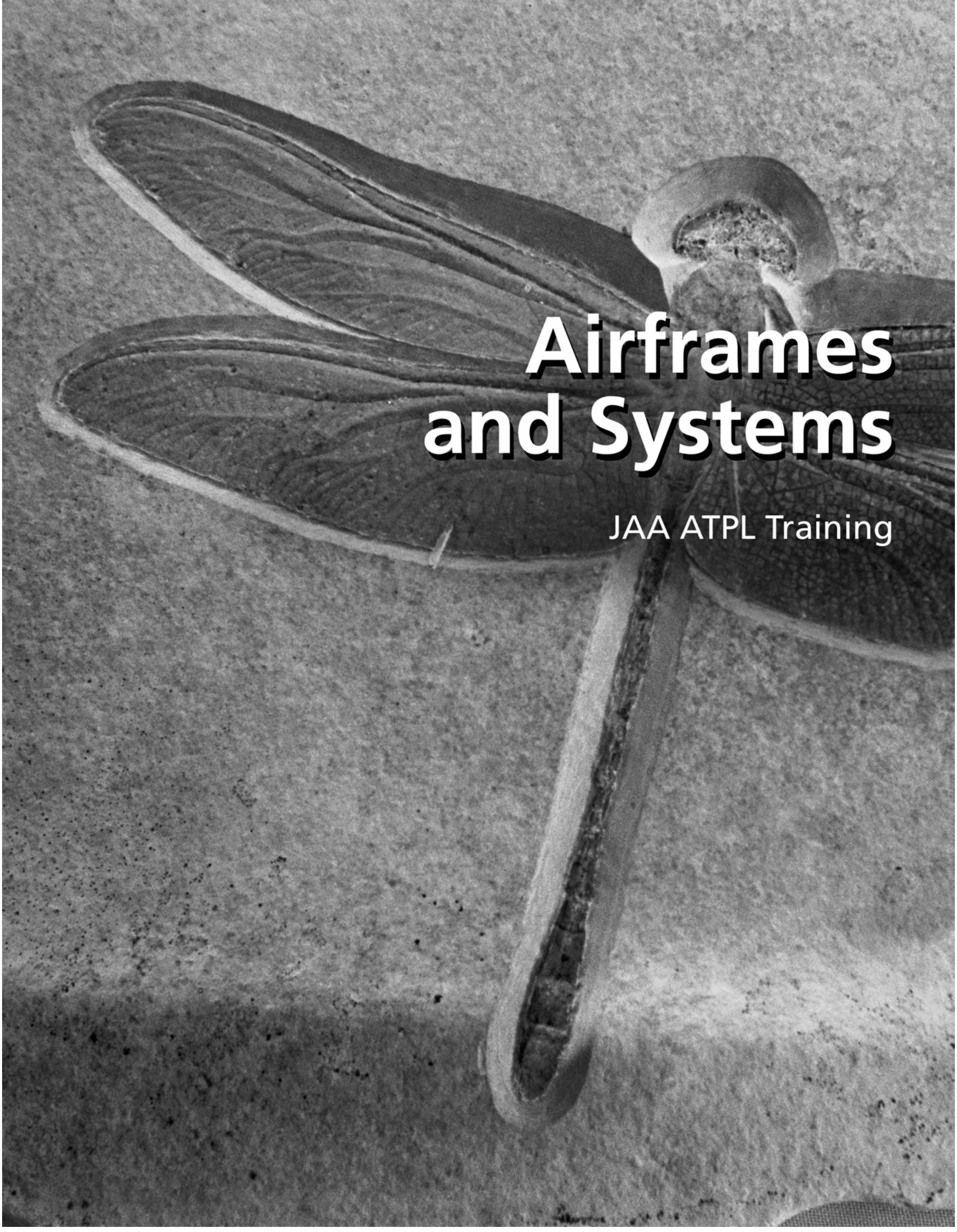
Airframes and Systems

JAA ATPL Training



This book has been written and published to assist students enrolled in an approved JAA Air Transport Pilot Licence (ATPL) course in preparation for the JAA ATPL theoretical knowledge examinations. Nothing in the content of this book is to be interpreted as constituting instruction or advice relating to practical flying.

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PREFACE

As the world moves toward a single standard for international pilot licensing, many nations have adopted the syllabi and regulations of the “Joint Aviation Requirements-Flight Crew Licensing" (JAR-FCL), the licensing agency of the Joint Aviation Authorities (JAA).

Though training and licensing requirements of individual national aviation authorities are similar in content and scope to the JAA curriculum, individuals who wish to train for JAA licences need access to study materials which have been specifically designed to meet the requirements of the JAA licensing system. The volumes in this series aim to cover the subject matter tested in the JAA ATPL ground examinations as set forth in the ATPL training syllabus, contained in the JAA publication, “JAR-FCL 1 (Aeroplanes)”.

The JAA regulations specify that all those who wish to obtain a JAA ATPL must study with a flying training organisation (FTO) which has been granted approval by a JAA-authorised national aviation authority to deliver JAA ATPL training. While the formal responsibility to prepare you for both the skill tests and the ground examinations lies with the FTO, these Jeppesen manuals will provide a comprehensive and necessary background for your formal training.

Jeppesen is acknowledged as the world's leading supplier of flight information services, and provides a full range of print and electronic flight information services, including navigation data, computerised flight planning, aviation software products, aviation weather services, maintenance information, and pilot training systems and supplies. Jeppesen counts among its customer base all US airlines and the majority of international airlines worldwide. It also serves the large general and business aviation markets. These manuals enable you to draw on Jeppesen’s vast experience as an acknowledged expert in the development and publication of pilot training materials.

We at Jeppesen wish you success in your flying and training, and we are confident that your study of these manuals will be of great value in preparing for the JAA ATPL ground examinations.

The next three pages contain a list and content description of all the volumes in the ATPL series.

ATPL Series

Meteorology (JAR Ref 050)

- The Atmosphere
- Wind
- Thermodynamics
- Clouds and Fog
- Precipitation
- Air Masses and Fronts
- Pressure System
- Climatology
- Flight Hazards
- Meteorological Information

General Navigation (JAR Ref 061)

- Basics of Navigation
- Magnetism
- Compasses
- Charts
- Dead Reckoning Navigation
- In-Flight Navigation
- Inertial Navigation Systems

Radio Navigation (JAR Ref 062)

- Radio Aids
- Self-contained and External-Referenced Navigation Systems
- Basic Radar Principles
- Area Navigation Systems
- Basic Radio Propagation Theory

Airframes and Systems (JAR Ref 021 01)

- Fuselage
- Windows
- Wings
- Stabilising Surfaces
- Landing Gear
- Flight Controls
- Hydraulics
- Pneumatic Systems
- Air Conditioning System
- Pressurisation
- De-Ice / Anti-Ice Systems
- Fuel Systems

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- Turbine Engine
- Engine Construction
- Engine Systems
- Auxiliary Power Unit (APU)

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- Direct Current
- Alternating Current
- Batteries
- Magnetism
- Generator / Alternator
- Semiconductors
- Circuits

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- Flight Instruments
- Automatic Flight Control Systems
- Warning and Recording Equipment
- Powerplant and System Monitoring Instruments

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- Laws and Definitions
- Aerofoil Airflow
- Aeroplane Airflow
- Lift Coefficient
- Total Drag
- Ground Effect
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- C_{LMAX} Augmentation
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- Boundary Layer
- High Speed Flight
- Stability
- Flying Controls
- Adverse Weather Conditions
- Propellers
- Operating Limitations
- Flight Mechanics

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- Multi-Engine Aeroplanes – Not certified under JAR/FAR 25 (Performance Class B)
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- Limits
- Loading
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- ICAO ATC Flight Planning
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- Jeppesen Airway Manual
- Meteorological Messages
- Point of Equal Time
- Point of Safe Return
- Medium Range Jet Transport Planning

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- Annex 8 – Airworthiness of Aircraft
- Annex 7 – Aircraft Nationality and Registration Marks
- Annex 1 – Licensing
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- Air Operations Certificate
- Flight Operations
- Aerodrome Operating Minima
- Low Visibility Operations
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- Transoceanic and Polar Flight

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- Definitions
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Chapter 1
Introduction to Structures

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We would like to thank and acknowledge:

For Diagram 1.9 — The estate of Mr. A. Whitlock for the kind permission to reproduce and use this drawing from his book “Behind the Cockpit Door”

INTRODUCTION

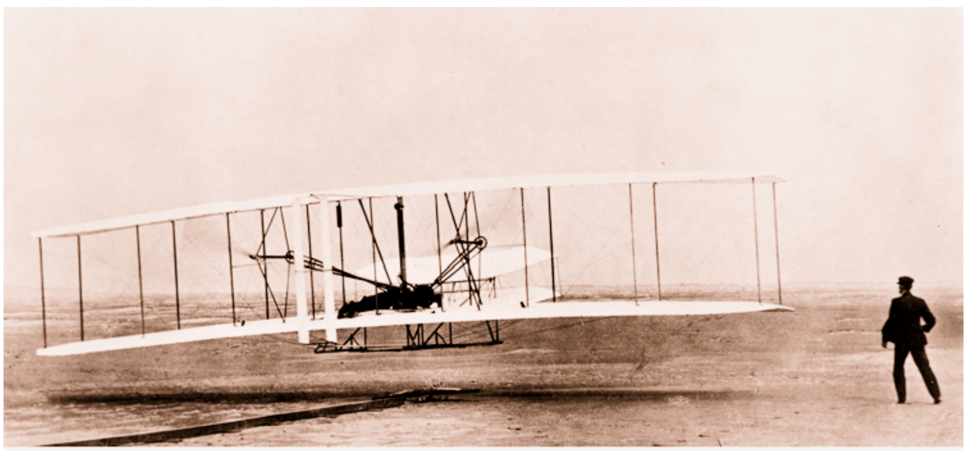


Diagram 1.1 Wright Flyer

Since the advent of powered flight, aircraft designers and manufacturers have made use of available technology, striving to build aircraft that are faster and can carry a greater load a greater distance. Competition to produce the best machine has fuelled research and development. Dramatic leaps in aviation technology were made in the First, Second, and Cold Wars of the Twentieth century.

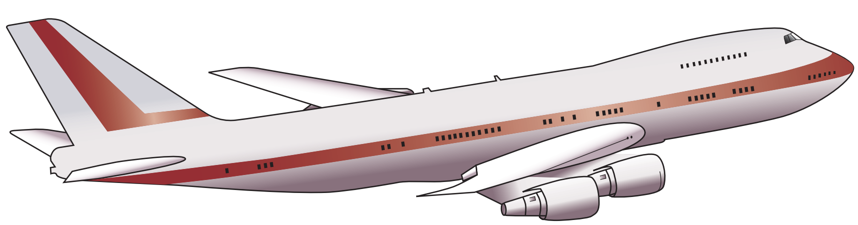


Diagram 1.2 Boeing 747 Air Transport Aircraft

Every aeroplane makes use of a horizontal aerofoil surface, **the wing**, to produce the lifting force to support its mass. Lift produced by the wing can also create a pitching moment. To balance this pitching moment, a second horizontal aerofoil is used. It can be placed in front of the main wing as a canard, as per diagram 1.1, or behind the wing as a tailplane, as per diagram 1.2. To give directional stability to the aircraft, a vertical aerofoil(s) fin is mounted behind the wing, usually on the fuselage.

Delta-wing aircraft

The design of a delta wing and its surface area are such that a horizontal tailplane is not required, however, a vertical fin is required.

V-tail aircraft

For the V-tailed aircraft, the designers have mounted the tailplane in the form of a V. This provides the balancing force for the wings and provides the directional stability of the fin, so a vertical fin is not required.

Before looking at the components that make up an aircraft's structure, this chapter describes:

- Physical strains that structures are subject to
- Loads that operating an aircraft applies to its structure
- Design philosophy that has evolved to ensure that aircraft are safe and that these loads do not cause catastrophic failure
- Materials used in manufacture

STRESSES AND STRAINS THAT AFFECT AIRCRAFT STRUCTURES

It has become fairly standard practice to interchange the words stress and strain. However, **strain** is the term used when an external force is applied to a structure that acts to deform it, while **stress** is the internal force within the structure that opposes the external force being applied. Listed below are the major stresses or strains that affect structures:

TENSION

In the diagram, the tensile load strain applied is trying to pull the rod and elongate it. The tensile stress in the rod is resisting this strain.

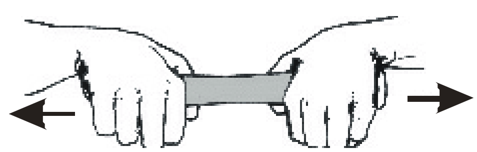


Diagram 1.3a Tension

COMPRESSION

In the diagram, the compressive load strain applied is trying to squeeze the rod and shorten its length. The compression stress within the rod is resisting this strain.

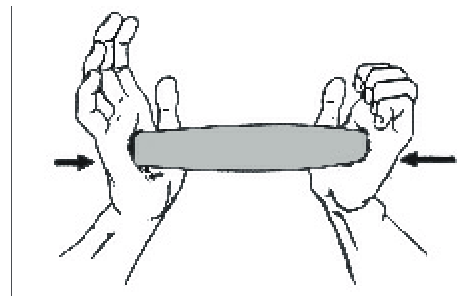


Diagram 1.3b Compression

TORSION

In the diagram, the torque load strain applied is trying to twist the rod along its length. The torsional stress within the rod is resisting this strain.

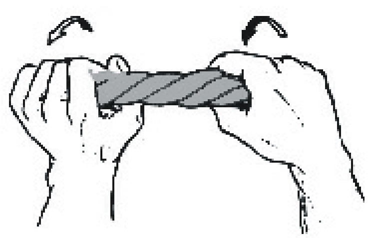


Diagram 1.3c Torsion

SHEAR

In the diagram, the fastening holding the two plates together is subject to cutting action as the two plates are subject to a tensile load. The resistance to the two opposite tensile loads is shear stress.

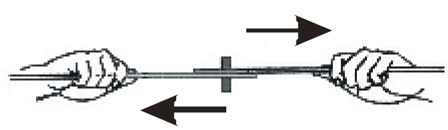


Diagram 1.3d Shear

COMBINATION STRESS

If a structure is bent, as per the diagram, it is subject to several different loads. The external radius of the bend is placed under tension, while the internal radius is placed under compression. A shear line is produced, shown by the red line, where the tension and compression act against each other.

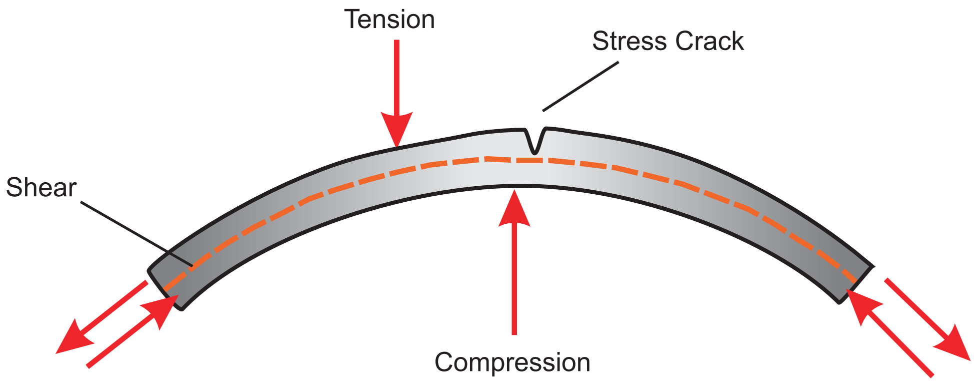


Diagram 1.3e Compound

HOOP STRESS

This is also referred to as radial stress, and is the stress raised in a container when it is filled, where the contents act to expand the container. For example, a balloon has hoop stress when inflated. Like a balloon, when an aircraft is pressurised, the pressure hull expands.

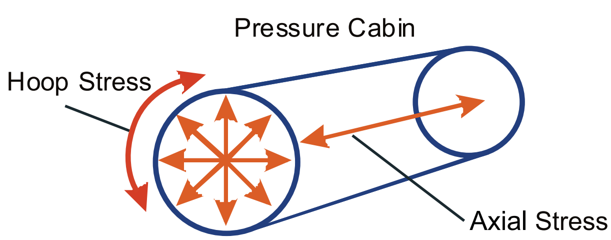


Diagram 1.3f Hoop and Axial Stress

AXIAL STRESS

Axial stress is also referred to as longitudinal stress. It occurs when the aircraft is pressurised and the pressure hull lengthens.

ELASTICITY OF MATERIAL

The materials that modern air transport aircraft are made from have a degree of elasticity. As with a rubber band when it is stretched and then released, it returns to its original size. If the rubber band is over stretched, it is beyond its elastic limits. When released, the rubber band does not return to its original size. This is **deformation**. In aircraft structures, deformation can take the form of bending, buckling, elongation, twisting, shearing, or cracking, which ultimately leads to fracture and creep in materials.

Where a material is subject to a load that is within its elastic limit but for a protracted period, the material can deform (elongation). This is termed **creep.** In Diagram 1.4, the bar is subject to a tension load and has increased in length.

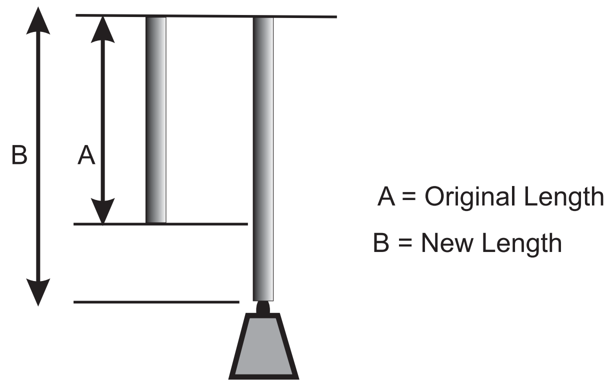


Diagram 1.4 Strain Ratio

The factors that affect creep are:

- Material type
- Load applied
- Duration of the load
- Temperature

STRAIN RATIO

When sufficient external force is applied to a structure so that it overcomes the structure's ability to withstand the force (e.g. external force greater than the structure's stress limitation), then the structure will deform due to **strain**. The amount by which the structure has deformed is found by comparing the original length with the deformed length. This is given as a ratio, which shows the magnitude of the strain.

SHOCK LOADS

When a structure has a sudden increase in the load that is being applied to it (e.g. by a bird strike directly on to the compressor blades of a jet engine), this is termed a shock load. A heavy landing also creates a shock load. When the resultant load applied to a structure is greater than the elastic property of the material it is made from, the affected structure(s) becomes permanently deformed.

FATIGUE IN MATERIALS

Fatigue is inevitable in materials that are subject to alternating loads (i.e. an old rubber band stretches and then fails before the normal load is applied). This is called fatigue failure. A structure subject to load reversals (tensile loads in the main), suffers fatigue failure more quickly than the same structure that has been subject to a continuous load. The effects of cyclical loads on structures are cumulative and reach a point where the structure fails even under normal loads.

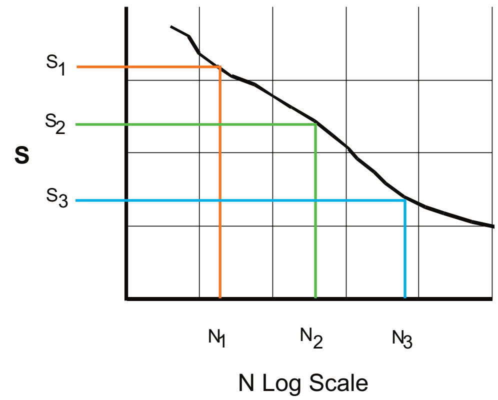


Diagram 1.5 Miner's Rule

Diagram 1.5 above is a graphical representation of Miner's rule. In this diagram, a log scale graph shows three levels of stress that a structure could be subject to. These are S1, S2, and S3. Along the lower axis are the number of cycles that the material can withstand before failure, shown by N1, N2, and N3.

To make an aircraft viable, the structure and the material that it is made from must be as light as possible. This results in the stress within the structure being high, especially as the Maximum Take-Off Mass (MTOM) may be up to two thirds of the structure failure load.

This combined with the varying **S** loads, which the aircraft is subject to throughout its operations cycles, have a cumulative effect on the fatigue within the structure. Consequently, all structures are given a fatigue life.

FATIGUE CRACKING

When a structure is subject to an average stress, scores, scratches, fastener holes, sharp edges, or sharp radial bends can build up local stress levels 2 or 3 times greater than the average. This leads to stress cracking as the stress level within the material tries to relieve itself. For example, if a sheet of paper is folded and a tear is started, further tension applied to the sheet across the tear line causes the tear to propagate. Each material has a critical crack length, CCL.

Up to the CCL, the material that the structure is made from maintains its integrity. When the CCL is reached, the crack propagates, leading to catastrophic failure of that structure, as the photograph of Aloha Airlines Boeing 737 (Diagram 1.6) graphically shows.

Diagram 1.6 Aloha B737 after a section of the cabin blew out when cracks in the aircraft's skin exceeded the material's CCL.

Photographer Unknown

FORCES THAT ACT ON STRUCTURES AND MATERIALS

In flight and on the ground, an aircraft is subject to several different forces, all of which act on the structure as a whole and individual components within the structure.

IN FLIGHT

LIFT

The lift generated by the wings in flight acts to pull the wings up and is evident by the flexing upward of the wing tips. This creates compression on the upper surfaces and tension in the lower surfaces. Due to lifting forces produced at the trailing edge, the wing also has a torsional force trying to rotate the leading edge nose down.

DRAG

The impedance of the airflow by components, such as fixed undercarriage legs, tries to bend the component backward out of the airflow.

MASS

The mass of the aircraft acts to pull the aircraft vertically downward.

WEIGHT

To calculate weight, multiply mass by acceleration due to gravity, which is assumed to be 9.81 m/s^2. (Some questions in the JAR examination specify 10 m/s^2 for acceleration due to gravity. If no value is given, use 9.81 m/s^2.)

Example:

A mass of 100 kg has a weight of .....? (gravitational constant of 10 m/s^2) = 1000 Newtons

A mass of 100 kg has a weight of .....? = 981 Newtons

When an aircraft is in straight and level unaccelerated flight, it is subject to **1g** (acceleration of 9.81 m/s^2). This is the same as when the aircraft is on the ground. However, as the aircraft turns or dives and pulls up or is displaced by a gust of wind, the aircraft feels a change in the gravitational force. A steeply banked turn increases **g**, while an up gust can reduce it. This alters the effective weight of the aircraft and all items within it.

ACCELERATION

Accelerating increases the drag on those components that are in the airflow. It also increases the momentum of the aircraft.

INERTIA

The structure and components of the aircraft want to continue in the original direction of travel when the aircraft turns, pulls up, dives, or levels out.

OTHER FACTORS FOR CONSIDERATION

AIRSPEED

At supersonic airspeeds, the friction of the air against the leading edges of the aircraft acts to heat them.

TEMPERATURE

A decrease in temperature makes many materials more brittle. An increase in temperature makes most materials more ductile.

ALTITUDE

An increase in altitude reduces the static ambient pressure acting against the aircraft. If the aircraft is pressurised, there is a differential acting across the skin that creates radial and axial load. Increasing the aircraft's altitude or increasing the cabin pressure, increases the differential and, therefore, the hoop and axial loads on the pressure hull.

ON THE GROUND

DURING LANDING

Momentum and inertia have a downward vertical component as well as a forward component as the aircraft touches down on the runway (e.g. the wings flexing downward on landing). As the runway is effectively in the way of the aircraft, an external force alters the aircraft's direction. This subjects the landing gear and its mounting structure to a compression load.

FRICTION

The friction created between the wheels and the ground (rolling friction) acts to hold the wheels back and, therefore, the lower leg of the undercarriage as the fuselage of the aircraft moves forward with its momentum. For a tricycle aircraft, the friction between the main gear wheels and the ground also acts to rotate the nose downward and if not controlled, the nose gear can be rotated into the ground at a greater rate than the structural limitation.

PRESSURISATION

If an aircraft lands with its cabin pressurised, the total load on the airframe is the sum of the landing load and the pressure load combined.

THRUST REVERSAL

When reverse thrust is selected, there is an angular change in direction of the thrust. This is 45° from normal thrust line. This retardation creates a load on the engine's mounting structures that have inertia from forward velocity.

BRAKING

Applying the main wheel brakes causes the nose to pitch down and the aircraft to slow. Items not fastened down continue to move forward due to momentum.

STATIONARY

When the aircraft is stationary, its undercarriage supports the total mass of the aircraft. Since it is stationary, the weight is its mass times 1g. When the aircraft starts to move, the increase in thrust is used to overcome inertia and accelerate the aircraft forward.

TAXIING

When the aircraft is taxiing over uneven surfaces, the wings flex up and down. The rolling friction between the ground and tyres also prevents the aircraft from turning and produces side loads on the gear legs during a turn.

ON TAKE-OFF

From a stationary start, the thrust of the engines has to overcome the inertia of the aircraft, the rolling friction until take-off, and the increase of drag as the speed increases.

DESIGN PHILOSOPHY

The design and construction of modern aircraft is controlled by the regulation detailed in JAR 23 for aircraft with a mass of 5700 kg and less, and in JAR 25 for air transport aircraft and aircraft of a mass greater than 5700 kg.

These regulations classify the aircraft's structure into three groups:

> **Primary structure**
> This is structure that is stressed. In the event it fails, the structural integrity of the aircraft would be compromised to such a point that the aircraft could suffer a catastrophic failure.
>
> **Secondary structure**
> This structure is stressed but to a lesser degree. In the event of a failure, the aircraft would not suffer a catastrophic failure, but could be limited in operation.
>
> **Tertiary structure**
> This structure is not stressed or nominally stressed and would not cause a catastrophic failure in the event it fails.

PARTS

Parts that make up a structure and systems are also categorised depending on the effect that their failure would have on a unit or system:

Critical parts must achieve and maintain a particularly high level of integrity if hazardous effects are not to occur at a rate in excess of "extremely remote". (See table on next page.)

The failing of a **major part** might adversely affect the operational integrity of the unit in which it is installed.

Below is a table compiled by the JAA on the probability of failure and the likely consequences:

Classification of failure condition	**JAR 25 Probability**	**Effect on aircraft and occupants**
Minor	Frequent	Probable Normal operation
Minor	Reasonably Probable 10^{-3}	Probable Operating Limitations Emergency Procedures
Major	Remote 10^{-5}	Improbable Significant reduction in safety margins Difficult for the crew to cope
Hazardous	Extremely Remote 10^{-7}	Improbable Large reductions in safety margins Crew extended because of workload Serious or fatal injuries to a small number of passengers
Catastrophic	Extremely Remote 10^{-9}	Extremely improbable Aircraft destroyed Multiple deaths

DESIGN LIMIT LOAD OR DLL

This is the maximum load that the aircraft designer or component manufacturer expects the airframe or component to be subject to in operation.

DESIGN ULTIMATE LOAD OR DUL

As it is feasible that an aircraft could experience loads in excess of the DLL, a further test is carried out where the minimum load applied to the structure must be 1.5 × DLL for three seconds. After the aircraft has been subject to this load, there may be permanent deformation of the aircraft's structure, but it must not collapse.

The difference between the DLL and the DUL is the **safety factor**. This is expressed as ratio of the DUL to DLL. In modern aircraft, the structural design and materials used far surpass the regulation on the safety factor.

CATASTROPHIC FAILURE

The regulations specify that the design and fabrication methods employed in the manufacture of an aircraft must show that during the operational life of the aircraft, there will be no catastrophic failures due to:

- Fatigue
- Corrosion
- Accidental damage

To comply with this requirement, designers must evaluate the materials and structures they intend to use and the loads the aircraft would be subject to during their operational life.

The result is the **safe life** philosophy.

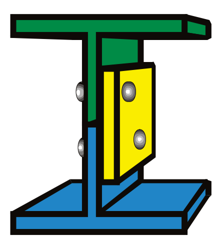
SAFE LIFE

The aircraft structure, as a whole, and components within the aircraft are given a safe life. This is based on one, several, or all of the following:

- Cumulative flying hours
- Landings
- Pressurisation cycles
- Calendar time

A structure's safe life is set so that it is removed from service before there is a possibility of it failing in operation. If in service there is degradation of the structure, this life can be reduced. Alternatively, the life of an aircraft or its components can be extended if the anticipated degradation does not occur. To achieve longevity of structure and comply with the regulations, designers make use of **fail-safe or damage-tolerant structures**.

FAIL-SAFE STRUCTURE



Fail-Safe Spar

Diagram 1.7 A Fail-Safe Design

To achieve a fail-safe structure, no one item within a structure takes the entire load. It is shared by several components, thus there are multiple load paths. This redundancy of items allows the structure to continue operating normally up to the static ultimate for a limited period. Damage (e.g. cracks) would have to be found before or during the next periodic servicing according to the design criteria. Furthermore, damage should not have propagated by much before being identified. In diagram 1.7 above, a fail-safe spar has been manufactured from several components so that if one should fail, the others are still operable.

However, the design incorporates extra structural items. These have a weight penalty, and this has led to a change toward designing damage-tolerant structures.

DAMAGE-TOLERANT STRUCTURE

Damage-tolerant structure is designed to spread the loads over a larger area than that of the fail-safe design. This load-spreading concept allows a lighter structure to be manufactured, which is tolerant of accidental damage, fatigue, and corrosion. Any damage to the structure should be found and repaired during normal inspection before there is degradation of the structure's integrity.

SERVICING CYCLE

Aircraft inspections are based on flying hour cycles, and a calendar date is used as a backstop. The calendar date is used to ensure that servicing is carried out at set periods, should the aircraft not have flown the requisite hours within that time period. Aircraft normally have a major servicing every five years.

AIRCRAFT CONSTRUCTION MATERIALS

METAL

Steel is a collective term as there are many different materials and alloys. Steels and alloys of steel are used in areas where strength is required and which are subject to high loading, wear, and high temperatures. Some of the types of steel and alloys used in aircraft construction are listed below:

High tensile steels serve to make fittings and fixings that are subject to tensile loads. Some high strength steels have poor crack properties, and loads in excess of the material's stress level results in cracks radiating along the stress path rather than deformation through elongation, bending, or buckling.

Stainless steels are expensive but have high strength-to-weight ratio and can withstand high temperatures. In many cases, stainless steel is corrosion resistant. These properties enable designers to use stainless steel in engine bays, hot air ducts, and leading edges.

Titanium alloys are extensively used in modern aircraft manufacture as they are resistant to high temperature, up to 400°C, have excellent corrosion resistant properties, and excellent strength-to-weight ratio. Titanium alloys are used to manufacture bulkheads, skins, structural castings, fire bulkheads, and in gas turbine engines.

Nickel Alloys are only used where the temperature environment is high and creep resistance is a major consideration. Their uses are normally restricted to the hotter parts of the gas turbine engine and its associated installation.

Light Alloy is the collective term given to alloys of aluminium and magnesium. Some of these materials have strengths equal to that of some steels but with about a third of the density.

Pure Aluminium, although it has corrosion resistant properties, is very soft and not very strong.

Aluminium Alloy is aluminium combined with different materials such as copper, chrome, zinc, manganese, silicon, magnesium, or a mixture of them. The resulting alloys have varying properties. When aluminium is alloyed with zinc (Al-Zn) the resulting alloy has a high strength-to-weight ratio and is used for static load conditions.

When aluminium is alloyed with **4%** copper (Al-Cu) the resulting alloy has a lower strength-to-weight ratio, a good fatigue resistance and is easier to use in manufacturing since it is softer than the Al-Zn alloys. This material is often called **Duralumin** and is extensively used in the production of aircraft.

Aluminium alloyed with magnesium, Al-Mg, results in a low strength alloy that can be welded.

Sheet aluminium alloys are coated with pure aluminium to improve the corrosion resistance. This is termed **Alclad**.

One of the properties of aluminum alloys is that their tensile strength increases as the temperature decreases. Another property is that the material's ability to elongate increases with a decrease in temperature. These properties, amongst others, mean that aluminum alloy is an excellent material from which to manufacture many aircraft parts.

Magnesium Alloy has a very good strength-to-weight ratio (aluminium is 1.5 times heavier), but it has very poor resistance to corrosion (avoid salt water). It also has very poor elastic properties, and both sheet metal and forging are prone to cracking when subject to vibration

Magnesium is a mono fuel which provides its own oxygen when burning. For this reason, components manufactured from magnesium need to be smothered should they catch fire. Do not bring a source of ignition in contact with magnesium.

COMPOSITE MATERIALS

Composite materials are manufactured from reinforcing fibres embedded in a bonding resin. As the materials can be moulded, they are described as **plastic**.

The main reinforcing materials are:

- Glass GFRP
- Carbon (graphite) CFRP
- Boron
- Aramid, known as **Kevlar**, KFRP, a synthetic material
- Lithium is being evaluated as a material

Some bonding materials are:

- Epoxy resin
- PTFE

ADVANTAGES OF COMPOSITE MATERIALS

The advantages of using composite material over alloys are:

- The ability to arrange the fibres to obtain directional properties consistent with the load
- The ability to make complex shapes, since the material is not homogeneous
- Weight savings
- Resistance to corrosion
- High specific strength
- High specific stiffness

DISADVANTAGES OF COMPOSITE MATERIALS

The disadvantages of composite materials are:

- They are quickly eroded by hail, sand, etc, so leading edges must be sheathed.
- They are difficult to repair.
- They can absorb moisture if the material is not correctly sealed.

Fibreglass, or Glass Fibre Reinforced Plastic (GFRP), has been used to produce the fuselage and wings of light aircraft. For air transport aircraft, it is normally used in the production of low pressure, low temperature ducts, and non-load bearing fairing panels.

Other composite materials that are lighter and stronger are replacing GFRP.

Graphite is stronger and stiffer than fibreglass but corrodes aluminium alloys if it comes into direct contact with them.

Kevlar, KFRP, has high tensile strength combined with low density. Items manufactured from Kevlar are tough and resist impact damage. Due to its qualities, an increasing number of aircraft components are being manufactured from Kevlar.

One of the properties of composite materials is that under load, they do not have the same fatigue profile as the alloys. Metals retain their structural strength under fatigue load conditions. However, they fail when the load reaches a critical point, whereas the composites have a gradual deterioration of their properties. Stress loads up to 80% of the ultimate load for the composite material are not normally considered in fatigue equations.

ELECTRICAL BONDING

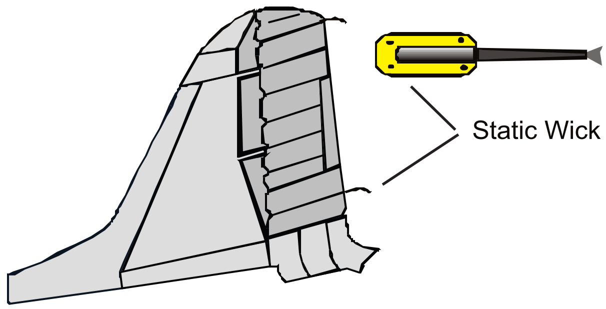


Diagram 1.8 Static Wick

To prevent dissimilar and electrolytic corrosion, rubberised sealant and paint primers separate aircraft skins and components from each other. To ensure that each part of the aircraft's structure has the same electrical potential, the components are bonded to each other.

This makes the structure a Faraday's cage that protects the occupants and systems from lightning strikes. Static wicks fitted to the trailing edges of the wings and empennage allow the static charges built up through flight to dissipate.

LIGHTNING STRIKES

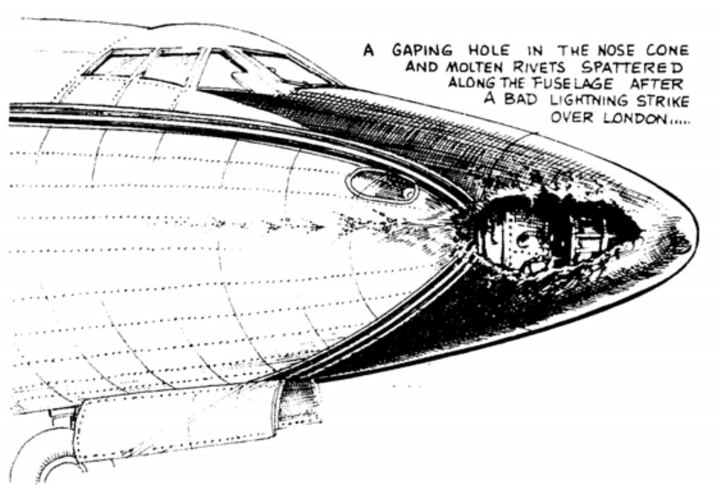


Diagram 1.9 After a Bad Lightning Strike

Lightning strikes on alloy aircraft normally track along the surface of the structure doing little or local damage to the skins due to the bonding. However, lightning strikes on composite materials can result in the lightning exploding the composite material at its exit point. If there is moisture present within the layers of composite material due to poor sealing, it can be turned to superheated steam, which delaminates the structure. If lightning strikes alloy materials that are not efficiently bonded, the intense heat can locally melt the materials. See diagram 1.9.

ACKNOWLEDGEMENT

We would like to thank and acknowledge:

For diagrams

2.5	Key Publications Ltd. for the right to use and adapt a cutaway diagram of a 737-300 aircraft drawn by Mike Bodrocke
2.18	Airbus Industry

INTRODUCTION

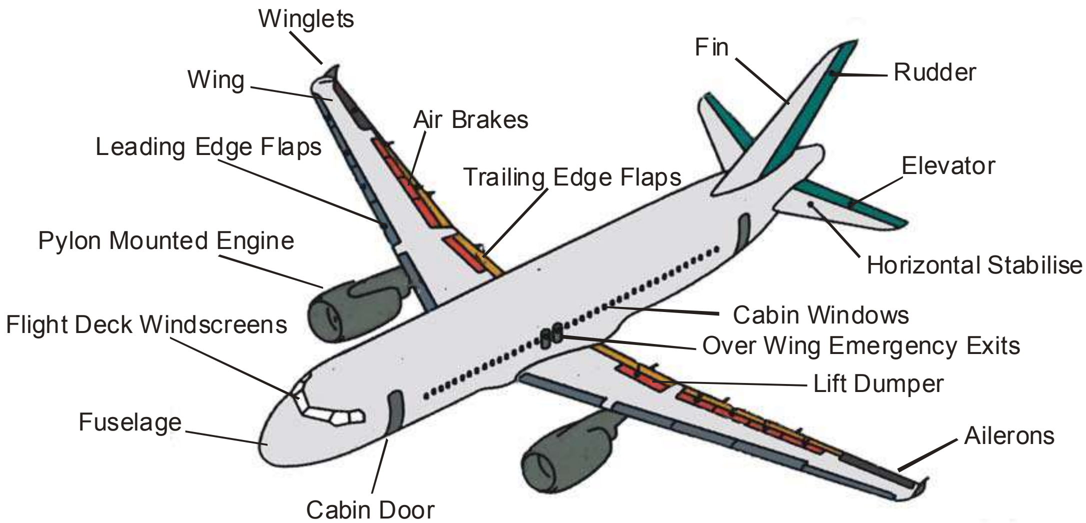


Diagram 2.1 An Air Transport Aircraft Structural Components

The fuselage forms the main body of the aircraft to which the wings, tail plane, canards, vertical fin, and engine (in the case of a single-engine aircraft) are attached. In some aircraft designs, the engines and landing gear also attach directly to the fuselage structure.

In modern Civil Air Transport Aircraft, as shown in diagram 2.1, the fuselage takes the form of a tube, which houses the flight deck, passenger cabin, freight holds, and the majority of the equipment required to operate the aircraft.

For passenger aircraft that fly above 10 000 ft, the fuselage also forms a pressure hull so that a cabin altitude of 8000 ft can be maintained throughout normal flight.

FRAME CONSTRUCTION

Diagram 2.2 Frame Structure Fuselage

In frame and fabric construction, the frame takes the loads. The fabric is coated with **dope,** which tautens and seals it, making it waterproof and preventing the air from flowing through it.

Frame construction and fabric covering is now only used in the construction of some light aircraft. It has largely been superseded by composite material such as fibreglass and Kevlar in this field.

In this type of construction, a rigid box work frame is made up of a series of vertical, horizontal, diagonal, and longitudinal tubular steel pipes that are welded together. The vertical and horizontal pipes act to position the longitudinal pipes in relation to each other and tie them together. Diagonal pipes brace the structure.

This design produces a square fuselage. To improve its profile, the designer would have to add formers and pipes to the structure, which would increase the overall mass. The framework is subject to all the flight loads and the ground loads, so it must withstand both compression and tensile loads.

MONOCOQUE STRUCTURES

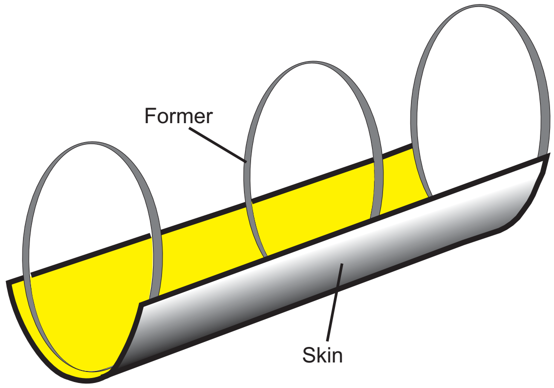


Diagram 2.3 Monocoque Structure

In a monocoque structure, the skin takes all the loads placed on the structure and the shape of the structure provides its strength and rigidity. Aluminium beverage cans and eggs are excellent examples. For an aircraft with a monocoque structure, the skin might be attached to formers to give it its basic shape, as per diagram 2.3, but the skin would carry all the flight and ground loads. As the strength of the whole fuselage is in the skin, any damage that deforms, dents, perforates, or creases it reduces its ability to carry these loads and can lead to structural failure.

Monocoque structure is not used in the production of current alloy-skinned aircraft as there are complications in the design of access hatches, doors, and their mountings, which impart extra stress to the skin. The stress of air loads and the need to carry a viable payload would require unacceptably thick and heavy skins, so most aircraft employ a semi-monocoque design.

SEMI-MONOCOQUE STRUCTURES

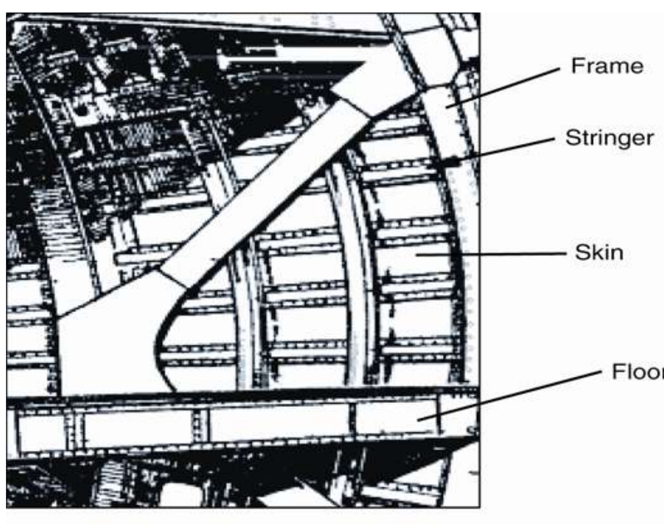


Diagram 2.4 Semi Monocoque structure

In the semi-monocoque design (refer to diagram 2.4), the loads imposed on the skin are shared by a series of frames, stringers, and formers that are attached to it. Frames act to strengthen the fuselage and spread the load. Stringers are lighter longitudinal members that reinforce the skin, and formers are used to maintain the skin's profile between frames.

This design is more tolerant of damage to the skin as the other structure shares the load path for the fuselage. This design overcomes the disadvantages of the monocoque structure while maintaining a good strength-to-weight ratio.

However, there is a requirement to meet the **fail-safe** philosophy for air transport aircraft. In this context, no one single component failure should cause the loss of the aeroplane. This is termed **redundancy**. Fail-safe structures need parallel load paths, and individual structural members need to be stronger so that if damage occurs (other than catastrophic), the remaining structure can bear the load, although the aircraft is impaired. The reinforced shell achieves this for the fuselage.

REINFORCED SHELL STRUCTURE

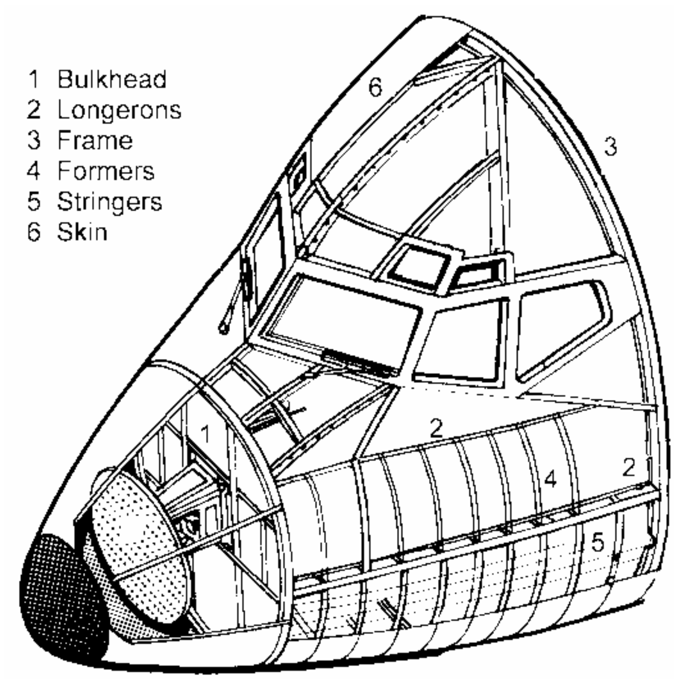


Diagram 2.5 Reinforced Shell

The fuselage structure consists of:

- Bulkheads
- Longerons
- Frames
- Formers
- Stringers
- Skin

In designing the fuselage, the designer reinforces the areas where there are going to be apertures for windows, doors, and access hatches. This reinforcing, in the form of a frame around the aperture, is to take the loads that would have been carried by the skin that has been cut away and heavier longitudinal members called longerons.

The fuselage frames are stiffened to transfer and share the loads and act to prevent crack propagation. Refer to Diagram 2.5.

RIVETING, BONDING, MILLING, AND ETCHING

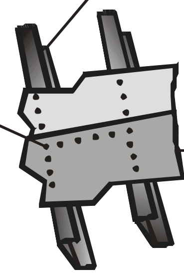


Diagram 2.6 Riveted Skin

Skins, frames, formers, etc, must be joined together using rivets or bonding. Riveting is time consuming and has the disadvantage of requiring a row of holes drilled through the skin and the frame to which it is attached. The weight of the material removed is less than the weight of the rivets used, see diagram 2.6. The modern method is to bond the two metallic parts together using an adhesive.

In older designs, manufactured aircraft structures were made up of individual parts, as listed above, which were riveted and later bonded together. Technological progress has allowed large billets of alloy to be machined (milled) to remove unnecessary material. At the same time, this process has allowed the retention of material to give the structure strength and rigidity. Therefore, it is not necessary to attach frames, stringers, and longerons to the sheet of alloy making the skin.

Chemical etching of the billet has further refined this process. The improvement of adhesives' shear strength has allowed large milled or etched structures to be bonded together, improving the strength-to-weight ratio and reducing the manufacturing costs.

PRESSURE BULKHEAD

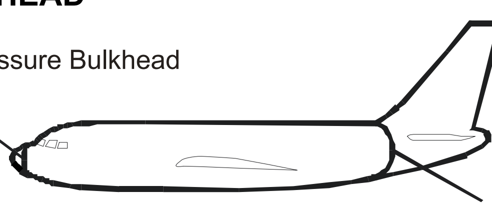


Diagram 2.7 Pressure Bulkheads

In pressurised aircraft, the pressure cabin terminates in front and rear pressure bulkheads, as per diagram 2.7. In large aircraft, the rear pressure bulkheads are usually domed to prevent the internal pressure from deforming them. In some designs, the passenger cabin floor is part of the pressure hull, and the void below is used as an un-pressurised cargo hold.

In modern designs, the passenger (pax) cabin and the hold are equally pressurised. The pressure load is equal around the circumference of the skin, and the floor panels do not have to be designed and manufactured to withstand the pressure differential.

CABIN FLOORS

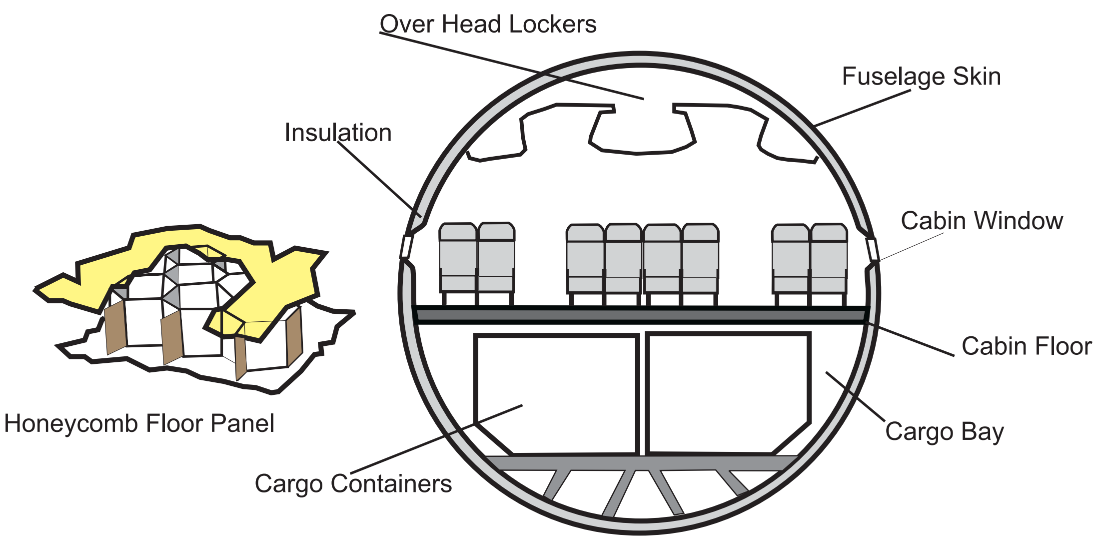


Diagram 2.8 Cross Section of Cabin

The cabin floor is manufactured from a series of panels attached to supporting beams and cross members. Aircraft that have pressurised passenger cabins and hold are frequently manufactured using a honeycomb structure, as shown in Diagram 2.8. This allows a lightweight structure to withstand compression loads and have rigidity without incurring a weight penalty. Honeycomb structures can be used in the manufacture of other panels and skins.

BLOW OUT BUNGS

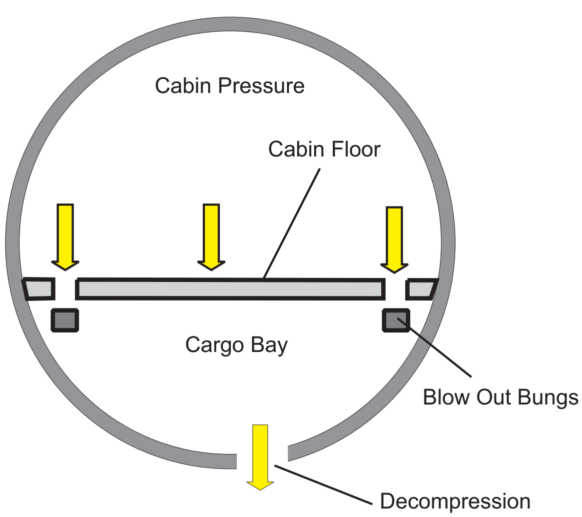


Diagram 2.9 Blow Out Bungs

Where a floor separates a pressurised hold from the pressure cabin (as per diagram 2.9), holes are let into the floor panels and plastic blow out bungs are press fitted into them. In the event that either compartment loses pressure, the bungs blow out due to the differential pressure across the floor. This allows the compartment pressures to equalise before the floor deforms with the possibility of injuring passengers, jamming flying control cables, rods, etc.

WINDOWS

FLIGHT DECK WINDOWS

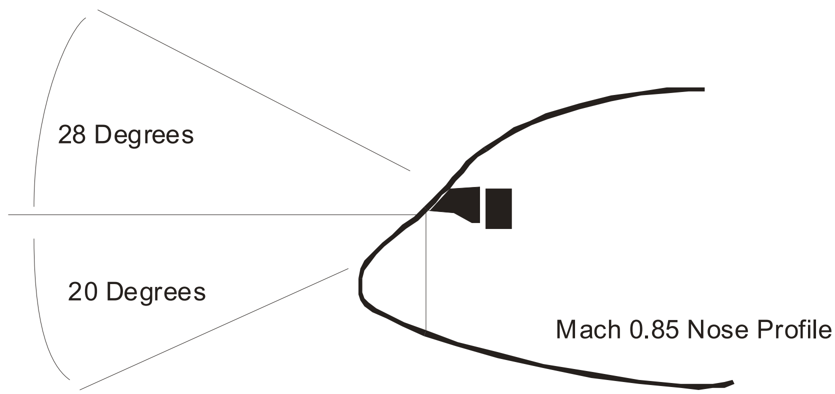


Diagram 2.10 Pilots vertical angles of vision from a semi-stepped nose of a modern airliner

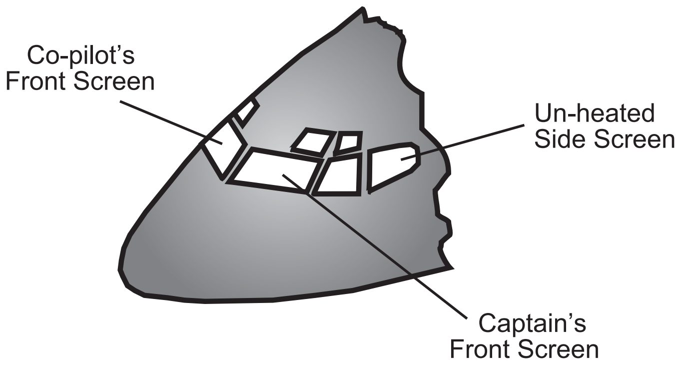
Modern subsonic and transonic air transport aircraft use a **stepped nose** profile; as shown in diagram 2.10. This allows the:

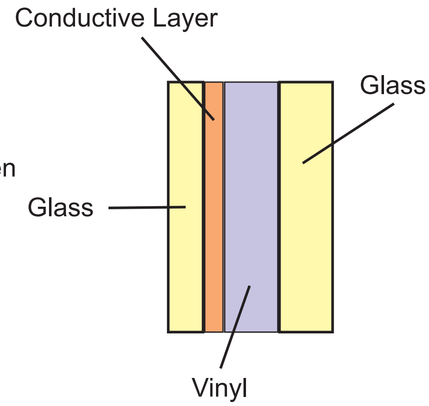
- Nose to be aerodynamically profiled for the aircraft's maximum operating speed
- Windscreens to be located in a position that is optically acceptable
- The pilot required vision for both ground and flight operations
- Reduction in the physical size of the windscreens

Angling the flight deck windscreens backward also allows them to shed rain and lessens the effect of a bird strike. While the structure surrounding the flight deck window apertures is strengthened, the amount of strengthening is less than that of the older bodyline designs, such as the Boeing Stratocruiser of 1945.

For pressurised aircraft, the stepped nose design with its reduced glazing area has the advantage of reducing the pressure loads applied to the windows. Therefore, the fuselage need not be further strengthened.

For example, every square foot of glazing in an aircraft pressurised to 9 psi above ambient is subject to a load of 1296 lb.





Front Screens

Diagram 2.11 Flight Deck Windows

Apart from optical qualities, air transport aircraft flight deck windows must be designed to withstand the pressure differential from cabin to ambient, the dynamic pressure of flight, and the changes in temperature. They must also be resistant to impact, such as bird strikes.

The requirements of the JARs for any flight deck windscreens or transparent panel that is inclined at 15° or more to the longitudinal axis of the aircraft is that they must be able to withstand the impact created by a 4 lb bird when the aircraft is flying at its designed cruising speed [V_c] at sea level, or the impact of the same bird at 8000 ft when the aircraft is travelling at 85% of its V_c, whichever is the more critical.

Laminated windows for air transport aircraft are manufactured from layers of soda lime glass, polyvinyl butyral (PVB), or vinyl materials of equivalent or greater strength. They are bonded together to form a one-piece laminated screen. The thin outer layer of glass is chemically hardened to increase its resistance to abrasion (see diagram 2.11).

The forward facing screens in all air transport aircraft are not only heated for de-icing purposes but also to ensure that the material does not become brittle as the temperature decreases, thus retaining flexibility in the event of a bird strike.

In the event of a bird strike, the outer layer might crack, but the bonding and laminations provide the strength to keep the window intact. If the window starts to delaminate, the stretching of the vinyl layers results in a milky white appearance in the affected areas. Ingress of moisture between the layers of laminate causes the affected area to take on a milky white appearance.

DIRECT VISION WINDOW

To comply with the regulations on maintaining clear vision in the event of a failure of the demisting system, one of the side panel windows on each side can be removed from within the flight deck to allow clear view. This is referred to as the direct vision window. If the openings are large enough, these can also be used as flight crew emergency exits. (See emergency equipment.)

PASSENGER CABIN WINDOWS

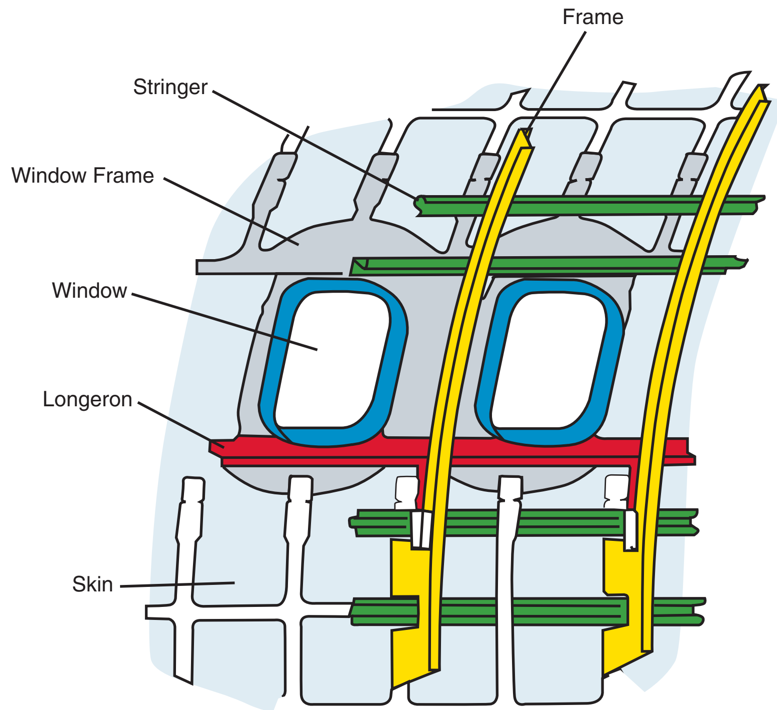


Diagram 2.11A Cabin Window Belt Structure

The cabin windows form part of the pressure shell of the fuselage. Diagram 2.11A shows the surrounding structure strengthened to form window frames and spread the loads. The windows are fitted from the inside. Since they are larger than the apertures, when the aircraft is pressurised, the pressure acts to hold the windows more tightly onto the frames.

To ensure that the windows are fail safe, they can be produced as two separate sheets of material bonded together with an air gap in between. The air gap is kept at cabin pressure using dried cabin air. This ensures that the windows do not fog up and that if passengers rest their heads against a window, they will not freeze onto the pane. If the external panel fails, the internal pane takes the pressure load.

AIRCRAFT DOORS CERTIFIED UNDER JAR 23 AND 25

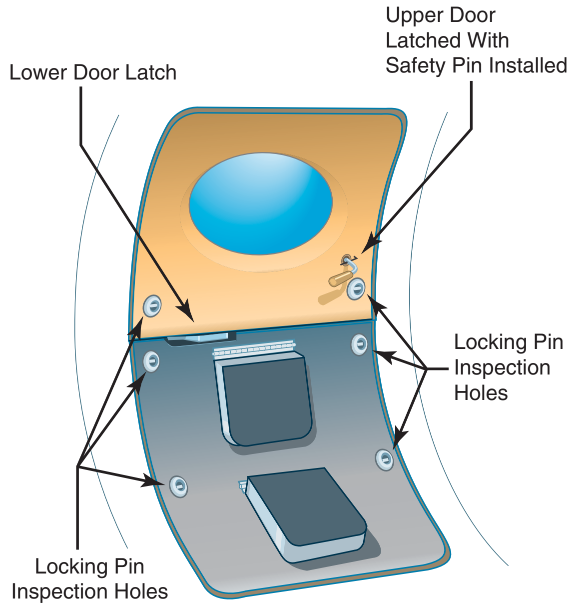


Diagram 2.12 Cabin Door

The size and classification of emergency exit doors and hatches is given in Emergency Equipment Doors.

The common door system for larger pressurised aircraft is the plug type door. These close from inside the aircraft and have locking pins that engage in the doorframe. Cabin air pressure, which is greater than ambient pressure, pushes the door more tightly onto its frame. When opened, these doors pull back inward before sliding upward or turning sideways to stow inside the cabin or tilt and move outside the cabin to give maximum room through the doorway.

The requirements for air transport aircraft doors are:

- Each passenger entry door in the side of the fuselage must qualify as a Type A, Type 1, or Type II passenger emergency exit.
- There must be a means of locking and safe-guarding passenger doors, cargo doors, and hatches against inadvertent operation in flight.
- The passenger door and those hatches that are denoted as emergency exits must be reasonably free from jamming after an emergency landing.
- Externally there must be a visual indication that doors and hatches are secured and correctly locked. This normally takes the form of a handle that stows flush with the skin only when the door or hatch is correctly locked.
- Internally there must be a visual indication that doors and hatches are secured and correctly locked. This normally takes the form of a warning light that illuminates on the Crew Warning Panel if any hatch or door is not closed and locked.
- The door must open externally as well as internally and the handles must fit flush to the skin.
- In the event of an emergency landing, operation of the external handle must be able to unlock and open the door.

- If the exit door is of the plug type that opens inward and stows inward, it must be able to be opened when people are crowding around it, or there must be a method of stopping people from crowding it.
- Pressurised aircraft external doors into the pressure cabin must have a means of preventing the aircraft from being pressurised if they are not closed and correctly secured.
- Passenger doors must not be located near the propeller disc or any other hazard.

SEALING

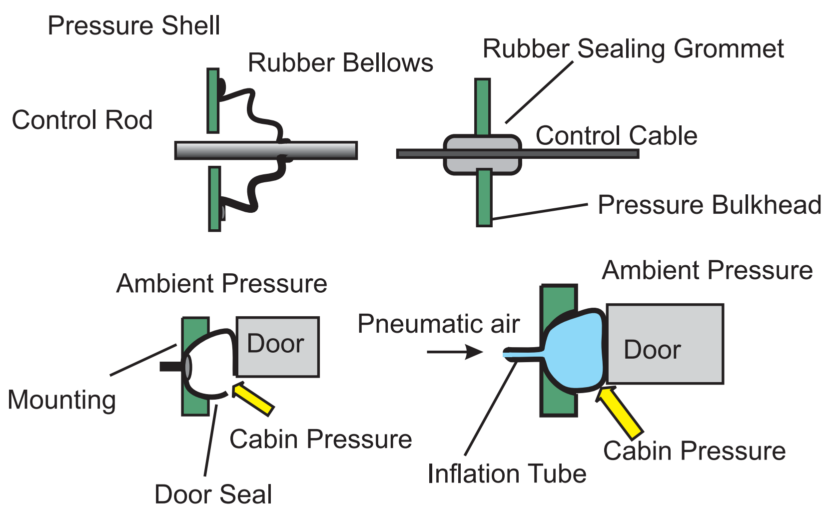


Diagram 2.13 Methods of Sealing

For pressurised aircraft, stopping cabin air from leaking out is very important, as this air has been conditioned, which takes engine power (cost). Therefore, every item that passes through a hole in the pressure cabin such as cabin doors, control rods, pipes, ducts, and cables must be sealed to prevent as much air loss as possible. Methods of sealing vary depending on the situation; see the section on pressurisation for details.

SEAT MOUNTING SUPPORT STRUCTURE

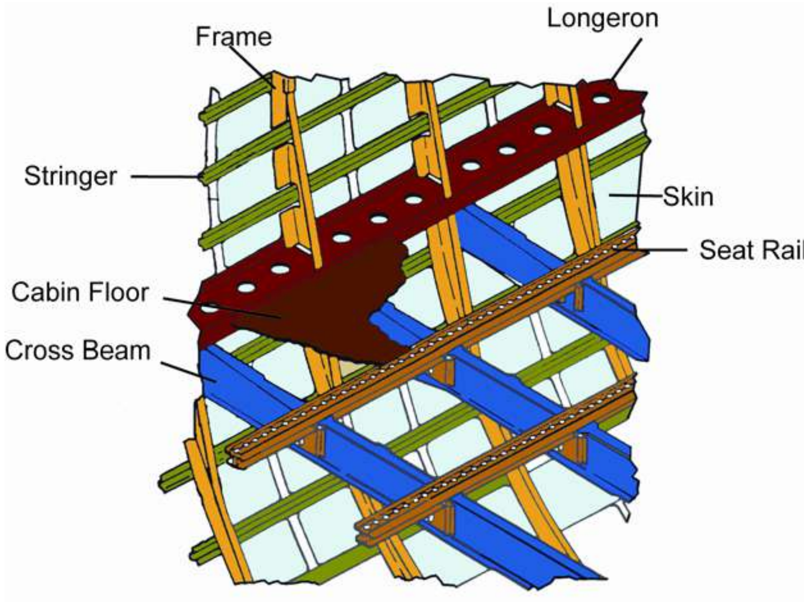


Diagram 2.14 Cabin Floor and Seat Supporting Structure

Seats and their associated mountings must withstand the ultimate loads as listed below. Therefore, the seat mounting rails are attached to the crossbeams beneath the floor panels, and the mounting rails are classified as primary structure.

For example, the following limits come from the JARs:

- Upward, 3.0g
- Forward, 9.0g
- Sideward, 3.0g on the airframe and 4.0g on the seats and their attachments
- Downward, 6.0g
- Rearward, 1.5g

SPAR ATTACHMENT

The strongest part of the fuselage is where the wings are attached, as this structure is the point where all the flight loads will be felt. This area is covered in the following chapter.

FUSELAGE SHAPE

The type of undercarriage and power plant has an effect on the external shape of the fuselage. The best designs have smooth contours and gradual changes in cross section so that the airflow is smooth, thus creating the minimum disturbance possible and reducing drag.

Diagram 2.15 Dakota

For early transport aircraft, this streamlined tapering design, of which the DC3 is one of the best known examples, remained the standard design, see diagram 2.15. To carry larger payloads, fuselages had to become more cylindrical and more power was required to overcome the increased mass and drag. This was achieved by fitting larger engines or more engines. Compare the profile of the DC 3 in diagram 2.15 to that of the DC 6 in diagram 2.16.

Diagram 2.16 DC 6

However, once a piston engine has reached a certain power output and size, to gain an increase in power, the engine would have to increase in both size and mass, which enters the law of diminishing returns. Bigger engines require more horsepower to operate and carry them. This effectively limited the motive power available to piston-engine aircraft, which in turn made the requirement for a smooth aerodynamic shape a necessity to reduce the drag created by the aircraft to a minimum. This limited the physical carrying capacity of these aircraft.

THE JET ENGINE

Diagram 2.17 Modern Jet Air Transport Aircraft

With the advent of the jet engine entering commercial service in the 1960s with its increased power, the modern air transport aircraft's fuselage has become basically a cylinder with a tapered tail and rounded nose. The power available exceeds the drag created by this profile.

The advantages of jet engine aircraft are:

- Easier manufacture resulting in lower cost
- Easier loading and unloading of cargo resulting in quicker turn around time
- Greater capacity for cargo resulting in increased revenue
- Easier seat pitching, allows fitting of more seats resulting in increased revenue

FUSELAGE PROFILE AND TRICYCLE UNDERCARRIAGE

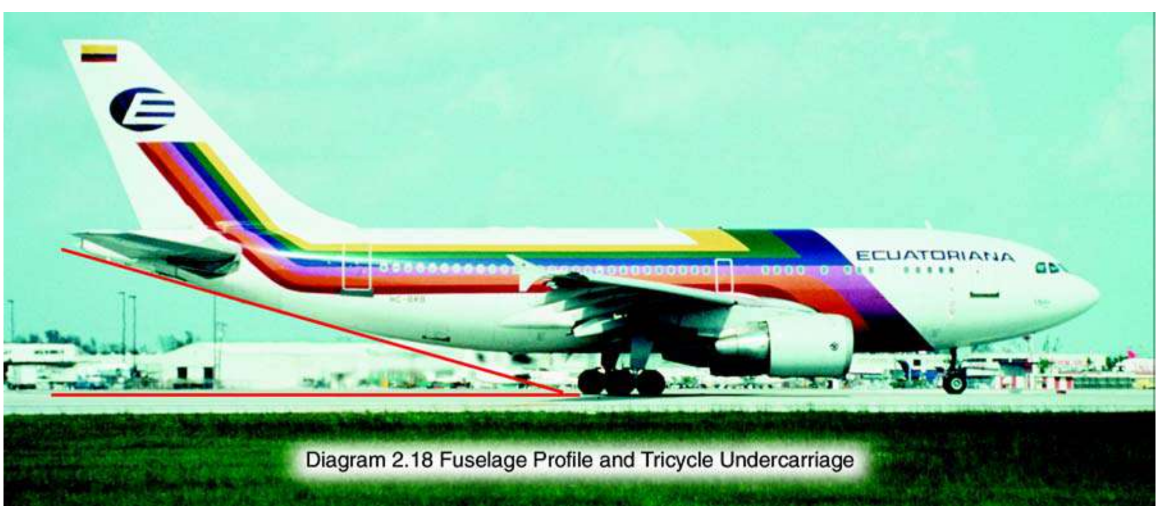


Diagram 2.18 Fuselage Profile and Tricycle Undercarriage

The use of tricycle undercarriages on large air transport aircraft has resulted in an alteration to the design of the rear of the fuselage. Diagram 2.18 shows how the lower surface of the fuselage tapers upward to the tip of the tailcone, while the top surface of the fuselage remains virtually unchanged.

This change was brought about by the difference in the way a tricycle-geared aircraft takes off compared to the take-off of a tail-wheeled aircraft. For tail-wheeled aircraft, the tail wheel lifts off during the take-off run. Then the aircraft reaches flying speed and lifts off or is rotated, by which time the tail plane is lowered back toward the runway, resulting in the plane lifting off.

For large tricycle-undercarriage aircraft, the aircraft reaches flying speed during the take-off run. To initiate the lift off, the pilot has to rotate the aircraft about the main gears, requiring the tail to go down as the nose is raised. The angle of the rear fuselage seen in diagram 2.18 gives an indication as to the degree of rotation.

TAIL BUMPER

To protect the rear fuselage from damage in cases where the pilot has over-rotated the aircraft, a tail bumper or skid is fitted to the structure. This is either fixed or retractable. Some aircraft have retractable tail bumpers with small wheels, which deploy with the landing gear.

FUSELAGE MOUNTED ENGINE

For propeller aircraft powered by a single engine, either piston or turbo-prop, the options for mounting the engine are:

- To a support frame attached to the front bulkhead as a tractor
- To a support frame attached to the rear bulkhead as a pusher

Whichever method is used, the mounting frame and bulkhead or pylon become a major structure. They take the strain of the mass of the engine when stationary and the mass, vibration, torque, and thrust when it is running.

Single gas turbine engines can be mounted within the fuselage. This configuration is seen in some military trainer aircraft.

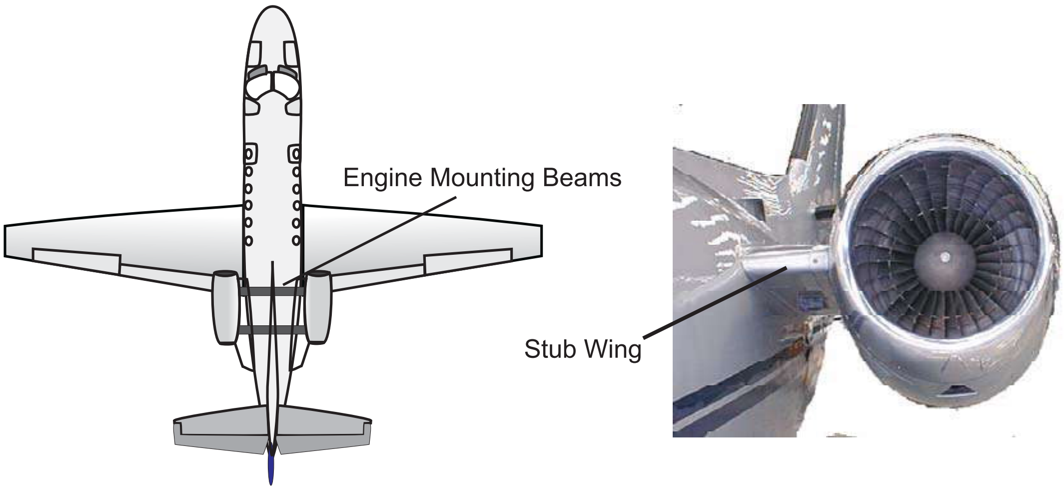


Diagram 2.19 Fuselage Mounted Engines and Stub Wings

For twin or four turbine-engine aircraft, the engines can be mounted to each side of the rear fuselage on mounting beams that carry the loads and transmit the thrust to the fuselage. These are referred to as **engine carry through beams**. These engines are mounted on **stub wings**, as shown in Diagram 2.19.

The advantage of mounting the engines at the rear of the fuselage is the reduction in cabin noise and drag compared to wing-mounted engines.

ACKNOWLEDGEMENTS

We would like to thank and acknowledge:

For Diagrams

3.7A 3.9 3.10 3.28 3.29	Key Publications Ltd. for the right to use and adapt a cutaway diagram of a 737-300 aircraft drawn by Mike Bodrocke
3.12	Photograph by kind permissions of Mr. Ashley Gibb
3.25	Aviation Partners Inc.

INTRODUCTION

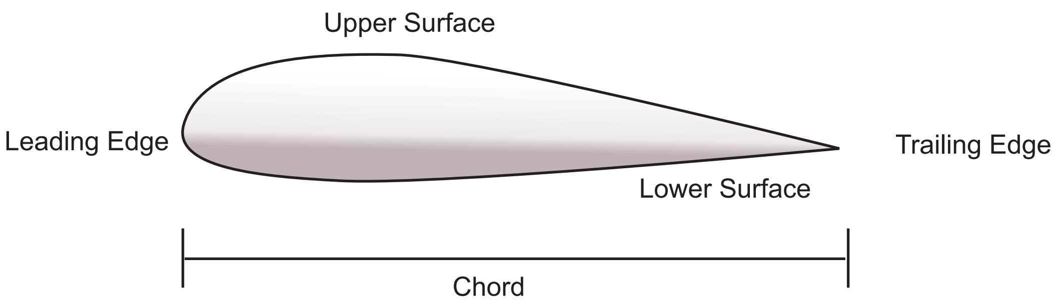


Diagram 3.1 Cross Section of a Wing

Wings were previously referred to as main planes. They produce the lift that supports the aircraft in flight. When air flows over the top surface of a wing from the leading edge, Le, to the trailing edge, Te, the distance that it has to travel is increased by the camber, or curve of the surface, compared to the distance that it would have to travel when passing under the wing, as shown in diagram 3.1. The air flowing over the cambered surface increases its velocity, which results in a decrease in the static pressure above the wing. The amount of lift produced is dependent on the speed of the air over the wing, the camber, and the surface area of the wing.

The internal construction and the shape of the wing depend on the intended speed and use of the aeroplane, such as supersonic, transonic or subsonic, general aviation, stunt, commuter, air transport, cargo, heavy-lift, and short take-off and landing (STOL). To classify a wing's lifting coefficient, its cross section, etc, it is given an NACA number or a Reynolds number.

The fitment of wings to fuselage falls into three basic categories:

- Biplane
- Braced monoplane
- Cantilever monoplane

BIPLANE

Diagram 3.2 Biplane

In the early days of flight, to get the lift required from slow moving aircraft, the wings of some were given a large camber. Many aircraft were biplanes, and some designers made triplanes. To make the structure as light and strong as possible with rigidity, while allowing a degree of flexibility, struts and wires braced the upper and lower wings, as shown in diagram 3.2.

Diagram 3.3 Biplane Wing Bracing

The structure created by the wings' internal spars, the interplane struts, and bracing wires acted to make a very strong latticework girder, which had high rigidity and resisted torsional loads. In flight, the stress was felt in the flying wires. These acted to hold the tips down and support the fuselage. When the aircraft was on the ground, the tension in the landing wires supported the wings, as shown in diagram 3.2 and 3.3.

By their design, biplanes are lower speed aircraft due to the drag created by the wings and bracing structures. Modern designs, such as the Pitts Special stunt aircraft, have replaced the wooden spars with modern materials and, in many cases, done away with the bracing wires. Their aerobatic displays show how strong the box structure can be.

BRACED MONOPLANES


Diagram 3.4 Braced High Wing Monoplane

Increase in airspeeds enabled designers to remove one wing and save the drag it created. However, to ensure that the wing has sufficient rigidity and can withstand the air loads, ground loads, and torsional loads without increasing its structural mass, a strut from the fuselage braces the wing. In flight, the strut of a high-wing monoplane is subject to tensile loads, acting as a tie. On the ground, the strut is subject to compression loads and acts as a brace.

This design is still in use for low-speed aircraft. In many cases, the bracing struts are connected to the structure supporting the main landing gear. When the aircraft is on the ground, the mass of the wings is placed directly on the landing gear.

To ensure that the bracing strut is **fail-safe,** it is manufactured from more than one tensile member. For example, in the Shorts 330 Sky Van, there are three such members per bracing strut.

CANTILEVER MONOPLANES

Diagram 3.4b Cantilever Monoplane

In this design, the wing is self-supporting and attached to the fuselage at one end. The designer must ensure that the wing's structure is capable of withstanding the torsional loads trying to twist the leading edge down, the loads created along its length in flight, and its mass on the ground.

COMPONENTS OF WING STRUCTURE

While the design and method of manufacturing wings has changed, the basic requirements and nomenclature remain the same. These are:

- Spars
- Ribs
- Stringers
- Skin
- End caps
- Wing tips
- Leading edge
- Trailing edge
- Root
- Torsion box
- Centre section

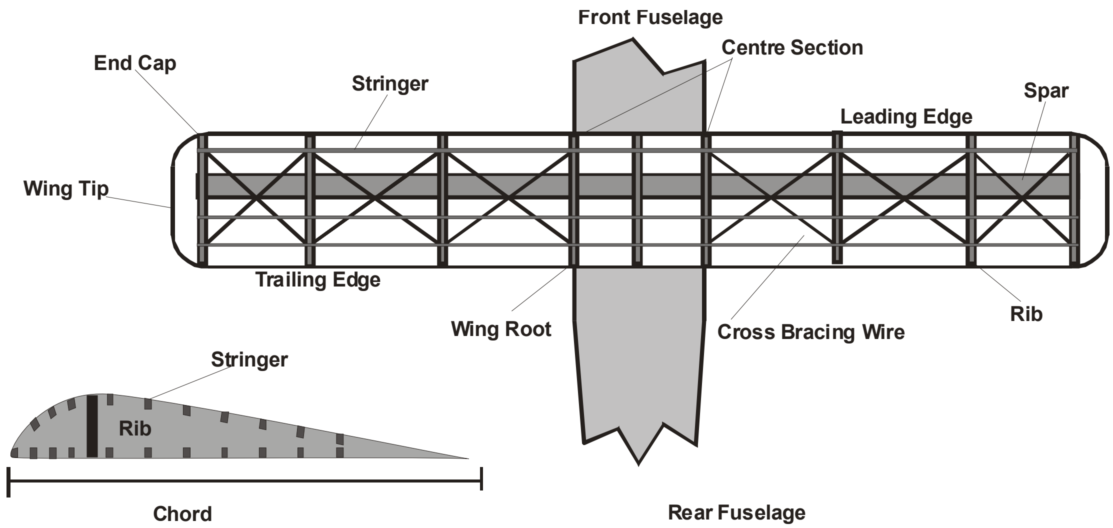


Diagram 3.5 Components of Wing Structure

Diagram 3.5 shows the basic components used to produce a fabric covered light aircraft wing. Extra torsional rigidity is achieved by using tension wires to brace the structure. The main structural component of a conventional wing, the spar, takes all the flight loads generated by the wing and transfers them to the fuselage structure. In very light aircraft, there may only be one spar. However, in medium transport aircraft, it is standard to fit two spars (see diagram 3.5A). For large air transport aircraft, three spars are normally used as seen in diagram 3.6.

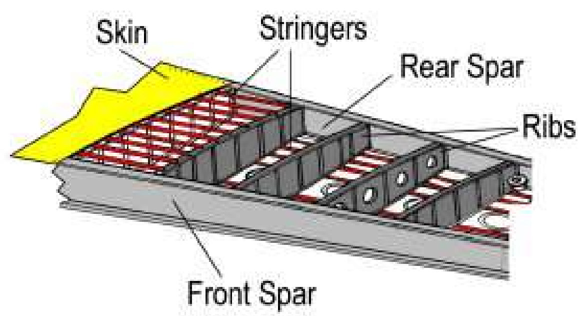


Diagram 3.5A Structure of a Modern Wing

STRINGERS

Stringers are smaller span-wise members that act to support the skin and increase the wing's rigidity when the surface they are attached to is in compression. For alloy-skinned aircraft, stringers are attached by either rivets or bonding to both ribs and skin.

RIBS

Ribs that run front to rear give the wing its aerofoil cross section. They act to brace the spar and support the skin and the stringers. They also act to spread the concentrated loads of wing-mounted engines, undercarriages, and control surfaces to the other structures. Wings can be manufactured as one complete assembly, normally for light aircraft, or in various sections that are joined together (e.g. Airbus).

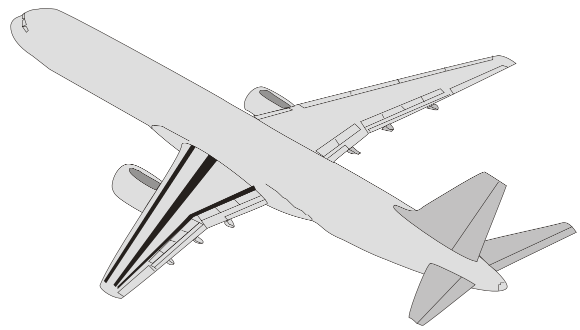


Diagram 3.6 Spar Location

SPARS

Diagram 3.6 shows the location of spars in large aircraft wing structure. There are normally three spars in a large aircraft:

- Front or forward spar
- Main spar
- Rear or auxiliary spar

The front spar is designed to take the air-load acting to bend the wing backward. Where external loads such as engines are mounted, the front spar shares some of the load.

The main spar is designed to be the main structural member of a conventional wing and it carries the majority of the loads placed on the wing. The rear spar acts as a mounting structure for the trailing edge flaps and ailerons. It also resists the upward-twisting action of the up-wash felt at the trailing edge. The spars also have to resist the torsional effect created at the wing tip by the movement of the aileron. When a wing flexes, the greatest motion is at the wing tip. However, the greatest strain is at the wing root.

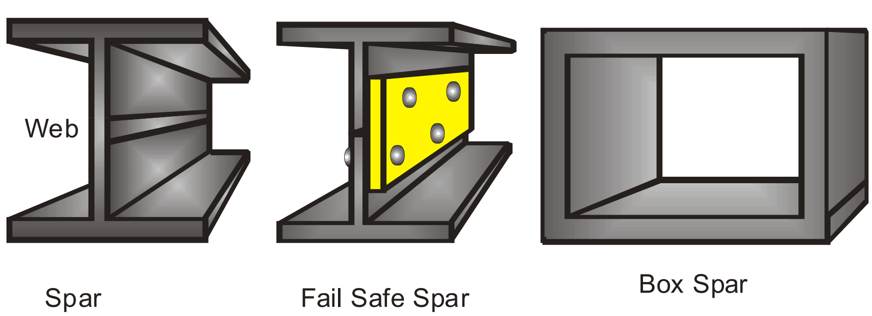


Diagram 3.7 Spars

As per diagram 3.7, a basic spar can be an I-beam, formed by a top and bottom cap with a central web. The **T** effect of the top and bottom caps gives the thin **I** web resistance to bending and buckling, which would otherwise have to come from an increase in material. However, should the strain placed on the beam be greater than its limits, the spar would crack, deform, and eventually fail.

Three separate parts joined together make up the fail-safe spar. This incurs a slight weight penalty for protection.

A Box spar gives great rigidity for minimum weight. In the early days, these were made of plywood. On modern air transport aircraft, the box spar has become the torsion box.

TORSION BOX

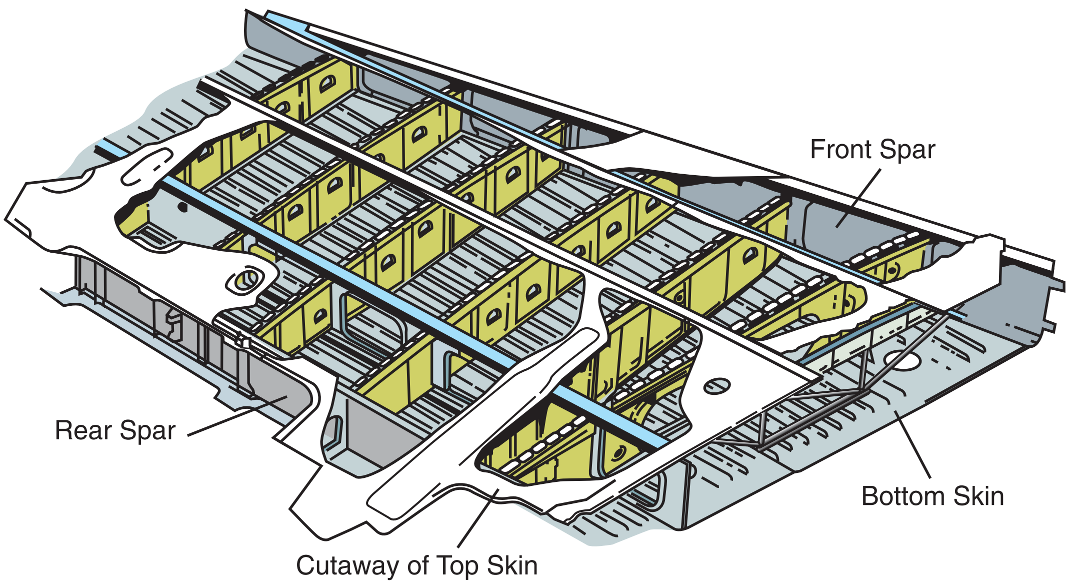


Diagram 3.7A

The upper and lower surfaces and the spars can form a torsion box. This structure increases the rigidity of the wing and resists the twisting and bending motion without increasing the mass of material used in construction. In diagram 3.7A, a cut-away drawing of the outer section of a 737-300 wing, the torsion box structure is formed between the front and rear spars. On larger and heavier aircraft, the torsion box encloses the front, main, and rear spars.

MODERN WING DESIGN

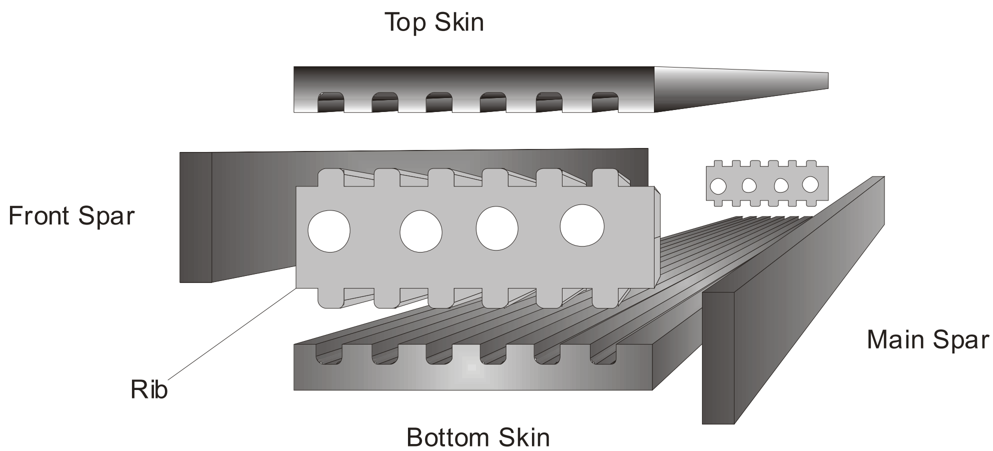


Diagram 3.8 Torsion Box

In modern wing design, as depicted in diagram 3.8, a torsion box is formed by the front and rear spars bonded onto machined skins. The stringers are formed when the skins are machined or etched by leaving span wise ridges of the sheet material. This produces a stronger, lighter, and stiffer wing. In these designs, the skin carries some of the flight loads.

WING LOADING

The amount of weight that a unit area of wing is supporting is termed the wing loading, and is found by dividing the load on the wing by the wing area. The higher the value is, the greater the structural strength of the wing needs to be. An aircraft like the Pitts Special has to have great strength due to the g-forces that act on the aircraft during aerobatic manoeuvres. Aircraft like the Tiger Moth and Piper Cub have a large wing area relative to their weight, which gives them a low wing loading. Air transport aircraft carry large payloads using a relatively small wing area, which gives them a high wing loading.

To find wing loading in straight and level flight, divide the aircraft's mass by the area of the wing.

> **For example:**
> An aircraft with a maximum take-off mass of 3300 lb supported by a wing with a surface area of 300 square feet has a wing loading of 11 lb per square foot at 1g.
>
> 3300 lb divided by 300 sq ft = 11 lb per square foot.

This would rise to 16.5 lb if the pilot pulled 1.5g

> 3300 X 1.5 = 4950 lb divided by 300 = 16.5 lb per square foot.

As can be seen from above, an increase in positive g-force due to manoeuvring increases the effective load on the aircraft's structure and the wing's loading.

Aircraft must be designed and engineered for the purpose for which they are intended and used within the limits of their design.

For air transport aircraft, the structure must be able to withstand a positive manoeuvring load factor of no less than +2.5g but not greater than +3.8g. Its structure must also be capable of withstanding a negative load factor of not less than –1g at speeds up to its cruise speed.

From JARs 23:

Aircraft are categorised into three groups:

- Normal
- Utility
- Aerobatic

Aircraft in the normal category are limited to non-aerobatic operations including:

- Any manoeuvre incidental to normal flying
- Stall except whip stalls
- Lazy eights, chandelles, steep turns, or similar manoeuvres in which the bank angle is limited to 60°

Aircraft in the utility category are limited to any of the above manoeuvres, in which the bank angle is limited to 90°, plus spins if A/C type approved.

Light aircraft normally belong to this category. However, for many light aircraft such as the Piper Warrior II, operating in utility mode is limited by weight and CG location. This is to ensure that the wing loading during manoeuvres does not exceed the wings' structural load limitation, and that the aircraft can recover from stalls and spins.

Aircraft in the aerobatic category are not bank-angle limited and are designed to withstand g-forces in excess of the utility category. These aircraft can normally enter extreme manoeuvres straight from take-off, legality permitting.

Commuter aircraft as well as transport aircraft are limited to any manoeuvre incidental to normal flying. This includes stalls, except whip stalls, and steep turns in which the angle of bank is not more than 60°.

WING MOUNTED LANDING GEAR

On light monoplanes that have fixed landing gear, such as the Piper Warrior, the main gear legs normally attach directly to the main spar, thereby transferring the fuselage's load via the spar to the main gear.

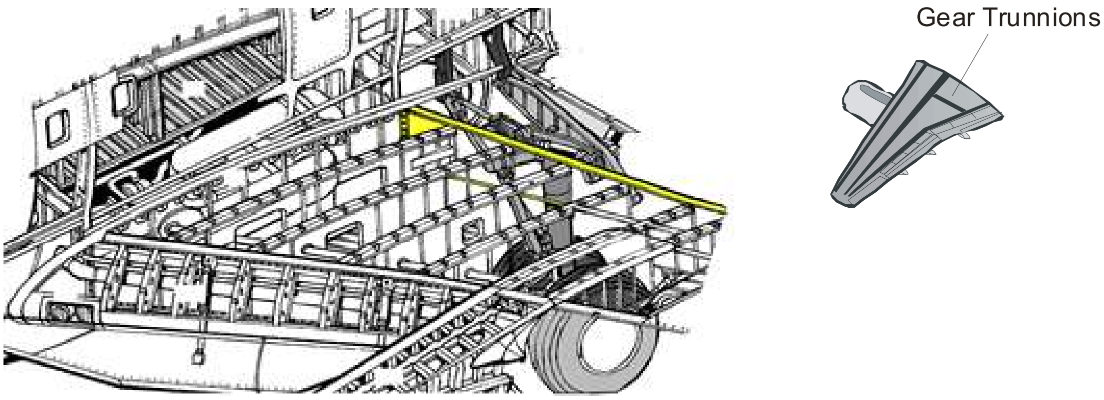


Diagram 3.9 Main Gear Mounting

Large aircraft that have retractable undercarriages mounted in the wings have one trunnion pivot point attached to the main spar and the other to supporting structures, as shown in diagram 3.9.

MONOPLANE CENTRE WING SECTION/FUSELAGE SECTION

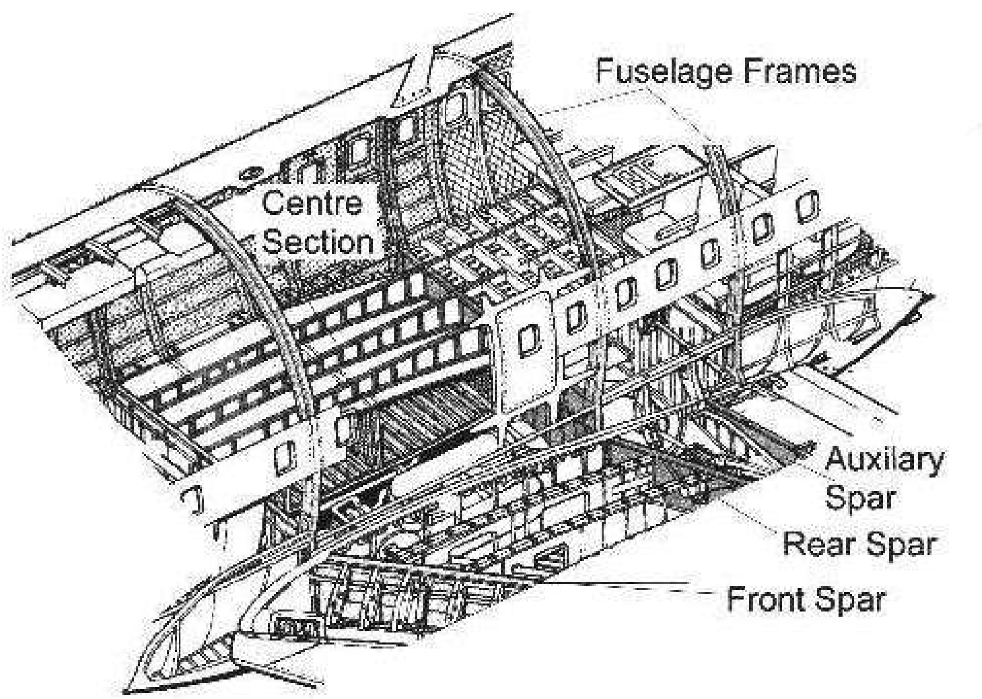


Diagram 3.10A

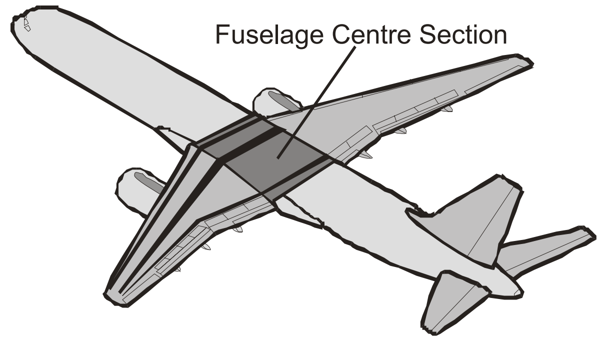


Diagram 3.10B Centre Section

Where an aircraft's wings are manufactured in two sections that are attached to the fuselage structure, the centre section is manufactured with major forgings to give the wings sufficient strength. These act to carry the loads through from one main spar to the other. In diagram 3.10A, the front, rear, and auxiliary spars are attached to major forgings that are built into the fuselage structure. These are attached to major frames that pass the loads from the fuselage to the wings' structure (see figure 3.10B). In light aircraft, the same concept is used. In these cases, the wing spars are inserted into a box section that is built into the structure of the fuselage and secured. This forms one continuous spar.

WING LOCATION

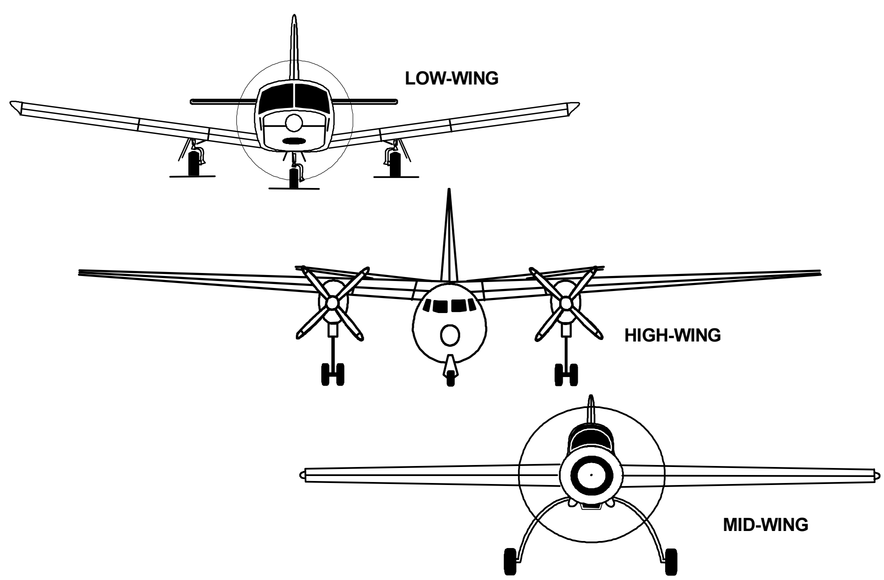


Diagram 3.11 Wing Locations

LOW-WING

When loaded with passengers/cargo, a low-wing aircraft will have a higher centre of gravity than that of a high-wing aircraft, due to the mass of the load. They are more crash worthy due to the spar box configuration within the central fuselage or the main spar of a single piece wing passing under the fuselage.

Where two wings are attached to major carry-through castings mounted within the fuselage, the floor level has to be higher than that of a high-wing aircraft. The landing gear legs of a low-wing aircraft are shorter than those of a high-wing aircraft. If an engine propeller-combination is to be mounted on the wings, the propeller size is limited by the ground clearance unless the legs are extended.

HIGH-WING

High-wing aircraft have the advantage of giving good downward visibility and make cargo loading and unloading easier due to the lower cabin floor.

Structurally, the wing can be manufactured in one section; the fuselage-to-wing joint must be strengthened as the fuselage and load hang from the wing. To give the aircraft crash-worthiness, the fuselage structure under the wing must be strengthened to prevent the wing from collapsing.

Where the landing gear is wing mounted, the main gear legs must be lengthened and strengthened. High-wing aircraft are used for heavy lift, as there are advantages aerodynamically in having a high wing. The Principles of Flight volume fully addresses these.

MID-WING

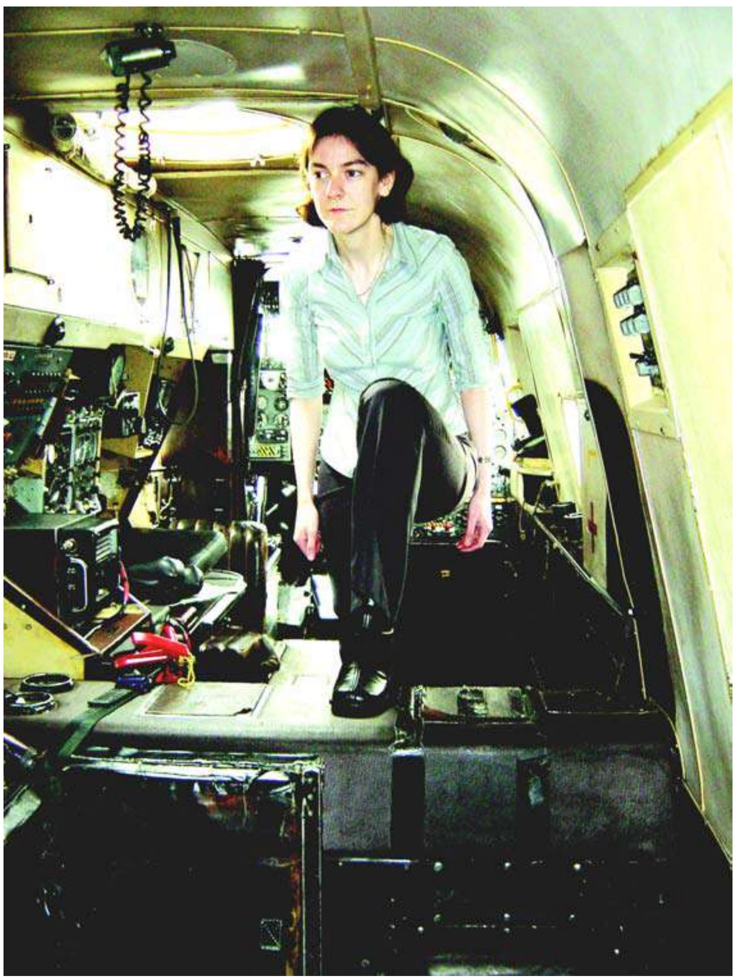


Diagram 3.12 Mid-Wing Spar

Mid-wing aircraft have the advantage of improved aerodynamics for high-speed flight but the disadvantage of having spars passing through the middle of the fuselage (see diagram 3.12 of a post-war Vickers Viking). This effectively limits their use, as this design is not acceptable under the JARs for a modern air transport aircraft.

HIP-WING

Hip-wing is the term used for wings that mount between the middle and bottom of the fuselage.

FAIRING PANELS

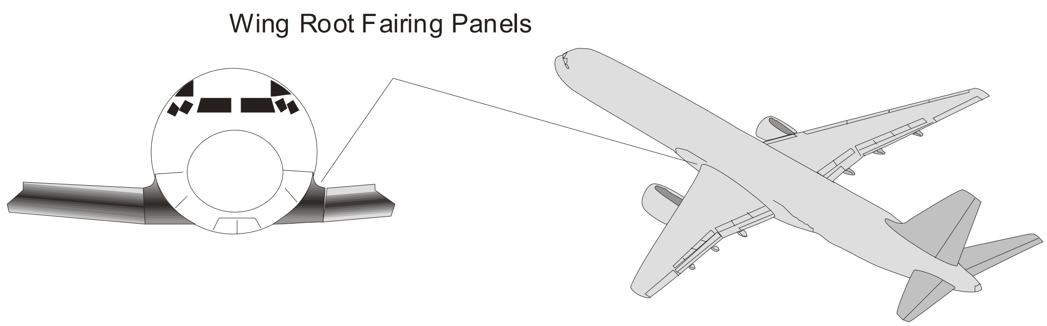


Diagram 3.13 Fairing Panels

The total drag acting against an aircraft in flight is the sum of the profile drag and lift-induced drag. A major part of the profile drag takes the form of interference drag. This is where two airflows interfere with each other. This results in a turbulent flow, which absorbs energy, thus creating drag.

Unless the joints between the wings and fuselage are carefully **faired** to allow a smooth transition from one streamlined shape to the other, interference drag is created. To reduce the interference drag, fairing panels, often referred to as **fillet panels**, are designed for high performance and larger aircraft. While these have little or no structural loading, they must be a good fit and kept in good condition to be effective. See diagram 3.13.

While the upper surface of the wing-to-fuselage joint is the most important, since it is the longest and has the most affected airflow, anywhere there is a junction between two external components, there is interference drag. However, junctions under the wings such as engine mounting pylons are not normally faired as they have a lesser effect. Refer to engine mounting later in this chapter.

Junctions for the empennage are not faired in most cases, as the airflow along the fuselage at this point is becoming turbulent and the additional structure would not result in any real aerodynamic gain.

DIHEDRAL ANGLE

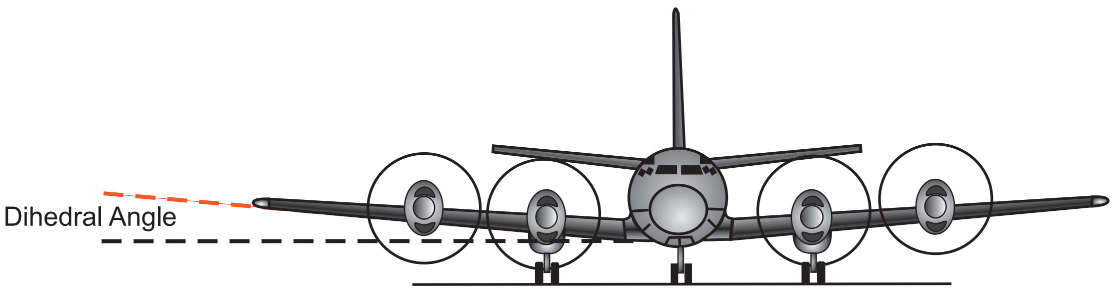


Diagram 3.15 Depicts the Dihedral Angle of a Low Winged Air Transport Aircraft

The dihedral angle is the angle between the wing and horizontal plane when the wing is above the horizontal plane. Refer to diagram 3.15.

ANHEDRAL ANGLE

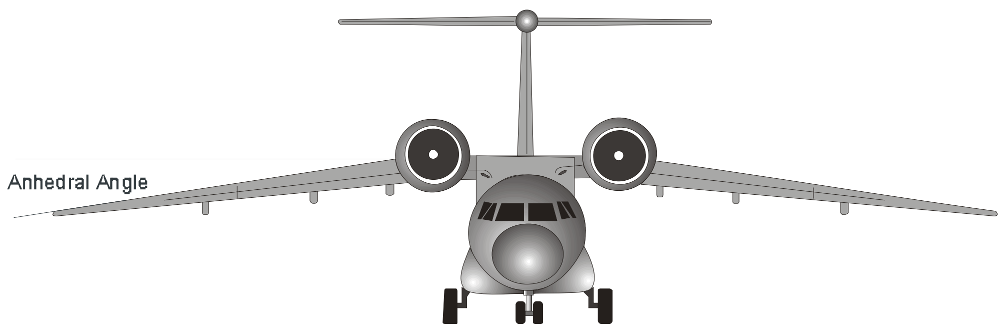


Diagram 3.16 Anhedral Angle

The anhedral angle is the angle between the wing and horizontal plane when the wing is mounted so that it is below the horizontal plane. Refer to diagram 3.16.

SWEEP BACK ANGLE

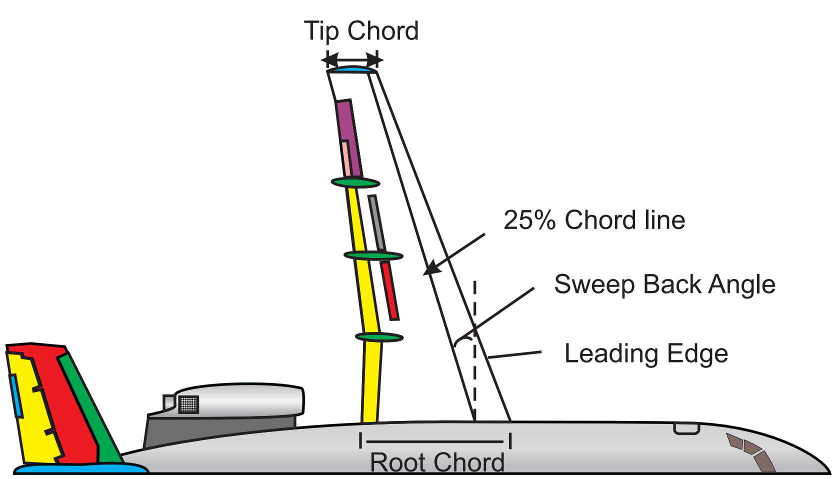


Diagram 3.17 Sweep Back Angle

This is the angle formed between the 25% chord line of the wing and a line that is at right angles to the wing root chord. This perpendicular line is located at the root's 25% chord point, as diagram 3.17 shows. Note that the 25% chord line is not parallel to the leading edge of the wing.

WASH OUT

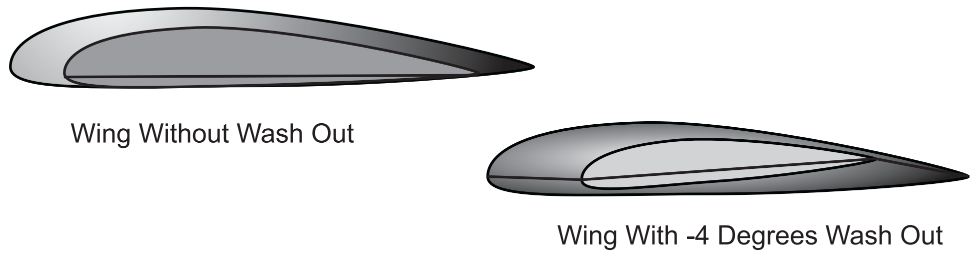


Diagram 3.18 Wash Out

Wash out is the term given to a downward negative twist of the wings and leading edge toward the wing tip, see diagram 3.18. At the wing tip, this washout is normally in the order of 3 to 4 degrees below the angle of incidence at which the wing is set to the fuselage. As this alters the angle of attack for the air passing over the outboard wing compared to the inboard wing, it assures that the inboard wing section stalls before the outer section. This in turn assures that the ailerons are in unstalled airflow.

WING TIPS AND WINGLETS

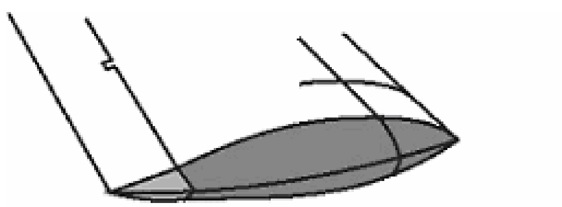


Diagram 3.19 Plain tip

The wing tip of a conventional wing acts as an aerodynamically shaped cover to the wing structure. In many light aircraft, they are made of GFRP to save weight and complication or light alloy. However, as detailed in the following paragraphs, the design of the wing tip radically affects the airflow across the wing, which in turn alters the lift produced by the wing.

WING TIP VORTEX

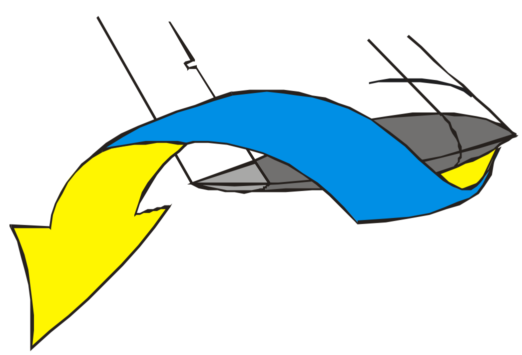


Diagram 3.20 Wing Tip Vortex

Aerodynamically, a plain rounded wing tip is not a good design as airflow from the underside of the wing, which is at a higher pressure, flows around the tip to the low-pressure area over the top surface of the wing. This sets up a flow pattern in the air travelling from front to rear across the wings. On the top surface, it is from outboard to inboard, and for the under surface, it is from inboard to outboard.

This flow pattern creates a spiralling vortex known as a **wing tip vortex** that absorbs energy from the airflow, thereby creating drag. The greater the tip length is, the larger the vortex effect.

This effect becomes greater as the differential between upper and lower surfaces increases, when the wing is at a large angle of attack, high co-efficient of lift value (e.g. approach to land), or when at cruise speed at high altitude.

What is required is some form of end plate that stops the leakage of air from the underside to the topside of the wing. However, a vertical plate just added to the wing would have to be quite large and would create its own parasitic drag resulting in an overall increase in drag.

There are several methods for reducing the vortex:

- High aspect ratio wings
- Elliptical wings
- Modified wing tips
- Wing tip tanks
- Winglets

HIGH ASPECT RATIO AND ELLIPTICAL WINGS

High aspect ratio wings, such as those used on gliders where the tip length is small compared to the root length, reduce the effect as do elliptical wing plan forms, such as that of the Supermarine Spitfire. However, the production of elliptical wings is complex and not cost effective for commercial aircraft.

MODIFIED TIP CAPS

Some light aircraft use modified tip caps to reduce the effect of the tip vortices, thereby reducing the induced drag. These take the forms of drooped, or down curve and up curve.

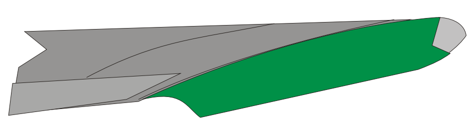


Diagram 3.21 Drooped or Down Turned Tip
as Viewed from the Trailing Edge

Drooped tips have a downward curve like the letter **'j'** on its side, as shown above. The down curve acts to give the under wing span wise flow a downward action. The tip vortex is formed away from the actual tip.

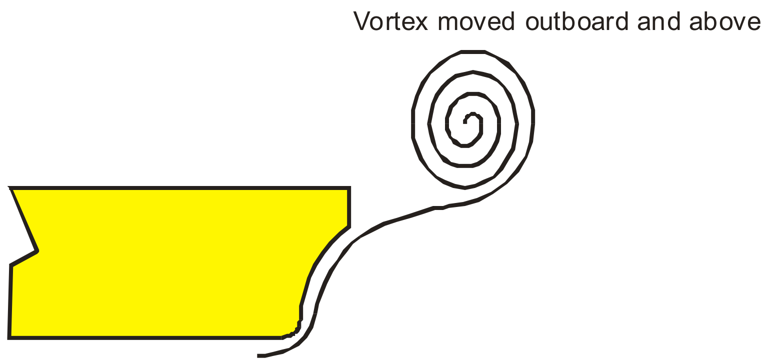


Diagram 3.22 Upturned Wing Tip of the Hoerner Design

Up curved tips, as designed by S Hoerner, are used on the Piper Warrior. The slight up sweep from the lower surface toward the upper surface acts to deflect the under wing span-wise flow upward and outward. The vortex is created slightly above and away from the wing tip. This displacement reduces the effect of the vortices on the wing's down wash and, therefore, reduces the induced drag.

TIP TANKS

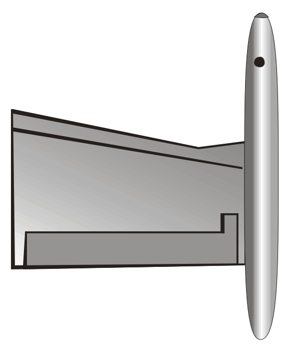


Diagram 3.23 Cylindrical Tip Tank

By attaching tip tanks to the wing tips, they effectively become end plates, reducing the amount of leakage by their presence. There are also further advantages of fitting tip tanks:

- Increased lift from the tank's profile
- Increased fuel capacity
- Increased down force at the wing's tip, resisting the upward bending motion of the wing tip due to the lift created

Diagram 3.24 Cessna 310

However, upswept tip tanks, as those fitted to the Cessna 310 aircraft, have the added advantage of displacing the vortices upward and outward in a similar manner as the Hoerner tip.

As the total effect of wing tip vortices on induced drag is determined by the distance between them, this shape of tip cap has the action of aerodynamically increasing the wing's span without a physical increase of the span. It also changes the wing's aspect ratio, hence reducing the induced drag.

WINGLETS

Diagram 3.25 Aviation Partners Winglets

Winglets are replacing wing tips on larger aircraft and high altitude aircraft. There is more to winglets than just a means of reducing wing tip vortices. Birds, such as crows and vultures, have been using a variation on the theme with their individual wing tip feathers for eons.

Winglets were originally fitted to Learjets, which made use of their enhanced lift at high attitude. Since the early days some twenty years ago, the aviation industry has refined the design, and they are now being retrofitted to older airliners. Given the major structural alteration and the cost of this work, it is evident that these items must have a significant effect on the performance of the aircraft. Expect future air transport aircraft to have winglets as standard equipment, as designing the winglet into a wing is easier than the retrofit.

A Cross section of Winglet

B Sweep 38 degrees

C Inclination from the vertical 15 degrees

D Span of the Winglet

E Wing's dihedral angle 7 degrees

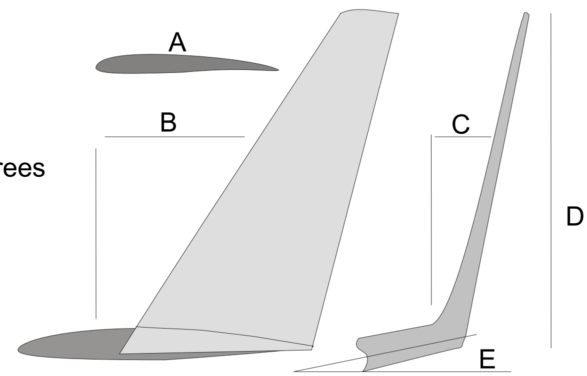


Diagram 3.26 A design of a Winglet

The advantage of winglets comes from their aerofoil cross-section (see diagram 3.26), and the way they are mounted. Winglets are normally swept back, with the tip inclined slightly outward. The leading edge is angled slightly outward, so that the winglet's chord line does not run parallel with the aircraft's longitudinal centre line.

The flow of air from the bottom surface of the wing to the upper surface and the inward flow of air across the top surface of the wing react with the normal free-stream airflow to create a small positive angle of attack for the winglet. This results in lift created over the inner surface of the winglet.

Lift is always at right angles to the airflow. A component of winglet lift acts in the direction of flight. This has the effect of negating some of the aircraft's drag. The downwash from the trailing edge of the winglet blocks or counteracts the flow of the main wingtip vortices, and even the vortex from the tip of the winglet is positioned to affect a portion of the main wingtip vortex.

The amount of lift that the winglet produces depends on the angle of attack at which the aircraft is being flown. High angles of attack, which produce the largest vortex effect, produce the maximum lift from the winglets.

As high co-efficient of lift conditions are encountered in take-off, landing, and high altitude cruise, one of the first aircraft to fit winglets as standard equipment was the Lear Jet 55. This aircraft was designed to be efficient at high altitudes.

For commercial aircraft, winglets may offer the operators:

- A decrease in the take-off and landing distance required for a given mass
- An increase in payload for a given take-off / landing distance available
- A reduction in specific fuel consumption

Winglets have certain disadvantages:

- At low co-efficient of lift values, they add to the aircraft's overall drag by increasing the parasitic drag. Induced drag also increases with the additional lift created by the winglets.
- They increase the aircraft's mass due to the need for them to be "manufactured so that they can withstand the aerodynamic loads that they could expect to be subject to".
- They impose an additional structural load that requires the wing to be strengthened, increasing the aircraft's mass.

The advantages outweigh the disadvantages for a commercial aircraft. To gain the same lift by increasing its overall span instead of the adding winglets would result in the addition of a much greater mass of material, as the structural complexity needed to support the extension would be greater than that of the winglet. Also from a commercial point of view, an aircraft with winglets still fits into the existing stands at airports around the world, whereas wingspan extensions would not.

WING MOUNTED ENGINES

Diagram 3.27 Comet — Wing Root Mounted Engines

The location and mounting system of aero engines affects the structural design of the wings. In diagram 3.27 of a Comet, the engines are located at the wing root. This requires the spars to form a spectacle shape to allow for the intakes and engine mounting. As these mountings over-complicate the wing design, it is standard for turbine aircraft with wing-mounted engines to have the engines mounted on pylons beneath the wings. One exception is the VFW/Fokker 614, which has engines mounted on over-wing pylons.

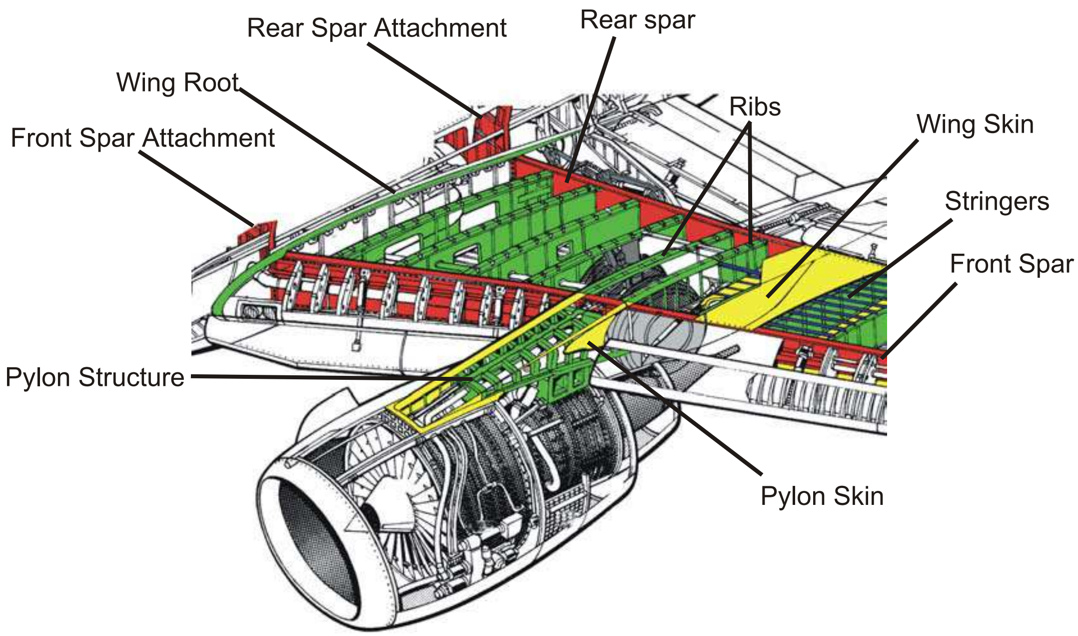


Diagram 3.28 Under Wing Pylon

The structure of the pylons is attached to the front and main spars to spread the load created by the engines to the wing (see diagram 3.28). As the engines protrude forward of the spars, their masses produce a downward twisting (torsional), moment. For the airflow, the pylons act as vertical plates that guide it rearward, reducing the under wing span wise flow. However, at the intersection of the pylon and the wing, a pressure increase can cause the oncoming airflow to lift upward and create a vortex across the top surface of the wing running from front to rear.

ENGINE CHINE/STRAKE

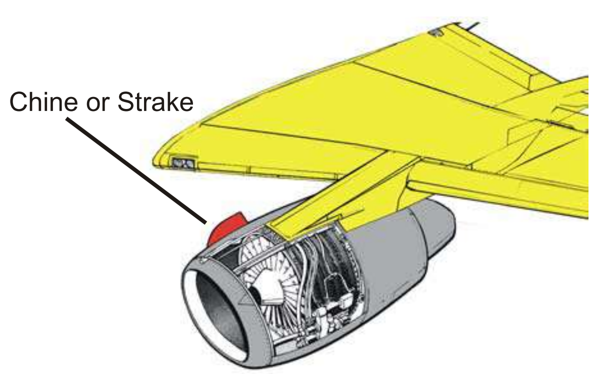


Diagram 3.29 Engine Cowling Chine or Strake

Local airflow around the engine cowlings can add to the problem of over-wing vortex and can be controlled by the addition of a Chine(s) or Strake(s) to guide the air backward and below the wing. Diagram 3.29 shows a single strake fitted to the inboard side of the engine cowling. Other aircraft may require one on each side.

Chapter 4
Empennage or Tailplane

ACKNOWLEDGEMENTS

We would like to thank and acknowledge:

For Diagrams

4.3 4.10	Key Publications Ltd. for the right to use and adapt a cutaway Diagram of a 737-300 aircraft drawn by Mike Bodrocke
4.15	Eclipse Aircraft USA
4.20	Ashley Gibbs

INTRODUCTION

On conventional aircraft, the empennage or tailplane normally takes the form of a single vertical fin and horizontal surface. These are fitted to give the aircraft directional and longitudinal static stability and are mounted at the tail (rear) of the aircraft.

POSITIVE STATIC STABILITY

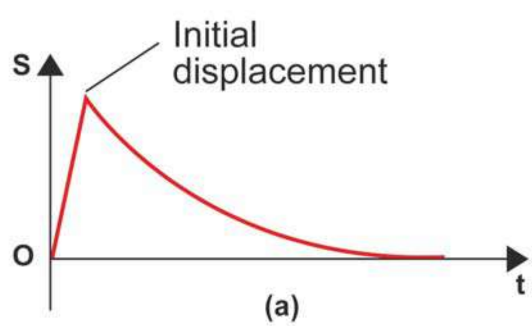


Diagram 4.1 Static Stability

Positive static stability is the natural tendency of the aircraft to return to its original equilibrium condition (usually straight and level) after it has been disturbed, without the pilot taking any corrective action. If an aircraft has neutral static stability, the aircraft would maintain its new attitude after any disturbance to its equilibrium. If an aircraft has static instability, any disturbance would result in the aircraft continuing to alter its heading or pitch.

STABILITY FUNCTION OF THE FIN

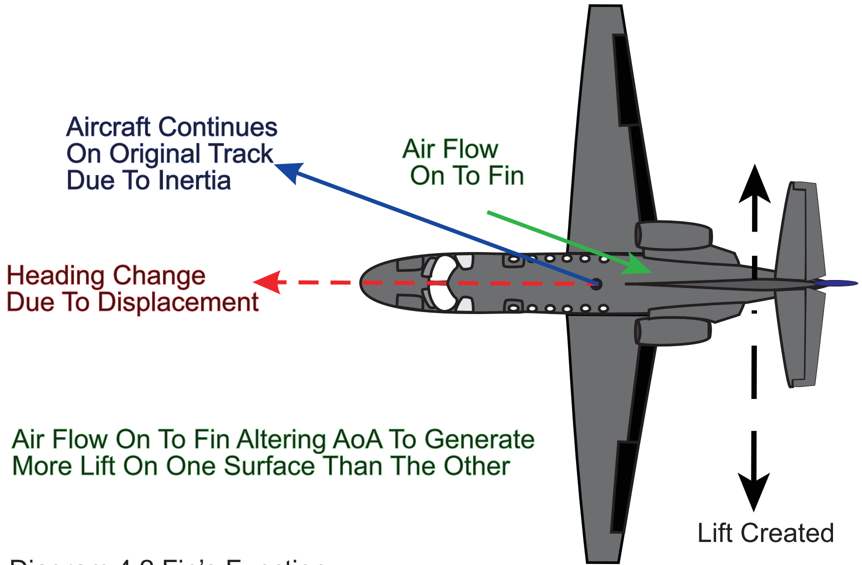


Diagram 4.2 Fin's Function

The fin's design gives the aircraft positive static directional stability about the vertical axis. If a gust of wind knocks the aircraft and causes it to yaw, it continues moving in the original direction due to the aircraft's inertia. This alters the airflow over the fin's surface, creating more lift on one side than the other and bringing the aircraft back onto its original track.

A sideslip alters the angle of attack (AoA) of the airflow over the fin. This results in a yawing moment being felt at the aircraft's CG, which acts to yaw the aircraft into the relative airflow. At small angles of sideslip, strong positive static stability forces are produced. As the sideslip angle increases, the static stability reduces to a neutral point. Any further increase in the sideslip angle results in the aircraft becoming statically unstable. JARs 23 and 25 require aircraft to have directional static stability for all normal flight conditions.

FIN STRUCTURE

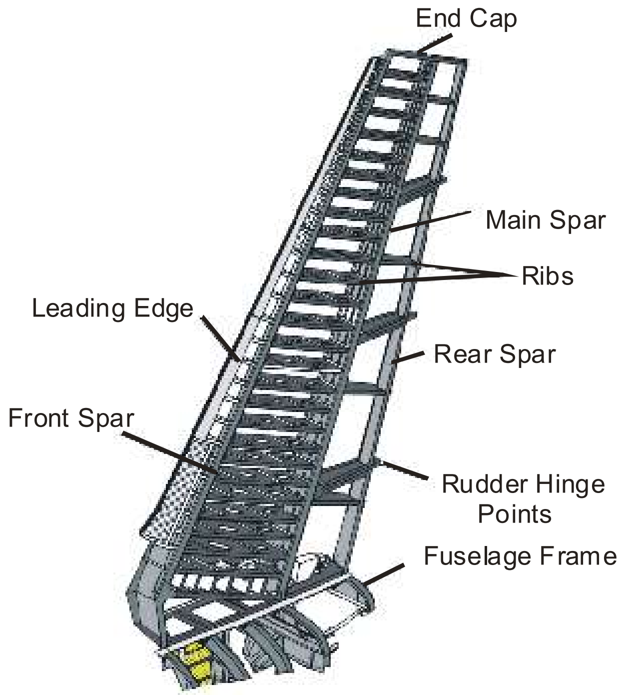


Diagram 4.3 Fin Structure

The fin, an aerofoil structure, consists of spars, stringers, ribs, and skin. These are subject to both tension and compression loads in flight. The rear spar is used as the mounting point for the rudder, the operation of which imparts further loads onto the fin's structure. The fin is attached to fuselage frames. For other than light GA aircraft, these frames are major structures, as any force acting against the fin's surface imparts a torsional stress in the fuselage.

FIN – FUSELAGE RELATIONSHIP

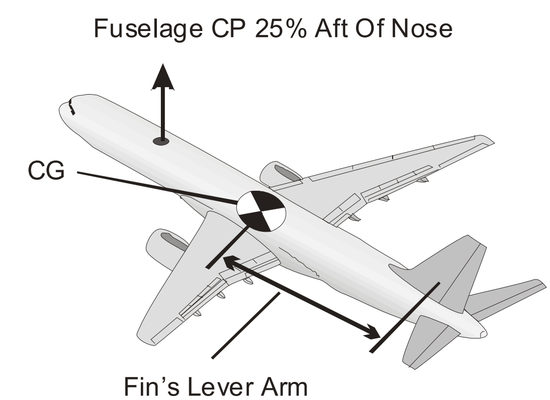


Diagram 4.3A Fin Fuselage Relationship

The fin is the primary source of directional static stability and directional control. The fuselage ahead of the CG acts in a small way to destabilise the aircraft. This is due to the fuselage's Centre of Pressure (CP) located in front or on a point ¼ the distance back from the nose for subsonic flight, which is ahead of the aircraft's Centre of Gravity (CG) as in diagram 4.3A.

NACELLES/SWEPT WINGS EFFECT ON STABILITY

Nacelles, such as engine nacelles, that protrude forward of the aircraft's wings act to reduce the aircraft's static stability. Swept back wings, on the other hand, impart a degree of directional stability to the aircraft. Listed below are some of the factors that are taken into account when an aircraft's fin is designed:

- The lift it produces
- The stalling angle of the fin
- Fin surface area
- The lever arm
- Motive power of the aircraft
- The aircraft's range of airspeed
- Wing plan form
- Horizontal tail plane location

LIFT

The fin produces lift like any other aerofoil surface. The amount of lift produced depends on the camber, angle of attack, the surface area, and the velocity of the airflow passing over it.

STALLING ANGLE

The fin, like other aerofoils, has an angle at which air flowing over it breaks away from the surface and becomes turbulent. The designer must ensure that the stalling angle of the fin is greater than the ordinary sideslip angles that the aircraft could encounter. Where aircraft are likely to experience large angles of sideslip, the fin could stall. To overcome this, dorsal fins can be fitted. See dorsal fin.

FIN SURFACE AREA

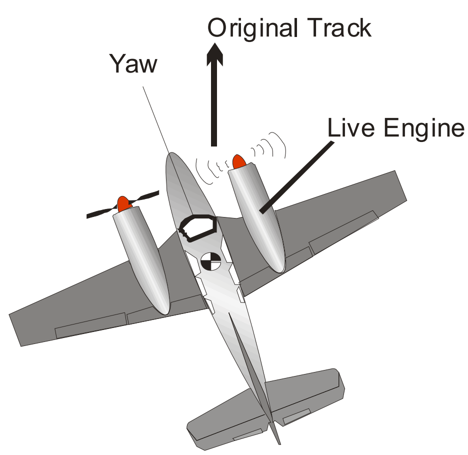


Diagram 4.4 Asymmetric Light Twin

While static directional stability is very important, controllability in asymmetric power condition is the predominant factor for multi-engine aircraft, given the engines' location from the centre of gravity (CG).

The overall length of the fuselage, the location of the wing in relation to the overall length, and the depth of the fuselage all have a bearing on the size of the fin required. The greater the length of fuselage ahead of the aircraft's CG and or the greater the keel depth of the fuselage, the more stability the fin must provide. Therefore, the fin must have either a larger surface area or longer lever arm.

LEVER ARM

The distance between the fin's location and the aircraft's operating CG range determines the turning moment for a given size of fin. To increase this moment, the fin's surface area would have to be increased. See multi-fin design.

MOTIVE POWER

There are two methods of aircraft propulsion to be considered, those that employ gas turbine engines to give thrust and those that use propellers. Where gas turbine engines are used as the method of propulsion, their efflux is both high velocity and hot. This could damage any structure that it impinged on. The designer ensures that the jet stream is kept clear of the aircraft's structure and, therefore, does not affect the airflow over the fin.

For aircraft that use propellers, they can be driven by either gas-turbine engines or piston engines. The propellers can be mounted either in front of the aircraft to pull the aircraft, known as a **tractor**, or can be mounted to push the aircraft instead of pulling. These aircraft are referred to as **pushers**.

Whichever method, pusher or tractor, the slipstream from the propeller can have an effect on the fin. Where the jet efflux from a turbo prop engine becomes part of the propeller slipstream, its temperature is reduced. See propeller effects below.

AIRSPEEDS

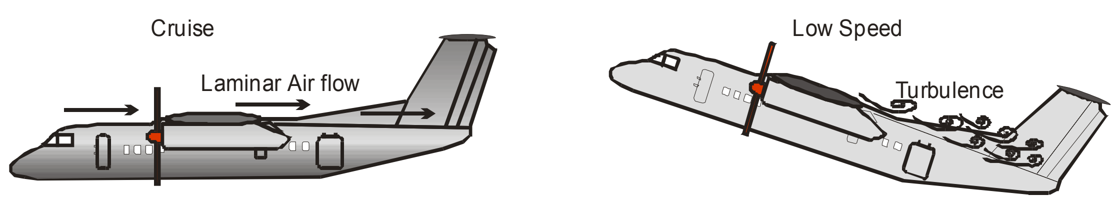


Diagram 4.5 Loss of Effective Fin Area

When aircraft fly at lower airspeeds, they require high angles of attack. The fuselage is also effectively at a higher angle of attack, so its boundary layer thickens. This increase in boundary layer thickness reduces the fin's effective surface area requiring an increase in fin area to obtain the same correcting action, see diagram 4.5. This can be achieved by using a larger fin, a ventral fin, or ventral strakes.

VENTRAL FIN AND STRAKES

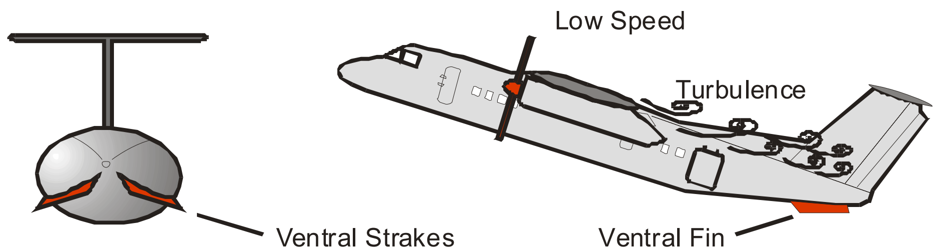


Diagram 4.5A Ventral Fin and Strakes

A ventral fin is a vertical plate attached to the rear fuselage under the fin. Aligned fore and aft with the aircraft's centre line in normal flight, the ventral fin's action is minimal, as it is in the slipstream of turbulent air flowing over the rear of the fuselage. However, when flying at high angles of attack, the action of pitching the nose up pushes the ventral fin down into clear air. This increases the surface area of the fin. Refer to the right picture of diagram 4.5A for an illustration.

Two ventral strakes can be fitted if a design requires a ventral fin and the designer wants to control the airflow around the tail through a range of airspeed. They can also be fitted if there is a possibility or problem of directional instability. The angle at which these are fitted and their length is dictated by the degree of stability required. These can also be referred to as ventral fins.

HIGH MACH NUMBERS

When an aircraft is flying at high Mach speeds, there is a reduction in the fin's ability to maintain static stability. This requires high-speed aircraft to have larger fins to make up for the reduction.

WING PLAN FORM

Sweptback wings achieve a small amount of static stability. This is due to the change in sweep back angle of the aircraft's wings when they yaw. This becomes important in tail-less aircraft, such as the flying wing. For conventional aircraft, the location of the wing is more important, as disturbed airflow from the wings can affect the airflow over the fin of the aircraft.

LOCATION OF TAILPLANE

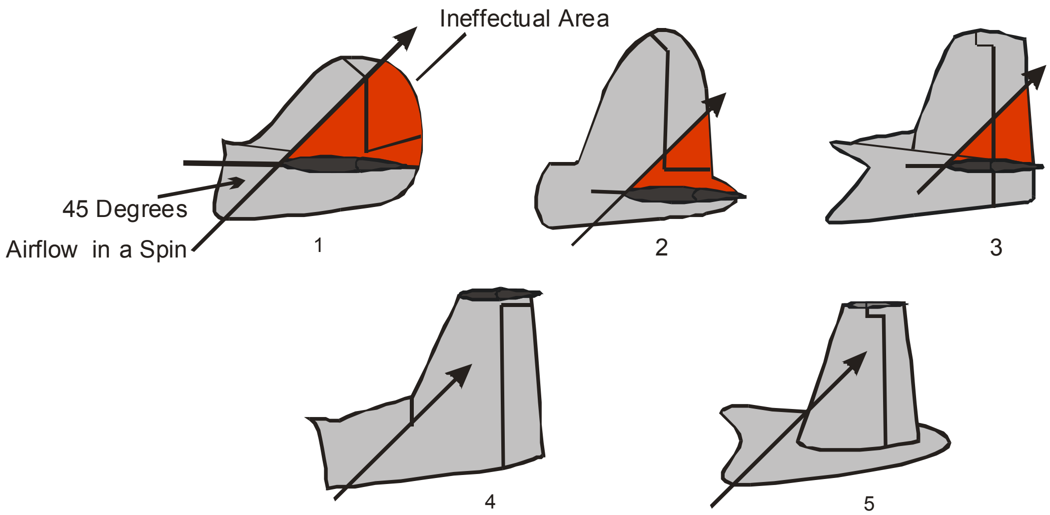


Diagram 4.6 Tailplane Location and Spin on Fin Effective Area

The location of the tailplane can have an effect on the airflow over the fin. In normal flight, this is due to the induced airflow over the tailplane's surface. As shown in diagram 4.6, the location of the tail plane can alter the effective area of the fin if the aircraft enters a spin.

From diagram 4.6:

1. 1920 – 1940 empennage (Spitfire fighter, etc.)
2. 1950 – 1960 empennage (DC 6, etc.)
3. 1960 – onward
4. T-tail
5. V-tail

PROPELLER EFFECTS

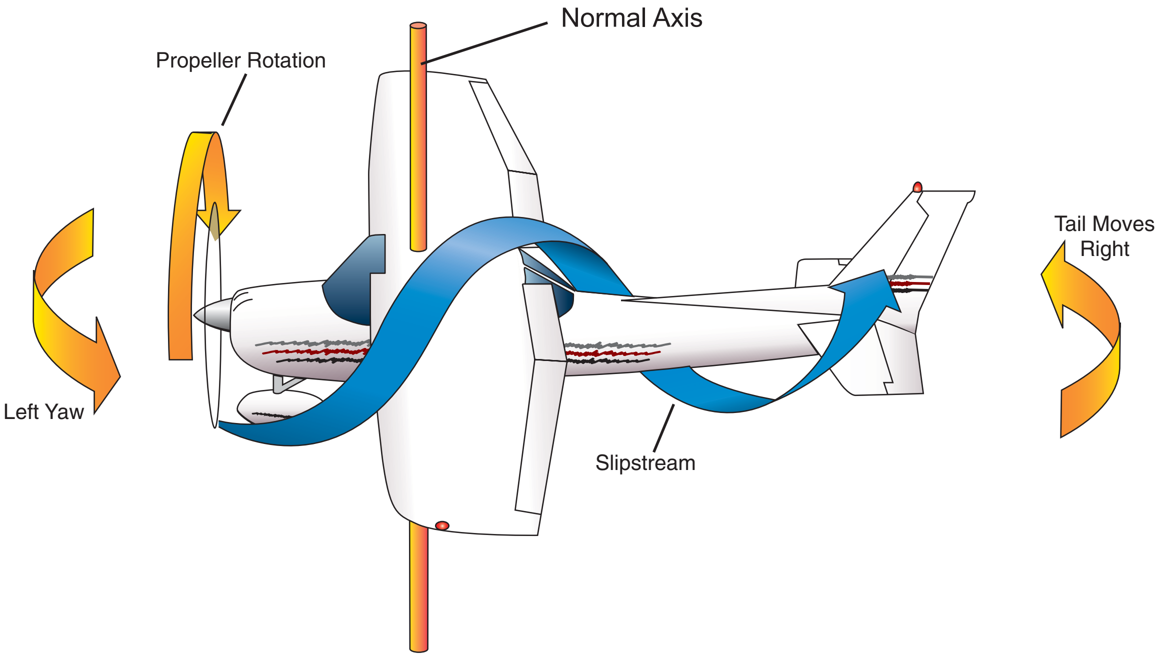


Diagram 4.7 Slipstream Effect on Propeller Wash

For single-engine tractor propeller-driven aircraft, the prop-wash is given a rotation. The effect of this is to cause the prop-wash to wrap itself around the fuselage and impinge on one side of the fin. In high-power situations, such as take-off in a single-engine aircraft, the magnitude and angle at which the prop-wash slipstream impinges on the fin act to push the fin around.

At lower power settings, velocity of the slipstream and the angle of attack that it has on the fin act to alter the airflow over the fin's surface, creating more lift on one side than on the other. The result is that the fin is pulled around in that direction.

To counter this lift, which alters with propeller speed, the designers can:

- Give the fin an asymmetric aerofoil shape
- Make the fin symmetrical and offset it when it is mounted onto the fuselage
- Make the fin symmetrical, mount it fore and aft in line with the aircraft's centre line and leave the pilot to fly with some rudder deflection applied
- Make the fin symmetrical, mount it fore and aft in line with the aircraft's centre line, and fit a fixed trim tab to the rudder, which results in the rudder being deflected in relation to the air flow passing over it

For pusher aircraft, where the propeller is mounted in front of the fin, the slipstream has the same effect as it would when rotating. For pusher aircraft where the propeller is mounted behind the fin, the action of the propeller is to induce the airflow across the fin.

