A torque tube is a hollow metal tube used to transmit a torsional, or twisting, force between the actuating control and the device being controlled. Large aeroplane control systems often use torque tubes between an electric or hydraulic motor and a jackscrew to actuate flaps, slats, and other control surfaces. Figure 9.9 shows an example of usage of a torque tube in a flight controlsystems.

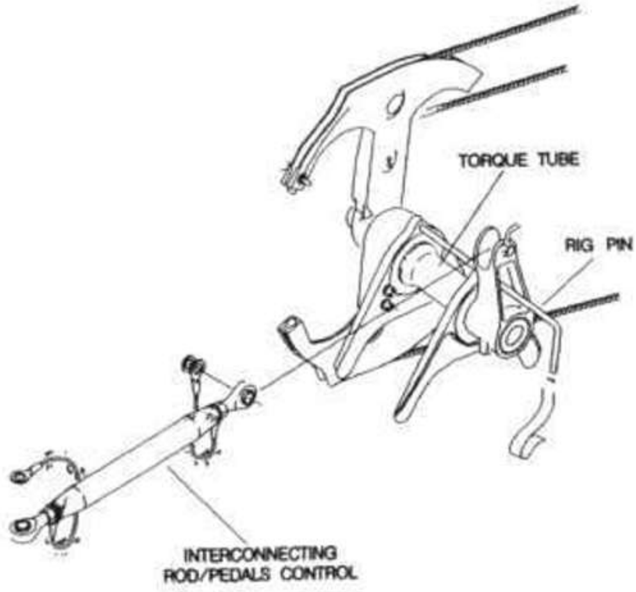


Fig. 9.9 Torque Tube Aerodynamically Controlled Systems

The control tabs are controlled by the control wheels in the flightdeck so that as one tab moves up, the opposite tab moves down. The ailerons are thus operated aerodynamically. When the control tabs are deflected, aerodynamic forces on the tabs move the ailerons in the oppositedirection.

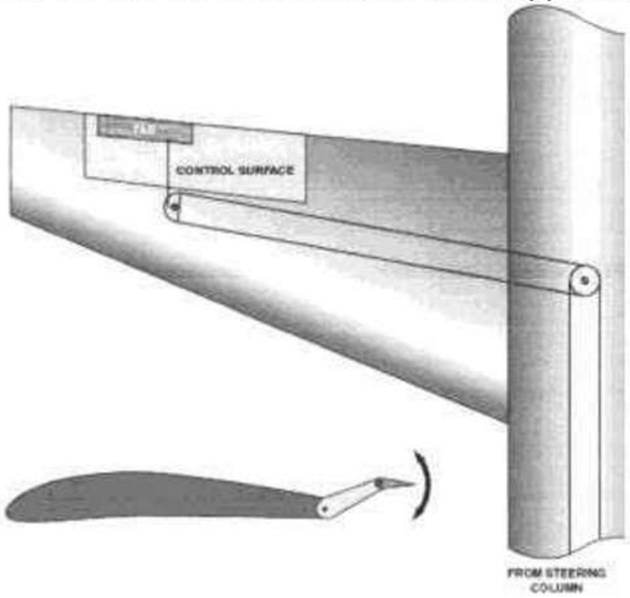


Fig. 9.10 aerodynamically controlled system

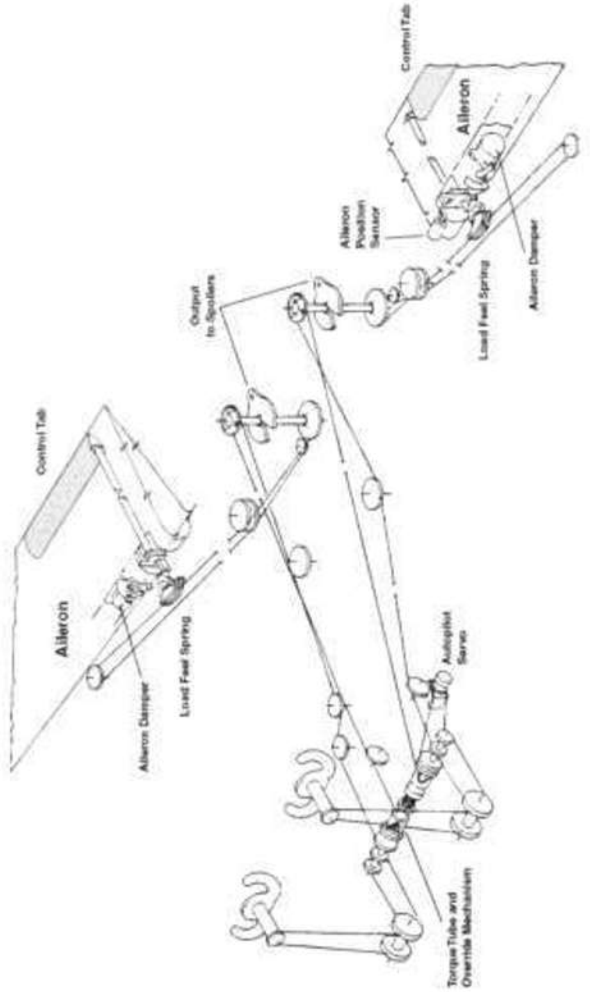


Fig. 9.11 Aerodynamically controlled aileron system Hydraulically Assisted Control System

As aircraft increase in size and weight, their controls become more difficult to operate and systems must be used to aid thepilot.

The power boosted control system is similar in principle to power steering in an automobile. A hydraulic actuator is in parallel with the mechanical operation of the controls, and in addition to moving the control surface, the normal control movement by the pilot also moves a control valve that directs hydraulic fluid to the actuator that moves thesurface.

Hydraulically Actuated System with Direct Cable Backup

On modern large aircraft, the previously described system is replaced with a more advanced architecture. In normal operation, the direct connection between the steering column and the control surface is in a disconnected mode, and the input coming from the flightdeck is only directed to the actuators control valve. In case of a hydraulic power failure, the hydraulic actuator is bypassed and the input force coming from the steering column is directed directly to the controlsurface.

It is obvious, that the necessary force to move the control surface is much higher in this mode.

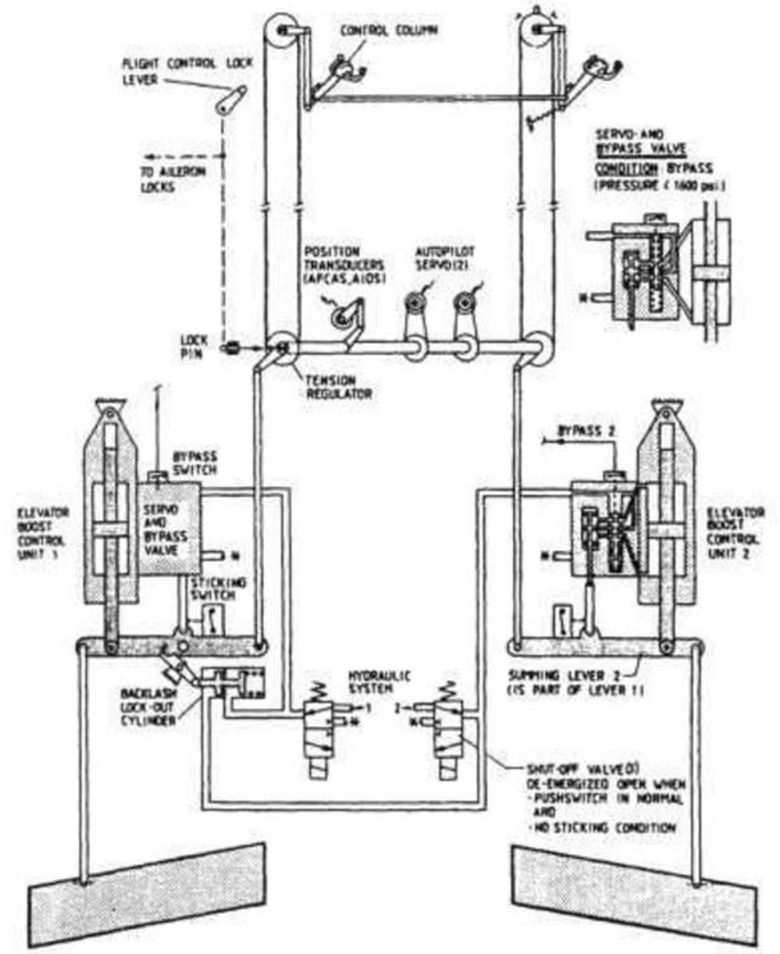


Fig. 9.12 Hydraulically Actuated System with Direct Cable Backup

Hydraulic Actuating System with Control Tab Backup

In the event of a hydraulic system failure, the control forces are too great for the pilot to manually move the surfaces, so, they are controlledwithservotabs.Inthemanualmodeofoperation,the

flight control column moves the tab on the control surface, and the aerodynamic forces caused by the deflected tab move the main control surface.

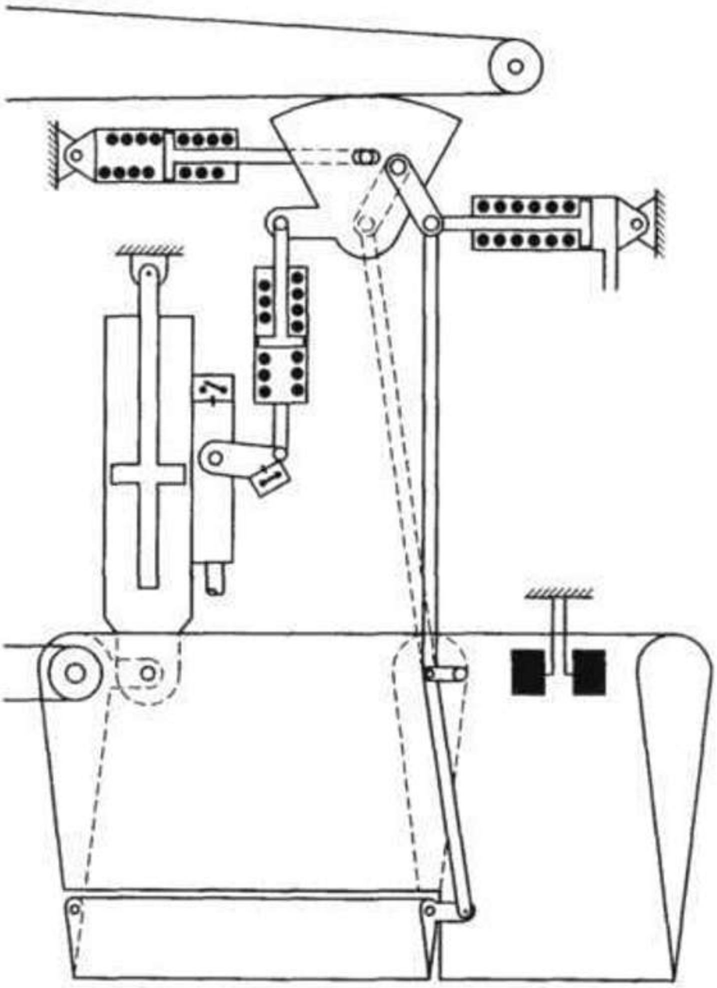


Fig. 9.13 Tab Backup

Hydraulic Power Operated Systems

Boosted flight control systems are found in aeroplanes lighter than approximately 60 tons, in heavier aeroplanes the servo tabs would be too big and need too much force for manual operation of the system. Another problem with a power-boosted control system is that during transonic flight shock waves form on the control surfaces and cause control surface buffeting, and this force is fed back into the control system. To prevent these forces reaching the pilot, many aeroplanes that fly in this region of airspeed use a power-operated irreversible control system. The flight controls in the cockpit actuate control valves which direct hydraulic fluid to control surface actuators.

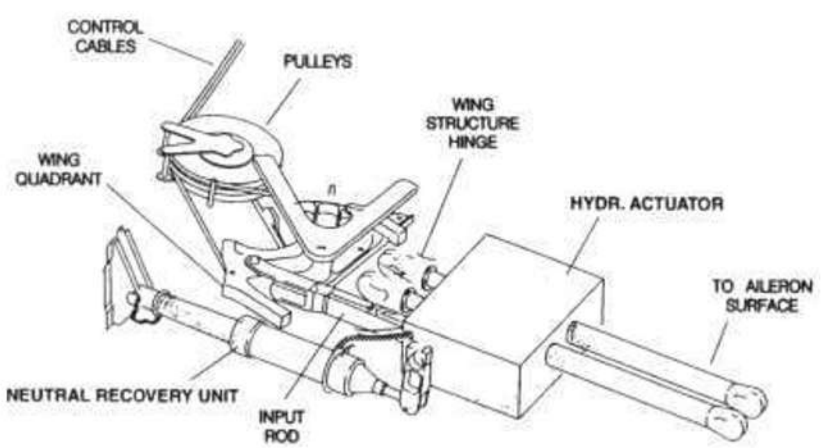


Fig. 9.14 Hydraulic Actuator with Control Cables Follow-up Control

Some kind of follow-up system is used to close the control valve of a servo control unit when it has reached the desired position. Figure 9.15 shows a typical mechanical follow-up system.

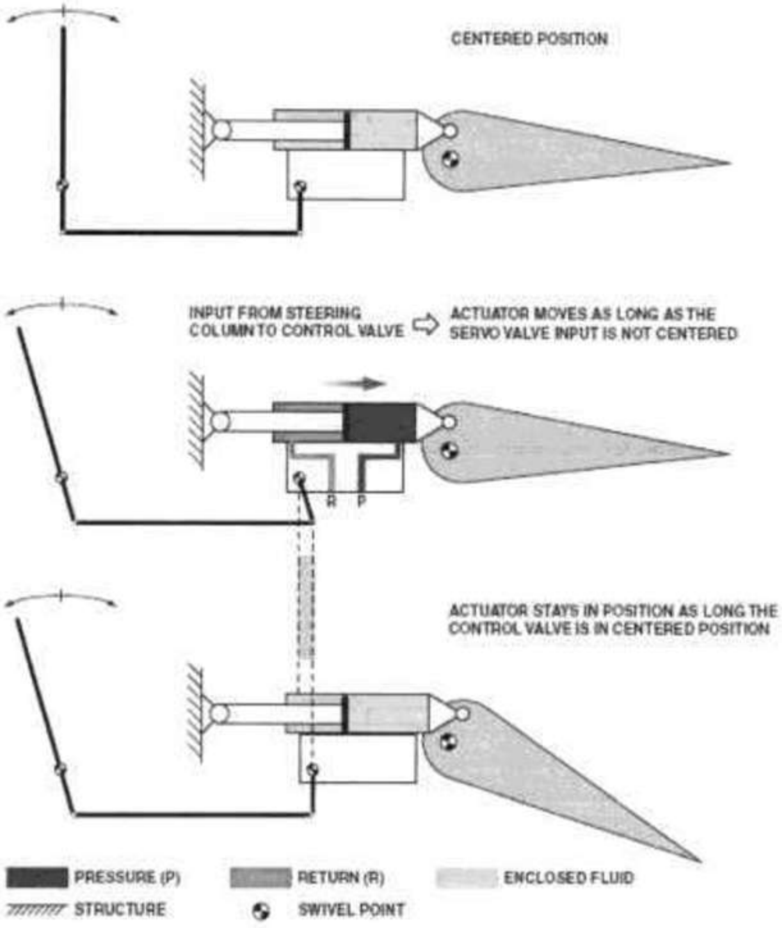


Fig. 9.15 Mechanical Follow-up Control Operation of a mechanical controlled actuator

Normal operation with the servo control pressurized. Refer to Figure 9.16

The moving servo control body is slaved to the mechanical input signal via input lever (13), travel of which is limited by stops (1). The lever actuates the control valve (8).
In the event of overpressure in one of the chambers, check valve
(7) holds supply pressure against one of the faces of pressure controlled relief valves (5).
Chamber overpressure is then applied to the other face of one valve (5) and causes it to move. The two chambers are then connected to the return line, thus ensuring anti-cavitation and over pressure limitation functions.

Operation in the event of control valve jamming

When the input force is higher than preloaded spring (4) force, roller (9) moves out of its cam.

This depresses microswitch (3) which transmits a warning signal to the crew via the jamming detection system. The crew must then shut off the power supply from the hydraulic system involved.

The warning is provided to prevent excessive fatigue loads on the structural attachments of the servo control, if the failure has remained hidden for several flights.

Operation on the ground, with the servo control depressurized

In the event of the control surface being subjected to external forces (e.g. gust forces), pressure increases in one of the chambers, thus causing one pressure controlled relief valve to open.

This chamber then empties into the return line via a small orifice

(6) which provides a ground gust damping function, for gust speeds up to 80 kts.

Ground testdevice

The jamming detection device can be tested by means of stop (2) which is extended when there is no pressure, and limits the travel of the controlvalve.

Operation of input lever (13) over a sufficient amount of travel then depresses microswitch (3) and allows it to be checked for correct operation.

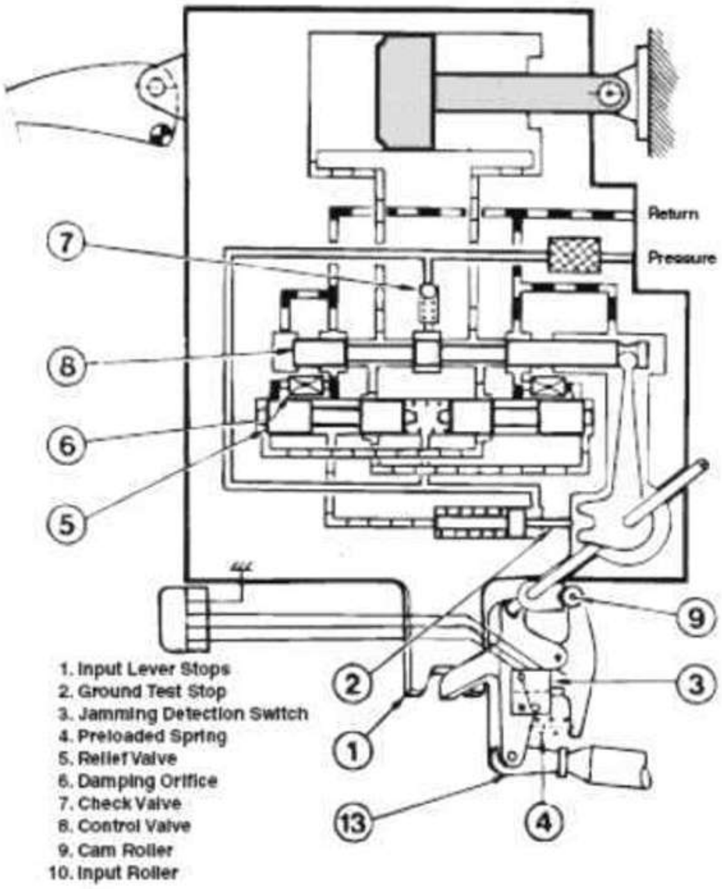


Fig. 9.16 Servo Control Trim Control

Transport aeroplanes are usually trimmable in all three axes.

Roll and Yaw Trimming Cable Controlled Flight Controls

Trim tabs are normally used for roll and yaw trimming on cable controlled aeroplanes what means, you will find them on the rudder and ailerons. They are controllable from the cockpit and allow the pilot to deflect the control surface just a small amount. This allows the aeroplane to be adjusted to fly straight and level with hands and feet off of the controls. Once a trim tab is adjusted, it maintains a fixed relationship with the control surface as it is moved as shown in Figure 9.18.

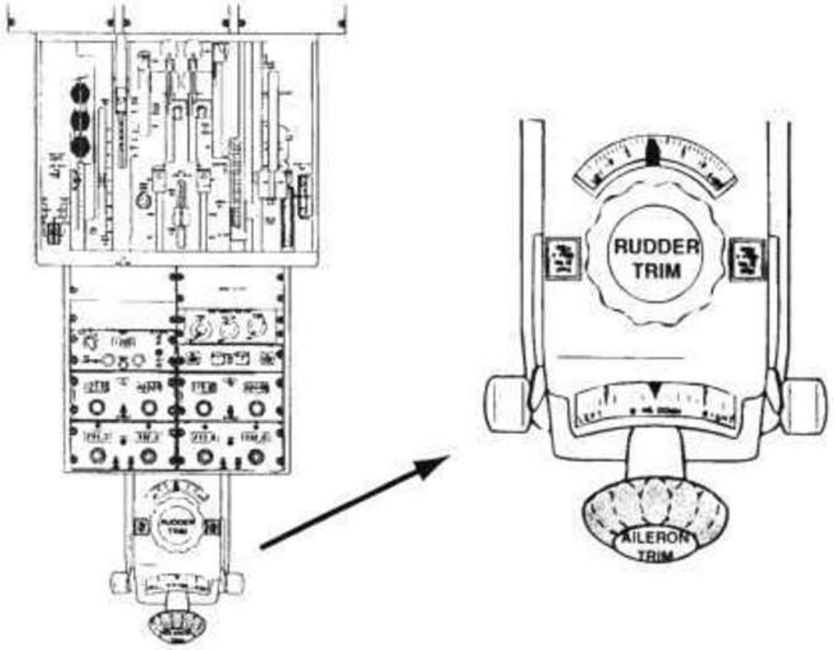
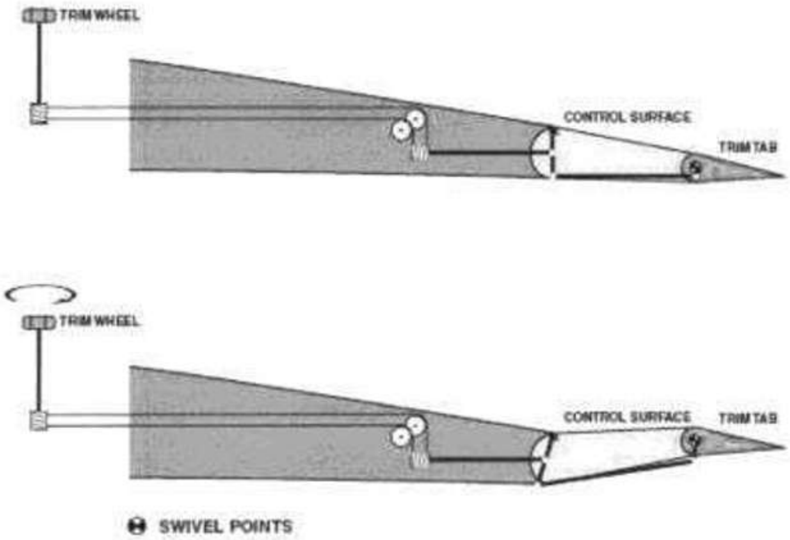


Fig. 9.17 Trim Tab Control

Boosted or hydraulically operated aileron and rudder systems are usually trimmed by means of moving the artificial feel and centring unit out of its centre position. This is done by a manually or electrical motor operated jack-screw. Fly-by-wire operated systems do not need special provision for trimming the ailerons. If an imbalance occurs during flight, the roll control computers send a re-positioning signal to



the aileron servo control units to achieve levelled flight again.

Fig. 9.18 Trim system

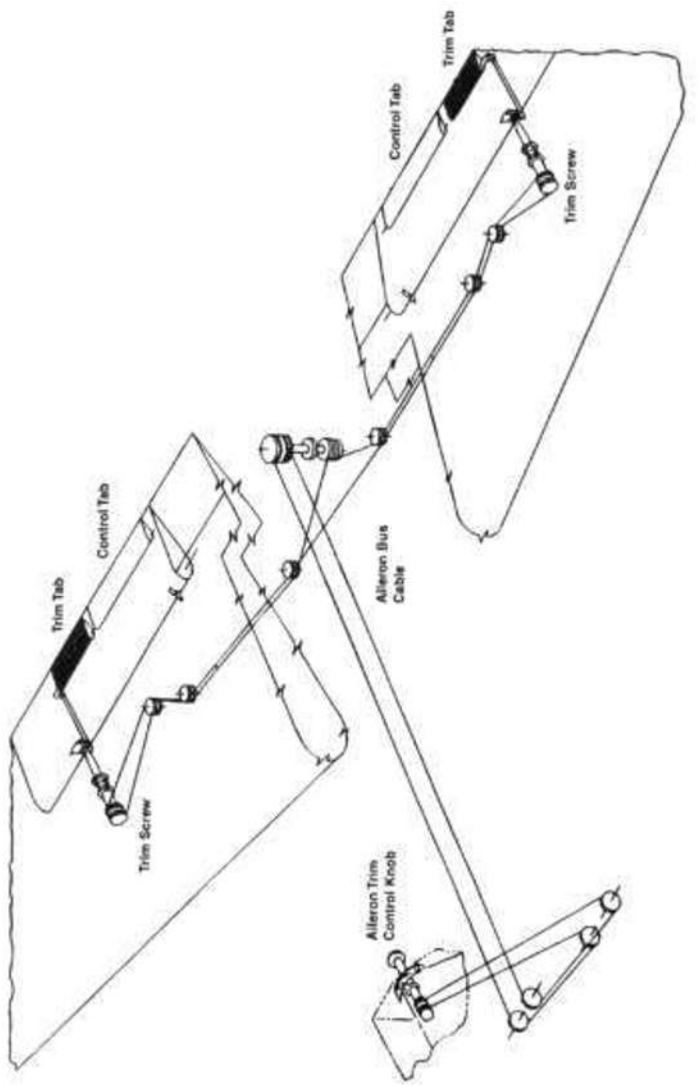


Fig. 9.19 Aileron Trim System on MD-80

Pitch Trimming

Larger transport aeroplanes are trimmed longitudinally by adjusting the position of the leading edge of the horizontal stabilizer. These stabilizers pivot about the rear spar, and a jack- screw controlled from the cockpit raises or lowers the leading edge. Raising the leading edge gives the aeroplane nose-down trim, and lowering the leading edge trims the aeroplane in a nose- up trim. The system layout is basically the same on most transport aeroplanes. Figure 9.21 shows the most common system lay-out. The motors can be electrically or hydraulically driven. For redundancy, different hydraulic systems supply the THS actuator.

Some other systems provide electrical actuators for backup operation

The trimmable horizontal stabiliser (THS) can be moved either manually or automatically.

Manual control is achieved from the cockpit by the different manual control wheels, switches or suitcase handles. Automatic control is achieved with signals from the autopilot or Flight Management systems which send signals direct to the THS- actuator.
During manual movement of the THS an audible signal sounds in the cockpit, so, the pilots can perceive motion of the system. This audible signal is also used to warn the pilots in case of dangerous THS runaway.

During manual movement of the THS an audible signal sounds in the cockpit, so, the pilots can perceive motion of the system. This audible signal is also used to warn the pilots in case of dangerous THS runaway.

Automatic Pitch Trim

In the auto flight mode the command signals from the autopilot are sent to the flight control computers where auto trim signals are calculated and sent to the THS actuator. On large aircraft, other systems are installed for pitch and centre of gravity control. These aircraft are equipped with a horizontal stabilizer designed as a fuel tank. This feature makes it possible to transfer a calculated amount of fuel into this tank to keep the centre of gravity in a safe and economic range during the flight. Therefore, extensive THS pitch angles can be avoided.

General

The pilots use the aileron control wheels to move the ailerons and flight spoilers. The autopilot, when engaged, automatically controls them.

Manual Operation - Control Wheels

The flight crew uses two conventional control wheels to control the roll attitude of the airplane. A transfer mechanism on the first officer side, supplies a mechanical link between the control wheels. If one control wheel cannot move, the other continues to control.

The control wheel gives mechanical input to the power control unit (PCU) through cables and linkages. Hydraulic pressure goes to the PCU and makes the housing move. The aileron PCU movement mechanically moves the aileron through the wing cables and thequadrant.

When the control wheel moves, it also gives a mechanical input to the roll control wheel steering (CWS) force transducer and the control wheel position sensor. The roll CWS force transducer sends signals to the flight control computers (FCC) proportional to control wheel turning forces. The control wheel position sensor sends signals to the flight data acquisition unit (FDAU) for control wheel position.
See the digital flight control system section for more information about the FCCs and the roll CWS force transducer. (SECTION 22-11)
See the flight data recorder section for more information about the aileron control wheel position sensor and FDAU. (SECTION31-31)

Manual Operation - Control Wheels – Manual Reversion

During manual reversion, the control wheel gives mechanical input to the power control unit (PCU) through cables and linkages. Mechanical stops in the PCU make the housing move. The aileron PCU movement mechanically moves the aileron through the wing cables and the quadrant.

Manual Operation - Aileron Trim

The aileron trim switches and the trim actuator let the flight crew trim out unwanted control wheel forces. When the pilots move the aileron trim switches on the aileron/rudder trim panel, the switches send a signal to the aileron trim actuator and FDAU. The trim actuator moves the feel and centering unit. This movement goes to the aileron PCU. Aileron PCU movement mechanically moves the aileron through the wing cables and quadrant.
During aileron trim, the control wheels also move to supply an indication on the top of the control wheel column.

Autopilot Operation

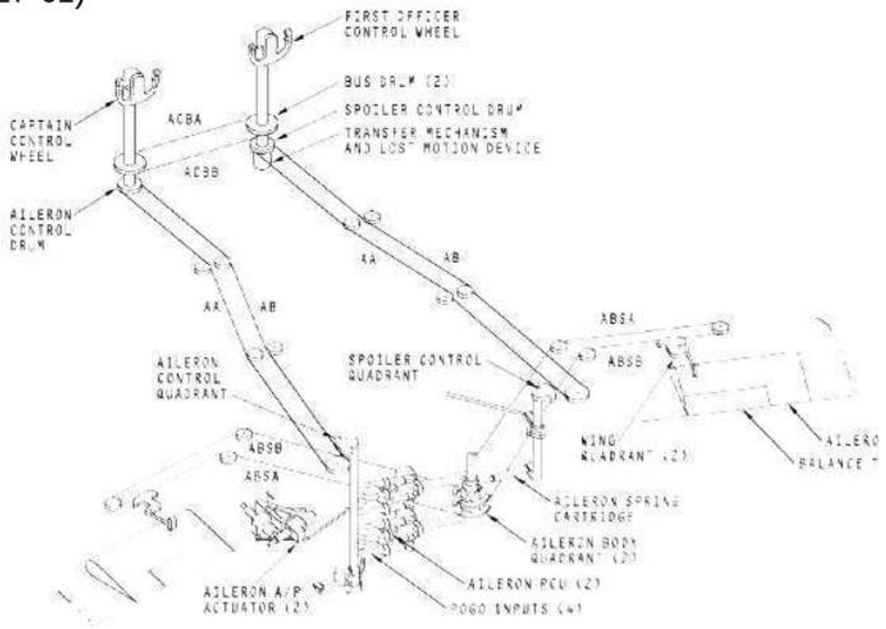
When engaged, the flight control computers control the aileron autopilot actuators. The actuators give mechanical input to the feel and centering unit and aileron position sensor. Movement of the feel and centering unit goes to the aileron PCU and signal from the aileron position sensor go to the FCCs. The

aileron PCU movement moves the aileron through the wing cables and the quadrant. The quadrant also gives mechanical input to the aileron position transmitter. The transmitter sends position signals to the(FDAU).
See the digital flight control system section for more information on the aileron autopilot operation and aileron position sensor. (SECTION22-11)

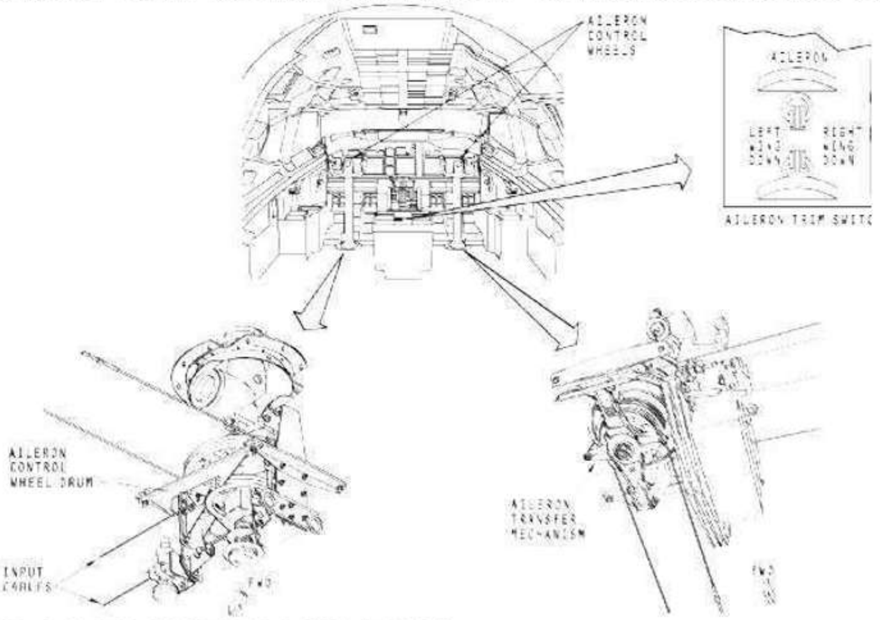
See the dig ital flight recorder system section for more information about the aileron position transmitter. (SECTION31-31)

Flight Spoilers
The flight spoilers also supply roll control. When the aileron PCU moves, it supplies an input to the flight spoilersystem.

See the flight spoiler control system section for more information on the spoiler and speed brake control. (SECTION 27-61)



AILERON AND AILERON TRIM CONTROL SYSTEM• GENERAL DESCRIPTION 2



AILERON AND AILERON TRIM CONTROL SYSTEM• COMPONENT LOCATIONS 1

Fig. 9.20 Different Manual Trim Control Inputs

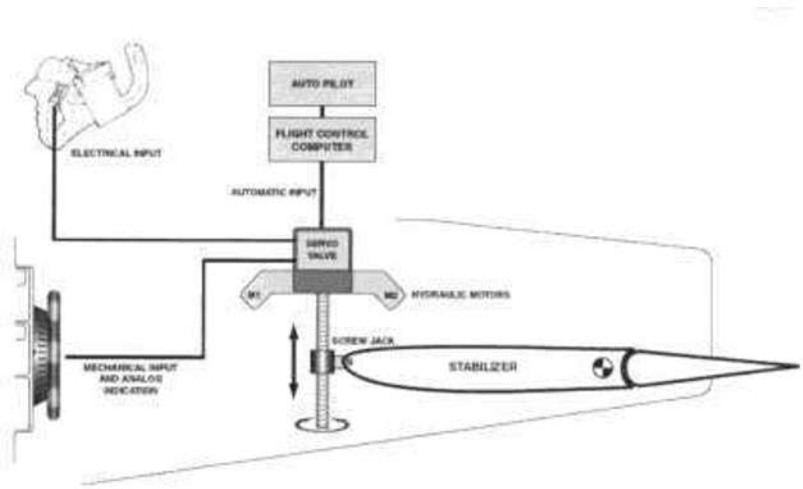


Fig. 9.21 Pitch Trim System Layout

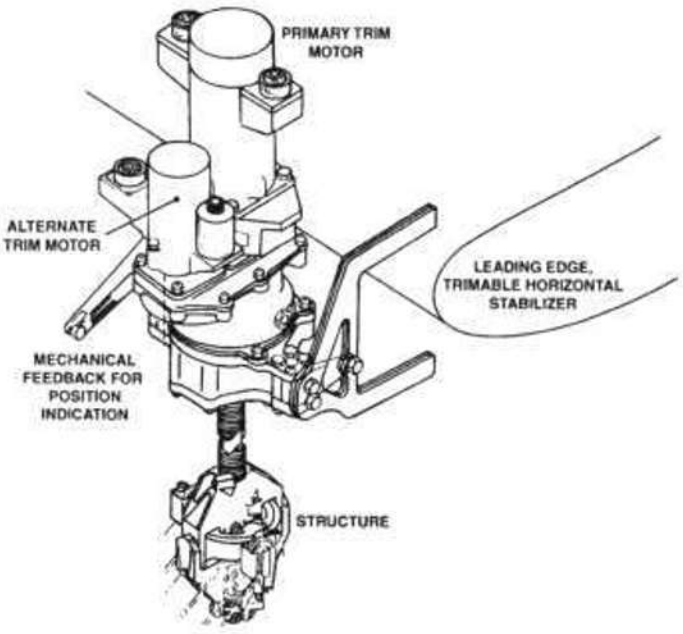
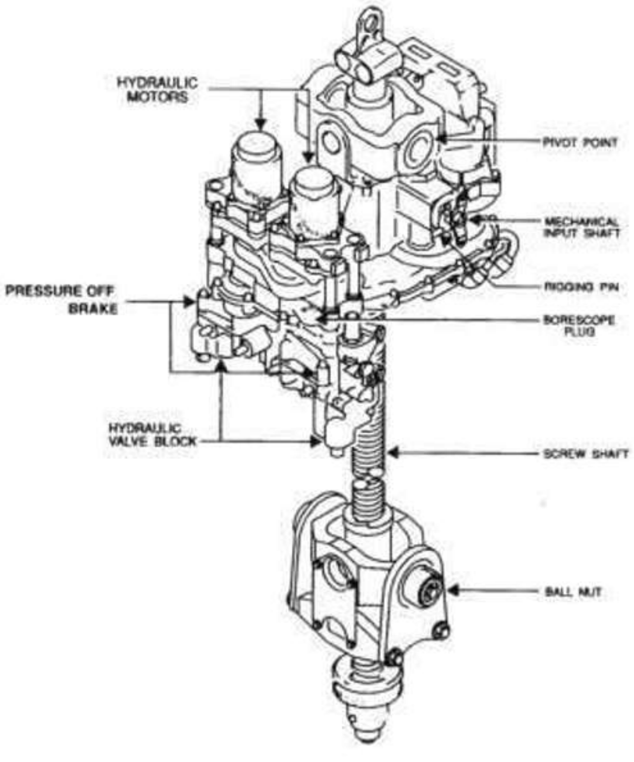


Fig. 9.22 Electrical Trim Actuator with Electric Backup



Actuator

Fig 9.23 Hydraulically driven Trim Actuator

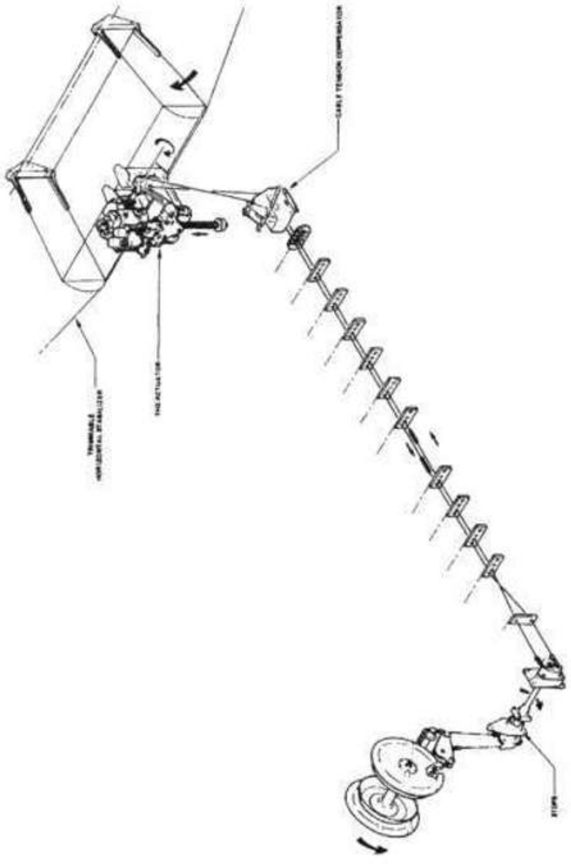


Fig. 9.24 Cable run of a mechanical controlled Hydraulic Actuator

Versine Signal

The vertical component of lift reduces as the angle of bank increases. Since it is the vertical component of lift which opposes mass, then, when the aircraft banks the vertical component of lift no longer balances mass and the aircraft would lose height. To counter this effect the angle of attack of the aircraft must be increased to make the vertical component equal to the mass.

In order properly to coordinate the turn entry and turn exit with rudder, the roll command signal is differentiated, and applied to the rudder servo, as is generally conventional practice. To prevent loss of altitude during banked turns, a lift compensation signal is supplied to the pitch servo. This signal is derived from the bank angle signal from a vertical gyro and is generally proportional to the versine (1 - cosine) of the bank angle.

Since, with the autopilot in the airspeed or Mach hold mode, the pitch up produced by the lift compensation signal tends to change the airspeed or Mach speed which the autopilot is trying to maintain, this signal is conflicting and not desired and is therefore inhibited in these modes.

Active Load Control

On the Airbus A320 and similar aircraft, the Active Load Control (also known as load alleviation function (LAF)) is accomplished by the electrical flight control system (EFCS). The LAF is implemented in the elevator and aileron computer (ELAC) and the spoiler elevator computer (SEC). The control surfaces used are both ailerons as well as spoilers 4 and 5 (i.e. the outboard pair on both sides) for up gusts.

There are four specific accelerometers that are installed in the

forward fuselage station to provide the electrical flight control computers with vertical acceleration values. These sense the up gust and deploy the spoilers to smooth out the normal result of an up gust of wind as described in the before mentioned example.
Four hydraulic accumulators are installed to provide the extra hydraulic flow needed to achieve the surface rates and duration of movement required for load alleviation as illustrated in Figure 9.25.

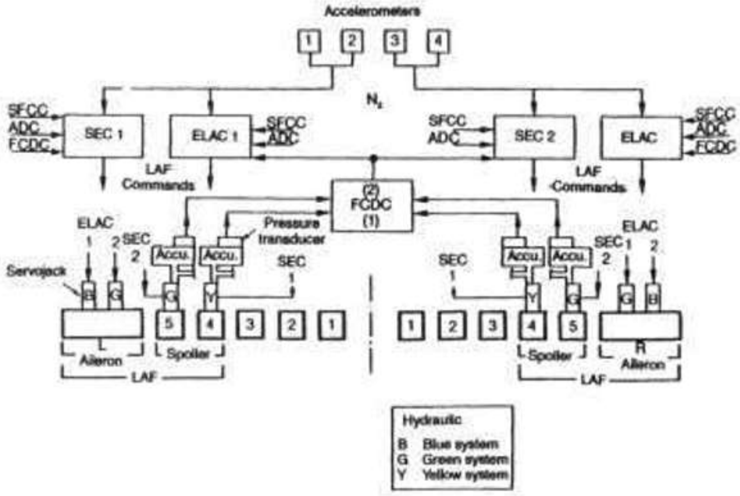


Fig. 9.25 Load Alleviation Function High Lift Devices

Trailing Edge Flaps

Flaps which only increase the camber have a fixed hinge point and are generally moved by hydraulic actuators. The trailing edge of this type of flap will move downwards immediately if a "flaps extend" selection is made on the flight deck.

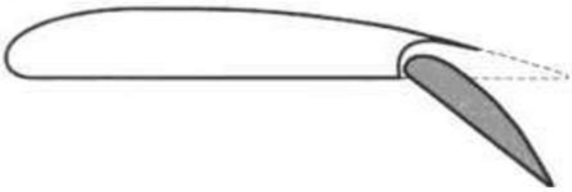


Fig. 9.26 A plain flap

Flaps which increase both the surface area and the camber have rollers and are installed in specially formed tracks.

After a "flaps extend" selection the flaps will first move backwards via the rollers and the tracks (increase of the wing area) followed by a trailing edge that moves down (camber increase).

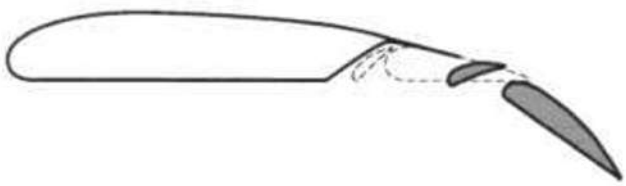


Fig. 9.27 A split Fowler flap

To perform this combined movement, a system of driving axles and screw spindles or rotary actuators is used. This system is usually driven by a hydraulic motor or by an electric motor depending on the type and size of aircraft. A screw type system is often used to prevent flap loads feeding back to the control system. Such flap system lay-outs are similar to the slat system lay-out. For some aircraft, both types of actuation are used. The flaps are operated with the flap handle on the flight deck. The flap handle can be set to different positions depending on the flight phase. This enables the pilot to select the most efficient flap position. See the typical system layout as shown in Figure9.29.

Different maximum speeds apply for each flap position setting. This means it is not allowed to extend the flaps in every flight phase. Structural damage could occur if the speed is too high. Therefore, the manufacturer defines the maximum speeds for each setting. These speed tables are usually printed in speed booklets. On modern aircraft, where most of the systems are computer controlled and monitored, limitation envelopes are integrated in the software. If the pilot tries to extend the high lift devices while the aircraft cruises with a speed higher than the allowed speed for flaps extension, the computer will not allow the extension and inhibits the operation accompanied by a warning. See Table 7.1 for an example of maximum allowedspeeds.

	Slat/Flap in °	Max IAS(knots)b
SLT/FLP	15/0	245
SLT/FLP	15/15	210
SLT/FLP	20/20	195
SLT/FLP Maximum Speed for slats/flap Indicated AirSpeed,	30/40 s extended	180

Table 7.1 Maximum speeds for slats/flaps VFEa

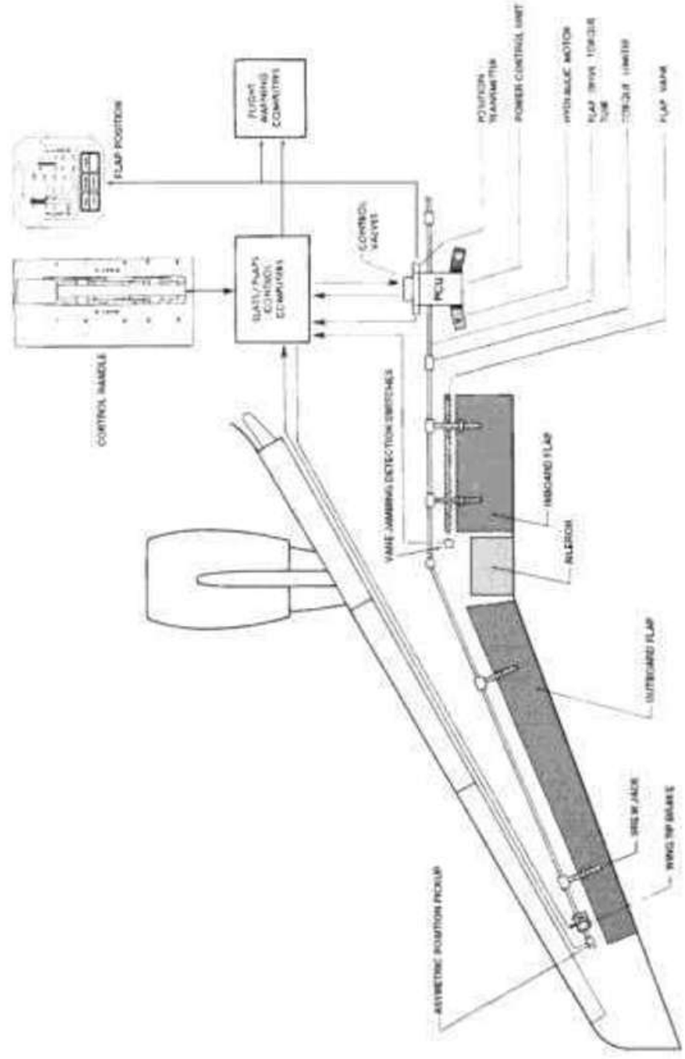
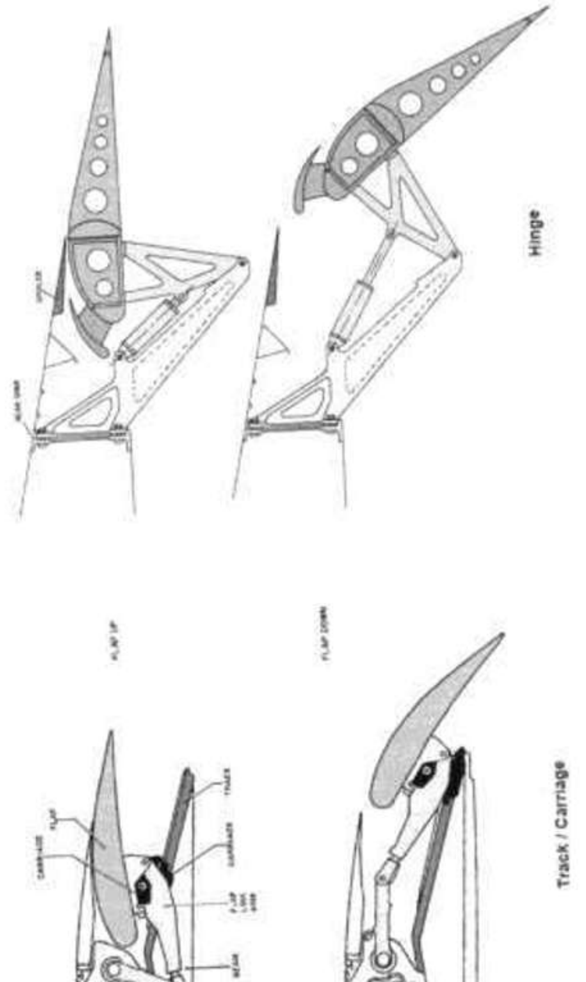


Fig. 9.28 Flap System Layout



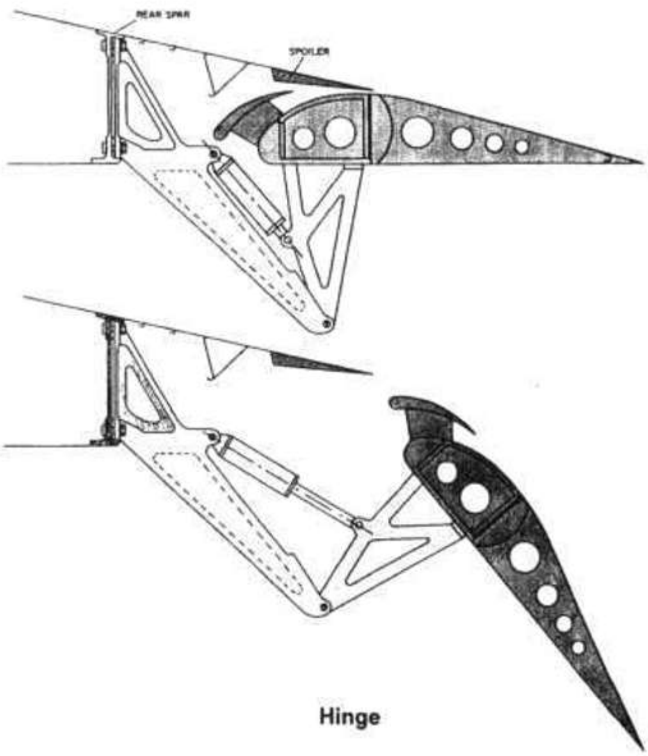


Fig. 9.29 Flaps Configurations

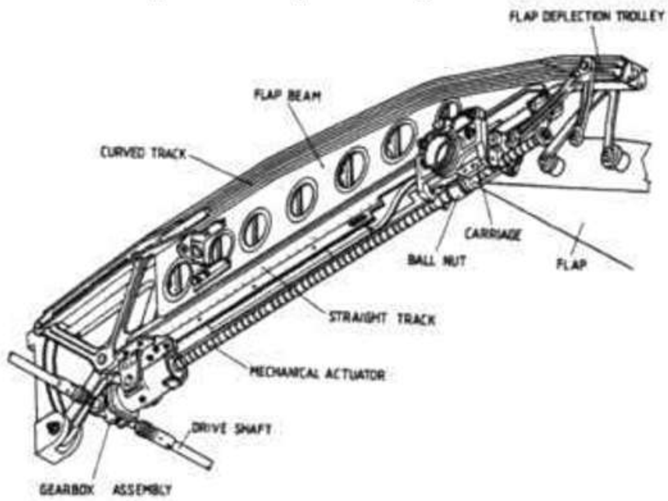


Fig. 9.30 Flap Track, Screw Jack Operated Leading Edge Flaps

Leading edge flaps, also named "Krueger Flaps", are installed below the forward wing edge as shown in Figure 9.31.

The flaps can have two positions, extended or retracted. The command for these positions comes from the flap handle or via the flap computer. Such leading edge flaps are driven hydraulically, pneumatically or electrically.

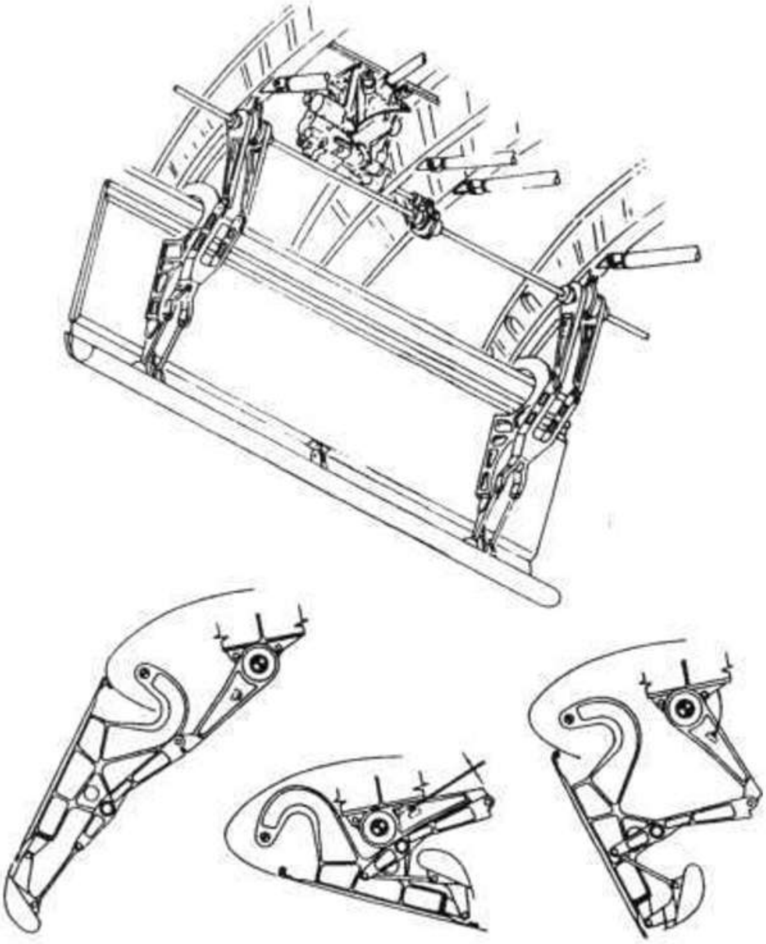


Fig. 9.31 Leading Edge Flaps Slats

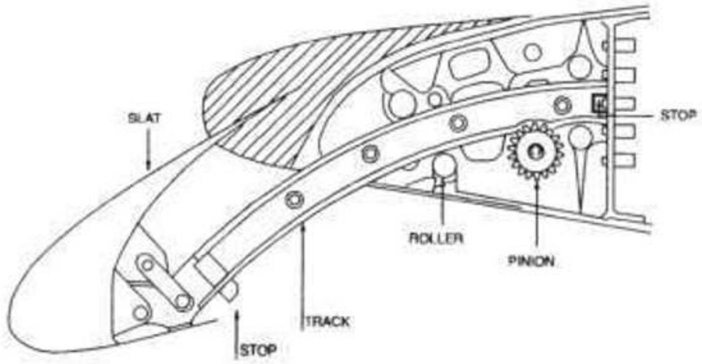
Together with the flaps, the slats are used for lift augmentation. The slats are installed on tracks which are attached to the leading edge of each wing. The operation of the slats is usually mechanical; by cables - hydro-mechanical or pneumatic. A

hydraulic or pneumatic power control unit (PCU) moves the mechanical transmission system which

operates the slats on each wing.

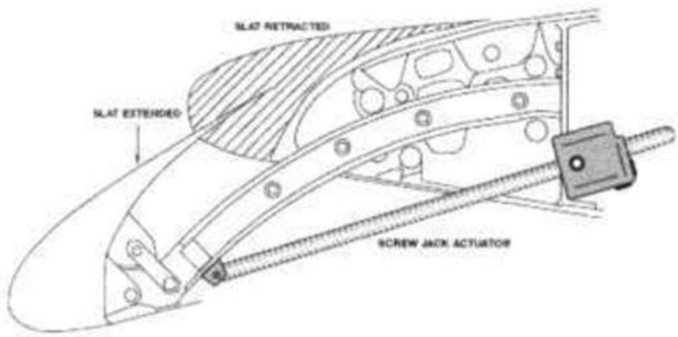
Fig. 9.32 Slats

Figure 9.33 shows the mechanism of a typical slat track. Generally the extension of the slats and the flaps must be done at the same time. Therefore, there is only one control handle in the cockpit to operate the flaps and slats. An indication system on the cockpit indicates the position of the slats, similar as the



flap system.

Fig. 9.33 Pinion Driven Slat



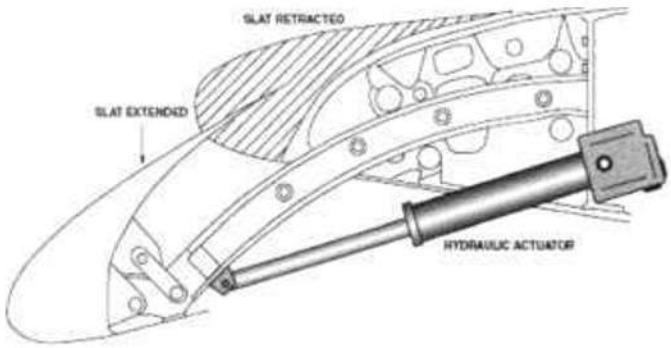


Fig. 9.34 Screw Jack Driven Slat

Fig. 9.35 Hydraulic Actuator Driven Slat

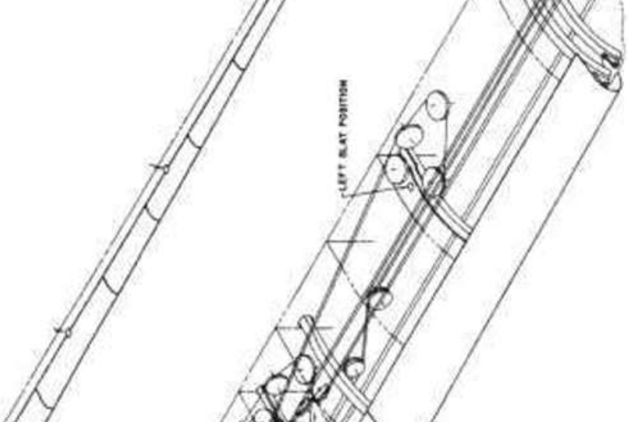


Fig. 9.36 Hydraulically Operated Slat Cable System

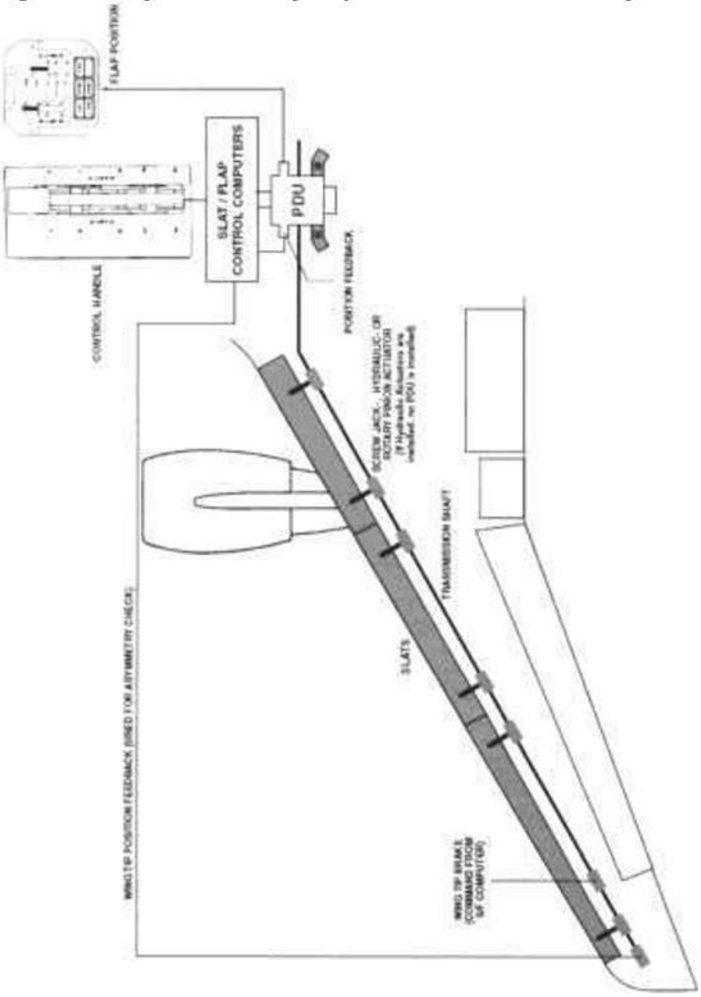


Fig. 9.37 Hydraulically Operated Slat Screw Jack System Flap/Slat Position Indication

Different methods and types of indicators are in use to show the actual position of the flaps to the flight crew. The signal transmission can be done either mechanically by steel cables or electrically by using a position pickup unit and an electrical indicator. The principle of slat/flap position transmission is usually similar.

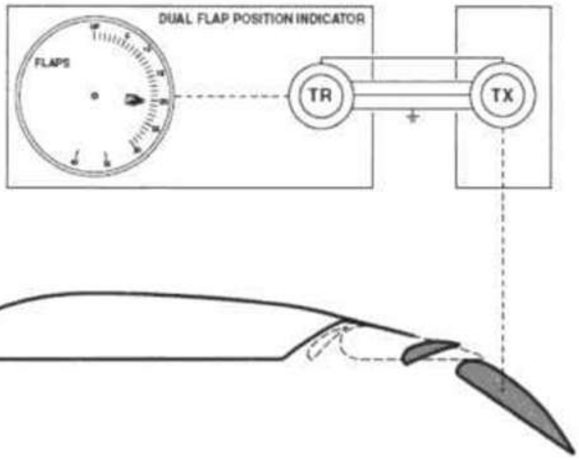


Fig. 9.38 Electrical Position Transmission

As you know, modern aircraft are equipped with displays instead of analogue indicators. Figure 9.39 shows a typical slat/flap indication.

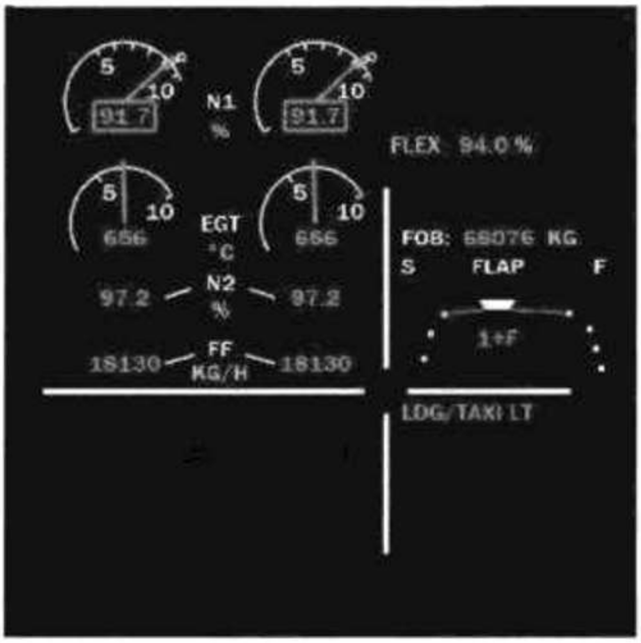


Fig. 9.39 Modern Indication Display Position and Asymmetry Monitoring

After a failure in the driving mechanism (spindle fracture, icing etc.), a difference in flap position might occur. This can cause a dangerous roll moment. To prevent this, the flaps can be connected to each other by means of a heavy steel cable (synchronization cable), especially in aircraft where the flaps are moved by means of hydraulic cylinders. For aircraft which have a driving axle system, asymmetry is measured electrically.

Monitoring of a Power Transmission Systems

See Figure 9.40

To monitor the power transmission system, the computers compare the commanded setting with the actual position data. The computers receive the position datafrom:

- the two APPUs (Asymmetry Position Pick-offUnit)
- the FPPU (Feedback Position Pick-offUnits)
- the Valve Blocks
- the flap-attachment failure detectionsensors

An APPU is installed at the end of the transmission system in each wing. The two synchro transmitters in each APPU send position data to each computer. The control valve of each valve block has a Linear Variable Differential Transducer (LVDT). The LVDT sends valve position data to its related computer. The position of the valve is directly related to the hydraulic pressure available at the valveblock.

Computers monitor the power transmission system continuously for failures. These are usually:
- asymmetry (a position difference between the twoAPPUs)
- runaway (a position difference between the APPUs and the FPPU)
- uncommanded movement (a movement in the wrong direction, or movement away from the last setposition)
- overspeed (the faster movement of one or morePPUs)
- systemjam
- control valveposition

As soon as a certain difference is measured between the left and the right flap, the hydraulic or the electric power supply to the driving motor is stopped, which stops the flap movement. For safety reasons, wing tip brakes are installed on each wing tip to stop and lock the transmission mechanically in case of certain dangerous failures.

