even further. The combined

energy loss from the air first driving the turbine and then expanding at the turbine outlet lowers the system air temperature to near freezing. (*Figure 4-10*)

Water Separator

The cool air from the air cycle machine can no longer hold the quantity of water it could when it was warm. A water separator is used to remove the water from the saturated air before it is sent to the aircraft cabin. The separator operates with no moving parts. Foggy air from the ACM enters and is forced through a fiberglass sock that condenses and coalesces the mist into larger water drops. The convoluted interior structure of the separator swirls the air and water. The water collects on the sides of the separator and drains down and out of the unit, while the dry air passes through. A bypass valve is incorporated in case of a blockage. (*Figure 4-11*)

Refrigeration Bypass Valve

As mentioned, air exiting the ACM turbine expands and cools. It becomes so cold, it could freeze the water in the water separator, thus inhibiting or blocking airflow. A temperature sensor in the separator controls a refrigeration bypass valve designed to keep the air flowing through the water separator above freezing temperature. The valve is also identified by other names such as a temperature control valve, anti-ice valve, and similar. It bypasses warm air around the ACM when opened. The air is introduced into the expansion ducting, just upstream of the water separator, where it heats the air just enough to keep it from freezing. Thus, the refrigeration bypass valve regulates the temperature of the ACM discharge air so it does not freeze when passing through the water separator.

This valve is visible in *Figure 4-5* and is diagrammed in the system in *Figure 4-6*. All air cycle air conditioning systems use at least one ram air heat exchanger and an air cycle machine with expansion turbine to remove heat energy from the bleed air, but variations exist. An example of a system different from that described above is found on the McDonnell Douglas DC-10. Bleed air from the pneumatic manifold is compressed by the air cycle machine compressor before it flows to a single heat exchanger. Condensed water from the water separator is sprayed into the ram air at its entrance to the exchanger to draw additional heat from the compressed bleed air as the water evaporates. A trim air valve for each cabin zone mixes bypassed bleed air with conditioned air in

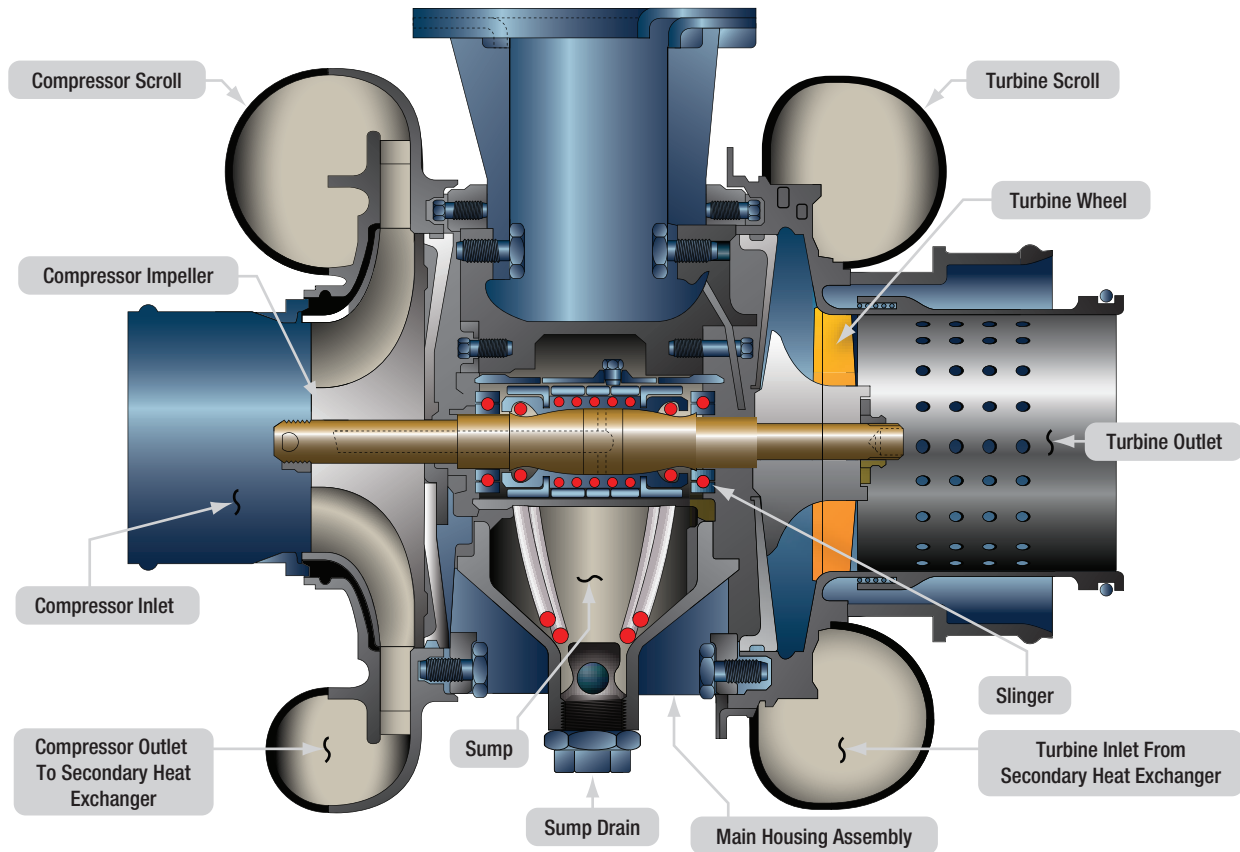


Figure 4-10. A cutaway diagram of an air cycle machine. The main housing supports the single shaft to which the compressor and turbine are attached. Oil lubricates and cools the shaft bearings.

response to individual temperature selectors for each zone. When cooling air demands are low, a turbine bypass valve routes some heat exchanger air directly to the conditioned air manifold. (*Figure 4-12*)

VAPOR CYCLE AIR CONDITIONING

Some turbine powered business class aircraft and older transport category aircraft use vapor cycle air conditioning. The absence of a bleed air source on reciprocating engine aircraft makes the use of an air cycle system impractical for conditioning cabin air. Vapor cycle air conditioning is used on most non-turbine aircraft that are equipped with air conditioning. However, it is not a source of pressurizing air as the air cycle system conditioned air is on turbine powered aircraft. The vapor cycle system only cools the cabin. If an aircraft equipped with a vapor cycle air conditioning system is pressurized, it uses a different source of air for pressurization. Vapor cycle air conditioning is a closed system used solely for the transfer of heat from inside the cabin to outside of the cabin. It can operate on the ground and in flight.

THEORY OF REFRIGERATION

Energy can be neither created nor destroyed; however, it can be transformed and moved. This is what occurs during vapor cycle air conditioning. Heat energy is moved from the cabin air into a liquid refrigerant. Due to the additional energy, the liquid changes into a vapor. The vapor is compressed and becomes very hot. It is removed from the cabin where the very hot vapor refrigerant transfers its heat energy to the outside air. In doing so, the refrigerant cools and condenses back into a liquid. The refrigerant returns to the cabin to repeat the cycle of energy transfer. (*Figure 4-13*)

Heat is an expression of energy, typically measured by temperature. The higher the temperature of a substance, the more energy it contains. Heat always flows from hot to cold. These terms express the relative amount of energy present in two substances. They do not measure the absolute amount of heat present. Without a difference in energy levels, there is no transfer of energy (heat).

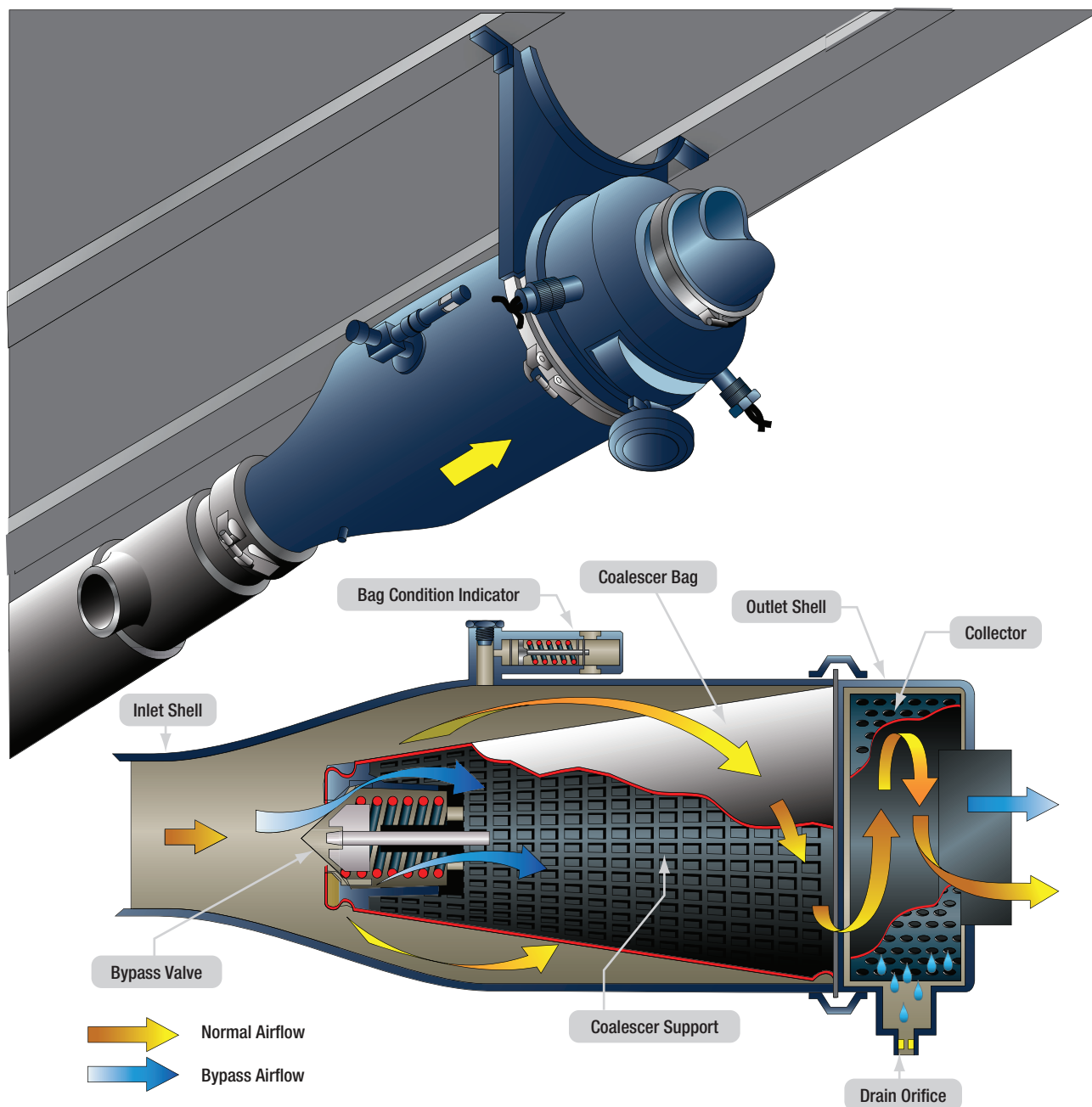


Figure 4-11. A water separator coalesces and removes water by swirling the air/water mixture from ACM expansion turbine. Centrifugal force sends the water to the walls of the collector where it drains from the unit.

Adding heat to a substance does not always raise its temperature. When a substance changes state, such as when a liquid changes into a vapor, heat energy is absorbed. This is called latent heat. When a vapor condenses into a liquid, this heat energy is given off. The temperature of a substance remains constant during its change of state. All energy absorbed or given off, the latent heat, is used for the change process. Once the change of state is complete, heat added to a substance raises the temperature of the substance.

After a substance changes state into a vapor, the rise in temperature of the vapor caused by the addition of still more heat is called superheat.

The temperature at which a substance changes from a liquid into a vapor when heat is added is known as its boiling point. This is the same temperature at which a vapor condenses into a liquid when heat is removed. The boiling point of any substance varies directly with pressure. When pressure on a liquid is increased, its boiling point increases, and when pressure on a liquid is

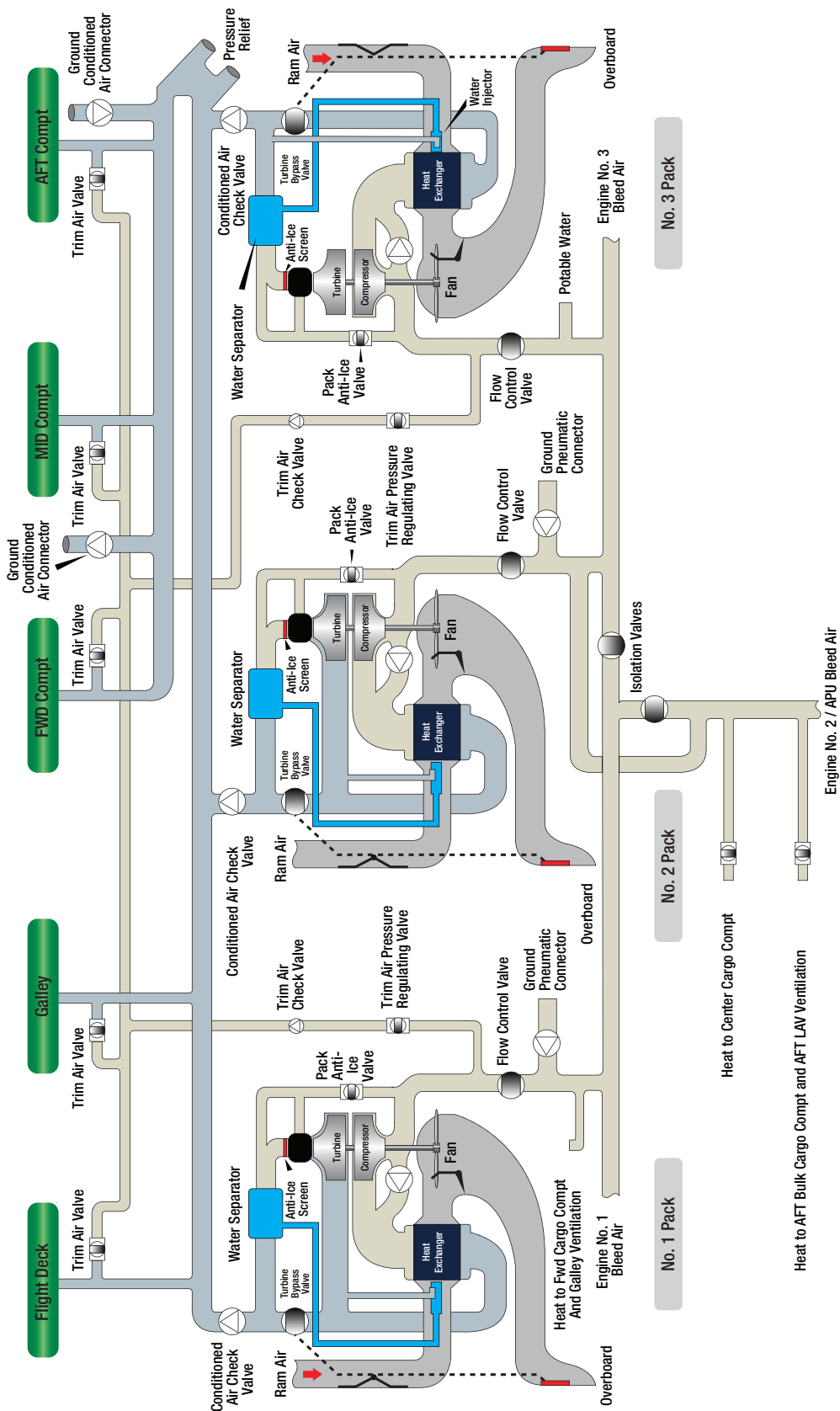


Figure 4-12. A water separator coalesces and removes water by swirling the air/water mixture from ACM expansion turbine. Centrifugal force sends the water to the walls of the collector where it drains from the unit.

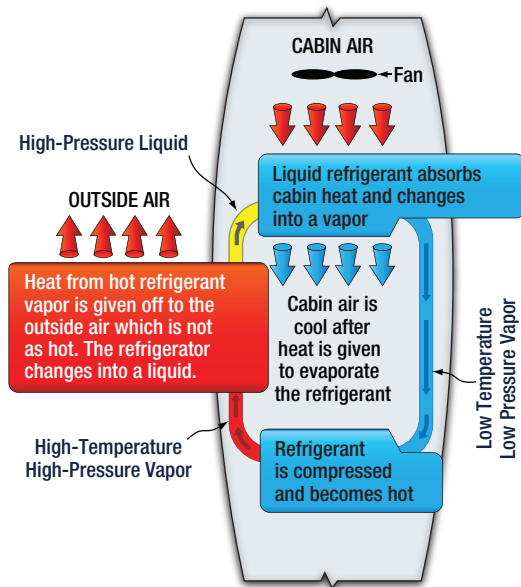


Figure 4-13. In vapor cycle air conditioning, heat is carried from the cabin to the outside air by a refrigerant which changes from a liquid to a vapor and back again.

decreased, its boiling point also decreases. For example, water boils at 100°C at normal atmospheric temperature (14.7 psi). When pressure on liquid water is increased to 20 psi, it does not boil at 100°C . More energy is required to overcome the increase in pressure. It boils at approximately 103°C . The converse is also true. Water can also boil at a much lower temperature simply by reducing the pressure upon it. With only 10 psi of pressure upon liquid water, it boils at 90°C .

(Figure 4-14)

Vapor pressure is the pressure of the vapor that exists above a liquid that is in an enclosed container at any given temperature. The vapor pressure developed by various substances is unique to each substance. A substance that is said to be volatile, develops high vapor

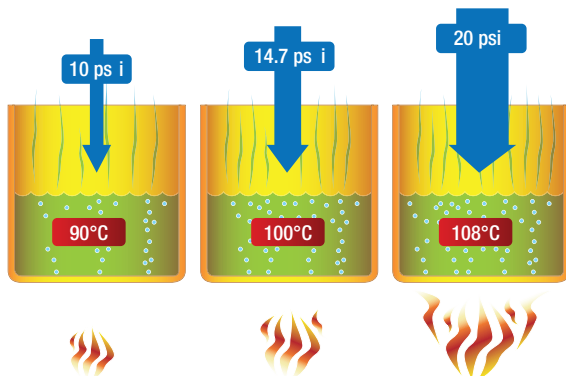


Figure 4-14. Boiling point of water changes as pressure changes.

pressure at standard day temperature (15°C). This is because the boiling point of the substance is much lower. The boiling point of tetrafluoroethane (R134a), the refrigerant used in most aircraft vapor cycle air conditioning systems, is approximately -26°C . Its vapor pressure at 15°C is about 71 psi. The vapor pressure of any substance varies directly with temperature.

BASIC VAPOR CYCLE

Vapor cycle air conditioning is a closed system in which a refrigerant is circulated through tubing and a variety of components. The purpose is to remove heat from the aircraft cabin. While circulating, the refrigerant changes state. By manipulating the latent heat required to do so, hot air is replaced with cool air in the aircraft cabin.

To begin, R134a is filtered and stored under pressure in a reservoir known as a receiver dryer. The refrigerant is in liquid form. It flows from the receiver dryer through tubing to an expansion valve. Inside the valve, a restriction in the form of a small orifice blocks most of the refrigerant. Since it is under pressure, some of the refrigerant is forced through the orifice. It emerges as a spray of tiny droplets in the tubing downstream of the valve. The tubing is coiled into a radiator type assembly known as an evaporator. A fan is positioned to blow cabin air over the surface of the evaporator. As it does, the heat in the cabin air is absorbed by the refrigerant, which uses it to change state from a liquid to a vapor. So much heat is absorbed that the cabin air blown by the fan across the evaporator cools significantly. This is the vapor cycle conditioned air that lowers the temperature in the cabin.

The gaseous refrigerant exiting the evaporator is drawn into a compressor. There, the pressure and the temperature of the refrigerant are increased. The high-pressure high-temperature gaseous refrigerant flows through tubing to a condenser. The condenser is like a radiator comprised of a great length of tubing with fins attached to promote heat transfer. Outside air is directed over the condenser. The temperature of the refrigerant inside is higher than the ambient air temperature, so heat is transferred from the refrigerant to the outside air. The amount of heat given off is enough to cool the refrigerant and to condense it back to a high-pressure liquid. It flows through tubing and back into the receiver dryer, completing the vapor cycle.

There are two sides to the vapor cycle air conditioning system. One accepts heat and is known as the low side. The other gives up heat and is known as the high side. The low and high refer to the temperature and pressure of the refrigerant. As such, the compressor and the expansion valve are the two components that separate the low side from the high side of the cycle. (*Figure 4-15*) Refrigerant on the low side is characterized as having low pressure and temperature. Refrigerant on the high side has high pressure and temperature.

VAPOR CYCLE AIR CONDITIONING SYSTEM COMPONENTS

By examining each component in the vapor cycle air conditioning system, greater insight into its function can be gained.

REFRIGERANT

For many years, dichlorodifluoromethane (R12) was the standard refrigerant used in aircraft vapor cycle air conditioning systems. Some of these systems remain in use today. R12 was found to have a negative effect on the environment; in particular, it degraded the earth's protective ozone layer. In most cases, it has been replaced by tetrafluoroethane (R134a), which is safer for the environment. R12 and R134a should not be mixed, nor should one be used in a system designed for the other. Possible damage to soft components, such as hoses and seals, could result causing leaks and or malfunction. Use only the specified refrigerant when servicing vapor cycle air conditioning systems. (*Figure 4-16*)

R12 and R134a behave so similarly that the descriptions of the R134a vapor cycle air conditioning system and components in the following paragraphs also apply to an R12 system and its components. R134a is a halogen compound (CF₃CFH₂). As mentioned, it has a boiling point of approximately -26°C. It is not poisonous to inhale in small quantities, but it does displace oxygen. Suffocation is possible if breathed in mass quantity.

Regardless of manufacturer, refrigerants are sometimes called Freon, which is a trade name owned by the Dupont Company. Caution should be used when handling any refrigerant. Because of the low boiling points, liquid refrigerants boil violently at typical atmospheric temperatures and pressure. They rapidly absorb heat energy from all surrounding matter. If

a drop lands on skin, it freezes, resulting in a burn. Similar tissue damage can result if a drop gets in one's eye. Gloves and other skin protection, as well as safety goggles, are required when working with refrigerant.

RECEIVER DRYER

The receiver dryer acts as the reservoir of the vapor cycle system. It is located downstream of the condenser and upstream of the expansion valve.

When it is very hot, more refrigerant is used by the system than when temperatures are moderate. Extra refrigerant is stored in the receiver dryer for this purpose. Liquid refrigerant from the condenser flows into the receiver dryer. Inside, it passes through filters and a desiccant material. The filters remove any foreign particles that might be in the system. The desiccant captures any water in the refrigerant. Water in the refrigerant causes two major problems. First, the refrigerant and water combine to form an acid. If left in contact with the inside of the components and tubing, the acid deteriorates the materials from which these are made. The second problem with water is that it could form ice and block the flow of refrigerant around the system, rendering it inoperative. Ice is particularly a problem if it forms at the orifice in the expansion valve, which is the coldest point in the cycle.

Occasionally, vapor may find its way into the receiver dryer, such as when the gaseous refrigerant does not completely change state to a liquid in the condenser. A stand tube is used to remove refrigerant from the receiver dryer. It runs to the bottom of the unit to ensure liquid is withdrawn and forwarded to the expansion valve. At the top of the stand tube, a sight glass allows the technician to see the refrigerant. When enough refrigerant is present in the system, liquid flows in the sight glass. If low on refrigerant, any vapor present in the receiver dryer may be sucked up the stand tube causing bubbles to be visible in the sight glass. Therefore, bubbles in the sight glass indicate that the system needs to have more refrigerant added. (*Figure 4-17*)

EXPANSION VALVE

Refrigerant exits the receiver dryer and flows to the expansion valve. The thermostatic expansion valve has an adjustable orifice through which the correct amount of refrigerant is metered to obtain optimal cooling. This is accomplished by monitoring the temperature of the

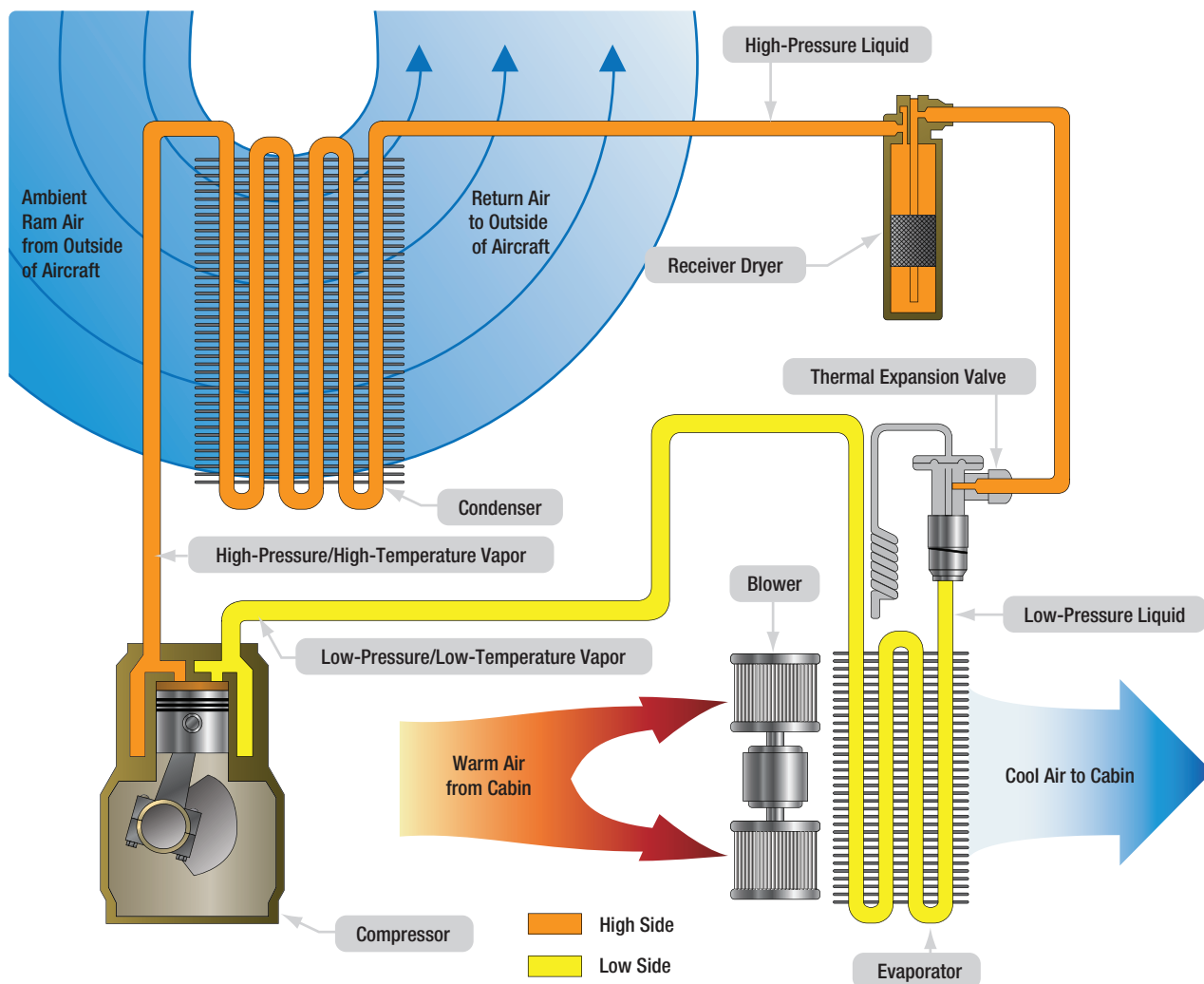


Figure 4-15. A basic vapor cycle air conditioning system. The compressor and the expansion valve are the two components that separate the low side from the high side of the cycle. This figure illustrates this division. Refrigerant on the low side is characterized as having low pressure and temperature. Refrigerant on the high side has high pressure and temperature.



Figure 4-16. A small can of R134a refrigerant used in vapor cycle air conditioning systems.

gaseous refrigerant at the outlet of the next component in the cycle, the evaporator. Ideally, the expansion valve should only let the amount of refrigerant spray into the evaporator that can be completely converted to a vapor.

The temperature of the cabin air to be cooled determines the amount of refrigerant the expansion valve should spray into the evaporator. Only so much is needed to completely change the state of the refrigerant from a liquid to a vapor. Too little causes the gaseous refrigerant to be superheated by the time it exits the evaporator. This is inefficient. Changing the state of the refrigerant from liquid to vapor absorbs much more heat than adding heat to already converted vapor (superheat). The cabin air blowing over the evaporator will not be cooled sufficiently if superheated vapor is flowing through the evaporator. If too much refrigerant is released by the expansion valve into the evaporator, some of it remains liquid when it exits the evaporator. Since it next flows to the compressor, this could be dangerous.

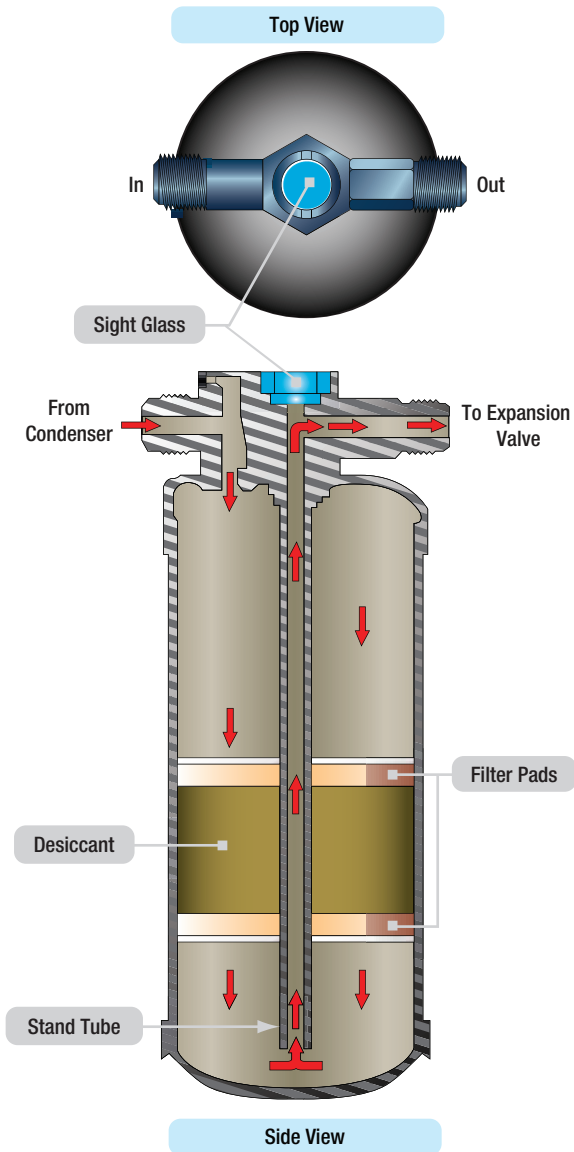


Figure 4-17. A receiver dryer acts as reservoir and filter in a vapor cycle system. Bubbles viewed in the sight glass indicate the system is low on refrigerant and needs to be serviced.

The compressor is designed to compress only vapor. If liquid is drawn in and attempts are made to compress it, the compressor could break, since liquids are essentially incompressible. The temperature of superheated vapor is higher than liquid refrigerant that has not totally vaporized. A coiled capillary tube with a volatile substance inside is located at the evaporator outlet to sense this difference. Its internal pressure increases and decreases as temperature changes. The coiled end of the tube is closed and attached to the evaporator outlet. The other end terminates in the area above a pressure diaphragm in the expansion valve. When superheated refrigerant vapor reaches the coiled end of the tube, its elevated temperature increases the pressure inside

the tube and in the space above the diaphragm. This increase in pressure causes the diaphragm to overcome spring tension in the valve. It positions a needle valve that increases the amount of refrigerant released by the valve. The quantity of refrigerant is increased so that the refrigerant only just evaporates, and the refrigerant vapor does not superheat.

When too much liquid refrigerant is released by the expansion valve, low-temperature liquid refrigerant arrives at the outlet of the evaporator. The result is low pressure inside the temperature bulb and above the expansion valve diaphragm. The superheat spring in the valve moves the needle valve toward the closed position, reducing the flow of refrigerant into the evaporator as the spring overcomes the lower pressure above the diaphragm. (Figure 4-18)

Vapor cycle air conditioning systems that have large evaporators experience significant pressure drops while refrigerant is flowing through them. Externally equalized expansion valves use a pressure tap from the outlet of the evaporator to help the superheat spring balance the diaphragm. This type of expansion valve is easily recognizable by the additional small-diameter line that comes from the evaporator into the valve (2 total). Better control of the proper amount of refrigerant allowed through the valve is attained by considering both the temperature and pressure of the evaporator refrigerant. (Figure 4-19)

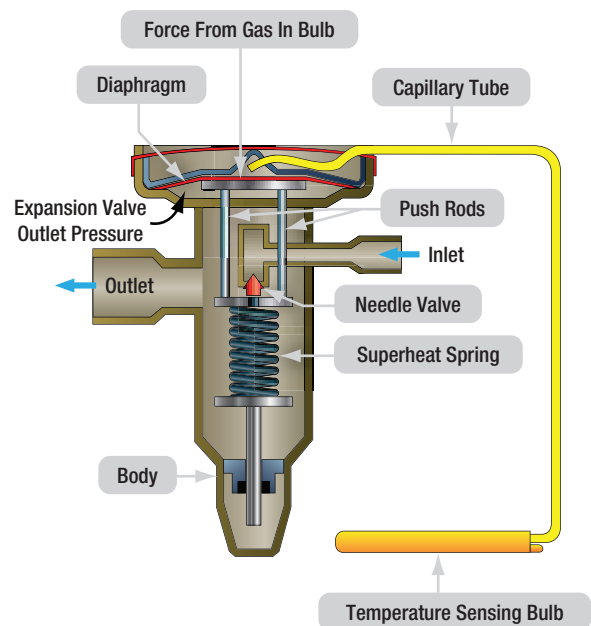


Figure 4-18. An internally equalized expansion valve.

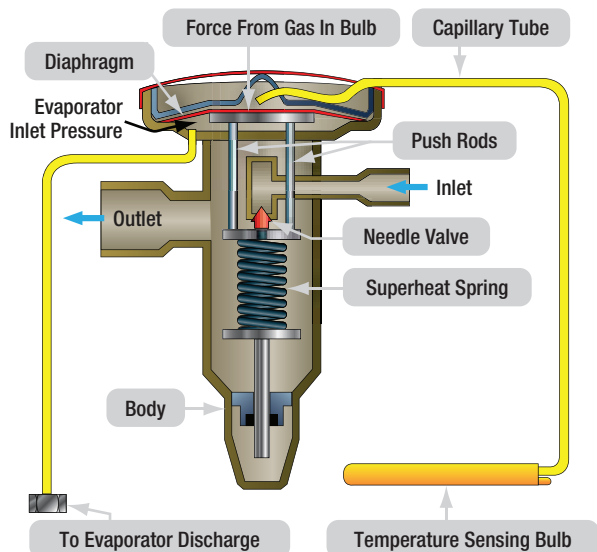


Figure 4-19. An externally equalized expansion valve uses evaporator discharge temperature and pressure to regulate the amount of refrigerant passing through the valve and into the evaporator.

EVAPORATOR

Most evaporators are constructed of copper or aluminum tubing coiled into a compact unit. Fins are attached to increase surface area, facilitating rapid heat transfer between the cabin air blown over the outside of the evaporator with a fan and the refrigerant inside.

The expansion valve located at the evaporator inlet releases high-pressure, high-temperature liquid refrigerant into the evaporator. As the refrigerant absorbs heat from the cabin air, it changes into a low-pressure vapor. This is discharged from the evaporator outlet to the next component in the vapor cycle system, the compressor. The temperature and pressure pickups that regulate the expansion valve are located at the evaporator outlet. The evaporator is situated in such a way that cabin air is pulled to it by a fan. The fan blows the air over the evaporator and discharges the cooled air back into the cabin. (*Figure 4-20*)

This discharge can be direct when the evaporator is located in a cabin wall. A remotely located evaporator may require ducting from the cabin to the evaporator and from the evaporator back into the cabin. Sometimes the cool air produced may be introduced into an air distribution system where it can blow directly on the occupants through individual delivery vents. In this manner, the entire vapor cycle air conditioning system may be located fore or aft of the cabin. A multi-position fan switch controlled by the pilot is usually available.

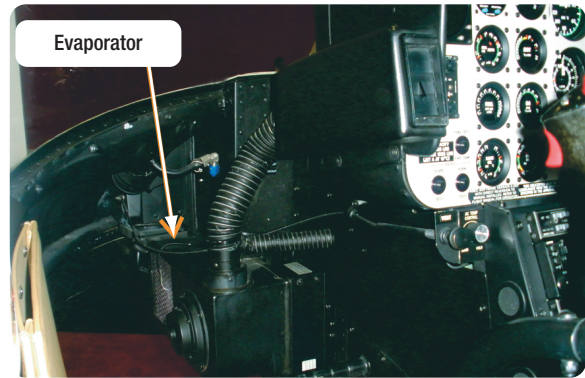


Figure 4-20. The evaporator of this aircraft's vapor cycle air conditioning system is visible in the forward cabin sidewall behind the right rudder pedal.

Figure 4-21 diagrams the vapor cycle air conditioning system in a Cessna Mustang very light jet. It has two evaporators that share in the cooling, with outlets integrated into a distribution system and cockpit mounted switches for the fans, as well as engaging and disengaging the system. When cabin air is cooled by flowing over the evaporator, it can no longer retain the water that it could at higher temperature. As a result, it condenses on the outside of the evaporator and needs to be collected and drained overboard. Pressurized aircraft may contain a valve in the evaporator drain line that opens only periodically to discharge the water, to maintain pressurization. Fins on the evaporator must be kept from being damaged, which could inhibit airflow. The continuous movement of warm cabin air around the fins keeps condensed water from freezing. Ice on the evaporator reduces the efficiency of the heat exchange to the refrigerant.

COMPRESSOR

The compressor is the heart of the vapor cycle air conditioning system. It circulates the refrigerant around the vapor cycle system. It receives low-pressure, low-temperature refrigerant vapor from the outlet of the evaporator and compresses it. As the pressure is increased, the temperature also increases. The refrigerant temperature is raised above that of the outside air temperature. The refrigerant then flows out of the compressor to the condenser where it gives off the heat to the outside air.

The compressor is the dividing point between the low side and the high side of the vapor cycle system. Often it is incorporated with fittings or has fittings in the connecting lines to it that are designed to service the

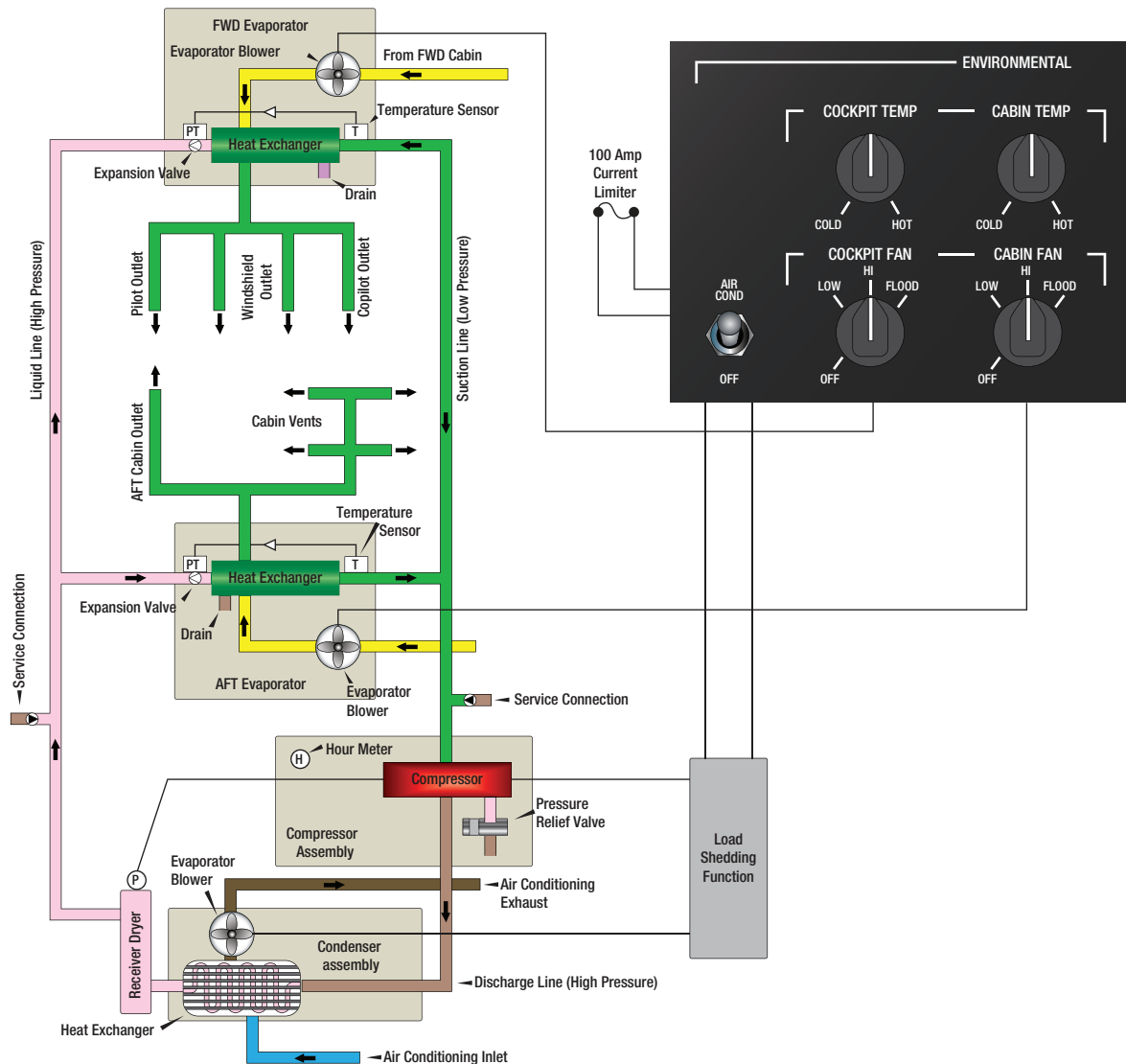


Figure 4-21. The vapor cycle air conditioning system on a Cessna Mustang has two evaporators, one for the cockpit and one for the cabin. Each evaporator assembly contains the evaporator, a blower, a thermal expansion valve and the temperature feedback line from the outlet of the evaporator to the expansion valve.

system with refrigerant. Access to the low and high sides of the system are required for servicing, which can be accomplished with fitting upstream and downstream of the compressor.

Modern compressors are either engine-driven or driven by an electric motor. Occasionally, a hydraulically driven compressor is used. A typical engine-driven compressor, similar to that found in an automobile, is located in the engine nacelle and operated by a drive belt off of the engine crankshaft. An electromagnetic clutch engages when cooling is required, which causes the compressor to operate. When cooling is sufficient, power to the clutch is cut, and the drive pulley rotates but the compressor does not. (*Figure 4-22*)

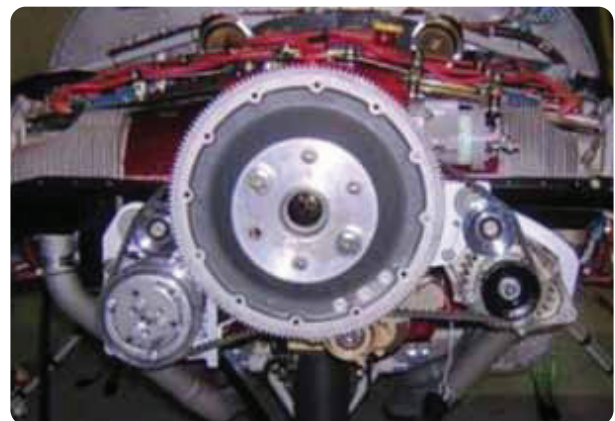


Figure 4-22. A typical belt drive engine driven compressor. The electromagnetic clutch pulley assembly in the front starts and stops the compressor depending on cooling demand.

Dedicated electric motor driven compressors are also used on aircraft. Use of an electric motor allows the compressor to be located nearly anywhere on the aircraft, since wires can be run from the appropriate bus to the control panel and to the compressor. (*Figure 4-23*) Hydraulically driven compressors are also able to be remotely located. Hydraulic lines from the hydraulic manifold are run through a switch activated solenoid to the compressor or bypasses it. This controls the operation of the hydraulically driven compressor.

Regardless of how the vapor cycle air conditioning compressor is driven, it is usually a piston type pump. It requires use of a lightweight oil to lubricate and seal the unit. The oil is entrained by the refrigerant and circulates with it around the system. The crankcase of the compressor retains a supply of the oil, the level of which can be checked and adjusted by the technician. Valves exist on some compressor installations that can be closed to isolate the compressor from the remainder of the vapor cycle system while oil servicing takes place.

CONDENSER

The condenser is the final component in the vapor cycle. It is a radiator-like heat exchanger situated so that outside air flows over it and absorbs heat from the high-pressure, high temperature refrigerant received from the compressor. A fan is usually included to draw the air through the condenser during ground operation. On some aircraft, outside air is ducted to the compressor. On others, the condenser is lowered into the airstream from the fuselage via a hinged panel. Often, the panel is controlled by a switch on the throttle levers. It is set to retract the compressor and streamline the fuselage when full power is required. (*Figure 4-24*)

The outside air absorbs heat from the refrigerant flowing through the condenser. The heat loss causes the refrigerant to change state back into a liquid. The high-pressure liquid refrigerant then leaves the condenser and flows to the receiver dryer. A properly engineered system that is functioning normally fully condenses all the refrigerant flowing through the condenser.

SERVICE VALVES

All vapor cycle air conditioning systems are closed systems; however, access is required for servicing. This is accomplished through the use of two service valves.

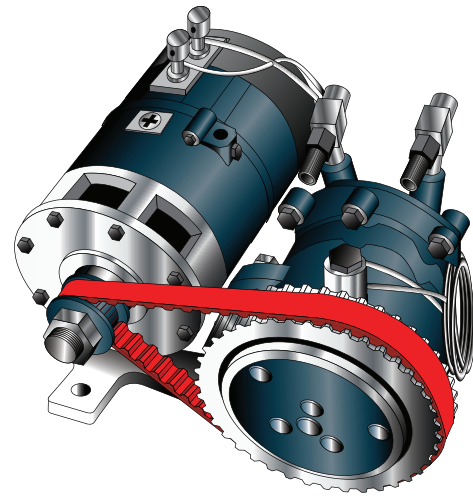


Figure 4-23. Examples of electric motor driven vapor cycle air conditioning compressors.



Figure 4-24. A vapor cycle air conditioning condenser assembly with an integral fan pulling outside air through the unit during ground operations.

One valve is located in the high side of the system and the other in the low side. A common type of valve used on vapor cycle systems that operate with R12 refrigerant is the Schrader valve. It is similar to the valve used to inflate tires. (*Figure 4-25*) A central valve core seats and unseats by depressing a stem attached to it. A pin in the servicing hose fitting is designed to do this when screwed onto the valve's exterior threads. All aircraft service valves should be capped when not in use.

R134a systems use valves that are very similar to the Schrader valve in function, operation, and location. As a safety device to prevent inadvertent mixing of refrigerants, R134a valve fittings are different from Schrader valve fittings and do not attach to Schrader valve threads. The R134a valve fittings are a quick disconnect type. Another type of valve called a

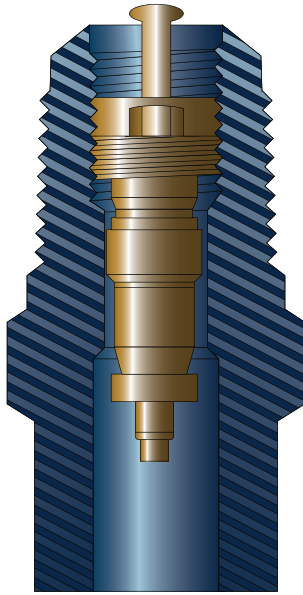


Figure 4-25. Cross-section of an R12 refrigerant service valve.

compressor isolation valve is used on some aircraft. It serves two purposes. Like the Schrader valve, it permits servicing the system with refrigerant. It also can isolate the compressor so the oil level can be checked and replenished without opening the entire system and losing the refrigerant charge. These valves are usually hard mounted to the inlet and outlet of the compressor.

A compressor isolation valve has three positions. When fully open, it back seats and allows the normal flow of refrigerant in the vapor cycle. When fully closed or front seated, the valve isolates the compressor from the rest of the system and servicing with oil, or even replacement of the compressor, is possible without losing the refrigerant charge. When in an intermediate position, the valve allows access to the system for servicing. The system can be operated with the valve in this position, but should be back seated for normal operation. The valve handle and service port should be capped when servicing is complete. (*Figure 4-26*)

VAPOR CYCLE AIR CONDITIONING SERVICING EQUIPMENT

Special servicing equipment is used to service vapor cycle air conditioning systems. The U.S. Environmental Protection Agency (EPA) has declared it illegal to release R12 refrigerant into the atmosphere. Equipment has been designed to capture the refrigerant during the servicing process. Although R134a does not have this restriction, it is illegal in some locations to release it to the atmosphere, and it may become universally so in the

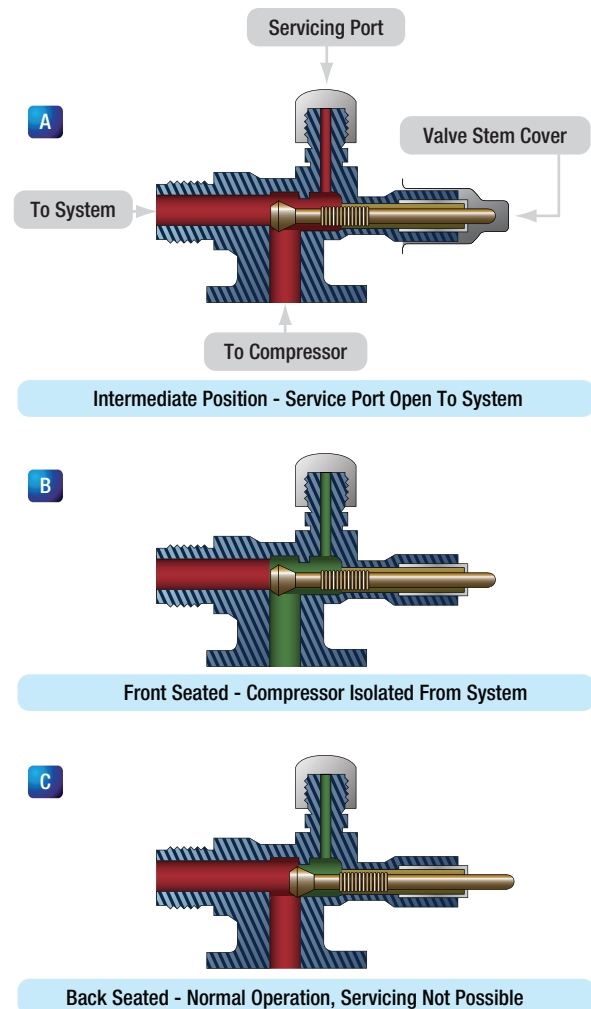


Figure 4-26. Compressor isolation valves isolate the compressor for maintenance or replacement. They also allow normal operation and servicing of the vapor cycle air conditioning system with refrigerant.

near future. It is good practice to capture all refrigerants for future use, rather than to waste them or to harm the environment by releasing them into the atmosphere. Capturing the refrigerant is a simple process designed into the proper servicing equipment. The technician should always be vigilant to use the approved refrigerant for the system being serviced and should follow all manufacturer's instructions.

MANIFOLD SET, GAUGES, HOSES, AND FITTINGS

In the past, the main servicing device for vapor cycle air conditioning systems was the manifold set. It contains three hose fittings, two O-ring sealed valves, and two gauges. It is essentially a manifold into which the gauges, fittings, and valves are attached. The valves are positioned to connect or isolate the center hose with either fitting.

Hoses attach to the right and left manifold set fittings and the other ends of those hoses attach to the service valves in the vapor cycle system. The center fitting also has a hose attached to it. The other end of this hose connects to either a refrigerant supply or a vacuum pump, depending on the servicing function to be performed. All servicing operations are performed by manipulating the valves. (*Figure 4-27*)

The gauges on the manifold set are dedicated - one for the low side of the system and the other for the high side. The low-pressure gauge is a compound gauge that indicates pressures above or below atmospheric pressure (0 gauge pressure). Below atmospheric pressure, the gauge is scaled in inches of mercury down to 30 inches. This is to indicate vacuum. 29.92 inches equals an absolute vacuum (absolute zero air pressure). Above atmospheric pressure, gauge pressure is read in psi. The scale typically ranges from 0 to 60 psi, although some gauges extend up to 150 psi. The high-pressure gauge usually has a range from zero up to about 500 psi gauge pressure. It does not indicate vacuum (pressure lower than atmospheric). These gauges and their scales can be seen in *Figure 4-28*.

The low-pressure gauge is connected on the manifold directly to the low side fitting. The high-pressure gauge connects directly to the high side fitting. The center fitting of the manifold can be isolated from either of the gauges or the high and low service fittings by the hand valves. When these valves are turned fully clockwise, the center fitting is isolated. If the low pressure valve is opened (turned counterclockwise), the center fitting is opened to the low pressure gauge and the low side service line. The same is true for the high side when the high pressure valve is opened. (*Figure 4-28*)

Special hoses are attached to the fittings of the manifold valve for servicing the system. The high pressure charging hose is usually red and attaches to the service valve located in the high side of the system. The low pressure hose, usually blue, attaches to the service valve that is located in the low side of the system. The center hose attaches to the vacuum pump for evacuating the system, or to the refrigerant supply for charging the system. Proper charging hoses for the refrigerant specific service valves must be used. When not using the manifold set, be sure the hoses are capped to prevent moisture from contaminating the valves.



Figure 4-27. A basic manifold set for servicing a vapor cycle air conditioning system.

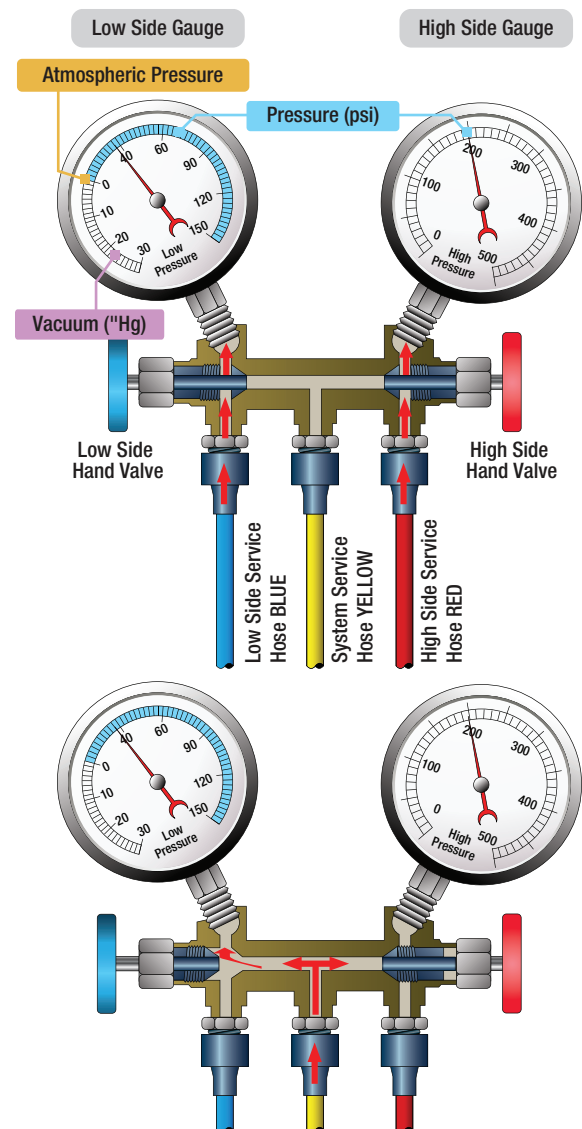


Figure 4-28. The internal workings of a manifold set with the center fitting isolated (top). Opening a valve connects the center hose to that side of the system and the gauge (bottom).

FULL SERVICE REFRIGERANT RECOVERY, RECYCLING, EVACUATION, AND RECHARGING UNITS

Regulations that require capture of all vapor cycle refrigerant have limited the use of the manifold set. It can still be used to charge a system. The refrigerant container is attached to the center hose and the manifold set valves are manipulated to allow flow into the low or high side of the system as required. But, emptying a system of refrigerant requires a service unit made to collect it. Allowing the refrigerant to flow into a collection container attached to the center hose will not capture the entire refrigerant charge, as the system and container pressures equalize above atmospheric pressure. An independent compressor and collection system is required.

Modern refrigeration recharging and recovery units are available to perform all of the servicing functions required for vapor cycle air conditioning systems. These all-in-one service carts have the manifold set built into the unit. As such, the logic for using a manifold set still applies. Integral solenoid valves, reservoirs, filters, and smart controls allow the entire servicing procedure to be controlled from the unit panel once the high side and low side services hoses are connected. A built-in compressor enables complete system refrigerant purging. A built-in vacuum pump performs system evacuation. A container and recycling filters for the refrigerant and the lubricating oil allow total recovery and recycling of these fluids. The pressure gauges used on the service unit panel are the same as those on a manifold set. Top-of-the-line units have an automatic function that performs all of the servicing functions sequentially and automatically once the hoses are hooked up to the vapor cycle air conditioning system and the system quantity of refrigerant has been entered. (Figure 4-29)

REFRIGERANT SOURCE

R134a comes in containers measured by the weight of the refrigerant they hold. Small 12-ounce to 2½-pound cans are common for adding refrigerant. Larger 30 and 50 pound cylinders equipped with shutoff valves are often used to charge an evacuated system, and they are used in shops that service vapor cycle systems frequently. (Figure 4-30)



Figure 4-29. A modern refrigerant recovery/recycle/charging service unit. Electronic control of solenoid activated valves combine with a builtin system for recovering, recycling, and recharging. A built-in vacuum pump and heated refrigerant reservoir are also included.



Figure 4-30. A 30 pound R134a refrigerant container with dual fittings. The fitting controlled by the blue valve wheel opens to the vapor space above the liquid refrigerant for connection to the low side of the vaporcycle system. The fitting controlled by the red valve wheel draws liquid refrigerant from the bottom of the cylinder through a stand tube. This fitting is connected to the system high side. On containers without dual fittings, the container must be inverted to deliver liquid refrigerant through a connected hose.

These larger cylinders are also used in the full servicing carts described above. The amount of refrigerant required for any system is measured in pounds. Check the manufacturer's service data and charge the system to the level specified using only the approved refrigerant from a known source.

VACUUM PUMPS

Vacuum pumps used with a manifold set, or as part of a service cart, are connected to the vapor cycle system so that the system pressure can be reduced to a near total vacuum. The reason for doing this is to remove all of the water in the system. As mentioned, water can freeze, causing system malfunction and can also combine with the refrigerant to create corrosive compounds.

Once the system has been purged of its refrigerant and it is at atmospheric pressure, the vacuum pump is operated. It gradually reduces the pressure in the system. As it does, the boiling point of any water in the system is also reduced. Water boils off or is vaporized under the reduced pressure and is pulled from the system by the pump, leaving the system moisture free to be recharged with refrigerant. (*Figure 4-31*)

The strength and efficiency of vacuum pumps varies as does the amount of time to hold the system at reduced pressure specified by manufacturers. Generally, the best established vacuum is held for 15-30 minutes to ensure all water is removed from the system. Follow the manufacturer's instructions when evacuating a vapor cycle air conditioning system. (*Figure 4-32*)

LEAK DETECTORS

Even the smallest leak in a vapor cycle air conditioning system can cause a loss of refrigerant. When operating normally, little or no refrigerant escapes. A system that requires the addition of refrigerant should be suspected of having a leak.

Electronic leak detectors are safe, effective devices used to find leaks. There are many types available that are able to detect extremely small amounts of escaped refrigerant. The detector is held close to component and hose connections where most leaks occur. Audible and visual alarms signal the presence of refrigerant. A detector specified for the type of refrigerant in the system should be chosen. A good leak detector is sensitive enough to detect leaks that would result in less than 1/2-ounce of refrigerant to be lost per year. (*Figure 4-33*)

Other leak detection methods exist. A soapy solution can also be applied to fittings and inspected for the formation of bubbles indicating a leak. Special leak detection dyes compatible for use with refrigerant can be injected into the vapor cycle system and can be seen when

Inches of Vacuum on Low Side Gauge (inches Hg)	Temperature at Which Water Boils (°C)	Absolute Pressure (psi)
0	100	14.696
4.92	96.1	12.279
9.23	90	10.152
15.94	80	6.866
20.72	70	4.519
24.04	60	2.888
26.28	50	1.788
27.75	40	1.066
28.67	30	0.614
28.92	26.7	0.491
29.02	24.4	0.442
29.12	22.2	0.393
29.22	20.6	0.344
29.32	17.8	0.295
29.42	15	0.246
29.52	11.7	0.196
29.62	7.2	0.147
29.74	0	0.088
29.82	-6.1	0.0049
29.87	-14.4	0.00245
29.91	-31.1	0.00049

Figure 4-31. When the temperature is low, a greater amount of vacuum is needed to boil off and remove any water in the vapor cycle system.



Figure 4-32. A vacuum pump is used to lower the pressure in the vapor cycle air conditioning system. This reduces the boiling point of water in the system, which vaporizes and is drawn out by the pump.



Figure 4-33. This electronic infrared leak detector can detect leaks that would lose less than 1/4 ounce of refrigerant per year.

they are forced out at a leak. Many of these are made to be visible under UV light. Occasionally, a leak can be detected upon close visual inspection. Oil in the system can be forced out of a leak, leaving a visible residue that is usually on the bottom side of a leaky fitting.

Old hoses may become slightly porous and leak a significant amount of refrigerant over time. Because of the length and area through which the refrigerant is lost, this type of leak may be difficult to detect, even with leak detecting methods. Visibly deteriorated hoses should be replaced.

SYSTEM SERVICING

Vapor cycle air conditioning systems can give many hours of reliable, maintenance-free service. Periodic visual inspections, tests, and refrigerant level and oil level checks may be all that is required for some time. Follow the manufacturer's instructions for inspection criteria and intervals.

VISUAL INSPECTION

All components of any vapor cycle system should be checked to ensure they are secure. Be vigilant for any damage, misalignment, or visual signs of leakage. The evaporator and condenser fins should be checked to ensure they are clean, unobstructed, and not folded over from an impact. Dirt and inhibited airflow through the fins can prevent effective heat exchange to and from the refrigerant. Occasionally, these units can be washed. Since the condenser often has ram air ducted to it or extends into the airstream, check for the presence of debris that may restrict airflow. Hinged units should be checked for security and wear. The mechanism to extend and retract the unit should function as specified, including the throttle position switch present on many systems. It is designed to cut power to the compressor clutch and retract the condenser at full power settings. Condensers may also have a fan to pull air over them during ground operation. It should be checked to ensure it functions correctly. (*Figure 4-34*)

Be sure the capillary temperature feedback sensor to the expansion valve is securely attached to the evaporator outlet. Also, check the security of the pressure sensor and thermostat sensor if the system has them. The evaporator should not have ice on the outside.

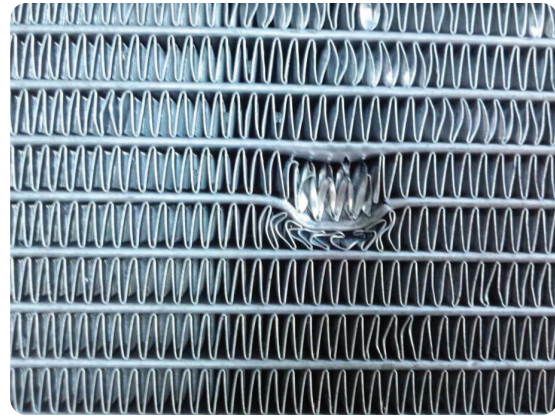


Figure 4-34. Damaged fins on a condenser.

This prevents proper heat exchange to the refrigerant from the warm cabin air blown over the unit. The fan blower should be checked to ensure it rotates freely. Depending on the system, it should run whenever the cooling switch is selected and should change speeds as the selector is rotated to more or less cooling. Sometimes systems low on refrigerant can cause ice on the evaporator, as can a faulty expansion valve or feedback control line. Ice formation anywhere on the outside of a vapor cycle air conditioning system should be investigated for cause and corrected. (*Figure 4-35*)

Security and alignment of the compressor is critical and should be checked during inspection. Belt-driven compressors need to have proper belt tension to function properly. Check the manufacturer's data for information on how to determine the condition and tension of the belt, as well as how to make adjustments. Oil level should be sufficient. Typically, 1/4 ounce of oil is added for each pound of refrigerant added to the system. When changing a component, additional oil may need to be



Figure 4-35. Ice on the evaporator coils is cause for investigation. It prevents proper heat exchange to the refrigerant.

added to replace that which is trapped in the replaced unit. Always use the oil specified in the manufacturer's maintenance manual.

LEAK TEST

As mentioned under the leak detector section above, leaks in a vapor cycle air conditioning system must be discovered and repaired. The most obvious sign of a possible leak is a low refrigerant level. Bubbles present in the sight glass of the receiver dryer while the system is operating indicate more refrigerant is needed. A system check for a leak may be in order. Note that vapor cycle systems normally lose a small amount of refrigerant each year. No action is needed if this amount is within limits.

Occasionally, all of the refrigerant escapes from the system. No bubbles are visible in the sight glass, but the complete lack of cooling indicates the refrigerant has leaked out. To locate the leak point, the system needs to be partially charged with refrigerant so leak detection methods can be employed. About 50 psi of refrigerant in the high and low sides should be sufficient for a leak check. By introducing the refrigerant into the high side, pressure indicated on the low side gauge verifies the orifice in the expansion valve is not clogged. When all refrigerant is lost due to a leak, the entire system should be checked. Each fitting and connection should be inspected visually and with a leak detector.

When a vapor cycle air conditioning system loses all of its refrigerant charge, air may enter the system. Water may also enter since it is in the air. This means that a full system evacuation must be performed after the leak is found and repaired. By establishing only a 50 psi charge in a depleted system, the leak(s) becomes detectable, but time and refrigerant are not wasted prior to evacuation. System evacuation is discussed below.

PERFORMANCE TEST

Verification of proper operation of a vapor cycle air conditioning system is often part of a performance test. This involves operating the system and checking parameters to ensure they are in the normal range. A key indication of performance is the temperature of the air that is cooled by the evaporator. This can be measured at the air outflow from the evaporator or at a nearby delivery duct outlet. An ordinary thermometer should read 40-50°F, with the controls set to full cold after the system has been allowed to operate for a few minutes.

Manufacturer's instructions include information on where to place the thermometer and the temperature range that indicates acceptable performance.

Pressures can also be observed to indicate system performance. Typically, low side pressure in a vapor cycle system operating normally is 10-50 psi, depending on ambient temperature. High side pressure is between 125 and 250 psi, again, depending on ambient temperature and the design of the system. All system performance tests are performed at a specified engine rpm (stable compressor speed) and involve a period of time to stabilize the operation of the vapor cycle. Consult the manufacturer's instructions for guidance.

FEEL TEST

A quick reference field test can be performed on a vapor cycle air conditioning system to gauge its health. In particular, components and lines in the high side (from the compressor to the expansion valve) should be warm or hot to the touch. The lines on both sides of the receiver dryer should be the same temperature. Low side lines and the evaporator should be cool. Ice should not be visible on the outside of the system. If any discrepancies exist, further investigation is needed. On hot, humid days, the cooling output of the vapor cycle system may be slightly compromised due to the volume of water condensing on the evaporator.

PURGING THE SYSTEM

Purging the system means emptying it of its refrigerant charge. Since the refrigerant must be captured, a service cart with this capability should be used. By connecting the hoses to the high side and low side service valves and selecting recover, cart solenoid valves position so that a system purging compressor pumps the refrigerant out of the vapor cycle system and into a recovery tank.

Vapor cycle systems must be properly purged before opening for maintenance or component replacement. Once opened, precautions should be taken to prevent contaminants from entering the system. When suspicion exists that the system has been contaminated, such as when a component has catastrophically failed, it can be flushed clean. Special fluid flush formulated for vapor cycle air conditioning systems should be used. The receiver dryer is removed from the system for flushing and a new unit is installed, as it contains fresh filters. Follow the aircraft manufacturer's instructions.

CHECKING COMPRESSOR OIL

The compressor is a sealed unit in the vapor cycle system that is lubricated with oil. Any time the system is purged, it is an opportunity to check the oil quantity in the compressor crankcase. This is often done by removing a filler plug and using a dip stick. Oil quantity should be maintained within the proper range using oil recommended by the manufacturer. Be certain to replace the filler plug after checking or adding oil. (*Figure 4-36*)

EVACUATING THE SYSTEM

Only a few drops of moisture can contaminate a vapor cycle air conditioning system. If this moisture freezes in the expansion valve, it could completely block the refrigerant flow. Water is removed from the system by evacuation. Anytime the system refrigerant charge falls below atmospheric pressure, the refrigerant is lost, or the system is opened, it must be evacuated before recharging.

Evacuating a vapor cycle air conditioning system is also known as pumping down the system. A vacuum pump is connected and pressure inside the system is reduced to vaporize any water that may exist. Continued operation of the vacuum pump draws the water vapor from the system. A typical pump used for evacuating an air conditioning system can reduce system pressure to about 29.62 "Hg (gauge pressure). At this pressure, water boils at 7.2°C. Operate the vacuum pump to achieve the recommended gauge pressure. Hold this vacuum for as long as the manufacturer specifies.

As long as a vapor cycle air conditioning system retains a charge higher than atmospheric pressure, any leak forces refrigerant out of the system. The system pressure prevents air (and water vapor) from entering. Therefore, it is permissible to recharge or add refrigerant to a system that has not dropped below atmospheric pressure without evacuating the system.

CHARGING THE SYSTEM

Charging capacity of a vapor cycle air conditioning system is measured by weight. The aircraft manufacturer's maintenance manual specifies this amount and the amount and type of oil to be put into the system when filling. Preweighing the refrigerant or setting the refrigerant weight into the servicing cart input ensures the system is filled to capacity.

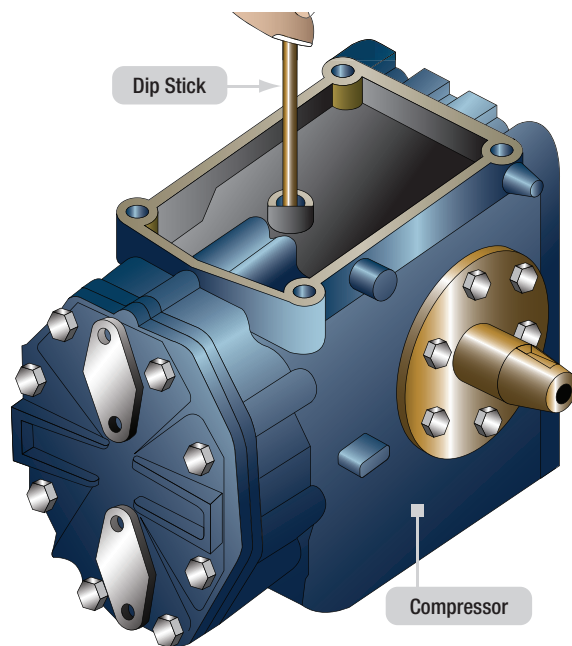


Figure 4-36. Checking the compressor oil when the system is open.

Charging a vapor cycle air conditioning system should be undertaken immediately after evacuation of the system is completed. With the hoses still connected to the high and low side service valves, selecting charge on the service cart panel positions solenoid operated valves so that the refrigerant supply is available. First, refrigerant is released into the high side of the system. Observe the low side gauge. When the low side gauge begins to indicate pressure, it is known that refrigerant is passing through the tiny orifice in the expansion valve. As pressure builds in the high side, the flow of refrigerant into the system stops.

To complete the charge of the system, refrigerant needs to be drawn in by the compressor. A major concern is to avoid damage to the compressor by having liquid refrigerant enter the compressor inlet. After the initial release of refrigerant into the high side, the high side service valve is closed and the remaining charge is made through the low side service valve. The engine is started and run at a specified rpm, usually a high idle speed. Full cool is selected on the air conditioning control panel in the cockpit. As the compressor operates, it draws vapor into the low side until the correct weighed amount of refrigerant is in the system. Charging is completed with a full performance test.

Charging with a manifold set is accomplished in the same way. The manifold center hose is connected to the refrigerant source that charges the system. After opening

the valve on the container (or puncturing the seal on a small can), the center hose connection on the manifold set should be loosened to allow air in the hose to escape. Once the air is bled out of the hose, the refrigerant can enter the system through whichever service valve is opened. The sequence is the same as above and all manufacturer instructions should be followed.

Oil quantity added to the system is specified by the manufacturer. Refrigerant premixed with oil is available and may be permissible for use. This eliminates the need to add oil separately. Alternately, the amount of oil to be put into the system can be selected on the servicing cart. Approximately 1/4-ounce of oil for each pound of refrigerant is a standard amount; however, follow the manufacturer's specifications.

DISTRIBUTION SYSTEMS

Distribution of cabin air on pressurized aircraft is managed with a system of air ducts leading from the pressurization source into and throughout the cabin. Typically, air is ducted to and released from ceiling vents, where it circulates and flows out floor-level vents. The air then flows aft through the baggage compartments and under the floor area. It exits the pressure vessel through the outflow valve(s) mounted low, on, or near the aft pressure bulkhead. The flow of air is nearly imperceptible. Ducting is hidden below the cabin floor and behind walls and ceiling panels depending on the aircraft and system design. Valves to select pressurization air source, ventilating air, temperature trim air, as well as in line fans and jet pumps to increase flow in certain areas of the cabin, are all components of the air distribution system. Temperature sensors, overheat switches, and check valves are also common.

On turbine powered aircraft, temperature controlled air from the air conditioning system is the air that is used to pressurize the cabin. The final regulation of the temperature of that air is sometimes considered part of the distribution system. Mixing air conditioned air with bleed air in a duct or a mixing chamber allows the crew to select the exact temperature desired for the cabin. The valve for mixing is controlled in the cockpit or cabin by a temperature selector. Centralized manifolds from which air can be distributed are common. (*Figure 4-37*)

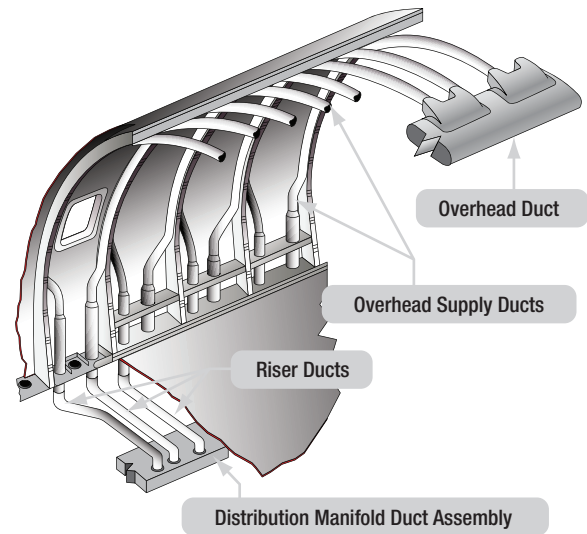


Figure 4-37. Centralized manifolds from which air can be distributed are common.

Large aircraft may be divided into zones for air distribution. Each zone has its own temperature selector and associated valve to mix conditioned and bleed air so that each zone can be maintain at a temperature independent of the others. The air distribution system on most aircraft makes provisions for ducting and circulating cooling air to electronics equipment bays. It also contains a gasper air system. This is air ducted from the cold air manifold or duct to an overhead adjustable delivery nozzle at each passenger station.

An inline fan controlled from the cockpit supplies a steady stream of gasper air that can be regulated or shut off with the delivery nozzle(s). (*Figure 4-38*)

FLOW CONTROL

Most commercial airliners have some sort of mass air flow control system. The parameters of flight constantly change. Adjustments to engine speeds affects bleed air flow. Output from the air conditioning pack varies. Altitude and air density vary. To ensure a stable mass of air flow through the cabin a mass airflow controller adjusts the flow to a predetermined level. Flow from the cabin air source is regulated to meet these demands. Flow of air leaving the cabin is also controlled. Mechanisms to control the flow of bleed air vary on different aircraft and different engines. During takeoff and climb, engine rpm is high. Air flow, temperature and pressure through the engine compressor are also high.

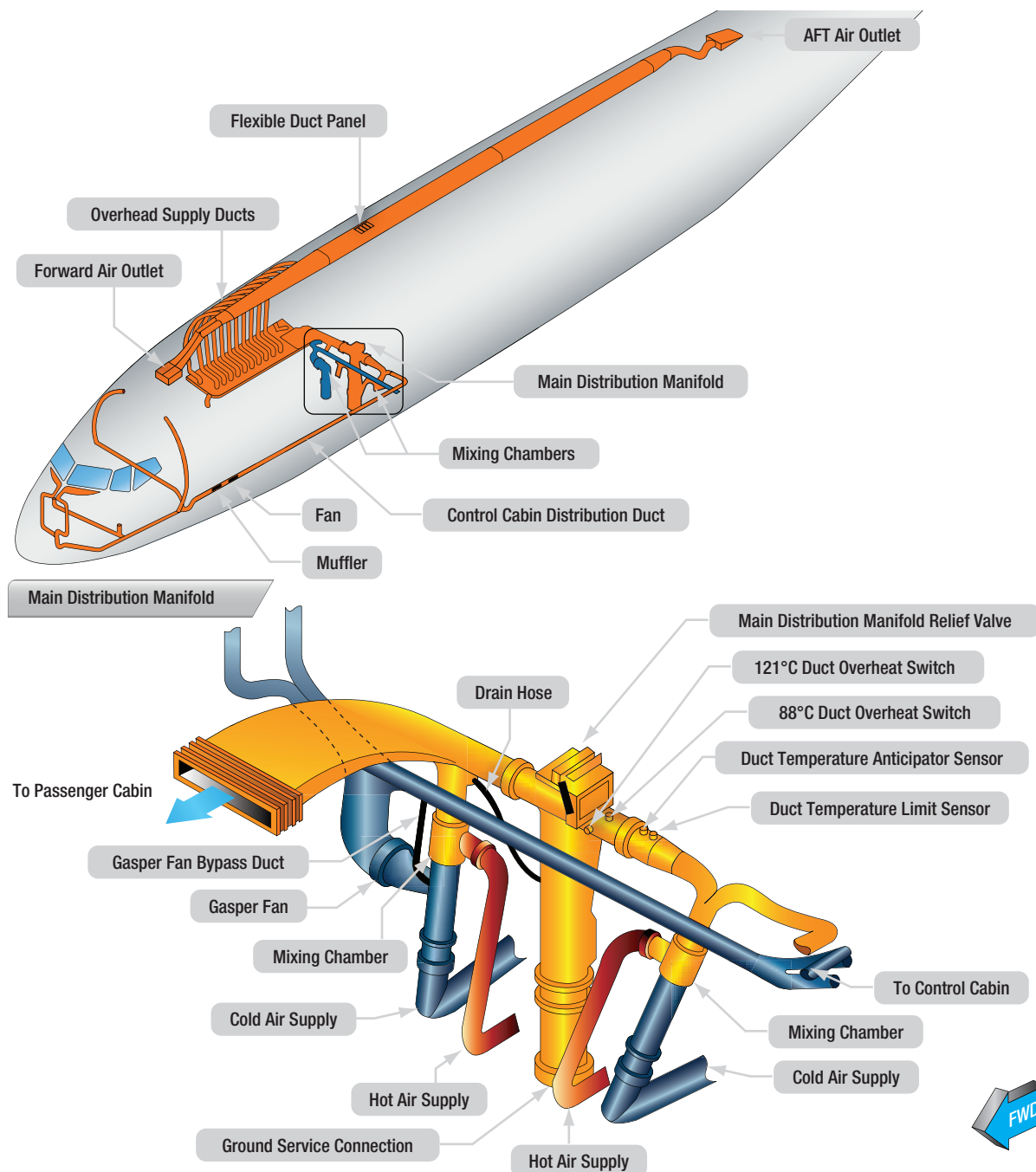


Figure 4-38. The conditioned air distribution system on a Boeing 737. The main distribution manifold is located under the cabin floor. Riser ducts run horizontally then vertically from the manifold to supply ducts, which follow the curvature of the fuselage carrying conditioned air to be released in the cabin.

Locating a bleed port at an early stage of compression provides sufficient bleed air for cabin environmental control systems and other needs. But as engine throttle settings are reduced during cruise and decent, a bleed port located on a higher stage of compression is required for the same volume and pressure of bleed air to be extracted. Bleed ports at three stages of the compressor is common on turbofan engines. Control of which bleed port delivers air to the pneumatic system is accomplished

in a variety of ways. Typically, a high pressure shutoff valve closes when low-stage bleed air pressure rises with throttle setting increase. This causes bleed air to automatically be extracted from a higher stage port.

TEMPERATURE CONTROL

Most cabin temperature control systems operate in a similar manner. Temperature is monitored in the cabin, cockpit, conditioned air ducts, and distribution air ducts.

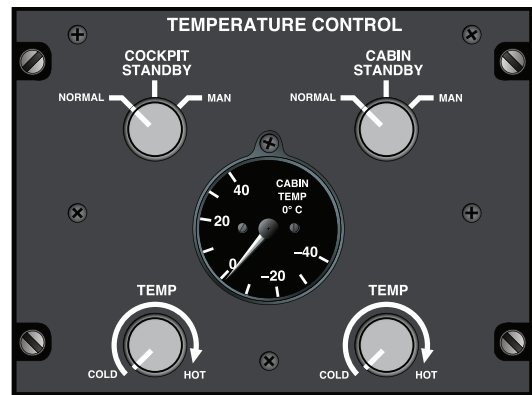


Figure 4-39. Typical temperature selectors on a transport category aircraft temperature control panel in the cockpit (left) and a business jet (right). On large aircraft, temperature selectors may be located on control panels located in a particular cabin air distribution zone.

These values are input into a temperature controller, or temperature control regulator, normally located in the electronics bay. A temperature selector in the cockpit can be adjusted to input the desired temperature. (Figure 4-39) The temperature controller compares the actual temperature signals received from the various sensors with the desired temperature input. Circuit logic for the selected mode processes these input signals. An output signal is sent to a valve in the air cycle air conditioning system.

This valve has different names depending on the aircraft manufacturer and design of the environmental control systems (i.e., mixing valve, temperature control valve, trim air valve). It mixes warm bleed air that bypassed the air cycle cooling process with the cold air produced by it. By modulating the valve in response to the signal from the temperature controller, air of the selected temperature is sent to the cabin through the air distribution system.

Cabin temperature pickup units and duct temperature sensors used in the temperature control system are thermistors. Their resistance changes as temperature changes. The temperature selector is a rheostat that varies its resistance as the knob is turned. In the temperature controller, resistances are compared in a bridge circuit. The bridge output feeds a temperature regulating function. An electric signal output is prepared and sent to the valve that mixes hot and cold air. On large aircraft with separate temperature zones, trim air modulating valves for each zone are used. The valves modulate to provide the correct mix required to match the selected temperature.

Cabin, flight deck, and duct temperature sensors are strategically located to provide useful information to control cabin temperature. (Figure 4-40)

HUMIDITY CONTROL

Control of cabin humidity primarily occurs during the air cycle air conditioning process. Moisture condensed from the cool air exiting the expansion turbine is removed by the water separator. Most water separators remove water by swilling the air in some fashion so that the heavier water can be drawn off.

However, at high altitudes, air from the air cycle air conditioners could be too dry for passenger and crew comfort. A water infiltration system can then be used to introduce moisture into the conditioned air. In a water infiltration system, the humidity level in the cabin is monitored. Water is pumped from a holding tank and sprayed into the conditioned air before it is directed into the cabin. An electronic humidity controller is used to signal the pump to operate.

PRESSURIZATION

PRESSURE OF THE ATMOSPHERE

The gases of the atmosphere (air), although invisible, have weight. A one square inch column of air stretching from sea level into space weighs 14.7 pounds. Therefore, it can be stated that the pressure of the atmosphere, or atmospheric pressure, at sea level is 14.7 psi. (Figure 4-41)

The weight exerted by a 1 square inch column of air stretching from sea level to the top of the atmosphere is what is measured when it is said that atmospheric pressure

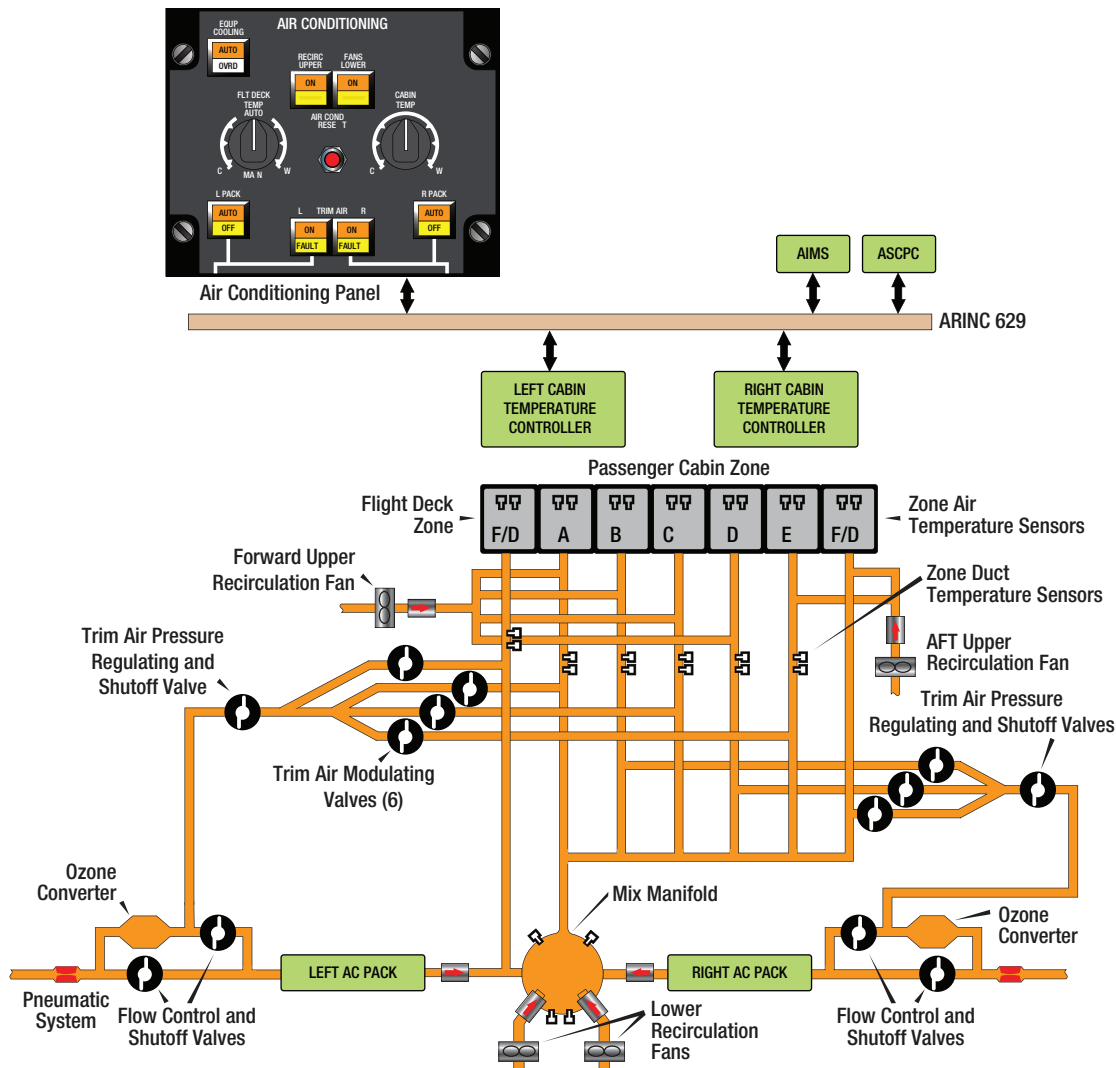


Figure 4-40. The temperature control system of a Boeing 777 combines the use of zone and duct temperature sensors with trim air modulating valves for each zone. Redundant digital left and right cabin temperature controllers process temperature input signals from the sensors and temperature selectors on the cockpit panel and throughout the aircraft to modulate the valves.

is equal to 14.7 pounds per square inch. Atmospheric pressure is also known as barometric pressure and is measured with a barometer. (*Figure 4-42*) Expressed in various ways, such as in inches of mercury or millimeters of mercury, these measurements come from observing the height of mercury in a column when air pressure is exerted on a reservoir of mercury into which the column is set. The column must be evacuated so air inside does not act against the mercury rising. A column of mercury 29.92 inches high weighs the same as a column of air that extends from sea level to the top of the atmosphere and has the same cross-section as the column of mercury.

Figure 4-42. The weight of the atmosphere pushes down on the mercury in the reservoir of a barometer, which causes mercury to rise in the column. At sea level, mercury is forced up into the column approximately

29.92 inches. Therefore, it is said that barometric pressure is 29.92 inches of mercury at sea level. Aviators often interchange references to atmospheric pressure between linear displacement (e.g., inches of mercury) and units of force (e.g., psi). Over the years, meteorology has shifted its use of linear displacement representation of atmospheric pressure to units of force. However, the unit of force nearly universally used today to represent atmospheric pressure in meteorology is the hectopascal (hPa). A hectopascal is a metric (SI) unit that expresses force in newtons per square meter. 1 013.2 hPa is equal to 14.7 psi. (*Figure 4-43*)

Atmospheric pressure decreases with increasing altitude. The simplest explanation for this is that the column of air that is weighed is shorter. How the pressure changes for a given altitude is shown in *Figure 4-44*.

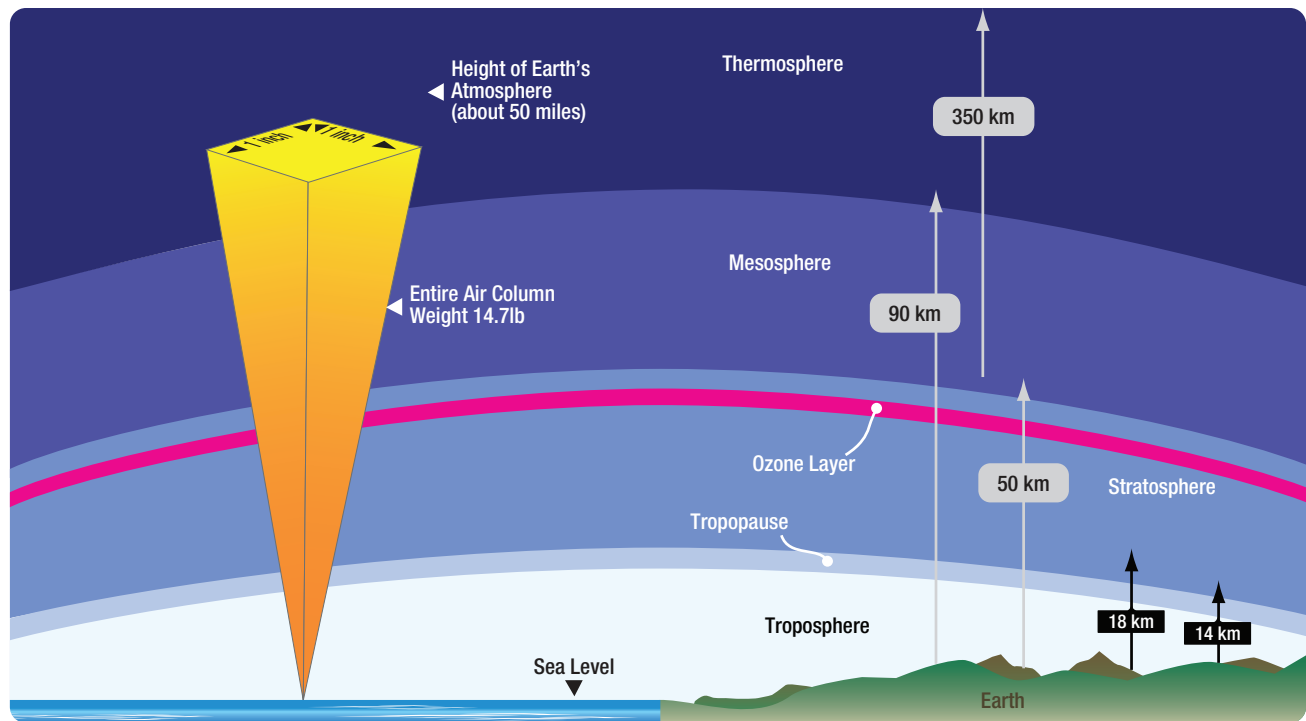


Figure 4-41. The weight exerted by a 1 square inch column of air stretching from sea level to the top of the atmosphere is what is measured when it is said that atmospheric pressure is equal to 14.7 pounds per square inch.

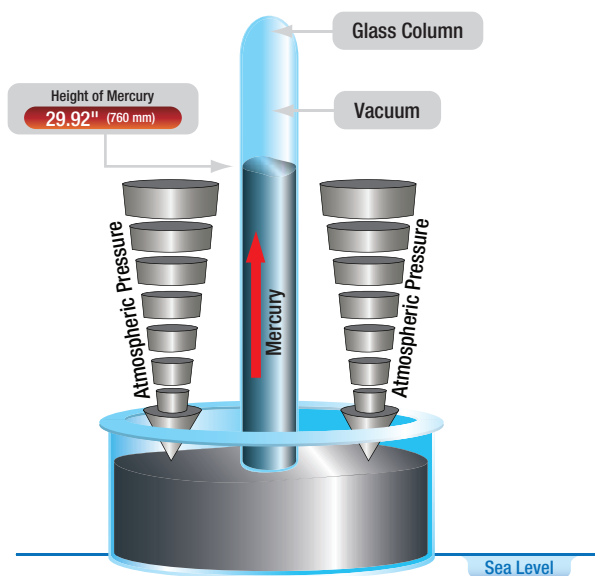


Figure 4-42. The weight of the atmosphere pushes down on the mercury in the reservoir of a barometer, which causes mercury to rise in the column. At sea level, mercury is forced up into the column approximately 29.92 inches. Therefore, it is said that barometric pressure is 29.92 inches of mercury at sea level.

The decrease in pressure is a rapid one and, at 50 000 feet, the atmospheric pressure has dropped to almost one tenth of the sea level value.

TEMPERATURE AND ALTITUDE

Temperature variations in the atmosphere are of concern to aviators. Weather systems produce changes in temperature near the earth's surface. Temperature also changes as altitude is increased. The troposphere is the lowest layer of the atmosphere. On average, it ranges from the earth's surface to about 38 000 feet above it. Over the poles, the troposphere extends to only 25 000 - 30 000 feet and, at the equator, it may extend to around 60 000 feet. This oblong nature of the troposphere is illustrated in *Figure 4-45*.

Most civilian aviation takes place in the troposphere in which temperature decreases as altitude increases. The rate of change is somewhat constant at about -2°C or -3.5°F for every 1 000 feet of increase in altitude. The upper boundary of the troposphere is the tropopause. It is characterized as a zone of relatively constant temperature of -57°C or -69°F . Above the tropopause lies the stratosphere. Temperature increases with altitude in the stratosphere to near 0°C before decreasing again in the mesosphere, which lies above it.

The stratosphere contains the ozone layer that protects the earth's inhabitants from harmful UV rays. Some civilian flights and numerous military flights occur in the stratosphere. *Figure 4-46* diagrams the temperature

Atmospheric Pressure

Standard atmospheric pressure at sea level is also known as 1 atmosphere, or 1 atm. The following measurements of standard atmospheric pressure are all equal to each other.

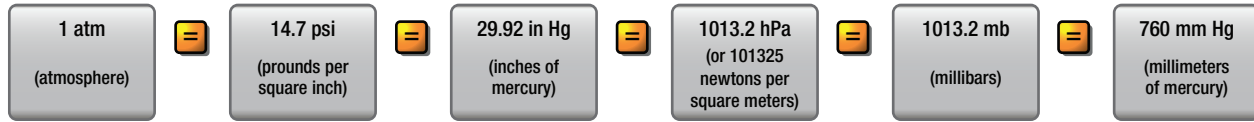


Figure 4-43. Various equivalent representations of atmospheric pressure at sea level.

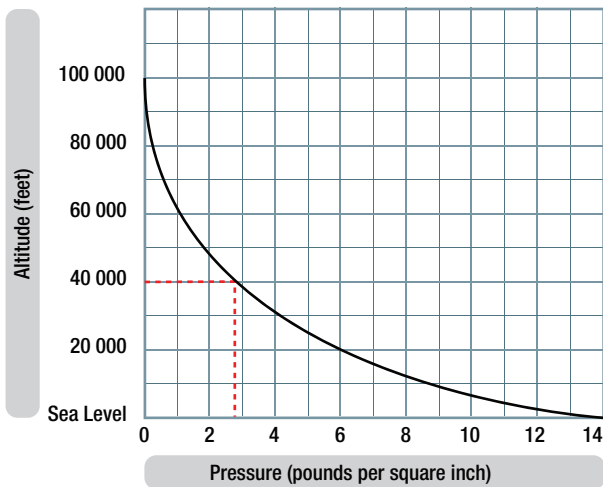


Figure 4-44. Atmospheric pressure decreasing with altitude.

At sea level the pressure is 14.7 psi, while at 40 000 feet, as the dotted lines show, the pressure is only 2.72 psi.

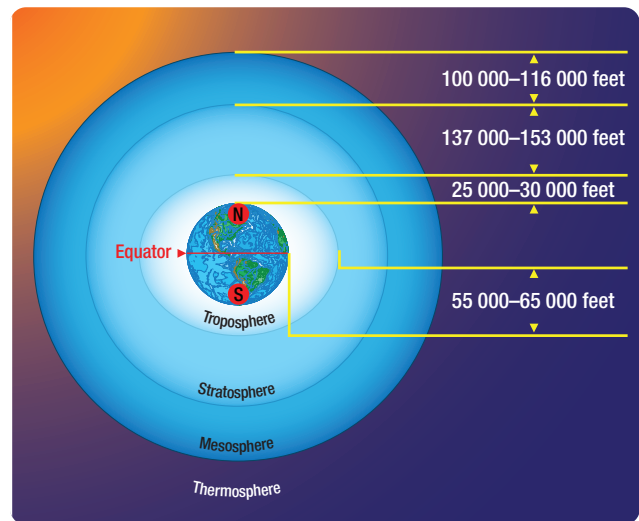


Figure 4-45. The troposphere extends higher above the earth's surface at the equator than it does at the poles.

variations in different layers of the atmosphere. When an aircraft is flown at high altitude, it burns less fuel for a given airspeed than it does for the same speed at a lower altitude.

This is due to decreased drag that results from the reduction in air density. Bad weather and turbulence can also be avoided by flying in the relatively smooth air above storms and convective activity that occur in the lower troposphere. To take advantage of these efficiencies, aircraft are equipped with environmental systems to overcome extreme temperature and pressure levels. While supplemental oxygen and a means of staying warm suffice, aircraft pressurization and air conditioning systems have been developed to make high altitude flight more comfortable.

Figure 4-47 illustrates the temperatures and pressures at various altitudes in the atmosphere.

PRESSURIZATION TERMS

The following terms should be understood for the discussion of pressurization and cabin environmental systems that follows:

1. Cabin altitude - given the air pressure inside the cabin, the altitude on a standard day that has the same pressure as that in the cabin. Rather than saying the pressure inside the cabin is 10.92 psi, it can be said that the cabin altitude is 8 000 feet (MSL).
2. Cabin differential pressure - the difference between the air pressure inside the cabin and the air pressure outside the cabin. Cabin pressure (psi) - ambient pressure (psi) = cabin differential pressure (psid or ? psi).
3. Cabin rate of climb - the rate of change of air pressure inside the cabin, expressed in feet per minute (fpm) of cabin altitude change.

PRESSURIZATION SYSTEMS

Pressurizing an aircraft cabin assists in making flight possible in the hostile environment of the upper atmosphere. The degree of pressurization and the operating altitude of any aircraft are limited by critical design factors. A cabin pressurization system must accomplish several functions if it is to ensure adequate passenger comfort and safety. It must be capable of maintaining a cabin pressure altitude of approximately 8 000 feet or lower regardless of the cruising altitude of the aircraft. This is to ensure that passengers and crew have enough oxygen present at sufficient pressure to facilitate full blood saturation. A pressurization system must also be designed to prevent rapid changes of cabin pressure, which can be uncomfortable or injurious to passengers and crew. Additionally, a pressurization system should circulate air from inside the cabin to the outside at a rate that quickly eliminates odors and to remove stale air. Cabin air must also be heated or cooled on pressurized aircraft. Typically, these functions are incorporated into the pressurization source.

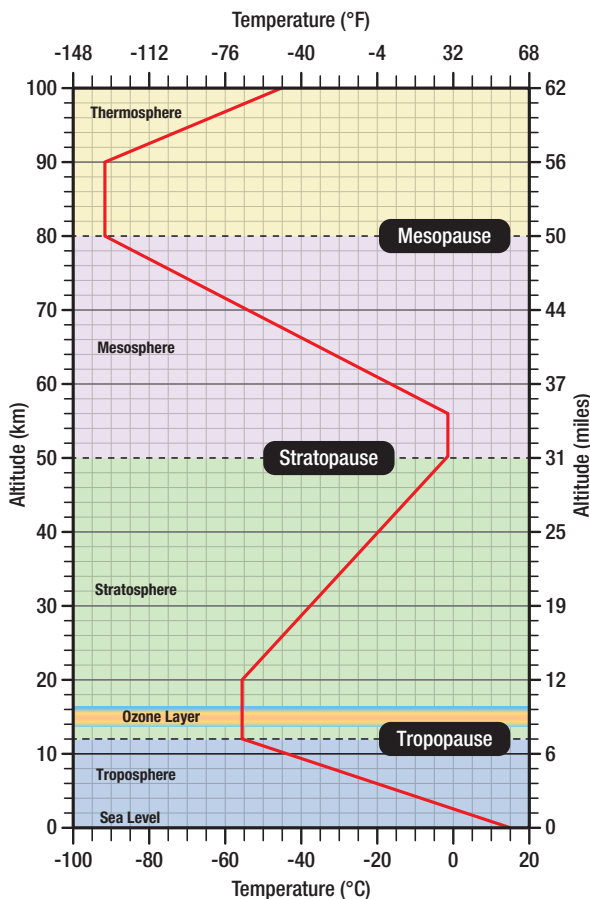


Figure 4-46. The atmospheric layers with temperature changes depicted by the red line.

To pressurize, a portion of the aircraft designed to contain air at a pressure higher than outside atmospheric pressure must be sealed. A wide variety of materials facilitate this. Compressible seals around doors combine with various other seals, grommets, and sealants to essentially establish an air tight pressure vessel. This usually includes the cabin, flight compartment, and the baggage compartments. Air is then pumped into this area at a constant rate sufficient to raise the pressure slightly above that which is needed. Control is maintained by adjusting the rate at which the air is allowed to flow out of the aircraft.

A key factor in pressurization is the ability of the fuselage to withstand the forces associated with the increase in pressure inside the structure versus the ambient pressure outside. This differential pressure can range from 3.5 psi for a single Engine reciprocating aircraft, to approximately 9 psi on high performance jet aircraft.

Altitude feet	Pressure			Temperature	
	psi	hPa	in Hg	°F	°C
0	14.69	1013.2	29.92	59.0	15
1 000	14.18	977.2	28.86	55.4	13
2 000	13.66	942.1	27.82	51.9	11
3 000	13.17	908.1	26.82	48.3	9.1
4 000	12.69	875.1	25.84	44.7	7.1
5 000	12.23	843.1	24.90	41.2	5.1
6 000	11.77	812.0	23.98	37.6	3.1
7 000	11.34	781.8	23.09	34.0	1.1
8 000	10.92	752.6	22.23	30.5	-0.8
9 000	10.51	724.3	21.39	26.9	-2.8
10 000	10.10	696.8	20.58	23.3	-4.8
12 000	9.34	644.4	19.03	16.2	-8.8
14 000	8.63	595.2	17.58	9.1	-12.7
16 000	7.96	549.2	16.22	1.9	-16.7
18 000	7.34	506.0	14.94	-5.2	-29.7
20 000	6.76	465.6	13.75	-12.3	-24.6
22 000	6.21	427.9	12.64	-19.5	-28.6
24 000	5.70	392.7	11.60	-26.6	-32.5
26 000	5.22	359.9	10.63	-33.7	-36.5
28 000	4.78	329.3	9.72	-40.9	-40.5
30 000	4.37	300.9	8.89	-48.0	-44.4
32 000	3.99	274.5	8.11	-55.1	-48.4
34 000	3.63	250.0	7.38	-62.2	-52.4
36 000	3.30	227.3	6.71	-69.4	-56.3
38 000	3.00	206.5	6.10	-69.4	-56.5
40 000	2.73	187.5	5.54	-69.4	-56.5
45 000	2.14	147.5	4.35	-69.4	-56.5
50 000	1.70	116.0	3.42	-69.4	-56.5

Figure 4-47. Cabin environmental systems establish conditions quite different from these found outside the aircraft.

(Figure 4-48) If the weight of the aircraft structure were of no concern, this would not be a problem. Making an aircraft strong for pressurization, yet also light, has been an engineering challenge met over numerous years beginning in the 1930s.

The development of jet aircraft and their ability to exploit low drag flight at higher altitude made the problem even more pronounced. Today, the proliferation of composite materials in aircraft structure continues this engineering challenge. In addition to being strong enough to withstand the pressure differential between the air inside and the air outside the cabin, metal fatigue from repeated pressurization and depressurization weakens the airframe. Some early pressurized aircraft structures failed due to this and resulted in fatal accidents.

Aging aircraft programs have been instituted to increase inspection scrutiny of older airframes that may show signs of fatigue due to the pressurization cycle. Aircraft of any size may be pressurized. Weight considerations when making the fuselage strong enough to endure pressurization usually limit pressurization to high performance light aircraft and larger aircraft. A few pressurized single-engine reciprocating aircraft exist, as well as many pressurized single engine turboprop aircraft.

CONTROL OF CABIN PRESSURE

PRESSURIZATION MODES

Aircraft cabin pressurization can be controlled via two different modes of operation. The first is the isobaric mode, which works to maintain cabin altitude at a single pressure despite the changing altitude of the aircraft.

For example, the flight crew may select to maintain a cabin altitude of 8 000 feet (10.92 psi). In the isobaric mode, the cabin pressure is established at the 8 000 foot level and remains at this level, even as the altitude of the aircraft fluctuates. The second mode of pressurization control is the constant differential mode, which controls cabin pressure to maintain a constant pressure difference between the air pressure inside the cabin and the ambient air pressure, regardless of aircraft altitude changes. The constant differential mode pressure differential is lower than the maximum differential pressure for which the airframe is designed, keeping the integrity of the pressure vessel intact.

When in isobaric mode, the pressurization system maintains the cabin altitude selected by the crew. This is the condition for normal operations. But when the aircraft climbs beyond a certain altitude, maintaining the selected cabin altitude may result in a differential pressure above that for which the airframe was designed.

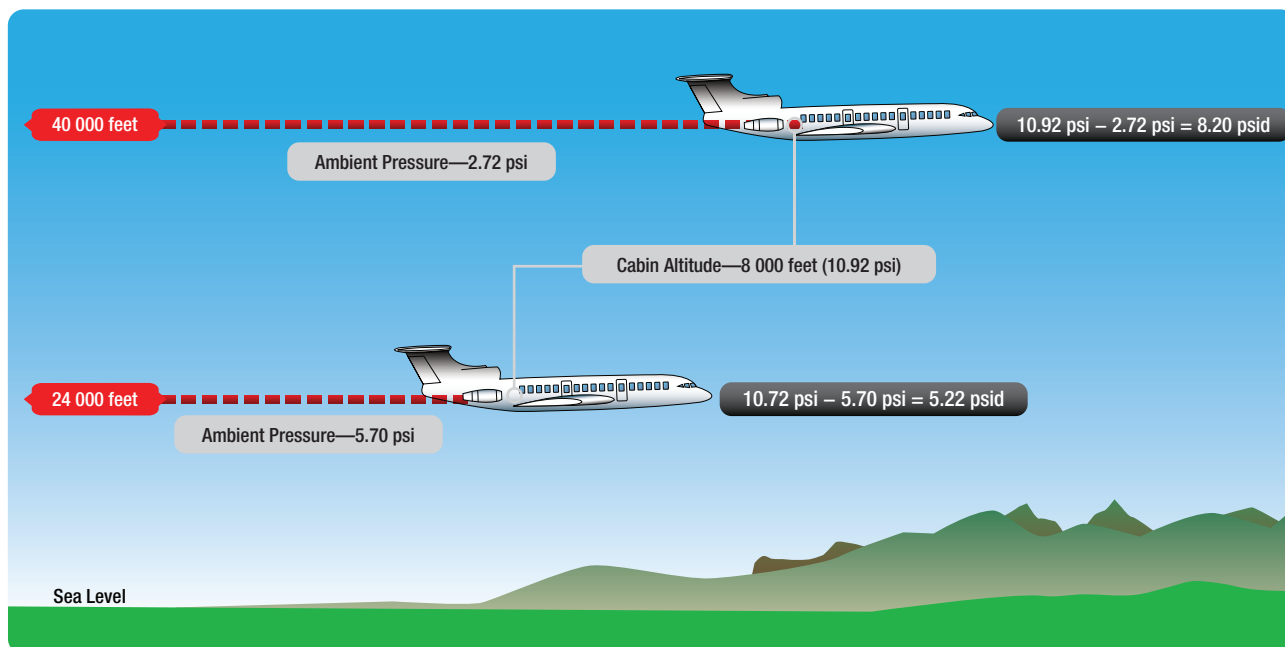


Figure 4-48. Differential pressure (psid) is calculated by subtracting the ambient air pressure from the cabin air pressure.

In this case, the mode of pressurization automatically switches from isobaric to constant differential mode. This occurs before the cabin's max differential pressure limit is reached. A constant differential pressure is then maintained, regardless of the selected cabin altitude. In addition to the modes of operation described above, the rate of change of the cabin pressure, also known as the cabin rate of climb or descent, is also controlled. This can be done automatically or manually by the flight crew. Typical rates of change for cabin pressure are 300 to 500 fpm. Also, note that modes of pressurization may also refer to automatic versus standby versus manual operation of the pressurization system.

CABIN PRESSURE CONTROLLERS

The cabin pressure controller is the device used to control the cabin air pressure. Older aircraft use strictly pneumatic means for controlling cabin pressure. Selections for the desired cabin altitude, rate of cabin altitude change, and barometric pressure setting are all made directly to the pressure controller from pressurization panel in the cockpit. (*Figure 4-49*)

Adjustments and settings on the pressure controller are the control input parameters for the cabin pressure regulator. The regulator controls the position of the outflow valve(s) normally located at the rear of the aircraft pressure vessel. Valve position determines

the pressure level in the cabin. Modern aircraft often combine pneumatic, electric, and electronic control of pressurization. Cabin altitude, cabin rate of change, and barometric setting are made on the cabin pressure selector of the pressurization panel in the cockpit. Electric signals are sent from the selector to the cabin pressure controller, which functions as the pressure regulator. It is remotely located out of sight near the cockpit but inside the pressurized portion of the aircraft. The signals are converted from electric to digital and are used by the controller. Cabin pressure and ambient pressure are also input to the controller, as well as other inputs. (*Figure 4-50*)

Using this information, the controller, which is essentially a computer, supplies pressurization logic for various stages of a flight. On many small transport and business jets, the controller's electric output signal drives a torque motor in the primary outflow valve. This modulates pneumatic airflow through the valve, which positions the valve to maintain the pressurization schedule.

On many transport category aircraft, two cabin pressure controllers, or a single controller with redundant circuitry, are used. Located in the electronics equipment bay, they receive electric input from the panel selector, as well as ambient and cabin pressure input. Flight altitude and landing field altitude information are often the crew selection choices on the pressurization control panel. Cabin altitude, rate of climb, and barometric setting are automatic through built-in logic and communication with the ADC and the flight management system (FMS). The controllers process the information and send electric signals to motors that directly position the outflow valve(s). (*Figure 4-51*)

Modern pressurization control is fully automatic once variable selections are made on the pressurization control panel if, in fact, there are any to be made. Entering or selecting a flight plan into the FMS of some aircraft automatically supplies the pressurization controller with the parameters needed to establish the pressurization schedule for the entire flight. No other input is needed from the crew.

All pressurization systems contain a manual mode that can override automatic control. This can be used in flight or on the ground during maintenance. The operator

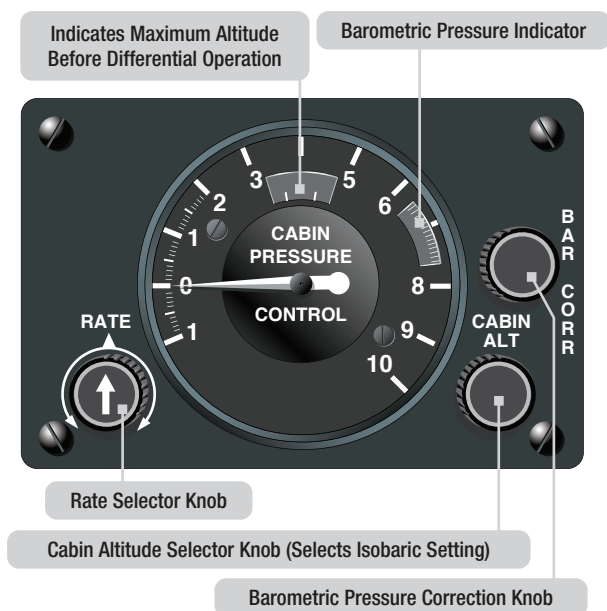


Figure 4-49. A pressure controller for an all pneumatic cabin pressure control system.

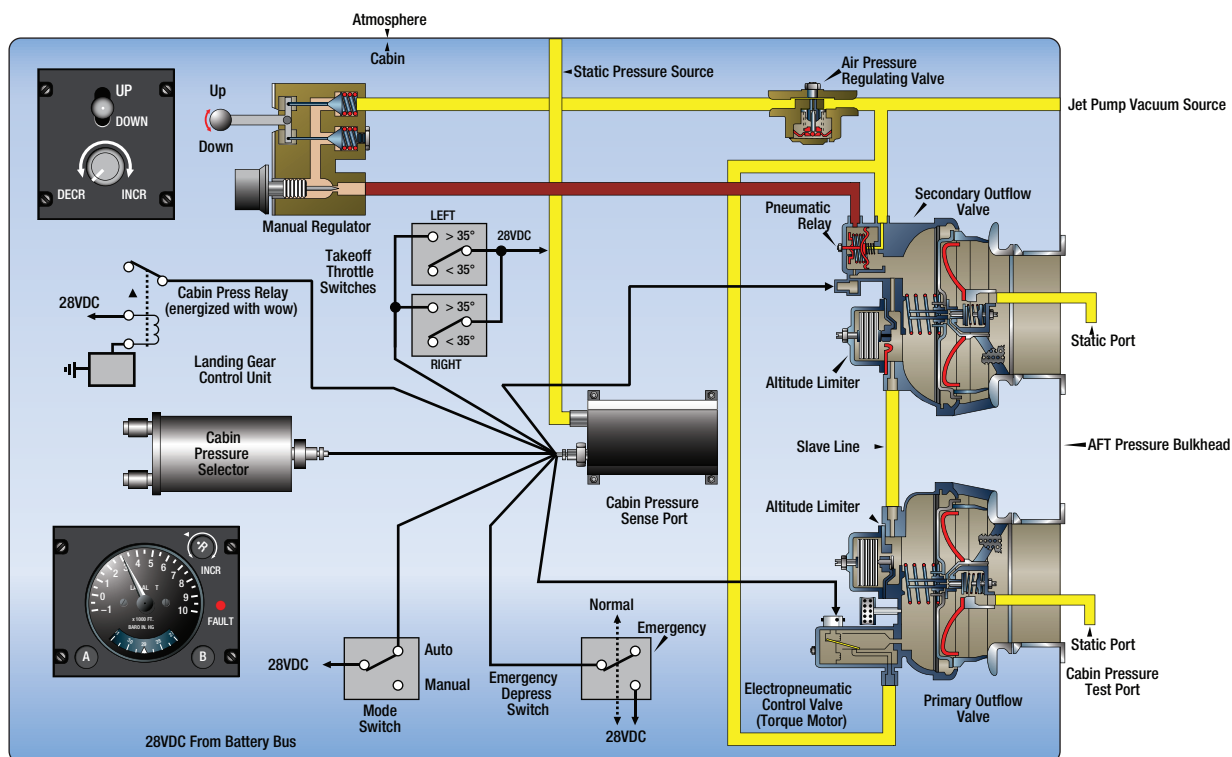


Figure 4-50. The pressurization control system on many small transports and business jets utilizes a combination of electronic, electric and pneumatic control elements.

selects the manual mode on the pressurization control panel. A separate switch is used to position the outflow valve open or closed to control cabin pressure. The switch is visible in *Figure 4-51*, as well as a small gauge that indicates the position of the valve.

CONTROL VALVE (OUTFLOW VALVE)

Controlling cabin pressurization is accomplished through regulating the amount of air that flows out

of the cabin. A cabin outflow valve opens, closes, or modulates to establish the amount of air pressure maintained in the cabin. Some outflow valves contain the pressure regulating and the valve mechanism in a single unit. They operate pneumatically in response to the settings on the cockpit pressurization panel that influence the balance between cabin and ambient air pressure. (*Figure 4-52*)



Figure 4-51. This pressurization panel from an 800 series Boeing 737 has input selections of flight altitude and landing altitude.

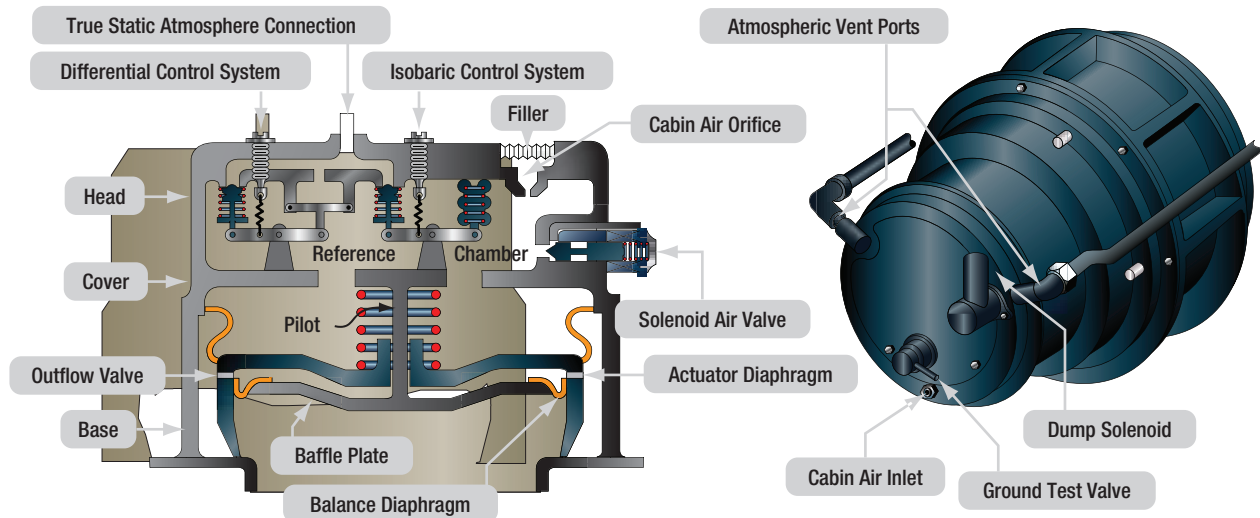


Figure 4-52. An all-pneumatic cabin pressure regulator and outflow valve.

Pneumatic operation of outflow valves is common. It is simple, reliable, and eliminates the need to convert air pressure operating variables into some other form. Diaphragms, springs, metered orifices, jet pumps, bellows, and poppet valves are used to sense and manipulate cabin and ambient air pressures to correctly position the outflow valve without the use of electricity. Outflow valves that combine the use of electricity with pneumatic operation have all-pneumatic standby and manual modes, as shown in *Figure 4-50*.

The pressure regulating mechanism can also be found as a separate unit. Many air transport category aircraft have an outflow valve that operates electrically, using signals sent from a remotely located cabin air pressure controller that acts as the pressure regulator. The controller positions the valve(s) to achieve the settings on the cockpit pressurization panel selectors according to predetermined pressurization schedules. Signals are sent to electric motors to move the valve as needed. On transports, often AC motors are used with a redundant DC motor for standby or manual operations. (*Figure 4-53*)

CABIN AIR PRESSURE SAFETY VALVE OPERATION

Aircraft pressurization systems incorporate various features to limit human and structural damage should the system malfunction or become inoperative. A means for preventing over pressurization is incorporated to ensure the structural integrity of the aircraft if control of the pressurization system is lost. A cabin air safety valve is a pressure relief valve set to open at a predetermined



Figure 4-53. This outflow valve on a transport category aircraft is normally operated by an ac motor controlled by a pressure controller in the electronics equipment bay. A second AC motor on the valve is used when in standby mode. A DC motor also on the valve is used for manual operation.

pressure differential. It allows air to flow from the cabin to prevent internal pressure from exceeding design limitations. *Figure 4-54* shows cabin air pressure safety valves on a large transport category aircraft. On most aircraft, safety valves are set to open between 8-10 psi.

Pressurization safety valves are used to prevent the over pressurization of the aircraft cabin. They open at a preset differential pressure and allow air to flow out of the cabin. Wide body transport category aircraft cabins may have more than one cabin pressurization safety valve.



Figure 4-54. Two pressurization safety valves on a Boeing 747.

Some outflow valves incorporate the safety valve function into their design. This is common on some corporate jets when two outflow valves are used. One outflow valve operates as the primary and the other as a secondary. Both contain a pilot valve that opens when the pressure differential increases to a preset value. This, in turn, opens the outflow valve(s) to prevent further pressurization. The outflow valves shown in **Figure 4-50** operate in this manner.

Cabin altitude limiters are also used. These close the outflow valves when the pressure in the cabin drops well below the normal cabin altitude range, preventing a further increase in cabin altitude. Some limiter functions are built into the outflow valve(s). An example of this can be seen in **Figure 4-50**. Other limiters are independent bellows units that send input to the outflow valve or are part of the cabin pressurization controller logic. A negative pressure relief valve is included on pressurized aircraft to ensure that air pressure outside the aircraft does not exceed cabin air pressure. The spring loaded relief valve opens inward to allow ambient air to enter the cabin when this situation arises. Too much negative pressure can cause difficulty when opening the cabin door. If high enough, it could cause structural damage since the pressure vessel is designed for cabin pressure to be greater than ambient.

Some aircraft are equipped with pressurization dump valves. These essentially are safety valves that are operated automatically or manually by a switch in the cockpit. They are used to quickly remove air and air pressure from the cabin, usually in an abnormal, maintenance, or emergency situation. Incorporation of an emergency pressurization mode is found on some

aircraft. A valve opens when the air conditioning packs fail or emergency pressurization is selected from the cockpit. It directs a mixture of bleed air and ram air into the cabin. This combines with fully closed outflow valves to preserve some pressurization in the aircraft.

PRESSURIZATION INDICATION

While all pressurization systems differ slightly, usually three cockpit indications, in concert with various warning lights and alerts, advise the crew of pressurization variables. They are the cabin altimeter, the cabin rate of climb or vertical speed indicator, and the cabin differential pressure indicator. These can be separate gauges or combined into one or two gauges. All are typically located on the pressurization panel, although sometimes they are elsewhere on the instrument panel. Outflow valve position indicator(s) are also common. (**Figure 4-55**)

On modern aircraft equipped with digital aircraft monitoring systems with LCD displays, such as Engine Indicating and Crew Alerting System (EICAS) or Electronic Centralized Aircraft Monitor (ECAM), the pressurization panel may contain no gauges. The environmental control system (ECS) page of the monitoring system is selected to display similar



Figure 4-55. This cabin pressurization gauge is a triple combination gauge. The long pointer operates identically to a vertical speed indicator with the same familiar scale on the left side of the gauge. It indicates the rate of change of cabin pressure. The orange PSI pointer indicates the differential pressure on the right side scale. The ALT indicator uses the same scale as the PSI pointer, but it indicates cabin altitude when ALT indicator moves against it.

information. Increased use of automatic redundancy and advanced operating logic simplifies operation of the pressurization system. It is almost completely automatic. The cabin pressurization panel remains in the cockpit primarily for manual control. (Figure 4-56)

PRESSURIZATION OPERATION

The normal mode of operation for most pressurization control systems is the automatic mode. A standby mode can also be selected. This also provides automatic control of pressurization, usually with different inputs, a standby controller, or standby outflow valve operation. A manual mode is available should the automatic and standby modes fail. This allows the crew to directly position the outflow valve through pneumatic or electric control, depending on the system.

Coordination of all pressurization components during various flight segments is essential. A weight-on-wheels (WOW) switch attached to the landing gear and a throttle position switch are integral parts of many pressurization control systems. During ground operations and prior to takeoff, the WOW switch typically controls the position of the pressurization safety valve, which is held in the open position until the aircraft takes off.

In an advanced system, the WOW switch may give input to the pressurization controller, which in turn controls the positions and operation of all pressurization components. In other systems, the WOW switch may

directly control the safety valve or a pneumatic source valve that causes the safety valve to be held open until the source is cut at takeoff when the WOW switch opens.

Throttle position switches can be used to cause a smooth transition from an unpressurized cabin to a pressurized cabin. A partial closing of the outflow valve(s) when the WOW switch is closed (on the ground) and the throttles are advanced gradually initiates pressurization during rollout. At takeoff, the rate of climb and the pressurization schedule require the outflow valve(s) to fully close. Passengers do not experience a harsh sensation from the fully closed valves because the cabin has already begun to pressurize slightly.

Once in flight, the pressurization controller automatically controls the sequence of operation of the pressurization components until the aircraft lands. When the WOW switch closes again at landing, it opens the safety valve(s) and, in some aircraft, the outflow valve(s) makes pressurizing impossible on the ground in the automatic pressurization mode. Maintenance testing of the system is done in manual mode. This allows the technician to control the position of all valves from the cockpit panel.

CABIN PRESSURIZATION TROUBLESHOOTING

While pressurization systems on different aircraft operate similarly with similar components, it cannot be assumed that they are the same. Even those systems constructed

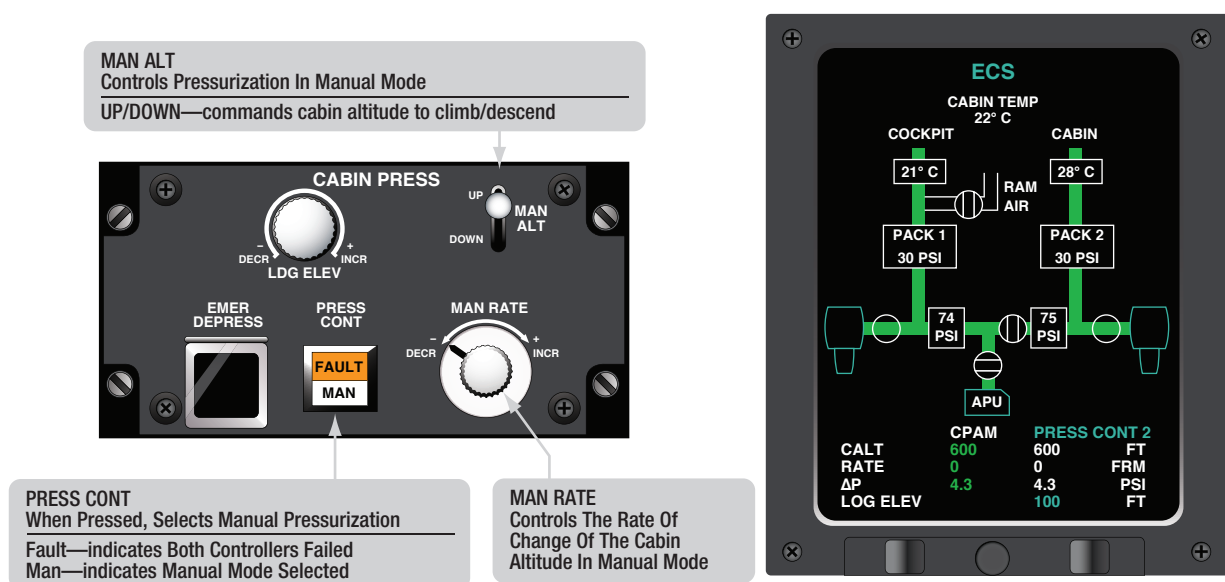


Figure 4-56. The pressurization panel and environmental control system page on a Bombardier CRJ200 50 passenger jet have no gauges. Traditional pressurization data is presented in digital format at the bottom of the page.

by a single manufacturer likely have differences when installed on different aircraft. It is important to check the aircraft manufacturer's service information when troubleshooting the pressurization system.

A fault, such as failure to pressurize or failure to maintain pressurization, can have many different causes. Adherence to the steps in a manufacturer's troubleshooting procedures is highly recommended to sequentially evaluate possible causes. Pressurization system test kits are available, or the aircraft can be pressurized by its normal sources during troubleshooting. A test flight may be required after maintenance.

Smoke detection is sometimes used in the cabin, cargo bays, lavatories and other areas of the aircraft with controlled operating environments. Warnings are indicated on the central warning system. Cabin altitude must be maintained in the range of human survival. Failure of the pressurization system could put this in jeopardy. Most pressurized aircraft are fitted with excess cabin altitude sensors. When cabin air pressure becomes insufficient at around 10 000 feet cabin altitude, a warning is annunciated on the central warning system. The warning may be aural or visual. Flight crew may immediately utilize emergency oxygen to maintain control of the aircraft and to avoid hypoxia.

SAFETY AND WARNING DEVICES

The correct functioning of cabin environmental systems is of obvious importance to passenger and crew safety. Various protection and warning devices are used to alert the crew of any malfunction that may require attention to insure safety.

PROTECTION AND WARNING DEVICES

One of the major concerns with pressurization and air conditioning systems is overheating. A leak anywhere in pneumatic system ducting can pose a fire hazard. Often a continuous loop fire detector will be run the length of pneumatic ducting or around a bay containing pneumatic lines such as the air conditioning and APU bays.

Warning annunciations occur in the cockpit on the central warning panel should an over heat caused by a pneumatic leak be detected. Typically the flow of pneumatic air in the indicated section of the pneumatic system is shut off. This is done by closing an isolation valve or shutoff valve. System redundancy permits safe operation with the remaining (functioning) portion of the pneumatic system.

Most air conditioning pack operation is protected by thermal switch monitoring. At the pack outlet before the air enters the distribution system, temperature detection switches shut down the pack should an overheat occur. The pack can typically be reset and operated again manually in case the overheat occurred due to a failure of the automatic temperature control system.

Question: 4-1

The volume of air for engine power production is _____ by bleeding air off of the compressor for air-conditioning and pressurization.

Question: 4-5

The thermostatic _____ has an adjustable orifice through which the correct amount of refrigerant is metered to obtain optimal cooling.

Question: 4-2

The heart of the air cycle air conditioning system is the refrigeration turbine unit, also known as the _____.

Question: 4-6

The outside air absorbs heat from the refrigerant flowing through the _____.

Question: 4-3

Vapor cycle air conditioning is a _____ system used solely for the transfer of heat from inside the cabin to outside of the cabin.

Question: 4-7

Servicing a vapor cycle air conditioning system is done with either a full service recovery, recycling, evacuation and recharging unit or a _____ set.

Question: 4-4

The low side of a vapor cycle air conditioning system has low _____ and low _____.

Question: 4-8

Water is removed from a vapor cycle air conditioning system by _____.

ANSWERS

Answer: 4-1
reduced.

Answer: 4-5
expansion valve.

Answer: 4-2
air cycle machine (ACM)

Answer: 4-6
condenser.

Answer: 4-3
closed.

Answer: 4-7
manifold.

Answer: 4-4
temperature.
pressure.

Answer: 4-8
evacuation (vacuuming down the system).

Question: 4-9

The temperature _____ compares the actual temperature signals received from the various sensors with the desired temperature input.

Question: 4-12

Controlling cabin pressurization is accomplished through regulating the amount of air that flows out of the cabin through the _____ valve.

Question: 4-10

The unit of force nearly universally used today to represent atmospheric pressure in meteorology is the _____.

Question: 4-13

Three pressurization parameters monitored on the flight deck include: cabin altitude, cabin rate of climb, and _____ pressure.

Question: 4-11

_____ is caused by repeated pressurization and depressurization of the fuselage pressure vessel.

ANSWERS

Answer: 4-9
controller.

Answer: 4-12
outflow.

Answer: 4-10
hectopascal (hPa).

Answer: 4-13
differential.

Answer: 4-11
Metal fatigue.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 05

INSTRUMENTS/AVIONIC SYSTEMS

Knowledge Requirements

11.5 - Instruments/Avionic Systems

11.5.1 - Instruments/Avionic Systems

Instrument Systems (ATA 31)

11.5.2 - Avionic Systems

Fundamentals of system layouts and operation of:

- Auto Flight (ATA 22),
- Communications (ATA 23),
- Navigation Systems (ATA 34).

2

1

INSTRUMENTS-AVIONIC
SYSTEMS

10.5 - INSTRUMENTS — AVIONICS SYSTEMS

Since the beginning of manned flight, it has been recognized that supplying the pilot with information about the aircraft and its operation could be useful and lead to safer flight. The Wright Brothers had very few instruments on their Wright Flyer, but they did have an engine tachometer, an anemometer (wind meter), and a stop watch. They were obviously concerned about the aircraft's engine and the progress of their flight. From that simple beginning, a wide variety of instruments have been developed to inform flight crews of different parameters. Instrument systems now exist to provide information on the condition of the aircraft, engine, components, the aircraft's attitude in the sky, weather, cabin environment, navigation, and communication. **Figure 5-1** shows various instrument panels from the Wright Flyer to a modern jet airliner.

The ability to capture and convey all of the information a pilot may want, in an accurate, easily understood manner, has been a challenge throughout the history of aviation. As the range of desired information has grown, so too have the size and complexity of modern aircraft, thus expanding even further the need to inform the flight crew without sensory overload or over cluttering the cockpit. As a result, the old flat panel in the front of the cockpit with various individual instruments attached to it has evolved into a sophisticated computer-controlled digital interface with flat-panel display screens and prioritized messaging. A visual comparison between a conventional cockpit and a glass cockpit is shown in **Figure 5-2**.

There are usually two parts to any instrument or instrument system. One part senses the situation and the other part displays it. In analog instruments, both of these functions often take place in a single unit or instrument (case). These are called direct-sensing instruments. Remote-sensing requires the information to be sensed, or captured, and then sent to a separate display unit in the cockpit. Both analog and digital instruments make use of this method. (**Figure 5-3**)

The relaying of important bits of information can be done in various ways. Electricity is often used by way of wires that carry sensor information into the cockpit. Sometimes pneumatic lines are used.



Figure 5-1. From top to bottom: instruments of the Wright Flyer, instruments on a World War I era aircraft, a late 1950s/early 1960s Boeing 707 airliner cockpit, and an Airbus A380 glass cockpit.



Figure 5-2. A conventional instrument panel of the C-5A Galaxy (top) and the glass cockpit of the C-5B Galaxy (bottom).

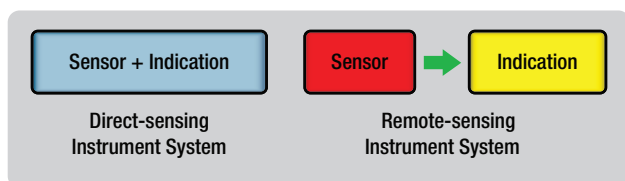


Figure 5-3. There are two parts to any instrument system the sensing mechanism and the display mechanism.

In complex, modern aircraft, this can lead to an enormous amount of tubing and wiring terminating behind the instrument display panel. More efficient information transfer has been accomplished via the use of digital data buses. Essentially, these are wires that share message carrying for many instruments by digitally encoding the signal for each. This reduces the number of wires and weight required to transfer remotely sensed information for the pilot's use. Flat-panel computer display screens that can be controlled to show only the information desired are also lighter in weight than the numerous individual gauges it would take to display the same information simultaneously.

An added bonus is the increased reliability inherent in these solid state systems. It is the job of the aircraft technician to understand and maintain all aircraft, including these various instrument systems. Accordingly, in this chapter, discussions begin with analog instruments and refer to modern digital instrumentation when appropriate.

CLASSIFYING INSTRUMENTS

There are three basic kinds of instruments classified by the job they perform: flight instruments, engine instruments, and navigation instruments. There are also miscellaneous gauges and indicators that provide information that do not fall into these classifications, especially on large complex aircraft. Flight control position, cabin environmental systems, electrical power, and auxiliary power units (APUs), for example, are all monitored and controlled from the cockpit via the use of instruments systems. All may be regarded as position/condition instruments since they usually report the position of a certain movable component on the aircraft, or the condition of various aircraft components or systems not included in the first three groups.

FLIGHT INSTRUMENTS

The instruments used in controlling the aircraft's flight attitude are known as the flight instruments. There are basic flight instruments, such as the altimeter that displays aircraft altitude; the airspeed indicator; and the magnetic direction indicator, a form of compass. Additionally, an artificial horizon, turn coordinator, and vertical speed indicator are flight instruments present in most aircraft. Much variation exists for these instruments, which is explained throughout this chapter. Over the years, flight instruments have come to be situated similarly on the instrument panels in most aircraft. This basic T arrangement for flight instruments is shown in **Figure 5-4**. The top center position directly in front of the pilot and copilot is the basic display position for the artificial horizon even in modern glass cockpits (those with solid-state, flat-panel screen indicating systems).

Original analog flight instruments are operated by air pressure and the use of gyroscopes. This avoids the use of electricity, which could put the pilot in a dangerous situation if the aircraft lost electrical power. Development of sensing and display techniques, combined with advanced aircraft electrical systems,



Figure 5-4. The basic T arrangement of analog flight instruments.

At the bottom of the T is a heading indicator that functions as a compass but is driven by a gyroscope and not subject to the oscillations common to magnetic direction indicators.



Figure 5-5. This electrically operated flat screen display instrument panel, or glass cockpit, retains an analog airspeed indicator, a gyroscope-driven artificial horizon, and an analog altimeter as a backup should electric power be lost, or a display unit fails.

Reciprocating Engines	Turbine Engines
Oil Pressure	Oil Pressure
Oil Temperature	Exhaust Gas Temperature (EGT)
Cylinder Head Temperature (CHT)	Turbine Inlet Temperature (TIT) or Turbine Gas Temperature (TGT)
Manifold Pressure	Engine Pressure Ratio (EPR)
Fuel Quantity	Fuel Quantity
Fuel Pressure	Fuel Pressure
	Fuel Flow
Tachometer	Tachometer (Percent Calibrated)
	N ₁ and N ₂ Compressor Speeds
Carburetor Temperature	Torquemeter (On Turboprop and Turboshaft Engines)

Figure 5-6. Common engine instruments. NOTE: For example purposes only. Some aircraft may not have these instruments or may be equipped with others.

has made it possible for reliable primary and secondary instrument systems that are electrically operated. Nonetheless, often a pneumatic altimeter, a gyro artificial horizon, and a magnetic direction indicator are retained somewhere in the instrument panel for redundancy. (*Figure 5-5*)

ENGINE INSTRUMENTS

Engine instruments are those designed to measure operating parameters of the aircraft's engine(s). These are usually quantity, pressure, and temperature indications. They also include measuring engine speed(s). The most common engine instruments are the fuel and oil quantity and pressure gauges, tachometers, and temperature gauges.

Figure 5-6 contains various engine instruments found on reciprocating and turbine-powered aircraft.

Engine instrumentation is often displayed in the center of the cockpit where it is easily visible to the pilot and copilot. (*Figure 5-7*) On light aircraft requiring only one flight crew member, this may not be the case. Multi-engine aircraft often use a single gauge for a particular engine parameter, but it displays information for all engines through the use of multiple pointers on the same dial face.

NAVIGATION INSTRUMENTS

Navigation instruments are those that contribute information used by the pilot to guide the aircraft along a definite course. This group includes compasses of various kinds, some of which incorporate the use of radio signals to define a specific course while flying the aircraft en route from one airport to another. Other navigational instruments are designed specifically to direct the pilot's approach to landing at an airport.



Figure 5-7. An engine instrumentation located in the middle of the instrument panel is shared by the pilot and co-pilot.

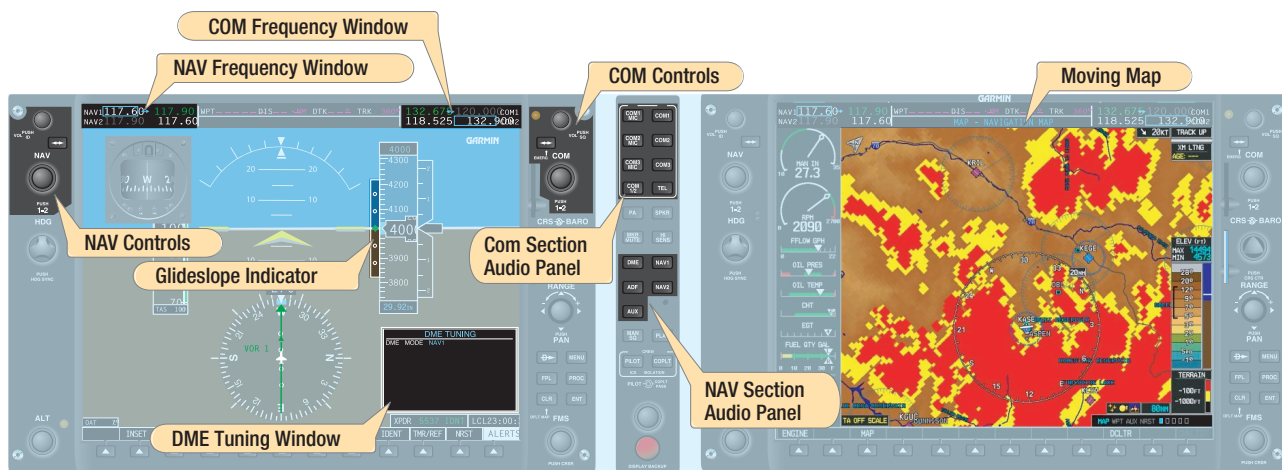


Figure 5-8. Navigation instruments.

Traditional navigation instruments include a clock and a magnetic compass. Along with the airspeed indicator and wind information, these can be used to calculate navigational progress. Radios and instruments sending locating information via radio waves have replaced these manual efforts in modern aircraft.

Global position systems (GPS) use satellites to pinpoint the location of the aircraft via geometric triangulation. This technology is built into some aircraft instrument packages for navigational purposes. An example of navigational information displayed on a glass cockpit display is illustrated in *Figure 5-8*.

CLASSIFICATION BY OPERATING PRINCIPLES

To understand how various instruments work and can be repaired and maintained, they can be classified according to the principle upon which they operate.

Some use mechanical methods to measure pressure and temperature. Some utilize magnetism and electricity to sense and display a parameter. Others depend on the use of gyroscopes in their primary workings.

Still others utilize solid state sensors and computers to process and display important information. In the following sections, the different operating principles for sensing parameters are explained. Then, an overview of many of the engine, flight, and navigation instruments is given.

PRESSURE MEASURING INSTRUMENTS

A number of instruments inform the pilot of the aircraft's condition and flight situations through the measurement of pressure. Pressure sensing instruments can be found in the flight group and the engine group. They can be either direct reading or remote-sensing.

These are some of the most critical instruments on the aircraft and must accurately inform the pilot to maintain safe operations. Pressure measurement involves some sort of mechanism that can sense changes in pressure. A technique for calibration and displaying the information is then added to inform the pilot. The type of pressure needed to be measured often makes one sensing mechanism more suited for use in a particular instance. The three fundamental pressure sensing mechanisms used in aircraft instrument systems are the Bourdon tube, the diaphragm or bellows, and the solid-state sensing device.

A Bourdon tube is illustrated in **Figure 5-9**. The open end of this coiled tube is fixed in place and the other end is sealed and free to move. When a fluid that needs to be measured is directed into the open end of the tube, the unfixed portion of the coiled tube tends to straighten out. The higher the pressure of the fluid, the more the tube straightens. When the pressure is reduced, the tube recoils. A pointer is attached to this moving end of the tube, usually through a linkage of small shafts and gears. By calibrating this motion of the straightening tube, a face or dial of the instrument can be created. Thus, by observing the pointer movement along the scale of the instrument face positioned behind it, pressure increases and decreases are communicated to the pilot.

The Bourdon tube is the internal mechanism for many pressure gauges used on aircraft. When high pressures need to be measured, the tube is designed to be stiff. Gauges used to indicate lower pressures use a more flexible tube that uncoils and coils more readily. Most Bourdon tubes are made from brass, bronze, or copper. Alloys of these metals can be made to coil and uncoil the tube consistently numerous times.

Bourdon tube gauges are simple and reliable. Some of the instruments that use a Bourdon tube mechanism include the engine oil pressure gauge, hydraulic pressure gauge, oxygen tank pressure gauge, and de-ice boot pressure gauge. Since the pressure of the vapor produced by a heated liquid or gas increases as temperature increases, Bourdon tube mechanisms can also be used to measure temperature. This is done by calibrating the pointer connecting linkage and relabeling the face of the gauge with a temperature scale. Oil temperature gauges often employ Bourdon tube mechanisms. (**Figure 5-10**)

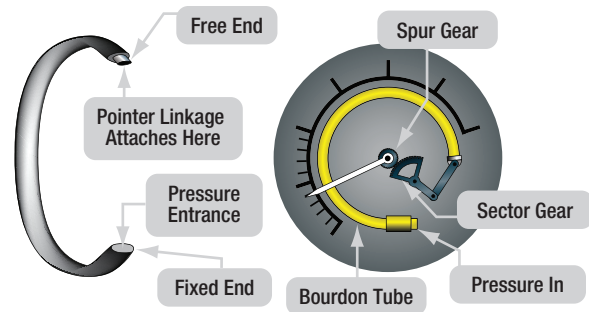


Figure 5-9. The Bourdon tube is one of the basic mechanisms for sensing pressure.

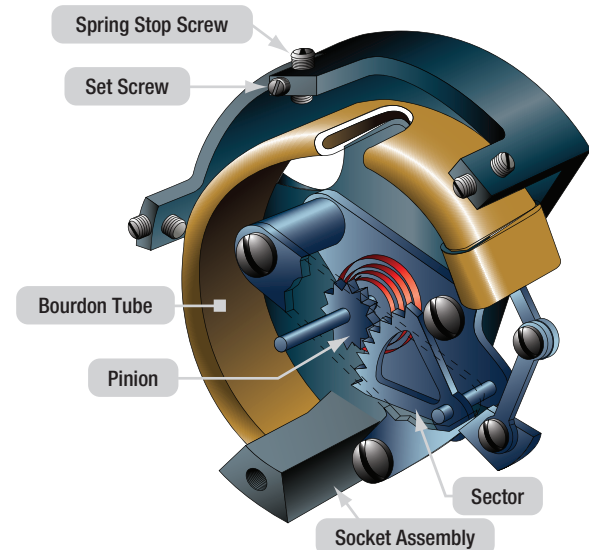


Figure 5-10. The Bourdon tube mechanism can be used to measure pressure or temperature by recalibrating the pointer's connecting linkage and scaling instrument face to read in degrees Celsius or Fahrenheit.

Since the sensing and display of pressure or temperature information using a Bourdon tube mechanism usually occurs in a single instrument housing, they are most often direct reading gauges. But the Bourdon tube sensing device can also be used remotely. Regardless, it is necessary to direct the fluid to be measured into the Bourdon tube. For example, a common direct-reading gauge measuring engine oil pressure and indicating it to the pilot in the cockpit is mounted in the instrument panel. A small length of tubing connects a pressurized oil port on the engine, runs through the firewall, and into the back of the gauge. This setup is especially functional on light, single-engine aircraft in which the engine is mounted just forward of the instrument panel in the forward end of the fuselage. However, a remote sensing unit can be more practical on twin engine aircraft where the engines are a long distance from the cockpit pressure display.

Here, the Bourdon tube's motion is converted to an electrical signal and carried to the cockpit display via a wire. This is lighter and more efficient, eliminating the possibility of leaking fluids into the passenger compartment of the aircraft.

The diaphragm and bellows are two other basic sensing mechanisms employed in aircraft instruments for pressure measurement. The diaphragm is a hollow, thin-walled metal disk, usually corrugated. When pressure is introduced through an opening on one side of the disk, the entire disk expands. By placing linkage in contact against the other side of the disk, the movement of the pressurized diaphragm can be transferred to a pointer that registers the movement against the scale on the instrument face. (*Figure 5-11*)

Diaphragms can also be sealed. The diaphragm can be evacuated before sealing, retaining absolutely nothing inside. When this is done, the diaphragm is called an aneroid. Aneroids are used in many flight instruments. A diaphragm can also be filled with a gas to standard atmospheric pressure and then sealed. Each of these diaphragms has their uses, which are described in the next section. The common factor in all is that the expansion and contraction of the side wall of the diaphragm is the movement that correlates to increasing and decreasing pressure.

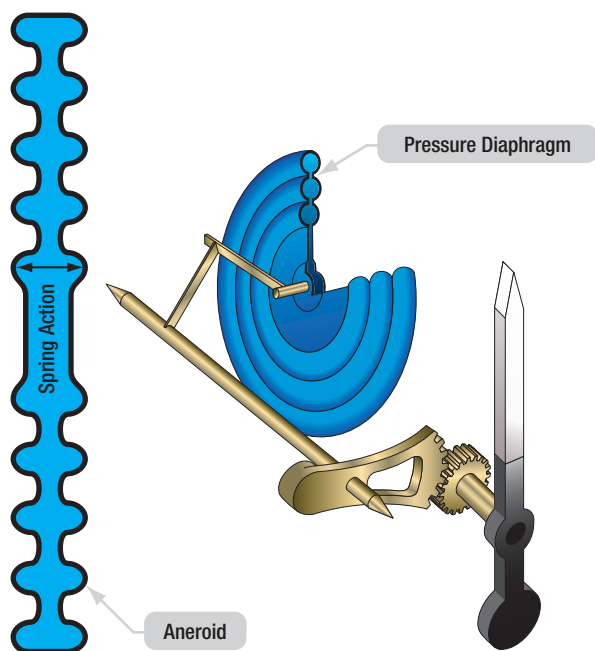


Figure 5-11. A diaphragm used for measuring pressure.
An evacuated sealed diaphragm is called an aneroid.

When a number of diaphragm chambers are connected together, the device is called a bellows. This accordion-like assembly of diaphragms can be very useful when measuring the difference in pressure between two gases, called differential pressure. Just as with a single diaphragm, it is the movement of the side walls of the bellows assembly that correlates with changes in pressure and to which a pointer linkage and gearing is attached to inform the pilot. (*Figure 5-12*)

Diaphragms, aneroids, and bellows pressure sensing devices are often located inside the single instrument housing that contains the pointer and instrument dial read by the pilot on the instrument panel. Thus, many instruments that make use of these sensitive and reliable mechanisms are direct reading gauges. But, many remote-sensing instrument systems also make use of the diaphragm and bellows. In this case, the sensing device containing the pressure sensitive diaphragm or bellows is located remotely on the engine or airframe. It is part of a transducer that converts the pressure into an electrical signal. The transducer, or transmitter, sends the signal to the gauge in the cockpit, or to a computer, for processing and subsequent display of the sensed condition.

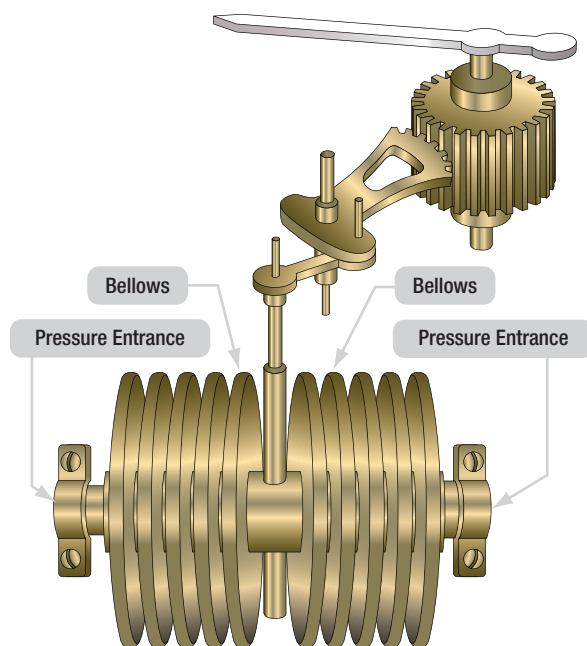


Figure 5-12. A bellows unit in a differential pressure gauge compares two different pressure values. End movement of the bellows away from the side with the highest pressure input occurs when the pressures in the bellows are not equal. The indicator linkage is calibrated to display the difference.

Examples of instruments that use a diaphragm or bellows in a direct reading or remote-sensing gauge are the altimeter, vertical speed indicator, cabin differential pressure gauge (in pressurized aircraft), and manifold pressure gauge.

Solid-state microtechnology pressure sensors are used in modern aircraft to determine the critical pressures needed for safe operation. Many of these have digital output ready for processing by electronic flight instrument computers and other on board computers. Some sensors send microelectric signals that are converted to digital format for use by computers. As with the analog sensors described above, the key to the function of solid-state sensors is their consistent property changes as pressure changes.

The solid-state sensors used in most aviation applications exhibit varying electrical output or resistance changes when pressure changes occur. Crystalline piezoelectric, piezoresistor, and semiconductor chip sensors are most common. In the typical sensor, tiny wires are embedded in the crystal or pressure sensitive semiconductor chip. When pressure deflects the crystal(s), a small amount of electricity is created or, in the case of a semiconductor chip and some crystals, the resistance changes. Since the current and resistance changes vary directly with the amount of deflection, outputs can be calibrated and used to display pressure values. Nearly all of the pressure information needed for engine, airframe, and flight instruments can be captured and/or calculated through the use of solid-state pressure sensors in combination with temperature sensors. But continued use of aneroid devices for comparisons involving absolute pressure is notable. Solid-state pressure sensing systems are remote-sensing systems. The sensors are mounted on the aircraft at convenient and effective locations.

TYPES OF PRESSURE

Pressure is a comparison between two forces. Absolute pressure exists when a force is compared to a total vacuum, or absolutely no pressure. It is necessary to define absolute pressure, because the air in the atmosphere is always exerting pressure on everything. Even when it seems there is no pressure being applied, like when a balloon is deflated, there is still atmospheric pressure inside and outside of the balloon. To measure that atmospheric pressure, it is necessary to compare it to a total absence of pressure, such as in a vacuum.

Many aircraft instruments make use of absolute pressure values, such as the altimeter, the rate-of-climb indicator, and the manifold pressure gauge. As stated, this is usually done with an aneroid.

The most common type of pressure measurement is gauge pressure. This is the difference between the pressure to be measured and the atmospheric pressure. The gauge pressure inside the deflated balloon mentioned above is therefore 0 pounds per square inch (psi). Gauge pressure is easily measured and is obtained by ignoring the fact that the atmosphere is always exerting its pressure on everything. For example, a tire is filled with air to 32 psi at a sea level location and checked with a gauge to read 32 psi, which is the gauge pressure. The approximately 14.7 psi of air pressing on the outside of the tire is ignored. The absolute pressure in the tire is 32 psi plus the 14.7 psi that is needed to balance the 14.7 psi on the outside of the tire. So, the tire's absolute pressure is approximately 46.7 psi. If the same tire is inflated to 32 psi at a location 10 000 feet above sea level, the air pressure on the outside of the tire would only be approximately 10 psi, due to the thinner atmosphere. The pressure inside the tire required to balance this would be 32 psi plus 10 psi, making the absolute pressure of the tire 42 psi. So, the same tire with the same amount of inflation and performance characteristics has different absolute pressure values. Gauge pressure, however, remains the same, indicating the tires are inflated identically. In this case, gauge pressure is more useful in informing us of the condition of the tire.

Gauge pressure measurements are simple and widely useful. They eliminate the need to measure varying atmospheric pressure to indicate or monitor a particular pressure situation. Gauge pressure should be assumed, unless otherwise indicated, or unless the pressure measurement is of a type known to require absolute pressure.

In many instances in aviation, it is desirable to compare the pressures of two different elements to arrive at useful information for operating the aircraft. When two pressures are compared in a gauge, the measurement is known as differential pressure and the gauge is a differential pressure gauge. An aircraft's airspeed indicator is a differential pressure gauge. It compares ambient air pressure with ram air pressure to determine how fast the aircraft is moving through the air.

A turbine engine's engine pressure ratio (EPR) gauge is also a differential pressure gauge. It compares the pressure at the inlet of the engine with that at the outlet to indicate the thrust developed by the engine. Both of these differential pressure gauges and others are discussed further in this chapter and throughout this module.

In aviation, there is also a commonly used pressure known as standard pressure. Standard pressure refers to an established or standard value that has been created for atmospheric pressure. This standard pressure value is 29.92 inches of mercury ("Hg), 1 013.2 hectopascal (hPa), or 14.7 psi. It is part of a standard day that has been established that includes a standard temperature of 15°C at sea level. Specific standard day values have also been established for air density, volume, and viscosity. All of these values are developed averages since the atmosphere is continuously fluctuating. They are used by engineers when designing instrument systems and are sometimes used by technicians and pilots. Often, using a standard value for atmospheric pressure is more desirable than using the actual value.

For example, at 18 000 feet and above, all aircraft use 29.92 "Hg as a reference pressure for their instruments to indicate altitude. This results in altitude indications in all cockpits being identical. Therefore, an accurate means is established for maintaining vertical separation of aircraft flying at these high altitudes.

PRESSURE INSTRUMENTS

Engine Oil Pressure

The most important instrument used by the pilot to perceive the health of an engine is the engine oil pressure gauge. (*Figure 5-13*) Oil pressure is usually indicated in psi. The normal operating range is typically represented by a green arc on the circular gauge. For exact acceptable operating range, consult the manufacturer's operating and maintenance data. In reciprocating and turbine engines, oil is used to lubricate and cool bearing surfaces where parts are rotating or sliding past each other at high speeds. A loss of pressurized oil to these areas would rapidly cause excessive friction and over temperature conditions, leading to catastrophic engine failure. As mentioned, aircraft using analog instruments often use direct reading Bourdon tube oil pressure gauges. *Figure 5-13* shows the instrument face of a typical oil

pressure gauge of this type. Digital instrument systems use an analog or digital remote oil pressure sensing unit that sends output to the computer, driving the display of oil pressure value(s) on the aircraft's cockpit display screens. Oil pressure may be displayed in a circular or linear gauge fashion and may even include a numerical value on screen. Often, oil pressure is grouped with other engine parameter displays on the same page or portion of a page on the display. *Figure 5-14* shows this grouping on a Garmin G1000 digital instrument display system for general aviation aircraft.

Manifold Pressure

In reciprocating engine aircraft, the manifold pressure gauge indicates the pressure of the air in the engine's induction manifold. This is an indication of power being developed by the engine. The higher the pressure of the



Figure 5-13. An analog oil pressure gauge is driven by a Bourdon tube. Oil pressure is vital to engine health and must be monitored by the pilot.



Figure 5-14. Oil pressure indication with other engine-related parameters shown in a column on the left side of this digital cockpit display panel.

fuel air mixture going into the engine, the more power it can produce. For normally aspirated engines, this means that an indication near atmospheric pressure is the maximum. Turbocharged or supercharged engines pressurize the air being mixed with the fuel, so full power indications are above atmospheric pressure.

Most manifold pressure gauges are calibrated in inches of mercury, although digital displays may have the option to display in a different scale. A typical analog gauge makes use of an aneroid described above. When atmospheric pressure acts on the aneroid inside the gauge, the connected pointer indicates the current air pressure. A line running from the intake manifold into the gauge presents intake manifold air pressure to the aneroid, so the gauge indicates the absolute pressure in the intake manifold. An analog manifold pressure gauge, along with its internal workings is shown in **Figure 5-16**. The digital presentation of manifold pressure is at the top of the engine instruments displayed on the Garmin G1000 multifunctional display in **Figure 5-14**. The aircraft's operating manual contains data on managing manifold pressure in relation to fuel flow and propeller pitch and for achieving various performance profiles during different phases of run-up and flight.

Engine Pressure Ratio (EPR)

Turbine engines have their own pressure indication that relates the power being developed by the engine. It is called the engine pressure ratio (EPR) indicator (EPR

gauge). This gauge compares the total exhaust pressure to the pressure of the ram air at the inlet of the engine. With adjustments for temperature, altitude, and other factors, the EPR gauge presents an indication of the thrust being developed by the engine. Since the EPR gauge compares two pressures, it is a differential pressure gauge. It is a remote-sensing instrument that receives its input from an engine pressure ratio transmitter or, in digital instrument systems displays, from a computer. The pressure ratio transmitter contains the bellows arrangement that compares the two pressures and converts the ratio into an electric signal used by the gauge for indication. (**Figure 5-15**)

Fuel Pressure

Fuel pressure gauges also provide critical information to the pilot. (**Figure 5-17**) Typically, fuel is pumped out of various fuel tanks on the aircraft for use by the engines. A malfunctioning fuel pump, or a tank that has been emptied beyond the point at which there is sufficient fuel entering the pump to maintain desired output pressure, is a condition that requires the pilot's immediate attention. While direct-sensing fuel pressure gauges using Bourdon tubes, diaphragms, and bellows sensing arrangements exist, it is particularly undesirable to run a fuel line into the cockpit, due to the potential for fire should a leak develop. Therefore the preferred arrangement is to have whichever sensing mechanism that is used be part of a transmitter device that uses electricity to send a signal to the indicator in the cockpit.

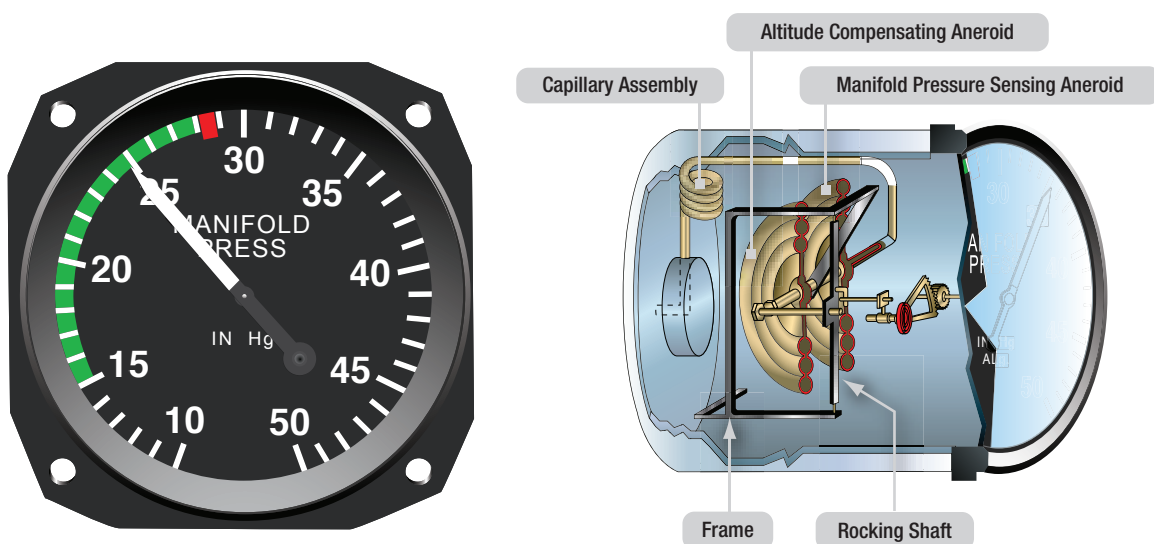
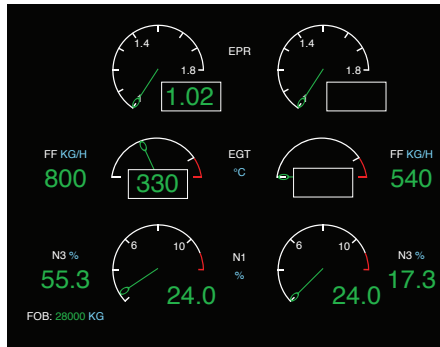


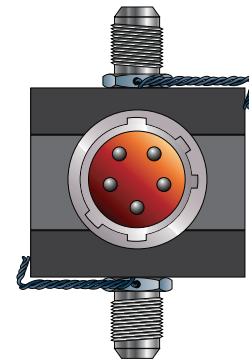
Figure 5-15. An analog manifold pressure indicator instrument dial calibrated in inches of mercury (left). The internal workings of an analog manifold pressure gauge are shown on the right. Air from the intake manifold surrounds the aneroid causing it to deflect and indicate pressure on the dial through the use of linkage to the pointer (right).



A. An analog EPR gauge from a turbine engine.



B. A digital EPR indication and other turbine engine parameters on a cockpit digital display screen.



C. Engine pressure ratio transducer.

Figure 5-16. Engine pressure ratio gauges.



Figure 5-17. A typical analog fuel pressure gauge.

Sometimes, indications monitoring the fuel flow rate are used instead of fuel pressure gauges. Fuel flow indications are discussed in the fuel system chapter of this handbook.

Hydraulic Pressure

Numerous other pressure monitoring gauges are used on complex aircraft to indicate the condition of various support systems not found on simple light aircraft. Hydraulic systems are commonly used to raise and lower landing gear, operate flight controls, apply brakes, and more. Sufficient pressure in the hydraulic system developed by the hydraulic pump(s) is required for normal operation of hydraulic devices. Hydraulic pressure gauges are often located in the cockpit and at or near the hydraulic system servicing point on the airframe. Remotely located indicators used by maintenance personnel are almost always direct reading Bourdon tube type gauges. Cockpit gauges usually have system pressure transmitted

from sensors or computers electrically for indication. **Figure 5-18** shows a hydraulic pressure transmitter in place in a high-pressure aircraft hydraulic system.

Vacuum Pressure

Gyro pressure gauge, vacuum gauge, or suction gauge are all terms for the same gauge used to monitor the vacuum developed in the system that actuates the air driven gyroscopic flight instruments. Air is pulled through the instruments, causing the gyroscopes to spin. The speed at which the gyros spin needs to be within a certain range for correct operation. This speed is directly related to the suction pressure that is developed in the system. The suction gauge is extremely important in aircraft relying solely on vacuum operated gyroscopic flight instruments.

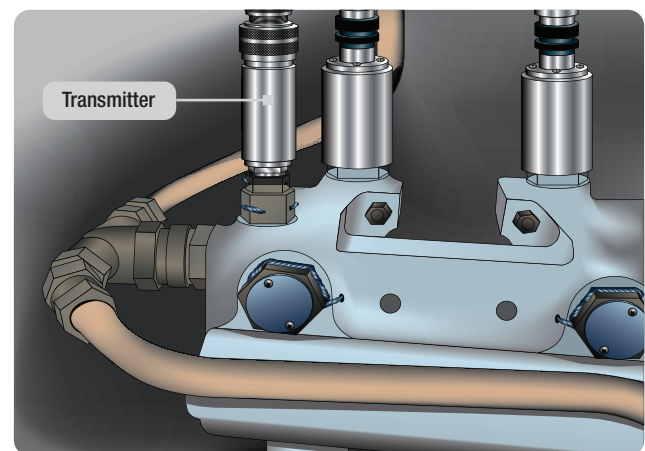


Figure 5-18. A hydraulic pressure transmitter senses and converts pressure into an electrical output for indication by the cockpit gauge or for use by a computer that analyzes and displays the pressure in the cockpit when requested or required.

Vacuum is a differential pressure indication, meaning the pressure to be measured is compared to atmospheric pressure through the use of a sealed diaphragm or capsule. The gauge is calibrated in inches of mercury. It shows how much less pressure exists in the system than in the atmosphere. **Figure 5-19** shows a suction gauge calibrated in inches of mercury.

Pressure Switches

In aviation, it is often sufficient to simply monitor whether the pressure developed by a certain operating system is too high or too low, so that an action can take place should one of these conditions occur. This is often accomplished through the use of a pressure switch. A pressure switch is a simple device usually made to open or close an electric circuit when a certain pressure is reached in a system. It can be manufactured so that the electric circuit is normally open and can then close when a certain pressure is sensed, or the circuit can be closed and then opened when the activation pressure is reached. (**Figure 5-20**)

Pressure switches contain a diaphragm to which the pressure being sensed is applied on one side. The opposite side of the diaphragm is connected to a mechanical switching mechanism for an electric circuit. Small fluctuations or a buildup of pressure against the diaphragm move the diaphragm, but not enough to throw the switch. Only when pressure meets or exceeds a preset level designed into the structure of the switch does the diaphragm move far enough for the mechanical device on the opposite side to close the switch contacts and complete the circuit. (**Figure 5-21**) Each switch is rated to close (or open) at a certain pressure, and must only be installed in the proper location.



Figure 5-19. Vacuum suction gauge.



Figure 5-20. A pressure switch can be used in addition to, or instead of, a pressure gauge.

A low oil pressure indication switch is a common example of how pressure switches are employed. It is installed in an engine so pressurized oil can be applied to the switch's diaphragm. Upon starting the engine, oil pressure increases and the pressure against the diaphragm is sufficient to hold the contacts in the switch open. As such, current does not flow through the circuit and no indication of low oil pressure is given in the cockpit. Should a loss of oil pressure occur, the pressure against the diaphragm becomes insufficient to hold the

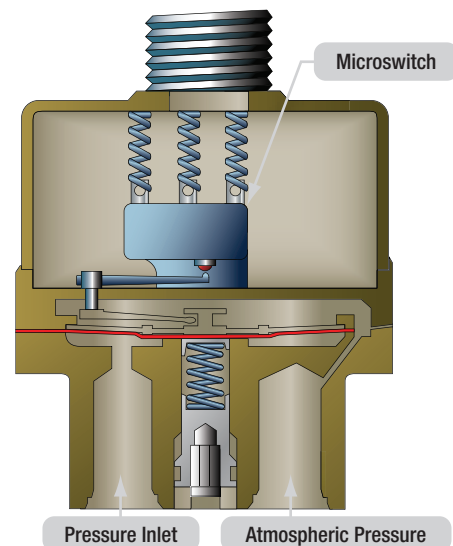


Figure 5-21. A normally open pressure switch positioned in an electrical circuit causes the circuit to be open as well. The switch closes, allowing electricity to flow when pressure is applied beyond the switch's preset activation point. Normally, closed pressure switches allow electricity to flow through the switch in a circuit but open when pressure reaches a preset activation point, thus opening the electrical circuit.

switched contacts open. When the contacts close, they close the circuit to the low oil pressure indicator, usually a light, to warn the pilot of the situation.

Pressure gauges for various components or systems work similarly to those mentioned above. Some sort of sensing device, appropriate for the pressure being measured or monitored, is matched with an indicating display system. If appropriate, a properly rated pressure switch is installed in the system and wired into an indicating circuit. Further discussion of specific instruments occurs throughout this handbook as the operation of various systems and components are discussed.

PITOT-STATIC SYSTEMS

Some of the most important flight instruments derive their indications from measuring air pressure. Gathering and distributing various air pressures for flight instrumentation is the function of the pitot-static system.

PITOT TUBES AND STATIC VENTS

On simple aircraft, this may consist of a pitot-static system head or pitot tube with impact and static air pressure ports and leak-free tubing connecting these air pressure pickup points to the instruments that require the air for their indications. The altimeter, airspeed indicator, and vertical speed indicator are the three

most common pitot-static instruments. *Figure 5-22* illustrates a simple pitot-static system connected to these three instruments.

A pitot tube is shown in *Figure 5-23*. It is open and faces into the airstream to receive the full force of the impact air pressure as the aircraft moves forward. This air passes through a baffled plate designed to protect the system from moisture and dirt entering the tube. Below the baffle, a drain hole is provided, allowing moisture to escape. The ram air is directed aft to a chamber in

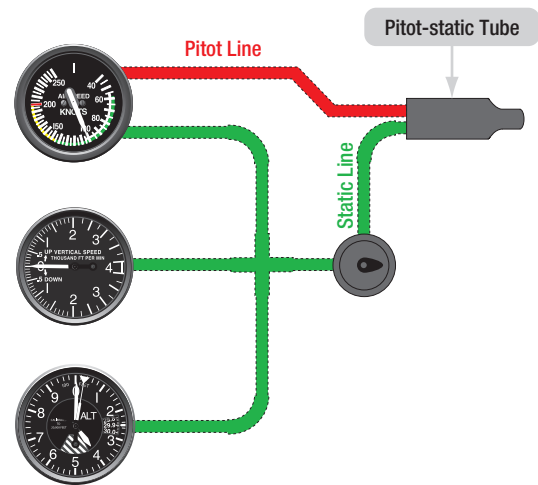


Figure 5-22. A simple pitot-static system is connected to the primary flight instruments.

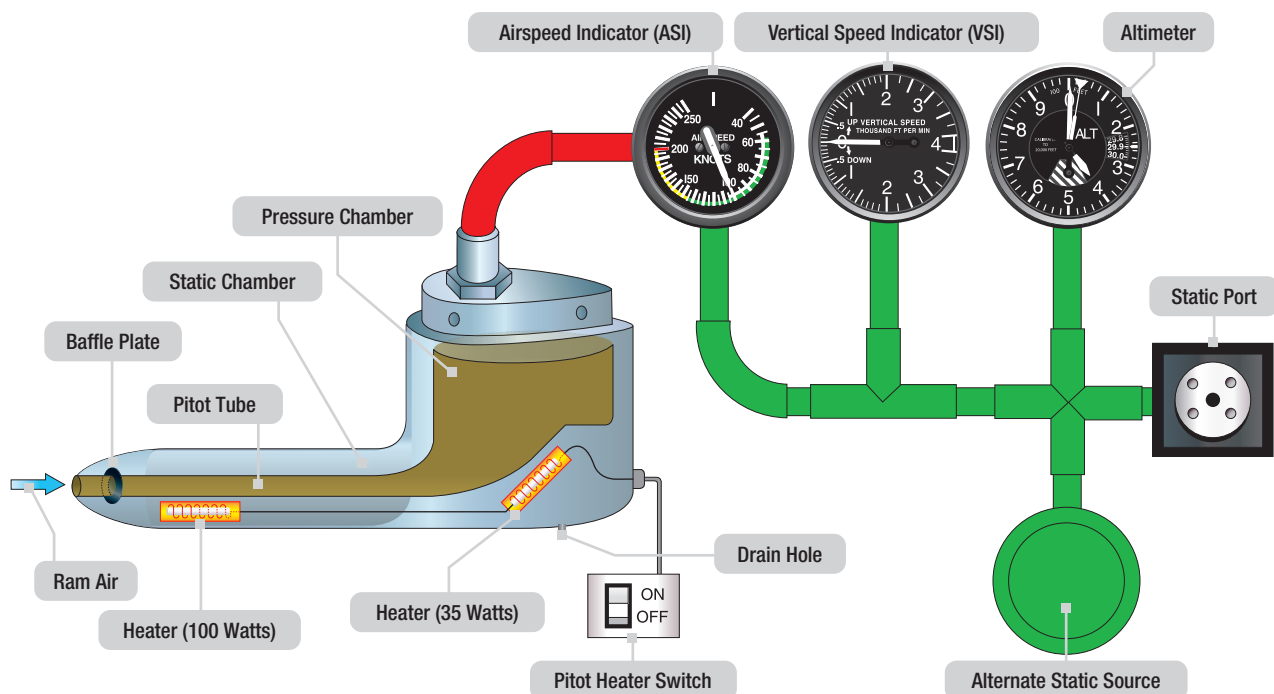


Figure 5-23. A typical pitot-static system head, or pitot tube, collects ram air and static pressure for use by the flight instruments.

the shark fin of the assembly. An upright tube, or riser, leads this pressurized air out of the pitot assembly to the airspeed indicator.

The aft section of the pitot tube is equipped with small holes on the top and bottom surfaces that are designed to collect air pressure that is at atmospheric pressure in a static, or still, condition. (*Figure 5-23*) The static section also contains a riser tube and the air is run out the pitot assembly through tubes and is connected to the altimeter, the airspeed indicator, and the vertical speed indicator.

Many pitot-static tube heads contain heating elements to prevent icing during flight. The pilot can send electric current to the element with a switch in the cockpit when ice-forming conditions exist. Often, this switch is wired through the ignition switch so that when the aircraft is shut down, a pitot tube heater inadvertently left on does not continue to draw current and drain the battery. Caution should be exercised when near the pitot tube, as these heating elements make the tube too hot to be touched without receiving a burn.

The pitot-static tube is mounted on the outside of the aircraft at a point where the air is least likely to be turbulent. It is pointed in a forward direction parallel to the aircraft's line of flight. The location may vary. Some are on the nose of the fuselage and others may be located on a wing. A few may even be found on the empennage. Various designs exist but the function remains the same, to capture impact air pressure and static air pressure and direct them to the proper instruments. (*Figure 5-24*)



Figure 5-24. Pitot static system heads, or pitot tubes, can be of various designs and locations on airframes.

Most aircraft equipped with a pitot-static tube have an alternate source of static air pressure provided for emergency use. The pilot may select the alternate with a switch in the cockpit should it appear the flight instruments are not providing accurate indications. On low-flying unpressurized aircraft, the alternate static source may simply be air from the cabin. (*Figure 5-25*)

On pressurized aircraft, cabin air pressure may be significantly different than the outside ambient air pressure. If used as an alternate source for static air, instrument indications would be grossly inaccurate. In this case, multiple static vent pickup points are employed. All are located on the outside of the aircraft and plumbed so the pilot can select which source directs air into the instruments. On electronic flight displays, the choice is made for which source is used by the computer or by the flight crew.

Another type of pitot-static system provides for the location of the pitot and static sources at separate positions on the aircraft. The pitot tube in this arrangement is used only to gather ram air pressure. Separate static vents are used to collect static air pressure information. Usually, these are located flush on the side of the fuselage. (*Figure 5-26*) There may be two or more vents. A primary and alternate source vent is typical, as well as separate dedicated vents for the pilot and first officer's instruments. Also, two primary vents may be located on opposite sides of the fuselage and connected with Y tubing for input to the instruments.



Figure 5-25. On unpressurized aircraft, an alternate source of static air is cabin air.



Figure 5-26. Heated primary and alternate static vents located on the sides of the fuselage.

This is done to compensate for any variations in static air pressure on the vents due to the aircraft's attitude. Regardless of the number and location of separate static vents, they may be heated as well as the separate ram air pitot tube to prevent icing.

The pitot-static systems of complex, multi-engine, and pressurized aircraft can be elaborate. Additional instruments, gauges, the autopilot system, and computers may need pitot and static air information. **Figure 5-27** shows a pitot-static system for a pressurized multi-engine aircraft with dual analog instrument panels in the cockpit. The additional set of flight instruments for the copilot alters and complicates the pitot-static system plumbing. Additionally, the autopilot system requires static pressure information, as does the cabin pressurization unit.

Separate heated sources for static air pressure are taken from both sides of the airframe to feed independent static air pressure manifolds; one each for the pilot's flight instruments and the copilot's flight instruments. This is designed to ensure that there is always one set of flight instruments operable in case of a malfunction.

AIR DATA COMPUTERS (ADC) AND DIGITAL AIR DATA COMPUTERS (DADC)

High performance and jet transport category aircraft pitot-static systems may be more complicated. These aircraft frequently operate at high altitude where the ambient temperature can exceed -45°C . The compressibility of air is also altered at high speeds and at high altitudes.

Airflow around the fuselage changes, making it difficult to pick up consistent static pressure inputs. The pilot must compensate for all factors of air temperature and density to obtain accurate indications from instruments. While many analog instruments have compensating devices built into them, the use of an air data computer (ADC) is common for these purposes on high-performance aircraft. Moreover, modern aircraft utilize digital air data computers (DADC). The conversion of sensed air pressures into digital values makes them more easily manipulated by the computer to output accurate information that has compensated for the many variables encountered. (**Figure 5-28**)

Essentially, all pressures and temperatures captured by sensors are fed into the ADC. Analog units utilize transducers to convert these to electrical values and manipulate them in various modules containing circuits designed to make the proper compensations for use by different instruments and systems. A DADC usually receives its data in digital format. Systems that do not have digital sensor outputs will first convert inputs into digital signals via an analog-to-digital converter.

Conversion can take place inside the computer or in a separate unit designed for this function. Then, all calculation and compensations are performed digitally by the computer. Outputs from the ADC are electric to drive servo motors or for use as inputs in pressurization systems, flight control units, and other systems. DADC outputs are distributed to these same systems and the cockpit display using a digital data bus. There are numerous benefits of using ADCs. Simplification of pitot-static plumbing lines creates a lighter, simpler, system with fewer connections, so it is less prone to leaks and easier to maintain.

One time compensation calculations can be done inside the computer, eliminating the need to build compensating devices into numerous individual instruments or units of the systems using the air data. DADC's can run a number of checks to verify the plausibility of data received from any source on the aircraft. Thus, the crew can be alerted automatically of a parameter that is out of the ordinary. Change to an alternate data source can also be automatic so accurate flight deck and systems operations are continuously maintained.

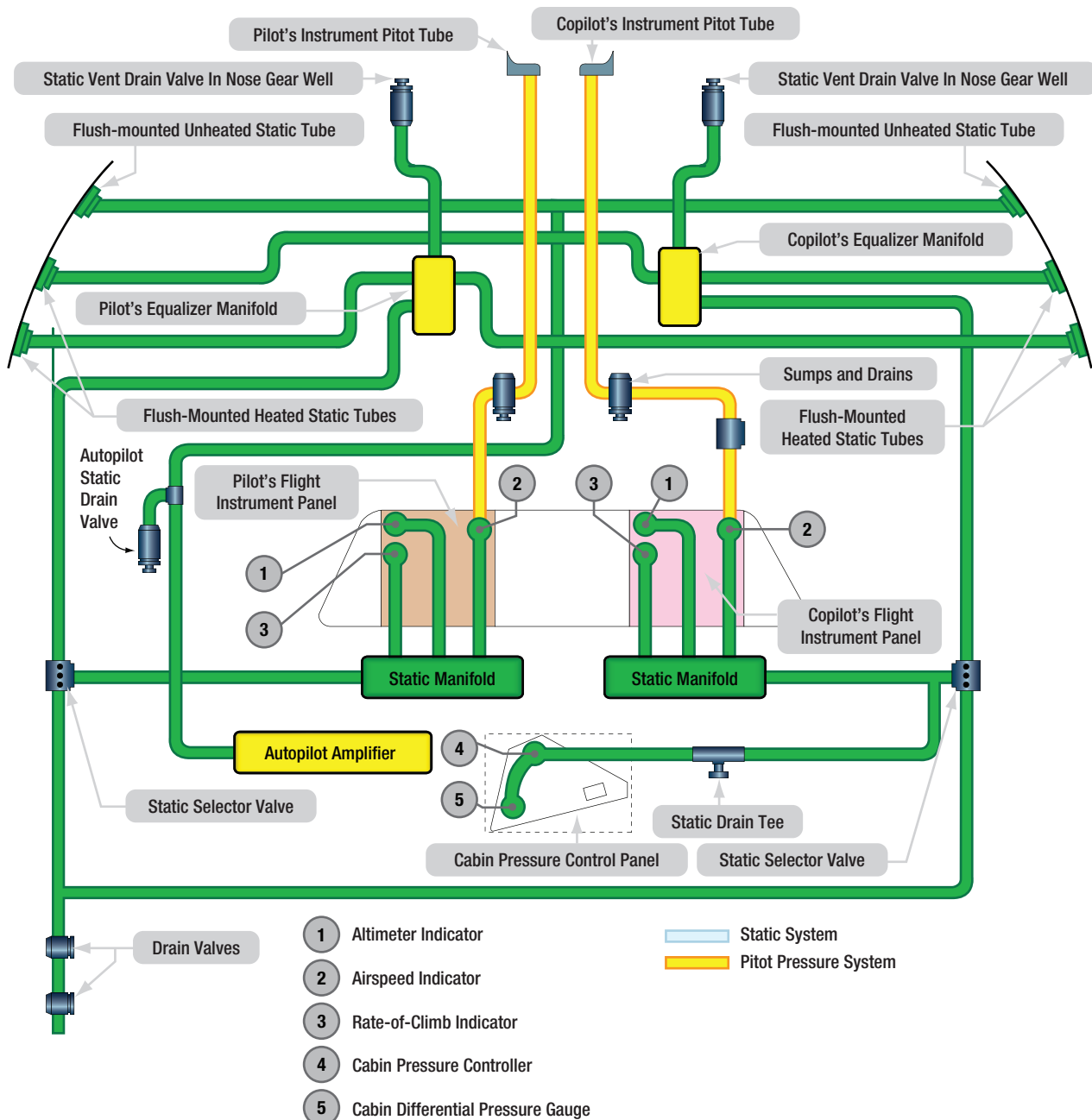


Figure 5-27. Schematic of a typical pitot-static system on a pressurized multi-engine aircraft.

In general, solid-state technology is more reliable and modern units are small and lightweight. *Figure 5-29* shows a schematic of how a DADC is connected into the aircraft's pitot-static and other systems.

PITOT-STATIC PRESSURE SENSING FLIGHT INSTRUMENTS

The basic flight instruments are directly connected to the pitot-static system on many aircraft. Analog flight instruments primarily use mechanical means to measure and indicate various flight parameters. Digital flight instrument systems use electricity and electronics



Figure 5-28. Teledyne's 90004 TAS/Plus air data computer (ADC) computes air data information from the pitot-static pneumatic system, aircraft temperature probe, and barometric correction device to help create a clear indication of flight conditions.

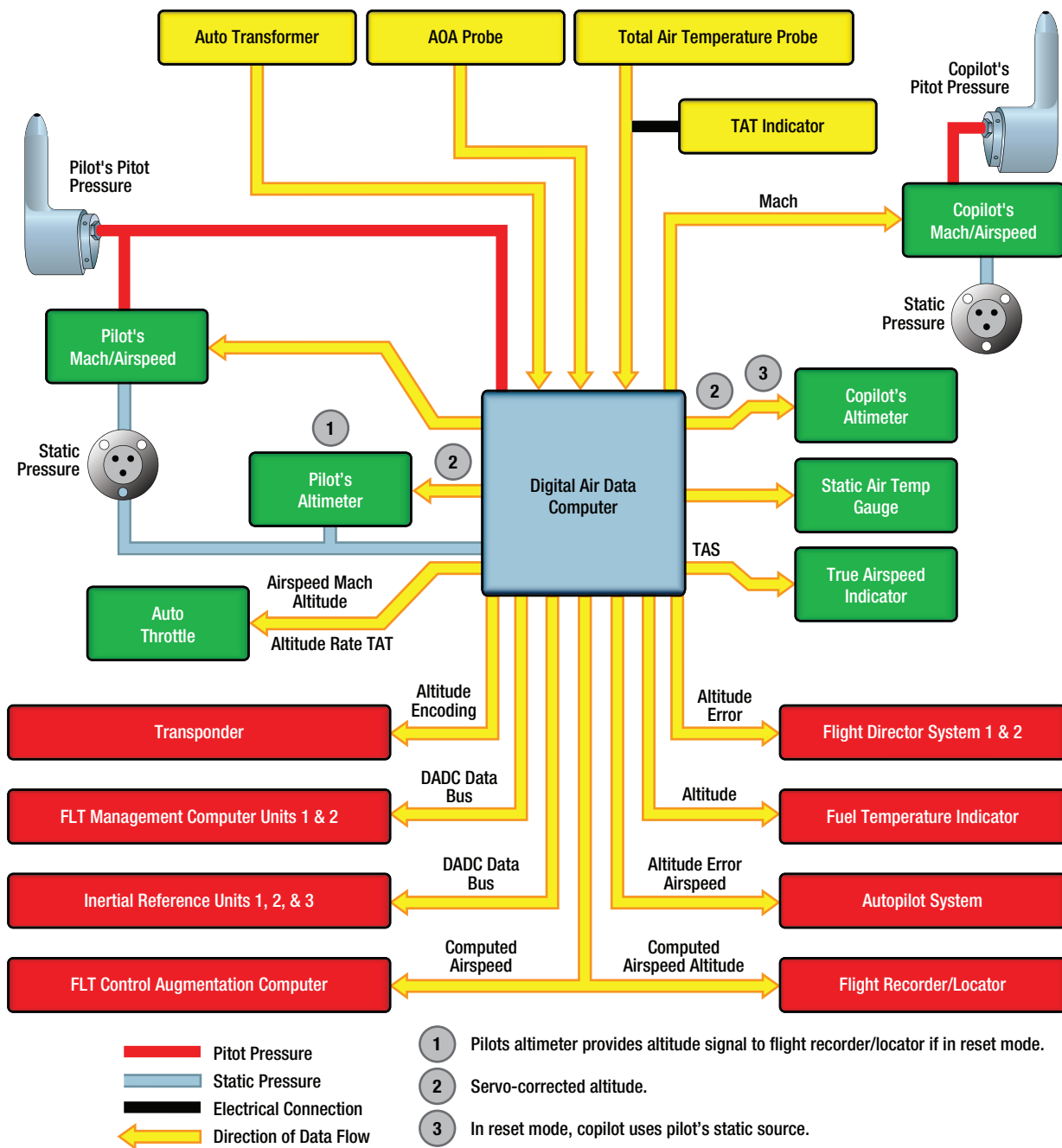


Figure 5-29. ADCs receive input from the pitot-static sensing devices and process them for use by numerous aircraft systems.

to do the same. Discussion of the basic pitot-static flight instruments begins with analog instruments to which further information about modern digital instrumentation is added.

Altimeters and Altitude

An altimeter is an instrument that is used to indicate the height of the aircraft above a predetermined level, such as sea level or the terrain beneath the aircraft. The most common way to measure this distance is rooted in discoveries made by scientists centuries ago. Seventeenth

century work proving that the air in the atmosphere exerted pressure on the things around us led Evangelista Torricelli to the invention of the barometer. Also in that century, using the concept of this first atmospheric air pressure measuring instrument, Blaise Pascal was able to show that a relationship exists between altitude and air pressure. As altitude increases, air pressure decreases. The amount that it decreases is measurable and consistent for any given altitude change. Therefore, by measuring air pressure, altitude can be determined. (*Figure 5-30*)

Altimeters that measure the aircraft's altitude by measuring the pressure of the atmospheric air are known as pressure altimeters. A pressure altimeter is

Atmosphere Pressure	
Altitude (ft)	Pressure (psi)
Sea Level	14.69
2 000	13.66
4 000	12.69
6 000	11.77
8 000	10.91
10 000	10.10
12 000	9.34
14 000	8.63
16 000	7.96
18 000	7.34
20 000	6.75
22 000	6.20
24 000	5.69
26 000	5.22
28 000	4.77
30 000	4.36
32 000	3.98
34 000	3.62
36 000	3.29
38 000	2.99
40 000	2.72
42 000	2.47
44 000	2.24
46 000	2.04
48 000	1.85
50 000	1.68

Figure 5-30. Air pressure is inversely related to altitude. This consistent relationship is used to calibrate the pressure altimeter.

made to measure the ambient air pressure at any given location and altitude. In aircraft, it is connected to the static vent(s) via tubing in the pitot-static system. The relationship between the measured pressure and the altitude is indicated on the instrument face, which is calibrated in feet. These devices are direct-reading instruments that measure absolute pressure. An aneroid or aneroid bellows is at the core of the pressure altimeter's inner workings. Attached to this sealed diaphragm are the linkages and gears that connect it to the indicating pointer. Static air pressure enters the airtight instrument case and surrounds the aneroid. At sea level, the altimeter indicates zero when this pressure is exerted by the ambient air on the aneroid. As air pressure is reduced by moving the altimeter higher in the atmosphere, the aneroid expands and displays altitude on the instrument by rotating the pointer. As the altimeter is lowered in the atmosphere, the air pressure around the aneroid increases and the pointer moves in the opposite direction. (*Figure 5-31*)

The face, or dial, of an analog altimeter is read similarly to a clock. As the longest pointer moves around the dial, it is registering the altitude in hundreds of feet. One complete revolution of this pointer indicates 1 000 feet of altitude. The second longest point moves more slowly. Each time it reaches a numeral, it indicates 1 000 feet of altitude. Once around the dial for this pointer is equal to 10 000 feet. When the longest pointer travels

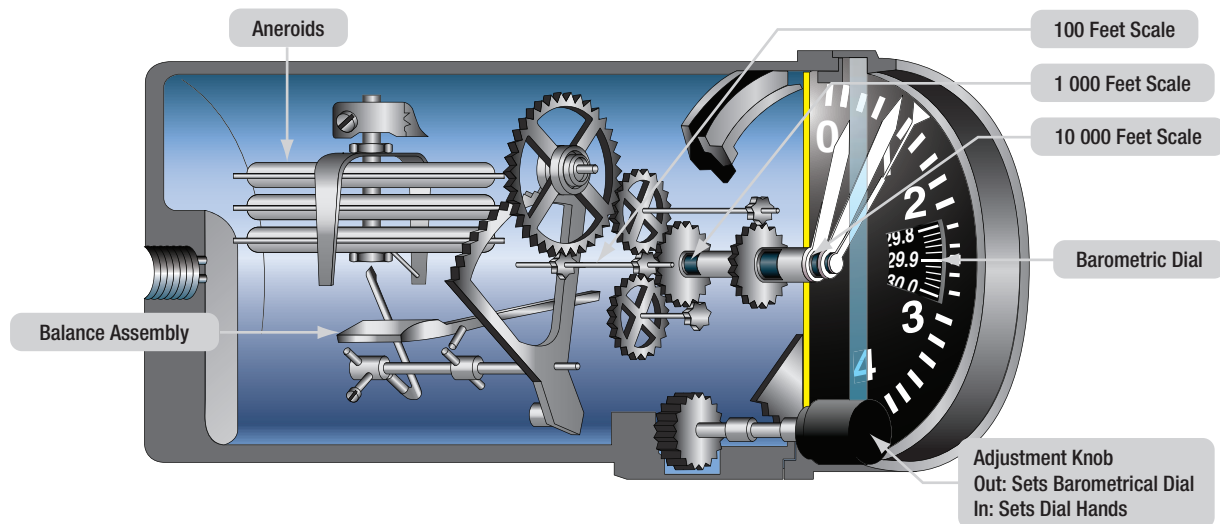


Figure 5-31. The internal arrangement of a sealed diaphragm pressure altimeter. At sea level and standard atmospheric conditions, the linkage attached to the expandable diaphragm produces an indication of zero. When altitude increases, static pressure on the outside of the diaphragm decreases and the aneroid expands, producing a positive indication of altitude. When altitude decreases, atmospheric pressure increases. The static air pressure on the outside of the diaphragm increases and the pointer moves in the opposite direction, indicating a decrease in altitude.

completely around the dial one time, the second-longest point moves only the distance between two numerals indicating 1 000 feet of altitude has been attained. If so equipped, a third, shortest or thinnest pointer registers altitude in 10 000 foot increments. When this pointer reaches a numeral, 10 000 feet of altitude has been attained. Sometimes a black-and-white or red-and-white cross-hatched area is shown on the face on the instrument until the 10 000 foot level has been reached. (*Figure 5-32*)

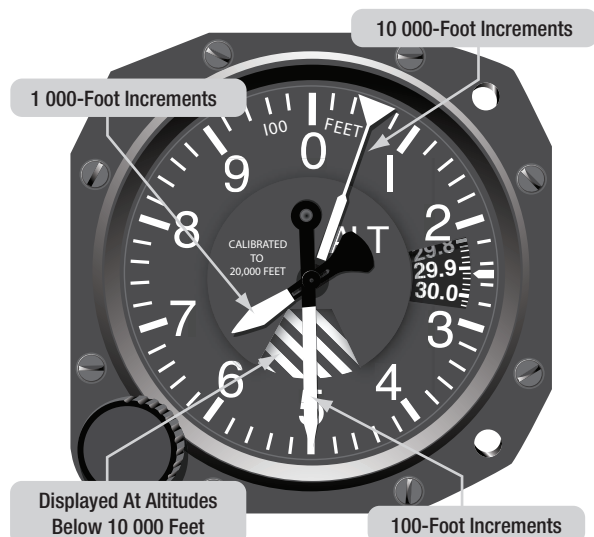


Figure 5-32. A sensitive altimeter with three pointers and a crosshatched area displayed during operation below 10 000 feet.

Many altimeters also contain linkages that rotate a numerical counter in addition to moving pointers around the dial. This quick reference window allows the pilot to simply read the numerical altitude in feet. The motion of the rotating digits or drum type counter during rapid climb or descent makes it difficult or impossible to read the numbers. Reference can then be directed to the classic clock style indication. *Figure 5-33* illustrates the inner workings behind this type of mechanical digital display of pressure altitude.

True digital instrument displays can show altitude in numerous ways. Use of a numerical display rather than a reproduction of the clock type dial is most common. Often a digital numeric display of altitude is given on the electronic primary flight display near the artificial horizon depiction. A linear vertical scale may also be presented to put this hard numerical value in perspective. An example of this type of display of altitude information is shown in *Figure 5-34*.

Accurate measurement of altitude is important for numerous reasons. The importance is magnified in instrument flight rules (IFR) conditions. For example, avoidance of tall obstacles and rising terrain relies on precise altitude indication, as does flying at a prescribed altitude assigned by air traffic control (ATC) to avoid colliding with other aircraft.

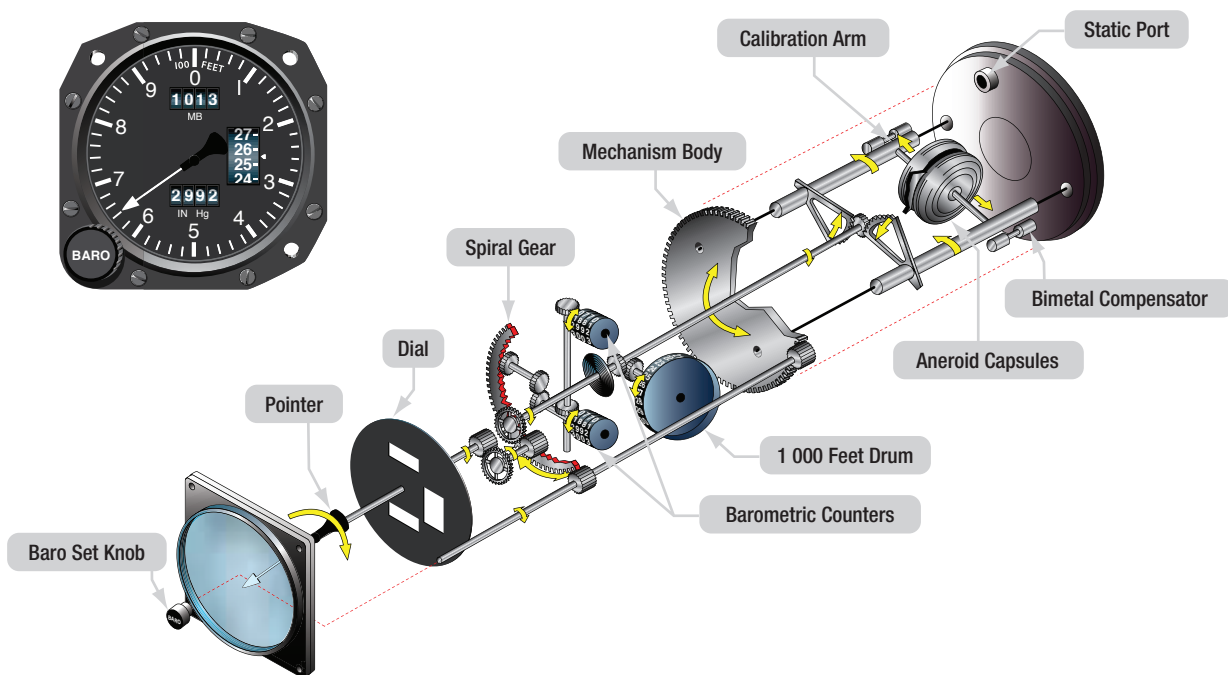


Figure 5-33. A drum-type counter can be driven by the altimeter's aneroid for numerical display of altitude. Drums can also be used for the altimeter's setting indications.



Figure 5-34. This primary flight display unit of a Garmin 1000 series glass cockpit instrumentation package for light aircraft indicates altitude using a vertical linear scale and a numerical counter. As the aircraft climbs or descends, the scale behind the black numerical altitude readout changes.

Measuring altitude with a pressure measuring device is fraught with complications. Steps are taken to refine pressure altitude indication to compensate for factors that may cause an inaccurate display. A major factor that affects pressure altitude measurements is the naturally occurring pressure variations throughout the atmosphere due to weather conditions.

Different air masses develop and move over the earth's surface, each with inherent pressure characteristics. These air masses cause the weather we experience, especially at the boundary areas between air masses known as fronts. Accordingly, at sea level, even if the temperature remains constant, air pressure rises and falls as weather system air masses come and go. The values in **Figure 5-30**, therefore, are averages for theoretical purposes.

To maintain altimeter accuracy despite varying atmospheric pressure, a means for setting the altimeter was devised. An adjustable pressure scale visible on the face of an analog altimeter known as a barometric or Kollsman window is set to read the existing atmospheric pressure when the pilot rotates the knob on the front

of the instrument. This adjustment is linked through gears inside the altimeter to move the altitude indicating pointers on the dial as well. By putting the current known air pressure (also known as the altimeter setting) in the window, the instrument indicates the actual altitude.

This altitude, adjusted for atmospheric pressure changes due to weather and air mass pressure inconsistency, is known as the indicated altitude. It must be noted that in flight the altimeter setting is changed to match that of the closest available weather reporting station or airport. This keeps the altimeter accurate as the flight progresses.

While there was little need for exact altitude measurement in early fixed wing aviation, knowing one's altitude provided the pilot with useful references while navigating in the three dimensions of the atmosphere. As air traffic grew and the desire to fly in any weather conditions increased, exact altitude measurement became more important and the altimeter was refined. In 1928, Paul Kollsman invented the means for adjusting an altimeter to reflect variations in air pressure from standard atmospheric pressure. The

very next year, Jimmy Doolittle made his successful flight demonstrating the feasibility of instrument flight with no visual references outside of the cockpit using a Kollsman sensitive altimeter.

The term pressure altitude is used to describe the indication an altimeter gives when 29.92 is set in the Kollsman window. When flying in U.S. airspace above 18 000 feet mean sea level (MSL), pilots are required to set their altimeters to 29.92. With all aircraft referencing this standard pressure level, vertical separation between aircraft assigned to different altitudes by ATC should be assured. This is the case if all altimeters are functioning properly and pilots hold their assigned altitudes. Note that the true altitude or actual height of an aircraft above sea level is only the same as the pressure altitude when standard day conditions exist. Otherwise, all aircraft with altimeters set to 29.92 "Hg could have true altitudes higher or lower than the pressure altitude indicated.

This is due to the pressure within the air mass in which they are flying being above or below standard day pressure (29.92). The actual or true altitude is less important than keeping aircraft from colliding, which is accomplished by all aircraft above 18 000 feet referencing the same pressure level (29.92 "Hg).

(Figure 5-35)

Temperature also affects the accuracy of an altimeter. The aneroid diaphragms used in altimeters are usually made of metal. Their elasticity changes as their temperature changes. This can lead to a false indication,

especially at high altitudes when the ambient air is very cold. A bimetallic compensating device is built into many sensitive altimeters to correct for varying temperature. **Figure 5-33** shows one such device on a drum type altimeter.

Temperature also affects air density, which has great impact on the performance of an aircraft. Although this does not cause the altimeter to produce an errant reading, flight crews must be aware that performance changes with temperature variations in the atmosphere. The term density altitude describes altitude corrected for nonstandard temperature. That is, the density altitude is the standard day altitude (pressure altitude) at which an aircraft would experience similar performance as it would on the non-standard day currently being experienced.

For example, on a very cold day, the air is denser than on a standard day, so an aircraft performs as though it is at a lower altitude. The density altitude is lower than that day. On a very hot day, the reverse is true, and an aircraft performs as though it were at a higher elevation where the air is less dense. The density altitude is higher than that day. Conversion factors and charts have been produced so pilots can calculate the density altitude on any particular day. Inclusion of nonstandard air pressure due to weather systems and humidity can also be factored. So, while the effects of temperature on aircraft performance do not cause an altimeter to indicate falsely, an altimeter indication can be misleading in terms of aircraft performance if these effects are not considered. (Figure 5-36)

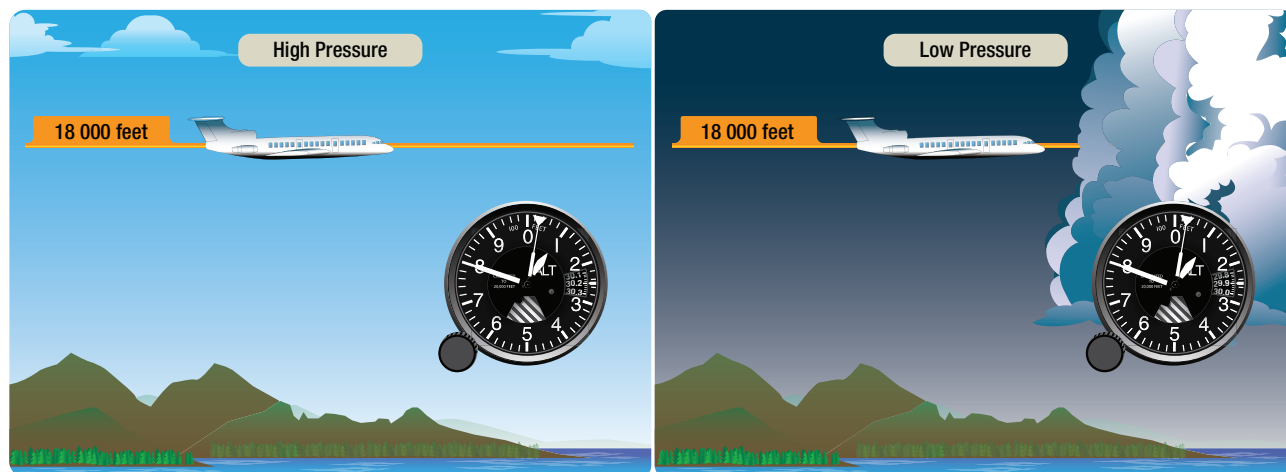


Figure 5-35. Above 18 000 feet MSL, all aircraft are required to set 29.92 as the reference pressure in the Kollsman window. The altimeter then reads pressure altitude. Depending on the atmospheric pressure that day, the true or actual altitude of the aircraft may be above or below what is indicated (pressure altitude).

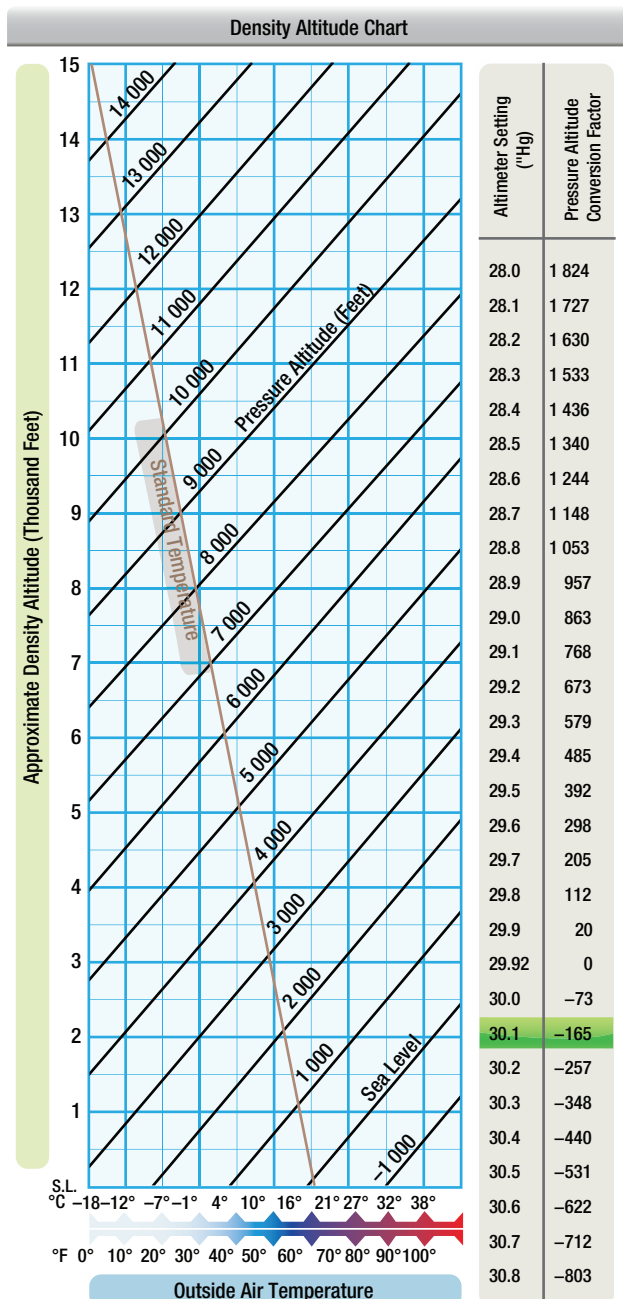


Figure 5-36. The effect of air temperature on aircraft performance is expressed as density altitude.

Other factors can cause an inaccurate altimeter indication. Scale error is a mechanical error whereby the scale of the instrument is not aligned so the altimeter pointers indicate correctly. Periodic testing and adjustment by trained technicians using calibrated equipment ensures scale error is kept to a minimum.

The pressure altimeter is connected to the pitot-static system and must receive an accurate sample of ambient air pressure to indicate the correct altitude. Position error, or installation error, is that inaccuracy caused by

the location of the static vent that supplies the altimeter. While every effort is made to place static vents in undisturbed air, airflow over the airframe changes with the speed and attitude of the aircraft. The amount of this air pressure collection error is measured in test flights, and a correction table showing the variances can be included with the altimeter for the pilot's use. Normally, location of the static vents is adjusted during these test flights so that the position error is minimal. (Figure 5-37) Position error can be removed by the ADC in modern aircraft, so the pilot need not be concerned about this inaccuracy.

Static system leaks can affect the static air input to the altimeter or ADC resulting in inaccurate altimeter indications. It is for this reason that static system maintenance includes leak checks every 24 months, regardless of whether any discrepancy has been noticed. See the instrument maintenance section toward the end of this chapter for further information on this mandatory check. It should also be understood that analog mechanical altimeters are mechanical devices that often reside in a hostile environment.

The significant vibration and temperature range swings encountered by the instruments and the pitot-static system (i.e., the tubing connections and fittings) can sometime create damage or a leak, leading to instrument malfunction. Proper care upon installation is the best preventive action. Periodic inspection and testing can also insure integrity.

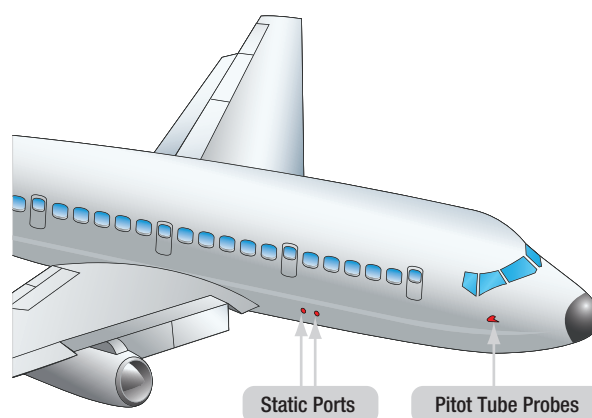


Figure 5-37. The location of the static vent is selected to keep altimeter position error to a minimum.

The mechanical nature of the analog altimeter's diaphragm pressure measuring apparatus has limitations. The diaphragm itself is only so elastic when responding to static air pressure changes. Hysteresis is the term for when the material from which the diaphragm is made takes a set during long periods of level flight. If followed by an abrupt altitude change, the indication lags or responds slowly while expanding or contracting during a rapid altitude change. While temporary, this limitation does cause an inaccurate altitude indication. It should be noted that many modern altimeters are constructed to integrate into flight control systems, autopilots, and altitude monitoring systems, such as those used by ATC. The basic pressure sensing operation of these altimeters is the same, but a means for transmitting the information is added.

Vertical Speed Indicator

An analog vertical speed indicator (VSI) may also be referred to as a vertical velocity indicator (VVI), or rate-of-climb indicator. It is a direct reading, differential pressure gauge that compares static pressure from the aircraft's static system directed into a diaphragm with static pressure surrounding the diaphragm in the instrument case.

Air is free to flow unrestricted in and out of the diaphragm but is made to flow in and out of the case through a calibrated orifice. A pointer attached to the diaphragm indicates zero vertical speed when the pressure inside and outside the diaphragm are the same. The dial is usually graduated in 100s of feet per minute. A zeroing adjustment screw, or knob, on the face of the instrument is used to center the pointer exactly on zero while the aircraft is on the ground. (*Figure 5-38*)

As the aircraft climbs, the unrestricted air pressure in the diaphragm lowers as the air becomes less dense. The case air pressure surrounding the diaphragm lowers more slowly, having to pass through the restriction created by the orifice. This causes unequal pressure inside and outside the diaphragm, which in turn causes the diaphragm to contract a bit and the pointer indicates a climb. The process works in reverse for an aircraft in a descent. If a steady climb or descent is maintained, a steady pressure differential is established between the diaphragm and case pressure surrounding it, resulting in an accurate indication of the rate of climb via graduations on the instrument face. (*Figure 5-39*)



Figure 5-38. A typical vertical speed indicator.

A shortcoming of the rate-of-climb mechanism as described is that there is a lag of six to nine seconds before a stable differential pressure can be established that indicates the actual climb or descent rate of the aircraft. An instantaneous vertical speed indicator (IVSI) has a built-in mechanism to reduce this lag. A small, lightly sprung dashpot, or piston, reacts to the direction change of an abrupt climb or descent. As this small accelerometer does so, it pumps air into or out of the diaphragm, hastening the establishment of the pressure differential that causes the appropriate indication. (*Figure 5-40*)

Gliders and lighter-than-air aircraft often make use of a variometer. This is a differential VSI that compares static pressure with a known pressure. It is very sensitive and gives an instantaneous indication. It uses a rotating vane with a pointer attached to it. The vane separates two chambers. One is connected to the aircraft's static vent or is open to the atmosphere.

The other is connected to a small reservoir inside the instrument that is filled to a known pressure. As static air pressure increases, the pressure in the static air chamber increases and pushes against the vane. This rotates the vane and pointer, indicating a descent since the static pressure is now greater than the set amount in the chamber with reservoir pressure. During a climb, the reservoir pressure is greater than the static pressure; the vane is pushed in the opposite direction, causing the pointer to rotate and indicate a climb. (*Figure 5-41*)

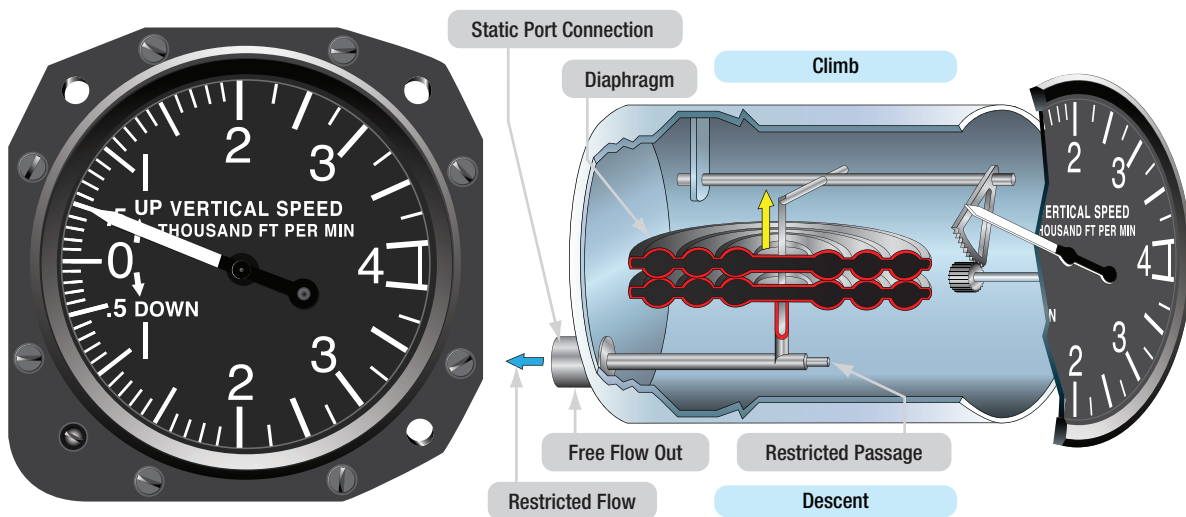


Figure 5-39. The VSI is a differential pressure gauge that compares free-flowing static air pressure in the diaphragm with restricted static air pressure around the diaphragm in the instrument case.

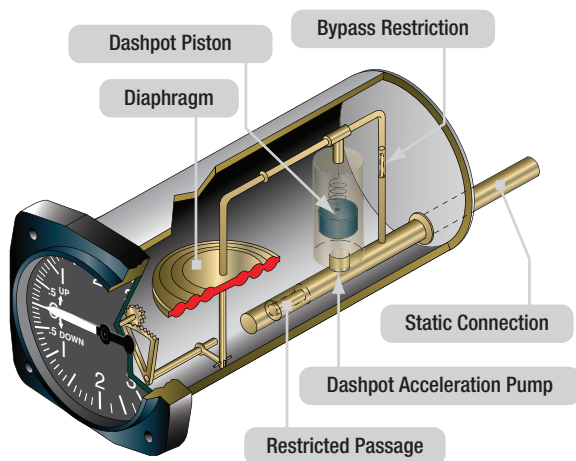


Figure 5-40. The small dashpot in this IVSI reacts abruptly to a climb or descent pumping air into or out of the diaphragm causing an instantaneously vertical speed indication.

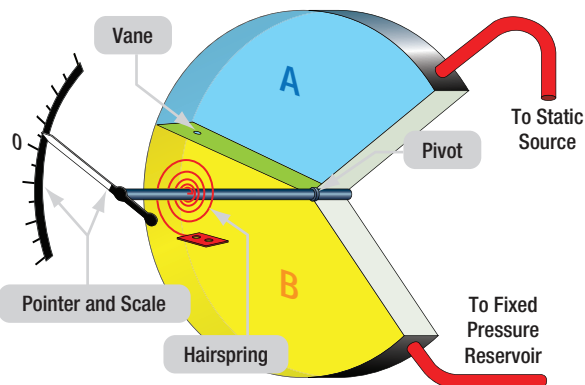


Figure 5-41. A variometer uses differential pressure to indicate vertical speed. A rotating vane separating two chambers (one with static pressure, the other with a fixed pressure reservoir), moves the pointer as static pressure changes.

The rate-of-climb indication in a digitally displayed instrument system is computed from static air input to the ADC. An aneroid, or solid-state pressure sensor, continuously reacts to changes in static pressure. The digital clock within the computer replaces the calibrated orifice found on an analog instrument. As the static pressure changes, the computer's clock can be used to develop a rate for the change. Using the known lapse rate conversion for air pressure as altitude increases or decreases, a figure for climb or descent in fpm can be calculated and sent to the cockpit. The vertical speed is often displayed near the altimeter information on the primary flight display. (*Figure 5-34*)

Airspeed Indicators

The airspeed indicator is another primary flight instrument that is also a differential pressure gauge. Ram air pressure from the aircraft's pitot tube is directed into a diaphragm in an analog airspeed instrument case. Static air pressure from the aircraft static vent(s) is directed into the case surrounding the diaphragm. As the speed of the aircraft varies, the ram air pressure varies, expanding or contracting the diaphragm. Linkage attached to the diaphragm causes a pointer to move over the instrument face, which is calibrated in knots or miles per hour (mph). (*Figure 5-42*)

The relationship between the ram air pressure and static air pressure produces the indication known as indicated airspeed. As with the altimeter, there are other factors that must be considered in measuring airspeed throughout all phases of flight. These can

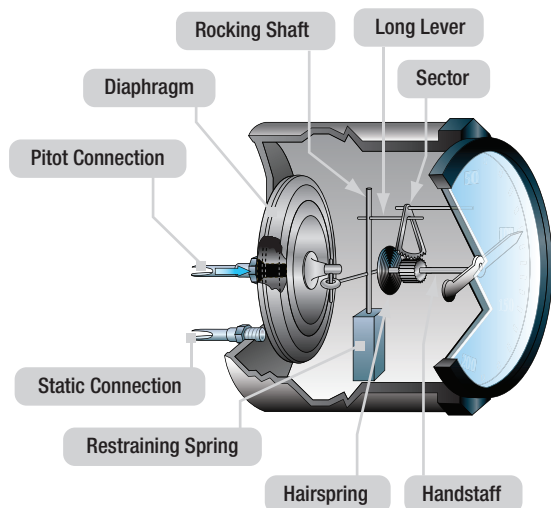


Figure 5-42. An airspeed indicator is a differential pressure gauge that compares ram air pressure with static pressure.

cause inaccurate readings or indications that are not useful to the pilot in a particular situation. In analog airspeed indicators, the factors are often compensated for with ingenious mechanisms inside the case and on the instrument dial face.

Digital flight instruments can have calculations performed in the ADC so the desired accurate indication is displayed. While the relationship between ram air pressure and static air pressure is the basis for most airspeed indications, it can be more accurate. Calibrated airspeed takes into account errors due to position error of the pitot-static pickups. It also corrects for the nonlinear nature of the pitot-static pressure differential when it is displayed on a linear scale. Analog airspeed indicators come with a correction chart that allows cross-referencing of indicated airspeed to calibrated airspeed for various flight conditions. These differences are typically very small and often are ignored. Digital instruments have these corrections performed in the ADC.

More importantly, indicated airspeed does not take into account temperature and air pressure differences needed to indicate true airspeed. These factors greatly affect airspeed indication. True airspeed, therefore, is the same as indicated airspeed when standard day conditions exist. But when atmospheric temperature or pressure varies, the relationship between the ram air pressure and static pressure alters. Analog airspeed instruments often include bimetallic temperature compensating devices that can alter the linkage movement between

the diaphragm and the pointer movement. There can also be an aneroid inside the airspeed indicator case that can compensate for non-standard pressures. Alternatively, true airspeed indicators exist that allow the pilot to set temperature and pressure variables manually with external knobs on the instrument dial. The knobs rotate the dial face and internal linkages to present an indication that compensates for nonstandard temperature and pressure, resulting in a true airspeed indication. (*Figure 5-43*)

Digital flight instrument systems perform all of the calculations for true airspeed in the ADC. Ram air from the pitot tube and static air from the static vent(s) are run into the sensing portion of the computer. Temperature information is also input. This information can be manipulated and calculations performed so a true airspeed value can be digitally sent to the cockpit for display. Refer to *Figure 5-34* for the display of airspeed information on the primary flight display on a light aircraft. Note that similar to its position in the standard T configuration of an analog cockpit, the airspeed indication is just left of the artificial horizon display.

Complications continue when considering airspeed indications and operating limitations. It is very important to keep high-speed aircraft from traveling faster than the speed of sound if they are not designed to do so. Even as an aircraft approaches the speed of sound,



Figure 5-43. An analog true airspeed indicator. The pilot manually aligns the outside air temperature with the pressure altitude scale, resulting in an indication of true airspeed.

certain parts on the airframe may experience airflows that exceed it. The problem with this is that near the speed of sound, shock waves can develop that can affect flight controls and, in some cases, can literally tear the aircraft apart if not designed for supersonic airflow. A further complication is that the speed of sound changes with altitude and temperature. So a safe true airspeed at sea level could put the aircraft in danger at altitude due to the lower speed of sound. (*Figure 5-44*)

In order to safeguard against these dangers, pilots monitor airspeed closely. A maximum allowable speed is established for the aircraft during certification flight testing. This speed is known as the critical Mach number or Mcrit. Mach is a term for the speed of sound. The critical Mach number is expressed as a decimal of Mach such as 0.8 Mach. This means 8/10 of the speed of sound, regardless of what the actual speed of sound is at any particular altitude. Many high performance aircraft are equipped with a Machmeter for monitoring Mcrit. The Machmeter is essentially an airspeed instrument that is calibrated in relation to Mach on the dial. Various scales exist for subsonic and supersonic aircraft. (*Figure 5-45*) In addition to the ram air, static air diaphragm arrangement, Machmeters also contain an altitude sensing diaphragm. It adjusts the input to the pointer so changes in the speed of sound due to altitude are incorporated into the indication.

Some aircraft use a Mach/airspeed indicator as shown in *Figure 5-46*. This two in one instrument contains separate mechanisms to display the airspeed and Mach number. A standard white pointer is used to indicate airspeed in knots against one scale. A red and white striped pointer is driven independently and is read against the Mach number scale to monitor maximum allowable speed.

REMOTE-SENSING AND INDICATION

It is often impractical or impossible to utilize direct reading gauges for information needed to be conveyed in the cockpit. Placing sensors at the most suitable location on the airframe or engine and transmitting the collected data electrically through wires to the displays in the cockpit is a widely used method of remote-sensing and indicating on aircraft. Many remote-sensing instrument systems consist simply of the sensing and transmitter unit and the cockpit indicator unit connected to each other by wires. For pressure flight instruments, the ADC and

Standard Altitude Temperature and the Speed of Sound		
Altitude (feet)	Temperature (°C)	Speed of Sound (knots)
Sea Level	15	661
2 000	11.11	657
4 000	8.88	652
6 000	3.33	648
8 000	1.11	643
10 000	-5	638
12 000	-8.88	633
14 000	-12.77	629
16 000	-16.66	624
18 000	-20.55	619
20 000	-24.44	614
22 000	-28.33	609
24 000	-32.77	604
26 000	-36.66	599
28 000	-40.55	594
30 000	-44.44	589
32 000	-48.33	584
34 000	-52.22	579
36 000	-56.11	574
38 000	-56.66	574
40 000	-56.66	574
42 000	-56.66	574
44 000	-56.66	574
46 000	-56.66	574
48 000	-56.66	574
50 000	-56.66	574

Figure 5-44. As temperatures fall at higher altitudes, the speed of sound is reduced.



Figure 5-45. A Machmeter indicates aircraft speed relative to the speed of sound.

pickup devices (pitot tubes, static vents, etc.) comprise the sensing and transmitter unit. Many aircraft collect sensed data in dedicated engine and airframe computers. There, the information can be processed.



Figure 5-46. A combination Mach/airspeed indicator shows airspeed with a white pointer and Mach number with a red and white striped pointer. Each pointer is driven by separate internal mechanisms.

An output section of the computer then transmits it electrically or digitally to the cockpit for display. Remote-sensing instrument systems operate with high reliability and accuracy. They are powered by the aircraft's electrical system.

Small electric motors inside the instrument housings are used to position the pointers, instead of direct-operating mechanical linkages. They receive electric current from the output section of the ADC or other computers. They also receive input from sensing transmitters or transducers that are remotely located on the aircraft. By varying the electric signal, the motors are turned to the precise location needed to reflect the correct indication. Direct electric transmission of information from different types of sensors is accomplished with a few reliable and relatively simple techniques. Note that digital cockpit displays receive all of their input from a DADC and other computers, via a digital data bus and do not use electric motors. The data packages transmitted via the bus contain the instructions on how to illuminate the display screen.

SYNCHRO TYPE REMOTE-INDICATING INSTRUMENTS

A synchro system is an electric system used for transmitting information from one point to another. The word synchro is a shortened form of the word synchronous and refers to any one of a number of similarly operating two unit electrical systems capable of measuring, transmitting, and indicating a certain

parameter on the aircraft. Most position-indicating instruments are designed around a synchro system, such as the flap position indicator. Fluid pressure indicators also commonly use synchro systems. Synchro systems are used as remote position indicators for landing gear, autopilot systems, radar, and many other remote-indicating applications. The most common types of synchro system are the autosyn, selsyn, and magnesyn synchro systems.

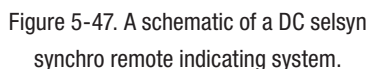
These systems are similar in construction, and all operate by exploiting the consistent relationship between electricity and magnetism. The fact that electricity can be used to create magnetic fields that have definite direction, and that magnetic fields can interact with magnets and other electromagnetic fields, is the basis of their operation.

DC Selsyn Systems

On aircraft with direct current (DC) electrical systems, the DC selsyn system is widely used. As mentioned, the selsyn system consists of a transmitter, an indicator, and connecting wires. The transmitter consists of a circular resistance winding and a rotatable contact arm.

The rotatable contact arm turns on a shaft in the center of the resistance winding. The two ends of the arm are brushes and always touch the winding on opposite sides. (*Figure 5-47*) On position indicating systems, the shaft to which the contact arm is fastened protrudes through the end of transmitter housing and is attached to the unit whose position is to be transmitted (e.g., flaps, landing gear). The transmitter is often connected to the moving unit through a mechanical linkage. As the unit moves, it causes the transmitter shaft to turn. The arm is turned so that voltage is applied through the brushes to any two points around the circumference of the resistance winding. The rotor shaft of DC selsyn systems, measuring other kinds of data, operates the same way, but may not protrude outside of the housing. The sensing device, which imparts rotary motion to the shaft, could be located inside the transmitter housing.

Referring to *Figure 5-47*, note that the resistance winding of the transmitter is tapped off in three fixed places, usually 120° apart. These taps distribute current through the toroidal windings of the indicator motor. When current flows through these windings, a magnetic field is created. Like all magnetic fields, a definite north



The resultant magnetic field created by current flowing through the indicator coils changes as each receives varied current from the tap-offs. The direction of the magnetic field also changes. Thus, the direction of the magnetic field across the indicating element corresponds in position to the moving arm in the transmitter. A permanent magnet is attached to the centered rotor shaft in the indicator, as is the indicator pointer. The magnet aligns itself with the direction of the magnetic field and the pointer does as well. Whenever the magnetic field changes direction, the permanent magnet and pointer realign with the new position of the field. Thus, the position of the aircraft device is indicated.

This changes the total resistance of that section. The result is a change in the current flowing through one of the indicator's motor coils. This, in turn, changes the magnetic field around that coil.

Therefore, the combined magnetic field created by all three motor coils is also affected, causing a shift in the direction of the indicator's magnetic field. The permanent magnet and pointer align with the new direction and shift to the locked position on the indicator dial. **Figure 5-48** shows a simplified diagram of a lock switch in a three wire selsyn system and an indicator dial.

Aircraft with alternating current (AC) electrical power systems make use of autosyn or magnasyn synchro remote indicating systems. Both operate in a similar way to the DC selsyn system, except that AC power is used. Thus, they make use of electric induction, rather than resistance current flows defined by the rotor brushes. Magnasyn systems use permanent magnet rotors such as those found in the DC selsyn system. Usually, the transmitter magnet is larger than the indicator magnet, but the electromagnetic response of the indicator rotor magnet and pointer remains the same. It aligns with the magnetic field set up by the coils, adopting the same angle of deflection as the transmitter rotor. (*Figure 5-49*)

Autosyn systems are further distinguished by the fact that the transmitter and indicator rotors used are electro-magnets rather than permanent magnets. Nonetheless, like a permanent magnet, an electro-magnet aligns with the direction of the magnetic field created by current flowing through the stator coils in the indicator. Thus, the indicator pointer position mirrors the transmitter rotor position. (*Figure 5-50*)

AC synchro systems are wired differently than DC systems. The varying current flows through the transmitter and indicator stator coils are induced as the AC cycles through zero and the rotor magnetic field flux is allowed to flow. The important characteristic of all synchro systems is maintained by both the autosyn and magnasyn systems. That is, the position of the transmitter rotor is mirrored by the rotor in the indicator. These systems are used in many of the same applications as the DC systems and more. Since they are usually part of instrumentation for high performance

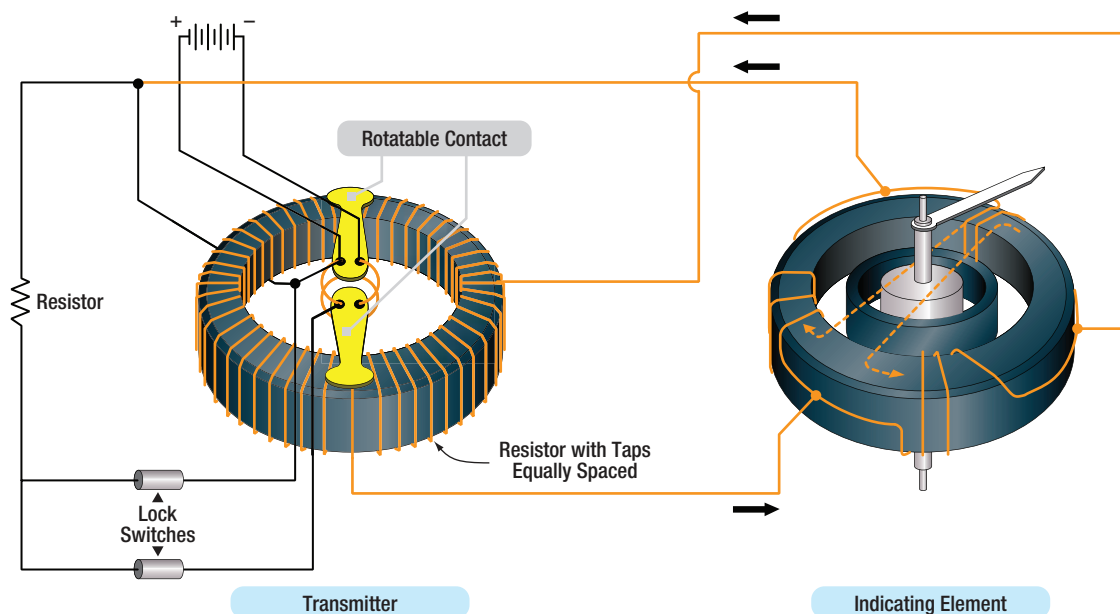


Figure 5-48. A lock switch circuit can be added to the basic DC selsyn synchro system when used to indicate landing gear position and up-and down-locked conditions on the same indicator.

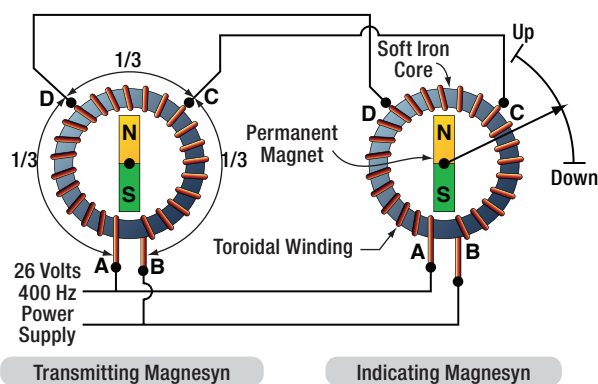


Figure 5-49. A magnasyn synchro remote-indicating system uses AC. It has permanent magnet rotors in the transmitter and indicator.

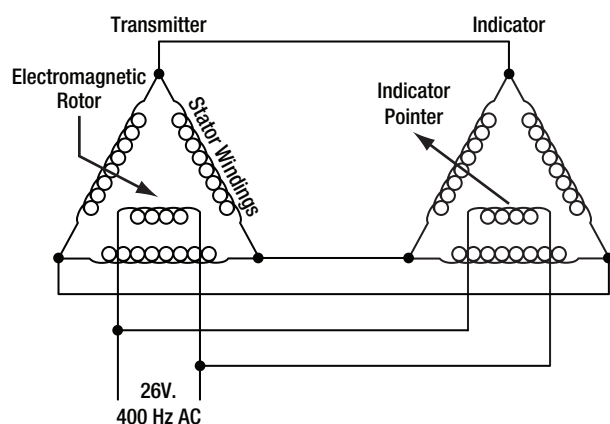


Figure 5-50. An autosyn remote-indicating system utilizes the interaction between magnetic fields set up by electric current flow to position the indicator pointer.

aircraft, adaptations of autosyn and magnasyn synchro systems are frequently used in directional indicators and in autopilot systems.

Remote Indicating Fuel and Oil Pressure Gauges

Fuel and oil pressure indications can be conveniently obtained through the use of synchro systems. As stated previously, running fuel and oil lines into the cabin to direct reading gauges is not desirable. Increased risk of fire in the cabin and the additional weight of the lines are two primary deterrents. By locating the transmitter of a synchro system remotely, fluid pressure can be directed into it without a long tubing run. Inside the transmitter, the motion of a pressure bellows can be geared to the transmitter rotor in such a way as to make the rotor turn. (Figure 5-51)

As in all synchros, the transmitter rotor turns proportional to the pressure sensed, which varies the voltages set up in the resistor windings of the synchro stator. These voltages are transmitted to the indicator coils that develop the magnetic field that positions the pointer.

Often on twin-engine aircraft, synchro mechanisms for each engine can be used to drive separate pointers on the same indicator. By placing the coils one behind the other, the pointer shaft from the rear indicator motor can be sent through the hollow shaft of the forward

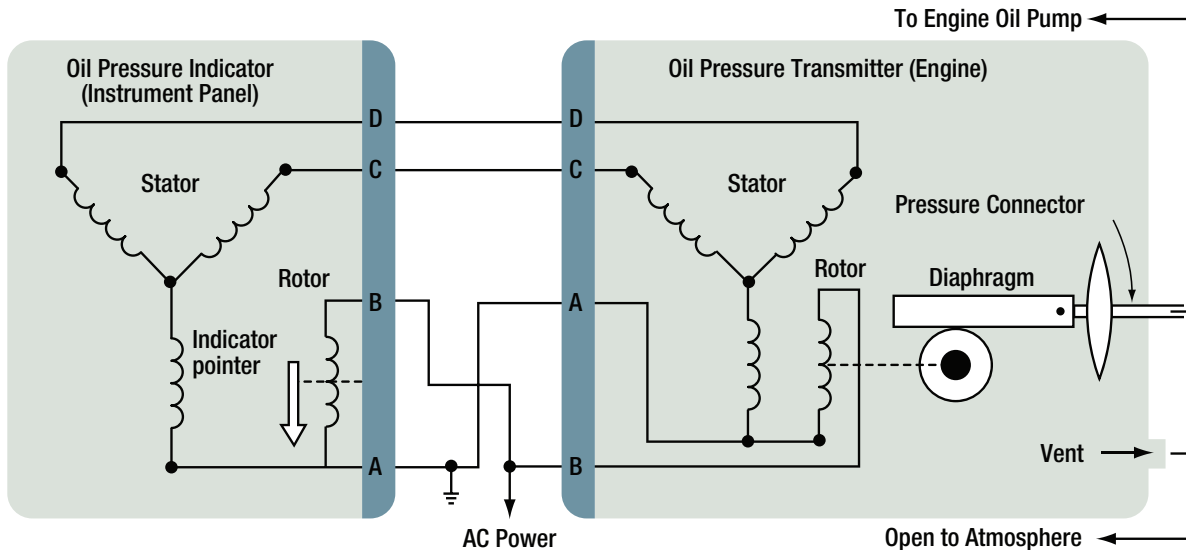


Figure 5-51. Remote pressure sensing indicators change linear motion to rotary motion in the sensing mechanism part of the synchro transmitter.

indicator motor. Thus, each pointer responds with the magnet's alignment in its own motor's magnetic field while sharing the same gauge housing. Labeling the pointers engine 1 or 2 removes any doubt about which indicator pointer is being observed.

A similar principle is employed in an indicator that has side-by-side indications for different parameters, such as oil pressure and fuel pressure in the same indicator housing. Each parameter has its own synchro motor for positioning its pointer.

Aircraft with digital instrumentation make use of pressure sensitive solid-state sensors that output digital signals for collection and processing by dedicated engine and airframe computers. Others may retain their analog sensors, but may forward this information through an analog to digital converter unit from which the appropriate computer can obtain digital information to process and illuminate the digital display. Many more instruments utilize the synchro remote indicating systems described in this section or similar synchros. Sometimes simple, more suitable, or less expensive technologies are also employed.

MECHANICAL MOVEMENT INDICATORS

There are many instruments on an aircraft that indicate the mechanical motion of a component, or even the aircraft itself. Some utilize the synchro remote-sensing and indicating systems described above. Other means for capturing and displaying mechanical movement

information are also used. This section discusses some unique mechanical motion indicators and groups instruments by function. All give valuable feedback to the pilot on the condition of the aircraft in flight.

TACHOMETERS

The tachometer, or tach, is an instrument that indicates the speed of the crankshaft of a reciprocating engine. It can be a direct-or remote-indicating instrument, the dial of which is calibrated to indicate revolutions per minutes (rpm). On reciprocating engines, the tach is used to monitor engine power and to ensure the engine is operated within certified limits. Gas turbine engines also have tachometers. They are used to monitor the speed(s) of the compressor section(s) of the engine. Turbine engine tachometers are calibrated in percentage of rpm with 100 percent corresponding to optimum turbine speed. This allows similar operating procedures despite the varied actual engine rpm of different engines. (*Figure 5-52*)

In addition to the engine tachometer, helicopters use a tachometer to indicator main rotor shaft rpm. It should also be noted that many reciprocating-engine tachometers also have built-in numeric drums that are geared to the rotational mechanism inside. These are hour meters that keep track of the time the engine is operated. There are two types of tachometer system in wide use today: mechanical and electrical.

MECHANICAL TACHOMETERS

Mechanical tachometer indicating systems are found on small, single-engine light aircraft in which a short distance exists between the engine and the instrument



Figure 5-52. A tachometer for a reciprocating engine is calibrated in rpm. A tachometer for a turbine engine is calculated in percent of rpm.

panel. They consist of an indicator connected to the engine by a flexible drive shaft. The drive shaft is geared into the engine so that when the engine turns, so does the shaft. The indicator contains a flyweight assembly coupled to a gear mechanism that drives a pointer. As the drive shaft rotates, centrifugal force acts on the flyweights and moves them to an angular position. This angular position varies with the rpm of the engine.

The amount of movement of the flyweights is transmitted through the gear mechanism to the pointer. The pointer rotates to indicate this movement on the tachometer indicator, which is directly related to the rpm of the engine. (Figure 5-53)

A more common variation of this type of mechanical tachometer uses a magnetic drag cup to move the pointer in the indicator. As the drive shaft turns, it rotates a permanent magnet in a close tolerance aluminum cup. A shaft attached to the indicating point is attached to the exterior center of the cup. As the magnet is rotated by the engine flex drive cable, its magnetic field cuts through the conductor surrounding it, creating eddy currents in the aluminum cup. This current flow creates its own magnetic field, which interacts with the rotating magnets flux field. The result is that the cup tends to rotate, and with it, the indicating pointer. A calibrated restraining spring limits the cups rotation to the arc of motion of the pointer across the scale on the instrument face. (Figure 5-54)

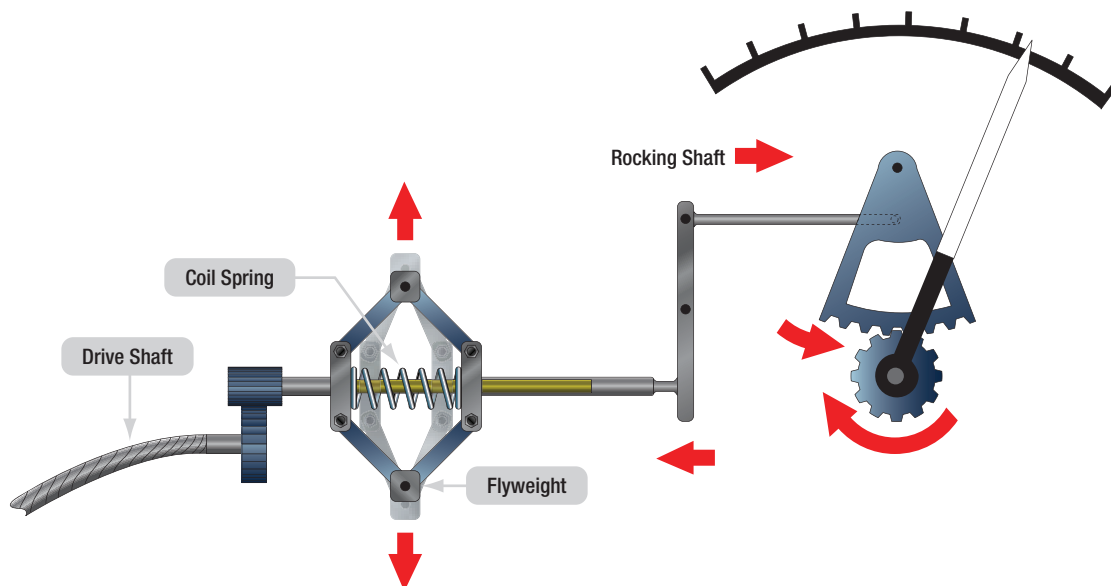


Figure 5-53. The simplified mechanism of a flyweight type mechanical tachometer.

ELECTRIC TACHOMETERS

It is not practical to use a mechanical linkage between the engine and the rpm indicator on aircraft with engines not mounted in the fuselage just forward of the instrument panel. Greater accuracy with lower maintenance is achieved through the use of electric tachometers.

A wide variety of electric tachometer systems can be employed, so manufacturers instructions should be consulted for details of each specific tachometer system. A popular electric tachometer system makes use of a small AC generator mounted to a reciprocating engine's gear case or the accessory drive section of a turbine engine. As the engine turns, so does the generator. The frequency output of the generator is directly proportional to the speed of the engine. It is connected via wires to

a synchronous motor in the indicator that mirrors this output. A drag cup, or drag disk link, is used to drive the indicator as in a mechanical tachometer. (*Figure 5-55*) Two different types of generator units, distinguished by their type of mounting system, are shown in *Figure 5-56*.