For multi-engine aircraft under normal conditions, the slipstream passing down each side of the fuselage is equal. Therefore, the best aerodynamic shape for the fin is that of a symmetrical aerofoil. This ensures that in normal flight, the lifting forces produced by each surface are equal.

MULTI-FIN DESIGN

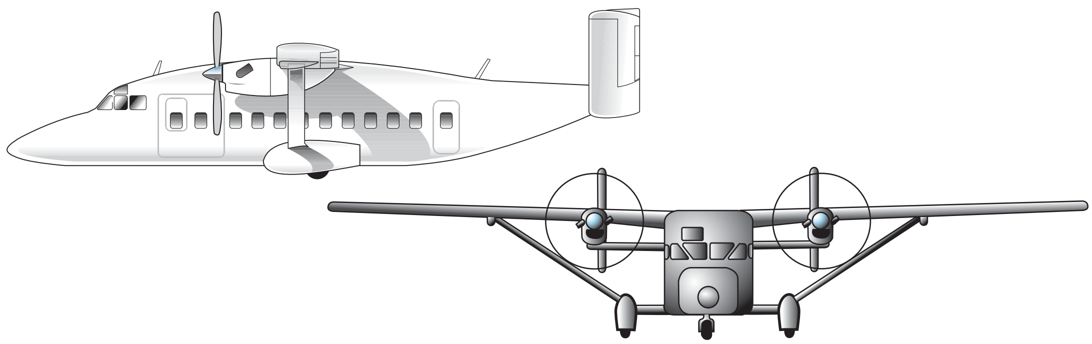


Diagram 4.8 Example of a Multi-Fin Aircraft

To ensure that the aircraft is directionally statically stable, the designer has to take into account the aircraft's operating airspeed and the length between the CG and the proposed location of the fin. The longer the aircraft's fuselage is aft of the CG, the greater the lever arm.

In shorter air transport aircraft, such as the Shorts Sky Van, the keel depth and overall length of the aircraft dictate that a larger fin surface is required to overcome the short lever arm. For practicality, to achieve this, the total area is divided into two sections (some designs use three sections) producing a multi-fin aircraft. In the case of the Sky Van, there is a large vertical fin attached to each tip of the horizontal stabiliser, as shown in diagram 4.8.

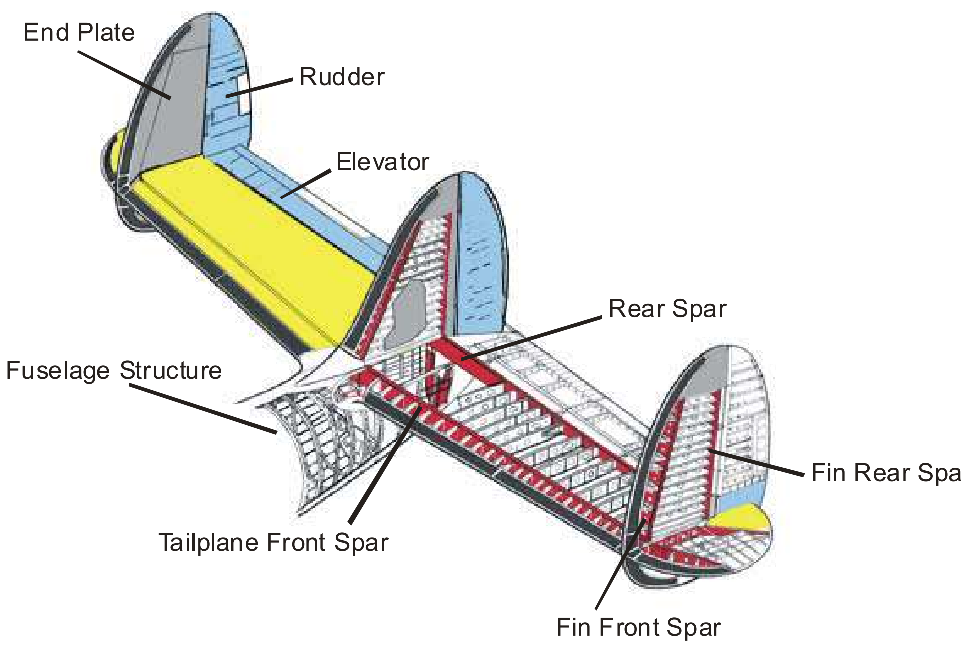


Diagram 4.9 Construction of Triple Fin

In other aircraft, a vertical fin is mounted on the fuselage with end plates mounted at the tips of the horizontal stabiliser. Diagram 4.9 shows the structure of a triple fin where the end plates and fins are mounted inboard of the tailplane tips. As can be seen from the diagram, any loads imposed on these end plates are fed into the tailplane structure and then into the fuselage structure. This complicates the design and increases the overall mass of the empennage.

DORSAL FIN

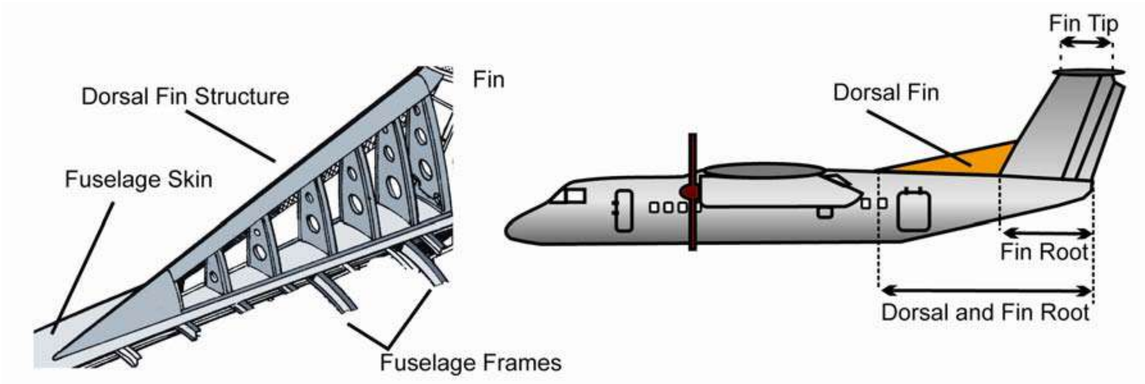


Diagram 4.10 Dorsal Fin

Dorsal fins are extensions running from the fin's lower leading edge to a point on the top surface of the fuselage. They act in two ways. They increase the stability of the fuselage at large sideslip angles, and they reduce the fin's effective aspect ratio, which in turn increases the fin's stalling angle.

COUPLE FORMED BY LIFT AND MASS

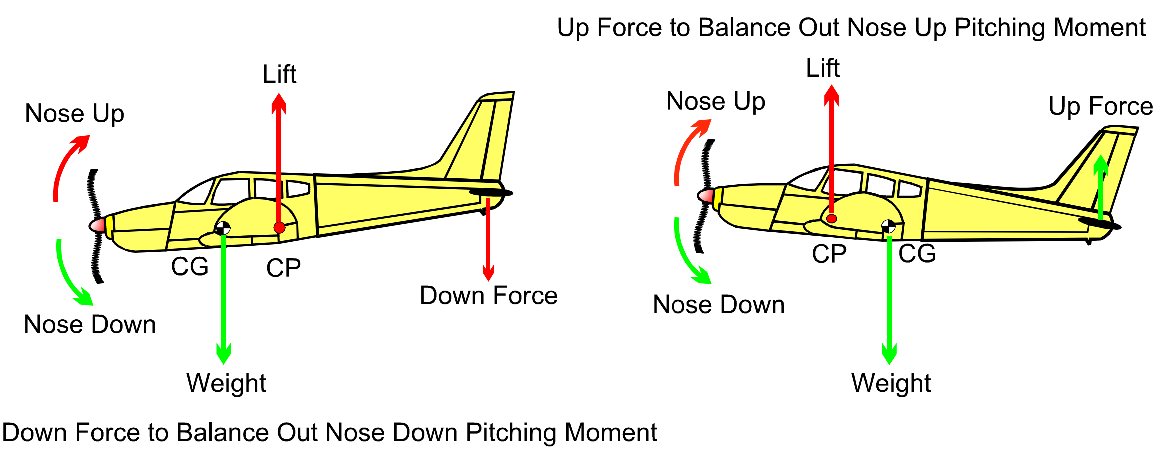


Diagram 4.11 Positive and Negative Pitching Moments

The total lift created by the aircraft's wing acts through the Centre of Pressure (CP). The total weight acts through the Centre of Gravity (CG). These two forces, which oppose each other, are termed a couple. In most cases, the CG and CP are longitudinally displaced from each other, so they create a pitching moment.

In normal flight conditions, the aircraft's CG is in front of the CP, creating a positive pitching moment. This acts to pitch the aircraft nose downward. If the CG were to move behind the CP, a negative pitching moment would be created, which would act to pitch the aircraft nose up.

COUPLE FORMED BY THRUST AND DRAG

The location of the aircraft's engines determines the thrust line. The total drag and the position through which it acts depend on the physical shape and streamlining of the aircraft. For example, if the same model of aircraft is available as either a land plane or float plane, the float plane has its drag line lower than the land plane due to the drag created by the floats.

Where the drag line is above the thrust line, a positive pitching moment is created, which acts to pitch the aircraft's nose upward. If the drag line is below the thrust line, a negative pitching moment is created, which acts to pitch the aircraft's nose downward.

During flight, the CG and CP move in relation to each other, depending on the aircraft speed, flap settings, pitch angle, and changes in CG due to the consumption of fuel. Changes in flaps and landing gear configuration alter the drag value, as increases and decreases of power alter thrust values. The tailplane acts to create a force that counter balances the pitching moments created by any imbalance between these two couples.

TAILPLANE INTRODUCTION

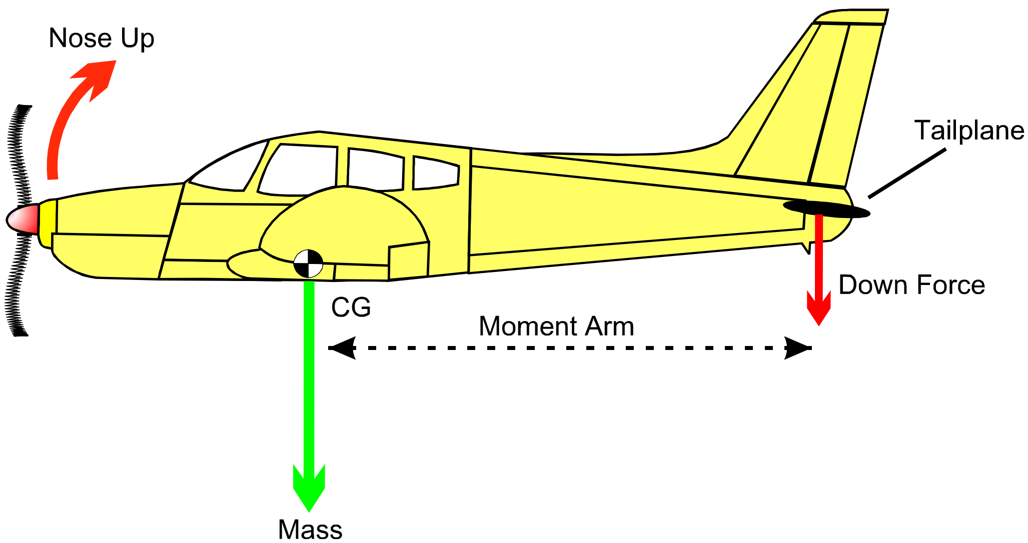


Diagram 4.11A Tailplane Function

The horizontal stabilising surface that is fitted at the rear of conventional aircraft can be referred to as the tailplane or horizontal stabiliser. Conventional aircraft require tailplanes due to the pitching moment created by the four forces of flight — lift and weight, and thrust and drag. The location of these two pairs acts to either pitch the aircraft's nose up or down.

STATIC STABILITY FUNCTION

The function of the tailplane is to provide positive longitudinal static stability. In the event that a gust of wind knocks the aircraft causing it to pitch nose up or down, the aircraft's inertia continues moving it in its original direction. This pitching would alter the airflow over the stabiliser's surfaces creating more lift on one side, bringing the aircraft back to its original attitude.

Sweptback wings add to the aircraft's static stability. When the aircraft's CG is ahead of the CP, this also contributes to the aircraft's static stability. If the CG and CP are co-located, the wings do not provide any static stability. Where the CG is behind the CP, the wings' contribution is unstable.

SIZE OF TAILPLANE

The factors that affect the size of the tailplane are governed by the:

- Length of the mean aerodynamic chord
- Sweepback of the wings
- Turning moment created by the lift from the wings
- Distance from the centre of gravity to the tailplane's location
- Range of movement centre of pressure

From this it can be seen that a comparatively small lifting surface can balance out the pitching moments created by a large wing due to the length of the lever arm.

FORCES ON THE TAILPLANE IN FLIGHT

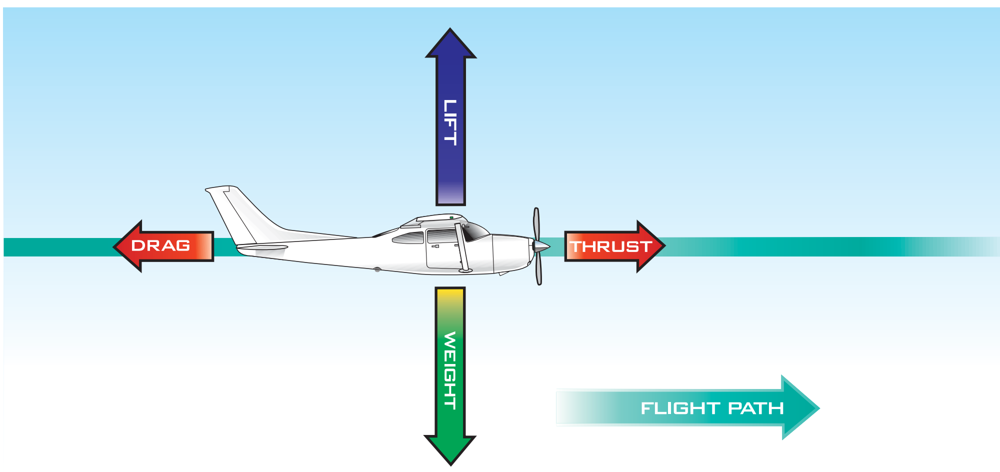


Diagram 4.12 Forces of Flight

If an aircraft were to fly straight and level in un-accelerated flight with all four forces of flight exactly balanced and without the need of elevator trim, there would be no pitching moment about the lateral axis as in diagram 4.12. In this situation, the tailplane has no load on its structure.

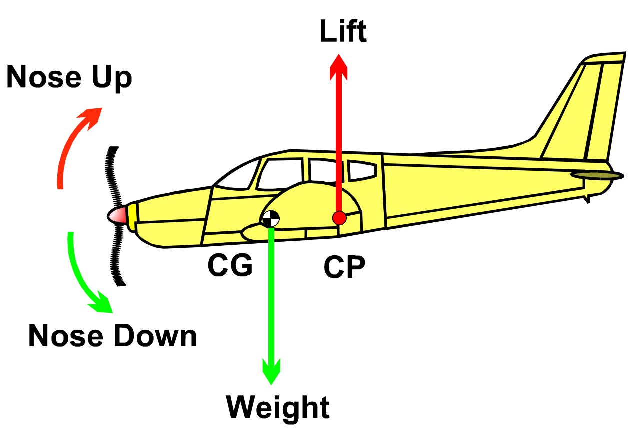


Diagram 4.13 Out of Balance Forces

In some flight conditions (e.g. slow speed), the aerodynamic force produced by the tailplane is insufficient to balance out the pitching moment created by the CP and CG location. In these cases, the elevator is deflected to alter the camber or curvature of the tailplane, thus increasing or reducing the aerodynamic forces.

The elevators are also deflected to enable the aircraft to climb or dive. Any deflection of the elevators creates drag. This is referred to as **trim drag**. Trim drag increases the aircraft's overall drag. Engine power is absorbed as drag is created through the aircraft's movement through the air.

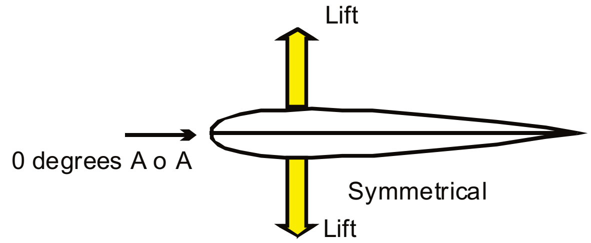


Diagram 4.14 Symmetrical

To achieve this for normal aircraft, the tailplane is manufactured as a symmetrical aerofoil as diagram 4.14 shows. If the angle of attack is zero degrees, then the lift value from each surface will be equal, giving no net lift. To ensure that there is a slight downforce from the tailplane, the leading edge is set approximately 2° down. See Angle of Incidence.

CONSTRUCTION

The tailplane is an aerofoil structure. It consists of spars, stringers, ribs, and skin. The rear spar of the tailplane acts as the hinge attachment point for the elevators. During flight, the tailplane undergoes changes in load. The tailplane is subject to compression and tension loads on both surfaces, as downforce or upforce is required.

H-TAIL

Aircraft that have vertical fins fitted as end plates require the tailplane structure to be strong enough to withstand the side loads imparted by yawing manoeuvres and natural gust loads. The tailplane structure must also withstand the loads imposed on the structure through the operation of the elevators.

LOCATION

Diagram 4.15a Mid-Fin Mounted Tailplane

The tailplane can be located at the:

- Mid-fuselage position
- Base of the fin
- Top of the fin, T-tailplane
- Mid-fin position

Placing the tailplane at either the mid-fuselage position, or base of the fin, simplifies the mounting of the tailplane structure to the fuselage and reduces the need to increase the structural strength of the fin.

T-TAIL

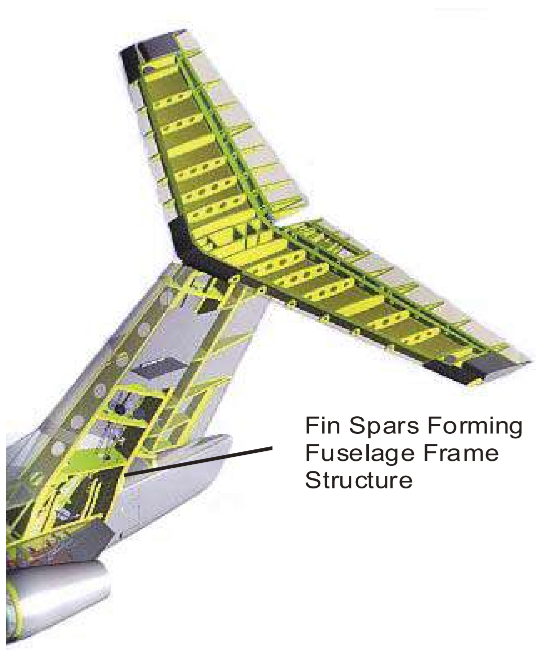


Diagram 4.15 T Tail Construction for the Eclipse 500

Placing the tailplane at the mid-fin position, or on top of the fin, puts the tailplane into less turbulent or free stream airflow. However, it complicates the mounting structure and requires the fin to be structurally strengthened.

PROPELLER WASH AND DOWNWASH

The designer takes into account the airflow across the tailplane's surface and tries to place the tailplane in the least turbulence or **free stream** airflow, where possible. Factors that affect the airflow across the tailplane for a propeller aircraft are:

- Downwash from the wings
- Wash from the propeller(s)

On most single engine propeller-driven aircraft, the tailplane is within the propeller wash unless mounted as a T tail on top of the fin. For some light twin-engine propeller aircraft, mounting the tailplane at the mid-fin position allows the tailplane to be out of the main stream of the prop wash. Mounting the tailplane at the base of the fin/mid fuselage would place the tailplane wholly within the prop wash.

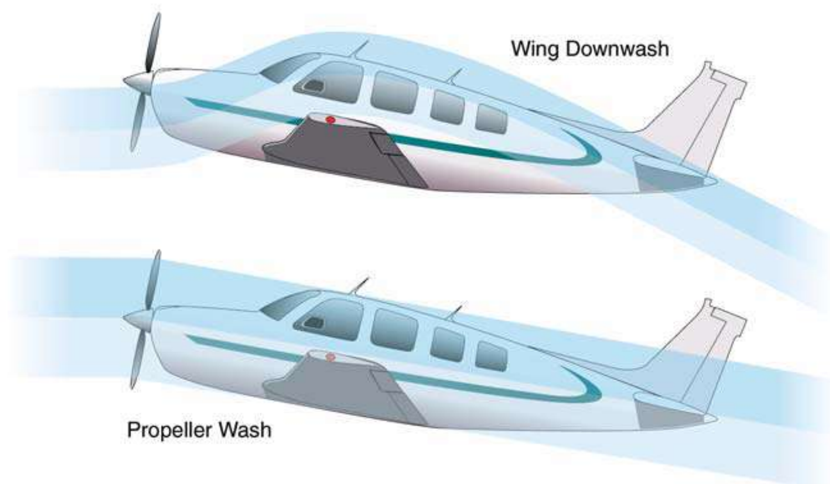


Diagram 4.16 Propeller Wash over Tailplane

Depending on the rpm of the engine and forward speed of the aircraft, the angle at which the prop wash strikes the tailplane alters the lifting characteristics of the tailplane.

Factors that affect the airflow across the tailplane for a turbine aircraft are:

- Jet efflux
- Location of the engines
- Downwash from the wings

One of the considerations taken into account in deciding the location of the tailplane of a turbine-powered aircraft is the temperature and velocity of the jet efflux. Hot gasses impinging on the tailplane could lead to structural damage.

Aircraft such as the MD 80, Eclipse 500, Galaxy, and Cessna Citation that have fuselage-mounted engines have their tailplanes mounted as either T-tails or mid-fins. For turbine aircraft where the engines are mounted under the wings, a mid-fuselage or base of fin-mounted tail plane is clear of the hot jet efflux. However, the velocity of the jet efflux passing under the tailplane can suck the free stream airflow with it. This alters the angle of attack at which the air strikes the leading edge of the tailplane. The amount of alteration is in proportion to the power setting.

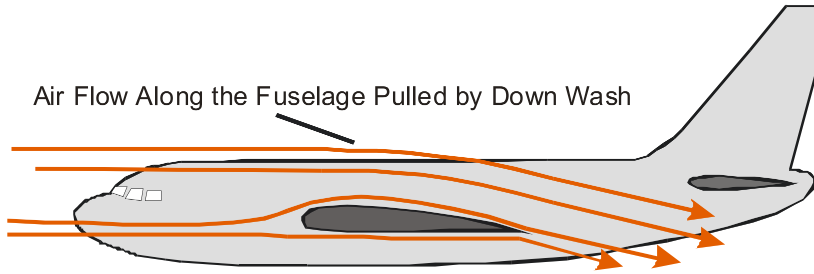


Diagram 4.17 Down Wash from the Wings

The other consideration is the effect of the downwash from the aircraft's wings. When the air passing over the upper surface of the wings has a greater velocity than that of the air passing underneath the wing, the two streams are deflected downward as they meet at the trailing edge. This is called downwash. It draws the free stream air down with it, which in turn alters the air's angle of attack on the tailplane.

ANGLE OF INCIDENCE

Where the tailplane is fitted within the slipstream effect of a propeller or the wing, the airflow onto and over the tailplane's surfaces is affected by the angle and velocity at which the slipstream hits the leading edge.

To overcome this; the tailplane can be fitted to the aircraft with its leading edge slightly below the horizontal. This is normally half the angle of incidence for the wings. This smaller angle, the tailplane's **angle of incidence** or **riggers angle**, is set to ensure that when the aircraft is balanced in the cruise, the load on the tailplane is minimal.

It also ensures that if a gust pitches the aircraft's nose upward or downward, the tailplane is able to restore the aircraft to the balanced condition. This is due to the angle of attack of the air on the tailplane acting over the lever arm to the CG.

DELTA-WING AIRCRAFT

Delta-wing aircraft do not require a tailplane to counter the pitching moments created by the interplay of lift and weight. The pitching moments are balanced within the one aerodynamic surface, as these wings have their trailing edges much further aft than a conventional wing.

V-TAIL

Diagram 4.18 V-Tail

Some light aircraft with conventional wings make use of a less conventional empennage in the form of the V-tail. The vertical fin and horizontal tailplane are replaced by two aerofoil surfaces mounted in the form of a V. These are constructed in the same manner as conventional fins/tailplanes and act to give the aircraft directional and pitch control.

CANARDS

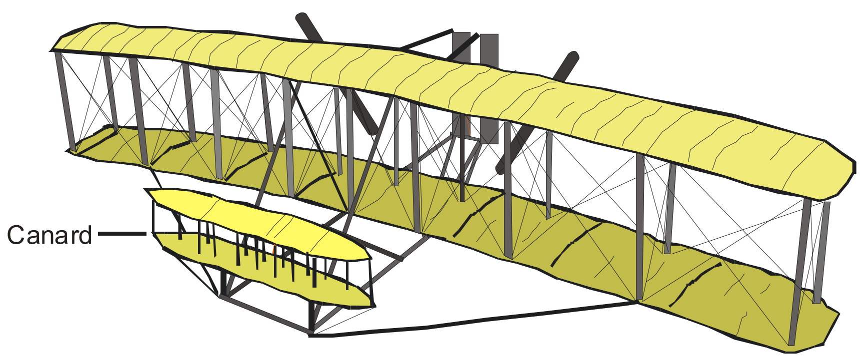


Diagram 4.19 Wright Flyer

Canard type aircraft do not require a conventional tailplane. In this design, the horizontal tailplane is replaced with horizontal surfaces mounted in front of the main wing. The Wright Brothers' Kitty Hawk Flyer, apart from being the first recorded powered aircraft, was a canard design. This design allowed the aircraft to pitch its nose up without having to lower a tail section.

Aircraft designed with canard surfaces have their main wings located toward the rear of the fuselage, with a canard or fore plane surface mounted ahead of the main wing. The rearward location of the main wing allows more of the fuselage to be used as cabin space when the wing is mounted at the mid-fuselage position.

Unlike the conventional tailplane, both the canard surface and the wing of the aircraft create an upward lifting force under all normal conditions of flight. The weight of the aircraft is shared over both surfaces, allowing a lighter wing loading to be obtained and a lighter structure to be used.

One of the effects of mounting a small wing forward of the main wing is that the lift from the canard counters the negative pitching moment of the wing. Canards are mounted with a slightly larger angle of incidence than that of the main wing. The result is that canards have a greater AoA than the main wings. Creation of a positive pitching moment by the wing lifts the nose of the aircraft, further increasing the AoA of the canard. If the pitching continues, the canard stalls before the wing stalls.

Diagram 4.20 Piaggio Avanti P180

While canards are not currently used in place of tailplanes on commercial airliners, several commuter type aircraft such as the Piaggio Avanti built under JAR 23 make use of both canards and a conventional tailplane. The advantage of this system is that all the horizontal surfaces become upward lifting surfaces in flight, allowing a lighter wing loading and a greater speed due to the increased surface area.

Chapter 5
Primary and Secondary Flight Controls

ACKNOWLEDGEMENTS

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For Diagrams

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INTRODUCTION

Aircraft have the ability to manoeuvre three dimensionally in flight. These manoeuvres are termed pitching, rolling, and yawing. This chapter examines the control surfaces and systems by which the pilot of an aircraft can control its flight. To assist students reading these notes before the aerodynamics notes, basic explanations of aerodynamic effect are added to explain how structural components achieve their function.

THE REFERENCE AXES

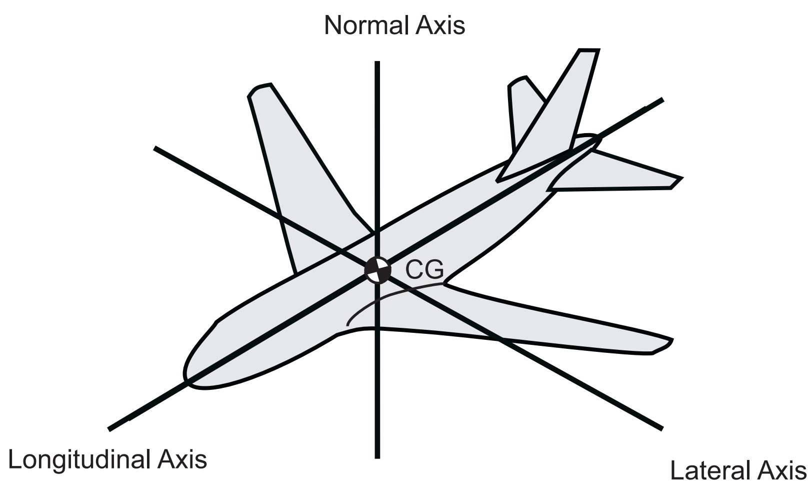


Diagram 5.1 Reference Axes

All three axes pass through the Centre of Gravity, the point through which the aircraft's total mass acts vertically downward, as diagram 5.1 shows.

The three axes or reference axes are:

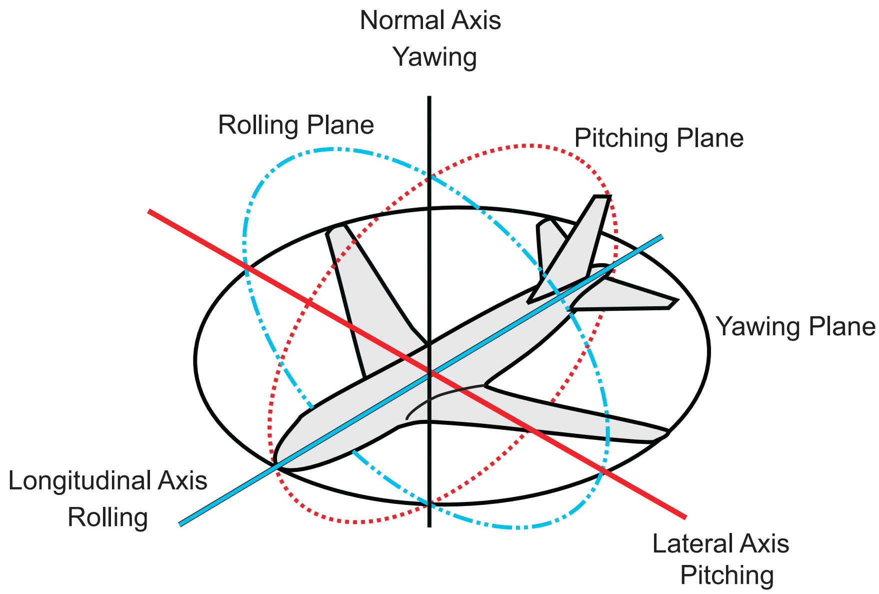


Diagram 5.2 Axes and Planes of Rotation

Longitudinal axis is a line that passes through the nose of the aircraft and exits through the tail. Movement about the longitudinal axis is termed rolling. It can also be referred to as the roll axis. Refer to diagram 5.2.

Lateral axis is a line that passes through the wing tip of one wing and exits through the other. Movement about the lateral axis is termed pitching. It is also referred to as the pitch axis. Refer to diagram 5.2.

Normal axis is a line that passes through the underside of the fuselage and exits through the top of the aircraft. Movement about the normal axis is termed yawing. It can also be referred to as the yaw axis. See diagram 5.2.

PRIMARY CONTROL SURFACES — CONVENTIONAL AEROPLANES

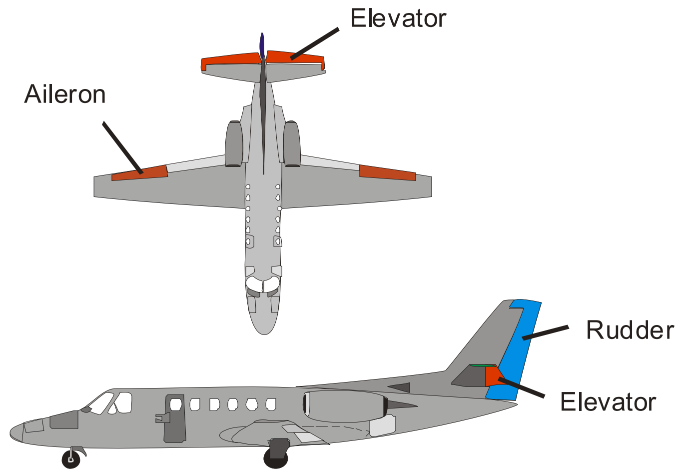


Diagram 5.3 Primary Control Surfaces on a Conventional Aircraft

Refer to diagram 5.3 for the control surfaces of a conventional aircraft with wings, fin, and tailplane. The following are control surfaces:

Ailerons act about the longitudinal axis to roll the aircraft in the rolling plane. They are aerodynamic surfaces attached to the trailing edge of the outboard section of the wings.

Elevators act about the lateral axis to pitch the aircraft in the pitching plane. They are aerodynamic surfaces attached to the trailing edge of the tailplane.

Rudders act about the normal axis to yaw the aircraft in the yawing plane. They are aerodynamic surfaces attached to the trailing edge of the fin.

FUNCTION OF THE PRIMARY CONTROL SURFACES

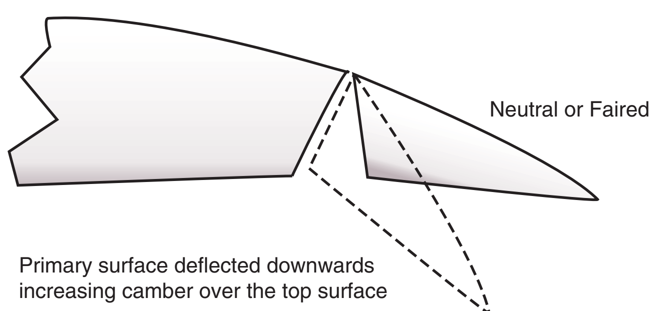


Diagram 5.4 Action of Primary Control Surfaces

Termed the **primary flight controls** or **primary controls**, each of these surfaces is effectively a small, hinged aerofoil capable of an angular movement, **deflection**.

Any deflection alters the original shape or contour of the aerofoil to which the primary surface is attached, as in diagram 5.4.

RANGE OF MOVEMENT

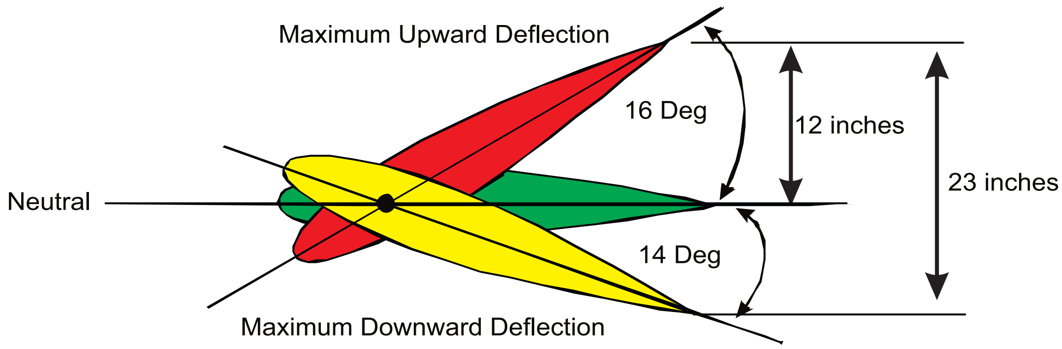


Diagram 5.5 Primary Control Surface Range of Movement

By altering the shape, the **camber** of the aerofoil, more lift is created on one side of the aerofoil, causing the aerofoil surface to move in the direction of the extra lift. The angular difference between full deflection in one direction, to full deflection in the opposite direction, is termed the control's **range of movement** or ROM. This can be expressed as a linear distance measured at the trailing edge. Refer to diagram 5.5.

STRUCTURE

The aerofoil cross-section of a control surface normally continues the cross-section profile of the aerofoil to which it attaches. When the control surface's profile exactly matches the main aerofoil so that there is no deflection in either direction, the control surface is **neutral** or **faired**. Refer to diagram 5.4.

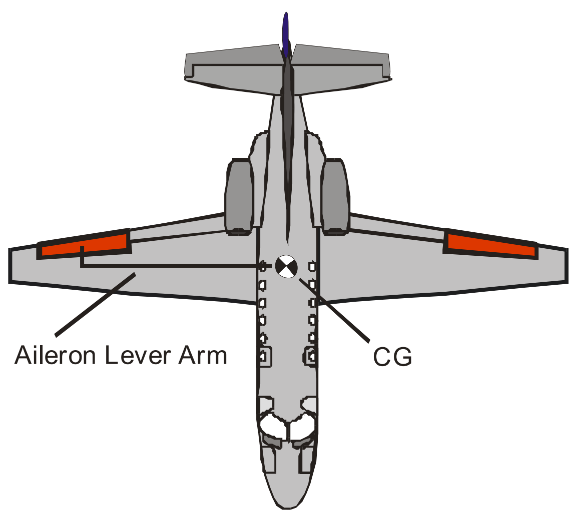


Diagram 5.6 Control Surface Lever Arm

The primary control surfaces are small in relation to the aerofoil surface to which they attach. The effect that they have is due to the length of the lever arm between their location and the centre of gravity. This allows a small angular deflection to create sufficient force to cause the aircraft to rotate about the reference axes. The exact size depends on the distance between their location and the aircraft's CG and the design speed of the aircraft. Diagram 5.6 illustrates this.

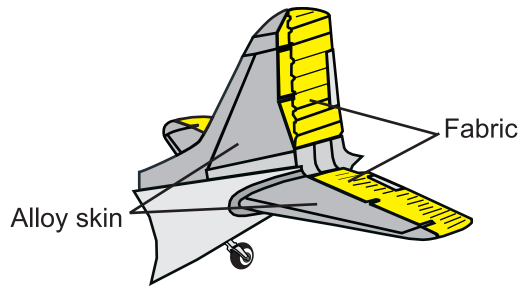


Diagram 5.7 Fabric Cover Frame for Elevators and Rudder

On slow speed aircraft where the air loads on the control surfaces are not very high, the primary and secondary control surfaces are manufactured as a frame of ribs, stringers, and spars covered in doped fabric. The Dakota is a classic example of this structural design. Sheet alloy replaces doped fabric for aircraft that have higher control surface loadings or if the designer wishes to use alloy, as shown in diagram 5.7.

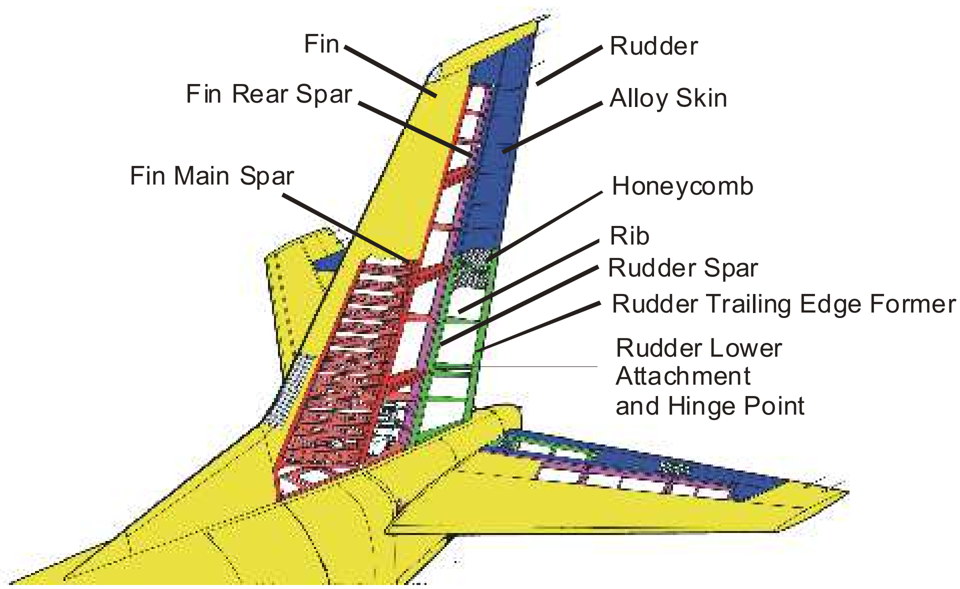


Diagram 5.8 Structural Parts of a Large Aircraft's Primary Control Surfaces

For high-speed aircraft and those with very high control surface air loads, such as the larger air transport aircraft, the control surfaces are manufactured by encapsulating a pre-formed shape cut from honeycomb. This increases the rigidity of the whole control surface without incurring a weight penalty. Whichever method is used, the structure is strengthened at the hinge attachment and control input points.

FLIGHT DECK CONTROLS

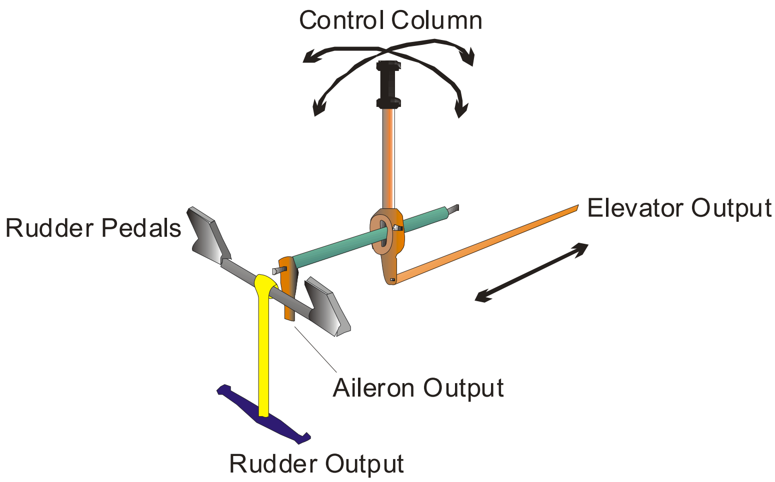


Diagram 5.9 Control Column and Rudder Pedals

The primary flight control surfaces link to the primary controls in the flight deck. These vary depending on the age, size, and complexity of the aircraft. On older light aircraft and fighters, these take the form of control column and rudder pedals, as in diagram 5.9.

CONTROL COLUMN

In this system, the elevators are operated by fore and aft movement of the control column, which is also referred to as the **stick**. Forward movement of the control column deflects the trailing edge of the elevators downward, altering the camber of the tailplane. This creates more lift on the top surface (low pressure), raising the tail of the aircraft and causing the nose to pitch down and vice versa.

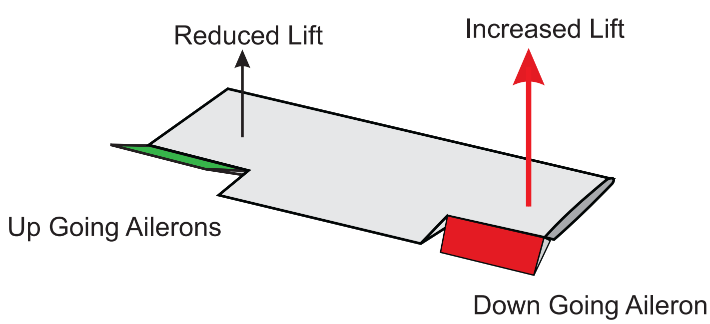


Diagram 5.10 Aileron Action

Side-to-side movement of the control column operates the ailerons. When the pilot moves the stick to the left, the left aileron deflects upward, reducing the lift created by the left wing. The right aileron deflects downward, increasing the lift created by the right wing, causing the aircraft to roll to the left and vice versa.

When the control surfaces are in the neutral position with the stick-type control column, the column is centralised. If the pilot moves the column diagonally, or at any angle other than straight fore and aft or side-to-side, the ailerons and the elevators move simultaneously.

RUDDER CONTROL

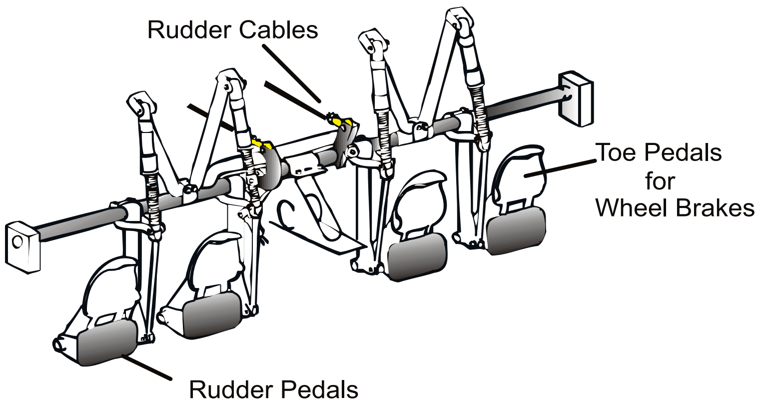


Diagram 5.11 Conventional Rudder Pedals

The pilots control the yawing action of the aircraft by pushing rudder pedals with their feet. Relaxing the pressure with the right foot and pushing forward with the left foot deflects the trailing edge of the rudder to the left, which alters the camber of the fin and the lift created over the right surface. The increase in pressure over the left surface acts to move the tail to the right, so the nose of the aircraft yaws to the left. When the rudder control surface is neutralised fore and aft with the fin, the rudder pedals are centralised.

While technology has altered the method by which the pilot makes aileron and elevator inputs from the control column (joystick) to fly-by-wire side stick, rudder controls remain basically the same for all conventional aircraft as pedals.

PUSH- PULL TYPE YOKE

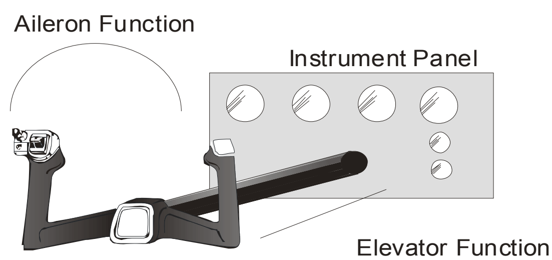


Diagram 5.12 Push Pull Type Control Wheel

For many modern light aircraft, the stick type control column has been replaced by the through instrument panel push/pull yoke or control wheel. In this design, the yoke mounts on a tube that passes through the instrument panel. Pulling the yoke back raises the elevator’s trailing edge and causes the aircraft to pitch nose up. Turning the yoke to the right raises the right aileron and lowers the left aileron, causing the aircraft to roll to the right and vice versa. Refer to diagram 5.12.

CONTROL WHEEL AND COLUMN

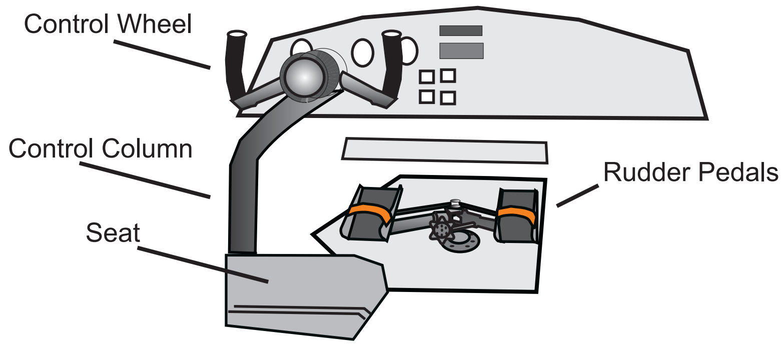


Diagram 5.13 Control Wheel and Control Column

For larger aircraft, a control column combined with a control wheel has replaced the push-pull type. The column acts to alter the elevators, and the control wheel alters the ailerons, as shown in diagram 5.13.

SIDE-STICK

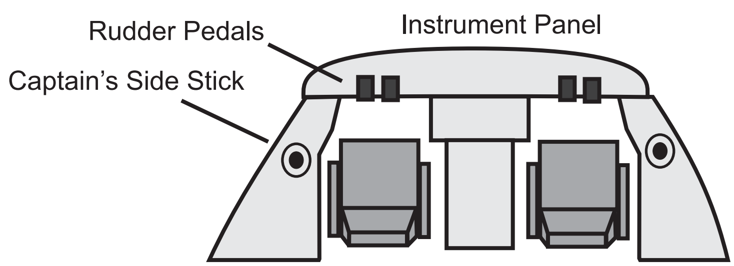


Diagram 5.13A Side Stick Flight Deck

Airbus has replaced the conventional control column and control wheel with a **side-stick** in their aircraft, making use of computers and fly-by-wire technology. As visible when comparing diagrams 5.13 and 5.13A, side-sticks provide the pilots more room as the bulk of the control column is removed. The side-stick acts in a similar manner to the original control column. This is covered more fully in the fly-by-wire topic covered later in this chapter.

CONTROL SYSTEM LINKAGE

For lighter aircraft and aircraft with slow airspeeds, the control surfaces, while being proportionately larger than those of faster aircraft, have less air loads acting on them when the control surface is deflected. This enables the designer to use a direct mechanical linkage between the controls and the control column/rudder pedals.

The types of direct mechanical linkage are control cables and control rods.

CABLE TENSION AND TURNBUCKLES

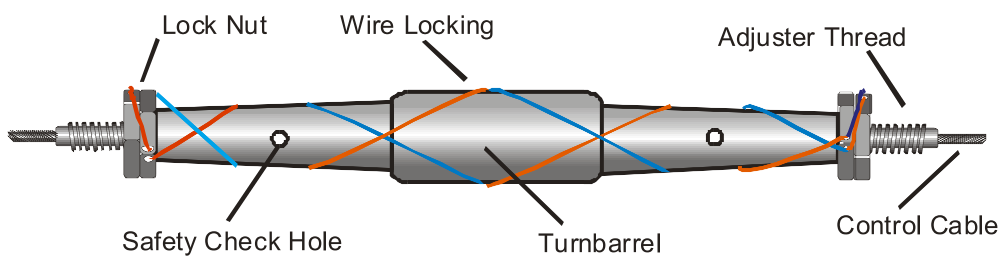


Diagram 5.14 Turnbarrel

To ensure that the pilot's control inputs are instantaneously transferred to the control surfaces in systems where control cables are used for primary or secondary control operation, the cable system is given a pre-tension. This tension is adjusted by the use of turnbuckles (turnbarrels), which are locked to ensure that they do not self adjust with the cable's operation. Diagram 5.14 shows this.

If the control cable system loses its tension and becomes slack, the control surfaces can make uncommanded movements and the system suffers from backlash.

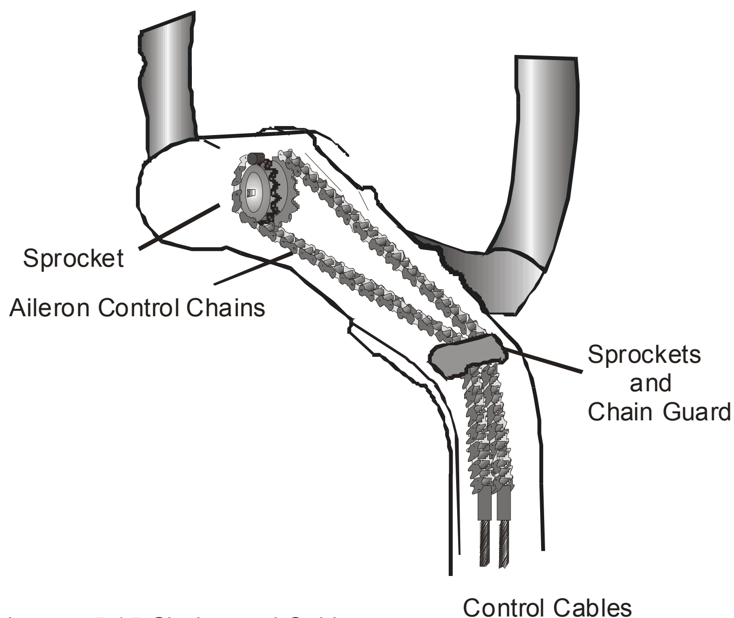


Diagram 5.15 Chains and Cables

In a cable system where both inputs and outputs require a rotary motion, the cables attach to a chain passing over a sprocket. Refer to diagram 5.15 showing a control wheel system.

CONTROL CABLES

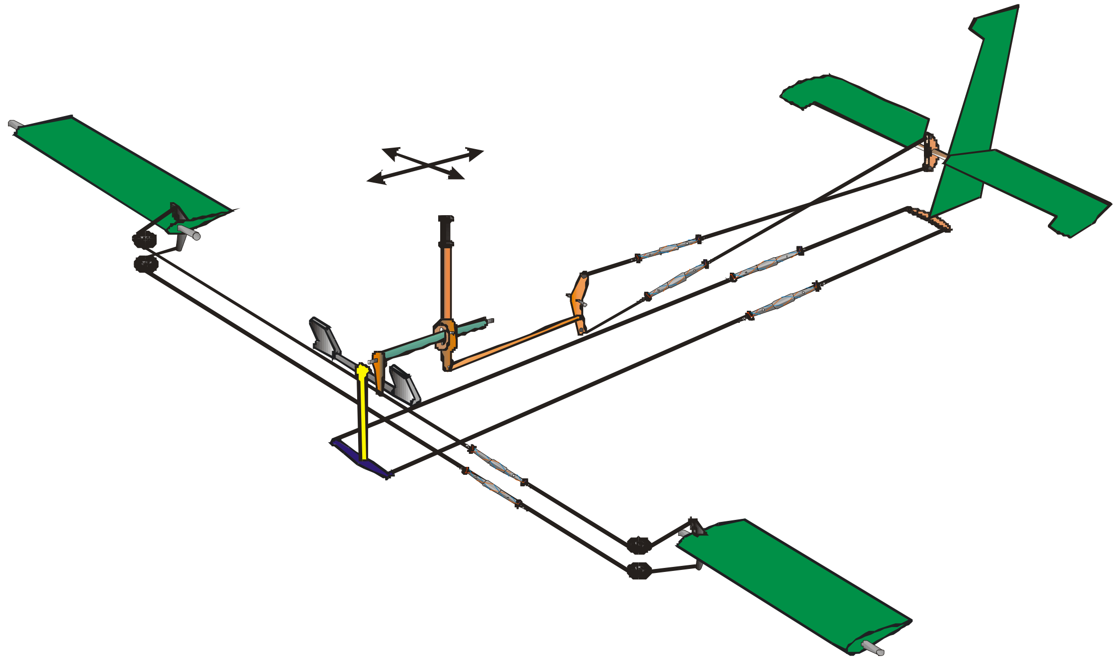


Diagram 5.16 Cable Operated Controls

Two control cables to the input control column/rudder pedals link each control surface, as cables can only operate in tension as in diagram 5.16.

When the pilot makes a control input, the cable under tension moves the control, and the second cable follows the movement of the control surface. When the reverse input occurs, the function of the cables reverses. Refer to diagram 5.16.

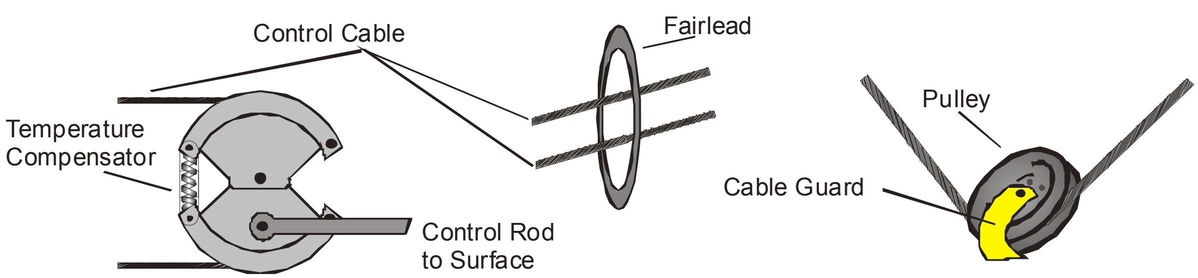


Diagram 5.16A Temperature Compensator, Fairlead and Pulley

For larger aircraft where the expansion and contraction of the fuselage and wings causes them to grow or shrink by several inches, cable systems require a thermal or spring compensator to be fitted. See diagram 5.16A for an illustration. This ensures that the pre-determined cable tension is maintained in the cable run.

To ensure that cables and airframe structures are not damaged by the back and forth movement of the cable, **fairleads** are fitted where the cable run passes through or by structure, such as frames. The JARs determine that fairleads must be installed, so that they do not cause a change in cable direction of more than 3°. It is normal practice for the cable to be routed through an enlarged aperture where fairleads of a softer material than the cable are fitted. When correctly fitted, the cable does not contact the fairleads, as diagram 5.16A shows.

To ensure that cables change direction smoothly and the system creates the minimum friction possible, pulley wheels are used. Guards are fitted to ensure that the cables and chains do not jump out of the pulley wheels and sprockets, see diagram 5.16A.

The pilot gains mechanical advantage by having the control cables fitted to the base of the control column and by lever horns attached to the rudder pedals. In some aircraft, the control cables directly attach to the control surfaces. For those aircraft where the air loads are slightly higher, the cables attach to bellcranks, and a control rod links the bellcrank to the control surface. See diagram 5.16 and 5.18. This system increases the pilot's leverage for a slight reduction in movement.

CONTROL RODS

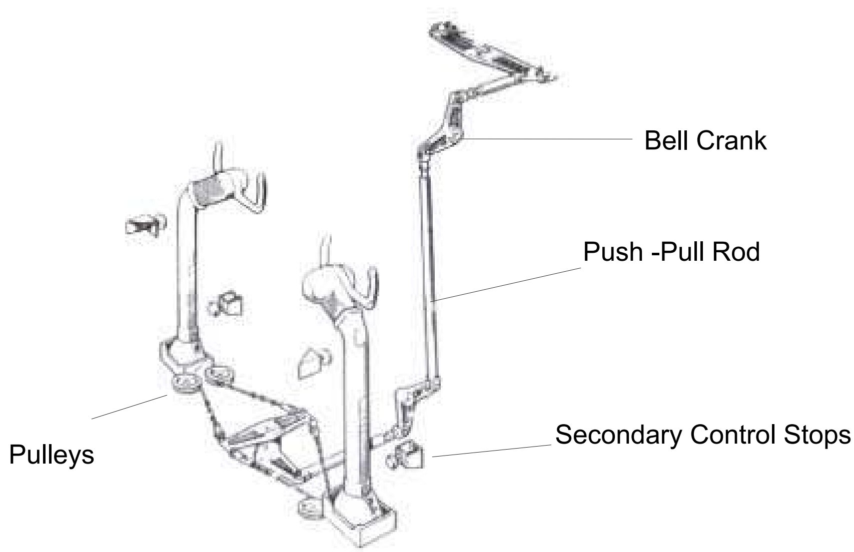


Diagram 5.17 Use of Control Rods in a System

The control rod system makes use of a tube that can withstand both compression and tensile strains, thus allowing a single rod to operate a control in both directions. Again, where rods pass through structure, the fairlead system is used. To support the rods along their length, idler levers can be used. When a control run using rods has to change direction, lever bell cranks are used, as can be seen in diagram 5.17. To compensate for thermal expansion and contraction, a **spring strut** can be inserted in the run.

CONTROL STOPS

Primary and secondary stops are fitted to prevent the pilot from inputting a control movement that is greater than the control surface's range of movement. This would either damage the surface or the linkages.

Primary Stops

Primary stops are located at the control surface end of the control run. These are adjusted to give the full range of movement for the surface, and then locked, so they do not alter. Diagram 5.18 shows the two primary stops limiting the range of movement for an elevator.

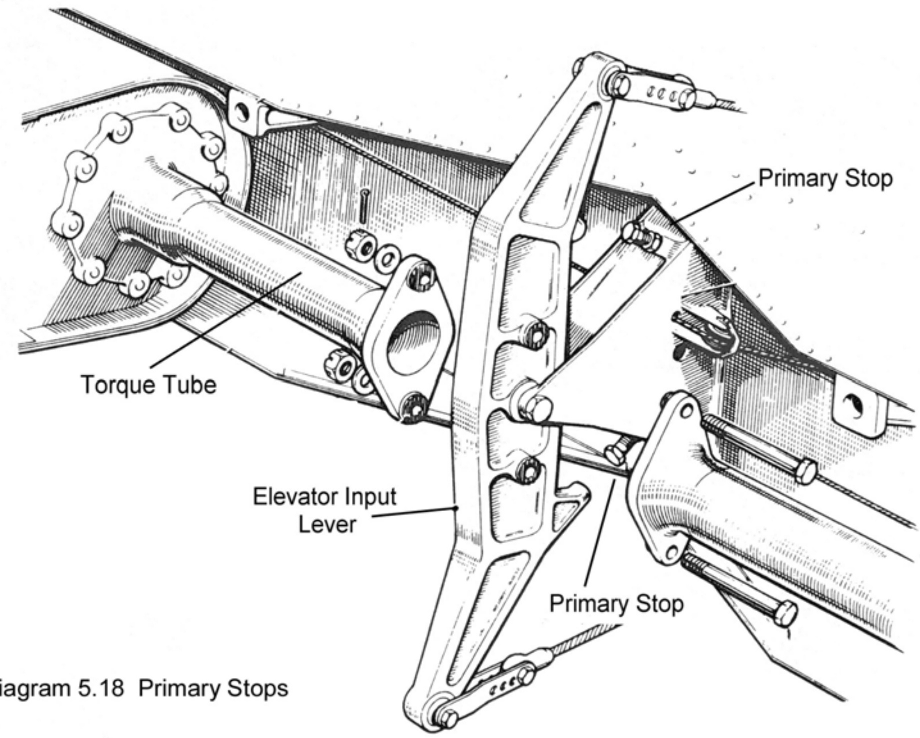


Diagram 5.18 Primary Stops

Secondary Stops

Secondary stops are located at the input end of the control system. These are adjusted so that there is a small clearance between them and the control system when the control surface is fully deflected, then locked.

These stops ensure that the pilot has full range of movement of the control. Should the pilot try to force the control column/rudder pedals beyond the control surface's limit, a physical stop prevents damage to the control linkages. Refer to diagram 5.17 for secondary control stops. These are not normally seen, as they fit below the flight deck floor.

HYDRAULIC CONTROL SYSTEMS

With the advent of faster aircraft and increased air loads over the control surfaces, hydraulic power was utilised to overcome the forces that would otherwise require excessive mechanical linkages. These systems take the form of Power Assist and Power Control Units.

POWER ASSIST OR BOOST

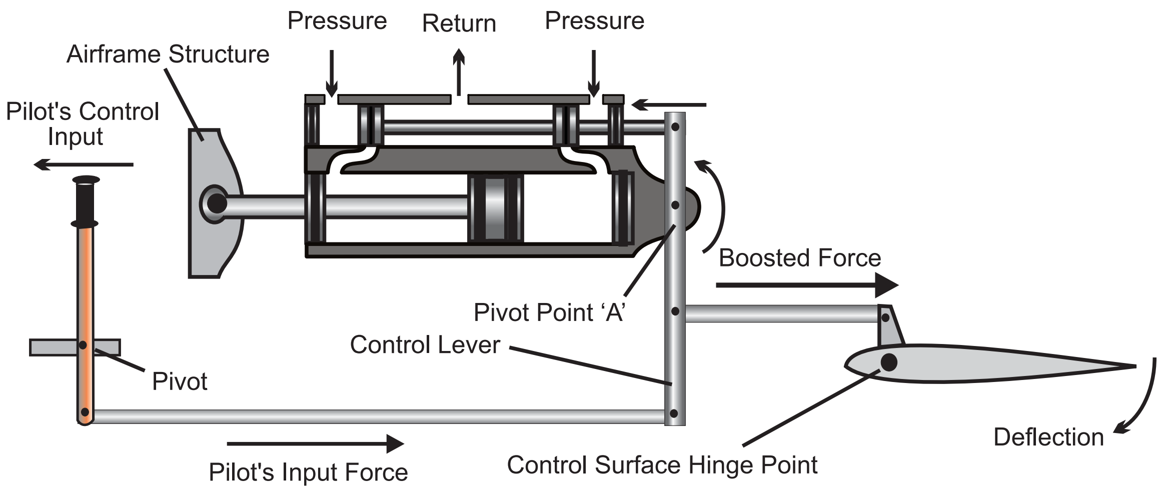


Diagram 5.19 Power Assisted Control System

If purely manual control systems were used on the faster and heavier aircraft, pilots would need to apply excessive control forces, so systems have been developed to help the pilot move the flight controls.

The first method of overcoming these forces is called **power assist** or **boost** control systems. In the power assist system, the pilot is still directly linked to the control surface. However, operating in parallel to the pilot's input is a hydraulically actuated **servo unit**. The ram of the servo unit is attached to the aircraft's structure, see diagram 5.19 above. The body of the servo unit is attached, via a control lever, to the control surface input. The control lever is also attached to a servo valve.

When the pilot makes a control input, muscle power initiates movement of the control surface. At the same time the control lever pivots about its joint to the servo unit's body [see pivot point A above] and displaces the servo valve. Movement of the servo valve ports hydraulic fluid at system pressure to one side of the servo unit's piston. The control rod linkage to the control lever becomes a pivot point and the servo unit moves the control lever increasing the force applied to the control surface, assisting the pilot to move the control surface.

When the pilot centralises the control, the servo unit works in reverse and neutralises the flying control surface. In this system, the pilot still provides a percentage of the power required to operate the control surfaces and through the control column/rudder pedals feels the effect of the air loads (airspeed x control deflection).

In the event of a hydraulic failure, the power-assist servo unit remains stationary, and the pilot reverts to direct manual operation of the controls. This system is not capable of handling the increased hinge moments created by large aircraft with bigger control surfaces flying at increasing airspeed and altitude. This is termed the **Mach effect**.

POWER CONTROL UNITS

On larger and faster air transport aircraft, the air loads acting on the control surfaces are greater than the force that a pilot and boost can feasibly apply. To gain mechanical advantage and operate the controls proportionately to the pilot's input, a hydraulically powered control unit (PCU) is used. This is fitted between the control surface input and the pilot's input. In this control system layout, the pilot's input operates a valve, which allows aircraft system hydraulic pressure to provide the motive force in moving the control surface.

PCUs physically attach to the aircraft's structure and the control surface input linkage. The pilot's control inputs operate a servo valve within the PCU. The PCU moves proportionately to the servo valve.

As there is no direct connection between the pilot and the control surface, failure of the PCU or loss of hydraulic pressure renders the control inoperative. To overcome this, the JAA regulations require a system of redundancy. This is normally achieved by having more than one PCU attached to the control surface, with each PCU operated from separate hydraulic power sources. This allows a failed PCU or a failed hydraulic system to become redundant.

PCU SCHEMATIC

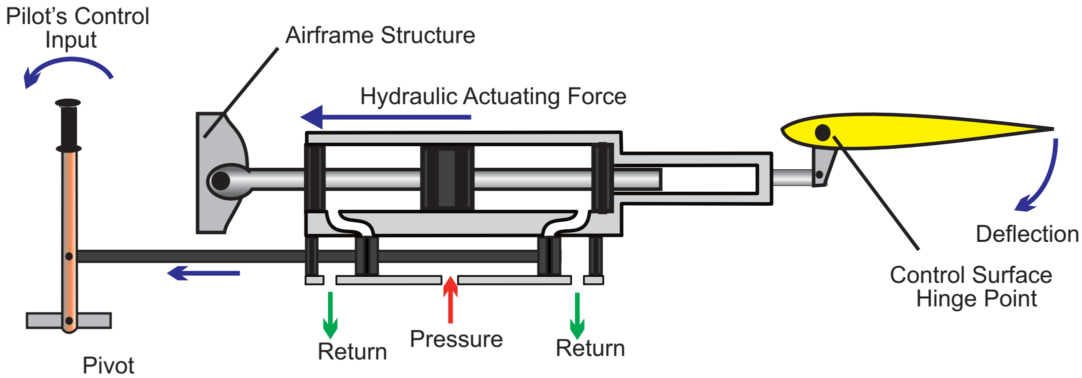


Diagram 5.20 Schematic of a Basic PCU

Diagram 5.20 is a schematic of a light PCU. To ensure proportionate movement, one end of a balanced double acting ram is attached to the aircraft structure. The PCU's body is attached to the input control of the control surface. The pilot's linkage is attached to a servo or spool valve. When the pilot's control is in the neutral position, the spool valve is also neutral. This locks off both the hydraulic pressure supply and fluid return from the ports leading to each side of the piston.

Considering the elevators, when the control column is moved forward (shown as left in the above diagram), the spool valve is deflected, opening the left port to system pressure and the right to return. As the ram is attached to the fuselage structure (it remains stationary), the body of the PCU moves to the left, giving an input to the control surface.

As the body is following the input from the servo valve, when the pilot stops moving the control column, the body rides over the stationary servo valve and blocks the ports. This is termed PCU follow up action, which traps fluid on each side of the piston to create a hydraulic lock.

PCU RELEASE UNIT

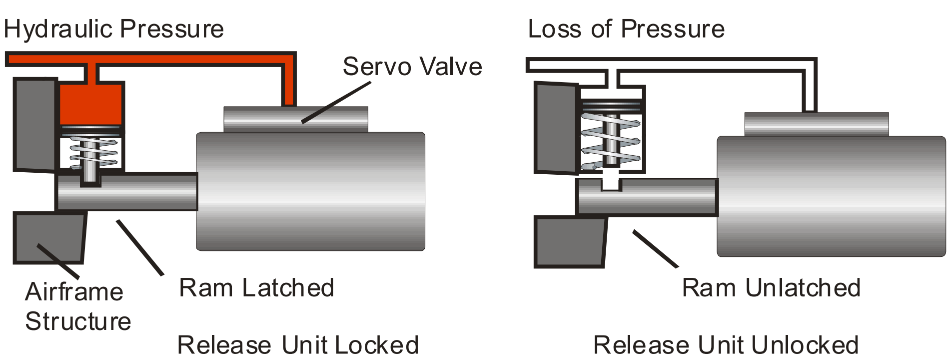


Diagram 5.21 PCU Release Unit

To allow a failed PCU to move freely and not impede the operation of serviceable PCUs attached to the same control surface, release units are used. In this design, the release unit is a single acting actuator, as in diagram 5.21. It consists of a piston-operated pawl and release spring mounted in a housing attached to the aircraft structure. Instead of directly attaching the ram to the structure, it passes through an orifice in the release unit. When hydraulic power is applied, the piston forces the pawl into a detent in the ram. If hydraulic pressure is lost or removed, the spring acts to disengage the pawl, allowing the PCU to move backward and forward freely.

PCU WITH INTERNAL INTERCONNECTING VALVES

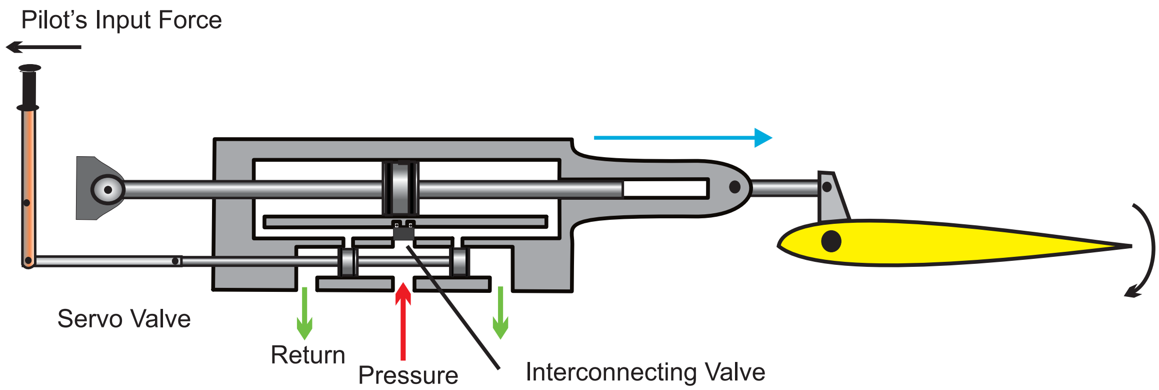


Diagram 5.22 PCU with Interconnecting Valve

Referring to diagram 5.22, in this design, the PCU's ram directly connects to the aircraft's structure, the body to the control surface. The control input is to a servo valve. Hydraulic pressure into the PCU servo valve acts on to an interconnecting valve, pushing it down against a spring, so that the interconnecting valve blocks a bypass passage that links one side of the PCU piston to the other. This isolates the two sides of the piston. Any deflection of the servo valve results in movement of the control surface.

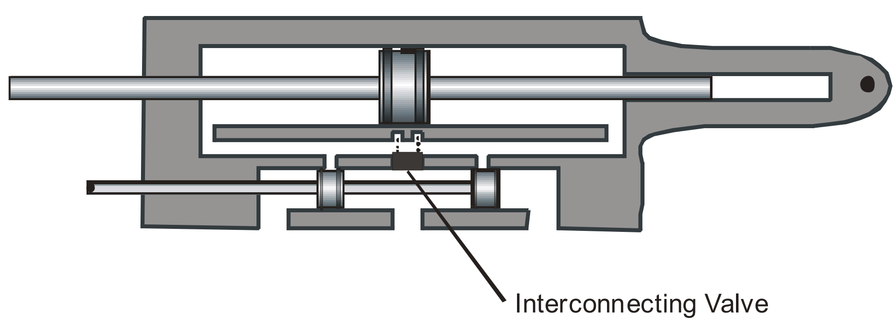


Diagram 5.23 PCU with Interconnecting Valve Open

In the event of a hydraulic failure, the spring pushes the interconnecting valve down. This allows both sides of the piston to become interconnected, breaking the hydraulic lock formed by the servo valve.

On light aircraft control systems where there is one PCU per control surface, the operation of the interconnecting valve allows the pilot to gain manual reversion. In this case, movement of the control column allows the pilot to physically move the PCU body with the servo valve. For air transport aircraft, where the force required to operate the control would be too great for manual reversion, there is more than one PCU per control surface. In this case, the redundant PCU floats freely.

PRIMARY CONTROL SURFACE DESIGN AND DEVELOPMENT

The Wright Brothers' aircraft and other early aircraft made use of wing warping to alter the camber of the wings to roll the aeroplane. This was soon found to be inefficient, and a section of the wing's outboard trailing edge was hinged to form ailerons.

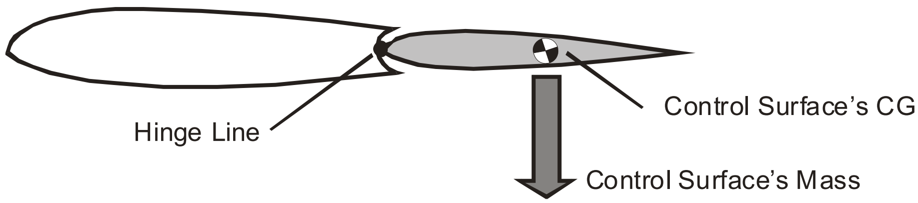


Diagram 5.24 Control Surface Leading Edge and Hinge in Alignment

As each control surface is effectively a small aerofoil, it has its own centre of gravity and centre of pressure. If the control surface were hinged at its leading edge with an input in line with the hinge, the mass of material behind the hinge line would act to make the controls heavy, increasing the effort required to move them. The pilot would have no mechanical advantage. This would increase the effort required for the pilot to deflect them, see diagram 5.24. To overcome these problems, the designer can use **aerodynamic balancing** and **mass balancing**.

FLUTTER

Control surface flutter is a condition where the trailing edge oscillates either side of its neutral point. There are several factors that contribute to flutter. These are:

- The elasticity of the aerofoil section to which the control is attached
- The location of the control surface C of G
- Rigidity of the control run

As an aerofoil, such as the wing, is subject to aerodynamic loads, due to its design, it bends and twists. This bending and twisting can create a vibration that acts backward to the control surface. If the C of G is aft of the hinge line, the trailing edge is deflected, although there is no control input. The deflection of the trailing edge creates bending within the control surface, which in turn springs the trailing edge in the opposite direction, and so on.

While this problem is normally associated with high-speed flight, GA aircraft with sloppy controls can achieve airspeeds in a dive where control surface flutter may occur. The designer must ensure that the natural vibration frequencies that induce flutter do not occur in the flight range of airspeeds. To overcome this, the designer can either move the CG on to the hinge line with mass balancing or make the wing/control surface more rigid.

MASS BALANCE

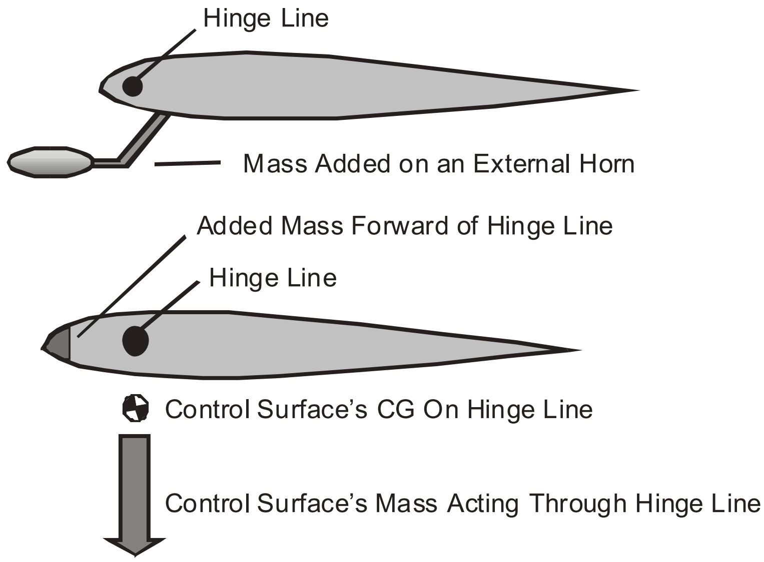


Diagram 5.25 Mass Balance

Mass balancing can take the form of an internal weight added to the leading edge of the control surface, referred to as **internal mass balance**. It can also take the form of an external weight fitted to a lever arm (horn) attached to the control surface's structure. Refer to diagram 5.25 for an illustration.

Where the hinge line of the control is at or very close to the control surface's leading edge or the mass of material required to be added internally would be disadvantageous (increased overall gross mass), the designer would opt to have a smaller mass acting over a longer lever arm mounted externally.

External mass balances are normally for use on slower aircraft and, if required, are fitted to the ailerons and rudder. Normally, there is sufficient space within the fuselage for the elevator's mass balance horn to move, thus eliminating the need for another external projection and its associated drag.

Where an external mass balance is used to balance the ailerons, it is standard for one horn to be attached to each aileron's structure on the underside. In operation, each balances its associated aileron. Being on the underside of the wing, it has less effect on the airflow across the control surface. If a rudder requires external mass balancing, and it is not feasible to mount a single horn from the top (head) of the rudder, then two smaller masses, one mounted each side of the rudder, ensure that it is correctly balanced.

AERODYNAMIC BALANCING

For manually actuated control surfaces hinged directly at their leading edges, the pilot has to apply all the effort required to deflect them. To reduce the effort required, aerodynamic balancing is used. This can take the form of inset hinges or a horn balance.

INSET HINGES

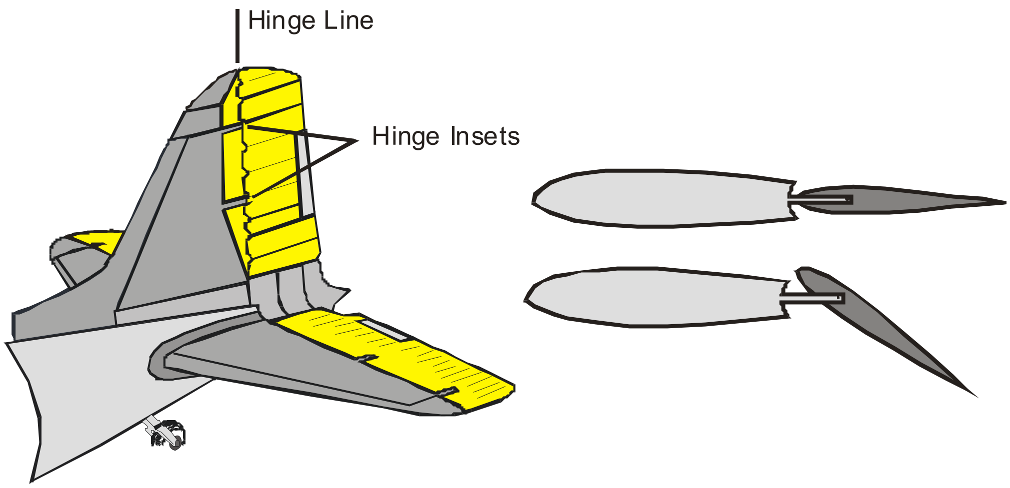


Diagram 5.26 Inset Hinges

In this design, the hinge line is located aft of the leading edge of the control surface. These are termed inset hinges. When the control is deflected in one direction, the leading edge of the control surface moves in the opposite direction and protrudes from the opposite side of the aerofoil surface to which it is attached. This has the effect of moving the leading edge into the airflow, resulting in a force being created that assists the pilot to move the control.

HORN BALANCE

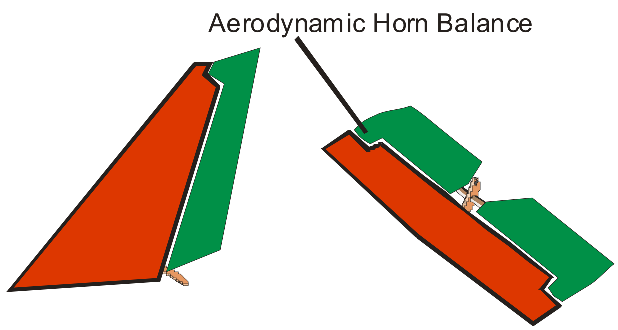


Diagram 5.27 Rudder and Elevator Aerodynamic Balance Horns

For some aircraft, this aerodynamic effect is achieved by having a balance horn protruding from the outboard edge of the control surface. This allows for a simpler leading edge hinge attachment. It is normally confined to elevators and rudders.

The use of an aerodynamic horn balance allows the designer to locate the balance mass for the control surface at its tip. This reduces the amount of material required compared with an internal balance fitted just ahead of the hinge line, as it is able to act over a lever arm. This makes the horn act as an external mass balance at the same time as providing aerodynamic balance.

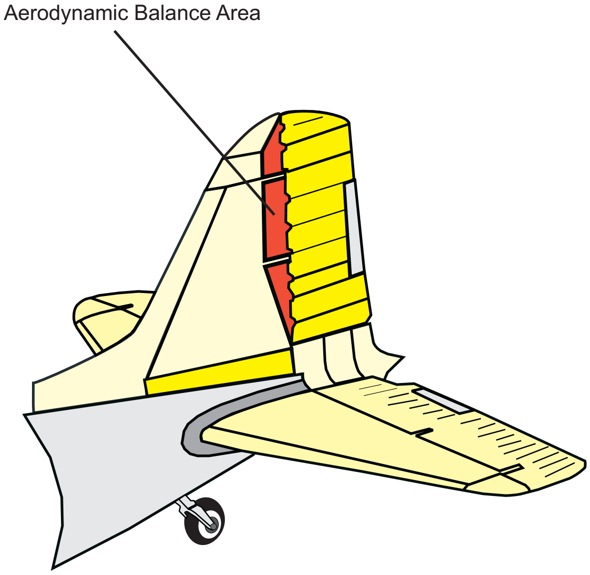


Diagram 5.28 Aerodynamic Balance Area

If the area ahead of an inset hinge or balance horn is too large, the action of deflecting the control results in the control wanting to move to full deflection. If the area ahead of the inset hinge or balance horn is sufficiently large that the aerodynamic balance created equals the hinge movement of the rest of the surface, the pilot loses the sense of feel for that control. If the balance area exceeds 1/5 of the total area, the control becomes over-balanced and any deflection results in the control surface driving itself to the fully deflected condition, as diagram 5.28 shows.

INTERNAL BALANCE PANEL

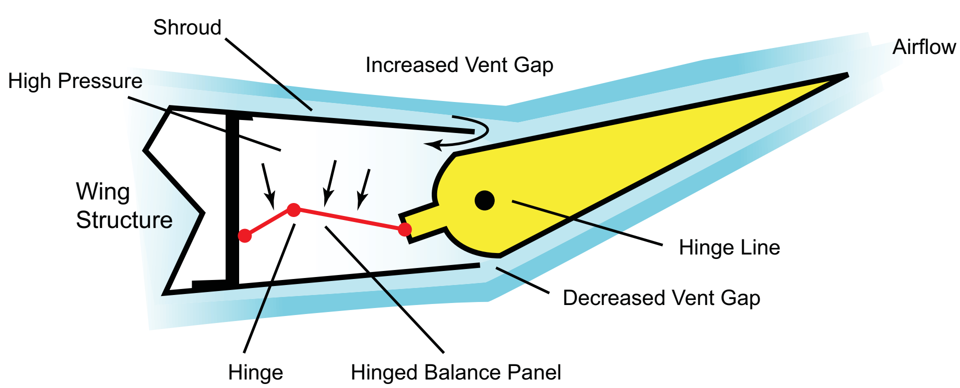


Diagram 5.29 Aileron with Internal Balance Panel

In this design, the leading edge of the aileron remains shrouded by the trailing edge of the wing throughout its range of movement. The gap between the Te of the wing and the Le of the aileron is filled with a panel that is hinged in three places:

1. At its junction with the trailing edge of the wing
2. At its junction with the leading edge of the aileron
3. At the mid point

This divides the shrouded area into two compartments, each of which is vented to atmosphere by the gap between the shroud and the surface of the aileron. When the aileron is deflected, for example upward, due to the curvature of the control surface, the gap between the up-going surface and the shroud increases. At the same time, the gap between the down-going aileron leading edge and the lower shroud decreases.

This alters the amount of airflow into the shrouded area, resulting in a pressure increase in the compartment above the balance panel. This difference in pressures assists movement of the controls. Since the opposite wing mirrors this, the total effect is to reduce the stick force required by the pilot to make the deflection.

HINGE MOMENT

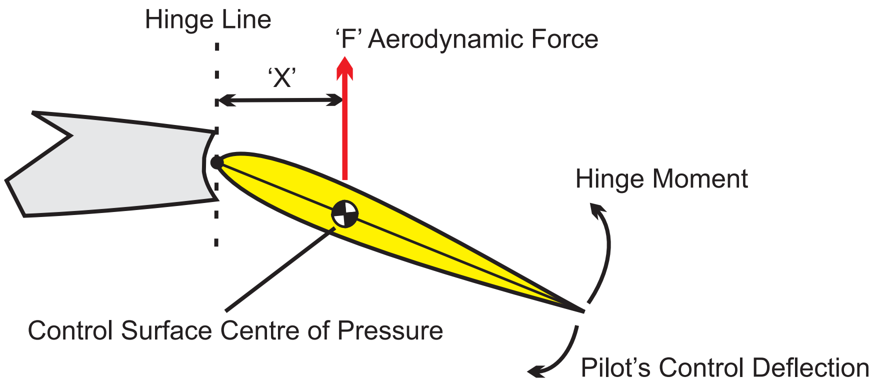


Diagram 5.30 Control Surface Hinge Moment

When a control surface is deflected, it produces an aerodynamic force, which acts in the opposite direction to the deflection and through the surface's centre of pressure. The hinge moment is the result of the aerodynamic force F and the distance between the CP and the hinge line X.

Hinge moment = FX

This hinge moment acts against the pilot. It increases with airspeed and control surface deflections. This is termed **stick force**, which is the force a pilot would have to exert to hold the stick in that position. It is also referred to as **feel**, as the pilot feels the air loads acting on the aircraft's controls.

- If the CP moves further aft, the hinge moment increases.
- If the CP moves forward, the hinge moment decreases.
- If the CP moves onto the hinge line, the hinge moment is zero.
- If the CP moves forward of the hinge line, the hinge moment increases and acts to move the control in the direction of the deflection.

STICK GEARING – MECHANICAL ADVANTAGE

Stick gearing can be used to obtain mechanical advantage and reduce the stick force. This is achieved by making the movement of the stick greater than the movement of the control surface deflection.

Stick gearing = stick movement/control surface deflection

2:1 = 20"/10"

ADVERSE YAWING EFFECT IN PLAIN AILERONS

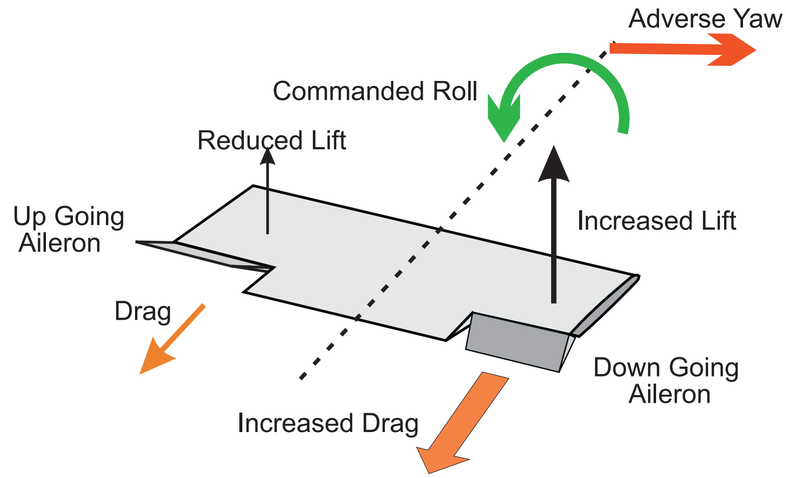


Diagram 5.31 Adverse Yaw

When a pilot starts a rolling manoeuvre by deflecting the ailerons, the down-going aileron increases the camber of the outer wing, while the up-going aileron decreases the camber of the inner wing. This increases the profile drag for both wings. However, the increased lift created by the down-going aileron increases the induced drag for that wing, while the induced drag for the up-going aileron decreases. This alteration of lift, drag, and angle of attack across the wings causes the aircraft to yaw in the opposite direction.

Hence, the secondary effect of roll is adverse yaw. To overcome the adverse yaw induced by roll, the pilot has to apply rudder to stop the aircraft yawing in the opposite direction of the roll. The effect of adverse yaw is greatest at full aileron deflection, which occurs at low airspeeds. There are three methods used to overcome this:

- Differential aileron
- Rudder aileron interlink
- Frise aileron

DIFFERENTIAL AILERONS

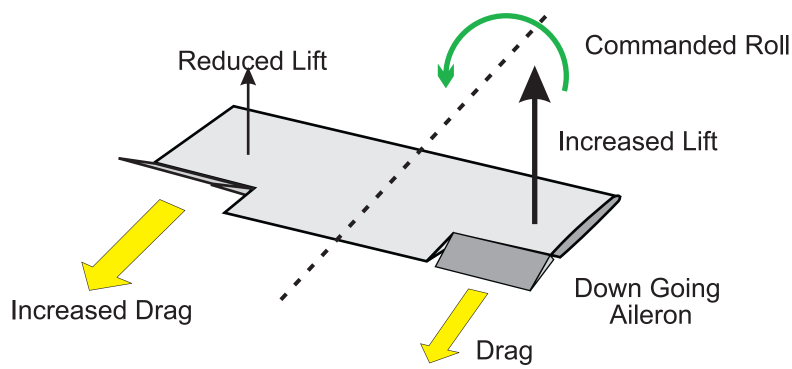


Diagram 5.32 Differential Ailerons

To overcome the secondary effect of rolling, the aileron movement is geared differentially. This results in the up-going aileron moving a greater distance than the down-going aileron.

The drag created by the up-going aileron is greater than that of the down-going aileron, which in turn helps counter the yawing effect.

RUDDER-AILERON INTERLINK

In this design, a pilot's control input to the aileron feeds a proportionate input to the rudder. As roll causes yaw, conversely yaw causes roll. When the pilot applies rudder, a degree of aileron deflection is also applied with the interlink system.

FRISE AILERONS

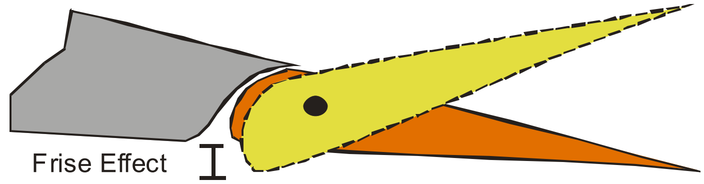


Diagram 5.33 Frise Aileron

In this system, which the Bristol Aircraft Company developed in 1923, the lower leading edge of the up-going aileron protrudes below the surface of the wing. This increases the drag created by the up-going aileron (down-going wing) to a value that is greater than the down-going aileron where the leading edge remains faired by the wing. This imbalance of drag counteracts the adverse yawing effect created by the up-going wing.

FIXED TABS

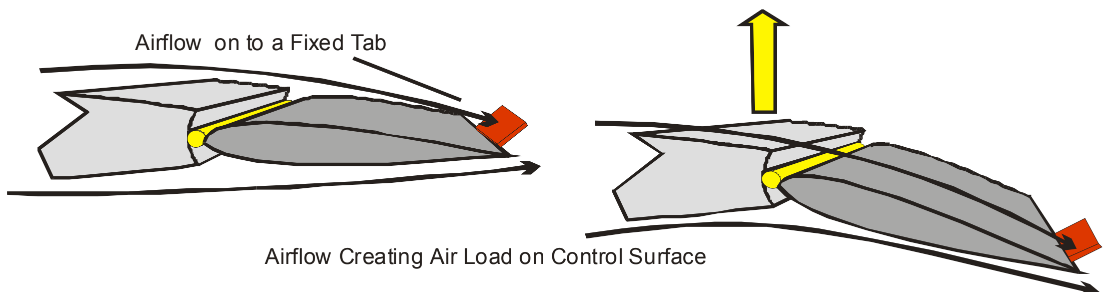


Diagram 5.34 Fixed Tab

Unless an aircraft's aerodynamic surfaces are exactly symmetrical, an aircraft can fly with one wing low. To lift the wing without the pilot having to hold the controls, a fixed tab of metal is attached to the trailing edge of the aileron. The angle at which this tab is bent up or down, depending on which aileron it is fitted, is determined after a flight test. The tabs are adjusted on the ground only by qualified engineers.

The effect of bending a tab upward, as per diagram 5.34, is to create a force that deflects (biases) the aileron downward. When the airload on the aileron equals the force created by the fixed tab, the aileron stops moving downward and remains at the deflected position. This method of lifting a wing is basic, and it can only be set for one effective airspeed. To improve the handling of the primary controls, balance tabs are fitted.

BALANCE TAB

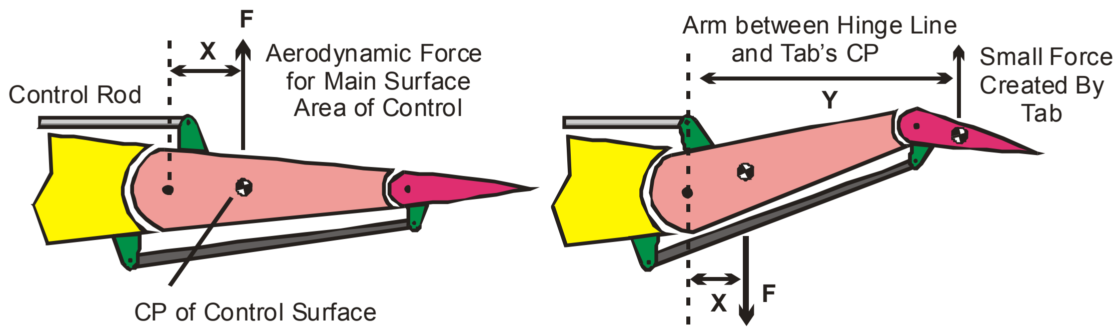


Diagram 5.35 Balance Tab

Hinging a small section of the control surface's trailing edge forms the balance tab, see diagram 5.35. An adjustable rod fitted to the wing structure connects the tab. As both ends of the connecting rod are on the same surface, when the control is given an input, the trailing edge of the balance tab is deflected in the opposite direction to the trailing edge of the control surface.

This action creates a small aerodynamic force, which acts in the opposite direction to the main aerodynamic force created by the control surface. However, as this is acting over a longer lever arm than that of the control surface, it reduces the hinge moment acting back to the pilot's controls. This balancing action is proportionate to the control surface's movement and is set up after a flight test.

TRIM TABS

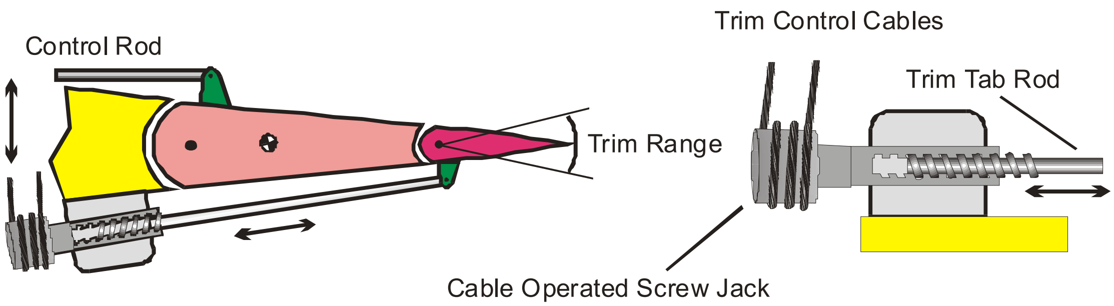


Diagram 5.36 Trim Tab

Fixed tabs, while correcting such conditions as a wing flying low, do not allow for the pilot to fly hands-off through a range of speed and CG conditions. To overcome this, a small section of the trailing edge of a control surface is hinged and can be adjusted by the pilot in flight. Trim tabs count as secondary control surfaces.

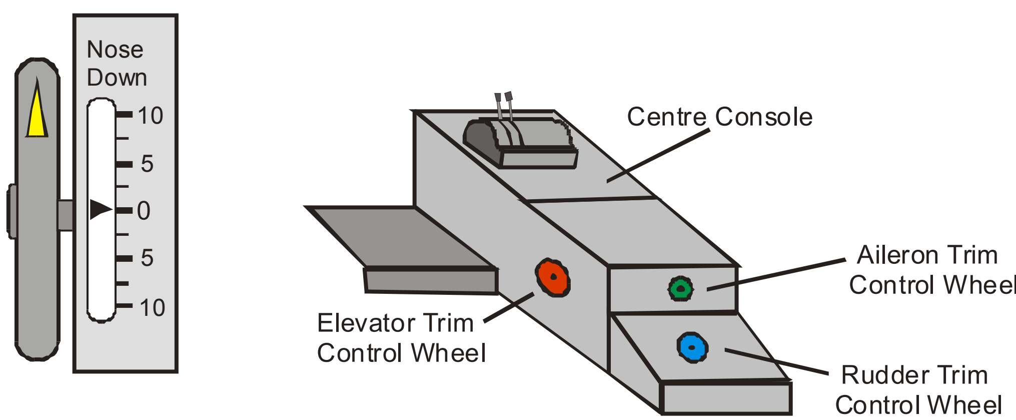
Diagram 5.36 shows a trim tab operated by a screw jack. Cables connected to the pilot's trim control rotate the screw jack. As the screw jack rotates in one direction, it lengthens the rod and vice versa. This action raises or lowers the trim tab's trailing edge from one maximum deflection to the other. This is termed the trim range.

The trim tab operates on the principle that when it is deflected, it creates an aerodynamic force that moves the control surface in the opposite direction. When the force created by the deflected control surface is equal to the force created by the trim tab, the control surface ceases to move and remains in that condition.

This allows the pilot, who must hold the stick out of the neutral position when the aircraft is flying straight and level, to adjust the trim tab and remove the forces that the control surfaces are applying back through the control stick (stick force). Therefore, the pilot can fly hands-off.

Diagram 5.37 Location of Trim Tabs

Trim tabs, normally found fitted to the elevators and rudders, can also be fitted to ailerons. As these are instinctive controls, the elevator trim tab control wheel mounts in the vertical plane and the rudder trim tab control wheel mounts in the horizontal plane. Rotating the elevator trim wheel forward pitches the nose down by raising the trailing edge of the elevator trim tab. This results in the elevator control surface deflecting downward.



Elevator Trim Wheel And Indicator

Diagram 5.38 Manual Trim Controls

To indicate that the trim control is in neutral on light aircraft, a line or index marker on the control wheel is aligned with an index marker on the console. For larger aircraft, a scale and pointer normally provide trim indication, as diagram 5.38 shows. Where a trim tab is fitted to an aircraft's primary controls, it must either be balanced to prevent it from fluttering or its control run designed to be irreversible. Trim tab flutter can cause control surface flutter.

TRIM FUNCTION FOR PCU OPERATED CONTROLS

Where hydraulic PCUs are used to operate primary flight surfaces, balance tabs, anti-balance tabs, and trim tabs cannot be used to relieve the aerodynamic load and allow hands-free flight. The surfaces actuated by PCUs hydraulically lock when the pilot has ceased moving them. Therefore, altering the geometry of the control run, which in turn repositions the stick, carries out the trim function for these controls. This is normally accomplished using electric trim motors.

ELECTRIC TRIM ACTUATORS

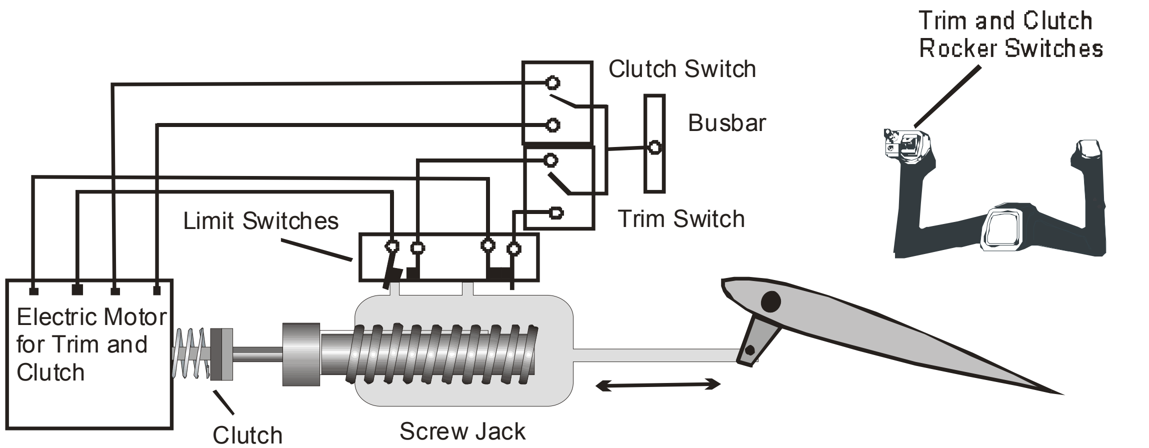


Diagram 5.39 Electric Trim

Both light and air transport aircraft make use of electric trim motors to make the task of trimming the flight control systems easier. For a light aircraft, the elevator or tailplane is normally the surface that has this system fitted. Air transport aircraft with autopilots have an electrical system for each flight control.

Requirements for powered trim systems are:

- Precautions must occur to prevent inadvertent, improper, or abrupt trim tab operation.
- The aeroplane is safely controllable.
- The pilot can perform all the manoeuvres and operations necessary to effect a safe landing following any probable powered trim system runaway that might reasonably be expected in service.

To achieve these requirements, it is standard for the pilot to have to operate two thumb-rocker switches simultaneously. One switch operates an electronic clutch, the other the trim motor. In the event of either switch or circuit failing, the power trim does not operate. Should, for any reason, the trim actuator continue to run after the switches are released, the pilot overcomes the force exerted by the trim on the system by manually moving the controls. In this case, many large air transport aircraft limit the range of the power trim.

SECONDARY CONTROL POSITION INDICATORS

For position indication on light GA aircraft, the hand wheel can have a simple groove filled with paint that aligns with a similar mark on the centre console. Large aircraft require a more refined method that presents a greater amount of information. This takes the form of a scale and pointer. The take-off safe range is marked off with a green band for the elevators and trimmable tail.

PRIMARY CONTROL SURFACES OPERATED BY SERVO TABS

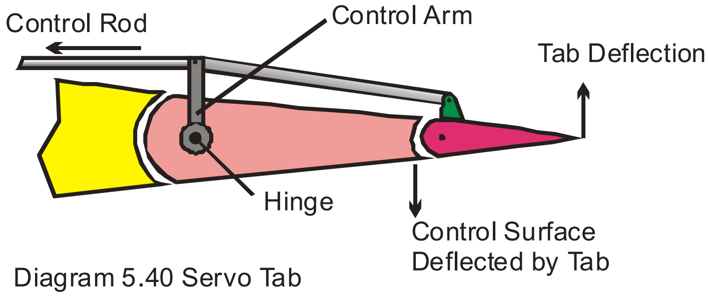


Diagram 5.40 Servo Tab

In this design, the primary control input directly links to a tab set in the trailing edge of the control surface, as diagram 5.40 illustrates. When the pilot makes a control input, the tab is deflected in the opposite direction to the required control surface deflection, resulting in the control surface moving in the intended direction.

This design allows manual control of surfaces that would otherwise require power-flying controls. When an aircraft fitted with these controls is in park, the control surfaces appear to hang down. When stationary, any movement of the control column results in only the servo tabs moving. On the take-off run, the control surfaces continue to be slack until sufficient airflow is passing over them. Due to the need for airflow across the tabs, aircraft fitted with servo tabs can suffer from low-speed handling problems.

SPRING SERVO TAB

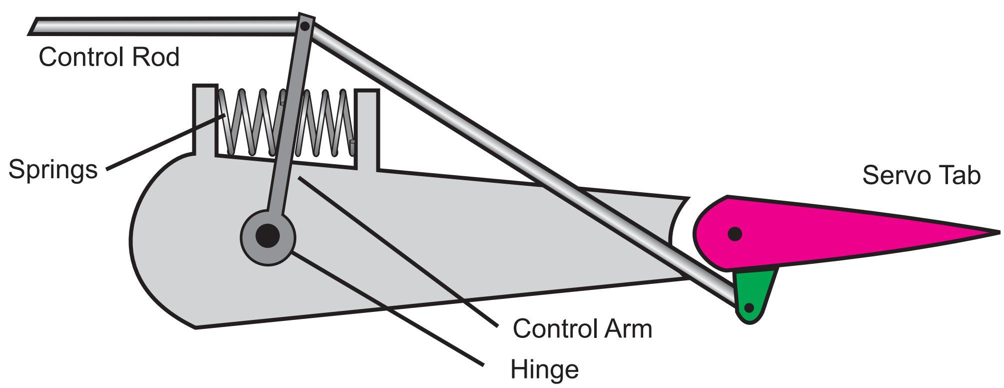


Diagram 5.41 Spring Servo Tab

The spring servo is designed to overcome the control problems associated with low-speed handling of the servo tabs. In this design, the control input linkage attaches to an input horn that is in line with the hinge line but not attached to the hinge or control surface. A linkage from this input horn goes diagonally to the servo tab.

Attached to the control surface and the input horn are two springs under compression. When the aircraft is manoeuvring at low airspeed, the force of the springs locks the control surface to the input horn. Therefore, any input made by the pilot moves the whole surface. When the air load becomes greater than the springs’ force, the control surface breaks free of the input horn, and the servo tab takes charge.

HORIZONTAL STABILATOR

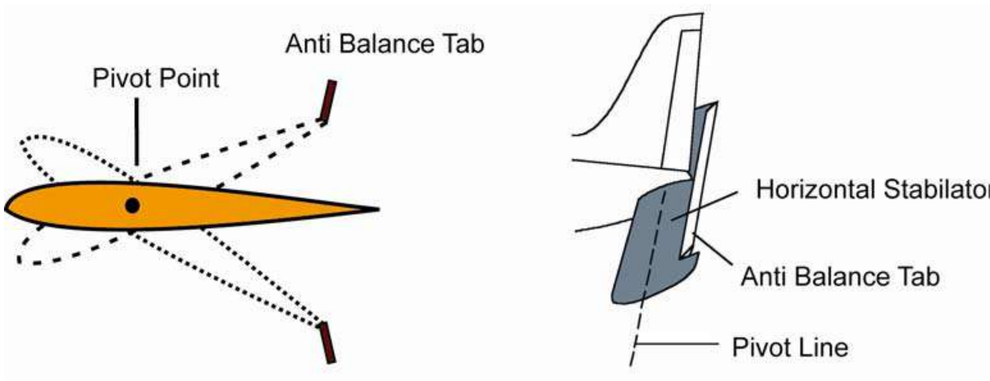


Diagram 5.42 Horizontal Stabilator and Anti Balance Tab

In this design, the fixed tailplane and pivoting elevator are replaced with a single pivoting aerofoil, see diagram 5.42. This gives greater controllability and is more responsive. However, as the stabilator surface has more authority than an elevator over the aircraft's pitching, an anti-balance tab system is used to prevent the pilot from making large control inputs.

It is conventional to talk about an elevator's movement by referring to the action of the trailing edge. For the horizontal stabilator, it is conventional to talk about the action of the leading edge. Therefore, when the control column is pushed forward, the trailing edge of the elevator is deflected down. For a horizontal stabilator, the leading edge is pitched up.

ANTI-BALANCE TAB

Anti-balance tabs can be fitted to control surfaces of light aircraft where it is necessary to increase the stick force. The most common control surface to have an anti-balance tab fitted is the horizontal stabilator, as per diagram 5.42.

In this design, a rod running diagonally to the fuselage connects a hinged section of the trailing edge from one surface of a horizontal stabilator. When the trailing edge of the stabilator moves upward, the anti-balance tab moves proportionately in the same direction. This increases the stick force feedback to the pilot.

ANTI-BALANCE TRIM TAB

The anti-balance trim tab combines both functions in the same tab, the length of the connecting rod setting the range of movement for the anti-balance action. To obtain the trim action, the pilot moves the hand wheel, thereby adjusting the anti-balance rod's attachment point to the fuselage. Both the trim action and the anti-balance action have set ranges of movement. By adjusting the trim, the pilot displaces the anti-balance tab.

SLAB OR ALL FLYING TAILPLANE

This design is for use in aircraft with powered flying controls. The tailplane, like the horizontal stabilator, is pivoted and rotates about a torque tube to alter its angle of attack. As hydraulic power moves the surface, artificial feel warns the pilot about the aerodynamic loads being created. Trim for this system is incorporated within the control run.

ELEVONS

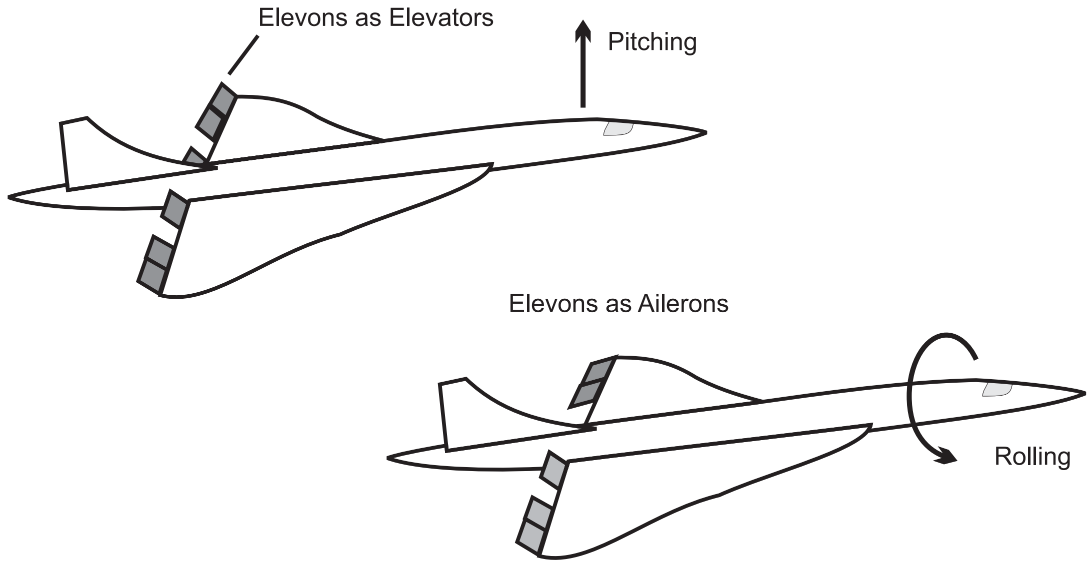


Diagram 5.43 Elevons

For delta-wing aircraft, such as the Concorde*,* which do not require a separate tailplane, the aileron and elevator functions combine into one hinged control an equal number of combined surfaces mounted on each wing, as in diagram 5.43. Movement of the control column in pitch control makes all the surfaces move together to act as an elevator. Roll is achieved by the elevon on one wing deflecting downward and the elevon on the opposite side deflecting upward.

When a pilot inputs both roll and pitch control at the same time, the control inputs pass through a **mixing unit**. This sends the input signal to the power flying control units, which deflects all the elevons in the same direction to achieve the pitch control but varies the amount by which they deflect to achieve the roll control.

VARIABLE INCIDENCE TAILPLANE — TRIMMING TAILPLANE

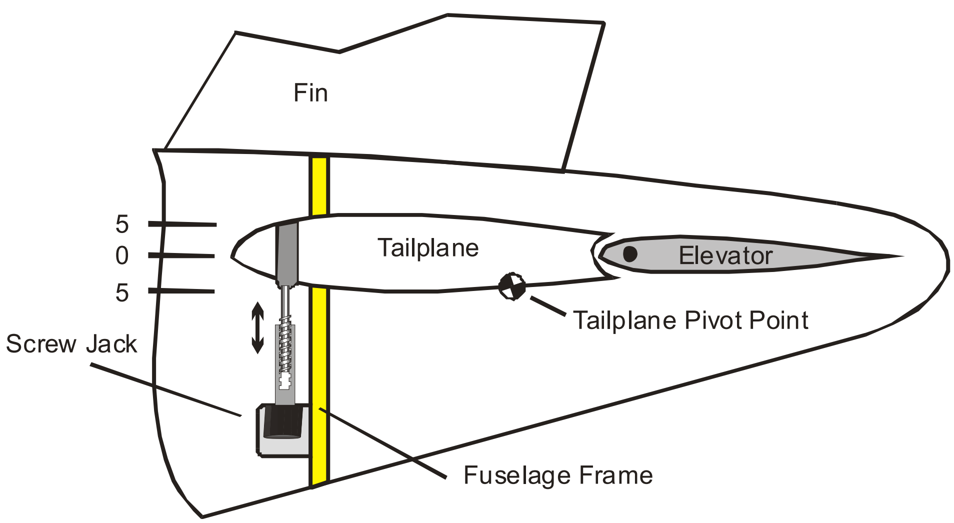


Diagram 5.44 Variable Incidence Tailplane

To reduce the tailplane load to the minimum in the cruise and to achieve the inbuilt stability function, the tailplane's leading edge is usually mounted 4 degrees below the horizontal. This is referred to as longitudinal dihedral. If the aircraft is correctly trimmed for economical flight the nose is slightly pitched up. This alters the angle of attack of the airflow onto the tailplane and creates the required downforce with minimal use of elevators, therefore decreasing trim drag.

If the aircraft's CG does not achieve the nose up attitude, the pilot has to apply up elevator and trim. This increases the elevator drag. To overcome this and allow the pilot to offload the tailplane, large air transport aircraft have trimmable tailplanes. In this design, the tailplane hinges at its rear attachment to the fuselage structure. The front of the tailplane connects to the fuselage by an actuator. This is normally an electric screw jack (actuator). Refer to diagram 5.44.

Operation of this actuator can be controlled both manually and automatically. In manual mode, the pilot is able to set the tailplane trim depending on the position of the CG and the trailing edge flap setting. This change in tailplane trim counters the downwash effect caused by the flaps and the lever arm from the CG to the tailplane CP.

In automatic mode, when the air loads acting on the tailplane, due to the elevator deflection, reach a pre-set level, the trim-jack is actuated and alters the leading edge's angle of attack, at the same time reducing elevator deflection. This results in the tailplane taking a more faired position for the normal condition of flight. This automatic action does not alter the pilot's control column position.

In high-speed high altitude cruise, the elevator remains locked with the tailplane, and both act as a single surface. This provides the required authority and prevents the elevator from flexing when deflected and also reduces drag. At low airspeed, the tailplane locks and the elevator deflects.

In the unlikely event that the tailplane trim-jack suffers a "runaway", and the leading edge goes fully up or down, the pilot must be able to handle the aircraft using the elevator controls only, although the elevator's authority is decreased.

BUTTERFLY OR V TAIL AIRCRAFT

Diagram 5.45 V Tails Function

Some light aircraft, such as the original Beech Bonanza 35, replaced the conventional tail with two aerofoil surfaces mounted in the form of a V. These act to provide directional stability and pitch stability. To allow pitch and roll control, control surfaces termed ruddervators hinge at the trailing edge of aerofoils. These are linked, via a mixing unit, to the control column and the rudder pedals.

For yaw control, operation of the rudder pedals deflects the ruddervators so that one is moving down and the other up. As the aerofoils are inclined, the movement is proportionate.

For pitch control, operation of the control column deflects the ruddervators in the same direction.

When both pitch and yaw inputs are made simultaneously, the mixing unit adjusts the input to each surface accordingly. Some designs replace the fixed aerofoil and control surfaces with all flying surfaces. These operate in a similar manner but pivot, as per diagram 5.45.

CONTROL LOCKS

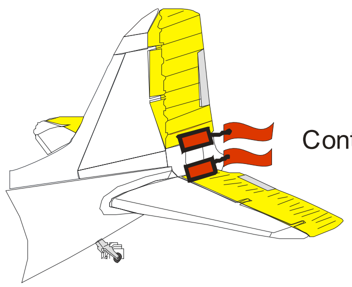


Diagram 5.46 Control Locks

Control locks are fitted to prevent gust damage to control systems and surfaces of parked aircraft, due to wind catching the surfaces and slamming them back and forth,. For light aircraft, these often take the form of a clamp fitted between the trailing edge of the control surface and the adjacent structure of the aerofoil. As diagram 5.46 shows, warning pennants attach to the locks, reminding pilots to remove them before flight.

Larger aircraft have integral control locking systems. These can either lock the control surface or the control run. The locking systems are designed so that a determined movement to unlatch the lever from the controls' unlocked detent is required as well as engaging the locking mechanism.

STICK POSITION STABILITY

If an aircraft requires the control column to be moved rearward to increase the AoA and trim acts to move the control column rearward to maintain a low airspeed; and conversely, if moving the control column forward decreases AoA, and trim acts to move the control column forward when transitioning to a higher airspeed, the aircraft is said to have **stick position stability**. If the aircraft requires moving the stick rearward to trim at high speed, the aircraft is said to have **stick position instability**.

STICK FORCE GRADIENT

If an aircraft demonstrates static stability in straight and level flight, it almost certainly demonstrates stability in manoeuvring flight and demonstrates a steady increase in stick force with an increase in load factor or **g**.

If the stick force gradient is too high, the aircraft will be tiring to fly. If it is too low, there is a danger that the pilot could over-stress the airframe. For transport aircraft, which have a high static stability, a high manoeuvre stick force gradient is applied to ensure that the limiting load factor is not exceeded.

The table below details the maximum stick forces allowed for an aircraft certified under JAR 23 and 25.

Force, in pounds, applied to the control wheel or rudder pedals	Pitch	Roll	Yaw
Control Column – JAR 23	60	30	
For short term application for pitch and roll control --two hands available for control - using a control wheel, JAR 23, 25	75	50	---
For short term application for pitch and roll control --one hand available for control - using a control wheel, JAR 23, 25	50	25	---
For short term application for yaw control – rudder pedals	----	---	150
For long term application using stick, control wheel and rudder pedals	10	5	20

STICK CENTRING SPRINGS

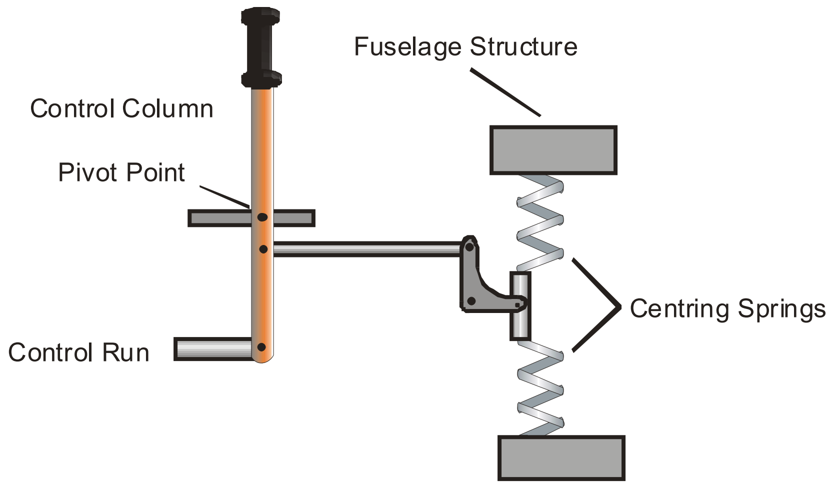


Diagram 5.47 Stick Centring Spring

In the simplified system shown in diagram 5.47, a tension spring is fitted on each side of a bellcrank and attached to the aircraft's structure. These springs act to centre the control surface and stick. For an equally geared control system, any movement of the control column applies an artificial feel to the pilot. This increases the stick force stability during manoeuvring. However, the effect is greatest at lower airspeed due to the larger deflections required. As the aircraft's airspeed increases, the required control deflections decrease as well as the stability effect of the springs.

This system is further simplified by putting springs inside a cylindrical container with the springs acting against a piston, which is linked to the control system. This is termed a spring feel unit.

DOWN SPRINGS

As a simple safety device to ensure that the elevator always wants to pitch the aircraft nose down, a long pre-tensioned spring is attached to the control run and the aircraft structure. As the spring is pre-tensioned, when the aircraft control column is released, the spring acts to move the elevator down. When the aircraft is in park, this is in the fully down position. As airspeed increases, the position the elevator takes is the balance between the down spring's force and the load on the elevator trying to streamline the control surface.

As the spring force is independent of airspeed, the spring adds to the stick force stability of the aircraft without altering the aircraft static stability. The stick force gradient is increased, which means that the pilot gets an increase in feel as airspeed increases. However, as the force from the spring is incremental and unaffected by airspeed or stick position, it does affect the aircraft manoeuvring stick force stability.

BOB WEIGHT

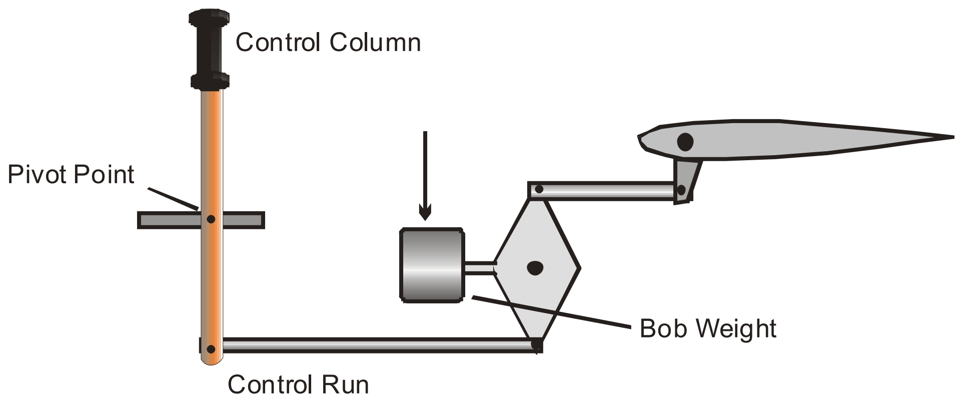


Diagram 5.48 Bob Weight

In this design of control system, a bob or counter-weight is added into the control run. When the aircraft is stationary or in unaccelerated level flight, the elevator is pulled downward due to gravity acting on the bob weight, resulting in increased stick force stability and stick force gradient. The effect is the same as the down spring.

However, the mass of the bob weight is subject to the same acceleration forces as the aircraft. The stick force that the bob weight applies increases in direct proportion to the acceleration force that the aircraft is subjected to. This, due to the linear action of the bob weight, increases the manoeuvring stick force. Therefore, a bob weight can be added to improve the manoeuvring stick force of an aircraft which otherwise has low values.

ARTIFICIAL FEEL

For larger aircraft where PCUs are used, the pilot has no direct feedback “feel”. Therefore, the designer has to use artificial feel to ensure that the pilot senses the magnitude of the effect that the control movements have. The simplest of these is the spring feel unit.

SPRING FEEL UNIT

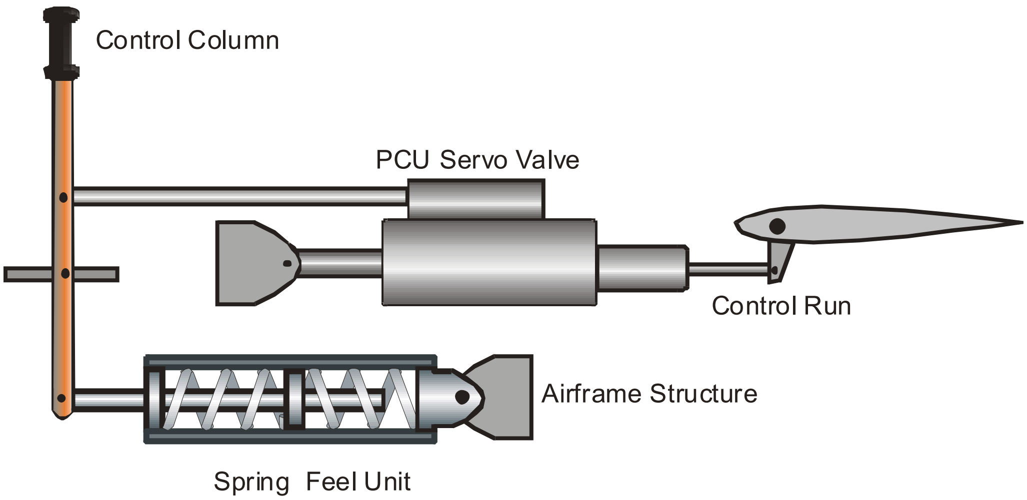


Diagram 5.49 Spring Feel Unit

In this design, the balance springs mount inside a container and act on a piston linked to the control input. As the pilot inputs a signal for the PCU to deflect the control surface, the resistance of the spring is felt. Greater input means greater resistance. Like the balance springs, this system is effective at lower airspeed where greater deflections are required. To create a realistic feel at higher airspeed, which acts as an indication to the pilot as to the air loads acting on the control surfaces, a system measuring and utilising dynamic pressure was devised. Dynamic pressure found by subtracting static pressure from pitot pressure is referred to as **Q**.

BASIC Q FEEL SYSTEM

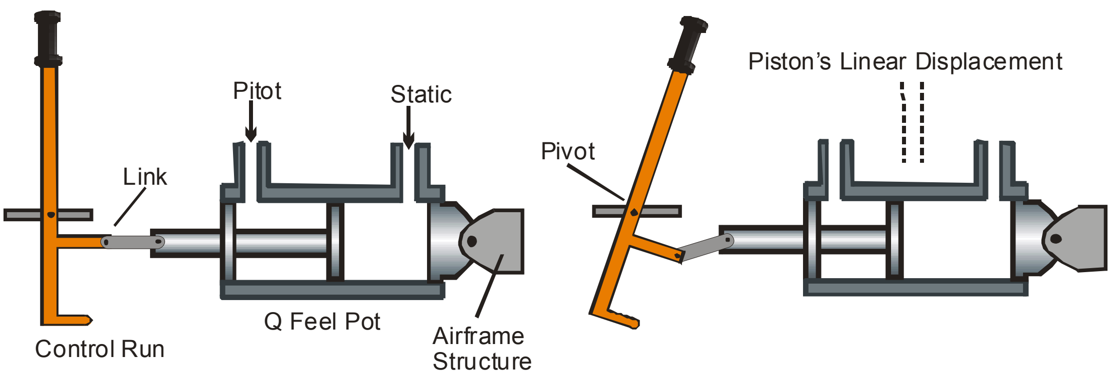


Diagram 5.50 Basic Q Feel Unit

In this design, pitot pressure is directed to one side of a double acting actuator and static pressure to the other. The resulting differential is dynamic pressure, which acts to bias the system, as diagram 5.50 shows. The body of the actuator attaches to the aircraft structure and the piston's ram is attached by a linking block to the control system. The actuator is frequently termed a **Q feel pot**.

The Q feel function is achieved by making any deflection away from neutral by moving the piston against the dynamic pressure. The method of achieving this for both fore and aft movement of the control column is to insert a bellcrank between the control column and link. By deflecting the stick an equal distance each side of neutral, the input rod to bellcrank moves linearly the same amount in each direction. The bellcrank turns about its pivot point and the output rod moves the same amount as the input rod but in the opposite direction, sending the signal to the PCU to deflect the control surface. At the same time, the horn of the bellcrank, attached to the Q feel pot, scribes an arc. As the ram attaches to the horn by a link, this arc is transcribed into a linear movement, pulling the piston against the dynamic pressure. The greater the movement, the greater the resistance feel. Any increase in speed increases the value of Q, which acts against the pilot, although requiring only small deflections.

HYDRAULIC Q FEEL

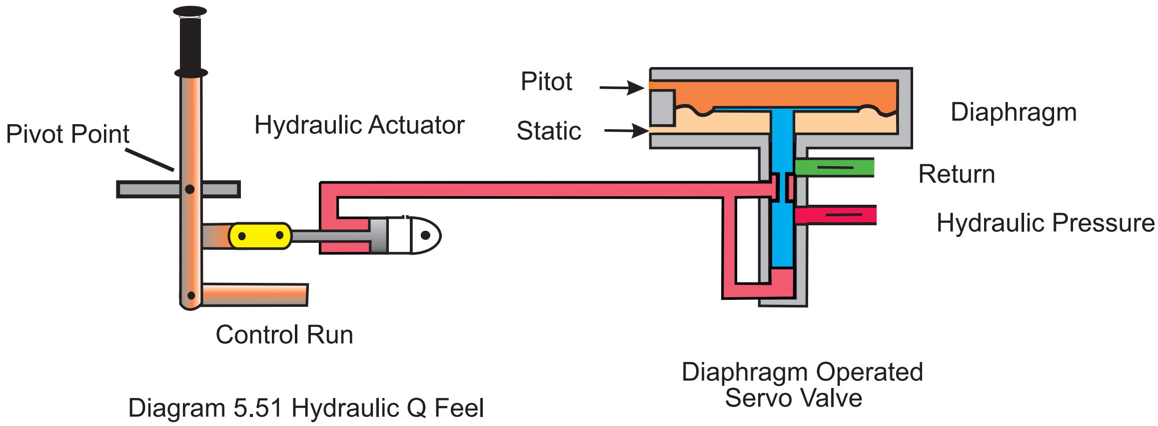


Diagram 5.51 Hydraulic Q Feel

In this design, the waste bin-sized Q feel pot is replaced with a small hydraulic actuator and a servo valve operated by dynamic pressure. Refer to diagram 5.51 for an illustration.

In this design, a small hydraulic servo valve is attached to a diaphragm separating pitot and static pressure. The dynamic pressure biases the servo valve, allowing hydraulic pressure into the actuator.

To make the feel proportionate, the hydraulic pressure that passes through the servo valve also acts up beneath the valve's stem. This neutralises the servo valve. A decrease in dynamic pressure biases the servo valve to release fluid, dropping the pressure and the force acting back on the stick. This system has a self-centring action.

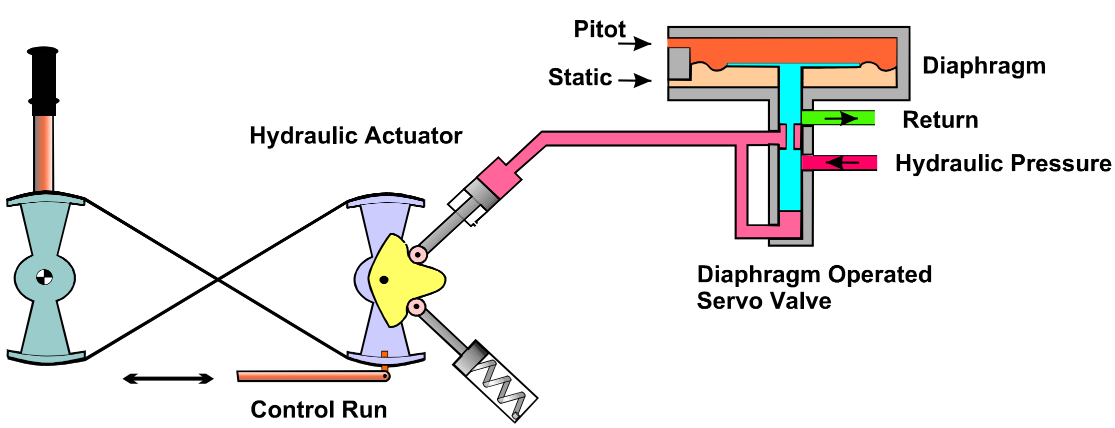


Diagram 5.52 – Spring-Feel Unit

A further improvement in the system is to incorporate a spring feel unit, so that artificial feel is given throughout the whole speed range. By mounting both the hydraulic feel actuator and the spring feel unit so that they act against a cam (see diagram 5.52), the feel effect is increased with increasing control deflection and a more positive centring action gained without the need to increase the size of the spring or actuator.

ROLL CONTROL FOR AIR TRANSPORT AIRCRAFT

As aircraft speed increases, the use of outboard ailerons for roll control can create two problems:

- Control reversal due to flexibility of the wing and control run.
- Excessive rolling moment due to the lever arm from the aileron to the aircraft CG.

To overcome this at high airspeeds, the outer ailerons are locked in the neutral position and the **inner ailerons**, or **flippers**, act to give roll control. As these are fitted to thick chord sections where there is a greater rigidity, the problem of aileron control reversal is overcome as is the rolling moment effect due to the fact that they are closer to the CG.

WING-MOUNTED SPEED BRAKES/LIFT DUMPERS

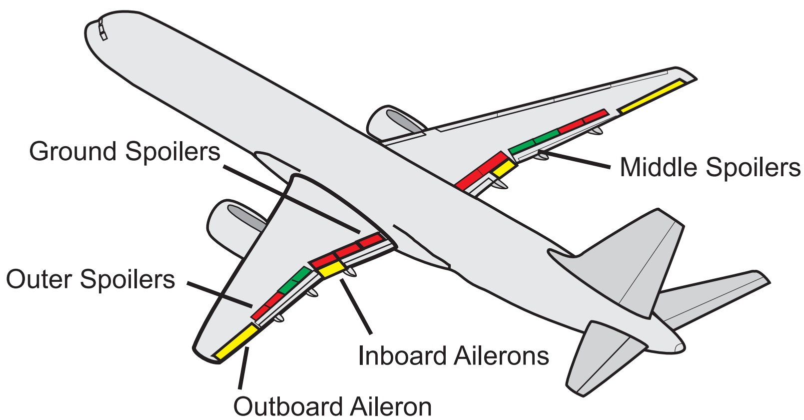


Diagram 5.53 Roll Control

There are two types of wing-mounted speed brakes. These are **vertical spoilers** and **hinged spoilers**. When deployed, the vertical spoiler is raised like a plate across the chord of the wing, creating drag and turbulence, thereby reducing the lift. When stowed, the upper surface of the spoiler is flush with the top surface of the wing.

Hinging and actuating the upper surface flap fairing shrouds can form hinged spoilers. This saves weight and is less complicated as the design adapts panels that were removable or openable for maintenance functions. Aircraft that make use of spoilers for roll control, such as air transport aircraft, normally use hinged spoilers.

Speed brakes are designed to increase drag and spoil the airflow over the wings on touch down, resulting in a slowing of the aircraft and a reduction in the landing run. Reducing the lift from the wings meant that the gear felt the full weight of the aircraft more quickly, which in turn increased the efficiency of the wheel brakes. Improvements on the basic design have allowed these panels to be used in flight as speed brakes and roll control.

The advantages of fitting spoilers are:

- As roll control, they allow the reduction in the size of the outboard aileron, which allows fitting of a longer trailing edge flap. This in turn allows achieving a lower landing speed.
- As lift dumpers, they allow lift/speed to be decreased without the reduction of engine power. This is an important function for turbine engines, which require time to recover after any power reduction.
- They allow a higher maximum control speed, as the air loads created by the deflection of mid and inboard wing mounted spoilers have less torsional effect on the wing's structure compared with those created by the outboard ailerons.

It is fairly standard for the outer panels to be used for roll control at lower airspeed, the middle panels for speed braking and lift dumping in flight and the inner panels as lift dumpers on landing. As the aircraft's speed increases, the outer spoilers cease to operate. See the section regarding Load Alleviation Function below for details.

FLIGHT OPERATION

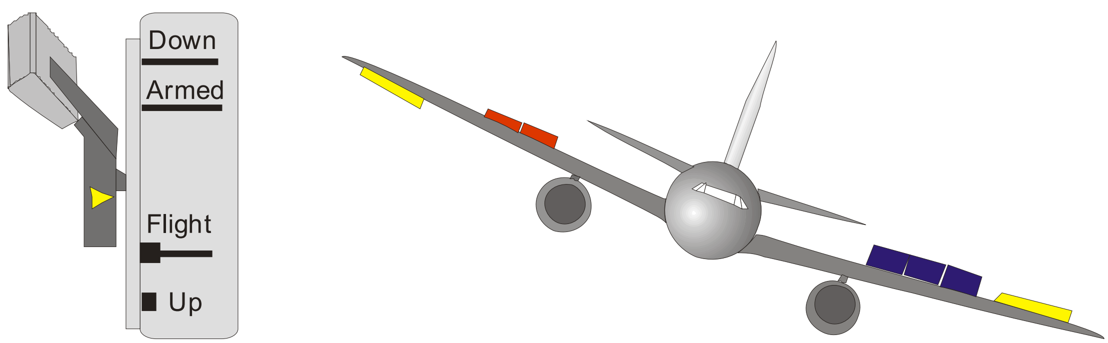


Diagram 5.54 Spoiler Operation In Flight

The spoiler can be used to reduce airspeed and/or increase the rate of descent. In this situation, the pilot can select the spoiler by moving the spoiler lever from the down (flush) detent position to the flight detent position, the maximum extended in flight position. The position of the lever between the two detents indicates the extension of the spoiler.

In this condition, the inner spoiler does not deflect as much as the mid-wing spoiler. This is to prevent tail plane buffet due to turbulent airflow. The outer wing spoilers do not deflect so as not to over-stress the wings. The total deflection is limited to prevent the aircraft from pitching nose up.

To improve roll control, many aircraft automatically link the control wheel input to the outer spoilers as well as the ailerons. In this design, when the pilot banks the aircraft, the outer spoilers on the down-going wing are partially extended to create drag, while the spoilers on the up-going wing remain retracted. This helps to counter adverse yaw and improve the rate of roll.

LOAD ALLEVIATION FUNCTION

As airspeed increases, the outboard ailerons and spoiler panels are locked out, so that they do not operate when the pilot inputs a roll command. In this case, the inboard aileron and mid-wing spoilers take on the function of roll control. This prevents excessive loads from being placed on the outer wing structure.

In situations where a pilot wishes to bank and descend, the input from the control wheel and the speedbrake lever are fed into a control or mixing unit. This acts to raise all the spoilers to achieve the desired slowing descent but makes the extension asymmetrical, so that the spoiler on the down-going wing deflects further.

AIRBRAKES AND HIGH SPEED FLIGHT AT HIGH ALTITUDE

Aircraft flying at high altitude and high speed require airbrakes to ensure that they can slow down so as not exceed the critical Mach speed of the wing. This can lead to a shock stall.

GROUND OPERATION

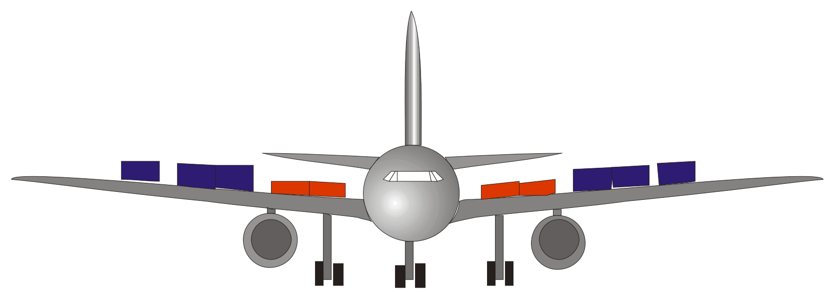


Diagram 5.55 Ground Operation of Spoilers

To achieve an 80% decrease in lift on landing, the pilot can select the airspeed brake lever to "armed." When the aircraft touches down and a series of conditions are met, the spoilers automatically go to full deflection, and the airbrake lever moves to the up detent. If for any reason, the pilot moves the thrust lever, apart from selecting reverse thrust to increase power, the spoilers automatically retract and the lever automatically moves to the down position.

Aircraft conditions to be met:

- Speed brake lever in armed position
- Thrust lever in flight idle position
- Main wheels rotating

FUSELAGE MOUNTED AIRBRAKES

Some aircraft such as the BAe 146 have hinged panels as part of the tail cone structure as well as spoilers mounted on the wings. These can be partly deployed to slow the aircraft in flight and fully deployed for landing.

DROOP AILERONS

Diagram 5.56 Droop Ailerons

The ailerons of some aircraft are designed to act as outboard wing flaps during landing. These are normally referred to as **droop ailerons**. In this design, when the pilot selects landing flap, both outboard ailerons automatically deflect downward, creating extra lift across the outboard wing section. Roll control inputs are fed to the roll control spoilers. This enables the aircraft to land at lower airspeeds.

FLY-BY-WIRE (FBW)

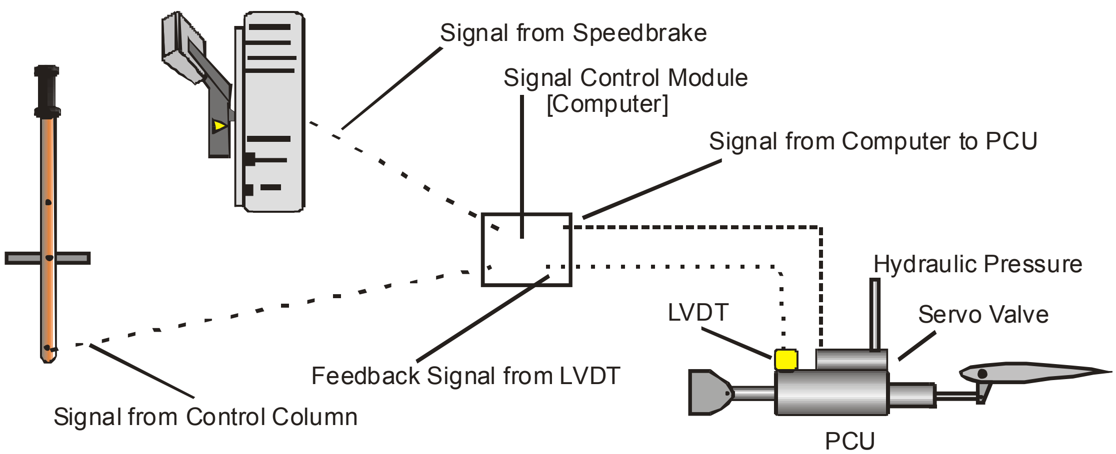


Diagram 5.57 A Basic Diagram of a Fly by Wire System

Modern aircraft make use of computer technology in the operation of the primary flight controls. In these systems, the pilot is not directly linked to the flight control surfaces, but through movement of the control column, inputs control demands into a series of computers. Within these computers, the software compares the pilot’s control input to the aircraft’s control parameters, its airspeed, and its angle of attack, then sends a processed command signal(s) to the appropriate controls to best obtain the desired results.