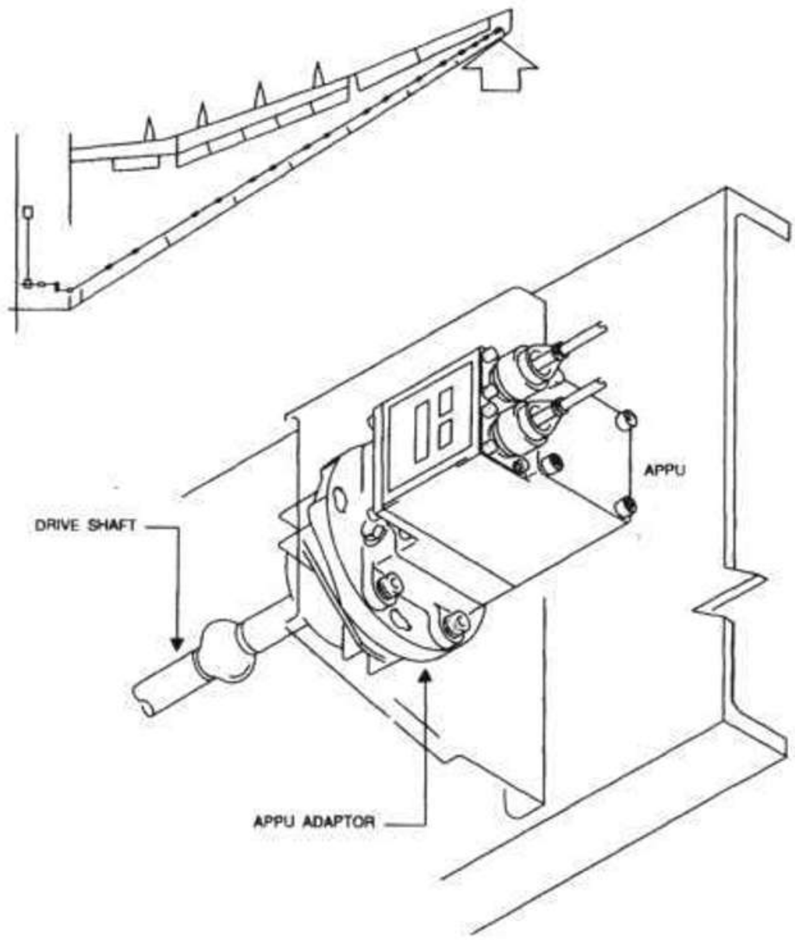


Fig. 9.40 Asymmetry Position Pick-off Unit Wing Tip Brake

The WTB is an electro-hydraulic pressure-on brake. It is installed near the end of the transmission system in each wing. The WTBs stop and hold the transmission if the computers find some given types of failures. If locked, the system can only be reset on ground for safety reasons.

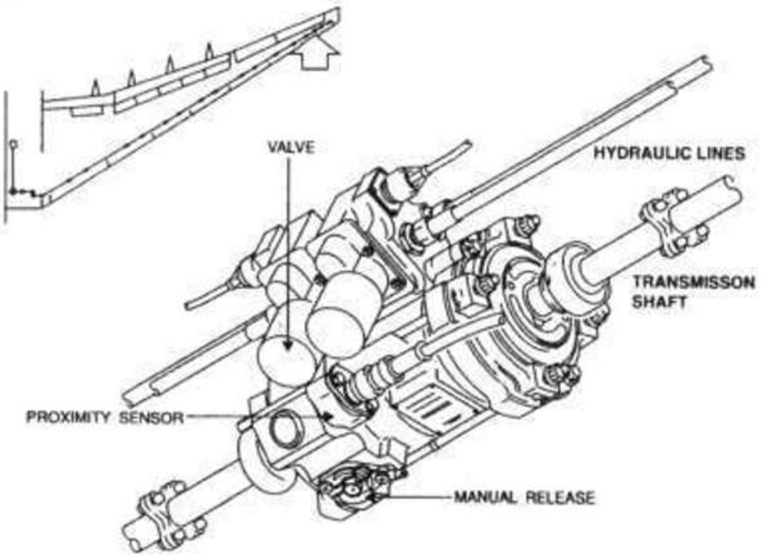


Fig. 9.41 Location of Wing Tip Brake

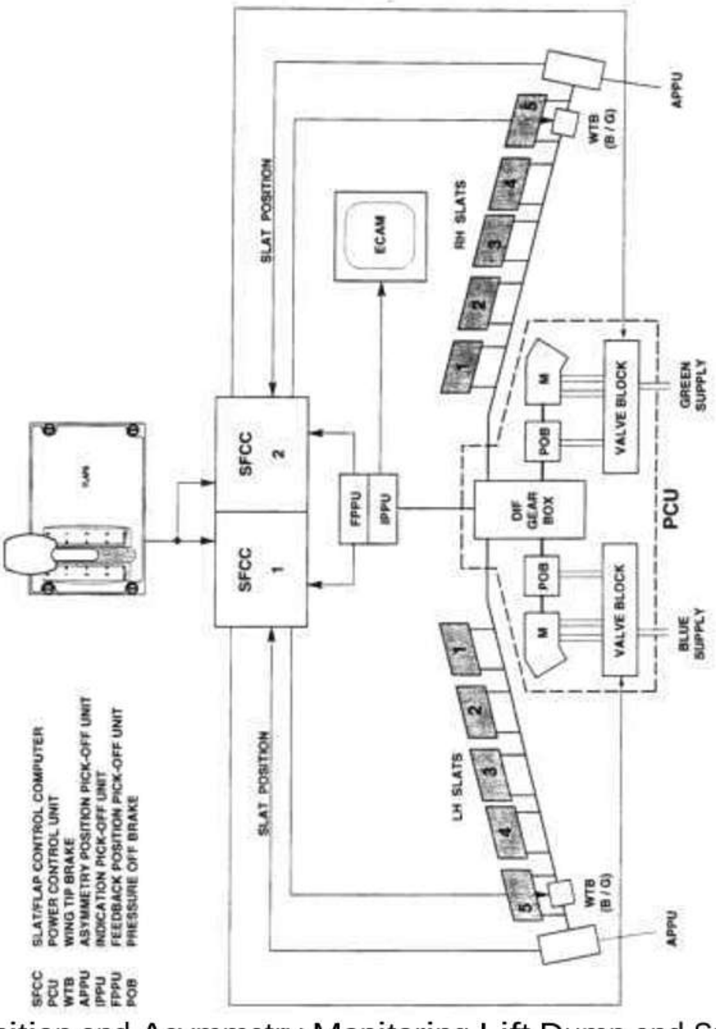


Fig. 9.42 Position and Asymmetry Monitoring Lift Dump and Speed Brakes

Wings of transport aeroplanes are equipped with spoiler surfaces (also known as lift dumpers). The different surfaces are used for different applications depending on the actual flight phase. The spoilers are usually grouped in relation to the different functions. This could look as follows:

Flight spoilers

- Roll spoilers (lateral control augmentation)
- Speed brakes (in-flight)

Ground spoilers (landing roll-out or take-off abort) See Figure 9.46 for a typical spoiler layout.
Roll Spoiler System Operation

Some of the spoiler surfaces on each wing assist the ailerons for lateral control. Movement of the down moving wing spoilers occur as the control wheel (or stick) actuates the ailerons. The aileron system is connected to the spoiler system in a mechanical or electrical way depending on theaeroplane.

Speed Brake System Operation

During flight, manually moving the pedestal-mounted lever aft will extend the flight spoilers on both wings to serve as speed brakes. Different from the roll spoilers, speed brake surfaces are symmetrically extended according to the handle position to a maximum of approximately 35° depending on the aero plane. Use of aileron control during speed brake operation results in asymmetrical extension of the spoilers to assist in lateral control.

Ground Spoiler System Operation

The system may be armed for automatic operation by selecting it on the speed brake lever. When the system is armed, all spoilers (flight and ground) will fully extend (about 60°) after a combination of signals from different systems which determines an unequivocal ground configuration of land-roll or take-off abort. The following list shows the most used signals from other systems used for automatic ground spoilerextension:

- Ground / flightsignal
- Wheel spinningsignal
- Throttle lever anglesignal
- Aeroplane speedsignal

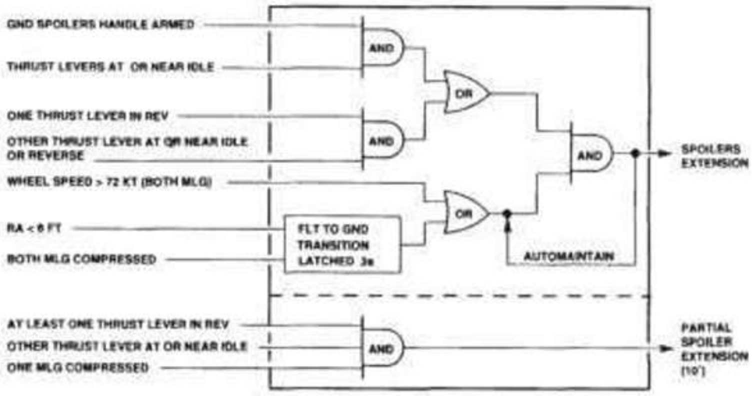


Fig. 9.43 Ground Spoiler Extension Logic (A320)

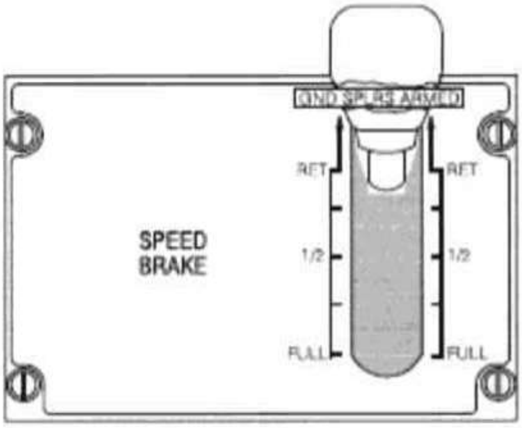


Fig. 9.44 Typical Speed Brake Handle

Fig. 9.45 Spoiler Indication

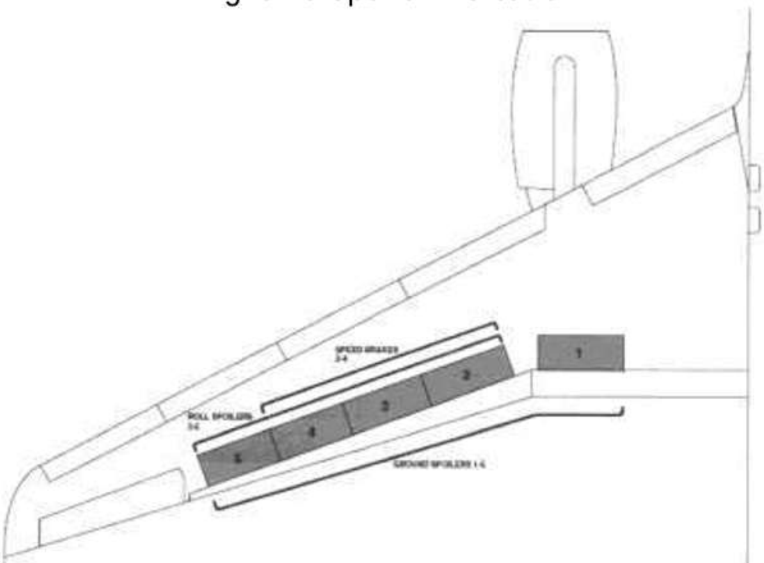


Fig. 9.46 Spoiler Layout

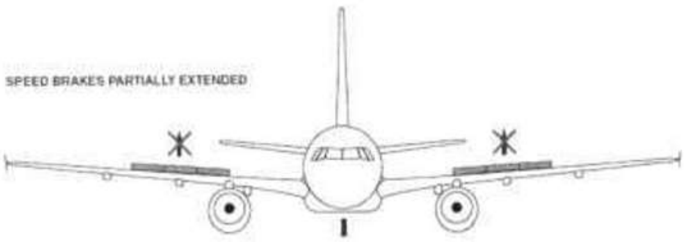


Fig. 9.47 Surface Configurations

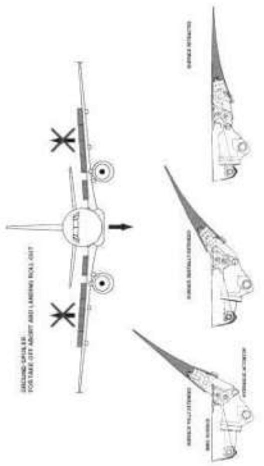


Fig. 9.48 Surface Configurations

On older aircraft, spoiler systems are controlled via a rather complex cable system. We took the MD80 as an example. This system features nearly the same functions as a modern system. Mechanical mixers control the spoilers" correct surface deflection to assist the aileron if a roll command is given.

As in modern systems, several criteria must be fulfilled to get a ground spoiler extension command, as ground spoilers destroy a large amount of lift.

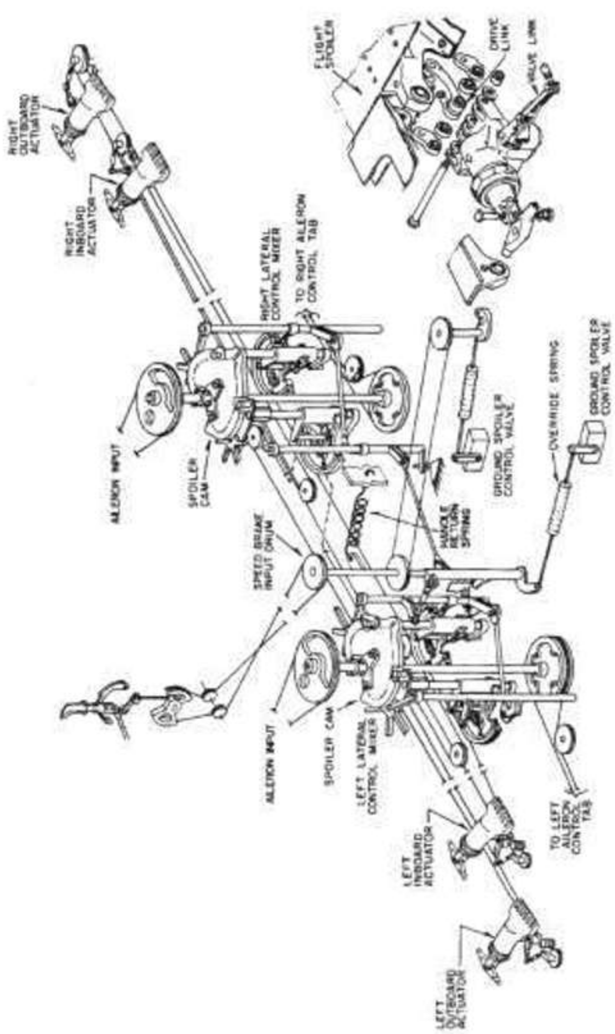


Fig. 9.49 Cable Controlled Spoiler System (MD80) Torque Limiting

In flight, particularly where high rates of control are to be produced, the movement of the flight control surfaces can result in loads which may impose excessive stresses on the aircraft structure. It is necessary, therefore, under automatically-controlled flight conditions, to safeguard against such stresses, and furthermore to safeguard against a servo system'runaway' condition which would cause control surfaces to be displaced to their maximum hardover positions. Such safeguards are implemented by limiting the torque applied by the servo-actuators, and also by allowing them either to slip, or to be completely disengaged, in the event that preset torque limits are exceeded. The methods adopted usually depend on either mechanical, electrical or electromechanicalprinciples.

Artificial Feel and Centring

Irreversible Systems

A hydraulically powered control system is said to be an „irreversible system" in that control forces from the pilot are transmitted, via the hydraulic system, onto the control surfaces. However, aerodynamic loads on the control surfaces due to airflow hitting them, are NOT transferred back to thepilot.

Any aerodynamic trim or control features, such as trim tabs or balance tabs, fitted to a hydraulically powered control system will have no effect. However, they may be fitted to a control system which is hydraulically assisted (and thus maintain some direct, or cable, input from pilot to control surface).

„Q' Feel

In a power-operated system the pilot"s control is connected to the control levers only, while the servo-unit is directly connected to the flight control surface. Thus, the effort required by the pilot to move the control column is simply that needed to move the control lever and control valve piston. It does not vary with the effort required to move the control surface which is supplied solely by servo-unit hydraulic power.

Since no forces are transmitted back to the pilot, the pilot has no "feel" of the aerodynamic loads acting on the control surfaces. It is necessary therefore, to incorporate an "artificial feel" device at a point between the pilot"s controls, and their connection to the servo-unit controllever.

A spring force is usually adequate. However, with elevators and rudders, it is common to have not only a static spring force but also a variable hydraulic force or a spring force more or less compressed by an electricactuator.

Another commonly used system for providing artificial feel, particularly in elevator and horizontal stabilizer control systems, is known as „q" feel. In this system, the feel force varies with the dynamic pressure of the air (i.e. 1/4 pV2 or „q") the pressurebeing sensed by a pitot-static capsule or bellows type sensing element connected in the hydraulic powered controls such that it monitors hydraulic pressure, and produces control forces dependent on the amount of control movement and forward speed of theaircraft.

Fig. 9.51 is a typical artificial feel system using both spring and hydraulic feel. The elevator system is shown, althoughthis artificial feel could be used in rudder or aileron systems. Artificial feel is essentially varied as a function ofairspeed.

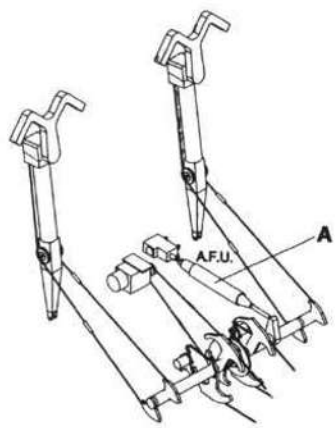


Fig. 9.50 Artificial Feel Unit

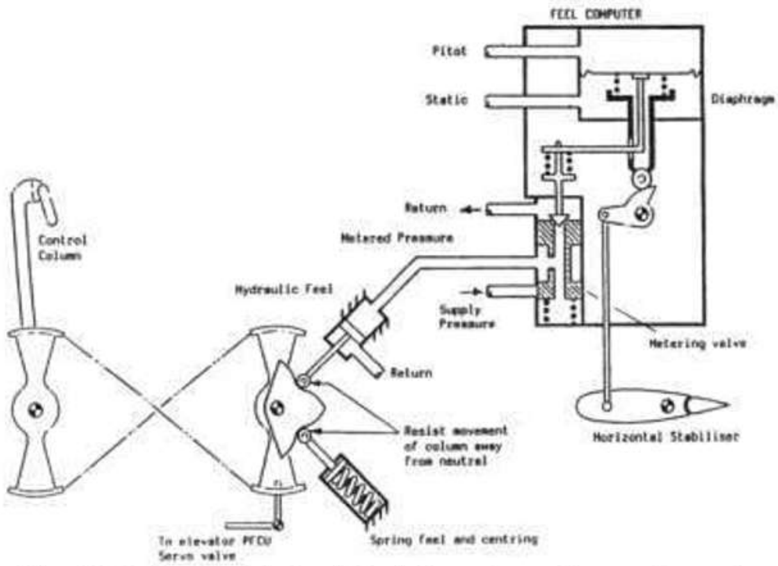


Fig. 9.51 Artificial Feel Unit Lay-Out Flutter Damping

General

A phenomenon known as control surface flutter occurs at specific speeds (usually high speeds) and can cause vibration of the airframe and in severe conditions can lead to structural divergence and catastrophic failure of the structure.

Mass Balance Damping

One remedy for this is to set a specific mass some distance ahead of the control surface hinge- line, such that its moment about the control surface hinge is equal and opposite to the mass- moment of the control surfaceitself.

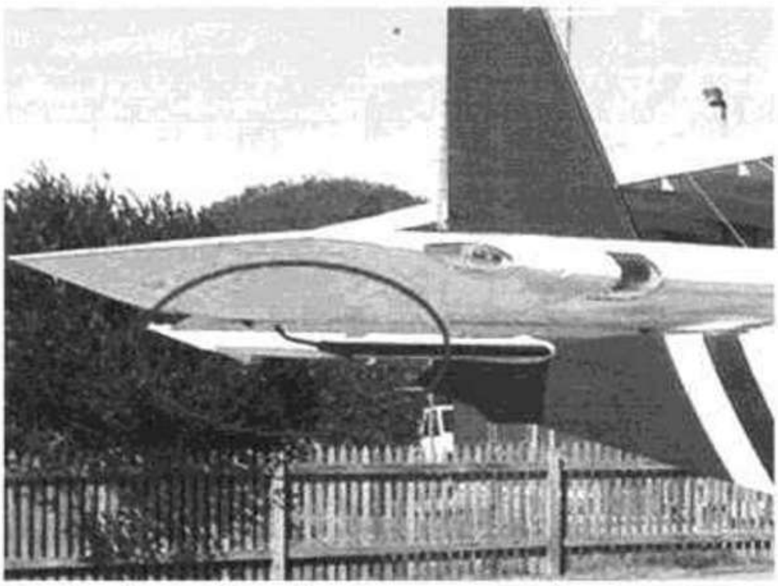


Fig. 9.52 The control surface mass balance can be seen on the Britten Norman Islander’s aileron, on the end of a rod projecting forward of the hinge line. Most other manufacturers hide the mass balance within the wing structure

Hydraulic Flutter Dampers

Some larger aircraft are fitted with dampers to prevent the control surface from fluttering. On direct operated cable systems, separate dampers are connected between the structure and the control surface.

When the surface is moved, the damper drive shaft is rotated moving the rotors in a silicone fluid. The stators and the fluid resist the movement of therotors.

The force necessary to rotate the drive shaft is equal to the internal

resistance of the damper and is proportional to the speed of the motion; the more rapid the motion, the greater the resistance that is applied to the rotors. Normal movement of a surface is slow and is met with little resistance from the dampers. Rapid movement such as that caused by flutter or gusts of wind is met with greater resistance from thedampers.

On hydraulic actuated systems, the damping function is carried out by the actuator itself. Since there is only one actuator in active mode in redundant systems, the other actuator acts as a damper. Most of the actuators have a built in reservoir to maintain the damping function in case of hydraulic powerloss.

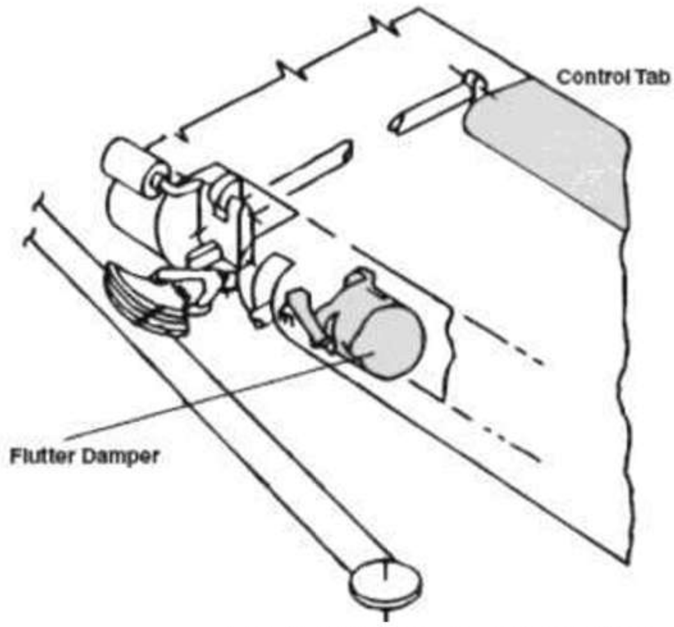


Fig. 9.53 Damper unit position

Fig. 9.54 Damper unit Yaw Damper General

Many of the high-speed jet aircraft with swept-back wings have the undesirable problem of Dutch Roll flight characteristics. This is an oscillatory flight condition that can be very uncomfortable for the passengers and, to counteract it, these aircraft are equipped with yawdampers.

All aircraft, particularly those having a swept-wing configuration, are subject to a yawing-rolling oscillation popularly known as 'Dutch Roll' but different aircraft exhibit varying degrees of damping, i.e. the inherent tendency to reduce the magnitude of oscillation and eventual return to straight flight varies. A sudden gust or a short uncoordinated rudder deflection produces a yawing motion, and this, in turn, initiates the Dutch Roll oscillation. The vertical stabiliser and the rudder (if kept in a fixed streamlined position) develop opposing forces that tend to offset the yawing motion, but as a result of the inertias of the aircraft"s motions, stabilisation is regained in the form of a damped oscillation.

As the aircraft recovers from the Dutch Roll, the magnitude of the oscillations gradually decreases. Thus, the Dutch Roll tendency may be comparatively mild in its effects and may, therefore, be tolerated without recourse to corrective action either manually or automatically.

For some aircraft, however, the natural damping of the Dutch Roll tendency is dependent not only on the size of the vertical stabiliser and rudder, but also on the aircraft's speed, the damping being more responsive at high speeds than at low speeds. It is, therefore, necessary in such cases for corrective action to be taken; such action requires displacement of the rudder in order to further assist the vertical stabiliser in its stabilisingfunction, and is referred to as yawdamping.

It is usual therefore, for these aircraft to utilise a two-axis automatic control system with control about the third axis, i.e. rudder control, being provided by a sub-system called a yaw damper.

The system is so designed that it can be operated independently of the automatic control system, so that in the event that the aircraft must be flown manually, Dutch Roll tendencies can still be counteracted. The system may be 'switched in' either by selecting a 'damper' position of the main engage switch on the automatic control system control panel, or by actuating a separately located yaw damperswitch.

In those aircraft having upper and lower rudder sections (e.g. Boeing 747) a yaw damper system is provided for each rudder section.

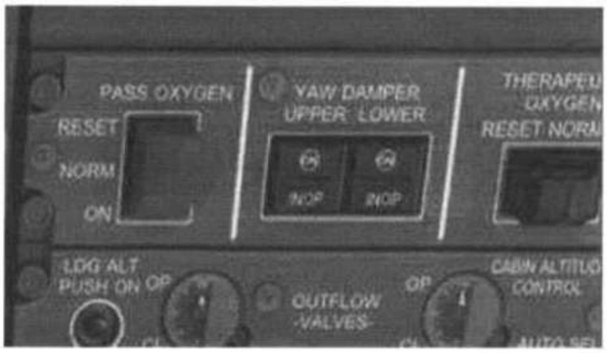


Fig. 9.55 Yaw damper switches for the two rudder sections of the B747

A rate gyro senses the rate of yaw of the aircraft and sends a signal to the rudder servo that provides exactly the right amount of rudder deflection to cancel the Dutch Roll before it gets enough amplitude to be disturbing.
The yaw damper system provides the following functions:

- Dutch rolldamping
- Turn coordination in low speed manual flight to reduce the sideslip induced by theturn.
- Engine failure compensation. A command is generated to the rudder to counteract sideslip during the transient induced by an enginefailure

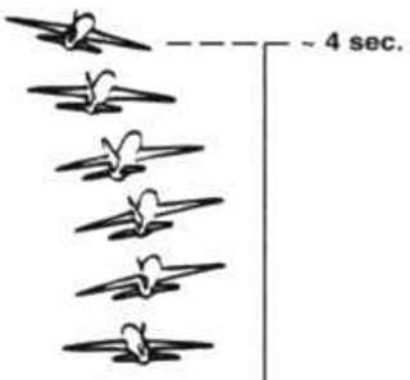


Fig. 9.56 Dutch Rolling Yaw Damper Systems

The operating fundamentals of yaw damper systems in general may be understood from Error! Reference source not found.. The principal component of a system is the yaw damper coupler which contains a rate gyroscope powered directly from the aircraft's 115 volts AC. supply, and the logic switching circuits relevant to filtering, integration, synchronising, demodulation, and servo signal amplification (gain). Servo amplifier output is supplied to the transfer valve of the rudder power control unit; this unit differs from those used for aileron and elevator control in that it has an additional actuator (the yaw damper actuator) and does not include the automatic control system engagemechanism.

The actuator is also mechanically linked with a valve that controls hydraulic pressure to the main actuator, the piston rod of which is extended or retracted to position the rudder left or right respectively. A lever which forms part of the linkage provides for mechanical summing of yaw damper input to the power control unit, and an input from the rudder trim control system. Under manual control, inputs from the rudder pedals also pass through the summing lever to operate the main actuator.

Before the system is switched on, the integration circuit within the coupler unit ensures that the system is in synchronism; the yaw damper fail light is illuminated at this time. When the yaw damper switch is at 'ON', DC power is applied to an engage relay in the accessory box, and on being energised, the relay completes a circuit to the actuator solenoid valve in the power control unit to allow hydraulic fluid to flow to the transfer valve; this is indicated by extinguishing of the fail light.

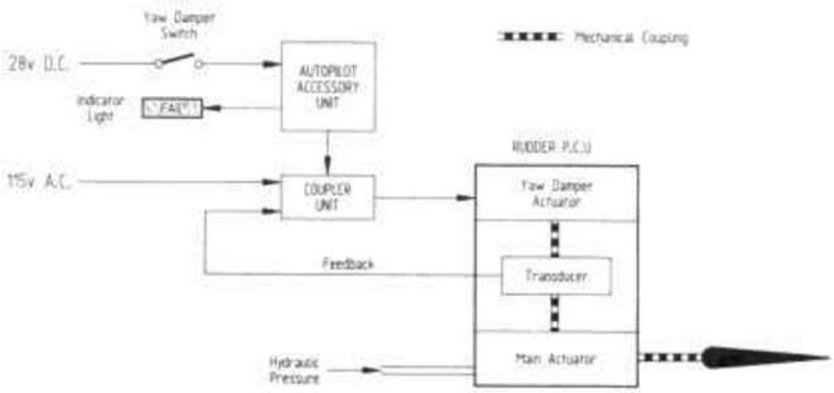


Fig. 9.57 Yaw damper system

Whenever a yawing motion of the aircraft occurs, the rate gyroscope will be precessed and its sensing element will produce an AC output appropriate to the direction and rate of movement.

The signal passes through a filtering circuit that discriminates between frequencies associated with flexing of the aircraft's fuselage, and with normal turns, so that only those frequencies related to Dutch Roll are allowed to pass. The signal is then demodulated and amplified, and applied to the transfer valve. The signal also passes through a circuit contained in the airspeed sensing section of an air data computer and is varied as an inverse function of airspeed. This will be used to control the amplifier gain of the correction circuit. The circuit may be of the potentiometric type, or of the switched gain type; the latter being used particularly in digital air datacomputers.

The transfer valve directs fluid under pressure to the additional yaw damper actuator. Movement of the actuator piston is transmitted to a control valve in the main actuator which, through its piston rod, then moves the rudder in the required direction. The yaw damper actuator piston also positions the core of the LVDT to produce a position feedback signal to cancel the rate gyroscope input when the actuator piston rod has moved the required amount. The feedback signal is also supplied to a position indicator, the display element of which moves left or right appropriate to the direction of control applied by the yaw damper actuator.

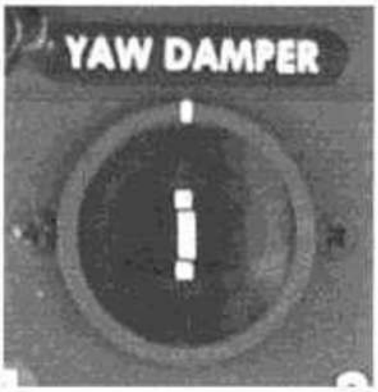


Fig. 9.58 Yaw damper position indicator (Boeing 737)

When yaw oscillations have been damped, a constant signal passes from the LVDT to an integrator in the coupler unit; the integrator output then builds up to assist the LVDT in centring the yaw damper actuator, and thereby returning the rudder to its neutral position. The rudder pedals are not displaced during yaw damper operation, i.e. the system is of the series- connectedtype.

A two-position test switch is provided to simulate the effects of oscillations, and when operated torques are applied to the rate gyroscope causing it to apply left or right rudder as appropriate, to the power control unit. Movements of the position indicator display element provide for monitoring of the position feedback signal from theLVDT.

In order to compensate for differences in aerodynamic damping which in some aircraft can arise between the landing flaps down and flaps up conditions the yaw rate gyroscope output signal is also passed through a gain change circuit controlled by a relay that is operated by a flap position switch. The control is such that energizing of the relay bypasses a resistance so as to produce a faster rate of response when the flaps are down.

An automatic flight control system may be used in all modes with the yaw damper system engaged; however the associated interlock circuit prevents the use of the control system when the yaw damper is disengaged. When the control system is operating in the localizer mode, the yaw damper is supplied with signals from the aileron control channel through a cross-feed circuit.

In manual control of the aircraft, if the pilot operates the rudder pedals at the same time that the yaw damper is applying input to the rudder, the two inputs will be summed and the input to the rudder will be modified accordingly.

Control Loop

The block diagram of Error! Reference source not found, shows a typical yaw damper control loop. The complete rudder channel may or may not perform other functions as seen before.
This is to say that the Dutch roll is only dampened, and therefore it is not eliminated.

In the signal source on the left of the block diagram, yaw rate or yaw acceleration is required. This signal is typically supplied by the inertial reference system, a yaw rate gyro or yaw accelerometers.

The shaper/processor accomplishes whatever is necessary in the way of conversion, smoothing, dampening, limiting, and gain control. Its output goes to the Dutch roll filter.

The Dutch roll filter is a band-pass filter, which attenuates all signals which are not at the frequency of the Dutch roll. Its output is a continuously changing command for left rudder, then right rudder, then back to left rudder and so on.

The servo amplifier amplifies this signal as required to control the servo and operate the rudder the correct amount to eliminate most of the Dutch roll.

The small amount of Dutch roll that is not eliminated is represented with a dashed line coupling the aeroplane to the signal source.

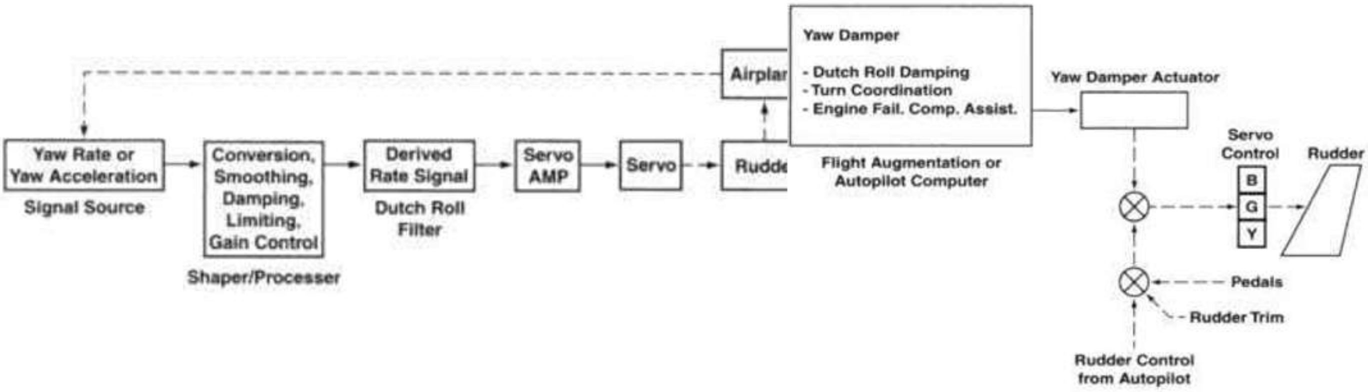


Fig. 9.59 Block Diagram

The yaw rate gyro senses the oscillations around the yaw (Z)axis.
The oscillation frequency which defines "Dutch Roll" isabout
0.25 Hz. The air data computer provides the computed airspeed for gain programming to the yaw damper computer. At high airspeed the correction is smoother than at lowairspeed.

The Y/D servo output is applied to the rudder actuator to control the rudder. The steering signal will not be felt at the pilot"spedals. The yaw damper authority on the rudder is about±2.5°.

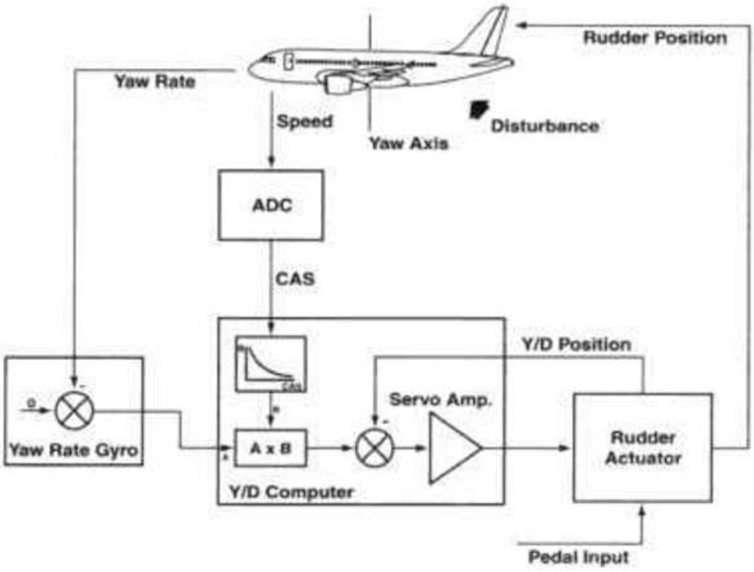


Fig. 9.60 Control Loop

Yaw Damper Signals

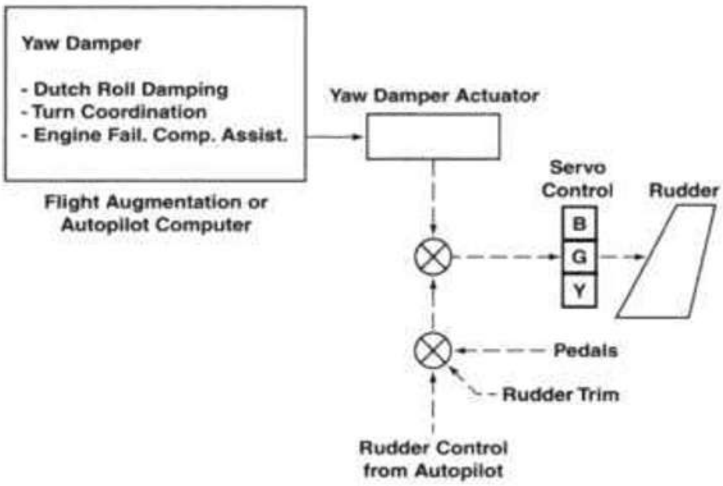


Fig. 3.1 Rudder Functions

The left diagram on Figure 9.61 illustrates an aeroplane flight path beginning at the left with a straight path. It soon changes to a constant rate of turn to the right. Near the end it resumes a straight path.

The rate gyro output represents the 400 Hertz synchro signal developed by the rate gyro during this flight path. When the aeroplane is flying straight ahead there is no output from the synchro.

During this time it is making a constant rate of turn, and the output is of a particular phase with constantamplitude.

The DC graph shows the demodulated and filtered output of a Dutch roll filter. Only during the time that the rate of turn is changing is there an output from the Dutch roll filter.

The right diagram on Figure 9.61 shows the flight path and the changing turns that occur during a Dutch roll. In a Dutch roll manoeuvre the rate of turn is constantly changing. Since the rate of turn is constantly changing, the output of the rate gyro is constantlychanging.

The DC graph is the Dutch roll filter output resulting from the rate gyro input. The DC polarities are greatest when the rate of turn is greatest, and reverse when the direction of turn (phase of gyro signal) reverses.

The Dutch roll filter is a narrow band pass filter designed to pass only signals which change at the frequency of the Dutch roll, which range from 1/5 Hz to 1/3 Hz. The rate gyro produces outputs for all turns, but only those related to Dutch roll will

appear at the input to the servo amplifier driving the rudder servo motor.

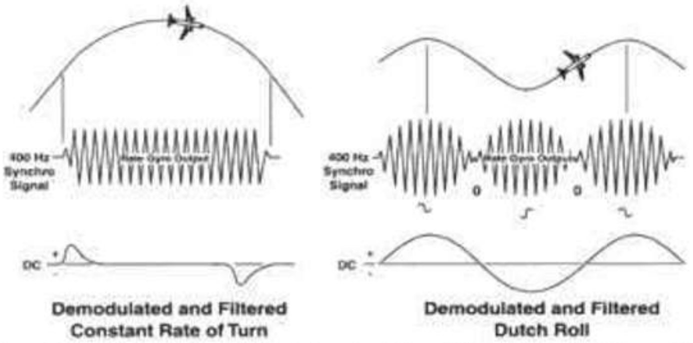


Fig. 9.61 Dutch Roll detected by Yaw Rate Signals

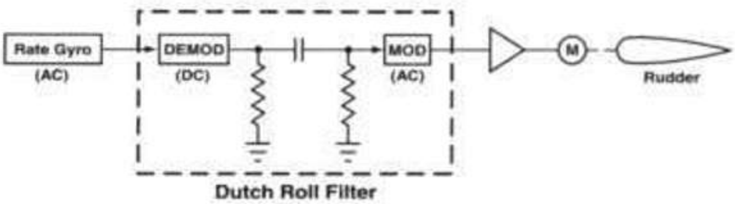


Fig. 9.62 Dutch role filter Yaw Damper Servo

This actuator moves the output rod in a linear direction. The input to the rudder system is in series mode. No movements at the rudder pedals are present. The rudder deflection is +/- 3-6°.

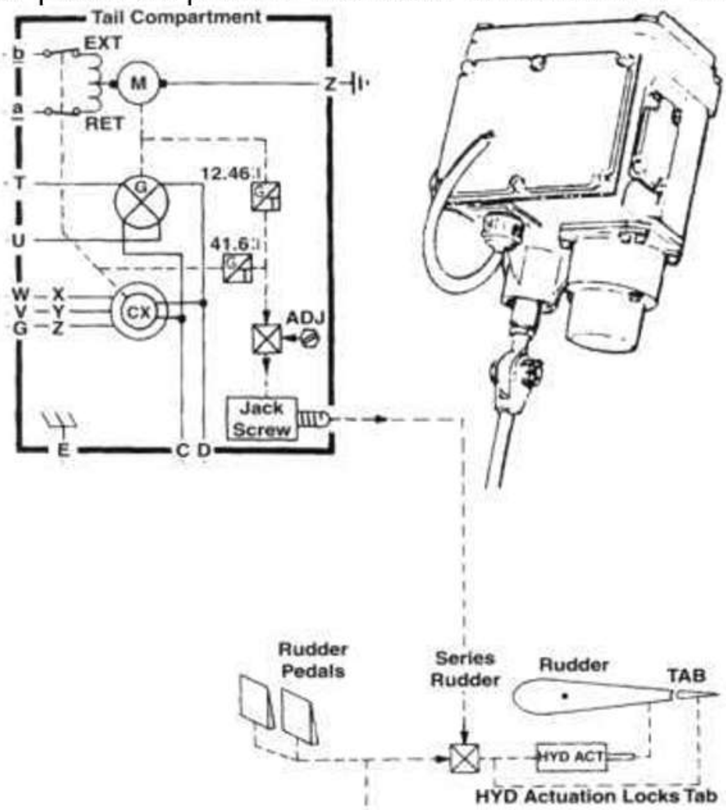


Fig. 9.63 Servo linear type Mach Trim Mach Tuck

Mach tuck (also known as "tuck-under") is dependent upon the dynamics of lift.

Mach tuck is the result of an aerodynamic stall due to an over- speed condition, rather than the more common stalls resulting from boundary layer separation due to insufficient airspeed, increased angle of attack, excessive load factors, or combination of those causes. As the aircraft's wing approaches its critical Mach number, the aircraft is travelling below Mach 1.0. However, the accelerated airflow over the upper surface of the cambered wing exceeds Mach 1.0 and a shock wave is created at the point on the wing where the accelerated airflow has gone supersonic. While the air ahead of the shock wave is in laminar flow, a boundary layer separation is created aft of the shock wave, and that section of the wing fails to produce lift.

In most aircraft susceptible to Mach tuck, the camber at the wing root, the section of the wing closest to the fuselage, is more pronounced than that of the wing tip. This design ensures that in a standard stall the root will stall before the tips. This allows the pilot to recognize the stall while still maintaining control of the ailerons to enhance stall recovery. However, this also means that when an airfoil exceeds its critical Mach number, the shock wave, and resulting stall condition, will begin to form at the root.

A second design element that leads to Mach tuck is that many aircraft which will approach the speed of sound are designed with swept wings. The centre of pressure of a wing is an imaginary point where the summation of all lifting forces across the wing's surface can be resolved into a single lift vector. When the wing root stalls, the centre of pressure of the (reduced) lift being generated by the wing is shifted towards the wing tip. With a swept wing, this also means that the centre of pressure travels aft (because it is travelling out from the wing root and therefore backwards as the wing sweeps). When the centre of pressure moves aft, its movement rearwards compared to the unmoving centre of mass of the aircraft will generate a force which will act to depress the nose of the aircraft; this nose down pitching moment is "Machtuck."

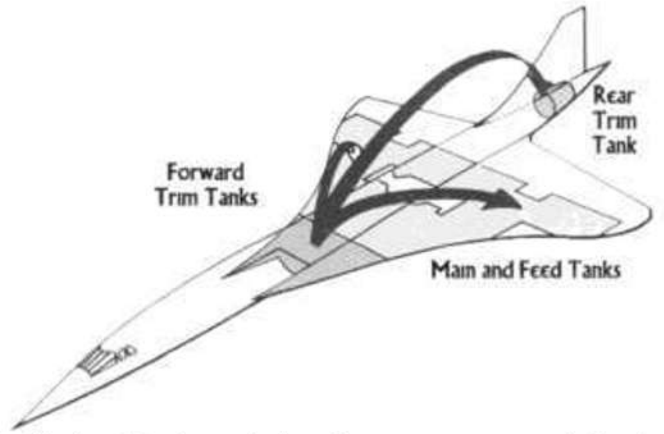
As the wing becomes more affected by the shock wave the centreof pressure will continue to travel aft, thereby causing a significantly higher nose-down force and requiring a nose-up input or trim to maintain level flight. Although Mach tuck develops gradually, if it is allowed to progress significantly, the centre of pressure can move so far rearward that there is no longer enough elevator authority available to counteract it, and the aeroplane enters a steep, sometimes unrecoverabledive.

In addition, until the aircraft goes supersonic, as the shock wave goes towards the rear, because of the faster flow there the top Shockwave will impinge upon the horizontal stabilizer and elevator control surfaces further back than the lower Shockwave; this can greatly exacerbate the nose down tendencies. The horizontal stabilizer at the tail of the aircraft generates a downward force, so loss of effective horizontal stabilizer area will reduce this downward force, so the tail will pitch up and the nose will pitch down. If the shock wave affects the elevators, it may reduce their effectiveness, making it impossible for the pilot to alter the aircraft'spitch.

Finally, there is a related condition that can exacerbate Mach tuck. If enough of the wing surface becomes engulfed in the shock wave, the wing will not produce enough lift to support the aircraft, and a standard stall will occur. This often fatal combination of overspeed and aerodynamic stall can most easily be avoided by not allowing the effects of Mach tuck to develop beyond its incipient stage. This is best accomplished by retarding the throttle, extending speed brakes, and if possible, extending the landing gear. Any actions, which would increase aerodynamic drag and thus reduce airspeed below critical Mach, will prevent further aggravation of thecondition.

Fuel Transfer

For aircraft such as supersonic fighters/bombers or supersonic transports such as Concorde that spend



long periods in supersonic flight, Mach tuck is often compensated for by moving fuel between tanks in the fuselage and tail (trim tank) to change the position of the centre of mass. This minimizes the amount of trim required and keeps the changing location of the centre of pressure within acceptable limits, and negates the need for any control surface movement to counter the Mach Tuck.

Fig. 9.64 Movement of fuel from forward tanks to aft tanks to move the centre of mass rearwards

Elevator/Stabilizer Trim

On most other large aircraft, Mach trim is automatically accomplished above about Mach 0.65 (depending on aircraft type) by adjusting the elevators with respect to the stabilizer as speed increases. The Flight Control Computers use Mach information from the ADIRU to compute a Mach trim actuator position. The Mach trim actuator repositions the elevator feel and centring unit which adjusts the control column"s neutral position.

The Mach Trim system actually "over" compensates and trims the nose up more than it should to just cancel the tuck-under. This is to comply with the EASA CS-25 requirement for minimum stick force stability.

The Specification is that at least 11b of stick force is required for every 6 kts change in speed. The normal reaction is termed as positive stick force stability. If speed increases the pilot must push forward on the stick by a minimum of 11b for every 6 kts.

As in the case of a yaw damper, it is a sub-system to an automatic control system in that it can be operational whether or not automatic control is engaged.

The principal components of the system are a trim coupler unit, an actuator, a test switch and a failure light. The coupler unit is electrically connected to an air data computer from which it receives signals corresponding to airspeed in terms of Mach number. It contains all the logic circuits necessary for the processing of the signals and their amplification before supplying them to the trim actuator.

The actuator consists of a 115 volts 400Hz single-phase motor which drives a screw type shaft through a gear train. The shaft is connected to the stabiliser via the neutral shift mechanism, while the actuator body is pivoted to the elevator feel and centring unit. A brake mechanism supplied with 28 volts DC from the trim coupler unit unlocks the actuator to allow its movement when engaged and operating. Switches are provided to limit actuator motion at its fully extended and retracted positions.

At airspeeds below the value preset for the aircraft type (0.715M in one particular case) the Mach trim actuator is inoperative and itsshaftisinthefullyextendedposition.Asspeedincreases and exceeds the set value a signal is supplied from the trim coupler unit to release the brake, and a speed signal from the air data computer is supplied to the motor which then rotates the screwed shaft. Since the shaft is connected to the stabiliser via the neutral shift mechanism the actuator body itself is traversed along the shaft. Movement of the body also rotates the feel and centring unit, and assuming that the automatic control system pitch channel is not engaged, the main control valve of the power control unit will be directly actuated and displacement of the unit will move the elevators up so that they counteract the „tuck under" effect. The Mach trim actuator motor drives a CX synchro that provides a position feedback signal to the coupler unit After demodulation it is supplied to the servo amplifier and when it opposes the command signal no further control is applied by the actuator motor. In addition to a pre-set 'start' value of Mach speed, there is also a corresponding value at which command signals arelimited,

e.g. the speed value of 0.715M referred to earlier has a corresponding limit value of 0.815M. The elevator correction versus the Mach number values quoted is shown graphically in Figure 9.66. The change in elevator correction commanded at 0.78M is produced by second stage amplification of the signal within the coupler unit.

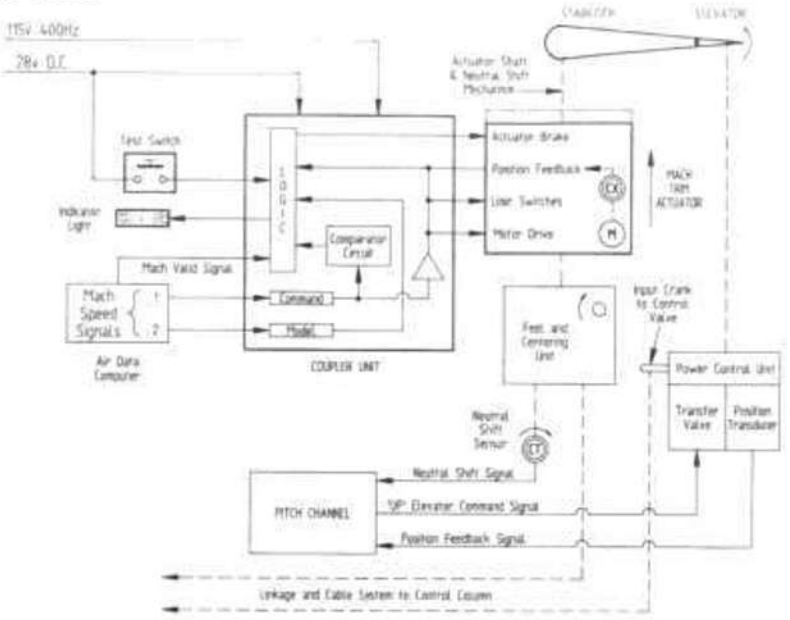


Figure 9.65: Mach trim circuit

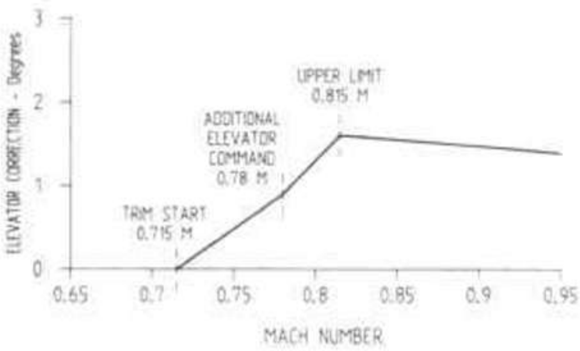


Fig. 9.66 Mach trim elevator correction

When the aircraft's speed decreases, the command signal also decreases, and so the position feedback signal now predominates to drive the trim actuator back to the extended position. When this

is reached, the circuit to the actuator motor is interrupted, and the elevators will at that time have been returned to their neutral position.

It will also be noted from Figure 9.65 that the air data computer provides two Mach speed signal data outputs to the trim coupler unit. These signals are of equal value and are supplied to a command channel and a 'model' channel. As the command channel is supplying a signal to the trim actuator, the 'model' channel furnishes a signal to a comparator circuit, in which it is compared to the feedback signal from the actuator position synchro. These two signals should be of the same amplitude, but of opposite sign. If the signals are not within tolerance, the circuit to the actuator motor is interrupted and this is indicated by illumination of the fail indicatorlight.

The system operates in a similar manner when the pitch channel of the automatic control system is engaged, except that the power control unit now moves the elevators up as a result of a command signal being applied to its transfer valve from the pitch channel This signal is balanced against a feedback signal from the power control unit's position transducer, and also a signal generated by the neutral shift sensor as it responds to movement of the feel and centringunit.

The test switch is provided for checking system operation and control movements by simulating the command signal input when speed is above the preset 'start' level, e.g. 0.715M. Since, under test conditions, the feed-back signal from the Mach trim actuator is not equal to the output from the 'model' channel of the coupler unit, the comparator circuit will cause the fail indicator light to illuminate.

Speed Trim

Some aircraft are fitted with a type of trim known as Speed Trim (mainly Boeing).

The speed trim system (STS) is a speed stability augmentation system designed to improve flight characteristics during operations with a low gross weight, aft centre of gravity and high thrust when the autopilot is not engaged. The purpose of the STS is to return the aeroplane to a trimmed speed by commanding the stabilizer in a direction opposite the speed change. The STS monitors inputs of stabilizer position, thrust lever position, airspeed and vertical speed and then trims the stabilizer using the autopilot stabilizer trim. As the aeroplane speed increases or decreases from the trimmed speed, the stabilizer is commanded in the direction to return the aeroplane to the trimmed speed. This increases control column forces to force the aeroplane to return to the trimmed speed. As the aeroplane returns to the trimmed speed, the STS commanded stabilizer movement isremoved.

STS operates most frequently during takeoffs, climb and go- arounds.

Rudder Limiter

The rudder control has the greatest aerodynamic authority over the aircraft‟s control. So much so that an aggressive rudder inputfrom the pilot can cause catastrophic structural damage of the tail section of theaeroplane.

Large aeroplanes are provided with a rudder travel limiter to protect the empennage from overload in case of inadvertent application of excessive rudder control at high speed. Some manufacturers call this „Rudder Ratio‟ changing.

In older aeroplanes, the limiter operates by ram air pressure from its own pitot tube installed usually in the leading edge of the vertical stabiliser. The higher the airspeed, the more ram pressure, resulting in proportional restriction of rudder movement.

Modern aeroplanes are provided with a computer controlled rudder travel limiter which receives airspeed electrical signals from the air data system for travel limiting computation (Figure 9.53).

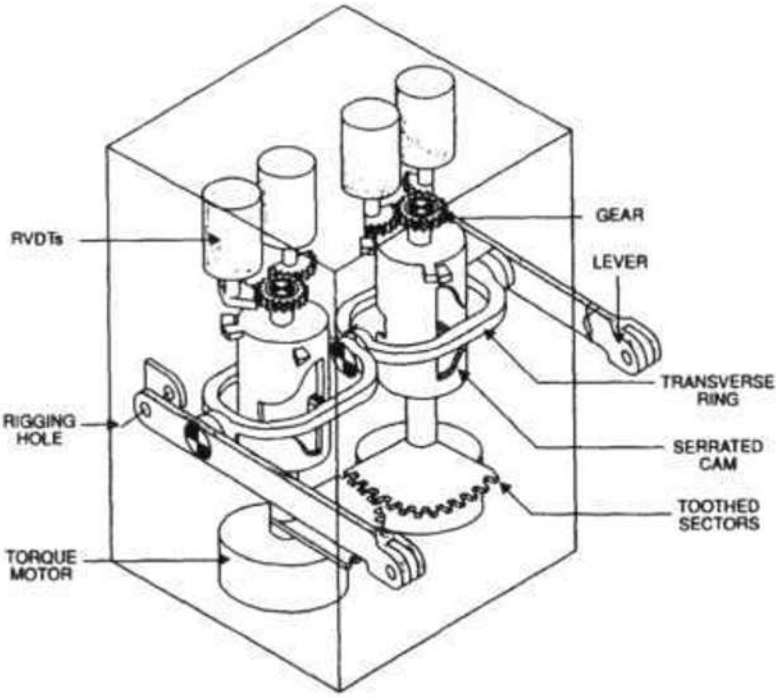


Fig. 9.67 Rudder travel limiter

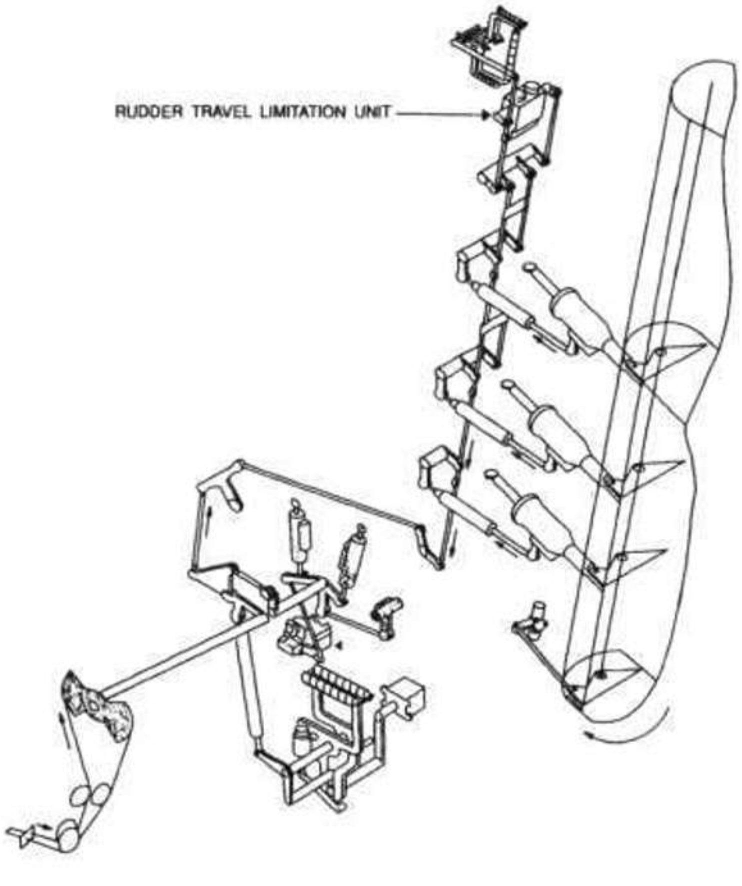


Fig. 9.68 Typical location of a Rudder Travel Limitation Unit

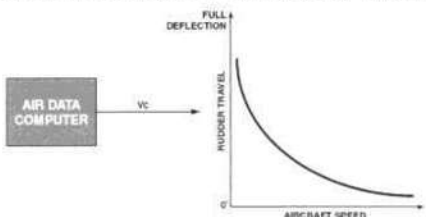


Fig. 9.69 Rudder Ratio Changer input

Gust Locks

Gust Locks lock the flight controls when the aircraft is parked to prevent them from moving around and causing damage to the flight controls and related system. Most modern large jet aircraft do not need them as the flight controls are hydraulically controlled and when the hydraulic system is 'at rest,' a dampening device in the actuator automatically locks the flight control in a neutral position, or the actuators themselves work as natural dampers if the hydraulic systems are depressurized.

Aircraft with cable operated control systems may be fitted with gust locks. The locking system consists of a lever on the rear of the pedestal connected to spring loaded pins on each of the control surfaces through a cable system. The pins are spring loaded to the off position in flight. Thus, if the cable fails in flight, the controls stay unlocked.

Rigging of Cable Operated Flight Controls

The cable system rigging should be checked at specified periods, system maintenance and repair, after heavy landings or abnormal flight loads to ensure that the components are not distorted and that the angles of the control surfaces are within permitted limits. The relevant figures together with permitted tolerances are specified in the appropriate manual for the aircraft concerned.

The usual method of checking rigging angles is by the use of special boards (or the equivalent) in which are incorporated or on which can be placed an instrument for determining the angle, i.e. a spirit level or clinometer as appropriate.

As different procedures apply for other aircraft types, adjustment of MD-80 ailerons has been taken as an example. See Figure 9.70.

The following description is an extract from a maintenance manual

A. Adjust the System from Aileron Control Wheel to Torque Tube

1. Insert rig pin (4-4) in Rig hole (R-3) in aileron bus torque tube (AA)

- Torque tube and turnbuckles, located below flight compartment floor, are accessible through electrical/electronic compartment

2. Adjust turnbuckles (1) until tension is between minimum and maximum load per cable tension table for 1/8-inch cables. (64- 68 lbs)

3. Differentially adjust turnbuckles (1) to align index marks (View A) on the control column and the control wheel is within 1/32inch

4. Safety all turnbuckles with clips

5. Remove rig pin (4-4) from the torque tube

6. Check aileron and control tab travel

B. Adjust System from Torque Tube to Lateral Control Mixer

Lateral control mixers are located in left and right main gear wheelwells.Turnbuckles(2)areaccessiblethroughforward lower cargo compartment ceiling panels 5151C for cables 7C, D. 5144C for cables B, C, D. 5156C for cables 8C, D, 9C, D

1. Open the main gear inboard doors and install the door safety locks

2. Insert rig pin (4-4) in rig hole (R-3) in torque tube(AA)

3. Adjust the turnbuckles (2) until tension is between minimum and maximum load per cable tension table for 1/8-inch cables. (64-68lbs)

4. Differentially adjust turnbuckles (2) until rig pin (4-10) can be easily inserted in the rig hole (R-7) in applicable lateral control mixer

5. Safety all turnbuckles withclips

6. Remove rig pin (4-4) and(4-10)

7. Check aileron and control tab travel

C. Adjust System from Lateral Control Mixer to Control Tab Sector

1. Extend Flaps to 40degree

2. Insert rig pin (4-10) in rig hole (R-7) in applicable lateral control mixer

3. Adjust turnbuckles (4) until tension is between minimum and maximum load per cable tension table for 1/8-inch cables. (64- 68 lbs)

4. Differentially adjust turnbuckles (4) until rig pin (8-7) can be easily inserted in the rig hole (R-4) through bracket, tab sector and bussector

5. Safety all turnbuckles withclips

6. Remove rig pin (8-7) and(4-10)

7. Check aileron and control tab travel

Close up must be carried out as described in the maintenance manual. A flight control checkmust be carried out if demanded in the manual.