The dual tachometer consists of two tachometer indicator units housed in a single case. The indicator pointers show simultaneously, on one or two scales, the rpm of two engines. A dual tachometer on a helicopter often shows the rpm of the engine and the rpm of the main rotor. A comparison of the voltages produced by the two tach generators of this type of helicopter indicator gives information concerning clutch slippage. A third indication showing this slippage is sometimes included in the helicopter tachometer. (*Figure 5-57*)

Some turbine engines use tachometer probes for rpm indication, rather than a tach generator system. They provide a great advantage in that there are no moving parts. They are sealed units that are mounted on a flange and protrude into the compressor section of the engine. A magnetic field is set up inside the probe that extends through pole pieces and out the end of the probe. A rotating gear wheel, which moves at the same speed as

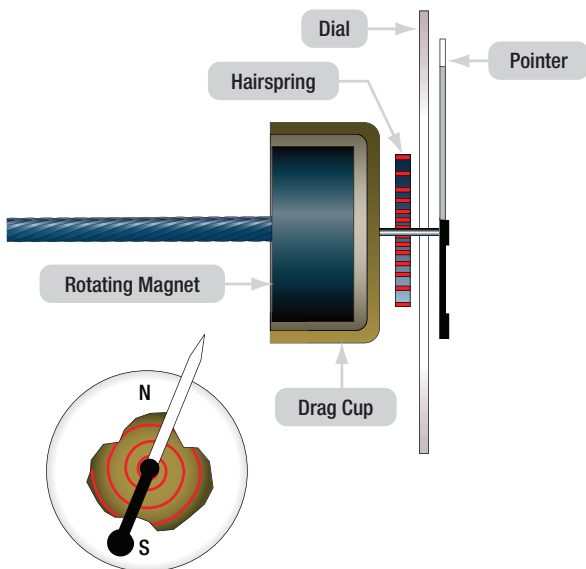


Figure 5-54. A simplified magnetic drag cup tachometer indicating device.

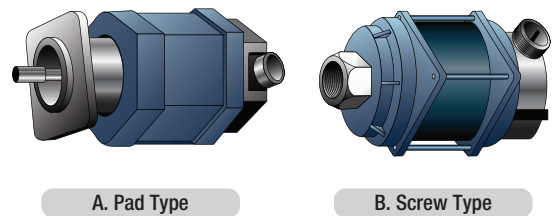


Figure 5-56. Different types of tach generators.

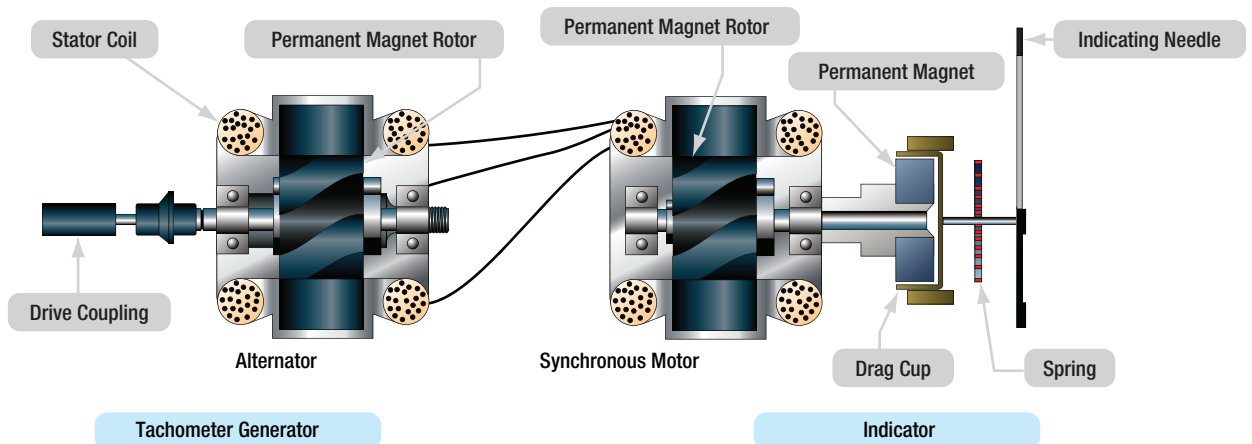


Figure 5-55. An electric tachometer system with synchronous motors and a drag cup indicator.

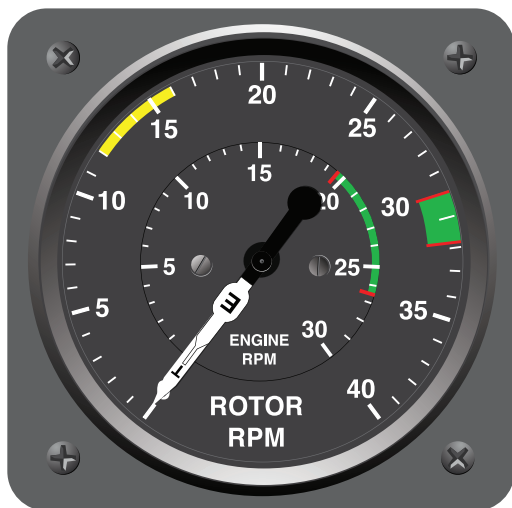


Figure 5-57. A helicopter tachometer with engine rpm, rotor rpm, and slippage indications.

the engine compressor shaft, alters the magnetic field flux density as it moves past the pole pieces at close proximity. This generates voltage signals in coils inside the probe. The amplitude of the EMF signals vary directly with the speed of the engine.

The tachometer probe's output signals need to be processed in a remotely located module. They must also be amplified to drive a servo motor type indicator in the cockpit. They may also be used as input for an automatic power control system or a flight data acquisition system. (*Figure 5-58*)

Synchroscope

The synchroscope is an instrument that indicates whether two or more rotating devices, such as engines, are synchronized. Since synchrosopes compare rpm, they utilize the output from tachometer generators. The instrument consists of a small electric motor that receives electrical current from the generators of both engines. Current from the faster running engine controls the direction in which the synchroscope motor rotates.

If both engines are operating at exactly the same speed, the synchroscope motor does not operate. If one engine operates faster than the other, its tach generator signal causes the synchroscope motor to turn in a given direction. Should the speed of the other engine then become greater than that of the first engine, the signal from its tach generator causes the synchroscope motor to reverse itself and turn in the opposite direction.

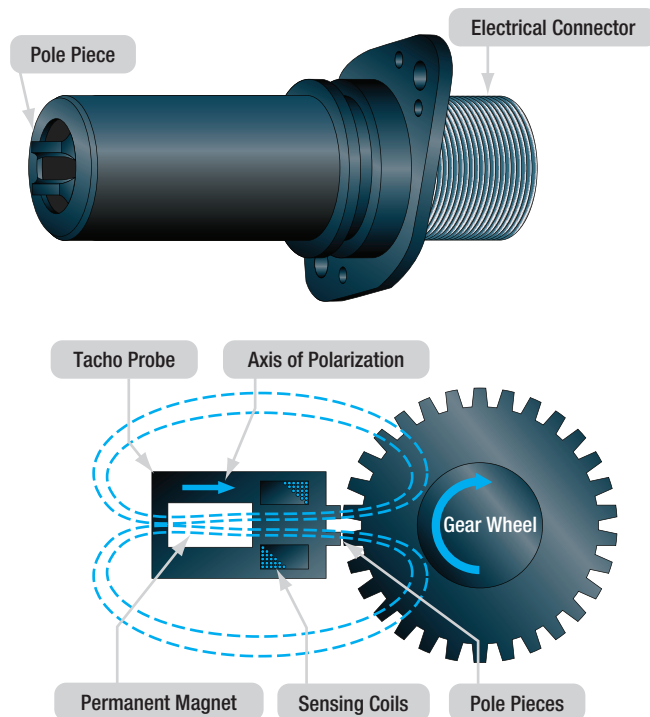


Figure 5-58. A tacho probe has no moving parts. The rate of magnetic flux field density change is directly related to engine speed.

The pilot makes adjustments to steady the pointer so it does not move. One use of synchroscope involve designating one of the engines as a master engine. The rpm of the other engine(s) is always compared to the rpm of this master engine. The dial face of the synchroscope indicator looks like *Figure 5-59*. "Slow" and "fast" represent the other engine's rpm relative to the master engine, and the pilot makes adjustments accordingly.

Accelerometers

An accelerometer is an instrument that measures acceleration. It is used to monitor the forces acting upon an airframe. Accelerometers are also used in inertial reference navigation systems. The installation of accelerometers is usually limited to high-performance and aerobatic aircraft.

Simple accelerometers are mechanical, direct-reading instruments calibrated to indicate force in Gs. One G is equal to one times the force of gravity. The dial face of an accelerometer is scaled to show positive and negative forces. When an aircraft initiates a rapid climb, positive G force tends to push one back into one's seat. Initiating a rapid decent causes a force in the opposite direction, resulting in a negative G force.



Figure 5-59. This synchroscope indicates the relative speed of the slave engine to the master.

Most accelerometers have three pointers. One is continuously indicating the acceleration force experienced. The other two contain ratcheting devices. The positive G pointer follows the continuous pointer and stay at the location on the dial where the maximum positive force is indicated. The negative G pointer does the same for negative forces experienced. Both max force pointers can be reset with a knob on the instrument face.

The accelerometer operates on the principle of inertia. A mass, or weight, inside is free to slide along a shaft in response to the slightest acceleration force. When a maneuver creates an accelerating force, the aircraft and instrument move, but inertia causes the weight to stay at rest in space. As the shaft slides through the weight, the relative position of the weight on the shaft changes. This position corresponds to the force experienced. Through a series of pulleys, springs, and shafts, the pointers are moved on the dial to indicate the relative strength of the acceleration force. (*Figure 5-60*)

Forces can act upon an airframe along the three axes of flight. Single and multi-axis accelerometers are available, although most cockpit gauges are of the single-axis type. Inertial reference navigation systems make use of multi-axis accelerometers to continuously, mathematically calculate the location of the aircraft in a three dimensional plane.

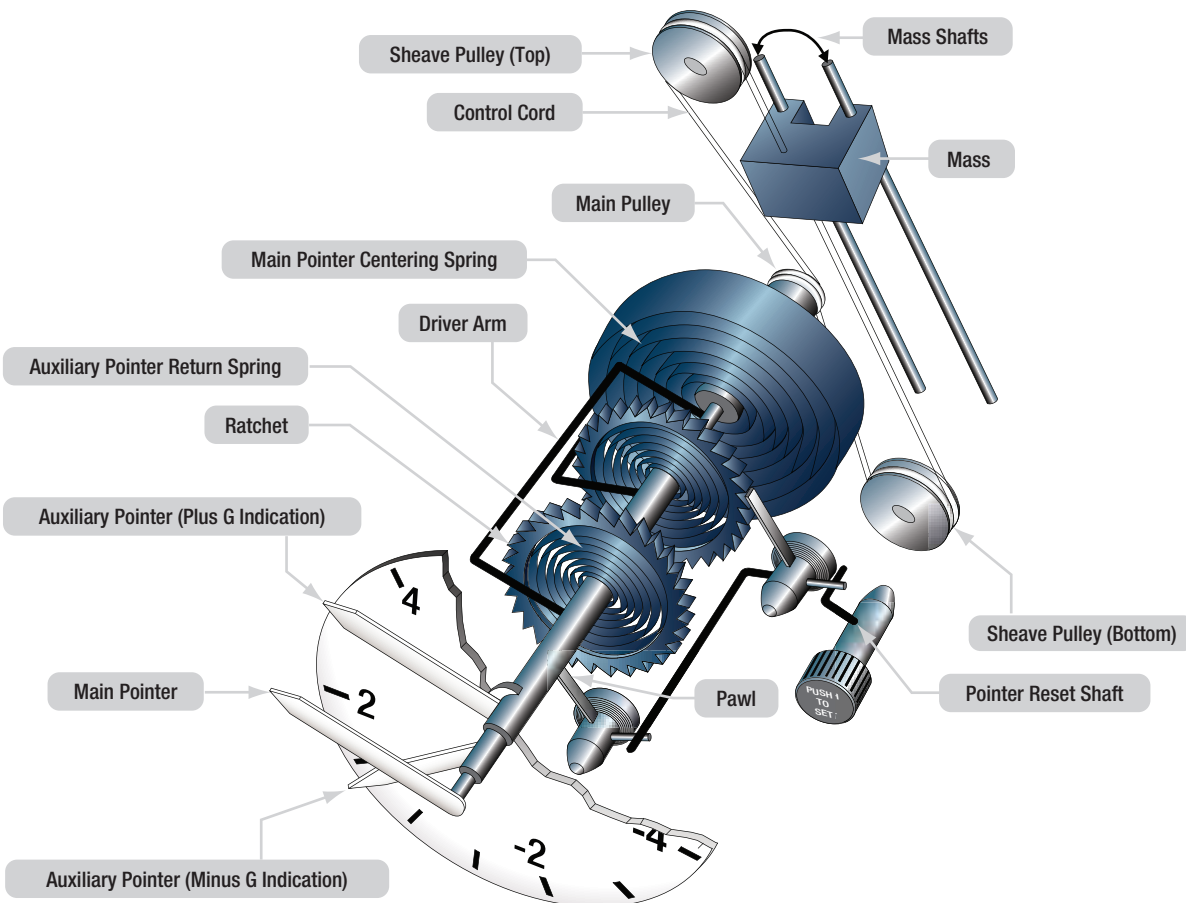


Figure 5-60. The inner workings of a mass type accelerometer.

Electric and digital accelerometers also exist. Solid-state sensors are employed, such as piezoelectric crystalline devices. In these instruments, when an accelerating force is applied, the amount of resistance, current flow, or capacitance changes in direct relationship to the size of the force. Microelectric signals integrate well with digital computers designed to process and display information in the cockpit.

STALL WARNING AND ANGLE OF ATTACK (AOA) INDICATORS

An aircraft's angle of attack (AOA) is the angle formed between the wing cord centerline and the relative wind. At a certain angle, airflow over the wing surfaces is insufficient to create enough lift to keep the aircraft flying, and a stall occurs. An instrument that monitors the AOA allows the pilot to avoid such a condition.

The simplest form of AOA indicator is a stall warning device that does not have a gauge located in the cockpit. It uses an aural tone to warn of an impending stall due to an increase in AOA. This is done by placing a reed in a cavity just aft of the leading edge of the wing. The cavity has an open passage to a precise point on the leading edge.

In flight, air flows over and under a wing. The point on the wing leading edge where the oncoming air diverges is known as the point of stagnation. As the AOA of the wing increases, the point of stagnation moves down below the open passage that leads inside the wing to the reed. Air flowing over the curved leading edge speeds up and causes a low pressure. This causes air to be sucked out of the inside of the wing through the passage. The reed vibrates as the air rushes by making a sound audible in the cockpit. (*Figure 5-61*)

Another common device makes use of an audible tone as the AOA increases to near the point where the aircraft will stall. This stall warning device includes an electric switch that opens and closes a circuit to a warning horn audible in the cockpit. It may also be wired into a warning light circuit.

The switch is located near the point of stagnation on the wing leading edge. A small lightly sprung tab activates the switch. At normal AOA, the tab is held down by air that diverges at the point of stagnation and flows under the wing. This holds the switch open so the horn



Figure 5-61. A reed type stall warning device is located behind this opening in the leading edge of the wing. When the angle of attack increases to near the point of a stall, low-pressure air flowing over the opening causes a suction, which audibly vibrates the reed.

does not sound nor the warning light illuminate. As the AOA increases, the point of stagnation moves down. The divergent air that flows up and over the wing now pushes the tab upward to close the switch and complete the circuit to the horn or light. (*Figure 5-62*)

A true AOA indicating system detects the local AOA of the aircraft and displays the information on a cockpit indicator. It also may be designed to furnish reference information to other systems on high-performance aircraft. The sensing mechanism and transmitter are usually located on the forward side of the fuselage.

It typically contains a heating element to ensure ice-free operation. Signals are sent from the sensor to the cockpit or computer(s) as required. An AOA indicator may be calibrated in actual angle degrees, arbitrary units, percentage of lift used, symbols, or even fast/slow. (*Figure 5-63*) There are two main types of AOA sensors in common use. Both detect the angular difference between the relative wind and the fuselage, which is used as a reference plane. One uses a vane, known as an alpha vane, externally mounted to the outside of the fuselage. It is free to rotate in the wind.

As the AOA changes, air flowing over the vane changes its angle. The other uses two slots in a probe that extends out of the side of the fuselage into the airflow. The slots lead to different sides of movable paddles in a chamber of the unit just inside the fuselage skin. As the AOA varies, the air pressure ported by each of the slots changes and the paddles rotate to neutralize the pressures.

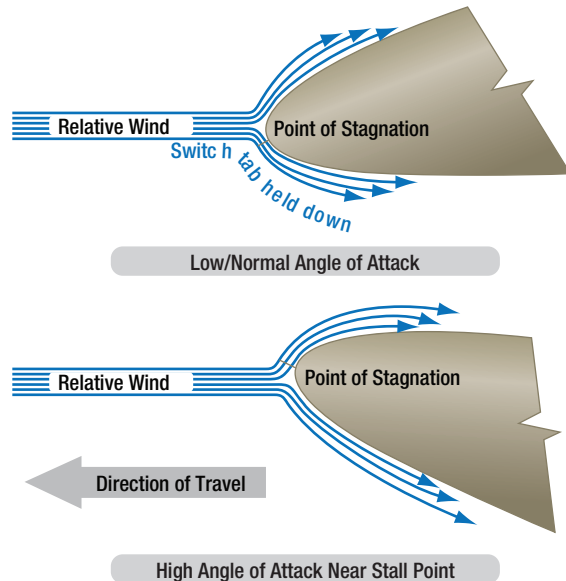
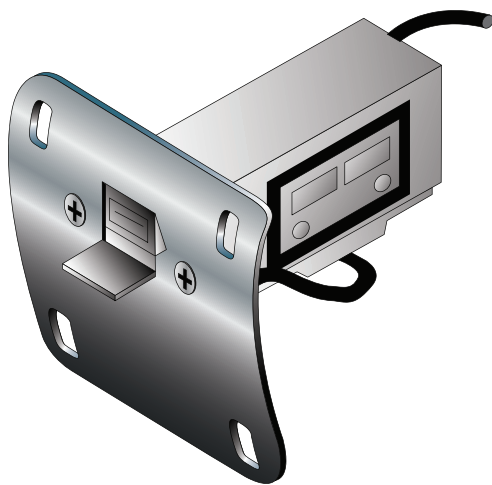


Figure 5-62. A popular stall warning switch located in the wing leading edge.



Figure 5-63. Angle of attack indicator.

The shaft upon which the paddles rotate connects to a potentiometer wiper contact that is part of the unit. The same is true of the shaft of the alpha vane. The changing resistance of the potentiometer is used in a balanced bridge circuit to signal a motor in the indicator to move the pointer proportional to the AOA. (Figure 5-64 and Figure 5-65)

Modern aircraft AOA sensor units send output signals to the ADC. There, the AOA data is used to create an AOA indication, usually on the primary flight display. AOA information can also be integrated with flap and slat position information to better determine the point of stall. Additionally, AOA sensors of the type described are subject to position error since airflow around the alpha vane and slotted probe changes somewhat with airspeed and aircraft attitude. The errors are small, but can be corrected in the ADC.

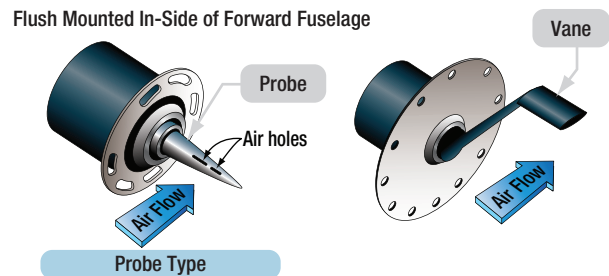


Figure 5-64. A slotted AOA probe and an alpha vane.

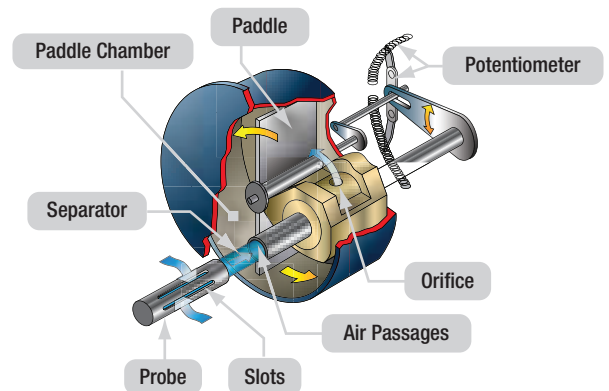


Figure 5-65. The internal structure of a slotted probe airstream direction detector.

To incorporate a warning of an impending stall, many AOA systems signal a stick shaker motor that literally shakes the control column to warn the pilot as the aircraft approaches a stall condition. Electrical switches are actuated in the AOA indicator at various preset AOA to activate the motor that drives an unbalanced weighted ring, causing the column to shake.

Some systems include a stick pusher actuator that pushes the control yoke forward, lowering the nose of the aircraft when the critical AOA is approached. Regardless of the many existing variations for warning of an impending stall, the AOA system triggers all stall warnings in high performance aircraft.

TEMPERATURE MEASURING INSTRUMENTS

The temperature of numerous items must be known for an aircraft to be operated properly. Engine oil, carburetor mixture, inlet air, free air, engine cylinder heads, heater ducts, and exhaust gas temperature of turbine engines are all items requiring temperature monitoring. Many other temperatures must also be known. Different types of thermometers are used to collect and present temperature information.

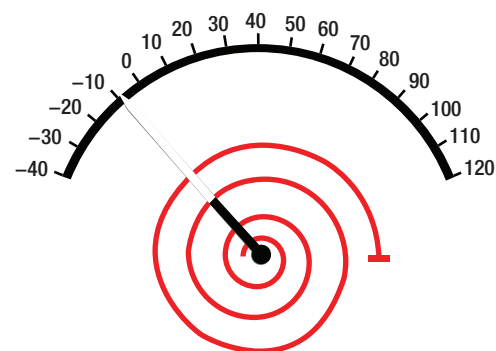
NON-ELECTRIC TEMPERATURE INDICATORS

The physical characteristics of most materials change when exposed to changes in temperature. The changes are consistent, such as the expansion or contraction of solids, liquids, and gases. The coefficient of expansion of different materials varies and it is unique to each material. Most everyone is familiar with the liquid mercury thermometer. As the temperature of the mercury increases, it expands up a narrow passage that has a graduated scale upon it to read the temperature associated with that expansion. The mercury thermometer has no application in aviation. A bimetallic thermometer is very useful in aviation. The temperature sensing element of a bimetallic thermometer is made of two dissimilar metals strips bonded together. Each metal expands and contracts at a different rate when temperature changes. One end of the bimetallic strip is fixed, the other end is coiled. A pointer is attached to the coiled end which is set in the instrument housing.

When the bimetallic strip is heated, the two metals expand. Since their expansion rates differ and they are attached to each other, the effect is that the coiled end tries to uncoil as the one metal expands faster than the other. This moves the pointer across the dial face of the instrument. When the temperature drops, the metals contract at different rates, which tends to tighten the coil and move the pointer in the opposite direction.

Direct reading bimetallic temperature gauges are often used in light aircraft to measure free air temperature or outside air temperature (OAT). In this application, a collecting probe protrudes through the windshield of the aircraft to be exposed to the atmospheric air. The coiled end of the bimetallic strip in the instrument head is just inside the windshield where it can be read by the pilot. (*Figure 5-66 and Figure 5-67*)

A bourdon tube is also used as a direct reading non-electric temperature gauge in simple, light aircraft. By calibrating the dial face of a bourdon tube gauge with a temperature scale, it can indicate temperature. The basis for operation is the consistent expansion of the vapor produced by a volatile liquid in an enclosed area. This vapor pressure changes directly with temperature. By filling a sensing bulb with such a volatile liquid and connecting it to a bourdon tube, the tube causes an



Bimetallic Temperature Gauge



Bimetallic coil of bonded metals with dissimilar coefficients of expansion.

Figure 5-66. A bimetallic temperature gauge works because of the dissimilar coefficients of expansion of two metals bonded together. When bent into a coil, cooling or heating causes the dissimilar metal coil to tighten, or unwind, moving the pointer across the temperature scale on the instrument dial face.



Figure 5-67. A bimetallic outside air temperature gauge and its installation on a light aircraft.

indication of the rising and falling vapor pressure due to temperature change. Calibration of the dial face in degrees Fahrenheit or Celsius, rather than psi, provides a temperature reading. In this type of gauge, the sensing bulb is placed in the area needing to have temperature measured. A long capillary tube connects the bulb to the bourdon tube in the instrument housing.

The narrow diameter of the capillary tube ensures that the volatile liquid is lightweight and stays primarily in the sensor bulb. Oil temperature is sometimes measured this way.

ELECTRICAL TEMPERATURE MEASURING INDICATION

The use of electricity in measuring temperature is very common in aviation. The following measuring and indication systems can be found on many types of aircraft. Certain temperature ranges are more suitably measured by one or another type of system.

Electrical Resistance Thermometer

The principle parts of the electrical resistance thermometer are the indicating instrument, the temperature-sensitive element (or bulb), and the connecting wires and plug connectors. Electrical resistance thermometers are used widely in many types of aircraft to measure carburetor air, oil, free air temperatures, and more. They are used to measure low and medium temperatures in the -70°C to 150°C range.

For most metals, electrical resistance changes as the temperature of the metal changes. This is the principle upon which a resistance thermometer operates. Typically, the electrical resistance of a metal increases

as the temperature rises. Various alloys have a high temperature-resistance coefficient, meaning their resistance varies significantly with temperature. This can make them suitable for use in temperature sensing devices. The metal resistor is subjected to the fluid or area in which temperature needs to be measured. It is connected by wires to a resistance measuring device inside the cockpit indicator. The instrument dial is calibrated in degrees Fahrenheit or Celsius as desired rather than in ohms. As the temperature to be measured changes, the resistance of the metal changes and the resistance measuring indicator shows to what extent.

A typical electrical resistance thermometer looks like any other temperature gauge. Indicators are available in dual form for use in multi-engine aircraft. Most indicators are self-compensating for changes in cockpit temperature. The heat-sensitive resistor is manufactured so that it has a definite resistance for each temperature value within its working range. The temperature sensitive resistor element is a length or winding made of a nickel/manganese wire or other suitable alloy in an insulating material. The resistor is protected by a closed-end metal tube attached to a threaded plug with a hexagonal head. (*Figure 5-68*) The two ends of the winding are brazed, or welded, to an electrical receptacle designed to receive the prongs of the connector plug.

The indicator contains a resistance-measuring instrument. Sometimes it uses a modified form of the Wheatstone bridge circuit. The Wheatstone bridge meter operates on the principle of balancing one unknown resistor against other known resistances. A simplified form of a Wheatstone bridge circuit is shown in *Figure 5-69*. Three equal values of resistance (*Figure 5-69A, B, and C*) are connected into a diamond shaped bridge circuit. A resistor with an unknown value (*Figure 5-69D*) is also part of the circuit. The unknown



Figure 5-68. An electric resistance thermometer sensing bulb.

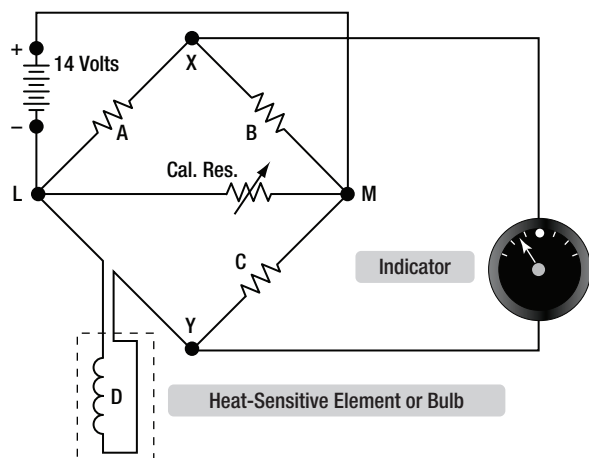


Figure 5-69. The internal structure of an electric resistance thermometer indicator features a bridge circuit, galvanometer.

A variable resistor or probe outside the indicator is in the form of the temperature sensor or probe.

resistance represents the resistance of the temperature bulb of the electrical resistance thermometer system. A galvanometer is attached across the circuit at points X and Y.

When the temperature causes the resistance of the bulb to equal that of the other resistances, no potential difference exists between points X and Y in the circuit. Therefore, no current flows in the galvanometer leg of the circuit. If the temperature of the bulb changes, its resistance also changes, and the bridge becomes unbalanced, causing current to flow through the galvanometer in one direction or the other. The galvanometer pointer is actually the temperature gauge pointer. As it moves against the dial face calibrated in degrees, it indicates temperature. Many indicators are provided with a zero adjustment screw on the face of the instrument. This adjusts the zeroing spring tension of the pointer when the bridge is at the balance point (the position at which the bridge circuit is balanced and no current flows through the meter).

Ratiometer Electrical Resistance Thermometers

Another way of indicating temperature when employing an electric resistance thermometer is by using a ratiometer. The Wheatstone bridge indicator is subject to errors from line voltage fluctuation. The ratiometer is more stable and can deliver higher accuracy. As its name suggests, the ratiometer electrical resistance thermometer measures a ratio of current flows. The resistance bulb sensing portion of the ratiometer

electric resistance thermometer is essentially the same as described above. The circuit contains a variable resistance and a fixed resistance to provide the indication. It contains two branches for current flow. Each has a coil mounted on either side of the pointer assembly that is mounted within the magnetic field of a large permanent magnet. Varying current flow through the coils causes different magnetic fields to form, which react with the larger magnetic field of the permanent magnet. This interaction rotates the pointer against the dial face that is calibrated in degrees Fahrenheit or Celsius, giving a temperature indication. (Figure 5-70)

The magnetic pole ends of the permanent magnet are closer at the top than they are at the bottom. This causes the magnetic field lines of flux between the poles to be more concentrated at the top. As the two coils produce their magnetic fields, the stronger field interacts and pivots downward into the weaker, less concentrated part of the permanent magnet field, while the weaker coil magnetic field shifts upward toward the more concentrated flux field of the large magnet. This provides a balancing effect that changes but stays in balance as the coil field strengths vary with temperature and the resultant current flowing through the coils.

For example, if the resistance of the temperature bulb is equal to the value of the fixed resistance (R), equal values of current flow through the coils. The torques, caused by the magnetic field each coil creates, are the same and cancel any movement in the larger magnetic field. The

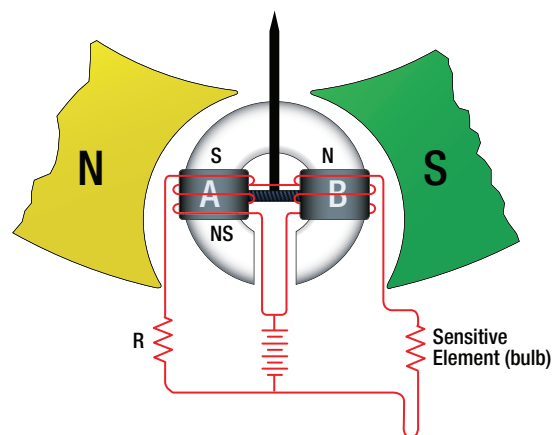


Figure 5-70. A ratiometer temperature measuring indicator has two coils. As the sensor bulb resistance varies with temperature, different amounts of current flow through the coils. This produces varying magnetic fields. These fields interact with the magnetic field of a large permanent magnet, resulting in an indication of temperature.

indicator pointer will be in the vertical position. If the bulb temperature increases, its resistance also increases. This causes the current flow through coil A circuit branch to increase. This creates a stronger magnetic field at coil A than at coil B. Consequently, the torque on coil A increases, and it is pulled downward into the weaker part of the large magnetic field. At the same time, less current flows through the sensor bulb resistor and coil B, causing coil B to form a weaker magnetic field that is pulled upward into the stronger flux area of the permanent magnet's magnetic field. The pointer stops rotating when the fields reach a new balance point that is directly related to the resistance in the sensing bulb. The opposite of this action would take place if the temperature of the heat-sensitive bulb should decrease.

Ratiometer temperature measuring systems are used to measure engine oil, outside air, carburetor air, and other temperatures in many types of aircraft. They are especially in demand to measure temperature conditions where accuracy is important, or large variations of supply voltages are encountered.

Thermocouple Temperature Indicators

A thermocouple is a circuit or connection of two unlike metals. The metals are touching at two separate junctions. If one of the junctions is heated to a higher temperature than the other, an electromotive force is produced in the circuit. This voltage is directly proportional to the temperature. So, by measuring the

amount of electromotive force, temperature can be determined. A voltmeter is placed across the colder of the two junctions of the thermocouple. It is calibrated in degrees Fahrenheit or Celsius, as needed. The hotter the high temperature junction (hot junction) becomes, the greater the electromotive force produced, and the higher the temperature indication on the meter. (Figure 5-71)

Thermocouples are used to measure high temperatures. Two common applications are the measurement of cylinder head temperature (CHT) in reciprocating engines and exhaust gas temperature (EGT) in turbine engines. Thermocouple leads are made from a variety of metals, depending on the maximum temperature to which they are exposed. Iron and constantan, or copper and constantan, are common for CHT measurement. Chromel and alumel are used for turbine EGT thermocouples.

The amount of voltage produced by the dissimilar metals when heated is measured in millivolts. Therefore, thermocouple leads are designed to provide a specific amount of resistance in the thermocouple circuit (usually very little). Their material, length, or cross-sectional size cannot be altered without compensation for the change in total resistance that would result. Each lead that makes a connection back to the voltmeter must be made of the same metal as the part of the thermocouple to which it is connected.

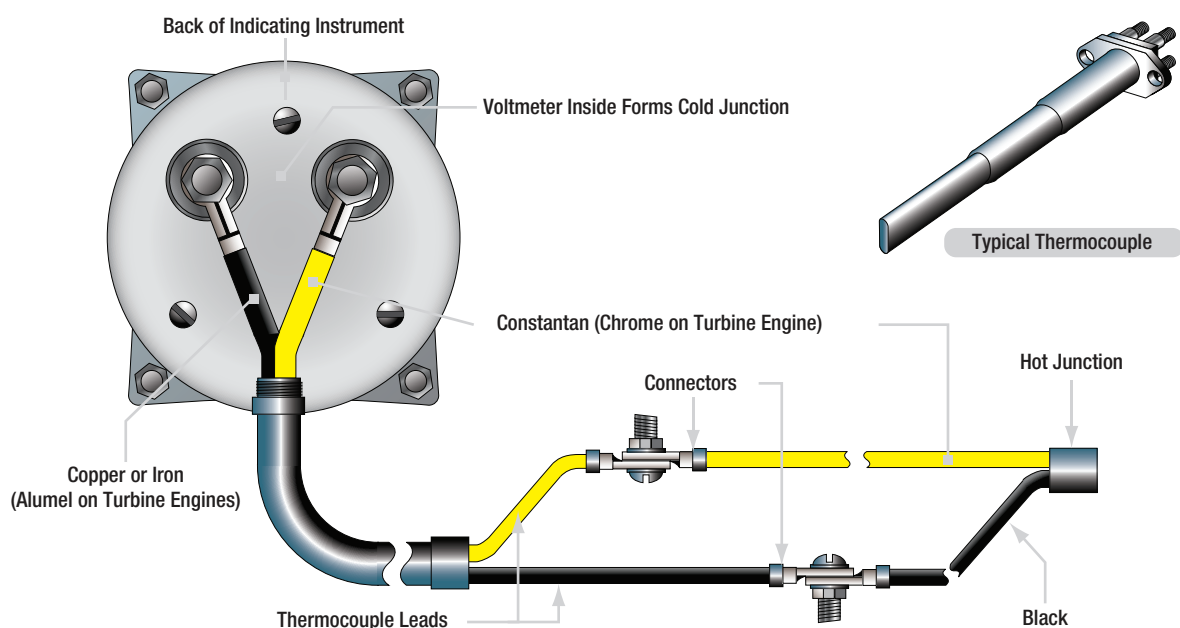


Figure 5-71. Thermocouples combine two unlike metals that cause current flow when heated.

For example, a copper wire is connected to the copper portion of the hot junction and a constantan wire is connected to the constantan part. The hot junction of a thermocouple varies in shape depending on its application. Two common types are the gasket and the bayonet.

In the gasket type, two rings of the dissimilar metals are pressed together to form a gasket that can be installed under a spark plug or cylinder hold down nut. In the bayonet type, the metals come together inside a perforated protective sheath. Bayonet thermocouples fit into a hole or well in a cylinder head. On turbine engines, they are found mounted on the turbine inlet or outlet case and extend through the case into the gas stream. Note that for CHT indication, the cylinder chosen for the thermocouple installation is the one that runs the hottest under most operating conditions. The location of this cylinder varies with different engines. (Figure 5-72)

The cold junction of the thermocouple circuit is inside the instrument case. Since the electromotive force set up in the circuit varies with the difference in temperature between the hot and cold junctions, it is necessary to compensate the indicator mechanism for changes in cockpit temperature which affect the cold junction. This is accomplished by using a bimetallic spring connected to

the indicator mechanism. This actually works the same as the bimetallic thermometer described previously. When the leads are disconnected from the indicator, the temperature of the cockpit area around the instrument panel can be read on the indicator dial. (Figure 5-73) Numeric LED indicators for CHT are also common in modern aircraft.

Turbine Gas Temperature Indicating Systems

EGT is a critical variable of turbine engine operation. The EGT indicating system provides a visual temperature indication in the cockpit of the turbine exhaust gases as they leave the turbine unit. In certain turbine engines, the temperature of the exhaust gases is measured at the entrance to the turbine unit. This is referred to as a turbine inlet temperature (TIT) indicating system.

Several thermocouples are used to measure EGT or TIT. They are spaced at intervals around the perimeter of the engine turbine casing or exhaust duct. The tiny thermocouple voltages are typically amplified and used to energize a servomotor that drives the indicator pointer. Gearing a digital drum indication off of the pointer motion is common. (Figure 5-74) The EGT indicator shown is a hermetically sealed unit. The

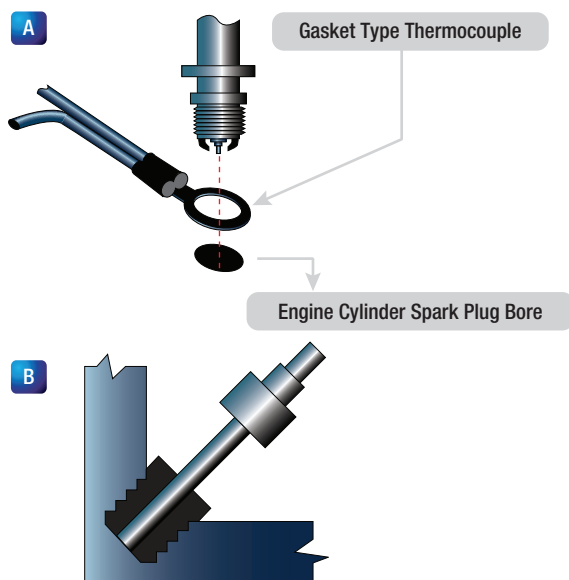


Figure 5-72. A cylinder head temperature thermocouple with a gasket type hot junction is made to be installed under the spark plug or a cylinder hold down nut of the hottest cylinder (A). A bayonet type thermocouple is installed in a bore in the cylinder wall (B).



Figure 5-73. Typical thermocouple temperature indicators.

instrument's scale ranges from 0°C to 1 200°C, with a vernier dial in the upper right-hand corner and a power off warning flag located in the lower portion of the dial.

A TIT indicating system provides a visual indication at the instrument panel of the temperature of gases entering the turbine. Numerous thermocouples can be used with the average voltage representing the TIT. Dual thermocouples exist containing two electrically independent junctions within a single probe. One set of these thermocouples is paralleled to transmit signals to the cockpit indicator. The other set of parallel thermocouples provides temperature signals to engine monitoring and control systems. Each circuit is electrically independent, providing dual system reliability.

A schematic for the turbine inlet temperature system for one engine of a four-engine turbine aircraft is shown in **Figure 5-75**. Circuits for the other three engines are identical to this system. The indicator contains a bridge circuit, a chopper circuit, a two phase motor to drive the pointer, and a feedback potentiometer. Also included are a voltage reference circuit, an amplifier, a power-off flag, a power supply, and an over temperature warning light. Output of the amplifier energizes the variable field of the two phase motor that positions the indicator main pointer and a digital indicator.

The motor also drives the feedback potentiometer to provide a humming signal to stop the drive motor when the correct pointer position, relative to the temperature signal, has been reached. The voltage reference circuit provides a closely regulated reference voltage in the bridge circuit to preclude error from input voltage variation to the indicator power supply.

The over temperature warning light in the indicator illuminates when the TIT reaches a predetermined limit. An external test switch is usually installed so that over temperature warning lights for all the engines can be tested at the same time. When the test switch is operated, an over temperature signal is simulated in each indicator temperature control bridge circuit.

Digital cockpit instrumentation systems need not employ resistance type indicators and adjusted servo driven thermocouple gauges to provide the pilot with temperature information. Sensor resistance and voltage values are input to the appropriate computer, where they are adjusted, processed, monitored, and output for display on cockpit display panels. They are also sent for use by other computers requiring temperature information for the control and monitoring of various integrated systems.

Total Air Temperature Measurement

Air temperature is a valuable parameter that many performance monitoring and control variables depend on. During flight, static air temperature changes continuously and accurate measurement presents challenges. Below 0.2 Mach, a simple resistance type or bimetallic temperature gauge can provide relatively accurate air temperature information. At faster speeds, friction, the air's compressibility, and boundary layer behavior make accurate temperature capture more complex. Total air temperature (TAT) is the static air temperature plus any rise in temperature caused by the high speed movement of the aircraft through the air. The increase in temperature is known as ram rise. TAT sensing probes are constructed specifically to accurately capture this value and transmit signals for cockpit indication, as well as for use in various engine and aircraft systems.

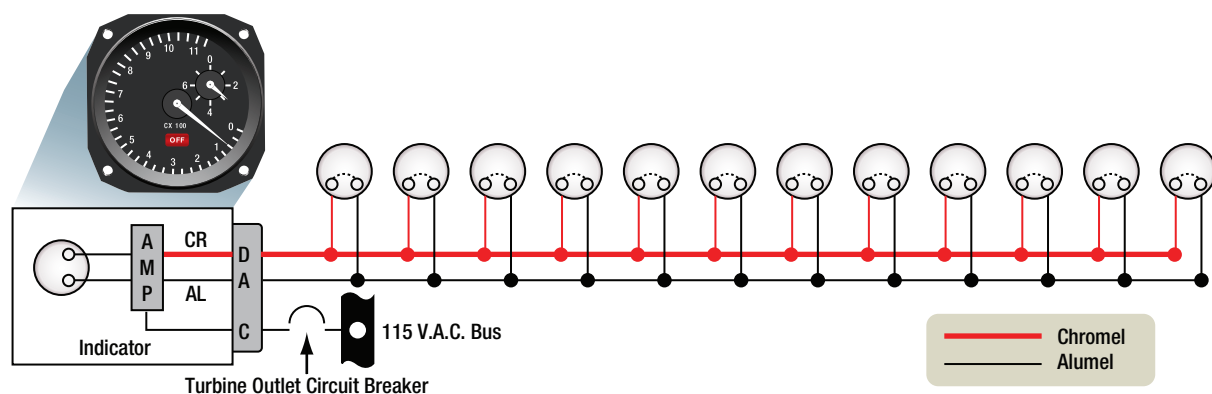


Figure 5-74. A typical exhaust gas temperature thermocouple system.

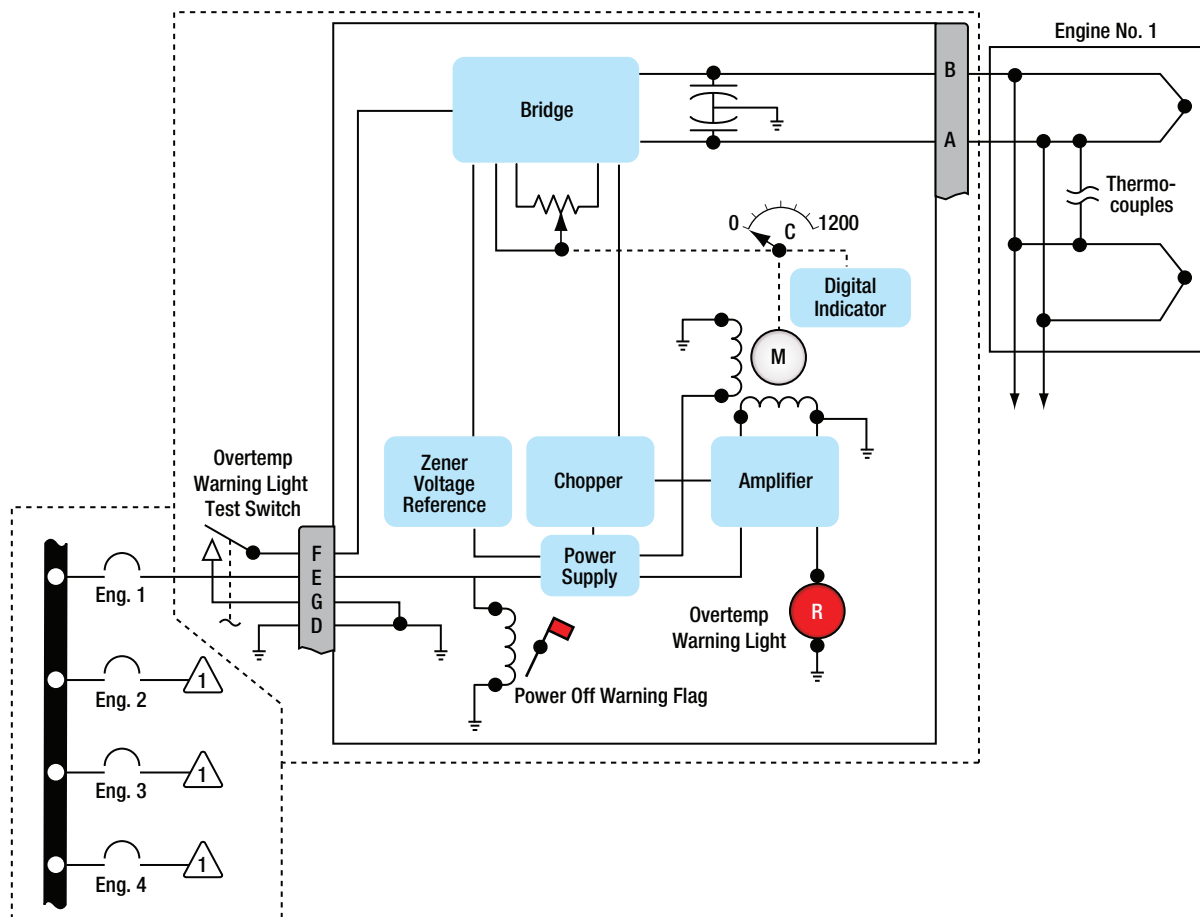


Figure 5-75. A typical analog turbine inlet temperature indicating system.

Simple TAT systems include a sensor and an indicator with a built-in resistance balance circuit. Air flow through the sensor is designed so that air with the precise temperature impacts a platinum alloy resistance element. The sensor is engineered to capture temperature variations in terms of varying the resistance of the element. When placed in the bridge circuit, the indicator pointer moves in response to the imbalance caused by the variable resistor.

More complex systems use signal correction technology and amplified signals sent to a servo motor to adjust the indicator in the cockpit. These systems include closely regulated power supply and failure monitoring. They often use numeric drum type readouts, but can also be sent to an LCD driver to illuminate LCD displays. Many LCD displays are multifunctional, capable of displaying static air temperature and true airspeed. In fully digital systems, the correction signals are input into the ADC. There, they can be manipulated appropriately for cockpit display or for whichever system requires temperature information. (Figure 5-76)

TAT sensor/probe design is complicated by the potential of ice forming during icing conditions. Left unheated, a probe may cease to function properly. The inclusion of a heating element threatens accurate data collection. Heating the probe must not affect the resistance of the sensor element. (Figure 5-77) Close attention is paid to airflow and materials conductivity during the design phase. Some TAT sensors channel bleed air through the units to affect the flow of outside air, so that it flows directly onto the platinum sensor without gaining added energy from the probe heater.

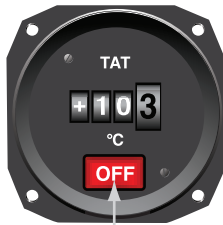
DIRECTION INDICATING INSTRUMENTS

A myriad of techniques and instruments exist to aid the pilot in navigation of the aircraft. An indication of direction is part of this navigation. While the next chapter deals with communication and navigation, this section discusses some of the magnetic direction indicating instruments. Additionally, a common, reliable gyroscopic direction indicator is discussed in the gyroscopic instrument section of this chapter.

Balanced Bridge Indicator



Servo Driven Indicator



LCD Indicator



Full Digital Display

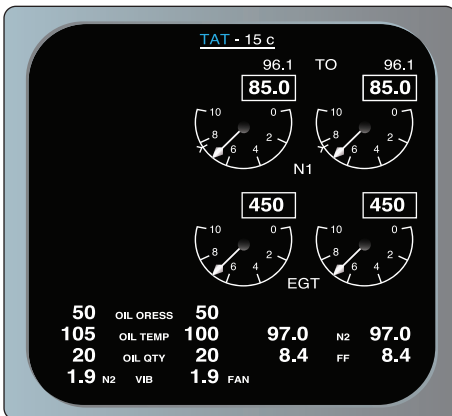


Figure 5-76. Different cockpit TAT displays.

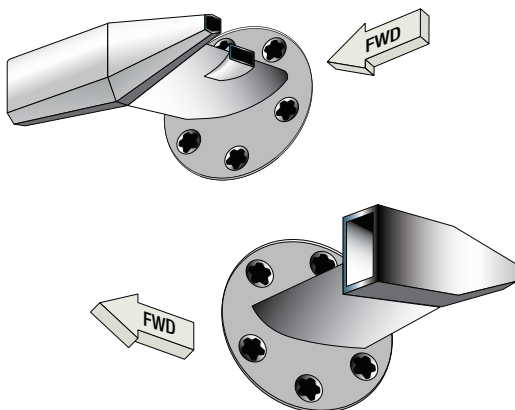


Figure 5-77. Total air temperature (TAT) probes.

DIRECT INDICATING MAGNETIC COMPASS

Having an instrument on board an aircraft that indicates direction can be invaluable to the pilot. In fact, it is a requirement that all certified aircraft have some sort of magnetic direction indicator. The magnetic compass is a direction finding instrument that has been used for navigation for hundreds of years. It is a simple instrument that takes advantage of the earth's magnetic field.

Figure 5-78 shows the earth and the magnetic field that surrounds it. The magnetic north pole is very close to the geographic North Pole of the globe, but they are not the same. An ordinary permanent magnet that is free to do so, aligns itself with the direction of the earth's magnetic field. Upon this principle, an instrument is constructed that the pilot can reference for directional orientation. Permanent magnets are attached under a float that is mounted on a pivot so it is free to rotate in the horizontal plane. As such, the magnets align with the earth's magnetic field.

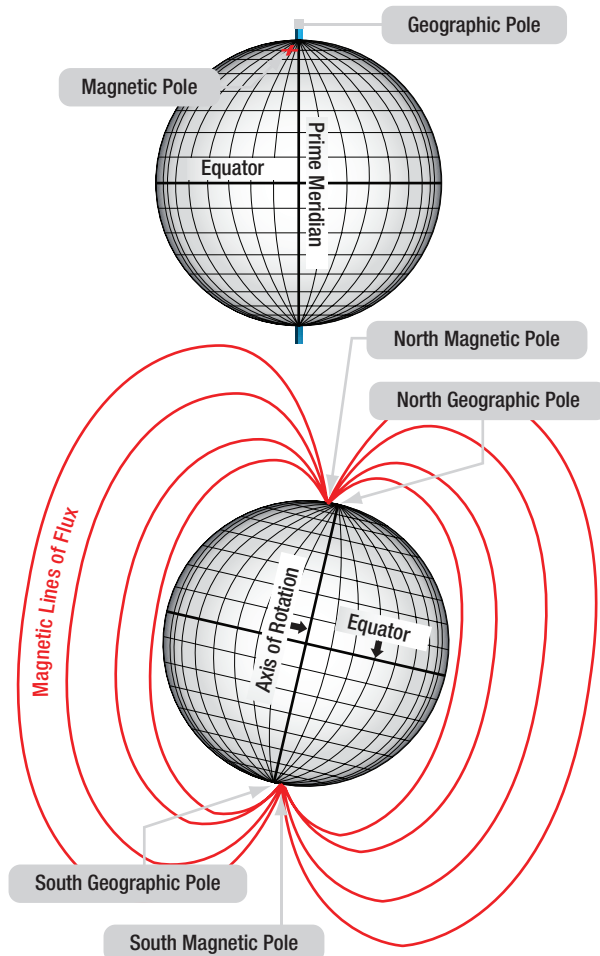


Figure 5-78. The earth and its magnetic field.

A numerical compass card, usually graduated in 5° increments, is constructed around the perimeter of the float. It serves as the instrument dial. The entire assembly is enclosed in a sealed case that is filled with a liquid similar to kerosene. This dampens vibration and oscillation of the moving float assembly and decreases friction.

On the front of the case, a glass face allows the numerical compass card to be referenced against a vertical lubber line. The magnetic heading of the aircraft is read by noting the graduation on which the lubber line falls. Thus, direction in any of 360° can be read off the dial as the magnetic float compass card assembly holds its alignment with magnetic north, while the aircraft changes direction. The liquid that fills the compass case expands and contracts as altitude changes and temperature fluctuates. A bellows diaphragm expands and contracts to adjust the volume of the space inside the case so it remains full. (*Figure 5-79*)

There are accuracy issues associated with using a magnetic compass. The main magnets of a compass align not only with the earth's magnetic field, they actually align with the composite field made up of all magnetic influences around them, meaning local electromagnetic influence from metallic structures near the compass and operation aircraft's electrical system. This is called magnetic deviation. It causes a magnet's alignment with the earth's magnetic field to be altered. Compensating screws are turned, which move small permanent magnets in the compass case to correct for this magnetic

deviation. The two set-screws are on the face of the instrument and are labeled N-S and E-W. They position the small magnets to counterbalance the local magnetic influences acting on the main compass magnets.

The process for knowing how to adjust for deviation is known as swinging the compass. It is described in the instrument maintenance pages near the end of this chapter. Magnetic deviation cannot be overlooked. It should never be more than 10 degrees. Using nonferrous mounting screws and shielding or twisting the wire running to the compass illuminating lamp are additional steps taken to keep deviation to a minimum.

Another compass error is called magnetic variation. It is caused by the difference in location between the earth's magnetic poles and the geographic poles. There are only a few places on the planet where a compass pointing to magnetic north is also pointing to geographic North. A line drawn through these locations is called the Agonic line. At all other points, there is some variation between that which a magnetic compass indicates is north and geographic (true) North. Isogonic lines drawn on aeronautical charts indicate points of equal variation. Depending on the location of the aircraft, airmen must add or subtract degrees from the magnetic indication to obtain true geographic location information.

(*Figure 5-80*)

The earth's magnetic field exits the poles vertically and arches around to extend past the equator horizontally or parallel to the earth's surface. (*Figure 5-78*)

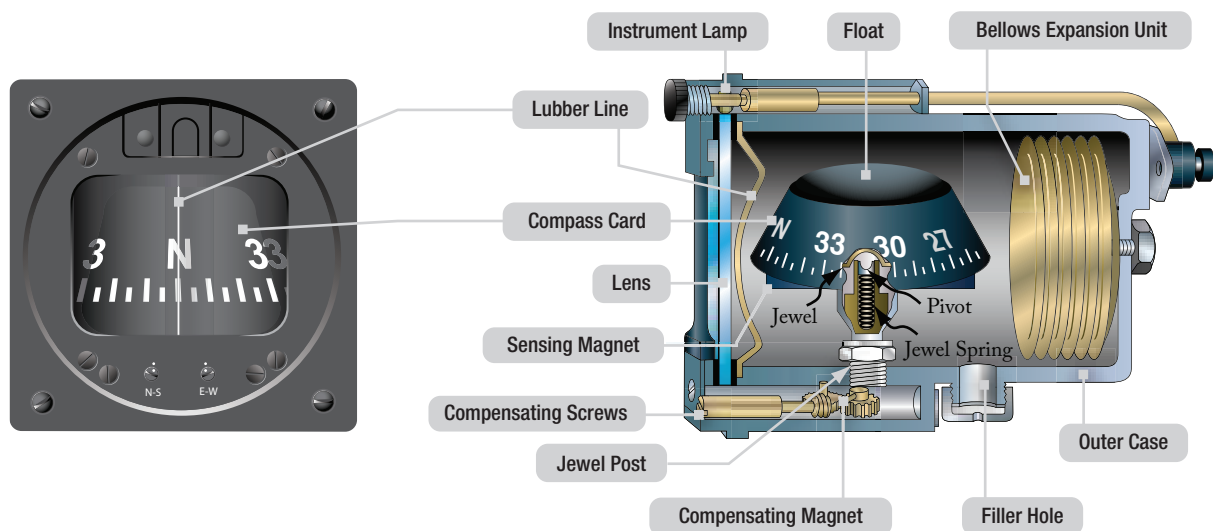


Figure 5-79. The parts of a typical magnetic compass.

Operating an aircraft near the magnetic poles causes what is known as dip error. The compass magnets pull downward toward the pole, rather than horizontally, as is the case near the equator. This downward motion causes inaccuracy in the indication. Although the compass float mechanism is weighted to compensate, the closer the aircraft is to the north or south magnetic poles, the more pronounced the errors.

Dip errors manifest themselves in two ways. The first is called acceleration error. If an aircraft is flying on an east-west path and simply accelerates, the inertia of the float mechanism causes the compass to swing to the north. Rapid deceleration causes it to swing southward. Second, if flying toward the North Pole and a banked turn is made, the downward pull of the magnetic field initially pulls the card away from the direction of the turn. The opposite is true if flying south from the North Pole and a banked turn is initiated. In this case, there is initially a pull of the compass indicator toward the direction of the turn. These kinds of movements are called turning errors.

Another peculiarity exists with the magnetic compass that is not dip error. Look again at the magnetic compass in *Figure 5-79*. If flying north or toward any indicated heading, turning the aircraft to the left causes a steady decrease in the heading numbers. But, before the turn is made, the numbers to the left on the compass card are actually increasing. The numbers to the right of the lubber line rotate behind it on a left turn. So, the compass card rotates opposite to the direction of the intended turn. This is because, from the pilot's seat, you are actually looking at the back of the compass card. While not a major problem, it is more intuitive to see the 360° of direction oriented as they are on an aeronautical chart or a hand-held compass.

Vertical Magnetic Compass

Solutions to the shortcomings of the simple magnetic compass described above have been engineered. The vertical magnetic compass is a variation of the magnetic compass that eliminates the reverse rotation of the compass card just described. By mounting the main indicating magnets of the compass on a shaft rather than a float, through a series of gears, a compass card can be made to turn about a horizontal axis.

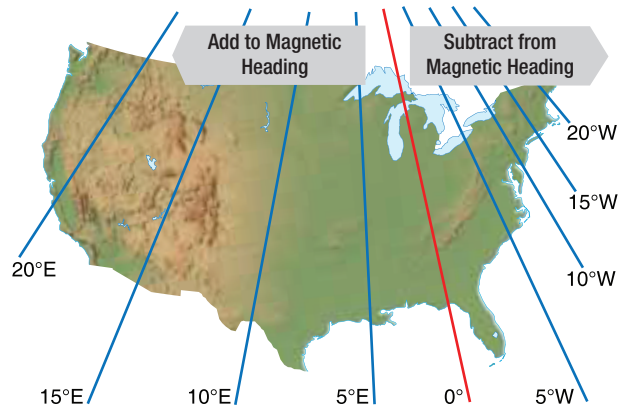


Figure 5-80. Aircraft located along the agonic line have 0° of variation between magnetic north and true north. Locations on and between the isogonic lines require addition or subtraction, as shown, to magnetic indications to arrive at a true geographic direction.

This allows the numbers for a heading, towards which the pilot wants to turn, to be oriented correctly on the indicating card. In other words, when turning right, increasing numbers are to the right; when turning left, decreasing numbers rotate in from the left. (*Figure 5-81*)

Many vertical magnetic compasses have also replaced the liquid-filled instrument housing with a dampening cup that uses eddy currents to dampen oscillations. Note that a vertical magnetic compass and a directional gyro look very similar and are often in the lower center position of the instrument panel basic T. Both use the nose of an aircraft as the lubber line against which a rotating compass card is read. Vertical magnetic compasses are characterized by the absence of the hand adjustment knob found on DGs, which is used to align the gyro with a magnetic indication.



Figure 5-81. A vertical magnetic direction indicator provides a realistic reference of headings.

REMOTE INDICATING COMPASS

Magnetic deviation is compensated for by swinging the compass and adjusting compensating magnets in the instrument housing. A better solution to deviation is to remotely locate the magnetic compass in a wing tip or vertical stabilizer where there is very little interference with the earth's magnetic field. By using a synchro remote indicating system, the magnetic compass float assembly can act as the rotor of the synchro system. As the float mechanism rotates to align with magnetic north in the remotely located compass, a varied electric current can be produced in the transmitter. This alters the magnetic field produced by the coils of the indicator in the cockpit, and a magnetic indication relatively free from deviation is displayed. Many of these systems are of the magnesyn type.

Remote Indicating Slaved Gyro Compass (Flux Gate Compass)

An elaborate and very accurate method of direction indication has been developed that combines the use of a gyro, a magnetic compass, and a remote indicating system. It is called the slaved gyro compass or flux gate compass system. A study of the gyroscopic instruments section of this chapter assists in understanding this device.

A gyroscopic direction indicator is augmented by magnetic direction information from a remotely located compass. The type of compass used is called a flux valve or flux gate compass. It consists of a very magnetically permeable circular segmented core frame or spider. The earth's magnetic field flows through this iron core and varies its distribution through segments of the core as the flux valve is rotated via the movement of the aircraft.

Pickup coil windings are located on each of the core's spider legs that are positioned 120° apart. (*Figure 5-82*)

The distribution of earth's magnetic field flowing through the legs is unique for every directional orientation of the aircraft. A coil is placed in the center of the core and is energized by AC current. As the AC flow passes through zero while changing direction, the earth's magnetic field is allowed to flow through the core. Then, it is blocked or gated as the magnetic field of the core current flow builds to its peak again. The cycle is repeated at the frequency of the AC supplied to the excitation coil. The result is repeated flow and non-flow of the earth's flux across the pickup coils.

During each cycle, a unique voltage is induced in each of the pickup coils reflecting the orientation of the aircraft in the earth's magnetic field.

The electricity that flows from each of the pickup coils is transmitted out of the flux valve via wires into a second unit. It contains an autosyn transmitter, directional gyro, an amplifier, and a triple wound stator that is similar to that found in the indicator of a synchro system. Unique voltage is induced in the center rotor of this stator which reflects the voltage received from the flux valve pickup coils sent through the stator coils. It is amplified and used to augment the position of the DG. The gyro is wired to be the rotor of an autosyn synchro system, which transmits the position of the gyro into an indicator unit located in the cockpit. In the indicator, a vertical compass card is rotated against a small aeroplane type lubber line like that in a vertical magnetic compass. (*Figure 5-83 and Figure 5-84*)

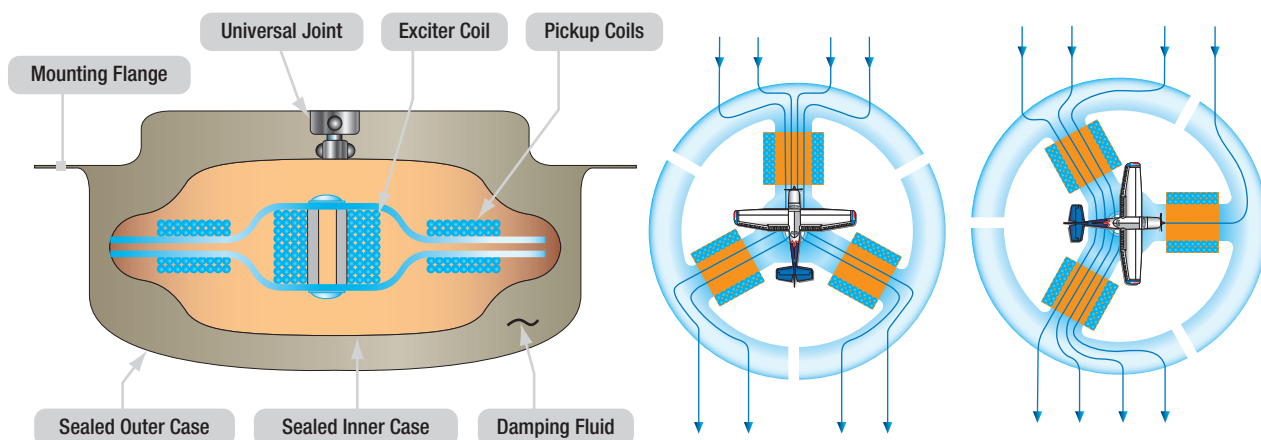


Figure 5-82. The simplified mechanism of a flyweight type mechanical tachometer.

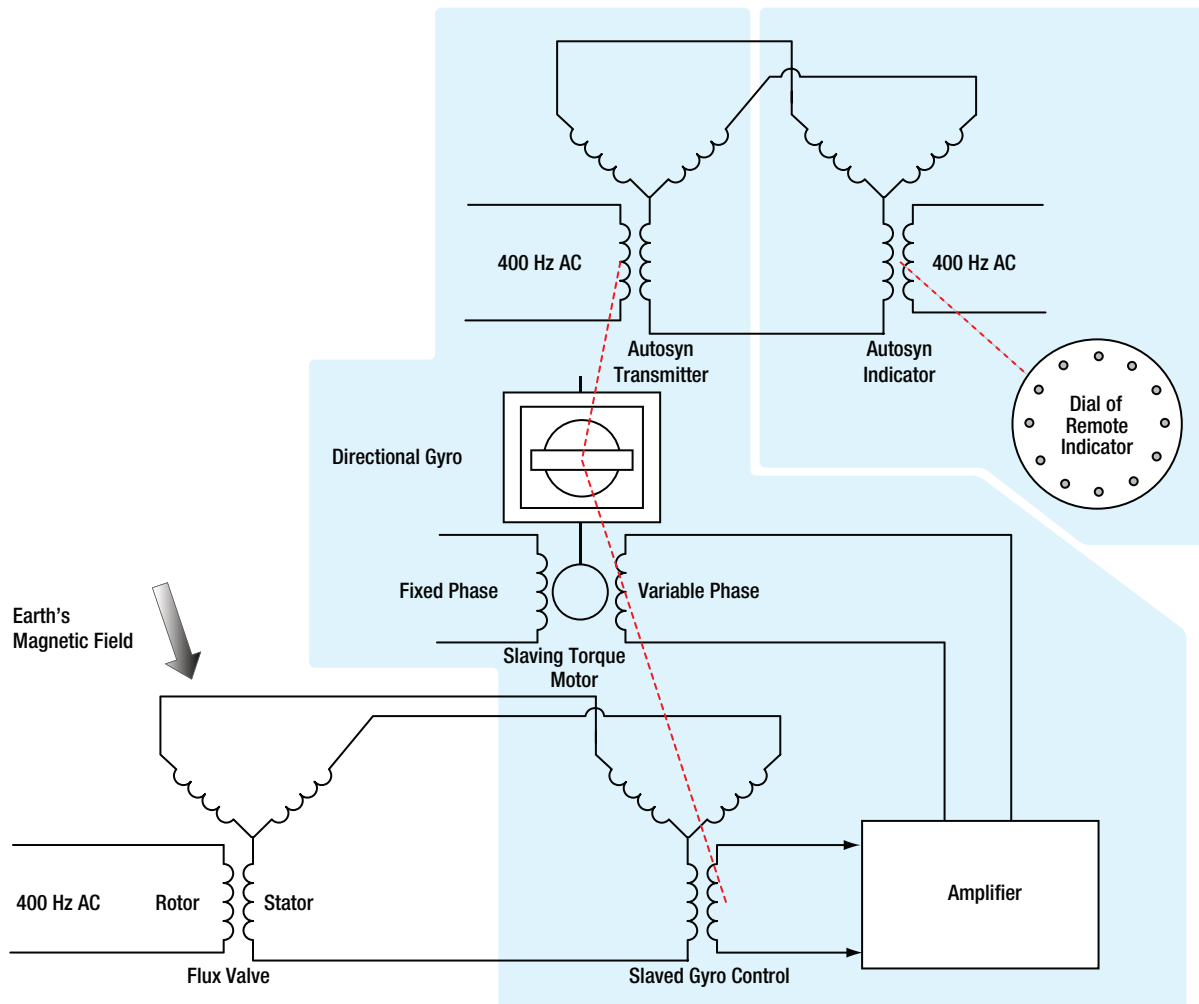


Figure 5-83. A simplified schematic of a flux gate, or slaved gyro, compass system.

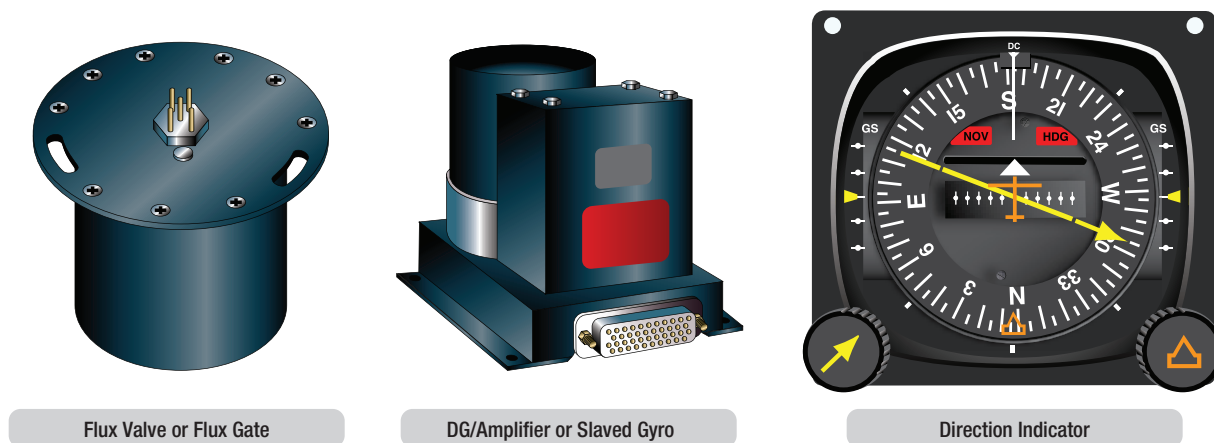


Figure 5-84. Solid state magnetometer units.

Further enhancements to direction finding systems of this type involving the integration of radio navigation aids are common. The radio magnetic indicator (RMI) is one such variation. (Figure 5-85) In addition to the rotating direction indicator of the slaved gyro compass,

it contains two pointers. One indicates the bearing to a very high frequency (VHF) omnidirectional range (VOR) station and the other indicates the bearing to a nondirectional automatic direction finder (ADF) beacon. These and other radio navigation aids are



Figure 5-85. A radio magnetic indicator (RMI) combines a slaved gyro heading indication (red triangle at top of gauge) with magnetic bearing information to a VOR station (solid pointer) and an ADF station (hollow pointer).

discussed further in the communications and navigation chapter of this handbook. It should also be noted that integration of slaved gyro direction indicating system information into auto pilot systems is also possible.

SOLID STATE MAGNETOMETERS

Solid state magnetometers are used on many modern aircraft. They have no moving parts and are extremely accurate. Tiny layered structures react to magnetism on a molecular level resulting in variations in electron activity. These low power consuming devices can sense not only the direction to the earth's magnetic poles, but also the angle of the flux field.

They are free from oscillation that plagues a standard magnetic compass. They feature integrated processing algorithms and easy integration with digital systems. (Figure 5-86)

GYROSCOPIC INSTRUMENTS

SOURCES OF POWER FOR GYROSCOPIC INSTRUMENTS

Gyroscopic instruments are essential instruments used on all aircraft. They provide the pilot with critical attitude and directional information and are particularly important while flying under IFR. The sources of power for these instruments can vary. The main requirement is to spin the gyroscopes at a high rate of speed. Originally, gyroscopic instruments were strictly vacuum driven.



Figure 5-86. Solid state magnetometer units.

A vacuum source pulled air across the gyro inside the instruments to make the gyros spin. Later, electricity was added as a source of power. The turning armature of an electric motor doubles as the gyro rotor. In some aircraft, pressure, rather than vacuum, is used to induce the gyro to spin. Various systems and powering configurations have been developed to provide reliable operation of the gyroscopic instruments.

PRESSURE DRIVEN GYROSCOPIC INSTRUMENT SYSTEMS

Gyroscopic instruments are finely balanced devices with jeweled bearings that must be kept clean to perform properly. When early vacuum systems were developed, only oil lubricated pumps were available. Even with the use of air-oil separators, the pressure outputs of these pumps contain traces of oil and dirt. As a result, it was preferred to draw clean air through the gyro instruments with a vacuum system, rather than using pump output pressure that presented the risk of contamination.

The development of self lubricated dry pumps greatly reduced pressure output contaminates. This made pressure gyro systems possible. At high altitudes, the use of pressure driven gyros is more efficient. Pressure systems are similar to vacuum systems and make use of the same components, but they are designed for pressure instead of vacuum. Thus, a pressure regulator is used instead of a suction relief valve. Filters are still extremely important to prevent damage to the gyros. Normally, air is filtered at the inlet and outlet of the pump in a pressure gyro system.

ELECTRICALLY-DRIVEN GYROSCOPIC INSTRUMENT SYSTEMS

A spinning motor armature can act as a gyroscope. This is the basis for electrically driven gyroscopic instruments in which the gyro rotor spin is powered by an electric

motor. Electric gyros have the advantage of being powered by battery for a limited time if a generator fails or an engine is lost. Since air is not sent through the gyro to spin the rotor, contamination worries are also reduced. Also, elimination of vacuum pumps, plumbing, and vacuum system components saves weight.

On many small, single-engine aircraft, electric turn-and-bank or turn coordinators are combined with vacuum-powered attitude and directional gyro instruments as a means for redundancy. The reverse is also possible. By combining both types of instruments in the instrument panel, the pilot has more options. On more complex multi-engine aircraft, reliable, redundant electrical systems make use of all electric powered gyro instruments possible. It should be noted that electric gyro instruments have some sort of indicator on the face of the dial to show when the instrument is not receiving power. Usually, this is in the form of a red flag or mark of some sort often with the word OFF written on it (or a similar word).

PRINCIPLES OF GYROSCOPIC INSTRUMENTS

Mechanical Gyros

Three of the most common flight instruments, the attitude indicator, heading indicator, and turn needle of the turn-and-bank indicator, are controlled by gyroscopes. To understand how these instruments operate, knowledge of gyroscopic principles and instrument power systems is required. A mechanical

gyroscope, or gyro, is comprised of a wheel or rotor with its mass concentrated around its perimeter. The rotor has bearings to enable it to spin at high speeds.

(*Figure 5-87A*)

Different mounting configurations are available for the rotor and axle, which allow the rotor assembly to rotate about one or two axes perpendicular to its axis of spin. To suspend the rotor for rotation, the axle is first mounted in a supporting ring. (*Figure 5-87B*)

If brackets are attached 90° around the supporting ring from where the spin axle attached, the supporting ring and rotor can both move freely 360°. When in this configuration, the gyro is said to be a captive gyro. It can rotate about only one axis that is perpendicular to the axis of spin. (*Figure 5-87C*)

The supporting ring can also be mounted inside an outer ring. The bearing points are the same as the bracket just described, 90° around the supporting ring from where the spin axle attached. Attachment of a bracket to this outer ring allows the rotor to rotate in two planes while spinning. Both of these are perpendicular to the spin axis of the rotor. The plane that the rotor spins in due to its rotation about its axle is not counted as a plane of rotation. A gyroscope with this configuration, two rings plus the mounting bracket, is said to be a free gyro because it is free to rotate about two axes that are both perpendicular to the rotor's spin axis. (*Figure 5-87D*)

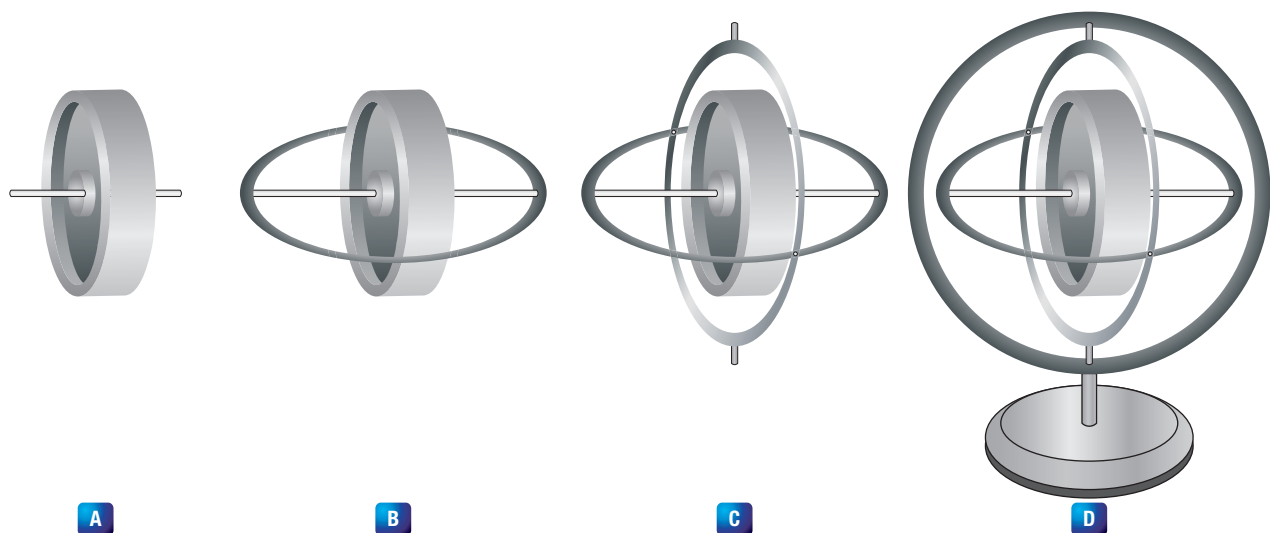


Figure 5-87. Gyroscopes.

As a result, the supporting ring with spinning gyro mounted inside is free to turn 360° inside the outer ring. Unless the rotor of a gyro is spinning, it has no unusual properties; it is simply a wheel universally mounted. When the rotor is rotated at a high speed, the gyro exhibits a couple of unique characteristics. The first is called gyroscopic rigidity, or rigidity in space. This means that the rotor of a free gyro always points in the same direction no matter which way the base of the gyro is positioned. (*Figure 5-88*)

Gyroscopic rigidity depends upon several design factors:

1. Weight for a given size - a heavy mass is more resistant to disturbing forces than a light mass.
2. Angular velocity - the higher the rotational speed, the greater the rigidity or resistance is to deflection.
3. Radius at which the weight is concentrated - maximum effect is obtained from a mass when its principal weight is concentrated near the rim, rotating at high speed.
4. Bearing friction - any friction applies a deflecting force to a gyro. Minimum bearing friction keeps deflecting forces at a minimum.

This characteristic of gyros to remain rigid in space is exploited in the attitude-indicating instruments and the directional indicators that use gyros.

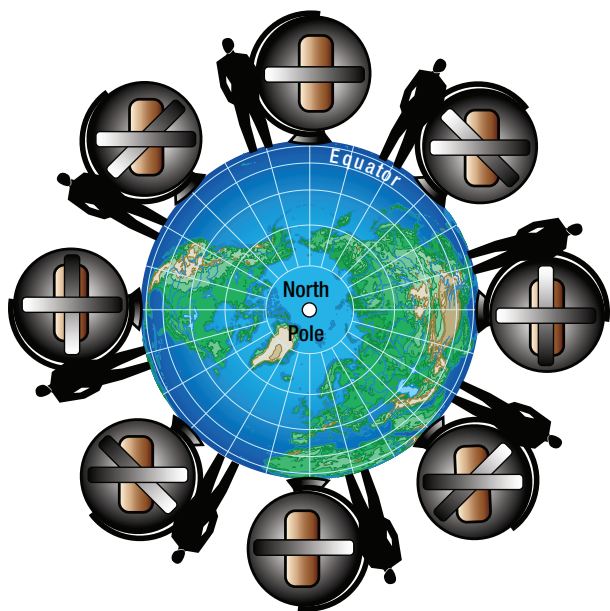


Figure 5-88. Once spinning, a free gyro rotor stays oriented in the same position in space despite the position or location of its base.

Precession is a second important characteristic of gyroscopes. By applying a force to the horizontal axis of the gyro, a unique phenomenon occurs. The applied force is resisted. Instead of responding to the force by moving about the horizontal axis, the gyro moves in response about its vertical axis. Stated another way, an applied force to the axis of the spinning gyro does not cause the axis to tilt. Rather, the gyro responds as though the force was applied 90° around in the direction of rotation of the gyro rotor. The gyro rotates rather than tilts. (*Figure 5-89*) This predictable controlled precession of a gyroscope is utilized in a turn and bank instrument.

SOLID STATE GYROS AND RELATED SYSTEMS

Improved attitude and direction information is always a goal in aviation. Modern aircraft make use of highly accurate solid state attitude and directional devices with no moving parts. This results in very high reliability and low maintenance.

RING LASER GYROS (RLG)

The ring laser gyro (RLG) is widely used in commercial aviation. The basis for RLG operation is that it takes time for light to travel around a stationary, non-rotating circular path. Light takes longer to complete the journey if the path is rotating in the same direction as the light is traveling. And, it takes less time for the light to complete the loop if the path is rotating in the direction opposite

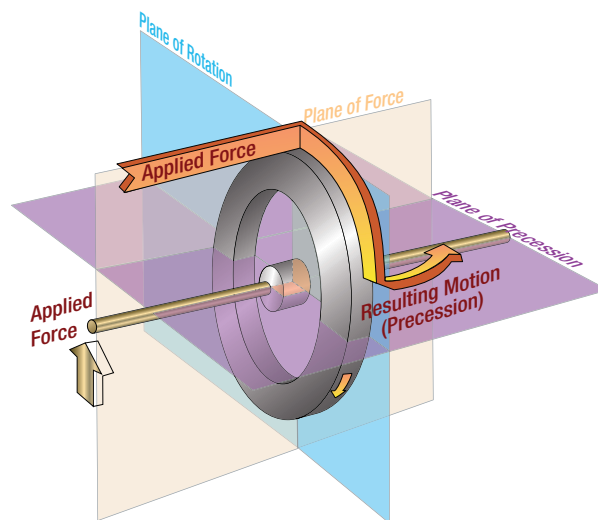


Figure 5-89. When a force is applied to a spinning gyroscope, it reacts as though the force came from 90° further around the rotor in the direction it is spinning. The plane of the applied force, the plane of the rotation, and the plane in which the gyro responds (known as the plane of precession), are all perpendicular to each other.

to that of the light. Essentially, the path is made longer or shorter by the rotation of the path. (*Figure 5-90*) This is known as the Sagnac effect.

A laser is light amplification by stimulated emission of radiation. A laser operates by exciting atoms in plasma to release electromagnetic energy, or photons.

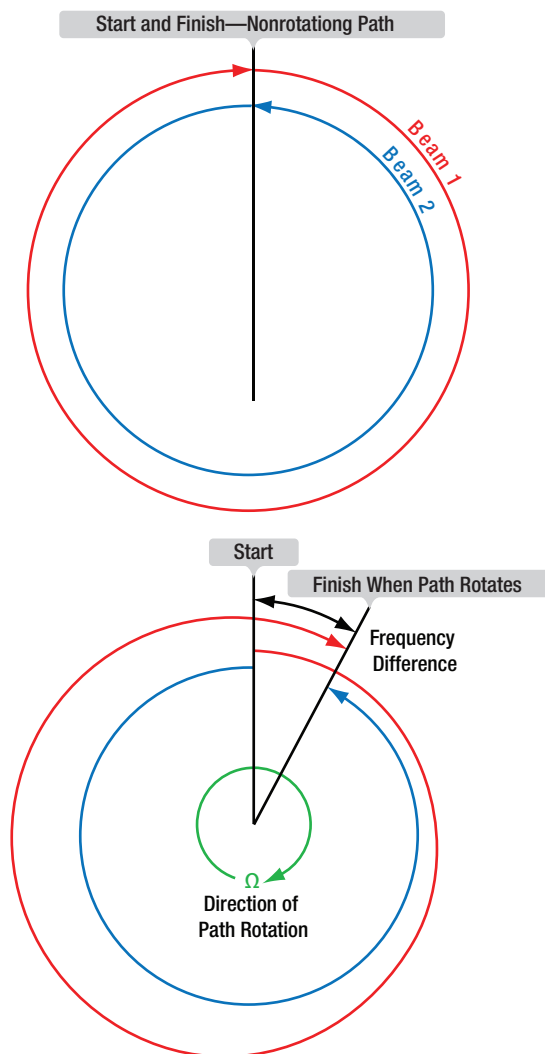
A ring laser gyro produces laser beams that travel in opposite directions around a closed triangular cavity. The wavelength of the light traveling around the loop is fixed. As the loop rotates, the path the lasers must travel lengthens or shortens. The light wavelengths compress

or expand to complete travel around the loop as the loop changes its effective length. As the wavelengths change, the frequencies also change.

By examining the difference in the frequencies of the two counter rotating beams of light, the rate at which the path is rotating can be measured. A piezoelectric dithering motor in the center of the unit vibrates to prevent lock-in of the output signal at low rotational speeds. It causes units installed on aircraft to hum when operating. (*Figure 5-91*)

An RLG is remotely mounted so the cavity path rotates around one of the axes of flight. The rate of frequency phase shift detected between the counter rotating lasers is proportional to the rate that the aircraft is moving about that axis. On aircraft, an RLG is installed for each axis of flight. Output can be used in analog instrumentation and autopilot systems. It is also easily made compatible for use by digital display computers and for digital autopilot computers.

RLGs are rugged and have a long service life with little maintenance due to their lack of moving parts. They measure movement about an axis extremely quickly and provide continuous output. They are extremely accurate and generally are considered superior to mechanical gyroscopes.



A Ring Laser Gyro Functions Due To The Sagnac Effect

Figure 5-90. Light traveling in opposite directions around a non-rotating path arrives at the end of the loop at the same time (top). When the path rotates, light traveling with the rotation must travel farther to complete one loop. Light traveling against the rotation completes the loop sooner (bottom).

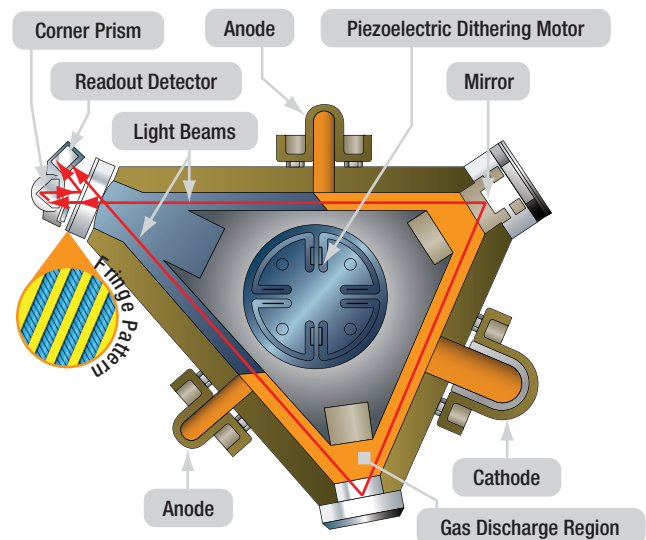


Figure 5-91. The ring laser gyro is rugged, accurate, and free of friction.

MICROELECTROMECHANICAL BASED ATTITUDE AND DIRECTIONAL SYSTEMS (MEMS)

On aircraft, microelectromechanical systems (MEMS) devices save space and weight. Through the use of solid state MEMS devices, reliability is increased primarily due to the lack of moving parts. The development of MEMS technology for use in aviation instrumentation integrates with the use of air data computers ADCs. This newest improvement in technology is low cost and promises to proliferate through all forms of aviation.

MEMS for gyroscopic applications are used in small, general aviation aircraft, as well as larger commercial aircraft. Tiny vibration-based units with resistance and capacitance measuring pick-offs are accurate and reliable and only a few millimeters in length and width. They are normally integrated into a complete micro-electronic solid-state chip designed to yield an output after various conditioning processes are performed. The chips, which are analogous to tiny circuit boards, can be packaged for installation inside a dedicated computer or module that is installed on the aircraft.

While a large mechanical gyroscope spins in a plane, its rigidity in space is used to observe and measure the movement of the aircraft. The basis of operation of many MEMS gyroscopes is the same despite their tiny size. The difference is that a vibrating or oscillating piezoelectric device replaces the spinning, weighted ring of the mechanical gyro. Still, once set

in motion, any out of plane motion is detectable by varying microvoltages or capacitances detected through geometrically arranged pickups.

Since piezoelectric substances have a relationship between movement and electricity, microelectrical stimulation sets a piezoelectric gyro in motion and the tiny voltages produced via the movement in the piezo are extracted. They are then input as the required variables needed to compute attitude or direction information. (Figure 5-92)

ATTITUDE HEADING AND REFERENCE SYSTEMS

In many modern aircraft, (AHRS) have taken the place of the gyroscope and other individual instruments. While MEMS devices provide part of the attitude information for the system, GPS, solid state magnetometers, solid state accelerometers, and digital air data signals are all combined in an AHRS to compute and output highly reliable information for display on a cockpit panel. (Figure 5-93)

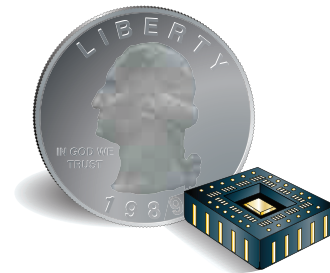


Figure 5-92. The relative scale size of a MEMS gyro.



Figure 5-93. Instrumentation displayed within a glass cockpit using an attitude heading and reference system (AHRS) computer.

COMMON GYROSCOPIC INSTRUMENTS

VACUUM-DRIVEN ATTITUDE GYROS

The attitude indicator, or artificial horizon, is one of the most essential flight instruments. It gives the pilot pitch and roll information that is especially important when flying without outside visual references. The attitude indicator operates with a gyroscope rotating in the horizontal plane. Thus, it mimics the actual horizon through its rigidity in space. As the aircraft pitches and rolls in relation to the actual horizon, the gyro gimbals allow the aircraft and instrument housing to pitch and roll around the gyro rotor that remains parallel to the ground. A horizontal representation of the aeroplane in miniature is fixed to the instrument housing. A painted semisphere simulating the horizon, the sky, and the ground is attached to the gyro gimbals. The sky and ground meet at what is called the horizon bar.

The relationship between the horizon bar and the miniature aeroplane are the same as those of the aircraft and the actual horizon. Graduated scales reference the degrees of pitch and roll. Often, an adjustment knob allows pilots of varying heights to place the horizon bar at an appropriate level. (*Figure 5-94*)

In a typical vacuum-driven attitude gyro system, air is sucked through a filter and then through the attitude indicator in a manner that spins the gyro rotor inside. An erecting mechanism is built into the instrument to assist in keeping the gyro rotor rotating in the intended

plane. Precession caused by bearing friction makes this necessary. After air engages the scalloped drive on the rotor, it flows from the instrument to the vacuum pump through four ports. These ports all exhaust the same amount of air when the gyro is rotating in plane.

When the gyro rotates out of plane, air tends to port out of one side more than another. Vanes close to prevent this, causing more air to flow out of the opposite side. The force from this unequal venting of the air re-erects the gyro rotor. (*Figure 5-95*)

Early vacuum driven attitude indicators were limited in how far the aircraft could pitch or roll before the gyro gimbals contacted stops, causing abrupt precession and tumbling of the gyro. Many of these gyros include a caging device. It is used to erect the rotor to its normal operating position prior to flight or after tumbling. A flag indicates that the gyro must be uncaged before use. More modern gyroscopic instruments are built so they do not tumble, regardless of the angular movement of the aircraft about its axes.

In addition to the contamination potential introduced by the air-drive system, other shortcomings exist in the performance of vacuum-driven attitude indicators. Some are induced by the erection mechanism. The pendulous vanes that move to direct airflow out of the gyro respond not only to forces caused by a deviation from the intended plane of rotation, but centrifugal force experienced during turns also causes the vanes to allow asymmetric porting of the gyro vacuum air.



Figure 5-94. A typical vacuum-driven attitude indicator shown with the aircraft in level flight (left) and in a climbing right turn (right).

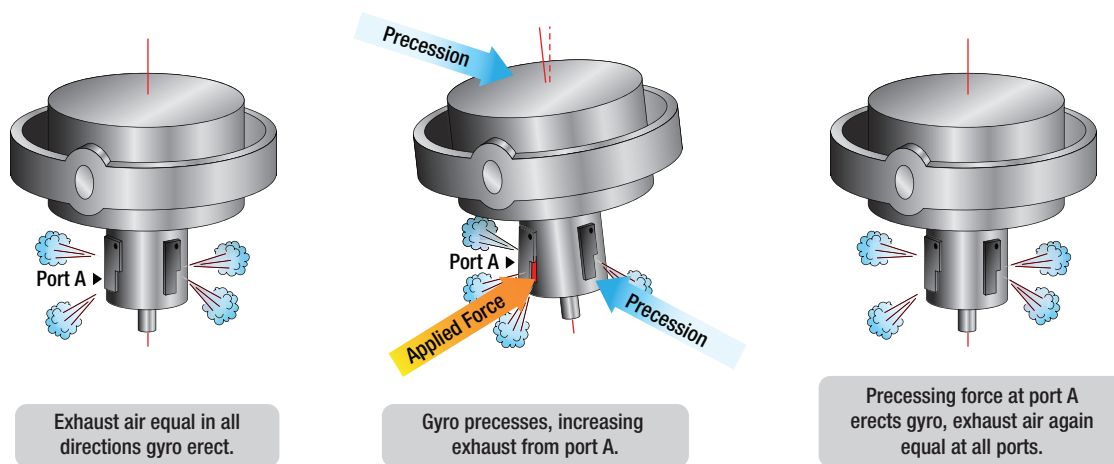


Figure 5-95. The erecting mechanism of a vacuum-driven attitude indicator.

The result is inaccurate display of the aircraft's attitude, especially in skids and steep banked turns. Also, abrupt acceleration and deceleration imposes forces on the gyro rotor. Suspended in its gimbals, it acts similar to an accelerometer, resulting in a false nose-up or nose-down indication. Pilots must learn to recognize these errors and adjust accordingly.

ELECTRIC ATTITUDE INDICATORS

Electric attitude indicators are very similar to vacuum driven gyro indicators. The main difference is in the drive mechanism. Inside the gimbals of an electric gyro, a small squirrel cage electric motor is the rotor. It is typically driven by 115-volt, 400-cycle AC. It turns at approximately 21 000 rpm.

Other characteristics of the vacuum-driven gyro are shared by the electric gyro. The rotor is still oriented in the horizontal plane. The free gyro gimbals allow the aircraft and instrument case to rotate around the gyro rotor that remains rigid in space. A miniature aeroplane fixed to the instrument case indicates the aircraft's attitude against the moving horizon bar behind it.

Electric attitude indicators address some of the shortcomings of vacuum-driven attitude indicators. Since there is no air flowing through an electric attitude indicator, air filters, regulators, plumbing lines and vacuum pump(s) are not needed. Contamination from dirt in the air is not an issue, resulting in the potential for longer bearing life and less precession. Erection mechanism ports are not employed, so pendulous vanes responsive to centrifugal forces are eliminated.

It is still possible that the gyro may experience precession and need to be erected. This is done with magnets rather than vent ports. A magnet attached to the top of the gyro shaft spins at approximately 21 000 rpm. Around this magnet, but not attached to it, is a sleeve that is rotated by magnetic attraction at approximately 44 to 48 rpm. Steel balls are free to move around the sleeve.

If the pull of gravity is not aligned with the axis of the gyro, the balls fall to the low side. The resulting precession re-aligns the axis of rotation vertically. Typically, electric attitude indicator gyros can be caged manually by a lever and cam mechanism to provide rapid erection. When the instrument is not getting sufficient power for normal operation, an off flag appears in the upper right hand face of the instrument. (*Figure 5-96*)

GYROSCOPIC DIRECTION INDICATOR OR DIRECTIONAL GYRO (DG)

The gyroscopic direction indicator or directional gyro (DG) is often the primary instrument for direction. Because a magnetic compass fluctuates so much, a gyro aligned with the magnetic compass gives a much more stable heading indication. Gyroscopic direction indicators are located at the center base of the instrument panel basic T. A vacuum-powered DG is common on many light aircraft. Its basis for operation is the gyro's rigidity in space. The gyro rotor spins in the vertical plane and stays aligned with the direction to which it is set. The aircraft and instrument case moves around the rigid gyro. This causes a vertical compass card that is geared to the rotor gimbal to move. It is calibrated in degrees, usually with every 30 degrees labeled. The nose of a small, fixed aeroplane on the instrument glass indicates the aircraft's heading. (*Figure 5-97*)

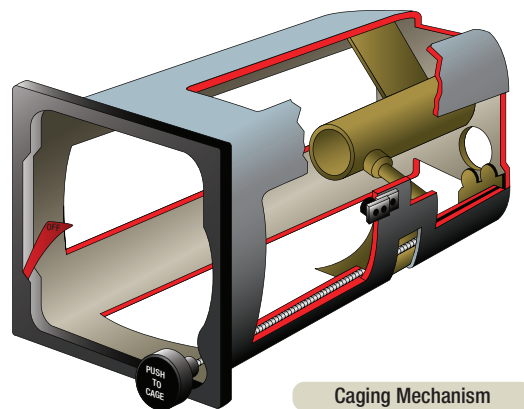
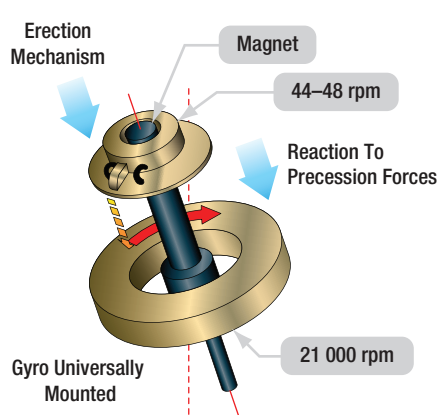


Figure 5-96. Erecting and caging mechanisms of an electric attitude indicator.

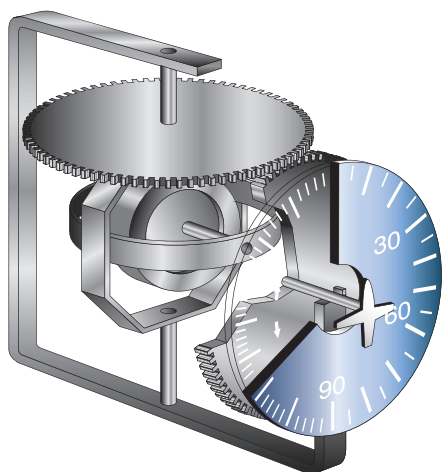


Figure 5-97. A typical vacuum-powered gyroscopic direction indicator, also known as a directional gyro.

Vacuum-driven direction indicators have many of the same basic gyroscopic instrument issues as attitude indicators. Built-in compensation for precession varies and a caging device is usually found. Periodic manual realignment with the magnetic compass by the pilot is required during flight.

TURN COORDINATORS

Many aircraft make use of a turn coordinator. The rotor of the gyro in a turn coordinator is canted upwards 30° . As such, it responds not only to movement about the vertical axis, but also to roll movements about the longitudinal axis. This is useful because it is necessary to roll an aircraft to turn it about the vertical axis. Instrument indication of roll, therefore, is the earliest possible warning of a departure from straight-and-level flight. Typically, the face of the turn coordinator has a small aeroplane symbol. The wing tips of the aeroplane provide the indication of level flight and the rate at which the aircraft is turning (*Figure 5-98*)

TURN-AND-SLIP INDICATOR

The turn-and-slip indicator may also be referred to as the turn-and-bank indicator, or needle-and-ball indicator. Regardless, it shows the correct execution of a turn while banking the aircraft and indicates movement about the vertical axis of the aircraft (yaw). Most turn-and-slip indicators are located below the airspeed indicator of the instrument panel basic T, just to the left of the direction indicator.

The turn-and-slip indicator is actually two separate devices built into the same instrument housing: a turn indicator pointer and slip indicator ball. The turn pointer is operated by a gyro that can be driven by a vacuum, air pressure, or by electricity. The ball is a completely independent device. It is a round agate, or steel ball, in a glass tube filled with dampening fluid. It moves in response to gravity and centrifugal force experienced in a turn. Turn indicators vary. They all indicate the rate at which the aircraft is turning.



Figure 5-98. A turn coordinator senses and indicates the rate of both roll and yaw.

Three degrees of turn per second cause an aircraft to turn 360° in 2 minutes. This is considered a standard turn. This rate can be indicated with marks right and left of the pointer, which normally rests in the vertical position. Sometimes, no marks are present and the width of the pointer is used as the calibration device. In this case, one pointer width deflection from vertical is equal to the 3° per second standard 2 minute turn rate. Faster aircraft tend to turn more slowly and have graduations or labels that indicate 4 minute turns. In other words, a pointer's width or alignment with a graduation mark on this instrument indicates that the aircraft is turning a $1\text{-}1/2^\circ$ per second and completes a 360° turn in 4 minutes. It is customary to placard the instrument face with words indicating whether it is a 2 or 4 minute turn indicator. (*Figure 5-99*)

The turn pointer indicates the rate at which an aircraft is turning about its vertical axis. It does so by using the precession of a gyro to tilt a pointer. The gyro spins in a vertical plane aligned with the longitudinal axis of the aircraft. When the aircraft rotates about its vertical axis during a turn, the force experienced by the spinning gyro is exerted about the vertical axis. Due to precession, the reaction of the gyro rotor is 90° further around the gyro in the direction of spin. This means the reaction to the force around the vertical axis is movement around the longitudinal axis of the aircraft. This causes the top of the rotor to tilt to the left or right. The pointer is attached with linkage that makes the pointer deflect in the opposite direction, which matches the direction of



Figure 5-99. Turn-and-slip indicator.

turn. So, the aircraft's turn around the vertical axis is indicated around the longitudinal axis on the gauge. This is intuitive to the pilot when regarding the instrument, since the pointer indicates in the same direction as the turn. (*Figure 5-100*)

The slip indicator (ball) part of the instrument is an inclinometer. The ball responds only to gravity during coordinated straight-and-level flight. Thus, it rests in the lowest part of the curved glass between the reference wires. When a turn is initiated and the aircraft is banked, both gravity and the centrifugal force of the turn act upon the ball. If the turn is coordinated, the ball remains in place. Should a skidding turn exist, the centrifugal force exceeds the force of gravity on the ball and it moves in the direction of the outside of the turn.

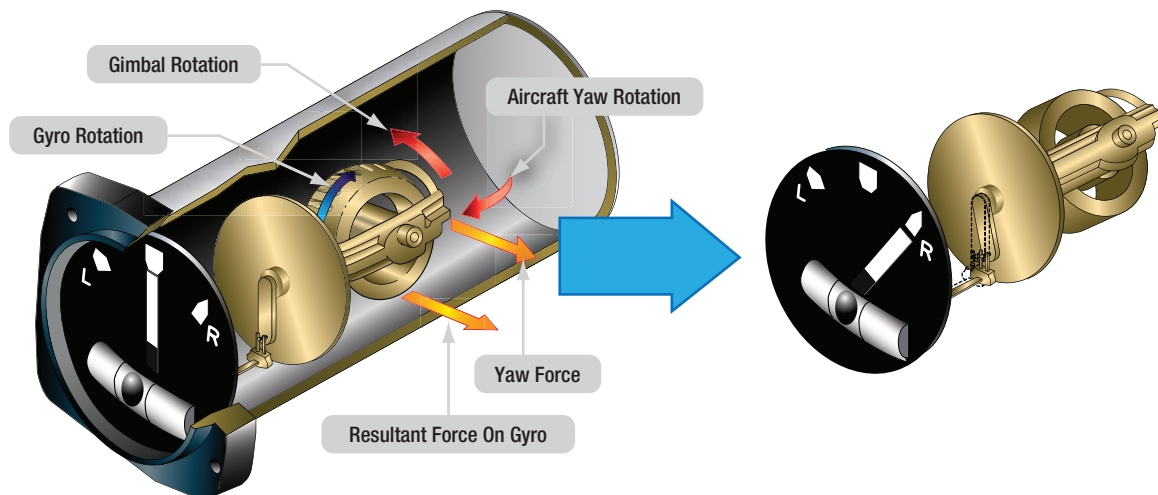


Figure 5-100. The turn-and-slip indicator's gyro reaction to the turning force in a right hand turn. The yaw force results in a force on the gyro 90° around the rotor in the direction it is turning due to precession. This causes the top of the rotor to tilt to the left. Through connecting linkage, the pointer tilts to the right.

During a slipping turn, there is more bank than needed, and gravity is greater than the centrifugal force acting on the ball. The ball moves in the curved glass toward the inside of the turn. As mentioned previously, often power for the turn-and-slip indicator gyro is electrical if the attitude and direction indicators are vacuum powered. This allows limited operation off battery power should the vacuum system and the electric generator fail. The directional and attitude information from the turn-and-slip indicator, combined with information from the pitot-static instruments, allow continued safe emergency operation of the aircraft.

Electrically powered turn-and-slip indicators are usually DC powered. Vacuum-powered turn-and-slip indicators are usually run on less vacuum (approximately 2" Hg) than fully gimballed attitude and direction indicators. Regardless, proper vacuum must be maintained for accurate turn rate information to be displayed.

GLASS COCKPIT

In an effort to increase the safety of operating complicated aircraft, computers and computer systems have been incorporated. Flight instrumentation and engine and airframe monitoring are areas particularly well suited to gain advantages from the use of computers. They contribute by helping to reduce instrument panel clutter and focusing the pilot's attention only on matters of imminent importance.

"Glass cockpit" is a term that refers to the use of flat-panel display screens in cockpit instrumentation. In reality, it also refers to the use of computer-produced images that have replaced individual mechanical gauges. Moreover, computers and computer systems monitor the processes and components of an operating aircraft beyond human ability while relieving the pilot of the stress from having to do so.

Computerized electronic flight instrument and maintenance systems have additional benefits. The solid-state nature of the components increases reliability. Also, microprocessors, data buses, and LCDs all save space and weight. Technicians interface with EICAS (Engine Indicating and Crew Alerting System) and ECAM (Electronic Centralized Aircraft Monitoring) systems through control panels to gather operating and maintenance data. (*Figure 5-101*)

These systems have been developed and utilized on aircraft for a number of years. New systems and computer architecture development is on going. Details on the operation and use of these glass cockpit maintenance aids are located in the manufacturer's maintenance manual.

AVIONICS SYSTEMS

AUTOFLIGHT

An aircraft automatic pilot system controls the aircraft without the pilot directly maneuvering the controls. The autopilot maintains the aircraft's attitude and/or direction and returns the aircraft to that condition when it is displaced from it. Automatic pilot systems are capable of keeping aircraft stabilized laterally, vertically, and longitudinally. On a large aircraft, the autopilot is engaged in auto flight mode. Automatic flight control systems encompass autopilot and related systems such as auto throttle and auto land. A discussion of the basics of an auto pilot systems follows.

The primary purpose of an autopilot system is to reduce the work strain and fatigue of controlling the aircraft during long flights. Most autopilots have both manual and automatic modes of operation. In the manual mode, the pilot selects each maneuver and makes small inputs into an autopilot controller. The autopilot system moves the control surfaces of the aircraft to perform the maneuver. In automatic mode, the pilot selects the attitude and direction desired for a flight segment. The autopilot then moves the control surfaces to attain and maintain these parameters.

Autopilot systems provide for one, two, or three axis control of an aircraft. Those that manage the aircraft around only one axis control the ailerons.

They are single-axis autopilots, known as wing leveler systems, usually found on light aircraft. (*Figure 5-102*) Other autopilots are two axis systems that control the ailerons and elevators. Three axis autopilots control the ailerons, elevators, and the rudder. Two and three axis autopilot systems can be found on aircraft of all sizes.

There are many autopilot systems available. They feature a wide range of capabilities and complexity. Light aircraft typically have autopilots with fewer capabilities than high performance and transport category aircraft.

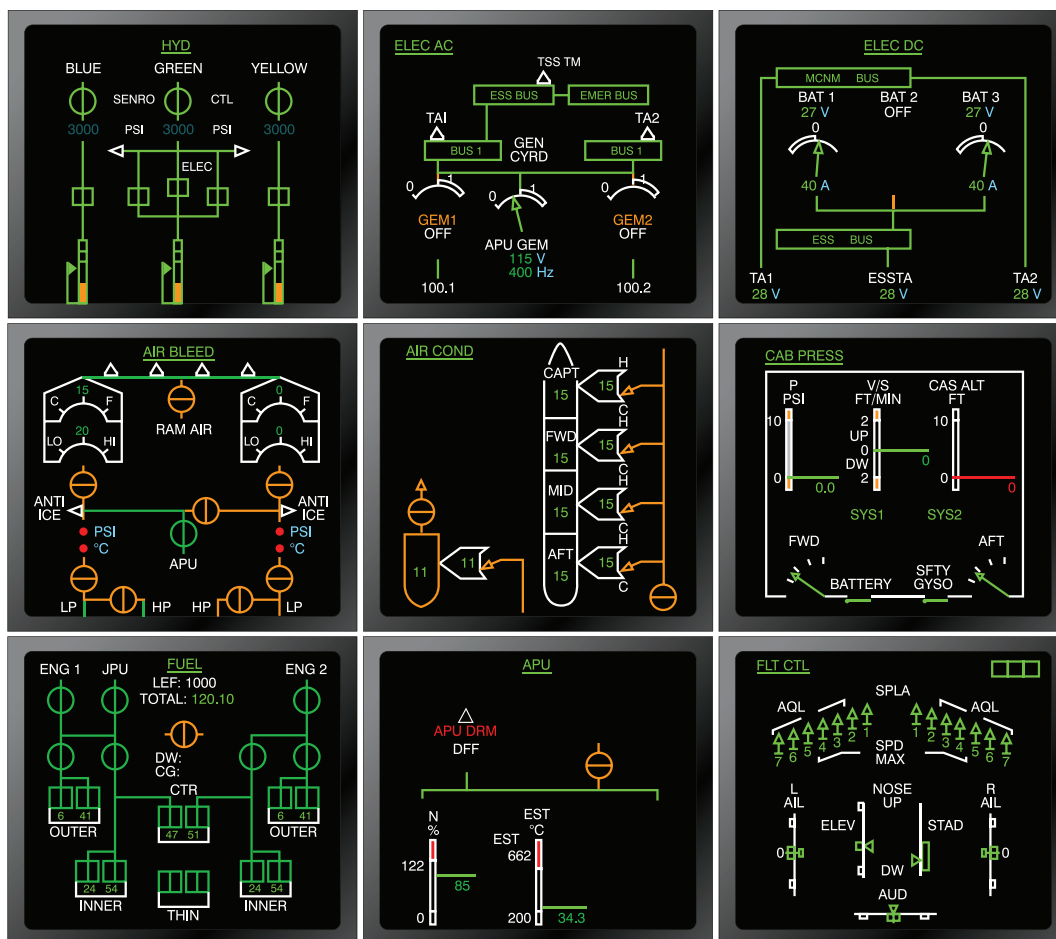


Figure 5-101. Glass cockpit displays allow system operating and maintenance data to be accessed by the technician. Nine of the 12 available system diagrams from an ECAM system are shown. The technician uses a control panel to select the desired system for display.

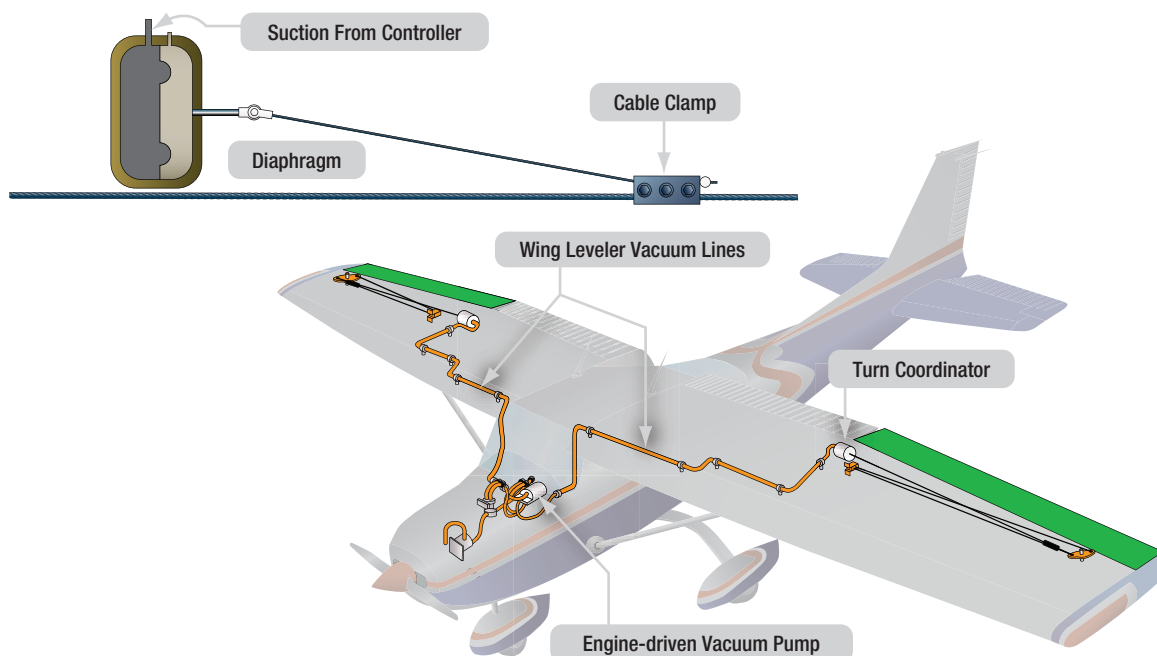


Figure 5-102. The wing leveler system on a small aircraft is a vacuum-operated single-axis autopilot. Only the ailerons are controlled. The aircraft's turn coordinator is the sensing element. Vacuum from the instrument vacuum system is metered to the diaphragm cable actuators to move the ailerons when the turn coordinator senses roll.

Integration of navigation functions is common, even on light aircraft autopilots. As autopilots increase in complexity, they not only manipulate the flight control surfaces, but other flight parameters as well.

Some modern small aircraft, high-performance, and transport category aircraft have very elaborate autopilot systems known as automatic flight control systems (AFCS). These three axis systems go far beyond steering the aeroplane. They control the aircraft during climbs, descents, cruise, and approach to landing. Some even integrate an auto throttle function that automatically controls engine thrust that makes auto landings possible. For further automatic control, flight management systems have been developed.

Through the use of computers, an entire flight profile can be programmed ahead of time allowing the pilot to supervise its execution. An FMS computer coordinates nearly every aspect of a flight, including the autopilot and auto throttle systems, navigation route selection, fuel management schemes, and more.

BASIS FOR AUTOPILOT OPERATION

The basis for autopilot system operation is error correction. When an aircraft fails to meet the conditions selected, an error is said to have occurred. The autopilot system automatically corrects that error and restores the aircraft to the flight attitude desired by the pilot. There are two basic ways modern autopilot systems do this. One is position based and the other is rate based.

A position based autopilot manipulates the aircraft's controls so that any deviation from the desired attitude of the aircraft is corrected. This is done by memorizing the desired aircraft attitude and moving the control surfaces so that the aircraft returns to that attitude.

Rate based autopilots use information about the rate of movement of the aircraft, and move control surfaces to counter the rate of change that causes the error. Most large aircraft use rate based autopilot systems. Small aircraft may use either.

AUTOPILOT COMPONENTS

Most autopilot systems consist of four basic components, plus various switches and auxiliary units. The four basic components are: sensing elements, computing element, output elements, and command elements. Many

advanced autopilot systems contain a fifth element: feedback or follow up. This refers to signals sent as corrections are being made by the output elements to advise the autopilot of the progress being made. (*Figure 5-103*)

Sensing Elements

The attitude and directional gyros, the turn coordinator, and an altitude control are the autopilot sensing elements. These units sense the movements of the aircraft. They generate electric signals that are used by the autopilot to automatically take the required corrective action needed to keep the aircraft flying as intended. The sensing gyros can be located in the cockpit mounted instruments. They can also be remotely mounted. Remote gyro sensors drive the servo displays in the cockpit panel, as well as provide the input signals to the autopilot computer.

Modern digital autopilots may use a variety of different sensors. MEMS gyros may be used or accompanied by the use solid state accelerometers and magnetometers. Rate based systems may not use gyros at all. Various input sensors may be located within the same unit or in separate units that transfer information via digital data bus. Navigation information is also integrated via digital data bus connection to avionics computers.

Computer and Amplifier

The computing element of an autopilot may be analog or digital. Its function is to interpret the sensing element data, integrate commands and navigational input, and send signals to the output elements to move the flight controls as required to control the aircraft. An amplifier is used to strengthen the signal for processing, if needed, and for use by the output devices, such as servo motors. The amplifier and associated circuitry is the computer of an analog autopilot system. Information is handled in channels corresponding to the axis of control for which the signals are intended (i.e., pitch channel, roll channel, or yaw channel). Digital systems use solid state microprocessor computer technology and typically only amplify signals sent to the output elements.

Output Elements

The output elements of an autopilot system are the servos that cause actuation of the flight control surfaces. They are independent devices for each of the control channels that integrate into the regular flight control system. Autopilot servo designs vary widely depending

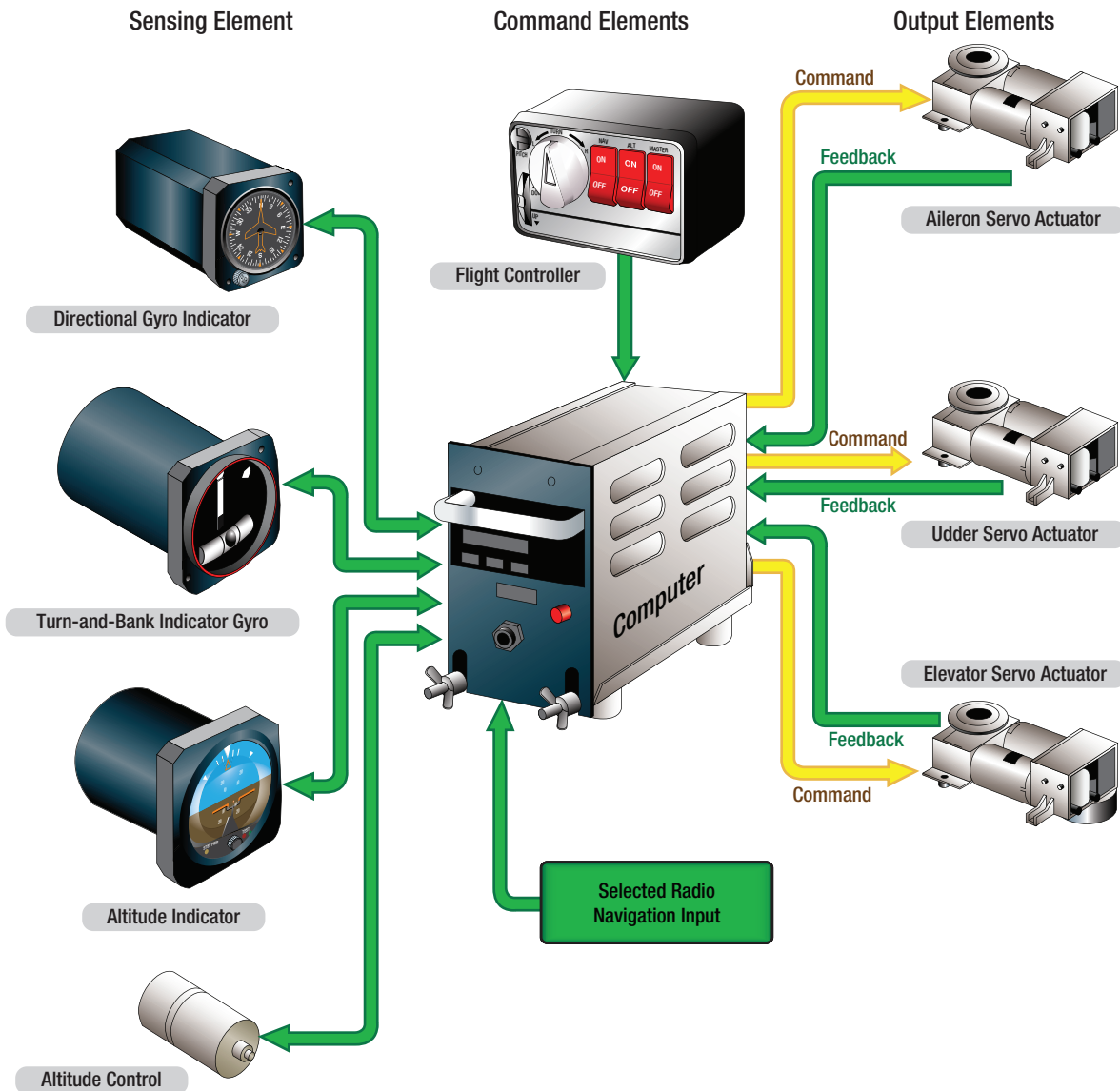


Figure 5-103. Typical analog autopilot system components.

on the method of actuation of the flight controls. Cable-actuated systems typically utilize electric servo motors or electro-pneumatic servos. Hydraulic actuated flight control systems use electro-hydraulic autopilot servos.

Digital fly-by-wire aircraft utilize the same actuators for carrying out manual and autopilot maneuvers. When the autopilot is engaged, the actuators respond to commands from the autopilot rather than exclusively from the pilot. Regardless, autopilot servos must allow unimpeded control surface movement when the autopilot is not operating.

Aircraft with cable actuated control surfaces use two basic types of electric motor operated servos. In one, a motor is connected to the servo output shaft through reduction

gears. The motor starts, stops, and reverses direction in response to the commands of autopilot computer. The other type of electric servo uses a constantly running motor geared to the output shaft through two magnetic clutches. The clutches are arranged so that energizing one clutch transmits motor torque to turn the output shaft in one direction; energizing the other clutch turns the shaft in the opposite direction. (*Figure 5-104*)

Electro-pneumatic servos can also be used to drive cable flight controls in some autopilot systems. They are controlled by electrical signals from the autopilot amplifier and actuated by an appropriate air pressure source. The source may be a vacuum system pump or turbine engine bleed air.

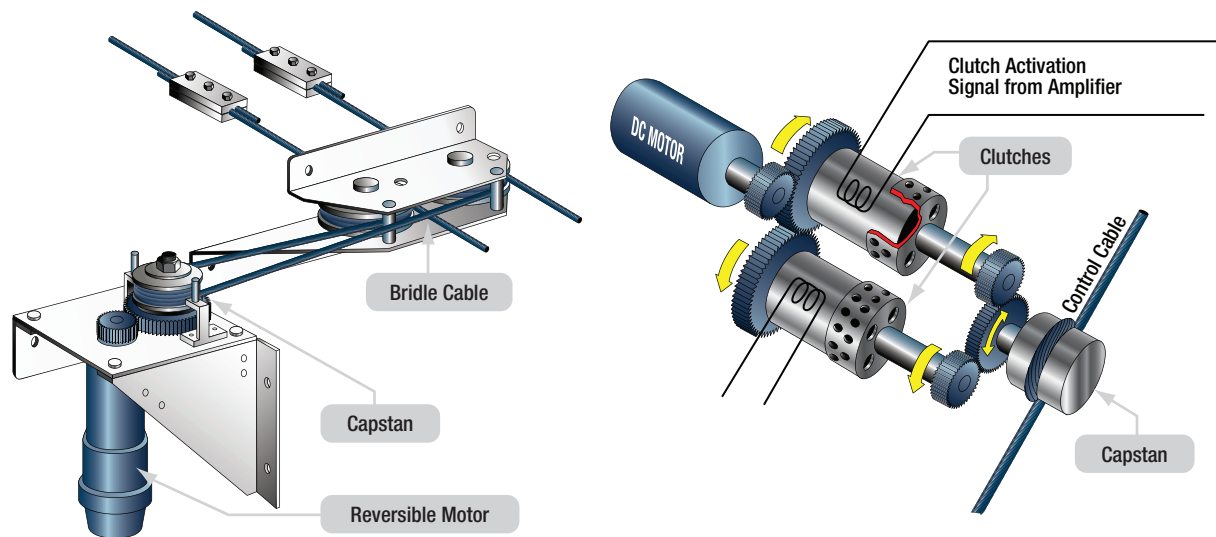


Figure 5-104. A reversible motor with capstan and bridle cable (left), and a single-direction constant motor with clutches that drive the output shafts and control cable in opposite directions (right).

Each servo consists of an electromagnetic valve assembly and an output linkage assembly. Aircraft with hydraulically actuated flight control systems have autopilot servos that are electro hydraulic. They are control valves that direct fluid pressure as needed to move the control surfaces via the control surface actuators.

They are powered by signals from the autopilot computer. When the autopilot is not engaged, the servos allow hydraulic fluid to flow unrestricted in the flight control system for normal operation. The servo valves can incorporate feedback transducers to update the autopilot of progress during error correction.

Command Elements

The command unit, called a flight controller, is the human interface of the autopilot. It allows the pilot to tell the autopilot what to do. Flight controllers vary with the complexity of the autopilot system. By pressing the desired function buttons, the pilot causes the controller to send instruction signals to the autopilot computer, enabling it to activate the proper servos to carry out the command(s). Level flight, climbs, descents, turning to a heading, or flying a desired heading are some of the choices available on most autopilots. Many aircraft make use of a multitude of radio navigational aids. These can be selected to issue commands directly to the autopilot computer. (*Figure 5-105*)

In addition to an on/off switch on the autopilot controller, most autopilots have a disconnect switch located on the control wheel(s). This switch, operated by

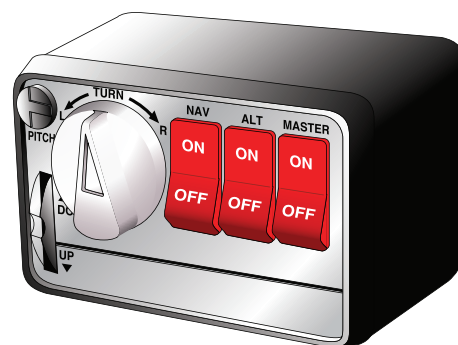


Figure 5-105. An autopilot controller of a simple autopilot system.

thumb pressure, can be used to disengage the autopilot system should a malfunction occur in the system or any time the pilot wishes to take manual control of the aircraft.

Feedback or Follow up Element

As an autopilot maneuvers the flight controls to attain a desired flight attitude, it must reduce control surface correction as the desired attitude is nearly attained so the controls and aircraft come to rest on course. Without doing so, the system would continuously over correct. Surface deflection would occur until the desired attitude is attained. But movement would still occur as the surface(s) returned to pre-error position. The attitude sensor would once again detect an error and begin the correction process all over again.

Various electric feedback, or follow up signals, are generated to progressively reduce the error message in the autopilot so that continuous over correction does

not take place. This is typically done with transducers on the surface actuators or in the autopilot servo units. Feedback completes a loop as illustrated in **Figure 5-106**.

A rate system receives error signals from a rate gyro that are of a certain polarity and magnitude that cause the control surfaces to be moved. As the control surfaces counteract the error and move to correct it, follow up signals of opposite polarity and increasing magnitude counter the error signal until the aircraft's correct attitude is restored. A displacement follow up system uses control surface pickups to cancel the error message when the surface has been moved to the correct position.

AUTOPILOT FUNCTIONS

The following autopilot system description is presented to show the function of a simple analog autopilot. Most autopilots are far more sophisticated; however, many of the operating fundamentals are similar.

The automatic pilot system flies the aircraft by using electrical signals developed in gyro-sensing units. These units are connected to flight instruments that indicate direction, rate of turn, bank, or pitch. If the flight attitude or magnetic heading is changed, electrical signals are developed in the gyros. These signals are sent to the autopilot computer/amplifier and are used to control the operation of servo units.

A servo for each of the three control channels converts electrical signals into mechanical force, which moves the control surface in response to corrective signals or pilot commands. The rudder channel receives two signals that determine when and how much the rudder moves. The first signal is a course signal derived from a compass system.

As long as the aircraft remains on the magnetic heading it was on when the autopilot was engaged, no signal develops. But, any deviation causes the compass system to send a signal to the rudder channel that is proportional to the angular displacement of the aircraft from the preset heading. The second signal received by the rudder channel is the rate signal that provides information anytime the aircraft is turning about the vertical axis. This information is provided by the turn-and-bank indicator gyro. When the aircraft attempts to turn off course, the rate gyro develops a signal proportional to the rate of turn, and the course gyro develops a signal

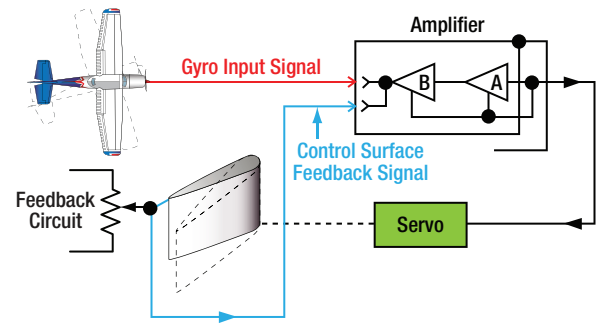


Figure 5-106. Basic function of an analog autopilot system including follow up or feedback signal.

proportional to the amount of displacement. The two signals are sent to the rudder channel of the amplifier, where they are combined and their strength is increased. The amplified signal is then sent to the rudder servo. The servo turns the rudder in the proper direction to return the aircraft to the selected magnetic heading.

As the rudder surface moves, a follow up signal is developed that opposes the input signal. When the two signals are equal in magnitude, the servo stops moving. As the aircraft arrives on course, the course signal reaches a zero value, and the rudder is returned to the streamline position by the follow up signal.

The aileron channel receives its input signal from a transmitter located in the gyro horizon indicator. Any movement of the aircraft about its longitudinal axis causes the gyro-sensing unit to develop a signal to correct for the movement. This signal is amplified, phase detected, and sent to the aileron servo, which moves the aileron control surfaces to correct for the error. As the aileron surfaces move, a follow up signal builds up in opposition to the input signal. When the two signals are equal in magnitude, the servo stops moving.

Since the ailerons are displaced from the streamline, the aircraft now starts moving back toward level flight with the input signal becoming smaller and the follow up signal driving the control surfaces back toward the streamline position. When the aircraft has returned to level flight roll attitude, the input signal is again zero. At the same time, the control surfaces are streamlined, and the follow up signal is zero.

The elevator channel circuits are similar to those of the aileron channel, with the exception that the elevator channel detects and corrects changes in pitch attitude

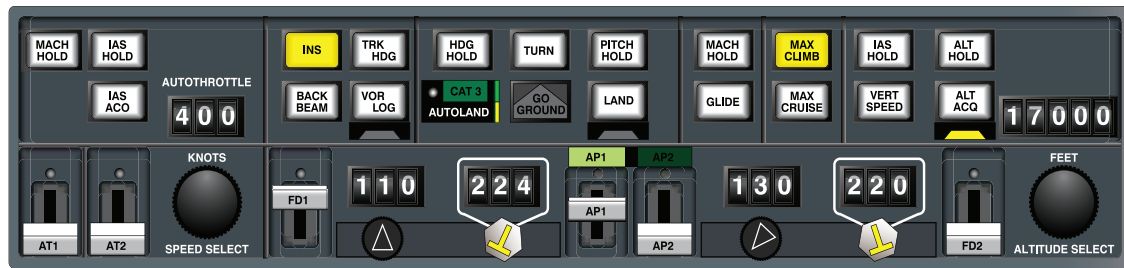


Figure 5-107. The AFCS control panel commands several integrated systems from a single panel including: flight directors, autopilots, autothrottles, autoland, and navigational aids. Mode selections for many features are made from this single interface.

of the aircraft. For altitude control, a remotely mounted unit containing an altitude pressure diaphragm is used. Similar to the attitude and directional gyros, the altitude unit generates error signals when the aircraft has moved from a preselected altitude. This is known as an altitude hold function. The signals control the pitch servos, which move to correct the error. An altitude select function causes the signals to continuously be sent to the pitch servos until a preselected altitude has been reached. The aircraft then maintains the preselected altitude using altitude hold signals.

Yaw Dampening

Many aircraft have a tendency to oscillate around their vertical axis while flying a fixed heading. Near continuous rudder input is needed to counteract this effect. A yaw damper is used to correct this motion. It can be part of an autopilot system or a completely independent unit. A yaw damper receives error signals from the turn coordinator rate gyro. Oscillating yaw motion is counteracted by rudder movement, which is made automatically by the rudder servo(s) in response to the polarity and magnitude of the error signal.

AUTOMATIC FLIGHT CONTROL SYSTEM

An aircraft autopilot with many features and various autopilot related systems integrated into a single system is called an automatic flight control system (AFCS). These were formerly found only on high-performance aircraft. Currently, due to advances in digital technology for aircraft, modern aircraft of any size may have AFCS.

AFCS capabilities vary from system to system. Some of the advances beyond ordinary autopilot systems are the extent of programmability, the level of integration of navigational aids, the integration of flight director and autothrottle systems, and combining of the command elements of these various systems into a single integrated flight control human interface. (*Figure 5-107*)

It is at the AFCS level of integration that an autothrottle system is integrated into the flight director and autopilot systems with glide scope modes so that auto landings are possible. Small general aviation aircraft being produced with AFCS may lack the throttle-dependent features.

Modern general aviation AFCS are fully integrated with digital attitude heading and reference systems (AHRS) and navigational aids including glideslope. They also contain modern computer architecture for the autopilot (and flight director systems) that is slightly different than described above for analog autopilot systems. Functionality is distributed across a number of interrelated computers and includes the use of intelligent servos that handle some of the error correction calculations. The servos communicate with dedicated avionics computers and display unit computers through a control panel, while no central autopilot computer exists. (*Figure 5-108*)

FLIGHT DIRECTOR SYSTEMS

A flight director system is an instrument system consisting of electronic components that compute and indicate the aircraft attitude required to attain and maintain a preselected flight condition. A command bar on the aircraft's attitude indicator shows the pilot how much and in what direction the attitude of the aircraft must be changed to achieve the desired result. The computed command indications relieve the pilot of many of the mental calculations required for instrument flights, such as interception angles, wind drift correction, and rates of climb and descent. Essentially, a flight director system is an autopilot system without the servos.

All of the same sensing and computations are made, but the pilot controls the aeroplane and makes maneuvers by following the commands displayed on the instrument panel. Flight director systems can be part of an autopilot system or exist on aircraft that do not possess full

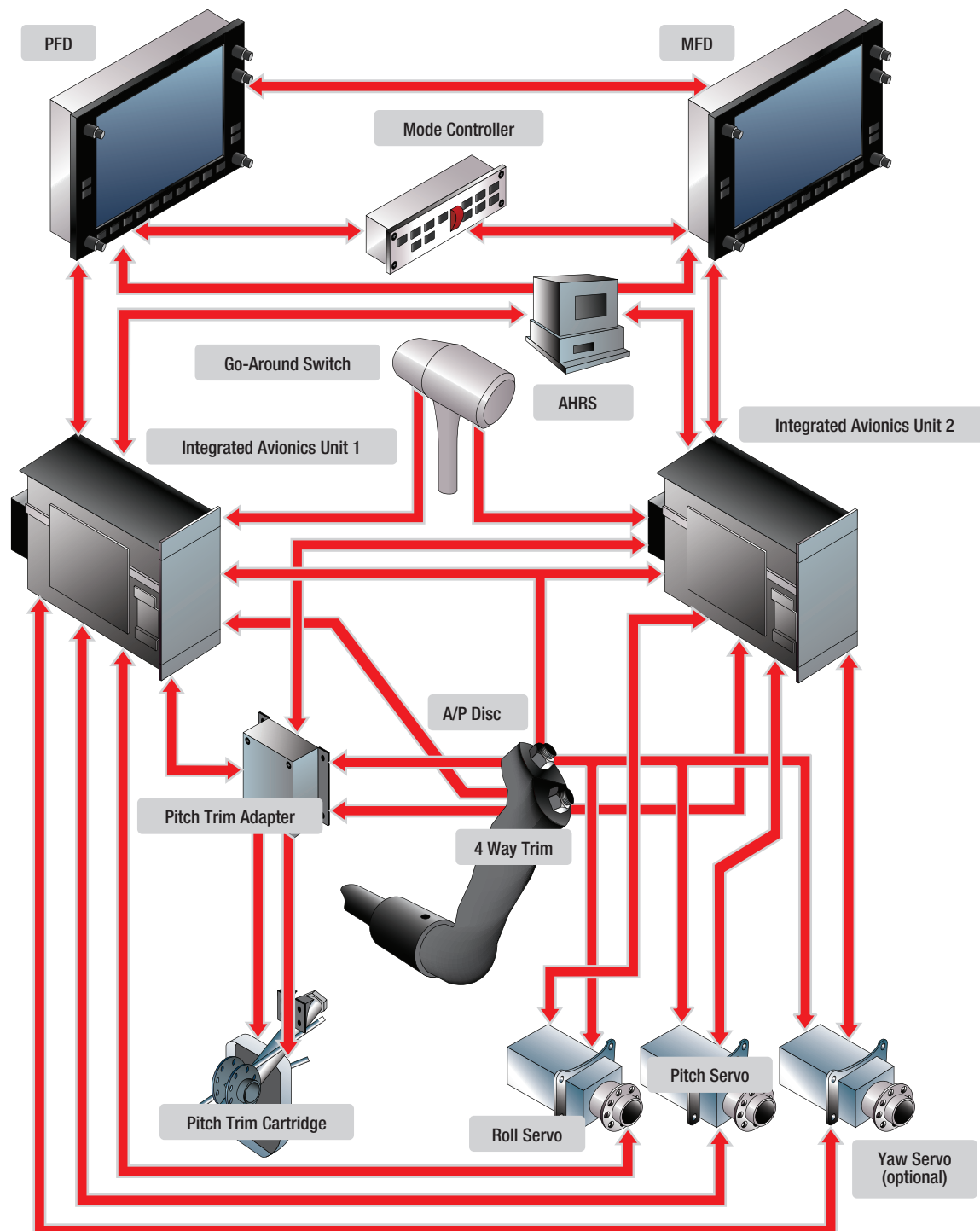


Figure 5-108. Automatic flight control system (AFCS) of a Garmin G1000 glass cockpit instrument system for a general aviation aircraft.

autopilot systems. Many autopilot systems allow for the option of engaging or disengaging a flight director display. Flight director information is displayed on the instrument that displays the aircraft's attitude. The process is accomplished with a visual reference technique. A symbol representing the aircraft is fit into a command bar positioned by the flight director in the proper location for a maneuver to be accomplished. The

symbols used to represent the aircraft and the command bar vary by manufacturer. Regardless, the object is always to fly the aircraft symbol into the command bar symbol. (*Figure 5-109*)

The instrument that displays the flight director commands is known as a flight director indicator (FDI), attitude director indicator (ADI), or electronic attitude

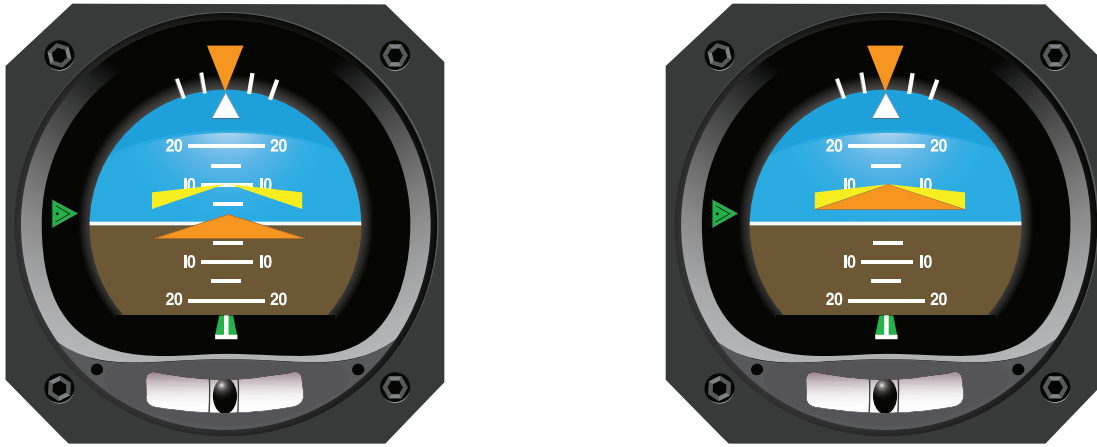


Figure 5-109. The flight director command bar signals the pilot how to steer the aircraft for a maneuver. By flying the aircraft so the triangular aeroplane symbol fits into the command bar, the pilot performs the maneuver calculated by the flight director. The instrument shown on the left is commanding a climb while the aeroplane is flying straight and level. The instrument on the right shows that the pilot has accomplished the maneuver.

director indicator (EADI). It may even be referred to as an artificial horizon with flight director. This display element combines with the other primary components of the flight director system. Like an autopilot, these consist of the sensing elements, a computer, and an interface panel. Integration of navigation features into the attitude indicator is highly useful. The flight director contributes to this usefulness by indicating to the pilot how to maneuver the aeroplane to navigate a desired course. Selection of the VOR function on the flight director control panel links the computer to the omnirange receiver. The pilot selects a desired course and the flight director displays the bank attitude necessary to intercept and maintain this course. Allocations for wind drift and calculation of the intercept angle is performed automatically.

Flight director systems vary in complexity and features. Many have altitude hold, altitude select, pitch hold, and other features. But flight director systems are designed to offer the greatest assistance during the instrument approach phase of flight. ILS localizer and glideslope signals are transmitted through the receivers to the computer and are presented as command indications. This allows the pilot to fly the aeroplane down the optimum approach path to the runway using the flight director system. With the altitude hold function engaged, level flight can be maintained during the maneuvering and procedure turn phase of an approach. Altitude hold automatically disengages when the glideslope is intercepted.

Once inbound on the localizer, the command signals of the flight director are maintained in a centered or zero condition. Interception of the glideslope causes a downward indication of the command pitch indicator. Any deviation from the proper glideslope path causes a fly-up or fly-down command indication. The pilot needs only to keep the aeroplane symbol fit into the command bar.

COMMUNICATIONS

Communication systems on large aircraft include radios for the flight crew to speak to air traffic control, ground operations and other aircraft while in flight. It also includes an interphone system that provides communication between the flight deck, and the passenger cabin while in flight and between ground crew personnel and the flight deck when the aircraft is on the ground. Modern aircraft may also include satellite communications technology (SATCOM) for communication with operations and for providing internet connections for passengers.

VERY HIGH FREQUENCY (VHF) RADIOS

Radio voice communication between an aircraft and air traffic control is maintained with very high frequency range (VHF) radios. There are typically a minimum of two such radios installed on an aircraft for redundancy. VHF radios are relatively short range radios in that they operate when within line-of-sight of each other. The frequencies used range from 118.0 MHz to 136.975 MHz. Seven hundred and twenty separate and distinct channels have been designated in this range with 25 kilohertz spacing between each channel. Further

division of the bandwidth is possible, such as in Europe where 8.33 kilohertz separate each VHF communication channel. VHF radios are used for communications between aircraft and air traffic control (ATC), as well as air-to-air communication between aircraft.

When using VHF, each party transmits and receives on the same channel. Only one party can transmit at any one time. VHF radios on transport aircraft are typically located in an avionics bay. (*Figure 5-110*) They are connected to a tuning and frequency display head on the flight deck and to the flight deck audio control panel via a data bus. The audio control panel illustrated in *Figure 5-111* allows any of three different VHF radios to be chosen.

HIGH FREQUENCY RADIOS

Some transoceanic aircraft may also use high frequency (HF) radios for long distance communication with an operator's base station. HF radio waves bounce off of the ionosphere layer of the atmosphere. This refraction extends the range of HF signals beyond line-of-sight. The frequency range is between 2 to 25 MHz.

SELECTIVE CALLING

Many commercial aircraft are equipped with a selective calling feature known as SELCAL. It works in conjunction with VHF or HF radio. When ground operations wish to contact the flight crew, the SELCAL panel on the flight deck illuminates and a tone is heard.

SATELLITE COMMUNICATION SYSTEMS

Use of satellite communication systems is common on large passenger aircraft. SATCOM uses a combination of ground stations and orbiting satellites to relay transmissions to and from the aircraft. INMARSAT is one such commercially available system that includes ten orbiting satellites used to provide a number of different services. Airline operators typically have SATCOM radios for enroute communication with ground stations. SATCOM is also used to connect passengers to the internet.

SERVICE INTERPHONE SYSTEM

Airliners are equipped with a service interphone system. This is an intercom type system that allows communication between the flight deck and the passenger cabin as well as connecting both of these areas with ground personnel outside the aircraft. Cabin



Figure 5-110. A transport aircraft VHF communications radio is located in the avionics bay with a control panel on the flight deck.



Figure 5-111. A typical airliner audio control panel.

phones (mikes and receivers) become active when picked up for use. To broadcast over the cabin speaker system, a switch must be thrown.

On the flight deck, crew typically can choose between communication with the passenger cabin or ground crew. The interphone system is accessed by maintenance personnel outside the aircraft by plugging a headset into one of the many jacks spread around the exterior of the aircraft. Exterior interphone jacks are typically located at the following areas:

- Fueling Station
- Cargo Bays
- Tail Cone
- APU (Compartment and Control Panel)
- Wheel Wells
- Equipment Bays
- Engine Nacelles

Modern interphone systems are controlled by an audio management unit (AMU) or similar "black box" located in the equipment bay. They are DC powered and often incorporate a ground crew call system. The ground crew

call system creates audible and visual alerts on the flight deck when a ground crew wishes to speak with someone on the flight deck using the interphone system. It can also be used by those on the flight deck to notify ground crew to plug in to the interphone system to communicate. A horn located on the exterior APU control panel sounds to gain the ground crew's attention when ground crew is selected in the cockpit. The same horn is used to notify ground crew that an equipment cooling failure has occurred on Boeing aircraft. It also sounds when the air data inertial reference unit is on and there is no AC power on the aircraft. An audio control panel or a control display unit is used for system selections.

NAVIGATION SYSTEMS

In the early years of aviation, a compass, a map, and dead reckoning were the only navigational tools. These were marginally reassuring if weather prevented the pilot from seeing the terrain below. Voice radio transmission from someone on the ground to the pilot indicating that the aircraft could be heard overhead was a preview of what electronic navigational aids could provide. For aviation to reach fruition as a safe, reliable, consistent means of transportation, some sort of navigation system needed to be developed. Early flight instruments contributed greatly to flying when the ground was obscured by clouds. Navigation aids were needed to indicate where an aircraft was over the earth as it progressed towards its destination. In the 1930s and 1940s, a radio navigation system was used that was a low frequency, four-course radio range system. Airports and selected navigation waypoints broadcast two Morse code signals with finite ranges and patterns.

Pilots tuned to the frequency of the broadcasts and flew in an orientation pattern until both signals were received with increasing strength. The signals were received

as a blended tone of the highest volume when the aircraft was directly over the broadcast area. From this beginning, numerous refinements to radio navigational aids developed.

Radio navigation aids supply the pilot with intelligence that maintains or enhances the safety of flight. As with communication radios, navigational aids are avionics devices, the repair of which must be carried out by trained technicians at certified repair stations. However, installation, maintenance and proper functioning of the electronic units, as well as their antennas, displays, and any other peripheral devices, are the responsibilities of the airframe technician.

VOR NAVIGATION SYSTEM

One of the oldest and most useful navigational aids is the VOR system. The system was constructed after WWII and is still in use today. It consists of thousands of land-based transmitter stations, or VORs, that communicate with radio receiving equipment on board aircraft. Many of the VORs are located along airways. The Victor airway system is built around the VOR navigation system. Ground VOR transmitter units are also located at airports where they are known as TVOR (terminal VOR). The U.S. Military has a navigational system known as TACAN that operates similarly to the VOR system. Sometimes VOR and TACAN transmitters share a location. These sites are known as VORTACs.

The position of all VORs, TVORs, and VORTACs are marked on aeronautical charts along with the name of the station, the frequency to which an airborne receiver must be tuned to use the station, and a Morse code designation for the station. Some VORs also broadcast a voice identifier on a separate frequency that is included on the chart. (*Figure 5-112*)



Figure 5-112. A VOR ground station.



Figure 5-113. V-shaped, horizontally polarized, bi-pole antennas are commonly used for VOR and VOR/glideslope reception. All antenna shown are VOR/glideslope antenna.

VOR uses VHF radio waves (108-117.95 MHz) with 50 kHz separation between each channel. This keeps atmospheric interference to a minimum but limits the VOR to line-of-sight usage. To receive VOR VHF radio waves, generally a V-shaped, horizontally polarized, bi-pole antenna is used. A typical location for the V dipole is in the vertical fin. Other type antennas are also certified. Follow the manufacturer's instructions for installation location. (*Figure 5-113*)

The signals produced by a VOR transmitter propagate 360° from the unit and are used by aircraft to navigate to and from the station with the help of an on board VOR receiver and display instruments. A pilot is not required to fly a pattern to intersect the signal from a VOR station since it propagates out in every direction. The radio waves are received as long as the aircraft is in range of the ground unit and regardless of the aircraft's direction of travel. (*Figure 5-114*)

A VOR transmitter produces two signals that a receiver on board an aircraft uses to locate itself in relation to the ground station. One signal is a reference signal. The second is produced by electronically rotating a variable signal. The variable signal is in phase with the reference signal when at magnetic north, but becomes increasingly out of phase as it is rotated to 180°. As it continues to rotate to 360° (0°), the signals become increasingly in phase until they are in phase again at magnetic north. The receiver in the aircraft deciphers the phase difference and determines the aircraft's position in degrees from the VOR ground based unit. (*Figure 5-115*)

Most aircraft carry a dual VOR receiver. Sometimes, the VOR receivers are part of the same avionics unit as the VHF communication transceiver(s). These are known as NAV/COM radios. Internal components are shared since frequency bands for each are adjacent. (*Figure 5-116*)

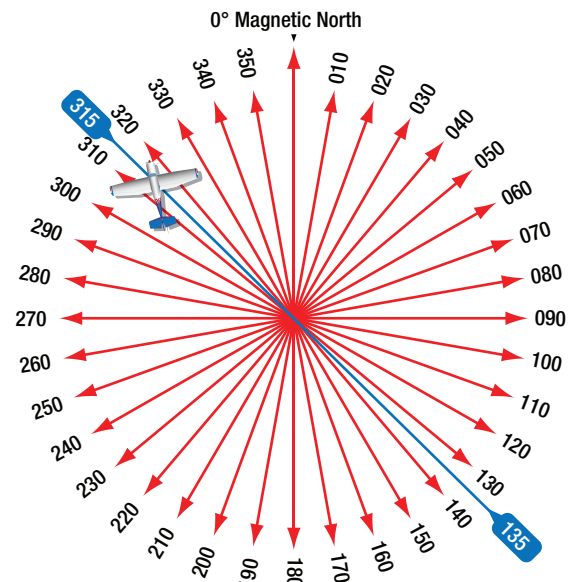


Figure 5-114. A VOR transmitter produces signals for 360° radials that an airborne receiver uses to indicate the aircraft's location in relation to the VOR station regardless of the aircraft's direction of flight. The aircraft shown is on the 315° radial even though it does not have a heading of 315°.

Large aircraft may have two dual receivers and even dual antennas. Normally, one receiver is selected for use and the second is tuned to the frequency of the next VOR station to be encountered en route. A means for switching between NAV 1 and NAV 2 is provided as is a switch for selecting the active or standby frequency. (*Figure 5-117*)

VOR receivers are also found coupled with instrument landing system (ILS) receivers and glideslope receivers. A VOR receiver interprets the bearing in degrees to (or from) the VOR station where the signals are generated. It also produces DC voltage to drive the display of the deviation from the desired course centerline to (or from) the selected station. Additionally, the receiver decides whether or not the aircraft is flying toward the VOR or

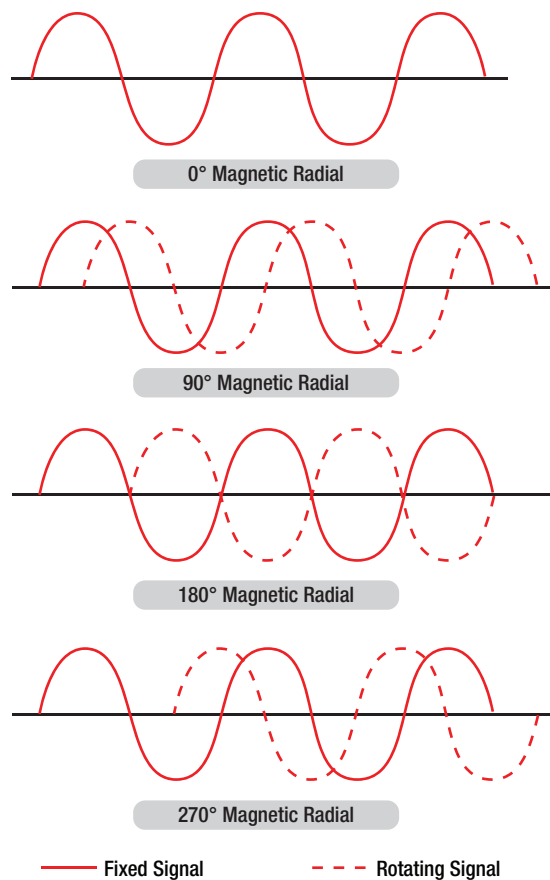


Figure 5-115. The phase relationship of the two broadcast VOR signals.



Figure 5-116. A NAV/COM receiver typically found in light aircraft.



Figure 5-117. An airliner VOR control head with two independent NAV receivers each with an active and standby tuning circuit controlled by a toggle switch.

away from it. These items can be displayed a number of different ways on various instruments. Older aircraft are often equipped with a VOR gauge dedicated to display only VOR information. This is also called an omni bearing selector (OBS) or a course deviation indicator (CDI). (*Figure 5-118*)

A separate gauge for the VOR information is not always used. As flight instruments and displays have evolved, VOR navigation information has been integrated into other instruments displays, such as the radio magnetic indicator (RMI), the horizontal situation indicator (HSI), an EFIS display or an electronic attitude director indicator (EADI). Flight management systems and automatic flight control systems are also made to integrate VOR information to automatically control the aircraft on its planned flight segments. Flat panel MFDs integrate VOR information into moving map presentations and other selected displays. The basic information of the radial bearing in degrees, course deviation indication, and to/from information remains unchanged however. (*Figure 5-119*)

At large airports, an instrument landing system (ILS) guides the aircraft to the runway while on an instrument landing approach. The aircraft's VOR receiver is used to interpret the radio signals. It produces a more sensitive course deviation indication on the same instrument display as the VOR CDI display. This part of the ILS is known as the localizer and is discussed below. While tuned to the ILS localizer frequency, the VOR circuitry of the VOR/ILS receiver is inactive.

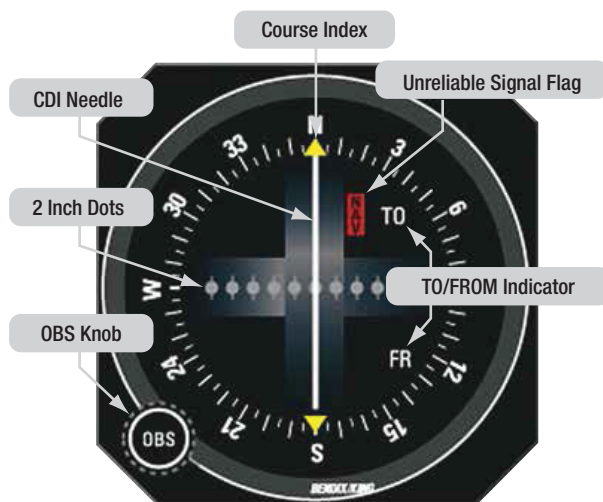


Figure 5-118. A traditional VOR gauge, also known as a course deviation indicator (CDI) or an omni-bearing selector (OBS).

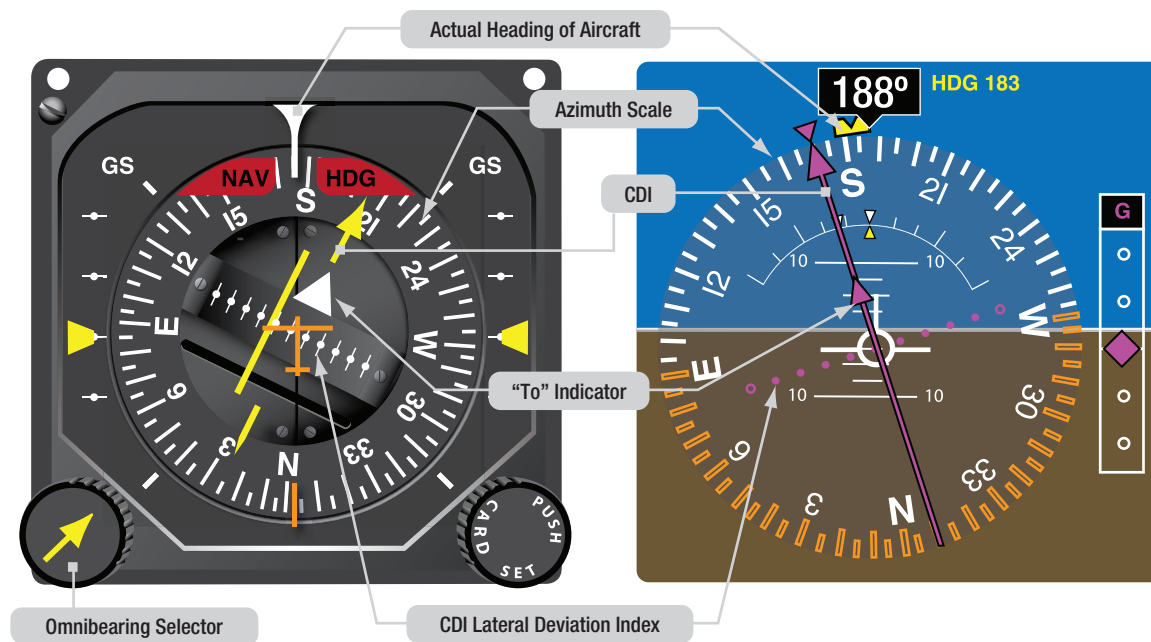


Figure 5-119. A mechanical HSI (left) and an electronic HSI (right) both display VOR information.

It is common at VOR stations to combine the VOR transmitter with distance measuring equipment (DME) or a nondirectional beacon (NDB) such as an ADF transmitter and antenna. When used with a DME, pilots can gain an exact fix on their location using the VOR and DME together. Since the VOR indicates the aircraft's bearing to the VOR transmitter and a co-located DME indicates how far away the station is, this relieves the pilot from having to fly over the station to know with certainty his or her location. These navigational aids are discussed separately in the following sections.

Functional accuracy of VOR equipment is critical to the safety of flight. VOR receivers are operationally tested using VOR test facilities (VOT). These are located at numerous airports that can be identified in the Airport Facilities Directory for the area concerned. Specific points on the airport surface are given to perform the test. Most VOTs require tuning 108.0 MHz on the VOR receiver and centering the CDI. The OBS should indicate 0° showing FROM on the indicator or 180° when showing TO. If an RMI is used as the indicator, the test heading should always indicate 180°. Some repair stations can also generate signals to test VOR receivers although not on 108.0 MHz. Contact the repair station for the transmission frequency and for their assistance in checking the VOR system.

A logbook entry is required. Note that some airborne testing using VOTs is possible by the pilot. An error of $\pm 4^\circ$ should not be exceeded when testing a VOR system with a VOT. An error in excess of this prevents the use of the aircraft for IFR flight until repairs are made. Aircraft having dual VOR systems where only the antenna is shared may be tested by comparing the output of each system to the other. Tune the VOR receivers to the local ground VOR station. A bearing indication difference of no more than $\pm 4^\circ$ is permissible.

AUTOMATIC DIRECTION FINDER (ADF)

An automatic direction finder (ADF) operates off of a ground signal transmitted from a NDB. Early radio direction finders (RDF) used the same principle.

A vertically polarized antenna was used to transmit LF frequency radio waves in the 190 kHz to 535 kHz range. A receiver on the aircraft was tuned to the transmission frequency of the NDB. Using a loop antenna, the direction to (or from) the antenna could be determined by monitoring the strength of the signal received. This was possible because a radio wave striking a loop antenna broadside induces a null signal. When striking it in the plane of the loop, a much stronger signal is induced. The NDB signals were modulated with unique Morse code pulses that enabled the pilot to identify the beacon to which he or she was navigating.

With RDF systems, a large rigid loop antenna was installed inside the fuselage of the aircraft. The broadside of the antenna was perpendicular to the aircraft's longitudinal axis. The pilot listened for variations in signal strength of the LF broadcast and maneuvered the aircraft so a gradually increasing null signal was maintained. This took them to the transmitting antenna. When over flown, the null signal gradually faded as the aircraft became farther from the station. The increasing or decreasing strength of the null signal was the only way to determine if the aircraft was flying to or from the NDB. A deviation left or right from the course caused the signal strength to sharply increase due to the loop antenna's receiving properties.

The ADF improved on this concept. The broadcast frequency range was expanded to include MF up to about 1 800 kHz. The heading of the aircraft no longer needed to be changed to locate the broadcast transmission antenna. In early model ADFs, a rotatable antenna was used instead. The antenna rotated to seek the position in which the signal was null. The direction to the broadcast antenna was shown on an azimuth scale of an ADF indicator in the flight deck. This type of instrument is still found in use today. It has a fixed card with 0° always at the top of a non-rotating dial. A pointer indicates the relative bearing to the station. When the indication is 0°, the aircraft is on course to (or from) the station. (Figure 5-120)

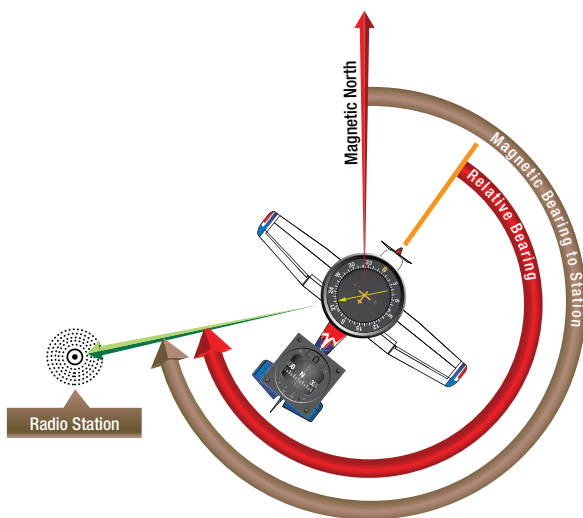


Figure 5-120. Older ADF indicators have nonrotating azimuth cards. 0° is fixed at the top of the instrument and the pointer always indicates the relative bearing to the ADF transmission antenna. To fly to the station, the pilot turns the aircraft until the ADF pointer indicates 0°.

As ADF technology progressed, indicators with rotatable azimuth cards became the norm. When an ADF signal is received, the pilot rotates the card so that the present heading is at the top of the scale. This results in the pointer indicating the magnetic bearing to the ADF transmitter. This is more intuitive and consistent with other navigational practices. (Figure 5-121)

In modern ADF systems, an additional antenna is used to remove the ambiguity concerning whether the aircraft is heading to or from the transmitter. It is called a sense antenna. The reception field of the sense antenna is omnidirectional. When combined with the fields of the loop antenna, it forms a field with a single significant null reception area on one side. This is used for tuning and produces an indication in the direction toward the ADF station at all times.

The on board ADF receiver needs only to be tuned to the correct frequency of the broadcast transmitter for the system to work. The loop and sense antenna are normally housed in a single, low profile antenna housing. (Figure 5-122)

Any ground antenna transmitting LF or MF radio waves in range of the aircraft receiver's tuning capabilities can be used for ADF. This includes those from AM radio stations. Audible identifier tones are loaded on the NDB carrier waves. Typically a two character Morse code designator is used.



Figure 5-121. A movable card ADF indicator can be rotated to put the aircraft's heading at the top of the scale. The pointer then points to the magnetic bearing the ADF broadcast antenna.

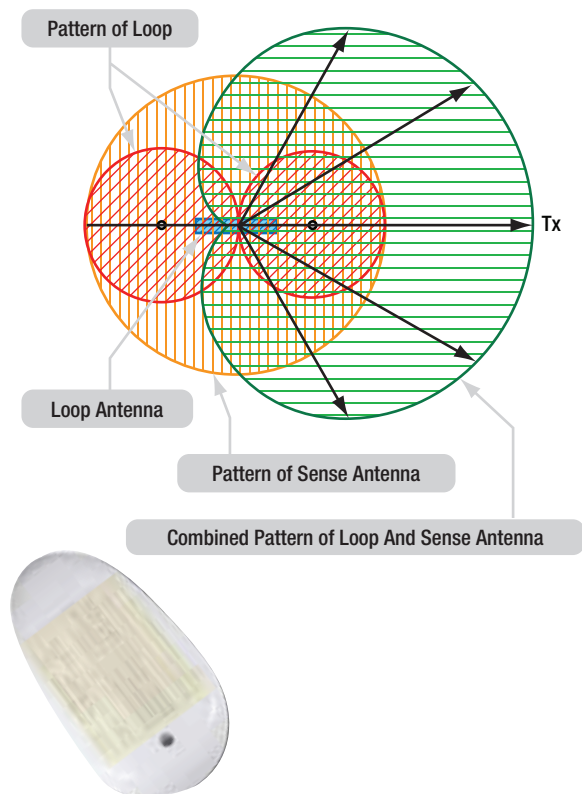


Figure 5-122. The reception fields of a loop and sense antenna combine to create a field with a sharp null on just one side. This removes directional ambiguity when navigating to an ADF station.

With an AM radio station transmission, the AM broadcast is heard instead of a station identifier code. The frequency for an NDB transmitter is given on an aeronautical chart next to a symbol for the transmitter. The identifying designator is also given. (*Figure 5-123*)

ADF receivers can be mounted in the flight deck with the controls accessible to the user. This is found on many general aviation aircraft. Alternately, the ADF receiver is mounted in a remote avionics bay with only the control head in the flight deck. Dual ADF receivers are common. ADF information can be displayed on the ADF indicators mentioned or it can be digital. Modern, flat, multipurpose electronic displays usually display the ADF digitally. (*Figure 5-124*)

When ANT is selected on an ADF receiver, the loop antenna is cut out and only the sense antenna is active. This provides better multi-directional reception of broadcasts in the ADF frequency range, such as weather or AWAS broadcasts.



Figure 5-123. Non-directional broadcast antenna in the LF and medium frequency range are used for ADF navigation.

When the best frequency oscillator (BFO) is selected on an ADF receiver/controller, an internal beat frequency oscillator is connected to the IF amplifier inside the ADF receiver. This is used when an NDB does not transmit a modulated signal. Continued refinements to ADF technology has brought it to its current state. The rotating receiving antenna is replaced by a fixed loop with a ferrite core. This increases sensitivity and allows a smaller antenna to be used. The most modern ADF systems have two loop antennas mounted at 90° to each other. The received signal induces voltage that is sent to two stators in a resolver or goniometer. The goniometer stators induce voltage in a rotor that correlates to the signal of the fixed loops. The rotor is driven by a motor to seek the null. The same motor rotates the pointer in the flight deck indicator to show the relative or magnetic bearing to the station. (*Figure 5-125*)



Figure 5-124. A cockpit mountable ADF receiver used on general aviation aircraft.

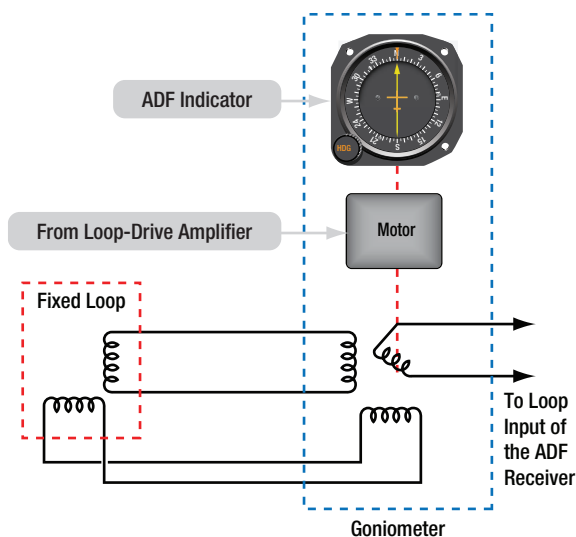


Figure 5-125. In modern ADF, a rotor in a goniometer replaces a the rotating loop antenna used in earlier models.

Technicians should note that the installation of the ADF antenna is critical to a correct indication since it is a directional device. Calibration with the longitudinal axis of the fuselage or nose of the aircraft is important. A single null reception area must exist in the correct direction. The antenna must be oriented so the ADF indicates station location when the aircraft is flying toward it rather than away. Follow all manufacturer's instructions. Radio Magnetic Indicator (RMI) To save space in the instrument panel and to consolidate related information into one easy to use location, the radio magnetic RMI combines indications from a magnetic compass, VOR, and ADF into one instrument. (*Figure 5-126*)

The azimuth card of the RMI is rotated by a remotely located flux gate compass. Thus, the magnetic heading of the aircraft is always indicated. The lubber line is usually a marker or triangle at the top of the instrument dial. The VOR receiver drives the solid pointer to indicate the magnetic direction TO a tuned VOR station. When the ADF is tuned to an NDB, the double, or hollow pointer, indicates the magnetic bearing TO the NDB.

Since the flux gate compass continuously adjusts the azimuth card so that the aircraft heading is at the top of the instrument, pilot workload is reduced. The pointers indicate where the VOR and ADF transmission stations are located in relationship to where the aircraft is currently positioned. Push buttons allow conversion of either pointer to either ADF or VOR for navigation involving two of one type of station and none of the other.

INSTRUMENT LANDING SYSTEMS (ILS)

An ILS is used to land an aircraft when visibility is poor. This radio navigation system guides the aircraft down a slope to the touch down area on the runway. Multiple radio transmissions are used that enable an exact approach to landing with an ILS. A localizer is one of the radio transmissions. It is used to provide horizontal guidance to the center line of the runway. A separate glideslope broadcast provides vertical guidance of the aircraft down the proper slope to the touch down point. Compass locator transmissions for outer and middle approach marker beacons aid the pilot in intercepting the approach navigational aid system. Marker beacons provide distance from the runway



Figure 5-126. A radio magnetic indicator (RMI) combines a magnetic compass, VOR, and ADF indications.

information. Together, all of these radio signals make an ILS a very accurate and reliable means for landing aircraft. (Figure 5-127)

Localizer

The localizer broadcast is a VHF broadcast in the lower range of the VOR frequencies (108 MHz-111.95 MHz) on odd frequencies only. Two modulated signals

are produced from a horizontally polarized antenna complex beyond the far end of the approach runway. They create an expanding field that is 2-1/2° wide (about 1 500 feet) 5 miles from the runway. The field tapers to runway width near the landing threshold. The left side of the approach area is filled with a VHF carrier wave modulated with a 90 Hz signal.

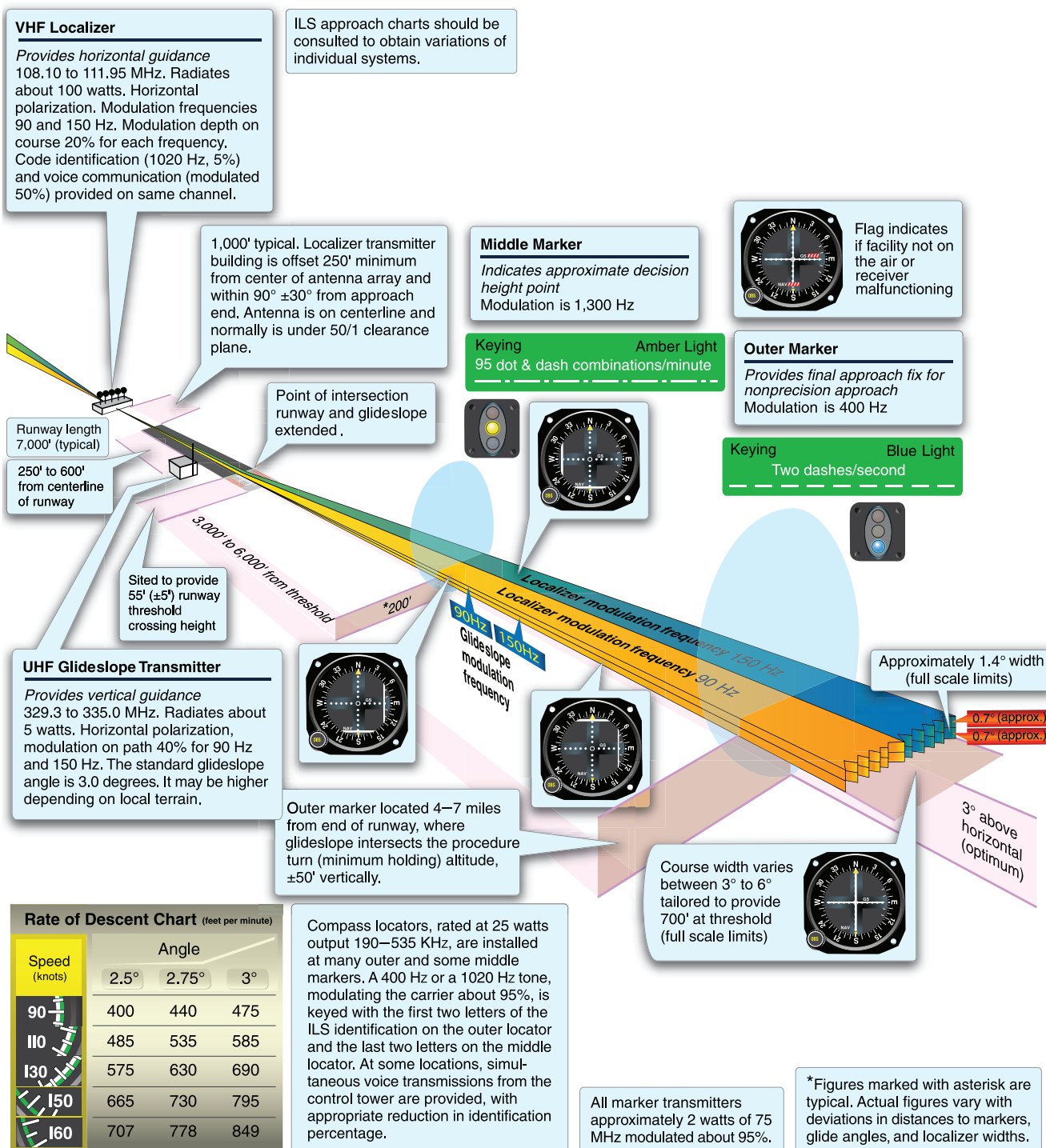


Figure 5-127. Components of an instrument landing system (ILS).

The right side of the approach contains a 150 MHz modulated signal. The aircraft's VOR receiver is tuned to the localizer VHF frequency that can be found on published approach plates and aeronautical charts. The circuitry specific to standard VOR reception is inactive while the receiver uses localizer circuitry and components common to both. The signals received are passed through filters and rectified into DC to drive the course deviation indicator.

If the aircraft receives a 150 Hz signal, the CDI of the VOR/ILS display deflects to the left. This indicates that the runway is to the left. The pilot must correct course with a turn to the left. This centers course deviation indicator on the display and centers the aircraft with the centerline of the runway. If the 90 Hz signal is received by the VOR receiver, the CDI deflects to the right. The pilot must turn toward the right to center the CDI and the aircraft with the runway center line. (*Figure 5-128*)



Figure 5-128. An ILS localizer antenna.

Glideslope

The vertical guidance required for an aircraft to descend for a landing is provided by the glideslope of the ILS. Radio signals funnel the aircraft down to the touchdown point on the runway at an angle of approximately 3° . The transmitting glideslope antenna is located off to the side of the approach runway approximately 1 000 feet from the threshold. It transmits in a wedge-like pattern with the field narrowing as it approaches the runway. (*Figure 5-129*)

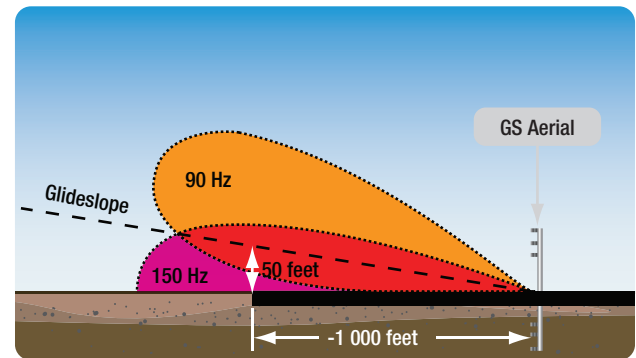


Figure 5-129. A glideslope antenna broadcasts radio signals to guide an aircraft vertically to the runway.

The glideslope transmitter antenna is horizontally polarized. The transmitting frequency range is UHF between 329.3 MHz and 335.0 MHz. The frequency is paired to the localizer frequency of the ILS.

When the VOR/ILS receiver is tuned for the approach, the glideslope receiver is automatically tuned. Like the localizer, the glideslope transmits two signals, one modulated at 90 Hz and the other modulated at 150 Hz. The aircraft's glideslope receiver deciphers the signals similar to the method of the localizer receiver. It drives a vertical course deviation indicator known as the glideslope indicator. The glideslope indicator operates identically to the localizer CDI only 90° to it. The VOR/ILS localizer CDI and the glideslope are displayed together on whichever kind of instrumentation is in the aircraft. (*Figure 5-130*)

The UHF antenna for aircraft reception of the glideslope signals comes in many forms. A single dipole antenna mounted inside the nose of the aircraft is a common option. Antenna manufacturers have also incorporated glideslope reception into the same dipole antenna used for the VHS VOR/ILS localizer reception. Blade type antennas are also used. (*Figure 5-131*)

Figure 5-132 shows a VOR and a glideslope receiver for a GA aircraft ILS.

Compass Locators

It is imperative that a pilot be able to intercept the ILS to enable its use. A compass locator is a transmitter designed for this purpose. There is typically one located at the outer marker beacon 4-7 miles from the runway threshold. Another may be located at the middle marker beacon about 3 500 feet from the threshold. The outer marker compass locator is a 25 watt NDB with a range of about 15 miles. It transmits omnidirectional LF radio waves (190 Hz to 535 Hz) keyed with the first two letters of the ILS identifier.



Figure 5-130. A traditional course deviation indicator is shown on the left. The horizontal white line is the deviation indicator for the glideslope. The vertical line is for the localizer. On the right, a Garmin G-1000 PFD illustrates an aircraft during an ILS approach. The narrow vertical scale on the right of the attitude indicator with the "G" at the top is the deviation scale for the glideslope. The green diamond moves up and down to reflect the aircraft being above or below the glidepath. The diamond is shown centered indicating the aircraft is on course vertically. The localizer CDI can be seen at the bottom center of the display. It is the center section of the vertical green course indicator. LOC1 is displayed to the left of it.

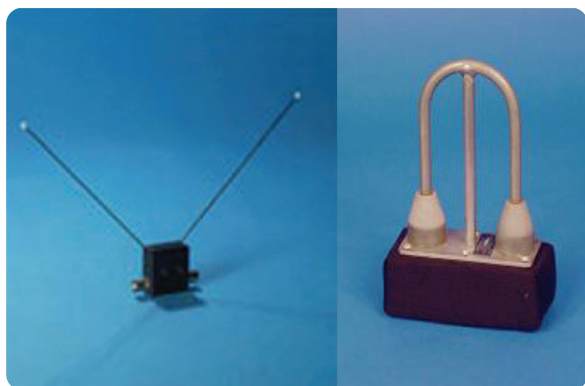


Figure 5-131. Glideslope antennas—designed to be mounted inside a non-metallic aircraft nose (left), and mounted inside or outside the aircraft (right).

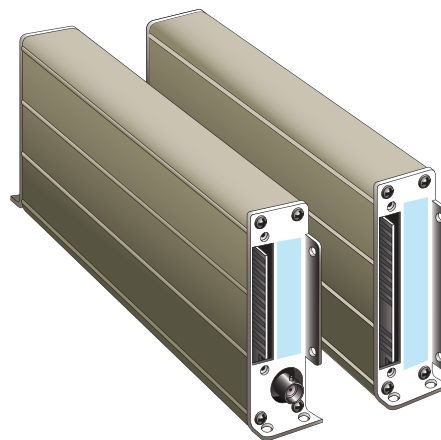


Figure 5-132. A localizer and glideslope receiver for a general aviation aircraft ILS.

The ADF receiver is used to intercept the locator so no additional equipment is required. If a middle marker compass locator is in place, it is similar but is identified with the last two letters of the ILS identifier. Once located, the pilot maneuvers the aircraft to fly down the glidepath to the runway. (*Figure 5-132*)

Marker Beacons

Marker beacons are the final radio transmitters used in the ILS. They transmit signals that indicate the position of the aircraft along the glidepath to the runway. As mentioned, an outer marker beacon transmitter is located 4-7 miles from the threshold. It transmits a 75 MHz carrier wave modulated with a 400 Hz audio tone in a series of dashes. The transmission is very narrow

and directed straight up. A marker beacon receiver receives the signal and uses it to light a blue light on the instrument panel. This, plus the oral tone in combination with the localizer and the glideslope indicator, positively locates the aircraft on an approach. (*Figure 5-133*)

A middle marker beacon is also used. It is located on approach approximately 3 500 feet from the runway. It also transmits at 75 MHz. The middle marker transmission is modulated with a 1 300 Hz tone that is a series of dots and dashes so as to not be confused with the all dash tone of the outer marker. When the signal is received, it is used in the receiver to illuminate an amber colored light on the instrument panel. (*Figure 5-134*)

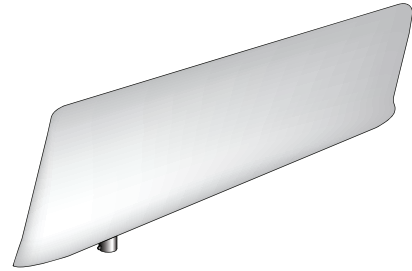
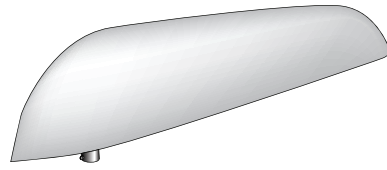
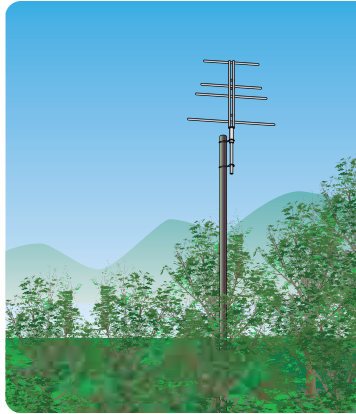


Figure 5-133. An outer marker transmitter antenna 4-7 miles from the approach runway transmits a 75 MHz signal straight up (left). Aircraft mounted marker beacon receiver antennas are shown (center and right).



Figure 5-134. Various marker beacon instrument panel display lights.



Figure 5-135. An ILS test unit.

Some ILS approaches have an inner marker beacon that transmits a signal modulated with 3 000 Hz in a series of dots only. It is placed at the land-or-go-around decision point of the approach close to the runway threshold. If present, the signal when received is used to illuminate a white light on the instrument panel. The three marker beacon lights are usually incorporated into the audio panel of a general aviation aircraft or may exist independently on a larger aircraft. Electronic display aircraft usually incorporate marker lights or indicators close to the glideslope display near attitude director indicator.

ILS radio components can be tested with an ILS test unit. Localizer, glideslope, and marker beacon signals are generated to ensure proper operation of receivers and correct display on flight deck instruments. (Figure 5-135)

DISTANCE MEASURING EQUIPMENT (DME)

Many VOR stations are co-located with the military version of the VOR station, which is known as TACAN. When this occurs, the navigation station is known as a VORTAC station. Civilian aircraft make use of one of the TACAN features not originally installed at civilian VOR stations - distance measuring equipment (DME). A DME system calculates the distance from the aircraft to the DME unit at the VORTAC ground station and displays it on the flight deck. It can also display calculated aircraft speed and elapsed time for arrival when the aircraft is traveling to the station.

DME ground stations have subsequently been installed at civilian VORs, as well as in conjunction with ILS localizers. These are known as VOR/DME and ILS/DME or LOC/DME. The latter aid in approach to the runway during landings. The DME system consists of an airborne DME transceiver, display, and antenna, as well as the ground based DME unit and its antenna. (Figure 5-136)



Figure 5-136. A VOR with DME ground station.

The DME is useful because with the bearing (from the VOR) and the distance to a known point (the DME antenna at the VOR), a pilot can positively identify the location of the aircraft. DME operates in the UHF frequency range from 962 MHz to 1 213 MHz. A carrier signal transmitted from the aircraft is modulated with a string of integration pulses. The ground unit receives the pulses and returns a signal to the aircraft. The time that transpires for the signal to be sent and returned is calculated and converted into nautical miles for display. Time to station and speed are also calculated and displayed. DME readout can be on a dedicated DME display or it can be part of an EHSI, EADI, EFIS, or on the primary flight display in a glass cockpit. (Figure 5-137)

The DME frequency is paired to the co-located VOR or VORTAC frequency. When the correct frequency is tuned for the VOR signal, the DME is tuned automatically. Tones are broadcast for the VOR station identification and then for the DME. The hold selector on a DME panel keeps the DME tuned in while the VOR selector is tuned to a different VOR. In most cases, the UHF of the DME is transmitted and received via a small blade type antenna mounted to the underside of the fuselage centerline. (Figure 5-138)

A traditional DME displays the distance from the DME transmitter antenna to the aircraft. This is called the slant distance. It is very accurate. However, since the aircraft is at altitude, the distance to the DME ground antenna from a point directly beneath the aircraft is shorter. Some modern DMEs are equipped to calculate this ground distance and display it. (Figure 5-139)



Figure 5-137. Distance information from the DME can be displayed on a dedicated DME instrument or integrated into any of the electronic navigational displays found on modern aircraft. A dual display DME is shown with its remote mounted receiver.



Figure 5-138. A typical aircraft mounted DME antenna.

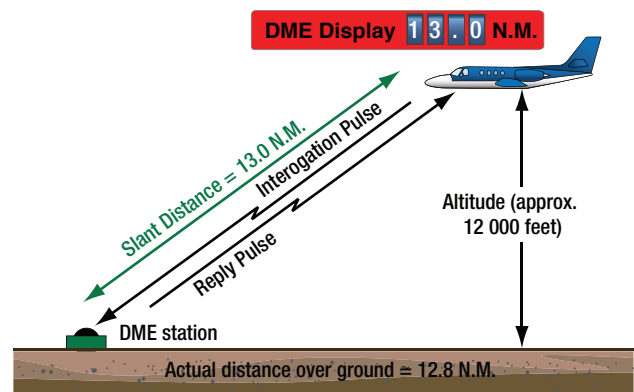


Figure 5-139. Many DMEs only display the slant distance, which is the actual distance from the aircraft to the DME station. This is different than the ground distance due to the aircraft being at altitude. Some DMEs compute the ground distance for display.

Area Navigation (RNAV)

Area navigation (RNAV) is a general term used to describe the navigation from point A to point B without direct over flight of navigational aids, such as VOR stations or ADF nondirectional beacons. It includes VORTAC and VOR/DME based systems, as well as systems of RNAV based around LORAN, GPS, INS, and the FMS of transport category aircraft. However, until recently, the term RNAV was most commonly used to describe the area navigation or the process of direct flight from point A to point B using VORTAC and VOR/DME based references which are discussed in this section.

All RNAV systems make use of waypoints. A waypoint is a designated geographical location or point used for route definition or progress-reporting purposes. It can be defined or described by using latitude/longitude grid coordinates or, in the case of VOR based RNAV, described as a point on a VOR radial followed by that point's distance from the VOR station (i.e., 200/25 means a point 25 nautical miles from the VOR station on the 200° radial).

Figure 5-140 illustrates an RNAV route of flight from airport A to airport B. The VOR/DME and VORTAC stations shown are used to create phantom waypoints

that are overflown rather than the actual stations. This allows a more direct route to be taken. The phantom waypoints are entered into the RNAV course-line computer (CLC) as a radial and distance number pair. The computer creates the waypoints and causes the aircraft's CDI to operate as though they are actual VOR stations. A mode switch allows the choice between standard VOR navigation and RNAV.

VOR based RNAV uses the VOR receiver, antenna, and VOR display equipment, such as the CDI. The computer in the RNAV unit uses basic geometry and trigonometry calculations to produce heading, speed, and time readouts for each waypoint. VOR stations need to be within line-of sight and operational range from the aircraft for RNAV use. (**Figure 5-141**)

RNAV has increased in flexibility with the development of GPS. Integration of GPS data into a planned VOR RNAV flight plan is possible as is GPS route planning without the use of any VOR stations.

RADAR BEACON TRANSPONDER

A radar beacon transponder, or simply, a transponder, provides positive identification and location of an aircraft on the radar screens of ATC. For each aircraft equipped with an altitude encoder, the transponder

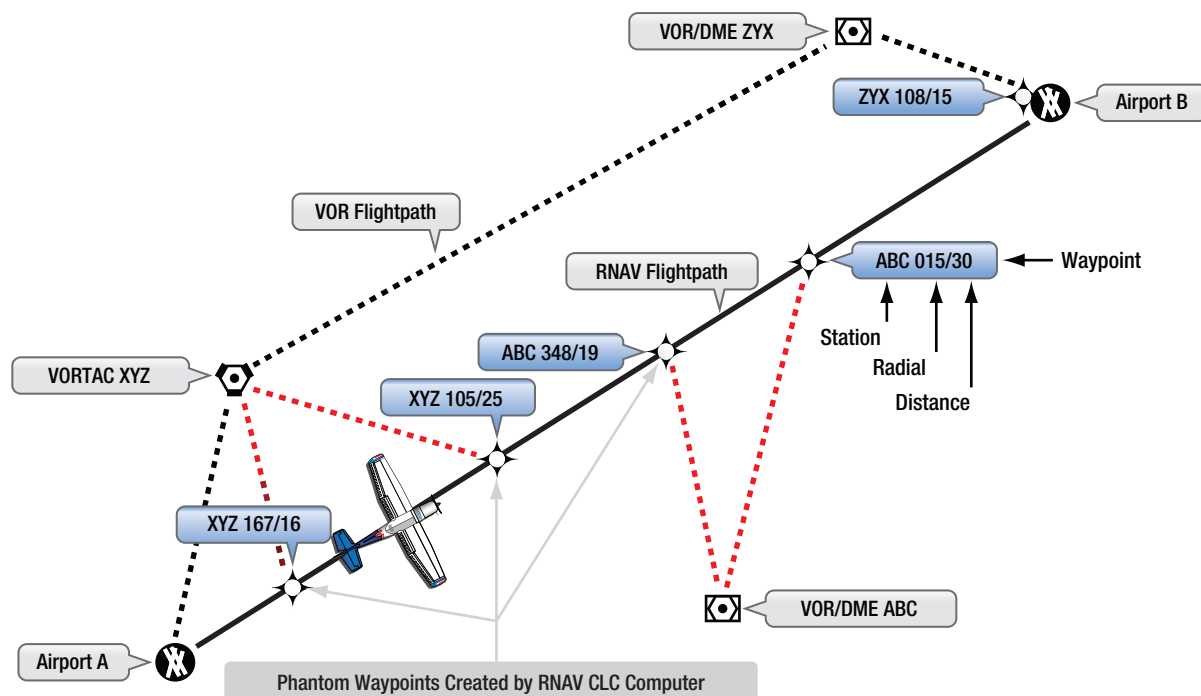


Figure 5-140. The pilot uses the aircraft's course deviation indicator to fly to and from RNAV phantom waypoints created by computer. This allows direct routes to be created and flown rather than flying from VOR to VOR.

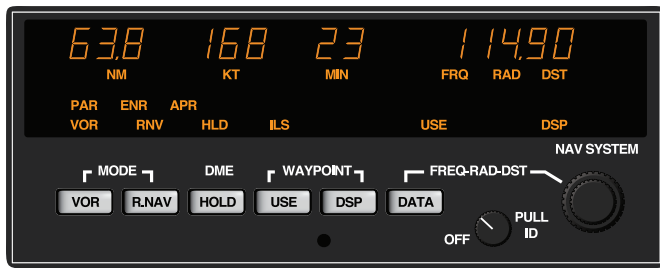


Figure 5-141. RNAV unit from a general aviation aircraft.

also provides the pressure altitude of the aircraft to be displayed adjacent to the on screen blip that represents the aircraft. (*Figure 5-142*)

Radar capabilities at airports vary. Generally, two types of radar are used by air traffic control (ATC). The primary radar transmits directional UHF or SHF radio waves sequentially in all directions. When the radio waves encounter an aircraft, part of those waves reflect back to a ground antenna. Calculations are made in a receiver to determine the direction and distance of the aircraft from the transmitter.

A blip or target representing the aircraft is displayed on a radar screen also known as a plan position indicator (PPI). The azimuth direction and scaled distance from the tower are presented giving controllers a two dimensional fix on the aircraft. (*Figure 5-143*)

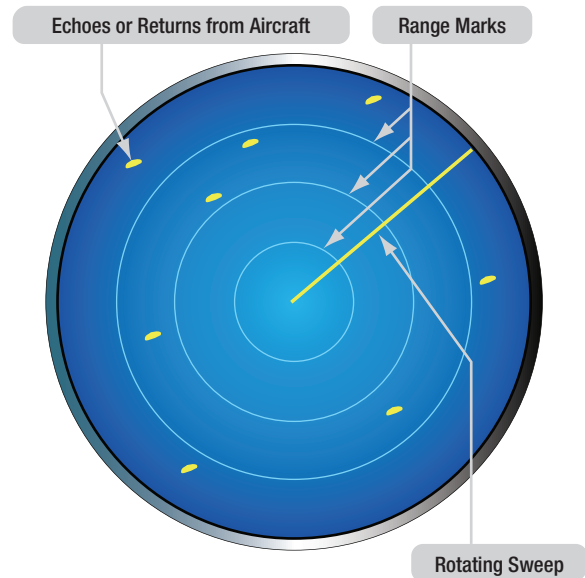


Figure 5-143. A plan position indicator (PPI) for ATC primary radar locates target aircraft on a scaled field.

A secondary surveillance radar (SSR) is used by ATC to verify the aircraft's position and to add the third dimension of altitude to its location. SSD radar transmits coded pulse trains that are received by the transponder on board the aircraft. Mode 3/A pulses, as they are known, aid in confirming the location of the aircraft.



Figure 5-142. A traditional transponder control head (A), a lightweight digital transponder (B), and a remote altitude encoder (C) that connects to a transponder to provide ATC with an aircraft's altitude displayed on a PPI radar screen next to the target that represents the aircraft.

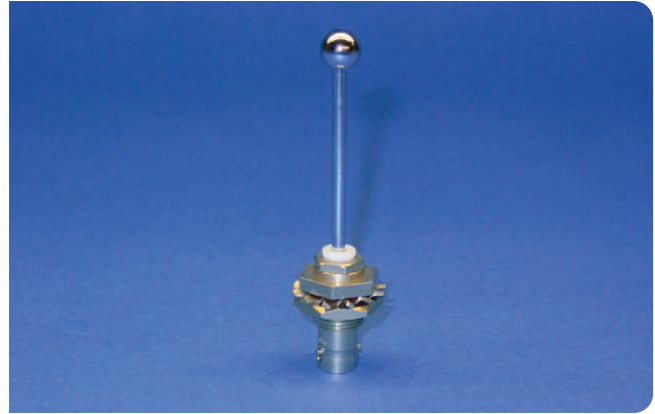


Figure 5-144. Aircraft radar beacon transponder antennas transmit and receive UHF and SHF radio waves.

When verbal communication is established with ATC, a pilot is instructed to select one of 4 096 discrete codes on the transponder. These are digital octal codes. The ground station transmits a pulse of energy at 1 030 MHz and the transponder transmits a reply with the assigned code attached at 1 090 MHz. This confirms the aircraft's location typically by altering its target symbol on the radar screen. As the screen may be filled with many confirmed aircraft, ATC can also ask the pilot to ident. By pressing the IDENT button on the transponder, it transmits in such a way that the aircraft's target symbol is highlighted on the PPI to be distinguishable.

To gain altitude clarification, the transponder control must be placed in the ALT or Mode C position. The signal transmitted back to ATC in response to pulse interrogation is then modified with a code that places the pressure altitude of the aircraft next to the target symbol on the radar screen. The transponder gets the pressure altitude of the aircraft from an altitude encoder that is electrically connected to the transponder. Typical aircraft transponder antennas are illustrated in *Figure 5-144*.

The ATC/aircraft transponder system described is known as Air Traffic Control Radar Beacon System (ATCRBS). To increase safety, Mode S altitude response has been developed. With Mode S, each aircraft is pre-assigned a unique identity code that displays along with its pressure altitude on ATC radar when the transponder responds to SSR interrogation. Since no other aircraft respond with this code, the chance of two pilots selecting the same response code on the transponder is eliminated.

A modern flight data processor computer (FDP) assigns the beacon code and searches flight plan data for useful information to be displayed on screen next to the target in a data block for each aircraft. (*Figure 5-145*)

Mode S is sometimes referred to as mode select. It is a data packet protocol that is also used in on board collision avoidance systems. When used by ATC, Mode S interrogates one aircraft at a time. Transponder workload is reduced by not having to respond to all interrogations in an airspace. Additionally, location information is more accurate with Mode S. A single reply in which the phase of the transponder reply is used to calculate position, called monopulse, is sufficient to locate the aircraft. Mode S also contains capacity for a wider variety of information exchange that is untapped potential for the future. At the same time, compatibility with older radar and transponder technology has been maintained.

Transponder Tests And Inspections

Because of the danger involved should a transponder malfunction and, for example, report the wrong altitude information, the functional condition of all transponders is of great concern to aviators. Relatively recent data suggests that such testing may not affect the number of transponder malfunctions. Widespread testing may be more of a problem in that, if not performed with strict adherence to manufacturer's testing guidelines, transponder radio signals may be transmitted into the atmosphere. Errant signal may cause aircraft to take evasive action or divert the attention of the flight crew from other critical matters.

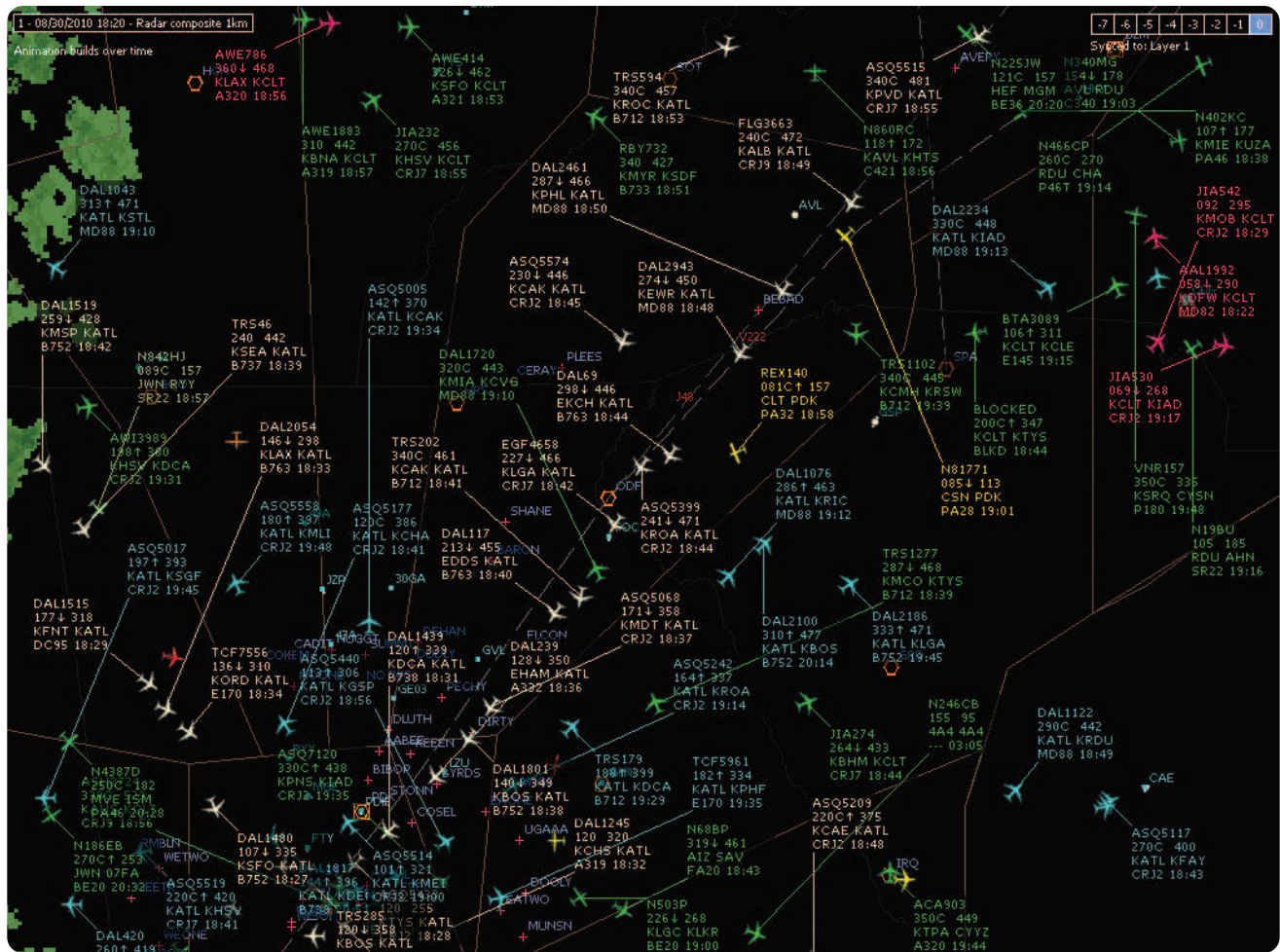


Figure 5-145. Air traffic control radar technology and an on board radar beacon transponder work together to convey and display air traffic information on a PPI radar screen. A modern approach ATC PPI is shown. Targets representing aircraft are shown as little aircraft on the screen. The nose of the aircraft indicates the direction of travel. Most targets shown above are airliners. The data block for each target includes the following information either transmitted by the transponder or matched and loaded from flight plans by a flight data processor computer: call sign, altitude/speed, origination/destination, and aircraft type/ETA (ZULU time). A "C" after the altitude indicates the information came from a Mode C equipped transponder. The absence of a C indicates Mode S is in use. An arrow up indicates the aircraft is climbing. An arrow down indicates a descent. White targets are arrivals, light blue targets are departures, all other colors are for arrivals and departures to different airports in the area.

Technician should follow the requirements for periodic testing of transponders issued by the NAA of the country of registration of the aircraft. They should also be sure to comply with any airworthiness directive of that country, EASA, and the country of aircraft manufacture. As with many radio-electronic devices, test equipment exists to test airworthy operation of a transponder. (Figure 5-146)

Operating a transponder in a hangar or on the ramp does not immunize it from interrogation and reply. Transmission of certain codes reserved for emergencies or military activity must be avoided. The procedure to select a code during ground operation is to do so with the transponder in the OFF or STANDBY mode to



Figure 5-146. A handheld transponder test unit.

avoid inadvertent transmission. Code 0000 is reserved for military use and is a transmittable code. Code 7500 is used in a hijack situation and 7600 and 7700 are also reserved for emergency use. Even the inadvertent transmission of code 1200 reserved for VFR flight not under ATC direction could result in evasion action. All signals received from a radar beacon transponder are taken seriously by ATC.

Altitude Encoders

Altitude encoders convert the aircraft's pressure altitude into a code sent by the transponder to ATC. Increments of 100 feet are usually reported. Encoders have varied over the years.

Some are built into the altimeter instrument used in the instrument panel and connected by wires to the transponder. Others are mounted out of sight on an avionics rack or similar out of the way place. These are known as blind encoders. On transport category aircraft, the altitude encoder may be a large black box with a static line connection to an internal aneroid. Modern general aviation encoders are smaller and more lightweight, but still often feature an internal aneroid and static line connection. Some encoders use microtransistors and are completely solid-state including the pressure sensing device from which the altitude is derived. No static port connection is required. Data exchange with GPS and other systems is becoming common. (*Figure 5-147*)

When a transponder selector is set on ALT, the digital pulse message sent in response to the secondary surveillance radar interrogation becomes the digital representation of the pressure altitude of the aircraft. There are 1 280 altitude codes, one for each 100 feet

of altitude between 1 200 feet mean sea level (MSL) and 126 700 feet MSL. Each altitude increment is assigned a code. While these would be 1 280 of the same codes used for location and IDENT, the Mode C (or S) interrogation deactivates the 4 096 location codes and causes the encoder to become active. The correct altitude code is sent to the transponder that replies to the interrogation. The SSR receiver recognized this as a response to a Mode C (or S) interrogation and interprets the code as altitude code.

COLLISION AVOIDANCE SYSTEMS

The ever increasing volume of air traffic has caused a corresponding increase in concern over collision avoidance. no longer adequate in today's increasingly crowded skies. On board collision avoidance equipment, long a staple in larger aircraft, is now common in general aviation aircraft. New applications of electronic technology combined with lower costs make this possible.

Traffic Collision Avoidance Systems (TCAS)

Traffic collision avoidance systems (TCAS) are transponder based air-to-air traffic monitoring and alerting systems. There are two classes of TCAS. TCAS I was developed to accommodate the general aviation community and regional airlines. This system identifies traffic in a 35-40 mile range of the aircraft and issues Traffic Advisories (TA) to assist pilots in visual acquisition of intruder aircraft. TCAS I is mandated on aircraft with 10 to 30 seats.

TCAS II is a more sophisticated system. It is required internationally in aircraft with more than 30 seats or weighing more than 15 000 kg. TCAS II provides the information of TCAS I, but also analyzes the projected flightpath of approaching aircraft. If a collision or near miss is imminent, the TCAS II computer issues a Resolution Advisory (RA). This is an aural command to the pilot to take a specific evasive action (i.e., DESCEND). The computer is programmed such that the pilot in the encroaching aircraft receives an RA for evasive action in the opposite direction (if it is TCAS II equipped). (*Figure 5-148*)

The transponder of an aircraft with TCAS is able to interrogate the transponders of other aircraft nearby using SSR technology (Mode C and Mode S). This is done with a 1030 MHz signal. Interrogated aircraft

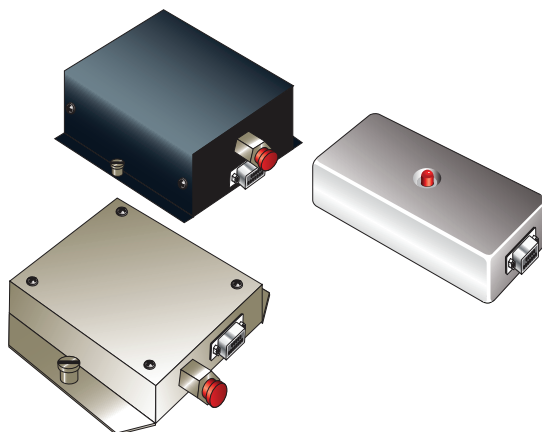


Figure 5-147. Modern altitude encoders for general aviation aircraft.

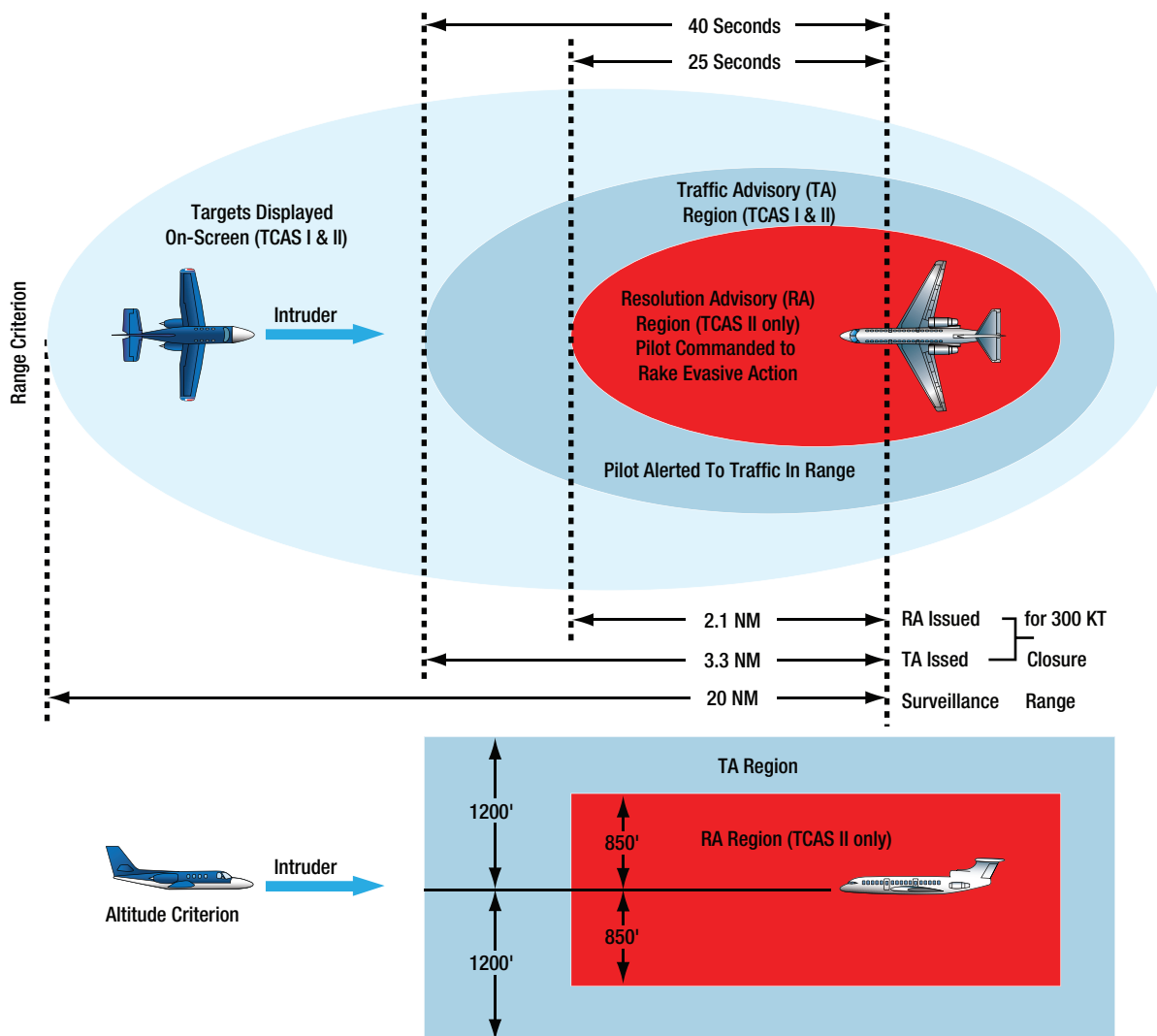


Figure 5-148. Traffic collision and avoidance system (TCAS) uses an aircraft's transponder to interrogate and receive replies from other aircraft in close proximity. The TCAS computer alerts the pilot as to the presence of an intruder aircraft and displays the aircraft on a screen in the cockpit. Additionally, TCAS II equipped aircraft receive evasive maneuver commands from the computer that calculates trajectories of the aircraft to predict potential collisions or near misses before they become unavoidable.

transponders reply with an encoded 1090 MHz signal that allows the TCAS computer to display the position and altitude of each aircraft. Should the aircraft come within the horizontal or vertical distances shown in **Figure 5-148**, an audible TA is announced. The pilot must decide whether to take action and what action to take. TCAS II equipped aircraft use continuous reply information to analyze the speed and trajectory of target aircraft in close proximity. If a collision is calculated to be imminent, an RA is issued.

TCAS target aircraft are displayed on a screen on the flight deck. Different colors and shapes are used to depict approaching aircraft depending on the imminent threat level. Since RAs are currently limited to vertical

evasive maneuvers, some stand-alone TCAS displays are electronic vertical speed indicators. Most aircraft use some version of an electronic HSI on a navigational screen or page to display TCAS information. (**Figure 5-149**)

A multifunction display may depict TCAS and weather radar information on the same screen. (**Figure 5-150**)

A TCAS control panel (**Figure 5-151**) and computer are required to work with a compatible transponder and its antenna(s). Interface with EFIS or other previously installed or selected display(s) is also required. TCAS may be referred to as airborne collision avoidance system (ACAS), which is the international name for the same system. TCAS II with the latest revisions is known as



Figure 5-149. TCAS information displayed on an electronic vertical speed indicator.



Figure 5-150. TCAS information displayed on a multifunction display. An open diamond indicates a target; a solid diamond represents a target that is within 6 nautical miles of 1 200 feet vertically. A yellow circle represents a target that generates a TA (25-48 seconds before contact). A red square indicates a target that generates an RA in TCAS II (contact within 35 seconds). A (+) indicates the target aircraft is above and a (-) indicates it is below. The arrows show if the target is climbing or descending.



Figure 5-151. This control panel from a Boeing 767 controls the transponder for ATC use and TCAS.

Version 7. The accuracy and reliability of this TCAS information is such that pilots are required to follow a TCAS RA over an ATC command.

ADS-B

Collision avoidance is a significant part of the FAA's NextGen plan for transforming the U.S. National Airspace System (NAS). Increasing the number of aircraft using the same quantity of airspace and ground facilities requires the implementation of new technologies to maintain a high level of performance and safety.

The successful proliferation of global navigation satellite systems (GNSS), such as GPS, has led to the development of a collision avoidance system known as automatic dependent surveillance broadcast (ADS-B). ADS-B is an integral part of NextGen program. The implementation of its ground and airborne infrastructure is currently underway. ADS-B is active in parts of the United States and around the world. (*Figure 5-152*)

ADS-B is considered in two segments: ADS-B OUT and ADS-B IN. ADS-B OUT combines the positioning information available from a GPS receiver with on-board flight status information, i.e. location including altitude, velocity, and time. It then broadcasts this information to other ADS-B equipped aircraft and ground stations. (*Figure 5-153*)



Figure 5-152. Low power requirements allow remote ADS-B stations with only solar or propane support. This is not possible with ground radar due to high power demands which inhibit remote area radar coverage for air traffic purposes.

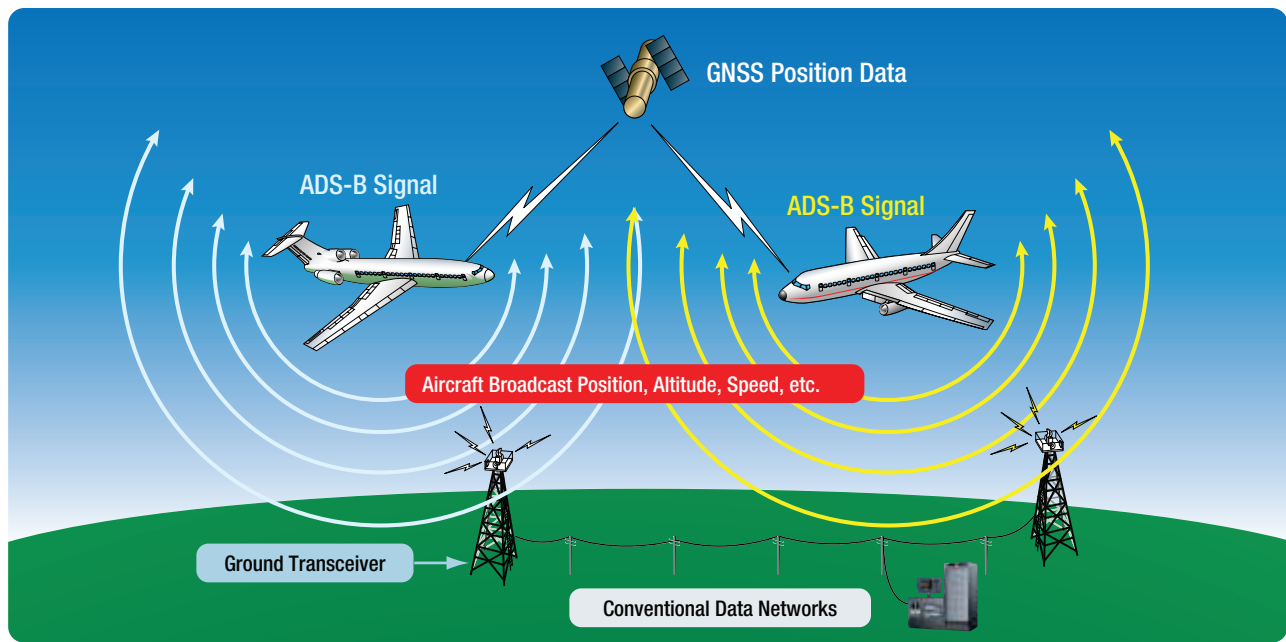


Figure 5-153. ADS-B OUT uses satellites to identify the position aircraft. This position is then broadcast to other aircraft and to ground stations along with other flight status information.

Two different frequencies are used to carry these broadcasts with data link capability. The first is an expanded use of the 1 090 MHz Mode-S transponder protocol known as 1 090 ES. The second, largely being introduced as a new broadband solution for general aviation implementation of ADS-B, is at 978 MHz. A 978 universal access transceiver (UAT) is used to accomplish this. An omni-directional antenna is required in addition to the GPS antenna and receiver. Airborne receivers of an ADS-B broadcast use the information to plot the location and movement of the transmitting aircraft on a flight deck display similar to TCAS. (*Figure 5-154*)

Inexpensive ground stations (compared to radar) are constructed in remote and obstructed areas to proliferate ADS-B. Ground stations share information from airborne ADS-B broadcasts with other ground stations that are part of the air traffic management system (ATMS). Data is transferred with no need for human acknowledgement. Microwave and satellite transmissions are used to link the network.

For traffic separation and control, ADS-B has several advantages over conventional ground-based radar. The first is the entire airspace can be covered with a much lower expense. The aging ATC radar system that is in place is expensive to maintain and replace. Additionally, ADS-B provides more accurate information since the

vector state is generated from the aircraft with the help of GPS satellites. Weather is a greatly reduced factor with ADS-B. Ultra high frequency GPS transmissions are not affected. Increased positioning accuracy allows for higher density traffic flow and landing approaches, an obvious requirement to operate more aircraft in and out of the same number of facilities.

The higher degree of control available also enables routing for fewer weather delays and optimal fuel burn rates. Collision avoidance is expanded to include runway incursion from other aircraft and support vehicles on the surface of an airport. ADS-B IN offers features not available in TCAS. Equipped aircraft are able to receive abundant data to enhance situational awareness.

Traffic information services-broadcast (TIS-B) supply traffic information from non-ADS-B aircraft and ADS-B aircraft on a different frequency. Ground radar monitoring of surface targets, and any traffic data in the linked network of ground stations is sent via ADS-B IN to the flight deck. This provides a more complete picture than air-to-air only collision avoidance. Flight information services-broadcast (FIS-B) are also received by ADS-B IN. Weather text and graphics, ATIS information, and NOTAMS (notices to airmen) are able to be received in aircraft that have 978 UAT capability. (*Figure 5-155*)



Figure 5-154. A cockpit display of ADS-B generated targets (left) and an ADS-B airborne receiver with antenna (right).

ADS-B test units are available for trained maintenance personnel to verify proper operation of ADS-B equipment. This is critical since close tolerance of air traffic separation depends on accurate data from each aircraft and throughout all components of the ADS-B system. (*Figure 5-156*)

RADIO ALTIMETER

A radio altimeter, or radar altimeter, is used to measure the distance from the aircraft to the terrain directly beneath it. It is used primarily during instrument approach and low level or night flight below 2500 feet. The radio altimeter supplies the primary altitude information for landing decision height. It incorporates an adjustable altitude bug that creates a visual or aural warning to the pilot when the aircraft reaches that altitude. Typically, the pilot will abort a landing if the decision height is reached and the runway is not visible.

Using a transceiver and a directional antenna, a radio altimeter broadcasts a carrier wave at 4.3 GHz from the aircraft directly toward the ground. The wave is frequency modulated at 50 MHz and travels at a known speed. It strikes surface features and bounces back toward the aircraft where a second antenna receives the return signal. The transceiver processes the signal by measuring the elapsed time the signal traveled and the frequency modulation that occurred. The display indicates height above the terrain also known as above ground level (AGL). (*Figure 5-157*)

A radar altimeter is more accurate and responsive than an air pressure altimeter for AGL information at low altitudes. The transceiver is usually located remotely from the indicator. Multifunctional and glass cockpit

displays typically integrate decision height awareness from the radar altimeter as a digital number displayed on the screen with a bug, light, or color change used to indicate when that altitude is reached.

Large aircraft may incorporate radio altimeter information into a ground proximity warning system (GPWS) which aurally alerts the crew of potentially dangerous proximity to the terrain below the aircraft. A decision height window (DH) displays the radar altitude on the EADI in *Figure 5-158*.

WEATHER RADAR

There are three common types of weather aids used in an aircraft flight deck that are often referred to as weather radar:

1. Actual on-board radar for detecting and displaying weather activity.
2. Lightning detectors.
3. Satellite or other source weather radar information that is uploaded to aircraft from an outside source.

On-board weather radar systems can be found in aircraft of all sizes. They function similar to ATC primary radar except the radio waves bounce off of precipitation instead of aircraft. Dense precipitation creates a stronger return than light precipitation. The on-board weather radar receiver is set up to depict heavy returns as red, medium return as yellow and light returns as green on a display in the flight deck. Clouds do not create a return. Magenta is reserved to depict intense or extreme precipitation or turbulence. Some aircraft have a dedicated weather radar screen. Most modern aircraft integrate weather radar display into the navigation display(s).

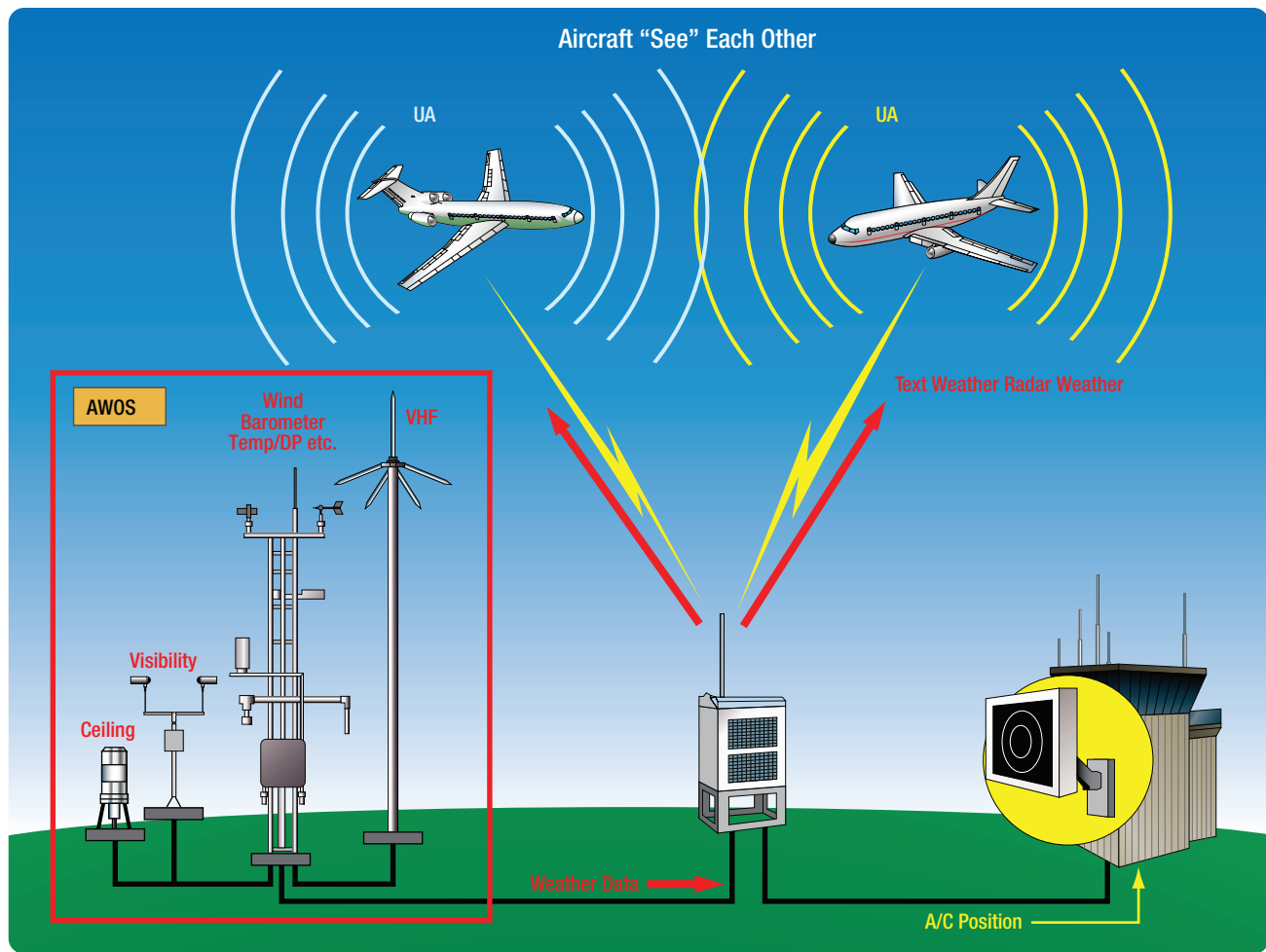


Figure 5-155. ADS-B IN enables weather and traffic information to be sent into the flight deck. In addition to AWOS weather, NWS can also be transmitted.



Figure 5-156. An ADS-B test unit.

Figure 5-159 illustrates weather radar displays found on aircraft. Radio waves used in weather radar systems are in the SHF range such as 5.44 GHz or 9.375 GHz. They are transmitted forward of the aircraft from a directional antenna usually located behind a nonmetallic nose cone. Pulses of approximately one microsecond in length are transmitted. A duplexer in the radar transceiver switches the antenna to receive for about 2500 micro seconds after a pulse is transmitted to receive and process any



Figure 5-157. A digital display radio altimeter (top), and the two antennas and transceiver for a radio/radar altimeter (bottom).

returns. This cycle repeats and the receiver circuitry builds a two dimensional image of precipitation for display. Gain adjustments control the range of the radar. A control panel facilitates this and other adjustments. (**Figure 5-160**)

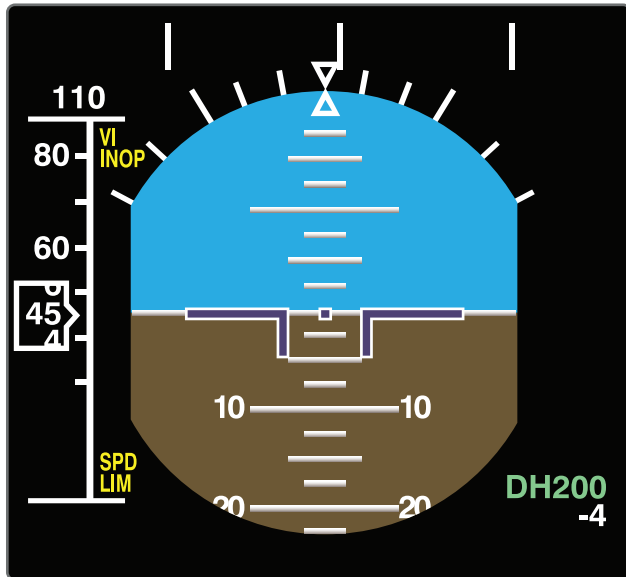


Figure 5-158. The decision height, DH200, in the lower right corner of this EADI display uses the radar altimeter as the source of altitude information.

Severe turbulence, wind shear, and hail are of major concern to the pilot. While hail provides a return on weather radar, wind shear and turbulence must be interpreted from the movement of any precipitation that is detected. An alert is annunciated if this condition occurs on a weather radar system so equipped. Dry air turbulence is not detectable. Ground clutter must also be attenuated when the radar sweep includes any terrain features. The control panel facilitates this. Special precautions must be followed by the technician during maintenance and operation of weather radar systems.

The radome covering the antenna must only be painted with approved paint to allow the radio signals to pass unobstructed. Many radomes also contain grounding strips to conduct lightning strikes and static away from the dome. When operating the radar, it is important to follow all manufacturer instructions. Physical harm is possible from the high energy radiation emitted, especially to the eyes and testes. Do not look into the antenna of a transmitting radar.

Operation of the radar should not occur in hangars unless special radio wave absorption material is used. Additionally, operation of radar should not take place while the radar is pointed toward a building or when refueling takes place. Radar units should be maintained and operated only by qualified personnel.

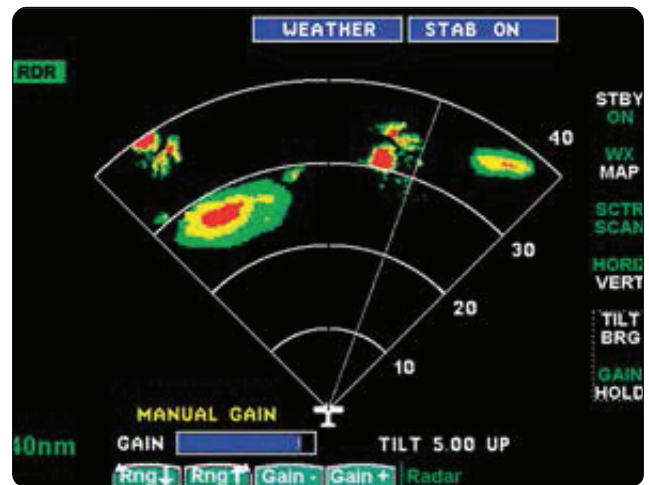


Figure 5-159. A dedicated weather radar display (top) and a multifunctional navigation display with weather radar overlay (bottom).

Lightning detection is a second reliable means for identifying potentially dangerous weather. Lightning gives off its own electromagnetic signal. The azimuth of a lightning strike can be calculated by a receiver using a loop type antenna such as that used in ADF. (Figure 5-161) Some lightning detectors make use of the ADF antenna. The range of the lightning strike is closely associated with its intensity. Intense strikes are plotted as being close to the aircraft.

Stormscope is a proprietary name often associated with lightning detectors. There are others that work in a similar manner. A dedicated display plots the location of each strike within a 200 mile range with a small mark on the screen. As time progresses, the marks may change color to indicate their age. Nonetheless, a number of lightning strikes in a small area indicates a storm cell, and the pilot can navigate around it. Lightning strikes can also be plotted on a multifunctional navigation display. (Figure 5-162)



Figure 5-160. A typical on-board weather radar system for a high performance aircraft uses a nose-mounted antenna that gimbals. It is usually controlled by the inertial reference system (IRS) to automatically adjust for attitude changes during maneuvers so that the radar remains aimed at the desired weather target. The pilot may also adjust the angle and sweep manually as well as the gain. A dual mode control panel allows separate control and display on the left or right HSI or navigational display.



Figure 5-161. A receiver and antenna from a lightning detector system.

A third type of weather radar is becoming more common in all classes of aircraft. Through the use of orbiting satellite systems and/or ground up-links, such as described with ADS-B IN, weather information can be sent to an aircraft in flight virtually anywhere in the world. This includes text data as well as real-time radar information for overlay on an aircraft's navigational display(s). Weather radar data produced remotely and sent to the aircraft is refined through consolidation of various radar views from different angles and satellite imagery. This produces more accurate depictions of actual weather conditions. Terrain databases are integrated to eliminate ground clutter. Supplemental data includes the entire range of intelligence available from the National Weather Service (NWS) and the National Oceanographic and Atmospheric Administration (NOAA).



Figure 5-162. A dedicated stormscope lightning detector display (left), and an electronic navigational display with lightning strikes overlaid in the form of green "plus" signs (right).

Figure 5-163 illustrates a plain language weather summary received in an aircraft along with a list of other weather information available through satellite or ground link weather information services. As mentioned, to receive an ADS-B weather signal, a 1090 ES or 970 UAT transceiver with associated antenna needs to be installed on board the aircraft. Satellite weather services are received by an antenna matched to the frequency of the service. Receivers are typically located remotely and interfaced with existing navigational and multifunction displays. Handheld GPS units also may have satellite weather capability. (**Figure 5-164**)

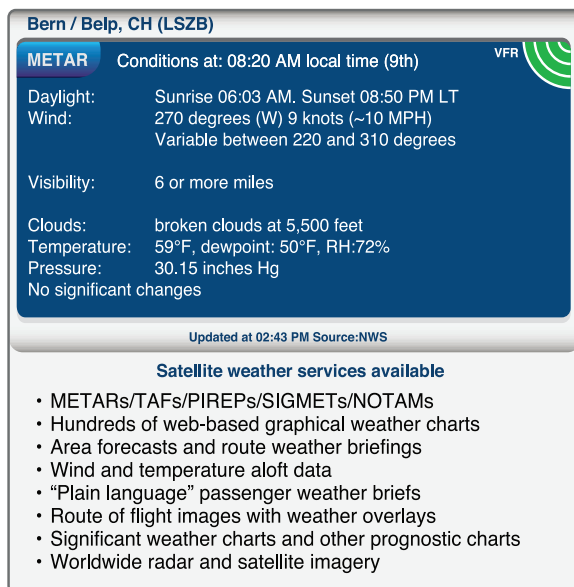


Figure 5-163. A plain language METAR weather report received in the cockpit from a satellite weather service for aircraft followed by a list of various weather data that can be radioed to the cockpit from a satellite weather service.

EMERGENCY LOCATOR TRANSMITTER (ELT)

An emergency locator transmitter (ELT) is an independent battery powered transmitter activated by the excessive G-forces experienced during a crash. It transmits a digital signal every 50 seconds on a frequency of 406.025 MHz at 5 watts for at least 24 hours.

The signal is received anywhere in the world by satellites in the COSPAS-SARSAT satellite system. Two types of satellites, low earth orbiting (LEOSATs) and geostationary satellites (GEOSATs) are used with different, complimentary capability. The signal is partially processed and stored in the satellites and then relayed to ground stations known as local user terminals (LUTs). Further deciphering of a signal takes place at the LUTs, and appropriate search and rescue operations are notified through mission control centers (MCCs) set up for this purpose. NOTE: Maritime vessel emergency locating beacons (EPIRBs) and personal locator beacons (PLBs) use the exact same system. **Figure 5-165** illustrates the basic components in the COSPAS-SARSAT system.

ELTs are required to be installed in aircraft according to FAR 91.207. This encompasses most general aviation aircraft not operating under Parts 135 or 121. ELTs must be inspected within 12 months of previous inspection for proper installation, battery corrosion, operation of the controls and crash sensor, and the presence of a sufficient signal at the antenna. Built-in test equipment facilitates testing without transmission of an emergency signal. The remainder of the inspection is visual. Technicians are cautioned to not activate the ELT and transmit an



Figure 5-164. A satellite weather receiver and antenna enable display of real-time textual and graphic weather information beyond that of airborne weather radar. A handheld GPS can also be equipped with these capabilities. A built-in multifunctional display with satellite weather overlays and navigation information can be found on many aircraft.

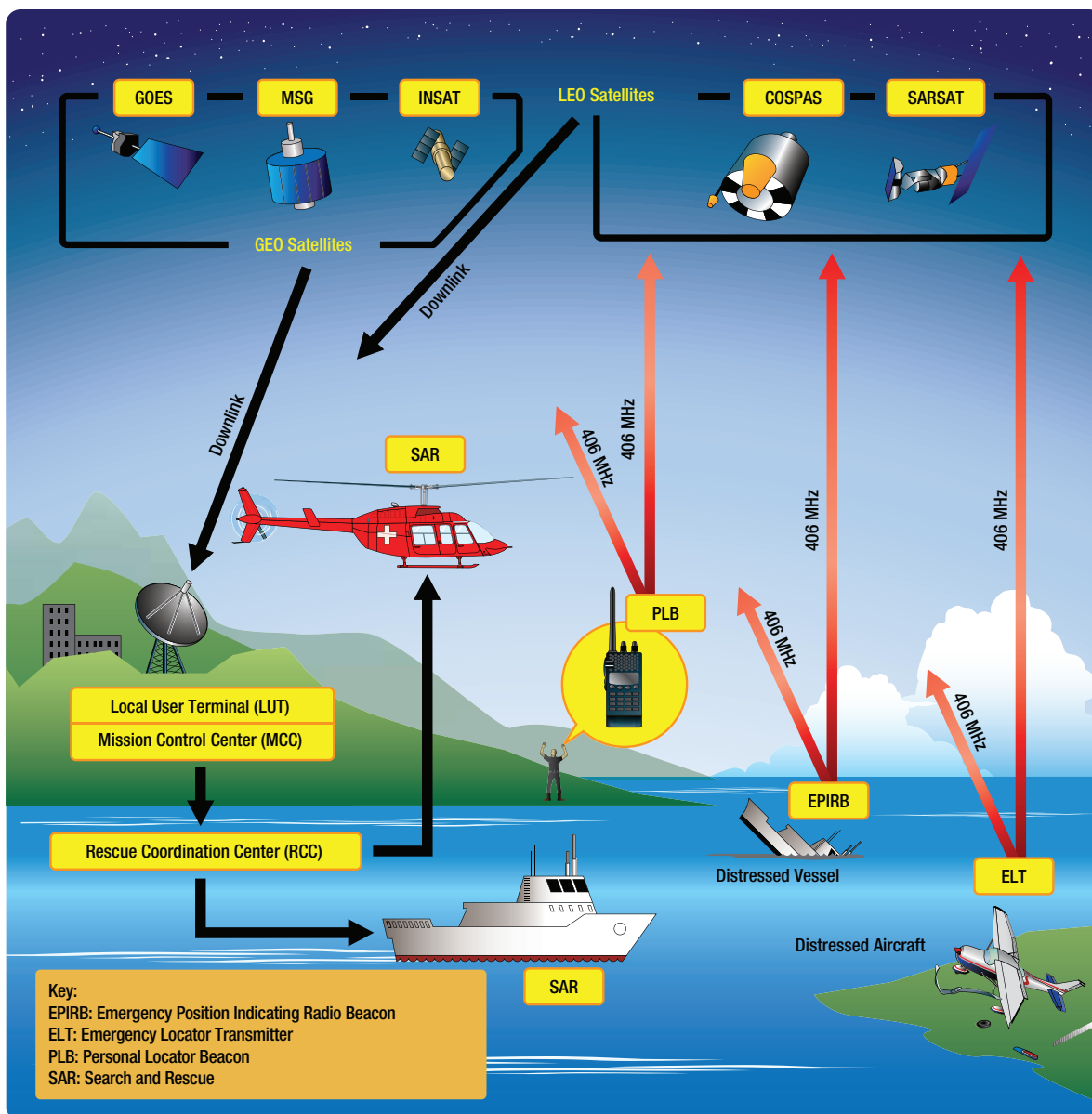


Figure 5-165. The basic operating components of the satellite-based COSPAS-SARSAT rescue system of which aircraft ELTs are a part.

emergency distress signal. Inspection must be recorded in maintenance records including the new expiration date of the battery. This must also be recorded on the outside of the ELT.

ELTs are typically installed as far aft in the fuselage of an aircraft as is practicable just forward of the empennage. The built-in G-force sensor is aligned with the longitudinal axis of the aircraft. Helicopter ELTs may be located elsewhere on the airframe. They are equipped with multidirectional activation devices. Follow ELT and airframe manufacturer's instructions for proper installation, inspection, and maintenance of all ELTs. *Figure 5-166* illustrates ELTs mounted locations.

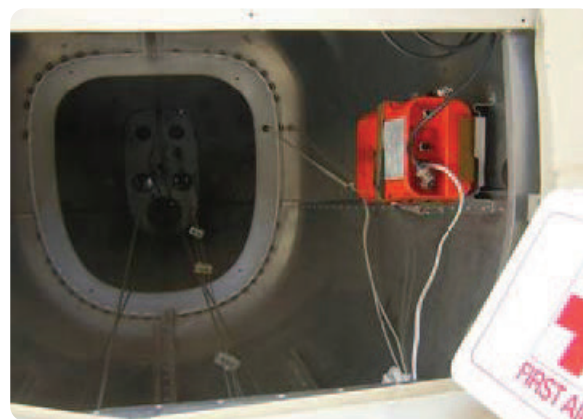


Figure 5-166. An emergency locator transmitter (ELT) mounting location is generally far aft in a fixed-wing aircraft fuselage in line with the longitudinal axis. Helicopter mounting location and orientation varies.

Use of Doppler technology enables the origin of the 406 MHz ELT signal to be calculated within 2 to 5 kilometers. Second generation 406 MHz ELT digital signals are loaded with GPS location coordinates from a receiver inside the ELT unit or integrated from an outside unit. This reduces the location accuracy of the crash site to within 100 meters. The digital signal is also loaded with unique registration information. It identifies the aircraft, the owner, and contact information, etc. When a signal is received, this is used to immediately research the validity of the alert to ensure it is a true emergency transmission so that rescue resources are not deployed needlessly. ELTs with automatic G-force activation mounted in aircraft are easily removable. They often contain a portable antenna so that crash victims may leave the site and carry the operating ELT with them. A flight deck mounted panel is required to alert the pilot if the ELT is activated. It also allows the ELT to be armed, tested, and manually activated if needed.

(Figure 5-167)

Modern ELTs may also transmit a signal on 121.5 MHz. This is an analog transmission that can be used for homing. Prior to 2009, 121.5 MHz was a worldwide emergency frequency monitored by the CORPAS-SARSAT satellites. However, it has been replaced by the 406 MHz standard. Transmission on 121.5 MHz are no longer received and relayed via satellite.



Figure 5-167. An ELT and its components including a cockpit-mounted panel, the ELT, a permanent mount antenna, and a portable antenna.

The 121.5 MHz frequency is still an active emergency frequency and is monitored by over flying aircraft and control towers. Technicians are required to perform an inspection/test of 121.5 MHz ELTs within 12 months of the previous one and inspect for the same integrity as required for the 406MHz ELTs mentioned above. However, older ELTs often lack the built-in test circuitry of modern ELTs certified to TSO C-126. Therefore, a true operational test may include activating the signal. This can be done by removing the antenna and installing a dummy load.

Any activation of an ELT signal is required to only be done between the top of each hour and 5 minutes after the hour. The duration of activation must be no longer than three audible sweeps. Contact of the local control tower or flight service station before testing is recommended.

It must be noted that older 121.5 MHz analog signal ELTs often also transmit an emergency signal on a frequency of 243.0 MHz. This has long been the military emergency frequency. Its use is being phased out in favor of digital ELT signals and satellite monitoring. Improvements in coverage, location accuracy, identification of false alerts, and shortened response times are so significant with 406 MHz ELTs, they are currently the service standard worldwide.

GLOBAL POSITIONING SYSTEM (GPS)

Global positioning system navigation (GPS) is the fastest growing type of navigation in aviation. It is accomplished through the use of NAVSTAR satellites set and maintained in orbit around the earth by the U.S. Government. Continuous coded transmissions from the satellites facilitate locating the position of an aircraft equipped with a GPS receiver with extreme accuracy. GPS can be utilized on its own for en route navigation, or it can be integrated into other navigation systems, such as VOR/RNAV, inertial reference, or flight management systems.

There are three segments of GPS: the space segment, the control segment, and the user segment. Aircraft technicians are only involved with user segment equipment such as GPS receivers, displays, and antennas. Twenty-four satellites (21 active, 3 spares) in six separate plains of orbit 12 625 feet above the planet comprise what is known as the space segment of the

GPS system. The satellites are positioned such that in any place on earth at any one time, at least four will be a minimum of 15° above the horizon. Typically, between 5 and 8 satellites are in view. (*Figure 5-168*)

Two signals loaded with digitally coded information are transmitted from each satellite. The L1 channel transmission on a 1 575.42 MHz carrier frequency is used in civilian aviation. Satellite identification, position, and time are conveyed to the aircraft GPS receiver on this digitally modulated signal along with status and other information. An L2 channel 1 227.60 MHz transmission is used by the military. The amount of time it takes for signals to reach the aircraft GPS receiver from transmitting satellites is combined with each satellite's exact location to calculate the position of an aircraft. The control segment of the GPS monitors each satellite to ensure its location and time are precise. This control is accomplished with five ground-based receiving stations, a master control station, and three transmitting antenna. The receiving stations forward status information received from the satellites to the master control station. Calculations are made and corrective instructions are sent to the satellites via the transmitters.

The user segment of the GPS is comprised of the thousands of receivers installed in aircraft as well as every other receiver that uses the GPS transmissions. Specifically, for the aircraft technician, the user section consists of a control panel/display, the GPS receiver circuitry, and an antenna. The control, display and receiver are usually located in a single unit which also may include VOR/ILS circuitry and a VHF communications transceiver. GPS intelligence is integrated into the multifunctional displays of glass cockpit aircraft. (*Figure 5-169*)

The GPS receiver measures the time it takes for a signal to arrive from three transmitting satellites. Since radio waves travel at 186 000 miles per second, the distance to each satellite can be calculated. The intersection of these ranges provides a two dimensional position of the aircraft. It is expressed in latitude/longitude coordinates. By incorporating the distance to a fourth satellite, the altitude above the surface of the earth can be calculated as well. This results in a three dimensional fix. Additional satellite inputs refine the accuracy of the position.