These command signals are routed via electrical cables to a servo valve attached to the hydraulic power control unit. The signal is converted into displacement of the servo valve, allowing the hydraulic flow and pressure to actuate the control surface. A return signal from either a **Linear Variable Differential Transformer**, **LVDT**, or **Rotary Variable Differential Transformer**, **RVDT**, is sent to the computers when the value from the LVDT or RVDT is equal to the value set by the computer. The signal to the servo valve is cancelled, and the control then hydraulically locks in that position.

As in the most sophisticated air transport aircraft, there is no manual reversion. The command signals are sent through 80 or more cables that take different routes from the input to the output to ensure that no single cable failure or damage results in the loss of a flight control.

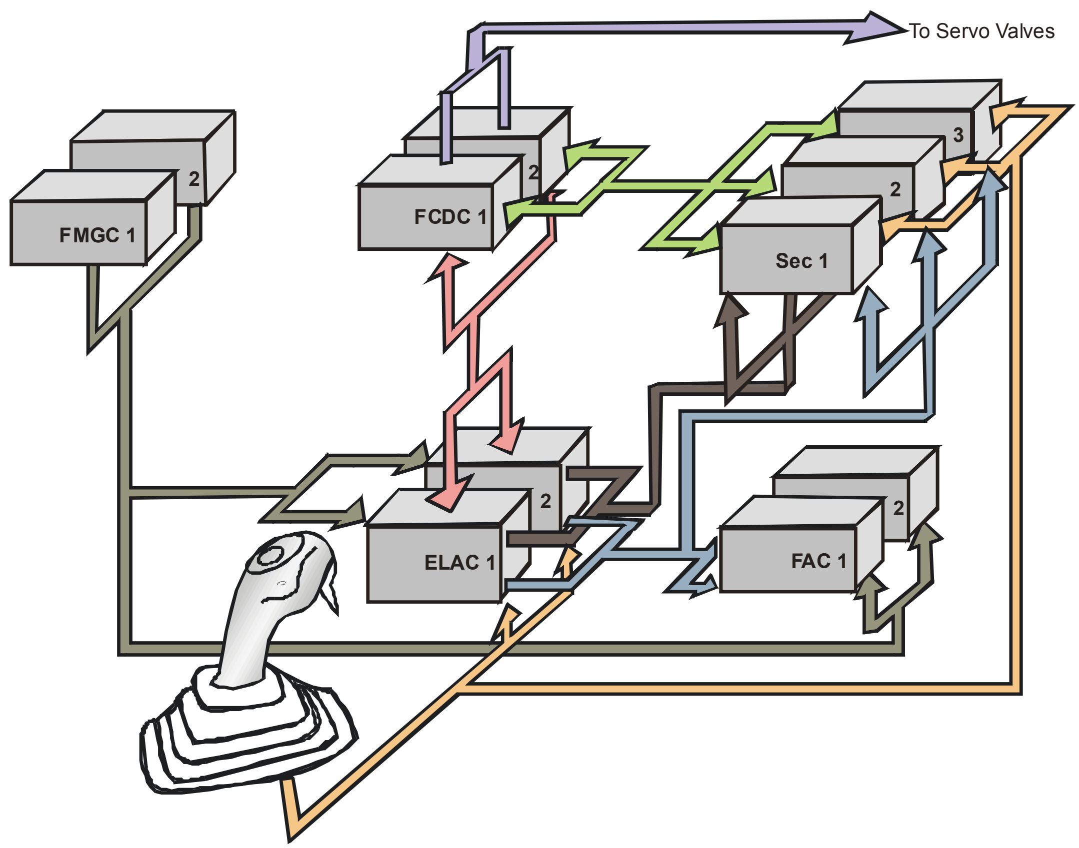


Diagram 5.58 Fly by Wire Computer System

Diagram 5.58 shows the computer system for a sophisticated fly by wire system. It includes:

- Two Elevator Aileron Computers, ELACs, and three Spoiler Elevator Computers, SECs, which control roll and pitch,
- Two **Flight Augmentation Computers**, **FACs**, which control the yaw axis,
- Two **Slat and Flap Control Computers**, **SFCCs**, used to control the slats and flaps
- Two **Flight Control Data Concentrators**, which interface between the ELACS, SECs and the servo valves. They also indicate the maintenance status.

In the event of failure, the aircraft has control authority if one ELAC and SEC are working.
There are many advantages of using fly by wire compared with mechanical systems. These are:

- A reduction in weight
- Reduced maintenance
- More rapid control response
- Built in control parameters
- Gust load alleviation
- Fuel saving

WEIGHT REDUCTION

For a large air transport aircraft, the weight of the mechanical components linking the primary flight controls to the control column is quite considerable. Replacing these with cables and computers saves weight, which increases the payload or saves fuel for the same payload.

REDUCED MAINTENANCE

The setting up and maintenance of mechanical control systems is expensive in maintenance time and parts, whereas the setting up and maintenance of a fly by wire system is simplified by the use of **Local Replacement Units**, **LRUs**, where a computer box can be replaced and calibrated.

CONTROL RESPONSE

As the speed of electricity is that of light, input signals can be processed instantaneously, allowing precise control inputs to be applied or cancelled more quickly than those of a purely mechanical system. This enhances control of the aircraft.

BUILT IN CONTROL PARAMETERS

Built into the software are the aircraft's safe flight envelope parameters. Any control input by the pilot that would cause the aircraft to exceed these limits is automatically cancelled, which ensures that the aircraft's structural limits are not exceeded.

GUST LOAD ALLEVIATION

By using the ability of the computer system to rapidly cycle control inputs, the aileron and spoilers can be actuated as soon as the air data computer senses the increased vertical acceleration created by encountering a gust. To prevent the sudden increase in lift that would cause the aircraft to climb at considerable velocity, the computers send a signal to the ailerons/spoilers of both wings to deflect upward to reduce the lift. This application reduces the strains created within the wing structures, which reduces the fatigue that the structure is subject to. This, in turn, allows the designer to reduce the structural strength required and, therefore, to reduce the mass of the structure.

For all air transport aircraft, there is the twin requirement of stability and controllability, given that a forward CG increases stability and decreases controllability and an aft CG is the reverse. The aft limit of the CG range of older aircraft with mechanical control systems was limited by the ability of the pilot to counter the increased instability in pitch. The use of fly by wire and the rapid response of the control surfaces has enabled the designers to move the CG rearward, making the aircraft more unstable but within safe limits for the computers to rapidly counter any pitch change.

This allows the aircraft to fly at high altitude with its nose pitched up. This creates a greater AoA at the wings, so that more lift is created but without increasing power so that fuel is saved in long-range cruise condition.

LIFT AUGMENTATION – HIGH LIFT DEVICES

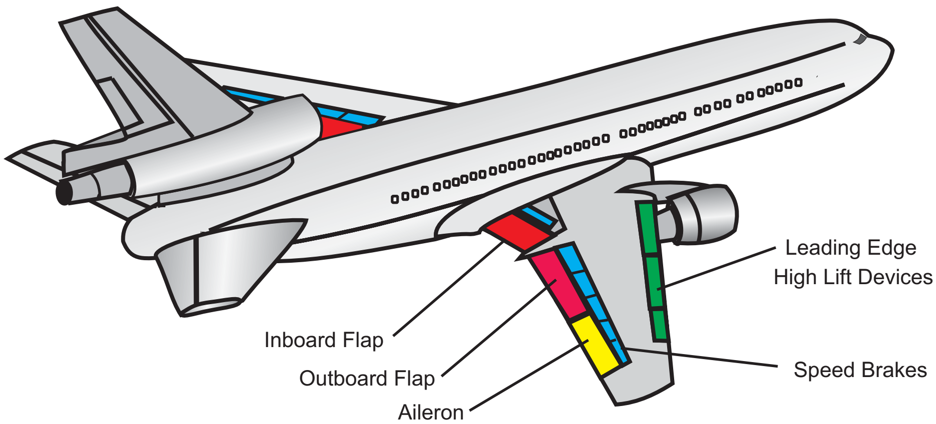


Diagram 6.1 High Lift Devices

Lift augmentation devices, also referred to as high lift devices, are a range of airflow-modifying attachments that are fitted to either the leading or trailing edges of the aircraft's wings. The spoilers/lift dumpers covered earlier fall within this group. Listed below are leading and trailing edge devices.

Trailing edge flap types are:

- Plain
- Split
- Slotted
- Blown
- Fowler
- Slotted Fowler

Leading edge devices are:

- Kruger flaps
- Slots
- Fixed slats
- Retractable slats
- Droop nose

TRAILING EDGE FLAPS

To improve the lifting capability of an aircraft's wing at slow speed, trailing edge flaps that alter camber and/or increase the surface area were devised. While these create lift, they also create drag, both of which can help or hinder depending on the situation. In take-off, the extra drag they create is a hindrance, thus flap extension is limited to gain an increase in lift with a lower drag penalty. In approach and landing, the drag created assists in slowing the aircraft down, and the increased lift allows the aircraft to be flown at a lower speed.

PLAIN FLAP

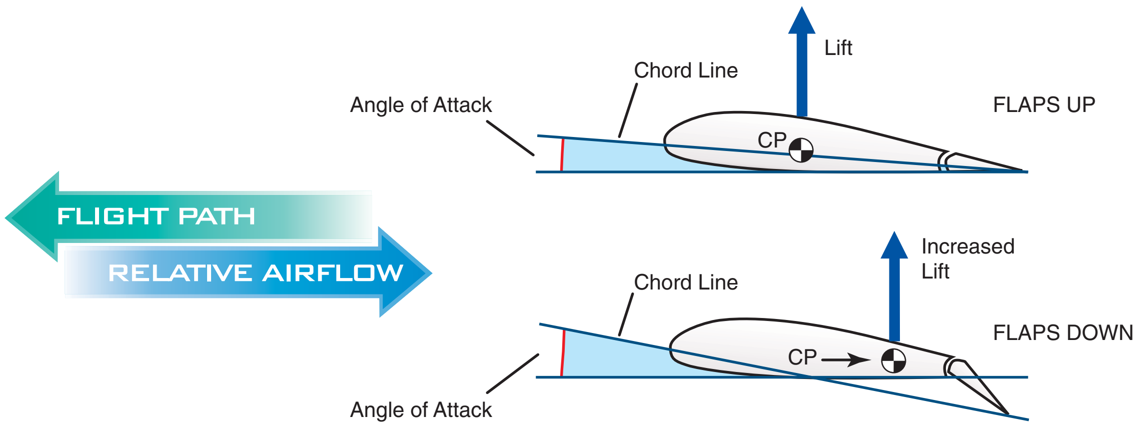


Diagram 6.2 Plain Flap

The plain flaps, as used on some light GA aircraft, are mounted inboard of the ailerons and have the same cross-section as the wing to which they are attached. The hinge line of these flaps passes through the flap structure, just aft of the flaps' leading edge. This results in the flaps' trailing edge aligning with the wings' trailing edge when the flaps are selected up.

As a flap is lowered, the trailing edge of the flap inscribes an arc, which shortens the wing's chord line while increasing the camber. Diagram 6.2 illustrates this. Lowering the flaps increases the wing's lift by up to 50%, which in turn moves the centre of pressure rearward, resulting in a nose down pitching moment. At the same time, the lowered flap creates a greater wake, which increases drag.

SPLIT FLAP

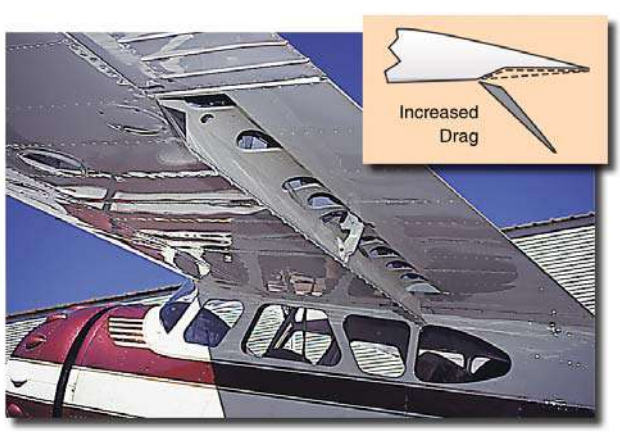


Diagram 6.3 Split Flap

In this design, a hinged panel is let into the rear under-surface of the wing, as shown in diagram 6.3. When the flaps are up (zero flap angle) the panel conforms to the contour of the wing. When the flap is lowered, the panel hinges downward, as per diagram 6.3. This results in creating both lift and drag. Lowering the flap increases the wing's camber without altering the length of the chord.

With the flap fully deflected for landing, there is a 60% increase in lift compared with the flap up. The resultant action of the flap is to pitch the aircraft's nose downward and slow it down. Many air transport aircraft incorporate split flaps, as they have both lift increasing and speed reducing properties. For some older aircraft, the application of full landing flap creates more drag than the lift created can support, so that even at full power, the aircraft descends.

SLOTTED FLAP

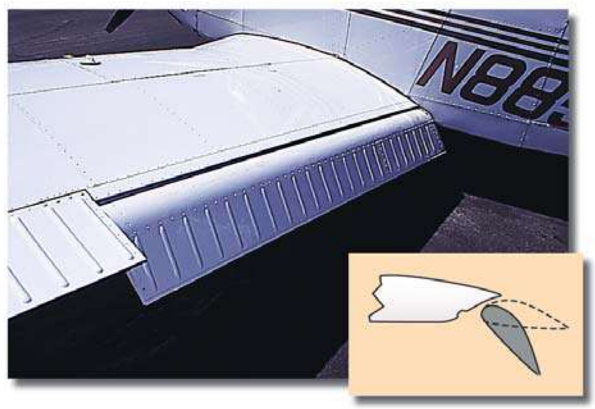


Diagram 6.4 Slotted Flap

In this design, a plain flap's hinge line is modified so that when it is lowered, a shaped slot is formed between the leading edge of the flap and the trailing edge of the wing, as in diagram 6.4. The slot guides high-energy air from under the wing to flow over the top surface of the flap, re-establishing a laminar flow. A wing with a slotted flap has a 65% increase in lift when the flap is fully deployed, with the additional advantage of delaying the onset of wing stall. However, the flap does not produce as much drag as a plain flap.

DOUBLE SLOTTED FLAP

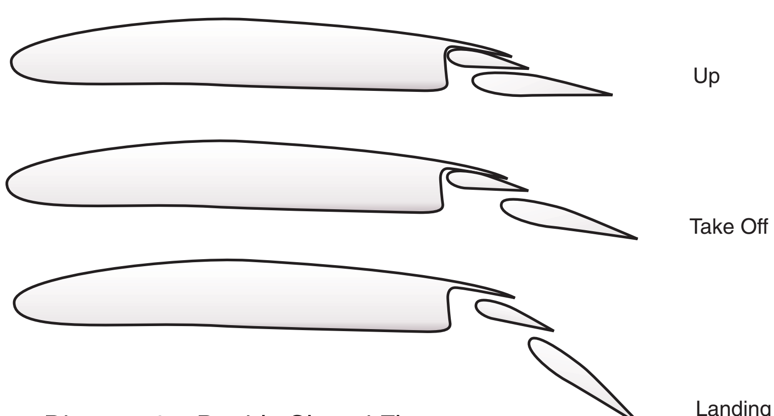


Diagram 6.5 Double Slotted Flaps

In this design, as well as the slot being formed between the leading edge of the flap and the trailing edge of the wing, a further slot is made through the flap behind its leading edge. As the flap is deployed successively from take-off to landing, both slots are brought into play, enhancing lift qualities. This type of flap gives a 70% increase in lift and delays the stalling angle to 18 degrees.

ZAP FLAPS

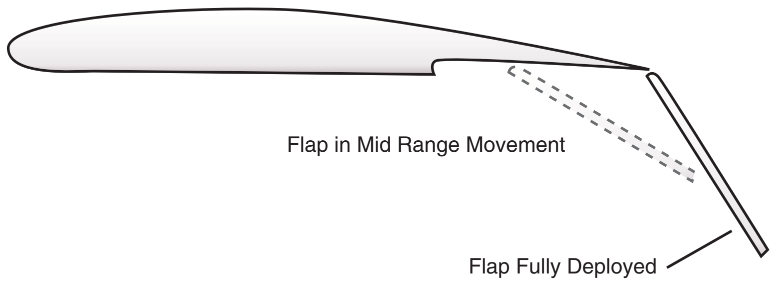


Diagram 6.6 Zap Flap

Zap flaps, pre-cursors of the Fowler flap, are designed to increase both wing area and camber. While increasing the lift of a basic wing by 90%, they reduce the stalling angle of the wing from 15° to 13°, create a nose down pitching moment, and create a lot of drag when fully deployed.

The flap formed by a section of the under surface of the wing moves backward and progressively downward with each increasing selection. Referring to diagram 6.6, showing a zap flap in the mid range and landing configuration, it can be seen that the operation of the zap flap is more complicated than that of the plain flap or split flap.

FOWLER FLAPS

Diagram 6.7 Fowler Flap

Fowler flaps increase a basic wing's lift by 90% without altering the stalling angle. This is achieved by increasing the camber and the wing's surface area. This design of flap is an aerofoil section, which when selected up, forms the rear lower surface of the wing, as diagram 6.7 shows. As the flaps are extended, they move progressively rearward and downward using a complicated system of rollers and guides.

DOUBLE SLOTTED FOWLER FLAPS

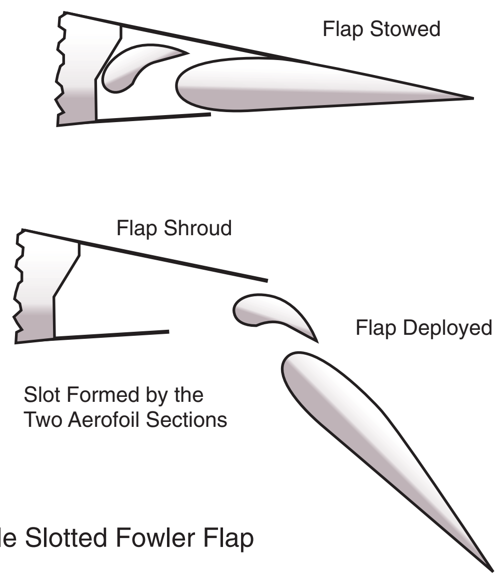


Diagram 6.8 Double Slotted Fowler Flap

In this design, a further small aerofoil section is mounted ahead of the main flap. As the flap is progressively deployed, initially it increases the surface area of the wing. Further deployment increases the camber and allows the high-energy air to flow through one slot.

Deployment toward landing flap further increases the camber and allows the high-energy air to flow through both slots. This flap system produces 100% extra lift and increases the stalling angle to 20°.

BLOWN FLAPS

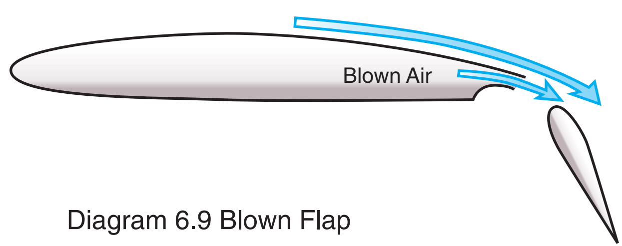


Diagram 6.9 Blown Flap

Slotted flaps make use of the high-energy air flowing under the wing by ducting it over the top surfaces of the flaps to re-energise the airflow coming from the top surface of the wing, pulling it downward and maintaining a laminar flow. The blown flap uses a jet of high velocity air taken as bleed air from the compressor of a turbine engine or dedicated blower and ducts this jet across the top surface of the flap. A further refinement of the blown flap is the jet flap. It completely replaces the mechanically moved aerofoil section with a linear vent that can be rotated, through which very high velocity air is blown. This draws the air flowing over the top surface of the wing down with it. This effectively increases the camber of the wing and the chord of the wing.

LEADING EDGE DEVICES

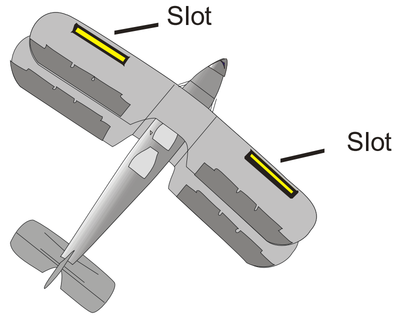


Diagram 6.10 Early Slots

The original leading edge devices were slots let into the aircraft's wing, aft of the leading edge, and in line with the ailerons, as in diagram 6.10. These ensured that when the wings were at high angles of attack, there was a laminar airflow across the ailerons, which maintained the ailerons' effectiveness.

SLOTS

Diagram 6.11 Fixed Slot

A shaped slot taken from the bottom surface to the top surface of the wing, just aft of the leading edge (as per diagram 6.11), ducts high-energy air over the top of the wing to re-energise the boundary layer. This ensures that a laminar flow passes across the remaining chord of the wing.

A basic slotted wing has 40% more lift than the same wing profile without the slot and increases the stalling angle to 20 degrees. The action of the slot does not alter the normal pitching moment of the wing. However, slots can cause an increase in drag for high-speed flight.

FIXED SLATS

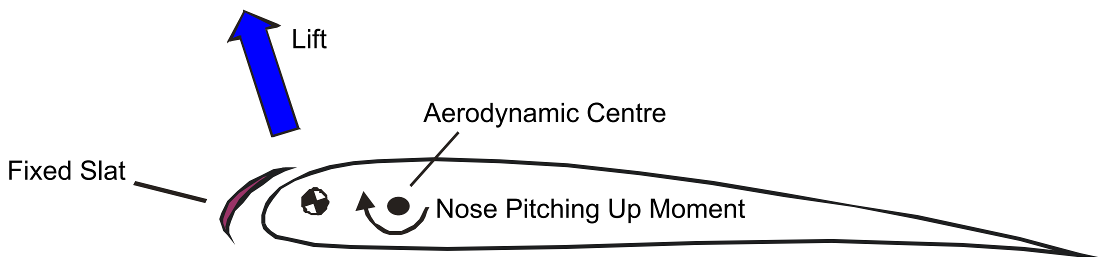


Diagram 6.12 Fixed Slat

To achieve the effectiveness of a slot along the leading edge of the whole wing, a small aerodynamically shaped strip of metal is fitted on brackets to the aircraft's leading edge. This re-energises the airflow across the whole wing.

The fixed slat produces 50% more lift than the basic wing. However, as the slot formed by the slat is forward of the leading edge, the effect of the accelerated airflow across the leading edge camber increases the lift forward of the wing's aerodynamic centre. This causes a nose up pitching moment. Fixed slats create drag at high airspeeds and are therefore not normally used on air transport aircraft. However, for aircraft where high lift and high angles of attack are required, such as crop sprayers, fixed slats are standard.

If slats were the only high lift devices fitted to the wing, they would cause design problems for the aircraft's undercarriage due to their effectiveness at higher angles of attack. It is therefore normal to use slots and slats in conjunction with trailing edge flaps.

MOVEABLE SLATS – CONTROLLED SLATS

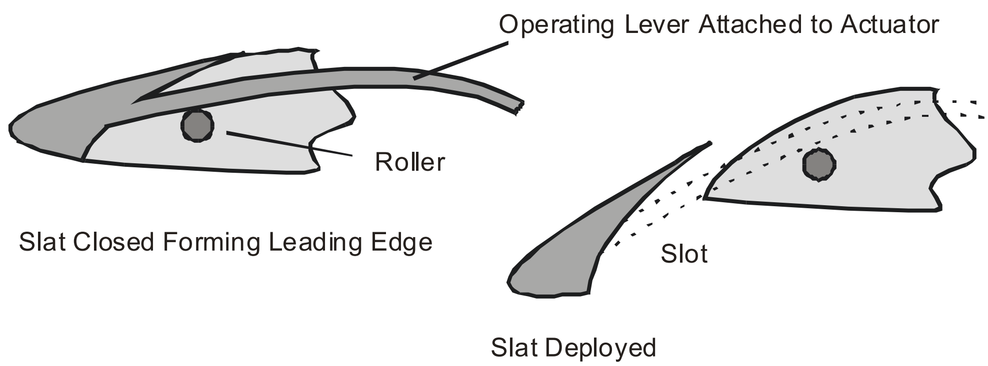


Diagram 6.13 Controlled Slats

Diagram 6.13 illustrates moveable, controlled slats. In this design, the pilot operates the slat. Its operation is linked to that of the trailing edge flaps to ensure that the aircraft is not subject to the nose up pitching action that slats on their own can create. In cruise, they are held against the wings, forming the leading edge.

MOVEABLE SLATS – AUTOMATIC SLATS

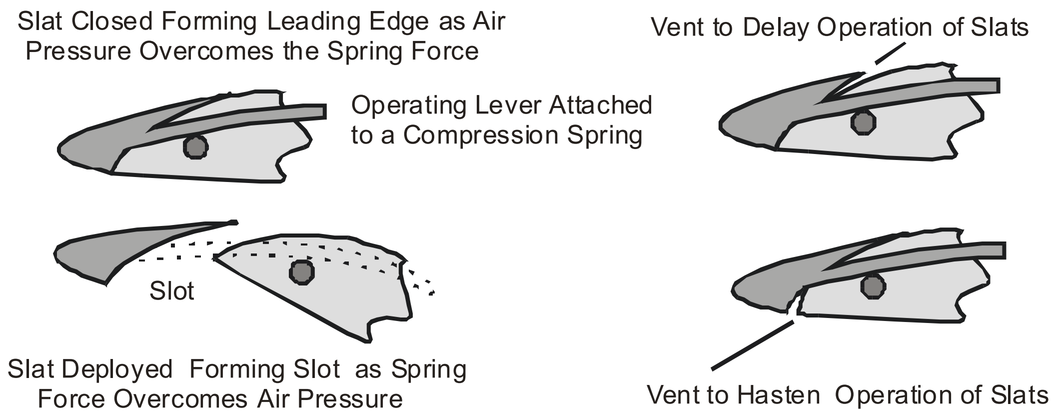


Diagram 6.14 Automatic Slats

In this design, the slat is held against the leading edge of the wing by air pressure. As the aircraft is pitched nose up, the pressure values alter, and the slat is moved forward by spring force.

If the slat has a vent formed along its top surface between its trailing edge and the wing's leading edge, the decrease in pressure over the cambered surface acts to hold the slat in the retracted position, delaying its opening. A vent formed on the under surface in the region of increased pressure under the wing acts to accelerate the slat's opening.

LEADING EDGE FLAP

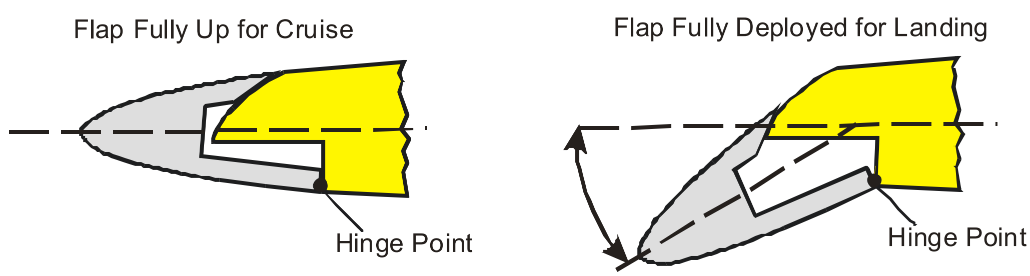


Diagram 6.15 Leading Edge Flap or Droop Snout

In the leading edge flap, or **droop snout** design, the forward section of the wing along its length is pivoted downward. This increases the wing's camber without unduly shortening the chord line. Like the leading edge slat, the effect is to increase the lift curve for a given wing section.

KRUGER FLAPS

Diagram 6.16 Kruger Flaps

Kruger flaps were specifically designed for turbine-powered aircraft, as they have thinner wing sections and were intended to run from wing root to almost the tip. However, they have been surpassed on modern air transport aircraft by the use of movable slats. If Kruger flaps are used, they are located inboard of the inner engines.

Note: Unless instructed to the contrary, for exam purposes, Kruger flaps are located between the wing root and the engine.

The Kruger flap is formed by hinging a panel along the under section of the leading edge. When the flap is deployed, it swings down and forward, as in diagram 6.16.

Its action is to vastly increase the camber of what would otherwise be a thin wing section, resulting when fully deployed in a 50% increase in lift. This creates a nose up pitching moment. If the Kruger flap is selected to intermediate settings, it can reduce the amount of lift produced by the wing.

BOUNDARY LAYER CONTROL

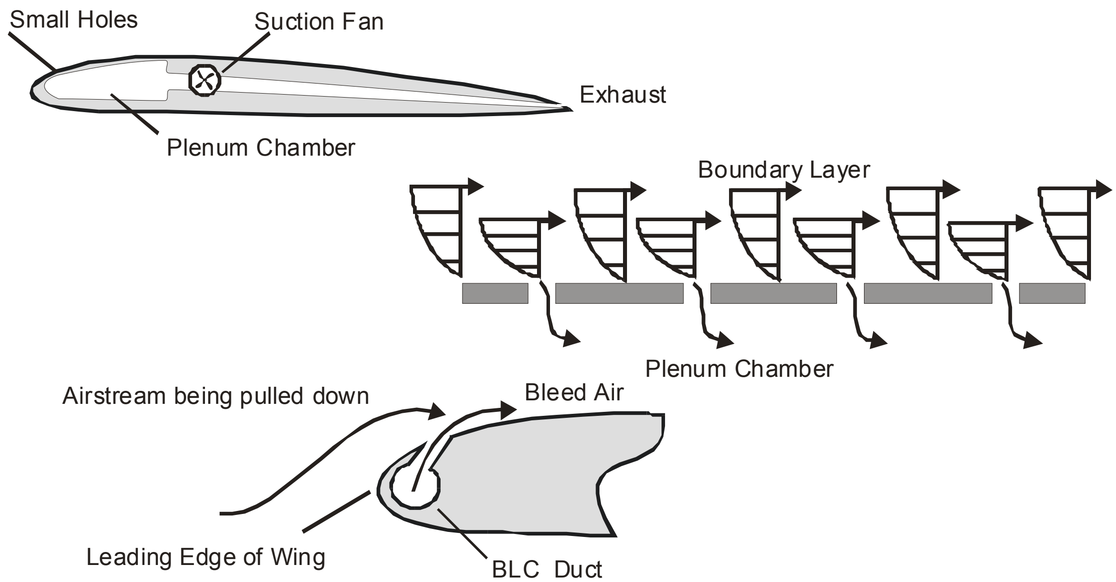


Diagram 6.17 Boundary Layer Control Via Suction and Blowing

Boundary layer control can be achieved by one of two methods, suction and blowing, as shown in diagram 6.17. In the suction method, air is drawn from the boundary layer, which has formed over the upper surface of the wing, through minute holes into a plenum chamber formed by having an inner skin. This reduces the thickness of the boundary layer and ensures that the airflow over the wing remains laminar longer. The air that has been drawn into the plenum chamber is ducted overboard at the trailing edge of the wing.

The disadvantage of this system is that it requires a myriad of minute holes (approximately 0.0025 inch diameter) through the wing's skin panels. These become blocked very easily, which reduces the efficiency of the system. Some air transport aircraft that make use of the suction method cover these panels with the top trailing edge of their slats.

Jetting high velocity air from vents through the wing's upper surface just behind the leading edge achieves Blowing. This accelerates the airflow close to the wing's skin and ensures that the boundary layer remains thin. The blown system is simpler but draws air from either a dedicated pump or engine. Both systems delay the onset of separation.

EFFECTIVENESS OF FLAPS AND SLATS

Where flaps and slats are deployed together, the nose down pitching moment of the trailing edge flaps is countered by the nose up pitching moment of the leading edge devices. The result is no effective pitching moment. The use of a slotted flap and slat combination increases the lift of the aerofoil by 75% and its stalling angle to 25 degrees. The use of a double-slotted fowler flap and slat combination increases the aerofoil's lift by 120% and the stalling angle to 28 degrees. This combination is frequently used on air transport aircraft.

FLAP OPERATION AND INDICATION

This next section looks at flap operation and indications for:

- Basic light aircraft
- Advanced light aircraft
- Air transport aircraft

BASIC FLAP SYSTEM FOR A LIGHT AIRCRAFT

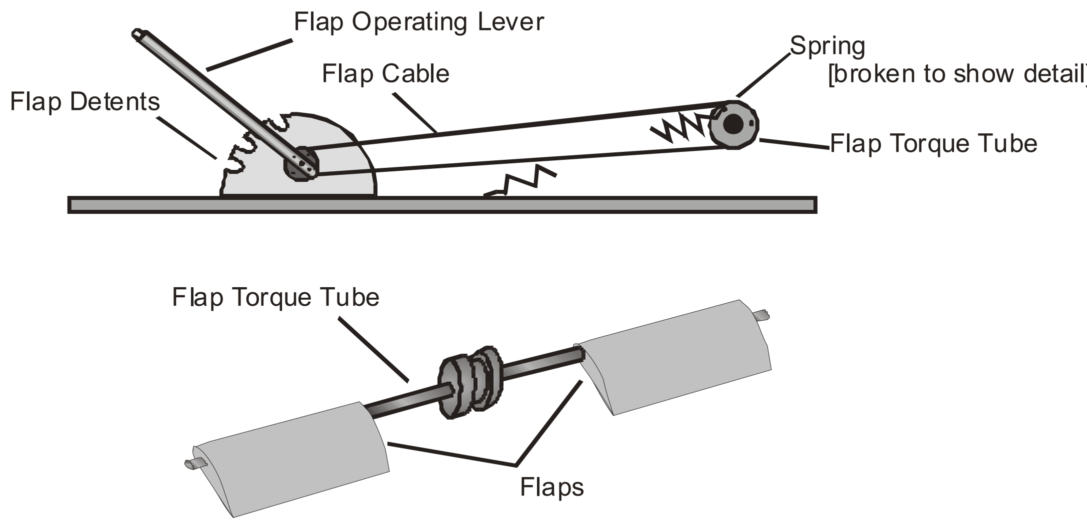


Diagram 6.18 Basic Light Aircraft Flap Arrangement

An unsophisticated flap system, similar to the one depicted in diagram 6.18, is adequate for basic light aircraft. The flaps are mounted on a common torque tube. This ensures that they function together, preventing asymmetry. Attached to the torque tube is a cable pulley. The flap operating cable is clamped to the pulley and the pulley attached to the flap-operating lever.

A tension spring is attached from the aircraft's structure to the torque tube pulley. This serves two functions, firstly in the event of a cable break, it retracts the flaps. Secondly, it increases the load on the pilot to give an indication of the loads being applied to the flaps' structure. Each flap setting from zero degrees to full landing flap has a detent position in the flap quadrant. The flap lever, similar to a car's hand-brake lever, has a push button to release the locking device so that the flap lever can be moved. The flap lever indicates the flaps' position (e.g. flat to the floor is zero flaps, etc.). In diagram 6.18, the flap is set to the third position.

FLAP CONTROLS FOR OTHER THAN BASIC LIGHT AIRCRAFT

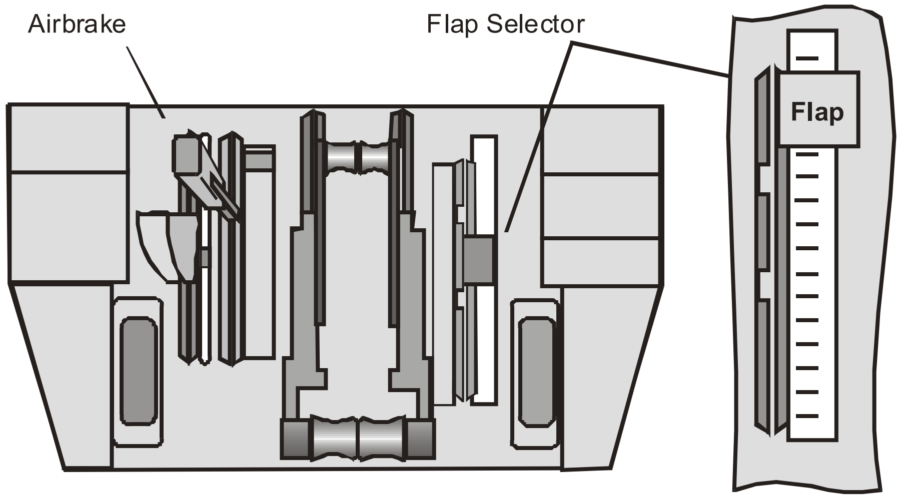


Diagram 6.19 Air Transport Aircraft Flap Selector

For flaps or other auxiliary devices, the selector-lever can be mounted on either the instrument panel or the centre console, as per diagram 6.19. As these controls must be instinctive, the selector lever must be:

- Forward or up for flaps up or auxiliary device stowed.
- Rearward or down for flaps down or auxiliary device deployed.

To ensure that the pilot can locate and identify the flap selector in the event of loss of cockpit lighting when flying in low visibility, the selector lever terminates in an aerofoil section knob.

ADVANCED LIGHT AIRCRAFT

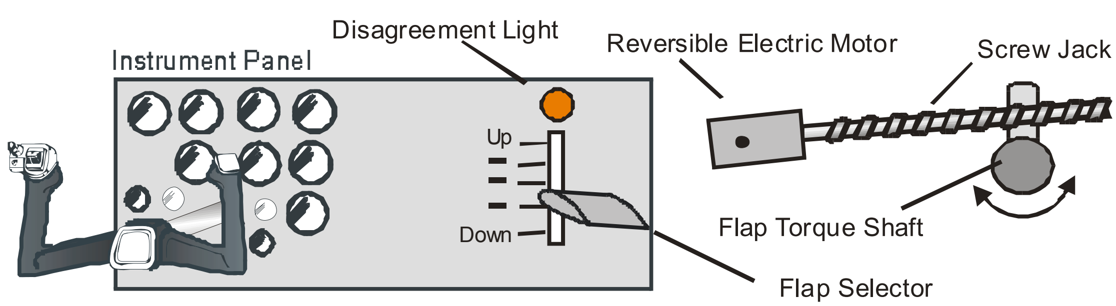


Diagram 6.20 Electrically Operated Flap System

In sophisticated light aircraft, electrical screw jacks are used to move the flaps then lock them in the chosen position. The floor-mounted lever, shown in diagram 6.18, is replaced with a sliding selector lever and disagreement light, mounted on the instrument panel (see diagram 6.20). Intermediate flap positions are marked off against the side of the slot, with each position detented.

The lever stops at each detent when the pilot selects a flap position. A purposeful move to the next setting is required. As the flap lags behind the selection, an amber disagreement light remains illuminated until the flap and the selector are in agreement. Once this occurs, the light goes out.

A reversible electric motor operates the flap screw jack. In many light aircraft, the actual switching mechanism is located on the flap's common torque shaft and takes the form of two limit switches, one for either direction. When the selector is moved away from a switch, it closes the circuit to the electric motor for that direction of movement, and the screw jack rotates until the torque tube reaches the location of the switch. As they contact, the power circuit opens and power is removed from the motor, locking the flaps in that position.

There are two failure conditions for this system. First, if power is lost to the screw jack's motor during selection, or if it jams. Second, in the event that the limit switch fails to stop the actuator at the selected position. In the first event, the disagreement light remains illuminated. To find the actual position of the flaps, the pilot moves the selector lever back toward the original position until the light goes out. The point at which the light goes out indicates the flaps' position. Each detent position is marked with the number of degrees. The pilot can thus work out the approximate deflection and adjust the aircraft's approach speed accordingly.

In the second condition, the motor runs until the internal load switch stops it. The flaps, therefore, move fully in the direction selected. In this case, the flaps' circuit breaker should be pulled to isolate the motor.

The simple flap lever system is sprung loaded to raise the flaps. For light aircraft with an electrically actuated flap system, play that is induced by wear can result in the flaps fluttering when at zero degrees. To overcome this, fixed tabs are fitted to both flaps, which act to hold the trailing edge up.

AIR TRANSPORT AIRCRAFT

Due to their larger flap areas and higher landing speeds, air transport aircraft require very powerful mechanisms to lower the flaps against the slipstream and to control their retraction.

This is normally achieved using the aircraft's hydraulic system, operating either linear hydraulic actuators or hydraulic motors driving drive shafts via a gearbox.

Where the wing flaps (leading edge and trailing edge) are fitted using separate drive sources, they must be synchronised by mechanical interconnection that is independent of the flap drive system or by an approved equivalent means. Where mechanical interconnection of surfaces is used, they are considered as a single surface.

However, the interconnection must be designed for the loads resulting when interconnected flap or slat surfaces on one side of the aircraft are jammed and immovable, while the surfaces on the other side are free to move, and the full power of the surface actuating system is applied. If a mechanical interconnection is not used, the designer has to show that the aircraft has safe flight characteristics with any combination of extreme positions of individual flaps. Designers of twin-engine aircraft have to ensure that the interconnection between the flaps is able to withstand asymmetrical loads resulting from one engine out and the other at take-off power.

FLAPS OPERATED BY INDEPENDENT HYDRAULIC LINEAR ACTUATOR

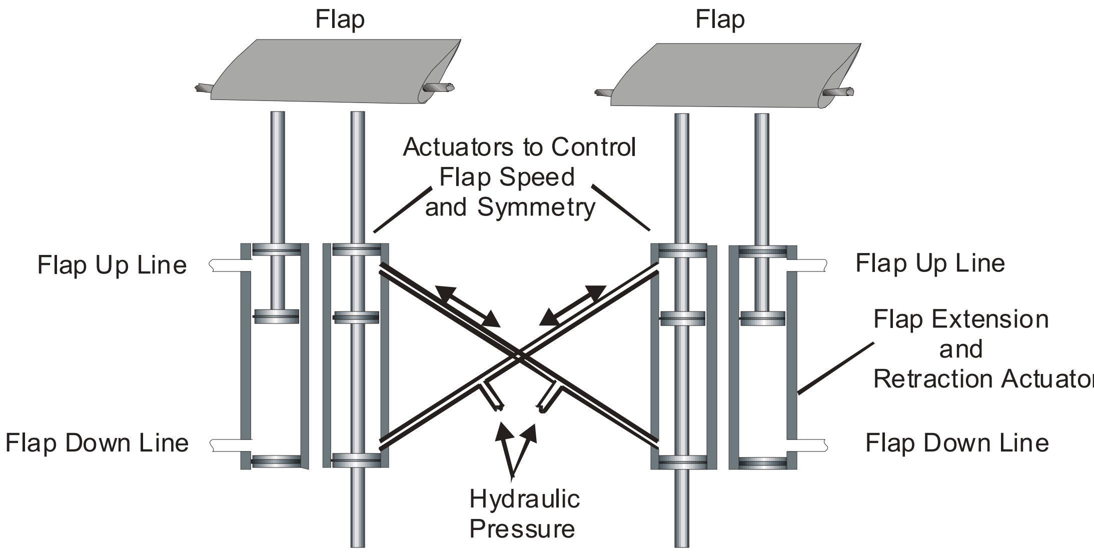


Diagram 6.21 Flaps Operated by Independent Hydraulic Actuators

Where independent hydraulic actuators are used (see diagram 6.21) to operate each wing flap, a system is required to ensure the flaps deploy at the same speed and to the same position. This is achieved by having two double acting balanced actuators, one attached to each flap and both actuators cross-connected hydraulically (see diagram above). As each flap is deflected, the fluid is displaced from both actuators and has to feed into the opposite chamber of the opposite actuator. This limits the speed and position of both flaps.

In this system, a pressure-relief valve termed a **blowback** valve is fitted in the downline. If the pilot over-speeds the flaps, the air pressure acting on the under surface could damage the flap or actuating mechanism. In this design, the increased air pressure felt back through the hydraulic system causes the blowback valve to unseat and the flaps are blown back (retracted), thus preventing damage.

To prevent the flaps from retracting too quickly, which could cause damage and stability problems, a one-way restrictor valve is fitted in the downline, allowing full flow into the actuator but restricting the flow from the actuator, thus slowing the flaps' movement.

FLAPS OPERATED BY POWER DRIVE UNIT

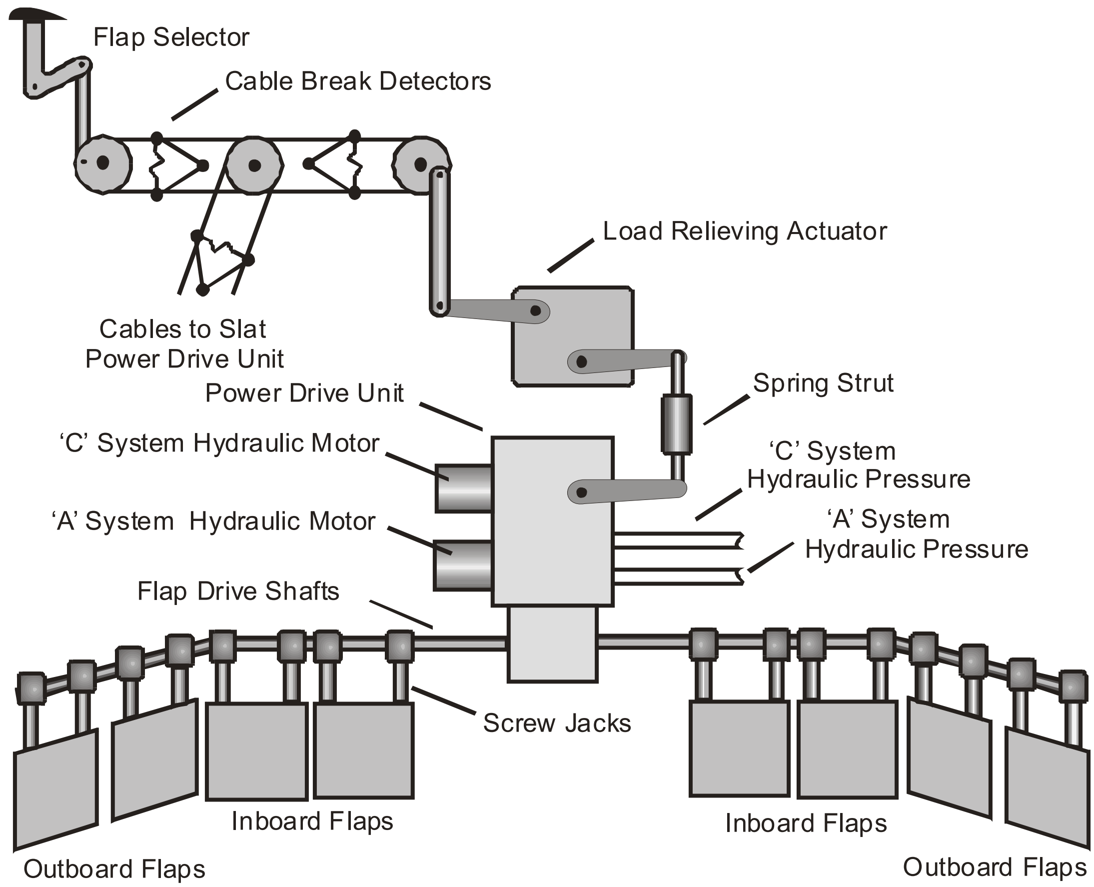


Diagram 6.22 Flaps Operated by Mechanical Interconnection

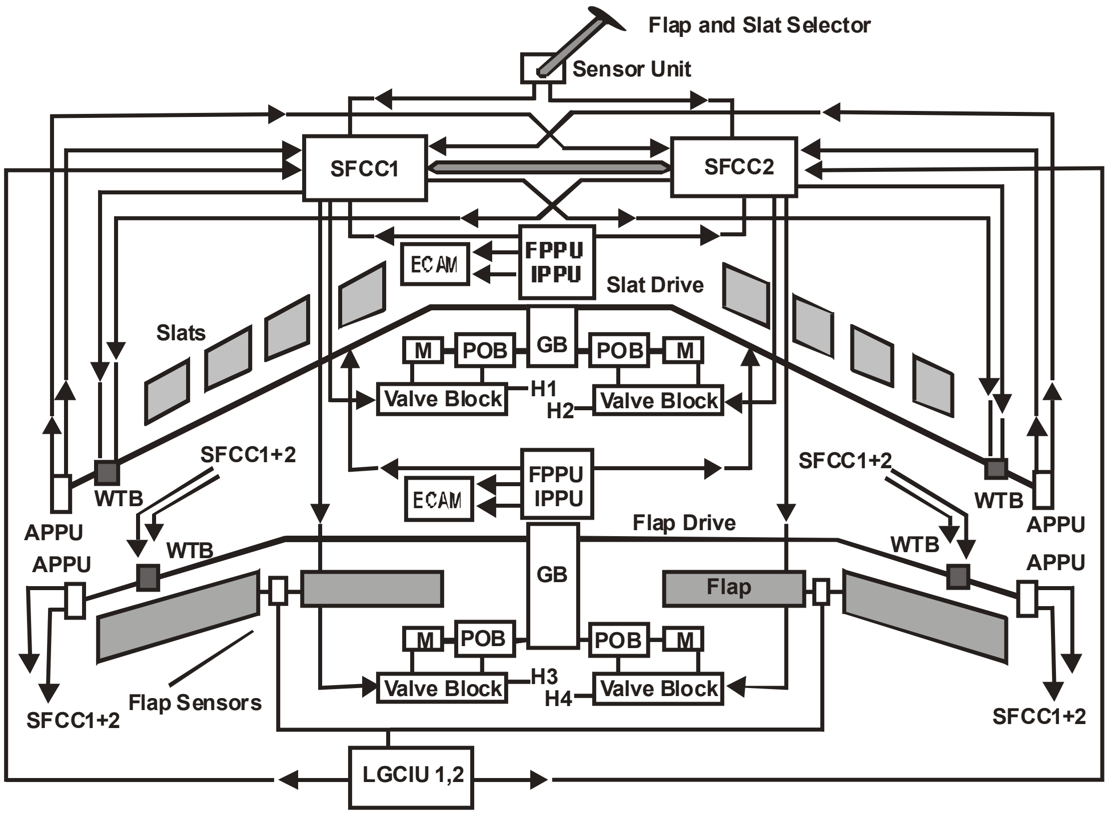
The more common system used on large air transport aircraft where Fowler or double slotted Fowler flaps are used is the Power Drive system. In this system, hydraulic motors drive screw jacks via a gearbox and shaft system as per diagram 6.22, which shows the older mechanical connection between the flight deck and the power-drive unit.

In this system, when a selection is made, both hydraulic motors drive the power-drive unit, which gears down the motors' rotation and increases the torque output. As all the screw jacks are on a common shaft, the trailing edge flaps deploy synchronously. Therefore, asymmetric flap selection is prevented. When the flaps reach the selected position, the hydraulic selector within the power drive unit cuts the flow to the hydraulic motors and applies a brake, thus locking the flaps in the correct position.

Failure of one of the hydraulic motors or loss of a hydraulic system reduces the speed of deployment but still allows full flap deflection. In the event of a cable break, or serious loss of cable tension, that would render the pilot's flap selector useless, a caution light illuminates on the flight deck. Some designs allow the flight crew to access the power drive unit and make a manual selection.

To prevent the flap structure and operating mechanism from becoming overstressed if lowered at too high an airspeed, the resistance to lowering in such a condition feeds back through the drive shafts to the power drive unit. Here a blowback mechanism allows the flaps to retract. As this is going against the pilot's selection, the spring strut compresses and absorbs any input signal. When the airload on the flaps decreases, the power-drive unit extends the flaps to the selected position.

FLY BY WIRE FLAP AND SLAT SYSTEM



Legend

SFCC = Slat and Flap Control Computer
FPPU = Feedback Position Pick up Unit
IPPU = Instrumentation Position Pick up Unit
APPU = Asymmetric Position Pick up Unit
POB = Power Off Brake
M = Motor
GB = Gearbox
H 1 etc = Hydraulic System 1
WTB = Wing Tip Brake
LGCU = Landing Gear Configuration Unit

Diagram 6.23 Fly by Wire Flat and Slat Control System Schematic

Diagram 6.23 is a schematic of a flap and slat control system for a fly by wire aircraft. Two identical computers, SFCC1 and SFCC2, form the heart of the system. Command inputs by the pilot are via the single flap/slat lever, the position of which is detected by the sensing unit.

The sensing unit sends the signal to both computers simultaneously. The two computers are cross-linked, so that they compare the input command and the feedback position information from the FPPUs. Both SFCCs send signals to the flap and slat PDUs, which consist of a hydraulic motor, power off brake, and valve block. The function of the valve block is to control the POB, speed, and direction of the motor. A separate hydraulic system supplies each PDU to enable system redundancy. The flap and slat PDUs are linked to the drive shafts via differential gearboxes. Both gearboxes are linked to feedback position pick up units and instrumentation position pick up units. The FPPUs send a signal back to each of the SFCCs, which compare the signals and adjust the valves in the valve blocks as required. The IPPUs send position signals to the pilots' ECAMs.

At the outer end of each drive shaft are a wing-tip brake and an asymmetric position pick up unit. The APPU sends signals to both SFCCs as to the actual position of the slats and flaps. If these signals vary across the wingspan, the SFCCs apply the WTB to lock the drive shaft and shut off the valves in the valve block. This action applies the POB.

Mounted on the flap drive shafts are flap position sensors. These are interlinked to the landing gear configuration unit which in turn signals the SFCCs. This is part of the Take-Off Configuration Warning System (TOCWS). In a fly by wire aircraft, this system warns the flight deck crew if they attempt a take-off without setting the aircraft up correctly (e.g. spoilers out, flaps at landing). If the condition is not corrected before brakes release, the system prevents take-off power from being applied to the engines.

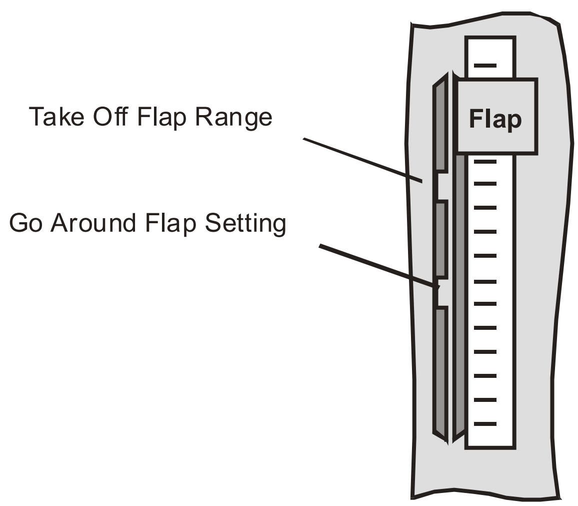


Diagram 6.24 Flap Selector

To assist the pilots, the flap selector lever has two detented or gated positions. The first position is for take-off flap. The second position is for go-around flap in the event of a baulked landing. These detents ensure that the pilot can position the flap selector and it is not inadvertently moved (see diagram 6.24).

FLAP AND SLAT POSITION INDICATIONS

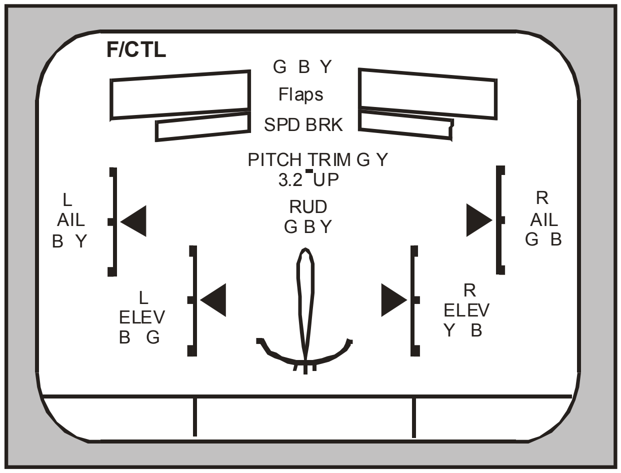


Diagram 6.25 ECAM Display of a Flight Control Systems Page

For basic light aircraft, the flap-selector lever can be used to inform the pilot of the actual position of the flaps. For larger aircraft, a position indicator is used. The older and more basic light twins can make use of a desynn transmitter and analogue gauge, which is calibrated with the flap positions from fully up to landing flap. For the fly by wire aircraft, the flap position information is displayed on the flight control systems page of the ECAM, as per diagram 6.25. This shows the positions of the controls and the hydraulic systems that are supplying pressure to them.

Chapter 7
Landing Gear — Undercarriages

ACKNOWLEDGEMENTS
We would like to thank and acknowledge:

For diagrams	7.20	Nose Undercarriage Leg	APPH Ltd.
	7.23	Body Gear	Dunlop Aircraft Tyres
	7.25	Landing Gear for a High Winged A/C	Dunlop Aircraft Tyres
	7.27	Bogie Layout	APPH Ltd.
	7.39	Floats	Cessna Aviation

INTRODUCTION
From the first flight of the Wright Brothers' Kitty Hawk Flyer, which took off from a small track-mounted trolley and landed on two skids, it became apparent that for practical purposes aircraft required wheels for ground operations. In these early days, bicycle wheels were used. As the Wright Brothers' engines improved, these wheels allowed the aircraft to take off without the need of the fixed rail and drop weight. Another benefit was in landing.

Since those first days, aircraft have been adapted to take off and land from the following surfaces:

- Water
- Snow and ice
- Unprepared hard surfaces
- Prepared hard surfaces

This allows aircraft to be categorised into three groups:

- Land planes
- Seaplanes/Floatplanes
- Amphibians

Land planes can take off from prepared or unprepared surfaces including snow and ice. Normally, land planes are fitted with wheeled landing gear. However, for operations on snow and ice, either **skis** or **bear paws** replace the wheels. In some cases, the skis or bear paws are retractable to allow the aircraft to operate from both snow and a hard surface.

Seaplanes are those where the fuselage is designed to be a floating hull. They only land and take off from water. **Floatplanes** are land planes where two floats, which allow the aircraft to take off and land on water, replace the wheeled undercarriage.

Amphibians are aircraft that can land or take off from water or prepared runways. Amphibious aircraft can either have a fuselage hull with retractable landing gear, or floats that incorporate retractable landing gear.

Originally, all land planes used fixed undercarriages. In the very early days, this amounted to a cross axle with two bicycle wheels attached to a V frame by bungee cords, which was in turn attached to the fuselage. The distance between the two wheels, the undercarriage **track**, was narrow, which led to poor landing and take-off stability. While the cross axle can still be found on some light fixed-gear aircraft, it is not in common use since, apart from the narrow track, the axle can create sufficient drag in long grass to tip a landing aircraft onto its nose.

With increases in aircraft weight, engine power, and aircraft performance, the designers replaced the V frames and bicycle wheels with wheels and tyres attached to undercarriage legs in an attempt to reduce the shock load when landing. In the quest for increased airspeed and operational efficiency, these gears were refined by fairing the landing gear legs and wheels to reduce the drag created at higher airspeeds. The ultimate drag reduction for landing gear is achieved by fully retracting it when not in use.

REQUIREMENTS FOR A MODERN UNDERCARRIAGE

The progress in the design of landing gear over the last century has led to the following specific requirements for air transport aircraft undercarriages that all designs must meet:

- Absorb the landing load and damp vibration
- Withstand side loads when landing and taxiing
- Support the aircraft on the ground when it is manoeuvring
- Provide minimum friction between the aircraft and the ground
- Possess a low coefficient of drag
- Withstand the flight air loads

DRAG AND UNDERCARRIAGES

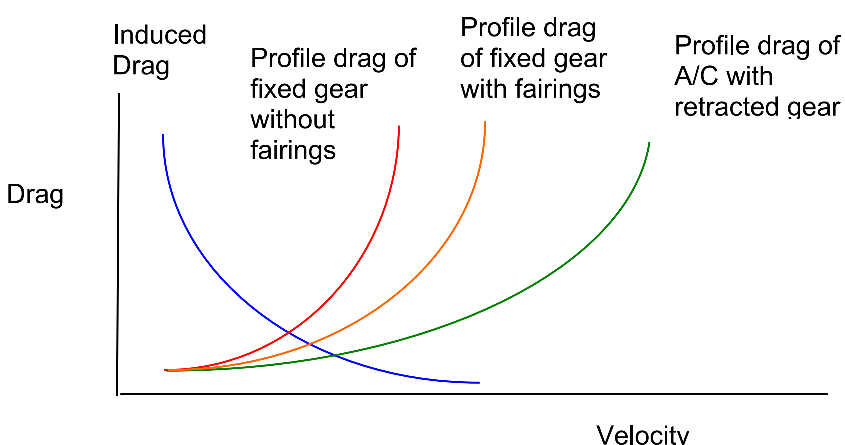


Diagram 7.1 Undercarriage Drag

As an aircraft's speed increases, its profile drag increases. To reduce drag and save fuel, the aircraft should be as streamlined as possible. For fixed landing gear, the method used to streamline the gear is to fit aerofoil cross-section **fairings** to the legs and more bulbous **spats** around the wheels, which conform where possible to the fineness ratio of 4:1.

Retracting the gear into wheel bays and fairing the apertures with doors removes the gear-created element of the profile drag. Diagram 7.1 represents the same aircraft fitted with a fixed gear without any fairing, then with the gear fully faired, and finally with a retractable gear raised.

LANDING

When an aircraft lands, there are three factors that come into play. These are mass, vertical velocity at touchdown (sink rate), and forward velocity at touchdown.

As mass times velocity equals force at touchdown, the aircraft exerts a downward force via its landing gear to the Earth. Obviously, the greater the landing mass or the greater the sink rate, the greater the force created. The Earth pushes back against the aircraft, as every force has an equal and opposite reaction. Unless controlled, this causes the aircraft to recoil or bounce back into the air before touching down again. Thus, a landing can continue in a series of kangaroo hops down the runway, or until the aircraft is pushed skyward but is lower than its stalling speed and crashes.

This is due to the fact that at the initial touchdown, the vertical component of the landing force is absorbed, leaving the mass. As the reaction is now greater than the mass alone, the aircraft bounces back into the air to restart the process.

SUSPENSION/SHOCK ABSORBER SYSTEMS FOR LIGHT AIRCRAFT

The original Wright Flyer had no shock absorbing system as its landing gear took the form of a wooden sledge. Attachment of pneumatic bicycle wheels to later versions gave very little suspension, for like all pneumatic tyres, the mass of the vehicle to which the tyres are fitted is supported by the gas pressure acting against the tyre cover.

To improve the suspension, the following systems were introduced:

- Bungee cords/bungee blocks
- Spring steel legs
- Torsion bar

BUNGEE CORDS/BUNGEE BLOCKS

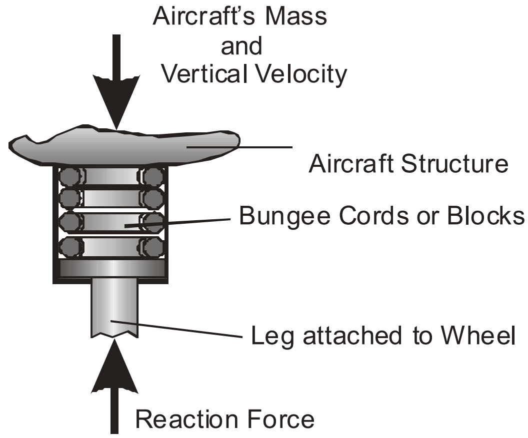


Diagram 7.2 Bungee Cords / Bungee Blocks

Diagram 7.2 illustrates bungee cords/bungee blocks, a system used on early aircraft and still in use on many aircraft. It consists of a series of bungee cord rings or rubber blocks (bungee blocks) mounted inside a tubular housing, which is connected to the aircraft's structure. The lower section of the leg with the wheel attached presses upward against the bungee blocks.

As the aircraft lands, the action of the upper and lower leg is to compress the bungee blocks. This act of compression absorbs energy from the touchdown and slows the sink rate. It also reduces the spine-jarring thump that early rigid systems subjected their pilots to, as part of the landing load has been absorbed into the bungees prior to it being fed into the aircraft's structure. As this system is un-damped, the reaction force from the touchdown and the recoil from the compressed bungees push the aircraft upward. However, under normal conditions, the aircraft bounces up and down without recoiling into the air. During taxiing, each bump causes the aircraft to sway and bounce.

SPRING STEEL LEG

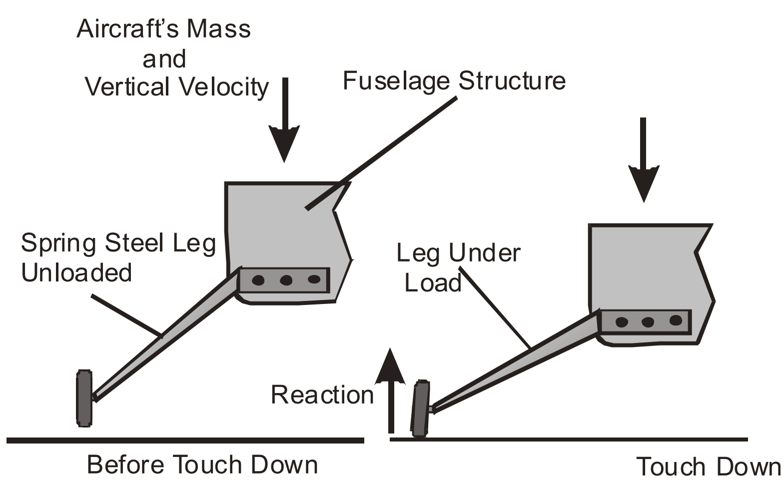


Diagram 7.3 Action of a Spring Steel Leg

This system is frequently used on very light G.A. aircraft. It consists of several strips of spring grade steel clamped together to form a bar. One end is attached to the fuselage structure, the other to a stub axle, as shown in diagram 7.3. On landing, the force of the aircraft sinking against the resistance of the runway causes the spring steel legs to start bending and absorbing the landing energy. As with the bungee system, energy absorbed by the spring steel is dissipated back into the fuselage structure at a controlled rate. Again, the aircraft sways and bounces as it taxis over uneven ground.

TORSION BAR

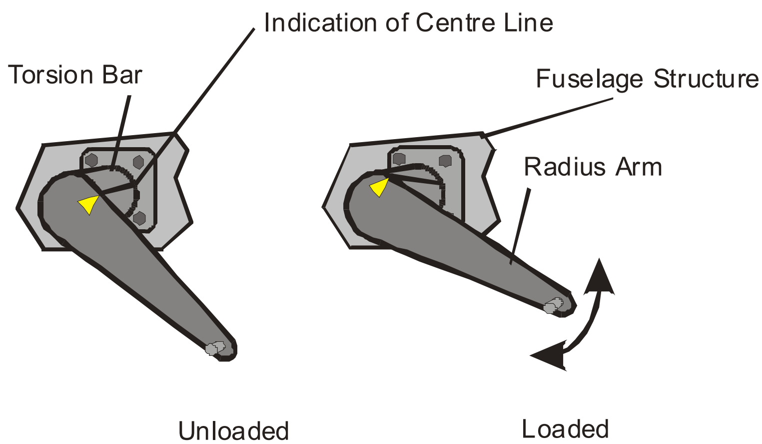


Diagram 7.4 Torsion Bar

The torsion bar, made of an elastic material that can withstand torsional loading, is attached to the fuselage structure at one end. A bearing supports the torsion bar where it passes through the fuselage skin. It is attached to a **radius arm**, so named as in its action it scribes an arc of a circle, as diagram 7.4 shows.

In flight, the torsion bar is unloaded. On touchdown, the radius leg moves backward and upward, applying a torque load to the torsion bar. As with the bungee system and spring leg system, the landing load energy absorbed by the torsion bar is dissipated back into the fuselage structure at a controlled rate. Again, the aircraft sways and bounces as it taxis over uneven ground.

As can be seen in diagram 7.4, these systems do not absorb the landing load other than controlling the rate at which the load is dissipated into the aircraft's structure. To control the landing load, absorb the shock, and control the rate of recoil, oleo-pneumatic shock absorbers are used.

BASIC PRINCIPLE OF AN OLEO-PNEUMATIC SHOCK ABSORBER

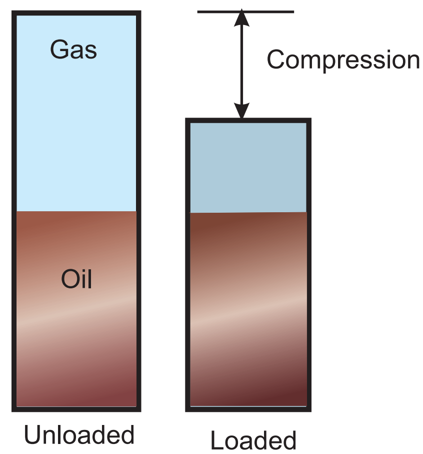


Diagram 7.5 Basic Principle of an Oleo

Oleo-pneumatic shock absorbers, generically referred to as **oleos**, function on the principle that fluid is considered incompressible, and that gas can be compressed. The pressure raised in the gas is equal to the force exerted in compressing it.

As the gas is compressed, raising its pressure, energy is absorbed and converted into heat. This is dissipated into the enclosing container by conduction and then to atmosphere by convection and radiation. Controlling the rate at which the gas can compress and expand allows the landing load and recoil action to be controlled. Oleo shock absorbers can only absorb vertical loads.

THE GAS CHARGE

The gas charge, normally nitrogen as it is inert (can be compressed air for light G.A. aircraft), supports the weight of the aircraft. The reason for using an inert gas is to prevent **dieseling**. If air is used in place of an inert gas, as the mineral fluid is kerosene based, there is a danger that the oil vapour within the leg will explode should the temperature and pressure reach a sufficient value. Aircraft operated in the air transport role must use nitrogen or another suitable inert gas for the gas charge.

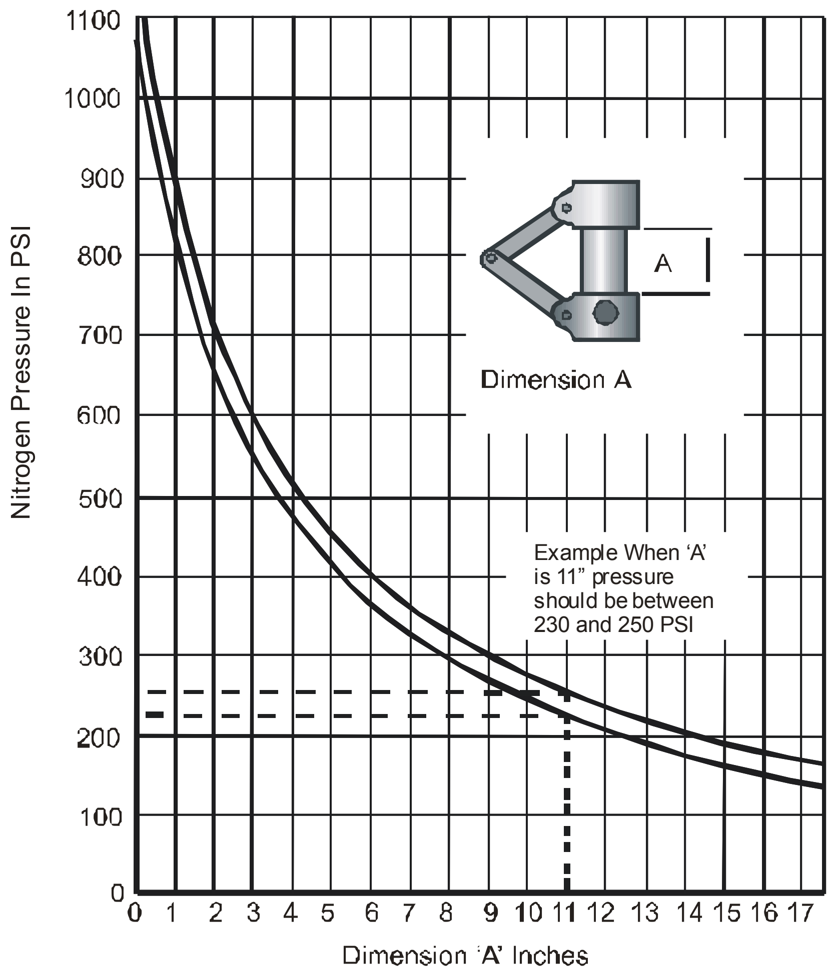


Diagram 7.6 Pressure / Extension Graph

A gas pressure versus extension graph, shown in diagram 7.6, allows the engineers to check that the oleo is correctly charged. Increasing the aircraft's mass decreases the oleo extension, and the gas charge pressure increases.

THE OIL CHARGE

An oil charge acts as a damper to control both the rate of compression during initial touchdown landing load and the recoil action of the leg. As the gas pressure increases, so does the pressure within the oil column.There are two main designs for oleos: separated and unseparated. This refers to whether the oil and gas interface or have a physical separator between them. Unseparated oleos are cheaper to manufacture and are frequently used on light aircraft with both fixed and retractable landing gear. One of the disadvantages of this system is that the gas/oil charges can mix, leading to poor damping (spongy operation).

UNSEPARATED OLEO

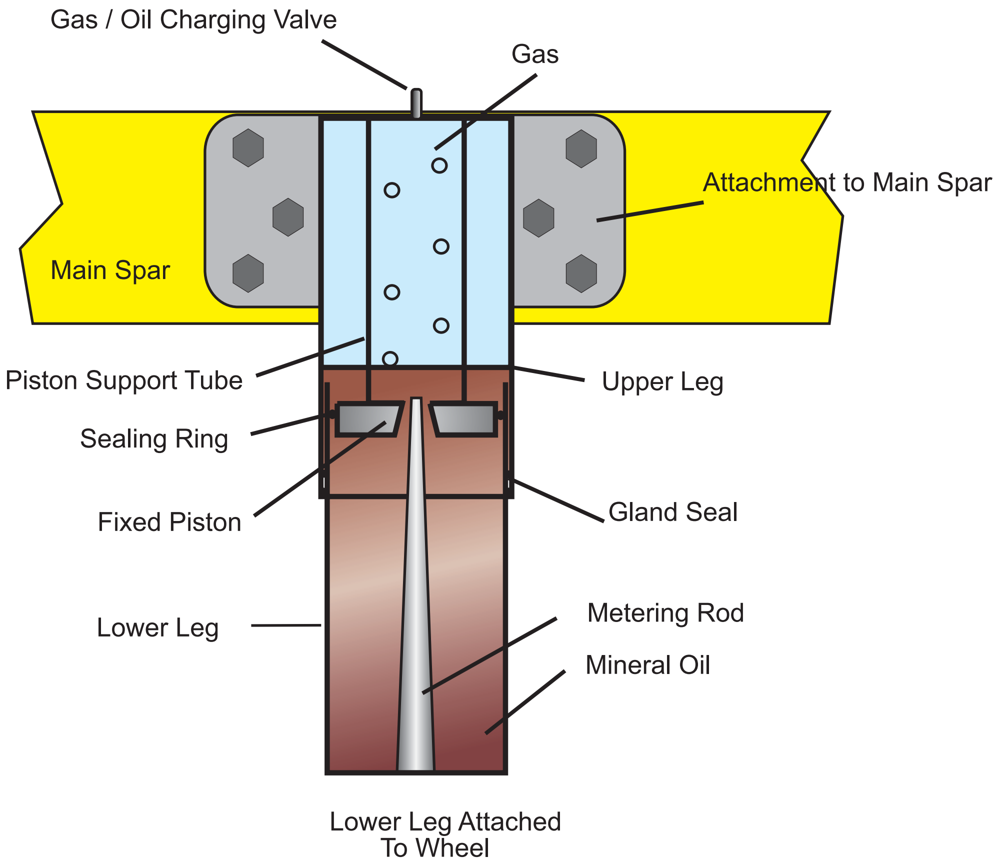


Diagram 7.7 Unseparated Oleo Pneumatic Strut

Diagram 7.7 shows a basic light aircraft unseparated oleo. The pressure in the gas charge supports the aircraft's mass. This acts directly on the fluid, raising its pressure. When the aircraft is in flight, the gas pressure and the weight of the lower leg and wheel assembly extend the oleo leg to its maximum length. As the wheels touch down, the lower leg is pushed upward as the upper leg moves downward. This action decreases the volume inside the cylinder.

As the fixed piston is attached to the upper leg, this compression forces the piston into the fluid. The pressure in the fluid increases. However, the fluid is able to flow from the chamber in the lower leg via an orifice in the fixed piston. The rate at which the fluid can transfer from the chamber in the lower leg is controlled by the **metering rod**, which is moving upward through the orifice as the leg becomes compressed. This acts as a variable restrictor, as it progressively reduces the size of the orifice.

This action allows the oleo to absorb the energy of the landing load by converting it into pressure. The pressure that has built up inside the gas charge acts to make the aircraft recoil. This is controlled by limiting the speed at which the oleo is able to extend. As the gas pressure acts to push the aircraft upward, the fluid flow rate through the restricted orifice provides this limiting action. Throughout these operations, energy is dissipated as heat into the leg castings. During ground manoeuvres, the oleos absorb any unevenness in the pavement surfaces.

Note: Some designs fit the metering rod in the top cylinder and the piston to the bottom cylinder. However, the function remains the same.

TORQUE LINK

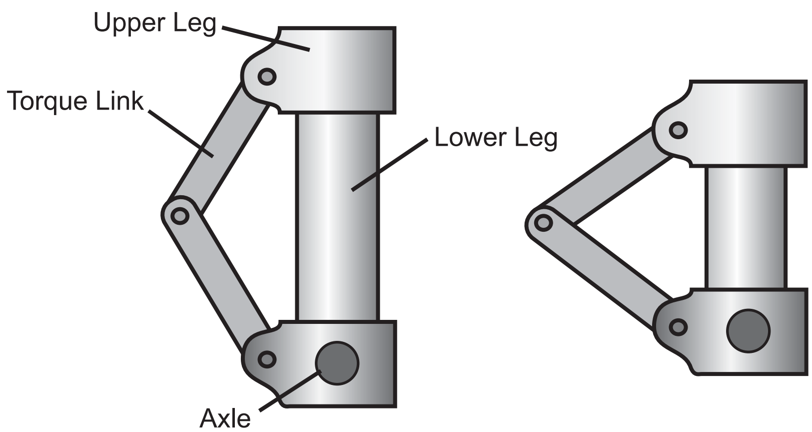


Diagram 7.8 Torque Links

The oleo leg is manufactured as two tubes, one sliding inside the other. As shown in diagram 7.8, a torque link is fitted to ensure that they remain in alignment (hence the wheel remains in alignment with the aircraft's centreline). The whole torque link has three pivot points, the upper link at its attachment to the upper leg, the lower link at its attachment to the lower leg, and in the middle where the two links are joined together.

This allows the torque link to move as the leg shortens and lengthens. The link is not designed to cope with the excessive side loads that can be created by trying to turn in a tighter circle than the specified minimum radius. This creates wear in the link joints that can lead to wheel shimmy.

HEAVY LANDINGS

In the event that the landing load is greater than the load that the oleo was designed for, the oleo can **bottom out**. This is where the leg's travel is greater than the available distance, and the top casting can physically touch the bottom casting.

Obviously, the oleo cannot function properly in this condition, and the shock that it would normally absorb is fed into the aircraft structure. In this situation, the aircraft needs to undergo heavy landing checks to ensure that it has not suffered any structural damage.

OIL CHARGE – LEAKS OR LOSSES

The most likely place for an oleo oil leak is from the gland seal at the bottom of the upper leg. As oil is lost, there is a loss in the oleo's damping action. The aircraft is spongy in the landing roll out and ground manoeuvres. The aircraft also appears to be unbalanced (i.e. nose down, wing down, etc.) on the ground. If an aircraft appears to be low on its oleos and this is remedied by adding nitrogen without ascertaining if the oil charge is correct (known as "adding an inch of nitrogen"), it can lead to the aircraft losing its damping action and bottoming out the oleo legs.

PROP CLEARANCE REQUIREMENT

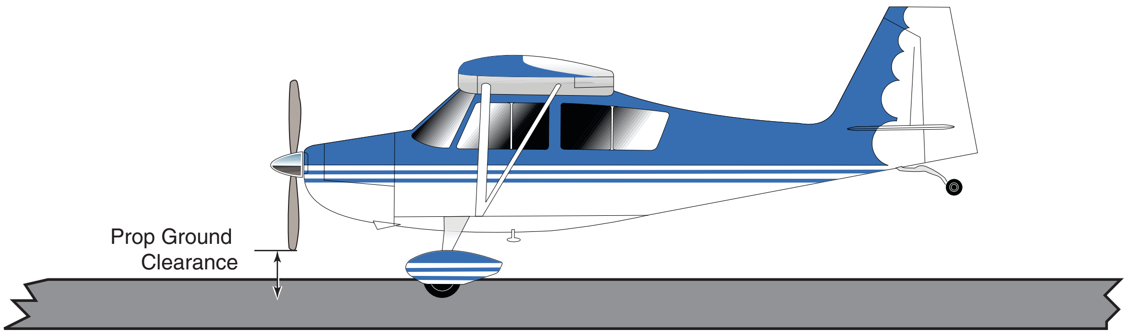


Diagram 7.9 Propeller Clearance for a Tail Dragger

The designers of JAR 23 and 25 certified aircraft that are piston or turboprop powered must take into account the effect of either the nose leg for a tricycle undercarriage aircraft, or the main gear for a taildragger, collapsing. This is to ensure that the propellers do not contact the ground in the landing run. Diagram 7.9 shows the prop clearance for a tail-wheel aircraft.

SEPARATED OLEO

Medium and large aircraft use the separated oleo system not only for safety against dieseling, but also because the system is more able to cope with the greater recoil force that these aircraft produce. Diagram 7.10 shows the basic component parts of this design. In this system, the upper and lower legs form a cylinder that is divided into two separate chambers by a free floating piston or separator. Above the separator is the fluid chamber, and below the separator is the gas chamber.

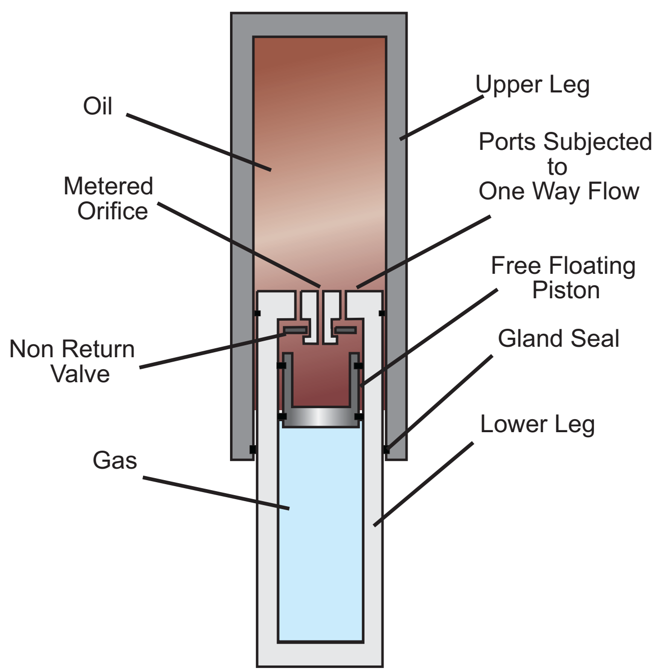


Diagram 7.10 Parts of a Separated Oleo

Attached to the lower leg, but in the fluid chamber, is a piston. This has a simple non-return valve that opens and closes a series of ports, and a fixed metered orifice allowing fluid to pass from one side to the other, at two differently controlled rates. The internal operation of the oleo in four situations is shown in diagram 7.11 and explained in the following paragraphs.

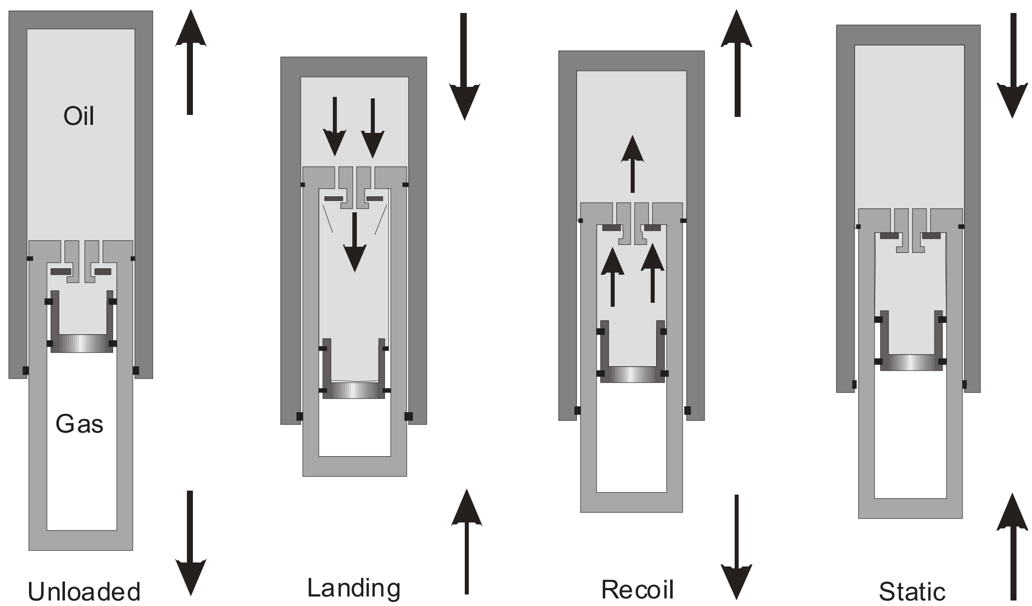


Diagram 7.11 Operation of a Separated Oleo Pneumatic Strut

UNLOADED

In flight with the gear lowered, the oleo is unloaded and extended to its maximum length, and the internal volume is at its greatest. However, as the oil charge is a fixed volume at a given pressure, the gas expands and pushes the piston upward, and the internal pressures equalise.

LANDING LOAD

On touchdown, the oleo shortens, forcing the piston upward into the oil. As oil is considered incompressible at these pressures, the upward movement of the piston forces the oil through the non-return valves and the metered orifice into the lower portion of its chamber in the lower leg. This has a smaller diameter, so the oil acts against the piston pushing it downward, compressing the gas and raising its pressure.

RECOIL

When the initial touchdown load has been absorbed and the recoil action starts, the gas pressure pushes the piston upward, trying to expel the fluid and lengthen the oleo. The rate of recoil is less than the original compression, as the non-return valve action blocks off all but the metered orifice, which has a fixed flow rate. As the fluid transfers across the piston, the pressure in the gas drops. In the landing run, the oleo goes through several progressively smaller compression/recoil cycles.

STATIC LOAD

When the aircraft is static (i.e. stationary on the ground and without any mass added or subtracted), the gas and fluid find their own level as the gas pressure, hence fluid pressure, equals the weight of the aircraft. Operators must be aware that any mass added to the aircraft results in the oleo shortening, as the gas compresses to equal the load applied. This basic fact catches many who leave equipment under an aircraft or boarding steps too close to the fuselage doors before a refuel commences.

TAIL WHEELS

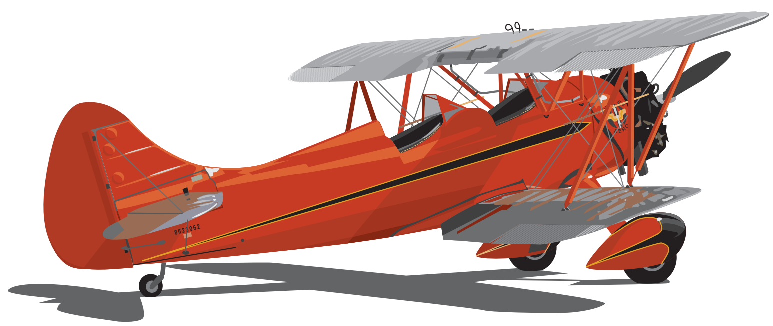


Diagram 7.12 Taildragger

In this design, the aircraft is supported by a small diameter wheel mounted at the rear of the fuselage with two main wheels located forward under the heaviest/strongest section of the wing, which simplifies attachment to the wing's structure. This also allows the main wheels' track to be increased, which adds to ground stability. To aid ground manoeuvring and allow the aircraft to be steered using the rudder, the tail wheel takes the form of a castor where the wheel's axle is located behind the pivot.

FULLY CASTORING TAIL WHEELS

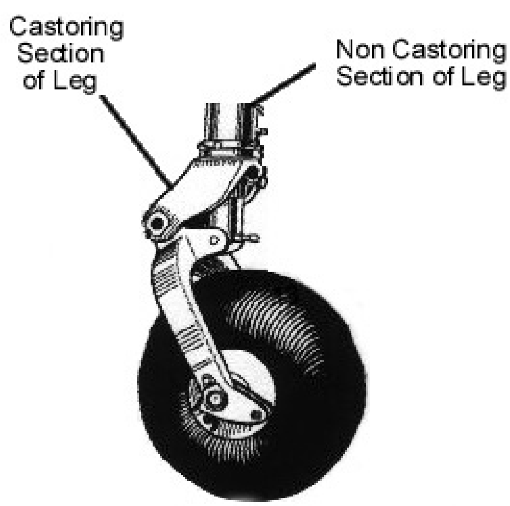


Diagram 7.13 Castoring Wheel

These early tail wheels were allowed to pivot freely, termed **fully castoring,** similar to the wheels of a shopping trolley. Diagram 7.13 shows a fully castoring wheel. This allows the pilot to give a burst of prop wash over a deflected rudder to turn the aircraft for taxiing. However, unless the aircraft takes off directly into wind, there is a danger that the tail will **weather cock** the aircraft into wind on the take-off or landing run. This became a problem for aircraft using prepared concrete runways in lieu of grass fields, and led to the introduction of the lockable tail wheel.

TAIL WHEEL — GROUND LOOPING

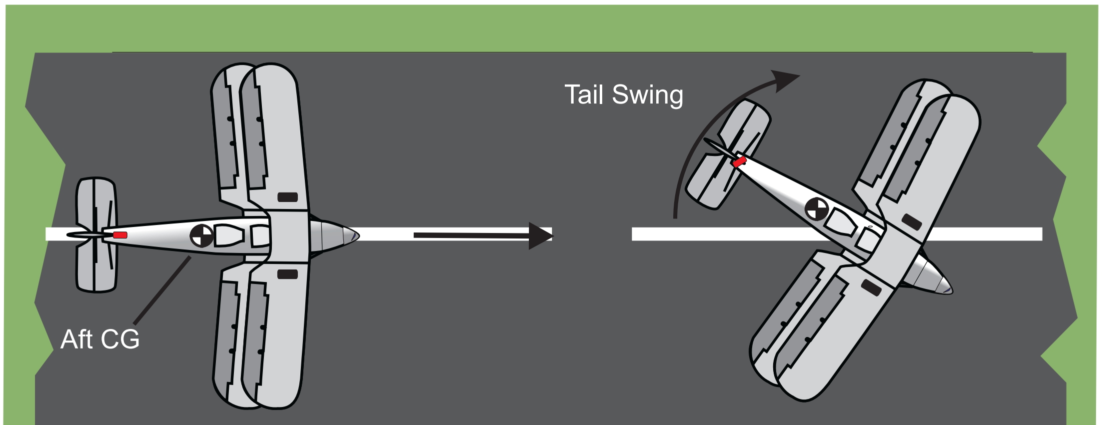


Diagram 7.14 Ground Looping

The other problem with fully castoring tail wheels is that any lateral movement of the rear of the fuselage results in the tail wheel pivoting and following the movement. If the turning moment at the tail of the aircraft is large enough, the aircraft can pivot around and face the direction from which it came. This is termed ground looping. The further aft the CG, the greater the turning moment created by any lateral movement of the tail, as in diagram 7.14.

LOCKABLE TAIL WHEEL — PILOT-OPERATED

The lockable tail wheel takes two forms, pilot-operated and load-operated. In the pilot-operated system used on larger aircraft, when both the aircraft and tail wheel are longitudinally aligned on the runway, the pilot releases a spring-loaded pin to lock the tail wheel fore and aft. As the tail wheel is unable to pivot, any tendency for the tail of the aircraft to swing laterally creates a side load on the wheel. The friction between the tyre and the ground resists this lateral turning moment, thus preventing the tail from swinging.

LOCKABLE TAIL WHEEL — LOAD-OPERATED

Diagram 7.15 Load-Operated Tail Wheel

In the load-operated design used on light aircraft, the tail wheel is spring-loaded to remain fore and aft until the side load applied to the wheel is greater than the spring's force. This normally occurs when the individual wheel brakes are applied, allowing the pilot to steer the aircraft on the ground. Otherwise, the wheel remains locked fore and aft. A reduction in the springs' force can result in the aircraft unexpectedly ground looping.

TAIL WHEELS AND NOSING OVER

Apart from ground looping, the other problem with tail wheel aircraft is the possibility of the aircraft tipping onto its nose when braking on landing. The challenges of handling tail wheel aircraft led to the adoption of nose wheels for air transport aircraft.

NOSE WHEELS

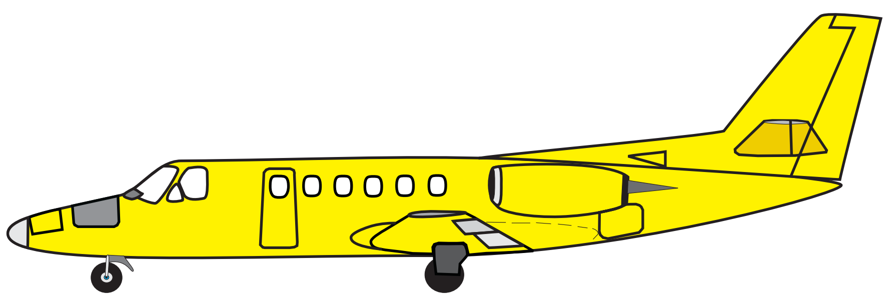


Diagram 7.16 Tricycle Undercarriage

Nose wheels, or in JAA terms, **auxiliary wheels**, allow aircraft to be almost horizontal when parked or ground manoeuvring. This gives the pilot the advantage of a clear view ahead without having to weave the aircraft. It also allows the aircraft to be steered so that the fuselage follows the nose, rather than the tail having to be swung in the opposite direction to the intended course. A tricycle arrangement is such that the nose wheel forms the apex of the landing gear triangle.

FULLY CASTORING NOSE WHEEL

For some light aircraft, the nose wheel's axle is located behind the vertical leg to simplify the steering. This arrangement allows the wheel to pivot like the front wheels of a shopping trolley. To steer the aircraft, the pilot uses differential braking, whereby one main wheel is braked more than the other. This results in the aircraft pivoting around the braked wheel. As the nose wheel is behind the leg, it pivots to follow the fuselage.

While this system is simple, it has the disadvantage of suffering from **shimmy**. To overcome this, a specialist tyre called a **Marstrand tyre** is used. Both shimmy and Marstrand tyres are covered later in these notes. As this form of steering is not precise, most modern tricycle undercarriage light aircraft use a direct steering system.

STEERABLE NOSE WHEELS — LIGHT AIRCRAFT WITH FIXED NDERCARRIAGES

Steerable nose wheels are directly linked to the pilot's rudder pedals. For most light aircraft, the pilot is able to steer the nose wheel through an arc 37° each side of the aircraft's longitudinal centreline. A **bungee spring** is incorporated into both steering rods. This prevents excessive force created by sudden large rudder pedal movements from being transmitted to the nose wheel steering system when the wheel is on the ground.

STEERABLE NOSE WHEELS — LIGHT AIRCRAFT WITH RETRACTABLE UNDERCARRIAGES

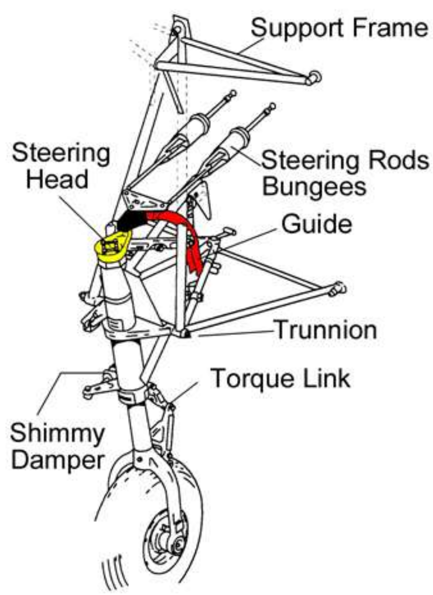


Diagram 7.17 Steering for a Retractable Nose

On all aircraft with retractable gear, the nose wheel bays are very narrow in relation to the nose gear itself. This requires the nose wheel to be in a fore and aft condition before it is raised. Otherwise, the wheel impacts against the bay's structure, damaging it and possibly jamming, thus preventing it from being fully retracted or lowered. During flight, the nose wheel's steering must be disconnected from the rudder pedals to prevent any input turning the wheel while it is retracted.

Referring to diagram 7.17, the steering head of the retractable leg engages into a pivoting link (blocked in black), which is connected to the steering rods. On retraction, the steering head moves out of the pivoting link and follows a guide, which centralises the wheel.

SHIMMY DAMPERS

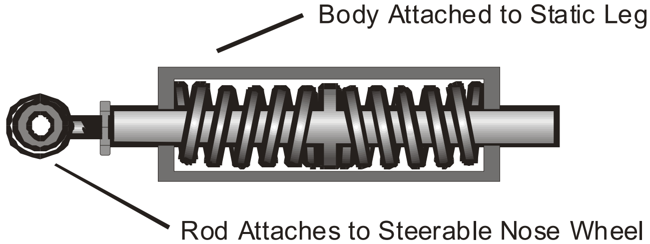


Diagram 7.18 Light Aircraft Shimmy Damper

Where the nose wheel is steerable, it is possible for the tyre to oscillate either side of the aircraft's track due to play in the linkages. To overcome this, a shimmy damper is fitted. On light aircraft, this consists of a housing attached to the upper leg, which contains a piston and two compression springs, as shown in diagram 7.18. The piston is connected to the steerable lower leg. When the wheel is fore and aft, the compression in the springs is equal. If the wheel starts to turn, it increases the compression, resisting its movement, which acts to return it to the fore and aft position. Worn shimmy dampers and torque links are the most common cause of nose wheel shimmy.

HYDRAULIC SHIMMY DAMPER

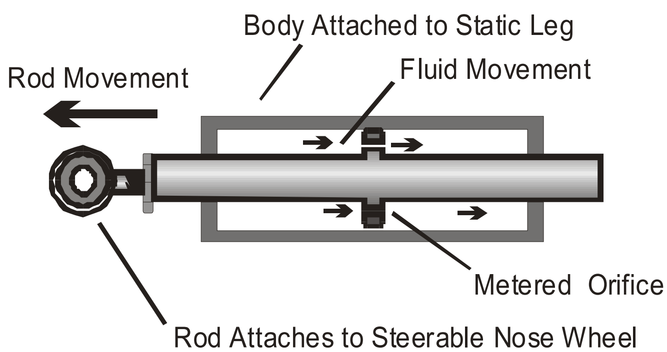


Diagram 7.19 Hydraulic Shimmy Damper

For larger aircraft, hydraulic damper struts are used similar to the one shown in diagram 7.19. This system uses a balanced actuator with metered orifices drilled through the piston. As the lower leg starts to oscillate, the piston is moved through the fluid. As the orifices limit the rate of transfer from one side of the piston to the other, damping occurs, and the pressure build-up against the piston resists the turning force of the oscillation. Attaching the shimmy damper to the steering ring allows the nose wheel to be turned and still maintain shimmy damping.

STEERABLE NOSE WHEELS FOR MEDIUM AND LARGE AIRCRAFT

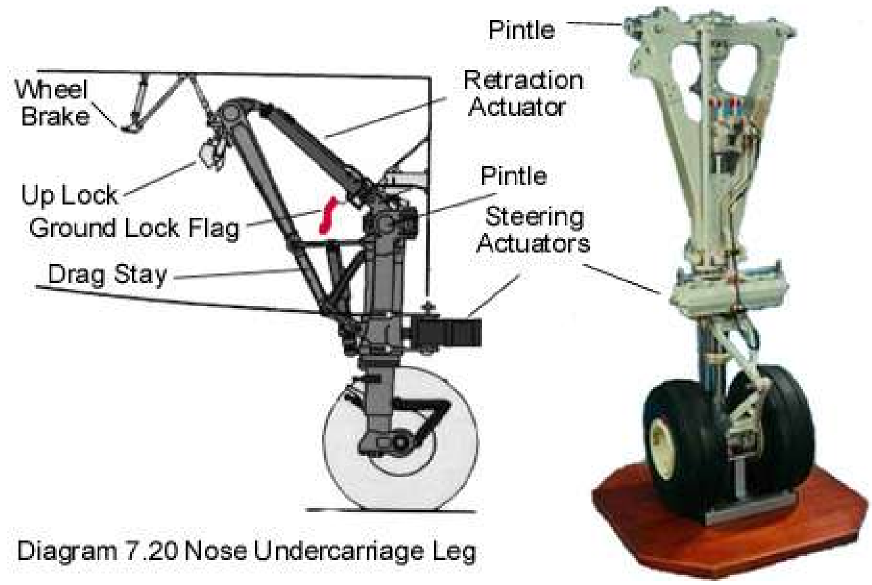


Diagram 7.20 Nose Undercarriage Leg

To overcome the greater forces acting on the nose wheel, these aircraft use hydraulic power systems. The pilots have two steering controls. For alignment of the aircraft to the runway during landing and take-off, the pilots' rudder pedals move the nose wheel through 7° of arc either side of the centreline.

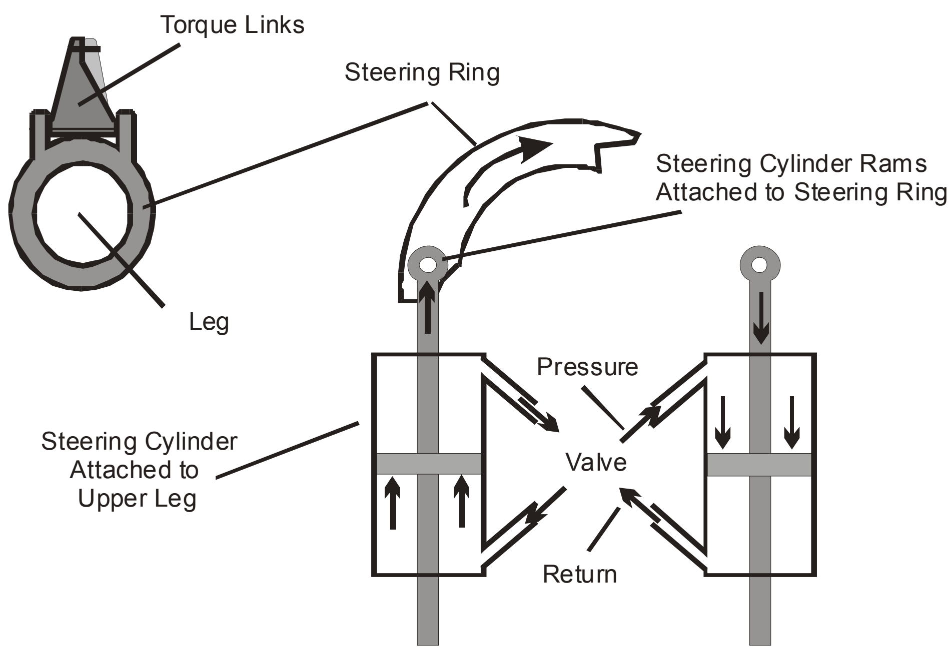


Diagram 7.21 Steering Cylinders

When the aircraft's speed falls below **60 kt,** the pilot's control wheel or tiller bar takes effect. On some older aircraft, this was a small hand wheel attached to the control column with a reference datum to show when the nose wheels were fore and aft. Modern aircraft mount the control wheel on the side panels. These are biased to neutral, and both captain and co-pilot have this facility. In diagram 7.20, the steering motor in the photograph is a rectangular box attached to the upper leg casting with the torque links below it. Depending on design, the lower leg can be rotated using two hydraulic linear actuators or a rotary actuator.

Where linear actuators are used, when the pilot makes an input to turn right, fluid from the aircraft's hydraulic power system is allowed to enter the first chamber of the right hand actuator and the second chamber of the left hand actuator, as diagram 7.21 illustrates. This turns the steering ring, and the lower leg is turned via the torque link.

When the wheels are in alignment with the centreline of the aircraft, hydraulic pressure can be brought into a central feed to all four chambers that are diagonally interlinked. As movement of the wheel in either direction results in expelling fluid from the first chamber of one actuator to the second chamber of the other actuator, the pressure resists this movement and damps the oscillations.

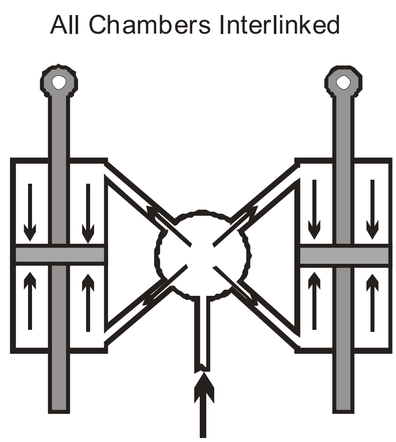


Diagram 7.22 Nose Wheel Centralised

As the pilot rotates the aircraft and the nose leg extends, an input is given to the steering to centralise the nose wheel in the fore and aft position. Refer to diagram 7.22 for an illustration. When undercarriage retraction is selected, an input hydraulically locks it in the fore and aft position.

NOSE WHEEL RUMBLE

As nose wheels are not braked, there is a possibility that the nose wheel will continue to rotate for a period of time after retraction, which creates a rumble and a gyroscopic mass. To prevent this rotation, a sprung steel strip that is fitted in the nose bay contacts the tyre and stops the wheel from turning. Diagram 7.20 shows this.

BODY GEAR STEERING

Diagram 7.23 Body Gear

For large air transport aircraft such as the Airbus A340, Boeing 747, Lockheed Tri Star, and McDonnell Douglas DC-10, the aircraft is supported on both wing gear and body gear. On these aircraft, the body gear is linked to the pilot's control wheel to allow the aircraft to turn. When the aircraft has slowed down below 60 kt, steering inputs to the nose wheel activate a proportional steering signal in the opposite direction for the body gear. This reduces the aircraft's turning radius. In diagram 7.23, the body gear can be seen between the wing gears.

TURNING CIRCLE

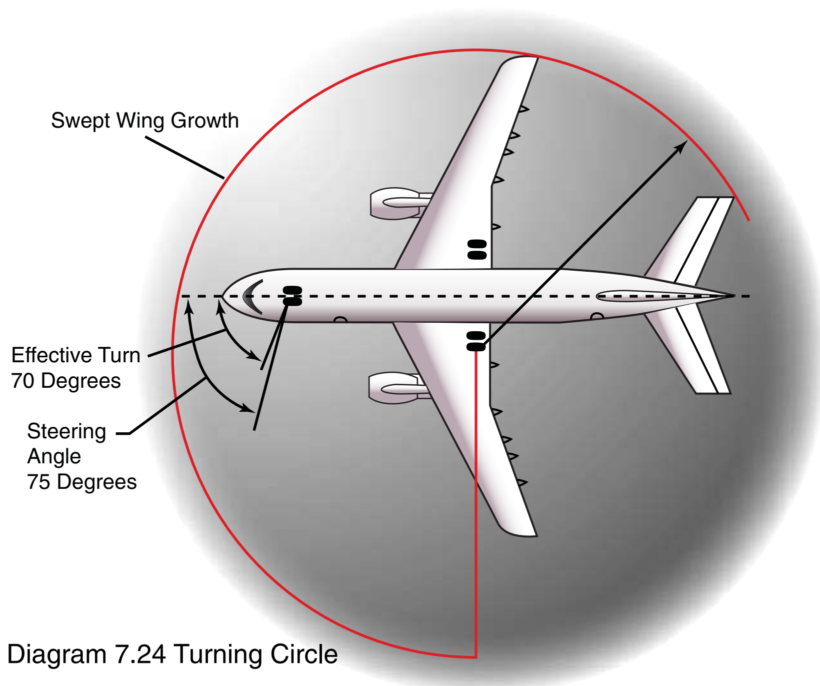


Diagram 7.24 Turning Circle

Diagram 7.24 shows a medium aircraft's turning circle. The nose wheel can be steered to 75° either side of the centreline, but the effective angle of steering is 70°. The aircraft manufacturer's handbook always quotes the aircraft's minimum turning radius. This should not be reduced, as it leads to excessive stresses being created in the inside leg. This smallest turning circle is achieved using nose wheel steering, inner wheel braking, and body gear steering where fitted.

Aircraft should always be turned using the largest turning radius practicable to reduce the stresses on the main undercarriage gear.

NOSE WHEEL STEERING DISCONNECT

To allow the nose wheel of aircraft fitted with powered steering to be turned when they are being towed or pushed, a steering disconnect mechanism is fitted. In many designs, it is not possible to attach the tow bar until the steering disconnect cover cap is removed and the steering disconnected.

As a safeguard, it is physically impossible to refit the cover cap without reconnecting the steering. The pilot is also informed by a caution caption if the steering is inoperative. Both of these measures ensure that the pilot has control of the steering prior to a take-off run.

MAIN GEAR BAYS — LOW-WING AIRCRAFT

On most modern medium/large air transport aircraft, the main undercarriage legs are attached to the wing main spar and rear or auxiliary spar. These legs retract inward into the wheel well. Originally, the wheel bay was part of the wing structure close to the wing root. However, as aircraft have become larger and heavier, the main gear has become larger. It is now common practice for the wheel bay to be part of the fuselage structure with the wheel bay doors forming part of the fairing around the fuselage/wing. Undercarriages that retract laterally have no effect on the aircraft’s longitudinal C of G.

HIGH-WING TURBOPROPS

Diagram 7.25 Landing Gear for High Wing Aircraft

In the case of high-wing turboprops, like the one shown in diagram 7.25, it is common practice for the undercarriage bay to be formed and faired as part of the engine nacelle. These aircraft undercarriages either retract forward or rearward, and extension and retraction cause the aircraft’s longitudinal C of G to move.

HIGH-WING TURBINE AIRCRAFT

For high-wing turbine powered aircraft, such as the BAe 146, the main gear is mounted and stowed in the fuselage. This requires the whole gear to articulate and fold up to take up the minimum space possible. These designs, by their nature, are complicated, and fuselage mounting reduces the wheel track.

WHEEL LAYOUTS

As aircraft developed and their weight increased, tyre pressures had to increase if the tyre size and undercarriage bay were not to become excessive (see tyres in the following chapter). As modern aircraft operate on high-pressure tyres (up to 315 psi), more wheels were required to be fitted to the undercarriage to support the weight of ever larger aircraft.

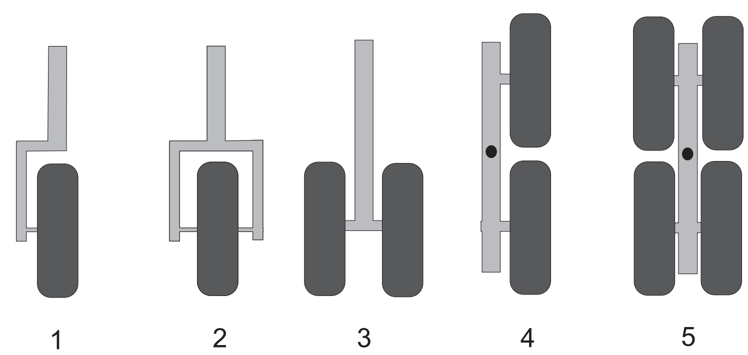


Diagram 7.26 Wheel Mounting Systems

Some of the wheel mounting systems are shown in diagram 7.26:

1	Half fork
2	Fork
3	Twin wheel, mounted on the same axle line
4	Tandem wheels
5	Bogies of four or six-wheel configuration for large air transport aircraft.

BOGIES

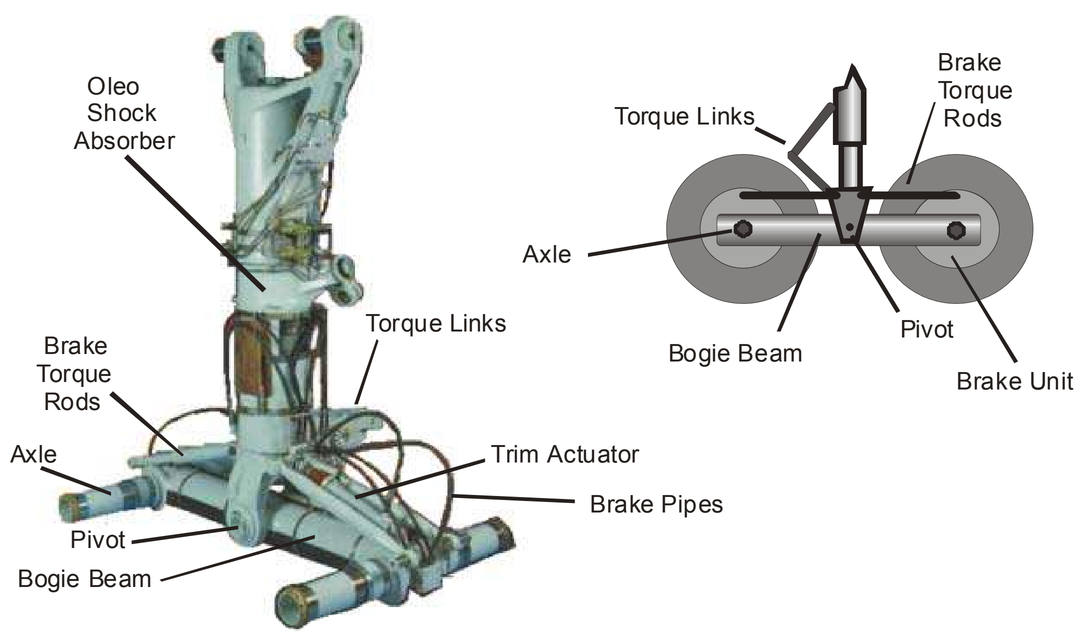


Diagram 7.27 Bogie Layout

The **bogie beam** attaches via a pivot to the bottom of the oleo leg. Refer to diagram 7.27. The lower leg is kept in fore and aft alignment by torque links. The function of the bogie beam is to locate and support the axles.

To lessen the friction created when the bogie wheels touch down, it is standard practice for one pair of bogie wheels to contact the runway first. This allows them to speed up and give the aircraft directional control. If all bogie wheels were to touch down simultaneously, it is possible that the friction created would pivot the nose gear onto the runway with such force as to cause structural damage.

It depends on the aircraft designer as to whether the front wheels (Airbus) or the rear wheels (Boeing) touch down first. One of the reasons for using the front wheels is that it reduces the lever arm between the touchdown point and the nose wheel.

BOGIE TRIM, BOGIE TRAIL

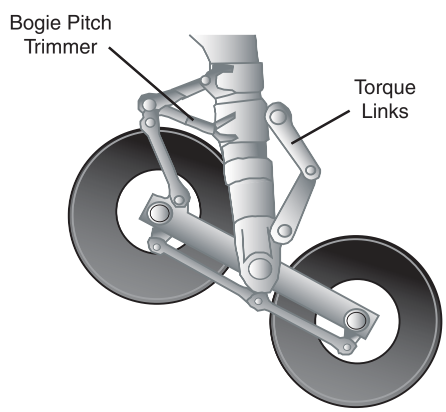


Diagram 7.28 Bogie Pitch Trimmer

To ensure that the bogie takes the correct position for landing (trail) and for retraction into the undercarriage bay, a hydraulic trim jack is used. Refer to diagram 7.28 for an illustration. This can also function as a hop damper.

HOP DAMPER

To prevent the bogie wheels from bouncing over rough surfaces and to ensure that the wheels remain in contact with the runway, a hop damper is pressurised and forms an actuator forcing the front wheels downward.

RETRACTABLE MAIN UNDERCARRIAGE FOR AIR TRANSPORT AIRCRAFT

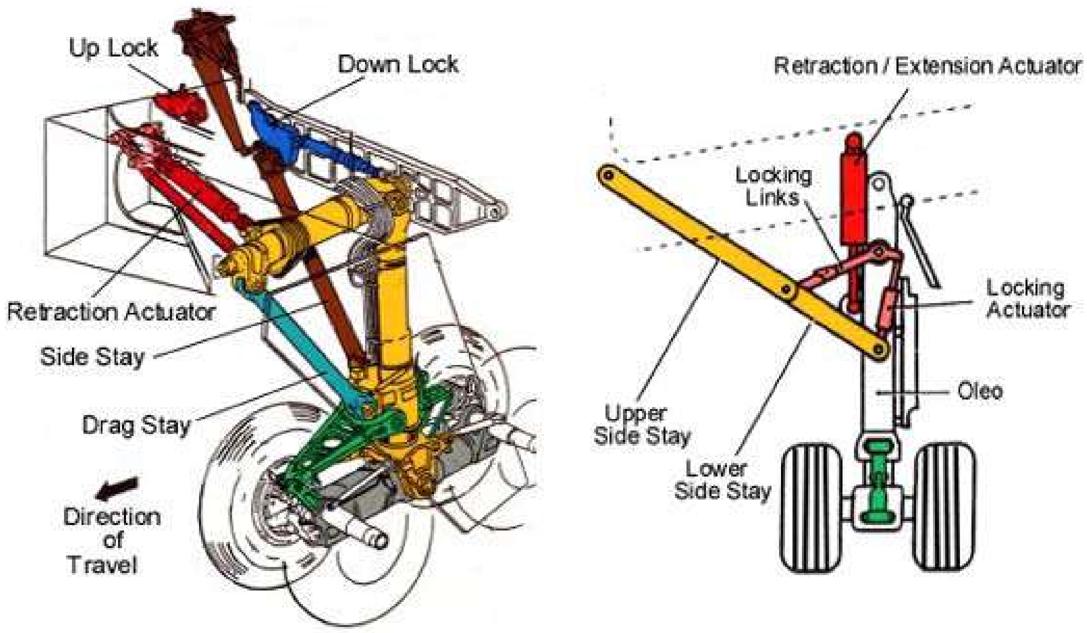


Diagram 7.29 Main Undercarriage Bracing

To support the lower leg against the forces of friction, a **fore stay** or **drag strut** is fitted, as in the left picture of diagram 7.29. To prevent the undercarriage from collapsing sideways, a **side stay** is fitted. As the gear has to retract sideways into the wheel bay, the side stay is articulated. In the left undercarriage of diagram 7.29, the side stay is held locked up and down by **hook locks**. This is not always the case, as in the right undercarriage of the diagram, where a **geometric locking** mechanism is used for the side stay.

A smaller cross-bracing geometric lock, actuated by a small hydraulic jack, can also lock the side stay. Using overcentre locks ensures that the gear remains down when hydraulic power has been removed. To retract the gear, hydraulic sequencing is required to operate the locking actuator first before the retraction actuator can retract the gear.

OVERCENTRE OR GEOMETRIC LOCKS

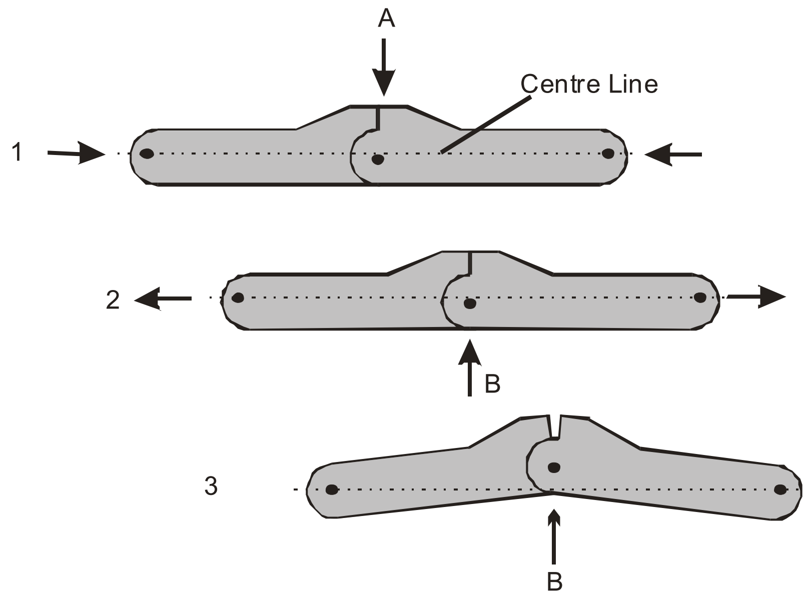


Diagram 7.30 Overcentre or Geometric Locks

To ensure that all the components of the side stay form a solid rod when extended, it is manufactured as an overcentre or geometric lock, as in diagram 7.30. The two parts of the lock are manufactured so that when they butt together at the hinge (as shown in 1), their hinge point is below a line drawn from the centre of each rod end. Any force applied from the direction of arrow A, locks the strut more firmly, as does compression force applied to each end.

To unlock the joint, a force in the direction of arrow B must overcome the compression force applied to the ends of the strut and be able to push the hinge up through the centreline (as shown in 2). The compression forces are a proportion of the mass of the aircraft. When the wheels have left the ground, the compression load is removed. A small hydraulic actuator can exert sufficient force in the direction of arrow B to push or pull the strut through the centreline, breaking the lock (as shown in 3).

Note: Overcentre locks are not to be used as up locks, only as down locks.

UP LOCKS

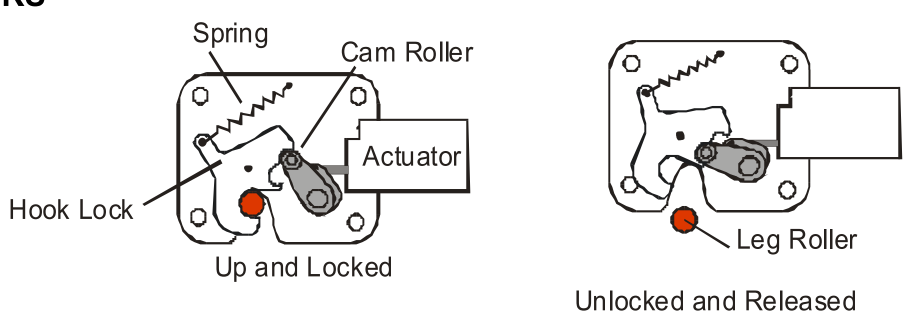


Diagram 7.31 Hook Locks

In an air transport aircraft, the undercarriage is held in the up position using mechanical hook locks. This ensures that the gear remains up in normal flight in the event of loss of hydraulic power, as the cam roller makes a physical lock preventing the hook from rotating open, as shown in the left of diagram 7.31.

The hook lock can take the form of a single hook (as shown in diagram 7.31) or a pincer. The hook lock has a pivot point and a profiled outline. A small hydraulic actuator operates a lever with a roller, which follows the cam profile on the hook. When the pilot selects undercarriage down, hydraulic pressure is sequenced into the up lock actuator, forcing the ram outward. The roller is forced to ride over the cam profile of the hook and moves into the lower profile cut out, where it pushes against the hook and with the action of the spring rolls the hook open.

On the “up” selection, the undercarriage roller strikes the hook and forces it to rotate back to the closed position, as the force produced by the retraction actuator is greater than the force produced by the up lock actuator.

EMERGENCY LOWERING

There is a JAA requirement that in the event of hydraulic failure, there must be a separate means of lowering the undercarriage. In some aircraft, this takes the form of compressed nitrogen or air. This is a one-shot system. If the pilot is unable to achieve satisfactory gear down indications, the undercarriage selector is left in the down position, and a guarded emergency selector is operated. Operating this selector allows compressed gas into the aircraft’s undercarriage down line. To ease the operation, fluid from the undercarriage up lines is jettisoned overboard. This ensures that any back pressure created by the fluid is minimal, and the gear can snap to the locked down, safe condition.

To comply with the JAA requirement, g force release force is an acceptable method. The pilot puts the aircraft through a series of manoeuvres, first creating negative g, then positive g (manoeuvres laid down in the aircraft flight manual). The positive g effectively increases the weight of the gear acting downward on the up lock hooks. As the pivot point of the hook is offset from the centre point of the undercarriage roller, a large turning moment is made. This forces the hook round and at the same time, due to the gear lever being in the down selected position, the cam roller is forced to turn. As the hook starts to rotate, the spring ensures the hook snaps clear, allowing the gear to drop. For the main gears, the mass times acceleration by gravity snaps the gear into the locked condition. Airflow aids extension of nose gears that retract forward.

MAIN UNDERCARRIAGE DOORS

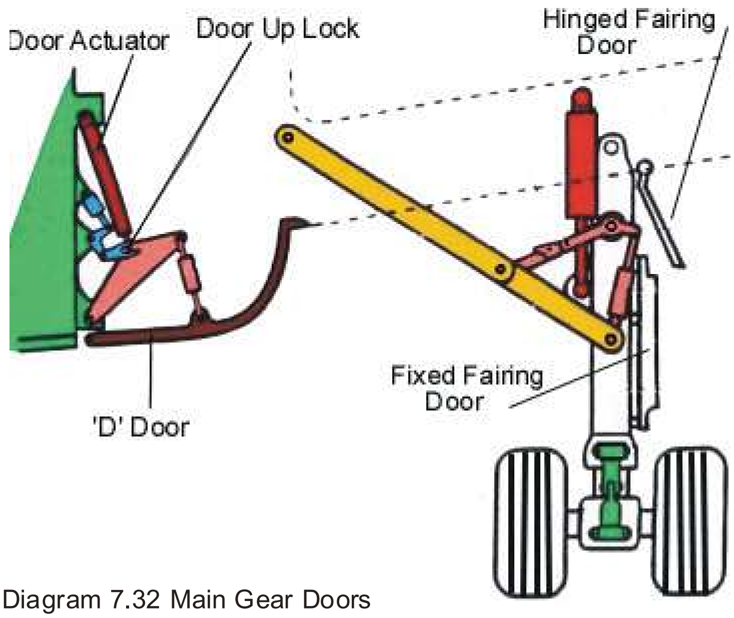


Diagram 7.32 Main Gear Doors

Wheel bay doors, frequently referred to as **D** doors, have their own up locks. These are sequenced to open before the main gear up lock opens and move clear as the wheels start to extend. Attached to the main legs are fixed fairing doors. When the gear has moved through an arc, the D doors are sequenced to close and lock. This reduces the aircraft's profile drag and prevents foreign objects (e.g. dirt, water, ice, snow, etc.) from being flung into the bay. In the event of emergency landing gear extension, these doors remain open.

LARGE AIRCRAFT UNDERCARRIAGE INDICATORS

Large aircraft with undercarriage doors and bogies have:

- 3 green lights to indicate that the gear is down and locked in a **safe** position
- A red **gear unsafe** light to indicate that part of the gear is unlocked
- A red **doors** light that indicates that the doors are in an unlocked position
- An amber **truck** light that indicates that the bogie is in an incorrect position for retraction

When all the gears are up and the doors are correctly locked, all the lights are out. On down selection, the gear unsafe light illuminates, and the red doors light illuminates. When the doors close, the red doors light extinguishes. When the gears lock down, the red gear unsafe light extinguishes and the 3 green lights illuminate.

If a door fails to lock, the red doors and gear lights remain illuminated. If the amber truck light illuminates after take-off, the undercarriage selector lever is physically prevented from moving to the up selection, as the truck would be in the wrong position. This could cause damage if the pilot attempted to retract the gear.

MECHANICAL INDICATION

Some aircraft have mechanical position indicators. These are only used by the flight crew to check on the condition of the landing gear in the event of an emergency and are normally located behind trim panels.

LARGE AIRCRAFT GEAR SELECTOR

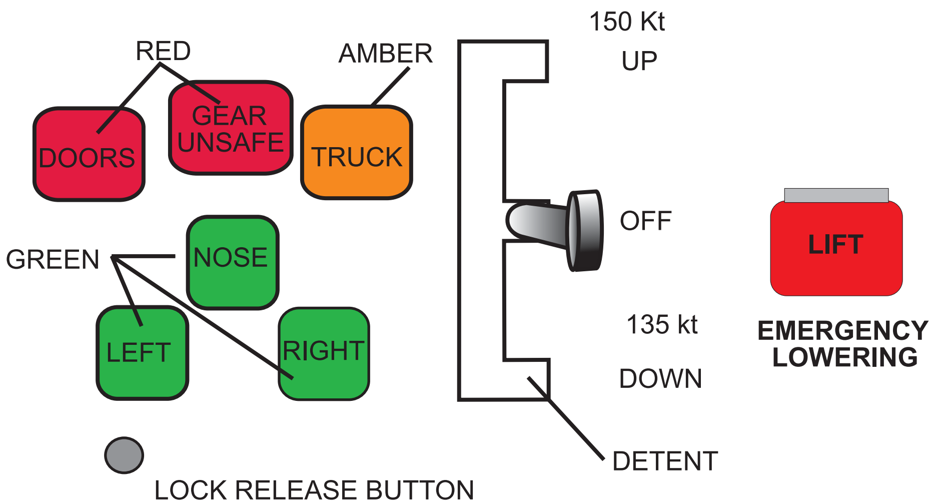


Diagram 7.33 Large Aircraft Undercarriage Selector and Indications

The gear selector must terminate in the shape of a wheel and the selector must be detented so as to require a determined movement. In some designs, the detent takes the form of horizontal slots (as shown in diagram 7.33), and the lever is biased to move into the slot and remain there. On larger aircraft, there is a third position labelled **off**. During the cruise, the selector is placed in this position. Gear selectors must be located forward of the throttles or thrust levers.

AIR/GROUND LOGIC

Air/Ground Logic is the term given to a series of switches that make or break depending on whether the aircraft undercarriage is compressed while on the ground or extended by being in flight. On light aircraft, this can take the form of micro switches (measuring small movements) operated by the torque links of the main gear.

For systems that use a power pack to retract the gear, the micro switch completes the circuit. It removes the electrical power supply to the power pack when the weight is on the gear. Thus, any movement of the selector lever will not cause the gear to collapse.

For large aircraft, Air/Ground Logic switches can take the following form:

- Micro switches
- Proximity switches
- Pressure switches

MICRO SWITCHES

As with the light aircraft, these switches register the movement of a plunger. As the plunger and its striker can wear, more modern aircraft are fitted with proximity switches so as not to have moving parts.

PROXIMITY SWITCHES

Proximity switches consist of 2 small plates, each mounted on a separate surface. One plate is made of magnetised stainless steel and the other plate incorporates a small field coil. When the plates are brought into close proximity to each other, a small EMF is induced in the coil. This is sensed by a detector unit, which supplies the signal to the Logic system.

PRESSURE SWITCHES

Some aircraft mount pressure sensitive switches inside the oleos. These sense the increasing pressure on touchdown and operate the Ground Logic.

Ground Logic on a large aircraft controls the operation of:

- Thrust reversers in conjunction with power setting
- Braking (bounce protection)
- Nose wheel steering in conjunction with groundspeed
- Automatic spoiler deployment when spoilers are set to **armed**
- Gear selector locking
- Thermal wing anti ice selection to **off**

UNDERCARRIAGE DOWN SAFETY LOCK

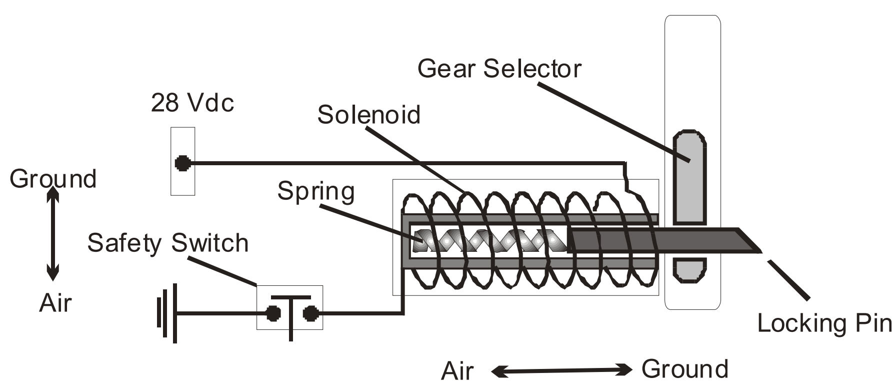


Diagram 7.34 Gear Down Safety Lock

To prevent inadvertent selection of an air transport aircraft undercarriage selector to retract, when the Logic system senses ground, a safety lock, in the form of a mechanical plunger operated by an electrical solenoid, locks the gear selector in the down detented position. Under normal operating circumstances, this plunger retracts only when the gear extends, and the system senses Air Logic. A safety override switch allows the pilot to withdraw the plunger and select "gear up" whilst on the ground (for emergency use only).

GROUND LOCKS

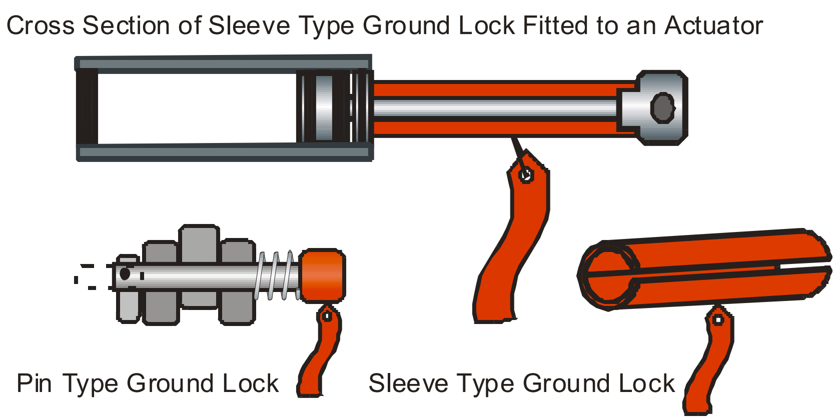


Diagram 7.35 Ground Locks

When aircraft are being towed or serviced without hydraulic power available, ground locks are fitted. These take the form of tubular clamps or small pins. To indicate that they are fitted, each locking device has a red or red and white diagonally striped flag (pennant) attached to it. When fitted, these locks physically prevent the movement of the gear and doors. Each aircraft carries its own set of locks. For an air transport aircraft, these must be mounted in an open-fronted or clear-fronted stowage cupboard, which the captain can check on the way to the flight deck.

Diagram 7.35 shows sleeves and pins, the two main types of ground locks used. Sleeves are tubes that are hinged along their length and clamp around the ram of an actuator when it is fully extended. The sleeve forms a physical lock to the ram's retraction. Pins, often used on geometric locks, fit through holes that align when the struts are in the overcentred condition and have a locking tag to prevent them from falling out.

Beware of ground locks that have lost their warning flags.

RETRACTABLE UNDERCARRIAGE FOR LIGHT AIRCRAFT

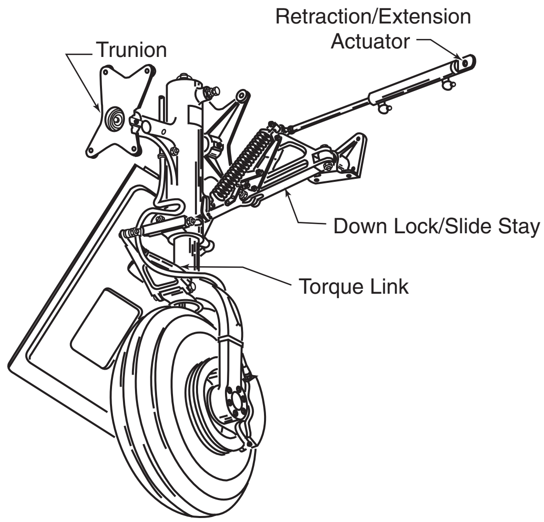


Diagram 7.36 Retractable Main Leg of a Light Aircraft

Diagram 7.36 shows a light aircraft's retractable main undercarriage leg showing the down lock/side stay which forms an overcentred lock, the springs that ensure that the side stay snaps locked, and the extension/retraction actuator. As these aircraft are light, the oleo leg does not require the additional bracing of drag stays. The hydraulic power source for the gear's extension and retraction is described below.

Many light aircraft retractable undercarriages use self-contained hydraulic **power packs**. Refer to the system schematic diagram 7.37. In this system, a spur gear pump, operated by a reversible electric motor, provides hydraulic power to raise and lower the aircraft's undercarriage. For protection in the event of a power failure, a freefall system is fitted. In this system a hydraulic lock holds the gear in the up position, and a mechanical lock holds it down.

A warning horn sounds to prevent inadvertent wheels up landings if the gear is not selected down when the flaps are set to landing, or the throttles retarded to flight idle. Selecting the gear down, raising the flaps, or advancing the throttles removes the warning.

OPERATION OF THE POWER PACK

A down selection causes the pump to rotate in an anti-clockwise direction. This draws unfiltered fluid from the reservoir and supplies it under pressure to the left-hand side of the unit. The fluid pressure closes off the return line, via the filter to the reservoir, and forces the gear up check valve piston to the right. This unseats the check valve, which breaks the hydraulic lock that held the gear up, and allows the fluid in the up line to enter the pump and be transferred to the down line side of the unit. The build up of pressure forces the shuttle valve to the left, blocking the return to the reservoir and supplying fluid to the down line.

As the gear is falling with gravity, and there are no mechanical locks to open, the down line functions at a low pressure between 400-800 psi. To control this pressure, a relief valve, termed a low pressure control valve, allows excess pressure to return to the reservoir.

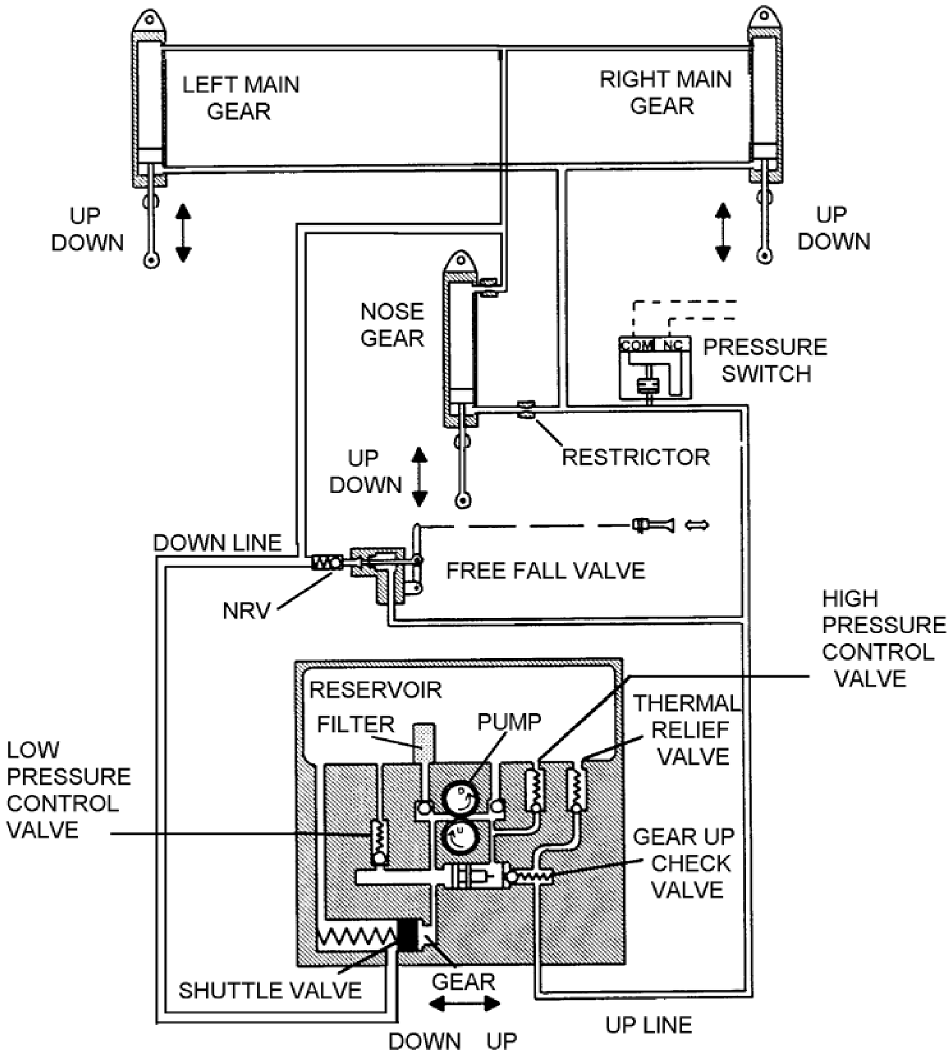


Diagram 7.37 Power Pack for a Light Aircraft

Restrictors in both the down line and the up line of the nose gear control the fluid supply to the three gears' actuators. These restrictions to flow allow the main gears to move first before the nose gear starts to move. This ensures that the drag created by lowering the gear is controlled when the main gears have locked down. Likewise, the nose gear lowered into wind is stopped from moving rearward too quickly by the restrictors. When all three gears are down and locked, the red gear unsafe light extinguishes, and the electric power to the pump is removed. As the pressure in the down line decays, the shuttle valve moves to the right by spring pressure.

On up selection, the pump rotates in a clockwise direction drawing fluid from the reservoir via the filter, the pressure closes off the return to the reservoir, and forces the gear up check valve piston to the left. Fluid pressure unseats the check valve, and fluid enters the up line. As the gears have to be raised against gravity, the piston areas are reduced by the rams, and the nose gear has to retract against the air flow, the operating pressure of the high pressure control valve is set at a nominal 1800 psi.

The initial action of the retraction actuators is to unlock the down lock/side stays mechanical hooks, and then pull the down locks through the overcentred condition, before starting to retract the gear. The main gear moves first due to the restrictors. When the gear is up and cannot travel any further, the pressure in the up line increases to a value of 1800 +/- 100 psi. At this point, the pressure switch in the up line removes the electrical power from the pump and cancels the red light. When pumping stops, the gear up check valve reseats, forming the hydraulic lock holding the gear up. If the pressure drops by a preset value, the pressure switch re-energises the pump motor, increasing the system pressure. Fluid from the down line is returned to the reservoir.

In the event that the pump fails to operate, the pilot selects down on the normal selector, isolates the pump electrically by pulling the circuit breakers, and operates the guarded emergency lowering selector. This opens the free fall valve breaking the hydraulic lock, which allows gravity and air flow to lower the gears. The down lock springs ensure that the gears snap locked.