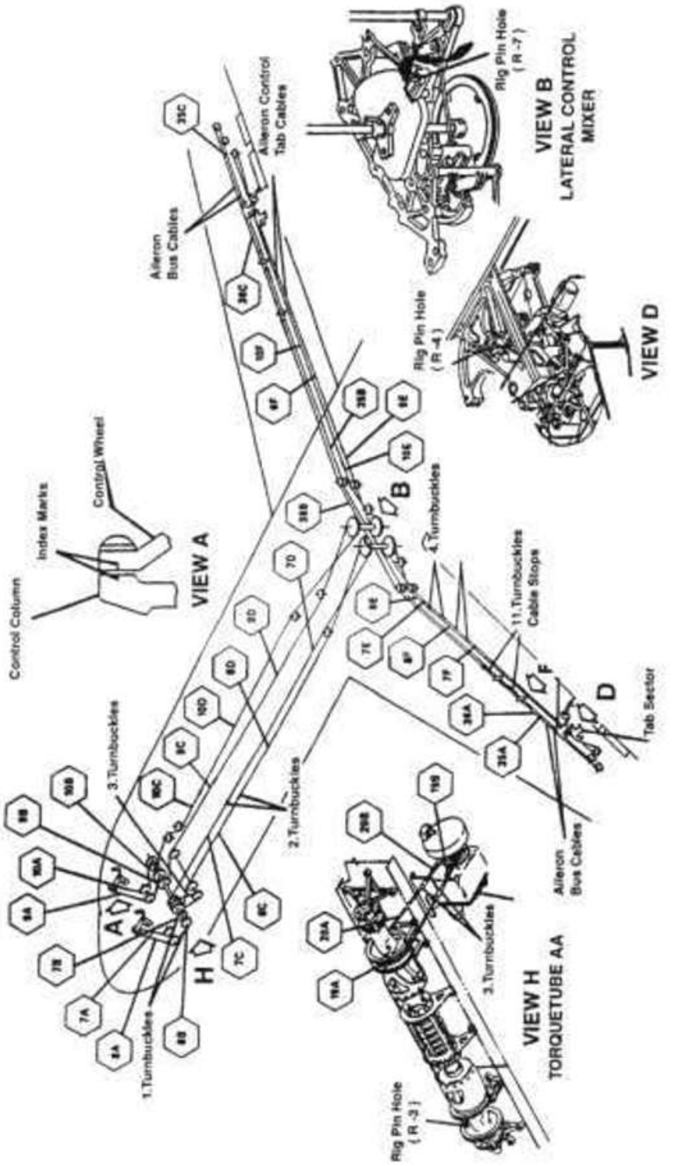


Fig. 9.70 Rigging of Cable operated Flight Controls

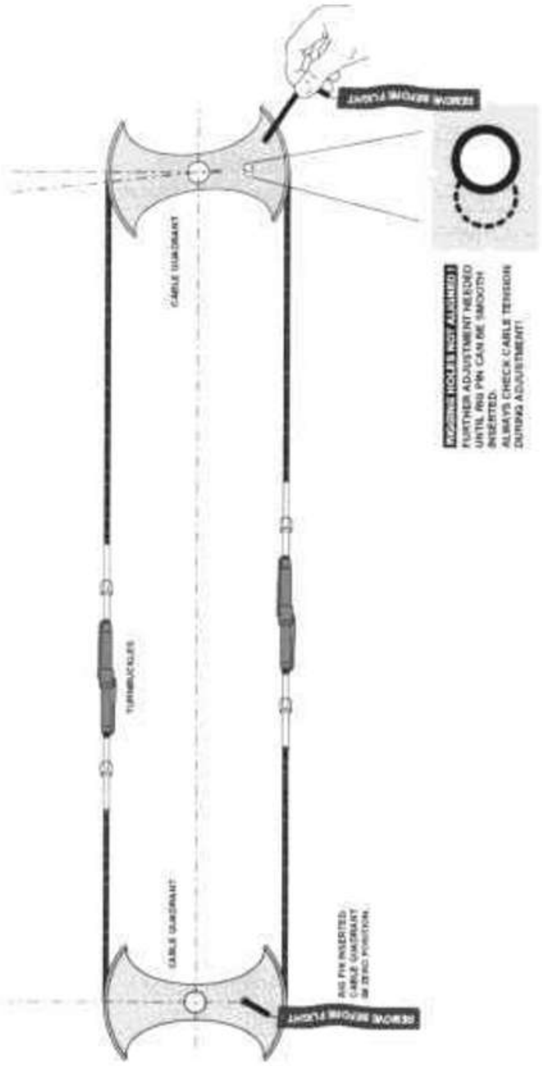


Fig. 9.71 Principle of Rigging

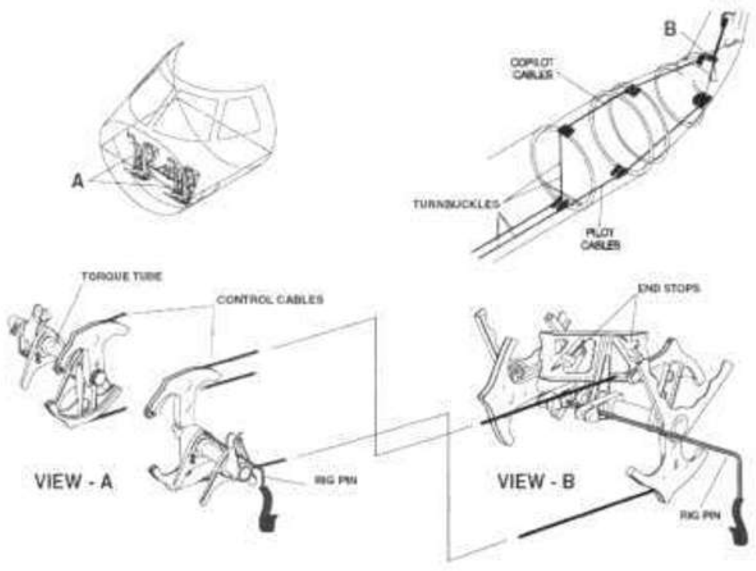


Fig. 9.72 Usage of Rig Pins

Stall Protection Systems General

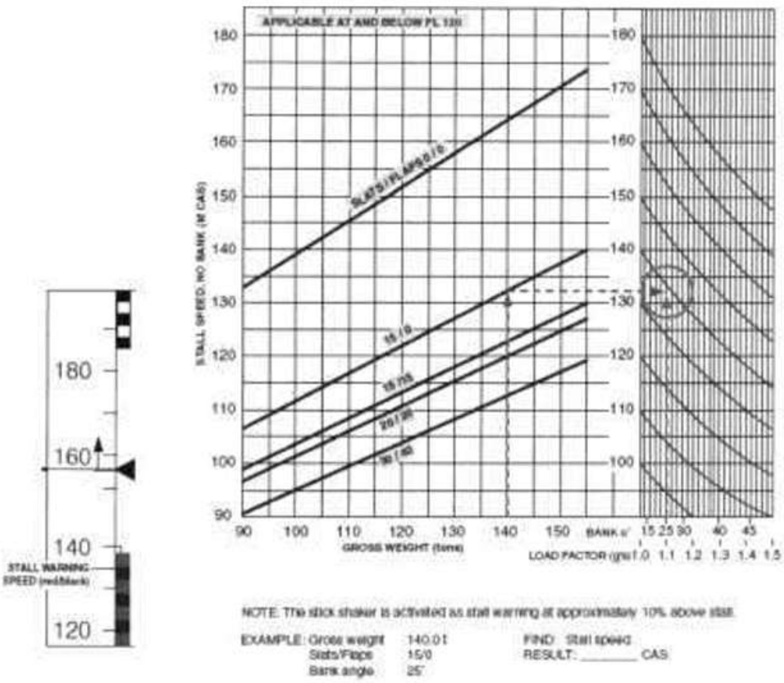
Vs (Stall Velocity) are the calibrated stalling speed, or the minimum steady flight speed, in knots, at which the aeroplane is controllable.

Most transport aeroplanes are equipped with a stall prevention and warning system. The system consists of computers which activate a prevention and warning mechanism as a function of angle of attack, flaps, slats, horizontal stabilizer position and airspeed.

On some aircraft, when it nears stall speed, the slats are automatically moved to a position to prevent stall. Manual control is then inhibited. At this stage the pilots are visual and tactile warned. A stick (control column) shaker is used for tactile warning.

Note that most stall warning systems operate as a function of aircraft indicated airspeed. Such a stall warning system will commence warning at a speed slightly above the stall speed, thus warning the pilot that if the aeroplane speed drops any further, a stall will occur.

Stall speeds differ according to type of aircraft. Charts can be found in the operation manual to find the minimum permitted speed with different configurations. On modern aircraft, the stall warning speed is calculated in real time depending on the actual configuration (slat, flap, landing gear).



See the stall speed chart in Figure 9.73 as an example.

Fig. 9.73 Stall Speed Chart (Airbus A310)

Stick Shaker

A stick shaker is a mechanical device to rapidly and noisily vibrate the control yoke (the "stick") of an aircraft to warn the pilot of an imminent stall. It is connected to the control column of most large civil aircraft.

The stick shaker is a component of the aircraft's stall protection system, which is composed of wing-mounted angle of attack (AoA) sensors that are connected to an avionics computer. The computer receives input from the AoA sensors and a variety of other flight systems. When the data indicate an imminent stall condition, the computer actuates both the stick shaker and an auditory alert.

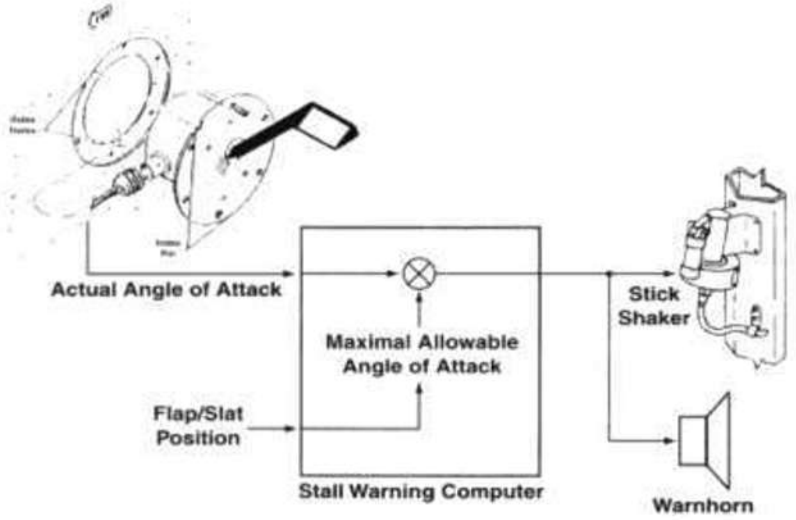


Fig. 9.74 Stick shaker system

The shaker itself is composed of an electric motor connected to a deliberately unbalanced flywheel. When actuated, the shaker

induces a forceful, noisy, and entirely unmistakable shaking of the control yoke. This shaking of the control yoke matches the frequency and amplitude of the stick shaking that occurs due to airflow separation in conventional aircraft as they approach the stall. The stick shaking is intended to act as a backup to the auditory stall alert, in cases where the flight crew may be distracted.

Stick Pusher

A stick pusher is a device installed in some fixed-wing aircraft to prevent the aircraft from entering an aerodynamic stall. Some large fixed-wing aircraft display poor post-stall handling characteristics or are vulnerable to deep stall. To prevent such an aircraft approaching the stall the aircraft designer may install a hydraulic or electro-mechanical device that pushes forward on the elevator control system whenever the aircraft"s angle of attack reaches the pre-determined value, and then ceases to push when the angle of attack falls sufficiently. A system for this purpose is known as a stickpusher.

When a stall condition is detected an aural warning is activated and in most aeroplanes a stick pusher actuator is activated, which moves the control column forward. This causes the aeroplane to reduce its angle of attack and gain speed again. When it recovers out of the stall condition all warnings are deactivated to permit the pilots" continuing manual flight.

Figure 9.75 shows a pneumatically operated stick pusher system. This system is controlled by two separate stall warning computers. To activate the stick pusher, both computers must detect a high angle of attack.

The safety requirements applicable to fixed-wing aircraft in the transport category, and also to many military aircraft, are very demanding in the area of pre-stall handling qualities andstall recovery. Some of these aircraft are unable to comply with these safety requirements relying solely on the natural aerodynamic qualities of the aircraft. In order to comply with the requirements aircraft designers may install a system that will constantly monitor the critical parameters and will automatically activate to reduce the angle of attack when necessary to avoid a stall. The critical parameters include the angle of attack, airspeed, wing flap setting and load factor. Action by the pilot is not required to recognise the problem or react toit.

Aircraft designers who install stick pushers recognise that there is the risk that a stick pusher may activate erroneously when not required to do so. The designer must make provision for the flight crew to deal with unwanted activation of a stick pusher. In some aircraft equipped with stick pushers, the stick pusher can be overpowered by the pilot. In other aircraft, the stick pusher system can be manually disabled by the pilot.

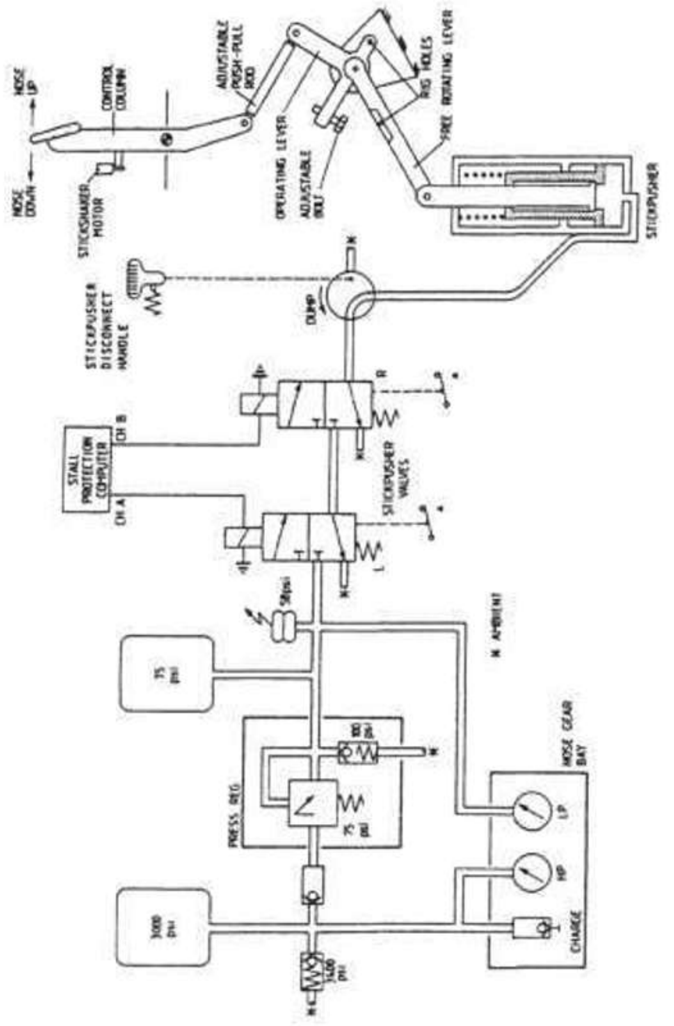
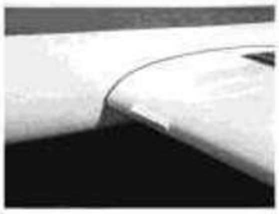


Fig. 9.75 Pneumatic Actuated Stick pusher System (Fokker 100)

Stall Strips

On some smaller aircraft, where no slats are installed, they can be equipped with stall strips at the wings" leading edge. The purpose of the strips is to produce a small vortex which reaches the horizontal stabilizer. If the angle of attack reaches an extreme angle, the pilot can feel the vibration caused by the vortex via the elevator surface.



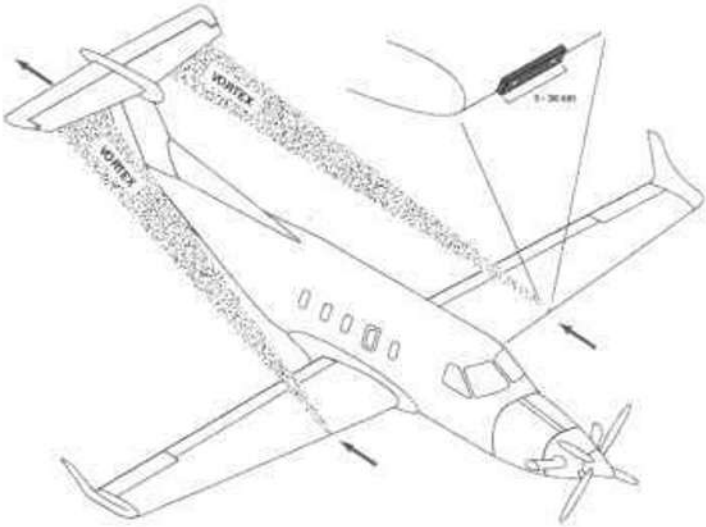


Fig. 9.76 Stall strips Fly By Wire Principle

The fly-by-wire system was designed and certified to render the new generation of aircraft even more safe, cost effective, and pleasant to fly.

Flight control surfaces are all

- Electrically-controlled, and
- Hydraulically-activated

The stabilizer and rudder can also be mechanically-controlled.

Pilots use side sticks to fly the aircraft in pitch and roll (and in yaw, indirectly, through turn coordination).

Computers interpret pilot input and move the flight control surfaces, as necessary to follow their orders.

However, when in "normal law", regardless of the pilot"s input, the computers will prevent excessive manoeuvres and exceedance of the safe envelope in pitch and roll axis. However, as on conventional aircraft, the rudder has no such protection.

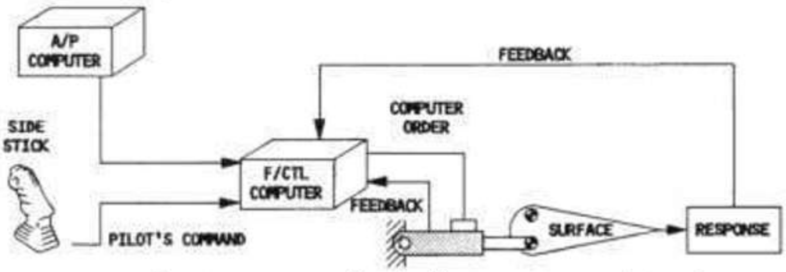


Fig. 9.77 FBW basic layout Control Surfaces

The flight controls are electrically or mechanically controlled as follows:

Pitch axis

- ElevatorElectrical
- Stabilizer Electrical for normal or alternatecontrol. Mechanical for manual trimcontrol

Roll axis

- AileronsElectrical
- SpoilersElectrical

Yaw axis

Rudder Mechanical, however control for yaw damping, turn coordination and trim is electrical.

Speed brakes

- Speed brakesElectrical

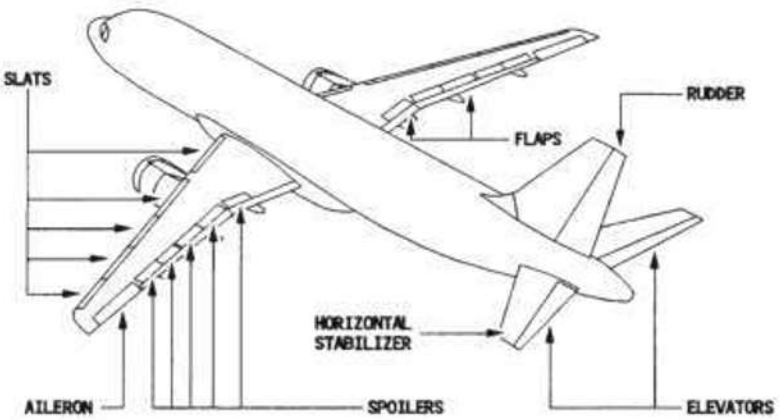


Fig. 9.78 Airbus A320 control surfaces

Note: All surfaces are hydraulically actuated Cockpit Controls
Each pilot has a sidestick controller with which to exercise manual control of pitch and roll. These are on their respective lateralconsoles.

The two side stick controllers are not coupled mechanically, and they send separate sets of signals to the flight control computers.

Two pairs of pedals, which are rigidly interconnected, give the pilot mechanical control of the rudder.

The pilots control speed brakes with a lever on the centre pedestal. The pilots use mechanically interconnected handwheels on each side of the centre pedestal to control the trimmable horizontal stabilizer.

The pilots use a single switch on the centre pedestal to set the rudder trim.
There is no manual switch for trimming the ailerons.

Computers

Seven flight control computers process pilot and autopilot inputs according to normal, alternate or direct flight control laws.

The computers are

2 ELACS (Elevator AileronComputer)
For: Normal elevator and stabilizer control. Aileron control

3 SECs (Spoilers Elevator Computer) For: Spoilerscontrol
Standby elevator and stabilizer control

2 FACs (Flight Augmentation Computer) For: Electrical rudder control

In addition 2 FCDC

Flight Control Data Concentrators (FCDC) acquire data from the ELACs and SECs and send it to the electronic instrument system (EIS) and the centralized fault display system (CFDS).

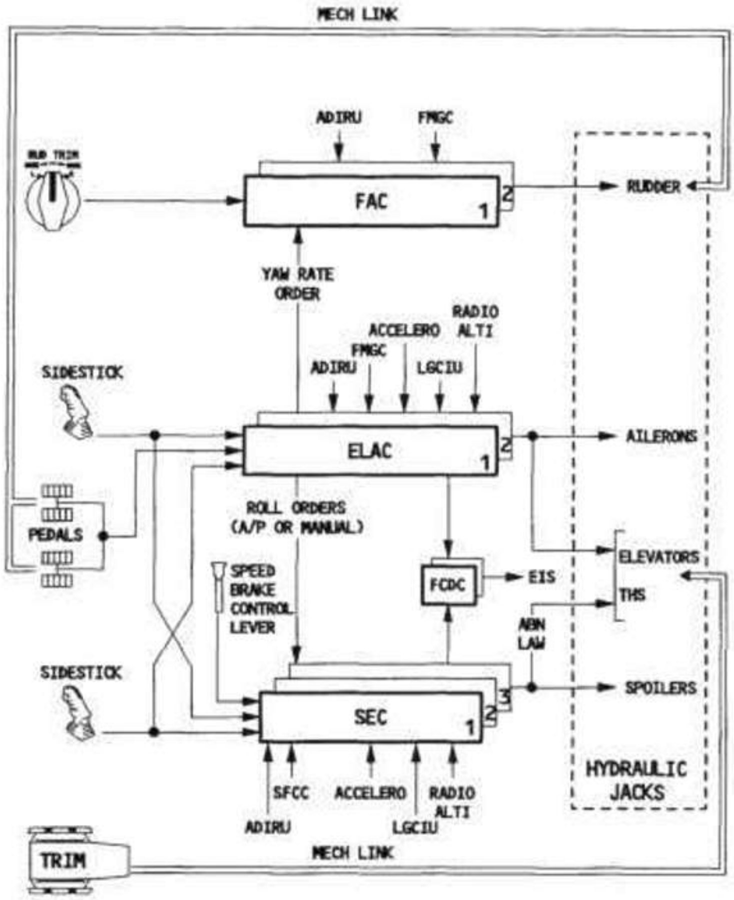


Fig. 9.79 FBW computer system Pitch Control

Two elevators and the Trimmable Horizontal Stabilizer(THS) control the aircraft in pitch. The maximum elevator deflection is 30° nose-up, and 11° nose-down. The maximum THS deflection is 13.5°nose-up, and 4°nose-down.

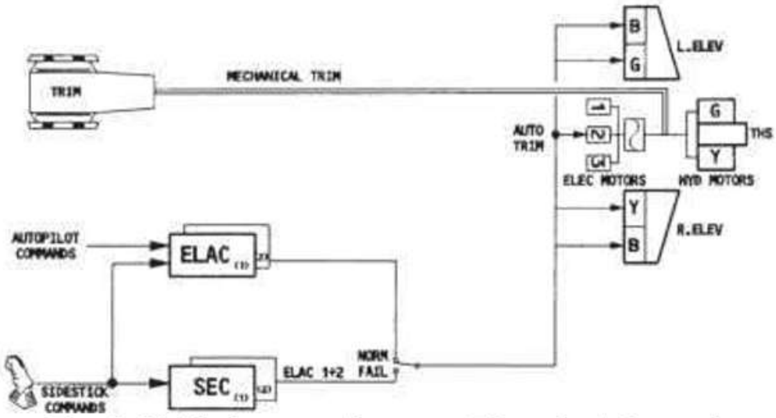


Fig. 9.80 Pitch control system Electrical Control

In normal operations, ELAC2 controls the elevators and the horizontal stabilizer, and the green and yellow hydraulic jacks drive the left and right elevator surfaces respectively. The THS is driven by N°1 of three electric motors.

If a failure occurs in ELAC2, or in the associated hydraulic systems, or with the hydraulic jacks, the system shifts pitch control to ELAC1. ELAC1 then controls the elevators via the blue hydraulic jacks and controls the THS via the N°2 electricmotor.

If neither ELAC1 nor ELAC2 is available, the system shifts pitch control either to SEC1 or to SEC2, (depending on the status of the associated circuits), and to THS motor N°2 or N°3.

Mechanical Control

Mechanical control of the THS is available from the pitch trim wheel at any time, if either the green or yellow hydraulic system is functioning.

Mechanical control from the pitch trim wheel has priority over electrical control.

Elevators

Two electrically-controlled hydraulic servojacks drive each elevator. Each servojack has three control modes.

- Active: The jack position is electrically-controlled.
- Damping: The jack follows surfacemovement.
- Centring: The jack is hydraulically retained in the neutral position.

In normal operation one jack is in active mode, the other jack is in damping mode. Somemanoeuvrescause the second jack to become active.

If the active servojack fails, the damped one becomes active, and the failed jack is automatically switched to the damping mode.

If neither jack is being controlled electrically, both are automatically switched to centringmode.

If neither jack is being controlled hydraulically, both are automatically switched to damping mode.

If one elevator fails, the deflection of the remaining elevator is limited in order to avoid putting excessive asymmetric loads on the horizontal tail plane or rear fuselage.

Stabilizer

A screwjack, driven by two hydraulic motors, drives the stabilizer. The two hydraulic motors are controlled by one of three electric motors, or the mechanical trim wheel.

Roll Control

One aileron and four spoilers on each wing control the aircraft about the roll axis. The maximum deflection of the ailerons is 25° The ailerons ext end 5°down when the flaps are extended (aileron droop). The maximum deflection of the spoilers is 35°.

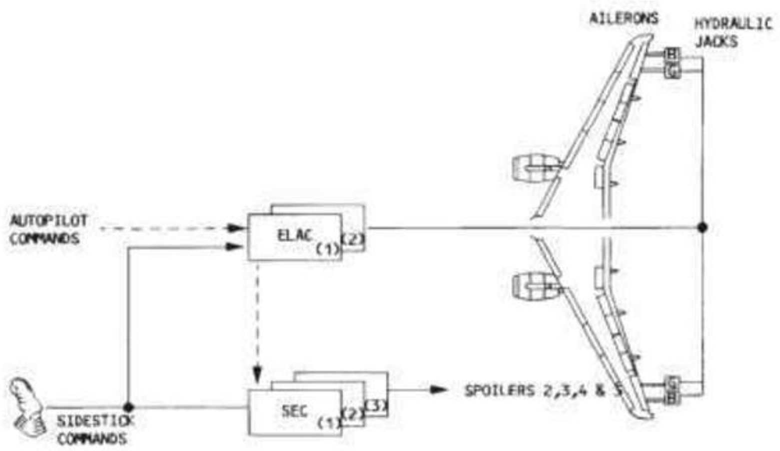


Fig. 9.81 Roll control system Electric Control

The ELAC 1 normally controls the ailerons. If ELAC1 fails, the system automatically transfers aileron control to ELAC2. If both ELACs fail, the ailerons revert to the damping mode.

SEC3 controls the N°2 spoilers, SEC1 the N°3 and 4 spoilers, and SEC2 the N°5 spoilers.If a SEC fails, the spoilers it controls are automatically retracted.

Actuation Ailerons

Each aileron has two electrically controlled hydraulic servojacks. One of these servojacks per aileron operates at a time. Each servojack has two control modes.

- Active: Jack position is controlledelectrically
- Damping: Jack follows surfacemovement

The system automatically selects damping mode, if both ELACs fail or in the event of blue and green hydraulic low pressure.

Spoilers

A servojack positions each spoiler. Each servojack receives hydraulic power from either the green, yellow, or blue hydraulic system, controlled by the SEC1, 2 or 3.

The system automatically retracts the spoilers to their zero position, if it detects a fault or loses electrical control.

If the system loses hydraulic pressure, the spoiler retains the deflection it had at the time of the loss, or a lesser deflection if aerodynamic forces push it down.

When a spoiler surface on one wing fails, the symmetric one on the other wing is inhibited.

Speed brakes and Ground Spoilers

Speed brake Control

The pilot controls the speedbrakes with the speed brake lever. The speedbrakes are actually spoilers 2, 3 and 4.

Speed brake extension is inhibited if:
- SEC1 and SEC3 both havefaults.
- An elevator (L or R) has a fault (in this case only spoilers 3 and 4 areinhibited).
- Angle-of-attack protection isactive.
- Flaps are in configurationFULL.
- Thrust levers above MCTposition
- Alpha flooractivation

If an inhibition occurs when the speedbrakes are extended, they retract automatically and stay retracted until the inhibition condition disappears and the pilots reset the lever. (The speedbrakes can be extended again 10 seconds or more after the lever is reset).

When a speedbrake surface on one wing fails, the symmetric one on the other wing is inhibited.

Note: 1. For maintenance purposes, the speedbrake lever will extend the N° 1surfaces when the aircraft is stopped on the ground, whatever the slat/flapconfiguration.

When the aircraft is flying faster than 315 knots or Mach 0.75 with the autopilot engaged, the speedbrake retraction rate is reduced (Retraction from FULL to in takes about 25seconds).

The maximum speedbrake deflection in manual flight is 40°for

spoilers 3 and 4, 20° for spoiler 2.

The maximum speedbrake deflection with the autopilot engaged is:
25°for spoilers 3 and 4, 12.5°for spoilers 2.

The maximum speedbrake deflection with the autopilot engaged is achieved with half speedbrake lever deflection.

For these surfaces (which perform both roll and speedbrake functions) the roll function has priority. When the sum of a roll order and a simultaneous speedbrake order on one surface is greater than the maximum deflection available in flight, the same surface on the other wing is retracted until the difference between the two surfaces is equal to the roll order.

Ground Spoiler Control

Spoilers 1 to 5 act as ground spoilers. When a ground spoiler surface on one wing fails, the symmetric one on the other wing is inhibited

Arming

The pilot arms the ground spoilers by pulling the speedbrake control lever up into the armed position.

Full extension

The ground spoilers automatically extend during rejected takeoff, at a speed greater than 72 knots, or at landing when both main landing gears have touched down, when ground spoilers are

armed and all thrust levers are at or near idle, or Reverse is selected on at least one engine (other thrust lever at or near idle), if ground spoilers were notarmed.

Note: In autoland, the ground spoilers fully extend at half speed one second after both main landing gear touch-down.

The spoiler roll function is inhibited when spoilers are used for the ground spoiler function.

Partial extension

The ground spoilers partially extend (101) when reverse is selected on at least one engine (other engine at idle), and one main landing gear strut is compressed. This partial extension, by decreasing the lift, eases the compression of the second main landing gear strut, and consequently leads to full ground spoiler extension.

Retraction

The ground spoilers retract after landing, or after a rejected takeoff, when the ground spoilers are disarmed.

Note: If ground spoilers are not armed, they extend at the reverse selection and retract when idle is selected.

Note: After an aircraft bounce, the ground spoilers remain extended with the thrust levers at idle.

The landing gear touchdown condition is triggered for both main landing gear either when their wheel speed is greater than 72

knots, or when their landing gear struts are compressed and theradio altitude is very low (RA < 6 feet).

For the ground spoiler logic, idle signifies thrust lever position < 4°or < 15°when below 10 ft.

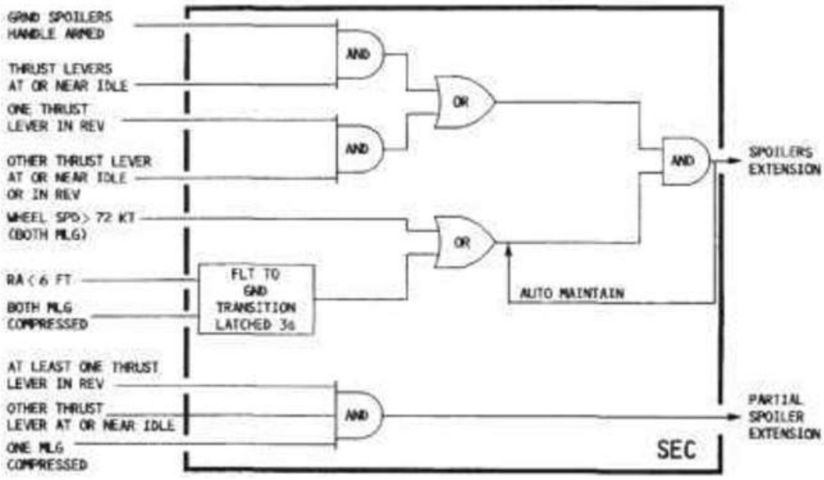


Fig. 9.82 Ground spoiler logic diagram Yaw Control

One rudder surface controls yaw

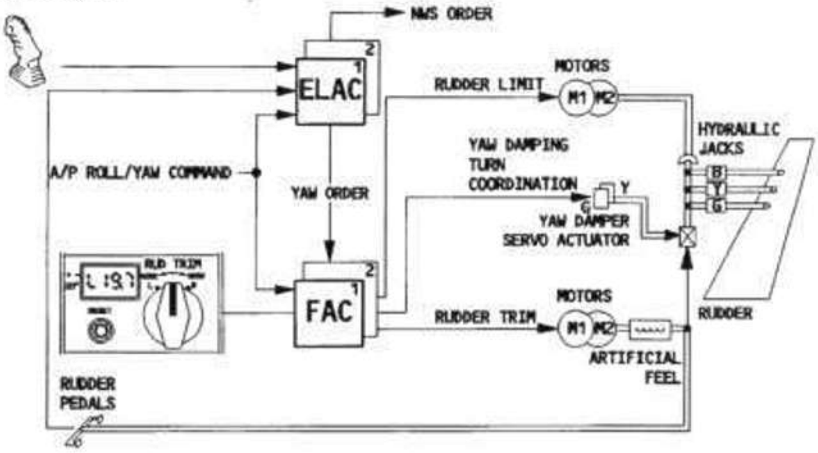


Fig. 9.83 Yaw control system Electrical Rudder Control

The yaw damping and turn coordination functions are automatic. The ELACs compute yaw orders for coordinating turns and damping yaw oscillations, and transmit them to the FACs.

Mechanical Rudder Control

The pilots can use conventional rudder pedals to control the rudder.

Rudder Actuation

Three independent hydraulic servojacks, operating in parallel, actuate the rudder. In automatic operation (yaw damping, turn coordination) a green servo actuator drives all three servojacks.

A yellow servo actuator remains synchronized and takes over if

there is a failure. There is no feedback to the rudder pedals from the yaw damping and turn coordination functions.

Protections

The normal law protects the aircraft throughout the flight envelope, as follows:

- load factorlimitation
- pitch attitudeprotection

- high-angle-of-attack (AOA)protection
- high-speedprotection

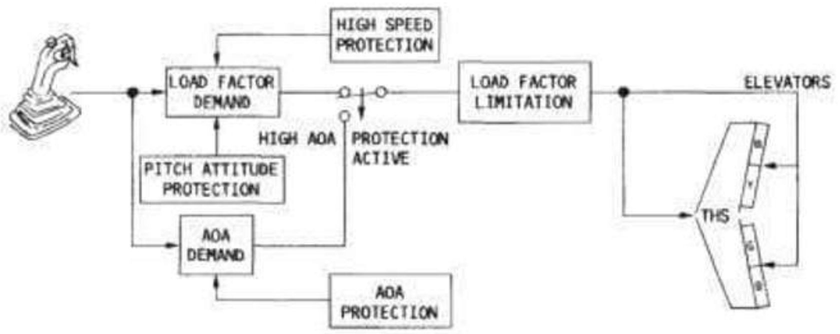


Fig. 9.84 FBW protection system Load Factor Limitation

The load factor is automatically limited to:

- +2.5 g to -1 g for cleanconfiguration.
- +2 g to 0 g for otherconfigurations

Pitch Attitude Protection

Pitch attitude is limited to:

- 30°nose up in conf. 0 to 3 (progressively reduced to 25°at low speed).
- 25° nose up in conf. FULL (progressively reduced to 20°at lowspeed).
- 13°nosedown(indicatedbygreensymbol“=“onthePFDs

pitch scale).

The flight director bars disappear from the PFD when the pitch attitude exceeds 25° up or 13°down. They return to the display when the pitch angle returns to the region between 22°up and 10°down.

High Angle of Attack Protection

Under normal law, when the angle of attack becomes greater than aprot, the system switches elevator control from normal mode to a protection mode, in which the angle of attack is proportional to sidestick deflection. That is, in the aprot range, from aprot to amax, the sidestickscommands directly. However, the angle of attack will not exceed amax, even if the pilot gently pulls the sidestick all the way back. If the pilot releases the sidestick, the angle of attack returns to aprot and staysthere.

This protection against stall and winds hear has priority over all other protections. The autopilot disconnects at aprot +1°.

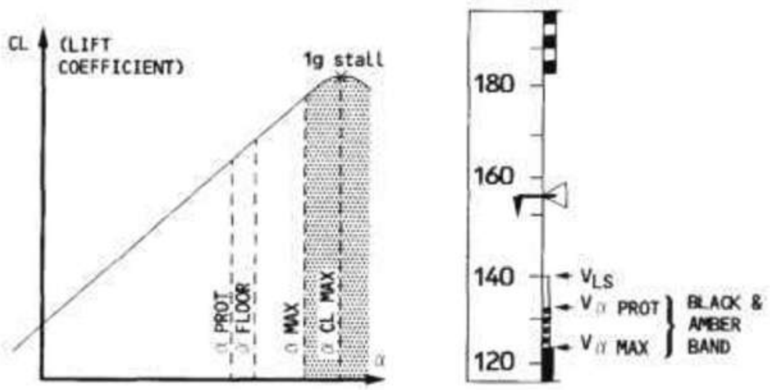


Fig. 9.85 Angle of attack protection

Vaprot, Vafioor, Vamax vary according to the weight and the configuration.

To deactivate the angle of attack protection, the pilot must push the sidestick:
- More than 8°forward,or
- More than 0.5°forward for at least 0.5 seconds, when a<amax

In addition, below 200 feet, the angle of attack protection is also deactivated, when
- Sidestick deflection is less than half nose-up, and
- Actual a is less than aprot-2°.

The Qfioor function is available from lift-off to 100 feet RA beforelanding.

High-Speed Protection

The aircraft automatically recovers following a high speed upset. Depending on the flight conditions (high acceleration, low pitch attitude), the High Speed Protection is activated at/or above Vmo/Mmo.

When it is activated, the pitch trim is frozen. Positive spiral static stability is introduced to 0° bank angle (instead of 33°in normal law), so that with the sidestick released, the aircraft always returns to a bank angle of 0° The bank angle limit is reduced from 67°to 45°. As the speedincreases above Vmo/Mmo, the sidestick nose- down authority is progressively reduced, and a permanent nose-up order is applied to aid recovery to normal flight conditions.

The High Speed Protection is deactivated when the aircraft speed decreases below Vmo/Mmo, where the usual normal control laws are recovered.

The autopilot disconnects when high speed protection goes active.

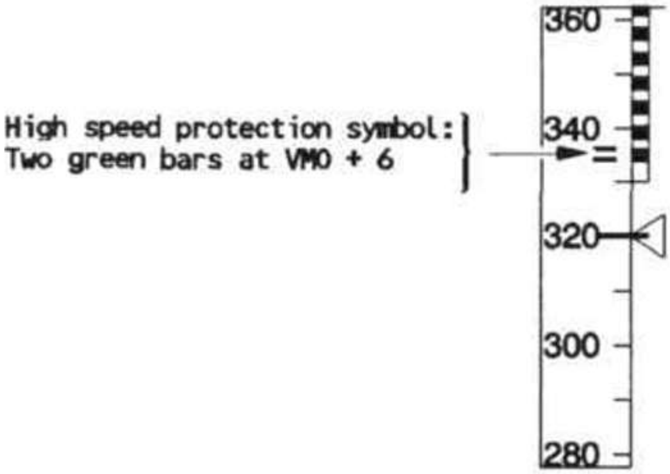


Fig. 9.86 High speed protection

Note: The ECAM displays an “0/SPEED” warning at VMo + 4 knots and MMo + 0.006.

Normal Law

When the aircraft is on the ground (in "on ground” mode), the sidestick commands the aileron and roll spoiler surface deflection. The amount of control surface deflection that results from a given amount of sidestick deflection depends upon aircraft speed. The pedals control rudder deflection through a direct mechanical linkage.

When the aircraft is in the “in flight” mode, normal law combines control of the ailerons, spoilers (except N°1 spoilers), and rud der (for turn coordination) in the sidestick. While the system thereby gives the pilot control of the roll and heading, it also limits the roll rate and bank angle, coordinates the turns, and damps the Dutch roll.

The roll rate requested by the pilot during flight is proportional to the sidestick deflection, with a maximum rate of 15°per second when the sidestick i s at the stop.

When the aircraft is in “flare” mode, the lateral control is the same as in "in flight” mode.

Bank Angle Protection

Inside the normal flight envelope, the system maintains positive spiral static stability for bank angles above 33° If the pilot releases the sidestick at a bank angle greater than 33° the bank angle automatically reduces to 33°. Up to 33°, the system holds the roll attitude constant when the sidestick is at neutral. If the pilot holds full lateral sidestick deflection, the bank angle goes to 67°(indicated by a pair of green bar lines “=“ on the PFD) and no further.

If the angle-of-attack protection or high speed protection is operative, the bank angle goes to 45°and no further, if the pilot holds full lateral sidestick deflection. If high speed protection is operative, the system maintains positive spiral static stability from a bank angle of 0° so that with the sidestick released, the aircraft always returns to a bank angle of 0°

When bank angle protection is active, auto trim is inoperative.

If the bank angle exceeds 45° the autopilot disconnects and the FD bars disappear. The FD bars return when the bank angle decreases to less than 40°.

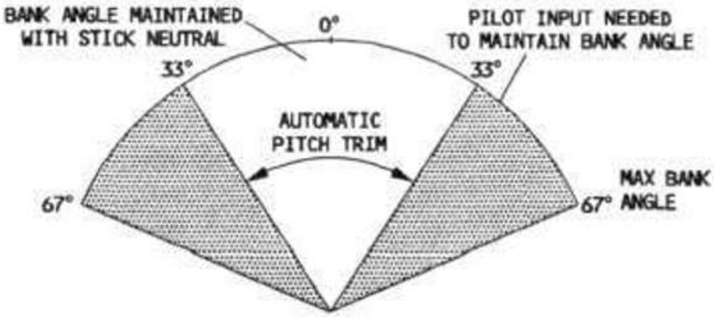


Fig. 9.87 Bank angle protection

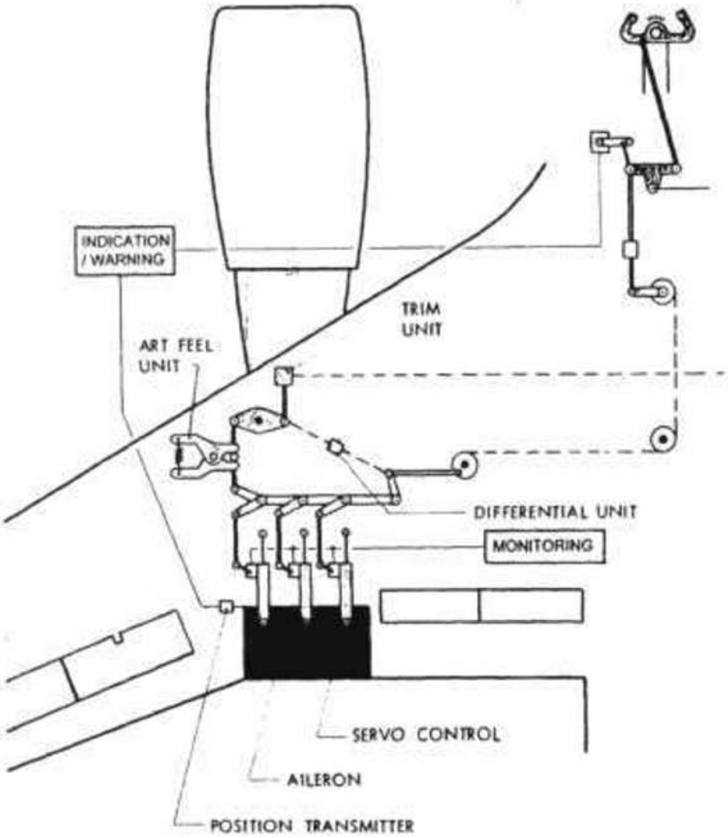


Fig. 9.88 Redundant Hydraulic Power Operated System

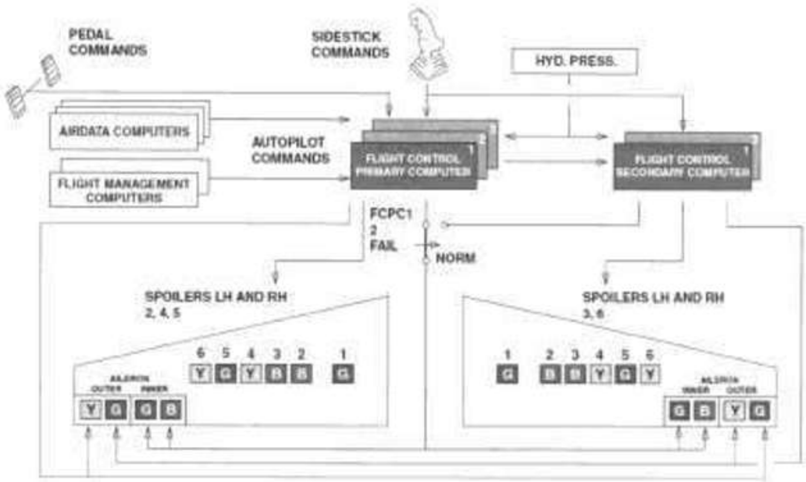


Fig. 9.89 Redundancy of Computers and Hydraulic Systems

Module 11.10: FUEL SYSTEMS (ATA 28)

Requirements General

All powered aircraft depend for their operation on the continuous flow of uncontaminated fuel under all operating conditions. Theweight of the fuel constitutes a good percentage of the total weight of the aircraft. This may range from about 10% of the gross weight of small personal aeroplanes, to more than 40% of the gross weight for some business jet aircraft used on long overseas flights.

The weight of the fuel means that the structure must be strong enough to carry it in all flight conditions. The fuel tanks must also be located such that the decreasing weight of the fuel will not cause balance problems as the fuel is beingused.

There have been more aircraft accidents caused by the improper management of the fuel system than those caused by failures of any other single system. Engine failure may be caused by using all of the fuel in tanks, but engines will also stop if an empty tank is selected when there is fuel in the other tanks.

Contamination in the fuel may clog strainers and shut off the flow of fuel to the engines. Water that condenses in partially filled tanks will stop the engine when it flows into the metering system. Water in turbine-powered aircraft is a special problem, as the more viscous jet fuel will hold water entrained in such tiny particles that it does not easily settle out. When the fuel temperature drops at high altitude, the water may form ice crystalswhichcanfreezeonthefuelfiltersandshutofftheflow

of fuel to the engine.

Regulatory Requirements

EASA Certification Specification CS-25. 25.951 - 25.1001

The requirements for the fuel system design are specified in detail in the EASA Certification Specifications under which the aircraft was built. Since the vast majority of aeroplanes in the general aviation fleet are built under EASA/FAR Part 23, "Airworthiness Standards: Normal, Utility, and Acrobatic Category Aeroplanes" and EASA/FAR Part 25 "Large Aircraft" we will list a few of the more basic requirements for the fuel system of theseaeroplanes.

- No pump can draw fuel from more than one tank at a time, and provisions must be made to prevent air from being drawn into the fuel supplyline.
- Turbine-powered aircraft must be capable of sustained operation when there is at least 0.75 cc of free water per gallon of fuel, and the fuel is cooled to its most critical condition for icing. The system must incorporate provisions to prevent the water which precipitates out of the fuel freezing on the filters and stopping fuel flow to theengine.
- Each fuel system of a multi-engine aircraft must be arranged in such a way that the failure of any one component (except the fuel tank) will not cause more than one engine to lose power.
- If multi-engine aircraft feed more than one engine from a single tank or assembly of interconnected tanks, each engine must have an independent tank outlet with a fuel shut off valve at thetank.
- Tanks used in multi-engine fuel systems must have two vents arranged so that they are not likely to both become pluggedat the sametime.

- Filler caps must be designed so that they are not likely to be installed incorrectly or lostin-flight.
- The fuel systems must be designed to prevent the ignition of fuel vapours bylightning.
- A gravity feed system must be able to flow 150% of the takeoff fuel flow when the tank contains the minimum fuel allowable, and when the aeroplane is positioned in the attitude that is most critical for fuelflow.
- A pump feed fuel system must be able to flow 125% of the takeoff fuel flow required for a jetengine.

- If the aircraft is equipped with a selector valve that allows the engine to operate from more than one fuel tank, the system must not cause a loss of power for more than ten seconds for a single-engine or twenty seconds for a multi-engine aeroplane, between the time one tank is allowed to run dry and the time at which the required power is supplied by the othertank.
- Turbine-powered aircraft must have a fuel system that will supply 100% of the fuel required for its operation in all flight attitudes, and the flow must not be interrupted, as the fuel system automatically cycles through all of the tanks or fuel cells in thesystem.
- If a gravity feed system has interconnected tank outlets, it should not be possible for fuel feeding from one tank to flow into another tank and cause it tooverflow.
- The amount of unusable fuel in an aircraft must be determined and this must be made known to the pilot. Unusable fuel is the amount of fuel in a tank when the first evidence of malfunction occurs. The aircraft must be in the attitude that is most adverse for fuelflow.
- The fuel system must be so designed that it is free from vapour lock when the fuel is at a temperature of 45'C under the most critical operatingconditions.

- Each fuel tank compartment must be adequately vented and drained so no explosive vapours or liquid canaccumulate.
- No fuel tank can be installed inside a personnel compartment of a multi-engineaircraft.
- Each fuel tank must have a 2% expansion space that cannot be filled with fuel, and it must also have a drainable sump where water and contaminants will normally accumulate when the aircraft is in its normal groundattitude.
- Provisions must be made to prevent fuel spilled during filling the tank from entering the aircraftstructure.
- The filler opening of an aircraft fuel tank must be marked with the word “FUEL”. For turbine-powered aircraft, the tank must be marked with the permissible fueldesignation.
- If the filler opening is for pressure fuelling, the maximum permissible fuelling and defuelling pressure must be specified.
- If more than one fuel tank has interconnected outlets, the airspace above the fuel must also beinterconnected.
- If the carburettor or fuel injection system has a vapour elimination system that returns fuel to one of the tanks, the returned fuel must go to the tank that is required to be used first.
- All fuel tanks are required to have a strainer at the fuel tank outlet or at the booster pump. For a reciprocating engine, the strainer should have an 8 to 16-mesh element, and for turbine engines, the strainer should prevent the passage of any object that could restrict the flow or damage any of the fuel system components.
- For engines requiring fuel pumps, there must be one engine driven fuel pump for eachengine.
- There must be at least one drain that will allow safe drainage of the entire fuel system when the aeroplane is in its normal ground attitude.

- If the design landing weight of the aircraft is less than that permitted for takeoff, there must be provisions in the fuel system for jettisoning fuel to bring the maximum weight down to the design landingweight.
- The fuel jettisoning valve must be designed to allow personnel to close the valve during any part of the jettisoningoperation.

Critical Design Configuration Control Limitations (CDCCLs)

Many maintenance tasks described in Chapter 28 (Fuel Systems) of the Aircraft Maintenance Manual (AMM) are designated as “CDCCL”.

Design features that are CDCCLs are defined and controlled by the FAA and EASA (see Part 145.A.45 and the associated AMC). CDCCLs are a means of identifying certain design configuration features intended to preclude a fuel tank ignition source for the operational life of theaeroplane.

CDCCLs are mandatory and cannot be changed or deleted without the approval of the Authority that is responsible for the aeroplane Type Certificate, or applicable regulatory agency. A critical fuel tank ignition source prevention feature may exist in the fuel system and its related installation or in systems that, if a failure condition were to develop, could interact with the fuel system in such a way that an unsafe condition would develop without this limitation. Strict adherence to configuration, methods, techniques, and practices as prescribed is required to ensure the CDCCL is complied with. Any use of parts, methods, techniques or practices not contained in the applicable CDCCL must be approved by the Authority that is responsible for the aeroplane Type Certificate, or applicable regulatory agency.

Turbine Engine Transport Aircraft Fuel Systems

The purpose of the fuel system is it to store a needed amount of fuel in the tanks and deliver a constant amount of fuel under pressure to one or more engines. In larger long range aircraft, additional tanks can be installed e.g. in the horizontal stabilizer for centre of gravity control.

In general, the fuel system consists of the following subsystems: storage system and fuel ventilation

- fuel feedsystem
- refuelling and defuellingsystem
- fuel quantity indicationsystem
- fuel jettison (dump)system
- longitudinal trim (centre of gravitycontrol)

Fuel Tanks General

The fuel that an aeroplane needs is stored in tanks located in the wings, in the fuselage and on larger aircraft also in the horizontal stabilizer as shown in Figure 10.1.

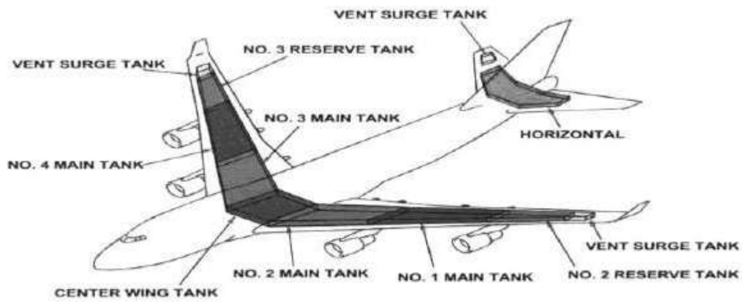


Fig. 10.1 Tank locations

Some types of aircraft have an auxiliary tank installed in front and/or the rear of the centre wing tank. For most aircraft, the number of engines determines the number of main tanks present in an aeroplane. In general, an aeroplane with two engines will have two main tanks; an aeroplane with three engines will have three main tanks, etc. Each engine is supplied with fuel from its own main tank. The auxiliary power unit is generally supplied with fuel from a main tank. If auxiliary tanks are filled with fuel, the rule generally applies that these should be emptied as quickly as possible by pumping the fuel to the main tanks.

Fuel Tank Construction

The location, size, shape, and construction of fuel tanks vary with the type and intended use of theaircraft.

Fuel tanks are manufactured from materials that will not react chemically with any aviation fuel and have a number of common features. Usually sumps and drains are provided at the lowest point in the tank, and the top of each tank is vented to the atmosphere. All except the smallest of tanks are fitted with baffles to resist fuel surging caused by changes in the attitude of the aircraft. An expansion space is provided in fuel tanksto allow for an increase in fuel volume due toexpansion.

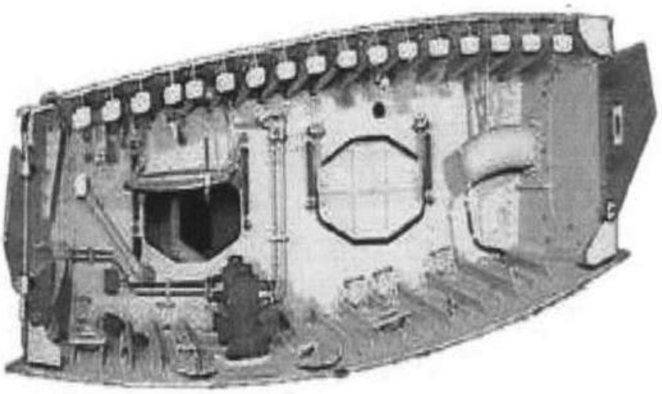
Some fuel tanks are equipped with dump valves that make it possible to jettison fuel during flight in order to reduce the weight of the aircraft to its specified landing weight. In aircraft equipped with dump valves, the operating control is located within reach of the pilot, co-pilot or flight engineer. Dump valves are designed and installed to afford safe, rapid discharge offuel.

The main tanks are of the integral type. Centre wing tanks are usually of the integral type. In some types of aircraft, they are of

the bladder type.

Integral Fuel Tanks

Rigid tanks require a large open space in the aircraft structure for their installation, and very few aircraft have this amount of space that is not crossed with structural members. Most wings have large empty spaces, however, and with the availability of some of the new space-age sealants, it has become standard practice for many of the aircraft manufacturers to seal off a portion of the wing to form a fuel tank. This



type of tank has the advantage of using the maximum amount of space for the fuel and of Fig. 10.2 Integral tank inner view

having a minimum amount of weight. This is the leading edge portion of the wing from the front spar forward, and it is sealed at both ends and all along the spar with a two-part sealant. All of the rivets and nutplates are sealed, and sealant is used around all of the inspection openings. The sealant is spread along each seam individually rather than sloshing the entire tank. See Figure10.3.

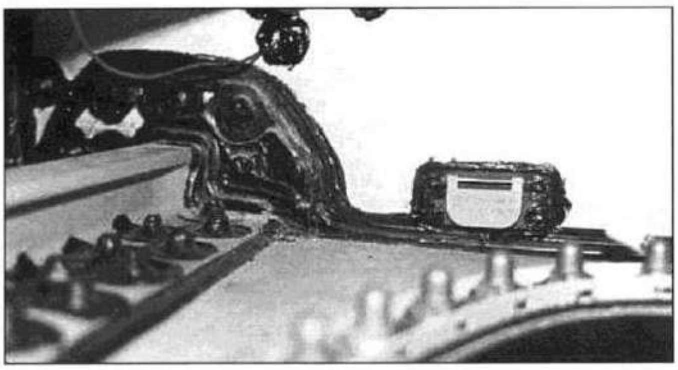


Fig. 10.3 Sealed seams

Fuel flows freely inboard towards the wing roots, but is restricted from flowing outboard by gravity-closed flapper valves located in the wing ribs. These are simple one way valves which prevent the fuel from surging towards the wing tip when the aircraft rolls.

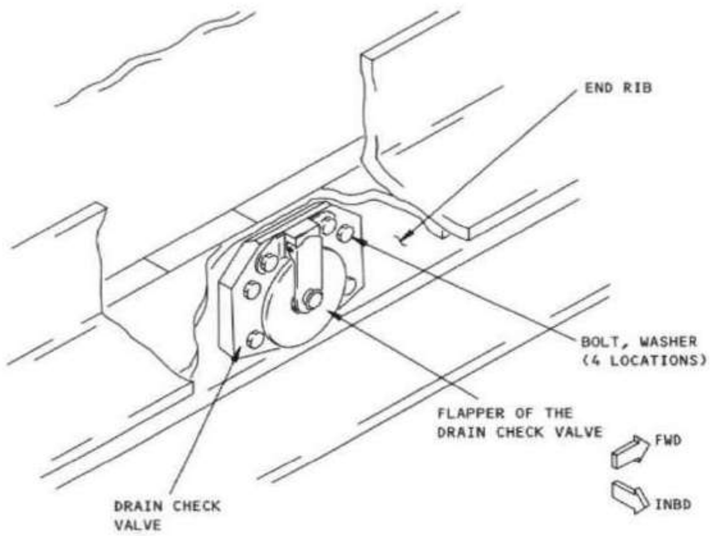


Fig.10.4 Fuel tank flapper valve

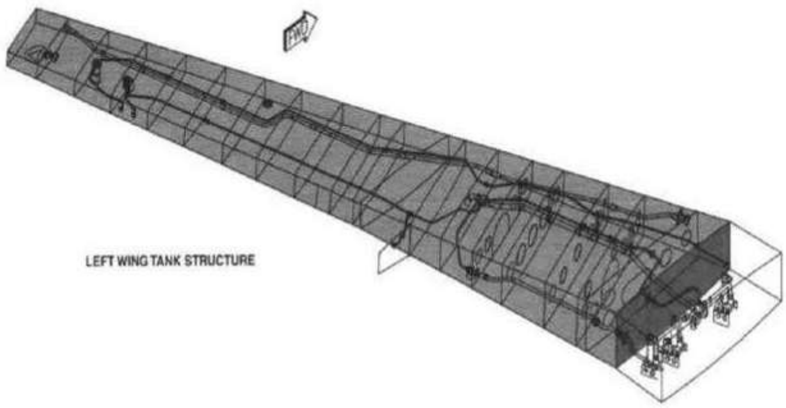


Fig. 10.5 Integral fuel tank structure Tank Access

For maintenance purposes, panels are provided to get access to fuel lines, components, sensors etc. Most of them are usually on the lower side of the tank, some on the upper side. The methods of sealing differ from type of aircraft. But the most common sealing methods are:

- O-Rings
- Gaskets

To simplify installation of components like overpressure protectors or NACA air intakes, such components are often attached to an access panel.

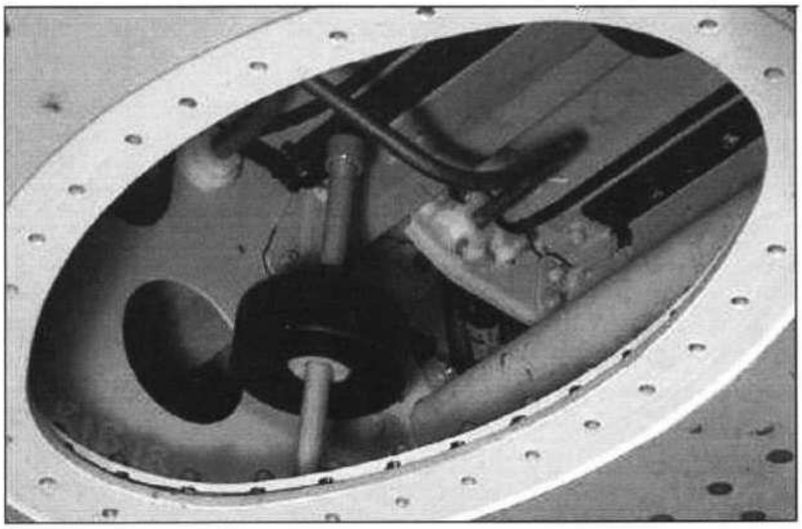


Fig.10.6 View inside a fuel tank through an open tank access Access Panel Installation

Several installation and sealing methods are common on integral tanks. This means, that panels can be installed from in- or outside. Also the sealing methods differ. The most popular these days is the O-ring type panel. The gasket type is not very often found but is still in service. The various methods are shown in Figure 10.7. When installing a removed tank access panel, a new gasket/O-ring must be installed and the screws evenly tightened in an order given in themanual.

Beside cleanliness during installation, it is important, to install the correct bolt length as described in the illustrated parts catalogue (IPC). Otherwise damage to the panel or dome nut will occur resulting in aleak.
Always apply the torque value as given in the maintenance manual. (Tank access panels are structural members of the wing.)

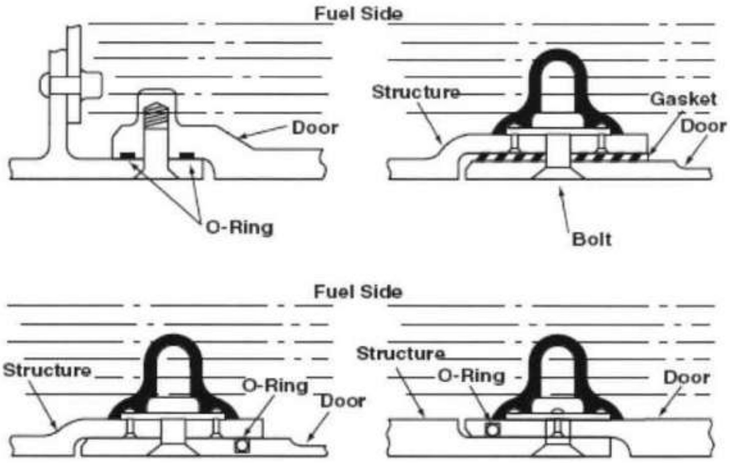


Fig. 10.7 Panel installations

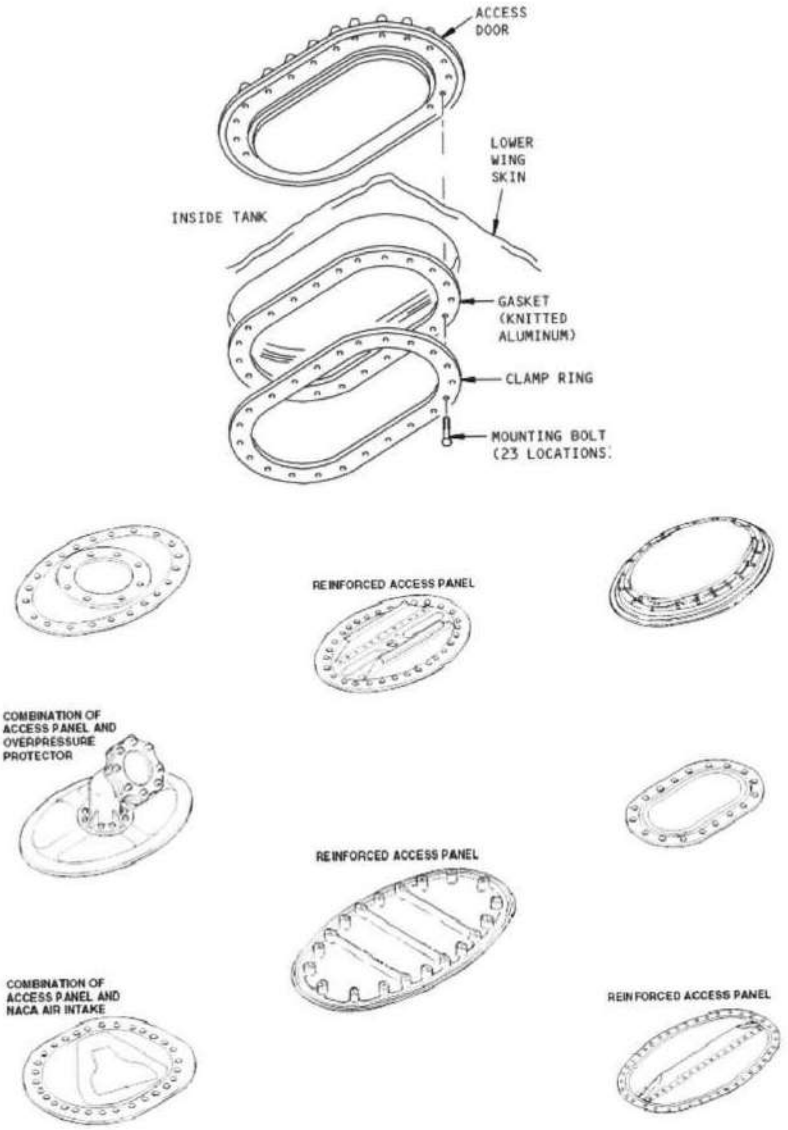


Fig. 10.8 Panel installations

Bladder Tanks

An excellent substitute for a welded fuel tank is the bladder tank that has been successfully used for both small and large aircraft. The fuel bay is prepared by covering all sharp edges of the metal structure with a chafe-resisting tape and installing a bladder made of thin fabric, impregnated with neoprene or some similar material that is impervious to thefuel.