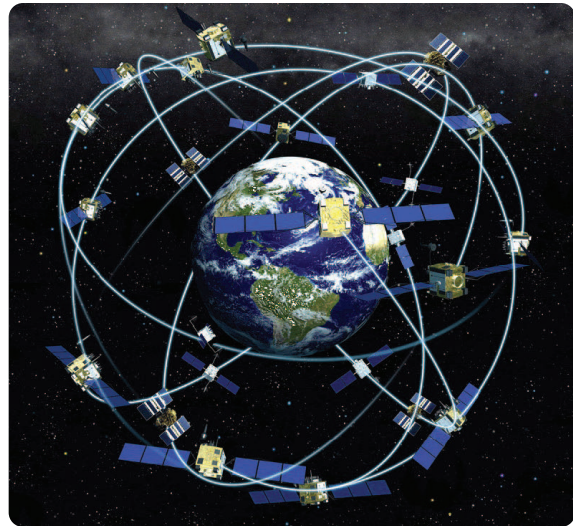


Figure 5-168. The space segment of GPS consists of 24 NAVSTAR satellites in six different orbits around the earth.



Figure 5-169. A GPS unit integrated with NAV/COM circuitry.

Having deciphered the position of the aircraft, the GPS unit processes many useful navigational outputs such as speed, direction, bearing to a waypoint, distance traveled, time of arrival, and more. These can be selected to display for use. Waypoints can be entered and stored in the unit's memory. Terrain features, airport data, VOR/RNAV and approach information, communication frequencies, and more can also be loaded into a GPS unit. Most modern units come with moving map display capability. A main benefit of GPS use is immunity from service disruption due to weather. Errors are introduced while the carrier waves travel through the ionosphere; however, these are corrected and kept to a minimum. GPS is also relatively inexpensive.

GPS receivers for IFR navigation in aircraft must be built to TSO-129A. This raises the price above that of handheld units used for hiking or in an automobile. But the overall cost of GPS is low due to its small infrastructure. Most of the inherent accuracy is built into the space and control segments permitting reliable positioning with inexpensive user equipment.

The accuracy of current GPS is within 20 meters horizontally and a bit more vertically. This is sufficient for en route navigation with greater accuracy than required. However, departures and approaches require more stringent accuracy. Integration of the wide area augmentation system (WAAS) improves GPS accuracy to within 7.6 meters and is discussed below. The future of GPS calls for additional accuracy by adding two new transmissions from each satellite. An L2C channel will be for general use in non-safety critical application. An aviation dedicated L5 channel will provide the accuracy required for category I, II, and III landings. It will enable the NEXTGEN NAS plan along with ADS-B.

WIDE AREA AUGMENTATION SYSTEM

To increase the accuracy of GPS for aircraft navigation, the wide area augmentation system (WAAS) was developed. It consists of approximately 25 precisely surveyed ground stations that receive GPS signals and ultimately transmit correction information to the aircraft. An overview of WAAS components and its operation is shown in *Figure 5-170*.

WAAS ground stations receive GPS signals and forward position errors to two master ground stations. Time and location information is analyzed, and correction instructions are sent to communication satellites in geostationary orbit over the NAS. The satellites broadcast GPS-like signals that WAAS enabled GPS receivers use to correct position information received from GPS satellites.

A WAAS enable GPS receiver is required to use the wide area augmentation system. If equipped, an aircraft qualifies to perform precision approaches into thousands of airports without any ground based approach equipment. Separation minimums are also able to be reduced between aircraft that are WAAS equipped. The WAAS system is known to reduce position errors to 1-3 meters laterally and vertically.

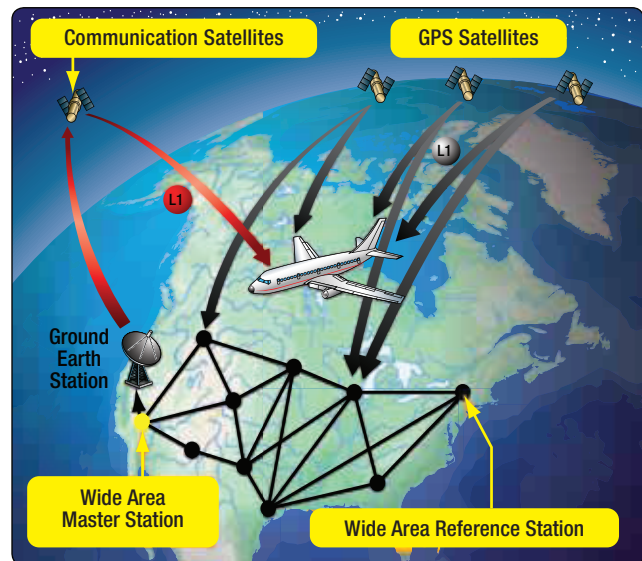


Figure 5-170. The wide area augmentation system (WAAS) is used to refine GPS positions to a greater degree of accuracy. A WAAS enabled GPS receiver is required for its use as corrective information is sent from geostationary satellites directly to an aircraft's GPS receiver for use.

INERTIAL NAVIGATION SYSTEM (INS) AND INERTIAL REFERENCE SYSTEM (IRS)

An inertial navigation system (INS) is used on some large aircraft for long range navigation. This may also be identified as an inertial reference system (IRS), although the IRS designation is generally reserved for more modern systems.

An INS/IRS is a self contained system that does not require input radio signals from a ground navigation facility or transmitter. The system derives attitude, velocity, and direction information from measurement of the aircraft's accelerations given a known starting point. The location of the aircraft is continuously updated through calculations based on the forces experienced by INS accelerometers. A minimum of two accelerometers is used, one referenced to north, and the other referenced to east. In older units, they are mounted on a gyro-stabilized platform. This averts the introduction of errors that may result from acceleration due to gravity. An INS uses complex calculation made by an INS computer to convert applied forces into location information.

An interface control head is used to enter starting location position data while the aircraft is stationary on the ground. This is called initializing. (*Figure 5-171*) From then on, all motion of the aircraft is sensed by the built-in accelerometers and run through the computer.



Figure 5-171. An interface panel for three air data and inertial reference systems on an Airbus. The keyboard is used to initialize the system. Latitude and longitude position is displayed at the top.

Feedback and correction loops are used to correct for accumulated error as flight time progresses.

The amount an INS is off in one hour of flight time is a reference point for determining performance. Accumulated error of less than one mile after one hour of flight is possible. Continuous accurate adjustment to the gyro-stabilized platform to keep it parallel to the Earth's surface is a key requirement to reduce accumulated error. A latitude/longitude coordinate system is used when giving the location output.

INS is integrated into an airliner's flight management system and automatic flight control system. Waypoints can be entered for a predetermined flightpath and the INS will guide the aircraft to each waypoint in succession. Integration with other NAV aids is also possible to ensure continuous correction and improved accuracy but is not required. Modern INS systems are known as IRS. They are completely solid-state units with no moving parts.

Three ring, laser gyros replace the mechanical gyros in the older INS platform systems. This eliminates precession and other mechanical gyro shortcomings. The use of three solid-state accelerometers, one for each plane of movement, also increases accuracy. The accelerometer and gyro output are input to the computer for continuous calculation of the aircraft's position.

The most modern IRS integrate is the satellite GPS. The GPS is extremely accurate in itself. When combined with IRS, it creates one of the most accurate navigation systems available. The GPS is used to initialize the IRS so the pilot no longer needs to do so. GPS also feeds data into the IRS computer to be used for error correction. Occasional service interruptions and altitude inaccuracies of the GPS system pose no problem for IRS/GPS. The IRS functions continuously and is completely self contained within the IRS unit. Should the GPS falter, the IRS portion of the system continues without it. The latest electronic technology has reduced the size and weight of INS/IRS avionics units significantly.

Figure 5-172 shows a modern micro-IRS unit that measures approximately 6-inches on each side.

COMMUNICATION AND NAVIGATION AVIONICS INSTALLATIONS

The aircraft maintenance technician may remove, install, inspect, maintain, and troubleshoot avionics equipment. It is imperative to follow all equipment and airframe manufacturers' instruction when dealing with an aircraft's avionics.

The installation of avionics equipment is partially mechanical, involving sheet metal work to mount units, racks, antennas, and controls. Routing of the interconnecting wires, cables, antenna leads, etc. is also an important part of the installation process. When a location for the equipment is selected by the manufacturer avionics radio equipment is securely mounted to the aircraft. All mounting bolts must be



Figure 5-172. A modern micro-IRS with built-in GPS.

secured by locking devices to prevent loosening from vibration. Adequate clearance between all units and adjacent structure is provided to prevent mechanical damage to electric wiring or to the avionics equipment from vibration, chafing, or landing shock. Combustible materials are kept away from avionics.

The performance and service life of most avionics equipment is seriously limited by excessive ambient temperatures. High performance aircraft with avionics equipment racks typically route air-conditioned air over the avionics to keep them cool. It is also common for non-air conditioned aircraft to use a blower or scooped ram air to cool avionics installations. Measures are also taken to prevent moisture from reaching the avionics equipment. The presence of water in avionics equipment areas promotes deterioration of the exposed components and could lead to failure.

Avionics equipment is sensitive to mechanical shock and vibration and is normally shock mounted to provide some protection against inflight vibration and landing shock. Vibration is a continued motion by an oscillating force. The amplitude and frequency of vibration of the aircraft structure will vary considerably with the type of aircraft. Special shock mounted racks are often used to isolate avionics equipment from vibrating structure.

(Figure 5-173)

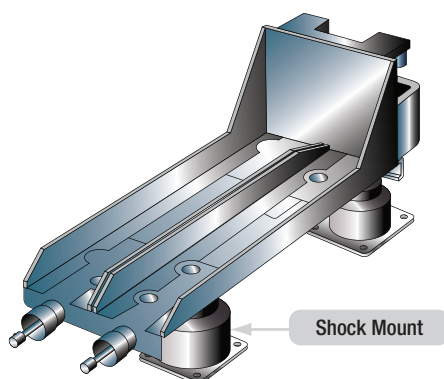


Figure 5-173. A shock mounted equipment rack is often used to install avionics.

Such mounts provide adequate isolation over the entire range of expected vibration frequencies. Periodic inspection of the shock mounts is required and defective mounts should be replaced with the proper type. The following factors to observe during inspection are:

1. Deterioration of the shock-absorbing material.
2. Stiffness and resiliency of the material.
3. Overall rigidity of the mount.

If the mount is too stiff, it may not provide adequate protection against the shock of landing. If the shock mount is not stiff enough, it may allow prolonged vibration following an initial shock

Shock-absorbing materials commonly used in shock mounts are usually electrical insulators. For this reason, each electronic unit mounted with shock mounts must be electrically bonded to a structural member of the aircraft to provide a current path to ground. This is accomplished by secure attachment of a tinned copper wire braid from the component, across the mount, to the aircraft structure as shown in **Figure 5-174**. Occasional bonding is accomplished with solid aluminum or copper material where a short flexible strap is not possible.

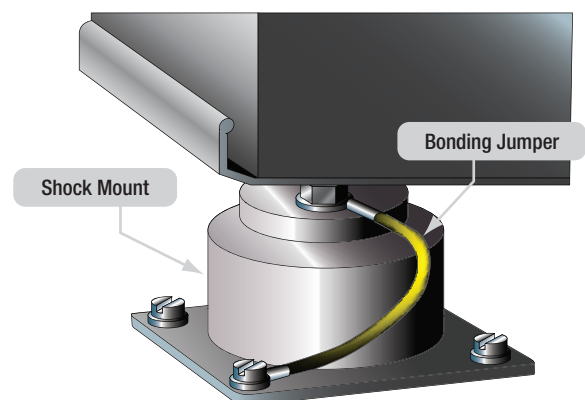


Figure 5-174. A bonding jumper is used to ground an equipment rack and avionics chassis around the non-conductive shock mount material.

Question: 5-1

The instruments used in controlling the aircraft's flight attitude are known as the _____.

Question: 5-6

An analog vertical speed indicator (VSI) may also be referred to as a _____ or _____.

Question: 5-2

Name three types of solid state pressure sensors on modern aircraft.

Question: 5-7

A primary flight instrument that is a differential pressure gauge comparing ram air pressure and static air pressure is called _____.

Question: 5-3

The turbine engine pressure indication that compares the total exhaust pressure to the pressure of the ram air at the inlet of the engine is called the _____ indicator.

Question: 5-8

The speed of sound changes with _____ and _____.

Question: 5-4

The three most common pitot-static instruments are the _____, the _____ and the _____.

Question: 5-9

A _____ is an electric system used for transmitting information from one point to another on non-digital aircraft.

Question: 5-5

The term density altitude describes altitude corrected for nonstandard _____.

Question: 5-10

The tachometer, or tach, is an instrument that indicates the speed(s) of the _____ of a turbine engine.

ANSWERS

Answer: 5-1

flight instruments.

Answer: 5-6

vertical velocity indicator (VVI).
rate-of-climb indicator.

Answer: 5-2

Crystalline piezoelectric.
Piezoresistor.
Semiconductor chip.

Answer: 5-7

an airspeed indicator.

Answer: 5-3

engine pressure ratio or EPR gauge.

Answer: 5-8

altitude.
temperature.

Answer: 5-4

altimeter.
airspeed indicator.
vertical speed indicator.

Answer: 5-9

synchro system.

Answer: 5-5

temperature.

Answer: 5-10

compressor section(s).

Question: 5-11

A rotating gear wheel that alters the magnetic field flux density as it moves past pole pieces is found in a _____.

Question: 5-16

An elaborate and very accurate method of direction indication that combines the use of a gyro, a magnetic compass, and a remote indicating system is called a _____ or a _____.

Question: 5-12

The two main types of angle of attack (AOA) sensors used on jet transport aircraft are the _____ and the _____.

Question: 5-17

In a _____ gyro, a vibrating or oscillating piezoelectric device replaces the spinning, weighted ring of the mechanical gyro.

Question: 5-13

For most metals, electrical _____ changes as the temperature of the metal changes.

Question: 5-18

In a turn, the ball of a turn-and-slip indicator responds to gravity and _____ force.

Question: 5-14

If one of the junctions is heated to a higher temperature than the other, an electromotive force is produced in the circuit. This describes the basic operation of a _____.

Question: 5-19

Name four basic elements (components) of an autopilot system.

Question: 5-15

Compensating screws are turned, which move small permanent magnets in the compass case to correct for magnetic _____.

Question: 5-20

Near continuous rudder input is needed to counteract the tendency of an aircraft to oscillate around their vertical axis while flying a fixed heading. A _____ is used to correct this motion.

ANSWERS

Answer: 5-11

tachometer probe.

Answer: 5-16

slaved gyro compass.

flux gate compass system.

Answer: 5-12

vane type.

slotted type.

Answer: 5-17

MEMS (microelectromechanical).

Answer: 5-13

resistance.

Answer: 5-18

centrifugal force.

Answer: 5-14

thermocouple.

Answer: 5-19

Sensing.

Command.

Output.

Computing.

Feedback or follow up.

Answer: 5-15

deviation.

Answer: 5-20

yaw damper.

Question: 5-21

An error of _____ should not be exceeded when testing a VOR system with a VOR test facility.

Question: 5-26

The traffic collision and avoidance system that is required internationally in aircraft with more than 30 seats or weighing more than 15 000 kg is _____.

Question: 5-22

An ADF (automatic direction finder) uses a _____ antenna.

Question: 5-27

The newest navigational aid that makes use of ground broadcasted information and on board GPS is called _____.

Question: 5-23

A Radio Magnetic Indicator combines indications from a magnetic compass, VOR, and ADF into one instrument.

Question: 5-28

Three common types of weather aids used on the aircraft flight deck are _____.

Question: 5-24

The radio navigation radio transmission that provides horizontal guidance to the centerline of the runway in an ILS (instrument landing system) is called the _____.

Question: 5-29

_____ ground stations receive GPS signals and forward position errors to two master ground stations.

Question: 5-25

An airborne _____ provides positive identification and location of an aircraft on the radar screens of ATC.

Question: 5-30

To receive _____ radio waves, generally a V-shaped, horizontally polarized, bi-pole antenna is used.

ANSWERS

Answer: 5-21
+ or - 4°.

Answer: 5-26
TCAS II.

Answer: 5-22
loop.

Answer: 5-27
ADS-B
(automatic dependent surveillance broadcast).

Answer: 5-23
magnetic compass.
VOR.
ADF.

Answer: 5-28
on-board radar.
lightning detectors.
satellite or other source weather radar information that
is uploaded to the aircraft from an outside source.

Answer: 5-24
localizer.

Answer: 5-29
WAAS (wide area augmentation system).

Answer: 5-25
transponder.

Answer: 5-30
VOR VHF.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 06

ELECTRICAL POWER (ATA 24)

Knowledge Requirements

11.6 - Electrical Power (ATA 24)

Batteries Installation and Operation; DC power generation;
AC power generation;
Emergency power generation; Voltage regulation;
Power distribution;
Inverters, transformers, rectifiers; Circuit protection;
External/Ground power.

3

ELECTRICAL POWER
(ATA 24)

10.6 - ELECTRICAL POWER

BATTERY

A battery is an energy storage device that relies on electrochemical reactions to deliver energy.

PRIMARY CELL BATTERIES

The dry cell is the most common type of primary-cell battery and is similar in its characteristics to that of an electrolytic cell. This type of a battery is basically designed with a metal electrode or graphite rod acting as the cathode (+) terminal, immersed in an electrolytic paste. This electrode/electrolytic build-up is then encased in a metal container, usually made of zinc, which itself acts as the anode (-) terminal. When the battery is in a discharge condition an electrochemical reaction takes place resulting in one of the metals being consumed. Because of this consumption, the charging process is not reversible. Attempting to reverse the chemical reaction in a primary cell by way of recharging is usually dangerous and can lead to a battery explosion. These batteries are commonly used to power items such as flashlights. The most common primary cells today are found in alkaline batteries, silver-oxide and lithium batteries. The earlier carbon-zinc cells, with a carbon post as cathode and a zinc shell as anode were once prevalent but are not as common.

SECONDARY CELL BATTERIES

A secondary cell is any kind of electrolytic cell in which the electrochemical reaction that releases energy is reversible. The lead-acid car battery is a secondary cell battery as are some aircraft batteries. The electrolyte is sulphuric acid (battery acid), the positive electrode is lead peroxide, and the negative electrode is lead. A typical lead-acid battery consists of six lead-acid cells in a case. Each cell produces 2 volts, so the whole battery produces a total of 12 volts. Other commonly used secondary cell chemistry types are nickel cadmium (NiCd), nickel metal hydride (NiMH), lithium-ion (Li-ion), and Lithium-ion polymer (Li-ion polymer).

Lead-acid batteries used in aircraft are similar to automobile batteries. The lead acid battery is made up of a series of identical cells each containing sets of positive and negative plates. **Figure 6-1** illustrates each cell contains positive plates of lead dioxide (PbO_2), negative plates of spongy lead, and electrolyte (sulfuric acid and water).

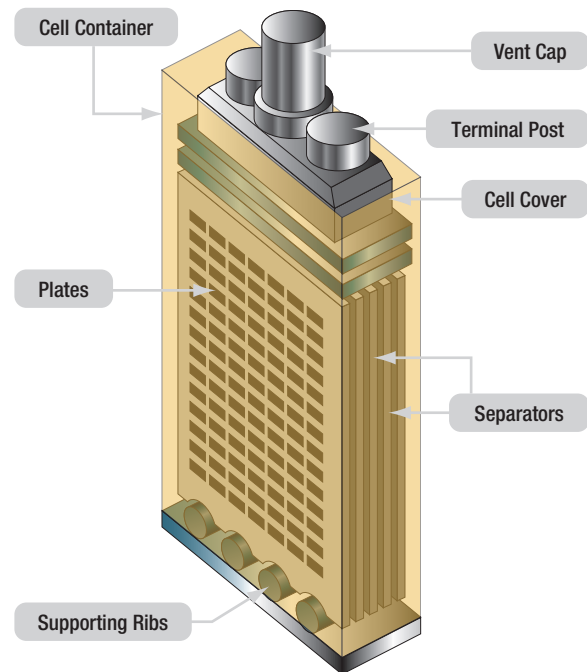


Figure 6-1. Lead-acid cell construction.

A practical cell is constructed with many more plates than just two in order to get the required current output. All positive plates are connected together as well as all the negatives. Because each positive plate is always positioned between two negative plates, there are always one or more negative plates than positive plates.

Between the plates are porous separators that keep the positive and negative plates from touching each other and shorting out the cell. The separators have vertical ribs on the side facing the positive plate. This construction permits the electrolyte to circulate freely around the plates. In addition, it provides a path for sediment to settle to the bottom of the cell.

Each cell is seated in a hard rubber casing through the top of which are terminal posts and a hole into which is screwed a non-spill vent cap. The hole provides access for testing the strength of the electrolyte and adding water. The vent plug permits gases to escape from the cell with a minimum of leakage of electrolyte, regardless of the position the airplane might assume.

Figure 6-2 shows the construction of the vent plug. In level flight, the lead weight permits venting of gases through a small hole. In inverted flight, this hole is covered by the lead weight.

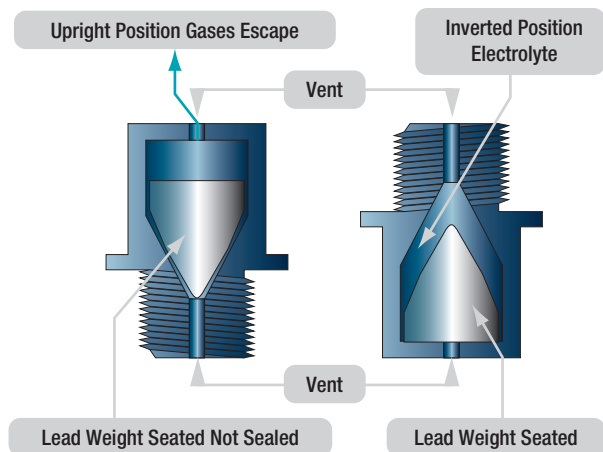


Figure 6-2. Nonspill battery vent plug.

Dry Charged (Flooded) Lead Acid Batteries

The electrolyte is added to the battery when it is placed in service, and battery life begins when the electrolyte is added. An aircraft storage battery consists of 6 or 12 lead-acid cells connected in series. The open circuit voltage of the 6 cell battery is approximately 12 volts, and the open circuit voltage of the 12-cell battery is approximately 24 volts. Open circuit voltage is the voltage of the battery when it is not connected to a load. When flooded (vented) batteries are on charge, the oxygen generated at the positive plates escapes from the cell. Concurrently, at the negative plates, hydrogen is generated from water and escapes from the cell. The overall result is the gassing of the cells and water loss. Therefore, flooded cells require periodic water replenishment. (*Figure 6-3*)

Valve-Regulated (Sealed) Lead-Acid Batteries (VRLA)

VRLA batteries contain all electrolyte absorbed in glass-mat separators with no free electrolyte and are sometimes referred to as sealed batteries. (*Figure 6-4*) The electrochemical reactions for VRLA batteries are the same as flooded batteries, except for the gas recombination mechanism that is predominant in VRLA batteries. These types of battery are used in general aviation and turbine powered aircraft and are sometimes authorized replacements for NiCd batteries.

When VRLA batteries are on charge, oxygen combines chemically with the lead at the negative plates in the presence of H_2SO_4 to form lead sulfate and water. This oxygen recombination suppresses the generation of hydrogen at the negative plates. Overall, there is no water loss during charging.



Figure 6-3. Lead acid battery installation.



Figure 6-4. Valve-regulated lead-acid battery (sealed battery).

A very small quantity of water may be lost as a result of self discharge reactions; however, such loss is so small that no provisions are made for water replenishment. The battery cells have a pressure relief safety valve that may vent if the battery is overcharged.

NICKEL CADMIUM (NICD) BATTERIES

A NiCd battery consists of a metallic box, usually stainless steel, plastic-coated steel, painted steel, or titanium containing a number of individual cells. (*Figure 6-5*) These cells are connected in series to obtain 12 volts or 24 volts. The cells are connected by highly conductive nickel copper links. Inside the battery box, the cells are held in place by partitions, liners, spacers, and a cover assembly. The battery has a ventilation system to allow the escape of the gases produced during an overcharge condition and provide cooling during normal operation. NiCd cells installed in an aircraft battery are typical of the vented cell type.

The vented cells have a vent or low pressure release valve that releases any generated oxygen and hydrogen gases when overcharged or discharged rapidly. This also



Figure 6-5. NiCd battery installation.

means the battery is not normally damaged by excessive rates of overcharge, discharge, or even negative charge. The cells are rechargeable and deliver a voltage of 1.2 volts during discharge.

Aircraft that are outfitted with NiCd batteries typically have a fault protection system that monitors the condition of the battery. The battery charger is the unit that monitors the condition of the battery and the following conditions are monitored.

1. Overheat condition.
2. Low temperature condition (below -40°C).
3. Cell imbalance.
4. Open circuit.
5. Shorted circuit.

If the battery charger finds a fault, it turns off and sends a fault signal to the Electrical Load Management System (ELMS).

NiCd batteries are capable of performing to its rated capacity when the ambient temperature of the battery is in the range of approximately $15\text{--}32^{\circ}\text{C}$. An increase or decrease in temperature from this range results in reduced capacity. NiCd batteries have a ventilation system to control the temperature of the battery. A combination of high battery temperature (in excess of 71°C) and overcharging can lead to a condition called thermal runaway. (*Figure 6-6*) The temperature of the battery has to be constantly monitored to ensure safe operation. Thermal runaway can result in a NiCd chemical fire and/or explosion of the NiCd battery under recharge by a constant-voltage source and is due to cyclical, ever increasing temperature and charging current.



Figure 6-6. Thermal runaway damage.

One or more shorted cells or an existing high temperature and low charge can produce the following cyclical sequence of events:

1. Excessive current,
2. Increased temperature,
3. Decreased cell(s) resistance,
4. Further increased current, and
5. Further increased temperature.

This does not become a self sustaining thermal chemical action if the constant voltage charging source is removed before the battery temperature is in excess of 71°C .

LITHIUM-ION BATTERIES

The most recent type of battery to be certified in aircraft is the lithium-ion battery. These batteries have greater capacity and weigh less than NiCd or lead acid types. They have no memory as NiCd batteries have and discharge less than half as slowly when not being used. The anode is a graphite layered structure capable of storing and releasing lithium ions. Cathode materials vary. The certified battery used on the Boeing 787 is made of lithium cobalt oxide (LiCoO_2) with an aluminum core.

A water-free electrolyte composed of organic carbonates resides between the anode and cathode. It functions as a transport medium for the lithium ions moving from the anode to the cathode during discharge and from the cathode to the anode during charging. A separator porous to the Li^+ ions is between the anode and cathode in each cell. The electrons that cause the lithium to be ions are the current used in the external circuit to power aircraft electrical buses and components.

Typical cell output voltage is between 3 and 4.2 volts depending primarily on the materials used to construct the cathode. Eight cells connected in series are typical as shown in *Figure 6-7*.

Lithium-ion aircraft batteries require built in safety devices to prevent overheating and thermal runaway. They are constructed with a wide variety of material choices that result in compromise between capacity, longevity, environmental endurance and operating range, current loading, specific energy, size and weight, etc. Additional current monitoring and other safety and alerting devices are included to warn flight crew of battery status and malfunction. Technicians must follow all manufacturer instructions when maintaining lithium-ion batteries.

BATTERY INSTALLATIONS AND OPERATION

Applicable to all installations are redundancy and the ability to maintain electric power to essential systems in case of electrical failure. Most modern airliners have at least two batteries - one for the aircraft and a separate, dedicated battery for APU starting. The main aircraft battery supplies the power for DC loads while the aircraft is on the ground. In the air, in the event of an APU failure, the batteries supply standby power until a Ram Air Turbine (RAT) can be deployed to cover the electrical load required by critical systems such as flight

control and instruments. The RAT generates power from the airstream due to the speed of the aircraft. In normal conditions the RAT is retracted into the fuselage. (*Figure 6-8*)

Usually, the same nickel-cadmium battery is used for both the main aircraft battery and the APU battery. They are interchangeable. These batteries weigh near 100 pounds (45 kg). Attach fittings on a battery facilitate the use of lifting equipment during removal and installation. Most aircraft use 28V DC batteries but configurations exist where two 14V batteries are connected in series to arrive at 28V for bus use.

Typical transport aircraft batteries have two connectors. The large connector is a terminal block which connects the high power output of the battery into the bus system. The smaller electrical connector is for battery control and status signals. Temperature sensors and overheat sensing are common. A cooling fan may be included in the installation. (*Figure 6-9*)

The battery system installation includes a battery charger for each battery. The chargers change AC power into DC power for DC buses in addition to keeping the batteries charged. Status monitoring and control signals are AC powered throughout the small electrical connector. The battery chargers have multiple charging modes. Boeing chargers in the 777 also have a transformer rectifier (TR) mode which converts the 115V AC input to DC power.



Figure 6-7. Cells and wiring in a lithium-ion aircraft battery.



Figure 6-8. A ram air turbine on a Boeing 757.



Figure 6-9. Nickel Cadmium aircraft battery.

Up to 65 amps may be drawn directly from the tightly controlled voltage output of this TR. Temperature sensing is important during battery charging. Built in sensors are used to isolate the battery charger should an over heat occur. Most airliner battery chargers are controlled by a series of relays so that the chargers only charge when power production is normal. In addition to a control relay in the charger, a battery thermal switch controlled by battery temperature, interrupts charging when the battery or the charger is hot. Charging may also be interrupted by relays controlling connectivity to the APU starter or the fueling panel. The manufacturer's wiring diagrams are the source for learning exactly how the batteries and charger are configured in relationship to other electrical components.

On a Boeing 777, an electrical load management system (ELMS) is used to control many aircraft electrical components. It is comprised of power distribution panels in a centralized location, each designed with a specific function within the entire power system of the aircraft. The panel are the left and right power panels, the left and right power management panels, the auxiliary power panel, the ground service/handling power panel and the standby power management panel. (*Figure 6-10*) The main aircraft battery is controlled through the standby power management panel. It is also directly or indirectly connected to all panels in the ELMS.

AC POWER GENERATION

An AC power system is the primary source of power on most transport aircraft.

These extremely reliable power distribution systems are computer controlled. Multiple power sources (AC alternators also known as AC generators) and a variety of distribution busses are used for redundancy. A typical airliner contains two or more main AC generators driven by the main engines, as well as more than one backup AC generator. DC systems are also employed on large aircraft and the aircraft battery may be used to supply emergency power in case of a multiple failures.

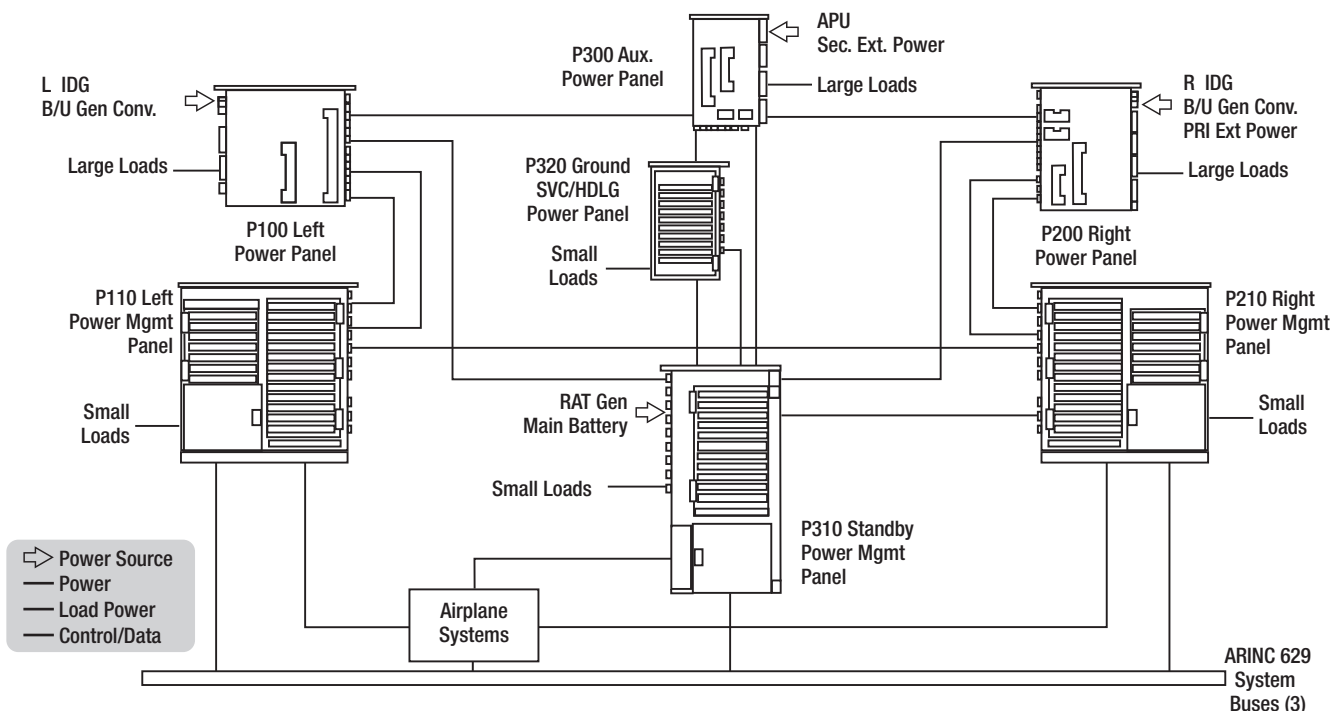


Figure 6-10. The main battery position in the aircraft electrical load management system is through the standby power management panel.

The typical AC generator produces three phase 115-volt AC at 400 Hz. AC generators were discussed previously in this module series. Since most modern transport category aircraft are designed with two engines, there are two main generators. The APU also drives a generator. This unit is available during flight if one of the main generators fails. The main and auxiliary generators may be similar in output capacity or back-up generators may have a low output. The main generators typically supply a maximum of 120 kilovolt amps (KVA). A fourth generator, driven by an emergency ram air turbine, is also available in the event the two main generators and one auxiliary generator fail. This emergency generator is typically smaller and produces less power. With four AC generators available, it is highly unlikely that a complete power failure occurs. However, if all AC generators are lost, the aircraft battery continues to supply DC electrical power to operate vital systems.

Note that on the latest generation of aircraft, there is even more redundancy for AC power production. The Boeing 777 has 2 integrated drive generators that are the primary AC generators but the aircraft also has two backup engine driven AC generators in case of failure of the primary generators. Each backup generator also contains two permanent magnet DC generators for powering the flight control power supply assemblies. Airbus aircraft have at least one backup AC generator that is powered by a hydraulic motor. Transport category aircraft use large amounts of electrical power for a variety of systems.

Passenger comfort requires power for lighting, audio visual systems, and galley power for food warmers and beverage coolers. A variety of electrical systems are required to fly the aircraft, such as flight control systems, electronic engine controls, communication, and navigation systems. The output capacity of one engine-driven AC generator can typically power all necessary electrical systems. A second engine-driven generator is operated during flight to share the electrical loads and provide redundancy. **Figure 6-11** illustrates the electrical power system of a Boeing 777.

The complexity of multiple generators and a variety of distribution busses requires several control units to maintain a constant supply of safe electrical power. The AC electrical system must maintain a constant output of 115 to 120 volts at a frequency of 400 Hz

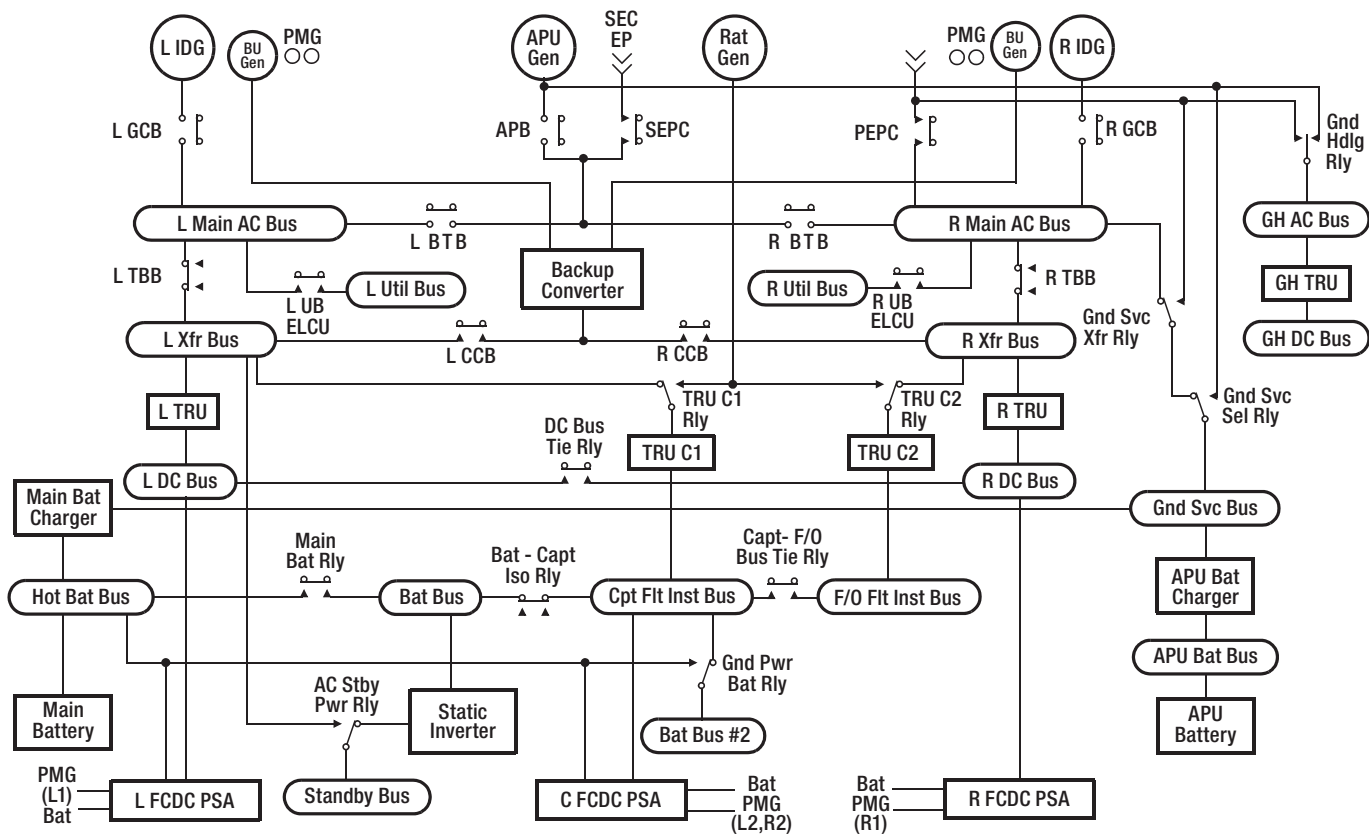
(± 10 percent). The system must ensure power limits are not exceeded. AC generators are connected to the appropriate distribution busses at the appropriate time, and generators are in phase when needed. There is also the need to monitor and control any external power supplied to the aircraft, as well as control of all DC electrical power.

Two electronic line replaceable units are used to control the electrical power on a typical large aircraft. The generator control unit (GCU) is used for control of AC generator functions, such as voltage regulation and frequency control. The bus power control unit (BPCU) is used to control the distribution of electrical power between the various distribution busses on the aircraft. The GCU and BPCU work together to control electrical power, detect faults, take corrective actions when needed, and report any defect to the pilots and the aircraft's central maintenance system. There is typically one GCU for each AC generator and at least one BPCU to control bus connections. These LRUs are located in the aircraft's electronics equipment bay and are designed for easy replacement.

When the pilot calls for generator power by activating the generator control switch on the flight deck, the GCU monitors the system to ensure correct operation. If all systems are operating within limits, the GCU energizes the appropriate generator circuits and provides voltage regulation for the system. The GCU also monitors AC output to ensure a constant 400-Hz frequency. If the generator output is within limits, the GCU then connects the electrical power to the main generator bus through an electrical contactor (solenoid). These contactors are often called generator breakers since they break (open) or make (close) the main generator circuit.

After generator power is available, the BPCU activates various contactors to distribute the electrical power. The BPCU monitors the complete electrical system and communicates with the GCU to ensure proper operation. The BPCU employs remote current sensors known as current transformers (CT) to monitor the system. (**Figure 6-12**)

A CT is an inductive unit that surrounds the main power cables of the electrical distribution system. As AC power flows through the main cables, the CT receives an induced voltage. The amount of CT voltage is



Electrical Power System Schematic

Figure 6-11. The electrical power system of a Boeing 777.

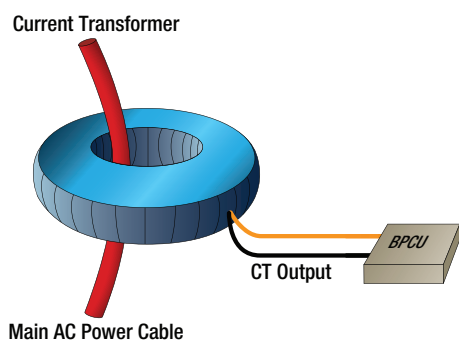


Figure 6-12. Current transformer.

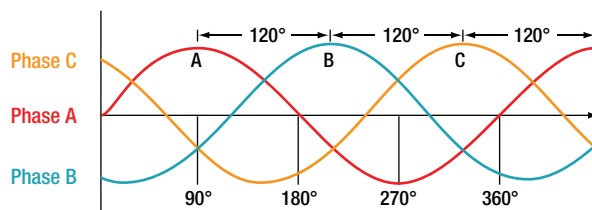
directly related to the current flowing through the cable. The CT connects to the BPCU, which allows accurate current monitoring of the system. A typical aircraft employs several CTs throughout the electrical system. The BPCU is a dedicated computer that controls the electrical connections between the various distribution busses found on the aircraft.

The BPCU uses contactors (solenoids) called bus tie breakers (BTB) for connection of various circuits. These BTBs open/close the connections between the busses as

needed for system operation as called for by the flight crew and the BPCU. This sounds like a simple task, yet to ensure proper operation under a variety of conditions, the bus system becomes very complex. There are three common types of distribution bus systems found on transport category aircraft: split bus, parallel bus, and split parallel. These are examined in the section on power distribution below.

AC ALTERNATORS

AC alternators produce a three phase AC output. For each revolution of the alternator, the unit produces three separate voltages. The sine waves for these voltages are separated by 120° . (*Figure 6-13*) This wave pattern is



One Full Rotation of the AC Alternator

Figure 6-13. AC alternator sine waves.

similar to those produced internally by a DC alternator; however, in this case, the AC alternator does not rectify the voltage and the output of the unit is AC.

The modern AC alternator does not utilize brushes or slip rings and is often referred to as a brushless AC alternator. This brushless design is extremely reliable and requires very little maintenance. In a brushless alternator, energy to or from the alternator's rotor is transferred using magnetic energy. In other words, energy from the stator to the rotor is transferred using magnetic flux energy and the process of electromagnetic induction. A typical large aircraft AC alternator is shown in **Figure 6-14**.

As seen in **Figure 6-15**, the brushless alternator actually contains three generators: the Exciter generator (armature and permanent magnet field), the Pilot exciter generator (armature and fields windings), and the main AC alternator (armature winding and field windings).

The need for brushes is eliminated by using a combination of these three distinct generators. The exciter is a small AC generator with a stationary field made of a permanent magnet and two electromagnets. The exciter armature is three phase and mounted on the rotor shaft. The exciter armature output is rectified and sent to the pilot exciter field and the main generator field.

The pilot exciter field is mounted on the rotor shaft and is connected in series with the main generator field. The pilot exciter armature is mounted on the stationary part of the assembly. The AC output of the pilot exciter armature is supplied to the generator control circuitry where it is rectified, regulated, and then sent to the exciter field windings. The current sent to the exciter field provides the voltage regulation for the main AC alternator. If greater AC alternator output is needed, there is more current sent to the exciter field and vice versa.

In short, the exciter permanent magnet and armature starts the generation process, and the output of the exciter armature is rectified and sent to the pilot exciter field. The pilot exciter field creates a magnetic field and



Figure 6-14. Large aircraft AC alternator.

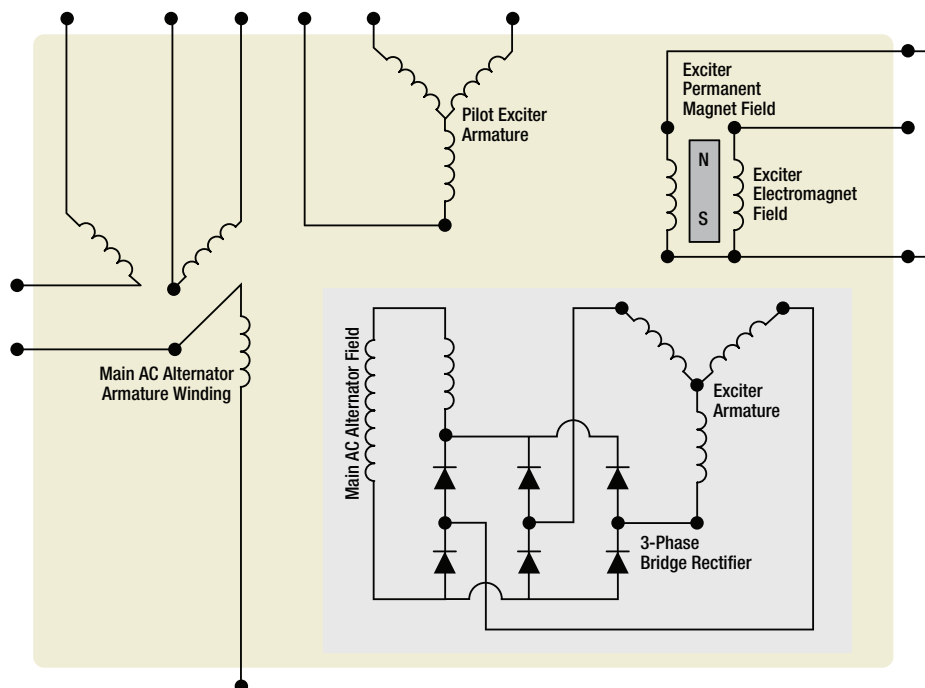


Figure 6-15. Schematic of an AC alternator.

induces power in the pilot exciter armature through electromagnetic induction. The output of the pilot exciter armature is sent to the main alternator control unit and then sent back to the exciter field. As the rotor continues to turn, the main AC alternator field generates power into the main AC alternator armature, also using electromagnetic induction. The output of the main AC armature is three phase AC and used to power the various electrical loads.

Some alternators are cooled by circulating oil through the internal components of the alternator. The oil used for cooling is supplied from the constant speed drive assembly and often cooled by an external oil cooler assembly. Located in the flange connecting the generator and drive assemblies, ports make oil flow between the constant speed drive and the generator possible. This oil level is critical and typically checked on a routine basis.

ALTERNATOR DRIVE

The unit shown in *Figure 6-16* contains an alternator assembly combined with an automatic drive mechanism. The automatic drive controls the alternator's rotational speed which allows the alternator to maintain a constant 400 Hz AC output.

All AC alternators must rotate at a specific rpm to keep the frequency of the AC voltage within limits. Aircraft AC alternators should produce a frequency of approximately 400 Hz. If the frequency strays more than 10 percent from this value, the electrical systems do not operate correctly. A unit called a constant-speed drive (CSD) is used to ensure the alternator rotates at the correct speed to ensure a 400 Hz frequency. The CSD can be an independent unit or mounted within the alternator housing. When the CSD and the alternator are contained within one unit, the assembly is known as an integrated drive generator (IDG).

The CSD is a hydraulic unit similar to an automatic transmission found in a modern automobile. The engine of the automobile can change rpm while the speed of the car remains constant. This is the same process that occurs for an aircraft AC alternator. If the aircraft engine changes speed, the alternator speed remains constant.

A typical hydraulic type drive is shown in *Figure 6-17*. This unit can be controlled either electrically or mechanically. Modern aircraft employ an electronic

system. The constant-speed drive enables the alternator to produce the same frequency at slightly above engine idle rpm as it does at maximum engine rpm.

The hydraulic transmission is mounted between the AC alternator and the aircraft engine. Hydraulic oil or engine oil is used to operate the hydraulic transmission, which creates a constant output speed to drive the alternator. In some cases, this same oil is used to cool the alternator as shown in the CSD cutaway view of *Figure 6-17*.

The input drive shaft is powered by the aircraft engine gear case. The output drive shaft, on the opposite end of the transmission, engages the drive shaft of the alternator. The CSD employs a hydraulic pump assembly, a mechanical speed control, and a hydraulic drive. Engine rpm drives the hydraulic pump, the hydraulic drive turns the alternator. The speed control unit is made up of a wobble plate that adjusts hydraulic pressure to control output speed.

Figure 6-18 shows a typical electrical circuit used to control alternator speed. The circuit controls the hydraulic assembly found in a typical CSD. As shown, the alternator input speed is monitored by a tachometer generator. The tach generator signal is rectified and sent to the valve assembly. The valve assembly contains three electromagnetic coils that operate the valve. The AC alternator output is sent through a control circuit that also feeds the hydraulic valve assembly.

By balancing the force created by the three electro-magnets, the valve assembly controls the flow of fluid through the automatic transmission and controls the speed of the AC alternator. It should be noted that an AC alternator also produces a constant 400 Hz if that alternator is driven directly by an engine that rotates at a constant speed.

On many aircraft, the auxiliary power unit operates at a constant rpm. AC alternators driven by these APUs are typically driven directly by the engine, and there is no CSD required. For these units, the APU engine controls monitor the alternator output frequency. If the alternator output frequency varies from 400 Hz, the APU speed control adjusts the engine rpm accordingly to keep the alternator output within limits.

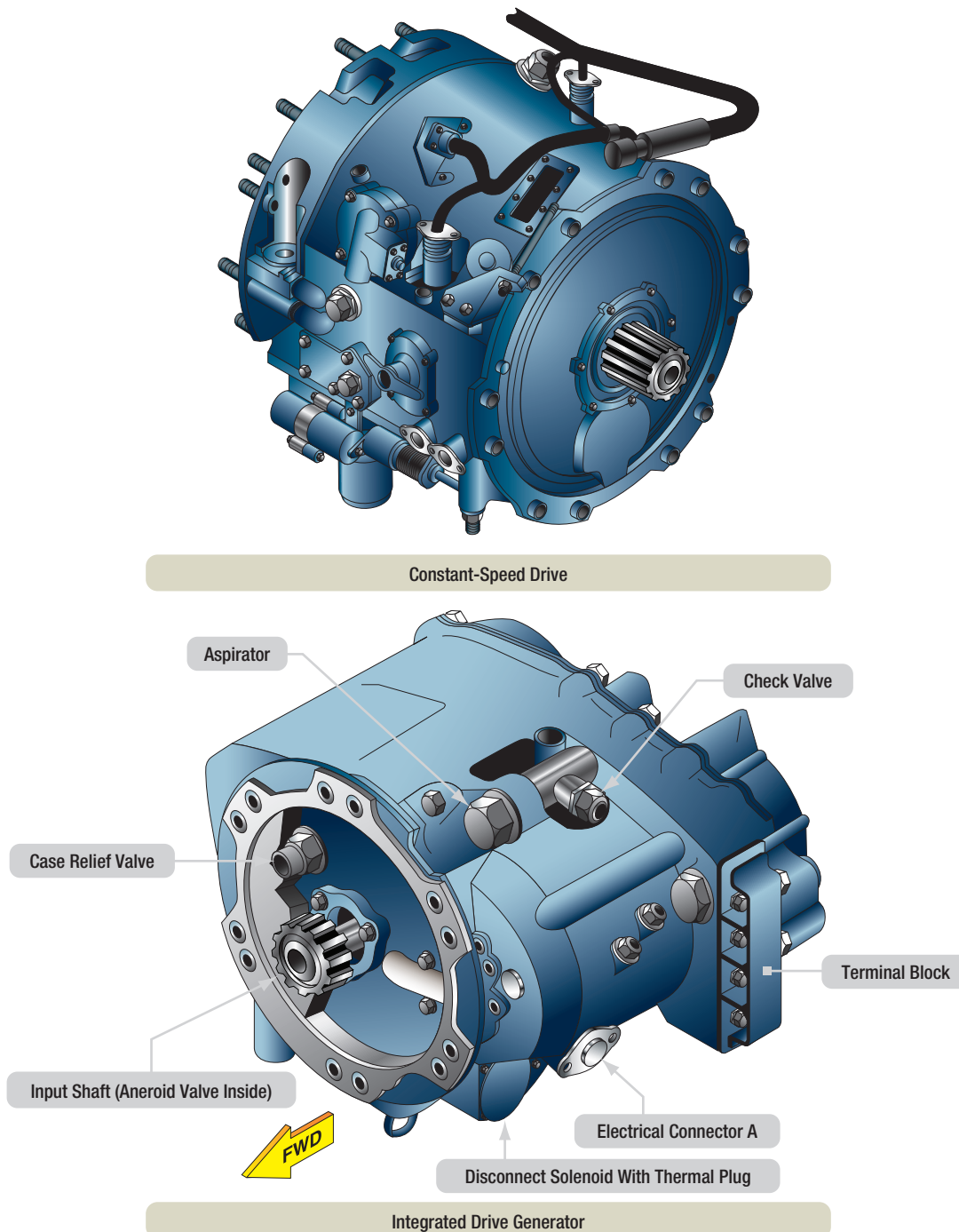


Figure 6-16 . Constant-speed drive (top) and integrated drive generator (bottom).

AC ALTERNATORS CONTROL SYSTEMS

Modern aircraft that employ AC alternators use several computerized control units, typically located in the aircraft's equipment bay for the regulation of AC power throughout the aircraft. **Figure 6-19** shows a photo of a typical equipment bay and computerized control units.

Since AC alternators are found on large transport category aircraft designed to carry hundreds of passengers, their control systems always have redundant computers that provide safety in the event of a system failure. Unlike DC systems, AC systems must ensure that the output frequency of the alternator stays within limits.

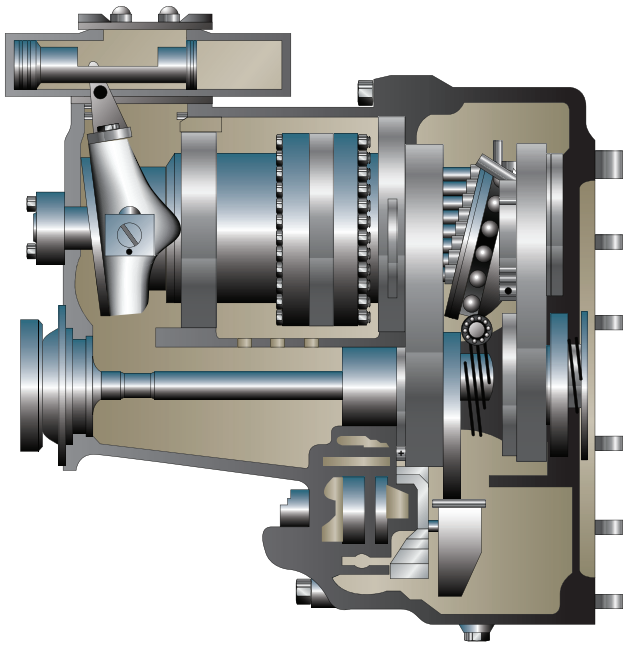


Figure 6-17. A hydraulic constant speed drive for an AC alternator.



Figure 6-19. Line replaceable units in an equipment rack.

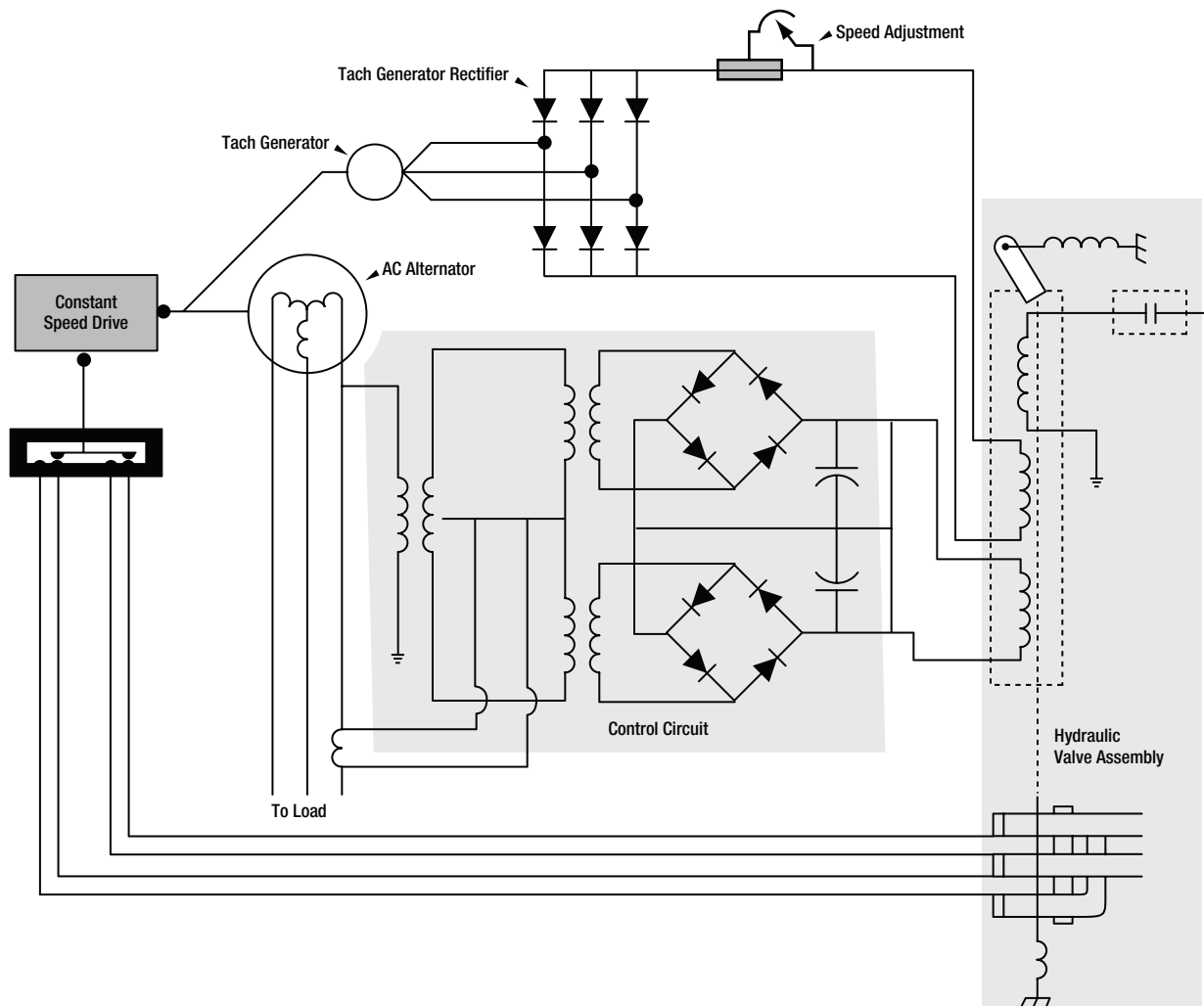


Figure 6-18. Speed control circuit.

If the frequency of an alternator varies from 400 Hz, or if two or more alternators connected to the same bus are out of phase, damage occurs to the system. All AC alternator control units contain circuitry that regulates both voltage and frequency. These control units also monitor a variety of factors to detect any system failures and take protective measures to ensure the integrity of the electrical system. The two most common units used to control AC alternators are the bus power control unit (BPCU) and the GCU. In this case, the term "generator" is used, and not alternator, although the meaning is the same. The GCU is the main computer that controls alternator functions.

The BPCU is the computer that controls the distribution of AC power to the power distribution busses located throughout the aircraft. There is typically one GCU used to monitor and control each AC alternator, and there can be one or more BPCUs on the aircraft. BPCUs are described later in this chapter; however, please note that the BPCU works in conjunction with the GCUs to control AC on modern aircraft.

A typical GCU ensures the AC alternator maintains a constant voltage, typically between 115 to 120 volts. The GCU ensures the maximum power output of the alternator is never exceeded. The GCU provides fault

detection and circuit protection in the event of an alternator failure. The GCU monitors AC frequency and ensures the output if the alternator remains 400 Hz. The basic method of voltage regulation is similar to that found in all alternator systems; the output of the alternator is controlled by changing the strength of a magnetic field. As shown in **Figure 6-20**, the GCU controls the exciter field magnetism within the brushless alternator to control alternator output voltage. The frequency is controlled by the CDS hydraulic unit in conjunction with signals monitored by the GCU.

The GCU is also used to turn the AC alternator on or off. When the pilot selects the operation of an AC alternator, the GCU monitors the alternator's output to ensure voltage and frequency are within limits. If the GCU is satisfied with the alternator's output, the GCU sends a signal to an electrical contactor that connects the alternator to the appropriate AC distribution bus. The contactor, often call the generator breaker, is basically an electromagnetic solenoid that controls a set of large contact points. The large contact points are necessary in order to handle the large amounts of current produced by most AC alternators. This same contactor is activated in the event the GCU detects a fault in the alternator output; however, in this case the contactor would disconnect the alternator from the bus.

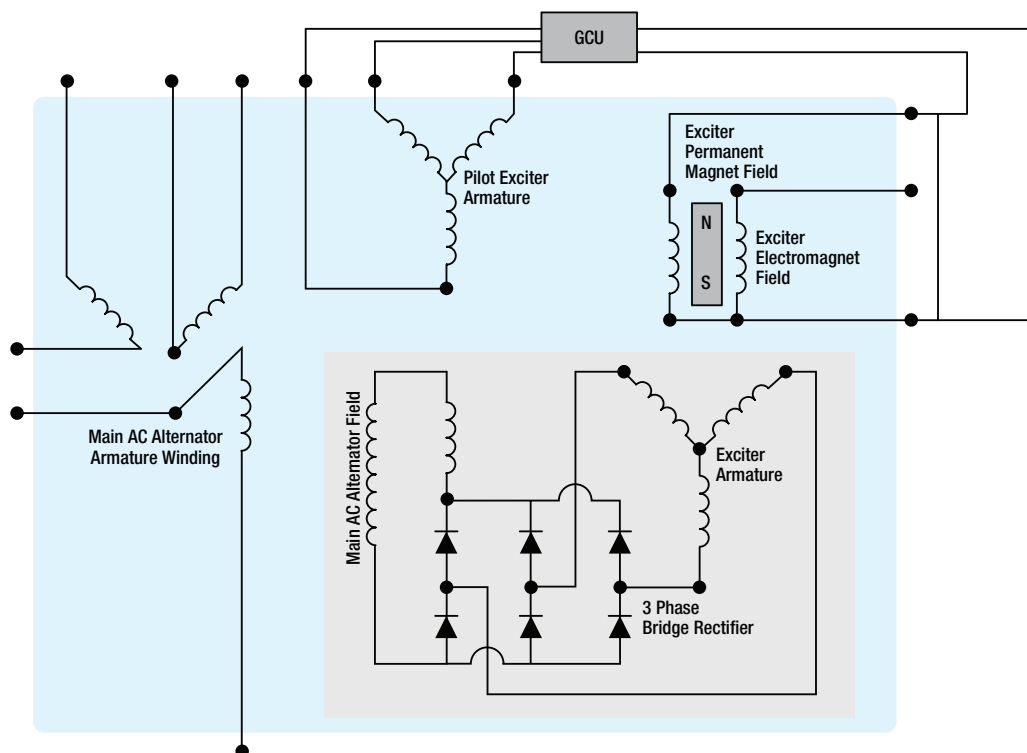


Figure 6-20. Schematic GCU control of the exciter field magnetism.

DC POWER GENERATION

Most modern airliners use AC generators for the primary source of power. However, there are still aircraft flying that may use DC generators for power. Smaller aircraft may also have DC generators or DC alternators. When present, DC generators normally have output controlled to 14 or 28V DC. The output is controlled by controlling field current strength. It is more common on large aircraft to produce DC power from AC generator power. Transformer rectifiers (TR) are used to convert the 115V AC to any DC voltage required - normally 28V DC. This DC powers various buses. Transformer rectifiers are also found in battery chargers.

An engine-driven DC generator requires a control circuit in order to ensure the generator maintains the correct voltage and current for the current electrical conditions of the aircraft. All aircraft are designed to operate within a specific voltage range (for example 13.5–14.5 volts) and since aircraft operate at a variety of engine speeds (remember, the engine drives the generator) and with a variety of electrical demands, all generators must be regulated by some control system. The generator control system is designed to keep the generator output within limits for all flight variables. Generator control systems are often referred to as voltage regulators or generator control units (GCU).

Aircraft generator output can easily be adjusted through control of the generator's magnetic field strength. Remember, the strength of the magnetic field has a direct effect on generator output. More field current means more generator output and vice versa. **Figure 6-21** shows a simple generator control used to adjust field current.

There are two basic types of generator controls: electromechanical and solid state (transistorized). The electromechanical type controls are found on older aircraft and tend to require regular inspection and maintenance. Solid-state systems are more modern and typically considered to have better reliability and more accurate generator output control.

FUNCTIONS OF GENERATOR CONTROL SYSTEMS

Most generator control systems perform a number of functions related to the regulation, sensing, and protection of the DC generation system.

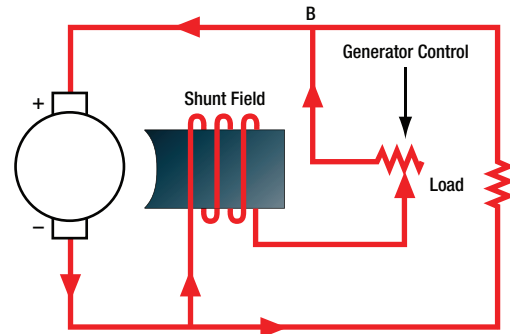


Figure 6-21. Regulation of generator voltage by field rheostat.

VOLTAGE REGULATION

The most basic of the GCU functions is that of voltage regulation. Regulation of any kind requires the regulation unit to take a sample of a generator output and compare that sample to a known reference. If the generator's output voltage falls outside of the set limits, then the regulation unit must provide an adjustment to the generator field current. Adjusting field current controls generator output.

OVER VOLTAGE PROTECTION

The over voltage protection system compares the sampled voltage to a reference voltage. The over voltage protection circuit is used to open the relay that controls the field excitation current. It is typically found on more complex generator control systems.

PARALLEL GENERATOR OPERATIONS

On multi-engine aircraft, a paralleling feature must be employed to ensure all generators operate within limits. In general, paralleling systems compare the voltages between two or more generators and adjust the voltage regulation circuit accordingly.

OVER EXCITATION PROTECTION

When one generator in a paralleled system fails, one of the generators can become overexcited and tends to carry more than its share of the load, if not all of the loads. Basically, this condition causes the generator to produce too much current. If this condition is sensed, the overexcited generator must be brought back within limits, or damage occurs. The over excitation circuit often works in conjunction with the overvoltage circuit to control the generator.

DIFFERENTIAL VOLTAGE

This function of a control system is designed to ensure all generator voltage values are within a close tolerance

before being connected to the load bus. If the output is not within the specified tolerance, then the generator contactor is not allowed to connect the generator to the load bus.

REVERSE CURRENT SENSING

If the generator cannot maintain the required voltage level, it eventually begins to draw current instead of providing it. This situation occurs, for example, if a generator fails. When a generator fails, it becomes a load to the other operating generators or the battery. The defective generator must be removed from the bus. The reverse current sensing function monitors the system for a reverse current. Reverse current indicates that current is flowing to the generator not from the generator. If this occurs, the system opens the generator relay and disconnects the generator from the bus.

GENERATOR CONTROLS FOR HIGH OUTPUT GENERATORS

Most modern high output generators are found on turbine powered corporate type aircraft. These small business jets and turboprop aircraft employ a generator and starter combined into one unit. This unit is referred to as a starter-generator. A starter-generator has the advantage of combining two units into one housing, saving space and weight. Since the starter-generator performs two tasks, engine starting and generation of electrical power, the control system for this unit is relatively complex.

A simple explanation of a starter-generator shows that the unit contains two sets of field windings. One field is used to start the engine and one used for the generation of electrical power. (*Figure 6-22*)

During the start function, the GCU must energize the series field and the armature causes the unit to act like a motor. During the generating mode, the GCU must disconnect the series field, energize the parallel field, and control the current produced by the armature. At this time, the starter-generator acts like a typical generator. Of course, the GCU must perform all the functions described earlier to control voltage and protect the system. These functions include voltage regulation, reverse current sensing, differential voltage, over excitation protection, over voltage protection, and parallel generator operations. A typical GCU is shown in *Figure 6-23*.

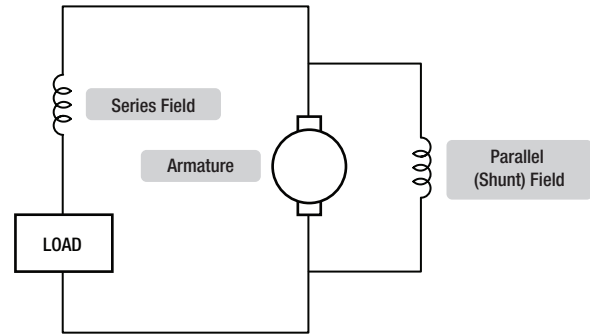


Figure 6-22. Starter-generator.



Figure 6-23. Generator control unit (GCU).

In general, modern GCUs for high-output generators employ solid-state electronic circuits to sense the operations of the generator or starter-generator. The circuitry then controls a series of relays and/or solenoids to connect and disconnect the unit to various distribution busses. One unit found in almost all voltage regulation circuitry is the zener diode. The zener diode is a voltage sensitive device that is used to monitor system voltage. The zener diode, connected in conjunction to the GCU circuitry, then controls the field current, which in turn controls the generator output.

OTHER VOLTAGE REGULATION

Small aircraft and older large aircraft that use DC generators for primary power must have a means for voltage regulation. The typical device for this is a generator control unit or voltage regulator that modifies current to the generator field to control generator output

power. As flight variables and electrical loads change, the voltage regulator monitors the electrical system and make the appropriate adjustments to ensure proper system voltage and current.

Voltage regulators found on older aircraft and small aircraft are for the low output generators used on these types of aircraft. They are typically electromechanical devices. Solid state units are found on more modern aircraft that employ DC alternators and not DC generators. The two most common types of voltage regulator are the carbon pile regulator and the three unit regulator. Each of these units controls field current using a type of variable resistor. Controlling field current then controls generator output. A simplified generator control circuit is shown in *Figure 6-24*.

CARBON PILE REGULATORS

The carbon pile regulator controls DC generator output by sending the field current through a stack of carbon disks (the carbon pile).

The carbon disks are in series with the generator field. If the resistance of the disks increases, the field current decreases and the generator output goes down. If the resistance of the disks decreases, the field current increases and generator output goes up. As seen in *Figure 6-25*, a voltage coil is installed in parallel with the generator output leads. The voltage coil acts like an electromagnet that increases or decrease strength as generator output voltage changes. The magnetism of the voltage coil controls the pressure on the carbon stack. The pressure on the carbon stack controls the resistance of the carbon; the resistance of the carbon controls field current and the field current controls generator output.

Carbon pile regulators require regular maintenance to ensure accurate voltage regulation; therefore, most have been replaced on aircraft with more modern systems.

THREE UNIT REGULATORS

The three unit regulator used with DC generator systems is made of three distinct units. Each of these units performs a specific function vital to correct electrical system operation. A typical three unit regulator consists of three relays mounted in a single housing. Each of the three relays monitors generator outputs and opens or closes the relay contact points according to system needs. A typical three unit regulator is shown in *Figure 6-26*.

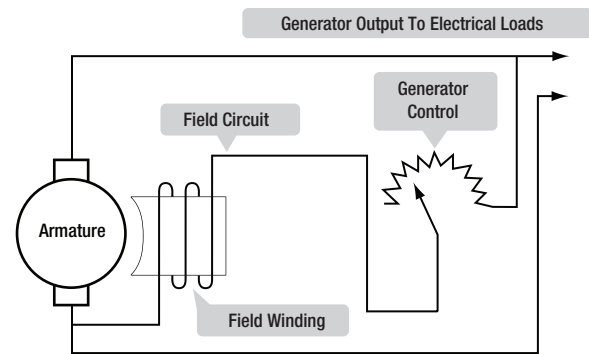


Figure 6-24. Voltage regulator for low-output generator.

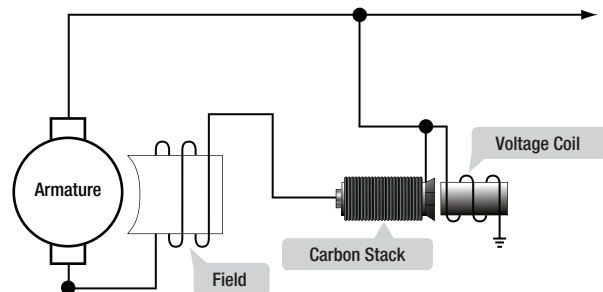


Figure 6-25. Carbon pile regulator.

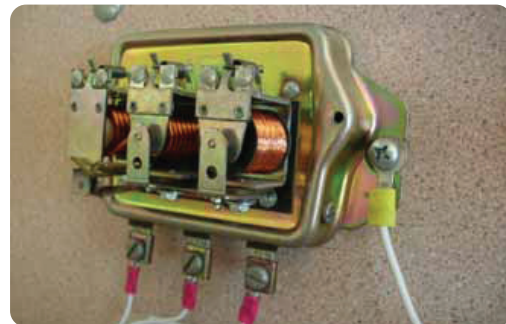


Figure 6-26. The three relays found on this regulator are used to regulate voltage, limit current, and prevent reverse current flow.

The voltage regulator section of the three unit regulator is used to control generator output voltage. The voltage regulator monitors generator output and controls the generator field current as needed. If the regulator senses that system voltage is too high, the relay points open and the current in the field circuit must travel through a resistor. This resistor lowers field current and therefore lowers generator output. Remember, generator output goes down whenever generator field current goes down.

As seen in *Figure 6-27*, the voltage coil is connected in parallel with the generator output, and it therefore measures the voltage of the system. If voltage gets beyond a predetermined limit, the voltage coil becomes

a strong magnet and opens the contact points. If the contact points are open, field current must travel through a resistor and therefore field current goes down. The dotted arrow shows the current flow through the voltage regulator when the relay points are open.

Since this voltage regulator has only two positions (points open and points closed), the unit must constantly be in adjustment to maintain accurate voltage control. During normal system operation, the points are opening and closing at regular intervals. The points are in effect vibrating. This type of regulator is sometimes referred to as a vibrating type regulator. As the points vibrate, the field current raises and lowers and the field magnetism averages to a level that maintains the correct generator output voltage. If the system requires more generator output, the points remain closed longer and vice versa.

There is a current limiter section of the three unit regulator. It is designed to limit generator output current. This unit contains a relay with a coil wired in series with respect to the generator output. As seen in **Figure 6-28**, all the generator output current must travel through the current coil of the relay. This creates a relay that is sensitive to the current output of the generator. That is, if generator output current increases, the relay points open and vice versa. The dotted line shows the current flow to the generator field when the current limiter points are open. It should be noted that, unlike the voltage regulator relay, the current limiter is typically closed during normal flight. Only during extreme current loads must the current limiter points open; at that time, field current is lowered and generator output is kept within limits.

The third unit of a three unit regulator is used to prevent current from leaving the battery and feeding the generator. This type of current flow would discharge the battery and is opposite of normal operation. It can be thought of as a reverse current situation and the third unit is known as reverse current relay. The simple reverse current relay shown in **Figure 6-29** contains both a voltage coil and a current coil.

The voltage coil is wired in parallel to the generator output and is energized any time the generator output reaches its operational voltage. As the voltage coil is energized, the contact points close and the current is then allowed to flow to the aircraft electrical loads,

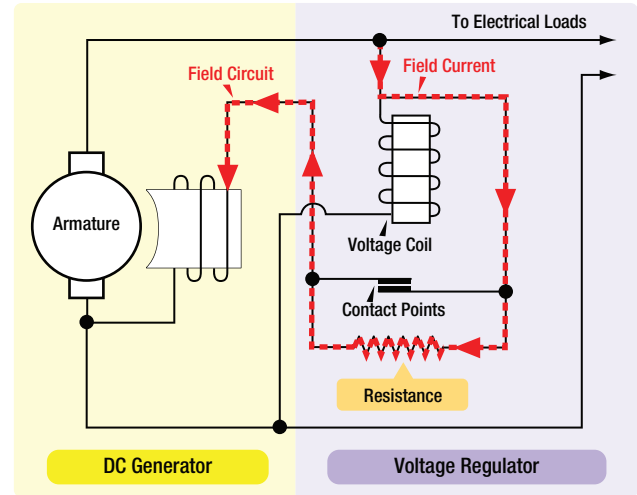


Figure 6-27. Voltage regulator.

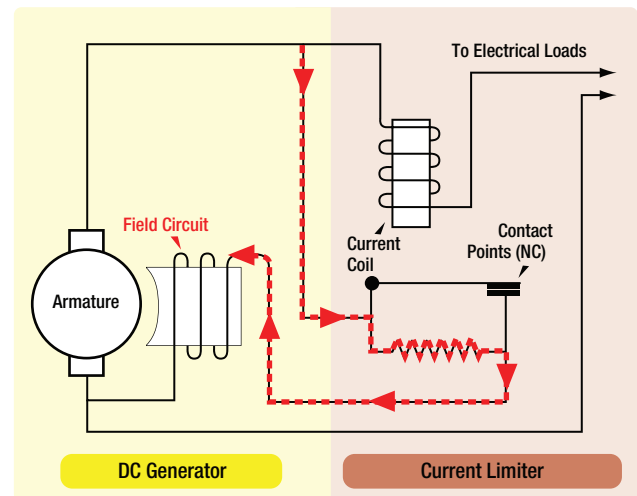


Figure 6-28. Current limiter.

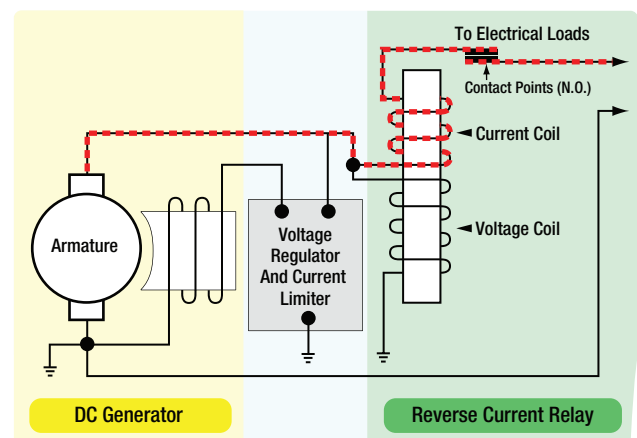


Figure 6-29. Reverse-current relay.

as shown by the dotted lines. The diagram shows the reverse current relay in its normal operating position; the points are closed and current is flowing from the

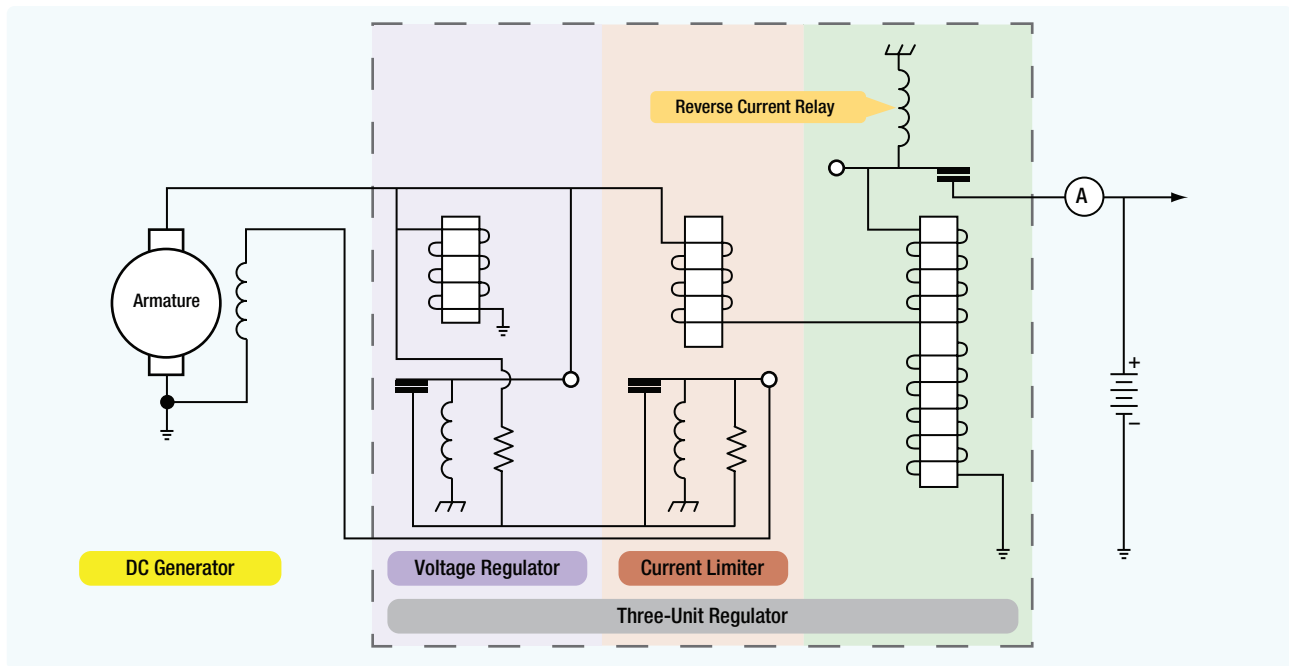


Figure 6-30. Three-unit regulator for variable speed generators.

generator to the aircraft electrical loads. As current flows to the loads, the current coil is energized and the points remain closed. If there is no generator output due to a system failure, the contact points open because magnetism in the relay is lost. With the contact points open, the generator is automatically disconnected from the aircraft electrical system, which prevents reverse flow from the load bus to the generator. A typical three unit regulator for aircraft generators is shown in **Figure 6-30**.

As seen in **Figure 6-30**, all three units of the regulator work together to control generator output. The regulator monitors generator output and controls power to the aircraft loads as needed for flight variables. Note that the vibrating regulator just described was simplified for explanation purposes. A typical vibrating regulator found on an aircraft would probably be more complex.

POWER DISTRIBUTION

POWER DISTRIBUTION ON SMALL MULTI-ENGINE AIRCRAFT

The power distribution systems found on modern multi-engine aircraft contain several distribution points (busses) and a variety of control and protection components to ensure the reliability of electrical power. As aircraft employ more electronics to perform various tasks, the electrical power systems becomes more complex and more reliable. One means to increase

reliability is to ensure more than one power source can be used to power any given load. Another important design concept is to supply critical electrical loads from more than one bus. Twin engine aircraft, such as a typical corporate jet or commuter aircraft, have two DC generators; they also have multiple distribution busses fed from each generator. **Figure 6-31** shows a simplified diagram of the power distribution system for a twin-engine turboprop aircraft.

This aircraft contains two starter generator units used to start the engines and generate DC electrical power. The system is typically defined as a split bus power distribution system since there is a left and right generator bus that splits (shares) the electrical loads by connecting to each sub-bus through a diode and current limiter. The generators are operated in parallel and equally carry the loads.

The primary power supplied for this aircraft is DC, although small amounts of AC are supplied by two inverters. The aircraft diagram shows the AC power distribution at the top and mid left side of the diagram. One inverter is used for main AC power and the second operated in standby and ready as a backup. Both inverters produce 26-volt AC and 115-volt AC. There is an inverter select relay operated by a pilot controlled switch used to choose which inverter is active.

The hot battery bus (right side of **Figure 6-31**) shows a direct connection to the aircraft battery. This bus is always hot if there is a charged battery in the aircraft. Items powered by this bus may include some basics like the entry door lighting and the aircraft clock, which should always have power available. Other items on this bus would be critical to flight safety, such as fire

extinguishers, fuel shut offs, and fuel pumps. During a massive system failure, the hot battery bus is the last bus on the aircraft that should fail.

If the battery switch is closed and the battery relay activated, battery power is connected to the main battery bus and the isolation bus. The main battery bus

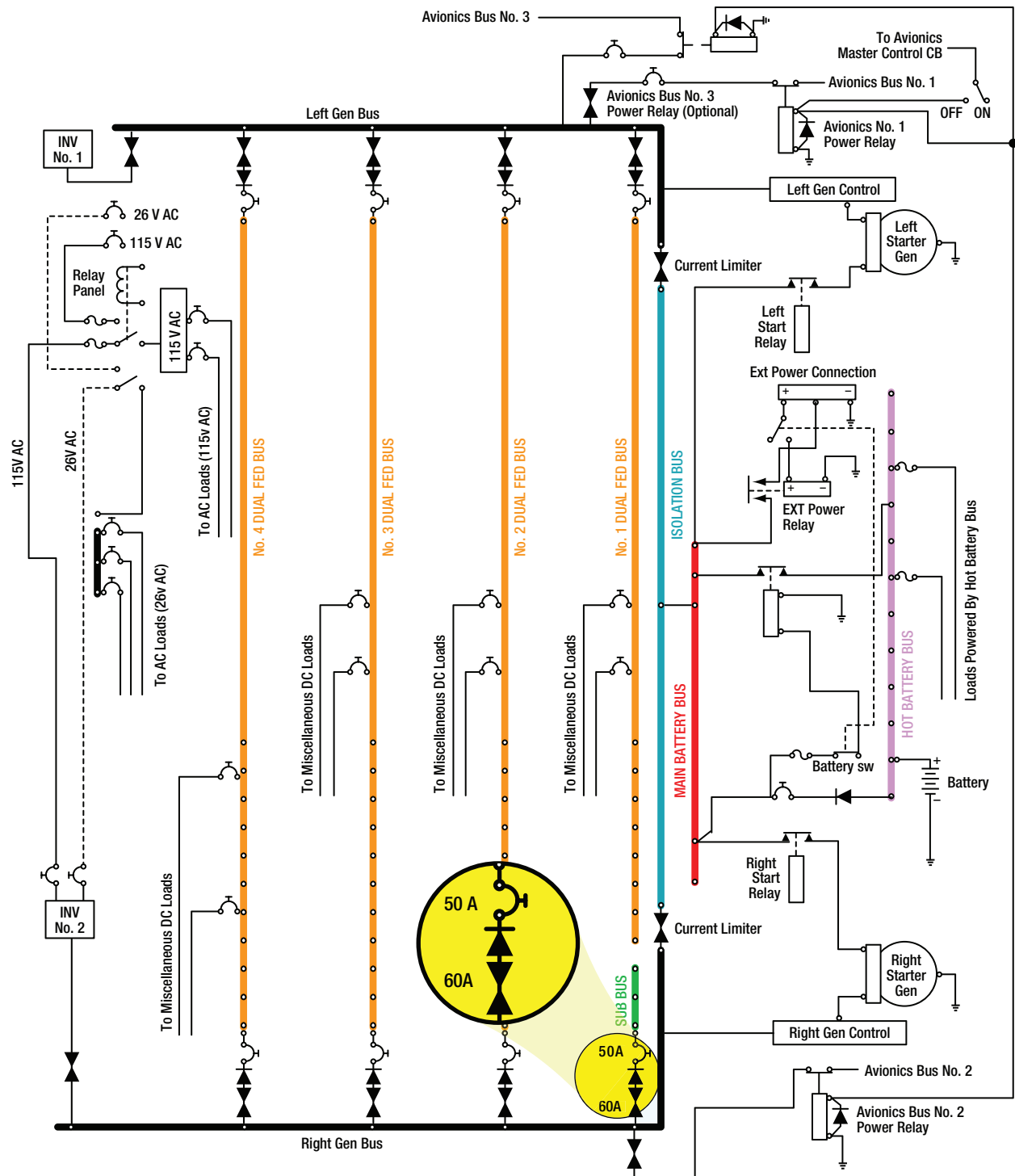


Figure 6-31. Diagram of the power distribution system for a twin-engine turboprop aircraft.

carries current for engine starts and external power. So the main battery bus must be large enough to carry the heaviest current loads of the aircraft. It is logical to place this bus as close as practical to the battery and starters and to ensure the bus is well protected from shorts to ground.

The isolation bus connects to the left and right busses and receives power whenever the main battery bus is energized. The isolation bus connects output of the left and right generators in parallel. The output of the two generators is then sent to the loads through additional busses. The generator busses are connected to the isolation bus through a fuse known as a current limiter. Current limiters are high amperage fuses that isolate busses if a short circuit occurs. There are several current limiters used in this system for protection between busses. As can be seen in **Figure 6-31**, a current limiter symbol looks like two triangles pointed toward each other. The current limiter between the isolation bus and the main generator busses are rated at 325 amps and can only be replaced on the ground. Most current limiters are designed for ground replacement only and only after the malfunction that caused the excess current draw is repaired.

The left and right DC generators are connected to their respective main generator busses. Each generator feeds its respective bus, and since the busses are connected under normal circumstances, the generators operate in parallel. Both generators feed all loads together. If one generator fails or a current limiter opens, the generators can operate independently. This design allows for redundancy in the event of failure and provides battery backup in the event of a dual generator failure.

In the center of **Figure 6-31** are four dual-feed electrical busses. These busses are considered dual-feed since they receive power from both the left and right generator busses. If a fault occurs, either generator bus can power any or all loads on a dual-feed bus. During the design phase of the aircraft, the electrical loads must be evenly

distributed between each of the dual-feed busses. It is also important to power redundant systems from different busses. For example, the pilot's windshield heat would be powered by a different bus from the one that powers the copilot's windshield heat. If one bus fails, at least one windshield heat continues to work properly, and the aircraft can be landed safely in icing conditions.

Notice that the dual-feed busses are connected to the main generator busses through both a current limiter and a diode. Remember, a diode allows current flow in only one direction. (**Figure 6-32**)

The current can flow from the generator bus to the dual-feed bus, but the current cannot flow from the dual fed bus to the main generator bus. The diode is placed in the circuit so the main bus must be more positive than the sub bus for current flow. This circuit also contains a current limiter and a circuit breaker. The circuit breaker is located on the flight deck and can be reset by the pilot. The current limiter can only be replaced on the ground by a technician. The circuit breaker is rated at a slightly lower current value than the current limiter; therefore, the circuit breaker should open if a current overload exists. If the circuit breaker fails to open, the current limiter provides backup protection and disconnects the circuit.

POWER DISTRIBUTION ON LARGE AIRCRAFT

SPLIT BUS SYSTEMS

Modern twin-engine aircraft, such as the Boeing 737, 757, 777, Airbus A-300, A-320, and A-310, employ a split bus power distribution system. During normal conditions, each engine-driven AC generator powers only one main AC bus. The busses are kept split from each other, and two generators can never power the same bus simultaneously. This is very important since the generator output current is not phase regulated. (If two out-of-phase generators were connected to the same bus, damage to the system would occur.) The split bus system

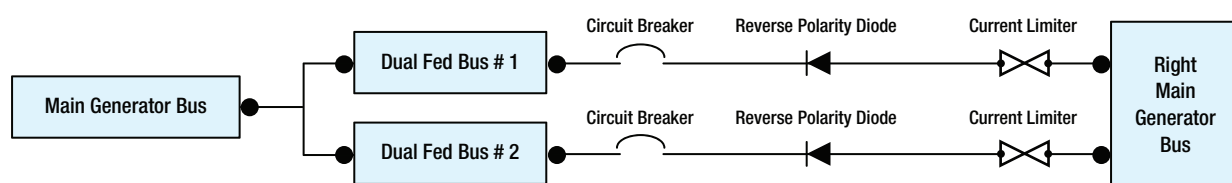


Figure 6-32. Dual-feed bus system.

does allow both engine driven generators to power any given bus, but not at the same time. Generators must remain isolated from each other to avoid damage. The GCUs and BPCU ensures proper generator operation and power distribution.

On all modern split bus systems, the APU can be started and operated during flight. This allows the APU generator to provide back-up power in the event of a main generator failure. A fourth emergency generator powered by the ram air turbine is also available if the other generators fail.

The four AC generators are shown at the bottom of **Figure 6-33**. These generators are connected to their respective busses through the generator breakers (GB's). For example, generator 1 sends current through GB1 to AC bus 1. AC bus 1 feeds a variety of primary electrical loads, and also feeds sub-busses that in turn power additional loads.

With both generators operating and all systems normal, AC bus 1 and AC bus 2 are kept isolated. Typically during flight, the auxiliary power breaker (APB) (bottom center of **Figure 6-33**) would be open and the APU generator off; the emergency generator (bottom right) would also be off and disconnected. If generator one should fail, the following happens:

1. The GB 1 is opened by the GCU to disconnect the failed generator.
2. The BPCU closes BTB 1 and BTB 2. This supplies AC power to AC bus 1 from generator 2.
3. The pilots start the APU and connect the APU generator. At that time, the BPCU and GCUs move the appropriate BTBs to correctly configure the system so the APU powers bus 1 and generator 2 powers bus 2. Once again, two AC generators operate independently to power AC bus 1 and 2.

If all generators fail, AC is also available through the static inverter (center of **Figure 6-33**). The inverter is powered from the hot battery bus and used for essential

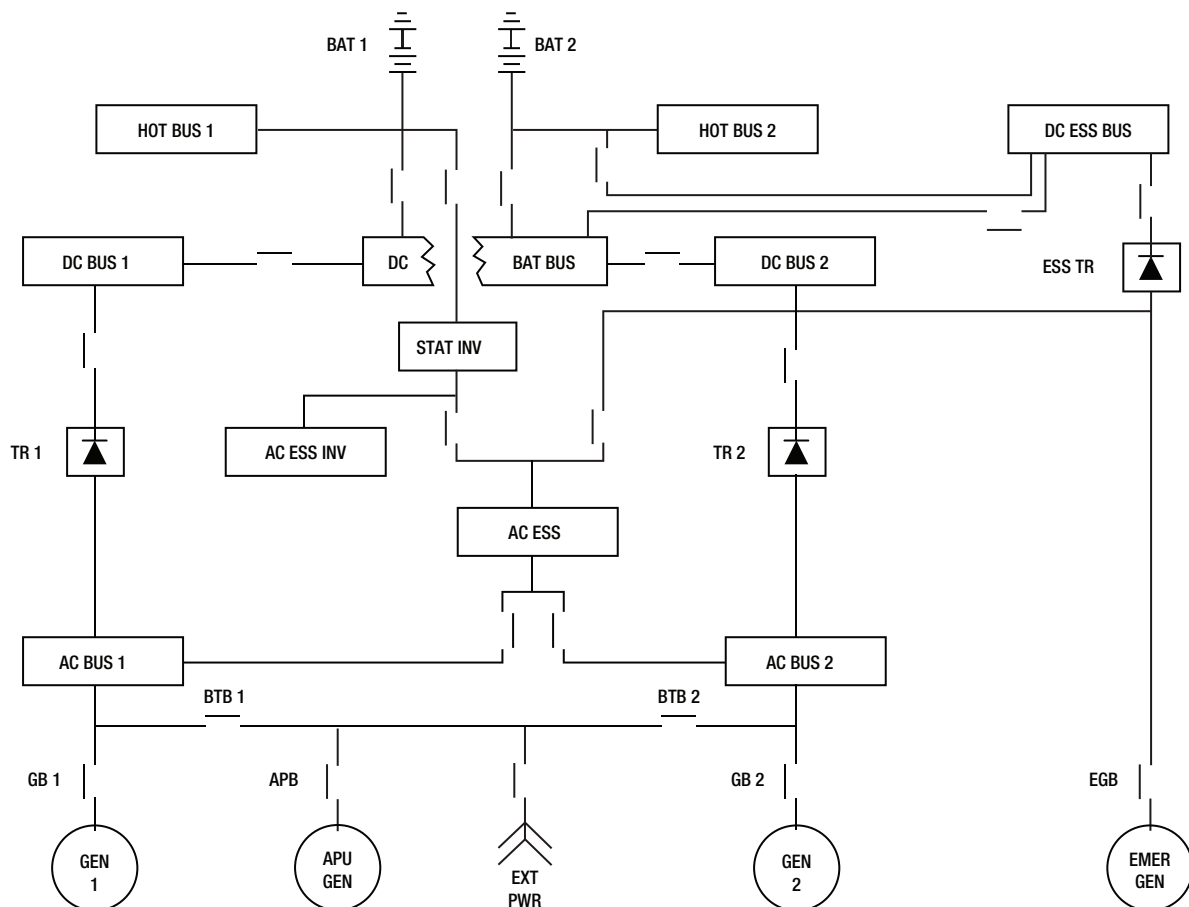


Figure 6-33. Schematic of split-bus power distribution system.

AC loads if all AC generators fail. Of course, the GCUs and BPCU take the appropriate actions to disconnect defective units and continue to feed essential AC loads using inverter power.

To produce DC power, AC bus 1 sends current to its transformer rectifier (TR), TR 1 (center left of **Figure 6-33**). The TR unit is used to change AC to DC. The TR contains a transformer to step down the voltage from 115-volt AC to 26-volt AC and a rectifier to change the 26-volt AC to 26-volt DC. The output of the TR is therefore compatible with the aircraft battery at 26-volt DC. Since DC power is not phase sensitive, the DC busses are connected during normal operation.

In the event of a bus problem, the BPCU may isolate one or more DC busses to ensure correct distribution of DC power. This aircraft contains two batteries that are used to supply emergency DC power.

PARALLEL BUS SYSTEMS

Multi-engine aircraft, such as the Boeing 727, MD-11, and the early Boeing 747, employ a parallel power distribution system. During normal flight conditions, all engine-driven generators connect together and power the AC loads. In this configuration, the generators are operated in parallel; hence the name parallel power distribution system. In a parallel system, all generator output current must be phase regulated. Before generators are connected to the same bus, their output frequency must be adjusted to ensure the AC output reaches the positive and negative peaks simultaneously. During the flight, generators must maintain this in-phase condition for proper operation.

One advantage of parallel systems is that in the event of a generator failure, the busses are already connected and the defective generator need only be isolated from the system. A paralleling bus, or synchronizing bus, is used to connect the generators during flight. The synchronizing bus is often referred to as the sync bus. Most of these systems are less automated and require that flight crew monitor systems and manually control bus contactors. BTBs are operated by the flight crew through the electrical control panel and used to connect all necessary busses. GBs are used to connect and disconnect the generators.

Figure 6-34 shows a simplified parallel power distribution system. This aircraft employs three main-engine driven generators and one APU generator. The APU (bottom right) is not operational in flight and cannot provide backup power. The APU generator is for ground operations only. The three main generators (**bottom of Figure 6-34**) are connected to their respective AC bus through GBs one, two, and three. The AC busses are connected to the sync bus through three BTBs. In this manner, all three generators share the entire AC electrical loads. Keep in mind, all generators connected to the sync bus must be in phase. If a generator fails, the flight crew would simply isolate the defective generator and the flight would continue without interruption.

The number one and two DC busses (**Figure 6-34 top left**) are used to feed the DC electrical loads of the aircraft. DC bus 1 receives power from AC bus 1 through TR1. DC bus 2 is fed in a similar manner from AC bus 2. The DC busses also connect to the battery bus and eventually to the battery. The essential DC bus (top left) can be fed from DC bus 1 or the essential TR. A diode prevents the essential DC bus from powering DC bus 1. The essential DC bus receives power from the essential TR, which receives power from the essential AC bus. This provides an extra layer of redundancy since the essential AC bus can be isolated and fed from any main generator. **Figure 6-34** shows generator 3 powering the essential AC bus.

SPLIT PARALLEL SYSTEMS

A split parallel bus basically employs the best of both split bus and the parallel-bus systems. The split parallel system is found on the Boeing 747-400 and contains four generators driven by the main engines and two APU-driven generators. The system can operate with all generators in parallel, or the generators can be operated independently as in a split bus system. During a normal flight, all four engine-driven generators are operated in parallel. The system is operated in split bus mode only under certain failure conditions or when using external power. The Boeing 747-400 split parallel system is computer controlled using four GCU and two BPCU. There is one GCU controlling each generator; BPCU 1 controls the left side bus power distribution, and BPCU 2 controls the right side bus power. The GCUs and BPCUs operate similarly to those previously discussed under the split bus system.

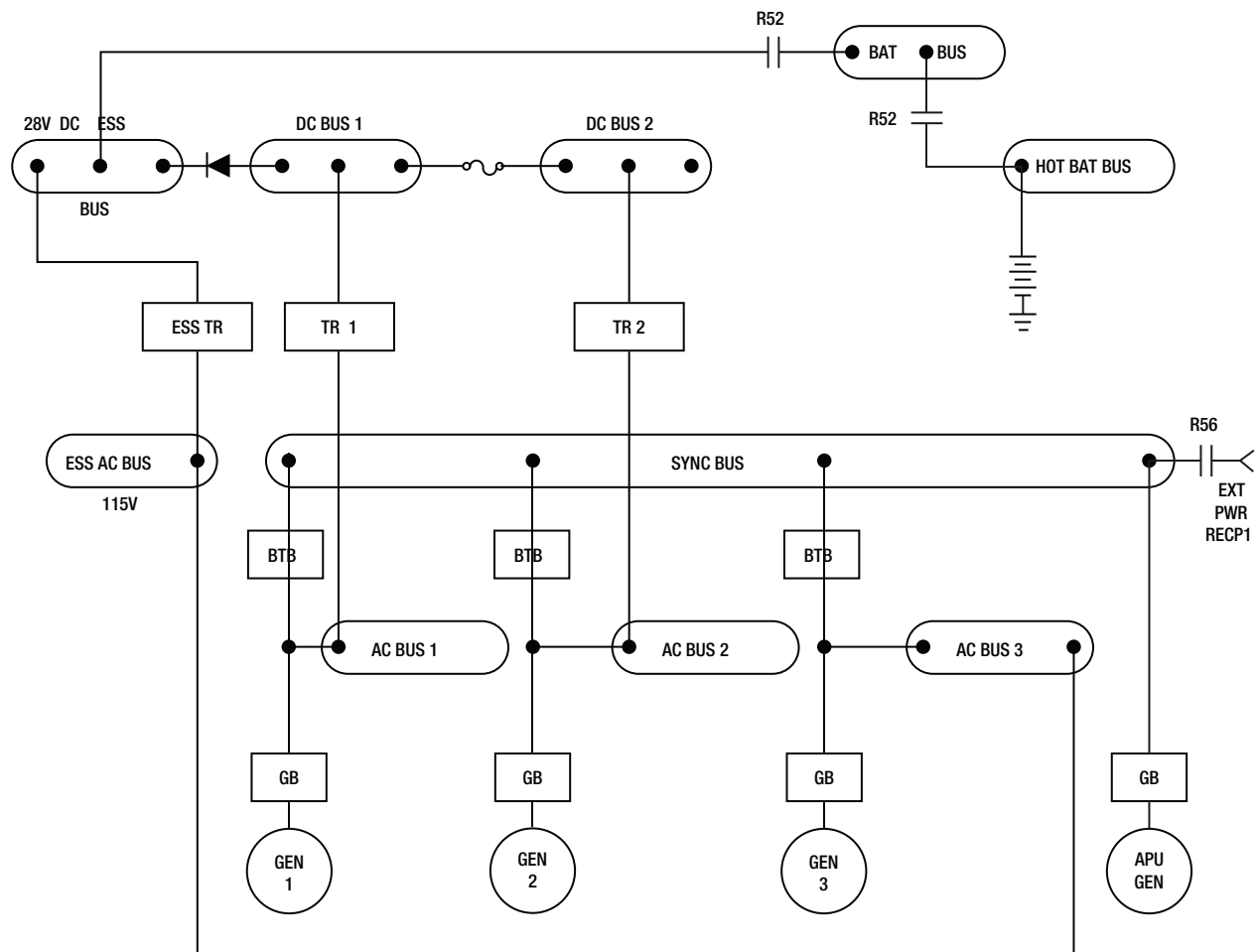


Figure 6-34. Parallel power distribution system.

Figure 6-35 shows a simplified split parallel power distribution system. The main generators (*top of Figure 6-35*) are driven by the main turbine engines. Each generator is connected to its load bus through a generator control breaker (GCB). The generator control unit closes the GCB when the pilot calls for generator power and all systems are operating normally. Each load bus is connected to various electrical systems and additional sub-busses. The BTB are controlled by the BPCU and connect each load bus to the left and right sync bus. A split systems breaker (SSB) is used to connect the left and right sync busses and is closed during a normal flight. With the SSB, GCBs, and BTBs, in the closed position the generators operate in parallel. When operating in parallel, all generators must be in phase.

If the aircraft electrical system experiences a malfunction, the control units make the appropriate adjustments to ensure all necessary loads receive electrical power. For example, if generator 1 fails, GCU 1 detects the fault and

command GCB 1 to open. With GCB 1 open, load bus 1 now feeds from the sync bus and the three operating generators. In another example, if load bus 4 should short to ground, BPCU 4 opens the GCB 4 and BTB 4. This isolates the shorted bus (load bus 4). All loads on the shorted bus are no longer powered, and generator 4 is no longer available. However, with three remaining generators operational, the flight continues safely.

As with all large aircraft, the Boeing 747-400 contains a DC power distribution system. The DC system is used for battery and emergency operations. The DC system is similar to those previously discussed, powered by TR units. The TRs are connected to the AC busses and convert AC into 26-volt DC.

The DC power systems are the final backups in the event of a catastrophic electrical failure. The systems most critical to fly the aircraft can typically receive power from the battery. This aircraft also contains two static inverters to provide emergency AC power when needed.

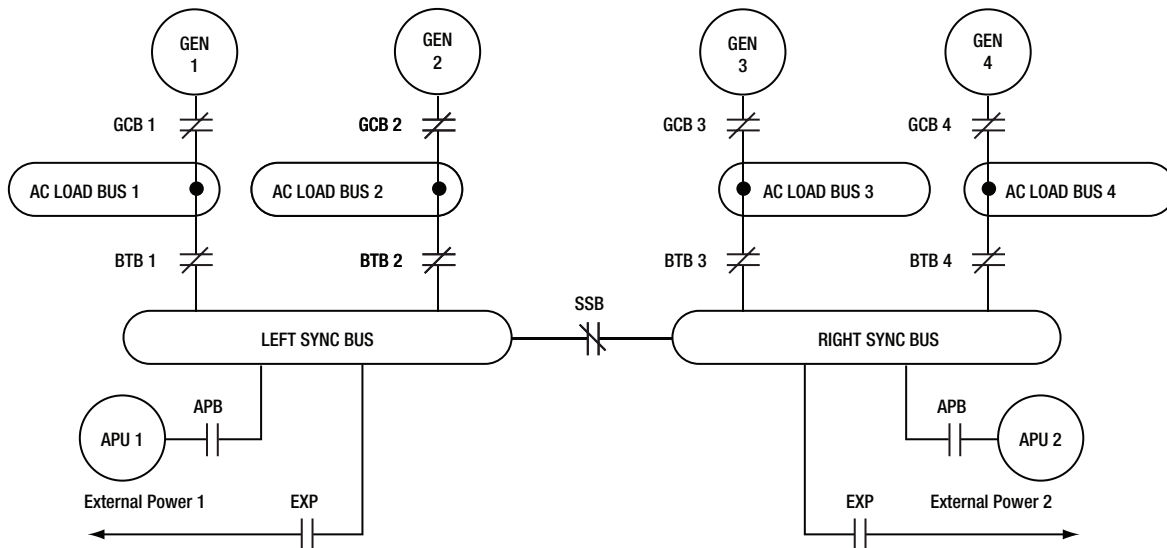


Figure 6-35. Split-parallel distribution system.

INVERTERS, TRANSFORMERS AND RECTIFIERS

INVERTERS

A modern inverter is a solid-state device that converts DC power into AC power. The electronic circuitry within an inverter is quite complex; however, for an aircraft technician's purposes, the inverter is simply a device that uses DC power, then feeds power to an AC distribution bus. Many inverters supply both 26-volt AC, as well as 115-volt AC. The aircraft can be designed to use either voltage or both simultaneously. If both voltages are used, the power must be distributed on separate 26- and 115-volt AC busses. Typical AC inverter output frequency is 400 cycles per second (cps). There are two basic types of inverters: the rotary and the static. Either type can be single phase or multiphase. The multiphase inverter is lighter for the same power rating than the single phase, but there are complications in distributing multiphase power and in keeping the loads balanced. Most modern aircraft use solid state static inverters rather than the rotary type.

ROTARY INVERTERS

There are many sizes, types, and configurations of rotary inverters. Such inverters are essentially AC generators and DC motors in one housing. The generator field, or armature, and the motor field, or armature, are mounted on a common shaft that will rotate within the housing. One common type of rotary inverter is the permanent magnet inverter.

Permanent Magnet Rotary Inverter

A permanent magnet inverter is composed of a DC motor and a permanent magnet AC generator assembly. Each has a separate stator mounted within a common housing. The motor armature is mounted on a rotor and connected to the DC supply through a commutator and brush assembly. The motor field windings are mounted on the housing and connected directly to the DC supply. A permanent magnet rotor is mounted at the opposite end of the same shaft as the motor armature, and the stator windings are mounted on the housing, allowing AC to be taken from the inverter without the use of brushes. **Figure 6-36** shows an internal wiring diagram for this type of rotary inverter. The generator rotor has six poles, magnetized to provide alternate north and south poles about its circumference.

When the motor field and armature are excited, the rotor will begin to turn. As the rotor turns, the permanent magnet will rotate within the AC stator coils, and the magnetic flux developed by the permanent magnets will be cut by the conductors in the AC stator coils. An AC voltage will be produced in the windings whose polarity will change as each pole passes the windings. This type inverter may be made multiphase by placing more AC stator coils in the housing in order to shift the phase the proper amount in each coil.

As the name of the rotary inverter indicates, it has a revolving armature in the AC generator section. The illustration in **Figure 6-37** shows the diagram of a revolving armature, three phase inverter.

The DC motor in this inverter is a four pole, compound wound motor. The four field coils consist of many turns of fine wire, with a few turns of heavy wire placed on top. The fine wire is the shunt field, connected to the DC source through a filter and to ground through a centrifugal governor. The heavy wire is the series field, which is connected in series with the motor armature. The centrifugal governor controls the speed by shunting a resistor that is in series with the shunt field when the motor reaches a certain speed.

The alternator is a three phase, four-pole, star-connected AC generator. The DC input is supplied to the generator field coils and connected to ground through a voltage regulator. The output is taken off the armature through three slip rings to provide three phase power.

The inverter would be a single-phase inverter if it had a single armature winding and one slip ring. The frequency of this type unit is determined by the speed of the motor and the number of generator poles.

Inductor Type Rotary Inverter

Inductor type inverters use a rotor made of soft iron laminations with grooves cut laterally across the surface to provide poles that correspond to the number of stator poles, as illustrated in **Figure 6-38**. The field coils are wound on one set of stationary poles and the AC armature coils on the other set of stationary poles. When DC is applied to the field coils, a magnetic field is produced. The rotor turns within the field coils and, as the poles on the rotor align with the stationary poles, a low reluctance path for flux is established from the field pole through the rotor poles to the AC armature pole and through the housing back to the field pole. In this circumstance, there will be a large amount of magnetic flux linking the AC coils.

When the rotor poles are between the stationary poles, there is a high reluctance path for flux, consisting mainly of air; then, there will be a small amount of magnetic flux linking the AC coils. This increase and decrease in flux density in the stator induces an alternating current in the AC coils.

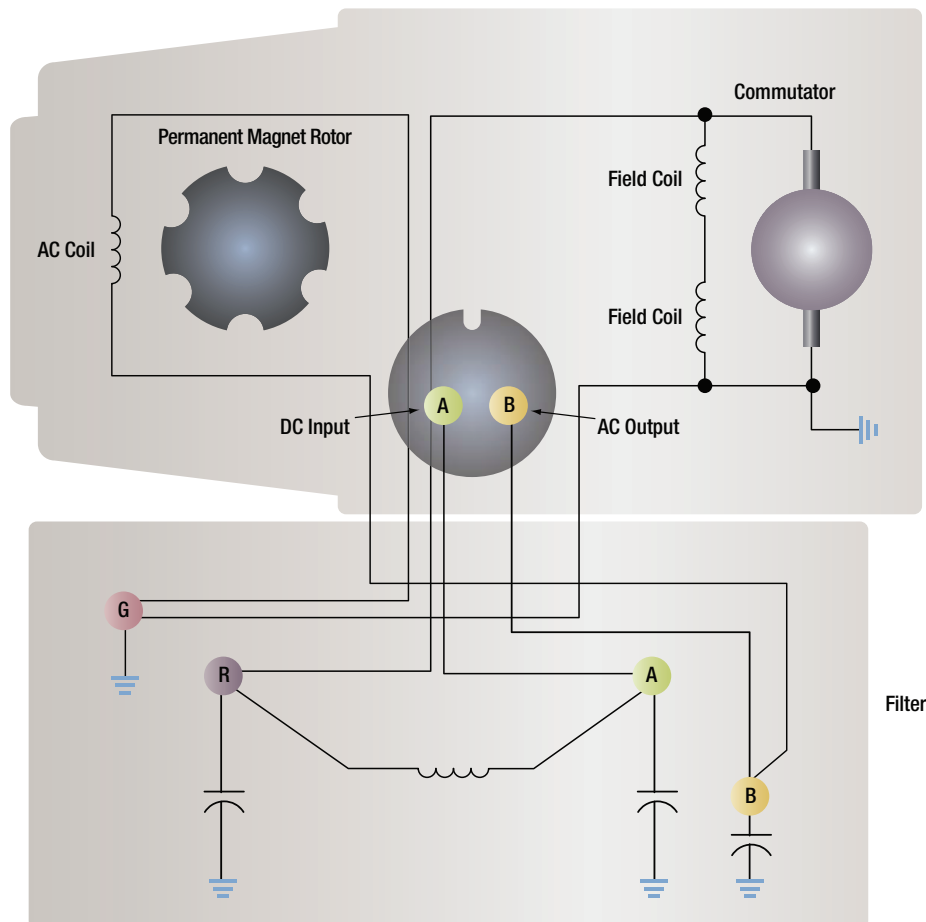


Figure 6-36. Internal wiring diagram of single-phase permanent magnet rotary inverter.

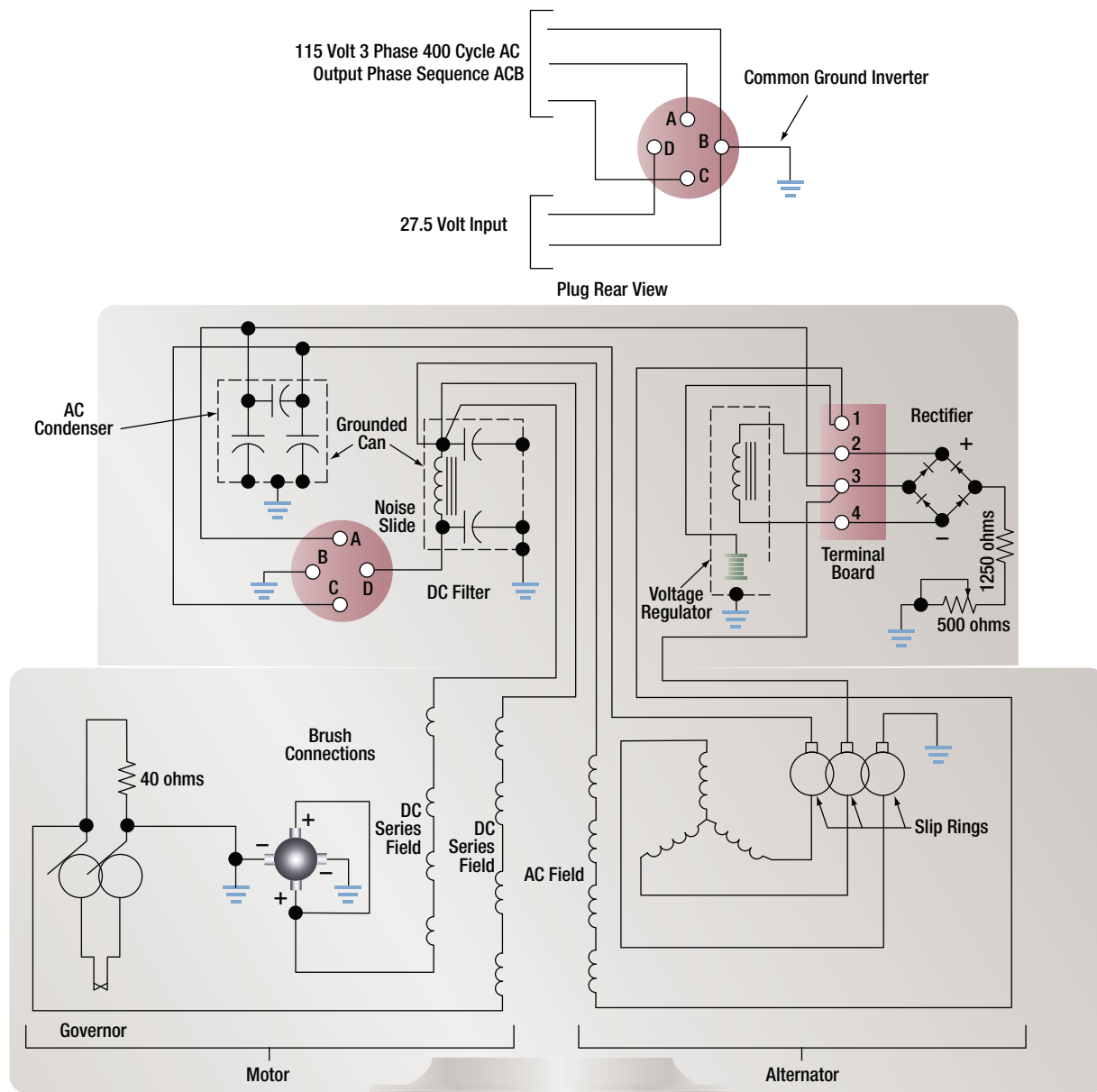


Figure 6-37. Internal wiring diagram of three-phase, revolving armature.

The number of poles and the speed of the motor determine the frequency of this type of inverter. The DC stator field current controls the voltage. A cutaway view of an inductor type rotary inverter is shown in *Figure 6-39*.

Figure 6-40 is a simplified diagram of a typical aircraft AC power distribution system, utilizing a main and a standby rotary inverter system.

STATIC INVERTERS

In many applications where continuous DC voltage must be converted to alternating voltage, static inverters are used in place of rotary inverters or motor generator sets.

The rapid progress made by the semiconductor industry is extending the range of applications of such equipment into voltage and power ranges that would have been impractical a few years ago. Some such applications are power supplies for frequency sensitive military and commercial AC equipment, aircraft emergency AC systems, and conversion of wide frequency range power to precise frequency power.

The use of static inverters in small aircraft also has increased rapidly in the last few years, and the technology has advanced to the point that static inverters are available for any requirement filled by rotary inverters. For example, 250 VA emergency AC supplies operated

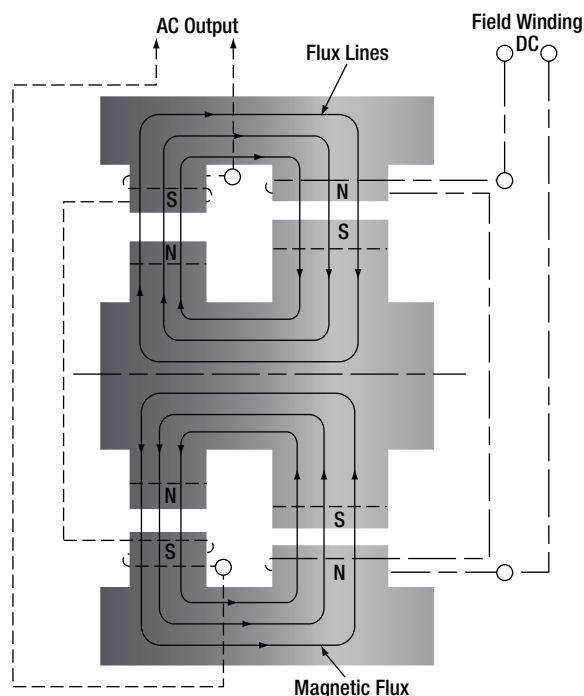


Figure 6-38. Diagram of basic inductor-type inverter.

from aircraft batteries are in production, as are 2 500 VA main AC supplies operated from a varying frequency generator supply. This type of equipment has certain advantages for aircraft applications, particularly the absence of moving parts and the adaptability to conduction cooling.

Static inverters, referred to as solid-state inverters, are manufactured in a wide range of types and models, which can be classified by the shape of the AC output waveform and the power output capabilities. One of the most commonly used static inverters produces a regulated sine wave output. A block diagram of a typical regulated sine wave static inverter is shown in *Figure 6-41*.

This inverter converts a low DC voltage into higher AC voltage. The AC output voltage is held to a very small voltage tolerance, a typical variation of less than 1 percent with a full input load change. Output taps are normally provided to permit selection of various

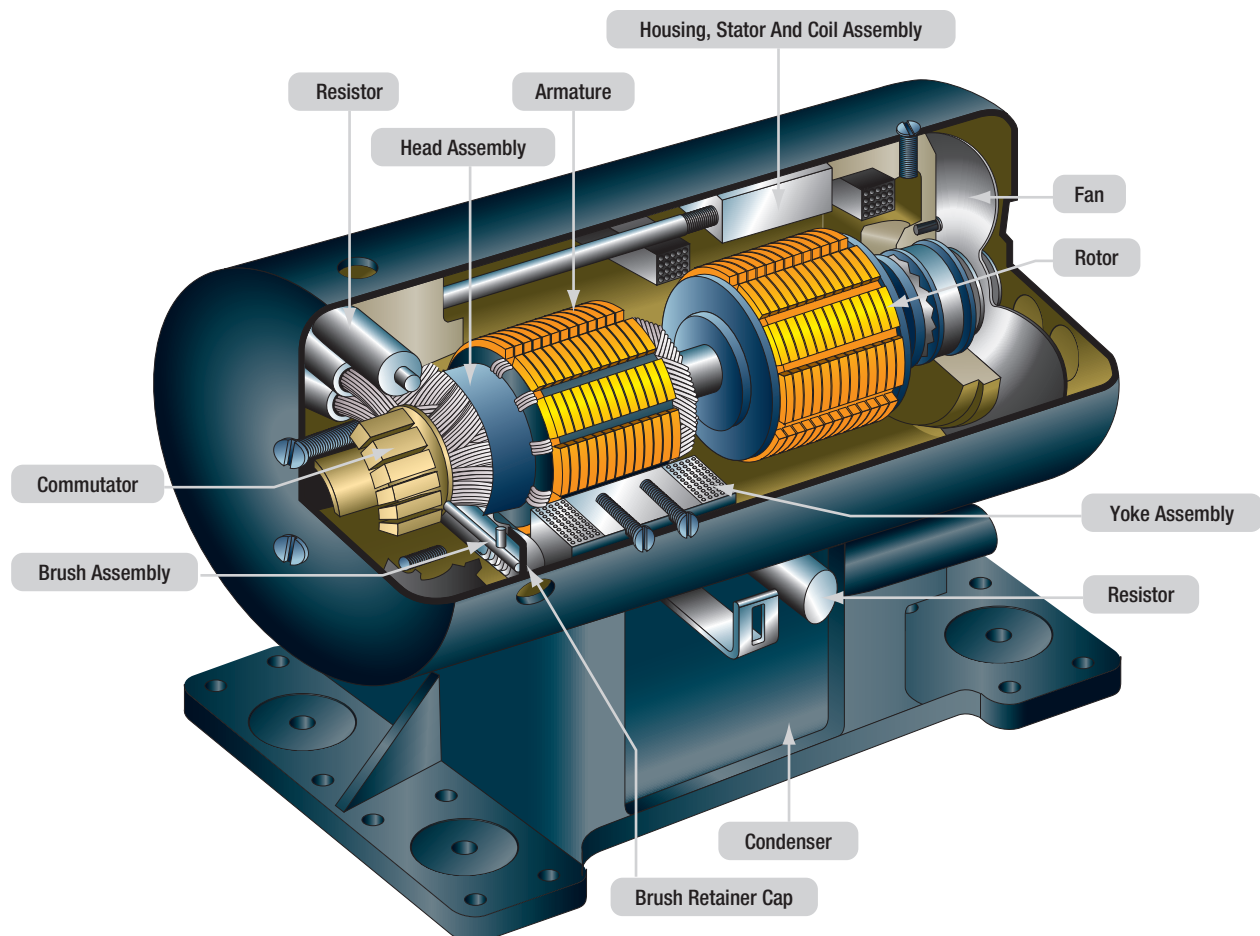
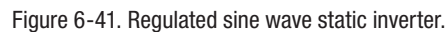


Figure 6-39. Cutaway view of inductor-type rotary inverter.



Variations of this type of static inverter are available, many of which provide a square wave output. Since static inverters use solid-state components, they are considerably smaller, more compact, and much lighter in weight than rotary inverters. Depending on the output

1. High efficiency.
2. Low maintenance, long life.
3. No warmup period required.
4. Capable of starting under load.
5. Extremely quiet operation.
6. Fast response to load changes.

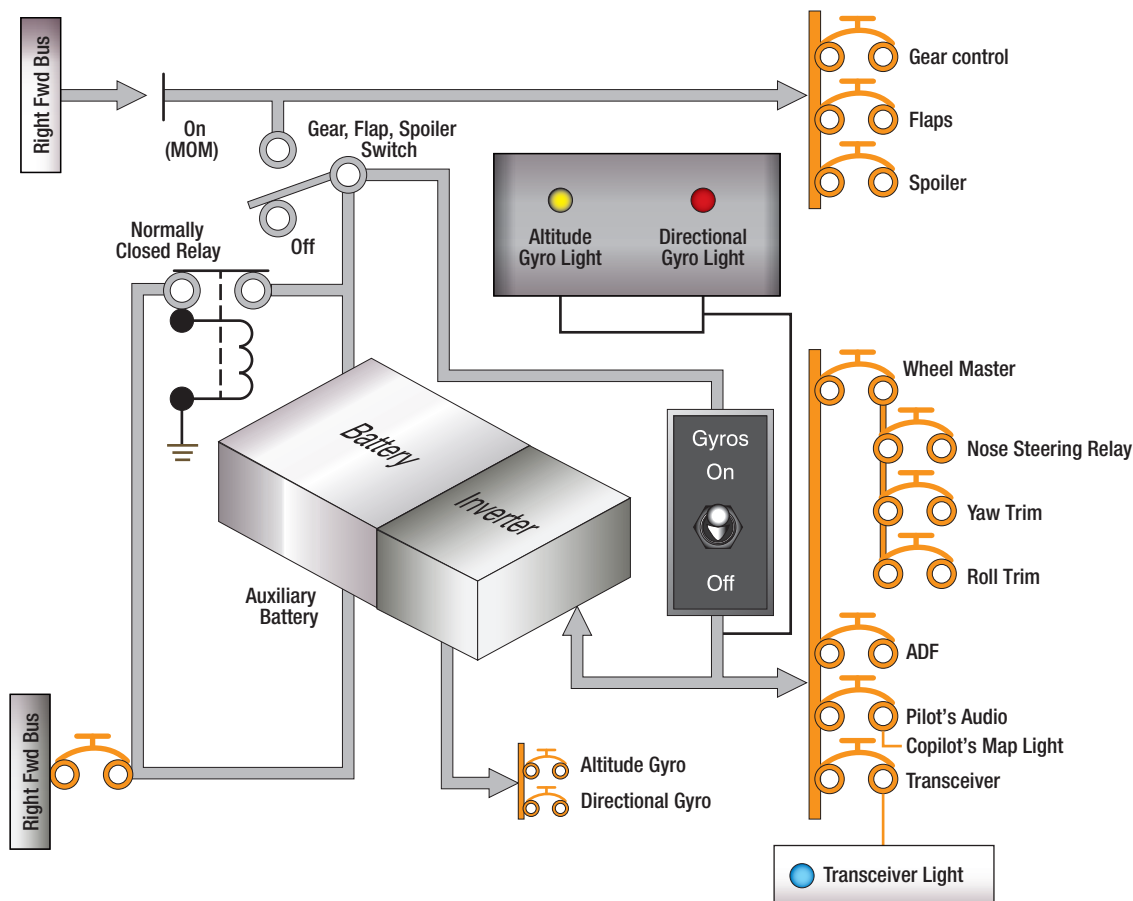


Figure 6-42. Auxiliary battery system using static inverter.

Static inverters are commonly used to provide power for such frequency sensitive instruments as the attitude gyro and directional gyro. They also provide power for autosyn and magnesyn indicators and transmitters, rate gyros, radar, and other airborne applications. **Figure 6-42** is a schematic of a typical small jet aircraft auxiliary battery system. It shows the battery as input to the inverter, and the output inverter circuits to various subsystems.

TRANSFORMERS

A transformer changes electrical energy of a given voltage into electrical energy at a different voltage level. It consists of two coils that are not electrically connected, but are arranged so that the magnetic field surrounding one coil cuts through the other coil. When an alternating voltage is applied to (across) one coil, the varying magnetic field set up around that coil creates an alternating voltage in the other coil by mutual induction. A transformer can also be used with pulsating DC, but a pure DC voltage cannot be used, since only a varying voltage creates the varying magnetic field that is the basis of the mutual induction process.

A transformer consists of three basic parts. (**Figure 6-43**) These are an iron core which provides a circuit of low reluctance for magnetic lines of force, a primary winding which receives the electrical energy from the source of applied voltage, and a secondary winding which receives electrical energy by induction from the primary coil.

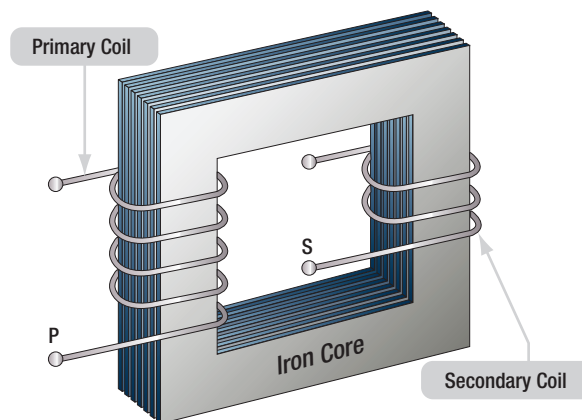


Figure 6-43. An iron-core transformer.

The primary and secondary of this closed core transformer are wound on a closed core to obtain maximum inductive effect between the two coils. There are two classes of transformers: (1) voltage transformers used for stepping up or stepping down voltages, and (2) current transformers used in instrument circuits.

In voltage transformers, the primary coils are connected in parallel across the supply voltage as shown in **Figure 6-44A**. The primary windings of current transformers are connected in series in the primary circuit (**Figure 6-44B**). Of the two types, the voltage transformer is the more common.

There are many types of voltage transformers. Most of these are either step-up or step-down transformers. The factor that determines whether a transformer is a step-up, or step-down type is the "turns" ratio. The turns ratio is the ratio of the number of turns in the primary winding to the number of turns in the secondary winding.

For example, the turns ratio of the step-down transformer shown in **Figure 6-45A** is 5 to 1, since there are five times as many turns in the primary as in the secondary. The step-up transformer shown in **Figure 6-45B** has a 1 to 4 turns ratio.

The ratio of the transformer input voltage to the output voltage is the same as the turns ratio if the transformer is 100 percent efficient. Thus, when 10 volts are applied to the primary of the transformer shown in **Figure 6-45A**, two volts are induced in the secondary. If 10 volts are applied to the primary of the transformer in **Figure 6-45B**, the output voltage across the terminals of the secondary will be 40 volts. No transformer can be constructed that is 100 percent efficient, although iron core transformers can approach this figure. This is because all the magnetic lines of force set up in the primary do not cut across the turns of the secondary coil.

A certain amount of the magnetic flux, called leakage flux, leaks out of the magnetic circuit. The measure of how well the flux of the primary is coupled into the secondary is called the "coefficient of coupling." For example, if it is assumed that the primary of a transformer develops 10 000 lines of force and only 9 000 cut across the secondary, the coefficient of coupling would be 0.9 or, stated another way, the transformer would be 90 percent efficient.

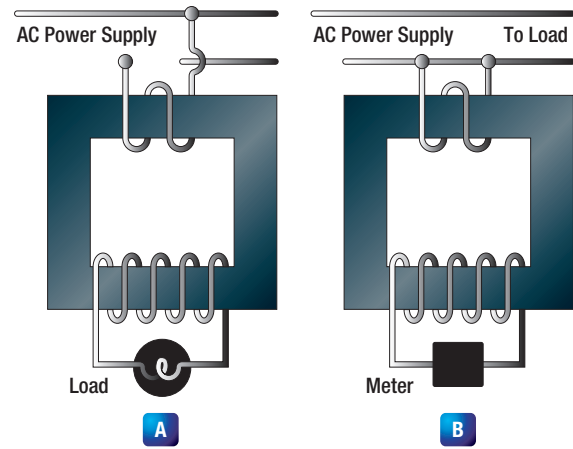


Figure 6-44. Voltage and current transformers.

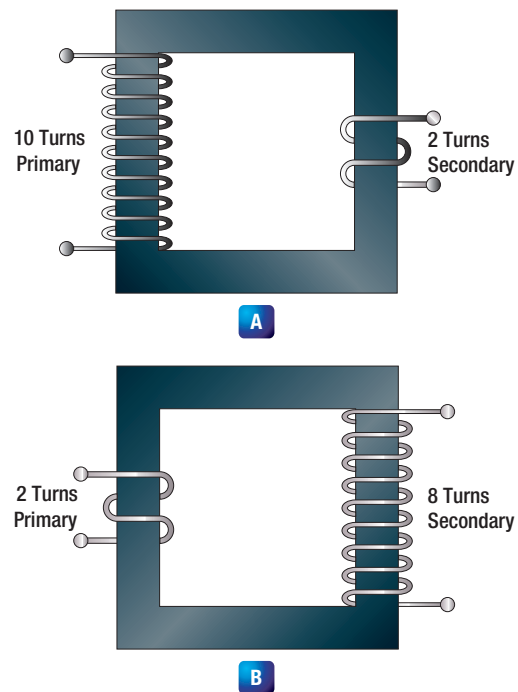


Figure 6-45. A step-down and a step-up transformer.

When an AC voltage is connected across the primary terminals of a transformer, an alternating current will flow and self induce a voltage in the primary coil that is opposite and nearly equal to the applied voltage. The difference between these two voltages allows just enough current in the primary to magnetize its core. This is called the exciting, or magnetizing, current. The magnetic field caused by this exciting current cuts across the secondary coil and induces a voltage by mutual induction.

If a load is connected across the secondary coil, the load current flowing through the secondary coil will produce a magnetic field which will tend to neutralize

the magnetic field produced by the primary current. This will reduce the self-induced (opposition) voltage in the primary coil and allow more primary current to flow. The primary current increases as the secondary load current increases, and decreases as the secondary load current decreases. When the secondary load is removed, the primary current is again reduced to the small exciting current sufficient only to magnetize the iron core of the transformer.

If a transformer steps up the voltage, it will step down the current by the same ratio. This should be evident if the power formula is considered, for the power ($I \times E$) of the output (secondary) electrical energy is the same as the input (primary) power minus that energy loss in the transforming process. Thus, if 10 volts and 4 amps (40 watts of power) are used in the primary to produce a magnetic field, there will be 40 watts of power developed in the secondary (disregarding any loss). If the transformer has a step-up ratio of 4 to 1, the voltage across the secondary will be 40 volts and the current will be 1 amp. The voltage is 4 times greater and the current is one fourth the primary circuit value, but the power ($I \times E$ value) is the same.

When the turns ratio and the input voltage are known, the output voltage can be determined as follows:

$$\frac{E_2}{E_1} = \frac{N_2}{N_1}$$

Where E is the voltage of the primary, E_2 is the output voltage of the secondary, and N_1 and N_2 are the number of turns of the primary and secondary, respectively. Transposing the equation to find the output voltage gives:

$$E_2 = \frac{E_1 N_2}{N_1}$$

The most commonly used types of voltage transformers:

1. Power transformers are used to step up or step down voltages and current in many types of power supplies. They range in size from the small power transformer shown in **Figure 6-46** used in a radio receiver to the large transformers used to step down high power line voltage to the 110 – 120 volt level used in homes.

Figure 6-47 shows the schematic symbol for an iron core transformer. In this case, the secondary is made up of three separate windings. Each winding supplies a different circuit with a specific voltage, which saves the weight, space, and expense of three separate transformers. Each secondary has a midpoint connection, called a "center tap," which provides a selection of half the voltage across the whole winding.

The leads from the various windings are color coded by the manufacturer, as labeled in **Figure 6-47**. This is a standard color code, but other codes or numbers may be used.

2. Audio transformers resemble power transformers. They have only one secondary and are designed to operate over the range of audio frequencies (20 to 20 000 cps).
3. RF transformers are designed to operate in equipment that functions in the radio range of frequencies. The symbol for the RF transformer is the same as for an RF choke coil. It has an air core as shown in **Figure 6-48**.

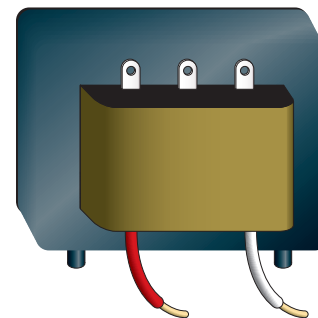


Figure 6-46. Power supply transformer.

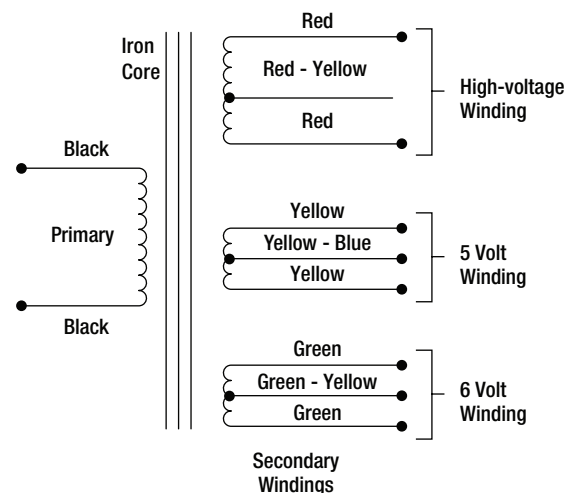


Figure 6-47. Schematic symbol for an iron-core power transformer.

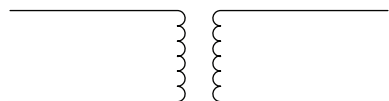


Figure 6-48. An air-core transformer.

4. Autotransformers are normally used in power circuits; however, they may be designed for other uses. Two different symbols for autotransformers used in power or audio circuits are shown in **Figure 6-49**. If used in an RF communication or navigation circuit (**Figure 6-49B**), it is the same, except there is no symbol for an iron core. The autotransformer uses part of a winding as a primary; and, depending on whether it is step up or step down, it uses all or part of the same winding as the secondary. For example, the autotransformer shown in **Figure 6-49A** could use the following the choices for primary and secondary terminals as shown.

Primary		used with		Secondary
1-2				1-3
1-2		" "		2-3
1-3		" "		1-2
1-3		" "		2-3
2-3		" "		1-3
2-3		" "		1-2

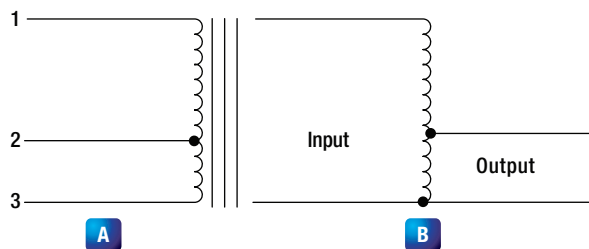


Figure 6-49. Autotransformers.

CURRENT TRANSFORMERS

Current transformers are used in AC power supply systems to sense generator line current and to provide a current, proportional to the line current, for circuit protection and control devices.

The current transformer is a ring type transformer using a current carrying power lead as a primary (either the power lead or the ground lead of the AC generator). The current in the primary induces a current in the secondary by magnetic induction.

The sides of all current transformers are marked "H1" and "H2" on the unit base. The transformers must be installed with the "H1" side toward the generator in the circuit in order to have proper polarity. The secondary of the transformer should never be left open while the system is being operated; to do so could cause dangerously high voltages, and could overheat the transformer. Therefore, the transformer output connections should always be connected with a jumper when the transformer is not being used but is left in the system.

TRANSFORMER LOSSES

In addition to the power loss caused by imperfect coupling, transformers are subject to "copper" and "iron" losses. The resistance of the conductor comprising the turns of the coil causes copper loss. The iron losses are

of two types called hysteresis loss and eddy current loss. Hysteresis loss is the electrical energy required to magnetize the transformer core, first in one direction and then in the other, in step with the applied alternating voltage. Eddy current loss is caused by electric currents (eddy currents) induced in the transformer core by the varying magnetic fields. To reduce eddy current losses, cores are made of laminations coated with an insulation, which reduces the circulation of induced currents.

POWER IN TRANSFORMERS

Since a transformer does not add any electricity to the circuit but merely changes or transforms the electricity that already exists in the circuit from one voltage to another, the total amount of energy in a circuit must remain the same. If it were possible to construct a perfect transformer, there would be no loss of power in it; power would be transferred undiminished from one voltage to another. Since power is the product of volts times amperes, an increase in voltage by the transformer must result in a decrease in current and vice versa. There cannot be more power in the secondary side of a transformer than there is in the primary. The product of amperes times volts remains the same.

The transmission of power over long distances is accomplished by using transformers. At the power source, the voltage is stepped up in order to reduce the line loss during transmission. At the point of utilization, the voltage is stepped down, since it is not feasible to use high voltage to operate motors, lights, or other electrical appliances.

RECTIFIERS

Rectifier circuits change AC voltage into DC voltage and are one of the most commonly used type of circuits in aircraft electronics. (**Figure 6-50**) The resulting DC waveform output is also shown. The circuit has a single semiconductor diode and a load resistor. When the AC voltage cycles below zero, the diode shuts off and does not allow current flow until the AC cycles through zero voltage again. The result is pronounced pulsating DC. While this can be useful, half of the original AC voltage is not being used.

A full wave rectifier creates pulsating DC from AC while using the full AC cycle. One way to do this is to tap the secondary coil at its midpoint and construct two circuits with the load resistor and a diode in each circuit. (**Figure 6-51**) The diodes are arranged so that when current is flowing through one, the other blocks current.

When the AC cycles so the top of the secondary coil of the transformer is positive, current flows from ground, through the load resistor (V_{RL}), Diode 1, and the upper half of the coil. Current cannot flow through Diode 2 because it is blocked. (**Figure 6-51A**) As the AC cycles through zero, the polarity of the secondary coil changes. (**Figure 6-51B**) Current then flows from ground, through the load resistor, Diode 2, and the bottom half of the secondary coil. Current flow through Diode 1 is blocked. This arrangement yields positive DC from cycling AC with no wasted current.

Another way to construct a full wave rectifier uses four semiconductor diodes in a bridge circuit. Because the secondary coil of the transformer is not tapped at the center, the resultant DC voltage output is twice that of the two diode full wave rectifier. (**Figure 6-52**) During the first half of the AC cycle, the bottom of the secondary coil is negative.

Current flows from it through diode (D1), then through the load resistor, and through diode (D2) on its way back to the top of the secondary coil. When the AC reverses its cycle, the polarity of the secondary coil changes. Current flows from the top of the coil through diode (D3), then through the load resistor, and through diode (D4) on its way back to the bottom of the secondary coil. The output waveform reflects the higher voltage achieved by rectifying the full AC cycle through the entire length of the secondary coil.

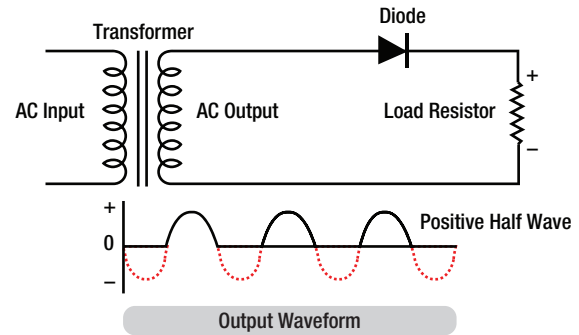


Figure 6-50. A half wave rectifier uses one diode to produce pulsating DC current from AC. Half of the AC cycle is wasted when the diode blocks the current flow as the AC cycles below zero.

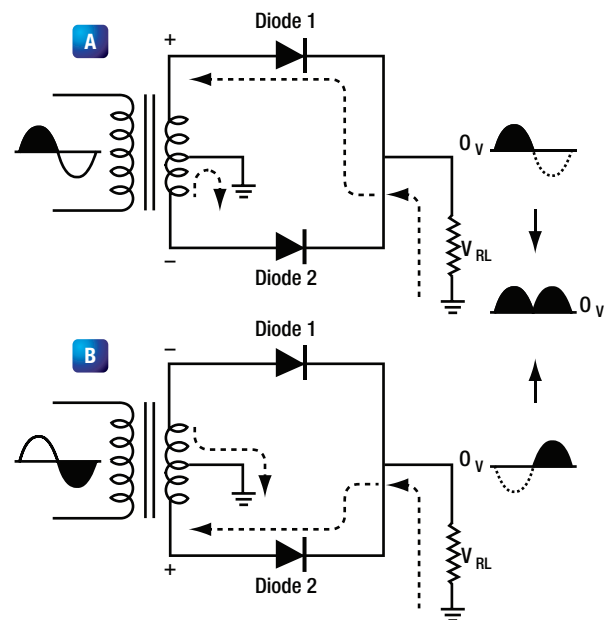


Figure 6-51. A full wave rectifier can be built by center tapping the secondary coil of the transformer and using two diodes in separate circuits. This rectifies the entire AC input into a pulsating DC with twice the frequency of a half wave rectifier.

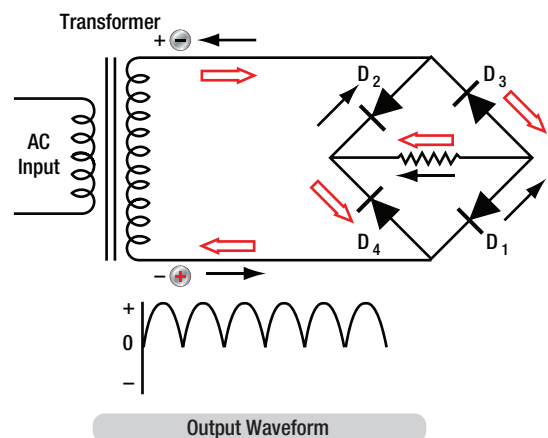


Figure 6-52. The bridge-type four-diode full wave rectifier circuit is most commonly used to rectify single-phase AC into DC avionics.

Use and rectification of three phase AC is also possible on aircraft with a specific benefit. The output DC is very smooth and does not drop to zero. A six diode circuit is built to rectify the typical three phase AC produced by an aircraft alternator. (*Figure 6-53*)

Each stator coil corresponds to a phase of AC and becomes negative for 120° of rotation of the rotor. When stator 1 or the first phase is negative, current flows from it through diode (D1), then through the load resistor and through diode (D2) on its way back to the third phase coil.

Next, the second phase coil becomes negative and current flows through diode (D3). It continues to flow through the load resistor and diode (D4) on its way back to the first phase coil. Finally, the third stage coil

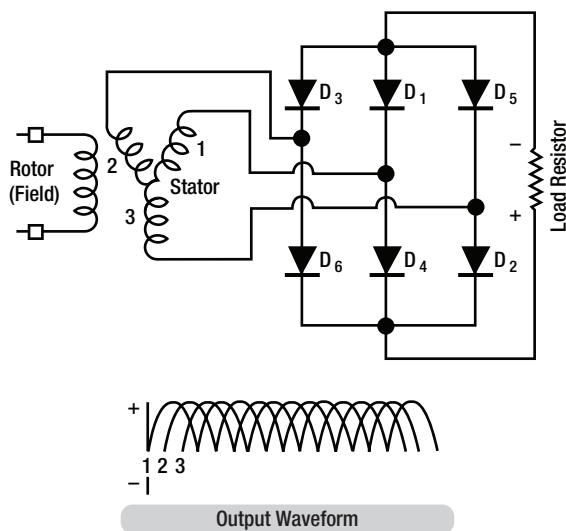


Figure 6-53. A six-diode three-phase AC rectifier.

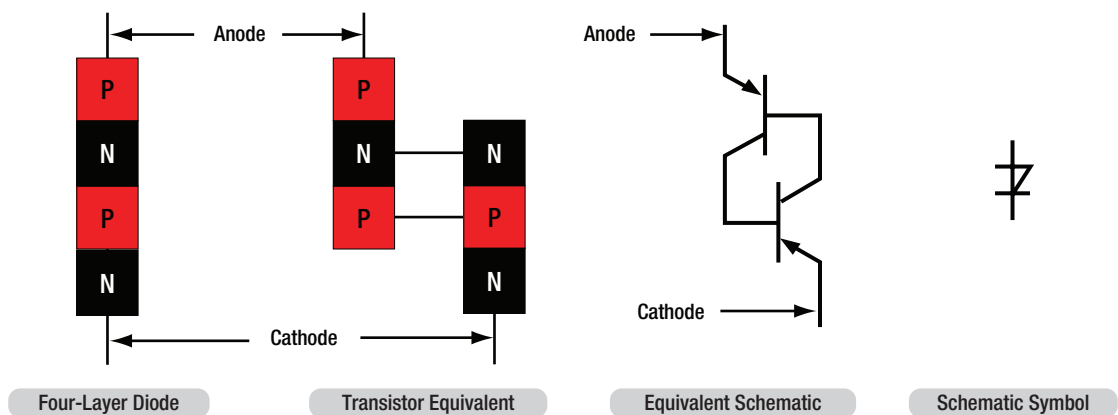


Figure 6-54. A four-layer semiconductor diode behaves like two transistors. When break-over voltage is reached, the device conducts current until the voltage is removed.

becomes negative causing current to flow through diode (D5), then the load resistor and diode (D6) on its way back to the second phase coil. The output waveform of this three phase rectifier depicts the DC produced. It is a relatively steady, non-pulsing flow equivalent to just the tops of the individual curves. The phase overlap prevents voltage from falling to zero producing smooth DC from AC.

SILICON CONTROLLED RECTIFIERS

Combination of semiconductor materials is not limited to a two type, three layer sandwich transistor. By creating a four-layer sandwich of alternating types of semiconductor material (i.e., PNP or NPN), a slightly different semiconductor diode is created. As is the case in a two layer diode, circuit current is either blocked or permitted to flow through the diode in a single direction.

Within a four-layer diode, sometimes known as a Shockley diode, there are three junctions. The behavior of the junctions and the entire four-layer diode can be understood by considering it to be two interconnected three layer transistors. (*Figure 6-54*)

Transistor behavior includes no current flow until the base material receives an applied voltage to narrow the depletion area at the base-emitter junction. The base materials in the four-layer diode transistor model receive charge from the other transistor's collector. With no other means of reducing any of the depletion areas at the junctions, it appears that current does not flow in either direction in this device. However, if a large voltage is applied to forward bias the anode or cathode, at some

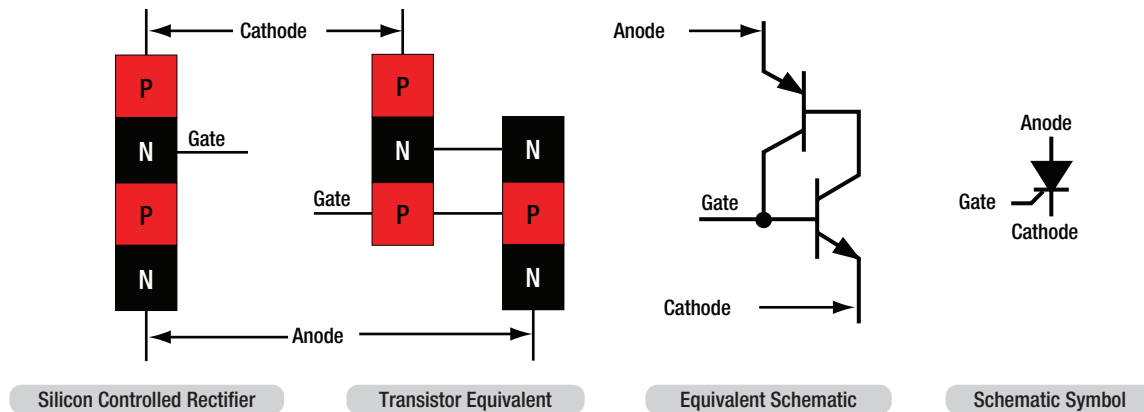


Figure 6-55. A silicon controlled rectifier (SCR) allows current to pass in one direction when the gate receives a positive pulse to latch the device in the on position. Current ceases to flow when it drops below holding current, such as when AC current reverses cycle.

point the ability to block flow breaks down. Current flows through whichever transistor is charged. Collector current then charges the base of the other transistor and current flows through the entire device.

Some caveats are necessary with this explanation. The transistors that comprise this four-layer diode must be constructed of material similar to that described in a zener diode. That is, it must be able to endure the current flow without burning out. In this case, the voltage that causes the diode to conduct is known as breakover voltage rather than breakdown voltage.

Additionally, this diode has the unique characteristic of allowing current flow to continue until the applied voltage is reduced significantly, in most cases, until it is reduced to zero. In AC circuits, this would occur when the AC cycles.

While the four-layer Shockley diode is useful as a switching device, a slight modification to its design creates a silicon controlled rectifier (SCR). To construct a SCR, an additional terminal known as a gate is added. It provides more control and utility. In the four-layer semiconductor construction, there are always two junctions forward biased and one junction reversed biased. The added terminal allows the momentary application of voltage to the reversed biased junction. All three junctions then become forward biased and current at the anode flows through the device. Once voltage is applied to the gate, the SCR become latched or locked on. Current continues to flow through it until the level drops off significantly, usually to zero. Then, another applied voltage through the gate is needed to reactivate the current flow. (*Figure 6-54 and Figure 6-55*)

Figure 6-56. Phase control is a key application for SCR. By limiting the percentage of a full cycle of AC voltage that is applied to a load, a reduced voltage results. The firing angle or timing of a positive voltage pulse through the SCR's gate latches the device open allowing current flow until it drops below the holding current, which is usually at or near zero voltage as the AC cycle reverses.

SCRs are often used in high voltage situations, such as power switching, phase controls, battery chargers, and inverter circuits. They can be used to produce variable DC voltages for motors and are found in welding power supplies. Often, lighting dimmer systems use SCR's to reduce the average voltage applied to the lights by only allowing current flow during part of the AC cycle. This is controlled by controlling the pulses to the SCR gate and eliminating the massive heat dissipation caused when using resistors to reduce voltage. **Figure 6-57** graphically depicts the timing of the gate pulse that limits full cycle voltage to the load. By controlling the phase during which time the SCR is latched, a reduced average voltage is applied.

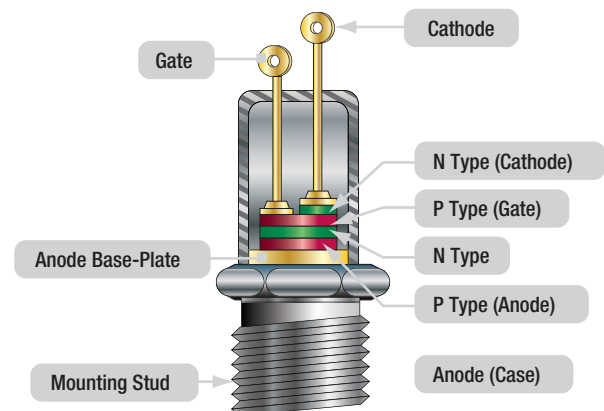


Figure 6-56. Cross-section of a medium-power SCR.

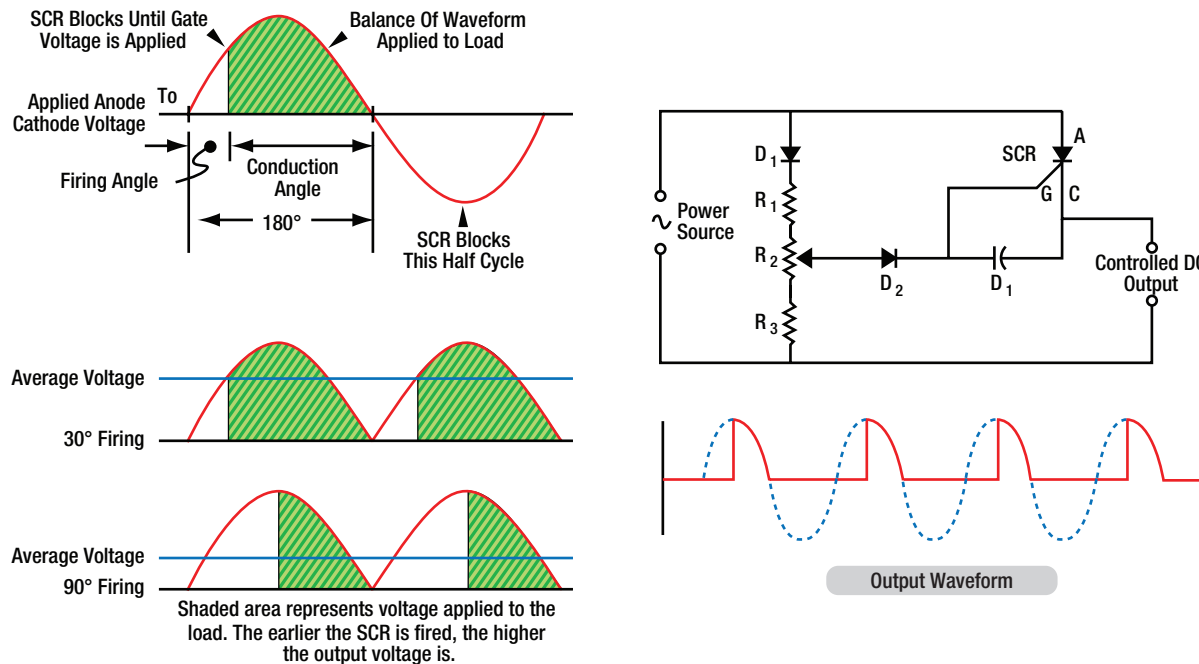


Figure 6-57. Phase control is a key application for SCR. By limiting the percentage of a full cycle of AC voltage that is applied to a load, a reduced voltage results. The firing angle or timing of a positive voltage pulse through the SCR's gate latches the device open allowing current flow until it drops below the holding current, which is usually at or near zero voltage as the AC cycle reverses.

TRANSFORMER RECTIFIERS

Control of the available electric power to numerous electrical devices on an aircraft in any and all situations makes the conversion from AC power to DC power common on modern airliners. Often, transformer rectifiers (TRs) are used for this purpose.

They are typically found between an AC power source and a DC bus and inside battery chargers. The TR not only rectifies AC to produce DC current. It also contains a transformer to adjust the DC output to the precise voltage required.

CIRCUIT PROTECTION

CURRENT LIMITING DEVICES

In addition to the protection proved by the GCUs and the BPCU, individual conductive circuits on an aircraft are protected with current limiting devices.

Conductors should be protected with circuit breakers or fuses located as close as possible to the electrical power source bus. Normally, the manufacturer of the electrical equipment specifies the fuse or circuit breaker to be used when installing equipment. The circuit breaker or fuse should open the circuit before the conductor emits smoke. To accomplish this, the

time current characteristic of the protection device must fall below that of the associated conductor. Circuit protector characteristics should be matched to obtain the maximum utilization of the connected equipment.

Figure 6-58 shows a chart used in selecting the circuit breaker and fuse protection for copper conductors.

This limited chart is applicable to a specific set of ambient temperatures and wire bundle sizes and is presented as typical only. It is important to consult such guides

Wire AN Gauge Copper	Circuit Breaker Amperage	Fuse Amperage
22	5	5
20	7.5	5
18	10	10
16	15	10
14	20	15
12	30	20
10	40	30
8	50	50
6	80	70
4	100	70
2	125	100
1		150
0		150

Figure 6-58. Wire and circuit protection chart.



Figure 6-59. Aircraft buss fuse.

before selecting a conductor for a specific purpose. For example, a wire run individually in the open air may be protected by the circuit breaker of the next higher rating to that shown on the chart.

FUSES

A fuse is placed in series with the voltage source and all current must flow through it. (*Figure 6-59*) The fuse consists of a strip of metal that is enclosed in a glass or plastic housing. The metal strip has a low melting point and is usually made of lead, tin, or copper. When the current exceeds the capacity of the fuse the metal strip heats up and breaks. As a result of this, the flow of current in the circuit stops.

There are two basic types of fuses: fast acting and slow blow. The fast-acting type opens very quickly when their particular current rating is exceeded. This is important for electric devices that can quickly be destroyed when too much current flows through them for even a very small amount of time. Slow blow fuses have a coiled construction inside. They are designed to open only on a continued overload, such as a short circuit.

CIRCUIT BREAKERS

A circuit breaker is an automatically operated electrical switch designed to protect an electrical circuit from damage caused by an overload or short circuit. Its basic function is to detect a fault condition and immediately discontinue electrical flow. Unlike a fuse that operates once and then has to be replaced, a circuit breaker can be reset to resume normal operation. All resettable circuit breakers should open the circuit in which they are installed regardless of the position of the operating control when an overload or circuit fault exists. Such circuit breakers are referred to as trip-free. (*Figure 6-60*)

Note that automatic reset circuit breakers automatically reset themselves. They should not be used as circuit protection devices in aircraft. When a circuit breaker trips, the electrical circuit should be checked and the fault removed before the circuit breaker is reset.

Sometimes circuit breakers trip for no apparent reason, and the circuit breaker can be reset one time. If the circuit breaker trips again, there exists a circuit fault and the technician must troubleshoot the circuit before resetting the circuit breaker. (*Figure 6-61*)

New aircraft designs use a digital circuit protection architecture. This system monitors the amperage through a particular circuit. When the maximum amperage for that circuit is reached, the power is rerouted away from the circuit. This system reduces the use of mechanical circuit breakers. The advantages are weight savings and the reduction of mechanical parts.

EMERGENCY POWER GENERATION

Power in an aircraft is distributed through various buses to permit control over small groups of electrical loads. This also protects against a single electrical power failure causing a severe loss of power. The main AC generators power the various buses including the DC buses which use transformer rectifiers to convert the AC to DC.

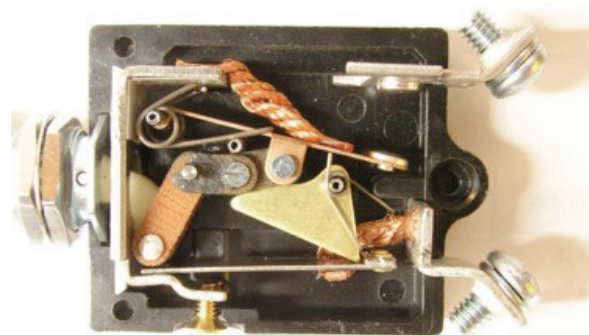


Figure 6-60. Circuit breaker panel.



Figure 6-61. Circuit breaker panel.

Modern aircraft typically have buses and loads divided so that the failure of a single generator is controlled by isolating the failed unit and use of the other main generator or back-up generators to power the buses of the failed unit. Automatic monitoring and switching of power sources is normal.

STANDBY POWER

When both of the main AC generators fail simultaneously, back-up or standby generators may still be used to power AC and DC buses. The failure of the main and back-up generators is rare. It presents the crew with a situation in which emergency power must be used. On most large turbine powered aircraft, a standby power bus is used for emergency power when the main sources of power fail. The standby bus is usually a hot bus directly connected to the main aircraft battery. It is powered by the APU generator, the ram air turbine generator or the aircraft battery.

Standby power from the standby power bus is DC. However, inverters are used to create AC power from the DC standby bus to power vital AC circuits. A limited number of components and circuits are powered off the standby bus. The idea is to be able to maintain vital systems for flight and not deplete the power and sources of power that remain in the emergency situation. The selection of standby power in an emergency is typically made by the crew with a switch on the flight deck. Automatic switching is more common on most modern aircraft.

EXTERNAL/GROUND POWER

Most aircraft employ an external power circuit that provides a means of connecting electrical power from a ground source to the aircraft. External power is often used for starting the engine or maintenance activities on the aircraft. This type of system allows operation of various electrical systems without discharging the battery. The external power systems typically consists of an electrical plug located in a convenient area of the fuselage, an electrical solenoid used to connect external power to the bus, and the related wiring for the system. A common external power receptacle is shown in *Figure 6-62*.

Figure 6-63 shows how the external power receptacle connects to the external power solenoid through a reverse polarity diode. This diode is used to prevent any accidental connection in the event the external power supply has the incorrect polarity (i.e., a reverse of the positive and negative electrical connections). A reverse polarity connection could be catastrophic to the aircraft's electrical system. If a ground power source



Figure 6-62. External power receptacle.

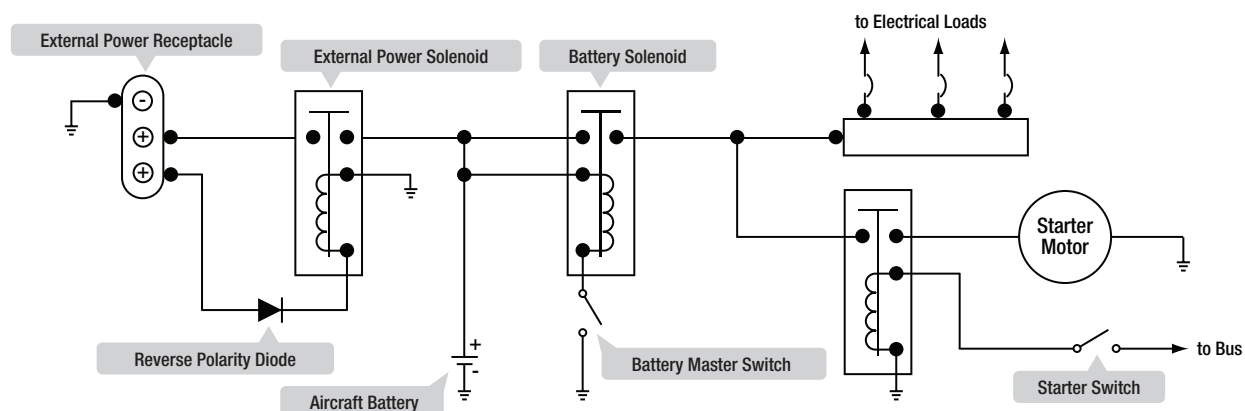


Figure 6-63. A simple external power circuit diagram.

with a reverse polarity is connected, the diode blocks current and the external power solenoid does not close. This diagram also shows that external power can be used to charge the aircraft battery or power the aircraft electrical loads. For external power to start the aircraft engine or power electrical loads, the battery master switch must be closed.

On most airliners, a separate ground handling bus and an APU battery bus are used. Many of these buses divide to power additional buses which distribute power strategically to sub-systems of electrical power.

On large aircraft, when the aircraft engines are running, AC generators mounted on and driven by the engines supply the DC buses through the use of transformer rectifiers units (TRU's). The TRU's convert the 115V AC generated into 28V DC power. When the aircraft is on the ground with external power connected, a separate TRU converts the AC from the ground power source to 28V DC to power the distribution buses. When the external power source is not being used, the main aircraft battery supplies the DC power buses.

Question: 6-1

What type of circuit breakers should never be used on aircraft?

Question: 6-6

How is the voltage output of an AC alternator controlled?

Question: 6-2

What two factors combine to cause a thermal runaway condition on a nickel-cadmium battery?

Question: 6-7

What is the basic electrical principle by which both DC alternators and transformers operate?

Question: 6-3

If both main generators and the APU generator fail, what is the next backup device to provide electrical power?

Question: 6-8

What must be done before two or more generators are connected to the same power bus?

Question: 6-4

Describe the power output of a typical turbine engine driven generator.

Question: 6-9

A transformer has 20 wire turns on its primary side and 4 wire turns on its secondary side. If 100 volts at 20 amps is fed into the primary side, what will be emitted from the secondary side?

Question: 6-5

On what factor does the frequency output of an AC alternator depend?

Question: 6-10

_____ convert AC power to DC, and
_____ convert DC power to AC.

ANSWERS

Answer: 6-1

Automatic reset types.

Answer: 6-6

By controlling the strength of its magnetic field.

Answer: 6-2

Heat and overcharging.

Answer: 6-7

Induction.

Answer: 6-3

Ram Air Turbine. (RAT).

Answer: 6-8

The output frequency phases must be equalized.

Answer: 6-4

115 volt AC at 400 Hz.

Answer: 6-9

20 volts at 100 amps (less its coefficient of efficiency).

Answer: 6-5

Its rotational speed.

Answer: 6-10

Rectifiers, inverters.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 07

EQUIPMENT AND FURNISHINGS (ATA 25)

Knowledge Requirements

11.7 - *Equipment and Furnishings (ATA 25)*

- (a) Emergency equipment requirements; Seats, harnesses and belts;
- (b) Cabin layout;
Equipment layout;
Cabin Furnishing installation;
Cabin entertainment equipment;
Galley installation;
Cargo handling and retention equipment;
Airstairs.

2

1

EQUIPMENT AND
FURNISHINGS (ATA 25)

11.7 - EQUIPMENT AND FURNISHINGS

EMERGENCY EQUIPMENT REQUIREMENTS

Since aircraft leave the surface of the planet and fly in the sky, emergency equipment such as fire extinguishers, life rafts, escape slides, first aid kits, etc. may be required by authorities to be on board for the protection of passengers and crew. Different size aircraft designed for different uses have emergency equipment requirements suitable for the intended purpose of the aircraft. For example, a small single engine aircraft has fewer required fire extinguishers on board than a 300 seat airliner.

The specifications for emergency equipment requirements on any given aircraft are found in EASA issued airworthiness codes called "certification specifications" (CS), i.e. CS-23, CS-25, etc. These certification standards are divided with separate sections devoted to different types of aircraft. CS-25 are those pertaining to large commercial aircraft. When an airliner is certified, it meets the requirements for the installation of emergency equipment in accordance with CS-25 specifications. For example, CS.25.851(a)(1) specifies the minimum number of portable fire extinguishers to be on board the aircraft. (**Figure 7-1**) To receive an airworthiness certificate, the manufacturer is required to make provisions for 4 extinguishers on a 250-seat aircraft. The aircraft operator must ensure the extinguishers are present and airworthy for each flight.

Many other specifications for emergency equipment are covered in CS-25 such as the number, type and location of emergency doors, seat belt strength, exit sign type, emergency lighting parameters and life raft locations. Details pertaining to the required emergency equipment itself may be found by reference to a European Technical Standard Order (ETSO). Acceptable Means for Compliance (AMC) for CS-25 (CS-25, Book 2) also contains details concerning components and procedures referenced in CS-25. Other documentation may also be referenced.

Thus, the required emergency equipment is delivered and stowed in place when an aircraft is placed into service. A primary function of the technician is to ensure all of the required equipment is in its specified location and serviceable. Note that these locations are specified in the certification standards. Furthermore, evacuation slides,

life rafts, fire extinguishers and oxygen bottles all have inspection requirements that include a pressure checked before each flight. These and other security and condition inspection items related to emergency equipment are written and require the signature of the technician who performs the check. This type of equipment is installed for a limited time period after which, it must be removed for in-depth inspection and recharging. Installation and removal procedures are detailed in the aircraft maintenance manual. (**Figure 7-2**)

There are many items on board an aircraft that contribute to safety via designs that are purposeful in an emergency. Some items regularly considered emergency equipment are:

- Fire Extinguishers
- First Aid Kits
- Life Vests
- Megaphones
- Flashlights
- Fire Axes

Passenger Capacity	Number of Extinguishers
7 to 30	1
31 to 60	2
61 to 200	3
201 to 300	4
301 to 400	5
401 to 500	6
501 to 600	7
601 to 700	8

Figure 7-1. Required minimum number of hand fire extinguishers.



Figure 7-2. Emergency slide inflation canister pressure is checked regularly on the gauge in the lower window (mostly yellow). A life-limit removal date sticker partially obscures the gauge in the photo.

- Defibrillator's
- Life Rafts
- Evacuation Slides
- Emergency Lights
- Emergency Exit Signs
- Escape Ropes
- Smoke Goggles and Hoods

Technicians must be aware that when working with pressurized raft and door-slide installations, specific instructions must be followed to prevent inadvertent deployment and injury. Typically, the devices must be armed after installation to operate as designed.

SEATS, HARNESES, AND BELTS

SEATS

Aircraft seats are constructed to be very strong yet lightweight. They are typically mounted in groups of two or three seats in a frame. The seat assembly is fitted with quick disconnect fittings for secure attachment into seat track mounting strips on the cabin floor. The seat tracks are bolted to structural fuselage frame members. The tracks allow adjustment of seats forward and aft in 1 inch increments before enabling a lock down of the seat in the track. The result is that seats may be arranged in different configurations (spacing). (*Figure 7-3*)

It is typical for seats used in 1st class and business class to be larger than coach seats. However, the seat frame for the larger seats, which may be more elaborate with full-recline capability, still connects into seat tracks in the same manner as the coach class seats. Passenger seat assemblies are fitted with arm rests and a reclining mechanism for the seat back. A formed metal luggage

restraint rail is part of the seat assembly. It prevents luggage stowed under the seat from sliding forward during abrupt deceleration. Each seat has retractable seat arms and a seat back tray table that unhinges from its stowage location in the seat back.

Seats on the flight deck for the captain and first officer are fully adjustable. Captain and first officer seats engage in floor-mounted tracks similar to the passenger seat assemblies. Five-point seat belts are attached to the seat frame assembly. (*Figure 7-4*) Observer seats typically fold down out of a wall or bulkhead and consist of a padded slab-style seats with safety harnesses.



Figure 7-3. Passenger seats are locked into a seat track on the floor of the cabin.

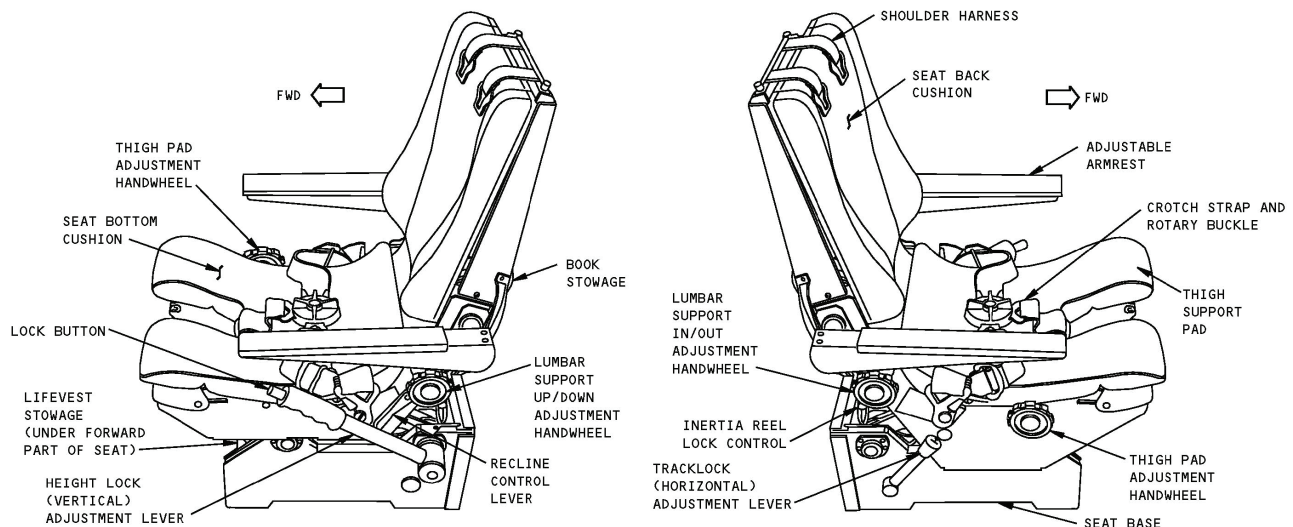


Figure 7-4. Typical captain's seat.

The most modern aircraft include an LCD viewing screen mounted in the back of each upper seat back for use by the passenger in the seat directly aft. Safety briefing announcements, movies and internet may be accessed depending on the in flight entertainment offered by the airline. A seat electronics unit (SEU) is typically mounted under each seat for facilitation of the inflight entertainment (IFE). A digital data bus runs the length of the cabin so that each individual SEU can connect to the IFE computer.

SEAT BELTS AND HARNESES

Seat belts for all passengers and seat harnesses for crew members are required on passenger service aircraft. If a seat belt is not functional and cannot be changed before flight, the seat must be placarded and not used. Seat belts are attached to the seat assembly/frame in most cases. Flight attendant and fold-down jump seats may attach the seat belts to a bulkhead. Typically passengers seats have a lap belt only. Seat belts that include shoulder harnesses are used by the crew. The captain and first officer's seat belts typically include both shoulder harness and crotch strap that fasten into a rotary buckle along with the lap belts. Seat belts and harnesses are constructed from nylon webbing which are woven to be extremely strong. Test procedures for seat belt and the forces they must withstand are also in CS-25.

CABIN LAYOUT

Operators may configure seats, galleys, bulkheads and lavatories in a variety of ways as long as CS-25 specifications are met. This does tend to limit configurations to those commonly found on airlines. Pragmatic location of galleys and lavatories also limits the cabin layout. Furthermore a lavatory for each seating zone (i.e. first class, coach, etc.) is required.

Flight attendants are charged with assisting passengers at exits in case of an emergency so their seat locations and control stations are generally limited to be near the exits. Needless to say, no exit or emergency escape

path may be blocked. Thus, cabin layout is a result of considerations for safety requirements and practical functional arrangements to serve the passengers.

Overhead storage bins run the length of the cabin. They are fastened to support structure attached to fuselage members. The exact location of the bins, however, is able to be modified to position them correctly in relation to the configuration of seats and other installations. Passenger service unit (PSU) locations are the same. Berths for crew changing or sleeping are located by design and access/egress requirements found in CS-25. These are usually at the forward or aft ends of the cabin or above the main flight and passenger cabins. The flight control cabin is always separated from the passenger cabin by a door. Closets, dividers, LCD monitors, video projectors and flight attendant locations are all slightly adjustable to accommodate different seating configurations.

Figure 7-5, illustrates the "flexibility zones" for lavatory location that Boeing provides on its 777-200 aircraft. Operators may configure the aircraft in numerous way as long as lavatories are installed in the zones illustrated. To facilitate this flexibility, the manufacturer makes available all water, air and electrical connections within the zones as well as engineered installation fittings and hardware. Galley installation is similarly flexible. A common adjustment to cabin layout and configuration is seat pitch. Pitch is the fore and aft distance between two rows of seating. A small seat pitch allows more seats to be installed on an aircraft. A large pitch is typically more comfortable. The seat pitch used is up to the operator within the confines of the engineering and certification specifications for the aircraft. The number of seats desired on a certain aircraft greatly influence the overall cabin layout.

EQUIPMENT LAYOUT

The location of various equipment on a airliner is not always left up to the operator. As stated, CS-25 specifies

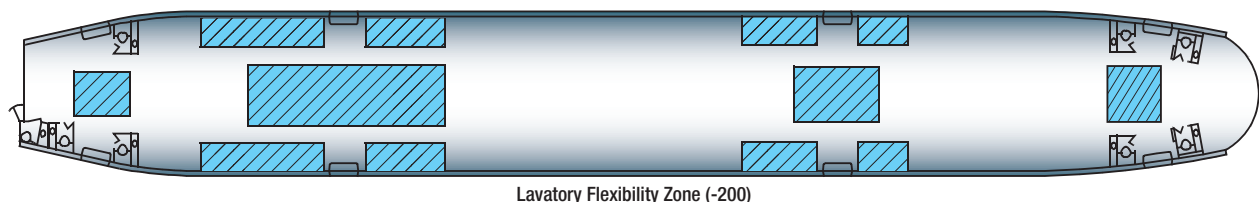


Figure 7-5. Flexibility zones for lavatory location.

many locations for emergency features and equipment. Some of the requirements for equipment location on the flight deck are described as "within arm's reach of the pilot while sitting". Obviously, this leaves the exact location for certain equipment open. However, when all location specifications are taken into account along with the practicality of the location of other equipment in the area, manufacturer's produce aircraft with fairly consistent locations for everything. It is typically the technicians job to identify the required equipment stowed in the chosen location and to inspect it for airworthiness. **Figure 7-6** illustrates an example of the equipment stowed on the sidewall next to the first officer seat on the flight deck of a 737.

It should be mentioned that much of an airliner's electronic equipment is installed in dedicated areas for avionics below the passenger cabin floor. Often, just the control interface is accessible in the passenger compartment or on the flight deck while the "black box" containing the electronics and computer(s) associated with a piece of electronic equipment is located on a rack of an equipment bay.

CABIN FURNISHING INSTALLATION

Cabin furnishings and installations are considered for safety attributes in addition to their just being able to function. In particular, materials are closely scrutinized and tested to ensure that specifications for flammability and smoke production are low. (**Figure 7-7**)

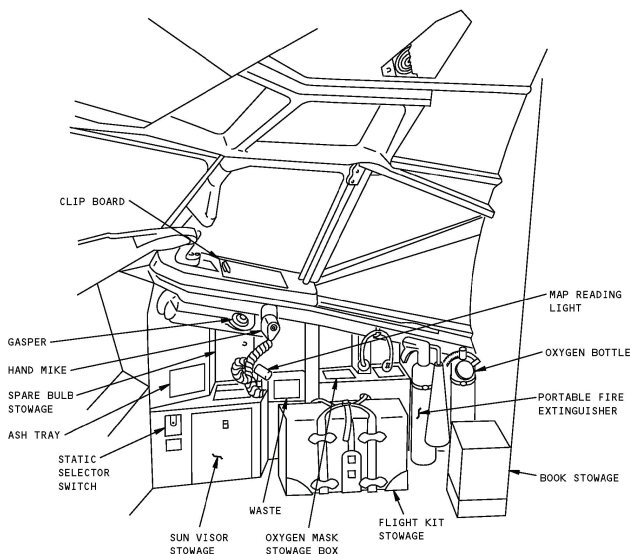


Figure 7-6. Equipment location on the flight deck.



Figure 7-7. A technician installs the first sidewall veneers within a large aircraft cabin.

Sidewall veneers, seat covers, curtains, partitions, carpet and nearly every other material from which a furnishing is constructed must adhere to certification standards specified in CS-25. Elaboration on CS-25 regulations appear in the acceptable means of compliance (AMC) for CS-25.

CABIN ENTERTAINMENT EQUIPMENT

Cabin entertainment is a growing area of attention for airline operators. Many independent vendors compete to sell airlines particular in flight entertainment packages. Individual selection of music and movies at each seat location is common. Internet service is also common. Use of digital data busing for cabin entertainment equipment is growing. As mentioned, individual seat electronics units (SEU's) are typically mounted under the seats in each seat row. These are part of a computerized network that allow passengers access to and control of the entertainment that the airline offers.

On many airliners, a cabin entertainment system with several LCD screens mounted throughout the cabin to be shared by all is used. Regardless whether individual or community systems are in place, maintenance information for cabin entertainment equipment is commonly the responsibility of the equipment vendor. See more about cabin entertainment equipment in *Sub-Module 20* in the book. (**Figure 7-8**)

GALLEY INSTALLATION

Galley installations on airliners are similar to lavatory installations. Areas of the aircraft are engineered to permit installation with fittings for connection into water and electrical systems. The galleys themselves



Figure 7-8. A modern cabin entertainment system with individual passenger programming selections.



Figure 7-9. Airbus A380 galley.

are preconstructed modular units. One or more units are used on any given aircraft and configuration. Galleys typically install into the same floor tracks as the passenger seats but may also have specially dedicated fittings for installation. Tie rods to structure at they top of the fuselage further secure the installation. Most galleys are installed at the aft or forward end of the passenger cabin. Between classes of the cabin (i.e. coach and first class) is also a common galley location. (*Figure 7-9*)

Large aircraft may have the galley installation on the lower level below the passenger cabin. An elevator is used to transport items to the passenger cabin. The specially designed lift system typically engages service carts securely to raise and lower food to and from the galley. Most galleys include one or more ovens to heat meals, a refrigerated area, coffee makers, potable water and drain lines, a trash collection area and lighting.

A control panel is installed with switches and circuit breakers for the electrical components. Often, flight attendant service carts are made to store in the galley module. Other features of a galley may include a fume extraction system, a trash compactor, an ice maker and an interphone system user's interface. Galley ovens and heating devices are electric and put a high demand on the aircraft electrical system. Buses and power supplied to galley are engineered accordingly. On some aircraft, full galley usage is restricted during periods of high electrical demand by critical aircraft systems. Typical galley power is 115 VAC. In recent years, aircraft manufacturers, especially Airbus, and vendors have redesigned the galley and its components to be lighter and more functional for flight attendants.

CARGO HANDLING AND RETENTION EQUIPMENT

Cargo on passenger airliners is handled in different ways depending on the aircraft and the options installed by the aircraft operator. Cargo compartments are located below the passenger cabin floor with doors to the exterior of the aircraft so that cargo is loaded into the compartments from the airport ramp. They are lined with fire resistant sidewall and ceiling panels.

(*Figure 7-10*)

Small passenger transports and many larger ones may simply have an empty compartment finished to accept loose cargo on a piece-by-piece basis. A major concern is the shift of cargo during flight maneuvers which could cause a partial loss of control or inhibit performance. Cargo nets and barriers are used to keep cargo from shifting in these wide open berths. Numerous fittings are installed on the floor, walls and ceilings of the cargo compartment into which nets and barriers attach with



Figure 7-10. A standard cargo container awaiting loading. Note the countoured sides to facilitate the shape of the compartment.

quick release fittings. Cargo is loaded into position with consideration to weight and balance requirements on the aircraft and then the barriers are installed to hold it in place.

Medium to large sized airliners and cargo-dedicated transports commonly use cargo loading systems to place and secure cargo in the cargo compartments. These systems are electrically operated. The most modern systems permit loading with only one or two people. Certified containers are used into which cargo is first loaded before the containers are loaded into the cargo compartment. These containers are purpose built in different sizes. Their shapes and dimensions are made to follow the contour of the aircraft fuselage so as to maximize the use of space in the cargo compartment. Pallets of different sizes may also be loaded into a cargo compartment as well as uncertified containers of various design. Cargo containers are known as unit load devices (ULDs). They are given different names depending on size. For example an LD-3 container is smaller than an LD-9. To enable loading of such heavy cargo containers, some type of automatic electric cargo system is installed in the aircraft. The controls for the system are typically located at the entrance to the cargo compartment.

Usually, the cargo loading system has a series of floor panels that install to cover the cargo compartment floor. These panels are equipped with a spread field of ball rollers upon which containers can slide. Cylindrical and wheel type rollers are also used. To move containers from side to side and fore and aft in the compartment, electric power drive units (PDU) are installed in the floor panels and controlled from the operator's control

panel. These are electric motors capable of driving the containers over the balls and rollers on the floor panels. Guides, restraints, stops and locks are all built in to assist in positioning and securing the loads. (**Figure 7-11**)

Freight aircraft have larger and often very flexible cargo loading systems on the main cabin deck as well as in the lower cargo compartments. Containerized, palletized, and free cargo are always secured and loaded with respect to weight and balance requirements of the given aircraft.

AIRSTAIRS

Many airliners have an on board set of stairs that can be deployed for use should an airport jetway malfunction or when no jetway service or portable stairs are available. These are called airstairs. They are typically an electrically deployed set of stairs stowed under the main cabin floor which extend out the fuselage from below a cabin door to the ground. (**Figure 7-12**) Boeing 727 aircraft have air stairs that extend from the aft end of the fuselage. Air stairs are rarely use on large wide-body aircraft due to the height above the ground of the main passenger cabin. The size and weight of stairs to reach this level makes them too cumbersome.

Business aircraft make extensive use of a type of stair that attaches to and folds down from the main cabin entry doors. These are generally not considered airstairs but they perform the same function. Incorporation and use of airstairs has become rare, primarily to save the weight and space that the stowed airstairs occupy. When fitted, typically a DC deployment and retraction motor is operated from a panel on the outside of the fuselage near the stair stowage compartment door.

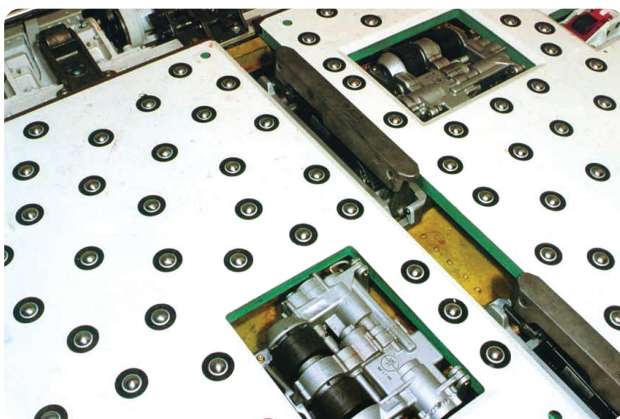


Figure 7-11. Cargo loading system floor panels with ball-rollers, electric power drive units and stops.



Figure 7-12. Airstairs on an airliner stow under the main cabin floor.

Question: 7-1

Specifications for emergency equipment requirements on any given aircraft are found in _____.

Question: 7-5

The black boxes that contain avionics computers are located _____.

Question: 7-2

Passenger seats on an airliner are installed by attachment into _____.

Question: 7-6

Cabin furnishings materials are tested for _____.

Question: 7-3

If a seat belt is not functional and cannot be changed before flight, the seat must _____.

Question: 7-7

Cabin entertainment equipment may use a _____ to provides services in each seat row.

Question: 7-4

A _____ for each seating class is required.

Question: 7-8

Aircraft galleys connect into _____ and _____ systems.

ANSWERS

Answer: 7-1

CS-25 (Certification Standard 25)

Answer: 7-5

on a rack in an equipment bay.

Answer: 7-2

seat track mounting strips.

Answer: 7-6

flammability and smoke production.

Answer: 7-3

be placarded and not used.

Answer: 7-7

a digital data bus.

Answer: 7-4

lavatory.

Answer: 7-8

water.

electrical.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 08

FIRE PROTECTION (ATA 26)

Knowledge Requirements

11.8 - Fire Protection (ATA 26)

- (a) Fire and smoke detection and warning systems;
Fire extinguishing systems;
System tests;
- (b) Portable fire extinguisher.

3

1

FIRE PROTECTION
(ATA 26)

11.8 - FIRE PROTECTION

A complete fire protection system on modern aircraft, and on many older aircraft, includes a fire detection system and a fire extinguishing system. Fire detection is accomplished in many different ways explained below. Fire extinguishing is accomplished with fixed and portable fire agent dispensing systems also explained in this *Sub-Module*.

REQUIREMENTS FOR FIRE TO OCCUR

Three things are required for a fire: (1) fuel - something that will, in the presence of heat combine with oxygen, thereby releasing more heat and as a result reduces itself to other chemical compounds; (2) heat - accelerates the combining of oxygen with fuel, in turn releasing more heat; and (3) oxygen—the element which combines chemically with another substance through the process of oxidation. Rapid oxidation, accompanied by a noticeable release of heat and light, is called combustion or burning. (**Figure 8-1**) Remove any one of these things and the fire extinguishes.

CLASSES OF FIRES

The following classes of fires that are likely to occur on board aircraft, as defined in the United States National Fire Protection Association (NFPA) Standard 10, Standard for Portable Fire Extinguishers, 2007 Edition, are:

1. Class A - fires involving ordinary combustible materials, such as wood, cloth, paper, rubber, and plastics.
2. Class B - fires involving flammable liquids, petroleum oils, greases, tars, oil-based paints, lacquers, solvents, alcohols, and flammable gases.
3. Class C - fires involving energized electrical equipment in which the use of an extinguishing media that is electrically non-conductive is important.
4. Class D - fires involving combustible metals, such as magnesium, titanium, zirconium, sodium, lithium, and potassium.

FIRE ZONES

Because fire is one of the most dangerous threats to an aircraft, the potential fire zones of modern multi-engine aircraft are protected by a fixed fire protection system.



Figure 8-1. The fire triangle; all three elements shown are required for fire to occur.

A fire zone is an area, or region, of an aircraft designed by the manufacturer to require fire detection and/or fire extinguishing equipment and a high degree of inherent fire resistance. The term "fixed" describes a permanently installed extinguishing system in contrast to any type of portable fire extinguishing equipment, such as a hand held Halon or water fire extinguisher. Typical zones on aircraft that have a fixed fire detection and/or fire extinguisher system are:

1. Engines and auxiliary power unit (APU)
2. Cargo and baggage compartments
3. Lavatories on transport aircraft
4. Electronic bays
5. Wheel wells
6. Bleed air ducts

NOTE: Fire zones are further classified by the airflow through them. The amount and flow characteristics through a zone greatly effect the characteristics of a fire there-in and the methods used to detect and extinguish a fire. The zones are as follows:

1. Class A zone - area of heavy airflow past regular arrangements of similarly shaped obstructions. The power section of a reciprocating engine is usually of this type.
2. Class B zone - area of heavy airflow past aerodynamically clean obstructions. Included in this type are heat exchanger ducts, exhaust manifold shrouds, and areas where the inside of the enclosing cowl or other closure is

smooth, free of pockets, and adequately drained so leaking flammables cannot puddle. Turbine engine compartments may be considered in this class if engine surfaces are aerodynamically clean and all airframe structural formers are covered by a fireproof liner to produce an aerodynamically clean enclosure surface.

3. Class C zone - area of relatively low airflow. An engine accessory compartment separated from the power section is an example of this type of zone.
4. Class D zone - area of very little or no airflow. These include wing compartments and wheel wells where little ventilation is provided.
- Class X zone - area of heavy airflow and of unusual construction, making uniform distribution of the extinguishing agent very difficult. Areas containing deeply recessed spaces and pockets between large structural formers are of this type. Tests indicate agent requirements to be double those for Class A zones.

FIRE PREVENTION

Leaking fuel, hydraulic, deicing, or lubricating fluids can be sources of fire in an aircraft. This condition should be noted and corrective action taken when inspecting aircraft systems. Minute pressure leaks of these fluids are particularly dangerous for they quickly produce an explosive atmospheric condition. Carefully inspect fuel tank installations for signs of external leaks. With integral fuel tanks, the external evidence may occur at some distance from where the fuel is actually escaping

Many hydraulic fluids are flammable and should not be permitted to accumulate in the structure. Sound proofing and lagging materials may become highly flammable if soaked with oil of any kind. Any leakage or spillage of flammable fluid in the vicinity of combustion heaters is a serious fire risk, particularly if any vapor is drawn into the heater and passes over the hot combustion chamber.

Oxygen system equipment must be kept absolutely free from traces of oil or grease, since these substances spontaneously ignite when in contact with oxygen under pressure. Oxygen servicing cylinders should be clearly marked so they cannot be mistaken for cylinders containing air or nitrogen, as explosions have resulted from this error during maintenance operations.

FIRE DETECTION AND WARNING SYSTEMS

To detect fires or overheat conditions, detectors are placed in the various zones to be monitored. The complete aircraft fire protection systems of most large turbine engine and high performance aircraft incorporate several of these different detection methods.

1. Rate-of-temperature-rise detectors.
2. Radiation sensing detectors.
3. Smoke detectors.
4. Overheat detectors.
5. Carbon monoxide detectors.
6. Combustible mixture detectors.
7. Optical detectors.
8. Observation of crew or passengers.

The types of detectors most commonly used for fast detection of fires are the rate-of-rise, optical sensor, pneumatic loop, and electric resistance systems.

REQUIREMENTS FOR OVERHEAT AND FIRE DETECTION SYSTEMS

Fire protection systems on current-production aircraft do not rely solely on observation by crew members as a primary method of fire detection. Regardless of the type, an ideal fire detector system includes as many of the following features as possible:

1. No false warnings under any flight or ground condition.
2. Rapid indication of a fire and accurate location of the fire.
3. Accurate indication that a fire is out.
4. Indication that a fire has re-ignited.
5. Continuous indication for duration of a fire.
6. Means for electrically testing the detector system from the aircraft cockpit.
7. Resists damage from exposure to oil, water, vibration, extreme temperatures, or handling.
8. Light in weight and easily adaptable to any mounting position.
9. Circuitry that operates directly from the aircraft power system without inverters.
10. Minimum electrical current requirements when not indicating a fire.
11. Cockpit light that illuminates, indicating the location of the fire, and with an audible alarm system.
12. A separate detector system for each engine.

THERMAL SWITCH SYSTEMS

A number of detectors, or sensing devices, are available. Many older-model aircraft still operating have some type of thermal switch system or thermocouple system. A thermal switch system has one or more lights energized by the aircraft power system and thermal switches that control operation of the light(s). These thermal switches are heat-sensitive units that complete electrical circuits at a certain temperature. They are connected in parallel with each other but in series with the indicator lights.

(Figure 8-2)

If the temperature rises above a set value in any one section of the circuit, the thermal switch closes, completing the light circuit to indicate a fire or overheat condition. No set number of thermal switches is required; the exact number is usually determined by the aircraft manufacturer. On some installations, all the thermal detectors are connected to one light; on others, there may be one thermal switch for each indicator light.

Some warning lights are push-to-test lights. The bulb is tested by pushing it in to check an auxiliary test circuit. The circuit shown in Figure 8-2 includes a test relay. With the relay contact in the position shown, there are two possible paths for current flow from the switches to the light. This is an additional safety feature. Energizing the test relay completes a series circuit and checks all the wiring and the light bulb. Also included in the circuit shown in Figure 8-2 is a dimming relay. By energizing the dimming relay, the circuit is altered to include a resistor in series with the light. In some installations, several circuits are wired through the dimming relay, and all the warning lights may be dimmed at the same time.

THERMOCOUPLE SYSTEMS

The thermocouple fire warning system operates on an entirely different principle from the thermal switch system. A thermocouple depends on the rate of temperature rise and does not give a warning when an engine slowly overheats or a short circuit develops. The system consists of a relay box, warning lights, and thermocouples. The wiring system of these units may be divided into the following circuits:

1. Detector circuit
2. Alarm circuit
3. Test circuit

These circuits are shown in Figure 8-3. The relay box contains two relays, the sensitive relay and the slave relay, and the thermal test unit. Such a box may contain from one to eight identical circuits, depending on the number of potential fire zones. The relays control the warning lights. In turn, the thermocouples control the operation of the relays. The circuit consists of several thermocouples in series with each other and with the sensitive relay.

The thermocouple is constructed of two dissimilar metals, such as chromel and constantan. The point at which these metals are joined and exposed to the heat of a fire is called a hot junction. There is also a reference junction enclosed in a dead air space between two insulation blocks. A metal cage surrounds the thermocouple to give mechanical protection without hindering the free movement of air to the hot junction. If the temperature rises rapidly, the thermocouple produces a voltage from the temperature difference between the reference junction and the hot junction. If both junctions are heated at the same rate, no voltage results.

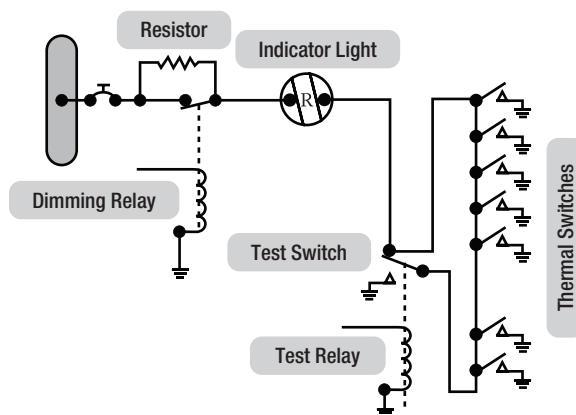


Figure 8-2. Thermal switch fire circuit.

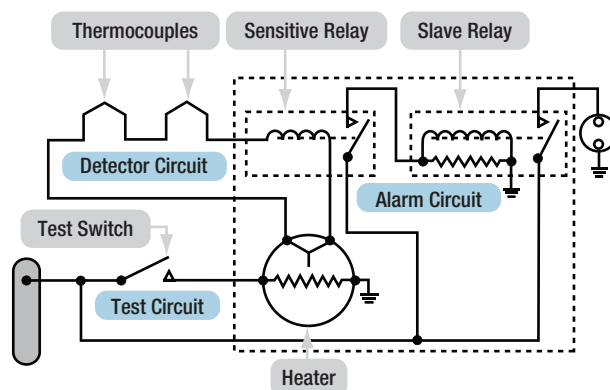


Figure 8-3. Thermocouple fire warning circuit.

In an engine compartment, there is a normal, gradual rise in temperature from engine operation; because it is gradual, both junctions heat at the same rate and no warning signal is given. If there is a fire, however, the hot junction heats more rapidly than the reference junction. The ensuing voltage causes a current to flow within the detector circuit. Any time the current is greater than 4 milliamperes (0.004 ampere), the sensitive relay closes. This completes a circuit from the aircraft power system to the coil of the slave relay.

The slave relay then closes and completes the circuit to the warning light to give a visual fire warning. The total number of thermocouples used in individual detector circuits depends on the size of the fire zones and the total circuit resistance, which usually does not exceed 5 ohms. As shown in **Figure 8-3**, the circuit has two resistors. The resistor connected across the slave relay terminals absorbs the coil's self induced voltage to prevent arcing across the points of the sensitive relay. The contacts of the sensitive relay are so fragile that they burn, or weld, if arcing is permitted.

When the sensitive relay opens, the circuit to the slave relay is interrupted and the magnetic field around its coil collapses. The coil then gets a voltage through self induction but, with the resistor across the coil terminals, there is a path for any current flow as a result of this voltage, eliminating arcing at the sensitive relay contacts.

CONTINUOUS LOOP SYSTEMS

Transport aircraft almost exclusively use continuous thermal sensing elements for powerplant and wheel well protection. These systems offer superior detection performance and coverage, and they have the proven ruggedness to survive in the harsh environment of modern turbofan engines. A continuous loop detector or sensing system permits more complete coverage of a fire hazard area than any of the spot type temperature detectors.

Two widely used types of continuous loop systems are the thermistor type detectors, such as the Kidde and the Fenwal systems, and the pneumatic pressure detector, such as the Lingberg system. (Lindberg system is also known as Systron-Donner and, more recently, Meggitt Safety Systems.)

FENWAL SYSTEM

The Fenwal system uses a slender Inconel tube packed with thermally sensitive eutectic salt and a nickel wire center conductor. (**Figure 8-4**) Lengths of these sensing elements are connected in series to a control unit. The elements may be of equal or varying length and of the same or different temperature settings.

The Fenwal system control unit, operating directly from the power source, applies a small voltage on the sensing elements. When an overheat condition occurs at any point along the element length, the resistance of the eutectic salt within the sensing element drops sharply, causing current to flow between the outer sheath and the center conductor. This current flow is sensed by the control unit, which produces a signal to actuate the output relay and activate the alarms. When the fire has been extinguished or the critical temperature lowered below the set point, the Fenwal system automatically returns to standby alert, ready to detect any subsequent fire or overheat condition. The Fenwal system may be wired to employ a loop circuit. In this case, should an open circuit occur, the system still signals fire or overheat. If multiple open circuits occur, only that section between breaks becomes inoperative.

KIDDE SYSTEM

In the Kidde continuous loop system, two wires are imbedded in an Inconel tube filled with a thermistor core material. (**Figure 8-5**) The two electrical conductors go through the length of the core. One conductor has a ground connection to the tube, and the other conductor connects to the fire detection control unit. As the temperature of the core increases, electrical resistance to the ground decreases.

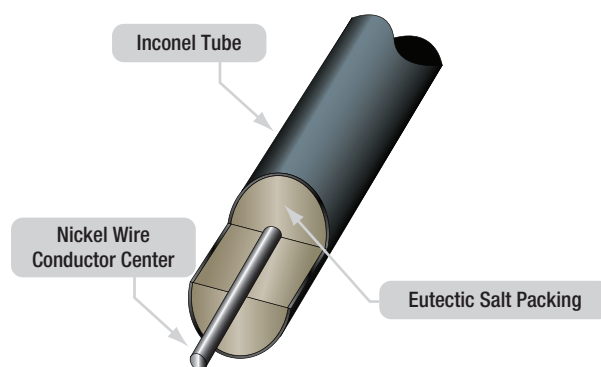


Figure 8-4. Fenwal sensing element.

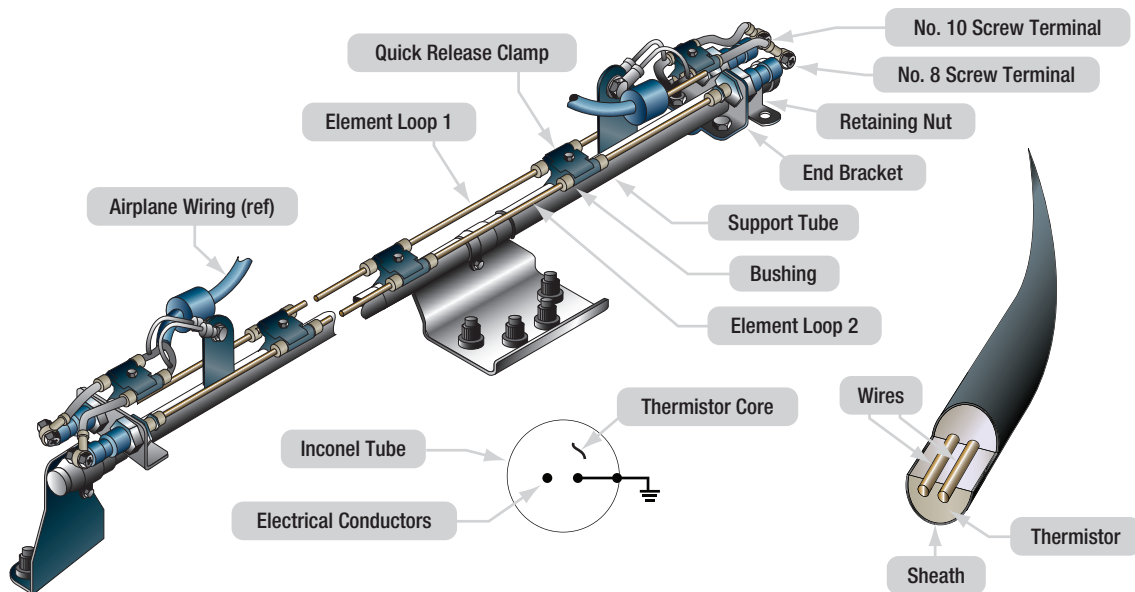


Figure 8-5. Kidde continuous-loop system.

The fire detection control unit monitors this resistance. If the resistance decreases to the overheat set point, an overheat indication occurs in the flight deck. Typically, a 10 second time delay is incorporated for the overheat indication. If the resistance decreases more to the fire set point, a fire warning occurs. When the fire or overheat condition is gone, the resistance of the core material in a Kidde detector system increases to the reset point and the flight deck indications disappear. The rate of change of resistance identifies an electrical short or a fire. The resistance decreases more quickly with an electrical short than with a fire. In some aircraft, in addition to fire and overheat detection, the Kidde continuous loop system supplies nacelle temperature data to the airplane condition monitoring function of the aircraft inflight monitoring system (AIMS).

Sensing Element

The resistance of a sensor varies inversely as it is heated; as sensor temperature is increased, its resistance decreases. Each sensor is composed of two wires embedded in thermistor material that is encased in a heavy wall Inconel tube for high strength at elevated temperatures. The electrical connectors at each end of the sensor are ceramic insulated. The Inconel tubes are shrouded in a perforated stainless steel tube and supported by Teflon impregnated asbestos bushings at intervals. The shroud protects the sensor from breakage due to vibration, abrasion against airplane structure, and damage from maintenance activity.

The resistance of a sensor also varies inversely with its length, the increments of length being resistances in parallel. The heating of a short length of sensor out of a given length requires that the short length be heated above the temperature alarm point so the total resistance of the sensor decreases to the alarm point. This characteristic permits integration of all temperatures throughout the length of the installation rather than sensing only the highest local temperature. The two wires encased within the thermistor material of each Inconel tube form a variable resistance network between themselves, between the detector wire and the Inconel tube, and between each adjacent incremental length of sensor. These variable resistance networks are monitored by the application of 28 volts direct current (DC) to the detector wire from the detector control unit.

Combination Fire and Overheat Warning

The analog signal from the thermistor-sensing element permits the control circuits to be arranged to give a two level response from the same sensing element loop. The first is an overheat warning at a temperature level below the fire warning indicating a general engine compartment temperature rise, such as would be caused by leakage of hot bleed air or combustion gas into the engine compartment. It could also be an early warning of fire and would alert the crew to appropriate action to reduce the engine compartment temperature. The second-level response is at a level above that attainable by a leaking hot gas and is the fire warning.

Temperature Trend Indication

The analog signal produced by the sensing element loop as its temperature changes is converted to signals suitable for flight deck display to indicate engine bay temperature increases from normal. A comparison of the readings from each loop system also provides a check on the condition of the fire detection system, because the two loops should normally read alike.

System Test

The integrity of the continuous loop fire detection system may be tested by actuating a test switch on the flight deck. This switches one end of the sensing element loop from its control circuit to a test circuit built into the control unit, which simulates the sensing element resistance change due to fire. (*Figure 8-6*)

If the sensing element loop is unbroken, the resistance detected by the control circuit is that of the simulated fire, and the alarm is activated. The test demonstrates, in addition to the continuity of the sensing element loop, the integrity of the alarm indicator circuit and the proper functioning of the control circuits. The thermistor properties of the sensing element remain unchanged for the life of the element (no irreversible changes take place when heated); the element functions properly as long as it is electrically connected to the control unit.

Fault Indication

Provision is made in the control unit to output a fault signal which activates a fault indicator whenever the short discriminator circuit detects a short in the sensing element loop. This is a requirement for transport category aircraft because such a short disables the fire detection system.

Dual Loop Systems

Dual loop systems are two complete basic fire detection systems with their output signals connected so that both must signal to result in a fire warning. This arrangement, called AND logic, results in greatly increased reliability against false fire warnings from any cause. Should one of the two loops be found inoperative at the preflight integrity test, a cockpit selector switch disconnects that loop and allows the signal from the other loop alone to activate the fire warning. Since the single operative loop meets all fire detector requirements, the aircraft can be safely dispatched and maintenance deferred to a more convenient time. However, should one of the two loops

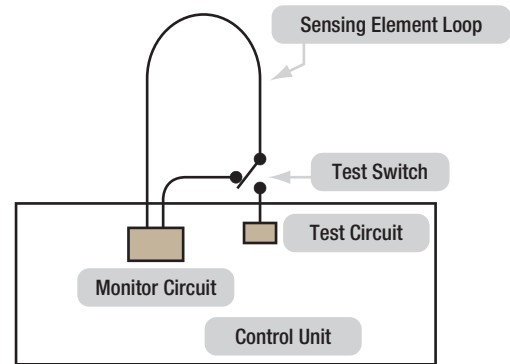


Figure 8-6. Continuously loop fire detection system test circuit.

become inoperative in flight and a fire subsequently occur, the fire signaling loop activates a cockpit fault signal that alerts the flight crew to select single loop operation to confirm the possible occurrence of fire.

Automatic Self Interrogation

Dual loop systems automatically perform the loop switching and decision making function required of the flight crew upon appearance of the fault indication in the cockpit, a function called automatic self interrogation. Automatic self interrogation eliminates the fault indication and assures the immediate appearance of the fire indication should fire occur while at least one loop of the dual loop system is operative. Should the control circuit from a single loop signal fire, the self-interrogation circuit automatically tests the functioning of the other loop. If it tests operative, the circuit suppresses the fire signal because the operative loop would have signaled if a fire existed. If, however, the other loop tests inoperative, the circuit outputs a fire signal. The interrogation and decision takes place in milliseconds, so that no delay occurs if a fire actually exists.

Support Tube Mounted Sensing Elements

For those installations where it is desired to mount the sensing elements on the engine, and in some cases, on the aircraft structure, the support tube mounted element solves the problem of providing sufficient element support points and greatly facilitates the removal and reinstallation of the sensing elements for engine or system maintenance.

Most modern installations use the support tube concept of mounting sensing elements for better maintainability, as well as increased reliability. The sensing element is attached to a prebent stainless steel tube by closely spaced clamps and bushings, where it is supported

from vibration damage and protected from pinching and excessive bending. The support tube mounted elements can be furnished with either single or dual sensing elements.

Being prebent to the designed configuration assures its installation in the aircraft precisely in its designed location, where it has the necessary clearance to be free from the possibility of the elements chafing against engine or aircraft structure. The assembly requires only a few attachment points and, should its removal for engine maintenance be necessary, it is quickly and easily accomplished. Should the assembly require repair or maintenance, it is easily replaced with another assembly, leaving the repair for the shop. Should a sensing element be damaged, it is easily replaced in the assembly.

Fire Detection Control Unit (Fire Detection Card)

The control unit for the simplest type of system typically contains the necessary electronic resistance monitoring and alarm output circuits housed in a hermetically sealed aluminum case fitted with a mounting bracket and electrical connector. For more sophisticated systems, control modules are employed that contain removable control cards with circuitry for individual hazard areas and/or unique functions. In the most advanced applications, the detection system circuitry controls all aircraft fire protection functions, including fire detection and extinguishing for engines, APUs, cargo bays, and bleed-air systems.

PRESSURE TYPE SENSOR RESPONDER SYSTEMS

Some smaller turboprop aircraft are outfitted with pneumatic single point detectors. The design of these detectors is based on the principles of gas laws. The sensing element consists of a closed, helium filled tube connected at one end to a responder assembly. As the element is heated, the gas pressure inside the tube increases until the alarm threshold is reached. At this point, an internal switch closes and reports an alarm to the cockpit. Continuous fault monitoring is included. This type of sensor is designed as a single sensor detection system and does not require a control unit.

PNEUMATIC CONTINUOUS LOOP SYSTEMS

The pneumatic continuous loop systems are also known by their manufacturers' names Lindberg, Systron-Donner, and Meggitt Safety Systems. These systems

are used for engine fire detection of transport type aircraft and have the same function as the Kidde system; however, they work on a different principle. They are typically used in a dual loop design to increase reliability of the system.

The pneumatic detector has two sensing functions. It responds to an overall average temperature threshold and to a localized discrete temperature increase caused by impinging flame or hot gasses. Both the average and discrete temperature are factory set and are not field adjustable. (*Figure 8-7*)

Averaging Function

The fire/overheat detector serves as a fixed-volume device filled with helium gas. The helium gas pressure inside the detector increases in proportion to the absolute temperature and operates a pressure diaphragm that closes an electrical contact, actuating the alarm circuit. The pressure diaphragm within the responder assembly serves as one side of the electrical alarm contact and is the only moving part in the detector. The alarm switch is preset at an average temperature. Typical temperature ranges for average temperature settings are 93°C to 454°C.

Discrete Function

The fire/overheat detector's sensor tube also contains a hydrogen-filled core material. (*Figure 8-8*) Large quantities of hydrogen gas are released from the detector core whenever a small section of the tube is heated to the preset discrete temperature or higher. The core out gassing increases the pressure inside the detector and actuates the alarm switch.

Both the averaging and discrete functions are reversible. When the sensor tube is cooled, the average gas pressure is lowered and the discrete hydrogen gas returns to the core material. The reduction of internal pressure allows the alarm switch to return to its normal position, opening the electrical alarm circuit.

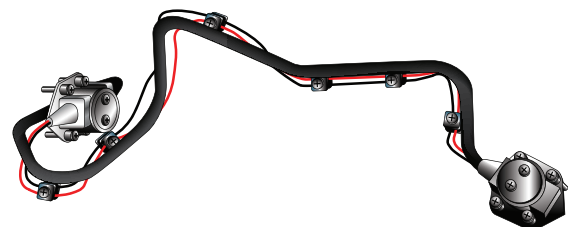


Figure 8-7. Pneumatic dual fire/overheat detector assembly.

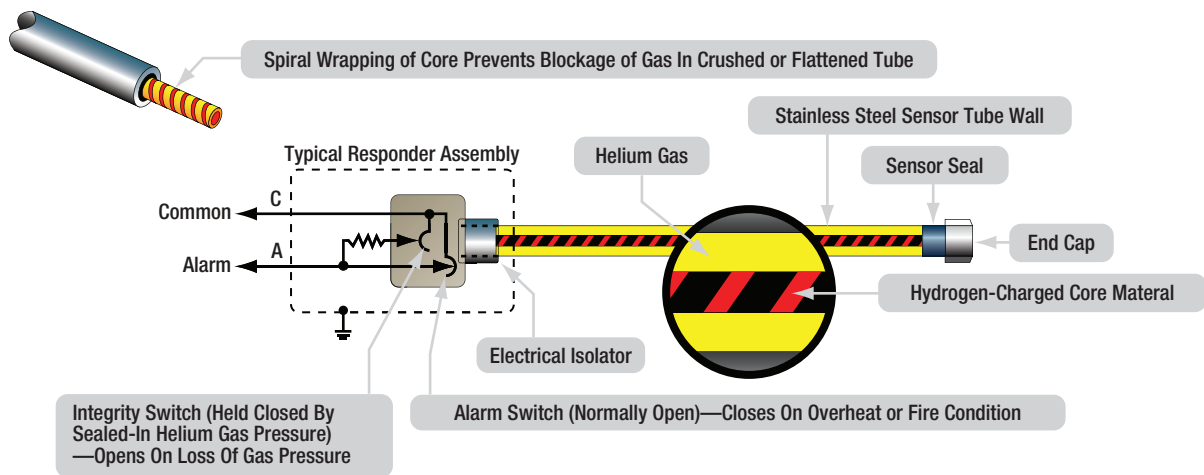


Figure 8-8. Pneumatic pressure loop detector system.

Figure 8-9 shows a typical aircraft fire detection system in which a control module monitors two loops of up to four pneumatic detectors each, connected in parallel. The control module responds directly to an alarm condition and continuously monitors the wiring and integrity of each loop.

The normally open alarm switch closes upon an overheat or fire condition, causing a short circuit between terminals A and C. During normal operation, a resistance value is maintained across the terminals by a normally closed integrity switch.

Loss of sensor gas pressure opens the integrity switch, creating an open circuit across the terminals of the faulted detector. In addition to the pressure activated alarm switch, there is a second integrity switch in the detector that is held closed by the averaging gas pressure at all temperatures down to -54°C .

If the detector should develop a leak, the loss of gas pressure would allow the integrity switch to open and signal a lack of detector integrity. The system then does not operate during test.

SMOKE, FLAME, AND CARBON MONOXIDE DETECTION SYSTEMS

SMOKE DETECTORS

A smoke detection system monitors the lavatories and cargo baggage compartments for the presence of smoke, which is indicative of a fire condition. Smoke detection instruments that collect air for sampling are mounted in the compartments in strategic locations. A smoke detection system is used where the type of fire anticipated is expected to generate a substantial amount of smoke before temperature changes are sufficient to actuate a heat detection system. Two common types used are light refraction and ionization.

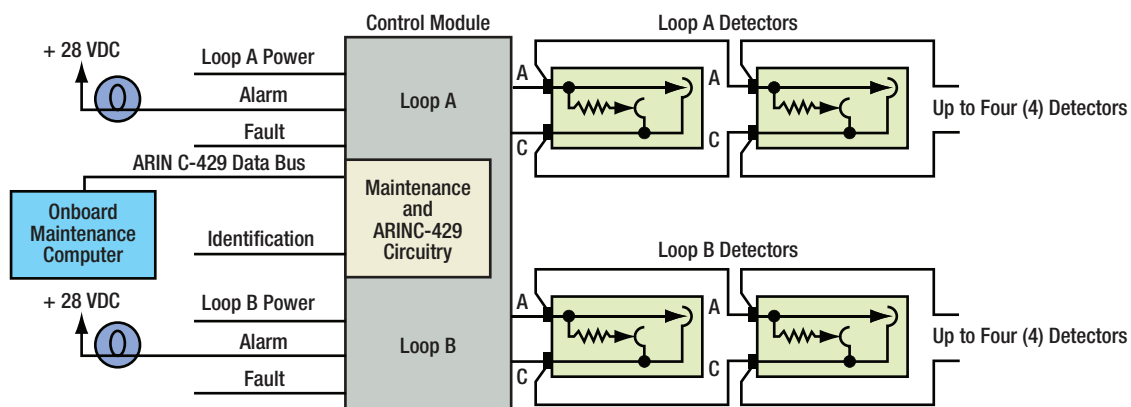


Figure 8-9. Aircraft detection system control module.

LIGHT REFRACTION TYPE

The light refraction type of smoke detector contains a photoelectric cell that detects light refracted by smoke particles. Smoke particles refract the light to the photoelectric cell and, when it senses enough of this light, it creates an electrical current that sets off a light.

IONIZATION TYPE

Some aircraft use an ionization type smoke detector. The system generates an alarm signal (both horn and indicator) by detecting a change in ion density due to smoke in the cabin. The system is connected to the 28 volt DC electrical power supplied from the aircraft. Alarm output and sensor sensitive checks are performed simply with the test switch on the control panel.

FLAME DETECTORS

Optical sensors, often referred to as flame detectors, are designed to alarm when they detect the presence of prominent, specific radiation emissions from hydrocarbon flames. The two types of optical sensors available are infrared (IR) and ultraviolet (UV), based on the specific emission wavelengths that they are designed to detect. IR-based optical flame detectors are used primarily on light turboprop aircraft and helicopter engines. These sensors have proven to be very dependable and economical for these applications.

When radiation emitted by the fire crosses the airspace between the fire and the detector, it impinges on the detector front face and window. The window allows a

broad spectrum of radiation to pass into the detector where it strikes the sensing device filter. The filter allows only radiation in a tight waveband centered on 4.3 micrometers in the IR band to pass on to the radiation-sensitive surface of the sensing device. The radiation striking the sensing device minutely raises its temperature causing small thermoelectric voltages to be generated. These voltages are fed to an amplifier whose output is connected to various analytical electronic processing circuits. The processing electronics are tailored exactly to the time signature of all known hydrocarbon flame sources and ignores false alarm sources, such as incandescent lights and sunlight. Alarm sensitivity level is accurately controlled by a digital circuit. (*Figure 8-10*)

CARBON MONOXIDE DETECTORS

Carbon monoxide is a colorless, odorless gas that is a byproduct of incomplete combustion. Its presence in the breathing air of human beings can be deadly. To ensure crew and passenger safety, carbon monoxide detectors are used in aircraft cabins and cockpits. They are most often found on reciprocating engine aircraft with exhaust shroud heaters and on aircraft equipped with a combustion heater. Turbine bleed air, when used for heating the cabin, is tapped off of the engine upstream of the combustion chamber. Therefore, no threat of carbon monoxide presence is posed.

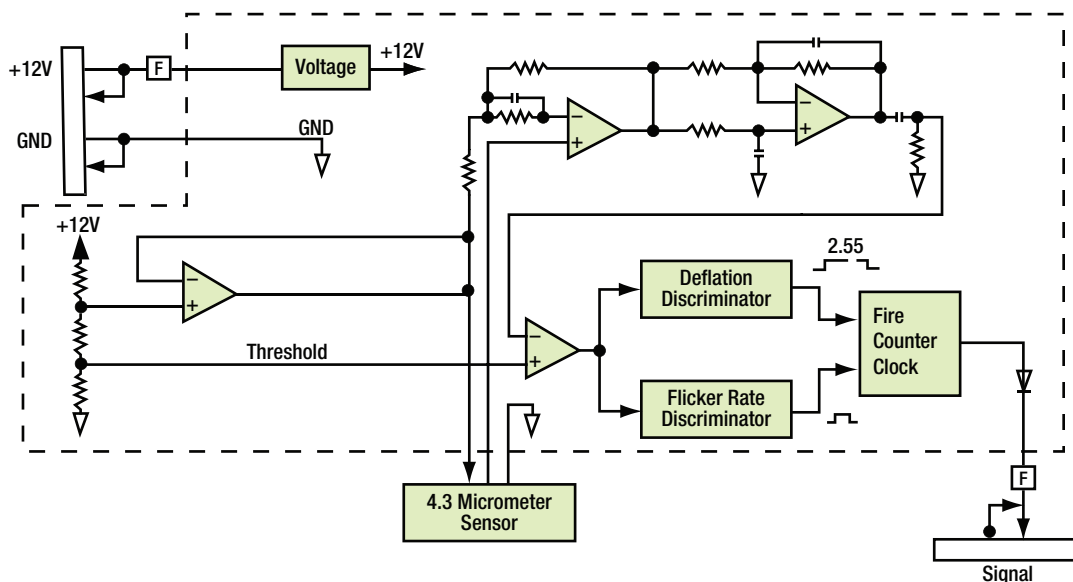


Figure 8-10. Infrared (IR) based optical flame detector.

Carbon monoxide gas is found in varying degrees in all smoke and fumes of burning carbonaceous substances. Exceedingly small amounts of the gas are dangerous if inhaled. A concentration of as little as 2 parts in 10 000 may produce headache, mental dullness, and physical lethargy within a few hours. Prolonged exposure or higher concentrations may cause death.

There are several types of carbon monoxide detectors. Electronic detectors are common. Some are panel mounted and others are portable. Chemical color-change types are also common. These are mostly portable. Some are simple buttons, cards, or badges that have a chemical applied to the surface. Normally, the color of the chemical is tan. In the presence of carbon monoxide, the chemical darkens to gray or even black. The transition time required to change color is inversely related to the concentration of CO present. At 50 parts per million, the indication is apparent within 15 to 30 minutes. A concentration of 100 parts per million changes the color of the chemical in as little as 2-5 minutes. As concentration increases or duration of exposure is prolonged, the color evolves from gray to dark gray to black.

FIRE EXTINGUISHING SYSTEMS

All types of aircraft store fire extinguishing agent on board to be accessed and deployed manually or automatically by the flight crew when a fire occurs. Storage of fire extinguishing agent is either in fixed containers, portable containers or both. Transport aircraft have both. Fixed containers are typically spherical in shape and are permanently installed in the aircraft. Apparatus to expel and direct the agent onto a fire is part of a fixed container system. Portable containers are also used. These are stored using quick release latches so that a user may quickly grab a container and hand carry it to the fire for deployment usually by squeezing a trigger type handle.

FIRE EXTINGUISHER AGENTS

Various agents are manufactured and used on aircraft. They are used in both fixed and portable systems. The following is a list of extinguishing agents and the class of fires for which each is appropriate.

1. Water - class A. Water deprives the fire of oxygen and cools the material below its ignition temperature. It soaks the burning material to

prevent it from igniting again once the fire is extinguished. A water fire extinguisher should only be used on Class A fires. Transport aircraft use portable water fire extinguishers in the passenger cabin, however not on an electrical fire. Since water is conductive, spraying water on an electrical fire could cause electrocution and will certainly cause damage to the electrical equipment. Note that water fire extinguishers have antifreeze as well as water inside to ensure service should temperatures drop below freezing 0°C.

2. Carbon dioxide (CO₂) - class B or C. Carbon dioxide acts as a blanketing agent. It smothers a fire and deprives it of oxygen. Caution must be exercised when using a CO₂ fire extinguisher in a confined area. The operator of the CO₂ fire extinguisher may also be deprived of oxygen. Because of this, CO₂ is not recommended for hand held fire extinguishers for internal aircraft use.

Carbon dioxide is an effective extinguishing agent. It is most often used in fire extinguishers that are available on the ramp to fight fires on the exterior of the aircraft, such as engine or APU fires. CO₂ has also been used for engine fire extinguishing on older transport aircraft. It can extinguish flammable fluid fires and fires involving electrical equipment although Halon is preferred for electrical fires. Carbon dioxide is noncombustible and does not react with most substances. It has a boiling point of 78.8°C. As such, it provides its own (vapor) pressure for discharge from the storage vessel, except in extremely cold climates where a booster charge of nitrogen may be added to winterize the system.

Carbon dioxide is about 1-1/2 times as heavy as air, which gives it the ability to replace air above burning surfaces and maintain a smothering atmosphere. CO₂ is effective as an extinguishing agent primarily because it dilutes the air and reduces the oxygen content so that combustion is no longer supported.

Under most conditions, some cooling effect is also realized. Carbon dioxide is considered only mildly toxic, but it can cause unconsciousness and death by suffocation if the victim is allowed to breathe CO₂ in fire extinguishing concentrations for 20 to 30 minutes. CO₂ is not effective as an extinguishing agent on fires

involving chemicals containing their own oxygen supply, such as cellulose nitrate (used in some aircraft paints). Also, fires involving magnesium and titanium cannot be extinguished by CO₂. Once used, a carbon dioxide fire extinguisher must be replaced.

3. Dry powder chemicals - class B, C or D. Dry powder extinguishers, while effective on Class B and C fires, are the best for use on Class D fires. The method of operation of dry powder fire extinguishers varies. Some containers use gas cartridge charges or store the agent under pressure within the container to force the powder charge out of the container onto the fire. Dry powder may also come in a large container or barrel from which it is applied by hand using a scoop or bucket. Examples of dry powder chemicals are sodium bicarbonate, potassium bicarbonate, and ammonium phosphate.
4. Halogenated hydrocarbons - class A, B, or C. Halogenated hydrocarbon (halon) fire extinguishing agents come in many chemical formulas. Halon 1211 (Bromochlorodifluoromethane, CBrClF₂) and Halon 1301 (Bromotrifluoromethane, CBrF₃) are commonly used in aviation depending on the application. Halon extinguishing agents smother a fire and deprive it of oxygen. They are volatile with part of their effect due to cooling of the burning materials through rapid expansion of the agent. Halon 1301 and Halon 1211 are less toxic than other halon formulas and are very effective. They are stored in pressurized containers. Halon 1301 creates satisfactory vapor pressure to expel itself. Halon 1211 has a higher boiling point and may require a nitrogen charge or a 1301 charge to pressurize adequately for effective discharge.

Dry powder is not recommended for use on aircraft fires except on metal fires. This is primarily because the leftover chemical residues and dust often make cleanup difficult, and can damage electronic or other delicate equipment. As such, dry powder is useful for Class D aircraft wheel and brake fires.

For over 45 years, halogenated hydrocarbons (Halons) have been practically the only fire extinguishing agents used in civil transport aircraft. However, Halon is an ozone depleting and global warming chemical, and its production has been banned by international agreement.

Although Halon usage has been banned in some parts of the world, aviation has been granted an exemption because of its unique operational and fire safety requirements. Halon replacement agents that have been found to be acceptable for environmental protection are available. To date, some of these are the halocarbons HCFC Blend B, HFC-227ea, and HFC-236fa.

Halon is extremely effective on a per unit weight basis over a wide range of aircraft environmental conditions. It is electrically nonconducting, evaporate rapidly, leaves no residue, and requires no cleanup or neutralization. NOTE: Do not use Halons on a Class D fire. Halon agents may react vigorously with the burning metal.

FIXED CONTAINER FIRE EXTINGUISHING SYSTEMS

Transport aircraft have fixed fire extinguishing systems installed in the following locations:

1. Turbine engine compartments.
2. APU compartments.
3. Cargo and baggage compartments.
4. Lavatories.

Older aircraft with reciprocating engines used CO₂ as an extinguishing agent, but all newer aircraft designs with turbine engines use Halon or equivalent extinguishing agent, such as halocarbon clean agents.

The fixed fire extinguisher systems used in most engine fire and cargo compartment fire protection systems are designed to dilute the atmosphere with an inert agent that does not support combustion. Many systems use perforated tubing or discharge nozzles to distribute the extinguishing agent. High rate of discharge (HRD) systems use open end tubes to deliver a large quantity of extinguishing agent in 1 to 2 seconds. The most common extinguishing agent still used today is Halon 1301.

CONTAINERS

Most fixed fire extinguishing agent containers on high performance and transport aircraft are the high-rate-of discharge (HRD) type. They typically store a liquid halogenated extinguishing agent (or other) and a pressurized gas (typically nitrogen) to assist in the propulsion of the agent from the container. The containers are normally manufactured from stainless steel. Depending upon design considerations, alternate materials are available, including titanium.

Fixed fire agent containers are available in a wide range of capacities. They are produced under department of transportation specifications or exemptions of the manufacturer's country. Most aircraft containers are spherical in design, which provides the lightest weight possible. However, cylindrical shapes are available where space limitations are a factor. Each container incorporates a temperature/pressure sensitive safety relief diaphragm that prevents container pressure from exceeding container test pressure in the event of exposure to excessive temperatures.

(Figure 8-11 and Figure 8-12)

DISCHARGE VALVES

Discharge valves are installed on the containers. A cartridge (squib) and frangible disk type valve are installed in the outlet of the discharge valve assembly. Special assemblies having solenoid operated or manually operated seat type valves are also available. Two types of cartridge disk release techniques are used. Standard release type uses a slug driven by explosive energy to rupture a segmented closure disc.

For high temperature or hermetically sealed units, a direct explosive impact type cartridge is used that applies fragmentation impact to rupture a prestressed corrosion resistant steel diaphragm. Most containers use conventional metallic gasket seals that facilitate refurbishment following discharge. (Figure 8-13)



Figure 8-11. Built-in non-portable fire extinguisher containers (HRD bottles) on an airliner.

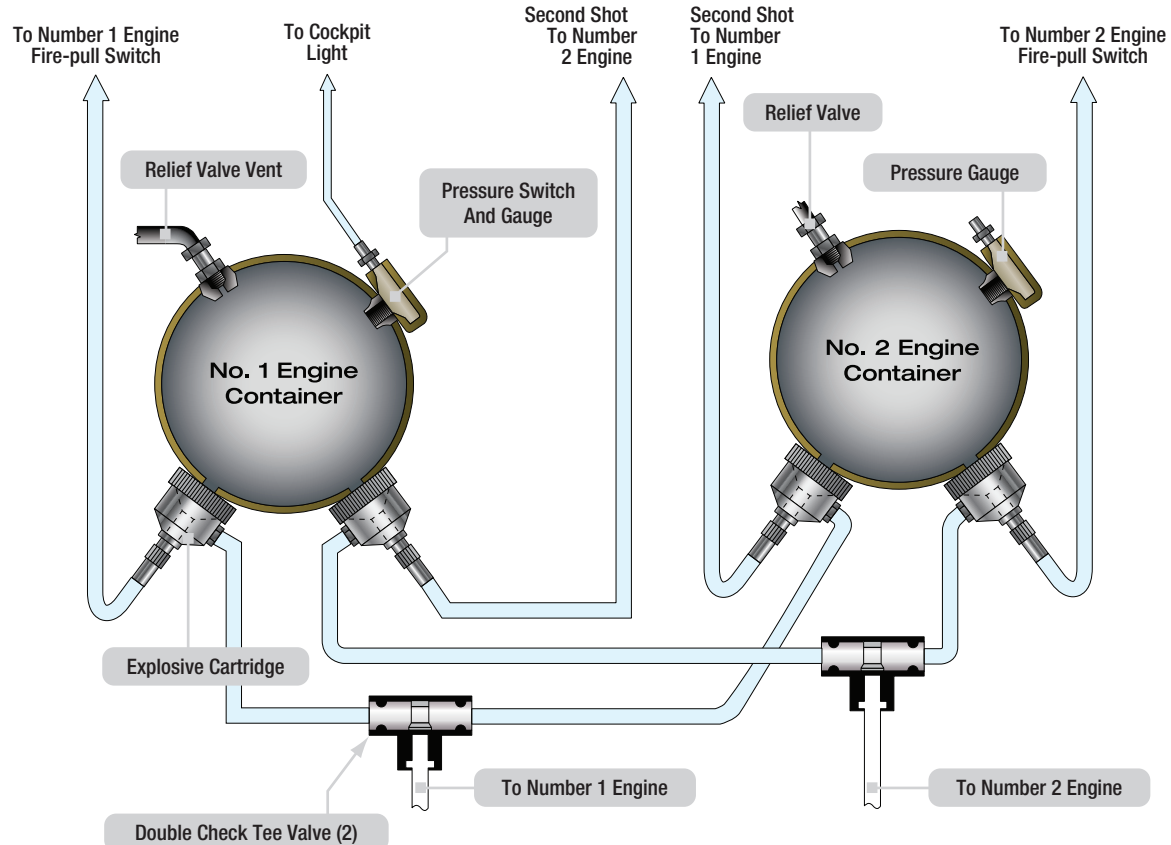


Figure 8-12. Diagram of fire extinguisher containers (HRD bottles).



Figure 8-13. Discharge valve (left) and cartridge, or squib (right).

PRESSURE INDICATION

A wide range of diagnostics is utilized to verify the fire extinguisher agent charge status. A simple visual indication gauge is available, typically a helical bourdon type indicator that is vibration resistant.

A combination gauge switch visually indicates actual container pressure and also provides an electrical signal if container pressure is lost, precluding the need for discharge indicators.

A ground checkable diaphragm type low pressure switch is commonly used on hermetically sealed containers. The Kidde system has a temperature compensated pressure switch that tracks the container pressure variations with temperatures by using a hermetically sealed reference chamber.

TWO WAY CHECK VALVE

Two way check valves are required in a two shot system to prevent the extinguisher agent from a reserve container from backing up into the previous emptied main container. Valves are supplied with either MS-33514 or MS-33656 fitting configurations.

DISCHARGE INDICATORS

Discharge indicators provide immediate visual evidence of container discharge on fire extinguishing systems. Two kinds of indicators can be furnished: thermal and discharge. Both types are designed for aircraft and skin mounting. (Figure 8-14)

Thermal Discharge Indicator (Red Disk)

The thermal discharge indicator is connected to the fire container relief fitting and ejects a red disk to show when container contents have dumped overboard due to excessive heat. The agent discharges through the opening left when the disk blows out. This gives the flight and maintenance crews an indication that the fire extinguisher container needs to be replaced before next flight.



Figure 8-14. Discharge indicators.

Normal Discharge Indicator (Yellow Disk)

If the flight crew activates the fire extinguisher system, a yellow disk is ejected from the skin of the aircraft fuselage. This is an indication for the maintenance crew that the fire extinguishing system was activated by the flight crew, and the fire extinguishing container needs to be replaced before next flight.

FIRE SWITCH

The engine and APU fire switches are typically installed on the center overhead panel or center console on the flight deck. (Figure 8-15) When an engine fire switch is activated, the following happens: the engine stops because the fuel control shuts off, the engine is isolated from the aircraft systems, and the fire extinguishing system is activated. Some aircraft use fire switches that need to be pulled and turned to activate the system, while others use a push type switch with a guard. To prevent accidental activation of the fire switch, a lock

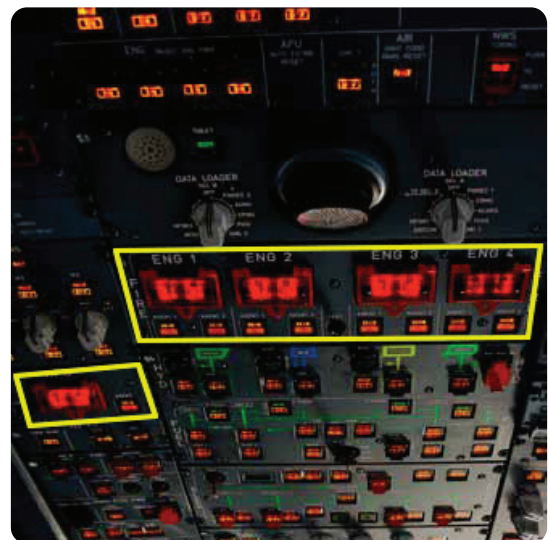


Figure 8-15. Engine and APU fire switches on the cockpit center overhead panel.

is installed that releases the fire switch only when a fire has been detected. This lock can be manually released by the flight crew if the fire detection system malfunctions. (Figure 8-16)

FIXED FIRE PROTECTION: CARGO COMPARTMENTS

Protection against fire in cargo compartments is mandatory. Transport aircraft need to have the following provisions for each cargo or baggage compartment:

1. The detection system must provide a visual indication to the flight crew within 1 minute after the start of a fire.
2. The system must be capable of detecting a fire at a temperature significantly below that at which the structural integrity of the airplane is substantially decreased.
3. There must be means to allow the crew to check, in flight, the functioning of each fire detector circuit.

CARGO COMPARTMENT CLASSIFICATION

A Class A cargo or baggage compartment is one in which the presence of a fire would be easily discovered by a crew member while at his or her station and each part of the compartment is easily accessible in flight.

A Class B cargo, or baggage compartment, is one in which there is sufficient access in flight to enable a crew member to effectively reach any part of the compartment with the contents of a hand fire extinguisher. When the access provisions are being used, no hazardous quantity of smoke, flames, or extinguishing agent enters any compartment occupied by the crew or passengers. There is a separate approved smoke detector or fire detector system to give warning at the pilot or flight engineer station.

A Class C cargo, or baggage compartment, is one not meeting the requirements for either a Class A or B compartment but in which:

1. There is a separate approved smoke detector or fire detector system to give warning at the pilot or flight engineer station.
2. There is an approved built-in fire extinguishing or suppression system controllable from the cockpit.
3. There are means to exclude hazardous quantities of smoke, flames, or extinguishing agent from any compartment occupied by the crew or passengers.
4. There are means to control ventilation and drafts within the compartment so that the extinguishing agent used can control any fire that may start within the compartment.

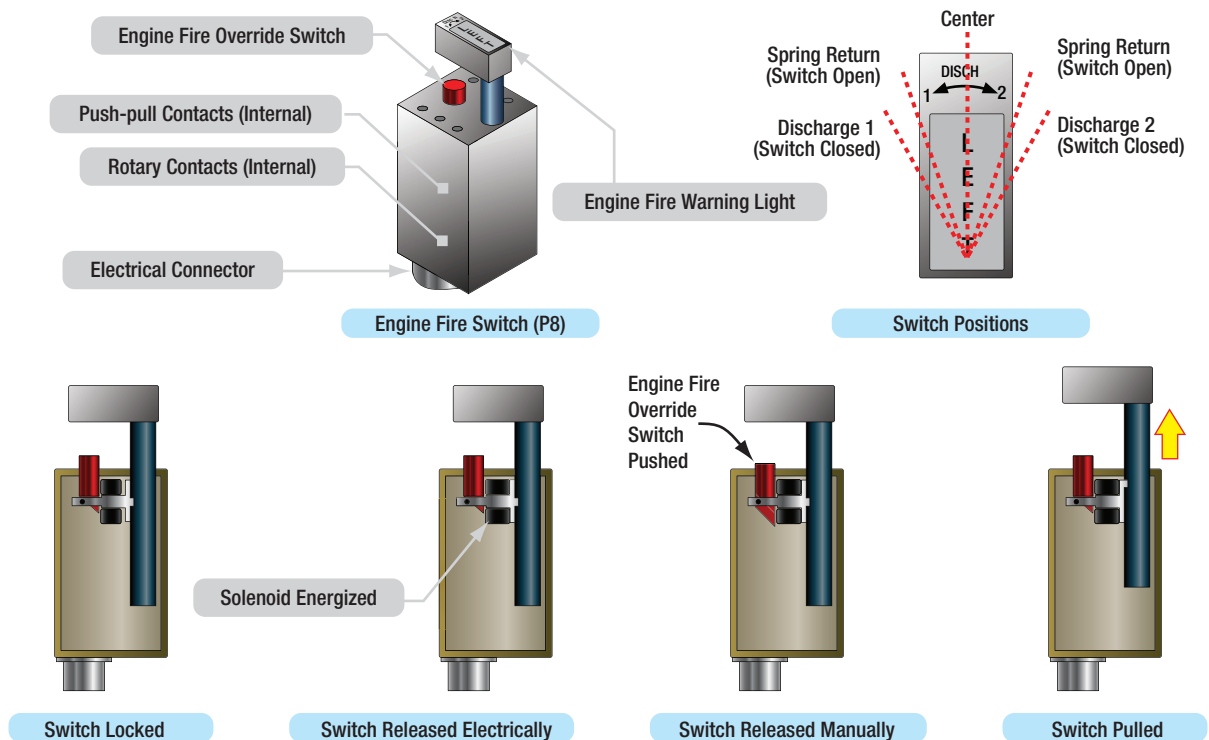


Figure 8-16. Engine fire switch operation.

A Class E cargo compartment is one on airplanes used only for the carriage of cargo and in which:

1. There is a separate approved smoke or fire detector system to give warning at the pilot or flight engineer station.
2. The controls for shutting off the ventilating airflow to, or within, the compartment are accessible to the flight crew in the crew compartment.
3. There are means to exclude hazardous quantities of smoke, flames, or noxious gases from the flight crew compartment.
4. The required crew emergency exits are accessible under any cargo loading condition.

CARGO COMPARTMENT FIRE PROTECTION AND WARNING

Cargo compartments typically combine a smoke type detection system with a fixed Halon extinguishing system. A smoke detector behind the sidewall in each compartment examines the air drawn through it by a circulation fan(s). If smoke is present, the system gives warnings on the flight deck. (*Figure 8-17*)

The following indications occur in the cockpit if there is smoke in a cargo compartment:

1. Master warning lights come on.
2. Fire warning aural operates.
3. A cargo fire warning message shows.
4. Cargo fire warning light comes on.

The master warning lights and fire warning aural are prevented from operating during part of the takeoff operation. (*Figure 8-18*)

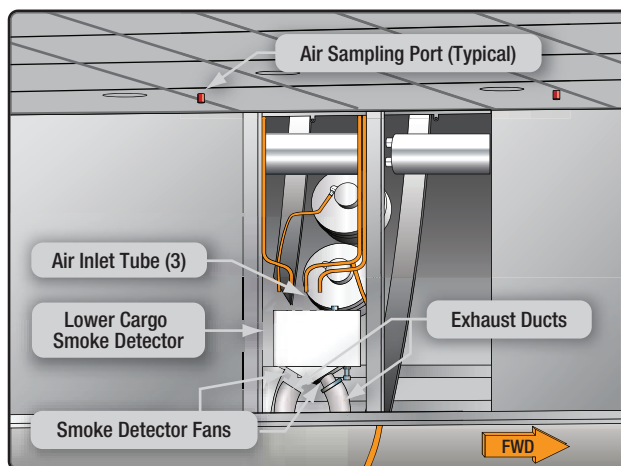


Figure 8-17. Smoke detector installation.