LIGHT AIRCRAFT UNDERCARRIAGE SELECTOR AND INDICATOR

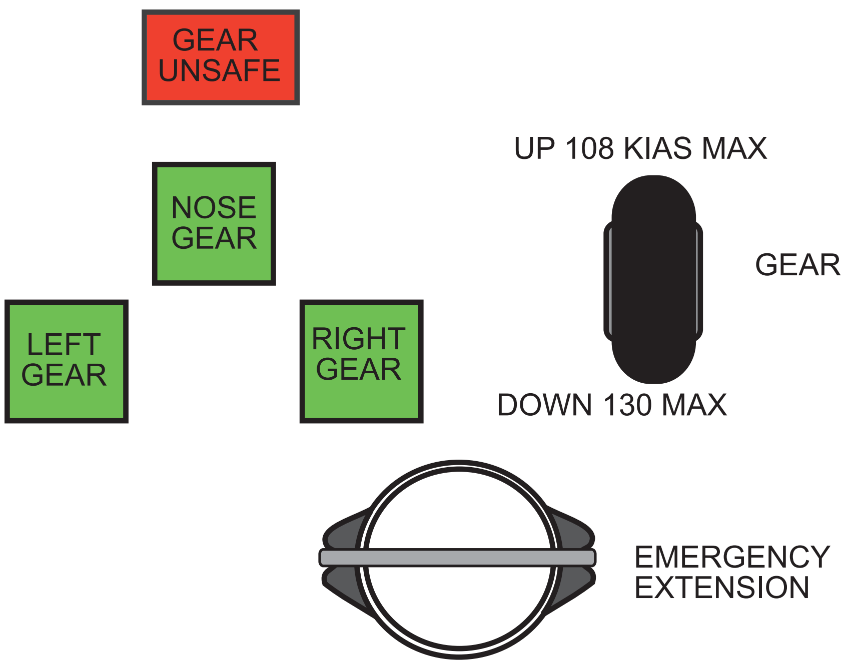


Diagram 7.38 Light Aircraft Gear Selector and Indications

JAR regulations require that the undercarriage selector must be forward of the throttle(s), where they are mounted on a quadrant. The gear selector is made in the shape of a wheel to differentiate it from any of the other controls. To prevent inadvertent operation, the selector has to have detents to lock it in the up or down position. This requires the pilot to pull or push the selector to unlock it before making a selection.

It is conventional for light aircraft to have three green lights in the form of a triangle, which only illuminate when the gear is down and locked. A red light (this can be part of the master caution panel) with the legend **gear unsafe** illuminates when the gear is unlocked and in transit.

When the gear is locked up, both the red light and the green light are extinguished. The green lights frequently take the form of block bulbs. In the event that an indicator fails, the pilot is able to interchange the blocks and see if it is a bulb failure or a condition.

If the pilot has to lower the gear through the emergency extension system, the indication is 3 greens when the gear is down and locked, and the red gear unsafe light illuminates to warn that the gear has been lowered by emergency means.

If the aircraft is parked with the throttles closed and electrical power is applied, should for any reason the gear selector be moved to gear up, the gear unsafe light illuminates, and the warning horn sounds. The ground logic system (micro switches) has isolated the power pack, preventing an inadvertent retraction.

SKIS AND BEAR PAWS

For aircraft that are operating from snow and ice runways, the aircraft's wheels are replaced with skis. The skis are attached to the undercarriage legs for shock load absorption. They have to be able to pivot about their attachment point to enable them to follow the contours of the ground over which they are running and to allow for the rotation of the aircraft. To prevent the skis from rotating so that the tips are down in flight, a steel cable is attached from the tips back to the aircraft's structure. To prevent the ski tips from rotating upward in flight, a bungee cord is attached between the aircraft's structure and the ski tails.

For aircraft that are operating between snow/ice runways and paved runways, the wheels remain attached. The skis are mounted on hydraulic actuators to allow them to be retracted for use on paved surfaces and extended for snow/ice surfaces. Again, the skis have to pivot and have attachment cables and bungees.

When operating from snow, the aircraft's weight is supported by the surface area of the skis. To achieve the same support, shorter but wider skis can be used. These are termed bear paws. Again they can be fixed or retractable for use on both paved and snow/ice surfaces.

FLOATS

Diagram 7.39 Amphibious Float Plane

To convert a land plane to operate exclusively from water, floats are fitted in place of the aircraft's standard undercarriage. Each main float must have:

- A buoyancy of 80% in excess of the maximum weight which that float is expected to carry in supporting the maximum weight of the seaplane or amphibian in fresh water

- At least four watertight compartments approximately equal in volume

In diagram 7.39, the shape of the floats with their distinctive cut out step, which allows the floats to plane during the take-off run, can be clearly seen as can the water rudders for water steerage. This aircraft, a Cessna Caravan, is an amphibian. The landing gear is retracted into the floats.

Chapter 8
Landing Gear — Wheels, Tyres, and Brakes

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For Reference Information

Dunlop Aircraft Tyres

Goodyear Tire Co.

NASA Tech Paper 3586, Tires

INTRODUCTION TO BRAKES
Heavier aircraft flying at higher airspeeds led to increased landing speeds, requiring a wheel braking system to be fitted. The original wheel brake system applied a braking force to both wheels equally at the same time. By the 1930s, differential-braking systems had become standard equipment for larger aircraft.

All aircraft wheel brake systems are a means of converting the aircraft's forward motion (kinetic energy) into heat energy via friction. A braking system's efficiency is determined by how rapidly kinetic energy can be converted into heat and the heat dissipated to other structures and the atmosphere.

TYPES OF WHEEL BRAKES

- Mechanical drum brakes — very old aircraft and cars only
- Hydraulic expander brakes — older aircraft only
- Pneumatic bag brakes — older aircraft
- Single disc — light GA aircraft
- Multi-disc — air transport aircraft

EXPANDER TYPE BRAKES

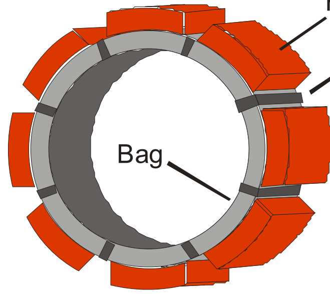


Diagram 8.1 Expander Type Brake

These are also referred to as bag brakes and consist of a semi-flexible torroidal tube ring that is located inside a drum attached to the axle. Attached to the outer surface of the ring are segments of friction material and return springs (see diagram 8.1). A pressurised supply of air in the pneumatic system or fluid in the hydraulic system expands the ring against the brake drum. Upon releasing the brake pedal, the return springs, which are under tension, act to collapse the bag, pulling the friction pads away from the drum. The pilot depresses toe pedals mounted in front of the rudder pedals to apply and control the wheel braking action (refer to diagram 8.2). Depressing the toe pedals operates a brake master cylinder, which is either attached to the toe brakes or mechanically operated by their action.

In the basic light aircraft version, the master cylinder incorporates a reservoir. In more sophisticated systems, the master cylinders are attached to a remote reservoir. This reduces the chance of oil contaminating the rudder pedals, etc, and the reservoir vent hole from becoming blocked.

BASIC BRAKE MASTER CYLINDER

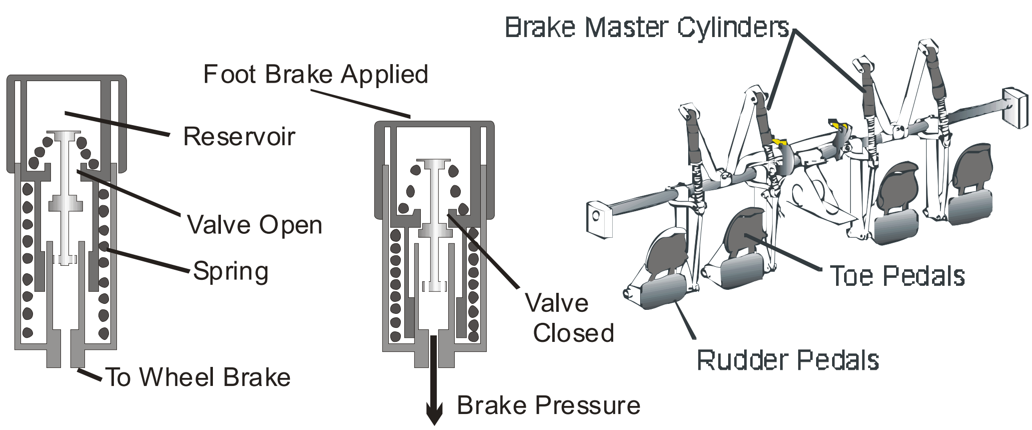


Diagram 8.2 Basic Brake Master Cylinders

Referring to diagram 8.2, when the brakes are not applied, the main spring extends the master cylinder. The valve is prevented from being closed by the upper spring by stops in the lower cylinder. This keeps the valve open and allows the reservoir to replenish the cylinder.

As the toe pedal is depressed, the cylinder shortens, compressing the main spring. The upper spring acts to close the valve, and the fluid displaces into the wheel brake system where it acts on the slave cylinder.

The fluid displaced into the brake pipes raises the pressure, which acts against the larger surface area of the piston, creating the force that is exerted against the disc. As the brake material and the disc progressively wear down, the amount of fluid returned to the reservoir reduces so the reservoir replenishes the cylinder when the brake is at rest.

SIMPLE HYDRAULIC BRAKING SYSTEM

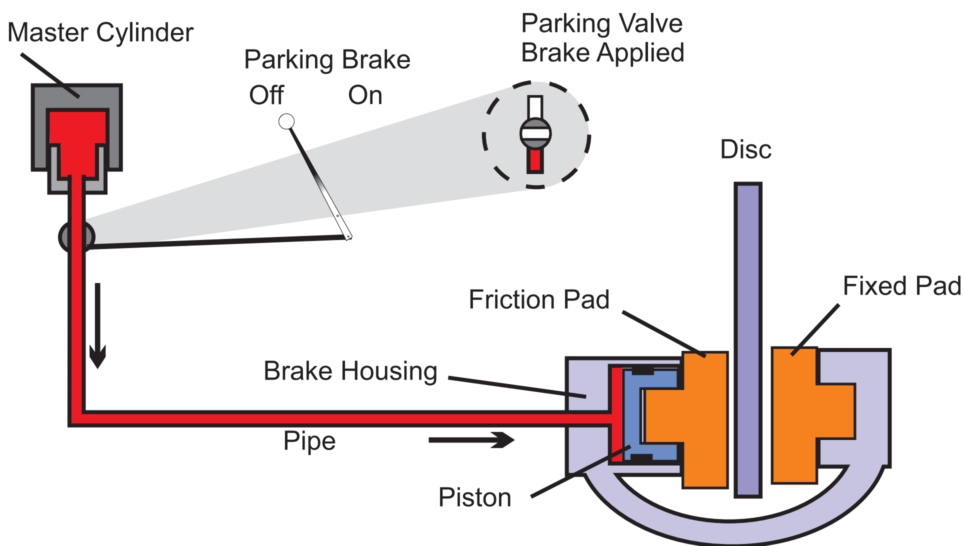


Diagram 8.3 Basic Brake Circuit

To achieve differential braking, each wheel has its own separate hydraulic circuit. Diagram 8.3 shows one wheel's circuit. Refer to the following text for explanations of the function and operation of both fixed and sliding discs.

LIGHT AIRCRAFT FIXED DISC

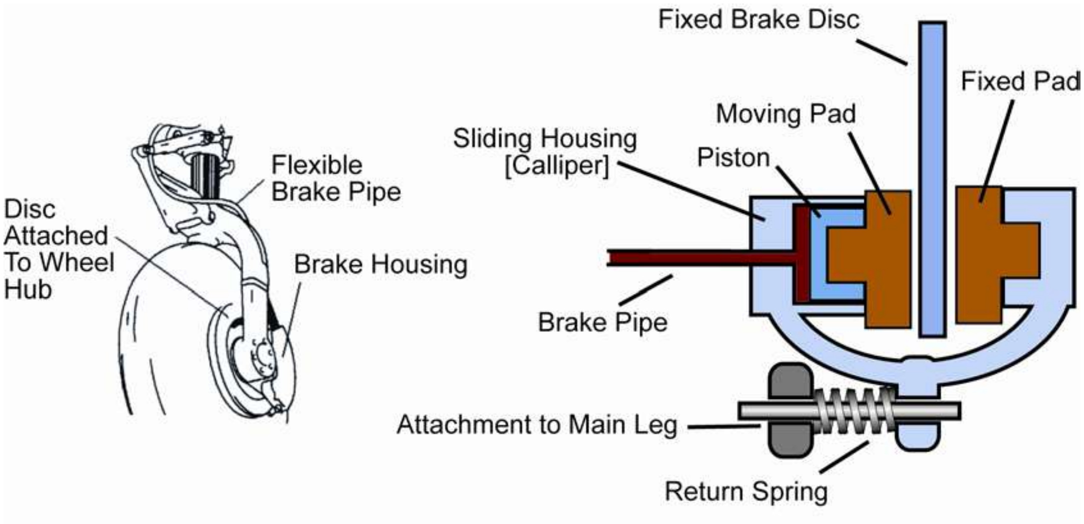


Diagram 8.4 Fixed Disc System for Light Aircraft

For light aircraft where the disc attaches to the wheel, a sliding calliper (similar to a car's) is used, as shown in diagram 8.4. Pressure exerts force equally in all directions, in this case, against the piston and the brake housing.

The displaced fluid moves the piston, pushing the puck against the disc. The brake housing slides to pull the fixed pad against the opposite side of the disc, thus creating friction and generating heat. Springs fitted between the calliper and its mounting points push the brake housing away as the brake pedal is released, returning the fluid to the reservoir.

To apply the parking brake, both toe brakes are depressed, and the parking brake lever is pulled and locked (refer to diagram 8.3). This creates a hydraulic lock in both systems simultaneously and traps pressure at the wheel brakes. Note that all wheel brake systems lose fluid/pressure back to their reservoirs over time and, therefore, must be reapplied regularly.

FIXED BRAKE HOUSING SLIDING DISC

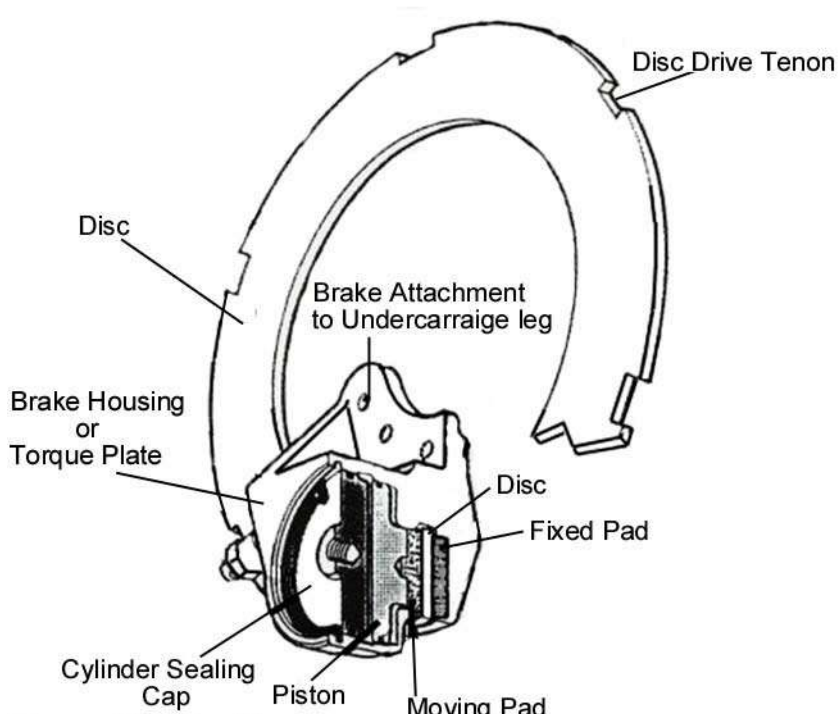


Diagram 8.5 Sliding Disc

For faster light aircraft, a single disc with castellations on its outer edge (refer to diagram 8.5) can be fitted inside the wheel. As the wheel rotates, the disc rotates. In this system, the caliper is fixed, and the action of the piston as it moves is to force the disc across onto the fixed pad.

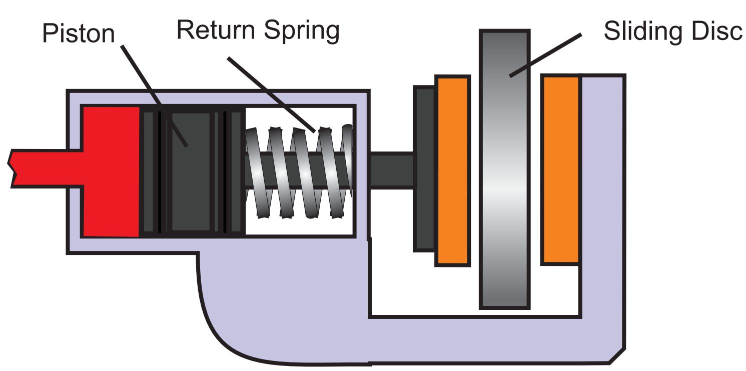


Diagram 8.6 Brake Release For A Sliding Disc Assembly

When the brake pedal releases, a spring fitted inside the piston housing acts to move the piston away from the disc. In diagram 8.6, the return spring appears as a compression spring. When the brake pedal releases and the piston returned, the disc rotation clears the fixed pad.

HEAVY DUTY BRAKE DISCS

For light single and twin piston-powered aircraft certified under JAR 23, it is possible to reduce landing distances by fitting heavy-duty brake discs. These are thicker than the standard disc and can absorb more heat, allowing the pilot to brake more heavily and stop in a shorter distance. Aircraft performance tables, as per those shown for the MEP1 in CAP 698, give the reduction in landing distance required.

BOOSTED BRAKES

Boosted brake systems are used for JAR 23 certified aircraft that have high landing speeds to increase the force applied at the brake pads against the discs. In the boosted brake system, when the pilot applies the toe brakes, aircraft hydraulic pressure proportionate to the toe brake deflection is fed into the master brake cylinders. This is similar to the servo assist braking on a car and acts to increase the force applied to the master cylinder's piston. The master cylinder piston moves the fluid into the brake lines. It does not allow aircraft hydraulic system pressure or fluid into the wheel brake units.

POWER BRAKES

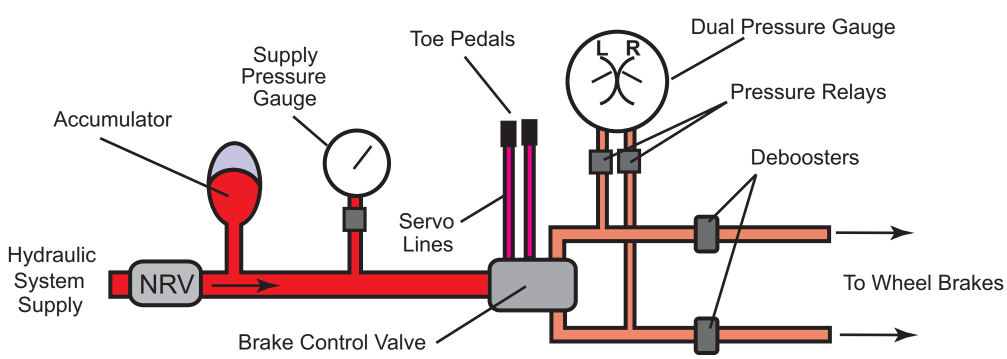


Diagram 8.7 Basic Power Brake System Components

For JAR 25 certified aircraft that are heavier, metered aircraft hydraulic system pressure serves to operate the brakes when landing to increase the braking force.

Diagram 8.7 shows the basic components used on an older air transport aircraft where the landing masses are greater than the 5700 kg, but the landing speeds are low. System pressure passes through a non-return valve to the brake control valve and charges the wheel brake accumulator. A direct reading gauge indicates the supply pressure available.

The brake control valve operates via servo pressure from the pilot's toe pedals. It directs hydraulic pressure to the appropriate wheel brakes proportionate to depression of the toe pedal. The servo lines are filled with hydraulic fluid, but this fluid does not mix with the system fluid, so that in the event of a servo line leaking in the flight deck, the pilot is not sprayed with high-pressure fluid.

A sensing line from each wheel brake allows the pilot to see the application of brake pressure to each wheel brake. Prior to take-off, the pilot fully depresses both toe pedals to ensure that equal pressure is supplied to the brake units. If the pressures are not even, the take-off must be cancelled and the system corrected.

As with the supply pressure gauge, where direct reading gauges are fitted, pressure relays are used. These ensure that any failure of the gauge does not result in high-pressure fluid spraying around the flight deck.

For power brake systems, the pilot receives artificial feel via the toe pedals to indicate the amount of braking applied. This can be achieved through backpressure in the servo lines acting on the master cylinder or compression springs.

In some older low-pressure braking systems, the aircraft hydraulic system pressure must be decreased. This is termed **deboosting**. A **debooster valve**, normally located on the main legs, reduces the pressure and increases the flow rate to the brakes

BRAKING REDUNDANCY

JAA regulations require that no single failure shall result in the loss of the braking system, and that the aircraft must come to a full stop in no more than 1.5 x the normal braking distance. To achieve this, air transport aircraft have a normal hydraulic supply and a separate hydraulic supply, or an emergency compressed gas supply.

This can take the form of:

- An accumulator that must have the capacity for six full brake applications
- An independent hydraulic power source
- A pressurised pneumatic supply

As the pilot's foot pressure is the controller of the brake pressure, it is possible to apply a pressure great enough to lock the brakes and prevent the wheel from turning, causing the wheel to skid. This damages the tyre to the point where it can burst (see tyres), reduce the braking efficiency, and lose directional stability. To overcome this, anti-skid braking systems are used on modern aircraft.

ANTI-SKID SYSTEMS

There are three broad categories of anti-skid:

- On/Off
  These are mechanical systems.

- Semi Modulating
  These are first generation electronic systems.

- Fully Modulating
  These are the modern electronic systems fitted to air transport aircraft.

JAR regulations require that:

- Aircraft fitted with an anti-skid system cannot take off unless it is serviceable.
- In the event of a power supply failure to an electronic anti-skid system, the pilot must be informed. Normally, a "fail" light illuminates.
- In the event of a loss of normal pressure supply to the brakes when an anti-skid device would be operating, provisions must occur for sufficient operation of the brakes to bring the aeroplane to rest when landing under runway surface conditions for which the aeroplane is certificated.
- If anti-skid devices are installed, the devices and associated systems must be designed so that no single probable malfunction, or failure, results in a hazardous loss of braking ability or directional control of the aeroplane.

MECHANICAL ON/OFF ANTI-SKID SYSTEM

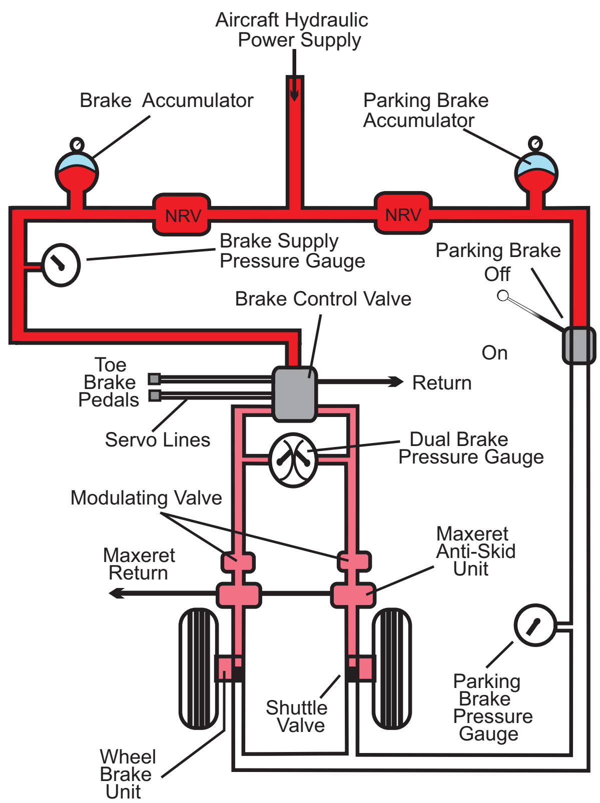


Diagram 8.8 Mechanical Anti-Skid System

On/Off systems are the simplest of the three types of anti-skid systems. In diagram 8.8, a mechanical unit called a **maxaret** provides the on/off anti-skid protection.

Fluid at supply pressure, nominally 3000 psi, is supplied to a wheel brake accumulator. This accumulator acts as an emergency reserve in the event of loss of hydraulic power, allowing the pilot six full applications of the brakes before it is exhausted.

The brake control valve directs the system pressure, proportionate to the toe pedal deflection, to the wheel brake units via modulating valves (see modulating valves below) and maxaret units. The control valve regulates the brake pressure to a max value of approximately 1750 psi.

For these systems, fully metered brake pressure as commanded by the pilot is applied until wheel locking is sensed. While the brakes are still being applied, the maxaret unit allows the wheel to spin back up by releasing pressure. When the system senses that the wheel is accelerating back to synchronous speed (i.e. groundspeed), full-metered pressure is again applied. The cycle of full pressure application/complete pressure release is repeated throughout the stop, or until the wheel ceases to skid with brake pressure applied.

In this system, the parking brake is applied when the wheel brakes have been released. The excess fluid in the brake housing is expelled by the return springs via the brake control valve. Operating the parking brake lever allows system fluid pressure to move the shuttle valves and apply the brake. In an emergency situation, if the power supply fails and the brake accumulator is exhausted, non-differential, non anti-skid braking is available using the parking brake lever.

MODULATING VALVES

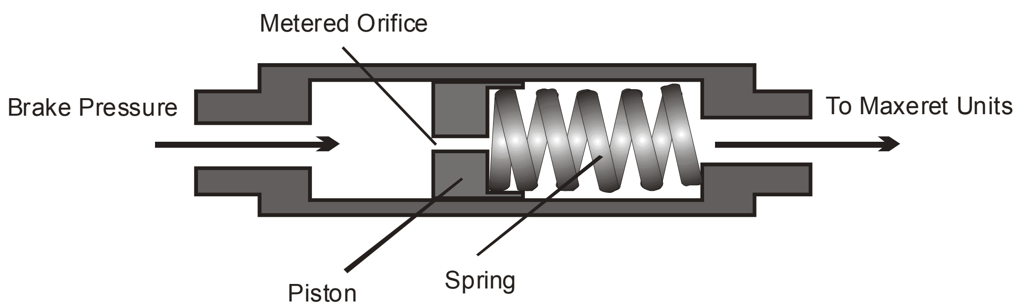


Diagram 8.9 Modulating Valve

Refer to diagram 8.9. Modulating valves are fitted between the brake control valve and the anti-skid units. As the brake pipeline and unit is full of fluid but unpressurised, pressure (flow) directs into the modulator when the pilot applies the brakes. Here it acts against the piston pushing it, compressing the spring, and filling all the brake slave cylinders. Once the cylinders have filled (taking up the clearance), the modulator piston becomes static. Any increase in brake pressure goes through the fluid column via the metering orifice.

If the anti-skid unit detects a skid condition, it releases a proportionate amount of pressure to reduce the brake friction force, allowing the wheel to speed up. This is taken down stream of the modulator. The piston prevents an excessive bleed-off of brake supply pressure (fluid) as the metered orifice limits the through flow. When the wheel speeds up, the anti-skid bleed-off stops and the through flow re-establishes brake pressure. On releasing the brakes, the modulator spring expands, pushing the piston to help return the brake to the off position.

As the electronic system reacts more quickly, the wheels are allowed to slow down to a point closer to a skid. This increases the braking action, reducing the landing roll at the expense of greater wear on the aircraft’s tyres, brake discs, and pads. Other advantages of anti-skid systems are that they prevent hydroplaning (aquaplaning) of wheels and protect the wheels from locking up if a main gear recoils off the runway. This is termed bounce protection.

ELECTRONIC ANTI-SKID SYSTEM

Mounted in each wheel axle and driven by the wheel is a small electrical tacho-generator. As the wheel turns faster, the signal produced by the tacho-generator increases and vice versa. This signal feeds to an anti-skid controller unit. When the anti-skid controller detects a skid condition, pressure from the affected brake unit is bled off to return. See semi-modulating and fully modulating anti-skid systems for their function and operation.

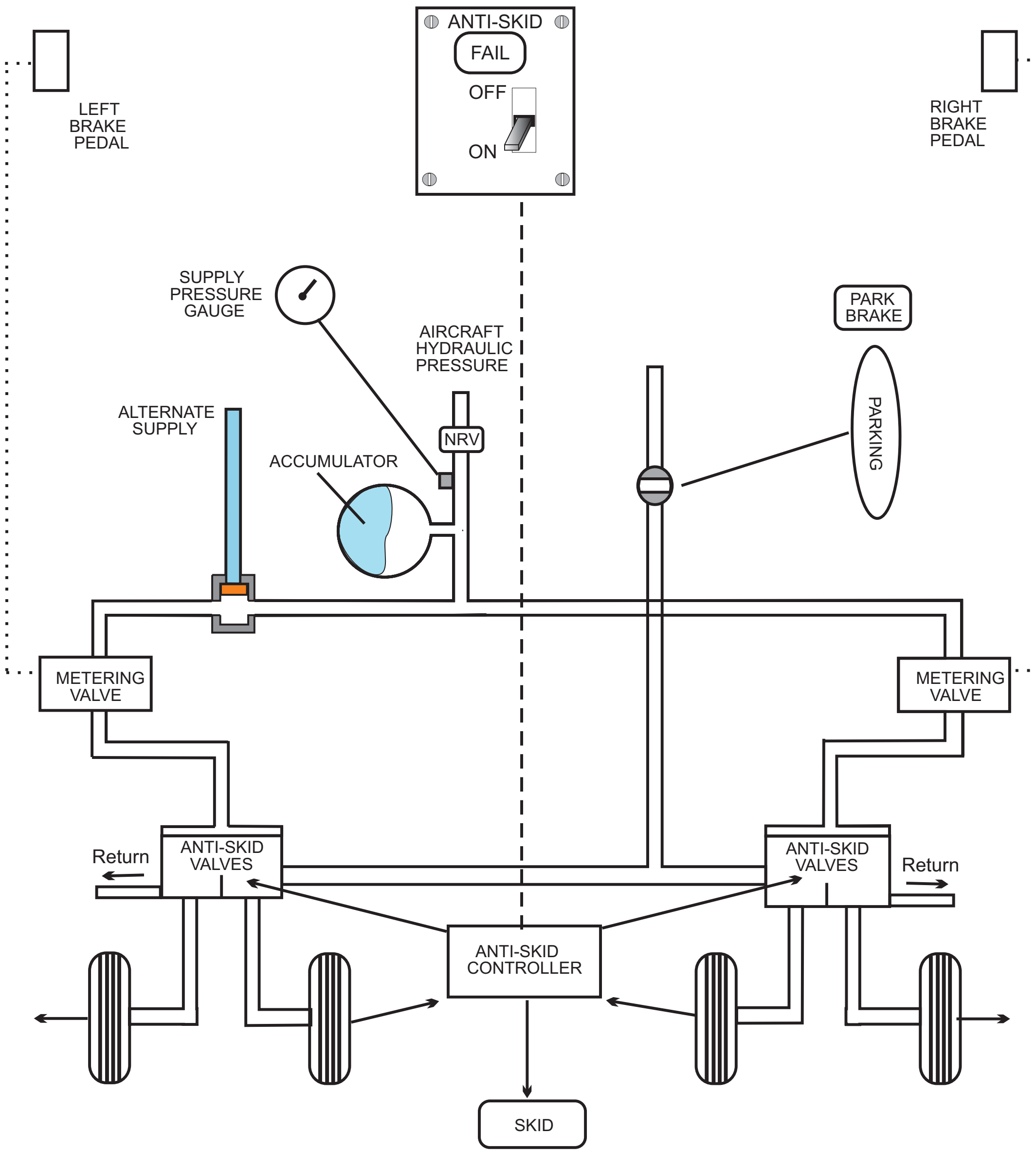


Diagram 8.10 Electronic Anti-Skid System Schematic

SEMI MODULATING SYSTEM

Semi modulating, or **quasi-modulating** systems attempt to continuously regulate brake pressure as a function of wheel speed. Typically, brake pressure releases when the wheel deceleration rate exceeds a pre-selected value. Brake pressure is re-applied at a lower level after a length of time appropriate to the depth of skid. Brake pressure then gradually increases until another incipient skid condition is sensed. In general, the corrective actions taken by these systems to exit the skid condition are based on a pre-programmed sequence rather than the wheel speed time history.

FULLY MODULATING SYSTEM

Fully modulating systems are a further refinement of the semi or quasi-modulating systems. The major difference between these two types of anti-skid systems is in the implementation of the skid control logic. During a skid, corrective action is based on the sensed wheel speed signal rather than a pre-programmed response. Specifically, the amount of pressure reduction or reapplication is based on the rate at which the wheel is going into or recovering from a skid. Also, higher fidelity transducers and upgraded control systems are used, which respond more quickly.

TOUCHDOWN PROTECTION AND BRAKING EFFICIENCY

Whichever electronic system is used, on touchdown the brakes are prevented from being applied until the wheels speed up to between 15 and 21 mph. If the pilot lands with the toe pedals fully depressed and anti-skid selected on, there is no braking action until the wheels have achieved this speed. As a precaution to prevent the nose leg from being slammed onto the runway, large aircraft braking systems are also prevented from applying full pressure until the air/ground logic system senses that the nose wheel is in contact with the ground.

To achieve maximum braking efficiency, the pilot also needs to apply the wheel brakes as early as possible, and increase the friction between the tyres and runway surface, by lowering the nose wheel onto the runway as soon as possible to initiate the braking action. Deploying the speed brakes and spoilers to disrupt the airflow over the wings destroys lift, which allows the aircraft's weight to sink onto the main wheels, increasing the friction between the tyres and runway surface.

AUTO BRAKE SYSTEM

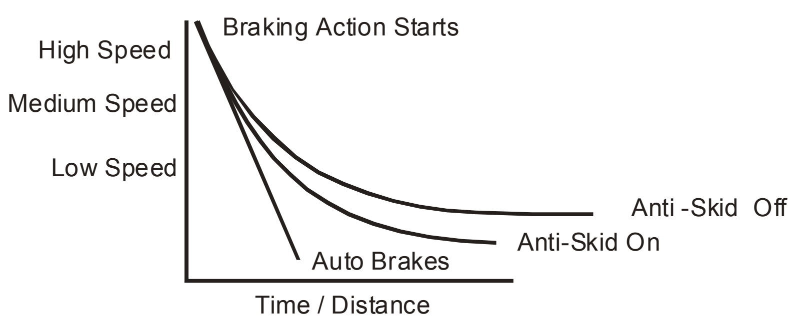


Diagram 8.11 Graphical Representation of the Effects of Auto Braking

Modern aircraft fitted with auto land systems are also fitted with auto brake systems. In the auto brake system, the pilot does not touch the toe pedals during the landing run. Normal braking is more effective at higher speeds and then tails off as the speed decreases and the brake temperature increases. Aircraft braking performance is shown in diagram 8.11. Anti-skid systems, whilst increasing the braking effectiveness, still suffer the same drop as manual braking. Auto brake systems, through electronic programming, maintain a constant rate of retardation g (deceleration). To achieve this and prevent the wheels from locking up, a serviceable anti-skid system must be set to the on position.

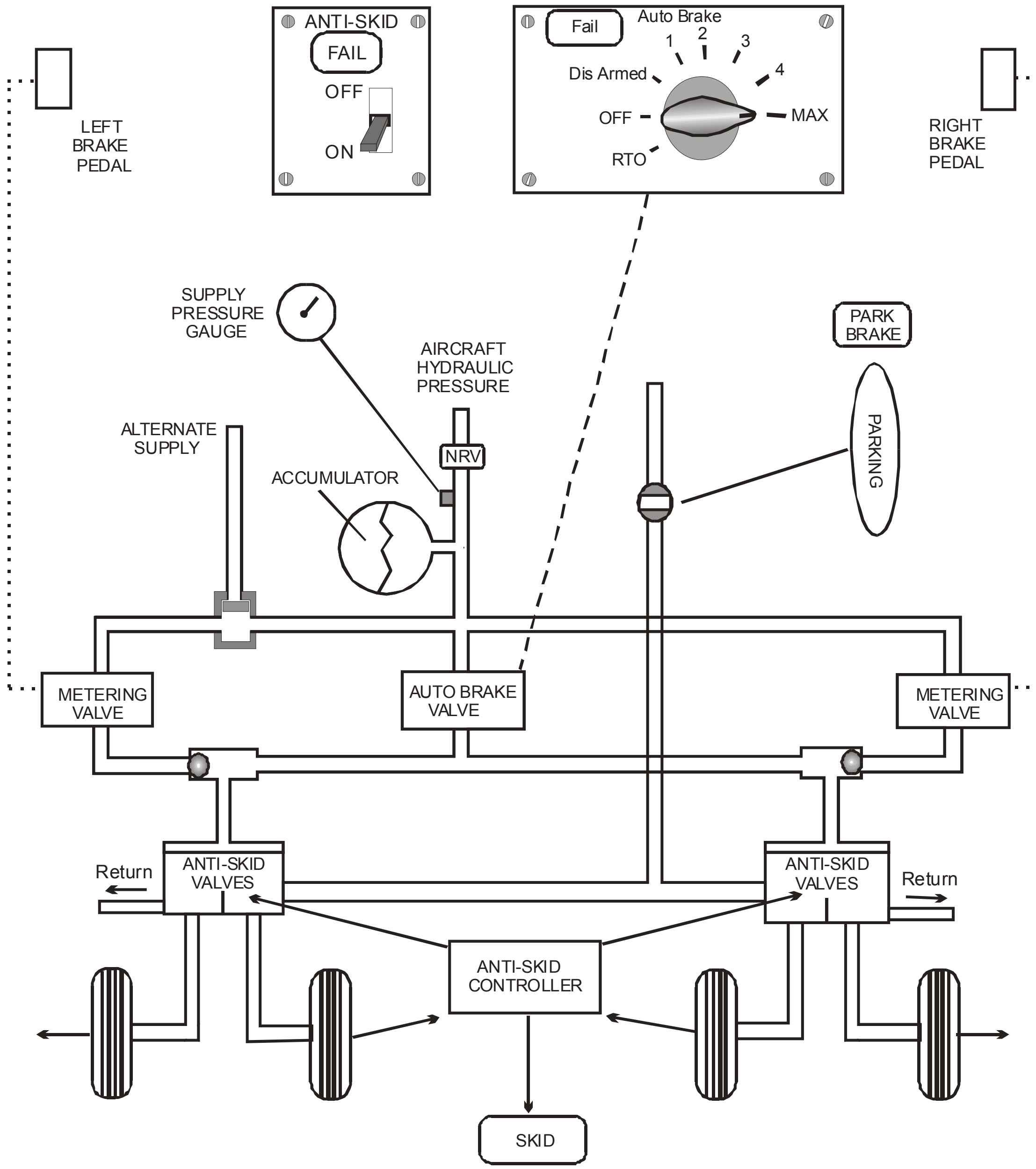


Diagram 8.12 Auto Brake Schematic

The pilot has a flight deck control panel from which to select the rate of deceleration, as shown in diagram 8.12. This can be altered throughout the landing run to increase or decrease the rate of braking. The auto brake system uses aircraft system hydraulic pressure and feeds it directly to the anti-skid valves and then the wheel brakes. This extra pressure increases the force of the friction pads on the brake disc and keeps the wheels at a point even closer to the skid than the normal anti-skid system. Operation of the auto brake system increases the wear and tear on tyre covers and brake units considerably.

In the event of an auto brake system failing, the braking returns to manual anti-skid. In the event of the anti-skid system failing, the auto brake system disconnects and the braking system returns to manual. Fail lights illuminate to warn the pilot of any unserviceable condition.

Auto brake selections, deceleration rates, and wheel brake pressure

Selection	**Rate of deceleration**	**Hydraulic pressure applied**
1	2 ft/sec	1100 psi
2	4.5 ft/sec	1350 psi
3	7 ft/sec	1600 psi
4	9.5 ft/sec	1800 psi
Max_1	12 ft/sec	2100 psi
Max_2	14.7 ft/sec	2100 psi
RTO	Uncontrolled rate of deceleration	3000 psi

Note: The table provides an indication of the type of deceleration rates and hydraulic pressure that are applied for a given auto brake selection. When Max is selected, the rate of deceleration increases as the aircraft's groundspeed decreases, hence the values for Max_1 and Max_2.

REJECTED TAKE-OFF

It is not a legal requirement to land using the auto brake system. However, if a serviceable auto brake system is fitted, it is required that the auto brake be set at Rejected Take-Off (RTO) for every take-off. Selecting this arms the braking system. If the pilot touches the toe brakes and applies brake pressure above a set value in the take-off run, the system disarms.

In the event of rejecting a take-off, for any reason, as the pilot retards the thrust lever, the brakes are applied at full power and bring the aircraft to a halt. Not all RTOs require removal of the main tyres. The determining factors are:

- Speed
- Load
- Distance

Where speeds remain below the normal landing speeds, consequently, normal braking energies are experienced. The tyres may remain in service. It is recommended that the tyres cool for approximately thirty minutes prior to recommencing normal operations. If higher than normal landing speeds are exceeded, as a consequence, higher than normal braking energies are experienced. In this case, the tyres require replacement.

MULTI-PLATE BRAKE DISC

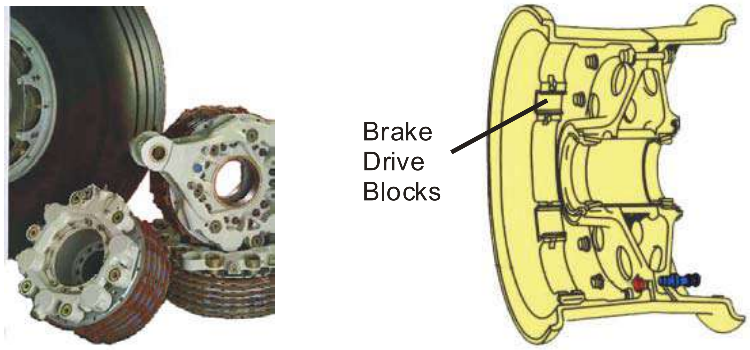


Diagram 8.13 Multi- Disc Brakes and Wheel Hub

To absorb the kinetic energy of larger aircraft, multi-plate disc brakes are used. These units fit inside the wheel hub (see diagram 8.13). **Tenon blocks** of the wheel hub drive the rotating discs, termed rotors.

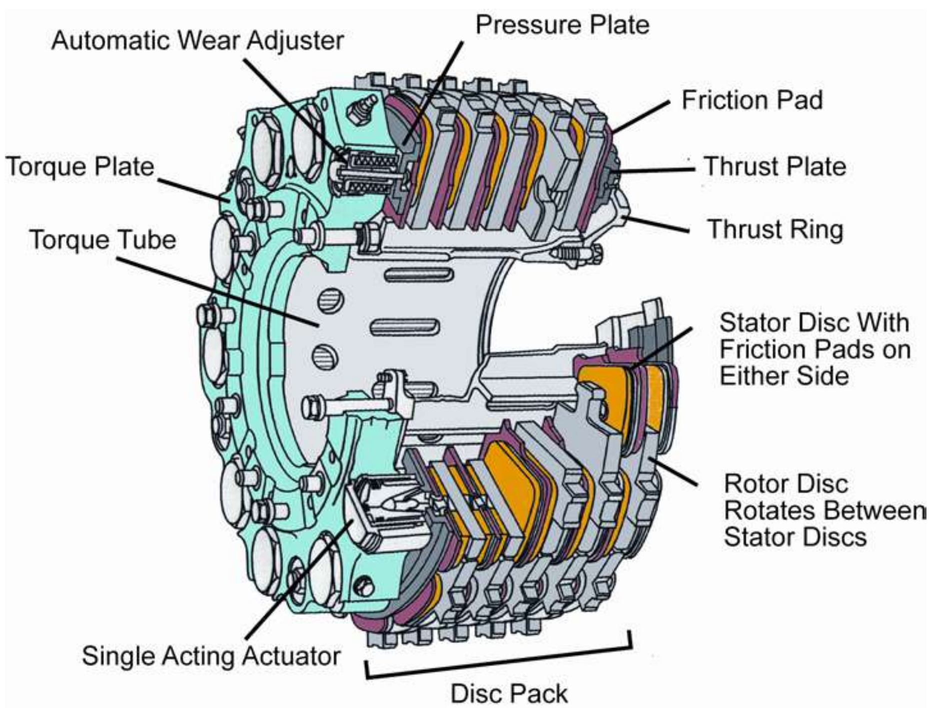


Diagram 8.14 Component Parts of a Multi-Disc Brake

Diagram 8.14 shows a multi-disc unit consisting of a torque plate connected to the undercarriage, housing a series of single acting actuators, a disc pack consisting of rotors and stators, a pressure plate, a torque tube, a thrust plate, and a thrust ring. The rotors, normally the disc material, are castellated and engage in the tenon drives on the wheel. To allow for expansion and prevent the discs from warping, the discs are made in interlocking segments (jigsaw).

The stators, normally the friction material, are produced as a ring with castellations on the inner diameter. These interlock with castellations of the torque tube and, while able to slide from side to side, cannot rotate. The torque tube is bolted to the torque plate and thrust ring which supports the thrust plate. The action of braking causes the single acting actuators to push the pressure plate against the disc pack, forcing it against the thrust plate. This increased contact area of friction pads and discs creates a high torque force.

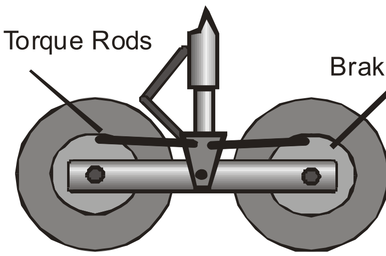


Diagram 8.15 Bogie Torque Rods

For large air transport aircraft with bogie undercarriages, this torque load transfers from the torque plate to the main gear legs via torque rods, as in diagram 8.15.

RETRACTION PIN/AUTOMATIC WEAR ADJUSTER

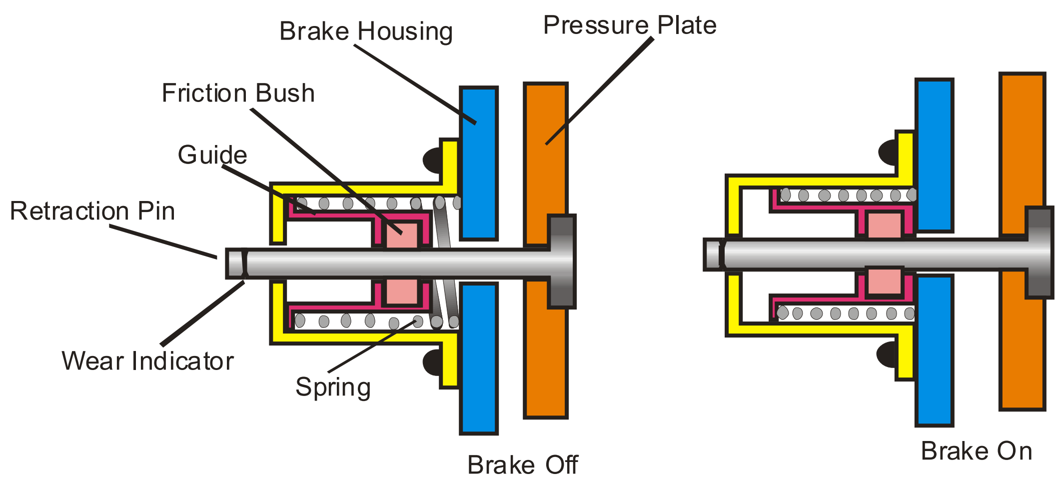


Diagram 8.16 Automatic Brake Wear Adjuster and Retraction Pin

As the brake cylinders are single acting, a mechanical force is required to release the brakes when the pilot's feet are removed from the toe pedals. This action can be incorporated with an automatic wear adjuster that ensures the brakes are fully released. The clearances between the discs and the pads are kept to a constant minimum (see diagram 8.16) by the operation of the **retraction pin** and the **friction bush**. As the brakes are applied, the pressure plate pulls the retraction pin. When the brake is released, the compression spring expands and pulls the pressure plate back, allowing the rotation of the wheel to separate the discs.

As the brake's discs and pads wear, the guide bottoms against the torque plate and the pressure plate pulls the pin through the friction bush. This allows the brake to take up the wear. Each time the brake is released, the brake discs clear by the same distance, thus ensuring that the braking action is smooth and that there is no lost motion (where the brake pedal moves a distance before the brakes are applied).

STICKING AND DRAGGING BRAKES

Hot brakes left on when the aircraft is parked can result in the discs and stators "gluing" themselves together, causing the brake to stick in the on position. If the brake disc and pads do not fully separate after operation of the brakes, the affected wheel turns more slowly. This is termed dragging.

BRAKE WEAR

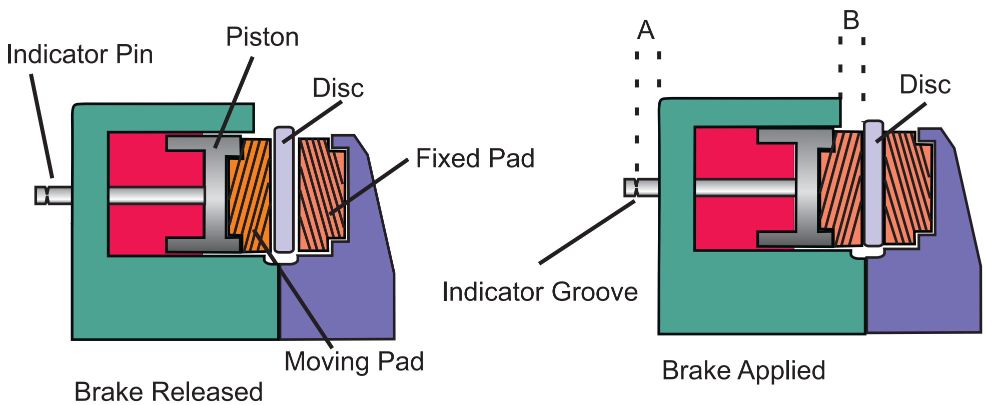


Diagram 8.17 Brake Wear Measurements

Brake wear is only measured when the brakes are fully applied. Depending on the brake unit, this measurement can be taken using a rule, a gauge or special tool, or via visual indication.

Refer to the Brake Applied diagram of 8.17. Whichever method is used, the wear is checked **between** the brake housing (torque plate) and the disc denoted as B. The actual measurement, and in the case of visual indication, can be taken between the indicator groove and the brake housing of the protruding indicator pin. The amount that the pin protrudes from the housing indicates how much material is left before the brake is worn out. It is the pilot's responsibility to ensure that there is sufficient brake wear left for the take-off and landing prior to making the flight.

BRAKE CHATTER/BRAKE FADE

As the brakes heat up through operation, they become less efficient as the disc's thermal capacity decreases. This can lead to brake chatter and squealing noises. Continued application of the brakes can lead to brake fade. This occurs when the brakes are applied but there is little or no braking action. A temperature sensor warns pilots of air transport aircraft as to the condition of the brakes.

BRAKE TEMPERATURE

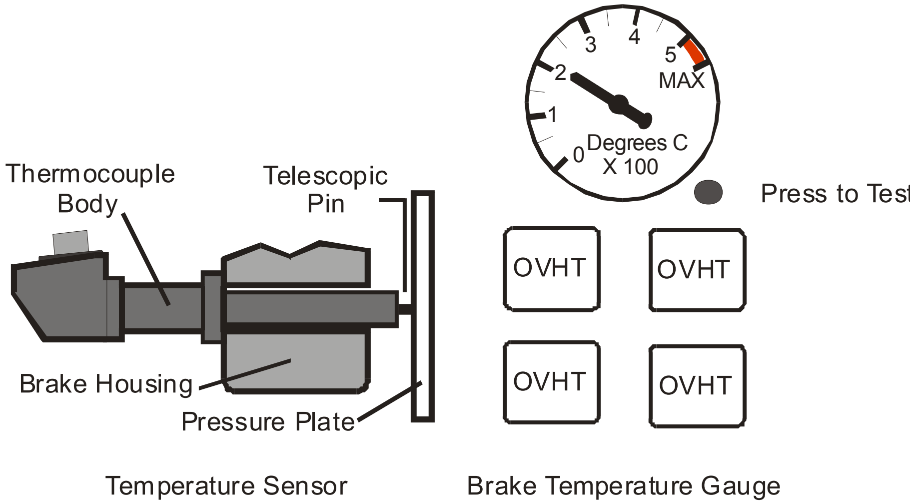


Diagram 8.18 Brake Temperature Indications

A sensor unit with a telescopic pin (see diagram 8.18) that is in contact with the pressure plate sends a signal to the flight deck (an analogue gauge showed temperature in older aircraft). This defaults to the highest wheel brake temperature. Each brake unit has an amber caution light with the legend **OVHT** (overheat). By pressing the appropriate caution light, the pilot can read that brake unit's temperature. Pressing the test button checks the continuity of the indications by illuminating the overheat lights and deflecting the gauge needle.

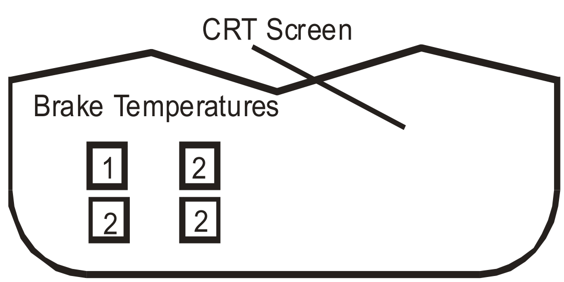


Diagram 8.19 Glass Cockpit Brake Temperature Indications

For modern aircraft with CRT screens, the brake temperature appears as a number inside a box. As the temperature increases, the number increases and the colour of the number and the outline box change. Prior to take-off, the pilot must ensure that the brakes can absorb the energy of a rejected take-off at that mass. If they cannot, the option is to reduce the mass or await a brake-cooling period.

RECOMMENDED TAXI SPEED

The JAA recommends that an aircraft taxis at 21 mph or less as taxiing and turning create more heat than would normally be expected. Brake units from the late 1960s were manufactured from steel alloys for the discs and ceramics for the pads. Modern brake units are manufactured from carbon fibre composites. These units operate more efficiently at the higher temperatures of fast/heavy weight landings than the older units, but are less tolerant of slow speed, low temperature braking.

BRAKE COOLING

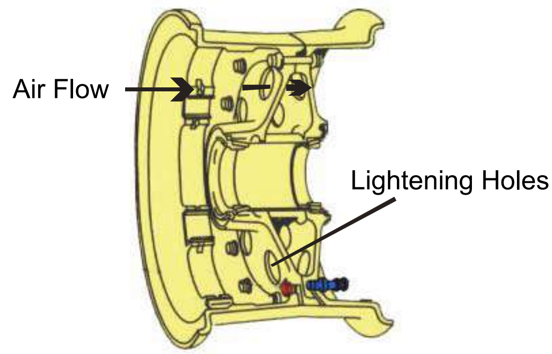


Diagram 8.20 Air Flow to a Brake Unit

As the wheel shields modern multi-disc brake units, the heat generated by the brakes can only escape via conduction and radiation into the surrounding structure and convection through any air passing across the brake unit. The airflow across the brake unit is restricted to that which enters through the wheel's lightening holes. To increase this airflow and cool the brakes more quickly, some large aircraft fit cooling fans in the wheel units. These cooling fans come on automatically upon reaching a threshold temperature.

BRAKE OVERHEAT

If an aircraft suffers a brake overheat, the airport Fire Chief must be informed and the aircraft cannot be refuelled until the Fire Chief provides permission. It is standard that an aircraft suffering from overheated brakes on landing must stop when they are clear of the runway to allow for inspection by the fire service and cooling, prior to taxiing to the stand.

SPONGY BRAKE OPERATION

If air or any other gas enters the hydraulic brake system, the gas compresses when the brakes are applied, leading to poor braking. This is termed spongy brake operation. Bleeding the brake system to remove the air remedies the condition.

WHEELS AND TYRES

TYPES OF WHEEL HUB

There are several designs of wheel hubs used in aviation. These are:

- Well base
- Detachable flange
- Split hub

WELL BASE

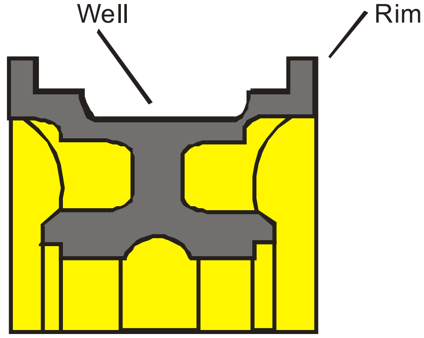


Diagram 8.21 Well Based Wheel Hub

Well-based wheel hubs are similar to car wheels in that they are one piece. This requires the tyre's internal diameter to be forced over the wheel rim, which makes them only suitable for use on light GA aircraft with low-pressure tyres.

DETACHABLE FLANGE

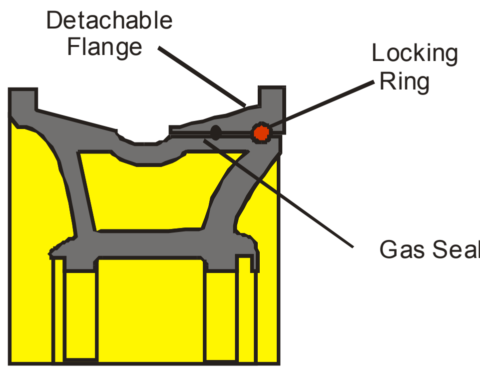


Diagram 8.22 Detachable Flange

Detachable flange wheels (see diagram 8.22) allows removal of the flange and the tyre to be taken off the wheel. The pressure in the tyre retains the flange, trying to force it over a locking ring. For tubeless tyres, a gas seal fits between the flange and the hub.

SPLIT HUB

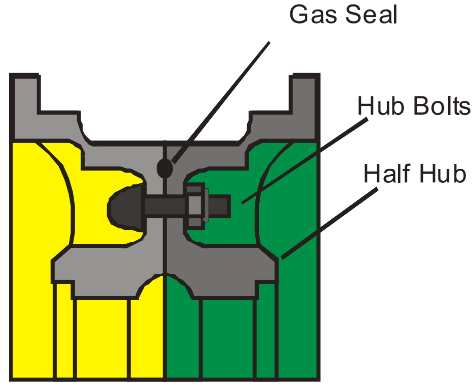


Diagram 8.23 Split Hub

These wheel hubs consist of two parts, referred to as half hubs, which are cross-bolted together. As this allows for easy fitment of tyre covers, it has become a common wheel type used on many GA and air transport aircraft. To make the hub gas tight where tubeless tyres are used, an O-ring seal is fitted between the two half hubs.

FUSIBLE PLUG

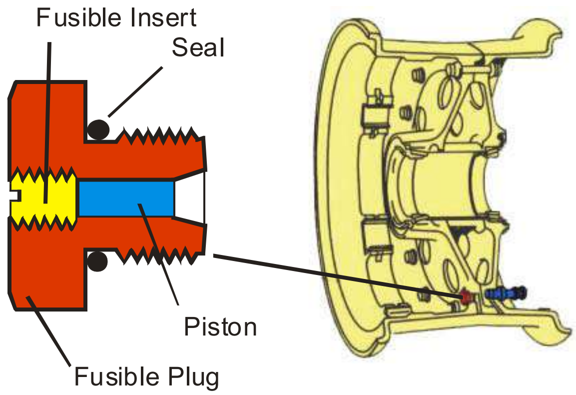


Diagram 8.24 Fusible Plug

As a safety precaution, to prevent over-pressurisation through overheating, fusible plugs are fitted in tubeless tyre wheel hub assemblies (see diagram 8.24). These plugs have a threaded insert of low melting point alloy. If the wheel temperature reaches a point where the fusible insert melts, the tyre inflation medium (nitrogen) is released at a controlled rate. This prevents tyre covers from exploding (with the force of a grenade) at high temperatures. The common value for an air transport aircraft fusible plug is 150°C. To indicate that the plug is set at this temperature, it is coloured red. Prolonged braking leads to slow tyre deflation. This works on the principle that prolonged braking generates excessive heat and the fusible plugs melt.

REGIONS OF A TYRE

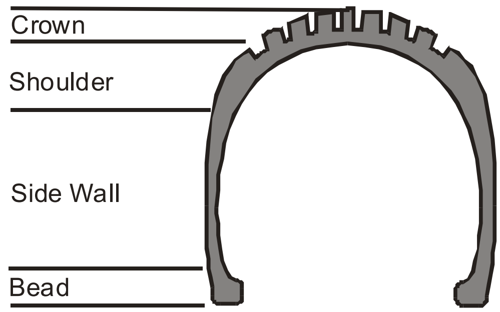


Diagram 8.25 Regions Of a Tyre

Referring to diagram 8.25, the tyre divides into 4 areas:

Crown
This area has the tyre tread and is designed to withstand the wear of normal operation.

Shoulder
This is a change in profile thickness from the crown and is not designed to take wear.

Sidewall
This is the thinnest and, therefore, weakest section of a tyre and is designed to flex when loads are applied.

Bead
This is designed to fit against the rim of the wheel, known as the bead seat.

STRUCTURE OF A TYRE

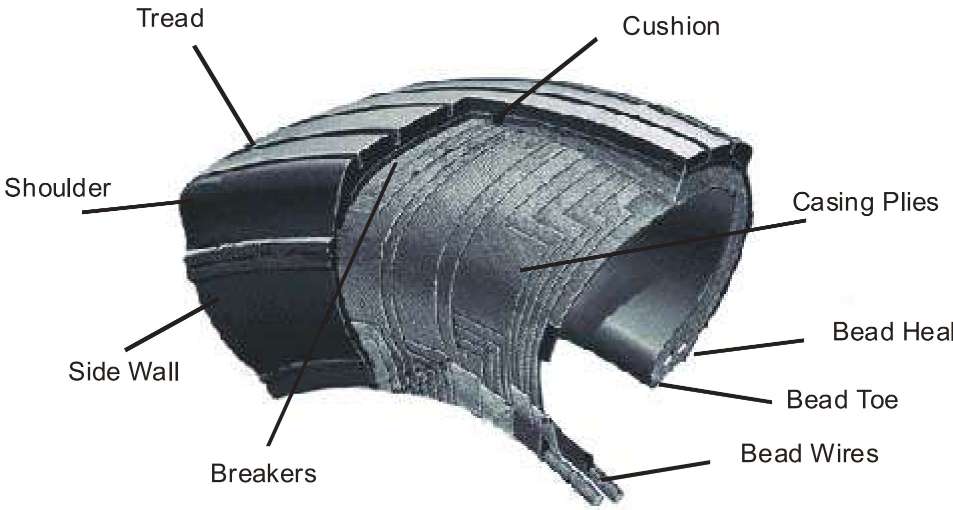


Diagram 8.26 Build Up Of A Cross Ply Tyre

Refer to diagram 8.26 of a cross ply tyre. The steel wire cores located in the bead form the backbone of the tyre. These cores consist of strands of high tensile steel coated in rubber and formed into a core. All other parts of the tyre anchor to these. Layers of casing plies provide the strength of the tyre.

Each ply is formed from a series of high modulus cords individually coated with rubber and laid side by side to form a sheet. Each sheet then attaches to the core wires, and the plies build up in successive layers.

There are two basic types of tyre lay ups. These are belted radial ply and biased cross ply. In the cross ply tyre, each casing ply is laid with the cords running at opposite bias angles. The number of plies and their angle dictate the tyre's strength and its load capability. During rotation under load, the bias ply tyre flexes to such an extent that the inter-ply friction creates heat within the tyre's structure. Adding reinforcing plies limits this flexing but increases the tyre's mass.

RADIAL BELTED TYRE

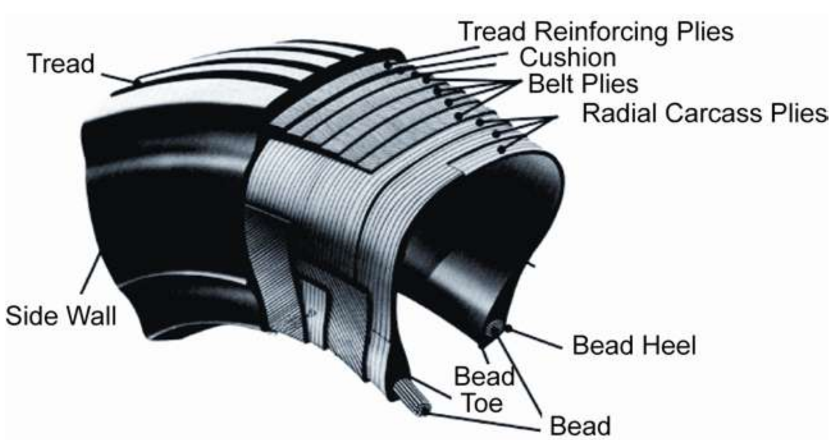


Diagram 8.27 Build Up Of A Radial Ply Tyre

Radial belted tyres have casing plies orientated in the plane of the tyre's cross-section. These are frequently made of steel, coated in rubber, and laid up as a sheet. Biased textile cord belts lay over the casing plies, and above this is a steel protector ply.

When the tyre is loaded, the tread is stabilised by the belt package. This reduces tyre wear due to the tread scrubbing against the runway. As there is a reduced number of plies, there is a reduction in inter-ply friction and, therefore, less heat build up within the tyre casing. When comparing two tyres, one as a bias ply, the other as a radial belted ply, the radial belted ply tyre can have a mass saving of approximately 20%.

FOOTPRINT AREA

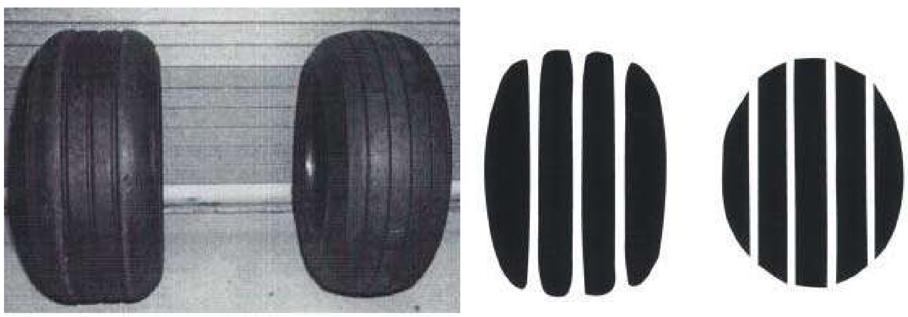


Diagram 8.28 Cross Ply and Radial Ply Tyres and Tread Footprints

Diagram 8.28 shows two tyres of the same size and their footprint area when equally loaded. The tyre on the left and the left footprint is of a cross ply tyre. The right tyre and footprint is a radial ply tyre. Comparing the two tyre footprints shows that the cross ply tyre has more tread in contact with the runway, therefore exerting the aircraft's weight over a larger area, which in turn reduces the friction between the tyre and the runway.

TUBED TYRES

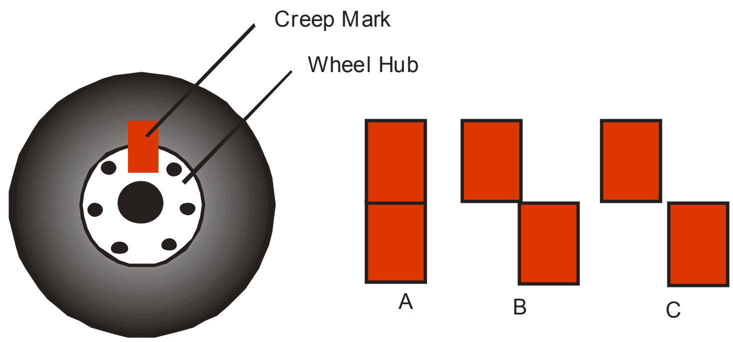


Diagram 8.29 Creep Marks on Tubed Tyres

Originally, all tyres were fitted with an inner tube to form a gas bladder with an inflation tube passing through the hub of the wheel. This increased the weight of the tyre and, due to the extra layer, created more heat when the tyre flexed. A further disadvantage was that when a new tyre was fitted to a wheel, due to the small surface area of the tyre in contact with the wheel and the friction created between the tyre and the runway, it resulted in the tyre moving circumferentially around the wheel. It takes up to five landings for the tyre to bed in against the wheel.

If the angular movement between the tyre and the wheel is sufficient, the inflation pipe can be ripped from the inner tube. To overcome this, creep marks, as per diagram 8.29, are painted across the tyre and the hub. For tyres up to 24” diameter, the creep mark is 1” across. For tyres over 24” diameter, the creep mark is 1.5” across. Tyres may continue to be used up until the point where the opposite edges of a displaced creep mark are in alignment (see B above). The wheel requires replacement for any movement past this (see C above).

TUBELESS TYRES

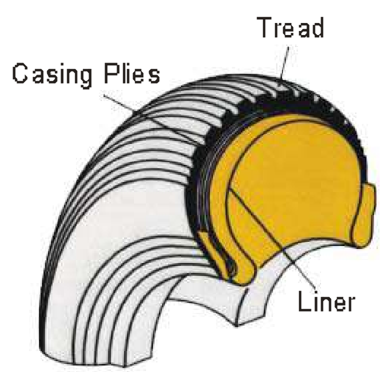


Diagram 8.30 Tubeless Tyre

In tubeless tyres, a gas tight rubber compound is manufactured as part of the tyre’s build up. This saves weight, and these tyres run cooler. The inflation adapter for tubeless tyres fits into the wheel hub. The tyre’s bead setting against the flange of the wheel forms a gas tight seal. Whilst these tyres also take time to bed in, tyre creep is not such a problem as it is for tubed tyres.

VENT HOLES — TUBELESS TYRES

Vent or **Awl** holes are small needle holes (the tool is termed an awl) made in the lower sidewall of the tyre above the wheel flange area. On tubeless tyres, the vent holes are marked with green litho ink or paint. They penetrate only to the mid casing. They provide an escape path to atmosphere for any inflation medium that has permeated naturally through the inner liner, which would otherwise build up within the tyre casing and cause ply or tread separation.

VENT HOLES — TUBED TYRES

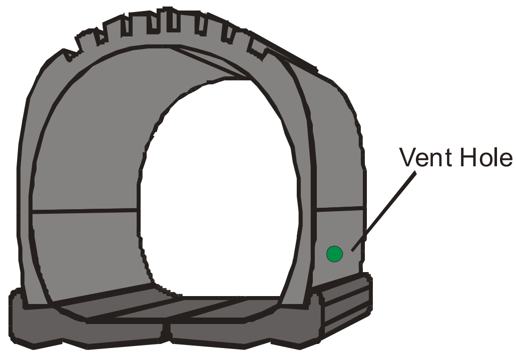


Diagram 8.31 Awl or Vent Holes

The vent holes of tubed tyres are marked with grey or silver paint or litho ink. They penetrate completely through the tyre casing. Their purpose is to allow the escape of any inflation medium that may become entrapped between the inner tube and the tyre wall.

TYRE MARKINGS

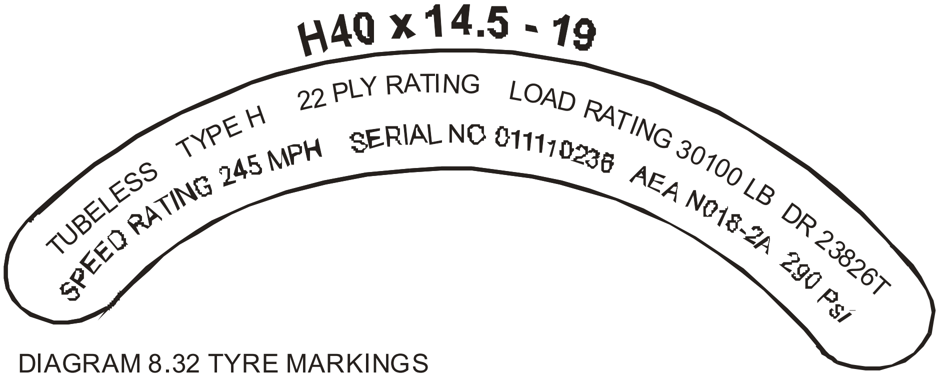


DIAGRAM 8.32 TYRE MARKINGS

Moulded into the external sidewall of the tyre is a series of markings, as per diagram 8.32. These include the:

Type of tyre
In this case, the tyre is tubeless type H.

Ply Rating
In this case, the tyre has the strength equal to 22 cotton plies.

> **Note:** The ply rating number does not indicate the physical number of piles. Together with the load rating, it indicates the strength and corresponding inflation pressures. See AEA No. below.

Load Rating
In this case, the tyre has maximum static load of 30 100 lb.

Part No.
This is a number specific to the company who manufactured the tyre.

Speed rating
In this case, 245 mph is the maximum groundspeed for which the tyre is tested and approved.

Tyre Pressure
This indicates the tyre pressure at which the tyre is inflated to prior to fitment to the aircraft.

If the tyre has been re-treaded, there is also an identifier as to the date and stage of re-tread.

BALANCE MARKER

The lightest point of a tyre cover is indicated by a red spot or triangle painted on the sidewall of the tyre. To statically balance the wheel assembly, the cover's light spot is located opposite the heavy spot of the wheel hub. For a tubeless tyre, unless otherwise indicated, this is normally the inflation adapter. For tubed tyres, this is the inflation adapter of the inner tube.

TYRE PRESSURE CLASSIFICATION

Aircraft tyres are subjected to high impact loads, high rolling speeds, and high temperatures. Therefore, they are manufactured to high specifications. Tyres are classed as either high pressure or low pressure. Exceeding the pressures or groundspeeds can cause the tyre to explode. See the Tyre Classification Table below.

Tyre Classification	Maximum Tyre Pressure	Maximum Groundspeed
High Pressure	315 psi	250 mph
Low Pressure	200 psi	120 mph

CIRCUMFERENTIALLY GROOVED TREAD

Diagram 8.33 Circumferentially Grooved Tyres

Tyres have tread to provide grip, clear water, and provide directional stability. The standard air transport aircraft tyre for landing on prepared surfaces is the circumferentially grooved tyre. In diagram 8.33, the left wheel assembly is for an air transport aircraft, the right wheel assembly is for GA aircraft.

TYRE WEAR

Landing and braking wear the tread areas down. This results in the tyre being less able to clear water from under the footprint area. The CAA specifies that circumferentially grooved tyres require replacement when the indicator groove(s) are 2 mm deep. This limitation is imposed due to the climatic and operational conditions that are conducive to aquaplaning.

CIRCUMFERENTIAL GROOVED TYRES WEAR INDICATION

For a tyre with an uneven number of circumferential grooves, the centre groove serves as the indicator. For tyres with an even number of circumferential grooves, the tread depth is measured across the central rib.

REINFORCED TREAD

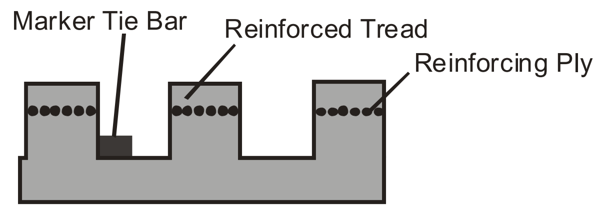


Diagram 8.34 Wear Indicators for Reinforced Tread

Some circumferentially grooved tyres have fabric-reinforcing plies built into the tread ribs. As these tyres wear down, the fabric plies show. To prevent replacement of these tyres due to misinterpretation, 2 mm high wear indicator blocks, termed **marker tie bars**, are moulded into the grooves at regular intervals, as diagram 8.34 shows.

BLOCK TREAD

Diagram 8.35 Block Tread Tyres

Tyre treads for landing on prepared and unprepared surfaces take the form of block tread to grip loose material, as shown in diagram 8.35. Block tread tyres are used until the outline of the tread block is just discernible on the cover.

MARSTRAND TYRES

Diagram 8.36 Marstrand Tyre

Diagram 8.36 shows twin contact tyres for use with fully castoring nose/tail wheels. They prevent the wheels from shimmying due to the two separate contact areas. The one contact area opposes the rotation of the other; hence, the tyre is longitudinally stable. Marstrand tyres are replaced when there are rolling contact wear marks appearing on the central groove.

CHINED TYRES

Diagram 8.37 Chined Tyre and Spray from a Nose Wheel

Chined tyres are designed to deflect the water cleared by the tyre treads back onto the runway, preventing a jet of water being ingested into the intake of rear engine aircraft. To give an indication of the problem of water ingestion, refer to the right picture in diagram 8.37 showing the water spray from the front wheels of an A340 during trials on a contaminated runway.

The chine takes the form of a lip attached to the shoulder of the tyre. In the picture on the left of a twin-wheeled leg in diagram 8.37, the chines are visible on the outer sidewall of the tyres. Single-nose wheel assemblies can have a chine on each side of the tyre cover. Normal limitations for tyre wear apply to the tread of these tyres. However, any damage to the chine renders the tyre unserviceable.

TYRE INFLATION MEDIUM

Nitrogen is the normal inflation medium for air transport aircraft tyres. It is a JAR requirement that air transport aircraft of more than 75 000 lb (34 090.91 kg) max take-off mass have their tyres inflated with dry nitrogen, or other gases shown to be inert, so that the gas mixture in the tyre does not contain oxygen in excess of 5% by volume.

In the event that dry nitrogen, or another inert gas, is not available, air may be used provided that either:

- The oxygen content after adjustment stays below 5% of volume, or
- In the following 15 flight hours, the tyre is purged and inflated with dry nitrogen in order to restore the oxygen content to a level below 5% of volume.

TYRE INFLATION FOR BEM LOAD

When the tyre is inflated to the pressure specified on the cover and then fitted to an aircraft, its pressure increases by approximately 4% due to the tyre's deflection. This deflection reduces the tyre inflation chamber volume.

If a tyre is inflated while under load, the inflation pressure marked on the cover must be increased by 4% to bring the tyre to the correct pressure for the aircraft's Basic Empty Mass static load.

TYRE STRETCH

After a new tyre is fitted to a wheel and inflated to the cover pressure, the tyre stretches, or grows, resulting in a drop in inflation pressure. This requires rectification before use.

DISSIPATION

For both tubed and tubeless tyres, there is an acceptable dissipation rate of 5% of the tyre's inflated pressure over a period of 24 hours. This is due to that fact that the molecules of nitrogen (or air) can pass through the material of the inner tube or tyre cover's liner.

EFFECTS OF AMBIENT TEMPERATURE

CHANGE ON TYRE PRESSURE

As an approximate guide, any increase in the ambient temperature by 3°C (5°F) causes the tyre's inflation pressure to increase by 1% and vice versa. If an aircraft is flying to a destination where the ambient temperature differs by 25°C, adjusting the tyre pressure for the cooler climate is necessary.

COLD TYRE PRESSURE CHECKS

Tyre pressures should be checked when the tyres are cold (at ambient temperature). See the table below for actions and pressures.

Pressure readings as a % of operational pressure	Tyre status	Action required
More than 105%	Over-inflated	Re-adjust pressure to maximum of normal operating range
105 — 100%	Normal operating pressure range	Do not adjust tyre pressure
100 — 95%	Acceptable daily pressure loss	Re-adjust pressure to maximum of normal operating pressure range
95 — 90%	Accidental pressure loss	Re-adjust pressure to maximum or normal operating pressure range Record action in Tech Log book Re-check pressure after 24 hours If pressure loss is greater than 5%, replace tyre assembly.
90 — 80%	Pressure loss	Remove assembly from aircraft
80 — 0%	Major pressure loss	If the tyre was rolling when the loss occurred, remove affected tyre and its axle companion if fitted as twin or bogie and scrap them. If the tyre was stationary, the tyre at the time of pressure loss can be reused after inspection. The likely cause is fusible plugs blowing.

HOT TYRE PRESSURE CHECKS

After landing, tyre pressures can increase by up to 10% due to the kinetic heating. If a single tyre's pressure is found to be above this limit, the wheel should be inspected for possible brake or bearing problems. Do not bleed off inflation pressure from a hot tyre.

Refer to the table below for pressures and correct course of action.

Axle type	Pressure readings as a % above operational pressure	Tyre status	Action required
Single wheel	Up to 10%	Normal	None
Twin wheel	Up to 10 %	Normal	If one tyre is down, top it up to the higher pressure. Record the action in the Tech Log and note the ambient temperature. If on the next *hot* check the pressure has dropped, replace the tyre assembly.
Bogies	Up to 10 %	Normal	All tyres on a bogie should be within 5% of the highest tyre pressure reading. Any tyre below this 5% range should be topped up to within 5% of the highest pressure reading. Record the action in the Tech Log and note the ambient temperature. If on the next *hot* pressure check a similar drop is noted, the tyre should be rejected.

FLIGHT DECK TYRE PRESSURE INDICATION

It is recommended, but not a JAA requirement, that aircraft should have a system to warn pilots of deflated tyres during flight, so that they can adjust the landing or divert, etc. On modern aircraft with glass cockpits, this information can be displayed on an ECAM systems page. These systems go a step further by giving current tyre pressure and temperature. This enables the pilot to calculate braking efficiency more accurately.

A sensor mounted on the wheel within the tyre cover detects the temperature and pressure, which is then relayed to a transducer mounted within the axle.

AQUAPLANING

The term given to a condition where the aircraft's tyres are riding on a liquid film and are not in direct contact with the runway surface is aquaplaning. The resulting effects are:

- Wheel skids, which damage or burst the affected tyre(s), due to the brakes locking the wheel(s)
- Increased landing roll, due to the loss of braking efficiency
- Loss of directional control

Aquaplaning, a European term, can also be referred to as **hydroplaning**, the American term.

There are three forms of aquaplaning:

Reverted rubber
This can occur on touchdown.

Dynamic
This can occur during the landing roll or take-off roll.

Viscous
This can occur during taxiing on taxiways or on stands.

Anti-skid systems aid in controlling aquaplaning conditions by modulating the wheel brakes and ensuring that brakes do not lock the wheels up, so that they can spin up and start to clear the water beneath them.

REVERTED RUBBER AQUAPLANING

Reverted rubber aquaplaning is less well known than dynamic aquaplaning. It is caused by the friction-generated heat that produces superheated steam at high pressure under the tyre's footprint. The high temperature causes the rubber to revert to its uncured state and form a seal around the tyre area that traps the high-pressure steam.

This condition occurs on damp runways or if the touchdown point is on an isolated damp spot of an otherwise dry runway. The effect is that the wheel does not spin up, which results in a reverted rubber skid. Refer to wet braking flat spots for the type of tyre damage that this can cause.

DYNAMIC AQUAPLANING

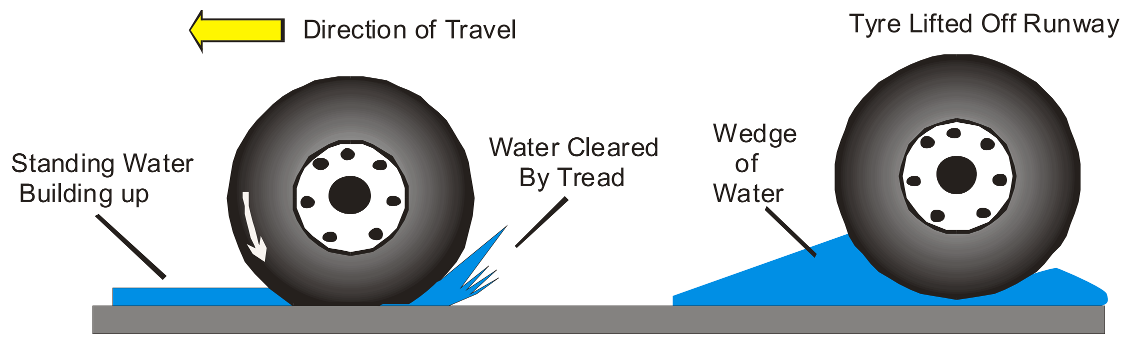


Diagram 8.38 Tyre Aquaplaning

Dynamic aquaplaning occurs when standing water on a wet runway is greater than the tread depth of the tyre. In this case, the tyre‘s treads cannot displace water from under the tyre fast enough to allow the tyre to make pavement contact over its total footprint area. If the tread depth of the tyres on an aircraft is greater than the depth of the water on the runway, then dynamic aquaplaning does not occur.

As the treads are unable to clear the water, it is pushed as a wedge in front of the tyre. When dynamic aquaplaning occurs, the tyre rides up onto the wedge of water, as in diagram 8.38. When the tyre is no longer in contact with the runway surface, the tyre is totally aquaplaning. In this situation, the centre of pressure in the tyre footprint area can move forward, and the wheel can stop rotating with the attendant losses of braking and directional stability.

CROSS PLY AND RADIAL PLY TYRES

As radial and cross ply tyres have different footprint areas and exert different ground pressures, they should not be mixed across an axle line. Either type may be used exclusively for all wheels or one type for the nose wheels and the other type for the mains.

AQUAPLANING SPEED FOR RIBBED TYRES

$$9 \times \sqrt{\frac{\text{Tyre Pressure in Psi}}{\text{SG of Contaminant}}}$$

$$34 \times \sqrt{\frac{\text{Tyre Pressure in Kg/Cm}^2}{\text{SG of Contaminant}}}$$

Aquaplaning speed links to tyre pressure. For the same tyre, the higher the pressure, the smaller the footprint area in contact with the runway. Therefore, the greater pressure per unit area (square inches or square centimetres) punches the tyre's tread through the standing water to contact the runway surface.

The above description of dynamic aquaplaning refers to standing water. However, this can occur whenever the runway is contaminated with ice, sleet, slush, or snow. As these are lighter than water, the aquaplaning formula can be modified to account for their lower Specific Gravity by dividing the tyre pressure by the given SG of the contaminant first.

The formula for working out the onset of aquaplaning speed appears in the formulas above. The SG for water is 1. When calculating the aquaplaning speed for a wet or water contaminated runway, the formula dispenses with the SG value. This is how most people remember it and discuss it.

In the JAA exams, most questions on aquaplaning refer to standing water. However, if a question should arise which details that the contaminant is any of the above and gives the SG value, use the full formula as it will be used in real life instances.

The Aircraft Performance Notes cover the different depths of allowed contaminant.

VISCOUS AQUAPLANING

Viscous aquaplaning is a low-speed phenomenon and can cause complete loss of braking action on a wet runway, taxiway, or stand if there is any contamination such as:

- A film of oil or grease
- A layer of dust
- Rubber from previous touchdowns or skids
- A smooth runways surface

The contamination combines with the water and creates a more viscous mixture. Note that viscous aquaplaning can occur with a water depth less than dynamic aquaplaning, and skidding can occur at lower speeds, like taxiing to the gate during light rain, applying the brakes, and rolling over a patch of spilt oil.

ACTIONS TO MINIMISE AQUAPLANING ON LANDING

- Avoid landing in heavy precipitation. Allow time for the runway to drain.
- Know the aquaplaning speed of the main tyres and nose wheels.
- Use flaps to land at the lowest practical speed.
- Do not perform a long flare or allow the aircraft to drift in the flare.
- Touch down firmly to punch the tyres through any moisture and do not allow the aircraft to bounce, as the distance covered in the bounce and the bounce protection system reduces the available braking distance.
- Keep the aircraft centreline aligned with the runway centreline.
- Apply anti-skid braking steadily to full pedal deflection when automatic ground spoilers deploy and main wheel spin-up occurs.
- Deploy ground spoilers manually if automatic deployment does not occur.
- Apply maximum reverse thrust as soon as possible after main gear touchdown. This is when it is most effective.
- Get the nose of the aircraft down quickly. Do not attempt to hold the nose off for aerodynamic braking.
- Apply forward column pressure as soon as the nose wheel is on the runway to increase weight on the nose wheel for improved steering effectiveness.
- If the aircraft is in a skid, align the aircraft centreline with the runway centreline.

TYRE DAMAGE

Inspecting the aircraft on a pre-flight includes checking the tyres. Pilots qualified on type are able to check and top tyre pressures if the operator and the authority agree. Some of the common causes of tyre damage include:

DRY BRAKING FLAT SPOT

Diagram 8.39 Dry Braking Flat Spot

Locked or non-rotating wheels on dry runways, shown in diagram 8.39, cause flat spots. The skid can wear through the tyre cover and has a distinctive onion ring pattern.

WET BRAKING FLAT SPOTS

Diagram 8. 40 Wet Braking Flat Spots

These occur on wet/damp runway surfaces when the wheel aquaplanes dynamically or through a reverted rubber skid. Diagram 8.40 shows the result of a wet skid. The affected area of the tyre has the appearance of melted rubber.

BULGES AND DELAMINATIONS

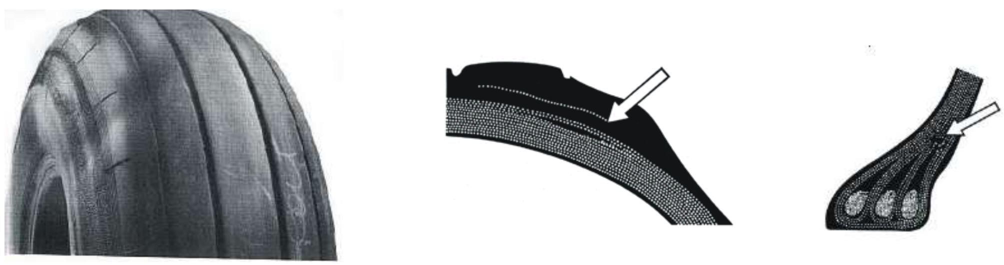


Diagram 8. 41 Tyre Bulges and Delaminations

Bulges on the treads or sidewalls of a tyre normally indicate that the internal components of the tyre are separating or delaminating. This can lead to a serious tyre failure. Do not puncture the blister or bulge as this can cause injury. The correct remedial action is to remove the tyre for further inspection.

The likely cause of tyre delamination is running on under-inflated tyres or taxiing at speed over long distances. Diagram 8.41 shows a tyre that has bulged, and two areas that have delaminated.

FOD

Diagram 8.42 FOD

Foreign object damage (FOD) describes items that should not have been there but were and have subsequently damaged an aircraft or its equipment. FOD also describes items that can present a hazard to an aircraft due to their location.

Diagram 8.42 shows a screw embedded in the central rib of a tyre. There is no way of knowing how long the item is, or whether it has penetrated the gas chamber.

Do not attempt to remove any item stuck in a tyre. It could be lethal. The correct course of action is to report it, so that the aircraft engineers can reduce the tyre pressure and replace the wheel. A screw has greater grip and sealing properties than a nail due to its thread.

BURST TYRES

Diagram 8.43 Burst Tyre

Tyres burst due to severe FOD damage, or from impact concussion, which accelerates the carcass fatigue on tyres that have been subjected to excessively fast taxiing over prolonged distance and/or under inflation.

As was illustrated by the Air France Concorde accident, tyres that burst at speed can project lumps of rubber with high kinetic energy into the wings' under surfaces. The wing structure must be strong enough to withstand the force of an exploding tyre for which the aircraft was designed and certified to use.

If the burst tyre was fitted on a twin axle or bogie, the companion tyre must be replaced if the burst occurred when the wheel was rolling.

CUT DAMAGE

Diagram 8.44 Severe Cuts

FOD often causes cuts to the tread and sidewalls of tyres.

The tyre requires removal if:

- Any cut penetrates the casing plies
- Cuts extend for more than 35 mm or 50% of any tread rib with a depth of 50% or more of the existing rib
- Any cut exposes the casing cords

TREAD CHUNKING

Diagram 8.45 Tread Chunking Cuts

Tread chunking is damage caused to the edge of a ribbed tread, where lumps or chunks are torn out of a rib as in diagram 8.45. This damage results from making tight radius turns at relatively high speeds, or operating from rough or unprepared runways. A tyre damaged in this way may remain in service if a chunk does not exceed $35mm^2$ of any rib, and the rib is not under cut.

CHEVRON CUTS

Diagram 8.46 Chevron Cuts

Chevron cuts occur as a result of wheel spin up when landing on a dry cross-cut runway. An affected tyre may remain in service if the cutting is less than the footprint area and does not undercut the tread.

HEAVY CROSS WIND LANDING

Diagram 8.47 Radial Scores

Tyres subjected to landing in strong crosswind conditions have score marks across the tread in a radial direction. If a tyre has these marks on the shoulder, it indicates that the aircraft has turned very tightly. A tyre subjected to this type of stress can suffer delamination later in service.

CROWN WEAR

The crown of the tyre is the area designed to withstand wear evenly under normal operating conditions. If the tyre is over inflated, the centre ribs of the crown wear more quickly than outer ribs.

SHOULDER WEAR

Diagram 8.48 Shoulder wear

Excessive shoulder wear on both shoulders of a tyre indicates severe tyre under inflation. Severe under inflation can also result in over deflection of the tyre's carcass and result in excessive heat build up in the shoulder regions. This can lead to delamination of the tyre in this region.

Wear on one shoulder indicates that the undercarriage leg is not correctly aligned when the gear is locked down.

SIDEWALL CRACKING

The sidewalls of a tyre may crack and craze during service. This can be due to:

- Weathering and ozone reacting with the rubber
- Over deflection of the tyre

Provided the cracks are confined to the sidewalls and no casing cords are visible, the tyre may continue in service.

EXCESSIVE BRAKE HEAT

If a tyre is subject to excessive brake heat (brake heat soaked), the heat is transferred via the wheel hub. This results in heat damage around the bead and sidewall. The tyre should be removed from service if:

- There are blisters in the region of the bead
- There is severe blueing of the bead and brittleness of the bead rubber
- There is the appearance of melted rubber in the bead area

CONTAMINATION OF TYRES

Hydraulic fluid, fuel, oil, and grease have a detrimental effect on tyre rubber. If allowed to remain in contact with a tyre, they cause the rubber to swell, becoming soft and spongy and resulting in a loss of strength. If these contaminants contact with the tyres, they must be removed with denatured alcohol or detergent and clean water.

FLAT SPOTS DUE TO NYLON SET

Aircraft tyres that include nylon in the carcass plies can develop flat spots if the aircraft is left stationary for any length of time. These flat spots are temporary and disappear during taxiing. This phenomenon is more noticeable in cold weather.

ACKNOWLEDGEMENT

We would like to thank and acknowledge:

For Diagram 9.1 Mr. Stuart Powney

INTRODUCTION

Diagram 9.1 Electra Taking Off

As seen from the previous chapters, hydraulic power enables pilots to remotely operate such services as the flying controls, flaps, undercarriage, and spoilers of large aircraft such as depicted in diagram 9.1.

This is achieved by passing a virtually incompressible fluid through a network of pipes to actuators that convert fluid pressure into mechanical force and motion. This chapter describes how hydraulic power is obtained, the different types of hydraulic systems, and the valves and components that make the system function.

BASIC PHYSICS OF HYDRAULICS

Before looking in some detail at how hydraulic power is generated, transmitted, and used, a review of basic hydraulic principles is required.

FLUIDS

All fluids are considered incompressible.

ENERGY

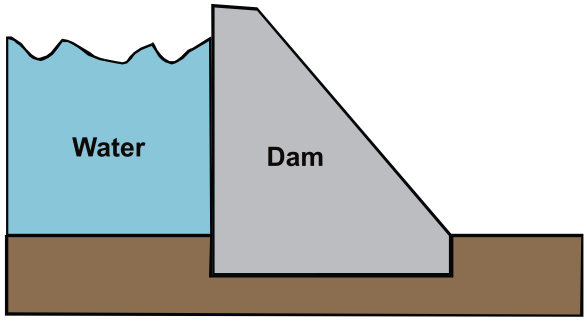


Diagram 9.2 Potential Energy

Potential energy
Water in a reservoir, as per diagram 9.2, has potential energy through static pressure.

Kinetic energy
Water flowing from the reservoir has kinetic energy through dynamic pressure.

STATIC PRESSURE

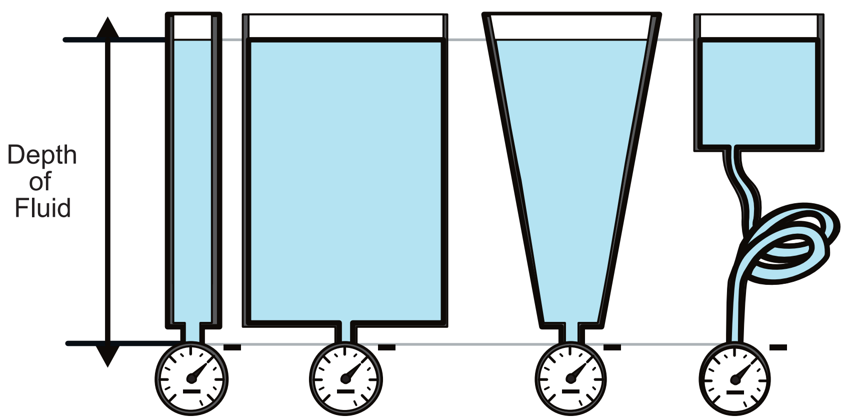


Diagram 9.3 Static Pressure Acts in a Vertical Column

The depth of fluid creates static pressure. In diagram 9.3, four containers of various shapes that have an equal depth of fluid register the same static pressure on the gauges at the base. This is due to the pressure acting as a vertical column.

PASCAL'S LAW

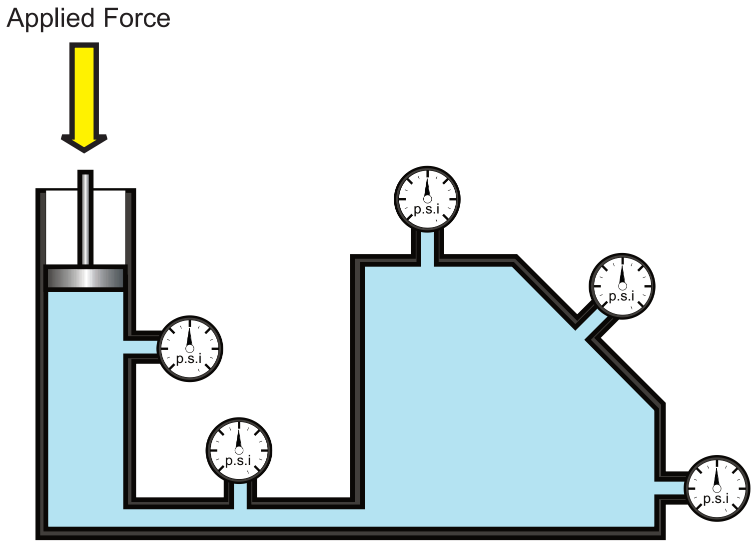


Diagram 9.4 Pascal's Law

Pascal's Law states, *"In an enclosed container, the pressure is equal throughout the fluid and acts equally in all directions and at right angles to the container's walls."* Diagram 9.4 shows Pascal's Law in action. All the gauges show the same reading.

RELATIONSHIP BETWEEN FORCE, AREA, AND PRESSURE

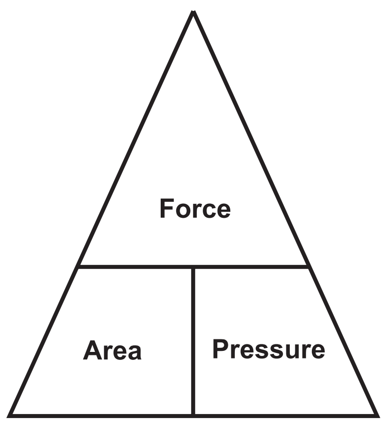


Diagram 9.5 Relationship between Force, Area, and Pressure

Force, in hydraulics, is the energy that is applied to act on the fluid, or the energy that is exerted when a fluid acts on a piston. Therefore, force is considered to be either an input or an output.

Area is the square surface area of a piston that is either used to transmit the input force into a fluid, or transmit the output force from a fluid.

Pressure is the name given to the energy (force) raised in the fluid. This is force/area. This relationship is normally expressed as: **F = A x P**

By altering the formula **A = F ÷ P**

Each part can be found **P = F ÷ A**

Use the triangle in diagram 9.5 as an aid to remember it.

The static pressure of a fluid within in a container increases when it is subjected to an external force. This is due to the individual molecules acting against each other as they resist the compression. As fluids are considered incompressible, all hydraulic systems raise fluid pressure by moving a volume of fluid and forcing it into a restrictive pipe. Hydraulic pressure cannot be raised without resistance.

THE SI SYSTEM — SYSTEM INTERNATIONALE

Force is expressed in Newtons	N
Area in square centimetres	m^2
Pressure in Newtons per sq cm	N/m^2, referred to as a Pascal (Pa)

Newtons — The ICAO definition of a Newton is the force which when applied to a body having a mass of one kilogram gives it an acceleration of 1 metre per second squared (m/s^2).

Acceleration — In the SI system it is given as the gravitational constant of 9.81 m/s^2.

Mass — In the SI system, the unit of mass is the kilogram.

Force — As force is equal to mass x acceleration, 1 kg mass x 9.81 m/s^2 = 9.81 Newtons.

Therefore, a 100 kg mass imparts a force of 981 Newtons.

THE IMPERIAL SYSTEM

While SI units are predominantly used in Europe, the UK and USA still use the Imperial system.

In the Imperial System, the measurements for force and mass are in pounds. Measurement of Area is in square inches and pressure in pounds per square inch as follows:

Force:	pounds
Area:	square inches
Pressure:	psi

For an input force of 100 lb acting on a piston of 10 square inches, a pressure of 10 psi is raised in the fluid. To express large pressures, the unit of **bar** is used, **1bar = 14.5 psi.**

UNIT CONVERSION FACTORS – TO 3 PLACES DECIMAL

Pressure		
Imperial	SI	Metric
psi	Pa	kg/cm^2
1 lb/in^2 (psi)	6894.757 Pa	0.070 kg/cm^2
1 bar (14.504 psi)	100.0 kPa	1.020 kg/cm^2

TRANSMISSION OF POWER

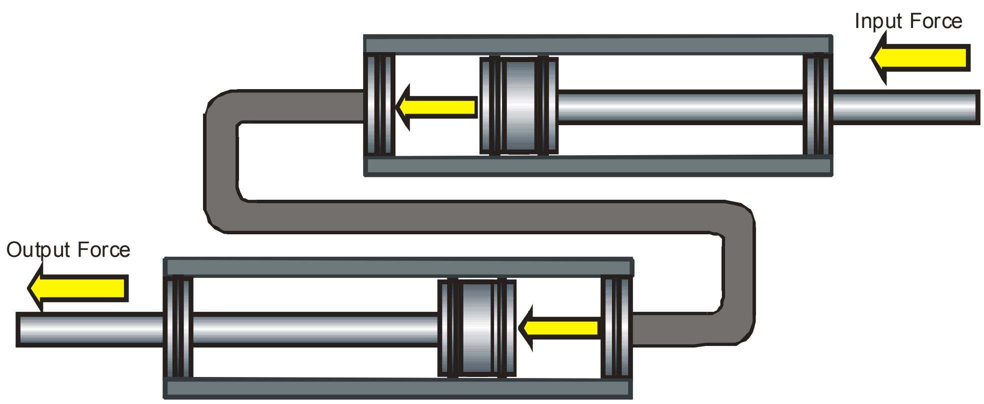


Diagram 9.6 Transmission of Power by Fluid

If two actuators with equal sized pistons are interconnected with a pipe, as per diagram 9.6, when an input force is applied to one piston, it results in an equal output force from the other piston. This system only allows the transmission of the input force and is, therefore, of limited use.

MULTIPLICATION OF FORCE

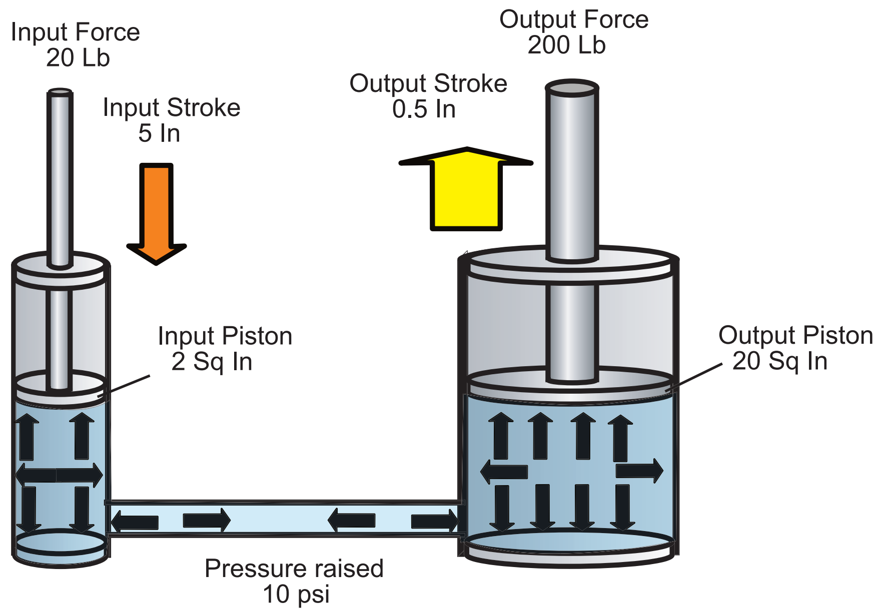


Diagram 9.7 Multiplication of Force using a Bramah's Press

Joseph Bramah (1749-1814), an engineer, worked out that if the input force was applied to a small piston and the pressure raised was able to act on the surface of a large piston, the output force would increase. This led to his invention called the Bramah's Press, depicted in diagram 9.7.

For hydraulic systems to be viable in aircraft, the output force must be greater than the required input force. This is achieved by linking a small piston (used for the input force) to a large piston (for the output force). Diagram 9.7 shows the multiplication of force using 20-pound force applied to a 2 square inch input piston and the resulting 200 lb output force from the 20 square inch output piston.
While **F = P x A**, movement of an output piston requires fluid flow.

The input piston's movement (its stroke) displaces a volume of fluid. This is transferred into the chamber beneath the output piston, where due to the larger cross sectional area, it spreads out, resulting in only a small movement of the output piston.

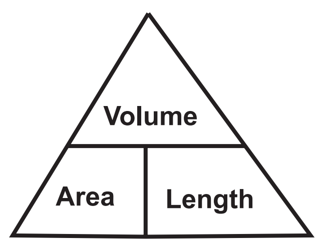


Diagram 9.8 Relationship between Volume, Area, and Length

1. There are two formulae to calculate the stroke of an output piston for a given stroke of an input piston.

   Volume = Length x Area: V = L x A
   A = V ÷ L
   L = V ÷ A

   Find the volume of fluid displaced by the stroke of the input piston (V = L x A). Then find the length of stroke of the output piston (L = V ÷ A).

   From diagram 9.7, the Bramah's Press, the volume moved by the input piston is 10 cubic inches = 5" x 2 sq in. The output piston moves 0.5" = 10 cubic inches ÷ 20 sq in.

2. Input force x input distance = output force x output distance.
For this formula, a cross multiplication is required.

(Using the same information from diagram 9.7)

Input Force 20 lb		Output Force 200 lb
x	=	x
5 in		Output distance
100 lb in		?

100 lb in	=	Output distance
Output Force 200 lb		

100 lb in ÷ 200 lb	=	0.5” output distance

Note: In some instances, candidates have to work out forces (input or output) before applying the above formula.

PASSIVE HYDRAULIC SYSTEMS

The Bramah’s Press (diagram 9.7) is an example of a passive hydraulic system, where pressure is only generated when the input piston is moved by an external force. Another example is a basic car or light aircraft braking system, where the brake pressure is raised by applying foot pressure on the brake pedal(s).

FLUID PRESSURE INTO MECHANICAL FORCE AND MOVEMENT

To convert the fluid pressure and flow, linear and rotary actuators are used. Linear actuators or actuators can also be termed jacks. Rotary actuators, termed hydraulic motors, are discussed later.

LINEAR ACTUATORS

There are three types of linear actuators to consider:

- Single-acting
- Double-acting unbalanced
- Double-acting balanced

SINGLE-ACTING ACTUATOR

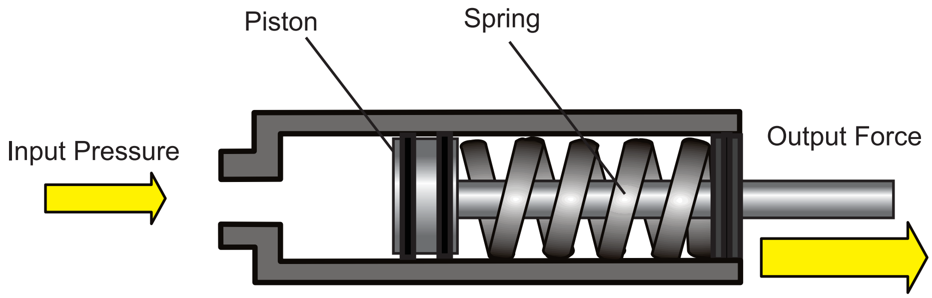


Diagram 9.9 Single-Acting Actuator

These actuators work hydraulically in one direction only. They require a mechanical force to return them to the original position. In diagram 9.9, the single-acting actuator makes use of a spring force.

DOUBLE-ACTING UNBALANCED ACTUATOR

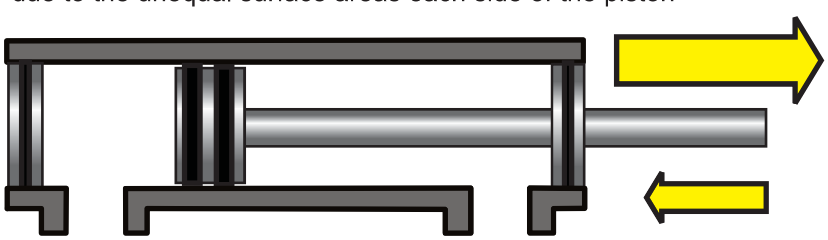


Diagram 9.10 Double-Acting Unbalanced Actuator

Double-acting unbalanced or uncompensated actuators operate hydraulically in both directions. As these actuators only have one ram attached to the piston, the surface area of each side of the piston varies. The variation is equal to the area of the ram. This results in the actuator producing a larger force on extension (ram extending) than on retraction (ram retracting) for a given pressure, see diagram 9.10.

In some exam questions, the candidate is asked to find the pressure required to achieve an equal force on both extension and retraction. Other questions provide an equal pressure to be simultaneously applied to both sides of the piston from a common supply and ask for the resulting action.

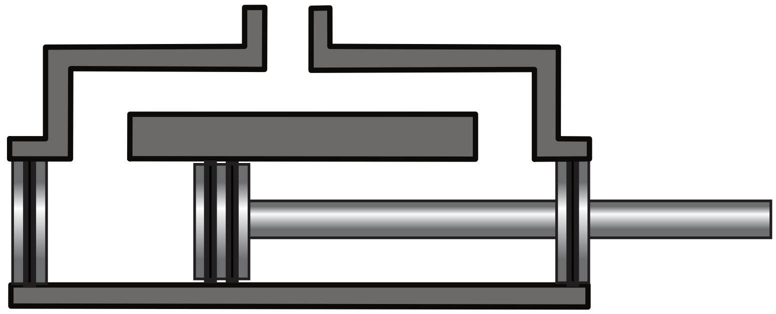


Diagram 9.11 Exam Style Double-Acting Actuator

Diagram 9.11 is a schematic representation of an uncompensated actuator for use with the above type question.

DOUBLE-ACTING BALANCED ACTUATOR

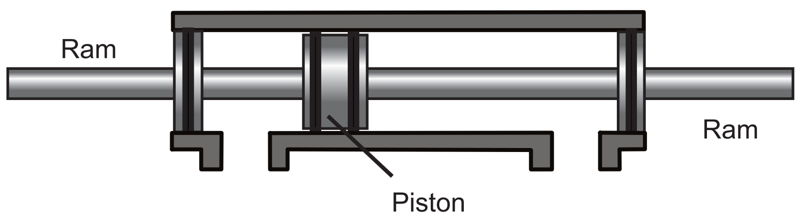


Diagram 9.12 Double-Acting Balanced Actuator

Double-acting balanced or compensating actuators have a ram of equal area attached to each side of the piston, as per diagram 9.12. For an equal pressure, they generate an equal force on both extension and retraction. This type of actuator can be used to move a loop of cable, operate two items in opposite directions simultaneously, or have a service operated by one ram only.

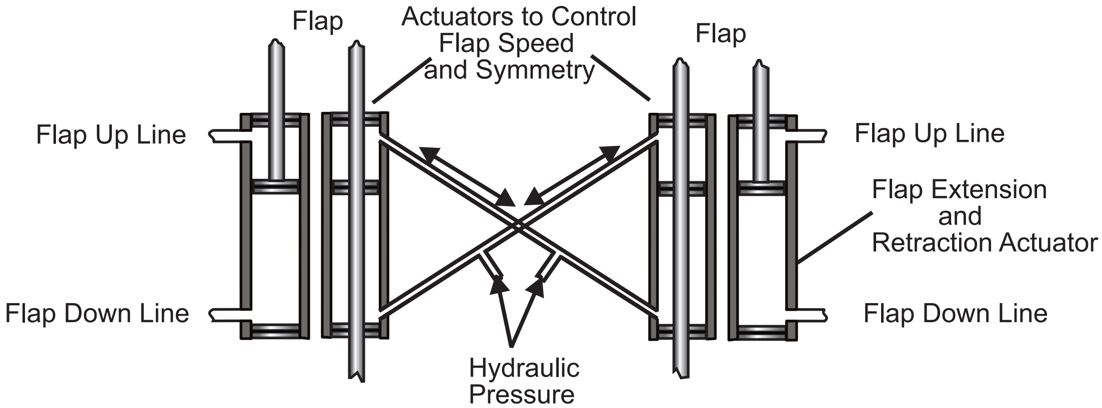


Diagram 9.13 Double-Acting Actuators Used to Control Speed and Symmetry

Diagram 6.21 of chapter 6 showed a further function of double-acting balanced actuators. A reduced version appears above as diagram 9.13, where they are used to synchronise the movement of two separate actuators. In this case, the trailing edge flaps are operated by two separate, double-acting unbalanced actuators supplied by a common power source. When the pilot selects flaps down, the control valve allows pressure/flow into these actuators.

As the flaps move downward, they pull the pistons of the double-acting balanced actuators with them. This action displaces fluid from one side of the actuator, which is linked to the opposite side of the other balanced actuator. The rate at which the fluid can leave the chamber of one actuator is controlled by the available volume in the other actuator. To get movement, both actuators have to move at the same time. This synchronises the flaps' operation.

PORTS

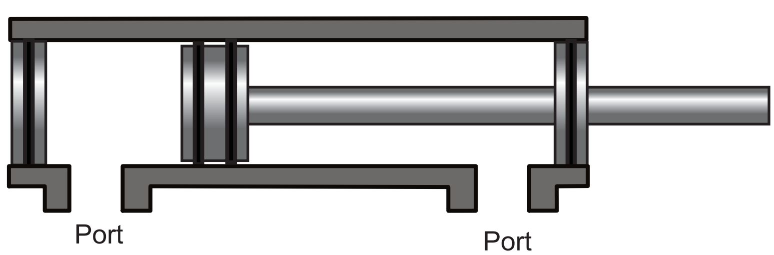


Diagram 9.14 Ports in a Double-Acting Actuator

The entry into and exit from a hydraulic component is commonly referred to as a port. Thus, a single-acting actuator is said to have one port and a double-acting actuator is said to have two ports, as per diagram 9.14.

SELECTORS

To control the operation of the hydraulic actuators, selector valves are used. These control the flow into and out of the actuators. These selector valves are fitted throughout the aircraft and therefore have to be operated remotely, either mechanically or electrically. Selectors come in two basic styles, rotary and linear.

Rotary Selectors come in three types:

- Two-port
- Four-port
- Open-centred

These are covered in the open-centred system explanation.

TWO-PORT ROTARY SELECTORS

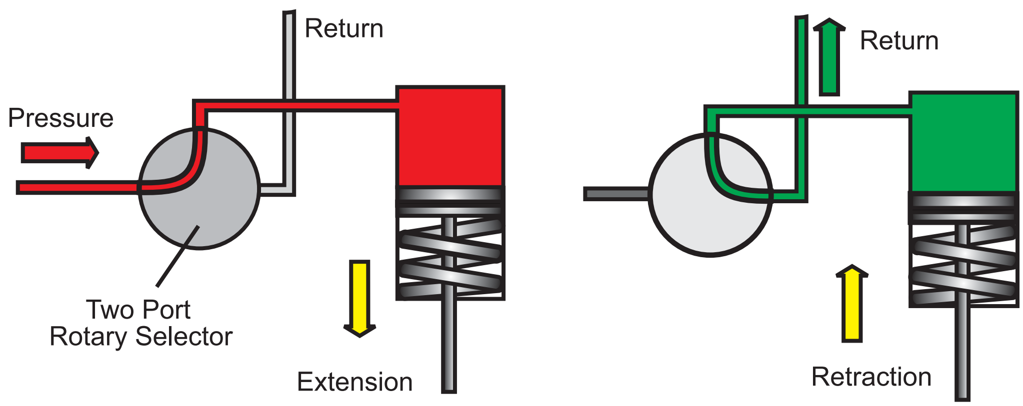


Diagram 9.15 Operation of a Two-Port Rotary Selector

A two port rotary selector, as shown in diagram 9.15, has one fluid path and is for use with a single-acting actuator.

FOUR-PORT ROTARY SELECTORS

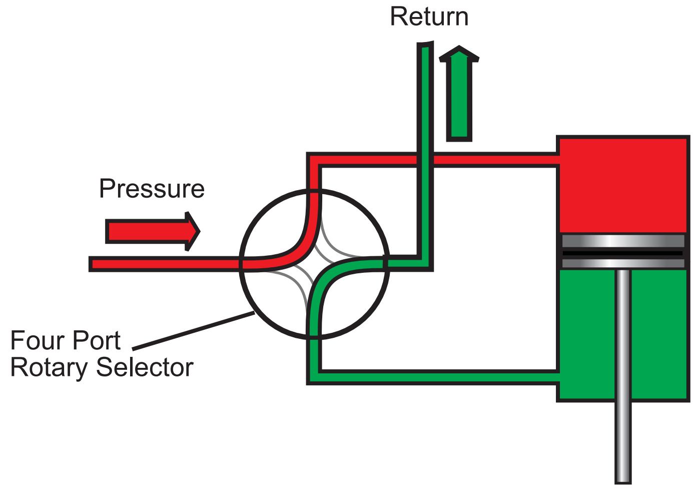


Diagram 9.16 Four-Port Rotary Selector

A four-port rotary selector has two fluid paths and is for use with a double-acting actuator. A four-port selector allows one path for fluid flow into the actuator and the other path for the return flow to the reservoir. Diagram 9.16 shows a four port rotary selector selected for ram extension. For ram retraction, the selector rotates so that the pressure and flow are directed into the lower chamber and the upper chamber is aligned with the return line. This is shown by the fainter lines in the selector.

LINEAR OR SPOOL VALVE SELECTORS

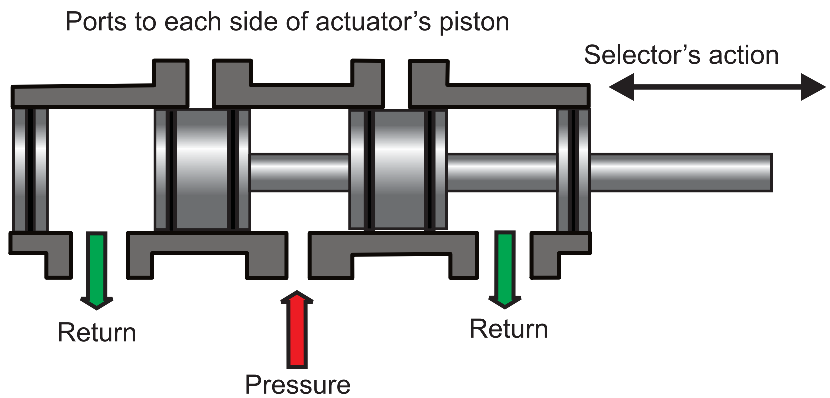


Diagram 9.17 Linear Selector for a Double-Acting Actuator

Another form of selector is the linear or spool valve. A schematic diagram of a spool valve appears in diagram 9.17. These valves operate using a linear motion. As pressure is supplied in between the two pistons, which are of equal area, the pressure does not cause the selector to move. To operate the selector, an external input is required to bias the spool in one direction or the other. This can be a mechanical actuation via a rod or cable.

For systems where the physical operation of the selector is not practical, the spool can be biased using an electro magnet. By placing a coil around a soft iron core that is attached to the spool, the whole valve is effectively turned into a solenoid and can be operated by moving a switch. The spool is centralised by springs when the electrical input is removed.

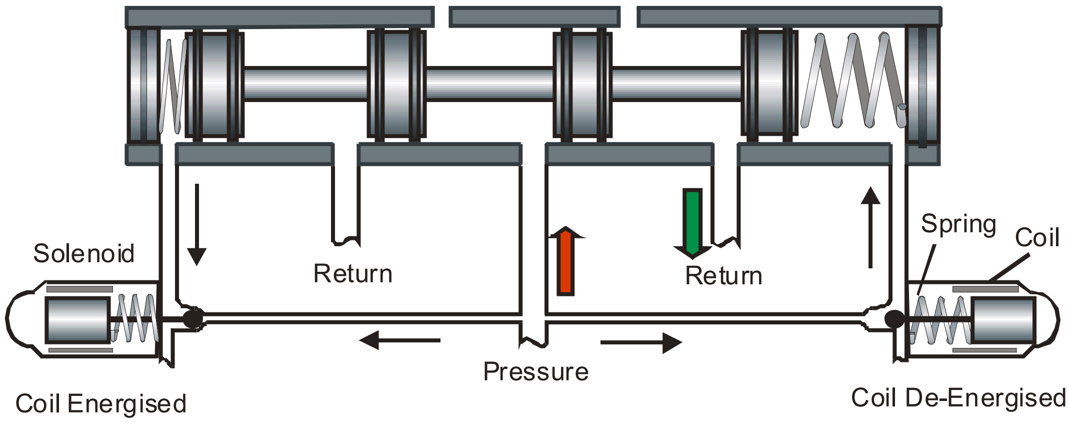


Diagram 9.18 Solenoid Operated Linear Selector

The most common method of operating these valves is to use solenoids to control the flow of hydraulic fluid from the pressure supply, which is ported to the pistons mounted at the ends of the selector. Diagram 9.18 shows a linear selector biased to the left. The solenoid on the left is energised, moving the valve to the right and blocking the supply being ported into the left chamber, which is opened to return.

The solenoid on the right is de-energised. The spring opens the valve allowing pressure to be ported onto the end of the right-hand spool to bias the spool to the left and operate a service using system pressure. When the actuator has completed its movement, electrical power is removed from the left solenoid, allowing fluid pressure back into the left chamber.

As the pressure acting on each piston is equal as are the pistons themselves, the centring springs mounted in each chamber act to centralise the valve, blocking off the ports to the actuator.

HYDRAULIC LOCK

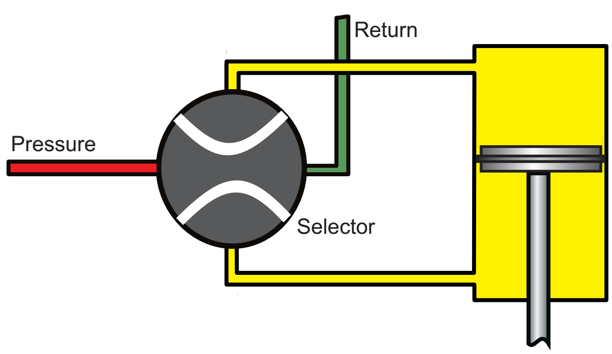
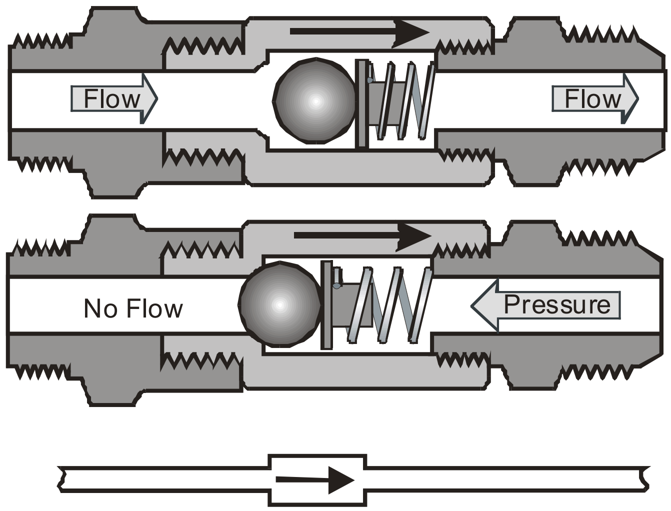


Diagram 9.19 Hydraulic Lock

If a selector, either rotary or spool valve, is placed so that there is neither flow into nor out of an actuator, the actuator is considered **hydraulically locked**. The ram is unable to move in either direction unless an input force exceeds 32 tons per square inch. The actuator and ram effectively become a solid rod. This allows the device to remain stationary in a set position.

NON-RETURN VALVES – NRV



Schematic Symbol for an NRV

Diagram 9.20 Operation of Typical Ball Type NRVs

Non-return valves are classed as flow control valves. They are incorporated into systems to prevent backflow and subsequent loss of pressure. A typical non-return valve is shown above. In this valve, a ball is held against a valve seat by the force of a spring. Fluid flow forces the ball from its seat, allowing the fluid to pass. When the flow stops, the ball re-seats and traps the fluid that is downstream of the valve. To prevent an NRV from being incorrectly fitted, an arrow is stamped into the body showing the direction of flow, as per diagram 9.20.

In schematic diagrams, an arrow in a box normally represents an NRV. The arrow indicates the direction of flow, as per diagram 9.20.

HAND PUMPS

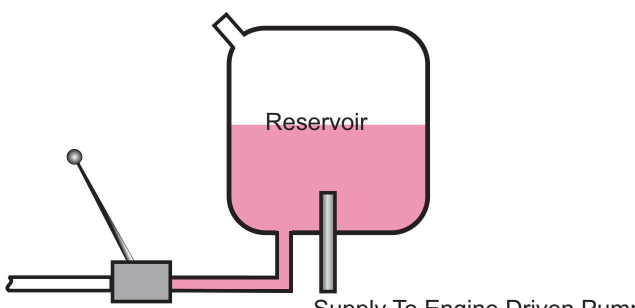


Diagram 9.21 Hand Pump Supply

Hand pumps are used for maintenance activities such as raising brake pressure without starting the engines, opening cargo doors, and can be used for emergency power in flight. They always take their supply of fluid from the bottom of the reservoir.

Two types of hand pump are used, single-acting and double-acting. Single-acting pumps only move fluid on one stroke of the handle, whereas double-acting pumps move fluid on both strokes of the handle.

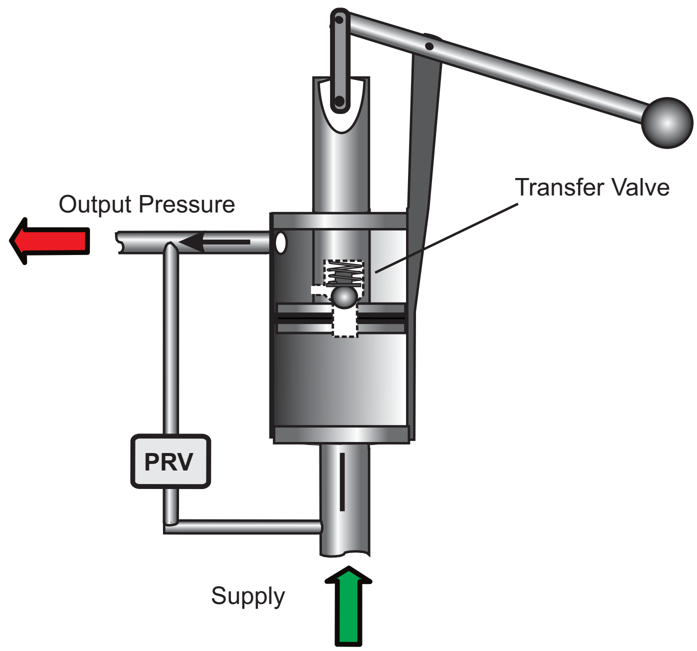


Diagram 9.22 Double-Acting Hand Pump

Diagram 9.22 shows a typical double-acting hand pump. A downward action of the handle draws fluid through the inlet non-return valve as the piston moves up. Upward movement of the handle forces the piston against the fluid, and the fluid pressure opens the transfer valve. As the piston moves down, fluid flows from the bottom chamber to the top chamber via the transfer valve.

Due to the ram displacement in the upper chamber, a volume of fluid flows out via the outlet non-return valve. On the next down stroke, the process repeats and the up-going piston ejects the fluid that has not been displaced by the ram. If the hand pump produces too much pressure, a relief valve opens allowing the excess fluid to return to the inlet side.

ACTIVE HYDRAULIC SYSTEMS

The next section shows how hydraulic pressure is raised by making use of mechanically-driven pumps. These systems are termed active, as they have a head of pressure available to operate sub systems.

PRESSURE GENERATION

Pressure in a hydraulic system is raised by drawing fluid from a system reservoir and forcing it into a restrictive pipe where the fluid's viscosity gives it resistance against flowing through pipes. It is this resistance that is measured and termed pressure.

The component that moves the fluid is referred to as a pump. As forcing fluid into a restrictive pipe raises pressure, the supply pipe to a pump is always larger than the pressure delivery pipe from the pump.

The most available and reliable source of power to drive a pump is the aircraft's engine. Any pump that is attached to the engine/gearbox is referred to as an engine-driven pump or EDP.

ELECTRICALLY OPERATED PUMPS

Electrically operated pumps are used in some light aircraft as the main power source. For air transport aircraft, they can be used as back-up or booster pumps, but not as the main source of hydraulic power, as the loss of the aircraft's electrical generators would also result in the loss of the hydraulic system.

Hydraulic systems fall into two main categories:

- Low pressure up to 2000 psi
- High pressure from 2000 to 4500 psi

The hydraulic pumps that are used to power these systems can be divided into two main categories:

- Constant volume (constant displacement)
- Constant pressure (variable displacement)

The constant volume type pumps can be sub divided into:

- Low-pressure pumps
- High-pressure pumps

LOW-PRESSURE SYSTEM PUMPS

There are two main types of low-pressure pumps used in aircraft hydraulic systems:

- Spur gear pumps
- Vane pumps (centrifugal)

The simplest and lightest of these is the spur gear pump. This type of pump is commonly used in aero-engine lubrication systems. The information given here as to its operation is also valid for the engine subjects.

Spur Gear Pumps

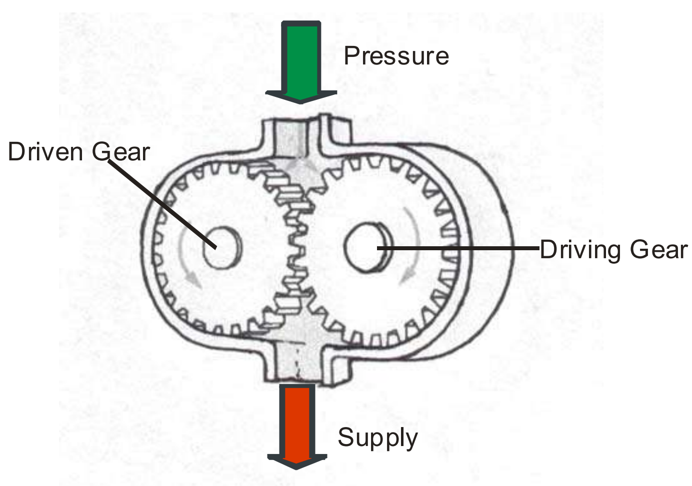


Diagram 9.23 Spur Gear Pump

Spur gear pumps are classed as constant volume pumps. They have the advantage of moving large volumes of fluid rapidly. The pressure they raise is produced by forcing fluid into a restrictive outlet pipe. This pressure also acts back (Pascal's Law) against the gears, which limits their use to low-pressure systems, normally up to 2000 psi.

Refer to diagram 9.23: The pump consists of two intermeshing gears located in one housing. The aircraft's engine or an electric motor rotates the driving gear, which in turn rotates the driven gear. Fluid is transported from the inlet port to the outlet port between the teeth and the housing. For one revolution, the pump moves a constant volume of fluid.

Fluid trapped between the intermeshing gear teeth would form a hydraulic lock and stall the pump. As this would result in damage to the drive and loss of the system, a port incorporating a pressure relief valve is located in the housing at the point where the teeth mesh. This relieves the fluid back to the inlet side of the pump and prevents a hydraulic lock from forming.

Vane Type Pumps

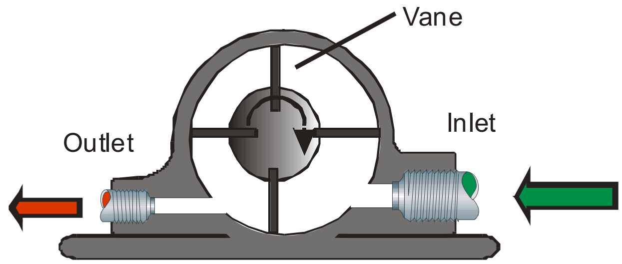


Diagram 9.24 Vane Type Low Pressure Pump

Vane type pumps are also considered constant volume pumps. They also move large volumes of fluid rapidly and raise pressure by forcing this volume into a restrictive outlet pipe. This pressure also acts back (Pascal's Law) against the vanes, which limits their use to low-pressure systems.

Diagram 9.24 shows a pump consisting of a cylindrical, offset shaft within a circular housing. The shaft is grooved to accept the vanes. These are sprung loaded to move out radially, following the contour of the housing as the shaft rotates. This action draws the fluid in from the inlet side and expels it into the outlet pipe. For one rotation of the shaft, the pump moves a set volume. A pressure relief valve can be incorporated to relieve excess outlet pressure back to the inlet side to prevent damage to the pump.

ENGINE AS THE POWER SOURCE

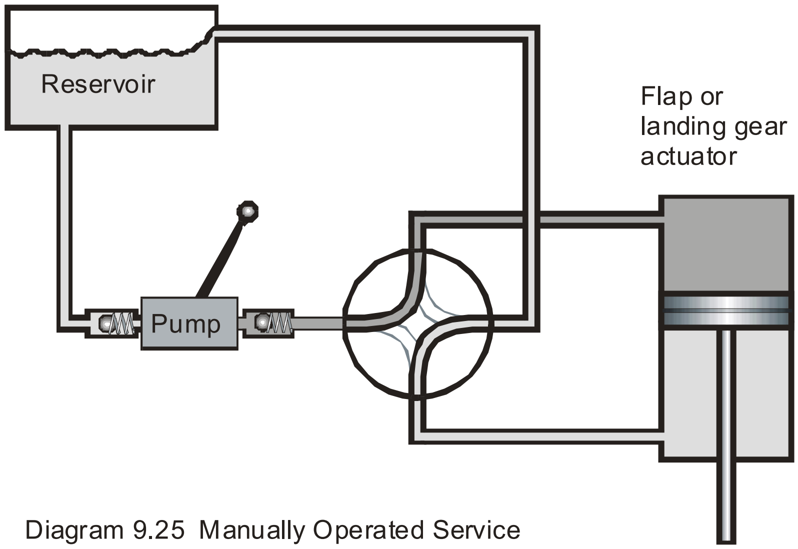


Diagram 9.25 Manually Operated Service

The early and basic aircraft hydraulic systems, as shown in diagram 9.25, required the pilot or co-pilot to put in a lot of effort with a hand pump to raise and lower the landing gear and flaps. By replacing the basic hand pump with a mechanical pump, both pilots could breathe a sigh of relief. These early systems, which were state of the art in the years between the First and Second World Wars, have long been surpassed. However, the basic principles by which these systems work are still in operation on aircraft today.

As EDPs are direct drive, (failure of a clutch would result in the loss of hydraulic power) they would start pumping fluid as soon as the engine is swung over on its starter until the engine stops.

This would:

- Continually increase the pressure within the system
- Raise the fluid’s temperature due to the increase in pressure
- Use engine power that could be required for the propeller
- Cause unnecessary wear in the pump

To overcome this, the open-centred system was devised.

OPEN-CENTRED SYSTEM USING OPEN-CENTRED SELECTORS

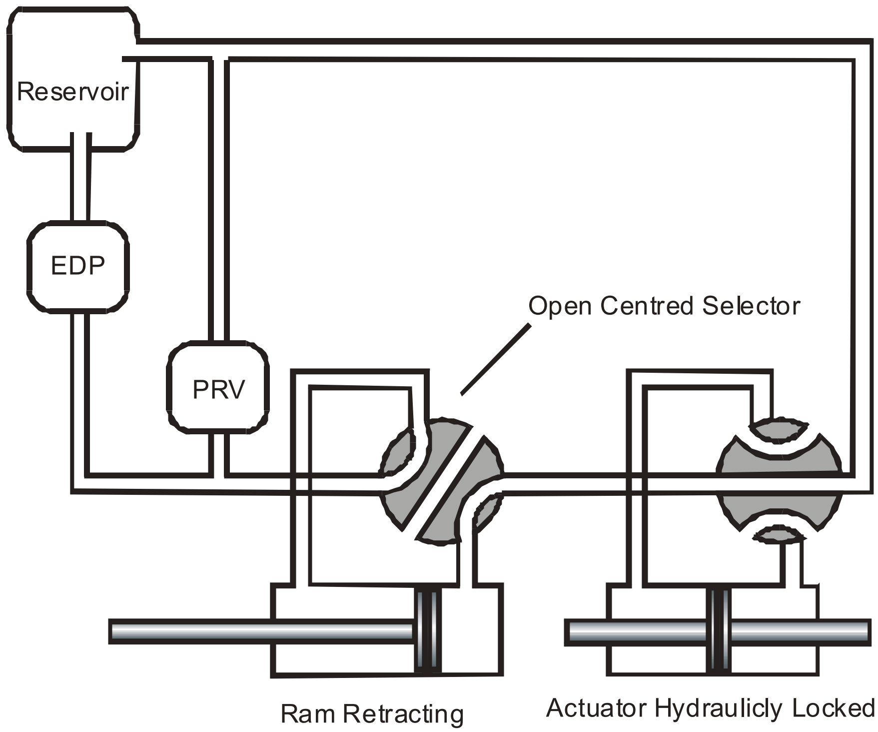


Diagram 9.26 Open Centred System in Operation

The open-centred system is a low-pressure system fitted to older light aircraft and used to operate the flaps, undercarriage, and air brakes. Only one actuator could be operated at any time, and the system was unable to operate flying controls. Open-centred selectors have an extra fluid path passing through their centres when the system is idling, as shown by the right-hand selector in diagram 9.26.

The system utilises a low-pressure pump, which pumps fluid continuously around the system through the centre of the open-centred selectors. The system pressure remains low until a selection is made. This off-loads the pump and hence, the engine, while the system is not required. Selecting a service, as shown by the left-hand selector in diagram 9.26, diverts the entire pump's output into the selected actuator.

Only the return flow passes through any selectors downstream. When the pressure reaches a set level, the selector automatically returns to the open position, which hydraulically locks the actuator. A pressure relief valve is fitted downstream of the pump to protect the system should the selector fail to return to the open position.

PRESSURE RELIEF VALVES — PRV

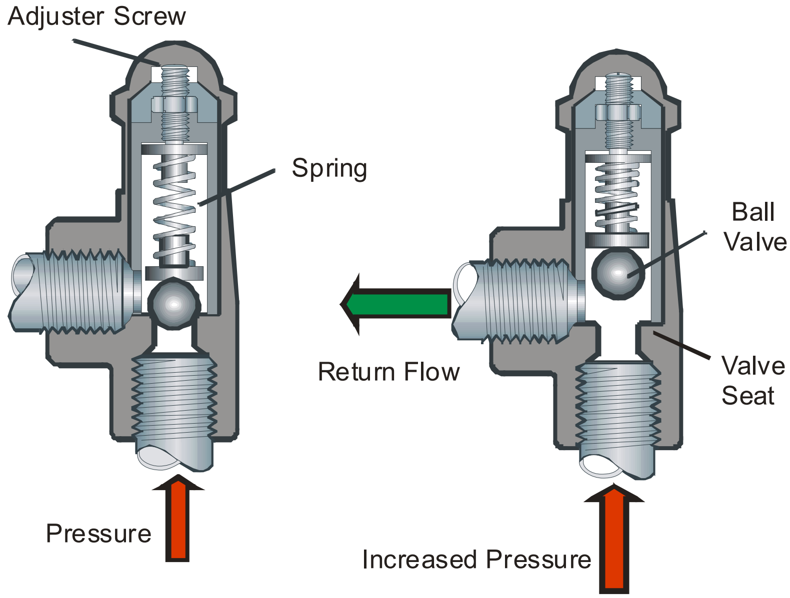


Diagram 9.27 Pressure Relief valve in Operation

Pressure relief valves protect hydraulic systems from over pressurisation. The ball type PRV is a common design, as shown in diagram 9.27. It consists of a ball that is held onto a valve seat by a spring. As only a very small area of the ball is subjected to fluid pressure, a spring of relatively low strength can exert sufficient force to keep the ball seated when the system is at normal working pressure.

When the fluid pressure increases above working pressure, the force it exerts also increases. At a set value, the fluid pressure lifts the ball off the valve seat. This is called the cracking pressure. Some fluid now flows back to the reservoir via a return line. If the pressure continues to increase, the ball lifts higher. If the PRV is able to relieve all the output of the pump, it is called a Full Flow Relief Valve (FFRV). Operation of a relief valve holds the hydraulic system at a higher pressure.

It also results in an increase in fluid temperature. When the pressure decreases below the cracking pressure, the spring re-seats the ball. The value at which this happens is called the re-seating pressure. If the spring tension is adjusted, the values for cracking, full flow, and re-seating pressures are also adjusted.

When PRVs operate, it normally means that the normal pressure regulation system has failed to function correctly. A higher pressure reading on the supply gauge and a higher temperature reading would indicate this.

Order of pressures for PRV from high to low are:

- Full Flow
- Cracking
- Re-Seat
- Normal

OPEN-CENTRED SYSTEM USING AN OFF-LOADING VALVE OR UNLOADING VALVE

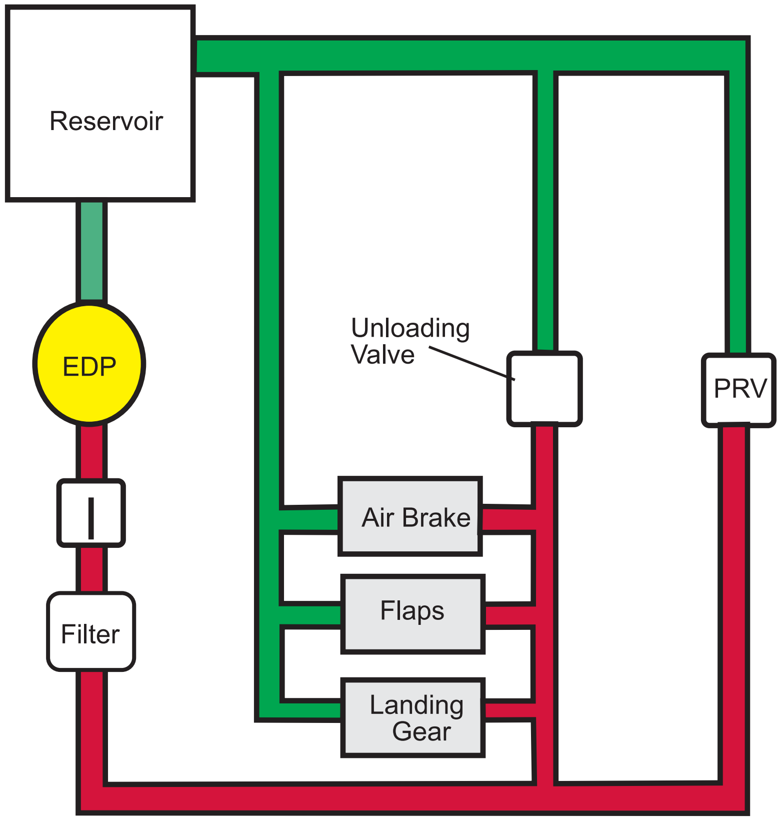


Diagram 9.28 Unloading Valve Circuit

The more modern version of the open-centred system, as shown above in diagram 9.28, is fitted to such aircraft as the Cessna Citation. In this system, the reservoir draws fluid by a low-pressure engine-driven pump and passes through a system bypass valve (off loading valve).