The bladder is put into the cavity prepared for it, by folding it and inserting it through an inspection opening. Then it is snapped or clipped in place, or in some instances, it is laced to the structure. An opening in the bladder is then secured to the inspection opening and it is covered with an inspectionplate.

These tanks should never be allowed to stand empty for an extended period of time. If it is necessary, the inside of the bladder should be treated according to the maintenance manual.

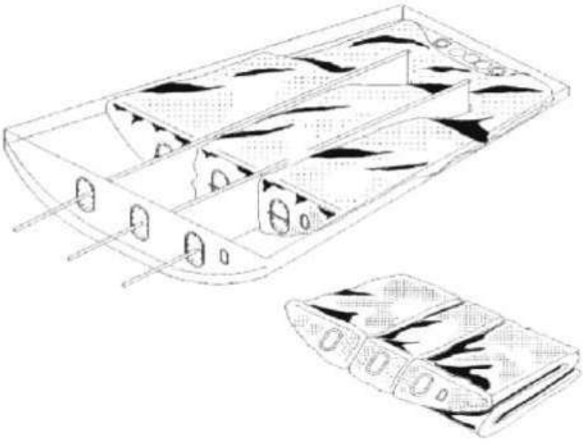


Fig. 10.9 Bladder tank

The ventilation system has been designed in such a way that,

damage to the fuel tanks by underpressure or over-pressure. Regulation by EASA/FAR CS-23 and CS-25 says that each fuel tank must have an expansion space of not less than 2% of the tank capacity, unless the tank vent discharges clear of the aeroplane.
This means that inflow and outflow of air is and always must be possible;
- when using thefuel;
- when refuelling anddefuelling.

when putting too much fuel in the tank (overflow), the fuel flows to the vent surge tanks via this system. If fuel exists in the surge tanks, they will be emptied when the engines arerunning.

The tank venting system provides positive venting of the tanks during flight. A ram air intake maintains a slight positive pressure in the vent system, thus decreasing fuel vaporization, and preventing negative pressures in the tanks through changes in aircraft attitude and fuel usage. In some aircraft, the vent system also prevents the building up of dangerous pressures in thetanksduringrefuelling,shouldtheautomaticcut-offfail,by dumping excess fuel. Generally, there are two vent pipes in each tank, the inboard vent is open-ended, but the outboard vent is fitted with a float valve, the purpose of which is to minimize fuel transfer both between tanks and into the vent/surge tank during changes of aircraft attitude. Fuel which is spilled into the venting system, collects in the vent/surge tank. On some aircraft the vent/surge tank drains under gravity into the main tanks, but on other aircraft an automatic pumping system is used. The pumping system may operate on a continuous basis, using 'jet1 pumps, or on an intermittent basis using float switches and a separate electrically-operated pump. In a jet pump, output from a normal booster pump passes through a jet nozzle, which is contained within a concentric pipe leading from the vent/surge tank. The flow of fuel through the jet nozzle automatically draws fuel from the vent/surge tank. With an intermittent system, a high-level float switch switches the transfer pump on, thus transferring fuel from the vent/surge tank to a main tank, and a low-level float switches the transfer pump off. A time-delay may be incorporated in the pump circuit, to prevent intermittent operation as a result of fuel surge.

Vent valves are generally either a caged cylindrical float which itself acts as a valve to close the vent, or a simple lever-type flap valve; typical examples are shown in Figure 10.12.

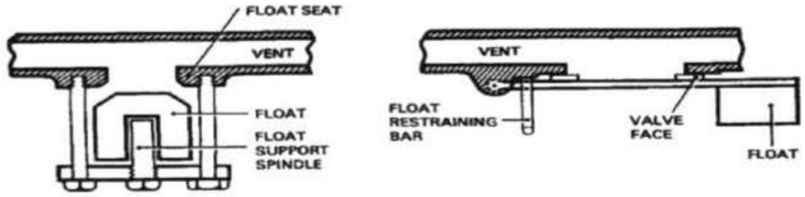


Fig.10.12 Vent valves

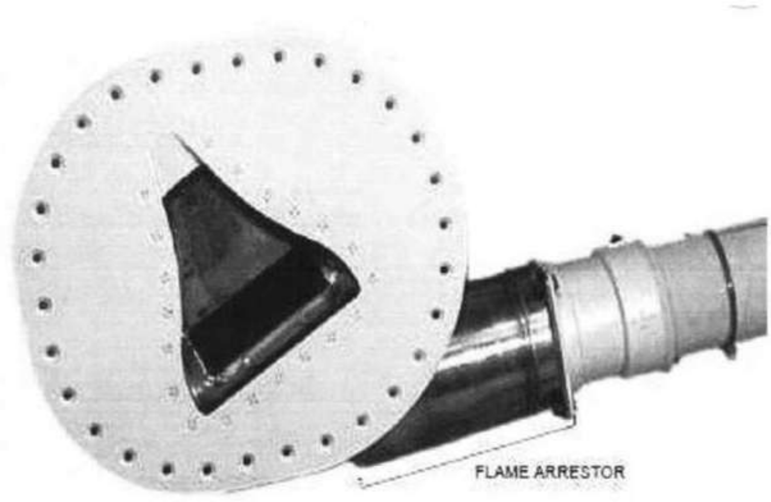


Fig. 10.13 NACA air intake with flame arrestor

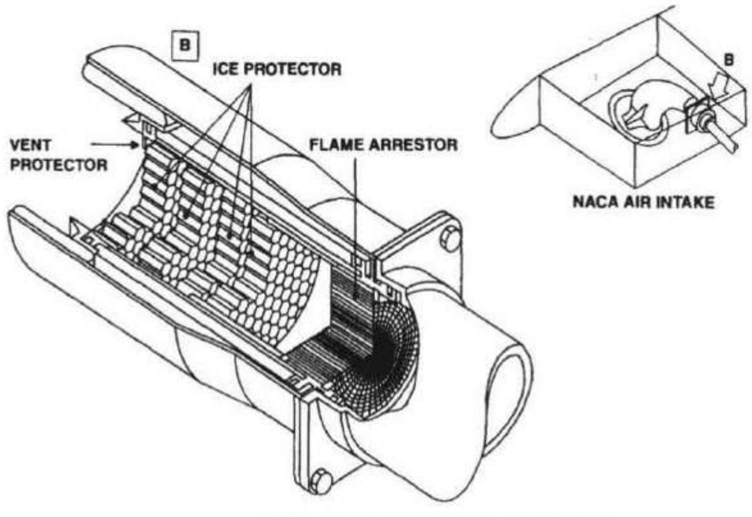


Fig. 10.14 Flame arrestor

Float Valve

The float valves have a float that senses the fuel level in the tank. In normal flight, the valve near the wing tip is open (the float is down). If an uncoordinated manoeuvre occurs, the fuel goes to the lower wing tip. Then the float valve senses the fuel level and closes itself. The other valve installed near the wing root opens. This keeps the tank open to the air and the tanks pressure differential at a permitted limit. Check valves ensure that the vent lines will be emptied back to the tank if ever fuel should be present in them.

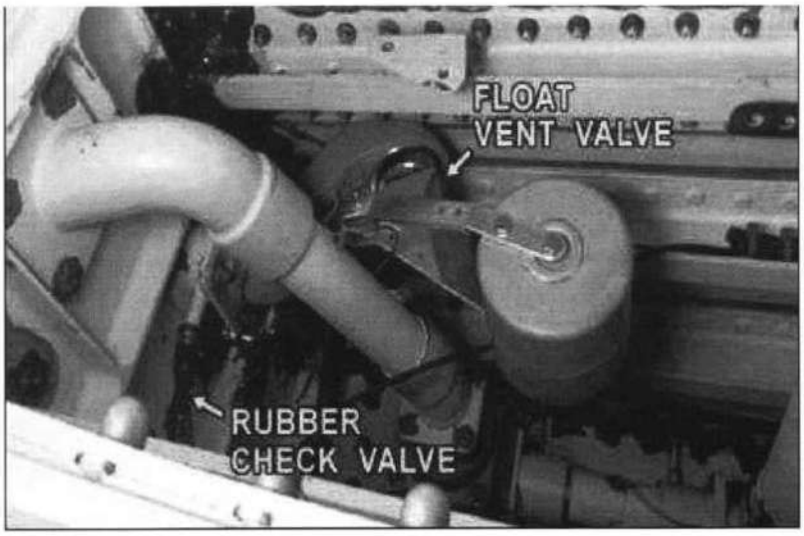


Fig. 10.15 Float valve

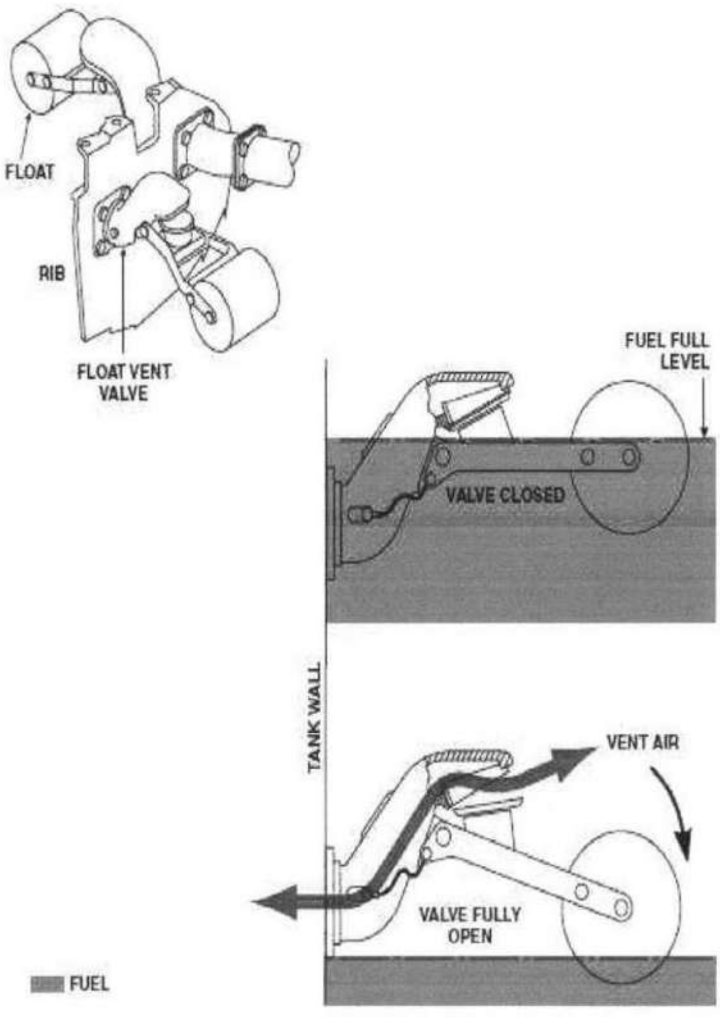


Fig.10.16 Float valve function

Burst Protection

In each vent surge tank, pressure relief valves and disks are usually installed. They open or break when the flame arrestor is clogged and the pressure difference (positive or negative) between atmospheric pressure and tank pressure becomes toolarge. Figure 10.17 shows a relief valve from one tank to another and a burst disk where the fuel or the air goes usually overboard. Where burst disks are used to relieve overboard, the disk is usually marked with a cross for better viewing from ground

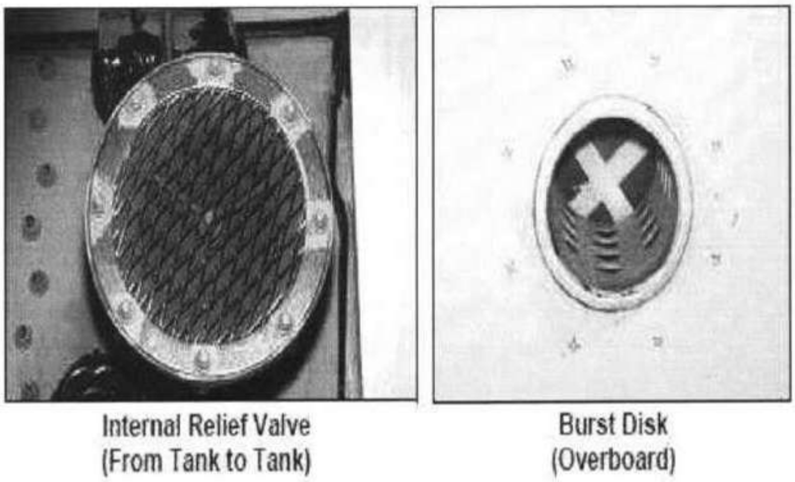


Fig.10.17 Relief valve and burst disk Tank Draining

Draining of sediment and fuel residue from the tank is done via a tank drain valve as shown in

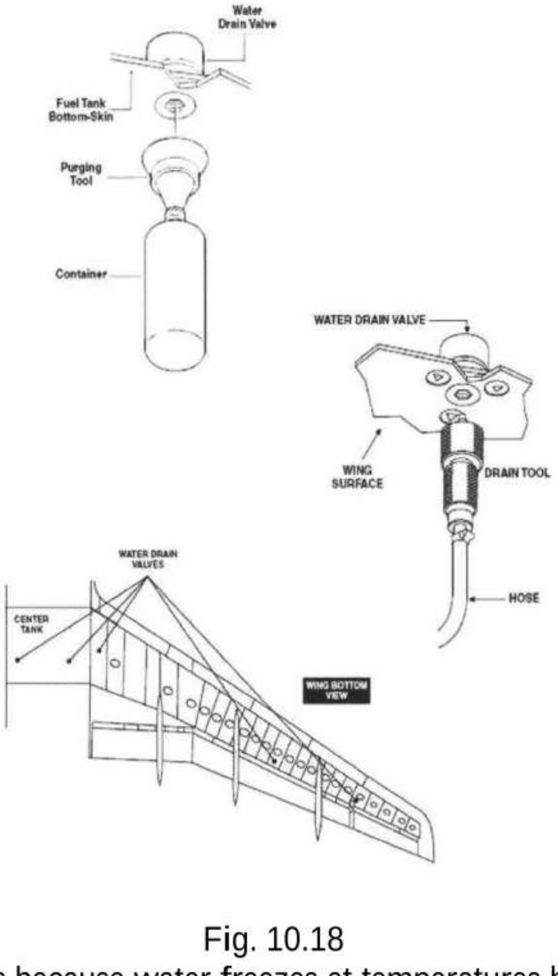


Fig. 10.18

It is also necessary to drain tanks because water freezes at temperatures below 0*0. The tank drain valve can be found on the underside of each wing or tank. Since water has a higher density than fuel, it will accumulate at the lowest point of the tank. Therefore, the valves are usually located at the lowest point of the tank. Where drain valves are not placed on the lowest point, indirect drain valves are used. To drain water properly from the tank, a vacuum apparatus must beused.

To properly drain any existing water from a fuel tank it is important to let the water settle for a certain time. As described before, it will accumulate at the lowest points in thetanks.

Drain valves are usually equipped with a check valve. This means if the valve assembly is leaking, it can be removed from the valve body for repair without empting the tank. See Figure 10.19.

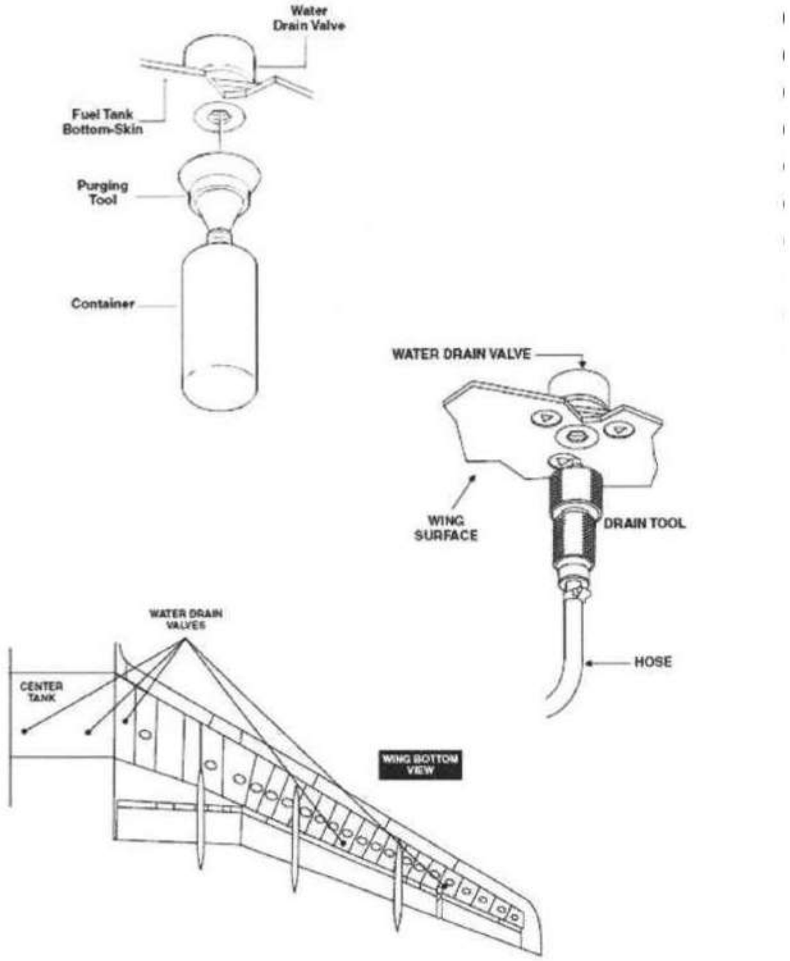


Fig. 10.19 Water drain valve location

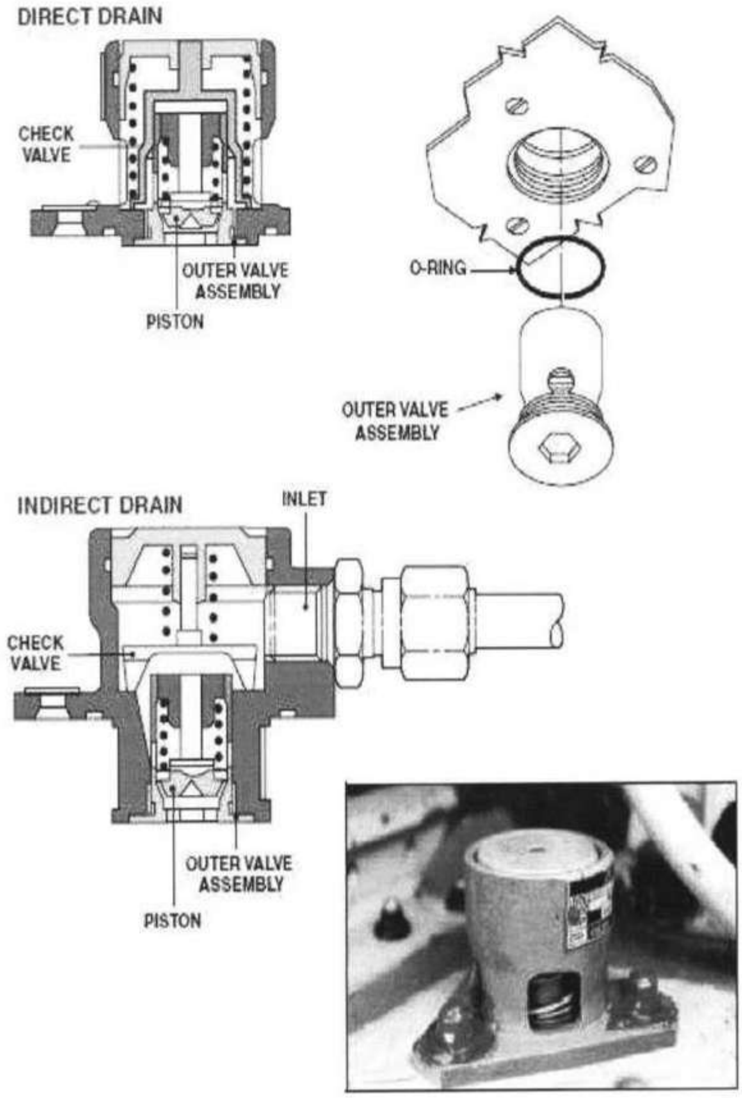


Fig.10.20 Waterdrainvalve Fig.

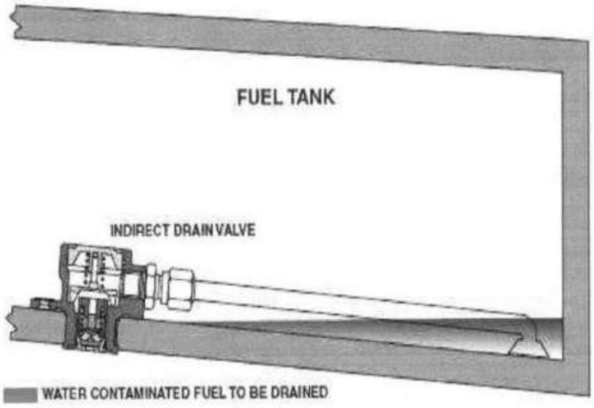
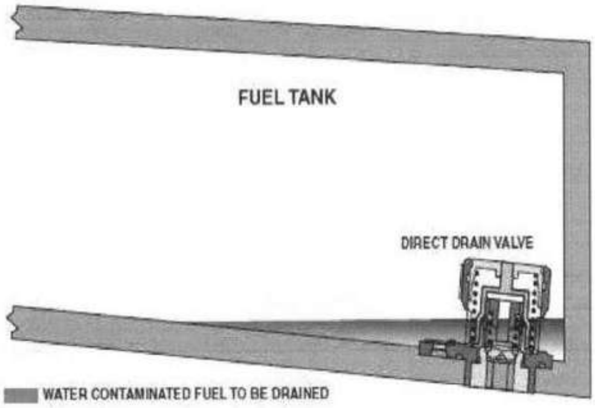


Figure 19: Arrangement of a Direct Drain Valve



10.21 Arrangement of an indirect drainvalve

Fuel Pressurization

Fuel pressurization is sometimes required to assist in forcing the fuel under relatively low pressure from certain tanks to others that are more strategically placed within the system. On some aircraft there may be no need for a pressurization system at all; it may be sufficient to gravity feed the fuel or rely on transfer pumps to move it around the system. On other aircraft ram air pressure may be utilized to give a low but positive pressure differential. Some fighter aircraft have a dedicated pressurization system using high-pressure air derived from the engine bleed system.

The engine bleed air pressure in this case would be reduced by means of pressure- reducing valves (PRVs) to a more acceptable level. For a combat aircraft which may have a number of external fuel tanks fitted the relative regulating pressure settings of the PRVs may be used to effectively sequence the transfer of fuel from the external and internal tanks in the desired manner. For example, on an aircraft fitted with under-wing and under- fuselage (ventral) tanks it may be required to feed from under- wing, then the ventral and finally the internal wing/fuselage tanks. The PRVs may be set to ensure that this sequence is preserved, by applying a higher differential pressure to those tanks required to transfer fuel first.

Fuel Feed System General

The fuel feed or supply means the fuel that is transported to the engines and the auxiliary power unit (APU). The feed system includes the following parts:

- fuelpumps
- checkvalves
- fire shut-offvalves

- crossfeed system consisting of crossfeed manifolds and crossfeedvalves.

Fuel Systems for Multi-Engined Aircraft

A fuel system for a typical multi-engined aircraft is illustrated in Figure 10.21 and Figure 10.22. The multiplicity of engines necessitates additional tanks, piping, valves and pumps. In addition, different venting and refuelling systems are necessary, and additional functions such as fuel jettisoning, fuel heating, cross-feeding, and instrumentation have to be provided for.

In modern turbine-powered aircraft, the fuel is usually contained in a number of integral tanks, in the wings and centre section, and, occasionally, in the fin and/or horizontal stabilizer. Individual engines are usually fed from an associated tank, or group of tanks, but crossfeed and interengine valves may be provided to enable the engines to be fed from any desired group of tanks, and also to permit fuel transfer between tanks. Fuel supplies for auxiliary power-units and combustion heaters, where fitted, are normally taken direct from a suitable tank or from a feed line.

pumped from the auxiliary tanks via transfer pumps to the main tanks and from there to the engines.

In the case of full tanks, the fuel from the main tanks must first be consumed before they are refilled from the auxiliarytanks.

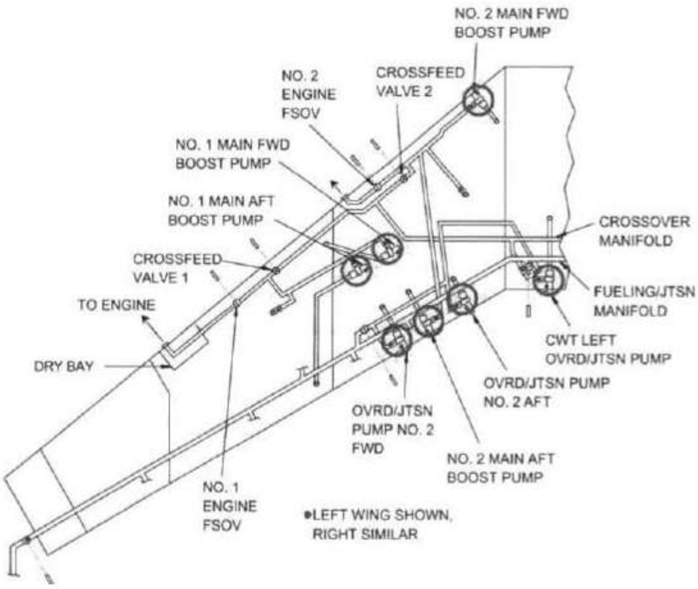


Fig. 10.22 Fuel supply system for a twin engined aircraft

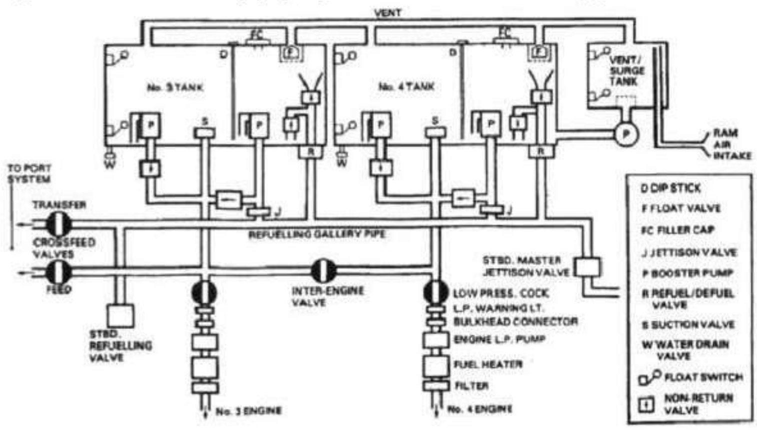


Fig. 10.23 Fuel supply system for a 4-engined aircraft

Booster (Low Pressure) Pumps

In the fuel system illustrated in Figure 10.22, two booster pumps (sometimes called 'low pressure pumps') are fitted in each tank. These pumps are designed for continuous operation, and either pump can supply the needs of any one engine. In the event of failure of both pumps in a tank, fuel is drawn from that tank by the associated engine-driven, low-pressure pump, via the suction valve, but in some cases this may be inadequate to provide full engine power at high altitude, and operating limitations may be imposed. The booster pumps are electrically operated, but, unlike the pumps fitted to light aircraft, may be operated by alternating current. They vary considerably in design, but are usually powered by induction motors, and may include a two stage impeller. In some instances the motor is of the flooded type, in which the motor runs submerged in fuel, thus obviating the need for seals. Overheat protectors are usually fitted, which cut off power to the motor when the pump temperature rises above a predetermined value. Pumps are often fitted in isolation chambers within the fuel tank, which enables them to be removed and re-fitted without draining thetank.

To prevent the return flow of fuel or power from one pump to another, the necessary check valves are installed in the system. Fuel moves from the tank by means of the fuel pumps, lines and via the fire shut-off valve to the corresponding engine or auxiliary power unit. The fuel is The purpose of the low pressure fuel pump is to feed the fuel from the tanks to the engine pump. This is done to support the engine pump and to prevent cavitation and the formation of vapour bubbles. The fuel pump consists usually of a three phase alternating current motor and a centrifugal pump.

The pump can be found in the pump housing (or canister). For quick replacement during ground time, the pump can be removed without having to drain the tanks. The housing can be closed manually or closes automatically when the pump is removed. In each tank, there are usually at least two fuel pumps. Each fuel pump must be capable of supplying one or more engines with sufficient fuel during all the various phases of the flight.

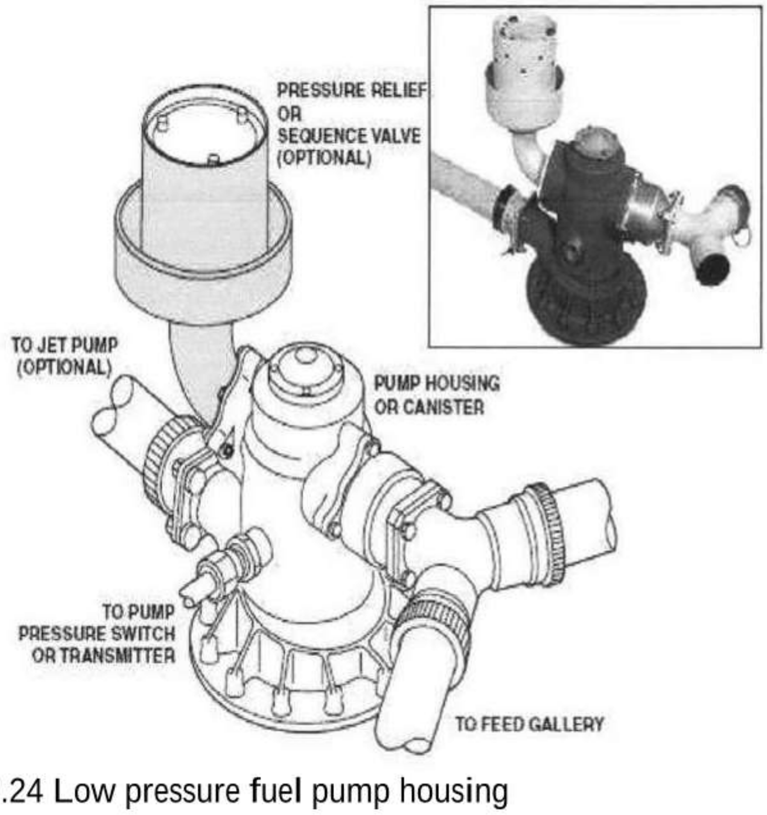


Fig. 10.24 Low pressure fuel pump housing

Fig.10.25 Low pressure fuel pump

Removal of the Fuel Pump Element

Fuel pumps can be usually removed without empting the tank.

On some modern aircraft, not even opening or entering the tank is necessary. Others provide access to the pump via a tank access panel from the upper side of the wing. In this case, the pump can be removed by using a special tool. A sliding valve is closing and prevents the fuel from entering the canister if the fuel pumpis
lowered. The removal procedure could look as follows:

1. Disconnect the electrical connector(17).
2. Put blanking caps on the disconnected electricalconnectors.
3. Cut, remove and discard the lockwire on the drain plug(18).
4. Fully loosen the screws(20).
5. Put the container below the pump(16).
6. Attach the special tool to the pumpflange.
7. Hold the handle of the tool and pull down, until the movement is stopped by the dowels in the canister. (Step1)
8. Loosen the drain plug (18) and let the fuel drain into the container.
9. When all the fuel has drained tighten the drain plug(18).
10. Hold the pump (16) and turn it counter clockwise. (Step2)
11. Hold the pump (16) and remove it from the canister. (Step3)

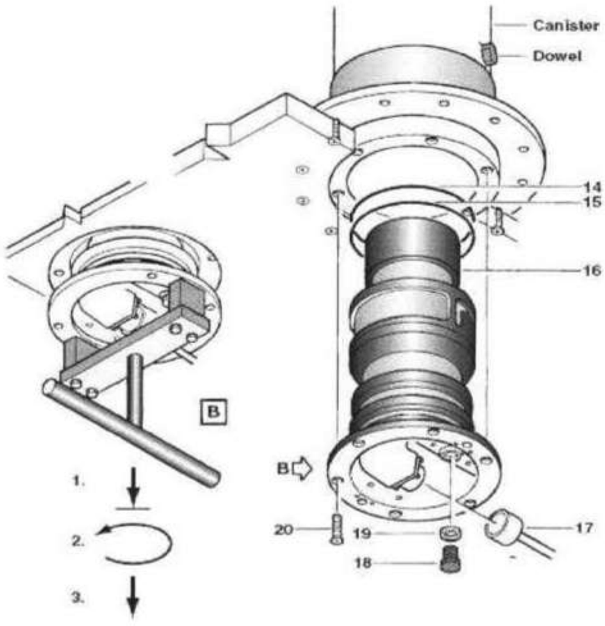


Fig. 10.26 Removal of pump element

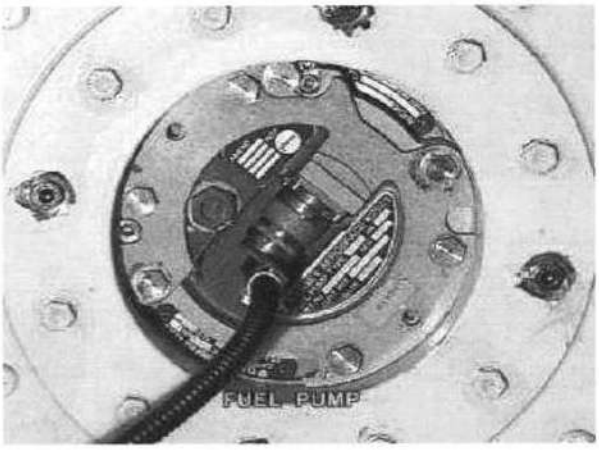


Fig. 10.27 Fuel pump (bottomview)

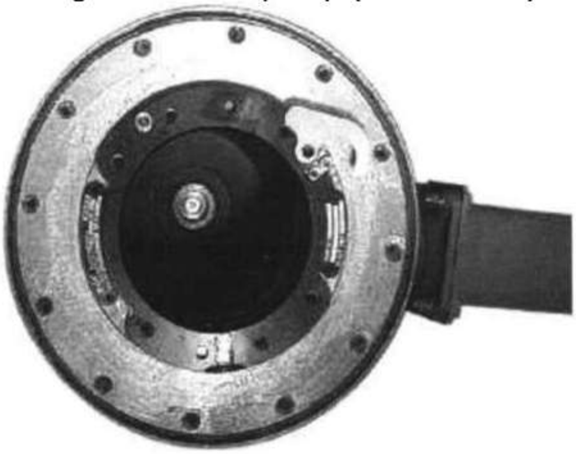


Fig. 10.28 Canister housing (pump elementremoved)

Fig. 10.29 Pump Element (removed)

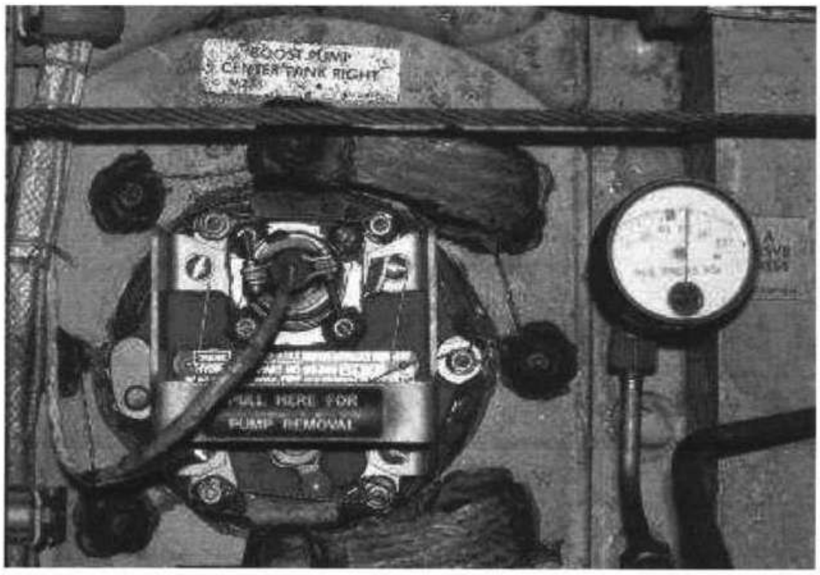


Fig.10.30 Fuel pump installation (B737) Jet Pumps

Jet pumps (sometimes called 'ejector pumps') are used where electrically driven fuel pumps are not essential. You may find them, for example, in vent surge tanks or centre tanks where no direct fuel feed to the engine takes place.

If ever fuel flows over into the vent surge tanks, jet pumps driven by the main fuel pumps bring the fuel back into its tank.

The same principle is used to empty centre tanks. The fuel flow of the wing tanks is used to drive the jet pump located in the centre tank.

To switch "on" and "off the jet pump, a shut off valve will be operated in front of the jet pump. See Figure 10.31.

The jet pump has two fuel inlets, the motive-flow inlet (from the electrically driven fuel pump) and the suction inlet (from fuel tank). When the jet pump is in operation the flow of fuel through the motive-flow nozzle causes a secondary flow (from the fuel tank). The mixed flow becomes stable in the mixing tube and slows down in the diffuser before it goes into the connected tank.

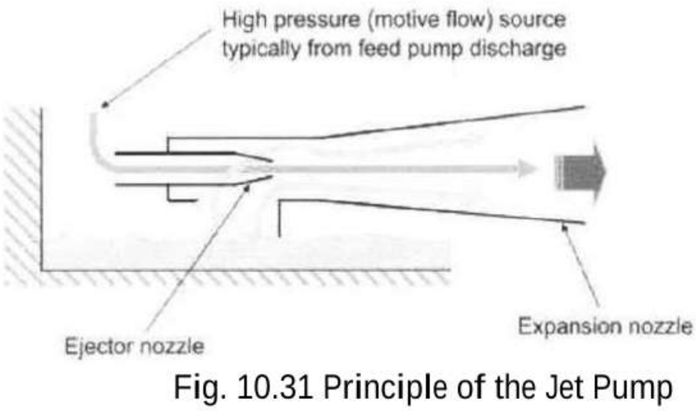


Fig. 10.31 Principle of the Jet Pump

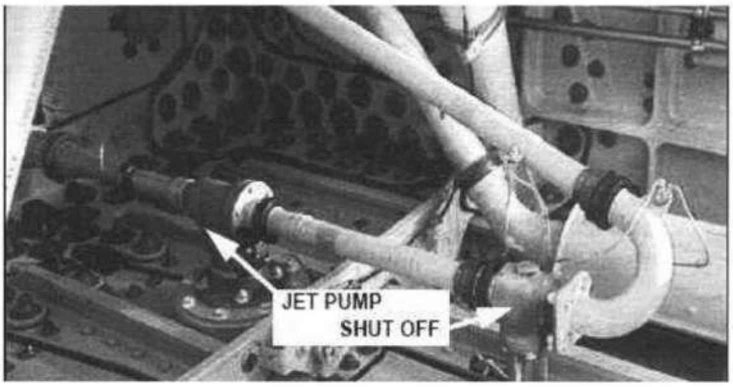


Fig. 10.33 Principle of the Jet Pump Fuel Switches

Float Switches

Float-operated switches are often of a magnetic type, similar to the one shown in Figure 10.33, and are designed to isolate the electrical mechanism from the fuel tank, for safety reasons. Upward movement of the float brings the armature closer to the magnet, and, at a predetermined fuel level, it has sufficient influence to attract the magnet, which results in operation of the micro-switch. As the fuel level and the float fall, the attraction of the armature is eventually overcome by the combined forces of the counterweight and the micro-switch spring, and the counterweight falls, changing the micro-switch circuit.

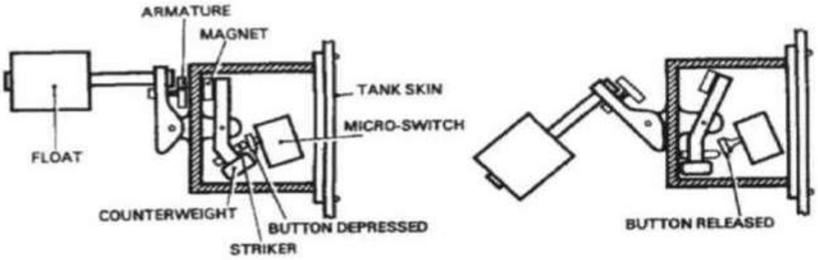


Fig. 10.34 Float switch Pressure Switches

Pressure switches are installed to monitor the output pressure of each fuel pump. For quick replacement without entering the tank they are often placed outside of the fuel tank. A pressure pipe connects the pressure switch to the fuel pump. If the outlet pressure from a pump decreases below a given threshold the pressure switch sends a signal to the flight deck where a warning to the flight crew is generated.

The primary components of the pressure switch are the body and the adapter. The body contains:

- a flexiblediaphragm
- a switchmechanism
- an electricalmicroswitch.

The flexible diaphragm isolates the switch mechanism and the microswitch from the fuel. When the fuel pressure increases to a given value, the flexible diaphragm moves to operate the switch mechanism, which opens the contacts of the microswitch. When the fuel pressure decreases the flexible diaphragm moves in the opposite direction to operate the switch mechanism and closes the contacts of themicroswitch.

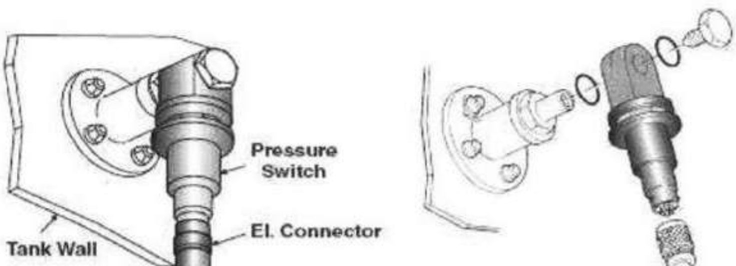


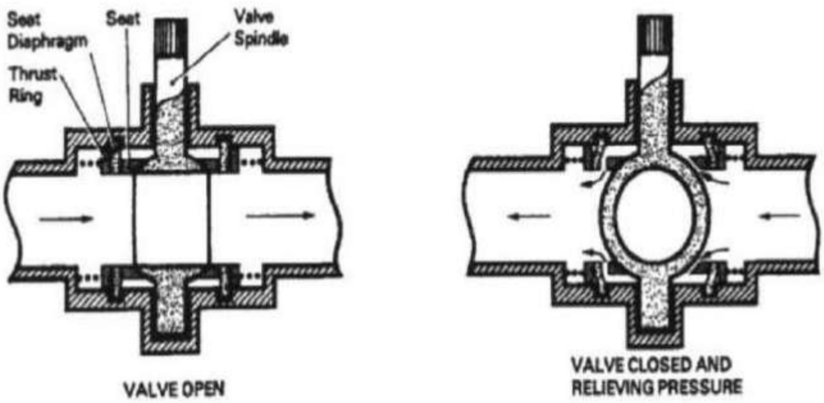
Fig. 10.35 Pressure switch

Additional to the pump pressure switches, the pressure in the fuel feed line will also be monitored and displayed to the flight crew. This can be done by using a switch or a pressure sensor. As seen before, a pressure switch can have two conditions; open or closed. Whereas a pressure sensor has a variable output depending on thepressure.

The location where the pressure is measured depends on the type of aircraft. Usually they can be found on the engine itself.

Valves LP Valves

Low-pressure valves, cross feed valves and inter-engine valves, are usually ball-type, full-flow valves, and may be either mechanically or electrically operated. A typical valve is illustrated in Figure 10.35; in this type a form of pressure relief is provided, to bleed off excess pressure which may occur, through variations of temperature downstream of the valve, when the valve is closed. This is a two position valve only, and either internal or external mechanical stops are provided, to limit movement to 90* a visual indication of valve position is also provided. When the valve is electrically operated, a reversible electric motor, equipped with an electromagnetic brake, is mounted on the valve casing, and .drives the valve through a gear train. Limit switches cut off power to the motor at the fully-open and fully-closed positions, and the brake operates automatically as the motor is de-energised; the brake is magnetically released when a reverse selection is made. The limit switches may also be used to operate position



indication lights or magnetic indicators in the crew compartment.

Fig. 10.36 Ball type valve Suction Valves

Suction valves are fitted to enable fuel to be drawn from the tanks by the engine-driven pumps; they are closed when booster pumps are operating normally. A suction valve is illustrated in Figure 10.36; it is a simple flap type valve, which closes when a pressure exists in the pipeline, and opens when suction is applied to the pipeline.

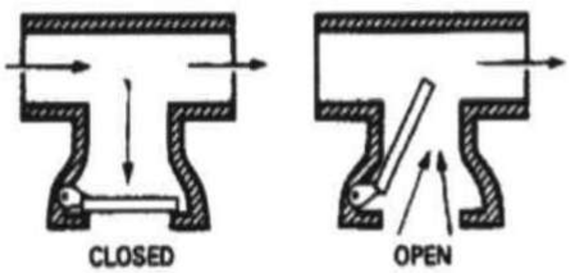


Fig. 10.37 Suction valve Non-Return Valve

Non-return valve ('check valve' in USA) may be fitted in several places in the fuel system, to provide flow in one directiononly. Atypicalnon-returnvalveisillustratedinFigure 10.37. The casing is marked with an arrow to show the direction of flow, and, in the valve illustrated, an interference spider is fitted to the inlet side, in order to prevent the valve from being fitted the wrong way round.

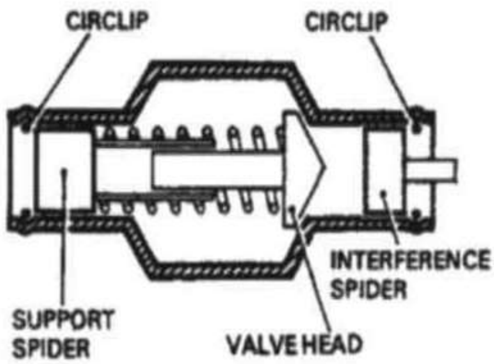


Fig. 10.38 Non-return valve

Air Release Valve

The air release valve releases air trapped in the engine fuel feedline.Itisinstalledatthehighestpointofthefuelline.

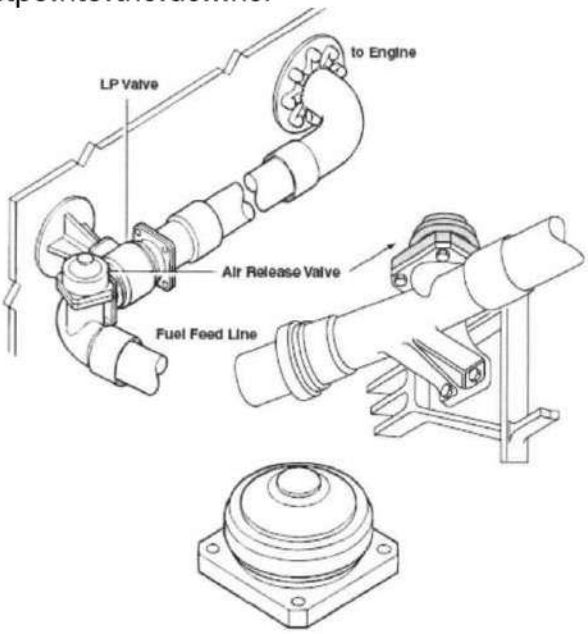


Fig. 10.39 Air release valve Fire Shut-off Valve

The function of the fire shut-off (often called "low pressure shut- off") valve is to be able to cut-off the fuel supply line to the engines in case of an external fire or any other defect which makes it necessary to isolate the engine. The valve can be closed either mechanically or electrically depending on the aircraft. There is at least one fire shut-off valve in each supply line. The fire shut-off valve of the auxiliary power unit (APU) is generally operated electrically. This is because the APU is electrically switched off by means of its own monitoring system.

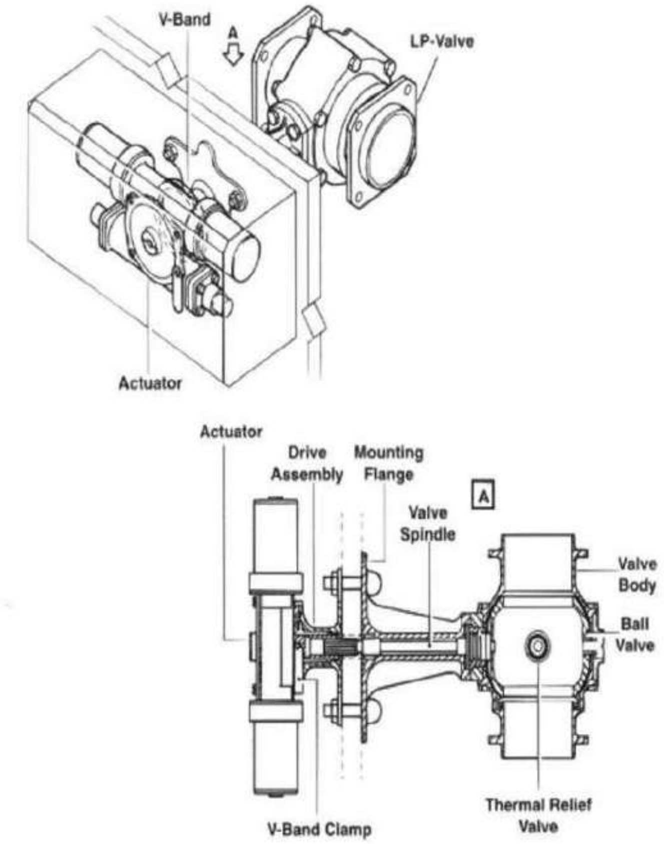


Fig.10.40 Shut-off valve Electrically Driven Valves

If valves are electrically driven, two actuators are installed for redundancy. They are supplied from different power sources. Figure 10.40 shows a simplified schematic.

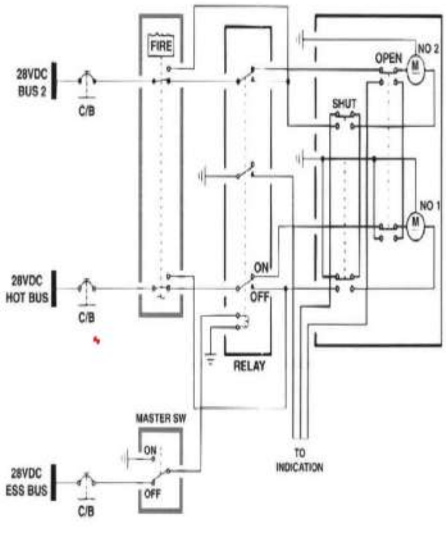


Fig. 10.41 Electrical supply Crossfeed System

In principle, each er … main tank to its engine stagnates, opening th … under pressure. Via this system the fuel is tra … . A number of crossfeed manifolds and cross … this system. Of course, systems differ from t…

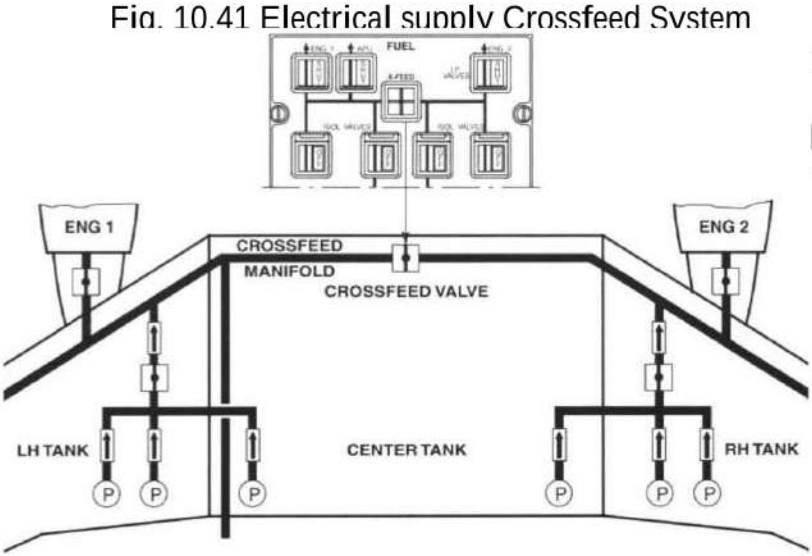


Fig. 10.42 Crossfeed system

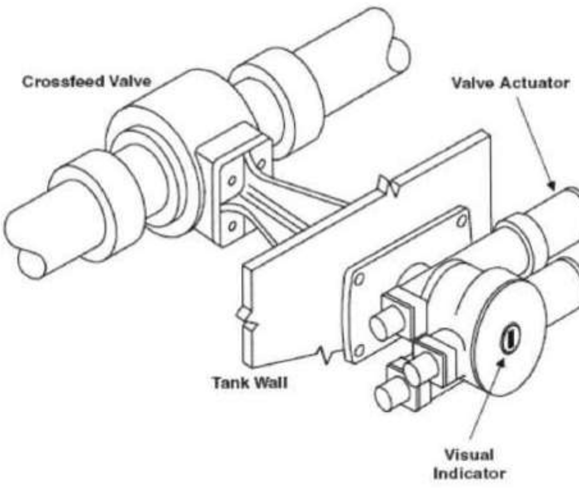


Fig. 10.43 Crossfeed valve

Fuelling and Defuelling the System General

Light aircraft fuel tanks are usually filled through over-wing filler caps, and drained by means of suitable cocks or plugs in the tanks or pipelines. These features are often retained on large transport aircraft for emergency refuelling and for draining individual tanks, but as these methods are very slow, refuelling and defuelling are normally carried out through pressure refuelling connections situated in the lower wing or fuselage surfaces. Using a tanker or hydrant, and delivery pressures of up to 350 kN/m2 (50 lbf/ins), refuelling rates of up to 1,000 gal/min (4500 litres/min) may be achieved; defuelling is carried out using the same system, but suction is then applied to the pressure connections. The system includes the pressure connections, individual refuel/defuel valves for each tank, a load control panel, and suitable pipelines and tank valves as illustrated in Figure 10.22. In systems fitted with electrically-operated refuel/defuel valves, the refuel/defuel valves are opened by selector switches on the load control panel, but may be closed by these switches, by the float switches in the tanks when the tanks are full, or by electronic controls on the load control panel when complete refuelling is not required. In systems fitted with mechanically-operated refuel/defuel valves, the valves are opened manually, and closed either manually, or by means of pressure operated valves in the tanks. The refuelling discharge pipes in the tanks are usually fitted with a diffuser, the purpose of which is both to prevent any erosion of, or damage to, the sealant, which may result from a high pressure jet, and also to prevent static discharge within thetank.

NOTE: Refuelling points should be marked with the type of fuel to be used, and overwing filling points should also be marked with the capacity of each tank. Similarly, refuelling/defueling containers and storage vehicles should be identified as to the type of fuel they contain.

A typical refuel/defuel valve is illustrated in Figure 10.43; it is actuated by either of two solenoids, one for refuelling and one for defuelling. When the refuel solenoid is energised, the associated plunger opens the passage from the inner cylinder to the exhaust port, and fuel pressure at the refuelling inlet opens the valve. When this solenoid is de-energised, the passage is closed, and pressure builds up on the inner face of the piston; the area of the piston is greater than that of the valve, so the valve closes. When the defuel solenoid is energised the inner cylinder is open, via the by-pass duct, to the refuelling inlet side of the valve, and suction applied to this side of the valve will create a pressure differential across the piston, moving the piston inwards and opening the valve. When the defuel solenoid is de-energised, the by-pass is closed, and fuel enters the inner cylinder by leakage past the piston; pressure builds up on the inner face of the piston and this pressure, assisted by the spring, closes the valve. Any pressure which builds up in the refuelling inlet line when the valve is closed, is relieved via the non-return valve and spring-loaded refuel solenoid plunger, to thetank.

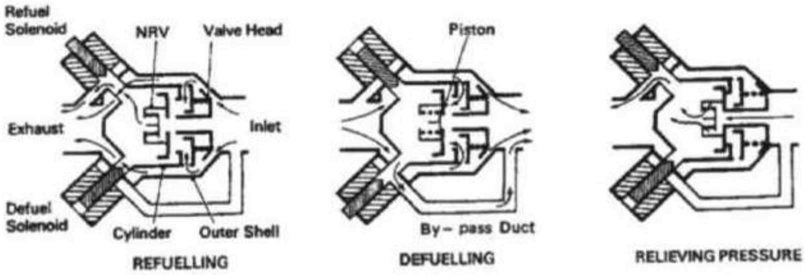


Fig. 10.44 Refuel / defuel valve

Operation
A refuel/defuel system controls the flow of fuel into or out of the aircraft. A refuel or defuel is controlled from a refuel control panel often located near the pressure coupling or at the fuselage where operation is convenient. Refuel/defuel couplings provide the interface between the refuel/defuel system and the external fuel source. Aircraft with a large tank capacity can have two refuel/defuel hoses connected to it to keep turnaround times as short aspossible.

When the aircraft is being refuelled, the fuelling hose is attached to the refuelling coupling. All the necessary manifold valves and tank valves are opened. The valve on the fuelling hose is opened and fuel flows into the tanks. When a tank is full, or when it reaches the level preset on the fuel control panel, the valve for that tank shuts off. When all the tanks have the correct amount of fuel in them, the system automatically shuts off. For alternative refuelling, overwing fuel ports are provided. Two different procedures to refuel an aircraft areavailable;

- the pressure refuel (automatic ormanual)
- the overwingrefuel.

Further, there are two procedures to defuel an aircraft:

- the pressure defuel (the aircraft fuel-pumps supply the fuel pressure for thedefuel)
- the suction defuel (the external defuel source supplies the suction to remove thefuel).

Depending of the architecture of the aircraft fuel system, these two procedures can often be used at the same time to increase the defuel rate.

The pressure fuelling system consists of;

- pressurecoupling
- a fuellingpanel;
- fuellingmanifolds;
- fuelling shut-offvalves
- a fuel quantitysystem
- a high level sensingsystem.

The fuel contents gauges on the load control panel are dual pointer instruments.

One pointer indicates actual tank contents, and the other pointer is used for preselecting the required quantity, whether refuelling or defuelling (automatic load control system). The preselector mechanism includes two micro-switches, one connected to the refuel solenoid of the refuel/defuel valve and the other connected to the defuel solenoid of the refuel/defuel valve. Pre-selection of a quantity greater than the actual tank contents will operate the micro-switch controlling the refuel solenoid, and pre-selection of a quantity smaller than the actual tank contents will operate the micro-switch controlling the defuel solenoid. When the pre- selected fuel quantity is obtained, the appropriate micro-switch circuit is broken and the refuel/defuel valvecloses.

When refuelling using the automatic load control system, the master switch is selected 'on', the function switch to 'refuel', and the loading switch to 'auto'. The contents indicator pre-selector pointers are set to the amount of fuel required in each tank, and the refuel/defuel valve selector switches moved to 'open'. The circuits to the refuel/defuel valves are now complete, and the valves open as pressure is applied.

When the quantity of fuel pre-selected for a particular tank has

been uplifted, the pointers on that indicator coincide, the appropriate micro-switch circuit is broken, and the associated refuel/defuel valve closes.

Manual refuelling is carried out by selecting 'on', 'refuel', and 'manual' on the appropriate switches, and filling the tanks individually by use of the refuel/defuel valve selector switches; the appropriate selector switch should be tripped as each tank contents indicator pointer reaches the fuel quantity required.

Defuelling is carried out in a manner similar to that described in the previous paragraphs, except that the function switch is selected to 'defuel'.

The system functions in the same way as for refuelling, the refuel/defuel valves being tripped either by the indicator micro- switch, or by the selector switches, when the excess fuel has beenoff-loaded.

Typical System Operation - Airbus A310
The Airbus A310 is a rather conventional fuel system; therefore it is suitable to describe all thefunctions.

See Figure 10.45.

A refuel/defuel manifold system is installed in the tanks. Inlet valves, (also known as refuel/defuel valves) control the flow of fuel to or from each tank. They are installed between the refuel manifold and the related tank. A fuel quantity computer controls the refuel procedure and closes the valves automatically if the preselected fuel quantity is reached.

In normal operation, no action is necessary by the ground engineer or flight crew while refuelling is in progress.

A high level protection is provided to prevent the tanks from overfilling. A wet high level sensor closes its inlet valve indicating this by a blue high level light on the fuel panel.

For manual operation, when defuelling the tanks as an example, switches are provided to override the computer and control the valves manually.

An inlet valve closes if one of these conditions occurs:

- the related fuel hi-level sensors become“wet”
- the refuel/defuel-valves switch is set to“shut"

• the fuel tank has got the set fuelload.

When electrical operation fails, most of the valves can be controlled manually. In this case, the fuel computer will not be able to prevent the tank from over-filling.

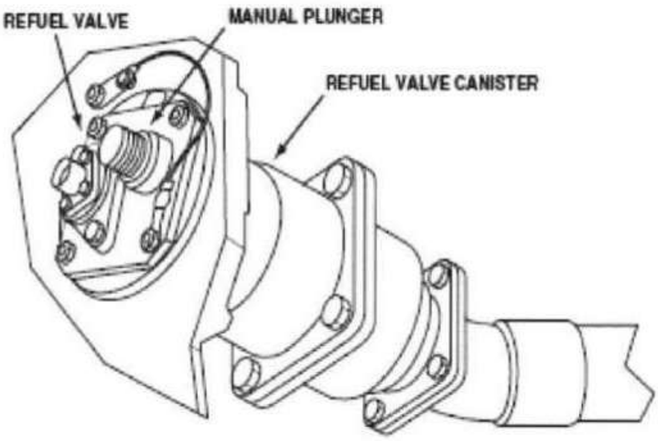


Fig. 10.45 Refuel or inlet valve

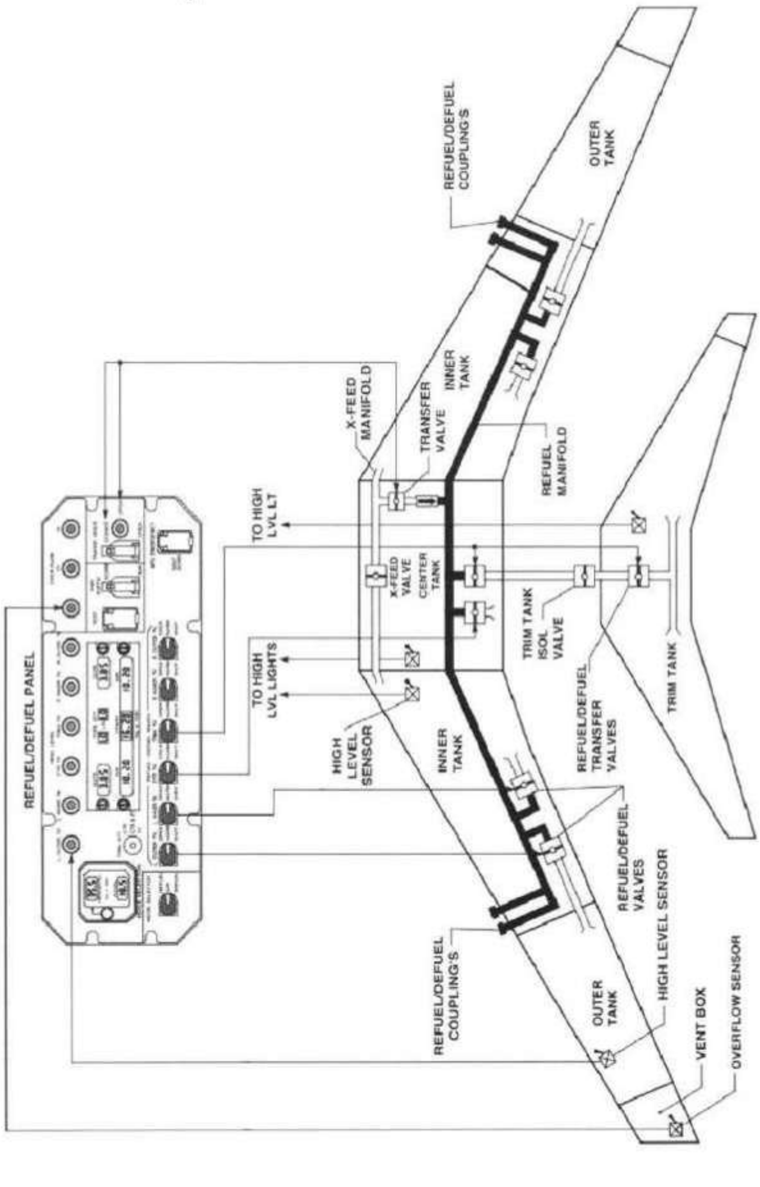


Fig. 10.46 Refuel / defuel system layout

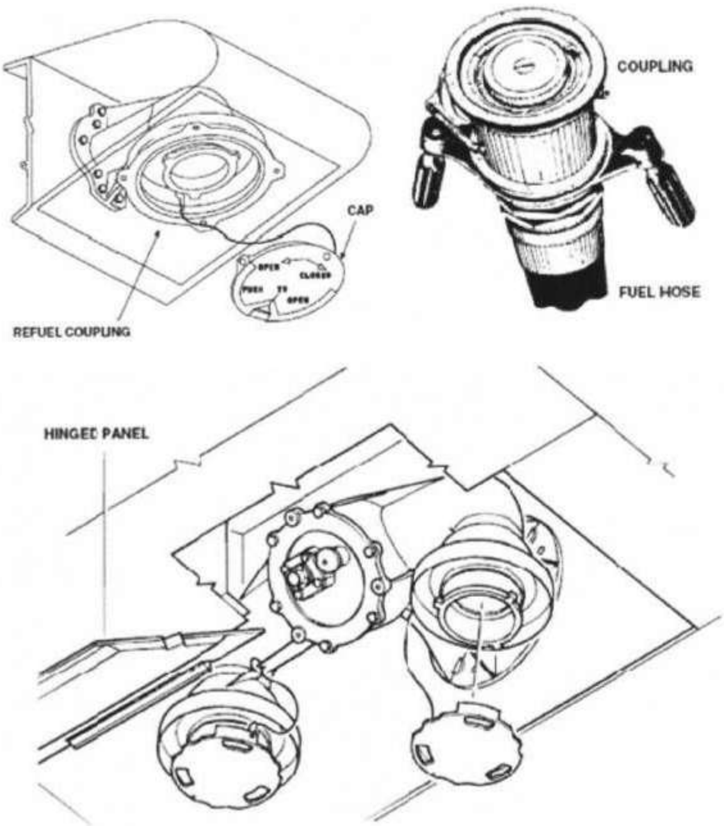


Fig. 10.47 Refuel / defuel panel and coupling Valve Actuation

Today, valves are driven electrically. Figure 10.47 shows a typical actuator. Pay special attention to the "see and feel" indicator. This feature makes it possible to determine the valve position when visible access to the actuator is limited. Depending on the importance of safety, some actuators are equipped with two electric motors. In this case, they are supplied from different electrical sources.