Smoke Detector Systems

The optical smoke detector consists of source light emitting diodes (LEDs), intensity monitor photodiodes, and scatter detector photodiodes. Inside the smoke detection chamber, air flows between a source (LED) and a scatter detector photodiode. Usually, only a small amount of light from the LED gets to the scatter detector. If the air has smoke in it, the smoke particles reflect more light on the scatter detector. This causes an alarm signal.

The intensity monitor photodiode makes sure that the source LED is on and keeps the output of the source LED constant. This configuration also finds contamination of the LED and photodiodes. A defective diode, or contamination, causes the detector to change to the other set of diodes. The detector sends a fault message if this occurs.



Figure 8-18. Cargo fire detection warning.

The smoke detector has multiple sampling ports. The fans draw air from the sampling ports through a water separator and a heater unit to the smoke detector. (Figure 8-19)

Cargo Compartment Extinguishing System

The cargo compartment extinguishing system is activated by the flight crew if the smoke detectors detect smoke in the cargo compartment. Some aircraft are outfitted with two types of fire extinguisher containers. The first system is the dump system that releases the extinguishing agent directly when the cargo fire discharge switch is activated. This action extinguishes the fire.

The second system is the metered system. After a time delay, the metered bottles discharge slowly and at a controlled rate through the filter regulator. Halon from the metered bottles replaces the extinguishing agent leakage. This keeps the correct concentration of extinguishing agent in the cargo compartment to keep the fire extinguished for 180 minutes.

The fire extinguishing bottles contain Halon 1301 or equivalent fire extinguishing agent pressurized with nitrogen. Tubing connects the bottles to discharge nozzles in the cargo compartment ceilings.

The extinguishing bottles are outfitted with squibs. The squib is an electrically operated explosive device. It is adjacent to a bottle diaphragm that can break. The diaphragm normally seals the pressurized bottle. When the cargo discharge switch is activated, the squib fires and the explosion breaks the diaphragm. Nitrogen pressure inside the bottle pushes the Halon through the discharge port into the cargo compartment. When the bottle discharges, a pressure switch is activated that

sends an indication to the flight deck that a bottle has been discharged. Flow control valves are incorporated if the bottles can be discharged in multiple compartments. The flow control valves direct the extinguishing agent to the selected cargo compartment. (Figure 8-20)

FIXED FIRE PROTECTION: LAVATORIES

Airplanes that have a passenger capacity of 20 or more are equipped with a smoke detector system that monitors the lavatories for smoke. Smoke indications provide a warning light in the cockpit or provide a warning light or audible warning at the lavatory and at flight attendant stations that would be readily detected by a flight attendant. Each lavatory must have a built in fire extinguisher that discharges automatically. The smoke detector is located in the ceiling of the lavatory. (Figure 8-21)

LAVATORY SMOKE DETECTOR AND WARNING SYSTEMS

Refer to Figure 8-22. The lavatory smoke detector is powered by the 28-volt DC left/right main DC bus. If there is smoke in the sensing chamber of the smoke detector, the alarm LED (red) comes on. The timing circuit makes an intermittent ground. The warning horn and lavatory call light operate intermittently. The smoke detection circuit makes a ground for the relay. The energized relay makes a ground signal for the overhead electronics unit (OEU) in the central monitoring systems (CMS).

This interface gives these indications: lavatory master call light flashes, cabin system control panel (CSCP) and cabin area control panel (CACP) pop-up window shows, and the lavatory call chime operates. Push the lavatory call reset switch or the smoke detector interrupt

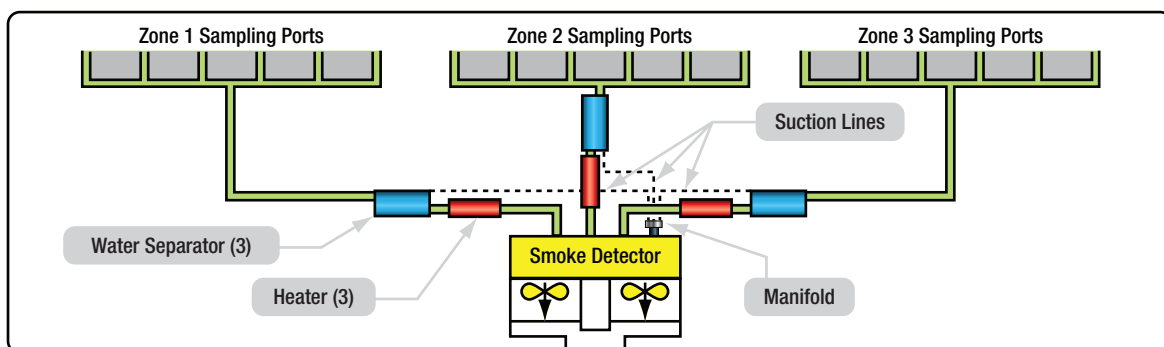


Figure 8-19. Cargo compartment smoke detector system.

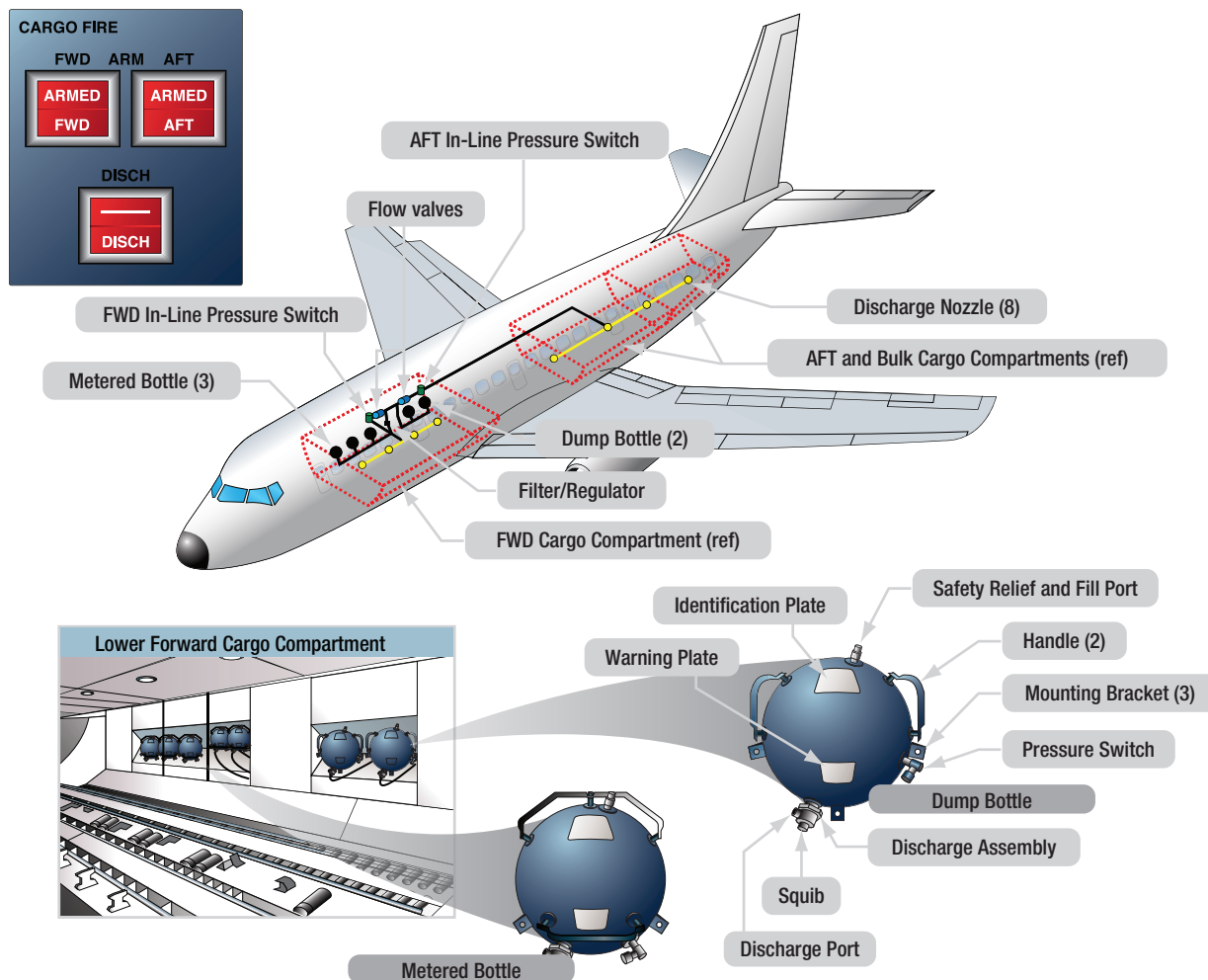


Figure 8-20. Cargo and baggage compartment extinguishing system.

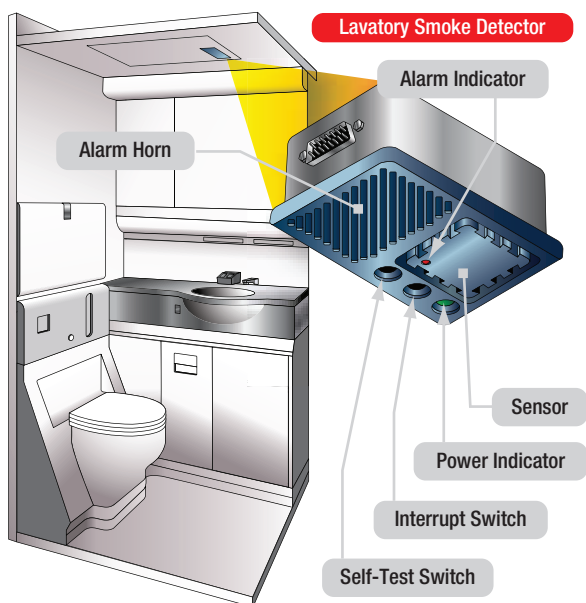


Figure 8-21. Lavatory smoke detector.

switch to cancel the smoke indications. If there is still smoke in the lavatory, the alarm LED (red) stays on. All smoke indications go away automatically when the smoke is gone.

LAVATORY FIRE EXTINGUISHER SYSTEMS

The lavatory compartment is outfitted with a fire extinguisher bottle to extinguish fires in the waste compartment. The fire extinguisher is a bottle with two nozzles. The bottle contains pressurized Halon 1301 or equivalent fire extinguishing agent. When the temperature in the waste compartment reaches approximately 76.66°C, the solder that seals the nozzles melt and the Halon is discharged. Weighing the bottle is often the only way to determine if the bottle is empty or full. (Figure 8-23)

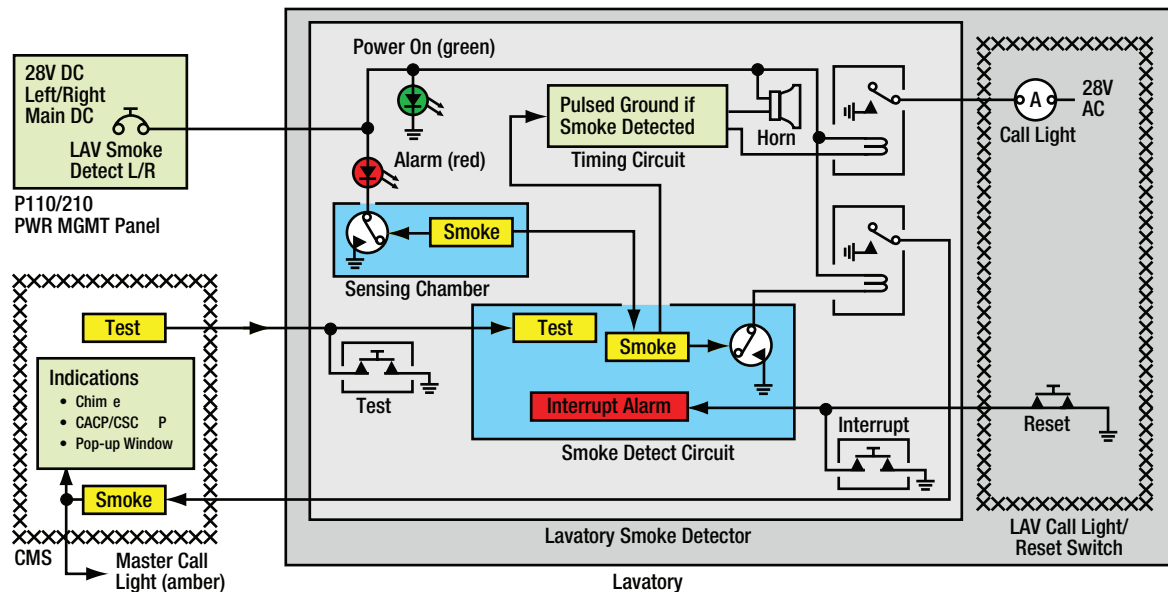


Figure 8-22. Lavatory smoke detector diagram.

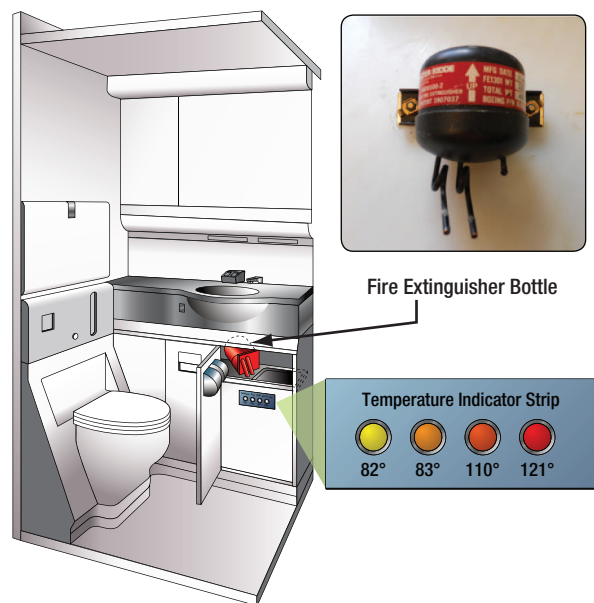


Figure 8-23. Lavatory fire extinguishing bottle.

FIRE DETECTION SYSTEM MAINTENANCE

Fire detector sensing elements are located in many high activity areas around aircraft engines. Their location, together with their small size, increases the chance of damage to the sensing elements during maintenance. An inspection and maintenance program for all types of continuous loop systems should include the following visual checks.

NOTE: These procedures are examples and should not be used to replace the manufacturer's instructions.

Sensing elements of a continuous loop system should be inspected for the following:

1. Cracked or broken sections caused by crushing or squeezing between inspection plates, cowl panels, or engine components.
2. Abrasion caused by rubbing of the element on cowlings, accessories, or structural members.
3. Pieces of safety wire, or other metal particles, that may short the spot-detector terminals.
4. Condition of rubber grommets in mounting clamps that may be softened from exposure to oils or hardened from excessive heat.
5. Dents and kinks in sensing element sections. Limits on the element diameter, acceptable dents and kinks, and degree of smoothness of tubing contour are specified by manufacturers. No attempt should be made to straighten any acceptable dent or kink, since stresses may be set up that could cause tubing failure. (*Figure 8-24*)
6. Nuts at the end of the sensing elements should be inspected for tightness and safety wire. (*Figure 8-25*) Loose nuts should be re-torqued to the value specified by the manufacturer's instructions. Some types of sensing element connection joints require the use of copper crush gaskets. These should be replaced any time a connection is separated.
7. If shielded flexible leads are used, they should be inspected for fraying of the outer braid. The braided sheath is made up of many fine metal strands woven into a protective covering

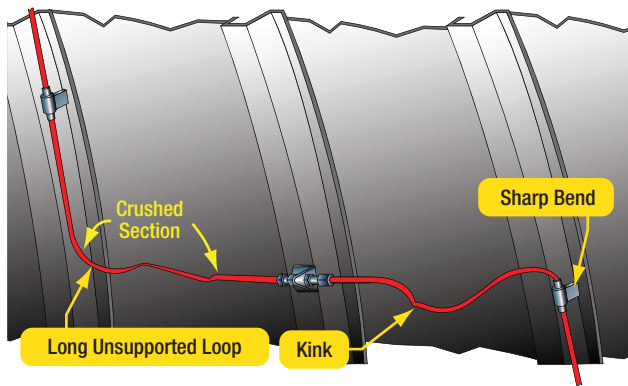


Figure 8-24. Sensing element defects.

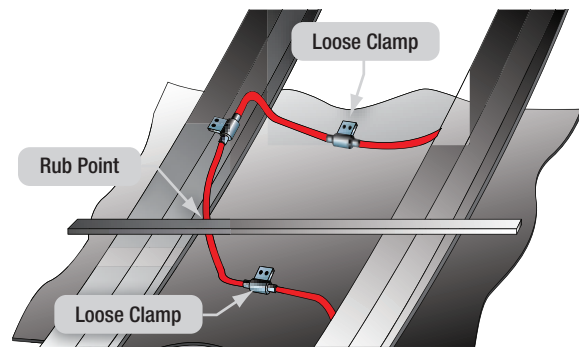


Figure 8-26. Rubbing interference.

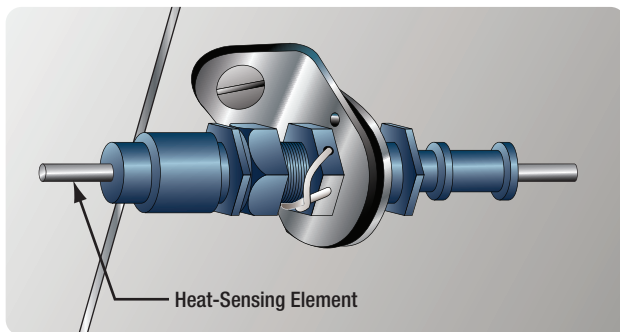


Figure 8-25. Connector joint fitting attached to the structure.

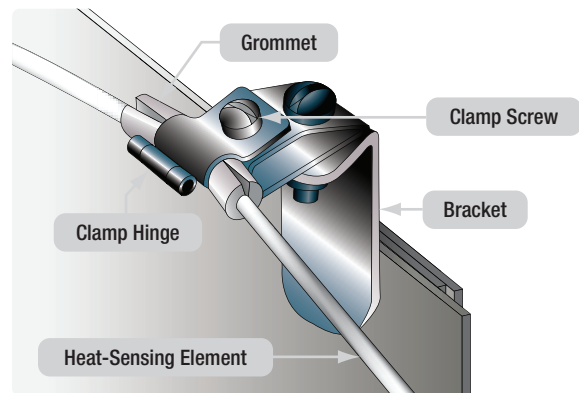


Figure 8-27. Inspection of fire detector loop clamp.

surrounding the inner insulated wire. Continuous bending of the cable or rough treatment can break these fine wires, especially those near the connectors.

8. Sensing element routing and clamping should be inspected carefully. (*Figure 8-26*) Long, unsupported sections may permit excessive vibration that can cause breakage. The distance between clamps on straight runs, usually about 20-25 cm, is specified by each manufacturer. At end connectors, the first support clamp usually is located about 12 cm from the end connector fittings. In most cases, a straight run of one inch is maintained from all connectors before a bend is started, and an optimum bend radius of 3 inches is normally adhered to.
9. Interference between a cowl brace and a sensing element can cause rubbing. This interference may cause wear and short the sensing element.
10. Grommets should be installed on the sensing element so that both ends are centered on its clamp. The split end of the grommet should face the outside of the nearest bend. Clamps and grommets should fit the element snugly. (*Figure 8-27*)

FIRE DETECTION SYSTEM TROUBLESHOOTING

The following troubleshooting procedures represent the most common difficulties encountered in engine fire detection systems:

1. Intermittent alarms are most often caused by an intermittent short in the detector system wiring. Such shorts may be caused by a loose wire that occasionally touches a nearby terminal, a frayed wire brushing against a structure, or a sensing element rubbing against a structural member long enough to wear through the insulation. Intermittent faults often can be located by moving wires to recreate the short.
2. Fire alarms and warning lights can occur when no engine fire or overheat condition exists. Such false alarms can be most easily located by disconnecting the engine sensing loop connections from the control unit. If the false alarm ceases when the engine sensing loop is disconnected, the fault is in the disconnected sensing loop, which should be examined for areas that have been bent into contact with hot parts of the engine. If no bent element can be found,

the shorted section can be located by isolating the connecting elements consecutively around the entire loop.

3. Kinks and sharp bends in the sensing element can cause an internal wire to short intermittently to the outer tubing. The fault can be located by checking the sensing element with an ohm meter while tapping the element in the suspected areas to produce the short.
4. Moisture in the detection system seldom causes a false fire alarm. If, however, moisture does cause an alarm, the warning persists until the contamination is removed, or boils away, and the resistance of the loop returns to its normal value.
5. Failure to obtain an alarm signal when the test switch is actuated may be caused by a defective test switch or control unit, the lack of electrical power, inoperative indicator light, or an opening in the sensing element or connecting wiring. When the test switch fails to provide an alarm, the continuity of a two wire sensing loop can be determined by opening the loop and measuring the resistance. In a single wire, continuous loop system, the center conductor should be grounded.

FIRE EXTINGUISHER SYSTEM MAINTENANCE

Regular maintenance of fire extinguisher systems typically includes such items as the inspection and servicing of fire extinguisher bottles (containers), removal and reinstallation of cartridge and discharge valves, testing of discharge tubing for leakage, and electrical wiring continuity tests. The following paragraphs contain details of some of the most typical maintenance procedures.

CONTAINER PRESSURE CHECK

Fire extinguisher containers are checked periodically to determine that the pressure is between the prescribed minimum and maximum limits. Changes of pressure with ambient temperatures must also fall within prescribed limits. The graph shown in *Figure 8-28* is typical of the pressure temperature curve graphs that provide maximum and minimum gauge readings. If the pressure does not fall within the graph limits, the extinguisher container is replaced.

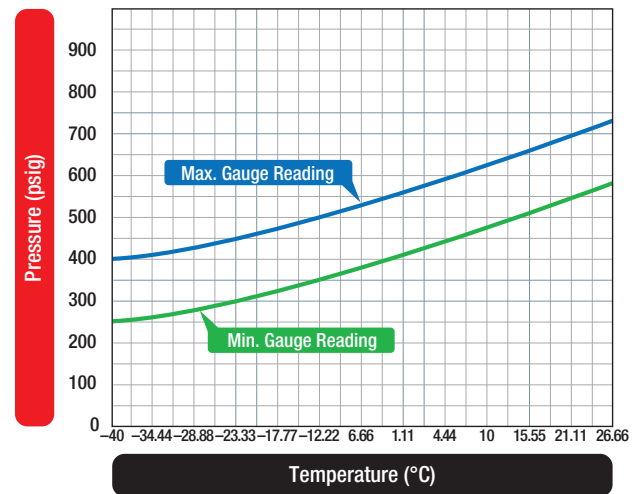


Figure 8-28. Fire extinguisher container pressure-temperature chart.

DISCHARGE CARTRIDGES

The service life of fire extinguisher discharge cartridges is calculated from the manufacturer's date stamp, which is usually placed on the face of the cartridge. The cartridge service life recommended by the manufacturer is usually in terms of years. Cartridges are available with a service life of 5 years or more. To determine the unexpired service life of a discharge cartridge, it is usually necessary to remove the electrical leads and discharge line from the plug body, which can then be removed from the extinguisher container.

AGENT CONTAINERS

Care must be taken in the replacement of cartridge and discharge valves. Most new extinguisher containers are supplied with their cartridge and discharge valve disassembled. Before installation on the aircraft, the cartridge must be assembled properly in the discharge valve and the valve connected to the container, usually by means of a swivel nut that tightens against a packing ring gasket. (*Figure 8-29*)

If a cartridge is removed from a discharge valve for any reason, it should not be used in another discharge valve assembly, since the distance the contact point protrudes may vary with each unit. Thus, continuity might not exist if a used plug that had been indented with a long contact point were installed in a discharge valve with a shorter contact point.

NOTE: The preceding material in this chapter has been largely of a general nature dealing with the principles involved and general procedures to be followed. When

actually performing maintenance, always refer to the applicable maintenance manuals and other related publications pertaining to a particular aircraft.

PORTABLE FIRE EXTINGUISHERS

There must be at least one hand held, portable fire extinguisher for use in the pilot compartment that is located within easy access of the pilot while seated. There must be at least one hand held fire extinguisher located conveniently in the passenger compartment of each airplane accommodating more than 6 and less than 30 passengers. Each extinguisher for use in a personnel compartment must be designed to minimize the hazard of toxic gas concentrations. The number of portable, hand held fire extinguishers for transport aircraft is shown in *Figure 8-30*.

PORTABLE EXTINGUISHER TYPES

All materials used in the cockpit and cabin must conform to strict standards to prevent fire. In case of a fire, several types of portable fire extinguishers are available to fight the fire. The most common types are Halon 1211 and water.

Portable fire extinguishers are used to extinguish fires in the passenger cabin and on the flight deck. *Figure 8-31* shows a Halon fire extinguisher used in a general aviation aircraft. The Halon extinguishers are used on electrical and flammable liquid fires. Some transport aircraft also use water fire extinguisher for use on nonelectrical fires. The following portable, hand-held fire extinguishers are unsuitable as cabin or cockpit equipment:

- CO₂
- Dry chemicals (due to the potential for corrosion damage to electronic equipment, the possibility of visual obscuration if the agent were discharged into the flight deck area, and the cleanup problems after their use)
- Specialized dry powder, however, it is suitable for use in ground operations.

Instructions for use of portable fire extinguishers are located on the container. Usually, this involves removal of a safetying device (pin or break away wire), direction of the extinguisher at the fire and squeezing a trigger handle. Other information on the label of a portable fire extinguisher includes the container approval number, weight and date of last service.

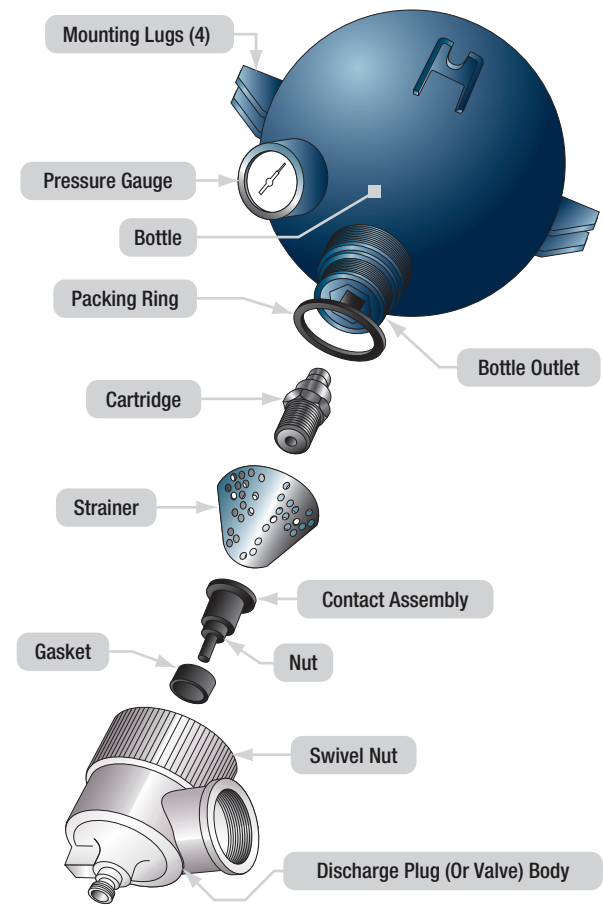


Figure 8-29. Components of fire extinguisher container.

Passenger Capacity	No. of Extinguishers
7 through 30	1
31 through 60	2
61 through 200	3
201 through 300	4
301 through 400	5
401 through 500	6
501 through 600	7
601 through 700	8

Figure 8-30. Hand held fire extinguisher requirement for transport aircraft.



Figure 8-31. Portable fire extinguisher.

Question: 8-1

A complete fire protection system on modern aircraft, and on many older aircraft, includes a fire _____ system and a fire _____ system.

Question: 8-6

The test circuit for a continuous loop fire detection element is located in the _____.

Question: 8-2

Name 8 means of fire detection used on large turbine-engine and high performance aircraft.

Question: 8-7

Most modern engine installations use a _____ to mount fire detection sensing elements for better maintainability, as well as increased reliability.

Question: 8-3

A _____ depends on the rate of temperature rise and does not give a warning when an engine slowly overheats or a short circuit develops.

Question: 8-8

A pneumatic fire detector system has two sensing functions. Name them.

Question: 8-4

A _____ permits more complete coverage of a fire hazard area than any of the spot type temperature detectors.

Question: 8-9

Storage of fire extinguishing agent is either in fixed containers or _____ containers or both.

Question: 8-5

In a Kidde continuous loop fire detection system, as the temperature of the detector loop core increases, electrical resistance to the ground _____.

Question: 8-10

The best fire extinguishing agent for a Class D fire is _____.

ANSWERS

Answer: 8-1

detection, extinguishing.

Answer: 8-6

control unit.

Answer: 8-2

1. Rate-of-temperature-rise detectors.
2. Radiation sensing detectors.
3. Smoke detectors.
4. Overheat detectors.
5. Carbon monoxide detectors.
6. Combustible mixture detectors.
7. Optical detectors.
8. Observation of crew or passengers.

Answer: 8-7

support tube.

Answer: 8-3

thermocouple.

Answer: 8-8

Overall average temperature threshold.

Localized discrete temperature increase.

Answer: 8-4

continuous loop detector or sensing system.

Answer: 8-9

portable.

Answer: 8-5

decreases.

Answer: 8-10

dry powder chemical.

Question: 8-11

The most commonly used Halon fire extinguishing agents are Halon _____ and Halon _____.

Question: 8-12

Most fixed fire extinguishing agent containers on high performance and transport aircraft are the _____ type.

Question: 8-13

Standard release type discharge valves use a slug driven by _____ to rupture a segmented closure disc.

Question: 8-14

A normal discharge of fire extinguishing agent will cause a _____ disk to be ejected from the skin of the aircraft fuselage.

Question: 8-15

Cargo compartments typically combine a _____ detection system with a fixed Halon extinguishing system.

Question: 8-16

In addition to a dump extinguisher, some cargo compartments have a _____ extinguishing system that discharge agent at a controlled rate.

Question: 8-17

_____ the bottle is often the only way to determine if the fire extinguishing agent bottle is empty or full.

Question: 8-18

Portable fire extinguishers are used to extinguish fires in the passenger caving and _____.

Question: 8-19

Information on the label of a portable fire extinguisher includes the container approval number, weight and _____.

Question: 8-20

When an engine fire switch is activated, the engine stops because the fuel control shuts off, the engine is isolated from the aircraft systems, and the _____ is activated.

ANSWERS

Answer: 8-11
1301.
1211.

Answer: 8-16
metered.

Answer: 8-12
high-rate-of discharge (HRD).

Answer: 8-17
Weighing.

Answer: 8-13
explosive energy.

Answer: 8-18
on the flight deck.

Answer: 8-14
yellow.

Answer: 8-19
date of last service.

Answer: 8-15
smoke type.

Answer: 8-20
fire extinguishing system.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 09

FLIGHT CONTROLS (ATA 27)

Knowledge Requirements

11.9 - *Flight Controls (ATA 27)*

Primary controls: aileron, elevator, rudder, spoiler; Trim control;
Active load control;
High lift devices;
Lift dump, speed brakes;
System operation: manual, hydraulic, pneumatic, electrical, fly-by-wire; Artificial feel, Yaw damper,
Mach trim, rudder limiter, gust lock systems;
Balancing and rigging;
Stall protection/warning system.

3

FLIGHT CONTROLS
(ATA 27)

11.9 - FLIGHT CONTROLS

PRIMARY FLIGHT CONTROLS

Since the dawn of heavier-than-air flight and the discovery of the three axis flight control network, airplanes continue to employ the three primary controls: elevator, aileron, and rudder. Before the discovery of the three axis control system, gliders and airplanes were very difficult to control during flight. It should be noted that the same control inputs used by the pilot to fly small airplanes are used to control large aircraft.

The primary flight controls provide the aerodynamic force necessary to make the aircraft follow a desired flight path. (*Figure 9-1*) The flight control surfaces are normally hinged or movable airfoils designed to change the attitude of the aircraft by changing the airflow over the aircraft's surfaces during flight. These surfaces are used for controlling the aircraft about its three axes. Typically, the ailerons and elevators are operated from the flight deck by means of a control stick, a control wheel, or yoke assembly and on some of the newer design aircraft, a joystick. Lateral control is the climb and dive movement or pitch of an aircraft that is controlled by the elevator.

To cause the airplane to ascend from a straight and level attitude, the pilot pulls back on the control yoke or stick. Pushing the control forward lowers the nose of the aircraft for making descents.

Longitudinal control is the banking movement or roll of an aircraft that is controlled by the ailerons. To roll the airplane around the longitudinal axis, the pilot rotates the control wheel or moves the stick to the left or right, as desired. When the control is moved to the left, the left aileron rises above the wing and the right aileron descends below the wing. This causes the left wing to drop and the right wing to ascend resulting in a left bank. Some aircraft may use multiple ailerons so that each wing includes an inboard and outboard aileron. In such instances, the control network may lock out the outboard ailerons during high-speed flight. The inboard ailerons may be designed to slightly droop when the trailing edge flaps are extended. In addition to ailerons, spoilers may also be incorporated into aileron system. Each aircraft may have specific features contained in the flight control system to enhance the operation of the airplane.

Directional control around the vertical axis is the left and right movement or yaw of an aircraft that is controlled by the rudder. Some aircraft may employ lower and upper rudder control surfaces where both rudders are deflected for control while flying at lower airspeeds and a single rudder is used for high-speed flight. Foot pedals normally control the position of the rudder. Stepping on the right rudder pedal deflects the rudder to make a right turn. Stepping on the left pedal causes the aircraft to turn left. Most often when making turns during flight, the application of the rudder is made in combination with the aileron control. When the proper proportion of rudder and ailerons are inputted into the control system for the purpose of banking through a turn, the airplane is in a coordinated turn.

SECONDARY FLIGHT CONTROLS

Large airplanes will often employ a series of secondary flight controls to augment the performance of the aircraft during takeoff and landing and to supplement the controllability of the airplane throughout the various flight parameters. Secondary flight controls include: spoilers, leading edge flaps, leading edge slats, trailing edge flaps, and speed brakes. The secondary flight controls may further be used for aerodynamic braking once the airplane has landed.

A common secondary flight control involves the use of spoilers to assist in controlling the bank of the airplane. The flight spoilers rise on the side of the airplane where the aileron is deflected up. They remained down on the wing where the aileron is deflected below the surface of the wing. (*Figure 9-2*)

TRIM CONTROLS

Trim systems are added to flight control members to assist the crew in controlling the aircraft. Trim systems may also be used to control the aircraft, to a degree, during emergencies when the primary flight control system(s) fail or develop a fault. Pilots learn early in their training how to trim an aircraft to relieve them of having to maintain physical pressure on the controls. Included in the trim controls are the trim tabs, servo tabs, balance tabs, and spring tabs. Trim tabs are small airfoils recessed into the trailing edges of the primary control surfaces. Trim tabs can be used to correct any tendency of the aircraft to move toward an undesirable

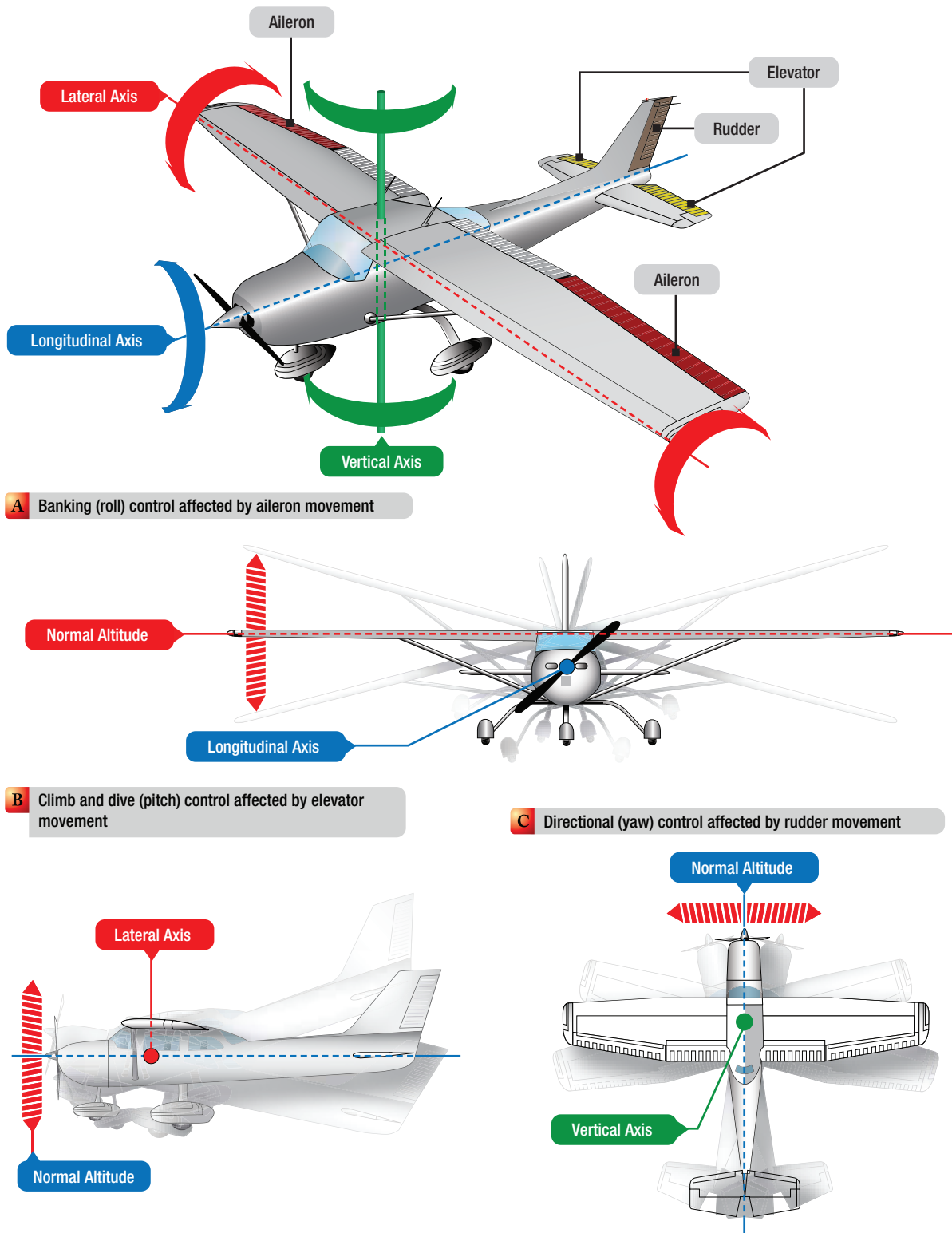


Figure 9-1. Aircraft 3-axis view with associated controls.

flight attitude. Their purpose is to enable the pilot to trim out any unbalanced condition that may exist during flight, without exerting any pressure on the primary flight controls.

Servo tabs, sometimes referred to as flight tabs, are used primarily on the large main control surfaces. They aid in moving the main control surface and holding it in the desired position. Only the servo tab moves in response to control movements inputted by the pilots.



Figure 9-2. Flight Spoilers. Note how the flight spoilers, outboard of the engine, move up with the aileron (shown near the tip of the wing).

Balance tabs are designed to move in the opposite direction of the primary flight control. Thus, aerodynamic forces acting on the tab assist in moving the primary control surface. (*Figure 9-3*)

Spring tabs are similar in appearance to trim tabs, but serve an entirely different function. Spring tabs are used for the same purpose as hydraulic actuators—to aid the pilot in moving the primary control surface. In the *Figure 9-4* note how each trim tab is hinged to its parent primary control surface, but is operated by an independent control.

ACTIVE LOAD CONTROL

As the aircraft maneuvers through the atmosphere, numerous dynamic loads are generated. To counter these loads to a degree, engineers have developed active load control techniques. Active load control



Figure 9-3. Balance Tab. Note that the balance tab moves in the opposite direction of the control surface. In this illustration the elevator is up while the balance tab is down.

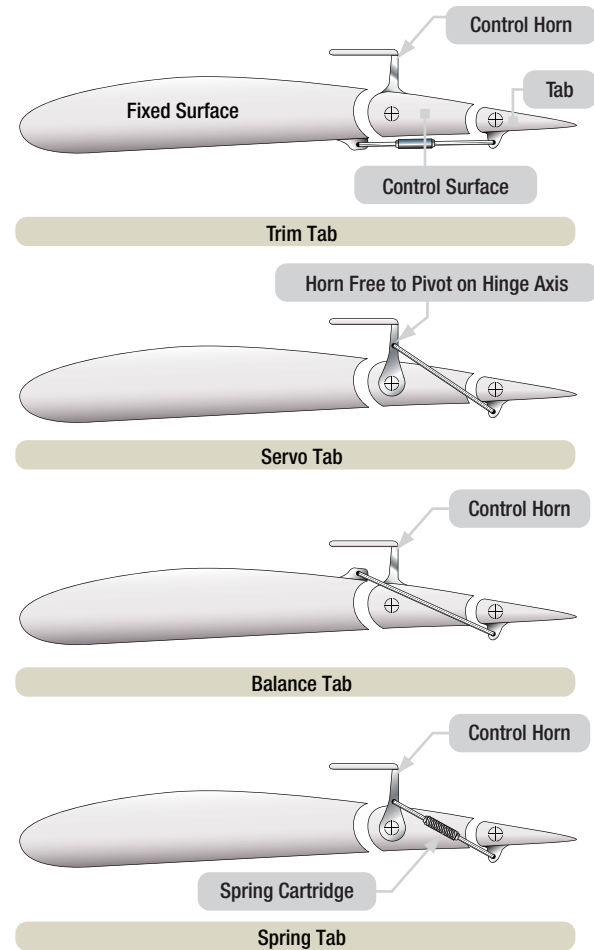


Figure 9-4. Types of trim tabs.

is a system whereby the stresses placed on the wings are redistributed to reduce the focus of the loads encountered during maneuvers involving rolls. During a roll, stresses on the wing increase as the g-load increases with bank angle. A standard approach to withstanding the additional loads is to make the wing stronger. Such reinforcement techniques add weight to the structure and empty weight of the airplane. Incorporating a system whereby the loads may be alleviated saves weight on the structural design of the aircraft. One example of an active load control system is the roll maneuver load alleviation (RMLA) approach. The RMLA network varies control surfaces deflections and extensions based roll command given by the pilot. Using the flexibility of the wing and torsional load placed on the wing structure, the RMLA system moderates or eliminates the deflection of the outboard aileron during rolls and relies on the deflection of the inboard ailerons and movement of leading edge flight control surfaces. A variety of scenarios must be met depending on the speed of the aircraft and aggressiveness of the control input.

In the realm of fighter aircraft, active load control systems may be implemented to increase the roll rate of the aircraft at high speeds. Such increases in the maneuverability of the aircraft present distinct advantages. In the future, the aviation community will likely encounter a host of flight control systems available through the incorporation of advanced construction techniques and electronics.

HIGH LIFT DEVICES

Included in the high lift devices group of flight control surfaces are the wing trailing edge flaps, slats, leading edge flaps, and slots. They may be used independently or in combination to improve the performance of the aircraft. (*Figure 9-5 and Figure 9-6*)

The trailing edge airfoils (flaps) increase the wing surface area when extended, thereby increasing lift on takeoff, and decreasing the speed of the airplane during landing. These airfoils are retractable and fair into the wing contour. Other flaps are simply portions of the lower skin that extend into the airstream, thereby slowing the aircraft. Leading edge flaps are airfoils extended from and retracted into the leading edge of the wing. Some installations create a slot (an opening between the extended airfoil and the wing leading edge). The flap (termed slat by some manufacturers) and slot create additional lift at the lower speeds used during takeoff and landing. (*Figure 9-7 and Figure 9-8*)

Other installations have permanent slots built in the leading edge of the wing. At cruising speeds, the trailing edge and leading edge flaps (slats) are retracted into the wing proper. Slats are movable control surfaces attached to the leading edges of the wings. When the slat is closed, it forms the leading edge of the wing. When in the open position (extended forward), a slot is created between the slat and the wing leading edge. At low airspeeds, this increases lift and improves handling characteristics, allowing the aircraft to be controlled at airspeeds below the normal landing speed.

(*Figure 9-9 and Figure 9-10*)

LIFT DUMP AND SPEED BRAKES

Lift decreasing devices are the speed brakes and spoilers. In some installations, there are two types of spoilers. Ground spoilers are extended only after the aircraft is on the ground, thereby assisting in the braking action. Flight spoilers assist in lateral control by being

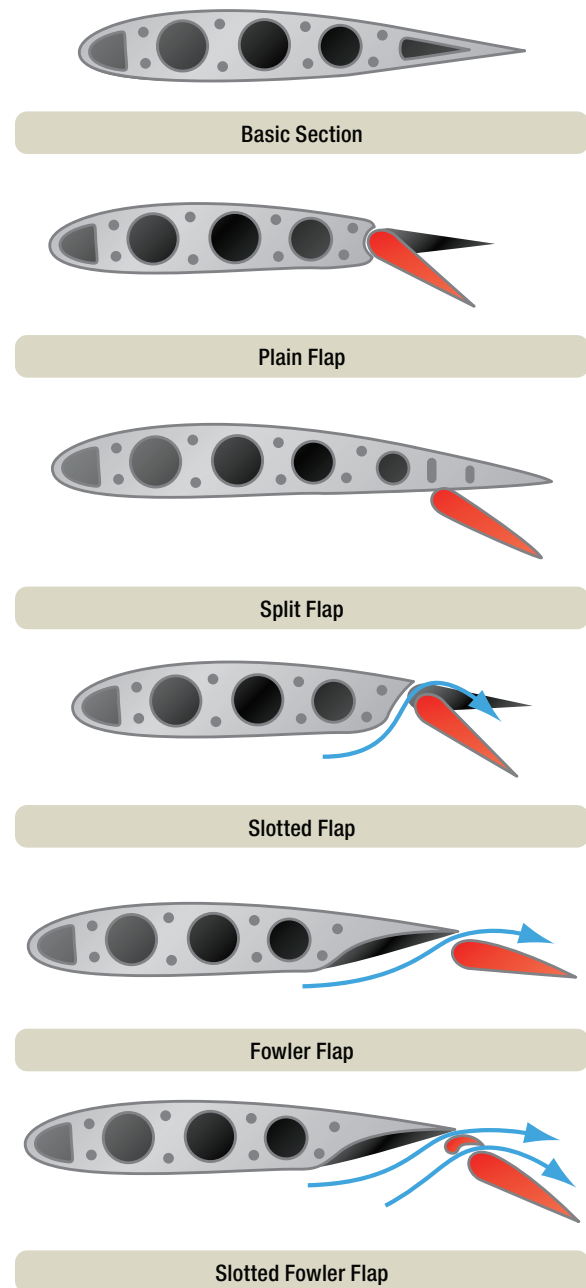


Figure 9-5. Types of flaps.

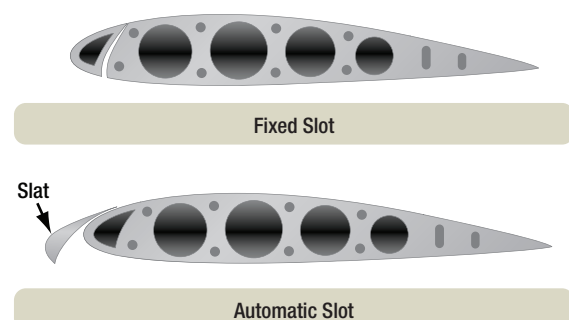


Figure 9-6. Slots.



Figure 9-7. Outboard Trailing Edge Flaps and Slats Extended. Note how the extension of the flaps and slats increase the surface area of the wing in addition to altering the shape of the wing to enhance low-speed performance and lift production.



Figure 9-9. Speed Brakes for Aerodynamic Braking. The flight spoilers, shown as the two panels in the center, join the ground spoilers to maximize aerodynamic braking after landing. Note: the aileron near wing tip is not above the wing with flight spoilers acting as speed brakes.



Figure 9-8. Leading edge and trailing edge inboard flaps extended. In addition to the outboard trailing edge flaps and slats shown in the previous illustration, leading edge flaps and inboard trailing edge flaps, seen behind the landing gear, are extended to further enhance low-speed performance.



Figure 9-10. Speed brake using the tail cone.

extended whenever the aileron on the same wing is deflected upward from neutral (*see Figure 9-2*). When actuated as speed brakes, the spoiler panels on both wings raise up. Inflight spoilers may also be located along the sides, underneath the fuselage, or back at the tail. In some aircraft designs, the wing panel on the up aileron side rises more than the wing panel on the down aileron side. This provides speed brake operation and lateral control simultaneously.

CONTROL SYSTEM OPERATION

MECHANICAL CONTROL

This is the basic type of system that was used to control early aircraft and is currently used in smaller aircraft where aerodynamic forces acting on the controls are not excessive. The controls are mechanical and manually operated by the pilot.

The mechanical system of controlling an aircraft can include cables, push-pull tubes, bell cranks, levers, jackscrews, cable drums, and torque tubes. The cable system is the most widely used because deflections of the structure to which it is attached

do not affect its operation. Some aircraft incorporate control systems that are a combination of mechanical control mechanisms. These systems incorporate cable assemblies, cable guides, linkages, bell cranks, push-pull tubes, torque tubes, adjustable stops, and control surface snubbers or mechanical locking devices. Surface locking devices, usually referred to as gust locks, limit external wind forces from damaging the control system while the aircraft is parked or tied down.

CONTROL CABLES

Control cables used in aviation are typically 7×7 and 7×19 flexible steel wires. Cables are very strong when placed under a tensile or pulling load. Flexible cables do not have strength when pushed. Consequently, when cables are used for flight controls, they often employ multiple cables so that one cable is under tension when the control input is made in one direction and the other cable is under tension when the control input is made in the opposite direction. Control cables may run the entire length from the control mechanism manipulated by the crew to the control quadrant, cable drum, torque tube, bell crank, or lever that connects to the control surface. Other cables may run from the pilot's control mechanism to hydraulic valves or other devices that ultimately deflect the control surfaces. (*Figure 9-11*)

Most manufacturers of large aircraft will include some means whereby cables may be identified through labeling. Through the use of this naming system, technicians are able to identify the function of a cable or identify the unit in need of service operations. New cables may be constructed using the data provided by the manufacturer. The data will have the size and type of cable, the types of fittings to attach to the ends of the cable, the specified length of the cable, and load testing requirements. It may be necessary to prestretch new cables as part of the installation and rigging process.

Throughout the length of the cable, pulleys may be used for changing the direction of the cable action or for cable support. Fairleads are also used to guide control cables along their length. Fairleads are generally made from plastic or other material that contacts the cable as it moves back and forth. These blocks of material are subject to wear over time. Cable guides are also used to protect cables from damage.

Cables that extend from pressurized portions of the aircraft to unpressurized areas use seals to prevent loss of cabin pressure. Significant air leaks at such locations may affect the operation of the pressurization system.

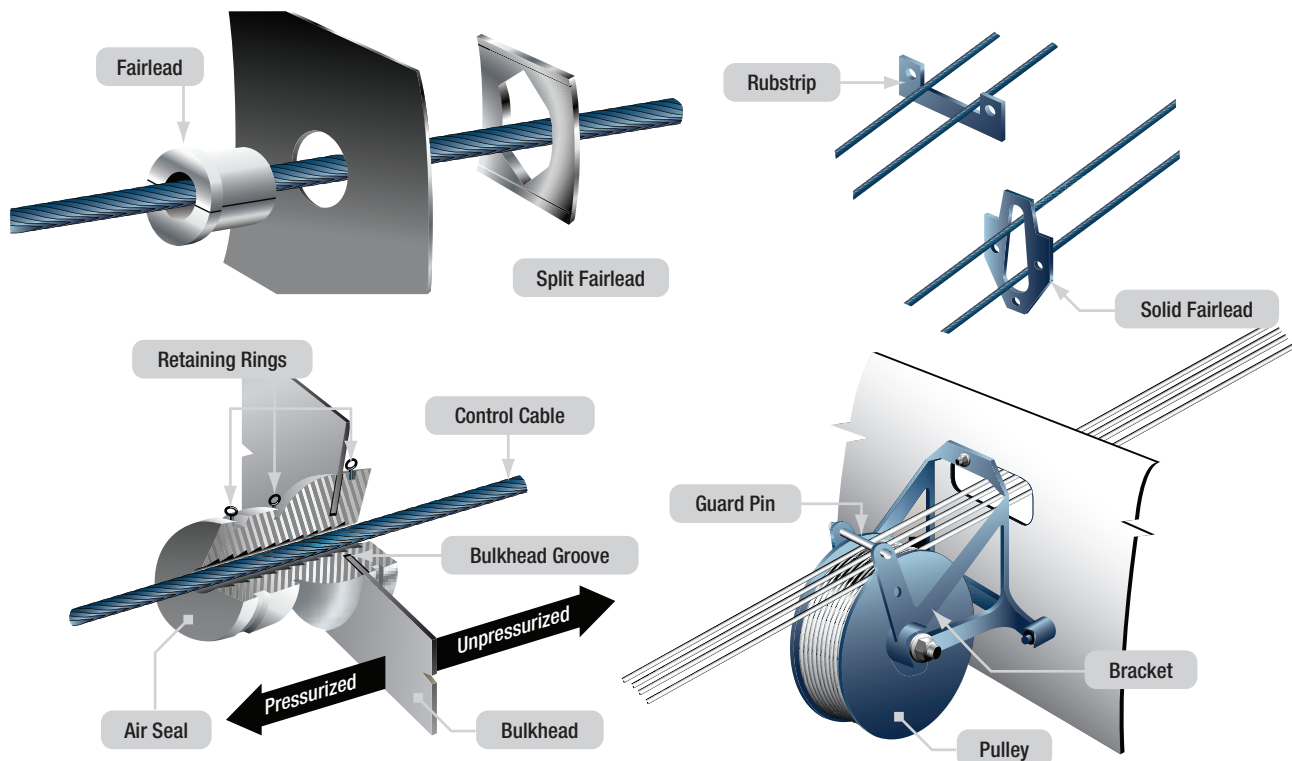


Figure 9-11. Fairleads, rubstrips, cable seals, and pulleys used to guide control cables.

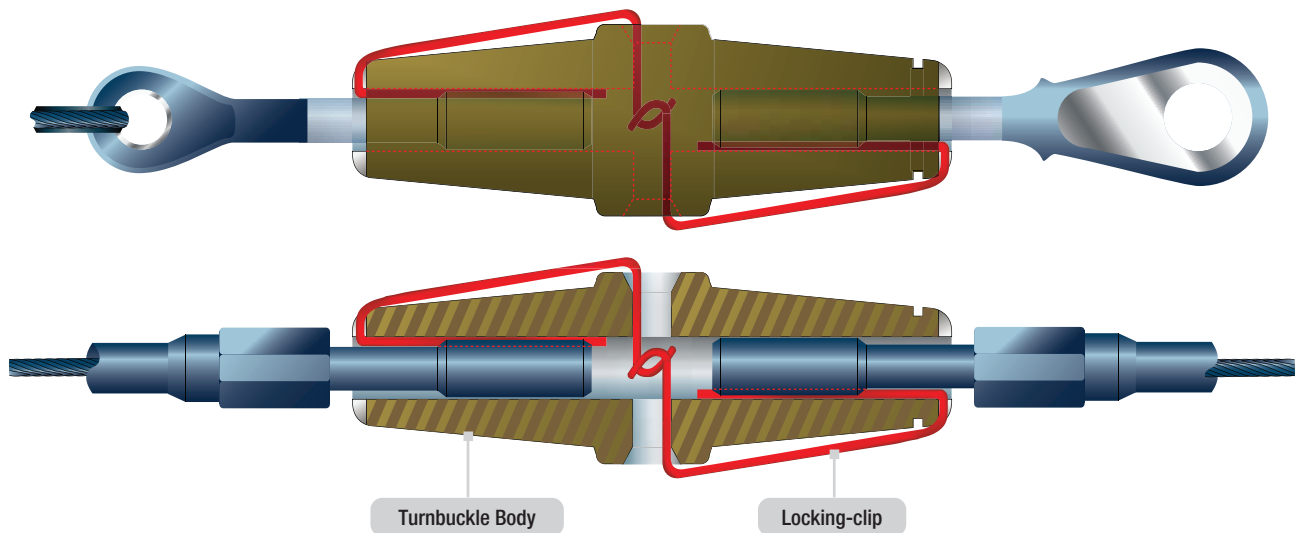
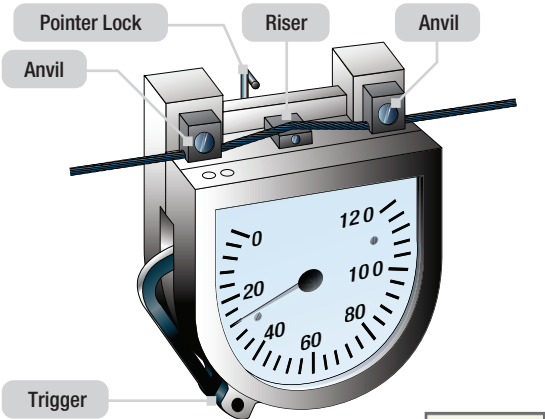


Figure 9-12. Turnbuckle with Safety Clips Installed Setting the proper tension on the cables is critical to the rigging process. Improper cable tension may cause loss of control travel or damage to components. Tensiometers are tools used to measure the tension placed on control cables.

Turnbuckles are normally included in the cable system for setting cable tension and serving as disconnect points. Turnbuckles are threaded devices that have an end with right-handed threads and the opposite end with left-handed threads. The cable terminals are threaded into the barrel of the turnbuckle. Turnbuckles are safetied using lockwire. Some turnbuckle and terminal ends have the option of being safetied with special clips. To allow for installation of the safety clips, a groove is cut into the threads of the turnbuckle and threaded terminal ends. The cut grooves must be aligned before the clip may be installed. (Figure 9-12)

Setting the proper tension on the cables is critical to the rigging process. Improper cable tension may cause loss of control travel or damage to components. Tensiometers are tools used to measure the tension placed on control cables. (Figure 9-13)

Because airplanes stretch and contract with changes in temperatures, some airplanes use cable tension regulators to maintain proper cable tension throughout the range of conditions. Such devices are needed as the expansion and contraction of airplane structures made of aluminum are greater than that of the steel cables.



No. 1			Riser	No. 2		No. 3	
Diameter			Tension (lb)	5/32	3/16	7/32	1/4
1/16	3/32	1/8					
12	16	21	30	12	20		
19	23	29	40	17	26		
25	30	36	50	22	32		
31	36	43	60	26	37		
36	42	50	70	30	42		
41	48	57	80	34	47		
46	54	63	90	38	52		
51	60	69	100	42	56		
			110	46	60		
			120	50	64		

Figure 9-13. Cable Tensiometer and conversion chart.

PUSH-PULL TUBES

Where cables only have strength when they are placed under tension, or pulled, push-pull rods are able to transmit force in either direction. Push-pull rods may be solid or hollow. The ends attached to the push-pull rods may be fixed or adjustable. Technicians must ensure that

adjustable ends have adequate thread engagement with the push-pull rod. Inadequate thread engagement may lead to part failure and loss of control. Witness holes are used to verify sufficient thread engagement. (Figure 9-14 and Figure 9-15)

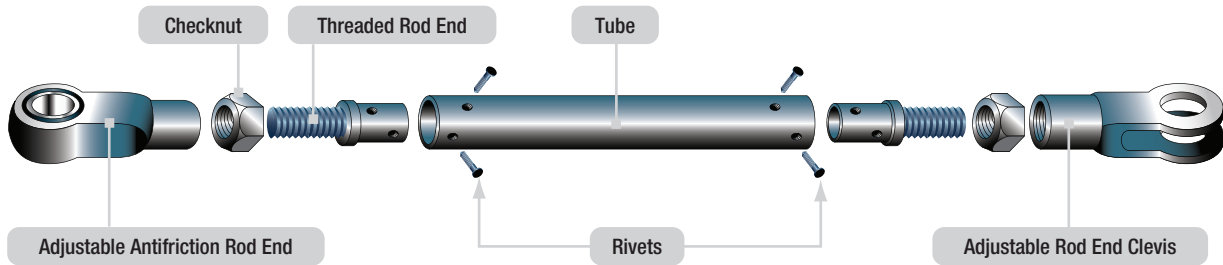


Figure 9-14. Exploded view of push-pull rod.



Figure 9-15. Witness hole in push-pull tube. When the terminal end is adequately threaded into the tube, the threads will block the witness hole.

BELLCRANKS AND LEVERS

Bellcranks are constructed so that a series of levers are able to receive an input signal and deliver an output. The output from a lever or bellcrank may amplify the input or vice-versa. Frequently, bellcranks change the direction of movement. The input signal may come from a lateral direction and the output motion made in a longitudinal direction and vice-versa. (Figure 9-16 and Figure 9-17)

JACKSCREWS

Jackscrews are commonly employed for moving surfaces that experience extreme aerodynamic loads, such as horizontal stabilizers and flaps. Jackscrews are threaded units that convert rotary motion into linear travel. The threads of jackscrews are quite coarse. Jackscrews are ideal for trim applications of large control surfaces.

TORQUE TUBES

Torque tubes are used in many areas of the flight control system. Torque tubes apply torsional, or rotating, motion to a member of the control system. Often torque tubes receive their input from control cables or push-pull rods. (Figure 9-18)

HYDROMECHANICAL CONTROL

As the size, complexity, and speed of aircraft increased, actuation of controls in flight became more difficult to perform strictly using physical strength. It soon became apparent that the pilot needed assistance to overcome the aerodynamic forces encountered by the

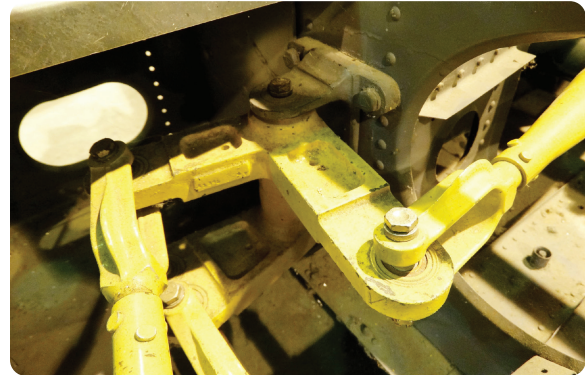


Figure 9-16. Bell crank with push-pull tubes.

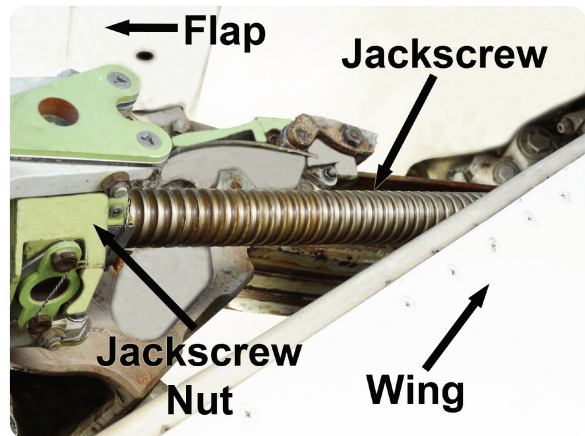


Figure 9-17. Jackscrew mechanism driving trailing edge flap.

control surfaces in order to control the aircraft. Spring tabs, which were operated by the conventional control system, were moved so that the airflow over them actually moved the primary control surface. This was sufficient for the aircraft operating in the lowest of the high speed ranges (250–300 mph). For higher speeds, a power-assisted (hydraulic) control system was designed and implemented.

Conventional cable or push-pull tube systems link the flight deck controls with the hydraulic system. With the system activated, the pilot's movement of a control

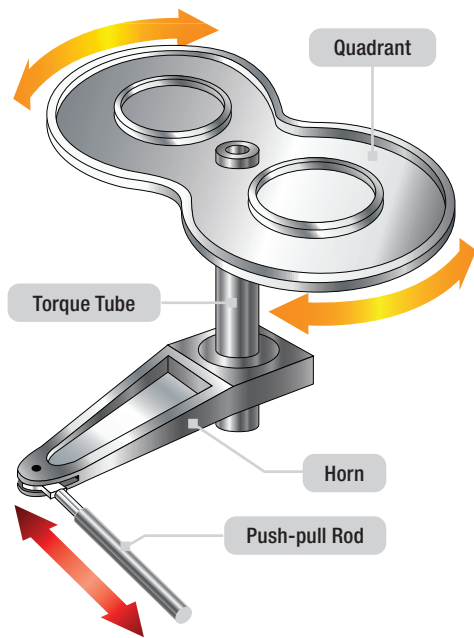


Figure 9-18. Torque tube transmitting input into rotary output.

causes the mechanical link to open and close servo valves, thereby directing hydraulic fluid to and from actuators, which convert hydraulic pressure into control surface movements.

Because of the mechanical advantage of the hydromechanical flight control system, the pilot cannot feel the aerodynamic forces acting on the control surfaces, and there is a risk of overstressing the structure of the aircraft. To overcome this problem, aircraft designers incorporated artificial feel systems into the design that provides increased resistance to the controls at higher speeds. In essence, the artificial feel simulates what the pilot would sense in terms of control system input if the aircraft did not have a hydraulic control network. Additionally, some aircraft with hydraulically powered control systems are fitted with a device called a stick shaker, which provides an artificial stall warning to the pilot.

Large aircraft often have the mechanical control network connected to the flight control as a back-up means of controlling the aircraft in the event of a hydraulic system failure or failure of the hydraulic control system. Often, aircraft are designed with multiple hydraulic actuation systems with the mechanical backup to ensure that the crew is able to control the aircraft.

ELECTRICAL AND ELECTRONIC CONTROLS

Modern aircraft have widely adopted electronics in their flight control systems. Normally multiple computers are incorporated in the control network with computers interfacing with autopilots, auto-landing, auto-speed braking, flaps, stall warning, ground proximity system, and etc. Regardless of the intricacy of computers involved in the control of the aircraft, their main function is to translate the control inputs made by the crew into actual control surface deflections. Electric trim is often found to control the position of the horizontal stabilizer. To make the trimming operation convenient, the switches to operate the trim are located in the control yoke as shown in **Figure 9-19**. Pilots must activate both switches simultaneously to engage the trim motor(s). A mechanical means of elevator trim is also provided on most aircraft. This mechanism is commonly found on the pedestal and drives the trim transmission using flexible cables. On many airplanes, the mechanical trim system moves when the horizontal stabilizer is trimmed via the pilot controlled electrical switches or when the autopilot trims the stabilizer.

Because the horizontal trim system is able to pitch the airplane nose up or down in a commanding fashion, some aircraft are equipped with a horizontal stabilizer trim brake system. This mechanism arrests the motion of the stabilizer during trimming operations when the movement of the elevator control inputted by the pilot opposes the direction of trim. Electrical controls may further serve as a back-up system. In **Figure 9-19**, the hydraulic motor normally drives the flap transmission. In the event of a complete hydraulic failure or fluid



Figure 9-19. Trim Switches for the Horizontal Stabilizer. Both switches must be activated simultaneously to engage the trim motor. Upward movement of the switches commands a nose down reaction and downward movement of the switches provide a nose up change in flight attitude.

depletion, the crew may operate the flaps using an electric motor to power the flap transmission. The ability to extend flaps for landing enhances the safety of the operation. (*Figure 9-20 to Figure 9-23*)

PNEUMATIC

Figure 9-24 shows another technique for assisting in the movement of a control surface of a large aircraft. It is called balance panel. Not visible when approaching the aircraft, it is positioned in the linkage that hinges the control surface to the aircraft. Balance panels have been constructed typically of aluminum skin-covered frame assemblies or aluminum honeycomb structures. The trailing edge of the location where the flight control is mounted is sealed to allow controlled airflow in and out of the hinge area where the balance panel is located. In essence, two chambers are established.

The pressure differential generated by the deflection of the control surface allows the balance panel to assist in the movement of the flight control. When the control surface is moved from the neutral position, differential pressure builds up across the balance panel. This differential pressure acts on the balance panel in a direction that assists in the control surface movement. For slight control surface movements, deflecting the control tab at the trailing edge of the surface is undemanding enough to not require significant assistance from the balance panel. But, as greater deflection is commanded, the force resisting control tab and control surface movement becomes greater and assistance from the balance panel is needed. The seals and mounting geometry allow the differential pressure of airflow on the balance panel to increase as deflection of the control surface is increased.

FLY-BY-WIRE CONTROL

The fly-by-wire (FBW) control system employs electrical signals that transmit the pilot's actions from the flight deck through a computer to the various flight control actuators. The FBW system evolved as a way to reduce the system weight of the hydromechanical system, reduce maintenance costs, and improve reliability. Electronic FBW control systems can respond to changing aerodynamic conditions by adjusting flight control movements so that the aircraft response is consistent for all flight conditions. Additionally, the computers can be programmed to prevent undesirable and dangerous characteristics, such as stalling and spinning.

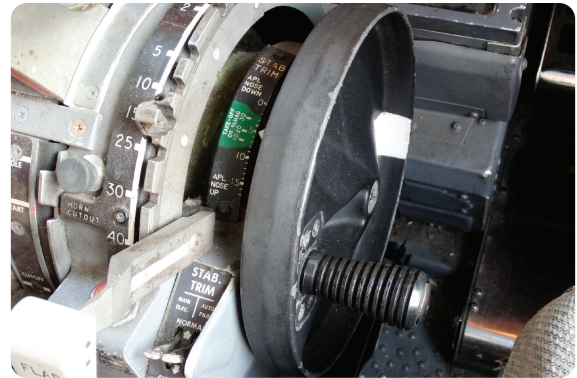


Figure 9-20. Mechanical Trim System and Trim Position Indicator for Horizontal Stabilizer. The protruding handle may be stowed within the trim wheel. The green band is used to signify the proper position of the horizontal stabilizer for takeoff. Trimming the horizontal stabilizer outside the green band provides an unsafe for takeoff warning scenario.

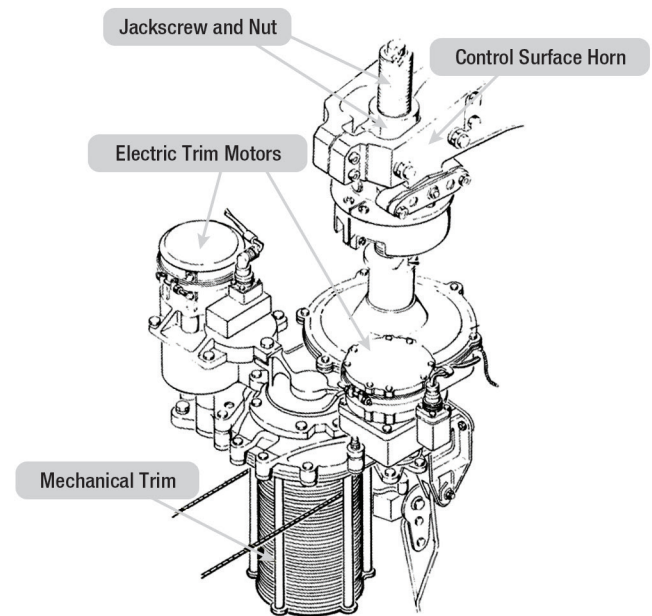


Figure 9-21. Electric and Mechanical Trim Systems for the Horizontal Stabilizer. The motor shown on the left is for manual trim operations from the switches on the control yokes, the motor on the right is used with the autopilot, and the cable drum is connected to the mechanical trim system on the pedestal.

Many of the later generation military high-performance aircraft are not aerodynamically stable. This characteristic is designed into the aircraft for increased maneuverability and responsive performance. Without the flight control computers reacting to the instability, the pilot would experience great difficulties controlling the aircraft.

These systems rely of the translation of control inputs into electrical signals and the conversion of the electrical signal into control surface movements. Without the



Figure 9-22. Horizontal stabilizer brake release knob.

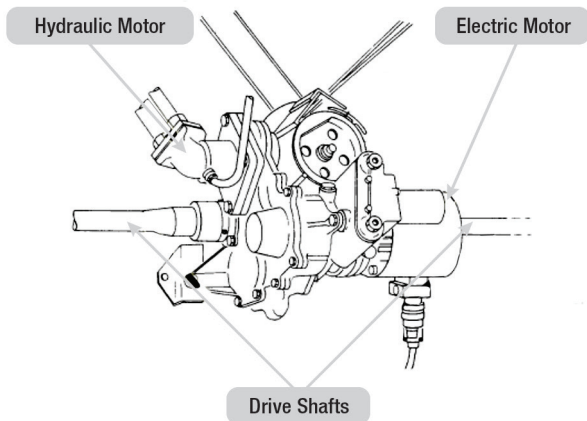


Figure 9-23. Flap Motor Drive. Flap system normally uses the hydraulic motor to extend and retract flaps. Electric motor serves as back-up system to ensure flaps are extended for landing.

inclusion of control cables and other mechanical components, transducers, rotary variable differential transformers (RVDTs), electronic display screens, sensors, artificial feel systems, power control units, power drive units, and other apparatuses are used for control operation.

With enhancements in flight control software, designers are able to use computers to send a host of signals to the various flight control members to reduce or eliminate flutter during flight. Traditional mechanical flight control mechanisms were unable to perform such feats.

The Airbus A-320 was the first commercial airliner to use FBW controls. Boeing used them in their 777 and newer design commercial aircraft. The Dassault Falcon 7X was the first business jet to use a FBW control system.

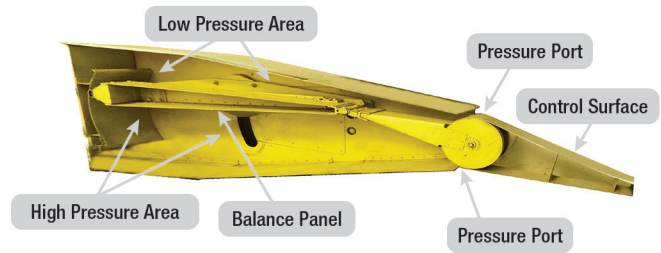


Figure 9-24. Balance Panel. In this illustration, the control surface is deflected down. The two pressure ports at the trailing edge of the wing have different pressures. The upper pressure port develops and transmits to the balance panel a low pressure while the lower pressure port develops a relatively higher pressure. The pressure differential acting on the balance panel assists in the movement of the flight control surface.

FLY-BY-OPTICS

Aircraft designers continue to enhance flight control systems. Where fly-by-wire systems are able to use computers to control the position of multiple flight control surfaces, fly-by-optics further improve the ability of the system to transfer data. Fly-by-optics networks are able to transfer data at higher speeds than wired systems. Fly-by-optics systems, also known as fly-by-light, are more immune to electrical interference that may affect fly-by-wire systems.

FLY-BY-WIRELESS

The next generation of flight controls include fly-by-wireless and fly-by-less wires systems. Similar to fly-by-wire systems, fly-by-wireless networks offer a reduction in weight of the aircraft by removing the extensive bundle of wires used in fly-by-wire aircraft. The weight savings further translate into a measure of efficiency. Another benefit of the fly-by-wireless system involves reduced maintenance. Through years of service, the fly-by-wire harnesses will develop maintenance issues with connectors, corrosion, broken wires, etc. Every connection becomes a potential point of failure. Removing the wires from the flight control network, or reducing the number of associated wires, saves maintenance costs over the life of the aircraft.

ARTIFICIAL FEEL

Aircraft that use purely mechanical flight control systems do not require artificial feel on the controls. The resistance transmitted through the control system provides the pilot with a natural feel regarding the magnitude of control input and associated stresses placed on the aircraft. (*Figure 9-25*)

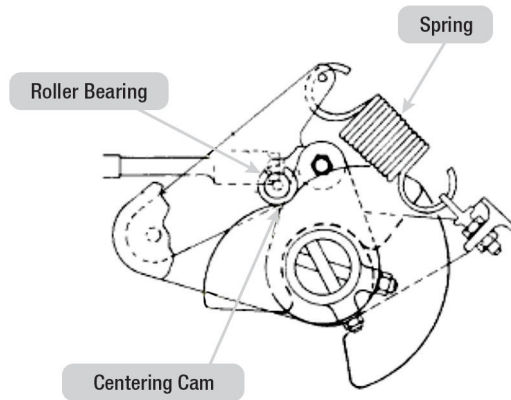


Figure 9-25. Artificial Feel Mechanism. As the pilot further deflects the control, the roller bearing rides up the side of the cam resulting in more spring opposition.

Aircraft that move control surfaces solely by hydromechanical and/or electromechanical means deprive the pilot of the feel of a mechanical control system. Consequently, as the load or resistance generated by the flight control surface as it is deflected into the airstream is not directly transmitted to the pilot. As a substitute, aircraft manufacturers have developed artificial feel systems to provide feedback regarding control input. Without artificial feel, pilots could generate high levels of loads on the aircraft structure without realizing it. Mechanisms used to produce artificial feel may be mechanical. A common approach is to use a spring-loaded roller that fits into the valley of a flattened v-shaped cam. As the control input is increased, the roller rides higher up the side of the cam, thereby increasing the spring resistance felt by the pilot.

YAW DAMPER

One common control system is the yaw damper used on many large aircraft. Typically associated with aircraft using swept wings that generate a motion referred to as a Dutch roll, the purpose of the yaw damper system is to counter the rolling tendency of the aircraft during flight. Yaw dampers work when the aircraft is controlled manually by the flight crew or during operations involving the autopilot. The yaw damper system provides inputs to the rudder in proportion to the yaw rate of the aircraft and in a direction that negates the oscillations that would otherwise take place during flight. Aside from increasing the stability of the aircraft, the yaw damper provides a smoother ride for the passengers.

MACH TRIM

Airfoils traveling at low subsonic speeds have a center of pressure acting on the wing that is approximately one quarter the distance of the chord, aft of the leading edge. The center of pressure does not move much until the aircraft begins traveling at high speeds. When the aircraft passes through the air at speeds around Mach 0.7 and above, the center of pressure begins to move aft on the wing. As aircraft approach the speed of sound, their form may further accelerate the air flowing over the wings and other portions of the aircraft. When the aircraft reaches its critical Mach number, shock waves may develop over the wing. The area in front of the shock waves develops high lift. This action continues to travel aft as the aircraft gains more speed. The rearward movement of the lift production causes the aircraft to experience Mach tuck resulting in a nose down flight attitude. To counter Mach tuck and keep the aircraft flying in a level attitude, Mach trim is incorporated in the control network. Mach, or the speed of sound, is not a constant value. The speed of sound varies largely with changes in temperature.

Another factor that enters the controllability of the airplane involves coffin corner. The operation of the airplane enters coffin corner when the stall speed of the aircraft flying at high altitudes for a given weight and load factor approaches the critical Mach number. Aircraft entering the coffin corner configuration may be very difficult to keep in stable flight. Any reduction of airspeed will cause the plane to stall and any increase of airspeed will generate a loss of lift due to entering critical Mach. Pilots strive to keep the airplane out of the portion of the flight envelope known as coffin corner.

Mach trim basically trims the nose of the aircraft up as Mach tuck begins to act on the aircraft. Most systems of Mach trim are automatic in that the flight crew does not have to manually change trim settings. The crew may notice changes in trim as the control network implements Mach trim input. To ensure the crew does not lose Mach trim during flight, airplanes will typically have redundant Mach trim systems.

RUDDER LIMITER

Airplanes that have a relatively low speed range (e.g., 200 knots) generally do not need flight control networks that limit control surface travel at higher speeds. The structure of such airplanes is capable of absorbing the

loads generated by large control surface deflections. But airplanes that are capable of traveling at high speeds (e.g., in excess of 350 knots) would require an extensive amount of structural reinforcement to handle the loads generated by large control deflections. Such addition to the structure results in extra weight. To combat the need for excess structure, many high speed aircraft resort to limiting control surface deflection during high speed operation. This is similar to operating an automobile. When traveling along a highway at high speeds, the driver does not apply large inputs to the steering wheel, but rather small inputs. The same automobile may need full steering deflection while traveling at low speeds as in the example of parking.

Some aircraft reduce the travel available to the rudder based on the speed of the aircraft. At low speeds the need for substantial rudder travel is required. At high speeds (e.g., above 250 knots) the effectiveness of the rudder is increased, thereby reducing the need for large deflections. For the same number of degrees of rudder deflection, the load placed on the structure increases with the speed of the aircraft. Consequently, aggressive rudder deflections at high speeds may exceed the structural limitation of the aircraft. To minimize the risk of exceeding structural limitations, aircraft may include rudder limiters that reduce rudder deflection at high speeds. In other words, full rudder deflection is only available at lower airspeeds. For example, an airplane may have 30° of rudder deflection in the left and right directions at low speeds, such as takeoff, landing, climb, etc., with full pedal travel. At cruise speeds the rudder limiter restricts the rudder deflection to 7° left and right with full pedal travel.

GUST LOCK SYSTEMS

Aircraft that use mechanical flight control systems will typically include a method for locking the controls when the aircraft is parked. Normally referred to as gust locks, these mechanism may either be separate from the control system or an integral part of the control. Separate gust locks may consist of a device that extends from a stationary part of the aircraft, such as the wing, and passes over and locks in place the flight control surface (e.g., the ailerons). Another technique is to lock the movement of the flight controls with pins and other devices. Rather than being on the exterior of the aircraft, such locking devices are installed in the flight compartment to keep the

controls from moving. By physically locking the flight controls in place, damage to the structure or control network is eliminated during times when the aircraft is parked and the wind acts to deflect the flight control surfaces. Gust locks will typically include a warning streamer with the following or similarly worded phrase: "REMOVE BEFORE FLIGHT."

Large aircraft that have hydraulic assist systems to move flight control surfaces often include gust dampers in their power control units. By using hydraulic fluid contained within the power control units that drive the flight control surfaces during flight, movement of the control surfaces by wind feeds a force into the hydraulic units. These mechanisms provide gust snubbing by forcing hydraulic fluid through special bypass valves and other devices. The end result is that the flight control surfaces are protected from wind gust damage.

BALANCING AND RIGGING

This section is presented for familiarization purposes only. Explicit instructions for the balancing of control surfaces are given in the manufacturer's maintenance and other technical publications for the specific aircraft and must be followed closely.

Any time repairs on a control surface add weight fore or aft of the hinge centerline; the control surface must be rebalanced or checked for proper balance. When an aircraft is repainted, the balance of the control surfaces must be checked. Any control surface that is out of balance is unstable and does not remain in a streamlined position during normal flight. For example, an aileron that is trailing edge heavy moves down when the wing deflects upward, and up when the wing deflects downward. Such a condition can cause unexpected and violent maneuvers of the aircraft. In extreme cases, fluttering and buffeting may develop to a degree that could cause the complete loss of the aircraft. Rebalancing a control surface could include both static and dynamic balancing.

STATIC BALANCE

Static balance is the tendency of an object to remain stationary when supported from its own CG. There are two ways in which a control surface may be out of static balance. They are called *underbalance* and *overbalance*.

When a control surface is mounted on a balance stand, a downward travel of the trailing edge below the horizontal position indicates underbalance. Some manufacturers indicate this condition with a plus (+) sign. An upward movement of the trailing edge, above the horizontal position indicates overbalance. This is designated by a minus (−) sign. These signs show the need for more or less weight in the correct area to achieve a balanced control surface, as shown in *Figure 9-26*.

A tail-heavy condition (static underbalance) causes undesirable flight performance and is not usually allowed. Better flight operations are gained by nose heavy static overbalance. Most manufacturers advocate the existence of nose-heavy control surfaces. The structural repair manual of large aircraft provides an extensive amount of data regarding repairs and balancing of control surfaces and tabs. There will be a section on repairs that do not require rebalancing. Another section will provide data on how to calculate control balance following a repair or repainting without removing the control surface from the aircraft. And there will be instructions for determining and correcting control surface balance with the surface removed and mounted on special tools. Many manufacturers produce balancing weights that are added or removed from the control surface. Often drawings providing details on producing corrective weights in the field are given in the structural repair manual.

DYNAMIC BALANCE

Dynamic balance is that condition in a rotating body wherein all rotating forces are balanced within themselves so that safe level of vibration is produced while the body is in motion. Dynamic balance as related to control surfaces is an effort to maintain balance when the control surface is submitted to movement on the aircraft in flight. It involves the placing of weights in the correct location along the span of the surfaces. The location of the weights is, in most cases, forward of the hinge centerline.

REBALANCING PROCEDURES

Repairs to a control surface or its tabs generally increase the weight aft of the hinge centerline, requiring static rebalancing of the control surface system, as well as the tabs. Control surfaces to be rebalanced should be removed from the aircraft and supported, from their own points, on a suitable stand, jig, or fixture.

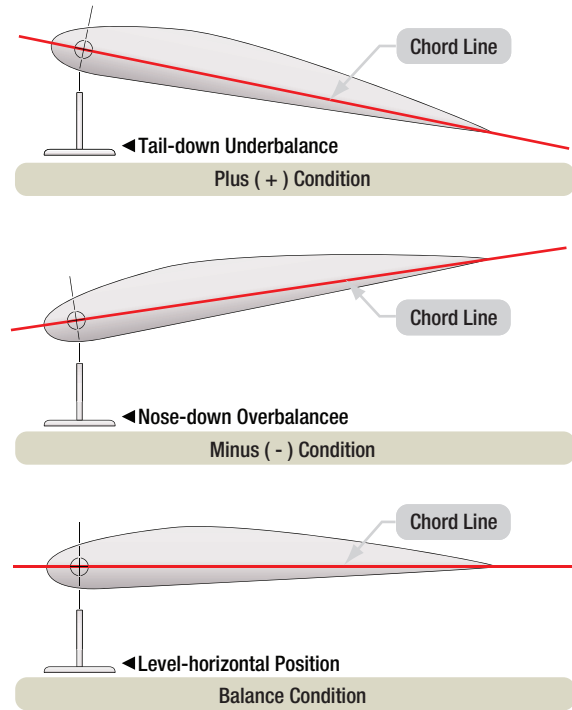


Figure 9-26. Over-, Under-, and Neutral-balanced control surfaces.

Trim tabs on the surface should be secured in the neutral position when the control surface is mounted on the stand. The stand must be level and be located in an area free of air currents. The control surface must be permitted to rotate freely about the hinge points without binding. Balance condition is determined by the behavior of the trailing edge when the surface is suspended from its hinge points. Any excessive friction would result in a false reaction as to the overbalance or underbalance of the surface. When installing the control surface in the stand or jig, a neutral position should be established with the chord line of the surface in a horizontal position. Use a bubble protractor, or suitable device, to determine the neutral position before continuing balancing procedures. (*Figure 9-27*)

Sometimes a visual check is all that is needed to determine whether the surface is balanced or unbalanced. Any trim tabs or other assemblies that are to remain on

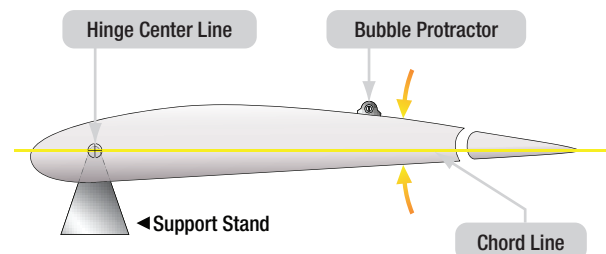


Figure 9-27. Control surface level on stand.

the surface during balancing procedures should be in place. If any assemblies or parts must be removed before balancing, they should be removed.

REBALANCING METHODS

Several methods of balancing (rebalancing) control surfaces are in use by the various manufacturers of aircraft. The most common are the calculation method, scale method, and the balance beam method.

The calculation method of balancing a control surface has one advantage over the other methods in that it can be performed without removing the surface from the aircraft. In using the calculation method, the weight of the material removed from the repair area and the weight of the materials used to accomplish the repair must be known. Subtract the weight removed from the weight added to get the resulting net gain in the amount of weight added to the surface. The distance from the hinge centerline to the center of the repair area is then measured. This distance must be determined to the nearest one hundredth of an inch or mm. Follow the instructions provided by the manufacturer as they will provide a series of formulas and other information regarding this process. (*Figure 9-28*)

The next step is to multiply the distance times the net weight of the repair. This results in an inch-pounds (in-lb), or similar unit, answer. If the result of the calculation is within specified tolerances, the control surface is considered balanced. If it is not within specified limits, consult the manufacturer's service manuals for the needed weights, material to use for weights, design for manufacturing corrective weights, and installation locations for the addition of weights. The scale method of balancing a control surface requires the use of a scale that is graduated in hundredths of a pound or similar metric unit. A support stand and balancing jigs for the surface are also required. The figure below illustrates a control surface mounted for rebalancing purposes. Use of the scale method requires the removal of the control surface from the aircraft. (*Figure 9-29*)

Some manufacturers use the balance beam method. This method often requires that a specialized tool be acquired or fabricated in the field. The manufacturer's maintenance manual typically provides specific instructions and dimensions to fabricate the tool.

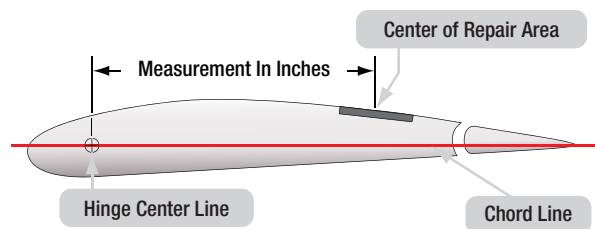


Figure 9-28. Calculating control balance.

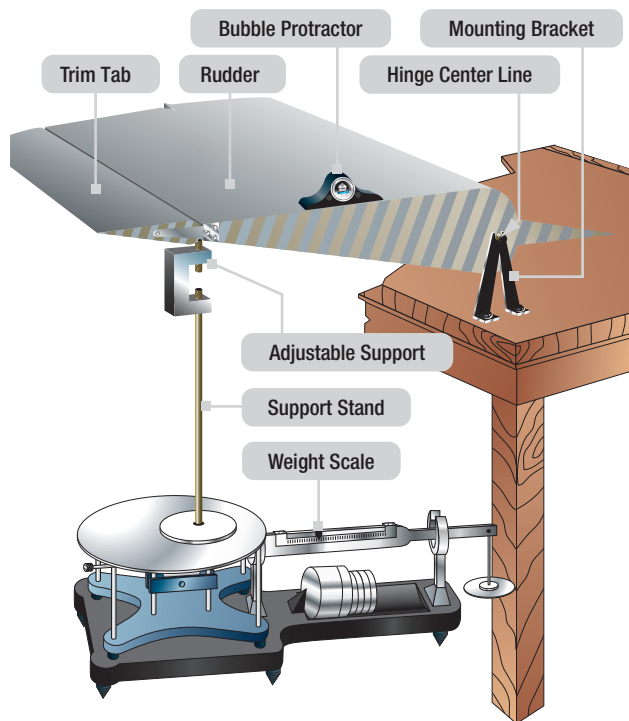


Figure 9-29. Scale measuring balance condition of control surface.

Once the control surface is placed on level supports, the weight required to balance the surface is established by moving the sliding weight on the beam. The maintenance manual indicates where the balance point should be. If the surface is found to be out of tolerance, the manual explains where to place weight to bring it into tolerance.

Aircraft manufacturers use different materials to balance control surfaces, the most common being lead or steel. Larger aircraft manufacturers may use depleted uranium because it has a heavier mass than lead. This allows the counterweights to be made smaller and still retain the same weight. Specific safety precautions must be observed when handling counterweights of depleted uranium because it is radioactive. The manufacturer's maintenance manual and service instructions must be followed and all precautions observed when handling the weights.

AIRCRAFT RIGGING

Aircraft rigging involves the adjustment and travel of movable flight control surfaces that are attached to major aircraft structures, such as wings and vertical and horizontal stabilizers. Ailerons, flaps, spoilers, and slats are attached to the wings, elevators are attached to the horizontal stabilizer, and the rudder is attached to the vertical stabilizer. Rigging involves setting cable tension, adjusting travel limits of flight controls and tabs, and setting travel stops. Newer generation aircraft incorporate various electronic controllers in the rigging process. Often the rigging process involves a number of special tools and pins needed to hold control wheels, bellcranks, and other control system devices in a certain position. Manufacturers typically will produce tools that are used to neutralize the position of the controls and measure travel with a scale. In some instances, technicians may construct similar tools or use substitutes to measure control surface and trim tab travel. (Figure 9-30) Rigging is not limited to the mechanical elements of the flight controls. Rigging of electronic equipment, if used, is also part of the process.

STALL PROTECTION AND WARNING SYSTEMS

Stall warning systems incorporated on modern day jetliners are far more advanced than those used on smaller general aviation airplanes. Stall warning systems typically involve multiple computers that monitor the configuration of the aircraft and flight data. Analyzing those bytes of information, the stall warning computers calculate when an airplane is nearing a stall condition. In such instances, a stick shaker that provides a violent shaking motion to the control yoke will give the crew a warning well in advance of a stall, usually. Some airplanes include a stick nudger or pusher that applies a nose down input to the elevator in an attempt to avoid the impending stall. The flight crew has the option of overcoming the input made by the stick nudger or pusher. The need for stick nudgers or pushers is due to the stall recovery characteristics of many larger airplanes. Where the stick nudger or pusher is designed to avoid a stall, the stick shaker is a stall warning mechanism. (Figure 9-31)

The stall speed of the aircraft is affected by a number of variables. In calculating the potential stall, the computers look at the position of the flaps, slats, speed brakes, airspeed, angle of attack, and other parameters. Failure to take the proper corrective action during a stall

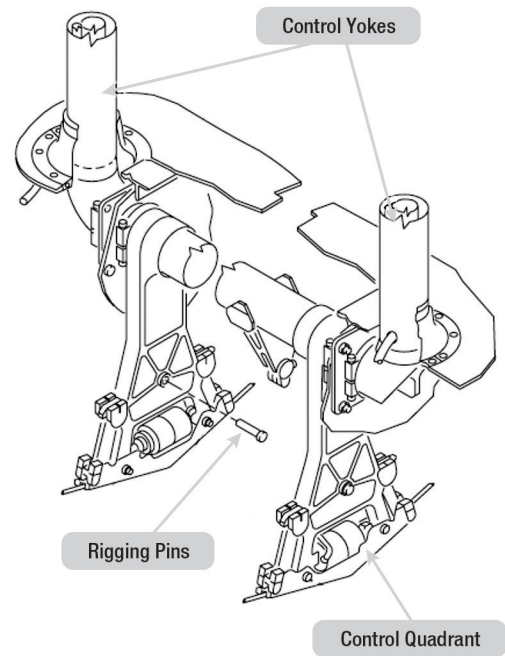


Figure 9-30. Rigging pin used to hold quadrant in correct position for control rigging.

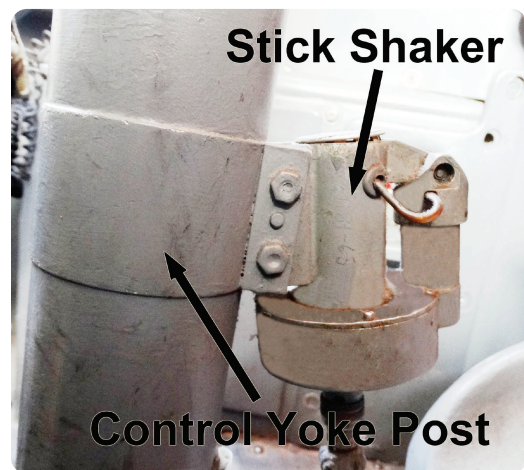


Figure 9-31. Stick shaker bolted to control yoke post. This motor violently shakes the control yoke.

may lead to serious consequences. Airplane stalls have claimed many lives over the history of flight. Angle of attack sensors commonly use a vane on the side of the fuselage that provides data regarding the angle that the aircraft is passing through the atmosphere. As the airplane changes its angle of attack, the vane reacts by rotating to a new position that is parallel to the airflow passing across it and sending a signal to the appropriate computer(s). As the data provided by the angle of attack sensor are critical to the safety of the aircraft during flight, the device is normally equipped with a heater to prevent the build-up of ice. (Figure 9-32)

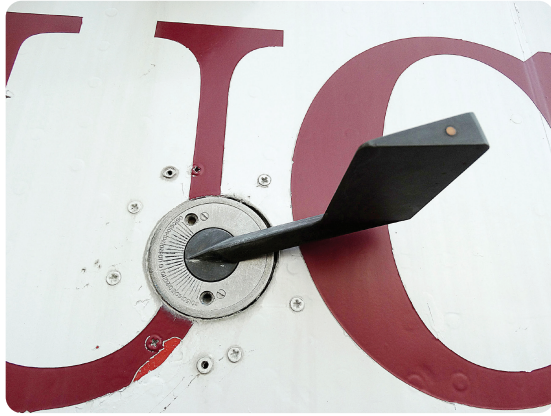


Figure 9-32. Angle of attack vane.

Unsafe for takeoff configuration warning is typically provided on large aircraft. This warning, often an aural warning sound in conjunction with a visual warning light(s), is given when there is an unsafe condition prior to takeoff. Such conditions include the improper position of the flaps or slats, the horizontal stabilizer position, the extension of speed brakes, the parking brake set, and so on. The warning is normally triggered when the crew advances the throttle and a problem is present. The value of this system is difficult to assess as attempting a takeoff when the airplane is improperly configured is likely to result in a tragic incident.

Landing configuration warning is provided when the airplane is improperly set up for landing. One common warning occurs when the all members of the landing gear are not locked in the down position and a throttle lever is reduced to a low power setting. A warning is frequently given when the flaps are extended for landing and the landing gear is not down and locked. The extension of spoilers at low altitudes is likely to provide an unsafe landing configuration warning. As with the unsafe for takeoff warning network, the crew receives an aural and visual warning when the landing configuration is improper.

Question: 9-1

Of the following, which are considered primary flight controls and which are considered secondary?

- Elevators
- Ailerons
- Speed Brakes
- Rudders
- Flaps
- Trim Tabs

Question: 9-5

Name two advantages of push-pull rods over control cables.

Question: 9-2

What effect does flap extension have during takeoff and landing?

Question: 9-6

What is the purpose of a horizontal stabilizer trim brake system?

Question: 9-3

In which mode of operation (push or pull) do flexible cables have strength?

Question: 9-7

What is the advantage of fly-by-optics control systems versus fly-by-wire?

Question: 9-4

What are the two functions of turnbuckles on an aircraft control system?

Question: 9-8

What are two benefits of a yaw damper system?

ANSWERS

Answer: 9-1

Elevators, rudder, and ailerons are primary, flaps, trim tabs, and speed brakes are secondary.

Answer: 9-5

Excerpt force in both pull and push directions; less weight per distance than steel cables.

Answer: 9-2

Increases lift allowing for lower takeoff, approach, and landing speeds.

Answer: 9-6

Arrests the motion of the stabilizer during trimming operations when the pilots elevator control movement opposes the trim direction.

Answer: 9-3

When being pulled.

Answer: 9-7

Eliminates electromagnetic interference.

Answer: 9-4

Setting control cable tension and serving as a system disconnect point.

Answer: 9-8

Increased stability for the pilot; smoother ride for the passengers.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 10

FUEL SYSTEMS (ATA 28)

Knowledge Requirements

11.10 - Fuel Systems (ATA 28)

- System lay-out;
- Fuel tanks;
- Supply systems;
- Dumping, venting and draining;
- Cross-feed and transfer;
- Indications and warnings;
- Refuelling and defueling;
- Longitudinal balance fuel systems.

3

10.10 - FUEL SYSTEMS

All powered aircraft require fuel on board to operate the engine(s). A fuel system consisting of storage tanks, pumps, filters, valves, fuel lines, metering devices, and monitoring devices is designed and certified under strict guidelines of the NAA of the manufacturing company. Each system must provide an uninterrupted flow of contaminant-free fuel regardless of the aircraft's attitude. Since fuel load can be a significant portion of the aircraft's weight, a sufficiently strong airframe must be designed. Varying fuel loads and shifts in weight during maneuvers must not negatively affect control of the aircraft in flight. Although the technician is rarely involved with designing fuel systems, a review of fuel system design criteria gives insight into how an aircraft fuel system operates.

Each fuel system must be constructed and arranged to ensure fuel flow at a rate and pressure established for proper engine and auxiliary power unit (APU) functioning under each likely operating condition. This includes any maneuver for which certification is requested during which the engine or APU may be in operation. (**Figure 10-1**) Each fuel system must be arranged so that no fuel pump can draw fuel from more than one tank at a time. There must also be a means to prevent the introduction of air into the system.

Each fuel system for a turbine engine powered airplane must meet applicable fuel venting requirements. A turbine engine fuel system must be capable of sustained operation throughout its flow and pressure range even though the fuel has some water in it.

The standard is that the engine continues to run using fuel initially saturated with water at 80°F having 0.75 cubic centimeters (cm) of free water per gallon added to it and then cooled to the most critical condition for icing likely to be encountered in operation.

FUEL SYSTEM INDEPENDENCE

Each fuel system for a multi-engine airplane must be arranged so that, in at least one system configuration, the failure of any one component (other than a fuel tank) does not result in the loss of power of more than one engine or require immediate action by the pilot to prevent the loss of power of more than one engine. If a single fuel tank (or series of fuel tanks inter connected to

function as a single fuel tank) is used on a multi-engine airplane, independent tank outlets for each engine, each incorporating a shut-off valve at the tank, must be provided. The shutoff valves may serve as firewall shutoff valves, which are also required.

However, note that if the line between the valve and the engine compartment contains more than one quart of fuel (or any greater amount shown to be safe) that can escape into the engine compartment, an additional firewall shutoff valve is needed. Lines and any components from each tank outlet to each engine must be completely independent of each other.

The fuel tank must have at least two vents arranged to minimize the probability of both vents becoming obstructed simultaneously. The filler caps must be designed to minimize the probability of incorrect installation or in flight loss.

FUEL SYSTEM LIGHTNING PROTECTION

The fuel system must be designed and arranged to prevent the ignition of fuel vapor within the system by direct lightning strikes or swept lightning strokes (where highly probable). Swept strokes occur when the lightning strike is deformed by interaction with aerodynamic forces and propagates in a unique manner due to the material and shape of the airframe surfaces. Corona and streamering must also be inhibited at fuel vent outlets since they may ignite the fuel-air mixture. A corona is a luminous discharge that occurs as a result of an electrical potential difference between the aircraft and the surrounding area.



Figure 10-1. Aircraft fuel systems must deliver fuel during any maneuver for which the aircraft is certified.

Streamer is a branch-like ionized path that occurs in the presence of a direct stroke or under conditions when lightning strokes are imminent. (*Figure 10-2*)

FUEL FLOW

The ability of the fuel system to provide fuel at a rate of flow and pressure sufficient for proper engine operation is vital in aircraft. Moreover, the fuel system must deliver the fuel at the aircraft attitude that is most critical with respect to fuel feed and quantity of unusable fuel. Tests are performed to demonstrate this performance. Fuel flowmeters are installed on most aircraft. During testing, the flowmeter is blocked and fuel must flow through or bypass the meter and still supply the engine at sufficient rate and pressure.

For gravity-flow fuel systems, the fuel flow rate must be 150 percent of the takeoff fuel consumption of the engine. For fuel pump systems, the fuel flow rate for each pump system (main and reserve supply) for each reciprocating engine must be 125 percent of the fuel flow required by the engine at the maximum takeoff power. However, the fuel pressure, with main and emergency pumps operating simultaneously, must not exceed the fuel inlet pressure limits of the engine. Auxiliary fuel systems and fuel transfer systems may operate under slightly different parameters. Turbine engine fuel systems must provide at least 100 percent required by the engine under each intended operating condition and maneuver. On aircraft with multiple fuel tanks, performance is monitored when switching to a new tank once fuel has been depleted from a tank.

FLOW BETWEEN INTERCONNECTED TANKS

If fuel can be pumped from one tank to another in flight, the fuel tank vents and the fuel transfer system must be designed so that no structural damage to any airplane component can occur because of overfilling of any tank.

UNUSABLE FUEL SUPPLY

The unusable fuel supply for each tank must be established. It cannot be less than that quantity at which the first evidence of malfunctioning appears under the most adverse fuel feed condition occurring under each intended operation and flight maneuver involving that tank. The effect on the usable fuel quantity as a result of a failure of any pump is also determined.



Figure 10-2. Lightning streamer at the wingtips of a jet fighter.

FUEL SYSTEM HOT WEATHER OPERATION

Each fuel system must be free from vapor lock when using fuel at its critical temperature, with respect to vapor formation, when operating the airplane in all critical operating and environmental conditions for which approval is requested. For turbine fuel, the critical temperature must be 43°C , -0°C , $+3^{\circ}\text{C}$ or the maximum outside air temperature for which approval is requested, whichever is more critical.

FUEL TANKS

Each fuel tank must be able to withstand, without failure, the vibration, inertia, fluid, and structural loads to which it may be subjected in operation. Fuel tanks with flexible liners must demonstrate that the liner is suitable for the particular application. The total usable capacity of any tank(s) must be enough for at least 30 minutes of operation at maximum continuous power. Each integral fuel tank must have adequate facilities for interior inspection and repair. Additionally, each fuel quantity indicator must be adjusted to account for the unusable fuel supply.

FUEL TANK TESTS

Aircraft fuel tanks must be able to withstand the forces that are encountered throughout the entire spectrum of operation. Various tank testing standards exist. A main focus is to ensure that tanks are strong enough to remain fully operational and not deform when under

various loads. Vibration resistance without leaking is also a concern. Tanks are tested under the most critical condition that may be encountered. Fuel tank supporting structure must be designed for the critical loads that could occur during flight or when landing with fuel pressure loads.

FUEL TANK INSTALLATION

Various standards exist for fuel tank installations. No fuel tank may be on the engine side of a firewall, and there must be at least ½-inch of clearance between the fuel tank and the firewall. Each tank must be isolated from personnel compartments of the aircraft by a fume proof and fuel-proof enclosure that is vented and drained to the exterior of the airplane. Pressurization loads should not affect the tank(s). Each tank compartment must be ventilated and drained to prevent the accumulation of flammable fluids or vapors. Compartments adjacent to tanks must also be ventilated and drained.

Aircraft fuel tanks must be designed, located, and installed to retain fuel when subjected to inertia loads resulting from ultimate static load factors, and under conditions likely to occur when the airplane lands on a paved runway at a normal landing speed with the landing gear retracted. They must also retain fuel if one of the gear collapses or if an engine mount tears away. (*Figure 10-3*)

Many aircraft have fuel tanks that are not metal. Bladder fuel tanks have their own standards of construction and installation. As with metal tanks, there must be pads to prevent any chafing between each tank and its supports. The padding must be nonabsorbent or treated to prevent the absorption of fuel.



Figure 10-3. Aircraft fuel tanks must be designed to retain fuel in the event of a gear-up landing. The fuel system drain valve should be located to prevent spillage.

Bladders must be supported so they are not required to support the entire fuel load. Surfaces adjacent to the liner must be smooth and free from projections that could cause wear. A positive pressure must be maintained within the vapor space of each bladder cell under any condition of operation, or it should be shown not to collapse under zero or negative pressure. Siphoning of fuel or collapse of bladder fuel cells should not result from improper securing or loss of the fuel filler cap. Bladder type fuel cells must have a retaining shell at least equivalent to a metal fuel tank in structural integrity.

FUEL TANK EXPANSION SPACE

Each fuel tank must have an expansion space of not less than two percent of the tank capacity. This is waived if the tank vent discharges clear of the airplane, in which case no expansion space is required. It must be impossible to fill the expansion space inadvertently with the airplane in the normal ground attitude.

FUEL TANK SUMP

Keeping contaminants out of the fuel delivered to the engine begins with the proper construction and installation of the fuel tank(s). Each tank must have a drainable sump with an effective capacity, in the normal ground and flight attitudes, of 0.25 percent of the tank capacity, or gallon, whichever is greater. Each fuel tank must allow drainage of any hazardous quantity of water from any part of the tank to its sump with the airplane in the normal ground attitude.

FUEL TANK FILLER CONNECTION

Each fuel tank filler connection must be specifically marked. Aircraft with engines that use only gasoline fuel must have filler openings no larger than 2.36 inches in diameter. Turbine fuel aircraft filler openings must be no smaller than 2.95 inches. Spilled fuel must not enter the fuel tank compartment or any part of the airplane other than the tank itself. Each filler cap must provide a fuel-tight seal for the main filler opening. However, there may be small openings in the fuel tank cap for venting purposes or for the purpose of allowing passage of a fuel gauge through the cap. Fuel filling points must have a provision for electrically bonding the airplane to ground fueling equipment (except pressure fueling connection points).

FUEL TANK VENTS

To allow proper fuel flow, each fuel tank must be vented from the top part of the expansion space. Vent outlets must be located and constructed in a manner that minimizes the possibility of being obstructed by ice or other foreign matter. Siphoning of fuel during normal operation must not occur. Venting capacity must allow the rapid relief of excessive differences of pressure between the interior and exterior of the tank. The airspaces of tanks with interconnected outlets must also be interconnected. There must be no point in any vent line where moisture can accumulate either on the ground or during level flight (unless drainage is provided by an accessible drain valve). Fuel tank vents may not terminate at a point where the discharge of fuel from the vent outlet constitutes a fire hazard or from which fumes may enter personnel compartments. The vents must be arranged to prevent the loss of fuel when the airplane is parked in any direction on a ramp having a one percent slope. Fuel discharged because of thermal expansion is allowed.

FUEL TANK OUTLET

There must be a fuel strainer for the fuel tank outlet or for the booster pump. It must also be accessible for inspection and cleaning. Turbine-engine aircraft fuel strainers must prevent the passage of any object that could restrict fuel flow or damage any fuel system component.

PRESSURE FUELING SYSTEMS

Pressure fueling systems are used on many large, high performance, and air carrier aircraft. Each pressure fueling system fuel manifold connection must have means to prevent the escape of hazardous quantities of fuel from the system if the fuel entry valve fails. A means for automatic shutoff must be provided to prevent the quantity of fuel in each tank from exceeding the maximum quantity approved for that tank. A means must also be provided to prevent damage to the fuel system in the event of failure of the automatic shutoff means prescribed in this section. All parts of the fuel system up to the tank that are subjected to fueling pressures must have a proof pressure of 1.33 times and an ultimate pressure of at least 2.0 times the surge pressure likely to occur during fueling.

FUEL PUMPS

Fuel pumps are part of most aircraft fuel systems. Standards exist for main pumps and emergency pumps. Operation of any fuel pump may not affect engine operation by creating a hazard, regardless of the engine power or thrust setting or the functional status of any other fuel pump. Turbine engines require dedicated fuel pumps for each engine. Any pump required for operation is considered a main fuel pump. The power supply for the main pump for each engine must be independent of the power supply for each main pump for any other engine. There must also be a bypass feature for each positive displacement pump.

Emergency pumps are used and must be immediately available to supply fuel to the engine if any main pump fails. The power supply for each emergency pump must be independent of the power supply for each corresponding main pump. If both the main fuel pump and the emergency pump operate continuously, there must be a means to indicate a malfunction of either pump to the appropriate flight crew member.

FUEL SYSTEM LINES AND FITTINGS

Even aircraft fuel system fluid lines and fittings have standards to ensure proper fuel system operation. Each fuel line must be installed and supported to prevent excessive vibration and to withstand loads due to fuel pressure and accelerated flight conditions. Lines connected to components of the airplane, between which relative motion could exist, must have provisions for flexibility. Flexible hose assemblies are used when lines may be under pressure and subject to axial loads. Any hose that is used must be shown to be suitable for a particular application. Where high temperatures may exist during engine operation or after shutdown, fuel hoses must be capable of withstanding these temperatures.

OTHER FUEL SYSTEM COMPONENTS

Fuel system components in an engine nacelle or in the fuselage must be protected from damage that could result in spillage of enough fuel to constitute a fire hazard as a result of a wheels-up landing on a paved runway.

FUEL VALVES AND CONTROLS

There must be a means to allow appropriate flight crew members to rapidly shut off the fuel to each engine

individually in flight. No shutoff valve may be on the engine side of any firewall. There must be means to guard against inadvertent operation of each shutoff valve and means to reopen each valve rapidly after it has been closed. Each valve and fuel system control must be supported so that loads resulting from its operation, or from accelerated flight conditions, are not transmitted to the lines connected to the valve. Gravity and vibration should not affect the selected position of any valve.

Fuel valve handles and their connections to valve mechanisms must have design features that minimize the possibility of incorrect installation. Check valves must be constructed to preclude incorrect assembly or connection of the valve. Fuel tank selector valves must require a separate and distinct action to place the selector in the OFF position. The tank selector positions must be located in such a manner that it is impossible for the selector to pass through the OFF position when changing from one tank to another.

FUEL STRAINER OR FILTER

In addition to fuel tank strainers already discussed, there must be a fuel strainer, or filter, between the fuel tank outlet and the inlet of either the fuel metering device or an engine driven positive displacement pump, whichever is nearer the fuel tank outlet. This fuel strainer, or filter, must be accessible for draining and cleaning and must incorporate a screen or element that is easily removable. The fuel strainer should have a sediment trap and drain, except that it need not have a drain if the strainer or filter is easily removable for drain purposes. The fuel strainer should also be mounted so that its weight is not supported by the connecting lines.

It should have the capacity to ensure that engine fuel system function is not impaired when fuel is contaminated to a degree that is greater than that established for the engine during its type certification. Commuter category airplanes must have a means to automatically maintain the fuel flow if ice clogs a filter.

AIRCRAFT FUEL SYSTEM LAYOUT

While each manufacturer designs its own fuel system, the basic fuel system requirements referenced at the beginning of this sub-module yield fuel systems of similar design and function in the field. In the following sections, representative examples of various

fuel systems are given. The fuel systems of other aircraft are similar but certainly not identical. Consult the manufacturer's maintenance manual for complete fuel system information on the aircraft upon which maintenance is being performed. Each aircraft fuel system must store and deliver clean fuel to the engine(s) at a pressure and flow rate able to sustain operations regardless of the operating conditions of the aircraft.

Fuel systems on high performance and large transport category jet aircraft are complex with some features and components not found in reciprocating-engine aircraft fuel systems. They typically contain more redundancy and facilitate numerous options from which the crew can choose while managing the aircraft's fuel load. Features like an on board APU, single point pressure refueling, and fuel jettison systems, which are not needed on smaller aircraft, add to the complexity of an airliner fuel system. Jet transport fuel systems can be regarded as a handful of fuel subsystems as follows:

1. Storage
2. Vent
3. Distribution
4. Feed
5. Indicating

FUEL TANKS

Most transport category aircraft fuel systems are very much alike. Integral fuel tanks are the norm with much of each wing's structure sealed to enable its use as a fuel tank. Center wing section or fuselage tanks are also common. These may be sealed structure or bladder type. Jet transport aircraft carry tens of thousands of pounds of fuel on board.

Figure 10-4 shows a diagram of a Boeing 777 fuel tank configuration with tank capacities. Note that there are optional fuel storage configurations available on the same model airliner. For example, airlines expecting to use an aircraft on transoceanic flights may order the aircraft with long-range auxiliary tanks. These additional tanks, usually located in the fuselage section of the aircraft, can alter fuel management logistics in addition to complicating the fuel system.

In addition to main and auxiliary fuel tanks, surge tanks may also be found on jet transports. These normally empty tanks located in the wing structure outboard of the main wing tanks are used for fuel overflow. A

Tank	Gallons	Pounds*
Left Main Tank	9 560	64 000
Right Main Tank	9 560	64 000
Center Tank	26 100	174 900
Total	45 200	302 900

* Usable fuel at level attitude.

Fuel density = 6.7 pounds per U.S. gallon.

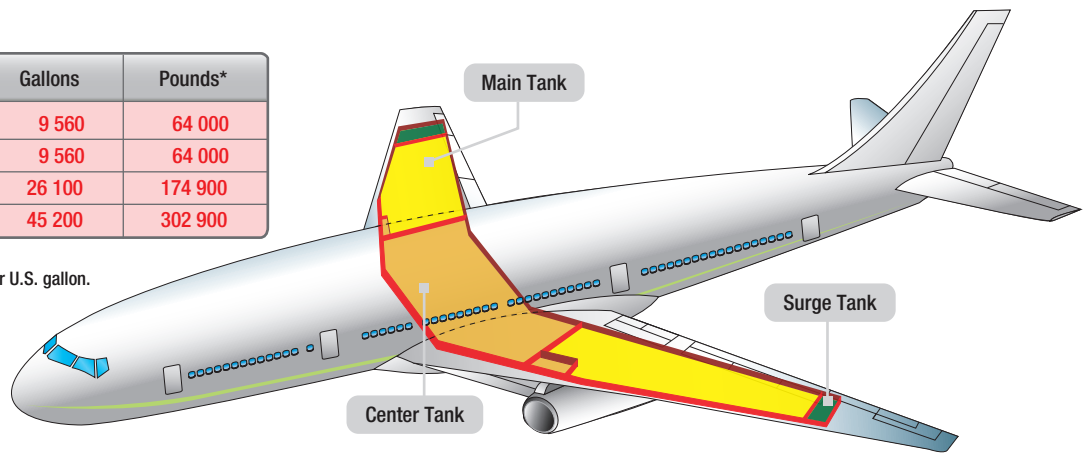


Figure 10-4. Boeing 777 fuel tank locations and capacities.

check valve allows the one way drainage of fuel back into the main tanks. Surge tanks are also used for fuel system venting.

INTEGRAL FUEL TANKS

On many aircraft, especially transport category and high performance aircraft, part of the structure of the wings or fuselage is sealed with a fuel resistant two part sealant to form a fuel tank. The sealed skin and structural members provide the highest volume of space available with the lowest weight. This type of tank is called an integral fuel tank since it forms a tank as a unit within the airframe structure.

Integral fuel tanks in the otherwise unused space inside the wings are most common. Aircraft with integral fuel tanks in the wings are said to have wet wings. For fuel management purposes, sometimes a wing is sealed into separate tanks and may include a surge tank or an overflow tank, which is normally empty but sealed to hold fuel when needed.

When an aircraft maneuvers, the long horizontal nature of an integral wing tank requires baffling to keep the fuel from sloshing. The wing ribs and box beam structural members serve as baffles and others may be added specifically for that purpose. Baffle check valves are commonly used. These valves allow fuel to move to the low, inboard sections of the tank but prevent it from moving outboard. They ensure that the fuel boost pumps located in the bottom of the tanks at the lowest points above the sumps always have fuel to pump regardless of aircraft attitude. (Figure 10-5)

Integral fuel tanks must have access panels for inspection and repairs of the tanks and other fuel system components. On large aircraft, technicians physically enter the tank for maintenance. Transport category aircraft often have more than a dozen oval access panels or tank plates on the bottom surface of the wing for this purpose. (Figure 10-6A) These aluminum panels are each sealed into place with an O-ring and an aluminum gasket for electrostatic bonding. An outer clamp ring is tightened to the inner panel with screws, as shown in Figure 10-6B.

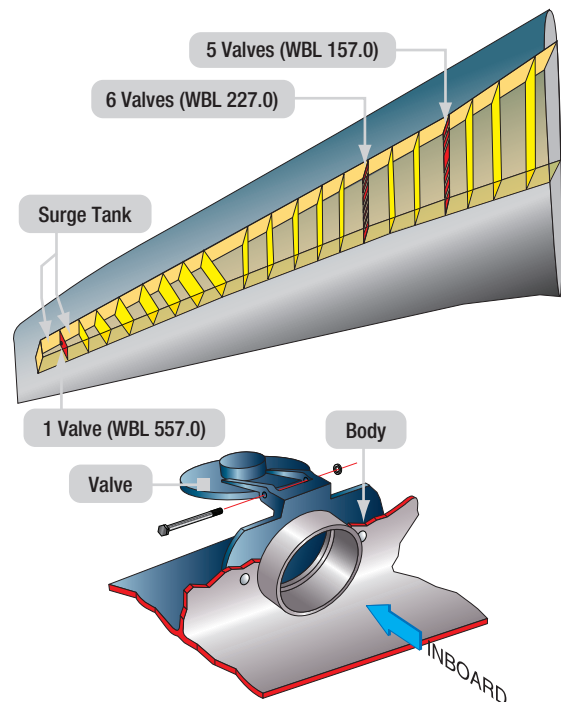


Figure 10-5. Baffle check valves are installed in the locations shown in the integral tank rib structure of a Boeing 737 airliner. Fuel is prevented from flowing outboard during maneuvers. The tank boost pumps are located inboard of WBL 157.

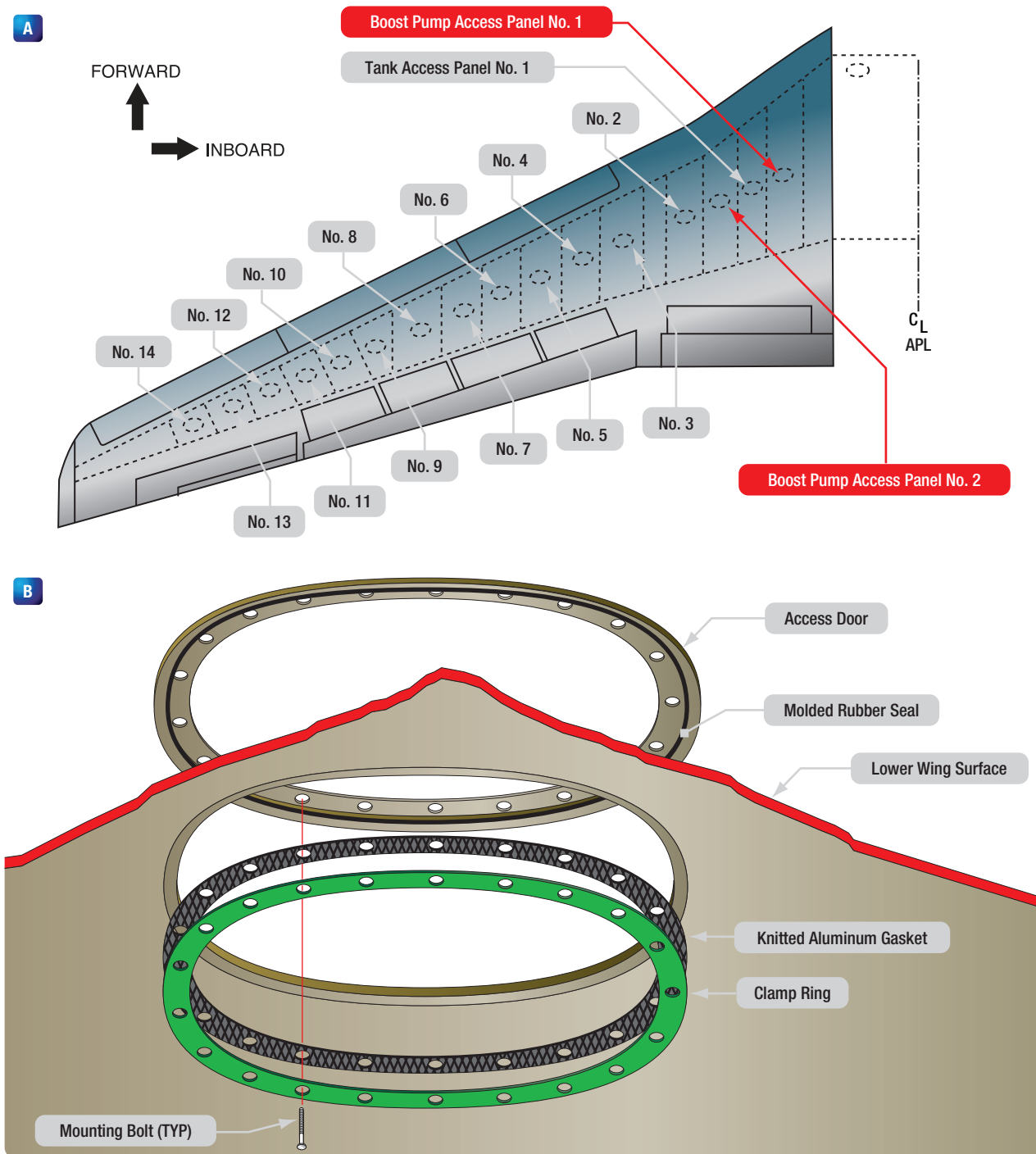


Figure 10-6. Fuel tank access panel locations on a Boeing 737 (A), and typical fuel tank access panel seals (B).

When entering and performing maintenance on an integral fuel tank, all fuel must be emptied from the tank and strict safety procedures must be followed. Fuel vapors must be purged from the tank and respiratory equipment must be used by the technician. A full-time spotter must be positioned just outside of the tank to assist if needed.

Aircraft using integral fuel tanks normally have sophisticated fuel systems that include in-tank boost pumps. There are usually at least two pumps in each tank that deliver fuel to the engine(s) under positive pressure. On various aircraft, these in-tank boost pumps are also used to transfer fuel to other tanks, jettison fuel, and defuel the aircraft.

BLADDER FUEL TANKS

A fuel tank made out of a reinforced flexible material called a bladder tank can be used instead of a rigid tank. A bladder tank contains most of the features and components of a rigid tank but does not require as large an opening in the aircraft skin to install. The tank, or fuel cell as it is sometimes called, can be rolled up and put into a specially prepared structural bay or cavity through a small opening, such as an inspection opening. Once inside, it can be unfurled to its full size.

Bladder tanks must be attached to the structure with clips or other fastening devices. They should lie smooth and unwrinkled in the bay. It is especially important that no wrinkles exist on the bottom surface so that fuel contaminants are not blocked from settling into the tank sump. (*Figure 10-7*)

Bladder fuel tanks that develop leaks can also be repaired. Most commonly, they are patched using patch material, adhesive, and methods approved by the manufacturer. As with soldered tanks, the patch has a required overlap of the damaged area. Damage that penetrates completely through the bladder is repaired with an external, as well as internal, patch. Synthetic bladder tanks have a limited service life. At some point, they seep fuel beyond acceptable limits and need to be replaced.

Bladder tanks are usually required to remain wetted with fuel at all times to prevent drying and cracking of the bladder material. Storage of bladder tanks without fuel can be accomplished by coating the tanks with a substance to prevent drying, such as clean engine oil that can be flushed from the tank when ready to return to service. Follow all manufacturer's instructions for the care and repair of these common tanks. It is important to ensure that bladder tanks are correctly secured in place with the proper fasteners when reinstalling them in the aircraft after a repair.

Bladder fuel tanks are used on aircraft of all size. They are strong and have a long life with seams only around installed features, such as the tank vents, sump drain, filler spout, etc. When a bladder tank develops a leak, the technician can patch it following manufacturer's instructions. The cell can also be removed and sent to a fuel tank repair station familiar with and equipped to perform such repairs.

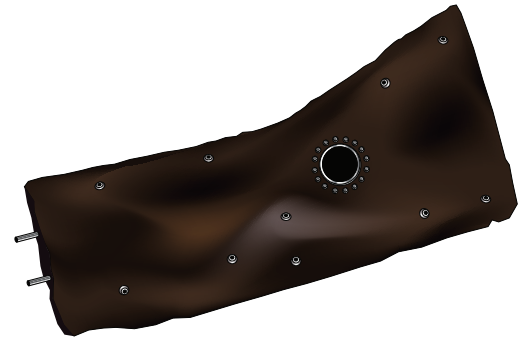


Figure 10-7. An aircraft bladder fuel tank.

The soft flexible nature of bladder fuel tanks requires that they remain wet. Should it become necessary to store a bladder tank without fuel in it for an extended period of time, it is common to wipe the inside of the tank with a coating of clean engine oil. Follow the manufacturer's instructions for the dry storage procedures for fuel cells.

FUEL SUPPLY SYSTEMS

Fuel is supplied from the fuel tanks of transport category aircraft to the engines and auxiliary power unit through a distribution system. This consists of fuel lines that connect various valves, pumps, heater exchangers and indication system components. The fuel feed, cross feed and transfer systems are all part of the distribution system. Pressure fueling and refueling components may also be considered part of the fuel supply/distribution system. They are covered in a separate section of this *Sub-Module*.

FUEL FEED

The fuel feed system is the heart of the fuel supply system since it delivers fuel to the engines. Jet transport aircraft supply fuel to the engines via in-tank fuel boost pumps, usually two per tank. They pump fuel under pressure through a shutoff valve for each engine. A manifold or connecting tubing typically allows any tank to supply any engine through the use of cross-feed valves.

Boost pump bypass valves allow fuel flow should a pump fail and check valves allow fuel flow only in the proper direction towards the engines. Note that the engines are designed to be able to run without any fuel boost pumps operating. However, each engine's fuel shutoff valve must be open to allow any flow to the engines from the tanks. (*Figure 10-8*)

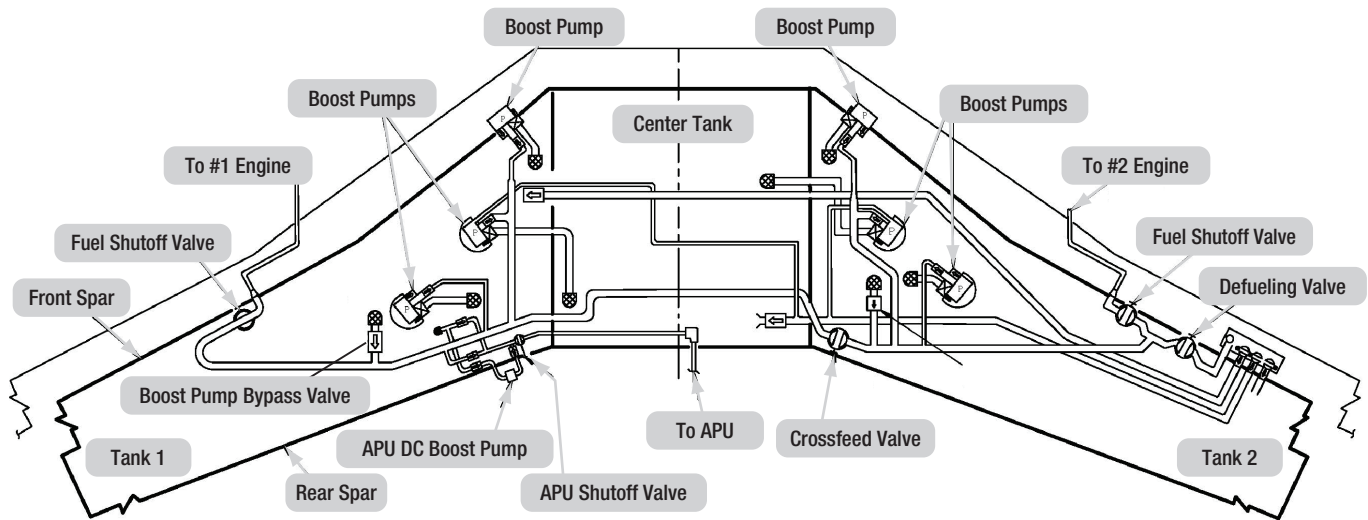


Figure 10-8. The fuel feed system of a Boeing 737.

CROSS-FEED AND TRANSFER SYSTEMS

It is important for the flight crew as well as technicians to manage the location of fuel between tanks. The fuel transfer system is the series of fuel lines and valves that permits movement of fuel from one tank to another on board the aircraft. Fuel boost pumps or dedicated transfer pumps move fuel from the tank in which the pump is located into a manifold containing valves for the other tanks. By opening the fuel valve (or refueling valve) to the tank into which fuel is desired, the fuel is transferred from one tank, into the manifold, and then into the desired tank.

Figure 10-9 shows the fuel system diagram for a DC-10. Dedicated transfer boost pumps move fuel into the transfer manifold. Opening the fuel valve on one of the tanks transfers the fuel into that tank. Note that the transfer manifold and boost pumps are also used to jettison fuel overboard by opening the proper dump valves with a transfer boost pump(s) operating. Additionally, the transfer system can function to supply the engines if the normal engine fuel feed malfunctions.

Not all jet transports have such fuel transfer capability. Through the use of a fuel feed manifold and cross-feed valves, some aircraft simply allow engines to be run off fuel from any tank as a means for managing fuel location. This is seen in *Figure 10-8*. The 737 does not have transfer pumps and a dedicated transfer manifold. It does have a fuel manifold made up of interconnect fuel lines from each boost pump. A cross feed valve separates the manifold into two halves. Normally, the left fuel

manifold is used to operate the left engine and the right fuel manifold is used to operate the right engine. The boost pumps on one side of the aircraft pump fuel to the engine on the opposite side when the cross-feed valve is open. Essentially, by opening the cross-feed valve, the fuel manifold can be pressurized and supplied with fuel from any boost pump on the aircraft. Thus, by opening the engine fuel shutoff valve to a particular engine, the engine can operate off of any boost pump in any tank.

FUEL DUMPING, VENTING AND DRAINING

FUEL SYSTEM DRAINS

Aircraft fuel systems must be fitted with at least one drain to allow safe drainage of the entire fuel system with the airplane in its normal ground attitude. The drain must discharge the fuel clear of all parts of the aircraft. A readily accessible drain valve that can easily be opened and closed is required. It must have a manual or automatic means for locking in the closed position, and it must be observable that it is closed. Fuel should be collectible from the system drain valve so it can be examined. The location of the valve should be such that spillage is prevented should a gear up landing be made.

FUEL VENT SYSTEMS

Transport category fuel systems require venting similar to reciprocating engine aircraft fuel systems. A series of vent tubing and channels exists that connects all tanks to vent space in the surge tanks (if present) or vent overboard. Venting must be configured to ensure the fuel is vented regardless of the attitude of the aircraft or

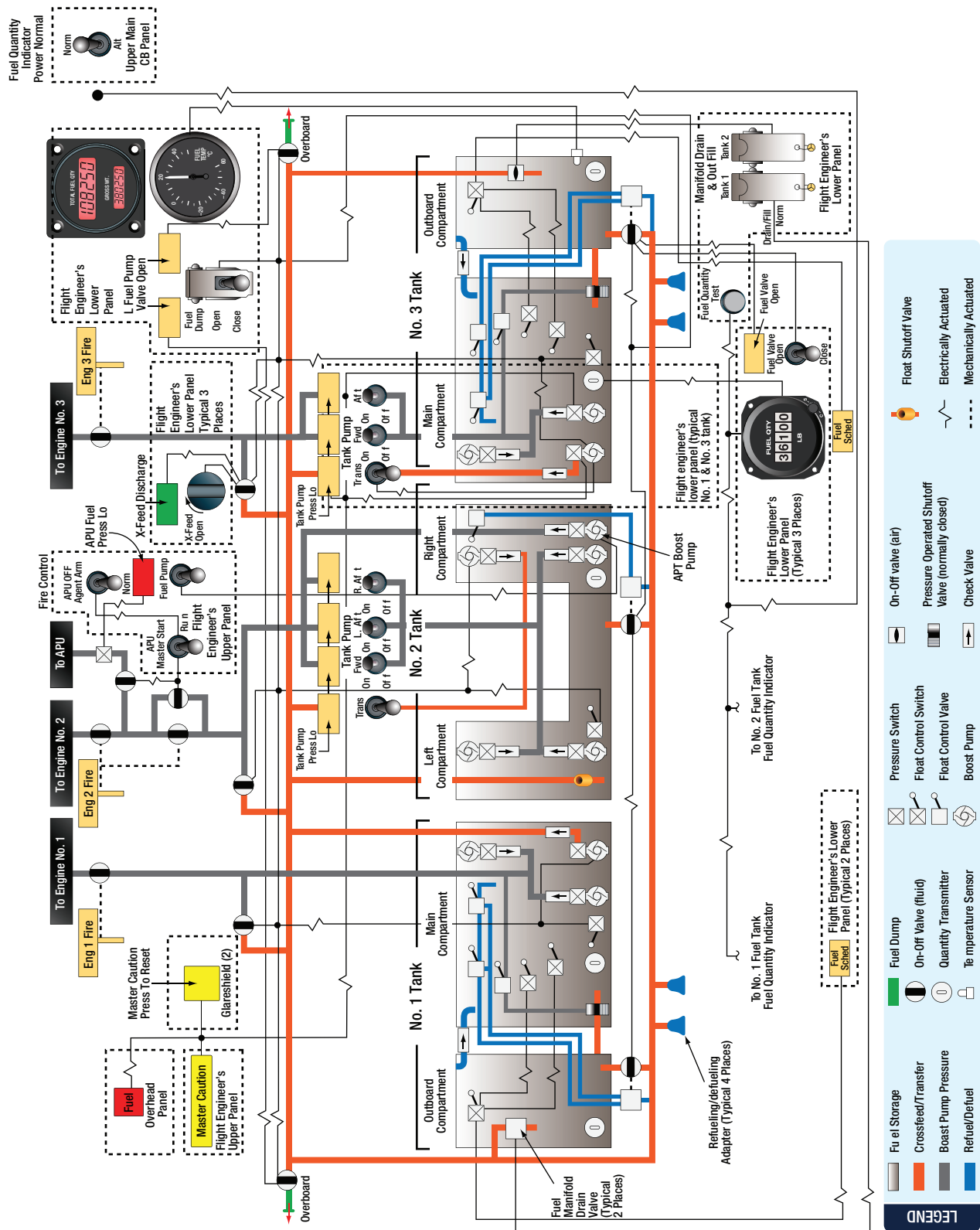


Figure 10-9. The fuel distribution systems, components, and cockpit controls of a DC-10 airliner. NOTE: Fuel transfer system components and lines are used to complete the fuel dump system, the refuel/defuel system, back-up fuel delivery system, and the fuel storage system.

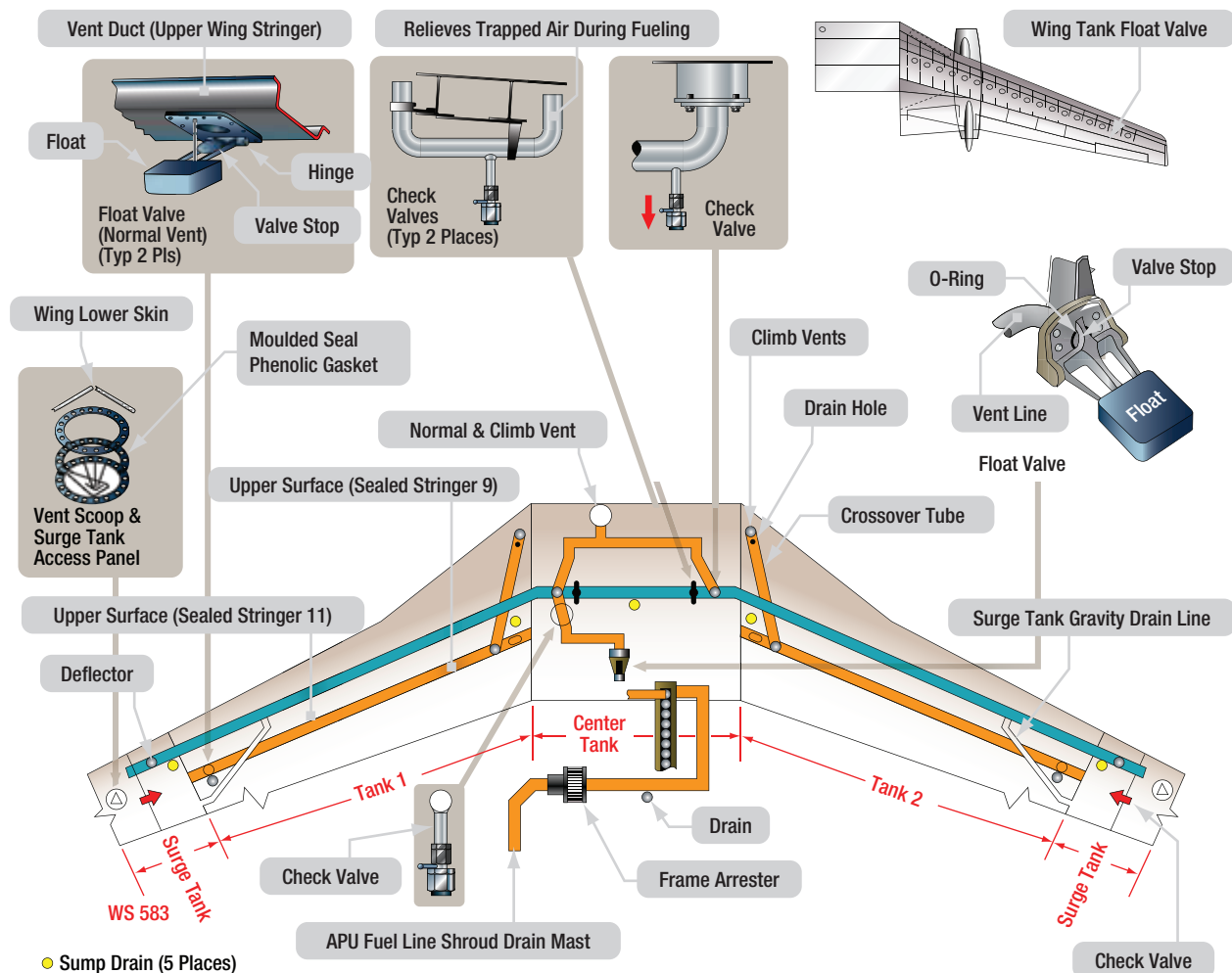


Figure 10-10. A fuel vent system with associated float and check valves that stop fuel and keep the tanks vented regardless of the aircraft attitude.

the quantity of fuel on board. This sometimes requires the installation of various check valves, float valves, and multiple vent locations in the same tank. **Figure 10-10** shows the fuel vent system of a Boeing 737.

FUEL JETTISONING SYSTEM

If an aircraft's design landing weight is less than that of the maximum takeoff weight, a situation could occur in which a landing is desired before sufficient fuel has burned off to lighten the aircraft. Fuel jettisoning systems are required on these aircraft so that fuel can be jettisoned in flight to avoid structural damage caused by landing the aircraft when it is too heavy. Fuel jettisoning systems are also referred to as fuel dump systems. (**Figure 10-11**)

Fuel jettisoning systems must meet several standards. The average rate of fuel jettisoning must be at least 1 percent of the maximum weight per minute, except that the time required to jettison the fuel need not be less

than 10 minutes. Fuel jettisoning must be demonstrated at maximum weight with flaps and landing gear up and in a power-off glide at 1.4 VS1. It must also be demonstrated during a climb with a critical engine inoperative and the remaining engines at maximum continuous power. Finally, the fuel jettisoning system must be performed during level flight at 1.4 VS1 if the glide and climb tests show that this condition could be critical.

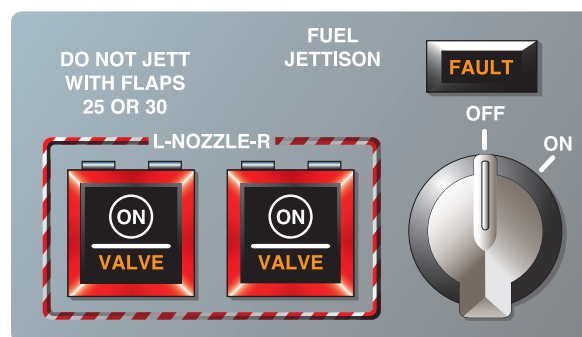


Figure 10-11. The fuel jettison panel on a Boeing 767.

During the demonstration of the fuel jettisoning system, it must demonstrate that it operates without fire hazard. No fuel or fumes can enter any part of the aircraft. The fuel must discharge clear of any part of the aircraft and the jettisoning operation must not adversely affect the controllability of the airplane. (*Figure 10-12*) The system must be designed so that any reasonably probable single malfunction in the system does not result in a hazardous condition due to unsymmetrical jettisoning of, or inability to jettison, fuel. The fuel jettisoning valve must be designed to allow flight crew members to close the valve during any part of the jettisoning operation.



Figure 10-12. Fuel being jettisoned free of the airframe on a transport category aircraft.

For turbine engine powered airplanes, the jettisoning system must be designed so that it is not possible to jettison fuel from the tanks used for takeoff and landing below the fuel level that would allow climb from sea level to 10 000 feet plus 45 minutes cruise at a speed for maximum range. If certain flight control configurations negatively affect jettisoning the fuel, a placard stating so must be posted next to the actuation control in the cockpit. An example of this is the "DO NOT JETT WITH FLAPS 25 OR 30" placard shown in *Figure 10-11*.

FUEL SYSTEM COMPONENTS

To better understand aircraft fuel systems and their operation, the following discussion of various components of aircraft fuel systems is included. Fuel tanks are a key component that have already been discussed above.

FUEL LINES AND FITTINGS

Aircraft fuel lines can be rigid or flexible depending on location and application. Rigid lines are often made of aluminum alloy and are connected with Army/Navy (AN) or military standard (MS) fittings. However, in the engine compartment, wheel wells, and other areas, subject to damage from debris, abrasion, and heat, stainless steel lines are often used.

Flexible fuel hose has a synthetic rubber interior with a reinforcing fiber braid wrap covered by a synthetic exterior. (*Figure 10-13*) The hose is approved for fuel and no other hose should be substituted. Some flexible fuel hose has a braided stainless steel exterior. (*Figure 10-14*) The diameters of all fuel hoses and line are determined by the fuel flow requirements of the aircraft fuel system. Flexible hoses are used in areas where vibration exists between components, such as between the engine and the aircraft structure.

Sometimes manufacturers wrap either flexible or rigid fuel lines to provide even further protection from abrasion and especially from fire. A fire sleeve cover is held over the line with steel clamps at the end fittings. (*Figure 10-15*)

As mentioned, aircraft fuel line fitting are usually either AN or MS fittings. Both flared and flareless fitting are used. Problems with leaks at fittings can occur. Technicians are cautioned to not over tighten a leaky fitting. If the proper torque does not stop a leak, depressurize the line, disconnect the fitting and visually inspect it for a cause. The fitting or line should be replaced if needed. Replace all aircraft fuel lines and fittings with approved replacement parts from the manufacturer. If a line is manufactured in the shop, approved components must be used.



Figure 10-13. A typical flexible aircraft fuel line with braided reinforcement.

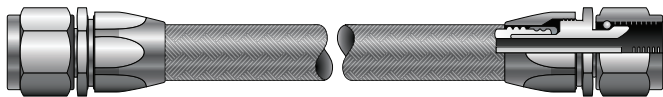


Figure 10-14. A braided stainless steel exterior fuel line with fittings.

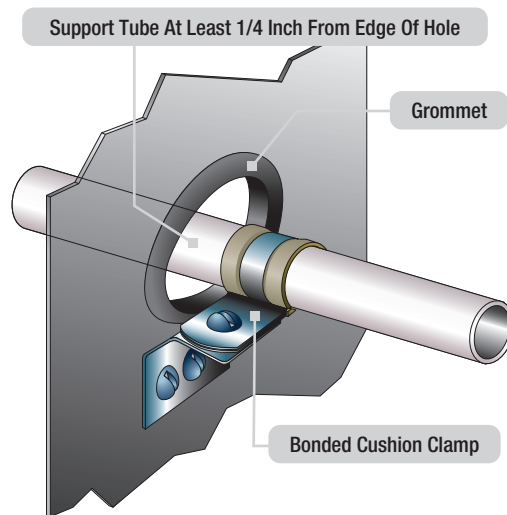


Figure 10-15. Exterior fuel hose wrap that protects from fire, as well as abrasion, shown with the clamps and pliers used to install it.

Several installation procedures for fuel hoses and rigid fuel lines exist. Hoses should be installed without twisting. The writing printed on the outside of the hose is used as a lay line to monitor fuel hose twist. Separation should be maintained between all fuel hoses and electrical wiring. Never clamp wires to a fuel line. When separation is not possible, always route the fuel line below any wiring. If a fuel leak develops, it does not drip onto the wires.

Metal fuel lines and all aircraft fuel system components need to be electrically bonded and grounded to the aircraft structure. This is important because fuel flowing through the fuel system generates static electricity that must have a place to flow to ground rather than build up. Special bonded cushion clamps are used to secure rigid fuel lines in place. They are supported at intervals shown in *Figure 10-16*.

All fuel lines should be supported so that there is no strain on the fittings. Clamp lines so that fittings are aligned. Never draw two fittings together by threading. They should thread easily and a wrench should be used only for tightening. Additionally, a straight



Tubing OD (inch)	Approximate Distance Between Supports (inches)
1/8 to 3/16	9
1/4 to 5/16	12
3/8 to 1/2	16
5/8 to 3/4	22
7 to 1-1/4	30
1-1/2 to 2	40

Figure 10-16. Rigid metallic fuel lines are clamped to the airframe with electrically bonded cushion clamps at specified intervals.

length of rigid fuel line should not be made between two components or fitting rigidly mounted to the airframe. A small bend is needed to absorb any strain from vibration or expansion and contraction due to temperature changes.

FUEL VALVES

There are many fuel valves used in aircraft fuel systems. They are used to shut off fuel flow or to route the fuel to a desired location. Large aircraft fuel systems have numerous valves. Most simply open and close and are known by different names related to their location and function in the fuel system (e.g., shutoff valve, transfer valve, cross-feed valve). Fuel valves can be manually operated, solenoid operated, or operated by electric motor. A feature of all aircraft fuel valves is a means for positively identifying the position of the valve at all times. Motor and solenoid-operated valves use position annunciator lights to indicate valve position in addition to the switch position. Flight management system (FMS) fuel pages also display the position of the fuel valves graphically in diagrams called up on the flat screen monitors. (*Figure 10-17*)



Figure 10-17. The graphic depiction of the fuel system on this electronic centralized aircraft monitor (ECAM) fuel page includes valve position information.

NOTE: Many valves have an exterior position handle, or lever, that indicates valve position. When maintenance personnel directly observe the valve, it can be manually positioned by the technician using this same lever. (Figure 10-18)

MANUALLY-OPERATED GATE VALVES

In complex fuel systems of transport category aircraft, fuel flow is controlled with a series of ON/OFF, or shutoff, type valves that are plumbed between system components. Hand-operated gate valves can be used, especially as fire control valves, requiring no electrical power to shutoff fuel flow when the emergency fire handle is pulled. The valves are typically positioned in the fuel feed line to each engine. Hand operated gate valves are also featured as ground-operated defuel valves and as boost pump isolation valves, which shut off the fuel to the inlet of the boost pump, allowing it to be changed without emptying the tank.

Gate valves utilize a sealed gate or blade that slides into the path of the fuel, blocking its flow when closed. **Figure 10-19** shows a typical hand-operated gate valve. When the handle is rotated, the actuating arm inside the valve moves the gate blade down between seals and into the fuel flow path. A thermal relief bypass valve is incorporated to relieve excess pressure buildup against the closed gate due to temperature increases.

MOTOR OPERATED VALVES

The use of electric motors to operate fuel system valves is common on large aircraft due to the remote location from the cockpit of fuel system components. The types



Figure 10-18. This motor-operated gate valve has a red position indicating lever that can be used by maintenance personnel to identify the position of the valve. The lever can be moved by the technician to position the valve.

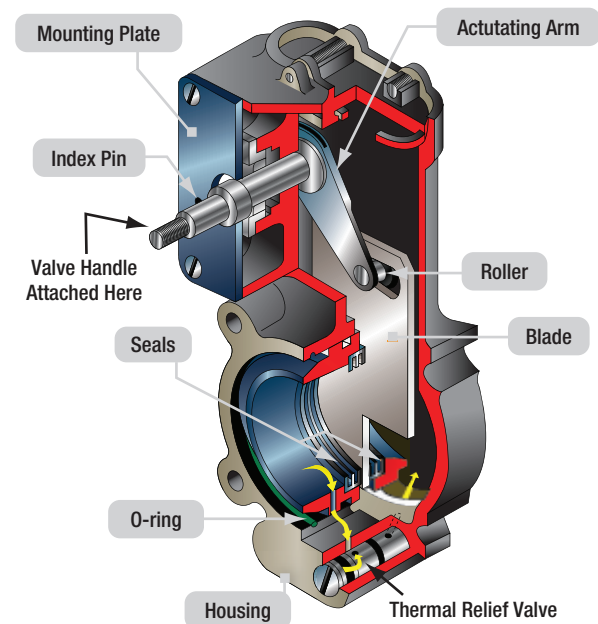


Figure 10-19. A hand-operated gate valve used in transport category aircraft fuel systems.

of valves used are basically the same as the manually operated valves, but electric motors are used to actuate the units. The two most common electric motor operated fuel valves are the gate valve and the plug type valve.

The motor operated gate valve uses a geared, reversible electric motor to turn the actuating arm of the valve that moves the fuel gate into or out of the path of the fuel. As with the manually operated gate valve, the gate or blade is sealed. A manual override lever allows the technician to observe the position of the valve or manually position it. (Figure 10-20)

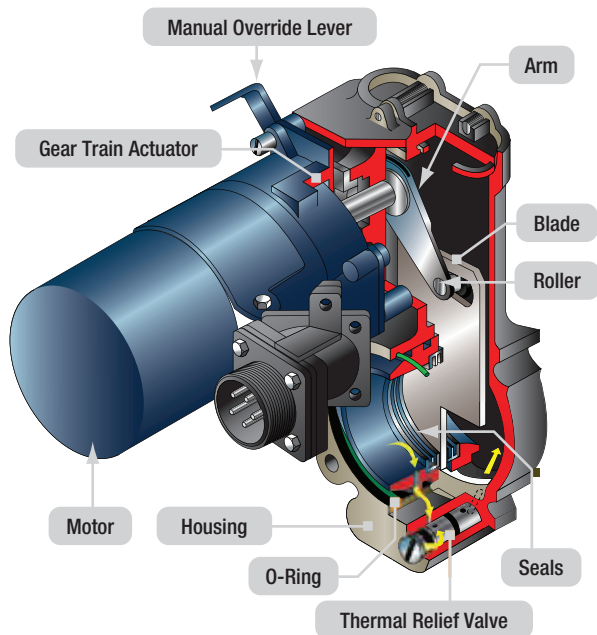


Figure 10-20. An electric motor-driven gate valve commonly used in large aircraft fuel systems.

Less common is the use of a motorized plug type fuel valve; an electric motor is used to rotate the plug or drum rather than it being rotated manually. Regardless of the type of valve used, large aircraft fuel system valves either allow fuel to flow or shut off flow.

SOLENOID-OPERATED VALVES

An additional way to operate a remotely located fuel valve is through the use of electric solenoids. A poppet type valve is opened via the magnetic pull developed when an opening solenoid is energized. A spring forces a locking stem into a notch in the stem of the poppet to lock the valve in the open position. Fuel then flows through the opening vacated by the poppet.

To close the poppet and shut off fuel flow, a closing solenoid is energized. Its magnetic pull overcomes the force of the locking stem spring and pulls the locking stem out of the notch in the poppet stem. A spring behind the poppet forces it back onto its seat. A characteristic of solenoid-operated fuel valves is that they open and close very quickly. (*Figure 10-21*)

FUEL PUMPS

Other than aircraft with gravity-feed fuel systems, all aircraft have at least one fuel pump to deliver clean fuel under pressure to the fuel metering device for each engine. Engine-driven pumps are the primary delivery device. Auxiliary pumps are used on many aircraft as

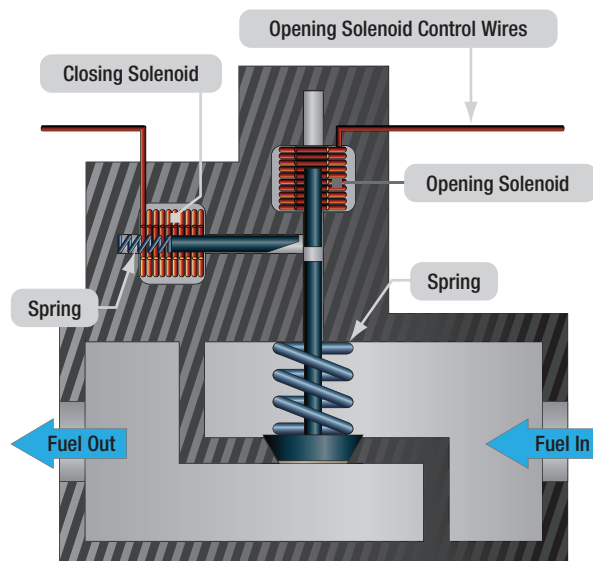


Figure 10-21. A solenoid-operated fuel valve uses the magnetic force developed by energized solenoids to open and close a poppet.

well. Sometimes known as booster pumps or boost pumps, auxiliary pumps are used to provide fuel under positive pressure to the engine-driven pump and during starting when the engine-driven pump is not yet up to speed for sufficient fuel delivery. They are also used to back up the engine-driven pump during takeoff and at high altitude to guard against vapor lock. On many large aircraft, boost pumps are used to move fuel from one tank to another. There are many different types of auxiliary fuel pumps in use. Most are electrically operated.

CENTRIFUGAL BOOST PUMPS

The most common type of auxiliary fuel pump used on aircraft, especially large and high-performance aircraft, is the centrifugal pump. It is electric motor driven and most frequently is submerged in the fuel tank or located just outside of the bottom of the tank with the inlet of the pump extending into the tank. If the pump is mounted outside the tank, a pump removal valve is typically installed so the pump can be removed without draining the fuel tank. (*Figure 10-22*) A centrifugal boost pump is a variable displacement pump. It takes in fuel at the center of an impeller and expels it to the outside as the impeller turns. (*Figure 10-23*)

An outlet check valve prevents fuel from flowing back through the pump. A fuel feed line is connected to the pump outlet. A bypass valve may be installed in the fuel feed system to allow the engine-driven pump to pull fuel from the tank if the boost pump is not operating.

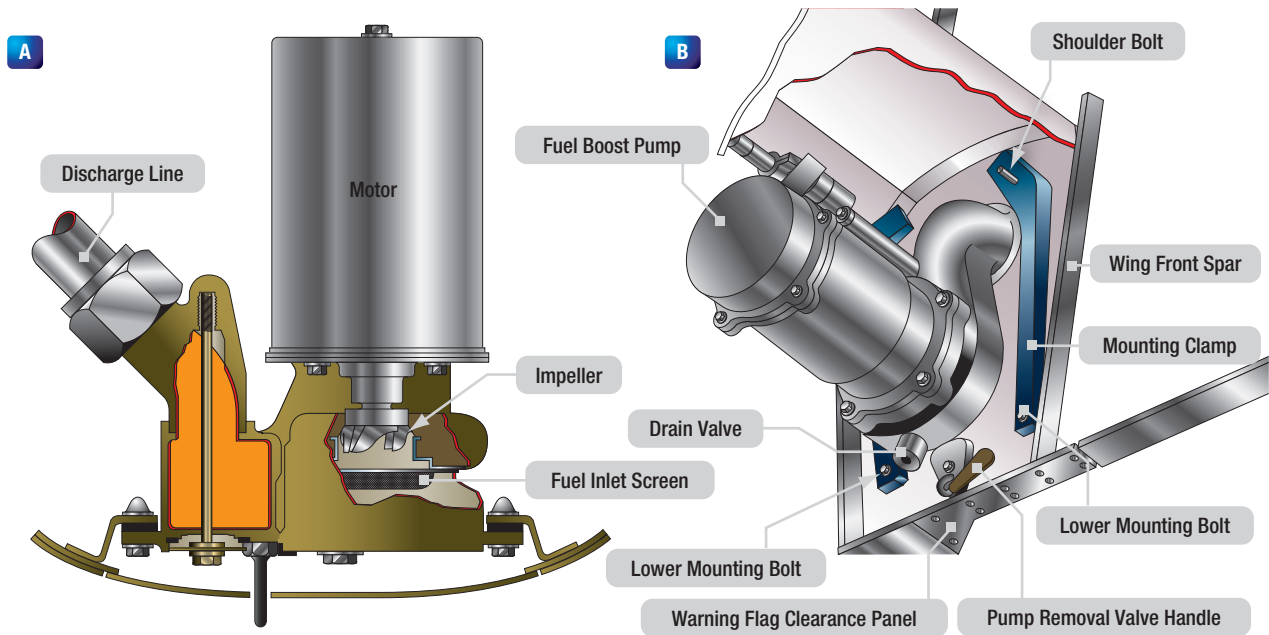


Figure 10-22. A centrifugal fuel boost pump can be submersed in the fuel tank (A) or can be attached to the outside of the tank with inlet and outlet plumbing extending into the tank (B). The pump removal valve handle extends below the warning flag clearance panel to indicate the pump inlet is closed.

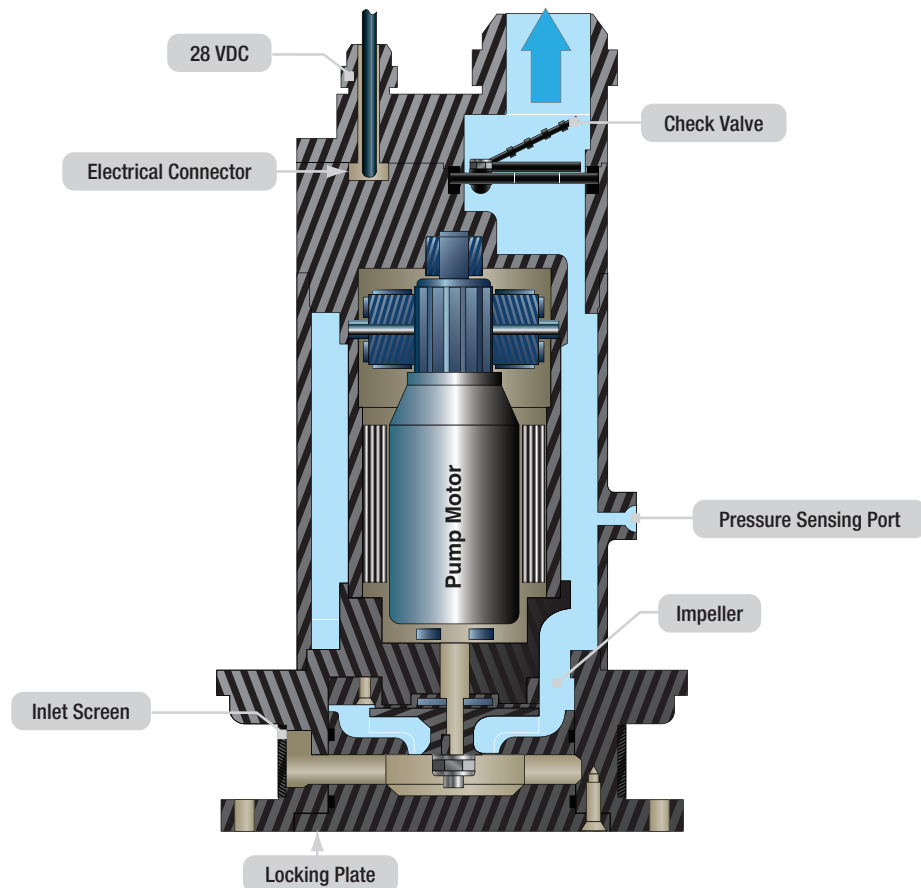


Figure 10-23. The internal workings of a centrifugal fuel boost pump. Fuel is drawn into the center of the impeller through a screen. It is moved to the outside of the case by the impeller and out the fuel outlet tube.

The centrifugal boost pump is used to supply the engine-driven fuel pump, back up the engine driven fuel pump, and transfer fuel from tank to tank if the aircraft is so designed.

Some centrifugal fuel pumps operate at more than one speed, as selected by the pilot, depending on the phase of aircraft operation. Single-speed fuel pumps are also common. Centrifugal fuel pumps located in fuel tanks ensure positive pressure throughout the fuel system regardless of temperature, altitude, or flight attitude thus preventing vapor lock. Submerged pumps have fuel proof covers for the electric motor since the motor is in the fuel. Centrifugal pumps mounted on the outside of the tank do not require this but have some sort of inlet that is located in the fuel. This can be a tube in which a shutoff valve is located so the pump can be changed without draining the tank. The inlet of both types of centrifugal pump is covered with a screen to prevent the ingestion of foreign matter. (*Figure 10-24*)

EJECTOR PUMPS

Fuel tanks with in-tank fuel pumps, such as centrifugal pumps, are constructed to maintain a fuel supply to the pump inlet at all times. This ensures that the pump does not cavitate and that the pump is cooled by the fuel. The section of the fuel tank dedicated for the pump installation may be partitioned off with baffles that contain check valves, also known as flapper valves. These allow fuel to flow inboard to the pump during maneuvers but does not allow it to flow outboard.

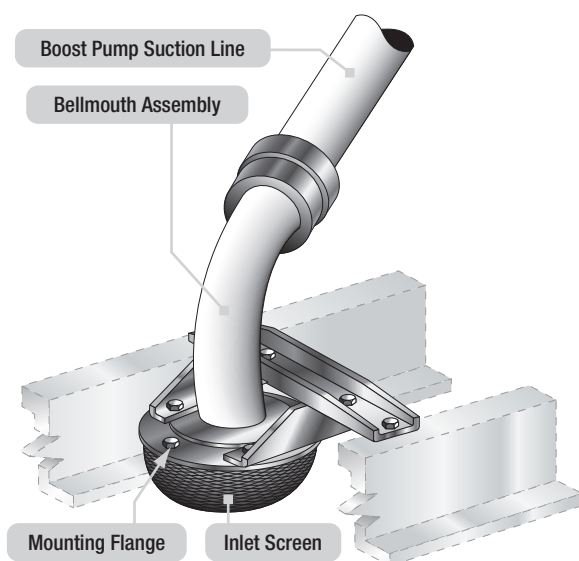


Figure 10-24. A typical fuel boost pump inlet screen installation for a centrifugal pump mounted outside of the bottom of the tank.

Some aircraft use ejector pumps to help ensure that liquid fuel is always at the inlet of the pump. A relatively small diameter line circulates pump outflow back into the section of the tank where the pump is located. The fuel is directed through a venturi that is part of the ejector. As the fuel rushes through the venturi, low pressure is formed. An inlet, or line that originates outside of the tank pump area, allows fuel to be drawn into the ejector assembly where it is pumped into the fuel pump tank section. Together, with baffle check valves, ejector pumps keep a positive head of fuel at the inlet of the pump. (*Figure 10-25*)

FUEL FILTERS

Two main types of fuel cleaning device are utilized on aircraft. Fuel strainers are usually constructed of relatively coarse wire mesh. They are designed to trap large pieces of debris and prevent their passage through the fuel system. Fuel strainers do not inhibit the flow of water. Fuel filters are usually fine mesh. In various applications, they can trap fine sediment that can be only thousands of an inch in diameter and also help trap water. The technician should be aware that the terms 'strainer' and 'filter' are sometimes used interchangeably even though they are not the same thing. Micronic filters are commonly used on turbine-powered aircraft. This is a type of filter that captures extremely fine particles in the range of 10-25 microns. A micron is 1/1000 of a millimeter. (*Figure 10-26*)

Turbine engine fuel control units are extremely close tolerance devices. It is imperative that fuel delivered to them is clean and contaminant free. The use of micronic filters makes this possible. The changeable cellulose filter mesh type shown in *Figure 10-27* can block particles 10-200 microns in size and absorbs water if it is present. The small size of the mesh raises the possibility of the filter being blocked by debris or water. Therefore, a relief valve is included in the filter assembly that bypasses fuel through the unit should pressure build up from blockage. In addition to a fuel filter installed between the fuel tank and the engine driven fuel pump, fuel filters are often used between the engine-driven fuel pump and the fuel metering device (fuel control) on turbine engine aircraft. While these are technically part of the engine fuel system, a common type used on turbine engines is shown in *Figure 10-28*.

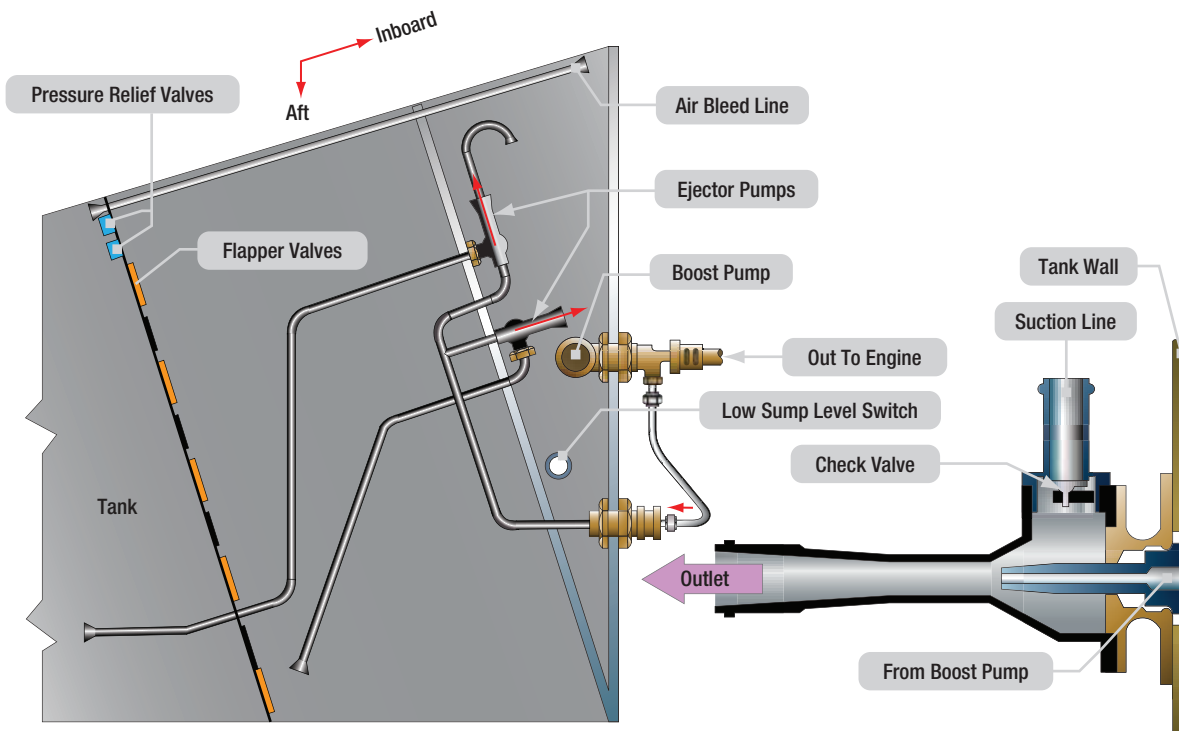


Figure 10-25. An ejector pump uses a venturi to draw fuel into the boost pump sump area of the fuel tank.

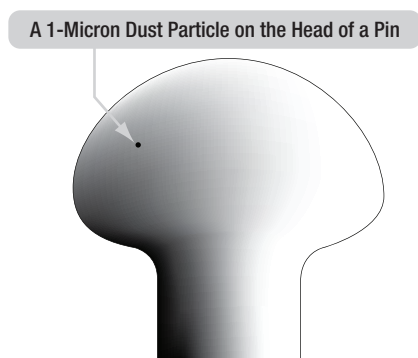


Figure 10-26. Size comparison of 1-micron dust particle and pin head.

It is also a micronic filter. It uses finely meshed disks or wafers stacked on a central core. These filters are able to withstand the higher pressure found in the engine fuel system downstream of the engine-driven pump. (*Figure 10-28*)

Many fuel filter assemblies contain a bypass valve that passes fuel around the filter when it is clogged. Indication of a filter blockage appear in the cockpit through the use of a bypass-activated switch or a pressure differential switch. The bypass valve physically activates a switch that closes the circuit to the annunciator in the first type. The differential pressure type indicator compares the input pressure of the fuel filter to the output pressure. A circuit is completed when a preset difference occurs.

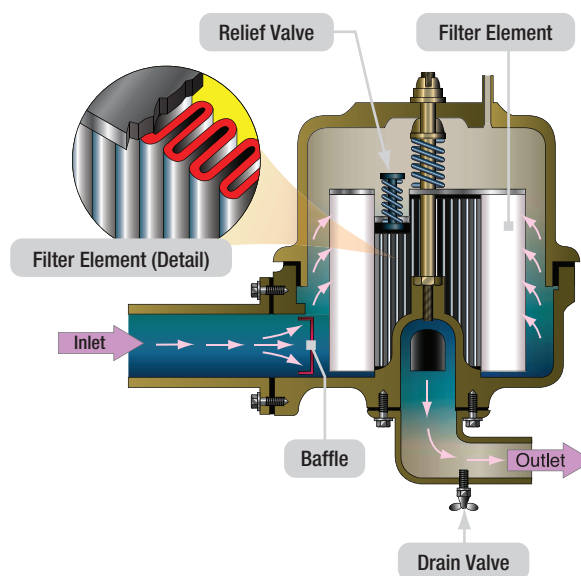


Figure 10-27. A typical micronic fuel filter with changeable cellulose filter element.

Thus, an indicator is illuminated should a blockage cause the bypass to open or the inlet and outlet pressures to vary significantly. Fuel temperature can also be monitored for the possibility of a blockage caused by frozen water. (*Figure 10-29*)

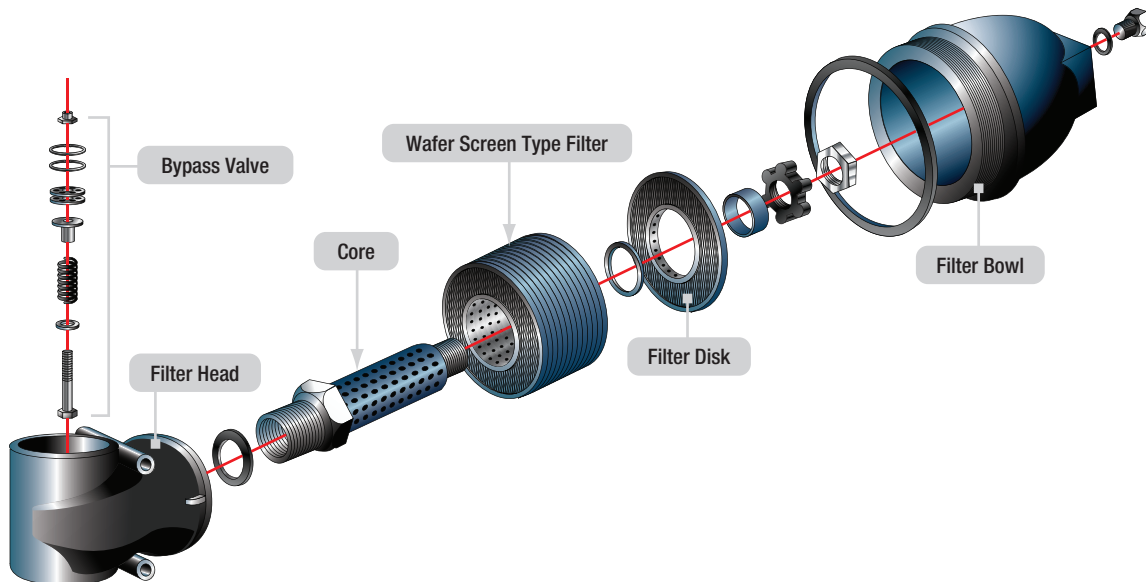


Figure 10-28. A micronic wafer filter uses multiple screen wafers through which fuel must pass to exit the filter through the core. A spring loaded bypass valve in the filter housing unseats when the filter is clogged to continue delivery of fuel.

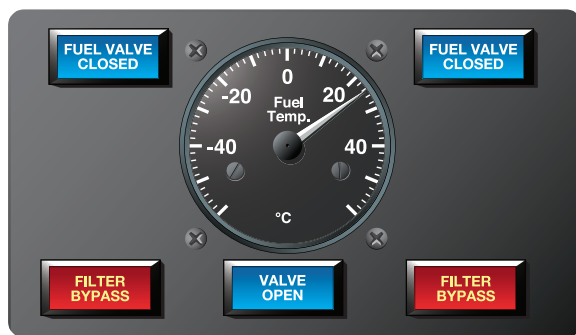


Figure 10-29. A Boeing 737 cockpit fuel panel showing illuminated valve position indicators and fuel filter bypass lights. The fuel temperature in tank No.1 is also indicated.

FUEL INDICATION AND WARNINGS

Fuel indicating systems on jet transport aircraft monitor a variety of parameters, some not normally found on general aviation aircraft. Business jet aircraft share many of these features. True fuel flow indicators for each engine are used as the primary means for monitoring fuel delivery to the engines. A fuel temperature gauge is common as are fuel filter bypass warning lights.

The temperature sensor is usually located in a main fuel tank. The indicator is located on the instrument panel or is displayed on a multifunction display (MFD). These allow the crew to monitor the fuel temperature during high altitude flight in extremely frigid conditions. The fuel filters have bypasses that permit fuel flow around

the filters if clogged. Indicator light(s) illuminate in the cockpit when this occurs. Valve position indicators and various warning lights and annunciations are also used.

Low fuel pressure warning lights are also common on jet transport aircraft. The sensors for these are located in the boost pump outlet line. They give an indication of possible boost pump failure.

FUEL QUANTITY INDICATING SYSTEMS

Fuel quantity gauges are important features on all aircraft. Indications exist for all tanks on a transport category aircraft. Often, these use a capacitance type fuel quantity indication system and a fuel totalizer as discussed below. The location of fuel instrumentation varies depending on the type of cockpit displays utilized on the aircraft.

RATIOMETER TYPE

Electric fuel quantity indicators are common in aircraft. Most of these units operate with direct current (DC) and use variable resistance in a circuit to drive a ratiometer type indicator. The movement of a float in the tank moves a connecting arm to the wiper on a variable resistor in the tank unit. This resistor is wired in series with one of the coils of the ratiometer type fuel gauge in the instrument panel. Changes to the current flowing through the tank unit resistor change the current flowing through one of the coils in the indicator. This

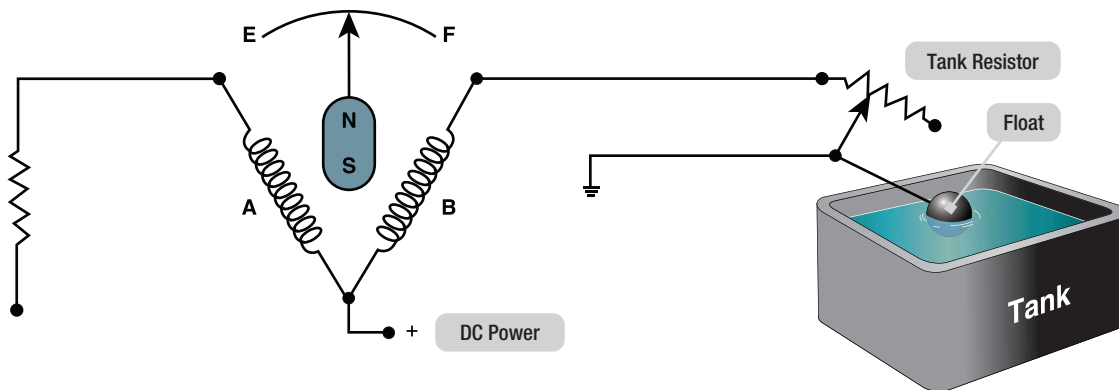


Figure 10-30. A DC electric fuel quantity indicator uses a variable resistor in the tank unit, which is moved by a float arm.

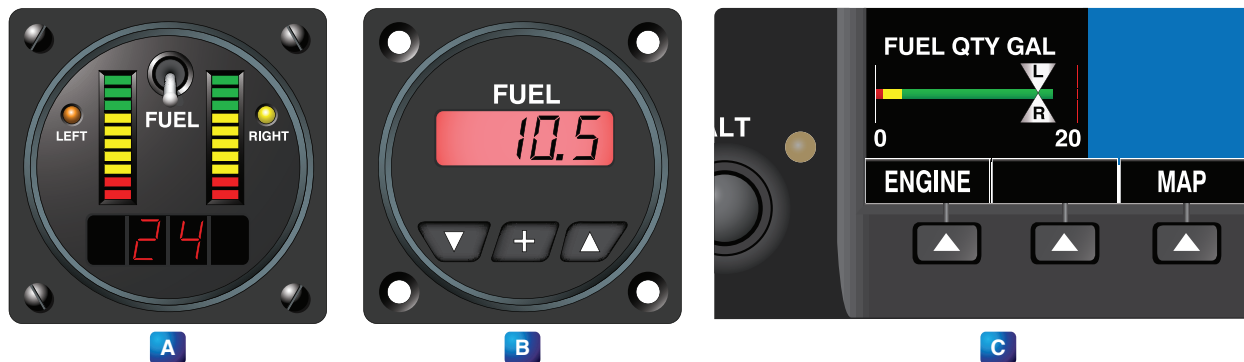


Figure 10-31. Digital fuel quantity gauges that work off of variable resistance from the tank unit are shown in A and B. The fuel quantity indication of a Garmin G-1000 flat screen display is shown in C.

alters the magnetic field in which the indicating pointer pivots. The calibrated dial indicates the corresponding fuel quantity. (*Figure 10-30*)

Digital indicators are available that work with the same variable resistance signal from the tank unit. They convert the variable resistance into a digital display in the cockpit instrument head. (*Figure 10-31*) Fully digital instrumentation systems, such as those found in a glass cockpit aircraft, convert the variable resistance into a digital signal to be processed in a computer and displayed on a flat screen panel.

CAPACITANCE TYPE

Large and high-performance aircraft typically utilize electronic fuel quantity systems. These more costly systems have the advantage of having no moving parts in the tank sending units. Variable capacitance transmitters are installed in the fuel tanks extending from the top to the bottom of each tank in the usable fuel. Several of these tank units, or fuel probes as they are sometimes called, may be installed in a large tank. (*Figure 10-32*)

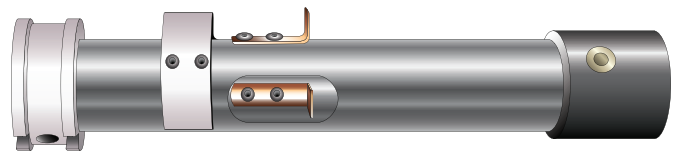


Figure 10-32. A fuel tank transmitter for a capacitance-type fuel quantity indicating system.

They are wired in parallel. As the level of the fuel changes, the capacitance of each unit changes. The capacitance transmitted by all of the probes in a tank is totaled and compared in a bridge circuit by a microchip computer in the tank's digital fuel quantity indicator in the cockpit. As the aircraft maneuvers, some probes are in more fuel than others due to the attitude of the aircraft. The indication remains steady, because the total capacitance transmitted by all of the probes remains the same. A trimmer is used to match the capacitance output with the precalibrated quantity indicator.

A capacitor is a device that stores electricity. The amount it can store depends on three factors: the area of its plates, the distance between the plates, and the

dielectric constant of the material separating the plates. A fuel tank unit contains two concentric plates that are a fixed distance apart. Therefore, the capacitance of a unit can change if the dielectric constant of the material separating the plates varies. The units are open at the top and bottom so they can assume the same level of fuel as is in the tanks. Therefore, the material between the plates is either fuel (if the tank is full), air (if the tank is empty), or some ratio of fuel and air depending on how much fuel remains in the tank. **Figure 10-33** shows a simplified illustration of this construction.

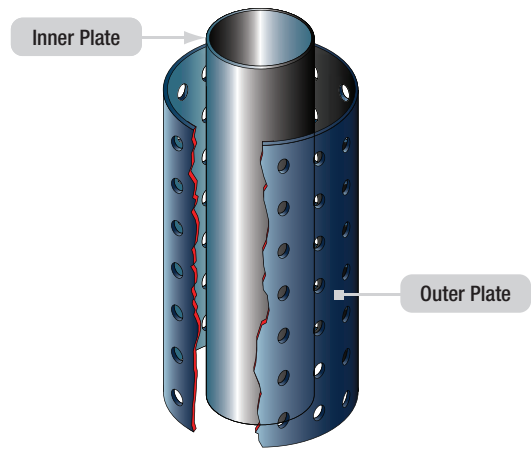


Figure 10-33. The capacitance of tank probes varies in a capacitance- type fuel tank indicator system as the space between the inner and outer plates is filled with varying quantities of fuel and air depending on the amount of fuel in the tank.

The bridge circuit that measures the capacitance of the tank units uses a reference capacitor for comparison. When voltage is induced into the bridge, the capacitive reactance of the tank probes and the reference capacitor can be equal or different. The magnitude of the difference is translated into an indication of the fuel quantity in the tank calibrated in pounds. **Figure 10-34** represents the nature of this comparison bridge circuit.

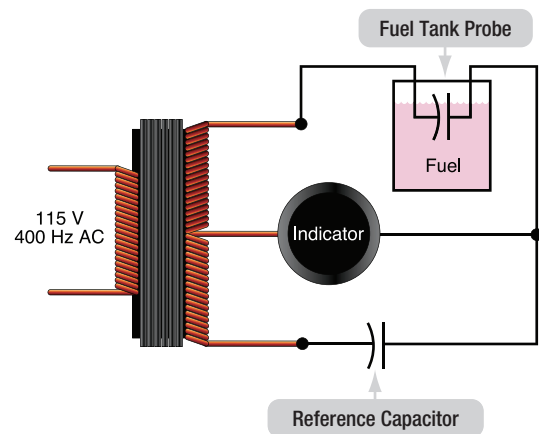


Figure 10-34. A simplified capacitance bridge for a fuel quantity system.

The use of tank unit capacitors, a reference capacitor, and a microchip bridge circuit in the fuel quantity indicators is complicated by the fact that temperature affects the dielectric constant of the fuel. A compensator unit (mounted low in the tank so it is always covered with fuel) is wired into the bridge circuit. It modifies current flow to reflect temperature variations of the fuel, which affect fuel density and thus capacitance of the tank units. (**Figure 10-35**)

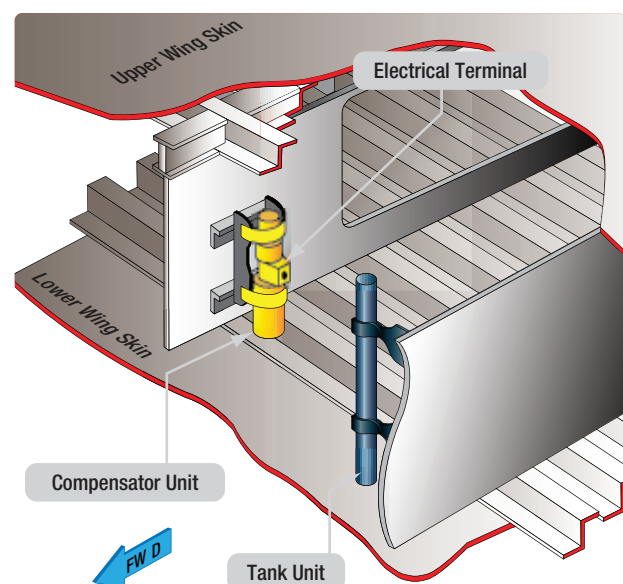


Figure 10-35. A fuel quantity tank unit and compensator unit installed inside a wing tank.

An amplifier is also needed in older systems. The amplitude of the electric signals must be increased to move the servo motor in the analog indicator. Additionally, the dielectric constant of different turbine-engine fuels approved for a particular aircraft may also vary. Calibration is required to overcome this. A fuel summation unit is part of the capacitance type fuel quantity indication system. It is used to add the tank quantities from all indicators. This total aircraft fuel quantity can be used by the crew and by flight management computers for calculating optimum airspeed and engine performance limits for climb, cruise, descent, etc. Capacitance type fuel quantity system test units are available for troubleshooting and ensuring proper functioning and calibration of the indicating system components.

MECHANICAL TYPE

Many aircraft with capacitance type fuel indicating systems also use a mechanical indication system to cross-check fuel quantity indications and to ascertain the amount of fuel on board the aircraft when electrical power is not available. A handful of fuel measuring sticks, or drip sticks, are mounted throughout each tank. When pushed and rotated, the drip stick can be lowered until fuel begins to exit the hole on the bottom of each stick. This is the point at which the top of the stick is equal to the height of the fuel. The sticks have a calibrated scale on them. By adding the indications of all of the drip sticks and converting to pounds or gallons via a chart supplied by the manufacturer, the quantity of the fuel in the tank can be ascertained. (*Figure 10-36*)

FUEL FLOWMETERS

A fuel flowmeter indicates an engine's fuel use in real time. This can be useful to the pilot for ascertaining engine performance and for flight planning calculations. The types of fuel flow meter used on an aircraft depends primarily on the powerplant being used and the associated fuel system. Measuring fuel flow accurately is complicated by the fact that the fuel mass changes with temperature or with the type of fuel used in turbine engines.

Turbine engine aircraft experience the greatest range of fuel density from temperature variation and fuel composition. An elaborate fuel flow device is used on these aircraft. It measures fuel mass for accurate fuel flow indication in the cockpit. The mass flow indicator takes advantage of the direct relationship between fuel mass and viscosity. Fuel is swirled by a cylindrical impeller that rotates at a fixed speed. The outflow deflects a turbine just downstream of the impeller. The turbine is held with calibrated springs. Since the impeller motor swirls the fuel at a fixed rate, any variation of the turbine deflection is caused by the volume and viscosity of the fuel. The viscosity component represents the mass of the fuel. (*Figure 10-37*)

An alternating current (AC) synchro system is part of the mass fuel flowmeter. It is used to position a pointer against the cockpit indicator scale calibrated in pounds per hour. With accurate fuel flow knowledge, numerous calculations can be performed to aid the pilot's situational awareness and flight planning. Most high performance aircraft have a fuel totalizer that electronically calculates

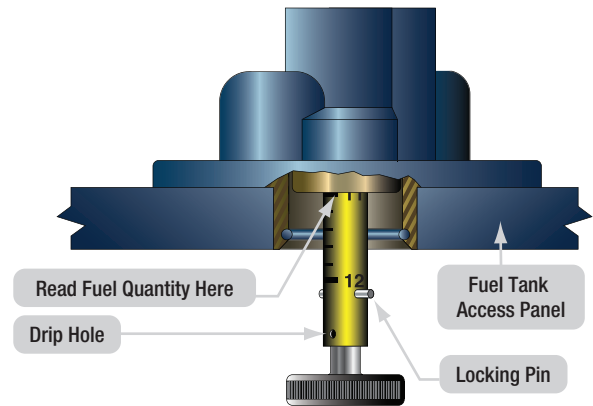


Figure 10-36: A fuel drip stick is lowered from the fuel tank bottom until fuel drips out the hole at the bottom. By reading the calibrated scale and adding readings from all tank drip sticks, a chart can be consulted to arrive at the total fuel quantity on the aircraft by weight or by volume.

and displays information, such as total fuel used, total fuel remaining on board the aircraft, total range and flight time remaining at the present airspeed, rate of fuel consumption, etc.

Relatively new types of fuel flow sensors/transmitters are available in new aircraft and for retrofit to older aircraft. Increasing use of microprocessors and computers on aircraft enable the integration of fuel temperature and other compensating factors to produce highly accurate fuel flow information. Fuel flow sensing with digital output facilitates this with a high degree of reliability.

Thermal dispersion technology provides flow sensing with no moving parts and digital output signals. The sensor consists of two resistance temperature detectors (RTDs). One is a reference RTD that measures the temperature of the fuel. The other is the active RTD. It is heated by an adjacent element to a temperature higher than the fuel.

As the fuel flows, the active element cools proportionally to the fuel flow. The temperature difference between the two RTDs is highest at no flow. The RTDs are connected to an electronic assembly that supplies power to the heater and uses sensing circuitry and a microprocessor to control a constant temperature difference between the heated and unheated RTDs. The electrical current to the heater is proportional to the mass flow of the fuel. As mentioned, the reference RTD is used as a temperature sensor to provide a temperature output and allow for temperature compensation of the flow measurement. (*Figure 10-38*)

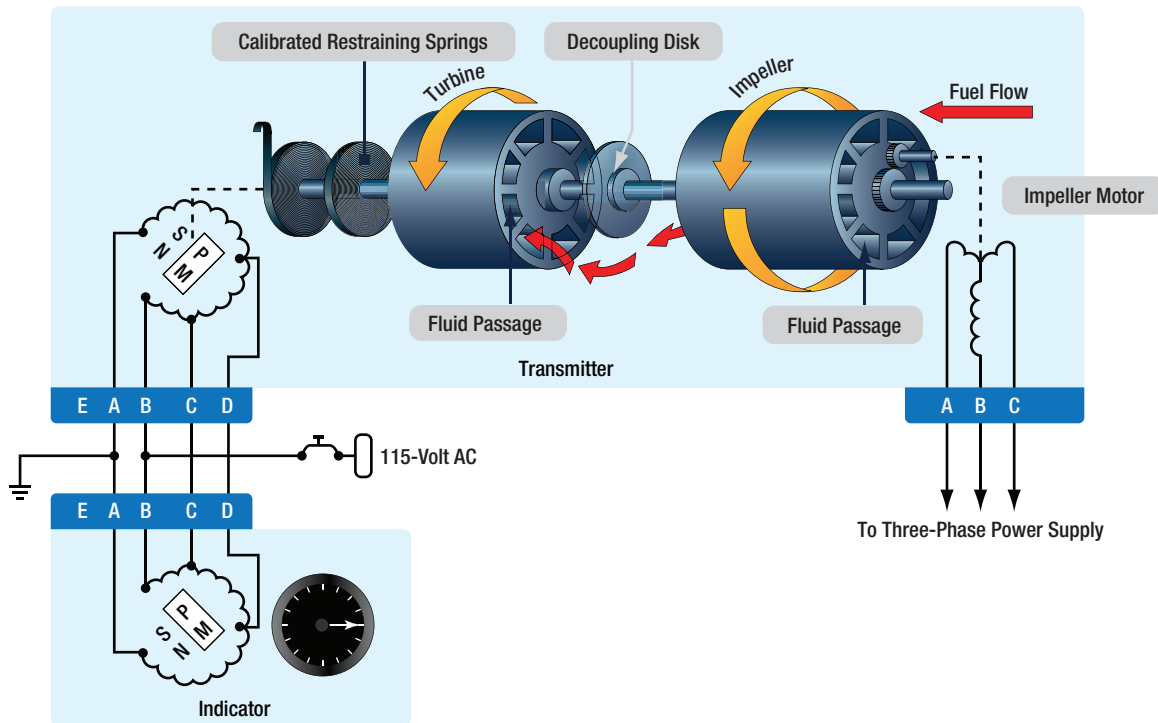


Figure 10-37. A mass flow fuel flow indicating system used on turbine-engine aircraft uses the direct relationship between viscosity and mass to display fuel flow in pounds per hour.



Figure 10-38. Fuel flow sensing units using thermal dispersion technology have no moving parts and output digital signals.

FUEL TEMPERATURE GAUGES

As previously mentioned, monitoring fuel temperature can inform the pilot when fuel temperature approaches that which could cause ice to form in the fuel system, especially at the fuel filter. Many large and high-performance turbine aircraft use a resistance type electric fuel temperature sender in a main fuel tank for this purpose. It can display on a traditional ratiometer gauge or can be input into a computer for processing and digital display. A low fuel temperature can be corrected with the use of a fuel heater if the aircraft is so equipped. Also as mentioned, fuel temperature can be integrated into fuel flow processing calculations. Viscosity differences at varying fuel temperatures that affect fuel flow sensing accuracy can be corrected via microprocessors and computers.

FUEL HEATERS

Turbine powered aircraft operate at high altitude where the temperature is very low. As the fuel in the fuel tanks cools, water in the fuel condenses and freezes. It may form ice crystals in the tank or as the fuel/water solution slows and contacts the cool filter element on its way through fuel filter to the engine(s). The formation of ice on the filter element blocks the flow of fuel through the filter. A valve in the filter unit bypasses unfiltered fuel when this occurs. Fuel heaters are used to warm the fuel so that ice does not form. These heat exchanger units also heat the fuel sufficiently to melt any ice that has already formed.

The most common types of fuel heaters are air/ fuel heaters and oil/fuel heaters. An air/fuel heater uses warm compressor bleed air to heat the fuel. An oil/ fuel exchanger heats the fuel with hot engine oil. This latter type is often referred to as a fuel-cooled oil cooler (FCOC). It not only heats the fuel but also cools the engine oil. (*Figure 10-39*)

Fuel heaters often operate intermittently as needed. A switch in the cockpit can direct the hot air or oil through the unit or block it. The flight crew uses the information supplied by the filter bypass indicating lights and fuel



Figure 10-39. Jet transport aircraft fly at high altitudes where temperatures can reach -45°C . Most have fuel heaters somewhere in the fuel system to help prevent fuel icing. This fuel-cooled oil cooler on an RB211 turbofan engine simultaneously heats the fuel while cooling the oil.

temperature gauge as seen in **Figure 10-29** to know when to heat the fuel. Fuel heaters can also be automatic. A built-in thermostatic device opens or closes a valve that permits the hot air or hot oil to flow into the unit to warm the fuel. (**Figure 10-40**)

NOTE: Some aircraft have a hydraulic fluid cooler in one of the aircraft fuel tanks. The fluid helps warm the fuel as it cools in this type of full-time heat exchanger.

FUEL PRESSURE GAUGES

Monitoring fuel pressure can give the pilot early warning of a fuel system related malfunction. Verification that the fuel system is delivering fuel to the fuel metering device can be critical. Simple light reciprocating-engine

aircraft typically utilize a direct reading Bourdon tube pressure gauge. It is connected into the fuel inlet of the fuel metering device with a line extending to the back of the gauge in the cockpit instrument panel. A more complex aircraft may have a sensor with a transducer located at the fuel inlet to the metering device that sends electrical signals to a cockpit gauge. (**Figure 10-41**)

In aircraft equipped with an auxiliary pump for starting and to backup the engine-driven pump, the fuel pressure gauge indicates the auxiliary pump pressure until the engine is started. When the auxiliary pump is switched off, the gauge indicates the pressure developed by the engine driven pump. Modern aircraft may use a variety of sensors including solid state types and those with digital output signals or signals that are converted to digital output. These can be processed in the instrument gauge microprocessor, if so equipped, or in a computer and sent to the display unit. (**Figure 10-42**)

PRESSURE WARNING SIGNAL

On aircraft of any size, visual and audible warning devices are used in conjunction with gauge indications to draw the pilot's attention to certain conditions. Fuel pressure is an important parameter that merits the use of a warning signal when it falls outside of the normal operating range. Low fuel pressure warning lights can be illuminated through the use of simple pressure sensing switches. (**Figure 10-43**) The contacts of the switch will close when fuel pressure against the diaphragm is insufficient to hold them open. This allows current to flow to the annunciator or warning light in the cockpit.

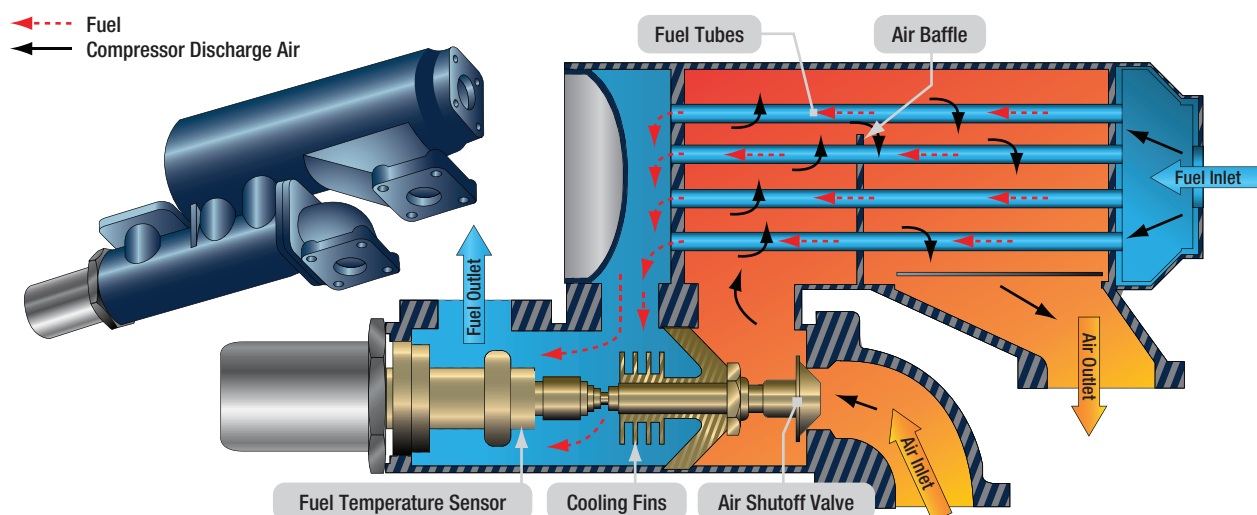


Figure 10-40. An air-fuel heat exchanger uses engine compressor bleed air to warm the fuel on many turbine engine powered aircraft.

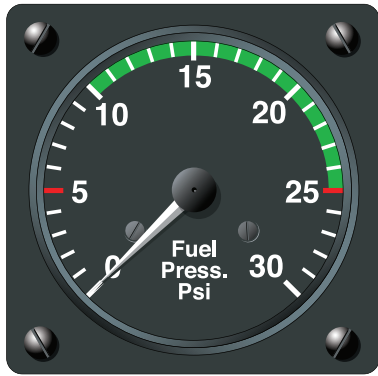


Figure 10-41. A typical fuel gauge that uses a signal from a sensing transducer to display fuel inlet pressure at the metering device.

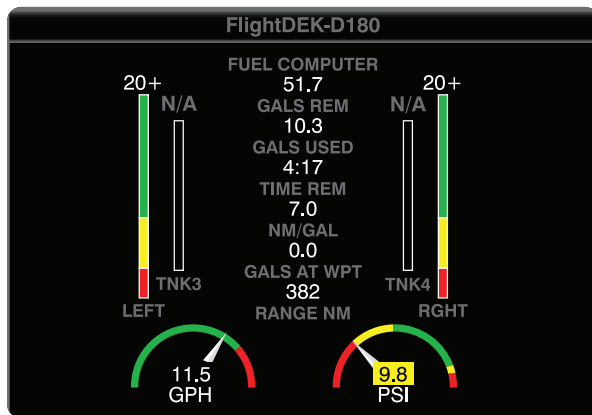


Figure 10-42. An electronic display of fuel parameters, including fuel pressure.

Most turbine-powered aircraft utilize a low pressure warning switch at the outlet of each fuel boost pump. The annunciator for each is typically positioned adjacent to the boost pump ON/OFF switch on the fuel panel in the cockpit. (*Figure 10-44*)

VALVE-IN-TRANSIT INDICATOR LIGHTS

Aircraft with multiple fuel tanks use valves and pumps to move fuel and to have it flow to desired locations, such as the engines, a certain tank, or overboard during fuel jettison. The functioning of the valves in the fuel system is critical. Some aircraft indicate to the crew when the valve is opening or closing with the use of valve-in-transit lights. Contacts in the valve control the lights that go out when the valve is fully open or when it is fully closed. Alternately, annunciator lights that show the valve position as OPEN or CLOSED are also used. Valve-in-transit and valve position indicators, or lights, are located on the fuel panel in the cockpit adjacent to the valve ON/OFF switches. (*Figure 10-45*)

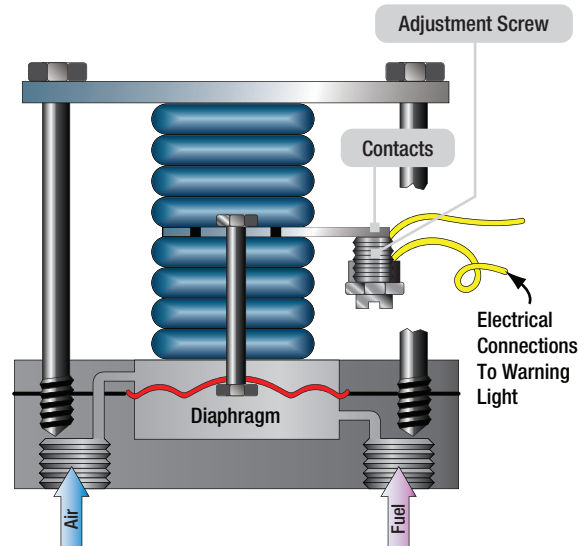


Figure 10-43. A fuel pressure warning signal is controlled by a switch that closes when fuel pressure is low.



Figure 10-44. A transport category aircraft fuel panel with low pressure warning lights for each fuel boost pump.

Sometimes the switch mechanism has the annunciator light built into it. Digital display systems graphically depict valve positions on screen.

REFUELING AND DEFUELING

Single-point pressure fueling at a fueling station accessible by ramp refueling trucks allows all aircraft fuel tanks to be filled with one connection of the fuel hose. Leading and trailing edge wing locations are common for these stations. *Figure 10-46* shows an airliner fueling station with the fueling rig attached.

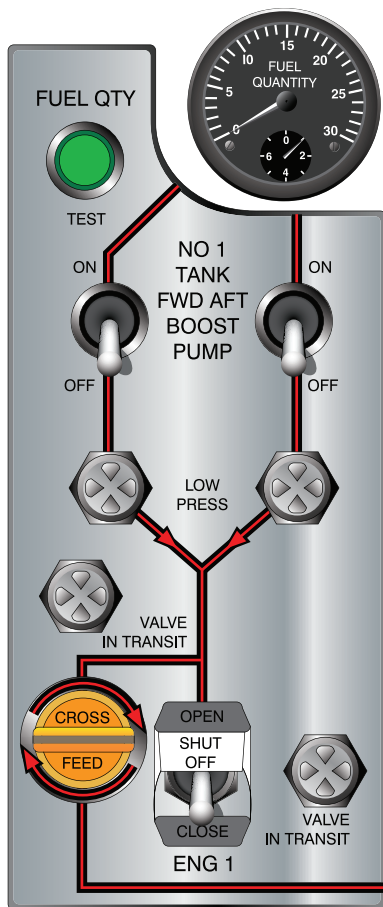


Figure 10-45. Valve-in-transit lights are used on this section of a transport category aircraft fuel panel. Low boost pump pressure lights that look the same are also on the panel.

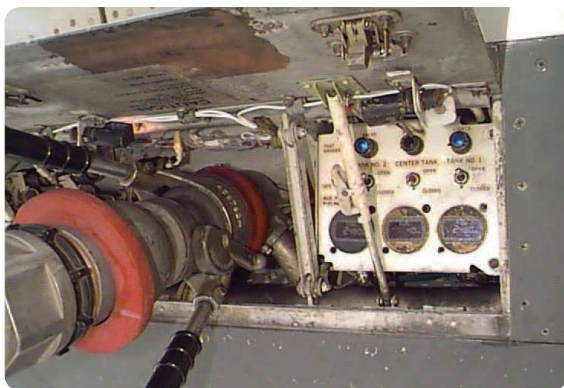


Figure 10-46. A central pressure refueling station on a transport category aircraft allows all fuel tanks to be filled from one position.

To fuel with pressure refueling, a hose nozzle is attached at the fueling station and valves to the tanks required to be filled are opened. These valves are called fueling valves or refueling valves depending upon the manufacturer's preference. Various automatic shutoff systems have been designed to close tank fueling valves

before the tanks overfill or are damaged. Gauges on the refueling panel allow refueling personnel to monitor progress. Occasionally, defueling the aircraft is required for an inspection or repair. The same fueling station is used, and the hose from the fuel truck is connected to same receptacle used to fuel the aircraft. To allow fuel to exit the aircraft, a defueling valve is opened. Fuel can either be pumped out of the aircraft using the boost pumps located in the tanks that need to be emptied, or the pump in the refueling truck can be used to draw the fuel out of the tanks.

Control over the operation is maintained by positioning various shutoff and crossfeed valves, as well as the defuel valve so that fuel travels from the tank to the fueling station and into the truck. Maintenance technicians are often asked to fuel or defuel aircraft. Fueling procedure can vary from aircraft to aircraft. Tanks may need to be fueled in a prescribed sequence to prevent structural damage to the airframe. The proper procedure should be confirmed before fueling an unfamiliar aircraft.

FUELING

Always fuel aircraft outside, not in a hangar where fuel vapors may accumulate and increase the risk and severity of an accident. Generally, there are two types of fueling process: over the wing refueling and pressure refueling.

OVER THE WING REFUELING

Over the wing refueling is accomplished by opening the fuel tank cap on the upper surface of the wing or fuselage, if equipped with fuselage tanks. The fueling nozzle is carefully inserted into the fill opening and fuel is pumped into the tank. This process is similar to the process used to refuel an automobile gas tank. When finished, the cap is secured and subsequent tanks are opened and refilled until the aircraft has the desired fuel load on board. Clean the area adjacent to the fill port when refueling over the wing. Ensure the fuel nozzle is also clean. Aviation fuel nozzles are equipped with static bonding wires that must be attached to the aircraft before the fuel cap is opened. (*Figure 10-47*)

Open the cap only when ready to dispense the fuel. Insert the nozzle into the opening with care. The aircraft structure is much more delicate than the fuel nozzle, which could easily damage the aircraft. Do not insert the neck of the nozzle deeply enough to hit bottom. This could dent the tank, or the aircraft skin, if it is an integral

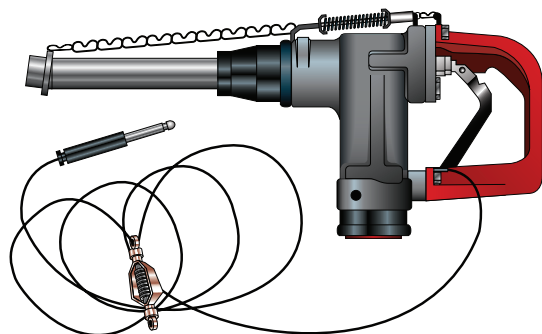


Figure 10-47. An AVGAS fueling nozzle with static bonding grounding wire.

tank. Exercise caution to avoid damage to the surface of the airframe by the heavy fuel hose. Lay the hose over a shoulder or use a refueling mat to protect the paint.

PRESSURE REFUELING

When pressure refueling, the aircraft receptacle is part of a fueling valve assembly. When the fueling nozzle is properly connected and locked, a plunger unlocks the aircraft valve so fuel can be pumped through it. Normally, all tanks can be fueled from a single point. Valves in the aircraft fuel system are controlled at the fueling station to direct the fuel into the proper tank. (*Figure 10-48*) Ensure that the pressure developed by the refueling pump is correct for the aircraft before pumping fuel. Note that, while similar, pressure fueling panels and their operation are different on different aircraft. Refueling personnel should be guided through the correct use of each panel. Do not guess at how the panel and associated valves operate.

When fueling from a fuel truck, precautions should be taken. If the truck is not in continuous service, all sumps should be drained before moving the truck, and the fuel should be visually inspected to be sure it is bright and clean. Turbine fuel should be allowed to settle for a few hours if the fuel truck tank has recently been filled or the truck has been jostled, such as when driven over a bumpy service road at the airport. Properly maneuver the fuel truck into position for refueling. The aircraft should be approached slowly. The truck should be parked parallel to the wings and in front of the fuselage if possible. Avoid backing toward the aircraft. Set the parking brake and chock the wheels.

Connect a static bonding cable from the truck to the aircraft. This cable is typically stored on a reel mounted on the truck. There are other miscellaneous good



Figure 10-48. This panel at the pressure refueling station has valve position switches and quantity gauges to be used during refueling. Valve open position lights are adjacent to the switches for each tank.

practices that should be employed when refueling an aircraft. A ladder should be used if the refuel point is not accessible while standing on the ground. Climbing on an expensive aircraft to access the fueling ports is possible but does not give the stability of a ladder and may not be appreciated by the aircraft owner. If it is necessary to walk on the wings of the aircraft, do so only in designated areas, which are safe.

Filler nozzles should be treated as the important tools that they are. They should not be dropped or dragged across the apron. Most have attached dust caps that should be removed only for the actual fueling process and then immediately replaced. Nozzles should be clean to avoid contamination of the fuel. They should not leak and should be repaired at the earliest sign of leak or malfunction. Keep the fueling nozzle in constant contact with the filler neck spout when fueling. Never leave the nozzle in the fill spout unattended. When fueling is complete, always double check the security of all fuel caps and ensure that bonding wires have been removed and stowed.

DEFUELING

Removing the fuel contained in aircraft fuel tanks is sometimes required. This can occur for maintenance, inspection, or due to contamination. Occasionally, a change in flight plan may require defueling. Safety procedures for defueling are the same as those for fueling. Always defuel outside. Fire extinguishers should be on hand. Bonding cables should be attached to guard against static electricity buildup.

Defueling should be performed by experienced personnel and inexperienced personnel must be checked out before doing so without assistance.

Remember that there may be a sequence in defueling an aircraft's fuel tanks just as there is when fueling to avoid structural damage. Consult the manufacturer's maintenance/operations manual(s) if in doubt. Pressure fueled aircraft normally defuel through the pressure fueling port. The aircraft's in-tank boost pumps can be used to pump the fuel out. The pump on a fuel truck can also be used to draw fuel out. These tanks can also be drained through the tank sump drains, but the large size of the tanks usually makes this impractical. Aircraft fueled over the wing are normally drained through the tank sump drains. Follow the manufacturer's procedure for defueling the aircraft.

What to do with the fuel coming out of a tank depends on a few factors. First, if the tank is being drained due to fuel contamination or suspected contamination, it should not be mixed with any other fuel. It should be stored in a separate container from good fuel, treated if possible, or disposed of properly. Take measures to ensure that contaminated fuel is never placed on board an aircraft or mixed with good fuel. Second, the manufacturer may have requirements for good fuel that has been defueled from an aircraft, specifying whether it can be reused and the type of storage container in which it must be stored. Above all, fuel removed from an aircraft must not be mixed with any other type of fuel.

Good fuel removed from an aircraft must be handled with all precautions used when handling any fuel. It must only be put into clean tanks and efforts must be made to keep it clean. It may be put back in the aircraft or another aircraft if the manufacturer allows. Large aircraft can often transfer fuel from a tank requiring maintenance to another tank to avoid the defueling process.

FIRE HAZARDS WHEN FUELING OR DEFUELING

Due to the combustible nature of AVGAS and turbine engine fuel, the potential for fire while fueling and defueling aircraft must be addressed. Always fuel and defuel outside, not in a hangar that serves as an enclosed area for vapors to build up to a combustible level.

Clothing worn by refueling personnel should not promote static electricity buildup. Synthetics, such as nylon, should be avoided. Cotton has proved to be safe for fuel handling attire.

As previously mentioned, the most controllable of the three ingredients required for fire is the source of ignition. It is absolutely necessary to prevent a source of ignition anywhere near the aircraft during fueling or refueling. Any open flame, such as a lit cigarette, must be extinguished. Operation of any electrical devices must be avoided. Radio and radar use is prohibited. It is important to note that fuel vapors proliferate well beyond the actual fuel tank opening and a simple spark, even one caused by static electricity, could be enough for ignition. Any potential for sparks must be nullified.

Spilled fuel poses an additional fire hazard. A thin layer of fuel vaporizes quickly. Small spills should be wiped up immediately. Larger spills can be flooded with water to dissipate the fuel and the potential for ignition. Do not sweep fuel that has spilled onto the ramp.

Class B fire extinguishers need to be charged and accessible nearby during the fueling and defueling processes. Fueling personnel must know exactly where they are and how to use them. In case of an emergency, the fuel truck, if used, may need to be quickly driven away from the area. For this reason alone, it should be positioned correctly on the ramp relative to the aircraft.

LONGITUDINAL BALANCE SYSTEMS

As large modern aircraft burn off fuel in flight, the center of gravity of the aircraft can change. This may cause the nose of the aircraft to pitch down. Use of elevator trim to compensate for the nose-down attitude is possible. However, when the amount of trim is increased, the amount of drag is also increased. The aircraft is designed to fly most efficiently without trim. Transferring fuel to maintain the desired aircraft center of gravity so that trim is not required for cruise flight is an option on some aircraft. Fuel tanks designated as trim tanks are used to carry the fuel that is transferred to maintain longitudinal balance.

To relieve the flight crew of the task of continually monitoring and transferring fuel to maintain a trim-free center of gravity, computerized center of gravity control systems are used. Fuel flow, fuel usage and tank volumes are monitored by the system computer so that valves and pumps are automatically operated to transfer fuel to the proper tank locations to maintain the desired CG. Pitch attitude and stabilizer position are also monitored by the

CGCC (center of gravity control computer). The CG is generally given as a range of the percentage of the mean aerodynamic chord (MAC).

The flight crew initiates system operation by entering variables such as passenger and cargo loads before flight. Once airborne, the As fuel is consumed, fuel management system and tank volume inputs are used by the CG control system to calculate and perform the fuel transfers that keep the CG within the desired range. Flight deck display of the aircraft weight and CG location allow crew surveillance of the automated transfers. Should the computerized CG control system fail, the system automatically switches to an alternate mode. It may also switch to a fault mode which automatically empties the trim tank.

FUEL SYSTEM REPAIR

The integrity of an aircraft fuel system is critical and should not be compromised. Any evidence of malfunction or leak should be addressed before the aircraft is released for flight. The danger of fire, explosion, or fuel starvation in flight makes it imperative that fuel system irregularities be given top priority. Each manufacturer's maintenance and operation instructions must be used to guide the technician in maintaining the fuel system in airworthy condition. Follow the manufacturer's instructions at all times. Component manufacturers and STC holder instructions should be used when applicable. Some general instructions for fuel system maintenance and repair are given in the following sections.

TROUBLESHOOT THE FUEL SYSTEM

Knowledge of the fuel system and how it operates is essential when troubleshooting. Manufacturers produce diagrams and descriptions in their maintenance manuals to aid the technician. Study these for insight. Many manuals have troubleshooting charts or flow diagrams that can be followed. As with all troubleshooting, a logical sequence of steps to narrow the problem to a specific component or location should be followed. Defects within the system can often be located by tracing the fuel flow from the tank through the system to the engine. Each component must be functioning as designed and the cause of the defect symptom must be ruled out sequentially.

LOCATION OF LEAKS AND DEFECTS

Close visual inspection is required whenever a leak or defect is suspected in a fuel system. Leaks can often be traced to the connection point of two fuel lines or a fuel line and a component. Occasionally, the component itself may have an internal leak. Fuel leaks also occur in fuel tanks and are discussed below. Leaking fuel produces a mark where it travels. It can also cause a stronger than normal odor. Gasoline may collect enough of its dye for it to be visible or an area clean of dirt may form. Jet fuel is difficult to detect at first, but it has a slow evaporation rate. Dirt and dust eventually settle into it, which makes it more visible.

When fuel leaks into an area where the vapors can collect, the leak must be repaired before flight due to the potential for fire or explosion. Repair could be deferred for external leaks that are not in danger of being ignited. However, the source of the leak should be determined and monitored to ensure it does not become worse. Follow the aircraft manufacturer's instructions on the repair of fuel leaks and the requirements that need to be met for airworthiness. Detailed visual inspection can often reveal a defect.

FUEL LEAK CLASSIFICATION

Four basic classifications are used to describe aircraft fuel leaks: stain, seep, heavy seep, and running leak. (*Figure 10-49*) In 30 minutes, the surface area of the collected fuel from a leak is a certain size. This is used as the classification standard. When the area is less than $\frac{3}{4}$ -inch in diameter, the leak is said to be a stain. From $\frac{3}{4}$ to 1Ω inches in diameter, the leak is classified as a seep. Heavy seeps form an area from 1Ω inches to 4 inches in diameter. Running leaks pool and actually drip from the aircraft. They may follow the contour of the aircraft for a long distance.

REPLACEMENT OF GASKETS, SEALS AND PACKINGS

A leak can often be repaired by replacing a gasket or seal. When this occurs or a component is replaced or reassembled after a maintenance operation, a new gasket, seal, or packing must be installed. Do not use the old one(s). Always be sure to use the correct replacement as identified by part number. Also, most gaskets, seals, and packings have a limited shelf life. They should be used only if they are within the service life stamped on the package.

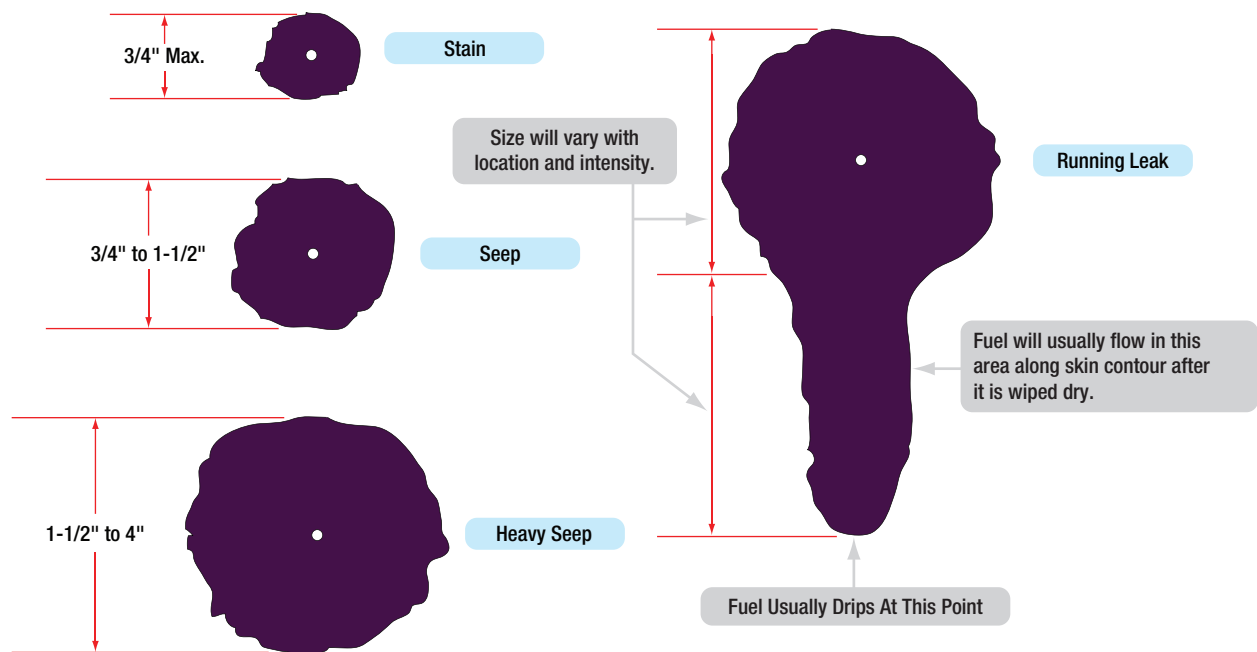


Figure 10-49. The surface area of collected fuel from a leak is used to classify the leak into the categories shown.

Remove the entire old gasket completely, and clean all mating surfaces. Clean surfaces and grooves allow a tight seal. Inspect new gaskets and seals for any flaws. Follow the manufacturer's instructions for replacement, including cleaning procedures and any sealing compound that you may need to apply during replacement. Torque assembly bolts evenly so as to provide even pressure and prevent pinching.

FUEL TANK REPAIR

Whether rigid removable, bladder type, or integral, all fuel tanks have the potential to develop leaks. Repair a tank according to the manufacturer's instructions. Some general notes for repair of each tank type follow. Note that at the time a tank is repaired, a thorough inspection should be made. Corrosion, such as that caused by water and microbes, should be identified and treated at this time, even if it is not the cause of the leak.

INTEGRAL TANKS

Occasionally, an integral tank develops a leak at an access panel. This can often be repaired by transferring fuel to another tank so the panel can be removed and the seal replaced. Use of the proper sealing compound and bolt torque are required. Other integral fuel tank leaks can be more challenging and time consuming to repair. They occur when the sealant used to seal the tank seams loses its integrity. To repair, fuel needs to be transferred or defueled out of the tank. The technician must enter

large tanks on transport category aircraft. Preparing the tank for safe entry requires a series of steps outlined by the aircraft manufacturer. These include drying the tank and venting it of dangerous vapors. The tank is then tested with a combustible gas indicator to be certain it can be entered safely. Clothing that does not cause static electricity and a respirator is worn. An observer is stationed outside of the tank to assist the technician in the tank. (*Figure 10-50*)

A continuous flow of ventilating air is made to flow through the tank. A checklist for fuel tank preparation for entry taken from a transport category maintenance manual is shown in *Figure 10-51*. The details of the procedures are also given in the manual.

Once the location of the leak is determined, the tank sealant is removed and new sealant is applied. Remove old sealant with a nonmetallic scraper. Aluminum wool can be used to remove the final traces of the sealant. After cleaning the area with the recommended solvent, apply new sealant as instructed by the manufacturer. Observe cure time and leak checks as recommended before refilling the tank.

BLADDER TANKS

Bladder fuel tanks that develop leaks can also be repaired. Most commonly, they are patched using patch material, adhesive, and methods approved by the

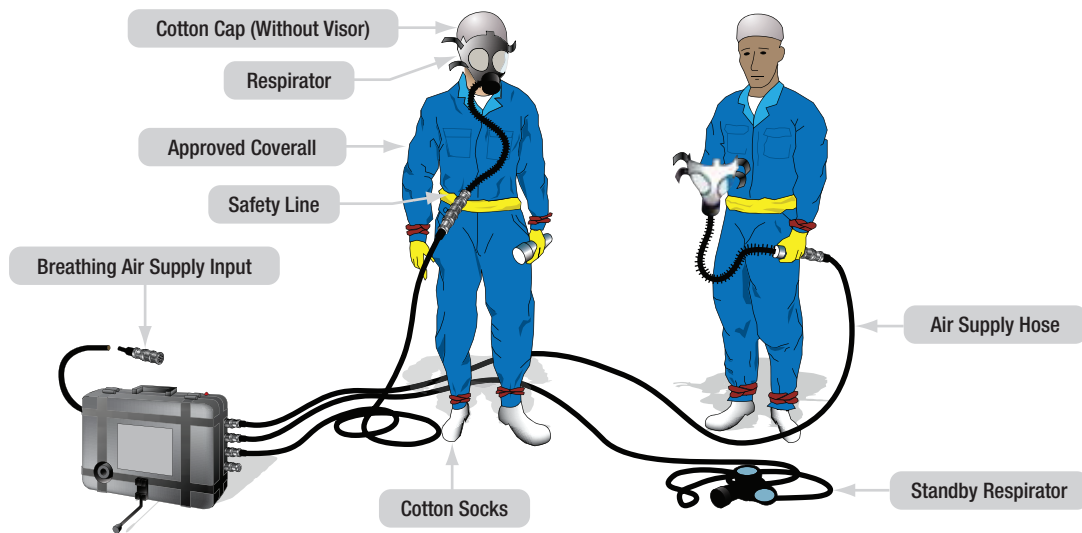


Figure 10-50. Wear a non-static protective suit and respirator when entering an integral fuel tank for inspection or repair.

This checklist must be completed prior to start of wet fuel cell entry and/or at shift change PRIOR to work assignment for the continuation of tank work started by a previous shift.

Wet fuel cell entry location

Area or building: _____ Stall: _____ Airplane: _____ Tank: _____
Shift: _____ Date: _____ Supervisor: _____

- ☐ 1. Airplane and adjacent equipment properly grounded.
- ☐ 2. Area secured and warning signs positioned.
- ☐ 3. Boost pump switches off and circuit breakers pulled and placarded.
- ☐ 4. No power on airplane: battery disconnected, external power cord disconnected from airplane, and external power receptacle placarded.
- ☐ 5. Radio and radar equipment off (see separation distance requirements).
- ☐ 6. Only approved explosion-proof equipment and tools will be used for fuel cell entry (lights, blowers, pressure and test equipment, etc.).
- ☐ 7. Ensure requirements listed on aircraft confined space entry permit are complied with, including appropriate personal protective equipment: OSH class 110 respirator at a minimum, approved coveralls, cotton cap and foot coverings, and eye protection.
- ☐ 8. Trained attendant and confined space logsheet required for all wet fuel cell entries.
- ☐ 9. Aerators checked for cleanliness prior to use.
- ☐ 10. Sponges available for residual fuel mop out.
- ☐ 11. All plugs in use have streamers attached.
- ☐ 12. Mechanical ventilation (venturis or blowers) installed to ventilate all open fuel cells.
Note: Ventilation system must remain in operation at all times while fuel cells are open. If ventilation system fails or any ill effects, such as dizziness, irritation, or excessive odors, are noted, all work shall stop and fuel cells must be evacuated.
- ☐ 13. Shop personnel entering cells and standby observers have current "fuel cell entry" certification cards. Certification requires the following training:
 - Aircraft confined space entry safety
 - Respirator use and maintenance
 - Wet fuel cell entry
- ☐ 14. Fire department notified.

Meter Reading

- ☐ 15. Oxygen reading (%): _____ By: _____
- ☐ 16. Fuel vapor level reading (ppm): _____ By: _____
- ☐ 17. Combustible gas meter (LEL) reading: _____ By (FD): _____

I confirm that all entry requirements were met prior to any entry.

Signature of supervisor or designee

Date

Figure 10-51. Fuel tank checklist entry.

manufacturer. As with soldered tanks, the patch has a required overlap of the damaged area. Damage that penetrates completely through the bladder is repaired with an external, as well as internal, patch. Synthetic bladder tanks have a limited service life. At some point, they seep fuel beyond acceptable limits and need to be replaced. Bladder tanks are usually required to remain wetted with fuel at all times to prevent drying and cracking of the bladder material. Storage of bladder tanks without fuel can be accomplished by coating the tanks with a substance to prevent drying, such as clean engine oil that can be flushed from the tank when ready to return to service. Follow all manufacturer's instructions for the care and repair of these common tanks. It is important to ensure that bladder tanks are correctly secured in place with the proper fasteners when reinstalling them in the aircraft after a repair.

FIRE SAFETY

Fuel vapor, air, and a source of ignition are the requirements for a fuel fire. Whenever working with fuel or a fuel system component, the technician must be vigilant to prevent these elements from coming together to cause a fire or explosion. A source of ignition is often the most controllable. In addition to removing all sources of ignition from the work area, care must be exercised to guard against static electricity. Static electricity can easily ignite fuel vapor, and its potential for igniting fuel vapor may not be as obvious as a flame or an operating electrical device. The action of fuel flowing through a fuel line can cause a static buildup as can many other situations in which one object moves past another. Always assess the work area and take steps to remove any potential static electricity ignition sources.

AVGAS is especially volatile. It vaporizes quickly due to its high vapor pressure and can be ignited very easily. Turbine engine fuel is less volatile but still possesses enormous capacity to ignite. This is especially true if atomized, such as when escaping out of a pressurized fuel hose or in a hot engine compartment on a warm day. Treat all fuels as potential fire hazards in all situations. As was discussed, empty fuel tanks have an extreme potential for ignition or explosion. Although the liquid fuel has been removed, ignitable fuel vapor can remain for a long period of time. Purging the vapor out of any empty fuel tank is an absolute necessity before any repair is initiated.

A fire extinguisher should be on hand during fuel system maintenance or whenever fuel is being handled. A fuel fire can be put out with a typical carbon dioxide (CO₂) fire extinguisher. Aim the extinguisher nozzle at the base of the flame and spray in a sweeping motion to have the agent fall over the flames to displace the oxygen and smother the fire. Dry chemical fire extinguishers rated for fuel can also be used. These leave behind a residue that requires cleanup that can be extensive and expensive. Do not use a water type extinguisher. Fuel is lighter than water and could be spread without being extinguished. Additional precautions used to prevent fire are discussed in the fueling/defueling section of this *Sub-Module*.

FUEL SERVICING AND CONTAMINATION

Maintaining aircraft fuel systems in acceptable condition to deliver clean fuel to the engine(s) is a major safety factor in aviation. Personnel handling fuel or maintaining fuel systems should be properly trained and use best practices to ensure that the fuel, or fuel system, are not the cause of an incident or accident.

FUEL AND FUEL SYSTEM CONTAMINANTS

Continuous vigilance is required when checking aircraft fuel systems for contaminants. Daily draining of strainers and sumps is combined with periodic filter changes and inspections to ensure fuel is contaminant free. Turbine powered engines have highly refined fuel control systems through which flow hundreds of pounds of fuel per hour of operation. Sumping alone is not sufficient. Particles are suspended longer in jet fuel due to its viscosity. Engineers design a series of filters into the fuel system to trap foreign matter. Technicians must supplement these with cautious procedures and thorough visual inspections to accomplish the overall goal of delivering clean fuel to the engines. Keeping a fuel system clean begins with an awareness of the common types of contamination. Water is the most common. Solid particles, surfactants, and microorganisms are also common. However, contamination of fuel with another fuel not intended for use on a particular aircraft is possibly the worst type of contamination.

WATER

Water can be dissolved into fuel or entrained. Entrained water can be detected by a cloudy appearance to the fuel.

Close examination is required. Air in the fuel tends to cause a similar cloudy condition but is near the top of the tank. The cloudiness caused by water in the fuel tends to be more towards the bottom of the tank as the water slowly settles out. As previously discussed, water can enter a fuel system via condensation. The water vapor in the vapor space above the liquid fuel in a fuel tank condenses when the temperature changes. It normally sinks to the bottom of the fuel tank into the sump where it can be drained off before flight. However, time is required for this to happen. (*Figure 10-52*)

On some aircraft, a large amount of fuel needs to be drained before settled water reaches the drain valve. Awareness of this type of sump idiosyncrasy for a particular aircraft is important. The condition of the fuel and recent fueling practices need to be considered and are equally important. If the aircraft has been flown often and filled immediately after flight, there is little reason to suspect water contamination beyond what would be exposed during a routine sumping.

An aircraft that has sat for a long period of time with partially full fuel tanks is a cause of concern. It is possible that water is introduced into the aircraft fuel load during refueling with fuel that already contains water. Any suspected contamination from refueling or the general handling of the aircraft should be investigated. A change in fuel supplier may be required if water continues to be an issue despite efforts are made to keep the aircraft fuel tanks full and sumps drained on a regular basis. Note that fuel below freezing temperature may contain entrained water in ice form that may not settle into the sump until melted. Use of an anti-icing solution in turbine fuel tanks helps prevent filter blockage from water that condenses out of the fuel as ice during flight.

Note that the fuel anti-ice additive level should be monitored so that recommended quantity for the tank capacity is maintained. After repeated fueling, the level can be obscured. A field hand-held test unit can be used to check the amount of anti-ice additive already in a fuel load. (*Figure 10-53*)

Strainers and filters are designed with upward flow exits to have water collect at the bottom of the fuel bowl to be drained off. This should not be overlooked. Entrained water in small quantities that makes it to



Figure 10-52. A sump drain tool used to open and collect fuel and contaminants from the fuel system sumps. Daily sump draining is part of the procedures needed to remove water from fuel that is to be delivered to the engine(s).



Figure 10-53. A hand-held refractometer with digital display measures the amount of fuel anti-ice additive contained in a fuel load.

the engine usually poses no problem. Large amounts of water can disrupt engine operation. Settled water in tanks can cause corrosion. This can be magnified by microorganisms that live in the fuel/water interface. High quantities of water in the fuel can also cause discrepancies in fuel quantity probe indications.

SOLID PARTICLE CONTAMINANTS

Solid particles that do not dissolve in the fuel are common contaminants. Dirt, rust, dust, metal particles, and just about anything that can find its way into an open fuel tank is of concern.

Filter elements are designed to trap these contaminants and some fall into the sump to be drained off. Pieces of debris from the inside of the fuel system may also accumulate, such as broken off sealant, or pieces of filter elements, corrosion, etc. Preventing solid contaminant introduction into the fuel is critical. Whenever the fuel system is open, care must be taken to keep out foreign matter. Lines should be capped immediately. Fuel tank caps should not be left open for any longer than required to refuel the tanks. Clean the area adjacent to wherever the system is opened before it is opened.

Coarse sediments are those visible to the naked eye. Should they pass beyond system filters, they can clog in fuel metering device orifices, sliding valves, and fuel nozzles. Fine sediments cannot actually be seen as individual particles. They may be detected as a haze in the fuel or they may refract light when examining the fuel. Their presence in fuel controls and metering devices is indicated by dark shellac-like marks on sliding surfaces.

The maximum amount of solid particle contamination allowable is much less in turbine engine fuel systems than in reciprocating-engine fuel systems. It is particularly important to regularly replace filter elements and investigate any unusual solid particles that collect therein. The discovery of significant metal particles in a filter could be a sign of a failing component upstream of the filter. A laboratory analysis is possible to determine the nature and possible source of solid contaminants.

SURFACTANTS

Surfactants are liquid chemical contaminants that naturally occur in fuels. They can also be introduced during the refining or handling processes. These surface active agents usually appear as tan to dark brown liquid when they are present in large quantities. They may even have a soapy consistency. Surfactants in small quantities are unavoidable and pose little threat to fuel system functioning. Larger quantities of surfactants do pose problems. In particular, they reduce the surface tension between water and the fuel and tend to cause water and even small particles in the fuel to remain suspended rather than settling into the sumps. Surfactants also tend to collect in filter elements making them less effective. Surfactants are usually in the fuel when it is introduced into the aircraft.

Discovery of either excessive quantities of dirt and water making their way through the system or a sudsy residue in filters and sumps may indicate their presence. The source of fuel should be investigated and avoided if found to contain a high level of these chemicals. As mentioned, slow settling rates of solids and water into sumps is a key indicator that surfactant levels are high in the fuel.

Most quality fuel providers have clay filter elements on their fuel dispensing trucks and in their fixed storage and dispensing systems. These filters, if renewed at

the proper intervals, remove most surfactants through adhesion. Surfactants discovered in the aircraft systems should be traced to the fuel supply source and the use and condition of these filters. (*Figure 10-54*)

MICROORGANISMS

The presence of microorganisms in turbine engine fuels is a critical problem. There are hundreds of varieties of these life forms that live in free water at the junction of the water and fuel in a fuel tank. They form a visible slime that is dark brown, gray, red, or black in color. This microbial growth can multiply rapidly and can cause interference with the proper functioning of filter elements and fuel quantity indicators. Moreover, the slimy water/microbe layer in contact with the fuel tank surface provides a medium for electrolytic corrosion of the tank. (*Figure 10-55*)

Since the microbes live in free water and feed on fuel, the most powerful remedy for their presence is to keep water from accumulating in the fuel. Fuel 100 percent free of water is not practicable. By following best practices for sump draining and filter changes, combined with care of fuel stock tanks used to refuel aircraft, much of the potential for water to accumulate in the aircraft fuel tanks can be mitigated. The addition of biocides to the fuel when refueling also helps by killing organisms that are present.

FOREIGN FUEL CONTAMINATION

Aircraft engines operate effectively only with the proper fuel. Contamination of an aircraft's fuel with fuel not intended for use in that particular aircraft can have



Figure 10-54. Clay filter elements remove surfactants. They are used in the fuel dispensing system before fuel enters the aircraft.



Figure 10-55. This fuel-water sample has microbial growth at the interface of the two liquids.

disastrous consequences. It is the responsibility of all aviators to put forth effort continuously to ensure that only the fuel designed for the operation of the aircraft's engine(s) is put into the fuel tanks. Each fuel tank receptacle or fuel cap area is clearly marked to indicate which fuel is required. (*Figure 10-56*)

If the wrong fuel is put into an aircraft, the situation must be rectified before flight. If discovered before the fuel pump is operated and an engine is started, drain all improperly filled tanks. Flush out the tanks and fuel lines with the correct fuel and then refill the tanks with the proper fuel. However, if discovered after an engine has been started or attempted to be started, the procedure is more in depth. The entire fuel system, including all fuel lines, components, metering device(s) and tanks, must be drained and flushed. If the engines have been operated, a compression test should be accomplished and the combustion chamber and pistons should be borescope inspected.

Engine oil should be drained and all screens and filters examined for any evidence of damage. Once reassembled and the tanks have been filled with the correct fuel, a full engine run-up check should be performed before releasing the aircraft for flight. Contaminated fuel caused by the introduction of small quantities of the wrong type of fuel into an aircraft may not look any different when visually inspected, making a dangerous situation more dangerous. Any person recognizing that this error has occurred must ground the aircraft. The lives of the aircraft occupants are at stake.

DETECTION OF CONTAMINANTS

Visual inspection of fuel should always reveal a clean, bright looking liquid. Fuel should not be opaque, which could be a sign of contamination and demands further



Figure 10-56. All entry points of fuel into the aircraft are marked with the type of fuel to be used. Never introduce any other fuel into the aircraft other than that which is specified.

investigation. As mentioned, the technician must always be aware of the fuel's appearance, as well as when and from what sources refueling has taken place. Any suspicion of contamination must be investigated.

In addition to the detection methods mentioned for each type of contamination above, various field and laboratory tests can be performed on aircraft fuel to expose contamination. A common field test for water contamination is performed by adding a dye that dissolves in water but not fuel to a test sample drawn from the fuel tank. The more water present in the fuel, the greater the dye disperses and colors the sample.

Another common test kit commercially available contains a gray chemical powder that changes color to pink or purple when the contents of a fuel sample contains more than 30 parts per million (ppm) of water. A 15 ppm test is available for turbine engine fuel. (*Figure 10-57*) These levels of water are considered generally unacceptable and not safe for operation of the aircraft. If levels are discovered above these amounts, time for the water to settle out of the fuel should be given or the aircraft should be defueled and refueled with acceptable fuel.

The presence and level of microorganisms in a fuel tank can also be measured with a field device. The test detects the metabolic activity of bacteria, yeast, and molds, including sulfate reducing bacteria, and other anaerobe microorganisms. This could be used to determine the amount of anti-microbial agent to be added to the fuel. The testing unit is shown in *Figure 10-58*.

Bug test kits test fuel specifically for bacteria and fungus. While other types of microorganisms may exist, this semiquantitative test is quick and easy to perform. Treat a fuel sample with the product and match the color of the sample to the chart for an indication of the level of



Figure 10-57. This kit allows periodic testing for water in fuel.

bacteria and fungus present. These are some of the most common types of microorganisms that grow in fuel; if growth levels of fungus and bacteria are acceptable, the fuel could be usable. (*Figure 10-59*)

Fuel trucks and fuel farms may make use of laser contaminant identification technology. All fuel exiting the storage tank going into the servicing hose is passed through the analyzer unit. Laser sensing technology determines the difference between water and solid particle contaminants. When an excessive level of either is detected, the unit automatically shuts off flow to the fueling nozzle. Thus, aircraft are fueled only with



Figure 10-58. A capture solution is put into a 1 liter sample of fuel and shaken. The solution is then put into the analyzer shown to determine the level of microorganisms in the fuel.



Figure 10-59. Fuel bug test kits identify the level of bacteria and fungus present in a fuel load by comparing the color of a treated sample with a color chart.

clean dry fuel. When surfactant filters are combined with contaminant identification technology and microorganism detection, chances of delivering clean fuel to the aircraft engines are good. (*Figure 10-60*)

Before various test kits were developed for use in the field by nonscientific personnel, laboratories provided complete fuel composition analysis to aviators. These services are still available. A sample is sent in a sterilized container to the lab. It can be tested for numerous factors including water, microbial growth, flash point, specific gravity, cetane index (a measure of combustibility and burning characteristics), and more. Tests for microbes involve growing cultures of whatever organisms are present in the fuel. (*Figure 10-61*)

FUEL CONTAMINATION CONTROL

A continuous effort must be put forth by all those in the aviation industry to ensure that each aircraft is fueled only with clean fuel of the correct type. Many contaminants, both soluble and insoluble, can contaminate an aircraft's fuel supply. They can be introduced with the fuel during fueling or the contamination may occur after the fuel



Figure 10-61. Laboratory tests of fuel samples are available.

is on board. Contamination control begins long before the fuel gets pumped into an aircraft fuel tank. Many standard petroleum industry safeguards are in place.

Fuel farm and delivery truck fuel handling practices are designed to control contamination. Various filters, testing, and treatments effectively keep fuel contaminant free or remove various contaminants once discovered. However, the correct clean fuel for an aircraft should never be taken for granted. The condition of all storage tanks and fuel trucks should be monitored. All filter changes and treatments should occur regularly and on time. The fuel supplier should take pride in delivering clean, contaminant free fuel to its customers.

On board aircraft fuel systems must be maintained and serviced according to manufacturer's specifications. Samples from all drains should be taken and inspected on a regular basis. Filters should be changed at the specified intervals. The fuel load should be visually inspected and tested from time to time or when there is a potential contamination issue. Particles discovered in filters should be identified and investigated if needed. Inspection of the fuel system during periodic inspections should be treated with highest concern.

Most importantly, the choice of the correct fuel for an aircraft should never be in question. No one should ever put a fuel into an aircraft fuel tank unless absolutely certain it is the correct fuel for that aircraft and its engine(s). Personnel involved in fuel handling should be properly trained. All potential contamination situations should be investigated and remedied.

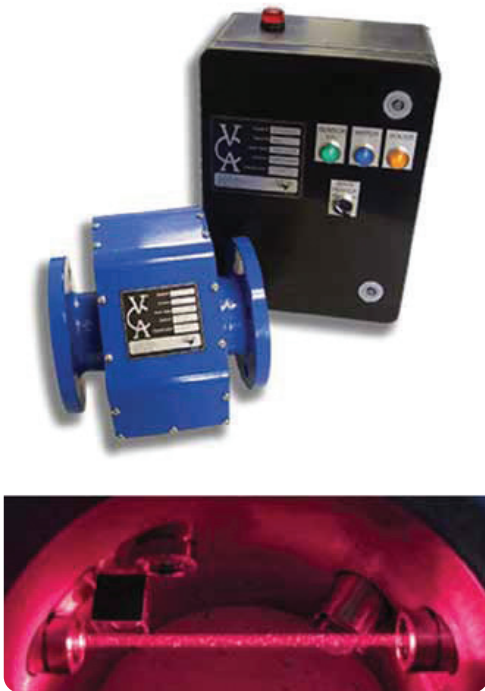


Figure 10-60. This contaminant analyzer is used on fuel supply source outflow, such as that on a refueling truck. Water and solid contaminant levels are detected using laser identification technology. The valve to the fill hose is automatically closed when levels of either are elevated beyond acceptable limits.

Question: 10-1

Each aircraft fuel system must be arranged so that no fuel pump can draw fuel from more than _____ tank(s) at a time.

Question: 10-6

A fuel vent system must be configured to ensure the fuel is vented regardless of the _____ of the aircraft or the quantity of fuel on board.