When a pilot makes a selection, the bypass valve closes, and the full flow is diverted to the appropriate service. However, as seen from the diagram, the services are in parallel with each other. This allows more than one service to be operated simultaneously.

Should the system bypass valve fail to open, the system does not generate any hydraulic pressure. If the valve fails closed, the system relief valve, an FFRV, protects the system but maintains a higher pressure. A filter is fitted down stream of the EDP, as this component is the most likely source of debris due to wear. A discussion of filters occurs later in these notes.

CLOSED-CENTRED SYSTEM

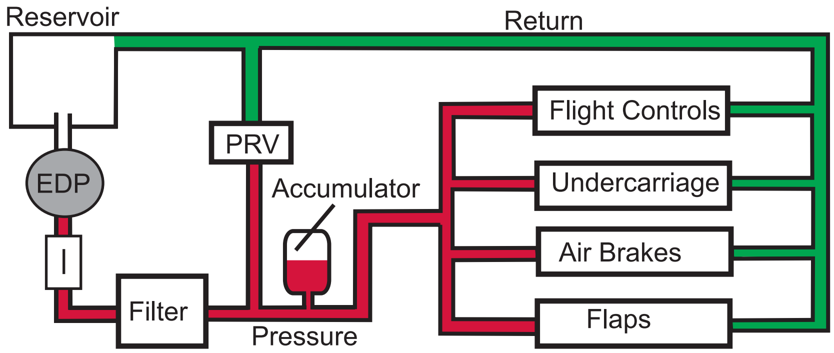


Diagram 9.29 Schematic Diagram of a Closed Centred System

For large transport aircraft, there is a requirement for power assistance to help the pilot move the control surfaces. These systems require a constant head of pressure and are termed **closed systems**, as diagram 9.29 shows. These allow the operation of multiple services at one time, give instant response to control inputs, and have a greater output force due to higher working pressures. Further requirements for large aircraft hydraulic systems are the ability to store energy for use in emergencies, produce power in the event of normal pump failure, and prioritise fluid flow to the primary services. These are the primary flying controls and the wheel brakes

PRESSURE GENERATION AND FLUID SUPPLY

Aircraft hydraulic systems range from low-pressure systems of 1000 psi, to very high pressures of 4500 psi, with the average standard pressure being 3000 psi. The advantages of using high-pressure hydraulic systems are:

- For a given output force, the size of the actuators can be reduced, with the resulting saving in weight and fluid volume required for their operation.
- Smaller actuators do not require such large volumes of fluid to operate them.
- Smaller diameter pipes can be used. These are easier to install throughout the structure of the aircraft.
- Smaller diameter pipes have reduced internal surface areas on which the pressure can act, therefore less force is acting on the pipe to expand/rupture it.
- The reduction in fluid volume, actuator, and pipe size saves weight.

The diameters of pipes used to return fluid from components to the reservoir are larger than those that supply pressure to the component. These larger bore pipes ensure that there is less resistance to flow, and subsequent back pressure, and are referred to as either the suction or return line.

CLOSED SYSTEMS

Closed systems (active systems) require a means of pressure regulation, which is achieved by one of two methods:

- Older systems make use of a constant volume high-pressure pump and pressure regulator sometimes called an automatic cut-out valve (ACOV).
- More modern systems use a variable volume pump which self regulates and removes the need for the ACOV.

HIGH PRESSURE ENGINE-DRIVEN PUMPS

All pumps driven directly from the engine or ancillary gearbox are referred to as Engine Driven Pumps (EDP). The high-pressure pumps used in the closed system are piston types. These normally revolve at about 4500 rpm. They move a smaller volume of fluid than the spur gear and vane type pumps but can operate at far higher pressures due to design.

FIXED VOLUME OR CONSTANT DISPLACEMENT PUMP

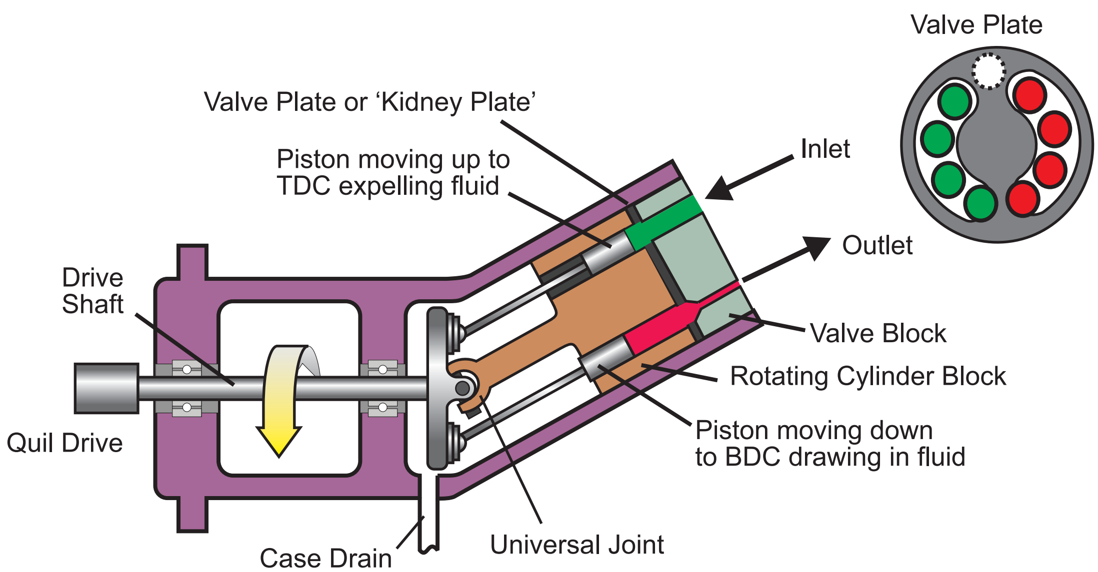


Diagram 9.30 Cross Section of a Constant Volume EDP

The fixed volume pump moves a constant volume of fluid into the system, regardless of the hydraulic pressure, with every revolution. They must be used with a pressure-regulating device to off load the system when the working pressure has been achieved.

Referring to diagram 9.30, fluid is transferred from the inlet side to the pressure side by means of reciprocating pistons. In this style of pump, the reciprocal action is achieved by angling the body in relation to the drive shaft. The drive is taken from the engine's ancillary gearbox via a quill drive. This is designed to be a weak point. In the event that the pump jams, the drive breaks, preventing more serious damage to the gearbox. The drive shaft terminates in a flat plate.

Linked by a universal joint to the centre of this plate is the cylinder block, which rotates at the same speed as the drive shaft. The cylinder block includes either seven or nine piston bores. The pistons operate through equal length rods, which terminate in shoes. As the drive shaft/cylinder block rotates, the cranked angle of the body creates the reciprocating action of the pistons. Each piston in turn travels from bottom dead centre (BDC) to top dead centre (TDC) and back to bottom dead centre in one revolution.

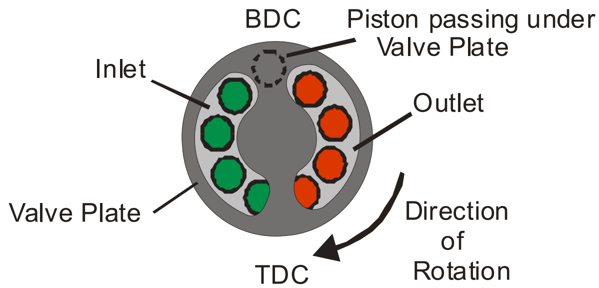


Diagram 9.31 Valve Plate

As the fluid is not compressible, a valve plate is located between the cylinder block and the valve block, as in diagram 9.31. Sometimes referred to as a kidney plate, this has two kidney-shaped ports.

The supply from the reservoir is linked to one port on the side of the down-going pistons. The opposite port is linked to the pressure supply (restrictive pipe) for the up-going pistons. When the pump rotates, fluid is drawn into the down-going pistons and transferred under the bridge formed by the valve plate at BDC. The up-going pistons act to dispel the fluid. System pressure builds in this fluid as soon as it passes BDC.

These pumps generate a lot of heat and require constant cooling and lubrication. This is achieved by allowing a bleed of fluid to spray around the inside of the pump's moving parts. This fluid is then removed via the case drain (see diagram 9.30) where it is either passed through a dedicated filter (case drain filter), or returned to the reservoir upstream of a low-pressure (return) filter.

FILTRATION

As the pump (either high or low pressure) is the most likely source of particle contaminants, it is common practice to place a filter unit immediately after the pump. However, some systems place a low-pressure filter in the return line just before the fluid enters the reservoir. Systems exist where both high-pressure and low-pressure filters are incorporated.

HIGH-PRESSURE FILTER

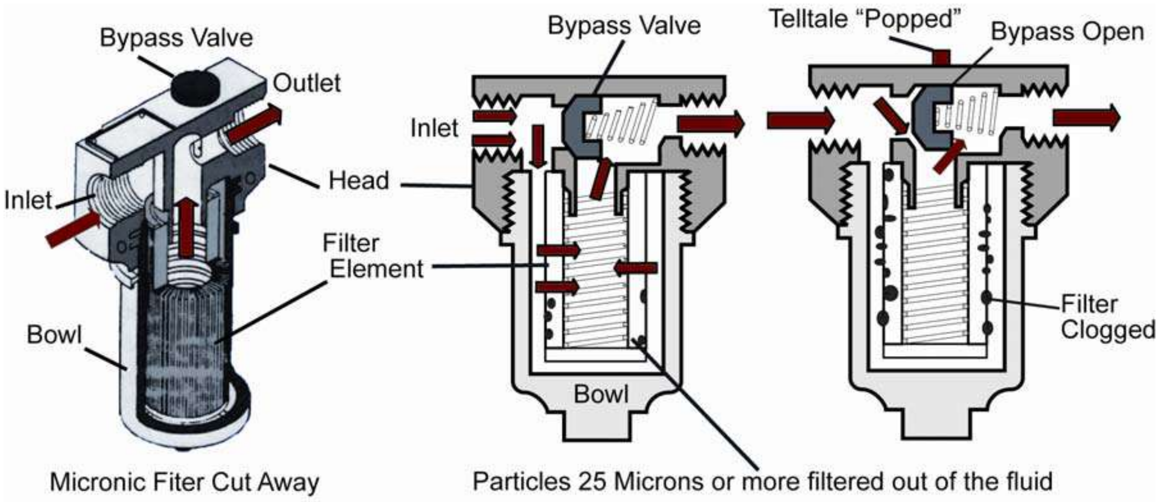


Diagram 9.32 Micronic Filter Operation

To prevent damage to selector valves and seals, even small particles of 25 microns or above have to be removed from the system fluid without restricting the flow.

To put it in context, there are 25 400 microns to the inch, a million microns in a meter, and the average human hair is 100 microns in diameter.

Pressure filters are normally referred to as Full Flow Micronic Pressure Filters, FFMPF, where the total output of the pump passes through the filter element before passing downstream to the rest of the system. In the cross section of a pressure filter, shown above as diagram 9.32, the pump's output enters the inlet in the filter head, flows down around, then through the micronic element before exiting from the centre of the element to the system.

As debris is deposited on the filter element, the restriction to through-flow creates a differential pressure across the element. This has the effect of increasing the pressure upstream of the filter. The reduced flow drops the pressure downstream of the filter. A pressure switch mounted each side of the filter is wired, so that this differential in pressure across the filter illuminates a 'filter warning light' on the flight deck.

Illumination of the filter warning light indicates that the filter is 'clogging' not that it has blocked. The appropriate flight manual details the action to take. If the clogging continues to a point where the filter would block, a bypass relief valve opens at a set pressure differential. This is maintained, allowing unfiltered fluid to enter the system.

Where filters mount, so that maintenance engineers can gain easy access, a red telltale button is ejected from the head when the bypass valve opens. This is filter popping. Filter elements are normally replaced on a flying-hours basis or at major servicing as a backstop.

PRESSURE REGULATION

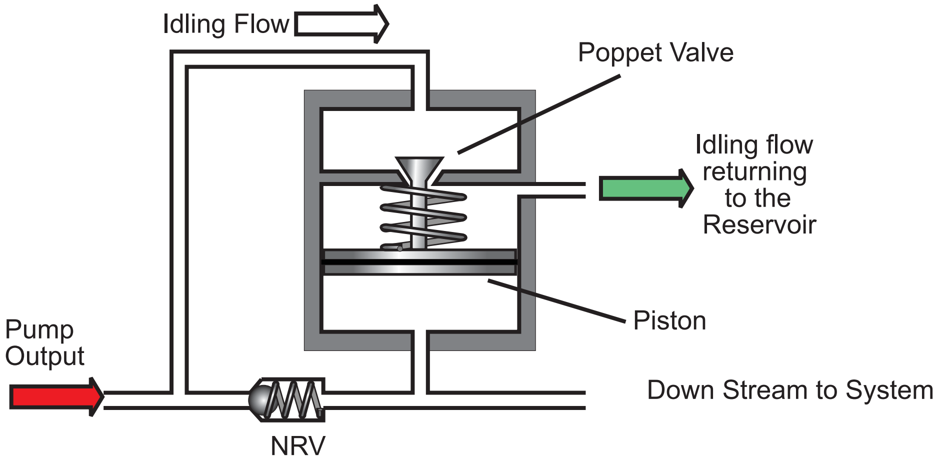


Diagram 9.33 ACOV in Kicked Out Condition

In a constant volume closed-centred system, pressure regulation is normally carried out by a hydro-mechanical valve called an Automatic Cut Out Valve or ACOV. See diagram 9.33 for its function and operation. The valve is located downstream of the pressure filter and is connected in parallel with a non-return valve.

The ACOV valve body is physically divided into two chambers, with the dividing wall having an orifice through it. Located in the valve is a combined piston and poppet valve. A compression spring trapped between the piston and chamber wall acts to close the poppet valve and forms three separate chambers. A tapping is taken from the pressure supply pipe before the NRV and leads to the poppet valve chamber. A tapping is taken after the NRV and leads to the piston chamber. An outlet port from the middle chamber is linked to the system reservoir.

From start, when the hydraulic system is slack, both the NRV and the poppet valve are seated by spring force. As the pump comes on line, and the pressure starts to increase, it is felt equally against the NRV and the poppet valve. When the pressure is sufficient, the NRV opens and fluid flows into the system raising the pressure. This pressure is now felt equally under the piston and over the poppet valve. The force created by the pressure on the poppet valve and the spring acting against the piston acts to hold the poppet valve closed. As the pressure continues to increase in the system, it creates disproportionate forces on poppet valve and piston due to their different surface areas.

When the force produced by the pressure in the system acting under the piston is equal to the ACOV spring, the only force holding the poppet valve closed is the fluid pressure acting down on its head. The disproportionate surface areas eventually results in overcoming the poppet valve and spring. At this point, the poppet valve is lifted, opening a path for the fluid in the upper chamber to flow to return. As this happens, the fluid pressure in the chamber drops, and the valve fully opens, allowing the full output from the pump to flow to return. At the same time as the poppet valve opens, the NRV closes, trapping fluid in the system downstream of itself. This maintains the pressure raised and keeps the poppet valve open.

In the above condition, the ACOV is said to be **kicked out**, and the system pressure is at its highest value. The opening of an idling circuit allows the pump to become off loaded, reducing the power required to operate the pump and the wear and tear on the pump. Any operation of a service downstream of the ACOV results in a drop in pressure. When the pressure has dropped below the force of the spring, the poppet valve re-seats, stopping the idling circuit. This is termed **kick-in** pressure. It is the lowest value of system working pressure. The system is regulated between these two pressures. The output of the pump now builds up against the NRV. When it is equal to the combined force of spring and pressure, it opens the NRV and re-pressurises the system.

Should the ACOV fail, an FFRV would maintain the system pressure at a value above the kick out pressure, resulting in higher pressure and temperature readings on the flight deck.

As actuators require pressure to produce the force and flow to produce the movement, a system using an ACOV must incorporate a hydraulic accumulator.

ACCUMULATORS

In active systems that use a constant displacement pump and pressure regulator, an accumulator must be fitted downstream of the NRV associated with the ACOV. For systems that use a self-regulating (constant pressure) pump, an accumulator is not required.

Hydraulic sub-systems such as the wheel brakes, flying controls, etc., have their own accumulators where required. These are dedicated to serve that sub-system.

All accumulators are designed to:

- Store energy (fluid under pressure)
- Provide a limited supply of fluid/pressure in the event of an emergency
- Dampen pressure fluctuations downstream of themselves
- Absorb fluid due to thermal expansion
- Cater for small internal leaks

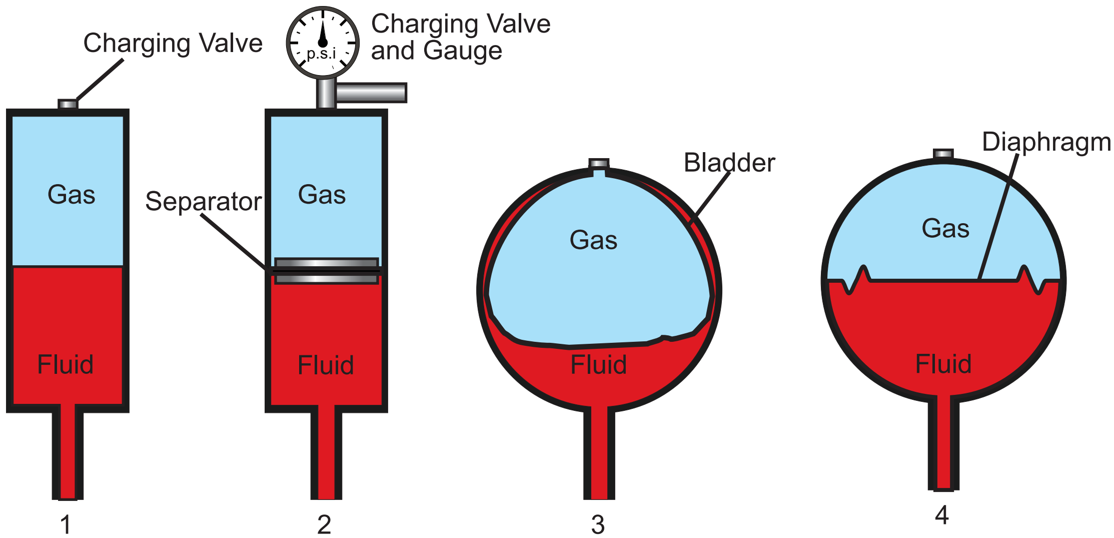


Diagram 9.34 Types Of Accumulator

Accumulators come in two different shapes, either spherical or cylindrical. Diagram 9.34 shows four separate arrangements for accumulators.

These are:

1. A cylindrical container without separator
2. A cylindrical container with a free floating piston or separator
3. A spherical container with an internal bladder
4. A spherical container with a flexible diaphragm

In early unseparated accumulators, the fluid and gas interfaced with each other. This resulted in the gas dissolving into the fluid, which created the problems of aeration (sluggish movement, etc.). It was also found that using compressed air could lead to the accumulator exploding as the hydraulic fluid (fuel) and compressed air (oxygen) dieseled due to the increase in pressure and temperature in operation. The introduction of a separator piston, bladder, or diaphragm prevented the gas and fluids from mixing. The use of an inert gas (nitrogen is preferred) eliminated both of these problems. For the gas to pass into the fluid or the fluid into the gas in a cylindrical accumulator with separator, the bore or piston seal would have to wear. In a spherical accumulator, the diaphragm or bladder material would have to become permeable to allow fluid/gas transfer.

In the event of the pressure pump failing, the NRV upstream of the accumulator would close, preventing loss of pressure back through the pump. The accumulator would give a limited emergency supply of pressure/fluid. In the wheel brake system, this is normally six full reversals of the brakes in manual application.

PRINCIPLE OF OPERATION

Accumulators make use of the principle that gas is compressible and fluid is incompressible. The accumulator is connected to the hydraulic pipe to allow pressurised fluid to flow in and out of the container. When the system is slack and the fluid is discharged back to the reservoir, a pre-charge of gas, nominally a third of the working pressure, is introduced into the cylinder.

As system pressure increases to the point where fluid pressure is greater than the pre-charge of gas, the fluid enters the container, thus reducing the volume available for the gas, which in turn raises the gas pressure. As the fluid flows into the container, the gas always equals the pressure of the fluid. When the system reaches working pressure, the fluid level/gas volume stabilises and their pressures are equal. In this condition, the accumulator is considered charged.

In the case of the fixed volume pump when the ACOV kicks out, as soon as a selector valve opens allowing system fluid into a service, the system pressure drops as the volume of the enclosing container increases. This, coupled with the absence of flow, causes the service to stutter until the ACOV kicks in and the flow and pressure increase. Therefore, where an ACOV is fitted, an accumulator must be fitted to prevent the ACOV from kicking in and out at such a frequency that the poppet valve would hammer and become damaged.

In the same system but with an accumulator fitted, when the ACOV is kicked out, the accumulator is fully charged. Any demand made on the system would cause the pressure in the fluid to drop. As the gas pressure when compressed was equal to the fluid pressure that compressed it, the gas pressure would expel fluid into the system giving the needed flow to create movement and pressure to create force.

As the fluid is expelled, the gas pressure and fluid pressure reduce. Starting with an initial pressure, or pre-charge, when the accumulator has discharged all the fluid into the system, the gas pressure still acts to pressure the fluid to this value. For safety reasons, the pre-charge or initial gas pressure must be equal to the lowest working pressure of the actuators in the system, nominally a third of the system working pressure.

For aircraft with hydraulically powered flying controls, upon shut down the pilot moves the controls through their full range of movements, counting the number of reversals obtained until the fluid pressure is dissipated and compares this with the flight manual specifications.

PRE-CHARGE INCORRECT

If the pre-charge is greater than the pressure specified in the maintenance manual, the volume of fluid that the accumulator can hold reduces. When fully charged, both gas and fluid are at system working pressure. This results in plenty of system pressure as the accumulator discharges but in a lack of fluid to operate the services, resulting in insufficient reversals.

Other effects are:

- More fluid left in the reservoir than designed, leading to an over-flow
- Lower fluid temperature due to excess fluid in the reservoir
- Lack of emergency reserve
- Frequency of ACOV kick in kick out, increasing to the point where fluctuations of pressure could cause damage (hammering)

PRE-CHARGING

Accumulator pre-charge is carried out when the system is slack. Most accumulators have a gas pressure gauge co-located with a charging adapter. The following sequence must occur to find the gas pressure for old style accumulators where there is no dedicated gas pressure gauge:

1. Release all system pressure to reservoir
2. Operate hand pump slowly while observing system pressure gauge
3. With each stroke, the gauge needle increases
4. When system pressure equals pre-charge pressure, the needle flickers

Heat created by the system's operation, or absorbed from ambient conditions, causes the fluid to expand and pressure within the pipes to rise. In systems with an accumulator, the fluid expansion is absorbed in the accumulator. The accumulator also irons out fluctuations of delivery pressure downstream of itself by absorbing or supplying fluid as required.

CONSTANT PRESSURE PUMP

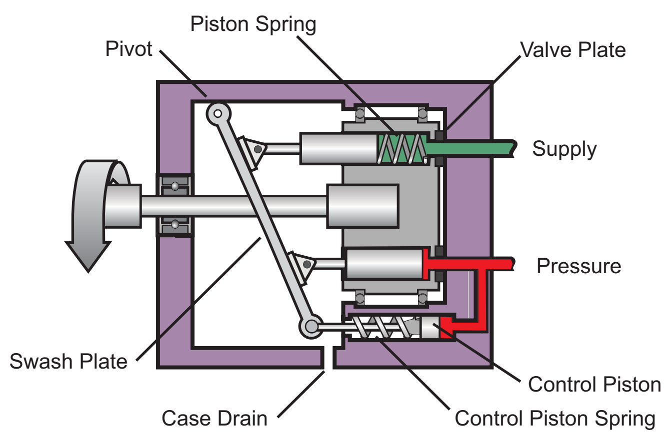


Diagram 9.35 Constant Pressure or Variable Volume Pump

In the other form of an active hydraulic system, constant pressure pumps are used.

These pumps effectively incorporate the pressure regulation within the pump housing. Diagram 9.35 shows a cross sectional view of a constant pressure pump. These can also be referred to as variable volume, variable displacement, or axial pumps.

The pump consists of a body, a cylinder block, and a valve plate and has either seven or nine pistons. In this design, the stroke of the pistons is controlled by the angle of a swash plate. As the pump rotates, the piston rods terminating in sliding shoes are forced to follow the swash plate by spring pressure.

The swash plate is controlled by the interaction of the control piston's spring force and the output pressure acting on the control piston. The spring force acts to move the swash plate to its maximum deflection. The output pressure acts to move the swash plate to the neutral position. The swash plate moves between these two positions. When supply and demand are equal, the spring force and the control piston force are in equilibrium and hold the swash plate steady.

When a demand is placed upon the system, the swash plate moves to increase the pumping capacity to maintain pressure. When the system is slack and the engines are switched off, the spring force sets the swash plate to maximum deflection or maximum stroke. As on starting the engine the starter motor will try to pressurise the whole system, a pump depressurisation/isolation system is used.

In normal operation, when full system pressure has been achieved and there are no demands on the system, the system pressure acting on the control piston moves it to the neutral position. As the input stroke and output stroke are equal, the pump is idling. Any demand on the system reduces the system pressure, resulting in the spring deflecting the swash plate. Fluid is pumped into the system until full system pressure is regained.

DEPRESSURISATION/ISOLATION

To enable the pump to be offloaded during the start cycle, the pilot selects depressurisation before initiating the start sequence. Diagram 9.36 shows a constant pressure pump with the associated circuitry for the blocking valve and depressurising solenoid in operation when the engine is in the process of starting.

Electrical power from the aircraft's vital bus bar (28 volts DC) is routed through two separate circuit breakers and two separate switches. A discussion of the fire switch occurs in depth in later notes. Under normal conditions, it is held in the normal position as shown. The EDP control switch has three positions. These are supply off, depressurise, and normal. In diagram 9.36, the switch is shown in the depressurise setting.

SUPPLY OFF

To isolate the reservoir from the pump when the system is slack, in an engine fire condition, or other abnormal condition, an electrically motored shut-off valve is fitted. When "supply off" is selected, the ganged switch moves down and the valve motor shuts. If the fire switch is operated, the valve automatically motor shuts.

DEPRESSURISING

This position is for use in engine start. The supply shut-off valve motors open and power is supplied to the depressurising solenoid valve, energising the field winding, opening the valve, and allowing the pump's output to act under the blocking valve, as shown in diagram 9.36.

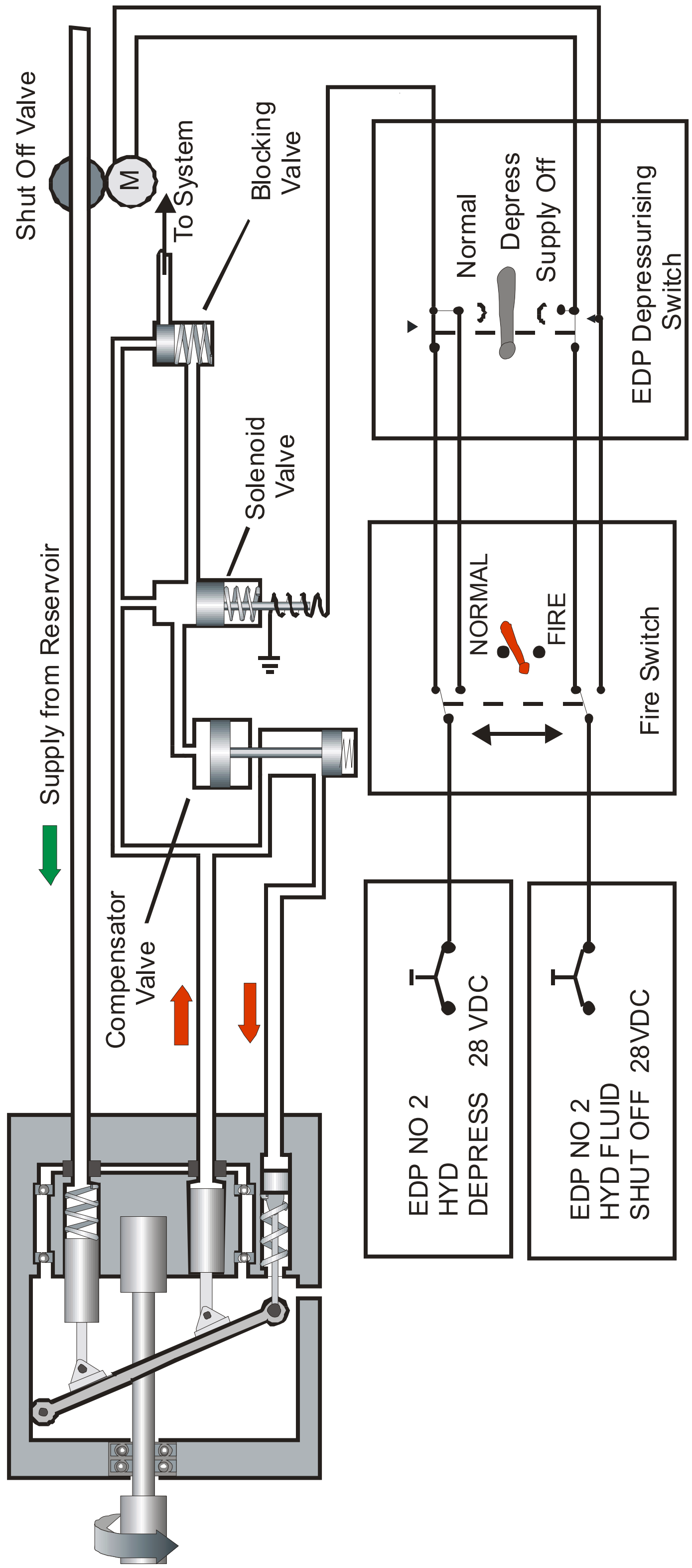


Diagram 9.36 Blocking Valve Operation

NORMAL

In the normal position, the solenoid of the depressurising valve is de-energised, and the valve is returned to the closed position by spring force. This ensures that in the event of a total electrical failure, hydraulic power is still available.

BLOCKING VALVE

The blocking valve itself is sprung loaded to close and isolate the pump from the hydraulic system. It is closed by its spring when the system is slack and is kept closed when “depress” is selected. When “normal” is selected, it is held open by fluid pressure.

COMPENSATING VALVE

The compensating valve acts to boost the initial pressure produced by the pump during start to ensure that the swash plate moves to the neutral position quickly.

CONDITION ON START

Before start with “depressurising” selected, the following conditions exist:

- The EDP at max deflection
- Reservoir shut off valve open
- Blocking valve closed
- Depressurise valve energised and open

As the pump starts to turn, pressure rises above the blocking valve and beneath the blocking valve due to the depressurising solenoid valve being open. Thus, the blocking valve remains closed.

Pressure starts to rise on the control piston and above the compensating valve piston, which in turn forces its smaller piston down. This increases the pressure/flow on the control piston, resulting in the control piston moving the swash plate to the neutral position, which causes the pump to idle. On the completion of a successful start and at the correct engine rpm, the pilot moves the control switch to “normal”, de-energising the depressurising solenoid valve, which closes under spring force and allows the fluid under the blocking valve to drain away. As the pressure under the blocking valve and above the compensating valve diminishes, the spring force deflects the swash plate, pumping fluid into the system. This in turn opens the blocking valve and pressure/flow enters the aircraft’s system.

In the event that the swash plate jams in the fully defected position, the pilot has the ability to select depressurise and isolate the pump. This prevents the possibility of damage to the system due to the very high temperatures that can be created when a simple pressure relief valve is used as the only means of relieving the pump’s output. In the event that the pilot operates the fire handle, both the shut-off valve and the depressurising solenoid valve operate, isolating the pump from its supply of fluid and the pump from the system.

FLUIDS

While all fluids are considered incompressible (up to 32 tons per square inch), the specifications for the ideal aircraft hydraulic fluid are extensive. The properties required are:

- Low viscosity
- Resistance to heat
- Chemically stable
- Non-flammable
- Resistance to foaming
- Good lubrication
- Corrosion resistance

A low viscosity is required to reduce the internal friction of the fluid, enabling it to flow through the system's pipes, hoses, and components. As the working pressures at which modern hydraulic systems operate have increased, so has the requirement of the hydraulic fluids used in them. Modern fluids have a greater resistance to the higher temperatures generated at these pressures. However, any increase in chemical stability at the upper end of the temperature range must not be at the expense of the lower end of the temperature range. The fluid has to be stable across a broad range of conditions and have a low freezing point and a high boiling point. Ideally, the fluid should be non-flammable.

Aircraft hydraulic fluid must be resistant to foaming. If bubbles of gas or air enter the supply to a hydraulic pump, they can create a cavitation directly in the inlet of the pump, which would stall the pump and stop pressure output. In addition, if air passes into the system, it can cause undemanded movement of actuators as the gas expands and contracts with changes in system pressure.

Foaming normally occurs in the reservoir, as any dissolved gasses in the returning fluid are able to expand as the pressure within the fluid dissipates in the reservoir. Anti-foaming agents are added to the fluids to prevent a froth forming that could be sucked down into the inlet of the pump and cause cavitation.

Within the system, many moving surfaces require cleaning and lubricating. The fluid itself is the medium for this lubrication. The chemical make-up of the fluid must not cause galvanic corrosion between dissimilar materials and should prevent corrosion.

There are three types of fluid:

- Vegetable-based fluid — not common in usage.
- Mineral-based fluid — used in military and light aircraft and for U/C shock absorbers.
- Synthetic-based fluid — the common fluid for commercial aircraft.

VEGETABLE-BASED FLUIDS

Vegetable-based fluid consists of a mixture of castor oil and alcohol. This fluid is almost colourless. Due to the alcohol, vegetable-based fluid has only a low tolerance to temperature making it unsuitable for high-pressure systems. The alcohol boils off at altitude and presents a fire risk, since it is flammable. Due to its chemical composition, this fluid must only be used with pure rubber seals and hoses.

MINERAL-BASED FLUID

Mineral-based fluid is a kerosene product and is dyed red for identification. This fluid has a higher temperature range and is suitable for use in high-pressure systems (3000 psi). It has good resistance to foaming, a high boiling point, and high flash point. However, it is flammable. This fluid must only be used with synthetic rubber seals and hoses.

SYNTHETIC-BASED FLUID — SKYDROL®

Skydrol® is a manufacturer's trade name that has become the generic name for fluids manufactured from phosphate ester by various companies. There are two grades of Skydrol® in current use:

- Type IV - Skydrol® LD-4 & 500B-4 – purple in colour and most common grade
- Type V - Skydrol® 5

Skydrol® fluids fulfill the complete list of requirements and, due to their chemical composition, can be used at very high temperatures and pressures. These fluids must be used with butyl rubber, ethylene propylene, or Teflon for the seals and hoses. Skydrol®, as with the other hydraulic fluids, is an irritant. Monsanto, the maker of Skydrol®, recommends that the user take proper health and safety precautions to prevent skin contact, eye contact, or inhalation.

FLUID TEMPERATURE

The hydraulic fluid temperature is normally sensed in the system reservoir and is displayed on the flight deck. All hydraulic fluids need to be kept within the working temperature range of the fluid so that the hydraulic system functions efficiently and the fluid remains chemically stable. If the fluid is too cold, then its viscosity increases, causing the operation of the services to become sluggish and too hot whereby the fluid can chemically break down. To maintain the fluid at its correct operating temperature, heat exchangers are used. These are fitted in the return or suction line of the system.

HEAT EXCHANGERS

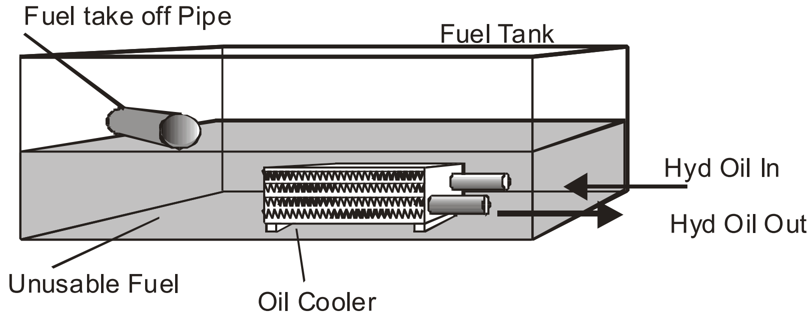


Diagram 9.37 Heat Exchanger

The heat exchangers can either be a ram air unit, where ambient ram air extracts heat from the fluid, or a unit mounted in the fuel tanks. For the latter, the units mount so that the unusable fuel constantly covers them, as per diagram 9.37. Hydraulic fluid/fuel coolers are common on turbine-powered aircraft as they make use of the unwanted heat from the hydraulic system to warm the aviation kerosene. This helps prevent the fuel from waxing or freezing as well as cooling the fluid.

Where the return line through a heat exchanger becomes blocked, a pressure sensing bypass valve allows the fluid to bypass the heat exchanger and return directly to the reservoir. This of course results in an increase of the fluid's temperature. If the hydraulic fluid overheats, the fluid thickens (becomes more viscous and darkens in colour).

Should a heat exchanger in a fuel tank leak, there is a risk that the aircraft's fuel would enter the hydraulic system, resulting in fluid contamination. This would alter the following fluid properties:

- Viscosity
- Resistance to heat
- Chemically stability
- Flammability
- Resistance to foaming
- Lubrication properties

A possible additional effect could be the seals breaking down, resulting in internal leaks and excessive wear of the EDP.

SEALS AND SEALING

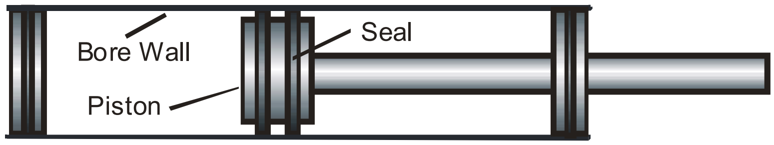


Diagram 9.38 Dynamic Seals

While the heat exchanger units remove the excess fluid temperature, the temperature of fluid also heats the hydraulic components. Due to the varying materials used and their different rates of expansion, there has to be a gap left between the moving surfaces to allow for this expansion. Seals are fitted to prevent the fluid leaking through the gap between the piston and cylinder walls. Where seals are fitted onto a moving surface (piston), they are termed dynamic, as in diagram 9.38.

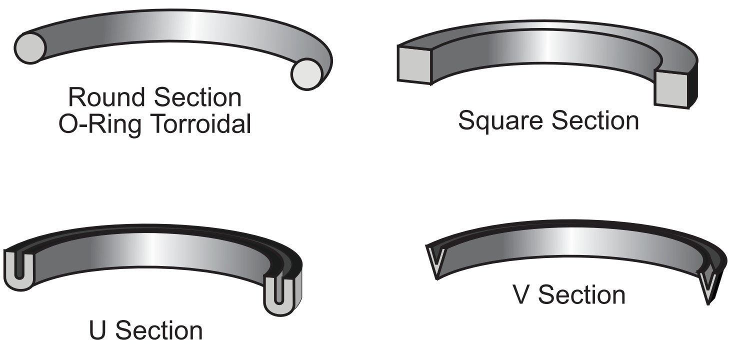


Diagram 9.39 Cross Section of Hydraulic Seals

Seals are also fitted between fixed parts. These are termed static. It is common practice to define a seal by referring to its cross section as shown in diagram 9.39.

STATIC SEALING

Round section and square section seals can be used to seal between static parts of a component. A flat sheet of sealing material can also be used. Static seals between mating surfaces are frequently termed **gaskets**. To be effective, seals have to be a tight fit between two surfaces. In the case of static sealing, the seal material is clamped and compressed by the two mating surfaces. This makes a fluid-tight joint.

DYNAMIC SEALING

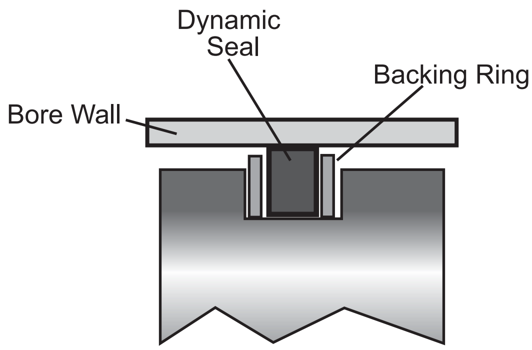


Diagram 9.40 Backing Rings

For dynamic sealing, the friction between the seal and cylinder wall must be kept to a minimum. Otherwise, the seal tries to remain stationary when the piston moves. This can result in the seal rolling out. To prevent this from happening and to support the seal, backing rings are fitted between the seal and the piston, as diagram 9.40 illustrates.

All the cross sections shown in diagram 9.39 can serve as dynamic seals. The round and square section seals seal hydraulically in both directions. U and V sections only seal hydraulically in one direction when the pressure acts into the section. Therefore, to seal a double-acting actuator, a minimum of two V or U-shaped seals would have to be fitted, one facing in either direction.

WIPER OR SCRAPER RINGS

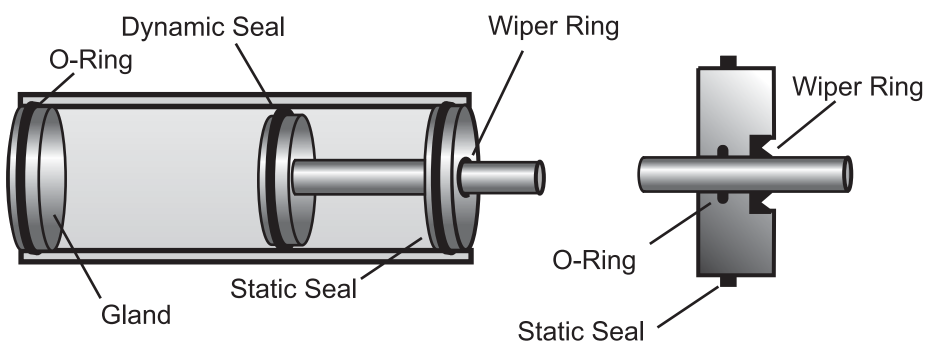


Diagram 9.41 Wiper Ring

To prevent dirt and grit that has adhered to an actuator's ram from damaging the dynamic seals on retraction, wiper or scraper rings are fitted. These are U or V-shaped seals, which wipe the dirt off the ram as it retracts and do not serve as hydraulic seals. Sealing is achieved via an O-ring or square seal acting against the ram, as diagram 9.41 shows.

INTERNAL LEAKAGE

Internal leakage occurs when hydraulic fluid leaks from the pressure side of the piston to the return side. This is normally due to a damaged seal or worn cylinder bore. It results in a drop in system pressure, the loss of a hydraulic lock, and causes a rise in system temperature as the pressure-regulating device has to function more frequently.

EXTERNAL LEAKAGE

External leakage occurs when fluid finds a path past a static seal or, in the case of an actuator, the dynamic seal around the ram. This results in a loss of system fluid, a rise in the remaining fluid's temperature, and depending on fluid type, increases the risk of fire and the possible shorting out of electrical equipment.

LEAK RATE

When a hydraulic system is unpressurised (slack), the force exerted on the mating surfaces and their seals and gaskets is reduced. This can result in a static leak. In this situation, the fluid runs or drips from the affected component. Many aircraft have set allowances termed a **leak rate** for this condition. When hydraulic pressure is re-applied, the sealing becomes fully effective and the leakage stops. If a component leaks when hydraulic pressure is applied, it is termed a **dynamic leak**. In this instance, the fluid runs freely or sprays from the affected component. Where static leaks might be acceptable, dynamic leaks are not.

SYSTEM RESERVOIRS

Most modern hydraulic systems are considered as **self bleeding** This means that the system removes the dissolved gases or air automatically. Older designs did not have this ability. Any air that had entered the system required manual purging (bleeding out) to prevent the incorrect operation of the system.

Reservoirs are found in many different designs, two of which are examined here. Whichever design is used, the reservoir must meet certain specifications. It must:

- Hold sufficient fluid for the entire system's use
- Hold an extra volume of fluid to cater for small external leaks
- Have sufficient volume to cater for jack ram displacement
- Have sufficient volume to cater for fluid expansion due to heating
- Have a means to remove air and dissolved gases from the returning fluid
- Hold an emergency reserve of fluid
- Ensure that only fluid can enter the pump inlet supply pipe
- Have a physical means to determine the fluid level when the system is slack
- Have a means of replenishment

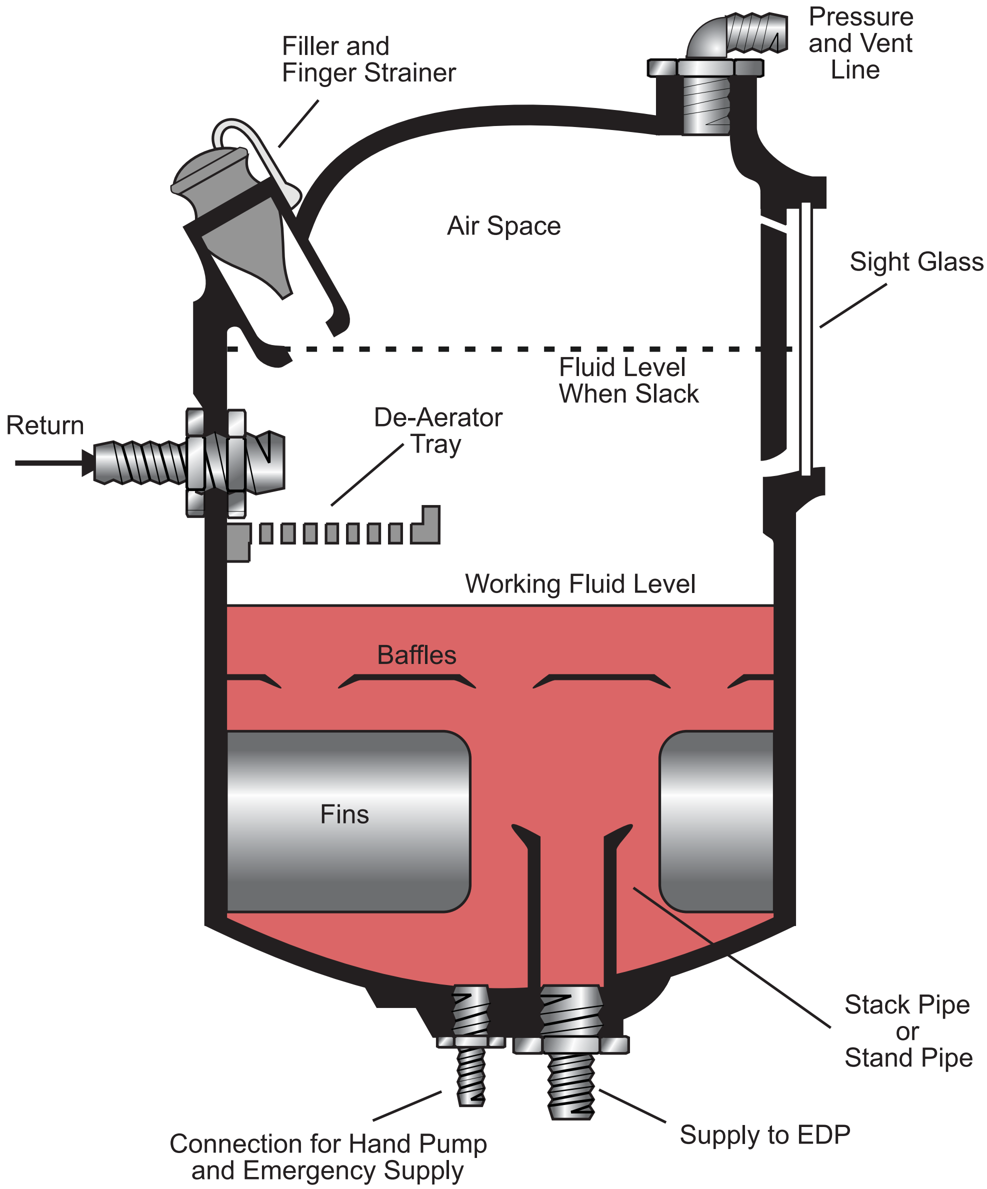


Diagram 9.42 Standard Diagram for a Pressurised Reservoir

Refer to diagram 9.42, which shows a cross section of a pressurised reservoir. The reservoir consists of a container that has a greater internal volume than the fluid it has to hold. This allows for both thermal expansion of the fluid and the change in fluid level when unbalanced double-acting actuators retract. The dashed line shows the slack fluid level. This condition only exists when the aircraft is put into the correct condition as per the Flight Manual, for example:

- Undercarriage down and locked
- Flaps retracted
- All accumulators dissipated and correctly pre-charged
- Reservoir unloaded

The lower line shows the working level. This is the level to which the fluid drops when the accumulators are fully charged. The fluid level now varies around this level as services are operated.

The method of checking the level is normally via a sight glass. This can take the form of a clear external tube or a clear circular disk. These are marked with upper and lower limit lines. The fluid level must be between these lines. The lower line (Lo) shows the level when the system is cold and the upper line (Hi) when the system is hot after operation. Some systems use a dipstick instead of the sight glass.

The fluid supply to the EDP is taken though a stack or stand pipe which ensures that should the main system leak, there is a reserve of fluid that cannot be drawn from the reservoir by the pump. The emergency reserve draws from the bottom of the reservoir, and this also supplies the hand pump. To ensure that clear fluid is always kept around the mouth of the stack pipe, fins and baffles are fitted. These prevent the fluid from swirling and surging.

DE-AERATOR

To remove air and dissolved gasses from the returning fluid, the fluid passes over a de-aerator tray. This can take several forms. The two common ones are:

- A flat tray which has a lip and is perforated with small holes
- A series of stepped small trays that are lipped and perforated with small holes

These allow the returning fluid to spread out and the dissolved gasses and air to boil out of the fluid. Clear fluid is able to drop back into the body of fluid below without enfolding air into it. The air is able to vent to airspace above the fluid and then to atmosphere.

The reservoir must be vented to atmosphere to ensure that a partial vacuum is not formed above the fluid as the level drops to the working line. If the reservoir is subject to only the ambient atmospheric pressure, as the aircraft climbs, the drop in pressure can result in:

- The fluid evaporating
- A corresponding drop in the supply pressure at the inlet to the pump
- Foaming due to the expansion of dissolved gasses within the fluid

To overcome this, in modern reservoirs the airspace above the fluid is pressurised. This is termed **loading** the reservoir. Conversely, when the pressure is removed, it is termed **unloading**. The air source can be taken from the aircraft's engines or a cylinder of compressed air. A vent valve is set to an upper limit and protects the reservoir from over pressurisation when the level increases. The values quoted for this are between 5 and 15 psi.

The main advantage is that the fluid supplied to the pump inlet is already under pressure, which prevents a cavitation from forming in the inlet pipe. This also allows the reservoir to be located below the pump and the air pressure forces the fluid up the supply pipe into the pump's inlet. Unloading the reservoir results in the fluid level in the reservoir rising as the fluid finds its own level.

If a reservoir mounted below its pump was to become unloaded during operation, the pump has to be able to draw the fluid from the reservoir and provide at least 75% normal power to the system.

The reservoir has a filler point for fluid replenishment. Located within the neck of the filler is a gauze filter to prevent the ingress of foreign objects. The type of fluid to be used is marked adjacent to the filler. This gives the person servicing the reservoir a final chance to check that the correct fluid is used.

FLUID CONTAMINATION

In the event that the wrong fluid is added to the reservoir, the system must not be operated nor must the supply cock be opened. The likely consequences of fluid contamination are:

- System failure as the seals deteriorate
- Internal leakage
- External leakage
- Increase in system temperature
- Chemical breakdown of the contaminating fluid resulting in cavitation as the fluid boils off within the system
- Chemical breakdown of the contaminating fluid resulting in an increase in viscosity as the fluid overheats

BOOSTED OR BOOT-STRAPPED RESERVOIR

Referring to diagram 9.43, the boosted reservoir, also termed a **boot-strapped** or **accumulator** reservoir, is similar to a cylindrical accumulator with separator. System fluid is stored on one side of the piston, which is acted on by a spring force and by bleed air or nitrogen gas pressure. These act to load the reservoir and ensure that pressurised fluid is supplied to the EDP. The piston or separator moves with the demands placed on the reservoir by the system. This movement allows for the ram displacement and thermal expansion.

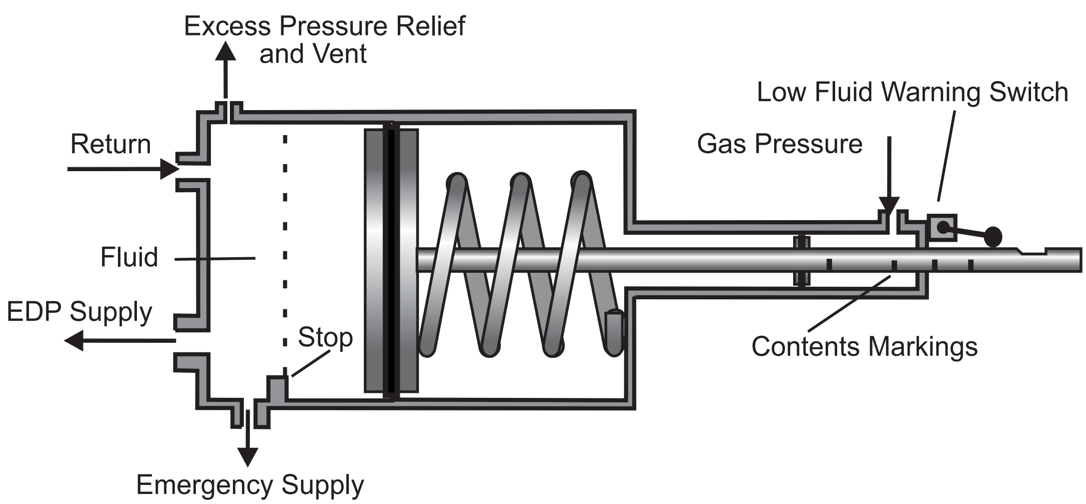


Diagram 9.43 Boosted or Boot Strapped Reservoir

The returning fluid is able to boil off any air that has been transported back, and this can vent through an overpressure relief valve over the side. The pump supply is taken from a position above the bottom of the reservoir. An internal stop for the piston ensures that there is an emergency reserve of fluid, denoted by the broken line in the diagram. A pipe located at the bottom of the reservoir supplies the hand pump and emergency pump.

To allow the fluid level of this type of reservoir to be determined, the piston is attached to a ram, which protrudes through the end of the reservoir. The ram is marked off in measurements that the particular system uses, enabling the person checking the reservoir to see the fluid level by how far the ram protrudes from the housing. A gas pressure gauge shows the base pressure. A co-located graph allows the checker to ensure that the correct fluid level coincides with the correct base pressure. A low contents warning switch illuminates a red warning light on the flight deck if the ram moves too far inward.

HYDRAULIC SYSTEMS

The previous sections discussed how hydraulic power is generated and the advantages of closed-centred systems. In this section, some of the systems and services that use hydraulics as the power source and the components that ensure that these systems function correctly are discussed.

Transport aircraft normally use hydraulics for the operation of:

- Power flying controls
- Artificial feel (Q feel)
- Trailing edge flaps
- Leading edge flaps
- Lift dumpers
- Spoilers and speed brakes
- Extension and retraction of landing gear
- Wheel brakes

As a means of linking the following text and putting the components in context, the following page shows a schematic diagram (9.44) of a typical twin-engine aircraft's hydraulic system.

In diagram 9.44, there are three hydraulic systems: blue, green, and yellow. It is fairly standard to refer to systems by colour to differentiate between them and to have multiple systems linked to one service. This complies with the ethos of redundancy where no one single failure should cause the loss of the aircraft.

PRIORITY SERVICES

The priority systems of a transport aircraft are the primary flying controls and the wheel brake systems. These systems must be supplied with whatever pressure and flow is available. To achieve this, Pressure Maintaining Valves (PMV), also known as priority valves, are fitted in the system. These isolate the other hydraulic services until a predetermined pressure is reached.

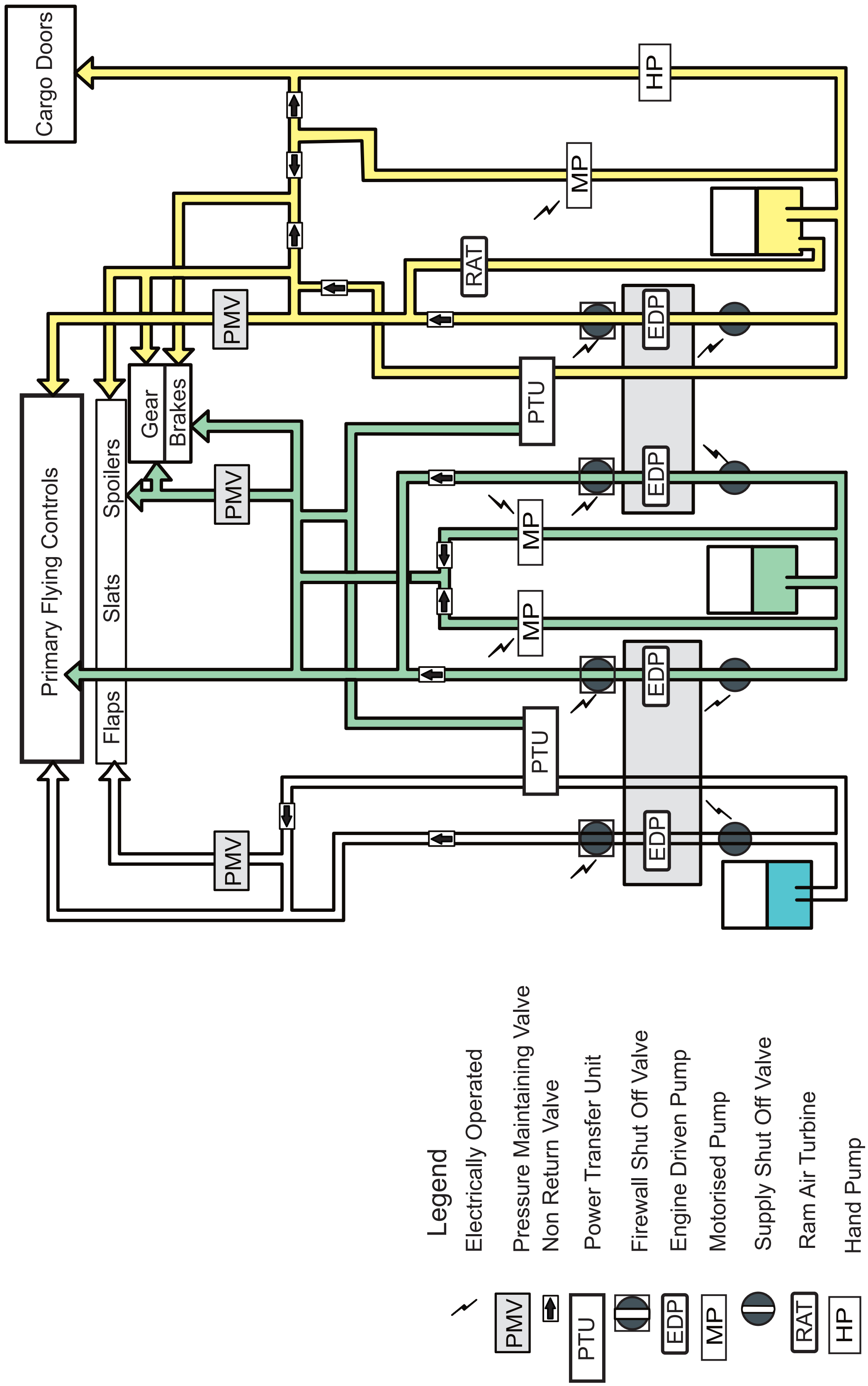


Diagram 9.44 Typical Hydraulic System for a Twin-Engine Aircraft

PRESSURE MAINTAINING VALVE OR PRIORITY VALVE

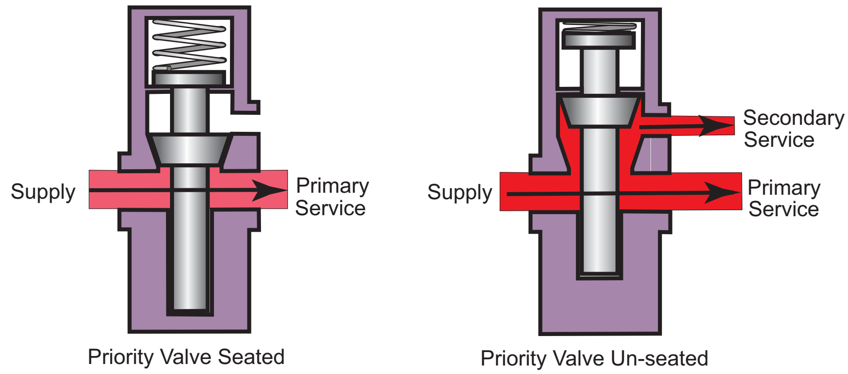


Diagram 9.45 Pressure Maintaining Valve

Pressure maintaining valves (PMVs) are fitted in the pressure line at a junction where supply is taken off for secondary services. See diagram 9.44 for schematic location. The function of the valve is to isolate the secondary service from the supply if the supply pressure drops below a predetermined value.

Diagram 9.45 illustrates the valve operation. The valve is held on its seat by spring force. When the supply pressure is sufficient, the valve is lifted off its seat and fluid flow/pressure is supplied to the secondary services. In a failing system, the priority valve closes and ensures that remaining pressure goes to the primary services by isolating the secondary services.

HYDRAULIC MOTORS

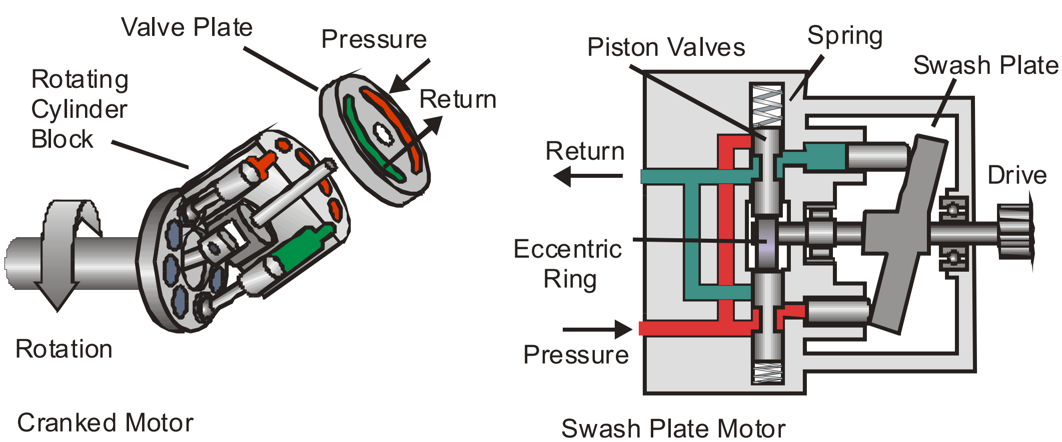


Diagram 9.46 Hydraulic Motors

Hydraulic motors are effectively piston pumps that work in reverse, where the fluid pressure pushes the pistons down. This linear movement is converted into rotary motion. There are two types, cranked and swash plate, as illustrated in diagram 9.46.

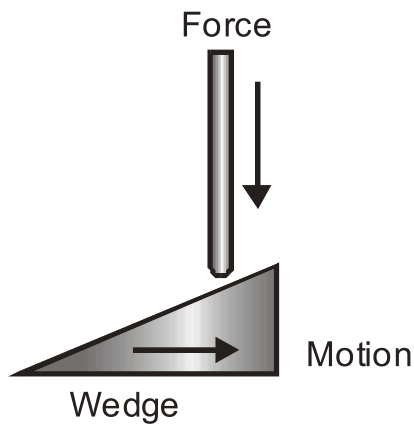


Diagram 9.47 Principle of a Hydraulic Motor

Both motors work on the principle (discounting friction) that a force applied to the sloping surface of a wedge causes the wedge to move away from the force. See diagram 9.47 for an illustration. In the case of hydraulic motors, the wedge is formed either by cranking the motor's body or a fixed swash plate. While the pressure provides the force to turn the motor and the motor's torque, the flow rate into the motor determines the speed of rotation.

Hydraulic motors can be connected to gearboxes to operate flaps or linked directly to a pump to make a **power transfer unit**. Operation of other services can cause fluctuation of flow rates within the system, which would cause the hydraulic motor to vary its output speed. Flow control valves are normally fitted upstream of a motor.

FLOW CONTROL VALVES

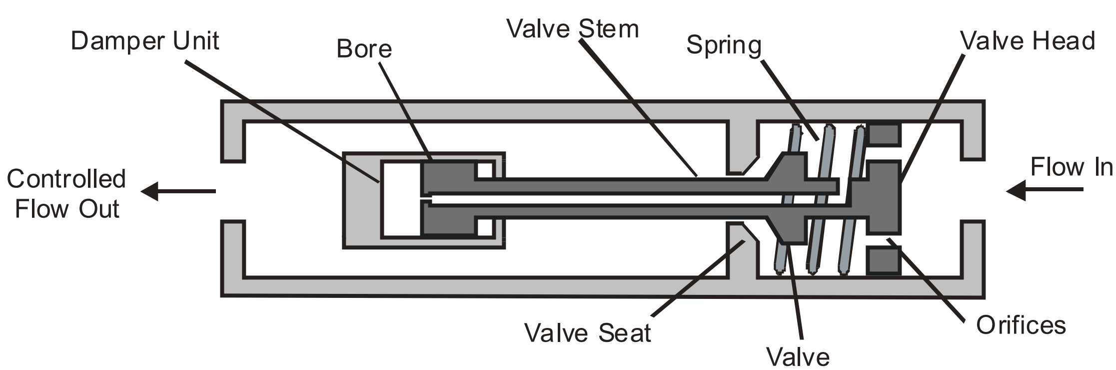


Diagram 9.48 Flow Control Valve

Diagram 9.48 shows a flow control valve, which are fitted upstream of components such as hydraulic motors to ensure that a constant flow rate is maintained to the motor.

For example, a motor requires a flow rate of 5 litres per minute, where the flow rate can vary between 5 to 10 litres per minute. When the flow rate in is equal to the required flow rate out, the spring acts against the valve head and holds the valve off the seat, allowing this flow to pass to the motor.

As the flow rate increases, the force on the valve head increases, resulting in the valve moving against the spring until the force in the spring and the valve head equalise. This action closes the gap between the valve and the valve seat, which limits the flow through the valve to the 5 litres per minute.

To prevent the valve from being reactioned into the valve seat when there are fluctuations in the flow rate, fluid pressure is fed into the damper unit via a small boring through the valve stem. This acts against the end of the valve stem and the body as a damper, which results in a slowing down of the valve's movement.

POWER TRANSFER UNIT

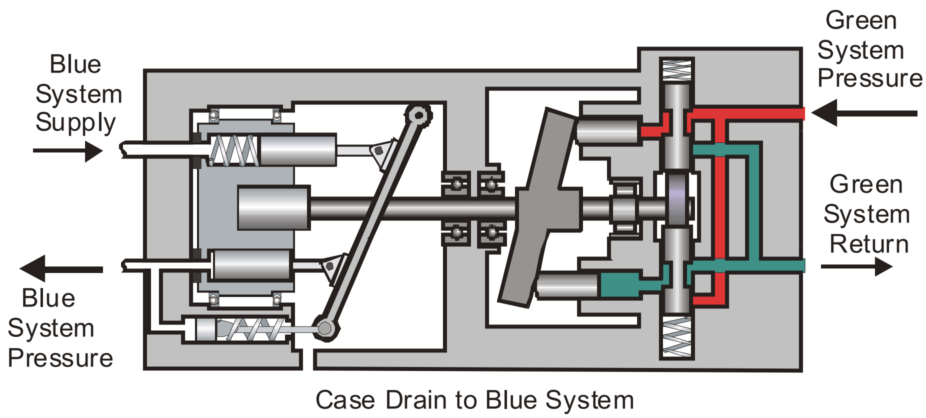


Diagram 9.49 Power Transfer Unit

Modern hydraulic systems make use of power transfer units. A hydraulic motor located in one system drives a hydraulic pump in another system. This enables a system where an EDP has failed to be brought back on line.

RAM AIR TURBINE

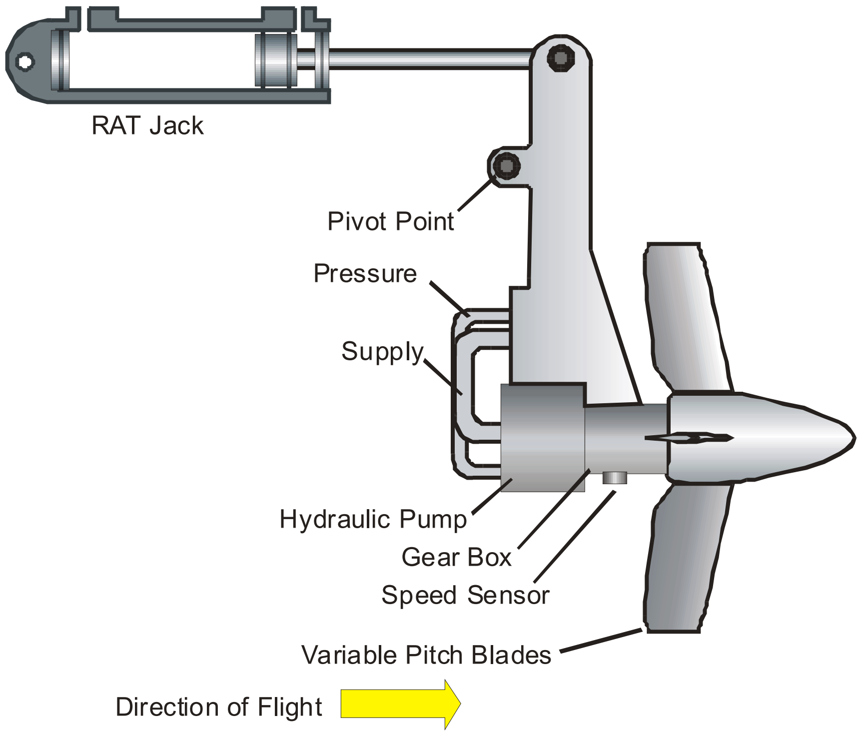


Diagram 9.50 Ram Air Turbine Deployed

The Ram Air Turbine, or RAT, is designed to give limited hydraulic power to the primary services in the event the normal hydraulic power generation system fails (see diagram 9.44). Diagram 9.50 shows the RAT, which consists of a variable pitch propeller driving a small hydraulic pump via a gearbox.

When not in use, the RAT is held in a bay, which is normally located in the underside of the aircraft, by main system pressure. When stowed, the RAT's propeller blades are kept in the feathered position. In the event of a drop in the system's pressure below a predetermined level and if the airspeed is below a set level, the RAT is deployed into the airflow, where the blades move from the feathered condition and start the windmilling action which drives the pump.

Altering the blade pitch angle controls the RAT's rpm. The first action of the RAT is to recharge its own accumulator and then charge the main system, where the priority valve ensures that the primary flying controls get the flow. As the wheel brakes are also primary hydraulic services, if the RAT is deployed in flight, it remains deployed on landing although its effectiveness drops as the aircraft loses speed.

In normal circumstances when an aircraft has landed and system pressure is dissipated, the RAT is held in its bay by interlocking switches that sense the aircraft is on its undercarriage.

RESTRICTORS

Restrictors serve to control fluid flow. In most cases, they serve as speed control devices. There are two basic types:

- Two-way or choke
- One-way

TWO-WAY RESTRICTORS

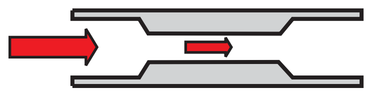
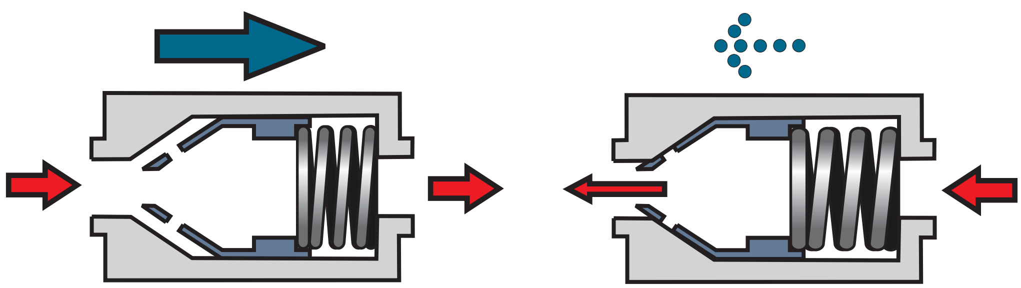


Diagram 9.51 Two-Way Restrictor

Reducing the bore of a pipe forms Two-way restrictors. This limits the flow of fluid in both directions equally. They can be referred to as a choke and are frequently fitted to light aircraft with retractable undercarriages. See the section on Power Packs in Chapter 7, Landing Gear, and diagram 7.37.

ONE-WAY RESTRICTORS



Un-Restricted Flow Restricted Flow

Diagram 9.52 One-Way Restrictor

A one-way restrictor is designed to allow full flow in one direction but restrict the return flow. Diagram 9.52 shows a cross section of a standard one-way restrictor. The left diagram shows the valve open to allow unrestricted flow. The right diagram shows the valve in the closed position where flow is reduced. To show the direction of full flow, a bold arrow is stamped on the case with an arrow made of dots showing the restricted flow.

The location of restrictors is very important. The forces acting on the service must also be taken into account (e.g. for use in controlling the speed of operation of the undercarriage and flaps as shown in diagram 9.53).

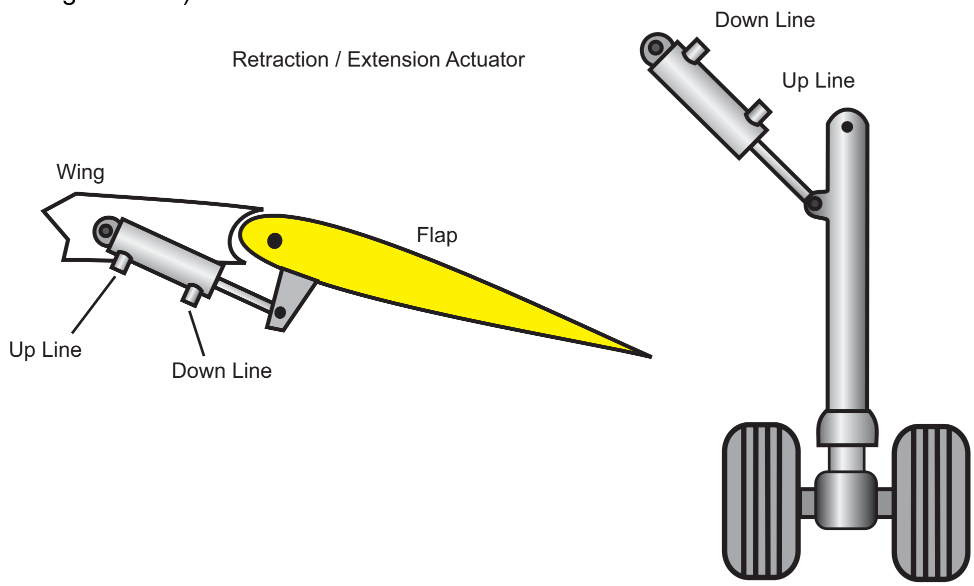


Diagram 9.53 Flap and Undercarriage

When lowering the undercarriage, the rate of extension has to be controlled, as the mass of the gear and gravity would want to take charge and accelerate it. The force, created by the mass times the acceleration, could be sufficiently large that damage could occur at the end of the gear's travel. To control this rate of extension a one-way restrictor is fitted in the "up" line, restricting the flow from the actuator. This creates a back pressure resisting the force of the fluid acting in the down side of the actuator and gravity. If the restrictor was fitted to control the flow of fluid entering the down line, the gear would fall with gravity and a cavitation would form in the down side of the actuator, as fluid could not enter fast enough. This would result in damage to the piston seals, etc.

For the flaps, the restrictor is fitted to control the rate of flap retraction by controlling the rate of flow from the downline of the flap actuator. This prevents the air load from acting on the flaps and moving them up faster than the fluid can flow into the actuator.

VALVES

THROTTLING VALVE

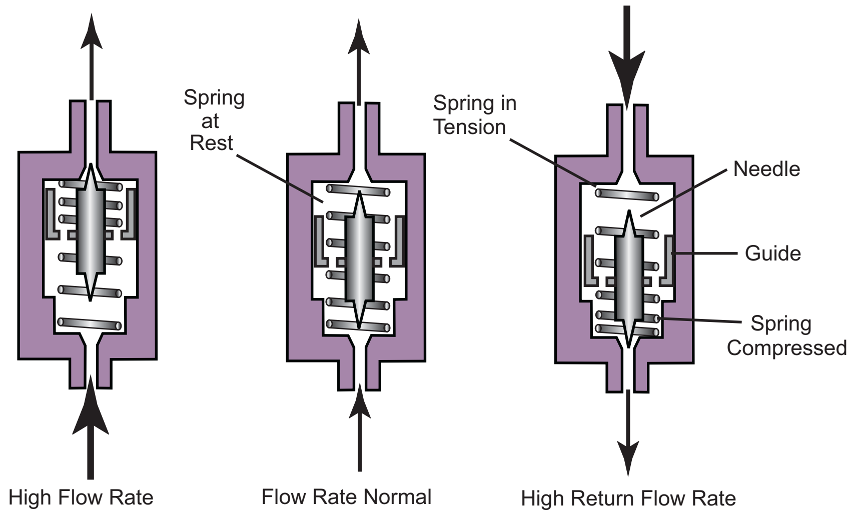


Diagram 9.54 Throttling Valve

Throttling valves can be used in place of restrictors. These valves ensure that the flow rate to and from a component is correct. Diagram 9.54 shows a cross section of a standard throttling valve. The core of the valve is held within the spirals of a spring. Mounted at right angles to the core is a circular plate with holes through it. The valve core is kept aligned by a guide attached to the plate.

An increase in the flow rate acts against the plate, deflecting the core. This force is balanced by the compression of the spring on one side and the tension in the spring on the other. The deflection moves the needle end of the core into and out of the corresponding shape in the body, thus making a variable restrictor, which controls the flow rate from the throttling valve. As the core is coned at both ends, the valve can be used to control inflow and outflow from a component. This makes the throttling valve suitable for replacing the standard one-way restrictor valve in an aircraft flap system.

SEQUENCE VALVES

Sequence valves are used to ensure that one operation is completed before another starts. In the operation of systems such as the retractable landing gear of large aircraft, there are several stages that must be completed in the correct order to ensure the correct operation of the system (e.g. when extending the landing gear, ensuring that the gear doors open before the gear is unlocked).

There are three types of sequence valves:

- **Hydraulic sequence valves** work by directing system fluid into one service ensuring its complete operation before the fluid flow is able to enter the second service.
- **Mechanical sequence valves** require a physical action to operate them.
- **Electrical sequence valves** are selectors that are electrically actuated when one action is completed. A micro switch makes and completes a circuit so that the sequence valve moves and directs the fluid to the next component.

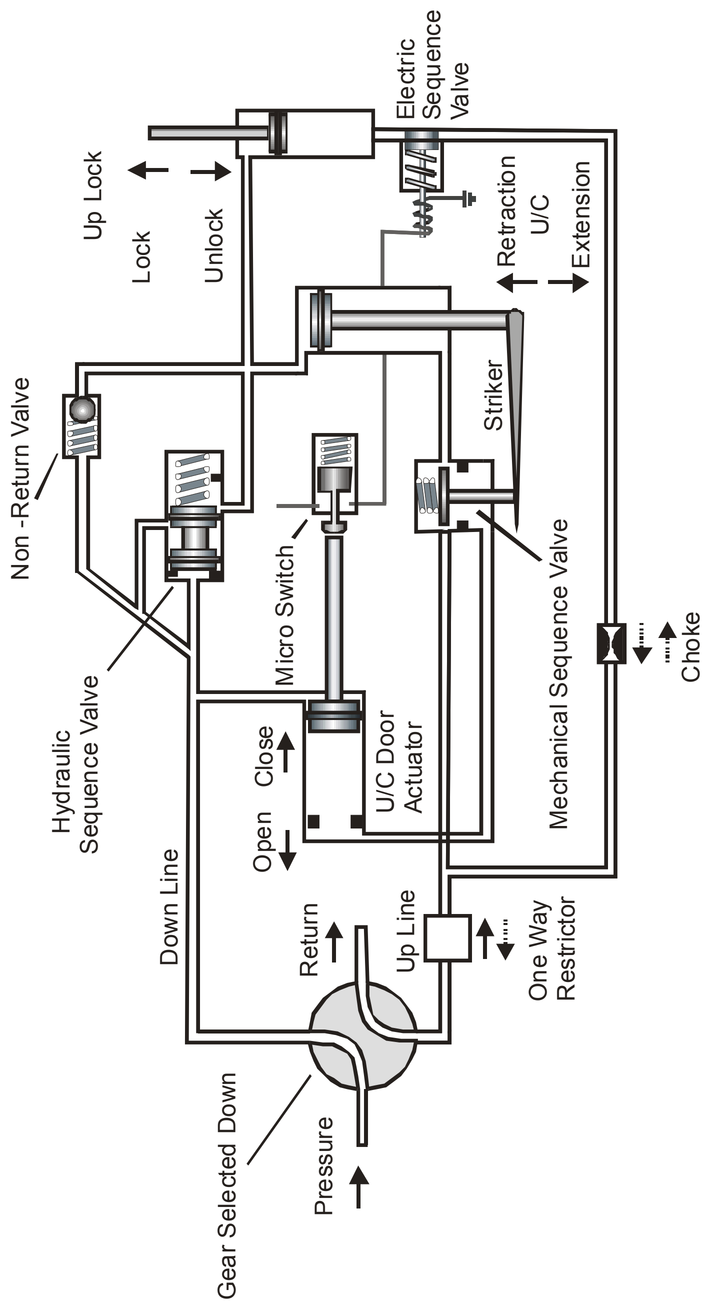


Diagram 9.55 Undercarriage Circuit Showing Sequence Valves

Diagram 9.55 is a created schematic diagram of an undercarriage circuit and only added to show the function and operation of the three types of sequence valves.

The starting condition for this diagram is:

- Gear retracted and locked up, D door closed
- NRV seated
- Hydraulic sequence valve biased by spring blocking port to actuator
- Micro switch depressed creating an open circuit
- Electric sequence valve closed by spring force, creating hydraulic lock in up lock actuator
- Mechanical sequence valve held open by striker
- Pilot just selecting Gear Down, opening the down line to pressure and the up line to return

With the selector in the gear down position, pressure is supplied to the rear of the NRV, the piston face, port of the hydraulic sequence valve, and the down line side of the of the D door actuator. The spring force in the hydraulic sequence valve is greater than the fluid's force. The fluid flow is directed into the door actuator opening the door. As the door starts to move, the micro switch extends and makes the circuit, which energises the electrical sequence valve's solenoid. This redraws the plunger, breaking the hydraulic lock in the up lock. The gear still remains locked up. The fluid expelled by the door actuator is allowed to return to the reservoir via the mechanical sequence valve, which is held open by the striker.

When the door actuator reaches the end of its travel, pressure builds until it overcomes the spring force in the hydraulic sequence valve. The valve is moved to the right and allows the ports to align. Pressure and flow is supplied to the gear actuator and the up lock actuator and is also felt on the ball face of the NRV. Although the fluid pressure each side of the NRV is equal, it remains seated due to the spring force. The up lock moves to the unlocked position and the gear actuator starts to extend. To prevent the fluid displaced by the gear actuator's piston from pressurising the fluid in the up line of the door actuator, which could cause the D door to start to close on the gear as it lowers, the mechanical sequence valve closes as the striker moves away from the plunger. Thus, all the displaced fluid is directed through the up line into the return line and the gear lowering sequence is correct.

When the gear is up, the down line is connected to return and the up line to pressure. As this happens, the spring in the hydraulic sequence valve biases the piston across to the left and blocks the fluid path from the down side of the gear actuator to return. As the mechanical sequence valve is closed, fluid flow is directed into the upside of the gear actuator. The choke in the lock side of the up lock prevents the lock from closing in conjunction with the build up in pressure created by the displaced fluid from the down side of the gear actuator.

As the fluid pressure in the gear actuator's down side increases, it unseats the NRV, allowing a controlled flow back to return. When the gear actuator reaches the end of its travel, the striker pushes the mechanical sequence valve plunger and opens the up line of the D door actuator to pressure. At the same time as the displaced fluid flow from the gear-up actuator ceases, the pressure build up in the up lock actuator closes the lock. The NRV seats follow. With all the flow into the door actuator, the door closes. As the door actuator reaches the end of its travel, it depresses the micro switch breaking the circuit to the solenoid. The spring closes the electrical sequence valve.

SHUTTLE VALVE

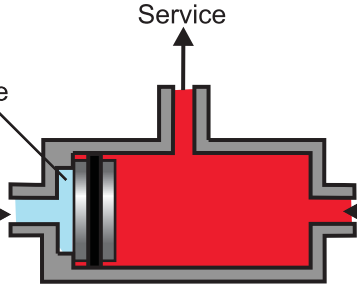


Diagram 9.56 Shuttle Valve

The shuttle valve, as shown in diagram 9.56, allows a service or system to be served by two independent pressure supplies. The normal systems that are considered when thinking about shuttle valves are the flaps and undercarriages lowered by emergency means or the wheel brakes when the normal pressure supply has failed. Refer to diagram 9.44. These can be operated using an alternative hydraulic supply or gas pressure.

> **Note:** The gas (normally nitrogen) is stored in a dedicated flask or bottle. The accumulator pre-charge is not used. Under normal circumstances, the shuttle, a free piston, is held by system pressure at one end of its container allowing the normal supply to enter the system or service.

In the event of supply failure, the pilot selects the alternative or emergency supply. The pressure acts against the piston, moving it across to block the normal supply port and allow the alternative or emergency supply to enter the service. If the service fails to respond after the alternative or emergency supply has been selected, this is most likely the fault of a stuck shuttle.

FLUID JETTISON VALVE

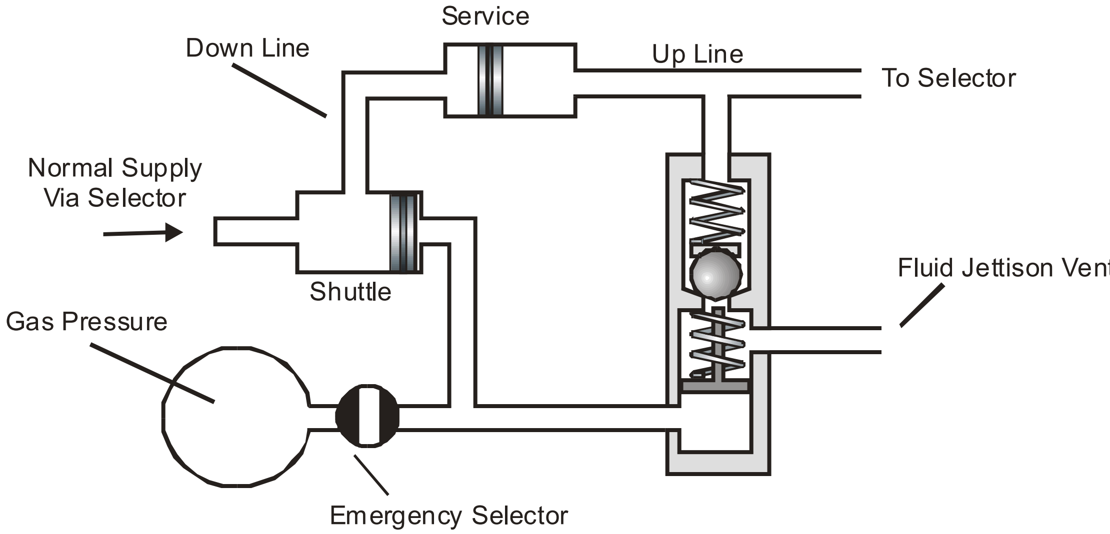


Diagram 9.57 Fluid Jettison Valve

For emergency lowering of the flaps and undercarriage systems where gas pressure is used, there is a need to ensure that the fluid being displaced by the actuator pistons does not create sufficient back pressure to impede their lowering.

To ensure that this does not happen, a fluid jettison valve, sometimes referred to as a fluid dump valve, is located in the up line, as in diagram 9.57. Under normal circumstances, as the diagram shows, the service operates using the normal supply via the selector. The shuttle is across, blocking the emergency supply. The ball of the jettison valve is seated and the emergency selector is off, isolating the gas supply from the system.

If the pilot operates the emergency selector, the gas pressure lifts the lower piston in the jettison valve, unseats the ball, opens the up line to the overboard vent, moves the shuttle across, and operates the service. The displaced fluid takes the path of least resistance and vents via the jettison valve overboard. After the operation of such a system, the hydraulic system must be bled to remove the gas.

PRESSURE REDUCING VALVE

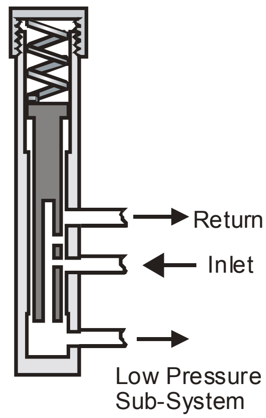


Diagram 9.58 Pressure Reducing Valve

Pressure reducing valves are fitted into the supply from the main system to sub-systems that operate at lower pressures (e.g. wheel brakes on some aircraft). Main system supply enters the valve through the inlet port, flows down through the stem of the valve and into the sub-system. As the pressure rises in the sub-system, the valve is forced up against the spring, reducing the size of the inlet until it matches the flow rate/pressure required by the sub-system. Any increase in sub-system pressure above this value lifts the valve further allowing fluid to return, thus balancing the sub-system pressure. When the sub-system ceases to operate, the increase in pressure under the valve lifts the valve and completely closes the inlet.

THERMAL RELIEF VALVE

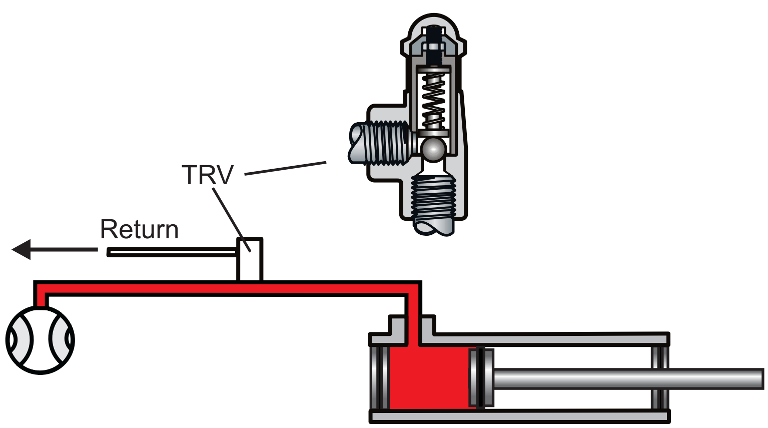


Diagram 9.59 Thermal Relief Valve

Thermal relief valves, as shown in diagram 9.59, are fitted into the system to protect it from the increase in pressure that comes with an increase in fluid temperature through either system operation or climate. They function in the same way as pressure relief valves. These valves are located in between components where fluid is trapped when a hydraulic lock is formed.

As the fluid temperature increases, its pressure increases until the valve cracks allowing a small spurt of fluid to enter the return. The valve then resets and the pressure in the pipe remains at this higher setting. While physically smaller than the PRV, the TRV operates at a higher pressure setting.

DIRECT READING GAUGES

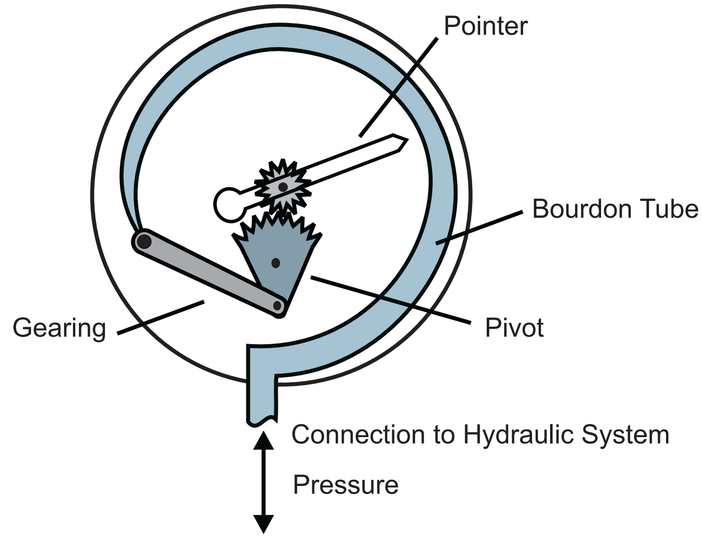


Diagram 9.60 Direct Reading Gauge

In older aircraft, direct reading gauges display the system pressure. These have a Bourdon tube inside them, which is filled with the system's fluid. Any increase in the system pressure also increases the pressure in the tube, causing it to try to straighten. This movement is translated via a gearing into movement of a needle. The rate of the tube's expansion is calibrated to give the reading.

PRESSURE RELAYS

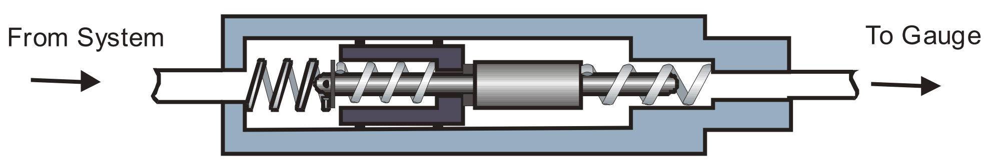


Diagram 9.61 Pressure Relay

Refer to diagram 9.61. Pressure relays are used in conjunction with direct reading gauges. They are fitted into pressure sensing lines between the system and direct reading gauges. The relay, in effect, is a piston within a cylinder. The piston acts to transmit the pressure from the system fluid to the gauge fluid.

The primary reason for fitting a pressure relay into the sensing line is to prevent a continued high-pressure jet of fluid coming from a leak in the sensing line. Leakage above the relay results in the piston moving across and sealing the outlet to the gauge. Gauge indication is lost, but the fluid loss is small.

PRESSURE TRANSMITTERS

Modern aircraft make use of pressure transmitters. These use a pressure-sensing capsule that is subject to the system pressure, compressing or expanding the capsule. Movement of the capsule creates or alters an electrical current, which reconverts into movement at the gauge.

HYDRAULIC FUSE

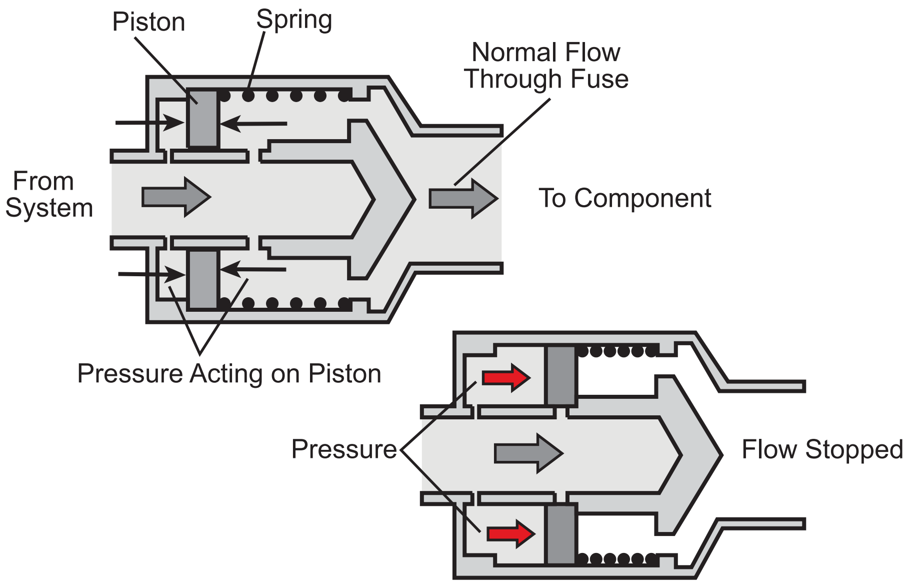


Diagram 9.62 Hydraulic Fuse

Hydraulic fuses, as shown in diagram 9.62, which sense increased flow rate are fitted upstream of components that could be a potential source of an external leak. Under normal conditions, the piston is held against its stops by a combination of fluid pressure and spring force.

If a leak occurs downstream of the fuse, a pressure differential occurs across the piston, resulting in the piston moving across and blocking the flow. While the service downstream of the fuse is lost, the other services supplied by the system remain serviceable.

ACKNOWLEDGEMENTS

We would like to thank and acknowledge:

For diagrams

Diagram 10.1	Mr. Stuart Powney
Diagram 10.5	Key Publications Ltd. for the right to use and adapt a cutaway diagram of a 737-300 aircraft drawn by Mike Bodrocke
Diagram 10.48 Diagram 10.49 Diagram 10.50	Fluid Transfer Ltd.

INTRODUCTION TO FUEL SYSTEMS

This chapter addresses three areas dealing with fuel. These are:

- The types of aviation fuels used and their physical properties
- The types of airframe fuel systems
- The safety regulation regarding refuelling of aircraft

While point two is the main area of the current learning objectives published by the JAA, the first point is added to emphasise the second. The third point is added to put the regulations that are questioned in Operational Procedures in context and to cover for any syllabus drift in the exam questions that could possibly occur.

There are two types of fuel currently used in aviation:

AVGAS — Aviation gasoline for use in conventional piston engines with ignition systems.

Avtur — Aviation kerosene for use in gas-turbine engines and the new diesel engines that are being developed and licensed for aircraft use.

Both fuels are distillates of mineral oil and have different properties. Aviation gasoline is a mixture of lighter hydrocarbons and manufactured to produce a fuel with a specific gravity, or SG, of 0.72 at 15°C. Aviation kerosene is a mixture of heavier hydrocarbons and produced with an SG range of 0.75 to 0.84.

All fuels possess the following properties in varying degrees:

Volatility is the measure of a fuel's tendency to change state from a liquid to a vapour.

Vapour Pressure is the term used to indicate the ambient pressure at which the fuel vaporises. A high vapour pressure indicates that the fuel vaporises at high atmospheric pressure.

Flash Point is the lowest temperature at which there are sufficient fuel vapours, due to the fuel's volatility, above the liquid to ignite. This results in combustion that burns all the vapours seen as a flash. There is not a sustained flame.

Fire Point is the lowest temperature at which the fuel can sustain combustion through vaporisation.

Auto-Ignition Temperature (AIT) is the temperature at which a fuel spontaneously ignites without the presence of an ignition source.

Freezing Point is the point at which the last ice crystal disappears from the fuel as it warms up. As aviation fuels are a mixture of hydrocarbons, they do not become a solid homogenous mass at one temperature like water, but individual hydrocarbons can form ice crystals.

AVIATION GASOLINE — AVGAS

The specifications for aviation gasoline were developed for the earlier carburetted aero engines and have been developed into three grades as engine power has increased. However, the basic specifications remain the same.

Aviation gasoline having a low density (compared to kerosene) is volatile and has a high vapour pressure. Gasoline easily vaporises (evaporates) at sea level on an ISA standard day. As the ambient pressure decreases, the volatility of the fuel increases and is said to be **boiling off** (e.g. at 10 000 ft, the ambient pressure has dropped by approximately 31% of the sea level value).

High ambient temperatures also increase the volatility of the fuel. Therefore, if an aircraft has been parked in high ambient temperatures (heat soaked), takes off and climbs rapidly to altitude, the boil-off is greater than the same aircraft taking off from the same airport when it has not been heat soaked. The actual amount lost through this process is small as it is the lightest hydrocarbons that vaporise at higher gas pressures. It is an indication of another problem called vapour-locking. If the fuel in a pipe or component is hot enough or at a sufficiently low pressure as to be able to vaporise, a bubble of vapour can form, preventing the flow of fuel.

This can be tackled by eliminating places within the fuel system where such vapour locking can occur, and by increasing the pressure within the fuel system to a value greater than the fuel's vapour pressure.

The flash point of AVGAS is approximately –40°C, thus the fire point is within the normal range of ambient atmospheric conditions. The freezing point of AVGAS has to be –58°C or lower.

This has several effects on the fuel. First, it makes starting easier and reduces the likelihood of running problems at low ambient temperatures. However, due to the high vapour pressure, AVGAS can easily vapour lock as temperatures increase or ambient pressure decreases.

The evaporation rate of a fluid is controlled not only by temperature and pressure but also by the surface area of a fluid in contact with the air. In the process of evaporating, fluids draw in heat from the surrounding area, as noticed when the skin is swabbed with surgical spirit before an injection. In a carburettor's venturi where the ambient pressure and temperature decrease as the velocity increases, the mist of fuel that enters the venturi is atomised.

The evaporation decreases the temperature in the surrounding structure further and any moisture that is carried in the air stream freezes on contact with the carburettor. The most likely item for the moisture to touch is the throttle plate. This condition is termed carburettor icing. See carburettor and fuel icing later in these notes for more detail.

STATIC DISSIPATERS

Whereas pure hydrocarbon is a complete insulator, AVGAS is a blend of hydrocarbons, which is not a complete insulator. The problem occurs where fuel is being pumped through pipes and filters with dissimilar materials. It can pick up a static electrical charge faster than the charge can dissipate to the surrounding structure. If the fuel is being pumped from one container to another, it is possible for the charge to spark out of the liquid and ignite the fuel-air vapour above its surface. This hazard increases if the fuel splashes or forms a mist. To overcome this, chemical compounds called static dissipaters can be added to the fuel before delivery to an aircraft. See Refuelling/Defuelling for the correct practice to minimise this potential hazard.

GRADES OF AVGAS

There are three grades of aviation gasoline:

- 80 grade
- 100 grade
- 100LL grade

80 GRADE

80 grade aviation gasoline is coloured red for identification purposes. It is for use in low-powered (low compression ratio) normally-aspirated (carburetted) aero engines. The grade number gives its octane rating, which is an indication of its resistance to detonation.

100 GRADE

100 grade fuel is dyed green for identification and is for use in higher-powered (compression) engines as it is more resistant to detonation. To improve the fuel's resistance to detonation liquid, tetraethyl lead is added. This of course has an environmental impact, so 100LL grade was introduced.

100LL GRADE

100LL is dyed blue for identification and has the same octane rating as 100 grade but has a Lower Lead content and has become the standard fuel grade for high-powered engines (injected, turbo-charged or super-charged engines).

CHEMICAL STABILITY

Where aviation gasoline is stored and has an interface with a change of air, the light hydrocarbons vaporise. The fuel can oxidise and two of the bi-products of this oxidation are soluble gums and insoluble black particulates. The tetraethyl lead oxidises into an insoluble white mass. The particulates and gums can block fuel filters or clog up carburettor jets and narrow bore pipes.

AVIATION KEROSENE — AVTUR

Aviation turbine fuel was developed from paraffin for use in gas-turbine engines. This fuel has a greater density than AVGAS and a lower vapour pressure (about 0.14 psi), so it is less volatile. This, whilst reducing the likelihood of vapour locks occurring and the loss of fluid via boil-off, requires the fuel to be conditioned to ensure that it flows and vaporises when sprayed through special injectors at high pressure into the combustion chamber of the engine.

Having a low vapour pressure results in the fuel having a flash point and fire point that is higher than gasoline (a minimum of 38°C), but it also results in Avtur having a higher freezing point. As with gasoline, the turbine fuels are a mixture of hydrocarbons, each with its own freezing point. As kerosene cools its viscosity increases. This makes it harder to pump and results in a decrease in fuel flow. When the fuel is cooled to its freezing point, the hydrocarbons with the highest freezing point turn into waxy crystals. Further cooling progressively increases the wax content as successive hydrocarbons freeze. This results in an increase in viscosity. However, kerosene remains pumpable to approx 6°C below its freezing point. Continued cooling converts kerosene into slush and eventually becomes a semi solid waxy block.

As it has to be pressurised prior to atomisation in the combustion chamber, the fuel has to pass through a high-pressure pump and fuel-metering system. These are covered in the gas turbine engine notes but are mentioned here to explain why turbine fuel must have lubricant properties. The fuel has to be chemically stable. However, if it is stored over a long period in conditions where oxidation can occur, soluble gums and insoluble particulates form which clog filter units and pipes.

GRADES OF AVTUR

There are three commercial grades of aviation turbine fuel available:

- Jet B
- Jet A
- Jet A1

Jet B

This was the original American aviation turbine fuel. It is what is termed a wide cut, wide range, or wide distillate as it is manufactured from mixing 70% gasoline with 30% kerosene. This has several very important effects:

- A high vapour pressure (2.6 psi)
- A low flash point (below 0°C)
- A low fire point (below 0°C)
- A low freezing point
- A lower SG
- Reduced lubricity
- Greater fuel loss through evaporation

As can be seen from the first three points, Jet B is a greater fire risk, which reduces crash survivability. Its use increases the wear in the fuel components, and if it is added to an aircraft that normally uses Jet A or Jet A1, it requires the fuel control system to be adjusted as its mass per unit of weight is less. Jet B is **not** used as a commercial fuel except in areas such as Alaska or Canada where its lower freezing quality is useful in arctic conditions.

Jet A

Jet A was the next generation of turbine fuel. This is termed **straight** kerosene, as there is no blending with gasoline. The flash point for this fuel is 38°C and its freezing point is –40°C. This fuel is normally used for domestic flights within the USA, as it is cheaper to produce than the current world standard commercial gas-turbine fuel Jet A1.

Jet A1

Jet A1 was developed for use in long-haul high-altitude commercial airliners. This was achieved by lowering the freezing point to –47°C and is considered by the JAA to be the preferred aviation kerosene fuel for commercial aviation worldwide.

FUEL CONTAMINATION

WATER CONTAMINATION

As aviation fuel is produced, it passes through a drying process that removes any water from the finished product. However, fuel is hygroscopic and absorbs moisture from the ambient atmosphere. As with the air, warm fuel supports more moisture than cold fuel and as warm air is more humid, there is more water in the air for the fuel to absorb.

DISSOLVED WATER

Water that is absorbed into a fuel from the water vapour in the atmosphere has been broken down into minute particles. When passing into the fuel, the particles are supported by the fuel due to their size. This is termed **dissolved water** or water in suspension. This water cannot be removed from the fuel without special equipment and causes fuel icing in the carburettor induction system above the throttle plate.

FREE WATER

Dissolved water that has precipitated out as the fuel cools (saturation point), or water that has entered the fuel on mass, due to its greater mass, collects at the bottom of the tank. This is termed **free water**. One of the mechanisms by which free water enters fuel is seen by the condensation that forms on the outside of a glass containing an iced drink or a chilled can taken from a refrigerator on a hot day. As fuels, especially AVGAS, are volatile, they absorb heat from the fuel tank structure in the process of their evaporation. Thus, any moisture in the airspace above the fuel condenses out and either runs down the tank walls through the fuel or drips into the fuel where it settles out at the bottom of the tank.

Tanks must have vents to ensure that a partial vacuum is not created as fuel is drawn off. For light aircraft with un-pressurised fuel tanks, this ensures that the air space is maintained at the ambient pressure at which the aircraft is flying, so that a pressure differential does not cause structural damage to the tank. There is always an interchange of moisture-laden air into the tank system. Therefore, from a water contamination perspective, it always pays to leave the aircraft with full fuel tanks to minimise the air gap above the fuel.

FUNGAL CONTAMINATION

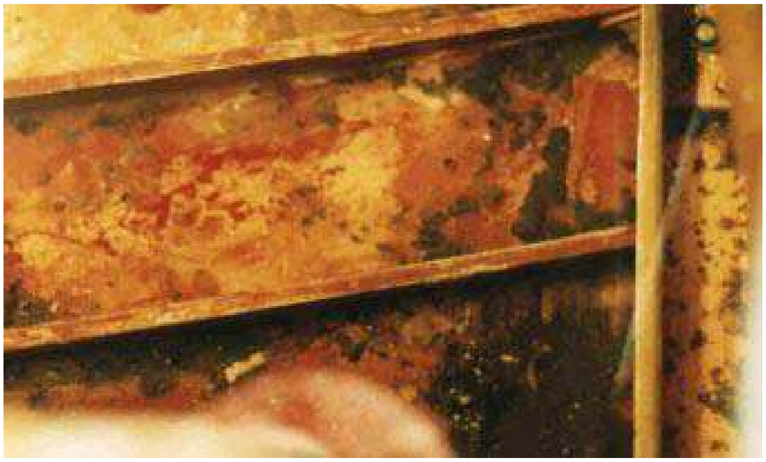


Diagram 10.1 Corrosion Caused by Cladisporium Resinae in an Integral Tank

An airborne fungus called cladisporium resinae can contaminate fuel tanks. The fungus exists in the fuel/water interface living in the fuel and deriving food from the hydrocarbon fuel. The fungus, which takes the form of blanket weed (as found in a pond), traps water within its mass of strands and can, therefore, survive for a prolonged period of time after the free water has been drained off until it builds up again. The fungus can block fuel filters and freeze due to the trapped water content blocking fuel pipes and filter units. It can cause corrosion also due to the water content where it remains in contact with alloy fuel tank structure. Refer to Diagram 10.1 where the corrosion appears as black patches.

For the JAA exams, the fungus is considered to be the particular problem for aircraft that use kerosene and have integral tanks. Although in reality, the fungus can affect gasoline-burning aircraft. To prevent infestation, a biocide can be mixed with the fuel. This also controls small, established amounts of the fungus. For large-scale infestations, the fuel system would have to be stripped and cleaned.

WATER SEDIMENT CHECKS

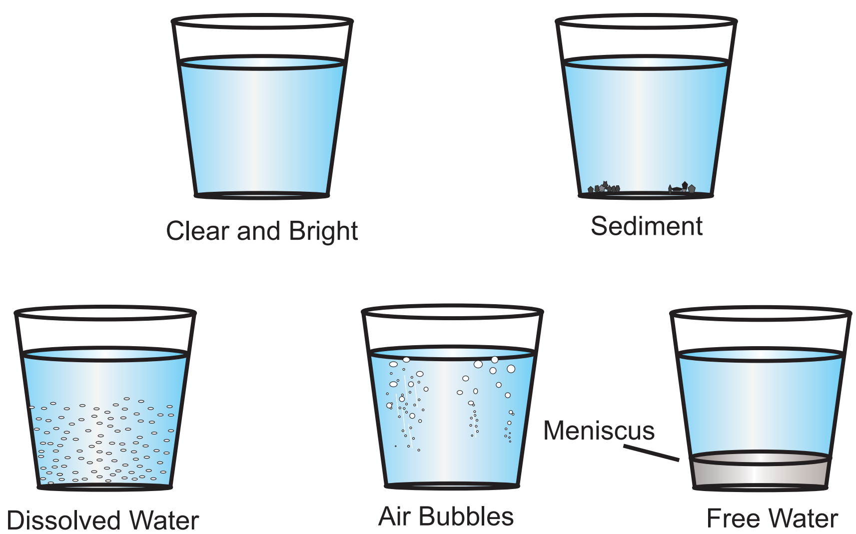


Diagram 10.2 Clear and Bright Water Sediment Checks

As detailed above, water in the fuel system can literally be a killer and has to be guarded against. The pilot’s main weapon in this is vigilance by carrying out water sediment checks before flight and by uplifting fuel from recognised suppliers who have a large volume turnover. This should ensure that if a problem exists, it is spotted before flight, and that the supply is unlikely to be contaminated with either water or fungus, as it does not remain in the fuel supplier’s depot very long.

For the water sediment check, a sample of fuel is drawn off in turn from the lowest point of each tank and the fuel filter(s) into a clear walled container. This is observed and held up to a good source of light for the clear and bright test. Refer to diagram 10.2 and the following text.

Clear refers to checking the fluid for traces of sediment, particulates, and other solid matter.

Bright refers to checking the fluid for traces of free water and dissolved water, which can also be termed **entrained water**.

When held up to the light, if the fluid has a hazy appearance, it can be due to the light refracting and reflecting as it passes from fuel into water and back into the fuel again. If the haziness clears from the bottom upward, then it was air that has become entrained into the fuel as the sample was taken. If the haziness remains or slowly starts to clear from the top of the sample downward, this is entrained water.

Free Water settles to the bottom of the sample, and the interface between the two fluids is seen as a meniscus.

Where sediment or water is found in the fuel samples, the affected tank(s) filter(s) must be drained off until a pure fuel sample is taken.

FUEL TANKS

TYPES

There are three types of fuel tanks used in aircraft for storing fuel:

- Rigid tanks
- Flexible tanks
- Integral tanks

RIGID TANKS

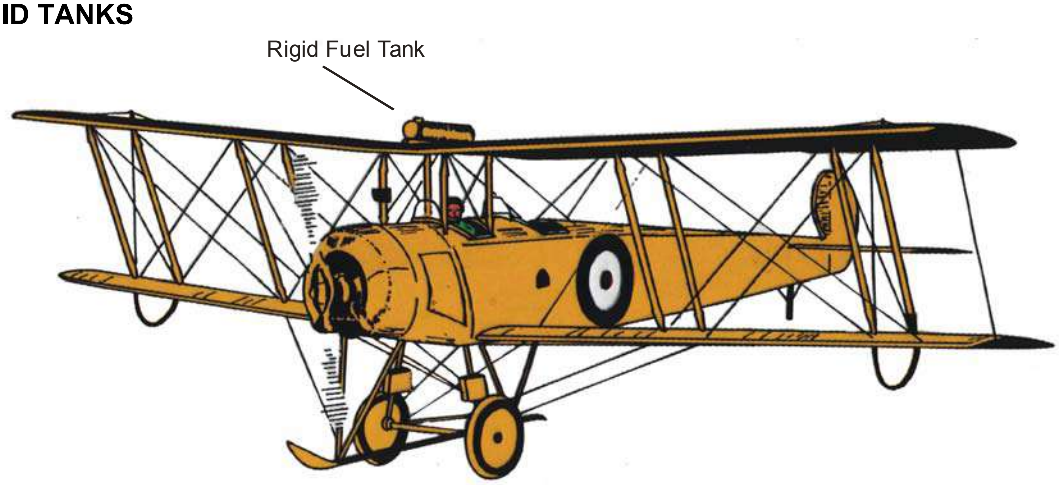


Diagram 10.3 Externally Mounted Rigid Fuel Tank

Rigid tanks were the first type of fuel tank used on aircraft, as shown in diagram 10.3. The fuel is carried in an externally mounted rigid tank allowing a gravity feed to the engine. While modern aircraft can incorporate rigid fuel tanks fitted internally, they are no longer the main fuel storage system as they have the following disadvantages:

- The rigid fuel tank and the extra strengthening required adds to the overall mass of the aircraft and reduces the useful load of the aircraft.
- As they are built as a separate structure, if they are large, they have to be built up in situ as the aircraft is manufactured, making maintenance replacement awkward.
- For practical and economic purposes, rigid tanks can only be fitted where there is sufficient space available for a large uncomplicated shape to be manufactured.
- They have to be tied to the aircraft's structure. This requires the surrounding structure to be strengthened to support the added mass and cater for the acceleration/de-acceleration loads imposed by the fuel tank and its contents.

FLEXIBLE TANKS

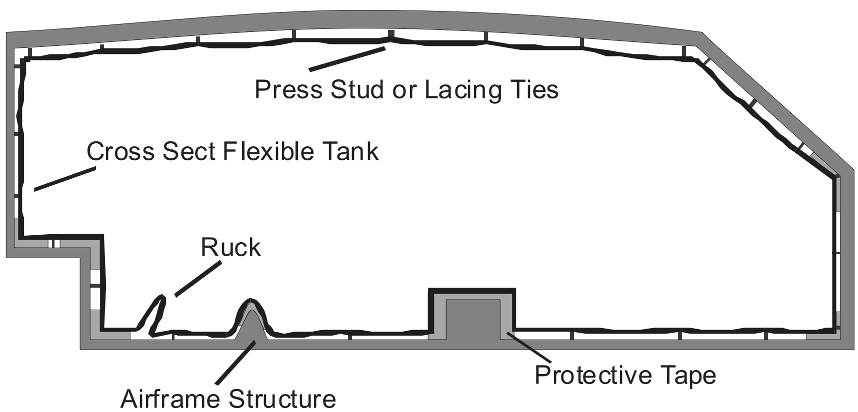


Diagram 10.4 Flexible Fuel Tank

Flexible tanks are made from a fuel resistant rubberised fabric, which has the advantage of being lighter than the comparable rigid tank. They can be manufactured to fit into areas where it would not be practicable to produce and fit a rigid tank. Diagram 10.4 shows this type of tank. Being flexible (to an extent) they are more crashworthy than either rigid or integral tanks provided that they do not get punctured.

Some tanks have an external layer of high-density closed cell foam that swells when in contact with fuel, so it self-seals (reducing the size of the leak). While modern aircraft can incorporate flexible fuel tanks fitted internally, they are no longer the main fuel storage system as they have the following disadvantages:

- There is no guarantee that the bottom of the tank is flat. It can be rucked, causing ridges that trap water.
- The fuel from a leaking flexible tank can run down the internal structure before showing on the exterior of the aircraft.
- The areas in which they are fitted have to be lined with tape to prevent any sharp edges from puncturing them.
- They have to be clipped or tied to the surrounding structure to preventing them from collapsing as the fuel is used.
- Once used with fuel, the tank must not be allowed to dry out as it can split and leak.
- Over a period of time, these tanks can become porous, so they have a finite life.

INTEGRAL TANKS

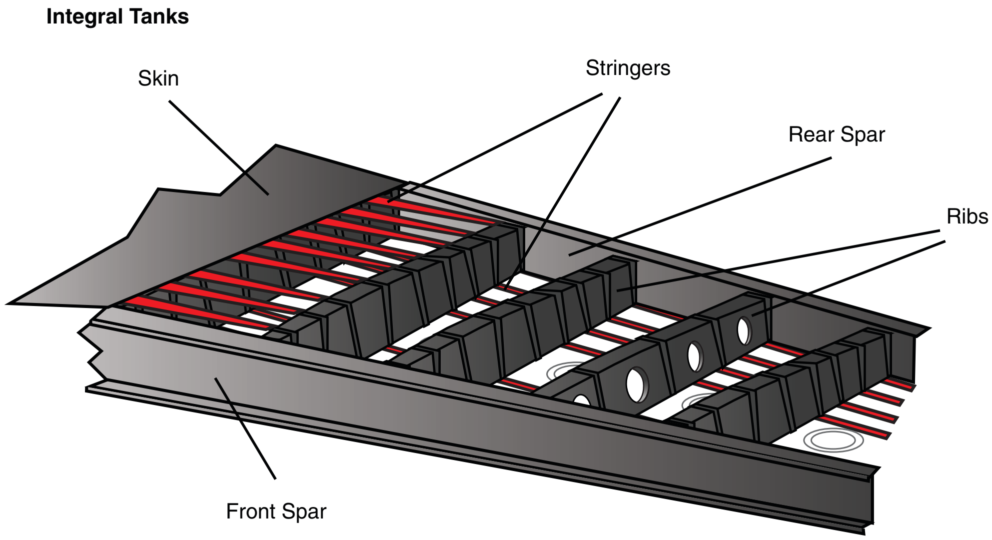


Diagram 10.5 Integral Tank made by sealing the joints and fastenings of wing's structure

Modern air transport aircraft use integral tanks as their main fuel storage system. These are formed by internally sealing structures. This system allows manufacturers to utilise areas that would otherwise not be viable as part of the aircraft’s fuel system.

As the main areas used are the aircraft’s wings, the system has been termed **wet wing**. In this system, sections of the wing’s structure are converted into fuel tanks. This is achieved by coating the joints with two or more layers of rubberised sealant, which can be painted on. The system has the advantage of being lighter and making use of structural bays that would otherwise be left as voids. The disadvantage, as with the flexible tank, is that the rubberised sealant must not be allowed to dry out. Otherwise, it can crack and cause the tank to leak.

This weight reduction in turn allows either more fuel to be carried to extend the aircraft’s range or more payload in place of the extra fuel. For modern turbine-powered air transport aircraft with thinner wing sections, the utilisation of the wet wing system allows fuel to be stored further toward the tips. Increasing the weight at the tips, counters the upward bending action created by lift.

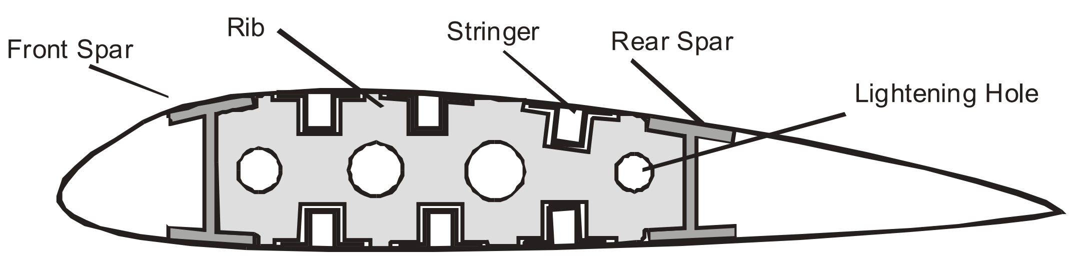


Diagram 10.6 Rib Acting as a Baffle Plate

As most modern air transport aircraft have wings that are straight or slightly dihedral, the fuel runs from the tips toward the wing root. Any manoeuvre that causes a wing to drop results in the fuel running back toward the wing tip. To control the movement of fuel within the tank, baffles can be fitted. In diagram 10.6, showing a cross section of an integral tank, the ribs have the same effect. Here the fuel is able to flow past the rib, as there are gaps between it and the surrounding structure.

FLAPPER VALVE

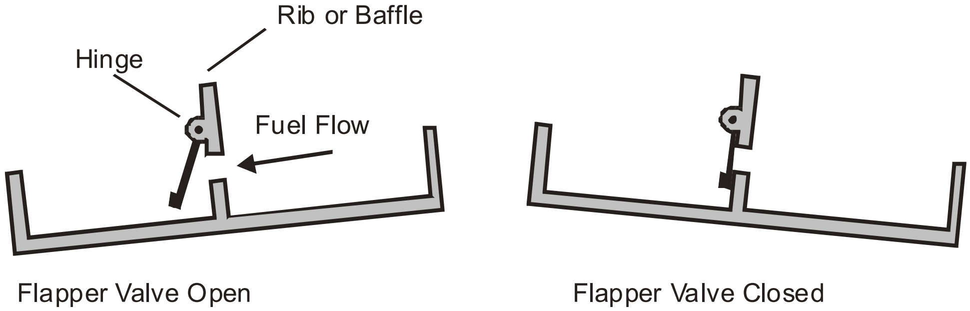


Diagram 10.7 Flapper Valve

Where the ribs or baffle plates are completely sealed and there is no provision for lightening holes, **flapper valves** are fitted to allow the fuel to pass in one direction only, as diagram 10.7 shows. The left diagram shows the flapper valve in the open condition when the aircraft is in a straight and level condition, allowing fuel to flow down to the fuel collector and then on to the engine. The right diagram shows the flapper valve closed when the aircraft's wing tip is lower than the fuel collector, trapping the fuel in the collector. This prevents fuel starvation of the engine.

UNUSABLE FUEL, STACK PIPE, AND SUMP

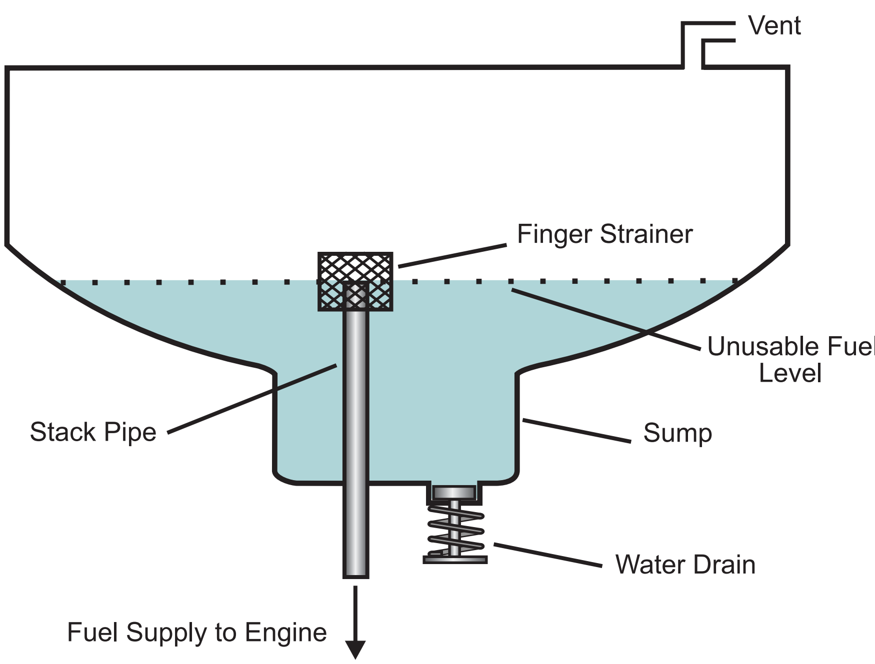


Diagram 10.8 Sump, Water Drain, Stack Pipe, Finger Strainer, and Vent

To prevent the risk of engine failure due to water or contaminants passing through the fuel line, the fuel supply to the engine is taken from above the bottom of the tank. Diagram 10.8 shows it being drawn off via a stack pipe. This leaves a volume of fuel that cannot be used by the engine and is termed unusable fuel. The aircraft's pilot's manual specifies this volume and its mass. To prevent any large item of debris that would block the fuel line, a coarse mesh (gauze) filter, referred to as a **finger strainer**, is fitted over the inlet of the stack pipe.

SUMPS

Each fuel tank must have a low point, which is termed a sump, to which any free water can drain down and can be drained from via a water drain valve. In diagram 10.8, the water drain valve is shown fitted in a distinct sump as per a rigid tank. For integral tanks, the sump is formed by the dihedral of the wing and the height of the take-off pipe above the bottom of the tank. For JAR 23 certified aircraft, the sump has a minimum capacity of 1/16 US gallon or 0.25% of the tanks capacity, whichever is greater.

WATER DRAIN VALVES

Water drain taps are fitted at the lowest point of the fuel tank and in filter units to allow any free water to be drained off when the pilot carries out the water sediment checks prior to flight. The water drain cock is sprung loaded to close and has to be pushed upward to open. Pilots must ensure that after operating the water drain tap, it seats correctly and stops the flow of fuel.

GRAVITY FEED FUEL SYSTEMS FOR A SINGLE-ENGINE LIGHT AIRCRAFT

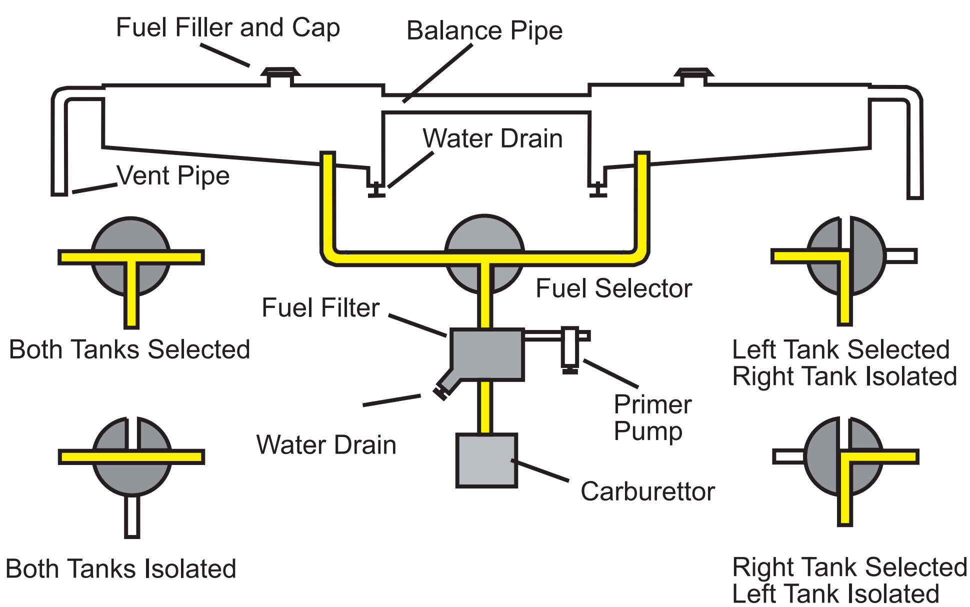


Diagram 10.9 Schematic of a Gravity Feed Fuel System

Gravity feed and pumped are two types of fuel systems. .High-wing single-engine light aircraft use the former, shown in diagram 10.9. These systems are used where the vertical height between the fuel tank and the engine's carburettor is sufficient to produce a sufficient head of pressure to supply the carburettor's float chamber at 1.5 times the flow rate required by the engine at take-off power. The latter are used where there is an insufficient head of pressure or the engines are mounted above the fuel tanks.

In the system, as shown by diagram 10.9, a rotary fuel selector allows the pilot to select both tanks simultaneously, or either the left or right tank individually, as well as isolate the fuel supply from the engine. Each tank has a vent pipe to ensure that the tank can vent to atmosphere. A pipe connecting the airspaces of both tanks together, termed a **balance pipe**, ensures that where either vent becomes blocked, fuel can be drawn from the tanks.

The fuel is passed through a filter unit upstream of the carburettor. This is also termed a **gascolator** (an American term) or **fuel strainer** (a British term). These are designed to remove fine particles that could block the carburettor's jets and any free water.

PRIMER PUMP

Hand-operated priming pumps are sometimes fitted to aircraft for cold weather starting or as an optional extra. They draw fuel from the top of the gascolator and deliver it into the cylinder head just before the inlet valve. The primer pump works on a pull-push action (in the same manner as a bicycle pump). Pulling the plunger out fills the pump, while pushing the plunger in delivers the fuel. If the pump's plunger is not locked in the closed position when the engine is running, the suction created in the manifold can draw fuel through the primer pump and cause the cylinders that are primed to run rich. See engine notes for further details.

MANUALLY OPERATED FUEL SELECTOR

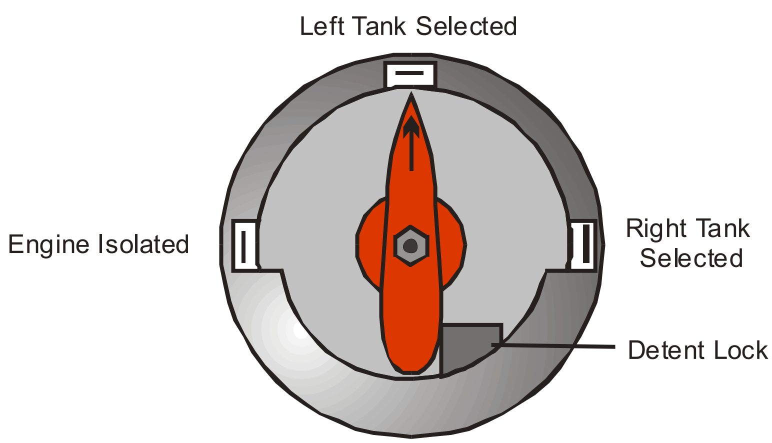


Diagram 10.10 Fuel Selector for a Light Aircraft

Diagram 10.10 shows a rotary selector of the type used on a Piper PA 28. It meets the JAR 23 requirements that the selector cannot be moved to off unless the detent lock is pushed down.

For light aircraft manufactured under JAR 23 using a manually operated fuel selector, the selector must comply with the following requirements:

- The fuel selector can be seen and operated by the pilot in flight without having to move the seat or the primary flight controls.
- Each engine can be isolated from or connected to the fuel supply in flight.
- The 'off' selection selector requires a separate and distinct action.
- The tank and off positions are marked and detented.
- The selector does not pass through 'off' when moving from one tank position to another.
- The selector's handle forms a pointer and indicates the position of the valve.

Where an aircraft has wing-mounted fuel tanks as per diagrams 10.9 and 10.11, the pilot uses the fuel selector to control the lateral movement of the aircraft's centre of gravity.

PRESSURE FEED FUEL SYSTEM FOR A SINGLE-ENGINE LIGHT AIRCRAFT

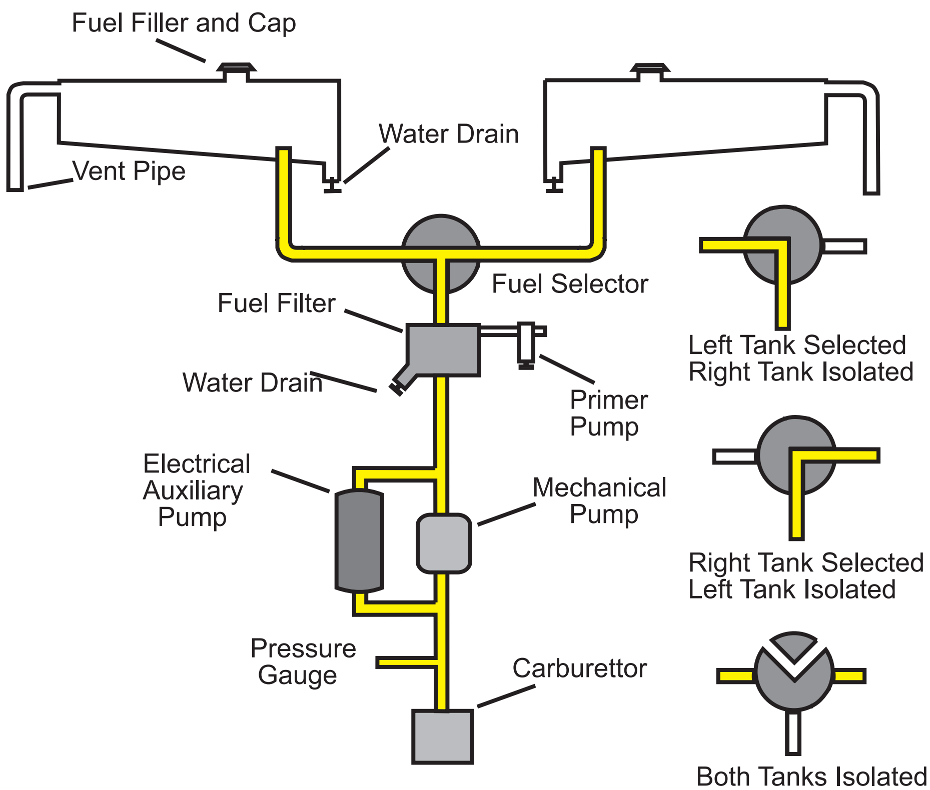


Diagram 10.11 Schematic of a Pump Fed Fuel System for Low-Wing Single-Engine Aircraft

For aircraft where the engine is mounted above or on the same level as the fuel tanks, the fuel system has to be able to deliver the fuel at all normal attitudes of flight. To meet this requirement, a mechanical pump operated by the engine draws the fuel from the tanks and supplies it under pressure to the carburettor.

To ensure that the engine continues to function in the event that the mechanical pump fails, an auxiliary electrical pump is mounted in parallel with the mechanical pump.

AUXILIARY FUEL PUMP

The auxiliary pump is selected on:

- For take-off
- For landing
- For tank changes
- In the event of low fuel pressure readings
- In the event of the engine running rough

By selecting the auxiliary pump ‘on’ during take-off and landing, the aircraft is protected from a sudden mechanical pump failure at a critical stage of flight. Using the auxiliary pump during tank changes ensures that a positive fuel flow is established at the point of change over.

FUEL PRESSURE INDICATIONS

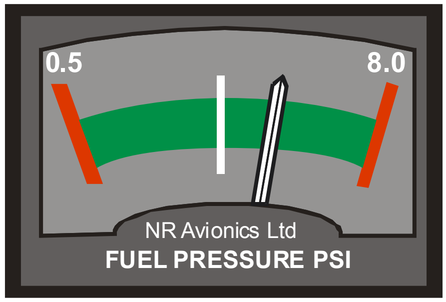


Diagram 10.12 Light Aircraft Fuel Pressure Gauge

For any fuel system that uses pumps to supply a carburettor or injector system, there has to be an indication of the supply pressure. Diagram 10.12 shows the type of gauge used on a small light aircraft system, as per diagram 10.11. This is a direct reading gauge where the fuel pressure delivered into the carburettor is tapped off just before the carburettor’s inlet and displayed to the pilot. These gauges are marked with the minimum and maximum fuel supply pressures for the engine, shown as red radials or red lines. A green arc depicts the normal operating pressures range. The standard fuel pressure range for a normally-aspirated light piston engine is between 0.5–8.0 psi.

The fuel system has to be designed so that the maximum supply pressure is not exceeded if both the main and auxiliary pumps are working normally and have been selected simultaneously.

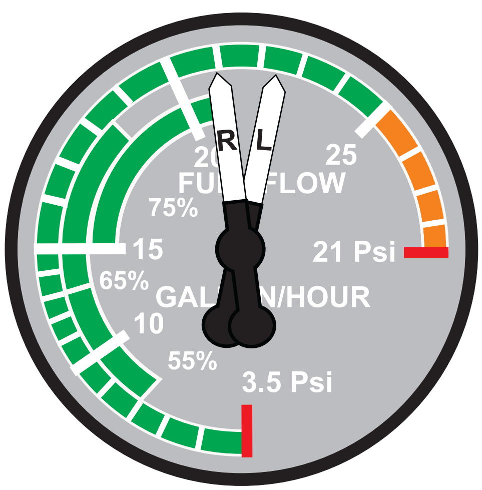


Diagram 10.13 Light Twin-Engine Aircraft Fuel Flow / Pressure Gauge

For more complex light aircraft (fuel-injected and turbo-charged), the requirement for fuel pressure indication can be combined with indications of fuel flow. As for any given temperature, the higher the fuel delivery pressure, the greater the fuel flow. Diagram 10.13 shows the type of gauge that can be used with twin-engine aircraft where a single gauge can have two needles. If both engines are set up to run equally, one needle is superimposed on the other. Any split between the needles shows a mismatch in conditions.

Referring to diagram 10.13, the upper and lower supply pressures are depicted with red radials. The outer green arc, calibrated in gallons per hour, shows the normal operating range. The amber arc depicts the cautionary range between the green arc and the upper radial. The inner green arcs show the ranges of fuel flow that can be set when percentages of power are used. For example, 75% power and fully rich have a greater fuel flow than 75% power and fully lean.

VAPOUR LOCK

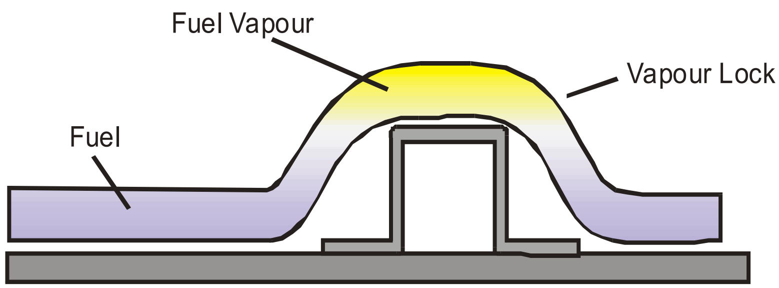


Diagram 10.14 Vapour Lock in a Fuel Pipe

One of the causes of a low fuel pressure reading is the formation of a vapour lock. Diagram 10.14 shows the formation of such a vapour lock, where the fuel vapour is at the same pressure as the fuel. This has collected in a bend of the fuel pipe and greatly reduces or completely stops the flow of fuel. Selecting the auxiliary pump increases the pressure in the fuel line, which should suppress the formation of the vapour and force the fuel through.

The most likely conditions for vapour locks forming are:

- High fuel temperature
- The critical fuel level, normally a low fuel level
- Low ambient pressure
- High angles of attack

These can be termed **critical operating conditions**.

FUEL SYSTEM FOR A LIGHT TWIN ENGINE AIRCRAFT

For light twin-engine aircraft, it is normal for the engine to be supplied with fuel taken from the tank(s) on the same wing. However, the system is designed for the fuel from one wing to supply the engine on the opposite wing. This is termed **cross feed** and is often abbreviated to **X feed**. Diagram 10.15 shows a schematic diagram of a light twin-engine aircraft.