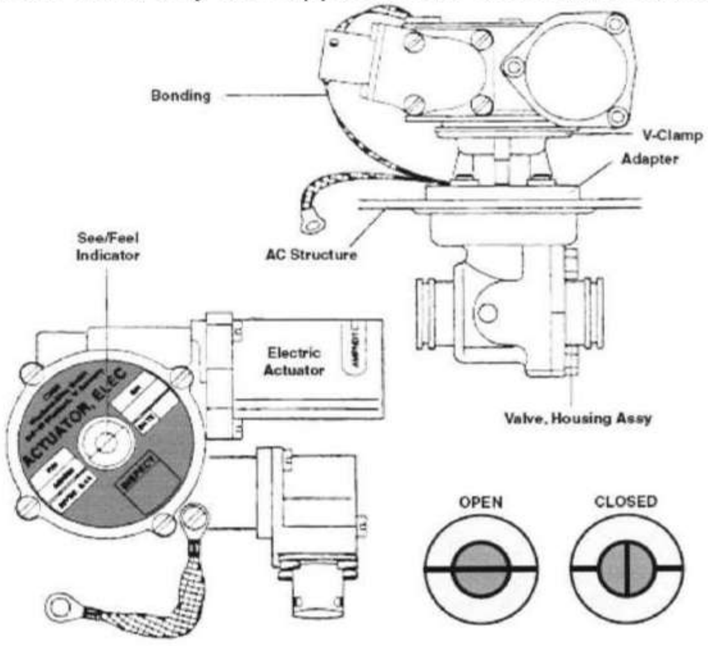


Fig. 10.48 Electrically actuated Fuel Valve

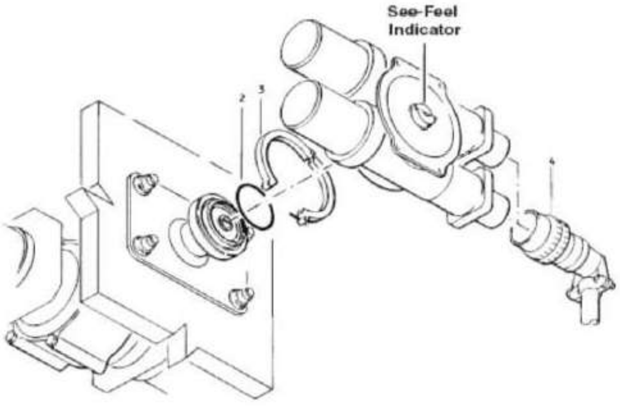


Fig. 10.49 Mounting principle

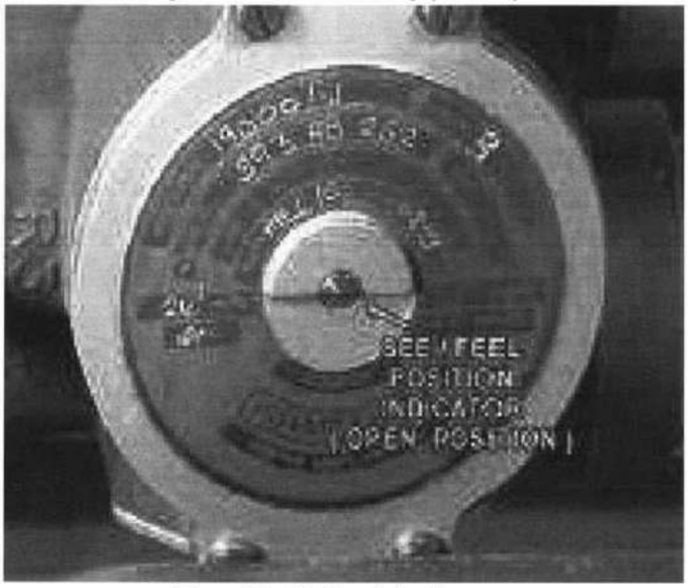


Fig. 10.50 Feel indicator

Fire Hazards when Fuelling or Defuelling

Aviation fuels are both highly flammable and volatile, and special care must be exercised when transferring them into or out of an aircraft. Be sure that the proper type of fire extinguisher is available at the aircraft and that it has been properly serviced and has not been used, even partially, since it was lastserviced.

Never service an aircraft with fuel inside a hangar or in any other closed area. If fuel is stored in containers other than the fuel service truck or the aircraft fuel tanks, be sure that the containers are closed, not only to prevent the entry of contaminants, but also to prevent the release of fuel vapours.

If any fuel is spilled, wipe it up immediately; or, if too much has been spilled to wipe up, use an approved procedure and materials for dealing with fuel spills. There are new absorbent materials available to contain and clean up hazardous materials.

Matches, cigarette lighters, smoking, open flames, and even backfires from malfunctioning vehicles are obvious sources of ignition that need no further mention. One source of ignition, however, is not so visible or obvious, it is the sparks created by static electricity.

Static electrical charges are generated in various degrees whenever one body passes through or against another. A greater generation of static electricity may be expected when handling turbine fuels than when handling aviation gasoline. A basic reason for this is related to the higher viscosity of this type of fuel. The high-speed fuelling rates and the flow through the ultrafine filter/separators required can create extremely high static electricalcharges.

To minimize this hazard, it is necessary to 'bleed off' static electrical charges before they build up to a high enough potential to create a static spark. This can be accomplished by bonding and grounding all components of the fuelling system together with static wires and allowing sufficient time for the charge to dissipate before performing any act which may draw a spark.The bleeding off of an electrical charge is not always an instantaneous act as is commonly believed. It may take several seconds to bleed off all the charge from somefuels.

Fuelling Procedures

From time to time the aircraft mechanic may be called upon to fuel or defuel aircraft or, because of their expertise, to assist in

the training of ground service personnel. The steps outlined below represent general procedures which should be carried out when fuelling anyaircraft.

1. Make sure of the grade and quantityrequired.
2. Make sure the fueller or system contains the correct grade and quantityrequired.
3. Check the fueller tank sumps for water before fuelling. Drain ifnecessary.
4. Approach the aircraft carefully. Try to position the fueller so that it can be quickly driven or pulled away in case of emergency. Avoid backing up to the aircraft; if absolutely necessary, have someone guide you from a position near the rear of the fueller. Set thebrake.
5. Bond and ground the aircraft and equipment in the proper sequence - fueller to ground, then fueller to aircraft. Before opening aircraft overwing fuel filler cap, connect the nozzle ground to the aircraft. Keep a constant contact between an overwing nozzle and the filler neck spout while filling. Grounding of underwing nozzles is not required. After fuelling, reverse the stepsabove.
6. Nozzles should never be propped open while fuelling aircraft or otherwise left unattended. Nozzles must never be dropped or dragged across the pavement. Nozzle dust caps should be installed immediately afterfuelling.
7. Leaving a filler cap off an aircraft fuel tank can be very dangerous. Never open a cap until you are actually ready to fuel that specific tank, then lock it and close the flap immediately after fuelling. Before leaving the wing, recheck each fillercap.

Fuel Quantity Indication Systems General

The principles of capacitive fluid contents measurement can be found in Section 13.8 (Instruments (ATA 31)). This section provides description of the application of these principles into actual aircraft systems, and descriptions of other fuel contents measuring systems.

Capacitive Systems

The Fuel Quantity Indication System measures the amount of fuel in each individual tank electrically by means of a probe (capacitor). The amount of fuel can be read on the flight deck and on the fuelling panel. In each tank, several probes are arranged to achieve accurate data. If one or more probes deliver incorrect data, the system is usually still operative and operates in a degraded mode. This means that the indication can still be available to the crew but inaccurate.

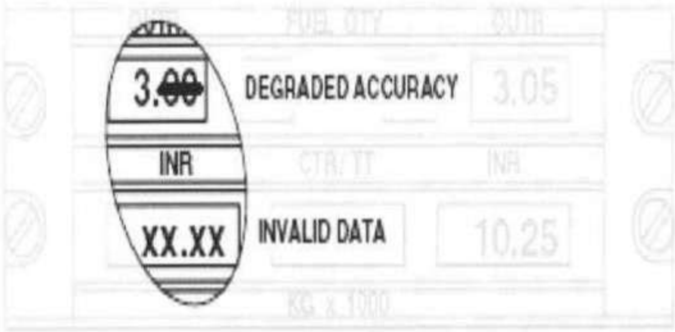


Fig. 10.51 Fuel quantity indication

If during refuelling an indication error occurs, the manual measuring method with the floatsticks is used as a cross check.

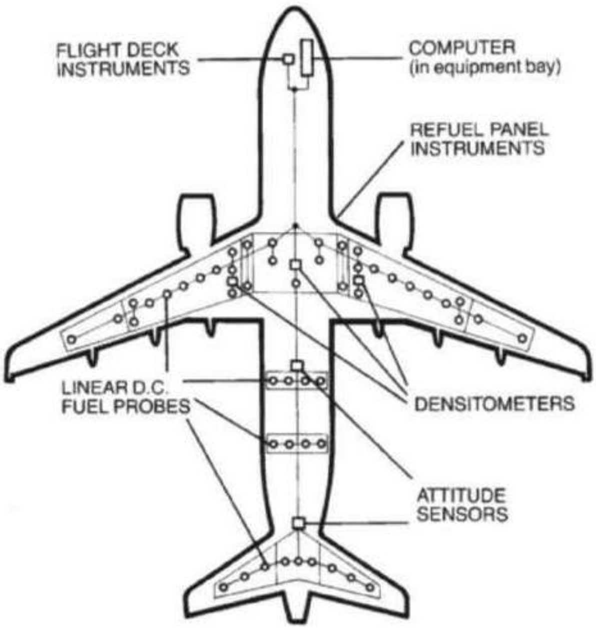


Fig. 10.52

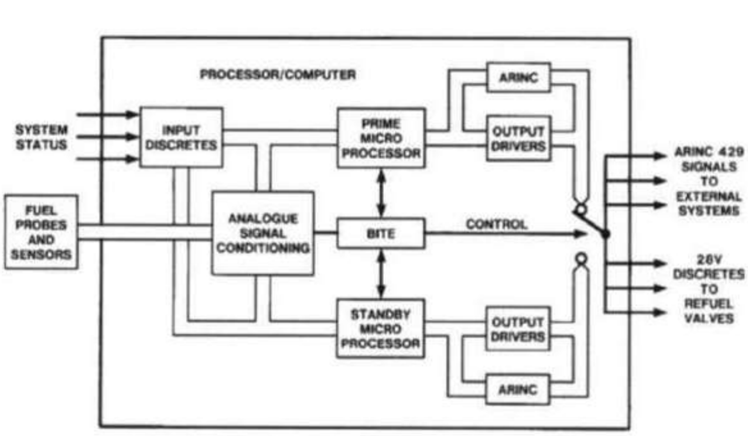


Fig. 10.53 Fuel quantity indication schematic

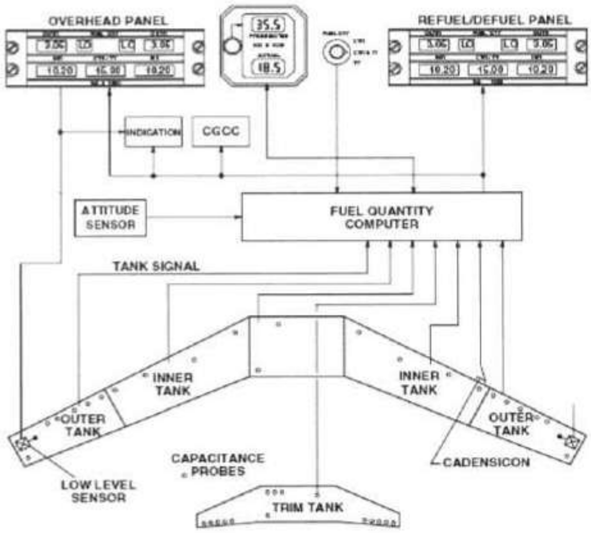


Fig. 10.54 Another typical system schematic

The capacitor-type fuel quantity system is an electronic fuel measuring device that accurately determines the weight of the fuel in the tanks.

The measuring element (probe) in a tank consists of two concentrically mounted tubes in open connection to the tank. These tubes form the plates of a capacitor.

The capacitance of a capacitor depends on three factors;

- the area of theplates;
- the distance between theplates;
- the dielectric constant of the material between theplates.

The only variable factor in a tank is the dielectric constant which depends on the ratio of fuel and air. A change in fuel level will

result in a change of capacitance.

The value of the capacitance is computed to a reading on the indicator.

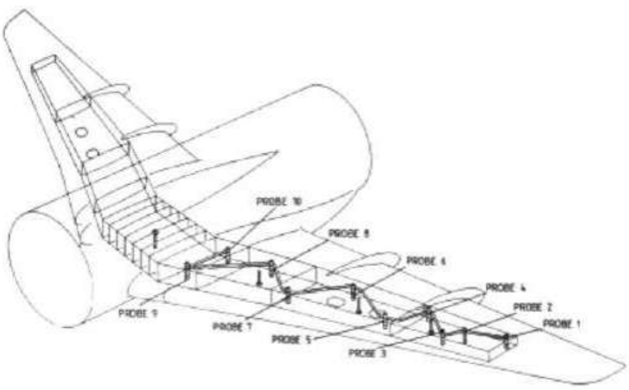


Fig. 10.55 Quantity probes

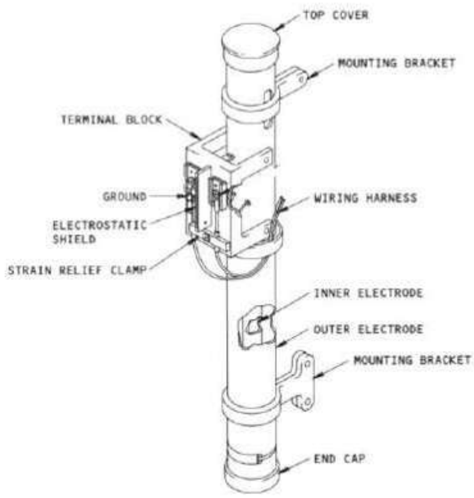


Fig. 10.56 Fuel capacitance probe

Dripsticks

A 'drip stick' (sometimes called 'dipstick') consists of a short outer tube, which is attached to an adaptor in the lower wing skin and protrudes upwards into the tank, and a long inner tube (calibrated in gallons or inches), which slides in the outer tube, and is secured to the adaptor by a bayonet fitting. The gap between the tubes is sealed against fuel leakage.

To check fuel contents, the inner tube is unlocked and slowly withdrawn downwards; when the top of this tube falls below the fuel level, fuel will flow through it, and drain out of a hole in its base. The length of tube protruding from the adaptor, will indicate the tank contents. The volume of fuel, in gallons, may be obtained from tables provided in the aircraft Maintenance Manual.

Floatsticks

The fuel floatstick (sometimes called 'magnetic dipstick') indicates the level of the fuel. It is used manually and no electricity is needed to operate the stick. It is operated from the outside of the aircraft, at the lower side of the wings and fuselage tanks. The fuel floatstick is less accurate in comparison to the capacity measuring element and is an alternative method for measuring the amount of fuel usually for cross checking. See Error! Reference source not found..

The floatstick is located in a housing in the tank and can move up and down freely. A permanent magnet is installed at the top of the stick (stick armature). The stick housing is surrounded by a ring-shaped float that contains a ring-shaped magnet. The floatstick is unlocked by means of pressing and twisting the stick (stick head andsocket).

When the floatstick is unlocked, it is pulled down to the point at

which the magnets stick to each other. Because one magnet floats on the fuel and the other is attached to the end of the stick, the level of the fuel in the tank is determined. A scale on the stick indicates this in various units, such as kilograms, inches or gallons.

Certain requirements must be met when using the floatstick. The position of the aeroplane must be known, the aeroplane may not move (due to influence of wind) and fuelling must be stopped. To determine the position of the aeroplane, a spirit level is located in the aeroplane (the exact location depends on the type of aeroplane). With the help of calibration tables (that can be found in Maintenance Manuals) the reading on the stick can be corrected. The contents of the tank can be determined by this information.

Which Stick to use?

Usually, it is not necessary to "pull" each stick since there are several located on each tank. But how do we know which stick to use for exact measuring?

As we know, the wing arrangement of commercial jetliners has normally a positive dihedral.

This means that the wing tip is higher than the wing root relative to the horizontal plane. If aeroplanes have negative dihedral, termed anhedral, the wing tip is lower than the wing root.

When refuelling the aircraft, the fuel collects at the lowest point and moves upward and outward on a positive dihedral wing. Therefore, the "float" of the sticks arein different positions; some at the bottom of the tank (not usable), some in a floating state (good for measuring) and others at the tank upper wall (not usable). To find the correct stick, you should at least approximately know how much fuel is in the tank by pulling each stick and find the one in the floating state. See

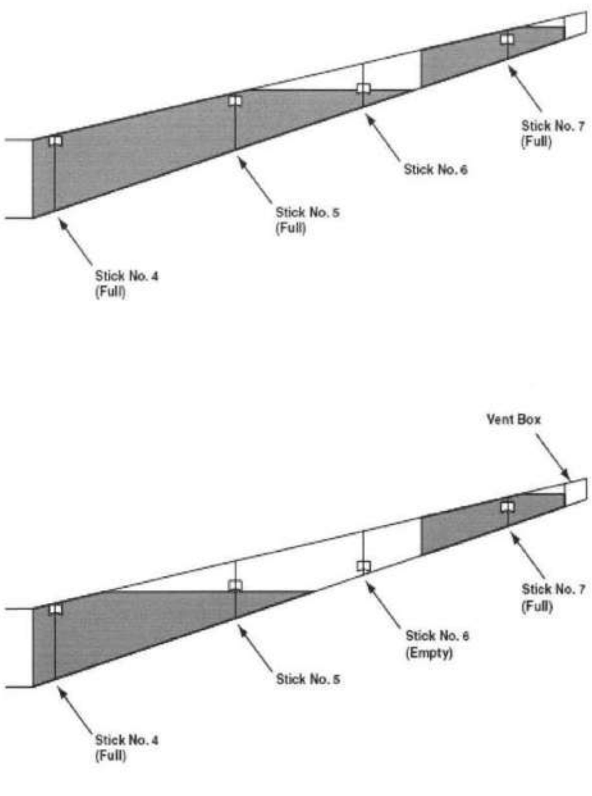


Fig. 10.57 which sticktoread?

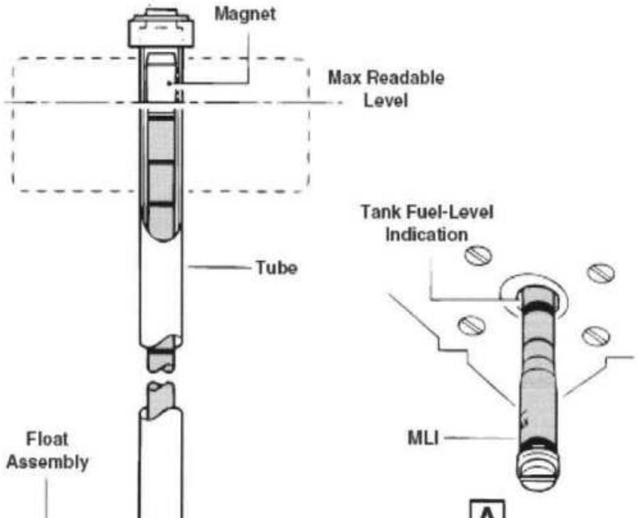


Fig. 10.58Floatstick

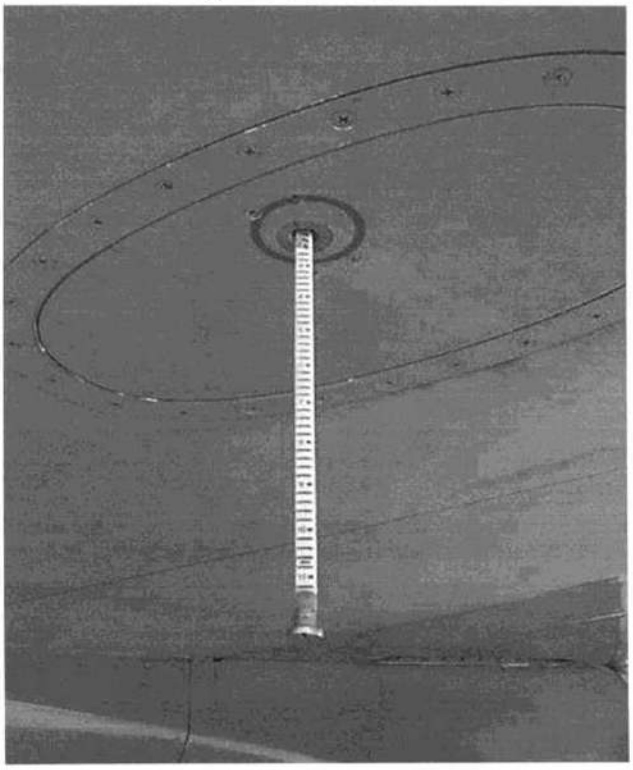


Fig. 10.59 Floatstick (Boeing 737) Ultrasonic Probes

All of the above systems use capacitive measurement techniques to sense fuellevel.
Ultrasonic techniques are now being developed which utilize ultrasonic transducers to measure fuel level instead of the conventional capacitive means. The sensor is located at the bottom of the waveguide. The waveguide arrangement at the base of the tank directs the ultrasonic transmission back to the transducer. To measure height with ultrasonics, the speed of sound in the fuel medium is required. This is generally

measured using a fixed reference in the waveguide. A portion of the ultrasonic wave is reflected directly back to the transducer and serves as a reference signal. The time taken for the signal to be reflected back from the fuel surface is measured and by using a simple ratiometriccalculation the fuel height may be determined. Fuel level may be measured by comparing the time of propagation for the reference signal with that for the fuel level reflected signal. This type of quantity measuring system was introduced on the Boeing777.

Fuel Jettison System
The Fuel Jettison System (also known as Dump System) makes it possible to bring the aircraft weight below the maximum landing weight, while keeping enough fuel in reserve for the landing. Dumping fuel is only done in case of serious malfunctions or passenger/crew medical reasons requiring a speedy landing. The dump system is operated from the flight deck. Dumping fuel is done by an automatic close system. Aircraft whose landing weights are close to the maximum take-off weight do not have a fuel jettison system.

Jettison systems;
- prevent high stress during overweight landings;
- ensure aircraft is light enough for a go-around in case of an enginefailure;
- make it impossible to empty all tanks during jettison (minimum fuel kept so the aircraft can reach the next airport).

Fuel is pumped or drained from each tank through a stand pipe, which ensures that a predetermined quantity of fuel remains. One type of system makes use of the refuelling gallery pipe, which is extended outboard to a position near each wing tip, and terminates in a large diameter open-ended pipe at each trailing edge. One of the booster pumps in each tank, which may be run at a higher speed for the jettisoning operation, is used to off-load the fuel, and the fuel feed to the engines is protected by non-return valves. Individual jettison valves are located at selected tank outlets, and a master jettison valve is located adjacent to each discharge nozzle; this type of system is illustrated inFigure
10.59.In another type of system, fuel is jettisoned through a pipe in each wing, the pipe being lowered into the airstream by an electrically-operated actuator. A short manifold is fitted between the main tanks in each wing, and a jettison valve controls flow from each tank into the manifold; auxiliary tanks are fed into the main tanks by the normal transfer valves, the transfer pumps being interconnected with the circuits operating the jettison valves. When the jettison pipe is in the retracted position it forms a seal at the manifold, and acts as a master jettison valve; the circuits to the jettison valves are not armed until this pipe is locked in the extended position. Both types of systems are controlled from a special panel at the crew station, which contains switches for the pumps and valves, and warning lamps or magnetic indicators to show the positions of the valves and the jettisonpipes.

Figure 10.58 shows a typical location of the jettison valve and pipe. In this example, the jettison pipe is routed through a flap track.

For economy reasons and to reduce workload, modern aircraft are equipped with automatic jettisonsystems.

The jettison gross weight will usually be determined prior to the flight. The value is usually entered by using either a rotating knob or the MCDU interface. If entered, the computer is able to shutthejettisonvalvesassoonthejettisongrossweightis reached.

In any case, the valves will close if a given fuel quantity minimum is reached.

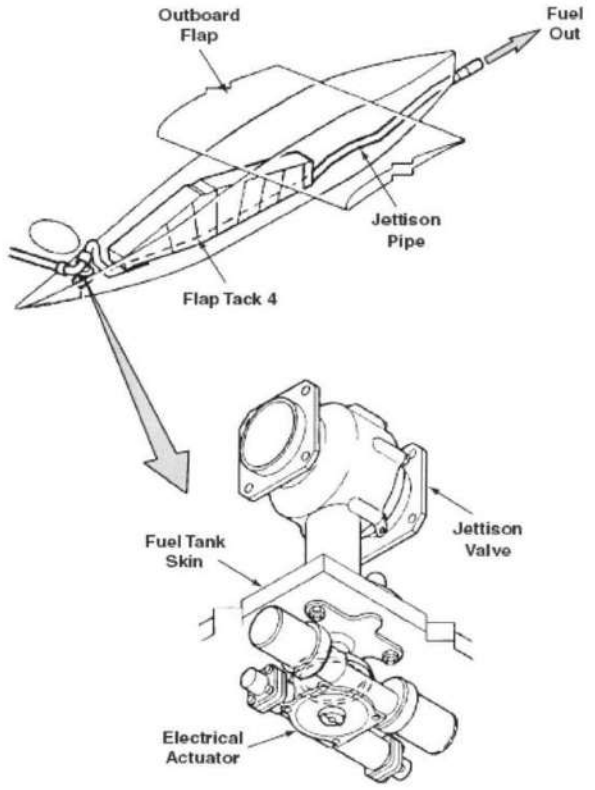


Fig. 10.60 Jettison valve and pipe

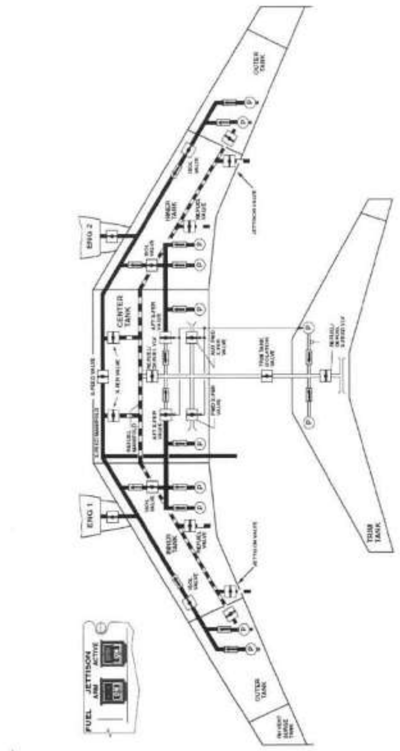
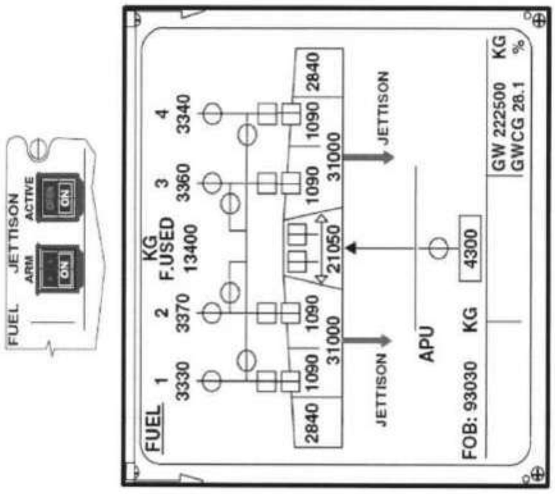
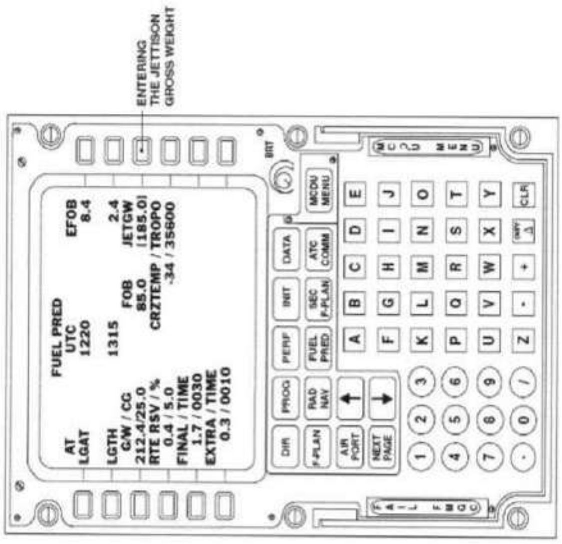


Fig. 10.61 Jettisonsystemlayout Fig. 10.62 Jettisoninterfaces





Fuel Heating

Water may enter the fuel system during refuelling, or as a result of condensation in the tanks, and, when the fuel temperature falls below OC, the suspended water droplets may freeze. These frozen droplets collect at the low pressure filters, and may restrict or block fuel flow to the engines. To prevent this, a filter by-pass and blockage indicator may be fitted, or a de-icing additive such as methyl alcohol may be used in the fuel. However, in most large aircraft provision is made for heating the fuel before it enters thefilters.

Fuel heaters are usually heat-exchangers, and may utilize hydraulic oil, engine oil, or air tapped from the engine compressors, as the heating medium. On some aircraft the engine oil coolers, which are in continuous use, are oil/fuel heat exchangers, and serve the additional purpose of heating the fuel. A heat exchanger operated by hot compressor air may be used in addition to the oil cooler, or may be used by itself for the purpose of heating the fuel. Oil/fuel heat exchangers are automatic in operation, oil flow being thermostatically controlled, but air/fuel heat exchangers may be either manually or automatically controlled.

A manually controlled fuel heating system usually consists of a pressure differential switch on the fuel filter, which operates a warning lamp in the crew compartment, and an electrically- operated valve on the heat exchanger, which is controlled by a switch adjacent to the warning lamp; a second warning lamp may also be included, to signify that the heating valve is open. When fuel flow through the filter becomes restricted by ice, the differential pressure across the filter increases, until it is sufficient to operate the icing warning lamp. The heat-exchanger valveshouldthenbeopenedtoadmithotcompressorairtothe

heat-exchanger and to warm the fuel. Fuel temperature on the outlet side of the filter is indicated by an instrument on the fuel control panel. With this type of system, the period and frequency of operation of the heat exchanger may be limited.

An automatically controlled fuel heating system consists of a thermostatically controlled air inlet valve on the heat exchanger, which progressively opens and closes to maintain fuel outlet temperature within pre-set limits above 0‘C. Actual fuel temperature is indicated on an instrument on the fuel control panel, but no action is required by the crew.

Longitudinal Balance Fuel Systems General

The centre of gravity (CG) of an aircraft is the point where all of the weight of the aircraft is considered to be located. Where the weight is placed in the aeroplane is a factor that has a tremendous effect on how well the aeroplane will fly. This is because the CG of the aeroplane must be maintained within certain limits prescribed by the manufacturer, in order for the aircraft to be flown safely. If the CG gets too far forward or too far backward the aircraft will be out of balance and difficult, if not impossible, tocontrol.

What we are interested in is CG control during flight. To operate an aircraft as economical as possible, the centre of gravity should be held in a range where the horizontal stabilizer is as streamlined as possible. This means that the stabilizer trim settings should be close to zero in cruise. This is done by transferringfuel.

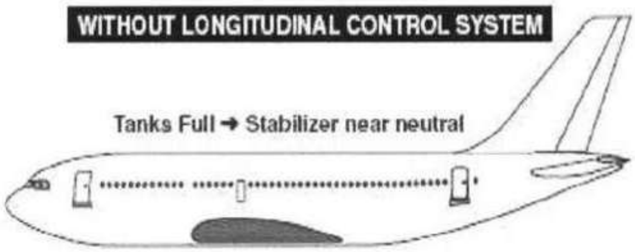


Fig.10.63

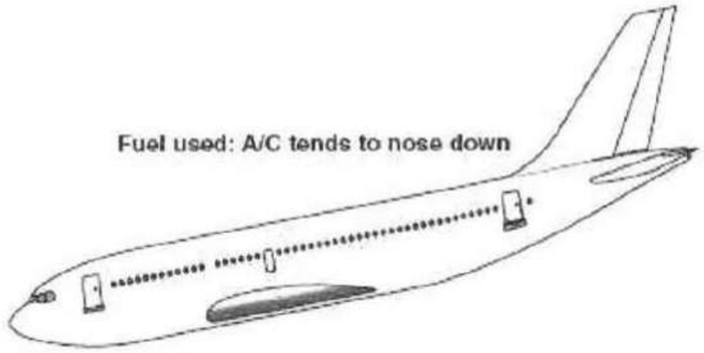


Fig.10.64

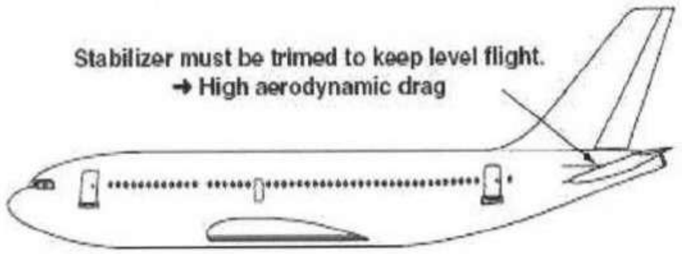


Fig.10.65

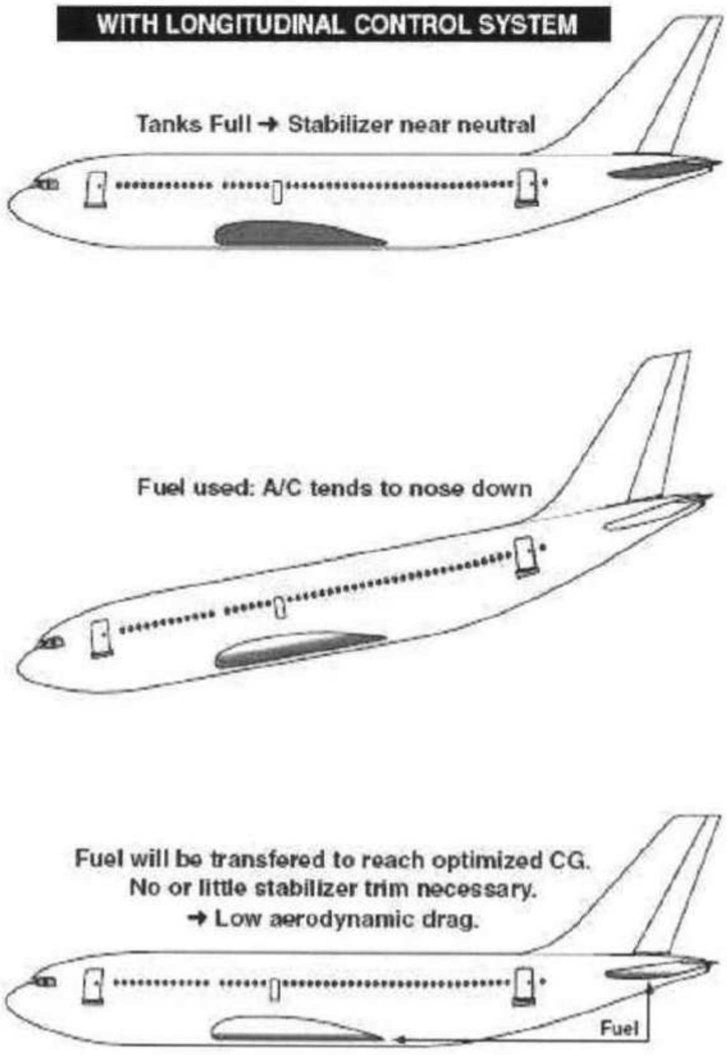


Fig. 10.66 Systemcomparison

Determining the CG

To control the centre of gravity, the system must know where the centre of gravity is before the engines are started. The aircraft's CG is determined by the manufacturer upon delivery and is used as the base for the loading sheets.

There are several parameters that change the centre of gravity:
- Passengerloading
- Cargoloading
- Refuelling

The resulting centre of gravity out of these parameters is called the zero fuel weight centre of gravity (ZFW-CG) is calculated by the flight crew prior to flight.

This data will be entered into the flight management computers and is used as the base reference.

Because the fuel weight is the only parameter that changes the centre of gravity in flight, the fuel system management and CG control uses the tank quantity for calculation.

Remember that the centre of gravity location is referred as a percentage of the mean aerodynamic chord (MAC). The target centre of gravity must always be forward of the certified aft-limit for the aircraft.

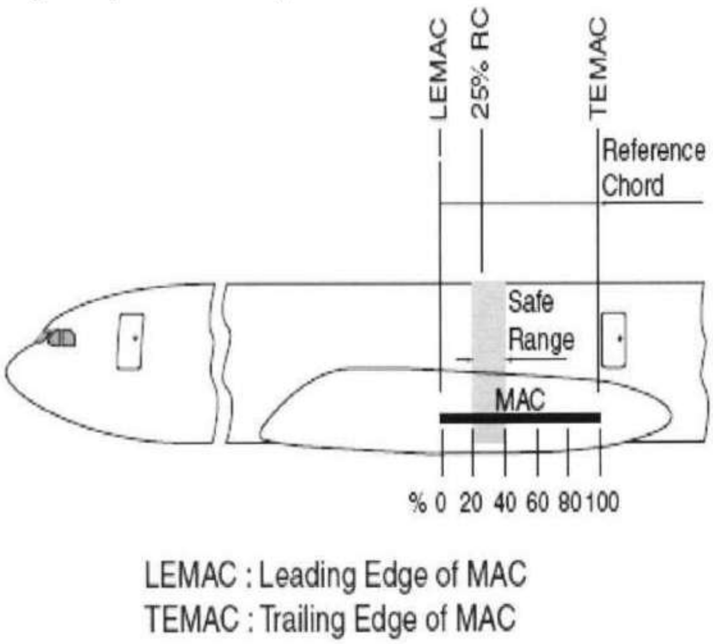


Fig.10.67 MAC

System Operation

Fuel control and monitoring computers and centre of gravity control computers control the forward and aft CG position by fuel transfer from and to the trim tank to improve aircraft performances.
- This dependson;
- zero fuelweight.
- zero fuel weight centre of gravity (aircraftloading).
- fuel quantity in eachtank.
- fuel flowsignals.
- The actual CG position and the gross weight (GW) are displayed

in the cockpit. The CG is indicated in % MAC, the GW in tons. CG control is started after take-off. The computer activates the corresponding valves and pumps and transfers fuel forward or aft (according to the fuel burn) in order to maintain the CG within a tolerance of MAC of the targetCG.

In case of a CG control system failure, the computer switches automatically (depending on the failure) to either an alternate mode or to a fault mode, which consists of emptying the trim tank.

Fig. 10.68 Indication

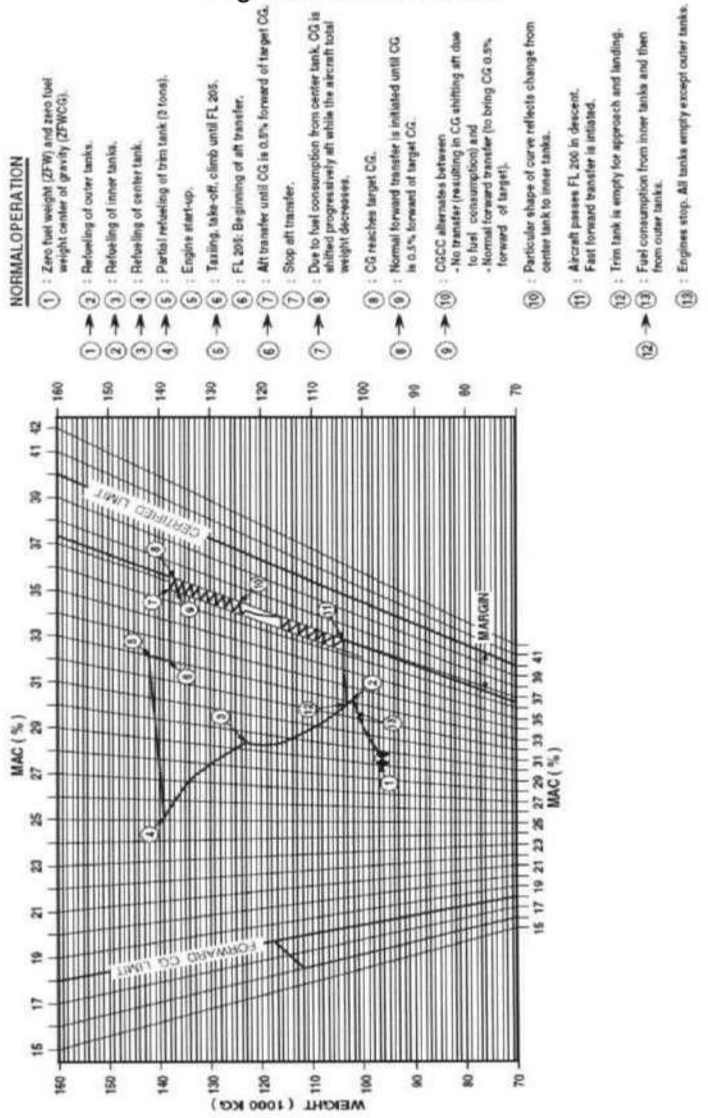


Fig. 10.69 Example of CG control during flight (Airbus A310)

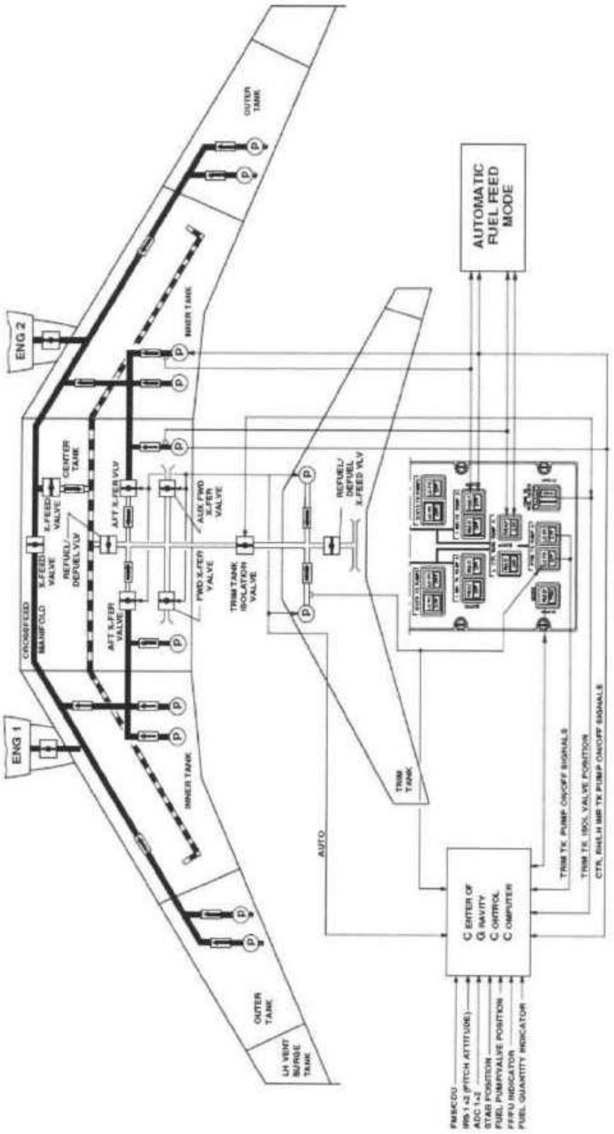


Fig. 10.70 Trim tank system layout

Fuel Contamination General

Everyone concerned with the handling and dispensing of aviation fuels should realize that the safety of an aircraft may depend upon their skill, knowledge and ability to deliver the correct grade of clean dry fuel into the aircraft. It is one of the prime factors contributing to flight safety.

Fuels, fuelling methods, and equipment are continually being developed and improved to meet the ever-increasing demands of modern aircraft and the aviation industry. However, one thing never changes - the vital importance of supplying the correct grade of uncontaminated fuel to the aircraft. The possibility of human error can never be eliminated, but it can be minimized through careful design of fuelling facilities, good operating procedure, and adequate training of personnel.

Checking for Fuel System Contaminants

Draining a sample of fuel from the main strainers of an aircraft has, in the past, been considered an acceptable method of assuring that the fuel in the system is clean. This practice is, in most cases, no longer considered adequate.

Quality control of aviation fuel has always been of particular concern, since the failure of an aircraft power plant during flight can be disastrous. Despite this, effective equipment and techniques for handling aviation gasoline have been relatively simple. The introduction of turbine-powered aircraft has made the need for fuel cleanliness much more important and at the same time more difficult to maintain.

The combustion process in the jet engine is one which must be carefully controlled. This requires complicated precision fuel control systems which are much more sensitive to fuel quality, and in particular, cleanliness, than those for piston engines. Besides this, the quantity of fuel which passes through these systems for each hour of flight is considerably greater than with the piston engines; hence, any slight contamination in the fuel accumulates at a much faster rate.

Along with the introduction of the more critical jet engine has come the utilization of a fuel which is harder to keep free of contamination. For example, a particle of dirt or rust, or a drop of water, settles out of aviation gasoline four times faster than it does in turbine fuels.

To better understand what is required to maintain fuel quality, it is first necessary to have a basic knowledge of aviation fuels, the common forms of fuel contaminants, how they get into the fuel, and how they can be detected and removed.

Types of Contaminants

The more common forms of aviation fuel contaminants are solids, water, surfactants, microorganisms, and miscellaneous ones including the intermixing of grades or types of fuel. Surfactants and micro-organisms, contaminants which are virtually unknown in aviation gasolines, have become critical with the advent of turbine fuels.

Solid Particles

Solid contaminants may be thought of as being those which will not dissolve in fuel. Most common are iron rust and scale, sand, and dirt. Also included, however, are such items as metal particles, dust, lint, particles of filter media, rubber, valve lubricants and even sludge produced by bacterial action. Solid

contaminants can be collected by the fuel at every stage of its movement from the refinery to the aircraft.

Surfactants

The term "surfactants" is a contraction of the words SURFaceACTiveAgeNTS. These are soap or detergent like materials that occur naturally in fuel or may be introduced by refining processes, by inclusion of certain additives into the fuel, or may be washed off internal surfaces by the passing of the fuel through pipelines or by storage in a tank or vessel which had previously handled other products.

Surfactants are usually more soluble in water than in fuel and reduce the interfacial tension between water and fuel, thereby stabilizing suspended water droplets and contaminants in the fuel. This ability to suspend water and dirt in fuel may disarm filter/separator action and permit these contaminants to get into the aircraft. This property has resulted in surfactants becoming one of the major contaminants in aviation turbine fuels, and can cause fuel gaugeproblems.

Surfactants, in large concentrated quantities, usually appear as a tan to dark brown liquid with a sudsy-like water/fuel interface.

Water

Although it has always been present in aviation fuel, water is now considered to be a major source of contamination. The ability of turbine fuel to entrain water and the possibility of ice crystals interrupting fuel flow makes water in the fuel a major hazard to flight safety.

Micro-Organisms

There are over 100 different species of micro-organisms which can live in the free water which accumulates in sumps and on the bottom of storage and aircraft tanks. Many of these microorganisms are airborne, while others are found in the soil. Thus, fuel is constantly exposed to inoculation with this type contaminant. The conditions most favourable to their growth are warm temperatures and the presence of iron oxides (rust) and mineral salts in thewater.

The principle effects of micro-organisms are:

- Formation of a sludge or slime which can foul filter/separators and aircraft fuellingmechanisms
- Emulsification of thefuel
- Creation of corrosive compounds and offensiveodours.

The brown-black debris on the bottom of this tank is made up of fungus and bacteria. Fuel microbial growth can clog fuel filters, causing the aeroplane fuel quantity indication system to read incorrect values, and eventually cause structural corrosion of the aluminium stringers and wing skin. Operators can greatly reduce the chance of microbial growth by draining water from aeroplane fuel tanks weekly and by testing for microbes annually.

Fig. 10.71 Microbial contamination in a wing tank

Severe corrosion of aircraft tanks has been attributed to micro- organisms and considerable expense has been incurred removing microbial growths and repairing or replacing corroded aluminium panels in wingtanks.

Microbial contamination in avgas is much less common than with jet fuel, although it does occur. The lower occurrence in avgas is due to the toxicity of tetraethyllead.

The best approach to microbial contamination is prevention. And the most important preventive step is keeping the amount of water in the fuel storage tank as low as possible.

Biocides can be used when micro-organisms reach problem levels. But biocides have their limits. A biocide may not work if a heavy biofilm has accumulated on the surface of the tank or other equipment, because then it doesn’t reach the organisms living deep within the biofilm. In such cases, the tank must be drained and mechanicallycleaned.

Even if the biocide effectively stops biogrowth, it still may be necessary to remove the accumulated biomass to avoid filter plugging. Since biocides are toxic, any water that contain biocides must be disposed of appropriately.

Detection of Contaminants

Because solid contaminants generally appear in relatively small numbers and sizes in relation to the volume of fuel, their detection can be difficult. Aviation gasoline is generally considered “clean” if a one-quart sample is clear of any sediment when viewed in a clean and dry glass container. It may be helpful to swirl the container so that a vortex is created. The solid contaminants, if present, will tend to collect at the bottom beneath the vortex.

Turbine fuels must of necessity be several orders of magnitude cleaner than aviation gasoline. While the above visual test is adequate for operational checks, it is necessary, from time to time, to check the operation efficiency and cleanliness level of a turbine fuel system with a tool which is more critical than a clear bottle. The aviation industry has adopted the Millipore test for this purpose.

The Millipore is a filter-type test capable of detecting microscopic solid contaminants down to 0.8 of a micron in size, which is approximately 1/120 the diameter of a human hair. An evaluation guide is provided, containing the instructions for conducting these tests, along with the means of evaluating the results.

The “white bucket" test is particularly helpful in detecting the presence of concentrations of surfactants in turbine fuel. All that is required is a clean white porcelain bucket and water which has been in contact with the fuel in tank bottoms, filter/separators or other points where surfactants are likely to accumulate. Surfactants, if present, will appear as a brown sudsy water layer on the bottom of the bucket or at the fuel-water interface.

Evidence of microbial growth or debris may appear as black sludge or slime, or even a vegetative-like mat growth. Growths also appear as dark brown spots on some filter/separator socks.

Human Error

Miscellaneous contaminants can include either soluble or insoluble materials or both. Fuel can be contaminated by mixing it with other

grades or types of fuels, by picking up compounds from concentrations in rust and sludge deposits, by additives, or by any other of a number of soluble materials.

The greatest single danger to aircraft safety from contaminated fuels cannot be attributed to solids, exotic micro-organisms, surfactants, or even water. It is contamination resulting from human error. It is the placing of the wrong grade or type of fuel into an aircraft, the mixing of grades, or any other type of human error that allows off specification fuels to be placed aboard the aircraft.

Any fuel which is suspected to be off-specification because of contaminants or mixing with other fuels should not be placed aboard an aircraft. If in doubt, immediately arrange for laboratory and other tests to definitely establish whether the fuel may be used for aviation purposes.

Tests for Detecting Presence of Water

Samples should be taken according to the regulations of the airport fuel companies and the aircraft operator. The following tests are used all over the world:

- Clear and Bright

When this term is applied to a fuel test sample taken in a clear glass jar, it means that the fuel is completely free of visible solid contamination and water (including any resting on the bottom or sides of the container). The sample must also possess an inherent brilliance and sparkle in the presence of light. (Cloudy or hazy fuel is caused, usually, by free and dispersed water but it can also occur because of finely divided dirtparticles.)

- Shell Water Detector(SWD)

This test consists of a small yellow capsule, fitted to a syringe,whichisthenexposedtoa5mlfuelsampledrawn through the syringe. The colour of the centre of the capsule changes according to water content, and changes to green when there is a positive indication of water contamination (around 30 ppm). Capsules may only be used once.

Fig.10.72 Shell Water Detector (SWD)

- Hydrokit

A “Go No-Go” type of water detector. The test consists of exposing a pre-measured fuel sample to a reactive powder, which is colour sensitive to free water in concentrations of about 30 ppm or more. The powder changes from white to pink if the fuel contains about 30 ppm or more of free water after two minutes of contact with thefuel.

Let’s go through a test using the Hydrokit:

- Take a sample from the delivery vehicle as example a fuel truck. The sample must be taken from the lowest point where a sample/drain valve is provided. See Figure10.68.

- Let the sample becomestable.
- If the sample moves into two layers, it contains water. Continue to drain from the water drain valve until it has no water.
- If the sample stays in one layer, it can be all fuel or allwater.

To find if the sample is fuel or water, do the test that follows:

- Use the Hydrokit to fill the test tube containing the reactive powder as shown in Figure10.70.
- If the powder stays white, the sample is fuel and the test is complete.
- If the white powder changes color to purple, the sample is water. Continue to take samples and test until all the water has beenremoved.

If you have a “no specific” hydrokit available, put the reactive powder direct into the sample.
Alternative Procedure
If you do not have a hydrokit available, add water to the sample. If the sample moves into two layers, the sample taken is all fuel and the test is complete. If the sample stays in one layer, the sample is water. Continue to take samples and test until all the water has been removed.
To discard the samples, refer to the local regulations.

Fig. 10.73 aking a sample from the fuel tanker / bowser

Fig. 10.74 Taking a sample from the fuel tanker / bowser

Fig. 10.75 Filling the test tube

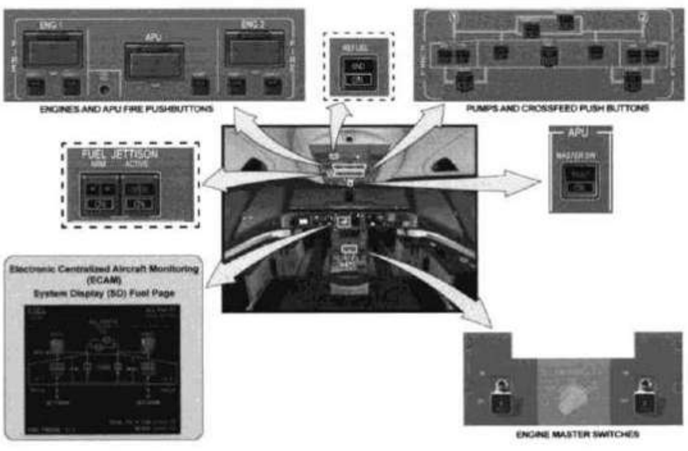


Fig. 10.76 Samples indicating clean and water-

contaminated fuel
Control and Indication Flightdeck
The cockpit controls for the fuel system include:

- Engine and APU are push buttons on the cockpit overhead panel
- Automatic REFUEL push buttons on the cockpit overhead panel(option)
- Pumps and crossfeed valves control push buttons on the fuel overheadpanel
- APU master Switch on the cockpit overheadpanel
- Engine Master Switches on the cockpitpedestal
- JETTISON ARM and ACTIVE push buttons on the cockpit overhead panel(option).

The cockpit indications for the fuel system are shown on the ECAM System Display (SD) Fuel page.

Fig.10.77 Fuel controls in the cockpit

Fig.14. Flightdeck interfaces External
The aircraft external controls for the fuel system include:

- APU emergency shutdown switch on the nose landing gear ground servicepanel
- Refuel/Defuel and transfer controls on the refuelpanel
- APU emergency shutdown switch on the refuelpanel

The external indications for the fuel system are shown on the refuel panel.

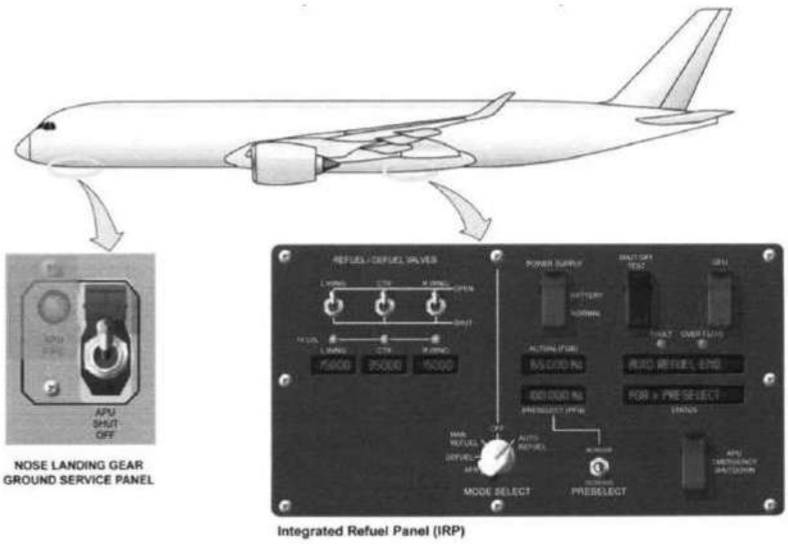


Fig. 10.78 Refuel panel Warnings
Fuel systems are provided with different warning sub-systems which generate cautions at different levels. They are shown in the cockpit, a few of them also on the fuelling panel. Normal fuel systems warn on the following parameters:

- tank level (high andlow).
- system components malfunctions (pump low press,etc.).
- fuelimbalance

- impossible fueltransfer

Warnings could look like the following examples:

Warning shown to flight	Description
L (R) TK PUMP 1 + 2 LO	Both tank pumps inoperative
L (R) XFR VALVE FAULT	Valve position disagree
L (R) WING TK	
AUTO TRANSFER FAULT	CTR TK >250kg and L orR WING TK <5000kg.
L (R) WING TK LOLVL	Low fuel level in either L or
ENG 1 (2) LP VALVE	Valve disagree in open
L (R) TK PUMP 1 (2)LO	One of two tank pump
X-FEED VALVE FAULT	Valve position disagree.
FUEL INBALANCE	Different fuel quantities between wing tanks.
etc.	

Fig. 10.79 Warnings

Fuel Tank Inerting System General

The Fuel Tank Inerting System (FTIS), or On-Board Inert Gas Generating System (OBIGGS, or IGGS), replaces the flammable gas space above the fuel tank (ullage) with a non-flammable atmosphere. Fuel Tank inerting systems in aircraft displace the oxygen in the tank with nitrogen.

Based on a hollow fibre separation process, the OBIGGS produces the flow of Nitrogen Enriched Air (NEA) to protect the aircraft. Part of the oxygen contained in the fuel tank is replaced by Nitrogen, keeping the vapours non-flammable.

The OBIGGS includes a filtration system, a pressure and temperature controller, an oxygen sensor and air separation modules. Depending on the applications, it can integrate a Temperature Management System to condition the bleed air.

Compressed air is directly tapped from the aircraft pneumatic system and transformed into NEA that is directly distributed to the fuel tanks.

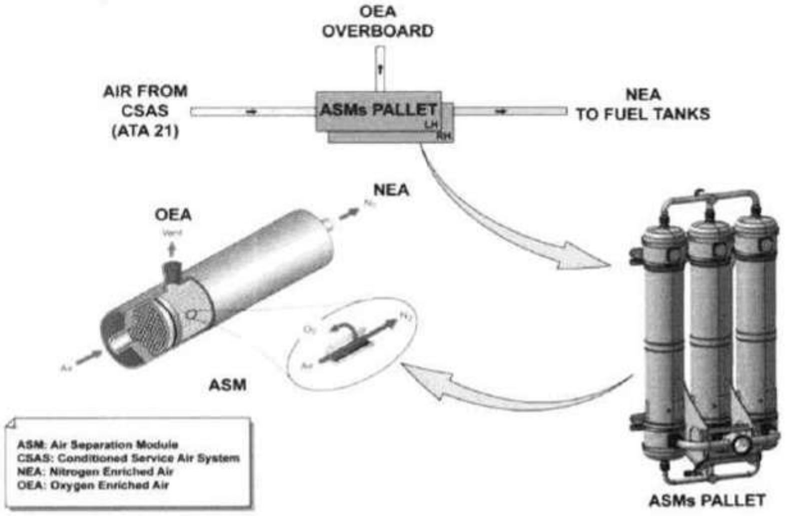


Fig. 10.80 On-Board Inert Gas Generating System (OBIGGS)

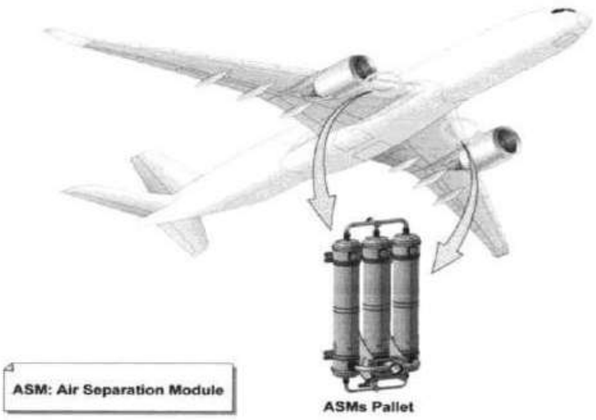


Fig. 10.81 Air Separation Modules location (Airbus A350) Typical System

The following is a description of the Inert Gas Distribution System (IGDS) as fitted to the Airbus A350:

The Inert Gas Distribution System (IGDS) supplies Nitrogen Enriched Air (NEA) to the three fuel tanks.

Distribution pipes allow NEA supply to fuel tanks and check valves prevent fuel reverse low from fuel tanks to IGDS.

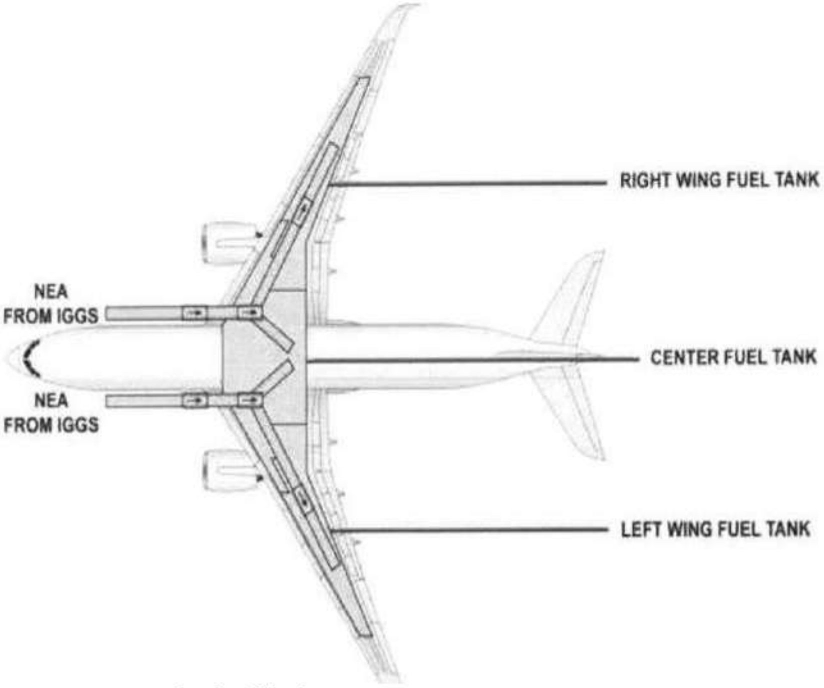
There are distribution pipes in the wings leading edge and inside the fuel tanks.

There are six check valves:

- One check valve in each wing leadingedge
- Four check valves inside the fueltanks.

The On-Board Inert Gas Generating System application

interfaces with the Landing Gears Extension/Retraction System application for flight/ground condition,



as the Inert Gas System operates only in flight.

Fig. 10.82 Inert Gas flow distribution (Airbus A 350)

The Inert Gas Control System controls the supply of conditioned air from pneumatic system to Air

Separation Modules (ASMs). It controls the NEA flow to the fuel tanks.