Question: 10-2

Aircraft fuel tanks must be constructed so as to retain fuel if one of the _____ collapses.

Question: 10-7

Fuel flowing through the fuel system generates _____ that must have a place to flow to ground rather than build up.

Question: 10-3

Name five fuel sub-systems found on jet transport aircraft.

Question: 10-8

The two most common electric motor operated fuel valves are the _____ and the plug type valve.

Question: 10-4

The long, horizontal nature of an integral wing tank requires _____ to keep the fuel from sloshing.

Question: 10-9

The most common type of auxiliary fuel pump used on large and high-performance aircraft is the _____ pump.

Question: 10-5

On what type of aircraft might a bladder fuel tank be found?

Question: 10-10

_____ filters are commonly used on turbine-powered aircraft.

ANSWERS

Answer: 10-1
one.

Answer: 10-6
attitude.

Answer: 10-2
landing gear.

Answer: 10-7
static electricity.

Answer: 10-3
Storage.
Vent.
Distribution.
Supply.
Feed.
Indicating.
Warning.

Answer: 10-8
gate valve.

Answer: 10-4
baffles.

Answer: 10-9
centrifugal.

Answer: 10-5
All sizes of aircraft.

Answer: 10-10
Micronic.

Question: 10-11

Name three types of fuel quantity indicator systems.

Question: 10-16

To fuel with pressure refueling, a hose nozzle is attached at the fueling station and _____ to the tanks required to be filled are opened.

Question: 10-12

The capacitance of a fuel quantity transmitter changes as the _____ of the material separating the plates varies.

Question: 10-17

In addition to the fuel hose, a _____ is connected between the fuel truck and the aircraft when refueling.

Question: 10-13

Measuring _____ accurately is complicated by the fact that the fuel mass changes with temperature or with the type of fuel used in turbine engines.

Question: 10-18

To avoid refueling, sometimes fuel tank maintenance can be performed by _____ fuel into a different tank.

Question: 10-14

The most common types of fuel heaters are air/ fuel heaters and _____ / fuel heaters.

Question: 10-19

A longitudinal balance system is preferred over trimming the aircraft as fuel is burnt off due to _____.

Question: 10-15

Fuel _____ is an important parameter that merits the use of a warning signal when it falls outside of the normal operating range.

Question: 10-20

True or False:

The technician must never enter the tank of a transport category aircraft.

ANSWERS

Answer: 10-11

Ratiometer type.
Capacitance type.
Mechanical type (drip stick).

Answer: 10-16

valves.

Answer: 10-12

dielectric constant.

Answer: 10-17

static bonding cable.

Answer: 10-13

fuel flow.

Answer: 10-18

transferring.

Answer: 10-14

oil.

Answer: 10-19

excess drag caused by trimming.

Answer: 10-15

pressure.

Answer: 10-20

False.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 11

HYDRAULIC POWER (ATA 29)

Knowledge Requirements

11.11 - Hydraulic Power (ATA 29)

System layout;
Hydraulic fluids;
Hydraulic reservoirs and accumulators;
Pressure generation: electric, mechanical, pneumatic; Emergency pressure generation;
Filters;
Pressure Control;
Power distribution;
Indication and warning systems;
Interface with other systems.

3

11.11 - HYDRAULIC POWER

The word "hydraulics" is based on the Greek word for water and originally meant the study of the physical behavior of water at rest and in motion. Today, the meaning has been expanded to include the physical behavior of all liquids, including hydraulic fluid. Hydraulic systems are not new to aviation. Early aircraft had hydraulic brake systems. As aircraft became more sophisticated, newer systems with hydraulic power were developed.

Hydraulic systems in aircraft provide a means for the operation of aircraft components. The operation of landing gear, flaps, flight control surfaces, and brakes is largely accomplished with hydraulic power systems. Hydraulic system complexity varies from small aircraft that require fluid only for manual operation of the wheel brakes to large transport aircraft where complex systems power flight controls, landing gear retraction, nose wheel steering, brakes and more. To achieve the necessary redundancy and reliability, the system may consist of several subsystems. Each subsystem has a power generating device (pump), reservoir, accumulator, heat exchanger, filtering system, etc. System operating pressure may vary from a couple hundred pounds per square inch (psi) in small aircraft and rotorcraft to 5 000 psi in large transports.

Hydraulic systems have many advantages as power sources for operating various aircraft units; they combine the advantages of light weight, ease of installation, simplification of inspection, and minimum maintenance requirements. Hydraulic operations are also almost 100 percent efficient, with only negligible loss due to fluid friction.

SYSTEM LAYOUT

Regardless of its function and design, every hydraulic system has a minimum number of basic components in addition to a means through which the fluid is transmitted. A basic system consists of a pump, reservoir, directional valve, check valve, pressure relieve valve, selector valve, actuator, and filter. (*Figure 11-1*)

OPEN-CENTER HYDRAULIC SYSTEMS

An open-center system is one having fluid flow, but no pressure in the system when the actuating mechanisms are idle. The pump circulates the fluid from the reservoir, through the selector valves, and back to the reservoir. (*Figure 11-2*) The open-center system may employ any number of subsystems, with a selector valve for each subsystem. Unlike the closed center system, the selector valves of the open center system are always connected in series with each other. In this arrangement, the system pressure line goes through each selector valve. Fluid is always allowed free passage through each selector valve and back to the reservoir until one of the selector valves is positioned to operate a mechanism. When one of the selector valves is positioned to operate an actuating device, fluid is directed from the pump through one of the working lines to the actuator. (*Figure 11-2B*)

With the selector valve in this position, the flow of fluid through the valve to the reservoir is blocked. The pressure builds up in the system to overcome the resistance and moves the piston of the actuating cylinder; fluid from the opposite end of the actuator returns to the selector valve and flows back to the reservoir.

Operation of the system following actuation of the component depends on the type of selector valve being used. Several types of selector valves are used in conjunction with the open center system. One type is both manually engaged and manually disengaged. First,

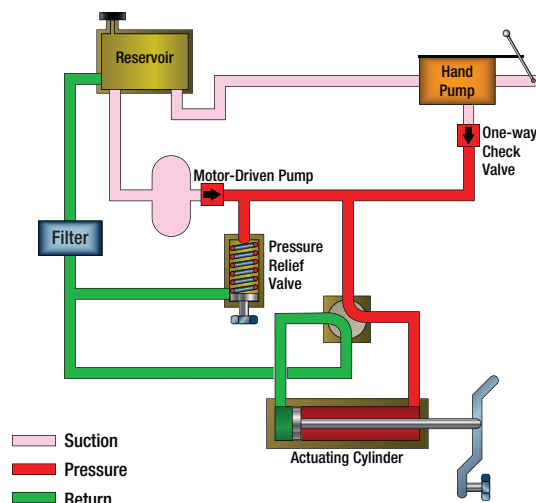


Figure 11-1. Basic hydraulic system.

the valve is manually moved to an operating position. Then, the actuating mechanism reaches the end of its operating cycle, and the pump output continues until the system relief valve relieves the pressure. The relief valve unseats and allows the fluid to flow back to the reservoir. The system pressure remains at the relief valve set pressure until the selector valve is manually returned to the neutral position. This action reopens the open center flow and allows the system pressure to drop to line resistance pressure

The manually engaged and pressure disengaged type of selector valve is similar to the valve previously discussed. When the actuating mechanism reaches the end of its

cycle, the pressure continues to rise to a predetermined pressure. When reached, the valve automatically returns to the neutral position and to open center flow.

CLOSED-CENTER HYDRAULIC SYSTEMS

In the closed-center system, the fluid is under pressure whenever the power pump is operating. The three actuators are arranged in parallel and actuating units B and C are operating at the same time, while actuating unit A is not operating. This system differs from the open-center system in that the selector or directional control valves are arranged in parallel and not in series. The means of controlling pump pressure varies in the closed-center system. If a constant delivery pump is used, the system pressure is regulated by a pressure regulator. A relief valve acts as a backup safety device in case the regulator fails.

If a variable displacement pump is used, system pressure is controlled by the pump's integral pressure mechanism compensator. The compensator automatically varies the volume output. When pressure approaches normal system pressure, the compensator begins to reduce the flow output of the pump. The pump is fully compensated (near zero flow) when normal system pressure is attained. When the pump is in this fully compensated condition, its internal bypass mechanism provides fluid circulation through the pump for cooling and lubrication. A relief valve is installed in the system as a safety backup.

(Figure 11-3)

An advantage of the open-center system over the closed-center system is that the continuous pressurization of the system is eliminated.

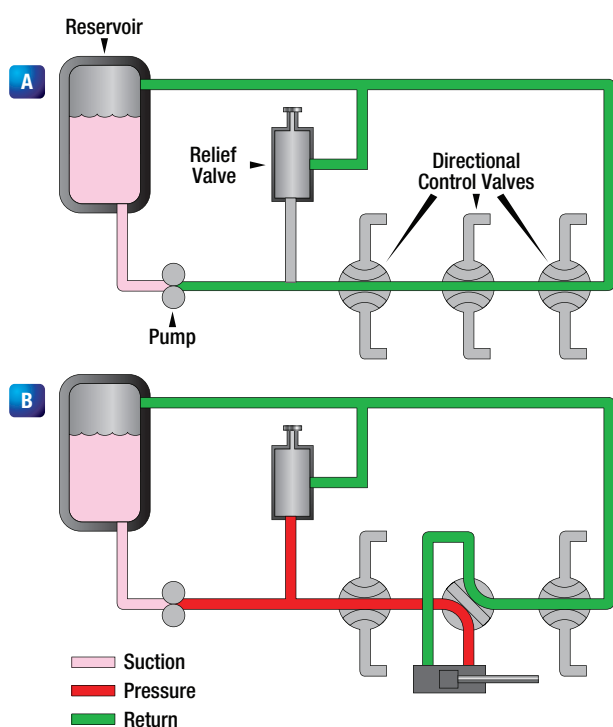


Figure 11-2. Open center hydraulic system.

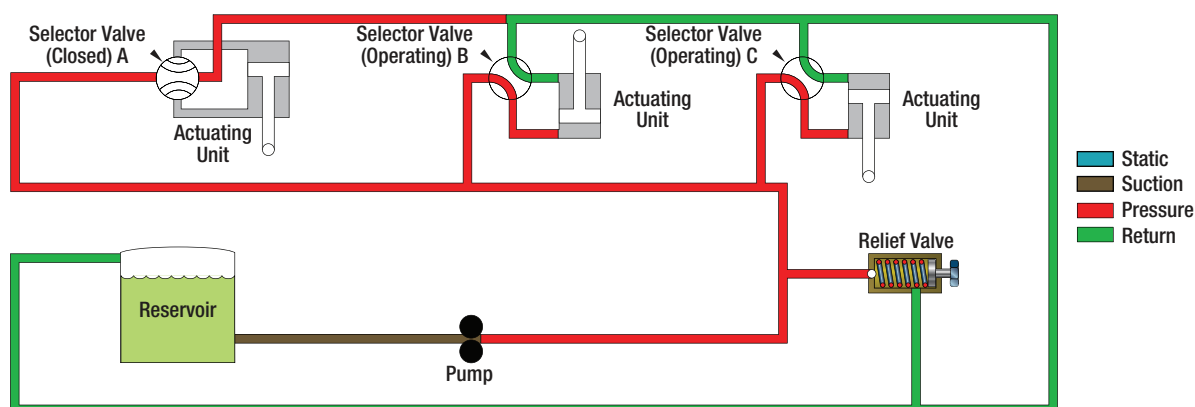


Figure 11-3. A basic closed-center hydraulic system with a variable displacement pump.

Since the pressure is built up gradually after the selector valve is moved to an operating position, there is very little shock from pressure surges. This action provides a smoother operation of the actuating mechanisms. The operation is slower than the closed-center system, in which the pressure is available the moment the selector valve is positioned. Since most aircraft applications require instantaneous operation, closed center systems are the most widely used.

EVOLUTION OF HYDRAULIC SYSTEMS

Smaller aircraft have relatively low flight control surface loads, and the pilot can operate the flight controls by hand. Hydraulic systems were utilized for brake systems on early aircraft. When aircraft started to fly faster and got larger in size, the pilot was not able to move the control surfaces by hand anymore, and hydraulic power boost systems were introduced. Power boost systems assist the pilot in overcoming high control forces, but the pilot still actuates the flight controls by cable or push rod. Small power packs are the latest evolution of the simple hydraulic system. They reduce weight by eliminating hydraulic lines and large quantities of hydraulic fluid.

HYDRAULIC POWER PACK SYSTEM

A hydraulic power pack is a compact unit that consists of an electric pump, a reservoir, valves, filters, and pressure relief valve all in one assembly. (*Figure 11-4*) The advantage of the power pack is that there is no need for a centralized hydraulic power supply system and long stretches of hydraulic lines. This reduces weight. Power packs are driven by either an engine gearbox or electric motor. Integration of essential valves, filters, sensors, and transducers virtually eliminates any opportunity for external leakage, and simplifies troubleshooting. Some power pack systems have an integrated actuator. These systems are used to control the stabilizer trim, landing gear, or flight control surfaces directly, thus eliminating the need for a centralized hydraulic system.

MODERN HIGH PERFORMANCE SYSTEMS

Many modern aircraft use a power supply system and fly-by-wire flight control. The pilot input is electronically sent to the flight control servos. The servos use hydraulic pressure to move the control surfaces. Cables or push rods are not used. Some manufacturers are reducing the use of hydraulic systems in their aircraft in favor of electrically controlled systems.



Figure 11-4. Hydraulic power pack.

The Boeing 787 is the first aircraft designed with more electrical systems than hydraulic systems. Large aircraft hydraulic systems are complexed close-center systems with a wide variety of components and, typically, triple redundancy. *Figure 11-5* is a typical example of a hydraulic system in a large commercial aircraft. After a discussion of hydraulic fluid and various hydraulic system components, large aircraft hydraulic systems will be discussed in greater detail later in this *Sub-Module*.

HYDRAULIC FLUID

Hydraulic system liquids are used primarily to transmit and distribute forces to various units to be actuated. Liquids are able to do this because they are almost incompressible. Pascal's Law states that pressure applied to any part of a confined liquid is transmitted with undiminished intensity to every other part. Thus, if a number of passages exist in a system, pressure can be distributed through all of them by means of the liquid. Manufacturers of hydraulic devices usually specify the type of liquid best suited for use with their equipment in view of the working conditions, the service required, temperatures expected inside and outside the systems, pressures the liquid must withstand, the possibilities of corrosion, and other conditions that must be considered.

If incompressibility and fluidity were the only qualities required, any liquid that is not too thick could be used in a hydraulic system. But a satisfactory liquid for a particular installation must possess a number of other properties. Some of the properties and characteristics that must be considered when selecting a satisfactory liquid for a particular system are discussed in the following paragraphs.

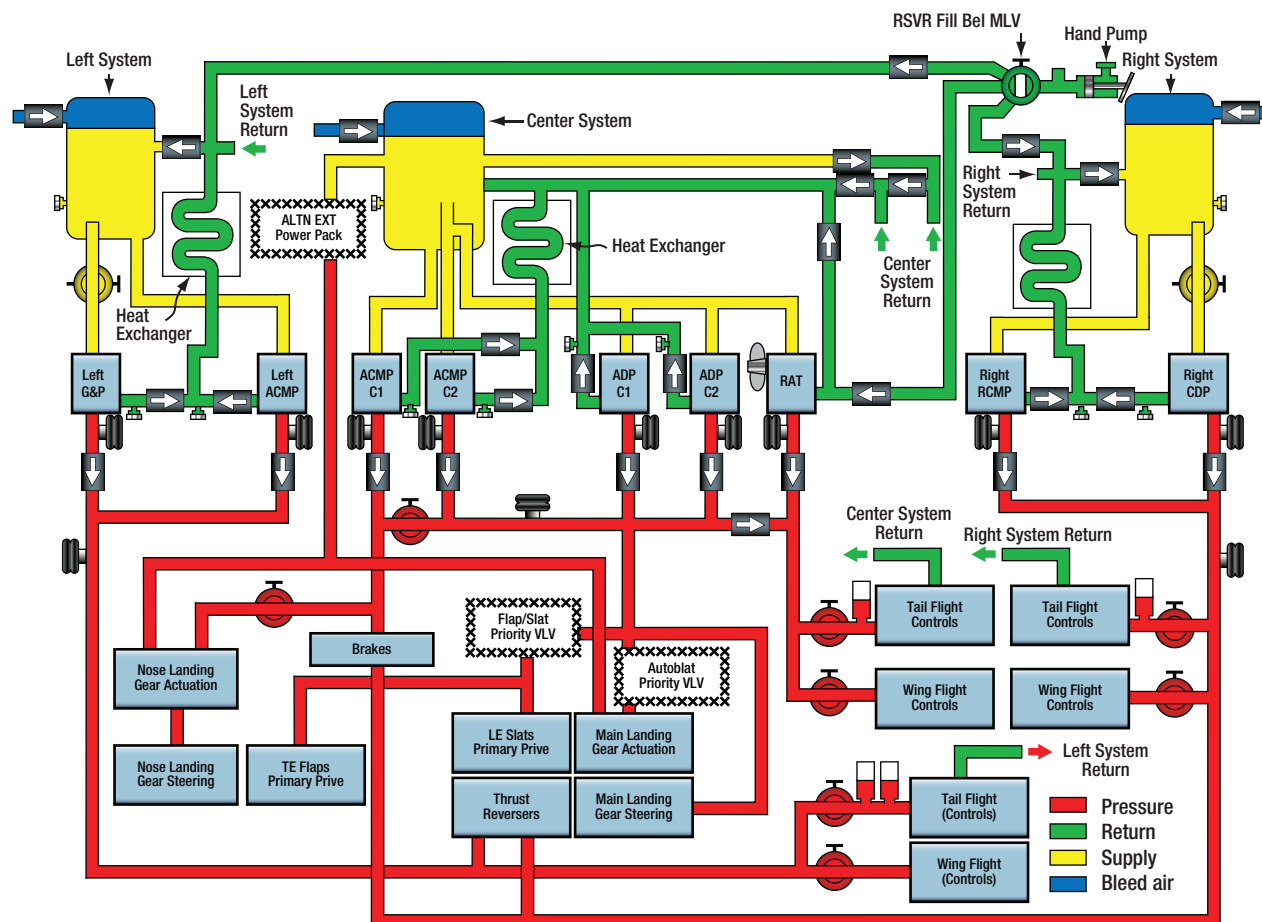


Figure 11-5. Large commercial aircraft hydraulic system.

PROPERTIES

VISCOSITY

One of the most important properties of any hydraulic fluid is its viscosity. Viscosity is internal resistance to flow. A liquid such as gasoline that has a low viscosity flows easily, while a liquid such as tar that has a high viscosity flows slowly. Viscosity increases as temperature decreases. A satisfactory liquid for a given hydraulic system must have enough body to give a good seal at pumps, valves, and pistons, but it must not be so thick that it offers resistance to flow, leading to power loss and higher operating temperatures. These factors add to the load and to excessive wear of parts. A fluid that is too thin also leads to rapid wear of moving parts or of parts that have heavy loads. The instruments used to measure the viscosity of a liquid are known as viscometers or viscosimeters.

Several types of viscosimeters are in use today. The Saybolt viscometer measures the time required, in seconds, for 60 milliliters of the tested fluid at 100°F to

pass through a standard orifice. The time measured is used to express the fluid's viscosity, in Saybolt universal seconds or Saybolt furol seconds.

(Figure 11-6)

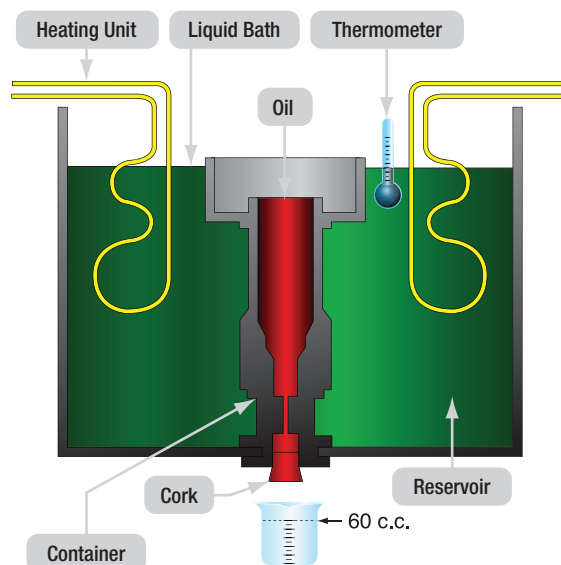


Figure 11-6. Saybolt viscosimeter.

CHEMICAL STABILITY

Chemical stability is another property that is exceedingly important in selecting a hydraulic liquid. It is the liquid's ability to resist oxidation and deterioration for long periods. All liquids tend to undergo unfavorable chemical changes under severe operating conditions. This is the case, for example, when a system operates for a considerable period of time at high temperatures. Excessive temperatures have a great effect on the life of a liquid. It should be noted that the temperature of the liquid in the reservoir of an operating hydraulic system does not always represent a true state of operating conditions. Localized hot spots occur on bearings, gear teeth, or at the point where liquid under pressure is forced through a small orifice. Continuous passage of a liquid through these points may produce local temperatures high enough to carbonize or sludge the liquid, yet the liquid in the reservoir may not indicate an excessively high temperature.

Liquids with a high viscosity have a greater resistance to heat than light or low-viscosity liquids that have been derived from the same source. The average hydraulic liquid has a low viscosity. Fortunately, there is a wide choice of liquids available for use within the viscosity range required of hydraulic liquids.

Liquids may break down if exposed to air, water, salt, or other impurities, especially if they are in constant motion or subject to heat. Some metals, such as zinc, lead, brass, and copper, have an undesirable chemical reaction on certain liquids. These chemical processes result in the formation of sludge, gums, and carbon or other deposits that clog openings, cause valves and pistons to stick or leak, and give poor lubrication to moving parts. As soon as small amounts of sludge or other deposits are formed, the rate of formation generally increases more rapidly. As they are formed, certain changes in the physical and chemical properties of the liquid take place. The liquid usually becomes darker in color, higher in viscosity, and acids are formed.

FLASH POINT

Flash point is the temperature at which a liquid gives off vapor in sufficient quantity to ignite momentarily or flash when a flame is applied. A high flash point is desirable for hydraulic liquids because it indicates good resistance to combustion and a low degree of evaporation at normal temperatures.

FIRE POINT

Fire point is the temperature at which a substance gives off vapor in sufficient quantity to ignite and continue to burn when exposed to a spark or flame. Like flash point, a high fire point is required of desirable hydraulic liquids.

TYPES OF HYDRAULIC FLUIDS

To assure proper system operation and to avoid damage to nonmetallic components of the hydraulic system, the correct fluid must be used. When adding fluid to a system, use the type specified in the aircraft manufacturer's maintenance manual or on the instruction plate affixed to the reservoir or unit being serviced. The three principal categories of hydraulic fluids are:

1. Minerals.
2. Polyalphaolefins.
3. Phosphate esters.

When servicing a hydraulic system, the technician must be certain to use the correct category of replacement fluid. Hydraulic fluids are not necessarily compatible. For example, contamination of the fire-resistant fluid MIL-H-83282 with MIL-H-5606 may render the MIL-H-83282 non fire-resistant.

MINERAL-BASED FLUIDS

Mineral oil-based hydraulic fluid (MIL-H-5606) is the oldest, dating back to the 1940s. It is used in many systems, especially where the fire hazard is comparatively low. MIL-H-6083 is simply a rust-inhibited version of MIL-H-5606. They are completely interchangeable. Suppliers generally ship hydraulic components with MIL-H-6083. Mineral-based hydraulic fluid (MIL-H-5606) is processed from petroleum. It has an odor similar to penetrating oil and is dyed red. Synthetic rubber seals are used with petroleum-based fluids.

POLYALPHAOLEFIN-BASED FLUIDS

MIL-H-83282 is a fire-resistant hydrogenated polyalphaolefin based fluid developed in the 1960s to overcome the flammability characteristics of MIL-H-5606. MIL-H-83282 is significantly more flame resistant than MIL-H-5606, but a disadvantage is the high viscosity at low temperature. It is generally limited to -40°C. However, it can be used in the same system and with the same seals, gaskets, and hoses as MIL-H-5606. MIL-H-46170 is the rust-inhibited version of MIL-H-83282.

Small aircraft predominantly use MIL-H-5606, but some have switched to MIL-H-83282 if they can accommodate the high viscosity at low temperature.

PHOSPHATE ESTER-BASED FLUID (SKYDROL®)

These fluids are used in most commercial transport category aircraft and are extremely fire-resistant. However, they are not fireproof and under certain conditions, they burn. The earliest generation of these fluids was developed after World War II as a result of the growing number of aircraft hydraulic brake fires that drew the collective concern of the commercial aviation industry. Progressive development of these fluids occurred as a result of performance requirements of newer aircraft designs. The airframe manufacturers dubbed these new generations of hydraulic fluid as types based on their performance.

Today, types IV and V fluids are used. Two distinct classes of type IV fluids exist based on their density: class I fluids are low density and class II fluids are standard density. The class I fluids provide weight savings advantages versus class II. In addition to the type IV fluids that are currently in use, type V fluids are being developed in response to industry demands for a more thermally stable fluid at higher operating temperatures. Type V fluids will be more resistant to hydrolytic and oxidative degradation at high temperature than the type IV fluids.

INTERMIXING OF FLUIDS

Due to the difference in composition, petroleum-based and phosphate ester-based fluids will not mix; neither are the seals for any one fluid usable with or tolerant of any of the other fluids. Should an aircraft hydraulic system be serviced with the wrong type fluid, immediately drain and flush the system and maintain the seals according to the manufacturer's specifications.

COMPATIBILITY WITH AIRCRAFT MATERIALS

Aircraft hydraulic systems designed around Skydrol® fluids should be virtually trouble-free if properly serviced. Skydrol® is a registered trademark of Monsanto Company. Skydrol® does not appreciably affect common aircraft metals - aluminum, silver, zinc, magnesium, cadmium, iron, stainless steel, bronze, chromium, and others - as long as the fluids are kept free of

contamination. Due to the phosphate ester base of Skydrol® fluids, thermoplastic resins, including vinyl compositions, nitrocellulose lacquers, oil-based paints, linoleum, and asphalt may be softened chemically by Skydrol® fluids. However, this chemical action usually requires longer than just momentary exposure, and spills that are wiped up with soap and water do not harm most of these materials. Paints that are Skydrol® resistant include epoxies and polyurethanes. Today, polyurethanes are the standard of the aircraft industry because of their ability to keep a bright, shiny finish for long periods of time and for the ease with which they can be removed.

Hydraulic systems require the use of special accessories that are compatible with the hydraulic fluid. Appropriate seals, gaskets, and hoses must be specifically designated for the type of fluid in use. Care must be taken to ensure that the components installed in the system are compatible with the fluid. When gaskets, seals, and hoses are replaced, positive identification should be made to ensure that they are made of the appropriate material. Skydrol® type V fluid is compatible with natural fibers and with a number of synthetics, including nylon and polyester, which are used extensively in most aircraft. Petroleum oil hydraulic system seals of neoprene or Buna-N are not compatible with Skydrol® and must be replaced with seals of butyl rubber or ethylene-propylene elastomers.

HYDRAULIC FLUID CONTAMINATION

Experience has shown that trouble in a hydraulic system is inevitable whenever the liquid is allowed to become contaminated. The nature of the trouble, whether a simple malfunction or the complete destruction of a component, depends to some extent on the type of contaminant. Two general contaminants are:

1. Abrasives, including such particles as core sand, weld spatter, machining chips, and rust.
2. Non-abrasives, including those resulting from oil oxidation and soft particles worn or shredded from seals and other organic components.

CONTAMINATION CHECK

Whenever it is suspected that a hydraulic system has become contaminated or the system has been operated at temperatures in excess of the specified maximum, a check of the system should be made. The filters in most hydraulic systems are designed to remove most

foreign particles that are visible to the naked eye. Hydraulic liquid that appears clean to the naked eye may be contaminated to the point that it is unfit for use. Thus, visual inspection of the hydraulic liquid does not determine the total amount of contamination in the system.

Large particles of impurities in the hydraulic system are indications that one or more components are being subjected to excessive wear. Isolating the defective component requires a systematic process of elimination. Fluid returned to the reservoir may contain impurities from any part of the system. To determine which component is defective, liquid samples should be taken from the reservoir and various other locations in the system. Samples should be taken in accordance with the applicable manufacturer's instructions for a particular hydraulic system. Some hydraulic systems are equipped with permanently installed bleed valves for taking liquid samples, whereas on other systems, lines must be disconnected to provide a place to take a sample.

HYDRAULIC SAMPLING SCHEDULE

- Routine sampling - each system should be sampled at least once a year, or every 3 000 flight hours, or whenever the airframe manufacturer suggests.
- Unscheduled maintenance - when malfunctions may have a fluid related cause, samples should be taken.
- Suspicion of contamination - if contamination is suspected, fluids should be drained and replaced, with samples taken before and after the maintenance procedure.

SAMPLING PROCEDURE

- Pressurize and operate hydraulic system for 10-15 minutes. During this period, operate various flight controls to activate valves and thoroughly mix hydraulic fluid.
- Shut down and depressurize the system.
- Before taking samples, always be sure to wear the proper personal protective equipment that should include, at the minimum, safety glasses and gloves.
- Wipe off sampling port or tube with a lint free cloth. Do not use shop towels or paper products that could produce lint. Generally speaking, the human eye can see particles down to about 40 microns in size. Since we are concerned with particles down to five microns in size, it is easy to contaminate a sample without ever knowing it.

- Place a waste container under the reservoir drain valve and open valve so that a steady, but not forceful, stream is running.
- Allow approximately one pint (250 ml) of fluid to drain. This purges any settled particles from the sampling port.
- Insert a precleaned sample bottle under the fluid stream and fill, leaving an air space at the top.
- Withdraw the bottle and cap immediately.
- Close drain valve.
- Fill out sample identification label supplied in sample kit, making sure to include customer name, aircraft type, aircraft tail number, hydraulic system sampled, and date sampled. Indicate on the sample label under remarks if this is a routine sample or if it is being taken due to a suspected problem.
- Service system reservoirs to replace the fluid that was removed.
- Submit samples for analysis to laboratory.

CONTAMINATION CONTROL

Filters provide adequate control of the contamination problem during all normal hydraulic system operations. Control of the size and amount of contamination entering the system from any other source is the responsibility of the people who service and maintain the equipment. Therefore, precautions should be taken to minimize contamination during maintenance, repair, and service operations. If the system becomes contaminated, the filter element should be removed and cleaned or replaced. As an aid in controlling contamination, the following maintenance and servicing procedures should be followed at all times:

- Maintain all tools and the work area (workbenches and test equipment) in a clean, dirt-free condition.
- A suitable container should always be provided to receive the hydraulic liquid that is spilled during component removal or disassembly procedures.
- Before disconnecting hydraulic lines or fittings, clean the affected area with dry cleaning solvent.
- All hydraulic lines and fittings should be capped or plugged immediately after disconnecting.
- Before assembly of any hydraulic components, wash all parts in an approved dry cleaning solvent.
- After cleaning the parts in the dry cleaning solution, dry the parts thoroughly and lubricate them with the recommended preservative or hydraulic liquid before assembly. Use only clean, lint free cloths to wipe or dry the component parts.

- All seals and gaskets should be replaced during the reassembly procedure. Use only those seals and gaskets recommended by the manufacturer.
- All parts should be connected with care to avoid stripping metal slivers from threaded areas. All fittings and lines should be installed and torqued in accordance with applicable technical instructions.
- All hydraulic servicing equipment should be kept clean and in good operating condition.

FILTERS

Contamination, both particulate and chemical, is detrimental to the performance and life of components in the aircraft hydraulic system. Contamination enters the system through normal wear of components, by ingestion through external seals during servicing, or maintenance, when the system is opened to replace/repair components, etc. To control the particulate contamination in the system, filters are installed in the pressure line, in the return line, and in the pump case drain line of each system. The filter rating is given in microns as an indication of the smallest particle size that is filtered out. The replacement interval of these filters is established by the manufacturer and is included in the maintenance manual.

In the absence of specific replacement instructions, a recommended service life of the filter elements is:

- Pressure Filters - 3 000 hours
- Return Filters - 1 500 hours
- Case Drain Filters - 600 hours

HYDRAULIC SYSTEM FLUSHING

When inspection of hydraulic filters or hydraulic fluid evaluation indicates that the fluid is contaminated, flushing the system may be necessary. This must be done according to the manufacturer's instructions; however, a typical procedure for flushing is as follows:

1. Connect a ground hydraulic test stand to the inlet and outlet test ports of the system. Verify that the ground unit fluid is clean and contains the same fluid as the aircraft.
2. Change the system filters.
3. Pump clean, filtered fluid through the system, and operate all subsystems until no obvious signs of contamination are found during inspection of the filters. Dispose of contaminated fluid and filter. **Note:** *A visual inspection of hydraulic filters is not always effective.*

4. Disconnect the test stand and cap the ports.
5. Ensure that the reservoir is filled to the full line or proper service level.

It is very important to check if the fluid in the hydraulic test stand, or mule, is clean before the flushing operation starts. A contaminated hydraulic test stand can quickly contaminate other aircraft if used for ground maintenance operations.

HEALTH AND HANDLING

Skydrol® fluids are phosphate ester-based fluids blended with performance additives. Phosphate esters are good solvents and dissolve away some of the fatty materials of the skin. Repeated or prolonged exposure may cause drying of the skin, which if unattended, could result in complications, such as dermatitis or even secondary infection from bacteria. Skydrol® fluids could cause itching of the skin but have not been known to cause allergic type skin rashes. Always use the proper gloves and eye protection when handling any type of hydraulic fluid. When Skydrol®/Hyjet mist or vapor exposure is possible, a respirator capable of removing organic vapors and mists must be worn.

Ingestion of any hydraulic fluid should be avoided. Although small amounts do not appear to be highly hazardous, any significant amount should be tested in accordance with manufacturer's direction, followed with hospital supervised stomach treatment.

HYDRAULIC RESERVOIRS AND ACCUMULATORS

RESERVOIRS

The reservoir is a tank in which an adequate supply of fluid for the system is stored. Fluid flows from the reservoir to the pump, where it is forced through the system and eventually returned to the reservoir. The reservoir not only supplies the operating needs of the system, but it also replenishes fluid lost through leakage. Furthermore, the reservoir serves as an overflow basin for excess fluid forced out of the system by thermal expansion (the increase of fluid volume caused by temperature changes), the accumulators, and by piston and rod displacement. The reservoir also furnishes a place for the fluid to purge itself of air bubbles that may enter the system.

Foreign matter picked up in the system may also be separated from the fluid in the reservoir or as it flows through line filters. Reservoirs are either pressurized or non-pressurized.

Baffles and/or fins are incorporated in most reservoirs to keep the fluid within the reservoir from having random movement, such as vortexing (swirling) and surging. These conditions can cause fluid to foam and air to enter the pump along with the fluid.

Many reservoirs incorporate strainers in the filler neck to prevent the entry of foreign matter during servicing. These strainers are made of fine mesh screening and are usually referred to as finger strainers because of their shape. Finger strainers should never be removed or punctured as a means of speeding up the pouring of fluid into the reservoir. Reservoirs could have an internal trap to make sure fluid goes to the pumps during negative-G conditions.

Most aircraft have emergency hydraulic systems that take over if main systems fail. In many such systems, the pumps of both systems obtain fluid from a single reservoir. Under such circumstances, a supply of fluid for the emergency pump is ensured by drawing the hydraulic fluid from the bottom of the reservoir. The

main system draws its fluid through a standpipe located at a higher level. With this arrangement, should the main system's fluid supply become depleted, adequate fluid is left for operation of the emergency system.

Figure 11-7 illustrates that the engine driven pump (EDP) is not able to draw fluid any more if the reservoir gets depleted below the standpipe. The alternating current motor driven pump (ACMP) still has a supply of fluid for emergency operations.

NON-PRESSURIZED RESERVOIRS

Non-pressurized reservoirs are used in aircraft that are not designed for violent maneuvers, do not fly at high altitudes, or in which the reservoir is located in the pressurized area of the aircraft. High altitude in this situation means an altitude where atmospheric pressure is inadequate to maintain sufficient flow of fluid to the hydraulic pumps. Most non-pressurized reservoirs are constructed in a cylindrical shape. The outer housing is manufactured from a strong corrosion-resistant metal. Filter elements are normally installed within the reservoir to clean returning system hydraulic fluid.

In some of the older aircraft, a filter bypass valve is incorporated to allow fluid to bypass the filter in the event the filter becomes clogged. Reservoirs can be serviced by pouring fluid directly into the reservoir through a filler

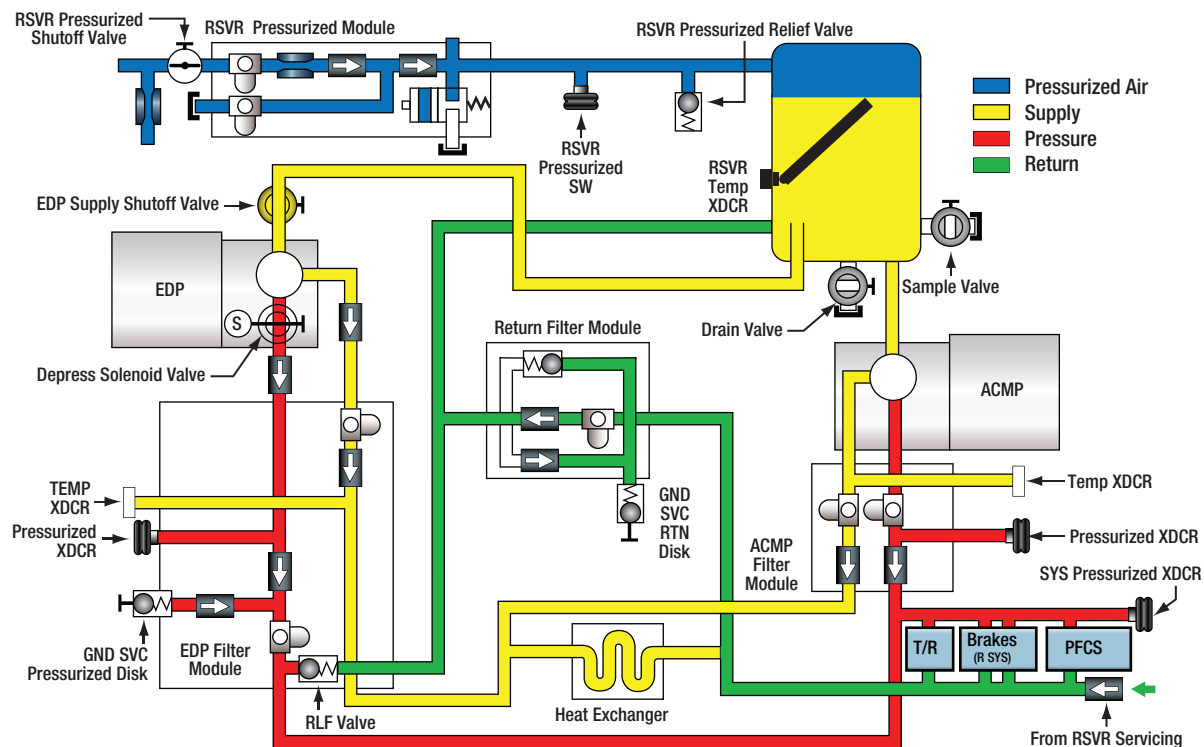


Figure 11-7. Hydraulic reservoir standpipe for emergency operations.

strainer (finger strainer) assembly incorporated within the filler well to strain out impurities as the fluid enters the reservoir. Generally, non-pressurized reservoirs use a visual gauge to indicate the fluid quantity. Gauges incorporated on or in the reservoir may be a direct reading glass tube type or a float type rod that is visible through a transparent dome. In some cases, the fluid quantity may also be read in the cockpit through the use of quantity transmitters. A typical non-pressurized reservoir is shown in *Figure 11-8*.

This reservoir consists of a welded body and cover assembly clamped together. Gaskets are incorporated to seal against leakage between assemblies. Non-pressurized reservoirs are slightly pressurized due to thermal expansion of fluid and the return of fluid to the reservoir from the main system. This pressure ensures that there is a positive flow of fluids to the inlet ports of the hydraulic pumps. Most reservoirs of this type are vented directly to the atmosphere or cabin with only a check valve and filter to control the outside air source. The reservoir system includes a pressure and vacuum relief valve.

The purpose of the valve is to maintain a differential pressure range between the reservoir and cabin. A manual air bleed valve is installed on top of the reservoir

to vent the reservoir. The valve is connected to the reservoir vent line to allow depressurization of the reservoir. The valve is actuated prior to servicing the reservoir to prevent fluid from being blown out of the filler as the cap is being removed. The manual bleed valve also needs to be actuated if hydraulic components need to be replaced.

PRESSURIZED RESERVOIRS

Reservoirs on aircraft designed for high-altitude flight are usually pressurized. Pressurizing assures a positive flow of fluid to the pump at high altitudes when low atmospheric pressures are encountered. On some aircraft, the reservoir is pressurized by bleed air taken from the compressor section of the engine. On others, the reservoir may be pressurized by hydraulic system pressure.

Air-Pressurized Reservoirs

Air-pressurized reservoirs are used in many commercial transport type aircraft. (*Figures 11-9 and 11-10*)

Pressurization of the reservoir is required because the reservoirs are often located in wheel wells or other non-pressurized areas of the aircraft and at high altitude there is not enough atmospheric pressure to move the fluid to the pump inlet. Engine bleed air is used to pressurize the reservoir. The reservoirs are typically cylindrical in shape. The following components are installed on a typical reservoir:

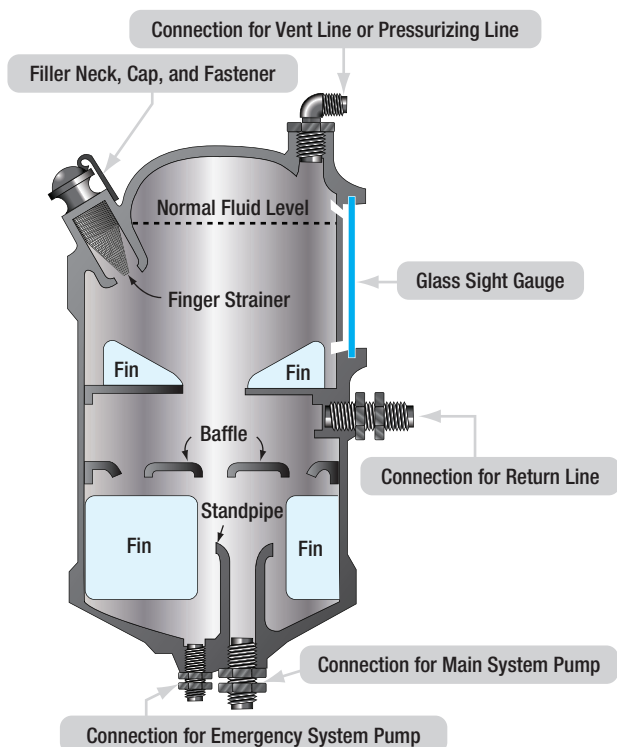


Figure 11-8. Non-pressurized reservoir.

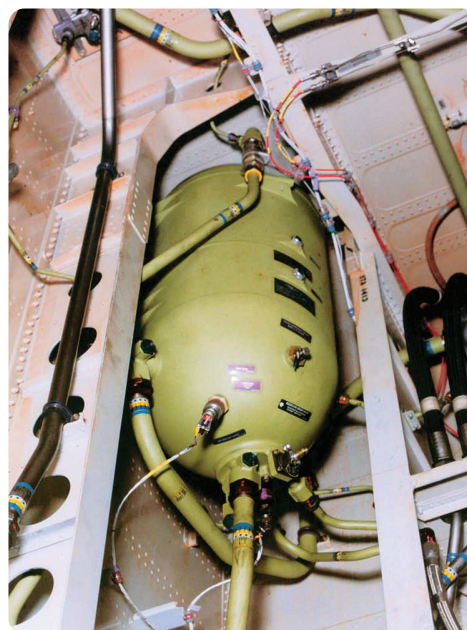


Figure 11-9. Air-pressurized reservoir.

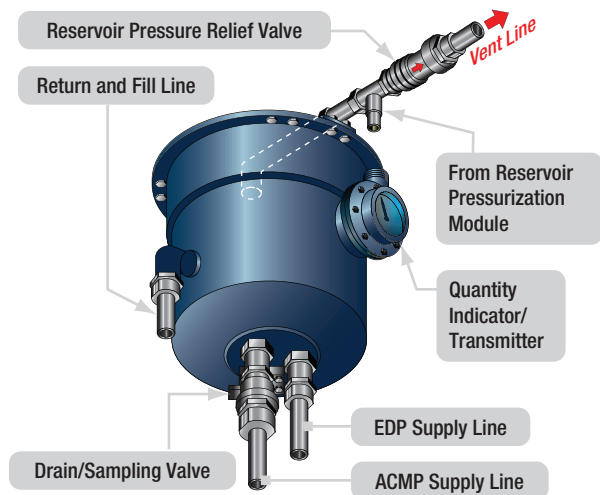


Figure 11-10. Components of an air-pressurized reservoir.

- Reservoir pressure relief valve - prevents over pressurization of the reservoir. Valve opens at a preset value.
- Sight glasses (low and overfull) - provides visual indication for flight crews and maintenance personnel that the reservoir needs to be serviced.
- Reservoir sample valve - used to draw a sample of hydraulic fluid for testing.
- Reservoir drain valve - used to drain the fluids out of the reservoir for maintenance operation.
- Reservoir temperature transducer - provides hydraulic fluid temperature information for the flight deck. (**Figure 11-11**)
- Reservoir quantity transmitter - transmits fluid quantity to the flight deck so that the flight crew can monitor fluid quantity during flight. (**Figure 11-11**)

A reservoir pressurization module is installed close to the reservoir. (**Figure 11-12**) The reservoir pressurization module supplies airplane bleed air to the reservoirs. The module typically consists of the following parts:

- Filters (2)
- Manual Bleed Valve
- Check Valves (2)
- Gauge Port
- Test Port

A manual bleeder valve is incorporated into the module. During hydraulic system maintenance, it is necessary to relieve reservoir air pressure to assist in the installation and removal of components, lines, etc. This type of valve is small in size and has a push button installed in the outer case. When the bleeder valve push button is pushed, pressurized air from the reservoir flows through the valve to an overboard vent until the air pressure is



Figure 11-11. Temperature and quantity sensors.

depleted or the button is released. When the button is released, the internal spring causes the poppet to return to its seat. Some hydraulic fluid can escape from the manual bleed valve when the button is depressed.

CAUTION: Put a rag around the air bleed valve on the reservoir pressurization module to catch hydraulic fluid spray. Hydraulic fluid spray can cause injuries to persons.

Fluid-Pressurized Reservoirs

Some aircraft hydraulic system reservoirs are pressurized by hydraulic system pressure. Regulated hydraulic pump output pressure is applied to a movable piston inside the cylindrical reservoir. This small piston is attached to and moves a larger piston against the reservoir fluid. The reduced force of the small piston when applied by the larger piston is adequate to provide head pressure for high altitude operation. The small piston protrudes out of the body of the reservoir. The amount exposed is used as a reservoir fluid quantity indicator.

Figure 11-13 illustrates the concept behind the fluid-pressurized hydraulic reservoir. The reservoir has five ports: pump suction, return, pressurizing, overboard drain, and bleed port. Fluid is supplied to the pump through the pump suction port.

Fluid returns to the reservoir from the system through the return port. Pressure from the pump enters the pressurizing cylinder in the top of the reservoir through the pressurizing port. The overboard drain port drains the reservoir, when necessary, while performing maintenance. The bleed port is used as an aid in servicing the reservoir. When servicing a system equipped with this type of reservoir, place a container under the bleed drain port. The fluid should then be pumped into the reservoir until air-free fluid flows through the bleed

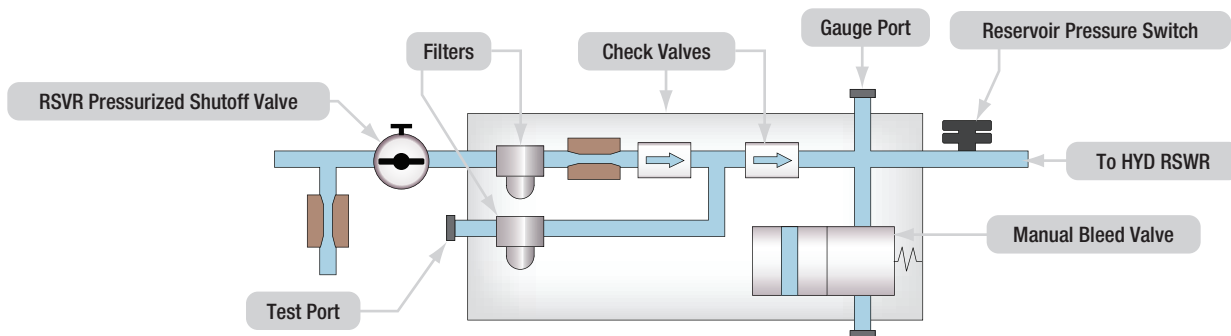


Figure 11-12. Reservoir pressurization module.

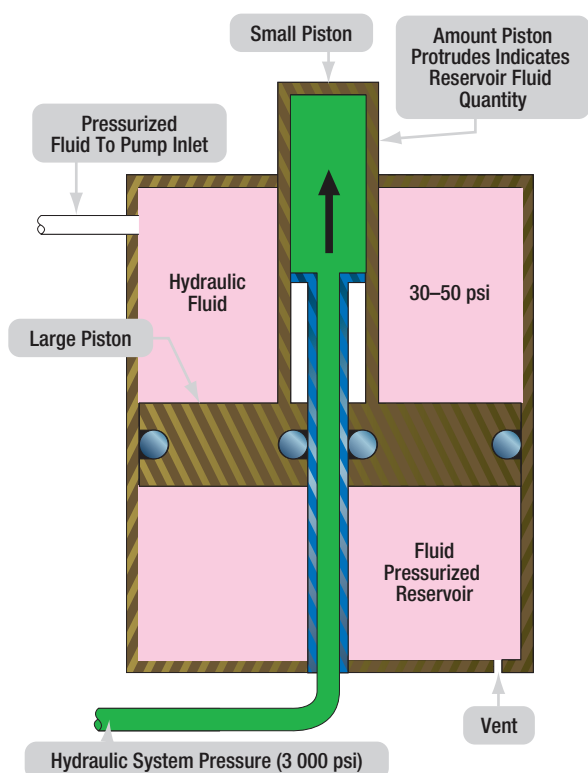


Figure 11-13. Operating principle behind a fluid-pressurized hydraulic reservoir.

drain port. The reservoir fluid level is indicated by the markings on the part of the pressurizing cylinder that moves through the reservoir dust cover assembly. There are three fluid level markings indicated on the cover: full at zero system pressure (FULL ZERO PRESS), full when system is pressurized (FULL SYS PRESS), and REFILL.

When the system is unpressurized and the pointer on the reservoir lies between the two full marks, a marginal reservoir fluid level is indicated. When the system is pressurized and the pointer lies between REFILL and FULL SYS PRESS, a marginal reservoir fluid level is also indicated.

RESERVOIR SERVICING

Non-pressurized reservoirs can be serviced by pouring fluid directly into the reservoir through a filler strainer (finger strainer) assembly incorporated within the filler well to strain out impurities as the fluid enters the reservoir. Many reservoirs also have a quick disconnect service port at the bottom of the reservoir. A hydraulic filler unit can be connected to the service port to add fluid to the reservoir. This method reduces the chances of contamination of the reservoir. Aircraft that use pressurized reservoirs often have a central filling station in the ground service bay to service all reservoirs from a single point. (Figure 11-14)

A built-in hand pump is available to draw fluid from a container through a suction line and pump it into the reservoirs. Additionally, a pressure fill port is available for attachment of a hydraulic mule or serving cart, which uses an external pump to push fluid into the aircraft hydraulic system. A check valve keeps the hand pump output from exiting the pressure fill port. A single filter is located downstream of both the pressure fill port and the hand pump to prevent the introduction of contaminants during fluid servicing. It is very important to follow the maintenance instructions when servicing the reservoir. To get the correct results when the hydraulic fluid quantities are checked or the reservoirs are to be filled, the airplane should be in the correct configuration. Failure to do so could result in over servicing of the reservoir. This configuration could be different for each aircraft.

The following service instructions are an example of a large transport type aircraft. Before servicing always make sure that the:

- Spoilers are retracted.
- Landing gear is down.
- Landing gear doors are closed.

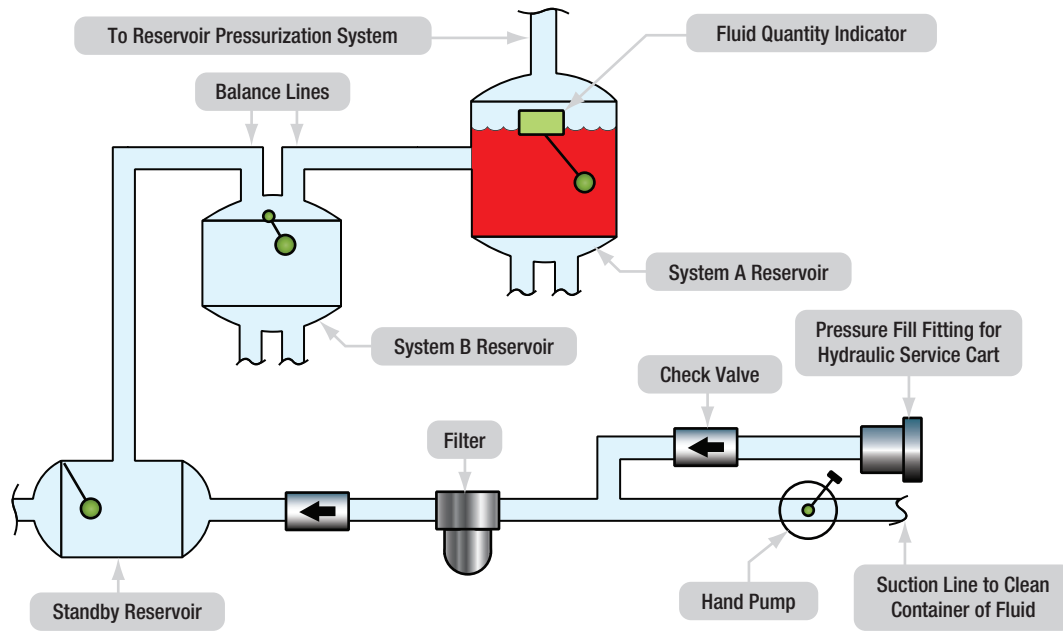


Figure 11-14. The hydraulic ground service station on a Boeing 737 provides for hydraulic fluid servicing with a hand pump or via an external pressure fluid source. All three reservoirs are serviced from the same location.

- Thrust reverser's are retracted.
- Parking brake accumulator pressure reads at least 2 500 psi.

ACCUMULATORS

The accumulator is a steel sphere divided into two chambers by a synthetic rubber diaphragm. The upper chamber contains fluid at system pressure, while the lower chamber is charged with nitrogen or air. Cylindrical types are also used in high-pressure hydraulic systems. Many aircraft have several accumulators in the hydraulic system. There may be a main system accumulator and an emergency system accumulator. There may also be auxiliary accumulators located in various sub-systems. The functions of an accumulator are to:

1. Dampen pressure surges in the hydraulic system caused by actuation of a unit and the effort of the pump to maintain pressure at a preset level.
2. Aid or supplement the power pump when several units are operating at once by supplying extra power from its accumulated, or stored, power.
3. Store power for the limited operation of a hydraulic unit when the pump is not operating.
4. Supply fluid under pressure to compensate for small internal or external (not desired) leaks that would cause the system to cycle continuously by action of the pressure switches continually kicking in.

TYPES OF ACCUMULATORS

There are two general types of accumulators used in aircraft hydraulic systems: spherical and cylindrical.

Spherical

The spherical type accumulator is constructed in two halves that are fastened and threaded, or welded, together. Two threaded openings exist. The top port accepts fittings to connect to the pressurized hydraulic system to the accumulator. The bottom port is fitted with a gas servicing valve, such as a Schrader valve. A synthetic rubber diaphragm, or bladder, is installed in the sphere to create two chambers. Pressurized hydraulic fluid occupies the upper chamber and nitrogen or air charges the lower chamber. A screen at the fluid pressure port keeps the diaphragm, or bladder, from extruding through the port when the lower chamber is charged and hydraulic fluid pressure is zero. A rigid button or disc may also be attached to the diaphragm, or bladder, for this purpose. (*Figure 11-15*)

The bladder is installed through a large opening in the bottom of the sphere and is secured with a threaded retainer plug. The gas servicing valve mounts into the retainer plug.

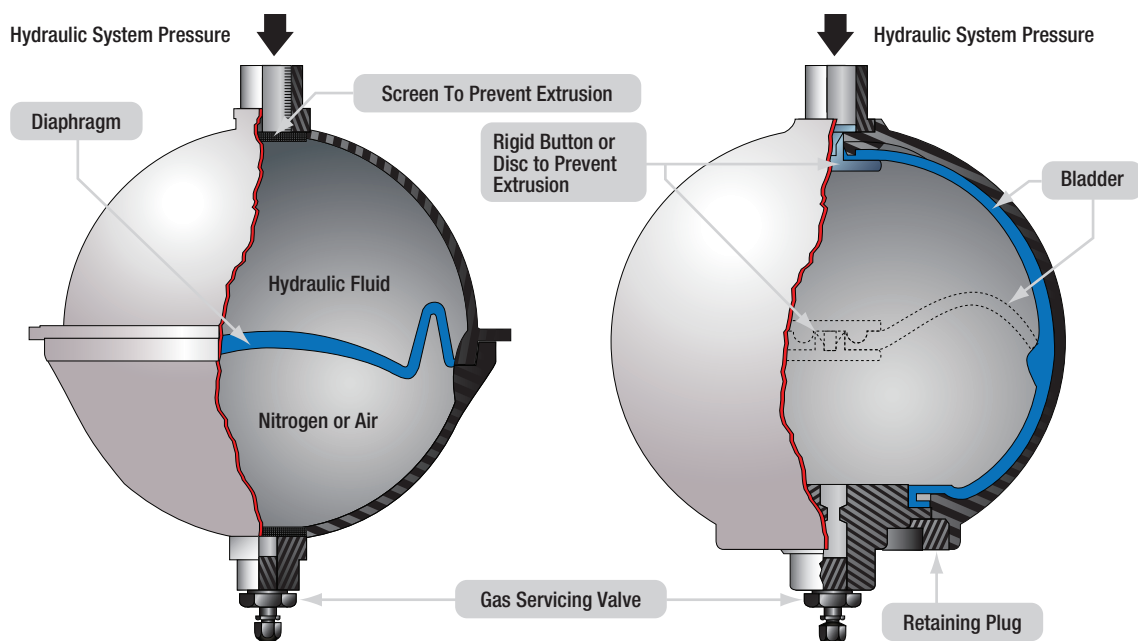


Figure 11-15. A spherical accumulator with diaphragm (left) and bladder (right). The dotted lines in the right drawing depict the bladder when the accumulator is charged with both hydraulic system fluid and nitrogen preload.

Cylindrical

Cylindrical accumulators consist of a cylinder and piston assembly. End caps are attached to both ends of the cylinder. The internal piston separates the fluid and air/nitrogen chambers. The end caps and piston are sealed with gaskets and packings to prevent external leakage around the end caps and internal leakage between the chambers. In one end cap, a hydraulic fitting is used to attach the fluid chamber to the hydraulic system. In the other end cap, a filler valve is installed to perform the same function as the filler valve installed in the spherical accumulator. (Figure 11-16)

In operation, the compressed-air chamber is charged to a predetermined pressure that is somewhat lower than the system operating pressure. This initial charge is referred to as the accumulator preload. As an example of accumulator operation, let us assume that the cylindrical accumulator is designed for a preload of 1 300 psi in a 3 000 psi system. When the initial charge of 1 300 psi is introduced into the unit, hydraulic system pressure is zero. As air pressure is applied through a gas servicing valve, it moves the piston toward the opposite end until it bottoms. If the air behind the piston has a pressure of 1 300 psi, the hydraulic system pump has to create a pressure within the system greater than 1 300 psi before the hydraulic fluid can actuate the piston. At 1 301 psi the piston starts to move within the cylinder, compressing the air as it moves. At 2 000 psi, it has

backed up several inches. At 3 000 psi, the piston has backed up to its normal operating position, compressing the air until it occupies a space less than one half the length of the cylinder.

When actuation of hydraulic units lowers the system pressure, the compressed air expands against the piston, forcing fluid from the accumulator. This supplies an instantaneous supply of fluid to the hydraulic system component. The charged accumulator may also supply fluid pressure to actuate a component(s) briefly in case of pump failure.



Figure 11-16. Cylindrical accumulator.

Maintenance of Accumulators

Maintenance consists of inspections, minor repairs, replacement of component parts, and testing. There is an element of danger in maintaining accumulators. Therefore, proper precautions must be strictly observed to prevent injury and damage.

Before disassembling any accumulator, ensure that all preload air (or nitrogen) pressure has been discharged. Failure to release the preload could result in serious injury to the technician. Before making this check, be certain you know the type of high-pressure air valve used. When you know that all air pressure has been removed, you can take the unit apart. Be sure to follow manufacturer's instructions for the specific unit you have.

HYDRAULIC PRESSURE GENERATION

All aircraft hydraulic systems have at least one power-driven pump and may include a hand pump as an additional unit when the power-driven pump is inoperative. The pump is the source of fluid flow, which when restricted, generates pressure in the hydraulic system. A hydraulic pump can be driven mechanically, electrically or with pneumatic air.

MECHANICAL, ELECTRICAL AND PNEUMATIC-DRIVEN PUMPS

Mechanically driven pumps are the primary source of pressure generation on most aircraft. Typically, the pump is mounted on the accessory gearbox of the main engine and is rotated by a shaft. When the engine is operating, the pump supplies ample fluid flow to generate pressure within the hydraulic system. Electrical motor driven pumps also exist. Often, these are the same pumps as the mechanically driven pumps but the drive shaft is turned by an electrical motor. As such, electrically driven pumps can be mounted away from the engine(s). On large aircraft, they are usually mounted in the wheel well or in a hydraulics bay near the root of the wings. Electrically driven pumps are installed for use in emergencies or during ground operation when engines are not running. Pneumatically driven pumps also exist on aircraft. Typically these pumps are used as demand pumps to supplement the primary pumps and are driven by air from the pneumatic system or by ram air in the case of a ram air turbine (RAT).

Modern transport aircraft use a combination of engine-driven power pumps, electrically-driven power pumps, pneumatically-driven power pumps, power transfer units (PTU), and a RAT-driven pump in a highly effective, fully redundant aircraft hydraulic system. For example, the Airbus A380 has two hydraulic systems, eight engine-driven pumps, and three electrical driven pumps. The Boeing 777 has three hydraulic systems with two engine driven pumps, four electrical driven pumps, two air driven pumps, and a hydraulic pump motor driven by the RAT. (*Figures 11-17 and 11-18*)

Many of the power driven hydraulic pumps of current aircraft are of variable delivery, compensator-controlled type. Constant delivery pumps are also in use. Principles of operation are the same for both types of pumps and are described in further detail below. A discussion of various types of pumps and how they work begins with hand pumps.



Figure 11-17. Engine-driven pump.



Figure 11-18. Electrically-driven pump.

HAND PUMPS

The hydraulic hand pump is used in some older aircraft for the operation of hydraulic subsystems and in a few newer aircraft systems as a backup unit. Hand pumps are generally installed for testing purposes, as well as for use in emergencies. Hand pumps are also installed to service the reservoirs from a single refilling station. The single refilling station reduces the chances for the introduction of fluid contamination.

Several types of hand pumps are used: single action, double action, and rotary. A single action hand pump draws fluid into the pump on one stroke and pumps that fluid out on the next stroke. It is rarely used in aircraft due to this inefficiency.

Double action hand pumps produce fluid flow and pressure on each stroke of the handle. (*Figure 11-19*) The double action hand pump consists essentially of a housing that has a cylinder bore and two ports, a piston, two spring-loaded check valves, and an operating handle. An O-ring on the piston seals against leakage between the two chambers of the piston cylinder bore. An O-ring in a groove in the end of the pump housing seals against leakage between the piston rod and housing.

When the piston is moved to the right, the pressure in the chamber left of the piston is lowered. The inlet port ball check valve opens and hydraulic fluid is drawn into the chamber. At the same time, the rightward movement of the piston forces the piston ball check valve

against its seat. Fluid in the chamber to the right of the piston is forced out of the outlet port into the hydraulic system. When the piston is moved to the left, the inlet port ball check valve seats. Pressure in the chamber left of the piston rises, forcing the piston ball check valve off of its seat. Fluid flows from the left chamber through the piston to the right chamber. The volume in the chamber right of the piston is smaller than that of the left chamber due to the displacement created by the piston rod. As the fluid from the left chamber flows into the smaller right chamber, the excess volume of fluid is forced out of the outlet port to the hydraulic system.

A rotary hand pump may also be employed. It produces continuous output while the handle is in motion. *Figure 11-20* shows a rotary hand pump in a hydraulic system.

CLASSIFICATION OF PUMPS

All pumps may be classified as either positive displacement or nonpositive displacement. Most pumps used in hydraulic systems are positive displacement. A non-positive displacement pump produces a continuous flow. However, because it does not provide a positive internal seal against slippage, its output varies considerably as pressure varies. Centrifugal and propeller pumps are examples of nonpositive displacement pumps. If the output port of a nonpositive displacement pump was blocked off, the pressure would rise and output would decrease to zero. Although the pumping element would

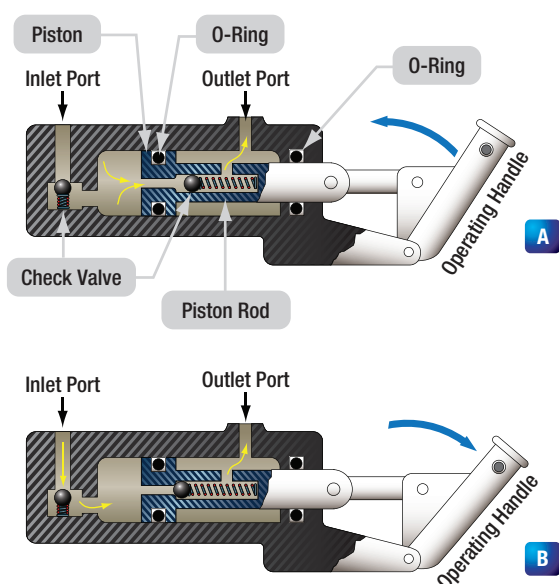


Figure 11-19. Double action hand pump.



Figure 11-20. Rotary hand pump.

continue moving, flow would stop because of slippage inside the pump. In a positive displacement pump, slippage is negligible compared to the pump's volumetric output flow. If the output port were plugged, pressure would increase instantaneously to the point that the pump pressure relief valve opens.

CONSTANT DISPLACEMENT PUMPS

A constant-displacement pump, regardless of pump rotations per minute, forces a fixed or unvarying quantity of fluid through the outlet port during each revolution of the pump. Constant displacement pumps are sometimes called constant-volume or constant-delivery pumps. They deliver a fixed quantity of fluid per revolution, regardless of the pressure demands. Since the constant-delivery pump provides a fixed quantity of fluid during each revolution of the pump, the quantity of fluid delivered per minute depends upon pump rotations per minute. When a constant displacement pump is used in a hydraulic system in which the pressure must be kept at a constant value, a pressure regulator is required.

Gear Type Power Pump

A gear type power pump is a constant displacement pump. It consists of two meshed gears that revolve in a housing. (*Figure 11-21*) The driving gear is driven by the aircraft engine or some other power unit. The driven gear meshes with, and is driven by, the driving gear. Clearance between the teeth as they mesh and between the teeth and the housing is very small. The inlet port of the pump is connected to the reservoir, and the outlet

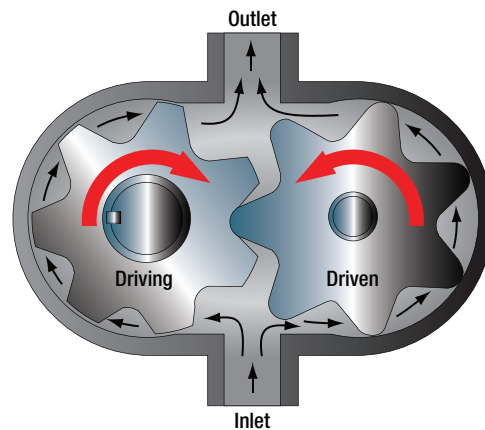


Figure 11-21. Gear-type power pump.

port is connected to the pressure line. When the driving gear turns, it turns the driven gear. Fluid is captured by the teeth as they pass the inlet, and it travels around the housing and exits at the outlet.

Gerotor Pump

A gerotor type power pump consists essentially of a housing containing an eccentric-shaped stationary liner, an internal gear rotor having seven wide teeth of short height, a spur driving gear having six narrow teeth, and a pump cover that contains two crescent-shaped openings. (*Figure 11-22*)

One opening extends into an inlet port and the other extends into an outlet port. During the operation of the pump, the gears turn clockwise together. As the pockets between the gears on the left side of the pump move

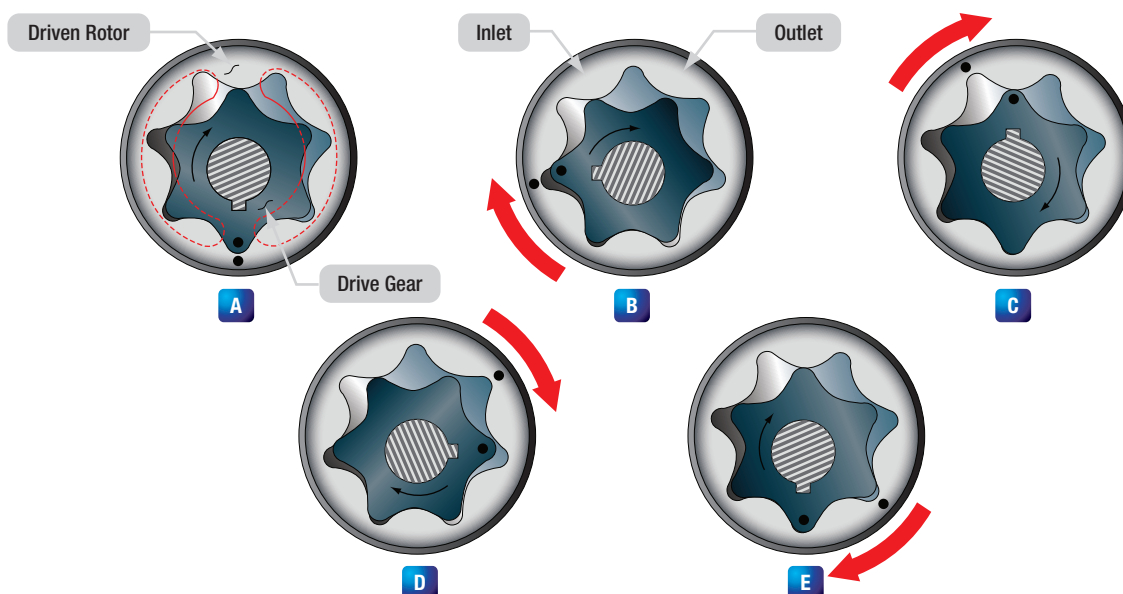


Figure 11-22. Gerotor pump.

from a lowermost position toward a topmost position, the pockets increase in size, resulting in the production of a partial vacuum within these pockets. Since the pockets enlarge while over the inlet port crescent, fluid is drawn into them. As these same pockets (now full of fluid) rotate over to the right side of the pump, moving from the topmost position toward the lowermost position, they decrease in size. This results in the fluid being expelled from the pockets through the outlet port crescent.

Piston Pumps

Piston pumps can be constant-displacement or variable displacement pumps. The common features of design and operation that are applicable to all piston type hydraulic pumps are described in the following paragraphs. Piston type power-driven pumps have flanged mounting bases for the purpose of mounting the pumps on the accessory drive cases of aircraft engines. A pump drive shaft, which turns the mechanism, extends through the pump housing slightly beyond the mounting base. Torque from the driving unit is transmitted to the pump drive shaft by a drive coupling. The drive coupling is a short shaft with a set of male splines on both ends. The splines on one end engage with female splines in a driving gear; the splines on the other end engage with female splines in the pump drive shaft. Pump drive couplings are designed to serve as safety devices. The shear section of the drive coupling, located midway between the two sets of splines, is smaller in diameter than the splines.

If the pump becomes unusually hard to turn or becomes jammed, this section shears, preventing damage to the pump or driving unit. (**Figure 11-23**) The basic pumping cylinder block, a piston for each bore, and a valve plate with inlet and outlet slots. The purpose of the valve plate slots is to let fluid into and out of the bores as the pump operates. The cylinder bores lie parallel to and symmetrically around the pump axis. All aircraft axial piston pumps have an odd number of pistons. (**Figure 11-24**)

Bent Axis Piston Pump

A typical constant-displacement axial type pump is shown in **Figure 11-25**. The angular housing of the pump causes a corresponding angle to exist between the cylinder block and the drive shaft plate to which the pistons are attached. It is this angular configuration of the pump that causes the pistons to stroke as the pump shaft is turned.

When the pump operates, all parts within the pump (except the outer races of the bearings that support the drive shaft, the cylinder bearing pin on which the cylinder block turns, and the oil seal) turn together as a rotating group. At one point of rotation of the rotating group, a minimum distance exists between the top of the cylinder block and the upper face of the drive shaft plate. Because of the angled housing at a point of rotation 180° away, the distance between the top of the cylinder block and the upper face of the drive shaft plate is at a maximum. At any given moment of operation, three of the pistons are moving away from the top face of the

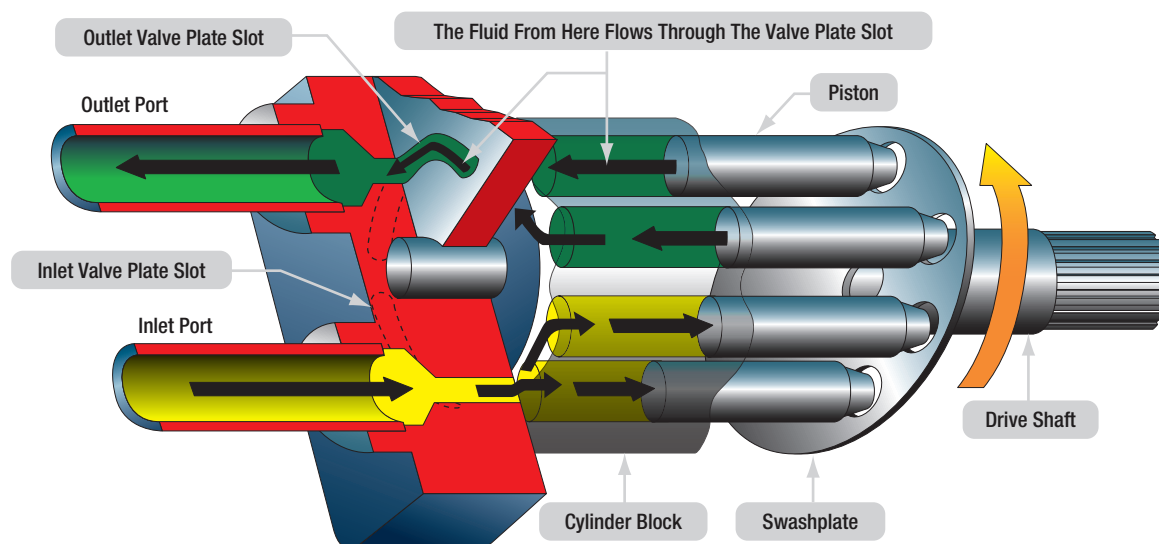


Figure 11-23. Axial inline piston pump.

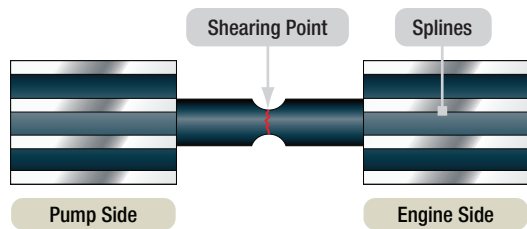


Figure 11-24. Hydraulic pump shear shaft.

cylinder block, producing a partial vacuum in the bores in which these pistons operate. This occurs over the inlet port, so fluid is drawn into these bores at this time. On the opposite side of the cylinder block, three different pistons are moving toward the top face of the block. This occurs while the rotating group is passing over the outlet port causing fluid to be expelled from the pump by these pistons. The continuous and rapid action of the pistons is overlapping in nature and results in a practically non-pulsating pump output.

Inline Piston Pump

The simplest type of axial piston pump is the swash plate design in which a cylinder block is turned by the drive shaft. Pistons fitted to bores in the cylinder block are connected through piston shoes and a retracting ring

so that the shoes bear against an angled swash plate. As the block turns, the piston shoes follow the swash plate, causing the pistons to reciprocate. The ports are arranged in the valve plate so that the pistons pass the inlet as they are pulled out, and pass the outlet as they are forced back in. In these pumps, displacement is determined by the size and number of pistons, as well as their stroke length, which varies with the swash plate angle.

Vane Pump

The vane type power pump is also a constant displacement pump. It consists of a housing containing four vanes (blades), a hollow steel rotor with slots for the vanes, and a coupling to turn the rotor. (*Figure 11-26*)

The rotor is positioned off center within the sleeve. The vanes, which are mounted in the slots in the rotor, together with the rotor, divide the bore of the sleeve into four sections. As the rotor turns, each section passes one point where its volume is at a minimum and another point where its volume is at a maximum. The volume gradually increases from minimum to maximum during the first half of a revolution and gradually decreases from maximum to minimum during the second half of the

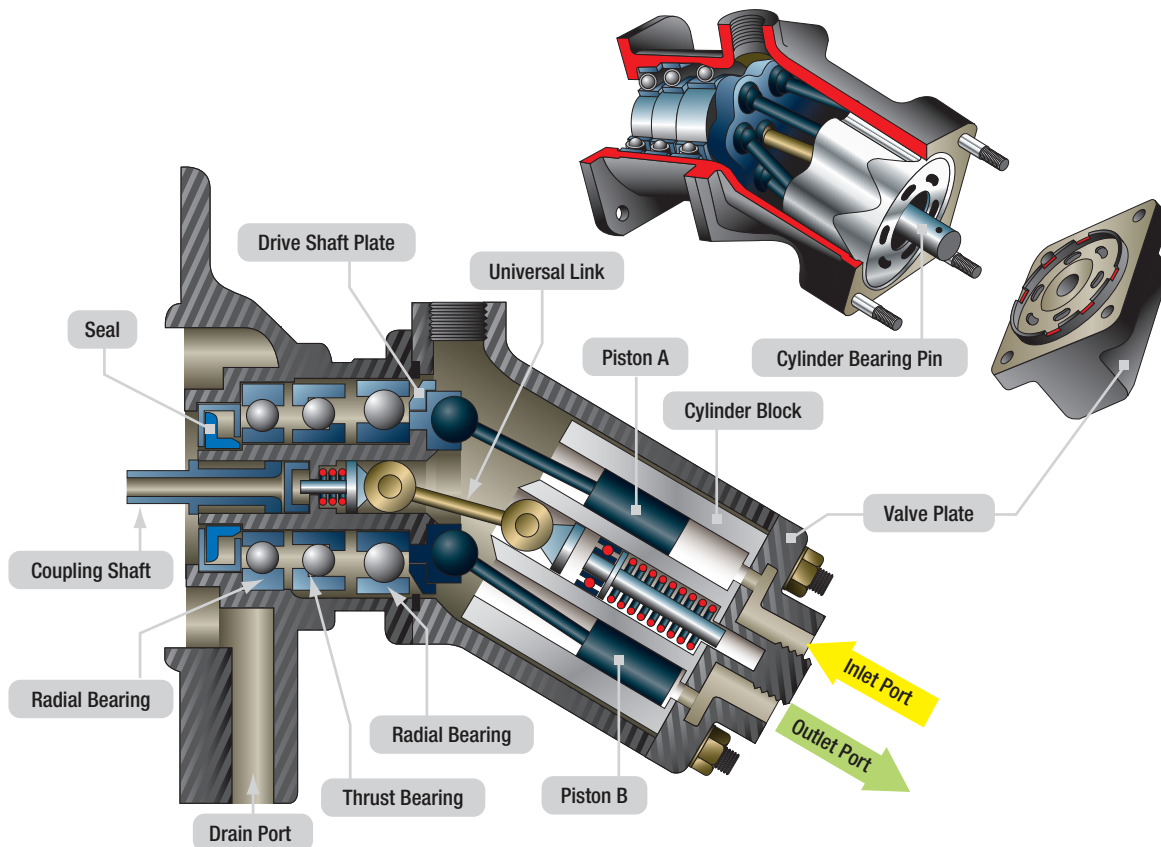


Figure 11-25. Bent axis piston pump.

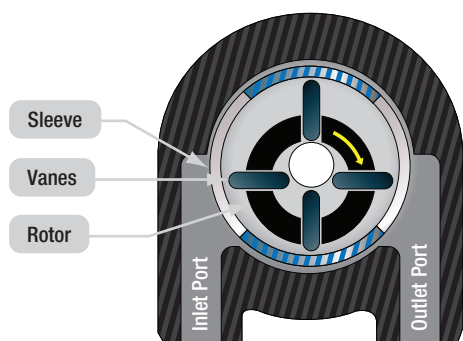


Figure 11-26. Vane-type power pump.

revolution. As the volume of a given section increases, that section is connected to the pump inlet port through a slot in the sleeve. Since a partial vacuum is produced by the increase in volume of the section, fluid is drawn into the section through the pump inlet port and the slot in the sleeve. As the rotor turns through the second half of the revolution and the volume of the given section is decreasing, fluid is displaced out of the section, through the slot in the sleeve aligned with the outlet port, and out of the pump.

VARIABLE DISPLACEMENT PUMPS

A variable displacement pump has a fluid output that is varied to meet the pressure demands of the system. The pump output is changed automatically by a pump compensator within the pump. The following paragraph discusses a two stage Vickers variable displacement pump. The first stage of the pump consists of a centrifugal pump that boosts the pressure before the fluid enters the piston pump. (*Figure 11-27*)

Basic Pumping Operation

The aircraft's engine rotates the pump drive shaft, cylinder block, and pistons via a gearbox. Pumping action is generated by piston shoes that are restrained and slide on the shoe bearing plate in the yoke assembly. Because the yoke is at an angle to the drive shaft, the rotary motion of the shaft is converted to piston reciprocating motion. As the piston begins to withdraw from the cylinder block, system inlet pressure forces fluid through a porting arrangement in the valve plate into the cylinder bore. The piston shoes are restrained in the yoke by a piston shoe retaining plate and a shoe plate during the intake stroke. As the drive shaft continues to turn the cylinder block, the piston shoe continues following the yoke bearing surface. This begins to return the piston into its bore (i.e., toward the valve block).

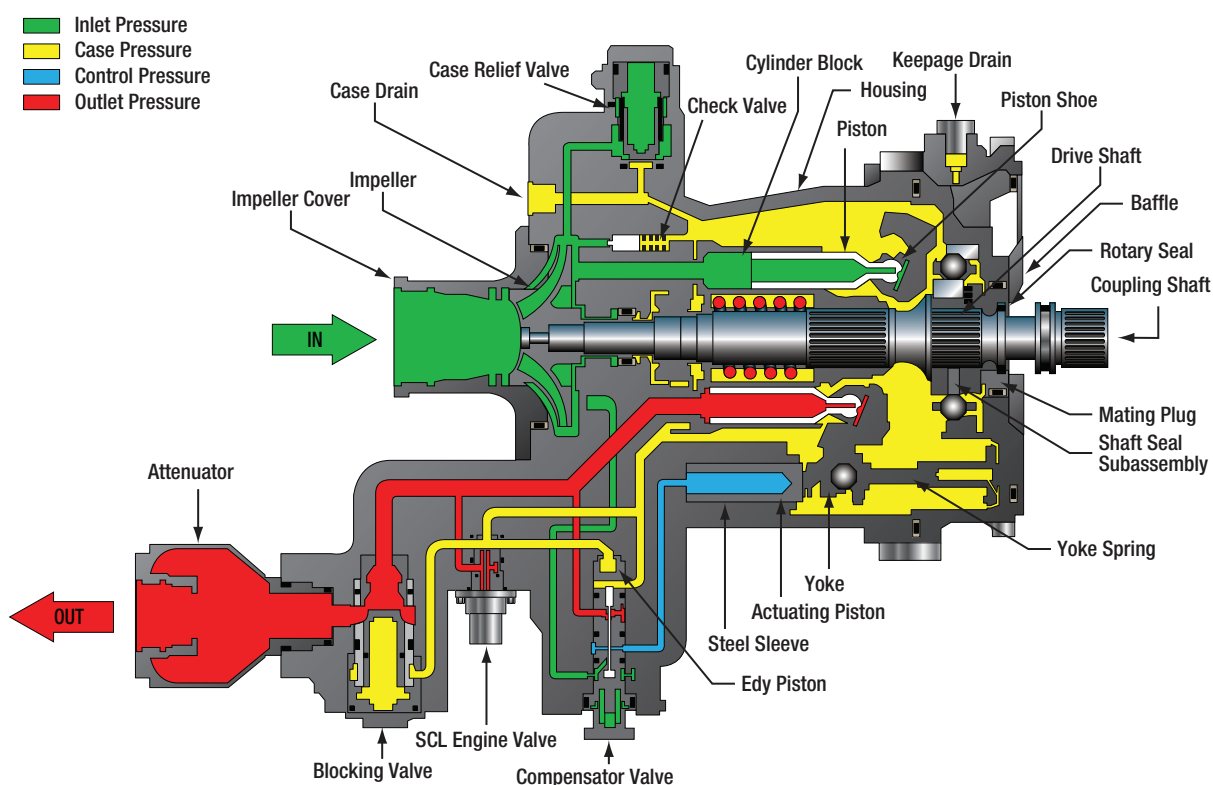


Figure 11-27. Variable displacement pump.

The fluid contained in the bore is precompressed, then expelled through the outlet port. Discharge pressure holds the piston shoe against the yoke bearing surface during the discharge stroke and also provides the shoe pressure balance and fluid film through an orifice in the piston and shoe subassembly. With each revolution of the drive shaft and cylinder block, each piston goes through the pumping cycle described above, completing one intake and one discharge stroke. High-pressure fluid is ported out through the valve plate, past the blocking valve, to the pump outlet. The blocking valve is designed to remain open during normal pump operation. Internal leakage keeps the pump housing filled with fluid for lubrication of rotating parts and cooling. The leakage is returned to the system through a case drain port. The case valve relief valve protects the pump against excessive case pressure, relieving it to the pump inlet.

Normal Pumping Mode

The pressure compensator is a spool valve that is held in the closed position by an adjustable spring load. (**Figure 11-28**) When pump outlet pressure (system pressure) exceeds the pressure setting (2 850 psi for full flow), the spool moves to admit fluid from the pump outlet against the yoke actuator piston. In **Figure 11-28**,

the pressure compensator is shown at cracking pressure; the pump outlet pressure is just high enough to move the spool to begin admitting fluid to the actuator piston.

The yoke is supported inside the pump housing on two bearings. At pump outlet pressures below 2 850 psi, the yoke is held at its maximum angle relative to the drive shaft centerline by the force of the yoke return spring. Decreasing system flow demand causes outlet pressure to become high enough to crack the compensator valve open and admit fluid to the actuator piston.

This control pressure overcomes the yoke return spring force and strokes the pump yoke to a reduced angle. The reduced angle of the yoke results in a shorter stroke for the pistons and reduced displacement. (**Figure 11-29**)

The lower displacement results in a corresponding reduction in pump flow. The pump delivers only that flow required to maintain the desired pressure in the system. When there is no demand for flow from the system, the yoke angle decreases to nearly zero degrees stroke angle. In this mode, the unit pumps only its own internal leakage. Thus, at pump outlet pressures above 2 850 psi, pump displacement decreases as outlet

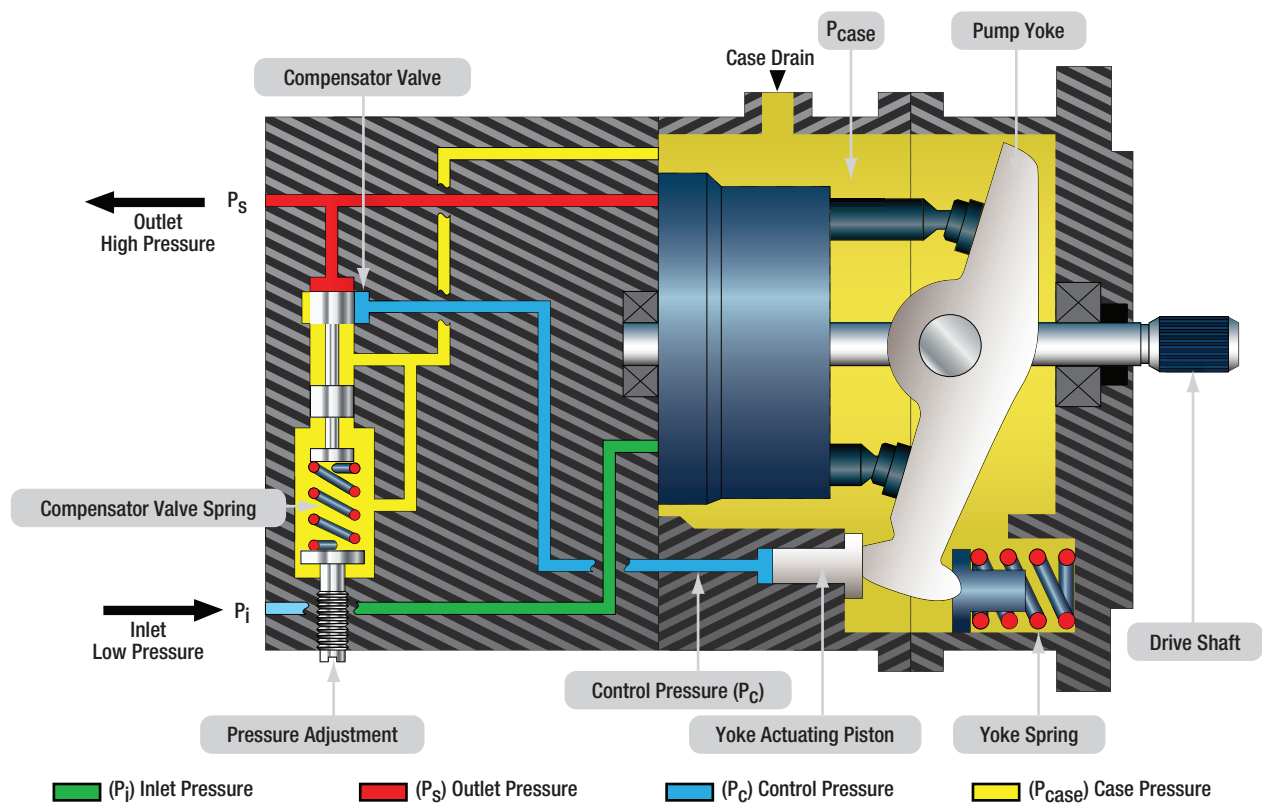


Figure 11-28. Normal pumping mode.

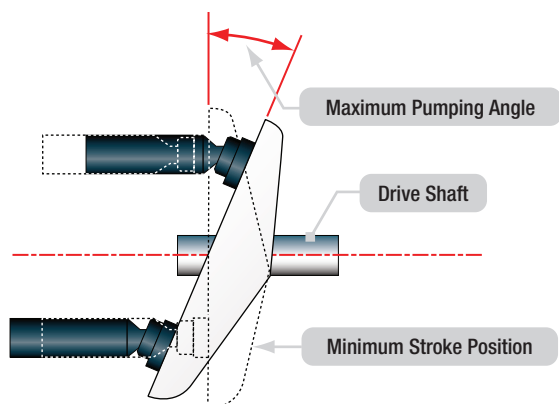


Figure 11-29. Yoke angle.

pressure rises. At system pressures below this level, no fluid is admitted through the pressure compensator valve to the actuator piston and the pump remains at full displacement, delivering full flow. Pressure is then determined by the system demand. The unit maintains zero flow at system pressure of 3 025 psi.

Depressurized Mode

When the solenoid valve is energized, the EDV solenoid valve moves up against the spring force and the outlet fluid is ported to the EDV control piston on the top of the compensator (depressurizing piston). (*Figure 11-30*)

The high-pressure fluid pushes the compensator spool beyond its normal metering position. This removes the compensator valve from the circuit and connects the actuator piston directly to the pump outlet. Outlet fluid is also ported to the blocking valve spring chamber, which equalizes pressure on both sides of its plunger.

The blocking valve closes due to the force of the blocking valve spring and isolates the pump from the external hydraulic system. The pump strokes itself to zero delivery at an outlet pressure that is equal to the pressure required on the actuator piston to reduce the yoke angle to nearly zero, approximately 1 100 psi.

This depressurization and blocking feature can be used to reduce the load on the engine during startup and, in a multiple pump system, to isolate one pump at a time and check for proper system pressure output.

EMERGENCY PRESSURE GENERATION

Generation of hydraulic pressure in emergency situations varies. Large aircraft with multiple hydraulic systems are designed to ensure hydraulic pressure to critical components even in the event of a complete system

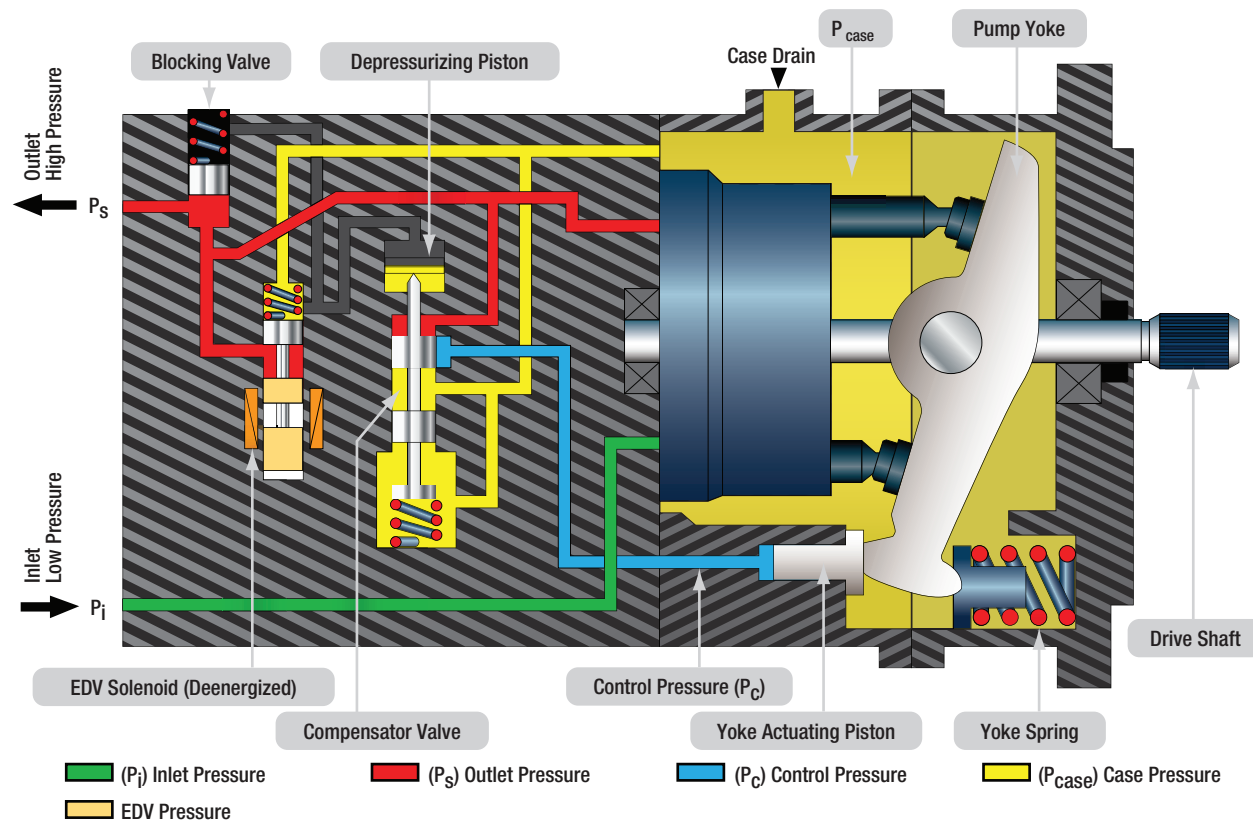


Figure 11-30. Depressurized mode.

failure or loss of engines. Generally, electrically-driven pumps are used when engine-driven pumps fail. Should there be no working engines, not only would the engine driven hydraulic pump be inoperative but electrical generator output would cease as well. This may leave only the aircraft batteries to provide electrical power to the electrically driven pumps. However, it is common for large transport aircraft to be fitted with a ram air turbine (RAT) for yet again an additional source of hydraulic and electric power.

RAM AIR TURBINE (RAT)

A RAT is installed in the aircraft to provide electrical and hydraulic power if the primary sources of aircraft power are lost. Ram air is used to turn the blades of a turbine that, in turn, operates a hydraulic pump and generator. The turbine and pump assembly is generally installed on the inner surface of a door installed in the fuselage.

The door is hinged, allowing the assembly to be extended into the slipstream by pulling a manual release in the flight deck. In some aircraft, the RAT automatically deploys when the main hydraulic pressure system fails and/or electrical system malfunction occurs. (Figure 11-31)

HYDRAULIC MOTORS

Just as a rotating shaft drives a hydraulic pump workings to move fluid, fluid forced through the pump can rotate the attached shaft. This is the principle behind a hydraulic motor. Hydraulic fluid forced through the

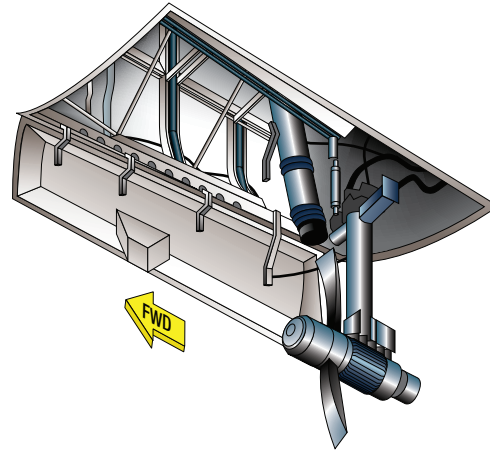


Figure 11-31. Ram air turbine.

pump rotates the shaft of the pump, which as a result, makes the pump a motor. The motion of the shaft is then used to drive something to which it is attached. Piston type motors are the most commonly used in hydraulic systems. (Figure 11-32)

They are basically the same as hydraulic pumps except they are used to convert hydraulic energy into mechanical (rotary) energy. Hydraulic motors are either of the axial inline or bent-axis type.

The most commonly used hydraulic motor is the fixed displacement bent axis type. These types of motors are used for the activation of trailing edge flaps, leading edge slats, and stabilizer trim. Some equipment uses a variable-displacement piston motor where very wide speed ranges are desired. Although some piston type hydraulic motors are controlled by directional control

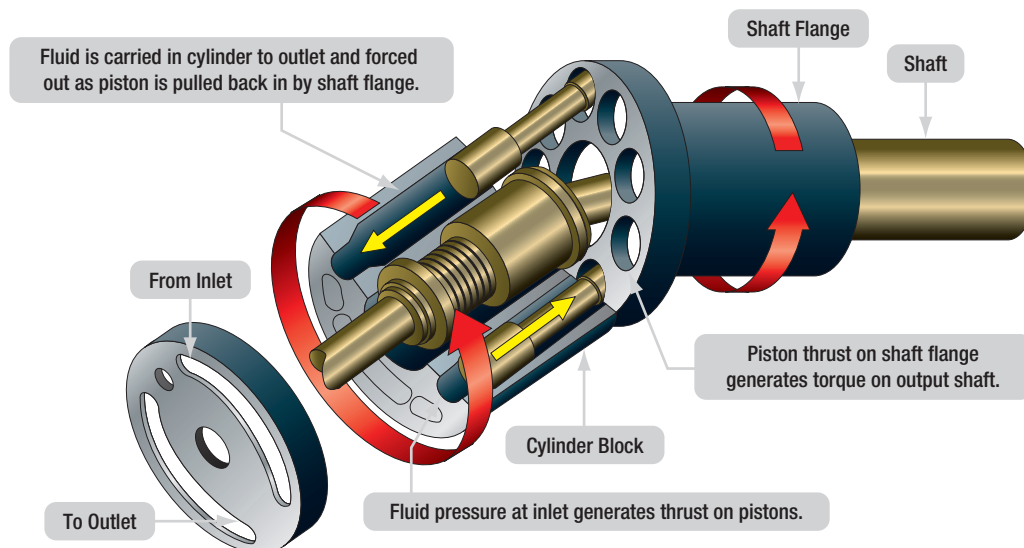


Figure 11-32. Bent axis piston motor.

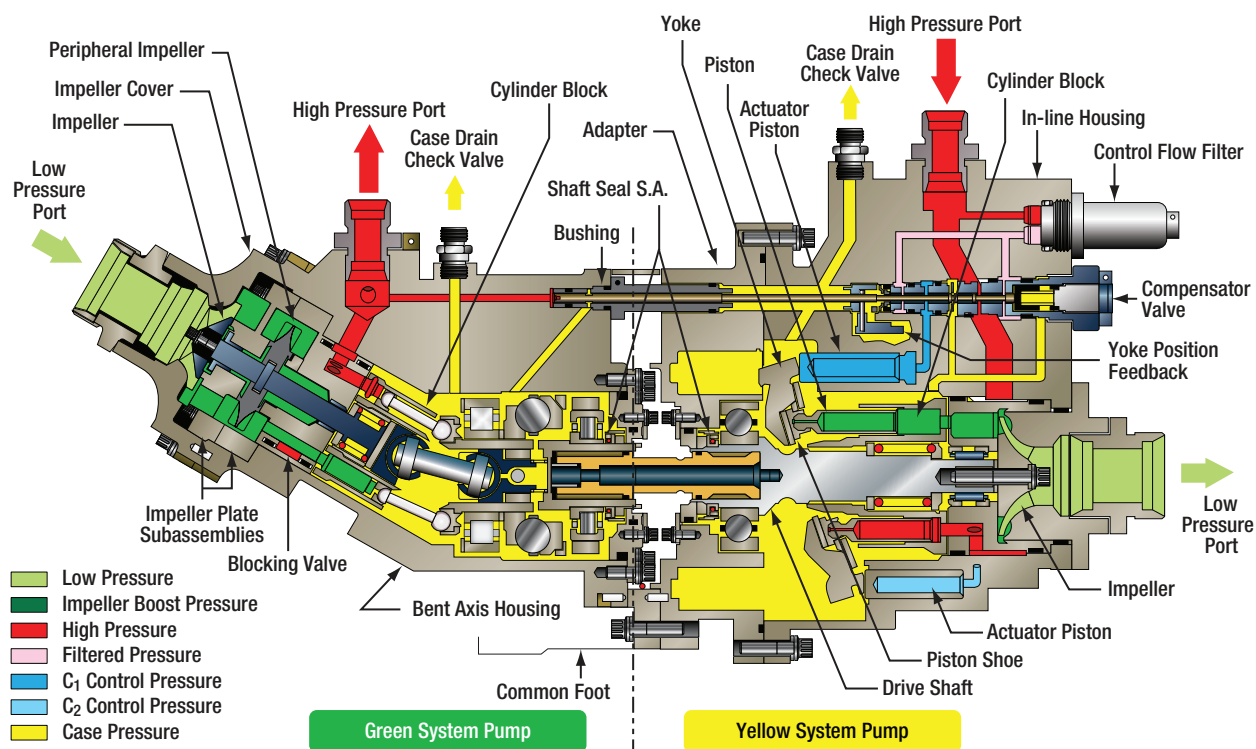


Figure 11-33. Power transfer unit.

valves, they are often used in combination with variable displacement pumps. This pump-motor combination is used to provide a transfer of power between a driving element and a driven element. Some applications for which hydraulic transmissions may be used are speed reducers, variable speed drives, constant speed or constant torque drives, and torque converters. Some advantages of hydraulic transmission of power over mechanical transmission of power are as follows:

- Quick, easy speed adjustment over a wide range while the power source is operating at a constant (most efficient) speed.
- Rapid, smooth acceleration or deceleration.
- Control over maximum torque and power.
- Cushioning effect to reduce shock loads.
- Smoother reversal of motion.

POWER TRANSFER UNITS (PTUS)

Hydraulic motors are also used in power transfer units (PTUs). In a PTU, two units, a hydraulic pump and hydraulic motor, are connected via a single drive shaft so that power can be transferred between two hydraulic systems. Depending on the direction of power transfer, each unit works as either a motor or a pump. The pressurized hydraulic system forces fluid through the motor which turns the shaft of the pump that moves fluid through the second hydraulic system. Thus, power

is transferred from one system to the other. While the PTU transfers power, it does not transfer any fluid from one system to the other. (*Figure 11-33*)

Different types of PTUs are in use; some can only transfer power in one direction while others can transfer power both ways. Some PTUs have a fixed displacement, while others use a variable displacement hydraulic pump. Regardless, the application of PTU's in aircraft allow component operation in a hydraulic system in which the pump has failed. The system with a working pump drives the motor of the PTU so that the pump shaft rotates in the system with the failed pump. Activation can be manual or automatic depending on the aircraft. In an automatically activated setup, a pressure switch is used to detect pump failure which opens a valve to divert fluid from the healthy system to the PTU.

HYDRAULIC MOTOR DRIVEN GENERATORS (HMDGS)

In case of an electrical failure, a hydraulic motor driven generator can be employed. An HMDG provides an alternative source of electrical power. The servo controlled variable displacement HMDG is an AC generator driven by the hydraulic motor portion of the unit. The generator part is typically designed to maintain the desired system output frequency of 400 Hz. Thus, an

aircraft with an HMDG can maintain electrical power should a primary generator fail through the use of the hydraulic system. Conversely, a hydraulic pump failure is backed up by an electrically driven hydraulic pump.

PRESSURE CONTROL

The safe and efficient operation of fluid power systems, system components, and related equipment requires a means of controlling pressure. There are many types of automatic pressure control valves designed for this purpose. Some of them are an escape for pressure that exceeds a set pressure; some only reduce the pressure to a lower pressure system or subsystem; and some keep the pressure in a system within a required range.

RELIEF VALVES

Hydraulic pressure must be regulated in order to use it to perform the desired tasks. A pressure relief valve is used to limit the amount of pressure being exerted on a confined liquid. This is necessary to prevent failure of components or rupture of hydraulic lines under excessive pressures. The pressure relief valve is, in effect, a system safety valve. The design of pressure relief valves incorporates adjustable spring loaded valves. They are installed in such a manner as to discharge fluid from the pressure line into a reservoir return line when the pressure exceeds the predetermined maximum for which the valve is adjusted. Various makes and designs of pressure relief valves are in use, but, in general, they all employ a spring loaded valving device operated by hydraulic pressure and spring tension. (*Figure 11-34*)

Pressure relief valves are adjusted by increasing or decreasing the tension on the spring to determine the pressure required to open the valve. They may be classified by type of construction or uses in the system. The most common types of valve are:

1. Ball type - in pressure relief valves with a ball type valving device, the ball rests on a contoured seat. Pressure acting on the bottom of the ball pushes it off its seat, allowing the fluid to bypass.
2. Sleeve type - in pressure relief valves with a sleeve type valving device, the ball remains stationary and a sleeve type seat is moved up by the fluid pressure. This allows the fluid to bypass between the ball and the sliding sleeve type seat.
3. Poppet type - in pressure relief valves with a poppet type valving device, a cone shaped poppet may have any of several design configurations;

however, it is basically a cone and seat machined at matched angles to prevent leakage. As the pressure rises to its predetermined setting, the poppet is lifted off its seat, as in the ball type device. This allows the fluid to pass through the opening created and out the return port.

Pressure relief valves cannot be used as pressure regulators in large hydraulic systems that depend on engine-driven pumps for the primary source of pressure because the pump is constantly under load and the energy expended in holding the pressure relief valve off its seat is changed into heat. This heat is transferred to the fluid and, in turn, to the packing rings, causing them to deteriorate rapidly. Pressure relief valves, however, may be used as pressure regulators in small, low-pressure systems or when the pump is electrically driven and is used intermittently.

Pressure relief valves may be used as:

1. System relief valve - the most common use of the pressure relief valve is as a safety device against the possible failure of a pump compensator or other pressure regulating device. All hydraulic systems that have hydraulic pumps incorporate pressure relief valves as safety devices.
2. Thermal relief valve - the pressure relief valve is used to relieve excessive pressures that may exist due to thermal expansion of the fluid. They are used where a check valve or selector valve

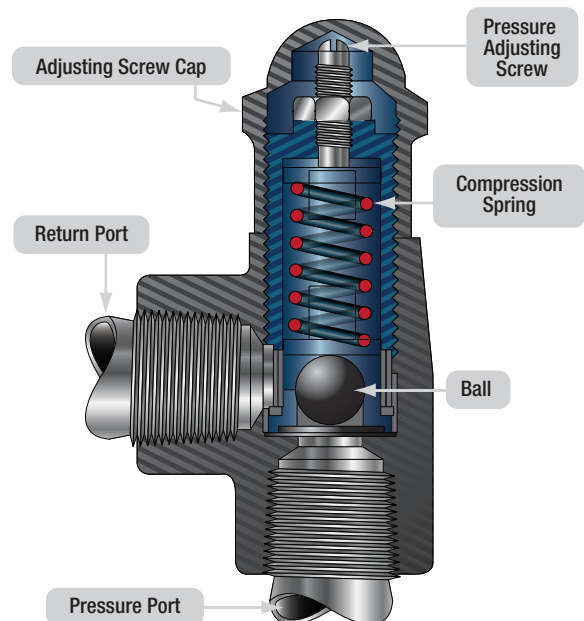


Figure 11-34. Pressure relief valve.

prevents pressure from being relieved through the main system relief valve. Thermal relief valves are usually smaller than system relief valves. As pressurized fluid in the line in which it is installed builds to an excessive amount, the valve poppet is forced off its seat. This allows excessive pressurized fluid to flow through the relief valve to the reservoir return line. When system pressure decreases to a predetermined pressure, spring tension overcomes system pressure and forces the valve poppet to the closed position.

PRESSURE REGULATORS

The term pressure regulator is applied to a device used in hydraulic systems that are pressurized by constant delivery type pumps. One purpose of the pressure regulator is to manage the output of the pump to maintain system operating pressure within a predetermined range. The other purpose is to permit the pump to turn without resistance (termed unloading the pump) at times when pressure in the system is within normal operating range. The pressure regulator

is located in the system so that pump output can get into the system pressure circuit only by passing through the regulator. The combination of a constant-delivery type pump and the pressure regulator is virtually the equivalent of a compensator controlled, variable-delivery type pump. (*Figure 11-35*)

PRESSURE REDUCERS

Pressure reducing valves are used in hydraulic systems where it is necessary to lower the normal system operating pressure by a specified amount. Pressure reducing valves provide a steady pressure into a system that operates at a lower pressure than the supply system. A reducing valve can normally be set for any desired downstream pressure within the design limits of the valve. Once the valve is set, the reduced pressure is maintained regardless of changes in supply pressure (as long as the supply pressure is at least as high as the reduced pressure desired) and regardless of the system load, if the load does not exceed the designed capacity of the reducer. (*Figure 11-36*)

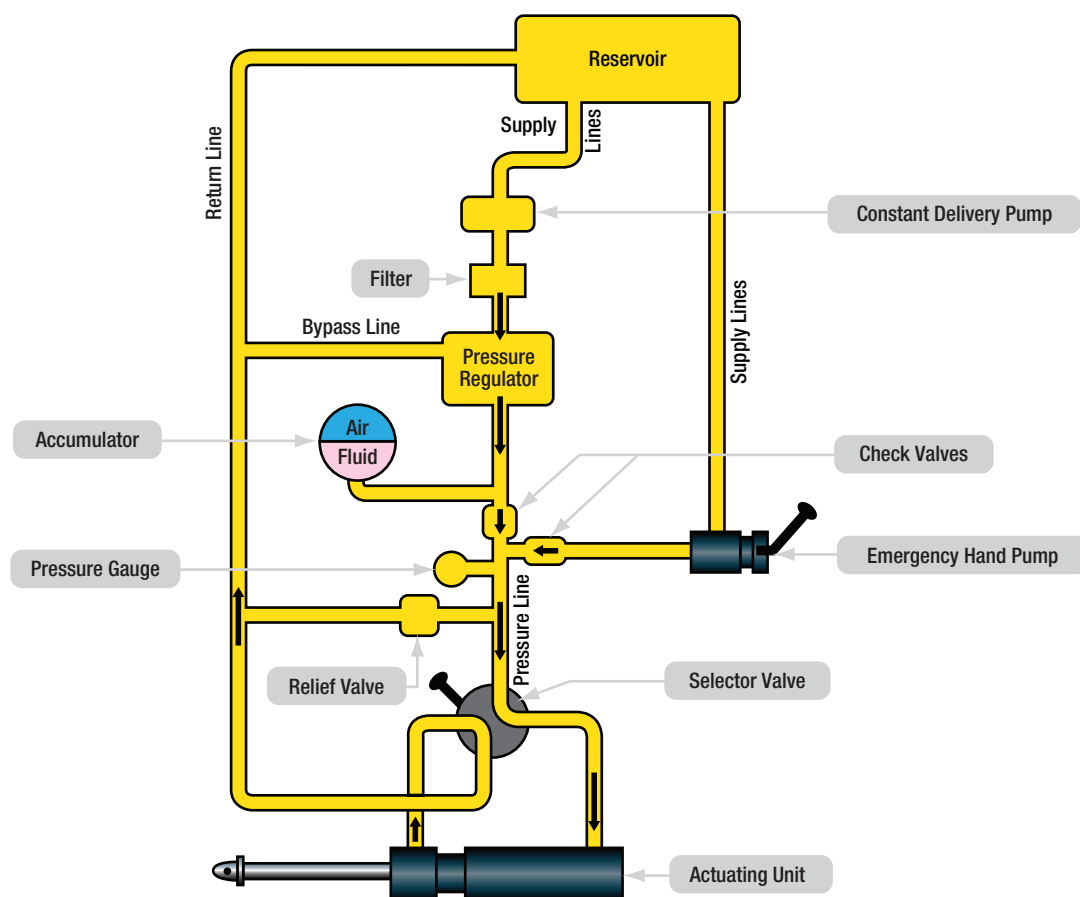


Figure 11-35. The location of a pressure regulator in a basic hydraulic system. The regulator unloads the constant delivery pump by bypassing fluid to the return line when the predetermined system pressure is reached.

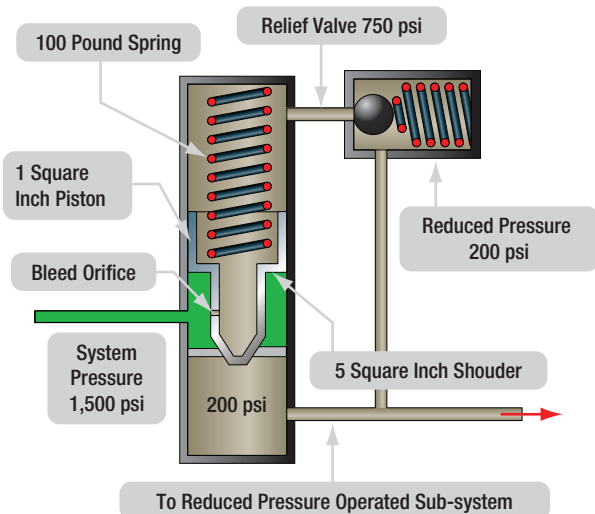


Figure 11-36. Operating mechanism of a pressure reducing valve.

POWER DISTRIBUTION

Power distribution in a hydraulic system is controlled through the use of variety of flow control valves. These valves control the speed and/or direction of fluid flow in the hydraulic system. They provide for the operation of various components when desired and the speed at which the component operates. Examples of flow control valves include: selector valves, check valves, sequence valves, priority valves, shuttle valves, quick disconnect valves, hydraulic fuses and shutoff valves.

SHUTOFF VALVES

Shutoff valves are used to shutoff the flow of fluid to a particular system, sub-system or component. In general, these types of valves are electrically powered. They are used to distribute hydraulic power to various components in the system. (*Figure 11-37*)

A shutoff valve may be used to create a priority in a hydraulic system. In this case, the valve is controlled by a pressure switch. The switch is adjusted so that if pressure drops below the set point, the valve closes. All components upstream of the shutoff valve are prioritized in that they receive system fluid pressure even when full pressure is not available.

SELECTOR VALVES

A selector valve is used to control the direction of movement of a hydraulic actuating cylinder or similar device. It provides for the simultaneous flow of hydraulic fluid both into and out of the unit. Hydraulic system pressure can be routed with the selector valve to operate the unit in either direction and a corresponding return



Figure 11-37. Shutoff valves.

path for the fluid to the reservoir is provided. There are two main types of selector valves: open-center and closed-center. An open center valve allows a continuous flow of system hydraulic fluid through the valve even when the selector is not in a position to actuate a unit. A closed-center selector valve blocks the flow of fluid through the valve when it is in the NEUTRAL or OFF position. (*Figure 11-38A*)

Selector valves may be poppet type, spool type, piston type, rotary type, or plug type. (*Figure 11-39*) Regardless, each selector valve has a unique number of ports. The number of ports is determined by the particular requirements of the system in which the valve is used. Closed-centered selector valves with four ports are most common in aircraft hydraulic systems. These are known as four-way valves.

Figure 11-38 illustrates how this valve connects to the pressure and return lines of the hydraulic system, as well as to the two ports on a common actuator. Most selector valves are mechanically controlled by a lever or electrically controlled by solenoid or servo. (*Figure 11-40*) The four ports on a four-way selector valve always have the same function. One port receives pressurized fluid from the system hydraulic pump. A second port always returns fluid to the reservoir. The third and fourth ports are used to connect the selector valve to the actuating unit. There are two ports on the actuating unit. When the selector valve is positioned to connect pressure to one port on the actuator, the other actuator port is simultaneously connected to the reservoir return line through selector valve. (*Figure 11-38B*)

Thus, the unit operates in a certain direction. When the selector valve is positioned to connect pressure to the other port on the actuating unit, the original port

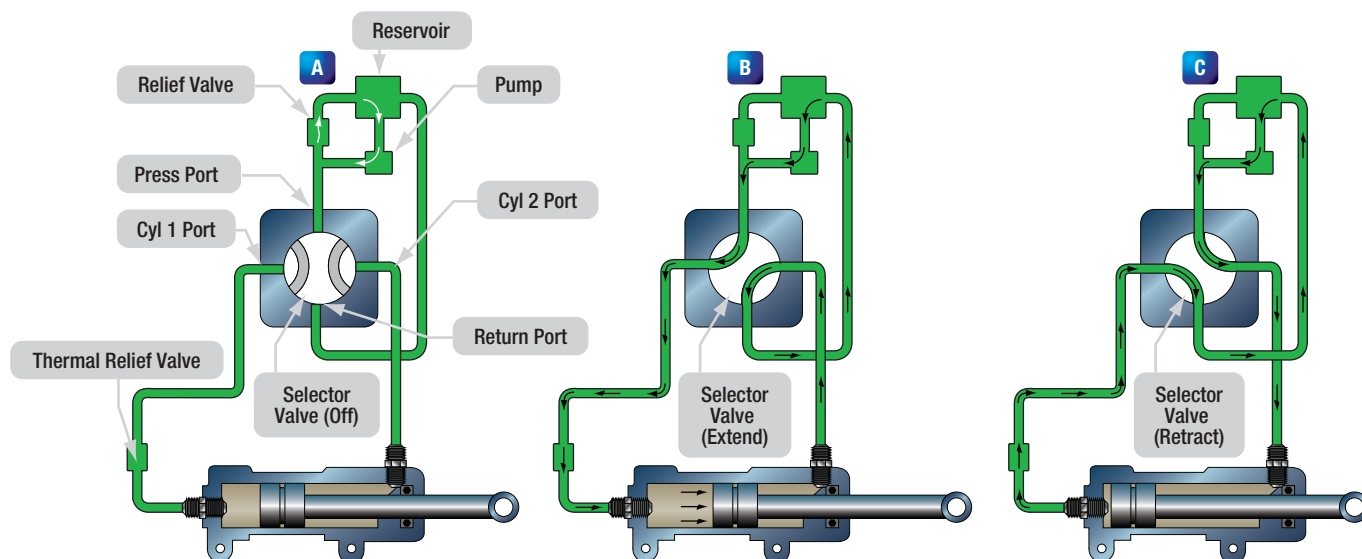


Figure 11-38. Operation of a closed-center four-way selector valve, which controls an actuator.

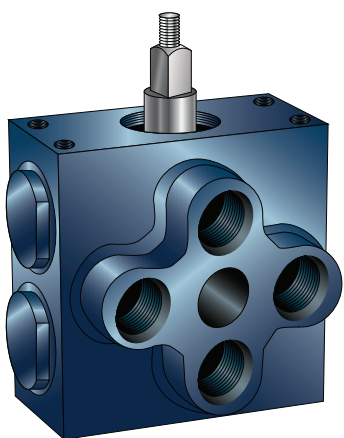


Figure 11-39. A poppet-type four-way selector valve.

is simultaneously connected to the return line through the selector valve and the unit operates in the opposite direction. (*Figure 11-38C*)

Figure 11-41 illustrates the internal flow paths of a solenoid operated selector valve. The closed center valve is shown in the NEUTRAL or OFF position. Neither solenoid is energized. The pressure port routes fluid to the center lobe on the spool, which blocks the flow. Fluid pressure flows through the pilot valves and applies equal pressure on both ends of the spool. The actuator lines are connected around the spool to the return line.

When selected via a switch in the cockpit, the right solenoid is energized. The right pilot valve plug shifts left, which blocks pressurized fluid from reaching the right end of the main spool. The spool slides to the right due to greater pressure applied on the left end of



Figure 11-40. Four-way servo control valve.

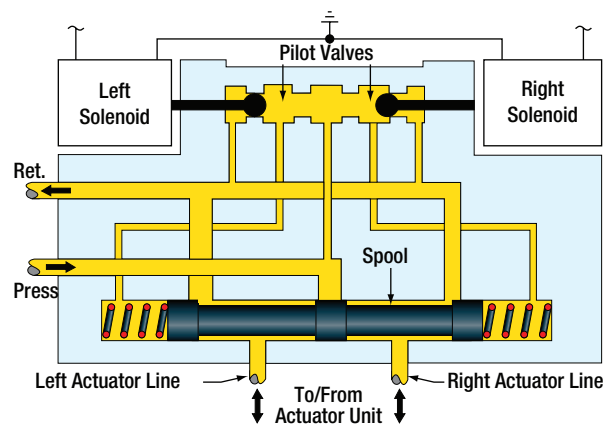


Figure 11-41. Servo control valve solenoids not energized.

the spool. The center lobe of the spool no longer blocks system pressurized fluid, which flows to the actuator through the left actuator line. At the same time, return flow is blocked from the main spool left chamber so the actuator (not shown) moves in the selected direction.

Return fluid from the moving actuator flows through the right actuator line past the spool and into the return line. (*Figure 11-42*)

Typically, the actuator or moving device contacts a limit switch when the desired motion is complete. The switch causes the right solenoid to de-energize and the right pilot valve reopens. Pressurized fluid can once again flow through the pilot valve and into the main spool right end chamber. There, the spring and fluid pressure shift the spool back to the left into the NEUTRAL or OFF position shown in *Figure 11-41*.

To make the actuator move in the opposite direction, the cockpit switch is moved in the opposite direction. All motion inside the selector valve is the same as described above but in the opposite direction. The left solenoid is energized. Pressure is applied to the actuator through the right port and return fluid from the left actuator line is connected to the return port through the motion of the spool to the left.

CHECK VALVES

Another common flow control valve in aircraft hydraulic systems is the check valve. A check valve allows fluid to flow unimpeded in one direction, but prevents or restricts fluid flow in the opposite direction. A check valve may be an independent component situated in-line somewhere in the hydraulic system or it may be built-in to a component. When part of a component, the check valve is said to be an integral check valve.

A typical check valve consists of a spring loaded ball and seat inside a housing. The spring compresses to allow fluid flow in the designed direction. When flow stops, the spring pushes the ball against the seat which prevents fluid from flowing in the opposite direction through the valve. An arrow on the outside of the housing indicated the direction in which fluid flow is permitted. (*Figure 11-43*) A check valve may also be constructed with spring loaded flapper or coned shape piston instead of a ball.

ORIFICE TYPE CHECK VALVE

Some check valves allow full fluid flow in one direction and restricted flow in the opposite direction. These are known as orifice type check valves, or damping valves. The valve contains the same spring, ball, and seat combination as a normal check valve but the seat area has a calibrated orifice machined into it. Thus fluid flow is unrestricted in the designed direction while the ball is

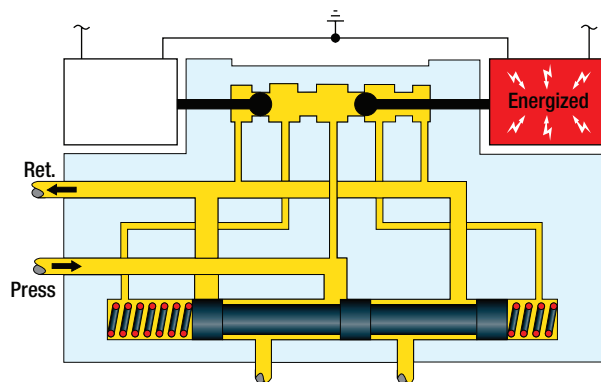


Figure 11-42. Servo control valve right solenoid energized.

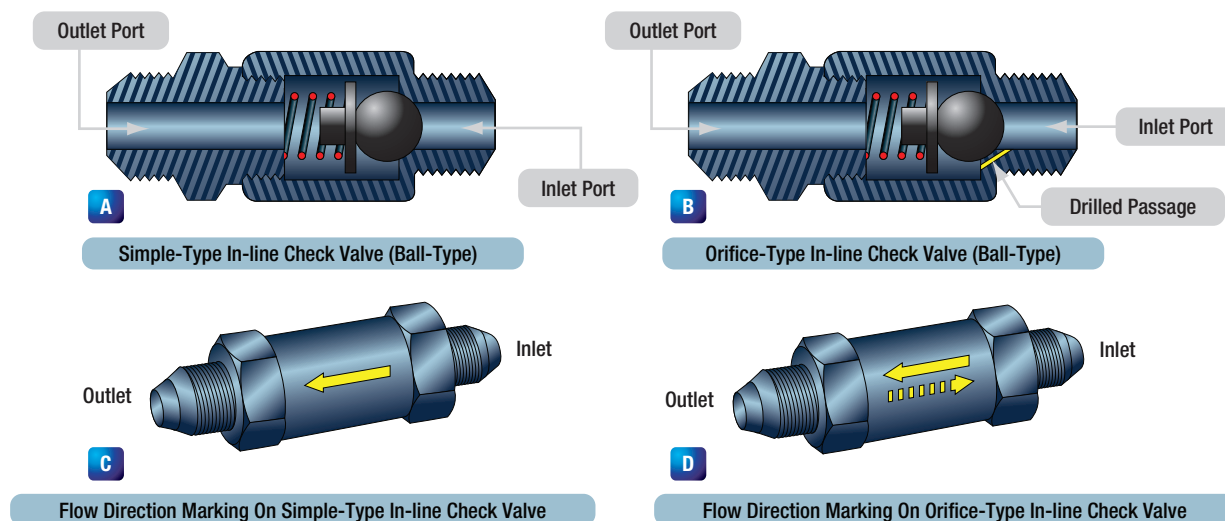


Figure 11-43. An in-line check valve and orifice type in-line check valve.

pushed off of its seat. The downstream actuator operates at full speed. When fluid back flows into the valve, the spring forces the ball against the seat which limits fluid flow to the amount that can pass through the orifice. The reduced flow in this opposite direction slows the motion, or dampens, the actuator associated with the check valve. (*Figure 11-43*)

An orifice check valve may be included in a hydraulic landing gear actuator system. When the gear is raised, the check valve allows full fluid flow to lift the heavy gear at maximum speed. When lowering the gear, the orifice in the check valve prevents the gear from violently dropping by restricting fluid flow out of the actuating cylinder.

SEQUENCE VALVES

Sequence valves control the sequence of operation between two branches in a circuit; they enable one unit to automatically set another unit into motion. An example of the use of a sequence valve is in an aircraft landing gear actuating system. In a landing gear actuating system, the landing gear doors must open before the landing gear starts to extend. Conversely, the landing gear must be completely retracted before the doors close. A sequence valve installed in each landing gear actuating line performs this function. A sequence valve is somewhat similar to a relief valve except that, after the set pressure has been reached, the sequence valve diverts the fluid to a second actuator or motor to do work in another part of the system. There are various types of sequence valves. Some are controlled by pressure, some are controlled mechanically, and some are controlled by electric switches.

PRESSURE CONTROLLED SEQUENCE VALVE

The operation of a typical pressure controlled sequence valve is illustrated in *Figure 11-41*. The opening pressure is obtained by adjusting the tension of the spring that normally holds the piston in the closed position. (Note that the top part of the piston has a larger diameter than the lower part.) Fluid enters the valve through the inlet port, flows around the lower part of the piston and exits the outlet port, where it flows to the primary (first) unit to be operated. (*Figure 11-44A*) This fluid pressure also acts against the lower surface of the piston.

When the primary actuating unit completes its operation, pressure in the line to the actuating unit increases sufficiently to overcome the force of the spring, and the piston rises. The valve is then in the open position. (*Figure 11-44B*) The fluid entering the valve takes the path of least resistance and flows to the secondary unit. A drain passage is provided to allow any fluid leaking past the piston to flow from the top of the valve. In hydraulic systems, this drain line is usually connected to the main return line.

MECHANICALLY OPERATED SEQUENCE VALVE

The mechanically operated sequence valve is operated by a plunger that extends through the body of the valve. (*Figure 11-45*) The valve is mounted so that the plunger is operated by the primary unit. A check valve, either a ball or a poppet, is installed between the fluid ports in the body. It can be unseated by either the plunger or fluid pressure. Port A and the actuator of the primary unit are connected by a common line. Port B is connected by a line to the actuator of the secondary unit.

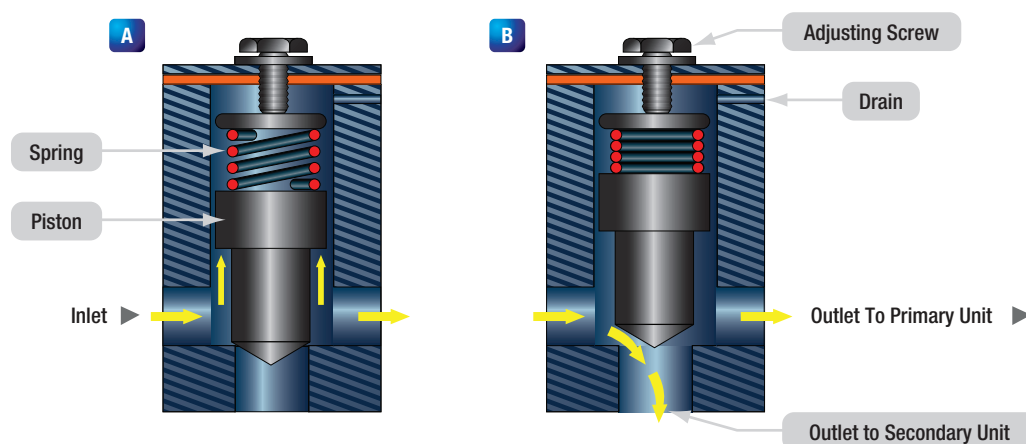


Figure 11-44. A pressure-controlled sequence valve.

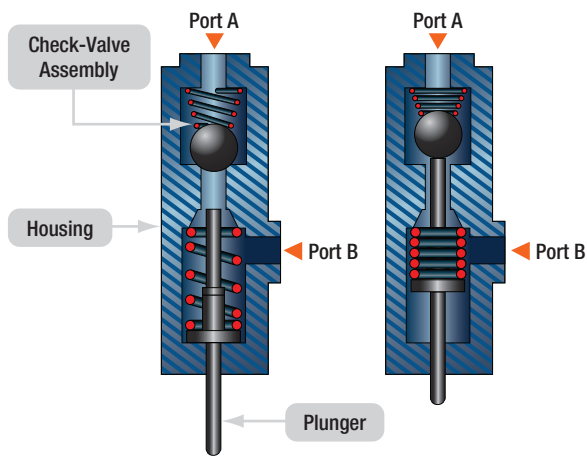


Figure 11-45. Mechanically operated sequence valve.

When fluid under pressure flows to the primary unit, it also flows into the sequence valve through port A to the seated check valve in the sequence valve. In order to operate the secondary unit, the fluid must flow through the sequence valve. The valve is located so that the primary unit moves the plunger as it completes its operation. The plunger unseats the check valve and allows the fluid to flow through the valve, out port B, and to the secondary unit.

PRIORITY VALVES

A priority valve gives priority to the critical hydraulic subsystems over noncritical systems when system pressure is low. For instance, if the pressure of the priority valve is set for 2 200 psi, all systems receive pressure when the pressure is above 2 200 psi. If the pressure drops below 2 200 psi, the priority valve closes and no fluid pressure flows to the noncritical systems. (Figure 11-46) Some hydraulic designs use pressure switches and electric shutoff valves to assure that the critical systems have priority over noncritical systems when system pressure is low.

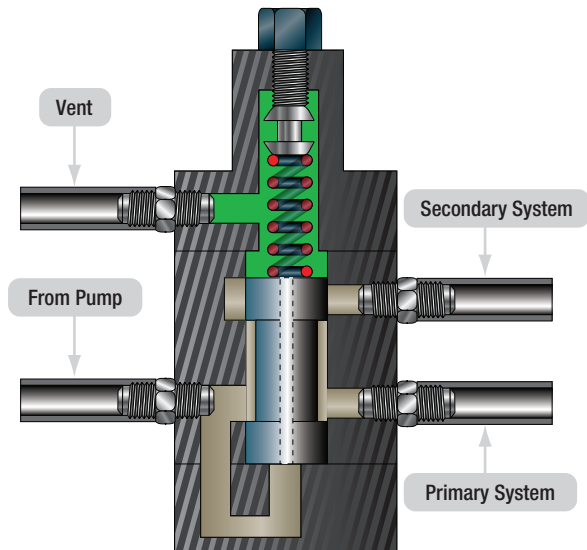


Figure 11-46. Priority valve.

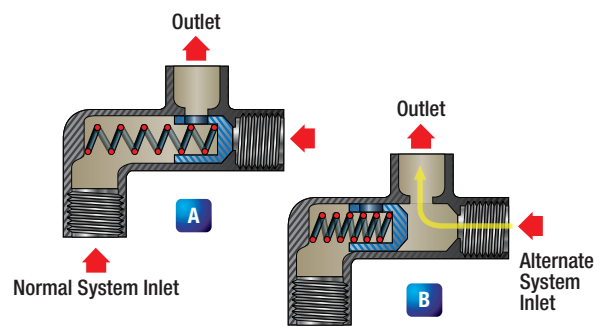


Figure 11-47. A spring-loaded piston-type shuttle valve in normal configuration (A) and with alternate/emergency supply (B).

The housing contains three ports - normal system inlet, alternate or emergency system inlet, and outlet. A shuttle valve used to operate more than one actuating unit may contain additional unit outlet ports.

SHUTTLE VALVES

In certain fluid power systems, the supply of fluid to a subsystem must be from more than one source to meet system requirements. In some systems, an emergency system is provided as a source of pressure in the event of normal system failure. The emergency system usually actuates only essential components. The main purpose of the shuttle valve is to isolate the normal system from an alternate or emergency system. It is small and simple; yet, it is a very important component. (Figure 11-47)

Enclosed in the housing is a sliding part called the shuttle. Its purpose is to seal off one of the inlet ports. There is a shuttle seat at each inlet port. When a shuttle valve is in the normal operation position, fluid has a free flow from the normal system inlet port, through the valve, and out through the outlet port to the actuating unit. The shuttle is seated against the alternate system inlet port and held there by normal system pressure and by the shuttle valve spring. The shuttle remains in this position until the alternate system is activated. This action directs fluid under pressure from the alternate system to the shuttle valve and forces the shuttle from the alternate system inlet port to the normal system inlet port. Fluid from the alternate system then has a free flow to the outlet port, but is prevented from entering the

normal system by the shuttle, which seals off the normal system port. The shuttle may be one of four types:

1. Sliding plunge.
2. Spring-loaded piston.
3. Spring-loaded ball.
4. Spring-loaded poppet.

In shuttle valves that are designed with a spring, the shuttle is normally held against the alternate system inlet port by the spring.

QUICK DISCONNECT VALVES

Quick disconnect valves are installed in hydraulic lines to prevent loss of fluid when units are removed. Such valves are installed in the pressure and suction lines of the system immediately upstream and downstream of the power pump. In addition to pump removal, a power pump can be disconnected from the system and a hydraulic test stand connected in its place. These valve units consist of two interconnecting sections coupled together by a nut when installed in the system. Each valve section has a piston and poppet assembly. These are spring loaded to the closed position when the unit is disconnected. (*Figure 11-48*)

HYDRAULIC FUSES

A hydraulic fuse is a safety device. Fuses may be installed at strategic locations throughout a hydraulic system. They detect a sudden increase in flow, such as a burst downstream, and shut off the fluid flow.

By closing, a fuse preserves hydraulic fluid for the rest of the system. Hydraulic fuses are fitted to the brake system, leading edge flap and slat extend and retract

lines, nose landing gear up and down lines, and the thrust reverser pressure and return lines. One type of fuse, referred to as the automatic resetting type, is designed to allow a certain volume of fluid per minute to pass through it. If the volume passing through the fuse becomes excessive, the fuse closes and shuts off the flow. When the pressure is removed from the pressure supply side of the fuse, it automatically resets itself to the open position. Fuses are usually cylindrical in shape, with an inlet and outlet port at opposite ends. (*Figure 11-49*)

OTHER HYDRAULIC SYSTEM COMPONENTS

HYDRAULIC ACTUATORS

A hydraulic actuating cylinder transforms energy in the form of fluid pressure into mechanical force, or action, to perform work. It is used to impart powered linear motion to some movable object or mechanism. A typical actuating cylinder consists of a cylinder housing, one or more pistons and piston rods, and some seals. The cylinder housing contains a polished bore in which the piston operates, and one or more ports through which fluid enters and leaves the bore. The piston and rod form an assembly. The piston moves forward and backward within the cylinder bore, and an attached piston rod moves into and out of the cylinder housing through an opening in one end of the cylinder housing.

Seals are used to prevent leakage between the piston and the cylinder bore and between the piston rod and the end of the cylinder. Both the cylinder housing and the piston rod have provisions for mounting and for attachment to an object or mechanism that is to be moved by the actuating cylinder.

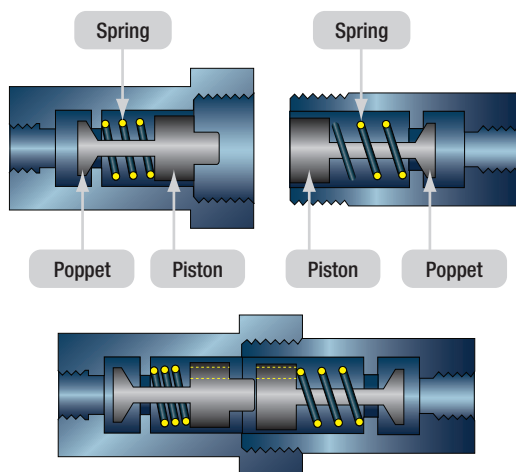


Figure 11-48. A hydraulic quick-disconnect valve.



Figure 11-49. Hydraulic fuse.

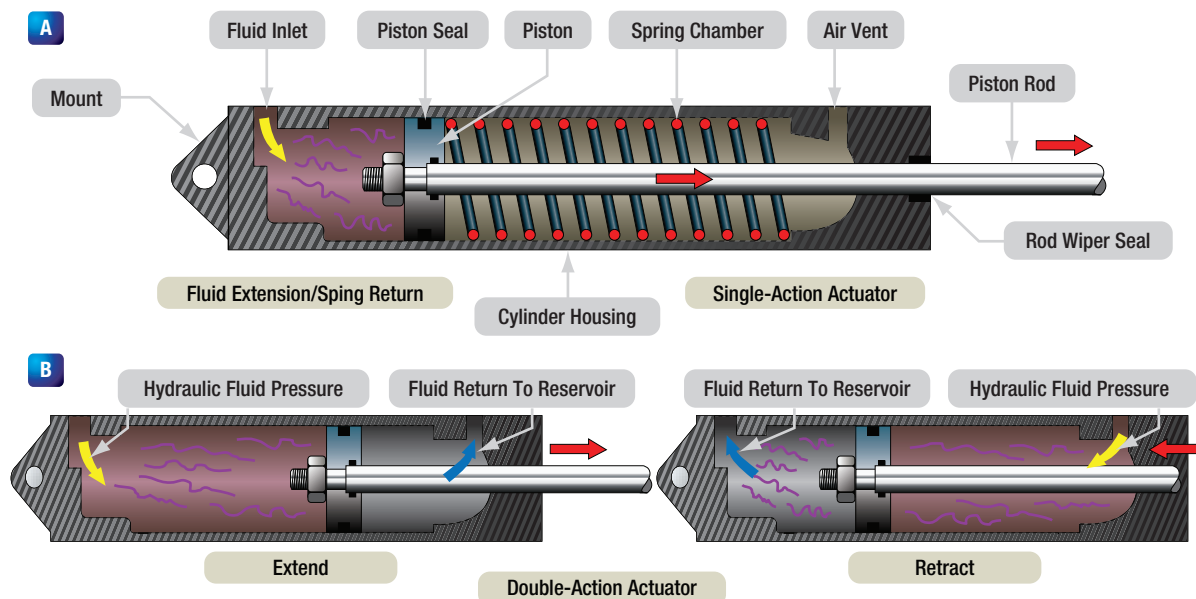


Figure 11-50. Linear actuator.

Actuating cylinders are of two major types: single action and double action. The single-action (single port) actuating cylinder is capable of producing powered movement in one direction only. The double-action (two ports) actuating cylinder is capable of producing powered movement in two directions.

LINEAR ACTUATORS

A single-action actuating cylinder is illustrated in **Figure 11-50A**. Fluid under pressure enters the port at the left and pushes against the face of the piston, forcing the piston to the right. As the piston moves, air is forced out of the spring chamber through the vent hole, compressing the spring. When pressure on the fluid is released to the point it exerts less force than is present in the compressed spring, the spring pushes the piston toward the left. As the piston moves to the left, fluid is forced out of the fluid port. At the same time, the moving piston pulls air into the spring chamber through the vent hole. A three way control valve is normally used for controlling the operation of a single-action actuating cylinder.

A double-action (two ports) actuating cylinder is illustrated in **Figure 11-50B**. The operation of a double-action actuating cylinder is usually controlled by a four-way selector valve.

Figure 11-51 shows an actuating cylinder interconnected with a selector valve. Operation of the selector valve and actuating cylinder is discussed below. When the selector

valve is placed in the ON or EXTEND position, fluid is admitted under pressure to the left-hand chamber of the actuating cylinder. (**Figure 11-51**)

This results in the piston being forced toward the right. As the piston moves toward the right, it pushes return fluid out of the right-hand chamber and through the selector valve to the reservoir. When the selector valve is placed in its RETRACT position, as illustrated in **Figure 11-16**, fluid pressure enters the right chamber,

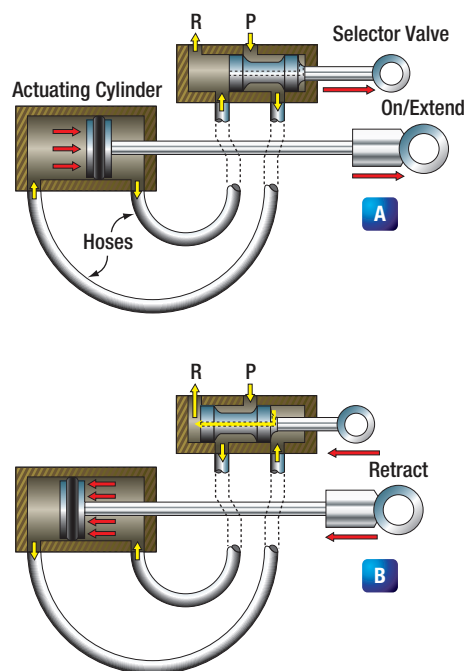


Figure 11-51. Linear actuator operation.

forcing the piston toward the left. As the piston moves toward the left, it pushes return fluid out of the left chamber and through the selector valve to the reservoir.

Besides having the ability to move a load into position, a double-acting cylinder also has the ability to hold a load in position. This capability exists because when the selector valve used to control operation of the actuating cylinder is placed in the off position, fluid is trapped in the chambers on both sides of the actuating cylinder piston. Internal locking actuators also are used in some applications.

ROTARY ACTUATORS

Rotary actuators can mount right at the part without taking up the long stroke lengths required of cylinders. Rotary actuators are not limited to the 90° pivot arc typical of cylinders; they can achieve arc lengths of 180°, 360°, or even 720° or more, depending on the configuration. An often used type of rotary actuator is the rack and pinion actuator used for many nose wheel steering mechanisms. In a rack-and-pinion actuator, a long piston with one side machined into a rack engages a pinion to turn the output shaft. (*Figure 11-52*) One side of the piston receive fluid pressure while the other side is connected to the return. When the piston moves, it rotates the pinion.

FILTERS

A filter is a screening or straining device used to clean the hydraulic fluid, preventing foreign particles and contaminating substances from remaining in the system. (*Figure 11-53*) If such objectionable material were not removed, the entire hydraulic system of the aircraft could fail through the breakdown or malfunctioning of a single unit of the system.

The hydraulic fluid holds in suspension tiny particles of metal that are deposited during the normal wear of selector valves, pumps, and other system components. Such minute particles of metal may damage the units and parts through which they pass if they are not removed by a filter. Since tolerances within the hydraulic system components are quite small, it is apparent that the reliability and efficiency of the entire system depends upon adequate filtering. Filters may be located within the reservoir, in the pressure line, in the return line, or in any other location the designer of the system decides that they are needed to safeguard the hydraulic

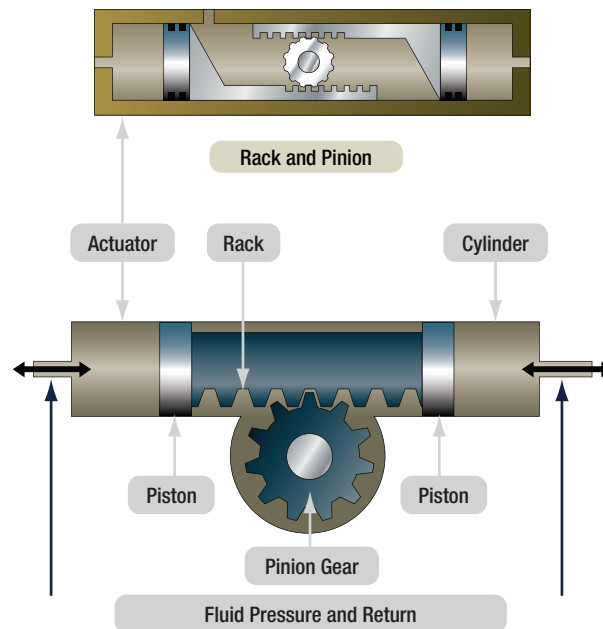


Figure 11-52. Rack and pinion gear.

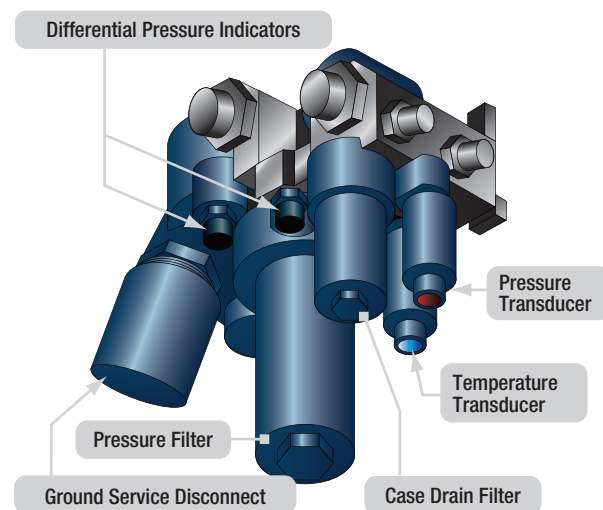


Figure 11-53. Filter module components.

system against impurities. Modern design often uses a filter module that contains several filters and other components. (*Figure 11-54*)

There are many models and styles of filters. Their position in the aircraft and design requirements determine their shape and size. Most filters used in modern aircraft are of the inline type. The inline filter assembly is comprised of three basic units: head assembly, bowl, and element.

The head assembly is secured to the aircraft structure and connecting lines. Within the head, there is a bypass valve that routes the hydraulic fluid directly from the inlet to the outlet port if the filter element becomes

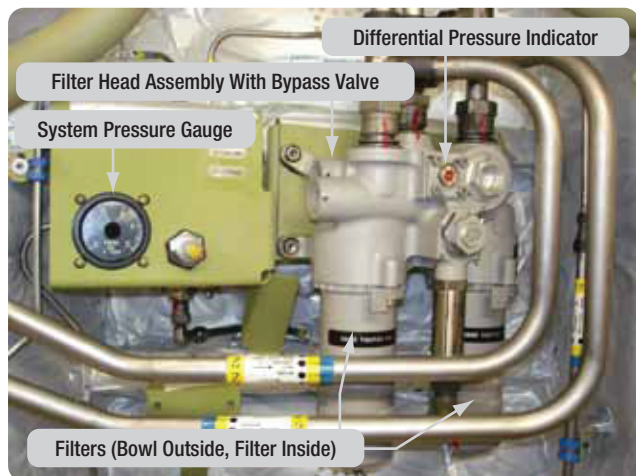


Figure 11-54. A transport category filter module with two filters.

clogged with foreign matter. The bowl is the housing that holds the element to the filter head and is removed when element removal is required. The element may be a micron, porous metal, or magnetic type. The micron element is made of a specially treated paper and is normally thrown away when removed. The porous metal and magnetic filter elements are designed to be cleaned by various methods and replaced in the system.

MICRON TYPE FILTERS

A typical micron type filter assembly utilizes an element made of specially treated paper that is formed in vertical convolutions (wrinkles). An internal spring holds the elements in shape. The micron element is designed to prevent the passage of solids greater than 10 microns (0.000 394 inch) in size. (*Figure 11-55*)

In the event that the filter element becomes clogged, the spring-loaded relief valve in the filter head bypasses the fluid after a differential pressure of 50 psi has been built up. Hydraulic fluid enters the filter through the inlet port in the filter body and flows around the element inside the bowl. Filtering takes place as the fluid passes through the element into the hollow core, leaving the foreign material on the outside of the element.

MAINTENANCE OF FILTERS

Maintenance of filters is relatively easy. It mainly involves cleaning the filter and element or cleaning the filter and replacing the element. Filters using the micron type element should have the element replaced periodically according to applicable instructions. Since reservoir filters are of the micron type, they must also be periodically changed or cleaned. For filters using

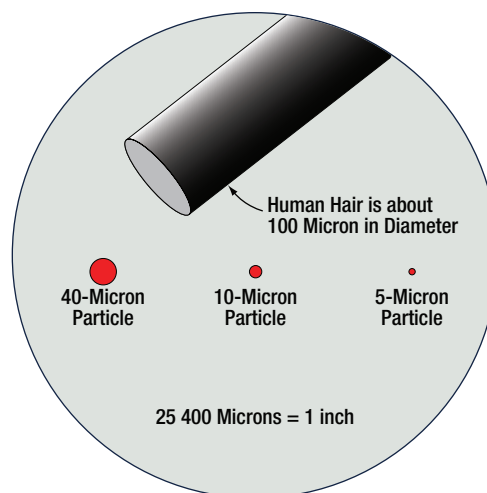


Figure 11-55. Size comparison in microns.

other than the micron type element, cleaning the filter and element is usually all that is necessary. However, the element should be inspected very closely to ensure that it is completely undamaged. The methods and materials used in cleaning all filters are too numerous to be included in this text. Consult the manufacturer's instructions for this information.

When replacing filter elements, be sure that there is no pressure on the filter bowl. Protective clothing and a face shield must be used to prevent fluid from contacting the eye. Replace the element with one that has the proper rating. After the filter element has been replaced, the system must be pressure tested to ensure that the sealing element in the filter assembly is intact. In the event of a major component failure, such as a pump, consideration must be given to replacing the system filter elements, as well as the failed component.

FILTER BYPASS VALVE

Filter modules are often equipped with a bypass relief valve. The bypass relief valve opens if the filter clogs, permitting continued hydraulic flow and operation of aircraft systems. Dirty oil is preferred over no flow at all. *Figure 11-56* shows the principle of operation of a filter bypass valve. Ball valve opens when the filter becomes clogged and the pressure over the filter increases.

FILTER DIFFERENTIAL PRESSURE INDICATORS

The extent to which a filter element is loaded can be determined by measuring the drop in hydraulic pressure across the element under rated flow conditions. This drop, or differential pressure, provides a convenient

means of monitoring the condition of installed filter elements and is the operating principle used in the differential pressure or loaded-filter indicators found on many filter assemblies.

Differential pressure indicating devices have many configurations, including electrical switches, continuous reading visual indicators (gauges), and visual indicators with memory. Visual indicators with memory usually take the form of magnetic or mechanically latched buttons or pins that extend when the differential pressure exceeds that allowed for a serviceable element. (*Figure 11-56, Top*)

When this increased pressure reaches a specific value, inlet pressure forces the spring-loaded magnetic piston downward, breaking the magnetic attachment between the indicator button and the magnetic piston. This allows the red indicator to pop out, signifying that the element must be cleaned. The button or pin, once extended, remains in that position until manually reset and provides a permanent (until reset) warning of a loaded element. This feature is particularly useful where it is impossible for an operator to continuously monitor the visual indicator, such as in a remote location on the aircraft.

Some button indicators have a thermal lockout device incorporated in their design that prevents operation of the indicator below a certain temperature. The lockout prevents the higher differential pressure generated at cold temperatures by high fluid viscosity from causing a false indication of a loaded filter element.

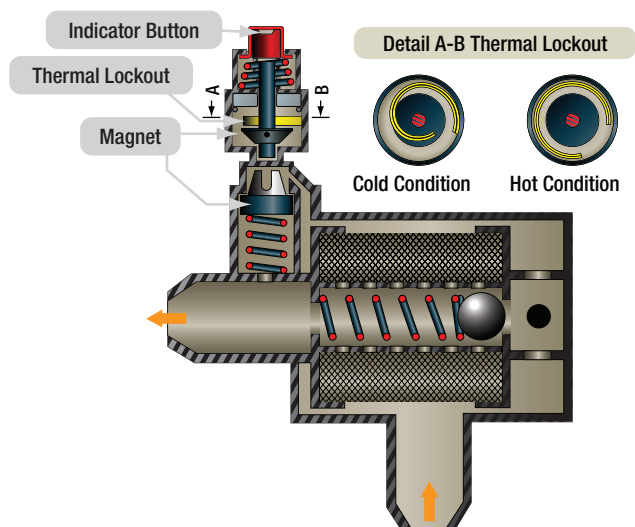


Figure 11-56. Filter bypass valve.

Differential pressure indicators are a component part of the filter assembly in which they are installed and are normally tested and overhauled as part of the complete assembly. With some model filter assemblies, however, it is possible to replace the indicator itself without removal of the filter assembly if it is suspected of being inoperative or out of calibration. It is important that the external surfaces of button type indicators be kept free of dirt or paint to ensure free movement of the button.

Indications of excessive differential pressure, regardless of the type of indicator employed, should never be disregarded. All such indications must be verified and action taken, as required, to replace the loaded filter element. Failure to replace a loaded element can result in system starvation, filter element collapse, or the loss of filtration where bypass assemblies are used.

Verification of loaded filter indications is particularly important with button type indicators as they may have been falsely triggered by mechanical shock, vibration, or cold start of the system. Verification is usually obtained by manually resetting the indicator and operating the system to create a maximum flow demand ensuring that the fluid is at near normal operating temperatures.

HEAT EXCHANGERS

Transport type aircraft use heat exchangers in their hydraulic power supply system to cool the hydraulic fluid from the hydraulic pumps. This extends the service life of the fluid and the hydraulic pumps. They are located in the fuel tanks of the aircraft. The heat exchangers use aluminum finned tubes to transfer heat from the fluid to the fuel. The fuel in the tanks that contain the heat exchangers must be maintained at a specific level to ensure adequate cooling of the fluid. (*Figure 11-57*)

HYDRAULIC SEALS

Seals are used to prevent fluid from passing a certain point, and to keep air and dirt out of the system in which they are used. The increased use of hydraulics and pneumatics in aircraft systems has created a need for packings and gaskets of varying characteristics and design to meet the many variations of operating speeds and temperatures to which they are subjected. No one style or type of seal is satisfactory for all installations. Some of the reasons for this are:

1. Pressure at which the system operates.
2. The type fluid used in the system.

3. The metal finish and the clearance between adjacent parts.
4. The type motion (rotary or reciprocating), if any.

Most seals are made from synthetic materials that are compatible with the hydraulic fluid used. Seals used for MIL-H-5606 hydraulic fluid are not compatible with Skydrol® and servicing the hydraulic system with the wrong fluid could result in leaks and system malfunctions. Seals for systems that use MIL-H-5606 are made of neoprene or Buna-N. Seals for Skydrol® are made from butyl rubber or ethylene-propylene elastomers.

Seals are divided into three main classes: packings, gaskets, and wipers. A seal may consist of more than one component, such as an O-ring and a backup ring, or possibly an O-ring and two backup rings. Hydraulic seals used internally on a sliding or moving assembly are normally called packings. (*Figure 11-58*) Hydraulic seals used between nonmoving fittings and bosses are normally called gaskets.

PACKINGS

V-RINGS

V-ring packings (AN6225) are one way seals and are always installed with the open end of the V facing the pressure. V-ring packings must have a male and

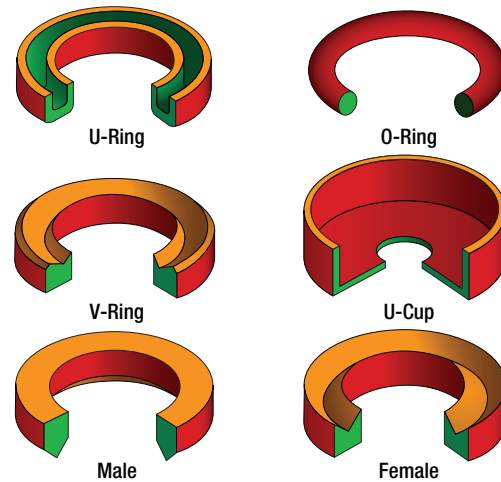


Figure 11-58. Packings.

female adapter to hold them in the proper position after installation. It is also necessary to torque the seal retainer to the value specified by the manufacturer of the component being serviced, or the seal may not give satisfactory service.

U-RING

U-ring packings (AN6226) and U-cup packings are used in brake assemblies and brake master cylinders. The U-ring and U-cup seals pressure in only one direction; therefore, the lip of the packings must face toward the pressure. U-ring packings are primarily low pressure packings to be used with pressures of less than 1 000 psi.

O-RINGS

Most packings and gaskets used in aircraft are manufactured in the form of O-rings. An O-ring is circular in shape, and its cross-section is small in relation to its diameter. The cross-section is truly round and has been molded and trimmed to extremely close tolerances. The O-ring packing seals effectively in both directions. This sealing is done by distortion of its elastic compound.

Advances in aircraft design have made new O-ring composition necessary to meet changing conditions. Hydraulic O-rings were originally established under Air Force-Navy (AN) specification numbers 6227, 6230, and 6290 for use in fluid at operating temperatures ranging from -54°F to 71°C. When new designs raised operating temperatures to a possible +135°C, more compounds were developed and perfected.

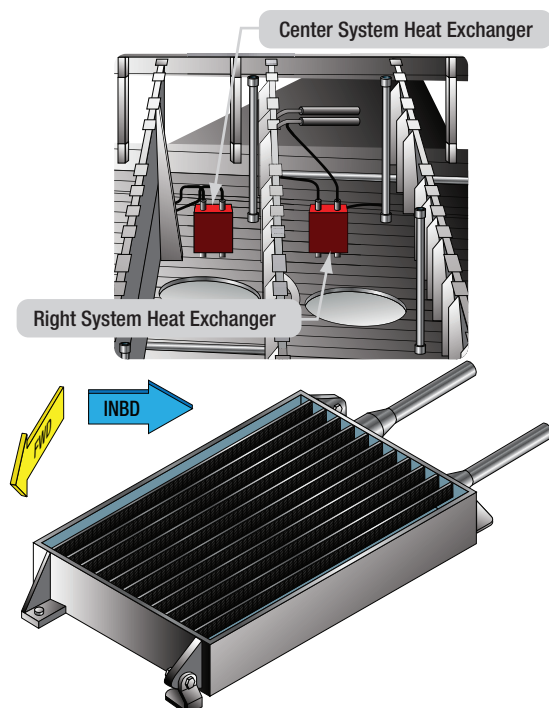


Figure 11-57. Hydraulic heat exchanger.

Recently, newer compounds were developed under Military Standard (MS) specifications that offered improved low-temperature performance without sacrificing high temperature performance. These superior materials were adopted in the MS28775 O-ring, which is replacing AN6227 and AN6230 O-rings, and the MS28778 O-ring, which is replacing the AN6290 O-ring. These O-rings are now standard for systems where the operating temperatures may vary from -54°C to 135°C.

O-Ring Color Coding

Manufacturers provide color coding on some O-rings, but this is not a reliable or complete means of identification. The color coding system does not identify sizes, but only system fluid or vapor compatibility and, in some cases, the manufacturer. Color codes on O-rings that are compatible with MIL-H-5606 fluid always contains blue, but may also contain red or other colors. Packings and gaskets suitable for use with Skydrol® fluid are always coded with a green stripe, but may also have a blue, gray, red, green, or yellow dot as a part of the color code. Color codes on O-rings that are compatible with hydrocarbon fluid always contain red, but never contain blue. A colored stripe around the circumference indicates that the O-ring is a boss gasket seal. The color of the stripe indicates fluid compatibility: red for fuel, blue for hydraulic fluid. The coding on some rings is not permanent. On others, it may be omitted due to manufacturing difficulties or interference with operation. Furthermore, the color coding system provides no means to establish the age of the O-ring or its temperature limitations. Because of the difficulties with color coding, rings are available in individual hermetically sealed envelopes labeled with all pertinent data. When selecting an O-ring for installation, the basic part number on the sealed envelope provides the most reliable compound identification.

Backup Rings

Backup rings (MS28782) made of Teflon™ do not deteriorate with age, are unaffected by any system fluid or vapor, and can tolerate temperature extremes in excess of those encountered in high pressure hydraulic systems. Their dash numbers indicate not only their size but also relate directly to the dash number of the O-ring for which they are dimensionally suited. They are procurable under a number of basic part numbers, but they are interchangeable; any Teflon™ backup ring

may be used to replace any other Teflon™ backup ring if it is of proper overall dimension to support the applicable O-ring. Backup rings are not color coded or otherwise marked and must be identified from package labels.

The inspection of backup rings should include a check to ensure that surfaces are free from irregularities, that the edges are clean cut and sharp, and that scarf cuts are parallel. When checking Teflon™ spiral backup rings, make sure that the coils do not separate more than ¼-inch when unrestrained. Be certain that backup rings are installed downstream of the O-ring. (*Figure 11-59*)

O-Ring Installation

When removing or installing O-rings, avoid using pointed or sharp-edged tools that might cause scratching or marring of hydraulic component surfaces or cause damage to the O-rings. Special tooling for the installation of O-rings is available. While using the seal removal and the installation tools, contact with cylinder walls, piston heads, and related precision components is not desirable.

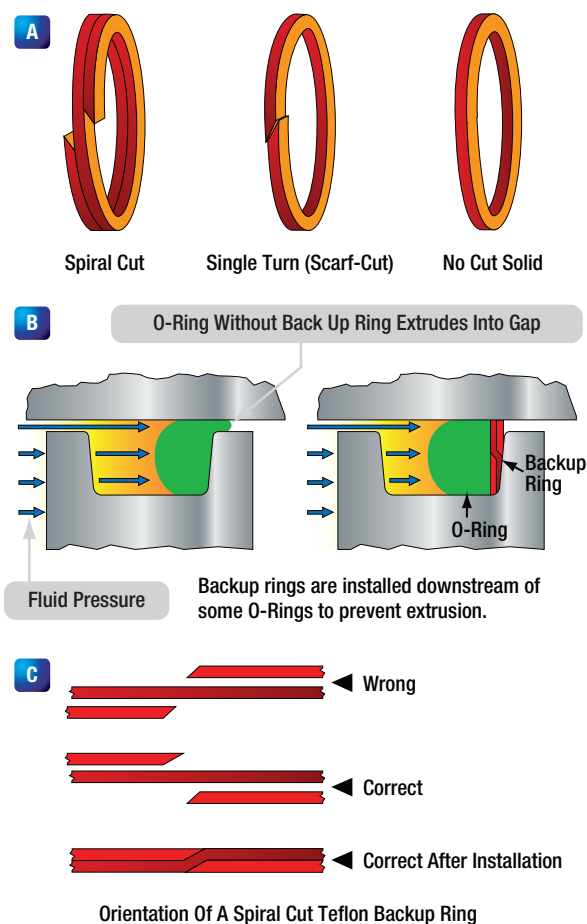


Figure 11-59. Backup O-rings installed downstream.

After the removal of all O-rings, the parts that receive new O-rings have to be cleaned and inspected to make sure that they are free from all contamination. Each replacement O-ring should be removed from its sealed package and inspected for defects, such as blemishes, abrasions, cuts, or punctures. Although an O-ring may appear perfect at first glance, slight surface flaws may exist. These flaws are often capable of preventing satisfactory O-ring performance under the variable operating pressures of aircraft systems; therefore, O-rings should be rejected for flaws that affect their performance. Such flaws are difficult to detect, and one aircraft manufacturer recommends using a four-power magnifying glass with adequate lighting to inspect each ring before it is installed. By rolling the ring on an inspection cone or dowel, the inner diameter surface can also be checked for small cracks, particles of foreign material, or other irregularities that cause leakage or shorten the life of the O-ring. The slight stretching of the ring when it is rolled inside out helps to reveal some defects not otherwise visible.

After inspection and prior to installation, immerse the O-ring in clean hydraulic fluid. During the installation, avoid rolling and twisting the O-ring to maneuver it into place. If possible, keep the position of the O-ring's mold line constant. When the O-ring installation requires spanning or inserting through sharply threaded areas, ridges, slots, and edges, use protective measures, such as O-ring entering sleeves, as shown in *Figure 11-60A*. After the O-ring is placed in the cavity provided, gently roll the O-ring with the fingers to remove any twist that might have occurred during installation. (*Figure 11-61*)

GASKETS

Gaskets are used as static (stationary) seals between two flat surfaces. Some of the more common gasket materials are asbestos, copper, cork, and rubber. Asbestos sheeting is used wherever a heat resistant gasket is needed. It is used extensively for exhaust system gaskets. Most asbestos exhaust gaskets have a thin sheet of copper edging to prolong their life.

A solid copper washer is used for spark plug gaskets where it is essential to have a non-compressible, yet semi-soft gasket. Cork gaskets can be used as an oil seal between the engine crankcase and accessories, and where a gasket is required that is capable of occupying an uneven or varying space caused by a rough surface or

expansion and contraction. Rubber sheeting can be used where there is a need for a compressible gasket. It should not be used in any place where it may come in contact with gasoline or oil because the rubber deteriorates very rapidly when exposed to these substances. Gaskets are used in fluid systems around the end caps of actuating cylinders, valves, and other units. The gasket generally used for this purpose is in the shape of an O-ring, similar to O-ring packings.

WIPERS

Wipers are used to clean and lubricate the exposed portions of piston shafts. They prevent dirt from entering the system and help protect the piston shaft against scoring. Wipers may be either metallic or felt. They are sometimes used together, a felt wiper installed behind a metallic wiper.

HYDRAULIC INDICATING AND WARNING

There are just a few hydraulic system indications on the flight deck. Fluid pressure and temperature are the primary parameters monitored as well as fluid quantity. Reservoir pressurization air pressure may also be monitored. Typically, electro-hydraulic transducers are mounted in the system in key locations so that hydraulic pressure and temperature can be displayed on a gauge or LCD screen. A separate transmitter and indication is used for brake pressure. For servicing and maintenance, direct reading indicators are installed so that maintenance technicians can observe system status while on the ramp.

System pressure sensors are often located at the hydraulic system pressure filter modules. Low pressure warning switches are located down stream of the pump outlet and may also be at the module. A hydraulic panel on the flight deck incorporates pump switches and temperature and pressure indications in older aircraft. Warnings and system indications are displayed on system status screens away from the switches on class cockpit aircraft. *Figure 11-62* illustrates a typical hydraulic control panel on the flight deck of an older Boeing aircraft.

Figure 11-63 illustrates system status, synoptic and maintenance page displays for the hydraulic system on a glass cockpit aircraft.

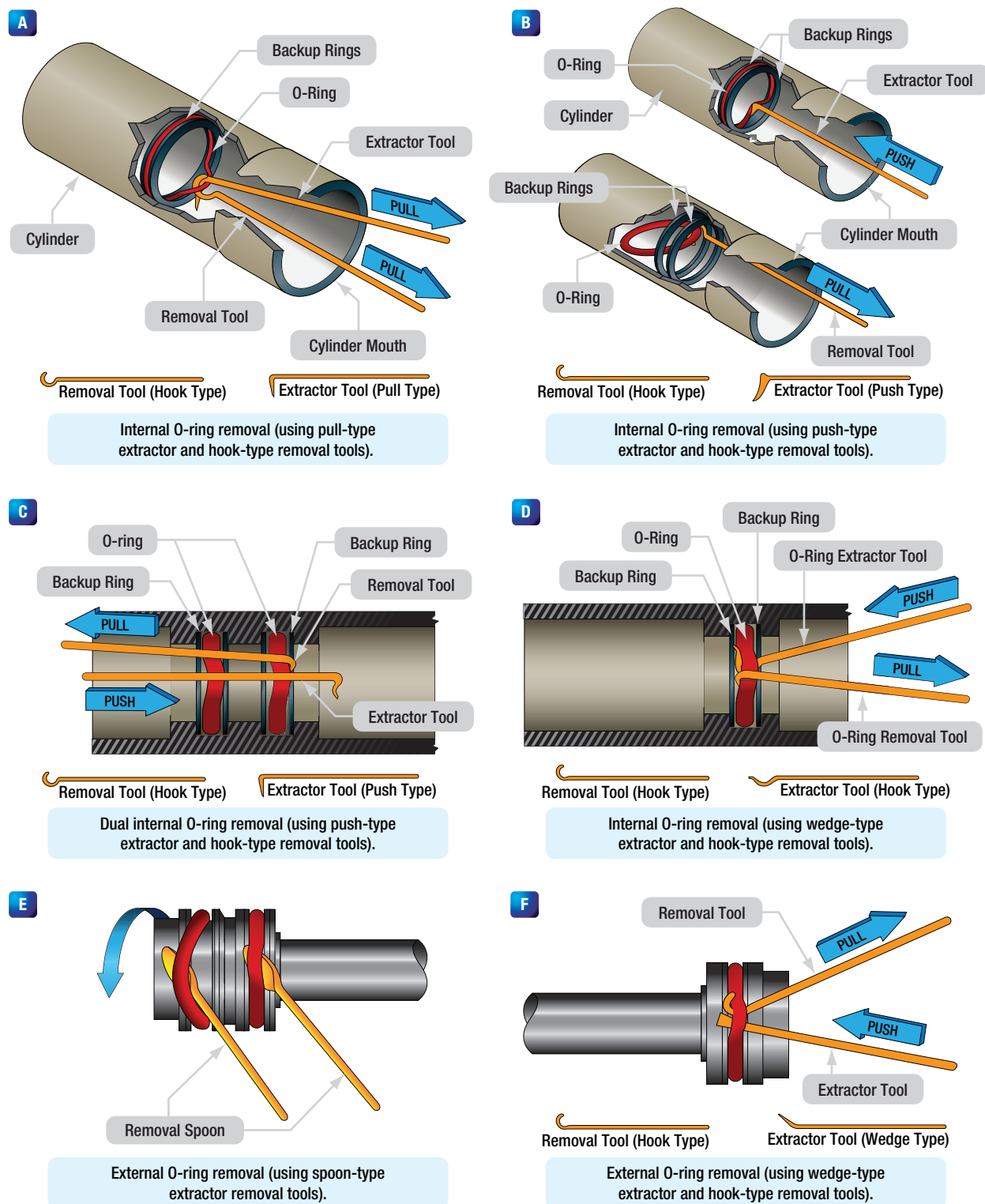


Figure 11-60. O-ring installation techniques.

Hydraulic fluid quantity is monitored at the reservoir through the use of a float gauge, sight glass, or other sensing mechanism which sends a signal to the flight deck for gauge or LCD display. A low quantity warning switch may be included in the system. Hydraulic system fluid temperature indication is usually limited to an

OVERHEAT annunciation for each pump/system. Temperature switches, often located in the return line as the fluid enters the reservoir, trip when a preset temperature is reached.

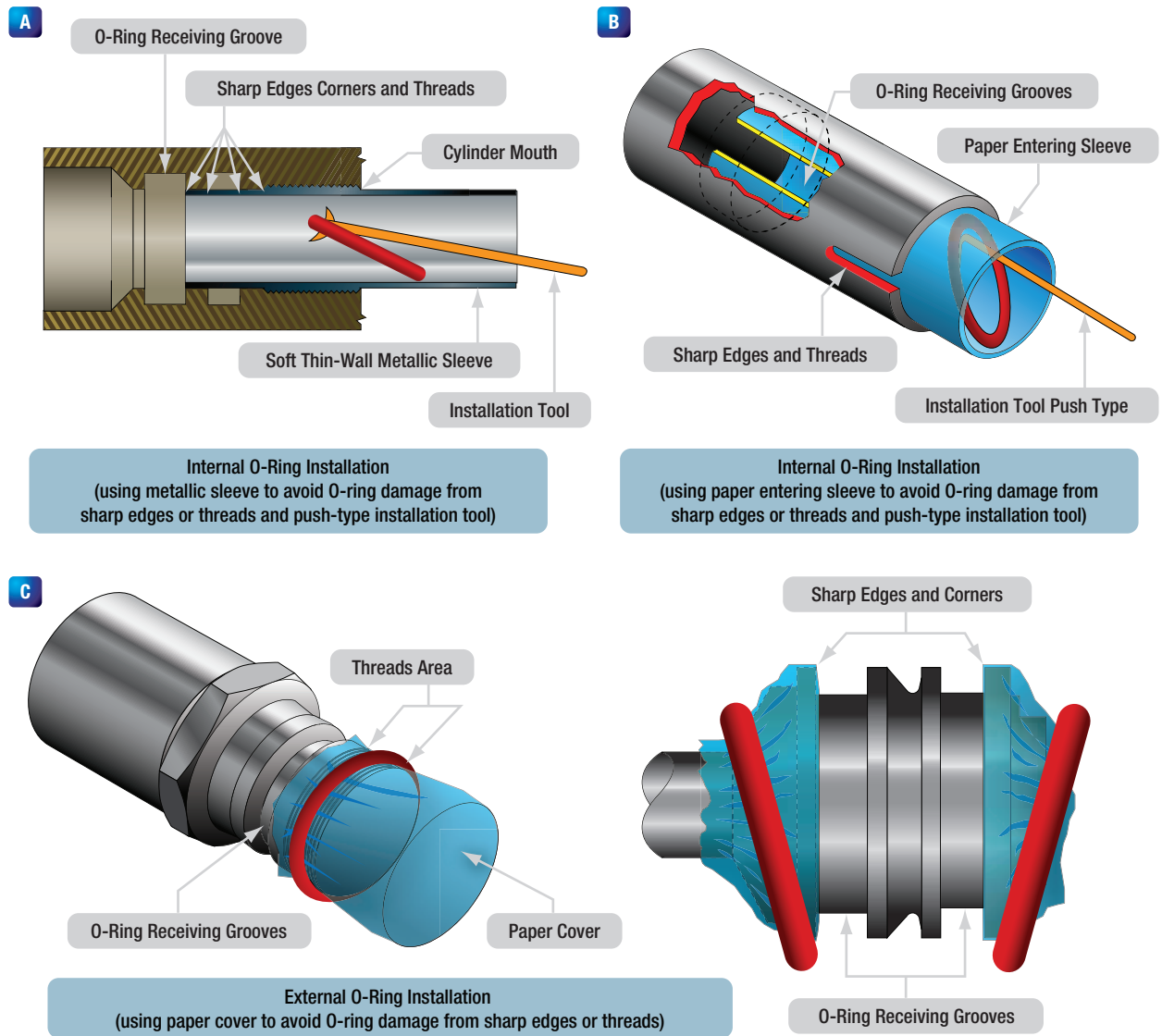


Figure 11-61. More O-ring installation techniques.



Figure 11-62. Hydraulic panel on an older Boeing transport aircraft.

A signal is sent to the flight deck for annunciation. Temperature sensors for hydraulic systems with electrically driven pumps serve as motor temperature

monitoring devices as well. Motor driven pumps are more likely to get hot than engine driven pumps. The hot motor transfers some of its heat to the fluid as it circulates.

Hydraulic system warnings include low pressure annunciations for each hydraulic system. Typically a lamp will illuminate, flash or change color on the flight deck when a pressure sensor sends an electric or electronic signal that a low pressure condition exists. Many indicators display a low pressure warning when the hydraulic pumps are OFF which goes away when the pumps are switched ON and operate normally.

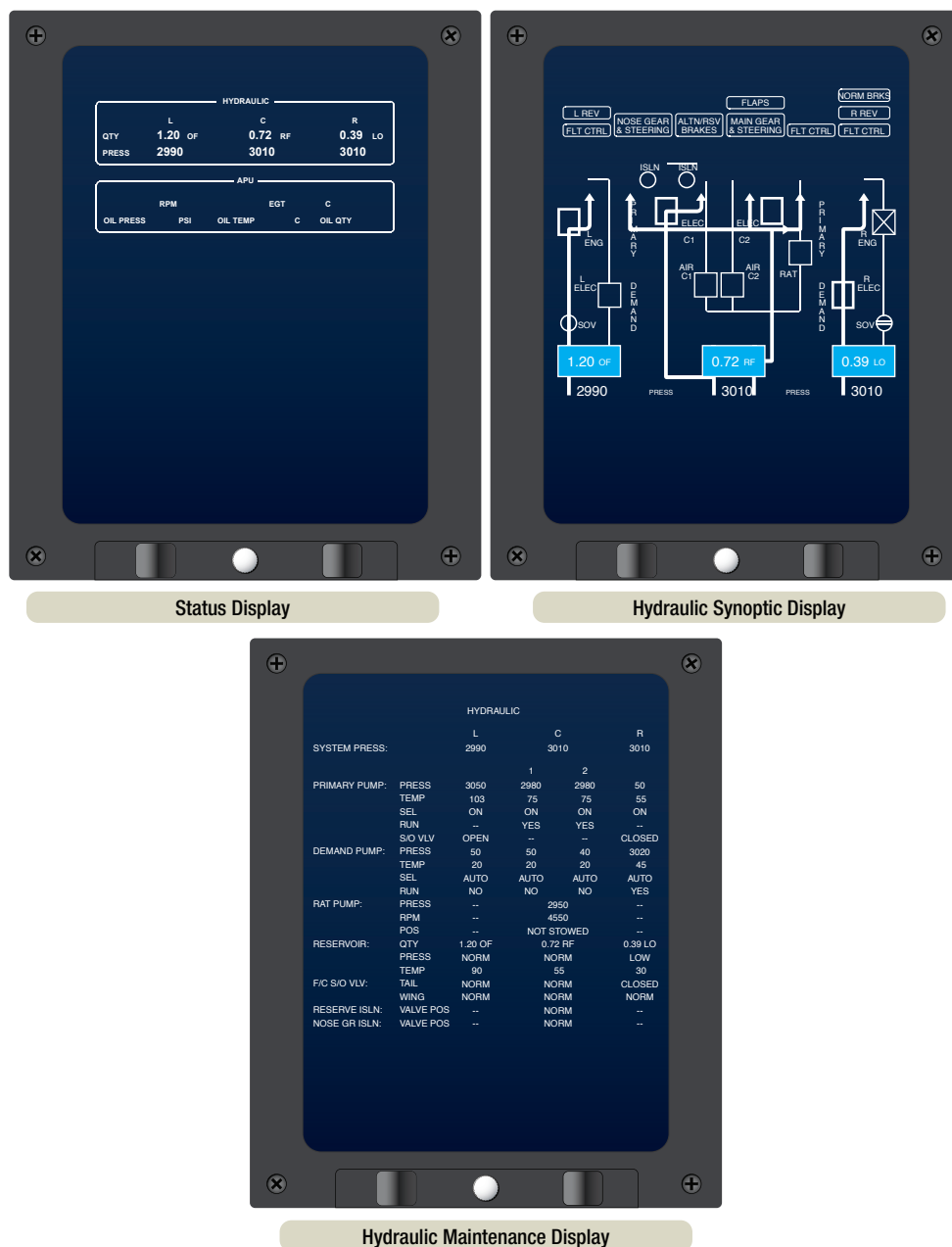


Figure 11-63. Hydraulic system information displayed on a digital flight deck flat panel screen.

INTERFACE WITH OTHER SYSTEMS

Many aircraft systems use hydraulic power such as landing gear extension and retraction, flight controls, and auto pilot. In most cases, the operational logic for these advanced systems are controlled by computer. To integrate the mechanical power of the hydraulic system, hydraulic system parameters and status condition must be input into the controlling computer. In the absence of any malfunction, the computer controller activates the correct hydraulic system components when needed.

Vital systems control logic can also dictate operation in alternate modes should the hydraulic system parameters be out of the normal operating range. For example, if the hydraulic pump used for normal operations is not maintaining acceptable system pressure, logic circuits reconfigure the operational mode from 'normal' to an alternate mode that utilizes the back up hydraulic pump. Hydraulic system parameters that are captured in analog format are converted to digital format for use in the control system logic.

Question: 11-1

Hydraulic operations are nearly ____% efficient.

Question: 11-6

Appropriate must be specifically designated for the type of hydraulic fluid in use.

Question: 11-2

Selector valves in an open-center hydraulic system are arranged in _____ with each other.

Question: 11-7

All hydraulic lines and fittings should be _____ immediately after disconnecting.

Question: 11-3

The most widely used type of hydraulic system used in aviation is the _____ - center system.

Question: 11-8

The _____ serves as an overflow basin for excess fluid forced out of the system by thermal expansion, the accumulators, and by piston and rod displacement.

Question: 11-4

Liquids are used to transmit and distribute force in a hydraulic system because they are basically _____.

Question: 11-9

Pressurizing the hydraulic reservoir assures _____ at high altitudes when low atmospheric pressures are encountered.

Question: 11-5

The type of hydraulic fluid used most in commercial jet transport aircraft is _____ - based.

Question: 11-10

Hydraulic reservoirs may be pressurized by bleed air or by _____.

ANSWERS

Answer: 11-1
100.

Answer: 11-6
seals, gaskets, and hoses.

Answer: 11-2
series.

Answer: 11-7
capped or plugged.

Answer: 11-3
closed.

Answer: 11-8
reservoir.

Answer: 11-4
incompressible.

Answer: 11-9
positive flow of fluid to the pump(s).

Answer: 11-5
phosphate ester.

Answer: 11-10
hydraulic system pressure.

Question: 11-11

Name 4 functions of accumulators.

Question: 11-16

Generally, _____ are used when engine-driven pumps fail.

Question: 11-12

The hydraulic pump is the source of fluid flow, which when restricted, generates _____ in the hydraulic system.

Question: 11-17

The most commonly used hydraulic _____ is the fixed-displacement bent-axis type.

Question: 11-13

When a constant displacement pump is used in a hydraulic system in which the pressure must be kept at a constant value, a _____ is required.

Question: 11-18

In a _____, two units, a hydraulic pump and hydraulic motor, are connected via a single drive shaft so that power can be transferred between two hydraulic systems.

Question: 11-14

Hydraulic pump drive couplings are designed to _____ if the pump becomes hard to turn or becomes jammed.

Question: 11-19

Name 3 types of relief valves.

Question: 11-15

Variable displacement hydraulic pump output is changed automatically by a pump _____ within the pump.

Question: 11-20

_____ valves are used in hydraulic systems where it is necessary to lower the normal system operating pressure by a specified amount.

ANSWERS

Answer: 11-11

Dampen pressure surges.

Supplement power pump when several hydraulic units are operated simultaneously.

Store of power for limited operation without a pump.

Compensate for small leaks in the system.

Answer: 11-16

electrically-driven pumps.

Answer: 11-12

pressure.

Answer: 11-17

motor.

Answer: 11-13

pressure regulator.

Answer: 11-18

PTU (power transfer unit).

Answer: 11-14

shear.

Answer: 11-19

Ball.

Sleeve.

Poppet.

Answer: 11-15

compensator.

Answer: 11-20

Pressure reducing.

Question: 11-21

A _____ is used to control the direction of movement of a hydraulic actuating cylinder or similar device.

Question: 11-26

Most hydraulic filters used in modern aircraft are of the _____ type.

Question: 11-22

A common flow control valve in aircraft hydraulic systems that allows fluid to flow unimpeded in one direction, but prevents or restricts fluid flow in the opposite direction is called a _____.

Question: 11-27

The _____ opens if the filter clogs, permitting continued hydraulic flow and operation of aircraft systems.

Question: 11-23

A sequence valve may be pressure controlled or _____ operated, or controlled by electric switches.

Question: 11-28

The extent to which a filter element is loaded can be determined by measuring the _____ in hydraulic pressure across the element under rated flow conditions.

Question: 11-24

_____ are installed in hydraulic lines to prevent loss of fluid when units are removed.

Question: 11-29

A heat exchanger used to keep hydraulic system fluid cool is typically located _____ on jet transport aircraft.

Question: 11-25

The operation of a double-action hydraulic actuating cylinder is usually controlled by a _____ selector valve.

Question: 11-30

Hydraulic seals are divided into three main classes: _____, _____ and _____.

ANSWERS

Answer: 11-21
selector valve.

Answer: 11-26
inline.

Answer: 11-22
check valve.

Answer: 11-27
bypass relief valve.

Answer: 11-23
mechanically.

Answer: 11-28
difference (drop).

Answer: 11-24
Quick disconnect valves.

Answer: 11-29
in a fuel tank.

Answer: 11-25
four-way.

Answer: 11-30
packings, gaskets, wipers.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 12

ICE AND RAIN PROTECTION (ATA 30)

Knowledge Requirements

11.12 - Ice and Rain Protection (ATA 30)

Ice formation, classification and detection;
Anti-icing systems: electrical, hot air and chemical;
De-icing systems: electrical, hot air, pneumatic and chemical;
Rain repellent;
Probe and drain heating;
Wiper systems.

3

11.12 - ICE AND RAIN PROTECTION

Rain, snow, and ice, are long time enemies of transportation. Flying has added a new dimension, particularly with respect to ice. Under certain atmospheric conditions, ice can build rapidly on airfoils and air inlets. On days when there is visible moisture in the air, ice can form on aircraft leading edge surfaces at altitudes where freezing temperatures start. Water droplets in the air can be supercooled to below freezing without actually turning into ice unless they are disturbed in some manner. This unusual occurrence is partly due to the surface tension of the water droplet not allowing the droplet to expand and freeze. However, when aircraft surfaces disturb these droplets, they immediately turn to ice on the aircraft surfaces.

In order to perform as designed, aircraft airfoils must be completely smooth and free of any irregularities or contamination in the form of ice, snow or frost. Even a small amount of surface contamination can reduce lift and raise the stall speed. Accidents have occurred due to airfoil surface roughness caused by frost. The additional weight caused by ice accumulation is also problematic. All aviators must be diligent to only dispatch an aircraft that is free from any type of ice, snow, or frost contamination. There are two types of ice are encountered during flight: clear ice and rime ice. Clear ice forms when the remaining liquid portion of the water drop flows out over the aircraft surface, gradually freezing as a smooth sheet of solid ice. Formation occurs when droplets are large, such as in rain or in cumuliform clouds. Clear ice is hard, heavy, and tenacious. Its removal by de-icing equipment is especially difficult. (Figure 12-1)



Figure 12-1. Formation of ice on aircraft leading edge.

Rime ice forms when water drops are small, such as those in stratified clouds or light drizzle. The liquid portion remaining after initial impact freezes rapidly before the drop has time to spread over the aircraft surface. The small frozen droplets trap air giving the ice a white appearance. Rime ice is lighter in weight than clear ice, however its weight is of little significance. The irregular shape and rough surface of rime ice decrease the effectiveness and efficiency of the aerodynamic airfoils. This reduces lift and increases drag. Rime ice is brittle and more easily removed than clear ice.

Mixed clear and rime icing can form rapidly when water drops vary in size or when liquid drops intermingle with snow or ice particles. Ice particles become embedded in clear ice, building a very rough accumulation sometimes in a mushroom shape on leading edges. Ice may be expected to form whenever there is visible moisture in the air and temperature is near or below freezing. An exception is carburetor icing, which can occur during warm weather with no visible moisture present.

ICING EFFECTS

Ice or frost forming on aircraft creates hazards detrimental to safe flight. The resulting malformation of the airfoil when ice adheres to it decreases the amount of lift. The additional weight and unequal formation of the ice may also cause unbalancing of the aircraft, making it hard to control. Enough ice to cause an unsafe flight condition can form in a very short period of time, thus some method of ice prevention or removal is necessary.

Ice buildup increases drag and reduces lift. It causes destructive vibration and hampers true instrument readings. Control surfaces become unbalanced or frozen. Fixed slots are filled and movable slots jammed. Radio reception is hampered and engine performance is affected. Ice, snow, and slush have a direct impact on the safety of flight. Not only because of degraded lift, reduced takeoff performance, and/ or maneuverability of the aircraft, but when chunks break off, they can also cause engine failures and structural damage. Fuselage aft-mounted engines are particularly susceptible to this foreign object damage (FOD) phenomenon. Wing mounted engines are not excluded however.

Ice can be present on any part of the aircraft and, when it

breaks off, there is some probability that it could go into an engine. The worst case is that ice on the wing breaks off during takeoff due to the flexing of the wing and goes directly into the engine, leading to surge, vibration, and complete thrust loss. Light snow that is loose on the wing surfaces and the fuselage can also cause engine damage leading to surge, vibration, and thrust loss. Whenever icing conditions are encountered, the performance characteristics of the airplane deteriorate. (*Figure 12-2*)

Increased aerodynamic drag increases fuel consumption, reducing the airplane's range and making it more difficult to maintain speed. Decreased rate of climb must be anticipated, not only because of the decrease in wing and empennage efficiency but also because of the possible reduced efficiency of the propellers and increase in gross weight. Abrupt maneuvering and steep turns at low speeds must be avoided because the airplane stalls at higher-than-published speeds with ice accumulation. On final approach for landing, increased airspeed must be maintained to compensate for this increased stall speed. After touchdown with heavy ice accumulation, landing distances may be as much as twice the normal distance due to the increased landing speeds. In this sub-module, ice prevention and ice elimination using electric, pneumatic and chemical systems is discussed.

The ice and rain protection systems used on aircraft keep ice from forming or remove ice on the following airplane components:

- Wing Leading Edges
- Horizontal and Vertical Stabilizer Leading Edges
- Engine Cowl Leading Edges
- Propellers
- Propeller Spinner
- Air Data Probes
- Flight Deck Windows
- Water and Waste System Lines and Drains
- Antenna

Figure 12-3 gives an overview of ice and rain protection systems installed in a large transport category aircraft. In modern aircraft, many of these systems are automatically controlled by the ice detection system and on board computers.

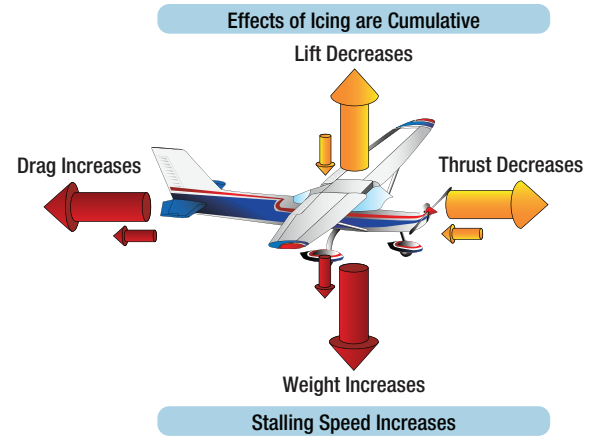


Figure 12-2. Effects of structural icing.

ICE DETECTION

Ice can be detected visually, but most modern aircraft have one or more ice detector sensors that warn the flight crew of icing conditions. Generally, an annunciator light comes on to alert the flight crew when ice is detected. In some aircraft models, multiple ice detectors are used, and the ice detection system automatically turns on certain anti-ice systems when icing is detected.

As can be seen in *Figure 12-4*, ice detectors are mounted on the sides of the forward fuselage to receive impact air as the aircraft moves forward. Two independent detectors and detector systems is normal. The typical anti-ice detector contains a probe which is vibrated at an established rate. For detectors on the Boeing 777, the vibration rate is 40 000 Hz. Inside the probe housing are circuit cards and a microprocessor. The probe is electrically connected to the ice protection control unit and the aircraft data buses.

Should ice collect on the ice detector probe, the established frequency of vibration decreases. A slight lowering of the frequency causes the integral electric probe heater to come ON. The heater quickly melts the ice in 5-7 seconds and shuts OFF. If ice reforms on the probe, the heater will cycle ON again, melt the ice, and turn OFF again. The control logic inside the detector unit monitors the heater cycles. When the probe heat cycles two or more times, the detector automatically sends an icing signal to the airfoil and cowl ice protection system (Boeing) and engine inlet anti-ice (EAI) is automatically turned ON. An annunciation is also made to alert the crew. Note that the aircraft must be in the air with the anti-ice switch on the flight deck set to AUTO.

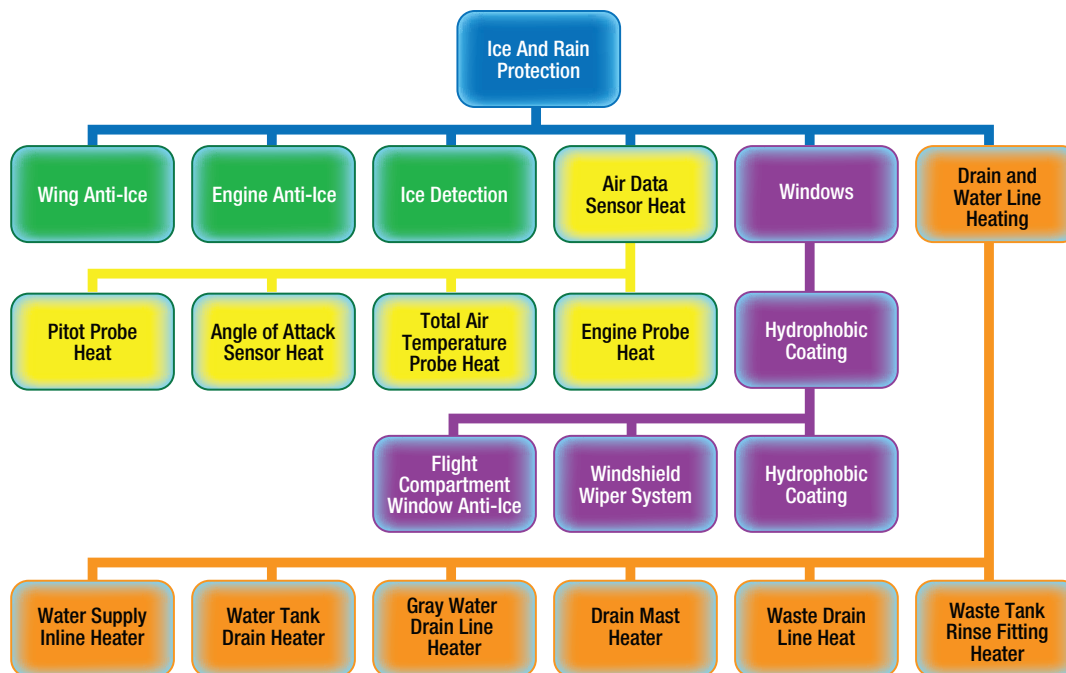


Figure 12-3. Ice and rain protection systems.

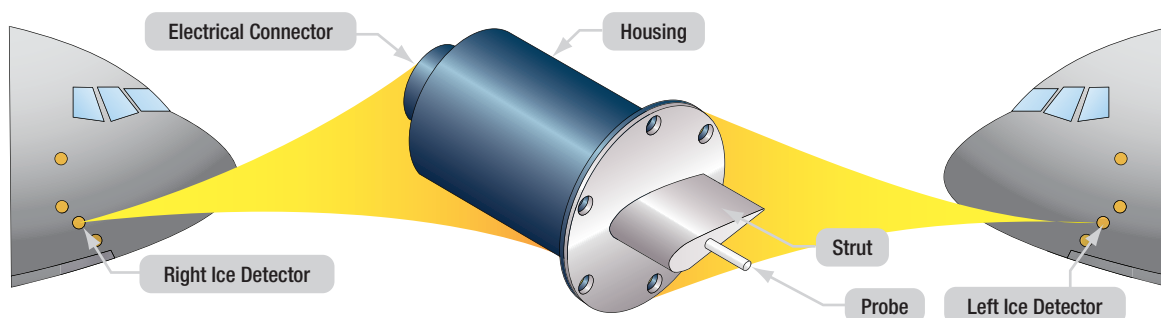


Figure 12-4. An ice detector alerts the flight crew of icing conditions and, on some aircraft, automatically activates ice protection systems. One or more detectors are located on the forward fuselage.

If the probe heat cycles ten or more times, an icing signal is sent and the wing anti-ice system (WIA) is also automatically turned ON. Note that if the probe heat is on for more than 15 seconds, the EAI comes ON. After 25 seconds, if the probe vibration does not return to its normal rate, a fault is annunciated and the heater is turned OFF.

Figure 12-5 illustrates the ice detector time cycles on a Boeing 777. Ice detection system logic and operation receives input from the air/ground sensing system. It does not function on the ground. However, the aircraft information management system displays the system information on the ice protection maintenance page and ground tests of the ice detectors can be initiated from there.

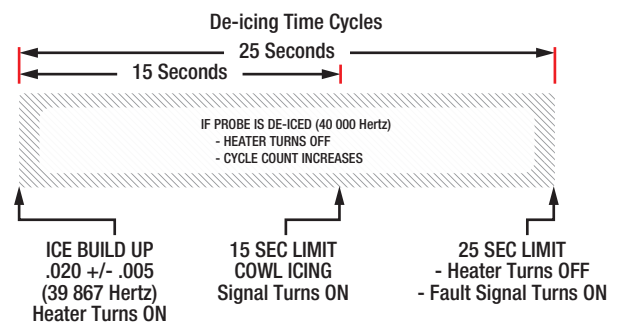


Figure 12-5. Deicing time cycle.

Detector probes get extremely hot when heated and cause serious burns. Also note that electrostatic handling precautions should be exercised when handling ice detectors. **Figure 12-6** illustrates the ice detection system on a Boeing 777.

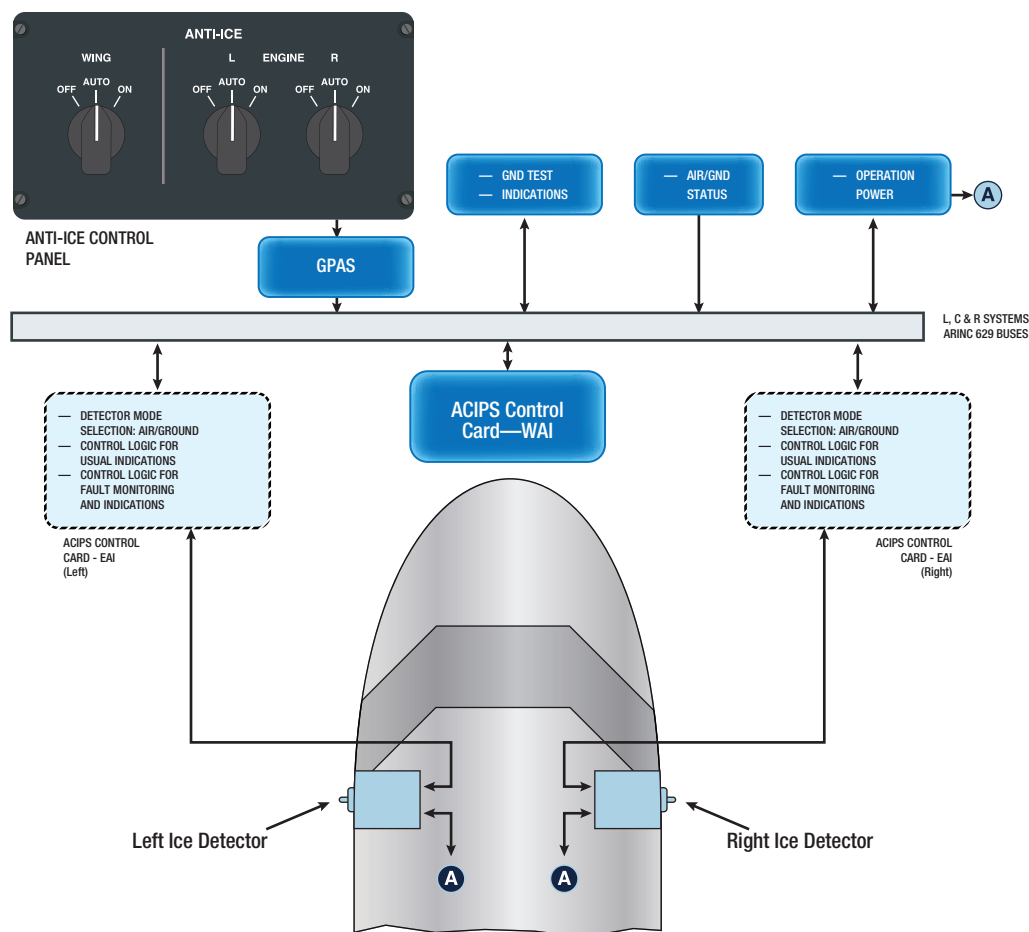


Figure 12-6. Forward fuselage ice detection system on a Boeing 777.

ANTI-ICE VERSUS DE-ICE

Ice control systems are designed for anti-icing or for de-icing. Anti-icing equipment is turned on before entering icing conditions and is designed to prevent ice from forming. A surface may be anti-iced by keeping it dry, by heating to a temperature that evaporates water upon impingement, or by heating the surface just enough to prevent freezing. De-Icing equipment is designed to remove ice after it begins to accumulate typically on the wings and stabilizer leading edges. Ice may be controlled on aircraft structures by the methods described in *Figure 12-7*.

Several means of preventing or removing ice are used depending on the type and location of the component that requires ice control. They basically include:

1. Heating the surfaces with hot air.
2. Heating by electrical elements.
3. Breaking up ice that has formed, usually with inflatable boots.
4. Application of chemicals to prevent adhesion of ice.

ANTI-ICING SYSTEMS

On large, high performance turbine powered aircraft, anti-ice is the preferred method of ice control. Thermal pneumatic anti-ice using engine bleed air is most common for large surfaces such as the leading edges of the wing, empennage and engine inlet cowling. Thermal electric anti-ice is most common on probes, drain, tanks and windshields. Chemical anti-icing is used on smaller turbine powered aircraft. Once activated, most systems are automatically operated.

THERMAL PNEUMATIC ANTI-ICING

Thermal pneumatic systems used for the purpose of preventing the formation of ice on airfoil leading edges usually use heated air ducted span-wise along the inside of the leading edge of the airfoil and distributed around its inner surface. These thermal pneumatic anti-icing systems are used for wings, leading edge slats, horizontal and vertical stabilizers, engine inlets, and more. As stated, the most common source of the heated air is the turbine engine compressor bleed air.

Location of Ice	Method of Control	
	Anti-Ice	De-Ice
Leading Edge of The Wing	Thermal Pneumatic	Pneumatic Boots
	Thermal Electric	Electric Boots
	Chemical	Chemical (Minor)
Engine Inlets	Thermal Pneumatic	
	Thermal Electric	
Leading Edge of Horizontal and Vertical Stabilizers	Thermal Pneumatic	Pneumatic Boots
	Thermal Electric	Electric Boots
	Chemical	
Pitot Tube, Static Ports, Air Data Sensors,	Thermal Electric	
Water Drains, Tanks and Lines		
Propeller		Electric Element Boots
		Chemical
Windshield/Flight Deck Windows	Thermal Pneumatic	Thermal Pneumatic
	Thermal Electric	(Primarily for De-fogging)
	Chemical	

Figure 12-7. Typical ice control methods.

WING ANTI-ICE (WAI) SYSTEM

Thermal wing anti-ice (WAI or TAI) systems for business jet and large-transport category aircraft take advantage of the relatively large amounts of very hot air that can be bled off of turbine engine compressors to provide a satisfactory source of anti-icing heat. The hot air is routed through ducting, manifolds, and valves to the leading edges of the wings. (*Figure 12-8*)

Figure 12-9 shows a typical WAI system schematic for a business jet. The bleed air is routed to each wing leading edge by an ejector in each wing inboard area. The ejector discharges the bleed air into piccolo tubes for distribution along the leading edge. Fresh ambient air is introduced into the wing leading edge by two flush-mounted ram air scoops in each wing leading edge, one at the wing root and one near the wingtip. The ejectors entrain ambient air, reduce the temperature of



Figure 12-8. Aircraft with thermal WAI system.

the bleed air, and increase the mass airflow in the piccolo tubes. The wing leading edge is constructed of two skin layers separated by a narrow passageway. (*Figure 12-10*)

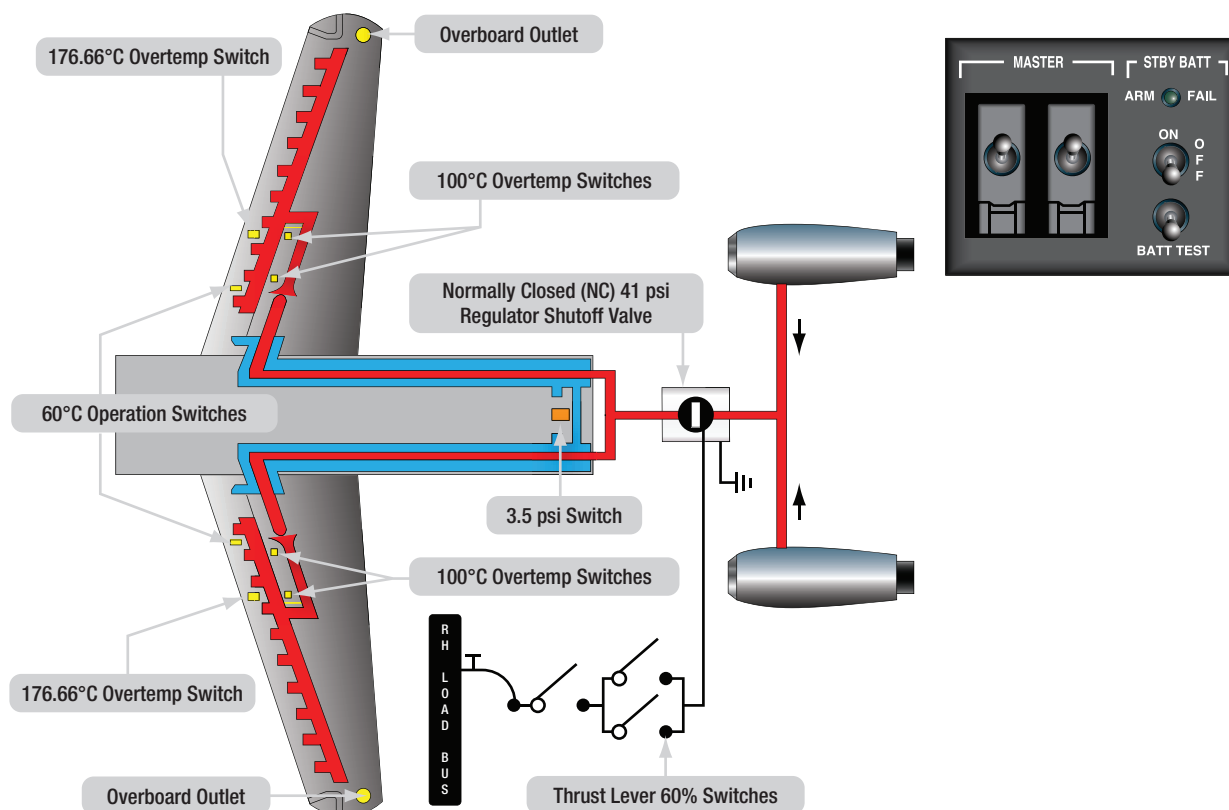


Figure 12-9. Thermal WAI system.

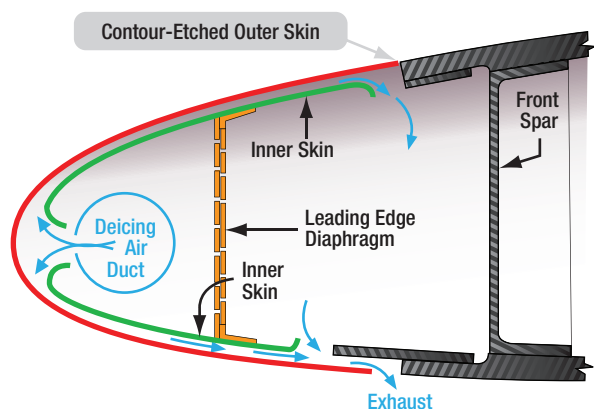


Figure 12-10. Heated wing leading edge.

The air directed against the leading edge can only escape through the passageway, after which it is vented overboard through a vent in the bottom of the wingtip. When the WAI switch is turned on, the pressure regulator is energized and the shutoff valve opens. When the wing leading edge temperature reaches approximately $+60^{\circ}\text{C}$, temperature switches turn ON the operation light above the switch. If the temperature in the wing leading edge exceeds approximately $+100^{\circ}\text{C}$ (outboard) or $+175^{\circ}\text{C}$ (inboard), the red WING OV HT warning light on the annunciator panel illuminates.

Large aircraft WAI systems are similar to the business jet system just described. However, the larger engines with multiple stage compressor bleed air tap-offs usually do not require that ambient air is mixed with the bleed air for temperature control.

The ducting of WAI systems on large aircraft usually consists of aluminum alloy, titanium, stainless steel, or molded fiberglass tubes. The tube, or duct, sections are attached to each other by bolted end flanges or by band type V-clamps. In some locations, the ducting is covered with a fire-resistant, heat-insulating material, such as fiberglass. Thin stainless steel expansion bellows are used at strategic positions in the ducting to absorb any distortion or expansion of the ducting that may occur due to temperature variations. The joined sections of ducting are hermetically sealed by sealing rings. These seals are fitted into annular recesses in the duct joint faces.

When installing a section of duct, make certain that the seal bears evenly against and is compressed by the adjacent duct's flange. When specified, the ducts should be pressure tested at the pressure recommended by the manufacturer of the aircraft concerned. Leak checks are

made to detect defects in the duct that would permit the escape of heated air. The rate of leakage at a given pressure should not exceed that recommended in the aircraft maintenance manual.

Air leaks can often be detected audibly and are sometimes revealed by holes in the lagging or thermal insulation material. However, if difficulty arises in locating leaks, a soap-and-water solution may be used. All ducting should be inspected for security, general condition, or distortion. Lagging or insulating blankets must be checked for security and must be free of flammable fluids, such as oil or hydraulic fluid.

Slat Leading Edges

Most large turbine aircraft are fitted with leading edge slats. This means that it may be the slats that actually receive thermal pneumatic wing anti-ice, not the wing proper. The WAI ducts move air from the pneumatic system through the wing leading edge to the leading edge slats. The ducting warms the cavities through which it is routed. Combined with telescoping ducts that direct air into the slats when extended, retracted or in transit, this is sufficient to keep ice from forming on the entire wing leading edge. Holes in the bottom of each slat allow the pneumatic air to exit the slats. The telescoping duct sections that direct air into the slats attached to the slat on one end, and slide over a narrow diameter "T" section that is connected into the WAI duct. A seal prevents any loss of air. (*Figure 12-11*)

WAI Operation

Large turbine powered aircraft wing anti-ice systems have automatic anti-ice control. The following paragraph explains the operation of the Boeing 777. Other Boeings and Airbus are similar. The primary components of the WAI system are the airfoil and cowl ice protection system (ACIPS) computer logic cards, the WAI valve and the WAI duct pressure sensor. Inputs include the ice detection system and aircraft status inputs. A switch on the flight deck turns the system ON, AUTO, or OFF. In the auto mode, the system turns on when the ice detection system detects ice. The OFF and ON positions are used for manual control of the WAI system. The WAI system is only used in the air, except for ground tests. The weight on wheels system (WOW) and/or airspeed data disarms the system when the aircraft is on the ground. *Figure 12-12* illustrates the WAI system on a modern airliner.

The WAI valve controls the flow of bleed air from the pneumatic system to the WAI ducts. The valve is electrically controlled and pneumatically actuated. The torque motor controls operation of the valve. With no electrical power to the torque motor, air pressure on one side of the actuator holds the valve closed. Electrical current through the torque motor allows air pressure to open the valve. As the torque motor current increases, the valve opening increases. (*Figure 12-13*)

The WAI pressure sensor senses the air pressure in the WAI duct after the WAI valve. The airfoil and cowl ice protection system (ACIPS) uses integrated circuit system logic cards in a card file system. An ACIPS control card uses the WAI pressure information to control the WAI system.

WAI Control

Modern aircraft use several on board computers to control aircraft systems. The WAI system is controlled by the ACIPS computer card. The ACIPS computer card controls both WAI valves. The required positions of the WAI valves change as bleed air temperature and altitude change. The left and right valves operate at the same time to heat both wings equally. This keeps the airplane aerodynamically stable in icing conditions. The WAI pressure sensors supply feedback information to the WAI ACIPS computer card for WAI valve control and position indication. If either pressure sensor fails, the WAI ACIPS computer card sets the related WAI valve to either fully open or fully closed. If either valve fails closed, the WAI computer card keeps the other valve closed.

As stated, there is one selector for the WAI system. The selector has three positions: AUTO, ON, and OFF. With the selector in AUTO and no operational mode inhibits, the WAI ACIPS computer card sends a signal to open the WAI valves when either ice detector detects ice. The valves close after a 3-minute delay when the ice detector no longer detects ice. The time delay prevents frequent ON/OFF cycles during intermittent icing conditions. With the selector ON and no operational mode inhibits, the WAI valves open. With the selector OFF, the WAI valves close. The operational mode for the WAI valves can be inhibited by many different sets of conditions. (*Figure 12-14*)

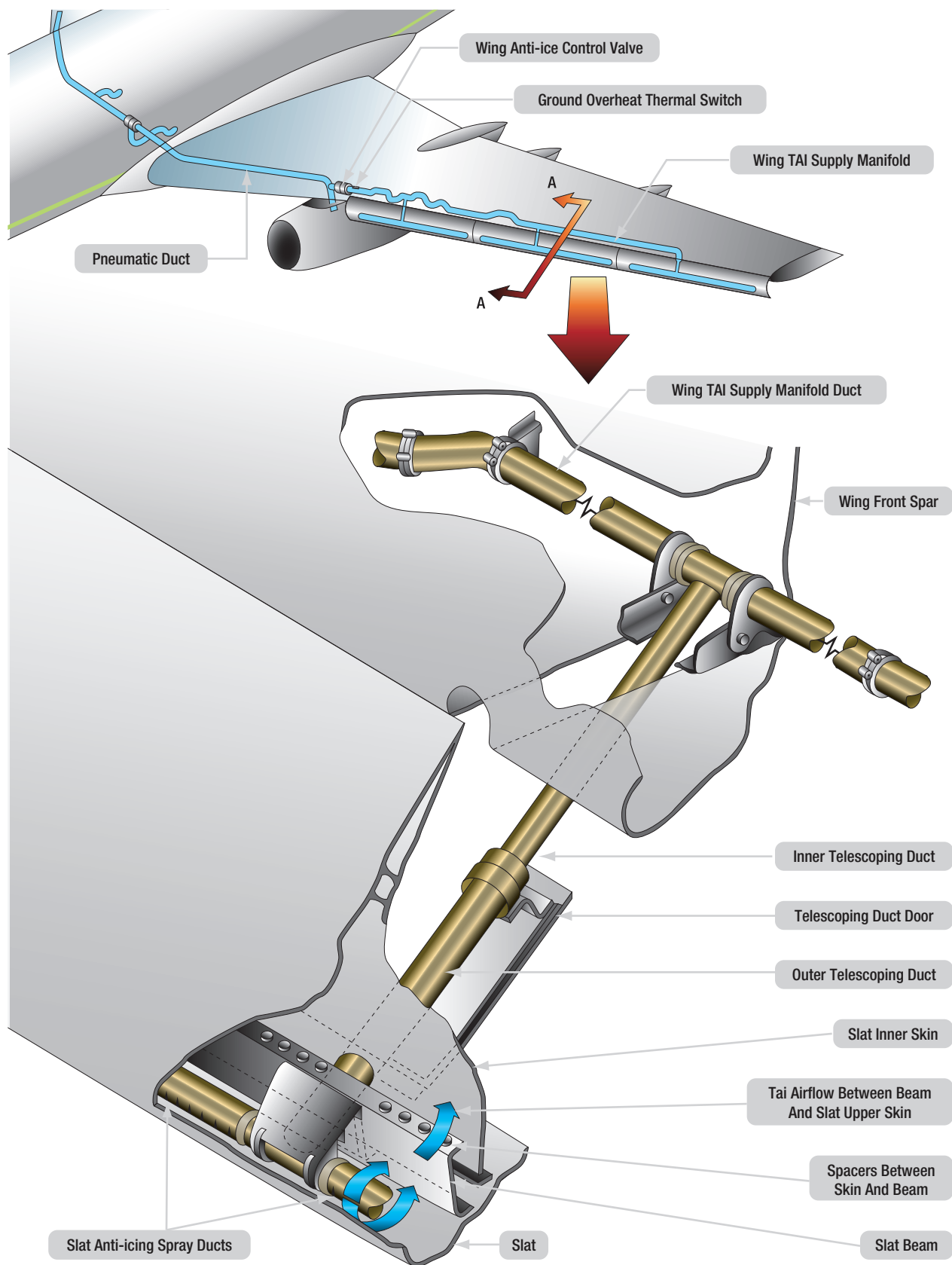


Figure 12-11. WAI ducting.

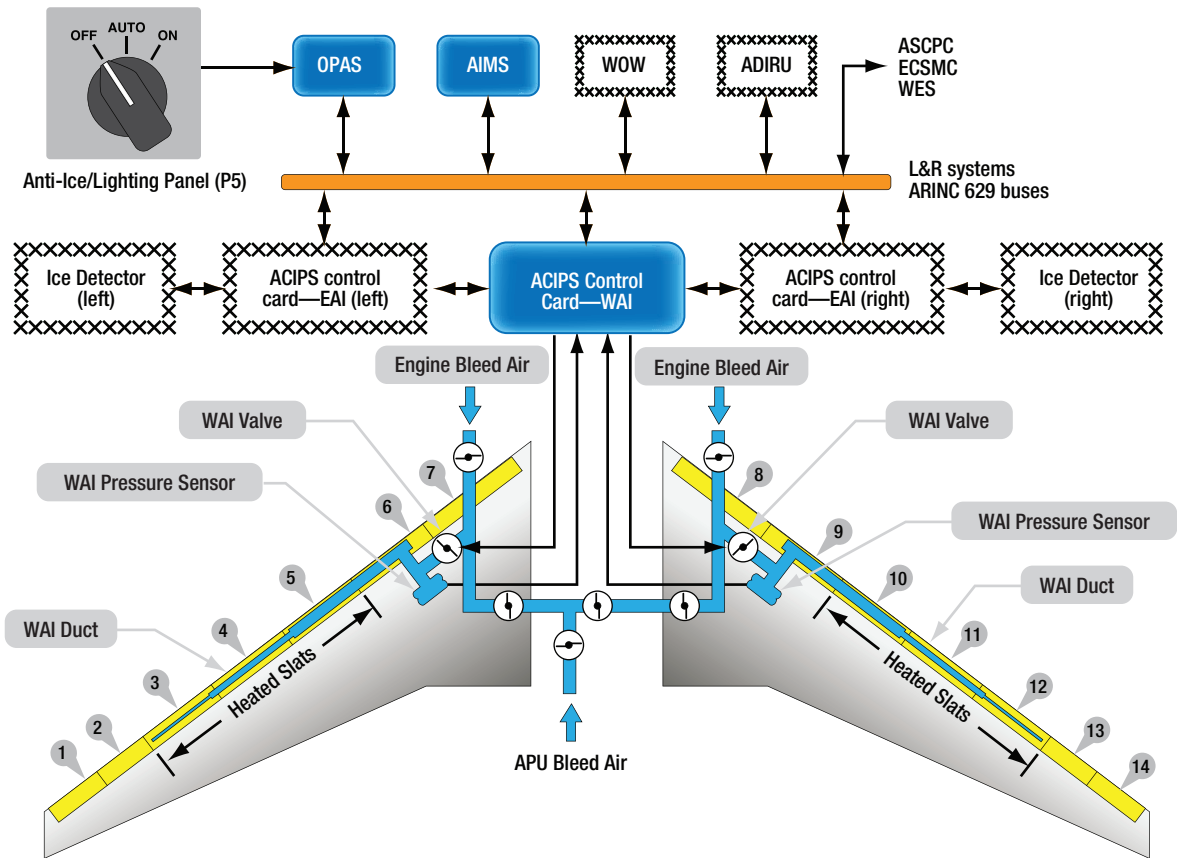


Figure 12-12. Wing leading edge slat anti-ice system.

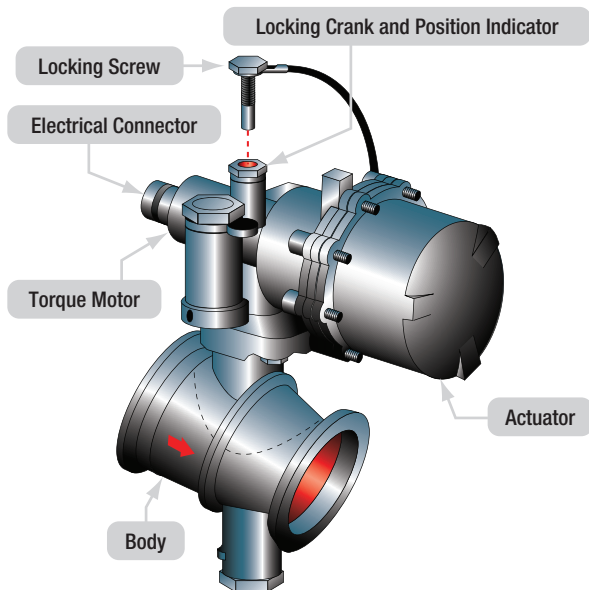


Figure 12-13. A wing anti-ice valve.

The operational mode is inhibited if all of these conditions occur:

- Auto mode is selected.
- Takeoff mode is selected.
- Airplane has been in the air less than 10 minutes.

With AUTO or ON selected, the operational mode is inhibited if any of these conditions occur:

- Airplane on the ground (except during an initiated or periodic built-in test equipment (BITE) test).
- Total Air Temperature (TAT) is more than 10°C and the time since takeoff is less than 5 minutes.
- Auto slat operation.
- Air driven hydraulic pump operation.
- Engine start.
- Bleed air temperature less than 93°C.

The WAI valves stay closed as long as the operational mode inhibit is active. If the valves are already open, the operational mode inhibit causes the valves to close.

WAI Indication System

The aircrew can monitor the WAI system on the on board computer maintenance page. (*Figure 12-15*)

The following information is shown:

- WING MANIFOLD PRESS - pneumatic duct pressure in PSIG
- VALVE - WAI valve open, closed, or regulating

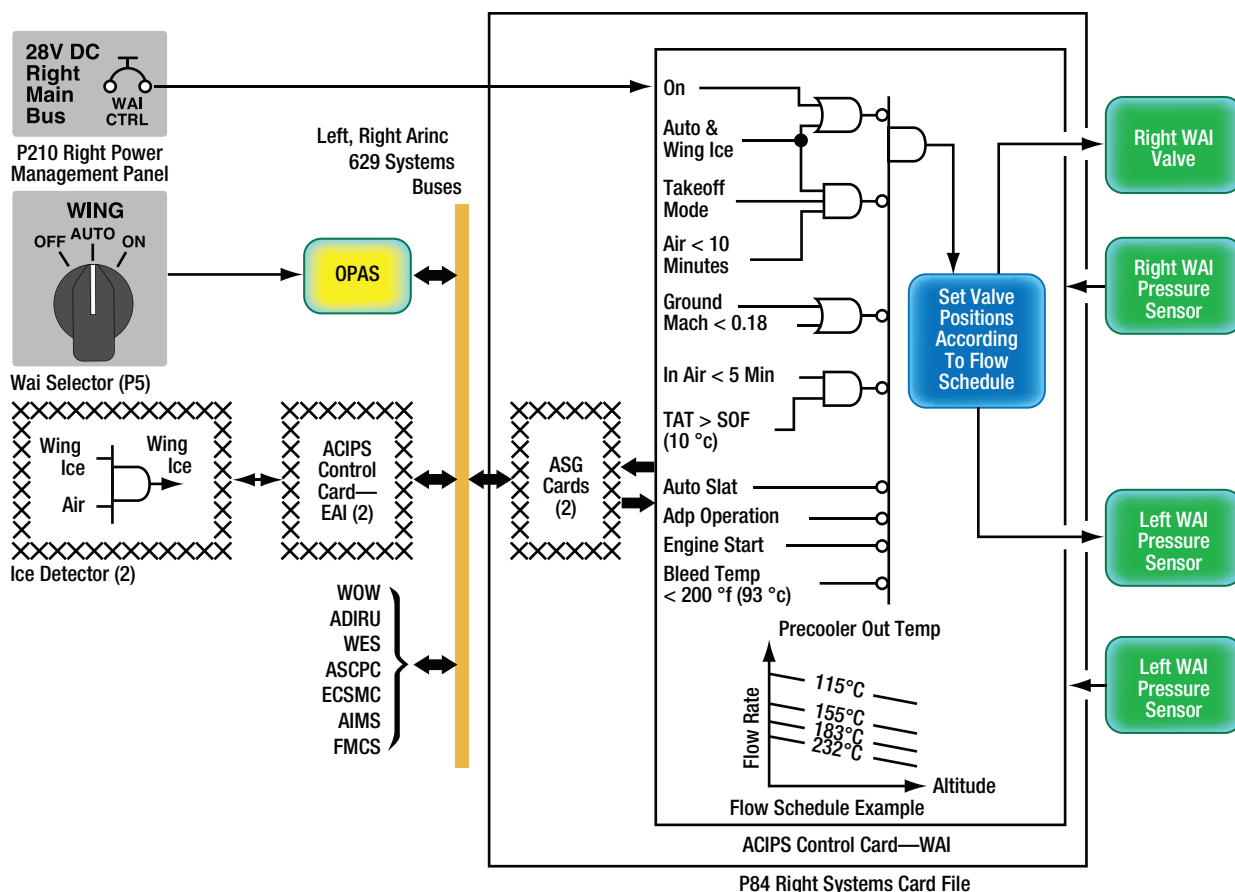


Figure 12-14. WAI inhibit logic schematic.

ICE PROTECTION		
ALTITUDE 10,000	ENG TYPE - -	
TAT -2		
ICE DETECTION:	L	R
ENGINE ANTI-ICE:	ENGINE/WING	ENGINE/WING
FINCISE DUCT LEAK SIGNAL	NORMAL	NORMAL
VALVE	REGULATING	REGULATING
SUPPLY AIR TEMP	884	884
AIR PRESSURE	13	13
AIR FLOW	13	13
WING ANTI-ICE:		
WING MANIFOLD PRESSURE	50	50
VALVE	REGULATING	REGULATING
AIR PRESSURE	19	19
AIR FLOW	85	85

Figure 12-15. Ice protection on board computer maintenance page.

- AIR PRESS - pressure downstream of the WAI valves in PSIG
- AIR FLOW - air flow through the WAI valves in pounds per minute

WAI System BITE Test

Built in test equipment (BITE) circuits in the WAI ACIPS computer card continuously monitor the WAI

system. Faults that affect the dispatch of the aircraft cause status messages. Other faults cause central maintenance computer system (CMCS) maintenance messages. The BITE in the WAI ACIPS computer card also performs automatic power-up and periodic tests. The power-up test occurs when the card gets power. BITE does a test of the card hardware and software functions and the valve and pressure sensor interfaces. The valves do not move during this test. The periodic test occurs when all these conditions are true:

- The airplane has been on the ground between 1 and 5 minutes.
- The WAI selector is set to AUTO or ON.
- Air driven hydraulic pumps are not in intermittent operation.
- Bleed pressure is sufficient to open the WAI valves.
- The time since the last periodic test is more than 24 hours.

During this test, the WAI valves cycle open and closed. This test makes sure that valve malfunctions are detected.

ENGINE ANTI-ICE (EAI)

In addition to thermal pneumatic wing anti-ice, large turbine powered aircraft also have thermal pneumatic engine inlet anti-ice (EAI). It is extremely important that ice not be allowed to build on the engine inlet cowl. Should ice form and then break off, it is ingested by the engine and could cause engine damage. This is why Boeing engine anti-ice is automatically turned on before wing anti-ice when the ice detection system begins to cycle the probe heat. The EAI operates similarly to the WAI. Bleed air supplied from a high stage compressor bleed port is ducted to the leading edge of the engine inlet cowls. It exits the cowl through overboard vents. A pneumatically actuated EAI valve controls the flow of the warm bleed air to the inlet cowl. The valve is supplied control pressure from an EAI controller. The controller has a torque motor that moves in response to ACIPS - EAI logic card signals. It regulates activation pressure to the EAI valve. (**Figure 12-16**)

Signals from the ice detection system are delivered to the EAI ACIPS logic control card along with EAI duct pressure information from sensors downstream of the EAI valve. The logic card circuits control the operation of the EAI controller which positions the EAI valve.

An air to air heat exchanger cools the bleed air used by the EAI controller, the pressure regulating and shutoff valve (PRSOV) and the high pressure and fan air controller (HPFAC). **Figure 12-17** illustrates the engine anti-ice system.

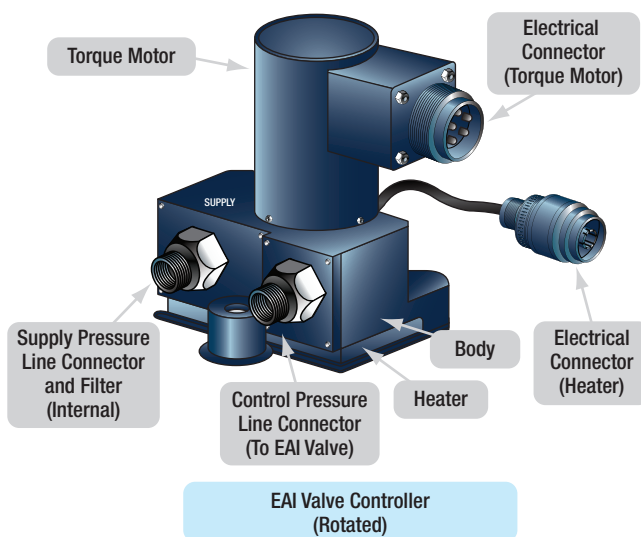


Figure 12-16. The EAI controller regulates air that operates the EAI anti-ice valve.

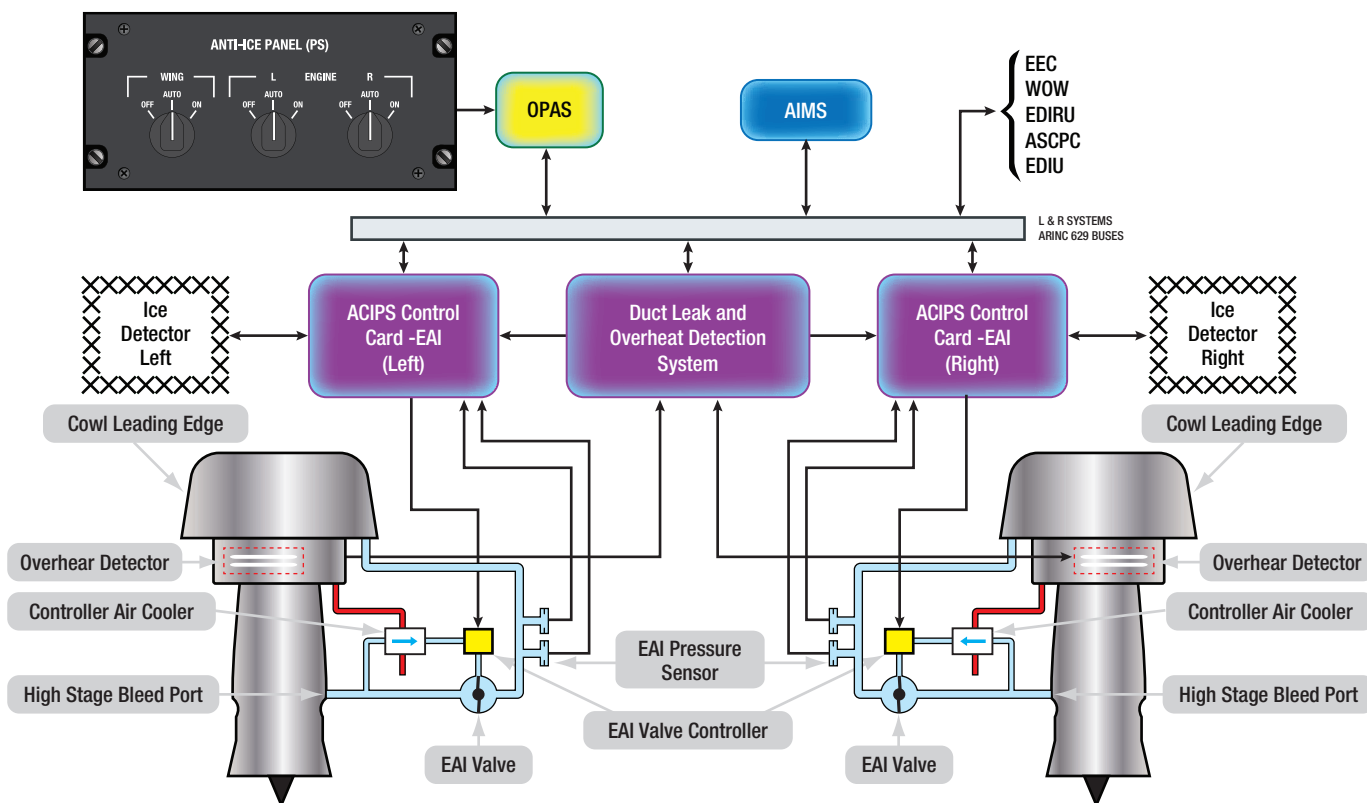


Figure 12-17. Thermal pneumatic engine anti-ice system on a Boeing 777.

EAI is automatic when the anti-ice switch is set to AUTO. To operate EAI on the ground, ON must be selected. Built in test Equipment BITE is active with the switch position ON or AUTO on the ground and in the air. EAI indications and warnings are displayed on the flight deck EICAS displays.

THERMAL ELECTRIC ANTI-ICING

Electricity is used to heat various components on an aircraft so that ice does not form. This type of anti-ice is typically limited to small components due to high amperage draw. Effective thermal electric anti-ice is used on most air data probes, such as pitot tubes, static air ports, TAT and AOA probes, ice detectors, and engine P2/T2 sensors. Water lines, waste water drains, and some turboprop inlet cowls are also heated with electricity to prevent ice from forming. Transport category and high performance aircraft use thermal electric anti-icing in windshields.

PROBE ANTI-ICE

In devices that use thermal electric anti-ice, current flows through an integral conductive element that produces heat. The temperature of the component is elevated above the freezing point of water so ice cannot form. Various schemes are used, such as an internal coil wire, externally wrapped blankets or tapes, as well as conductive films and heated gaskets. A basic discussion of probe heat follows.

Data probes that protrude into the ambient airstream are particularly susceptible to ice formation in flight. **Figure 12-18** illustrates the types and location probes that use thermal electric heat on one airliner. A pitot tube, for example, contains an internal electric element that is controlled by a switch in the cockpit. Use caution checking the function of the pitot heat when the aircraft is on the ground. The tube gets extremely hot since it must keep ice from forming at altitude in temperatures near -45°C at speeds possibly over 500 miles per hour. An ammeter or load meter in the circuit can be used as a substitute to touching the probe, if so equipped. Simple probe heat circuits exist on most with a switch and a circuit breaker to activate and protect the device.

Advanced aircraft may have more complex circuitry in which control is by computer and flight status condition of the aircraft is considered before thermal electric heaters are activated automatically. **Figure 12-19** shows such a circuit for a pitot tube. The primary flight computer (PFC) supplies signals for the air data card (ADC) to energize ground and air heat control relays to activate probe heat. Information concerning speed of the aircraft, whether it is in the air or on the ground, and if the engines are running are factors considered by the ADC logic. Similar controls are used for other probe heaters.

WATER SYSTEM AND DRAIN ANTI-ICE

Transport type aircraft have water and waste systems on board, and electrical heaters are often used to prevent

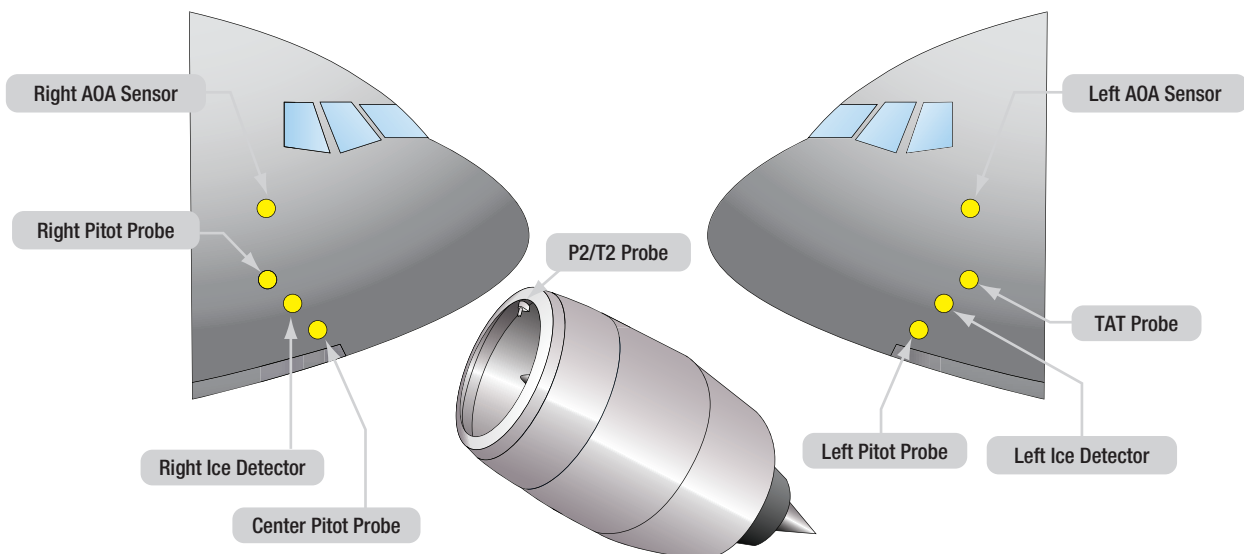


Figure 12-18. Probes with thermal electric anti-icing on one commercial airliner.

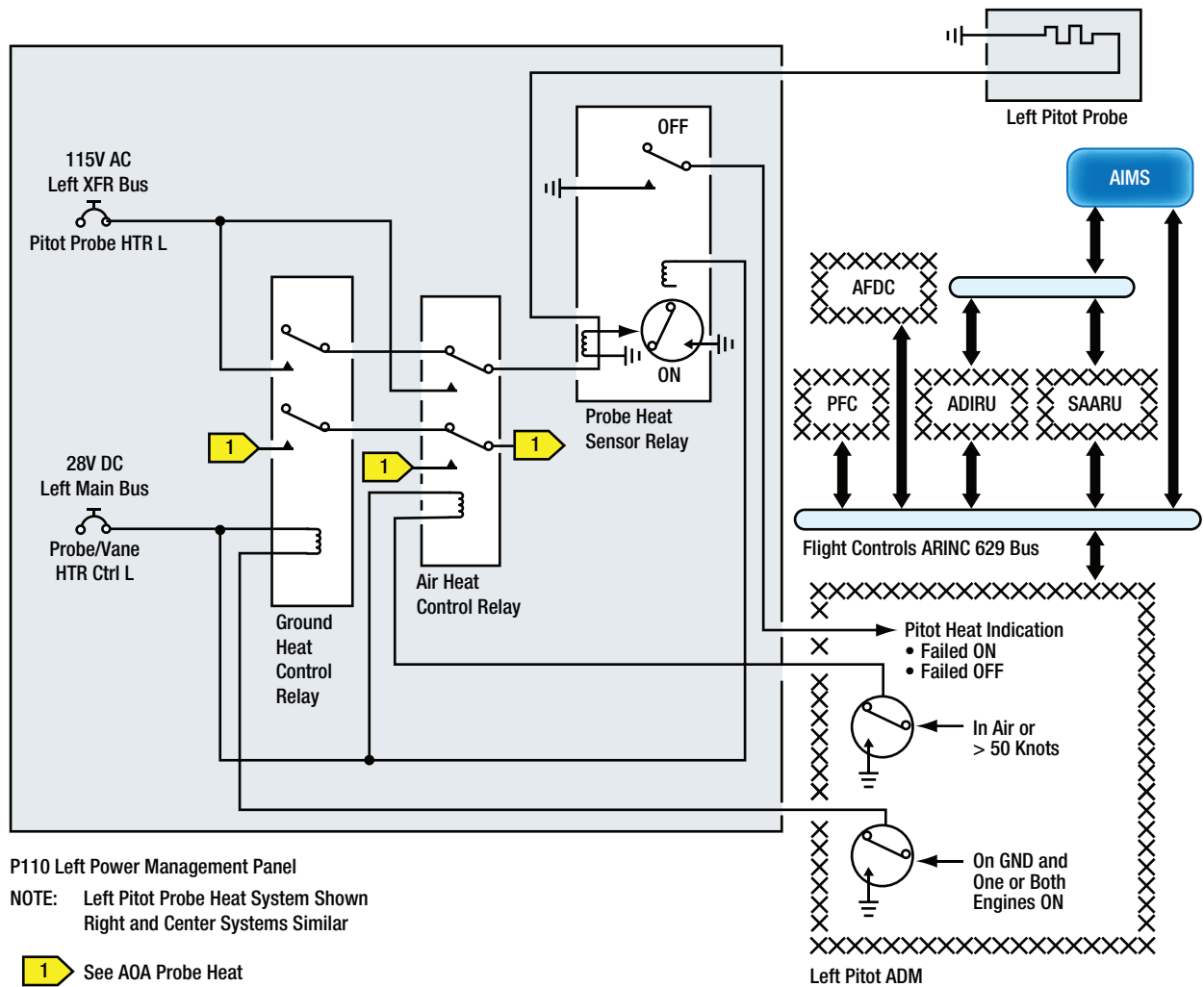


Figure 12-19. Pitot probe heat system.

the formation of ice in the water lines of these systems. Water lines carry water from the potable tanks to the lavatories and galleys. The waste water tanks collect the gray water from the galleys and lavatories. Heater blankets, in-line heaters, or heater boots are often used to heat the water supply lines, water tank drain hoses, waste drain lines, waste tank rinse fittings, and drain masts. Thermostats in the water lines supply temperature data to the control unit that turns the electrical heaters on and off. When the temperature falls below freezing, the electrical heaters turn on and stay on until the temperature reaches a safe temperature. **Figure 12-20** is a schematic of a water supply line heater system, and **Figure 12-21** shows the location of the waste water tanks and heater blanket.

On modern aircraft, the particular heating device and the thermostat that controls it are line replaceable units and easily changed by the technician if inoperative.

Drain mast electric heating elements are integral and require that the mast be replaced. Drain line heating elements are either flexible wrap type or integral. Consult the manufacturer's maintenance and parts manual for replacement information.