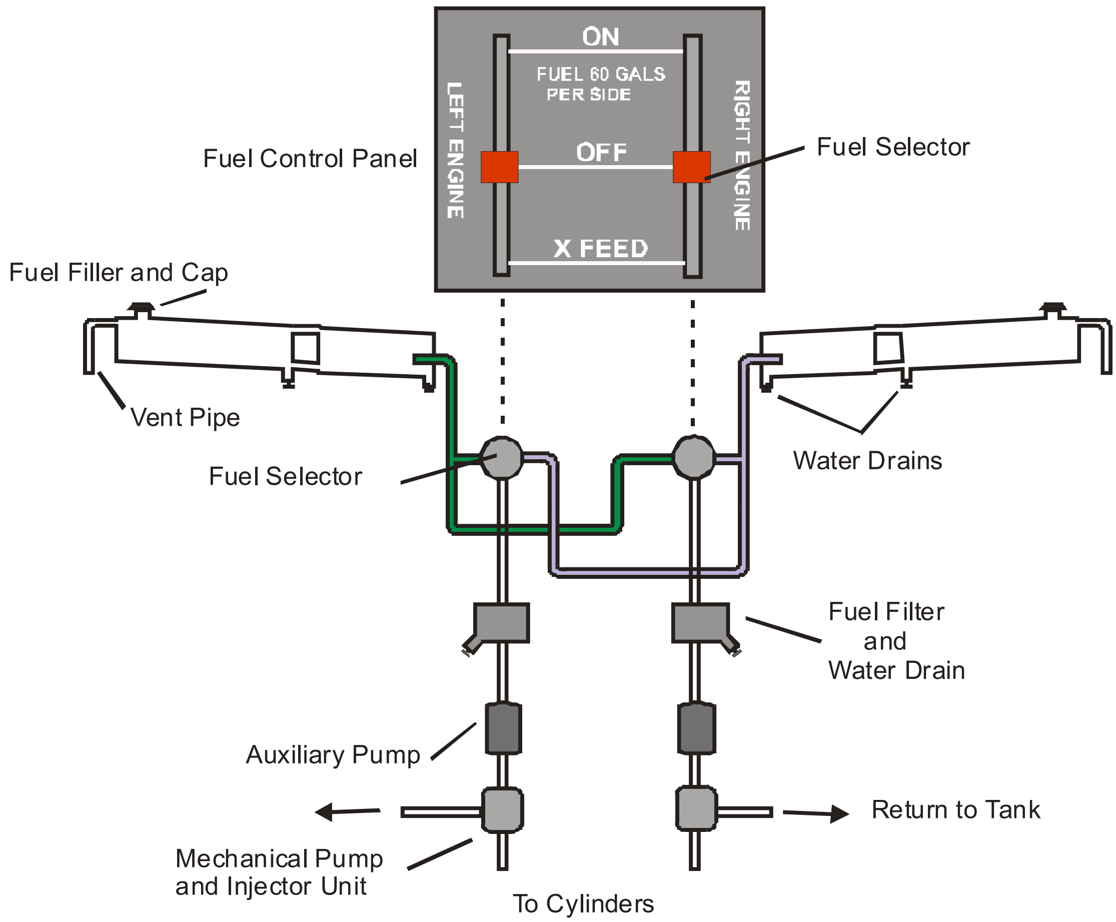


Diagram 10.15 Schematic of a Fuel System for a Light Twin

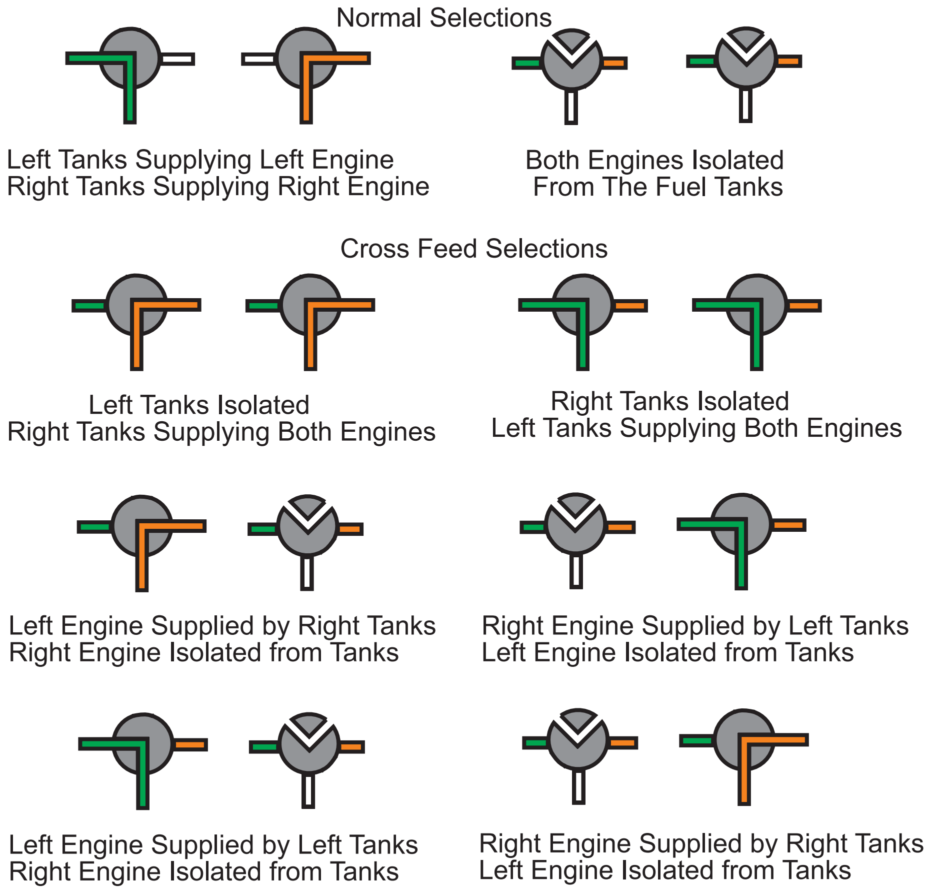


Diagram 10.16 Schematic for a Light Twin-Engine Aircraft

The regulations governing the use of cross feed are:

- Aircraft are not to operate with both selectors set on X feed as diagnosing a fuel-related problem and taking remedial action would be very difficult.
- Aircraft are not to take off with X feed selected.
- Under normal circumstances, aircraft are not to land with X feed selected.
- Cross feed is used to control fuel imbalances by supplying both engines from the heaviest wing.

In the event of a single engine failure, the remaining engine can draw the fuel from the dead engine's tank. This is used to control the lateral imbalance, increase endurance and extend range.

On aircraft that have fuel injector systems, or carburettor systems that have a vapour return facility, the mechanical fuel pumps draw more fuel from the tanks than the engines use except when set at full power and fully rich. The excess fuel is returned to the first tank to be used on the same side as the engine.

The aircraft's handbook specifies how much fuel must be used from the working engine's tanks before the engine can be cross-fed from the dead engine's tanks (normally specified as a length of time that must elapse). Failure to observe this, results in the live engine's tanks filling to capacity and the returned fuel being lost overboard via the tank vent.

VENTING OF LIGHT AIRCRAFT TANKS

For light aircraft with un-pressurised fuel tanks, there has to be an expansion space above the fuel of not less than 2% of the tanks total volume. This has to be vented to atmosphere via a vent pipe or in some designs, vented filler caps are used.

The requirements for fuel vents are:

- They must be designed and located to minimise the possibility of them becoming blocked with ice or debris.
- The capacity of the vent must be large enough to rapidly relieve any excessive pressure differential between the interior and exterior of the tank.
- The vent must prevent siphoning or loss of fuel during normal operation. Fuel may discharge through the vent due to thermal expansion only.
- Where more than one tank is used to supply an engine and they are interconnected, as per diagram 10.16, the expansion space must also be interconnected.
- The vent pipe must be designed to prevent the accumulation of moisture either on the ground or in flight. If this is not possible, a drain for the vent pipe must be fitted.
- The fuel venting must be arranged so that any discharge of fuel or fumes cannot enter the cabin or constitute a fire hazard.
- For aerobatic aircraft, the venting system must be designed to prevent excessive loss of fuel during manoeuvres including short periods of inverted flight. They must also prevent any continued loss of fuel due to siphoning after completing any manoeuvre.

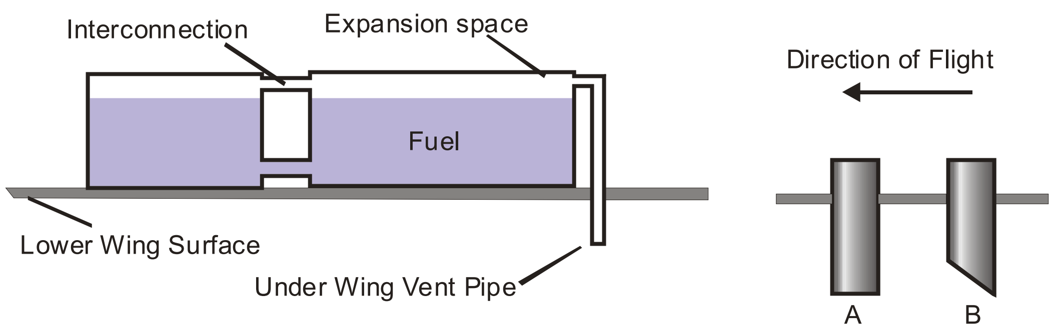


Diagram 10.17 Under Wing Vent Pipes

Diagram 10.17 depicts an under wing vent pipe system that complies with the requirements given above. As the vent pipe protrudes from beneath the wing, it is clear of any ice accumulations building up on the under surface of the wing, and the pipe drains any moisture that collects in it to atmosphere.

By terminating the vent pipe with a forward facing chamfer, as per B of diagram 10.17, the air space above the fuel is slightly above ambient pressure due to the ram effect into the vent pipe. This reduces the loss of fuel through evaporation and applies a slight head of pressure into the fuel pipe, aiding the pump and reducing the likelihood of vapour locks.

LIGHT AIRCRAFT FUEL FILLERS AND CAPS

Light aircraft use the same system of refuelling that is currently used on cars. A filler cap is removed from an aperture at the top of the fuel tank, and the fuel is delivered via a nozzle connected to a flexible hose. This is termed **open line refuelling**. The safety issues regarding open line refuelling are covered later in this chapter.

A fuel tank filler cap must provide a fuel-tight seal, and the fuel tank filler connection must prevent the entry of fuel into any part of the aeroplane other than the tank itself. If the filler point is recessed, the recess must have a drain that discharges clear of the aeroplane. Each fuel filling point must have a provision for electrically bonding the aeroplane to the ground fuelling equipment.

Each fuel tank filler connection must be marked as prescribed in *JAR 23.1557 (c).*

- For gasoline powered aircraft, the word **AVGAS** and the minimum fuel grade must be marked adjacent to the fuelling point.
- For turbine-engine powered aeroplanes, the words **Jet Fuel** and the permissible fuel designations, or references to the Aeroplane Flight Manual (AFM) for permissible fuel designations, must be marked adjacent to the fuelling point.

LIGHT AIRCRAFT FUEL QUANTITY INDICATIONS

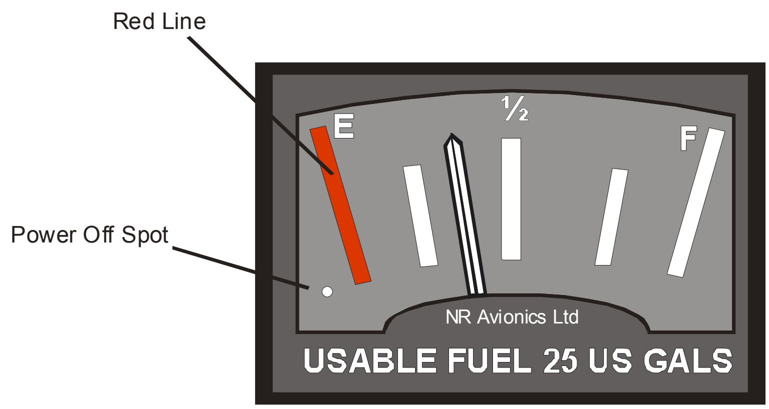


Diagram 10.18 Light Aircraft Electrically Operated Fuel Gauge

Each separate tank that feeds an engine must have a fuel gauge. Where two tanks are interconnected, as per diagram 10.17, they count as a single tank and only require one gauge. Thus, the light twin, as depicted in diagram 10.15, requires two gauges. Each gauge must have a calibrated red line or red radial showing that the tank is empty of usable fuel when in straight and level flight, as in diagram 10.18. The majority of modern gauges are operated electrically. When power is removed, the gauge needle moves to a point below the empty mark. Diagram 10.18 shows a power off spot. The needle would park here when power is turned off.

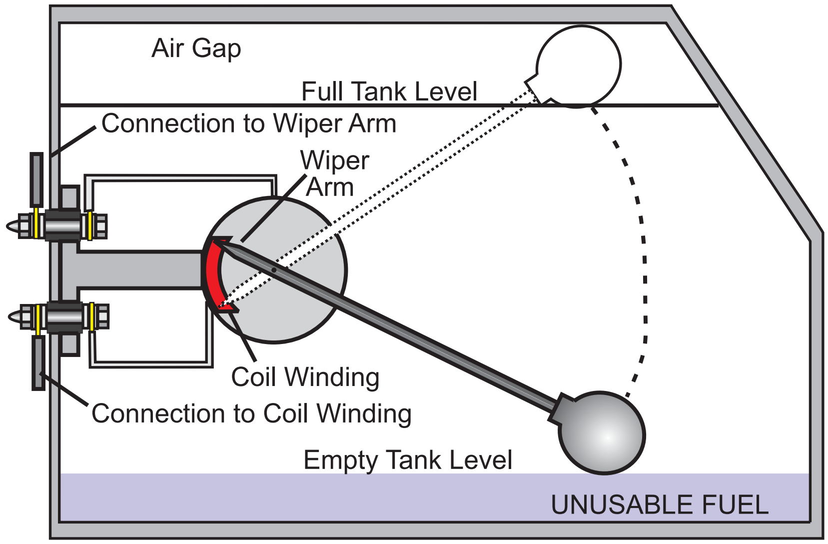


Diagram 10.19 Fuel Tank Sender Unit

For cost and simplicity, the majority of light aircraft fuel gauges are operated by float switches termed **sender units**. Diagram 10.19 shows such a float switch. As fuel is an electrical insulator, very low voltages and amperages can be used within the fuel tanks. However, all items must be intrinsically safe. They must not produce a spark that could ignite the fuel/air vapour mix in the tank in the event of failure.

In these systems, the float is linked to a wiper arm, which passes across a coil winding. As the fuel level increases or decreases, the float's movement alters the position of the wiper arm.

This can be utilised in two ways, either to give a voltage resistance reading or a voltage potentiometer. Light aircraft normally utilise the voltage resistance for fuel indication.

RESISTANCE FUEL GAUGING SYSTEM

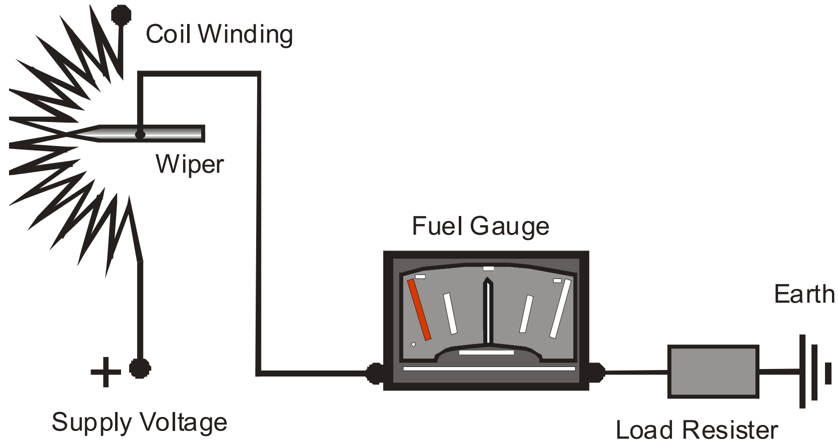


Diagram 10.20 Resistive Fuel Gauge Circuit

Diagram 10.20 illustrates a resistive fuel gauge circuit. In this system, a voltage is applied to the coil winding, and the wiper arm is connected to earth via an ammeter and load resister. As the wiper arm moves across the coil winding, it varies the resistance. This is shown as a deflection on the ammeter, which is converted into a fuel contents gauge by having a dial or face that is calibrated with units of volume.

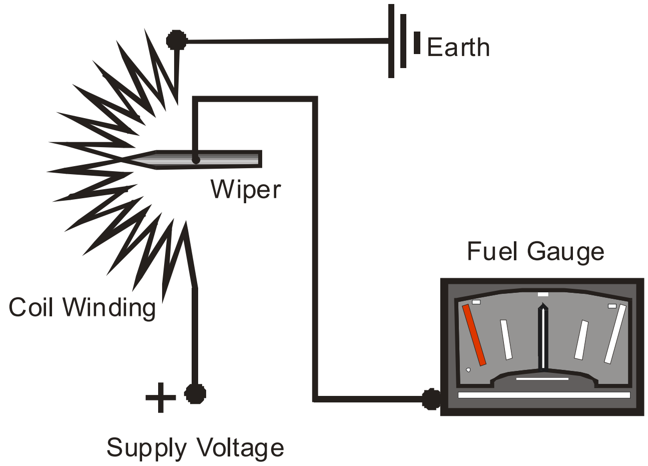


Diagram 10.21 Float Gauge Circuit as a Potentiometer

As a potentiometer, the circuit is altered as per diagram 10.21. In this system, the coil winding is connected to earth, and the winding acts as a resister dropping the voltage by a set value at each point. As the wiper arm passes across the coil winding, it detects the voltage at that exact point. This is shown on a voltmeter that again is calibrated with units of volume.

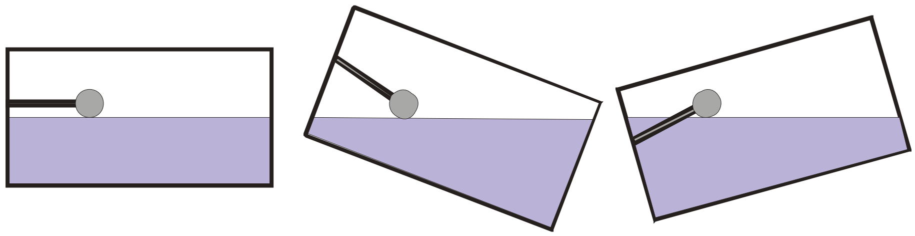


Diagram 10.22 Effects of Manoeuvres on Fuel Indications

As the resistive system uses a float to find the level of the fuel, it is only ever accurate when the aircraft is stationary on the ground or in straight and level un-accelerated flight in still air. Refer to diagram 10.22, where the tank is half full. Any manoeuvre or acceleration which causes the fuel to move within the tank can give false indications.

Large aircraft do not use the resistive system (volume) due to this lack of accuracy. They make use of a system that determines the contents of the fuel tank as a mass via the fuel's density. This is termed a **capacitive** system and is covered later in these notes.

A further disadvantage of the volume system of tank contents gauging is that the volume of a fluid alters with temperature by expanding with an increase in temperature and contracting with a decrease in temperature. If a light aircraft is refuelled when the fuel is cold and is parked in a warm hanger (overnight), this expansion normally is seen as a puddle of fuel on the floor under the fuel tank vents. This is irritating, a waste of money, and a potential fire risk but not a serious risk to flight.

The problem for larger aircraft is that with their larger tanks, the contraction of the fuel due to low temperatures allows a much greater mass of fuel to be loaded for any given volume. Therefore, if all the tanks were filled to capacity on a cold night and the aircraft were to take off in the early morning before the ambient temperature had risen, the aircraft would be heavier than anticipated and would suffer degradation in its performance.

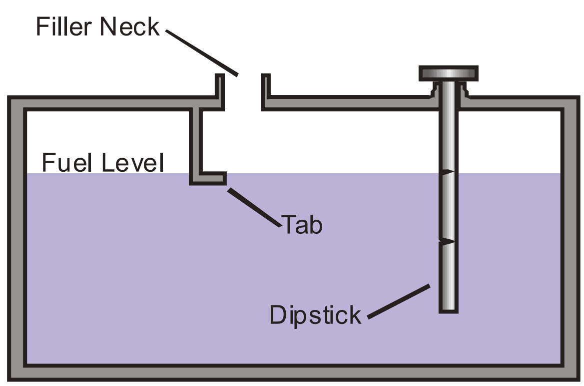


Diagram 10.23 Dipstick and Tabs for Light Aircraft

There is also a requirement for aircraft to have a manual means of determining the fuel contents when the aircraft is on the ground. For light aircraft, this requirement is normally satisfied by the use of a dipstick or fixed tabs located within the tank and visible through the filler orifice when the cap is removed.

LARGE AIRCRAFT FUEL SYSTEMS

INTRODUCTION

The following section of this chapter covers the general design requirements for air transport aircraft airframe fuel systems, which are regulated by JAR 25. An airframe fuel system is considered to stop at the Low Pressure fuel cock (LP fuel cock). As the AGK exam covers both airframe systems and power plants in one paper, the author has included the engine's High Pressure fuel cock (HP fuel cock) in certain diagrams for completeness. The components that constitute the high-pressure system and its operation are covered in the engine notes.

FUEL TANKS

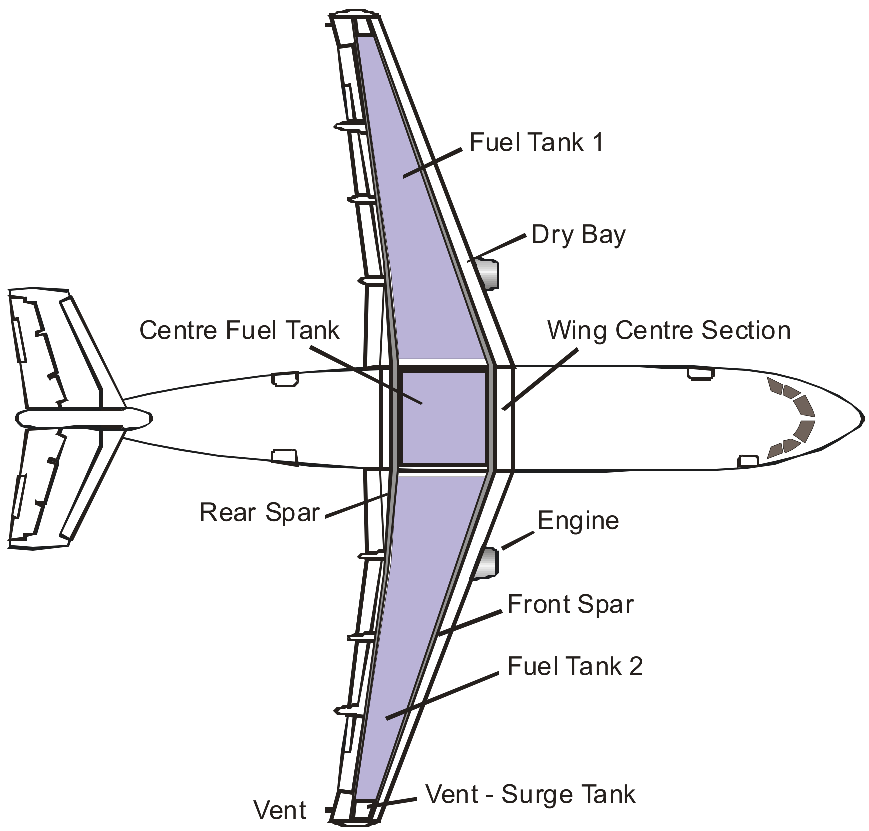


Diagram 10.24 Fuel Tank Layout for a Twin Jet

The majority of air transport aircraft use integral fuel tanks where areas of the wing's structure have been converted into fuel tanks. The advantages were outlined earlier. However, to increase the aircraft's fuel capacity for long haul flights, large air transport aircraft are fitted with a centre tank, as viewed in diagram 10.24. Additional tanks termed auxiliaries can be mounted within the fuselage.

Large transonic and supersonic aircraft often have an auxiliary tank mounted within the tailplane to allow fuel to be transferred and assist in trimming the aircraft. This is covered later in these notes.

DRY BAYS

To prevent fuel tanks butting up against structures that could be subjected to flames, termed **fire zones**, a void is left between the fuel tank structure and the firewall structure. This is termed a dry bay. Refer diagram 10.24. Dry bays and structural voids around fuel tanks must have open drainage holes, not only to ensure that any fuel leakage drains but also to ventilate them.

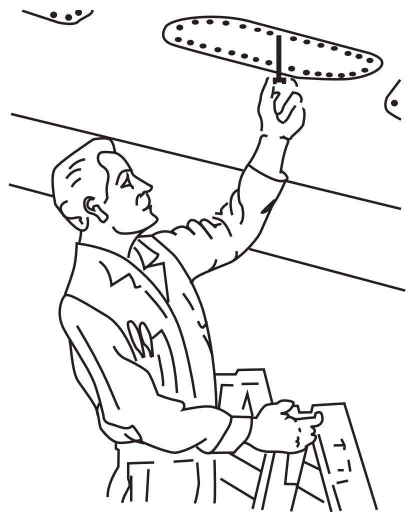


Diagram 10.25 Under Wing Access Panels

Integral fuel tanks must have facilities for interior inspection. This requires access hatches. These are normally located in the under side of the wing (see diagram 10.25). Fuel tank access covers have failed due to impact with high speed objects, such as failed tyre tread material and engine debris following engine failures.

FUEL TANK SUMPS

Fuel tanks must have a sump with an effective capacity, in the normal ground attitude, of not less than the greater of 0.10% of the tank capacity or one-quarter of a litre. This collects any hazardous quantity of water from any part of the tank when the aircraft is in the ground attitude.

Each sump must have an accessible drain that allows complete drainage of the sump on the ground and has manual or automatic means for positive locking in the closed position. The drain must be designed to prevent fuel spillage in the event of a landing with landing gear retracted.

CENTRE TANK

As aircraft have a maximum structural take-off mass and a maximum structural landing mass, operators try to reduce the fuel load within the legal requirements to enable them to carry the maximum payload possible. As seen in the Mass and Balance subject, adding fuel into fuselage tanks has the disadvantage of increasing the fuselage mass and, in many cases, there is a maximum fuel load limit for the centre tank, unless the wing tanks have been filled.

Therefore, many aircraft take off either with the centre tank empty or with a minimum fuel load within it. The crew controls the fuel feed to the engines to ensure that the fuselage centre fuel tank and auxiliary tanks are used first, thus lightening the fuselage and maintaining the mass within the wings.

TANK PRESSURISATION AND VENTING

To prevent fuel loss through evaporation at high altitude flight levels, and to assist in preventing vapour locks by ensuring a positive head of fluid pressure into the inlet of the tank-mounted booster pumps, the aircraft's fuel tanks are slightly pressurised using bleed air taken from the compression stage of the aircraft's gas turbine engines. This bleed air is pressure regulated to ensure that an increase of engine rpm does not create pressure fluctuations within the fuel tanks.

The tanks are vented to atmosphere via relief valves. These ensure that if the air pressure within the tank increases above a set level, the air and fuel vapour are relieved to atmosphere. In the event of a greater fuel draw off than the bleed air regulation can cope with, an inward relief valve allows ambient air into the tanks so that the fuel tank is not subjected to a negative pressure that could damage the tank's structure and stop or reduce fuel flow. It is normal for the bleed air regulation to cope with all normal fuel draw off, transfers, etc.

VENT-SURGE TANKS

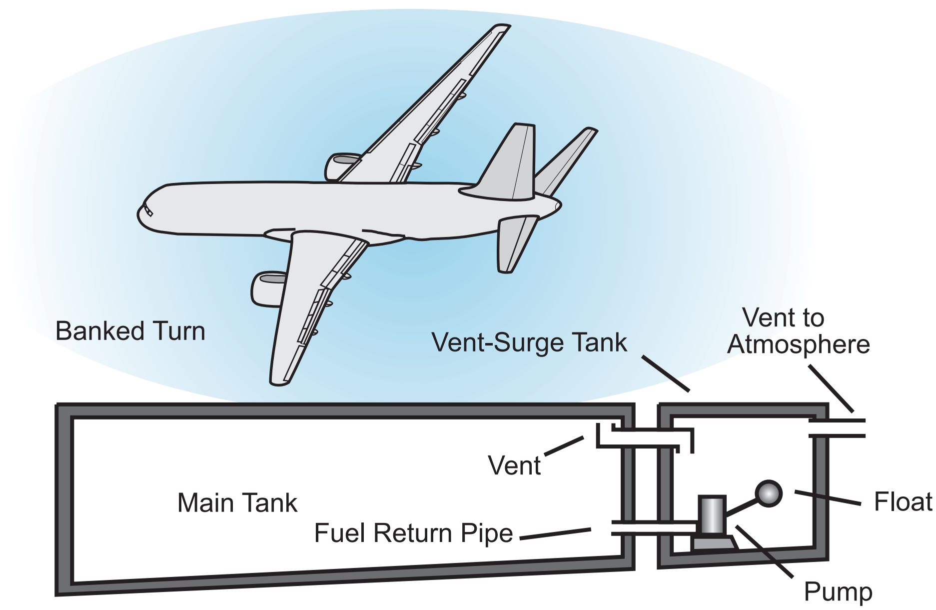


Diagram 10.26 Vent-Surge Tank

Vent-surge tanks are an integral part of the fuel system on large aircraft. These are normally located between the main fuel tanks and the wing tips (see diagram 10.24). They are not filled as part of the aircraft's Take-Off Fuel (TOF) load but act as the vent link between the main tank and atmosphere. In the event of the fuel volume increasing through thermal expansion after refuelling, the fuel can expand into the vent-surge tank. During flight when the aircraft banks, the surge of fuel toward the wing tip is stopped from venting to atmosphere via draining into the vent-surge tank (see diagram 10.26).

As the venting from pressurised fuel tanks can include droplets of fuel, the vent-surge tanks are designed to minimise the loss of fuel. Each vent-surge tank has a method of returning the liquid fuel to the main tanks. Diagram 10.26 shows a float switch operated electrical pump. Other methods can be a **jet pump** (see ejector pump later in these notes) or a simple flapper valve for aircraft with a large dihedral.

FUEL SYSTEM FOR A TWIN JET

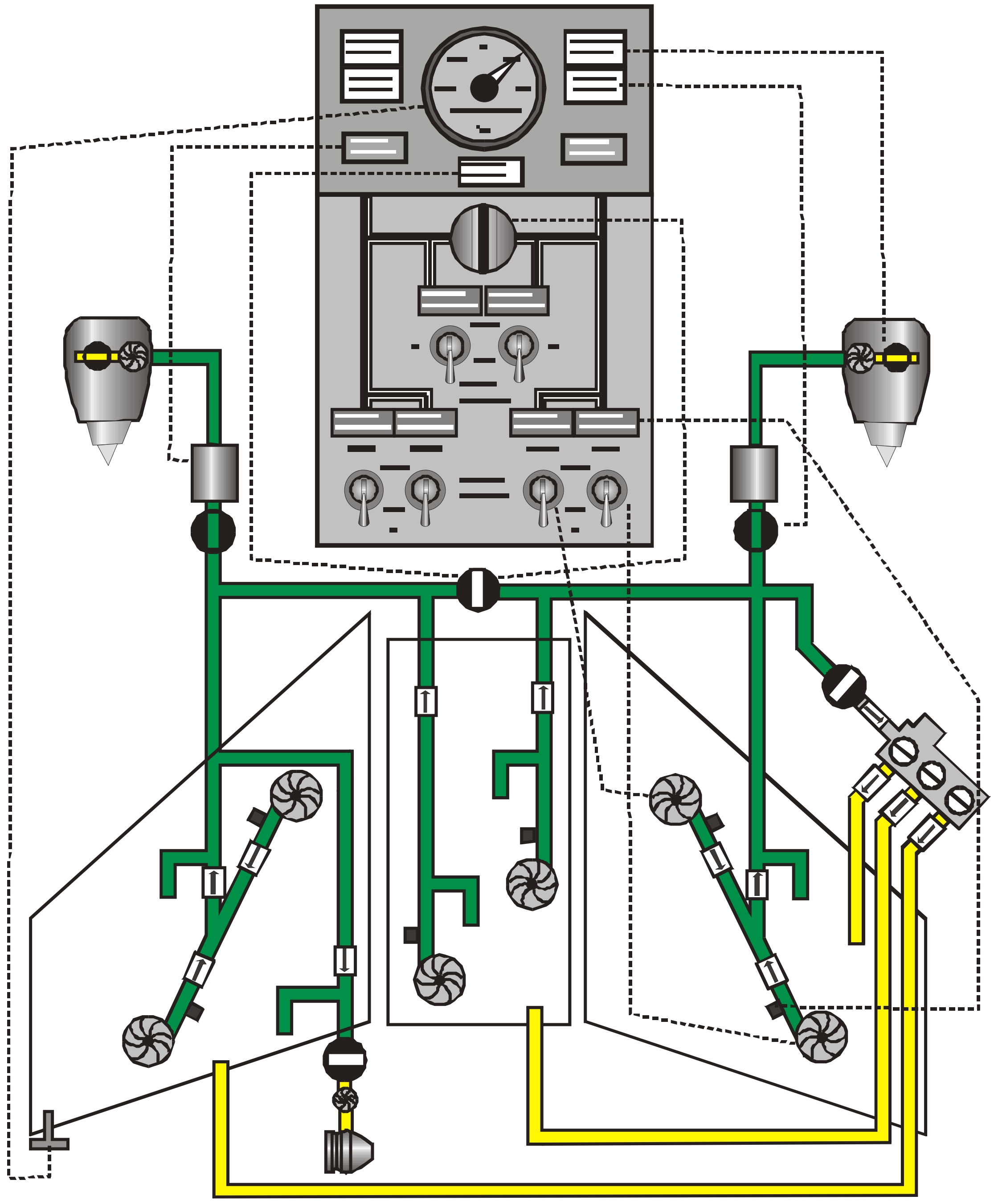


Diagram 10.27 Fuel System Schematic of a Twin Jet

The fuel system for a regional twin-engine aircraft is shown in schematic form in diagram 10.27, as would be found in an aircraft-type training manual. This diagram is loosely based on that of the Boeing 737-400 series aircraft, which is termed MRJT in the JAA syllabus and is only included for ATPL(A) and CPL(A) training purposes. As per all schematics, it is not strictly accurate but explains the basis on which a system works.

A variation of the complete schematic diagram 10.27 is sometimes used as an annex in Aircraft General Knowledge exam with one or more questions based on it. Apart from giving the candidate a preview of this schematic, the author uses it to amplify the JAR 25s relating to fuel systems and indications. Therefore, for clarity and ease of explanation, this diagram is broken down into its component parts.

ENGINE FUEL SUPPLY

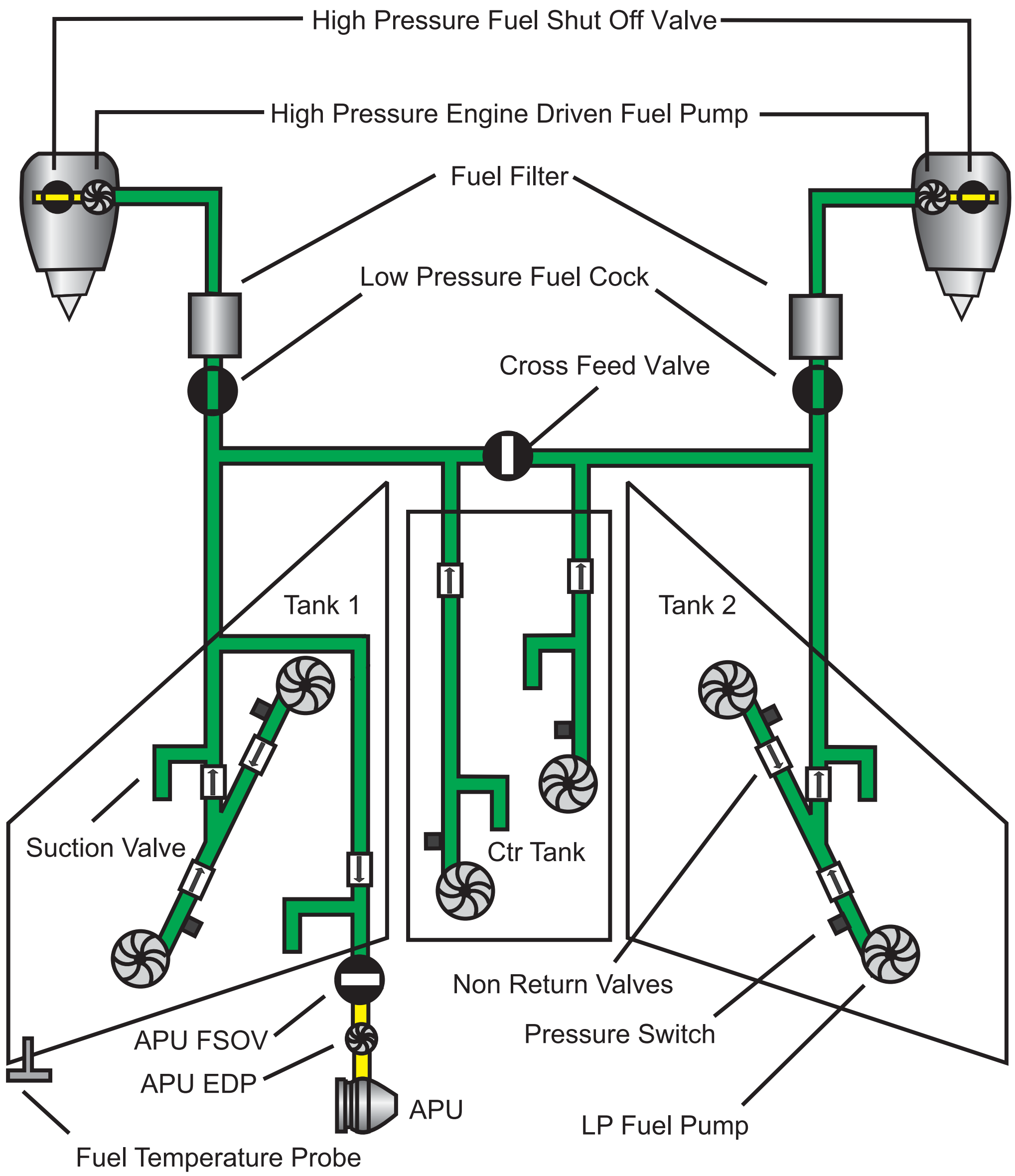


Diagram 10.28 Fuel Supply System Schematic for a Twin Jet

The JAR 25 regulations require that:

- Each fuel system is able to supply at least 100% of the fuel flow required under each intended operating condition and manoeuvre.
- The fuel is delivered to each engine at a pressure within the tolerance specified for the engine fuel system.
- Fuel systems other than gravity feed must be fitted with a main pump. Where a main pump is required, a back up main pump or emergency pump must be fitted. An emergency pump must capable of providing the 100% fuel flow.
- Each engine, in addition to having appropriate manual switching capability, must be designed to prevent interruption of fuel flow to that engine without attention by the flight crew, when any tank supplying fuel to that engine is depleted of usable fuel during normal operation, and any other tank that normally supplies fuel to that engine alone contains usable fuel.

Diagram 10.28 shows how these requirements are met. Effectively there are two separate fuel systems; one for each engine, comprising the centre tank and the wing tank in the same wing as that on which the engine is mounted. In the event of a fuel imbalance or an emergency, fuel from the opposite wing tank (via the cross feed valve) is drawn from the tanks by low pressure (LP) pumps, also termed **booster pumps**.

There are two pumps per engine in each wing tank and one per side in the centre tank. The two LP pumps per wing tank are required to meet the JAR requirement of 100% back up. The centre tank only requires one LP pump per engine as the crew can select the engine’s wing tanks or open the cross feed valve and supply both engines from the remaining LP pump in the event of a failure.

To meet the fourth requirement, the output pressure of the centre tank booster pumps is greater than that of the wing tank booster pumps. So, when all the pumps are selected ‘on’, both engines are supplied from the centre tank first. The wing tanks are effectively idling. In the event of a centre tank pump failing or the tank running dry, the engines are automatically and without interruption supplied by the wing tanks.

BOOSTER PUMPS

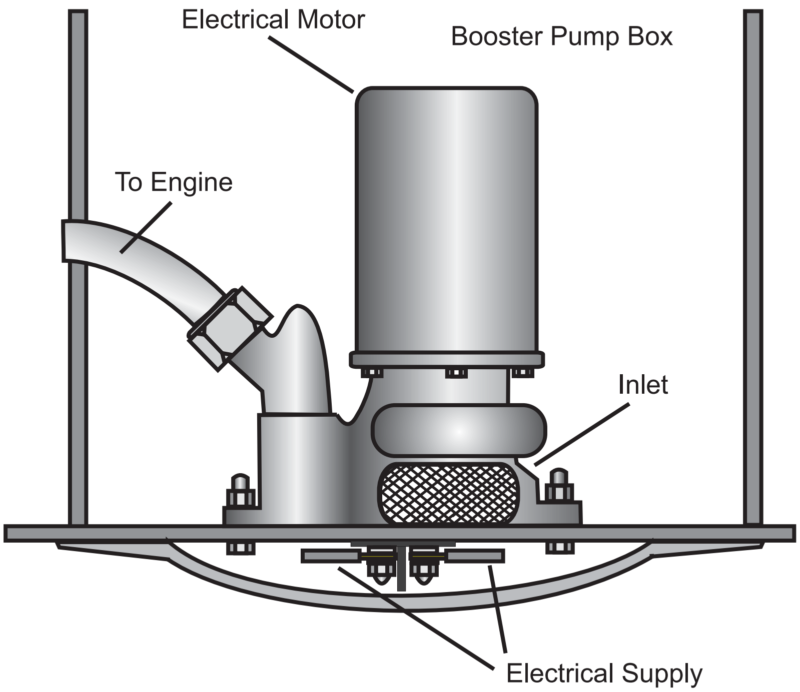


Diagram 10.29 Booster Pump

Booster pumps are normally centrifugal pumps powered by intrinsically safe electrical motors. Mounting the motors within the fuel tank, or passing some of the pumped fuel around galleries within the body of the motor, allows the motor and its bearings to be cooled. To prevent any ingress of large debris, a coarse mesh filter screen is fitted across the booster pump's inlet. This inlet must be five times larger than the outlet to ensure that fuel starvation does not occur.

BOOSTER PUMP BOX

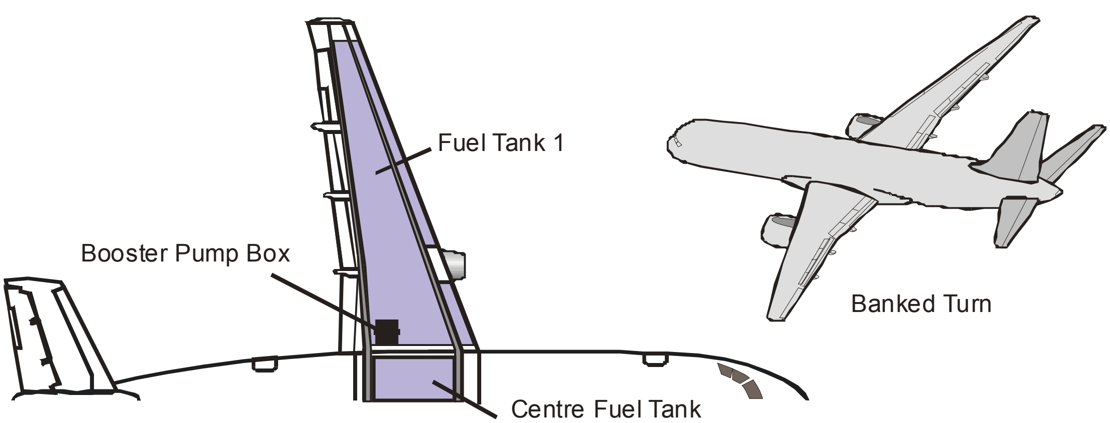


Diagram 10.30 Booster Pump Location

In diagram 10.28, the two booster pumps, as required by JAR 25, are shown as being separate in the main wing tanks. In reality, they are mounted side by side in a booster box as per diagrams 10.29 and 10.30. The booster box is located at the lowest point of the wing, by the wing root close to the rear of the tank. The function of the box is to keep a supply of fuel around the pump inlets at all normal flight conditions. Scavenge pumps are used to ensure that the box is kept filled when the aircraft's tanks effectively become lower than the box (e.g. a banked turn). These can either be electrical or ejector pumps.

EJECTOR PUMP

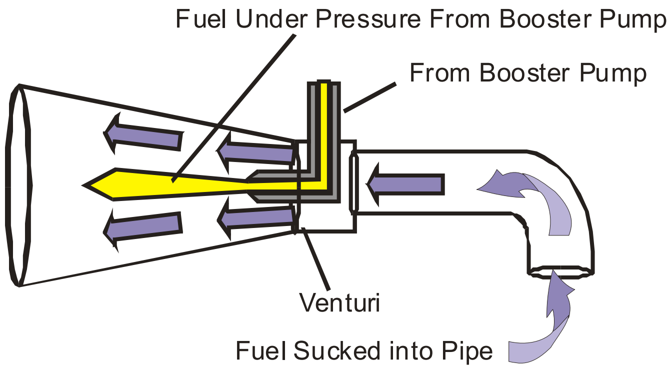


Diagram 10.31 Ejector Type Scavenge Pump

Ejector pumps work on the principle of the Venturi effect. Refer to diagram 10.31. A jet of pressurised fuel taken from either the booster pump's output or a dedicated pump streams through a cone (trumpet) shaped bell. The action of this jet is to pull the surrounding fluid with it. While the ejector pump cannot raise a great head of pressure, it is capable of moving large volumes of fluid many times the volume of the pressurised jet. For this reason, this type of simple pump can be used as a scavenge pump, drawing all usable fuel from different sections of a tank or different tanks to a central point.

Systems with dedicated pumps have a float switch in the booster box. If the fuel level drops, the dedicated pump is activated until the increased fuel level switches the pump off. The flight crew are able to override the float switch and select the scavenge pumps on at any stage of flight. Systems that use the fuel taken from the booster pumps can either be constantly operating when the pumps are selected on or have an on/off cock. This allows the jet pump to be operated by a float switch and/or pilot. The advantage of having constant movement of the fuel is the evening out of temperature, decreasing the likelihood of waxing.

SCROLL TYPE BOOSTER PUMP

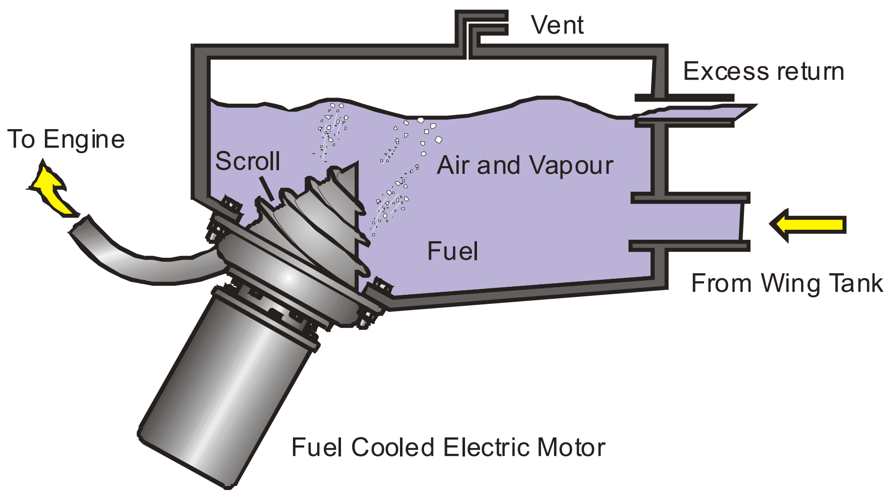


Diagram 10.32 Scroll Type Booster Pump

Another type of centrifugal pump is the scroll pump (see diagram 10.32). In this design, an electric motor turns a conical impeller that has an external helical thread like a screw. As the impeller rotates (clockwise in diagram) it pulls the fuel downward and forces it into the engine's feed pipe.

One of the advantages of this type of pump is that the action of the scroll in drawing the fuel downward separates out any air or fuel vapour. These are given a twist and stream upward from the scroll, as shown in diagram 10.32, thus ensuring that only liquid enters the supply pipe.

COLLECTOR TANK

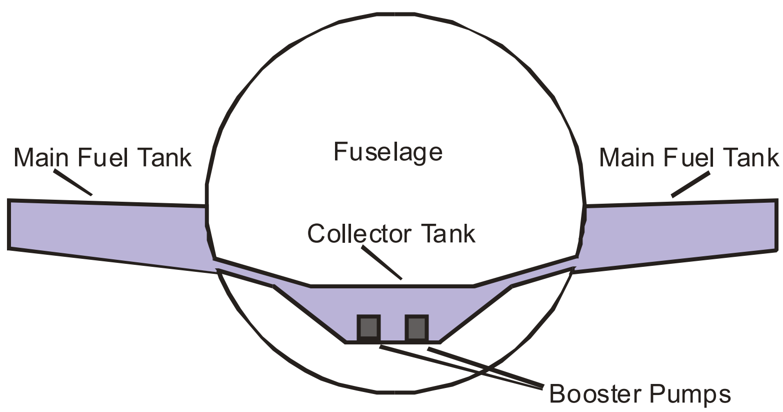


Diagram 10.33 Collector Tank

Other systems make use of collector tanks, where the fuel drains down from the wing tanks into a small fuselage mounted tank, which houses the booster pumps and supplies the engine. Refer to diagram 10.33. Whichever system is used, where more than one tank supplies fuel to an engine, it must be capable of supplying fuel to the engine without the intervention of the crew when one of the tanks has exhausted its supply of usable fuel.

PRESSURE SWITCHES

Refer to diagram 10.28 for the location of low-pressure switches. Each booster pump has a pressure switch located downstream of it. This is to give the pilot a warning via an amber light of a drop in fuel supply pressure. The exact indications are covered later in this text with diagram 10.35.

NON-RETURN VALVES

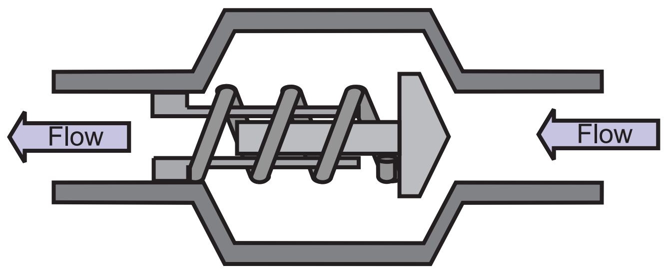


Diagram 10.34 Fuel System Non-Return Valve

Referring to diagram 10.28, downstream of each booster pump is a non-return valve (NRV) depicted as an arrow. These are located to prevent fuel pumped by one booster pump from re-circulating through the intake of the other. Diagram 10.34 shows a typical fuel system NRV.

THERMAL RELIEF

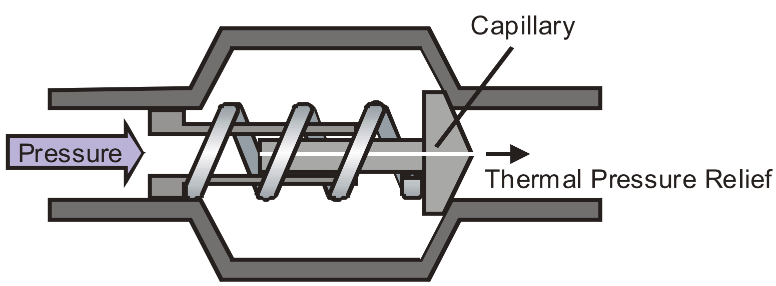


Diagram 10.35 Fuel System NRV withThermal Relief

Where fuel can be trapped in a fuel supply line between components, there is a possibility that due to thermal expansion, the fluid pressure would rise and damage either the component or the pipe. To overcome this and provide thermal relief, some NRVs have a capillary size hole bored through the centre of the valve (see diagram 10.35). When the valve is closed under normal circumstances, the fluid flow through the capillary bore is negligible. However, with an increase in temperature and pressure, a small volume of fluid passes through the capillary bore, which relieves the pressure of the fluid trapped downstream.

SUCTION VALVE

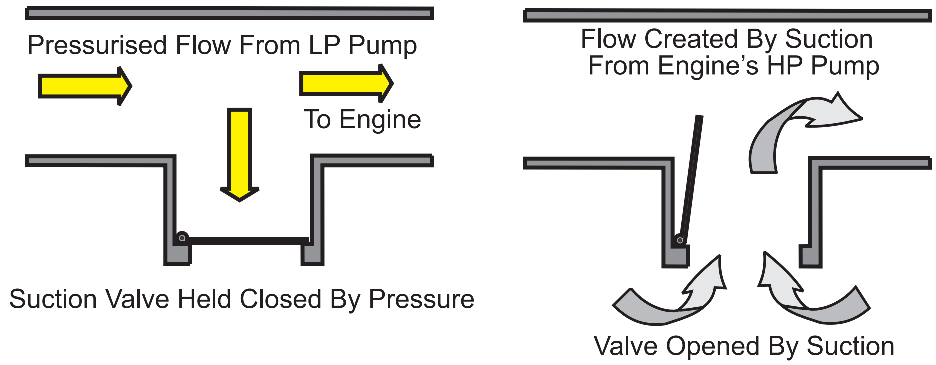


Diagram 10.36 Suction Valve

Refer to diagram 10.28. A suction valve is downstream of the NRVs. These valves are fitted to meet the requirement that if both booster pumps fail, the engine's High Pressure (HP) pump must be able to draw sufficient fuel to maintain at least 75% of its power output when the system is functioning normally.

Referring to diagram 10.36, as the impellers of the booster pumps would cause a restriction, a valve with a minimum of five times the cross sectional area of the supply pipe is fitted into the supply pipe. Under normal conditions, the pressurised flow from the LP booster pump(s) holds the valve plate closed, preventing re-circulation of the fuel. If the supply pressure drops, the suction created by the HP pump and the pressure of the fuel in the tank opens the valve plate, allowing the engine to draw its fuel supply.

LOW PRESSURE FUEL COCK — LP COCK

The LP Cock is designed to isolate the fuel tanks from the engine. It should be located as close to the fuel tanks as practicably possible and must be located so that the valve's operation is not affected by the engine's mountings suffering a structural failure. If there is any risk that the LP Cock is subjected to flame, it must be either fire resistant or protected from the flames. (Some aircraft manuals refer to them as spar shut-off valves).

Closing of any fuel shut-off valve (FSOV) for any engine must not stop the fuel flow to any other engine. Operation of a FSOV must not prevent the operation of other equipment in an emergency, such as the means for feathering the propeller for a turboprop. The system must allow the crew to reopen the shut-off means in flight after it has been closed.

HIGH PRESSURE FUEL COCK — HP FUEL COCK

The HP cock is designed to isolate a gas turbine's combustion section from the engine's high-pressure fuel system and is located downstream of the HP EDP. The engine fuel system is covered in the Propulsion notes. However, the regulations governing the LP Cock's controls also cover the HP Cock and are listed below:

- Each control must be located so that persons entering, leaving, or moving normally in the cockpit cannot inadvertently operate it.
- For manual valves, positive stops, or in the case of fuel valves, suitable index provisions must exist in the open and closed positions.
- Control must be able to maintain any set position without constant attention by flight crewmembers and without creep due to control loads or vibration.
- The knob of these control levers must be coloured red.

For electrical or electronic fuel selector:

- Digital controls or electrical switches must be properly labelled.
- If the fuel valve selector handle or electrical or digital selection is also a fuel shut-off selector, the 'off' position marking must be coloured red.
- Means must be provided to indicate to the flight crew the tank or function selected. Selector switch position is not acceptable as a means of indication. The 'off' or 'closed' position must be indicated in red.
- Electrically operated valves must have a valve position indicator, which senses directly that the valve has attained the position selected.
- If a separate emergency shut-off means is provided, it also must be coloured red.

FUEL FILTER

Referring to diagram 10.28, apart from the finger screens fitted around the tank outlets, each fuel system requires a fuel filter or strainer unit fitted as close to the fuel tank outlet as practicable. This unit must have a sump to trap any free water that entered the system and a drain.

FLIGHT DECK FUEL CONTROL PANEL

Refer to diagram 10.37 for a representation of a 737-400 flight deck fuel control panel. This is mounted on the captain's overhead panel (above the pilot's front windscreen). The need for this type of panel was caused by the introduction of the two crew cockpit. The panel is a schematic representation of the aircraft's fuel system, which allows the crew to see and control the fuel system.

JARs require that:

- Each fuel tank selector control must be marked to indicate the position corresponding to each tank and to each existing cross feed position.
- If safe operation requires the use of any tanks in a specific sequence, that sequence must be marked on, or adjacent to, the selector for those tanks.
- Each valve control for each engine must be marked to indicate the position corresponding to each engine controlled.

These requirements are met by the placards on the panel denoting which switches align with which engine's fuel system and the associated condition indicated by the operation of the lights.

These are:

Low pressure Lights	Illuminates when power is applied, the pumps are selected to 'ON', and the pressure is low
Centre Tank	Extinguishes when the pressure is normal or the selection is 'OFF'
Low pressure Lights	Illuminates when the pressure is low or the switch is 'OFF'
Main Tank	Extinguishes when the pressure is normal and the selection is 'ON'
Cross Feed Light	Extinguished when the valve is closed Dim when the valve is open (cross feed selected) Bright when the valve is in transit and not in agreement with the selector's position
Filter By-Pass Light	Illuminates when the filter is clogging and sufficient differential pressure exists across the filter for the pressure switch to activate and indicate an impending by-pass of the filter
LP Valve Closed Light	Dim when valve is closed Bright when in transit
Engine Valve Closed Light	Extinguished when valve open Bright when valve closed

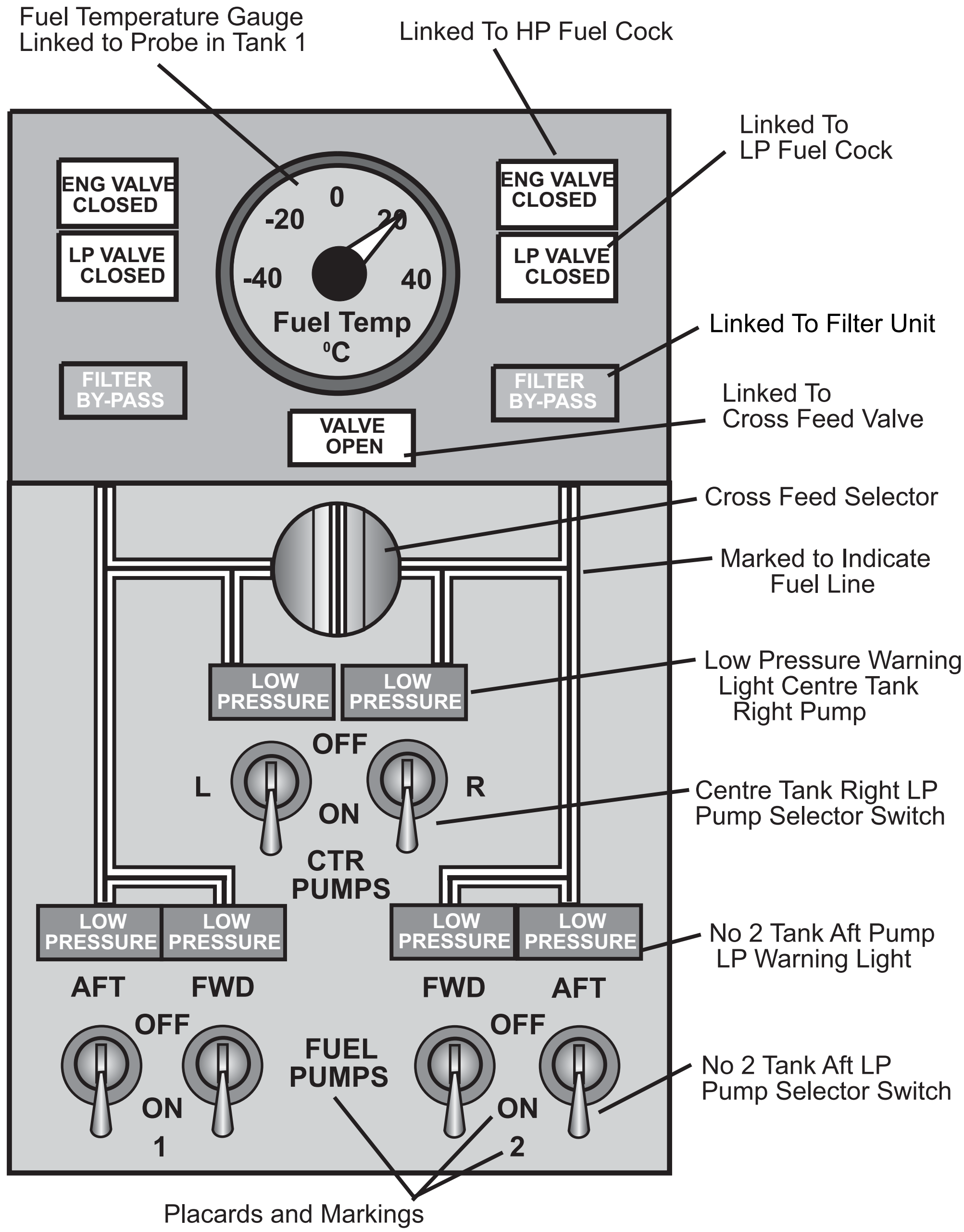


Diagram 10.37 Flight Deck Fuel Control Panel For Twin Jet

FUEL TEMPERATURE GAUGE

Refer to diagram 10.37. The fuel temp gauge is linked to the fuel probe in the No. 1 tank (refer to diagram 10.28 for location. This indicates the overall temperature of the fuel to the crew and warns the pilots of the need to increase the heat required for conditioning the fuel.

Apart from utilising the heat sink ability of fuel to remove waste heat from the hydraulic system, the heat gained helps to prevent the fuel from dropping to its freezing point. Some aircraft have dedicated heater units mounted in the fuel tanks to overcome fuel freezing problems.

AUXILIARY POWER UNIT FUEL SUPPLY

Refer to diagram 10.28 to locate and trace the fuel supply for the APU. Fuel can be supplied from the No. 1 tank by either LP pump, or drawn from the No. 1 tank by the APU's HP pump, via a dedicated suction valve (75% power). The unit can also be supplied from the centre tank by running the left LP pump, or from either LP pump in the right tank when the cross feed cock is opened.

For ground operations, it is normal for the APU to be supplied with fuel from the centre tank, as this prevents the aircraft from gaining a fuel imbalance with prolonged operation. The APU has its own dedicated control panel. The controls and indications alter depending on whether the APU can only be operated on the ground, whether it can be started and operated in flight, or operated in flight if started prior to take-off.

UNDER WING SINGLE POINT PRESSURISED REFUEL SYSTEM

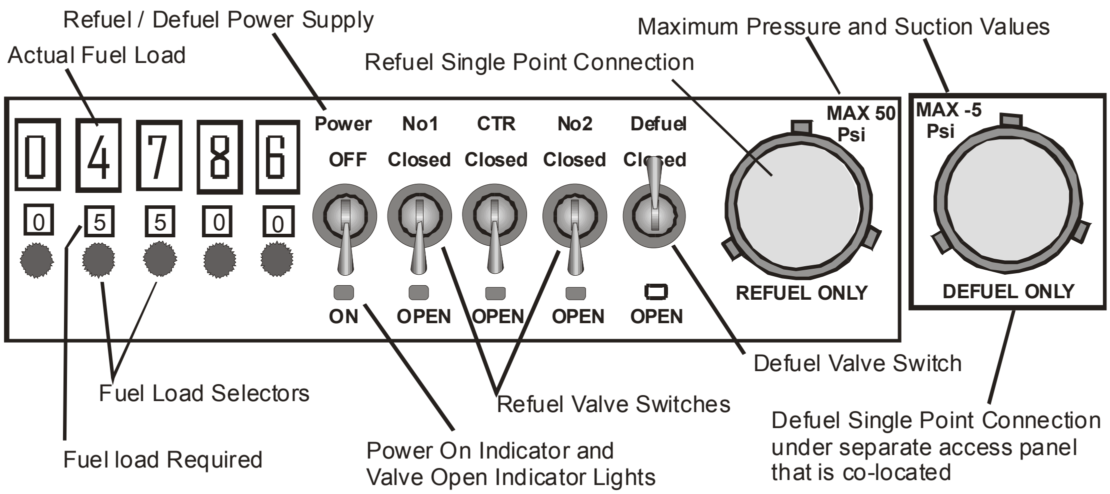


Diagram 10.38 Refuel Panel

Large aircraft make use of single point pressurised refuelling systems. As the name implies, all the tanks can be filled from a single connection point. This is normally accessible via a panel on the underside of the wing on the opposite side of the aircraft to the main entrance doors. Diagram 10.38 shows a digital refuel panel. First generation systems used analogue gauges. These systems operate using fuel mass, not volume, and utilise the aircraft's fuel tank sender units to display the actual fuel tank contents. Large aircraft fuel content measurement and indications are covered later in these notes. The 737-400 (JAR exam MRJT) uses kilograms for units of mass. Other aircraft might use pounds.

Defuelling of large aircraft is also carried out via a single hose connection. This is normally located alongside the refuelling panel but accessed through a separate panel, as diagram 10.38 shows. In legislation, the term refuelling also covers defuelling.

Standard refuelling connections must be able to withstand a pressure of 50 psi and the defuel system, a suction pressure of minus 5 psi.

The system can be operated in several ways. Diagram 10.39 shows the system set up to refuel the aircraft to a total fuel load of 5500 kg, first with power on. The total mass of fuel required for the aircraft can be entered via the fuel load selectors. With all the refuelling valve switches set to open, fuel is pumped into the aircraft's tanks via the refuelling galleries.

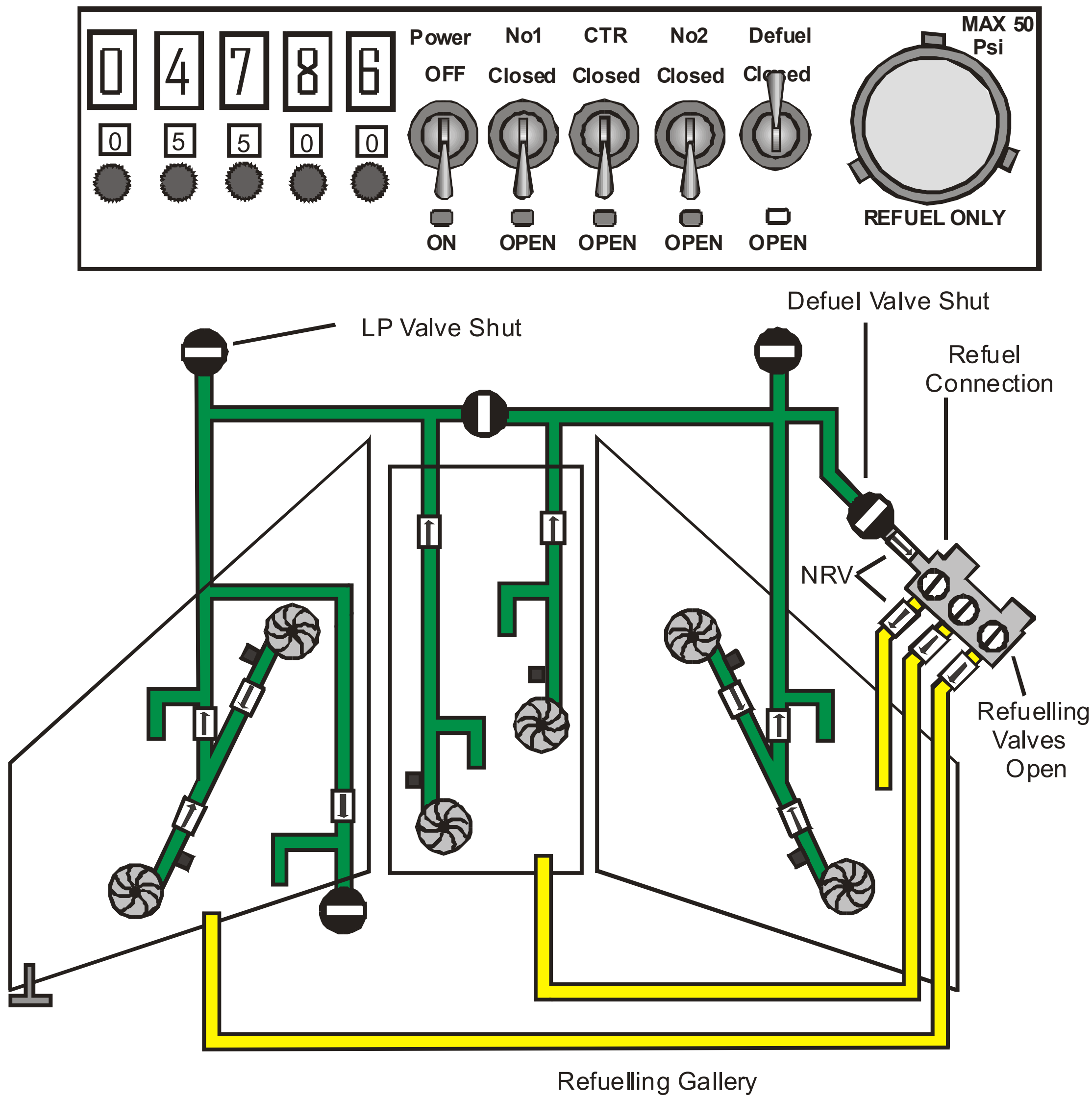


Diagram 10.39 Refuelling System

The system is designed to evenly fill the wing tanks and ensure that the centre tank is not filled beyond the mass specified before the wing tanks are filled. When the fuel load is correct (the actual and the selected values are the same), the valves are automatically shut off. The switches return to the closed position and the valve open lights extinguish. The power on light remains illuminated.

If the wing tanks alone are to be filled, the refueller would leave the centre tank refuelling valve closed, select the total fuel mass for the aircraft, and then start pumping the fuel on board. If one tank is to be brought up to a set value, the refueller can add fuel to that tank alone.

As the refuel/defuel circuit draws power without making the battery master switch, many aircraft refuelling point access panels are designed to either move the power switch to the 'off' position or prevent the panel from closing until the switch is placed in the 'off' position. This is to ensure that the circuit is isolated after refuelling.

REFUELLING GALLERY PIPES

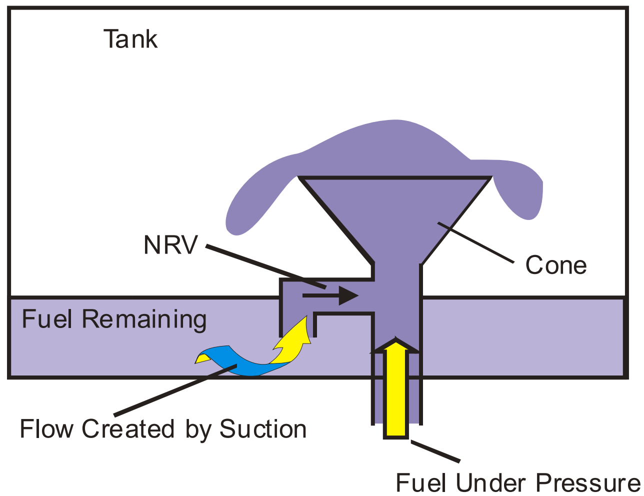


Diagram 10 40 Refuel Gallery Pipe in Tank

The following paragraph is taken directly from JARs for fuel system design.

"*With standard refuelling equipment and standard aircraft turbine fuels, voltages high enough to cause sparking may be induced between the surface of the fuel and the metal parts of the tank at refuelling rates above approximately 250 gal/min. These induced voltages may be increased by the presence of additives and contaminants (e.g. anti-corrosion inhibitors, lubricating oil, free water) and by splashing or spraying of the fuel in the tank.*"

As can be seen from above, refuelling has a potential fire/explosion risk due to sparks created by dissimilar static electrical charges within the fluid body and the surrounding structure because of the insulating qualities of hydrocarbons. To minimise this risk, JARs require that the fuel is added at the lowest possible level in a tank and is prevented from splashing.

One of the methods of meeting this requirement is shown in diagram 10.40. The fuel under pressure enters the tank via the gallery pipe, which terminates in a cone. This allows the pressure to drop and the fuel to bubble over the edge of the cone rather than jetting up into the tank, splashing and spraying around. A pipe incorporating an NRV acts to draw the remaining fuel and added fuel from the lowest practical level in the tank via suction. This ensures that the fuel, the fuel that is added, and the contents of the tank are mixed, allowing any different potentials within the fluid body to even out before the fuel cascades over the rim of the cone.

FUEL IMBALANCE ON THE GROUND

Referring to diagram 10.41, this system allows a fuel imbalance between wing tanks to be reconciled without needing to defuel one tank and refuel the other. In diagram 10.41, tank 1 is over fuelled. To even the fuel load laterally requires each tank to have 3250 kg in it.

At the refuelling panel, the following selections would be made:

- Turn power on to the refuel/defuel circuit
- Set 3250 kg on the load indicator
- Select tank 2's refuel valve to open
- Set the defuel valve open
- On the flight deck fuel control panel (diagram 10.37) the X-feed valve would be selected to cross feed and power applied to one of the LP pumps in tank 1.

Fuel from tank 1 transfers via the supply gallery through the defuel valve to tank 2. When the correct mass of fuel has transferred, tank 2 refuel valve automatically closes. The defuel valve and power can be set to 'off', and the flight deck controls can be reset.

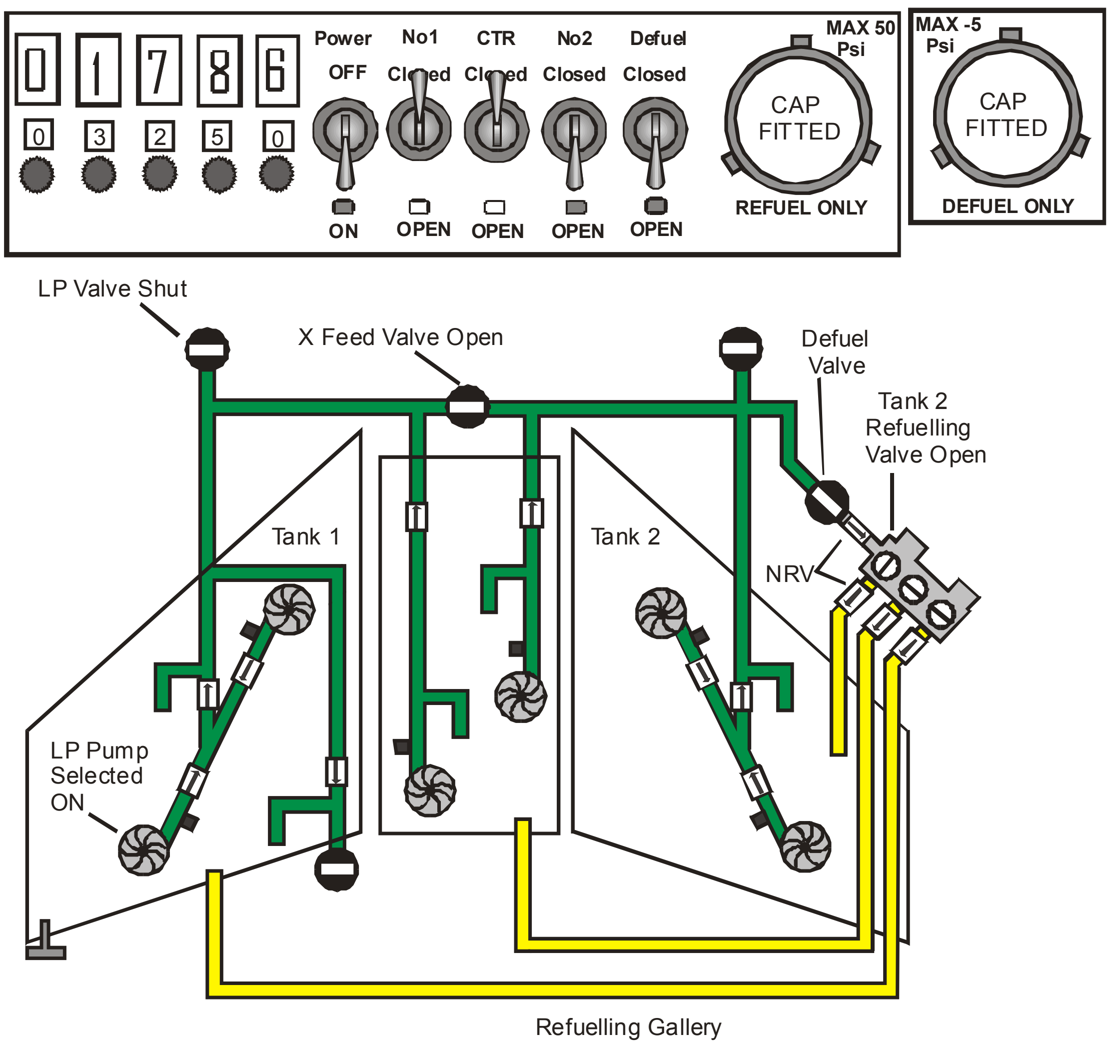


Diagram 10.41 Reconciling a Fuel Imbalance

DEFUEL

The LP pumps are used to defuel a single tank or the whole aircraft. In this case, the refuel valves remain closed and the bowser is connected to the defuel point. The bowser's pumps apply suction pressure of not more than –5 psi. Any greater suction pressure could damage the fuel system. Defuelling can only remove the usable fuel. Draining the unusable fuel requires the use of the water drain valves.

CAPACITIVE FUEL CONTENTS MEASURING SYSTEM

Refer to diagram 10.42 (A). If a tank is filled with a mass of fuel that is equal to half its capacitive volume is tipped as per (B), the mass and the volume remain constant.

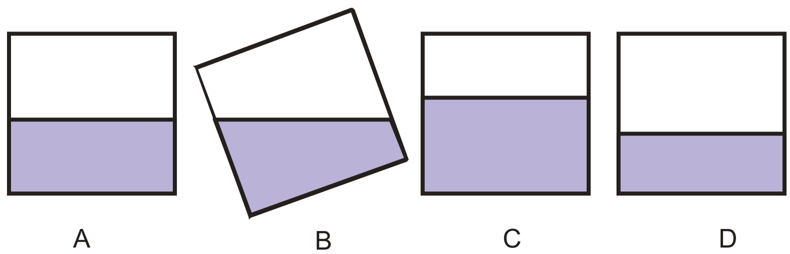


Diagram 10.42 Fuel Tank Conditions

Should the fuel heat and expand as per (C), or cool and contract as per (D), the SG and the volume alter but the mass remains constant. Using a volumetric system, such as a float, and resistive system of indication would give false indications as to the quantity of fuel on board an aircraft. This led to the adoption of the capacitive contents gauging system for air transport aircraft, which effectively weighs the quantity of fuel on board the aircraft.

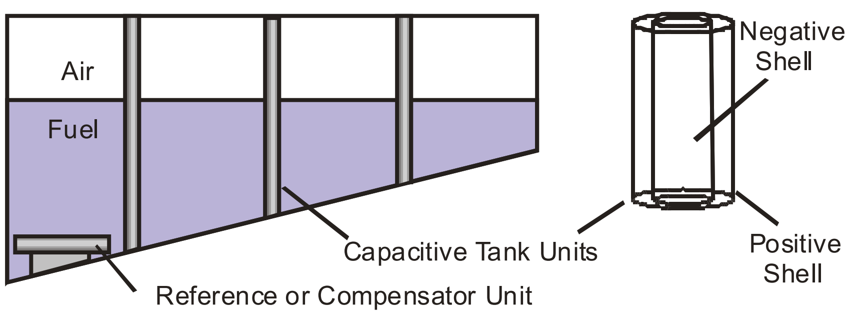


Diagram 10.43 Capacitive Tank Units

The capacitive content measuring system functions on the principle that air and fuel have different di-electric values. Diagram 10.43 depicts a typical capacitive system. This consists of a series of vertically mounted tank units. These alter in length as the depth of the tank varies. A reference unit also referred to as a compensator unit is mounted, so that it is always submerged beneath the fuel. An amplifier and contents indicator complete the system, which is supplied with 28 Volts DC.

The capacitance of a capacitor is determined by the area of the capacitor's plates, the distance between them and the permeability (conductivity) of the material between them. In this case, each tank unit consists of two concentric tubes, one located inside the other but electrically insulated from the other as per diagram 10.43. As the total area of all the tubes is known, the distance between the inner and outer tubes is set. In addition, all the tubes are connected in parallel to the compensator unit, effectively forming one large capacitor. The depth of fuel determines the total capacitance of the system.

The outer tube of each tank unit is perforated to allow the fuel to move between the two tubes, so that it is at the same level as the rest of the tank. Some designs have an extra tube or shell mounted around the unit. It acts as an earth screen to prevent arcing in the event of an electrical failure but is not part of the measuring system.

An aircraft's system is set to read zero when the tanks are empty of usable fuel, unusable fuel remaining. As the designers know the volume of the fuel tank and the standard SG of the fuel to use, they can calibrate the system from zero to full and calibrate the indication system in units of mass, either kilograms or pounds.

When the aircraft banks, although the fuel flows to the lowest point, the same surface area of the tank units is covered in fuel, so the capacitance and the indication remain the same. As these systems units are designed to work within a range of Specific Gravities, by locating the reference unit under the fuel, if there is a change of SG and the fuel expands or contracts, the compensator unit is also subjected to it**.** Although the depth has varied, the mass has remained constant. Due to the change in SG, the capacitance also remains the same. Therefore the indicator gives the same mass reading as before.

If the SG of the fuel put into the tank is different from the range or setting for the compensator, there can be an error in the quantity indicated (e.g. if a wide cut fuel was used in place of JET A1, a greater mass would be shown than actually carried). In the event of a failure in the fuel indication system, the indicator must drop to zero.

FLIGHT DECK CONTENTS INDICATIONS

Older aircraft used individual analogue gauges on the flight engineer's panel to indicate fuel contents. To reduce the number of gauges required for two-crew operation, the totaliser system was used. In this system, all the inputs from the various tanks feed to a single amplifier, which combines them and gives a single output, allowing one gauge to show the total fuel contents of the aircraft. To comply with the regulation, which requires the pilot to be able to ascertain the contents of each fuel tank, a momentary selector switch adjacent to the gauge allows the crew to check the contents of each tank. After releasing the selector, the gauging system defaults back to total.

It is easier to give a quick visual indication as to a position or value with an analogue dial, but is more accurate to read a digital indication. A hybrid system was used in which the quantity of fuel was indicated by both digital and analogue methods on the same gauge.

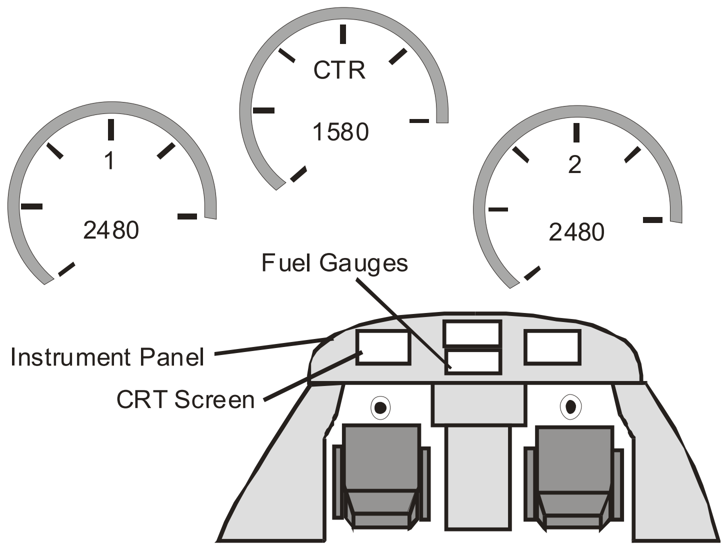


Diagram 10.44 Fuel Contents Indication on Modern Flight Deck

The advent of the glass cockpit has updated and enhanced the hybrid system (see diagram 10.44) of a typical MRJT aircraft. In this system, the fuel content is indicated by three analogue/digital displays on a CRT screen. The gauges are set out as per the tank numbering system. The digital readout and a white arc show the mass of each tank.

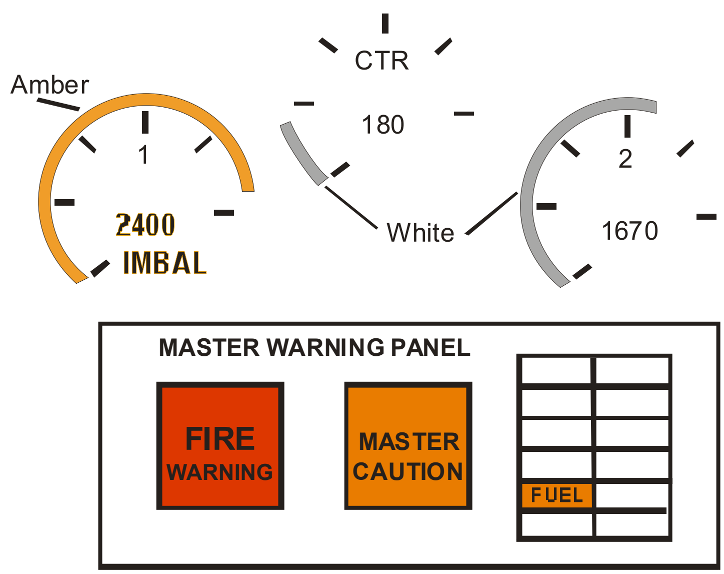


Diagram 10.45 Abnormal Fuel Indication on Modern Flight Deck

Refer to diagram 10.45. In flight to warn the crew of an abnormal condition (e.g. a fuel imbalance, low quantity, or greater fuel quantity in the centre tank than the wings) the white arcs turn amber on the affected tank and the condition is noted in text. The master caution amber would flash, drawing the crew's attention to the central warning panel and a fuel warning. The master caution and the central warning can be cancelled by depressing the master caution light, but the indication on the gauge remains until the situation is corrected. Any other fuel condition results in the illumination of the master warning and the central warning panel.

FUEL RECONCILIATION ON THE GROUND

JARs require that a physical check of the fuel quantity be carried out on the ground without applying power.

This can be accomplished by dipstick from the top of the tanks, but in most cases, large aircraft use under wing units. There are a variety of these, however, the JAA FCL syllabus 021 01 11 04 only specifies the dipstick. The exams have been questioning the drip stick, shown in diagram 10.46.

The drip stick is a hollow tube with a very small hole in its side, which passes through a gland let into the lower skin of the wing tank. When not in use, a tap mounted at the lower section of the drip stick ensures that fuel is not lost, the unit is locked up, and a panel covers its exterior.

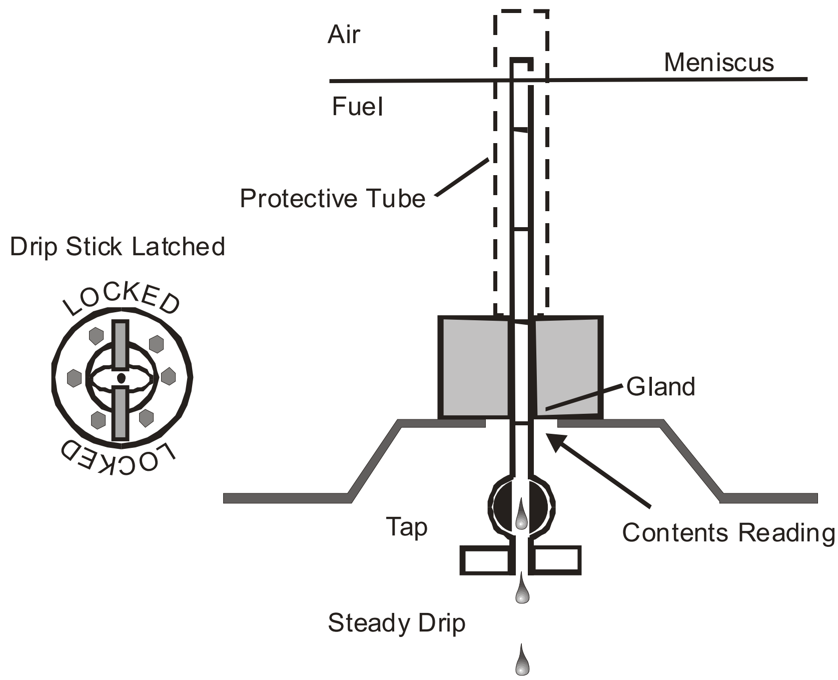


Diagram 10.46 Drip Stick

When a reading is to be taken, the tap is opened, and any fuel in the stick is allowed to drain out. With the tap open, the stick is slowly pulled down until a steady drip of fluid comes from the bottom of the stick. This indicates that the orifice is at the same level as the fuel's surface. If the orifice is lower than the fuel's meniscus, it results in a steady trickle, increasing to a steady run. When the surface level has been found, the quantity of fuel in that compartment can be read off from the contents graduations marked on the tube referenced to the junction of the fitting. There are several drip sticks per wing tank. The total quantity of fuel is the sum of all the readings.

FUEL JETTISONING

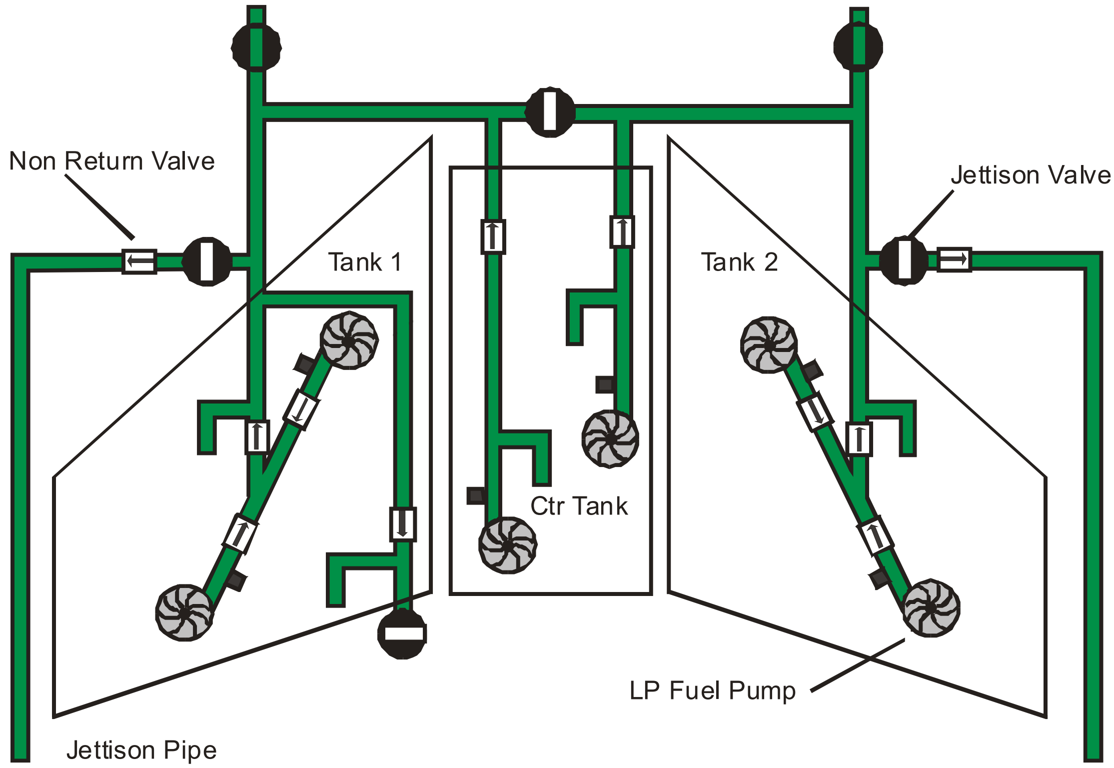


Diagram 10.47 Fuel Jettison System

Most air transport aircraft have a maximum take-off mass much greater than their maximum landing mass. In an emergency, if they require to land shortly after take-off, they would be grossly overweight. This is likely to result in structural failure, which could compound the problem of safely evacuating the passengers and crew.

To overcome this problem, aircraft certified under JARs 23 and 25 can be required to have a means of jettisoning fuel. Diagram 10.47 depicts such a system, and the following text denotes when the JARs require a jettison system to be fitted.

AIRCRAFT CERTIFIED UNDER JAR 23

A twin-engine aircraft certified under JAR 23 requires a jettison system that is able to jettison enough fuel to bring the aircraft's mass down from the MTOM to the MLM. The rate of fuel jettisoning must average 1% of the MTOM per minute. However, the minimum time for jettisoning this fuel does not have to be less than 10 minutes.

A jettison system has to be fitted if:

- The aircraft's maximum landing mass is less than 25% of the total fuel mass for the aircraft, or
- Its maximum landing mass is 95% of the MTOM, and the minimum fuel mass is equal to 1.5 hours operation at max continuous power plus a mass of fuel equal to the difference between MTOM and MLM.

AIRCRAFT CERTIFIED UNDER JAR 25

For JAR 25 certified aircraft, a fuel jettisoning system must be installed, unless it is shown that the aeroplane can meet the climb requirements of a climb speed which is:

- Not less than 1.08 V_{SR} for aeroplanes with four engines, on which the application of power results in a significant reduction in stall speed, or
- 1.13 V_{SR} for all other aeroplanes
- Not less than V_{MCL}
- Not greater than V_{REF}

At maximum take-off mass, less the actual or computed weight of fuel necessary for:

- A 15-minute flight comprised of:
  - A take-off
  - A go-around
  - A landing at the airport of departure

If the aircraft cannot meet the climb requirement, a jettison system must be fitted that from maximum take off mass is capable of jettisoning enough fuel within 15 minutes to allow the aircraft to meet the climb requirements. The jettison system must be designed so that the fuel is discharged clear of any part of the aircraft, so that fuel or fumes do not enter any part of the aircraft, and that the system is free of fire hazard. During fuel jettisoning, the controllability of the aeroplane must not be adversely affected.

The system must automatically stop jettisoning fuel when the level in the tanks is equal to the quantity of fuel needed for a climb from sea level to 10 000 ft and cruise for 45 minutes at the aircraft's speed for maximum range. However, the fuel jettisoning system must be designed to allow the flight crew to stop the jettisoning of fuel at any time in the operation.

The control for jettisoning fuel must be coloured red to denote that it is an emergency system. It must not be co-located to normal controls where it could be operated inadvertently. If an auxiliary control independent of the main jettisoning control is fitted (also coloured red), the system can be designed to jettison all the usable fuel.

These systems have to be designed so that any single failure does not result in a hazardous condition due to unsymmetrical jettisoning of or inability to jettison fuel. If the operation of slots, slats, or flaps alters the airflow to such an extent that it affects fuel jettisoning, a placard must be placed adjacent to the jettisoning control to warn the flight crew against jettisoning, while they are in use.

To meet these requirements, the control's levers are detented. Push button or toggle switches are normally under a guard to prevent inadvertent operation. In older aircraft fuel systems, a dedicated float switch is used to close the jettison valve when the minimum fuel quantity is reached.

Modern systems use the fuel capacitance system and the Flight Management System to determine the amount of fuel to remain. This system takes the ambient conditions and engine performance into account. Please note that the LP pumps supply the jettison system.

Other considerations are that jettisoning must not take place in precipitation or close to a thunderstorm and must be carried out at altitudes where the fuel evaporates before it reaches the ground.

FUEL TRIMMING

Large air transport aircraft and supersonic transports use fuel as a means of trimming the aircraft. By having fuel tanks in the tailplane or rear fuselage, the pilot can pump fuel from the main tanks to the rear tanks to move the aircraft's CG. For large transport aircraft, the increase in mass at the tailplane reduces the need for large-scale elevator deflections at low airspeeds such as landing approach, allowing the V_{mca} to be reduced as more of the controls' range of movement remains available. It also allows the crew to trim the aircraft without having to apply elevator or tailplane trim at high altitude and high transonic speeds, thus reducing trim drag. Supersonic aircraft, such as Concorde, use fuel trimming to move the CG to stop the Mach tuck under effect.

REFUELLING AND DEFUELLING

There are two methods of refuelling an aircraft: open line or pressure line. There are three methods of delivering the fuel: bowser, dispenser, and hydrant. The refuelling of light aircraft via open line can be carried out using either bowsers or dispensers. The refuelling of large aircraft using pressure line is with either bowsers or hydrants. Defuelling of large aircraft is via bowser.

This section looks at the systems and the safety regulations starting with open line for light piston engine aircraft.

OPEN LINE REFUELLING

Diagram 10.48 Fuel Bowser

The aircraft can either be taxied to a dispenser, similar in design to a filling station's petrol pump, or parked on the apron and have a fuel bowser (see diagram 10.48) deliver to the aircraft.

Before refuelling may commence, the person refuelling the aircraft must check that:

- The engines and all electrical equipment are switched off.
- No passengers are on board.
- The bowser that arrives has the correct fuel grade markings.
- The aircraft is earthed.
- The bowser has a drag strip or chain in contact with the ground. This is to ensure that any static charge created up by the vehicle in movement is dissipated.
- The vehicle can drive away from the aircraft in an emergency without having to reverse.
- The bowser and the aircraft are bonded together to ensure equal polarity. This is normally done using a bonding lead that is part of the bowser equipment. It must be attached to a designated earth point on the aircraft (see aircraft's AFM).

Before the tank filler caps are removed, the bonding lead attached to the open line discharge nozzle must be connected to the designated earth adjacent to the filler cap. This lead must remain attached until the filler caps are refitted. Refuelling is considered to have commenced once the caps have been removed, even if fuel has yet to be added. During refuelling, the person operating the nozzle should ensure that the fuel entering the tank does not splash around.

For open line refuelling from a dispenser, the aircraft is parked so that it can be easily pushed away from the pump station in the event of an emergency. For earthing polarity, the aircraft is earthed to a point, which forms part of the dispenser stand. The nozzle must be bonded to the aircraft before the caps are removed and remain bonded until caps are fitted.

PRESSURE REFUELLING FROM A BOWSER

The regulations concerning pressure refuelling when using a bowser compared with open line refuelling differ in only one point. The pressure refuelling connector makes a metal-to-metal connection to the aircraft, so the pressure line does not require a separate bonding connection. Diagram 10.49 is included to show the reader the usual controls found on a bowser, not for exam purposes.

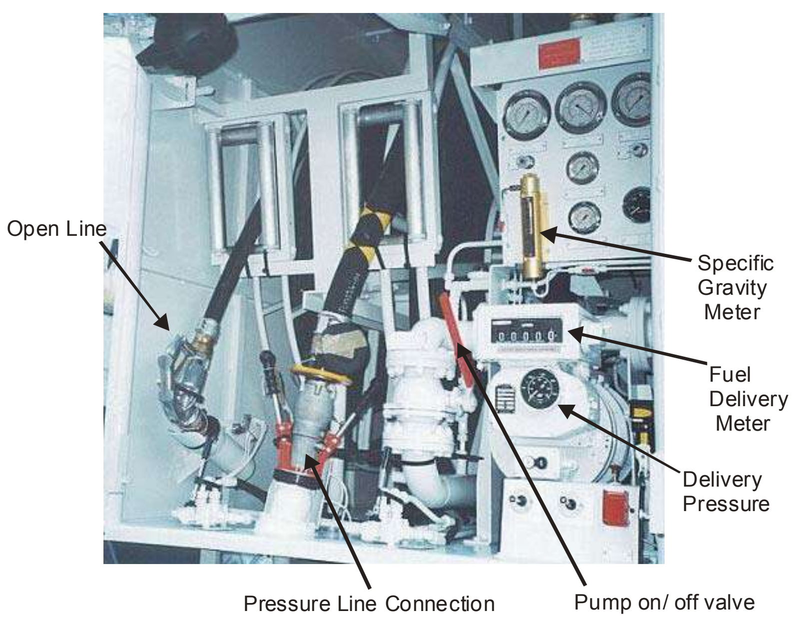


Diagram 10.49 Bowser's Refuelling Controls

Chapter 11
Ice and Rain Protection

ACKNOWLEDGEMENTS

We would like to thank and acknowledge:

For Diagrams:

11.3	The estate of Mr. A. Whitlock for the kind permission to reproduce and use this drawing from his book “Behind the Cockpit Door”
11.9	Mr. Stuart Powney

INTRODUCTION

In the early days of commercial aviation after the First World War, many aviators lost their lives when flight into bad weather led to the build-up of ice (ice accretion) on their aircraft. As instruments were rudimentary in the early 1920s, some airliner designs put the passengers inside a cabin while the pilots sat in an open cockpit. This allowed them to see and experience the atmospheric conditions in which they were flying. The only solutions to icing were either to avoid flying in those conditions that create icing, or minimise the time spent in them to reduce the build-up of ice on the airframe. Improvement in instrumentation allowed piston-powered aircraft to fly in weather conditions where the pilot cannot see the ground (Instrument Meteorological Conditions, or IMC). The gas-turbine engine has allowed aircraft to cruise above the bad weather conditions, whereas piston and turboprop aircraft, which fly in the lower levels of the atmosphere, remain in bad weather. However, remember that the gas-turbine powered aircraft have to pass through the lower-level weather on the ascent and descent.

EFFECTS OF RAIN IN FLIGHT

Leaving aside the effects of carburettor icing (which is covered in the engine notes), moderate to heavy rainfall can reduce the amount of lift created by the wings. This is best visualised by watching the rain splash on a flat surface. Every drop that hits the surface bounces some more water upward. This disturbance on a wing in flight causes the boundary layer to thicken, resulting in a loss of lift. This loss becomes significant in heavy rain where the splashing is continuous and water is running across the surface. The raindrops displacing air molecules also reduce lift—humidity decreases density.

Water on the windscreen decreases and distorts forward visibility. To overcome this on light aircraft, the designers arrange the windscreen to shed the rain by shaping it. A curved and rear-sloping windscreen enables the slipstream to carry the rain droplets rearward. The effectiveness of this method depends on the speed of the aircraft (slipstream) and the aircraft’s angle of attack. High angles of attack can create a turbulent airflow over the windscreen, which reduces the effectiveness of the slipstream in removing the rainwater.

WINDSCREEN RAIN REMOVAL SYSTEMS

From JAR Ops: *"An operator shall not operate an aeroplane with a maximum certificated take-off mass of more than 5700 kg unless it is equipped at each pilot station with a windshield wiper or equivalent means to maintain a clear portion of the windshield during precipitation."*

WINDSHIELD WIPERS

The normal system is the use of windscreen wipers. For screens that are narrow, these move from side to side across the forward window panels. For wider screens, these can be arranged to sweep like those on a car. Wipers can be electro/mechanical, electro/hydraulic, or hydraulically actuated and have two or more speeds. They are only used below certain airspeeds and are parked against stops for high-speed flight. Operation of the wipers when the windscreens are dry results in damage to the wiper blade and can score the screen. Some aircraft have a liquid de-icer system screen, which also acts as a screen wash system.

RAIN REPELLENT

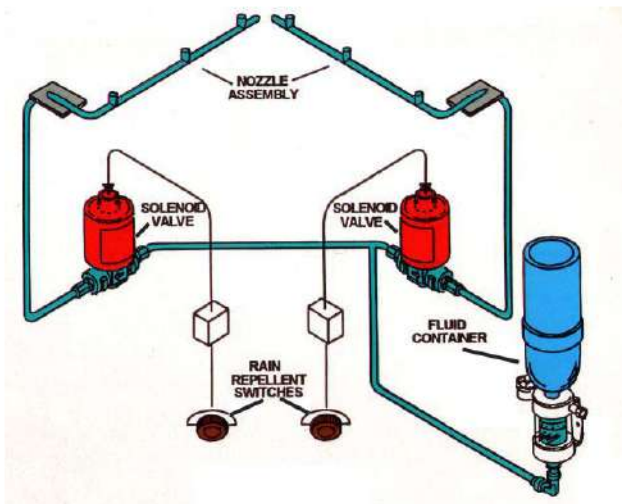


Diagram 11.1 Rain Repellent System

For aircraft that operate in climatic conditions of torrential rain (monsoons, etc.), use of a rain repellent system can assist the wipers to clear the large volume of water that would otherwise inundate them. Rain repellent is a very viscous oil sprayed on to the windscreen and spread by the wiper blades. This reduces the surface tension, so that the water flows off more easily, allowing the wipers to clear larger volumes of water than normal.

Rain repellent is stored in a pressurised container and is delivered in single shots on to the windscreens via momentary switches. Even in torrential rain, a shot of repellent reduces vision through the selected screen to effectively nil for 15 seconds until the fluid has spread out. For this reason, normally only one pilot applies the repellent at a time. The second pilot waits until the first pilot has re-established vision before applying repellent.

This fluid is very toxic and has a distinctive odour. In the event that the flight crew believes that they can smell the vapours from this fluid, they should automatically go on to 100% oxygen. If rain repellent is selected when the windscreen is dry, the wipers of the affected screen should not be used as the repellent smears and obscures vision. A disadvantage of this fluid is that it is very viscous and smears backward aft of the windscreens. Like all oils and greases, it is a magnet for dust and dirt.

BLEED AIR RAIN REMOVAL

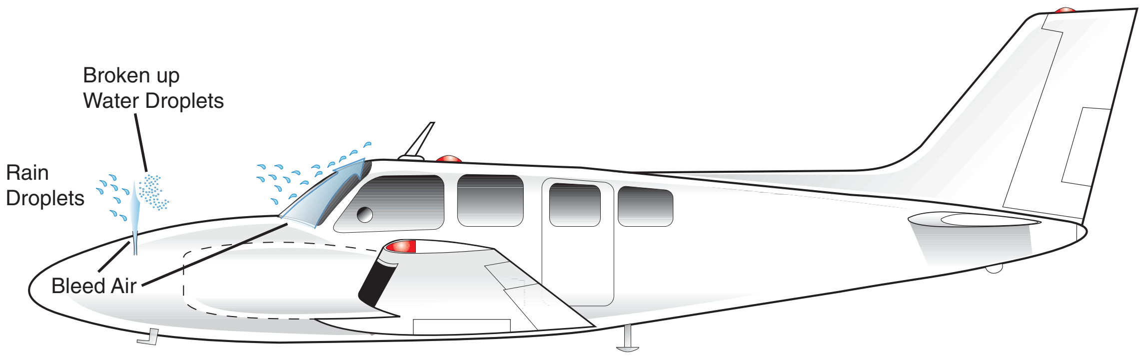


Diagram 11.2 Bleed Air Rain Dispersal

The bleed air rain removal system can be used to supplement the windscreen wipers or as the "equivalent means" to maintain a clear portion of the windshield during precipitation. In this system, air taken from a dedicated blower or bleed from the compression phases of a turbine engine is directed up the face of the windscreens to maintain a laminar flow of high velocity air over the windscreens. This sweeps the rain upward and off the panels. A bleed air duct can also be located forward of the windscreen ducting a high velocity jet upward. This acts to break up the size of the rain droplets and disperse them before they reach the windshield. Diagram 11.2 illustrates both methods.

EFFECTS OF ICE ACCRETION ON AN AIRCRAFT

Diagram 11.3 Ice

Any deposit of ice, snow, or frost on the external surfaces of an aeroplane may drastically affect its flying qualities, due to a combination of the following effects. The aircraft's weight increases by approx. 10 lb per gallon (1 kg per litre) of water that freezes on to the aircraft's surfaces. Ice results in a reduction in lift and an increase in drag due to the alteration in the profile of the lifting surfaces. Due to the changes in weight, lift, and drag, the aircraft's stability and controllability also alter.

The aircraft's performance degrades as the depth of ice increases. To overcome the effect of the loss of lift, extra weight, and drag, more power is required. This results in an increase in fuel consumption, decrease in range, endurance, and service ceiling as a minimum and can, if not controlled, lead to the aircraft crashing.

There is a real danger that ice can form in the joints between the primary control surfaces and the wings and empennage, freezing the controls solid. There is also the possibility that water, slush, or snow may be sprayed up into the undercarriage bay during take-off, freezing and locking the landing gear in the up position.

Another possible hazard is the ingestion of lumps of ice into the intake of gas-turbine engines, damaging the compressor blades, causing an out of balance rotation. Formation of ice on airscrews can have the same effect. Ice can block pitot and static ports and other sensors, which results in the failure of instruments, giving rise to false indications. It can also form around aerials, reducing the effectiveness of their transmission and reception (attenuation of signal). Ice formation in piston-engine carburetion is covered in the piston engine notes.

As seen above, ice and aircraft do not mix.

ICE FORMATIONS

There are four main types of icing: hoarfrost, rime, mixed and clear, or glaze. Each type is associated with different meteorological conditions, which are dependent on temperature and precipitation. A brief explanation of each type and the basics of their formation follow as a means of introducing the student to the process of ice detection and ice protection. Full details of icing conditions are given in the meteorological subject matter and are examined separately.

FROST/HOAR FROST

Ice crystals form via sublimation when water vapour freezes on the ground or on any other exposed object whose temperature is at or below 0°C. This forms a white crystalline structure. While this does not form a heavy blanket, it does roughen the surface of the aircraft and is akin to sticking sheets of coarse sand paper all over the skin.

In the UK, this type of ice normally forms on a clear winter night as the ambient temperature drops due to temperature radiation into space. The same type of ice formation can occur on aircraft in flight, either when a cold-soaked aircraft descends through warm clear humid air, or a cold aircraft takes off and passes through a surface inversion from cold air at ground level into warmer humid air.

RIME ICE

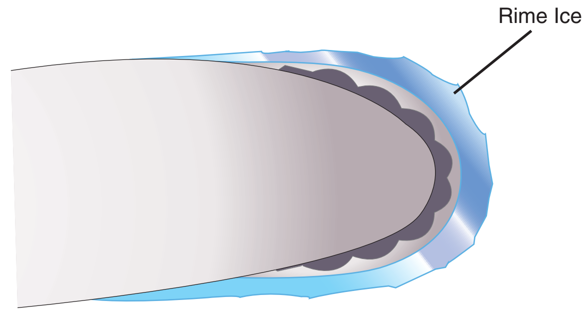


Diagram 11.4 Rime Ice Build Up on Leading Edge of Wing

Rime ice is the term given to a rough opaque white structure. This ice can form at ground level or at altitude. At ground level, it forms when the air temperature drops to the dew point causing fog, then falls below 0°C creating freezing fog. Droplets of water that are super cooled (water that has a temperature below freezing but not frozen), form the fog. However, any cold object that these droplets touch causes the droplets to freeze on contact, building up an opaque, white rime, which has an open structure and is porous. It has the same consistency as the frost build-up that forms around the icebox of a refrigerator/deepfreeze. If the fog is moving with a breeze, the windward side of the object gets the thickest coating.

In flight, this type of ice formation forms on the leading edges of an aircraft when it flies through a low-density cloud of small super cooled water droplets. The initial droplets freeze on contact with the Le, and subsequent droplets freeze on contact with the ice that has formed, trapping air between them. This gives the opaque white effect and forms a very rough surface that disrupts the airflow (see diagram 11.4).

GLAZE ICE

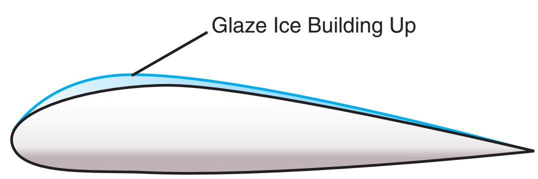


Diagram 11.5 Glaze Ice Build Up on Top Surface of Wing

Glaze ice is also known as **clear ice** or **rain ice**. This type of ice forms a transparent air free sheet and can be formed on the ground by freezing rain. On contact with an object, the water droplet starts to freeze but not instantaneously. This allows the air to bubble to the surface and dissipate.

In flight, clear ice forms when larger super cooled droplets impact on the airframe when the ambient air temperature is between -3°C and -8°C. As the droplet impacts the airframe, it starts to freeze, but the action of freezing releases heat (the latent heat of freezing) into the rest of the droplet, preventing the whole droplet from freezing instantaneously. This allows the remains of the droplet to run backward with the slip-stream, freezing more slowly as it goes, which allows any air trapped by the impact to bubble out, leaving a clear smooth sheet of ice that has very high adhesive properties. See diagram 11.5 for an illustration.

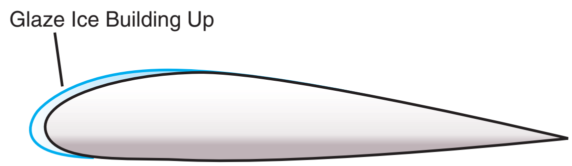


Diagram 11.6 Glaze Ice Build Up on Leading Edge of Wing

Clear ice can also be formed by an aircraft flying into a dense cloud of small super cooled water droplets. These freeze closer to the point of impact and build up a sheath of clear ice, see diagram 11.6.

MIXED ICING

Mixed icing is the term given to the formation of both glaze and rime ice. This is formed when the aircraft is flying in the transition phase between the two conditions. The surface is rough and predominantly opaque white but with clear patches.

RUN BACK ICING

Small water droplets can be super cooled to –40°C, but larger water droplets do not super cool to the same degree. The largest droplet might only be just below freezing. In this case, the droplet takes longer to freeze on impact and runs further backward. This increases the amount of ice forming on the aircraft and can result in ice forming across the gap between the wings/tailplane and the control surfaces. The ambient temperatures at which run back icing is likely to occur are between +10°C and -3°C.

ICING CONDITIONS

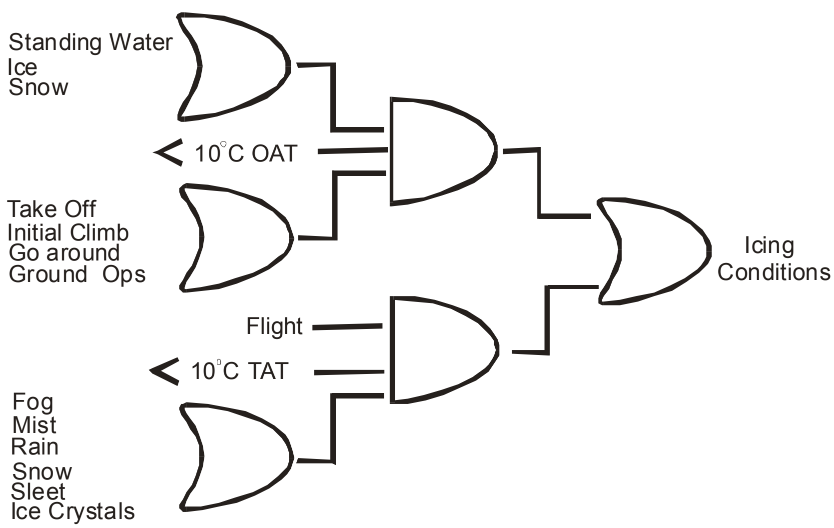


Diagram 11.7 Icing Conditions

Diagram 11.7 shows the conditions in which icing conditions exist.

Diagram 11.8 gives the International Standard Atmosphere temperatures from the Earth's surface to the Tropopause.

International Standard Atmosphere Temperature Table

Feet	Deg C	Feet	Deg C
-1000	16.98	20 000	-14.72
0	**15.00**	25 000	-24.62
+1000	13.02	30 000	-34.53
+2000	11.04	35 000	-44.44
+3000	9.06	36 000	-54.34
+4000	7.08	**36 090**	-56.32
+5000	5.09	**40 000**	**-56.5**
+6000	3.11	45 000	Temperature remains constant at **–56.5** degrees Celsius from the tropopause 36 090 ft to an altitude of 65 617 ft where it starts to rise.
+7000	1.13	50 000	
+8000	**-0.85**	55 000	
10 000	-2.83	60 000	
15 000	-4.81	65 000	

Diagram 11.8 ISA Temperature Table

From diagram 11.8, it can be seen that for a standard day at 8000 ft, the air is below freezing and drops to –56.5°C at 40 000 ft. Above 40 000 ft, all humidity has been converted into ice crystals. These do not adhere to a cold-soaked airframe.

FREEZING CONDITIONS

Freezing conditions occur when the outside air temperature is below +3°C and there is visible moisture in any form such as:

- Fog with visibility below 1.5 km
- Rain, snow, sleet, or ice crystals
- Standing water, slush, ice, or snow is present on the runway

The most critical ambient temperature range is between +3°C and -10°C. However, at much higher ambient temperatures (possibly up to +15°C or higher), ice may form on the top and underside of fuel tanks containing large quantities of cold fuel.

FREEZING DRIZZLE

Freezing drizzle is fairly uniform precipitation composed exclusively of fine drops very close together, which freeze upon impact with the ground or other exposed objects.

FREEZING FOG

Freezing fog is a suspension of numerous minute water droplets, which freeze upon impact with ground or other exposed objects forming a coat of glaze or rime ice. This suspension generally reduces the horizontal visibility at the Earth's surface to less than 1 km (5/8 mile).

FREEZING PRECIPITATION
Freezing precipitation corresponds to freezing rain or freezing drizzle.

LIGHT FREEZING RAIN
This is precipitation of liquid water particles, which freeze upon impact with exposed objects.

SLEET
Sleet is precipitation in the form of a mixture of rain and snow. For operation in light sleet, treat as light freezing rain.

SLUSH
Slush is snow or ice reduced to a soft watery mixture by rain, warm temperature, and/or chemical treatment.

SNOW
Snow is precipitation of ice crystal, most of which are branched, star-shaped, or mixed with unbranched crystal. At a temperature higher than -5°C, the crystals are generally agglomerated into snowflakes.

COLD-SOAKED WING
Rain or high humidity on a cold-soaked wing results in the formation of ice or frost on the wing surface when the temperature of the aeroplane's wing surface is at or below 0°C.

CLEAN AIRCRAFT POLICY

Diagram 11.9 Snow Contaminated Dakota

The JAA and FAA have a Clean Aircraft Policy. JAR-OPS 1.345-Ice and other Contaminants provides this stating:

"An operator shall establish procedures to be followed when ground de-icing and anti-icing and related inspections of the aeroplane(s) are necessary."

"A Commander shall not commence take-off unless the external surfaces are clear of any deposit, which might adversely affect the performance and/or controllability of the aeroplane except as permitted in the Aeroplane Flight Manual."

"A Commander shall not commence a flight under known or expected icing conditions unless the aeroplane is certificated and equipped to cope with such conditions."

This puts the onus on operator and captain to ensure that the whole aircraft is clear and free of ice and snow before it starts its take-off run. Operators and pilots must not try to rely on loose snow being removed by the slipstream in the take-off run.

PRE-TAKE-OFF CHECK

This check ensures that the representative surfaces of the aeroplane are free of ice, snow, slush, or frost just prior to take-off. This check should be accomplished as close to the time of take-off as possible and is normally made from within the aeroplane by visually checking the wings or other critical surfaces, defined by the aeroplane manufacturer.

GROUND DE-ICING AND ANTI-ICING

To meet the clean aircraft policy, aircraft have to have all snow and ice removed before flight and if icing conditions exist, ice must be prevented from reforming. The term de-icing is applied to removing ice and snow that already exists, and anti-icing or ant-icing refers to the prevention of further contamination. Below are the various areas and the method of removing contaminants.

REMOVING VERY DRY SNOW FROM A COLD-SOAKED AIRCRAFT

In a dry continental climate with very low sub zero temperatures, the snow is very dry (i.e. when a handful of snow is squeezed, it does not form a snowball). As the snow does not adhere to the aircraft, the external surfaces can be cleaned with a soft brush. If the temperature remains below zero and there is no precipitation, anti-icing action is not required.

Care must be taken to ensure that all snow is removed from the hinge joints of the control surfaces, wheel bays, air intakes, pitot, and static vents. This type of snow blows. As a precaution, when parking, all blanks and bungs should be fitted to stop the ingress of moisture or blown snow that can freeze and block sensing lines and air intakes.

REMOVAL OF SNOW, SLEET, SLUSH, AND ICE

In maritime climates where temperatures at sea level do not fall to the sub-arctic temperatures required to produce dry snow, the snow in these regions is moister and forms snowballs. It can also fall as sleet, which is a mixture of wet snow and rain. Temperatures can rise and fall subsequent to a snowfall, semi-melting the snow to form slush, and then refreezing it.

In these situations, brushing is of no use and chemical de-icing fluids are used to melt and wash away the contaminants. These chemicals are referred to as **freezing point depressants** (FPDs) as they lower the freezing point of the water. There are different types of de-icing fluid. They are designated by International Standards Organisation (ISO) by number (i.e. ISO type 1 fluid, etc.), each type of fluid having different properties. As with car radiator anti-freeze, these fluids can be diluted with water before application. The OAT at the airport at which the de-icing/anti-icing operation is to be carried out governs the type of fluid, its dilution strength, and method of application.

ONE STEP DE-ICING/ANTI-ICING

In this system, fluid is pressure sprayed onto the aircraft. The fluid can be heated so that de-icing is effected in three ways: pressure, heat, and lowering the freezing point. With a one-step de-icing/anti-icing procedure, the holdover time begins at the commencement of de-icing/anti-icing. If the aircraft takes off within the time limit of the holdover, no further treatment is required.

TWO STEP DE-ICING/ANTI-ICING

In this system, a fluid is pressure-sprayed onto the aircraft. The fluid can be heated, so that a triple effect is gained to remove all ice and snow, etc., which the pressure, the heat, and the FPD effect. Following this, when the aircraft has been cleared of snow and ice, a further treatment is carried out using a different or more concentrated fluid to give a longer holdover time.

With a two-step procedure, the holdover time begins at the commencement of the second (anti-icing) step.

HOLDOVER TIME

When de-icing fluid is sprayed onto an aircraft, it becomes diluted and due to the curvature of the fuselage and wings, it and the melt water run off. The time the fluid remaining on the airframe remains active (anti-icing) is termed **holdover time**.

Holdover times for a fluid vary with the prevalent climatic conditions, the dilution strength of the fluid used, the type of aircraft, and its location in relation to other aircraft (jet blast). The holdover time, in effect, runs out at the commencement of take-off roll or when frozen deposits start to form/accumulate at aeroplane surfaces. Aircraft must take off within the holdover time of the fluid or they have to be re-treated.

However, even if aircraft take off well within the holdover time, it (the holdover time) is not meant to imply that flight is safe in the prevailing conditions, as these fluids are displaced by the slipstream and evaporate due to the increase in airflow.

TYPE I FLUID

Type I fluids have a low viscosity with a minimum of 80% glycol, the FPD agent. This results in the fluid forming a thin, liquid-wetting film, which runs off the surfaces to which it is applied. This, combined with the fact that melting snow/ice easily dilutes the glycol and it readily evaporates, means that the fluid has limited holdover capabilities. Prevailing weather conditions can further reduce these.

With ISO Type I fluids, increasing the concentration of fluid in the fluid/water mix does not provide any increase in holdover time. Where type I fluid is used as the first step of a two-step operation, its freezing point must not be higher than 3°C above the ambient temperature and as the second step, or as the fluid in a one-step operation, not higher than 10°C below the ambient temperature.

ISO TYPE I FLUID HOLDOVER TIMES

OAT	Approximate Holdover Times Under Various Weather Conditions Holdover Time in Minutes					
Celsius	Frost *	Freezing Fog	Snow	Freezing Drizzle **	Light Freezing Rain	Rain or Cold Soaked Wing
above 0	45	12 to 30	6 to 15	5 to 8	2 to 5	2 to 5
0 to -10	45	6 to 15	6 to 15	5 to 8	2 to 5	
below -10	45	6 to 15	6 to 15			
* During conditions that apply to aircraft protection for active frost						
** Use light freezing rain holdover times if positive identification of freezing drizzle is not possible						
Caution The time of protection is shortened in heavy weather conditions. Heavy precipitation rates or high moisture content, high wind velocities, or jet blast may reduce holdover time below the lowest stated in the range. Holdover time may also be reduced when aircraft skin temperature is lower than OAT. Therefore, the indicated times should be used only in conjunction with a pre-take-off check						

Diagram 11.10 Table 1 Guidelines for Holdover Times for ISO Type 1 Fluid Mixtures as a Function of Weather Conditions and OAT from Pink AIC169

TYPE II/IV FLUID

Type II/IV fluids have a minimum of 50% glycol, the FPD agent, and contain a thickening agent, which increases the fluids viscosity and enables them to form a thicker liquid-wetting film on surfaces to which it is applied, which remains on the airframe for a longer period.

The thickening agent also acts to slow the evaporation rate of the glycol, enabling the fluid to provide longer holdover times than Type I fluids in similar conditions. With this type of fluid, the holdover time can be raised by increasing the concentration of fluid in the fluid/water mix, up to the maximum holdover time available from undiluted fluid.

Mixture Strength Fluid/Water	Lower Temperature Limit for Application OAT
% glycol/ hot water 0/100	-3°C
25/75	-6°C
50/50	-13°C
75/25	-23°C

Diagram 11.11 Anti-Ice Mixture/Temperature Table

Diagram 11.11 shows the mixtures that are used for de-icing and the lowest temperature to which they can be used (i.e. an aircraft can be de-iced with hot water only down to an OAT of -3°C after which it would have to be anti-iced with a 50/50 mixture, as illustrated in diagram 11.12).

Mixture Strength Fluid/Water	Lower Temperature Limit for Application OAT
50/50	-3°C
75/25	-14°C
100/0	-25°C

Diagram 11.12 Anti-Ice Mixture/Temperature Table

ISO TYPE II FLUID HOLDOVER TIMES

OAT	Fluid Concentration	Approximate Holdover Times Under Various Conditions Holdover Time in (Hour : Minutes)					
Celsius	Fluid /Water Vol/ Vol %	Frost *	Freezing Fog	Snow	Freezing Drizzle ***	Light Freezing Rain	Rain or Cold Soaked Wing
above 0	100 / 0	18.00	2:30 - 3:00	0:45 - 1:25	0:40 - 1:00	0:35 - 0:55	0:10 - 0:50
	75 / 25	6:00	1:05 - 2:00	0:20 - 0:40	0:30 - 1:00	0:15 - 0:30	0:05 - 0:35
	50 / 50	4:00	0:20 - 0:45	0:05 - 0:20	0:10 - 0:20	0:05 - 0:10	
0 to -3	100 / 0	12.00	2:20 - 3:00	0:35 - 1:00	0:40 - 1:00	0:35 - 0:55	
	75 / 25	5:00	1:05 - 2:00	0:20 - 0:35	0:30 - 1:00	0:15 - 0:30	
	50 / 50	3:00	0:20 - 0:45	0:05 - 0:15	0:10 - 0:20	0:05 - 0:10	
-3 to -14	100 / 0	12.00	0.40 - 3:00	0:20 - 0:40	0:30 - 1:00**	0:35 - 0:45**	
	75 / 25	5:00	0:35 - 2:00	0:15 - 0:30	0:30 - 1:00**	0:35 - 0:45**	
-14 to -25	100 / 0	12.00	0.20 - 3:00	0:15 - 0:30			
below -25	100 / 0	ISO Type II Fluid may be used below -25^{o}C provided that the freezing point of the fluid is at least 7^{o}C below the actual OAT and the aerodynamic acceptance criteria are met. Consider using ISO type I when ISO type II fluid cannot be used. See table 1					

* During conditions that apply to aircraft protection for active frost

** The lowest used temperature is limited to -10^{o}C

*** Use light freezing rain hold over times if positive identification of freezing drizzle is not possible

Caution The time of protection will be shortened in heavy weather conditions. Heavy precipitation rates or high moisture content, high wind velocities or jet blast may reduce holdover time below the lowest stated in the range.

Holdover time may also be reduced when aircraft skin temperature is lower than OAT. Therefore the indicated times should be used only in conjunction with a pre- take off check

ISO Type II fluids used during ground de-icing / anti-icing are not intended for and do not provide ice protection during flight

Diagram 11.13 Table 2 Guidelines for Holdover Times for ISO Type II Fluid Mixtures as a Function of Weather Conditions and OAT from Pink AIC 169

ISO TYPE IV FLUID HOLDOVER TIMES

OAT	Fluid Concentration	Approximate Holdover Times Under Various Conditions Holdover Time in (Hour : Minutes)					
Celsius	**Fluid /Water Vol/ Vol %**	**Frost** *	**Freezing Fog**	**Snow**	**Freezing Drizzle** ***	**Light Freezing Rain**	**Rain or Cold Soaked Wing**
above 0	**100 / 0**	18.00	2:30 - 3:00	0:45 - 1:25	0:40 - 1:00	0:35 - 0:55	0:10 - 0:50
	75 / 25	6:00	1:05 - 2:00	0:20 - 0:40	0:30 - 1:00	0:15 - 0:30	0:05 - 0:35
	50 / 50	4:00	0:20 - 0:45	0:05 - 0:20	0:10 - 0:20	0:05 - 0:10	
0 to -3	**100 / 0**	12.00	2:20 - 3:00	0:35 - 1:00	0:40 - 1:00	0:35 - 0:55	
	75 / 25	5:00	1:05 - 2:00	0:20 - 0:35	0:30 - 1:00	0:15 - 0:30	
	50 / 50	3:00	0:20 - 0:45	0:05 - 0:15	0:10 - 0:20	0:05 - 0:10	
-3 to -14	**100 / 0**	12.00	0.40 - 3:00	0:20 - 0:40	0:30 - 1:00**	0:35 - 0:45**	
	75 / 25	5:00	0:35 - 2:00	0:15 - 0:30	0:30 - 1:00**	0:35 - 0:45**	
-14 to -25	**100 / 0**	12.00	0.20 - 3:00	0:15 - 0:30			
		ISO Type II Fluid may be used below –25°C provided that the freezing point of the fluid is at least 7°C below the actual OAT and the aerodynamic acceptance criteria are met. Consider using ISO type I when ISO type II fluid cannot be used. See table 1					
below -25	**100 / 0**						

* During conditions that apply to aircraft protection for active frost

** The lowest used temperature is limited to –10°C

*** Use light freezing rain hold over times if positive identification of freezing drizzle is not possible

Caution The time of protection will be shortened in heavy weather conditions. Heavy precipitation rates or high moisture content, high wind velocities or jet blast may reduce holdover time below the lowest stated in the range.

Holdover time may also be reduced when aircraft skin temperature is lower than OAT. Therefore the indicated times should be used only in conjunction with a pre- take off check

ISO Type IV fluids used during ground de-icing / anti-icing are not intended for and do not provide ice protection during flight

Diagram 11.14 Table 3 Guidelines for Holdover Times for ISO Type II Fluid Mixtures as a Function of Weather Conditions and OAT from Pink AIC 169

APPLICATION OF DE-ICING AND ANTI-ICING FLUIDS

De-icing and anti-icing operations can be carried out either when the aircraft is stationary and shut down or with passengers and crew embarked and the engines running. At some locations when the holdover times are low, a further application of anti-icing fluid is applied just prior to the aircraft entering the runway to line up for take-off. Some large airports in the USA have fixed installations in the form of gantries from which anti-icing fluid can be sprayed onto the upper surfaces of air transport aircraft.

The following precautions must be taken when spraying these fluids onto an aircraft to ensure that it is not damaged or the crew and passengers intoxicated with the fumes from the fluid.

- Except when the engines are running, all intakes must be blanked.
- Bleed air valves for air conditioning must be closed.
- Fluid must not be directed into:
  - Wheel bays
  - Orifices and vents
  - Pitot tubes
  - Static vents
  - Sensing lines
  - Gaps between trailing edges of wings, etc. and control surfaces or hinge lines.
- Exposed linkages must be checked to ensure that the fluid has not de-greased them.
- Fluid must not be used on cabin windows unless it is compatible with the window material.

HOT WATER DE-ICING

The water is heated to a maximum temperature of 95°C and pressure sprayed onto the airframe to melt and blast away the snow/ice. After a surface has been de-iced, it must either be dried or treated within three minutes.

FLUID DE-ICING

De-icing fluids are normally applied heated to a minimum temperature of 60°C at the nozzle in order to assure maximum efficiency and sprayed at a pressure of 100 psi.

FLUID ANTI-ICING

Normally 100% concentrations of type II/IV fluids are used for anti-icing purposes on uncontaminated aeroplane surfaces. Anti-icing fluid is normally applied unheated. However, in extremes of climate, they can be heated to a max of 60°C. If the fluid is overheated in application, the thickening agent forms into a gel. This can have a more detrimental effect on the aircraft than the ice it was supposed to prevent.

Operators and pilots must also be aware that if a thickened de-icing/anti-icing fluid applied to an aircraft is allowed to dry on, it can become a hazard in its own right, as it dries with an uneven surface and holds dirt, resulting in disruption of the airflow over the lifting surfaces, which should be smooth.

COLD FUEL TANKS

If an aircraft lands after a sector with a large quantity of fuel remaining, and has become cold soaked due to temperature of altitude at which it flew, the surface of the wings can be lower than that of the airport's ambient air temperature. In this case, a stronger mix of anti-icing fluid would be required than that indicated by the OAT. This difference makes a buffer in maintaining the freezing point of the fluid and the surfaces of the wing under and over the fuel tanks.

UNDERCARRIAGE BAYS

To prevent the landing gear from becoming locked in the retracted condition by ice, the bays must be cleared of ice, sleet, snow, and any pools of water prior to take-off. De-icing fluid must not be used. While this melts the contamination, not all the water/fluid drains from the bay. This mixture is either very diluted (the small amount of fluid soon evaporates), or insufficient to contend with the reduced OAT as the aircraft climbs, thus turning to ice and creating the situation that was trying to be averted.

Undercarriage bays must either be cleaned by hand or by using a warm air heater to drive the contaminant out and dry the bay. If the aircraft is taking off from a contaminated runway, the pilot should cycle the undercarriage several times to allow the slipstream to blow any contamination away before retracting the gear for the flight.

FIXED UNDERCARRIAGES

Aircraft, with fixed undercarriages and wheel fairings (spats), that land on contaminated runways or un-prepared surfaces in icing conditions must have any mud or contamination removed from the inside of the fairing before the next flight, as this could freeze and lock the wheel resulting in a tyre blow out on take-off or landing. Pilots may remove and clean the spats, but an engineer must refit them.

TAXIING

In icing conditions, pilots must be aware of the jet blast and prop wash hazard to their aircraft and should avoid taxiing or holding close to other aircraft. The increased velocity of the jet blast and prop wash can lift snow and ice from the taxiway and fling it in to the following aircraft's engines/props. It is a fallacy to believe that the heat of jet blast prevents ice from forming on a following aircraft. It drives the anti-icing fluid off the wings by evaporation and slipstream, leaving the aircraft unprotected.

ICING CONDITIONS IN FLIGHT — INTENSITY

Apart from the different types of ice that form in different temperatures, there are three categories of rates of ice build-up: light, moderate, and heavy (severe). Each is determined by the rate at which it forms on the airframe. Each aircraft is certified as to which conditions it can fly in. Aircraft that are not certified for light icing must not intentionally enter any meteorological conditions that could lead to icing. For air transport aircraft certified under JAR 25, there are two categories: continuous maximum icing and intermittent maximum icing.

DE-ICING PASTE

To afford some protection from icing to light aircraft that are not equipped with de-icing or anti-icing systems should they encounter icing condition in flight, a thin smear of a paste containing an FPD can be applied to the leading edges. The paste acts in two ways. The FPD lowers the freezing point of any moisture in contact with it and prevents ice that has formed from adhering to the leading edges. The paste erodes over time and must be freshly applied before each flight. This is not to allow a pilot to fly into known icing conditions but to allow the pilot to get out of icing conditions that might be encountered during a flight.

ICE DETECTION IN FLIGHT

As ice forms on thin edges first, each aircraft has points, such as the windscreen wiper arms, where the formation of ice indicates that the aircraft has entered into icing conditions. Locations on the airframe where ice build-up affects the controllability or stability of an aircraft are termed critical points, such as the leading edge of the wings, tailplane, and fin.

As ice forms on thin edges more quickly, and the leading edge of the tail and fin are normally thinner than those of the wings, the leading edge of the tailplane is often one of the first critical points to become iced. Pilots must be aware of the effect that icing on the tailplane's leading edge can have on the airflow. This is especially important as the lowering of the trailing edge flaps can cause the aircraft to violently pitch nose down. The only remedy is to raise the flaps.

Pilots are advised not to use the autopilot in icing conditions, as it will constantly correct by trimming the out of balance condition created by build-up of ice, then either fail as they reach the end of the trim range, or give the unwary crew an unpleasant surprise when they go to manual. This recommendation is a result of the ATR 42 accident at Roselawn in the USA.

To assist the pilot, there are several different types of ice detector that can be fitted to aircraft to assist the pilot in identifying icing conditions. These operate in different ways and are listed below:

1. Icing conditions exist:
    a. Temperature/moisture probes

2. Icing has formed:
    a. Visual
        i. Ice detection light
        ii. Hot rod detector
    b. Pressure
        i. Smiths ice detector
    c. Torque
        i. Serrated rotor
    d. Vibration frequency
        i. Vibrating rod ice detector

TEMPERATURE/MOISTURE PROBES

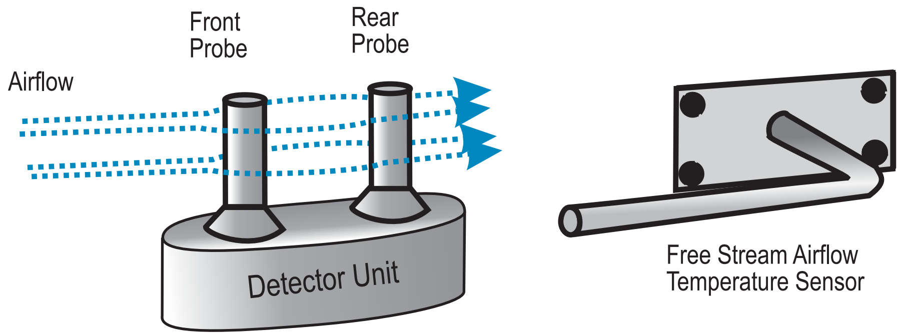


Diagram 11.15 Temperature and Moisture Probes

Unlike the other detectors, the Sangamo Western Ice Detector works on the principle of detecting when icing conditions exist, not when ice has actually formed. Working on the principle that ice cannot form unless there is moisture and freezing temperature, the detector system consists of two separate units. These are a moisture detector and a temperature sensor termed a **thermal switch**. See diagram 11.15.

The moisture detector consists of a unit that has two cylindrical rods that are mounted close together. When they protrude into the free stream airflow, the forward rod screens the rear rod. Both rods are heated resistance bulbs. As the forward bulb shields the rear bulb, it cools more quickly than the rear bulb. Any moisture in the free stream airflow impinges on the forward bulb only. This acts to cool the forward bulb down even more.

A controller built into the base of the moisture probe senses the temperature difference between the two probes. If it drops to a predetermined value, it sends a signal to the thermal switch. The temperature sensor of the thermal switch is exposed to the ambient temperature of the free stream airflow. While the OAT remains above freezing, the switch remains open and prevents the signal from being sent to the flight deck. As soon as the temperature drops, the switch closes, and any moisture signal illuminates an amber ice warning light on the flight deck.

ICE DETECTION LIGHT

Unless an aircraft is not certified to operate at night, or is prohibited from night operations, it must be equipped with a light that illuminates those parts of the wing that are deemed critical for ice accumulation. The lighting must not cause glare or reflection that would interfere with the flight crew. For large air transport aircraft, this is achieved by mounting a spot lamp in the wing root fillet panel ahead of the leading edge of the wing, so that it illuminates the length of the wing.

HOT ROD ICE DETECTOR

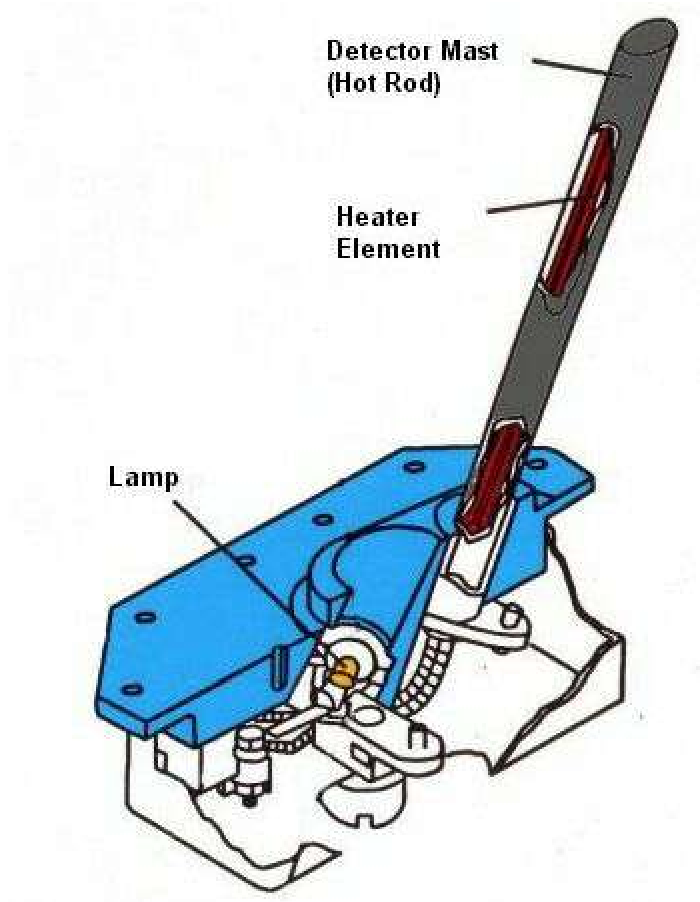


Diagram 11.16 Hot Rod Ice Detector

The Hot Rod Ice Detector, proper name being the **Teddington ice detector**, is a visual indicator, as seen in diagram 11.16. It consists of a thin rod with an aerofoil cross section with a heater element running down its centre, which is controlled by an on/off switch on the flight deck. The rod is angled backward and protrudes into the airflow. A small lamp mounted in the unit's housing illuminates the rod for nighttime operations. Either one rod is mounted where both pilots can observe it or two rods are fitted. Each is located where a pilot can see it but so that it does not interfere with normal vision.

When flying in possible icing conditions, the pilot keeps a regular watch on the rod. Ice forms on its thin edge before forming on the lifting surfaces of the aircraft. When ice is observed, the rate of accretion indicates the severity of the conditions. The pilot can apply heat to the rod or mast to clear it, then switch the heat off, and watch how quickly the ice reforms.

SMITHS ICE DETECTOR

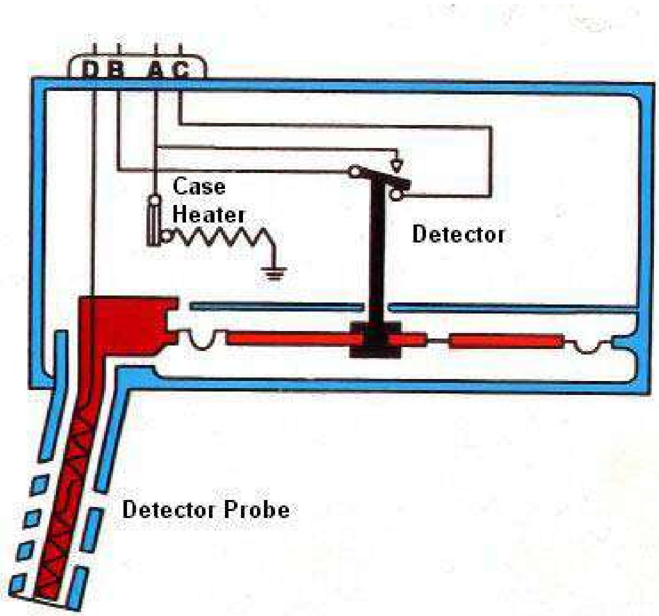


Diagram 11.17 Smiths Pressure-Operated Ice Detector

The Smiths ice detector, shown in diagram 11.17, is a pressure-operated unit that detects the presence of ice. In this design, a detector housing mounts inside the fuselage with a probe protruding into the free stream airflow. The probe, a hollow cylindrical tube with one end sealed and heater element core, has four small holes facing forward and two large holes facing aft.

In non-icing conditions, the positive air pressure (pitot and static) in the detector unit holds the switch in the open condition. In icing conditions, the formation of ice on the front edge quickly blocks the four small holes. This results in a reduction of pressure in the detector (static only), which results in the switch closing. As the switch closes, an amber ice indication light on the flight deck illuminates and power is applied to the heater in the probe. The power is supplied to both the heater and the ice warning light until the ice melts and the positive pressure is re-established. The severity of icing is indicated by the frequency at which the ice warning light illuminates.