Temperature Isolation Valves (TIV) controls the conditioned air supply to ASMs and Dual Flow Shut Off Valves (DFSOV) controls the NEA flow to fuel tanks.

Pipelines and Couplings General

Pipelines in aircraft fuel systems are not subjected to high pressures, and rigid pipes are generally manufactured from aluminium alloy tubing, although fire resistant and fireproof materials, such as stainless steel or titanium, must be used forward of the engine bulkhead and in other specified areas. Pipe ends are flared or beaded to accept the specified type of coupling. Some vent and jettison pipes are built into the structure, and in certain cases are of square section for ease of manufacture. Standard AGS or AS pipe couplings are available in sizes up to 2.5 in. diameter, and these are often used in aircraft fuel systems; however, where flexibility is required in joints, because of flight loads and temperature variations, specially designed couplings may be employed. A number of non-standard couplings are described and illustrated in the followingpages.

Flexible Couplings

Two types of flexible coupling are illustrated in Figure 10.78. Diagram (A) shows a coupling which has provision for a certain amount of misalignment, as well as both angular and axial movement of the pipes. The pipe ends are beaded, and the surfaces within the joint are smooth and polished, so that the seals may slide freely over the pipes. A split retainer encloses the beads. When the coupling nut is tightened on the body, the O- rings are squeezed between the gland washers and the split retainer, and expand to form a seal between the body and the pipes. Sketch (B) shows a coupling which is less flexible, but which has provision for a limited amount of misalignment and movement. When the inner and outer sleeves are screwed together pressure is applied to the split collars, and the rubber sealissqueezedouttoformasealbetweentheinnersleeve and the pipe beads.

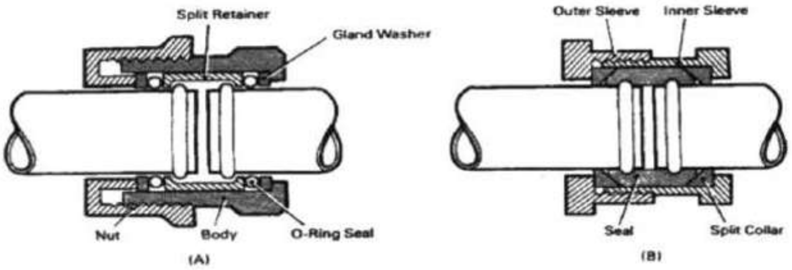


Fig. 10.83 Flexible coupling Vee-clamp Couplings

Figure 10.79 illustrates a typical Vee-clamp coupling. With this coupling a special fitting is welded to each pipe end, and the two fittings are held together by a pair of vee-section, semicircular clamps. The seal is formed by an O-ring, which is located in a groove in one fitting, and is pressed against the face of the other fitting when the clamps are tightened. In some instances, fail- safe links are fitted to the clamps.

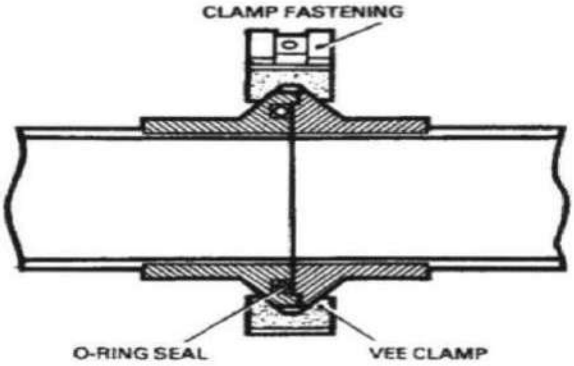


Fig. 10.84 Vee-clampcoupling

Sliding Coupling

Where only air or vapour passes through a pipe, a sliding coupling (Figure 10.80) may be used. As with vee-clamps, a special fitting is welded to each pipe end. An O-ring forms the seal, and the coupling is assembled by sliding the inner sleeve into the outer sleeve, so that the O-ring is located centrally.

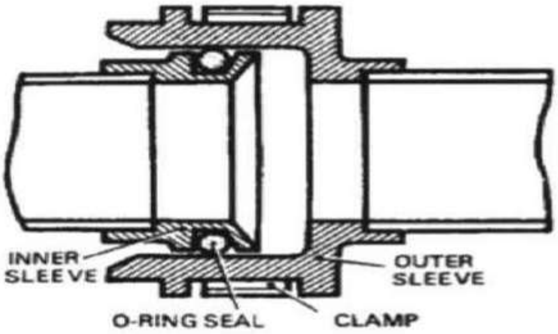


Fig. 10.85 Sliding coupling Bonding and Drip Shields

Bonding of fuel system pipes is very important, since many of these are contained within the fuel tanks, and static electricity must be prevented from causing sparks in this explosive atmosphere. Bonding strips or cables are used to form a conducting path across couplings, and between pipes and adjacent structure. A typical bonding installation is included in Figure10.81.

In certain positions in the aircraft, couplings may be enclosed in drip shields, or heat shields, for safety reasons. Draining facilities are often provided on these shields, and a typical installation is shown in Figure10.81.

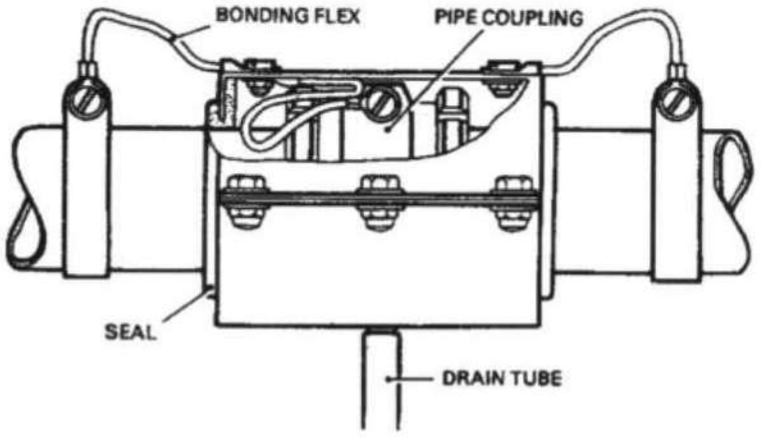


Fig. 10.86 Drip shield, also showing bonding connection Identification

Pipes are marked for identification purposes. The identification labels are taped at regular intervals along thefuel pipeline. The label identifying fuel is red, with a line of 4-point stars, and is marked with the word"FUEL".

Fig. 10.87 Fuel pipeline identification label

Maintenance

The fuel system is very important to the safe and efficient operation of an aircraft, and particular care must be taken to ensure that the instructions and precautions contained in the relevant manuals, schedules, and servicing instructions, are properly carriedout.

SafetyPrecautions

The flammability of a fuel depends to a large extent on its flash point, and the different types of fuel vary considerably in this respect. Kerosene is far safer to handle than gasoline, but, regardless of the type of fuel used in a particular system, it is essential that precautions are taken to prevent the combustion of fuel vapours during servicing operations. In addition, precautions must be taken to prevent the harmful effects to health which may result from handling fuel or inhaling fuel vapour.

The following general precautions should be observed whenever the fuel system is being worked on, and the relevant manuals should be consulted for any requirements which are applicable to a particular aircraft or fuel system.

- The aircraft should be electrically earthed, and any ground equipment or containers should be earthed to theaircraft.
- Suitable and adequately manned fire-fighting equipment should be available, and suitable warning notices should be prominentlydisplayed.
- Aircraft electric supplies should be switched off, and no live electrical cables should be leftdisconnected.
- Only spark-proof electrical equipment should be operated in the vicinity of theaircraft.
- Explosion-proof lamps and torches should beused.
- Whendrainingfuel,anyprecautionsdetailedintherelevant

Maintenance Manual regarding centre-of-gravity movement or maximum permitted jack loads, must be observed.

- To prevent undue spillage of fuel, tanks and pipes should be drained or isolated as appropriate, before breaking a connection or removing acomponent.
- Air-fed respirators should be worn in areas of high vapour concentration, e.g. near an open tank accesshole.

Refuelling/Defuelling

When the aircraft is to be refuelled or defueled, precautions must be taken to provide a path to earth for any static electricity which may be present, or which may build up as a result of the fuel flow. Refuelling and defuelling should normally be carried out in the open air, and suitable fire extinguishing equipment should be available and adequately manned. Both the aircraft and the refuelling vehicle should be earthed to a point which is known to be satisfactory, and an 'escape route' for the refuellingvehicle should be kept clear. When the aircraft is to be pressure refuelledthe earthing wire on the refuelling pipe should be connected to the earth point provided on the aircraft, before connecting the refuelling pipe, and when the aircraft is to be refuelled through the over-wing filler port, the earthing wire on the refuellingpipe should be connected to the earth point provided on the aircraft before removing the filler cap and inserting the nozzle. The earthing wire should remain in position until after the refuelling pipe is disconnected, or the filler cap replaced, as appropriate. Similarly, when defuelling, the earthing wire should be connected first and disconnected last. No radio or radar equipment should be operated while refuellingor defuelling is taking place, and only those electrical circuits concerned with these operations should be switchedon.

NOTE: Since pressure refuelling rates are high, the failure of an associated float switch or fuel level shut-off valve could cause a rapid build-up in pressure, and possibly serious damage to the tanks. Persons refuelling an aircraft should be prepared to shut off the supply instantly, should the automatic cut-off system fail tooperate.

Fuel Leakage

When leakage or spillage of fuel has occurred, care must be taken to ensure that all traces of fuel and vapour are removed. Where lagging has become contaminated with fuel in areas adjacent to passenger cabins and crew compartments, the lagging should be removed and cleaned, and any residual fuel should be mopped up. Where fuel has leaked into a compartment which is vented and drained, the venting and drainage arrangements should be checked to ensure that they are functioning correctly, and that there is a flow of air through the compartment. It is sometimes specified that a check of the venting system of such a compartment should be carried out with the cabin pressurised. In the event of a gross leakage, consideration should be given to the effects that fuel may have on other materials and components, such as cable insulation, seals, transparencies and bearings

Cleanliness

Scrupulous cleanliness is essential for correct and safe operation of an aircraft fuel system. This applies not only to the installed fuel system, but also to any ground equipment (e.g. test rigs, containers, refuelling vehicles and storage tanks) used in connection with it; a check for contamination in ground equipment should be carried out on a regular, planned basis. Foreign matter or contaminants in an aircraft fuel system can cause serious corrosion or damage to tanks and components, and may result in engine malfunction.

Whenever an orifice or a connection in a fuel system has unavoidably to be left open, protection against the entry of foreign matter must be provided, by the use of blanks or specially designed covers; these blanks or covers must remain in place until the orifice or connection is finally closed. The blanking of all openings in components removed from a fuel system is equally important, especially when the components are being returned for investigation; any foreign matter retained in the component may provide a clue as to the cause of the failure ormalfunction.

In many cases, standard AGS blanks, made from rubber, plastic or metal, may be suitable for use in a fuel system, but many components have non-standard connections, and these necessitate the use of blanks which are specially designed for the purpose, and made from a material which is compatible with the fuel. No material from which particles are easily detached (e.g. cotton, paper, wood and cork) should be used for blanking purposes. Blanks must be designed so that it is impossible to re- connect the attaching component with a blank inplace.

When jointing compound is used during installation or assembly of a component, care must be taken not to use an excessive amount, otherwise surplus compound may enter the system, and block or damage components such as valves, pumps and filters. Surplus compound should be wiped off whilst stillwet.

The tanks, filters, and the lowest points in a number of feed and vent pipes, are fitted with drain valves, by means of which fuel samples may be taken after refuelling, and at the periods specified in the approved Maintenance Schedule. Fuel samples are normally collected in a glass jar, and should be inspected for the presence of free water, sediment and microbiological contamination. If excessive free water or sediment is found in the samples, all fuel should be drained from the system, and the tanks should be partially filled with clean fuel. This fuel should then be drained through the drain valves until samples are satisfactory. If microbiological contamination is found in the samples, the tanks should be visually inspected for fungal deposits, which, if present, must be removed. After removal of fungus, the fuel feed system must be flushed through to remove microbiological debris. Whenever excessive contamination of any sort has been found, the system filters should be inspected, and cleaned or replaced asnecessary.

Component Removal and Installation

In order to remove a component (except for those in the top of a tank) it will usually first be necessary to drain some, if not all, of the fuel in a tank, or to close the low pressure, transfer, or servicing valves, so as to isolate part of the system. The normal defueling system should be used to remove large quantities of fuel, but small quantities may be drained into suitable containers, using the water drain valves. The fire and safety should be observed when defueling and when working on the fuel system. Actual procedures for removing or installing components should be obtained from the appropriate Maintenance Manual, but the following general points should be taken into account.

Removal

• Any electrical circuits which have to be disconnected, should first be isolated by removing the associated fuse, or by tripping the associated circuit breaker, as applicable.

• Care should be taken not to twist or strain the pipes, when removing union nuts; the use of two spanners is recommended wherever possible.

• In order to remove a component, it may be necessary to remove adjacent pipe clips in order that pipes may be withdrawn without damage. Care must be taken not to dent or score pipe flares or mating surfaces when removing the associated component.

• Provision should be made for the collection of any fuel which may drain from the pipes when they are disconnected. Any spillage should be moppedup.

• Any nuts and washers which are removed should be retained, but seals and gaskets should bediscarded.

• Blanks or covers should be installed on openings and pipe ends, including those on the component which wasremoved.

Installation

• A component which is drawn from stores for installation on an aircraft, should be checked to ensure that it is the correct part, is to the required modification standard, and has the appropriate test and inspection markings; the remaining life of any seals or rubber components should also bechecked.

• An inspection should be made for any damage or corrosion which may have occurred during storage. Any position indicators on the component should be checked for correct setting.

• Components such as valves, which have adjustable stops, should be checked to ensure that the stops are adjusted to the correct position of thevalve.

• Any component which is treated with inhibiting oil, should be thoroughly flushed with system fluid, and dried with a lint-freecloth.

• New seals and gaskets should be fitted, blanks should be removed, and the component should be installed in position. Care should be taken not to damage associated joint faces, pipes, or threads. Mating parts should be checked for alignment and fit; they should not be forced intoposition.

• Modern aircraft fuel system components are so designed that it should be impossible to install them incorrectly. However, when fitting physically reversible units, or components with adjacent unions of similar size, care should be taken to ensure that the pipes are correctlyconnected.

• Where recommended, anti-seize compound should be applied to threads. To prevent

contamination of the system, the compound should be applied sparingly, and should only be applied to the male thread on a pipe union. Seals should normally be lubricated with mineral jelly or an approved alternative; joint gaskets are normally fitted dry, but use of a sealant or jointing compound may be recommended in some instances.
• Nuts, bolts and pipe unions should be tightened to the recommended torque values, and the bonding wires and dips which were removed from adjacent parts should be replaced as originallyinstalled.
• Any electrical connections to the component should be made before re-connecting the supply, and the unit should, where appropriate, be checked for full and free movement in the correctsense.
• Manual controls on valves should be checked to ensure that the valve operates in the correct sense, and reaches its stops before the associated control; the clearance between the control and its stops should be checked and adjusted to within the limits specified in the relevant manual. Controls should be locked afteradjustment.
• The operation of limit switches on electrically-operated mechanisms, should be checked against the position of the component, and should be adjusted asnecessary.
• Any indicators in the crew compartment, such as magnetic indicators and warning lamps, which are associated with the component being installed, should be checked forcorrect

operation.
• Bonding should betested
• When installation is complete a flow test and/or pressure test should be carriedout.

Filters

A light aircraft fuel filter normally consists of a housing, a filter element, a sediment bowl, and a drain valve. Water and sediment may be drained from the bowl prior to flight, and the bowl should be removed periodically for cleaning and inspection of the filter element. These filters are often placed in the suction line to the pump and, when replacing the bowl, care should be taken to ensure that it forms a good seal with the housing; a leak could result in inadequate fuel supply to theengine.

The main filters fitted to turbine engined aircraft areusually fitted to the engine, and consist of a housing, a felt or paper filter element, a filter case, and a drain valve; in some aircraft the housing incorporates a differential pressure switch, which operates an icing warning lamp in the flight compartment. The drain valve may be used to take fuel samples, but precautions may need to be taken to avoid the need to bleed the engine fuel system; these precautions normally include dosing the high pressure fuel cock, opening the low pressure fuel cock, and using the tank booster pump to discharge the fuel sample. New filter elements should be fitted at the periods specified in the approved Maintenance Schedule, and whenever breakdown, repairs, or contamination of the system have taken place. To remove a filter dement, the high and low pressure fuel cocks should be dosed, the associated electrical services should be isolated, and the element and case should be removed; any debris should be examined, and the source located. Before fitting a newfilter element,thecaseshouldbewashedoutwithkeroseneor trichloroethylene, and old seals or gaskets should be replaced; it is usually recommended that seals are lubricated with kerosene or petrolatum prior to installation. After fitting a filter element, it is usually necessary to run the associated engine, and to check the system for satisfactory operation and freedom fromleaks.

Pressure Tests

Pressure tests are normally required at regular intervals, after repairs, modifications, and replacement of components, and whenever leakage is suspected. In those vent systems which utilise part of the wing structure (e.g. top hat sections) to form the vent duct, vent pressure tests may also be required after structural repairs. The tests required will be specified in the relevant Maintenance Manual, and should be carefully carried out. Test rigs, capable of supplying fuel or air under pressure, are required, and should include an accurate pressure gauge, a relief valve, and, in the case of a fuel pumping rig, a flowmeter. All test rigs should be clearly identified with the certification (or re- certification) date. In addition, special blanks, plugs, cover plates, and dummy components may be required. The vent, feed, and transfer systems are usually tested separately since different test pressures are generallyprescribed.

Vent System Pressure Test

For this test, the vent system on each side of the aircraft should normally be tested separately. All vent openings should be blanked, and it will often be necessary to gag float-operated valves, or to replace them with dummy components. Alternative means of venting the tanks during the tests should be provided. Air pressure should be applied to the system either through a water drain valve, or through an adaptor fitted to one of the blanks, and the pressure should be slowly raised to the test pressure quoted in the relevant Maintenance Manual. When the

air pressure supply cock is turned off, any decrease in pressure will indicate leakage, and the drop in pressure over a prescribed time should be noted. The source of any leakage in excess of that permitted should be traced and rectification action should betaken

Feed System Pressure Test

The feed system from a tank to its associated engine should be tested individually. Crossfeed and inter-engine valves should be closed, and the low-pressure cock should be opened. On some aircraft the feed systems are pressurized by switching on both pumps in the tank concerned, whilst on others the booster pumps are replaced by dummy components, and fuel pressure is applied by means of an external test rig. In some systems there will be flow through the bleed hole in the suction valve, and this must be within prescribed limits. Rates of flow indicated on the test rig flowmeter, which are in excess of these limits, will be indicative of either an internal or external fuel leak. All pipes, connections, and valves should be checked visually for signs of leakage under pressure; no leakage is normallypermitted.

NOTE: The systems in which drip shields or heat shields are fitted to some couplings, the test pressure must be applied for a sufficient length of time to enable any leakage to collect and flow through the drain. Alternatively, a separate pressure test of the drip shield may be specified, or the shield may be required to be removed for thetest.

Transfer System Pressure Test

The pipes and couplings in the fuel transfer system may be pressure tested in a similar manner to the feed system. Pipes should be disconnected and blanked at the positions specified in the relevant Maintenance Manual, and fuel pressure should be applied by means of the transfer pump, or by use of an external test rig, supplying fuel through a dummy pump. No leaks should be evident, and no fuel flow should be recorded on the test rig Flowmeter.

Additional Pressure Tests

A number of other pressure tests may be specified, in order to ensure that there is no leakage which could prove hazardous, or prevent proper operation of the fuel system. One example is the pressure testing of conduits which pass through the fuel tanks, and house electrical cables. These conduits are usually- sealed by means of a pressure bung or pressure seal, and are tested by applying air pressure to the inside, through a drain pipe, or special adaptor. When the air supply is shut off, there should be no drop in pressure over a prescribed period of time. If leakage is evident at the pressure bung, it is usually permissible to apply sealant to seal the bung and the holes through which the cables pass

Flow Tests

Flow tests should be carried out in accordance with the relevant Maintenance Manual, as and when required by the approved Maintenance Schedule, or when necessitated by repairs, replacements or modifications. The tests are designed to ensure that the system will provide a fuel flow to each engine which is in excess of the requirements of the engine when it is operating at maximum power, and at a pressure suitable for proper operation of the carburettor or engine-driven pump, as appropriate. For all tests the aircraft should be levelled laterally and longitudinally, and the fuel tanks should contain the minimum quantity of fuel (i.e. unusable fuel plus sufficient for the test only); tank vents should be clear, and overwing filler capsshouldbefitted.Allequipmentusedshouldbebondedand electrically earthed.

Full Flow Test

A full flow test is normally only required after initial installation or major breakdown of the system. Fuel flow test rigs are required for the test, and should be located adjacent to each engine, with the test rig pump at the same level as the engine- drivenpump.

The rig inlet hose is usually connected to a self-sealing coupling on the engine bulkhead, and the outlet directed to a suitable container. An external electrical supply should be connected to the aircraft, in order to operate the fuel system valves and to check operation of the associated warning lamps and indicators. The test includes suction feed operation (using the test rig pump), pressure feed operation (using the aircraft booster pumps), and all possible combinations of cross-feeding, to ensure that fuel flow is satisfactory under all flight conditions. The schedule of test operations, and the flow rates and pressures which should be achieved, are detailed in the relevant MaintenanceManual.

For the suction test, the test rig pump is used to draw fuel from the tanks. Valve selections should be made according to the test schedule, and the flow rates and pressures obtained at each stage of the test should be recorded. These results should be within the limitations prescribed for the suction test.

For the pressure test, the aircraft booster pumps should be used to pump fuel from thetank.

The test rig pump is switched off, and its by-pass opened. Selections of pumps and valves should be made in accordance with the test schedule, and the flow rates and pressures obtained at each stage of the test should be recorded. These results should be within the limitations prescribed for the pressure test

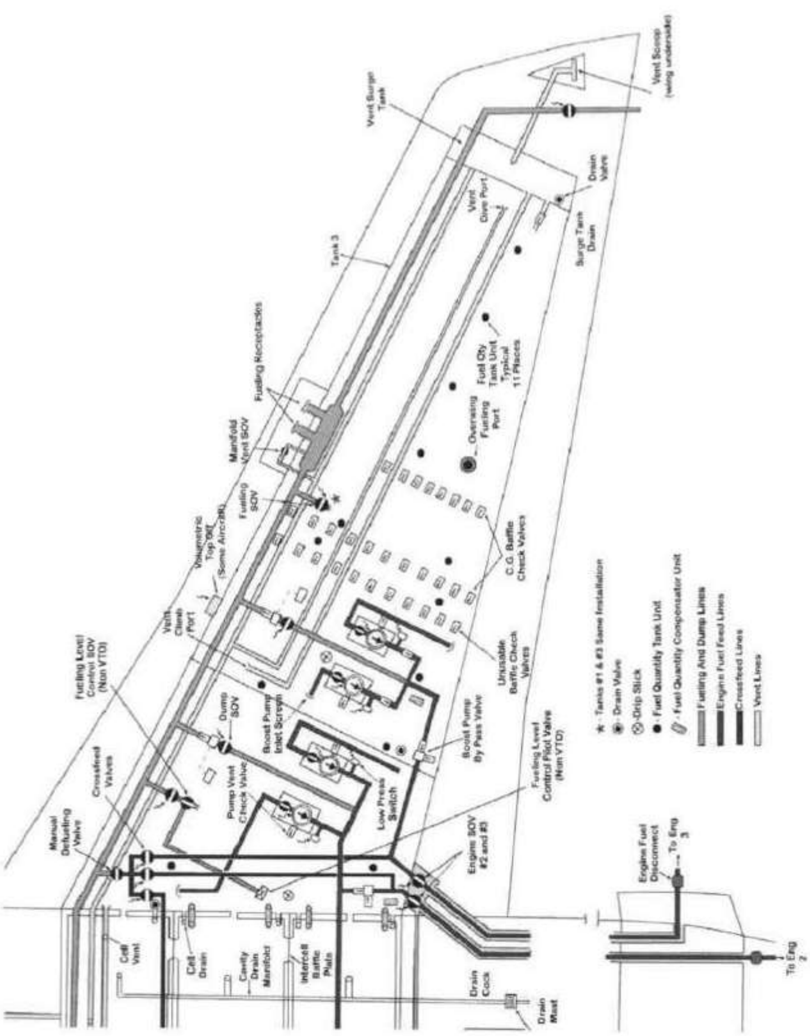
Limited Flow Test

A limited flow test is often considered as a satisfactory method of checking a fuel system after a component has been changed; only that part of the system affected by the component change needs to be tested. The fuel feed pipe is disconnected at the engine, or, in some instances, a drain pipe is connected to a special drain valve at the engine, and a suitable container is positioned to catch the drainedfuel.

The appropriate low pressure cock should be turned on, and the flow rates should be checked with the associated pumps operating separately and together. For each part of the test, when the fuel flow is free from bubbles, it should be directed into a calibrated container, and the time taken to pump a given quantity of fuel should be recorded. These figures should be converted to flow rates, which should not be less than the minimum flow rates specified in the relevant MaintenanceManual

Gravity Feed Test

To check agravity feed system such as is fitted to some light aircraft, the feed pipe should be disconnected at the carburettor, and a suitable container should be positioned below the engine. With the fuel outlet positioned at the same height as the carburettor, and the fuel valve turned on, the fuel should be checked for freedom from bubbles and for full-bore flow, then directed into a calibrated container. The time taken to drain a given quantity of fuel should be recorded, and the equivalent flow rate should not be less than the minimum flow rate specified in the relevant MaintenanceManual.



Boeing 727 Fuel System

Fig. 10.88

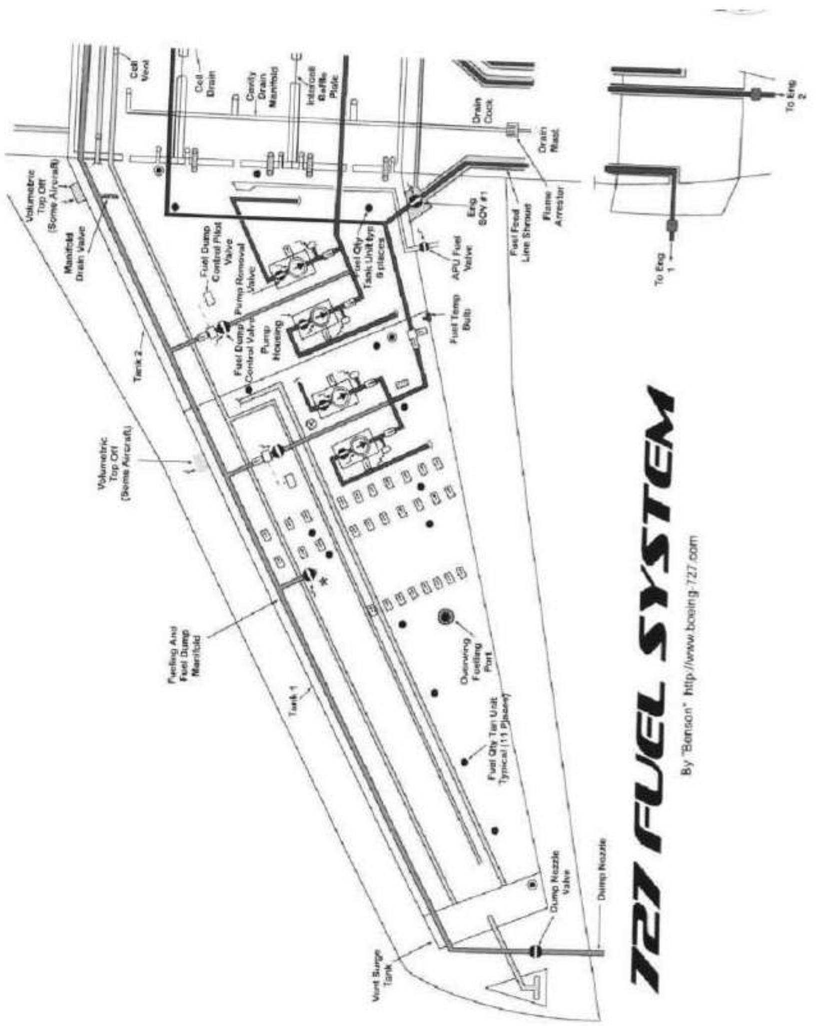


Fig.10.89 Boeing 727 Complete Fuel System

11.12: Ice and Rain Protection (ATA 30)

1. INTRODUCTION

The first electrical flight control system for a civil aircraft was designed by Aerospatiale and installed on the Concorde. This is an analog, full-authority system for all control surfaces. The first generation of electrical flight control systems with digital technology appeared on several civil aircraft at the start of the 1980s with the Airbus A310 program. These systems control the slats, flaps, and spoilers. The Airbus A320 (certified in early 1988) is the first example of a second generation of civil electrical flight control aircraft, rapidly followed by the A340 aircraft (certified at the end of 1992). The distinctive feature of these aircraft is that all control surfaces are electrically controlled and that the system is designed to be available under all circumstances. "Fly-by-wire" technology translates the pilot's actions into electronic signals, which computers use to manipulate flight controls. The computers constantly monitor pilot input and prevent the aircraft from exceeding its flight envelope, thereby increasing safety. And because fly-by-wire replaces heavy, complex mechanical linkages with lighter electrical wires, it is more efficient.

2. NEED OF FLY BY WIRESYSTEM

2.1. CONVENTIONAL PRIMARY FLIGHT CONTROLS SYSTEMS

This system employ hydraulic actuators and control valves controlled by cables that are driven by the pilot controls. These cables run the length of the airframe from the cockpit area to the surfaces to be controlled. This type of system, while providing full airplane control over the entire flight regime, does have some distinct drawbacks.

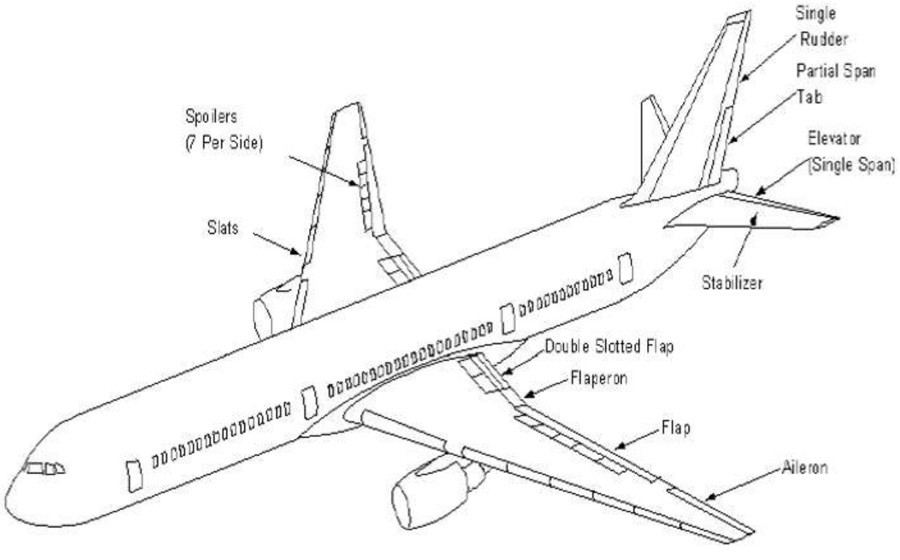


Fig. 12.1 Aircraft Flight Control Surfaces

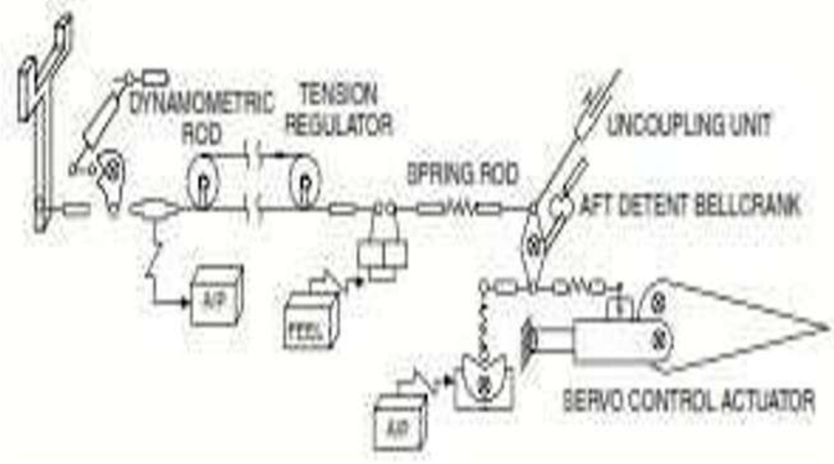


Fig. 12.2 Aircraft Mechanical Flight Controls

The cable-controlled system comes with a weight penalty due to the long cable runs, pulleys, brackets, and supports needed. The

Ice and Rain Protection (ATA 30)

Ice Formation Icing Hazards

In-flight icing is a serious hazard. It destroys the smooth flow of air, increasing drag, degrading control authority and decreasing the ability of an airfoil to lift. The actual weight of the ice on the aeroplane is secondary to the airflow disruption it causes. As power is added to compensate for the additional drag and the nose is lifted to maintain altitude, the angle of attack increases, allowing the underside of the wings and fuselage to accumulate additional ice. Ice accumulates on every exposed frontal surface of the aeroplane - not just on the wings, propeller, and windshield, but also on the antennas, vents, intakes, and cowlings. It builds in flight where no heat or boots can reach it. It can cause antennas to vibrate so severely that they break. In moderate to severe conditions, a light aircraft can become so iced up that continued flight is impossible. The aeroplane may stall at much higher speeds and lower angles of attack than normal. It can roll or pitch uncontrollably, and recovery may beimpossible.

Kinds of Ice and Its Effect on Flight

Structural ice adheres to the external surfaces of the aeroplane. It is described as rime, clear or glaze, or mixed.

- Rime ice has a rough, milky white appearance. Much of it can be removed by de-ice systems or prevented by anti- ice.
- Clear or glaze ice is smooth and generally followsthe

contours of the surface closely, however after further accumulation, it can form ridges. It is hard to remove.

- Mixed ice is a combination of rime and clearice.

Ice distorts the flow of air over the wing, diminishing the wing's maximum lift, reducing the angle of attack for maximum lift, adversely affecting aeroplane handling qualities, and significantly increasing drag. Wind tunnel and flight tests have shown that frost, snow, and ice naccumulations(on the leading edge or upper surface of the wing) no thicker or rougher than a piece of coarse sandpaper can reduce lift by 30 percent and increase drag up to 40 percent.

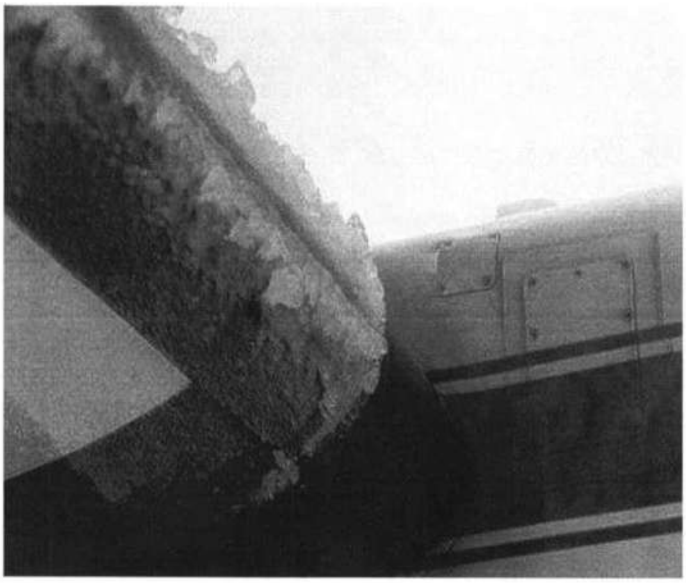


Fig.12.3 Ice formation on a wing leading edge

Larger accretions can reduce lift even further and increase drag by 80 percent or more. Even aircraft equipped for flight into icing conditions are significantly affected by ice accumulation on the unprotected areas. A NASA study revealed close to 50 percent of the total drag associated with an ice encounter remained after all the protected surfaces were cleared. Unprotected surfaces include antennas, flap hinges, control horns, fuselage frontal area, windshields, windshield wipers, wing struts, fixed landing gear,etc.

Ice forms on aircraft surfaces at 0 degrees Celsius (00) or colder when liquid water is present. However even the best plans have some variables. The following table illustrates the icing risk in terms of cloud type and ambienttemperature:

Ice Formation in Flight

Icing on aircraft is caused primarily by the presence of supercooled water droplets in the atmosphere. Ice will only form on an aircraft when it is flying in cloud or precipitation where the water droplets are supercooled. The term supercooled means the water is present in liquid form at a temperature below the freezing point of 00. In order to freeze, water must lose its Latent Heat to its surroundings: a droplet at rest can lose heat to the surrounding air slowly and so there is a tendency for a droplet to remain supercooled. When it strikes an aircraft however, the metal structure conducts away the Latent Heat and so itfreezes.

If the droplets impinge on the forward facing surfaces of an aircraft, they freeze and cause a build up of ice which may seriously alter the aerodynamic qualities. This applies particularly to small objects, which have a higher catch rate

efficiency than large ones, as small amounts of ice will produce relatively bigger changes in shape. The actual amount and shape of the ice build up depends on surface temperature, which results from an energy balance arising from heat input from viscous or kinetic air heating, kinetic heating by water droplets and the latent heat of fusion, and losses from evaporation or sublimation, convection and by warming the impingingdroplets.

Three different situations arise, depending on whether the surface temperature is less than, equal to or greater than 00. When the temperature is less than 00 all the impinging water droplets are frozen, and when it is above 00 none are frozen. However, for a particular set of atmospheric conditions and altitude it is found that there is quite a wide aircraft speed range over which the energy balance gives a skin temperature of 00 and this energy balance occurs at one end of the speed range by all the droplets freezing and at the other by none freezing.

The potential "catch rate” or "impingement rate" and the actual icing rate are thus not simply related in this region. The "no icing hazard" speed depends, therefore, upon the free water content of the atmosphere as well as the temperature and altitude. For severe conditions it is about the maximum speed of subsonic aircraft. The final influencing factor of note is that icing does not occur above about 12,000 m (40,000 ft) since the droplets are all frozen and in the form of ice crystals and will not adhere to the aircraft'ssurface.

Aero foils, engines, propellers, windows and radio antenna are included in those features of the aircraft which are required to be protected from iceformation.

Ice in almost any form constitutes a hazard to flight and it must be removed before the flight can be safely conducted.

Frost will form on any surface, if both the air temperature and surface temperature are at or below freezing. The amount of frost depends on the moisture content of the air. Frost does not add appreciable weight but it must be removed before flight because the boundary layer can be severely disrupted causing aero foil efficiency to be severelyreduced.

Aircraft that fly into clouds when the temperature is below freezing are liable to encounter supercoiled water droplets (water which is below freezing point but which is still liquid). When these droplets hit a cold soaked airframethey instantly form ice crystals. If sufficient ice crystals are formed the shape of the aero foil will be altered and efficiency willbe lost.

Icing in flight predominantly forms on leading edges and protrusions such as Pitot probes or antenna. Ice and Frost must always be removed before takeoff.

The Effects of Icing on an Aircraft duringFlight

During flight in certain conditions, ice may accumulate on all the forward facingareas of the aircraft. These accumulations have the following effects on anaircraft:

- A decrease in lift due to a change in shape of the wing aerofoilsection and loss of the streamline flow of air around the leading edges and topsurfaces.

- An increase in drag for the same reasons that cause a loss oflift.Theroughsurfaceprovidesincreasedskinfriction.

- Decreases propeller efficiency because of the change in bladeshape. There is the possibility of damage to the fuselage as a result of ice being flung off the propellers. There may be loss of control due to ice restricting or preventing movement of the controlsurfaces.

- An increase in weight causing loss of height. A change in the position of the weight can cause a change in the trim of the aircraft and possibly a loss ofstability.

- Blockage of pitot/staticports.

• Loss of vision through the cockpit windows orwindshields.

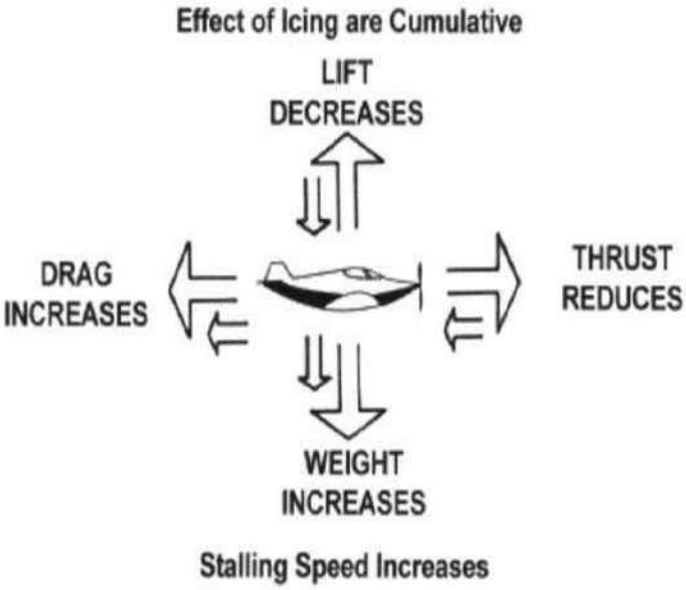


Fig.12.4 Effects of Icing on Aircraft during Flight

Classification of Ice Clear Ice
Clear ice will occur in very dense clouds where the water
particles are large and only slightly supercooled or in supercooled rain where the water drops are of large size. The kinetic energy of the water droplets produces a certain amount of heat on impact, and this may delay the freezing of the droplet. The water remains a liquid for a period and runs back over the surface until it has lost enough heat to freeze. The ice formed under these circumstances is a continuous smooth layer known as glaze ice or clear ice and is the most dangerous form of icing because it is difficult tosee.
Rime Ice
If the air temperature is very low and the cloud particles are small the water droplets striking an unheated surface freeze immediately on impact and produce a rough, relatively opaque ice formation known as rime ice.

Gleam Ice

This type of ice will form when complete freezing of the water particles on impact takes longer than in the case of rime ice. Such ice will form when the water particles are large and the air temperature very low. The remaining water will freeze sufficiently rapidly to trap some air, giving the ice an opaque appearance. The delay in the freezing of the residual water gives it time to flow back. The ice that forms will extend further back over the leading edge and the surface will not be as rough as rimeice.
Dry Ice

Icing does not occur above about 12,000 meters (40,000 feet) since the droplets are all frozen and in the form of ice crystals and will not adhere to the aircraft's surface. However, if the ice crystals are present in sufficient volume, they can accumulate in auxiliary cooling ducts and engine air intakes causingblockage.

Hoarfrost

Hoarfrost occurs on a surface which is below the freezing point of 00. It forms in clean air, water vapour being converted directly into ice crystals forming a white feathery coating. It sometimes occurs on the surface of the wings where integral fuel tanks arefitted.

Definitions

De-Icing - In the de-icing method the protection system is automatically switched on and off at regular intervals. During the 'off' period a certain amount of ice deposit is allowed to accumulate which will not seriously affect the aerodynamic shape of the surface. The ice deposit is then removed by operating the system for a short time. Typical De-Icing Systems are usedfor:

- Tail Leading Edges, Cooling TurbineOutlets

Anti-Icing- Where the anti-icing method is used, the protection system is switched on prior to encountering icing conditions and it remains on so that no ice is allowed to form on thesurface.

Typical Anti-Icing Systems are used for:

- Wing Leading Edges, Engine Air Intakes, AirIntakes
- Air Data Sensors, Cockpit Windows and Windshields, WaterOutlets

Airframe Icing Problem Areas
The main airframe icing problems are:

- Intakes: It has been found that some intakes, although heated, allow ice to form. Generally, engine intakes must be very clean in design, avoiding any projections; even rivet heads will cause sufficient turbulence to form an accretion point If the intakes are hinged to give engine access, the sealing at the hinge point must not offer any leakage.

- Windscreen Anti-Icing: Electrically- heated windscreens are completely satisfactory and also reliable, even in the most severeconditions.

- Outside Air Temperature (OAT) Gauge: Once in the icing range, temperatures are critical and an OAT gauge that is accurate to one degree isessential.

- Pitot/Static Systems: Most pitot heads are heated and operate satisfactorily in icing conditions. The combined pitot/static probe is excellent because both its sources are combined and the wholeheated.

- Grills: Most helicopters are fitted with a grill that may cover a fire- lighting access point or serve to ventilate a small gearbox. These grilles are usually made ofexpanded metal or wire mesh and are natural catchments and ice traps.

Ice Detection
The purpose of an ice detection system is to warn the flight crew of an impending ice builds up on the airframe during flight.Icing, if severe enough, can and has caused fatal accidents if notdetected.

Visual Method for Ice Detection
On some aircraft the flight crew must visually monitor ice build up on the airframe. Helpful tools for visual ice detection are:

- Ice detection spotlights
- Illuminated stick in front of the wind shieldframe

Ice Detection Spot Lights

Many aircraft have two ice formation spot lightsmounted one each side of the fuselage, in such a position as to light up the leading edges of the main-planes, when required, to allow visual examination for iceformation.

NOTE: In some aircraft, this may be the only method of ice detection.

Fig.12.5 Ice detection spot light areas

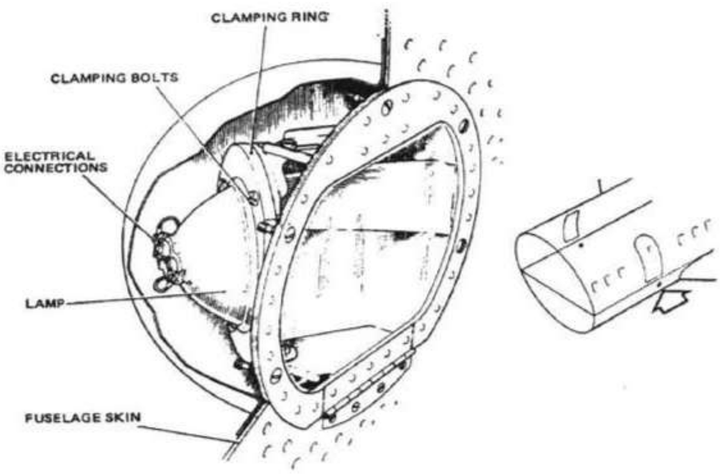


Fig.12.6 Ice detection spot light

Pressure Operated Ice Detector

The pressure operated ice detector consists of an elliptically shaped tube. Mounted on the base is a sensitive pressure switch, actuated by a diaphragm. The detector is mounted into the airstream. In the leading edge of the tube small holes are drilled, connected to the lower side of diaphragm. The big hole is connected to the upper side of the diaphragm. The total area of the small holes in the leading edge exceeds that of the big hole.

In case of ice accumulation on the leading edge, the small holes are blocked faster than the big hole. Thereforethe ram air pressure on the upper side of the diaphragm overcomes the pressure on the lower side and the pressure switch will be actuated.

When no ice is accumulated on the detector the higher pressure at the lower side of the diaphragm keeps the pressure switch open.

If the pressure switch closes due to ice accumulation the ice warning relay will be energized and the "ICE" warning light and the heater in the detector is activated. After approximately 20 seconds the ice is meltedand the pressure switch reopens. The ice detector is now ready for a new ice warning cycle. This cycling will continue until such time that the icing conditions no longerexist.

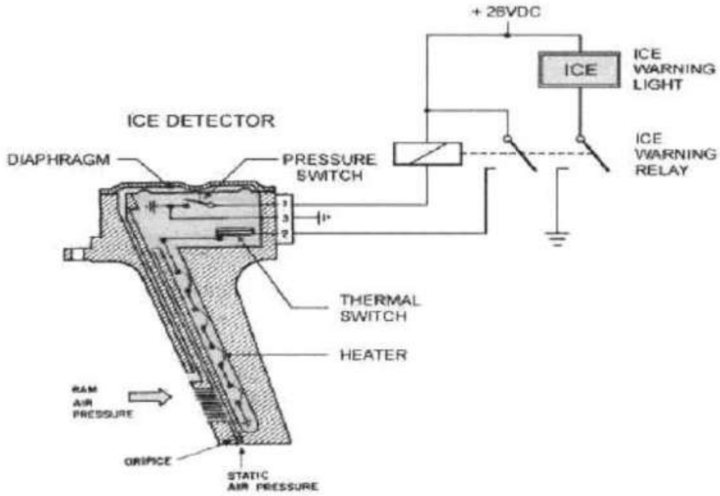


Fig.12.7 Pressure Operated Ice Detector Operation Vibrating Rod Ice Detector
This ice detector senses the presence of icing conditions and provides an indication in the flight compartment that such conditions exist. The system consists of a solid-state ice detector and advisory warning light. The ice detector is attached to the fuselage with its probe protruding through the skin. The ice detector probe (exposed to the airstream) is an ice sensing element that ultrasonically vibrates in an axial mode of its own resonant frequency of approximately 40KHz.

This vibration is monitored by a reference coil and compared to a reference oscillator in an ice detectorcontroller.

For redundancy normally two similar ice detection systems are installed. There are also other vibration rod ice detector systems with the sensor and the controller in the same housing.

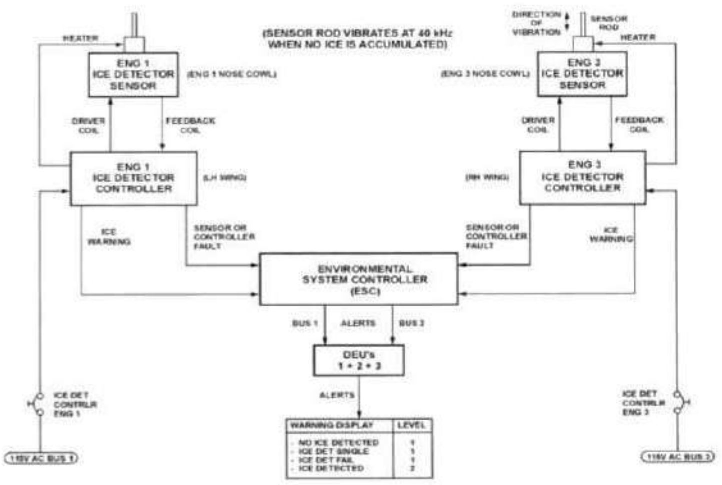


Fig.12.8 Vibration Rod Ice Detector (Typical for MD-11)

When ON, both detection systems with no ice present, the rods vibrate at the same frequency as the reference signal and the message NO ICE DETECTED comes on the warning display if still wing or tail or engine anti-ice systems are in use.

If ice accumulates on the rod, its weight will change and this will alter the vibrationfrequency.
If the discrepancy between the two signals exceeds a set amount the warning ICE DETECTED appears until all anti-ice systems are selected to ON and a heater is switched on to melt the ice on the probe to reset the detector. The heater will remain on until the two signals match again. Failures in one or both detection systems will set the warning ICE DET SINGLE or ICE DETFAIL.
CAUTION: Do not touch the vibration rod it gets hot veryfast.

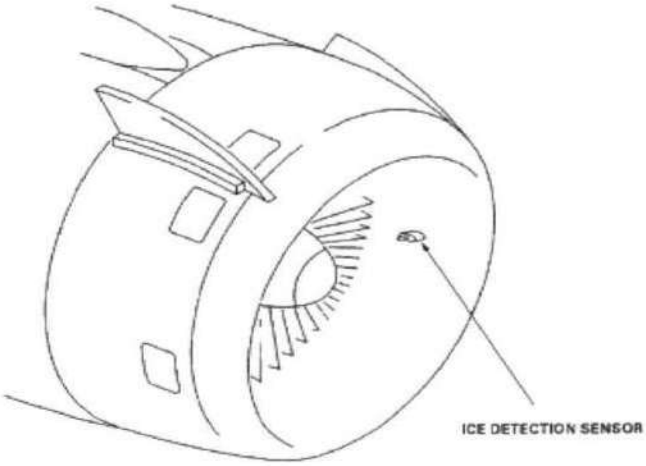


Fig.12.9 Ice Detector Sensor Installation Hot Rod Ice Detector

This consists of an aluminiumalloy oblong base (called the plinth) on which is mounted a steel tube detector mast of aerofoil section, angled back to approximately 30°from the vertical, mounted on the side of the fuselage, so that it can be seen from the flight compartment windows. The mast houses a heating element, and in the plinththere is a built- infloodlight.

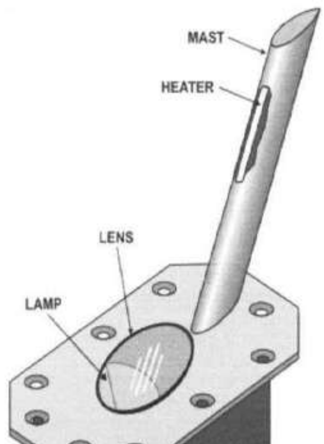


Fig.12.10 Hot rod ice detector

The heating element is normally off and when icing conditions are met ice accretes on the leading edge of the detector mast. This can then be observed by the flight crew. During night operations the built-in floodlight may be switched on to illuminate the mast. By manual selection of a switch to the heating element the formed ice is dispersed for further observance.

Serrated Rotor Ice Detector Head

This consists of a serrated rotor, incorporating an integral drive shaft coupled to a small AC motor via a reduction gearbox, being rotated adjacent to a fixed knife-edge cutter. The motor casing is connected via a spring-tensioned toggle bar to a micro-switch assembly. The motor and gearbox assembly is mounted on a static spigot attached to the motor housing, and together with the micro-switch assembly, is enclosed by a cylindrical housing. The detector is mounted through the fuselage side so that the inner housing is subjected to the ambient conditions with the outer being sealed from the aircraft cabinpressure.

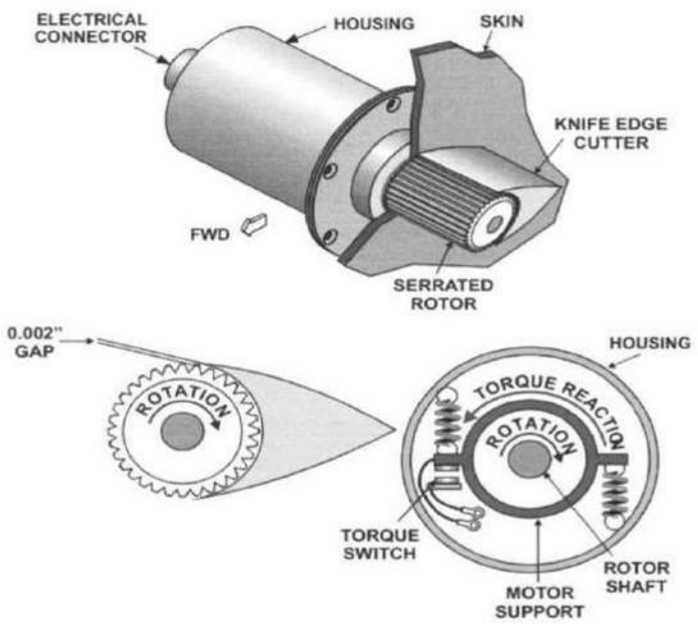


Fig.12.11 Hot rod ice detector

The serrated rotor on the detector head is continuously driven by the electrical motor so that its periphery rotates within 0.050 mm (0.002 in) of the leading edge of the knife-edge cutter. The torque therefore required to drive the rotor under non-icing conditions will be slight, since bearing friction only has to be overcome. Under icing conditions, however, ice will accrete on the rotor until the gap between the rotor and knife edge is filled, whereupon a cutting action by the knife edge will produce a substantial increase in the required torque causing the toggle bar to move against its spring mounting and so operate the micro switch, to initiate a warning signal. Once icing conditions cease, the knife-edge cutter will no longer shave ice, torque loading will reduce and allow the motor to return to its normal position and the micro switch will open-circuit the ice warningindicator.

Torque Testing of Serrated Rotor

The functional testing of the serrated rotor ice detector head is carried out with the use of a torque tester of the type specified by the relevant aircraft Maintenance Manual. Care should be taken so that sufficient torque only is applied to cause the warning system to operate but not enough to stall the motor as this may cause overheating with a subsequent electricalfailure.

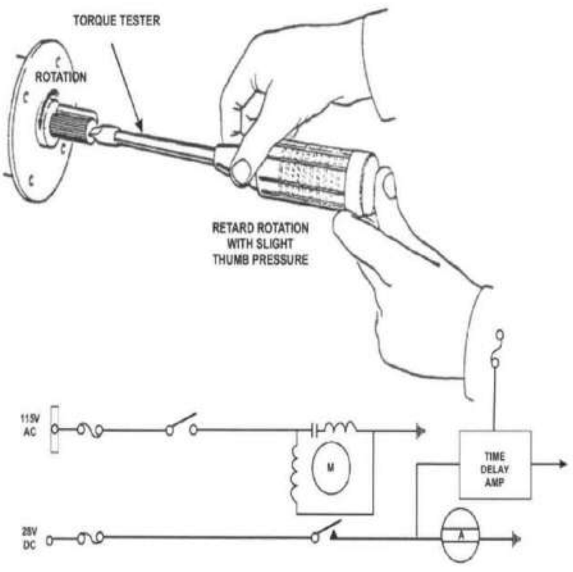


Fig.12.12 Torque testing of serrated rotor ice detector head Radioactive Ice Detector System

The detection system operates on the principle that a stream of
beta particles is attenuated according to the density or thickness of absorbing medium (air or ice). The triggering threshold corresponds to 0.4 mm (0.015 in.) ice thickness on the sensing surface of the probe.

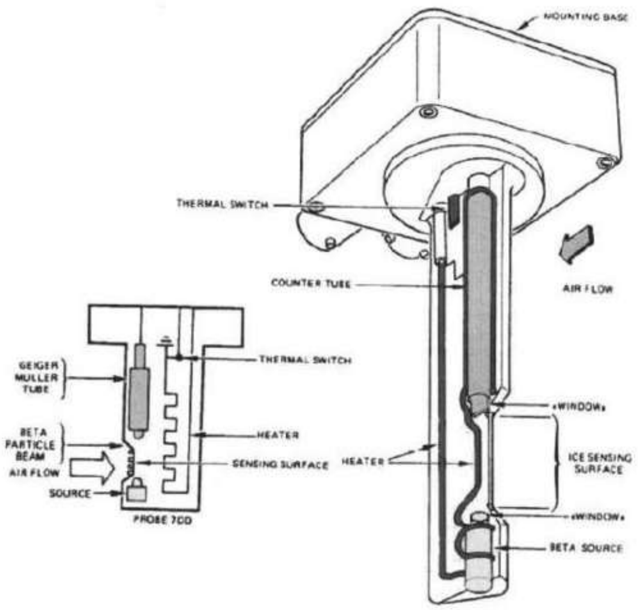


Fig.12.13 Radio Active Ice Detection Probe

A beta source sends radioactive radiation via the ice sensing surface to a Geiger Mueller (GM) tube. If ice is accumulated on the sensing surface of the ice detection probe, the GM tube sends a detection signal to the ice detection controller which provides an "ICE" signal to a pulse generator to turn on an ICE warning light in the cockpit. Simultaneously the controller sends a de-icing current to a heating element in the ice detection probe to melt the accumulated ice. If the is melted, the detection signal of the GM probe changes and the "ICE" signal in the controller is cancelled. Therefore the controller stops the de-icing current to the heater to allow new ice accumulation on the ice detector and removes the

"ICE" signal tothepulsegenerator.Iftheaeroplaneleavestheicingareathe pulse generator does not get a new "ICE" signal and switch off the ICE warning light in the cockpit until after 1minute.

In case of a probe overheat a thermal switch sends an overheat signal to the controller overheat latching circuit.
Then the controller inhibits heating current and activates the OVHT light in the ICE DET P/B to inform the crew that the ice detection system has failed. After cool-down of the probe, the overheat latching circuit can be reset by pressing the ICE DET P/B. Then the OVHT light extinguishes and the lockout of the heater circuit is cancelled.

An ICE DET TEST P/B allows a functional test of the controller and ICE warninglight.

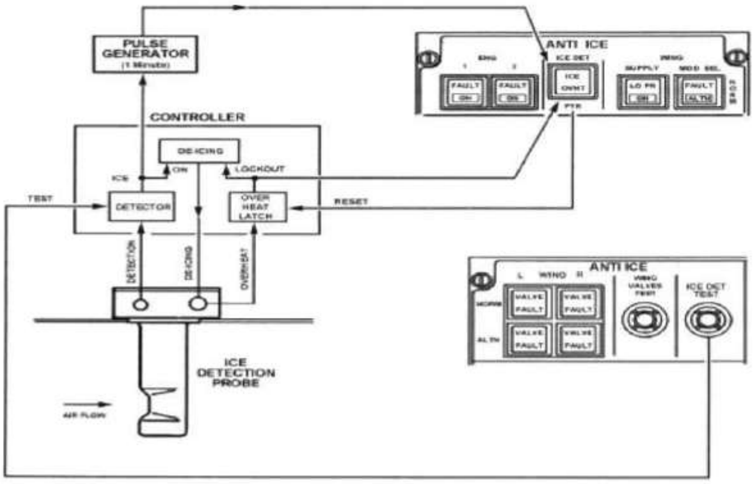


Fig.12.14 Radioactive Ice Detection System - Control and Monitor Logic

Anti-Icing/De-Icing Systems Aerofoil Ice Protection
There are two primary methods used for aerofoil iceprotection.

- Thermal (by ducting hot air along the inner surfaces of the aerofoilleadingedges)
- Electrical (by fitting heater elements to the aerofoilleading edges).

Thermal (Hot-Air) Systems

In systems of this type the leading edges of wings and tail units are usually provided with a second inner skin positioned to form a smallgap.
Heated air is ducted to the wings and tail units and passes into
the gap, providing sufficient heat in the outer skin of the leading edge to melt ice already formed and prevent further ice formation. The air is exhausted to atmosphere through outlets in the skin surfaces and also in some cases, in the tips of wings and tailunits.
AirSupplies
There are several methods by which the heated air can be supplied and these include bleeding of air from a turbine engine compressor, heating of ram air by passing it through a heat exchanger located in an engine exhaust gas system and combustion heating of ramair.

In a compressor bleed system the hot air is tapped directly from a compressor stage and after mixing with a supply of cool air in a mixing chamber it passes into the main ducting. In some systems, equipment, e.g. safety shut-off valves, is provided to ensure that an air mass flow sufficient for all de-icing requirements is supplied within pressure limits acceptable to duct and structurallimitations.

The heat exchanger method of supplying warm air is employed in some types of aircraft powered by turbo-propeller engines. The heat exchanger unit is positioned so that exhaust gases can be diverted to pass between tubes through which outside air enters the main supply ducts. The supply of exhaust gases is usually regulated by a device such as a thermostatically controlled flap fitted in the ducting between the exhaust unit and the heatexchanger.

In a combustion heating system ram air is passed through a cylindrical jacket enclosing a sealed chamber in which a fuel/air mixture is burned, and is heated by contact with the chamber walls. Air for combustion is derived from a separate air intake and is supplied to the chamber bymeans of a blower.

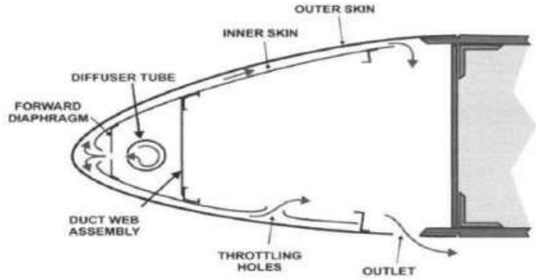


Fig.12.15 Typical heated Leading Edge

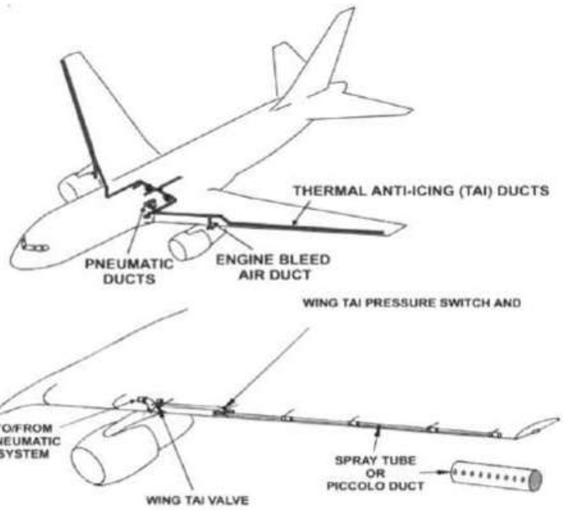


Fig. 12.16 Heated air sources

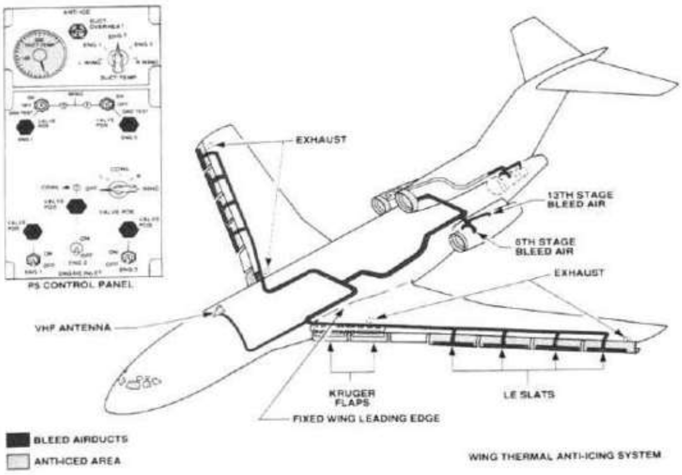


Fig. 12.17 Wing thermal anti-icing system

Ducting

The type of ducting, materials used, methods of inter connection and disposition in an aircraft vary between de-icing systems, and reference should therefore always be made to the relevant aircraft Maintenance Manual fordetails.