WINDSHIELD ANTI-ICE

High performance and transport category aircraft windshields are typically made of laminated glass, polycarbonate, or similar ply material. Typically clear vinyl plies are also included to improve performance characteristics. The laminations create the strength and impact resistance of the windshield assembly. These are critical feature for windshields as they are subject to a wide range of temperatures and pressures. They must also withstand the force of a 4 pound bird strike at cruising speed to be certified. The laminated construction facilitates the inclusion of electric heating elements into the glass layers, which are used to keep

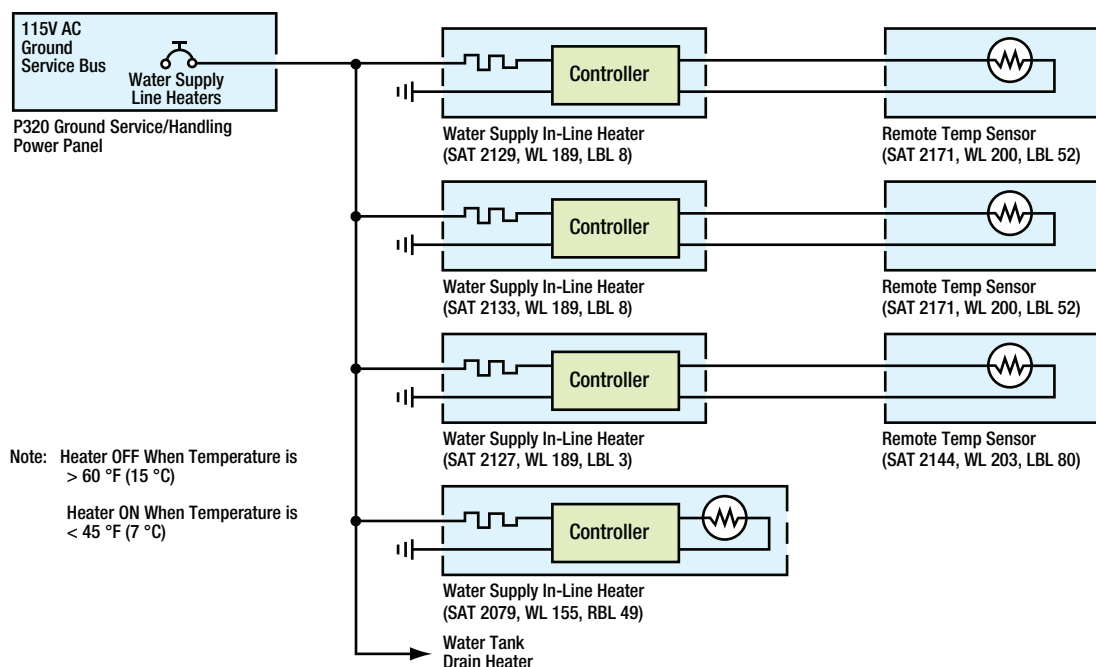


Figure 12-20. Water supply line heater system.

the windshield clear of ice, frost, and fog. The elements can be in the form of resistance wires or a transparent conductive material may be used as one of the window plies. To ensure enough heating is applied to the outside of the windshield, heating elements are placed on the inside of the outer glass ply. Windshields are typically bonded together by the application of pressure and heat without the use of cement. **Figure 12-22** illustrates the plies in one transport category aircraft windshield.

Whether resistance wires or a laminated conductive film is used, aircraft window heat systems have window heat control units to supply power and feedback mechanisms, such as thermistors, to provide the window heat control units with information used to keep operating temperature within acceptable limits. Most systems are automatic once switched with switches on the flight deck. Separate circuits for pilot and co-pilot are common to ensure visibility in case of a malfunction.

Consult the manufacturer's maintenance information for details on the particular window heat system in question. Some windshield heating systems can be operated at two heat levels. On these aircraft, NORMAL heating supplied heat to the broadest area of windshield. HIGH heating supplies a higher intensity of heat to a smaller but more essential viewing area. Typically, this window heating system is always on and set in the NORMAL position.

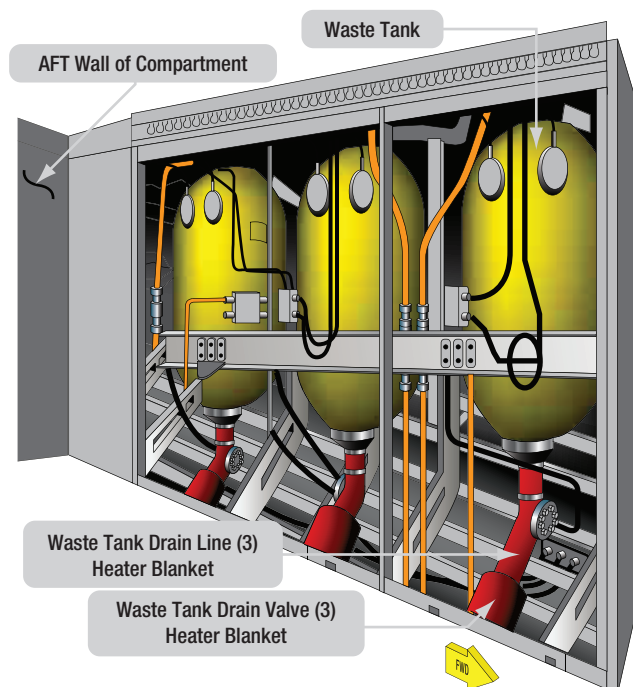


Figure 12-21. Waste water tanks and tank drain valve heater blankets.

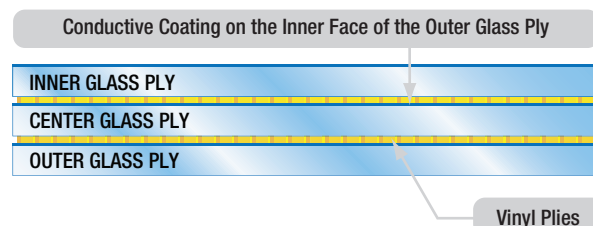


Figure 12-22. Cross-section of a transport category windshield.

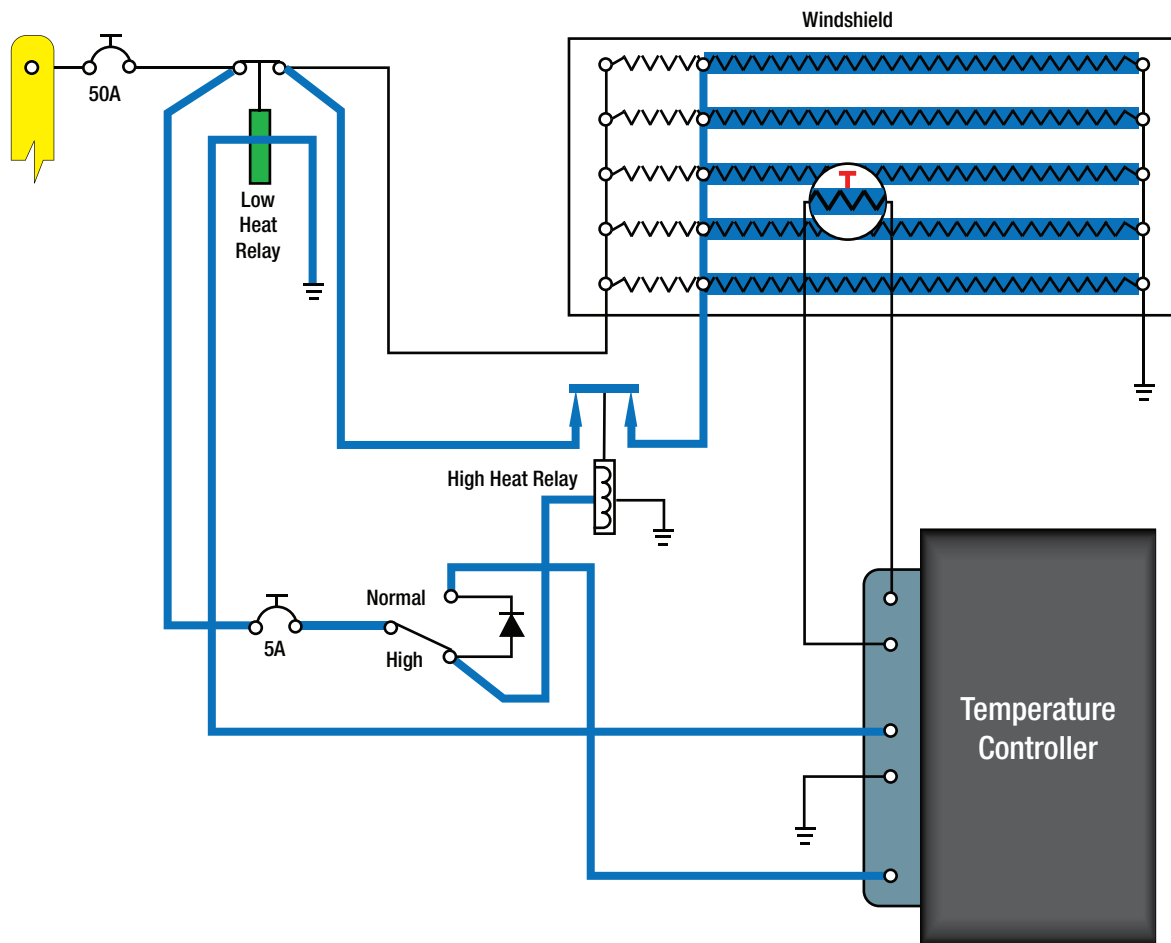


Figure 12-23. Electric windshield heat schematic.

Figure 12-23 illustrates a simplified windshield heat system of this type. Window heat anti-ice systems typically BITE test automatically. They are connected to an aircraft data bus for communication with the aircraft information management system.

Figure 12-24 is a synoptic diagram of a Boeing 777 window heat system. On this aircraft, the back-up window heat is always ON. When selecting window heat, the crew is actually asking for more heat from the window heat control units.

PROPELLER ANTI-ICE

Many propellers use thermal electric boots to remove any ice that forms on the blades (de-ice). However some aircraft permit operation of the electric heating element boots to prevent ice from forming. In this case, the boots perform an anti-icing function. More information on this type of propeller ice control is given in the de-ice section of this *Sub-Module*.

CHEMICAL ANTI-ICING

Chemical anti-icing is used in some aircraft to anti-ice the leading edges of the wing, stabilizers, windshields, and propellers. The wing and stabilizer systems are often

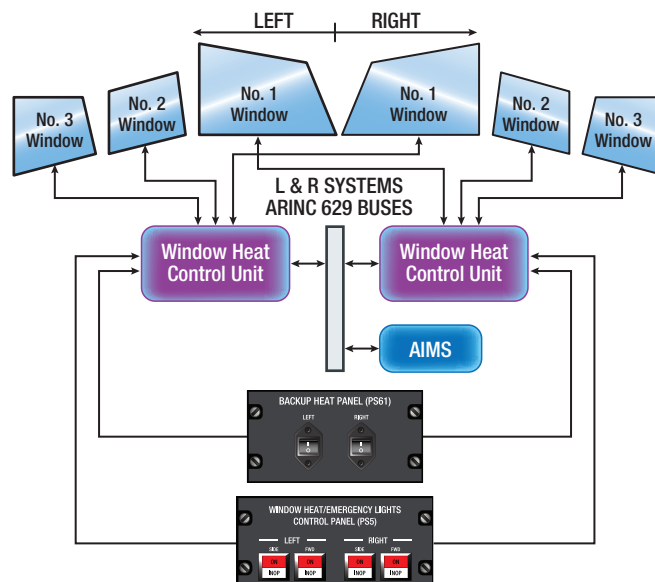


Figure 12-24. Electric window heat synoptic diagram for a Boeing 777.

called weeping wing systems or are known by their trade name of TKS™ systems. Ice protection is based upon the freezing point depressant concept. An antifreeze solution is pumped from a reservoir through a mesh screen embedded in the leading edges of the wings and stabilizers. Activated by a switch in the cockpit, the liquid flows over the wing and tail surfaces, preventing the formation of ice as it flows. The solution mixes with the supercooled water in the cloud, depresses its freezing point, and allows the mixture to flow off of the aircraft without freezing.

The system is designed to anti-ice, but it is also capable of de-icing an aircraft as well. When ice has accumulated on the leading edges, the antifreeze solution chemically breaks down the bond between the ice and airframe. This allows aerodynamic forces to carry the ice away. Thus, the system clears the airframe of accumulated ice before transitioning to anti-ice protection. **Figure 12-25** shows a chemical anti-ice system. The TKS™ weeping wing system contains formed titanium panels that are laser drilled with over 800 tiny holes (.002 5 inch diameter) per square inch.

These are mated with non-perforated stainless steel rear panels and bonded to wing and stabilizer leading edges. As fluid is delivered from a central reservoir and pump, it seeps through the holes. Aerodynamic forces cause the fluid to coat the upper and lower surfaces of the airfoil. The glycol based fluid prevents ice from adhering to the aircraft structure. Some aircraft with weeping wing systems are certified to fly into known icing conditions.

Others use it as a hedge against unexpected ice encountered in flight. The systems are basically the same. Reservoir capacity permits 1-2 hours of operation. TKS™ weeping wings are used primarily on reciprocating aircraft that lack a supply of warm bleed air for the installation of a thermal anti-ice system. However, the system is simple and effective leading to its use on some turbine powered corporate aircraft as well.

GROUND APPLIED ANTI-ICE

When aircraft surfaces are contaminated by frozen moisture, they must be deiced prior to dispatch. When freezing precipitation exists and there is a risk of recontamination of the surface before takeoff, aircraft surfaces must be anti iced.

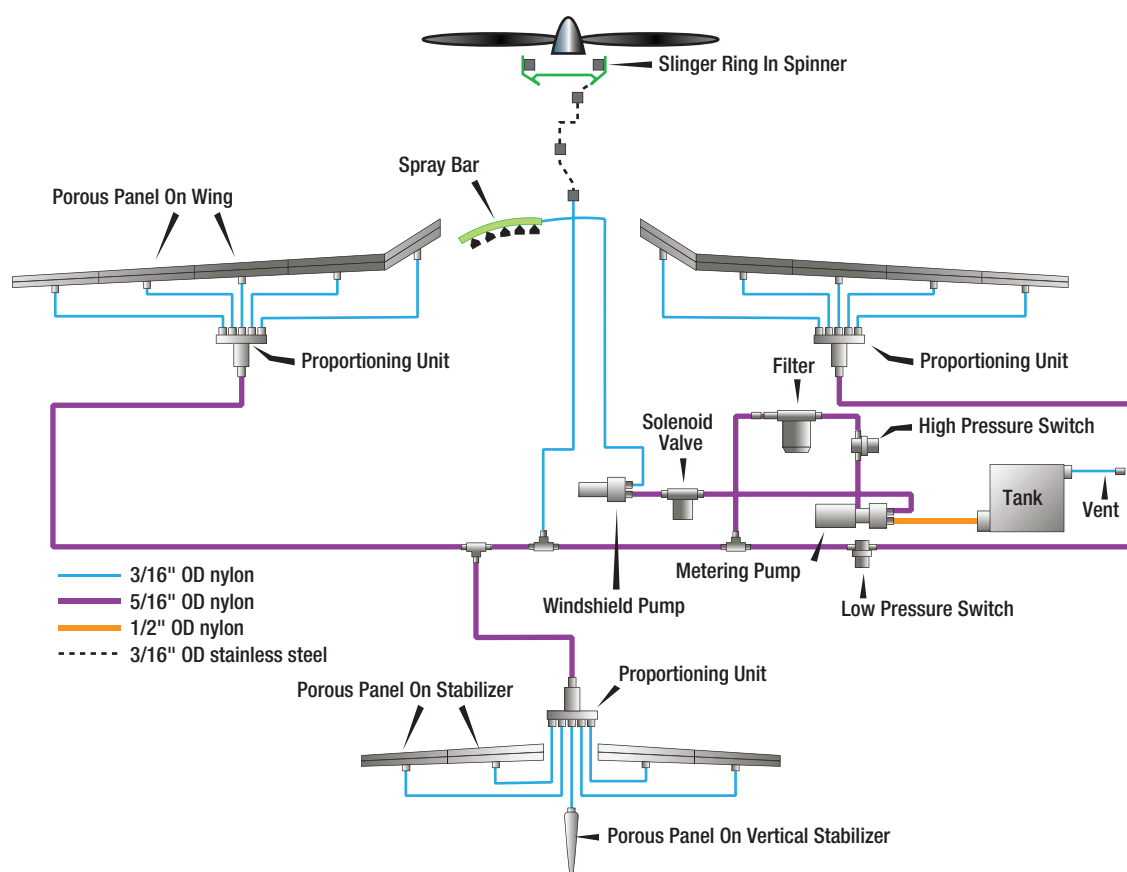


Figure 12-25. Chemical anti-ice system.

Ground de-icing of an aircraft is performed by a crew outside the aircraft equipped with spray equipment and often a boom truck to facilitate access. There are many formulas for deicing fluid. An ethylene glycol or propylene glycol based liquid is typical. It is mixed with hot water and sprayed on the aircraft being careful to avoid spraying it into critical areas of the aircraft such as:

- Engine inlets.
- Probes and ports.
- Air conditioning inlets and exits.
- APU inlet.
- Cooler and heat exchanger inlets.
- Fuel tank vents.

Type I deice fluid has few additives. It is an effective de-icing agent depending on the ratio of water to glycol and the temperature of the fluid when applied as well as the ambient conditions. A smooth thin layer of Type I fluid remains on the aircraft after the de-icing application which acts as an anti-icing agent. However, Type I fluid has low viscosity and its anti-ice capacity lasts only a few minutes.

Type II and Type IV de-ice fluids are commonly used on large turbine powered aircraft. Type II fluid is propylene glycol based fluid with molecular polymers added as a thickener. It becomes thixotropic fluid which becomes less viscous as stress is applied. As such, when applied to de-ice the aircraft, Type II becomes a thicker coat adhering to the airfoil surfaces and protects against new ice, snow or frost from forming. As the aircraft airspeed increases during takeoff rollout and flight, the force of the air against the fluid layer decreases its viscosity and it is blown off.

Type IV fluid is similar to Type II fluid in that it has significant additives and leaves a thixotropic coating once applied. The anti-icing capabilities are greater for Type IV fluid which is evident by longer holdover times. A holdover time is the time between when the anti-ice fluid is applied and when it is no longer effective. This is discussed further in the section on ground de-icing.

DE-ICING SYSTEMS

When ice, snow or frost are allowed to accumulate on aircraft surfaces and then are removed, the process is known as de-icing. Smaller turbine powered aircraft and reciprocating aircraft often incorporate de-ice systems rather anti-ice system although some aircraft may use a combination of de-ice and anti-ice for overall ice protection.

PNEUMATIC DE-ICE BOOTS

The most common means for de-icing wings and stabilizers on small turbine powered aircraft and reciprocating engine aircraft is with pneumatic de-ice boots. The leading edges of the wings and stabilizers have inflatable boots attached to them. The boots expand when inflated by pneumatic pressure, which breaks away ice accumulated on the boot. Most boots are inflated for 6 to 8 seconds. They are deflated by vacuum suction. The vacuum is continuously applied to hold the boots tightly against the aircraft while not in use.

CONSTRUCTION AND INSTALLATION OF DE-ICE BOOTS

De-icer boots are made of soft, pliable rubber, or rubberized fabric, and contain tubular air cells. The outer ply of the de-icer boot is of conductive neoprene to provide resistance to deterioration by the elements and many chemicals. The neoprene also provides a conductive surface to dissipate static electricity charges. These charges, if allowed to accumulate, would eventually discharge through the boot to the metal skin beneath, causing static interference with the radio equipment. (*Figure 12-26*)

On modern aircraft, the de-icer boots are bonded with an adhesive to the leading edge of wing and tail surfaces. The trailing edges of this type boot are tapered to provide a smooth airfoil. Elimination of fairing strips, screws, and rivnuts used on older types of de-icing boots reduces the weight of the de-ice system. The de-icer boot air cells are connected to system pressure and vacuum lines by non-kinking flexible hose.

When gluing the de-ice boots to the leading edge of wings and stabilizers, the manufacturer's instruction must be strictly followed. The glue is typically a contact cement normally spread on both the airfoil and the boot and allowed to become tacky before mating the surfaces. Clean, paint-free surfaces are required for the glue to adhere properly. Removal of old boots is performed by re-softening the cement with solvent.

SOURCES OF OPERATING AIR

The source of operating air for de-ice boot systems varies with the type of powerplant installed on the aircraft. Reciprocating engine aircraft typically use a dedicated engine-driven air pump mounted on the accessory drive gear box of the engine.



Figure 12-26. Deicing boots inflated (left) and deflated (right).

The suction side of the pump is used to operate the gyroscopic instruments if installed on the aircraft. It is also used to hold the de-ice boots tight to the aircraft when they are not inflated. The pressure side of the pump supplies air to inflate the de-ice boots, which breaks up ice that has formed on the wing and stabilizer leading edges. The pump operates continuously. Valves, regulators, and switches in the cockpit are used to control the flow of source air to the system.

Turbine Engine Bleed Air

The source of de-ice boot operating air on turbine engine aircraft is typically bleed air from the engine compressor(s). A relatively low volume of air on an intermittent basis is required to operate the boots. This has little effect on engine power enabling use of bleed air instead of adding a separate engine-driven air pump. Valves controlled by switches in the cockpit deliver air to the boots when requested.

RECIPROCATING ENGINE AIRCRAFT DE-ICE SYSTEMS

General aviation reciprocating engine aircraft, especially twin-engine models, are commonly equipped with pneumatic de-icer systems. Rubber boots are attached with glue to the leading edges of the wings and stabilizers. These boots have a series of inflatable tubes. During operation, the tubes are inflated and deflated in an alternating cycle. (*Figure 12-27*) This inflation and deflation causes the ice to crack and break off. The ice is then carried away by the airstream. Boots used in GA aircraft typically inflate and deflate along the length of the wing.

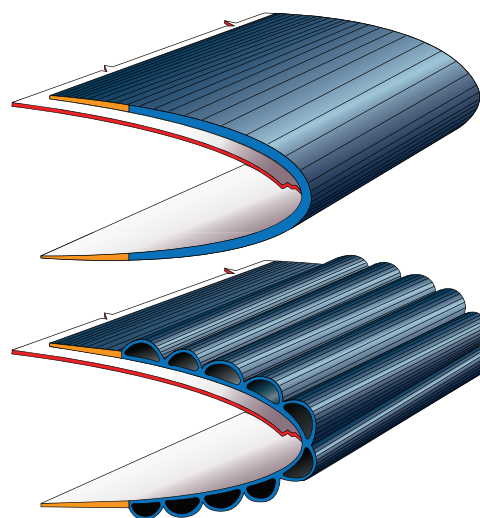


Figure 12-27. Pneumatic deicing system for a twin engine GA aircraft.

In larger turbo prop aircraft, the boots are installed in sections along the wing with the different sections operating alternately and symmetrically about the fuselage. This is done so that any disturbance to airflow caused by an inflated tube is kept to a minimum by inflating only short sections on each wing at a time.

System Operation

Figure 12-28 shows a de-ice system used on a GA twin-engine aircraft. In normal flight, all of the components in the de-ice system are de-energized. Discharge air from the dry air pumps is dumped overboard through the de-ice control valves. The deflate valve is open connecting the de-ice boots to the suction side of the pump through the check valve manifold and the vacuum regulator. The gyroscopic instruments are also connected to the vacuum side of the dry air pump. The vacuum regulator is set to supply the optimum suction for the gyros, which is sufficient to hold the boots tightly against the airfoil surfaces.

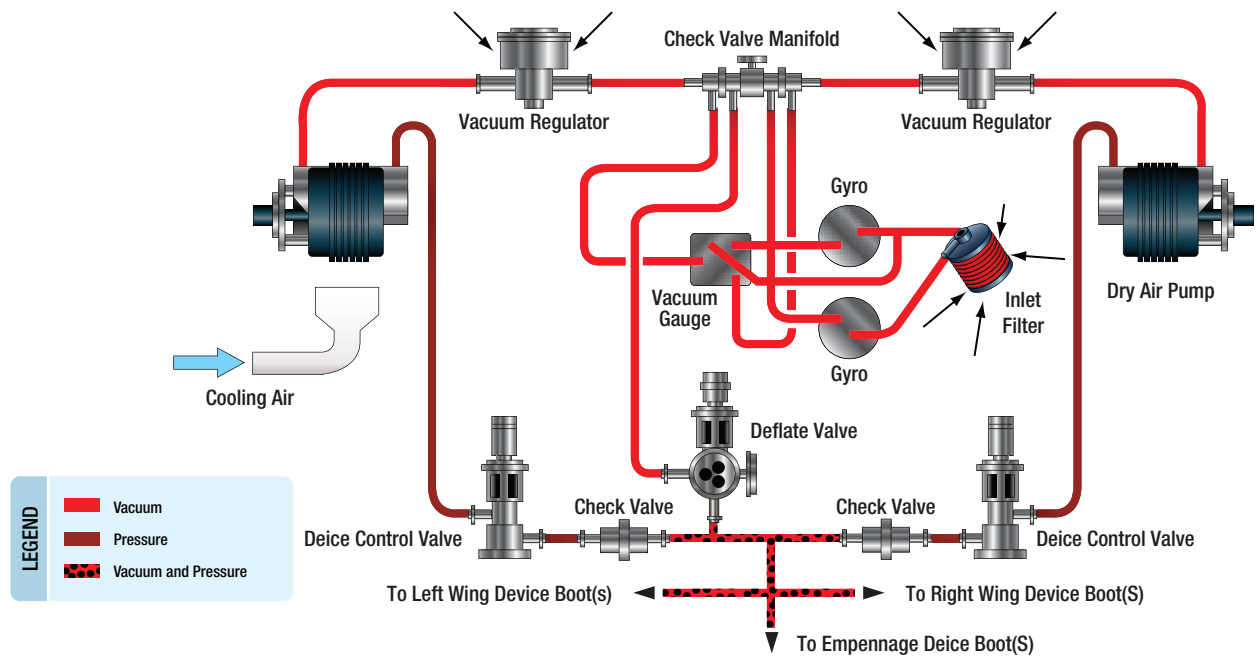


Figure 12-28. Pneumatic deicing system for a twin engine GA aircraft.

When the switch shown in *Figure 12-29* is pushed ON, the solenoid-operated de-ice control valves in each nacelle open and the deflate valve energizes and closes. Pressurized air from the discharge side of the pumps is routed through the control valves to the de-ice boot. When the system reaches 17 psi, pressure switches located on the deflate valve de-energize the de-ice control valve solenoids. The valves close and route pump air output overboard. The deflate valve opens and the boots are again connected to vacuum. On this simple system, the pilot must manually start this inflation/deflation cycle by pushing the switch each time de-ice is required.

Larger aircraft with more complex systems may include a timer, which will cycle the system automatically until turned OFF. The use of distributor valves is also common. A distributor valve is a multi-position control valve controlled by the timer. It routes air to different de-ice boots in a sequence that minimizes aerodynamic disturbances as the ice breaks off the aircraft. Boots are inflated symmetrically on each side of the fuselage to maintain control in flight while de-icing occurs. Distributor valves are solenoid operated and incorporate the deflate valve function to reconnect the de-ice boots with the vacuum side of the pump after all have been inflated. Combining functional components of a de-ice system into a single unit is fairly common.



Figure 12-29. Wing deice switch.

Figure 12-30 illustrates the right side of de-ice boot system. The left side is the same. In addition to the distributor valves, which combine functions of a control valve and deflate valve, the system also uses a combination unit. This unit combines the functions of a shutoff control valve for all pump supply air, as well as a pressure regulator for the system. It also contains a secondary air filter.

DE-ICE SYSTEM FOR TURBOPROP AIRCRAFT

Figure 12-31 shows a pneumatic de-ice system used on a turboprop aircraft. The source of pneumatic air is engine bleed air, which is used to inflate two inboard wing boots, two outboard boots, and horizontal stabilizer boots. Additional bleed air is routed through the brake de-ice valve to the brakes. A three position

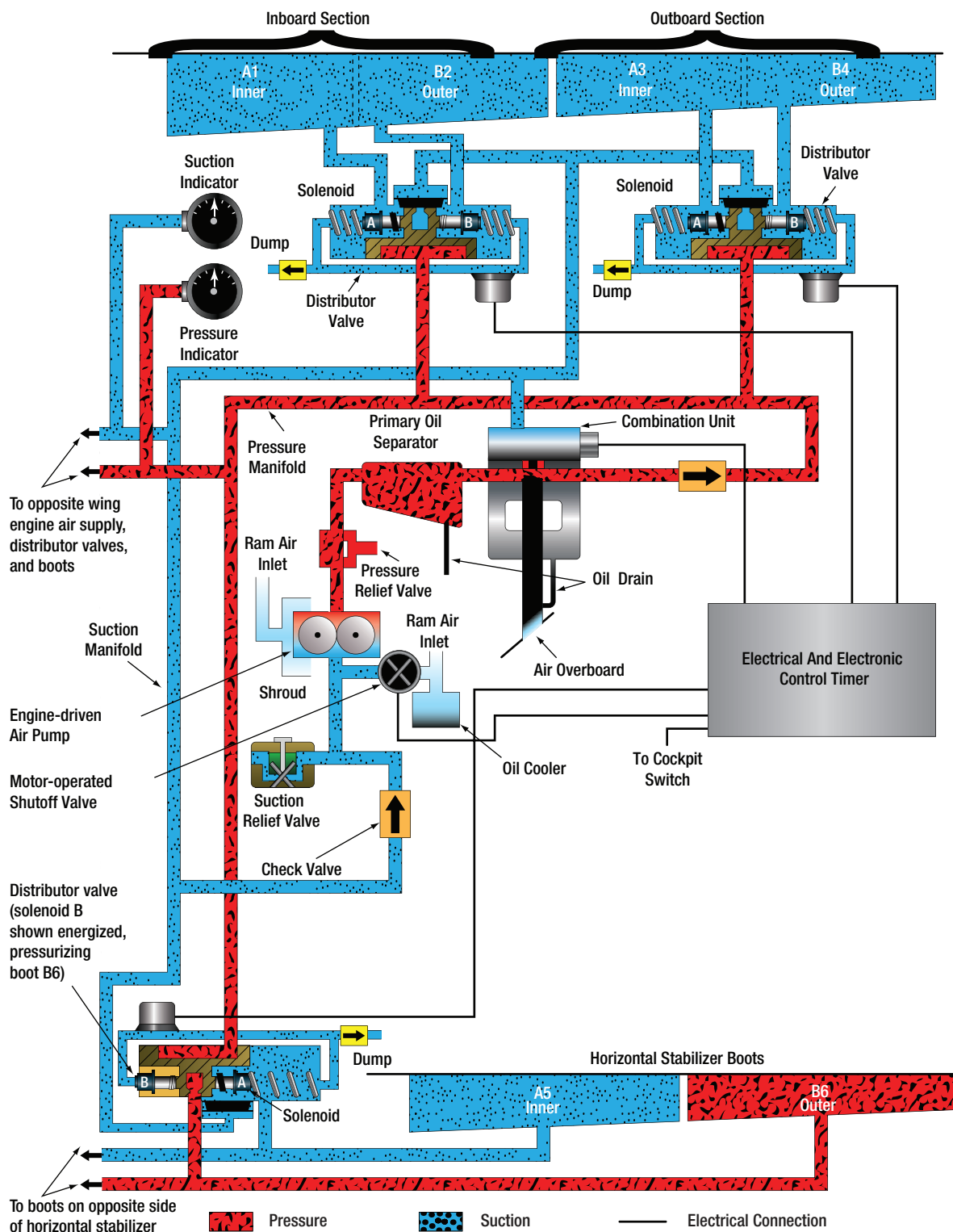


Figure 12-30. Right-side deice boot system on a large aircraft (left side similar).

switch controls the operation of the boots. This switch is spring loaded to the center OFF position. When ice has accumulated, the switch should be selected to the single cycle (up) position and released. (*Figure 12-32*)

Pressure regulated bleed air from the engine compressors supply air through bleed air flow control units and pneumatic shutoff valves to a pneumatic control assembly that inflates the wing boots. After an inflation period of 6 seconds, an electronic timer switches the distributor in the control assembly to deflate the wing

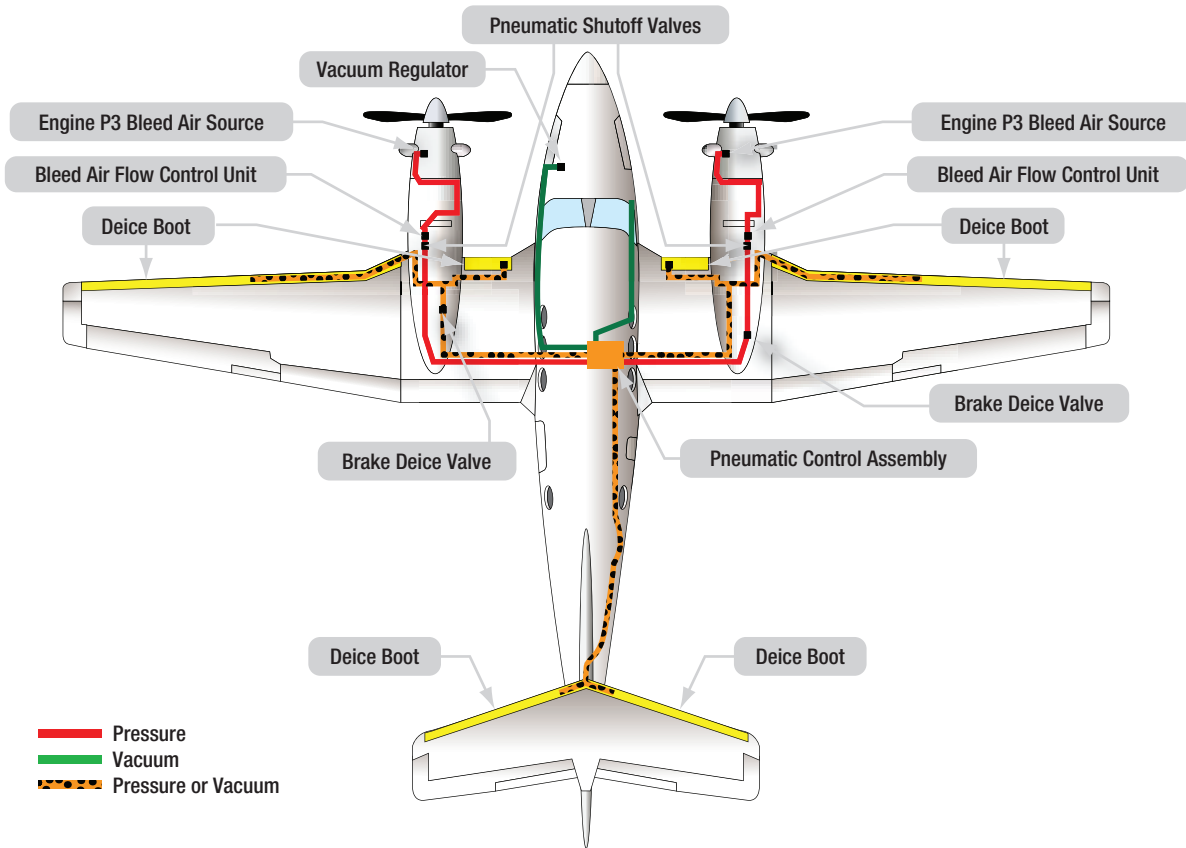


Figure 12-31. Wing deice system for turboprop aircraft.



Figure 12-32. Ice protection panel on a turboprop aircraft with deice boots.

boots, and a 4 second inflation begins in the horizontal stabilizer boots. After these boots have been inflated and deflated, the cycle is complete, and all boots are again held down tightly against the wings and horizontal stabilizer by vacuum.

The spring-loaded switch must be selected up again for another cycle to occur. Each engine supplies a common bleed air manifold. To ensure the operation of

the system, if one engine is inoperative, a flow control unit with check valve is incorporated in the bleed air line from each engine to prevent the loss of pressure through the compressor of the inoperative engine. If the boots fail to function sequentially, they may be operated manually by selecting the DOWN position of the same de-ice cycle switch.

Depressing and holding it in the manual DOWN position inflates all the boots simultaneously. When the switch is released, it returns to the (spring-loaded) off position, and each boot is deflated and held by vacuum. When operated manually, the boot should not be left inflated for more than 7 to 10 seconds, as a new layer of ice may begin to form on the expanded boots and become unremovable. If one engine is inoperative, the loss of its pneumatic pressure does not affect boot operation. Electric power to the boot system is required to inflate the boots in either single-cycle or manual operation. When electric power is lost, the vacuum holds the boots tightly against the leading edge.

INSPECTION, MAINTENANCE, AND TROUBLESHOOTING OF RUBBER DE-ICER BOOT SYSTEMS

Maintenance on pneumatic de-icing systems varies with each aircraft model. The instructions of the airframe or system components manufacturer should be followed in all cases. Depending on the aircraft, maintenance usually consists of operational checks, adjustments, troubleshooting, and inspection.

Operational Checks

An operational check of the system can be made by operating the aircraft engines or by using an external source of air. Most systems are designed with a test plug to permit ground checking the system without operating the engines. When using an external air source, make certain that the air pressure does not exceed the test pressure established for the system. Before turning the de-icing system on, observe the vacuum operated instruments.

If any of the gauges begin to operate, it is an indication that one or more check valves have failed to close and that reverse flow through the instruments is occurring. Correct the difficulty before continuing the test. If no movement of the instrument pointers occurs, turn on the de-icing system.

With the de-icer system controls in their proper positions, check the suction and pressure gauges for proper indications. The pressure gauge fluctuates as the de-icer tubes inflate and deflate. A relatively steady reading should be maintained on the vacuum gauge. It should be noted that not all systems use a vacuum gauge. If the operating pressure and vacuum are satisfactory, observe the de-icers for actuation. With an observer

stationed outside the aircraft, check the inflation sequence to be certain that it agrees with the sequence indicated in the aircraft maintenance manual.

Check the timing of the system through several complete cycles. If the cycle time varies more than is allowable, determine the difficulty and correct it. Inflation of the de-icers must be rapid to provide efficient de-icing. Deflation of the boot being observed should be completed before the next inflation cycle. (*Figure 12-33*)

Adjustments

Examples of adjustments that may be required include adjusting the de-icing system control cable linkages, adjusting system pressure relief valves, and de-icing system vacuum(suction) relief valves. A pressure relief valve acts as a safety device to relieve excess pressure in the event of regulator valve failure.

To adjust this valve, operate the aircraft engines and adjust a screw on the valve until the de-icing pressure gauge indicates the specified pressure at which the valve should relieve. Vacuum relief valves are installed in a system that uses a vacuum pump to maintain constant suction during varying vacuum pump speeds. To adjust a vacuum relief valve, operate the engines. While watching the vacuum (suction) gauge, an assistant should adjust the suction relief valve adjusting screw to obtain the correct suction specified for the system.

Troubleshooting

Not all troubles that occur in a de-icer system can be corrected by adjusting system components. Some troubles must be corrected by repair or replacement of system components or by tightening loose connections.

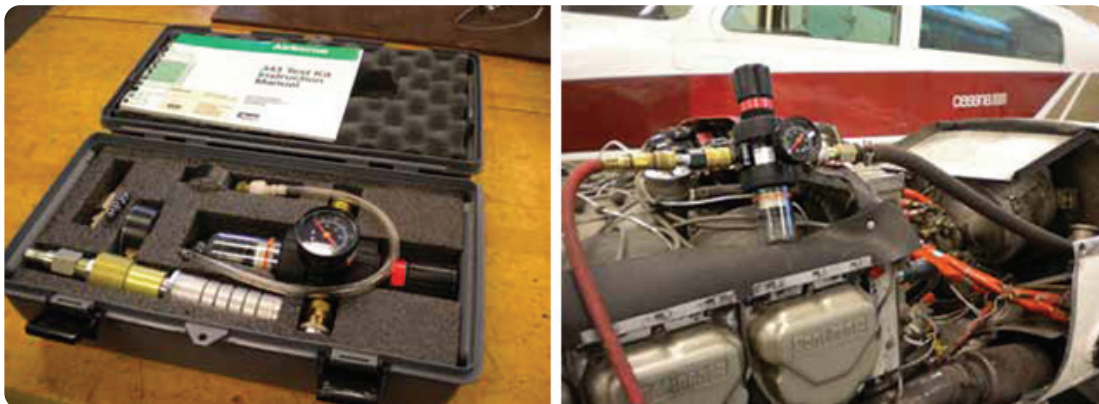


Figure 12-33. Test equipment used to test a wing deice system (left), and test equipment installed in the aircraft for testing (right).

Problem	Causes (most of which can be identified with a 343 Test Kit)	Corrective Action(s)
Boots Do Not Inflate	<ul style="list-style-type: none"> Open circuit breaker Faulty deflate valve Solenoid inoperable: <ol style="list-style-type: none"> Improper voltage at solenoid Blocked air vent in solenoid Inoperative plunger Diaphragm not seated <ol style="list-style-type: none"> Blocked vent orifice located in rivet bottom at center of diaphragm Dirty diaphragm seal area Diaphragm ruptured Two faulty deice control valves of faulty two-stage regulators Faulty check valve Relay not functioning Leak in system boots 	<ul style="list-style-type: none"> Reset circuit breaker Check deflate valves as follows: <ul style="list-style-type: none"> Solenoid inoperable: <ol style="list-style-type: none"> Correct electrical system Clean with alcohol or replace Clean with alcohol or replace Diaphragm not seated <ol style="list-style-type: none"> Clean with .010 diameter wire and alcohol Clean with blunt instrument and alcohol Replace valve Clean or replace valve assembly as noted above Replace check valve Check wiring or replace relay Repair as needed
Slow Boot Inflation	<ul style="list-style-type: none"> Lines blocked or disconnected Low air pump capacity One or more deice control valves not functioning properly Deflate valve not fully closed Ball check in deflate valve inoperative Leaks in system or boots 	<ul style="list-style-type: none"> Check and replace lines Replace air pump Clean or replace valve assembly as noted above Clean or replace valve assembly as noted above Clean check valve or replace deflate valve Repair as needed
System Will Not Cycle	<ul style="list-style-type: none"> Pressure in system not reaching specified psi to activate pressure switch Leak in system or boots Pressure switch on deflate valve inoperative 	<ul style="list-style-type: none"> Clean or replace deice control valve as noted above Clean or replace deflate valve, as noted above Repair as needed, tighten all hose connections Replace switch
Slow Deflation	<ul style="list-style-type: none"> Low vacuum Faulty deflate valve (indicated by temporary reduction in suction gauge reading) 	<ul style="list-style-type: none"> Repair as needed Clean or replace valve assembly as noted above
No Vacuum For Boot Hold Down	<ul style="list-style-type: none"> Malfunctioning deflate valve or deice valve Leak in system or boots 	<ul style="list-style-type: none"> Clean or replace valve assembly as noted above Repair as needed
Boots Will Not Deflate During Cycle	<ul style="list-style-type: none"> Faulty deflate valve 	<ul style="list-style-type: none"> Check and replace valve
Boots Appear To Inflate On Aircraft Climb	<ul style="list-style-type: none"> Vacuum source for boot holddown inoperative Lines running through pressurized cabin loose or disconnected 	<ul style="list-style-type: none"> Check operation of ball check in deflate valve Check for loose or disconnected vacuum lines and repair

Figure 12-34. Troubleshooting guide for wing deice system.

Several troubles common to pneumatic de-icing systems are shown in the left-hand column of the chart in *Figure 12-34*.

NOTE: The probable causes and the remedy of each trouble listed in the chart. In addition to using troubleshooting charts, operational checks are sometimes necessary to determine the possible cause of trouble.

Inspection

During each preflight and scheduled inspection, check the de-icer boots for cuts, tears, deterioration, punctures, and security; during periodic inspections, go a little further and check de-icer components and lines for cracks. If weather cracking of rubber is noted, apply a coating of conductive cement. The cement, in addition to sealing the boots against weather, dissipates static

electricity so that it does not puncture the boots by arcing to the metal surfaces.

De-Ice Boot Maintenance

The life of the de-icers can be greatly extended by storing them when they are not needed and by observing these rules when they are in service:

1. Do not drag gasoline hoses over the de-icers.
2. Keep de-icers free of gasoline, oil, grease, dirt, and other deteriorating substances.
3. Do not lay tools on or lean maintenance equipment against the de-icers.
4. Promptly repair or resurface the de-icers when abrasion or deterioration is noted.
5. Wrap de-ice boots in paper or canvas when storing.

Thus far, preventive maintenance has been discussed. The actual work on the de-icers consists of cleaning, resurfacing, and repairing. Cleaning should ordinarily be done at the same time the aircraft is washed, using a mild soap and water solution. Grease and oil can be removed with a cleaning agent, such as naphtha, followed by soap and water scrubbing. Whenever the degree of wear is such that it indicates that the electrical conductivity of the de-icer surface has been destroyed, it may be necessary to resurface the de-icer. The resurfacing substance is a black, conductive neoprene cement. Prior to applying the resurfacing material, the de-icer must be cleaned thoroughly and the surface roughened.

The de-icer must be relieved of its installed tension before applying the patch. The area to be patched must be clean and buffed to roughen the surface slightly. Patches are glued in place. Follow manufacturer's instructions for all repairs.

ELECTRICAL DE-ICING SYSTEMS

De-Icing is accomplished with electricity on propellers and, occasionally on airfoils with electric de-ice boots.

ELECTRIC PROPELLER DE-ICE

The formation of ice on the propeller leading edges, cuffs, and spinner reduces the efficiency of the powerplant system. De-ice systems using electrical heating elements are common.

Many propellers are de-iced by an electrically heated boot on each blade. The boot, firmly cemented in place, receives current from a slip ring and brush assembly on the spinner bulkhead. The slip ring transmits current to the de-ice boot. The centrifugal force of the spinning propeller and air blast breaks the ice particles loose from the heated blades. (*Figure 12-35*)

On one aircraft model, the boots are heated in a preset sequence, which is an automatic function controlled by a timer. This sequence is as follows: 30 seconds for the right prop outer elements; 30 seconds for the right prop inner elements; 30 seconds for the left prop outer elements; and, 30 seconds for the left prop inner elements. Once the system is turned on for automatic is activated, it cycles continuously. A manual bypass of the timer is incorporated. (*Figure 12-36*)

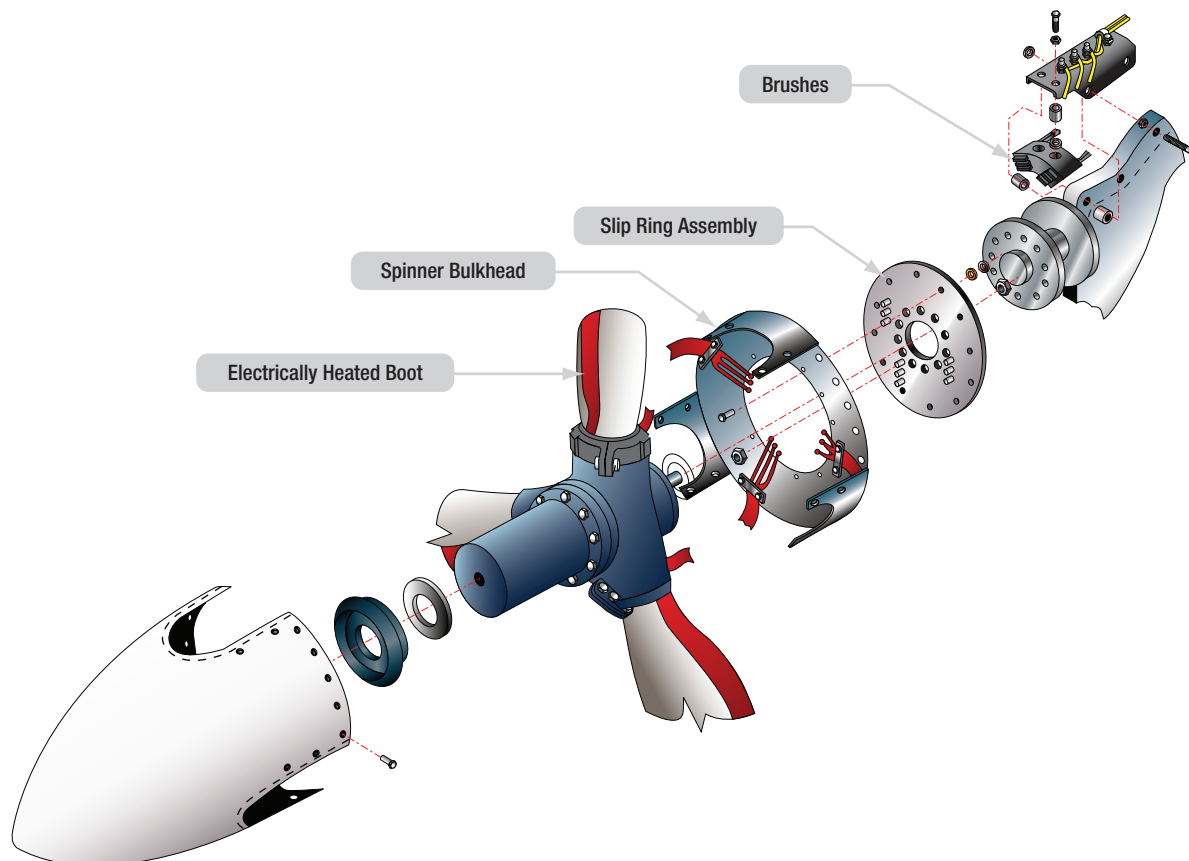


Figure 12-35. Electro thermal propeller de-ice system components.

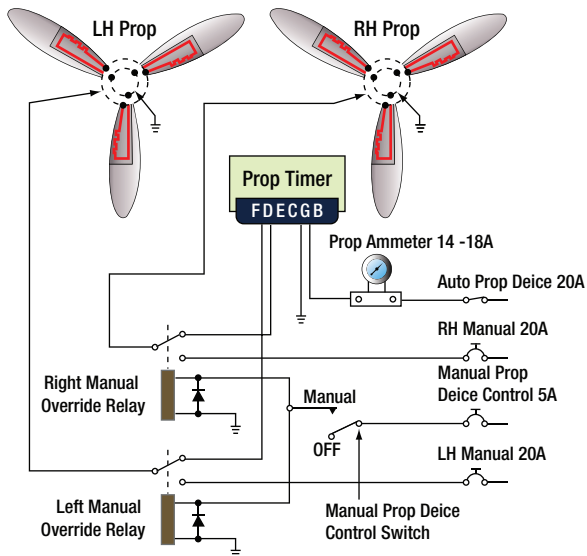


Figure 12-36. Propeller electrical deice system schematic.

ELECTRIC AIRFOIL DE-ICE BOOTS

A few modern aircraft are equipped with electric de-ice boots on wing sections or on the horizontal stabilizer. These boots contain electric heating elements which are bonded to the leading edges similarly to pneumatic de-ice boots. When activated, the boots heat up and melt the ice off of leading edge surfaces. The

elements are controlled by a sequence timer in a de-ice controller. Ice detector and ram air temperature probe inputs initiate operation when other flight condition parameters exist. The boot elements turn ON and OFF in paired sections to avoid aerodynamic imbalance. The system is inoperative while the aircraft is on the ground. (Figure 12-37)

HOT AIR DE-ICE/DE-FOG

Some laminated windshields on older aircraft have a space between the plies that allows the flow of hot air to be directed between the glass to keep it warm and fog free. The source of air is bleed air or conditioned air from the environmental control system. Small aircraft may utilize ducted warm air, which is release to flow over the windshield inner surface to defrost and defog. These systems are similar to those used in automobiles. The source of air could be ambient (defog only), the aircraft's heating system, or a combustion heater. While these pneumatic windshield heat systems are effective for de-icing/defogging the aircraft on which they are installed, they are not approved for flying into known icing conditions and, as such, are not effective for anti-ice.

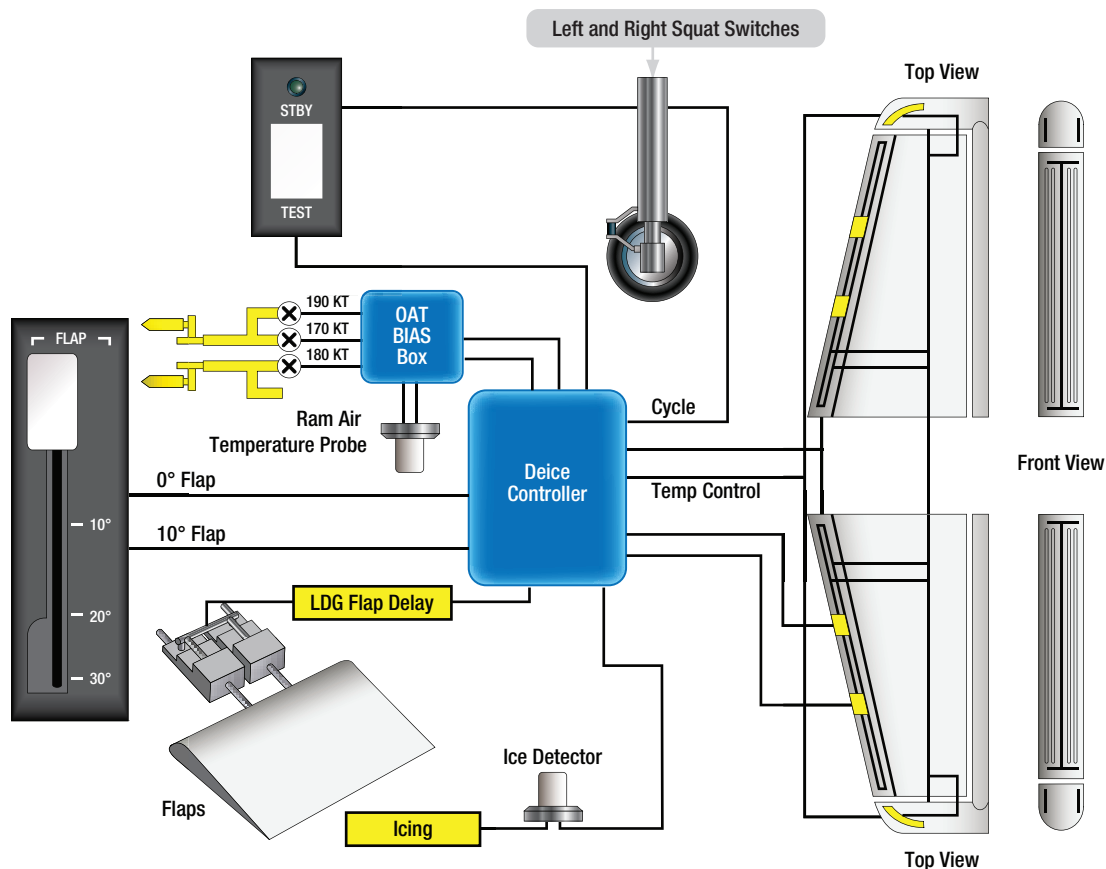


Figure 12-37. Electric stabilizer deice system.

NOTE: Large aircraft equipped with pneumatic jet blast rain repellent systems achieve some hot air anti-icing effects from operating this system although electric windshield heat.

CHEMICAL DE-ICE SYSTEMS

WINDSHIELD CHEMICAL DE-ICE

As previously mentioned in this chapter, chemical anti-ice systems exist generally for small aircraft. This type of anti-ice is also used on windshields.

The liquid chemical is sprayed through a nozzle onto the outside of the windshield which prevents ice from forming. The chemical can also de-ice the windshield of ice that may have already formed. Systems such as these have a fluid reservoir, pump, control valve, filter, and relief valve. Other components may exist. **Figure 12-38** shows a set of spray tubes for application of chemical anti-ice on an aircraft windshield.

CHEMICAL PROPELLER DE-ICE

Some aircraft models, especially single-engine GA aircraft, use a chemical de-icing system for the propellers. Ice usually appears on the propeller before it forms on the wing. The glycol-based fluid is metered from a tank by a small electrically driven pump through a micro filter to the slinger rings on the prop hub. The propeller system can be a stand-alone system, or it can be part of a chemical wing and stabilizer de-icing system such as the TKS™ weeping system.

CHEMICAL GROUND DE-ICING OF AIRCRAFT

The presence of ice on an aircraft may be the result of direct precipitation, formation of frost on integral fuel tanks after prolonged flight at high altitude, or accumulations on the landing gear following taxiing through snow or slush. The aircraft must be free of all frozen contaminants adhering to the wings, control surfaces, propellers, engine inlets, or other critical surfaces before takeoff.

Any deposits of ice, snow, or frost on the external surfaces of an aircraft may drastically affect its performance. This may be due to reduced aerodynamic lift and increased aerodynamic drag resulting from the disturbed airflow over the airfoil surfaces, or it may be due to the weight of the deposit over the whole aircraft. The operation of

an aircraft may also be seriously affected by the freezing of moisture in controls, hinges, valves, microswitches, or by the ingestion of ice into the engine. When aircraft are hangared to melt snow or frost, any melted snow or ice may freeze again if the aircraft is subsequently moved into subzero temperatures. Any measures taken to remove frozen deposits while the aircraft is on the ground must also prevent the possible refreezing of the liquid.

Frost Removal

Frost deposits can be removed by placing the aircraft in a warm hangar or by using a frost remover or de-icing fluid. These fluids normally contain ethylene glycol and isopropyl alcohol and can be applied either by spray or by hand. It should be applied within 2 hours of flight. De-Icing fluids may adversely affect windows or the exterior finish of the aircraft, only the type of fluid recommended by the aircraft manufacturer should be used. Transport category aircraft are often de-iced on the ramp or a dedicated de-icing location on the airport. De-Icing trucks are used to spray the de-icing and/or anti-icing fluid on aircraft surfaces. (**Figure 12-39**)

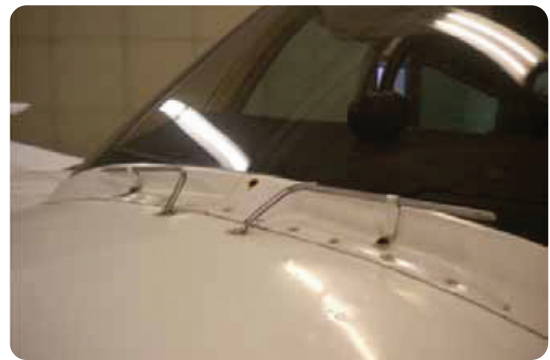


Figure 12-38. Chemical deicing spray tubes.



Figure 12-39. Boom truck de-icing of an airliner.

De-Icing Fluid

As mentioned previously, various de-icing fluids exist for ground de-icing and anti-icing of aircraft. A de-icing fluid must be accepted according to its type for holdover times, aerodynamic performance, and material compatibility. The coloring of these fluids is standardized. In general, straight glycol is colorless, Type-I fluids are orange, Type-II fluids are white/pale yellow, and Type-IV fluids are green.

The color for Type-III fluid has not yet been determined. It is a fluid compound manufactured for use on slow aircraft that rotate at less than 100 knots. However, Type III de-ice fluid is approved for use on Boeing 757 and 767 aircraft.

As explained in the anti-ice section of this sub-module, Type I fluid is an ethylene or propylene glycol based de-ice fluid that contains wetting agents and corrosion inhibitors. This is a low viscosity fluid that leaves a thin film on aircraft surfaces. The film quickly flows off the de-iced surface resulting in minimal anti-icing properties. It is common for Type I fluid to be mixed with hot water in different ratios depending on weather/icing conditions. Type II and IV de-icing fluids are thixotropic. They share the property of high viscosity until acted upon by an outside force. So once applied, a thick layer of Type II and Type IV fluid results on the aircraft surface. This gives high anti-icing protection even when frozen precipitation is falling when de-icing is done.

If both de-icing and anti-icing are required, the procedure may be performed in one or two steps. The selection of a one or two step process depends upon weather conditions, available equipment, available fluids, and the holdover time to be achieved. A one step approach when anti-icing is desirable or required must be done with Type II or Type IV de-ice fluid since Type I fluid does not provide significant anti-icing. A two step approach allows options. For example, The first step, de-icing, could be done with Type I fluid. With a high concentration of water to fluid, Type I fluid used to de-ice is less expensive than other types of fluid. Then, a second application can be made using Type II or Type IV fluid. Since the aircraft is already de-iced, a smaller quantity of these more expensive fluids can be used to give the thixotropic anti-ice properties needed.

Holdover Time (HOT)

Holdover Time (HOT) is the estimated time that de-icing/anti-icing fluid prevents the formation of frost or ice and the accumulation of snow on the critical surfaces of an aircraft. HOT begins when the final application of de-icing and anti-icing fluid commences and expires when the de-icing and anti-icing fluid loses its effectiveness. **Figure 12-40** shows a holdover timetable for Type IV fluid. HOT guidelines for other fluids are available and must be used when comparing options for different fluid use and options considering weather and traffic conditions.

Critical Surfaces

Basically, all surfaces that have an aerodynamic, control, sensing, movement, or measuring function must be clean. These surfaces cannot necessarily be cleaned and protected in the same conventional de-icing/anti-icing manner as the wings. Some areas require only a cleaning operation, while others need protection against freezing. The procedure of de-icing may also vary according to aircraft limitations. The use of hot air may be required when de-icing (e.g., landing gear or propellers). **Figure 12-41** shows critical areas on an aircraft that should not be sprayed directly with de-ice fluid. Some critical elements and procedures that are common for most aircraft are:

1. De-Icing/anti-icing fluids must not be sprayed directly on wiring harnesses and electrical components (e.g., receptacles, junction boxes), onto brakes, wheels, exhausts, or thrust reversers.
2. De-Icing/anti-icing fluid shall not be directed into the orifices of pitot heads, static ports, or directly onto airstream direction detectors probes/angle of attack airflow sensors.
3. All reasonable precautions shall be taken to minimize fluid entry into engines, other intakes/outlets, and control surface cavities.
4. Fluids shall not be directed onto flight deck or cabin windows as this can cause crazing of acrylics or penetration of the window seals.
5. Any forward area from which fluid can blow back onto windscreens during taxi or subsequent takeoff shall be free of residues prior to departure.
6. If Type II, III, or IV fluids are used, all traces of the fluid on flight deck windows should be removed prior to departure, particular attention being paid to windows fitted with wipers.

FAA Type IV Holdover Time Guideline

Guidelines for holdover times anticipated for SAE type IV fluid mixtures as function of weather conditions and OAT.

CAUTION: This table is for use in departure planning only, and it should be used in conjunction with pretakeoff check procedures.

OAT		SAE type IV fluid concentration neat fluid water (vol. %/vol.%)	Approximate holdover times under various weather conditions (hours:minutes)						
°C	°F		Frost*	Freezing Fog	Snow◇	Freezing drizzle***	Light free rain	Rain on cold soaked wing	Other*
Above 0	Above 32	100/ 0	18:00	1:05–2:15	0:35–1:05	0:40–1:10	0:25–0:40	0:10–0:50	CAUTION: No Holdover Time Guidelines Exist
		72/25	6:00	1:05–1:45	0:30–1:05	0:35–0:50	0:15–0:30	0:05–0:35	
		50/50	4:00	0:15–0:35	0:05–0:20	0:10–0:20	0:05–0:10	CAUTION: Clear Ice May Require Louch For Confirmation	
0 through –3	32 through 27	100/ 0	12:00	1:05–2:15	0:30–0:55	0:40–1:10	0:15–0:40		
		75/25	5:00	1:05–2:15	0:25–0:50	0:35–0:50	0:15–0:30		
		50/50	3:00	1:15–0:35	0:05–0:15	0:10–0:20	0:05–0:15		
below –3 through –14	below 27 through 7	100/ 0	12:00	0:20–0:50	0:20–0:40	**0:20–0:45	**0:10–0:25		
		75/25	5:00	0:25–0:50	0:15–0:25	**0:15–0:30	**0:10–0:20		
		below –14 through –25	below 7 through –13	100/ 0	12:00	0:15–0:40	0:15–0:30		
below –25	below –13	100/ 0	SAE type IV fluid may be used below –25 °C(–13 °F) if the freezing point of the fluid is at least 7 °C(13 °F) below the OAT and the aerodynamic acceptance criteria are met. Consider use of SAE type I when SAE type IV fluid cannot be used.						

°C = Degrees Celsius
°F = Degrees Fahrenheit
OAT= Outside Air Temperature
VOL = Volume

The responsibility for the application of these data remains with the user.

* During conditions that apply to aircraft protection for ACTIVE FROST

** No holdover time guidelines exist for this condition below –10 °C (14 °F)

*** Use light freezing rain holdover times if positive identification of freezing drizzle is not possible

‡ Snow pellets, ice pellets, heavy snow, moderate and heavy freezing rain, hail.

◊ Snow includes snow grains

CAUTIONS:

- The time of protection will be shortened in heavy weather conditions: heavy precipitation rates or high moisture contents.
- High wind velocity or jet blast may reduce holdover time below the lowest time stated in the range.
- Holdover time may be reduced when aircraft skin temperature is lower than OAT.

Figure 12-40. FAA deice holdover time guidelines.

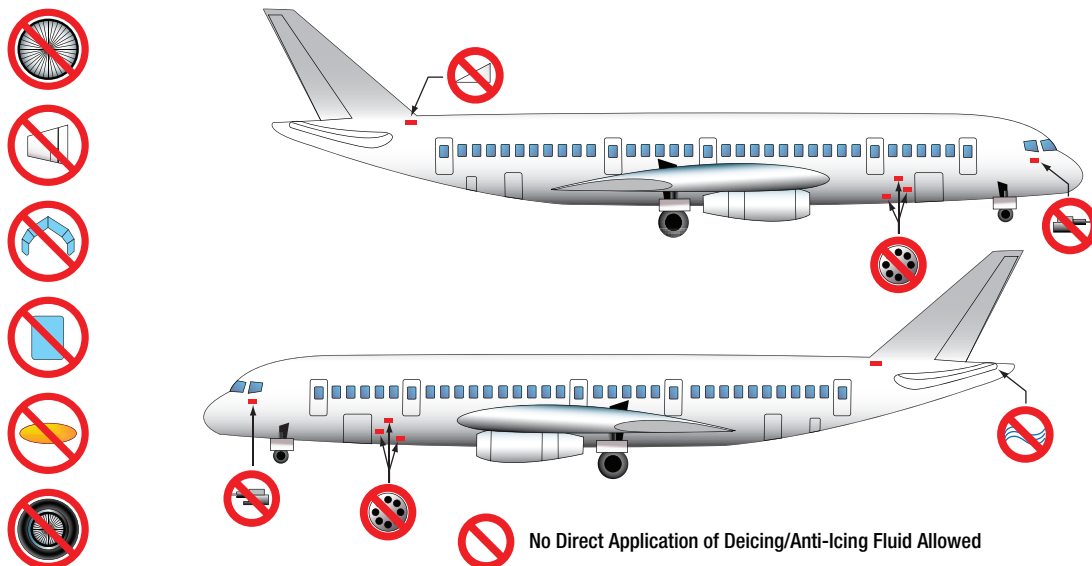


Figure 12-41. No direct application of deicing/anti-icing fluid allowed.

- Landing gear and wheel bays shall be kept free from buildup of slush, ice, or accumulations of blown snow.
- When removing ice, snow, slush, or frost from aircraft surfaces, care shall be taken to prevent it entering and accumulating in auxiliary intakes

or control surface hinge areas (e.g., manually remove snow from wings and stabilizer surfaces forward toward the leading edge and remove from ailerons and elevators back towards the trailing edge).

Ice and Snow Removal

Probably the most difficult deposit to deal with is deep, wet snow when ambient temperatures are slightly above the freezing point. This type of deposit should be removed with a soft brush or squeegee. Use care to avoid damage to antennas, vents, stall warning devices, vortex generators, etc., that may be concealed by the snow. Light, dry snow in subzero temperatures should be blown off whenever possible; the use of hot air is not recommended, since this would melt the snow, which would then freeze and require further treatment.

Moderate or heavy ice and residual snow deposits should be removed with a de-icing fluid. No attempt should be made to remove ice deposits or break an ice bond by force. After completion of de-icing operations, inspect the aircraft to ensure that its condition is satisfactory for flight. All external surfaces should be examined for signs of residual snow or ice, particularly in the vicinity of control gaps and hinges. Check the drain and pressure sensing ports for obstructions.

When it becomes necessary to physically remove a layer of snow, all protrusions and vents should be examined for signs of damage. Control surfaces should be moved to ascertain that they have full and free movement. The landing gear mechanism, doors and bay, and wheel brakes should be inspected for snow or ice deposits and the operation of uplocks and microswitches checked. Snow or ice can enter turbine engine intakes and freeze in the compressor. If the compressor cannot be turned by hand for this reason, hot air should be blown through the engine until the rotating parts are free.

RAIN CONTROL SYSTEMS

There are several different ways to remove the rain from the windshields. Most aircraft use one or a combination of the following systems: windshield wipers, chemical rain repellent, pneumatic rain removal (jet blast), or windshields treated with a hydrophobic surface seal coating.

WINDSHIELD WIPER SYSTEMS

In an electrical windshield wiper system, the blades are driven by an electric motor(s) that receive (s) power from the aircraft's electrical system. On some aircraft, the pilot's and copilot's windshield wipers are operated by separate systems to ensure that clear vision is maintained through one of the windows should one system fail. Each windshield wiper assembly consists of a wiper, wiper arm and a wiper motor/converter. Almost all windshield wiper systems use electrical motors. Some older aircraft might be equipped with hydraulic wiper motors. (*Figure 12-42*)

Maintenance performed on windshield wiper systems consists of operational checks, adjustments, and troubleshooting. An operational check should be performed whenever a system component is replaced or whenever the system is suspected of not working properly. During the check, make sure that the windshield area covered by the wipers is free of foreign matter and is kept wet with water. Adjustment of a windshield wiper system consists of adjusting the wiper blade tension, the angle at which the blade sweeps across the windshield, and proper parking of the wiper blades.

CHEMICAL RAIN REPELLENT

Water poured onto clean glass spreads out evenly. Even when the glass is held at a steep angle or subjected to air velocity, the glass remains wetted by a thin film of water. However, when glass is treated with certain chemicals, a transparent film is formed that causes the water to behave very much like mercury on glass. The water draws up into beads that cover only a portion of the glass and the area between beads is dry. The water is readily removed from the glass. This principle lends itself quite naturally to removing rain from aircraft windshields. The high-velocity slipstream continually removes the water beads, leaving a large part of the window dry.

A rain repellent system permits application of the chemical repellent by a switch or push button in the cockpit. The proper amount of repellent is applied regardless of how long the switch is held. On some systems, a solenoid valve controlled by a time delay module meters the repellent to a nozzle which sprays it on the outside of the windshield. Separate systems exist for the forward glass of the pilot and copilot. (*Figure 12-43*)

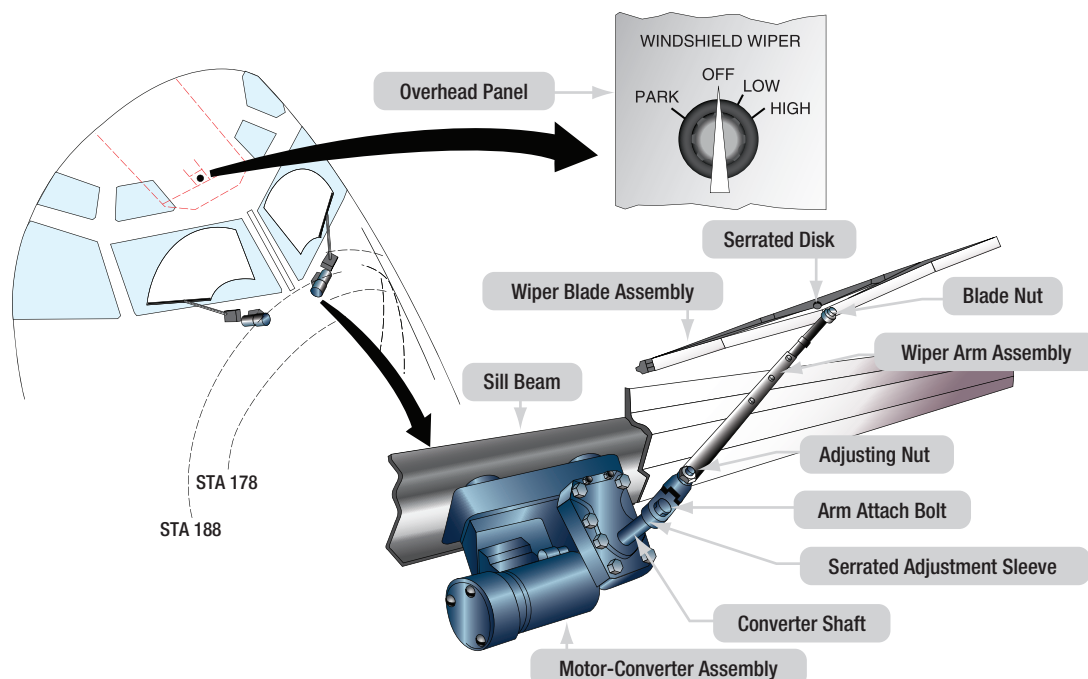


Figure 12-42. Windshield wiper assembly/installation on a transport category aircraft. The motor-converter is mounted under the aircraft skin.

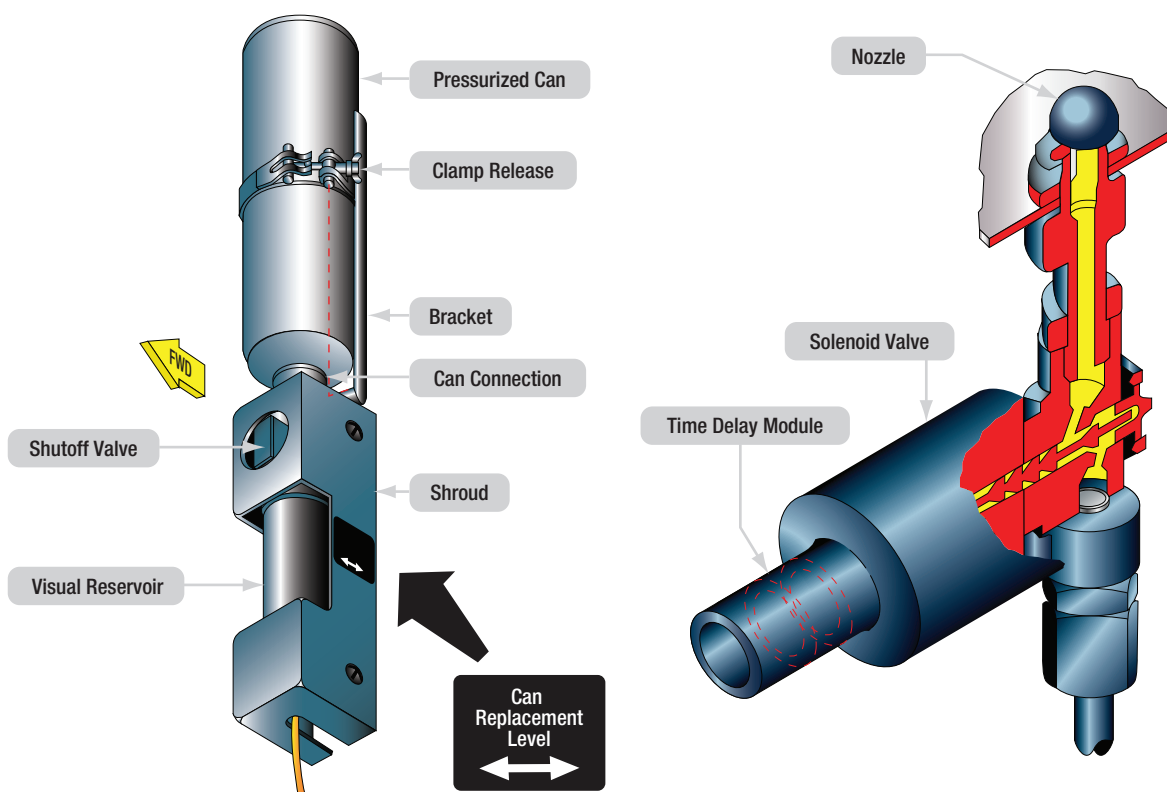


Figure 12-43. Cockpit rain repellent canister and reservoir.

This system should only be used in very wet conditions. The rain repellent system should not be operated on dry windows because heavy undiluted repellent restricts window visibility. Should the system be operated inadvertently, do not operate the windshield wipers or

rain clearing system as this tends to increase smearing. Also, the rain repellent residues caused by application in dry weather or very light rain can cause staining or minor corrosion of the aircraft skin. To prevent this, any concentrated repellent or residue should be removed by

a thorough fresh water rinse at the earliest opportunity. After application, the repellent film slowly deteriorates with continuing rain impingement. This makes periodic reapplication necessary. The length of time between applications depends upon rain intensity, the type of repellent used, and whether windshield wipers are used.

WINDSHIELD SURFACE SEAL COATING

Some aircraft models use a surface seal coating, also called hydrophobic coating that is on the outside of the pilot's/copilot's windshield. (*Figure 12-44*) The word hydrophobic means to repel or not absorb water. The windshield hydrophobic coating is on the external surface of the windows (windshields). The coatings cause raindrops to bead up and roll off, allowing the flight crew to see through the windshield with very little distortion. The hydrophobic windshield coating reduces the need for wipers and gives the flight crew better visibility during heavy rain. Most new aircraft windshields are treated with surface seal coating.

The manufacturer's coating process deeply penetrates the windshield surface providing hydrophobic action for quite some time. When effectiveness declines, products made to be applied in the field are used. These liquid treatments rubbed onto the surface of the windshield maintain the beading action of rain water. They must be applied periodically or as needed.

PNEUMATIC RAIN REMOVAL SYSTEMS

Windshield wipers characteristically have two basic problem areas. One is the tendency of the slipstream aerodynamic forces to reduce the wiper blade loading pressure on the window, causing ineffective wiping or streaking. The other is in achieving fast enough wiper oscillation to keep up with high rain impingement rates during heavy rain falls. As a result, most aircraft wiper systems fail to provide satisfactory vision in heavy rain.

The rain removal system shown in *Figure 12-45* controls windshield icing and removes rain by directing a flow of heated air over the windshield. This heated air serves two purposes. First, the air breaks the rain drops into small particles that are then blown away. Secondly, the air heats the windshield to prevent the moisture from freezing. The air can be supplied by an electric blower or by bleed air.

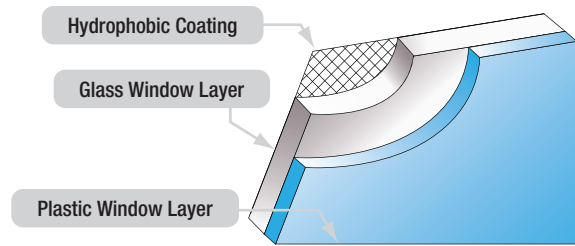


Figure 12-44. Hydrophobic coating on windshield.

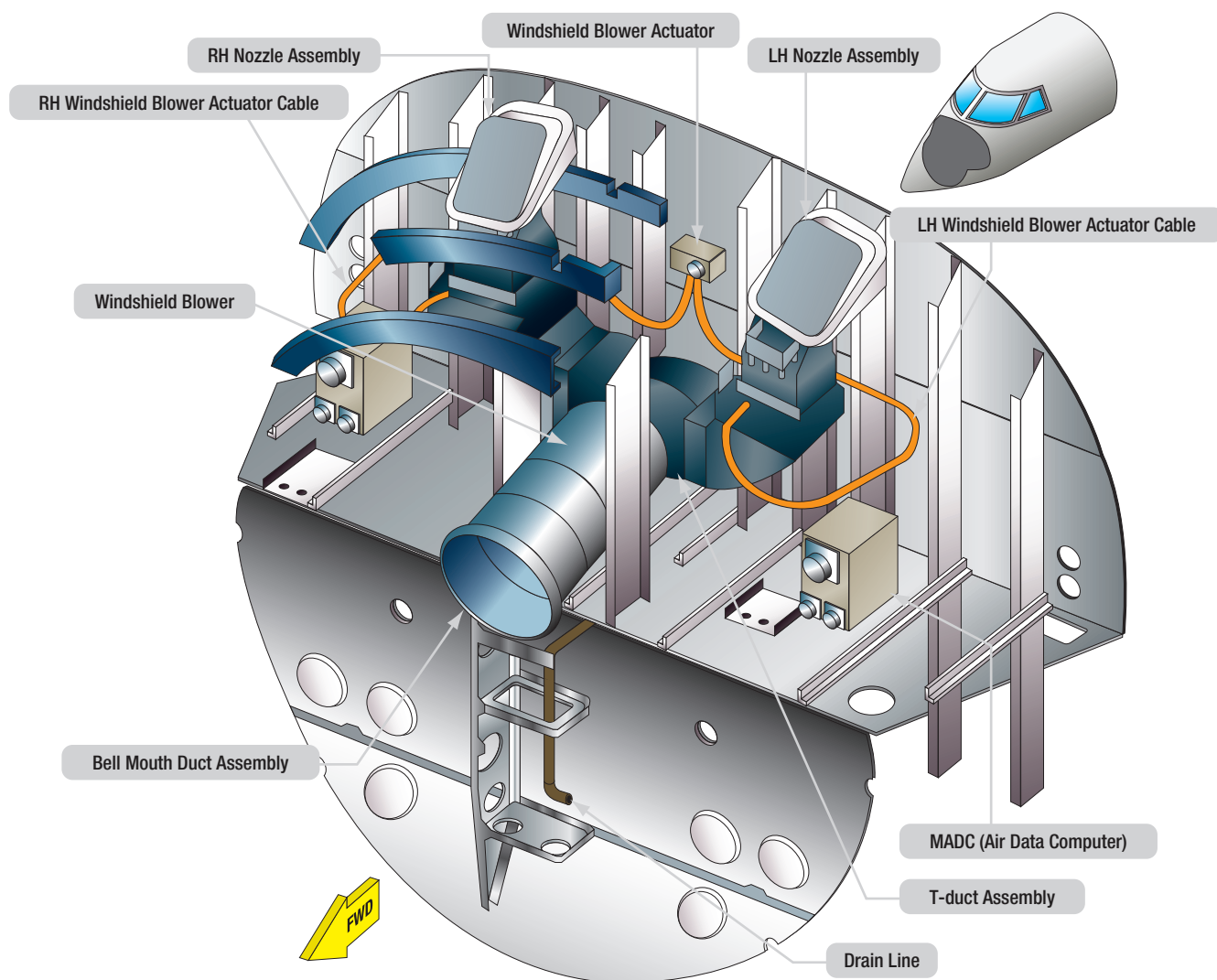


Figure 12-45. Windshield rain and frost removal system.

Question: 12-1

The two types of ice are encountered during flight are _____ ice and _____ ice.

Question: 12-6

_____ that protrude into the ambient airstream are particularly susceptible to ice formation in flight.

Question: 12-2

When ice collects on an ice detector probe, the established _____ of vibration decreases.

Question: 12-7

The purpose of _____ is to create the strength and impact resistance of the windshield assembly.

Question: 12-3

The most common source of heated air for wing anti-icing is turbine engine _____.

Question: 12-8

Wing and stabilizer chemical anti-ice systems are often called _____ systems.

Question: 12-4

The _____ controls the flow of bleed air from the pneumatic system to the WAI ducts.

Question: 12-9

On modern aircraft, the de-icer boots are bonded with _____ to the leading edge of wing and tail surfaces.

Question: 12-5

An EAI _____ regulates air that operates the EAI anti-ice valve.

Question: 12-10

The source of air for pneumatic de-ice boots on turboprop aircraft is _____.

ANSWERS

Answer: 12-1
clear.
rime.

Answer: 12-6
Data probes.

Answer: 12-2
frequency.

Answer: 12-7
laminations.

Answer: 12-3
compressor bleed air.

Answer: 12-8
weeping wing.

Answer: 12-4
WAI valve (wing anti-ice valve).

Answer: 12-9
an adhesive.

Answer: 12-5
controller.

Answer: 12-10
engine bleed air.

Question: 12-11

A _____ may be used to ensure de-ice boots are held tightly deflated against the aircraft structure to provide the significant change in size and shape needed to break off accumulated ice when the boots inflate.

Question: 12-12

An operational check of a de-ice boot system can be made by operating the engines or by using _____.

Question: 12-13

A propeller de-ice boot receives current from a _____ on the spinner bulkhead.

Question: 12-14

Deposits of ice, snow, or frost on the external surfaces of an aircraft may reduce aerodynamic lift and increase _____ resulting from the disturbed airflow over the airfoil surfaces.

Question: 12-15

Deep, wet snow when ambient temperatures are slightly above the freezing point should be removed from aircraft surfaces with _____.

Question: 12-16

Some aircraft models use a surface seal coating, also called _____ that is on the outside of the pilot's/ copilot's windshield.

ANSWERS

Answer: 12-11
deflate valve.

Answer: 12-14
aerodynamic drag.

Answer: 12-12
an external source of air.

Answer: 12-15
a soft brush or squeegee.

Answer: 12-13
slip ring and brush assembly.

Answer: 12-16
hydrophobic coating.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 13

LANDING GEAR (ATA 32)

Knowledge Requirements

11.13 - Landing Gear (ATA 32)

- Construction, shock absorbing;
- Extension and retraction systems: normal and emergency;
- Indications and warning;
- Wheels, brakes, anti-skid and auto-braking;
- Tires;
- Steering;
- Air-ground sensing.

3

LANDING GEAR
(ATA 32)

11.13 - LANDING GEAR

Aircraft landing gear supports the entire weight of an aircraft during landing and ground operations. They are attached to primary structural members of the aircraft. The type of gear depends on the aircraft design and its intended use. Most landing gear have wheels to facilitate operation to and from hard surfaces, such as airport runways. Other gear feature skids for this purpose, such as those found on helicopters, balloon gondolas, and in the tail area of some tail dragger aircraft.

Aircraft that operate to and from frozen lakes and snowy areas may be equipped with landing gear that have skis. Aircraft that operate to and from the surface of water have pontoon type landing gear. Regardless of the type of landing gear utilized, shock absorbing equipment, brakes, retraction mechanisms, controls, warning devices, cowling, fairings, and structural members necessary to attach the gear to the aircraft are considered parts of the landing gear system. (*Figure 13-1*)

Numerous configurations of landing gear types can be found. Additionally, combinations of two types of gear are common. Amphibious aircraft are designed with gear that allow landings to be made on water or dry land. The gear features pontoons for water landing with extendable wheels for landings on hard surfaces. A similar system is used to allow the use of skis and wheels on aircraft that operate on both slippery, frozen surfaces and dry runways. Typically, the skis are retractable to allow use of the wheels when needed. *Figure 13-2* illustrates this type of landing gear.

NOTE: References to auxiliary landing gear refer to the nose gear, tail gear, or outrigger type gear on any particular aircraft. Main landing gear is the two or more large gear located close to the aircraft's center of gravity.

LANDING GEAR CONFIGURATIONS

Three basic arrangements of landing gear are used: tail wheel type landing gear (conventional gear), tandem landing gear, and tricycle type landing gear.

TAIL WHEEL TYPE LANDING GEAR

Tail wheel type landing gear is also known as conventional gear because many early aircraft use this type of arrangement. The main gear are located forward of the center of gravity, causing the tail to require support from



Figure 13-1. Basic landing gear types include those with wheels (A), skids (B), skis (C), and floats or pontoons (D).



Figure 13-2. An amphibious aircraft with retractable wheels (left) and an aircraft with retractable skis (right).



Figure 13-3. Tail wheel configuration landing gear on a DC-3 (top) and a STOL Maule MX-7-235 Super Rocket.

a third wheel assembly. A few early aircraft designs use a skid rather than a tail wheel. This helps slow the aircraft upon landing and provides directional stability. The resulting angle of the aircraft fuselage, when fitted with conventional gear, allows the use of a long propeller that compensates for older, underpowered engine design. The increased clearance of the forward fuselage offered by tail wheel type landing gear is also advantageous when operating in and out of non-paved runways. Today, aircraft are manufactured with conventional gear for this reason and for the weight savings accompanying the relatively light tail wheel assembly. (*Figure 13-3*)

The proliferation of hard surface runways has rendered the tail skid obsolete in favor of the tail wheel. Directional control is maintained through differential braking until the speed of the aircraft enables control with the rudder. A steerable tail wheel, connected by cables to the rudder or rudder pedals, is also a common design. Springs are incorporated for dampening. (*Figure 13-4*)

TANDEM LANDING GEAR

Few aircraft are designed with tandem landing gear. As the name implies, this type of landing gear has the main



Figure 13-4. The steerable tail wheel of a Pitts Special.

gear and tail gear aligned on the longitudinal axis of the aircraft. Sailplanes commonly use tandem gear, although many only have one actual gear forward on the fuselage with a skid under the tail. A few military bombers, such as the B-47 and the B-52, have tandem gear, as does the U2 spy plane. The VTOL Harrier has tandem gear but uses small outrigger gear under the wings for support. Generally, placing the gear only under the fuselage facilitates the use of very flexible wings. (*Figure 13-5*)



Figure 13-5. Tandem landing gear along the longitudinal axis of the aircraft permits the use of flexible wings on sailplanes (left) and select military aircraft like the B-52 (center). The VTOL Harrier (right) has tandem gear with outrigger-type gear.

TRICYCLE TYPE LANDING GEAR

The most commonly used landing gear arrangement is the tricycle type landing gear. It is comprised of main gear and nose gear. (*Figure 13-6*) Tricycle type landing gear is used on large and small aircraft with the following benefits:

1. Allows more forceful application of the brakes without nosing over when braking, which enables higher landing speeds.
2. Provides better visibility from the flight deck, especially during landing and ground maneuvering.
3. Prevents ground looping of the aircraft. Since the aircraft center of gravity is forward of the main gear, forces acting on the center of gravity keep the aircraft moving forward rather than looping, such as with a tail wheel type landing gear.

The nose gear of a few aircraft with tricycle type landing gear is not controllable. It simply casters as steering is accomplished with differential braking during taxi. However, nearly all aircraft have steerable nose gear. On light aircraft, the nose gear is directed through mechanical linkage to the rudder pedals. Heavy aircraft typically utilize hydraulic power to steer the nose gear. Control is achieved through an independent tiller in the flight deck. (*Figure 13-7*)



Figure 13-6. Tricycle-type landing gear with dual main wheels on a Learjet (top) and a Cessna 172, also with tricycle gear (bottom).

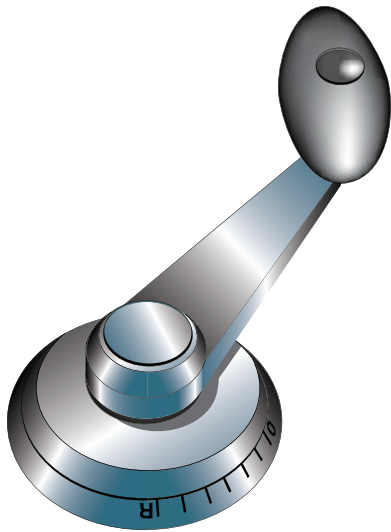


Figure 13-7. A nose wheel steering tiller located on the flight deck.

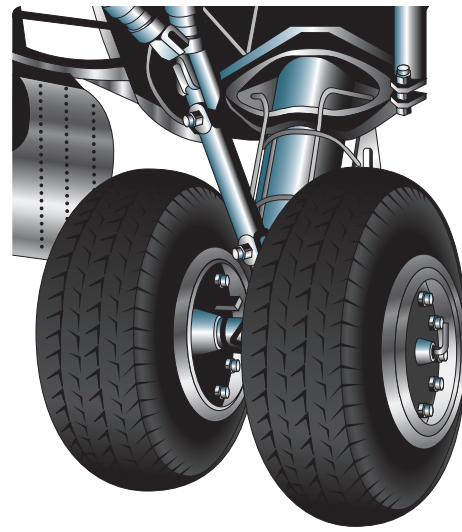


Figure 13-8. Dual main gear of a tricycle-type landing gear.

The main gear on a tricycle type landing gear arrangement is attached to reinforced wing structure or fuselage structure. The number and location of wheels on the main gear vary. Many main gear have two or more wheels. (*Figure 13-8*)

Multiple wheels spread the weight of the aircraft over a larger area. They also provide a safety margin should one tire fail. Heavy aircraft may use four or more wheel assemblies on each main gear. When more than two wheels are attached to a landing gear strut, the attaching mechanism is known as a bogie. The number of wheels included in the bogie is a function of the gross design weight of the aircraft and the surface type on which the loaded aircraft is required to land. *Figure 13-9* illustrates the triple bogie main gear of a Boeing 777.

The tricycle type landing gear arrangement consists of many parts and assemblies. These include air/oil shock struts, gear alignment units, support units, retraction and safety devices, steering systems, wheel and brake assemblies, etc. A main landing gear of a transport category aircraft is illustrated in *Figure 13-10* with many of the parts identified as an introduction to landing gear nomenclature.

FIXED AND RETRACTABLE LANDING GEAR

Further classification of aircraft landing gear can be made into two categories: fixed and retractable. Many small, single engine light aircraft have fixed landing gear, as do a few light twins. This means the gear is attached to the airframe and remains exposed to the



Figure 13-9. Triple bogie main landing gear assembly on a Boeing 777.

slipstream as the aircraft is flown. As the speed of an aircraft increases, so does parasite drag. Mechanisms to retract and stow the landing gear to eliminate parasite drag add weight to the aircraft. On slow aircraft, the penalty of this added weight is not overcome by the reduction of drag, so fixed gear is used. As the speed of the aircraft increases, the drag caused by the landing gear becomes greater and a means to retract the gear to eliminate parasite drag is required, despite the weight of the mechanism.

Retractable landing gear stow in fuselage or wing compartments while in flight. Once in these wheel wells, gear are out of the slipstream and do not cause parasite drag. Most retractable gear have a close fitting panel attached to them that fairs with the aircraft skin when the gear is fully retracted. (*Figure 13-11*) Other aircraft have separate doors that open, allowing the gear to enter or leave, and then close again.

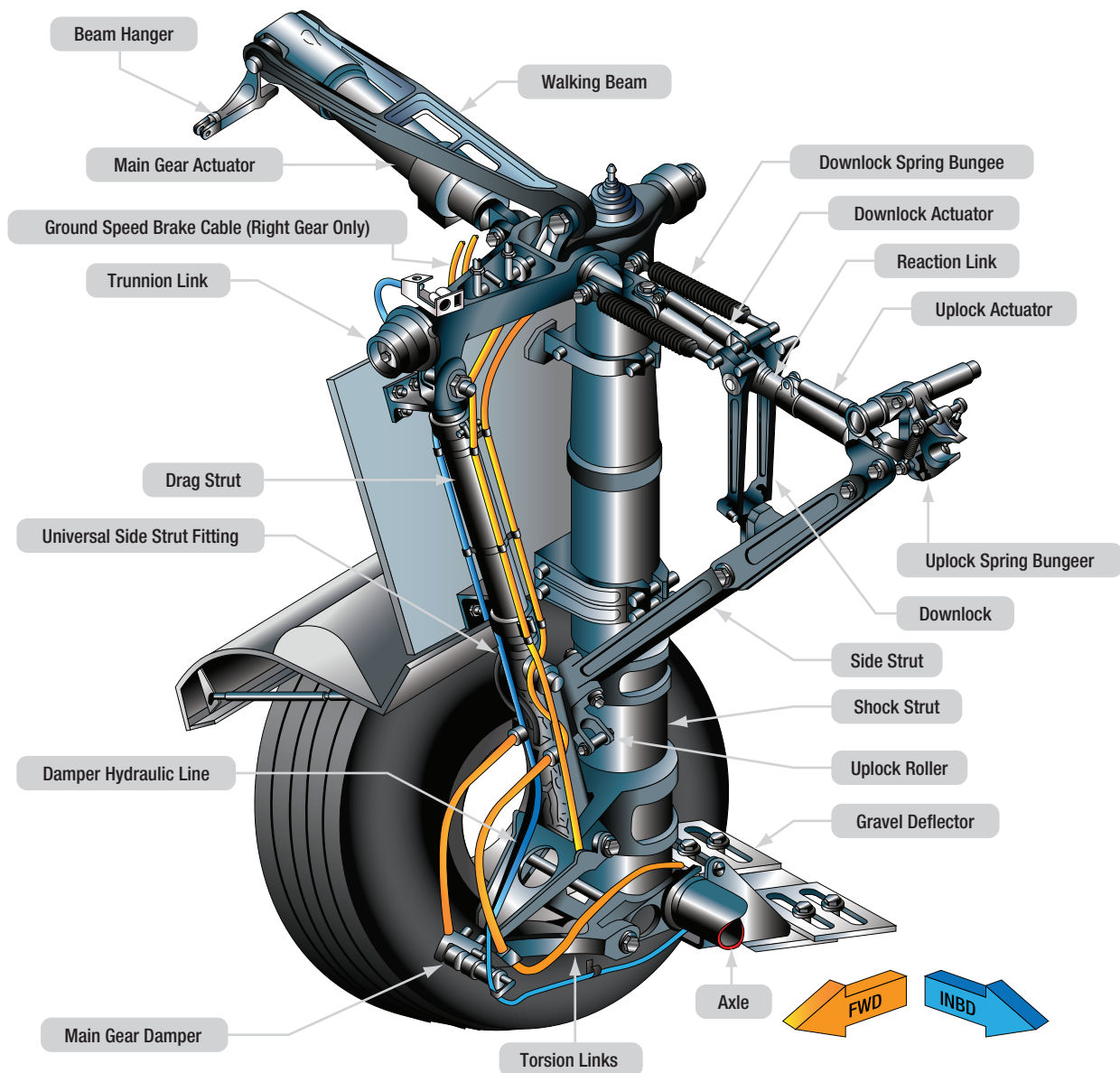


Figure 13-10. Nomenclature of a main landing gear bogie truck.



Figure 13-11. The retractable gear of a Boeing 737 fair into recesses in the fuselage. Panels attached to the landing gear provide smooth airflow over the struts. The wheel assemblies mate with seals to provide aerodynamic flow without doors.

NOTE: The parasite drag caused by extended landing gear can be used by the pilot to slow the aircraft. The extension and retraction of most landing gear is usually accomplished with hydraulic power. Landing gear retraction systems are discussed below.

LANDING GEAR ALIGNMENT AND SUPPORT

Retractable landing gear consist of several components that enable it to function. Typically, these are the torque links, trunnion and bracket arrangements, drag strut linkages, electrical and hydraulic gear retraction devices, as well as locking, sensing, and indicating components. Additionally, nose gear have steering mechanisms attached to the gear.

ALIGNMENT

As previously mentioned, a torque arm or torque links assembly keeps the lower strut cylinder from rotating out of alignment with the longitudinal axis of the aircraft. In some strut assemblies, it is the sole means of retaining the piston in the upper strut cylinder. The link ends are attached to the fixed upper cylinder and the moving lower cylinder with a hinge pin in the center to allow the strut to extend and compress.

Alignment of the wheels of an aircraft is also a consideration. Normally, this is set by the manufacturer and only requires occasional attention such as after a hard landing. On some smaller aircraft, the main wheels must be inspected and adjusted, if necessary, to maintain the proper tow-in or tow-out and the correct camber.

Tow-in and tow-out refer to the path a main wheel would take in relation to the airframe longitudinal axis or centerline if the wheel was free to roll forward.

Three possibilities exist. The wheel would roll either: 1) parallel to the longitudinal axis (aligned); 2) converge on the longitudinal axis (tow-in); or 3) veer away from the longitudinal axis (tow-out). (*Figure 13-12*)

The manufacturer's maintenance instructions give the procedure for checking and adjusting tow-in or tow-out. A general procedure for checking alignment on a light aircraft follows. To ensure that the landing gear settle properly for a tow-in/tow-out test, especially on spring steel strut aircraft, two aluminum plates separated with grease are put under each wheel. Gently rock the aircraft on the plates to cause the gear to find the at rest position preferred for alignment checks. A straight edge is held across the front of the main wheel tires just below axle height.

A carpenter's square placed against the straight edge creates a perpendicular that is parallel to the longitudinal axis of the aircraft. Slide the square against the wheel assembly to see if the forward and aft sections of the tire touch the square. A gap in front indicates the wheel is towed-in. A gap in the rear indicates the wheel is towed-out. (*Figure 13-13*)

Camber is the alignment of a main wheel in the vertical plain. It can be checked with a bubble protractor held against the wheel assembly. The wheel camber is said

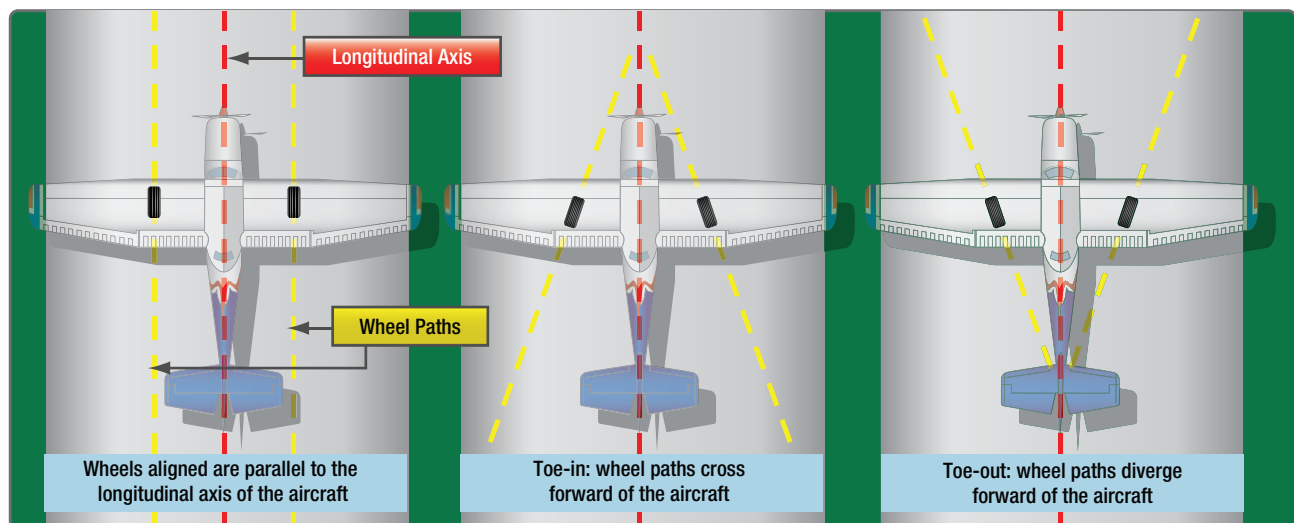


Figure 13-12. Wheel alignment on an aircraft.

to be positive if the top of the wheel tilts outward from vertical. Camber is negative if the top of the wheel tilts inward. (*Figure 13-14*)

Adjustments can be made to correct small amounts of wheel misalignment. On aircraft with spring steel gear, tapered shims can be added or removed between the bolt-on wheel axle and the axle mounting flange on the strut. Aircraft equipped with air/oil struts typically use shims between the two arms of the torque links as a means of aligning tow-in and tow-out. Follow all manufacturer's instructions. (*Figure 13-15*)

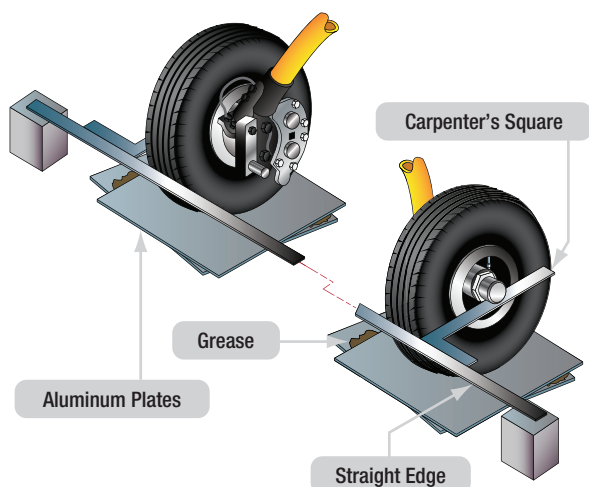


Figure 13-13. Finding toe-in and toe-out on a light aircraft with spring steel struts.

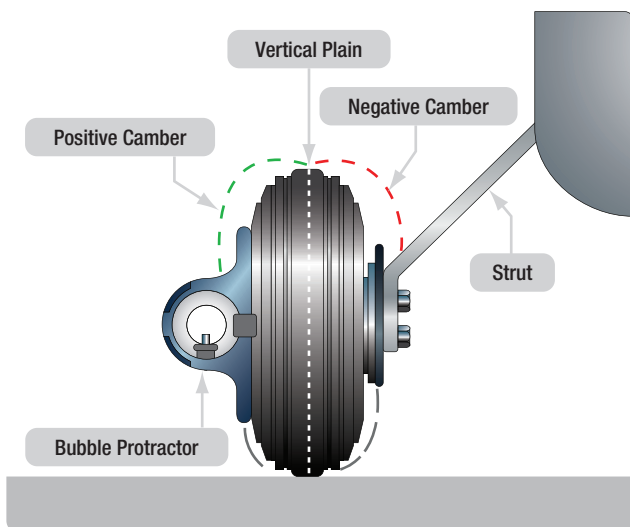


Figure 13-14. Camber of a wheel is the amount the wheel is tilted out of the vertical plain. It can be measured with a bubble protractor.

SUPPORT

Aircraft landing gear are attached to the wing spars or other structural members, many of which are designed for the specific purpose of supporting the landing gear. Retractable gear must be engineered in such a way as to provide strong attachment to the aircraft and still be able to move into a recess or well when stowed.

A trunnion arrangement is typical. The trunnion is a fixed structural extension of the upper strut cylinder with bearing surfaces that allow the entire gear assembly to move. It is attached to aircraft structure in such a way that the gear can pivot from the vertical position required for landing and taxi to the stowed position used during flight. (*Figure 13-16*)

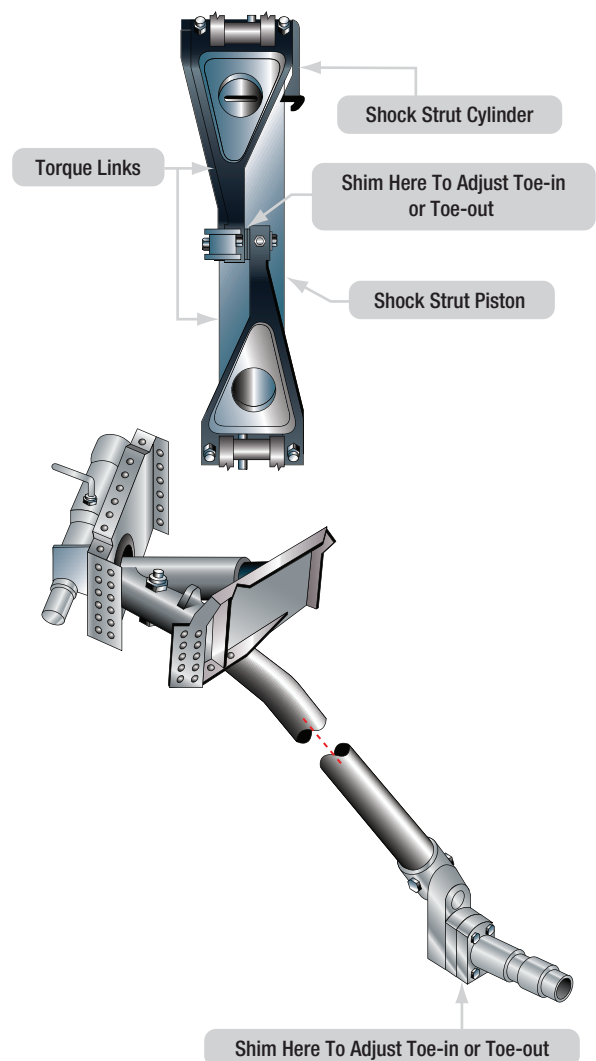


Figure 13-15. Toe-in and toe-out adjustments on small aircraft with spring steel landing gear are made with shims behind the axle assembly. On shock strut aircraft, shims are placed where the torque links couple.

While in the vertical gear down position, the trunnion is free to swing or pivot. Alone, it cannot support the aircraft without collapsing. A drag brace is used to restrain against the pivot action built into the trunnion attachment. The upper end of the two piece drag brace is attached to the aircraft structure and the lower end to the strut. A hinge near the middle of the brace allows the brace to fold and permits the gear to retract. For ground operation, the drag brace is straightened over center to a stop, and locked into position so the gear remains rigid. (*Figure 13-17*)

The function of a drag brace on some aircraft is performed by the hydraulic cylinder used to raise and lower the gear. Cylinder internal hydraulic locks replace the over center action of the drag brace for support during ground maneuvers.

SHOCK ABSORBING

In addition to supporting the aircraft for taxi, the forces of impact on an aircraft during landing must be controlled by the landing gear. This is done in two ways: 1) the shock energy is altered and transferred throughout the airframe at a different rate and time than the single strong pulse of impact, and 2) the shock is absorbed by converting the energy into heat energy. A variety of non-shock absorbing land gear are used on general aviation aircraft such as leaf type spring gear, gear with bungee cords and even rigid steel landing gear on early aircraft and aircraft with skids. Most turbine powered aircraft use shock strut landing gear.

SHOCK STRUTS

True shock absorption occurs when the shock energy of landing impact is converted into heat energy, as in a shock strut landing gear. This is the most common method of landing shock dissipation in aviation. It is used on aircraft of all sizes. Shock struts are self-contained hydraulic units that support an aircraft while on the ground and protect the structure during landing. They must be inspected and serviced regularly to ensure proper operation. There are many different designs of shock struts, but most operate in a similar manner. The following discussion is general in nature. For information on the construction, operation, and servicing of a specific aircraft shock, consult the manufacturer's maintenance instructions.

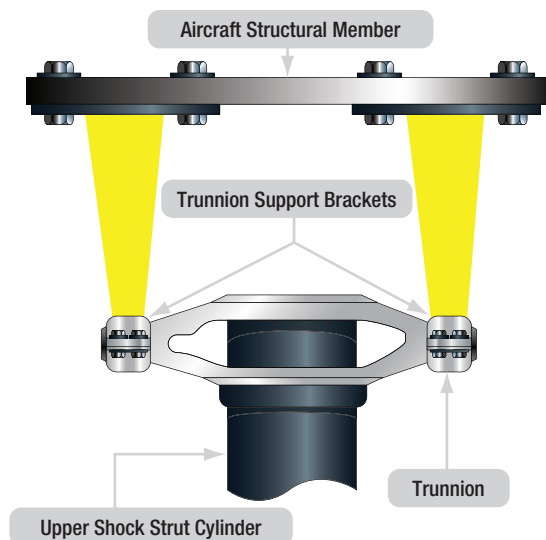


Figure 13-16. The trunnion is a fixed structural support that is part of or attached to the upper strut cylinder of a landing gear strut. It contains bearing surfaces so the gear can retract.

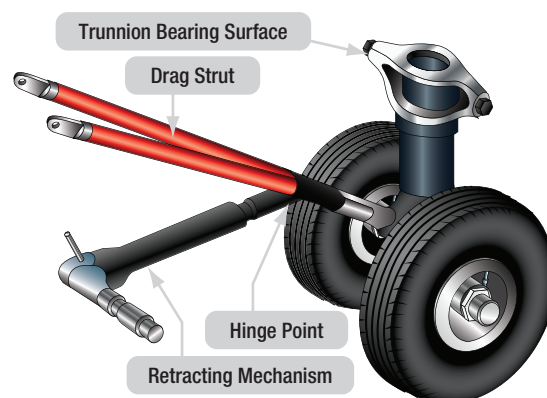


Figure 13-17. A hinged drag strut holds the trunnion and gear firm for landing and ground operation. It folds at the hinge to allow the gear to retract.

A typical pneumatic/hydraulic shock strut uses compressed air or nitrogen combined with hydraulic fluid to absorb and dissipate shock loads. It is sometimes referred to as an air/oil or oleo strut. A shock strut is constructed of two telescoping cylinders or tubes that are closed on the external ends. The upper cylinder is fixed to the aircraft and does not move. The lower cylinder is called the piston and is free to slide in and out of the upper cylinder. Two chambers are formed. The lower chamber is always filled with hydraulic fluid and the upper chamber is filled with compressed air or nitrogen. An orifice located between the two cylinders provides a

passage for the fluid from the bottom chamber to enter the top cylinder chamber when the strut is compressed. (*Figure 13-18*)

Most shock struts employ a metering pin similar to that shown in *Figure 13-18* for controlling the rate of fluid flow from the lower chamber into the upper chamber. During the compression stroke, the rate of fluid flow is not constant. It is automatically controlled by the taper of the metering pin in the orifice. When a narrow portion of the pin is in the orifice, more fluid can pass to the upper chamber. As the diameter of the portion of the

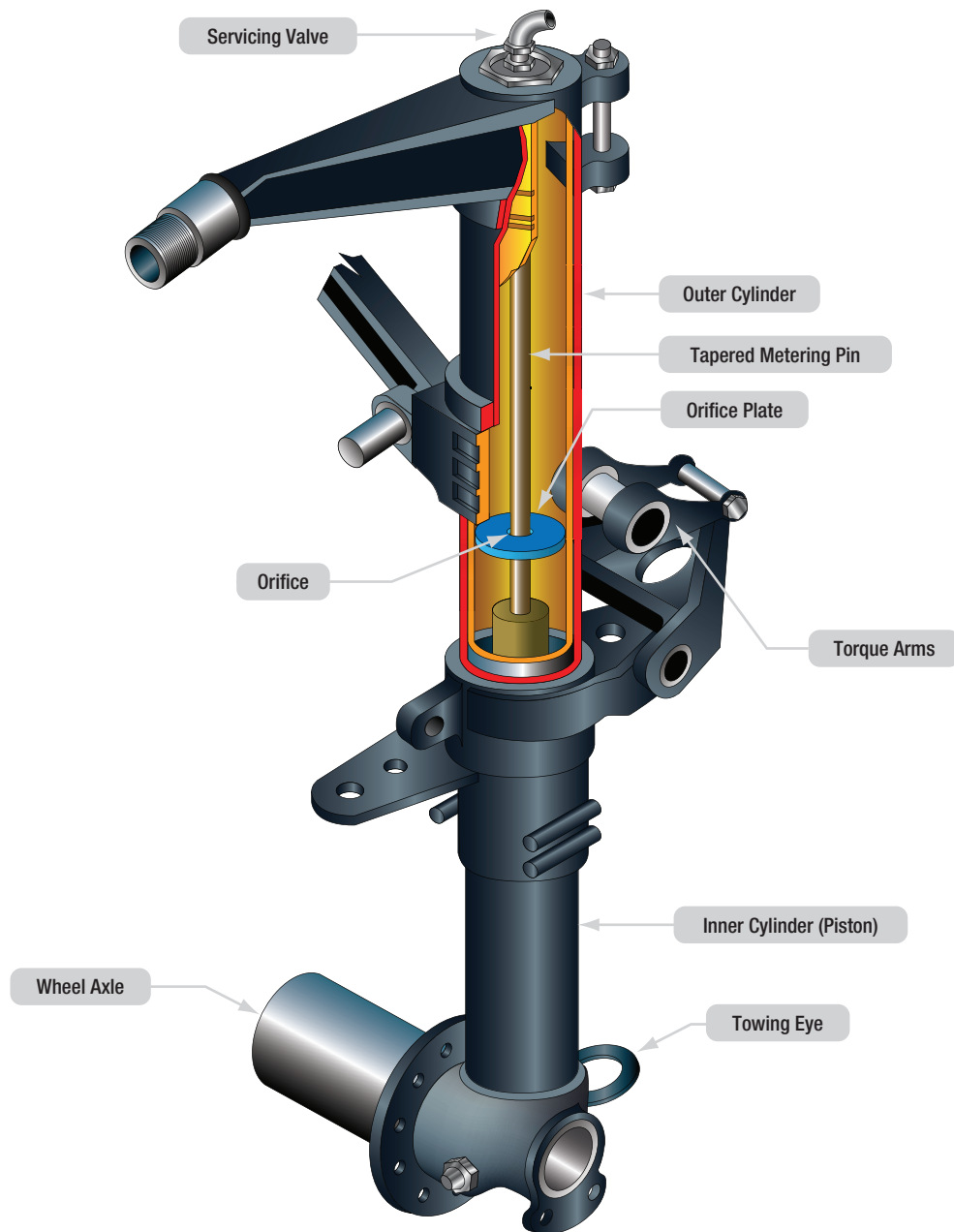


Figure 13-18. Some landing gear shock struts use an internal metering tube rather than a metering pin to control the flow of fluid from the bottom cylinder to the top cylinder.

metering pin in the orifice increases, less fluid passes. Pressure build-up caused by strut compression and the hydraulic fluid being forced through the metered orifice causes heat. This heat is converted impact energy. It is dissipated through the structure of the strut.

On some types of shock struts, a metering tube is used. The operational concept is the same as that in shock struts with metering pins, except the holes in the metering tube control the flow of fluid from the bottom chamber to the top chamber during compression. *(Figure 13-19)*

Upon lift off or rebound from compression, the shock strut tends to extend rapidly. This could result in a sharp impact at the end of the stroke and damage to the strut. It is typical for shock struts to be equipped with a damping or snubbing device to prevent this.

A recoil valve on the piston or a recoil tube restricts the flow of fluid during the extension stroke, which slows the motion and prevents damaging impact forces. Most shock struts are equipped with an axle as part of the lower cylinder to provide installation of the aircraft wheels. Shock struts without an integral axle have provisions on the end of the lower cylinder for installation of the axle assembly. Suitable connections are provided on all shock strut upper cylinders to attach the strut to the airframe. *(Figure 13-20)*

The upper cylinder of a shock strut typically contains a valve fitting assembly. It is located at or near the top of the cylinder. The valve provides a means of filling the strut with hydraulic fluid and inflating it with air or nitrogen as specified by the manufacturer. A packing gland is employed to seal the sliding joint between the upper and lower telescoping cylinders. It is installed in the open end of the outer cylinder. A packing gland wiper ring is also installed in a groove in the lower bearing or gland nut on most shock struts. It is designed to keep the sliding surface of the piston from carrying dirt, mud, ice, and snow into the packing gland and upper cylinder. Regular cleaning of the exposed portion of the strut piston helps the wiper do its job and decreases the possibility of damage to the packing gland, which could cause the strut to a leak.

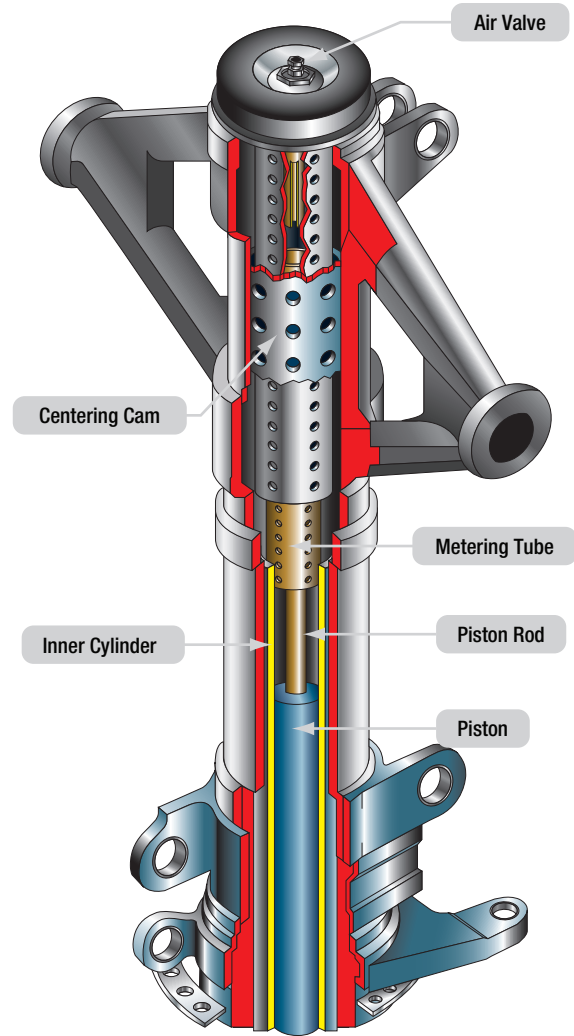


Figure 13-19. Some landing gear shock struts use an internal metering tube rather than a metering pin to control the flow of fluid from the bottom cylinder to the top cylinder.

To keep the piston and wheels aligned, most shock struts are equipped with torque links or torque arms. One end of the links is attached to the fixed upper cylinder. The other end is attached to the lower cylinder (piston) so it cannot rotate. This keeps the wheels aligned. The links also retain the piston in the end of the upper cylinder when the strut is extended, such as after takeoff. *(Figure 13-21)*

Nose gear shock struts are provided with a locating cam assembly to keep the gear aligned. A cam protrusion is attached to the lower cylinder, and a mating lower cam recess is attached to the upper cylinder. These cams line up the wheel and axle assembly in the straight-ahead position when the shock strut is fully extended. This allows the nose wheel to enter the wheel well when the nose gear is retracted and prevents structural damage

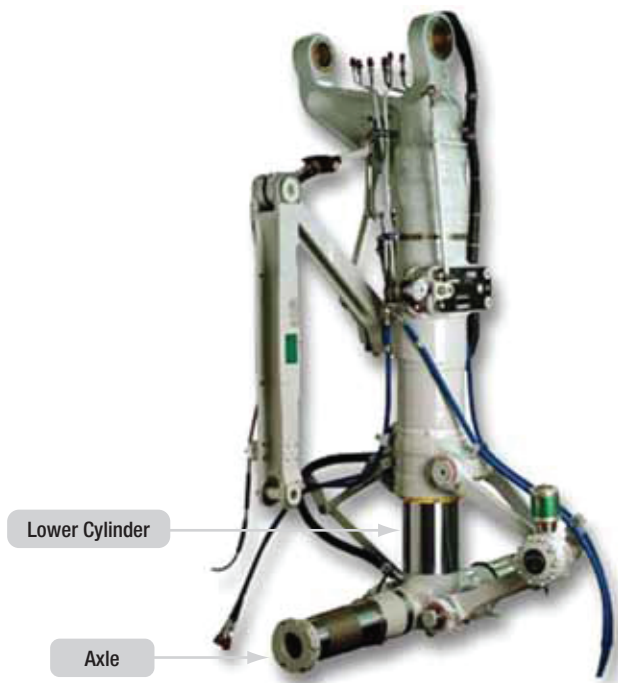


Figure 13-20. Axles machined out of the same material as the landing gear lower cylinder.



Figure 13-21. Torque links align the landing gear and retain the piston in the upper cylinder when the strut is extended.

to the aircraft. It also aligns the wheels with the longitudinal axis of the aircraft prior to landing when the strut is fully extended. (*Figure 13-22*) Many nose gear shock struts also have attachments for the installation of an external shimmy damper. (*Figure 13-23*)

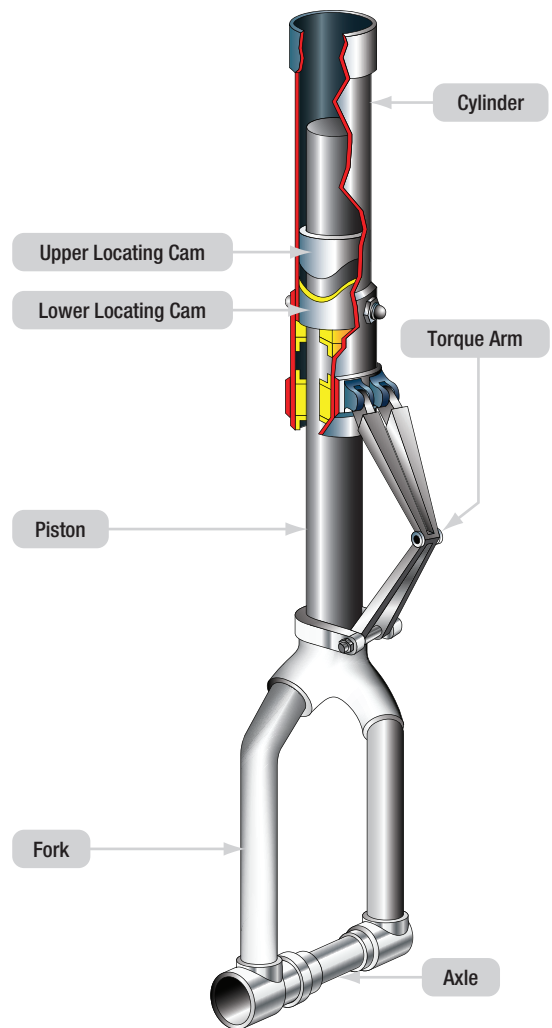


Figure 13-22. An upper locating cam mates into a lower cam recess when the nose landing gear shock strut is extended before landing and before the gear is retracted into the wheel well.



Figure 13-23. A shimmy damper helps control oscillations of the nose gear.

Nose gear struts are often equipped with a locking or disconnect pin to enable quick turning of the aircraft while towing or positioning the aircraft when on the ramp or in a hangar. Disengagement of this pin allows the wheel fork spindle on some aircraft to rotate 360°, thus enabling the aircraft to be turned in a tight radius. At no time should the nose wheel of any aircraft be rotated beyond limit lines marked on the airframe. Nose and main gear shock struts on many aircraft are also equipped with jacking points and towing lugs. Jacks should always be placed under the prescribed points. When towing lugs are provided, the towing bar should be attached only to these lugs. (*Figure 13-24*)

Shock struts contain an instruction plate that gives directions for filling the strut with fluid and for inflating the strut. The instruction plate is usually attached near filler inlet and air valve assembly. It specifies the correct type of hydraulic fluid to use in the strut and the pressure to which the strut should be inflated. It is of utmost importance to become familiar with these instructions prior to filling a shock strut with hydraulic fluid or inflating it with air or nitrogen.

SHOCK STRUT OPERATION

Figure 13-25 illustrates the inner construction of a shock strut. Arrows show the movement of the fluid during compression and extension of the strut. The compression stroke of the shock strut begins as the aircraft wheels touch the ground. As the center of mass of the aircraft moves downward, the strut compresses, and the lower cylinder or piston is forced upward into the upper cylinder. The metering pin is therefore moved

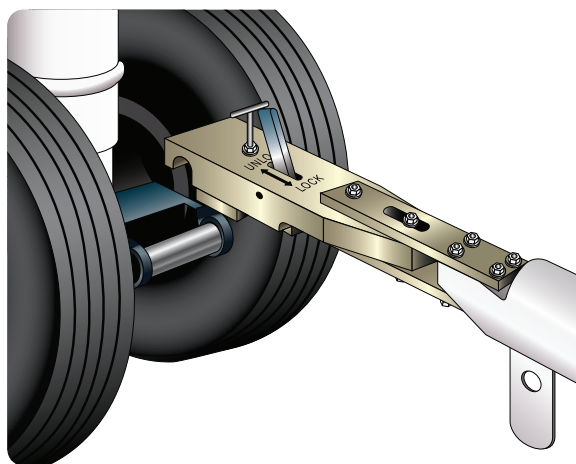


Figure 13-24. A towing lug on a landing gear is the designed means for attaching a tow bar.

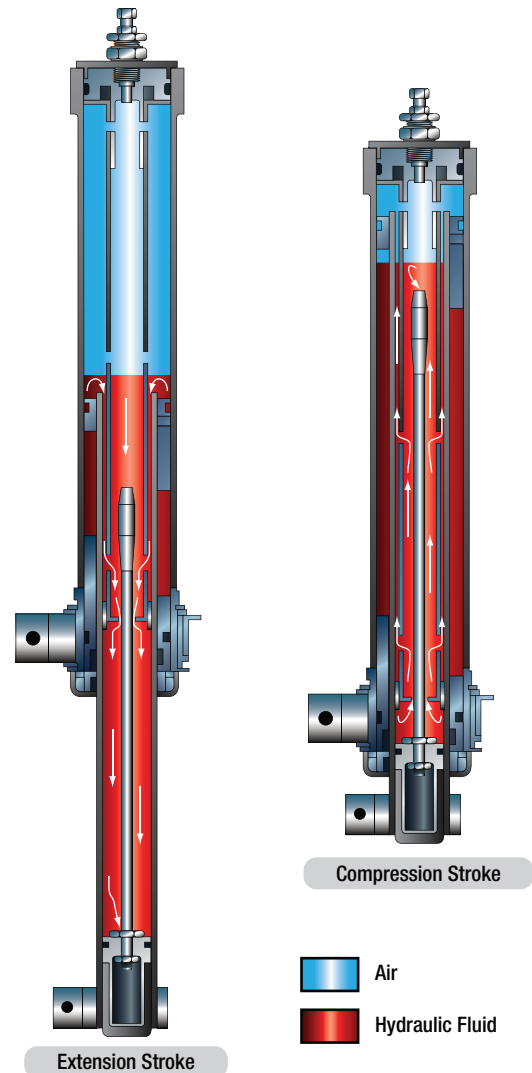


Figure 13-25. Fluid flow during shock strut operation is controlled by the taper of the metering pin in the shock strut orifice.

up through the orifice. The taper of the pin controls the rate of fluid flow from the bottom cylinder to the top cylinder at all points during the compression stroke. In this manner, the greatest amount of heat is dissipated through the walls of the strut. At the end of the downward stroke, the compressed air in the upper cylinder is further compressed which limits the compression stroke of the strut with minimal impact. During taxi operations, the air in the tires and the strut combine to smooth out bumps.

Insufficient fluid, or air in the strut, cause the compression stroke to not be properly limited. The strut could bottom out, resulting in impact forces to be transferred directly to the airframe through the metallic structure of the strut. In a properly serviced strut, the extension stroke of the shock strut operation occurs at the end of the

compression stroke. Energy stored in the compressed air in the upper cylinder causes the aircraft to start moving upward in relation to the ground and lower strut cylinder as the strut tries to rebound to its normal position. Fluid is forced back down into the lower cylinder through restrictions and snubbing orifices. The snubbing of fluid flow during the extension stroke dampens the strut rebound and reduces oscillation caused by the spring action of the compressed air. A sleeve, spacer, or bumper ring incorporated into the strut limits the extension stroke.

Efficient operation of the shock struts requires that proper fluid and air pressure be maintained. To check the fluid level, most struts need to be deflated and compressed into the fully compressed position. Deflating a shock strut can be a dangerous operation. The technician must be thoroughly familiar with the operation of the high pressure service valve found at the top of the strut's upper cylinder. Refer to the manufacturer's instructions for proper deflating technique of the strut in question and follow all necessary safety precautions.

Two common types of high pressure strut servicing valves are illustrated in **Figure 13-26**. The AN6287-1 valve in **Figure 13-26** has a valve core assembly and is rated to 3 000 pounds per square inch (psi). However, the core itself is only rated to 2 000 psi. The MS28889-1

valve in **Figure 13-26B** has no valve core. It is rated to 5 000 psi. The swivel nut on the AN6287-1 valve is smaller than the valve body hex. The MS28889-1 swivel nut is the same size as the valve body hex. The swivel nuts on both valves engage threads on an internal stem that loosens or draws tight the valve stem to a metal seat.

SERVICING SHOCK STRUTS

The following procedures are typical of those used in deflating a shock strut, servicing it with hydraulic fluid, and reinflating the strut.

1. Position the aircraft so that the shock struts are in the normal ground operating position. Make certain that personnel, work stands, and other obstacles are clear of the aircraft. If the maintenance procedures require, securely jack the aircraft.
2. Remove the cap from the air servicing valve. (**Figure 13-27A**)
3. Check the swivel nut for tightness.
4. If the servicing valve is equipped with a valve core, depress it to release any air pressure that may be trapped under the core in the valve body. (**Figure 13-27B**) Always be positioned to the side of the trajectory of any valve core in case it releases. Propelled by strut air pressure, serious injury could result.
5. Loosen the swivel nut. For a valve with a valve core (AN2687-1), rotate the swivel nut one turn

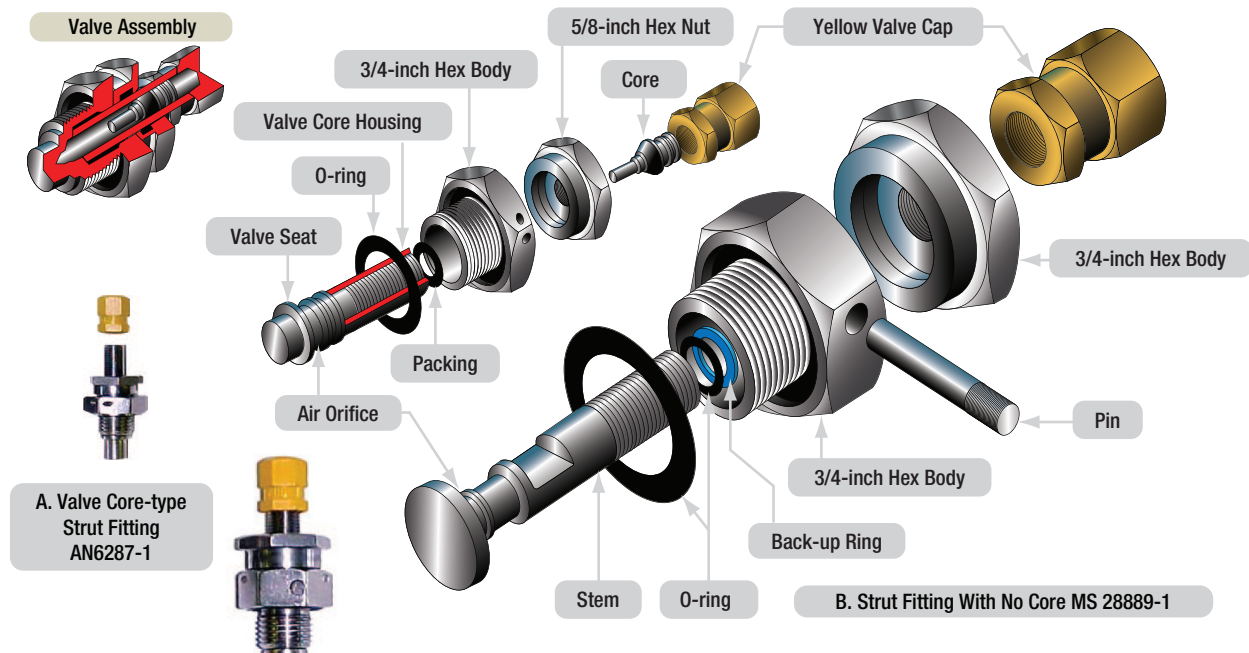


Figure 13-26. Valve core-type (A) and core-free valve fittings (B) are used to service landing gear shock struts.

- (counter clockwise). Using a tool designed for the purpose, depress the valve core to release all of the air in the strut. For a valve without a valve core (MS28889), rotate the swivel nut sufficiently to allow the air to escape.
6. When all air has escaped from the strut, it should be compressed completely. Aircraft on jacks may need to have the lower strut jacked with an exerciser jack to achieve full compression of the strut. (*Figure 13-28*)
 7. Remove the valve core of an AN6287 valve (*Figure 13-27D*) using a valve core removal tool. (*Figure 13-29*) Then, remove the entire service valve by unscrewing the valve body from the strut. (*Figure 13-27E*).
 8. Fill the strut with hydraulic fluid to the level of the service valve port with the approved hydraulic fluid.
 9. Reinstall the air service valve assembly using a new O-ring packing. Torque according to applicable manufacturer's specifications. If an AN2687-1 valve, install a new valve core.
 10. Inflate the strut. A threaded fitting from a controlled source of high pressure air or nitrogen should be screwed onto the servicing valve. Control the flow with the service valve swivel nut. The correct amount of inflation is measured in psi on some struts. Other manufacturers specify struts to be inflated until extension of the lower strut is a certain measurement. Follow

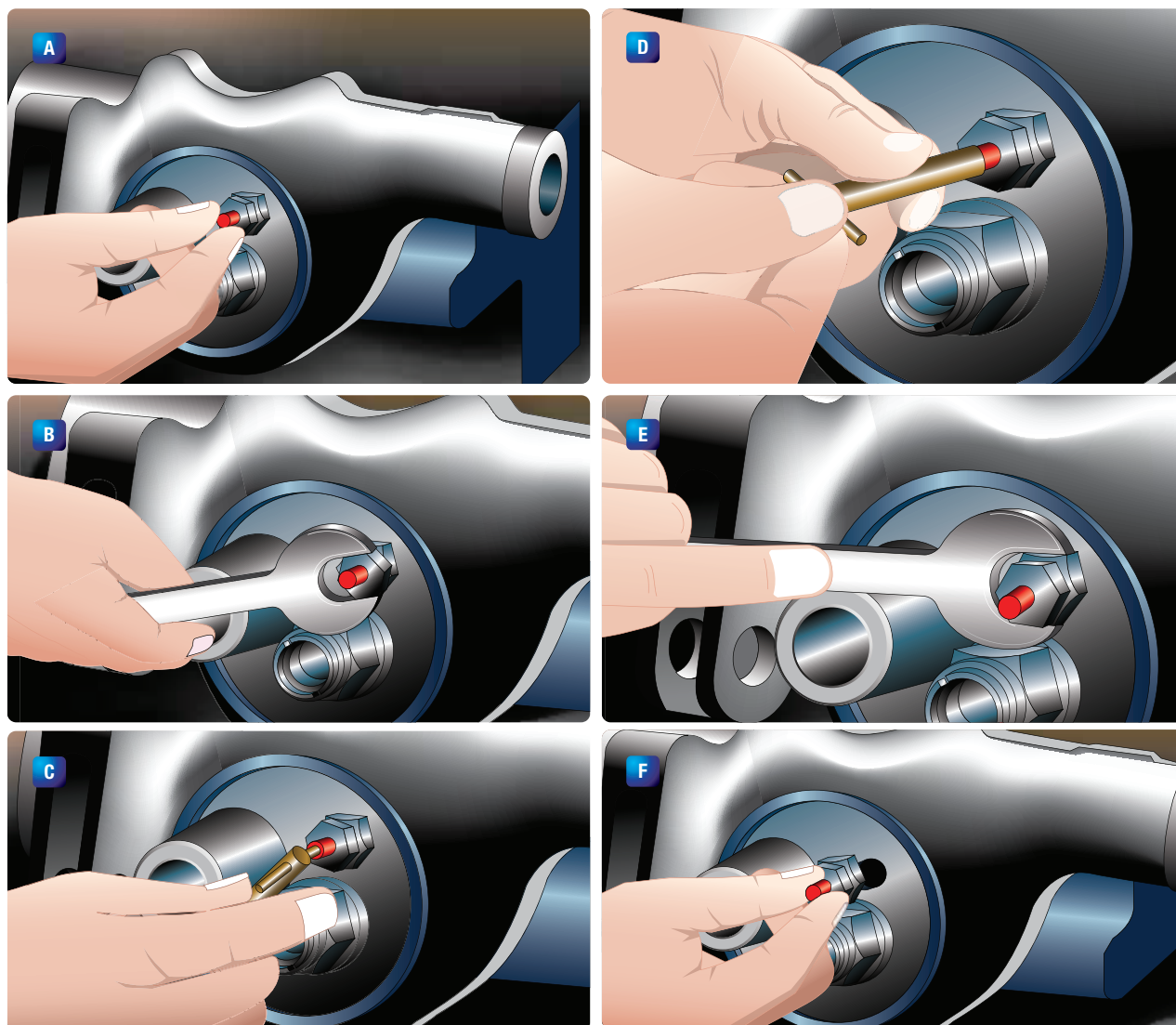


Figure 13-27. Steps in servicing a landing gear shock strut include releasing the air from the strut and removing the service valve from the top of the strut to permit the introduction of hydraulic fluid. Note that the strut is illustrated horizontally. On an actual aircraft installation, the strut is serviced in the vertical position (landing gear down).

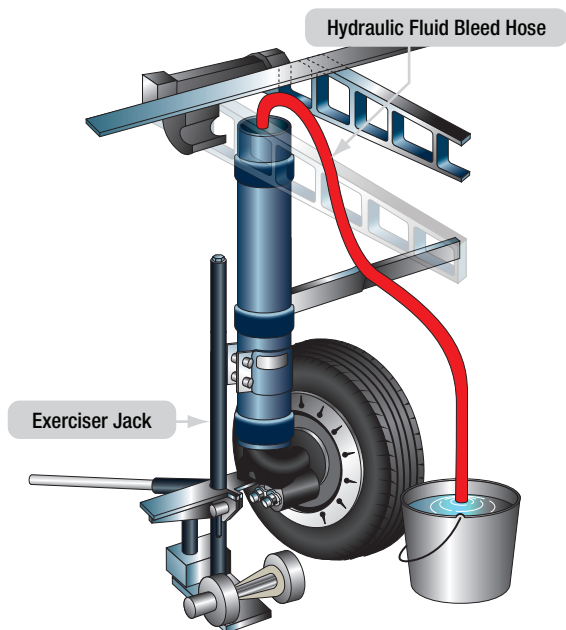


Figure 13-28. Air trapped in shock strut hydraulic fluid is bled by exercising the strut through its full range of motion while the end of an air-tight bleed hose is submerged in a container of hydraulic fluid.

manufacturer's instructions. Shock struts should always be inflated slowly to avoid excess heating and over inflation.

11. Once inflated, tighten the swivel nut and torque as specified.
12. Remove the fill hose fitting and finger tighten the valve cap of the valve.

BLEEDING SHOCK STRUTS

It may be necessary to bleed a shock strut during the service operation or when air becomes trapped in the hydraulic fluid inside the strut. This can be caused by low hydraulic fluid quantity in the strut. Bleeding is normally done with the aircraft on jacks to facilitate repeated extension and compression of the strut to expel the entrapped air. An example procedure for bleeding the shock strut is as follows.

1. Construct and attach a bleed hose containing a fitting suitable for making an airtight connection at the shock strut service valve port. Ensure a long enough hose to reach the ground while the aircraft is on jacks.
2. Jack the aircraft up until the shock struts are fully extended.
3. Release any air pressure in the shock strut.
4. Remove the air service valve assembly.
5. Fill the strut to the level of the service port with approved hydraulic fluid.



Figure 13-29. This valve tool features internal and external thread chasers, a notched valve core removal/installation tool, and a tapered end for depressing a valve core or clearing debris.

6. Attach the bleed hose to the service port and insert the free end of the hose into a container of clean hydraulic fluid. The hose end must remain below the surface of the fluid.
7. Place an exerciser jack or other suitable jack under the shock strut jacking point. Compress and extend the strut fully by raising and lowering the jack. Continue this process until all air bubbles cease to form in the container of hydraulic fluid. Compress the strut slowly and allow it to extend by its own weight.
8. Remove the exerciser jack. Lower the aircraft and remove all other jacks.
9. Remove the bleed hose assembly and fitting from the service port of the strut.
10. Install the air service valve, torque, and inflate the shock strut to the manufacturer's specifications.

EXTENSION AND RETRACTION SYSTEMS: NORMAL

Large aircraft extension and retraction systems are nearly always powered by hydraulics. Typically, the hydraulic pump is driven off of the engine accessory drive. Auxiliary electric hydraulic pumps are also common. Other devices used in a hydraulically-operated retraction system include actuating cylinders, selector valves, uplocks, downlocks, sequence valves, priority valves, tubing, and other conventional hydraulic system components. These units are interconnected so that they permit properly sequenced retraction and extension of the landing gear and the landing gear doors.

The correct operation of any aircraft landing gear retraction system is extremely important. **Figure 13-30** illustrates an example of a simple large aircraft hydraulic landing gear system. The system is on an aircraft that has doors that open before the gear is extended and close after the gear is retracted. The nose gear doors operate via mechanical linkage and do not require hydraulic power. There are many gear and gear door arrangements on various aircraft. Some aircraft have gear doors that close to fair the wheel well after the gear is extended. Others have doors mechanically attached to the outside of the gear so that when it stows inward, the door stows with the gear and fairs with the fuselage skin.

In the system illustrated in **Figure 13-30**, when the flight deck gear selector is moved to the gear-up position, it positions a selector valve to allow pump pressure from the hydraulic system manifold to access

eight different components. The three downlocks are pressurized and unlocked so the gear can be retracted. At the same time, the actuator cylinder on each gear also receives pressurized fluid to the gear-up side of the piston through an unrestricted orifice check valve. This drives the gear into the wheel well. Two sequence valves (C and D) also receive fluid pressure. Gear door operation must be controlled so that it occurs after the gear is stowed. The sequence valves are closed and delay flow to the door actuators. When the gear cylinders are fully retracted, they mechanically contact the sequence valve plungers that open the valves and allow fluid to flow into the close side of the door actuator cylinders. This closes the doors. Sequence valves A and B act as check valves during retraction. They allow fluid to flow one way from the gear-down side of the main gear cylinders back into the hydraulic system return manifold through the selector valve.

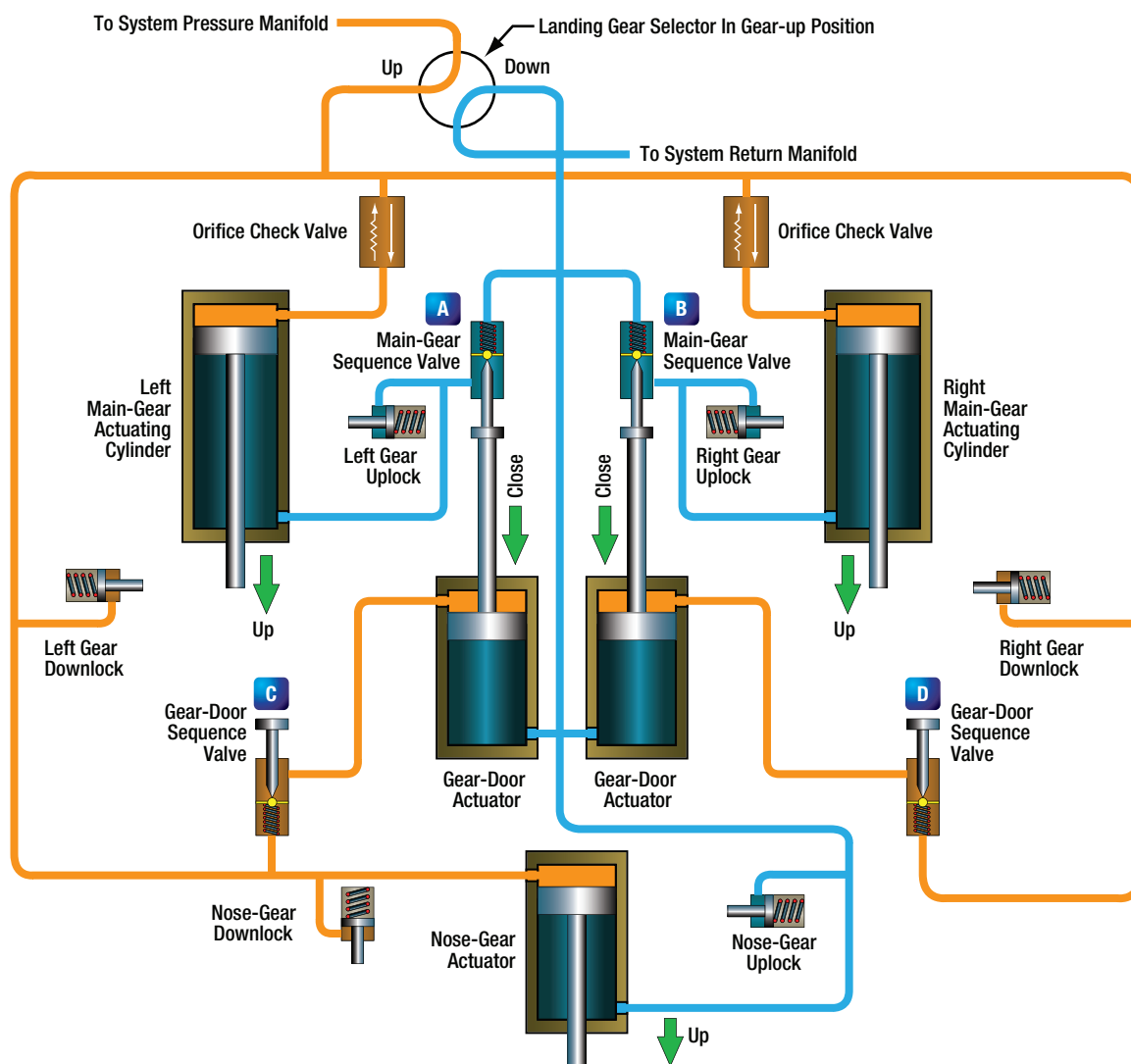


Figure 13-30. A simple large aircraft hydraulic gear retraction system.

To lower the gear, the selector is put in the gear-down position. Pressurized hydraulic fluid flows from the hydraulic manifold to the nose gear uplock, which unlocks the nose gear. Fluid flows to the gear-down side of the nose gear actuator and extends it. Fluid also flows to the open side of the main gear door actuators. As the doors open, sequence valves A and B block fluid from unlocking the main gear uplocks and prevent fluid from reaching the down side of the main gear actuators. When the doors are fully open, the door actuator engages the plungers of both sequence valves to open the valves. The main gear uplocks, then receives fluid pressure and unlock. The main gear cylinder actuators receive fluid on the down side through the open sequence valves to extend the gear. Fluid from each main gear cylinder up-side flows to the hydraulic system return manifold through restrictors in the orifice check valves. The restrictors slow the extension of the gear to prevent impact damage.

There are numerous hydraulic landing gear retraction system designs. Priority valves are sometimes used instead of mechanically operated sequence valves. This controls some gear component activation timing via hydraulic pressure. Particulars of any gear system are found in the aircraft maintenance manual.

The aircraft technician must be thoroughly familiar with the operation and maintenance requirements of this crucial system.

EMERGENCY EXTENSION SYSTEMS

The emergency extension system lowers the landing gear if the main power system fails. There are numerous ways in which this is done depending on the size and complexity of the aircraft. Some aircraft have an emergency release handle in the flight deck that is connected through a mechanical linkage to the gear uplocks. When the handle is operated, it releases the uplocks and allows the gear to free-fall to the extended position under the force created by gravity acting upon the gear.

Other aircraft use a non-mechanical back-up, such as pneumatic power, to unlatch the gear. Large and high performance aircraft are equipped with redundant hydraulic systems. This makes emergency extension less common since a different source of hydraulic power can

be selected if the gear does not function normally. If the gear still fails to extend, some sort of unlatching device is used to release the uplocks and allow the gear to free fall. (*Figure 13-31*)

Consult the aircraft maintenance manual for all emergency landing gear extension system descriptions of operation, performance standards, and emergency extension tests as required.

LANDING GEAR RETRACTION TEST

The proper functioning of a landing gear system and components can be checked by performing a landing gear retraction test. This is also known as swinging the gear. The aircraft is properly supported on jacks for this check, and the landing gear should be cleaned and lubricated if needed. The gear is then raised and lowered as though the aircraft were in flight while a close visual inspection is performed. All parts of the system should be observed for security and proper operation. The emergency back-up extension system should be checked whenever swinging the gear.

Retraction tests are performed at various times, such as during annual inspection. Any time a landing gear component is replaced that could affect the correct functioning of the landing gear system, a retraction test



Figure 13-31. These emergency gear extension handles in a Boeing 737 are located under a floor panel on the flight deck. Each handle releases the gear uplock via a cable system so the gear can freefall into the extended position.

should follow when adjustments to landing gear linkages or components that affect gear system performance are made. It may be necessary to swing the gear after a hard or overweight landing. It is also common to swing the gear while attempting to locate a malfunction within the system. For all required retraction tests and the specific inspection points to check, consult the manufacturer's maintenance manual for the aircraft in question as each landing gear system is unique.

The following is a list of general inspection items to be performed while swinging the gear:

1. Check the landing gear for proper extension and retraction.
2. Check all switches, lights, and warning devices for proper operation.
3. Check the landing gear doors for clearance and freedom from binding.
4. Check landing gear linkage for proper operation, adjustment, and general condition.
5. Check the alternate/emergency extension or retraction systems for proper operation.
6. Investigate any unusual sounds, such as those caused by rubbing, binding, chafing, or vibration.

INDICATIONS AND WARNING

Control of the landing gear and annunciating its position is done through a system of switches. Solid state circuits are controlled with solid state proximity switches located on the gear so that the position of the gear is known at all times. The condition of the gear is also known. For example, DOWN and LOCKED versus DOWN and NOT LOCKED.

Landing gear position indicators are located on the instrument panel adjacent to the gear selector handle. They are used to inform the pilot of gear position status. There are many arrangements for gear indication. Usually, there is a dedicated light for each gear. The most common display for the landing gear being down and locked is an illuminated green light. Three green lights means it is safe to land. All lights out typically indicates that the gear is up and locked, or there may be gear up indicator lights. Gear in transit lights are used on some aircraft as are barber pole displays when a gear is not up or down and locked. Blinking indicator lights also indicate gear in transit. Some manufacturer's use a gear disagree annunciation when the landing gear is not in the same position as the selector. Many aircraft monitor

gear door position in addition to the gear itself. Consult the aircraft manufacturer's maintenance and operating manuals for a complete description of the landing gear indication system. (*Figure 13-32*)

LANDING GEAR SAFETY DEVICES

There are numerous landing gear safety devices. Most common are those that prevent the gear from retracting or collapsing while on the ground (ground locks).

A nose wheel centering device prevents damage to the fuselage and gear by aligning the nose gear with the wheel well bay before retraction. Various safety



Figure 13-32. Landing gear selector panels with position indicator lights. The Boeing 737 panel illuminates red lights above the green lights when the gear is in transit.

and proximity switches and circuits that ensure sequential operation of the landing gear and other system components dependent on the air-ground status of the aircraft are also common. As mentioned above, gear position indicators are another safety device used to communicate to the pilot the position status of each individual landing gear at any time.

GROUND LOCKS

Ground locks are commonly used on aircraft landing gear as extra insurance that the landing gear will remain down and locked while the aircraft is on the ground. They are external devices that are placed in the retraction mechanism to prevent its movement. A ground lock can be as simple as a pin placed into the predrilled holes of gear components that keep the gear from collapsing. Another commonly used ground lock clamps onto the exposed piston of the gear retraction cylinder that prevents it from retracting.

All ground locks should have a red streamers attached to them so they are visible and removed before flight. Ground locks are typically carried in the aircraft and put into place by the flight crew during the post landing walk-around. (*Figure 13-33*)



Figure 13-33. Gear pin ground lock devices.

NOSE WHEEL CENTERING

Since most aircraft have steerable nose wheel gear assemblies for taxiing, a means for aligning the nose gear before retraction is needed. Centering cams built into the shock strut structure accomplish this. An upper cam is free to mate into a lower cam recess when the gear is fully extended. This aligns the gear for retraction. When weight returns to the wheels after landing, the shock strut is compressed, and the centering cams separate allowing the lower shock strut (piston) to rotate in the upper strut cylinder. This rotation is controlled to steer the aircraft. (*Figure 13-34*) Small aircraft sometimes incorporate an external roller or guide pin on the strut.

As the strut is folded into the wheel well during retraction, the roller or guide pin engages a ramp or track mounted to the wheel well structure. The ramp/track guides the roller or pin in such a manner that the nose wheel is straightened as it enters the wheel well.

SAFETY SWITCHES

At least one landing gear squat switch, or safety switch, is found on most aircraft. This is a switch positioned to open and close depending on the extension or compression of the main landing gear strut. (*Figure 13-35*) The squat switch is wired into any number of system operating circuits. One circuit prevents the gear from being retracted while the aircraft is on the ground. There are different ways to achieve this lockout. A solenoid that extends a shaft to physically disable the gear position selector is one such method found on many aircraft. When the landing gear is compressed, the squat safety switch is open, and the center shaft of the solenoid

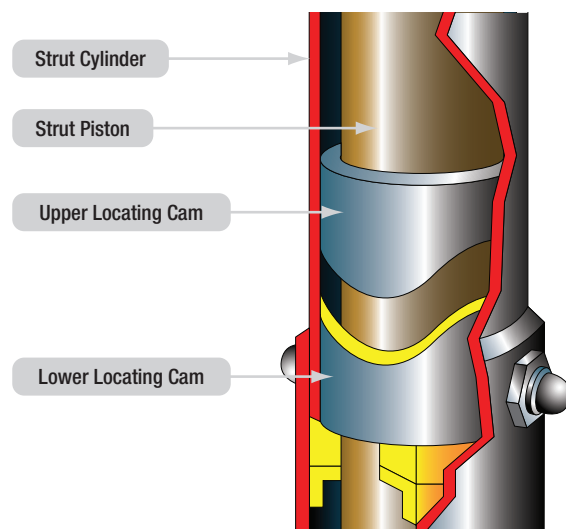


Figure 13-34. A cutaway view of a nose gear internal centering cam.

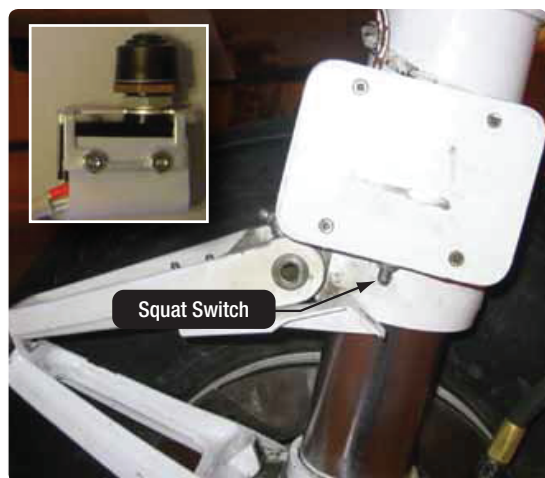


Figure 13-35. Typical landing gear squat switches.

protrudes a hardened lock-pin through the landing gear control handle so that it cannot be moved to the up position. At takeoff, the landing gear strut extends.

The safety switch closes and allows current to flow in the safety circuit. The solenoid energizes and retracts the lock-pin from the selector handle. This permits the gear to be raised. (*Figure 13-36*)

PROXIMITY SENSORS

The use of proximity sensors for gear position safety switches is common in high-performance aircraft. An electromagnetic sensor returns a different voltage to a gear logic unit depending on the proximity of a conductive target to the switch. No physical contact is made. When the gear is in the designed position, the metallic target is close to the inductor in the sensor which reduces the return voltage to an electronic logic unit located in the equipment bay.

This type of sensing is especially useful in the landing gear environment where switches with moving parts can become contaminated with dirt and moisture from runways and taxi ways. The technician is required to ensure that sensor targets are installed the correct distance away from the sensor. Go-No-Go gauges are often used to set the distance. (*Figure 13-37*)

On the latest models of airline aircraft, use of proximity sensors has been expanded to monitor the position of cabin entry doors, cargo doors, access doors and thrust reversers. Two proximity sensor electronic units (PSEUs), each containing two integrated logic circuit cards, receive the proximity sensor information. It is then communicated to systems throughout the aircraft, typically through ARINC 629 or 429 data buses.

Condition warning annunciations originate in this manner as well as permissions for various systems to operate or not operate depending on the condition status of the proximity sensors. The following is a list of the systems that make use of the proximity sensor data from the PSEU's on the Boeing 777:

- Airplane information management system (AIMS)
- Electrical load management system (ELMS)
- Brake system control unit (BSCU)
- Cabin system management unit (CSMU)
- Cargo smoke detection system (CSDS)
- Audio management unit (AMU)

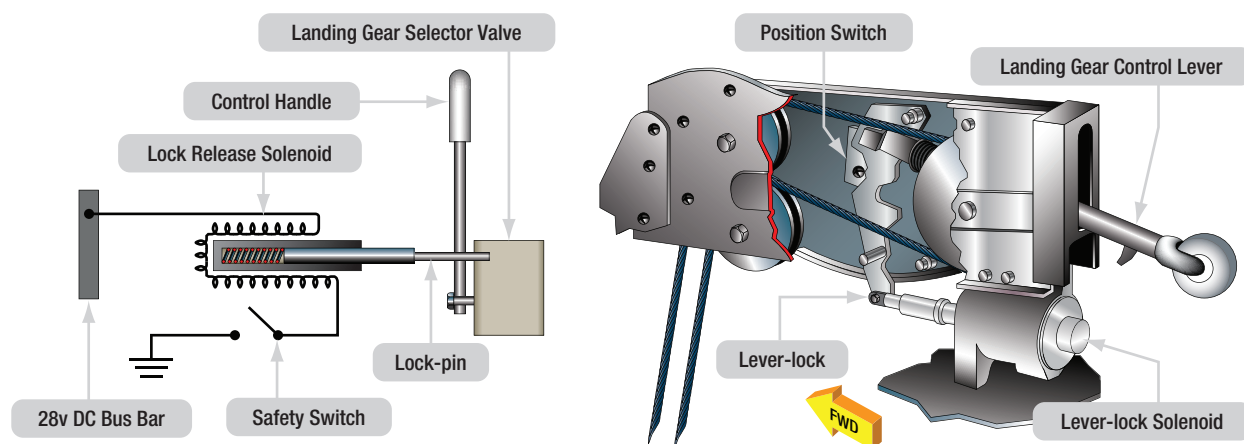


Figure 13-36. A landing gear safety circuit with solenoid that locks the control handle and selector valve from being able to move into the gear up position when the aircraft is on the ground. The safety switch, or squat switch, is located on the aircraft landing gear.

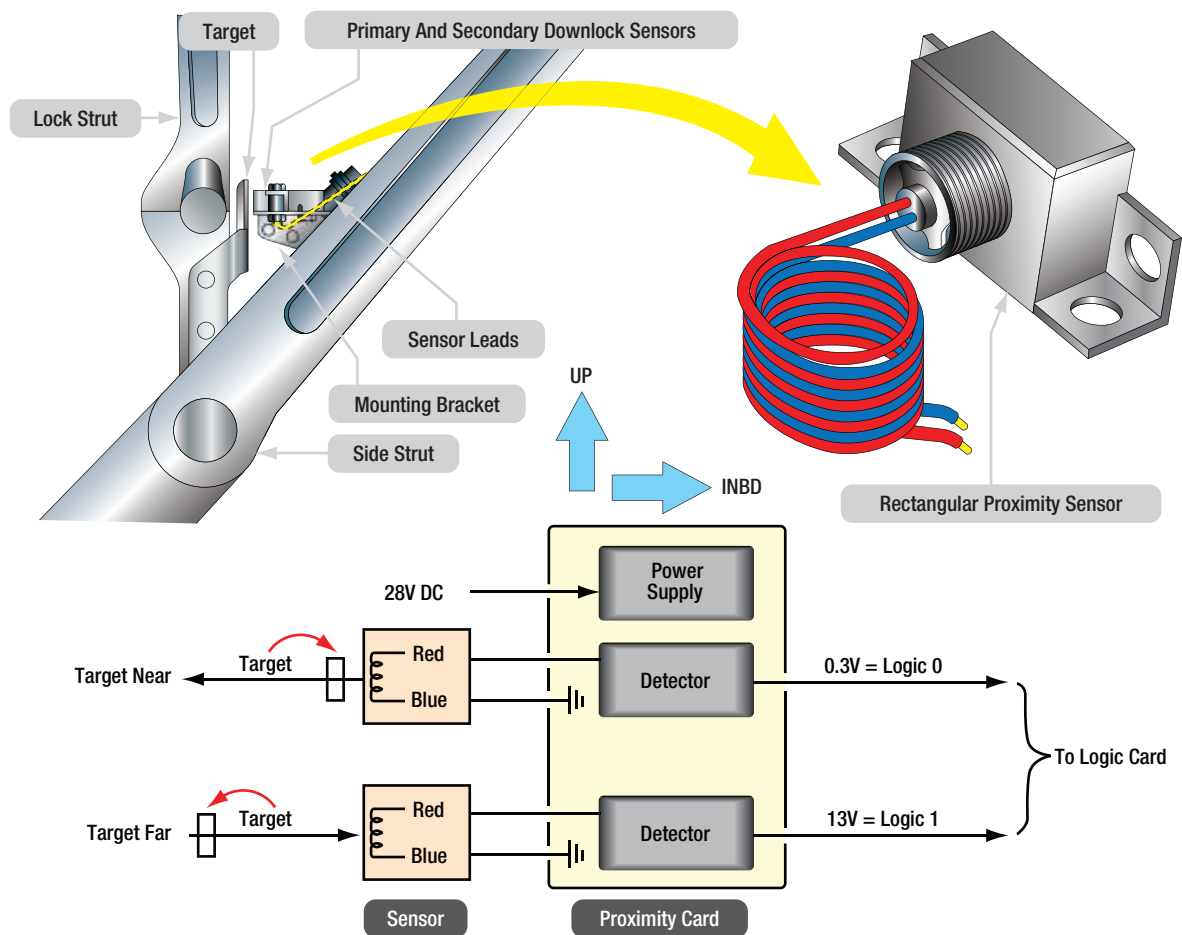


Figure 13-37. Proximity sensors are used instead of contact switches on many landing gear.

AIR-GROUND SENSING

Air-ground sensing in modern aircraft is part of an aircraft-wide control system. The operation (or the prevention of operation) of many on-board systems depends on whether or not the aircraft is in the air or on the ground. The following is a list of some of the systems on a large passenger aircraft that may require this status information from the air-ground sensing system:

- Airplane Information Management System (AIMS)
- Airborne Vibration Monitor Signal Conditioner Unit (AVMSCU)
- Autopilot Flight Director Computer (AFDC)
- Audio Management Unit (AMU)
- APU Controller (APUC)
- Air Supply Cabin Pressure Controller (ASCPC)
- Backup Generator Converter
- Bus Power Control Unit (BPCU)
- Cabin System Management Unit (CSMU)
- Cabin Temperature Controllers (CTC)
- Control Display Unit (CDU)
- Engine Data Interface Unit (EDIU)
- Electrical Load Management System (ELMS)

- Fuel Quantity Indication System (FQIS)
- Flap Slat Electronic Unit (FSEU)
- Generator Control Unit (GCU)
- Overhead Panel Bus Controller (OPBC)
- Passenger Address/Cabin Interphone (PA/CI)
- Proximity Sensor Electronics Unit (PSEU)
- Warning Electronic Unit (WEU)

As described above, squat switches have been used to initiate air ground status for aircraft. The newest airliners use strain gauge load sensors mounted on the landing gear beam structure to detect when the weight of the aircraft is on the main landing gear wheels (ground condition). As the gear beam structure bends under the weight of the aircraft, corresponding electrical signals from the load sensors are sent to a unit or card file in the equipment bay for processing. There, weight on wheels (WOW) integrated circuit logic cards communicate the air-ground information with the other systems on the aircraft through a digital data bus. Responding to signal from the WOW cards, air-ground relays control the distribution of electrical power to the

appropriate systems and devices on the aircraft. Note that the WOW cards are electrostatic sensitive and all electrostatic sensitive material precautions must be observed by the technician.

The WOW load sensors are two piece, variable reluctance strain measurement devices. They are attached by mounting brackets to the landing gear beam. As the gear beam bends with the weight of the aircraft, the distance between the two pieces changes. The distance between sensor parts is what varies the signal sent to the WOW cards because one of the sensor parts is a target and the other contains two electric coils. Thus, the varied electric signal sent to the WOW cards is analog. Communication with most other aircraft systems by the WOW cards is digital. The load sensors are covered for protection from the elements. Replacement of a sensor requires calibration of the sensor. Replacement of the WOW cards requires that all sensors be calibrated.

The use of load sensors instead of proximity switches for air-ground sensing adds the capability of capturing the weight of the aircraft when it is on the ground. Air-ground output from the sensors and cards are displayed on the flight deck on a landing gear maintenance page that is part of the aircraft information management system. Testing in air mode and/ or ground mode of various air-ground related systems can be initiated here. Status of the air-ground system messages also appear on a landing gear maintenance display.

STEERING

NOSE WHEEL STEERING

The nose wheel on most aircraft is steerable from the flight deck via a nose wheel steering system. This allows the aircraft to be directed during ground operation. A few simple aircraft have nose wheel assemblies that caster. Such aircraft are steered during taxi by differential braking.

Most small aircraft have steering capabilities through the use of a simple system of mechanical linkages connected to the rudder pedals. Due to their mass and the need for positive control, large aircraft utilize a power source for nose wheel steering. Hydraulic power predominates. There are many different designs for large aircraft nose steering systems. Most share similar characteristics and components. Control of the steering is from the flight

deck through the use of a small wheel, tiller, or joystick typically mounted on the left side wall. Switching the system on and off is possible on some aircraft.

Mechanical, electrical, or hydraulic connections transmit the controller input movement to a steering control unit. The control unit is a hydraulic metering or control valve. It directs hydraulic fluid under pressure to one or two actuators designed with various linkages to rotate the lower strut. An accumulator and relief valve, or similar pressurizing assembly, keeps fluid in the actuators and system under pressure at all times. This permits the steering actuating cylinders to also act as shimmy dampers. A follow up mechanism consists of various gears, cables, rods, drums, and/or bell-crank, etc. It returns the metering valve to a neutral position once the steering angle has been reached. Many systems incorporate an input subsystem from the rudder pedals for small degrees of turns made while directing the aircraft at high speed during takeoff and landing. Safety valves are typical in all systems to relieve pressure during hydraulic failure so the nose wheel can swivel. The following explanation is accompanied by *Figure 13-38*, *Figure 13-39*, and *Figure 13-40*, which illustrate a large aircraft nose wheel steering system and components. These figures and explanation are for instructional purposes only.

The nose wheel steering wheel connects through a shaft to a steering drum located inside the flight deck control pedestal. The rotation of this drum transmits the steering signal by means of cables and pulleys to the control drum of the differential assembly. Movement of the differential assembly is transmitted by the differential link to the metering valve assembly where it moves the selector valve to the selected position. This provides the hydraulic power for turning the nose gear. (*Figure 13-38*)

As shown in *Figure 13-39*, pressure from the aircraft hydraulic system is directed through the open safety shutoff valve into a line leading to the metering valve. The metering valve then routes the pressurized fluid out of port A, through the right turn alternating line, and into steering cylinder A. This is a one port cylinder and pressure forces the piston to begin extension. Since the rod of this piston connects to the nose steering spindle on the nose gear shock strut which pivots at point X, the extension of the piston turns the steering spindle

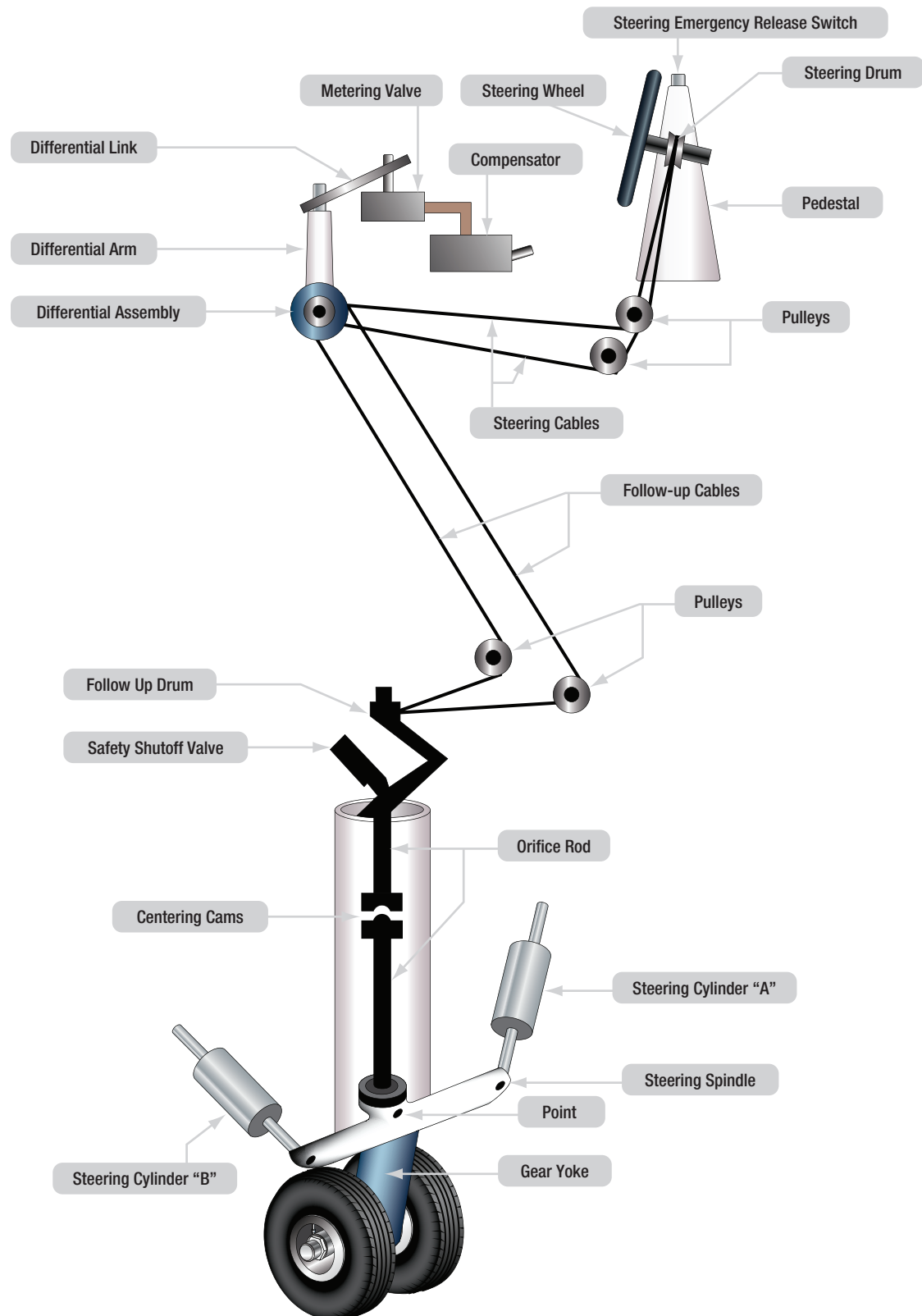


Figure 13-38. Example of a large aircraft hydraulic nose wheel steering system with hydraulic and mechanical units.

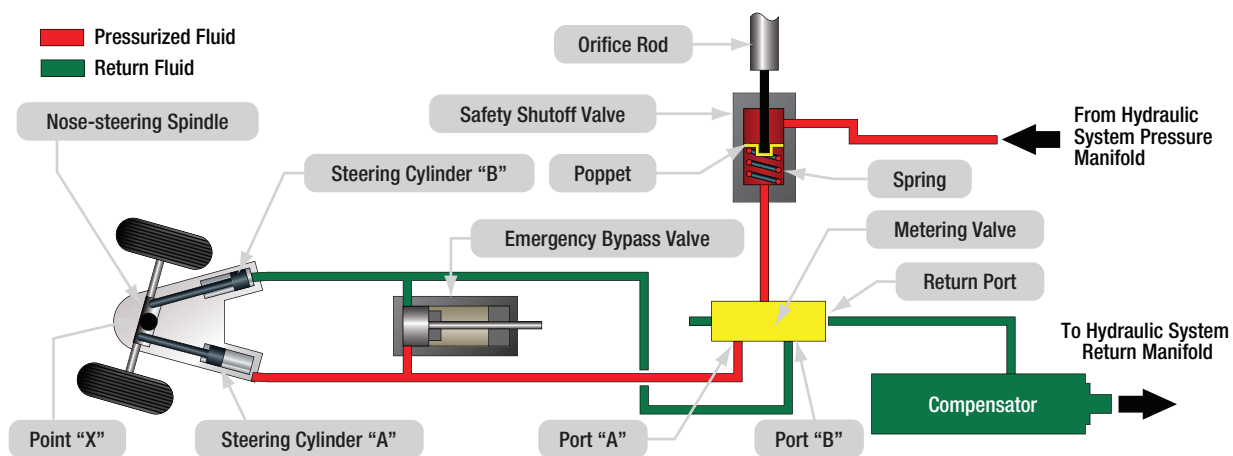


Figure 13-39. Hydraulic system flow diagram of large aircraft nose wheel steering system.

gradually toward the right. As the nose wheel turns, fluid is forced out of steering cylinder B through the left turn alternating line and into port B of the metering valve. The metering valve directs this return fluid into a compensator that routes the fluid into the aircraft hydraulic system return manifold.

As described, hydraulic pressure starts the nose gear turning. However, the gear should not be turned too far. The nose gear steering system contains devices to stop the gear at the selected angle of turn and hold it there. This is accomplished with follow up linkage. As stated, the nose gear is turned by the steering spindle as the piston of cylinder A extends. The rear of the spindle contains gear teeth that mesh with a gear on the bottom of the orifice rod. (*Figure 13-38*)

As the nose gear and spindle turn, the orifice rod also turns but in the opposite direction. This rotation is transmitted by the two sections of the orifice rod to the scissor follow up links located at the top of the nose gear strut. As the follow up links return, they rotate the connected follow up drum, which transmits the movement by cables and pulleys to the differential assembly. Operation of the differential assembly causes the differential arm and links to move the metering valve back toward the neutral position.

The metering valve and the compensator unit of the nose wheel steering system are illustrated in *Figure 13-40*. The compensator unit system keeps fluid in the steering cylinders pressurized at all times. This hydraulic unit consists of a three port housing that encloses a spring-loaded piston and poppet. The left port is an air vent that prevents trapped air at the rear of the piston

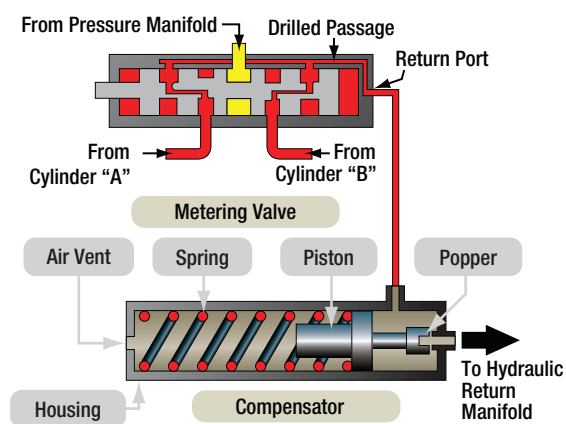


Figure 13-40. Hydraulic system flow diagram of large aircraft nose wheel steering system.

from interfering with the movement of the piston. The second port located at the top of the compensator connects through a line to the metering valve return port. The third port is located at the right side of the compensator. This port connects to the hydraulic system return manifold. It routes the steering system return fluid into the manifold when the poppet valve is open.

The compensator poppet opens when pressure acting on the piston becomes high enough to compress the spring. In this system, 100 psi is required. Therefore, fluid in the metering valve return line is contained under that pressure. The 100 psi pressure also exists throughout the metering valve and back through the cylinder return lines. This pressurizes the steering cylinders at all times and permits them to function as shimmy dampers.

STEERING DAMPER

As mentioned above, large aircraft with hydraulic steering hold pressure in the steering cylinders to

provide the required damping. This is known as steering damping. Some older transport category aircraft have steering dampers that are vane type. Nevertheless, they function to steer the nose wheel, as well as to dampen vibration.

SHIMMY DAMPERS

Torque links attached from the stationary upper cylinder of a nose wheel strut to the bottom movable cylinder or piston of the strut are not sufficient to prevent most nose gear from the tendency to oscillate rapidly, or shimmy, at certain speeds. This vibration wheel shimmy must be controlled through hydraulic damping. The damper can be built integrally within the nose gear, but most often it is an external unit attached between the upper and lower shock struts. It is active during all phases of ground operation while permitting the nose gear steering system to function normally.

PISTON TYPE

Aircraft not equipped with hydraulic nose wheel steering utilize an additional external shimmy damper unit. The case is attached firmly to the upper shock strut cylinder. The shaft is attached to the lower shock strut cylinder and to a piston inside the shimmy damper. As the lower strut cylinder tries to shimmy, hydraulic fluid is forced through a bleed hole in the piston. The restricted flow through the bleed hole dampens the oscillation. (*Figure 13-41*)

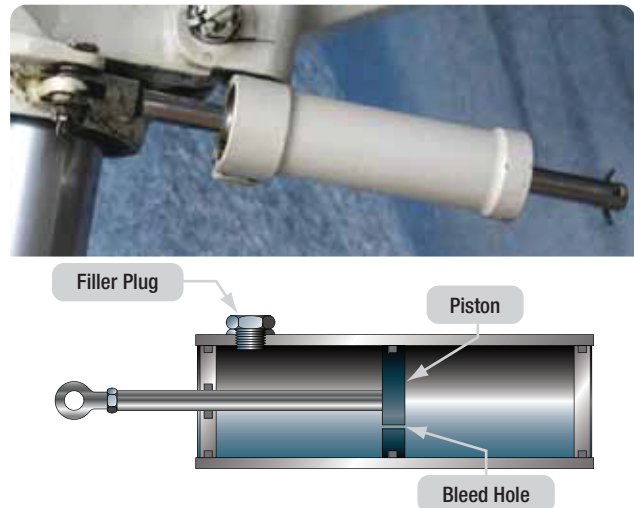


Figure 13-41. A shimmy damper on the nose strut of a small aircraft. The diagram shows the basic internal arrangement of most shimmy dampers. The damper in the photo is essentially the same except the piston shaft extends through both ends of the damper cylinder body.

A piston type shimmy damper may contain a fill port to add fluid or it may be a sealed unit. Regardless, the unit should be checked for leaks regularly. To ensure proper operation, a piston type hydraulic shimmy damper should be filled to capacity.

VANE TYPE

A vane type shimmy damper uses (*Figure 13-42*) fluid chambers created by the vanes separated by a valve orifice in a center shaft. As the nose gear tries to oscillate, vanes

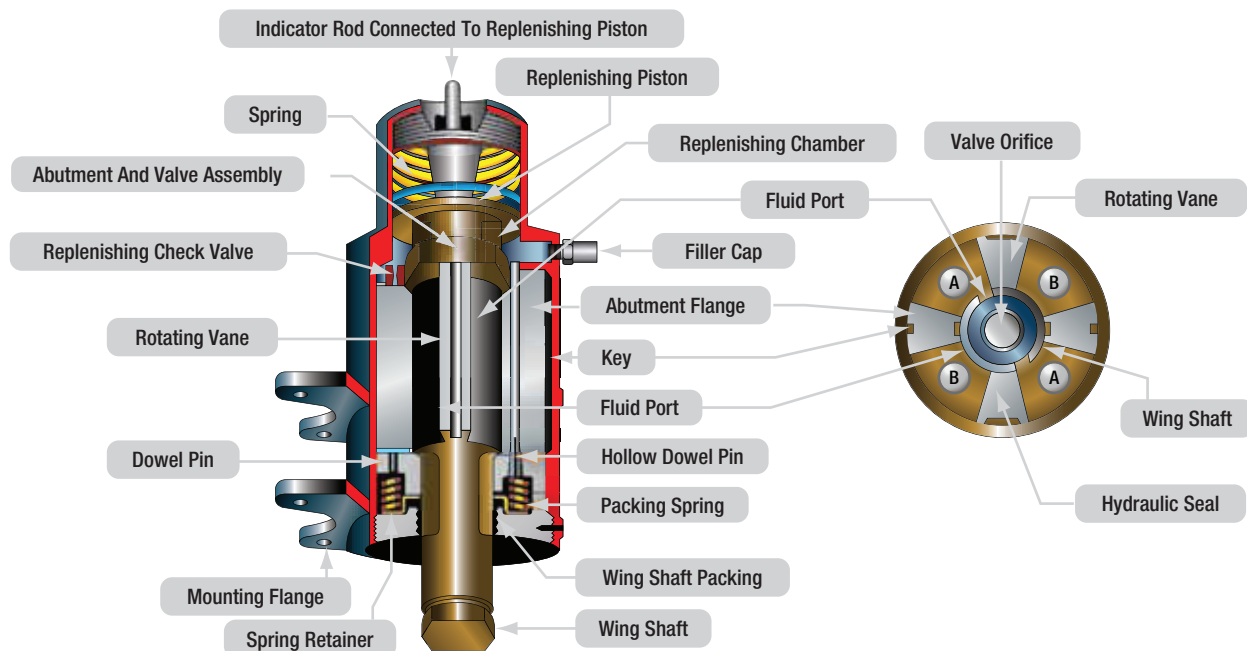


Figure 13-42. A typical vane-type shimmy damper.

rotate to change the size of internal chambers filled with fluid. The chamber size can only change as fast as the fluid can be forced through the orifice. Thus, the gear oscillation is dissipated by the rate of fluid flow. An internal spring loaded replenishing reservoir keeps pressurized fluid in the working chambers and thermal compensation of the orifice size is included. As with the piston type shimmy damper, the vane type damper should be inspected for leaks and kept serviced. A fluid level indicator protrudes from the reservoir end of the unit.

NON-HYDRAULIC SHIMMY DAMPER

Non-hydraulic shimmy dampers are currently certified for many aircraft. They look and fit similar to piston type shimmy dampers but contain no fluid inside.

In place of the metal piston, a rubber piston presses out against the inner diameter of the damper housing when the shimmy motion is received through the shaft. The rubber piston rides on a very thin film of grease and the rubbing action between the piston and the housing provides the damping. This is known as surface effect damping. The materials use to construct this type of shimmy damper provide a long service life without the need to ever add fluid to the unit. (*Figure 13-43*)

LANDING GEAR SYSTEM MAINTENANCE

The moving parts and dirty environment of the landing gear make this an area of regular maintenance. Because of the stresses and pressures acting on the landing gear, inspection, servicing, and other maintenance becomes a continuous process. The most important job in the maintenance of the aircraft landing gear system is thorough accurate inspections. To properly perform inspections, all surfaces should be cleaned to ensure that no trouble spots are undetected.



Figure 13-43. A non-hydraulic shimmy damper uses a rubber piston with lubricant that dampens via motion against the inner diameter of the unit housing.

Periodically, it is necessary to inspect shock struts, trunnion and brace assemblies and bearings, shimmy dampers, wheels, wheel bearings, tires, and brakes. Landing gear position indicators, lights, and warning horns must also be checked for proper operation. During all inspections and visits to the wheel wells, ensure all ground safety locks are installed.

Other landing gear inspection items include checking emergency control handles and systems for proper position and condition. Inspect landing gear wheels for cleanliness, corrosion, and cracks. Check wheel tie bolts for looseness. Examine antiskid wiring for deterioration. Check tires for wear, cuts, deterioration, presence of grease or oil, alignment of slippage marks, and proper inflation. Inspect landing gear mechanism for condition, operation, and proper adjustment. Lubricate the landing gear, including the nose wheel steering. Check steering system cables for wear, broken strands, alignment, and safetying. Inspect landing gear shock struts for such conditions as cracks, corrosion, breaks, and security. Where applicable, check brake clearances and wear.

Various types of lubricant are required to lubricate points of friction and wear on landing gear. Specific products to be used are given by the manufacturer in the maintenance manual. Lubrication may be accomplished by hand or with a grease gun. Follow manufacturer's instructions. Before applying grease to a pressure grease fitting, be sure the fitting is wiped clean of dirt and debris, as well as old hardened grease. Dust and sand mixed with grease produce a very destructive abrasive compound. Wipe off all excess grease while greasing the gear. The piston rods of all exposed strut cylinders and actuating cylinders should be clean at all times.

LANDING GEAR RIGGING AND ADJUSTMENT

Occasionally, it becomes necessary to adjust the landing gear switches, doors, linkages, latches, and locks to ensure proper operation of the landing gear system and doors. When landing gear actuating cylinders are replaced and when length adjustments are made, over travel must be checked. Over travel is the action of the cylinder piston beyond the movement necessary for landing gear extension and retraction. The additional action operates the landing gear latch mechanisms.

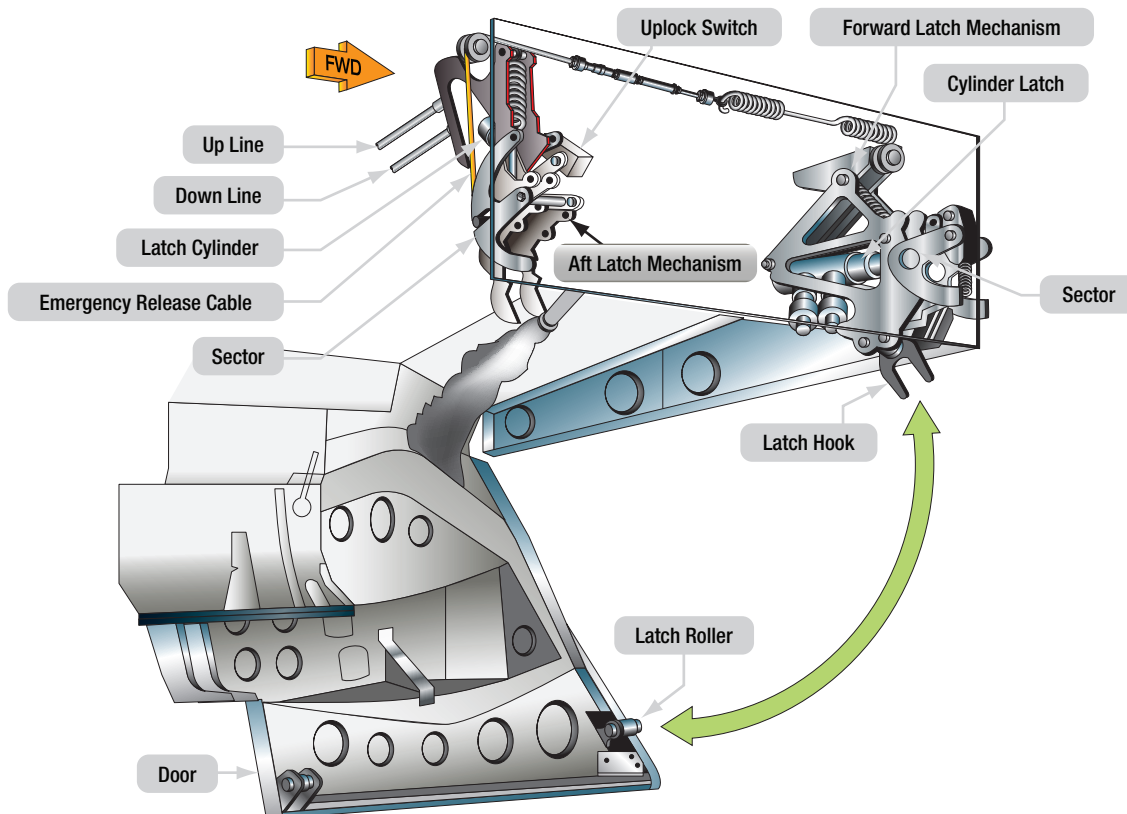


Figure 13-44. An example of a main landing gear door latch mechanism.

A wide variety of aircraft types and landing gear system designs result in procedures for rigging and adjustment that vary from aircraft to aircraft. Uplock and downlock clearances, linkage adjustments, limit switch adjustments, and other adjustments must be confirmed by the technician in the manufacturer's maintenance data before taking action. The following examples of various adjustments are given to convey concepts, rather than actual procedures for any particular aircraft.

ADJUSTING LANDING GEAR LATCHES

The adjustment of various latches is a primary concern to the aircraft technician. Latches are generally used in landing gear systems to hold the gear up or down and/or to hold the gear doors open or closed. Despite numerous variations, all latches are designed to do the same thing. They must operate automatically at the proper time, and they must hold the unit in the desired position. A typical landing gear door latch is examined below. Many gear up latches operate similarly. Clearances and dimensional measurements of rollers, shafts, bushings, pins, bolts, etc., are common.

On this particular aircraft, the landing gear door is held closed by two latches. To have the door locked securely, both latches must grip and hold the door tightly against the aircraft structure. The principle components of each latch mechanism are shown in *Figure 13-44*. They are a hydraulic latch cylinder, a latch hook, a spring loaded crank-and-lever linkage with sector, and the latch hook. When hydraulic pressure is applied, the cylinder operates the linkage to engage (or disengage) the hook with (or from) the roller on the gear door. In the gear-down sequence, the hook is disengaged by the spring load on the linkage. In the gear-up sequence, when the closing door is in contact with the latch hook, the cylinder operates the linkage to engage the latch hook with the door roller. Cables on the landing gear emergency extension system are connected to the sector to permit emergency release of the latch rollers. An uplock switch is installed on, and actuated by, each latch to provide a gear up indication in the flight deck.

With the gear up and the door latched, inspect the latch roller for proper clearance as shown in *Figure 13-45A*. On this installation, the required clearance is $1/8 + 3/32$ -inch. If the roller is not within tolerance, it may be adjusted by loosening its mounting

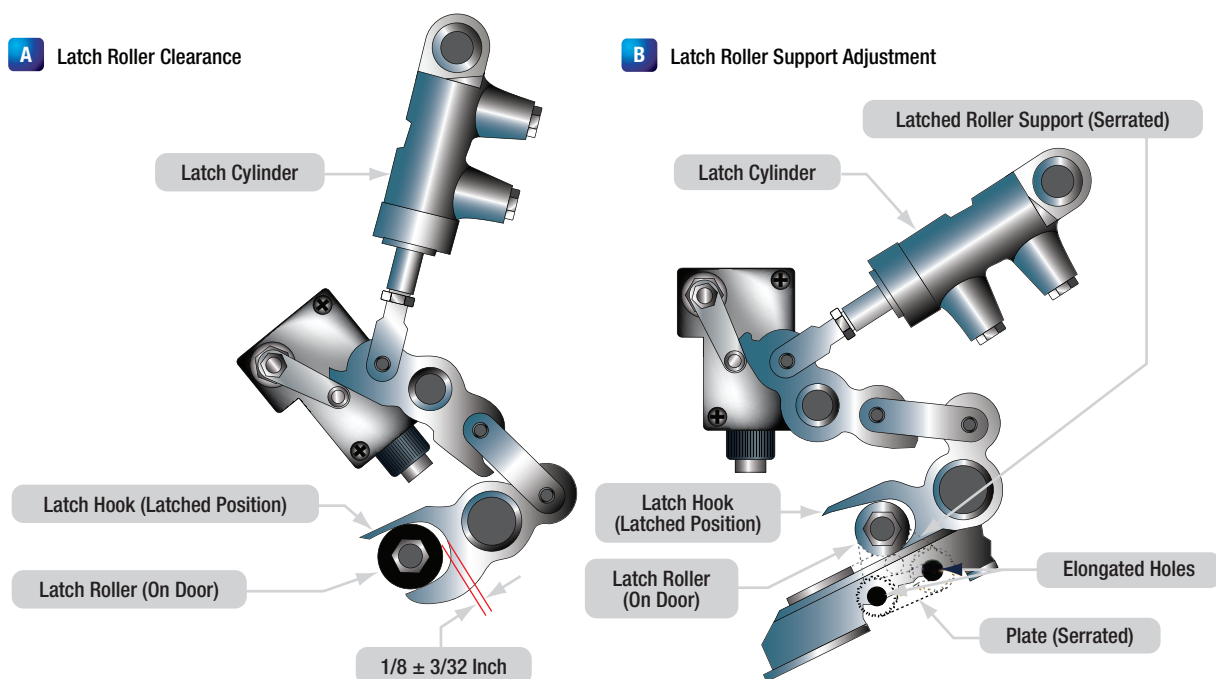


Figure 13-45. Main landing gear door latch roller clearance measurement and adjustment.

bolts and raising or lowering the latch roller support. This is accomplished via the elongated holes and serrated locking surfaces of the latch roller support and serrated plate. (*Figure 13-45B*)

GEAR DOOR CLEARANCES

Landing gear doors have specific allowable clearances between the doors and the aircraft structure that must be maintained. Adjustments are typically made at the hinge installations or to the connecting links that support and move the door. On some installations, door hinges are adjusted by placing a serrated hinge with an elongated mounting hole in the proper position in a hinge support fitting. Using serrated washers, the mounting bolt is torqued to hold the position. *Figure 13-46* illustrates this type of mounting, which allows linear adjustments via the elongated hole.

The distance landing gear doors open or close may depend upon the length of the door linkage. Rod end adjustments are common to fit the door. Adjustments to door stops are also a possibility. The manufacturer's maintenance manual specifies the length of the linkages and gives procedure for adjusting the stops. Follow all specified procedures that are accomplished with the aircraft on jacks and the gear retracted. Doors that are too tight can cause structural damage. Doors that are too loose catch wind in flight, which could cause wear and potential failure, as well as parasite drag.

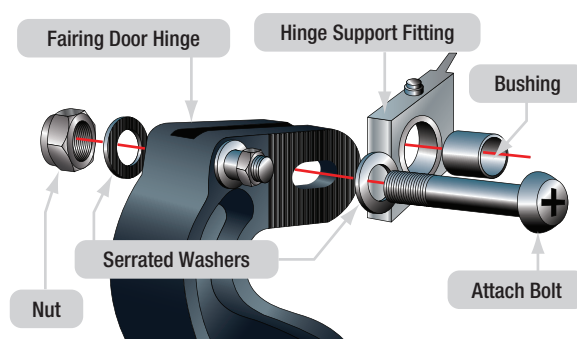


Figure 13-46. An adjustable door hinge installation for setting door clearance.

DRAW AND SIDE BRACE ADJUSTMENT

Each landing gear has specific adjustments and tolerances per the manufacturer that permit the gear to function as intended. A common geometry used to lock a landing gear in the down position involves a collapsible side brace that is extended and held in an over center position through the use of a locking link. Springs and actuators may also contribute to the motion of the linkage. Adjustments and tests are needed to ensure proper operation.

Figure 13-47 illustrates a landing gear on a small aircraft with such a side brace. It consists of an upper and lower link hinged at the center that permits the brace to jackknife during retraction of the gear. The upper end pivots on a trunnion attached to structure in the wheel well overhead.

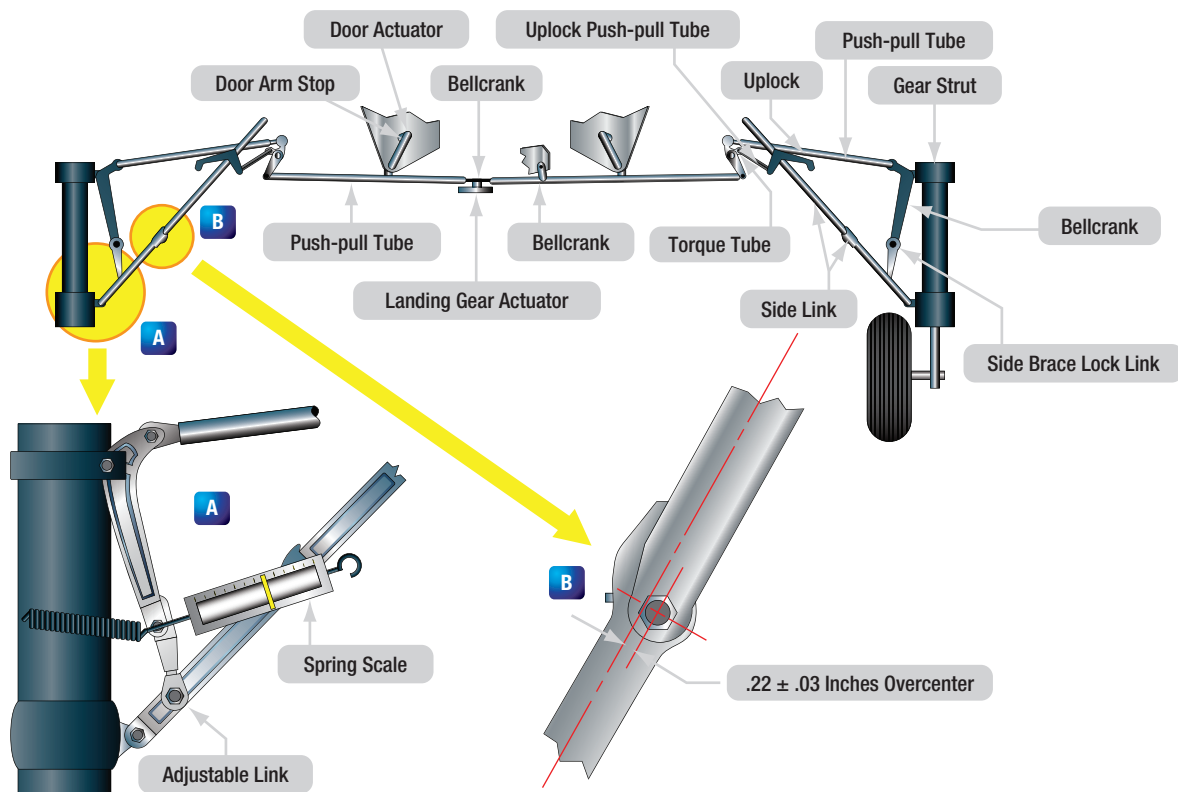


Figure 13-47. Over-center adjustments on a small aircraft main gear.

The lower end is attached to the shock strut. A locking link is incorporated between the upper end of the shock strut and the lower drag link. It is adjustable to provide the correct amount of over center travel of the side brace links. This locks the gear securely in the down position to prevent collapse of the gear. To adjust the over center position of the side brace locking link, the aircraft must be placed on jacks. With the landing gear in the down position, the lock link end fitting is adjusted so that the side brace links are held firmly over center. When the gear is held inboard six inches from the down and locked position and then released, the gear must free fall into the locked down position.

In addition to the amount the side brace links are adjusted to travel over center, down lock spring tension must also be checked. This is accomplished with a spring scale. The tension on this particular gear is between 40 and 60 pounds. Check the manufacturer's maintenance data for each aircraft to ensure correct tensions exist and proper adjustments are made.

AIRCRAFT WHEELS

Aircraft wheels are an important component of a landing gear system. With tires mounted upon them, they support the entire weight of the aircraft during

taxi, takeoff, and landing. The typical aircraft wheel is lightweight, strong, and made from aluminum alloy. Some magnesium alloy wheels also exist. Early aircraft wheels were of single piece construction, much the same as the modern automobile wheel. As aircraft tires were improved for the purpose they serve, they were made stiffer to better absorb the forces of landing without blowing out or separating from the rim. Stretching such a tire over a single piece wheel rim was not possible. A two piece wheel was developed. Early two piece aircraft wheels were essentially one piece wheels with a removable rim to allow mounting access for the tire. These are still found on older aircraft. (*Figure 13-48*)

Later, wheels with two nearly symmetrical halves were developed. Nearly all modern aircraft wheels are of this two piece construction. (*Figures 13-49 and 13-50*)

WHEEL CONSTRUCTION

The typical modern two piece aircraft wheel is cast or forged from aluminum or magnesium alloy. The halves are bolted together and contain a groove at the mating surface for sealing the assembly for use with tubeless tires. The bead seat area of a wheel is where the tire actually contacts the wheel. It is the critical area that accepts the significant tensile loads from the tire during

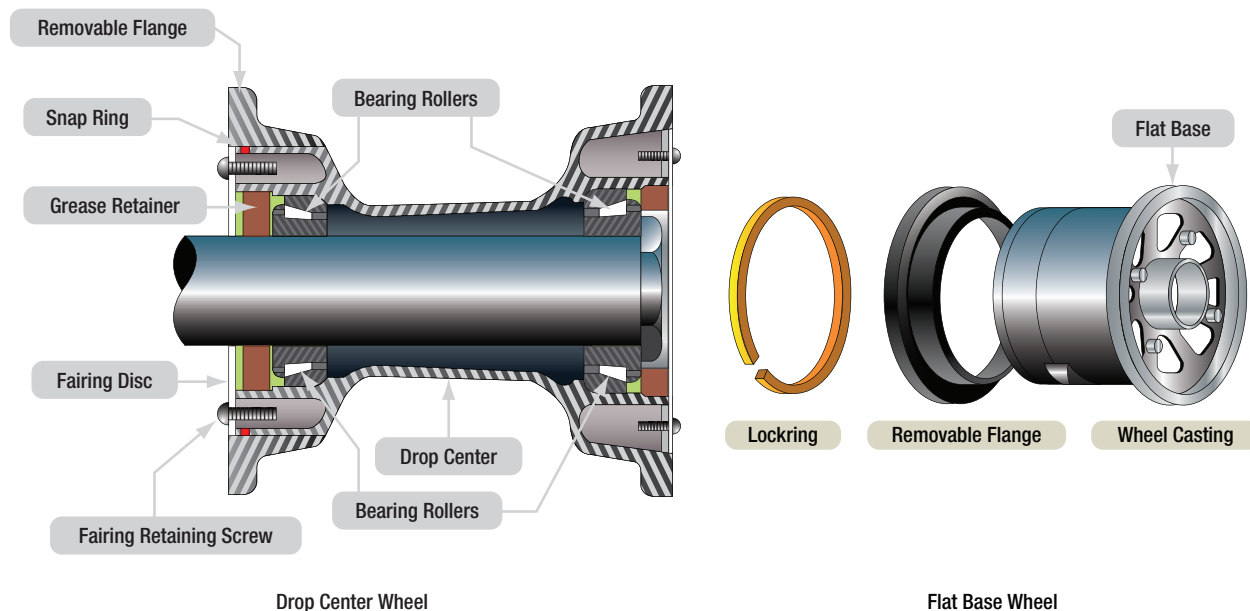


Figure 13-48. Removable flange wheels found on older aircraft are either drop center or flat base types.

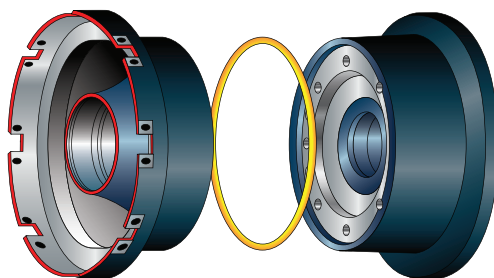


Figure 13-49. Two-piece split-wheel aircraft wheels found on modern light aircraft.

landing. To strengthen this area during manufacturing, the bead seat area is typically rolled to prestress it with a compressive stress load.

INBOARD WHEEL HALF

Wheel halves are not identical. The primary reason for this is that the inboard wheel half must have a means for accepting and driving the rotor(s) of the aircraft brakes that are mounted on both main wheels. Tangs on the rotor are fitted into steel reinforced keyways on many wheels. Other wheels have steel keys bolted to the inner wheel halves. These are made to fit slots in the perimeter of the brake rotor. Some small aircraft wheels have provisions for bolting the brake rotor to the inner wheel half. Regardless, the inner wheel half is distinguishable from the outer wheel half by its brake mounting feature. (Figure 13-51)

Both wheel halves contain a bearing cavity formed into the center that accepts the polished steel bearing cup, tapered roller bearing, and grease retainer of a typical wheel bearing set-up. A groove may also be machined to accept a retaining clip to hold the bearing assembly in place when the wheel assembly is removed. The wheel bearings are a very important part of the wheel assembly and are discussed in a later section of this chapter. The inner wheel half of a wheel used on a high performance aircraft is likely to have one or more thermal plugs. (Figure 13-52)

During heavy braking, temperatures can become so great that tire temperature and pressure rise to a level resulting in explosion of the wheel and tire assembly. The thermal plug core is filled with a low melting point alloy. Before tire and wheel temperatures reach the point of explosion, the core melts and deflates the tire. The tire must be removed from service, and the wheel must be inspected in accordance with the wheel manufacturer's instructions before return to service if a thermal plug melts. Adjacent wheel assemblies should also be inspected for signs of damage. A heat shield is commonly installed under the inserts designed to engage the brake rotor to assist in protecting the wheel and tire assembly from overheating.

An over inflation safety plug may also be installed in the inner wheel half. This is designed to rupture and release all of the air in the tire should it be over inflated. The fill

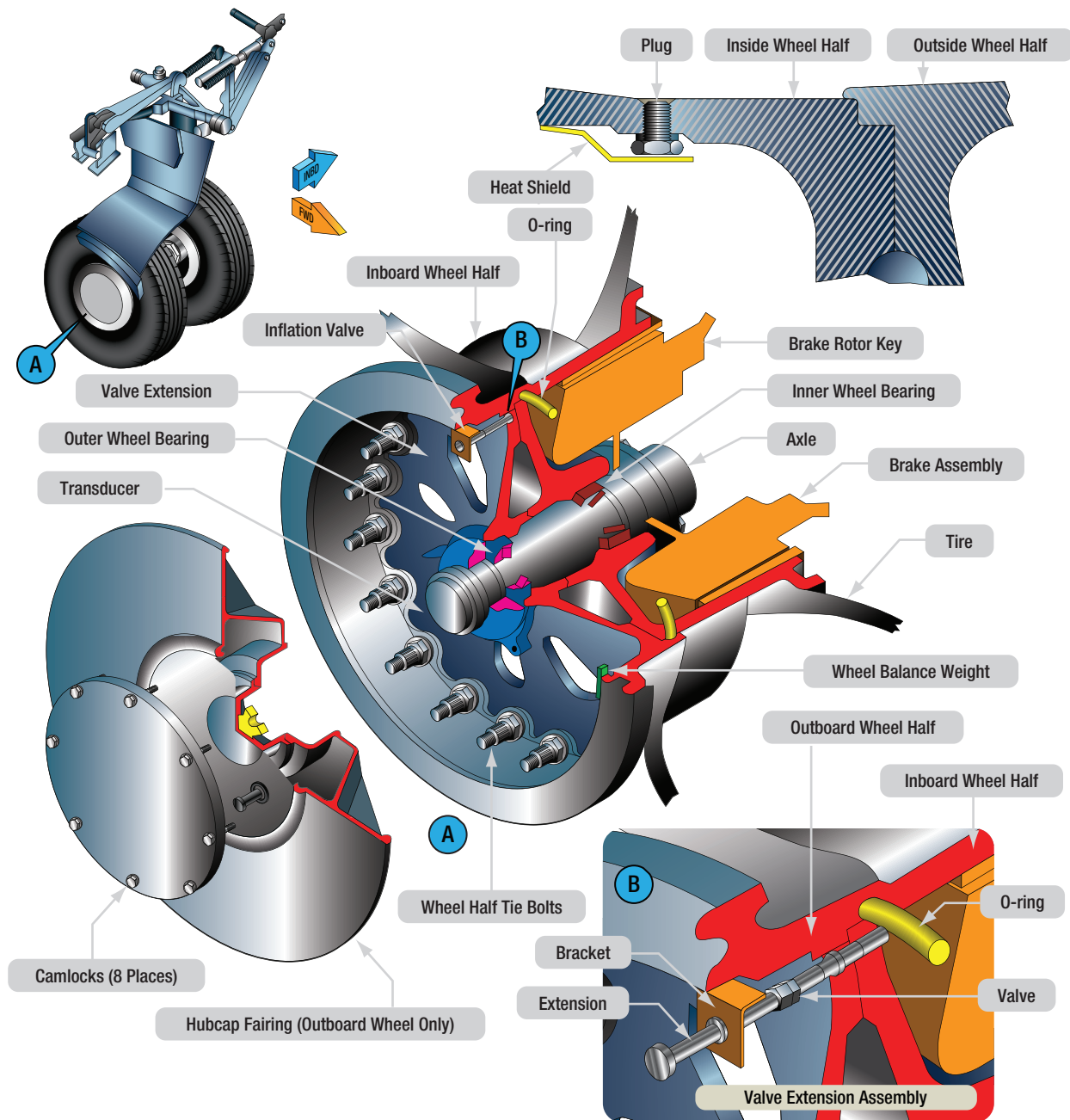


Figure 13-50. Features of a two piece aircraft wheel found on a modern airliner.



Figure 13-51. Keys on the inner wheel half of an aircraft wheel used to engage and rotate the rotors of a disc brake.

valve is also often installed in the inner wheel half with the stem extending through holes in the outer wheel half to permit access for inflation and deflation.

OUTBOARD WHEEL HALF

The outboard wheel half bolts to the inboard wheel half to make up the wheel assembly upon which the tire is mounted. The center boss is constructed to receive a bearing cup and bearing assembly as it does on the inboard wheel half. The outer bearing and end of the axle is capped to prevent contaminants from entering this area.

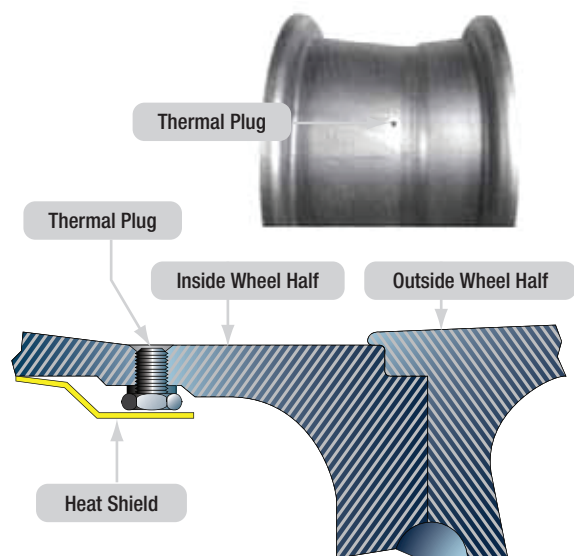


Figure 13-52. Heavy use of the aircraft brakes can cause tire air temperature and pressure to rise to a level resulting in explosion of the wheel assembly. To alleviate this, thermal plug(s) mounted in the inner wheel half of a high performance aircraft wheels are made with a fusible core that melts and releases the air from the tire before explosion.

Aircraft with antiskid brake systems typically mount the wheel spin transducer here. It is sealed and may also serve as a hub cap. The 737 outer wheel half illustrated in *Figure 13-50* also has a hub cap fairing over the entire wheel half. This is to fair it with the wind since the outer wheel half does not close behind a gear door on this aircraft. Hub caps may also be found on fixed gear aircraft.

The outboard wheel half provides a convenient location of the valve stem used to inflate and deflate tubeless tires. Alternately, it may contain a hole through which a valve stem extension may pass from the inner wheel half or the valve stem itself may fit through such a hole if a tube type tire is used.

WHEEL INSPECTION

An aircraft wheel assembly is inspected while on the aircraft as often as possible. A more detailed inspection and any testing or repairs may be accomplished with the wheel assembly removed from the aircraft.

ON AIRCRAFT INSPECTION

The general condition of the aircraft wheel assemblies can be inspected while on the aircraft. Any signs of suspected damage that may require removal of the wheel assembly from the aircraft should be investigated.

Proper Installation

The landing gear area is such a hostile environment that the technician should inspect the landing gear including the wheels, tires, and brakes whenever possible.

Proper installation of the wheels should not be taken for granted. All wheel tie bolts and nuts must be in place and secure. A missing bolt is grounds for removal, and a thorough inspection of the wheel halves in accordance with the wheel manufacturer's procedures must be performed due to the stresses that may have occurred. The wheel hub dust cap and anti skid sensor should also be secure. The inboard wheel half should interface with the brake rotor with no signs of chafing or excessive movement. All brake keys on the wheel must be present and secure.

Examine the wheels for cracks, flaked paint, and any evidence of overheating. Inspect thermal plugs to ensure no sign of the fusible alloy having been melted. Thermal plugs that have permitted pressure loss in the tire require that the wheel assembly be removed for inspection. All other wheels with brakes and thermal plugs should be inspected closely while on the aircraft to determine if they too have overheated. Each wheel should be observed overall to ensure it is not abnormally tilted. Flanges should not be missing any pieces, and there should be no areas on the wheel that show significant impact damage.

Axle Nut Torque

Axle nut torque is of extreme importance on an aircraft wheel installation. If the nut is too loose, the bearing and wheel assembly may have excessive movement. The bearing cup(s) could loosen and spin, which could damage the wheel. There could also be impact damage from the bearing rollers which leads to bearing failure. (*Figure 13-53*) An over torqued axle nut prevents the bearing from properly accepting the weight load of the aircraft. The bearing spins without sufficient lubrication to absorb the heat caused by the higher friction level. This too leads to bearing failure. All aircraft axle nuts must be installed and torqued in accordance with the airframe manufacturer's maintenance procedures.

OFF AIRCRAFT WHEEL INSPECTION

Discrepancies found while inspecting a wheel mounted on the aircraft may require further inspection with the wheel removed from the aircraft. Other items such as

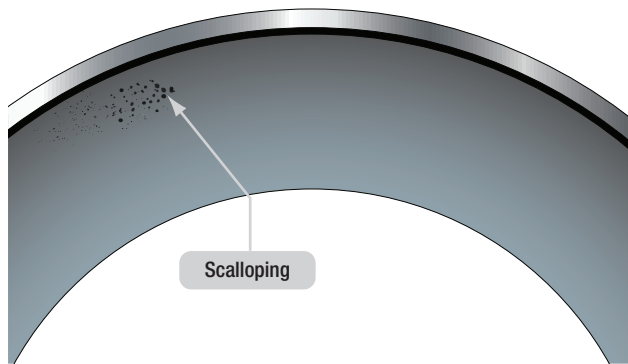


Figure 13-53. Improper loose torque on the axle nut can cause excessive end play leading to bearing race damage known as scalloping.

bearing condition, can only be performed with the wheel assembly removed. A complete inspection of the wheel requires that the tire be removed from the wheel rim. Observe the following caution when removing a wheel assembly from an aircraft.

CAUTION: Deflate the tire before starting the procedure of removing the wheel assembly from the aircraft. Wheel assemblies have been known to explode while removing the axle nut, especially when dealing with high pressure, high performance tires. The torque of the nut can be the only force holding together a defective wheel or one with broken tie bolts. When loosened, the high internal pressure of the tire can create a catastrophic failure that could be lethal to the technician. It is also important to let aircraft tires cool before removal. Three hours or more is needed for cool down. Approach the wheel assembly from the front or rear, not broadside.

Do not stand in the path of the released air and valve core trajectory when removing air from the tire as it could seriously injure the technician should it release from the valve stem.

NOTE: As a precautionary measure, remove only one tire and wheel assembly from a pair at a time. This leaves a tire and wheel assembly in place should the aircraft fall off its jack, resulting in less chance of damage to the aircraft and injury to personnel.

Loosening the Tire from the Wheel Rim

After inflation and usage, an aircraft tire has a tendency to adhere to the wheel, and the bead must be broken to remove the tire. There are mechanical and hydraulic presses designed for this purpose. In the absence of a device specifically made for the job, an arbor press can be used with patience working sequentially around the wheel as close as possible to the bead. (*Figure 13-54*)

There should be no air pressure in the tire while it is being pressed off of the wheel. Never pry a tire off of the rim with a screwdriver or other device. The wheels are relatively soft. Any nick or deformation causes a stress concentration that can easily lead to wheel failure.

Disassembly of the Wheel

Disassembly of the wheel should take place in a clean area on a flat surface, such as a table. Remove the wheel bearing first and set aside for cleaning and inspecting. The tie bolts can then be removed. Do not use an impact tool to disassemble the tie bolts. Aircraft wheels are made of relatively soft aluminum and magnesium alloys.



Figure 13-54. Tire beads must be broken from the wheel to remove the tire. A mechanical removal tool designed for breaking the bead is shown in (A); a hydraulic press designed with the capacity for large aircraft wheels is shown in (B); and an arbor press is shown in (C). All are tools available to the technician for this purpose.

They are not designed to receive the repeated hammering of an impact tool and will be damaged if used.

Cleaning the Wheel Assembly

Clean the wheel halves with the solvent recommended by the wheel manufacturer. Use of a soft brush helps this process. Avoid abrasive techniques, materials, and tools, such as scrapers, capable of removing the finish off of the wheel. Corrosion can quickly form and weaken the wheel if the finish is missing in an area. When the wheels are clean, they can be dried with compressed air.

Inspection of the Wheel Halves

A thorough visual inspection of each wheel half should be conducted for discrepancies specified in the wheel manufacturer's maintenance data. Use of a magnifying glass is recommended.

Corrosion is one of the most common problems encountered while inspecting wheels. Locations where moisture is trapped should be checked closely. It is possible to dress out some corrosion according to the manufacturer's instructions. An approved protective surface treatment and finish must be applied before returning the wheel to service. Corrosion beyond stated limits is cause for rejection of the wheel. In addition to corrosion, cracks in certain areas of the wheel are particularly prevalent. One such area is the bead seat area. (*Figure 13-55*)

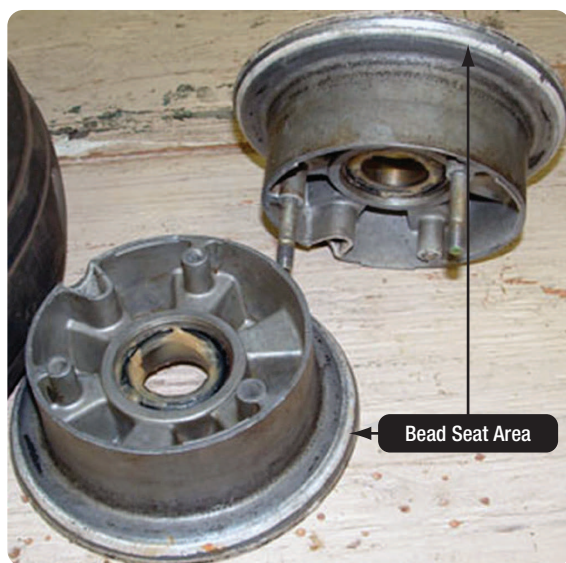


Figure 13-55. The bead seat areas of a light aircraft wheel set. Eddy current testing for cracks in the bead seat area is common.

The high stress of landing is transferred to the wheel by the tire in this contact area. Hard landings produce distortion or cracks that are very difficult to detect. This is a concern on all wheels and is most problematic in high-pressure, forged wheels. Dye penetrant inspection is generally ineffective when checking for cracks in the bead area. There is a tendency for cracks to close up tightly once the tire is dismounted, and the stress is removed from the metal. Eddy current inspection of the bead seat area is required. Follow the wheel manufacturer's instruction when performing the eddy current check.

The wheel brake disc drive key area is another area in which cracks are common. The forces experienced when the keys drive the disc against the stopping force of the brakes are high. Generally, a dye penetrant test is sufficient to reveal cracks in this area. All drive keys should be secure with no movement possible. No corrosion is permitted in this area. (*Figure 13-56*)

Wheel Tie Bolt Inspection

Wheel half tie bolts are under great stress while in service and require inspection. The tie bolts stretch and change dimension usually at the threads and under the bolt head. These are areas where cracks are most common. Magnetic particle inspection can reveal these cracks. Follow the maintenance manual procedures for inspecting tie bolts.

Key and Key Screw Inspection

On most aircraft inner wheel halves, keys are screwed or bolted to the wheel to drive the brake disc(s). The drive keys are subject to extreme forces when the brakes are applied. As mentioned, there should be no movement between the wheel and the keys. The bolts should be checked for security, and the area around the keys should be inspected for cracks. There is also a limitation on how worn the keys can be since too much wear

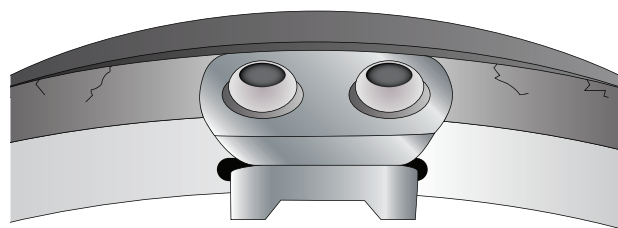


Figure 13-56. Inspection for cracks in the wheel disc drive key area is performed with dye penetrant on many wheels.

allows excessive movement. The wheel manufacturer's maintenance instructions should be used to perform a complete inspection of this critical area.

Fusible Plug Inspection

Fusible plugs or thermal plugs must be inspected visually. These threaded plugs have a core that melts at a lower temperature than the outer part of the plug. This is to release air from the tire should the temperature rise to a dangerous level. A close inspection should reveal whether any core has experienced deformation that might be due to high temperature. If detected, all thermal plugs in the wheel should be replaced with new plugs. (*Figure 13-57*)

Balance Weights

The balance of an aircraft wheel assembly is important. When manufactured, each wheel set is statically balanced. Weights are added to accomplish this if needed. They are a permanent part of the wheel assembly and must be installed to use the wheel. The balance weights are bolted to the wheel halves and can be removed when cleaning and inspecting the wheel. They must be re-fastened in their original position. When a tire is mounted to a wheel, balancing of the wheel and tire assembly may require that additional weights be added. These are usually installed around the circumference of the outside of the wheel and should not be taken as substitutes for the factory wheel set balance weights. (*Figure 13-58*)

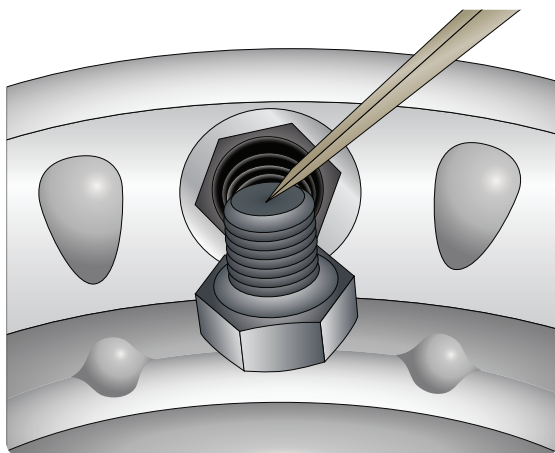


Figure 13-57. Visually inspect the core of a thermal or fusible plug for deformation associated with heat exposure. Replace all of the plugs if any appear to have begun to deform.

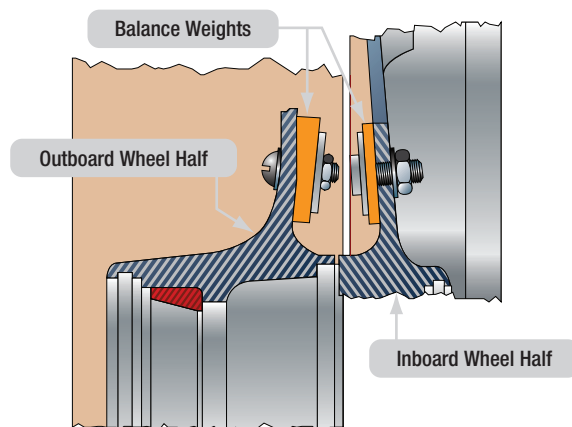


Figure 13-58. Two piece aircraft wheels are statically balanced when manufactured and may include weights attached to each wheel half that must stay with the wheel during its entire serviceable life.

WHEEL BEARINGS

Cleaning the Wheel Bearings

The bearings should be removed from the wheel to be cleaned with the recommended solvent, such as varsol, naphtha, or Stoddard™ solvent. Soaking the bearings in solvent is acceptable to loosen any dried-on grease. Bearings are brushed clean with a soft bristle brush and dried with compressed air. Never rotate the bearing while drying with compressed air. The high speed metal to metal contact of the bearing rollers with the race causes heat that damages the metal surfaces. The bearing parts could also cause injury should the bearing come apart. Always avoid steam cleaning of bearings. The surface finish of the metals will be compromised leading to early failure.

Inspection of Wheel Bearings

Once cleaned, the wheel bearing is inspected. There are many unacceptable conditions of the bearing and bearing cup, which are grounds for rejection. In fact, nearly any flaw detected in a bearing assembly is likely to be grounds for replacement. Common conditions of a bearing that are cause for rejection are as follows:

- **Galling** - caused by rubbing of mating surfaces. The metal gets so hot it welds, and the surface metal is destroyed as the motion continues and pulls the metal apart in the direction of motion. (*Figure 13-59*)
- **Spalling** - a chipped away portion of the hardened surface of a bearing roller or race. (*Figure 13-60*)
- **Overheating** - caused by lack of sufficient lubrication results in a bluish tint to the metal surface. The ends



Figure 13-59. Galling is caused by rubbing of mating surfaces. The metal gets so hot it welds, and the surface metal is destroyed as the motion continues and pulls the metal apart in the direction of motion.



Figure 13-60. Spalling is a chipped away portion of the hardened surface of a bearing roller or race.

of the rollers shown were overheated causing the metal to flow and deform, as well as discolor. The bearing cup raceway is usually discolored as well. (*Figure 13-61*)

- Brinelling - caused by excessive impact. It appears as indentations in the bearing cup raceways. Any static overload or severe impact can cause true brinelling that leads to vibration and premature bearing failure. (*Figure 13-62*)
- False Brinelling - caused by vibration of the bearing while in a static state. Even with a static overload, lubricant can be forced from between the rollers and the raceway. Submicroscopic particles removed at the points of metal-to-metal contact oxidize. They work to remove more particles spreading the damage. This is also known as frictional corrosion. It can be identified by a rusty coloring of the lubricant. (*Figure 13-63*)
- Staining and surface marks - located on the bearing cup as grayish black streaks with the same spacing as the rollers and caused by water that has gotten into the bearing. It is the first stage of deeper corrosion that follows. (*Figure 13-64*)
- Etching and corrosion - caused when water and the damage caused by water penetrates the surface treatment of the bearing element. It appears as a reddish/brown discoloration. (*Figure 13-65*)

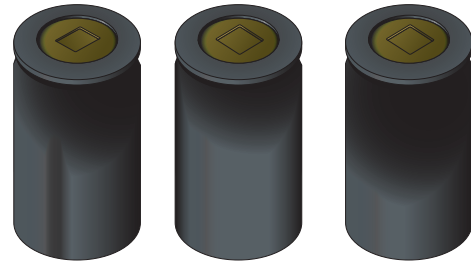


Figure 13-61. Overheating caused by lack of sufficient lubrication results in a bluish tint to the metal surface. The ends of the rollers shown were overheated causing the metal to flow and deform, as well as discolor. The bearing cup raceway is usually discolored as well.

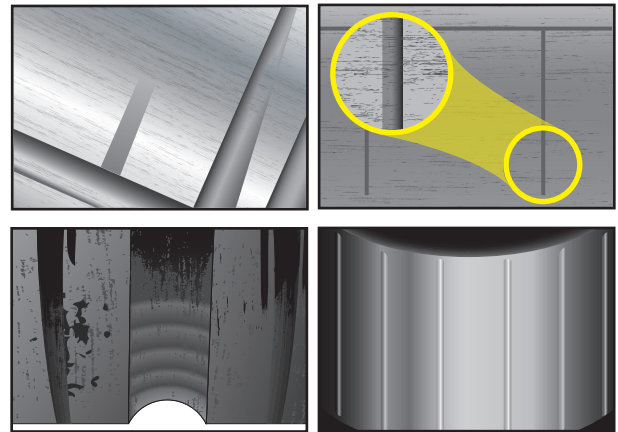


Figure 13-62. Brinelling is caused by excessive impact. It appears as indentations in the bearing cup raceways. Any static overload or severe impact can cause true brinelling, which leads to vibration and premature bearing failure.

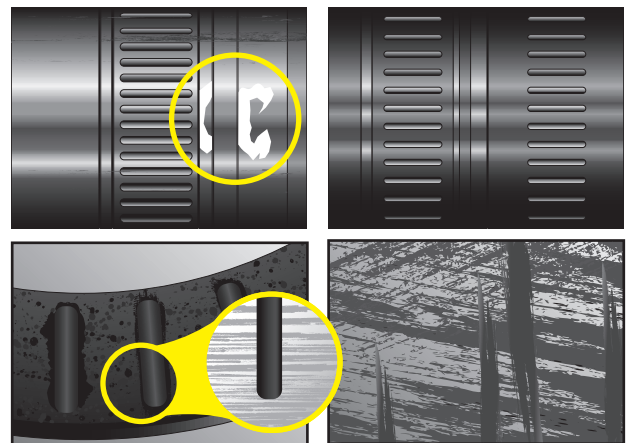


Figure 13-63. False brinelling is caused by vibration of the bearing while in a static state. Even with a static overload, it can force the lubricant from between the rollers and the raceway. Submicroscopic particles removed at the points of metal-to-metal contact oxidize. They work to remove more particles spreading the damage. This is also known as frictional corrosion. It can be identified by a rusty coloring of the lubricant.

- Bruising - caused by fine particle contamination possibly from a bad seal or improper maintenance of bearing cleanliness. It leaves a less than smooth surface on the bearing cup. (*Figure 13-66*)

The bearing cup does not require removal for inspection; however, it must be firmly seated in the wheel half boss. There should be no evidence that a cup is loose or able to spin. (*Figure 13-67*) The cup is usually removed by heating the wheel in a controlled oven and pressing it out or tapping it out with a non-metallic drift.

The installation procedure is similar. The wheel is heated and the cup is cooled with dry ice before it is tapped into place with a non-metallic hammer or drift. The outside of the race is often sprayed with primer before insertion. Consult the wheel manufacturer's maintenance manual for specific instructions.

BEARING HANDLING AND LUBRICATION

Periodically, wheel bearings must be removed, cleaned, inspected, and lubricated. When cleaning a wheel bearing, use the recommended cleaning solvent.

Do not use gasoline or jet fuel. Dry the bearing by directing a blast of dry air between the rollers. Do not direct the air so that it spins the bearing as without lubrication, this could cause the bearing to fly apart resulting in injury. When inspecting the bearing, check for defects that would render it unserviceable, such as cracks, flaking, broken bearing surfaces, roughness due to impact pressure or surface wear, corrosion or pitting, discoloration from excessive heat, cracked or broken bearing cages, and scored or loose bearing cups or cones that would affect proper seating on the axle or wheel. If any discrepancies are found, replace the bearing with a serviceable unit. Bearings should be lubricated immediately after cleaning and inspection to prevent corrosion.

Handling of bearings is of the utmost importance. Contamination, moisture, and vibration, even while the bearing is in a static state, can ruin a bearing. Avoid conditions where these may affect bearings and be sure to install and torque bearings into place according manufacturer's instructions.



Figure 13-64. Staining and surface marks on the bearing cup that are grayish black streaks with the same spacing as the rollers are caused by water that has gotten into the bearing. It is the first stage of deeper corrosion that will follow.

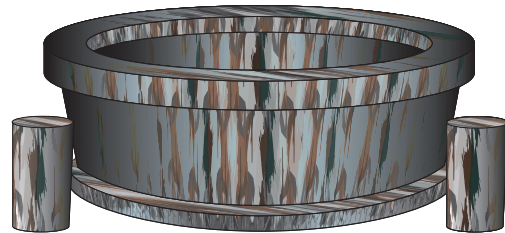


Figure 13-65. Etching and corrosion is caused when water, and the damage caused by water, penetrates the surface treatment of the bearing element. It appears as a reddish/brown discoloration.



Figure 13-66. Bruising is caused by fine particle contamination possibly from a bad seal or improper maintenance of bearing cleanliness. It leaves a less than smooth surface on the bearing cup.

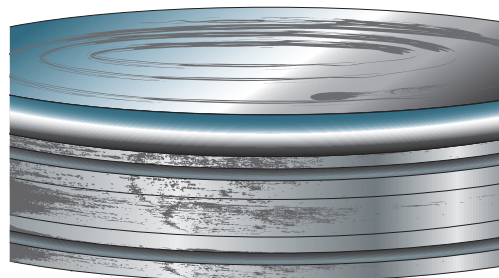


Figure 13-67. Bearing cups should be tight in the wheel boss and should never rotate. The outside of a bearing cup that was spinning while installed in the wheel is shown.

Proper lubrication is a partial deterrent to negative environmental impacts on a bearing. Use the lubricant recommended by the manufacturer. Use of a pressure bearing packing tool or adapter is also recommended as the best method to remove any contaminants from inside the bearing that may have remained after cleaning. (Figure 13-68)

To lubricate a tapered roller bearing without the use of a bearing lubrication tool, place a small amount of the approved grease on the palm of the hand. Grasp the bearing with the other hands and press the larger diameter side of the bearing into the grease to force it completely through the space between the bearing rollers and the cone. Gradually turn the bearing so that all of the rollers have been completely packed with grease. (Figure 13-69)

AIRCRAFT BRAKES

Very early aircraft have no brake system to slow and stop the aircraft while it is on the ground. Instead, they rely on slow speeds, soft airfield surfaces, and the friction developed by the tail skid to reduce speed during ground operation. Brake systems designed for aircraft became common after World War I as the speed and complexity

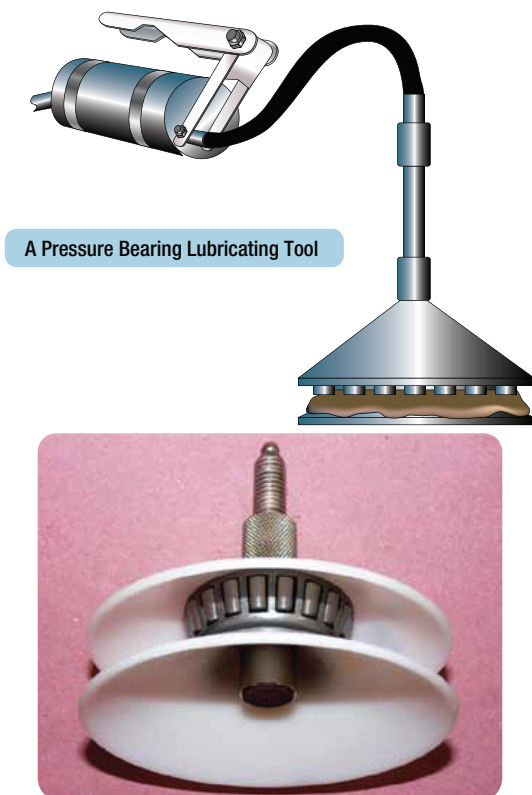


Figure 13-68. A pressure bearing lubricating tool.



Figure 13-69. Packing grease into a clean, dry bearing can be done by hand in the absence of a bearing grease tool. Press the bearing into the grease on the palm of the hand until it passes completely through the gap between the rollers and the inner race all the way around the bearing.

of aircraft increased and the use of smooth, paved runway surfaces proliferated. All modern aircraft are equipped with brakes. Their proper functioning is relied upon for safe operation of the aircraft on the ground. The brakes slow the aircraft and stop it in a reasonable amount of time. They hold the aircraft stationary during taxi.

On most aircraft, each of the main wheels is equipped with a brake unit. The nose wheel or tail wheel does not have a brake. In the typical brake system, mechanical and/or hydraulic linkages to the rudder pedals allow the pilot to control the brakes. Pushing on the top of the right rudder pedal activates the brake on the right main wheel(s) and pushing on the top of the left rudder pedal operates the brake on the left main wheel(s). The basic operation of brakes involves converting the kinetic energy of motion into heat energy through the creation of friction. A great amount of heat is developed and forces on the brake system components are demanding. Proper adjustment, inspection, and maintenance of the brakes is essential for effective operation.

TYPES AND CONSTRUCTION OF AIRCRAFT BRAKES

Modern aircraft typically use disc brakes. The disc rotates with the turning wheel assembly while a stationary caliper resists the rotation by causing friction against the disc when the brakes are applied. The size, weight, and landing speed of the aircraft influence the design and complexity of the disc brake system. Single, dual, and multiple disc brakes are common types of brakes. Segmented rotor brakes are used on large aircraft. Expander tube brakes are found on older large aircraft. The use of carbon discs is increasing in the modern aviation fleet.

SINGLE DISC BRAKES

Small, light aircraft typically achieve effective braking using a single disc keyed or bolted to each wheel. As the wheel turns, so does the disc.

Braking is accomplished by applying friction to both sides of the disc from a non-rotating caliper bolted to the landing gear axle flange. Pistons in the caliper housing under hydraulic pressure force wearable brake pads or linings against the disc when the brakes are applied. Hydraulic master cylinders connected to the rudder pedals supply the pressure when the upper halves of the rudder pedals are pressed.

FLOATING DISC BRAKES

A floating disk brake is illustrated in *Figure 13-70*. A more detailed, exploded view of this type of brake is shown in *Figure 13-71*. The caliper straddles the disc. It has three cylinders bored through the housing, but on other brakes this number may vary. Each cylinder accepts an actuating piston assembly comprised mainly of a piston, a return spring, and an automatic adjusting pin. Each brake assembly has six brake linings or pucks. Three are located on the ends of the pistons, which are in the outboard side of the caliper. They are designed to move in and out with the pistons and apply pressure to the outboard side of the disc. Three more linings are located opposite of these pucks on the inboard side of the caliper. These linings are stationary.



Figure 13-70. A single disc brake is a floating-disc, fixed caliper brake.

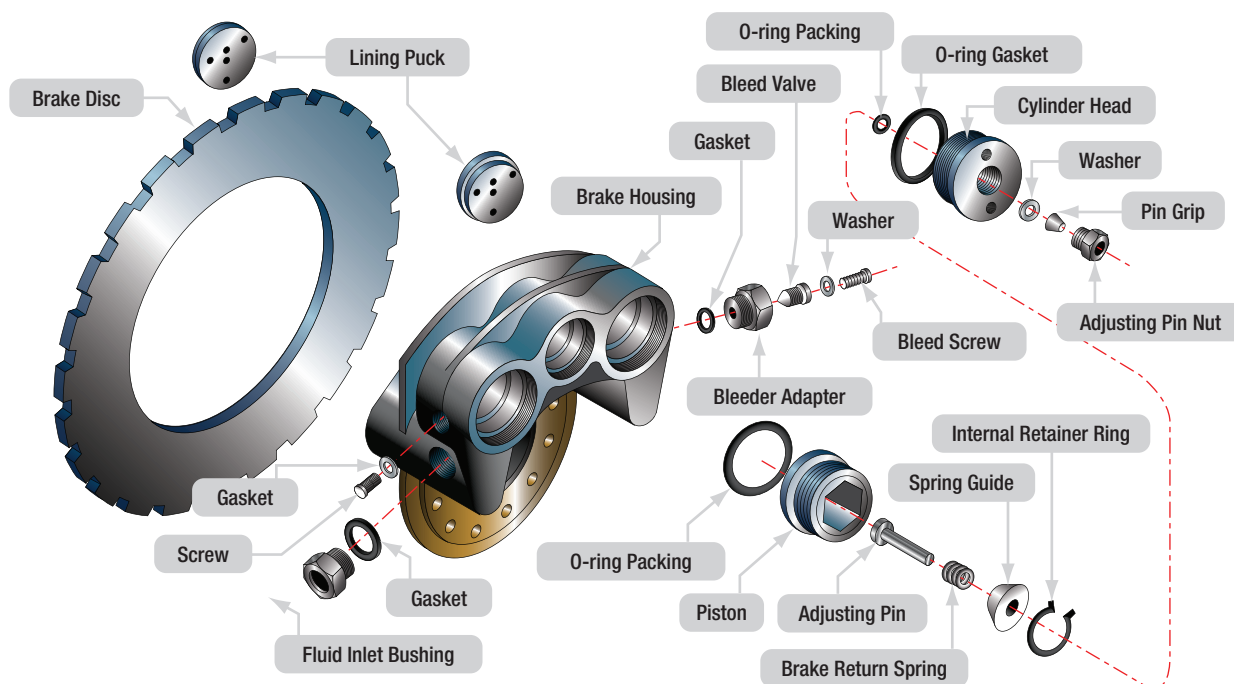


Figure 13-71. An exploded view of a single-disc brake assembly found on a light aircraft.

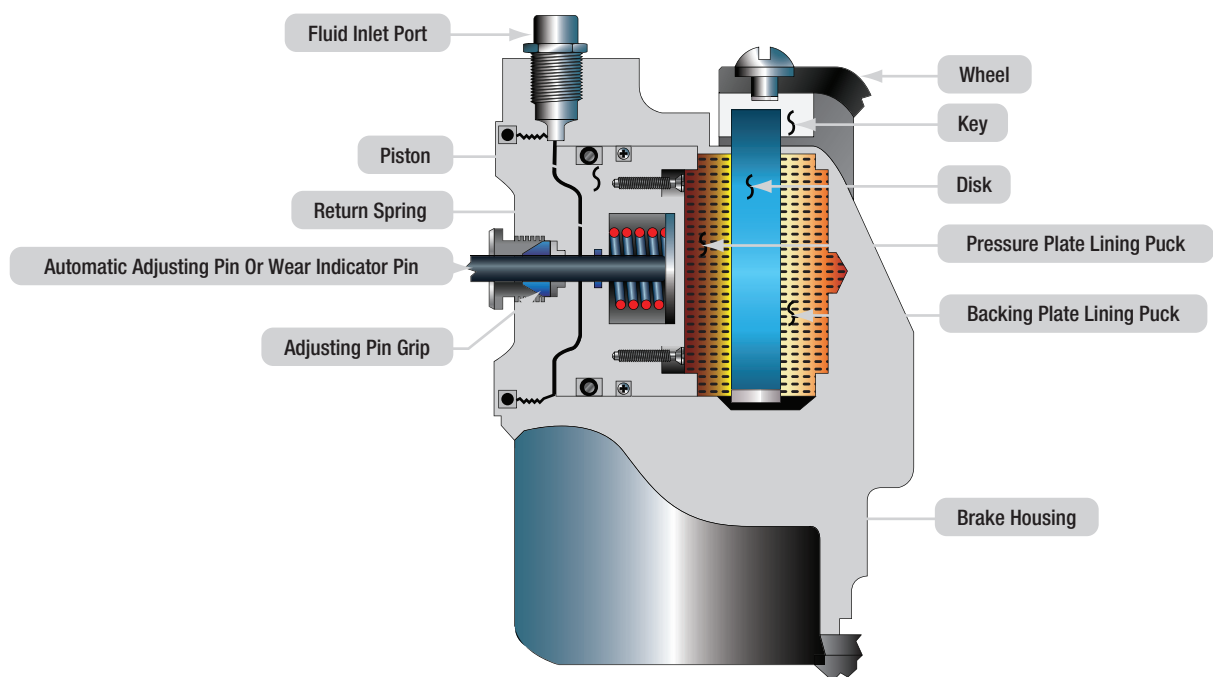


Figure 13-72. A cross-sectional view of a Goodyear single-disc brake caliper illustrates the adjusting pin assembly that doubles as a wear indicator.

The brake disc is keyed to the wheel. It is free to move laterally in the key slots. This is known as a floating disk. When the brakes are applied, the pistons move out from the outboard cylinders and their pucks contact the disc. The disc slides slightly in the key slots until the inboard stationary pucks also contact the disc. The result is a fairly even amount of friction applied to each side of the disc and thus, the rotating motion is slowed.

When brake pressure is released, the return spring in each piston assembly forces the piston back away from the disc. The spring provides a preset clearance between each puck and the disc. The self adjusting feature of the brake maintains the same clearance, regardless of the amount of wear on the brake pucks. The adjusting pin on the back of each piston moves with the piston through a frictional pin grip.

When brake pressure is relieved, the force of the return spring is sufficient to move the piston back away from the brake disc, but not enough to move the adjusting pin held by the friction of the pin grip. The piston stops when it contacts the head of the adjusting pin. Thus, regardless of the amount of wear, the same travel of the piston is required to apply the brake. The stem of the pin protruding through the cylinder head serves as a wear indicator. The manufacturer's maintenance information states the minimum length of the pin that needs to be protruding for the brakes to be considered airworthy. (*Figure 13-72*)

The brake caliper has the necessary passages machined into it to facilitate hydraulic fluid movement and the application of pressure when the brakes are utilized. The caliper housing also contains a bleed port used by the technician to remove unwanted air from the system. Brake bleeding, as it is known, should be done in accordance with the manufacturer's maintenance instructions.