SERRATED ROTOR ICE DETECTOR

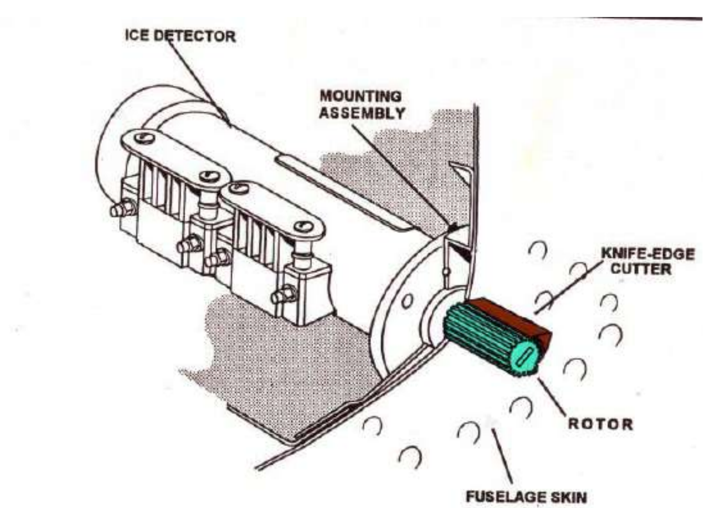


Diagram 11.18 Serrated Rotor Ice Detector

Refer to diagram 11.18. The serrated rotor ice detector consists of a housing mounted inside the airframe with a protruding serrated rotor and fixed knife, which is mounted at right angles to the aircraft's skin with the rod in front of the knife. During flight, an electric motor inside the housing rotates the serrated rotor at a constant rate toward the leading edge of the knife. These have a minimum clearance. In non-icing conditions, the torque required to turn the rotor is at a low level. However, as ice forms on the rotor, the torque increases as the knife shaves the ice off. The increased torque makes a micro switch and illuminates an ice warning light on the flight deck. This remains on as long as ice is forming on the rotor and is shaved off.

VIBRATING ROD ICE DETECTOR

The Vibrating Rod is the most modern of the ice detection systems and consists of a rod that protrudes into the free stream airflow from a detector unit housing. See diagram 11.19. These are normally mounted on the underside of the fuselage. The rod is vibrated by an electrical impulse at a set frequency of 40 kHz.

In non-icing conditions, the rod maintains this frequency. However, as ice forms on the rod, its mass increases and vibration frequency decreases. A drop in frequency activates the ice warning light on the flight deck and applies power to a heater element within the rod. When the ice has been cleared, the vibration frequency of the rod is restored and power to both the warning light and the indication is removed. The rod continues to vibrate at 40 kHz until ice forms on the rod again. Severity of icing is determined by the frequency at which the warning light illuminates.

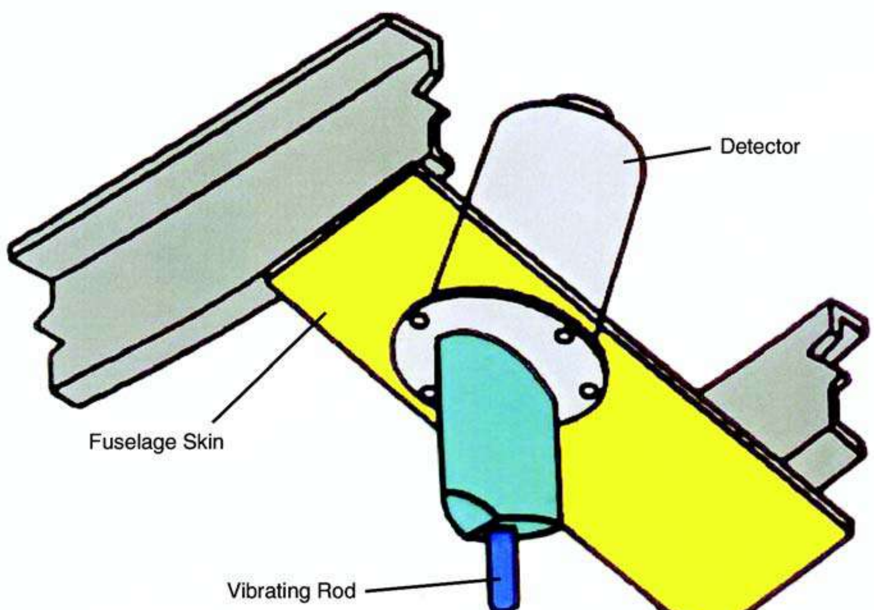


Diagram 11.19 Vibrating Rod Ice Detector

ICE PROTECTION

There are several different systems for protecting aircraft from ice build-up, which can be classified into two groups, de-icing systems and anti-icing systems:

- Pneumatic de-icing
- Liquid de-icing
- Electrical de-icing
- Electrical anti-icing
- Thermal anti-icing

The areas to be protected are:

- Leading edges of the wings, tailplane, and fin
- Jet engine air intakes
- Windshield
- Pitot probes
- Propellers

LEADING EDGE PROTECTION

For leading edge protection, slower aircraft such as light piston powered and twin-engine turboprop aircraft normally use pneumatic de-icing boots. Large, older multi piston-engine aircraft and small light corporate jets often use liquid de-icing systems. Large air transport aircraft use thermal anti icing.

PNEUMATIC DE-ICING BOOTS

Pneumatic boots, often referred to as just boots, are a mechanical de-icing system fitted to the leading edge of the aircraft's wings, tailplane, and fin. The boots consist of a shaped rubber sleeve (boot) that conforms to the leading edge of the surface to which it is attached. Within the boot are a series of tubes that can be inflated. For narrow wing profiles such as light aircraft and outer wing sections of larger aircraft, these tubes are orientated with the wing's span and are termed spanwise. For thicker cross sections, the tubes are orientated at right angles to the wingspan and termed chordwise.

When the tubes are inflated, they expand outward. This action cracks the ice that has formed on the leading edges, allowing the slipstream to catch the ice and break it off in chunks. To be effective, the ice must be between ¼- and ½-inch thick over the boots. If the boots are operated before the ice is ¼-inch thick, the rime ice is sufficiently flexible to expand with the boots without cracking and form what is termed a **bridge**. The boots continue to operate within the arch of the bridge without breaking it, allowing further ice accretion. If the ice is thicker than ½-inch, it can become too strong and prevent the boots from inflating.

The boots have to be inflated with dry air to prevent any ingress of moisture that could freeze and prevent the boots from inflating. When the boots are being deflated, or when they are not being used, a vacuum suction pressure is applied to them to ensure that they remain fully deflated to maintain the profile of the leading edge.

Before flight, the boots should be inspected for signs of scuff damage or holes. If within limits laid down by the manufacturer, they can be repaired by a vulcanised patch (like an inner tube). All patches must be secure. The boots have an outer sheath of neoprene rubber (which is annually polished to maintain them) to ensure that they are conductive to static electricity and are bonded to the airframe.

LIGHT AIRCRAFT DE-ICER BOOTS

Diagram 11.20 shows de-icer boots fitted to a light twin piston-engine aircraft. This system is manually operated with a momentary switch when the pilot detects that the ice has built up to the correct depth. The boots are pneumatically connected to the instrument air vacuum manifold downstream of the instruments, and the exhaust air from the engine-driven vacuum pumps via an inflation-deflation valve.

When not in operation, the valve applies the vacuum pressure of 2.2 psi from the instrument manifold to the boots. When the pilot selects the de-ice, the valve closes the suction side and diverts the exhaust air from the vacuum pump into the pneumatic system to inflate all the spanwise boots simultaneously.

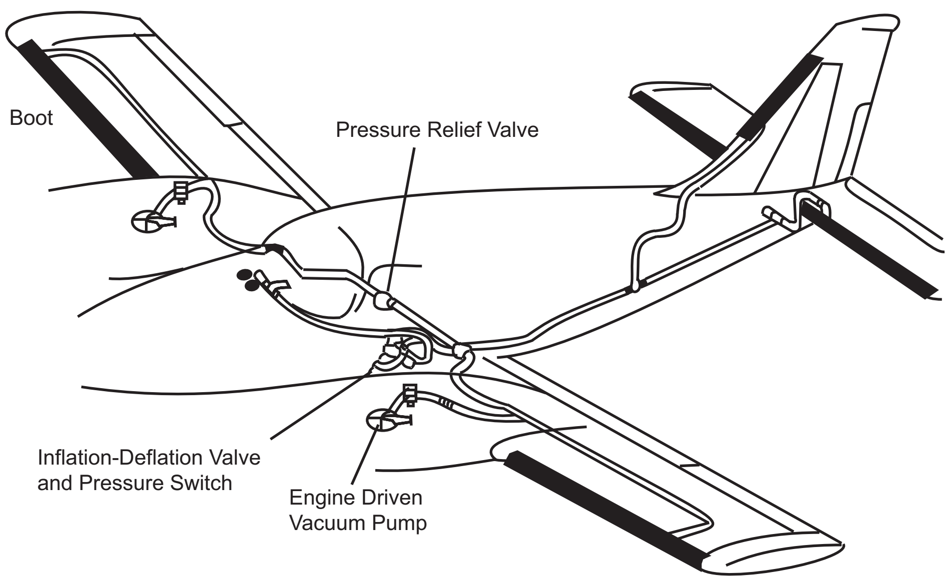


Diagram 11.20 De-icer Boots on a Light Twin

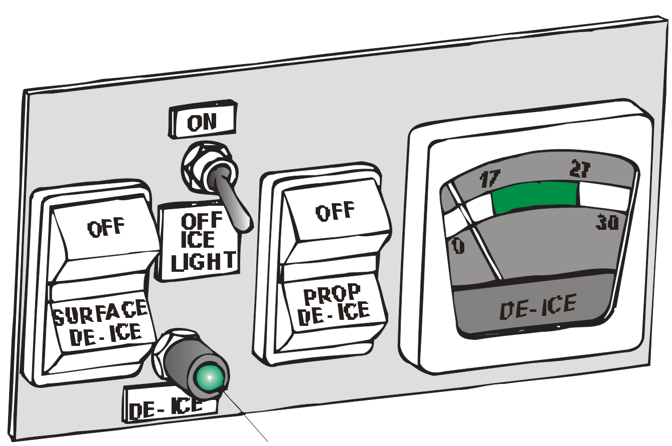


Diagram 11.21 De-Icing Control Panel for a Light Aircraft

The boots are inflated to a nominal pressure of 18 psi. At this pressure, a pressure switch then closes the inflation valve and opens the deflation valve. To confirm to the pilot that they are functioning correctly, a green light (see diagram 11.21) illuminates at 8 psi and remains illuminated until the pressure drops below 8 psi. A **timer unit** starts to count down to prevent them from remaining inflated should the pressure not reach 18 psi, or the pressure switch fails as the inflation valve opens. If the inflation sequence has not started before the timer unit reaches zero, it automatically opens the deflation valve and closes the inflation valve. The whole operation from start of inflation to complete deflation should not exceed 34 seconds. If the pilot notices that the green 8 psi light remains illuminated, the circuit breakers for the surface de-icer should be pulled.

TWIN-ENGINE TURBOPROPS

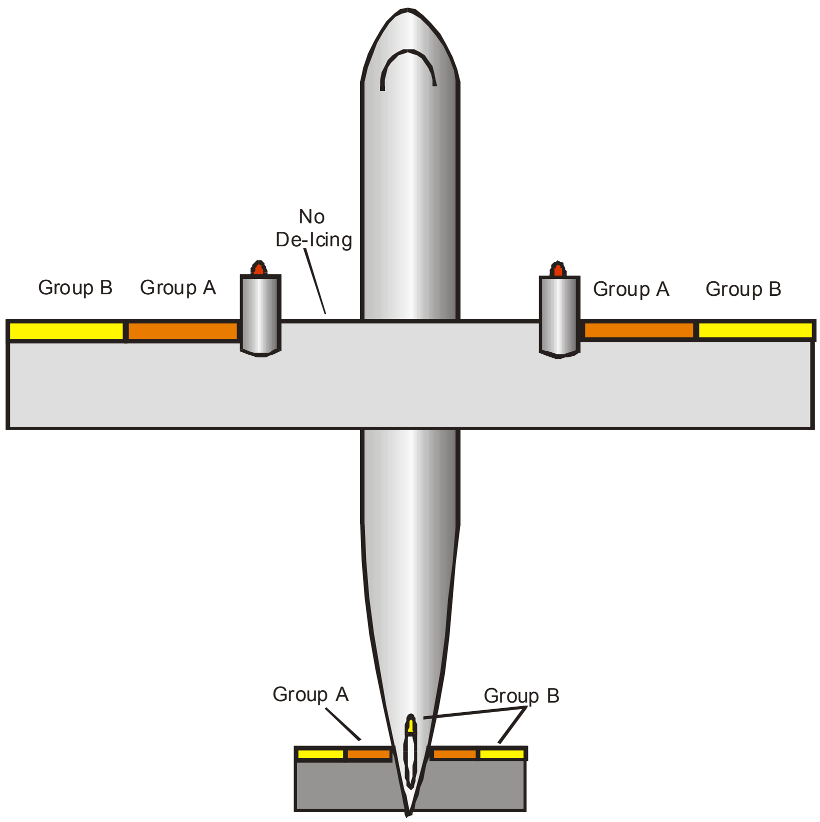


Diagram 11.21a Turboprop Aircraft Fitted with Boots

For twin-engine turboprops and larger piston-engine aircraft, operating all the boots at one time requires too much air. Therefore, the boots divide into groups, as diagram 11.21a shows. All the boots belonging to a group are inflated simultaneously to ensure that the aircraft remains balanced. As group A is deflated, group B is inflated. This ensures that the whole aircraft is de-iced within 34 seconds. Where boots are fitted around large chords, a further refinement is to make alternate chordwise tubes within a group inflate and deflate.

As these aircraft remain in icing conditions for greater periods of time, they have two ways of controlling the operation of the boots; a momentary selection giving one complete cycle referred to as manual, and continuous cycling. Continuous cycling has two positions, one each for light icing and heavy icing. In light icing conditions, the accretion rate is slower. Therefore, a cyclic timer rests the boots for 206 seconds after the boots have deflated before starting the next inflation cycle.

For heavy icing where the catch rate is higher, the rest period is reduced to 26 seconds. If the cyclic timer fails, the crew has to operate the manual switch. Air for the operation of the boots can be taken from a dedicated compressor or as bleed air from the turboprop's engine. The wing section inboard of each engine is not usually fitted with de-icer boots, as the ice does not readily form here due to the thickness of the chord and the blast of prop wash, which is accelerated by the proximity of the fuselage.

AIRFRAME DE-ICING FLUID SYSTEM

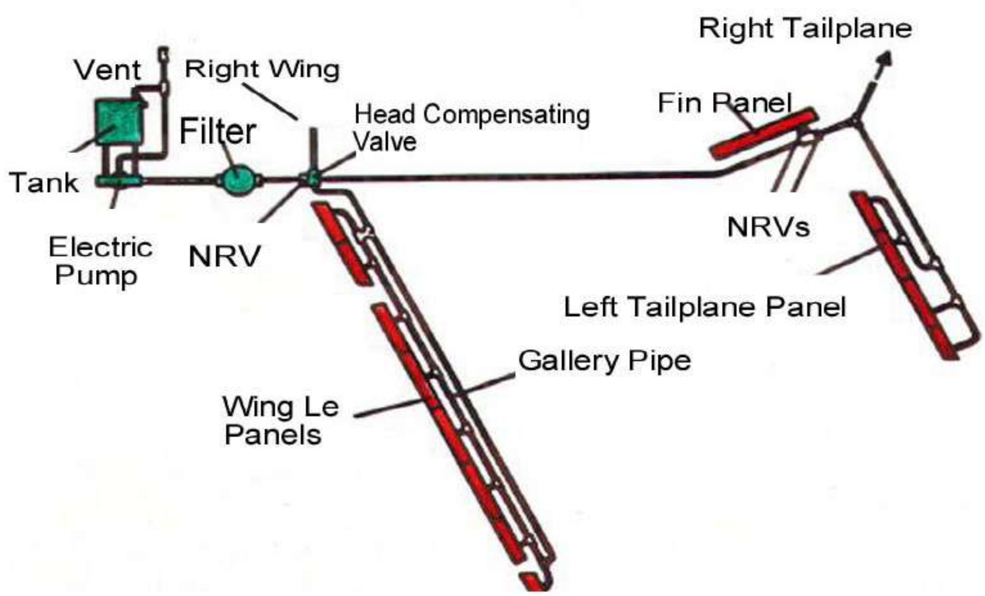


Diagram 11.22 Fluid De-Icing System

Diagram 11.22 is a schematic representation of a T-tailed aircraft's fluid de-icing system. These systems are also referred to as **weeping** or **wet wings**. The system consists of a fluid reservoir, an electrical pump, filter, head-compensating valve, NRVs, micro porous panels, and pipe work. The pilot's controls consist of a rotary timing switch, which times from 0 to 8 minutes, a chime that sounds when the switch reaches zero, a contents warning light that illuminates when the tank's capacity has 30 minutes of fluid left, and a low pressure warning light.

When ice has formed, the pilot selects the appropriate amount of fluid to be dispensed by using the timer. This activates the pump, and Isopropyl Alcohol is pumped to the panels where it oozes or weeps out of the micro pores onto the skin's surface. A head-compensating valve is fitted to the aircraft to ensure that fluid is supplied to the fin and, if used, T-tail, as well as the wings.

The pilot must be aware of the effect of the airflow across the aircraft's wings at different angles of attack, as this alters the distribution pattern of the fluid. The system suffers from the pores becoming blocked by impact with insects and fine airborne particles. To overcome this, the system has to function regularly to clean the pores.

THERMAL ANTI-ICING SYSTEM

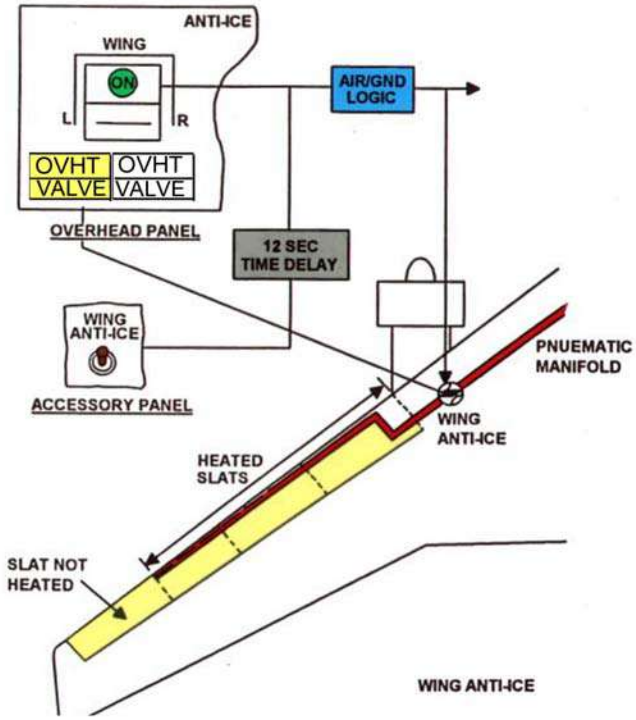


Diagram 11.23 Thermal Anti-Icing

Thermal anti-ice uses a very large quantity of hot air. Only aircraft with large turbine engines (or dedicated heaters) can use this system, as the air drawn off from the HP compressor results in less thrust as it reduces the engine pressure ratio (EPR) and raises the exhaust gas temperature (EGT), which has to be considered at high altitude and when flight planning.

Refer to diagram 11.23. These systems must have an overheat sensor in the supply and shut off valve as the temperature could damage the structure of the leading edge. In flight, the airflow across the wings and any moisture in the atmosphere dissipates the heat in the leading edge structures. When the aircraft is on the ground, this cooling airflow is not available. To prevent structural damage, the air/ground logic circuit closes the anti-icing valves.

It is standard for leading edge anti-icing to be selected “on” prior to take-off. To prevent the engines from losing thrust during the initial climb as the air/ground logic switches to air logic, a 12-second time delay operates. This keeps the valves closed until the aircraft has stabilised. In modern systems, this is also linked to the radio altimeter and cuts out if the aircraft reaches 400 ft radio altitude before the 12 seconds has elapsed.

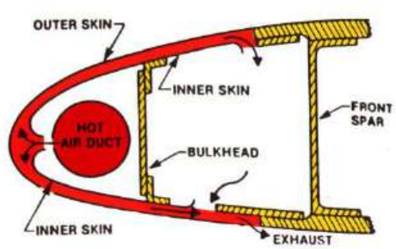


Diagram 11.24 Thermally Heated Leading Edge

Diagram 11.24 shows how the hot bleed air is ducted through tubes that have holes through them (termed piccolo tubes) into galleries formed by an inner skin and then ducted out under the wing.

JET ENGINE AIR INTAKES

Part of the certification process for jet engines for use in aviation is an ice ingestion test, as the effect of a block of ice hitting a high-speed rotor off centre is to put it out of balance. The most likely place for ice to form on an engine's nacelle is the intake lip where the airflow stagnates, allowing ice to form easily. Ice also forms on the tip of the engine's rotating bullet, inlet guide vane (IGV), and at the root end of the first/second row(s) of compressor blades. Apart from the hazard of lumps of ice forming on the lip, then breaking off and being ingested, ice formations disrupt the airflow into the engines. This can lead to the engines stalling in certain flight conditions such as high pitch angles. There are two systems for protecting air intakes. For large turbine aircraft, this is normally done using bleed air anti-icing. For smaller turboprop engines, this is normally achieved by a mixture of electrical anti-icing and de-icing.

ENGINE NACELLE BLEED AIR ANTI-ICING

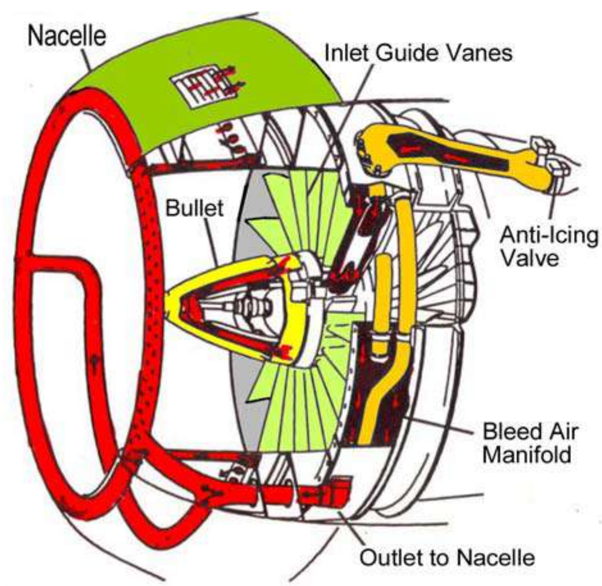


Diagram 11.25 Intake Bleed Air Anti-Icing

Diagram 11.25 shows how the intake lip of a turbine engine is anti-iced using bleed air. When engine anti-icing is selected, the anti-icing valves open, allowing bleed air to enter the bleed air manifold. Where IGVs are fitted, some air is passed down through them to prevent ice from forming on them. This air can be used to anti-ice the bullet. A piccolo tube behind the intake lip heats the lip structure preventing the formation of ice. If the air used for anti-icing is exhausted overboard, there is reduction in EPR. However, if the bleed air is exhausted back into the intake (close to the blade roots), it ensures that ice does not form at the blade roots but raises the temperature of the intake air, which reduces its density. Engine anti-icing can be selected and operated on the ground due to the large volume of airflow passing over the intake lips, even when the aircraft is stationary and the engines are at ground idle.

ELECTRICAL DE-ICING AND ANTI-ICING

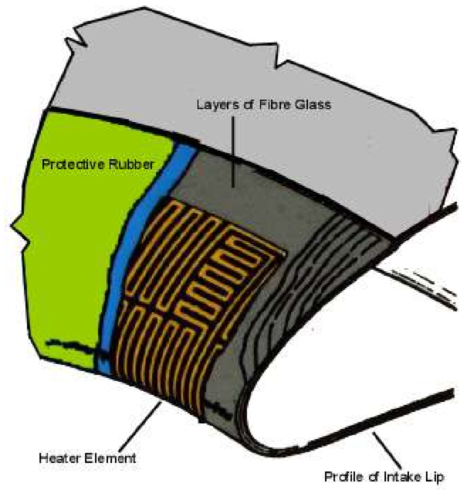


Diagram 11.26 Heater Mat

Diagram 11.26 shows the construction of a heater mat. A heating element manufactured from a thin sheet of etched brass is embedded into a layer of mastic over a profile of glass fibre and covered by layers of heat-resistant rubber. The whole mat is mounted and bonded onto the intake lip.

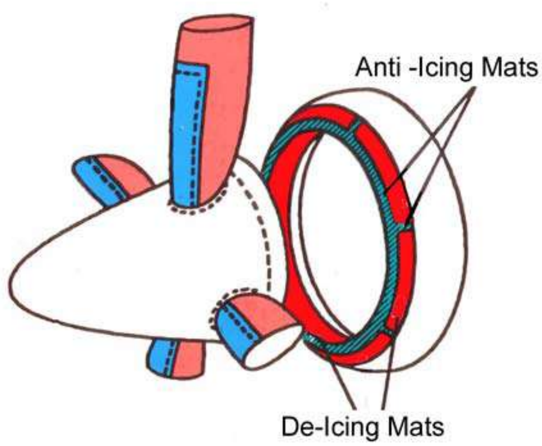


Diagram 11.27 Electrical Anti-Icing and De-Icing of an Intake Lip

Diagram 11.27 shows how anti-ice and de-ice heater mats protect the intake of a turboprop. The whole mat is divided into sections. Some sections are heated continuously, which form the anti-icing element. Other sections are heated cyclically. These form the de-icing element. Inside the intake, aft of the stagnation zone, is de-iced as the velocity of the air being sucked into the intake helps prevent ice forming. A section of the outer surface of the nacelle aft of the stagnation zone is protected by de-icing elements. To ensure that any ice that has formed on this area is fully removed, anti-icing elements bisect them. These are termed **breaker strips.**

WINDSHIELDS FLUID DE-ICING

Some light aircraft use a liquid de-icing fluid. This is sprayed onto the bottom of the windscreen and is distributed by the airflow. The pilot controls the amount of fluid applied by operating a momentary switch (similar to a car's windscreen wash), often referred to by the manufacturer's initials as TKS systems. The AFM details how much fluid the reservoir holds and how many applications this gives when applied for so many seconds.

WINDSHIELDS ELECTRICAL ANTI-ICING

As light aircraft have a limited generating capacity, electrical anti-icing is limited to a small panel in front of the captain only.

For large aircraft with their greater generation power, the forward screens are heated to 35°C by passing a current through a very thin electrical element that is part of the clear view panel. This is either gold or iridium trioxide. When power is applied to the screen, a green light illuminates. When the temperature is reached, a sensing element cuts the power. The sensing element cycles power to the screen to reduce the load. In extremely cold conditions, the pilot can select "high". This increases the power supply to the screen, speeding the heating.

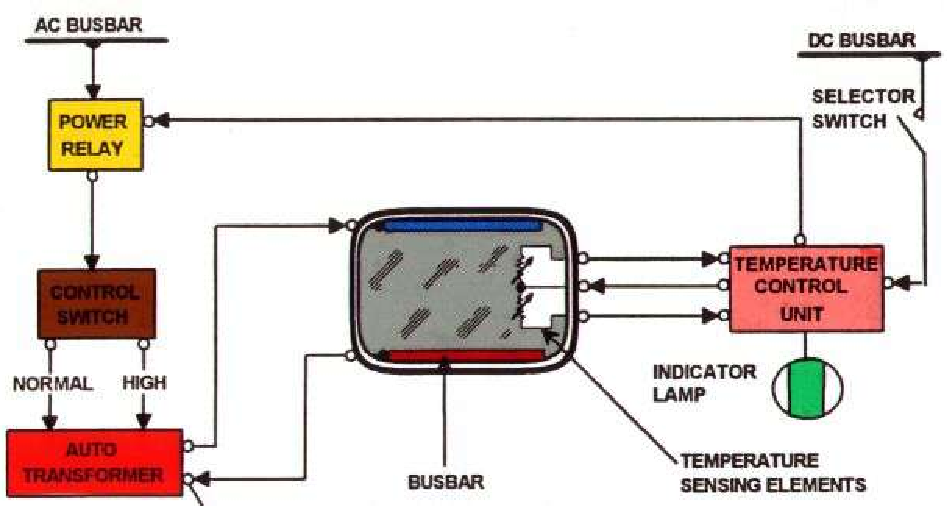


Diagram 11.28 Schematic for Electrical Anti-Icing of Flight Deck Windscreens

Diagram 11.28 shows how these units have an overheat cut out that automatically operates to remove power to the screen in the event of an over temperature. Once the screen has cooled, it resets. To reapply power, the crew must select "normal" first regardless of the ambient conditions. Otherwise, there is a risk of thermal shock. Heating the windscreen units not only anti-ices them but makes them more malleable and resistant to bird strikes.

PITOT PROBES

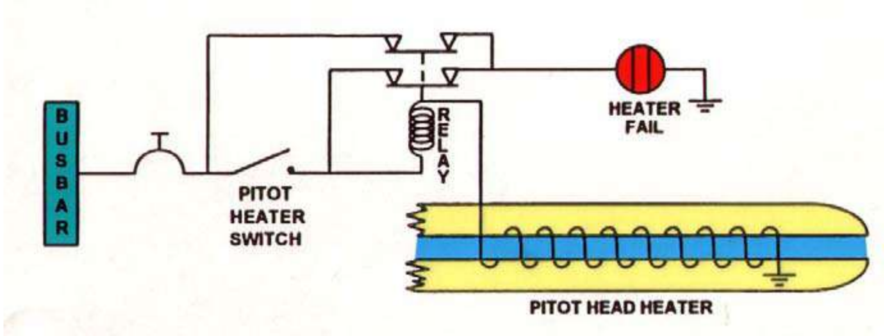


Diagram 11.29 Pitot Heater

All pitot probes and some static vents have heating elements. Some large aircraft have two elements for redundancy. As a precaution, air transport aircraft have a heater fail light (see diagram 11.29). The pitot is heated to prevent the ingress of moisture and icing, and is selected ON before take-off. Pilots must check that the heater element is operating before flight.

Caution: Do not touch the probe. It will burn. Always offer the back of the hand to the underside or side of the probe. If the probe is inadvertently touched, the natural reaction is to pull the hand away. Never cup the hand around the probe, as the natural reaction is to grasp. This leads to a severe burn of the palm and fingers.

PROPELLER DE-ICING

Propeller ice formation is a hazard. Unless removed, there is a loss of efficiency and a danger that the propellers will become unbalanced, creating vibration. This could cause structural damage to the engine mounts. Ice forms on back of the blades (the curved surface of the blade that a pilot of a single-engine aircraft cannot see when sitting at the controls) for the first 1/4-1/3 of the blades' length from the hub. After this, the rotational velocity of the blade stops the ice from being able to adhere. There are two systems for removing ice that has formed on the blades, a fluid system and an electrical system.

PROPELLER FLUID DE-ICING SYSTEM

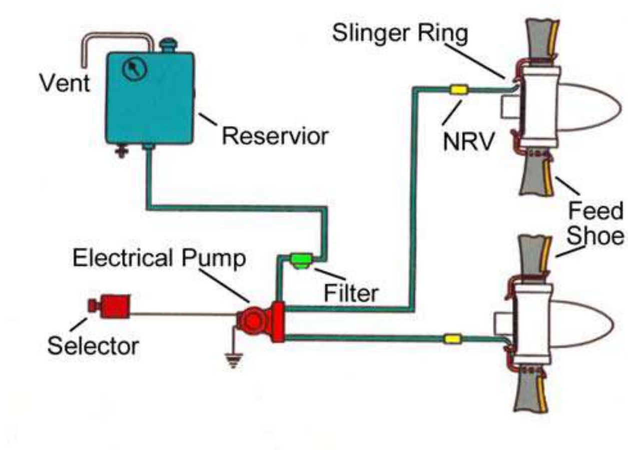


Diagram 11.30 Fluid Propeller De-Ice System

Diagram 11.30 shows a schematic for a propeller fluid de-icing system. The fluid is stored in a reservoir or tank and is filtered before being pumped. An electrical pump is controlled by a timer switch and pumps the fluid via NRVs to propellers. To overcome the difficulty of getting fluid from a fixed pipe to the rotating propeller, a simple system termed a **slinger ring** is used.

The slinger ring, a plate with a returned lip, is attached to the rear face of the propeller hub. The fluid is pumped onto this plate, and centrifugal force slings it outward and down the feed pipes to plastic feed shoes that are fitted to the leading edge of the blades. These shoes are tapered thicker at the hub and have micro grooves that guide the flow of fluid outward where it is then spread by the rotation and airflow. While these systems are termed de-icing, the fluid needs to be applied before the aircraft encounters icing conditions.

PROPELLER ELECTRICAL DE-ICING SYSTEM

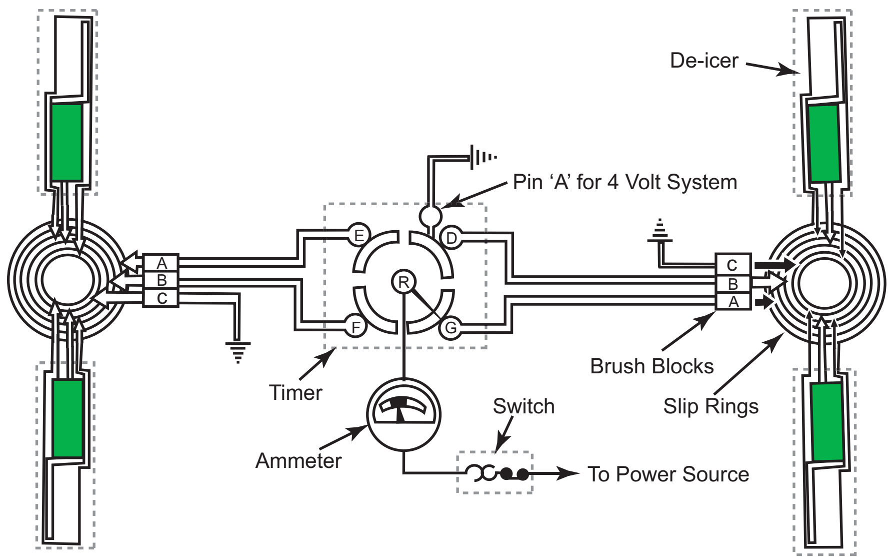


Diagram 11.31 Light Aircraft Electrical De-icing System

While propellers require de-icing for up to the first 1/3 of the blade, some designs fit a single heater mat to cover this area. Other designs use two separate heater mats to cover up to 50% of the blade's leading edge. This is done to increase the de-icing efficiency and reduce the electrical load, as each of these mats is smaller than the single heater mat.

Diagram 11.31 shows an electrical de-icing system for a light aircraft using the twin-heater mat system. The mats are constructed in a similar manner to the intake heater mats, and are bonded to the leading edge of the blades. The heater mats are electrically connected to concentric copper rings, termed **slip rings** (for a twin-mat system, there are three rings; supply to the inner mats, supply to the outer mats, and earth). Three fixed, graphite brushes are mounted on the engine. These are spring-loaded, so that they constantly touch the slip rings.

The brushes are connected to terminals in a **cyclic timer**. An ammeter measures the supply. When icing conditions are encountered, the pilot turns the prop de-icing on. Power is supplied to the cyclic timer, which applies power to the inner heater mat of one propeller set, then to the outer mats of the same set before switching over to the inner set of the second propeller set, followed by the outer set.

This switching reduces the electrical load on the aircraft's alternators and by heating the inner set first, the ice's adhesion is broken more quickly and the balance of the propeller is maintained. Where an aircraft possesses an uneven number of blades, all the blades must be de-iced simultaneously to maintain balance.

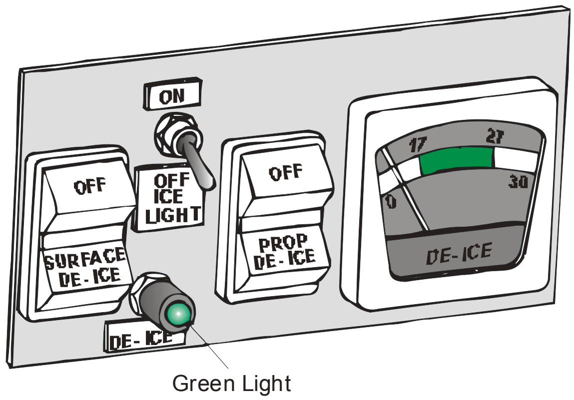


Diagram 11.32 Propeller De-Icing controls and Indications

The ammeter indicates to the pilot that the propeller de-icing system is operating correctly by dropping to zero each time the cyclic timer moves from one mat to another (referred to as a **null**), and by showing the load placed on the electrical system (see diagram 11.32).

TURBOPROP ELECTRICAL ANTI-ICING AND DE-ICING OPERATION AND INDICATIONS

As seen above, these aircraft use both electrical anti-icing and de-icing systems. To indicate to the crew the status and operation of these systems, a green light illuminates continuously when the anti-icing system is functioning correctly. This same light flashes when a de-icing system is functioning. As turboprops are likely to encounter both light and heavy icing, their propeller de-icing systems have two settings. These are:

- Slow cycle for heavy icing
- Fast cycle for light icing

Refer to diagram 11.33, which shows the timings for the fast and slow cycles and the operation of the green lamp.

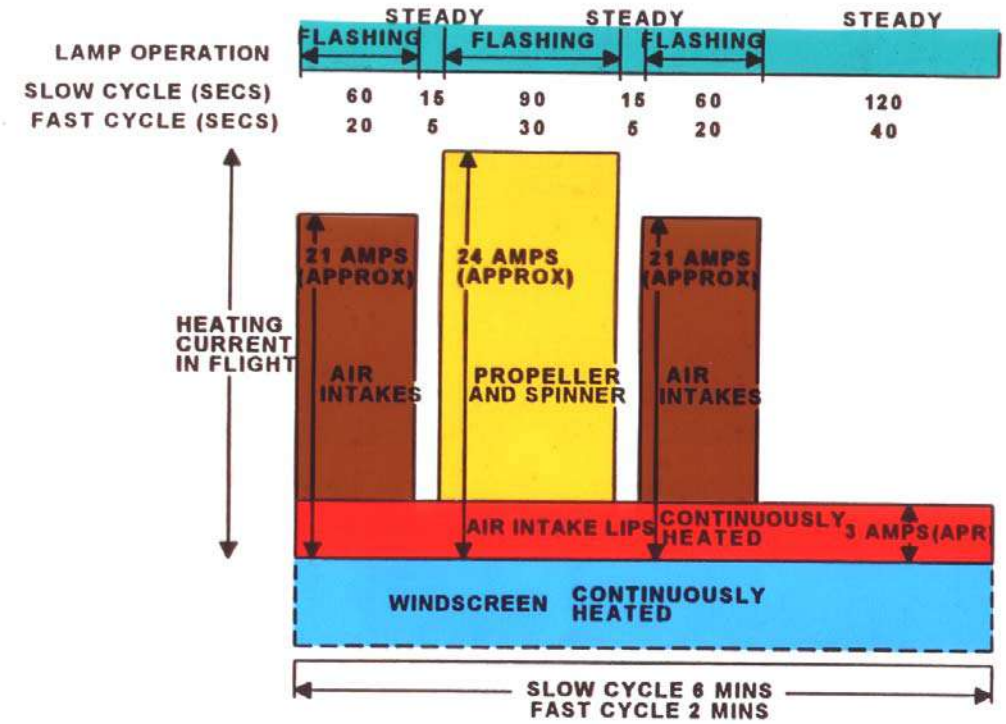


Diagram 11.33 Electrical Anti and De-Icing Timings and Indications

Chapter 12

Environmental Control Systems: Air Conditioning

ACKNOWLEDGEMENT

We would like to thank and acknowledge:

For Diagrams and assistance with technical information:

12.5 Janitrol

INTRODUCTION

The Earth’s atmosphere is approximately 348 miles (560 kilometres) thick and is comprised of nitrogen (N_2) 78%, oxygen (O_2) 21%, and a mixture of other gases (1%).

The atmosphere can be divided into four distinct layers. These are the troposphere, the stratosphere, the mesosphere, and the thermosphere or ionosphere.

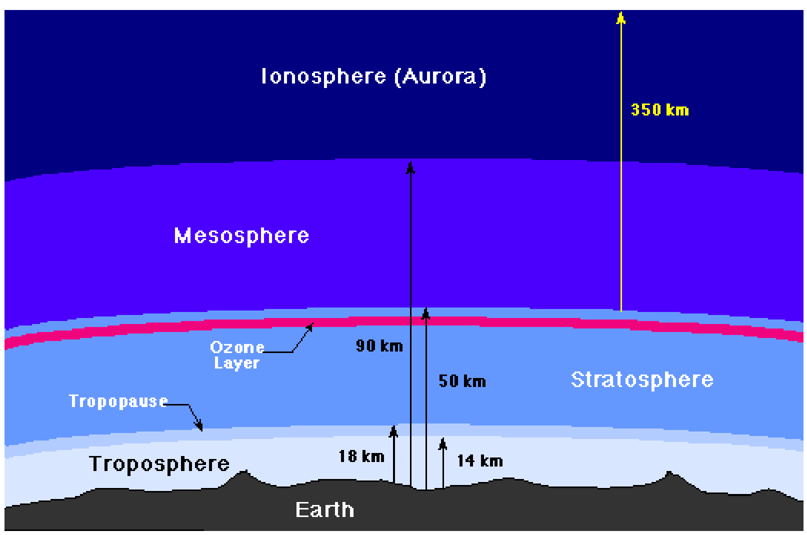


Diagram 12.1 Layers of the Atmosphere

Commercial flight takes place in the troposphere and lower stratosphere. The troposphere begins at the Earth's surface and extends 5 to 9 miles high. The stratosphere begins just above the troposphere and extends to 31 miles high. The tropopause is the boundary between the two layers.

Refer to the ISA table below. Diagram 12.2 shows the effect of increasing altitude on temperature.

Altitude		**Temperature**		
Feet	**Metres**	**Deg K**	**Deg C**	**Deg F**
-1000	-304.8	290.13	16.98	62.6
0	**0**	**288.15**	**15.0**	**59.0**
+1000	+304.8	286.17	13.02	55.4
2000	609.6	284.19	11.04	51.9
3000	914.4	282.21	9.06	48.3
4000	1219.2	280.23	7.08	44.7
5000	1524.0	278.24	5.09	41.2
6000	1828.08	276.26	3.11	37.6
7000	2133.6	274.28	1.13	34.0
8000	**2438.4**	**272.30**	**-0.85**	**30.5**
9000	2743.2	270.32	-2.83	26.9
10 000	3048.0	268.34	-4.81	23.3
15 000	4572.0	258.43	-14.72	5.5
20 000	6096.0	248.53	-24.62	-12.3
25 000	7620.0	238.62	-34.53	-30.2
30 000	9144.0	228.71	-44.44	-48.0
35 000	10 668.0	218.81	-54.34	-65.8
36 000	10 972.8	216.83	-56.32	-69.4
36 090	**11 000.2**	**216.65**	**-56.5**	**-69.7**
40 000	12 192.0	Temperature remains constant at **–56.5°** Celsius from the tropopause to an altitude of 65 000 ft, where it starts to rise.		
45 000	13 716.0			
50 000	15 240.0			
55 000	16 764.0			
60 000	18 288.0			
65 000	19 812.0			

Diagram 12.2 ICAO International Standard Atmosphere

Refer to highlighted and emboldened lines in diagram 12.2. The temperature and pressure given in the line for zero feet (msl) are the conditions covered by the term ISA day, a temperature of 15°C, a pressure of 14.69 psi equating to 1013.2 millibars, said as “ten thirteen point two”. The imperial value of 14.69 psi is often rounded up to 14.7 psi, or on occasion to 15 psi.

Climbing through the air column to 8000 ft, the ambient temperature falls to -0.85°C, and the pressure decreases to 10.92 psi (753 millibars). The temperature decreases until at 36 090 ft, and the ambient temperature has dropped to –56.5°C. This is the tropopause, the lower edge of the stratosphere. The temperature remains constant in the lower stratosphere to an altitude of about 65 000 ft, where it starts to rise.

From the above table, it is evident that heat must be brought into the aircraft's cabin for both comfort and survival. Aviation medicine has determined that the average person (post World War II test of fit young service men, which is currently under revision) requires:

- A pound of air per minute, 1 ppm for ventilation
- A temperature between 18 and 24°C, the comfort zone
- An average humidity of between 30 and 60% at 18°C

This chapter examines how aircraft certified under JAR 23 and 25 achieve temperature control, ventilation, and humidity control.

TEMPERATURE CONTROL

RAM AIR MUFFLER TYPE HEAT EXCHANGER

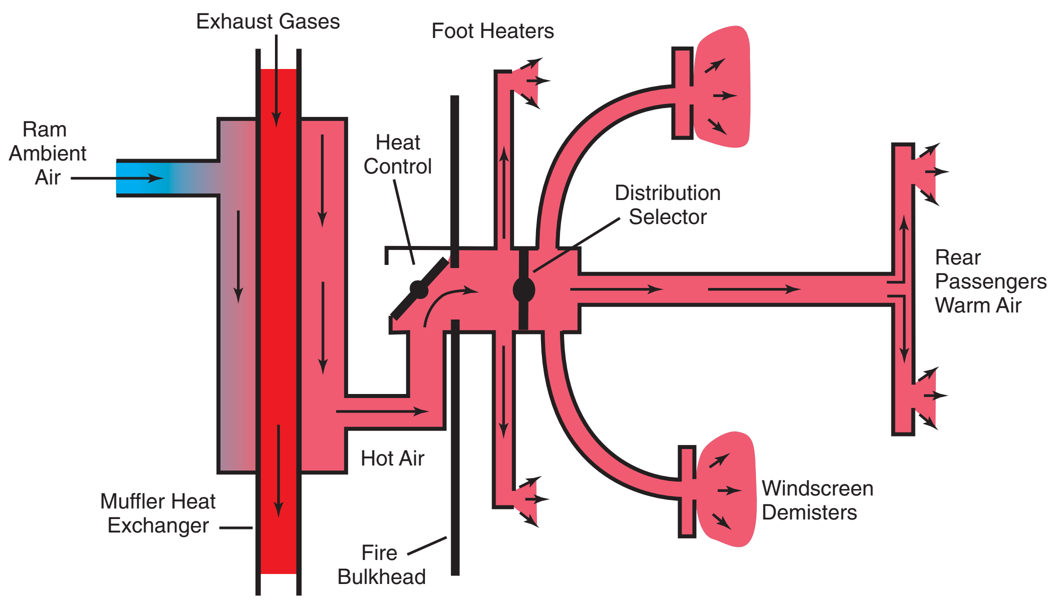


Diagram 12.3 Light Aircraft RamAir Muffler Type Heat Exchanger

Diagram 12.3 illustrates the type of system found on many typical light training aircraft such as the PA-28 Piper Warrior, etc. In this system, ram air ducted from a scoop located on the chin of the lower engine cowling is directed into a muffler that encloses the engine's exhaust system. Here the engine's exhaust gases heat the ram ambient air. Note that the gas flow and the airflow are separated by the exhaust pipe. If the exhaust pipe were to develop a crack, there is the very real risk of the pilot suffering from carbon monoxide poisoning.

The air is ducted to a control box located on the forward face of the front fire bulkhead. As there is always a flow through the muffler, the control box can direct all the heated air overboard, into the cabin, or proportionately into the cabin and overboard depending on the heat selection the pilot makes. As the heat control box effectively opens or closes a hole through the fire bulkhead, in the event of an engine fire or smoke in the cabin, the pilot must close the hole by selecting "heat off".

From the heat control box, the warm/hot air can be directed to vents in front of the windscreen to aid defogging, to the pilot's foot warmers, and to the rear passengers foot warmers. Heat is applied at a lower level as this warms the extremities and aids comfort. In general, passengers prefer cool air directed at their faces to keep them alert. The air is not dried and the ambient humidity is that which enters the aircraft. Drain holes in the duct before the muffler allow condensed moisture to drain away. The forward velocity of the aircraft and the temperature selection determine the volume of air and the velocity at which it enters the cabin. The amount of heat available to warm the ram air depends on the amount of work the engine is doing. In the climb, the engine works harder, so the exhaust gas temperature and quantity of gas increase. In the cruise, the engine's temperature reduces and the rpm decreases. To gain the same level of heat in the cabin, the temperature control must direct more air into the cabin. In descent with the engine power reduced, the temperature of the exhaust gases and their quantity greatly decreases, requiring a higher selection to maintain cabin temperature.

COMBUSTION HEATER

More sophisticated light aircraft can use a dedicated combustion heater to heat ram air. These systems can normally operate up to altitudes of 16 000 ft, and can be left operating on the ground to warm the cabin prior to flight. The disadvantages of these systems are that they draw fuel from one of the fuel tanks at a rate of 0.5 US gal/hour (which can eat into the aircraft's trip fuel), they could be a source of carbon monoxide poisoning, and are a potential fire risk.

Diagram 12.4 shows a schematic for a typical combustion heater system. The unit located within the fuselage has to be supplied with two sources of air. The air to be heated, ram air, is taken from an inlet at the base of the fin. The air for use in the combustion chamber, combustion air, is taken from an inlet on the side of the aircraft, so that it is at static pressure.

An electrically powered combustion blower draws ambient air in through the static inlet, which supplies the combustion chamber with the correct weight of air for complete combustion of the fuel (aviation gasoline). A pressure switch in the duct between the combustion blower and the combustion chamber senses the pressure in the duct and ensures that the correct pressure is maintained.

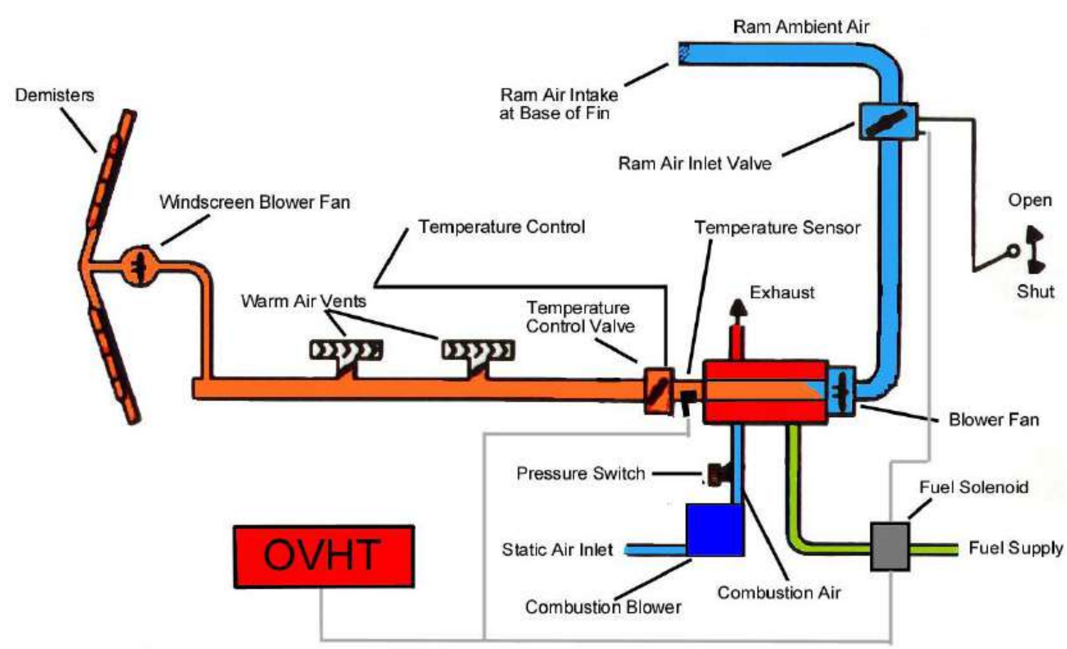


Diagram 12.4 Combustion Heater Schematic

In flight, the air to be heated is supplied as ram air through the forward motion of the aircraft. For operation on the ground, an electrically powered blower fan is fitted between the heater and the intake. This induces a flow through the duct. The power supply for the blower fan is routed through a squat switch on the undercarriage, so that after take–off, the blower fan is automatically switched off.

A valve is fitted into the ram air duct, which allows the pilot to control the rate of flow into the combustion heater. A micro switch mounted by the valve and linked to a fuel shut-off valve prevents the heater from operating until the valve is opened to the minimum airflow position. This prevents damage to the heater caused by overheating.

Diagram 12.5 shows the design of a combustion chamber and the airflow. The fuel drawn from one of the aircraft's fuel tanks at the cross feed pipe by a dedicated electrically powered pump is sprayed by an injector nozzle into the stream of air from the combustion blower and is ignited by a sparkplug. This creates a swirling flame, which ensures that all the fuel is combusted.

The central chamber is interlinked with two concentric chambers, so that the hot exhaust gases have to pass back and forth before venting to atmosphere via an exhaust pipe or mast that protrudes below the aeroplane. This design ensures that the ram air, which passes between the heated chambers, extracts the maximum heat possible. The **solenoid valve** acts as a fuel shut-off.

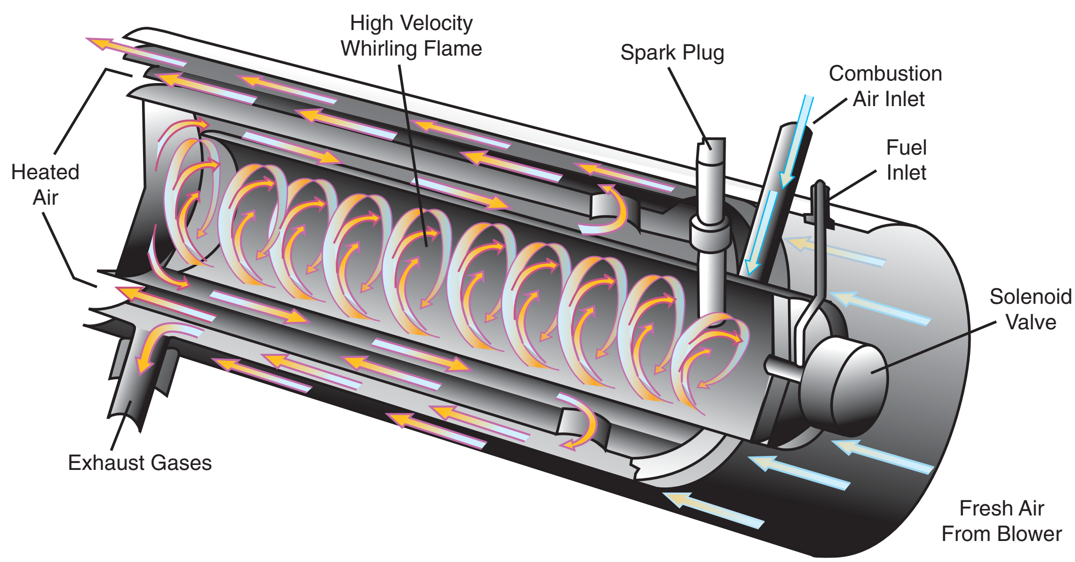


Diagram 12.5 Janitrol Combustion Heater

A valve plate in the output duct downstream of the heater unit controls temperature. To get hotter air, the pilot closes the valve, slowing the through flow of air from the heater unit, which allows more time for heat transference. A temperature **thermistor** is located in the outlet duct. This serves two functions, first as overheat protection, second as a temperature/economiser control.

In the event the outlet air temperature exceeds preset limits, the thermistor closes the solenoid valve, de-energising the fuel pump and the ignition system, which shuts the heater down. At the same time, the master caution and a heater overheat indicator on the central warning panel illuminate. If this situation occurs, the heater can only be reset on the ground.

To make the heater economical, the thermistor registers the temperature as set by the pilot's positioning of the valves and cycles the heater to maintain the temperature of the outlet air within a tolerance. When the upper temperature is reached, the heater unit closes down until the temperature falls and then cuts back in again automatically.

If the heater is shut down in flight, the pilot keeps the ram air valve open for a minute to allow the unit to cool. When the heater is to be shut down on the ground, the pilot must keep the blower fan running with the ram air valve open for approximately 3 minutes to ensure that the unit cools after the heater is turned off. Failure to do this can lead to the unit cracking with attendant dangers of carbon monoxide poisoning and fire.

RAM AIR HEATING FROM A TURBOCHARGER SYSTEM

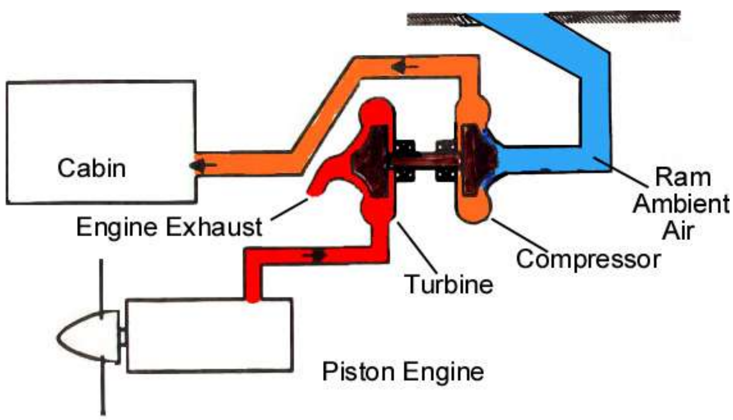


Diagram 12.6 Turbocharger Heating

Some aircraft use the engines exhaust gases to drive a compressor via a turbine. Diagram 12.6 shows a simple schematic of such a system. This system cannot be used to pre-heat the cabin unless the engine is running, and the temperature output depends on the engine's rpm. All the piston-engine ram air systems suffer a decrease in heating efficiency as the aircraft climbs due to the drop in density.

PRESSURISED AIRCRAFT

Pressurised cabins and compartments to be occupied must be equipped to provide a cabin pressure altitude of not more than 8000 ft at the maximum operating altitude of the aeroplane under normal operating conditions. If certification for operation over 25 000 ft is requested, the aeroplane must be able to maintain a cabin pressure altitude of not more than 15 000 ft in the event of any reasonably probable failure or malfunction in the pressurisation system.

For larger and faster pressurised aircraft, it is standard to fit two air conditioning units (referred to as **air conditioning packs,** abbreviated to **ACS packs** or just **packs**) to serve the system. This allows for redundancy, as one pack is able to maintain the minimum conditions required by the regulation.

The air conditioning system is the source of air that maintains normal cabin pressure (8000 ft). A single pack's through flow must enable the aircraft to maintain the limiting cabin altitude in the event of a pressurisation system failure. This is 15 000 ft for aircraft certified to operate above 25 000 ft.

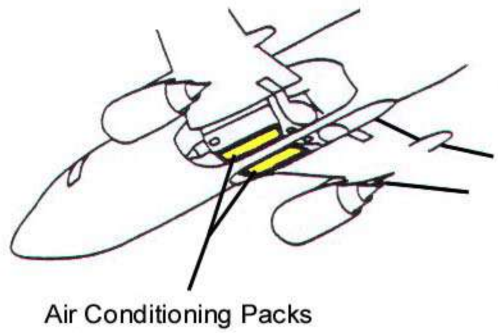


Diagram 12.7 Location of Air Conditioning Packs

In modern medium and large air transport aircraft, it is standard to locate the air conditioning packs beneath the centre fuel tank (see diagram 12.7). This has two advantages. First, the fuel tank is used as a heat sink for the air conditioning packs. Second, the fuel in the tank is heated by the excess heat, which helps to condition it.

BASIC PRINCIPLES OF AIRCRAFT AIR CONDITIONING

As aircraft are going to be operated at different flight levels in different temperature zones around the world, the aircraft's air conditioning system must be capable of taking extremely cold air and warming it, or extremely hot humid air and cooling and dehumidifying it. As the ambient temperature is a variable, these systems use heated ambient air. The heating of the ambient air is either by dedicated means, or hot bleed air is taken from a gas turbine's compressor. This heated air, often referred to as **charge air**, is then split and a proportion cooled, before it is mixed together to achieve the required temperature.

Cooling the charge air is a major function of the pack. There are two different methods in which this can be achieved, the use of air as a cooling medium referred to as **air cycle,** or the use of a refrigerant referred to as **vapour cycle**.

AIR CYCLE MACHINES

The component that cools the charge air is termed a **Cold Air Unit** or **CAU**. There are three different designs. Each comes under the heading of **air cycle machines**.

The types are:

- The bootstrap
- The brake turbine
- The turbo fan

BOOTSTRAP IN CONJUNCTION WITH A MECHANICAL BLOWER

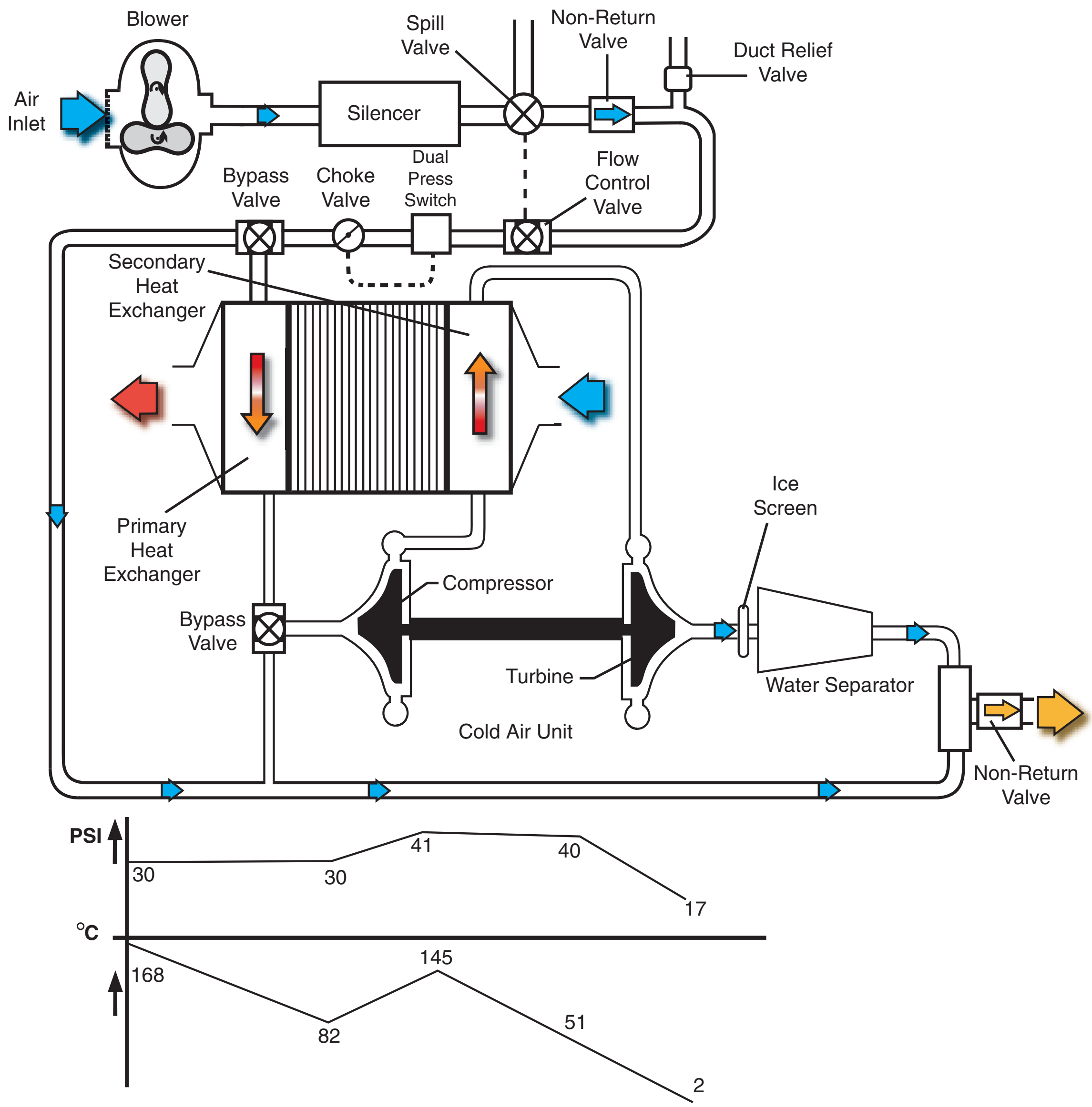


Diagram 12.8 Bootstrap and Blower

The schematic diagram 12.8 shows a bootstrap cold air unit used in conjunction with a dedicated blower. This type of system is used on larger piston-engine aircraft and smaller turboprop aircraft, where the latter's engines are not designed to supply bleed air for the aircraft's air condition system. The function and operation of this system is given next.

BLOWER

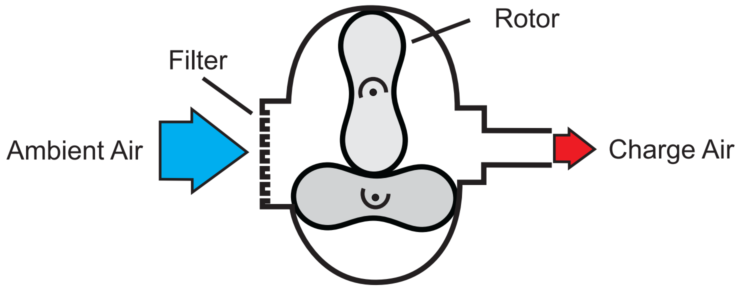


Diagram 12.9 Roots Blower

Ambient air is drawn into a blower. The type illustrated is a **Roots Blower** (the designer/manufacturer). Refer to diagram 12.9. The blower consists of two wasted lobes which are engine driven, and geared to rotate and mesh. This draws a large volume of air in and forces it into the supply duct. The restriction of the duct raises both air pressure and temperature. Refer to the following graph, diagram 12.8. Since the blower is mechanical, it requires lubrication. Failure of the oil seals can result in blue smoke in the aircraft as the oil vaporises in the hot air. The rotation of the lobes creates a pulsing, changing air pressure (whomp-whomp effect), which is removed by the silencer unit located downstream.

SPILL VALVE AND FLOW CONTROL VALVE

The spill valve is designed to allow charge air to bleed overboard in different conditions and is linked to the flow control valve. For older aircraft during the take-off run when the propellers require all the available power, the spill valve opens to allow all the charge air to vent to atmosphere. This off loads the blower and reduces the load on the engine.

During flight, the flow control valve, also referred to as a **mass flow controller**, determines the correct mass of air passing through the system to ventilate the aircraft. This is done by venting charge air to atmosphere. As the aircraft climbs and the ambient density decreases, the flow control valve progressively closes the spill valve. Engine rpm, thus blower rpm, also alters the spill valve's position.

In the event of an engine fire, to prevent contamination of the cabin air, the spill valve is fully opened when the pilot operates the engine's fire handle. In the event that the cabin air becomes contaminated with smoke from the air conditioning system, the pilot can open the spill valve manually to isolate the system. A non-return valve (NRV) is fitted downstream of the spill valve to prevent loss of cabin air pressure in the event that the spill valve is opened or failure of the blower.

DUCT RELIEF VALVE

A duct relief valve is located downstream of the NRV. The function of this valve is to protect the duct from over pressurisation. There is a real danger that if the duct ruptures, high temperature air could play onto fuel lines or electrical cables and start a fire. The relief valve is set to operate at 10 psi above the ducts normal pressure, refer to the graph on diagram 12.8. The standard used in examination questions is the valve's value (10 psi).

CHOKE VALVE AND DUAL PRESSURE SWITCH

The choke valve is fitted as a means of increasing the charge air's temperature in certain conditions, by restricting the through flow and creating a backpressure. The choke valve only restricts the airflow when the bypass valve downstream is fully open. The dual pressure switch is located upstream of the choke valve and monitors the duct pressure. If the pressure rise increases to 8.5 psi above the normal value, the dual pressure switch signals the choke valve, preventing the choke valve closing any further. In the event that the pressure continues to rise, at 9.5 psi the dual pressure switch signals the choke valve and drives the valve fully open.

BYPASS VALVES

There are two bypass valves fitted in this system, one downstream of the choke valve and a second downstream of the primary heat exchanger. They function as temperature control valves. The first bypass valve can direct all the air through the heat exchanger or allow a percentage of air to bypass the heat exchanger. The second valve controls the amount of air that enters the CAU. Both valves are controlled by temperature sensors, either mounted in the aircraft's cabin or in the duct leading into the aircraft's cabin.

PRIMARY HEAT EXCHANGER

The system has two heat exchangers, which act as radiators. Charge air from the first bypass valve is ducted into the primary heat exchanger, also known as a **pre-cooler**. In the heat exchanger, the hot charge air is passed through a matrix of small-bore pipes, while ram ambient air passes around them. As the heat exchanger is open ended, this results in adiabatic cooling, where the temperature decreases, but there is no significant change in the pressure.

BOOTSTRAP

The bootstrap consists of three components in the following order: compressor, heat exchanger, and turbine. The compressor and turbine are linked together and form one CAU. The system is referred to as a bootstrap as it is able to self-start. As soon as there is air flowing across the turbine, it starts to revolve itself and the compressor.

As the air from the compressor drives the turbine, the turbine tries to accelerate due to an increase in airflow from the compressor. However, the compressor's rotation applies a load back on the turbine, thus they self regulate. Due to the compression and work done by the cold air unit and the speed of rotation, these units must be lubricated. Failure of the oil seals can result in blue smoke entering the cabin.

As diagram 12.8 shows, air that has been cooled by the pre-cooler is directed by the second bypass valve into to the eye of the CAU's centrifugal compressor. Here, it is compressed, raising both pressure and temperature. The output from the compressor is then passed through the secondary heat exchanger, also referred to as an intercooler, before being ducted on to the edge of a turbine.

In its passage across the turbine, the air is made to work by rotating the turbine and compressor. This work absorbs pressure energy and, at the same time, the air is able to expand. The combined effect reduces the temperature of the air, resulting in a stream of cold air leaving the turbine. To get more cold air, more hot air is ducted into the eye of the compressor and vice versa. The speed of the CAU is determined by the temperature requirements of the system and the air's density.

WATER EXTRACTOR

At lower altitudes, the moisture content (humidity) is higher than that found at high altitudes. Excess humidity inside the aircraft would manifest itself as condensation or even water droplets falling from the air conditioning low-pressure ducts. This would lead to discomfort for the passengers and crew, as well as the possible shorting of electrical circuits, corrosion, and an increase in mass over time as the insulation blankets become sodden.

To remove this excess moisture, water extractors, also known as **water separators**, are fitted downstream of the cold air unit. In some systems, more than one water extractor is used as warm air can support more moisture than cold air.

Refer to diagram 12.8. The extractor is fitted in the cold air supply downstream of the CAU. Between them is an ice screen. This is fitted to prevent ice pellets from damaging the water extractor. Ice pellets form because the air leaving the turbine contains moisture that is below the dewpoint.

There are different designs of water extractors. However, they all work on the same basic principle of diffusion, coalition, and extraction. As the air enters the water extractor, it passes through a diffuser section that slows the airflow and guides it over a **coalescer** section (see note). Here, the moisture is coalesced (merged) into larger droplets.

The droplets are carried by the airflow to the extractor section. Here the water impinges on a series of loose rods, forming large droplets which run down to a collector. The rods are loose to enable them to vibrate and shake the water downward. Dried air leaves the extractor to be mixed with the other streams of air.

In turboprop systems and older gas turbine aircraft systems, the water that has been collected is drained overboard or collected and drained by ground staff. In modern air transport aircraft, this moisture can be collected and reintroduced into the air stream when the aircraft is flying at high altitude where the humidity is low.

Water extractors incorporate an internal bypass valve. In the unlikely event of the water extractor becoming blocked by ice, the bypass valve opens and allows un-dried air to pass into the air conditioning system.

Note: In some designs, this takes the form of a fabric sock over a frame. One side of the fabric has the appearance of the loops used in Velcro®. This is always placed outward. Other systems use wire gauze.

BLEED AIR BOOTSTRAP

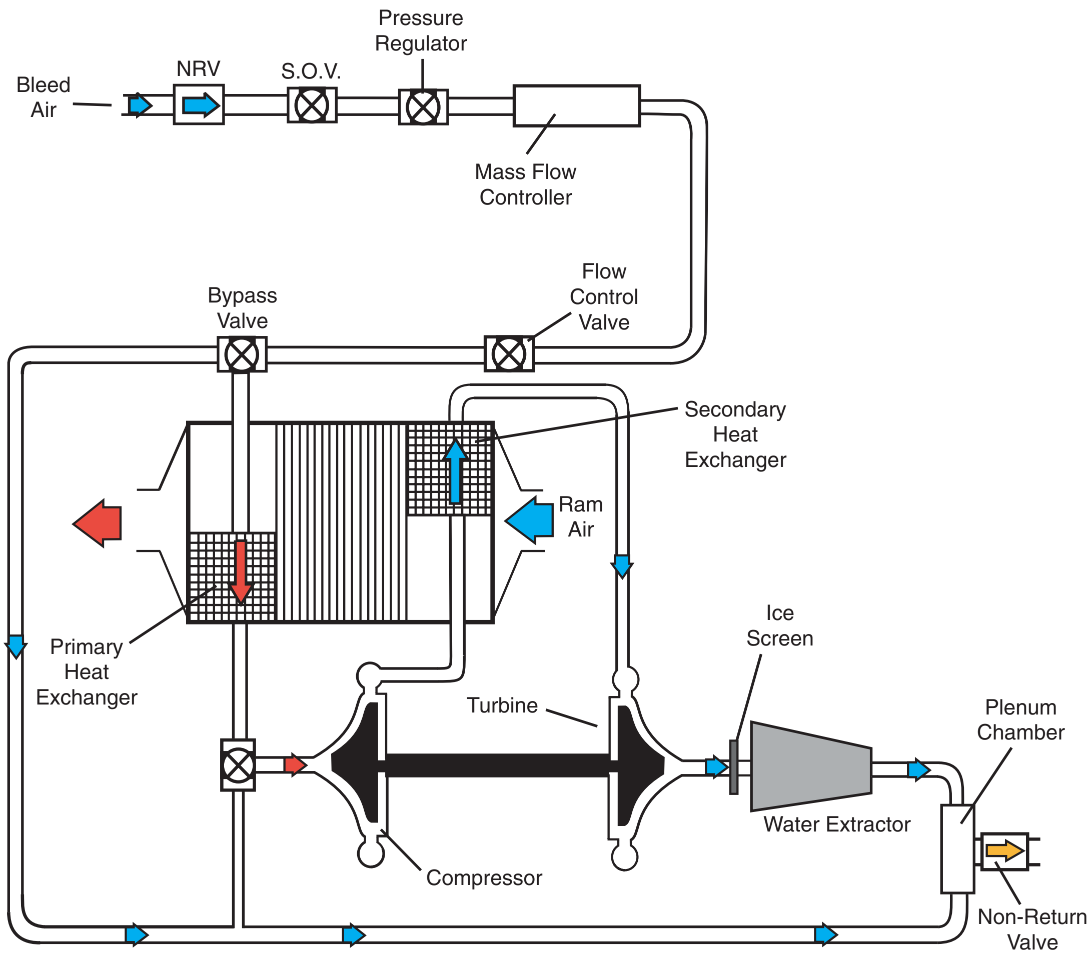


Diagram 12.10 Bleed Air Bootstrap System

Diagram 12.10 is a schematic for a bootstrap used in conjunction with bleed air from a gas-turbine engine. Turbine powered aircraft, where the compressor section can supply more air than the core engine requires, are able to supply bleed air to the packs.

This air has been heated due to compression. Therefore, in this system, there is no requirement for a blower and silencer. In the event of any problem, the bleed air to the air conditioning pack can be cut by closing the bleed air shut off valve (SOV). This isolates the engine, and as the air-entering the engine passes through it, there is no need for a spill valve.

A pressure relief valve protects the ducts from over pressure due to increased engine rpm. A mass flow controller linked to a flow control valve ensures that the correct mass of air is supplied to the system as the aircraft changes altitudes and engine rpm settings.

The bootstrap functions in the same manner as before, heating the air before passing onto a turbine to ensure that there is an adequate temperature loss across the turbine. In these systems, if hotter air is required in the system, the bleed air can be taken from latter stages of compression in the engine. As before, the streams of air are mixed in a plenum chamber before passing into the aircraft cabin.

BRAKE TURBINE

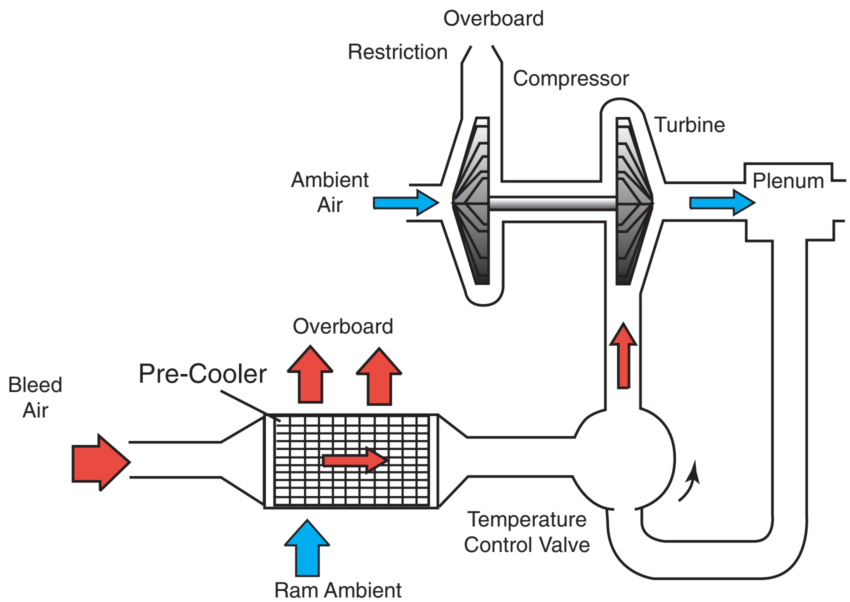


Diagram 12.11 Brake Turbine Air Cycle Machine

Diagram 12.11 is a brake turbine. In this system, the bleed air is passed through a pre-cooler to obtain adiabatic cooling and then to a temperature control valve, or TCV. This directs the pre-cooled air to the turbine or plenum chamber. To ensure that the air passing across the turbine loses pressure and temperature, a compressor draws in ambient air at static pressure by taking its supply from within a vented bay. This air passes across the compressor and is dumped overboard via a restricted pipe.

The restriction creates a backpressure that acts to slow the compressor and place a load on the turbine. The speed of these machines is self-regulating and is determined by the mass of air that passes across the turbine and the air's density. They can be turning at 40 000 to 45 000 rpm at high altitude.

TURBO FAN

Diagram 12.12 shows a typical turbo fan system, using bleed air from a gas turbine's compressor as the source of charge air. The air is rooted to a TCV, another term for a by-pass. The air to be cooled is passed through a heat exchanger before being split by a second TCV.

Air to form the cold stream is passed across a turbine before mixing with the warm and hot streams. If the mixing takes place before the water extractor, the ice screen can be dispensed with, as the temperature is greater than the freezing point of the moisture entrained within it.

In this design, the compressor of the cold air unit is replaced with a fan (a drum with vanes). This is placed in the stream of ram air that exits from the heat exchanger. In flight, the fan turning in the exhausted ram air imposes a load on the turbine, ensuring that the air passing across the turbine has to work and lose temperature. See the graph in diagram 12.12.

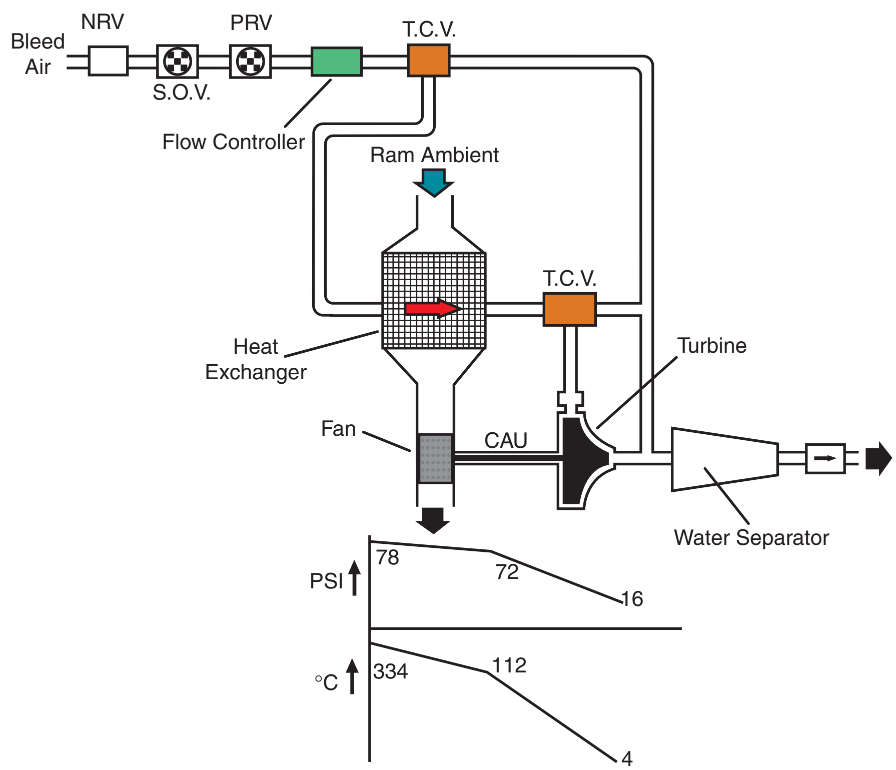


Diagram 12.12 Typical Turbo Fan Systems

On the ground, the fan acts to induce airflow through the heat exchanger allowing the system to function correctly.

INDUCERS

If a pack is operated on the ground without a sufficient airflow passing through the heat exchangers, the temperature of the charge air is sufficient to damage the heat exchangers and other components. Apart from the turbo fan, the other air cycle machines require a means to draw air through their heat exchangers. These can take the form of electrically operated fans or the use of high-pressure bleed air injected into the throat of a venturi in the duct from the heat exchanger to overboard. Both systems draw ambient air through the heat exchanger, allowing correct operation.

VAPOUR CYCLE

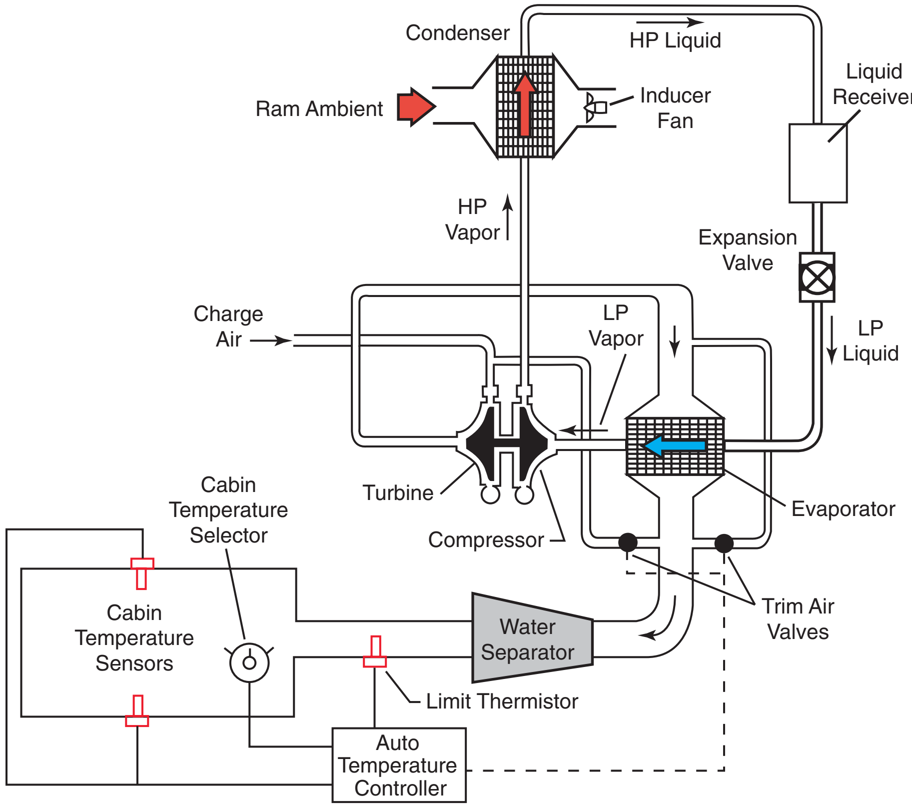


Diagram 12.13 Vapour Cycle Cooling Pack

Diagram 12.13 shows the standard schematic for a vapour cycle cooling pack or VCCP system. The system is divided into two sub systems, one containing the refrigerant Freon, the other is the air system. In the liquid system, the Freon, which in its natural state is a gas having a low boiling and evaporation point, is stored as a liquid under its own gas pressure in a reservoir termed a **liquid receiver**.

When the system is operating, low-pressure liquid Freon is drawn from the reservoir via an expansion valve and passes through the evaporator. The evaporator is a heat exchanger. The Freon draws heat from the air and changes state from a low-pressure liquid to a low-pressure vapour.

The low-pressure vapour then passes through a compressor, which is driven by a turbine in the charge air supply. The vapour is further heated and pressurised into a high-pressure vapour and passes into a condenser, a second heat exchanger which is cooled by ram ambient air. The Freon undergoes a further change in state through loss of temperature, reverts to a liquid, but remains under high pressure where it flows back into the liquid receiver.

The motive power for the system is provided by airflow across the turbine. The cooled air is then allowed to expand and pass across the evaporator unit, emerging as a stream of cold air. To obtain the required cabin temperature, hot and warm air can be mixed into the cold air stream via trim air valves. The process of correcting the temperature is termed **trimming**.

If the charge air is very hot (or the cold stream needs to be increased), the Freon entering the evaporator changes state more quickly. This alters the pressure in the low-pressure liquid pipe. The expansion valve set to maintain this pressure opens, allowing more Freon to enter the evaporator. From this, it can be concluded that the expansion valve is part of the temperature control system.

HUMIDITY AND HUMIDIFIERS

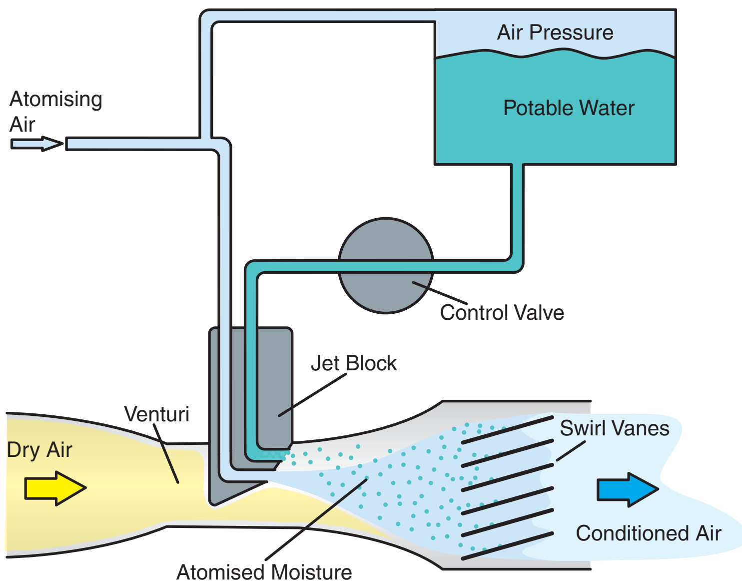


Diagram 12.14 Humidifier

Whereas aircraft flying at the lower levels have a need to remove excess humidity from the air to condition it, aircraft flying at high altitudes draw cold dry air into their air conditioning systems. This is not normally a problem, as the moisture exhaled by the passengers and crew maintains the cabin's relative humidity.

However, for flights of six hours duration or more, there is a need to supplement the cabin's humidity. This is done by humidifiers. See diagram 12.14 for a schematic of a humidification system. Engine bleed air, or air from another source (termed **atomising air**), is supplied to provide a head of pressure in a tank and to a jet block, which is located in a venturi downstream of the water separator.

The tank contains potable water, the term used for water that has been chemically treated to ensure that there are no water-borne "bugs" etc. When the humidity in the cabin drops to a set value, the control valve is automatically opened, allowing water to be expelled from the jet block. The block and the second stream of air are designed to break the water up, atomising it. The venturi is used to decrease the static pressure of the dry air, assisting in the air absorbing the very fine water mist.

The swirl vanes, which are covered in fabric, are designed to impart a swirl into the airflow to ensure that all of the air moisture mixes and to absorb any excess moisture. This is absorbed into the airflow when the humidifier switches off, prolonging the humidifying effect.

CONDITIONED AIR

Conditioned air means that the air has had its temperature and humidity corrected. While the cabin temperature range is between 18 and 24°C, the temperature in the supply ducts is higher as the final cooling of the air takes place in the cabin as the air expands. The temperature in the ducts is limited between 18 and 27°C and is termed the **control range**.

DUCTING

There are three different types of material used in the manufacture of air conditioning ducts:

- Stainless steel for high temperature high pressure air
- Light alloy for medium temperature medium pressure air
- GRP/plastic for low temperature pressure air

AIR DISTRIBUTION

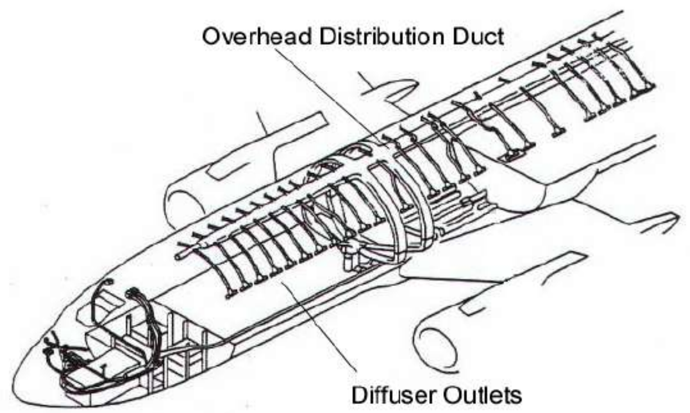


Diagram 12.15 Air Supply to Passenger Cabin

In a passenger air transport aircraft, the conditioned air is brought into the cabin at ceiling level along the length of the passenger cabin via overhead supply ducts and branch pipes to diffuser outlets. Air brought in at this level is often termed as coming in at the hat rack. See diagram 12.15 and 12.16 for an illustration.

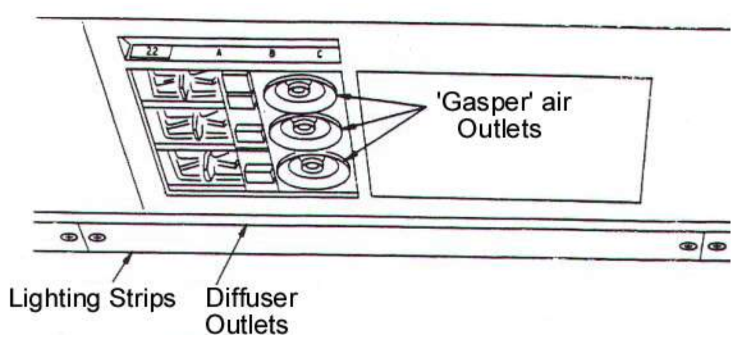


Diagram 12.16 Ventilation for Passengers

Smaller pipes from the diffuser branch pipes take air to the passengers' overhead service panel punka louvres, also known as **gaspers**, which allow the users to direct air onto their faces.

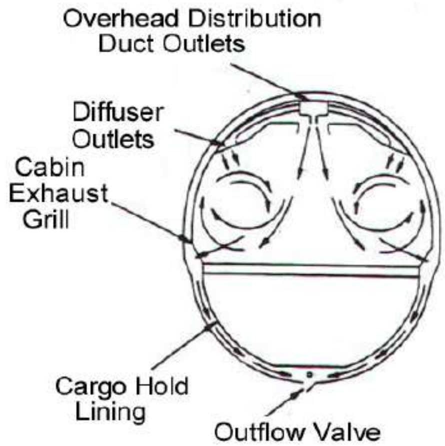


Diagram 12.17 Cabin Air Circulation

Diagram 12.17 shows the circulation pattern for an air transport aircraft, which ensures that the whole cabin is correctly ventilated.

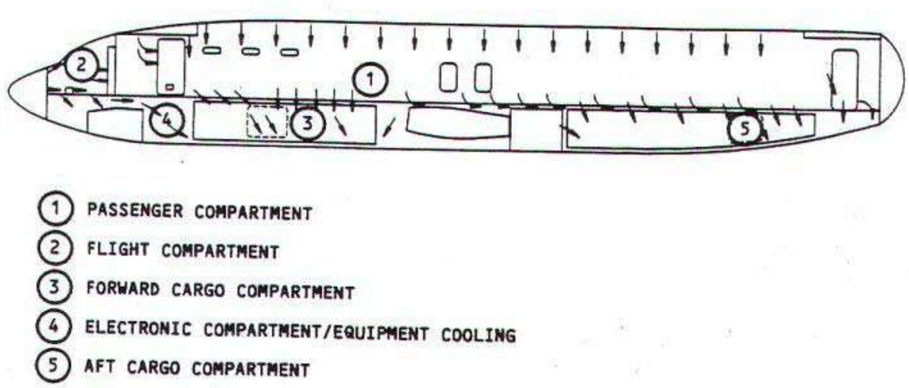


Diagram 12.18 Airflow through a Medium Range Twin Jet

Diagram 12.18 shows the through flow of ventilating air. The flight deck always has a separate distribution supply from the passenger cabin. In large aircraft, the passenger cabin can be divided into zones with each zone having a separate supply duct. See diagram 12.19 for a schematic of the complete air conditioning system of a modern aircraft.

Where cargo holds are heated by warm air, they always use the air that has already passed through the cabin. The exhaust from the flight deck is used to ventilate and cool the aircraft's avionics that are normally located in a bay beneath and behind the flight deck. On medium and large aircraft, there is normally an access hatch from the flight deck into this bay.

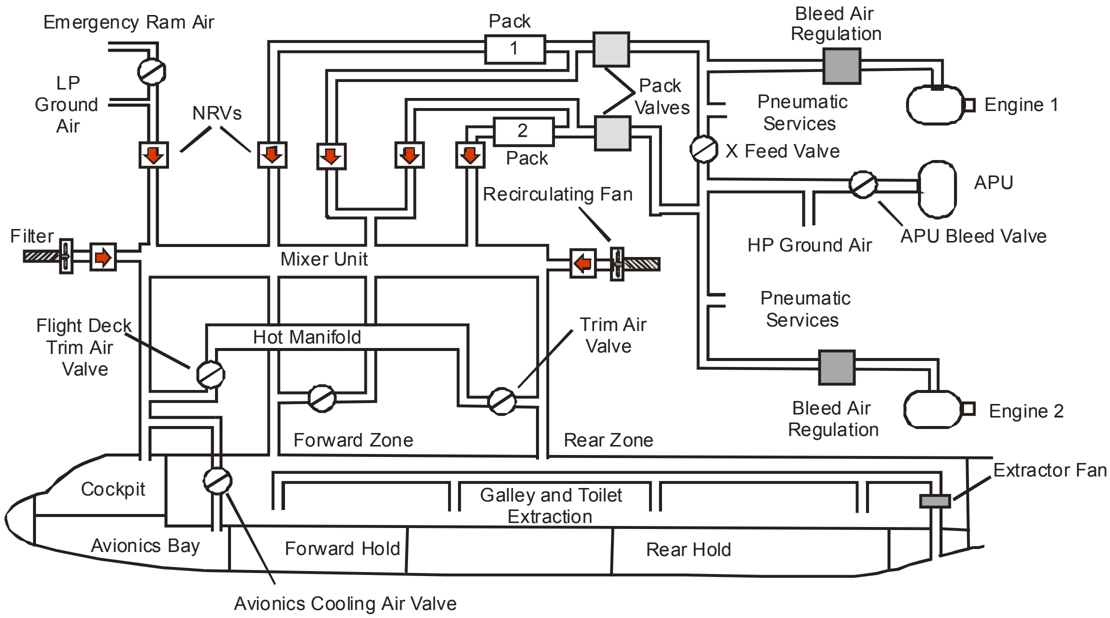


Diagram 12.19 Air Conditioning System Schematic

Diagram 12.19 shows the type of system schematic that is used in aircraft type training notes. While not strictly correct, it allows the pilot to understand the system that is explained.

This aircraft has two air conditioning packs, packs 1 and 2. In normal operation, pack 1 is supplied with bleed air from the No. 1 engine and pack 2 from engine No. 2. Therefore, if there is any problem with an engine (e.g. forward spool oil seal failing, allowing oil into the compression cycle), the engine can be isolated from the pack.

In the event of an engine failing, the crew can elect to open the cross-feed valve and operate both packs from the single engine. This does have ramifications on the aircraft's and engine's performance.

On the ground, the Auxiliary Power Unit or APU normally supplies pack 1 with bleed air to condition the cabin. However, the APU can supply pack 2 via the cross-feed valve. If the APU is unserviceable, hot, high-pressure air from a ground conditioning trolley can be fed into the system. A further connection allows low-pressure pneumatic conditioned air from a cart to be fed directly into the aircraft's distribution system.

The bleed-air regulation valves determine the amount of bleed air and the point from which it is taken. This depends on the engine's rpm and the amount of air required by the different pneumatic systems. At lower rpm, or the need for high temperature, bleed air is drawn from the engine's High Pressure Compressor section. The pack valves allow the pilot to isolate the packs from the bleed air supply.

RECYCLED CABIN AIR

Cabin air is drawn by electric fans through HEPA filter units, which are designed to remove harmful bacteria. The air is then passed into a mixing unit, where it is mixed with air from the packs. This re-oxygenates the recycled cabin air, while the recycled air helps to humidify the fresh air. Air from toilets and galleys is drawn by a separate duct and vented directly to the atmosphere.

Re-circulation of cabin air allows operators to reduce their direct operating costs, as the reduction of air taken from the engines increases the engines' efficiency, allowing a reduction in fuel burn for a given flight.

MINIMUM VENTILATION RATE

The minimum fresh airflow rate for any air transport aircraft is 0.4 lb/min per person for any period exceeding five minutes. This includes systems where the re-circulation of cabin air is counted in the supply of fresh air and is then failed.

TEMPERATURE CONTROL

Diagram 12.20 Cabin Temperature Control Panel

Air that bypasses the packs is fed into a hot manifold, where it mixes and is used to trim the temperature of the air taken from the mixing unit, before it is supplied to the zones in the cabin and the flight deck.

Regulations require that where the total volume of the passenger cabin and flight deck exceeds 800 cubic feet, the flight deck crew (also referred to as the control cabin crew) must have a separate means to control their environmental temperature. This can be seen in diagram 12.20. The flight deck is denoted as CON CAB and in the previous system schematic diagram 12.19.

SMOKE CONTAMINATION

In the schematic diagram 12.21, if smoke is detected on the flight deck, the source is indeterminable as the cross-feed valve is open. For this reason, it is standard practice to keep the two supplies separate.

In the event that smoke is detected in the cabin, the first action is to switch off the re-circulation fans, as this would spread the smoke from one zone to another. Increasing the outflow from the aircraft assists in removing the smoke from the passengers.

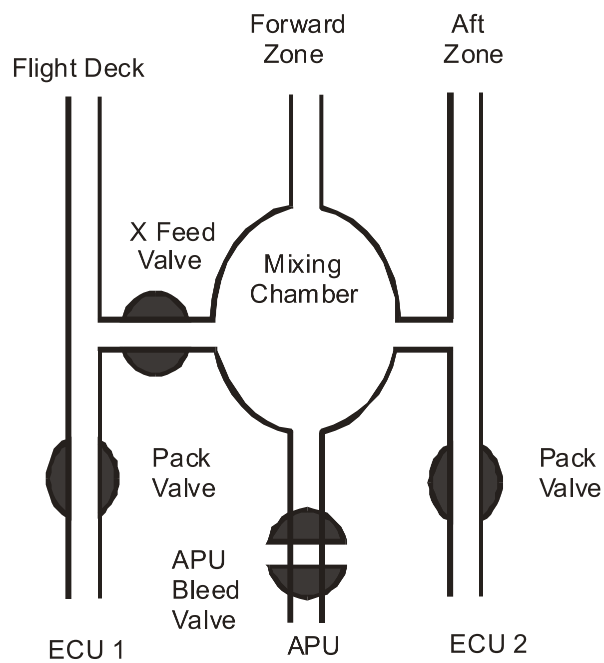


Diagram 12.21 Exam Question Type Schematic

Chapter 13
Pressurisation and Oxygen

ACKNOWLEDGEMENTS

We would like to thank and acknowledge:

For Diagrams and assistance with technical information:

13.1	The estate of Mr. A. Whitlock for the kind permission to reproduce and use this drawing from his book “Behind the Cockpit Door”
13.18	Unknown Photographer
13.25	Virgin Atlantic Airways with special thanks to Mr J Jasper of Virgin Atlantic’s Cabin Crew Training, Horley
13.34	Dräger
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INTRODUCTION

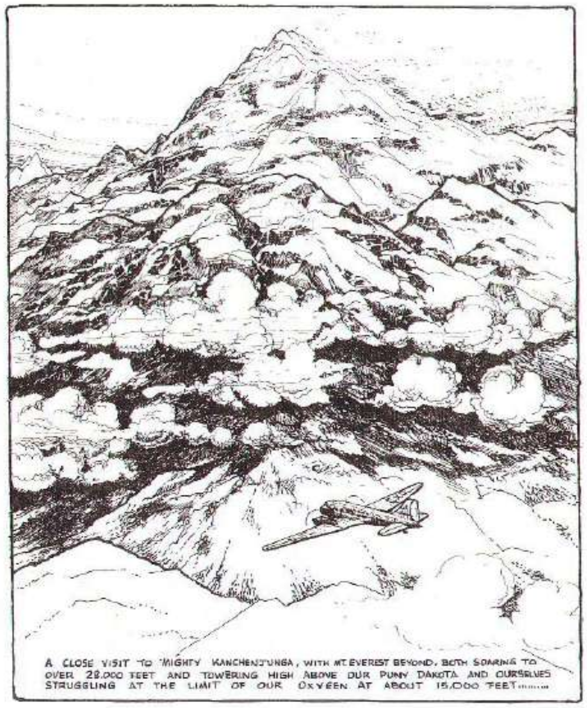

Diagram 13.1 Mount Everest

The human species is very adaptable. It has been able to colonise most parts of the world from the freezing arctic to heated deserts. However, permanent inhabitation stops at 17 000 feet above sea level due to the drop in atmospheric density, affecting the blood's ability to absorb oxygen. See diagram 13.2, ICAO table.

A few very fit individuals have been able to climb Mount Everest (29 029 ft) without carrying supplementary oxygen. Although they have trained over years and ascended slowly, allowing their bodies to acclimatise, they still suffer **mountain sickness**, hypoxia. Modern air transport aircraft can climb to the equivalent altitude in a matter of minutes.

To safeguard passengers and crew from the effects of hypoxia, the Joint Aviation Authority have outlined basic parameters in JARs 23 and 25 that aircraft and system manufacturers must meet. To expand these regulations, the JAA and FAA have adopted the aerospace recommended practice from the Society of Automotive Engineers Inc. (SAE), ARP 1270 aircraft cabin pressurisation control criteria paper. These regulations are laid out in these notes. After reading the main text, refer to Appendix 1 and 2 for JAR-OPS 1 requirements. These can form the basis of JAR–FCL questions.

MAXIMUM CABIN ALTITUDE FOR PRESSURISED AIRCRAFT

The maximum safe altitude for pilots to operate without supplementary oxygen is 15 000 ft. However, as commercial airlines carry tens of millions of passengers annually, who vary in age and fitness, the passenger cabin and the flight deck are designed to maintain a cabin altitude of 6000 to 8000 ft above mean sea level (amsl). Eight thousand feet is considered to be the maximum altitude where the pilot's flying skills and judgement are not seriously impaired, and the partial pressure of oxygen in the air is sufficient for the passengers.

Altitude		Temperature			Pressure	
Feet	Metres	Deg K	Deg C	Deg F	Lb/sq.in.	Millibars
-1000	-304.8	290.13	16.98	62.6	15.24	1050.4
0	0	288.15	15.0	59.0	14.69	1013.2
+1000	+304.8	286.17	13.02	55.4	14.17	977.1
2000	609.6	284.19	11.04	51.9	13.66	942.1
3000	914.4	282.21	9.06	48.3	13.17	908.1
4000	1219.2	280.23	7.08	44.7	12.69	875.1
5000	1524.0	278.24	5.09	41.2	12.23	843.0
6000	1828.08	276.26	3.11	37.6	11.78	811.9
7000	2133.6	274.28	1.13	34.0	11.34	781.8
8000	2438.4	272.30	-0.85	30.5	10.92	752.6
9000	2743.2	270.32	-2.83	26.9	10.51	724.3
10 000	3048.0	268.34	-4.81	23.3	10.11	696.8
15 000	4572.0	258.43	14.72	5.5	8.29	571.7
20 000	6096.0	248.53	-24.62	-12.3	6.75	465.6
25 000	7620.0	238.62	-34.53	-30.2	5.45	375.9
30 000	9144.0	228.71	-44.44	-48.0	4.36	300.9
35 000	10 668.0	218.81	-54.34	-65.8	3.46	238.4
36 000	10 972.8	216.83	-56.32	-69.4	3.29	227.3
36 090	11 000.2	216.65	-56.5	-69.7	3.28	226.3
40 000	12 192.0	Temperature remains constant at **-56.5°C** from the tropopause to an altitude of 65 000 ft, where it starts to rise.			2.7	187.5
45 000	13 716.0				2.14	147.5
50 000	15 240.0				1.68	115.9
55 000	16 764.0				1.32	91.2
60 000	18 288.0				1.04	71.7
65 000	19 812.0				0.82	56.4

Diagram 13.2 ICAO Table

CABIN AMBIENT PRESSURE

Refer to diagram 13.2 for the external ambient conditions at 8000 ft. Chapter 12 explained that the temperature of the cabin needs to be kept between 18 and 24°C. The pressure within the cabin must be maintained at 10.92 psi or higher, which is referred to as cabin ambient. An increase in cabin ambient pressure relates to a decrease in **cabin altitude**, and an increase in cabin altitude relates to a decrease in cabin ambient pressure.

CABIN DIFFERENTIAL PRESSURE — ΔP

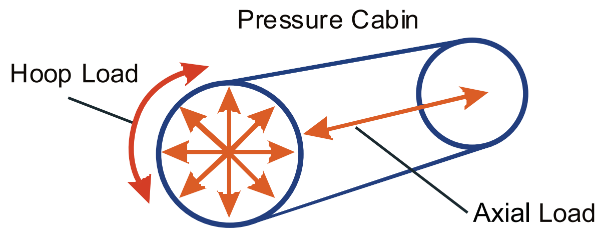

Diagram 13.3 Hoop and Axial Loads

If an aircraft is flying at 40 000 ft, the ambient air pressure around the aircraft is 2.7 psi. The differential between the two pressures (10.92 – 2.7) is 8.22 psi, giving a pressure of 8.22 pounds per square inch. This may not seem much, but this pressure differential creates the hoop and axial loads in the fuselage skins. For example, if one square foot were considered, the air load would be 1183.7 pounds [(12x12) x 8.22]. This is only added to give the reader a feel for the loads applied. **ΔP** is the symbol used for differential pressure.

MAXIMUM DIFFERENTIAL

The maximum differential, or max diff, as it is also referred to, is determined by the aircraft's structural strength. For air transport aircraft, this is currently considered by JAR FCL to be 9.5 pounds. Of course in reality, some aircraft exceed this. With new materials and building systems, the max diff of aircraft will continue to rise. Existing aircraft could have been built with higher max diffs using existing materials but there would have been unacceptable weight penalties.

From the point of view of the cabin structure, the greater the max diff, the higher the aircraft can fly while maintaining sea level conditions in the cabin. For example, if one aircraft has a max diff of 6.4 psi and another a max diff of 8.6 psi, the first aircraft would be able to fly at a maximum of 15 000 ft and maintain sea level conditions, whereas the second aircraft would be able to maintain sea level cabin pressure at a maximum of 22 500 ft.

NEGATIVE DIFFERENTIAL

Pressurised aircraft are designed to act as pressure containers, withstanding a higher internal pressure than that of the surrounding atmosphere. They are not designed to withstand higher atmospheric ambient conditions than cabin ambient. If these conditions develop and are allowed to increase, there is a real danger of structural damage due to the crushing effect of the pressure differential.

SAFETY

To protect the aircraft from structural damage due to excessive ΔP caused by the failure of the normal pressure control system, two outward pressure relief valves termed **safety valves** and two inward pressure relief valves termed **inward relief valves** are fitted.

SAFETY VALVES

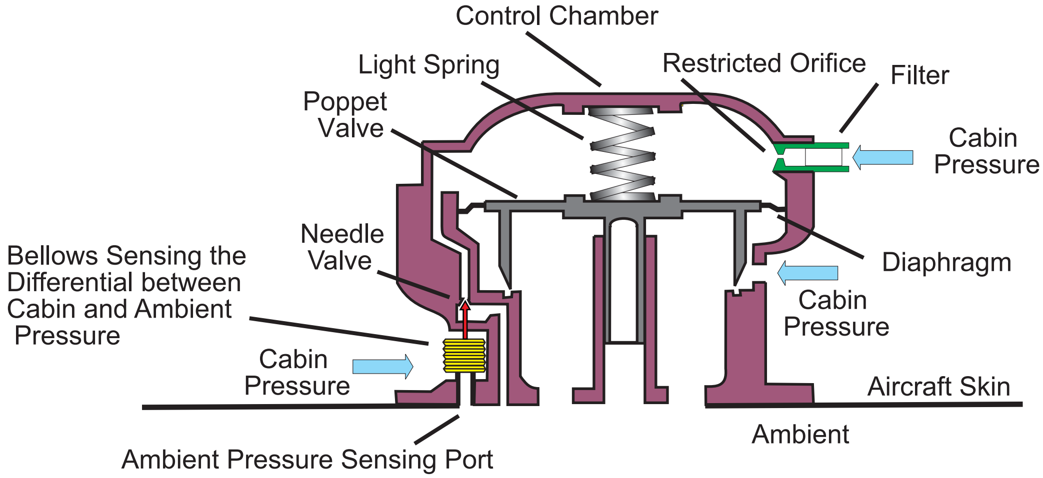

Diagram 13.4 Safety Valve

Two safety valves are fitted to an aircraft's pressure hull. They function independently of each other and the normal pressure control system. Their outflow size is determined by the amount of air that they would have to vent if any of the air pressure ducts within the pressure cabin burst. These valves normally operate at 0.5 psi above the aircraft's maximum differential pressure.

Diagram 13.4 shows a cross section of a safety valve. The diaphragm attached to the poppet valve and the valve case forms a flexible wall between the control chamber and the ambient atmosphere. The control chamber is open to the pressure cabin via a restricted orifice and filter. This allows cabin air pressure to build up within the control chamber but at a slower rate due to the restriction.

The poppet valve is held closed by the combined force of the light spring and the air pressure in the control chamber, which acts over the surface area of the poppet valve and diaphragm. The cabin pressure acting up under the reduced surface area of the diaphragm and poppet valve acts to try to lift the poppet valve, opening a vent to atmosphere.

The bellows senses the differential between cabin pressure, which acts on its external surface and atmospheric ambient pressure, which in turn acts inside the bellows via a sensing port. Under normal conditions, the strength of the bellows holds a needle valve in a vent line from the control chamber to ambient against its seat, separating the control chamber from ambient.

If for any reason the cabin pressure increases above the maximum differential, the differential across the bellows causes it to collapse, opening the control chamber to atmosphere. The amount the bellows collapses is determined by the amount that the aircraft's max diff is exceeded. As the control chamber pressure drops, the force holding the poppet valve closed decreases and the poppet valve lifts, allowing cabin air to vent to atmosphere. A restricted flow from the cabin enters the control chamber. The culmination of these factors is to hold the cabin pressure 0.5 psi higher than normal.

INWARD RELIEF VALVES

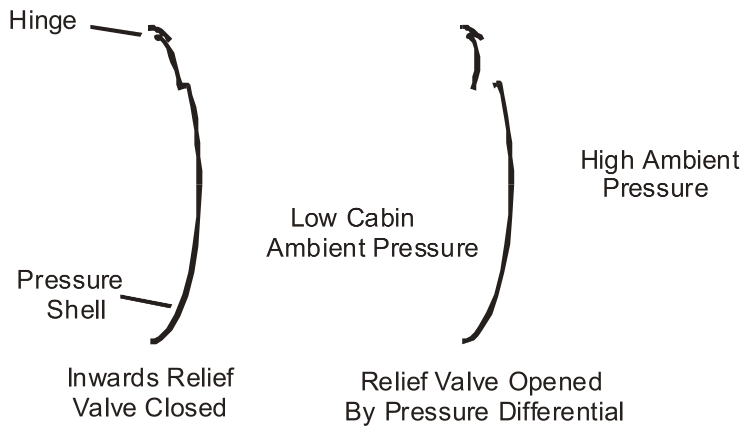

Diagram 13.5 Inward Relief Valve

Inward relief valves can take the form of a simple hinged plate, which is fitted on the inside of the pressure hull and lightly spring-loaded to shut, as seen in diagram 13.5. Under normal conditions, the plate is held closed by the light spring and cabin pressure. In the event that the aircraft's ambient pressure rises above the cabin by 0.5 psi, the pressure differential across the valve plate opens the valve, allowing the cabin air pressure to equalise.

CABIN PRESSURE

The cabin is pressurised using the ventilation air from the air conditioning systems. The flow rate of this is controlled by the mass flow controller (in some aircraft, the crew can alter the rates within set tolerances) and, therefore, is considered fixed. The cabin pressure (ambient altitude) is determined by controlling the outflow of this air. The less air that is allowed to escape, the greater the cabin pressure, and vice versa.

During the aircraft's climb, air has to be allowed to escape from the cabin at a greater rate than inflow from the air conditioning system to allow the cabin altitude to climb. When the aircraft and the cabin arrive at their predetermined level, the outflow must equal the inflow. During the descent, the inflow must be greater than the outflow to enable the cabin pressure to rise.

OUTFLOW

While the mechanics of cabin pressurisation are separate from the air conditioning system, both systems are interrelated for the design of an aircraft. The major component in the cabin pressurisation system is the outflow or discharge valve. The outflow valve must be matched to the aircraft to ensure complete pressure control at the aircraft's max diff if one of the air conditioning packs fails.

Some aircraft designs fit two outflow valves in place of a single valve. Where two or more valves are used, the system must permit complete pressure control over the full pressure range when all inflows are operating and one outflow valve has failed.

RATE OF CHANGE

The term rate of change, or ROC, is given to the value by which the cabin altitude is allowed to ascend or descend. This is normally given in feet per minute or fpm. However, ROC can also be used as **rate of climb** and ROD used for **rate of descent**. The aircraft also has a rate of change. The rate of change of the aircraft and the cabin are interrelated. See the Maximum Rates of Change graph below. Be very careful when reading JAR FCL questions as to which ROC is being considered.

MAXIMUM RATES OF CHANGE

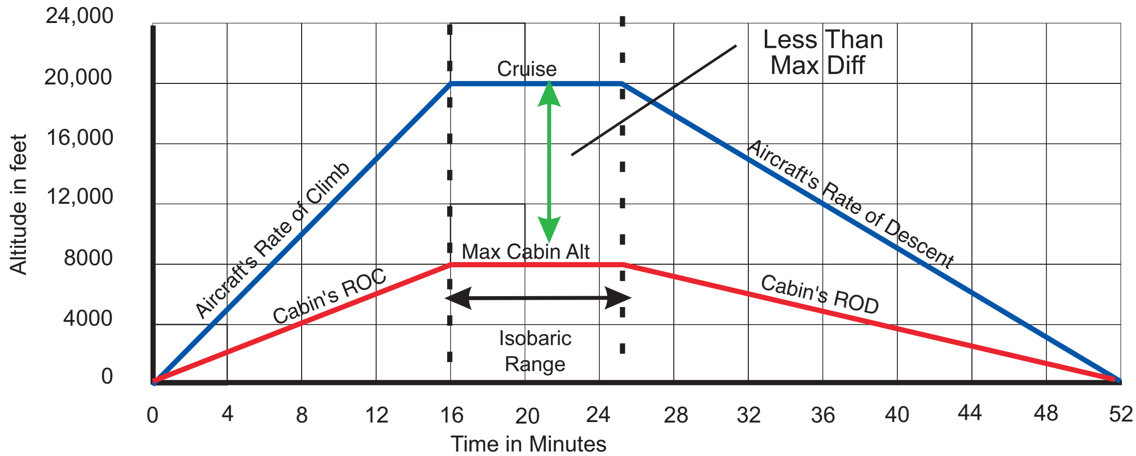

Diagram 13.6 Maximum Rate of Change

The maximum rate of ascent is 500 fpm and 300 fpm for descent, 152m/60s and 91m/60s respectively. These rates have been determined by passenger comfort due to the human ear physiology.

In diagram 13.6, the lower line depicts the maximum rate of climb, the maximum cabin altitude, and the maximum rate of descent for an air transport aircraft's passenger cabin. As the max cabin altitude is 8000 ft and the max ROC 500 fpm, the minimum time for the cabin to reach the max cabin altitude is 16 minutes (8000 ft/500 fpm=16 min).

For physiological and psychological reasons, it is standard for the aircraft to level out from its climb at the same time as the cabin levels off at its cruise altitude. Therefore, the aircraft's rate of climb can be determined by dividing the level-off altitude by the time taken, in this case 20 000 ft/16 min=1250 fpm.

The same method is used to calculate the start of the let down (the maximum ROD for the aircraft is 300 fpm). If a slow aircraft descent is planned, a new cabin ROD can be calculated. Where aircraft are taking off or landing at elevations that are above sea level, these values must be subtracted from the aircraft and cabin height to climb or descend.

For example, if an aircraft is to take off from an airport that is 3000 ft above sea level and land at an airport that is 5000 ft above sea level and have a max cabin altitude of 7000 ft, the pilot uses the max rates of change allowed and levels off to cruise at 14 000 ft.

Cabin Climb		Aircraft Climb	
Max Cabin Alt	7000 ft	Level Off Alt	14 000 ft
Airport Alt	-3000 ft	Airport Alt	-3000 ft
Alt to Climb	4000 ft	Alt to Climb	11 000 ft
ROC 500 fpm	8 minutes	ROC	1375 fpm

Cabin Descent		Aircraft Descent	
Max Cabin Alt	7000 ft	Level Off Alt	14 000 ft
Airport Alt	- 5000 ft	Airport Alt	- 5000 ft
Alt to Descend	2000 ft	Alt to Descend	9000 ft
ROD 300 fpm	6.67 minutes	Time 6.67 min	1349 fpm

EUSTACHIAN TUBE

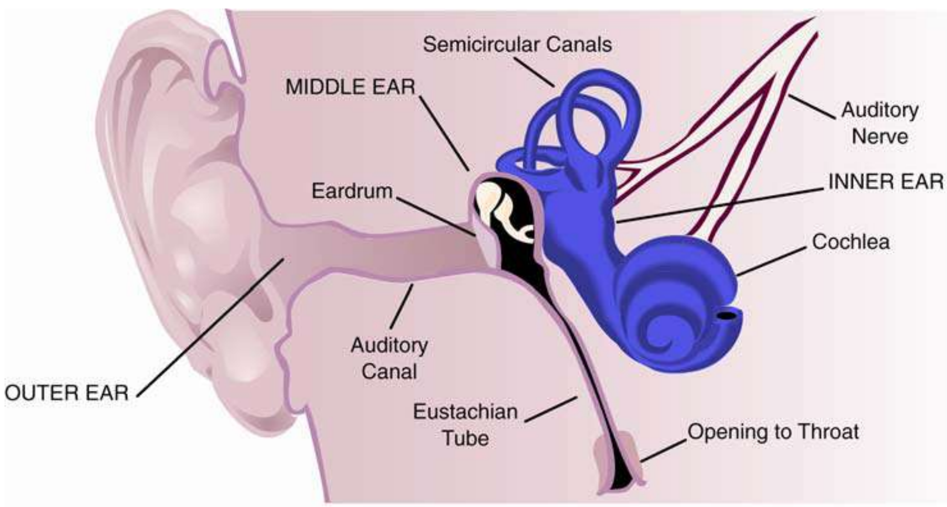

Diagram 13.7 Human Ear

Diagram 13.7 shows the composition of the human ear. The Eustachian tube or internal auditory passage connects the middle ear to the nasopharynx (nasal passage). The Eustachian tube at the nasal end is normally held closed by a muscle and fleshy membranes that surround it. This forms an air chamber with the membrane of the eardrum as the point at which the differential between the ambient pressure and the internal pressure is felt.

In normal circumstances, the Eustachian tube muscle contracts occasionally, allowing the pressure differential across the eardrum to equalise. Swallowing or yawning can also induce this equalisation. Airline passengers are not trained in techniques of opening the tubes, and may be suffering from colds or other medical conditions that cause the membrane around the lower end of the Eustachian tube to become swollen, blocking the tube.

Any increase in the rates of change above the maximum of 500 fpm for ascent, or 300 fpm for descent, results in an increase in discomfort for passengers. At high rates of change, discomfort turns to pain, and the pressure differential can cause burst eardrums.

CABIN PRESSURE CONTROL SYSTEMS

There are three types of cabin control system used in pressurised air transport aircraft. These are:

- Pneumatic pressure control system
- Electro-pneumatic pressure control system
- Electronic pressure control system

The other systems must be designed to meet the following criteria in order of precedence:

- Safety
- Passenger comfort
- Economic factors, crew convenience

The design of any system must also consider the following:

- High rates of change of cabin inflow which may be caused by rapid engine or cabin source variation
- High rates of change of local ambient static pressure at the overboard location of the cabin air outflow valve, such as may occur during aircraft rotation just prior to take-off
- Lack of responsiveness of the cabin pressure control system

The crew must be provided with the means to monitor the following:

- Cabin pressure rate of change
- Cabin pressure altitude
- Cabin to ambient pressure differential
- The outflow valve position

VARIABLE ISOBARIC CONTROLS

All air transport aircraft cabin pressure control systems must incorporate an altitude selector permitting the crew to vary the selected isobaric cabin altitude based on the flight plan and the aircraft's structural safety pressure differential limitation. The selector for the isobaric control must be located in a position that is accessible to the crew. This normally forms part of the cabin pressurisation panel, which also incorporates the means to monitor the items listed above.

The range of adjustable isobaric control for aircraft that operate at airports that are below 10 000 ft amsl is between -1000 ft to +10 000 ft. For aircraft that are certified to operate to and from airports above 10 000 ft amsl the adjustable range should be between -1000 ft to +14 000 ft.

PNEUMATIC PRESSURE CONTROL SYSTEM

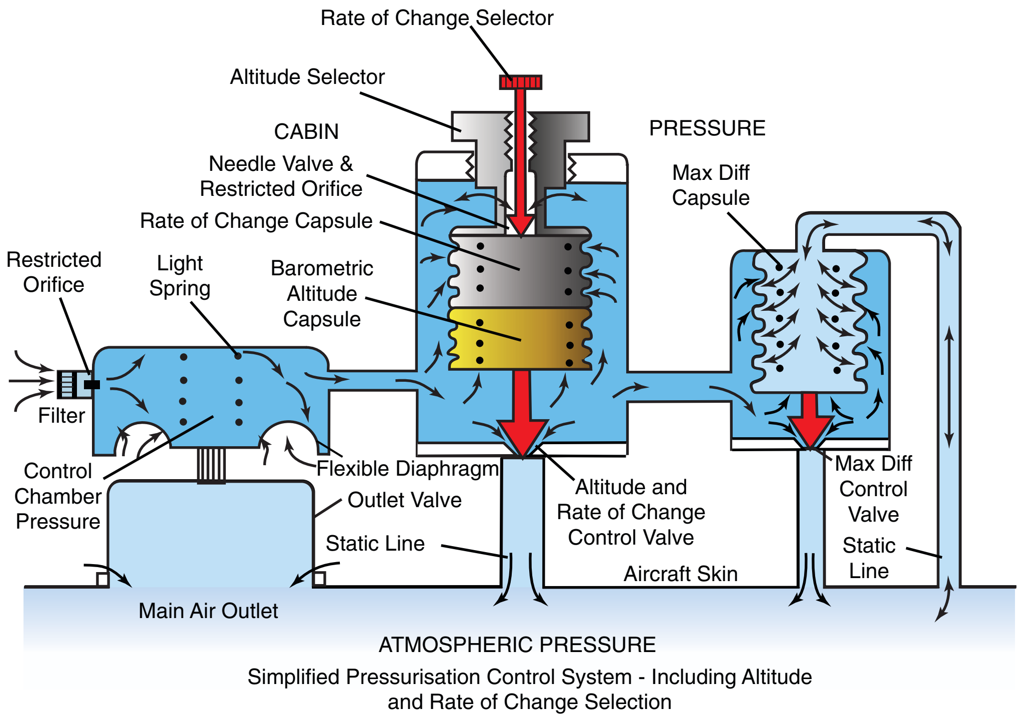

Diagram 13.8 Simplified Pneumatic Pressure Control System

Diagram 13.8 shows a simplified schematic of a pneumatic pressure control system. This is used as a vehicle for the explanations of how the following function and operate:

- The discharge valve
- The cabin pressure controller
- Altitude capsule
- Rate of change capsule
- The max differential capsule

Explanations on climbing, cruise, and descent follow.

DISCHARGE VALVE

Diagram 13.8 shows the control chamber of the discharge valve pneumatically linked to the cabin pressure controller. Within the control chamber, a light spring acts to hold the discharge valve closed when there is no pressure differential and to assist the control chamber pressure.

The valve is located within the pressure hull and is subjected to cabin pressure, which acts under the flexible diaphragm that is connected to the outflow valve. A restricted orifice allows cabin air to enter the control chamber, but the control chamber is always lagging behind the cabin pressure by 0.25 psi due to the restriction.

CABIN PRESSURE CONTROLLER

Referring to diagram 13.8, the cabin pressure controller consists of two separate controls, the cabin rate of change and the cabin altitude, that function together to give one output.

The cabin altitude selector consists of a barometric capsule connected to a combined rate of change and altitude needle valve at one end and a threaded selector knob at the other. If the cabin altitude is set at one altitude (e.g. 7000 ft) and the crew then elect to lower cabin altitude, the knob is screwed inward, and vice versa.

Located within the altitude selector is a rate of change selector. This takes the form of a threaded needle valve, which acts as a variable restrictor in the orifice of the rate of change capsule. Screwing the ROC needle valve inward increases the restriction, slowing the airflow into or from the capsule, which reduces the rate of change.

Control chamber pressure acts around the rate of change capsule and the barometric altitude capsule. Each has a small internal spring that prevents it from collapsing. The barometric capsule has a fixed internal pressure (8000 ft) and reacts to any increase or decrease in the surrounding air by expanding or contracting.

If the barometric capsule is considered fixed at one end, any expansion or contraction alters its overall length, moving the control valve closer to or further away from its valve seat. However, in this case, the barometric capsule is attached to the rate capsule and some of the barometric capsule movement is lost as the rate capsule expands and contracts. The two capsules work in opposition to each other, the rate of opposition determined by the selected cabin ROC.

INITIAL INFLATION PRESSURE

Before engine start for a take-off from a sea level airport, with the cabin doors open, the airport ambient pressure is felt inside the cabin, within the outflow valve's control chamber, as well as outside the hull. The light spring ensures that the valve is seated.

On start up, with the cabin doors closed and air conditioning selected on, the cabin pressure rises more quickly than the pressure in the control chamber due to the restrictor. When the pressure differential across the diaphragm reaches 0.25 psi, the outflow valve opens. This is termed **initial inflation** and 0.25 psi is the initial inflation pressure.

THE CLIMB

The crew selects the cabin altitude and rate of climb for the flight plan prior to the take-off run. For example, if cabin altitude of 8000 ft at a ROC of 500 fpm is selected, the cabin pressure has to decrease by 3.77 psi (from 14.69 psi to 10.92 psi) over 16 minutes. This requires a pressure drop of 0.236 psi per minute.

The outflow valve, which has to be sufficiently open to allow the total inflow of air to exhaust to prevent any increase in cabin pressure, has to open further to achieve the selected rate of climb. This is achieved by the control chamber pressure acting on the barometric altitude capsule and the ROC capsule. The capsules react together holding the control needle valve off the seat, resulting in the control chamber pressure decreasing. This in turn makes it possible for the outflow to open, further allowing the extra mass of air equivalent to the 0.236 psi to vent to atmosphere.

UNANTICIPATED STEPPED CLIMB

Diagram 13.9 is included to illustrate one of the disadvantages of the pneumatic controller. The cabin rate of change can become the same as the aircraft's rate of climb if the aircraft is put on hold at an altitude below the pre-selected cabin altitude.

For the example shown in Diagram 13.9, the crew of an aircraft with an ROC of 1250 fpm have selected a cabin altitude of 8000 ft and cabin ROC of 500 fpm. After take-off, the crew is told to hold at 5000 ft until given clearance to continue the climb.

The aircraft reaches the hold level in 4 minutes. If the crew does not take any action, the cabin reaches this altitude in 10 minutes. Now as ambient equals cabin altitude, a signal from the controller, which is used to open the valve fully on landing to ensure that the aircraft does not land pressurised, opens the discharge valve fully.

For the example, the crew has been cleared to continue the climb at the 20-minute point. This could be at any point after the cabin has equalised with ambient. As the aircraft recommences its climb at the original rate, the fully open outflow valve allows the cabin to climb at the same rate as the aircraft until it passes through 8000 feet, where the cabin altitude capsule re-establishes control and closes the outflow valve.

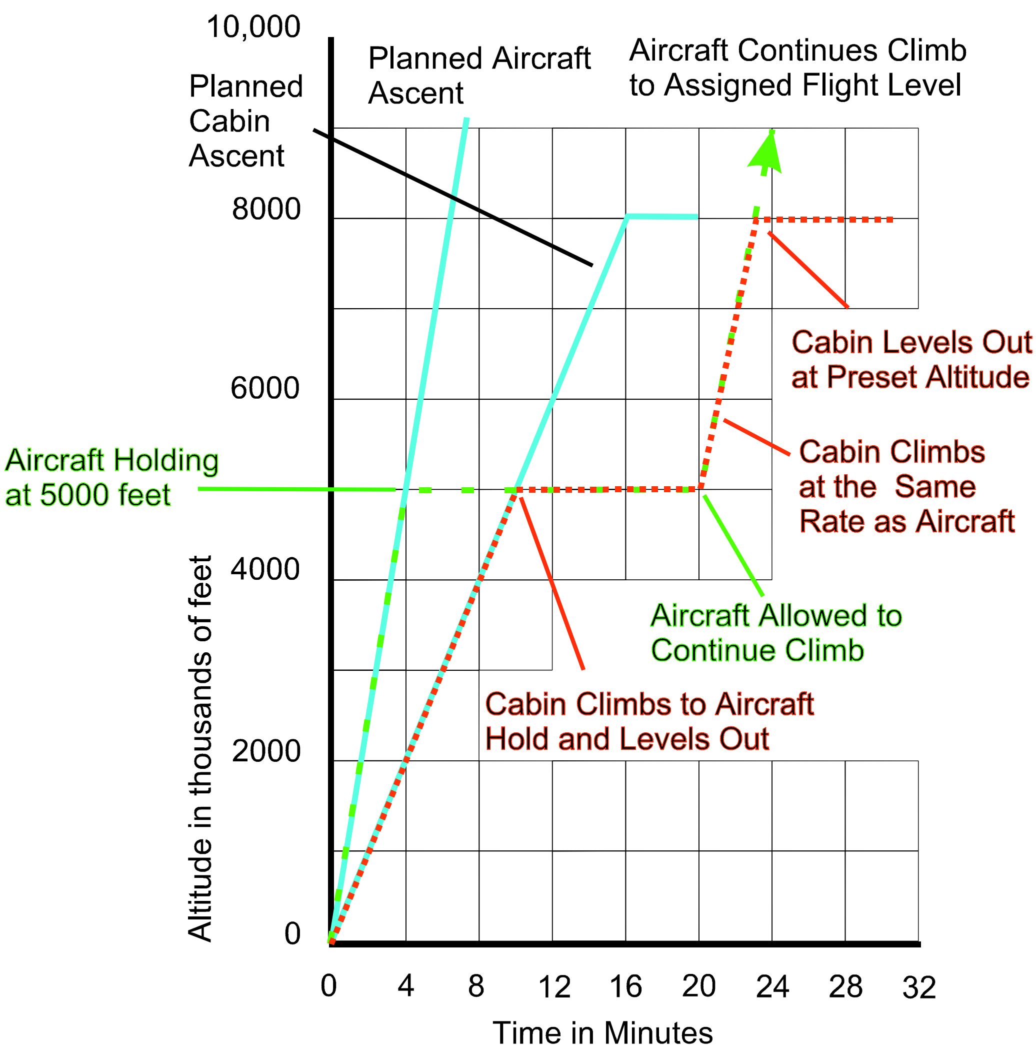

Diagram 13.9 Unanticipated Stepped Climb

This leaves the passengers and crew dazed and surprised. The pressure control system functions correctly for the rest of the flight and there will be no fault found on investigation after landing.

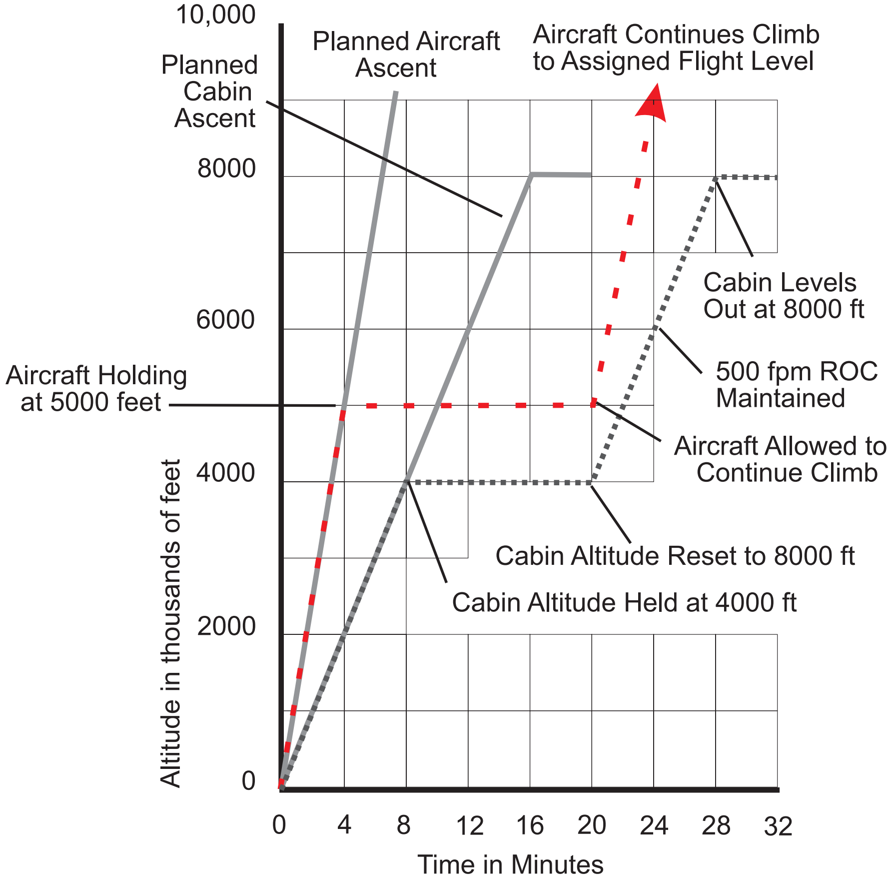

Diagram 13.10 Correct Procedure for Stepped Climb

To overcome this, the crew must select a cabin altitude below the hold level when informed of the hold. The cabin reaches this at the predetermined rate and the controller maintains this cabin altitude during the hold. When the crew is told to resume the climb, they reset the cabin altitude to the desired level and climb the aircraft as planned. The cabin rises at the correct rate to the selected altitude, as in diagram 13.10.

CRUISE — ISOBARIC CONTROL

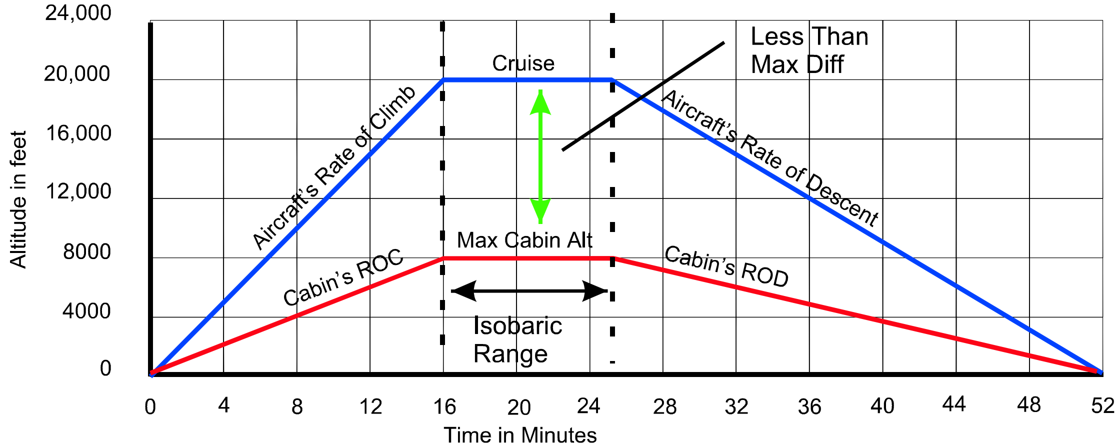

Diagram 13.11 Isobaric Range

After 16 minutes, the ROC capsule has equalised with the reduced control chamber pressure. At this point, the altitude capsule equals the control chamber pressure and maintains the outflow valve in a set position. During the cruise, the barometric pressure of 1013.2 is set on the controller to ensure that it operates from msl. Any alteration in cabin pressure results in the altitude capsule either being compressed or expanding.

This movement is termed **isobaric control**, as the control system acts to maintain the selected cabin pressure. Isobaric control functions between the point where the cabin levels off from the climb to its point of descent (see diagram 13.11) provided that the differential between the cabin altitude and the aircraft is less than its maximum.

PARTIALLY OPEN

The outflow valve is open in one position, maintaining the cabin at the selected altitude. This is termed **partially open**. Any increase in cabin pressure results in the outflow valve moving further toward open, and, conversely, any reduction in cabin pressure results in the outflow valve moving further toward closed.

As JAR FCL questions can be worded around the expected movement of the outflow valve, the term partially open is taken as meaning the valve is just off its valve seat. Do not fall for this condition being termed partially closed.

MAXIMUM DIFFERENTIAL CAPSULE

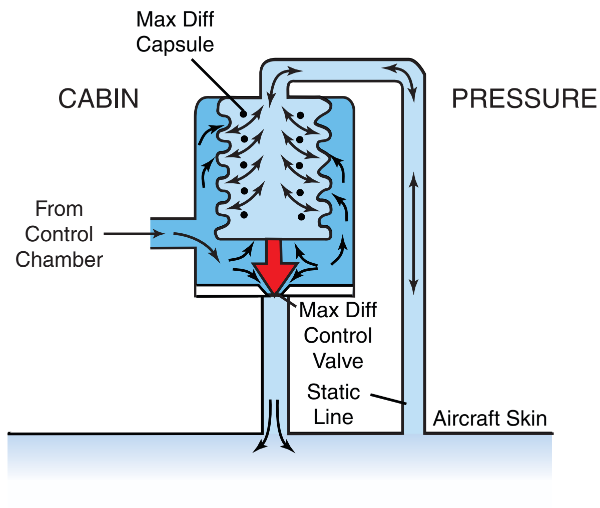

Diagram 13.12 Maximum Differential Capsule

Diagram 13.12 shows a max diff capsule. The capsule consists of a bellows with an internal light spring attached to a needle valve at one end and the control chamber at the other. The capsule is subjected to control chamber pressure externally and ambient pressure internally.

MAX DIFF PROFILES — OFF SCHEDULE DESCENT

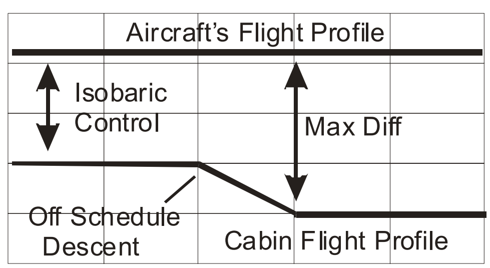

Diagram 13.13 Max Diff Control of an Off Schedule Descent Profile

Refer to Diagram 13.13. If the control chamber pressure increases above the max diff value, indicating a cabin descent while the aircraft maintains its flight level (therefore, ambient pressure does not change), the valve collapses, allowing control chamber air to vent to atmosphere. This opens the discharge valve, venting cabin air to atmosphere to maintain the cabin pressure at the max diff value.

MAX DIFF PROFILES — AIRCRAFT CLIMB

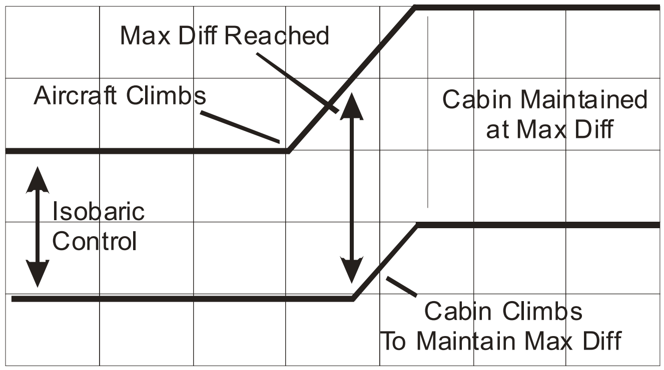

Diagram 13.14 Max Diff Control of Cabin Pressure if Aircraft Climbs

If an aircraft with its cabin controlled by the isobaric range has to climb to an altitude where the selected cabin altitude cannot be maintained without exceeding max differential, the max diff capsule opens the discharge valve at the point where the isobaric control ends. This allows the cabin altitude to rise to the max diff and maintain this higher cabin altitude until the aircraft descends.

PRESSURE BUMPING

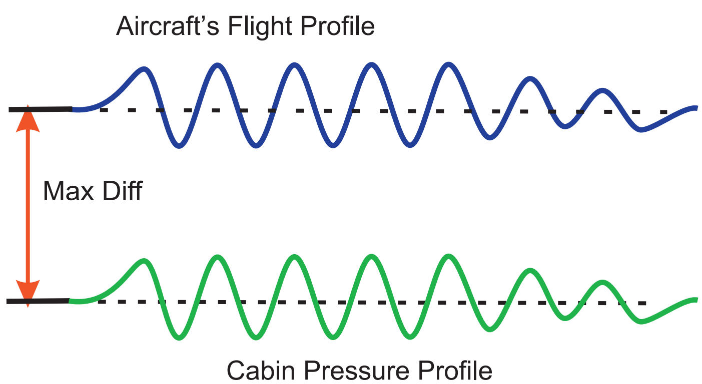

Diagram 13.15 Cabin Pressure Bumping

If an aircraft is flown so that the differential between the cabin altitude and the aircraft's altitude is exactly max, as per diagram 13.15, any disturbance to the aircraft which results in it climbing, results in the cabin altitude also climbing to maintain the max diff. For aircraft that fly with a natural phugoid or those that are returning to stability after an upset, the result is a constant changing of pressure within the cabin as the discharge valve "hunts" to find the correct position. This creates discomfort for both crew and passengers.