Light alloy and stainless steel are materials normally used in construction, stainless steel being adopted principally in compressor bleed systems. Flanged and bolted end fittings, or band-type vee-clamps with interposed sealing rings are common methods of connecting duct sections together, and in some cases an additional means of sliding duct sections one end into the other and securing by adjustable clamps may be adopted.

In some installations in which ducting passes through the fuselage, joints between duct sections are sealed to prevent loss of cabin air pressure. Fuselage ducting may, in some types of aircraft, comprise an inner stainless steelduct surroundedby an outer fibreglass duct. The two ducts are approximately 13 mm (1/2 in) apart and the interspace is filled with glass wool to provide thermal insulation. The purpose of this ducting arrangement is to serve as a leak warning system by venting interspace air through venturis which operate pressure switches and a warninglight.

Expansion and contraction of ducting is catered for by bellows or gimbal type expansion joints and in aircraft having variable incidence tailplanesand other moveable aerofoil surfaces such as leading edge slats and Kruger flaps, swivel joints and telescopic joints are fitted in the ducts supplying air to these surfaces.

In some installations, ducting in certain areas is lagged with a fire-resisting, heat-insulating material, normally fibreglass held in place by glass-cloth bound with glasscord.

Temperature Control

The control of the air temperature within ducting and leading edgesections is an important aspect of thermal de-icing system operation and the methods adopted depend on the type ofsystem.

In a typical compressor bleed system, control is effectedby temperature sensing units which are located at various points in the leading edge ducting and by valves in the main air supply ducting. The sensing units and valves are electrically interconnected so that the valves are automatically positioned to regulate the flow of heated air to the system thus maintaining the temperature within a predetermined range. Indications of air temperature conditions are provided by resistance type temperature sensing elements and indicators, temperature sensitive switches and overheat warning lights. On some aircraft the electrical supplies to the valves are interrupted by landing gear controlledrelays when the aircraft is on theground. Under these conditions, valve operation is accomplished by holding the system control switch to a TEST position.

When heat exchangers are employed, temperature control is usually obtained by the use of adjustable flaps and valves to decrease or increase the supply of heating and cooling air passed across the exchangers. The method of controlling the flaps and valves varies with different aircraft, but a typical system incorporates an electric actuator, which is operated automatically by an inching device controlled by atemperature sensing element fitted in the duct on the warm air outlet side of the heat exchanger. In some systems, actuators are directly controlled by thermal switches, so that the flaps or valves are automatically closed when a predetermined temperature is reached. Indications of air temperature conditions are provided by resistance type temperature sensing elements and indicators, temperature sensitive switches and overheat warninglights.

In systems incorporating combustion heaters, the temperature is usually controlled by thermal cyclic switches located in the heater outlet ducts, so that when the temperature reaches a predetermined maximum the fuel supply to the heaters is automatically switchedoff.

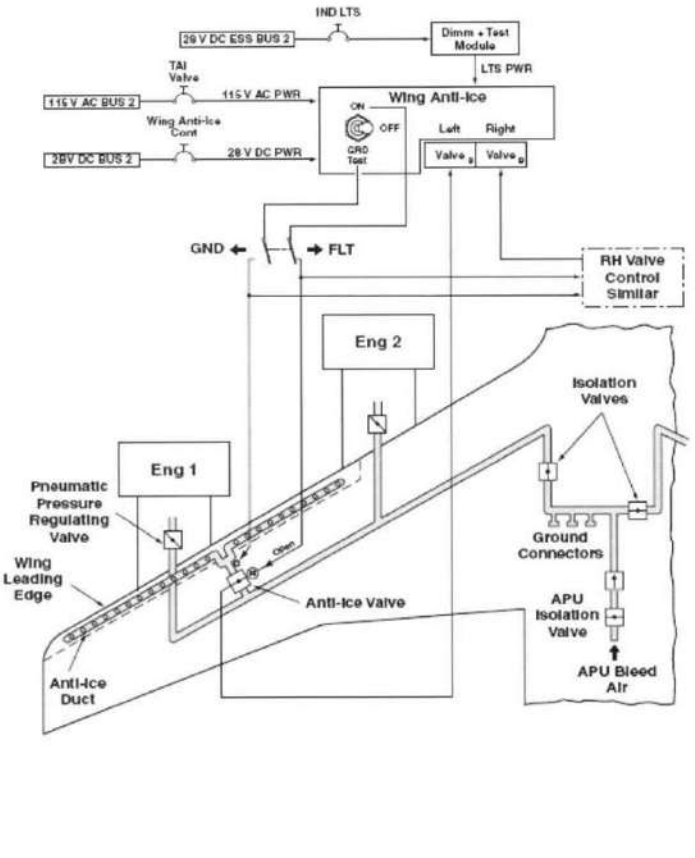


Fig. 12.18 Typical Wing Anti-Ice System Electrical Systems

There are various methods of utilizing electricity for the prevention or removal of ice on components, e.g. by use of spray mats and heater mats. Each mat is designed for a specific application; the heat output being obtained from whatever electrical source isavailable.

Mats are available for both anti-icing and de-icing. Anti-icing mats are supplied continuously with electricity while the de- icing mat is intermittently heated.

Wing anti-ice systems should not be used on the ground, to prevent wing leading edge overheat, except for short functional tests.

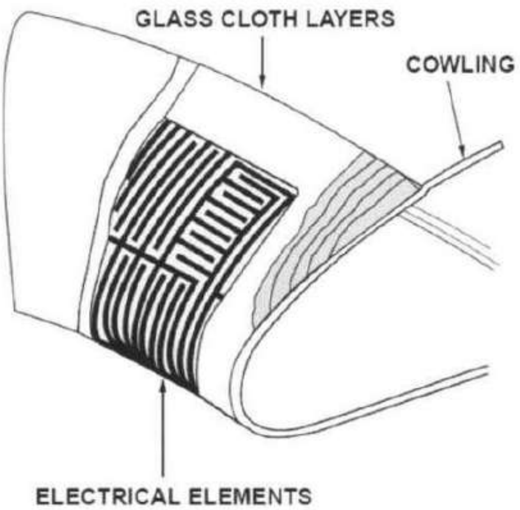


Fig. 12.19 Typical electrical heating matconstruction

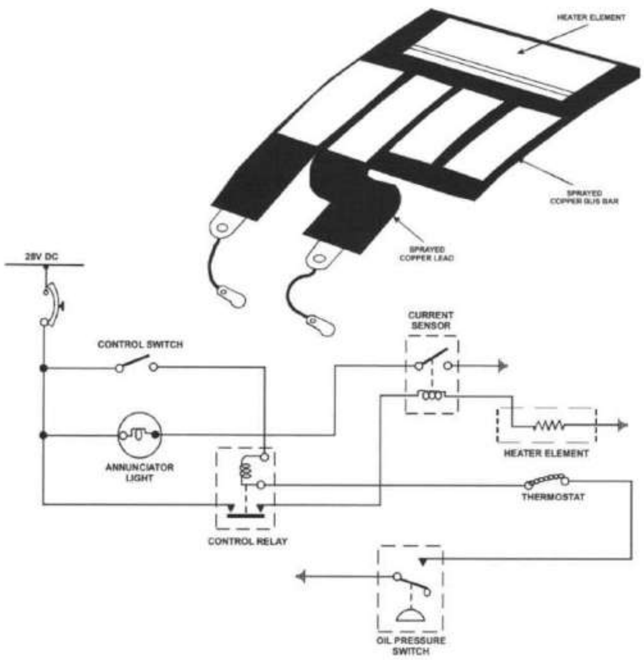


Fig. 12.20 Typical electrical heating mat circuit Fluid De-Icing Systems

In systems of this type, a de-icing fluid is drawn from a storage tank by an electrically driven pump and fed through micro filters to a number of porous metal distributor panels. The panels are formed to the profiles of the wing and tail unit leading edges into which they are fitted. At each panel the fluid passes into a cavity, and then through a porous plastic sheet to a porousstainlesssteelouterskin.Asthefluidescapesitbreaks the bond between ice and the outer skin and the fluid and ice together are directed rearward, by the airflow, over the aerofoil.

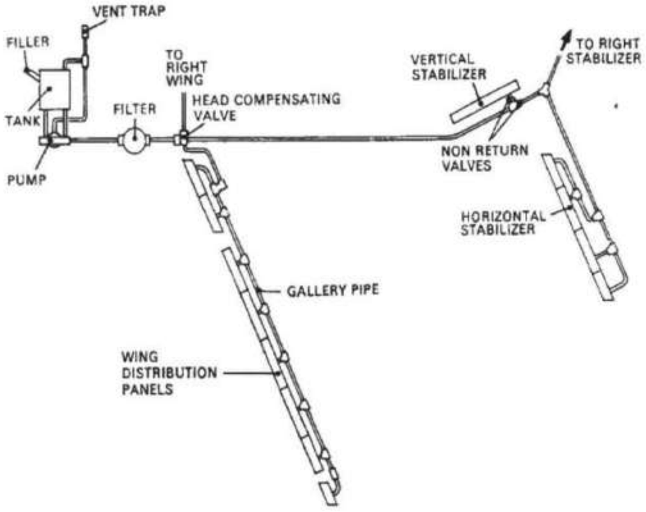


Fig. 12.21Typical fluid de-icing system

The interconnection of components of a typical fluid de-icing system is shown in Figure I. The head compensating valve is fitted in some types of aircraft to correct for variations in system pressure (head effect) due to differences in level between the wings, horizontal and vertical stabilisers. The non- return valves prevent back flow when the system is inoperative. Nylon pipelines are usually used throughout the system; those for the main fluid supply being of 8 mm (5/16 in) inch outside diameter and those for connections to individual distributor panels of 4-7 mm (1/4 in) outsidediameter.

A sectional view of a typical distributor panel is shown in Figure 12.20. The connector contains a metering tube which is accurately calibrated to provide the required rate of fluid flow through the distributor. In some aircraft the metering of fluid to the distributor panels is done via proportioning units containing the corresponding number of meteringtubes.

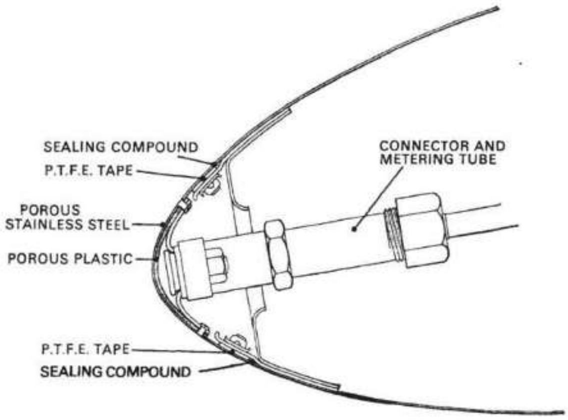


Fig. 12.22 Section of a typical distributor panel

To prevent electrolytic corrosion, plastic sealing strips are interposed between the stainless steel panel and the metal used in the aerofoil structure. In some installations an epoxy resin sealing compound is used, and to facilitate the removal of a panel it is sprayed along its edges with a thin coating of polytetrafluourethylene (PTFE) to act as a release agent. In addition, a strip of PTFE tape may be laid along the mating surfaces of the aerofoilstructure.

Pneumatic De-Icing System

Pneumatic de-icing systems are employed in certain types of piston-engined and twin turbopropeller aircraft. The number of components comprising a system varies with the operating principle. The typical de-icing system schematically illustrated in Figure 12.21 comprises the following:-

- The air supplysystem.
- The air distributionsystem.
- The pneumatic de-icerboots.
- The controls andindicators.

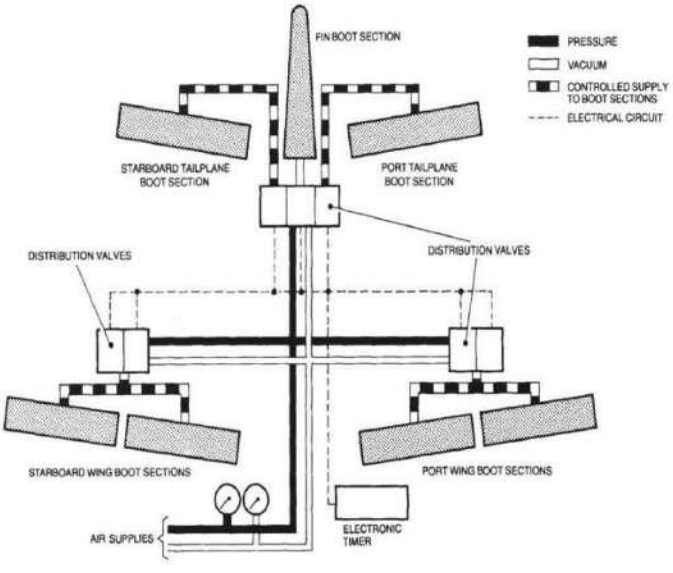


Fig. 12.23 Schematic diagram of a pneumatic de-icing system

When the system is switched on, pressure is admitted to the pneumatic de-icer boot sections to inflate the tubes. This weakens the bond between the ice and de-icerboot surfaces, causing the ice to break away. At the end of the inflation stage of the operating sequence, the air in the inflation tubes is dumped to atmosphere through automatic opening valves, the tubes are then fully deflated by the vacuum supply. This inflation and deflation cycle is repeated whenever the system is in operation. When the system is switched off, vacuum is continually supplied to all inflation tubes within the de-icer boot sections to hold them flat against the wing and tail unit leading edges, thereby minimising aerodynamicdrag.

The method of sequencing usually varies with the method of air. In most installations, sequencing control is effected by means of an electronic device, however reference should always be made to the relevant Aircraft Maintenance Manual for details of the method of control and operating timecycles.

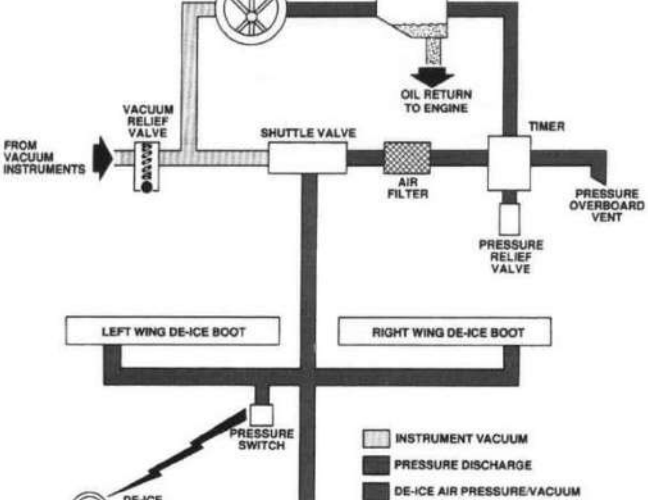


Fig. 12.24 Schematic diagram of a pneumatic de-icing system Air Supplies and Distribution

The tubes in a pneumatic de-icer boot section are normally inflated by one of the following methods:-

- Air pressure from the pressure side of an engine-driven vacuumpump.
- Air pressure from a high pressure air reservoir,or

- as fitted to some types of turbo-propeller aircraft, air pressure from a tapping at an engine compressorstage.

Whenever the system is switched 'OFF' or at the end of the inflation stage of the operating sequence, the de-icerboots are deflated by vacuum derived from either a vacuum pump, or, in systems utilising an engine compressor tapping, from the venturi section of an ejector nozzle. The method of distributing air supplies to the de-icerboots, depends on the type of de- icing system adopted for a particular type of aeroplane, but in general, there are three methods in use. One method employs shuttle valves which are controlled by a separate solenoid valve. The second method distributes air to each de-icerboot by individually solenoid-controlled valves; in the third method, distribution is effectedby a motor-drivenvalve

Pneumatic De-icer Boots

Pneumatic de-icer boots, or overshoes, consist of layers of natural rubber and rubberisedfabric between which are disposed flat inflatable tubes that are closed at the ends. The inflatable tubes are made of rubberised fabric which are vulcanisedinside the natural rubberlayers.

When the de-icerboots are in position on a wing, or tailplaneleading edge, the tubes either run parallel to the span, or as in other arrangements, parallel to the chord. The inflatable tubes are connected to the air supply pipelines from the distribution valves by short lengths of flexible hose. These in turn are secured to connectors on the de-icerboots and air supply pipelines, by hose clips. The external surfaces of the de-icerboots are coated with a film of conductive material to discharge any accumulations of staticelectricity.

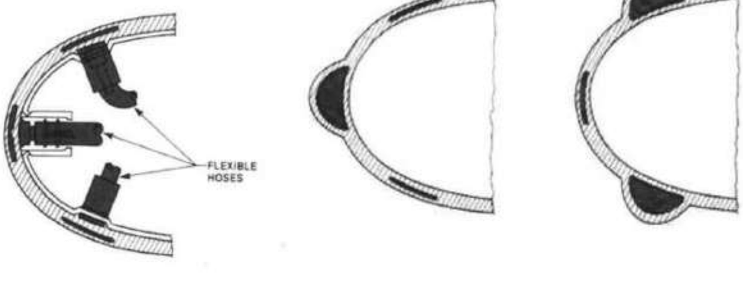


Fig. 12.25 De-icer boot operation

A de-icer boot may be attached to the leading edge of an aerofoil either by, screw fasteners (rivnuts), or adhesive cement. To cover the edges of a screw-fastened de-icer boot, metal fairing strips are used on the upper and lower surfaces. To secure and prevent inward creep of the de-icerboot, end strips are also fitted. The strips are secured by the same screws and rivnuts as those used to secure the edges of the de-icer boots to the leadingedge.

Controls and Indicator

The controls and indicators required for the operation of a de- icing system depend upon the type of aircraft, and the particular arrangement of its de-icing system. In the basic arrangement, a main ON-OFF switch, pressure and vacuum gauges, or indicating lights, form part of the controllingsection.
Installation

For full details of the checks to be carried out on pneumatic de-icing system components prior to installation, and the installation methods to be adopted, reference should alwaysbe

made to the relevant Aircraft and Aircraft Component Maintenance Manuals. The information provided in the following paragraphs is therefore of a general nature, and intended only as a guide to installationpractices.

Pneumatic De-icer Boots

Screw-fastened De-icer Boots. The following points should be observed when installing de-icer boots of this type:-

- The mating surfaces of the leading edges should beclean

and free from projections or rough edges which may chafe the under-surfaces of the boot. Projections such as rivets may be covered with adhesivetape.

- LeadingedgesshouldbepaintedwithamixtureofFrench

Chalk and lead-free petrol or Methylated Spirit and allowed to dry thoroughly. This will leave a smooth even film of French Chalk on the leading edge skin to lubricate movement of the rubber de-icerboot.

- The under-surface of the de-icer boot should also be thoroughly dusted with FrenchChalk.

NOTE: The liquid mixture as applied to leading edges should not beused.

To prevent chafing of the aerofoil surfaces, a narrow strip of adhesive tape should also be placed on the underside of the trailing edges of the fairing strips, taking care to ensure that the tape does not extend beyond the edge of the fairing strips.
The de-icer boot should then be positioned adjacent to leading edges, and the hoses from the air supply system fitted to the correct de-icer boot connections.

When fitting a screw-fastened de-icer boot, it is immaterial

whether the upper or lower edge is attached first. The edge chosen should be the one most convenient to reach and perform the stretching operation described in paragraphs (g) and (h). The de-icer boot and a fairing strip should be loosely secured, along one edge, to the aerofoil. The screws should not be tightened until the other edge has beensecured.

To assist the ice shedding action and deflation, de-icerboots are secured under tension. This is effected by stretching the second edge into position. A special tool for gripping the edge of the de-icer boot is normally used to facilitate stretching, together with a number of pegs to locate the de-icerboot mounting holes over therivnuts.

When the de-icer boot is stretched back over the rivnuts, a peg should be forced through the de-icerboot mounting hole and into the appropriate rivnut. At least two additional pegs should be inserted in adjacent rivnuts, before releasing the tension. This operation should be repeated until the entire edge of the de-icer boot is attached. As the tension and pegging operation progresses, there may be a tendency for wrinkling to develop. This may be worked out by slapping the de-icerboot surface with the gloved hand which will also help to keep lengthwise shrinkage of the de-icer boot to aminimum.

When all the pegs have been inserted, the fairing strip should be installed over the pegs, and working from one end, the pegs replaced with the appropriate attachment screws. These, together with those screws securing the other edge of the de- icing boot, should then be tightened and the end fairing strips fitted.

Cemented De-icer Boots

The following points should be observed when installing de- icer boots of thistype:

- All paint should be removed from the complete leading edge skin surface of an aerofoil, which together with the under-surfaces of the de-icer boot, should be cleaned by swabbing the skin surface with a clean, lint-free cloth soaked in Toluol and then wiped dry with a clean dry cloth before the solvent has had time to evaporate. Cloths should be changed frequently to avoid re-contamination of the cleaned areas, and to ensure a completely clean surface, cleaning operations should be carried out at leasttwice.

- Immediately after the final drying operation, an even coat of the cement specified for use should be applied to the leading edge skin surface and under-surface of the de-icer boot. The surfaces should then be allowed to dry before a second coat of cement is applied and left to dry until completely tack-free. Drying times depend on temperature and humidity conditions. At 15C or above with a relative humidity of up to 75%, a minimum of one hour drying time should be allowed. When the relative humidity is between 75% to 90%, a drying time of two hours between coats should beallowed.

- Before cementing a de-icerboot to the leading edge, it should be rolled up (undersurface outwards) and a check made to ensure that its air supply connecting tubes do not foul the edges of the holes provided in the leading edge. A check should also be made that when unrolled from this position, the de-icer boot will be correctly aligned with the entireleadingedge.Thissecondcheckisfacilitatedby marking a centre line down the length of the de-icer boot on the cemented surface, and a corresponding centre line along the front of the leading edge.

The de-icer boot should be folded back at the air connection end, over a distance of 229 mm (9 in) to 305 mm (12 in), and tacky surfaces reproduced in this area on the de-icerboot and leading edge, by swabbing with a clean lint-free cloth moistened with Toluol. Excessive swabbing should be avoided to prevent removal of cement from the surfaces. The two tacky surfaces should be positioned together and the de-icer boot rolled down firmly with a 51 mm (2 in) wide rubber covered handroller.

NOTE: Two sizes of hand roller are normally specified for use, a 51 mm (2 in) wide roller for main rolling and a 19 mm (3/4 in) wide roller for rolling between the inflatable tubes. Ail rolling must be carried out parallel to the inflatabletubes.

The de-icer boot should then be rolled back from the free end, and tacky surfaces reproduced over the de-icerboot and leading edge. The surfaces should then be joined and rolled together, working progressively in strips 305 mm (12 in) wide, until the de-icer boot is attached to the leading edge. If the de- icer boot becomes attached to the leading edge in an incorrect position, the de-icer boot should be pulled up smartly and repositioned immediately avoiding any twisting orbending.

NOTE: If a blister appears during rolling operations a piece of wire should be gently inserted between the open end of the de-icer boot and the leading edge until it enters the blister. The wire should then be removed and the air rolled out along the "tunnel" formed by the wire. In areas clear of the inflation tubes, a blister may be dispersed by inserting a hypodermic needle intoit.

In some installations, fairing strips are fitted along one edge of a de-icer boot surface to permit flush fitting of the strip. Care must be taken when cutting to ensure that it is not done too deeply; a cut should be made one layer at atime.

When the de-icer boots are finally cemented in position, the air supply connections should be made secure, and the ends of de-icer boots trimmed so that they lie flush with the end of the fairing strips and all excess cement cleaned away from the de- icer boot and metal surrounds. Additionally the de-icerboots should be sealed along their edges and ends by applying a special sealing cement. Reference should be made to the Aircraft Maintenance Manual for preparation details of the cement and the number of coats to be applied. When the final coat of sealing cement is dry, it should be 'capped' by the application of cement to the same specification as that to restore the conductivity of thesurface.

Finally, a test for the satisfactory adhesion of the de-icerboots to leading edge sections should be carried out in the manner specified in the relevant Aircraft Maintenance Manual, which also includes a check to ensure that the forces required to separate the specially prepared test pieces are within permissiblelimits.

Inspection and Maintenance

The majority of inspection and maintenance procedures associated with pneumatic de-icing systems are related to the de-icer boots, since their location on an aircraft makes them vulnerable to damage which may be caused under actual operating conditions or during ground handling and servicing operations. Therefore, the information on inspection and maintenance procedures detailed in the following paragraphs is primarily concerned with de-icer boots and is intended to serve only as a general guide. Reference should therefore always be made to the relevant Aircraft and Component Maintenance Manuals and Approved Maintenance Schedules appropriate to the specific aircraft type.

At each inspection careful consideration should be given to the overall condition of de-icer boots. This applies particularly to the soft pliable rubber which may easily be damaged during servicing operations. The following are general precautions which should beobserved:—

- Refuelling hose and other equipment must not be dragged over the surfaces of the de- icerboots.
- Ladders or service platforms which are placed near the de- icer boots during servicing operations must have sponge- rubber pads fitted to preventdamage.
- Oil or grease found on the surface of de-icerboots must be removed as soon as possible with soap and water or with a clean rag moistened with a lead-free petrol. Petrol should not be allowed to dry on the surfaces; it should be wiped off immediately with a cleancloth.

NOTE: Surfaces should not be rubbed hard during cleaning as damage to the conductive film may result.

- Personnel must not tread on de-icerboots during servicing operations.

- In tropical areas de-icerboots should not be exposed to sunlight for longperiods.

Conductive Surface Deterioration

The conductive surface of cemented de-icerboots deteriorates slowly in service through general abrasion and this will be evident by the abraded condition and rough appearance of the surface. The method of restoring the conductivity of the surface is asfollows:-

- Carefully clean the outer surface of the de-icerboot using soap and water or a clean lint- free cloth moistened with lead-free petrol. Liquids should be used sparinglyand surfaces dried offimmediately.

NOTE: The de-icerboot surfaces should not be rubbed hard during cleaning or damage to the conductive surfaces will result.

- Apply one coat of the conductive cement specified for the de-icerboots, ensuring that the identification and serial reference number details printed on a small part of the de- icer boot surface, are not obliterated. The cement should be applied evenly and sparingly over the surface, care being taken to prevent excess quantities running down and forming rippled ridges across it. While the cement is drying, the aircraft should remain in a warm, dry area. In dusty conditions, the de-icer boots should be covered loosely with paper which is sealed with tape at theedges.

NOTE: Conductive cement normally requires about 24 hours to dry. Therefore, whenever possible, de-icerboots should be resurfaced soon after the start of any periodic maintenance check on theaircraft.

De-icer Boot Deterioration

Continued operational use and exposure to intensesunlight

causes the surface of de-icerboots to deteriorate by general crazing. If the surface is extremely crazed, with the cracks extending to the fabric reinforcing, the de-icerboot should be replaced. If the damage due to crazing does not extend into the natural rubber plies of the de-icerboot, then a coating of conductive cement will normally suffice in restoring the de- icer boot to a serviceablecondition.

Repairs

Damage to de-icerboots found during inspections will vary from minor cuts, holes and scratches; which may be easily repaired, to extensive splitting of the tube areas which are beyond repair. Minor tears and holes can usually be detected by visual means or the sound of escaping air, or by system pressure not building up to the required operational value as indicated by the system pressure gauge. If there is difficulty in locating a leak, the de-icerboot should be inflated, and a soap solution applied to the suspectedarea.

Repair schemes are devised, and detailed in Maintenance Manuals and in some cases, Structural Repair Manuals, for the relevant aircraft type. The repair methods to be adopted and the nature of the work involved, depends largely on the extent of the damage to the de-icer boot. Kits are provided for carrying out cold patch repairs in-situ which are normally confined to two size ranges of damage; cuts, holes or cracks up to 19 mm (3/4 in) long, and from 19 mm (3/4 in) to 51 mm (2 in) long. The kits normally contain moulded rubber patches of various sizes, cements, rubber and rubberisedfabric sheets, and the appropriate tools. Where larger cuts, holes, or cracks have occurred, or where a portion of a de-icerboot has been torn away, a temporary emergency repair may be made with the aid of a cold patch repairkit.
Numerous cold patches greatly affect the efficiency of a de- icerboot, therefore these repairs must only be regarded asa

temporary measure. When a de-icer boot has been extensively patched it must be removed from a leading edge in order that permanent repair by a vulcanising process may be carried out. Some important aspects of de-icer boot repair methods are summarised as follows:-

- Patches should be approximately 16 mm (5/8 in) larger all round than the cut. In the case of cuts which range in size from 19 mm (3/4 in) to 51 mm (2 in), and for the emergency repair of more extensively damaged areas, the rubber or rubberisedfabric sheet material should be cut 38 mm (1 1/2 in) larger allround.
- Care should be taken to remove only the conductive coating from an area of the de-icerboot surface slightly larger than the patch required. To assist in this operation, a special buffing shield

(supplied with the repair kit) with various sized apertures is held over the damage area in the selectedposition.

• After the removal of the conductive coating, the cleaned surface should be roughened with a wire brush and then smoothed as evenly as possible with an emery buffing stick. The surface should be finally cleaned with lead-free petrol and allowed todry.

• The patch and damage area should be coated with cement, and after the specified air- drying time has elapsed, the patch should be applied to the de-icerboot. To ensure that the patch adheres centrally over the damage, onecorner
should be lightly applied and the remainder pressed down at the same time exerting a slight pulling action. The tension thus created will help to close up the puncture in the de-icerboot surface when the patch is finally in position.

• Patches should be pressed down carefully to prevent entrapmentofairandrolledthoroughlywiththesteelroller

supplied with the repair kit, ensuring that the edges of the patch have adhered closely to the de-icer boot surface.

• After allowing the appropriate time for setting, the patch and surrounding area should be wiped with a clean lint-free cloth moistened with lead-free petrol to remove all surplus cement. When the surfaces are thoroughly dry and clean, a coat of conductive cement should be applied to restore the conductivesurface.

• If the damage to a de-icerboot is such that an inflatable tube is cut completely through, the inside surface should be patched first and then the outside. The patches must be cut from rubberised fabric sheet which is manufactured such that it stretches in one direction only. When in position, the patches should stretch normal to the inflation tubes, i.e. for de-icer boots using spanwise inflation tubes, the fabric stretch should be chordwise and viceversa.

NOTE: A pressure test must be carried out on the de-icerboot in accordance with the procedures specified in the Aircraft Maintenance Manual after repair to the inflatabletubes.

Functioning Tests

Functioning tests must be carried out at the check periods specified in the Approved Maintenance Schedules, or when a system malfunction occurs, or a major component (e.g. a distributor valve, regulator valve or de-icerboot) has been replaced, and also after repairs to a de-icerboot. The method of testing a system depends mainly on the type of aircraft and precise details must therefore be obtained from the relevant Maintenance Manual. The checks which in general form part of functioning test methods, are outlined asfollows:-

• Testsmaybecarriedoutusingeithertheaircraftenginesor

by deriving the requisite air supplies from a ground test trolley. If a system which uses engine compressor bleed air is to be tested by using a test trolley, the air supply must be clean, moisture-free and at a delivery pressure approximating that normally obtained at the engine compressortapping.

• Pressure and vacuum indicators should be checked to ensure that supplies are maintained at the specified values. Adjustments should be made, where necessary, to relevant regulating and reliefvalves.

• With a system selected 'ON', de-icerboots should be checkedtoensurethattheyinflateanddeflateinthecorrect
sequence and for the correct periods of time determined by the appropriate timing device. During this check it should also be noted that the pressure gauge pointer fluctuates simultaneously with de-icer boot inflation and deflation.

• If the de-icerboot section inflates and deflates sluggishly; eventhoughthecorrectpressureisindicated,itmaybethe
result of an obstruction in certain of the pipelines (e.g. those leading to the de-icerboot sections or to

the distributor valves). If the distributor valves are of the electrically controlled type, sluggish inflation and deflation may also be due to stickingsolenoids.

- Jointsandsectionsofpipelinesshouldbecheckedforleaks under operating conditions.
- When the system is selected to the 'OFF' position, the de- icer boot sections should deflate completely and lie flat against the leading edge of the appropriateaerofoil.

Storage of De-icer Boots

Before storing, the surface conditions of the de-icerboot should be inspected carefully, to establish that there are no operating defects. They should be cleaned and repaired, and where necessary lightly dusted with French Chalk. Connectors should be blanked off and the de-icerboots rolled up commencing at approximately 153 mm (6 in) diameter. Rolling should be commenced at the end remote from valve connectors so that they are on the outside of the finished roll. Where connectors are located near the centre of the de-icerboot, a pad of corrugated paper should be placed over the connectors to protect the contacting surface. The rolled de-icerboot should then be thoroughly wrapped up in a heavy paper to exclude all light, and stored in a cool, dry, dark place, where it will not be crushed or wrinkled. In cases where de-icerboots are pre- cemented to detachable leading edge sections, the latter should be wrapped up and stored in such a manner that they are supported on their trailingedges.

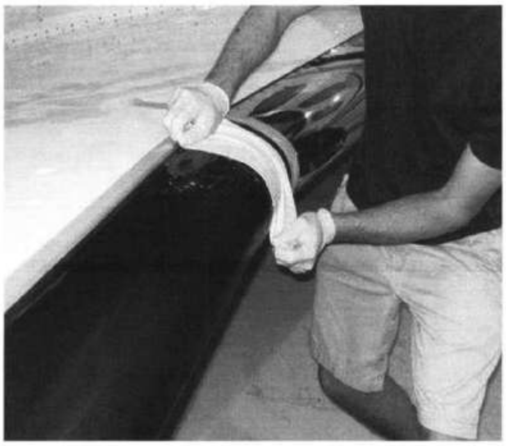


Fig. 12.26 De-icer boot cleaning and polishing Engine Anti-Ice Systems General

Engine anti-ice systems should prevent ice accumulation on the engine airintakes.

Hot air from an engine compressor high pressure stage is used to heat the engine nose cowl. The engine anti-ice system should always be used if icing conditions exist to prevent possible damage to a running engine. Engine anti-ice systems are manually controlled by the flightcrew.
Turbine engines are susceptible to damage from chunks of ice that get into the compressor, so anti-icing systems are used to prevent the formation of ice ahead of the compressor inlet.

Many aircraft have air passages in the compressor inlet case, inlet guide vanes, nose dome, and nose cowling. Hot compressor bleed air flows through these passages to prevent the formation of ice. Ice can form when the engine is operated at high speed on the ground when the temperature is as high as 100 if the air is moist. The high velocityof the inlet air creates a pressure drop that lowers the temperature of the air enough for ice to form. In flight the anti-icing system is turned on before entering areas of visible moisture (rain or clouds) when the inlet temperature is between about 100 and -150. Below - 150, there is so little moisture in the air that ice is not likely to form. Sometimes turbine-powered aircraft sit on the ground and water collects in the compressor and freezes. If this should happen, direct a flow of warm air through the engine until all of the ice is melted and the rotating parts turnfreely.

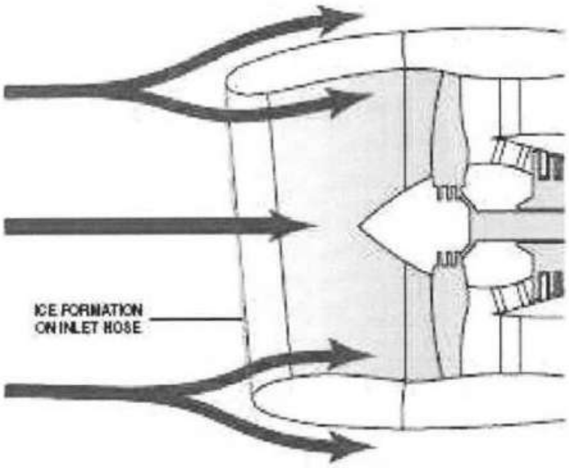


Fig. 12.27 Ice formation on engine inlet

When an anti-ice system is required to provide protection against ice formation in the inlet, engine bleed air is extracted from the compressor of the engine being anti-iced and routed to the inlet area through external piping and control valves. The air is extracted from the point in the engine which will provide the correct pressure and temperature to satisfy the needs of the engine during both ground and flightoperations.

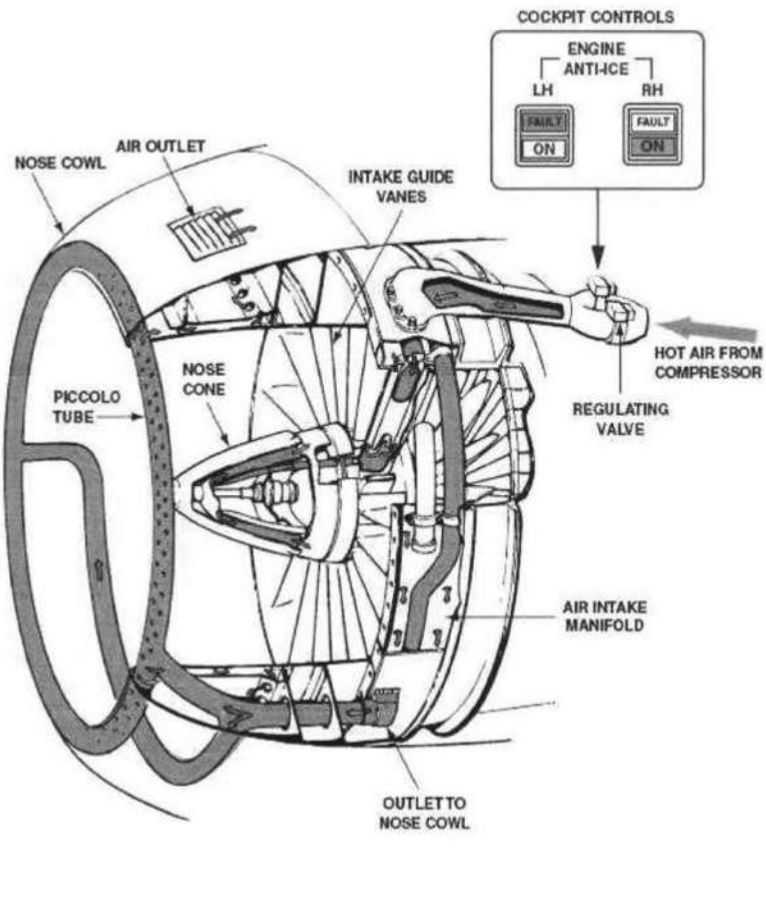


Fig. 12.28 Engine anti-ice system

Components

The main components of an engine anti-ice system are:

- The engine anti-icevalve
- The engine anti-ice push button (P/B) orswitch
- The swirl tube or a piccoloduct

Engine Anti-Ice Valves

Depending on engine type the valves are shutoff or pressure regulating valves. Pressure regulating valves are normally pneumatic actuated and controlled to close by an electrical solenoid.

Shutoff valves are normally actuated by an electrical AC or DC motor. Both types contain limit switches or also pressure switches for the function monitoring. The valves are also equipped with visual position indicator and manual locking device.

Engine Anti-Ice P/B or Switch

Both types have two fix positions ("ON" and "OFF'). The Engine Anti-Ice P/B contains an "ON" and a "DISAG' or 'FAULT" light to indicate the switch position and the status of the valve.

On older aircraft with Engine Anti-Ice switches the respective
indication lights are installed on the pilot's overhead panel. Swirl Tube or Piccolo Duct
To distribute the hot air a swirl tube or a piccolo duct is installed in the nosecowl.

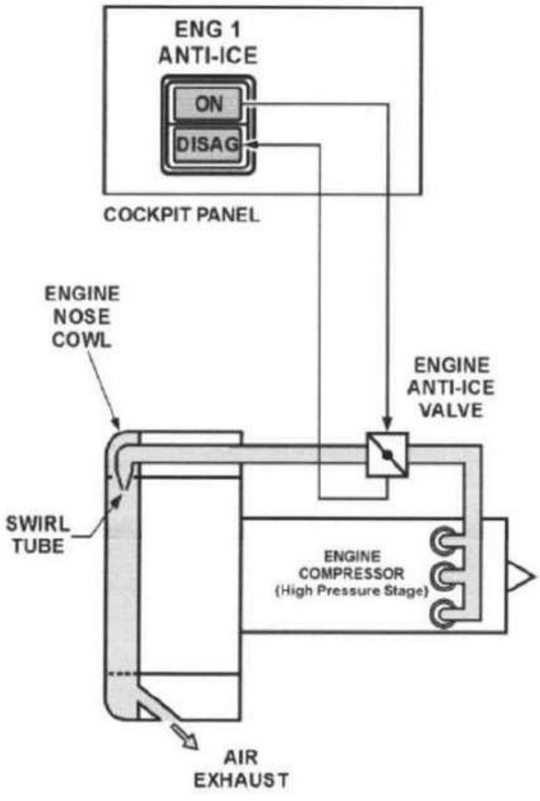


Fig. 12.29 Engine Anti-Ice System

Windshield De-icing and Anti-icing Systems Fluid De- Icing System

The method employed in this system is to spray the windscreen panel with a methyl-alcohol based fluid. The principal components of the system are a fluid storage tank, a pump which may be a hand-operated or electrically- operated type, supply pipe lines and spray tube unit. Figure 12.28 illustrates the interconnection of components based on a typical aircraft system in which fluid is supplied to the spray tubes by two electrically- operated pumps. The system may be operated using either of the pumps or both, according to the severity oficing.

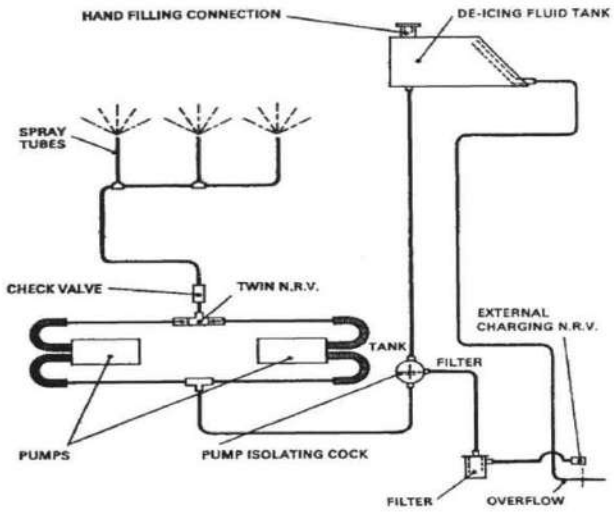


Fig. 12.30 Typical fluid de-icing system Electrical Anti-icing System

This system employs a windscreen of special laminated construction heated electrically to prevent, not only the formation of ice and mist, but also to improve the impact resistance of the windscreen at lowtemperatures.

The film-type resistance element is heated by alternating current supplied from the aircraft's electrical system. The power required for heating varies according to the size of the panel and the heat required to suit the operating conditions. Details of these requirements are given in the relevant aircraft Maintenance Manual.

The circuit embodies a controlling device, the function of which is to maintain a constant temperature at the windscreen and also to prevent overheating of the vinyl interlayer which would cause such permanent damage as vinyl 'bubbling' and discolouration. In a typical antiicingsystem shown schematically in Figure 12.29, the controlling device is connected to two temperature sensing elements laminated into the windscreen. The elements are usually in the form of a fine wire grid, the electrical resistance of which varies directly with the windscreen temperature. One sensing element is used for controlling the temperature at a normal setting and the other is used for overheat protection. A system of warning lights and, in some cases, magnetic indicators, also forms part of the control circuit and provides visual indications of circuit operating conditions, e.g. 'normal', 'off or'overheat'.

When the power is applied via the system control switch and power relay, the resistance element heats the glass. When it attains a temperature pre-determined for normal operation the change in resistance of the control element causes the control deviceorcircuittoisolate,orinsomecasestoreduce,thepower supply to the heater element. When the glass has cooled through a certain range of temperature, power is again applied and the cycle is repeated. In the event of a failure of the controller, the glass temperature will rise until the setting of the overheat sensing element is attained. At this setting an overheat control circuit cuts off the heating power supply and illuminates a warning light. The power is restored and the warning light extinguished when the glass has cooled through a specific temperature range. In some systems a lock-out circuit may be incorporated, in which case the warning light will remain illuminated and power will only be re-applied by cycling the system control switch to 'OFF' and back to'ON'.

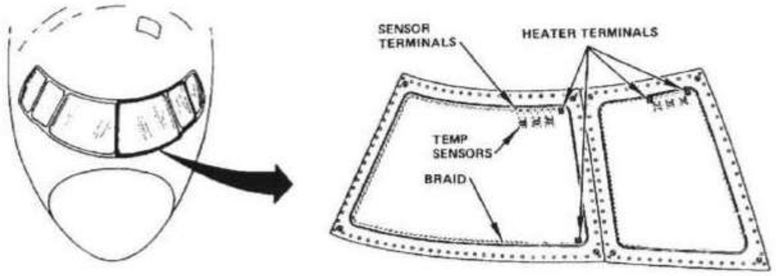


Fig. 12.31 Electrical windshield anti-icing

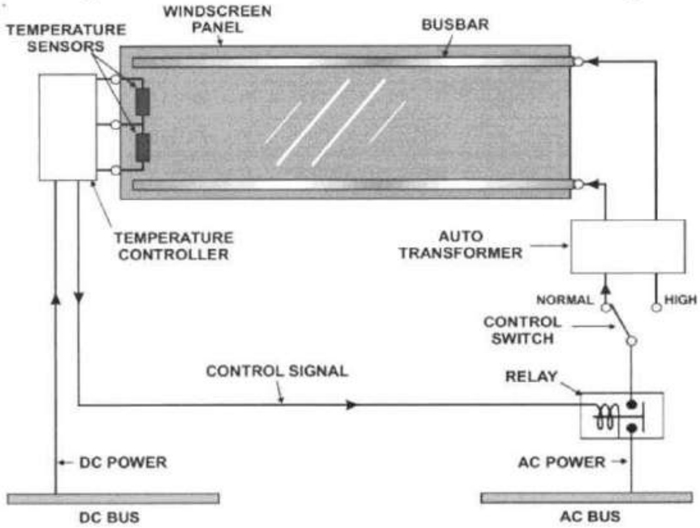


Fig. 12.32 Typical electrical anti-icing system

In addition to the normal temperature control circuit it is usual to incorporate a circuit which supplies more heating power under severe icing conditions when heat losses are high. When the high power setting is selected, the supply is switched to higher voltage output tappingsof an auto transformer which also forms part of an anti-icing system circuit thus maintaining the normal operating temperature. The temperature is controlled in a manner similar to that of the normal control temperature circuit.

For ground testing purposes, the heating power supply circuit may also be controlled by landing gear shock-strut microswitches in such a way that the voltage applied to the resistance elements is lower than that normally available in flight.

Testing of Systems

The method of testing varies between individual systems; reference should therefore be made to relevant aircraft Maintenance Manuals for details of procedure to be adopted, specific test equipment required and also for any precautions to be observed. There are, however, aspects of testing and certain precautions which are of a standard nature, and these are summarised for guidance as follows:-

- In general, test procedures are principally concerned with the checking of the electrical resistance of heating films and temperature sensing elements, checking of the voltages applied at selected system operating conditions, e.g. 'normal', 'low' and 'high' settings of a system control switch and also checking of insulation resistance betweencircuits.
- Electrical power should always be applied initially atlow intensity and the windscreen allowed to warm up gradually thus minimising the effects of thermal shock stresses.
- When carrying out resistance and voltage checks of some anti-icingsystemsitisnecessarytoisolatetheoverheat sensing element circuit. In such cases, the period of time during which power is applied to the heating

elements must be kept to a minimum to avoid overheating of the windscreens.

- In systems incorporating electrically heated direct vision and other side windows, the circuits to thesewindows must
be checked at the same time as the windscreen anti-icing systemchecks.

During ground testing attention should be paid to the effect of ambient temperature and strong sunlight on the behaviour of temperature control systems. Ambient temperatures approachingthoseofthenormaloperatingtemperatureof windscreens will result in a very brief application of power, followed by no power for a considerable period. In some instances power will not be applied at all. It is possible, therefore, to be misled into believing that a serviceable system is malfunctioning. Where it is necessary to carry out system tests and checks in such conditions, it is recommended that the aircraft be positioned in the shade or in a hangar if practicable, and also that cool wet cloths be applied to the windscreens thereby lowering their temperature prior to switching on the power. Bearing in mind thermal shock stresses, the use of ice should beavoided.

Windshield Wiper System

Rain removal systems are used in larger aircraft to keep the windshield free of water so the pilot can see for the approach and to manoeuvre the aircraft safely on theground.

Small general aviation aircraft have acrylic windshields that are easy to scratch, so windshield wipers are not used. Rain is prevented from obstructing visibility on these aircraft by keeping the windshield waxed with a good grade of paste wax. Water does not spread out on the waxed surface, but balls up and is blown away by the propellerblast.

Mechanical systems use windshield wipers similar to those used on automobiles except that they are able to withstand the high air loads caused by the speed of theaircraft.

The wipers for the pilot and the co-pilot are driven independently, so if one drive malfunctions, there will still be clear visibility on the other side. The wipers may be driven by electric motors or hydraulic or pneumatic actuators, and all systems have speed controls and a position on the control switch that drives the blades to a stowed, or parked,position.

Windshield wipers should never be operated on a dry windshield because they will scratch the expensive glass. When you must operate them for maintenance purposes, flush the windshield with water and operate the wipers while the glass iswet.

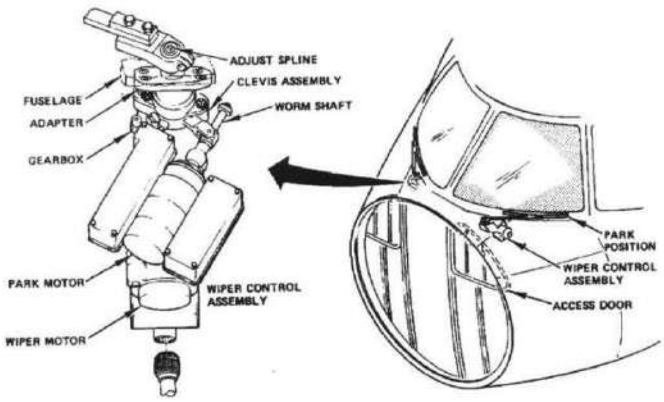


Fig. 12.33 Windshield wiper mechanism

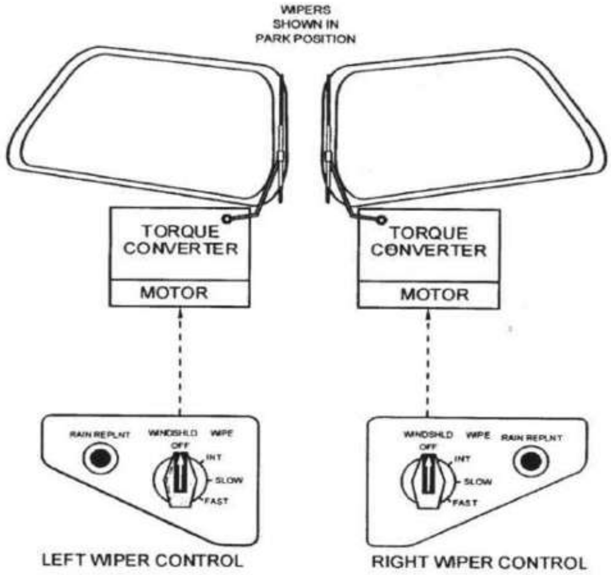


Fig. 12.35 Typical windshield wiper system

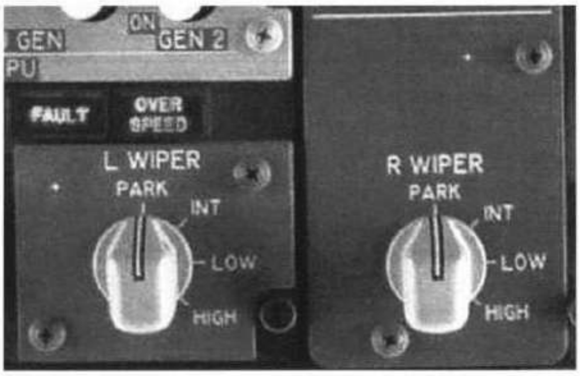


Fig. 12.36 Typical windshield wiper control switches (Boeing 737)

Rain Repellent System

Chemical rain repellent is a syrupy liquid carried in pressurized cans in the rain repellent system. When flying in heavy rain with the windshield wipers operating, the pilot depresses the rain repellent buttons. This opens solenoid valves for a specific length of time and allows the correct amount of liquid to spray out along the lower portion of thewindshield.

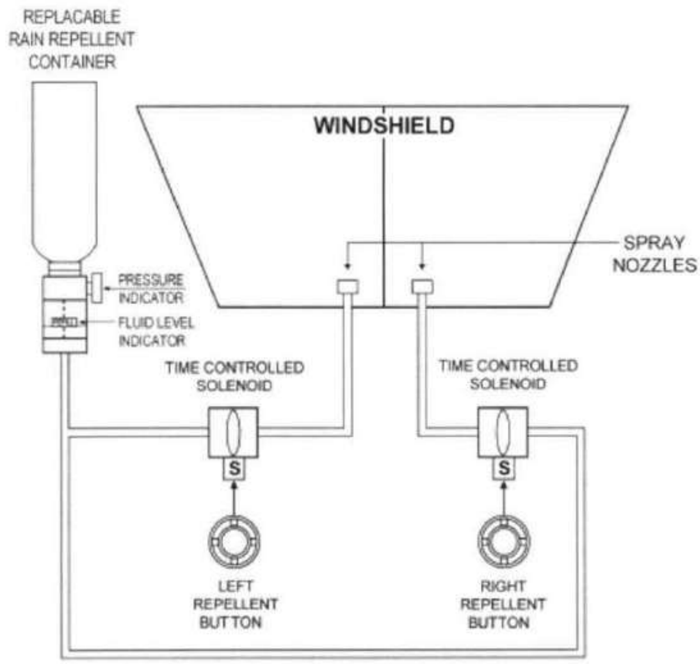


Fig. 12.37 Rain repellent system

The windshield wipers then spread the repellent evenly over the glass, and when rain strikes the treated surface it balls up rather than spreading out. The water is carried away by the high velocity of the air flowing over the windshield. Chemical rain repellent should not be discharged onto a dry windshield because it will smear and be difficult to remove. It can restrict visibility if it is sprayed on the windshield when there is not enough rain to allow it to be spread outsmoothly.

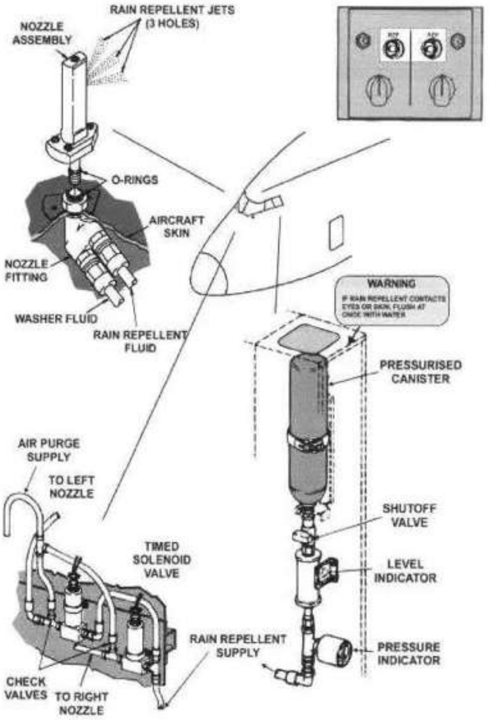


Fig. 12.38 Rain repellent container and pressure / quantity indicator

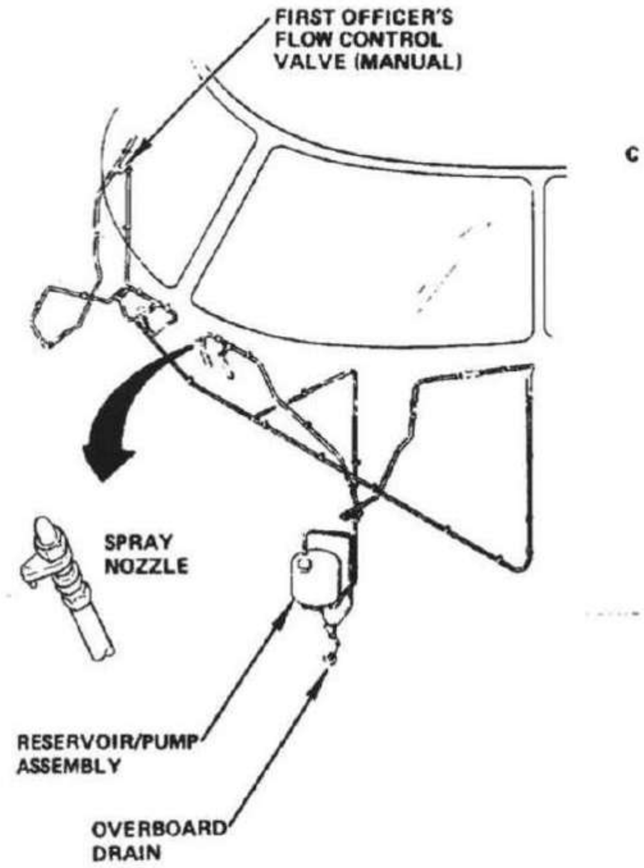


Fig. 12.39 Rain repellent system Rain Repellent Concerns

The rain repellent has been removed due to worries about the environmental effects of the "RainBoe" fluid used as it contains CFC's. It is also poisonous and in 1991 Boeing added D- limonine which has a strong smell of orange peel into RainBoesothatleakagecouldbedetected.Therearenoplans to replace the rain repellent with another liquid product even though there are safe alternatives e.g. "LeBozec".

On 25 May 1982, a 737-200 Adv (PP-SMY) was written off by a heavy landing in a rainstorm. One report stated that "The pilots misuse of rain repellent caused an opticalillusion".

Since early 1994 most aircraft have been built with a coated windshield (such as 'Surface Seal' coated glass from PPG Industries) which has a hydrophobic coating. The coating does deteriorate with time depending upon wiper use and windscreen cleaning methods etc, butcan bere-applied.

Probes and Drain HeatingIntroduction

Icing of air data probes may have fatal effects for safe aircraft operation. Therefore the probes are electrically heated to prevent ice accumulation.

Heated Air Data Probes
The following probes are usually heated:

- PitotProbes
- Staticports
- Angle of AttackSensors
- Temperature Probes (SAT) or(TAT)

General Safety Precautions

- Never touch heated probes you may burn yourfingers.
- Do not operate probe heating systems on ground, except for short test function. The probes may be damaged byoverheat.
- Remove protective covers prior probe heat systemoperation.

Typical Probe Heating System
To control and monitor a probe heating system different techniques areapplied:

Current transformers or shunts in each probe heater circuit supply, via a rotary selector, and an indicator to monitor the respective heater current. Current monitoring relays in the probe heater circuits activate a warning light if the rotary selector is in "OFF" position or insufficient heat current for a probe is detected.

Current sensing relays for each heated probe can trigger a warning light or warning message in the flight compartment if the heater current is insufficient.

Probe heat computers control and monitor the operation of the air data sensors.

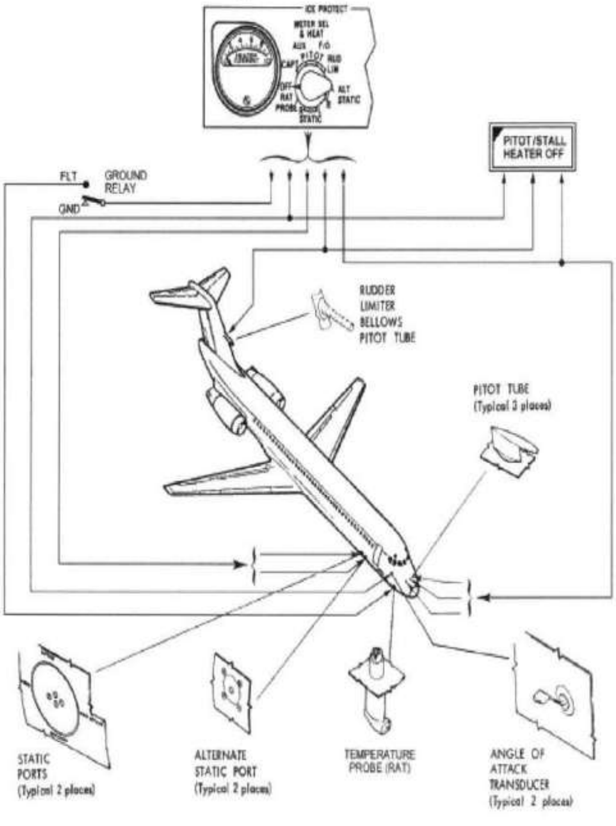


Fig. 12.40 Typical probe heating system

Probe Heating System Monitored by Current Sensing

Relays

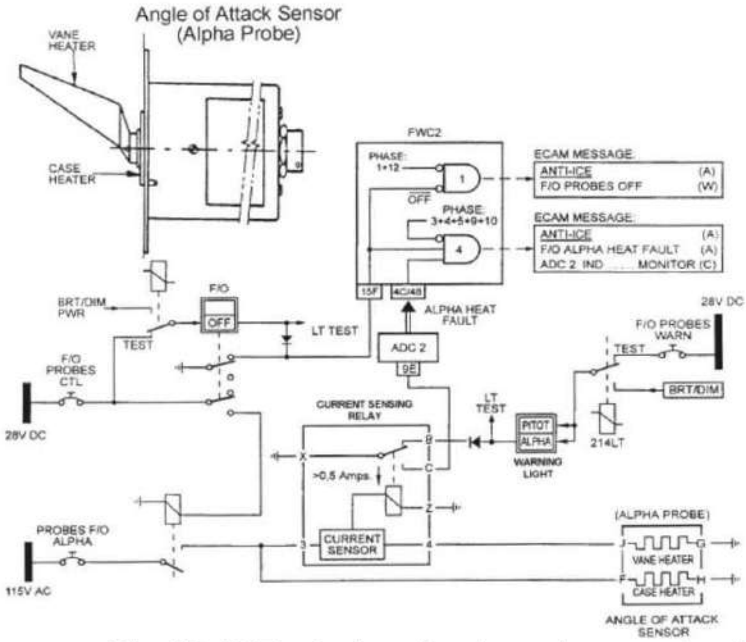


Fig. 12.41 Typical angle of attack sensor heating system

Probe Heating System controlled by Probe Heat Computer Control

Probes and static ports are automatically heated when an engine is running or the aircraft is in flight condition. The PROBES/WINDOW HEAT P/B overrides the automatic operation. On ground, pitot heating is reduced and TAT heating is cutoff.

Probe Heat Computer (PHC)

The Probe Heat Computer controls and monitors the heating current for all probes.

In case of a probe heating fault the Probe Heat Computer (PHC) sends a warning signal to the ECAM system.

The PHC also sends fault messages to the Centralized Maintenance Computer.

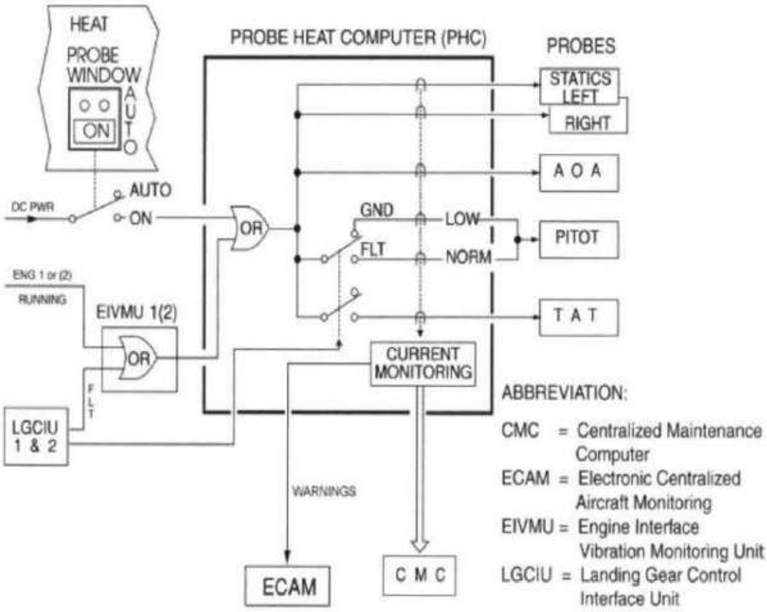


Fig. 12.42 Typical probe heating system

Potable and Waste Water Heating Systems Water and Toilet Drain Heaters

Electrical heaters are provided for toilet drain lines, water lines, drain masts, and waste water drains when they are located in an area that is subjected to freezing temperatures in flight or onground.

The types of heaters used are integrally heated hoses, ribbon, blanket, or patch heaters that wrap around the lines, and gasket heaters. Thermostats are provided in heater circuits where excessive heating is undesirable or to reduce power consumption.