FIXED DISC BRAKES

Even pressure must be applied to both sides of the brake disc to generate the required friction and obtain consistent wear properties from the brake linings. The floating disc accomplishes this as described above. It can also be accomplished by bolting the disc rigidly to the wheel and allowing the brake caliper and linings to float laterally when pressure is applied. This is the design of a common fixed disc brake used on light aircraft. The brake is manufactured by the Cleveland Brake Company and is shown in *Figure 13-73*. An exploded detail view of the same type of brake is shown in *Figure 13-74*.

The fixed-disk, floating-caliper design allows the brake caliper and linings to adjust position in relationship to the disc. Linings are riveted to the pressure plate and backplate. Two anchor bolts that pass through the pressure plate are secured to the cylinder assembly.

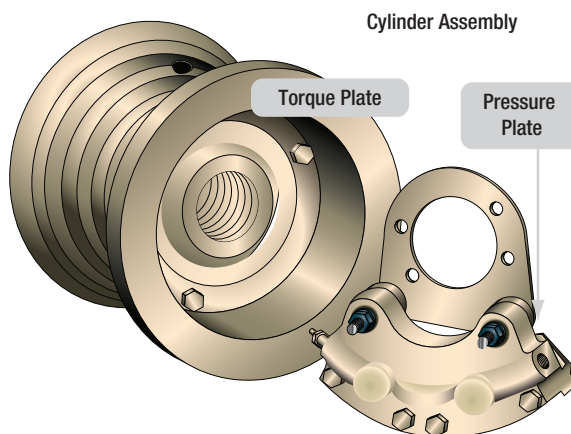
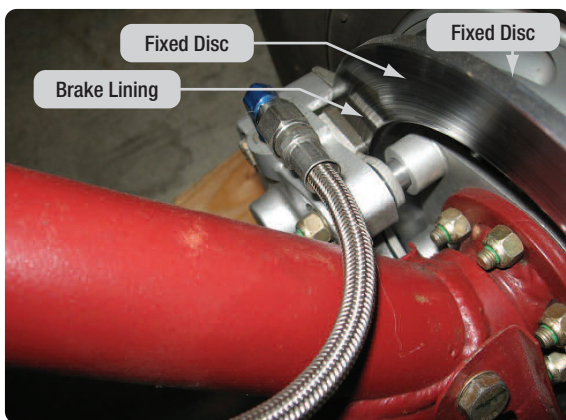


Figure 13-73. A Cleveland brake on a light aircraft is a fixed-disc brake. It allows the brake caliper to move laterally on anchor bolts to deliver even pressure to each side of the brake disc.

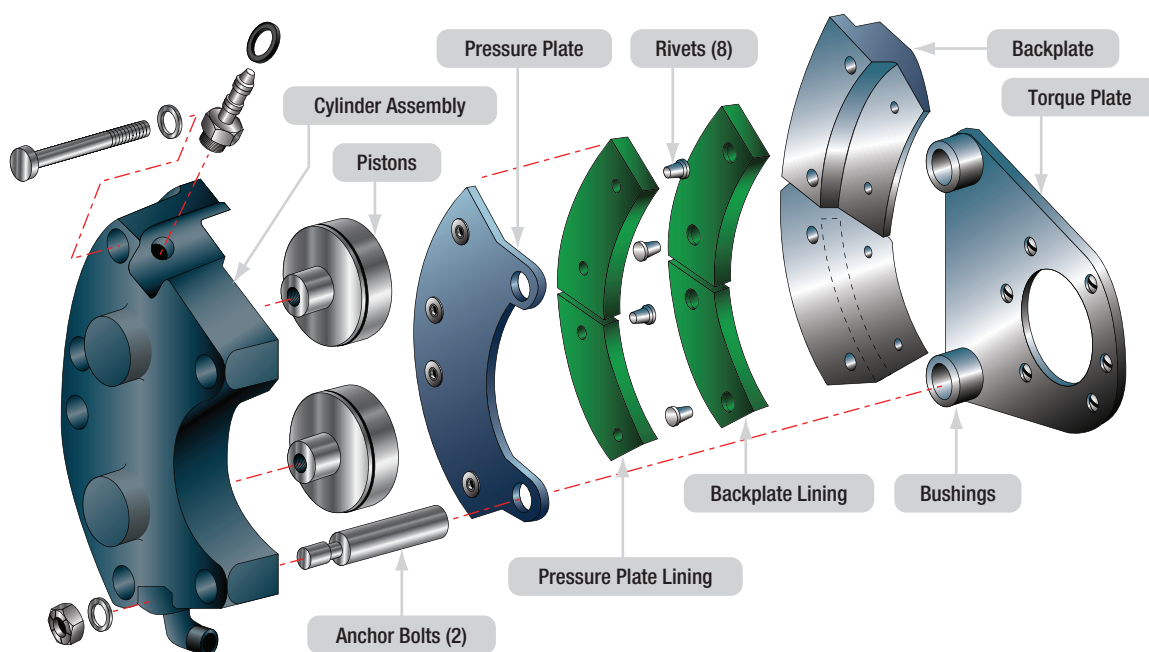


Figure 13-74. An exploded view of a dual-piston Cleveland brake assembly.

The other ends of the bolts are free to slide in and out of bushings in the torque plate, which is bolted to the axle flange. The cylinder assembly is bolted to the backplate to secure the assembly around the disc. When pressure is applied, the caliper and linings center on the disc via the sliding action of the anchor bolts in the torque plate bushings. This provides equal pressure to both sides of the disc to slow its rotation. A unique feature of the Cleveland brake is that the linings can be replaced without removing the wheel. Unbolting the cylinder assembly from the backplate allows the anchor bolts to slide out of the torque plate bushings. The entire caliper assembly is then free and provides access to all of the components.

Maintenance requirements on all single disc brake systems are similar to those on brake systems of any type. Regular inspection for any damage and for wear on the linings and discs is required. Replacement of parts worn beyond limits is always followed by an operational check. The check is performed while taxiing the aircraft. The braking action for each main wheel should be equal with equal application of pedal pressure. Pedals should be firm, not soft or spongy, when applied. When pedal pressure is released, the brakes should release without any evidence of drag.

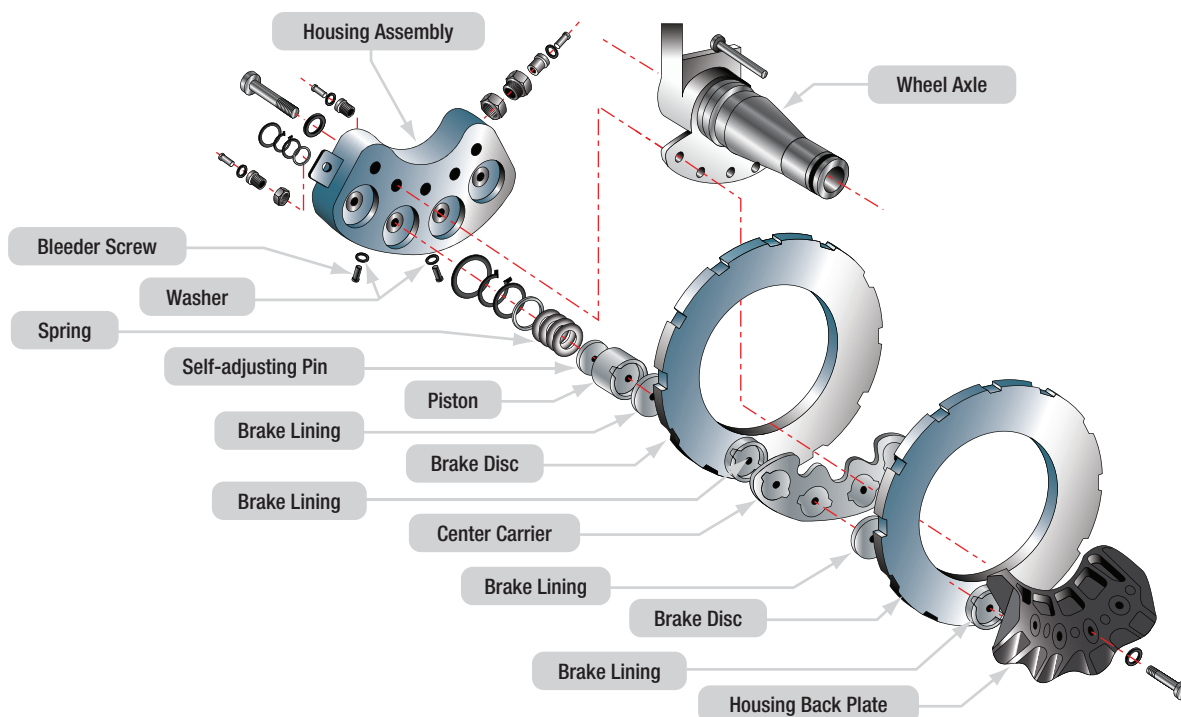


Figure 13-75. A dual-disc brake is similar to a single-disc brake. It uses a center carrier to hold brake linings against each of the discs.

DUAL DISC BRAKES

Dual disc brakes are used on aircraft where a single disc on each wheel does not supply sufficient braking friction. Two discs are keyed to the wheel instead of one. A center carrier is located between the two discs. It contains linings on each side that contact each of the discs when the brakes are applied. The caliper mounting bolts are long and mount through the center carrier, as well as the backplate which bolts to the housing assembly. (Figure 13-75)

MULTIPLE DISC BRAKES

Large, heavy aircraft require the use of multiple disc brakes. Multiple disc brakes are heavy duty brakes designed for use with power brake control valves or power boost master cylinders, which is discussed later in this chapter. The brake assembly consists of an extended bearing carrier similar to a torque tube type unit that bolts to the axle flange. It supports the various brake parts, including an annular cylinder and piston, a series of steel discs alternating with copper or bronze plated discs, a backplate, and a backplate retainer. The steel stators are keyed to the bearing carrier, and the copper or bronze plated rotors are keyed to the rotating wheel. Hydraulic pressure applied to the piston causes the entire stack of stators and rotors to be compressed. This creates enormous friction and heat and slows the rotation of the wheel. (Figure 13-76)

As with the single and dual disc brakes, retracting springs return the piston into the housing chamber of the bearing carrier when hydraulic pressure is relieved. The hydraulic fluid exits the brake to the return line through an automatic adjuster. The adjuster traps a predetermined amount of fluid in the brakes that is just sufficient to provide the correct clearances between the rotors and stators. (Figure 13-77)

Brake wear is typically measured with a wear gauge that is not part of the brake assembly. These types of brake are typically found on older transport category aircraft. The rotors and stators are relatively thin, only about $\frac{1}{8}$ -inch thick. They do not dissipate heat very well and have a tendency to warp.

SEGMENTED ROTOR DISC BRAKES

The large amount of heat generated while slowing the rotation of the wheels on large and high performance aircraft is problematic. To better dissipate this heat, segmented rotor disc brakes have been developed. Segmented rotor disc brakes are multiple disc brakes but of more modern design than the type discussed earlier. There are many variations. Most feature numerous elements that aid in the control and dissipation of heat. Segmented rotor disc brakes are heavy-duty brakes especially adapted for use with the high pressure hydraulic systems of power brake systems. Braking is

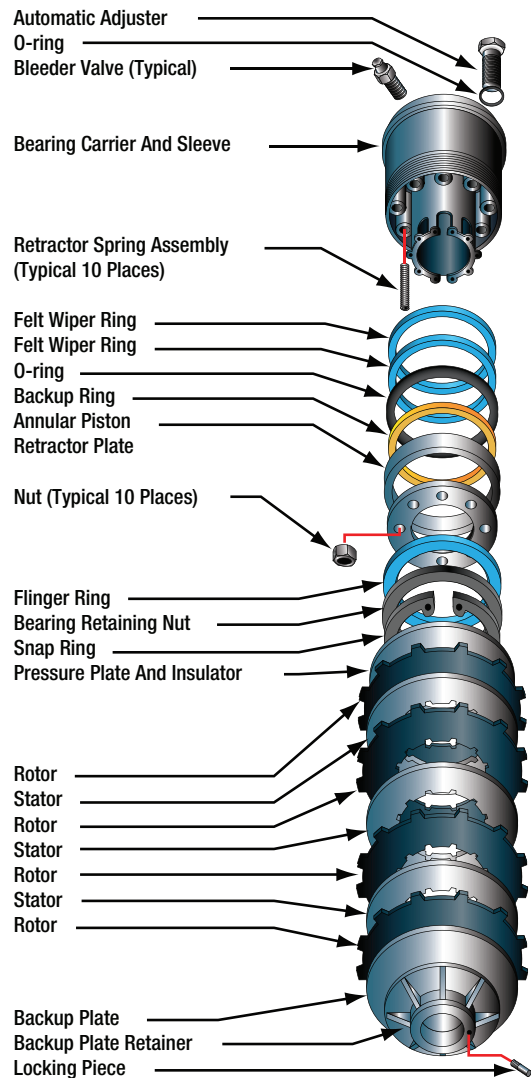


Figure 13-76. A multiple disc brake with bearing carrier upon which the parts of the brake are assembled including an annular cylinder and piston assembly that apply pressure evenly to a stack of rotors and stators.

accomplished by means of several sets of stationary, high friction type brake linings that make contact with rotating segments. The rotors are constructed with slots or in sections with space between them, which helps dissipate heat and give the brake its name. Segmented rotor multiple disc brakes are the standard brake used on high performance and air carrier aircraft. An exploded view of one type of segmented rotor brake assembly is shown in *Figure 13-78*.

The description of a segmented rotor brake is very similar to the multiple disc type brake previously described. The brake assembly consists of a carrier, a piston and piston cup seal, a pressure plate, an auxiliary stator plate, rotor segments, stator plates, automatic adjusters, and a backing plate.

The carrier assembly, or brake housing with torque tube, is the basic unit of the segmented rotor brake. It is the part that attaches to the landing gear shock strut flange upon which the other components of the brake are assembled. On some brakes, two grooves or cylinders are machined into the carrier to receive the piston cups and pistons. (*Figure 13-78*) Most segmented rotor disc brakes have numerous individual cylinders machined into the brake housing into which fit the same number of actuating pistons. Often, these cylinders are supplied by two different hydraulic sources, alternating every other cylinder from a single source. If one source fails, the brake still operates sufficiently on the other. (*Figure 13-79*) External fittings in the carrier or brake housing admit the hydraulic fluid. A bleed port can also be found.

A pressure plate is a flat, circular, high-strength steel, non-rotating plate notched on the inside circumference to fit over the stator drive sleeves or torque tube spines. The brake actuating pistons contact the pressure plate. Typically, an insulator is used between the piston head and the pressure plate to impede heat conduction from the brake discs. The pressure plate transfers the motion of the pistons to the stack of rotors and stators that compress to slow the rotation of the wheels. On most designs, brake lining material attached directly to the pressure plate contacts the first rotor in the stack to transfer the motion of the piston(s). (*Figure 13-78*) An auxiliary stator plate with brake lining material on the side opposite the pressure plate can also be used.

Any number of alternating rotors and stators are sandwiched under hydraulic pressure against the backing plate of the brake assembly when the brakes are applied. The backing plate is a heavy steel plate bolted to the housing or torque tube at a fixed dimension from the carrier housing. In most cases, it has brake lining material attached to it and contacts the last rotor in the stack. (*Figure 13-78*) Stators are flat plates notched on the internal circumference to be held stationary by the torque tube spines. They have wearable brake lining material riveted or adhered to each side to make contact with adjacent rotors. The liner is typically constructed of numerous isolated blocks. (*Figure 13-78*) The space between the liner blocks aids in the dissipation of heat. The composition of the lining materials vary. Steel is often used.

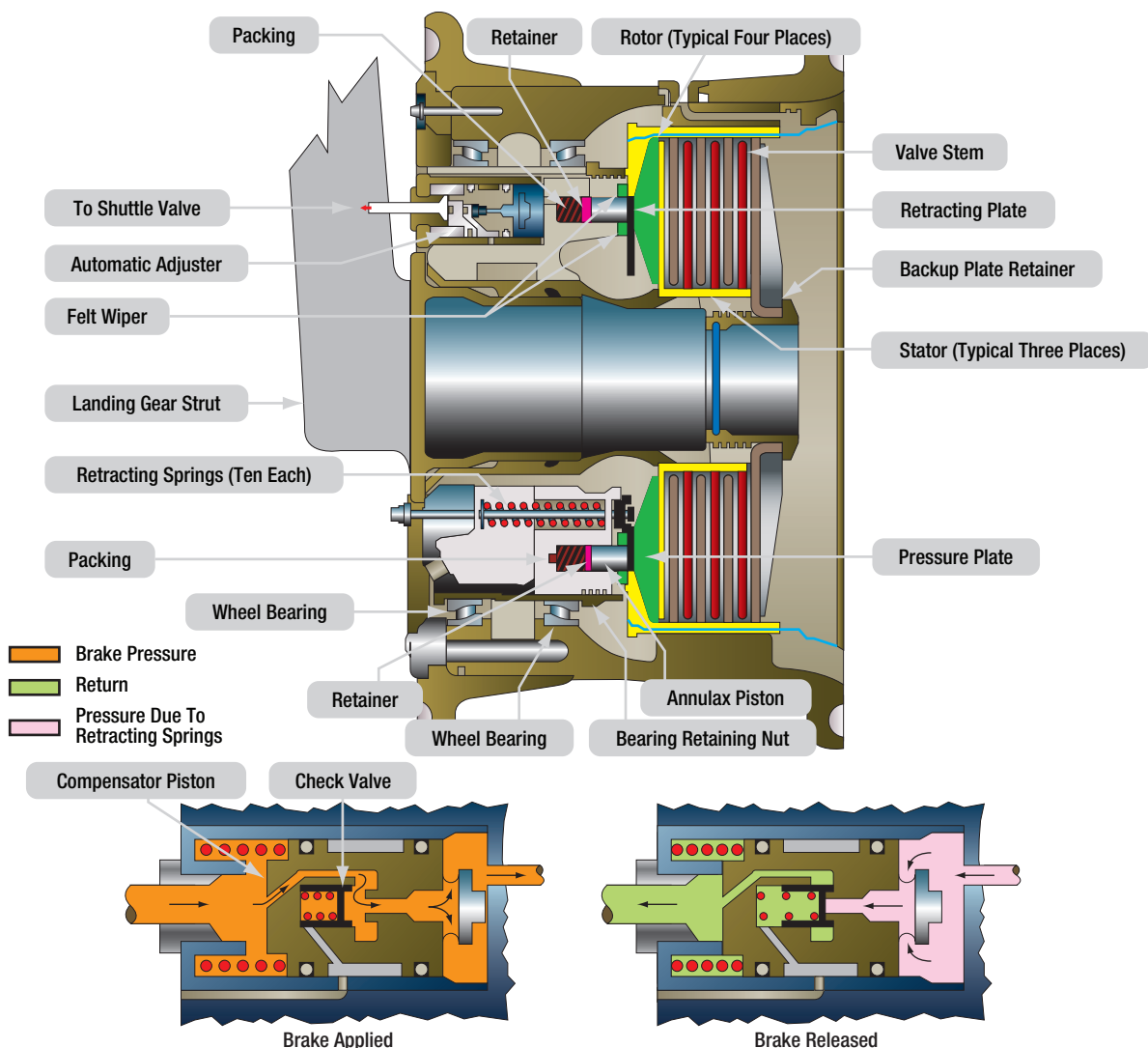


Figure 13-77. A multiple-disc brake with details of the automatic adjuster.

Rotors are slit or segmented discs that have notches or tangs in the external circumference that key to the rotating wheel. Slots or spaces between sections of the rotor create segments that allow heat to dissipate faster than it would if the rotor was solid. They also allow for expansion and prevent warping. (*Figure 13-78*)

Rotors are usually steel to which a frictional surface is bonded to both sides. Typically, sintered metal is used in creating the rotor contact surface. Segmented multiple disc brakes use retraction spring assemblies with auto clearance adjusters to pull the backplate away from the rotor and stator stack when brake pressure is removed. This provides clearance so the wheel can turn unimpeded by contact friction between the brake parts, but keeps the units in close proximity for rapid contact and braking when the brakes are applied. The number of retraction devices varies with brake design.

Figure 13-80 illustrates a brake assembly used on a Boeing 737 transport category aircraft. In the cutaway view, the number and locations of the auto adjustment retraction mechanisms can be seen. Details of the mechanisms are also shown. Instead of using a pin grip assembly for auto adjustment, an adjuster pin, ball, and tube operate in the same manner. They move out when brake pressure is applied, but the ball in the tube limits the amount of the return to that equal to the brake lining wear. Two independent wear indicators are used on the brake illustrated. An indicator pin attached to the backplate protrudes through the carrier. The amount that it protrudes with the brakes applied is measured to ascertain if new linings are required.

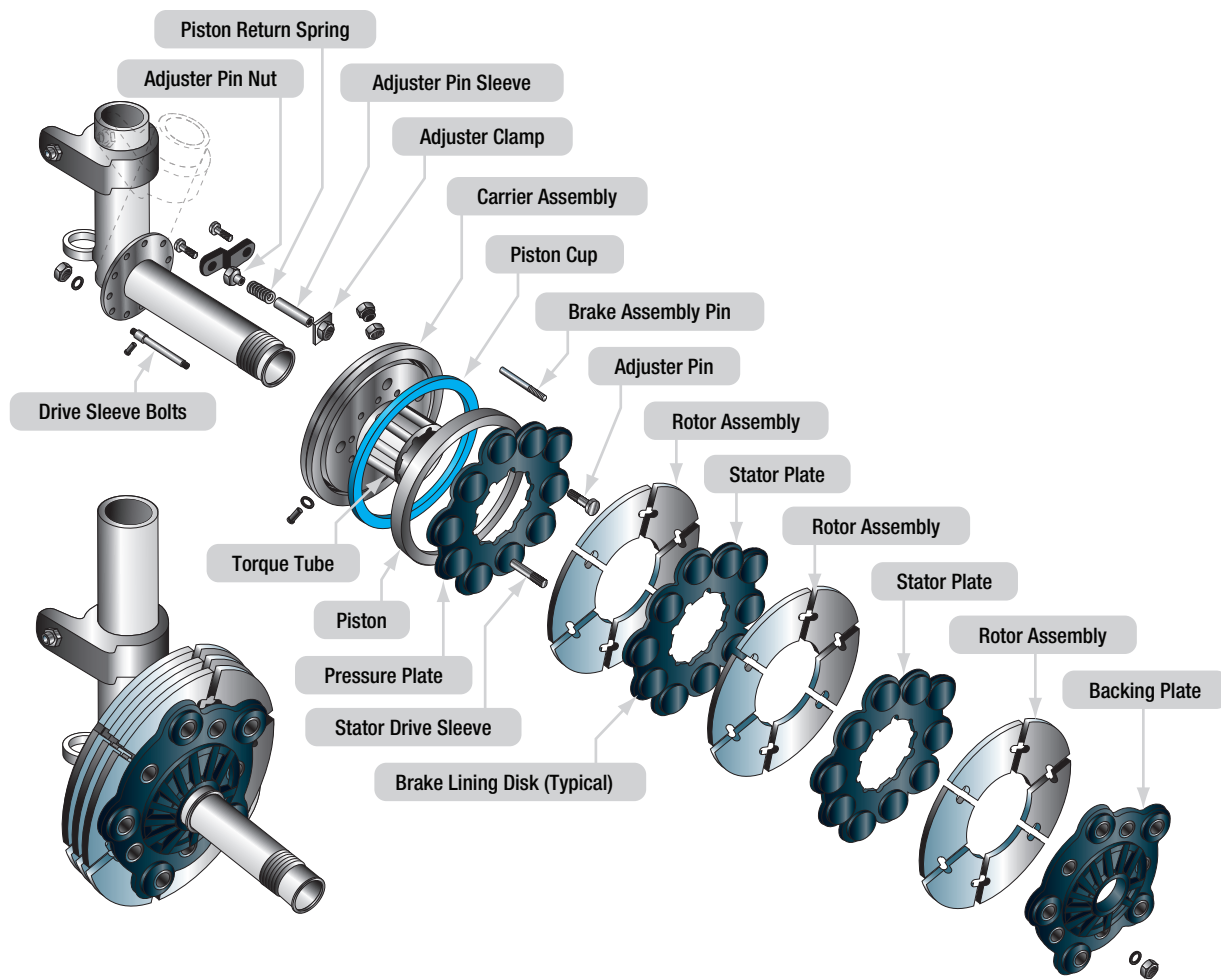


Figure 13-78. Exploded and detail views of segmented rotor brakes.

NOTE: The other segmented multiple disc brakes may use slightly different techniques for pressure plate retraction and wear indication. Consult the manufacturer's maintenance information to ensure wear indicators are read correctly.

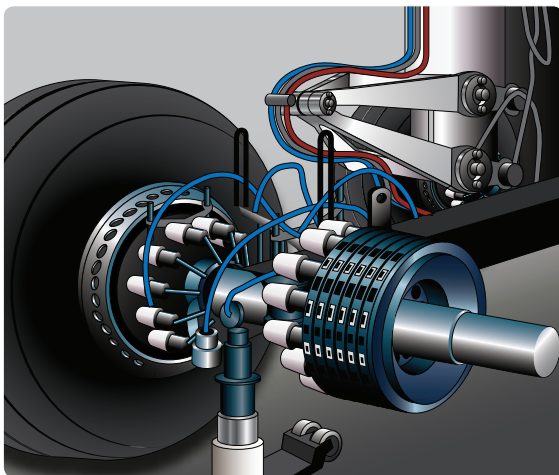


Figure 13-79. Many modern segmented rotor disc brakes use a housing machined to fit numerous individual actuating pistons.

CARBON BRAKES

The segmented multiple disc brake has given many years of reliable service to the aviation industry. It has evolved through time in an effort to make it lightweight and to dissipate the frictional heat of braking in a quick, safe manner. The latest iteration of the multiple disc brake is the carbon disc brake. It is currently found on high performance and air carrier aircraft.

Carbon brakes are so named because carbon fiber materials are used to construct the brake rotors. (*Figure 13-81*) Carbon brakes are approximately forty percent lighter than conventional brakes. On a large transport category aircraft, this alone can save several hundred pounds in aircraft weight. The carbon fiber discs are noticeably thicker than sintered steel rotors but are extremely light.

They are able to withstand temperatures fifty percent higher than steel component brakes. The maximum designed operating temperature is limited by the ability of

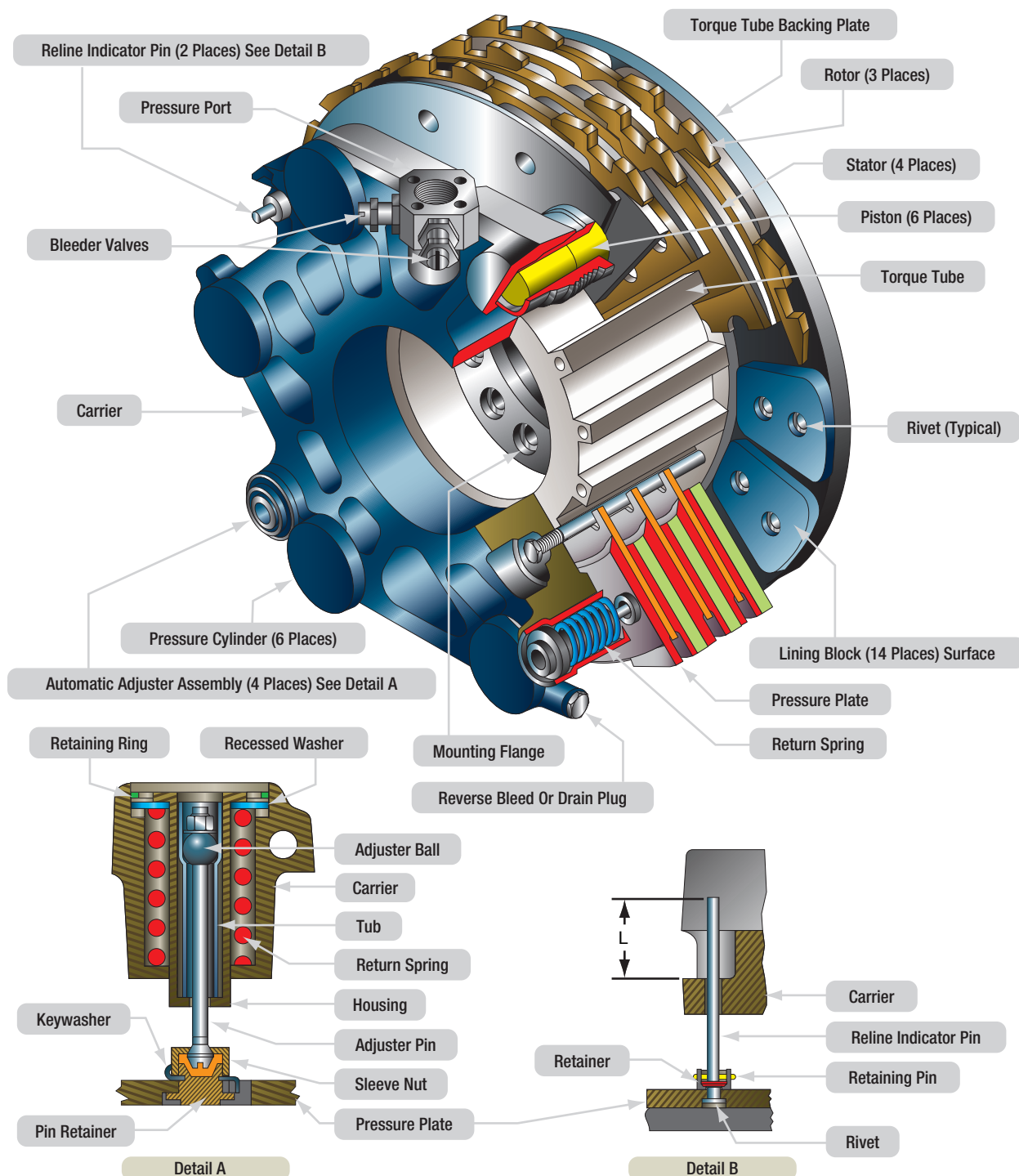


Figure 13-80. The multiple-disk brake assembly and details from a Boeing 737.

adjacent components to withstand the high temperature. Carbon brakes have been shown to withstand two to three times the heat of a steel brake in non-aircraft applications. Carbon rotors also dissipate heat faster than steel rotors. A carbon rotor maintains its strength and dimensions at high temperatures. Moreover, carbon brakes last twenty to fifty percent longer than steel brakes, which results in reduced maintenance.

The only impediment to carbon brakes being used on all aircraft is the high cost of manufacturing. The price is expected to lower as technology improves and greater numbers of aircraft operators enter the market.

EXPANDER TUBE BRAKES

An expander tube brake is a different approach to braking that is used on aircraft of all sizes produced

in the 1930s-1950s. It is a lightweight, low pressure brake bolted to the axle flange that fits inside an iron brake drum. A flat, fabric-reinforced neoprene tube is fitted around the circumference of a wheel like torque flange. The exposed flat surface of the expander tube is lined with brake blocks similar to brake lining material. Two flat frames bolt to the sides of the torque flange. Tabs on the frames contain the tube and allow evenly spaced torque bars to be bolted in place across the tube between each brake block. These prevent circumferential movement of the tube on the flange. (*Figure 13-82*)

The expander tube is fitted with a metal nozzle on the inner surface. Hydraulic fluid under pressure is directed through this fitting into the inside of the tube when the brakes are applied. The tube expands outward, and the brake blocks make contact with the wheel drum causing friction that slows the wheel. As hydraulic pressure is increased, greater friction develops. Semi-elliptical springs located under the torque bars return the expander tube to a flat position around the flange when hydraulic pressure is removed. The clearance between the expander tube and the brake drum is adjustable by rotating an adjuster on some expander tube brakes. Consult the manufacturer's maintenance manual for the correct clearance setting.

Figure 13-83 gives an exploded view of an expander tube brake, detailing its components. Expander tube brakes work well but have some drawbacks. They tend to take a setback when cold. They also have a tendency to swell with temperature and leak. They may drag inside the drum if this occurs. Eventually, expander brakes were abandoned in favor of disc brake systems.

BRAKE ACTUATING SYSTEMS

The various brake assemblies, described in the previous section, all use hydraulic power to operate. Different means of delivering the required hydraulic fluid pressure to brake assemblies are discussed in this section. There are three basic actuating systems:

1. An independent system not part of the aircraft main hydraulic system.
2. A booster system that uses the aircraft hydraulic system intermittently when needed.
3. A power brake system that only uses the aircraft main hydraulic system(s) as a source of pressure.

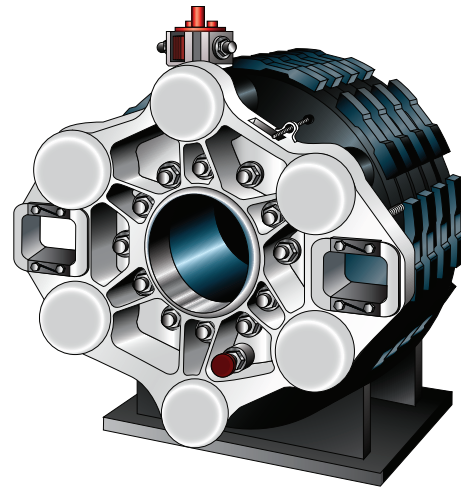


Figure 13-81. A carbon brake for a Boeing 737.

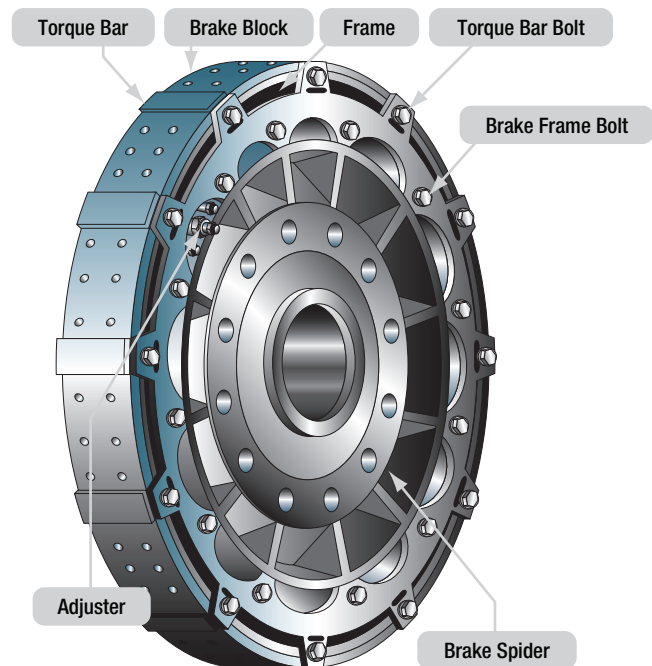


Figure 13-82. An expander tube brake assembly.

Systems on different aircraft vary, but the general operation is similar to those described.

INDEPENDENT MASTER CYLINDERS

In general, small, light aircraft and aircraft without hydraulic systems use independent braking systems. An independent brake system is not connected in any way to the aircraft hydraulic system. Master cylinders are used to develop the necessary hydraulic pressure to operate the brakes. This is similar to the brake system of an automobile. In most brake actuating systems, the pilot pushes on the tops of the rudder pedals to apply the brakes. A master cylinder for each brake is mechanically

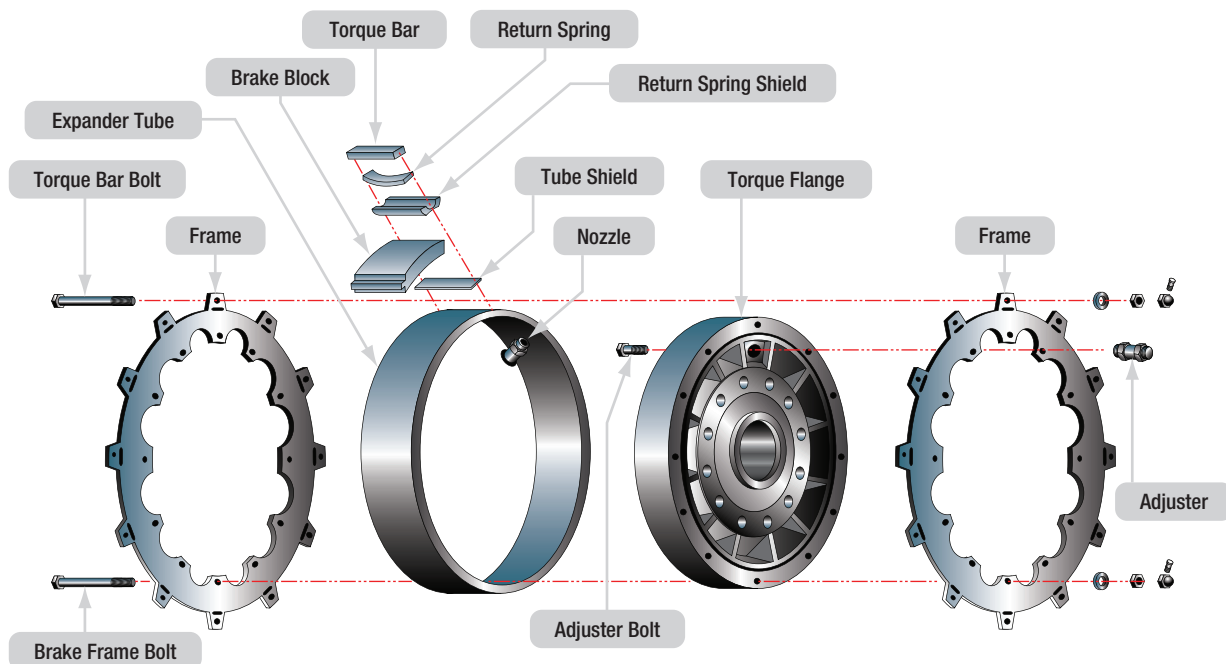


Figure 13-83. An exploded view of an expander tube brake.

connected to the corresponding rudder pedal (i.e., right main brake to the right rudder pedal, left main brake to the left rudder pedal). (*Figure 13-84*)

When the pedal is depressed, a piston inside a sealed fluid-filled chamber in the master cylinder forces hydraulic fluid through a line to the piston(s) in the brake assembly. The brake piston(s) push the brake linings against the brake rotor to create the friction that slows the wheel rotation. Pressure is increased throughout the entire brake systems and against the rotor as the pedal is pushed harder.

Many master cylinders have built-in reservoirs for the brake hydraulic fluid. Others have a single remote reservoir that services both of the aircraft's two master cylinders. (*Figure 13-85*) A few light aircraft with nose wheel steering have only one master cylinder that actuates both main wheel brakes. This is possible because steering the aircraft during taxi does not require differential braking. Regardless of the set-up, it is the master cylinder that builds up the pressure required for braking.

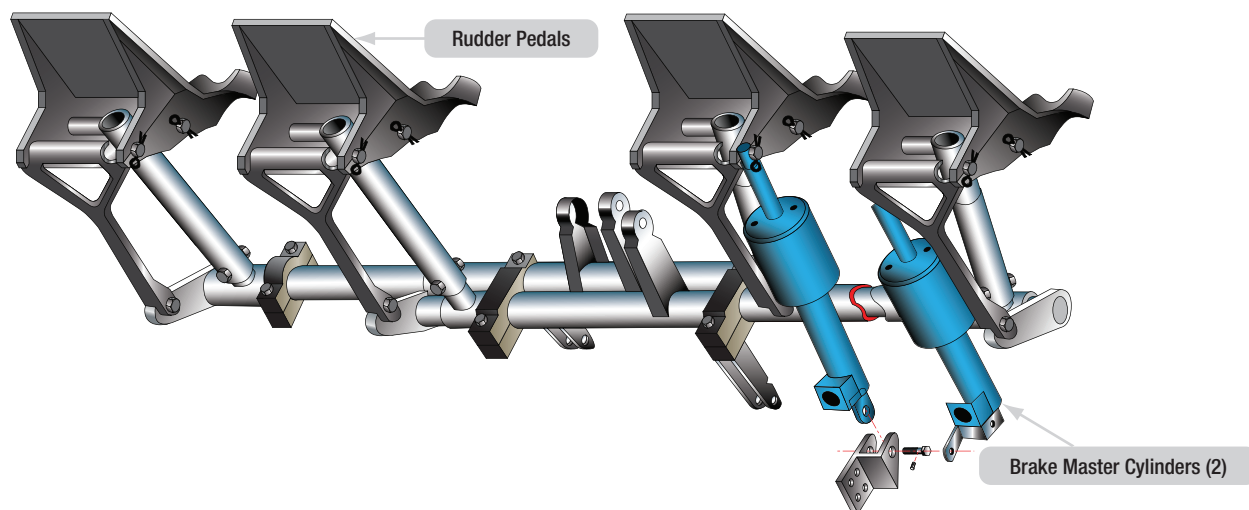


Figure 13-84. Master cylinders on an independent brake system are directly connected to the rudder pedals or are connected through mechanical linkage.

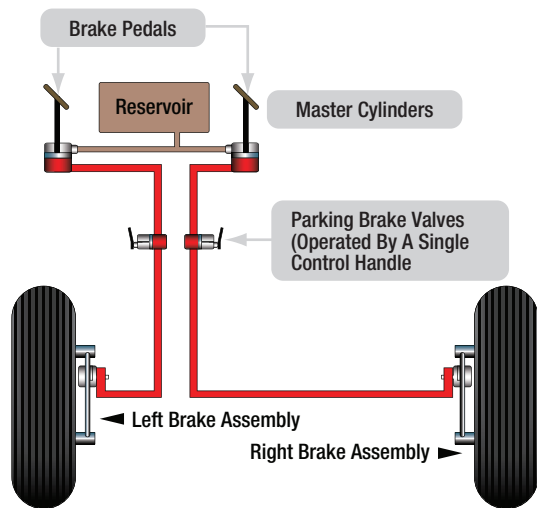


Figure 13-85. A remote reservoir services both master cylinders on some independent braking systems.

A master cylinder used with a remote reservoir is illustrated in **Figure 13-86**. This particular model is a Goodyear master cylinder. The cylinder is always filled with air-free, contaminant-free hydraulic fluid as is the reservoir and the line that connects the two together. When the top of the rudder pedal is depressed, the piston arm is mechanically moved forward into the master cylinder. It pushes the piston against the fluid, which is forced through the line to the brake. When pedal pressure is released, the return springs in the brake assembly retract the brake pistons back into the brake housing. The hydraulic fluid behind the pistons is displaced and must return to the master cylinder. As it does, a return spring in the master cylinder move the piston, piston rod and rudder pedal back to the original position (brake off, pedal not depressed). The fluid behind the master cylinder piston flows back into the reservoir. The brake is ready to be applied again. The

forward side of the piston head contains a seal that closes off the compensating port when the brakes are applied so that pressure can build. The seal is only effective in the forward direction. When the piston is returning, or is fully retracted to the off position, fluid behind the piston is free to flow through piston head ports to replenish any fluid that may be lost downstream of the master cylinder. The aft end of the master cylinder contains a seal that prevents leakage at all times. A rubber boot fits over the piston rod and the aft end of the master cylinder to keep out dust.

A parking brake for this remote reservoir master cylinder brake system is a ratcheting mechanical device between the master cylinder and the rudder pedals. With the brakes applied, the ratchet is engaged by pulling the parking brake handle. To release the brakes, the rudder pedals are depressed further allowing the ratchet to disengage. With the parking brake set, any expansion of hydraulic fluid due to temperature is relieved by a spring in the mechanical linkage.

A common requirement of all braking systems is for there to be no air mixed in with the hydraulic fluid. Since air is compressible and hydraulic fluid essentially is not, any air under pressure when the brakes are applied causes spongy brakes. The pedals do not feel firm when pushed down due to the air compressing. Brake systems must be bled to remove all air from the system. Instructions for bleeding the brakes are in the manufacturer's maintenance information. Brake systems equipped with Goodyear master cylinders must be bled from the top down to ensure any air trapped behind the master cylinder piston is removed.

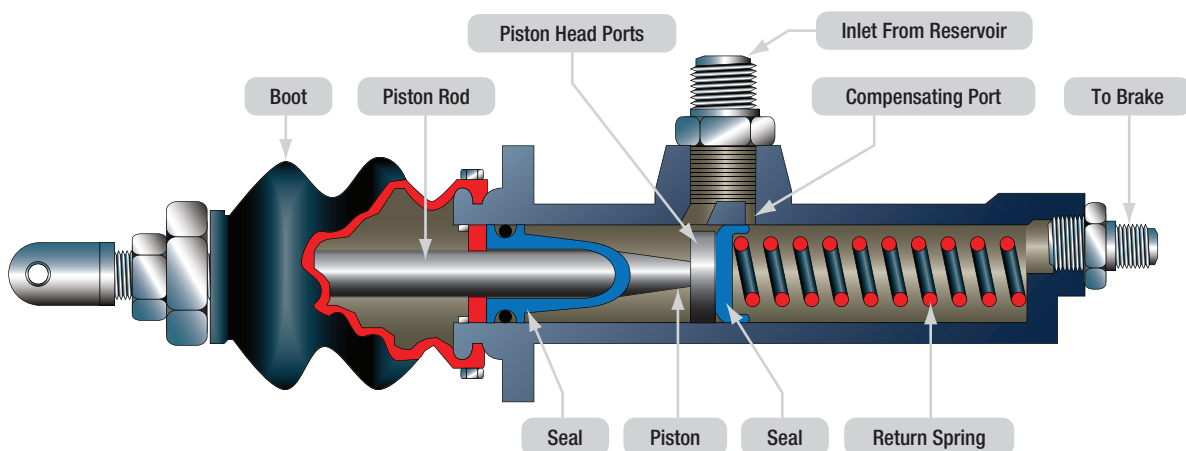


Figure 13-86. A Goodyear brake master cylinder from an independent braking system with a remote reservoir.

An alternative common arrangement of independent braking systems incorporates two master cylinders, each with its own integral fluid reservoir. Except for the reservoir location, the brake system is basically the same as just described. The master cylinders are mechanically linked to the rudder pedals as before. Depressing the top of a pedal causes the piston rod to push the piston into the cylinder forcing the fluid out to the brake assembly. The piston rod rides in a compensator sleeve and contains an O-ring that seals the rod to the piston when the rod is moved forward. This blocks the compensating ports. When released, a spring returns the piston to its original position which refills the reservoir as it returns. The rod end seal retracts away from the piston head allowing a free flow of fluid from the cylinder through the compensating ports in the piston to the reservoir. (Figure 13-87)

The parking brake mechanism is a ratcheting type that operates as described. A servicing port is supplied at the top of the master cylinder reservoir. Typically, a vented plug is installed in the port to provide positive pressure on the fluid.

BOOSTED BRAKES

In an independent braking system, the pressure applied to the brakes is only as great as the foot pressure applied to the top of the rudder pedal. Boosted brake actuating systems augment the force developed by the pilot with hydraulic system pressure when needed. The boost is only during heavy braking. It results in greater pressure applied to the brakes than the pilot alone can provide. Boosted brakes are used on medium and larger aircraft that do not require a full power brake actuating system. A boosted brake master cylinder for each brake is mechanically attached to the rudder pedals. The boosted brake master cylinder operates differently. (Figure 13-88)

When the brakes are applied, the pressure from the pilot's foot through the mechanical linkage moves the master cylinder piston in the direction to force fluid to the brakes. The initial movement closes the compensator poppet used to provide thermal expansion relief when the brakes are not applied. As the pilot pushes harder on the pedal, a spring loaded toggle moves a spool valve in the cylinder. Aircraft hydraulic system pressure flows through the valve to the back side of the piston. Pressure is increased, as is the force developed to apply the brakes.

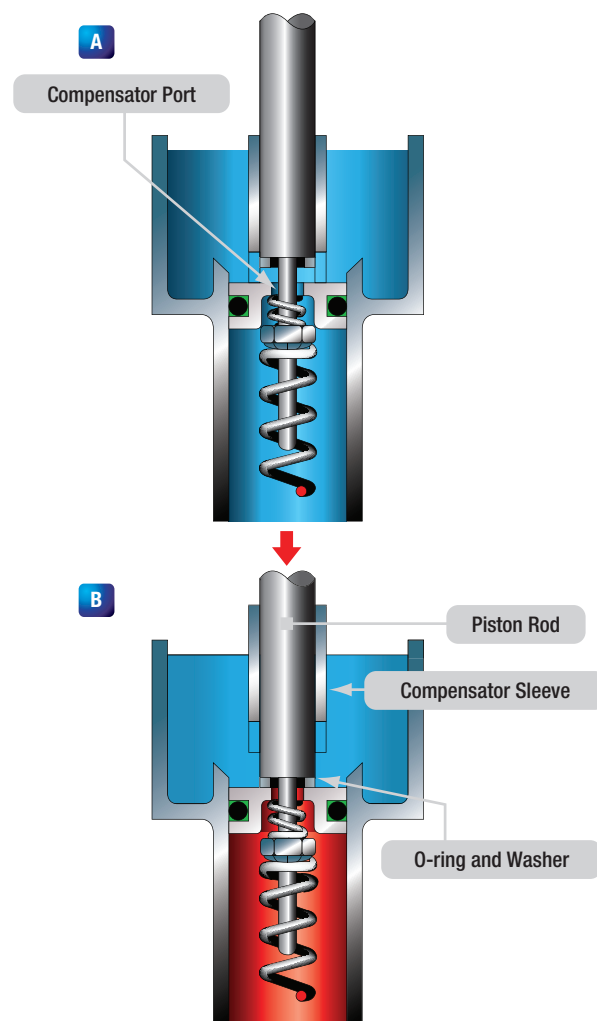


Figure 13-87. A common master cylinder with built-in reservoir. Illustration A depicts the master cylinder when the brakes are off. The compensating port is open to allow fluid to expand into the reservoir should temperature increase. In B, the brakes are applied. The piston rod-end seal covers the compensating port as it contacts the piston head.

When the pedal is released, the piston rod travels in the opposite direction, and the piston returns to the piston stop. The compensating poppet reopens. The toggle is withdrawn from the spool via linkages, and fluid pushes the spool back to expose the system return manifold port. System hydraulic fluid used to boost brake pressure returns through the port.

POWER BRAKES

Large and high performance aircraft are equipped with power brakes to slow, stop, and hold the aircraft. Power brake actuating systems use the aircraft hydraulic system as the source of power to apply the brakes. The pilot presses on the top of the rudder pedal for braking as with the other actuating systems. The volume and pressure of hydraulic fluid required cannot be produced

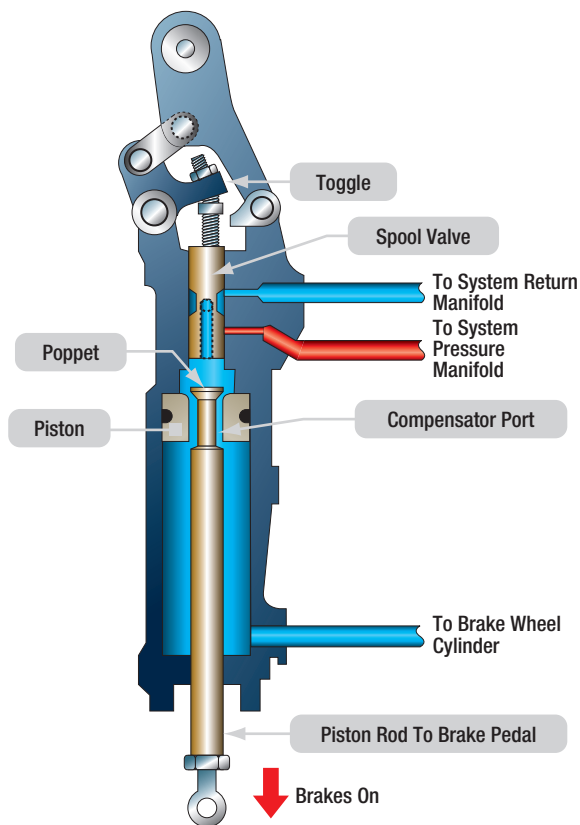


Figure 13-88. A master cylinder for a boosted brake system augments foot pedal pressure with aircraft system hydraulic pressure during heavy braking.

by a master cylinder. Instead, a power brake control valve or brake metering valve receives the brake pedal input either directly or through linkages. The valve meters hydraulic fluid to the corresponding brake assembly in direct relation to the pressure applied to the pedal. Many power brake system designs are in use.

Most are similar to the simplified system illustrated in *Figure 13-89A*. Power brake systems are constructed to facilitate graduated brake pressure control, brake pedal feel, and the necessary redundancy required in case of hydraulic system failure. Large aircraft brake systems integrate antiskid detection and correction devices. These are necessary because wheel skid is difficult to detect on the flight deck without sensors. However, a skid can be quickly controlled automatically through pressure control of the hydraulic fluid to the brakes.

Hydraulic fuses are also commonly found in power brake systems. The hostile environment around the landing gear increases the potential for a line to break or sever, a fitting to fail, or other hydraulic system malfunctions to occur where hydraulic fluid is lost en route to the brake

assemblies. A fuse stops any excessive flow of fluid when detected by closing to retain the remaining fluid in the hydraulic system. Shuttle valves are used to direct flow from optional sources of fluid, such as in redundant systems or during the use of an emergency brake power source. An airliner power brake system is illustrated in *Figure 13-89B*.

Brake Control Valve/ Brake Metering Valve

The key element in a power brake system is the brake control valve, sometimes called a brake metering valve. It responds to brake pedal input by directing aircraft system hydraulic fluid to the brakes. As pressure is increased on the brake pedal, more fluid is directed to the brake causing a higher pressure and greater braking action.

A brake metering valve from a Boeing 737 is illustrated in *Figure 13-90*. The system in which it is installed is diagrammed in *Figure 13-91*. Two sources of hydraulic pressure provide redundancy in this brake system. A brake input shaft, connected to the rudder/brake pedal through mechanical linkages, provides the position input to the metering valve. As in most brake control valves, the brake input shaft moves a tapered spool or slide in the valve so that it allows hydraulic system pressure to flow to the brakes. At the same time, the slide covers and uncovers access to the hydraulic system return port as required.

When the rudder/brake pedal is depressed, the slide in the metering valve moves to the left. (*Figure 13-90*) It covers the return port so pressure can build in the brake system. The hydraulic supply pressure chamber is connected to the brake system pressure chamber by the movement of the slide, which due to its taper, unblocks the passage between these two. As the pedal is depressed further, the valve slide moves farther to the left. This enables more fluid to flow to the brakes due to the narrowing shape of the slide. Brake pressure increases with the additional fluid.

A passage in the slide directs brake pressure fluid into a compensating chamber at the end of the slide. This acts on the end of the slide creating a return force that counters the initial slide movement and gives feel to the brake pedal. As a result, the pressure and return ports are closed and pressure proportional to the foot pressure on the pedal is held on the brakes. When the pedal is

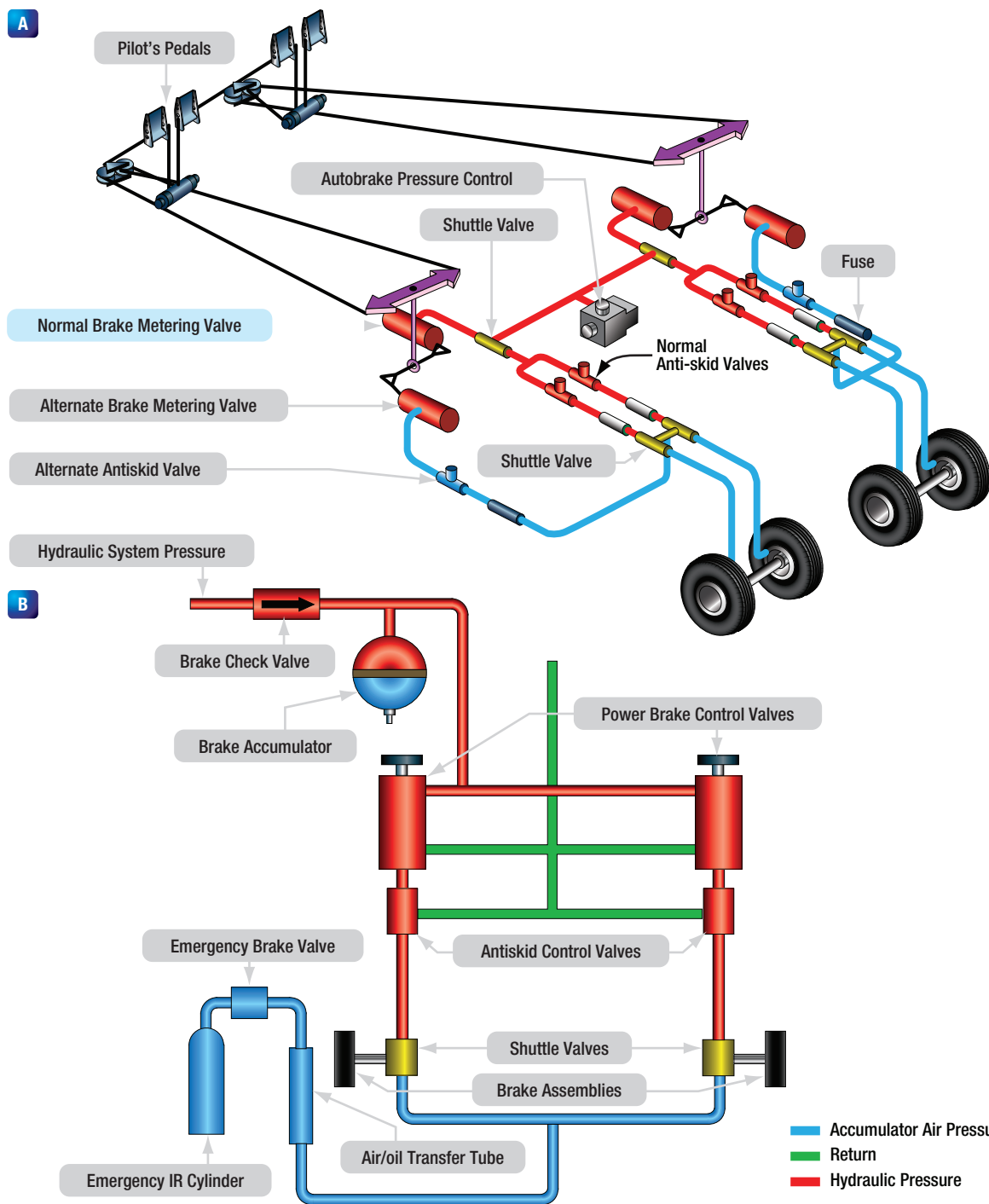


Figure 13-89. The orientation of components in a basic power brake system is shown in A.
The general layout of an airliner power brake system is shown in B.

released, a return spring and compensating chamber pressure drive the slide to the right into its original position (return port open, supply pressure chamber and brake pressure chambers blocked from each other). The metering valve operates as described simultaneously for the inboard and the outboard brakes. (*Figure 13-90*) The design of the link assembly is such that a single side of the metering valve can operate even if the other fails.

Most brake control valves and metering valves function in a similar manner, although many are single units that supply only one brake assembly.

The auto brake, referenced in the metering valve diagram, is connected into the landing gear retraction hydraulic line. Pressurized fluid enters this port and drives the slide slightly to the left to apply the brakes

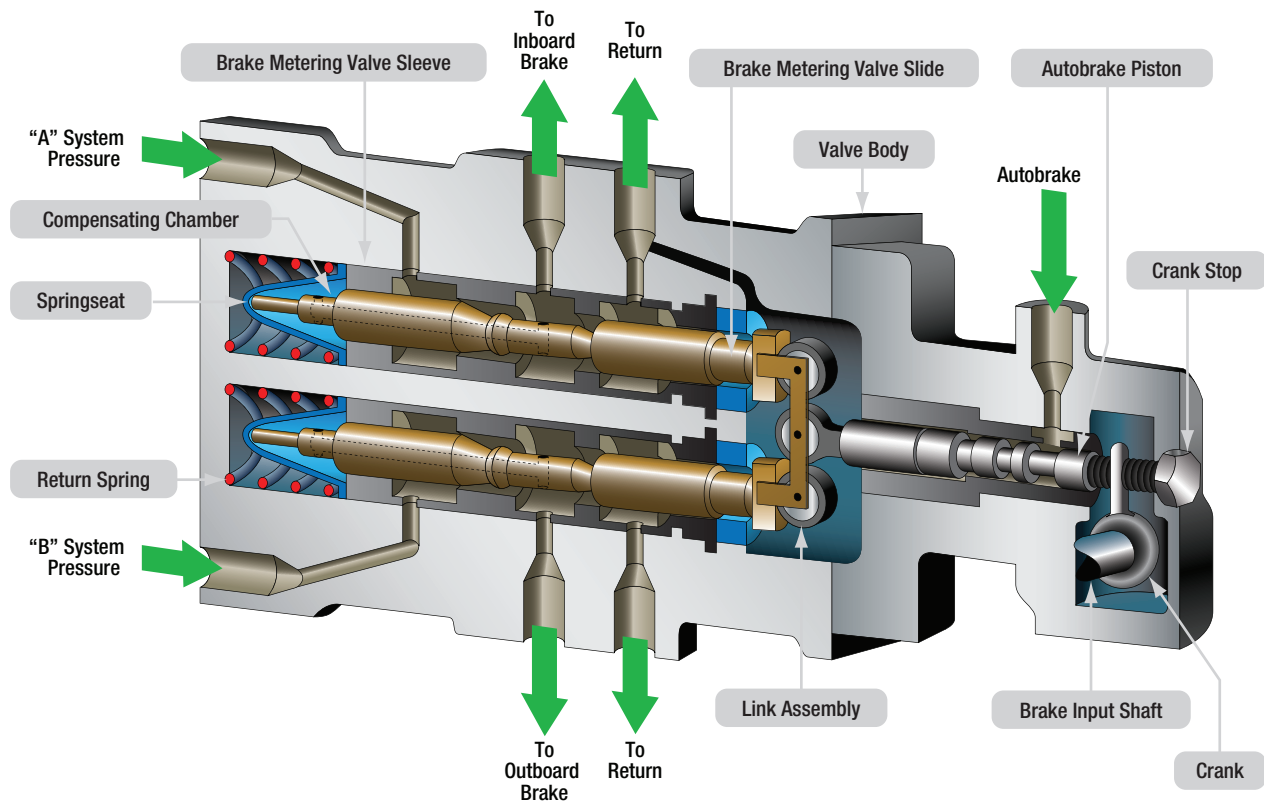


Figure 13-90. A brake metering valve from a Boeing 737. A machined slide or spool moves laterally to admit the correct amount of hydraulic system fluid to the brakes. The pressure developed is in proportion to the amount the rudder/brake pedal is depressed and the amount the slide is displaced. The slide/spool also simultaneously controls the return of fluid to the hydraulic system return manifold when brake pressure is released.

automatically after takeoff. This stops the wheels from rotating when retracted into the wheel wells. Auto brake pressure is withheld from this port when the landing gear is fully stowed since the retraction system is depressurized.

The majority of the rudder/brake pedal feel is supplied by the brake control or brake metering valve in a power brake system. Many aircraft refine the feel of the pedal with an additional feel unit. The brake valve feel augmentation unit, in the above system, uses a series of internal springs and pistons of various sizes to create a force on the brake input shaft movement. This provides feel back through the mechanical linkages consistent with the amount of rudder/brake pedal applied. The request for light braking with slight pedal depression results in a light feel to the pedal and a harder resistance feel when the pedals are pushed harder during heavy braking. (*Figure 13-92*)

EMERGENCY BRAKE SYSTEMS

As can be seen in *Figure 13-99*, the brake metering valves not only receive hydraulic pressure from two

separate hydraulic systems, they also feed two separate brake assemblies. Each main wheel assembly has two wheels. The inboard wheel brake and the outboard wheel brake, located in their respective wheel rims, are independent from each other. In case of hydraulic system failure or brake failure, each is independently supplied to adequately slow and stop the aircraft without the other. More complicated aircraft may involve another hydraulic system for back-up or use a similar alternation of sources and brake assemblies to maintain braking in case of hydraulic system or brake failure.

NOTE: In the segmented rotor brake section above, a brake assembly was described that had alternating pistons supplied by independent hydraulic sources. This is another method of redundancy particularly suitable on, but not limited to, single main wheel aircraft.

In addition to supply system redundancy, the brake accumulator is also an emergency source of power for the brakes in many power brake systems. The accumulator is precharged with air or nitrogen on one side of its internal diaphragm. Enough hydraulic fluid is contained on the

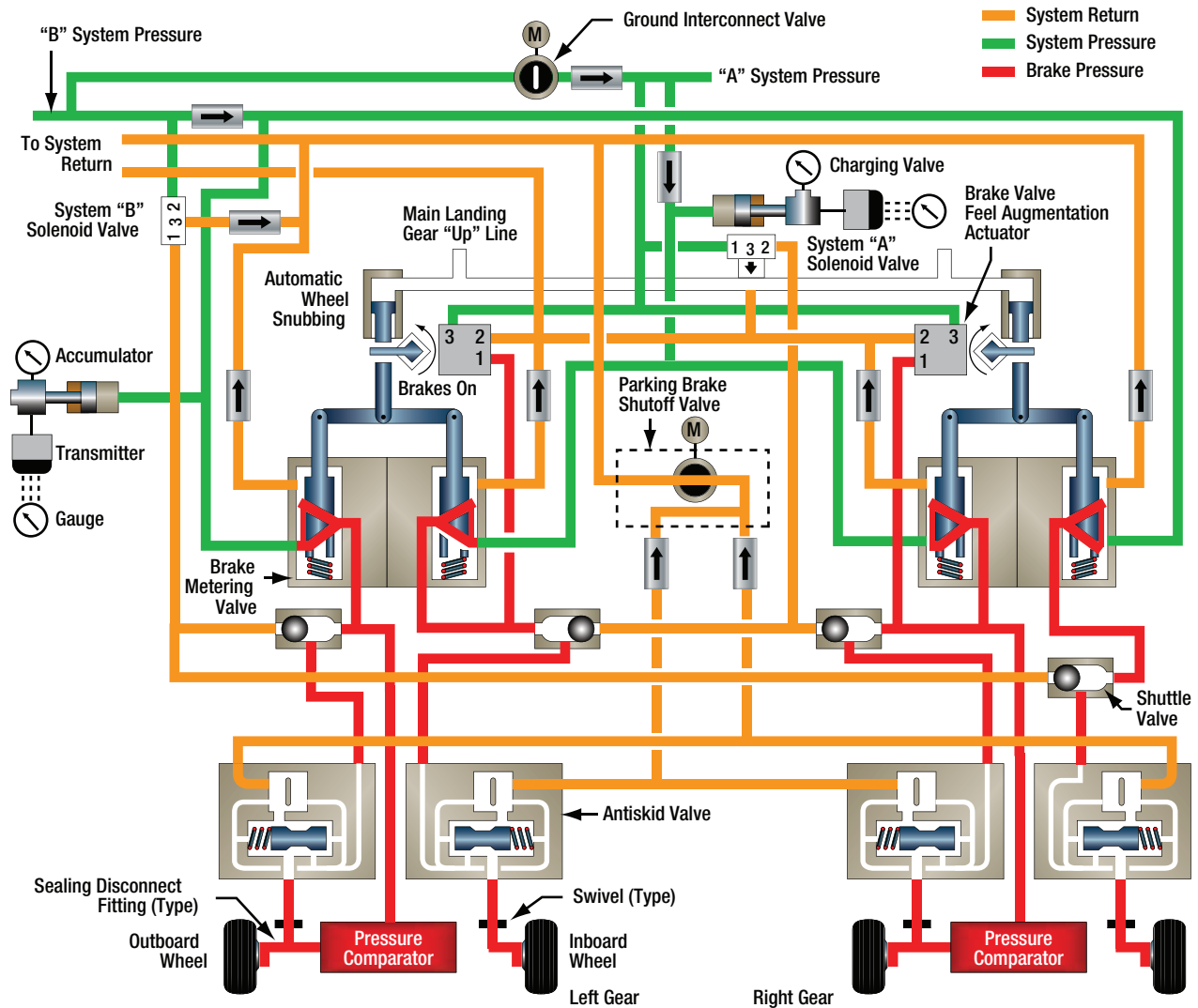


Figure 13-91. The power brake system on a Boeing 737.

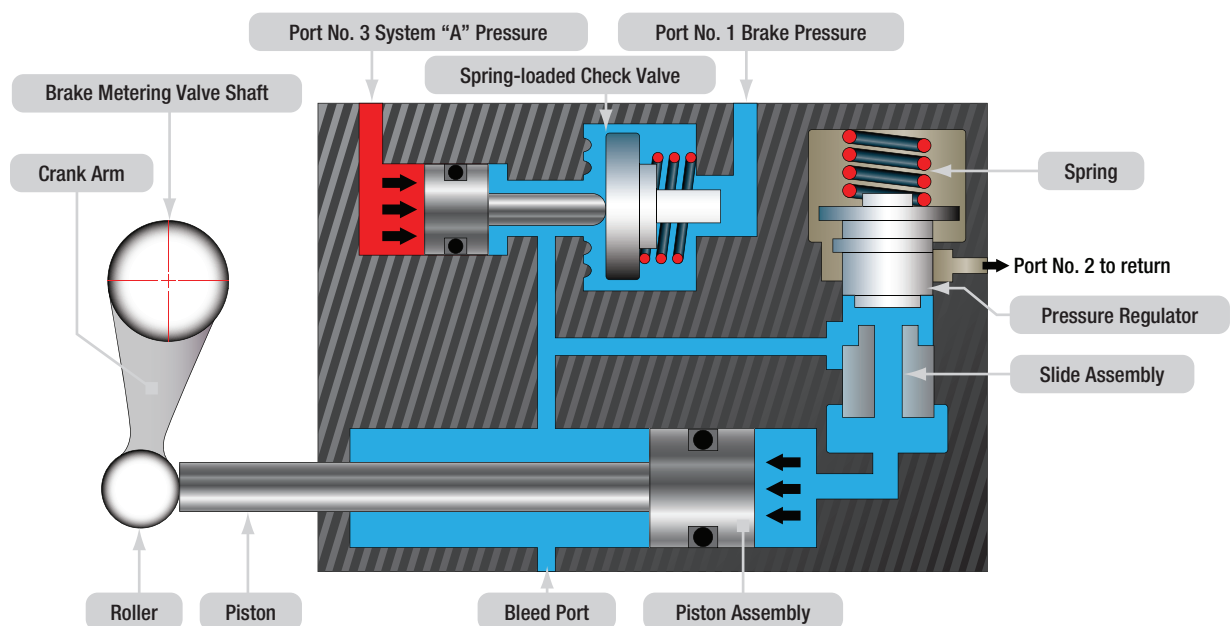


Figure 13-92. The power brake system on a Boeing 737.

other side of the diaphragm to operate the brakes in case of an emergency. It is forced out of the accumulator into the brakes through the system lines under enough stored pressure to slow the aircraft. Typically, the accumulator is located upstream of the brake control/metering valve to capitalize on the control given by the valve. (Figure 13-93)

Some simpler power brake systems may use an emergency source of brake power that is delivered directly to the brake assemblies and bypasses the remainder of the brake system completely. A shuttle valve immediately upstream of the brake units shifts to accept this source when pressure is lost from the primary supply sources. Compressed air or nitrogen is sometimes used. A precharged fluid source can also be used as an alternate hydraulic source.

PARKING BRAKE

The parking brake system function is a combined operation. The brakes are applied with the rudder pedals and a ratcheting system holds them in place when the parking brake lever on the flight deck is pulled. (Figure 13-94)

At the same time, a shut-off valve is closed in the common return line from the brakes to the hydraulic system. This traps the fluid in the brakes holding the rotors stationary. Depressing the pedals further releases the pedal ratchet and opens the return line valve.

BRAKE DEBOOSTERS

Some aircraft brake assemblies that operate on aircraft hydraulic system pressure are not designed for such high pressure. They provide effective braking through a power brake system but require less than maximum hydraulic system pressure. To supply the lower pressure, a brake deboster cylinder is installed downstream of the control valve and antiskid valve. (Figure 13-95)

The deboster reduces all pressure from the control valve to within the working range of the brake assembly. Brake debosters are simple devices that use the application of force over different sized pistons to reduce pressure. (Figure 13-96) Their operation can be understood through the application of the following equation:

$$\text{Pressure} = \text{Force}/\text{Area}$$

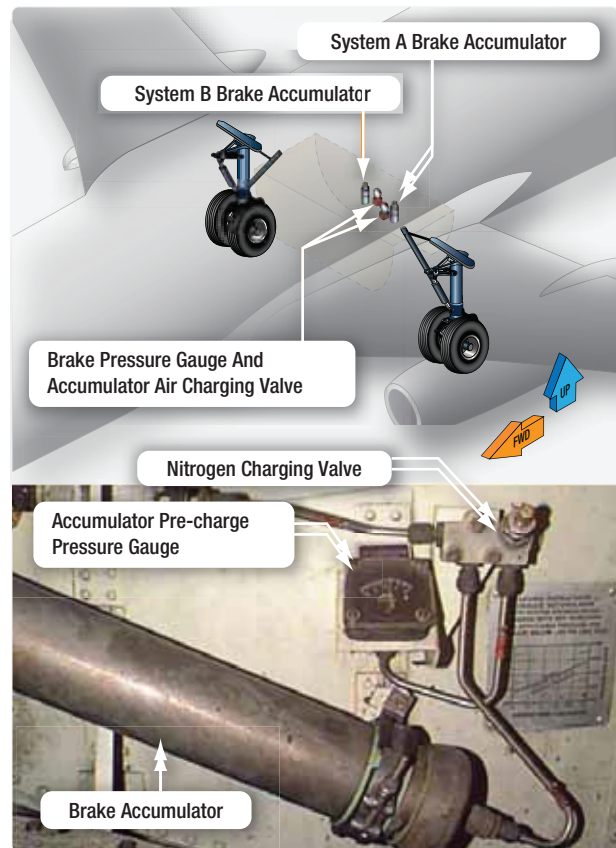


Figure 13-93. Emergency brake hydraulic fluid accumulators are precharged with nitrogen to deliver brake fluid to the brakes in the event normal and alternate hydraulic sources fail.

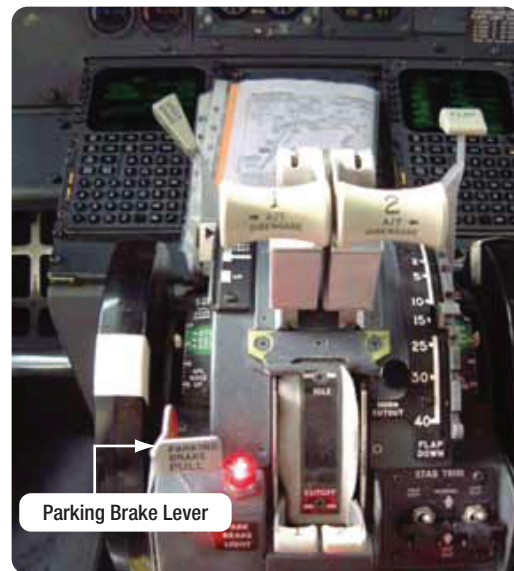


Figure 13-94. The parking brake lever on a Boeing 737 center pedestal throttle quadrant.

High-pressure hydraulic system input pressure acts on the small end of a piston. This develops a force proportional to the area of the piston head. The other

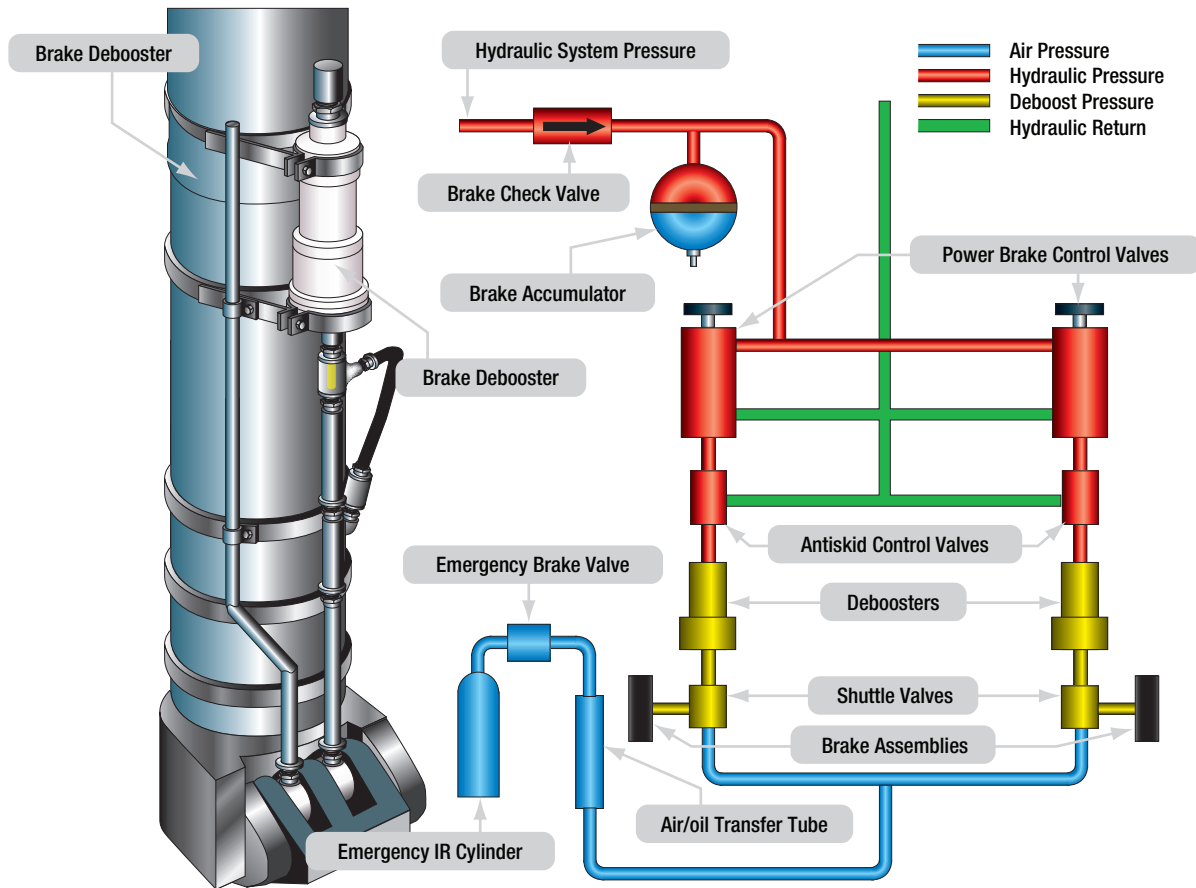


Figure 13-95. The location of a brake deboost cylinder on a landing gear strut and the deboost's position in relation to other components of a power brake system.

end of the piston is larger and housed in a separate cylinder. The force from the smaller piston head is transferred to the larger area of the other end of the piston. The amount of pressure conveyed by the larger end of the piston is reduced due to the greater area over which the force is spread. The volume of output fluid increases since a larger piston and cylinder are used. The reduced pressure is delivered to the brake assembly.

The spring in the deboost aids in returning the piston to the ready position. If fluid is lost downstream of the deboost cylinder, the piston travels further down into the cylinder when the brakes are applied. The pin unseats the ball and allows fluid into the lower cylinder to replace what was lost. Once replenished, the piston rises up in the cylinder due to pressure build-up. The ball reseats as the piston travels above the pin and normal braking resumes. This function is not meant to permit leaks in the brake assemblies. Any leak discovered must be repaired by the technician.

A lockout deboost functions as a deboost and a hydraulic fuse. If fluid is not encountered as the piston moves down in the cylinder, the flow of fluid to the brakes is stopped. This prevents the loss of all system hydraulic fluid should a rupture downstream of the deboost occur. Lockout deboosts have a handle to reset the device after it closes as a fuse. If not reset, no braking action is possible.

ANTISKID

Large aircraft with power brakes require antiskid systems. It is not possible to immediately ascertain in the flight deck when a wheel stops rotating and begins to skid, especially in aircraft with multiple wheel main landing gear assemblies. A skid not corrected can quickly lead to a tire blowout, possible damage to the aircraft, and control of the aircraft may be lost.

SYSTEM OPERATION

The antiskid system not only detects wheel skid, it also detects when wheel skid is imminent. It automatically relieves pressure to the brake pistons of the wheel in

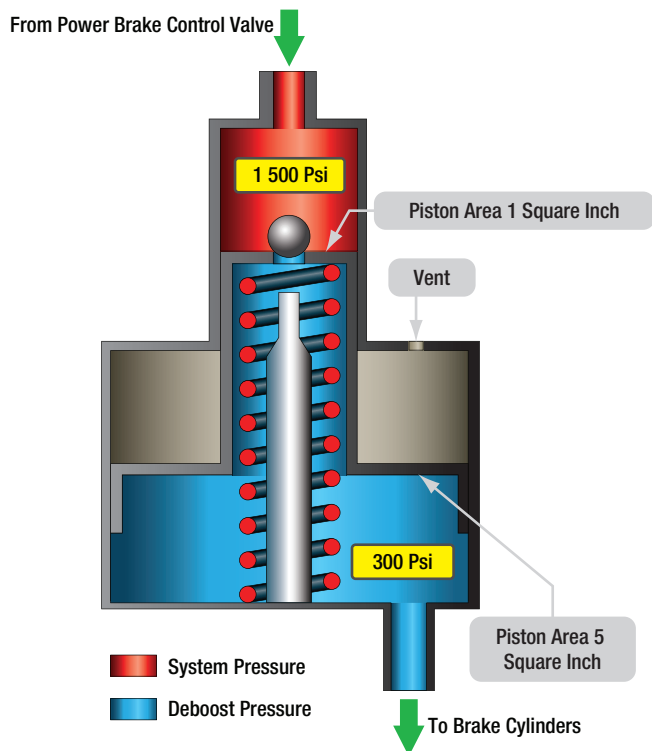


Figure 13-96. Brake boosters.

question by momentarily connecting the pressurized brake fluid area to the hydraulic system return line. This allows the wheel to rotate and avoid a skid. Lower pressure is then maintained to the brake at a level that slows the wheel without causing it to skid.

Maximum braking efficiency exists when the wheels are decelerating at a maximum rate but are not skidding. If a wheel decelerates too fast, it is an indication that the brakes are about to lock and cause a skid. To ensure that this does not happen, each wheel is monitored for a deceleration rate faster than a preset rate. When excessive deceleration is detected, hydraulic pressure is reduced to the brake on that wheel. To operate the antiskid system, flight deck switches must be placed in the ON position. (Figure 13-97)

After the aircraft touches down, the pilot applies and holds full pressure to the brake pedals. The antiskid system then functions automatically until the speed of the aircraft has dropped to approximately 20 mph. The system returns to manual braking mode for slow taxi and ground maneuvering.

There are various designs of antiskid systems. Most contain three main types of components: wheel speed sensors, antiskid control valves, and a control unit. These

units work together without human interference. Some antiskid systems provide complete automatic braking. The pilot needs only to turn on the auto brake system, and the antiskid components slow the aircraft without pedal input. (Figure 13-97)

Ground safety switches are wired into the circuitry for antiskid and auto brake systems. Wheel speed sensors are located on each wheel equipped with a brake assembly. Each brake also has its own antiskid control valve. Typically, a single control box contains the antiskid comparative circuitry for all of the brakes on the aircraft. (Figure 13-98)

WHEEL SPEED SENSORS

Wheel speed sensors are transducers. They may be alternating current (AC) or direct current (DC). The typical AC wheel speed sensor has a stator mounted in the wheel axle. A coil around it is connected to a controlled DC source so that when energized, the stator



Figure 13-97. Antiskid switches in the cockpit.

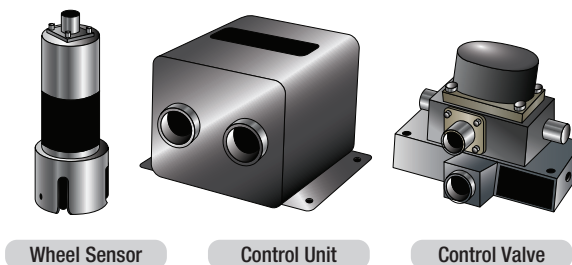


Figure 13-98. A wheel sensor (left), a control unit (center), and a control valve (right) are components of an antiskid system. A sensor is located on each wheel equipped with a brake assembly. An antiskid control valve for each brake assembly is controlled from a single central control unit.

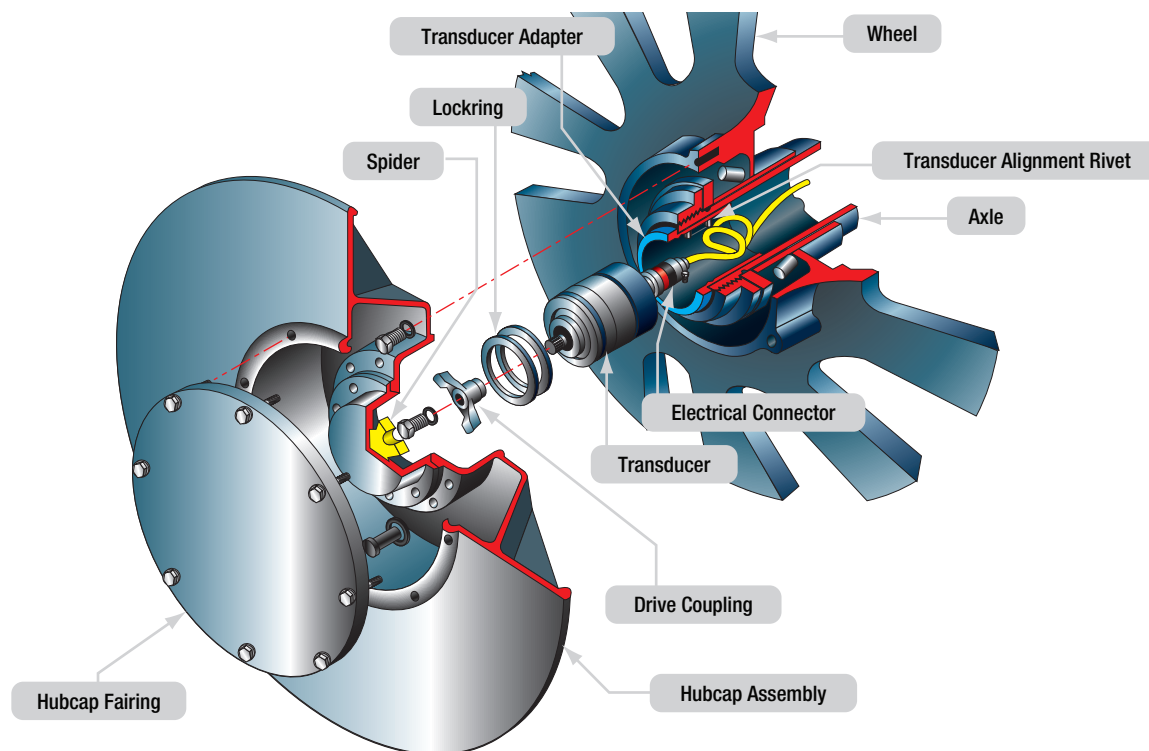


Figure 13-99. The stator of an antiskid wheel sensor is mounted in the axle, and the rotor is coupled to the wheel hub spider that rotates with the wheel.

becomes an electromagnet. A rotor that turns inside the stator is connected to the rotating wheel hub assembly through a drive coupling so that it rotates at the speed of the wheel. Lobes on the rotor and stator cause the distance between the two components to constantly change during rotation. This alters the magnetic coupling or reluctance between the rotor and stator. As the electromagnetic field changes, a variable frequency AC is induced in the stator coil. The frequency is directly proportional to the speed of rotation of the wheel. The AC signal is fed to the control unit for processing. A DC wheel speed sensor is similar, except that a DC is produced, the magnitude of which is directly proportional to wheel speed. (*Figure 13-99*)

CONTROL UNITS

The control unit can be regarded as the brain of the antiskid system. It receives signals from each of the wheel sensors. Comparative circuits are used to determine if any of the signals indicate a skid is imminent or occurring on a particular wheel. If so, a signal is sent to the control valve of the wheel to relieve hydraulic pressure to that brake which prevents or relieves the skid. The control unit may or may not have external test switches and status indicating lights. It is common for it to be located in the avionics bay of the aircraft. (*Figure 13-100*)



Figure 13-100. A rack mounted antiskid control unit from an airliner.

The Boeing antiskid control valve block diagram in *Figure 13-101* gives further detail on the functions of an antiskid control unit. Other aircraft may have different logic to achieve similar end results. DC systems do not require an input converter since DC is received from the wheel sensors, and the control unit circuitry operates primarily with DC.

Only the functions on one circuit card for one wheel brake assembly are shown in *Figure 13-101*. Each wheel has its own identical circuitry card to facilitate simultaneous operation. All cards are housed in a single control unit that Boeing calls a control shield. The converter shown changes the AC frequency

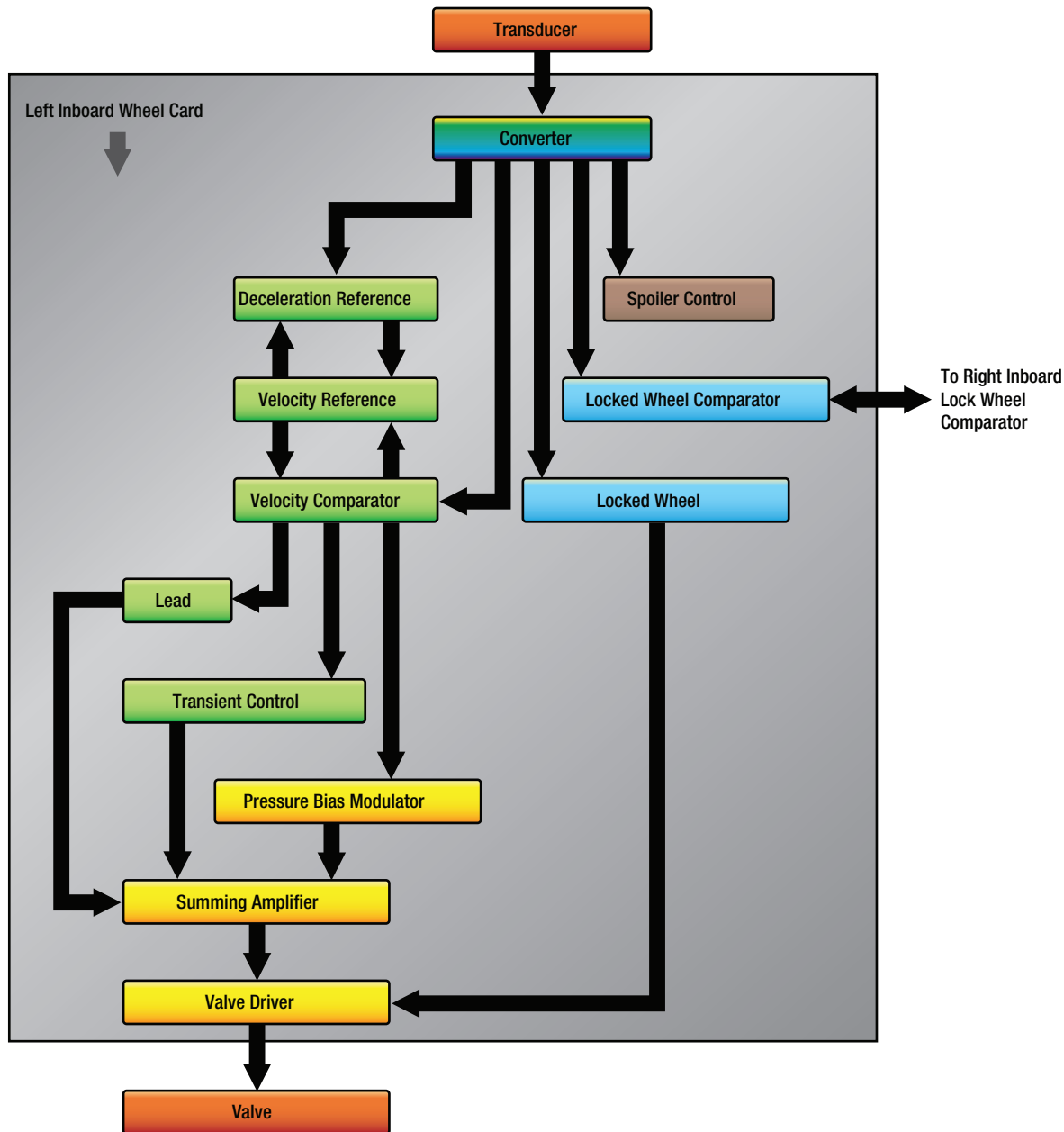


Figure 13-101. A Boeing 737 antiskid control unit internal block diagram.

received from the wheel sensor into DC voltage that is proportional to wheel speed. The output is used in a velocity reference loop that contains deceleration and velocity reference circuits.

The converter also supplies input for the spoiler system and the locked wheel system, which is discussed at the end of this section. A velocity reference loop output voltage is produced, which represents the instantaneous velocity of the aircraft. This is compared to converter output in the velocity comparator. This comparison of voltages is essentially the comparison of the aircraft speed to wheel speed. The output from the velocity

comparator is a positive or negative error voltage corresponding to whether the wheel speed is too fast or too slow for optimum braking efficiency for a given aircraft speed.

The error output voltage from the comparator feeds the pressure bias modulator circuit. This is a memory circuit that establishes a threshold where the pressure to the brakes provides optimum braking. The error voltage causes the modulator to either increase or decrease the pressure to the brakes in attempt to hold the modulator threshold. It produces a voltage output that is sent to the summing amplifier to do this. A lead output from

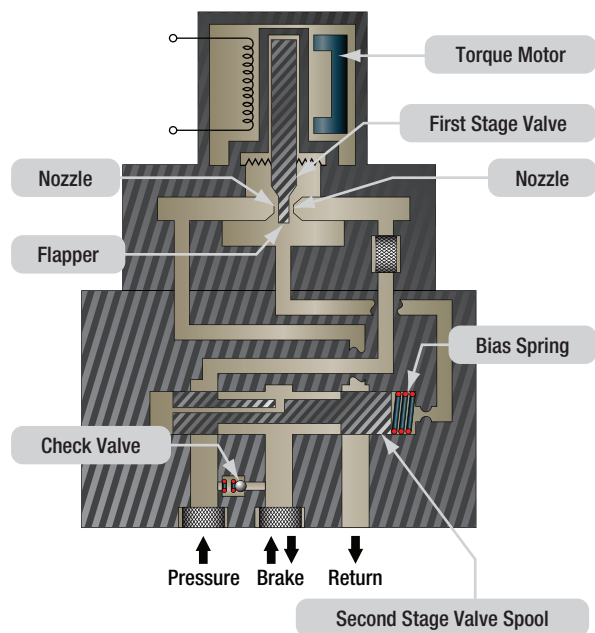


Figure 13-102. An antiskid control valve uses a torque motor controlled flapper in the first stage of the valve to adjust pressure on a spool in the second stage of the valve to build or relieve pressure to the brake.

the comparator anticipates when the tire is about to skid with a voltage that decreases the pressure to the brake. It sends this voltage to the summing amplifier as well. A transient control output from the comparator designed for rapid pressure dump when a sudden skid has occurred also sends voltage to the summing amp. As the name suggests, the input voltages to the amplifier are summed, and a composite voltage is sent to the valve driver. The driver prepares the current required to be sent to the control valve to adjust the position of the valve. Brake pressure increases, decreases, or holds steady depending on this value.

ANTISKID CONTROL VALVES

Anti-skid control valves are fast-acting, electrically controlled hydraulic valves that respond to the input from the antiskid control unit. There is one control valve for each brake assembly. A torque motor uses the input from the valve driver to adjust the position of a flapper between two nozzles. By moving the flapper closer to one nozzle or the other, pressures are developed in the second stage of the valve. These pressures act on a spool that is positioned to build or reduce pressure to the brake by opening and blocking fluid ports. (Figure 13-102)

As pressure is adjusted to the brakes, deceleration slows to within the range that provides the most effective braking without skidding. The wheel sensor

signal adjusts to the wheel speed, and the control unit processes the change. Output is altered to the control valve. The control valve flapper position is adjusted and steady braking resumes without correction until needed. Anti-skid control valves are typically located in the main wheel for close access to hydraulic pressure and return manifolds, as well as the brake assemblies. (Figure 13-103)

Systematically, they are positioned downstream of the power brake control valves but upstream of deboosters cylinders if the aircraft is so equipped as was shown in Figure 13-95.

TOUCH DOWN AND LOCK WHEEL PROTECTION

It is essential that the brakes are not applied when the aircraft contacts the runway upon landing. This could cause immediate tire blowout. A touchdown protection mode is built into most aircraft antiskid systems to prevent this. It typically functions in conjunction with the wheel speed sensor and the air/ground safety switch on the landing gear strut (squat switch). Until the aircraft has weight on wheels, the detector circuitry signals the antiskid control valve to open the passage between the brakes and the hydraulic system return, thus preventing pressure build-up and application of the brakes. Once the squat switch is open, the antiskid control unit sends a signal to the control valve to close and permit brake pressure build-up. As a back-up and when the aircraft is on the ground with the strut not compressed enough to open the squat switch, a minimum wheel speed sensor signal can override and allow braking. Wheels are often

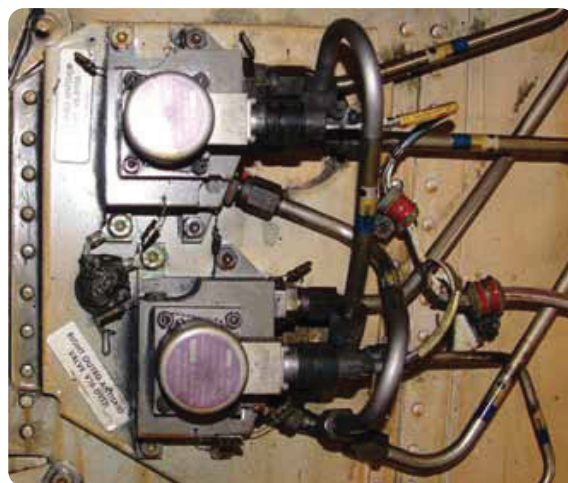


Figure 13-103. Two antiskid control valves with associated plumbing and wiring.

grouped with one relying on the squat switch and the other on wheel speed sensor output to ensure braking when the aircraft is on the ground, but not before then.

Locked wheel protection recognizes if a wheel is not rotating. When this occurs, the antiskid control valve is signaled to fully open. Some aircraft antiskid control logic, such as the Boeing 737 shown in *Figure 13-102*, expands the locked wheel function. Comparator circuitry is used to relieve pressure when one wheel of a paired group of wheels rotates 25 percent slower than the other. Inboard and outboard pairs are used because if one of the pair is rotating at a certain speed, so should the other. If it is not, a skid is beginning or has occurred.

On takeoff, the antiskid system receives input through a switch located on the gear selector that shuts off the antiskid system. This allows the brakes to be applied as retraction occurs so that no wheel rotation exists while the gear is stowed.

ANTISKID SYSTEM TESTS

It is important to know the status of the antiskid system prior to attempting to use it during a landing or aborted takeoff. Ground tests and inflight tests are used. Built-in test circuits and control features allow testing of the system components and provide warnings should a particular component or part of the system become inoperative. An inoperative antiskid system can be shut off without affecting normal brake operation.

GROUND TEST

Ground tests vary slightly from aircraft to aircraft. Consult the manufacturer's maintenance manual for test procedures specific to the aircraft in question. Much of the antiskid system testing originates from testing circuits in the antiskid control unit. Built-in test circuits continuously monitor the antiskid system and provide warning if a failure occurs. An operational test can be performed before flight. The antiskid control switch and/or test switch is used in conjunction with system indicator light(s) to determine system integrity.

A test is first done with the aircraft at rest and then in an electrically simulated antiskid braking condition. Some antiskid control units contain system and component testing switches and lights for use by the technician. This accomplishes the same operational verification, but allows an additional degree of troubleshooting.

Test sets are available for antiskid systems that produce electric signals that simulate speed outputs of the wheel transducer, deceleration rates, and flight, and ground parameters.

INFLIGHT TEST

Inflight testing of the antiskid system is desirable and part of the pre-landing checklist so that the pilot is aware of system capability before landing. As with ground testing, a combination of switch positions and indicator lights are used according to information in the aircraft operations manual.

ANTISKID SYSTEM MAINTENANCE

Anti-skid components require little maintenance. Troubleshooting antiskid system faults is either performed via test circuitry or can be accomplished through isolation of the fault to one of the three main operating components of the system. Anti-skid components are normally not repaired in the field. They are sent to the manufacturer or a certified repair station when work is required. Reports of antiskid system malfunction are sometimes malfunctions of the brake system or brake assemblies. Ensure brake assemblies are bled and functioning normally without leaks before attempting to isolate problems in the antiskid system.

WHEEL SPEED SENSOR

Wheel speed sensors must be securely and correctly mounted in the axle. The means of keeping contamination out of the sensor, such as sealant or a hub cap should be in place and in good condition. The wiring to the sensor is subject to harsh conditions and should be inspected for integrity and security. It should be repaired or replaced if damaged in accordance with the manufacturer's instructions. Accessing the wheel speed sensor and spinning it by hand or other recommended device to ensure brakes apply and release via the antiskid system is common practice.

CONTROL VALVE

Anti-skid control valve and hydraulic system filters should be cleaned or replaced at the prescribed intervals. Follow all manufacturer's instructions when performing this maintenance. Wiring to the valve must be secure, and there should be no fluid leaks.

CONTROL UNIT

Control units should be securely mounted. Test switches and indicators, if any, should be in place and functioning. It is essential that wiring to the control unit is secure. A wide variety of control units are in use. Follow the manufacturer's instructions at all times when inspecting or attempting to perform maintenance on these units.

AUTO BRAKING

Aircraft equipped with auto brakes typically bypass the brake control valves or brake metering valves and use a separate auto brake control valve to provide this function. In addition to the redundancy provided, auto brakes rely on the antiskid system to adjust pressure to the brakes if required due to an impending skid.

Figure 13-104 shows a simplified diagram of the Boeing 757 brake system with the auto brake valve in relation to the main metering valve and antiskid valves

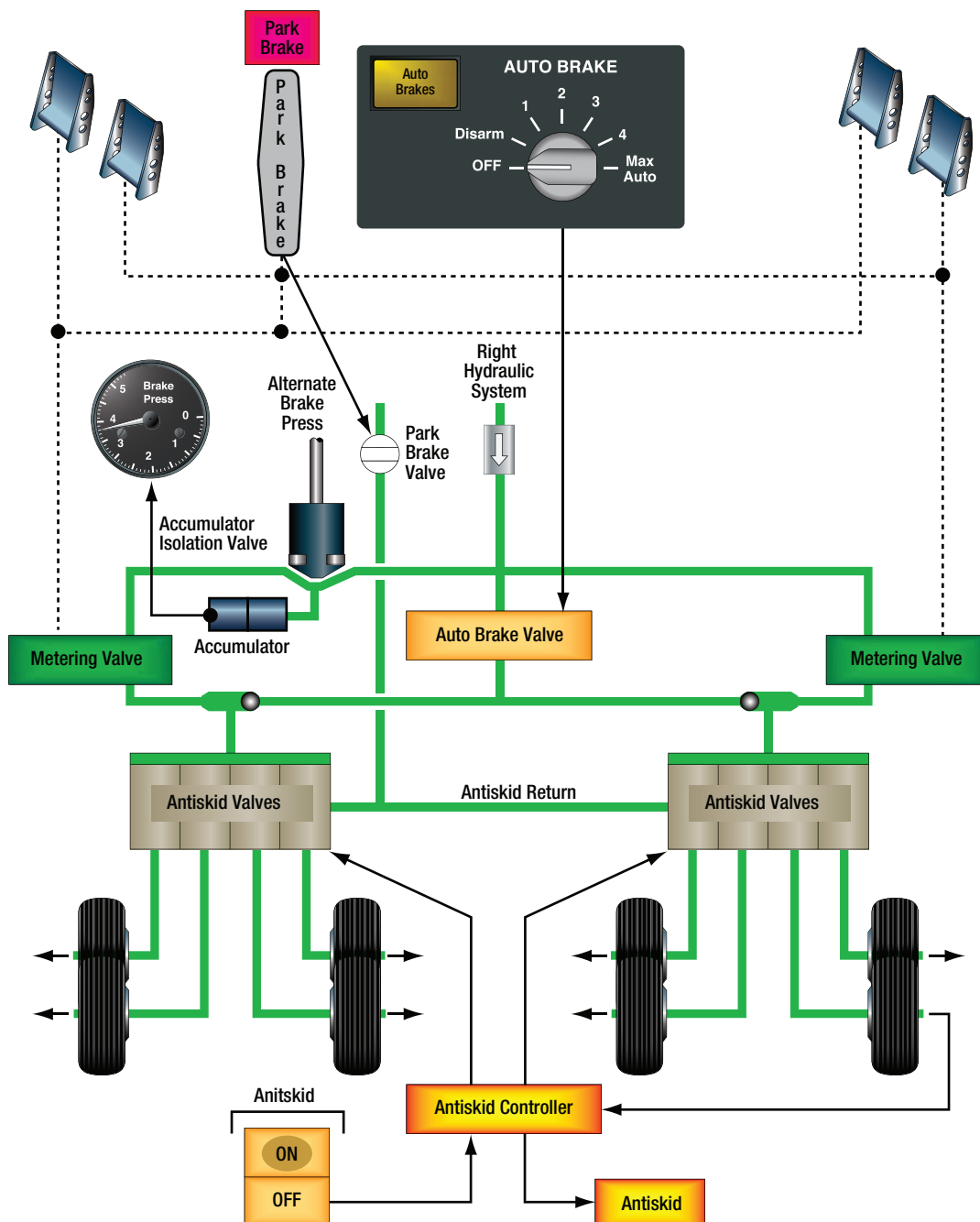


Figure 13-104. The Boeing 757 normal brake system with auto brake and antiskid.

in this eight-main wheel system. Note that in an auto brake system, the antiskid system must be functioning properly for the auto brake system to function.

Auto braking is activated by a switch on the flight deck. It can be set to various levels of deceleration. Typically, when turned ON, the auto brake system brakes automatically when weight is on the wheels and the throttles are in the idle position or thrust reversers are activated. Auto brake deactivates if the crew steps on the rudder/brake pedals, or the aircraft slows beyond a predetermined speed.

An auto brake control box controlled by the selector switch regulates the function of the auto brake valve when auto braking is selected. It meters hydraulic fluid to the brakes instead of the regular brake metering valves. Downstream of the auto brake valve, the antiskid functions normally. Should a malfunction in the auto brake system occur, a warning light or flat screen annunciation will illuminate on the flight deck. The system will not function in this case and braking must be accomplished with the rudder brake pedals.

BRAKE INSPECTION AND SERVICE

Brake inspection and service is important to keep these critical aircraft components fully functional at all times. There are many different brake systems on aircraft. Brake system maintenance is performed both while the brakes are installed on the aircraft and when the brakes are removed. The manufacturer's instructions must always be followed to ensure proper maintenance.

ON AIRCRAFT SERVICING

Inspection and servicing of aircraft brakes while installed on the aircraft is required. The entire brake system must be inspected in accordance with manufacturer's instructions. Some common inspection items include: brake lining wear, air in the brake system, fluid quantity level, leaks, and proper bolt torque.

LINING WEAR

Brake lining material is made to wear as it causes friction during application of the brakes. This wear must be monitored to ensure it is not worn beyond limits and sufficient lining is available for effective braking. The aircraft manufacturer gives specifications for lining wear in its maintenance information. The amount of wear can

be checked while the brakes are installed on the aircraft. Many brake assemblies contain a built-in wear indicator pin. Typically, the exposed pin length decreases as the linings wear, and a minimum length is used to indicate the linings must be replaced. Caution must be used as different assemblies may vary in how the pin is measured. On the Goodyear brake described above, the wear pin is measured where it protrudes through the nut of the automatic adjuster on the back side of the piston cylinder. (*Figure 13-105*)

The Boeing brake illustrated in *Figure 13-80* measures the length of the pin from the back of the pressure plate when the brakes are applied (dimension L). The manufacturer's maintenance information must be consulted to ensure brake wear pin indicators on different aircraft are read correctly. On many other brake assemblies, the distance between the disc and a portion of the brake housing when the brakes are applied is sometimes used. As the linings wear, this distance increases. The manufacturer specifies at what distance the linings should be changed. (*Figure 13-106*)

Multiple disc brakes typically are checked for lining wear by applying the brakes and measuring the distance between the back of the pressure plate and the brake housing. (*Figure 13-107*)

Regardless of the method particular to each brake, regular monitoring and measurement of brake wear ensures linings are replaced as they become unserviceable. Linings worn beyond limits usually require the brake assembly to be removed for replacement.

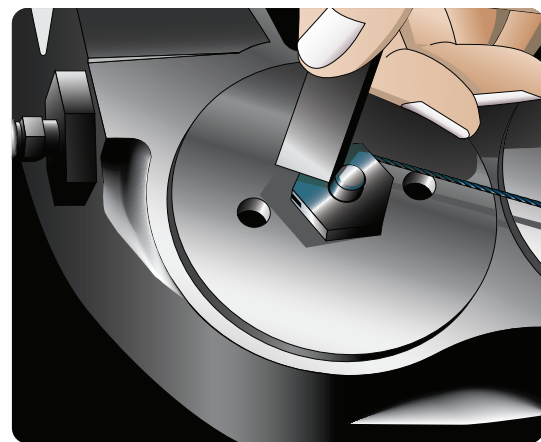


Figure 13-105. Brake lining wear on a Goodyear brake is ascertained by measuring the wear pin of the automatic adjuster.

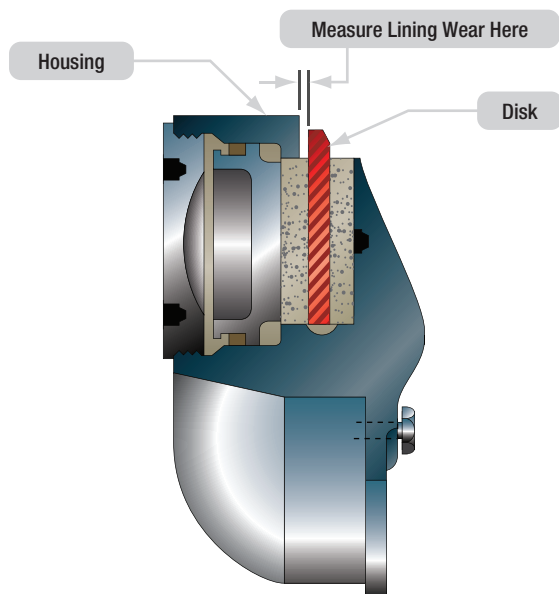


Figure 13-106. The distance between the brake disc and the brake housing measured with the brakes applied is a means for determining brake lining wear on some brakes.

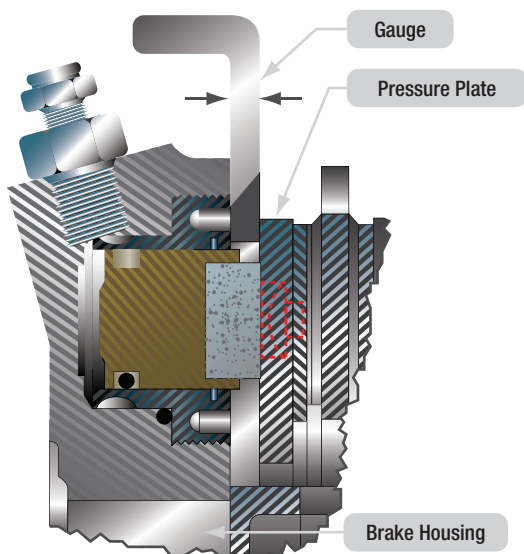


Figure 13-107. The distance between the brake housing and the pressure plate indicates lining wear on some multiple disc brakes.

AIR IN THE BRAKE SYSTEM

The presence of air in the brake system fluid causes the brake pedal to feel spongy. The air can be removed by bleeding to restore firm brake pedal feel. Brake systems must be bled according to manufacturers' instructions. The method used is matched to the type of brake system. Brakes are bled by one of two methods: top down, gravity bleeding or bottom up pressure bleeding. Brakes are bled when the pedals feel spongy or whenever the brake system has been opened.

BLEEDING MASTER CYLINDER BRAKE SYSTEMS

Brake systems with master cylinders may be bled by gravity or pressure bleeding methods. Follow the instructions in the aircraft maintenance manual. To pressure bleed a brake system from the bottom up, a pressure pot is used. (*Figure 13-108*)

This is a portable tank that contains a supply of brake fluid under pressure. When dispersing fluid from the tank, pure air-free fluid is forced from near the bottom of the tank by the air pressure above it. The outlet hose that attaches the bleed port on the brake assembly contains a shut-off valve. Note that a similar source of pure, pressurized fluid can be substituted for a pressure tank, such as a hand-pump type unit found in some hangars.

The typical pressure bleed is accomplished as illustrated in *Figure 13-109*. The hose from the pressure tank is attached to the bleed port on the brake assembly. A clear hose is attached to the vent port on the aircraft brake fluid reservoir or on the master cylinder if it incorporates the reservoir. The other end of this hose is placed in a collection container with a supply of clean brake fluid covering the end of the hose. The brake assembly bleed port is opened. The valve on the pressure tank hose is then opened allowing pure, air-free fluid to enter the brake system. Fluid containing trapped air is expelled through the hose attached to the vent port of the reservoir. The clear hose is monitored for air bubbles. When they cease to exist, the bleed port and pressure

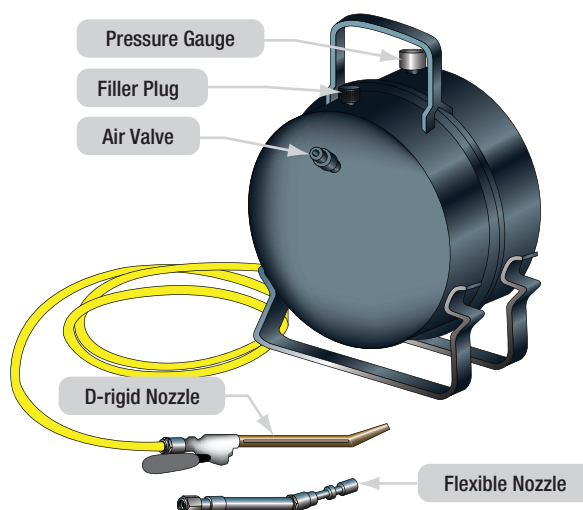


Figure 13-108. A typical brake bleeder pot or tank contains pure brake fluid under pressure. It pushes the fluid through the brake system to displace any air that may be present.

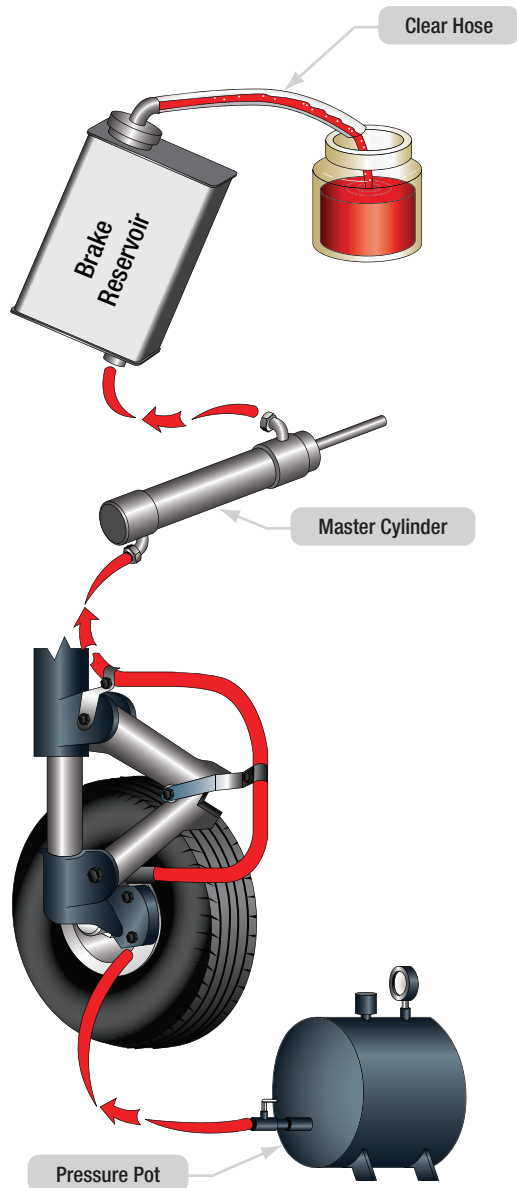


Figure 13-109. Arrangement for bottom-up pressure bleeding of aircraft brakes. Fluid is pushed through the system until no air bubbles are visible in the hose at the top.

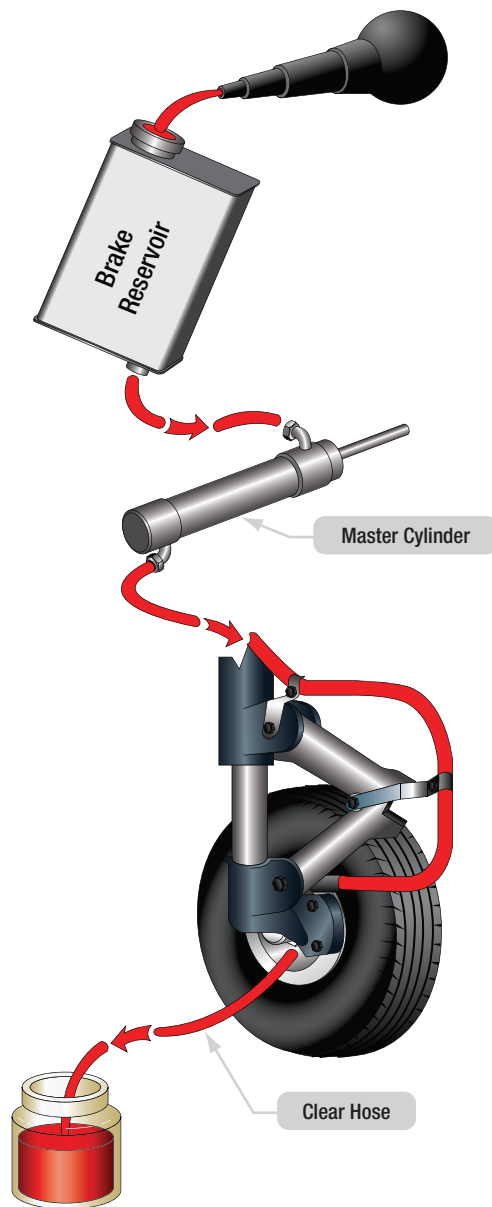


Figure 13-110. Arrangement for top down or gravity bleeding of aircraft brakes.

tank shutoff are closed and the pressure tank hose is removed. The hose at the reservoir is also removed. Fluid quantity may need to be adjusted to assure the reservoir is not over filled. Note that it is absolutely necessary that the proper fluid be used to service any brake system including when bleeding air from the brake lines.

Brakes with master cylinders may also be gravity bled from the top down. This is a process similar to that used on automobiles. (*Figure 13-110*) Additional fluid is supplied to the aircraft brake reservoir so that the quantity does not exhaust while bleeding, which would cause the reintroduction of more air into the system. A

clear hose is connected to the bleed port on the brake assembly. The other end is submersed in clean fluid in a container large enough to capture fluid expelled during the bleeding process. Depress the brake pedal and open the brake assembly bleed port. The piston in the master cylinder travels all the way to the end of the cylinder forcing air fluid mixture out of the bleed hose and into the container. With the pedal still depressed, close the bleed port. Pump the brake pedal to introduce more fluid from the reservoir ahead of the piston in the master cylinder. Hold the pedal down, and open the bleed port on the brake assembly. More fluid and air is expelled through the hose into the container. Repeat this process

until the fluid exiting the brake through the hose no longer contains any air. Tighten the bleed port fitting and ensure the reservoir is filled to the proper level.

Whenever bleeding the brakes, ensure that reservoirs and bleed tanks remain full during the process. Use only clean, specified fluid. Always check the brakes for proper operation, any leaks when bleeding is complete, and assure that the fluid quantity level is correct.

BLEEDING POWER BRAKE SYSTEMS

Top down brake bleeding is used in power brake systems. Power brakes are supplied with fluid from the aircraft hydraulic system. The hydraulic system should operate without air in the fluid as should the brake system. Therefore, bottom up pressure bleeding is not an option for power brakes; the air trapped in the brake system would be forced into the main hydraulic system, which is not acceptable. Many aircraft with power brake systems accept the connection of an auxiliary hydraulic mule that can be used to establish pressure in the system for bleeding. Regardless, the aircraft system must be pressurized to bleed power brake systems. Attach a clear hose to the brake bleed port fitting on the brake assembly and immerse the other end of the hose in a container of clean hydraulic fluid. With the bleeder valve open, carefully apply the brake to allow aircraft hydraulic fluid to enter the brake system. The fluid expels the fluid contaminated with air out of the bleed hose into the container.

When air is no longer visible in the hose, close the bleed valve and restore the hydraulic system to normal operation configuration. Power brake systems on different aircraft contain many variations and a wide array of components that may affect the proper bleeding technique to be followed.

Consult the manufacturer's maintenance information for the correct bleeding procedure for each aircraft. Be sure to bleed auxiliary and emergency brake systems when bleeding the normal brake system to ensure proper operation when needed.

FLUID QUANTITY AND TYPE

As mentioned, it is imperative that the correct hydraulic fluid is used in each brake system. Seals in the brake system are designed for a particular hydraulic fluid. Deterioration and failure occurs when they are exposed

to other fluids. Mineral based fluid, such as MIL-H-5606 (red oil), should never be mixed with phosphate-ester based synthetic hydraulic fluid such as Skydrol®. Contaminated brake/hydraulic systems must have all of the fluid evacuated and all seals replaced before the aircraft is released for flight.

Fluid quantity is also important. The technician is responsible for determining the method used to ascertain when the brake and hydraulic systems are fully serviced and for the maintenance of the fluid at this level. Consult the manufacturer's specifications for this information.

INSPECTION FOR LEAKS

Aircraft brake systems should maintain all fluid inside lines and components and should not leak. Any evidence of a leak must be investigated for its cause. It is possible that the leak is a precursor to more significant damage that can be repaired, thus avoiding an incident or accident. (*Figure 13-111*)

Many leaks are found at brake system fittings. While this type of leak may be fixed by tightening an obviously loose connection, the technician is cautioned against over tightening fittings. Removal of hydraulic pressure from the brake system followed by disconnection and inspection of the connectors is recommended. Over tightening of fitting can cause damage and make the leak worse. MS flareless fittings are particularly sensitive to over tightening. Replace all fittings suspected of damage. Once any leak is repaired, the brake system must be repressurized and tested for function as well as to ensure the leak no longer exists.



Figure 13-111. The cause of all aircraft brake leaks must be investigated, repaired, and tested before releasing the aircraft for flight.

Occasionally, a brake housing may seep fluid through the housing body. Consult the manufacturer's maintenance manual for limits, and remove any brake assembly that seeps excessively.

PROPER BOLT TORQUE

The stress experience by the landing gear and brake system requires that all bolts are properly torqued. Bolts used to attach the brakes to the strut typically have the required torque specified in the manufacturer's maintenance manual. Check for torque specifications that may exist for any landing gear and brake bolts, and ensure they are properly tightened. Whenever applying torque to a bolt on an aircraft, use of a calibrated torque wrench is required.

OFF AIRCRAFT BRAKE SERVICING AND MAINTENANCE

Certain servicing and maintenance of an aircraft brake assembly is performed while it has been removed from the aircraft. A close inspection of the assembly and its many parts should be performed at this time. Some of the inspection items on a typical assembly follow.

BOLT AND THREADED CONNECTIONS

All bolts and threaded connections are inspected. They should be in good condition without signs of wear. Self-locking nuts should still retain their locking feature. The hardware should be what is specified in the brake manufacturer's parts manual. Many aircraft brake bolts, for example, are not standard hardware and may be of closer tolerance or made of a different material. The demands of the high stress environment in which the brakes perform may cause brake failure if improper substitute hardware is used. Be sure to check the condition of all threads and O-ring seating areas machined into the housing. The fittings threaded into the housing must also be checked for condition.

DISCS

Brake discs must be inspected for condition. Both rotating and stationary discs in a multiple disc brake can wear. Uneven wear can be an indication that the automatic adjusters may not be pulling the pressure plate back far enough to relieve all pressure on the disc stack. Stationary discs are inspected for cracks. Cracks usually extend from the relief slots, if so equipped. On multiple disc brakes, the slots that key the disc to the torque tube

must also be inspected for wear and widening. The discs should engage the torque tube without binding. The maximum width of the slots is given in the maintenance manual. Cracks or excessive key slot wear are grounds for rejection. Brake wear pads or linings must also be inspected for wear while the brake assembly is removed from the aircraft. Signs of uneven wear should be investigated and the problem corrected. The pads may be replaced if worn beyond limits as long as the stationary disc upon which they mount passes inspection. Follow the manufacturer's procedures for inspections and for pad replacement.

Rotating discs must be similarly inspected. The general condition of the disc must be observed. Glazing can occur when a disc or part of a disc is overheated. It causes brake squeal and chatter. It is possible to resurface a glazed disc if the manufacturer allows it. Rotating discs must also be inspected in the drive key slot or drive tang area for wear and deformation. Little damage is allowed before replacement is required. The pressure plate and back plate on multiple disc brakes must be inspected for freedom of movement, cracks, general condition, and warping. New linings may be riveted to the plates if the old linings are worn and the condition of the plate is good.

NOTE: Replacing brake pads and linings by riveting may require specific tools and technique as described in the maintenance manual to ensure secure attachment. Minor warping can be straightened on some brake assemblies.

AUTOMATIC ADJUSTER PINS

A malfunctioning automatic adjuster assembly can cause the brakes to drag on the rotating disc(s) by not fully releasing and pulling the lining away from the disc. This can lead to excessive, uneven lining wear and disc glazing. The return pin must be straight with no surface damage so it can pass through the grip without binding. Damage under the head can weaken the pin and cause failure. Magnetic inspection is sometimes used to inspect for cracks. The components of the grip and tube assembly must be in good condition. Clean and inspect in accordance with the manufacturer's maintenance instructions. The grip must move with the force specified and must move through its full range of travel.

TORQUE TUBE

A sound torque tube is necessary to hold the brake assembly stable on the landing gear. General visual

inspection should be made for wear, burrs, and scratches. Magnetic particle inspection is used to check for cracks. The key areas should be checked for dimension and wear. All limits of damage are referenced in the manufacturer's maintenance data. The torque tube should be replaced if a limit is exceeded.

BRAKE HOUSING AND PISTON CONDITION

The brake housing must be inspected thoroughly. Scratches, gouges, corrosion, or other blemishes may be dressed out and the surface treated to prevent corrosion. Minimal material should be removed when doing so. Most important is that there are no cracks in the housing. Fluorescent dye penetrant is typically used to inspect for cracks. If a crack is found, the housing must be replaced. The cylinder area(s) of the housing must be dimensionally checked for wear. Limits are specified in the manufacturer's maintenance manual.

The brake pistons that fit into the cylinders in the housing must also be checked for corrosion, scratches, burrs, etc. Pistons are also dimensionally checked for wear limits specified in the maintenance data. Some pistons have insulators on the bottom. They should not be cracked and should be of a minimal thickness. A file can be used to smooth out minor irregularities.

SEAL CONDITION

Brake seals are very important. Without properly functioning seals, brake operation will be compromised or the brakes will fail. Over time, heat and pressure mold a seal into the seal groove and harden the material. Eventually, resilience is reduced and the seal leaks. New seals should be used to replace all seals in the brake assembly.

Acquire seals by part number in a sealed package from a reputable supplier to avoid bogus seals and ensure the correct seals for the brake assembly in question. Check to ensure the new seals have not exceeded their shelf life, which is typically three years from the cure date. Many brakes use back-up rings in the seal groove to support the O-ring seals and reduce the tendency of the seal to extrude into the space which it is meant to seal. These are often made of Teflon® or similar material. Back-up seals are installed on the side of the O-ring away from the fluid pressure. (*Figure 13-112*) They are often reusable.

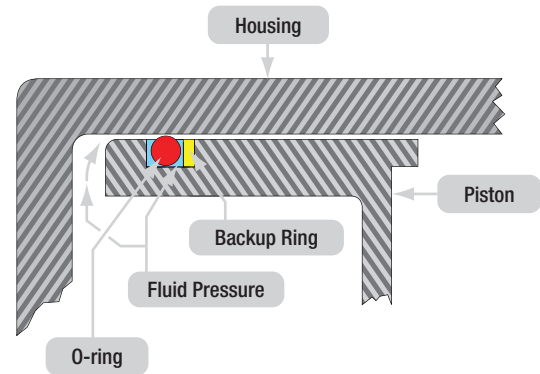


Figure 13-112. Back-up rings are used to keep O-rings from extruding into the space between the piston and the cylinder. They are positioned on the side of the O-ring away from the fluid pressure.

REPLACEMENT OF BRAKE LININGS

In general aviation, replacement of brake linings is commonly done in the hangar. On large aircraft, brake assemblies are typically sent to a repair operation or back shop that specializes in brake overhaul including brake lining replacement. Should the situation arise when immediate replacement of brake linings is required of hangar or line technicians, follow the brake manufacturer's instructions precisely on how to do so.

BRAKE MALFUNCTIONS AND DAMAGE

Aircraft brakes operate under extreme stress and varied conditions. They are susceptible to malfunction and damage. A few common brake problems are discussed in this section.

OVERHEATING

While aircraft brakes slow the aircraft by changing kinetic energy into heat energy, overheating of the brakes is not desirable. Excessive heat can damage and distort brake parts weakening them to the point of failure. Protocol for brake usage is designed to prevent overheating. When a brake shows signs of overheating, it must be removed from the aircraft and inspected for damage. When an aircraft is involved in an aborted takeoff, the brakes must be removed and inspected to ensure they withstood this high level of use. The typical post-overheat brake inspection involves removal of the brake from the aircraft and disassembly of the brakes. All of the seals must be replaced. The brake housing must be checked for cracks, warping, and hardness per the maintenance manual. Any weakness or loss of heat treatment could cause the brake to fail under high pressure braking. The brake discs must also be inspected.

They must not be warped, and the surface treatment must not be damaged or transferred to an adjacent disc. Once reassembled, the brake should be bench tested for leaks and pressure tested for operation before being installed on the aircraft.

DRAGGING

Brake drag is a condition caused by the linings not retracting from the brake disc when the brakes are no longer being applied. It can be caused by several different factors. Brakes that drag are essentially partially on at all times. This can cause excessive lining wear and overheating leading to damage to the disc(s).

A brake may drag when the return mechanism is not functioning properly. This could be due to a weak return spring, the return pin slipping in the auto adjuster pin grip, or similar malfunction. Inspect the auto adjuster(s) and return units on the brake when dragging is reported. An overheated brake that has warped the disc also causes brake drag. Remove the brake and perform a complete inspection as discussed in the previous section. Air in the brake fluid line can also cause brake drag. Heat causes the air to expand, which pushes the brake linings against the disc prematurely. If no damage has been caused when reported, bleed the brakes to remove the air from the system to eliminate the drag. At all times, the technician should perform inspections to ensure the proper parts are used in the brake assembly. Improper parts, especially in the retraction/adjuster assemblies, can cause the brakes to drag.

CHATTERING OR SQUEALING

Brakes may chatter or squeal when the linings do not ride smoothly and evenly along the disc. A warped disc(s) in a multiple brake disc stack produces a condition wherein the brake is actually applied and removed many times per minute. This causes chattering and, at high frequency, it causes squealing. Any misalignment of the disc stack out of parallel causes the same phenomenon. Discs that have been overheated may have damage to the surface layer of the disc. Some of this mix may be transferred to the adjacent disc resulting in uneven disc surfaces that also leads to chatter or squeal. In addition to the noise produced by brake chattering and squealing, vibration is caused that may lead to further damage of the brake and the landing gear system. The technician must investigate all reports of brake chattering and squealing.

AIRCRAFT TIRES AND TUBES

Aircraft tires may be tube type or tubeless. They support the weight of the aircraft while it is on the ground and provide the necessary traction for braking and stopping. The tires also help absorb the shock of landing and cushion the roughness of takeoff, rollout, and taxi operations. Aircraft tires must be carefully maintained to perform as required. They accept a variety of static and dynamic stresses and must do so dependably in a wide range of operating conditions.

TIRE CLASSIFICATION

Aircraft tires are classified in various ways including by: type, ply rating, whether they are tube type or tubeless, and whether they are bias ply tires or radials. Identifying a tire by its dimensions is also used. Each of these classifications is discussed as follows.

TYPES

A common classification of aircraft tires is by type as classified by the United States Tire and Rim Association. While there are nine types of tires, only Types I, III, VII, and VIII, also known as a Three Part Nomenclature tires, are still in production.

- Type I tires are manufactured, but their design is no longer active. They are used on fixed gear aircraft and are designated only by their nominal overall diameter in inches. These are smooth profile tires that are obsolete for use in the modern aviation fleet. They may be found on older aircraft.
- Type III tires are common general aviation tires. They are typically used on light aircraft with landing speeds of 160 miles per hour (mph) or less. Type III tires are relatively low pressure tires that have small rim diameters when compared to the overall width of the tire. They are designed to cushion and provide flotation from a relatively large footprint. Type III tires are designated with a two number system. The first number is the nominal section width of the tire, and the second number is the diameter of the rim the tire is designed to mount upon. (*Figure 13-113*)
- Type VII tires are high performance tires found on jet aircraft. They are inflated to high-pressure and have exceptional high load carrying capability. The section width of Type VII tires is typically narrower than Type III tires. Identification of Type VII aircraft tires involves a two number system. An X is used between the two numbers. The first number designates the nominal overall diameter of the tire.

The second number designates the section width. (*Figure 13-114*)

- Type VIII aircraft tires are also known as three part nomenclature tires. (*Figure 13-115*) They are inflated to very high-pressure and are used on high-performance jet aircraft. The typical Type

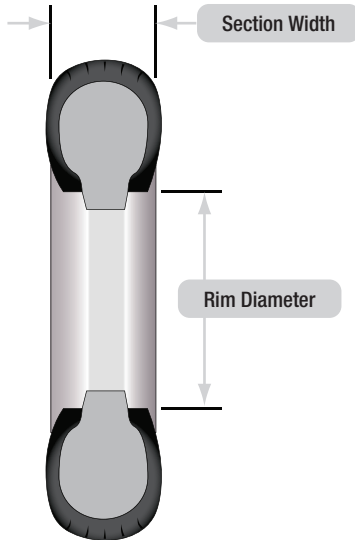


Figure 13-113. Type III aircraft tires are identified via a two-number system with a (-) separating the numbers. The first number is the tire section width in inches. The second number is the rim diameter in inches. For example: 6.00 X 6 is a Cessna 172 tire that is 6.00 inches wide and fits on a rim that has a diameter of 6 inches.

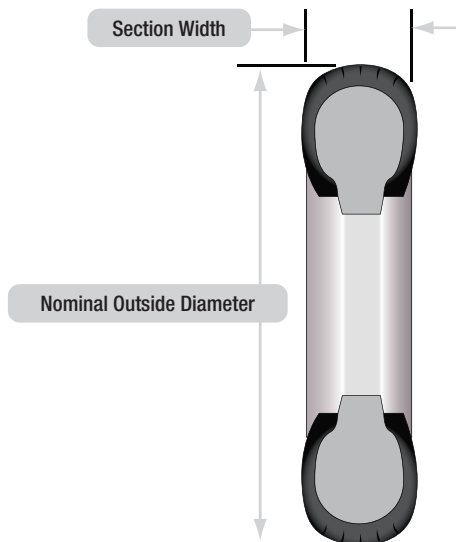


Figure 13-114. A Type VII aircraft tire is identified by its two-number designation. The first number represents the tire's overall diameter in inches and the second number represents the section width in inches. Type VII designators separate the first and second number an "X." For example: 26 x 6.6 identifies a tire that is 26 inches in diameter with a 6.6-inch nominal width.

VIII tire has relatively low profile and is capable of operating at very high speeds and very high loads. It is the most modern design of all tire types. The three part nomenclature is a combination of Type III and Type VII nomenclature where the overall tire diameter, section width, and rim diameter are used to identify the tire. The X and — symbols are used in the same respective positions in the designator.

When three part nomenclature is used on a Type VIII tire, dimensions may be represented in inches or in millimeters. Bias tires follow the designation nomenclature and radial tires replace the "-" with the letter R. For example, 30 x 8.8 R15 designates a Type VIII radial aircraft tire with a 30 inch tire diameter, an 8.8 inch section width to be mounted on a 15 inch wheel rim.

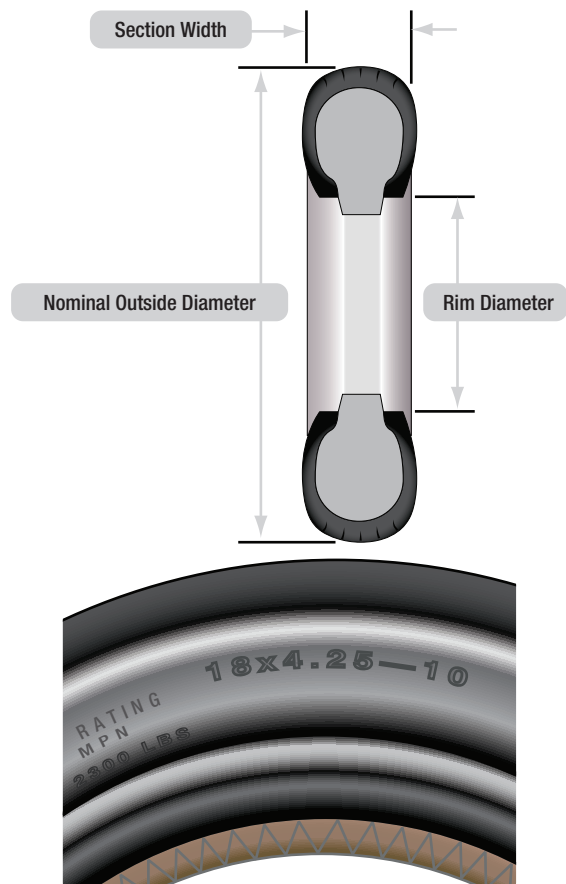


Figure 13-115. A Type VIII or three-part nomenclature tire is identified by 3 parameters: overall diameter, section width, and rim diameter.

They are arranged in that order with the first two separated by an "X" and the second two separated by a "-." For example: 18 X 4.25-10 designates a tire that is 18 inches in diameter with a 4.25-inch section width to be mounted on a 10-inch wheel rim.

A few special designators may also be found for aircraft tires. When a B appears before the identifier, the tire has a wheel rim to section width ratio of 60 to 70 percent with a bead taper of 15 degrees. When an H appears before the identifier, the tire has a 60 to 70 percent wheel rim to section width ratio but a bead taper of only 5 degrees.

PLY RATING

Tire plies are reinforcing layers of fabric encased in rubber that are laid into the tire to provide strength. In early tires, the number of plies used was directly related to the load the tire could carry. Nowadays, refinements to tire construction techniques and the use of modern materials to build up aircraft tires makes the exact number of plies somewhat irrelevant when determining the strength of a tire. However, a ply rating is used to convey the relative strength of an aircraft tire. A tire with a high ply rating is a tire with high strength able to carry heavy loads regardless of the actual number of plies used in its construction.

TUBE TYPE OR TUBELESS

As stated, aircraft tires can be tube type or tubeless. This is often used as a means of tire classification. Tires that are made to be used without a tube inserted inside have an inner liner specifically designed to hold air. Tube type tires do not contain this inner liner since the tube holds the air from leaking out of the tire. Tires that are meant to be used without a tube have the word tubeless on the sidewall. If this designation is absent, the tire requires a tube. Consult the aircraft manufacturer's maintenance information for any allowable tire damage and the use of a tube in a tubeless tire.

BIAS PLY OR RADIAL

Another means of classifying an aircraft tire is by the direction of the plies used in construction of the tire, either bias or radial. Traditional aircraft tires are bias ply tires. The plies are wrapped to form the tire and give it strength. The angle of the plies in relation to the direction of rotation of the tire varies between 30° and 60°. In this manner, the plies have the bias of the fabric from which they are constructed facing the direction of rotation and across the tire. Hence, they are called bias tires. The result is flexibility as the sidewall can flex with the fabric plies laid on the bias. (*Figure 13-116*)

Some modern aircraft tires are radial tires. The plies in radial tires are laid at a 90° angle to the direction of rotation of the tire. This configuration puts the non-stretchable fiber of the plies perpendicular to the sidewall and direction of rotation. This creates strength in the tire allowing it to carry high loads with less deformation. (*Figure 13-117*)

TIRE CONSTRUCTION

An aircraft tire is constructed for the purpose it serves. Unlike an automobile or truck tire, it does not have to carry a load for a long period of continuous operation. However, an aircraft tire must absorb the high impact loads of landing and be able to operate at high speeds even if only for a short time. The deflection built into an aircraft tire is more than twice that of an automobile tire. This enables it to handle the forces during landings

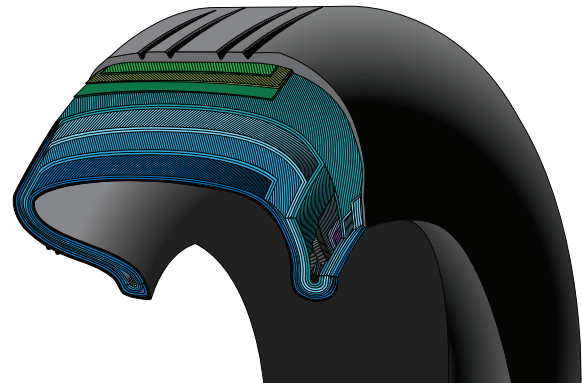


Figure 13-116. A bias ply tire has the fabric bias oriented with and across the direction of rotation and the sidewall. Since fabric can stretch on the bias, the tire is flexible, and can absorb loads. Strength is obtained by adding plies.

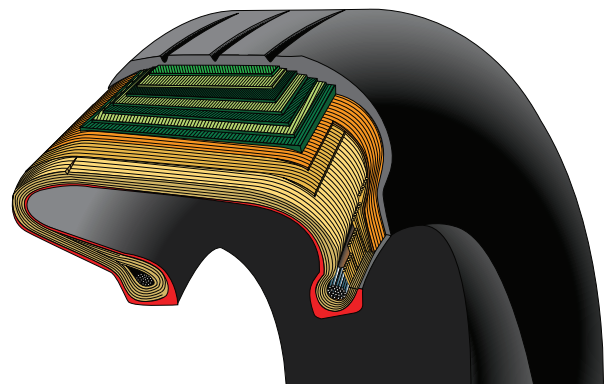


Figure 13-117. A radial tire has the fiber strands of the ply fabric oriented with and at 90 degrees to the direction of rotation and the tire sidewall. This restricts flexibility directionally and the flexibility of the sidewall while it strengthens the tire to carry heavy loads.

without being damaged. Only tires designed for an aircraft as specified by the manufacturer should be used. It is useful to the understanding of tire construction to identify the various components of a tire and the functions contributed to the overall characteristics of a tire. Refer to **Figure 13-118** for tire nomenclature used in this discussion.

BEAD

The tire bead is an important part of an aircraft tire. It anchors the tire carcass and provides a dimensioned, firm mounting surface for the tire on the wheel rim. Tire beads are strong. They are typically made from high-strength carbon steel wire bundles encased in rubber. One, two, or three bead bundles may be found on each side of the tire depending on its size and the load it is designed to handle. Radial tires have a single bead bundle on each side of the tire. The bead transfers the impact loads and deflection forces to the wheel rim. The bead toe is closest to the tire centerline and the bead heel fit against the flange of the wheel rim.

An apex strip is additional rubber formed around the bead to give a contour for anchoring the ply turn-ups. Layers of fabric and rubber called flippers are placed around the beads to insulate the carcass from the beads and improve tire durability. Chafer strips made of fabric or rubber are laid over the outer carcass plies after the plies are wrapped around the beads. The chafers

protect the carcass from damage during mounting and demounting of the tire. They also help reduce the effects of wear and chafing between the wheel rim and the tire bead especially during dynamic operations.

CARCASS PLYS

Carcass plies, or casing plies as they are sometimes called, are used to form the tire. Each ply consists of fabric, usually nylon, sandwiched between two layers of rubber. The plies are applied in layers to give the tire strength and form the carcass body of the tire. The ends of each ply are anchored by wrapping them around the bead on both sides of the tire to form the ply turn-ups. As mentioned, the angle of the fiber in the ply is manipulated to create a bias tire or radial tire as desired. Typically, radial tires require fewer plies than bias tires.

Once the plies are in place, bias tires and radial tires each have their own type of protective layers on top of the plies but under the tread of the running surface of the tire. On bias tires, these single or multiple layers of nylon and rubbers are called tread reinforcing plies. On radial tires, an under tread and a protector ply do the same job. These additional plies stabilize and strengthen the crown area of the tire. They reduce tread distortion under load and increase stability of the tire at high speeds. The reinforcing plies and protector plies also help resist puncture and cutting while protecting the carcass body of the tire.

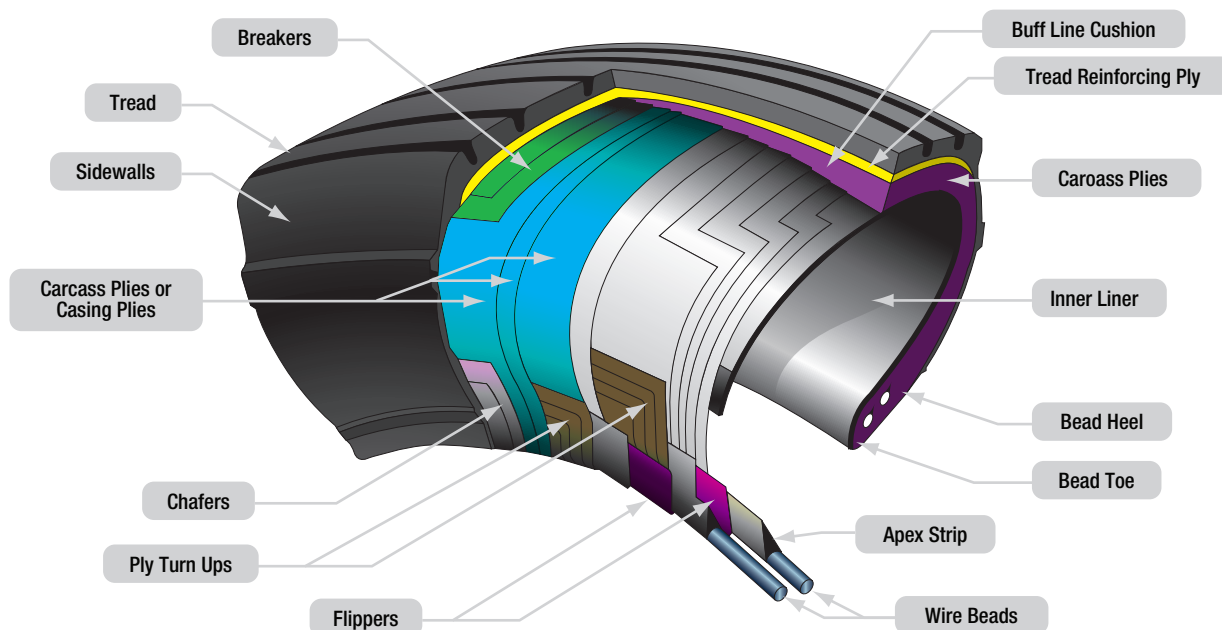


Figure 13-118. Construction nomenclature of an aircraft tire.



Figure 13-119. Aircraft tire treads are designed for different uses. A is a rib tread designed for use on paved surfaces. It is the most common aircraft tire tread design. B is a diamond tread designed for unpaved runways. C is an all weather tread that combines a ribbed center tread with a diamond tread pattern of the edges. D is a smooth tread tire found on older, slow aircraft without brakes designed for stopping. E is a chine tire used on the nose gear of aircraft with fuselage mounted jet engines to deflect runway water away from the engine intake(s).

TREAD

The tread is the crown area of the tire designed to come in contact with the ground. It is a rubber compound formulated to resist wear, abrasion, cutting, and cracking. It also is made to resist heat buildup. Most modern aircraft tire tread is formed with circumferential grooves that create tire ribs. The grooves provide cooling and help channel water from under the tire in wet conditions to increase adhesion to the ground surface. Tires designed for aircraft frequently operated from unpaved surfaces may have some type of cross-tread pattern. Older aircraft without brakes or brakes designed only to aid in taxi may not have any grooves in the tread. An all-weather tread may be found on some aircraft tires. This tread has typical circumferential ribs in the center of the tire with a diamond patterned cross tread at the edge of the tire. (*Figure 13-119*)

The tread is designed to stabilize the aircraft on the operating surface and wears with use. Many aircraft tires are designed with protective undertread layers as described above. Extra tread reinforcement is sometimes accomplished with breakers. These are layers of nylon cord fabric under the tread that strengthen the tread while protecting the carcass plies. Tires with reinforced tread are often designed to be retreaded and used again once the tread has worn beyond limits. Consult the tire manufacturer's data for acceptable tread wear and retread capability for a particular tire.

SIDEWALL

The sidewall of an aircraft tire is a layer of rubber designed to protect the carcass plies. It may contain compounds designed to resist the negative effects of

ozone on the tire. It also is the area where information about the tire is contained. The tire sidewall imparts little strength to the cord body. Its main function is protection.

The inner sidewall of a tire is covered by the tire inner liner. A tube type tire has a thin rubber liner adhered to the inner surface to prevent the tube from chafing on the carcass plies. Tubeless tires are lined with a thicker, less permeable rubber. This replaces the tube and contains the nitrogen or inflation air within the tire and keeps from seeping through the carcass plies. The inner liner does not contain 100 percent of the inflation gas. Small amounts of nitrogen or air seep through the liner into the carcass plies. This seepage is released through vent holes in the lower outer sidewall of the tires. These are typically marked with a green or white dot of paint and must be kept unobstructed. Gas trapped in the plies could expand with temperature changes and cause separation of the plies, thus weakening the tire leading to tire failure. Tube type tires also have seepage holes in the sidewall to allow air trapped between the tube and the tire to escape. (*Figure 13-120*)

CHINE

Some tire sidewalls are mounded to form a chine. A chine is a special built-in deflector used on nose wheels of certain aircraft, usually those with fuselage mounted engines. The chine diverts runway water to the side and away from the intake of the engines. (*Figure 13-119E*) Tires with a chine on both sidewalls are produced for aircraft with a single nose wheel.



Figure 13-120. A sidewall vent marked by a colored dot must be kept free from obstruction to allow trapped air or nitrogen to escape from the carcass plies of the tire.

TIRE INSPECTION ON THE AIRCRAFT

Tire condition is inspected while mounted on the aircraft on a regular basis. Inflation pressure, tread wear and condition, and sidewall condition are continuously monitored to ensure proper tire performance.

INFLATION

To perform as designed, an aircraft tire must be properly inflated. The aircraft manufacturer's maintenance data must be used to ascertain the correct inflation pressure for a tire on a particular aircraft. Do not inflate to a pressure displayed on the sidewall of the tire or by how the tire looks. Tire pressure is checked while under load and is measured with the weight of the aircraft on the wheels. Loaded versus unloaded pressure readings can vary as much as 4 percent. Tire pressure measured with the aircraft on jacks or when the tire is not installed is lower due to the larger volume of the inflation gas space inside of the tire. On a tire designed to be inflated to 160 psi, this can result in a 6.4 psi error. A calibrated pressure gauge should always be used to measure inflation pressure. Digital and dial type pressure gauges are more consistently accurate and preferred. (*Figure 13-121*)

Aircraft tires disperse the energy from landing, rollout, taxi, and takeoff in the form of heat. As the tire flexes, heat builds and is transferred to the atmosphere, as well as to the wheel rim through the tire bead. Heat from braking also heats the tire externally. A limited amount of heat is able to be handled by any tire beyond which structural damage occurs.

An improperly inflated aircraft tire can sustain internal damage that is not readily visible and that can lead to tire failure. Tire failure upon landing is always dangerous. An aircraft tire is designed to flex and absorb the shock

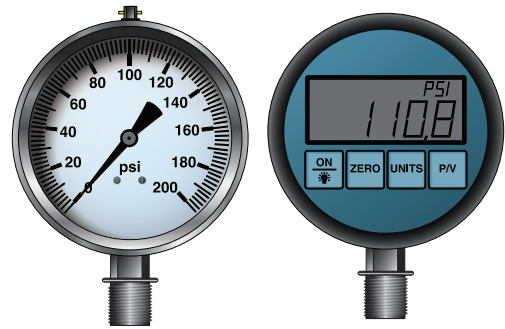


Figure 13-121. A calibrated bourdon tube dial-type pressure gauge or a digital pressure gauge is recommended for checking tire pressure.

of landing. Temperature rises as a result. However, an under inflated tire may flex beyond design limits of the tire. This causes excessive heat build-up that weakens the carcass construction. To ensure tire temperature is maintained within limits, tire pressure must be checked and maintained within the proper range on a daily basis or before each flight if the aircraft is only flown periodically. Tire pressure should be measured at ambient temperature. Fluctuations of ambient temperature greatly affect tire pressure and complicate maintenance of pressure within the allowable range for safe operation. Tire pressure typically changes 1 percent for every 2.8°C of temperature change. When aircraft are flown from one environment to another, ambient temperature differences can be vast. Maintenance personnel must ensure that tire pressure is adjusted accordingly.

For example, an aircraft with the correct tire pressure departing Phoenix, Arizona where the ambient temperature is 38°C arrives in Vail, Colorado where the temperature is 10°C. The 28°C difference in ambient temperature results in a 10 percent reduction in tire pressure. Therefore the aircraft could land with under inflated tires that may be damaged due to over temperature from flexing beyond design limits as described above. An increase in tire pressure before takeoff in Phoenix, Arizona prevents this problem as long as the tires are not inflated beyond the allowable limit provided in the maintenance data.

When checking tire pressure, allow 3 hours to elapse after a typical landing to ensure the tire has cooled to ambient temperature. The correct tire pressure for each ambient temperature is typically provided by the manufacturer on a table or graph.

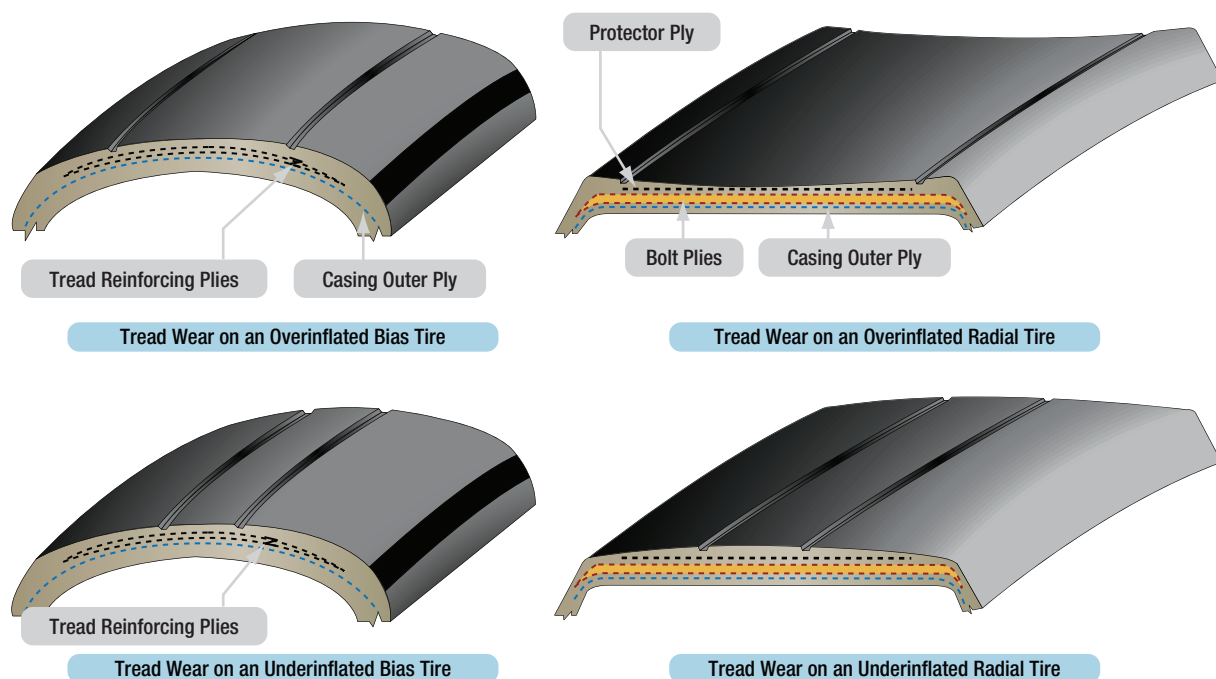


Figure 13-122. Tires that are overinflated lack adherence to the runway and develop excess tread wear in the center of the tread. Tires that are under inflated develop excess tread wear on the tire shoulders. Overheating resulting in internal carcass damage and potential failure are possible from flexing the tire beyond design limits.

In addition to overheating, under inflated aircraft tires wear unevenly, which leads to premature tire replacement. They may also creep or slip on the wheel rim when under stress or when the brakes are applied. Severely under inflated tires can pinch the sidewall between the rim and the runway causing sidewall and rim damage. Damage to the bead and lower sidewall area are also likely. This type of abuse like any over flexing damages the integrity of the tire and it must be replaced. In dual-wheel setups, a severely under inflated tire affects both tires and both should be replaced. Over inflation of aircraft tires is another undesirable condition. While carcass damage due to overheating does not result, adherence to the landing surface is reduced. Over a long period of time, over inflation leads to premature tread wear. Therefore, over inflation reduces the number of cycles in service before the tire must be replaced. It makes the tire more susceptible to bruises, cutting, shock damage, and blowout. (*Figure 13-122*)

TREAD CONDITION

Condition of an aircraft tire tread is able to be determined while the tire is inflated and mounted on the aircraft. The following is a discussion of some of the tread conditions and damage that the technician may encounter while inspecting tires.

Tread Depth and Wear Pattern

Evenly worn tread is a sign of proper tire maintenance. Uneven tread wear has a cause that should be investigated and corrected. Follow all manufacturer instructions specific to the aircraft when determining the extent and serviceability of a worn tire. In the absence of this information, remove any tire that has been worn to the bottom of a tread groove along more than 1/8 of the circumference of the tire. If either the protector ply on a radial tire or the reinforcing ply on a bias tire is exposed for more than 1/8 of the tire circumference, the tire should also be removed. A properly maintained evenly worn tire usually reaches its wear limits at the centerline of the tire. (*Figure 13-123*)

Asymmetrical tread wear may be caused by the wheels being out of alignment. Follow the manufacturer's instructions while checking caster, camber, tow-in, and tow-out to correct this situation. Occasionally, asymmetrical tire wear is a result of landing gear geometry that cannot, or is not, required to be corrected. It may also be caused by regular taxiing on a single engine or high speed cornering while taxiing. It is acceptable to remove the tire from the wheel rim, turn it around, and remount it to even up tread wear if the tire passes all other criterion of inspection for serviceability. Removal of a tire before it is worn beyond limits to

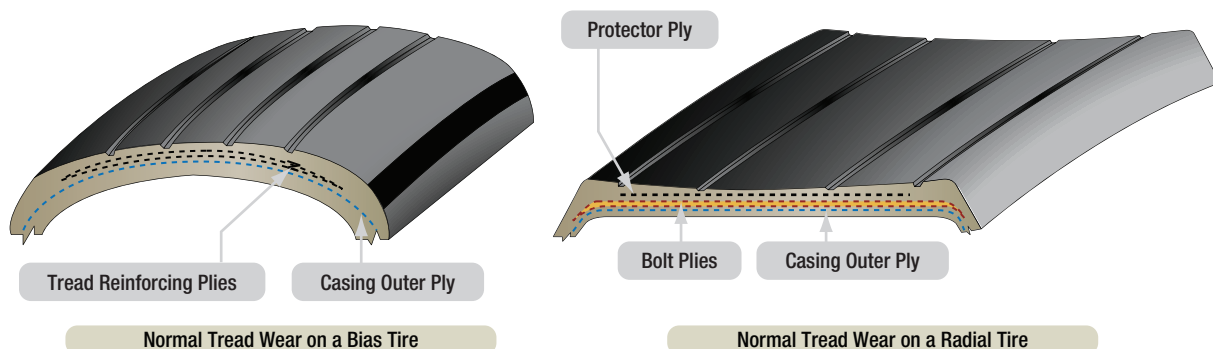


Figure 13-123. Normal tire wear.

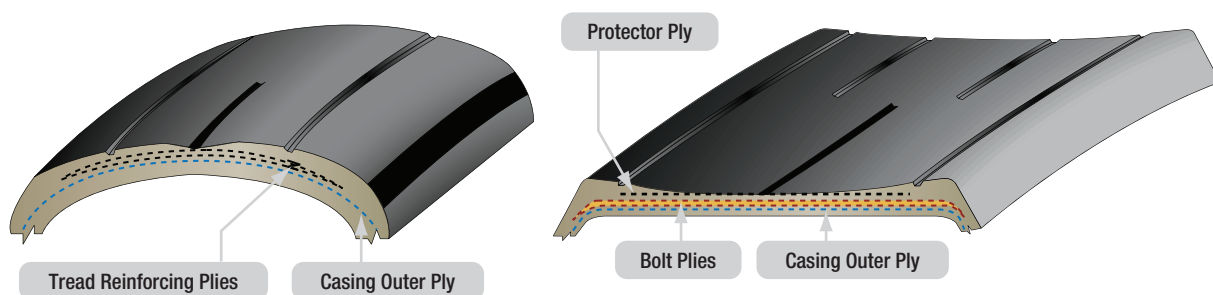


Figure 13-124. Tread wear on a bias ply tire (left) and a radial tire (right) show wear beyond limits of serviceability but still eligible to be retreaded.

be eligible for retreading is cost effective and good maintenance practice. Considerable traction is lost when tire tread is severely worn and must also be considered when inspecting a tire for condition. (Figure 13-124) Consult airframe manufacturer and tire manufacturer specifications for wear and retread limitations.

Tread Damage

In addition to tread wear, an aircraft tire should be inspected for damage. Cuts, bruises, bulges, imbedded foreign objects, chipping, and other damage must be within limits to continue the tire in service. Some acceptable methods of dealing with this type of damage are described below. All damage, suspected damage, and areas with leaks should be marked with chalk, a wax marker, paint stick, or other device before the tire is deflated or removed. Often, it is impossible to relocate these areas once the tire is deflated. Tires removed for retread should be marked in damaged areas to enable closer inspection of the extent of the damage before the new tread is installed. (Figure 13-125)

Foreign objects imbedded in a tire's tread are of concern and should be removed when not imbedded beyond the tread. Objects of questionable depth should only be removed after the tire has been deflated. A blunt

awl or appropriately sized screwdriver can be used to pry the object from the tread. Care must be exercised to not enlarge the damaged area with the removal tool. (Figure 13-126)



Figure 13-125. Marking of damaged area to enable closer inspection.

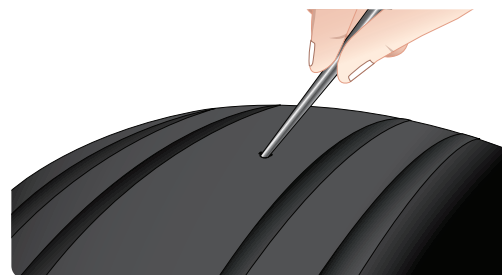


Figure 13-126. Deflate a tire before removing or probing any area where a foreign object is lodged.

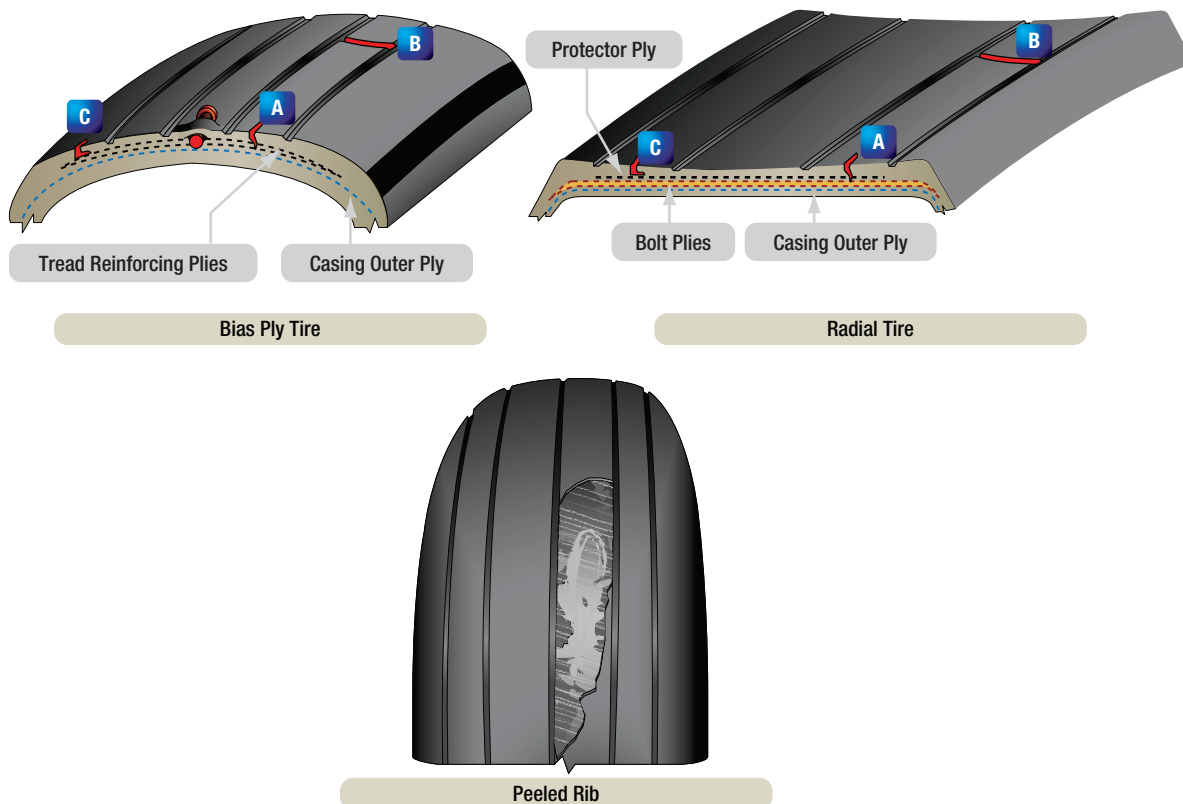


Figure 13-127. Remove an aircraft tire from service when the depth of a cut exposes the casing outer plies of a bias ply tire or the outer belt layer of a radial tire (A); a tread rib has been severed across the entire width (B); or, when undercutting occurs at the base of any cut (C). These conditions may lead to a peeled rib.

Once removed, assess the remaining damage to determine if the tire is serviceable. A round hole caused by a foreign object is acceptable only if it is 3/8 inch or less in diameter. Embedded objects that penetrate or expose the casing cord body of a bias ply tire or the tread belt layer of a radial tire cause the tire to become unairworthy and it must be removed from service. Cuts and tread undercutting can also render a tire unairworthy. A cut that extends across a tread rib is cause for tire removal. These can sometimes lead to a section of the rib to peel off the tire. (*Figure 13-127*)

Consult the aircraft maintenance manual, airline operations manual, or other technical documents applicable to the aircraft tire in question. A flat spot on a tire is the result of the tire skidding on the runway surface while not rotating. This typically occurs when the brakes lock on while the aircraft is moving. If the flat spot damage does not expose the reinforcing ply of a bias tire or the protector ply of a radial tire, it may remain in service. However, if the flat spot causes vibration, the tire must be removed. Landing with a brake applied can often cause a severe flat spot that exposes the tire

under tread. It can also cause a blowout. The tire must be replaced in either case. (*Figure 13-128*)

A bulge or separation of the tread from the tire carcass is cause for immediate removal and replacement of the tire. Mark the area before deflation as it could easily become undetectable without air in the tire. (*Figure 13-129*)

Operation on a grooved runway can cause an aircraft tire tread to develop shallow chevron shaped cuts. These cuts are allowed for continued service, unless chunks or cuts into the fabric of the tire result. Deep chevrons



Figure 13-128. Landing with the brake on causes a tire flat spot that exposes the under tread and requires replacement of the tire.



Figure 13-129. Bulges and tread separation are cause for removal of a tire from service.

that cause a chunk of the tread to be removed should not expose more than 1 square inch of the reinforcing or protector ply. Consult the applicable inspection parameters to determine the allowable extent of chevron cutting. (*Figure 13-130*)

Tread chipping and chunking sometimes occurs at the edge of the tread rib. Small amounts of rubber lost in this way are permissible. Exposure of more than 1 square inch of the reinforcing or protector ply is cause for removal of the tire. (*Figure 13-131*)

Cracking in a tread groove of an aircraft tire is generally not acceptable if more than 1/4 inch of the reinforcing or protector ply is exposed. Groove cracks can lead to undercutting of the tread, which eventually can cause the entire tread to be thrown from the tire. (*Figure 13-132*)

Oil, hydraulic fluid, solvents, and other hydrocarbon substances contaminate tire rubber, soften it, and make it spongy. A contaminated tire must be removed from service. If any volatile fluids come in contact with the tire, it is best to wash the tire or area of the tire with

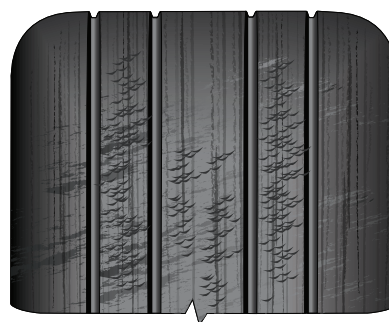


Figure 13-130. Chevron cuts in a tire are caused by operation on grooved runway surfaces. Shallow chevron cuts are permitted on aircraft tires.

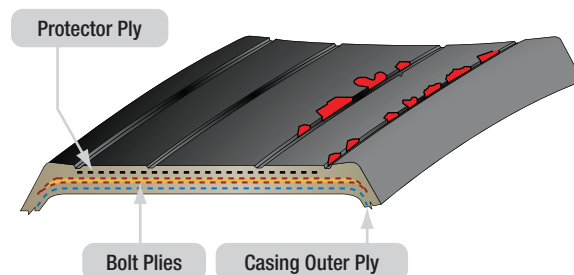


Figure 13-131. Tread chipping and chunking of a tire requires that the tire be removed from service if more than 1 square inch of the reinforcing ply or protector ply is exposed.

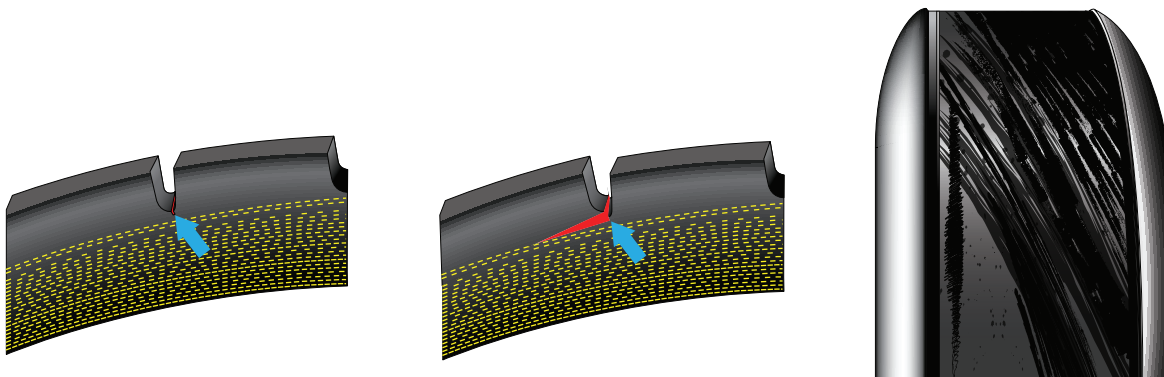


Figure 13-132. A thrown tread can result from a groove crack or tread undercutting and must be removed from service.

denatured alcohol followed by soap and water. Protect tires from contact with potentially harmful fluids by covering tires during maintenance in the landing gear area. Tires are also subject to degradation from ozone and weather. Tires on aircraft parked outside for long periods of time can be covered for protection from the elements. (*Figure 13-133*)

SIDEWALL CONDITION

The primary function of the sidewall of an aircraft tire is protection of the tire carcass. If the sidewall cords are exposed due to a cut, gouge, snag, or other injury, the tire must be replaced. Mark the area of concern before removal of the tire. Damage to the sidewall that does not reach the cords is typically acceptable for service. Circumferential cracks or slits in the sidewall are unacceptable. A bulge in a tire sidewall indicates possible delamination of the sidewall carcass plies. The tire must immediately be removed from service.

Weather and ozone can cause cracking and checking of the sidewall. If this extends to the sidewall cords, the tire must be removed from service. Otherwise, sidewall checking as show in *Figure 13-134* does not affect the performance of the tire and it may remain in service.

TIRE REMOVAL

Removal of any tire and wheel assembly should be accomplished following all aircraft manufacturer's instructions for the procedure. Safety procedures are designed for the protection of the technician and the maintenance of aircraft parts in serviceable condition. Follow all safety procedures to prevent personal injury and damage to aircraft parts and assemblies. An aircraft tire and wheel assembly, especially a high pressure assembly that has been damaged or overheated, should be treated as though it may explode. Never approach such a tire while its temperature is still elevated above ambient temperature. Once cooled, approach a damage tire and wheel assembly from an oblique angle advancing toward the shoulder of the tire. (*Figure 13-135*)

Deflate all unserviceable and damaged tires before removal from the aircraft. Use a valve core/deflation tool to deflate the tire. Stand to the side - away from the projectile path of the valve core. A dislodged valve core propelled by internal tire pressure can cause serious human injury.



Figure 13-133. Cover tires to protect from harmful chemicals and from the elements when parked outside for long periods of time.



Figure 13-134. Cracking and checking in the sidewall of a tire is acceptable for service as long as it does not extend to or expose the sidewall carcass plies.

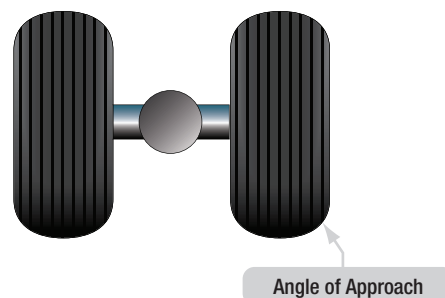


Figure 13-135. To avoid potential injury, approach a tire/wheel assembly that has damage or has been overheated at an angle toward the tire shoulder only after it has cooled to ambient temperature.

When completely deflated, remove the valve core. (*Figure 13-136*) A tire and wheel assembly in airworthy condition may be removed to access other components for maintenance without deflating the tire. This is common practice, such as when accessing the brake when the wheel assembly is immediately reinstalled. For



Figure 13-136. The tire valve core should be removed after the tire is completely deflated and before the tire and wheel assembly is removed from the aircraft.

tracking purposes, ensure damaged areas of a tire are marked before deflation. Record all known information about an unserviceable tire and attach it to the tire for use by the retread repair station.

Once removed from the aircraft, a tire must be separated from the wheel rim upon which is mounted. Proper equipment and technique should be followed to avoid damage to the tire and wheel. The wheel manufacturer's maintenance information is the primary source for dismounting guidelines. The bead area of the tire sits firmly against the rim shoulder and must be broken free. Always use proper bead breaking equipment for this purpose. Never pry a tire from a wheel rim as damage to the wheel is inevitable. The wheel tie bolts must remain installed and fully tightened when the bead is broken from the rim to prevent damage to the wheel half mating surfaces.

When the bead breaking press contact surface is applied to the tire, it should be as close to the wheel as possible without touching it during the entire application of pressure. Tires and rims of different sizes require contact pads suitable for the tire. Hand presses and hydraulic presses are available. Apply the pressure and hold it to allow the bead to move on the rim. Gradually

progress around the rim until the tire bead is broken free. Ring type bead breakers apply pressure around the circumference of the entire sidewall so rotation is not required. (Figure 13-137) Once the bead is broken free, the wheel halves may be disassembled. (Figure 13-138)

Radial tires have only one bead bundle on each side of the tire. The sidewall is more flexible in this area than a bias ply tire. The proper tooling should be used, and pressure should be applied slowly to avoid heavy distortion of the sidewall. Lubrication may be applied and allowed to soak into the tire-wheel interface. Only soapy tire solutions should be used. Never apply a hydrocarbon based lubricant to an aircraft tire as this contaminates the rubber compound used to construct the tire. Beads on tube type and tubeless tires are broken free in a similar manner.

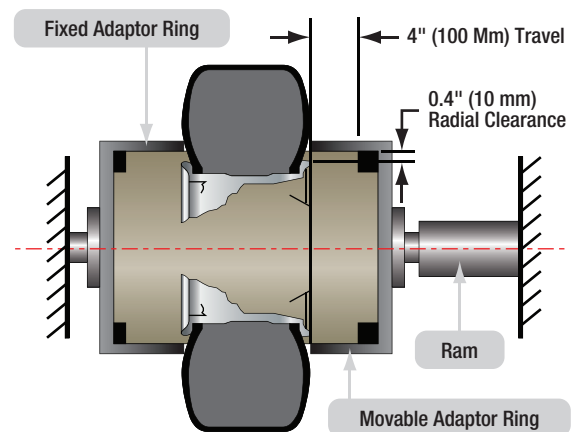


Figure 13-137. A ring adapter applies pressure around the entire circumference of the lower sidewall of the tire to break the bead free from the wheel rim. The diameter of the adapter must be correct for the tire and the travel limited so as to not injure the tire.



Figure 13-138. An electrohydraulic tire bead breaker (left) used on large tires and a manual tire bead breaker (right) used on small tires.

TIRE INSPECTION OFF AIRCRAFT

Once a tire has been removed from the wheel rim, it should be inspected for condition. It may be possible to retread the tire at an approved repair station and return it to service. A sequential inspection procedure helps ensure no parts of the tire are overlooked. Mark and record the extent of all damage. Follow manufacturer's guidelines for inspection. Tires must only be repaired by those with the experience and equipment to do so. Most tire repairs are accomplished at a certified tire repair facility.

When inspecting a tire removed from the aircraft, pay special attention to the bead area since it must provide an air tight seal to the wheel rim and transfer forces from the tire to the rim. Inspect the bead area closely as it is where the heat is concentrated during tire operation. Surface damage to the chafer is acceptable and can be repaired when the tire is retreaded. Other damage in the bead area is usually cause for rejection. Damage to the turnups, ply separation at the bead, or a kinked bead are examples of bead area damage that warrant the tire be discarded. The bead area of the tire may sustain damage or have an altered appearance or texture on a tire that has been overheated. Consult a certified tire repair station or retread facility when in doubt about the condition observed. The wheel rim must also be inspected for damage. An effective seal without slippage, especially on tubeless tires, is dependent on the condition and integrity of the wheel in the bead seat area.

Overheating of a tire weakens it even though the damage might not be obvious. Any time a tire is involved in an aborted takeoff, severe braking, or the thermal plug in the wheel has melted to deflate the tire before explosion, the tire must be removed. On a dual installation, both tires must be removed. Even if only one tire shows obvious damage or deflates, the loads experienced by the mate are excessive. Internal damage such as ply separation, is likely. The history of having been through an overheat event is all that is required for the tire to be discarded.

Damaged or suspected damaged areas of the tire should be re-inspected while the tire is off the aircraft. Cuts can be probed to check for depth and extent of damage below the tread. In general, damage that does not exceed 40 percent of the tire plies can be repaired when the tire is retreaded. Small punctures with a diameter on the

tire inner surface of less than 1/8 inch and a diameter on the outer surface of less than 1/4 inch can also be repaired and retreaded. A bulge caused by ply separation is reason to discard a tire. However, a bulge caused by tread separation from the tire carcass may be repairable during retread. Exposed sidewall cord or sidewall cord damage is unacceptable and the tire cannot be repaired or retreaded. Consult the tire manufacturer or certified retreader for clarification on damage to a tire.

TIRE REPAIR AND RETREADING

The technician should follow airframe and tire manufacturer instructions to determine if a tire is repairable. Many example guidelines have also been given in this section. Nearly all tire repairs must be made at a certified tire repair facility equipped to perform the approved repair. Bead damage, ply separation, and sidewall cord exposure all require that the tire be scrapped. Inner liner condition on tubeless tires is also critical. Replacing the tube in a tube type tire is performed by the technician as are mounting and balancing all types of aircraft tires.

Aircraft tires are very expensive. They are also extremely durable. The effective cost of a tire over its life can be reduced by having the tread replaced while the carcass is still sound and injuries are within repairable limits. Certified tire retread repair stations, often the original equipment manufacturer (OEM), do this work. The technician inspects a tire to prequalify it for retread so that the cost of shipping it to the retread repair facility is not incurred if there is no chance to retread the tire. The tire retreader inspects and tests every tire to a level beyond the capability of the hangar or line technician. Shearography, an optical nondestructive testing method that provides detailed information about the internal integrity of the tire, is used by tire retread repair facilities to ensure a tire carcass is suitable for continued service.

Tires that are retreaded are marked as such. They are not compromised in strength and give the performance of a new tire. No limits are established for the number of times a tire can be retreaded. This is based on the structural integrity of the tire carcass. A well maintained main gear tire may be able to be retreaded a handful of times before fatigue renders the carcass unairworthy. Some nose tires can be retread nearly a dozen times.

TIRE STORAGE

An aircraft tire can be damaged if stored improperly. A tire should always be stored vertically so that it is resting on its treaded surface. Horizontal stacking of tires is not recommended. Storage of tires on a tire rack with a minimum 3/4-inch flat resting surface for the tread is ideal and avoids tire distortion.

If horizontal stacking of tires is necessary, it should only be done for a short time. The weight of the upper tires on the lower tires cause distortion possibly making it difficult for the bead to seat when mounting tubeless tires. A bulging tread also stresses rib grooves and opens the rubber to ozone attack in this area. (*Figure 13-139*)

Never stack aircraft tires horizontally for more than 6 months. Stack no higher than four tires if the tire is less than 40-inches in diameter and no higher than three tires if greater than 40-inches in diameter. The environment in which an aircraft tire is stored is critical. The ideal location in which to store an aircraft tire is cool, dry, and dark, free from air currents and dirt.

An aircraft tire contains natural rubber compounds that are prone to degradation from chemicals and sunlight. Ozone (O_3) and oxygen (O_2) cause degradation of tire compounds. Tires should be stored away from strong air currents that continually present a supply of one or both of these gases. Fluorescent lights, mercury vapor lights, electric motors, battery chargers, electric welding equipment, electric generators, and similar shop equipment produce ozone and should not be operated near aircraft tires. Mounted inflated tires can be stored with up to 25 percent less pressure than operating pressure to reduce vulnerability from ozone attacks. Sodium vapor lighting is acceptable. Storage of an aircraft

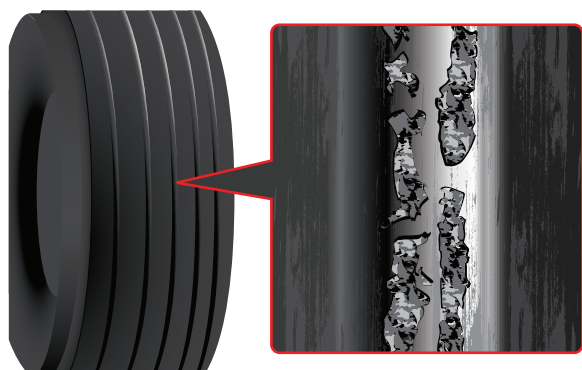


Figure 13-139. Ozone cracking in a tire tread groove is facilitated by horizontal stacking.

tire in the dark is preferred to minimize degradation from ultraviolet (UV) light. If this is not possible, wrap the tire in dark polyethylene or paper to form an ozone barrier and to minimize exposure to UV light.

Common hydrocarbon chemicals, such as fuels, oils, and solvents, should not contact a tire. Avoid rolling tires through spills on the hangar or shop floor and be sure to clean any tire immediately if contaminated. Dry the tire and store all tires in a dry place away from any moisture, which has a deteriorating effect on the rubber compounds. Moisture with foreign elements may further damage the rubber and fabric of a tire. Dirty areas must be avoided.

Tires are made to operate in a wide range of temperatures. However, storage should be at cool temperatures to minimize degradation. A general range for safe aircraft tire storage is between 0°C and 40°C. Temperatures below this are acceptable but higher temperatures must be avoided.

AIRCRAFT TUBES

Many aircraft tires accept a tube inside to contain the inflation air. Tube type tires are handled and stored in similar fashion as tubeless tires. A number of issues concerning the tubes themselves must be addressed.

TUBE CONSTRUCTION AND SELECTION

Aircraft tire tubes are made of a natural rubber compound. They contain the inflation air with minimal leakage. Unreinforced and special reinforced heavy duty tubes are available. The heavy duty tubes have nylon reinforcing fabric layered into the rubber to provide strength to resist chafing and to protect against heat such as during braking. Tubes come in a wide range of sizes. Only the tube specified for the applicable tire size must be used. Tubes that are too small stress the tube construction.

TUBE STORAGE AND INSPECTION

An aircraft tire tube should be kept in the original carton until put into service to avoid deterioration through exposure to environmental elements. If the original carton is not available, the tube can be wrapped in several layers of paper to protect it. Alternately, for short time periods only, a tube may be stored in the correct size tire it is made for while inflated just enough to round out the tube. Application of talc to the inside

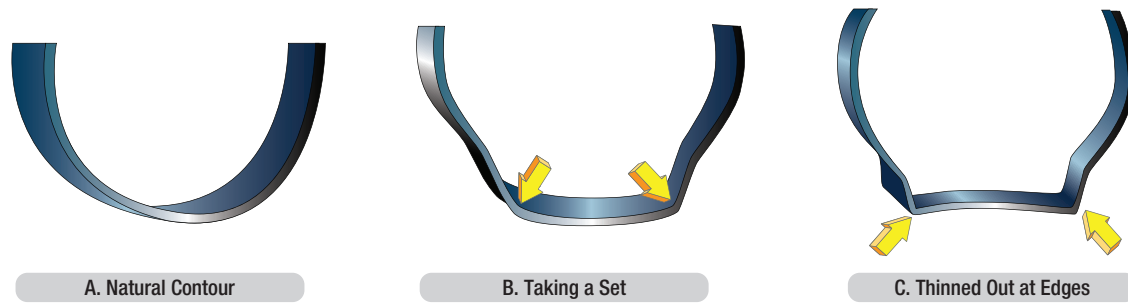


Figure 13-140. During inspection, an aircraft tire tube should retain its natural contour. Tubes with thinned areas or that have taken a set should be discarded and replaced.

of the tire and outside of the tube prevents sticking. Remove the tube and inspect it and the tire before permanently mounting the assembly. Regardless of storage method, always store aircraft tubes in a cool, dry, dark place away from ozone producing equipment and moving air. When handling and storing aircraft tire tubes, creases are to be avoided. These weaken the rubber and eventually cause tube failure. Creases and wrinkles also tend to be chafe points for the tube when mounted inside the tire. Never hang a tube over a nail or peg for storage.

An aircraft tube must be inspected for leaks and damage that may eventually cause a leak or failure. To check for leaks, remove the tube from the tire. Inflate the tube just enough to have it take shape but not stretch. Immerse a small tube in a container of water and look for the source of air bubbles. A large tube may require that water be applied over the tube. Again look for the source of bubbles. The valve core should also be wetted to inspect it for leaks.

There is no mandatory age limit for an aircraft tire tube. It should be elastic without cracks or creases in order to be considered serviceable. The valve area is prone to damage and should be inspected thoroughly. Bend the valve to ensure there are no cracks at the base where it is bonded to the tire or in the area where it passes through the hole in the wheel rim. Inspect the valve core to ensure it is tight and that it does not leak. If an area of a tube experiences chafing to the point where the rubber is thinned, the tube should be discarded. The inside diameter of the tube should be inspected to ensure it has not been worn by contact with the toe of the tire bead. Tubes that have taken an unnatural set should be discarded. (*Figure 13-140*)

TUBE TIRE INSPECTION

It is important to inspect the inside of a tube type tire before installing a tube for service. Any protrusions or rough areas should be cause for concern, as these tend to abrade the tube and may cause early failure. Follow the tire, tube, and aircraft manufacturer's inspection criterion when inspecting aircraft tires and tubes.

TIRE MOUNTING

A certified technician may be called upon to mount an aircraft tire onto the wheel rim in preparation for service. In the case of a tube type tire, the tube must also be mounted. The following section presents general procedures for these operations using tube type and tubeless tires. Be sure to have the proper equipment and training to perform the work according to manufacturer's instructions.

TUBELESS TIRES

Aircraft tire and wheel assemblies are subject to enormous stress while in service. Proper mounting ensures tires perform to the limits of their design. Consult and follow all manufacturer's service information including bolt torques, lubrication and balancing requirements, and inflation procedures. As mentioned, a wheel assembly that is to have a tire mounted upon it must be thoroughly inspected to ensure it is serviceable. Pay close attention to the bead seat area, which should be smooth and free from defects. The wheel half mating surface should be in good condition. The O-ring should be lubricated and in good condition to ensure it seals the wheel for the entire life of the tire. Follow the manufacturer's instructions when inspecting wheels and the tips provided earlier in this chapter. (*Figure 13-141*)

A final inspection of the tire to be mounted should be made. Most important is to check that the tire is specified for the aircraft application. It should say

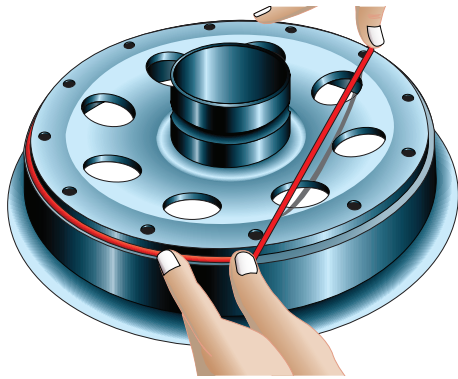


Figure 13-141. The wheel half O-ring for a tubeless tire wheel assembly must be in good condition and lubricated to seal for the entire life of the tire. The mating surfaces of the wheel halves must also be in good condition.

tubeless on the sidewall. The part number, size, ply rating, speed rating, and technical standard order (TSO) number should also be on the sidewall and be approved for the aircraft installation. Visually check the tire for damage from shipping and handling. There should be no permanent deformation of the tire. It should pass all inspections for cuts and other damage discussed in the previous sections of this chapter. Clean the tire bead area with a clean shop towel and soap and water or denatured alcohol. Inspect the inside of the tire for condition. There should be no debris inside the tire. Tire beads are sometimes lubricated when mounted on aluminum wheels. Follow the manufacturer's instructions and use only the non-hydrocarbon lubricant specified. Never lubricate any tire bead with grease. Do not use lubricants with magnesium alloy wheels. Most radial tires are mounted without lubricant. The airframe manufacturer may specify lubrication for a radial tire in a few cases.

When the wheel halves and tires are ready to be mounted, thought must be given to tire orientation and the balance marks on the wheel halves and tire. Typically, the tire serial number is mounted to the outboard side of the assembly. The marks indicating the light portion of each wheel half should be opposite each other. The mark indicating the heavy spot of the wheel assembly should be mounted aligned with the light spot on the tire, which is indicated by a red mark. If the wheel lacks a mark indicating the heavy spot, align the red spot on the tire (the light point) with the valve fitting location on the wheel. A properly balanced tire and wheel assembly improves the overall performance of the tire. It promotes smooth operation free from vibration, which results in uniform tread wear and extended tire life.

When assembling the wheel halves, follow manufacturer's instructions for tie bolt tightening sequences and torque specification. Anti-seize lubricants and wet-torque values are common on wheel assemblies. Use a calibrated hand torque wrench. Never use an impact wrench on an aircraft tire assembly.

For the initial inflation of an aircraft tire and wheel assembly, the tire must be placed in a tire inflation safety cage and treated as though it may explode due to wheel or tire failure. The inflation hose should be attached to the tire valve stem, and inflation pressure should be regulated from a safe distance away. A minimum of 30 feet is recommended. Air or nitrogen should be introduced gradually as specified. Dry nitrogen keeps the introduction of water into the tire to a minimum, which helps prevent corrosion. Observe the tire seating progress on the wheel rim while it inflates. Depressurize the tire before approaching it to investigate any observed issue. (*Figure 13-142*)



Figure 13-142. Modern tire inflation cages have been tested to withstand catastrophic failure of a tire and wheel assembly during inflation. All newly mounted tires should be inflated in such a cage.

Aircraft tires are typically inflated to their full specified operating pressure. Then, they are allowed to remain with no load applied for 12 hours. During this time, the tire stretches and tire pressure decreases. A 5-10 percent reduction is normal. Upon bringing the tire up to full pressure again, less than five percent loss per day of pressure is allowable. More should be investigated.

TUBE TYPE TIRES

Wheel and tire inspection should precede the mounting of any tire, including tube type tires. The tube to be installed must also pass inspection and must be the correct size for the tire and tire must be specified for the aircraft. Tire talc is commonly used when installing tube type tires to ensure easy mounting and free movement between the tube and tire as they inflate. (*Figure 13-143*)

The technician should lightly talc the inside of the tire and the outside of the tube. Some tubes come from the factory with a light talc coating over the outside of the tube. Inflate the tube so that it just takes shape with minimal pressure. Install the tube inside the tire. Tubes are typically produced with a mark at the heavy spot of the tube. In the absence of this balance mark, it is assumed that the valve is located at the heaviest part of the tube. For proper balance, align the heavy part of the tube with the red mark on the tire (the light spot on the tire). (*Figure 13-144*)

Once wheel balance is marked and the tube balance mark and the tire balance mark are all positioned correctly, install the outboard wheel half so the valve stem of the tube passes through the valve stem opening. (*Figure 13-145*)



Figure 13-143. Tire talc is used on the inside of tube-type tires and the outside of aircraft tubes. This prevents binding and allows the tube to expand without stress into place within the tire.

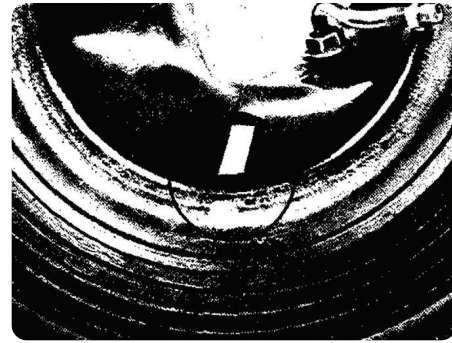


Figure 13-144. When assembling a tube into a tire, the heavy balance mark on the tube is aligned with the light balance mark on the tire.



Figure 13-145. Mounting a tube type tire with the tube valve stem positioned to pass through the outboard wheel half.

Mate the inboard wheel half to it, being careful not to pinch the tube between the wheel rims. Install the tie bolts, tighten, and torque as specified. Inflate the assembly in a tire inflation cage. The inflation procedure for a tube type tire differs slightly from that of a tubeless tire. The assembly is slowly brought up to full operating pressure. Then, it is completely deflated. Reinflate the tire/tube assembly a second time to the specified operating pressure and allow it to remain with no load for 12 hours. This allows any wrinkles in the tube to smooth out, helps prevent the tube from being trapped under a bead, and generally evens how the tube lays within the tire to avoid any stretched areas and thinning of the tube. The holding time allows air trapped between the tube and the tire to work its way out of the assembly, typically through the tire sidewall or around the valves stem.

TIRE BALANCING

Once an aircraft tire is mounted, inflated, and accepted for service, it can be balanced to improve performance. Vibration is the main result of an imbalanced tire and wheel assembly. Nose wheels tend to create the greatest disturbance in the cabin when imbalanced. Static balance is all that is required for most aircraft tires and wheels.

A balance stand typically accepts the assembly on cones. The wheel is free to rotate. The heavy side moves to the bottom. (*Figure 13-146*) Temporary weights are added to eliminate the wheel from rotating and dropping the heavy side down. Once balanced, permanent weights are installed. Many aircraft wheels have provisions for securing the permanent weight to the wheel. Weights with adhesive designed to be glued to the wheel rim are also in use. Occasionally, a weight in the form of a patch glued to the inside of the tire is required. Follow all manufacturer's instructions and use only the weights specified for the wheel assembly. (*Figure 13-147*)

Some aviation facilities offer dynamic balancing of aircraft tire and wheel assemblies. While this is rarely specified by manufacturers, a well balanced tire and wheel assembly helps provide shimmy free operation and reduces wear on brake and landing gear components, such as torque links.



Figure 13-146. A typical aircraft tire and wheel balancing stand.

OPERATION AND HANDLING TIPS

Aircraft tires experience longer life if operated in a manner to conserve wear and minimize damage. The most important factor impinging on tire performance and wear, as well as resistance to damage is proper inflation. Always inflate tires to the specified level before flight for maximum performance and minimal damage. An improperly inflated tire has increased potential to fail upon landing due to the high impact loads experienced. The following sections include other suggestions that can extend the life and the investment made in aircraft tires.

TAXIING

Needless tire damage and excessive wear can be prevented by proper handling of the aircraft during taxi. Most of the gross weight of an aircraft is on the main landing gear wheels. Aircraft tires are designed and inflated to absorb the shock of landing by deflection of the sidewalls two to three times as much as that found on an automobile tire. While this enables the tire to handle heavy loads, it also causes more working of the tread and produces scuffing action along the outer edges of the tread that results in more rapid wear. It also leaves the tire more prone to damage as the tread compound opens during this flexing.

An aircraft tire that strikes a chuck hole, a stone, or some other foreign object is more likely to sustain a cut, snag, or bruise than an automobile tire due to its more flexible nature. There is also increased risk for internal tire injury when a tire leaves the paved surface of the taxi way. These incidents should be avoided. Dual or multiple wheel main gear should be operated so that all tires remain on the paved surface so the weight of the aircraft is evenly distributed between



Figure 13-147. A tire balancing patch (left), adhesive wheel weights (center), and a bolted wheel weight (right) are all used to balance aircraft tire and wheel assemblies per the manufacturer's instructions.

the tires. When backing an aircraft on a ramp for parking, care should be taken to stop the aircraft before the main wheels roll off of the paved surface.

Taxiing for long distances or at high speeds increase the temperature of aircraft tires. This makes them more susceptible to wear and damage. Short taxi distances at moderate speeds are recommended. Caution should also be used to prevent riding the brakes while taxiing, which adds unnecessary heat to the tires.

BRAKING AND PIVOTING

Heavy use of aircraft brakes introduces heat into the tires. Sharp radius turns do the same and increase tread abrasion and side loads on the tire. Plan ahead to allow the aircraft to slow without heavy braking and make large radius turns to avoid these conditions. Objects under a tire are ground into the tread during a pivot. Since many aircraft are primarily maneuvered on the ground via differential braking, efforts should be made to always keep the inside wheel moving during a turn rather than pivoting the aircraft with a locked brake around a fixed main wheel tire.

LANDING FIELD AND HANGAR FLOOR CONDITION

One of the main contributions made to the welfare of aircraft tires is good upkeep of airport runway and taxiway surfaces, as well as all ramp areas and hangar floors. While the technician has little input into runway and taxiway surface upkeep, known defects in the paved surfaces can be avoided and rough surfaces can be negotiated at slower than normal speeds to minimize tire damage. Ramps and hangar floors should be kept free of all foreign objects that may cause tire damage. This requires continuous diligence on the part of all aviation personnel. Do not ignore foreign object damage (FOD). When discovered, action must be taken to remove it. While FOD to engines and propellers gains significant attention, much damage to tires is avoidable if ramp areas and hangar floors are kept clean.

TAKEOFFS AND LANDINGS

Aircraft tires are under severe strain during takeoff and landing. Under normal conditions, with proper control and maintenance of the tires, they are able to withstand these stresses and perform as designed.

Most tire failures occur during takeoff which can be extremely dangerous. Tire damage on takeoff is often the result of running over some foreign object. Thorough preflight inspection of the tires and wheels, as well as maintenance of hangar and ramp surfaces free of foreign objects, are keys to prevention of takeoff tire failure. A flat spot caused on the way to the runway may lead to tire failure during takeoff. Heavy braking during aborted takeoffs is also a common cause of takeoff tire failure. (*Figure 13-148*)

Tire failure upon landing can have several causes. Landing with the brakes on is one. This is mitigated on aircraft with antiskid systems, but can occur on other aircraft. Other errors in judgment, such as landing too far down the runway and having to apply the brakes heavily, can cause overheating or skidding. This can lead to flat-spotting the tires or blow out.

HYDROPLANING

Skidding on a wet, icy, or dry runway is accompanied by the threat of tire failure due to heat build-up and rapid tire wear damage. Hydroplaning on a wet runway may be overlooked as a damaging condition for a tire. Water building up in front of the tire provides a surface for the tire to run on and contact with the runway surface is lost. This is known as dynamic hydroplaning. Steering ability and braking action is also lost. A skid results if the brakes are applied and held.

Viscous hydroplaning occurs on runways with a thin film of water that mixes with contaminants to cause an extremely slick condition. This can also happen on a very smooth runway surface. A tire with a locked brake during viscous hydroplaning can form an area of

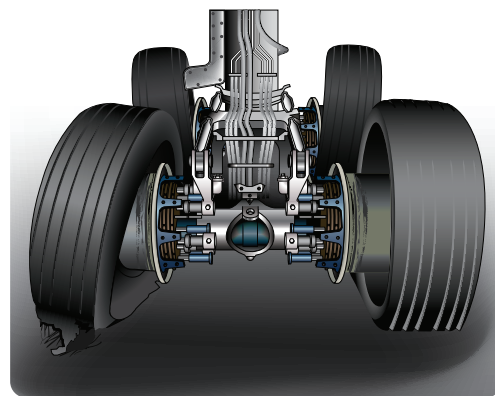


Figure 13-148. Heavy braking during an aborted takeoff caused these tires to fail.

reverted rubber or skid burn in the tread. While the tire may continue in service if the damage is not too severe, it can be cause for removal if the reinforcing tread or protector ply is penetrated. The same damage can occur while skidding on ice.

Modern runways are designed to drain water rapidly and provide good traction for tires in wet conditions. A compromise exists in that crosscut runways and textured runway surfaces cause tires to wear at a greater rate than a smooth runway. (*Figure 13-149*)

A smooth landing is of great benefit to any tire. Much aircraft tire handling and care is the responsibility of the pilot. However, the technician benefits from knowing the causes of tire failure and communicating this knowledge to the flight crew so that operating procedures can be modified to avoid those causes.



Figure 13-149. Crosscut runway surfaces drain water rapidly but increase tire wear.

Question: 13-1

Three basic arrangements of landing gear are the tail wheel type landing gear (also known as conventional gear), tandem landing gear, and _____ landing gear.

Question: 13-2

When more than two wheels are attached to a landing gear strut, the attaching mechanism is known as a _____.

Question: 13-3

Aligned, _____ and _____ refer to the path a main wheel would take in relation to the airframe longitudinal axis or centerline if the wheel was free to roll forward.

Question: 13-4

The _____ is a fixed structural extension of the upper strut cylinder with bearing surfaces that allow the entire gear assembly to move.

Question: 13-5

A typical pneumatic/hydraulic landing gear shock strut uses _____ air or _____ combined with hydraulic fluid to dissipate shock loads.

Question: 13-6

To keep the piston and wheels aligned, most shock struts are equipped with _____.

Question: 13-7

Shock strut servicing is accomplished through the high-pressure service valve found at the _____ of the strut's upper cylinder.

Question: 13-8

It may be necessary to _____ a shock strut during the service operation or when air becomes trapped in the hydraulic fluid inside the strut.

Question: 13-9

The proper functioning of a landing gear system and components can be checked by performing a _____ test, which is also known as "swinging" the gear.

Question: 13-10

_____ are commonly used on aircraft landing gear as extra insurance that the landing gear will remain down and locked while the aircraft is on the ground.

ANSWERS

Answer: 13-1
tricycle type.

Answer: 13-6
torque links or torque arms.

Answer: 13-2
bogie.

Answer: 13-7
top.

Answer: 13-3
tow-in.
tow-out.

Answer: 13-8
bleed.

Answer: 13-4
trunnion.

Answer: 13-9
landing gear retraction.

Answer: 13-5
compressed.
nitrogen.

Answer: 13-10
Ground locks.

Question: 13-11

Weight on wheels (WOW) integrated circuit logic cards communicate air-ground status information with the other systems on the aircraft through a _____.

Question: 13-12

Large aircraft with hydraulic steering hold pressure in the steering cylinders to provide the required _____.

Question: 13-13

A common geometry used to lock a landing gear in the down position involves a collapsible _____ that is extended and held in an over center position through the use of a locking link.

Question: 13-14

Aircraft _____ support the entire weight of the aircraft during taxi, takeoff, and landing.

Question: 13-15

A _____ wheel half bolt is grounds for removal, and a thorough inspection of the wheel halves.

Question: 13-16

The bearing _____ does not require removal for inspection; however, it must be firmly seated in the wheel half boss.

Question: 13-17

The basic operation of brakes involves converting the _____ energy of motion into _____ energy through the creation of friction. text.

Question: 13-18

Large, heavy aircraft require the use of _____ brakes.

Question: 13-19

A _____ is a flat, circular, high-strength steel, non-rotating plate notched on the inside circumference to fit over the stator drive sleeves or torque tube spines.

Question: 13-20

Power brake systems on large aircraft use the _____ as a source of power.

ANSWERS

Answer: 13-11
digital data bus.

Answer: 13-16
cup (race).

Answer: 13-12
damping.

Answer: 13-17
kinetic.
heat.

Answer: 13-13
side brace.

Answer: 13-18
multiple disc.

Answer: 13-14
wheels.

Answer: 13-19
pressure plate.

Answer: 13-15
bolt.

Answer: 13-20
aircraft main hydraulic system.

Question: 13-21

The majority of the rudder/brake pedal feel is supplied by the brake control or brake _____ in a power brake system.

Question: 13-26

The presence of air in the brake system fluid causes the brake pedal to feel _____.

Question: 13-22

Brake _____ are simple devices that use the application of force over different sized pistons to reduce pressure.

Question: 13-27

_____ pressure bleeding is not an option for power brakes; the air trapped in the brake system would be forced into the main hydraulic system, which should operate without air in the fluid.

Question: 13-23

Three main components of pst brake antiskid systems are _____, _____ and _____.

Question: 13-28

Many brakes use _____ in the seal groove to support the O-ring seals and reduce the tendency of the seal to extrude into the space which it is meant to seal.

Question: 13-24

Antiskid control valves are fast-acting, _____ controlled hydraulic valves that respond to the input from the antiskid control unit.

Question: 13-29

Tire _____ are reinforcing layers of fabric encased in rubber that are laid into the tire to provide strength.

Question: 13-25

Auto braking is activated by _____.

Question: 13-30

The main function of an aircraft tire sidewall is _____.

ANSWERS

Answer: 13-21
metering valve.

Answer: 13-26
spongy.

Answer: 13-22
deboosters.

Answer: 13-27
Bottom up.

Answer: 13-23
wheel speed sensors.
antiskid control valves.
control unit.

Answer: 13-28
back-up rings.

Answer: 13-24
electrically.

Answer: 13-29
plies.

Answer: 13-25
a switch on the flight deck.

Answer: 13-30
protection.

Question: 13-31

Evenly worn tread is a sign of _____.

Question: 13-34

A wheel half mating surface O-ring should be _____ and in good condition to ensure it seals the wheel for the entire life of the tire.

Question: 13-32

An aircraft tire and wheel assembly, especially a high pressure assembly that has been damaged or overheated, should be treated as though it may _____.

Question: 13-35

_____ is the main result of an imbalanced tire and wheel assembly.

Question: 13-33

The effective cost of a tire over its life can be reduced by having the tread _____ while the carcass is still sound and injuries are within repairable limits.

Question: 13-36

_____ is caused when water builds up in front of a tire and provides a surface for the tire to run on so that contact with the runway surface is lost.

ANSWERS

Answer: 13-31
proper tire maintenance.

Answer: 13-34
lubricated.

Answer: 13-32
explode.

Answer: 13-35
Vibration.

Answer: 13-33
replaced.

Answer: 13-36
Dynamic hydroplaning.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 14

LIGHTS (ATA 33)

Knowledge Requirements

11.14 - Lights (ATA 33)

External: navigation, anti collision, landing, taxiing, ice;

Internal: cabin, cockpit, cargo;

Emergency.

3

LIGHTS (ATA 33)

11.14 - LIGHTS

AIRCRAFT LIGHTING SYSTEMS

Aircraft lighting systems provide illumination for both exterior and interior use. Lights on the exterior provide illumination for such operations as landing at night, inspection of icing conditions, and safety from midair collision. Interior lighting provides illumination for instruments, cockpits, cabins, and other sections occupied by crew-members and passengers. Certain special lights, such as indicator and warning lights, indicate the operation status of equipment.

EXTERIOR LIGHTS

Position, anti-collision, landing, and taxi lights are common examples of aircraft exterior lights. Some lights are required for night operations. Other types of exterior lights, such as wing inspection lights, are of great benefit for specialized flying operations.

NAVIGATION/POSITION LIGHTS

Aircraft operating at night must be equipped with position lights that meet minimum requirements. A set of position lights consist of one red, one green, and one white light. (*Figure 14-1* and *Figure 14-2*)

On some types of installations, a switch in the cockpit provides for steady or flashing operation of the position lights. On many aircraft, each light unit contains a single lamp mounted on the surface of the aircraft. Other types of position light units contain two lamps and are often streamlined into the surface of the aircraft structure. The green light unit is always mounted at the extreme tip of the right wing. The red unit is mounted in a similar position on the left wing. The white unit is usually located on the vertical stabilizer in a position where it is clearly visible through a wide angle from the rear of the aircraft. *Figure 14-3* illustrates a schematic diagram of a position light circuit. Position lights are also known as navigation lights.

There are, of course, many variations in the position light circuits used on different aircraft. All circuits are protected by fuses or circuit breakers, and many circuits include flashing and dimming equipment. Small aircraft are usually equipped with a simplified control switch and circuitry. In some cases, one control knob or switch is used to turn on several sets of lights; for example, one type utilizes a control knob, the first movement of which

turns on the position lights and the instrument panel lights. Further rotation of the control knob increases the intensity of only the panel lights. A flasher unit is seldom included in the position light circuitry of very light aircraft but is used in small twin-engine aircraft. Traditional position lights use incandescent light bulbs. LED lights have been introduced on modern aircraft because of their good visibility, high reliability, and low power consumption.

ANTI-COLLISION LIGHTS

An anti-collision light system may consist of one or more lights. They are rotating beam lights that are usually installed on top of the fuselage or tail in such a location that the light does not affect the vision of the crew member or detract from the visibility of the position lights. Large transport type aircraft use an anti-collision light on top and one on the bottom of the aircraft. *Figure 14-4* shows a typical anti-collision light installation in a vertical stabilizer.



Figure 14-1. A left wing tip position light (red) and a white strobe light.

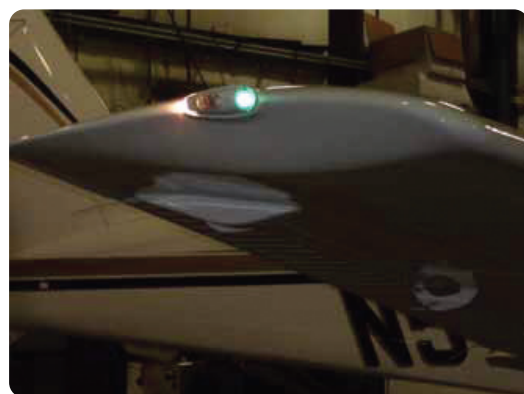


Figure 14-2. A right wing tip position light, also known as a navigation light.

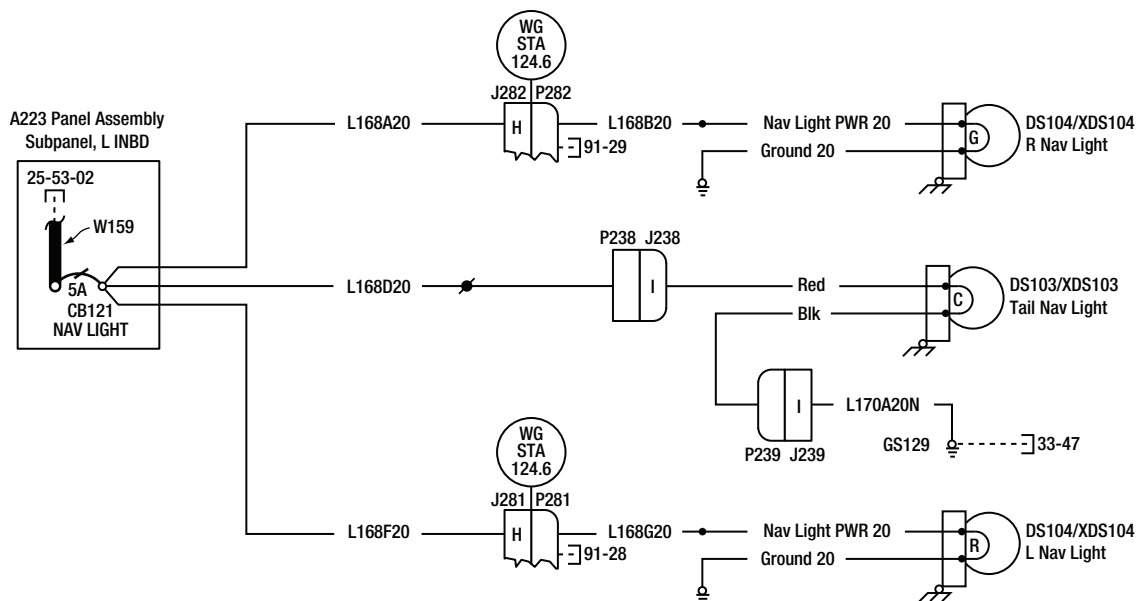


Figure 14-3. Navigation light system schematic.

An anti-collision light unit usually consists of one or two rotating lights operated by an electric motor. The light may be fixed but mounted under rotating mirrors inside a protruding red glass housing. The mirrors rotate in an arc, and the resulting flash rate is between 40 and 100 cycles per minute. Newer aircraft designs use a LED type of anti-collision light. The anti-collision light is a safety light to warn other aircraft, especially in congested areas.

A white strobe light is a second type of anti-collision light that is also common. Usually mounted at the wing tips and, possibly, at empennage extremities, strobe lights produce an extremely bright intermittent flash of white light that is highly visible. The light is produced by a high voltage discharge of a capacitor. A dedicated power pack houses the capacitor and supplies voltage to a sealed xenon-filled tube. The xenon ionizes with a flash when the voltage is applied. A strobe light is shown in *Figure 14-1*.

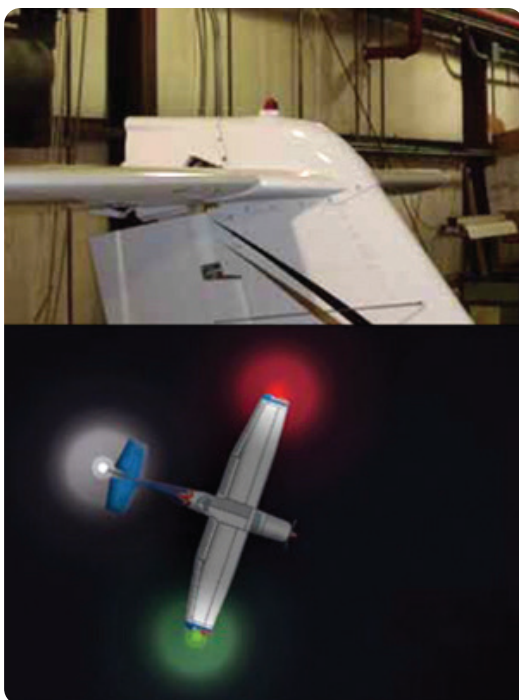


Figure 14-4. Anticollision lights.

LANDING LIGHTS

Landing lights are installed in aircraft to illuminate runways during night landings. These lights are very powerful and are directed by a parabolic reflector at an angle providing a maximum range of illumination. Landing lights of smaller aircraft are usually located midway in the leading edge of each wing or streamlined into the aircraft surface. Landing lights for larger transport category aircraft are usually located in the leading edge of the wing close to the fuselage. Each light may be controlled by a relay, or it may be connected directly into the electric circuit. On some aircraft, the landing light is mounted in the same area with a taxi light. (*Figure 14-5*) A sealed beam, halogen, or high intensity xenon discharge lamp is used.

TAXI LIGHTS

Taxi lights are designed to provide illumination on the ground while taxiing or towing the aircraft to or from a runway, taxi strip, or in the hangar area. (*Figure 14-6*)



Figure 14-5. Landing lights.

Taxi lights are not designed to provide the degree of illumination necessary for landing lights. On aircraft with tricycle landing gear, either single or multiple taxi lights are often mounted on the non-steerable part of the nose landing gear. They are positioned at an oblique angle to the center line of the aircraft to provide illumination directly in front of the aircraft and also some illumination to the right and left of the aircraft's path. On some aircraft, the dual taxi lights are supplemented by wingtip clearance lights controlled by the same circuitry. Taxi lights are also mounted in the recessed areas of the wing leading edge, often in the same area with a fixed landing light.

Many small aircraft are not equipped with any type of taxi light, but rely on the intermittent use of a landing light to illuminate taxiing operations. Still other aircraft utilize a dimming resistor in the landing light circuit to provide reduced illumination for taxiing. A typical circuit for taxi lights is shown in **Figure 14-7**. Some large aircraft are equipped with alternate taxi lights located on the lower surface of the aircraft, aft of the nose radome. These lights, operated by a separate switch from the main taxi lights, illuminate the area immediately in front of and below the aircraft nose.

WING ICE INSPECTION LIGHTS

Some aircraft are equipped with wing inspection lights to illuminate the leading edge of the wings to permit observation of icing and general condition of these areas during flight. These lights permit visual detection of ice formation on wing leading edges while flying at night. They are usually controlled through a relay by an on/off toggle switch in the cockpit.



Figure 14-6. Taxi lights.

Some wing inspection light systems may include or be supplemented by additional lights, sometimes called nacelle lights, that illuminate adjacent areas, such as a cowl flaps or the landing gear. These are normally the same type of lights and can be controlled by the same circuits.

INTERIOR LIGHTS

Aircraft are equipped with interior lights to illuminate the cabin. (**Figure 14-8**) Often white and red light settings are provided on the flight deck. Commercial aircraft have a variety of independent lighting systems that illuminate the flight deck, instrument panels, passenger cabin, cargo compartments and more. Interior lights incorporate the use of both incandescent and fluorescent lights that operate off a variety of AC and DC electrical buses.

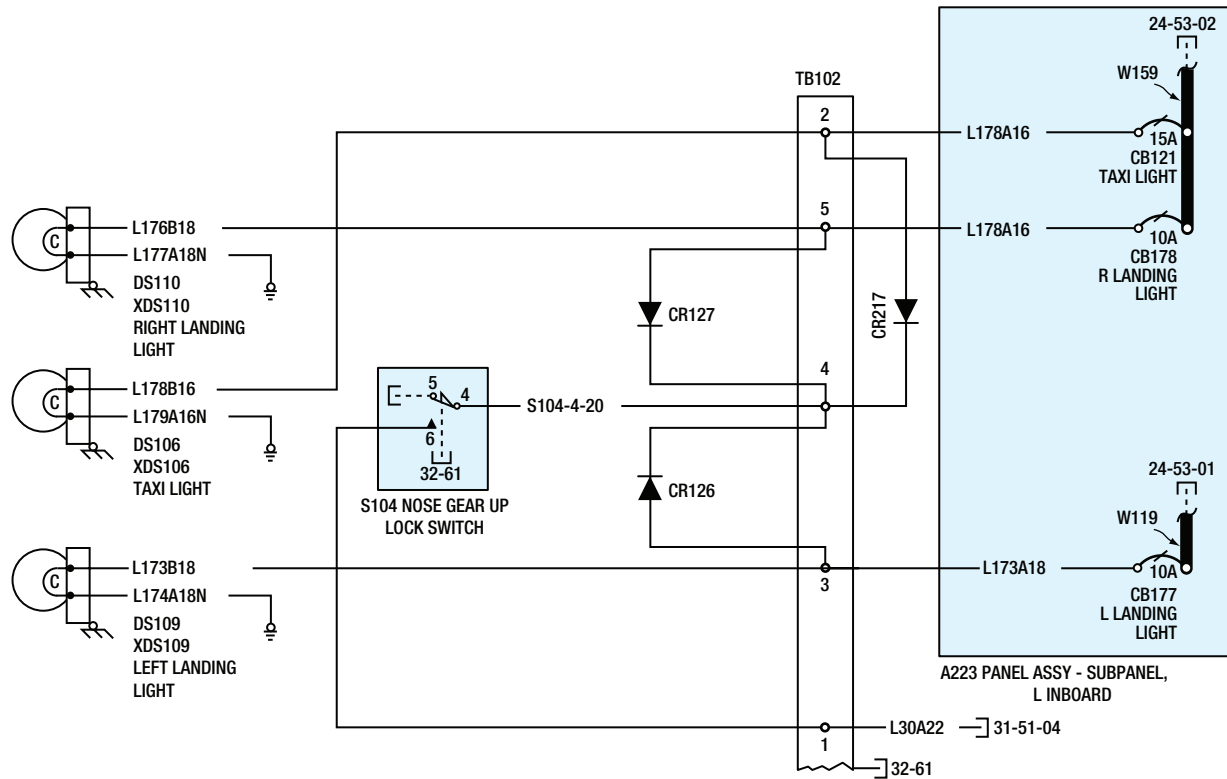


Figure 14-7. Taxi light circuit.



Figure 14-8. Interior cockpit and cabin light system.

PASSENGER CABIN LIGHTS

Independent lighting systems are used in the passengers cabin. A combination of incandescent and fluorescent lights in overhead and window positions provide general illumination. (*Figure 14-9*) These normally use AC power. Threshold and doorway entry lights are used as well as a variety of illuminated information signs. Galley and lavatories have their own lighting circuits. Overhead passenger service units (PSUs) in each seat row contain independent reading lights and service call lights for each seat. On the most modern aircraft such as the Boeing 777, the myriad of lights in the passenger

FLIGHT DECK/CONTROL CABIN LIGHTING

On an airliner flight deck, it is normal to have lighting for general illumination of the control cabin as well local lighting for panels, instruments and controls. Fluorescent background lights are also used. A centrally located panel, typically an overhead panel, houses the controls for many interior and exterior lights. Independent light controls may also be located on appropriate panels.

Normal lighting requirements are met using 28 VAC power with key lights positioned for part-power and no power situations typically run off a 28 VDC bus. On Boeing Aircraft, 115 VAC is used for fluorescent lights.



Figure 14-9. Incandescent and fluorescent lights are used to illuminate modern aircraft cabins.

compartment are controlled by a central control unit such as the Cabin Service System (CSS). Interface panels are located for easy access by the cabin crew. A menu provides numerous options for not only which lights are illuminated but also light intensity.

CARGO COMPARTMENT LIGHTING

Cargo and service compartments also have lighting. Dome lights, flood lights and explosion-proof lights as required are installed with independent circuits protected by circuit breakers. The lights are controlled by switches near the entrance to each area or inside the compartments. Often, a control panel for a cargo area includes light switches in addition to door and cargo system operating controls. Sidewall, overhead and door mounted lights are common. Door and door sill lights are positioned so that they illuminate the cargo compartment doorway as well as the area just outside the compartment to facilitate work while loading cargo.

EMERGENCY LIGHTING

Emergency lights are installed in the cabin to illuminate escape routes for passengers and crew during a failure of AC power systems. Lighting strips in the floor and exit lights automatically illuminate when power is lost. (**Figure 14-10**) Emergency lights are used to illuminate the over wing area at the emergency exits and on the escape slides. Lavatories and the control cabin also have emergency lighting.

Various configuration exist for automatic switching of certain emergency lights to the hot DC battery bus (or similar) in case of partial electrical failure. Some interior lighting is designed to always be connected to a DC bus so no switching is required. Total electrical failure causes most emergency lights to revert to dedicated batteries that are an integral part of the lighting installation. Emergency EXIT/area lights for example, may contain a battery in the assembly that includes the lamps, cover lens, solid state switching logic and battery-charging control circuits. In some cases, the light/battery assembly can be removed from its mounted location and used as a portable flashlight. NiCad batteries are typical. In other configurations, the dedicated emergency light battery is remotely located in the same area as the light.

Regardless, emergency lighting is ARMED by a switch on the flight deck or at the passenger cabin lighting control panel. The all-in-one emergency light assemblies also have a switch that must be set to ARMED when the unit is installed. Inspection of an aircraft's emergency lighting system normally includes checking the condition and security of all visible wiring, connections, terminals, fuses, and switches and light units. A continuity light or meter can be used in making these checks, since the cause of many troubles can often be located by systematically testing each circuit for continuity.

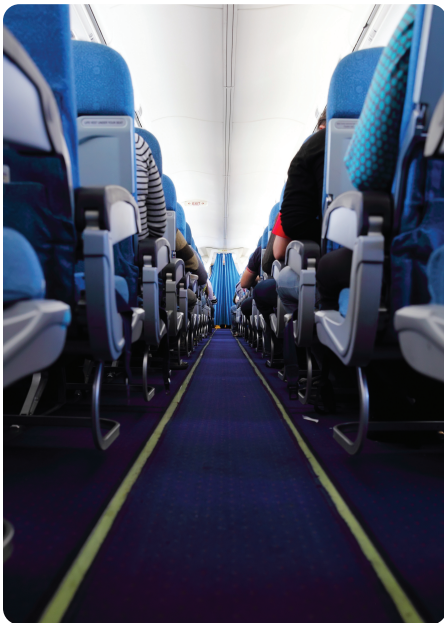


Figure 14-10. Emergency light strips in the aisle floor guide passengers and crew to the exits in case of emergency.

Question: 14-1

Another name for a “navigation light” is a _____ light.

Question: 14-5

A light that illuminates the leading edge of the wing is used to observe _____ as well as the general condition in this area during flight.

Question: 14-2

A _____ navigation light is always mounted on the extreme tip of the right wing.

Question: 14-6

Cabin lighting in normal situations is typically powered by _____ current.

Question: 14-3

A rotating light on the exterior of an aircraft is a(n) _____ light.

Question: 14-7

Cargo compartment lighting is controlled from _____.

Question: 14-4

Typically, aircraft have _____ lights because landing lights are too bright and focused at an angle beneficial for seeing the runway from the air.

Question: 14-8

Cabin exit emergency lights often contain integral _____.

ANSWERS

Answer: 14-1
position.

Answer: 14-5
ice.

Answer: 14-2
green.

Answer: 14-6
alternating.

Answer: 14-3
anti-collision light.

Answer: 14-7
a control panel near the entrance to the compartment.

Answer: 14-4
taxi.

Answer: 14-8
battery.
solid state switching logic circuits.
battery charging circuits.
operational arming switches.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 15

OXYGEN (ATA 35)

Knowledge Requirements

11.15 - Oxygen (ATA 35)

System lay-out: cockpit, cabin;
Sources, storage, charging and distribution;
Supply regulation;
Indications and warnings.

3

OXYGEN (ATA 35)

11.15 - OXYGEN

OXYGEN AND THE ATMOSPHERE

The mixture of gases that make up the earth's atmosphere is commonly called air. It is composed principally of 78 percent nitrogen and 21 percent oxygen. The remaining 1 percent is made up of various gases in smaller quantities. Some of these are important to human life, such as carbon dioxide, water vapor, and ozone. **Figure 15-1** indicates the respective percentage of the quantity of each gas in its relation to the total mixture.

As altitude increases, the total quantity of all the atmospheric gases reduces rapidly. However, the relative proportions of nitrogen and oxygen remain unchanged up to about 50 miles above the surface of the earth. The percentage of carbon dioxide is also fairly stable. The amounts of water vapor and ozone vary. Nitrogen is an inert gas that is not used directly by man for life processes; however, many compounds containing nitrogen are essential to all living matter.

The small quantity of carbon dioxide in the atmosphere is utilized by plants during photosynthesis. Thus, the food supply for all animals, including man, depends on it. Carbon dioxide also helps control breathing in man and other animals. The amount of water vapor in the atmosphere is variable but, even under humid conditions at sea level, it rarely exceeds 5 percent. Water also occurs in the atmosphere as ice crystals. All forms of water in the atmosphere absorb far more energy from the sun than do the other gases. Water plays an important role in the formation of weather.

Ozone is a form of oxygen. It contains three oxygen atoms per molecule, rather than the usual two. Most of the atmosphere's ozone is formed by the interaction of oxygen and the sun's rays near the top of the stratosphere in an area called the ozone layer. This is important to living organisms because ozone filters out most of the sun's harmful ultraviolet (UV) radiation. Ozone is also produced by electrical discharges, such as lightning strikes. It has a faint odor, somewhat like that of weak chlorine, that may be detected after a thunderstorm. Auroras and cosmic rays may also produce ozone. Ozone is of great consequence to living creatures on earth and to the circulation of the upper atmosphere.

HUMAN RESPIRATION AND CIRCULATION

The second most prevalent substance in the atmosphere, oxygen, is essential for most living processes. Without oxygen, humans and animals die very rapidly. A reduction in the normal oxygen supply alters the human condition. It causes important changes in body functions, thought processes, and the maintainable degree of consciousness. The resultant sluggish condition of mind and body produced by insufficient oxygen is called hypoxia.

There are several scenarios that can result in hypoxia. During aircraft operations, it is brought about by a decrease in the pressure of oxygen in the lungs at high altitudes. The air contains the typical 21 percent of oxygen, but the rate at which oxygen can be absorbed into the blood depends upon the oxygen pressure. Greater pressure pushes the oxygen from the lung alveoli into the bloodstream. As the pressure is reduced, less oxygen is forced into and absorbed by the blood.

At sea level, oxygen pressure in the lungs is approximately three pounds per square inch (psi). This is sufficient to saturate the blood with oxygen and permit the mind and body to function normally. As altitude is increased, this pressure decreases. Below 7 000 feet above sea level, the available oxygen quantity and pressure remain sufficient for saturation of the blood with oxygen. Above 7 000 feet, however, the oxygen pressure becomes increasingly insufficient to saturate the blood. At 10 000 feet mean sea level (MSL), saturation of the blood with oxygen is only about 90 percent of normal. Long durations at

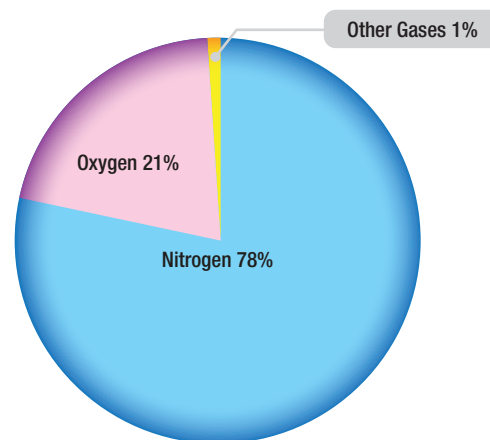


Figure 15-1. The percentage of the various gases that comprise the atmosphere.

this altitude can result in headache and fatigue, both symptoms of hypoxia.

At 15 000 feet MSL, oxygen transfer to the bloodstream drops to 81 percent of saturation. This typically results in sleepiness, headache, blue lips and fingernails, and increased pulse and respiration. Worse yet, vision and judgment become impaired and safe operation of an aircraft becomes compromised. Higher in the atmosphere, decreasing pressure causes even less oxygen to enter the bloodstream; only 68 percent saturation at 22 000 feet MSL. Remaining at 25 000 feet MSL for 5 minutes, where oxygen transfer to the blood is reduced to approximately 50 percent saturation, causes unconsciousness. (*Figure 15-2*)

The negative effects of reduced atmospheric pressure at flight altitudes, forcing less oxygen into the blood, can be overcome. There are two ways this is commonly done:

1. Increase the pressure of the oxygen.
2. Increase the quantity of oxygen in the air mixture.

Large transport category and high performance passenger aircraft pressurize the air in the cabin. This serves to push more of the normal 21 percent oxygen found in the air into the blood for saturation. When utilized, the percentage of oxygen available for breathing remains the same; only the pressure is increased.

By increasing the quantity of oxygen available in the lungs, less pressure is required to saturate the blood. This is the basic function of an aircraft oxygen system.

Altitude MSL (feet)	Oxygen Pressure (psi)
0	3.08
5 000	2.57
10 000	2.12
15 000	1.74
20 000	1.42
25 000	1.15
30 000	0.92
35 000	0.76
40 000	0.57

Figure 15-2. Oxygen pressure in the atmosphere at various altitudes.

Increasing the level of oxygen above the 21 percent found in the atmosphere can offset the reduced pressure encountered as altitude increases.

Oxygen may be regulated into the air that is breathed so as to maintain a sufficient amount for blood saturation. Normal mental and physical activity can be maintained at indicated altitudes of up to about 40 000 feet with the sole use of supplemental oxygen.

Oxygen systems that increase the quantity of oxygen in breathing air are most commonly used as primary systems in small and medium size aircraft designed without cabin pressurization. Pressurized aircraft utilize oxygen systems as a means of redundancy should pressurization fail. Portable oxygen equipment may also be aboard for first aid purposes.

SOURCES OF OXYGEN

Oxygen is a colorless, odorless, and tasteless gas at normal atmospheric temperatures and pressures. It transforms into a liquid at -183°C (its boiling point). Oxygen combines readily with most elements and numerous compounds. This combining is called oxidation. Typically, oxidation produces heat. When something burns, it is actually rapidly combining with oxygen. Oxygen itself does not burn because it does not combine with itself, except to form oxygen or ozone. But, pure oxygen combines violently with petroleum products creating a significant hazard when handling these materials in close proximity to each other. Nevertheless, oxygen and various petroleum fuels combine to create the energy produced in internal combustion engines.

Production of gaseous oxygen for commercial or aircraft cylinders is often through a process of liquefying air. By controlling temperature and pressure, the nitrogen in the air can be allowed to boil off leaving mostly pure oxygen. Oxygen may also be produced by the electrolysis of water. Passing electric current through water separates the oxygen from the hydrogen. One further method of producing gaseous oxygen is by separating the nitrogen and oxygen in the air through the use of a molecular sieve. This membrane filters out nitrogen and some of the other gases in air, leaving nearly pure oxygen for use. On board oxygen sieves, or oxygen concentrators as they are sometimes called, are used on some military aircraft. Their use in civil aviation is expected.

Another source of oxygen used frequently for emergency passenger oxygen on airline aircraft is chemical in nature. The chemical combination with oxygen when burning some materials causes a release of excess oxygen that can be captured and used. This unique production of oxygen is explained further below.

OXYGEN STORAGE

GASEOUS OXYGEN TANKS

Pure gaseous oxygen, or nearly pure gaseous oxygen, is stored and transported in high-pressure cylinders that are typically painted green. Technicians should be cautious to keep pure oxygen away from fuel, oil, and grease to prevent unwanted combustion. Not all oxygen in containers is the same. Aviator's breathing oxygen is tested for the presence of water. This is done to avoid the possibility of it freezing in the small passage ways of valves and regulators. Ice could prevent delivery of the oxygen when needed. Aircraft often operate in subzero temperatures, increasing the possibility of icing. The water level should be a maximum of 0.02 ml per liter of oxygen. The words "Aviator's Breathing Oxygen" should be marked clearly on any cylinders containing oxygen for this purpose. (*Figure 15-3*)

Traditionally, the cylinders used to store high pressure oxygen have been heavy steel tanks rated for 1 800-1 850 psi of pressure and capable of maintaining pressure up to 2 400 psi. While these performed adequately, lighter weight tanks have been developed. Some newer cylinders are comprised of a lightweight aluminum shell wrapped

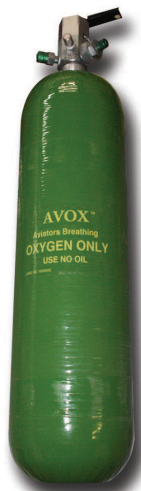


Figure 15-3. "Aviator's breathing oxygen" is marked on all oxygen cylinders designed for this purpose.

by Kevlar™. These cylinders are capable of carrying the same amount of oxygen at the same pressure as steel tanks, but weigh much less.

Also available are heavy-walled all-aluminum cylinders. These units are common as carry-on portable oxygen used in light aircraft. Most oxygen storage cylinders are painted green, but yellow and other colors may be used as well. The tanks are typically certified to department of transportation (DOT) specifications in the country of manufacture. To ensure continued serviceability, cylinders must be hydrostatically tested periodically. In general, a hydrostatic test consists of filling the container with water and pressurizing it to 5/3 of its certified rating. It should not leak, rupture, or deform beyond an established limit. *Figure 15-4* shows a hydrostatic cylinder testing apparatus.

Most cylinders also have a limited service life after which they can no longer be used. After a specified number of filling cycles or calendar age, the cylinders must be removed from service. The most common high-pressure steel oxygen cylinders used in aviation are the 3AA and the 3HT. They come in various sizes



Figure 15-4. This test stand is used for hydrostatic testing of oxygen cylinders. The water-filled cylinder is lowered into the barrel on the left where it is pressurized to the proper level as monitored via gauges mounted on the control panel. A displacement container on the top left of the control board collects water from the barrel to measure the expansion of the cylinder when pressurized to ensure it is within limits.

Certification Type	Material	Rated Pressure (psi)	Required Hydrostatic Test	Service Life (years)	Service Life (fillings)
DOT 3AA	Steel	1 800	5	Unlimited	N/A
DOT 3HT	Steel	1 850	3	24	4 380
DOT-E-8162	Composite	1 850	3	15	N/A
DOT-SP-8162	Composite	1 850	5	15	N/A
DOT 3AL	Aluminum	2 216	5	Unlimited	N/A

Figure 15-5. Common cylinders used in aviation with some certification and testing specifications.

but are certified to the same specifications. Cylinders certified under U.S. DOT-E-8162 are popular for their extremely light weight. These cylinders typically have an aluminum core around which Kevlar™ is wrapped. The DOT-E- 8162 approved cylinders are also approved under DOT-SP-8162 specifications. The SP certification has extended the required time between hydrostatic testing to 5 years (previously 3 years). (*Figure 15-5*)

The manufactured date and certification number is stamped on each cylinder near the neck. Subsequent hydrostatic test dates are also stamped there as well. Composite cylinders use placards rather than stamping. The placard must be covered with a coat of clear epoxy when additional information is added, such as a new hydrostatic test date.

Oxygen cylinders are considered empty when the pressure inside drops below 50 psi. This ensures that air containing water vapor has not entered the cylinder. Water vapor could cause corrosion inside the tank, as well as presenting the possibility of ice forming and clogging a narrow passageway in the cylinder valve or oxygen system. Any installed tank allowed to fall below this pressure should be removed from service.

CHEMICAL OR SOLID OXYGEN

Sodium chlorate has a unique characteristic. When ignited, it produces oxygen as it burns. This can be filtered and delivered through a hose to a mask that can be worn and breathed directly by the user. Solid oxygen candles, as they are called, are formed chunks of sodium chlorate wrapped inside insulated stainless steel housings to control the heat produced when activated. The chemical oxygen supply is often ignited by a spring-loaded firing pin that when pulled, releases a hammer that smashes a cap creating a spark to light the candle. Electric ignition via a current-induced hot wire also exists. Once lit, a sodium chlorate oxygen generator

cannot be extinguished. It produces a steady flow of breathable oxygen until it burns out, typically generating 10-20 minutes of oxygen. (*Figure 15-6*)

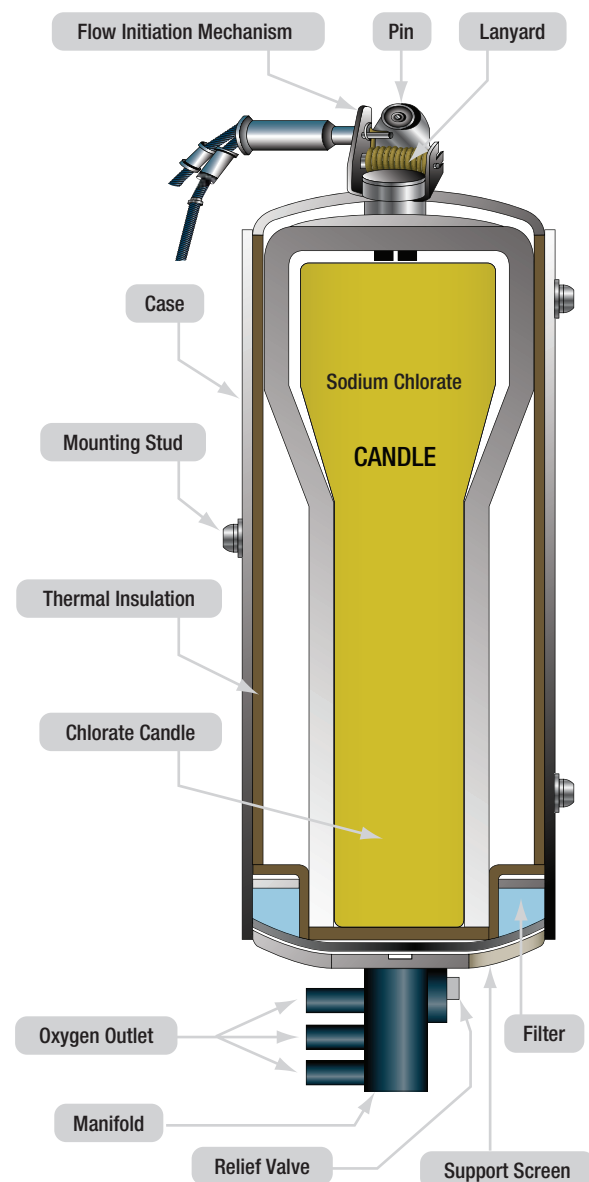


Figure 15-6. A sodium chlorate solid oxygen candle is at the core of a chemical oxygen generator.

Solid oxygen generators are primarily used as backup oxygen devices on pressurized aircraft. They are one third as heavy as gaseous oxygen systems that use heavy storage tanks for the same quantity of oxygen available. Sodium chlorate chemical oxygen generators also have a long shelf life, making them perfect as a standby form of oxygen. They are inert below 204°C and can remain stored with little maintenance or inspection until needed, or until their expiration date is reached.

The feature of not extinguishing once lit limits the use of solid oxygen since it becomes an all-or-nothing source. The generators must be replaced if used, which can greatly increase the cost of using them as a source of oxygen for short periods of time. Moreover, chemical oxygen candles must be transported with extreme caution and as hazardous materials. They must be properly packed and their ignition devices deactivated.

ON BOARD OXYGEN GENERATING SYSTEMS (OBOGS)

The molecular sieve method of separating oxygen from the other gases in air has application in flight, as well as on the ground. The sieves are relatively light in weight and relieve the aviator of a need for ground support for the oxygen supply. On board oxygen generating systems on military aircraft pass bleed air from turbine engines through a sieve that separates the oxygen for breathing use. Some of the separated oxygen is also used to purge the sieve of the nitrogen and other gases that keep it fresh for use. Use of this type of oxygen production in civilian aircraft is anticipated. (*Figure 15-7*)

LIQUID OXYGEN

Liquid oxygen (LOX) is a pale blue, transparent liquid. Oxygen can be made liquid by lowering the temperature to below -183°C or by placing gaseous oxygen under pressure. A combination of these is accomplished with a Dewar bottle. This special container is used to store and transport liquid oxygen. It uses an evacuated, double-walled insulation design to keep the liquid oxygen under pressure at a very low temperature. (*Figure 15-8*)

A controlled amount of oxygen is allowed to vaporize and is plumbed into a gaseous oxygen delivery system downstream of a converter that is part of the container assembly. A small quantity of LOX can be converted to an enormous amount of gaseous oxygen, resulting in the use of very little storage space compared to that needed



Figure 15-7. This on-board oxygen generating system uses molecular sieve technology.



Figure 15-8. A spherical liquid oxygen on-board container used by the military.

for high-pressure gaseous oxygen cylinders. However, the difficulty of handling LOX, and the expense of doing so, has resulted in the container system used for gaseous oxygen to proliferate throughout civilian aviation. LOX is used nearly exclusively in military aviation.

OXYGEN CHARGING

Charging procedures for oxygen systems vary. Many general aviation aircraft are set up to simply replace an empty cylinder with one that is fully charged. This is also the case with a portable oxygen system. High performance and air transport category aircraft often have built-in oxygen systems that contain plumbing designed to refill gaseous oxygen cylinders while they are in place. A general discussion of the procedure to fill this type of installation follows.

Before charging any oxygen system, consult the aircraft manufacturer's maintenance manual. The type of oxygen to be used, safety precautions, equipment to be used, and the procedures for filling and testing the system must be observed. Several general precautions should also be observed when servicing a gaseous oxygen system. Oxygen valves should be opened slowly

and filling should proceed slowly to avoid overheating. The hose from the refill source to the oxygen fill valve on the aircraft should be purged of air before it is used to transfer oxygen into the system. Pressures should also be checked frequently while refilling.

Airline and fixed-base operator maintenance shops often use oxygen filler carts to service oxygen systems. These contain several large oxygen supply cylinders connected to the fill cart manifold. This manifold supplies a fill hose that attaches to the aircraft. Valves and pressure gauges allow awareness and control of the oxygen dispensing process. (*Figure 15-9*)

Be sure all cylinders on the cart are aviator's breathing oxygen and that all cylinders contain at least 50 psi of oxygen pressure. Each cylinder should also be within its hydrostatic test date interval. After a cart cylinder has dispensed oxygen, the remaining pressure should be recorded. This is usually written on the outside of the cylinder with chalk or in a cylinder pressure log kept with the cart. As such, the technician can tell at a glance the status of each oxygen bottle. No pump or mechanical device is used to transfer oxygen from the fill cart manifold to the aircraft system. Objects under pressure flow from high pressure to low pressure. Thus, by connecting the cart to the aircraft and systematically opening oxygen cylinders with increasingly higher pressure, a slow increase in oxygen flow to the aircraft can be managed.