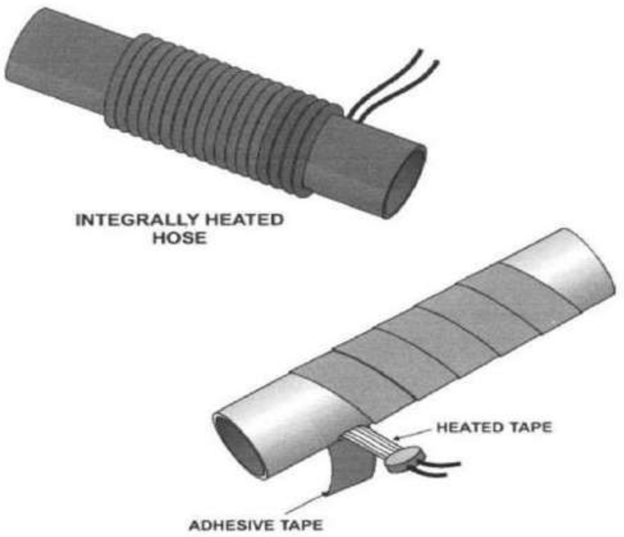


Fig. 12.43 Typical water and drain line heaters Caution:

Some heaters have higher power settings in flight. Therefore, prior to flight simulation on ground or with the aircraft jacked, the heater circuit breakers must be pulled to prevent overheating.

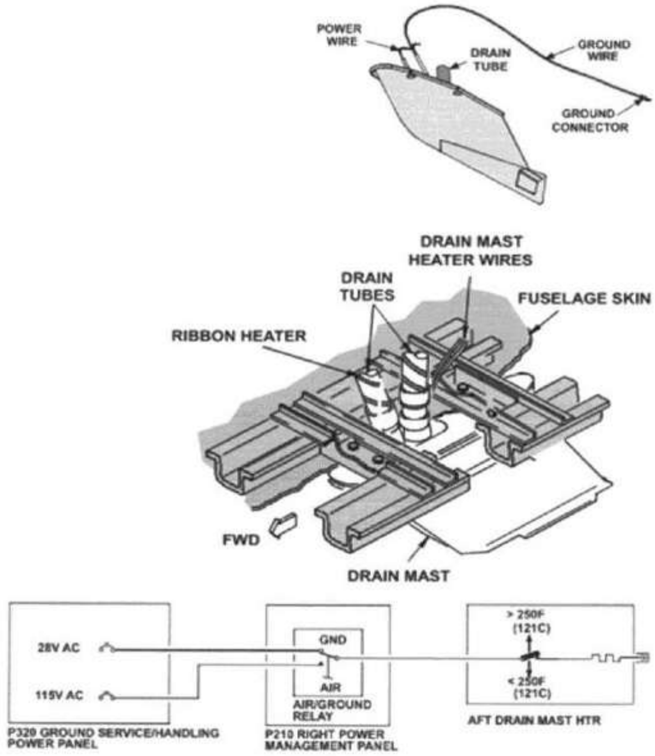


Fig. 12.44 Typical heater types & drain mast heater

1. INTRODUCTION

Landing-gear designs for aircraft vary from simple, fixed arrangements to very complex retractable systems involving many hundreds of parts. The landing gear of an aircraft serves a number of very important functions and it is classified by number of different characteristics. This chapter examines typical examples of landing gear and the systems by which they areoperated.
Purpose of Landing-gear:

- Supports the airplane during groundoperations.
- Dampens the vibrations when the airplane is being taxied or towed & cushions the landingimpact.
- Performs the function of steering andbraking.

Landing gear is attached to primary structural members of the aircraft. The type of gear depends on the aircraft design and its intended use. Most landing gear has 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 landinggear.

Figure 1- Basic landing gear types include those with wheels (a), skids (b), skis (c), and floats or pontoons (d).

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. The landing gear of an airplane consists of main and auxiliary units, either of which may be fixed or retractable. The main landing gear supports aircraft on land or water. It may include a combination of wheels, floats, skis, shock absorbing equipment, brakes, retracting mechanism, controls, warning devices, cowling, fairing and structural members needed for attachment to the primary structure of theaircraft.
The auxiliary landing gear consists of tail or nose wheel installations, skis, outboard pontoons, etc with necessary cowling and reinforcements.

Landing Gear Arrangement

Three basic arrangements of landing gear are used: tail wheel type landing gear (also known as 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 a third wheelassembly.

Figure 2- Tail wheel configuration landing gear on a DC-3 (left) and a STOL Maule MX-7-235 Super Rocket

Advantage-

- Reduced drag in theair.
- Reduced landing gearweight.

Disadvantage-

- Loss of forward visibility during flight maneuvering on ground due to nosehigh.
- Less stable on ground and requires more skill during taxing andtakeoff.

Tandem Landing Gear

Few aircraft are designed with tandem landing gear. As the name implies, this type of landing gear has the main 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 underthefuselagefacilitatestheuseofveryflexiblewings.

Figure 3- 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 4-Tricycle-type landing gear with dual main wheels on a Learjet (left) and a Cessna 172, also with tricycle gear (right).

Tricycle-type landing gear is used on large and small aircraft with the following benefits:
- Allows more forceful application of the brakes without nosing over when braking, which enables higher landingspeeds.
- Provides better visibility from the flight deck, especially during landing and groundmaneuvering.
- Prevents ground-looping of the aircraft. Since the aircraftcenterofgravityisforwardofthemaingear,

forces acting on the center of gravity tend to 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 isachieved through an independent tiller in the flight deck. 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 gearvary.

Many main gears have two or more wheels. 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 bogie. The number of wheels included in bogie is a function of the gross design weight of the aircraft and the surface type on which the loaded aircraft is required toland.

Figure 5- Triple bogie main landing gear assembly on a Boeing777.
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 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. A great deal of the parasite drag caused by light aircraft landing gear can be reduced by building gear as aerodynamically as possible and by adding fairings or wheel pants to streamline the airflow past the protruding assemblies. A small, smooth profile to the oncoming wind greatly reduces landing gear parasitedrag.Thethincrosssectionofthespringsteelstruts

combine with the fairings over the wheeland brake assemblies to raise performance of the fixed landing gear by keeping parasite drag to a minimum. Retractable landing gear stows in fuselage or wing compartments while in flight. Once in these wheel wells, gears are out of the slipstream and do not cause parasites drag. Most retractable gear has a close fitting panel attached to them that fairs with the aircraft skin when the gear is fully retracted. Other aircraft have separate doors that open, allowing the gear to enter or leave, and then closeagain.
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 hydraulics. Landing gear retraction systems are discussed later in thischapter.

Shock Absorbing and Non-Shock Absorbing Landing Gear

In addition to supporting the aircraft for taxi, the forces of impact on an aircraft during landing must be controlled by the landinggear.

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 ofimpact

2. The shock is absorbed by converting the energy into heat energy

Leaf-Type Spring Gear

Many aircraft utilize flexible spring steel, aluminum, or composite struts that receive the impact of landing and return ittotheairframetodissipateataratethatisnotharmful.

The gear flexes initially and forces are transferred as it returns to its original position. The most common example of this type of non-shock absorbing landing gear is the thousands of single-engine Cessna aircraft that use it. Landing gear struts of this type made from composite materials are lighter in weight with greater flexibility and do notcorrode.

Rigid type landing gear

The rigid landing gear is commonly found on helicopters and sailplanes. The gear is rigidly mounted to the aircraft with no specific components to cushion the ground contact other than through the flexing of the landing gear or airframe structure.

Figure 6- Non-shock absorbing struts made from steel, aluminum, or composite material transfer the

impact forces of Landing to the airframe at a non-damagingrate

Shock-Absorbing Landing Gear

Shock-Absorbing Landing Gear dissipates the impact energy of landing through some means. Most of this landing gear does this by forcing a fluid through restriction. The movement of this fluid generates heat, and the heat is radiated into the surrounding atmosphere, thus dissipating the landingenergy.

Two types of shock absorbing landing gear commonly used are the spring-oleo and the air -oleotypes. Spring-oleo struts found in modern aircraft consist of a piston type structure and a heavy coiled spring. The piston and cylinder arrangement provides an oil chamber and an orifice through which oil is forced during landing.

As the strut collapse the coil spring is compressed thus providing additional cushioning. Spring supports the aircraft weight on theground and during taxiing and the oleo strut absorbs the shock of landing.

Air-oleo struts consist of a fluid metering unit. During compression of the strut at landing an orifice provides a restriction of fluid flow and this reduces the rate at which the piston can move into the cylinder. This provides a cushioning effect to reduce the shock of landing.
Compressed air acts as a shock absorber during the time that the aircraft is taxiing.

Landing gear components

Landing gear assemblies are made up of various components designed to stabilize the assembly.

TRUNNION

It is the Portion of the landing gear assembly attached to the airframe. The trunnion is supported at its end by bearing assemblies, which allow the gear to pivot during retraction and extension. The landing gear strut extends down from the approximate center of thetrunnion.

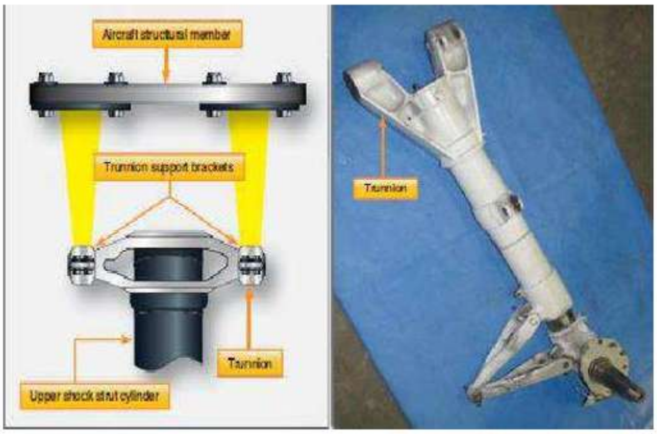


Figure 7- The trunnion is a fixed structural support that is part of or attached to the upper strut cylinder of a landing gearstrut.Itcontainsbearingsurfacessothegearcanretract.

SHOCK STRUT

True shock absorption occurs when the shock energy of landing impact is converted into heat energy, as in a shock strutlandinggear.Thisisthemostcommonmethodof 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 properoperation.

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, consults the manufacturer's maintenance Instructions. 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 filledwith 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 iscompressed.

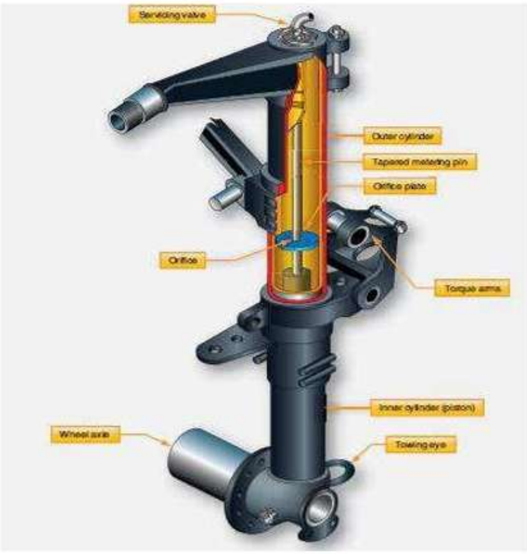


Figure 8- A landing gear shock strut with a metering pin to control the flow of hydraulic fluid from the lower chamber to the upper chamber during compression

Most shock struts employ a metering pin to that control 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 metering pin in the orifice increases, less fluid passes. Pressure build-up caused by strut compression and thehydraulic 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 theholesinthemeteringtubecontroltheflowoffluidfrom

the bottom chamber to the top chamber during 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. The upper cylinder of a shock strut typically contains a valve fittingassembly.

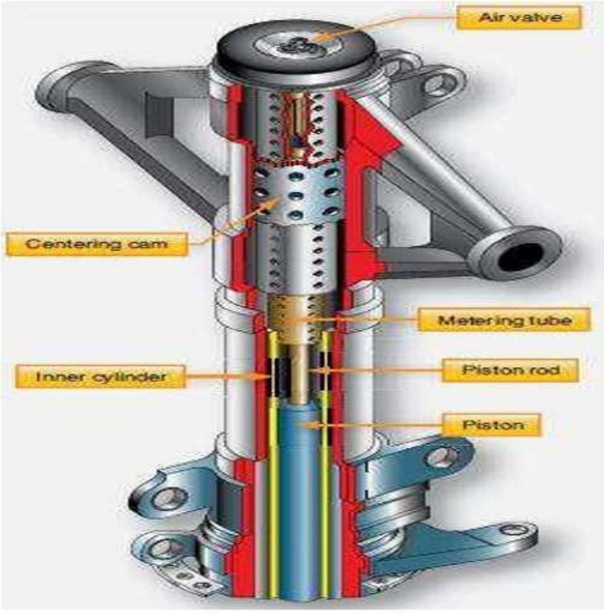


Figure 9- Some landing gear shock struts use an internal metering tube rather than a metering pin to control the flow of fluid from the bottom cylinder to the topcylinder.

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. 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.

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 to the aircraft. It also aligns the wheelswiththelongitudinalaxisoftheaircraftpriorto landing when the strut is fully extended. Many nose gear shock struts also have attachments for the installation of an external shimmydamper.

Figure 10- Torque links align the landing gear and retain the piston in the upper cylinder when the strut is extended

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 theselugs.

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 ornitrogen.

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Figure 11- A shimmy damper helps control oscillations of the nosegear. Figure 12- A towing lug on a landing gear is the designed means for attaching a tow bar.

Figure 13- 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.

Shock Strut Operation

Figure 14 shown below 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 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 outbumps.

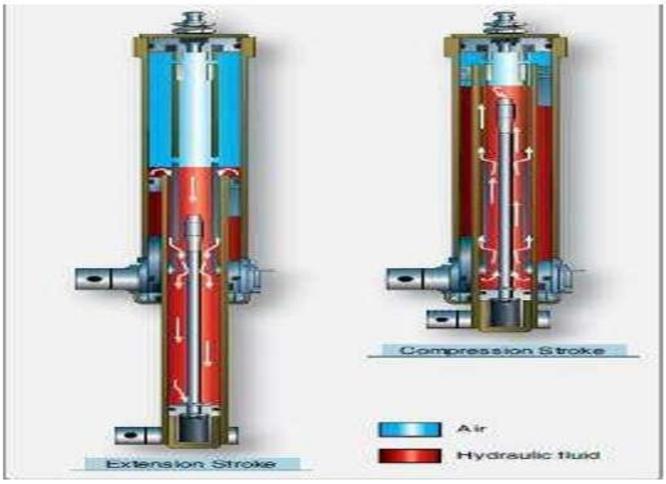


Figure 14- Fluid flow during shock strut operation is controlled by the taper of the metering pin in the shock strut orifice.

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.

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 entrappedair.

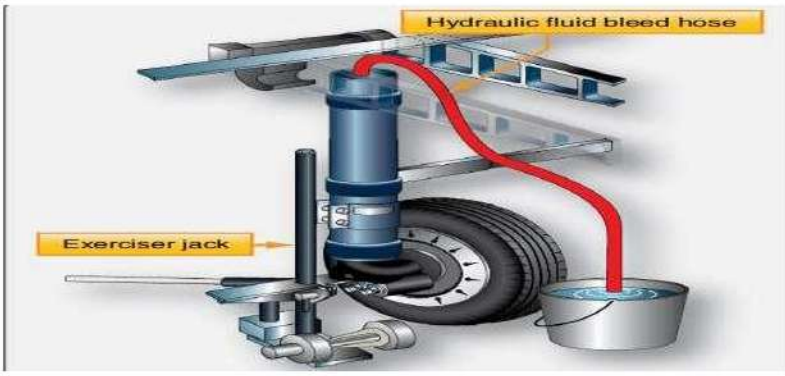


Figure-18 Air trapped in shock strut hydraulic fluid is bled by exercising the strut through its full range of motion while the end of an air-tight bleed hose is submerged in a container of hydraulicfluid.

Landing Gear Alignment, Support, and Retraction

Retractable landing gear consists 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 has steering mechanisms attached to thegear.

Alignment

As previously mentioned, a torque arm or torque links assemblykeepsthelowerstrutcylinderfromrotatingoutof 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. The aircraft's main wheels must be inspected and adjusted, if necessary, to maintain the proper tow-in or two- out and the correct camber. Tow-in and two-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. Veerawayfromthelongitudinalaxis(two-out).[Figure-19]

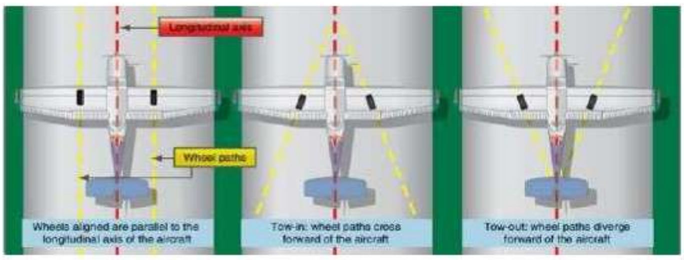


Figure-19Wheel alignment on an aircraft.

The manufacturer's maintenance instructions give the procedure for checking and adjusting tow-in or two-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 towedout.[Figure-20]

Figure-20 Finding tow-in and two-out on a light aircraft with spring steel struts.

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 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-21] 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 two-out.[Figure-31] Follow all manufacturers Instructions.

Figure-21 Camber of a wheel is the amount the wheel is tilted out of the vertical plain. It can be measured with a bubbleprotractor.

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 duringflight.

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 theaircraft

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-22] 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 groundmaneuvers.

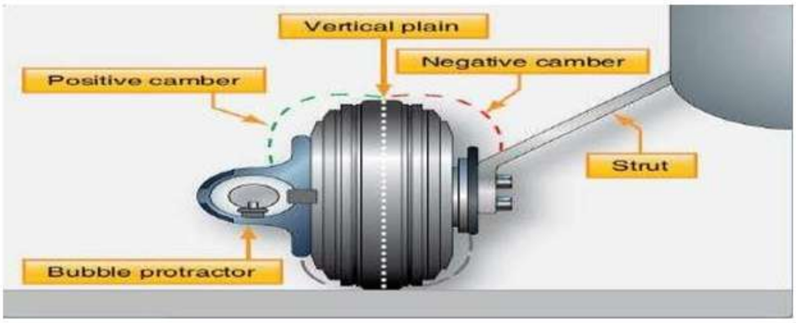


Figure-22 A hinged drag strut holds the trunnion and gear firm for landing and ground operation. It folds at the hinge to allow thegear toretract.
Small Aircraft Retraction Systems

As the speed of a light aircraft increases, there reaches a point where the parasite drag created by the landing gear in the wind is greater than the induced drag caused by the added weight of a retractable landing gear system. Thus, many light aircraft have retractable landing gear. There are many unique designs. The simplest contains a lever in the flight deck mechanically linked to the gear. Through mechanical advantage, the pilot extends and retracts the landing gear by operating the lever. Use of a roller chain, sprockets, and a hand crank to decrease the required force iscommon.

Electrically operated landing gear systems are also found on light aircraft. An all-electric system uses an electric motor and gear reduction to move the gear. The rotary motion of the motor is converted to linear motion to actuate the gear. This is possible only with the relatively lightweight gear found on smaller aircraft. An all-electric gear retraction system is illustrated inFigure-23.

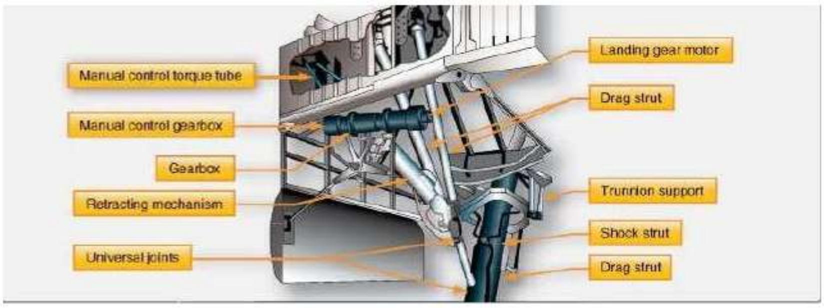


Figure 23- A geared electric motor landing gear retraction system.

A more common use of electricity in gear retraction systems is that of an electric/hydraulic system found in many Cessna and Piper aircraft. This is also known as a power pack system. A small lightweight hydraulic power pack contains several components required in a hydraulic system. These include the reservoir, a reversible electric motor-driven hydraulic pump, a filter, high-and-low pressure control valves, a thermal relief valve, and a shuttle valve. Some power packs incorporate an emergency hand pump. A hydraulic actuator for each gear is driven to extend or retract the gear by fluid from the powerpack.

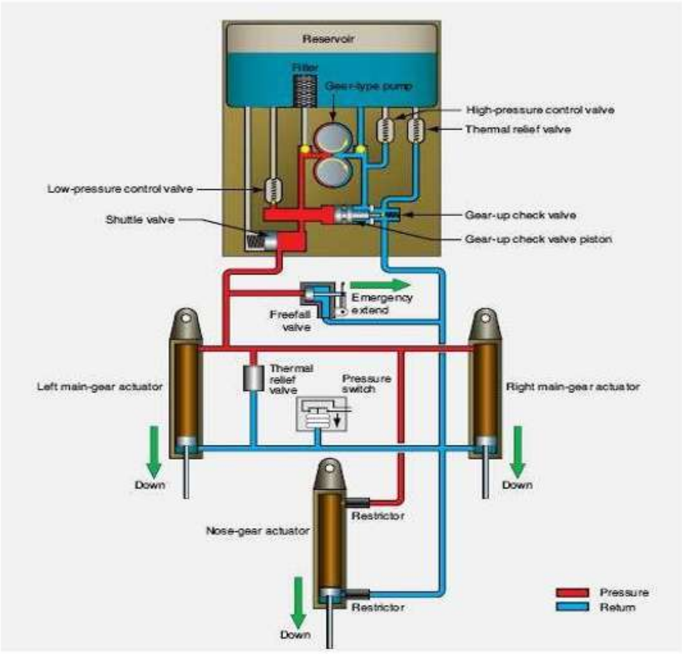


Figure 24 A popular light aircraft gear retraction system that uses a hydraulic power pack in the gear down condition.

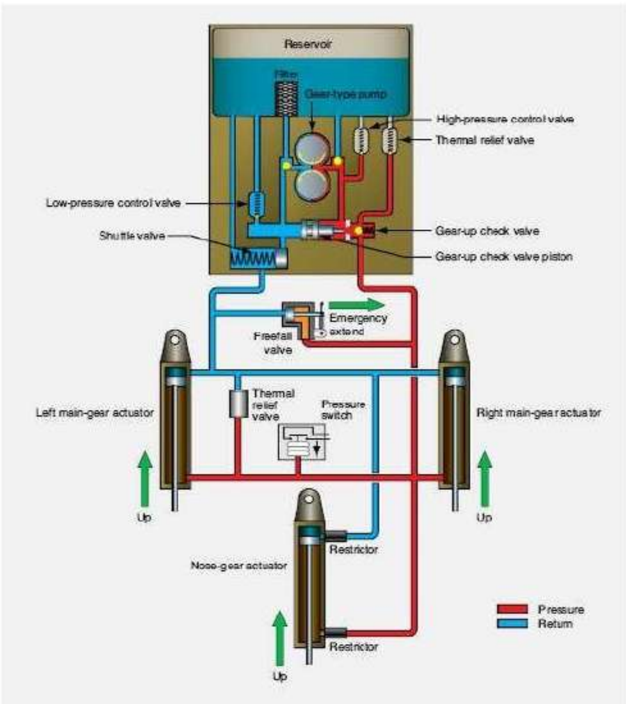


Figure 25 A hydraulic power pack gear retraction system in the gear upcondition.

When the flight deck gear selection handle is put in the gear down position, a switch is made that turns on the electric motor in the power pack. The motor turns in the direction to rotate the hydraulic gear pump so that it pumps fluid to the gear-down side of the actuating cylinders. Pump pressure moves the spring-loaded shuttle valve to the left to allow fluid to reach all three actuators. Restrictors are used in the nose wheel actuator inlet and outlet ports to slow down the motion of this lighter gear. While hydraulic fluid is pumped to extend the gear, fluid from the upside of the actuators returnstothereservoirthroughthegear-upcheckvalve.

When the gear reaches the down and locked position, pressure builds in the gear-down line from the pump and the low-pressure control valve unseats to return the fluid to the reservoir. Electric limit switches turn off the pump when all three gears are down and locked. To raise the gear, the flight deck gear handle is moved to the gear-up position. This sends current to the electric motor, which drives the hydraulic gear pump in the opposite direction causing fluid to be pumped to the gear-up side of theactuators.

In this direction, pump inlet fluid flows through the filter. Fluid from the pump flows thought the gear-up check valve to the gear-up sides of the actuating cylinders. As the cylinders begin to move, the pistons release the mechanical down locks that hold the gear rigid for ground operations. Fluid from the gear-down side of the actuators returns to the reservoir through the shuttle valve. When the three gears are fully retracted, pressure builds in the system, and a pressure switch is opened that cuts power to the electric pump motor. The gear is held in the retracted position with hydraulic pressure. If pressure declines, the pressure switch closes to run the pump and raise the pressure until the pressure switch opensagain.

Large Aircraft Retraction Systems

Large aircraft 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, up locks, down locks, sequence valves, priority valves, tubing, and other conventional hydraulic system components. These unitsare 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-26 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-26, 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 down locks 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

Manifoldthroughtheselectorvalve.Tolowerthegear,the selector is put in the gear-down position. Pressurized hydraulic fluid flows from the hydraulic manifold to the nose gear up lock, which unlocks the nose gear. Fluid flows to the gear-down side of the nose gear actuator and extends it. Fluid alsoflowstotheopensideofthemaingeardooractuators.

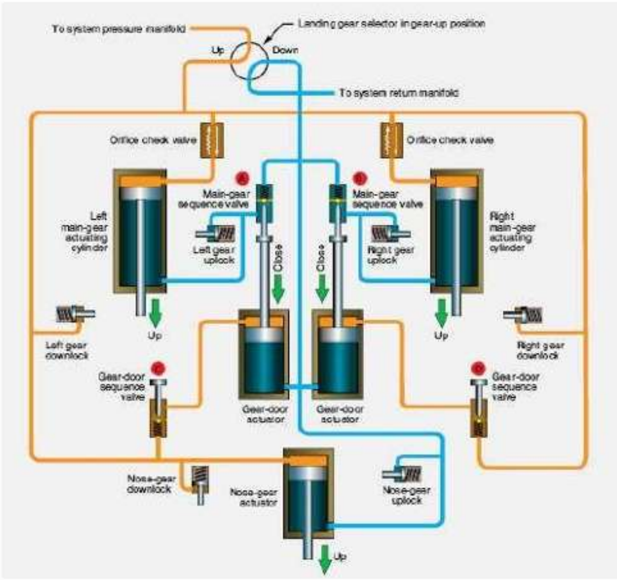


Figure 26 A simple large aircraft hydraulic gear retraction system.

As the doors open, sequence valves A and B block fluid from unlocking the main gear up locks 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 up locks 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 fromeach 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. These 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 crucialsystem.

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 up locks. When the handle is operated, it releases the up locks 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. The popular small aircraft retraction system shown in Figure-24 and 25 uses a free-fall valve for emergency gear extension. Activated from the flight deck, when the free-fall valve is opened, hydraulic fluid is allowed to flow from the gear-up side of the actuators to the gear-down side of the actuators, independent of the power pack. Pressure holding the gear up is relieved, and the gear extends due to its weight. Air moving past the gear aids in the extension and helps push the gear into the down-and-locked position. Large and high performance aircraft are equipped with redundanthydraulic

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 up locks and allow the gear to freefall.

In some small aircraft, the design configuration makes emergency extension of the gear by gravity and air loads alone impossible or impractical. Force of some kind must therefore be applied. Manual extension systems, wherein the pilot mechanically cranks thegear into position, are common. 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 Safety Devices

There are numerous landing gear safety devices. The most common are those that prevent the gear from retracting or collapsing while on the ground. Gear indicators are another safety device. They are used to communicate to the pilot the position status of each individual landing gear at any time. A further safety device is the nose wheel centering device mentioned previously in this chapter.

Safety Switch

A 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-27]

Figure 13-27. Typical landing gear squatswitches.

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 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 beraised.

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 totheswitch.Nophysicalcontactismade.Whenthegearis

in the designed position, the metallic target is close to the inductor in the sensor which induces the return voltage. 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– nogogaugesareoftenusedtosetthedistance.[Figure-28]

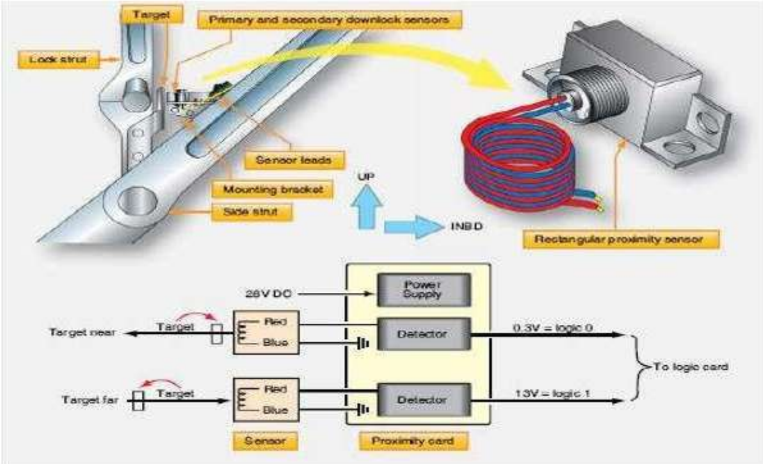


Figure 28 Proximity sensors are used instead of contact switches on much landing gear.

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 pre-drilled holes of gear components that keep the gear from collapsing. Another commonly used groundlock 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-29]

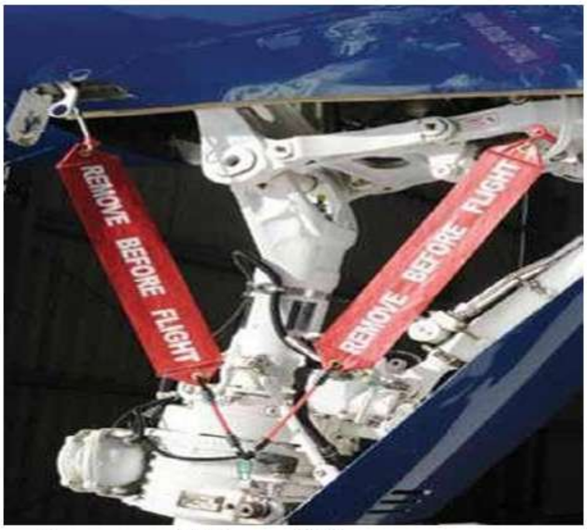


Figure 29 Gear pin ground lock devices.

Landing Gear Position Indicators

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 mean it is safe to land. All lightsouttypicallyindicatesthatthegearisupandlocked,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.

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 theaircraft.

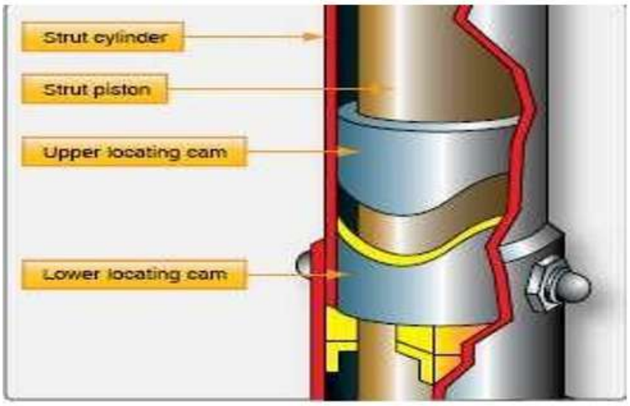


Figure 30 A cutaway view of a nose gear internal centering cam.

[Figure-30] 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 amp/track guides the roller or pin in such a manner that the nose wheel is straightened as it enters the wheelwell.

Nose Wheel Steering Systems

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 differentialbraking.

Small Aircraft

Most small aircraft have steering capabilities through the use of a simple system of mechanical linkages connected to the rudder pedals. Push-pull tubes are connected to pedal horns on the lower strut cylinder. As the pedals are depressed, the movement is transferred to the strut piston axle and wheel assembly which rotates to the left orright.

Large Aircraft

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 canswivel. The following explanation accompanies Figure-31, 32, 33, 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 nosegear.AsshowninFigure-52,pressurefromtheaircraft 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 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 returnmanifold.

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 Aextends.

Figure 31 Example of a large aircraft hydraulic nose wheel steering system with hydraulic and mechanical units.

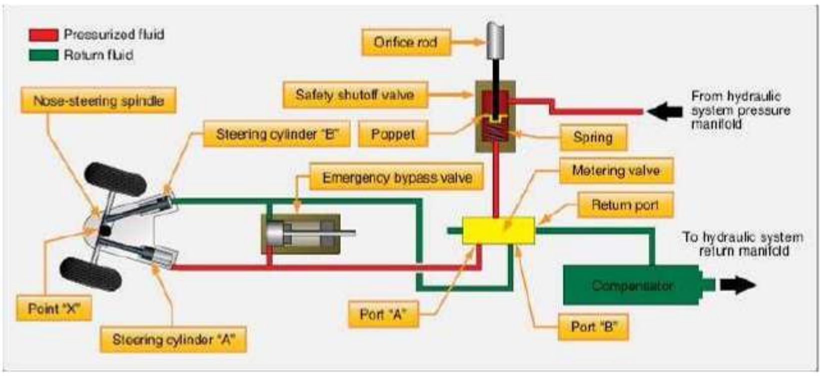


Figure-32 Hydraulic system flow diagram of large aircraft nose wheel steeringsystem

The rear of the spindle contains gear teeth that mesh with a gear on the bottom of the orifice rod. [Figure-31] .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 neutralposition.

The metering valve and the compensator unit of the nose wheel steering system are illustrated in [Figure-33]. 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 therearofthepistonfrominterferingwiththemovementof 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 shimmydampers.

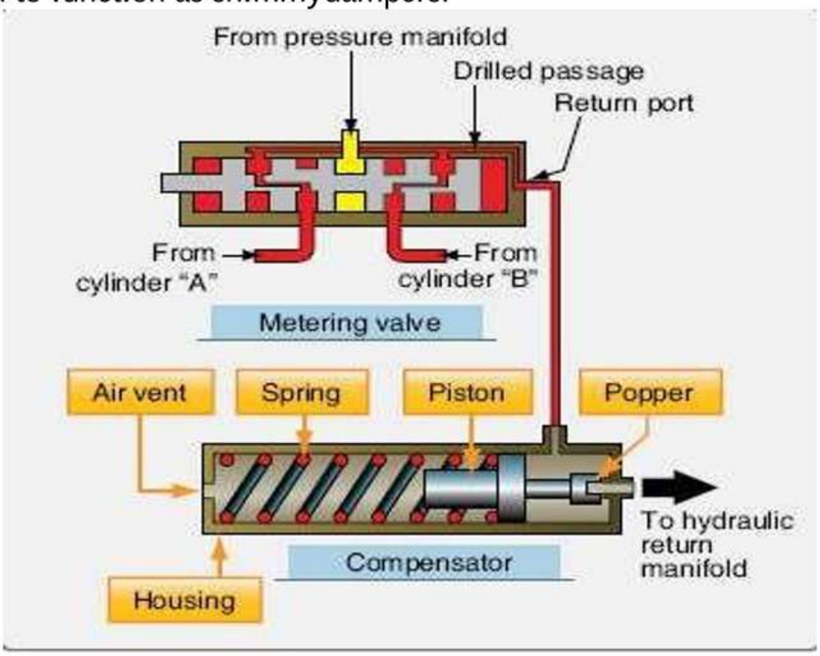


Figure-33Hydraulic system flow diagram of large aircraft nose wheel steeringsystem.

Shimmy Dampers

Torque links attached from the stationary upper cylinder of a nose wheel strut to the bottom moveable 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 must be controlled through the use of a shimmy damper. A shimmy damper controls nose wheel shimmy 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.

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 dampenvibration.

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 diameterofthedamperhousingwhentheshimmymotionis 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 used to construct this type of shimmy damper provide a long service life without the need to ever add fluid to the unit.[Figure-36]

Figure-36 A non-hydraulic shimmy damper uses a rubber piston with lubricant that dampens via motion against the inner diameter of the unithousing.

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 onolder aircraft. Later, wheels with two nearly symmetrical halves were developed. Nearly all modern aircraft wheels are of this two piece construction. [Figure37]

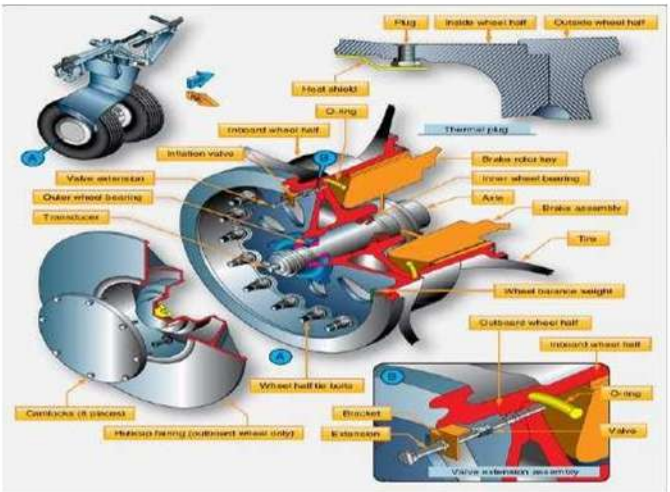


Figure-37 Features of a two piece aircraft wheel found on a modern airliner.

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 an O- ring, which seals the rim since most modern aircraft utilize 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 landing. To strengthen this area during manufacturing, the bead seat area is typically rolled to press tress it with 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 key ways 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-38]

Figure-38 Keys on the inner wheel half of an aircraft wheel used to engage and rotate the rotors of a discbrake.

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 thewheel

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 thermalplugs.

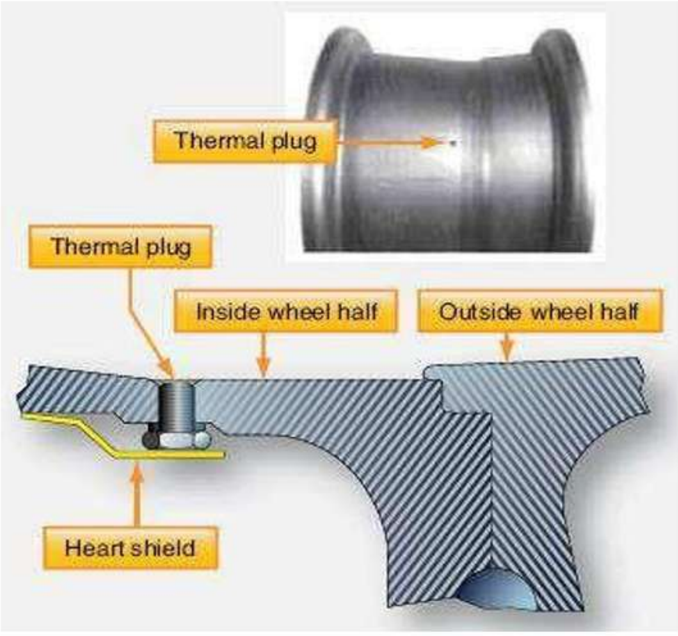


Figure-39 Heavy use of the aircraft brakes can cause tire air temperature and pressure to rise to a level resulting in explosion of the wheel assembly. To alleviate this, thermal plug(s) mounted in the inner wheel half of a high performance aircraft wheels are made with a fusible core that melts and releases the air from the tire beforeexplosion.

[Figure-39]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 pointalloy.

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 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 anddeflation.

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. Aircraft with anti-skid brake systems typically mount the wheel-spin transducer here. It is sealed and may also serve as a hub cap. The 737 outer wheels half illustrated in Figure-59 also have 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 valvestemextensionmaypassfromtheinnerwheelhalfor
the valve stem itself may fit through such a hole if a tube- type tire isused.

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 theaircraft.

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 beinvestigated.

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 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 tiebolts. 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 valvestem.

NOTE: As a precautionary measure, remove only one tire and wheel assembly from a pair at a time. This leaves a tire injury topersonnel.

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-63] As stated above, 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 wheelfailure.

Dis-assembly of the Wheel

Dis-assembly of the wheel should take place in a clean area onaflatsurface,suchasatable.Removethewheelbearing

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. They are not designed to receive the repeated hammering of an impact tool and will be damaged ifused.

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 compressedair.

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 metalsurfaces.

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.

Wheel Bearing Inspection

Once cleaned, the wheel bearing is inspected. There are many unacceptable conditions of the bearing and bearing cup, which aregrounds 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 asfollows:

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. [Figure41]

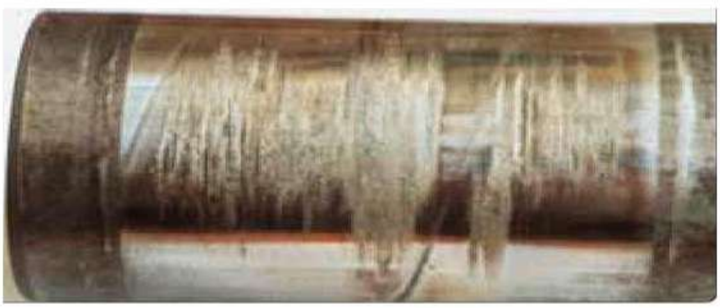


Figure 41 Galling is caused by rubbing of mating surfaces. The metal gets so hot it welds, and the surface metal is destroyed as the motion continues and pulls the metal apart in the direction ofmotion.

Spalling— A chipped away portion of the hardened surface of a bearing roller orrace.

Figure 42Spalling is a chipped away portion of the hardened surface of a bearing roller or race.

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 aswell.

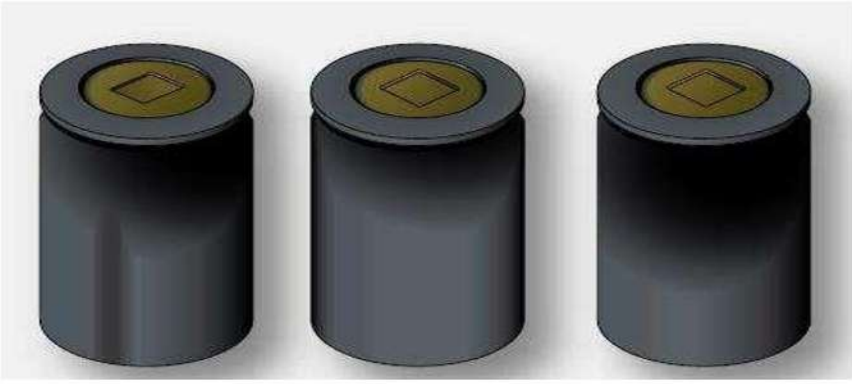


Figure 43 Overheating caused by lack of sufficient lubrication results in a bluish tint to the metal surface. The ends of the rollers shown were overheated causing the metal to flow and deform, as well as discolor. The bearing cup raceway is usually discolored aswell.

Brinelling— it is 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-44]

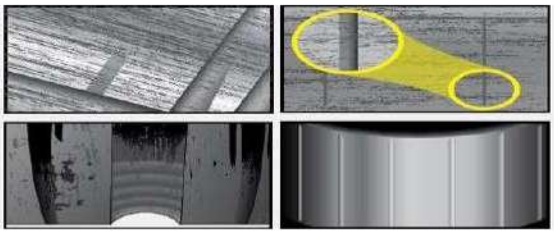


Figure 44- Brinelling Indications

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-45]

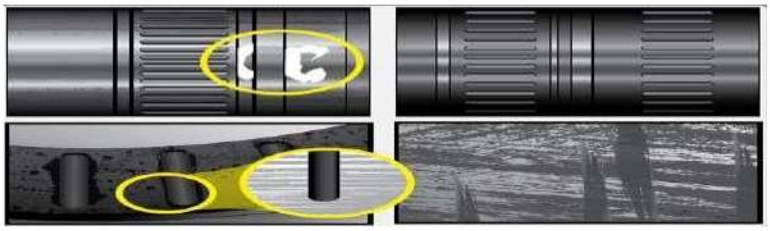


Figure-45False brinelling is caused by vibration of the bearing while in a staticstate.

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-46]

Figure 46 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.

Etching and corrosion—caused when water and the damage caused by water penetrates the surface treatment of the bearingelement.Itappearsasareddish/browndiscoloration.

Figure 47 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.

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-48]

Figure 48- 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 bearingcup.

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-49] The cup is usually removed by eating 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 specificinstructions.

Figure-49 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.

Bearing Handling and Lubrication

Handling of bearings 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. Proper lubrication isa 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-50]

Figure-50 A pressure bearing lubricating tool.

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. 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 mounted, 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.Nocorrosionispermittedinthisarea.[Figure-51]

Figure-51 Inspection for cracks in the wheel disc drive key area is performed with dye penetrant on many wheels.

Wheel Tie Bolt Inspection

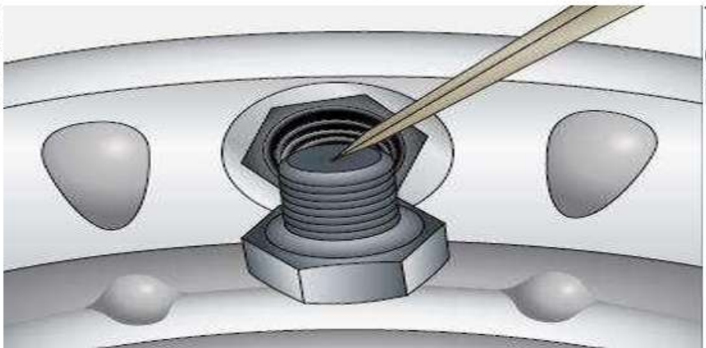
Wheel half tie bolts are under great stress while in service and require inspection. The tie bolts stretch and change dimension usually at the threads and under the bolt head. These are areas where cracks are most common. Magnetic particle inspection can reveal these cracks. Follow the maintenance manual procedures for inspecting tiebolts.

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 allows excessive movement. The wheel manufacturer's maintenance instructions should be used to perform a complete inspection of this criticalarea.

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-52]



Figue-52 Visually inspects the core of a thermal or fusible plug for deformation associated with heat exposure. Replace all of the plugs if any appear to have begunto deform.

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 tousethewheel.Thebalanceweightsareboltedtothewheel

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- 53]

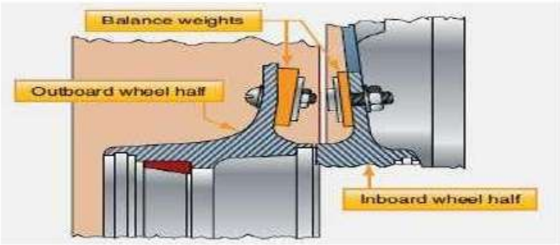


Figure-53 Two piece aircraft wheels are statically balanced when manufactured and may include weights attached to each wheel half that must stay with the wheel during its entire serviceablelife.

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 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 reasonableamountoftime.Theyholdtheaircraftstationary during engine run-up and, in many cases, steer the aircraft 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 maintenanceofthebrakesisessentialforeffectiveoperation.

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. Ex pander tube brakes are found on older large aircraft. The use of carbon discs is increasing in the modern aviationfleet.

Single Disc Brakes

Small, light aircraft typically achieve effective braking using a single disc keyed or bolted to each wheel. As thewheel

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 arepressed.

Floating Disc Brakes

A floating disk brake is illustrated in Figure-54 a more detailed; exploded view of this type of brake is shown in Figure-79. 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 aredesigned 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 arestationary.

Figure 54A single disc brake is a floating-disc, fixed caliper brake.

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-55]

Figure 55 An exploded view of a single-disc brake assembly found on a light aircraft.

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 maintenanceinstructions.

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 accomplishesthisasdescribedabove.Itcanalsobe 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-56.
An exploded detail view of the same type of brake is shown in Figure-57. 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 back plate. Two anchor bolts that pass through the pressure plate are secured to the cylinder 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 back plate 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 back plate 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 liningsanddiscsisrequired.Replacementofpartsworn

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.

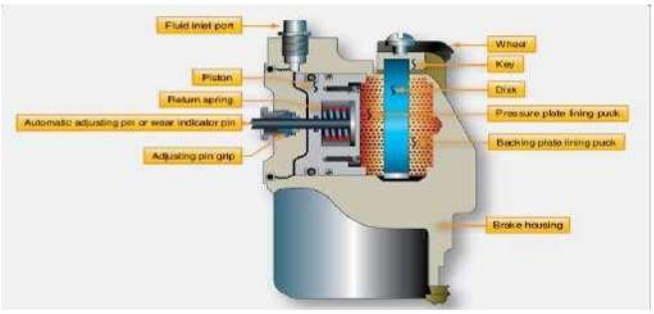


Figure 56 A cross-sectional view of a Goodyear single-disc brake caliper illustrates the adjusting pin assembly that doubles as a wearindicator.

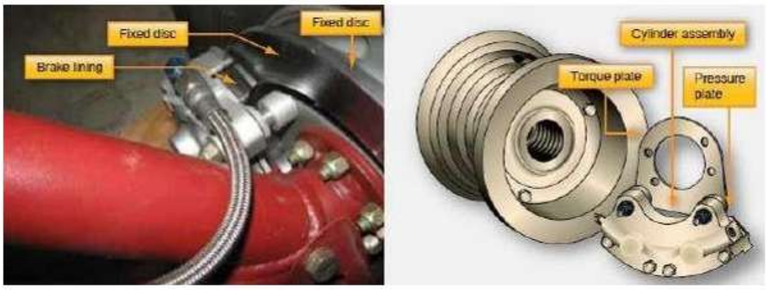


Figure 57-A Cleveland brake on a light aircraft is a fixed- disc brake. It allows the brake caliper to move laterally on anchorboltstodeliverevenpressuretoeachsideofthebrake disc.

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.

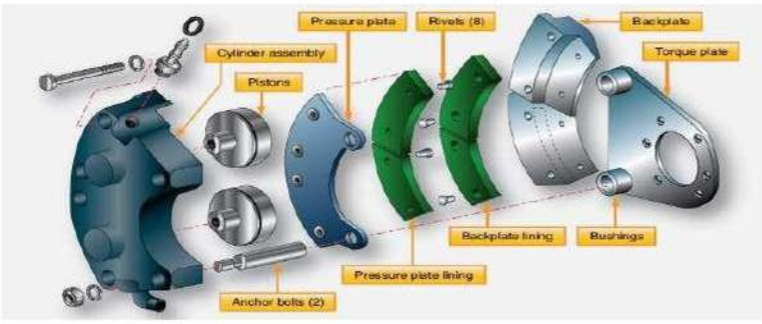


Figure 58 An exploded view of a dual-piston Cleveland brakeassembly.

The caliper mounting bolts are long and mount through the center carrier, as well as the back plate which bolts to the housing assembly.

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 back plate, and a back plate 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. [Figure59]

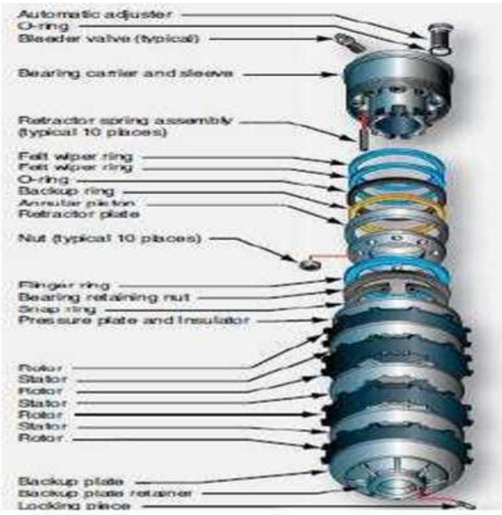


Figure 59 - A multiple disc brake with bearing carrier upon which the parts of the brake are assembled including an annular cylinder and piston assembly that apply pressure evenly to a stack of rotors andstators.

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-60] 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 1/8-inch thick. They do notdissipateheatverywellandhaveatendencytowarp.

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 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 thebrakeitsname.Segmentedrotormultiple-discbrakesare 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-86. 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 andpistons.

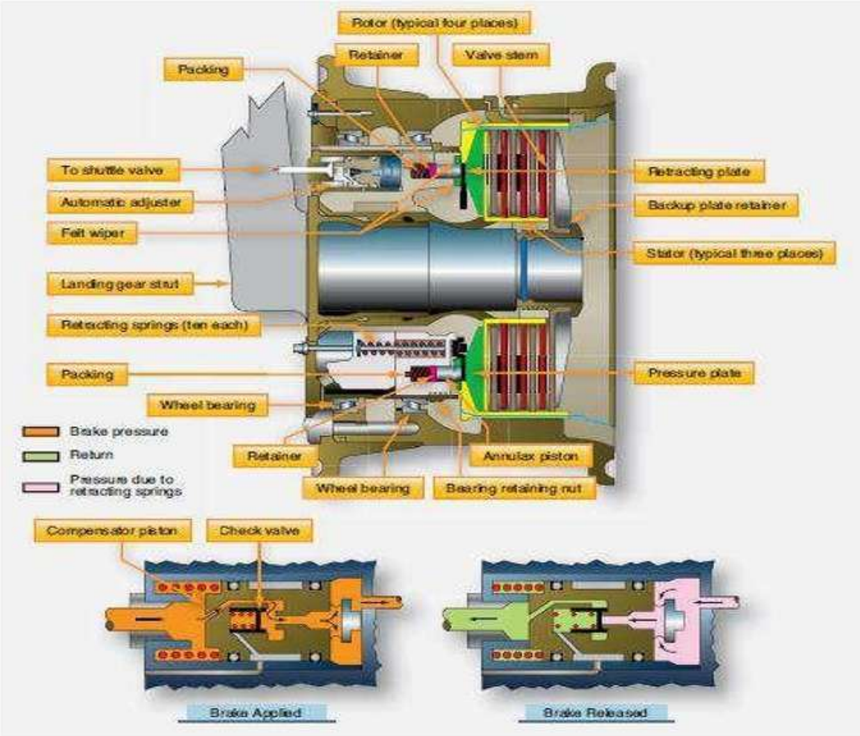
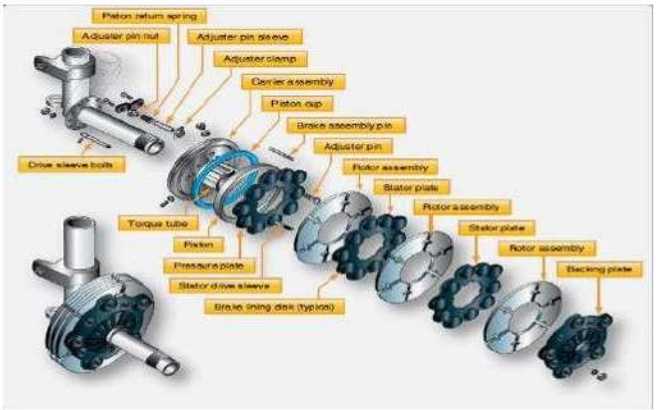


Figure 60 A multiple-disc brakes with details of the automaticadjuster.
Figure-61]Most segmented rotor-disc brakes have numerous individual cylinders machined into the brake housing into which fit the same number of actuating pistons. Often, these cylinders are supplied by two different hydraulic sources, alternating every other cylinder from a single source. If one source



fails, the brake still operates sufficiently on the other.

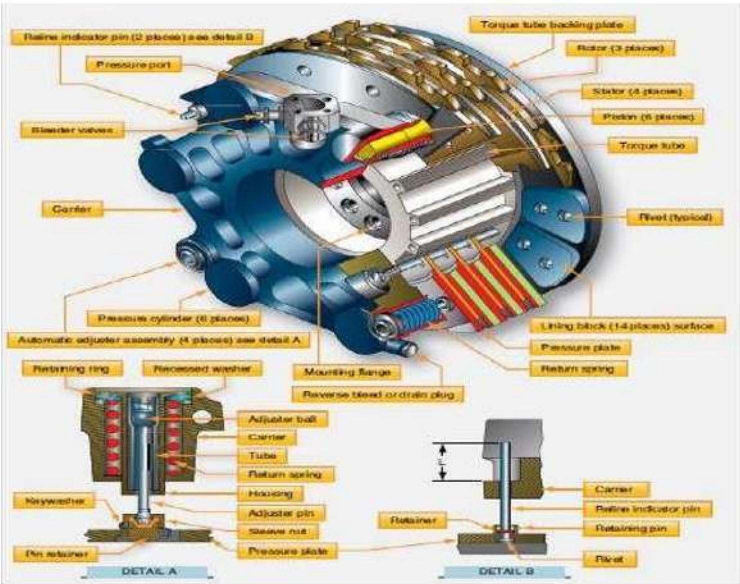
61 Exploded and detail views of segmented rotor brakes.

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 brakediscs.
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- 61] 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 aresandwiched under hydraulic pressure against the backing plate of the brake assembly when the brakes areapplied.

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-61] 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-61] The space between the liner blocks aids in the dissipation of heat. The composition of the lining materials vary. Steel is often used. Rotors are slit or segmented discs that have notches or tangs in the external circumference that key to the rotating wheel.

Figure 62 The multiple-disk brake assemblyand details from a Boeing737.

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- 86] 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 back-plate 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 devicesvaries with brake design. [Figure-62] 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 back plate protrudes through the carrier. The amount that it protrudes with the brakes appliedismeasuredtoascertainifnewliningsarerequired.

NOTE: 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 readcorrectly.

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-63] Carbon brakes are approximately fortypercent

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 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-aircraftapplications.

Figure 13-63 A carbon brake for a Boeing 737.

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 themarket.

Expander Tube Brakes

An expander tube brake is a different approach to breaking 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 liningmaterial.

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-90] 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 expandertubeandthebrakedrumisadjustablebyrotatingan adjuster on some expander tube brakes.

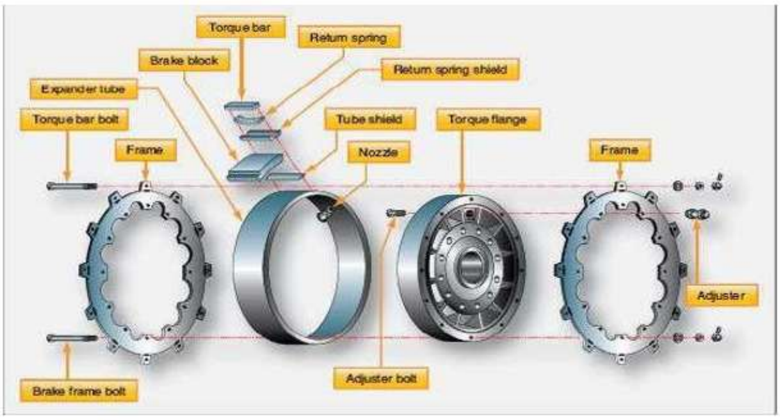


Figure 64 An expanded view of expander tube brake assembly

Consult the manufacturer's maintenance manual for the correct clearance setting. Figure-91 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 brakesystems.

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 break assembliesarediscussedinthissection.Therearethreebasic actuating systems:

1. An independent system not part of the aircraft main hydraulicsystem;
2. A booster system that uses the aircraft hydraulic system intermittently when needed;and
3. A power brake system that only uses the aircraft main hydraulic system(s) as a source ofpressure.

Systems on different aircraft vary, but the general operation is similar to thosedescribed.

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 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-65]

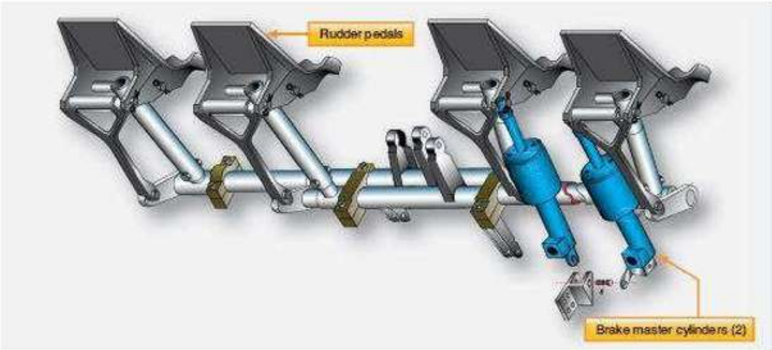


Figure 65 Master cylinders on an independent brake system are directly connected to the rudder pedals or are connected through mechanicallinkage.

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-64]

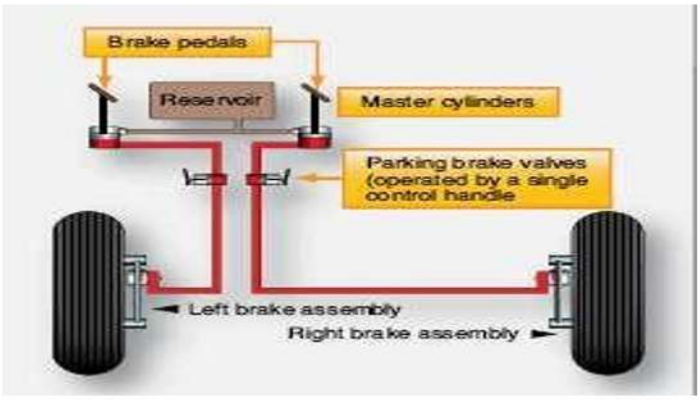


Figure 66 A remote reservoir services both master cylinders on some independent brakingsystems.
A few light aircraft with nose wheel steering have only one master cylinder that actuates both main wheel brakes. This is possible because steering the aircraft during taxi does not require differential braking Regardless of the set-up; it is the master cylinder that builds up the pressure required for braking.

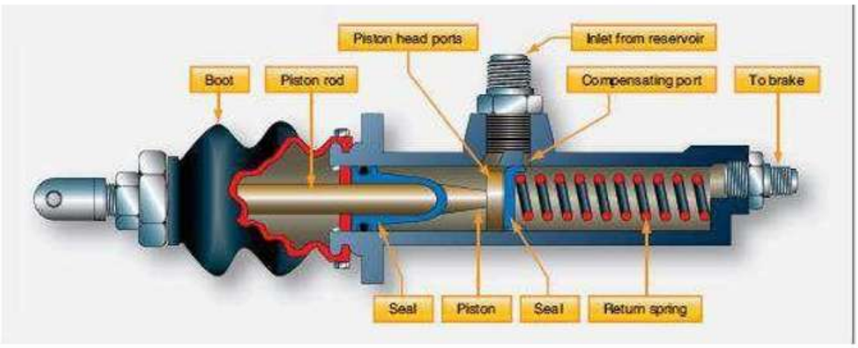


Figure 67 A Goodyear brake master cylinder from an

independent braking system with a remote reservoir.

A master cylinder used with a remote reservoir is illustrated in Figure 67. 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. Hydraulic fluid expands as temperature increases. Trappedfluid
Can cause a brake to drag against the rotor(s). Leaks may also result. When the brakes are not applied, fluid must be allowed to expand safely without causing these issues. A compensating port is included in most master cylinders to facilitate this. In the master cylinder in Figure 67, this port is opened when the piston is fully retracted. Fluid in the brake system is allowed to expand into the reservoir, which has the capacity to accept the extra fluid volume. The typical reservoirisalsoventedtotheatmospheretoprovidepositive pressure on the fluid. 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 temperatureisrelievedbyaspringinthemechanicallinkage.
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. 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-95] 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 thefluid.

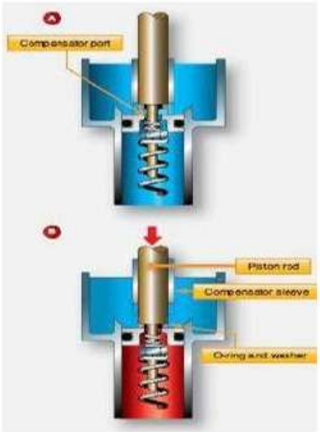


Figure 68-A common master cylinder with built-in reservoir is shown. Illustration A depicts the master cylinder when the brakes are off. The compensating port is open to allow fluid to expand into the reservoir should temperature increase. In B, the brakes are applied. The piston rod-end seal covers the compensating port as it contacts the pistonhead.

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 cylinderforeachbrakeismechanicallyattachedtotherudder pedals. However, the boosted brake master cylinder operates differently. [Figure-69]

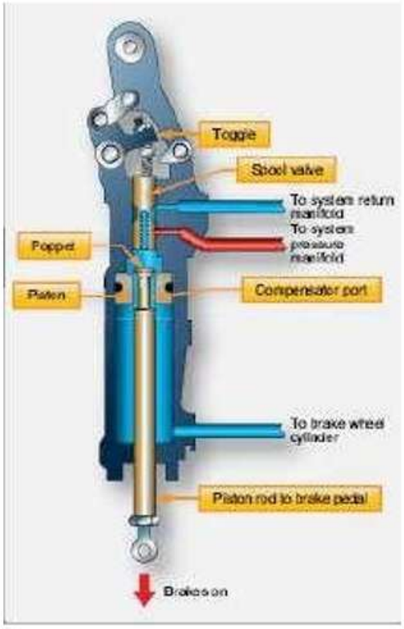


Figure 69 A master cylinders for a boosted brake system augments foot pedal pressure with aircraft system hydraulic pressure during heavy braking.

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. When the pedal is released, the piston rod travels in the opposite direction, and the piston returnstothepistonstop.Thecompensatingpoppetreopens.

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 theport.

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 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 inFigure-70A.