The following is a list of steps to safely fill an aircraft oxygen system from a typical oxygen refill cart.

1. Check hydrostatic dates on all cylinders, especially those that are to be filled on the aircraft. If a cylinder is out of date, remove and replace it with a specified unit that is serviceable.
2. Check pressures on all cylinders on the cart and in the aircraft. If pressure is below 50 psi, replace the cylinder(s). On the aircraft, this may require purging the system with oxygen when completed. Best practices dictate that any low-pressure or empty cylinder(s) on the cart should also be removed and replaced when discovered.
3. Take all oxygen handling precautions to ensure a safe environment around the aircraft.
4. Ground the refill cart to the aircraft.
5. Connect the cart hose from the cart manifold to the aircraft fill port. Purge the air from the refill hose with oxygen before opening the refill valve on the aircraft. Some hoses are equipped with purge valves to do this while the hose is securely attached to the aircraft. Others hoses need to be purged while attached to the refill fitting but not fully tightened.
6. Observe the pressure on the aircraft bottle to be filled. Open it. On the refill cart, open the cylinder with the closest pressure to the aircraft cylinder that exceeds it.
7. Open the aircraft oxygen system refill valve. Oxygen will flow from cart cylinder (manifold) into the aircraft cylinder.
8. When the cylinder pressures equalize, close the cylinder on the cart, and open the cart cylinder with the next highest pressure. Allow it to flow into the aircraft cylinder until the pressures equalize and flow ceases. Close the cart cylinder, and proceed to the cart cylinder with the next highest pressure.
9. Continue the procedure in step 8 until the desired pressure in the aircraft cylinder is achieved.
10. Close the aircraft refill valve, and close all cylinders on the cart.
11. The aircraft oxygen cylinder valve(s) should be left in the proper position for normal operations. Remotely mounted cylinders are usually left open.
12. Disconnect the refill line from the refill port on the aircraft. Cap or cover both.
13. Remove the grounding strap.



Figure 15-9. Typical oxygen servicing cart used to fill an aircraft system.

Temperature has a significant effect on the pressure of gaseous oxygen. Manufacturers typically supply a fill chart or a placard at the aircraft oxygen refill station to guide technicians in compensating for temperature/

pressure variations. Technicians should consult the chart and fill cylinders to the maximum pressure listed for the prevailing ambient temperature. (*Figure 15-10*)

When it is hot, oxygen cylinders are filled to a higher pressure than 1 800 psi or 1 850 psi, the standard maximum pressure ratings of most high-pressure aircraft oxygen cylinders. This is allowable because at altitude the temperature and pressure of the oxygen can decrease significantly. Filling cylinders to temperature compensated pressure values ensures a full supply of oxygen is available when needed.

When filling cylinders on a cold day, compensation for temperature and pressure changes dictates that cylinders be filled to less than the maximum rated capacity to allow for pressure increases as temperature rises. Strict adherence to the temperature/pressure compensation chart values is mandatory to ensure safe storage of aircraft oxygen.

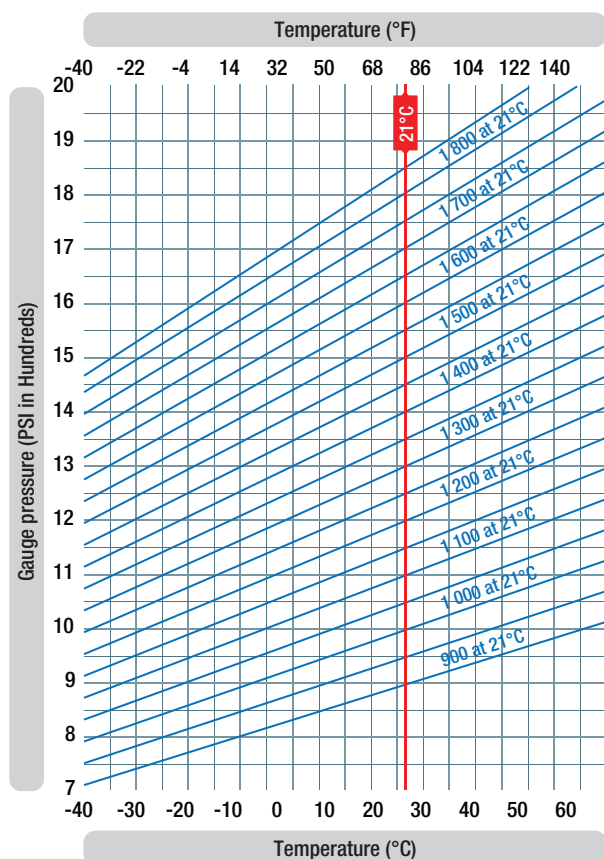


Figure 15-10. A temperature-compensating pressure refill chart is used by the technician to ensure proper oxygen cylinder pressure in the aircraft system.

NOTE: Some aircraft have temperature compensation features built into the refill valve. After setting the ambient temperature on the valve dial, the valve closes when the correct amount of oxygen pressure has been established in the aircraft cylinder. A chart can be used to ensure proper servicing.

OXYGEN SYSTEMS AND SUPPLY REGULATION

The use of gaseous oxygen in aviation is common; however, applications vary. On a light aircraft, it may consist of a small carry-on portable cylinder with a single mask attached via a hose to a regulator on the bottle. Larger portable cylinders may be fitted with a regulator that divides the outlet flow for 2-4 people. Built-in oxygen systems on high performance and light twin-engine aircraft typically have a location where oxygen cylinders are installed to feed a distribution system via tubing and a regulator. The passenger compartment may have multiple breathing stations plumbed so that each passenger can individually plug in a hose and mask if oxygen is needed. A central regulator is normally controlled by the flight crew who may have their own separate regulator and oxygen cylinder. Transport category aircraft may use a more elaborate built-in gaseous oxygen system as a backup system to cabin pressurization. In all of these cases, oxygen is stored as a gas at atmospheric temperature in high-pressure cylinders. It is distributed through a system with various components that are described in this section.

The design of the various oxygen systems used depends largely on the type of aircraft, its operational requirements, and whether the aircraft has a pressurization system. Systems are often characterized by the type of regulator used to dispense the oxygen: continuous-flow and demand flow. In some aircraft, a continuous-flow oxygen system is installed for both passengers and crew. The pressure demand system is widely used as a crew system, especially on the larger transport aircraft. Many aircraft have a combination of both systems that may be augmented by portable equipment.

CONTINUOUS-FLOW SYSTEMS

In its simplest form, a continuous-flow oxygen system allows oxygen to exit the storage tank through a valve and passes it through a regulator/reducer attached to the top of the tank. The flow of high-pressure oxygen passes through a section of the regulator that reduces

the pressure of the oxygen, which is then fed into a hose attached to a mask worn by the user. Once the valve is opened, the flow of oxygen is continuous. Even when the user is exhaling, or when the mask is not in use, a preset flow of oxygen continues until the tank valve is closed. On some systems, fine adjustment to the flow can be made with an adjustable flow indicator that is installed in the hose in line to the mask. A portable oxygen setup for a light aircraft exemplifies this type of continuous-flow system and is shown in **Figure 15-11**.

A more sophisticated continuous-flow oxygen system uses a regulator that is adjustable to provide varying amounts of oxygen flow to match increasing need as altitude increases. These regulators can be manual or automatic in design. Manual continuous-flow regulators are adjusted by the crew as altitude changes. Automatic continuous-flow regulators have a built in aneroid. As the aneroid expands with altitude, a mechanism allows more oxygen to flow through the regulator to the users. (**Figure 15-12**)

SYSTEM LAYOUT: CABIN (CONTINUOUS FLOW)

Many continuous-flow systems include a fixed location for the oxygen cylinders with permanent delivery plumbing installed to all passenger and crew stations in the cabin. Fully integrated oxygen systems usually have separate, remotely mounted components to reduce pressure and regulate flow. A pressure relief valve is also typically installed in the system, as is some sort of filter and a gauge to indicate the amount of oxygen pressure remaining in the storage cylinder(s). **Figure 15-13** diagrams the type of continuous-flow system that is found on small to medium sized aircraft.

Built-in continuous-flow gaseous oxygen systems accomplish a final flow rate to individual user stations through the use of a calibrated orifice in each mask. Larger diameter orifices are usually used in crew masks to provide greater flow than that for passengers. Special oxygen masks provide even greater flow via larger orifices for passengers traveling with medical conditions requiring full saturation of the blood with oxygen.

Allowing oxygen to continuously flow from the storage cylinder can be wasteful. Lowest sufficient flow rates can be accomplished through the use of rebreather apparatus. Oxygen and air that is exhaled still contains usable



Figure 15-11. A typical portable gaseous oxygen cylinder complete with valve, pressure gauge, regulator/reducer, hose, adjustable flow indicator, and rebreather cannula. A padded carrying case/bag can be strapped to the back of a seat in the cabin to meet certification and testing specifications.



Figure 15-12. A manual continuous flow oxygen system may have a regulator that is adjusted by the pilot as altitude varies. By turning the knob, the left gauge can be made to match the flight altitude thus increasing and decreasing flow as altitude changes.

oxygen. By capturing this oxygen in a bag, or in a cannula with oxygen absorbing reservoirs, it can be inhaled with the next breath, reducing waste. (**Figure 15-14**)

The passenger section of a continuous-flow oxygen system may consist of a series of plug-in supply sockets fitted to the cabin walls adjacent to the passenger seats to which oxygen masks can be connected. Flow is

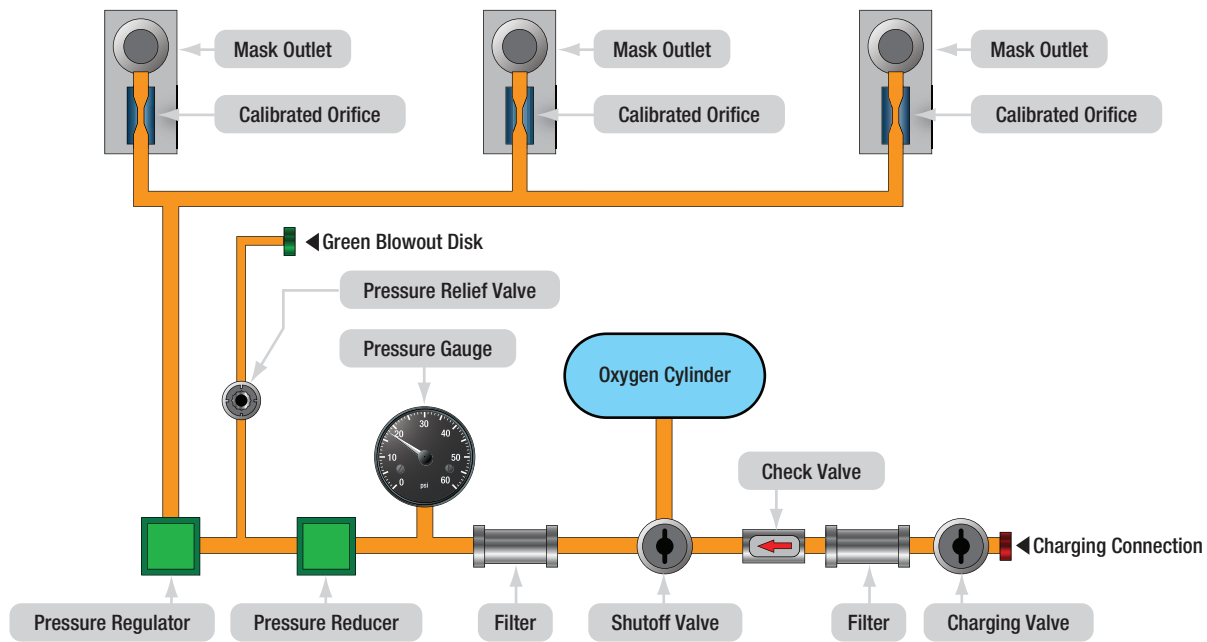


Figure 15-13. Continuous flow oxygen system found on small to medium size aircraft.

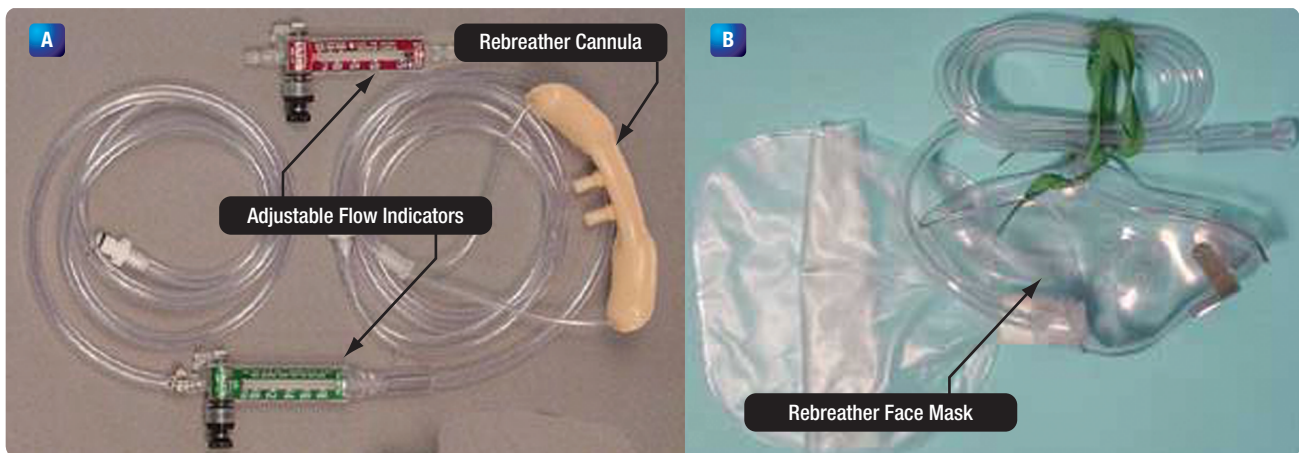


Figure 15-14. A rebreather cannula (A) and rebreather bag (B) capture exhaled oxygen to be inhaled on the next breath. This conserves oxygen by permitting lower flow rates in continuous flow systems. The red and green devices are optional flow indicators that allow the user to monitor oxygen flow rate. The type shown also contains needle valves for final regulation of the flow rate to each user.

inhibited until a passenger manually plugs in. When used as an emergency system in pressurized aircraft, such as an airliner, depressurization automatically triggers the deployment of oxygen ready continuous-flow masks at each passenger station. A lanyard attached to the mask turns on the flow to each mask when it is pulled toward the passenger for use. The masks are normally stowed overhead in the passenger service unit (PSU). (*Figure 15-15*) Deployment of the emergency continuous-flow passenger oxygen masks may also be controlled by the crew. (*Figure 15-16*)

Continuous-flow oxygen masks are simple devices made to direct flow to the nose and mouth of the wearer. They fit snugly but are not air tight. Vent holes allow cabin air to mix with the oxygen and provide escape for exhalation. In a rebreather mask, the vents allow the exhaled mixture that is not trapped in the rebreather bag to escape. This is appropriate, because this is the air-oxygen mixture that has been in the lungs the longest, so it has less recoverable oxygen to be breathed again. (*Figure 15-17*)

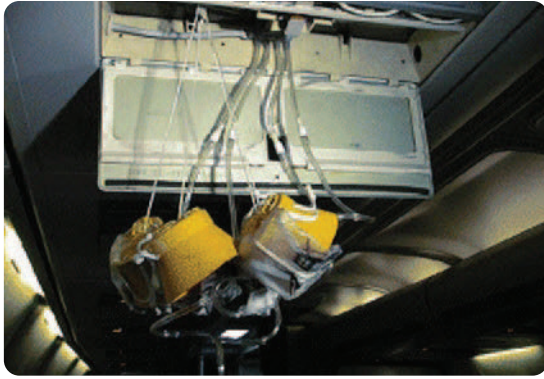


Figure 15-15. A passenger service unit (psu) is hinged over each row of seats in an airliner. Four yellow continuous flow oxygen masks are shown deployed. They are normally stored behind a separate hinged panel that opens to allow the masks to fall from the PSU for use.



Figure 15-17. Examples of different continuous-flow oxygen masks.



Figure 15-16. The crew can deploy passenger emergency continuous-flow oxygen masks and supply with a switch in the cockpit.

DEMAND-FLOW SYSTEMS

When oxygen is delivered only as the user inhales, or on demand, it is known as a demand-flow system. During the hold and exhalation periods of breathing, the oxygen supply is stopped. Thus, the duration of the oxygen supply is prolonged as none is wasted.

Demand-flow systems are similar to continuous-flow systems in that a cylinder delivers oxygen through a valve when opened. The tank pressure gauge, filter(s),

pressure relief valve, and any plumbing installed to refill the cylinder while installed on the aircraft are all similar to those in a continuous flow system. The high-pressure oxygen also passes through a pressure reducer and a regulator to adjust the pressure and flow to the user. But, demand-flow oxygen regulators differ significantly from continuous-flow oxygen regulators. They work in conjunction with close-fitting demand type masks to control the flow of oxygen. (*Figure 15-18*)

A pressure reduction occurs at the inlet of each individual demand regulator by limiting the size of the inlet orifice at the pressure reducing inlet valve. When the user inhales (demands) oxygen through the mask, this valve unseats and allows oxygen to flow through the regulator. When the user exhales, the valve reseats to conserve oxygen. There are two types of individual regulators: the diluter-demand type and the pressure demand type. (*Figure 15-19*)

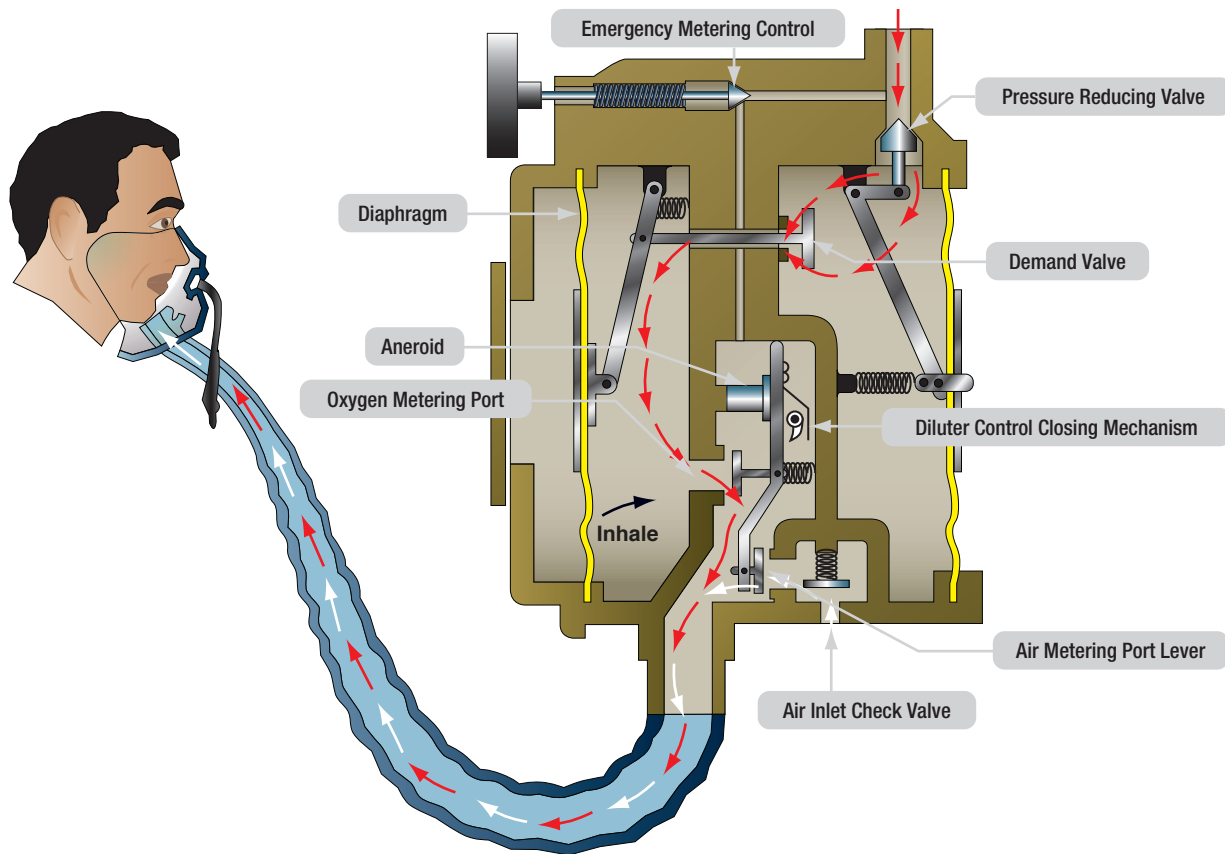


Figure 15-18. A demand regulator and demand-type mask work together to control flow and conserve oxygen. Demand-flow masks are close fitting so that when the user inhales, low pressure is created in the regulator, which allows oxygen to flow. Exhaled air escapes through ports in the mask, and the regulator ceases the flow of oxygen until the next inhalation.

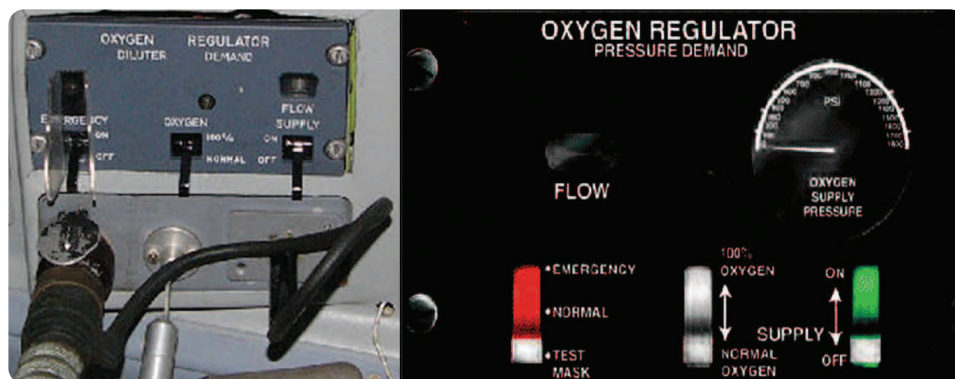


Figure 15-19. The two basic types of regulators used in demand flow oxygen systems. The panel below the diluter demand regulator on the left is available for mask hose plug in (left), lanyard mask hanger (center), and microphone plug in (right). Most high performance demand type masks have a microphone built-in.

DILUTER-DEMAND

The diluter-demand type regulator holds back the flow of oxygen until the user inhales with a demand type oxygen mask. The regulator dilutes the pure oxygen supply with cabin air each time a breath is drawn.

With its control toggle switch set to normal, the amount of dilution depends on the cabin altitude. As altitude increases, an aneroid allows more oxygen and less cabin air to be delivered to the user by adjusting flows through a metering valve. At approximately 34 000 feet, the diluter-demand regulator meters 100 percent oxygen. This should not be needed unless cabin pressurization fails.

Additionally, the user may select 100 percent oxygen delivery at any time by positioning the oxygen selection lever on the regulator. A built-in emergency switch also delivers 100 percent oxygen, but in a continuous flow as the demand function is bypassed. (*Figure 15-20*)

PRESSURE DEMAND

Pressure demand oxygen systems operate similarly to diluter demand systems, except that oxygen is delivered through the individual pressure regulator(s) under higher pressure. When the demand valve is unseated, oxygen under pressure forces its way into the lungs of the user. The demand function still operates, extending the overall supply of oxygen beyond that of a continuous-flow system. Dilution with cabin air also occurs if cabin altitude is less than 34 000 feet.

Pressure demand regulators are used on aircraft that regularly fly at 40 000 feet and above. They are also found on many airliners and high-performance aircraft that may not typically fly that high. Forcing oxygen into the lungs under pressure ensures saturation of the blood, regardless of altitude or cabin altitude. Both diluter-demand and pressure demand regulators also come in mask-mounted versions. The operation is essentially the same as that of panel-mounted regulators. (*Figure 15-21*)

SYSTEM LAYOUT: COCKPIT

Diluter-demand and pressure demand flow systems are used most frequently by the crew on high performance and air transport category aircraft. (*Figure 15-22*)

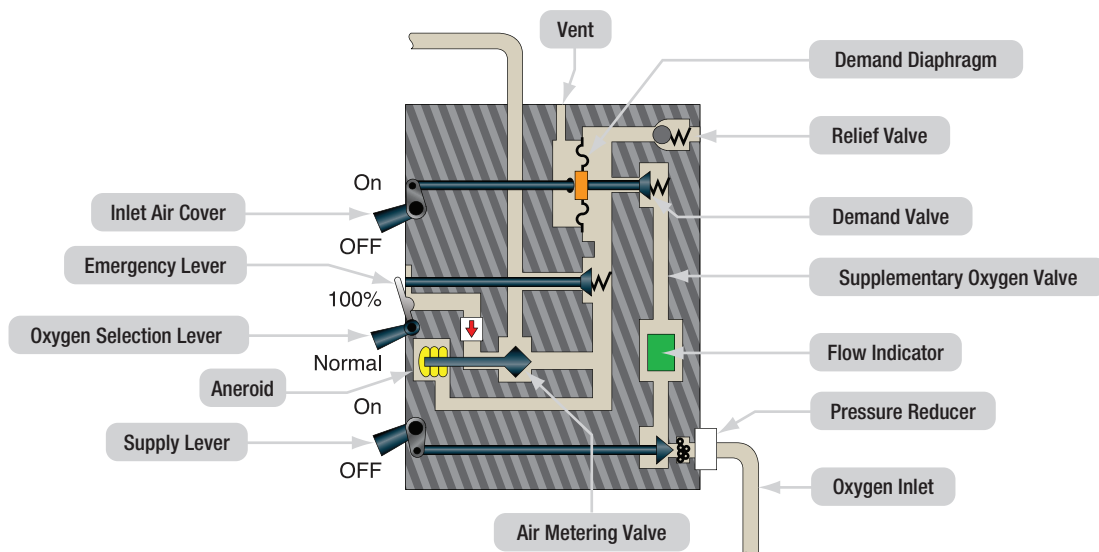


Figure 15-20. A diluter-demand regulator operates when low pressure caused by inhalation moves the demand diaphragm. A demand valve connected to the diaphragm opens, letting oxygen flow through the metering valve. The metering valve adjusts the mixture of cabin air and pure oxygen via a connecting link to an aneroid that responds to cabin altitude.



Figure 15-21. A mask-mounted version of a miniature diluter-demand regulator designed for use in general aviation (left), a mechanical quick-donning diluter-demand mask with the regulator on the mask (center), and an inflatable quick-donning mask (right). Squeezing the red grips directs oxygen into the hollow straps.

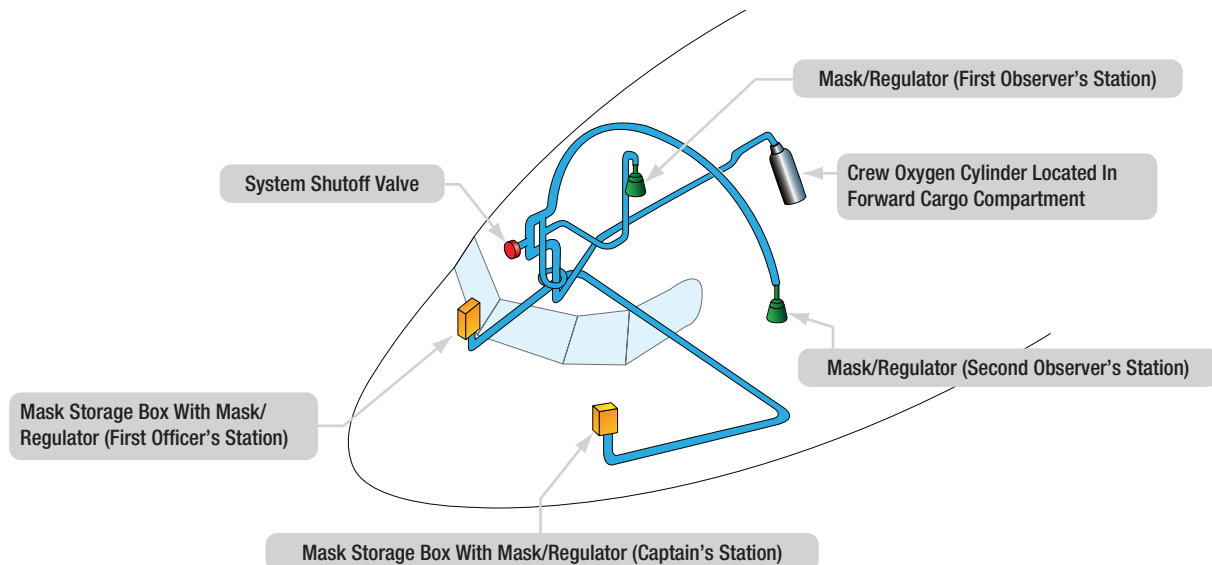


Figure 15-22. Location of demand-flow oxygen components on a transport category aircraft.

The tanks and servicing panel are remotely mounted, often near the forward baggage compartment. In large aircraft, separate storage cylinders for crew and passengers are typical. Also located near the tank is the system pressure reducing valve, sometimes called a pressure regulator. This device lowers the oxygen pressure from the storage cylinder(s) to roughly 60-85 psi. This oxygen is then distributed to the flight deck through a system of tubing and valves and into the individual demand regulators located at each crew station.

Each demand regulator is controlled by the user. A system shutoff valve is located upstream of the station regulators and masks. An electric switch, often on the flight deck overhead panel, controls the shutoff valve. Tank pressure indicators may be located near the switch. Manual/cable activation can be achieved through use of a handle below a panel in the flight deck flooring on Boeing aircraft.

CHEMICAL OXYGEN SYSTEMS

The two primary types of chemical oxygen systems are the portable type, much like a portable carry-on gaseous oxygen cylinder, and the fully integrated supplementary oxygen system used as backup on pressurized aircraft in case of pressurization failure. (*Figure 15-23*)

This latter use of solid chemical oxygen generators is most common on airliners. The generators are stored in the overhead PSU attached to hoses and masks for every passenger on board the aircraft. When a depressurization occurs, or the flight crew activates a



Figure 15-23. An oxygen generator mounted in place in an overhead passenger service unit of an air transport category aircraft.

switch, a compartment door opens and the masks and hoses fall out in front of the passengers. The action of pulling the mask down to a usable position actuates an electric current, or ignition hammer, that ignites the oxygen candle and initiates the flow of oxygen. Typically, 10 to 20 minutes of oxygen is available for each user. This is calculated to be enough time for the aircraft to descend to a safe altitude for unassisted breathing. Chemical oxygen systems are unique in that they do not produce the oxygen until it is time to be used. This allows safer transportation of the oxygen supply with less maintenance.

Chemical oxygen-generating systems also require less space and weigh less than gaseous oxygen systems supplying the same number of people. Long runs of tubing, fittings, regulators, and other components

are avoided, as are heavy gaseous oxygen storage cylinders. Each passenger row grouping has its own fully independent chemical oxygen generator. The generators, which often weigh less than a pound, are insulated and can burn completely without getting hot. The size of the orifice opening in the hose-attach nipples regulates the continuous flow of oxygen to the users.

ELECTRONIC PULSE DEMAND SYSTEMS

A recent development in general aviation oxygen systems is the electronic pulse demand oxygen delivery system (EDS). A small, portable EDS unit is made to connect between the oxygen source and the mask in a continuous flow oxygen system. It delivers timed pulses of oxygen to the wearer on demand, saving oxygen normally lost during the hold and exhale segments of the breathing cycle. Advanced pressure sensing and processing allows the unit to deliver oxygen only when an inhalation starts. It can also sense differences in users' breathing cycles and physiologies and adjust the flow of oxygen accordingly. A built-in pressure sensing device adjusts the amount of oxygen released as altitude changes. (*Figure 15-24*)

Permanently mounted EPD systems are also available. They typically integrate with an electronic valve/regulator on the oxygen cylinder and come with an emergency bypass switch to provide continuous-flow oxygen should the system malfunction. A liquid crystal display (LCD) monitor/control panel displays numerous system operating parameters and allows adjustments to the automatic settings. This type of electronic metering of oxygen has also been developed for passenger emergency oxygen use in airliners. (*Figure 15-25*)

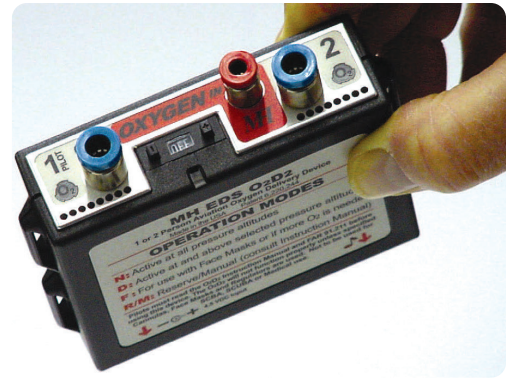


Figure 15-24. A portable two-person electronic pulse-demand (EPD) oxygen regulating unit.

LOX SYSTEMS

LOX systems are rarely used in civilian aviation. They may be encountered on former military aircraft now in the civilian fleet. As mentioned, the storage of LOX requires a special container system. The plumbing arrangement to convert the liquid to a usable gas is also unique. It basically consists of a controlled heat exchange assembly of tubing and valves. Overboard pressure relief is provided for excessive temperature situations. Once gaseous, the LOX system is the same as it is in any comparable gaseous oxygen delivery system. Use of pressure demand regulators and masks is common. Consult the manufacturer's maintenance manual for further information if a LOX system is encountered.

OXYGEN SYSTEM DISTRIBUTION

Tubing and fittings make up most of the oxygen system distribution plumbing and connect the various components of the oxygen system. Most oxygen lines are metal in permanent installations. High-pressure lines are usually stainless steel. Tubing lines in the low-pressure parts of the oxygen system are typically aluminum.

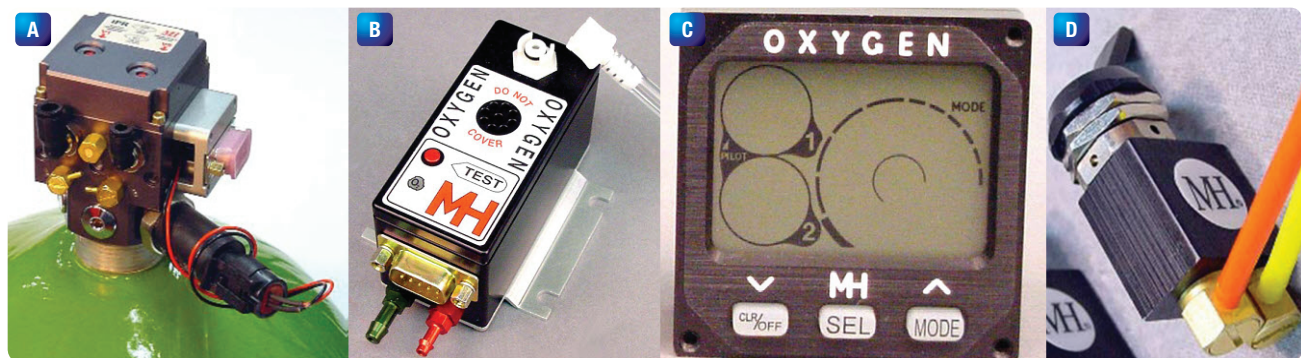


Figure 15-25. The key components of a built-in electronic pulse demand oxygen metering system: (A) electronic regulator, (B) oxygen station distributor unit, (C) command/display unit, (D) emergency bypass switch.

Flexible plastic hosing is used to deliver oxygen to the masks; its use is increasing in permanent installations to save weight. Installed oxygen tubing is usually identified with color coded tape applied to each end of the tubing, and at specified intervals along its length. The tape coding consists of a green band overprinted with the words "BREATHING OXYGEN" and a black rectangular symbol overprinted on a white background border strip. (*Figure 15-26*)



Figure 15-26. Color-coded tape used to identify oxygen tubing.

Tubing to tubing fittings in oxygen systems are often designed with straight threads to receive flared tube connections. Tubing to component fittings usually have straight threads on the tubing end and external pipe threads (tapered) on the other end for attachment to the component. The fittings are typically made of the same material as the tubing (i.e., aluminum or steel). Flared and flareless fittings are both used, depending on the system. Five types of valves are commonly found in high pressure gaseous oxygen systems: filler, check, shutoff, pressure reducer, and pressure relief. They function as they would in any other system with one exception: oxygen system shutoff valves are specifically designed to open slowly.

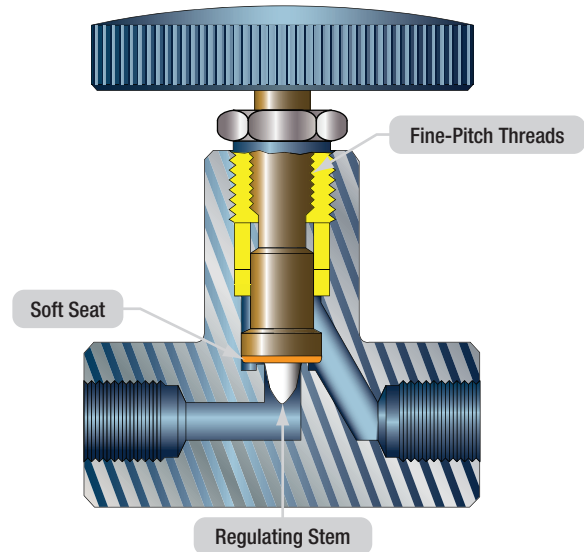


Figure 15-27. This high-pressure oxygen system shutoff valve has fine-pitch threads and a regulating stem to slow the flow of oxygen through the valve. A soft valve seat is also included to assure the valve closes completely.

The ignition point for any substances is lower in pure oxygen than it is in air. When high pressure oxygen is allowed to rush into a low pressure area, its velocity could reach the speed of sound. If it encounters an obstruction (a valve seat, an elbow, a piece of contaminant, etc.), the oxygen compresses. With this compression, known as adiabatic compression (since it builds so quickly no heat is lost to its surroundings), comes high temperature. Under pressure, this high temperature exceeds the ignition point of the material the oxygen encounters and a fire or explosion results.

A stainless steel line, for example, would not normally burn and is used for carrying numerous fluids under high pressure. But under high pressure and temperature in the presence of 100 percent oxygen, even stainless steel can ignite. To combat this issue, all oxygen shutoff valves are slow, opening valves designed to decrease velocity. (*Figure 15-27*)

Additionally, technicians should always open all oxygen valves slowly. Keeping oxygen from rushing into a low pressure area should be a major concern when working with high-pressure gaseous oxygen systems.

OXYGEN INDICATION AND WARNINGS

Oxygen cylinder valves and high-pressure systems are often provided with a relief valve should the desired pressure be exceeded. Often, the valve is ported to an indicating or blowout disk. This is located in a conspicuous place, such as the fuselage skin, where it can be seen during walk-around inspection. Most blowout disks are green. The absence of the green disk indicates the relief valve has opened, and the cause should be investigated before flight. (*Figure 15-28*) In addition to the blowout disc, various indications and warning are included in oxygen systems.

Flow indicators, or flow meters, are common in all oxygen systems. They usually consist of a lightweight object, or apparatus, that is moved by the oxygen stream. When flow exists, this movement signals the user in some way. (*Figure 15-29*)

Many flow meters in continuous flow oxygen systems also double as flow rate adjusters. Needle valves fitted

into the flow indicator housing can fine adjust the oxygen delivery rate. Demand-flow oxygen systems usually have flow indicators built into the individual regulators at each user station. Some contain a blinking device that activates when the user inhales and oxygen is delivered. Others move a colored pith object into a window. Regardless, flow indicators provide a quick verification that an oxygen system is functioning. (*Figure 15-29*)

Different types of independent flow indicators are used to provide verification that the oxygen system is functioning. Many demand-flow system flow indicators are built into the individual oxygen regulators as was seen *Figure 15-19*.

The amount of pressure in the supply tanks of an airliner using gaseous oxygen is an indication of the quantity of oxygen on board. Most tanks contain a direct reading gauge on the tank outlet. Maintenance personnel use this



Figure 15-28. An oxygen blowout plug on the side of the fuselage indicates when pressure relief has occurred and should be investigated.

gauge. The flight crew relies on a transducer that sends an electric signal to a gauge on the flight deck as was seen in *Figure 15-16*. On modern aircraft, transducer output is sent to the aircraft information management system for display of pressure on a multifunctional display screen. The system status page must be selected on the display select panel. (*Figure 15-30*)

Note that some aviators flying in smaller aircraft may use a portable pulse oximeters. This device measures the oxygen saturation level of the blood. With this information, adjustments to the oxygen flow rates of on board oxygen equipment can be made to prevent hypoxia. *Figure 15-31* shows an oximeter into which a finger is inserted to measure oxygen saturation of the blood in percentage. Heart rate is also displayed.

OXYGEN SYSTEM MAINTENANCE

All oxygen systems require servicing and maintenance. Various procedures and requirements used to perform common maintenance procedures are covered in this section.

LEAK TESTING GASEOUS OXYGEN SYSTEMS

Leaks in a continuous-flow oxygen system may be difficult to detect because the system is open at the user end. Blocking the flow of oxygen allows pressure to build and leak check procedures can be followed that are similar to those used in the high pressure sections of the systems.



Figure 15-29. Different flow indicators are used to provide verification that the oxygen system is functioning: continuous-flow, in-line (left); continuous-flow, in-line with valve adjuster (center); and old style demand flow (right).



Status Display

Figure 15-30. The display select panel and oxygen system status display on a Boeing 777 multifunctional display.



Figure 15-31. A portable pulse-type oximeter displays percentage of oxygen saturation of the blood and heart rate. Pilots can adjust oxygen supply levels to maintain saturation and avoid hypoxia.

Detection of leaks should be performed with oxygen-safe leak check fluid. This is a soapy liquid free from elements that might react with pure oxygen or contaminate the system. As with leak detection on an inflated tire or tube assembly, the oxygen leak detection solution is applied to the outside of fittings and mating surfaces. The formation of bubbles indicates a leak. (*Figure 15-32*)



Figure 15-32. Oxygen system leak check solution.

Careful assembly of oxygen components and fittings without over tightening or under-tightening is required. If a leak is found at a fitting, it should be checked for the proper torque. Tightening may not always stop the leak. If the fitting is torqued properly and a leak still exists, pressure must be released from the system and the fitting must be examined for flaws or contamination. If necessary, the fitting must be replaced. All system components, lines, and fittings must be replaced with the proper parts, which should be cleaned and inspected thoroughly before installation. Follow the manufacturer's instructions and repeat the leak check when completed.

Use caution when maintaining the high pressure portion of a gaseous oxygen system. An open tank valve pressurizes the lines and components with up to 1 850 pounds per square inch (psi) of oxygen. Identify the high-pressure section of the system as that portion upstream of the reducer or regulator that has stainless steel tubing.

NOTE: No attempt should be made to tighten a leaky oxygen fitting while the system is charging. The oxygen supply should be isolated in the cylinder and the system depressurized to reduce the consequences of a spark or to minimize spillage and injury should a complete fitting failure occur.

DRAINING AN OXYGEN SYSTEM

The biggest factor in draining an oxygen system is safety. The oxygen must be released into the atmosphere without causing a fire, explosion, or hazard. Draining outside is highly recommended. The exact method

of draining can vary. The basic procedure involves establishing a continuous flow in a safe area until the system is empty.

If the cylinder valve is operative, close the valve to isolate the oxygen supply in the cylinder. All that remains is to empty the lines and components. This can be done without disassembling the system by letting oxygen flow from the delivery point(s). If the environment is safe to receive the oxygen, positioning a demand-flow regulator to emergency delivers a continuous flow of oxygen to the mask when plugged in. Hang the mask(s) out of a window while the system drains. Plug in all mask(s) to allow oxygen to drain from a continuous-flow oxygen system. Systems without check valves can be drained by opening the refill valve.

PURGING AN OXYGEN SYSTEM

The inside of an oxygen system becomes completely saturated with oxygen during use. This is desirable to deliver clean, odor-free oxygen to the users and to prevent corrosion caused by contamination. An oxygen system needs to be purged if it has been opened or depleted for more than 2 hours, or if it is suspected that the system has been contaminated. Purging is accomplished to evacuate contaminants and to restore oxygen saturation to the inside of the system. The main cause of contamination in an oxygen system is moisture. In very cold weather, the small amount of moisture contained in the breathing oxygen can condense. With repeated charging, a significant amount of moisture may collect. Additionally, systems that are opened contain the moisture from the air that has entered. Damp charging equipment, or poor refill procedures, can also introduce water into the system. Always follow manufacturer's instructions when performing maintenance, refilling, or purging an oxygen system.

Cumulative condensation in an oxygen system cannot be entirely avoided. Purging is needed periodically. The procedure for purging may vary somewhat with each aircraft model. Generally speaking, oxygen is run through a sound oxygen system for a number of minutes at a given pressure to perform the purging. This can be as little as 10 minutes at normal delivery pressure. Other systems may require up to 30 minutes of flow at an elevated pressure. Regardless, the removal of contaminants and the re-saturation of the inside of the system with oxygen is the basis for purging. It is

acceptable to use nitrogen, or dry air, to blow through lines and components when performing maintenance. However, a final purging with pure oxygen is required before the system is serviceable for use.

It is important to ensure storage cylinders are refilled if they are used during the purging process. Be certain that there are no open lines and all safety caps are installed before returning the aircraft to service.

INSPECTION OF MASKS AND HOSES

The wide varieties of oxygen masks used in aviation require periodic inspection. Mask and hose integrity ensure effective delivery of oxygen to the user when it is needed. Sometimes this is in an emergency situation. Leaks, holes, and tears are not acceptable. Most discrepancies of this type are remedied by replacement of the damaged unit.

Some continuous-flow masks are designed for disposal after use. Be sure there is a mask for each potential user on board the aircraft. Masks designed to be reused should be clean, as well as functional. This reduces the danger of infection and prolongs the life of the mask. Various mild cleaners and antiseptics that are free of petroleum products can be used. A supply of individually wrapped alcohol swabs are often kept in the cockpit.

Built-in microphones should be operational. Donning straps and fittings should be in good condition and function so that the mask is held firm to the user's face. Note that the diameter of mask hoses in a continuous flow system is quite a bit smaller than those used in a demand flow system. This is because the inside diameter of the hose aids in controlling flow rate. Masks for each kind of system are made to only connect to the proper hose.

Smoke masks are required on transport aircraft and are used on some other aircraft as well. These cover the eyes, as well as the user's nose and mouth. Smoke masks are usually available within easy grasp of the crew members. They are used when the situation in the cockpit demands the increased level of protection offered. Smoke mask hoses plug into demand regulators in the same port used for regular demand type masks and operate in the same manner. Most include a built-in microphone. (*Figure 15-33*) Some portable oxygen systems are also fitted with smoke masks.



Figure 15-33. Smoke masks cover the eyes as well as the nose and mouth of the user.

REPLACEMENT OF TUBING, VALVES, AND FITTINGS

The replacement of aircraft oxygen system tubing, valves, and fittings is similar to the replacement of the same components in other aircraft systems. There is, however, an added emphasis on cleanliness and compatible sealant use. Any oxygen system component should be cleaned thoroughly before installation. Often tubing comes with leftover residue from the bending or flaring processes. Cleaning should be accomplished with non-petroleum-based cleansers. Trichlorethylene, acetone, and similar cleaners can be used to flush new tubing. Tubing should be blown or baked dry before installation. Follow the manufacturer's procedures for cleaning oxygen system components.

Some oxygen components make use of tapered pipe fittings. This type of connection is usually sealed with the application of thread lubricant/sealant. Typical thread sealers are petroleum based and should not be used; only oxygen compatible thread lubricant/sealers should be used. Alternatively, Teflon™ tape is also used on oxygen pipe fitting connections. Be sure to begin wrapping the Teflon™ tape at least two threads from the end of the fitting. This prevents any tape from coming loose and entering the oxygen system.

OXYGEN SYSTEM SAFETY

Precautions must be observed when working with or around pure oxygen. It readily combines with other substances, some in a violent and explosive manner. As mentioned, it is extremely important to keep distance between pure oxygen and petroleum products. When allowed to combine, an explosion can result. Additionally, there are a variety of inspection and maintenance practices that should be followed to ensure safety when working with oxygen and oxygen systems. Care should be used and, as much as possible, maintenance should be done outside.

When working on an oxygen system, it is essential that the warnings and precautions given in the aircraft maintenance manual be carefully observed. Before any work is attempted, an adequate fire extinguisher should be on hand. Cordon off the area and post NO SMOKING placards. Ensure that all tools and servicing equipment are clean and avoid power on checks and use of the aircraft electrical system.

When working around oxygen and oxygen systems, cleanliness enhances safety. Clean, grease-free hands, clothes, and tools are essential. A good practice is to use only tools dedicated for work on oxygen systems. There should be absolutely no smoking or open flames within a minimum of 50 feet of the work area. Always use protective caps and plugs when working with oxygen cylinders, system components, or plumbing. Do not use any kind of adhesive tape. Oxygen cylinders should be stored in a designated, cool, ventilated area in the hanger away from petroleum products or heat sources.

Oxygen system maintenance should not be accomplished until the valve on the oxygen supply cylinder is closed and pressure is released from the system. Fittings should be unscrewed slowly to allow any residual pressure to dissipate. All oxygen lines should be marked and should have at least 2 inches of clearance from moving parts, electrical wiring, and all fluid lines. Adequate clearance must also be provided from hot ducts and other sources that might heat the oxygen. A pressure and leak check must be performed each time the system is opened for maintenance. Do not use any lubricants, sealers, cleaners, etc., unless specifically approved for oxygen system use.

Question: 15-1

The atmosphere of the earth contains about _____ % oxygen.

Question: 15-5

When _____ is ignited, it produces oxygen as it burns.

Question: 15-2

What are two ways the negative effects of reduced atmospheric pressure at flight altitudes forcing less oxygen into the blood can be overcome?

Question: 15-6

Built-in oxygen systems contain _____ designed to refill gaseous oxygen cylinders while they are in place.

Question: 15-3

Pure oxygen combines violently with _____, which creates a significant hazard when handling these materials in close proximity to each other.

Question: 15-7

_____ has a significant effect on the pressure of gaseous oxygen.

Question: 15-4

To ensure continued serviceability, oxygen cylinders must be _____ periodically.

Question: 15-8

Transport category aircraft use a more elaborate, built-in _____ oxygen system as a backup system to cabin pressurization.

ANSWERS

Answer: 15-1

21.

Answer: 15-5

sodium chlorate.

Answer: 15-2

Increase the pressure of the oxygen or,
Increase the quantity of oxygen in the air mixture.

Answer: 15-6

plumbing.

Answer: 15-3

petroleum products.

Answer: 15-7

Temperature.

Answer: 15-4

hydrostatically tested.

Answer: 15-8

gaseous.

Question: 15-9

Built-in _____ gaseous oxygen systems accomplish the final flow rate to an individual user station through the use of a calibrated orifice in each mask.

Question: 15-13

An oxygen _____ usually consists of a lightweight object, or apparatus, that is moved by the oxygen stream.

Question: 15-10

When oxygen is delivered only as the user inhales, it is known as a _____ system.

Question: 15-14

To purge an oxygen system, pure _____ is used.

Question: 15-11

Two types of oxygen regulators for flight crew use on jet transport aircraft are _____ and _____.

Question: 15-15

Oxygen fittings should be unthreaded _____ to allow any residual pressure to dissipate.

Question: 15-12

High-pressure oxygen distribution lines are usually made from _____.

Question: 15-16

Detection of leaks should be performed with _____.

ANSWERS

Answer: 15-9
continuous-flow.

Answer: 15-13
flowmeter (flow indicator).

Answer: 15-10
demand-flow.

Answer: 15-14
oxygen.

Answer: 15-11
demand flow.
diluter demand.
pressure demand.

Answer: 15-15
slowly.

Answer: 15-12
stainless steel.

Answer: 15-16
oxygen-safe leak check fluid.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 16

PNEUMATIC/VACUUM (ATA 36)

Knowledge Requirements

11.16 - Pneumatic/Vacuum (ATA 36)

System lay-out: cockpit, cabin;
Sources, storage, charging and distribution;
Supply regulation;
Indications and warnings;
Interfaces with other systems.

3

PNEUMATIC/VACUUM
(ATA 36)

11.16 - PNEUMATIC/VACUUM

VACUUM SYSTEMS

Large transport aircraft powered by reciprocating engines may contain original instrument technology that includes gyroscopic instruments powered by a vacuum system

SYSTEM LAYOUT

Twin-engine aircraft vacuum systems contain an engine-driven vacuum pump on each engine. The associated lines and components for each pump are isolated from each other and act as two independent vacuum systems. The vacuum lines are routed from each vacuum pump through a vacuum relief valve and through a check valve to the vacuum four-way selector valve. The four-way valve permits either pump to supply a vacuum manifold.

From the manifold, flexible hoses connect the vacuum operated instruments into the system. To reduce the vacuum for the turn and bank indicators, needle valves are included in both lines to these units. Lines to the artificial horizons and the directional gyro receive full vacuum. From the instruments, lines are routed to the vacuum gauge through a turn and bank selector valve.

This valve has three positions: main, left turn and bank (T&B), and right T&B. In the main position, the vacuum gauge indicates the vacuum in the lines of the artificial horizons and directional gyro. In the other positions, the lower value of vacuum for the turn and bank indicators are displayed.

A schematic of this type of twin-engine aircraft vacuum system is shown in *Figure 16-1*. Note the following components: two engine-driven pumps, two vacuum relief valves, two flapper type check valves, a vacuum manifold, a vacuum restrictor for each turn and bank indicator, an engine four-way selector valve, one vacuum gauge, and a turn-and-bank selector valve. Not shown are system and individual instrument filters. A drain line may also be installed at the low point in the system.

COCKPIT

Much of the vacuum system is located behind the instrument panel. Mounted in the panel are the 4-way selector valve, turn and bank selector valve and a system vacuum gauge. Tubing and connections for the gyroscopic instruments are behind the instrument panel.

CABIN

No vacuum system components in the passenger cabin.

PNEUMATIC SYSTEMS

Pneumatic and hydraulic systems are similar in that they use confined fluids. Since liquids and gases flow, they are both considered fluids; however, there is a great difference in the characteristics of the two. Liquids are practically incompressible; a quart of water still occupies about a quart of space regardless of how hard it is compressed. But gases are highly compressible; a quart of air can be compressed into a thimbleful of space. In spite of this difference, gases and liquids are both fluids which are confined and made to transmit power. The type of unit used to provide pressurized air for pneumatic systems is determined by the system's air pressure requirements.

HIGH PRESSURE SYSTEMS

In the past, some aircraft manufacturers equipped their aircraft with a high pressure pneumatic system (3 000 psi). An aircraft that utilizes this type of system is the Fokker F27. Such systems operate a great deal like hydraulic systems, except they employ air instead of a liquid for transmitting power. High pressure pneumatic systems are sometimes used for:

- Brakes.
- Opening and closing doors.
- Driving hydraulic pumps, alternators, starters, water injection pumps, etc.
- Operating emergency devices (brakes-landing gear).

SYSTEM LAYOUT

Many high pressure pneumatic systems are installed for one time emergency or back-up use and are completely discharged when used. They use pressurized air or nitrogen containers with no on-board means provided to repressurize the system once deployed. Other high pressure pneumatic systems use pressurized containers that are re-charged during flight through the action of compressors installed in the system. This type of installation allows the pneumatic system to operate components repeatedly rather than just once in a manner similar to a hydraulic system. *Figure 16-2* shows a typical layout of a high pressure pneumatic system on a twin reciprocating engine aircraft equipped with on-board compressors.

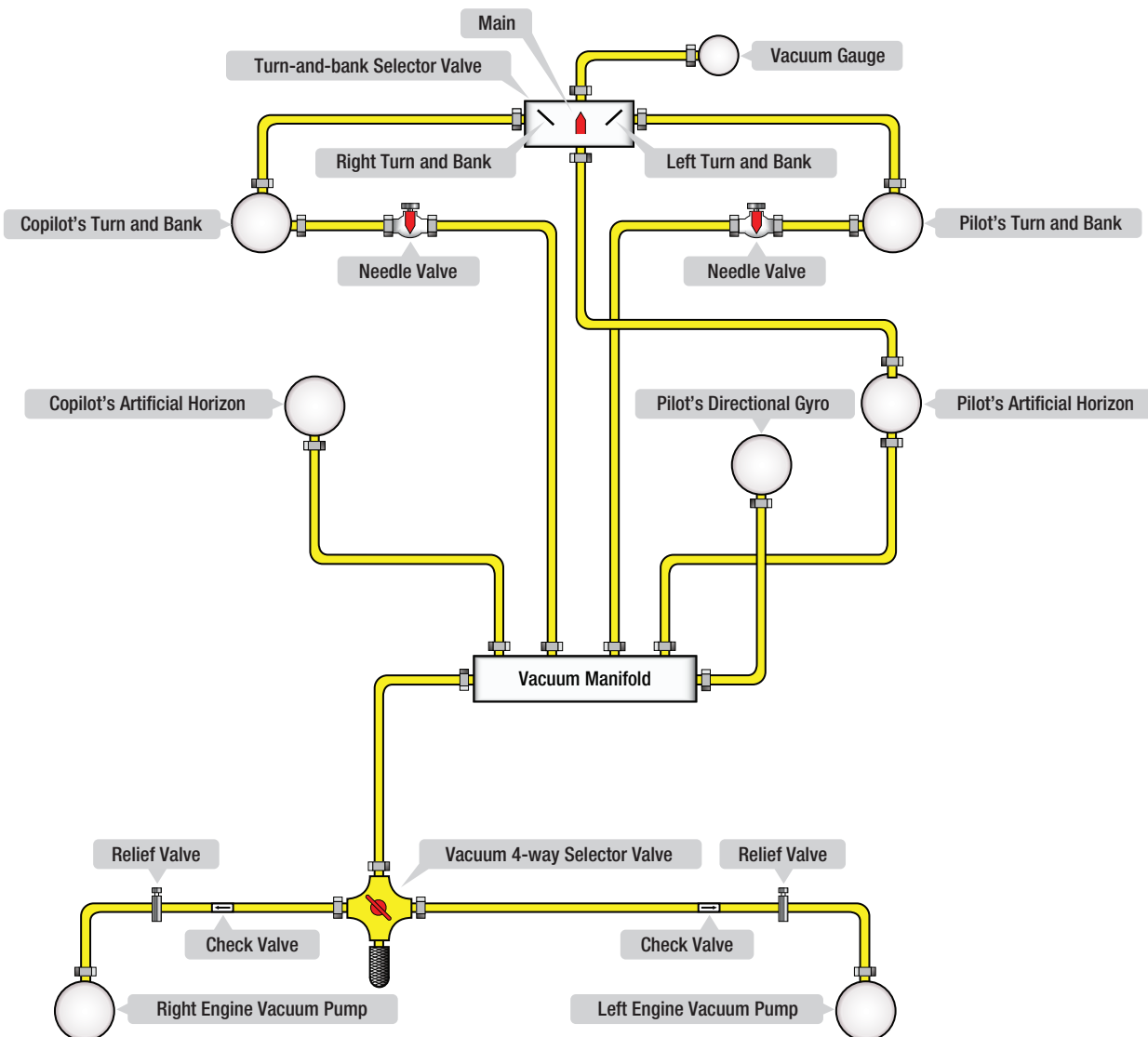


Figure 16-1. An example of a twin-engine instrument vacuum system.

SOURCES

Sources for high pressure pneumatic systems include engine-driven and other on-board compressors, ground air, and ground nitrogen sources.

As stated, some aircraft employ permanently installed air compressors which recharge air bottles whenever pressure is used for operation of a unit. Several types of compressors are used for this purpose. Some have two stages of compression, while others have three, depending on the maximum desired operating pressure. Details on compressor operation are found in the aircraft maintenance manual. They are typically oil lubricated thus the system plumbing may contain an oil separator of some type as well as a means for removing moisture in the system.

Alternately, air and nitrogen storage containers for pneumatic systems are filled on the ground with either a ground-based compressor or a high pressure bottle transfer for nitrogen.

STORAGE

For high-pressure systems, air is usually stored in metal bottles at pressures ranging from 1 000 to 3 000 psi, depending on the particular system. The high pressure storage cylinder is typically a light weight, wire-wrapped, steel-constructed reservoir. The date of manufacture and safe working pressure should be on the reservoir as well as a date stamped for the performance of the last hydrostatic test. It is common practice for these high pressure containers to be inspected often and removed periodically for hydrostatic checks.

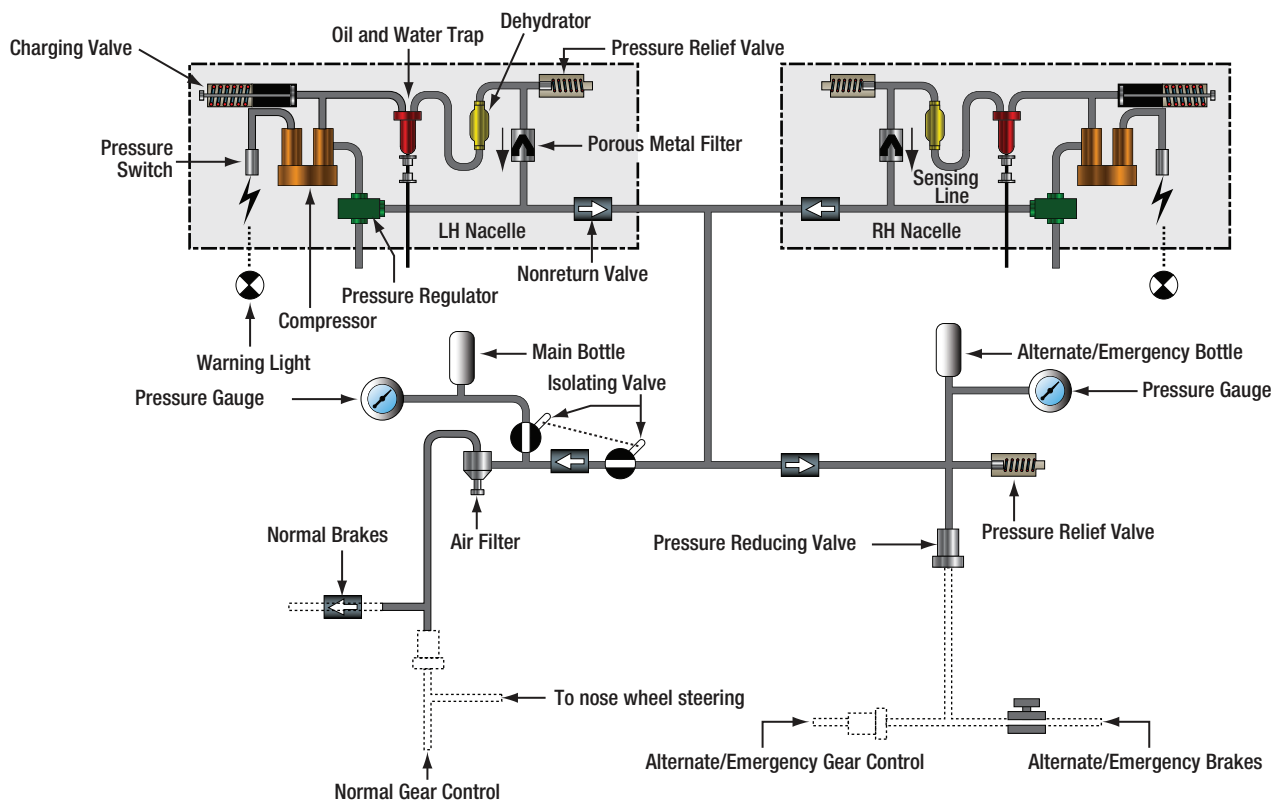


Figure 16-2. High-pressure pneumatic system.

A standpipe is commonly used at the discharge port to prevent any water that has collected inside the container from entering the system. Air flowing out of the container must go through the end of the standpipe which is elevated above any conceivable water level. This type of container is used in both one-time and multi-deployment systems.

CHARGING

Charging of high pressure bottles is done with either an on-board compressor or a ground source. The typical high-pressure storage bottle has two ports, one of which is equipped with a charging valve. A ground-operated compressor or air bottle can be connected to this valve to add air. Nitrogen may also be introduced through this valve.

For on-board charging, the charging valve is plumbed to the compressor outlet. The other valve on a typical high pressure pneumatic reservoir is a control valve. It acts as a shutoff valve, keeping air trapped inside the bottle until the system is operated. This valve must be opened when installed in a chargeable system. Reservoir contents stay held in the bottle with system pressure. A pressure switch is used for flight deck warnings.

DISTRIBUTION

Pneumatic power is distributed through high pressure steel or stainless steel lines. The use of check valves is common to prevent back flow. The lines are routed in the same manner as hydraulic lines to reach the components. In systems that operate one time and emergency systems, a shuttle valve is often used to close off the normal system flow and allow flow of high pressure pneumatic air to operate the component.

SUPPLY REGULATION

A pressure regulator maintains system pressure with a relief valve to limit pressure in case of regulator failure. Check valves are used to prevent back flow to the compressor. In addition to the use of a selector valve or control valve to direct the air to the portion of the system through which it must be distributed, isolation valves are often installed in the distribution system to isolate working components from those that are inoperative or to isolate part of the system that has a leak.

NOTE: All components in a high pressure pneumatic system do not necessarily operate at full system pressure. Pressure reducing valves are used to lower the system pressure to that required by a particular component or

sub system. Restrictors and variable restrictors are used to control the speed of the component(s) operated by pneumatic. (*Figure 16-3*)

The few high pressure pneumatic systems on aircraft that the technician encounters are either one time use or multi-use or both. *Figure 16-4* illustrates part of a pneumatic system that uses a rechargeable system for normal operation of gear extension and retraction as well as brake operation. For emergency brake application, a completely redundant distribution system supplies high pressure air from a reservoir (not shown) independent of the normal system.

EMERGENCY BACK-UP SYSTEMS AND PNEUDRAULICS

Many aircraft use a high-pressure pneumatic back-up source of power to extend the landing gear or actuate the brakes if the main hydraulic braking system fails. High pressure nitrogen is not directly used to actuate the landing gear actuators or brake units but, instead, it applies the pressurized nitrogen to move hydraulic fluid to the actuator. This process is called pneudraulics. The following paragraph discusses the components and operation of an emergency pneumatic landing gear extension system used on a business jet. (*Figure 16-5*)

Nitrogen Bottles

Nitrogen used for emergency landing gear extension is stored in two bottles, one bottle located on each side of the nose wheel well. Nitrogen from the bottles is released by actuation of an outlet valve. Once depleted, the bottles must be recharged by maintenance personnel. Fully serviced pressure is approximately 3 100 psi at 70°F (21°C) enough for only one extension of the landing gear.

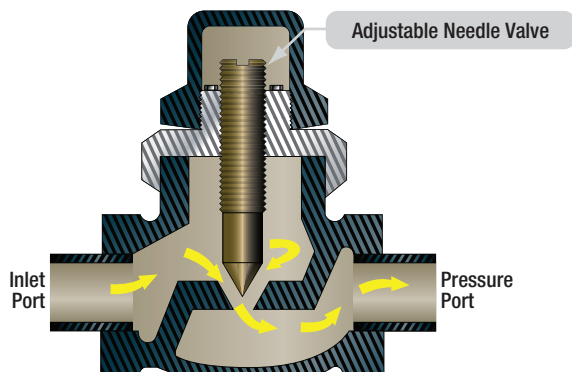


Figure 16-3. Variable pneumatic restrictor.

Gear Emergency Extension Cable and Handle

The outlet valve is connected to a cable and handle assembly. The handle is located on the side of the copilot's console and is labeled EMER LDG GEAR. Pulling the handle fully upward opens the outlet valve, releasing compressed nitrogen into the landing gear extension system. Pushing the handle fully downward closes the outlet valve and allows any nitrogen present in the emergency landing gear extension system to be vented overboard. The venting process takes approximately 30 seconds.

Dump Valve

As compressed nitrogen is released to the landing gear selector/dump valve during emergency extension, the pneudraulic pressure actuates the dump valve portion of the landing gear selector/dump valve to isolate the landing gear system from the remainder of hydraulic system. When activated, a blue DUMP legend is illuminated on the LDG GR DUMP V switch, located on the flight deck overhead panel. A dump valve reset switch is used to reset the dump valve after the system has been used and serviced.

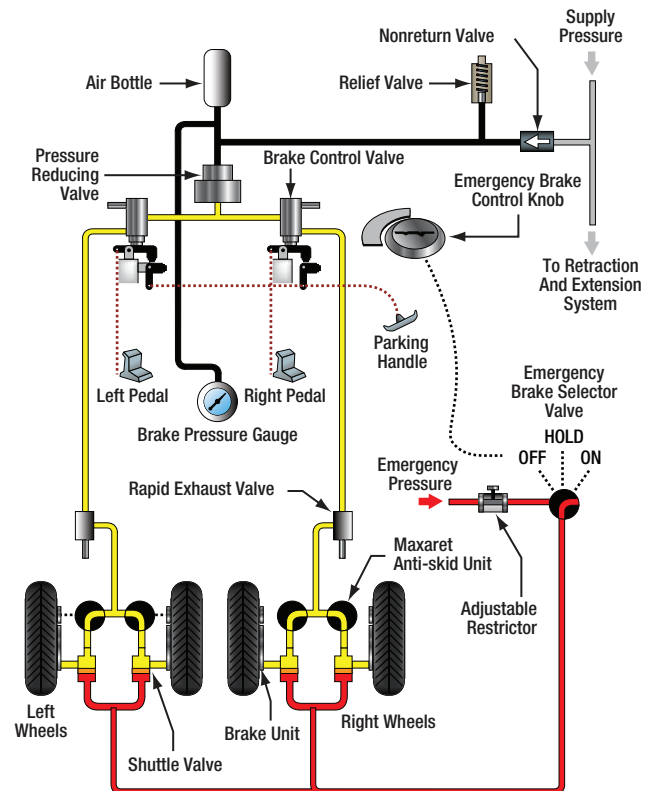


Figure 16-4. Normal rechargeable and emergency non-rechargeable pneumatic brake systems on the same aircraft.

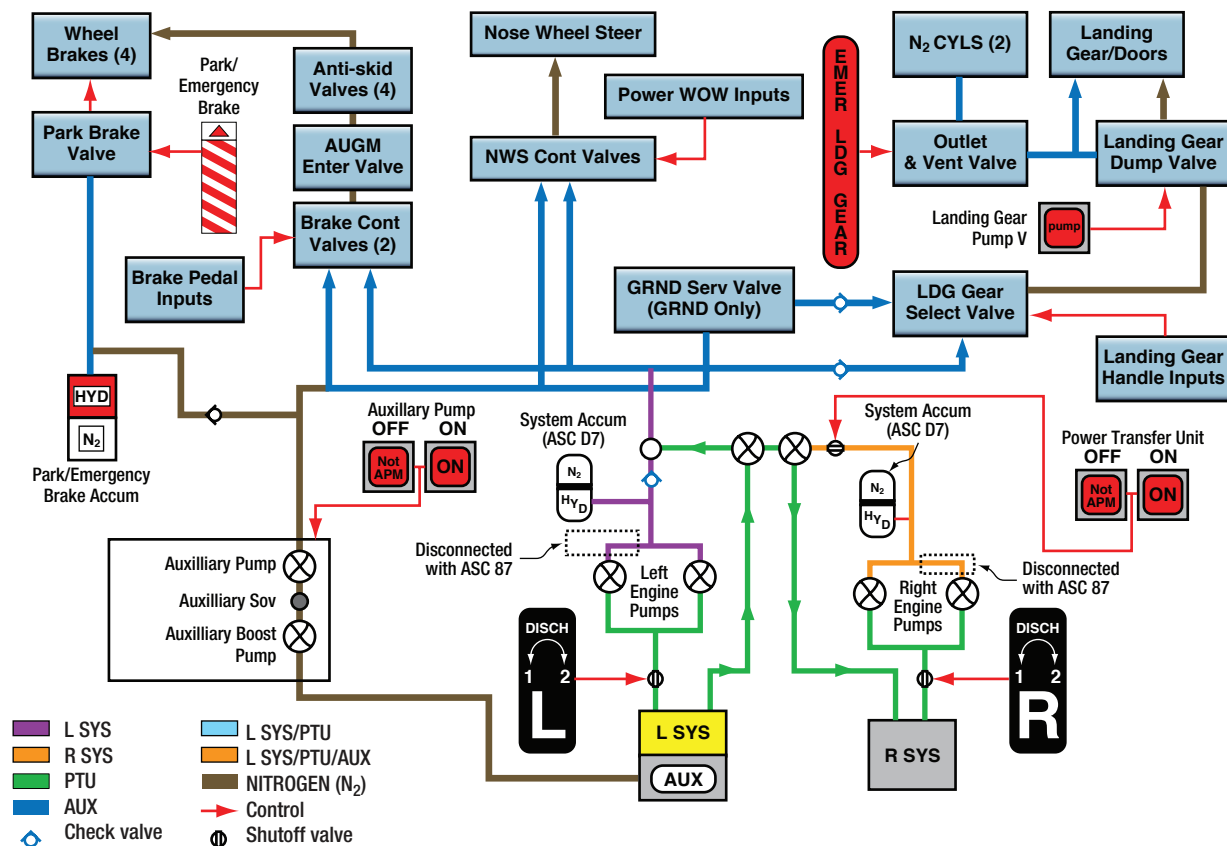


Figure 16-5. Pneumatic emergency landing gear extension system.

Emergency Extension Sequence:

1. Landing gear handle is placed in the DOWN position.
2. Illuminated Red light in the landing gear control handle.
3. EMER LDG GEAR handle is pulled fully outward.
4. Compressed nitrogen is released to the landing gear selector/dump valve.
5. Pneudraulic pressure actuates the dump valve portion of the landing gear selector/dump valve.
6. Blue DUMP legend is illuminated on the LDG GR DUMP switch.
7. Landing gear system is isolated from the remainder of hydraulic system.
8. Pneudraulic pressure is routed to the OPEN side of the landing gear door actuators, the UNLOCK side of the landing gear uplock actuators, and the EXTEND side of the main landing gear sidebrace actuators and nose landing gear extend/retract actuator.
9. Landing gear doors open.
10. Uplock actuators unlock.
11. Landing gear extends down and locks.

12. Three green DOWN AND LOCKED lights on the landing gear control panel are illuminated.
13. Landing gear doors remain open.

HIGH PRESSURE PNEUMATIC POWER SYSTEM MAINTENANCE

Maintenance of high pressure pneumatic power system consists of servicing, troubleshooting, removal, and installation of components, and operational testing.

The air compressor's lubricating oil level should be checked daily in accordance with the applicable manufacturer's instructions. The oil level is indicated by means of a sight gauge or dipstick. When refilling the compressor oil tank, the oil (type specified in the applicable instructions manual) is added until the specified level. After the oil is added, ensure that the filler plug is torqued and safety wire is properly installed.

The pneumatic system should be purged periodically to remove the contamination, moisture, or oil from the components and lines. Purging the system is accomplished by pressurizing it and removing the plumbing from various components throughout the

system. Removal of the pressurized lines causes a high rate of airflow through the system, causing foreign matter to be exhausted from the system. If an excessive amount of foreign matter, particularly oil, is exhausted from any one system, the lines and components should be removed and cleaned or replaced. Upon completion of pneumatic system purging and after reconnecting all the system components, the system air bottles should be drained to exhaust any moisture or impurities that may have accumulated there.

After draining the air bottles, service the system with nitrogen or clean, dry, compressed air. The system should then be given a thorough operational check and an inspection for leaks and security.

MEDIUM PRESSURE PNEUMATIC SYSTEMS

SYSTEM LAYOUT

Medium pressure pneumatic systems on large passenger aircraft are typically designed around the sources for pneumatic air that feed a common manifold. Each engine contains an independent bleed air subsystem that is designed to extract and regulate pneumatic bleed air from the engine. It is then forwarded to the pneumatic manifold for use. The pneumatic manifold contains the control valves that are operated to supply the systems that require pneumatic power.

An isolation valve separates the pneumatic manifold from each engine bleed air supply and regulation subsystem so as to be able to turn the supply ON and OFF from that engine. The APU is similarly designed although the APU may turn a dedicated load compressor to supply the air rather than tapping bleed air off the compressor section of the engine. A pneumatic power supply cart provides already regulated air pressure. When it is used to supply the manifold, the aircraft engines are not operated. A ground pneumatic air supply adapter with check valve is located directly in the pneumatic manifold. Closing the engine and APU isolation valves isolates the ground air supply. The supply cart must be powered down to deenergize the pneumatic manifold and remove the hose.

SOURCES

A medium pressure pneumatic system (35-150 psi) does not include an air bottle/storage reservoir. Instead, it draws air from the compressor section of a turbine engine. This is known as bleed air and is used to provide pneumatic power for engine starts, engine de-icing, wing de-icing, air conditioning and more. In some cases, it provides hydraulic power to the aircraft systems (if the hydraulic system is equipped with an air driven hydraulic pump). Engine bleed air is also used to pressurize the aircraft's hydraulic reservoirs, anti-ice the TAT probe and other applications specific to particular aircraft.

Ground sources of pneumatic air also are used. Fixed and portable cart type units containing engine-driven air supply compressors are connected into the pneumatic manifold to power the pneumatic system without running the engines. A ground air supply connector and check valve is provided in the manifold for the duct-diameter sized hose from the ground source.

STORAGE

Bleed air pneumatic systems normally do not store pneumatic air in any particular container like the reservoir bottles of a high pressure pneumatic systems. Each turbine engine and the APU supply the bleed air. A shutoff or regulating and shutoff valve is typically located between the engine bleed air tap-offs and the pneumatic ducting that makes up the pneumatic manifold. A shutoff type valve is also used to control the flow of pneumatic air from the APU.

Thus, the pneumatic manifold, which is typically 4 inch diameter ducting, may be considered a storage location. It is located downstream of the pneumatic shutoff valves from the engines and APU. Control valves allow pneumatic air to be routed from the manifold into pneumatically powered components such as engine starters, pneumatically driven hydraulic pumps, and into the wing anti-ice ducts and air conditioning packages.

PRESSURE CONTROL

Airliner pneumatic system pressure control begins with control of engine compressor bleed air. Intermediate-stage compressor bleed air normally supplies the bulk of the pneumatic system demand. However, in times of high demand or reduced engine throttle, a second, and sometimes a third tap off of high stage compressor bleed air is combined with intermediate stage air to

main sufficient volume for operating pneumatic system component demands. Various pressure regulating and sourcing valves are used to deliver the optimum volume of air into the pneumatic manifold at any given time.

On the most modern aircraft, regulation is maintained electronically. Digital data buses supply inputs to central pneumatic system control computers. The computers set the position of the various valves in the system to meet demand. The dominant use of pneumatic air is cabin air conditioning. On a Boeing 777, therefore, the computer that regulates the pneumatic system is called the air supply cabin pressure controller (ASCPC). There are two ASCPCs located in the main equipment center. They use data about the air sources and air end user components and sub-systems to select the regulating valve positions. Environmental conditions, engine, APU and airframe status conditions as well as flight status conditions are all factors considered for control.

The data for automatic operation comes from these:

- Airfoil and cowl ice protection system (ACIPS)
- Autopilot flight director system (AFDS)
- Airplane information management system (AIMS)
- Air supply cabin pressure controllers (ASCPC)
- Auxiliary power unit controller (APUC)
- Cabin temperature controller (CTC)
- Duct leak and overheat detection (DLODS)
- ECS miscellaneous card (ECSMC)
- Electronic engine control (EEC)
- Electrical load management system (ELMS)
- Flap slat electronics unit (FSEU)
- Hydraulic interface module (HYDIM) cards
- Overhead panel ARINC 629 system (OPAS)
- Warning electronic unit (WEU)
- Weight on wheels (WOW) cards

Figure 16-6 illustrates the pneumatic system on a 777. It shows how the data buses provide a flow of information to and from the pneumatic system, its components and the myriad of other related systems on the aircraft listed above. Older airliners have fewer inputs to regulating and shutoff valves than modern aircraft. Most valves modulate with a handful or fewer inputs from sensors, sensing lines, related valves, components and pneumatic manifold pressure. Systems are designed so that pneumatic manifold pressure remains relatively constant despite the demands of pneumatic components. Cabin air conditioning, in particular requires a relatively large

continuous flow of pneumatic air. Reduced pressure in the pneumatic manifold is fed back to the mechanism that controls the valve positions of regulating and shutoff valves for various compressor bleed air stages. Typically these are positioned to maintain volume requirements by manipulating valve positions to maintain a set pressure in the manifold. (**Figure 16-7**) On this aircraft, the 13th stage modulating and shutoff valve supplies any additional air required to the pneumatic manifold.

Pressure relief valves are provided to protect pneumatic ducts from excessive pressure. Typically, a relief valve is located in each section of the pneumatic manifold separated by an isolation or shutoff valve. The APU supply duct may also have a relief valve. Relief settings on some Boeing 737's are in the 80-110 psi range.

Without the benefit of computer control found on the most modern aircraft, many aircraft control and operate bleed air pneumatic system pressure regulating valves solely with pneumatic pressure. No electricity is needed. Pneumatic pressure in the pneumatic manifold is routed to a pressure regulator that also receives pneumatic/bleed air input from other locations. Pneumatic temperature inputs are also used. The regulator's internal mechanism balances the air inputs as required by system demands. This is largely done with springs and chambers with diaphragms for comparing pressures. Pneumatic lines then run from the regulator output to pressure regulating and shutoff valve mechanism(s) which modulate valve position accordingly using the pressurized air signal from the regulator.

DISTRIBUTION

The engine bleed air distribution system interconnects the engine bleeds of the engines and APU and contains the necessary valves to shut off bleed air at each engine and isolate various ducts. The medium pressure pneumatic system is generally characterized by the use of 3-4 inch diameter ducting. The pneumatic manifold, which is itself ducting, distributes the air through the use of control valves leading to various pneumatic systems components and sub-systems. The ducts into which the control valves direct the air are of various sizes. High volume ducting (3-4 inch diameter) is used for engine starting and wing anti-ice and air conditioning. Smaller diameter ducting is used for many other components such as windscreen anti fogging and total air temp gauge anti-ice.

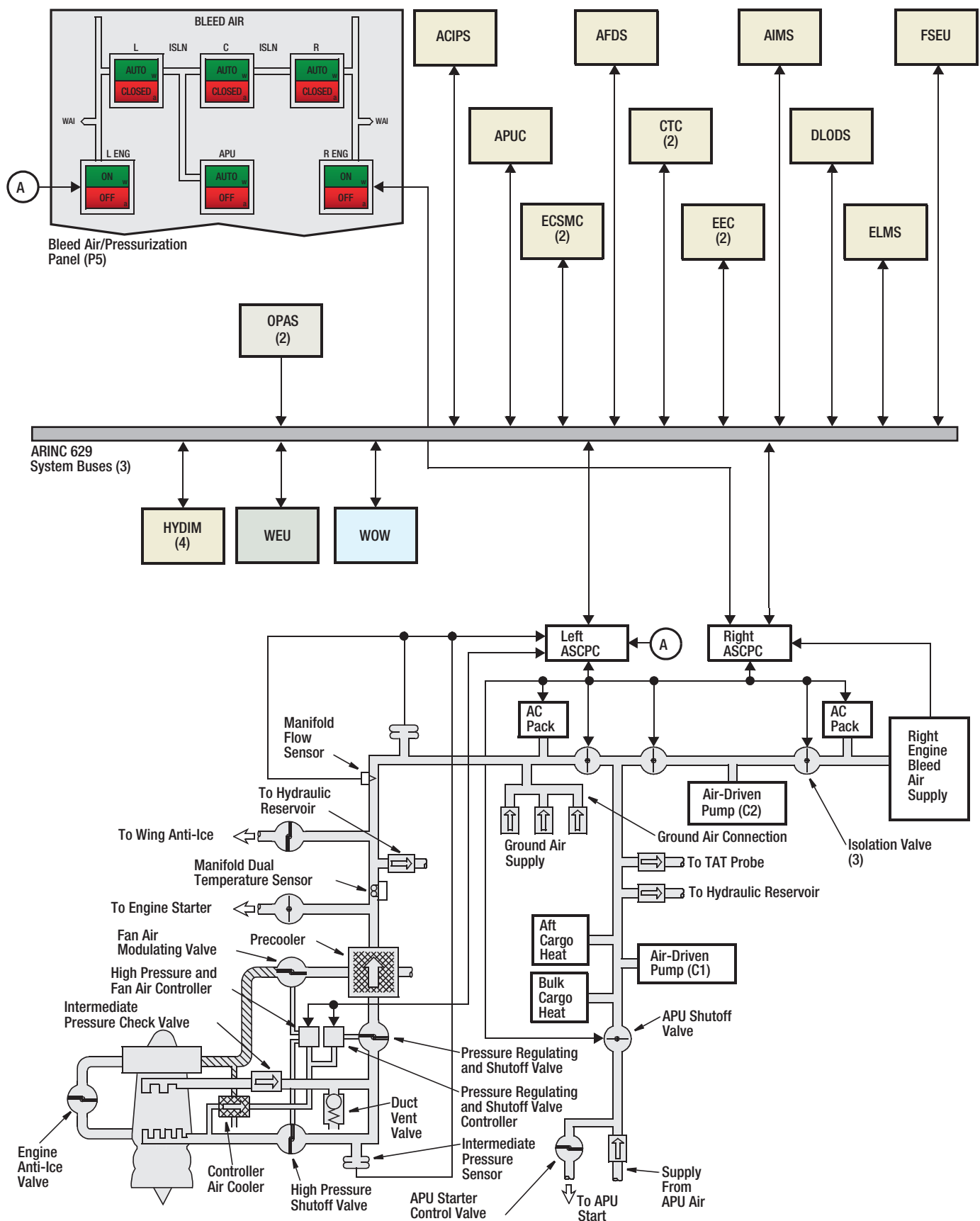


Figure 16-6. Pneumatic system components and integration on a Boeing 777.

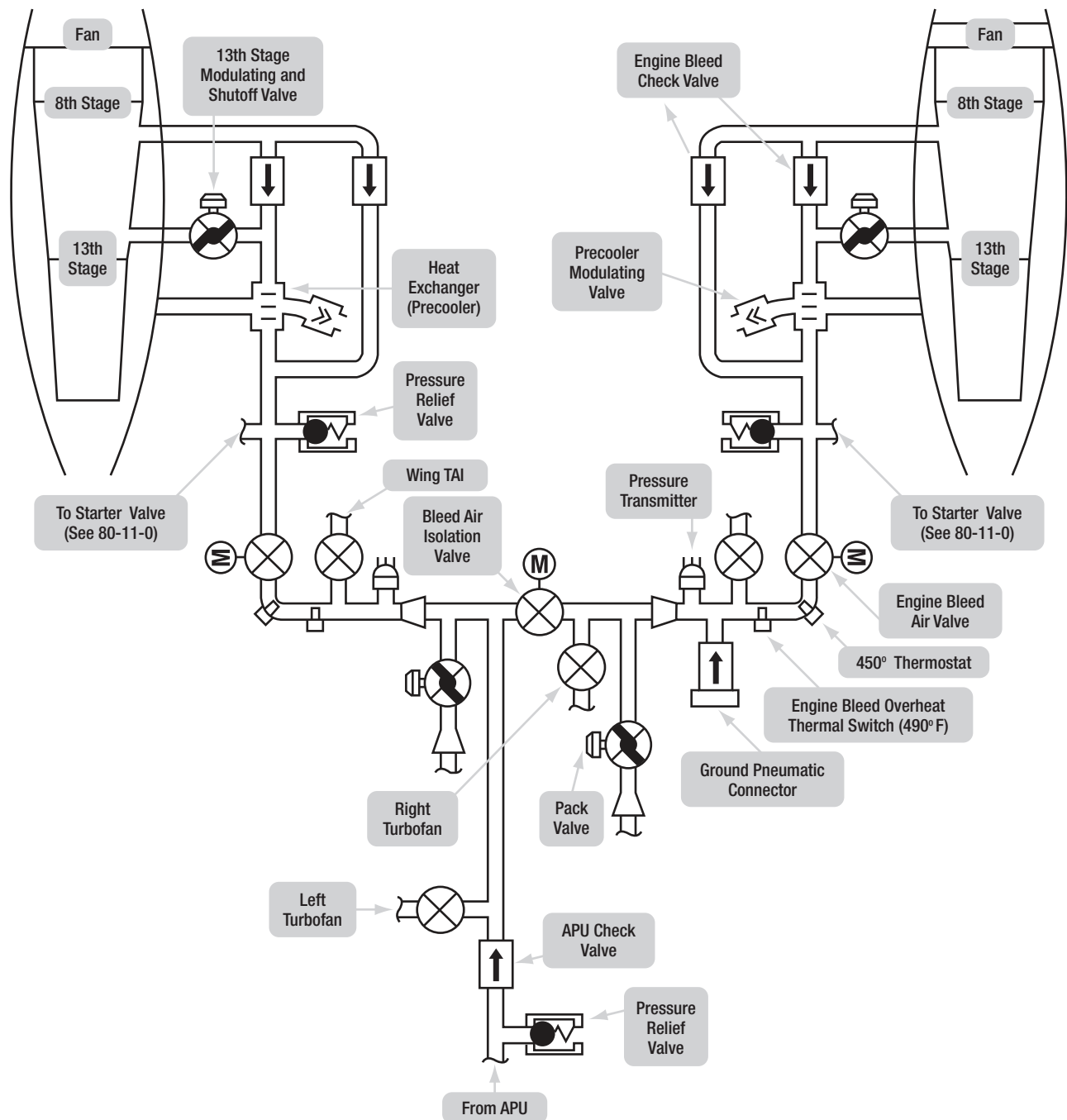


Figure 16-7. Engine bleed air distribution system schematic.

The temperature of pneumatic air is controlled within an acceptable range. Typically, some sort of heat exchanger is located in the bleed air portion of the pneumatic system for this purpose. A working temperature in the pneumatic manifold of close to 200°C is normal. By controlling the quantity of the overall volume of air that passes through the heat exchanger, the pneumatic manifold temperature is regulated.

Air conditioning systems accepted air from the pneumatic manifold that is too hot to be released directly into the cabin. Air conditioning packages use heat exchangers and well as an air cycle machine to adjust the temperature of the pneumatic air so that it is comfortable in the cabin. Wing anti-ice and engine starter air temperatures are not as critical and make use of pneumatic manifold air without further temperature adjustment.

There is a limit to the amount of ozone present in the pneumatic air processed by the air conditioning system that is sent to the cabin. Above the limit, passengers experience symptoms such as headache, respiratory problems, and even cancer with long term exposure. An ozone converter in the distribution line to the cabin reduces the ozone concentration in the pneumatic air. The ozone converter uses special, active, oxide-coated surfaces to remove the ozone. Some converters are constructed to remove hydrocarbons and carbon monoxide contaminants/odors as well. The process involves a chemical reaction between the coated metal plates and the hot air. **Figure 16-8** illustrates a typical ozone converter in a pneumatic duct.

Note that the air cleaners shown in **Figure 16-8** do not affect ozone levels. Instead this type of cleaner in a pneumatic system is used to remove airborne particles. A swirling motion is induced such that the heavier particulates are separated. They are moved to the outside of the cleaner housing by centrifugal force. An electrically controlled pneumatically operated purge valve removes the particles.

PRESSURE AND VACUUM PUMPS

OVERVIEW

Pumps used in hydraulic power and those used in pneumatic power, have a lot in common; as do pumps used in engine oil and aircraft fuel systems. The biggest difference with pneumatic pumps is the fluid they handle is a gas and not a liquid.

The purpose of the pump in each of these systems is the same; to increase the pressure of the fluid and to use that pressure to accomplish a task such as move a flight control, raise the landing gear, pressurize the airplane, start a turbine engine, or countless other possibilities. When studying a large commercial airplane, one discovers that a greater variety of things are accomplished using pneumatic power than with hydraulic power.

All pumps, whether mechanical in nature (gear-type, gerotor-type, vane-type, piston-type, diaphragm-type, centrifugal-type, roots blower type) or those based on Bernoulli's principle (ejector-type), operate under the same basic concept. By forcing a fluid out at the discharge side of the pump, a partial vacuum is created

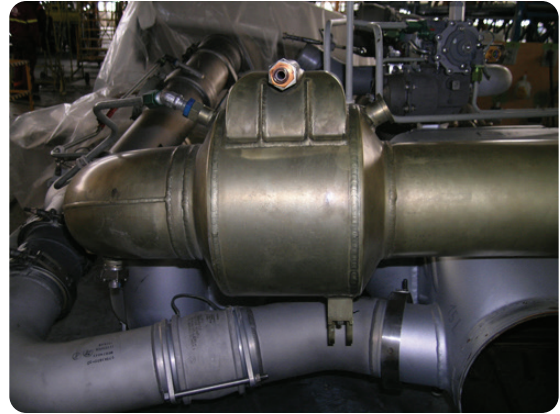


Figure 16-8. Catalytic ozone converter.

at the inlet side of the pump. The supply of fluid the pump is designed to handle is at normal atmospheric pressure, or possibly higher, and this higher pressure forces the fluid into the pump's inlet. This concept can be viewed when someone sucks on a straw inserted into a glass of water and creates a partial vacuum in the straw. The liquid in the glass is at normal atmospheric pressure, which forces the liquid into the straw and we are able to drink.

Pumps are generally viewed as devices that add pressure to a fluid to accomplish some type of work. Most pressure pumps can also be viewed as vacuum pumps. The discharge side of the pump is at a pressure higher than ambient, but the inlet side of the pump is at a pressure lower than ambient (partial vacuum). So pressure pumps and vacuum pumps are not different types, the classification is simply based on what the pump is being used for. In a small airplane, for example, the inlet side of a vane type pump can create the vacuum that powers the gyros in attitude instruments and at the same time, the discharge side can send pressurized air to the cabin. In this case, the device is acting as both a vacuum and a pressure pump.

CLASSIFICATION OF PUMPS

The pumps used in pneumatic systems can be classified as positive or non-positive displacement. Positive displacement means that in one complete revolution of the pump, a fixed amount of fluid will be discharged. If one revolution produces 2 ounces of fluid, then five revolutions displaces 10 ounces. A positive displacement pump is often called a constant-displacement, constant-volume or constant-delivery type of pump. As long as there is no slippage of fluid within the pump, the output per revolution will remain the same. A non-positive

displacement pump, such as a centrifugal impeller type, experiences a lot of fluid slippage as the pressure increases and it cannot maintain a consistent output per revolution.

TYPES OF PUMPS

When it comes to the pneumatic system of the aircraft, the types of pumps typically used are the wet-type or dry-type vacuum pump, the multiple piston pump, the centrifugal impeller pump and the roots blower pump. A very common source of pneumatic power is the compressor of an aircraft's turbine engine, which in addition to pumping air into the engine's combustion chamber can also pump air into the aircraft.

Vane Pump, Wet-Type and Dry-Type

The vane-type power pump is a constant-displacement pump. It consists of a housing containing four vanes (blades), a hollow steel rotor with slots for the vanes, and a coupling to turn the rotor. (**Figure 16-9**) The only difference between the wet or dry-type pump is the method used to lubricate the rotating vanes.

The rotor is positioned off center within the sleeve. The vanes, which are mounted in the slots in the rotor, together with the rotor, divide the bore of the sleeve into four sections. As the rotor turns, each section passes one point where its volume is at a minimum and another point where its volume is at a maximum. The volume increases from minimum to maximum during the first half of a revolution and decreases from maximum to minimum during the second half. As the volume of a

given section increases, that section is connected to the pump inlet through a slot in the sleeve. Since a partial vacuum is produced by the increase in volume of the section, fluid is drawn into the section through the pump inlet and the slot in the sleeve. As the rotor turns through the second half of the revolution and the volume of the given section is decreasing, fluid is displaced out of the section through the slot in the sleeve aligned with the outlet port, and out of the pump. This type of pump moves four pockets of fluid in one revolution.

Piston Pump

For many turbine engine powered aircraft, a high pressure bottle filled with compressed gas is carried on board. In an emergency, the high pressure gas can be used to apply the brakes or force the landing gear to extend. The source to fill these high pressure bottles comes from a multi-stage piston pump, which is typically land based so the bottle cannot be refilled in flight.

For the Fokker F-27 airplane, many of the aircraft systems are operated with high pressure pneumatics, such as the landing gear and brakes. The turboprop engines on this airplane drive four stage piston pneumatic pumps, so there is a continuous supply of high pressure air. (**Figure 16-10**) A four stage pneumatic pump has four separate cylinders and pistons, with stage one being the largest in diameter, and each successive stage becoming smaller. Stage one compresses the air and then supplies it to a smaller diameter stage two, which supplies it to stage three and eventually to stage four. By the time it leaves stage four, the pressure of the air is higher than 3 000 psi.

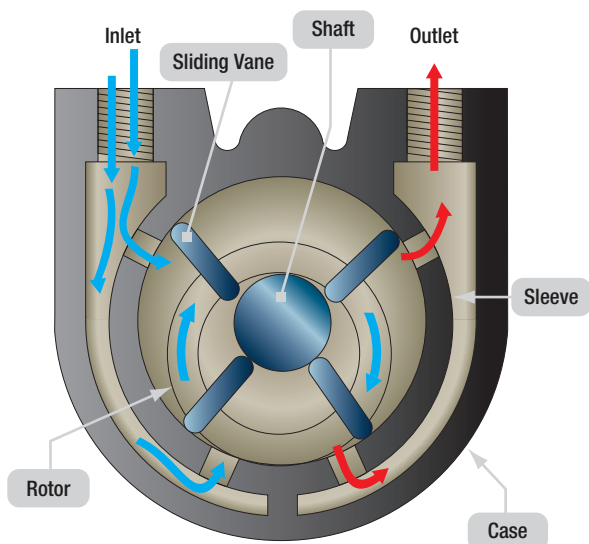


Figure 16-9. Vane-type pneumatic pump.

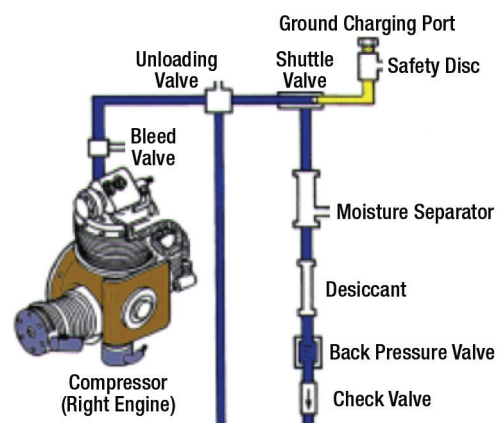


Figure 16-10. Four-stage piston pneumatic pump for Fokker F27.

Centrifugal Impeller Pump

Pressurized airplanes that have turbocharged reciprocating engines often use the centrifugal compressor (impeller) in the turbocharger to both pressurize the airplane and to supercharge the engine. The compressor takes in outside air and directs it into the center of the rotating impeller. (*Figure 16-11*) The impeller throws the air to the outside (centripetal acceleration), increasing both the velocity and the pressure of the air. The fins on the impeller form diverging passages when flowing from the center to the outside, and this diverging shape is why the pressure increases in addition to the velocity increasing. The increased velocity is converted to additional pressure as the air leaves the impeller and flows through another diverging passageway. This high-pressure air is used to increase the engine's power and to pressurize the airplane.

Turbine Engine Compressor

Most commercial airliners rely on the compressor found in the turbine engines as a source of pneumatic power. For large airplanes the compressor would be an axial flow type, utilizing airfoil shaped blades and vanes to pump the air from front to back while steadily increasing its pressure. (*Figure 16-12*) Some of this air can be bled away and sent to the airframe to be used for a variety

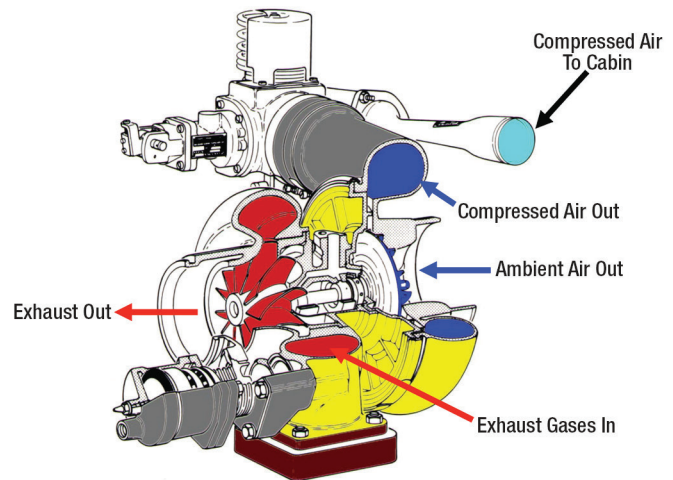


Figure 16-11. Turbocharger for pressurization (pneumatic system pump).

of purposes, to include cabin pressurization, wing anti-icing, pressurizing hydraulic reservoirs and pressurizing water tanks. The turbine engines in smaller aircraft might use a centrifugal impeller type of compressor, which was discussed earlier.

On the Boeing 787, no bleed air is taken from the engines. Instead, a variable speed electric motor is used to drive a stand-alone centrifugal compressor, which serves to pressurize the airplane. By not using bleed air from the engine, the weight and complexity of ducting the air from the engines to the airplane is eliminated.

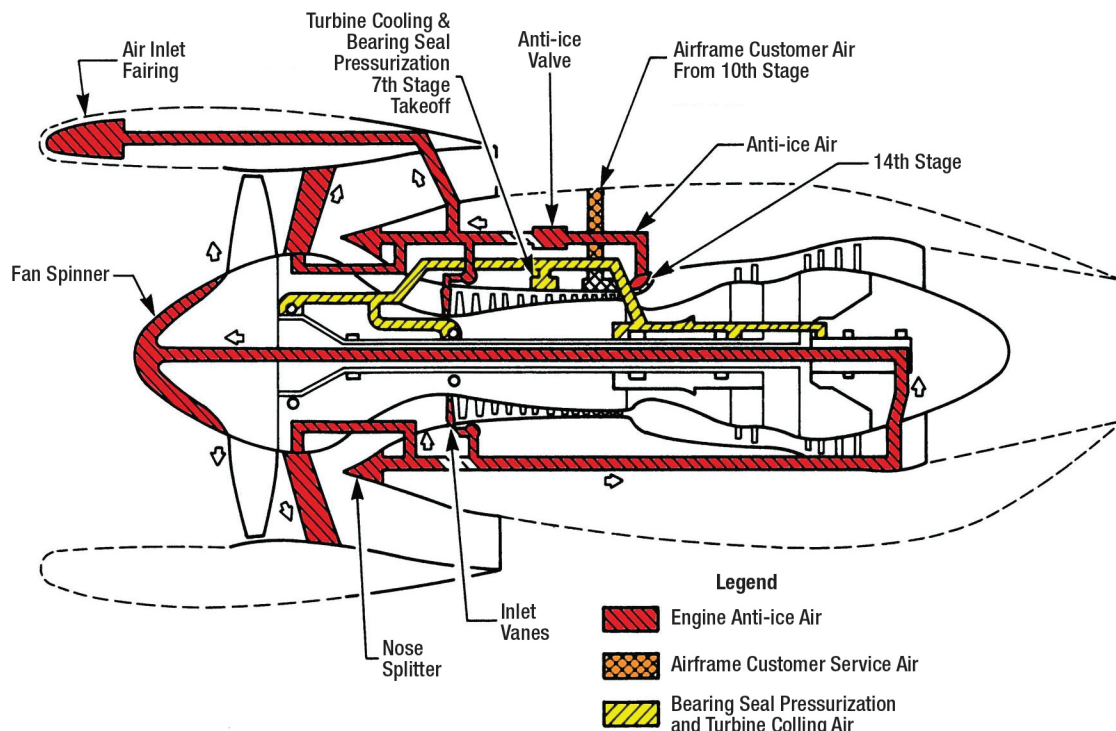


Figure 16-12. Turbine engine compressor (pneumatic system pump).

Turbine powered airplanes, from regional jets to large mainline carriers, all have an auxiliary power unit (APU) installed in the airplane. The APU is a small turboshaft engine utilizing a centrifugal compressor. One purpose of the APU is to provide bleed air for the aircraft's pneumatic system. As identified earlier, the centrifugal compressor uses a rotating impeller that brings the air in to the center and accelerates it to the outside. (*Figure 16-13*) Acting as a pneumatic pump, the pressurized air coming from the APU can be used to power the air conditioning system and also provide air for the engine's air turbine starters.

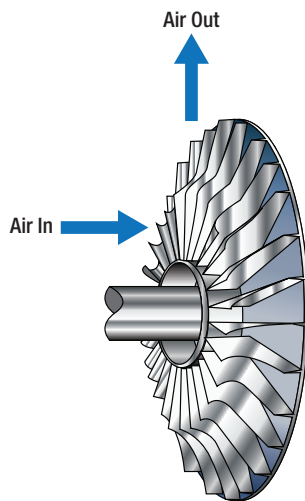


Figure 16-13. Centrifugal compressor impeller.

Roots Type Blower

Some small turbine powered airplanes, have used a roots type blower as a source of pneumatic power. This device is made up of two lobed rotors that rotate very close to each other, without touching, with air entering the space between the lobes and being compressed. (*Figure 16-14*) The early Beech King Air twin turboprop airplane used a roots type blower for pressurization.

Vacuum Pumps

Almost all pressure pumps are capable of functioning as a vacuum pump, because the inlet side of a pressure pump is under a partial vacuum when it is operating. The vane pump, (*Figure 16-9*), is the most common type. The suction side of this pump is typically used to pull air through the gyros in attitude indicating instruments. The output of the pump, which is at an increased pressure, can either be dumped overboard or used to perform tasks like inflating wing deicer boots or pressurizing a small airplane.

FLIGHT DECK INDICATIONS

On the flight deck of an aircraft with a pneumatic power system, the following indications might be seen.

1. A gauge showing the pressure in the pneumatic manifold.
2. A gauge showing the temperature of the air in the pneumatic manifold.

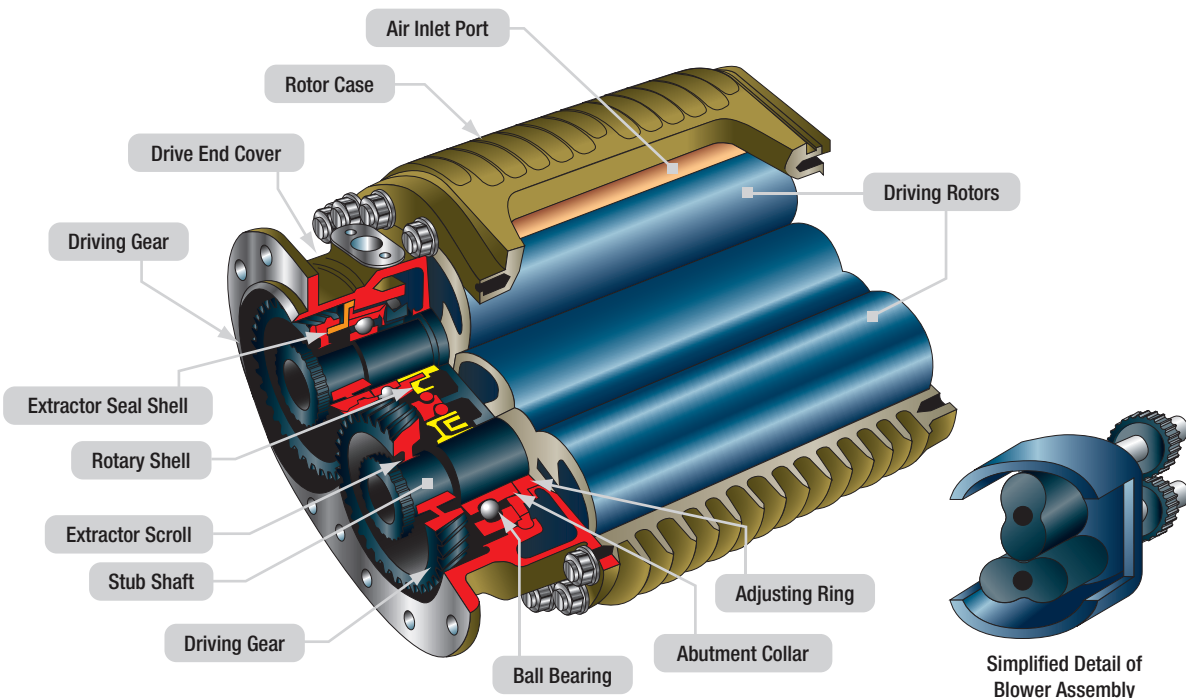


Figure 16-14. Roots type blower (pneumatic system pump).

3. An advisory light to indicate if the pressure or temperature in the pneumatic manifold is not within limits.
4. If the pneumatic system includes a filter, there might be an advisory light to indicate if the filter becomes clogged and starts bypassing.

INDICATIONS AND WARNINGS

There are few indications and warnings associated with the pneumatic system. Pneumatic manifold pressure is a key parameter monitored on the flight deck. Twin engine airliners typically have a pressure transmitter mounted in each section of the pneumatic manifold associated with an engine. These transducers send an electric signal to a dual gauge on the pneumatic control panel. Isolation valve control switches are located nearby. A low or no pressure situation can be handled by closing an isolation valve and using the remaining pressure to supply all pneumatic requirements. One transmitter is used for each engine. Both pressure transmitters are connected to a dual pressure indicator on the overhead panel.

Engine bleed pneumatic system temperature is also monitored. On a Boeing 737, indication consists of over temperature switches located in the ducting system. The switches are wired to trip lights on the flight deck overhead panel. The temperature switches illuminate the trip lights and the corresponding engine bleed valve closes automatically when the bleed air temperature exceeds approximately 490°F.

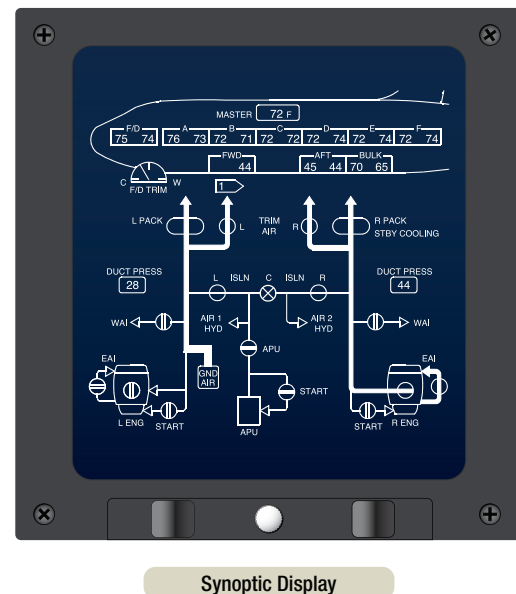
On digital aircraft, pneumatic system operation is completely automatic. Redundant computer controlled systems factor all supply and demand parameters including pneumatic pressure and temperature as in a non-digital aircraft. Valve positions and flow data are also factored. Over pressure and over temperature bleed air and duct leak conditions cause protective shutdown of the affected part of the system. BITE equipped components provide self monitoring information to the central maintenance computer system. Synoptic display of pneumatic air user system parameters and those of the pneumatic system itself is given via a multi-functional display panel. (*Figure 16-15*)

For the pneumatic system, the air synoptic display shows the following information:

- Ground air in use.
- Duct pressures.

- Engine bleed air pressure regulating and shutoff valve position.
- Isolation valve position.
- APU shutoff valve position.

An × on an isolation valve symbol or the APU shutoff valve symbol shows the valve has failed or the switch on the bleed air/pressurization panel for the valve is in the non-normal position. An air supply maintenance page viewable on the multi-functional display is shown in *Figure 16-16*.



Synoptic Display

Figure 16-15. Synoptic display of a modern airliner pneumatic system.



Pneumatic Air Supply Display

Figure 16-16. Pneumatic air supply maintenance page on a digital airliner.

INTERFACE WITH OTHER SYSTEMS

Pneumatic systems interface with other types of systems on many aircraft. The most common interface is with portions of the aircraft hydraulic system. As mentioned in the discussion on high pressure pneumatic systems, interface of emergency high pressure pneumatic system air with a normally hydraulic braking system is common.

Use of a shuttle valve prioritizes the flow of air and directs it into the brake actuating mechanisms. The seals installed for hydraulic use are sufficient for use in the one type deployment of emergency brakes by pressurized air. Hydraulic system actuators are designed primarily for use with hydraulic fluid. Use with air is limited to one time emergency operations.

However, pneumatic power may be used to supplement and backup hydraulic system components without loss of performance. This is done by turning a hydraulic pump with pneumatic power. The pneumatically driven hydraulic pump then supplies the hydraulic system components with fluid in the usual engineered manner. Cross utilizing hydraulic components with pneumatic air is eliminated. Traditional benefits of hydraulic power are retained such as those from the incompressibility of the fluid.

Control of a pneumatically driven hydraulic pump is through the use of a control valve in the pneumatic manifold. Selection of the pump via a switch on the flight deck causes the control valve to open and supply pneumatic air to drive the pump. All hydraulic system controls are then operated normally either by the flight crew or automatically by computer. The Boeing 777 is an aircraft that incorporates pneumatically driven hydraulic pumps.

Question: 16-1

Twin-engine aircraft vacuum systems contain an _____ on each engine.

Question: 16-5

_____ -stage compressor bleed air normally supplies the bulk of the pneumatic system demand.

Question: 16-2

The pressure developed in a high pressure pneumatic system is approximately _____ psi.

Question: 16-6

A _____ is located in each section of the pneumatic manifold separated by an isolation or shutoff valve.

Question: 16-3

When high pressure pneumatic air or nitrogen is used to move hydraulic fluid to an actuator, it is called _____.

Question: 16-7

Pneumatic manifold _____ is a key parameter monitored on the flight deck.

Question: 16-4

The _____ contains the control valves that are operated to supply the systems that require pneumatic power.

Question: 16-8

Pneumatic power may be used to back-up hydraulic system components without loss of performance by turning a _____ pump.

ANSWERS

Answer: 16-1
engine-driven vacuum pump.

Answer: 16-5
Intermediate.

Answer: 16-2
3 000.

Answer: 16-6
relief valve.

Answer: 16-3
pneudraulics.

Answer: 16-7
pressure.

Answer: 16-4
pneumatic manifold.

Answer: 16-8
hydraulic.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 17

WATER/WASTE (ATA 38)

Knowledge Requirements

11.17 - Water/Waste (ATA 38)

Water system lay-out, supply, distribution, servicing and draining;
Toilet system lay-out, flushing and servicing;
Corrosion aspects.

3

WATER/WASTE
(ATA 38)

11.17 - WATER/WASTE SYSTEMS

Large passenger aircraft are fitted with food preparation galleys and lavatories for passenger comfort. To support these installations, potable water, toilet and waste water/drainage systems are also installed.

LAYOUT

The water and waste system is composed of related systems as mentioned. The potable water system stores, delivers, monitors and controls potable (drinkable) water for galley and lavatory components. The toilet system provides sanitary toilets in the lavatory compartments and a means to dispose of toilet waste. The waste water system disposes of all waste water from the lavatory compartments. A water tank pressurization system is used to pressurize the potable water system. **Figure 17-1** illustrates these systems.

SUPPLY

The potable water system is the water supply system for the aircraft. Potable water is stored below the cabin floor in a single or multiple tanks. Typically, the aircraft pneumatic system pressurizes the tank(s). Pressure inside the tanks pushes the water through distribution lines to the lavatories and galleys.

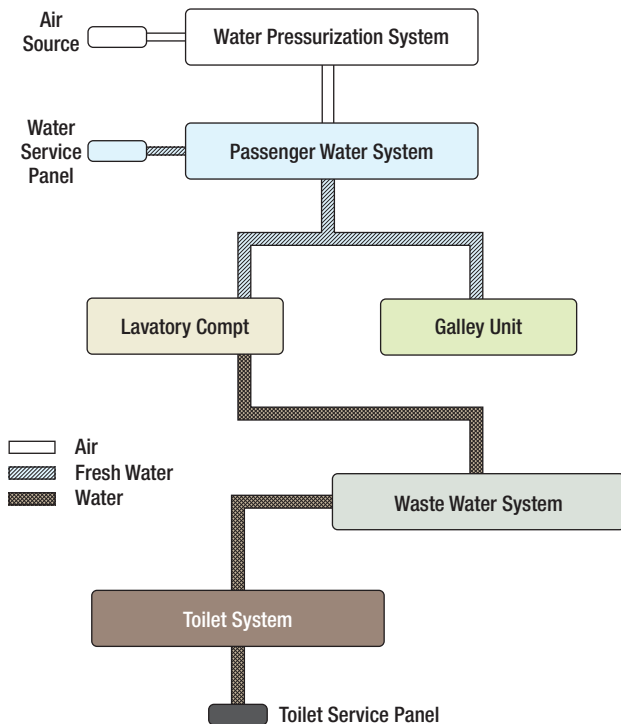


Figure 17-1. Related water/waste systems.

Potable water also goes to the toilets in lavatories equipped with vacuum flush toilet bowls. A fitting for pressurizing the tank on the ground with an external air source is available on the ground service panel. This is also where the tank can be filled and flushed. **Figure 17-2** illustrates a typical water servicing panel located behind a hinged door on the exterior of the aircraft. The water servicing panels are usually heated to prevent icing. Note that on some aircraft, a separate electrically powered compressor is used to pressurize the potable water tanks.

The water tank pressurization system passes air through an air filter and pressure regulator on way to the water tank. The air filter prevents contamination. A replaceable 10-micron filtration cellulose element is common. The pressure regulator reduces the air pressure and maintains approximately 20-50 psi in the water tank depending on the system. Check valves in the system prevent reverse pressurization. A system pressure relief valve prevents damage to the tank from over pressurization.

Figure 17-3 illustrates a typical potable water system. Most potable water supply systems are fitted with a pressure transmitter for tank pressure indication. Tank water quantity indication is also common.

DISTRIBUTION

Plumbing lines connect the water supply tank(s) to the lavatory and galley sink faucets. They are typically reinforced plastic hoses sometimes enclosed in protective conduit. Inline electric element heaters are installed just upstream of all hot water faucets. The units have a small reservoir and high wattage electric heating elements with

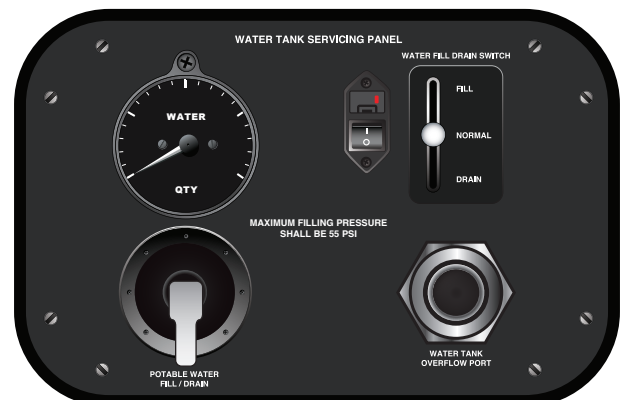


Figure 17-2. A typical water servicing panel on an airliner.

overheat protection. Distribution lines are typically run behind the cabin ceiling panels. Shutoff valves are used to isolate sections of the water system for maintenance.

Since galleys and lavatories are modular units, often the technician must connect or disconnect the water distribution lines to the modular units. Plumbing inside the units includes lines, valves, heaters, etc. as well as components of the water drain and waste systems.

Potable water distribution plumbing spans a great portion of the fuselage. Water lines pass near or through cold sections of the fuselage. In these areas, the plumbing must be insulated or heated. Heat tapes, inline heaters and heated hoses are all used to prevent ice in the water system.

SERVICING

It is necessary to completely drain the water system before disinfecting it or when parking the airplane in freezing weather. The water should be drained at least every 3 days and fresh water added to the system to prevent the growth of bacteria. Use of a disinfectant regularly in the potable water system is standard.

Disinfection is also accomplished after completion of certain maintenance procedures or if the water system is contaminated.

The water servicing panel contains a fitting for the attachment of a fill hose. Once the hose is attached, a fill valve is positioned to allow the water to flow into the tank. The quantity indicator on the panel is used to fill the tank. An overflow drain line is plumbed from the top of the tank to the aircraft exterior. When the water reaches the level of the overflow fitting it spills into the overflow line and is drained overboard. Often, tanks are filled until water comes out of the overflow. Note that a maximum water fill pressure must be observed to avoid damage to the tank. The fill valve handle located on the service panel must be returned to the closed position for flight. Often the service panel door cannot be closed until the fill valve handle is in the proper position.

The fill/overflow valve allows filling of the passenger water system. On many aircraft, the valve is a four-port machined body with a rotary core. The rotary core is spring-loaded to ensure positive seating and prevent leakage. The valve is operated by a handle attached to

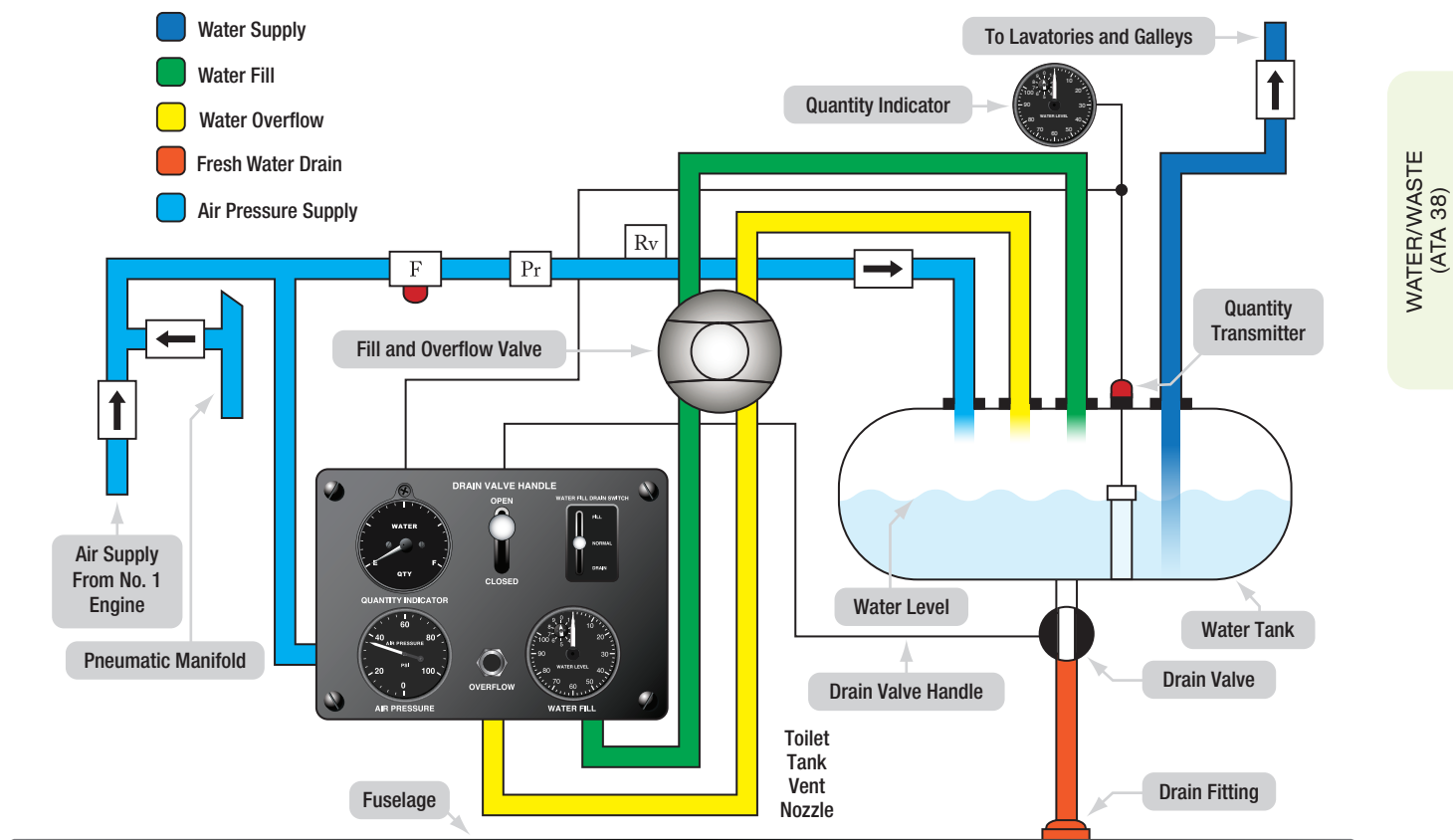


Figure 17-3. A typical potable water supply system.

the rotary core by a short stem. It is a 2-way valve with an OPEN and a CLOSED position. As stated, during flight the valve must be in the CLOSED position. The OPEN position permits filling of the water tank. There is typically only one water service panel installed, which allows filling, monitoring and draining of the system. The panel is located in the lower side of the fuselage, usually positioned in an area that does not interfere with other servicing and loading activities during turn-around checks.

DRAINING

A potable water tank drain valve is a two port valve designed to permit drainage of the potable water tank. It is typically located below the tank and is controlled from the water service panel.

Each lavatory is also provided with a drain valve which controls potable water drainage of lavatory compartment plumbing. This valve typically has 3 positions: OFF, DRAIN and ON. In the OFF position all lines are completely closed. In the DRAIN position all lines are completely open. In the ON (normal) position, the water inlet and outlet lines are open and the drain line is closed. The valve is accessible in the lower part of the lavatory sink cabinet below the water heater. Galley and lavatory units are usually built in such a way as to allow any spills and leaks to drain through a port in the floor of the unit. This water is typically drained overboard through a heated drain mast.

Waste water is either drained directly overboard or to a holding tank. The tank is usually the same tank used for collecting toilet waste. The waste water drainage system disposes of all waste water from the lavatory compartments sinks. The system consists of plumbing necessary to drain the waste water into the toilet tank and moisture condensation and seepage to the toilet drain tube. Sink stoppers are spring loaded to the CLOSED position to maintain the integrity of cabin pressurization. Check valves are commonly used to prevent odors from backing up into the lavatory. Cabin pressure is used to close the drain system until needed and to force waste water overboard through a drain mast(s) mounted on the lower outside of the fuselage.

(Figure 17-4)

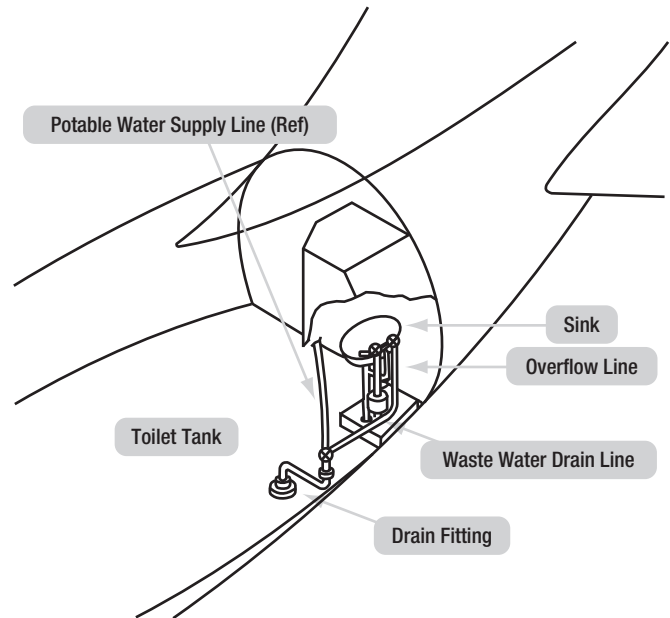


Figure 17-4. A typical waste water installation on an airliner.

On some aircraft, waste water is known as gray water. This does not include toilet waste which is collected in a tank and emptied from the ground lavatory service panel when the aircraft is on the ground. The gray water is used to assist toilet flushing or is drained overboard.

(Figure 17-5)

Waste water drain masts are heated either electrically or with warm pneumatic system air. As with the water distribution lines, waste water lines are also heated in cold areas to prevent freezing. Heaters in some the gray water drain masts give high heat in flight and low heat on the ground. Heated gaskets are used in some installations to protect the waste drains. Heater blankets are sometimes used to heat the waste tank drain lines.

TOILET SYSTEMS

Large passenger aircraft and freight aircraft are fitted with lavatories. These lavatories include a flushable toilet system. For many years, toilet flushing systems used an electric motor driven flushing mechanism to remove toilet waste into a holding tank positioned directly under the toilet. The most modern aircraft incorporate a vacuum flush system with remote waste holding tank(s).

As mentioned, modular lavatory installations are common. Most have all of the components immediately needed for flushing installed inside the unit including the waste holding tank. A waterproof floor pan or catch basin with drain makes up the floor of the lavatory unit

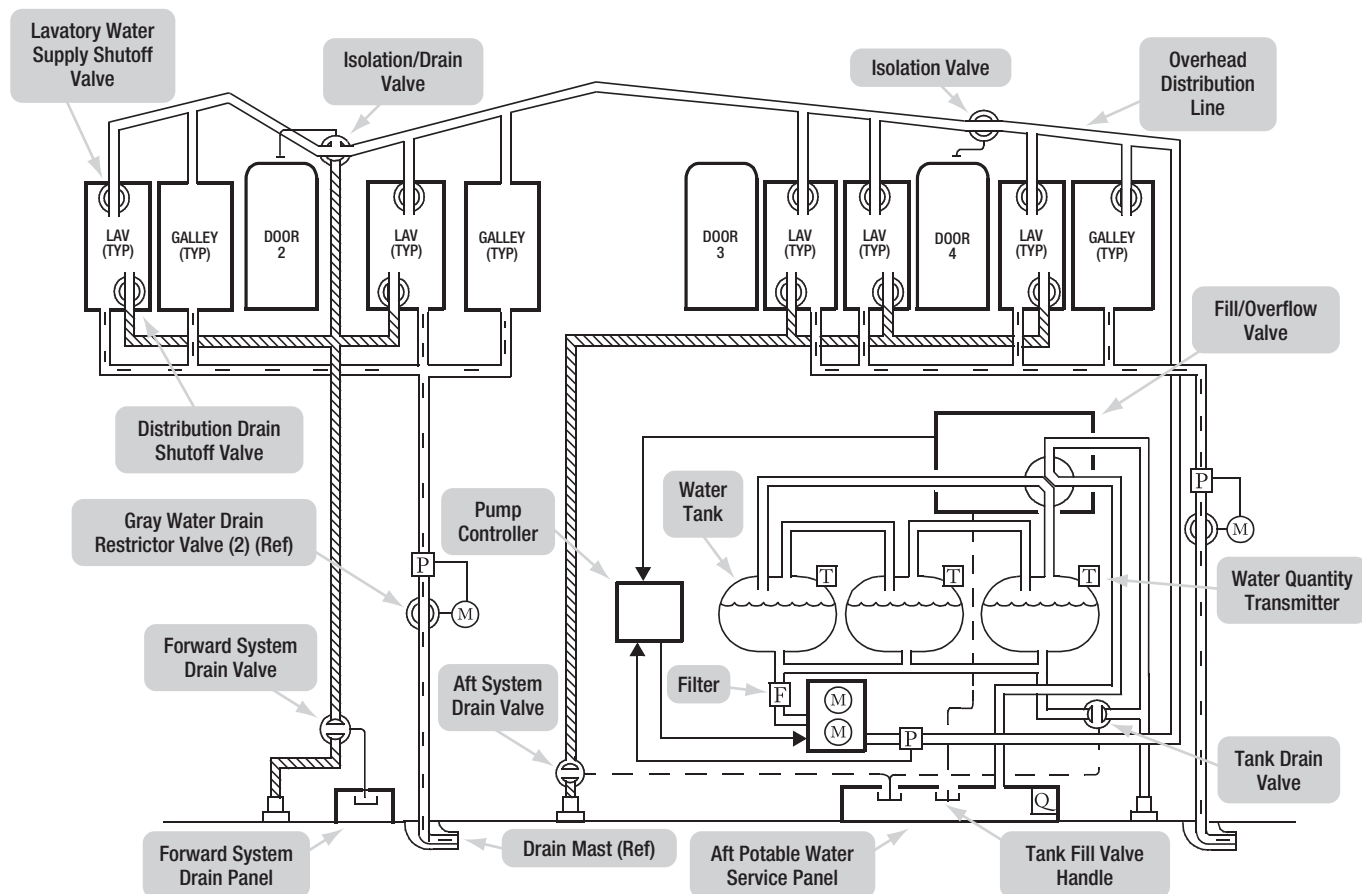


Figure 17-5. Gray water and potable water layout on a modern airliner.

which is installed on top of the passenger cabin floor. Tiedowns attach the unit to aircraft structure. Drain and vent lines extend through the floor to the outside of the aircraft. (*Figure 17-6*)

FLUSHING

All flushing system components are installed in the lavatory assembly. Typically, the toilet flush handle activates a timer which controls a toilet flush motor mounted on top of the toilet waste holding tank. When flushed, the motor turns a pump that sprays flushing agent from the tank through a filter and into the toilet bowl. Nozzles direct the spray in a swirling motion. After the set amount of flush time (usually 8-12 seconds) the waste and agent drain into the tank below the toilet. A separator between the tank and the bowl prevents splash and vision of tank contents.

Modern vacuum toilets are of different construction than the electric flush unit described and are not interchangeable. (*Figure 17-7*) The vacuum flush toilet is electronically controlled by a flush control unit mounted close to the flush valve at the outlet of the toilet. In

addition to opening and closing the valve, the flush control module opens a rinse valve that releases potable water to rinse the bowl as the waste is removed and is connected into the aircraft's central monitoring system and the cabin services system.

Vacuum waste systems remove waste from the toilet with suction and deposits it in a waste holding tank. Potable water is injected for approximately one second to rinses the toilet when flushed. The waste tanks are remotely located below the cabin floor. The suction for flushing is made by an electric vacuum blower below 16 000 feet and by the differential pressure between the cabin pressure and the outside air pressure above 16 000 feet. A barometric switch controls the blower. Vacuum check valves are used to prevent reverse airflow through the vacuum blower bypass lines.

The electronic flush control unit controls a normally closed flush valve to open when the flush handle is activated. The waste is sucked from the toilet when flushed and sent to the remote waste storage tank. The tank is an integral part of the vacuum waste system. It

contains a quantity sensing system and a separate sensor to indicate the tank is full. An electronic control module shuts the system down when the tank is full. It also provides indications that are sent to the flight attendants' cabin control panel as well as the EICAS maintenance page displayed on the flight deck. The tank contains a liquid separator at the inlet (top) of the tank. The liquid separator removes both moisture and waste particles from the waste tank air before venting the air overboard. The waste remains in the tank(s).

LAYOUT

There are a variety of lavatory system layouts owing to the number and configuration of lavatories on any particular aircraft. The basic systems remains as described.

Each lavatory is positioned near an external service panels to which the waste tank drain and flush lines as well as the waste tank vent line are routed. Waste is removed from the waste tanks through the waste service panel. A "T" handle in the service panel opens the waste tank drain ball valve which permits the waste tank to drain. The flush line is used to rinse the waste holding tank.

Figure 17-8 illustrates a vacuum flush waste system. A vacuum blower is associated with each waste tank regardless of the number of toilets that drain to the tank. Drain and flush lines run from the tank to the exterior servicing panel as usual. Also plumbed into the system are the potable water lines used to delivery a small quantity of water with each flush.

SERVICING

Servicing components in a toilet system allow ground draining and cleansing of the toilet units. The waste tank is drained by pulling the waste drain valve handle on the toilet servicing panel after attaching the ground service cart to the 4 inch drain outlet and removing the drain plug. The tank is cleaned by attaching water pressure to the ground flush connection. The water enters the tank through spray nozzles that rinse the insides of the filter and tanks. The latest toilet waste holding tanks have built in separators to permit particle free vacuum air to be expelled overboard as waste is drawn into the tank from the toilet. A waste tank rinse nozzle is fitted to the top of the tank to rinse it when servicing. (**Figure 17-9**)



Figure 17-6. Self contained modular lavatory.



Figure 17-7. A vacuum flush system toilet.

CORROSION ASPECTS

The presence of water in lavatories and galleys makes the structure around and below these areas highly susceptible to corrosion. Most structure is aluminum although some manufacturer's include titanium structure because it is much more resistant to corrosive effects. Waterproof flooring, built in catch basins and drains help reduce corrosion by controlling where water can flow. But operation with thousands of passengers over time typically results in the loss of water and corrosion of the entire area around a lav or galley installation. Anti-corrosion measures are increased around and under lavatories and galleys. Some common actions are as follows:

- Corrosion Preventative Fluids (CPFs) applied to the structure.
- Floor panel clips taped to prevent scratching.
- Floor panels cushioned to prevent damage.
- Floor panels sealed to prevent fluid getting into the structure.

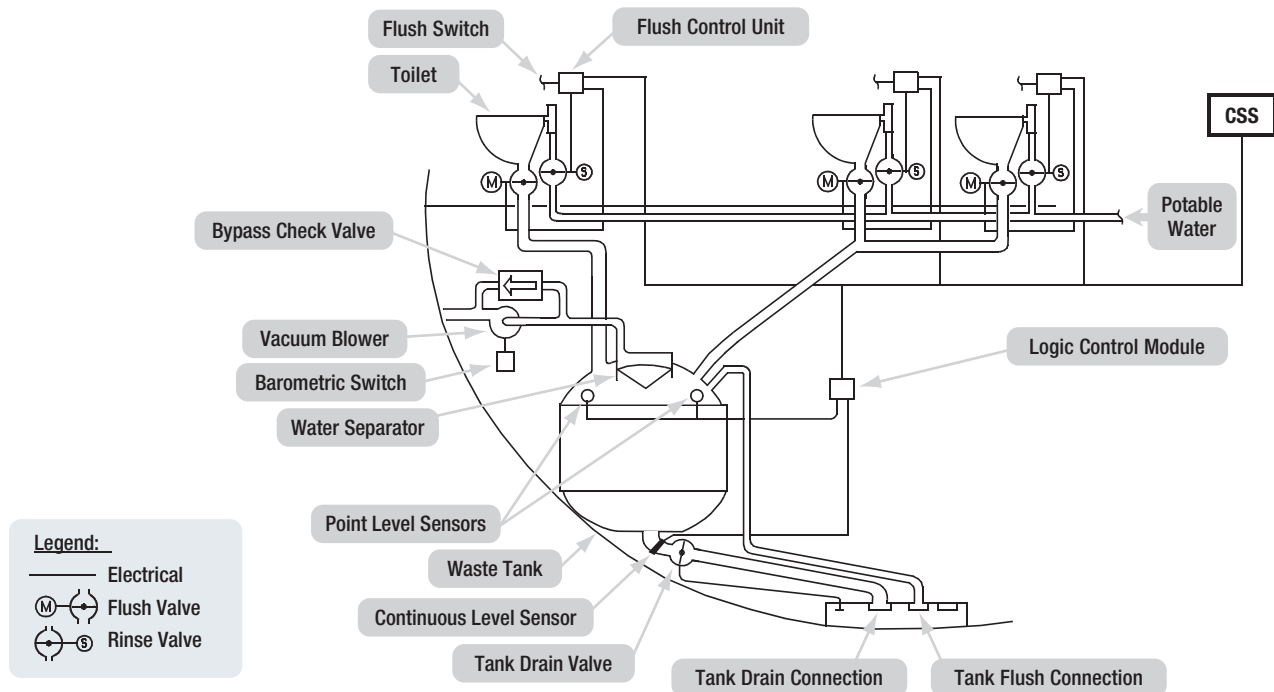


Figure 17-8. Toilet waste system on a vacuum flush aircraft.

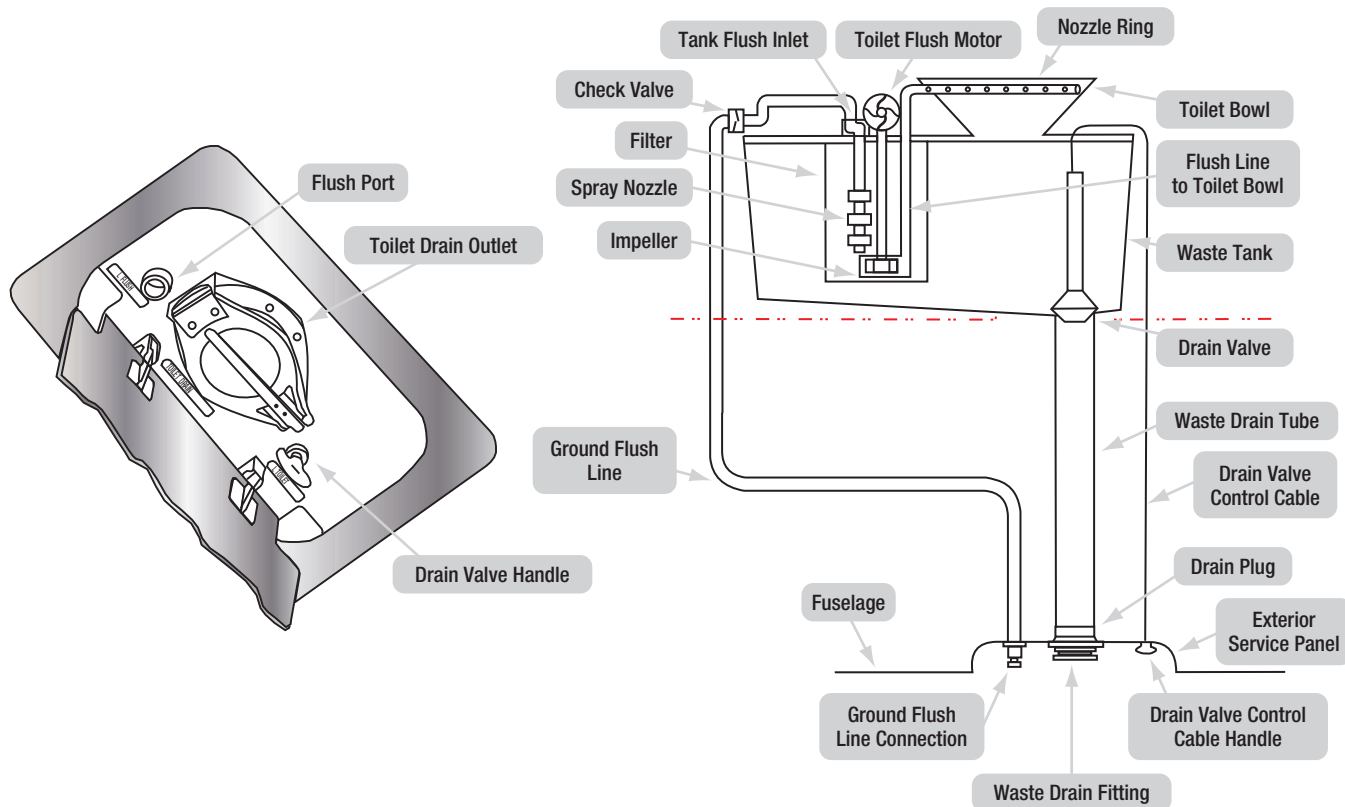


Figure 17-9. A typical toilet waste system on an airliner.

- Moisture barriers applied over the entire area.
- Leaks in the toilet system traced and cured at an early stage.

CPF is a water displacing fluid that adheres to the surface of clean (or primed) aluminum and its alloys. It is a slightly waxy substance that can either be sprayed by aerosol can or by a 5 gallon (221) dispenser onto the structure. Care needs to be taken during this operation, as the fluid spray will damage the lungs if breathed in. Use only in well ventilated areas and wear face masks.

A thickened version of the fluid is available for brushing. Consult the AMM prior to application. The biggest draw-back with CPF is that it acts as a debris collector. It will collect dust and debris as it is very sticky. During routine maintenance and certainly prior to any structural inspection this needs to be removed. Always record any work done (re-application, etc.) and clear all work with a CRS signature. In addition to CPF, sealing of the floor panels in the area of lavatories and galleys is common. This is done in a variety of ways including taping and two part sealants. Consult the airframe manufacturer's maintenance manual for the correct materials and procedures. In addition to sealing the floor panels, some operations utilize a Mylar moisture barrier over the floor before lavatories and galleys are installed.

A caution with this method is that protection may be provided by the impermeable Mylar, however water may migrate to the edge of the protected area and cause structural damage there. The most effective means of corrosion control is to prevent the liquid getting to the floor in the first place. Vacuum flush systems are a great improvement over the re-usable water toilet systems. The under floor structure on these aircraft is usually in much better condition.

On all toilet systems it is important that leaks are detected and rectified early. Often the first sign that there is trouble under the toilets is water coming from the lower bilge drains on the underside of the fuselage. Any liquid from these needs immediate investigation. Liquid coming from the bilge drain could be (a) clean water (b) liquid from some cargo on board, or (c) toilet water. The clean water could be rain water, condensation water or potable water. In all cases, the source must be found and the fault rectified.

Question: 17-1

_____ inside the potable water tank(s) pushes the water through distribution lines to the lavatories and galleys.

Question: 17-5

_____ are commonly used to prevent odors from backing up into the lavatory.

Question: 17-2

What is used to heated potable water on passenger aircraft?

Question: 17-6

Waste water drain masts are heated either _____ or with _____.

Question: 17-3

The fill valve handle located on the water service panel must be _____ for flight.

Question: 17-7

Vacuum waste systems remove waste from the toilet with _____ and deposits it in a waste holding tank.

Question: 17-4

Galley and lavatory units are usually built in such a way as to allow any spills and leaks to drain through _____.

Question: 17-8

Waterproof flooring, built in catch basins and drains help reduce _____ by controlling where water can flow.

ANSWERS

Answer: 17-1

Air pressure.

Answer: 17-5

Check valves.

Answer: 17-2

In-line electric element heaters.

Answer: 17-6

electrically.

warm pneumatic system air.

Answer: 17-3

closed.

Answer: 17-7

suction.

Answer: 17-4

a port in the floor of the unit.

Answer: 17-8

corrosion.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 18

ON BOARD MAINTENANCE SYSTEMS (ATA 45)

Knowledge Requirements

11.18 - On Board Maintenance Systems (ATA 45)

- Central maintenance computers;
- Data loading system;
- Electronic library system;
- Printing;
- Structure monitoring (damage tolerance monitoring).

2

11.18 - ON BOARD MAINTENANCE SYSTEMS

On board maintenance systems (also called central maintenance systems) are electronic systems used to facilitate the maintenance of modern aircraft. The exact configuration of on board maintenance systems varies from one aircraft model to another. However, the core functions of these systems are the same. They monitor the aircraft for faults, record and store the fault data, and provide information about these faults to flight crews and maintenance personnel.

The data collected by on board maintenance systems can be accessed both in flight and on the ground. In flight, the system advises the flight crew of faults that may affect aircraft operation. On the ground, maintenance crews use the system for testing and troubleshooting purposes. In some applications, the aircraft can radio fault information to the ground while in flight.

On board maintenance systems also electronically store data contained in maintenance manuals, flight manuals, and other publications. The systems allow maintenance personnel to access these publications without having to carry books and papers to the aircraft. On board maintenance systems allow such technical data to be uploaded, downloaded, viewed, and printed by maintenance personnel. The systems are used for both line and base maintenance. For the most part, the information provided here is general. When aircraft specific information is given, it should be noted that system details and terminology differ somewhat among the various aircraft manufacturers.

CENTRAL MAINTENANCE COMPUTERS

The central maintenance computer (CMC) is the main processing unit for an on board maintenance system. Like all computers, the CMC has inputs and outputs and it is programmable. Inputs to the CMC come from the various systems being monitored, which are located all over the aircraft. Outputs from the CMC are provided in the form of visual displays, printed text, and digital data that may be downloaded. **Figure 18-1** shows a block diagram of the Airbus A330 central maintenance system.

It is common for an aircraft's on board maintenance system to have two redundant CMCs installed. With dual CMCs, all data inputs are available to both units. One CMC will be "active" at any given time. The other CMC is on "standby." The active CMC is the one providing outputs. If the active CMC should fail, the standby can be immediately substituted for it. In some systems this changeover occurs automatically. The system monitors itself, and when it senses the failure of the active CMC, it automatically switches to the standby CMC. The switching can also be done manually.

CMCs are controlled by control units in the cockpit, such as the MCDUs (multipurpose control display units) found on the Airbus A330. **Figure 18-2** shows the locations of the MCDUs on the pedestal of the A330. The MCDUs enable the user to navigate the on board maintenance system by selecting from various on screen menus. These menus allow the user to view both current faults and fault history. Current faults are, of course, important for determining the aircraft's current status. Fault history can be useful for monitoring trends, such as recurring failures of a particular component. In addition, a CMC menu permits the user to check the current status of individual systems, even if no fault condition is present.

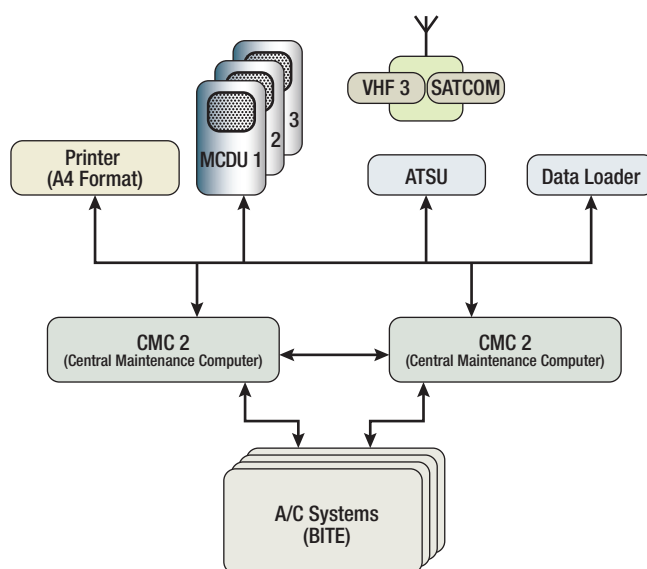


Figure 18-1. Airbus A330 central maintenance system.

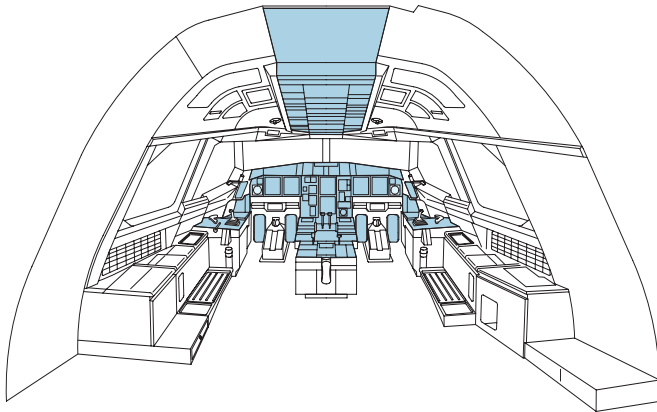


Figure 18-2. A330 Pedestal, showing the locations of the multipurpose control display units (MCDUs).

Figure 18-3 shows an example of navigating the central maintenance system menus on an A330's MCDU. In this example, there are three faults: A bleed air system fault, and anti-ice system fault, and an electrical system fault. The menus allow the user to access specific information about each fault that was sensed, such as the date and time the fault occurred.

In addition to the cockpit-mounted control units, some on board maintenance systems allow for the connection of a personal computer. When the personal computer is connected, it can be used to access the data stored in the CMC. Reports of current faults and fault history can be downloaded to the personal computer.

Some types of system failures will immediately affect the operational capability of the aircraft, while other failures have no immediate impact. Because of the redundancy designed into aircraft for safety purposes, some faults can be tolerated. The minimum equipment list (MEL) determines which faults must be corrected before further flight, and which faults may be deferred for correction later. The fault indications provided by the CMC should be compared with the MEL to determine whether the aircraft can be dispatched.

The CMC classifies faults according to their severity. More severe faults will trigger cockpit indications for the flight crew so that their effect on operations can be evaluated. Less severe faults will not be displayed to the flight crew; that fault information will simply be stored and dealt with by maintenance personnel after the flight.

As an example, the CMC in the Airbus A330 divides faults into three classes: Class 1, Class 2, and Class 3. Class 1 faults are the most serious, and involve something listed in the MEL. A Class 1 fault may ground the aircraft (a NO GO condition), or it may limit the conditions under which the aircraft may fly (a GO IF condition). An example of a GO IF Class 1 fault would be the failure of a pressurization system component that limits the aircraft to unpressurized operation only. Class 1 faults are indicated in the cockpit by warning or

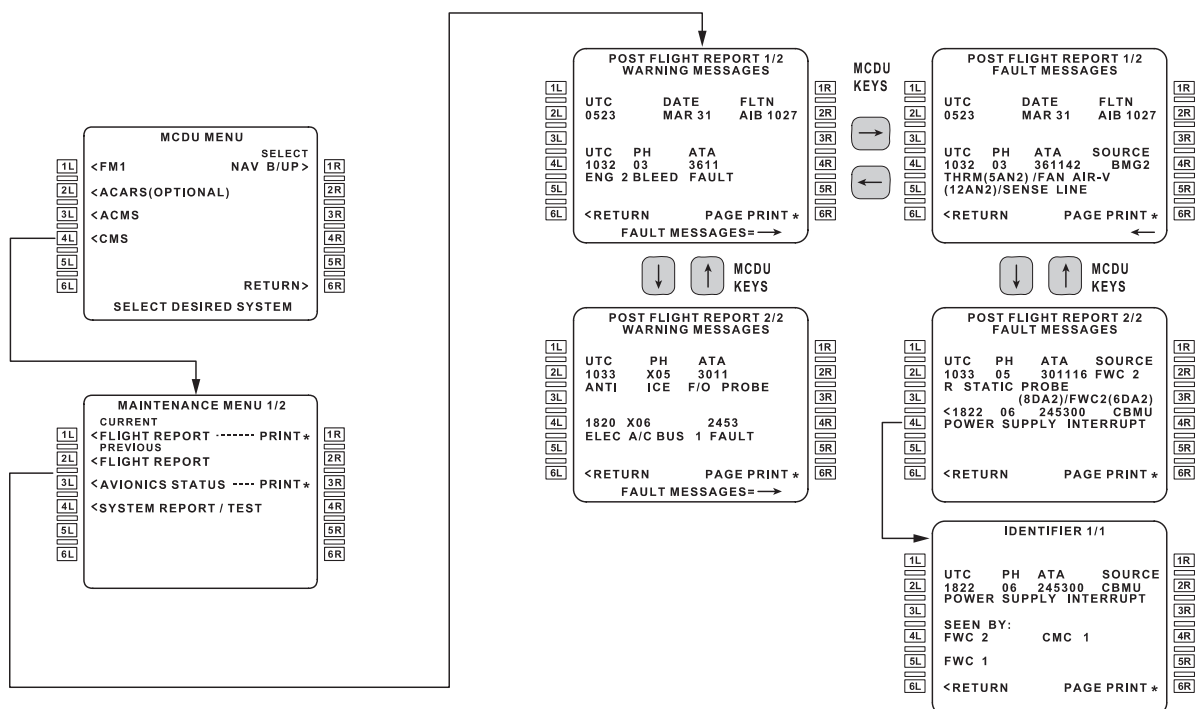


Figure 18-3. Maintenance Menus on the A330.

caution lights, by failure messages on indicator screens, or by flags on the flight instruments. These indications are referred to as "flight deck effects."

Class 2 faults are less serious. The aircraft can be dispatched with a Class 2 fault because operational capability is not compromised (a GO condition). When a Class 2 fault exists in a system, the system is still functioning normally, although it may not have full redundancy. The flight crew is provided with a notification that the fault has occurred, in the form of a "maintenance status" message. Repair of Class 2 faults may be deferred for a period of time in accordance with the operator's approved maintenance program.

Class 3 faults are minor discrepancies within a monitored system. They do not affect the operation of the aircraft. The flight crew is not notified of Class 3 faults. Maintenance personnel can access the CMCs record of Class 3 faults and repair them when convenient.

The information gathered by central maintenance computers can be relayed to the ground through the ACARS system. ACARS stands for aircraft communications reporting and addressing system.

ACARS is a data link system that uses the aircraft's VHF communications radio, and in some aircraft a SATCOM (satellite communications) radio. Worldwide, a network of ground stations is able to communicate digitally with aircraft using the system. **Figure 18-4** shows the ACARS system.

Although the on board maintenance system can be connected to it, ACARS is a stand-alone system. It can be used by the flight crew to send messages manually, and it automatically sends reports about occurrences not associated with the on board maintenance system. ACARS sends data automatically when the on board maintenance system detects a serious fault. This alerts maintenance personnel about the faults before the aircraft arrives, allowing them more time to prepare for dealing with the faults. Parts or LRUs (line replaceable units) can be pulled from stock, and can be waiting for the aircraft when it lands. This can reduce down time, helping to keep flights on schedule.

BITE

Units that are monitored by the CMC contain special circuits known as built in test equipment (BITE). BITE is installed in many systems throughout the



Figure 18-4. The ACARS system relays inflight maintenance information by both satellite and surface links.

aircraft including navigation systems, flight control systems, environmental control systems, and others. Within each system, the BITE circuitry tests numerous individual parameters to determine whether the system is functioning properly. The individual system BITE circuits are connected to the CMC by a digital data bus. ARINC 429 buses are used for this purpose in many aircraft. Other data buses, such as ARINC 629, may also be used.

Whenever a system that contains BITE is first powered on, the BITE automatically performs a test of that system. This is referred to as an initialization test or a power-up check. If any fault is detected by the BITE during this test, an output is generated and sent to the CMC. If the system passes the initial test, BITE begins its regular monitoring of the system parameters. This monitoring is sometimes referred to as a "watchdog" function. During operation, the monitoring process is continuous. If anything that is being monitored fails, BITE will alert the CMC automatically.

In some aircraft, the user can run the BITE power-up check for a given system from the CMC control unit at any time. This capability is provided as a CMC menu item. This function can be useful when troubleshooting the system. Some LRUs containing BITE have indicator lights that indicate the status of the LRU. Green lights indicate a normal condition, red lights indicate that the BITE detected a fault in the LRU. **Figure 18-5** shows an LRU with BITE indicators.

BITE systems also have the capability of storing fault history. The history is kept in non-volatile memory. Non-volatile memory holds the stored information even after the system has been powered off.



Figure 18-5. Built In Test Equipment (BITE) Indicators.

DATA LOADING SYSTEMS

An aircraft's data loading system provides a means to upload data to, and download data from, the on board maintenance system. The data loading system connects to other on board systems, as well. The data loading system can be used with any digital system that requires data uploads and downloads while installed in the aircraft.

Early data loading systems used floppy disks as the data storage medium. An example of this is the MDDU (multipurpose disk drive unit) used on many Airbus models. The MDDU uses 3.5 inch floppy disks for uploading, downloading, and data storage. In the Airbus system, a Data Loader Selector switches the MDDU to the various systems that require a data upload or download. On the Boeing 777, data loading is accomplished through a maintenance access terminal (MAT) on the flight deck. **Figure 18-6** shows the MAT.

Data loading systems also allow for the use of other forms of storage media. Newer systems can be connected to a laptop computer through a USB (universal serial bus) cable. A CD-ROM disk, or a USB memory stick or "flash drive" may also be used. In some aircraft, there are multiple locations to connect external devices to the data loading system. For example, the 777 has two laptop maintenance access terminal interfaces. One is located on the flight deck, and one is located in the main equipment center below the flight deck.

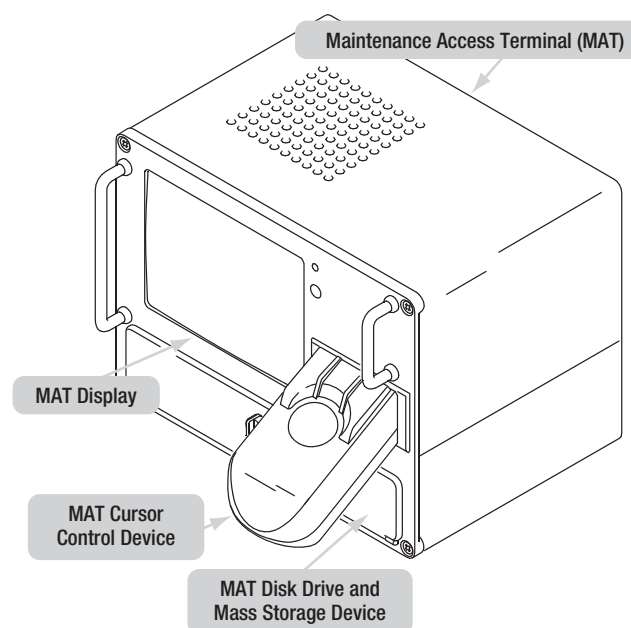


Figure 18-6. Boeing 777 Maintenance Access Terminal (MAT).

The primary uses for the data loading system are the uploading of program updates, the uploading of database updates, and the downloading of reports. An example of a unit requiring program updates is the central maintenance computer, which contains an operating program that is upgraded from time to time. The program upgrades to the CMC are input through the data loading system. The same is true for other aircraft systems with internal programming. The number of systems that require program updating varies from aircraft to aircraft.

An example of a database that requires updating is the navigation database which forms a part of the flight management system (FMS). The navigation database contains a great deal of information used by the flight crew. This includes the locations of airports, airways, way points, and intersections, the locations and frequencies of radio navigation aids, and other information needed to create and follow a flight plan. Because changes to this information occur from time to time, the navigation database requires periodic updates. These updates are uploaded through the data loading system. The standard frequency for navigation database updates is every 28 days. **Figure 18-7** shows examples of navigation database update software.

The data loading system can also be used to download reports from the aircraft. An example of this is the report of faults stored within the central maintenance computer. Reports on both current faults and fault history can be downloaded.

ELECTRONIC LIBRARY SYSTEM

An electronic library system (ELS) consists of databases containing information used by flight crews and maintenance personnel. These databases can include



Figure 18-7. Navigation system update software.

maintenance manuals, illustrated parts catalogs, wiring diagram manuals, flight manuals, service bulletins, and many other kinds of documentation from the manufacturer or the aircraft operator. The ELS takes the place of paper manuals. This results in a weight savings, and can make accessing the information in the manuals quicker and easier. The system has the capability of storing the equivalent of hundreds of pounds of paper manuals in its computer memory.

The databases in an electronic library system can be accessed through an on board display terminal and keyboard. They can also be accessed by an external personal computer, or through another digital device such as a tablet or smart phone. The laptop or other external device is typically connected to the system using a serial bus cable.

The databases in an ELS must be updated periodically as revisions are made to the technical data contained in the manuals. These revisions can be input through the data loading system.

PRINTING

Many aircraft have capability to print out paper copies of reports from the on board maintenance system, as well as other documents. Aircraft printers typically conform to ARINC Standard 744A, which gives technical requirements for such printers. These printers are able to print high-resolution alphanumeric text, as well as graphical images. The printers can print on paper up to 8.5 inches wide. **Figure 18-8** shows an example of an aircraft printer.

The speeds of aircraft printers vary, depending on the specific model of printer, and on what is being printed. Text generally prints faster, and images take longer. Some printers can print a page of text in as little as 5 seconds, while others are slower. Print resolution also varies. A standard resolution is 300 dots per inch (dpi), but some printers are capable of greater resolution. The paper supply for aircraft printers comes in the form of rolls. The paper rolls are typically 150 feet long, and may be perforated or non-perforated.

Inside the printer, an electric motor is used to advance the paper. The printer uses a thermal print head, and the paper is heat sensitive. For this reason, care must be taken to keep the paper away from heat sources and

out of direct sunlight while it is being stored. Exposure to heat can darken the paper, making it unusable for printing.

Aircraft printers receive input from CMCs, the ACARS system, and other sources by means of data lines, which may be ARINC 429 buses or Ethernet cables. Some printers are capable of receiving input wirelessly, and operated as part of a wireless LAN (local area network).

A typical aircraft printer is equipped with an indicator light to show whether the power is on or off. It will also give an alert when the paper supply is running low. Some printers perform a self-test on power-up, and will provide an indication if a fault is found during the test.

STRUCTURE MONITORING

Structure monitoring, also known as damage tolerance monitoring, has been recognized as an important function in aircraft maintenance. As aircraft age, their structures become more susceptible to damage caused by fatigue. Repeated cabin pressurization cycles cause fatigue. Repairs and alterations can change the structural characteristics of an aircraft, introducing different stresses than were present with the original design. Corrosion can seriously weaken an aircraft's structure. Also, events such as hard landings can lead to structural damage which may be difficult to detect.

Certification regulations require aircraft manufacturers to identify critical areas of the aircraft's structure. These areas are known as fatigue critical structures (FCS). These critical structures are identified by performing fatigue testing on test articles, which are subjected to repeated load cycles until they fail. The results of this testing are analyzed to determine the FCS for the aircraft.

Aircraft operators are required to monitor all FCS on their aircraft. This monitoring is intended to detect cracks and other structural deformations before they reach critical proportions, resulting in catastrophic failure. The FCS monitoring process is accomplished by performing damage tolerance inspections (DTIs). DTIs are inspections focused specifically on fatigue critical structures. The aircraft's DTI program will state where to inspect, how to inspect, and how often to repeat the required inspections.

DTI inspections may be accomplished using visual inspection, eddy current, penetrant, X-ray, or other methods. In addition to these, strain sensors may be used for structure monitoring. A strain sensor is a device that is bonded to a critical point on the structure. If the structure at that point becomes deformed, the strain sensor also becomes deformed. This deformation changes the electrical characteristics (typically the resistance) of the sensor. When the electrical characteristics of the sensor are measured, the changes indicate that the structure has deformed.

Effective structure monitoring is crucial for preventing accidents caused by structural failure. For this reason, all data gathered during damage tolerance inspections must be recorded and carefully evaluated to ensure that the aircraft remains structurally sound.



Figure 18-8. An aircraft printer.

Question: 18-1

The _____ is the main processing unit for an on board maintenance system.

Question: 18-5

An aircraft's _____ provides a means to upload data to, and download data from, the on board maintenance system.

Question: 18-2

On an Airbus 330, the CMCs are controlled by _____ on the flight deck.

Question: 18-6

Many aircraft have _____ to print out paper copies of reports from the on board maintenance system, as well as other documents.

Question: 18-3

The information gathered by central maintenance computers can be relayed to the ground through the _____ system.

Question: 18-7

Structure monitoring is also known as _____ monitoring.

Question: 18-4

The individual system BITE circuits are connected to the CMC by a _____.

Question: 18-8

The fatigue critical structure (FCS) monitoring process is accomplished by performing _____.

ANSWERS

Answer: 18-1

central maintenance computer (CMC).

Answer: 18-5

data loading system.

Answer: 18-2

MCDUs (multipurpose control display units)

Answer: 18-6

printers.

Answer: 18-3

ACARS.

Answer: 18-7

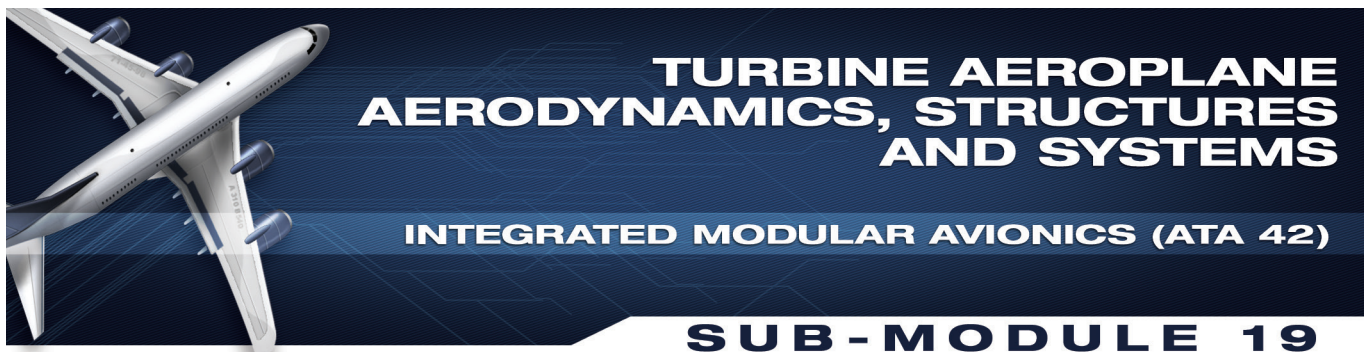
damage tolerance.

Answer: 18-4

digital data bus.

Answer: 18-8

damage tolerance inspections (DTIs).



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 19

INTEGRATED MODULAR AVIONICS (ATA 42)

Knowledge Requirements

11.19 - Integrated Modular Avionics (ATA 42)

Functions that may be typically integrated in the Integrated Modular Avionic (IMA) modules are, among others:

Bleed Management, Air Pressure Control, Air Ventilation and Control, Avionics and Cockpit Ventilation Control, Temperature Control, Air Traffic Communication, Avionics Communication Router, Electrical Load Management, Circuit Breaker Monitoring, Electrical System BITE, Fuel Management, Braking Control, Steering Control, Landing Gear Extension and Retraction, Tire Pressure Indication, Oleo Pressure Indication, Brake Temperature Monitoring, etc.,
Core System;
Network Components.

2

11.19 - INTEGRATED MODULAR AVIONICS

INTRODUCTION

This section describes integrated modular avionics (IMA). As the name indicates, an aircraft with IMA has avionics systems that are integrated and modular. Integrated means that multiple functions are combined into a single piece of equipment. Modular refers to a design method that allows the system to be divided into separate, replaceable modules. The term Avionics itself derives from aviation electronics, and it refers to electronic systems used in aircraft. Avionics encompasses a wide range of systems. Avionics are used for navigation, communication, control of the aircraft, and other purposes. On a modern aircraft, there are dozens of systems that can be considered avionics systems.

Integrated modular avionics is a design methodology, not an avionics system per se. In other words, it describes the way the avionics systems are put together, not the specific functions of the individual avionics systems. IMA represents an advance in avionics technology. Aircraft with IMA can realize reductions in the bulk and weight of their avionics systems. Also, the overall reliability of the avionics can be improved.

The following is just a partial list of functions that may be Integrated into an IMA System:

- Bleed Management
- Air Pressure Control
- Air Ventilation and Control
- Avionics and Cockpit Ventilation Control
- Air Traffic Communication
- Avionics Communication Router
- Electrical Load Management
- Circuit Breaker Monitoring
- Electrical System Built In Test Equipment (BITE)
- Fuel Management
- Braking Control
- Steering Control
- Landing Gear Extension and Retraction
- Tire Pressure Indication
- Brake Temperature Monitoring

INTEGRATION OF AVIONICS

Design methods for avionics systems have evolved over time. Initially, avionics systems were discrete, stand alone systems. This means that each system was separate.

For example, an aircraft's compass system might have consisted of a flux detector in the wing, a gyroscope in the avionics compartment, and a heading indicator on the instrument panel. These components were used only by the compass system. They were not shared by other systems on the aircraft. Although the compass system's components were connected to each other with wiring, the compass system itself was not connected to any of the other systems on the aircraft.

Traditionally, on non-IMA aircraft, each avionics system had its own separate indicator and its own separate controls. As more and more avionics systems were developed and installed in aircraft, more indicators and controls had to be installed. Instrument panels became more complex and crowded. *Figure 19-1* shows such an instrument panel. In addition, as more avionics systems were developed and installed, more LRUs (line replaceable units) or "black boxes" were installed in avionics compartments. More wiring was needed to interconnect these LRUs with their associated cockpit controls and indicators. More electrical power was needed to operate the systems.

Each additional indicator, LRU, and wire that is installed on an aircraft takes up space and adds weight. Because both space and weight-carrying capability are at a premium, it is desirable to keep the number of indicators, LRUs, and wires to a minimum. In the case of indicators, engineers began to develop designs that used the same indicator to display information from more than one system. For example, older designs had separate indicators for the compass system, the radio navigation system, and the weather radar system.



Figure 19-1. An older instrument panel with many discrete indicators.

In newer designs, these systems are all connected to single, "integrated" indicator such as a navigation display (ND). The use of integrated indicators saves space and weight, and it streamlines pilot workload by reducing the number of indicators that must be scanned during flight. **Figure 19-2** shows a modern, integrated instrument panel. The design concept that was first used to combine indicator functions in the cockpit has been carried further with integrated modular avionics. In an aircraft with IMA, the same concept – integration – is applied to many of the LRUs as well. Instead of individual, independent LRUs, an aircraft with IMA uses modules which are integrated into a single system. The modules perform the functions formerly performed by the independent LRUs, but they are not completely independent. They share circuitry.

Traditional (non-IMA) avionics suites have many separate LRUs ("black boxes") located in an avionics compartment. In such systems, there is a considerable amount of duplication among the black boxes. For example, each black box in the avionics compartment typically contains its own power supply. These power supplies receive aircraft power and use it to provide the various voltages needed by the circuits within that box. Also, each power supply is connected to the aircraft's electrical power system by a separate wire.



Figure 19-2. An instrument panel with fewer, integrated instruments.

These power supplies might be functionally identical to each other. In a non-IMA aircraft, there could be twenty identical power supplies in twenty separate boxes in the avionics compartment. However, if the power supply in one of the black boxes fails, that system fails because that box cannot use power from another black box.

In an aircraft with integrated modular avionics, some of the self-contained black boxes are replaced by modules. The modules form part of an integrated system because they are plugged into a mainframe or rack which is a single piece of equipment. This results in a bulk and weight reduction because some circuitry is now shared among the various modules.

For example, instead of having twenty duplicate, non-redundant power supplies, an IMA aircraft might have three redundant power supplies that are each capable of supplying all twenty modules. If one of the IMA power supplies fails, there are still two redundant power supplies that can power the modules. A failure of one or even two of the power supplies does not result in the failure of any of the avionics modules. Thus, the IMA aircraft is more fault-tolerant, resulting in better reliability.

The integration concept reduces the total number of LRUs needed. Often, the same kind of data processing circuitry is required for different avionics functions. In previous (non-IMA) designs, this data processing circuitry had to be duplicated in separate LRUs for each separate system. With IMA, the data processing circuitry can be contained in fewer LRUs (modules), and it is shared among the systems that require it. This combining of functions results in an overall reduction in the number of separate LRUs. Less wiring is needed in an IMA aircraft. This is partially due to the integration of functions that eliminates unnecessary duplication of circuitry. Using the example of the power supplies, with IMA there is no need to provide twenty separate wires from the aircraft electrical system for power supplies because there are only three power supplies for the IMA modules. In addition to this, the fact that IMA aircraft make extensive use of digital data buses also reduces the amount of wiring needed.

DIGITAL DATA BUS USE REDUCES WIRING

The use of digital data buses can result in a tremendous reduction in the amount of wiring used in an aircraft. Digital data buses are used to transfer information from one piece of equipment to another, using far fewer wires

than were previously required. A number of different digital data bus systems are used in various aircraft. Some of the more important ones are ARINC 429, ARINC 629, and AFDX (Avionics Full Duplex).

An example that illustrates this reduction in wiring is radio tuning. When a radio is tuned, frequency information must be transferred from a radio tuning unit to the radio receiver or transceiver that is being tuned. This frequency information might consist of four or more digits. Say a pilot wishes to tune a VHF communications transceiver to the frequency 128.35 MHz. The pilot enters this frequency into the tuning unit in the cockpit. From there, it must be carried to the VHF transceiver located in the avionics compartment (*Figure 19-3*).

With traditional avionics systems, those not using digital data buses, each piece of information to be transferred from one location to another requires at least one separate wire. Often, a far larger number of wires is needed. Using our example of a pilot tuning the VHF communications transceiver to 128.35 MHz, each digit of the selected frequency must be transferred from the tuning unit to the transceiver. Because there are five digits in this frequency, it might seem that five wires would be needed to transfer this data. However, because the information being transferred is complex (each digit might be anything from a zero through a nine), even more wires are needed. A typical pre-data-bus method of accomplishing this transfer was the ARINC "two out of five" or 2 × 5 tuning scheme. *Figure 19-4* shows the 2 × 5 tuning scheme.

2 × 5 tuning requires the use of five wires for each digit of information being transferred. Of these five wires, two are be connected to ground and three are not. The particular wires which are grounded determine whether the digit transferred is a zero, a one, a two, etc.

Since all VHF communication frequencies begin with the number "1," it is not necessary to transfer that digit from the tuning unit to the transceiver. However, each of the other four digits in the selected frequency must be transferred. Four digits at five wires per digit results in 20 wires. A ground wire is also needed, bringing the total number of tuning wires to 21. *Figure 19-5* illustrates this.

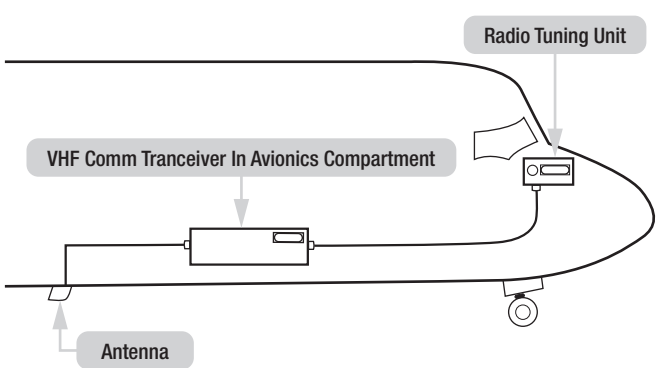


Figure 19-3. VHF Comm system interconnection.

2 x 5 Tuning Scheme

	0	1	2	3	4	5	6	7	8	9
A		+	+						+	+
B	+	+		+	+					
C			+	+		+	+			
D					+	+		+	+	
E	+						+	+		+

*Wires marked with + are grounded.

Figure 19-4. Two out of five tuning scheme.

Using a digital data bus like ARINC 429, this same tuning information can be transferred using only two wires. With ARINC 429, the two information-carrying wires are twisted together, and they are covered with a braided shield. The shield is usually kept grounded, and protects the two inner wires from electromagnetic interference. The two inner wires are referred to as a "twisted pair" (*Figure 19-6*).

The reason it is possible to transfer so much information on only two wires is that the information is sent serially. This means that the same two wires carry one bit of information at a time. One bit of information is sent, then another, and then another. Soon, all the required information has been transferred from the tuning unit to the radio. The information transfer is done at a rapid rate: ARINC 429 can transfer up to 100 000 bits of information each second. ARINC 629 can transfer up to two million bits per second. AFDX is a newer digital data bus system used that can transfer data at rates up to 100 million bits per second. Because far fewer wires are needed for each system when using digital data buses, a substantial weight savings is realized.

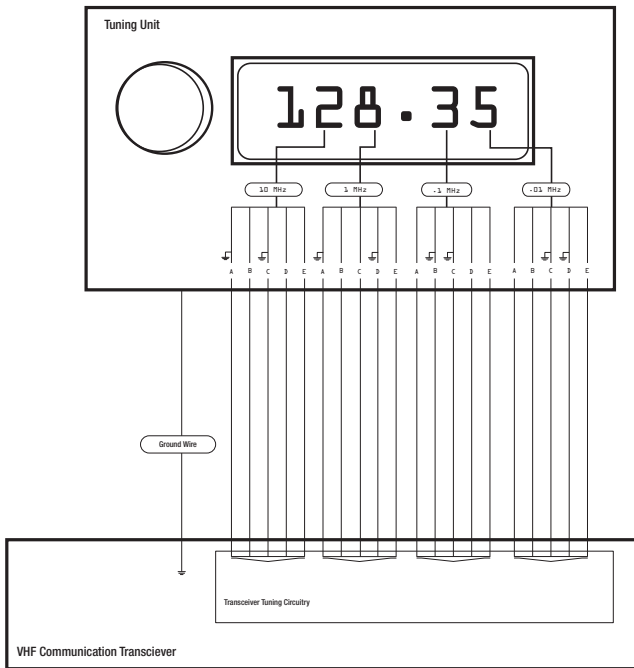


Figure 19-5. VHF Comm system tuning wires using two out of five tuning.

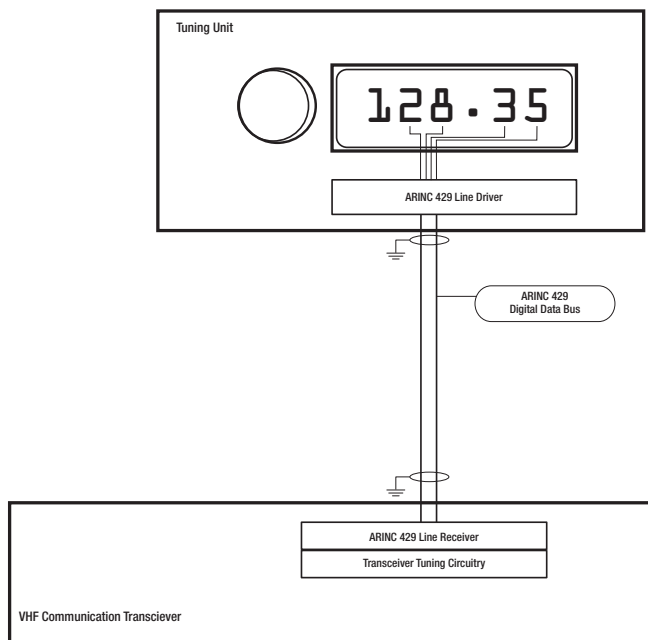


Figure 19-6. VHF Comm system tuning wires using ARINC 429 digital data bus.

CORE SYSTEM

IMA systems consist of a core system and network components. The core system contains data processing circuitry that processes many different kinds of information. This processing circuitry is shared by the various avionics systems that have been integrated. Information from various sensors, controls, and LRUs

is brought into the core system for processing, then sent out from the core system to displays, actuators, and other places in the aircraft where that information is used.

The core system uses the same computer processors for many different purposes. For example, a core system can use the same processor for such tasks as calculating throttle settings for best fuel economy, calculating the amount of rudder deflection needed to coordinate a turn, monitoring an instrument landing system receiver for malfunctions, determining if the stall warning system needs to be activated, and many others. The core system's processing power is shared among these various functions. This sharing eliminates the need to have a processor in each and every system.

COMMON CORE SYSTEMS CONCEPTS

This IMA system consists of a common core system with network components. The core system contains a central and backup computer and data processing circuitry. This processing circuitry is shared by the various avionics systems which have been integrated. Information from various sensors, controls, and line replaceable units (LRUs) are brought into the core system from data concentration modules. Data is then sent out from the core system to displays, actuators, and other places on the aircraft where that information is used. The core system uses the same computer processors for many different purposes.

CORE PROCESSOR INPUT/OUTPUT MODULE (CPIOM)

The CPIOM is a standard hardware platform designed to host several independent functions. For example CPIOMs for the utilities domain perform fuel management, measurement, and display; and can also control the landing gear extension systems, braking, and nose wheel steering software. CPIOMs in the energy domain use the standardized architecture to control electrical power distribution. CPIOMs in the cabin domain host interrelated functions such as cabin pressure control, air conditioning and ventilation, and can also integrate such systems as fire and smoke detection as well as the monitoring of aircraft doors and evacuation slides. Today, the CPIOM operative system is based on the latest version of the ARINC 653 standard. ARINC 653 manages time and space between safety avionics systems and normal operative systems.

NETWORK COMPONENTS

Network components are the parts of the IMA system that allow data to be transferred into and out of the core system. These components include input and output devices and data bus wiring. Input devices receive data from a digital data bus and couple it to the core system processing circuits. Output devices prepare core system output data for transmission along a digital data bus. The exact properties of the input and output devices vary with the particular kind of data transfer system that is being used. As mentioned above, there are several different data transfer systems used in aircraft, some of the more common ones being ARINC 429, ARINC 629, and AFDX.

777 AIMS

An example of an aircraft that uses integrated modular avionics is the Boeing 777. (*Figure 19-7*) In this aircraft, Boeing calls its IMA system the airplane information management system (AIMS). The core system of the 777 AIMS system is contained in two cabinets which are located in the aircraft's main equipment center. Each of the AIMS cabinets has ten line replaceable modules (LRMs) which plug into it. These modules are of four different types. The cabinets themselves contain network components and circuitry that is shared by all the LRMs installed in each cabinet. This common

circuitry is contained in a "backplane bus." The cabinets are also networked with other LRUs in the aircraft via digital data buses.

In the 777, the Line Replaceable Modules in the AIMS process information for the following avionics systems:

- Primary Display System (PDS)
- Flight Management Computer System (FMCS)
- Thrust Management Computer System (TMCS)
- Central Maintenance Computer System (CMCS)
- Airplane Condition Monitoring System (ACMS)
- Flight Data Recorder System (FDRS)
- Data Communication Management System (DCMS)

The AIMS is interconnected with many other units in the aircraft. It receives input data from these units, and it provides output data to them (*Figure 19-8*).

The AIMS uses several methods for sending and receiving information. These include six different digital data transfer formats, analog data transfer, and wireless RF links.

The primary data transfer format used by the AIMS is ARINC 629. The AIMS is connected to 66 LRUs located throughout the aircraft via ARINC 629. ARINC 629 uses a bi-directional data bus. This means

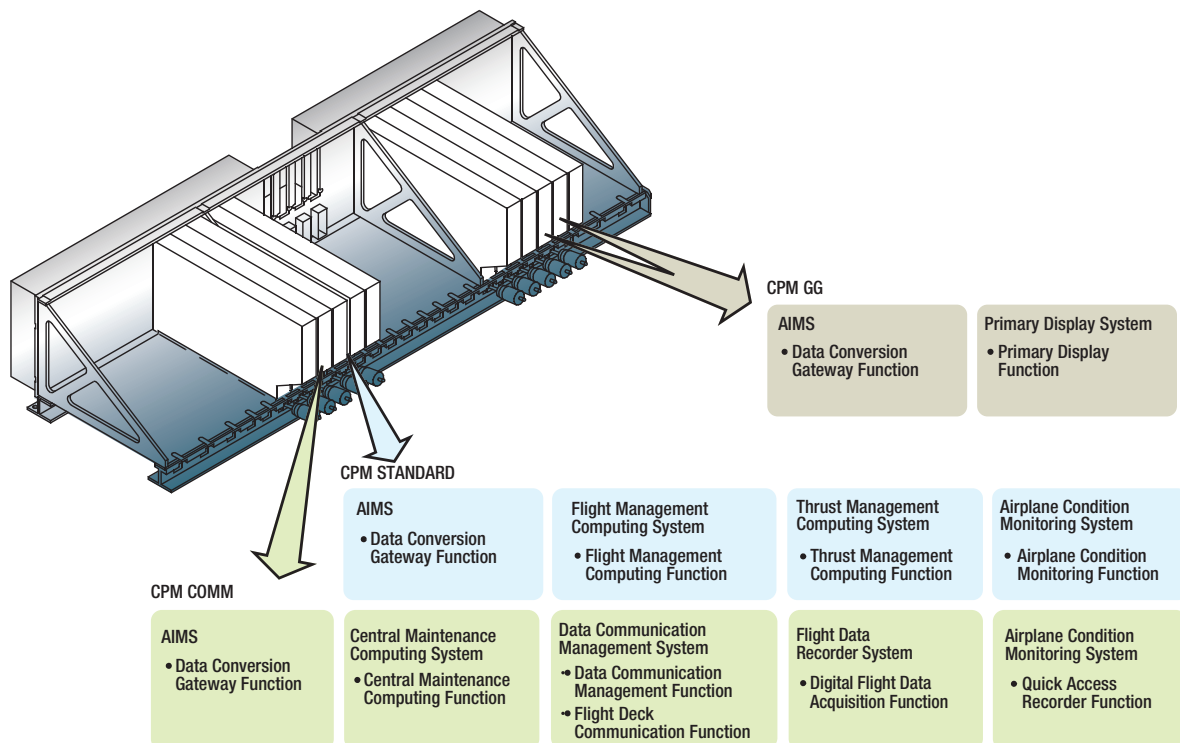


Figure 19-7. Boeing 777 Airplane Information Management System (AIMS) cabinet.

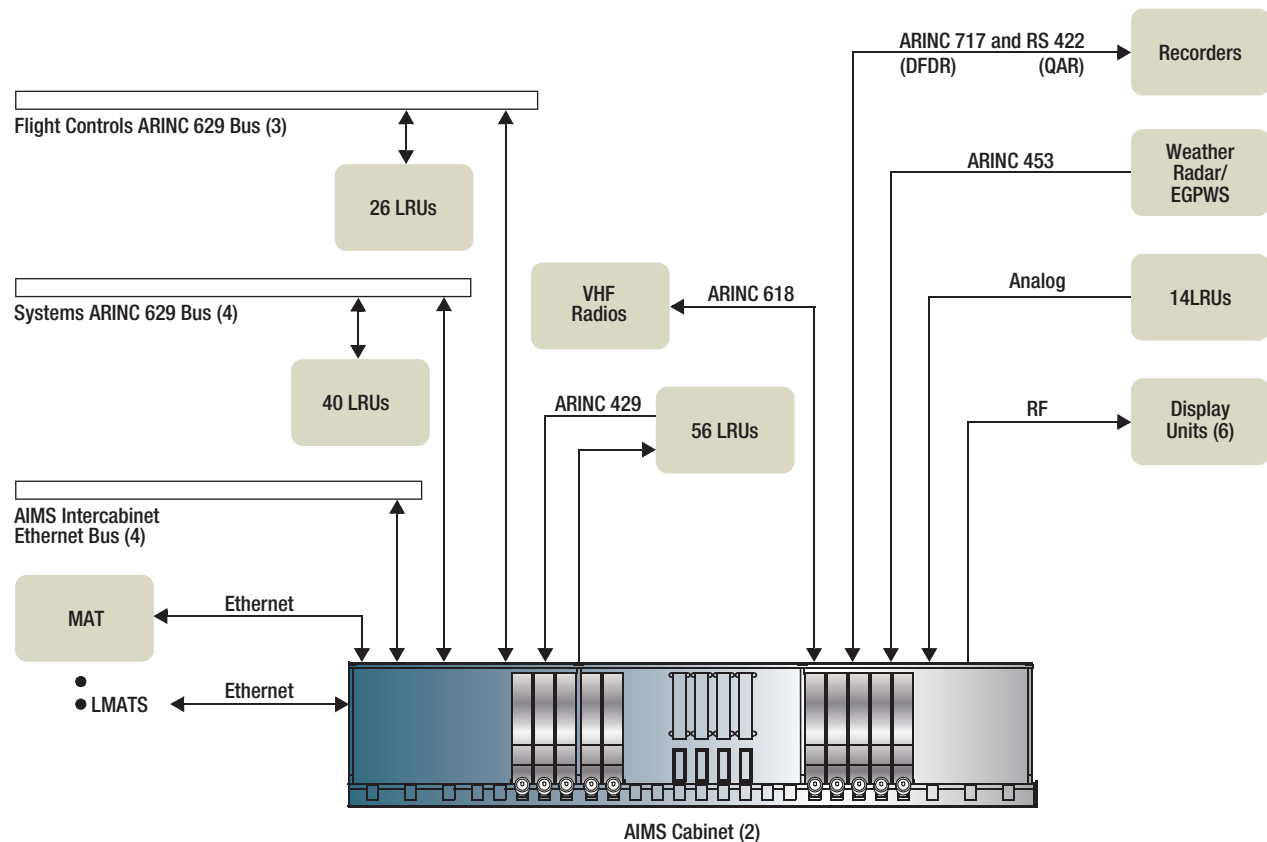


Figure 19-8. Boeing 777 AIMS Interfaces.

that data can be sent both ways on the same data bus. The AIMS can both send information to an LRU, and receive information from that LRU, using a single data bus connected between the two. ARINC 629 uses a more rapid data transfer rate (up to 2 Mbps) than some other digital data transfer systems.

The AIMS also uses the ARINC 429 data format to communicate with 56 LRUs throughout the aircraft. Data transfer using ARINC 429 is unidirectional. This means that information can only be transferred in one direction on a single ARINC 429 bus. If an LRU needs to send information to the AIMS, one bus is required.

If the AIMS needs to send information back to that LRU, a second bus is required. ARINC 429 data transfer rates are somewhat slower than those used with ARINC 629. The transfer rates range from 12 kbps to 100kbps. ARINC 429 is a widely used data transfer system that was first developed in the 1970s. It is used in many makes and models of aircraft.

The AIMS displays information on six flat-panel display units (DUs) (*Figure 19-9*). These units are located on the instrument panel in the cockpit. They show the information that has been gathered from the various systems connected to the AIMS, and which has been processed by the AIMS. These DUs are the main displays used by the flight crew. The DUs give the following types of displays:

- Primary Flight Display (PFD)
- Navigation Display (ND)
- Multifunction Display (MFD)
- Engine Indicating and Crew Alerting System (EICAS)

Switches are provided in the cockpit that allow the flight crew to control the DUs. These switches allow a particular type of display to be shown in different positions on the instrument panel. If one of the DUs fails, its function can be taken over by another functional DU. The system thus provides a great deal of flexibility.

Figure 19-10 shows a DU operating as a PFD. It shows aircraft pitch and roll attitude, airspeed, altitude, vertical speed, heading, and other information.

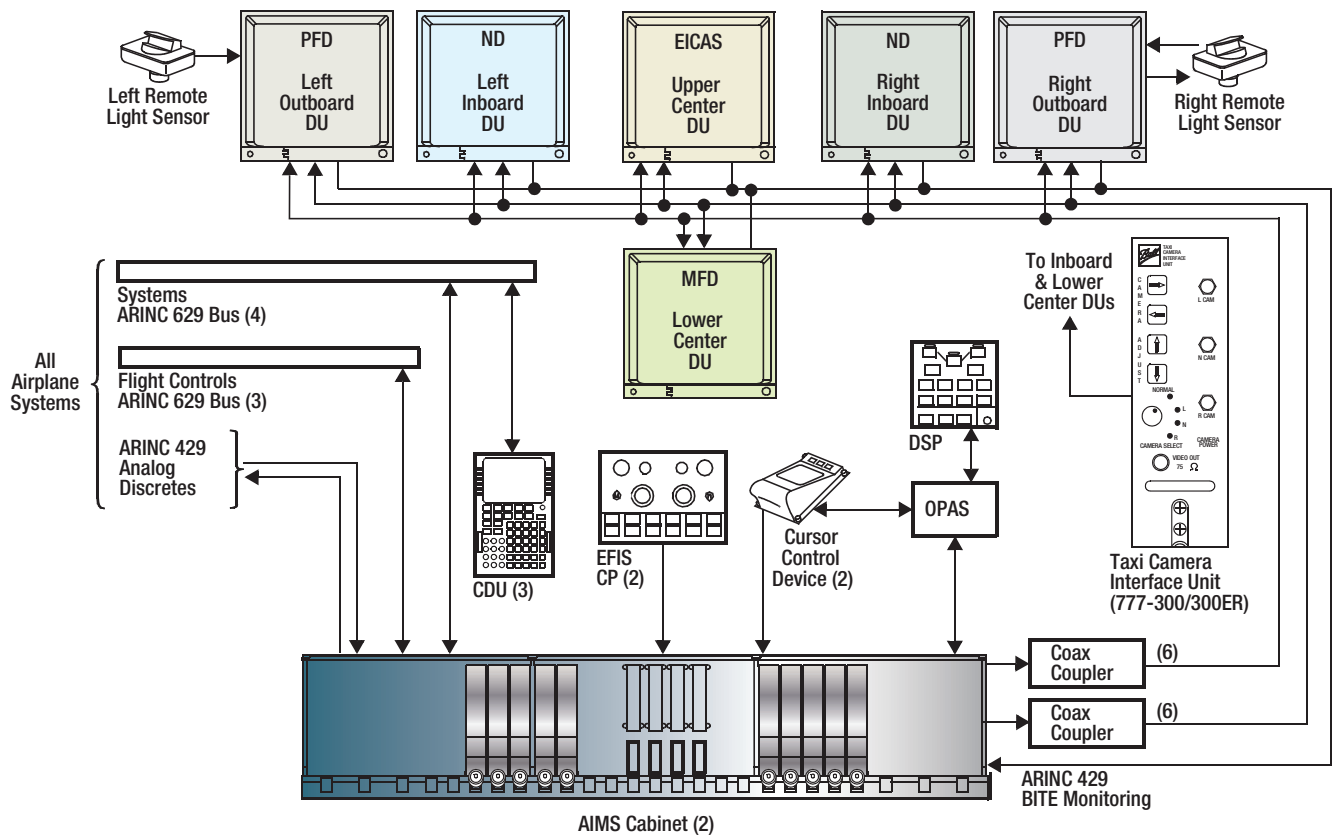


Figure 19-9. Boeing 777 Primary Display System, showing Display Units (DUs).

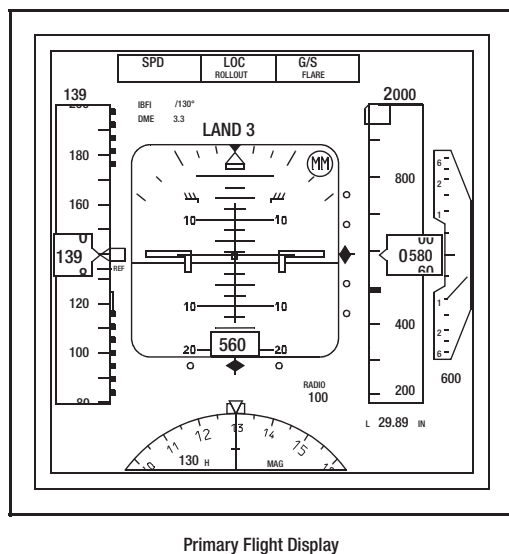
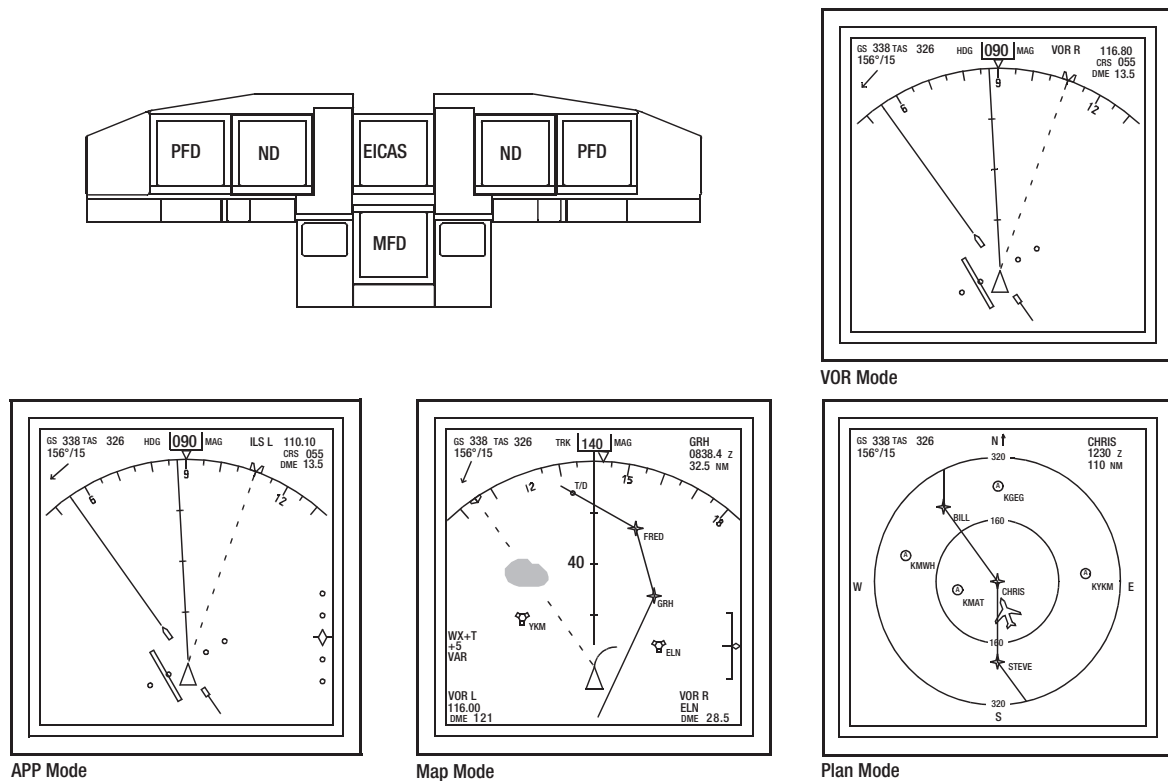


Figure 19-10. Display Unit operating as a Primary Flight Display (PFD).

Figure 19-11 shows a DU operating as a ND. The ND can display data in four different modes. These modes are VOR, Plan, Map, and Approach. The displays are laid out differently for each mode, and different information is displayed in different modes. The ND shows heading, VOR, DME, localizer, glideslope, TCAS, along with other information.

Figure 19-12 shows a DU operating as an EICAS display. This display shows engine operating parameters such as EPR, N1, EGT, etc. It shows fuel quantity, landing gear position, flap/slat position, and other aircraft system information. It also provides warnings, cautions and advisories to the flight crew.

Figure 19-13 shows a DU operating as an MFD. The MFD can display many different kinds of information. Different screens (formats) can be selected. These formats are Secondary Engine Display, Status Display, Synoptic Display, Maintenance Page, Communication Display, Electronic Checklist, and Ground Maneuver Camera Display.



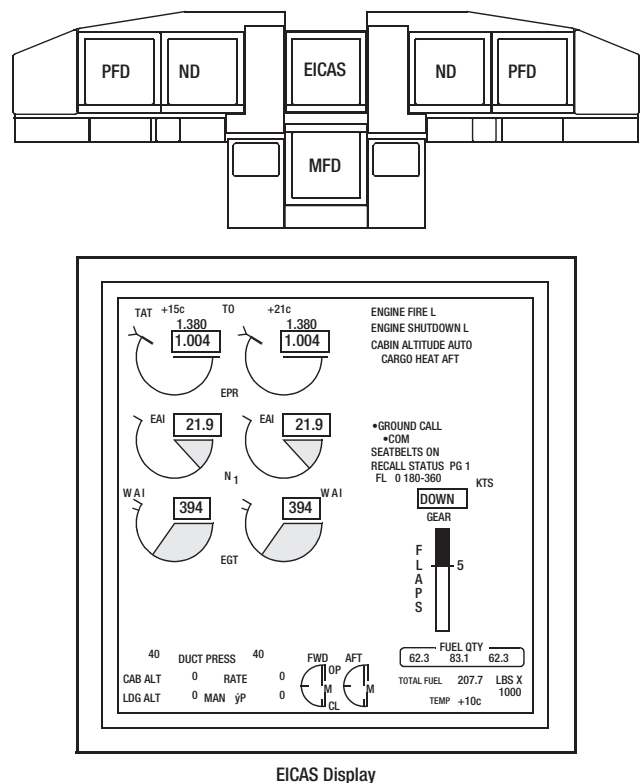
Navigation Display

Figure 19-11. Display Unit operating as a Navigation Display (ND).

The Boeing 777 AIMS is one implementation of an integrated modular avionics system. Other aircraft's IMA systems vary somewhat in their specifics. The exact system architecture and the terminology used varies from one manufacturer to another, and from one model to another. However, the basic features of IMA are:

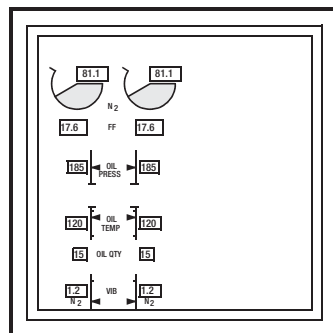
- Modular design
- Integration of modules within the system
- Use of digital data buses for interconnection
- Fault tolerance
- Flexibility

These features are found in any IMA system.



EICAS Display

Figure 19-12. Display Unit operating as an Engine Indicating and Crew Alerting System (EICAS) display.



HYDRAULIC			
L	C	R	
QTY 0.91	0.90	0.72	RF
PRESS 3000	3000	3000	

JPU			
RPM 100.1	EST 560	C	
OL PRESS 82 PSI	OL TEMP 75C	OL QTY 7.9	

OXYGEN	
CREW PRESS 1950	

ELEC GEN OPS 1
FLAPS/LAT CONTROL #2

The schematic diagram illustrates a three-channel test system. It consists of three main sections labeled 2990, 5010, and 3010, each connected to a common ground and a common output line. Each section contains a 5.00V source, a 200V source, and a 100V source. The 2990 section also includes a 1.00V source and a 1.00V source. The 5010 section includes a 1.00V source and a 1.00V source. The 3010 section includes a 1.00V source and a 1.00V source. The channels are connected to a common ground and a common output line.

[illegible]

ATC	FLIGHT INFORMATION	COMPANY
REVIEW	MANAGER	NEW MESSAGES

[illegible]

Module 11A B1 - Turbine Aeroplane Aerodynamic, Structures and Systems

Question: 19-1

Aircraft with IMA can realize reductions in the bulk and _____ of their avionics systems?

Question: 19-5

The wires inside the braided shield of an ARINC 429 data bus are referred to as a _____.

Question: 19-2

The use of _____ indicators saves space and weight, and it streamlines pilot workload by reducing the number of indicators that must be scanned during flight.

Question: 19-6

The _____ uses the same computer processors for many different purposes.

Question: 19-3

IMA aircraft are typically _____ fault-tolerant than traditional aircraft.

Question: 19-7

_____ include input and output devices and data bus wiring.

Question: 19-4

The use of digital data buses _____ the amount of wiring used in an aircraft.

Question: 19-8

Display units (DUs) use _____ to select the particular type of display desired.

ANSWERS

Answer: 19-1
weight.

Answer: 19-5
twisted pair.

Answer: 19-2
integrated.

Answer: 19-6
core system.

Answer: 19-3
more.

Answer: 19-7
Network components.

Answer: 19-4
decreases.

Answer: 19-8
switches.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1

Sub-Module 20

CABIN SYSTEMS (ATA 44)

Knowledge Requirements

11.20 - Cabin Systems (ATA 44)

The units and components which furnish a means of entertaining the passengers and providing communication within the aircraft (Cabin Intercommunication Data System) and between the aircraft cabin and ground stations (Cabin Network Service). Includes voice, data, music and video transmissions.

The Cabin Intercommunication Data System provides an interface between cockpit/cabin crew and cabin systems. These systems support data exchange of the different related LRU's and they are typically operated via Flight Attendant Panels. The Cabin Network Service typically consists on a server, typically interfacing with, among others, the following systems:

- Data/Radio Communication, In-Flight Entertainment System.
- The Cabin Network Service may host functions such as:
 - Access to predeparture/departure reports,
 - E-mail/intranet/internet access,
 - Passenger database,

Cabin Core System;

In-flight Entertainment System;

External Communication System;

Cabin Mass Memory System;

Cabin Monitoring System;

Miscellaneous Cabin System.

2

11.20 - CABIN SYSTEMS

INTRODUCTION

Cabin Systems include those used to communicate both within the aircraft cabin and between the cabin and the cockpit. They also include systems for inflight entertainment (IFE) of passengers, and systems used by passengers to communicate with the ground.

Like most aircraft systems, cabin systems have evolved and upgraded over the years. Older aircraft used analog electronics, while newer designs use digital electronics. In the case of IFE video, early systems used video projectors and large screens visible to all passengers. Video was stored on reels, and later on videocassettes. There was only one choice of video to watch. Newer aircraft are equipped with individual flat-panel video displays at each seat, and video is stored digitally. Passengers can select from many different video sources. Early systems used pneumatic headsets for passenger IFE audio; modern aircraft use electronic headsets.

As with all complex systems, there are variations in cabin systems. Different models of aircraft have different systems and different features. Some of the features that can be included in cabin systems are:

- Passenger Address - for addressing passengers through overhead cabin speakers
- Cabin Interphone - for communication among the cabin crew members, and also between the cabin and the flight deck
- Flight Attendant Calling - allows passengers to call for assistance from the cabin crew
- Audio and Video Entertainment
- Moving Map Displays - showing the aircraft's current location, altitude, and airspeed
- Telephone, Fax and Internet Service.

CABIN INTERCOMMUNICATION DATA SYSTEM

A cabin intercommunication data system is used by flight attendants, pilots, and passengers. Typical user interfaces for the system are flight attendant panels, cabin handsets, cockpit handsets, and flight attendant call buttons at passenger seats. There are also speakers and passenger information lights/signs.

Flight attendants can use the system to call each other in various parts of the cabin. For example, a flight attendant located at the galley at the front of the cabin can call another flight attendant who is at the rear of the cabin. Also, the pilots on the flight deck can call the flight attendants at their stations in the cabin, and vice versa. Flight attendants can make general announcements to the passengers. Passengers can activate flight attendant call lights. An example of a cabin intercommunication system is described below under 777 Cabin Services System.

CABIN NETWORK SERVICE

A cabin network service is a digital system that is typically hosted on a server within the aircraft. It provides services, such as access to email accounts and the Internet. It may also provide access to data stored in databases. A cabin network service is essentially a computer Local Area Network (LAN) within the aircraft, which can interface with external networks

The interconnections within the LAN can be wired, or they can be wireless. The design trend is toward wireless connections. Because of the potential for interference with other systems, the use of wireless is restricted to certain phases of flight. Operation of wireless systems is not permitted during takeoffs and landings.

Although systems vary, and new features are added to each new version, typical uses of the cabin network service are for passengers to connect to the internet while in flight, and to access In Flight Entertainment (IFE). In fact, on some aircraft the cabin network service is integrated with the IFE into a single system. Some systems also permit interconnection to satellite communications systems for inflight telephone calls.

Cabin Networks can transfer large volumes of data to many different locations by using multiplexing and de-multiplexing techniques, which are described below under IFE.

CABIN CORE SYSTEM

777 CABIN SERVICES SYSTEM

A specific example of an aircraft with a multi-featured cabin system is the Boeing 777. In the 777, the system is called the cabin services system (CSS). It includes both the cabin intercommunication data system and the cabin network service. **Figure 20-1** shows a block diagram of the 777 Cabin Services System.

As can be seen in **Figure 20-1**, the system is digital and makes extensive use of digital data buses. The system uses three different data bus systems: ARINC 629, the CSS Intersystem Bus, and ARINC 628. The ARINC 629 bus

connects to the aircraft information management system (AIMS), which is the primary information management system for the aircraft. The CSS intersystem bus is a data transfer bus internal to the cabin system. ARINC 628 is a data interface specifically designed for use with inflight entertainment (IFE) systems. There are several available IFE systems for an operator (airline) to choose from, and the 777 can accommodate any IFE system that uses the ARINC 628 interface.

The central processor of the system is the cabin systems management unit (CSMU). This unit is connected to the aircraft's main ARINC 629 data bus. The CSMU connects to the IFE system through a different data

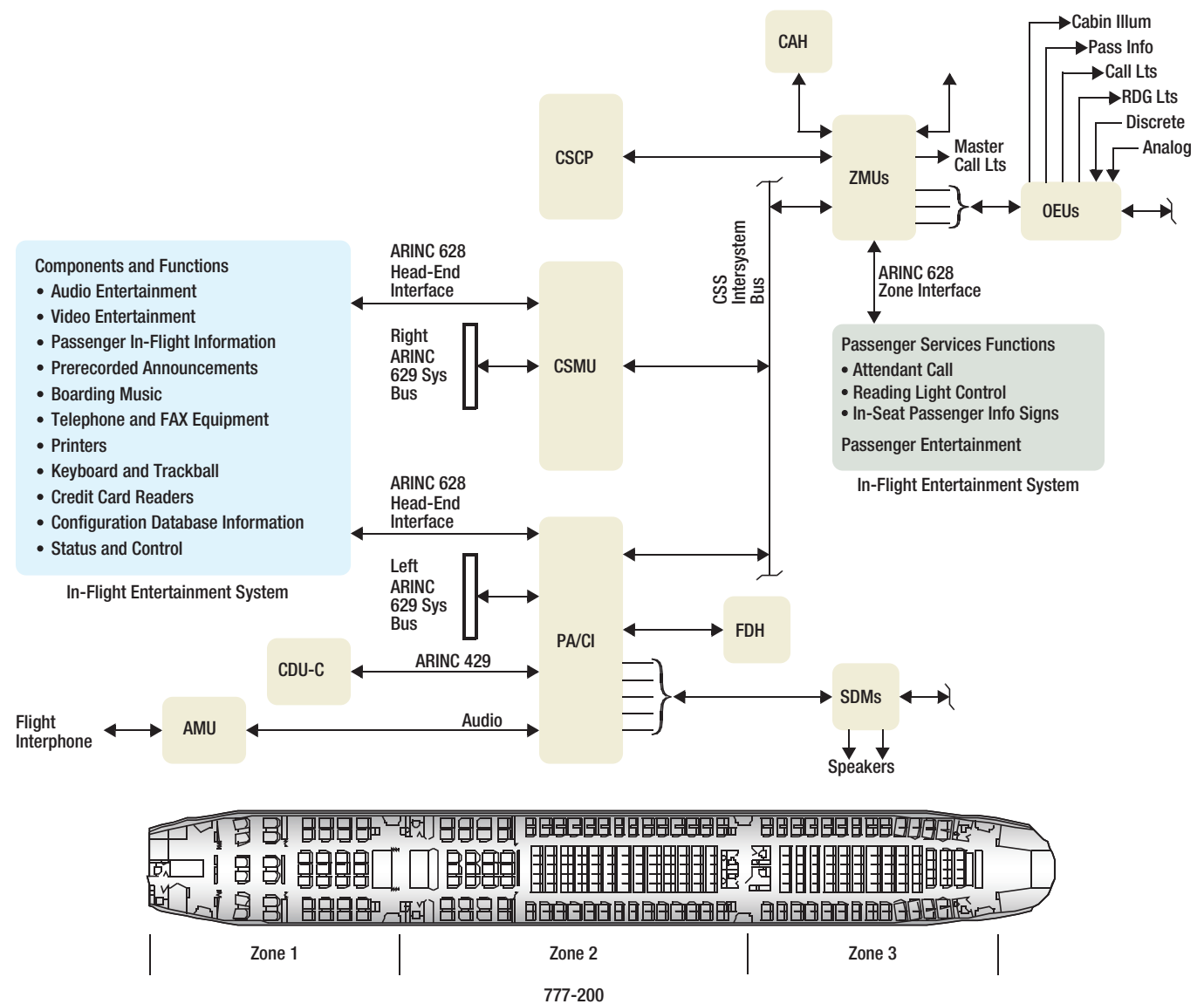


Figure 20-1. Cabin services system.

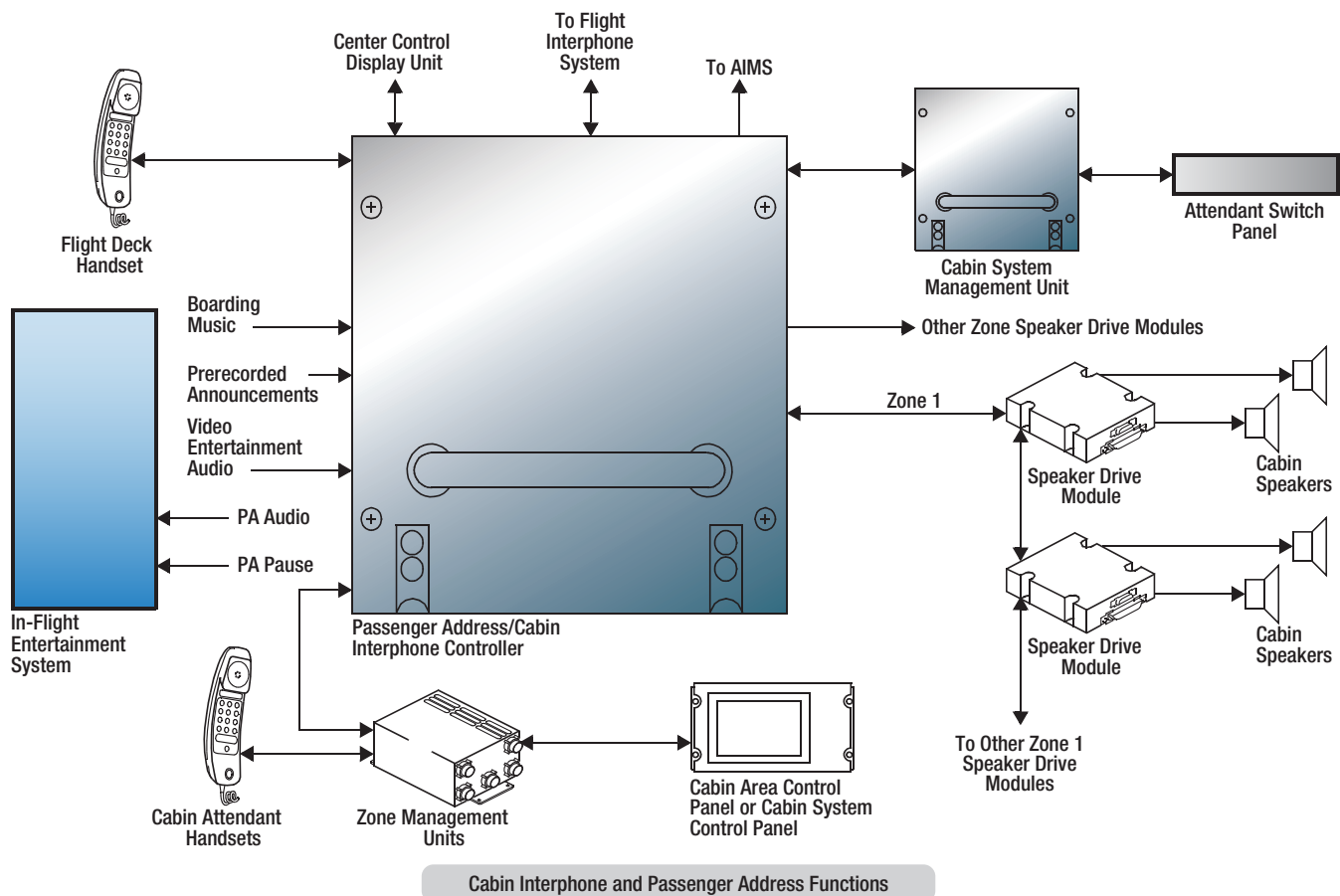


Figure 20-2. Cabin interphone and passenger address functions.

bus—the ARINC 628 bus. There is also a passenger address /cabin interphone controller, which the cabin crew uses to control the passenger address and cabin interphone functions. The cabin systems control panel is an interface used to load data into the system. The aircraft's cabin is divided into zones. Each zone has a zone management unit (ZMU), which is a processor connected to the rest of the system through a data bus called the "CSS intersystem bus." Each ZMU is controlled by a cabin area control panel. Cabin attendant handsets are connected to the ZMUs. These handsets are similar to telephones. They are used by the flight attendants to communicate with each other, and with the flight crew. They are also used for making announcements to the passengers. (*Figure 20-2*)

Each ZMU is connected to a series of overhead electronic units (OEUs) within its zone. At each passenger seat there are reading lights and attendant call lights. These lights are connected to the OEUs, which in turn are connected to the ZMU for the entire zone.

INFLIGHT ENTERTAINMENT (IFE)

In flight entertainment has been a rapid development area in the past decades. Original IFE consisted of a few audio channels for the passenger to select. If they missed the start, they had no option but to begin listening part way through. It then advanced to a visual aspect, with the first systems being a projector in the ceiling with throw down screens at the front of the cabin. With the development of video and screen technology, the choice increased for passengers. Multiple tape players were introduced providing a choice of films sent to small screens or audio to headsets. It was not until the introduction of CDs, DVDs and solid state memory did the passengers begin to get video or audio on demand.

Today's modern systems house hundreds of film and audio files with each passenger able to watch or listen to what they want at any time from the beginning. Further additions have now been put to IFE with passengers now having the ability to phone, text, or email from their seats together with the ability to browse the internet, make purchases on-board, and view flight details.

Each passenger seat in the 777 has IFE equipment. The IFE equipment typically includes a video display, an audio headphone jack, and controls for selecting specific video or audio content to be enjoyed by the passenger. It can also include other items, such as a credit card reader, or a port for connecting the passenger's own personal equipment to the system. The system may allow for connection of a device to the internet.

These functions are provided by a central source. This central source may be called the main IFE computer. As stated previously, various versions of IFE systems are available from various manufacturers, and their terminology and features differ somewhat.

In the 777 system, the main IFE computer is connected to the Cabin Services System via an ARINC 628 data bus. The data routes from the IFE computer, through the CSMU and the ZMUs, to the individual passenger seats. This is a two way connection. Passenger requests for particular content (e.g. specific audio, video, games) are sent to the main IFE computer, and the content is then sent from the main IFE computer to the passenger.

In flight, the main IFE computer is constantly receiving requests for different content from different passengers, and it must route the requested content to the correct passenger. This content, which may be audio, video, text, or something else, has been stored as digital data. It will be transferred as a data stream. The computer may be called upon to send out a large number of different data streams to the various passenger seats, all at the same time. This is accomplished by breaking up the data into small packets, attaching an address to each packet, then sending the packets at high speed down the data bus. The addresses indicate which passenger seat will receive which data packets. The data packets for all the seats are placed onto the data bus. The order in which they are sent does not matter, because the addresses will allow them to be sorted out at the receiving end. This process is known as multiplexing.

At the receiving end, the data packets are sorted by address. All the data packets labeled with a particular address (for a particular passenger) are reassembled. This is known as de-multiplexing. The data is converted back into audio, video, text, or whatever content it originally represented and delivered to the passenger. Each passenger receives only the data that was addressed to

his or her seat. A high-speed digital data transfer system can multiplex and de-multiplex many channels, and carry information to many destinations simultaneously.

SIMPLE IFE SYSTEM

Figure 20-3 shows a simple IFE system which provides audio on demand and video on demand.

IFE SERVER

The IFE server houses the data files. The user (passenger) demands content from the server via their hand set. The requested content is sent to the multiplexer. The job of the multiplexer is to share the available bandwidth with all the users. The content will be split into packets and will be allocated a time slot to send the data to the user. This is possible as we do not need a constant connection for each user. As long as the user keeps receiving packets, it looks as if it is one continuous stream. This splitting up of the data and interleaving it with other users is called multiplexing.

EXTERNAL COMMUNICATION SYSTEM

Although a large amount of data can be stored on board the aircraft and accessed by the cabin system, it is also useful to communicate with points outside the aircraft. This involves connecting with the internet, and with the telephone system. To do this, the aircraft's external communication system is used to connect with either a ground-based network or a satellite-based network. There are several commercial networks available, and each of them has advantages and disadvantages. Changes and upgrades to these networks occur frequently.

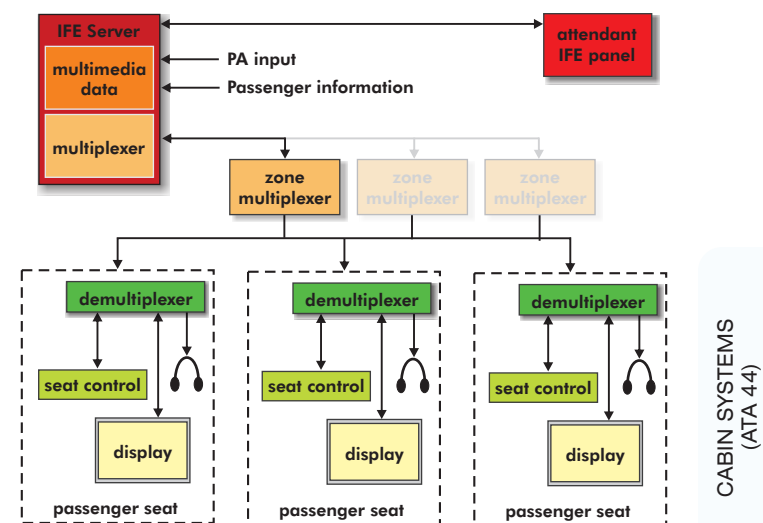


Figure 20-3. A simple IFE system.

For connecting with the external networks, the aircraft uses a router that is connected to dedicated radio transmitting and receiving equipment. This radio equipment uses various frequencies, depending on which external network is being accessed. Antenna size and shape also varies according to the network being used. For satellite-based systems, the antennas must be mounted to the upper surface of the aircraft, typically on the top of the fuselage or in the tail section. For ground-based systems, they will be mounted on the lower surface of the aircraft, usually on the belly of the fuselage.

Ground-based networks use radio towers to communicate with the aircraft. These towers have a limited range, so ground-based systems can provide coverage only over limited geographical areas. Ground-based systems tend to have a wide bandwidth, allowing for higher-speed data transfer. This makes web pages load faster, and results in good-quality audio and streaming video.

Several satellite systems are available for use by aircraft systems. The coverage areas and capabilities of these systems vary. For example, the Inmarsat system uses geostationary satellites, which orbit at an altitude of approximately 22 000 miles (approximately 36 000 km). Inmarsat provides coverage over most of the earth's surface, but not in the polar regions. The Iridium system uses low-earth orbit (LEO) satellites, which orbit at around 500 miles (800 km) altitude. With over 60 satellites in orbit, Iridium satellite coverage is truly worldwide. Satellite networks, while providing a larger coverage area than ground-based networks, tend to operate at somewhat slower speeds, and with narrower bandwidths than ground-based networks.

(Figure 20-4)

CABIN MASS MEMORY SYSTEM

The cabin mass memory system is where the data that is used in the cabin is stored. The mass memory system can be programmed with cabin configuration information. This is information about the way the seats are configured, for example, which seats are first class, and how many zones the cabin is divided into. The cabin configuration programming varies, even the same make and model of aircraft can have several possible cabin configurations.

The mass memory system can store many types of data files. This includes audio and video files that are used for passenger briefing and announcements, and can include

IFE such as prerecorded music. An aircraft's mass memory system is accessible to the cabin crew through user interfaces (control panels). Although it is stored in a central location, data from the mass memory system can be transferred to access points around the cabin using digital data buses. The system can be updated when necessary through a data loading system.

CABIN MONITORING SYSTEM

Cabin monitoring systems are used to monitor various conditions in the passenger cabin. These conditions can include (among others): Cabin temperature, Lighting, Status of passenger warning lights such as "Fasten Seat Belts" and "No Smoking" lights, cabin entry and exit door status (open or closed), status of smoke and fire detectors in lavatories and elsewhere in the cabin, and Status of galley equipment, such as water tanks.

(Figure 20-5)

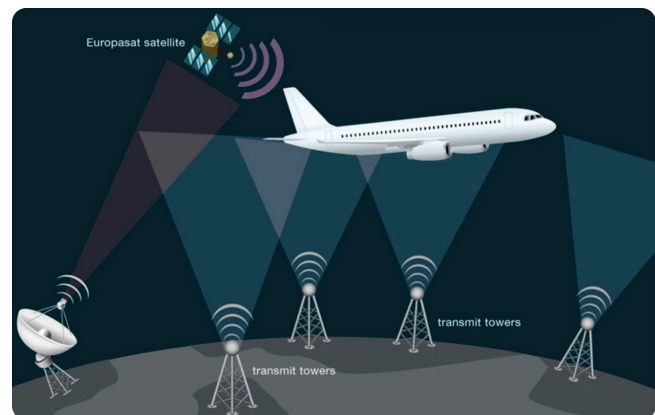


Figure 20-4. Inmarsat's air-to-ground network for inflight internet.

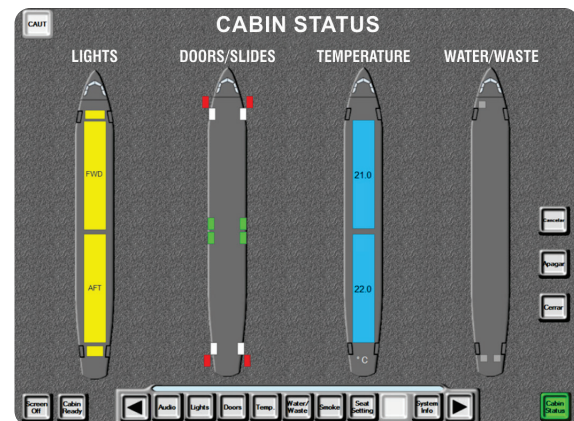


Figure 20-5. A monitoring panel accessible for flight attendants gives quick access to basic cabin functions.

Information about the conditions being monitored can be accessed by flight attendants in real time. It can also be stored and accessed later for purposes such as troubleshooting and trend monitoring.

Depending on the complexity of the aircraft, these panels may range from just simple intercom stations to the inclusion of many cabin wide functions such as lighting control, access to passenger databases, environmental conditions, water and waste holding tank status and other features.

MISCELLANEOUS CABIN SYSTEMS

In the new generation of Airbus Aircraft, Cabin Intercommunication Data Systems (CIDS) operates and monitors various passenger and crew functions such as:

- Passenger/cabin announcements.
- Cabin temperature control including interfaces with the air conditioning system to set and monitor cabin temperature, heaters, and dehumidifiers.
- Water/waste tank level indication including functions such as depressurization, auto-flush, and system shutdown.
- Cabin illumination controlled by zone, area, or seat
- Emergency and evacuation signals.
- Lavatory smoke warning and indication.
- Aircraft doors and slides status.
- IFE system status.
- Lighted signs including no smoking, fasten seat belt, and lavatory occupied signs.

Most of these functions were once controlled exclusively from panels installed in the cockpit. Now, with the use of touch screens, it is possible for the cabin crew to perform these adjustments in real time for each seat, row, or area inside the aircraft.

Question: 20-1

A _____ is used by flight attendants, pilots, and passengers. Typical user interfaces for the system are flight attendant panels, cabin handsets, cockpit handsets, and flight attendant call buttons at passenger seats.

Question: 20-5

To connect with the external networks, aircraft use a _____ that is connected to dedicated radio transmitting and receiving equipment.

Question: 20-2

A _____ is essentially a computer Local Area Network (LAN) within the aircraft, one that can interface with external networks.

Question: 20-6

Passenger briefings and announcement can be stored in the _____.

Question: 20-3

On a Boeing 777, the cabin system management unit contains the _____ of the cabin services system.

Question: 20-7

Cabin entry and exit door status is monitored by the _____.

Question: 20-4

When an inflight entertainment computer is called upon to send out a large number of different data streams to various passenger seats, all at the same time, a process called _____ is used in conjunction with a digital data bus.

ANSWERS

Answer: 20-1

cabin intercommunication data system

Answer: 20-5

router

Answer: 20-2

cabin network service

Answer: 20-6

cabin mass memory system.

Answer: 20-3

central processor

Answer: 20-7

cabin monitoring system.

Answer: 20-4

multiplexing



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →

B1**Sub-Module 21****INFORMATION SYSTEMS (ATA 46)**

Knowledge Requirements

11.21 - Information Systems (ATA 46)

The units and components which furnish a means of storing, updating and retrieving digital information traditionally provided on paper, microfilm or microfiche. Includes units that are dedicated to the information storage and retrieval function such as the electronic library mass storage and controller. Does not include units or components installed for other uses and shared with other systems, such as flight deck printer or general use display.

Typical examples include Air Traffic and Information Management Systems and Network Server Systems:

Aircraft General Information System;

Flight Deck Information System;

Maintenance Information System;

Passenger Cabin Information System;

Miscellaneous Information System.

2

11.21 - INFORMATION SYSTEMS

AIRCRAFT GENERAL INFORMATION SYSTEMS

Aircraft operations and maintenance involve dealing with large quantities of information. This information must be stored in some manner, and, ideally, it should be capable of being accessed as efficiently as possible. Also, the information changes from time to time. To keep up with the changes, there must be a way to update the information stored in an information system.

Aircraft information systems have evolved over the years. Initially, paper was the storage medium used. Manuals, drawings, charts, and other publications were available only in printed form. Information was retrieved by physically locating and reading the pages containing the information needed. Paper information sources have the advantage of being self-contained. No special equipment is needed to access the information. When revisions are required, new pages are printed and distributed. The old pages are removed and discarded, and are replaced by the new, revised pages.

Because large amounts of paper are bulky and heavy, other methods for storing large quantities of information were developed. Microfilm and microfiche were methods that saved space and weight. These methods involved using tiny photographic images of the pages contained in manuals, drawings, charts, etc. A disadvantage of microfilm and microfiche was that special equipment was needed to magnify the images so that they could be viewed. The microfilm or microfiche was useless if the viewing equipment malfunctioned, or if it was not available when the information needed to be accessed. Updates to microfilm and microfiche came in the form of new rolls of microfilm, or new microfiche sheets. After the new rolls or sheets were received, the old media was removed and discarded.

Digital computers represent a major advance in information technology. Modern computers can store and process very large amounts of information. They are compact, and they are very lightweight compared to other storage media, such as printed books. Computers store information in a type of memory known as ROM (read-only memory). This ROM can be internal to the computer, as in a hard drive. It can also be portable, contained in compact disks, flash drives, and

other forms. The information contained in the digital memory can be accessed through display screens. It can be transferred in and out of the computer on wired data buses and wirelessly, and it can be printed. When updates are needed to the information contained in ROM, the memory can be electronically erased and written over. There is no need to physically remove and replace paper or film. Aircraft information systems can be used to store and retrieve many kinds of information for various users. Examples are flight deck information systems, maintenance information systems, and passenger cabin information systems.

Memories used on modern aircraft are solid state drives that have completely replaced the older hard disk drive systems. These on board information systems (OIS) provide a method of storing, updating, and retrieving digital information. Applications for these functions are hosted on three sub networks or "domains" of a network server system known as avionics domain, flight operations domain, and communications and cabin domain.

THE AIRBUS SYSTEM

The architecture selected by Airbus is based on a system of networked real-time servers and routers combined with the centralized acquisition of secure digital communications. Although open to the world via digital radio links, the entire on-board system is designed to be highly secure both from the point of view of computer security and operational availability. The Airbus system collects, centralizes, and compiles all the data related to the flight and provides external communication, data calculation and storage capacities. This modular central system also hosts application unique to Airbus and airline companies that deal with the actual operation of the plane from services to passengers to on board documentation, navigation, performance calculations, flight logs, and more. The information system is made up of four components that operate in an integrated way.

Network Server System (NSS)

The network server is the system's backbone. One part of the NSS is highly secured and strictly devoted to avionics. Another part, containing information and documents related to flight operations is connected to the outside world; for example in-flight entertainment and wireless connections).

Secure Communication Interface

The secure communication interface is a link between the avionics and open world. As a basic component for the whole network's security, it guarantees the security of information exchanged between the IFE and the avionics systems, as well as the security of the ground-to-air and air-to-ground communications.

Central Data Acquisition Module (CDAM)

The central data acquisition module (CDAM) is a maintenance monitoring system capable of recording and analyzing up to a million parameters. It can generate over a hundred different maintenance reports concerning the condition of the aircraft and possible technical failures. Operators can program and configure the CDAM based on their needs, but also decode and display reports generated on board using ground programming and reading tools.

Data Loading And Configuration Systems

Airbus aircraft are also equipped with a unique data loading and configuration system. This is an application software for downloading and managing the configuration of various on board computer software.

FLIGHT DECK INFORMATION SYSTEM

777 ELECTRONIC FLIGHT BAG

An example of a flight deck information system is the Electronic Flight Bag (EFB), an optional system used on the Boeing 777 and on other aircraft (*Figure 21-1*). The flight crew uses the EFB to access information that would traditionally have been found in various printed publications and carried aboard in a flight bag. Such publications include sectional charts and approach plates. The system also provides advanced capabilities beyond those available in printed publications.

On a 777 equipped with the EFB, the system displays information on the two Display Units (DUs) that are installed on the flight deck. One DU is for the Captain and one is for the First Officer. These two DUs are touch-screen devices, and operate independently of each other. The system can be navigated by the touch-screen DUs, and through keyboards and cursor-control devices. The DUs are connected to two electronics units (EUs) located on the airplane information management system (AIMS) rack in the main equipment center. The AIMS is an integrated system for processing information from many sources in the aircraft.

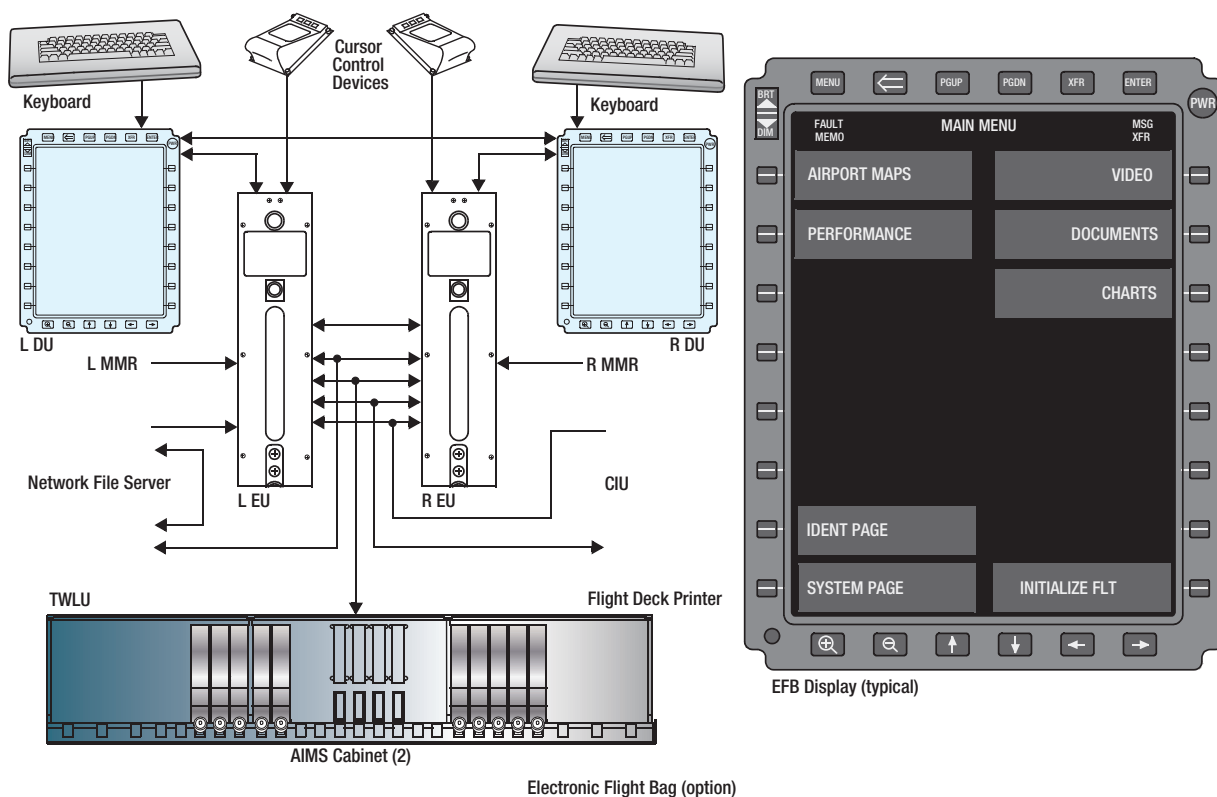


Figure 21-1. Electronic flight bag.

In the 777 system, the EUs send information stored in databases to the DUs through fiber-optic cables. The two DUs are connected to each other through wired connections. Databases accessible through the EFB include aeronautical charts, airport maps and charts with real-time position monitoring, manuals, minimum equipment lists, and logbooks. The system can also display video. The EFB system receives data from the AIMS and from multi-mode receivers (MMRs). MMRs are GPS receivers designed for use in instrument landings. The MMRs provide aircraft position information that is extremely precise. This allows the EFB to pinpoint the aircraft's position on an airport map. The system also receives video from the Camera Interface Unit (CIU). This video is from the flight deck entry surveillance camera.

The 777 Electronic Flight Bag's databases can be updated through the aircraft's data loader, but they can also be updated wirelessly. For wireless updates, the system uses a Terminal Wireless LAN Unit (TWLU). The TWLU contains a radio transmitter and receiver that creates a Local Area Network (LAN) between the airplane and a ground-based network. This allows the databases to be updated wirelessly while the airplane is parked at the gate.

MAINTENANCE INFORMATION SYSTEM

The 777 EFB is one example of an aircraft information system. That particular system is used by the flight crew. However, the same kind of technology is also used by maintenance personnel. Flight crews need to access the EFB's information while in flight, but maintenance personnel need to access different kinds of information while the aircraft is on the ground.

Maintenance Information Systems work along similar lines as a flight deck information system (such as the EFB), but the information being stored is different. Instead of maps and charts, maintenance crews use maintenance manuals, illustrated parts catalogs, wiring diagram manuals, service bulletins, and other technical data. Maintenance Information Systems provide access to these publications electronically, just as Flight Deck Information Systems electronically provide the flight crew with access to maps and charts.

An advantage of using an electronic format, in addition to the space and weight savings it provides, is the ability to quickly locate the desired information. Instead of leafing through a large paper maintenance manual, for example, a technician using a maintenance information system uses hyperlinks which allow easy navigation within the system.

A typical method of achieving this is to use a menu containing links to each of the ATA 100 chapters within the manual. Within each chapter, the table of contents contains links that will quickly access a particular page. This allows the technician to locate the desired page with a few "clicks."

Aircraft manufacturers, which previously published their manuals only on paper or microfilm/microfiche, now offer their manuals in electronic format. Laptop computers are very well suited for storing and retrieving maintenance information in the aviation maintenance environment. Laptops can be taken practically anywhere on the aircraft that the technician might need to go while performing maintenance. For this reason, laptop computers are widely used to access maintenance information (*Figure 21-2*).

PASSENGER CABIN INFORMATION SYSTEM

Flight attendants need to be able to access information to do their jobs, just as pilots and maintenance technicians do. In paper form, a typical flight attendant's manual weighs about five pounds (2.3 kg). Each flight attendant is required to carry the manual while serving as cabin



Figure 21-2. Laptop computers are used to access maintenance information.

crew. The flight attendant's manual contains vital information for cabin crew members, such as checklists, procedures, security information, and information about safety devices and systems on the aircraft. Passenger cabin information systems provide the same advantages to the cabin crew that Flight Deck Information Systems provide to the flight crew, and that maintenance information systems provide to the maintenance crew.

The use of the digital electronics allows flight attendants to carry a small device, such as a tablet, that contains the manual they are required to have on hand (*Figure 21-3*).

The device weighs much less than a paper manual. This saves the airline money, as even small reductions in weight affect fuel economy. The airline also saves money on the cost of printing revisions to the flight attendant's manual.

MISCELLANEOUS INFORMATION SYSTEMS

New information systems are introduced from time to time, and improvements to current information systems are constantly being developed. Whether the information is used by the flight crew, or by maintenance personnel, or by any other user, the basics of an information system are the same. Each system contains digital memory to store the desired information. There is a set of user controls, which allow the user to locate and retrieve the information. There must also be a means of displaying the information to the user. And the system must have a means to update the information, to ensure that the most up-to-date version is being used. These will be the core



Figure 21-3. Passenger Cabin Information can be accessed with a computer tablet.

Question: 21-1

Examples are aircraft general information systems include: flight deck information systems, maintenance information systems, and _____ information systems.

Question: 21-4

Maintenance crews use maintenance manuals, illustrated parts catalogs, wiring diagram manuals, service bulletins, and other technical data stored in the _____.

Question: 21-2

An electronic flight bag (EFB) is part of the _____ information system.

Question: 21-5

A _____ computer can be used to access electronically stored maintenance information of the aircraft.

Question: 21-3

On a 777 equipped with the EFB, the system displays information on _____ that are installed on the flight deck.

Question: 21-6

Passenger cabin information systems provide access to an electronic version of the flight attendants' _____.

ANSWERS

Answer: 21-1
passenger cabin.

Answer: 21-4
maintenance information system.

Answer: 21-2
flight deck.

Answer: 21-5
laptop.

Answer: 21-3
the two Display Units (DUs).

Answer: 21-6
manual.

AC	/	Alternating Current
ACARS	/	Aircraft Communications Addressing and Reporting System
ACU	/	Alternator Control Unit
ADC	/	Air Data Computer
ADF	/	Automatic Direction Finder
AFCS	/	Automatic Flight Control System
AOA	/	Angle Of Attack
APB	/	Auxiliary Power Breaker
ATC	/	Air Traffic Control
ATE	/	Automatic Test Equipment
BPCU	/	Bus Power Control Unit
BITE	/	Built In Test Equipment
BTB	/	Bus Tie Breakers
CDU	/	Control Display Unit
CMC	/	Central Maintenance Computer
CRT	/	Cathode Ray Tube
CSD	/	Constant Speed Drive
CT	/	Current Transformer
DC	/	Direct Current
DG	/	Directional Gyro
DME	/	Distance Measuring Equipment
EASA	/	European Aviation Safety Administration
EICAS	/	Engine Indication And Crew Alerting System
ELMS	/	Electrical Load Management System
ELT	/	Emergency Locator Transmitter
EPR	/	Engine Pressure Ratio
FAA	/	Federal Aviation Administration
FMS	/	Flight Management System
GB	/	Generator Breaker
GCB	/	Generator Control Breaker
GCU	/	Generator Control Unit
GPS	/	Global Positioning System
HF	/	High Frequency
HMDGS	/	Hydraulic Motor Driven Generators
HOT	/	Hold Over Time
IDG	/	Integrated Drive Generator
IFR	/	Instrument Flight Rules
ILS	/	Instrument Landing System
INS	/	Inertial Navigation System
IT	/	Internal Tolerances
KVA	/	Kilovolt Amp
LCD	/	Liquid Chrystal Display
LOX	/	Liquid Oxygen
LRU	/	Line Replaceable Unit
LSK	/	Line Select Key
NAV/COM	/	Navigation / Communication

OBOGS	/	On board Oxygen Generation System
PTU	/	Power Transfer Unit
RAT	/	Ram Air Turbine
RLG	/	Ring Laser Gyros
RNAV	/	Area Navigation
SATCOM	/	Satellite Communication System
SCR	/	Silicon Controlled Rectifier
SSB	/	Split Systems Breaker
SSTD	/	Spread Spectrum Time Domain Reflectometer
TCAS	/	Traffic Collision Avoidance System
TDR	/	Time Domain Reflectometer
TR	/	Transformer Rectifier
VHF	/	Very High Frequency
VME	/	Versa Model Eurocard
VRLA	/	Valve Regulated Lead-Acid Battery
VXI	/	VME Expansion for Instrumentation

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