To prevent this from occurring, the aircraft should be flown at a level lower than the max diff altitude by some 600 to 1000 ft, allowing the aircraft to move up and down while the cabin remains at the preset altitude.

DESCENT

Prior to descent, the flight crew alter the barometric setting to that of the destination airport. The control system now works in reverse, slowly closing the outflow valve to build up cabin pressure. In diagram 13.11, the indication is given that the cabin arrives at the airfield elevation at the same time as the aircraft touches down.

For older and smaller aircraft of the type that use the pneumatic pressure controllers, landing with the aircraft pressurised would increase the structural load on the airframe to a point where its fatigue life would be shortened or structural damage would occur. As the pressure controller has an allowed tolerance 125 ft each side of the selected cabin altitude, crew reliance on the pressure controller to open the outflow valve at the exact moment of touch down could lead to the aircraft landing pressurised, albeit unknowingly.

To overcome this, a barometric pressure, some 700 ft above that of the landing field, is set on the cabin pressure controller. As the aircraft descends, its rate of change increases the cabin pressure. When the aircraft passes through the 700 ft mark on its final approach, the cabin pressure and the ambient pressure are equal, so the outflow valve is driven open. The cabin's final rate of descent is that of the aircraft, which is likely to be less than 200 fpm, which is within the body's acceptable range for pressure change, as seen in diagram 13.16.

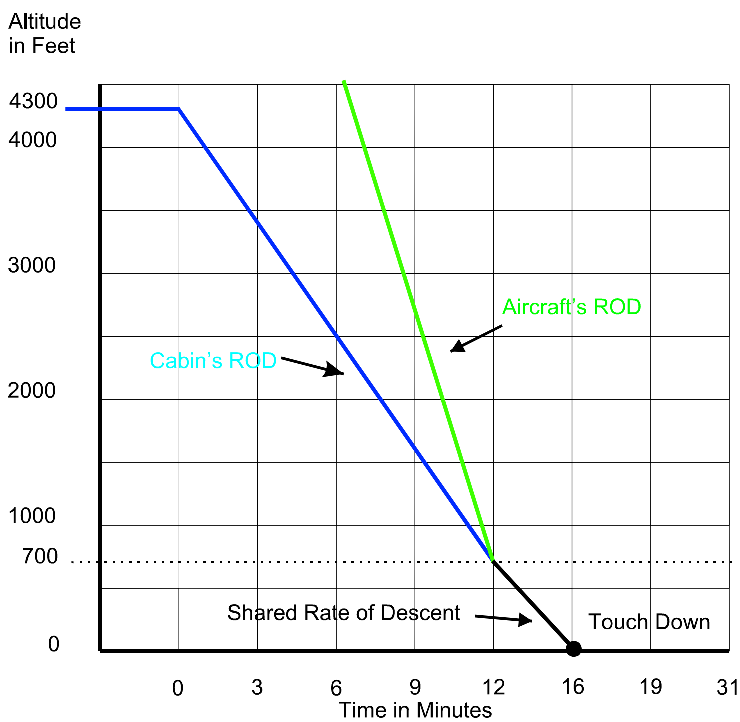

Diagram 13.16 Pneumatic Controller Landing Profile

Diagram 13.16 illustrates the following example. An aircraft with a cabin altitude of 4300 ft is to land at a sea level airport. The crew set the barometric pressure for the airport +700 ft. The cabin has to descend 3600 ft (4300 ft – 700 ft) at a ROD of 300 fpm. This requires a time of 12 minutes. The aircraft's ROD is 2000 fpm. As the cabin and aircraft pass through the 700 feet mark, they share the final ROD of 200 fpm, taking 3.5 minutes from this point to touch down.

GO AROUND

In the event of an aborted landing, termed a **go around** or **landing climb**, the crew must reset the cabin altitude above the airport's altitude as the aircraft climbs to minimum safe altitude, (MSA), to prevent the cabin climbing at the same rate as the aircraft.

DUMP VALVE

Cabin pressure control systems that use pneumatics to operate discharge valves require the fitting of a dump valve, as seen in diagram 13.17. In the event of the pneumatic controller or discharge valve failing, by driving the discharge valve shut, termed an **off schedule descent**, the cabin pressure can be controlled manually by adjusting the position of the dump valve. This is termed **inching**.

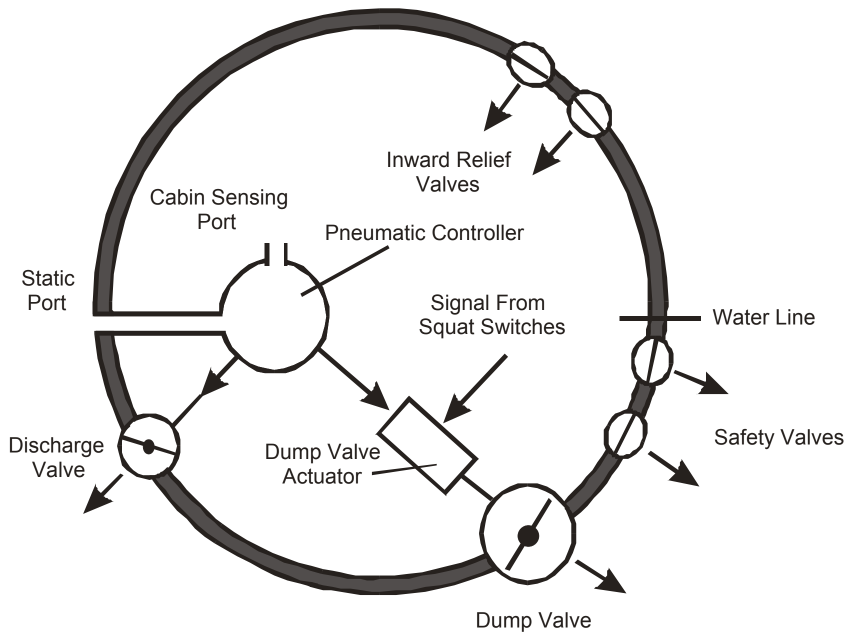

Diagram 13.17 Dump Valve

To clear smoke or fumes from the aircraft, the crew can inch the dump valve open to increase the through flow, whilst maintaining cabin pressure within limits. On landing, the air/ground logic (squat switch, weight on wheels, weight on ground) switch signals the dump valve to drive fully open. This ensures that the aircraft is depressurised on touchdown. Dump valve control switches are guarded, and the guard is normally coloured red to indicate to the crew its importance.

DITCHING

Diagram 13.18 Boeing 707 Afloat After Ditching Through Lack of Fuel

To assist in keeping a ditched aircraft afloat for as long as possible, all valves that would be below the aircraft's water line (see diagram 13.17) must be able to be physically closed and held closed against the water pressure. This can take the form of a mechanical linkage that closes and holds the valve closed or a means to direct water into a valve's control chamber, so that the water pressure itself holds the valve closed.

The ditching control is guarded and the guard, normally a spring-loaded cover, is coloured red to indicate to the crew its importance and to aid location. All the valves that are affected are closed by the operation of a single switch. However, operation of the ditching switch does not prevent the ingress of water through leakage. Designers keep this to a minimum by fitting seals and gaskets.

PNEUMATIC CONTROLLER AND INDICATIONS

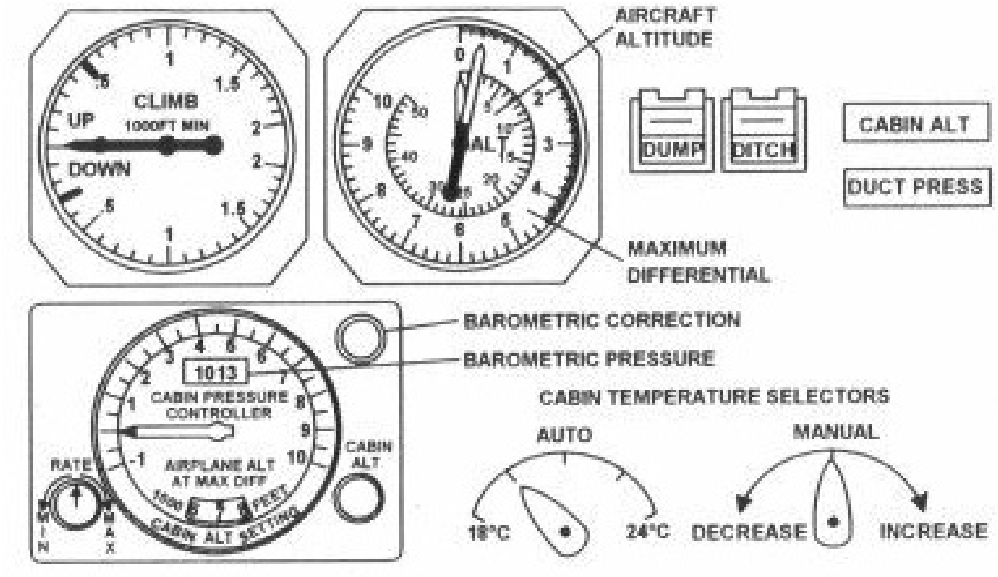

Diagram 13.19 Pneumatic Pressure Controller

Diagram 13.19 shows a typical cabin pressurisation panel for a pneumatic controller. The top left instrument is the cabin's vertical speed indicator (VSI). This is marked with the maximum accepted rates of change. This VSI only shows the aircraft's rate of change if the cabin and ambient pressures are the same.

The second instrument is a combined differential gauge and aircraft altimeter. The differential gauge registers the differential across the aircraft's pressure hull and has a green arc up to the max diff, which is denoted with a red line. Below them is the pressure controller itself. This has an outer scale showing the cabin pressure. The pilot uses the cabin alt selection knob, lower right, to move the pointer to the selected cabin altitude. As the cabin altitude is selected, the maximum aircraft altitude at maximum differential is displayed in the lower window. This allows flight crew to check planned flight levels with selected cabin altitudes.

The window at the top of the gauge shows the barometric pressure as set by the barometric correction knob. The cabin rate of change knob is located to the left of the main dial, while the only indications are max and min. The APR specifies that the rate control should be smoothly adjustable from a minimum rate of 50 fpm to not less than 750 fpm.

ELECTRO-PNEUMATIC PRESSURE CONTROL SYSTEM

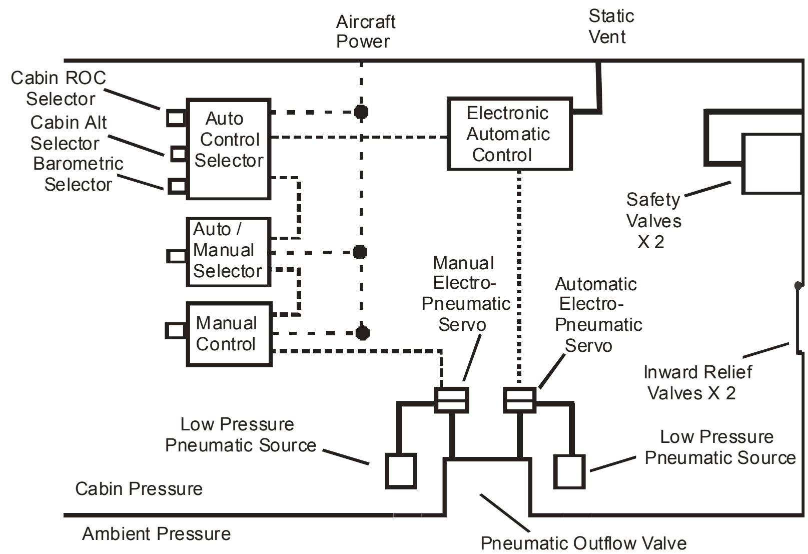

Diagram 13.20 Electro-Pneumatic Pressure Control System

These systems, often termed hybrid systems, came as an intermediate stage between the full pneumatic controller and the electronic controller, as used on modern large air transport aircraft. These systems are commonly used on commuter and executive aircraft due to their low costs. Diagram 13.20 depicts a schematic layout for an electro-pneumatic system, showing the outflow valve being operated by low-pressure pneumatics that are controlled by electrical signals. If the manual selector is placed in manual, the manual electro-pneumatic servo operates the outflow valve.

ELECTRONIC PRESSURE CONTROL SYSTEM

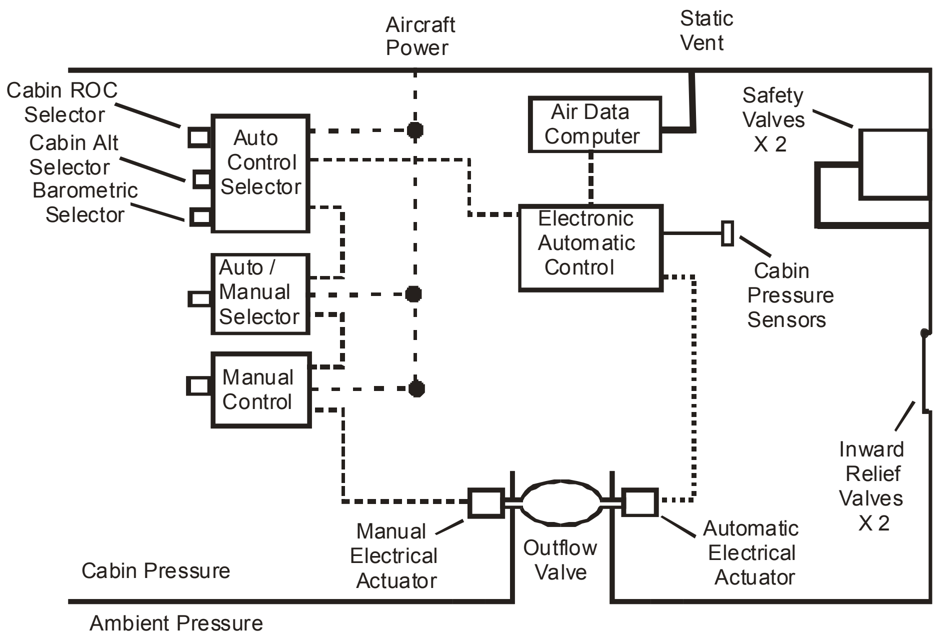

Diagram 13.21 Electronic Pressure Control System

Diagram 13.21 depicts the basic layout of an electronic pressure controller. Here, the pneumatically operated discharge valve is replaced with a totally electrically operated system using two independent electrical motors. These have different levels of sophistication and automation.

Two of the advantages of using automatic electronic systems are the reduction in the crew's in-flight work load, and the ability, when coupled with the increase in fuselage structural strength of large modern air transport aircraft, to take off and land pressurised. Therefore, these controllers must operate to provide the desired level of comfort during the following segments of flight:

- Door closure and air inflow initiation
- Taxi from ramp
- Take-off and transition from ground to flight mode
- Climb
- Cruise or hold
- Descent and barometric correction
- Landing and transition from flight to ground mode
- Taxi to ramp
- Door opening

The profile in Diagram 13.22 shows how a modern aircraft starts to pressurise the cabin once the doors are closed and the engines are running. During the taxi out to the take-off point, the aircraft is held at an altitude below that of the airfield from which it is departing. If the crew elect to fly the cabin at max diff or just under, the cabin pressure is increased in the take-off climb, pushing the cabin below sea level or below field/airport elevation.

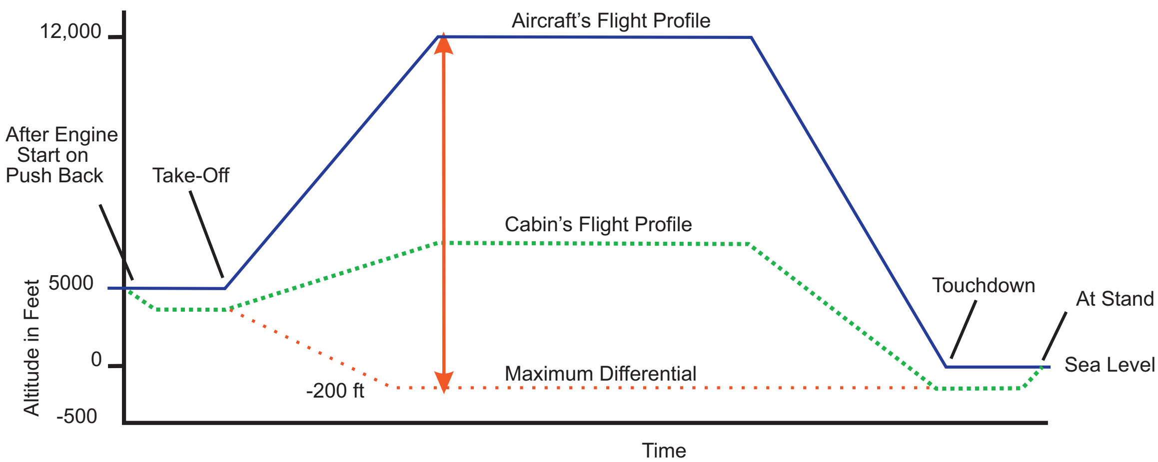

Diagram 13.22 Profiles for an Aircraft Taking-Off and Landing Pressurised

Prior to touchdown, the cabin altitude is held at a pressure of 0.1 psi below that of the landing field. This equates to 200 ft. This pressure is maintained as the aircraft touches down and is allowed to rise as the aircraft taxis to the stand, so that when the doors are opened, the cabin pressure has equalised with the airfield ambient pressure.

It is more likely that the crew will fly the aircraft with the cabin at a higher altitude to save fuel. This appears as the cabin's flight profile in diagram 13.22. After take-off, the cabin altitude is allowed to rise under rate control and is maintained for the cruise under isobaric control. In descent, the cabin again is under rate control but again is taken to a pressure altitude some 200 ft below the landing field.

ELECTRONIC CONTROLLER OPERATION

Diagram 13.23 depicts different operational functions of an automatic pressure controller as fitted to a medium range, twin jet type aircraft. The outflow valve is of the sugar scoop type that is fitted toward the rear of the underside of the fuselage. This type of outflow valve is designed to minimise drag and assist in energy recovery.

Either of two electric motors mechanically actuate the valve. To build in redundancy, AC powers one motor while DC powers the other. Both motors are connected to the cabin pressure controller. The cabin pressure controller is linked to the air data computer. This feeds information on the ambient pressure. The cabin pressure is also fed to the cabin pressure controller, as is a signal from the undercarriage air/ground logic system.

The actual control panel is divided into four sections. The left-hand section is for automatic operation. The middle section is the standby section, which allows the crew to have semi-automatic operation in the event of automatic control failure.

The right-hand section allows the crew to manually control the outflow valve in the event of failure of both automatic and standby mode. The bottom section, the mode selector panel, allows the crew to select the different modes of operation.

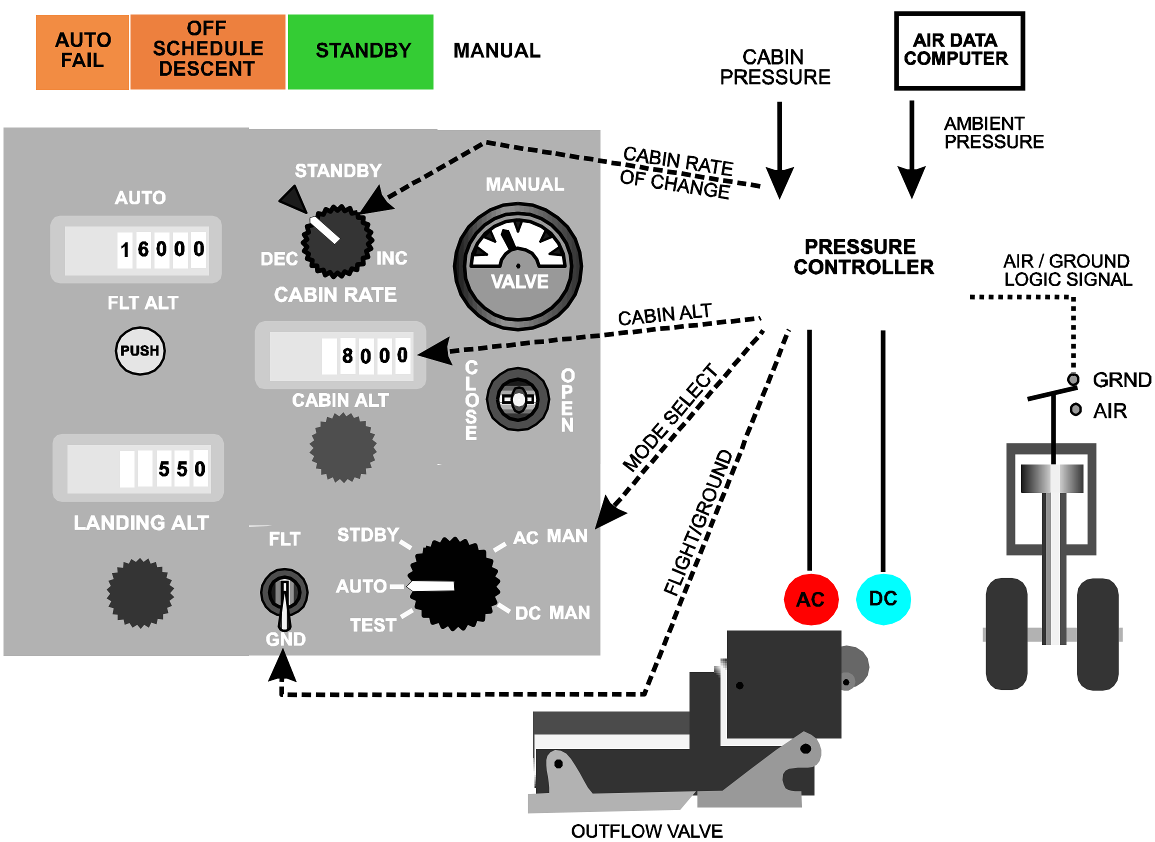

Diagram 13.23 Electronic Controller Operation

AUTOMATIC OPERATION

Refer to diagram 13.23. Prior to take-off, the crew sets the flight level and the destination airfield's elevation on the left-hand panel, the cabin altitude on the standby panel, and auto on the mode selector, and move the ground/flight switch to flight. If there are no alterations to the flight plan or system failures, this is the extent of the crew's cabin control workload apart from monitoring.

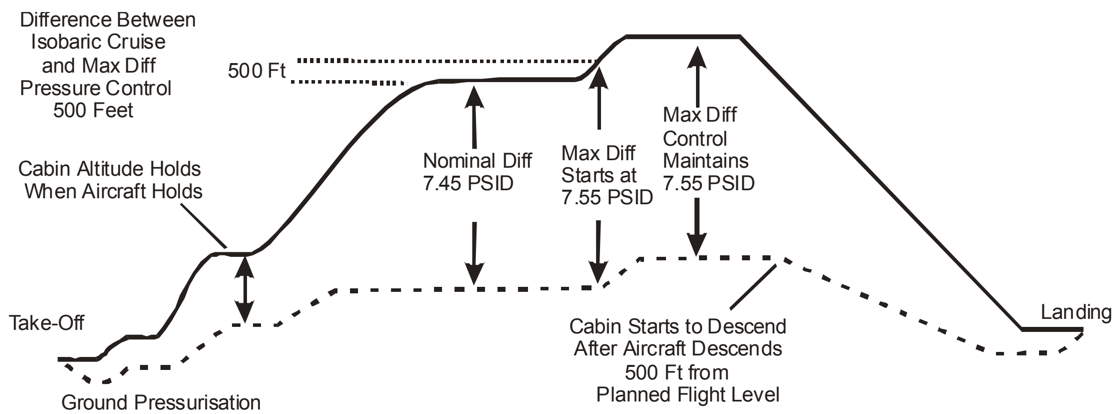

Diagram 13.24 Stepped Climb Profile for a MRJT type Aircraft

Diagram 13.24 depicts how an electronic pressure controller on automatic maintains the cabin pressure. As the system is operating on a computer program, any aircraft holds in the climb automatically cause the cabin altitude to hold. The profile also shows when the cabin is programmed to start its descent. The term **psid** stands for **psi differential**, a fairly common term.

If the aircraft has to return to the departure airfield, the air data has the ambient conditions programmed into it, so there is no action required. If the aircraft has to divert, the new landing field elevation has to be entered. More modern systems are programmed through the EFIS system and, therefore, all information is accessed via the computer.

If the crew selects the FL, cabin Alt, landing Alt, and the mode but fail to select air on the air/ground selector, as the gear leaves the runway, the pressurisation system automatically goes to air mode. In the event of a return to the departure airport, the crew have to set the landing altitude.

STANDBY

If the auto system fails, an amber "auto fail" light illuminates. The crew switches the mode selector to standby. This cancels the auto fail light and illuminates a green standby mode light. In the standby mode, the crew use the cabin rate of change control knob to adjust the cabin pressure. The machine still maintains isobaric control and max diff control.

OFF SCHEDULE DESCENT

If in standby mode the cabin pressure starts to increase, an amber off schedule descent warning light illuminates. This indicates a failure in the control of the outflow valve. In this event, the crew must select manual mode.

MANUAL MODE

The manual selector has two manual positions, one for AC, the other for DC. Under normal circumstances, all outflow valve operations are carried out using AC via the AC motor. When manual mode is selected, the green standby light and amber off schedule descent lights are extinguished, and a white “manual” lamp illuminates.

In manual mode, the crew have to inch the outflow valve to obtain the cabin pressure required and the ROC. This is done using the momentary switch to operate the selected valve motor with reference to the valve position-indicating gauge.

While the manual system should be capable of making a 2000 fpm cabin rate of change, the maximum permissible rate of change is 1500 fpm. This equates to a pressure change of 0.16 psi.

As all electronic outflow valves have two power sources and many systems have two outflow valves, aircraft fitted with electronic controllers and electrically powered outflow valves do not require dump valves.

CABIN PRESSURE INDICATOR

Diagram 13.25 Residual Pressure Indicators

As modern aircraft land pressurised, a residual cabin pressure indicator is located at each cabin door that is opened for access. This flashes red if the cabin pressure is above ambient, and the doors are placed in manual prior to opening. In the event of this warning, the cabin crew are instructed not to open the doors but to inform the captain.

All external doors and hatches that give access into the pressure hull must have provision to prevent the initiation of pressurisation of the aeroplane to an unsafe level if the door is not fully closed and locked.

LEAKAGE AND SEALING

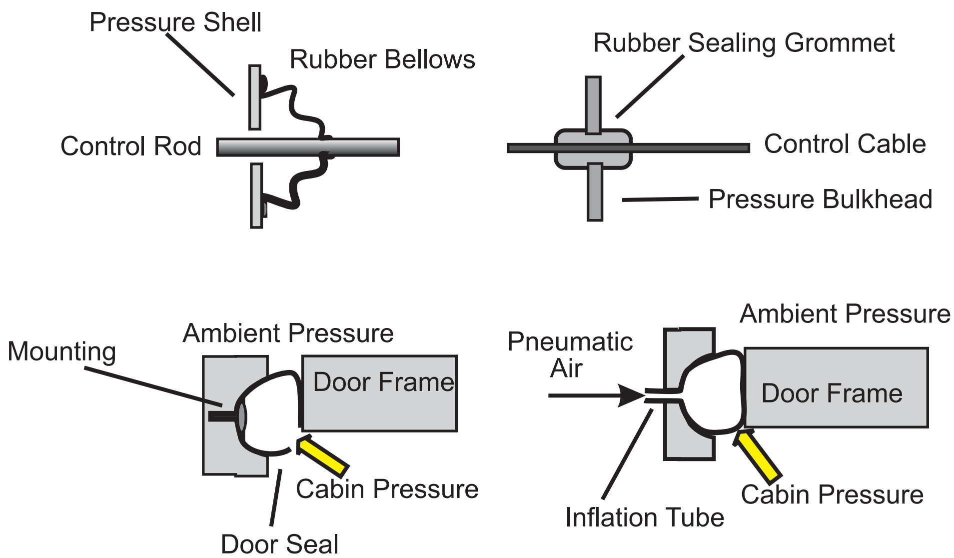

Diagram 13.26 Sealing

All aircraft lose cabin pressure through leakage. Designers try to keep this to a minimum by fitting seals, as the air that is lost has to be supplied by the engines. Diagram 13.26 shows some of the sealing methods used. For control rods, rubber **gaiters**, or bellows can be fitted to the structure and clamped to the control rod. This allows the rod to move back and forth without loss of air pressure. Cables can be sealed by passing them through a rubber grommet and greasing the length of cable that moves backward and forward through the grommet.

Doors and hatches in the pressure hull have to be sealed. An older system used an inflatable tube like an oversized bicycle inner tube, which was attached to the door structure and inflated when the cabin doors were closed. A more rigid rubber seal, that is mounted on the door, has largely replaced this system. While this seal is tubular, it has a series of small holes that face into the cabin, allowing cabin air pressure to enter the seal tube. This pushes the seal against the doorframe as the lower external pressure tries to extrude the seal past the door. Failure of these seals makes a continuous loud noise but should not affect the cabin pressure.

DECOMPRESSION

Loss of cabin pressure is termed **decompression**. There are three levels of decompression:

- Explosive
- Rapid
- Normal

Decompression rates are those based on the maximum ROC permitted.

EXPLOSIVE DECOMPRESSION

An explosive decompression would be due to a catastrophic failure of the aircraft's structure, such as the loss of a door or large hatch. In an explosive decompression, the cabin pressure would equalise with the ambient between 0 and 4 seconds. At the top end, lung and ear damage would occur as the cabin decompresses faster than the lungs can decompress. At the lower end of the time scale, the aircraft would suffer severe, if not catastrophic, structural damage.

RAPID DECOMPRESSION

A rapid decompression occurs when the cabin pressure decreases to ambient in a period of 5 to 7 seconds. During a rapid decompression, passengers and crew may: be aware of noise as the air escapes; feel a breeze, as the air rushes past them to ambient; be aware of dust and dirt particles suspended in the air; notice a momentary fog as the moisture vapour in the cabin expands as the pressure drops; and feel dazed.

As the pressure drops, air and gases within the body expand and rush to atmosphere. Normally, air rushes from the mouth and nasal passages, allowing the lungs and middle ear to equalise. As the lungs can decompress faster than the cabin's rate of decompression, there is no likelihood of damage to the lung tissue.

The main danger is hypoxia. Unless rapid utilisation of the aircraft's supplementary oxygen system is made, unconsciousness occurs. This is hastened due to the decompression causing the lungs to vent and can lead to a reduction of effective consciousness by between 1/2 and 1/3 the normal time.

If the flight crew believe that they are in danger of a decompression (cracked windscreen, etc.), they must place themselves on oxygen, initiate a let-down, and raise the cabin altitude to minimise the differential to reduce the effect of any subsequent decompression.

If an aircraft suffers decompression at high altitude, the maximum rate of descent that the crew can ever initiate is V_d or dive velocity.

CABIN ALTITUDE IN THE EVENT OF A PRESSURE FAILURE

Aircraft that are certified to operate at altitudes above 25 000 ft must be able to maintain a cabin pressure altitude of not more than 15 000 ft in the event of any probable failure or malfunction in the pressurisation system.

In the event of cabin pressure failure, as the cabin altitude rises, the following warnings and actions occur:

- 10 000 ft above msl, an audible and red visual flight deck warning occurs
- 13 000 ft, outflow valves drive shut automatically
- 14 000 ft, passenger oxygen masks deploy automatically to the half-hung position

The 10 000 ft audible and visual warning occurs to alert the crew of possible problems, so that they have time to correct where possible, to minimise passenger discomfort and possibly to prevent passenger oxygen masks from dropping.

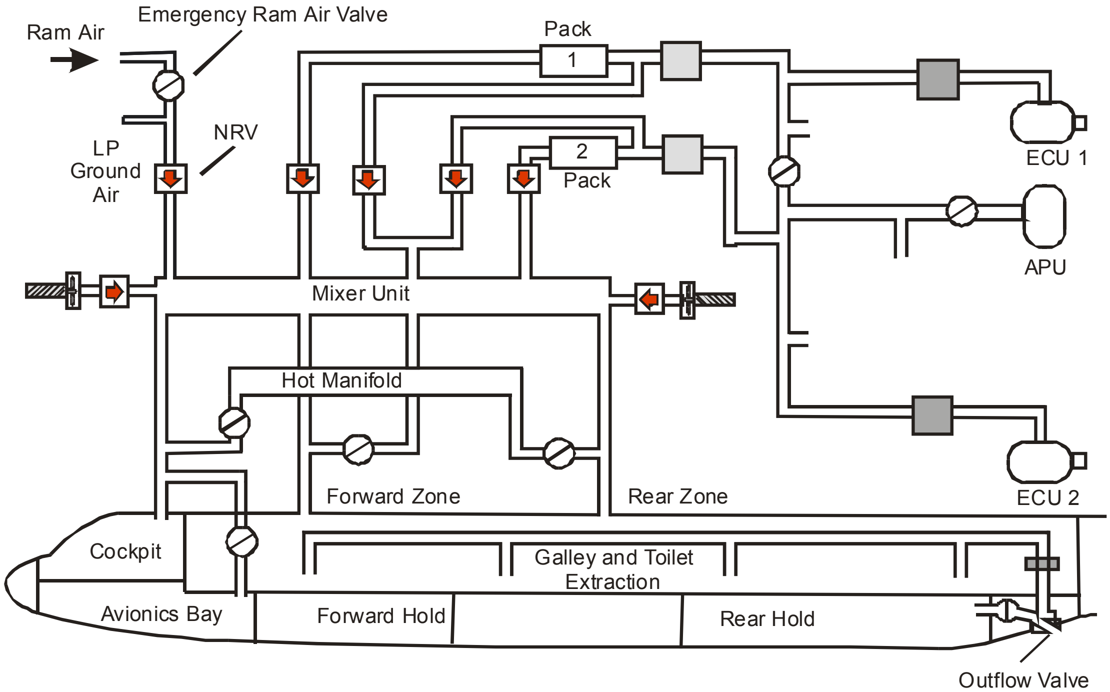

Diagram 13.27 Emergency Ram Air

If the problem cannot be solved, the pressure controller signals the outflow valves to close to minimise the loss of cabin pressure. If both inflows are lost, or the outflow is greater than the maximum inflow, the crew can open the emergency ram air valve. This allows ram ambient air into cabin via the cabin air conditioning distribution ducts.

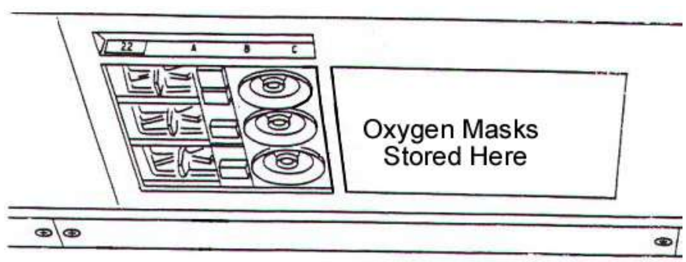

Diagram 13.28 Passengers Overhead Service Unit

If the cabin altitude reaches 14 000 ft, the passenger oxygen masks that are stored in the passengers' overhead service unit (PSU), see diagram 13.28, are deployed to the half-hung position by a baro-static controller. This is done at this altitude to ensure that supplementary oxygen is available before the cabin reaches 15 000 ft.

GRAPHICAL REPRESENTATION OF CABIN DIFFERENTIAL

Depending on the originator of the information, some graphical representations of cabin altitude versus aircraft altitude use a logarithmic scale, see diagram 13.29 for details. Care must be taken when determining the cabin altitude when interpolating a differential that is found between the given guidelines.

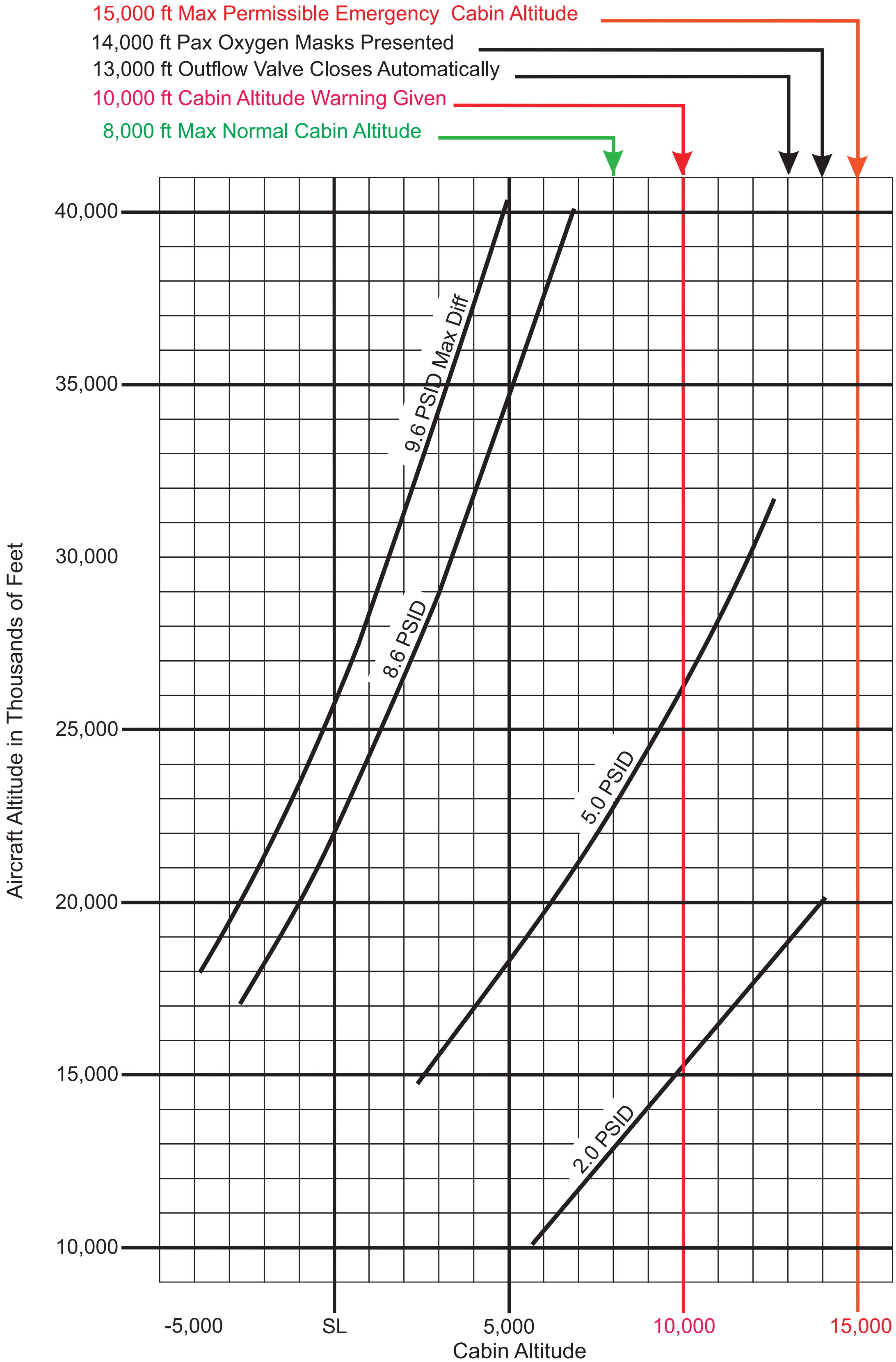

Diagram 13.29 Graphical Representation of Cabin Differential

OXYGEN

This and the previous chapter have outlined the regulations and the basis of how aircraft are ventilated and pressurised to maintain an adequate supply of oxygen to both passengers and crew. This section examines the JAA regulations for the provision of passenger and crew supplementary oxygen for both pressurised and un-pressurised aircraft, first aid oxygen, and the methods by which the regulations are achieved.

HYPOXIA

Hypoxia is a lack of oxygen in the body. Its onset can be insidious as it affects the decision making process to a point where the person affected cannot make rational decisions or even remember his name, and it can lead to un-consciousness and finally death if not reversed.

OXYGEN SATURATION

Altitude	Useful Consciousness	Physiological and Psychological Effect	Saturation
Sea level		Normal body and mental conditions	100%
0 – 7000 ft		Normal body and mental conditions	100–95%
10 000 ft		Long exposure results in fatigue and headaches	90%
15 000 ft		Drowsiness, blue lips and fingernails, impaired vision, increased respiration and pulse rates. Certain personality changes, over-confidence, and lack of judgement	81%
18 000 ft	30 minutes		
22 000 ft		Loss of coordination, double vision, convulsions, danger of collapse	68%
25 000 ft	2-3 minutes	Loss of consciousness followed by death	55-50%
30 000 ft	45 - 75 seconds	Maximum of 75 seconds before unconsciousness and death	
33 000 ft		Breathing 100% oxygen at ambient pressure has the effect of restoring the body's physiological and psychological to sea level conditions	
35 000 ft	30 - 60 seconds	Maximum of 1 minute before lapsing into unconsciousness followed by death	
40 000 ft		The maximum altitude for breathing 100% oxygen at ambient pressure, above this altitude positive pressure breathing is required	
45 000 ft	Approx 12 seconds	Maximum of 12 seconds before unconsciousness and death	
65 000 ft		Without cabin pressurisation or a full pressure suit (space suit) the body fluids boil	

Diagram 13.30 Effects of Altitude and Oxygen

As oxygen is effectively 21% of the atmospheric gases and the ambient pressure is 14.69 psi at sea level, the pressure exerted by the oxygen alone is 2.94 psi (normally rounded to 3 psi). This is termed a **partial pressure**. The partial pressure of oxygen in the lungs forces the oxygen across the lung tissue into the blood stream. At sea level, blood has an oxygen saturation of 100%. The level of saturation decreases with an increase in altitude. This is due to the decrease in barometric pressure.

The hypoxia effects shown in Diagram 13.30 are exacerbated by length of exposure, altitude at which the aircraft is flying, and the rapidity of the cabin's rate of climb to that altitude.

NUMBER OF PASSENGER MASKS AND DISTRIBUTION

JAR-OPS 1 states that as of November 9, 1998, aircraft that operate above 25 000 ft, or those that operate at or below 25 000 ft but cannot descend safely within 4 minutes to 13 000 ft, must be fitted with automatically deployable oxygen equipment immediately available to each occupant, wherever seated. The total number of masks must exceed the number of seats by at least 10%. The extra masks are to be evenly distributed throughout the cabin.

These extra masks are to enable cabin crew or passengers who are away from their seats to gain immediate access to oxygen. As crew or passengers might be in the aircraft lavatories when oxygen is required, each aircraft lavatory must have two face masks.

For aircraft to fly above 30 000 ft, these masks must be able to be automatically deployed before the cabin altitude exceeds 15 000 ft, and the crew must be provided with a manual means to release the masks in the event of failure of the automatic system. Indication of the operation of the automatic presentation system should be provided on the flight deck.

OXYGEN

In aviation, there are three physical states for the transportation of oxygen:

- Liquid
- Gaseous
- Chemical

Liquid oxygen is not used in the civil aviation industry, as it is very expensive and poses handling, storage, and safety problems. Flight crew are always supplied with gaseous oxygen, as this is the most economic and effective way to meet the regulations. Depending on design, some air transport aircraft have gaseous supplementary oxygen systems for the passengers. However, it is more common to find that passenger oxygen is produced by chemical oxygen generators as these are cheaper to produce, have a five-year shelf life, and require no servicing as they are replaced, not serviced.

GASEOUS OXYGEN SYSTEMS

There are two gaseous oxygen systems in current use:

- The continuous flow system
- The diluter demand system

The continuous flow system is normally used in light un-pressurised aircraft intending to fly above 10 000 ft, or as the passenger supplementary oxygen system for some pressurised aircraft. The diluter demand system, a more sophisticated and more expensive system, is used for flight crew of air transport aircraft.

CONTINUOUS FLOW SYSTEM

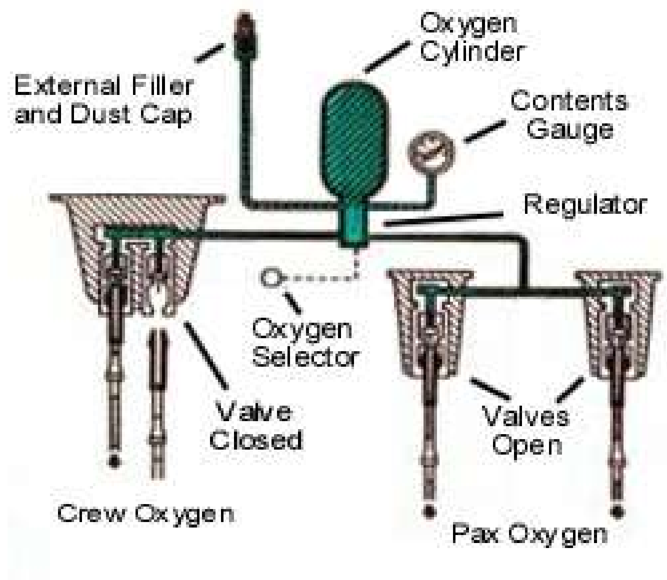

Diagram 13.31 Simple Continuous Flow System

In these systems, the oxygen is stored in a pressurised cylinder with a one-stage regulator that drops the pressure from 1800 psi to 10 psi.

Selected low-pressure oxygen is fed to all mask outlet connectors. When a mask is plugged into the connector, a continuous flow of oxygen enters the mask regardless of whether the user is inhaling or exhaling. An outward relief valve fitted in the mask allows the user to breathe out. Breathing out using these systems requires effort. When not in use, the masks are unplugged and stored.

CONTINUOUS FLOW PASSENGER SUPPLEMENTARY OXYGEN SYSTEM

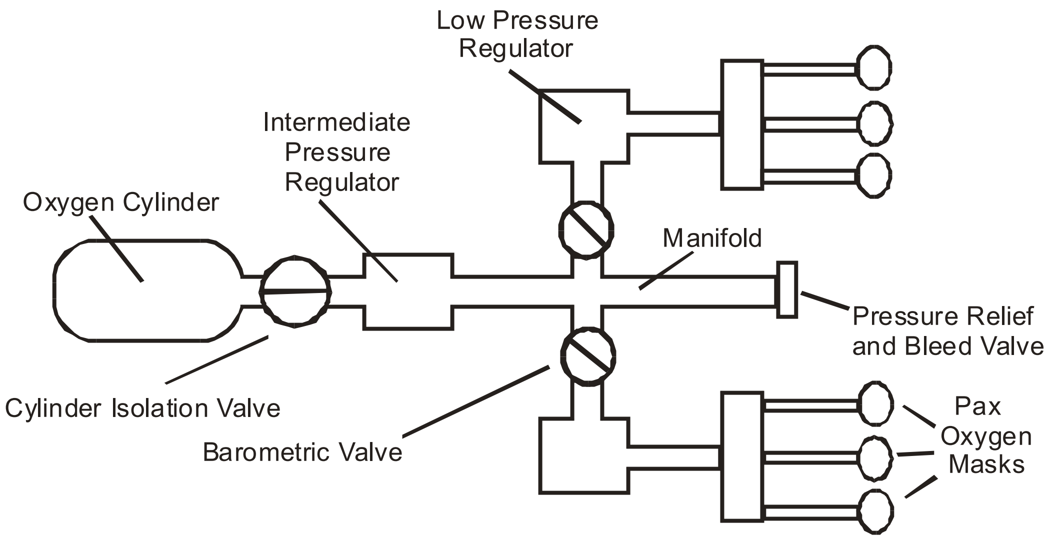

Diagram 13.32 Continuous Flow Passenger Supplementary Oxygen System

In these systems, gaseous oxygen stored in a cylinder at 1800 psi is passed through an intermediate pressure regulator, where the pressure is dropped to between 80–100 psi. It is then fed into a ring main or manifold. A barometric valve prevents the oxygen from flowing to the passenger masks.

When the cabin altitude exceeds 14 000 ft, the barometric valve opens and allows oxygen to pass into the low-pressure regulator. At the same time, pneumatic pressure opens a latch allowing the PSU door to open and deploy the masks in the half-hung condition. The act of pulling the face mask down opens the valve to the mask, allowing a continuous flow of oxygen into a one-size-fits-all rubber cup that covers the mouth and nose and has an elasticised head strap.

The manifold is kept pressurised to ensure that passengers at the far end of the manifold receive oxygen at the same instant as the passengers closest to the supply and to prevent the ingress of moisture. There is always a slight bleed off from the manifold. This loses oxygen, for which the operator has had to pay. This, coupled with the extra weight of storage cylinders, pipe work, and regulators, as well as the cost of their servicing, has resulted in operators specifying chemical oxygen generators for passenger oxygen in most modern aircraft purchases.

FLIGHT CREW'S DILUTER DEMAND SYSTEM

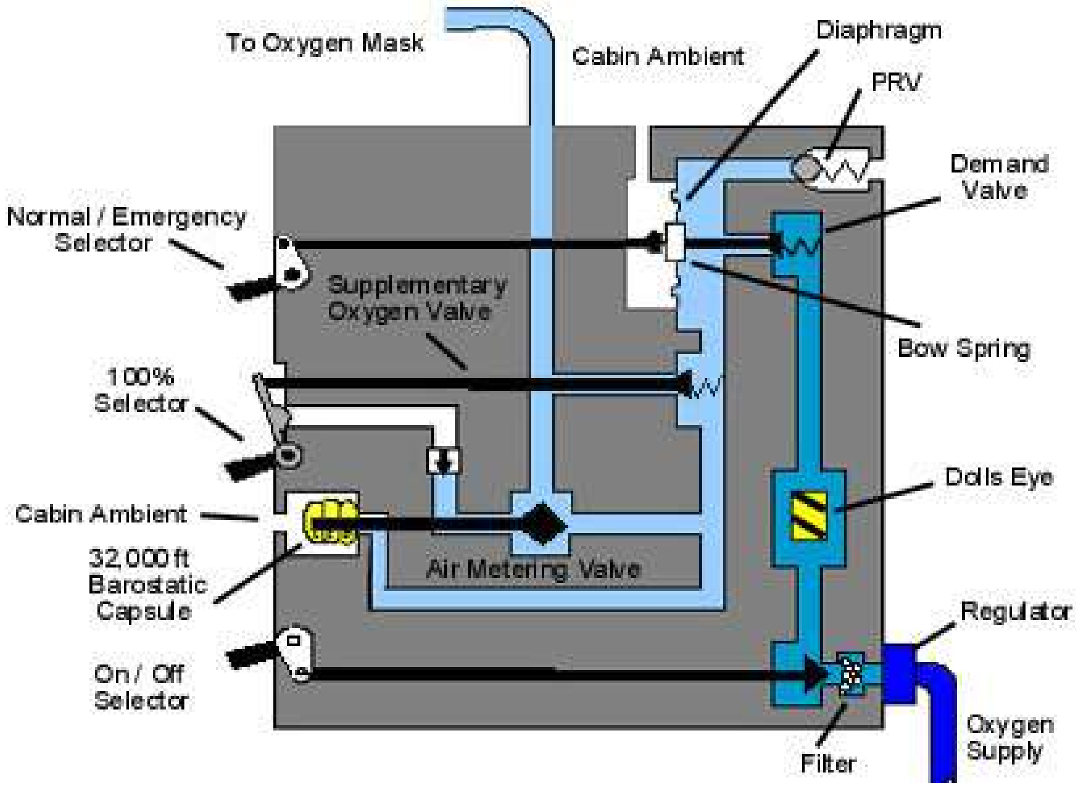

Diagram 13.33 Diluter Demand System

Diagram 13.33 is a simplified schematic of a diluter demand system as fitted to air transport aircraft. Oxygen from the cylinders is supplied to the diluter demand unit at high pressure. This is regulated to 400 psi and passes through an inline filter unit to an on/off valve.

From the on/off valve, the oxygen flows past a flow indicator or **doll's eye**. This gives the crew a visual indication that oxygen is flowing when the unit is in the diluter mode, as oxygen is odourless and tasteless.

Low-pressure regulation is controlled by the demand valve, diaphragm, and bow spring. When "normal" is selected, and the pilot is not breathing in or in the process of exhaling, the demand valve is closed by spring and oxygen pressure. As the pilot inhales, a partial vacuum is formed in the mask and is felt by the diaphragm.

The other side of the diaphragm is subjected to cabin ambient. The differential created biases the diaphragm across to the right. The bow spring, oxygen pressure acting on the demand valve, and the demand valve's spring limit its movement. This allows oxygen to flow past the demand valve. Pressure is maintained at approx 8 psi as the pilot inhales. As the pilot's rate of inhalation slows, the pressure builds up against the diaphragm and acts to close the demand valve. This is the demand action of the regulator. In this system, the pilot has to make the effort during the inhalation rather than the exhalation.

In certain conditions, the flight crew can require a supplementary supply of oxygen when the cabin altitude is below 33 000 ft. Therefore, to supply 100% oxygen on demand would be wasteful. To overcome this, a port allows cabin ambient air to enter the regulator via a non-return valve, so that as the pilot inhales, a mixture of predominantly cabin air and oxygen is supplied to the mask.

In slow decompression, as the cabin altitude rises, the baro-static valve, which is set for 32 000 ft, starts to close the air-metering valve, reducing the ratio of cabin air to oxygen. At 32 000 ft, the air-metering valve is fully closed, ensuring that the pilot is on 100% oxygen before 33 000 ft. A further baro-static capsule (not shown) increases the flow rate to ensure that the pilot is on pressure breathing (continuous flow) at 40 000 ft.

At normal cabin altitudes, if the ventilation air is contaminated with smoke and fumes, using the mask with the regulator in the “normal” mode would subject the pilot to smoke inhalation via the air-metering valve. To overcome this, the pilot selects 100% oxygen. This closes the cabin air supply port and opens a supplementary oxygen valve, which makes up the volume as the air-metering valve is still in the position of reducing the oxygen flow. The regulator still functions as a demand system.

Pilots can select emergency oxygen in the event of thick smoke, high cabin altitude, or stress. When emergency is selected, the diaphragm is biased to the right (refer to diagram 13.33) unseating the demand valve, allowing a constant flow of oxygen at a pressure higher than the demand supply (nominally 12 psi) to enter the mask. This gives a positive pressure within the face mask, which ensures that smoke is kept out and increases the oxygen saturation of the pilot’s blood.

There is one further position of the normal/emergency selector. This is “mask test”. By holding the selector in this position, the demand valve is opened even further, allowing a much higher pressure to enter the system. This is used during pre-flight to check and purge the mask line. No attempt should be made to wear the mask or use this pressure for breathing as it can damage the lungs. Any over pressure in the demand regulator causes the pressure relief valve to open.

PASSENGER CHEMICAL OXYGEN GENERATORS

Passenger chemical oxygen generators and the passenger face masks are stored above the passengers’ heads in the PSU (see diagram 13.34). The generators work on the principle that some mono fuels, when ignited, produce more oxygen than the combustion process requires, and that this oxygen can be utilised.

Diagram 13.34 Passenger Oxygen Mask and Generator Stowage

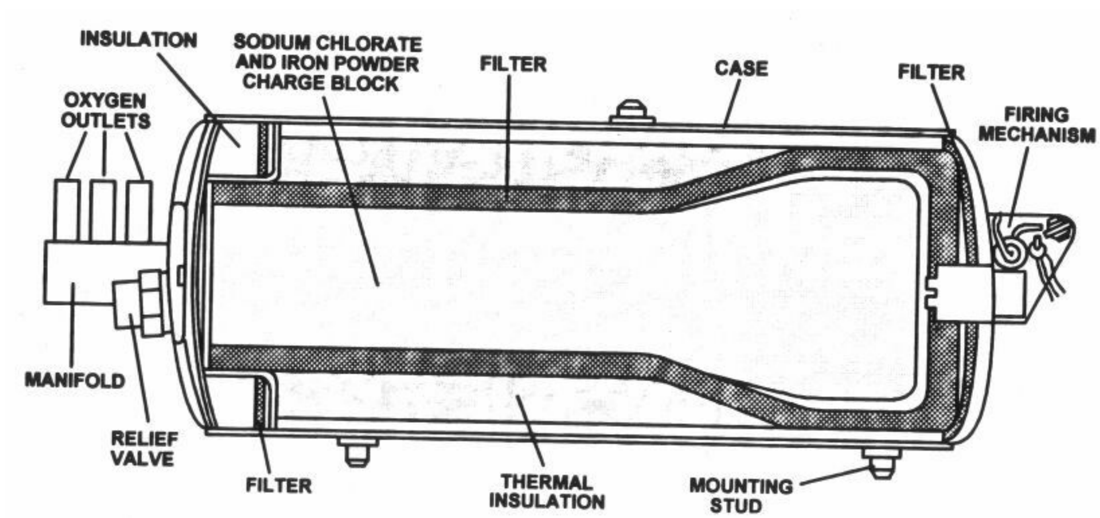

Diagram 13.35 Cross Section of a Chemical Oxygen Generator

Diagram 13.25 shows a cross section of a typical aviation chemical oxygen generator. Inside a cylindrical case is a charge block or candle. This is the mono fuel. It consists of a mixture of sodium chlorate and iron powder (not iron filings). This is surrounded by a filter material and thermal insulation.

When ignited, the charge block smoulders at a temperature of 400°F and burns for 15 minutes minimum, releasing about 45% of its weight as usable oxygen. Cooling of the oxygen takes place in the supply pipe and reservoir bag, which is located just before the face mask. The oxygen must be no more than 10°C above the ambient temperature when it enters the face mask.

Diagram 13.36 Dräger Chemical Oxygen Generators

Each unit incorporates a pressure relief valve to prevent excessive internal pressure. Once ignited, the unit continues to smoulder until the entire charge block has been consumed. To indicate if the unit has been used, a band of heat sensitive paint is painted around the cylinder with adjacent band(s) of black paint to aid identification (see diagram 13.36).

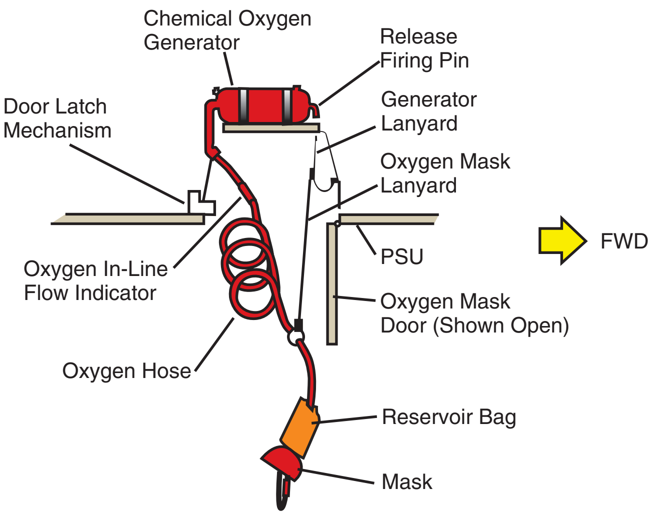

Diagram 13.37 Half-Hung Position

For chemical oxygen generators, the door latches of the PSUs are released by a 28-volt DC electrical signal from a baro-static unit. In the event of an incipient decompression, the pilot can release the latches by the operation of a guarded switch on the flight deck. For generators that supply more than one mask, the first passenger to pull a mask down from the half-hung position initiates activation.

In the older units, as the mask was pulled, a lanyard attached to the supply pipe released a spring-loaded striker on to a fulminate, see diagram 13.37. Modern systems use a 28-volt DC squib to ignite the charge block. This is still activated by the act of pulling the mask downward.

PAX MASKS AND SMOKE

The passenger face masks are designed to allow ambient cabin air into the mask as the wearer inhales. This air is supplemented by the continuous flow from the oxygen generator or the gaseous system. This precludes their use in smoke-filled atmospheres as the oxygen enrichment results in the wearer's trachea opening, allowing deeper smoke inhalation.

CABIN CREW

Cabin crew positions are provided with the same supplementary oxygen system as the passengers. The additional 10% of passenger masks ensure that cabin crew who are away from their seats can access oxygen regardless of their location at the time of cabin pressurisation failure.

Where a cabin crewmember's work area is not within easy reach of the supplementary oxygen mask provided at the seat station, additional mask units should be provided at workstations such as galleys.

PROTECTIVE BREATHING EQUIPMENT

Protective breathing equipment, or PBE, is the term given to equipment that is for protection against smoke, fumes, and other harmful gases, by protecting the eyes, nose, and mouth and provides oxygen for a period of not less than 15 minutes.

PBEs intended for flight crew use must be conveniently located on the flight deck and be easily reached by the crewmembers when seated in their normal positions. Cabin crew PBEs are located at their normal crew duty stations.

Additional PBE must be located at or adjacent to cabin mounted hand fire extinguishers. For B class compartments, where the hand held fire extinguisher is located inside the compartment door, PBE must be located adjacent to the entrance but stowed outside the door. PBE can take two forms, either a portable gaseous cylinder with a full face mask or a smoke hood with a chemical oxygen generator.

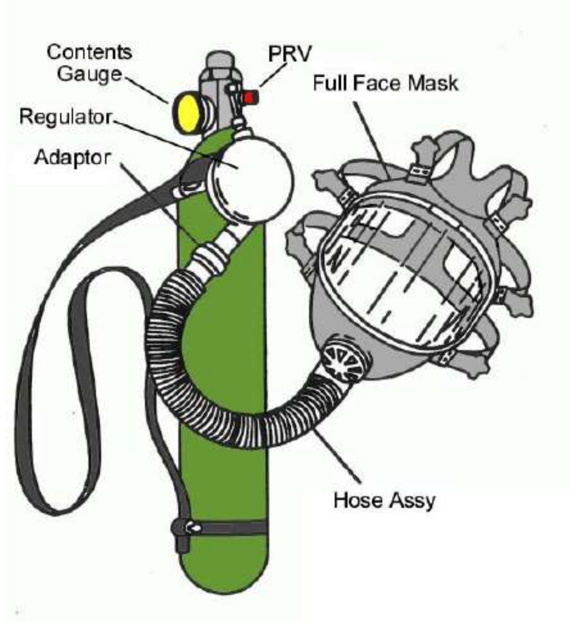

Diagram 13.38 Portable Gaseous Oxygen Walk Round Kit

Diagram 13.38 shows an American-style single cylinder gaseous oxygen portable PBE or walk round kit. These typically hold 120 litres of oxygen at 1800 psi and have either two or three rates of continuous supply. These are:

- Normal 2 litres per minute giving 60 minutes duration
- High 4 litres per minute giving 30 minutes duration
- Emergency 10 litres per minute giving 12 minutes duration

Each cylinder has a pressure gauge, which can be calibrated either in pressure or volume by being divided into eighths. The first eighth is coloured red to indicate that the pressure is very low. Each cylinder has an on/off selector and flow rate selector. Some cylinders have the emergency flow rate, although this extra rate is not mandatory. The full-face plate protects the eyes, nose, and mouth from smoke and fumes, and the strap allows the user to carry the cylinder and have both hands free.

Each cylinder must be equipped with a pressure relief valve to prevent the cylinder bursting in the event that a fire prevents its removal from its storage position. The release of oxygen in a fire situation aids combustion. The rate of relief is set so that the internal pressure is prevented from rising above the cylinder's maximum structural pressure, while preventing excessive oxygen enrichment of the cabin atmosphere.

Diagram 13.39 Dräger Oxycrew Smoke Hood

Diagram 13.39 shows a Dräger smoke hood. In the event of a fire, these not only protect the crew's eyes, mouths, and noses from smoke but also their heads from molten plastic, while allowing them free use of their hands. As far as JAR FCL questioners are concerned, these smoke hoods contain a chemical oxygen generator that is initiated by pulling a red toggled cord downward. This then provides the minimum of 15 minutes of oxygen required.

In reality, modern smoke hoods like the one illustrated above are chemically treated on the inside and activate via the wearer's body heat and exhalation, providing a minimum of 20 minutes operation.

FLIGHT CREW MASKS

Flight crew masks must be located within immediate reach of crewmembers while seated at their duty stations. For pressurised aircraft that are certified to operate above 25 000 ft, the flight crew oxygen masks must be of the **quick donning type**. That is they must be able to be fitted one handed, fully functional within 5 seconds, and leave both hands free. They must be able to be put on over spectacles without disturbing them. These masks must not interfere with immediate communication between the flight crewmembers and other crewmembers over the aeroplane intercommunication system or inhibit radio communications.

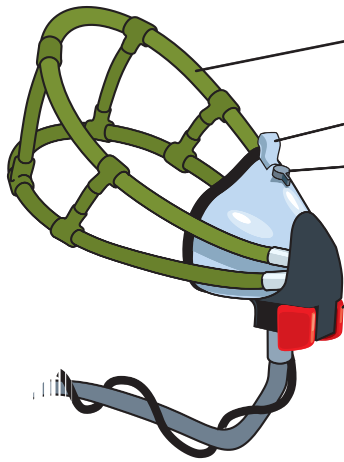

Diagram 13.40 Flight Crew Face Mask

Diagram 13.40 shows a typical face mask. These require smoke goggles to turn them into the flight crew's duty station PBE. The face mask has inflatable head straps, which are inflated for fitment by squeezing two toggles, one on each side of the mask, together. When they are released, the harness deflates and holds the mask onto the face.

UN-PRESSURISED AIRCRAFT

An operator shall not operate a non-pressurised aeroplane at altitudes above 10 000 ft unless supplemental oxygen equipment capable of storing and dispensing the oxygen supplies required is provided.

When un-pressurised aircraft are operated above 10 000 ft but below 13 000 ft for more than 30 minutes, or whenever the aircraft exceeds 13 000 ft, all flight crew on duty must use supplemental oxygen continuously. The commander is responsible for ensuring that this occurs.

As of April 1, 2000, an operator of an un-pressurised aircraft with an MTOM exceeding 5700 kg or 20 or more passenger seats must ensure that each flight crewmember has a 15-minute (min) supply of supplementary oxygen and has equipment to protect the eyes, nose, and mouth. In addition, when the flight crew is more than one and a cabin crewmember is not carried, portable PBE must be carried to protect one member of the flight crew. Refer to Appendix 2 for JAR – OPS 1 requirements.

FIRST AID OXYGEN

First aid oxygen is required to be carried on pressurised aircraft that operate above 25 000 ft and have cabin crew. First aid oxygen is an undiluted supply for passengers who, for physiological reasons, might require oxygen following a cabin depressurisation. This can be delivered from a portable container(s) or from fixed first aid outlets from a ring main system.

The minimum number of outlets is two, and each outlet should have a normal minimum mass flow rate of 4 litres of oxygen per minute. However, there may be a means to decrease this flow to not less than 2 litres per minute at any cabin altitude.

GASEOUS OXYGEN SYSTEM COMPONENTS

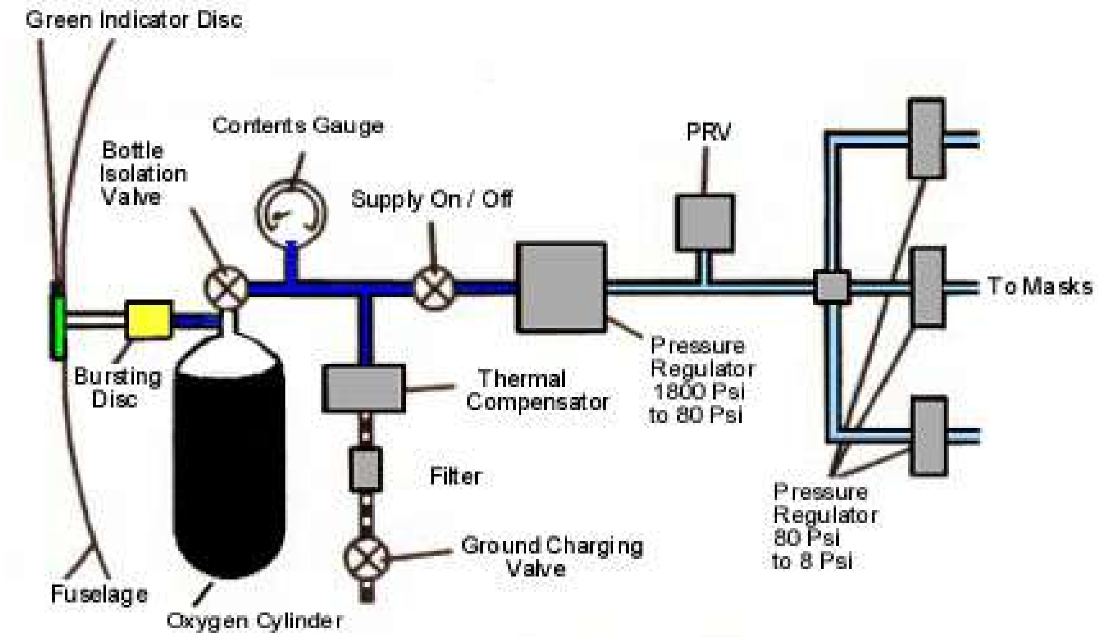

Diagram 13.41 Basic Oxygen System

As gaseous oxygen is stored under pressure (normally 1800 psi) in strong steel alloy cylinders with hemispherical ends, the cylindrical section of some of these storage cylinders are wire bound to increase resistance to fragmentation in the event of a catastrophic failure of the cylinder. To denote that they contain oxygen, they are colour coded. Those of American origin are painted green. Those of UK origin are painted black with a white neck and shoulder. Refer to diagram 13.41 and previous diagram 13.38.

Cylinders are also marked with “use no oil or grease” in red paint and “aviation oxygen” in white paint and have their manufacture and test dates stamped around the neck. Each main storage cylinder has an isolation valve attached to the neck of the cylinder, which is only closed by ground engineers. Otherwise, they are secured open.

Refer to diagram 13.41. As there is a risk that these cylinders could explode if subjected to excessive temperature, each cylinder is connected to an external vent via a bursting disc.

If the oxygen pressure due to thermal heating exceeds the bursting disc’s design maximum pressure, it ruptures and allows the entire contents of the cylinder to vent to atmosphere. To indicate that this has occurred, a green plastic indicator disc is blown out of the vent pipe’s skin fitting, leaving a red bowl visible.

Rather than have a single large cylinder store all the gaseous oxygen required, several smaller cylinders are normally used. Each is connected to a main manifold via a non-return valve, with each cylinder having its own bursting disc system, so that in the event of any single failure, not all the oxygen is lost overboard.

An oxygen contents gauge is mounted on the flight deck. Older systems used a direct reading gauge. Modern systems use pressure transducers and either display the content on a dedicated gauge or as part of the systems information EFIS screen. This information can be displayed as a pressure or as volume content by dividing volume into eighths. The first eighth is coloured red to indicate that the pressure is very low. Due to the physics of gasses, while 1800 psi equates to full, 900 psi does not equate to half full but less than half full.

A supply on/off selector is fitted downstream of the gauge take-off point, so actual system contents can be observed without allowing supply to the regulator. The flight crew uses this selector to isolate the regulators from the storage cylinders.

Pressure regulation can be carried out in one combined unit, or there can be several stages of regulation. The supply pressure of 1800 psi is dropped to the intermediate pressure of 80–100 psi with the final pressure drop of 8-10 psi made at the individual crewmember's regulator. To protect the crew from the supply pressure in the event of the intermediate pressure regulator failing, a pressure relief valve is fitted in the intermediate pressure line.

REPLENISHING AIRCRAFT OXYGEN SYSTEMS

Replenishing or charging of any aircraft gaseous system must be undertaken with care and caution. This is especially so with aircraft oxygen systems. The rate of charging must be slow, allowing the oxygen pressure in the aircraft's storage cylinders to build up slowly. If a fast rate of charge is used, the actual flow and the change of pressure heat the gas and the cylinder, giving rise to false full reading.

As gas temperature varies, the actual volume that cylinders can hold alters. Many aircraft have a thermometer and pressure versus volume graph located adjacent to the aircraft's oxygen charging point. (It is assumed that the ground supply cylinders and the contents are at ambient temperature when the aircraft is being charged). This enables accurate content reading.

THERMAL COMPENSATOR

To even out any temperature fluctuations in the oxygen while charging the system, a thermal compensator can be fitted in the charging line downstream of an inline filter, see Diagram 13.41. This unit consists of a tube with numerous fine copper wires (like hair) radiating inward from the body of the tube. These quickly heat up and equally quickly dissipate their heat, thus evening out the temperature of the oxygen as it flows past.

OXYGEN BACTERIAL CONTAMINATION

Each oxygen charging point has a captive filler blank. This must be fitted after each replenishment to prevent the ingress of dirt and moisture. Apart from the danger of ice crystals forming in the oxygen system, if moisture is allowed to enter the oxygen system, bacteria can grow and multiply. This gives the affected oxygen the smell and taste of rotten eggs. To prevent this, all charging hoses must be purged before use, and all cylinders must maintain a positive oxygen pressure to prevent the ingress of moisture.

FIRE AND EXPLOSION

Oxygen itself does not burn until extremely high temperatures are achieved, but it promotes combustion. Explosions are a very rapid growth in combustion. The effect that oxygen has is proportionate to the ratio of oxygen and its pressure. At high concentrations and pressures, it makes materials that are not normally considered combustible burn.

Of a particular hazard are petroleum-based oils and greases. They absorb pure oxygen at such a high rate, that at a molecular level, they vibrate and produce such heat that they spontaneously ignite.

LUBRICATION OF OXYGEN COMPONENTS

The only mediums that can be used to lubricate oxygen system components and threads are graphite, carbon, or Teflon-based materials.

Appendix 1

Appendix 1 To JAR-OPS 1.770-Oxygen-Minimum Requirements for Supplemental Oxygen for Pressurised Aeroplanes	
Supply for	Duration and Cabin Altitude
All occupants of flight deck seats on flight deck duty	Entire flight time when the cabin pressure altitude exceeds 13 000 ft and entire flight time when the cabin pressure altitude exceeds 10 000 ft but does not exceed 13 000 ft after the first 30 minutes at those altitudes, but in no case less than: 30 minutes for aeroplanes certificated to fly at altitudes not exceeding 25 000 ft (Note 2) 2 hours for aeroplanes certificated to fly at altitudes more than 25 000 ft (Note 3)
All required cabin crew members	Entire flight time when cabin pressure altitude exceeds 13 000 ft but not less than 30 minutes (Note 2), and entire flight time when cabin pressure altitude is greater than 10 000 ft but does not exceed 13 000 ft after the first 30 minutes at these altitudes.
100% of passengers (Note 5)	Entire flight time when the cabin pressure altitude exceeds 15 000 ft but in no case less than 10 minutes. (Note 4)
30% of passengers (Note 5)	Entire flight time when the cabin pressure altitude exceeds 14 000 ft but does not exceed 15 000 ft.
10% of passengers (Note 5)	Entire flight time when the cabin pressure altitude exceeds 10 000 ft but does not exceed 14 000 ft after the first 30 minutes at these altitudes.

Note 1: The supply provided must take account of the cabin pressure altitude and descent profile for the routes concerned.

Note 2: The required minimum supply is that quantity of oxygen necessary for a constant rate of descent from the aeroplane's maximum certificated operating altitude to 10 000 ft in 10 minutes and followed by 20 minutes at 10 000 ft.

Note 3: The required minimum supply is that quantity of oxygen necessary for a constant rate of descent from the aeroplane's maximum certificated operating altitude to 10 000 ft in 10 minutes and followed by 110 minutes at 10 000 ft. The oxygen required in JAR-OPS 1.780(a)(1) may be included in determining the supply required.

Note 4: The required minimum supply is that quantity of oxygen necessary for a constant rate of descent from the aeroplane's maximum certificated operating altitude to 15 000 ft in 10 minutes.

Note 5: For the purpose of this table, "passengers" means passengers actually carried and includes infants.

Appendix 2

Appendix 2 To JAR-OPS 1.775-Supplemental Oxygen for Non-Pressurised Aeroplanes	
Supply for	Duration and Cabin Altitude
All occupants of flight deck seats on flight deck duty	Entire flight time at pressure altitudes above 10 000 ft
All required cabin crew members	Entire flight time at pressure altitudes above 13 000 ft and for any period exceeding 30 minutes at pressure altitudes above 10 000 ft but not exceeding 13 000 ft.
100% of passengers (See Note)	Entire flight time at pressure altitudes above 13 000 ft.
10% of passengers (See Note)	Entire flight time after 30 minutes at pressure altitudes greater than 10 000 ft but not exceeding 13 000 ft.

Note: For the purpose of this table, "passengers" means passengers actually carried and includes infants under the age of 2.

ACKNOWLEDGEMENTS

We would like to thank and acknowledge:

For Diagrams and assistance with technical information

14.11	Kidde-Graviner
14.12	Mathisen Way
14.13	Poyle Rd
14.16	Colnbrook
14.26	Slough
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INTRODUCTION

In most cases, the tendency is to visualise leaping flames when talking about fire. It is also necessary to be aware that a smouldering material is on fire and presents an equal hazard if not discovered and extinguished.

Before looking at the methods of fire detection, crew alerting, and extinguishing a fire, it would be advantageous to re-examine the basic chemistry of fire, review the classifications used, and look at the JAA definitions and regulations.

BASIC CHEMISTRY

There are three conditions that must be present for fire to start and take hold:

- Fuel
- Heat
- Oxygen

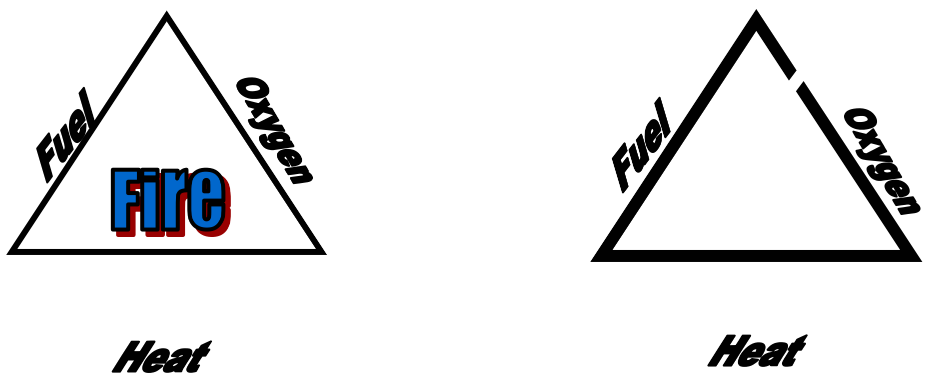


Diagram 14.1 The Fire Triangle

THE FIRE TRIANGLE

When these conditions are present, they are referred to as the fire triangle, as shown in the left-hand picture of diagram 14.1. Any break in the triangle, as shown in the right-hand diagram, results in:

- An existing fire being extinguished
- The prevention of one starting

FUELS

As either liquid or solid materials are heated, they start to give off gases. It is these gases that actually combust. Therefore, the lower the temperature at which a material can be converted into a gas, the more flammable it becomes and vice versa. This willingness of a fuel to change state from the liquid or solid form into its gaseous form is termed **volatility**.

HEAT

The heat (ignition source) that starts a sustained fire must as a minimum be the temperature at which the fuel is readily converted into a gas, ignite it, and continue the conversion process. After the initial ignition, the temperature of the fuel rises, which in turn increases the production of flammable gases, so the fire spreads over the fuel.

OXYGEN

For fire to start and continue, the correct ratio of oxygen is required. The oxygen itself does not burn until extremely high temperatures are reached, but oxygen promotes the combustion of other materials. The exception to this rule is a mono fuel, which produces its own oxygen during the process of combustion.

COMBUSTION PRODUCTS

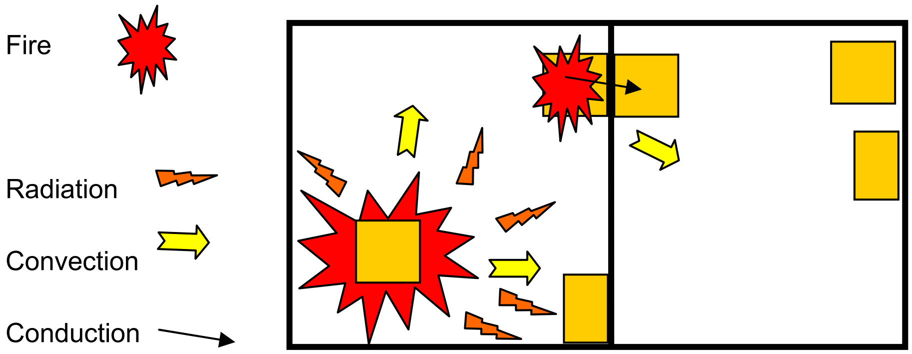
The two principal products of combustion are heat and smoke.

HEAT

The heat output of a fire depends on the surface area of the fuel that is burning, the fuel's calorific content, and the oxygen supply. For example, an oxy-acetylene welding torch increases the intensity of the acetylene flame by adding extra oxygen. However, if too much oxygen is added, the flame can be snuffed out.

Heat is transferable by conduction, convection, and radiation.

Conduction	As felt when touching a hot item
Convection	As seen when warm air is rising from a hot surface
Radiation	The sun's heat is transferred to earth via radiation through the vacuum of space.



Where a fire takes place within a confined space as per the diagram above, the heat from the fire is transferable via conduction, convection, and radiation to any other objects within the room and the surrounding structures, thus raising their temperatures.

If the heat transferred is sufficient to raise these objects to their fire point, they ignite, causing the fire to spread as in the diagram above. A fire that started in the bottom left corner has caused the package in the top right corner to ignite.

Heat is also being transmitted through the compartment's walls into the adjoining compartment. As there are items in an adjoining compartment, this transmission of heat raises their temperature to a point where they can spontaneously ignite and spread the fire.

SMOKE

Smoke is normally taken as the visible vapour created by particles of carbon from the combustion of a fuel. However, the process of heating a fuel before and during combustion results in the production of gases, which can be toxic depending on the type of material that is being heated.

The volume and density of these gases depend on the material and the temperatures involved. Some of these gases are combustible in their own right. Therefore, it is more accurate to think of all products of combustion as smoke whether it is seen or not. The old saying "*There's no smoke without fire"* is a warning to the unwary, but there are combustible gases before flames.

Where fire takes place within a confined space, the action of combustion reduces the oxygen content. The smoke produced reduces the oxygen content further and reduces visibility.

The reduction of oxygen and the body's natural reaction to smoke inhalation affect breathing, vision, and the mental process. Vision is also affected by the density of the vapours and the body's natural reaction to smoke in the eyes. There is also the attendant danger that the material being burnt gives off toxic gases, which could either incapacitate or kill.

CLASSIFICATION OF FIRES

Class A: Fires that involve solid materials, predominantly of an organic kind such as paper, cotton, and wood also form glowing embers

The means of extinguishing these fires is to cool them. The use of water also prevents re-ignition by soaking the fuel.

Class B: Fires that involve liquids or liquefiable solids

The means of extinguishing this fire is foam, dry powder, carbon dioxide, and Halon.

Class C: Fires that involve gases or liquefied gases such as butane, propane, and methane, etc., resulting from spillage or leakage.

The means of extinguishing these fires is to smother them with foam or dry powder and use water to cool any leaking container.

Class D: Fires that involve metals such as aluminium

The means of extinguishing these fires is to smother them with a special dry powder.

Note: For theoretical purposes in the JAA exam, fires that involve magnesium chips or flares would be smothered using sand.

FIRES INVOLVING ELECTRICITY

There is no categorisation of electrical fire as there used to be, as electricity does not burn, but any of the above fire classes may involve an electrical supply that must be isolated before water or foam is used.

DOMESTIC SITUATIONS

Carbon dioxide, dry powder, and Halon 1301 and 1211 can be used on fires that involve electrical equipment before the power supply is turned off. However, if a fire involves electrical equipment, it is safer to isolate it from the supply first where possible.

FIRE PREVENTION BY DESIGN

Obviously, the best way to fight a fire is to prevent it from starting in the first place. Ideally, all the materials that are used in the production of aircraft would be fireproof. In reality, for practical purposes some of the materials used in the manufacture of aircraft are combustible, others are fire resistant. To minimise the potential fire risk, aircraft manufacturers have to comply with regulations laid down in the JAR 23 and 25.

JAR 23 and 25 classify compartments, detail engine fire zones, and detail design requirements to prevent the spread of fire and smoke, so that any fire does not result in the unintentional operation of or inability to operate essential services or equipment.

JAR DEFINITIONS

JAR 1 defines the difference between fireproof, fire resistant, flame resistant, flash resistant, and flammable. These definitions appear below.

FIREPROOF

Fireproof materials, components, and equipment, must be capable of withstanding a flame temperature of 1100°C ±80°C for a period of 15 minutes without any failure that would create a hazard to the aircraft.

FIRE RESISTANT

Fire resistant materials, components, and equipment, must be capable of withstanding a flame temperature of 1100°C ± 80°C, for a period of 5 minutes without any failure that would create a hazard to the aircraft.

FLAME RESISTANT

Flame resistant materials are not susceptible to combustion to the point of propagating a flame, beyond safe limits, after the ignition source is removed.

FLASH RESISTANT

Flash resistant means not susceptible to burning violently when ignited.

FLAMMABLE

Flammable, with respect to a fluid or gas, means susceptible to igniting readily or exploding.

FUSELAGE CARGO COMPARTMENT CLASSIFICATION

Cargo compartment classifications for aircraft certified under JAR 23 and 25 are listed below:

Class A: is one where a crewmember at his/her duty station would easily discover the presence of a fire, and each part of the compartment is easily accessible in flight.

Class B: is one where there is sufficient access in flight for a crewmember to effectively reach any part of the compartment with the contents of a hand-held fire extinguisher.

Where on accessing the compartment, no hazardous quantity of smoke, flame, or extinguishing agent enters any compartment occupied by the crew or passengers.

A smoke or fire detector is fitted to warn the flight crew as to the presence of a fire.

Class C: is a compartment not classed as either A or B.

It requires a fire or smoke detection system that warns the flight crew and a fixed fire-extinguishing system operated by flight crew.

The ventilation rate can be controlled so that the fire-extinguishing agent used can control the fire.

There are means to exclude hazardous quantities of smoke, flame, and extinguishing agent from any compartment occupied by the crew or passengers.

Class D: are compartments that would be completely confined without endangering the safety of the aeroplane or the occupants if a fire were to break out. The volume of the compartment does not exceed 1000 cubic ft (28.3 cubic metres).

There are means to exclude hazardous quantities of smoke or other noxious gases from any compartment occupied by the crew or passengers.

Heat from a fire within a class D compartment must not affect critical parts of the aircraft.

The ventilation and draught are controlled within each compartment, so that any fire likely to occur in the compartment does not progress beyond safe limits.

The ventilation rate is found using the following formula $W = 2000 - V$, where W is the ventilation and leak rate in cubic ft per hour and V is the volume in cubic feet.

For example, for a 500 cubic ft compartment, the ventilation rate of 1500 cubic ft per hour is acceptable.

Class E: classification is used for air cargo aircraft only where the main cabin is turned into freight bay.

They must have a smoke or fire detection system that warns the crew.

The crew must be able to shut off the ventilation airflow to and from the compartment.

There are means to exclude hazardous quantities of smoke or other noxious gases from any compartment occupied by the crew.

The required emergency exits are accessible by the crew under any cargo condition.

LAVATORY COMPARTMENTS

Aircraft lavatory compartments must have an ashtray fitted inside the compartment, adjacent to the door, and a further ashtray fitted outside the compartment adjacent to the entry side of the door, regardless as to whether smoking is permitted on the aircraft. The lavatory entry door must be designed so that it can be opened from the outside without the use of a special tool.

FIRE ZONES

Fire zones are those areas that contain the ingredients for a fire, such as the main engines, auxiliary power units (APUs), and combustion heaters. Main engines can be divided into reciprocating and turbine.

GAS-TURBINE ENGINES

Gas-turbine engines normally divide into three zones, as shown in diagram 14.2. These are:

- Zone 1, the fire zone
- Zone 2, the cold zone
- Zone 3, the hot exhaust zone

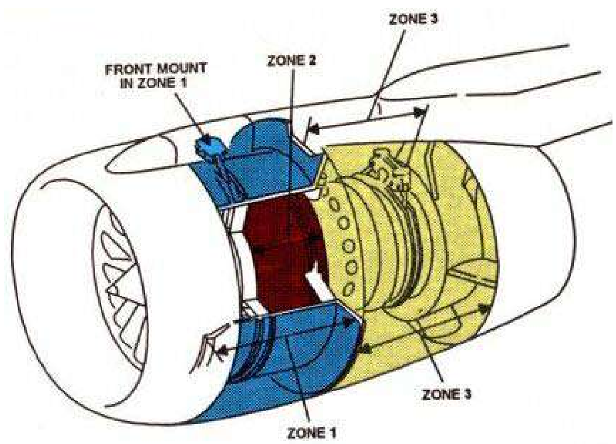


Diagram 14.2 Zones of a Gas-Turbine Engine

ZONE 1

Zone 1 is defined as the compressor and accessory section where there is isolation from the rest of the engine. The accessory section is where the fuel pumps, hydraulic pumps, engine lubricating oil, generators, and gearboxes are located. This zone requires fire detection and protection systems, as a fire here would rapidly get out of control.

ZONE 2

Zone 2 is the area around the compressor spool where there is no fuel or source of ignition. A fire in this zone would be a metal fire due to the compressor blades rubbing against the containment casing. This would be indicated by a severe vibration and engine shutdown removing the source of ignition.

ZONE 3

Zone 3 is the hot zone from the combustion chambers rearward. Normally, this is considered an overheat zone as there is no fuel, but a cracked exhaust pipe could lead to hot gases jetting into the surrounding structure.

However, if either of the following conditions exist, they are included in the zone 1 category.

- The combustor, turbine, and tail section are included in zone 1 if there are any lines or components that carry flammable fluid or gases.
- Any complete power plant compartment in which there is no isolation between the compressor, accessory, combustor, turbine, and tailpipe section.

AUXILIARY POWER UNIT — APU

Diagram 14.3 APU Bay

Auxiliary power units are normally located in the tail of an aircraft, where they are isolated from the rest of the aircraft's structure by a firewall. In this case, the whole of the APU and compartment are classed as a fire zone 1. Refer to diagram 14.3.

COMBUSTION HEATERS

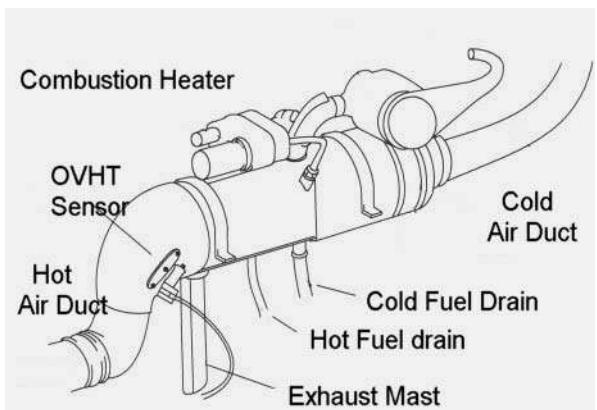


Diagram 14.4 Combustion Heater

Combustion heaters, as per diagram 14.4, can be fitted to light or large aircraft to heat ambient air being supplied to the aircraft's cabin, etc. These systems burn aviation fuel within a chamber if there is any other source of flammable fluid in the region of the combustion heater that could:

- Be damaged by a heater malfunction
- Allow flammable fluid or vapour to reach the heater if there was a leak
- If the heater's fuel system has fittings, should they leak, allow fuel vapour to enter the region around the heater or the ventilation air passage around the heater.

The heater must have a means of being shut down independently of the normal controls in the case of either the combustion heater overheating or the ventilation air overheating.

AIR INTAKES

There must be means to prevent hazardous quantities of fuel entering the engine air intakes due to leakage or overflowing drains, vents, or other components of flammable fluid systems.

Ventilation and combustion air intakes must be located so that no flammable fluid or vapour can enter them under normal operating conditions, or in the event of a malfunction of any other equipment.

DRAINAGE AND VENTILATION OF FIRE ZONES

There must be complete drainage of each part of each fire zone. The drains must be arranged so that they do not create an additional fire hazard. Each fire zone must ventilate to prevent the accumulation of flammable vapour. The ventilation openings must not allow entry of flammable fluids, vapours, or flame from other zones. Each means of ventilation must be arranged so that discharged vapours do not cause an additional fire hazard.

FIREWALLS

Diagram 14.5 Firewall of a Light Aircraft

Firewalls are designed to separate fire zones from the rest of the aircraft's structure. Any opening in a firewall, such as a hole to allow an electric cable, etc., to pass through, must be sealed with a fireproof grommet or fitting.

FLAMMABLE FLUID FIRE PROTECTION

Where storing and movement of flammable fluids occurs via pipes and components outside of a fire zone, the designer must take into account the resulting hazard if the fluid should leak and ignite. One of the methods of containing fluid is to place one pipe inside another, taking into account:

- The flammability of the fluid
- The likely source of a leak
- The likely fluid's path
- The possible ignition sources
- The means of controlling a fire or extinguishing it (i.e. shut off valves, containment, etc.)

ENGINE AND COMBUSTION HEATER FIRE SHUT-OFF VALVES

Each engine installation must have a means to shut off fuel, oil, de-icer, and other flammable fluids from flowing into, through, or within the fire zone. This does not include wet sumps. Refer to Chapter 9, Constant Pressure Pump-Depressurisation/Isolation for isolation of the hydraulic system and Chapter 10, Low Pressure Fuel Cock-LP Cock for the fuel systems fire shut-off valve.

However, when the means of shut-off is activated, it must not interfere with the later emergency operation of other equipment, such as the means for feathering the propeller.

There must be a way to guard against inadvertent operation of the shut-off means and to make it possible for the crew to reopen the shut-off means in flight after it has been closed.

FLAMMABLE FLUID — RESERVOIRS AND PIPES IN A FIRE ZONE

Apart from engine-lubricating oil that is stored in an integral sump, any flammable fluids or gases that are to be stored in a fire zone must be in flameproof containers and pipe work. There must be a means of shutting off the flow from the reservoir, and the whole design should provide the same level of safety as if the reservoir was not there.

Flammable fluid pipelines or components that are located in any fire zone must be fire resistant unless failure of the pipeline would result in spread of fire into other regions of the aircraft, when they must be fireproof. The fuel pipes are covered with a sleeve, often a dull ochre colour in the engine bay of light aircraft.

FLIGHT CONTROLS AND ENGINE MOUNTS WITHIN A FIRE ZONE

Essential flight controls and engine mountings that are located within a fire zone must be manufactured in fireproof materials or shielded to ensure that they are capable of withstanding the effects of fire.

DRY BAYS

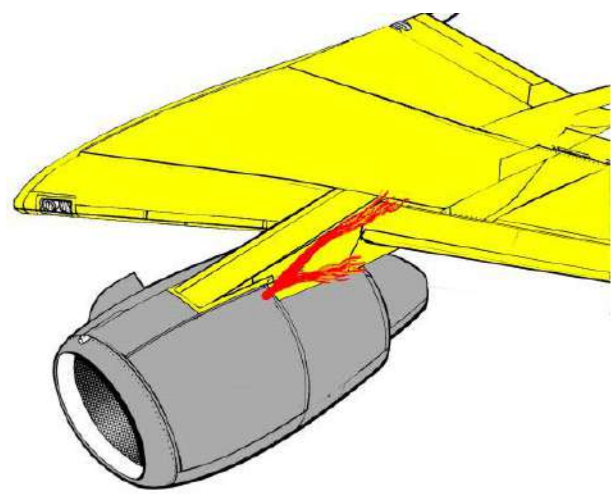


Diagram 14.6 Dry Bays

For aircraft that have engines mounted on pylons below the wings, there is a risk that a fire starting in the engine could travel up through the pylon into the wing via the conduit for the fuel supply pipes and other services. Flames issuing from the engine cowling could also lick over the leading edge of the wing, as diagram 14.6 shows. If this area were part of the fuel tank system, this might result in a catastrophic fire in the wing tanks. To prevent this, apart from fitting fireproof bungs in the pylon, the area of the wing adjacent to the pylon is sealed off and kept free of flammable fluids, and is termed a dry bay.

INTRODUCTION TO FIRE DETECTION

There are three methods of detecting a fire. These are heat detection, smoke detection, and flame detection. Heat detection is normally used in areas that have large changes of air due to the ventilation rates. These are the fire zones around engines, APUs, and main undercarriage wheel bays. Smoke detection is normally used in cargo compartments (B-E) and lavatory compartments. Flame detection requires rapid response.

ENGINE FIRE DETECTION WARNING TIME

Flame detection systems that are fitted to engine bays must be able to detect a fire within 5 seconds of it starting.

CARGO COMPARTMENT FIRE WARNING TIMES

Fires in Class C cargo compartments must be detected within one minute of the fire's initial ignition. This ensures that the crew has sufficient time to take remedial action before the fire can cause structural damage. As this is the fire's nascent phase, where it is more likely that an item has started to smoulder rather than burst into flame, the common method of detection is the smoke detector.

However, to ensure that in the case of an intense rapid growth fire condition existing with limited smoke production, modern smoke detectors, such as those manufactured by Walter Kidde Aerospace, incorporate a heat-sensing element, making them dual units.

SMOKE DETECTORS

The most commonly used smoke detectors for Class B and C cargo areas are those that sense the presence of airborne particulates and aerosols. These are as follows:

- Radioactive ionisation
- Photoelectric light scatter
- Photoelectric light attenuation

RADIOACTIVE IONISATION DETECTORS

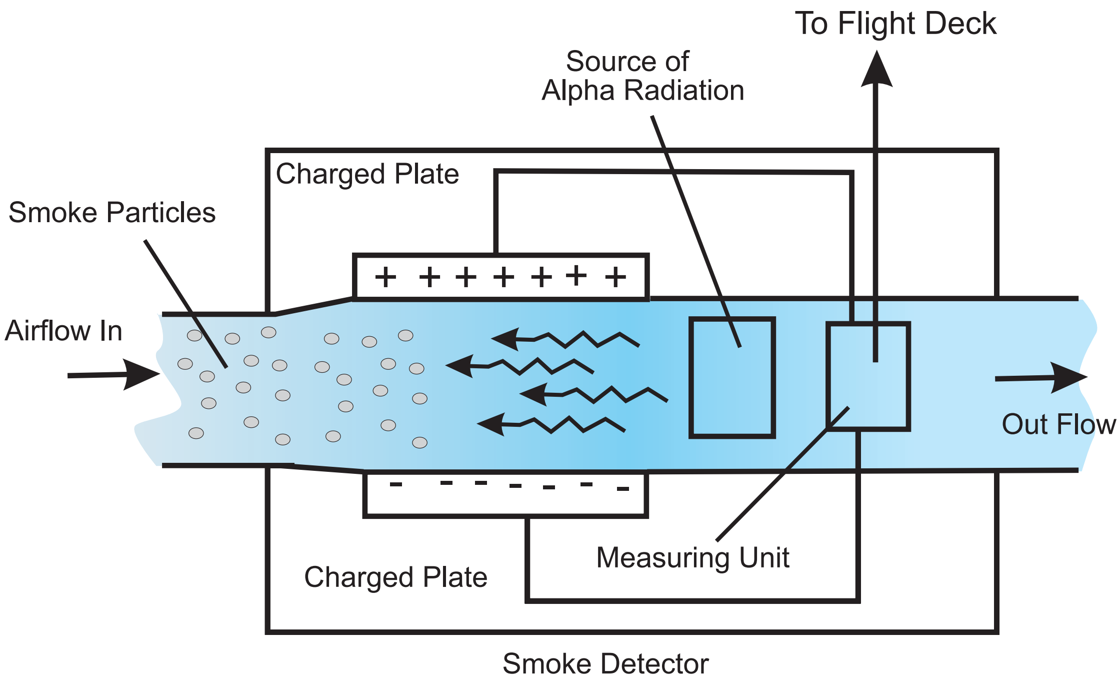


Diagram 14.7 Ionised Radiation Smoke Detector

Ionised radiation smoke detectors operate in a similar manner to the common household smoke detector and are able to detect small amounts of smoke as given off by a flaming fire. They use an ionisation chamber, as in diagram 14.7, a source of ionising radiation, which emits alpha particles. The alpha particles ionise the oxygen and nitrogen atoms of the air in the ionisation chamber.

The act of ionisation removes an electron from the atom. Electrons have a negative charge while the remaining atoms have a positive charge. Two separate plates are across the chamber; one has a negative voltage, while the other a positive voltage supplied from the aircraft's electrical system. When power is applied to these plates, they act as electro-magnets, which attract the ionised particles of the opposite charge.

The electronics within the detection circuit sense the small amount of electrical current that is created by the electrons and ions moving toward the plates. When smoke particles enter the ionisation chamber, they disrupt the current being created by attaching themselves to the ions, which neutralises their potential. In this situation, the detection circuit senses the drop in current between the plates. This triggers the warning.

PHOTOELECTRIC LIGHT SCATTER DETECTORS

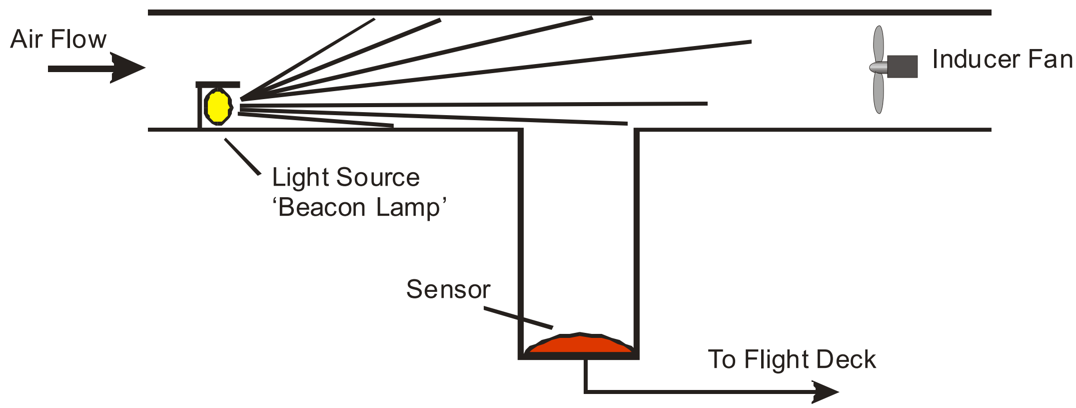


Diagram 14.8 Labyrinth Type Photoelectric Light Scatter Smoke Detector

As light travels in a straight line through clear air, a photoelectric light sensor is placed out of line from a light source in these detectors. Under normal conditions, the sensor is hidden from the light source. When smoke enters the chamber, the light is scattered through refraction and reflection, allowing some of the light to fall on the sensor.

Diagram 14.8 is a schematic of a labyrinth type smoke detector. In this design, the light source is a high-powered light emitting diode (LED). This emits infrared (IR) and is installed at 90° to a photodiode (the sensor). When an internal fan draws air through the labyrinth chamber, any smoke reflects some of the light from the LED onto the photodiode, causing it to give an output voltage. Any increase in the volume of smoke results in a further increase in the output voltage. The internal electronics give an alarm warning when the output voltage reaches a calibrated level.

To prevent spurious warnings caused by dust, filters are fitted to the air intakes, and the labyrinth chamber is heated to stop condensation forming in humid conditions.

These units self test and are designed to prevent spurious warnings from being given on the flight deck. In the event that the unit fails in operation, a fail light warns the pilot.

PHOTOELECTRIC LIGHT ATTENUATION DETECTORS

In these smoke detectors, a light source is in direct line with a sensor, as diagram 14.9 illustrates. Under normal conditions, a beam of light, a known value from the light source, shines on the sensor cell. As the sensor cell is photoelectric, the light creates an electrical voltage that is measured and compared against a set value.

When smoke enters the detector chamber, it starts to obscure the light (attenuation). The subsequent reduction in light falling on the sensor drops the voltage output of the photoelectric cell. The measuring circuit senses the drop in voltage and triggers the flight deck warning. This type of detector requires a greater volume of smoke than either of the previous designs.

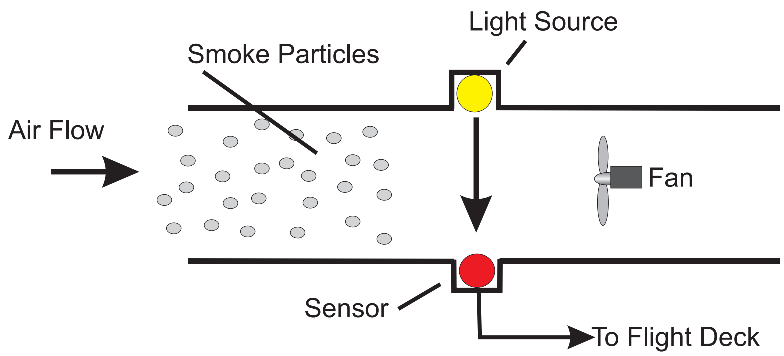


Diagram 14.9 Light Attenuation Smoke Detector

LOCATION AND REDUNDANCY OF SMOKE DETECTORS

Smoke detectors can either be ceiling mounted in the areas that they monitor, or remotely located. Whichever method is utilised, the system installed must continue to function fully in the event of a failure.

To ensure that a single failure in a detection system does not cause the loss of fire warning, the systems are duplicated, so that there are two separate detection systems monitoring the same area.

CEILING MOUNTED

Where they are ceiling mounted, there must be sufficient detectors to enable any source of smoke to register within one minute. A large open cargo hold or smaller hold that is subdivided requires several detectors. The radioactive ionisation type smoke detector with its high sensitivity detection is ideally suited for this type of installation.

REMOTELY MOUNTED

Another method is to mount a pair of smoke detectors in a remote location. Air from several points within the cargo bay is drawn through them, allowing two detectors to cover a large open hold, etc. Optical type smoke detectors function more efficiently when air is passed through their chambers. Labyrinth type smoke detectors are being used remotely in diagram 14.10. To ensure redundancy, two units are mounted in parallel, both taking air drawn from the same sources. The method of drawing the air through the chambers is an electric fan. Again, this is duplicated and in the event of the main fan failing or overheating, the second fan is automatically selected as power is removed from the primary fan.

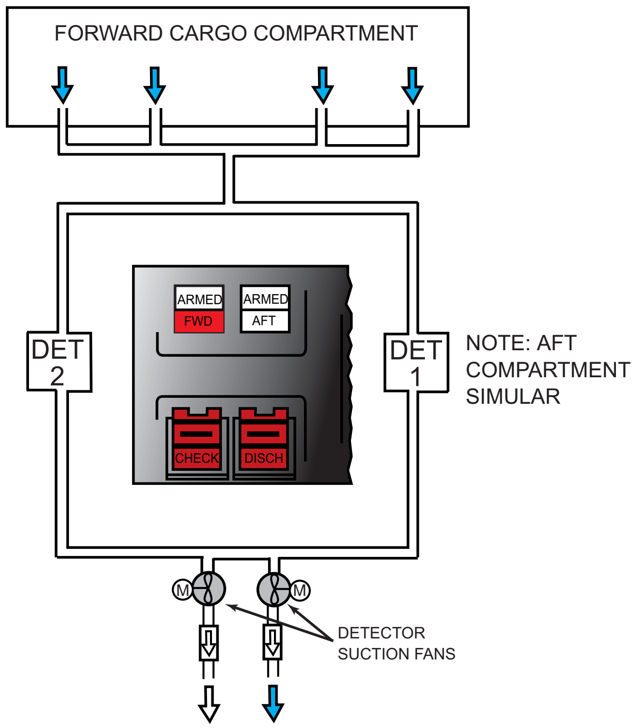


Diagram 14.10 Pair of Remotely Mounted Smoke Detectors

OVERHEAT DETECTORS — FENWAL THERMOSWITCH®

Overheat detectors are fitted in areas where there is the possibility of high temperature air or gases from a ruptured duct impinging on structure and damaging it or starting a fire. These areas include fire zone 3 of a gas-turbine engine, the area surrounding a jet exhaust pipe, and the areas surrounding thermal anti-icing ducting.

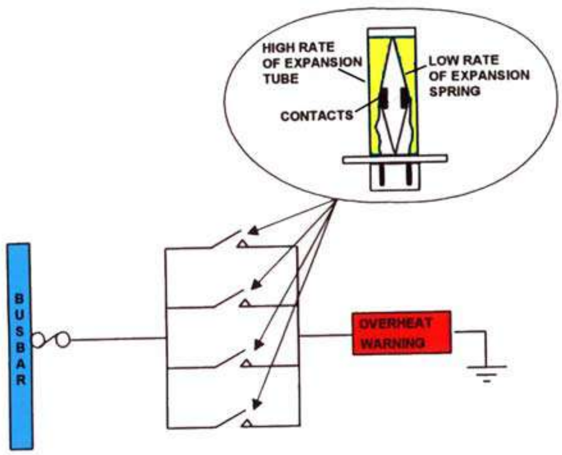


Diagram 14.11 Overheat Detectors

Diagram 14.11 shows a typical overheat sensor. In this design, a tube with a high co-efficient of expansion is mounted on a base plate. There are two sprung elements inside the tube, bowed away from each other by the compression effect of the tube. These sprung elements are made of a material with a low co-efficient of expansion.

These are electrically insulated from each other. One end is attached to the base plate and the other to the sealed end of the tube. Attached to each element is an electrical wire. Thus, the sensor forms an open electrical switch. When the temperature surrounding the sensor is heated, the tube expands faster than the sprung elements, resulting in them being pulled together. When the temperature reaches a predetermined level, the two elements make contact, completing the circuit.

In areas such as zone 3 of an engine bay, several of these detectors are mounted on the surrounding structure around the combustion chamber/turbine. They are wired in parallel to ensure that other units pick up the condition and give the alarm warning should a lancing flame from a cracked casing burn through one of the cables to a sensor.

This sensor reacts to a 1°F change in temperature and responds very rapidly to a sudden increase in temperature due to its design.

FIRE DETECTORS

Fire detectors can be divided into two types; those that respond to an increase in heat above the background level, and those that respond to the optical properties of a fire. The latter are normally referred to as flame detectors.

There are several designs of thermal fire detectors:

- Resistive firewire systems
- Capacitive firewire systems
- Gas pressure systems
- Thermocouple systems

FIREWIRE ™ KIDDE GRAVINER

Firewire™ is the registered trademark of Kiddie Graviner and operates in a dual mode, sensing both resistance and capacitance simultaneously. Other companies manufacture similar style sensors, but these only function as either resistive or capacitative systems.

Note: As it is highly unlikely that the JAA–FCL has made such a distinction, these notes explain both the combined function of the Kidde Graviner Firewire™ system and the older individual system, so that the student is acquainted with modern technology and past technologies.

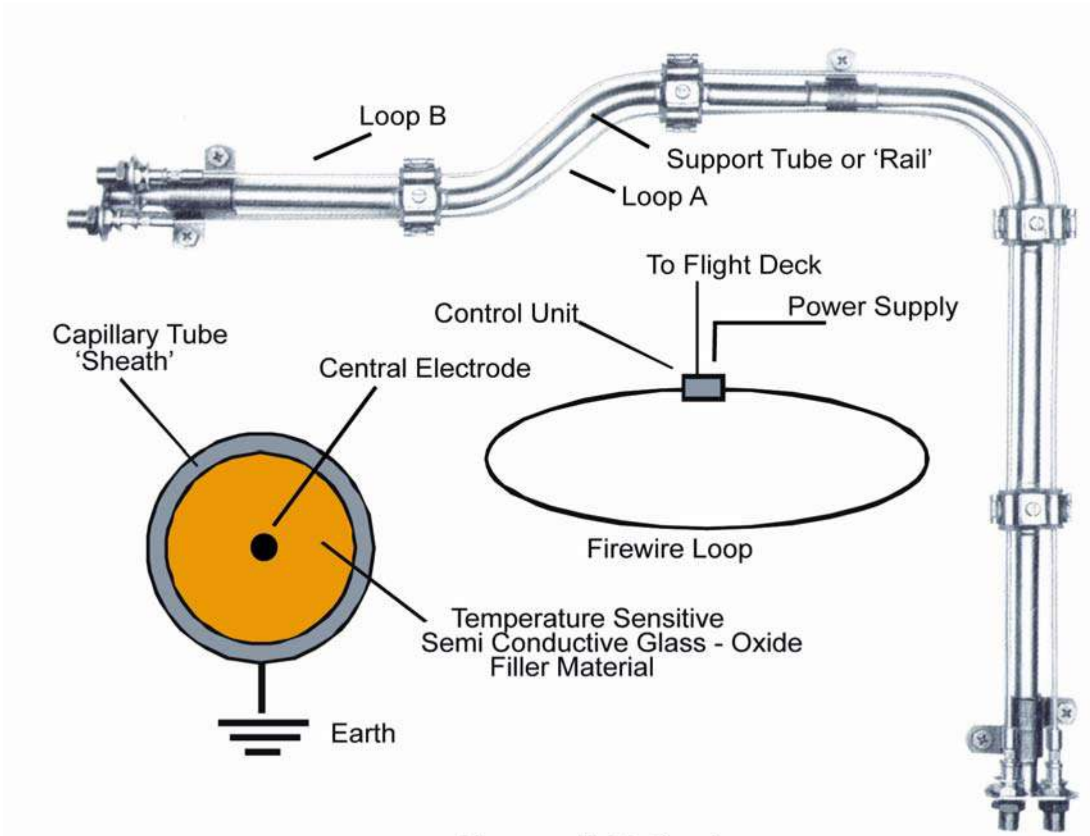


Diagram 14.12 Firewire

Firewire is a term used for a fine central electrode surrounded with a filler material inside a slim stainless steel capillary tube or sheath, which is earthed to the aircraft. Refer to diagram 14.12 showing a cross section of a Firewire. Firewire is mounted as a continuous loop in areas where the outbreak of fire is possible. While Firewire can be turned through a radius of 25 mm, it can be damaged through kinking, crushing, or excessive vibration. To prevent it from becoming kinked or subjected to excessive vibration, clamps along its length support the Firewire. These are normally mounted on a stainless steel support tube or "rail", which in turn is attached to the surrounding structure.

In this design of Firewire™, the central wire is surrounded by a glass/oxide filler material which is temperature-sensitive and semi-conductive, having a negative co-efficient of resistance. The Firewire™ loop is linked to a control unit that measures both resistance and capacitance.

The resistance of the unit is measured between the central wire and the capillary tube. Application of heat either in one point as a "hot spot" or over a length results in the filler material's resistance decreasing. This decrease of resistance is accompanied by an increase in capacitance.

The control unit monitors a combination of resistance and capacitance and signals a warning when the resistance reaches its operating value. This is normally set at 1000 ohms provided that the capacitance is sufficiently high. This prevents spurious warnings in the event of a drop in resistivity.

Both ends of the Firewire™ connect to a control unit. In the event of any break in the Firewire™, each part of the Firewire™ loop still functions as a detector, provided that the central element is not touching the outer capillary tube, as this is earthed.

If there are multiple breaks in the Firewire™ loop, provided that the central element is not touching the outer capillary tube, the parts of the loop that are still attached to the control unit still function as a fire detector.

The control unit checks:

- That the elements and their wiring are continuous
- That the insulation resistance of the element is within limits
- The integrity of the system

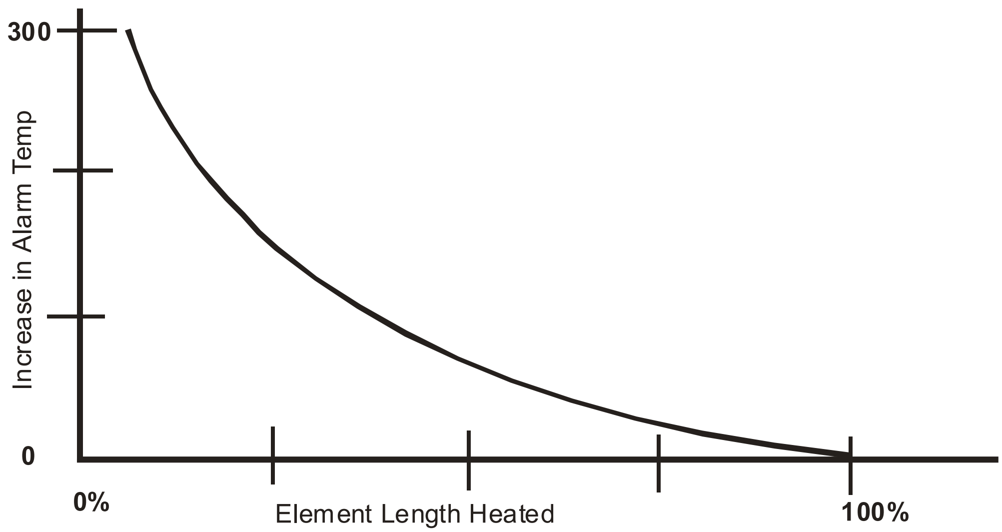


Diagram 14.13 Relationships Between Fire Temperature and Element Length Heated

The Firewire™ is designed to be an averaging system. This means that the greater the length of Firewire™ that is heated, the lower the temperature required to signal the alarm. Refer to diagram 14.13, which gives an indication of the relationship between the temperature in a fire condition that the firewire is exposed to and the element length.

As different areas that need fire protection can have different background temperatures, such as engine bays and wheel bays, Firewire™ is manufactured in three different temperature elements. The maximum average temperature of the area to be protected should be 75°C below the Firewire™ element's operating level.

FIREWIRE AS A GENERIC TERM

Other companies manufacture similar types of capillary-tubed fire sensors. These are generically referred to as **firewire**. There are two separate systems covered by this term. The first is purely a resistive system, while the second is purely a capacitive system. Both use a capillary tube central electrode and filler material.

RESISTIVE FIREWIRE

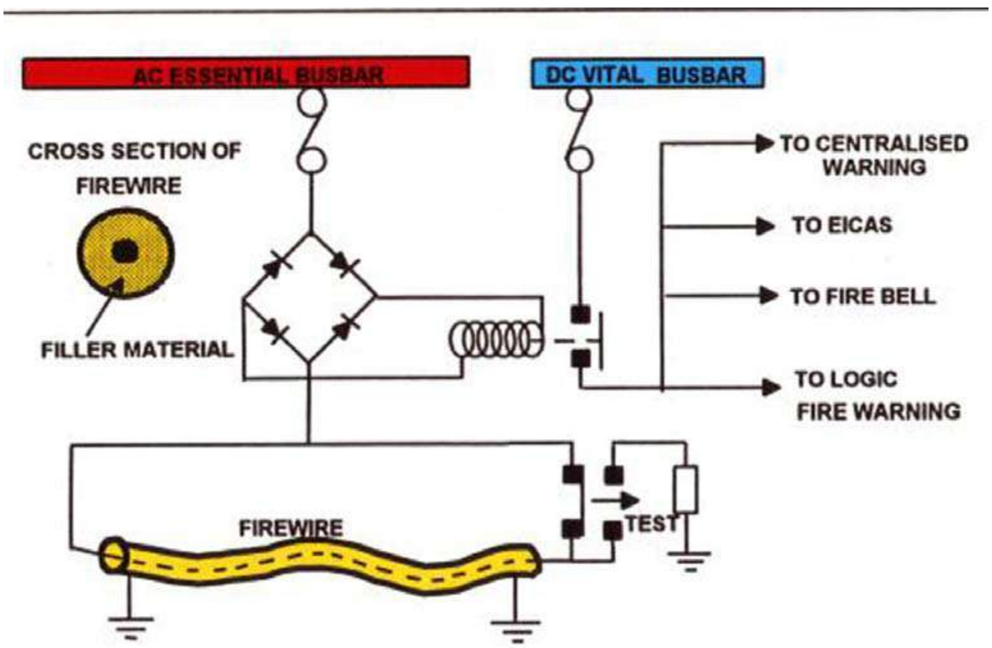


Diagram 14.14 Resistive Firewire

Refer to diagram 14.14. An element similar in construction to the Kidde Graviner Firewire ™ is purely activated by the drop in the filler material's resistance, which acts to earth out the central element. This results in the sensing unit becoming unbalanced and triggers the alarm by closing a solenoid operated switch sending a signal to the flight deck.

If this sensor becomes crushed, there is the likelihood of the central element touching the capillary tube, instantly earthing it out, which causes a spurious fire warning. If the sensor is not completely crushed, there is still the possibility of the background heat within the bay causing a spurious fire warning, due to the closer proximity of the outer tube to the inner element and reduction in the filler material between the two.

To test the system, the charge is earthed out to simulate a fire condition. Passing a current through the element checks its integrity.

CAPACITIVE FIREWIRE

To overcome the deficiency in the purely resistive system, the manufacturers designed the capacitive system, see diagram 14.15. In this system, the firewire loop is connected to a controller. This allows a charge into the central electrode for a set period and then discharges it in to a measuring circuit.

In the event of a fire, the capacitance of the central element increases, allowing a greater charge into the element within the same period. When the system discharges, it is this greater capacitance that activates the fire warning. In the event of this system becoming crushed and earthing out, the controller senses that there is no discharge and illuminates a fail light in the flight deck. This is sometimes referred to as a **no fault system** as it cannot give a spurious warning.

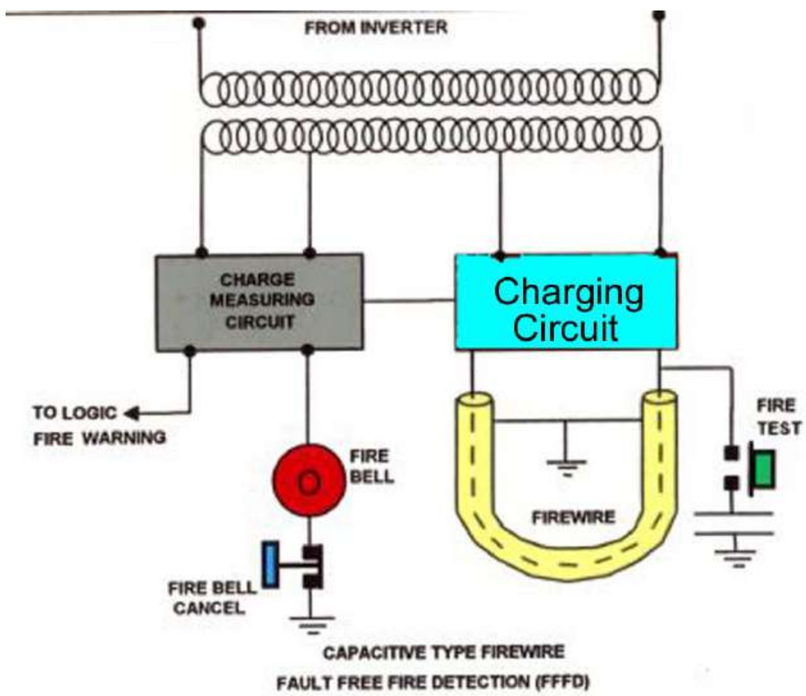


Diagram 14.15 Capacitive Type Firewire

When the system is fault tested, the charge of the central electrode is earthed out. This should illuminate a fail light. The functional test passes a large capacity into the firewire, then into the measuring circuit.

SYSTRON DONNER PRESSURE-OPERATED FIRE/OVERHEAT DETECTOR

This system operates on the principle that gases expand with an increase in temperature and that if these gases are trapped within an enclosed container, their expansion results in an increase in pressure. This can be used to close a pressure sensitive switch.

Refer to diagram 14.16. In this design, a sensor tube that is sealed at one end is connected to two pressure switches mounted in a housing, which is termed a responder unit. Within the sensor tube is an open spiral wire of titanium hydride. Titanium hydride is a metal that acts as a “sponge” to hydrogen when cool but releases the hydrogen as a gas when heated.

The sensor tube is also charged with helium gas to a pressure above one atmosphere. This pressure acts to hold a pressure switch, termed an **integrity** or **Built-In Test (BIT) switch**, closed. In the event of the helium pressure dropping below a pre-determined level, the switch opens and illuminates a fail light in the flight deck.

In overheat conditions as the general temperature of the area increases, the helium gas pressure increases and closes the alarm switch, triggering the fire warning on the flight deck.

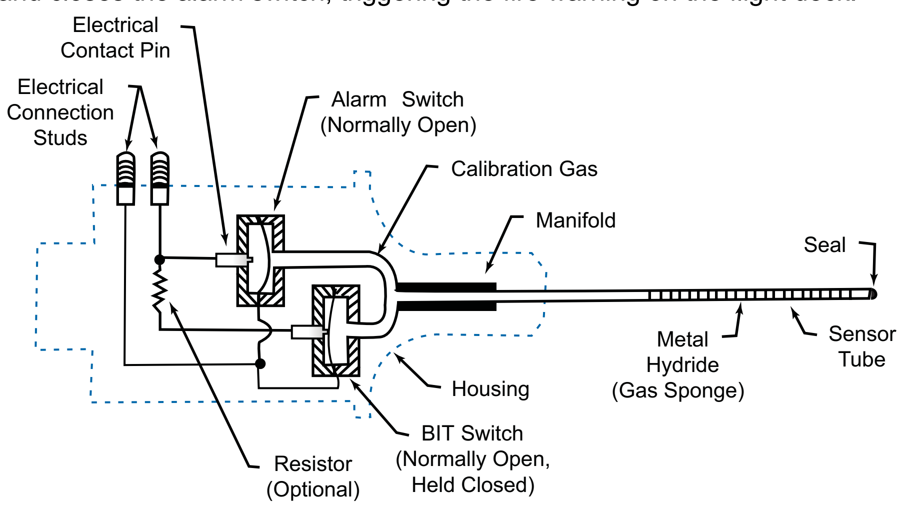


Diagram 14.16 Pressure Operated Fire/Overheat Switch

In the event of a localised “hotspot” or discrete” fire where a flame impinges on the sensor tube, heating up a small area before raising the average temperature of the bay, the hydrogen is liberated from the metal hydride. This increases the pressure, closes the alarm switch, giving the flight deck fire warning.

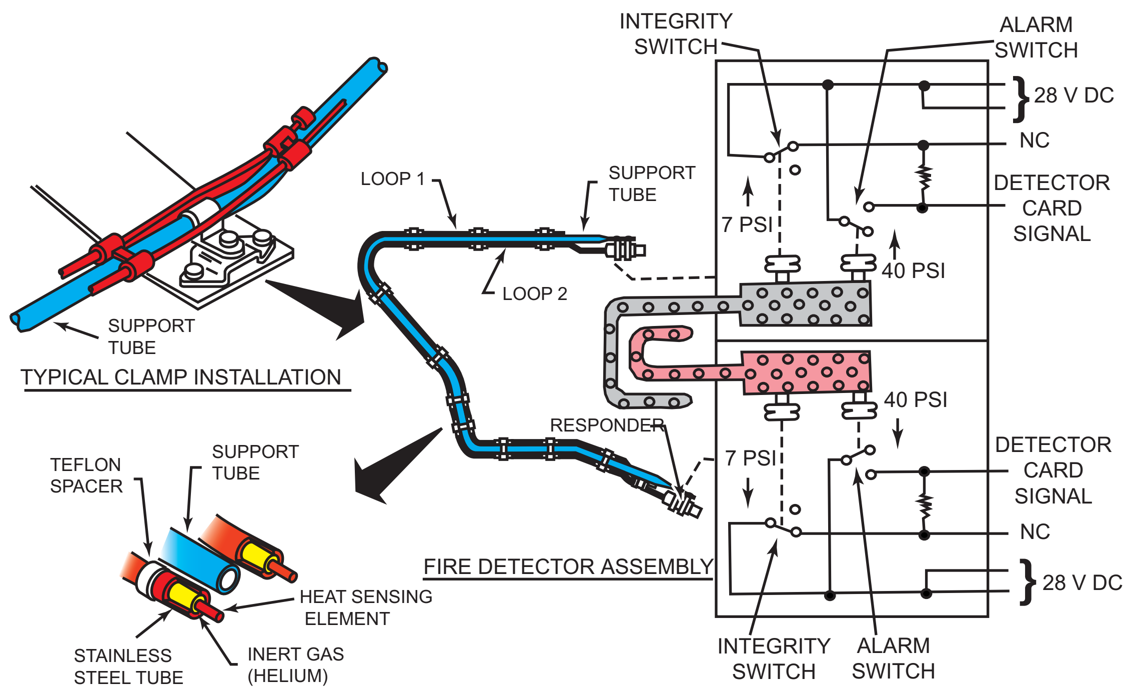


Diagram 14.17 Systron Donner Detector

The electrical supply for the system is taken from the aircraft’s vital busbar via a detection card control circuit and is passed to the open circuit alarm switch through the closed integrity switch. Allowing the supply to pass through a resistor to the detection card completes the circuit. This results in a drop in voltage in the supply when it returns to the control circuit.

In the event of the loss of calibration gas pressure, the integrity switch opens and the circuit is broken. The control circuit noting the loss triggers the failure light. In the event of the alarm switch closing, the full voltage is able to pass back to the control circuit, triggering the flight deck warning.

THERMOCOUPLE FIRE DETECTOR

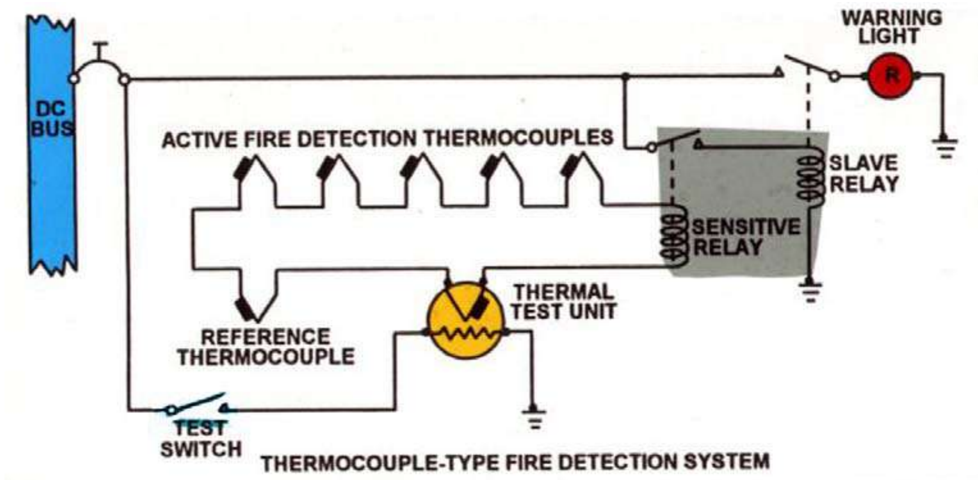


Diagram14.18 Thermocouple Type Fire Detector

This is an older design of fire detector that uses the electrical potential produced by two strips of dissimilar metals that are connected at the ends and where one end is heated.

In this design, several thermocouples are connected in series. One thermocouple is located in an area that has the same background temperature as the area to be monitored but unlikely to be subjected to an overheat or fire situation. See diagram 14.18.

When an overheat situation occurs, the difference in temperature between the reference unit and the sensing units acts to create a current flow; the larger the temperature differential or the more units that are affected, the greater the flow from hot to cold. This current energises a sensitive relay, which in turn makes the warning circuit.

To carry out an integrity test, a further thermocouple is located in the same area as the reference unit. A heating element is co-located to the unit. When the pilot selects “test”, the element heats and the differential temperature causes the current flow, triggering a fire warning. In the event that a connecting cable between units severs, only those units connected to the cold end function as fire detectors.

JAA REQUIREMENTS FOR FIRE WARNING

The flight crew of twin-engine aircraft (gas turbine or reciprocating fitted with a turbo charger), commuter aircraft, or aircraft where the engine is not readily visible from the flight deck must receive warning if a fire or overheat condition exists in:

- Engine fire zone 1
- The hot zone 2 of a gas-turbine engine

  - The combustion section
  - The turbine section
  - The tailpipe

- APUs
- Any other fire zone
- Main wheel bays
- Cargo compartments (B-E)
- Restroom compartments

The JAR requirements state that an aircraft that requires fitting of a fire or overheat warning cannot take off unless it has serviceable fire detection and a suitable suppression system is fitted and functional.

DUAL LOOPS

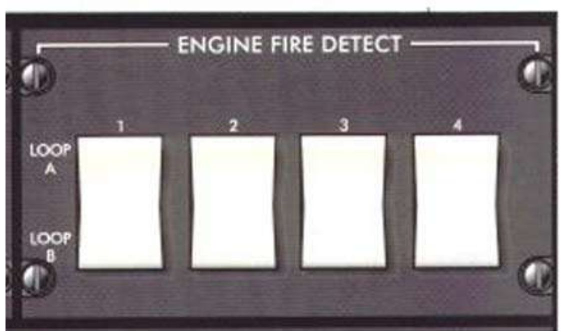


Diagram 14.19 Fire Detector Selections

Many aircraft have two sensing systems fitted to monitor one area. These are normally referred to as loops and are normally designated as A and B. A three-position rocker switch, as per diagram 14.19, or a rotary switch are usually used, allowing the selection of:

- Loop A only, loop B only
- Loops A + B together

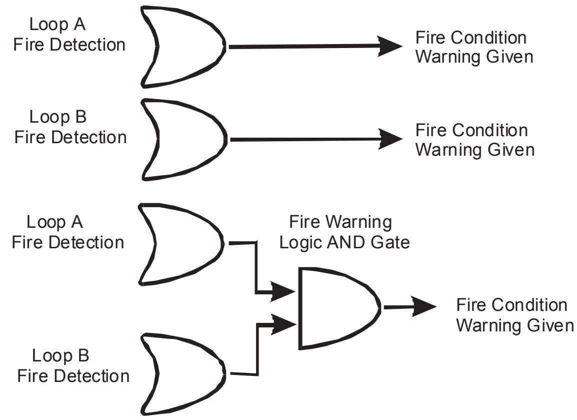


Diagram 14.20 Fire Warning Logic

Diagram 14.20 shows the fire warning logic of selecting the three different conditions. If loops A or B only are selected, anything that triggers the sensor sends the alarm signal to the flight deck. By selecting Loops A + B, the signal from each loop is sent to an AND gate, where the alarm signal is not sent until both loops detect a fire condition.

A and B work together to reduce the possibility of spurious alarms. In the event of a loop failing during flight, the crew can cancel the failed loop and revert to a single loop. The JAR Minimum Equipment Lists (MELs) only require a serviceable fire detection system at dispatch. Therefore, a dual system allows an operator to continue flying an aircraft for revenue with one failed loop.

BUILT-IN TEST — BIT

Modern fire detection systems have built-in test circuits. When electrical power is applied to the systems, they constantly monitor the loops for integrity of the whole system. If the test circuit detects a fault, an amber “Fault” light illuminates on the appropriate fire-warning panel. This alerts the pilot to select the serviceable loop.

However, in the unlikely event of a fault occurring in one loop where both have been selected exactly at the same time as a fire occurs, modern solid-state controllers monitoring the systems are designed to give a fire warning.

Older systems required the pilot to frequently press a test button in flight to determine that the system had integrity.

PRE-FLIGHT TEST

The commander of an aircraft with a fire warning system must test the system’s integrity before take-off to ensure compliance with the regulations.

AURAL AND VISUAL ALERTING

For fire, the aural warning can take the form of a bell, horn, or chime. The crew can cancel this. Once cancelled, if another fire warning is given, the aural warning sounds again.

To visually alert the pilot to a problem, attention getters flash red, which the pilot can cancel. As with the aural warning, if a further fire condition occurs, the attention getters flash again.

FIRE WARNING — INDICATING THE SOURCE

A constant red light illuminates on the appropriate fire-warning panel, see diagram 14.21 for an example.

The pilot cannot cancel this and the light remains illuminated until the fire condition ceases to exist, when the fire sensor resets itself. If another fire condition for the same area were to exist, the warnings would start again after it has reset.

Diagram 14.21 Central Warning Panel

Diagram 14.22 Fire Handles Illuminate to Warn of Engine Fires

For large aircraft built under JAR 25, it is standard for an engine's fire warning to be given by illuminating the appropriate fire handle, see diagram 14.22. The function of the fire handle is covered later in these notes.

OVERHEAT WARNING

Where pilots have to be warned of overheat conditions existing, such as an increase in temperature around a gas turbine exhaust pipe, etc., an amber/yellow overheat light illuminates on the appropriate warning panel, not in the fire handles. See diagram 14.21 for pylon overheat. The "abnormal procedures" of the flight manual would detail the remedial action.

FIRE WARNINGS FOR LAVATORY COMPARTMENTS

For aircraft with 20 passenger seats or more, each lavatory must be equipped with a smoke detector system or equivalent that provides:

- A warning light in the cockpit, or
- A warning light or audible warning in the passenger cabin that would be readily detected by a cabin crewmember.

FIRE SUPPRESSION

This next section of the chapter details how to achieve in-flight fire suppression and the regulations in force. This is subdivided into fixed fire extinguisher systems and hand-held fire extinguishers as it aligns itself with further emergency equipment. The fire extinguishing agents used in aircraft are discussed first.

FIRE EXTINGUISHING AGENTS

Agents that can be used for aircraft fixed fire extinguisher systems are:

Agent	Known as	Chemical Formula	Usage
Carbon Dioxide	CO_2	CO_2	Cargo compartments Engine bays
Bromochlorodifluoromethane	BCF Halon 1211	$CBrClF_3$	Flight deck Passenger Compartments Cargo Compartments Engine bays
Bromotrifluoromethane	Halon 1301	CF_3Br	Cargo Compartments Engine bays
Water		H_2O	Passenger Compartments
Methyl Bromide	MB	CH_3Br	Engine bays

CARBON DIOXIDE — CO_2

CO_2 is a gas that extinguishes fire by dispelling the oxygen from the immediate area. It has a toxic effect and, therefore, is not normally used in passenger cabins (see note).

While CO_2 is effective at smothering a fire, the cooling effect of CO_2 is very strong. This can lead to freeze burns to flesh and cause thermal shock to hot metals. The JAA regulations state that no more than five pounds of CO_2 may be discharged into any fuselage compartment. Therefore, CO_2 is not commonly used to extinguish an aircraft's fixed fire.

The CO_2 is stored in the fire extinguisher as a liquid but due to its volatility, it does not require an additional gas pressure to expel it from the container.

Note: Although there is a small percentage of CO_2 in the lungs to regulate breathing, haemoglobin finds it easier to absorb CO_2 than oxygen. Therefore, any increase in CO_2 in the local atmosphere can be toxic to us.

BROMOCHLORODIFLUOROMETHANE — CBRCIF3

BCF–Halon 1211 is a halogenated hydrocarbon. Chemicals of this group are also referred to as **freons**. BCF is a non-corrosive chemical that forms a blanketing mist when released, which deprives the fire of oxygen and interferes with the combustion process, preventing re-ignition.

It is stored as a liquefied gas kept under pressure by nitrogen, which also starts the expulsion of the liquid from the container when the fire extinguisher is operated. BCF does not cause cold burns or thermally shock heated metals and has a lesser toxicity than CO_2. It also has the advantage of being directed as a stream from a hand-held fire extinguisher, allowing the user to fight fires from a safe distance. It is this extinguishant that the regulators have in mind when determining if a bay is B or C.

BROMOTRIFLUOROMETHANE — CF_3BR

BTM-Halon 1301 has the same fire knock down properties as Halon 1211 but is less toxic than BCF. It is stored as a liquefied gas kept under pressure by nitrogen, which also starts the expulsion of the liquid Halon 1301 from the container when the fire extinguisher is operated.

However, Halon 1301 readily converts to a gas as per CO_2 and is less directable than BCF. It is not suitable as the agent in a hand-held fire extinguisher.

This agent is the standard agent that manufacturers such as Kidde Graviner supply for engine and cargo fixed fire suppression installations.

WATER — H_2O

Water filled hand-held fire extinguishers are carried in the passenger cabins to fight Class A fires. The water is expelled from the extinguisher by nitrogen gas pressure.

METHYL BROMIDE — MB

Methyl Bromide is stored as a liquefied gas kept under pressure by nitrogen, which also starts the expulsion of the liquid from the container when the fire extinguisher is operated. It is an older agent that is highly toxic and corrosive to aluminium alloys, magnesium alloys, and zinc. If an engine bay is contaminated by error with this agent, the engine must be run to heat the area and evaporate the agent and checks on the surrounding structure are required.

Methyl Bromide has toxicity such that exposure to concentrations of 0.5 to 1% by volume over a period of half an hour can be fatal.

Methyl Bromide is the most harmful of the agents available and is being phased out of service as many manufacturers do not supply or service these units. However, be aware that some aircraft might still have this agent on board.

PROTECTIVE BREATHING EQUIPMENT FOR FLIGHT CREW

Diagram 14.23
Oxygen and Smoke Goggles

Where an aircraft is fitted with an extinguishing system, regulations require that there be protective breathing equipment for each flight crewmember on flight deck duty, as illustrated in diagram 14.23. The first action of a flight crew on receiving and confirming a fire warning is to don protective breathing equipment to ensure that either toxic fumes or fire extinguishing agent do not incapacitate them.

BUILT-IN FIRE EXTINGUISHER SYSTEMS

For areas that a crewmember cannot access with a hand-held fire extinguisher, a fixed fire suppression system must be fitted. The regulations state that:

- Fuel burning heaters (combustion heaters) may be protected by an individual one-shot system.
- For each other designated fire zone, two discharges must be provided, each of which produces adequate agent concentration.
- The capacity of each required built-in fire extinguishing system must be adequate for any fire likely to occur in the compartment where used, considering the volume of the compartment and the ventilation rate.
- The fire-extinguishing system for a nacelle must be able to simultaneously protect each zone of the nacelle for which protection is provided.

To comply with this requirement, the system must have sufficient agent to:

- Knock the fire down.
- Blanket the area to prevent re-ignition while it cools.

However, to alleviate the weight penalty of carrying two full-size extinguishers per cargo hold or engine bay, the authorities allow the use of a master and back up extinguisher system that can cross feed between two holds or engines, see Cross Feed System below.

FIXED FIRE EXTINGUISHERS — SPHERICAL

Modern fixed fire extinguishers, **fire bottles**, are spherical in shape, as they hold the larger volume for a smaller size compared with the older cylindrical designs.

SQUIBS AND SQUIB TEST

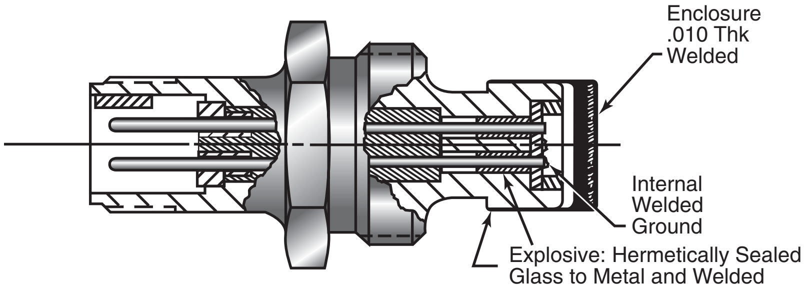


Diagram 14.24 Squibs

Each outlet incorporates a frangible disc type valve that is ruptured by a small explosive charge called a **squib**, as shown in diagram 14.24. When the fire extinguisher is operated, an electrical current fires the charge, causing it to detonate. This explosion either ruptures the sealing disc directly or drives a slug through it, releasing the agent.

To ensure the integrity of each firing circuit, the crew carries out a continuity test before flight. When the pilot depresses the squib test button, a micro voltage is passed through the squib. This tests the complete circuit without detonating the squib. If the system is serviceable, a green or yellow/green light illuminates, as diagram 14.25 shows. An unsatisfactory test precludes flight until completion of remedial action.

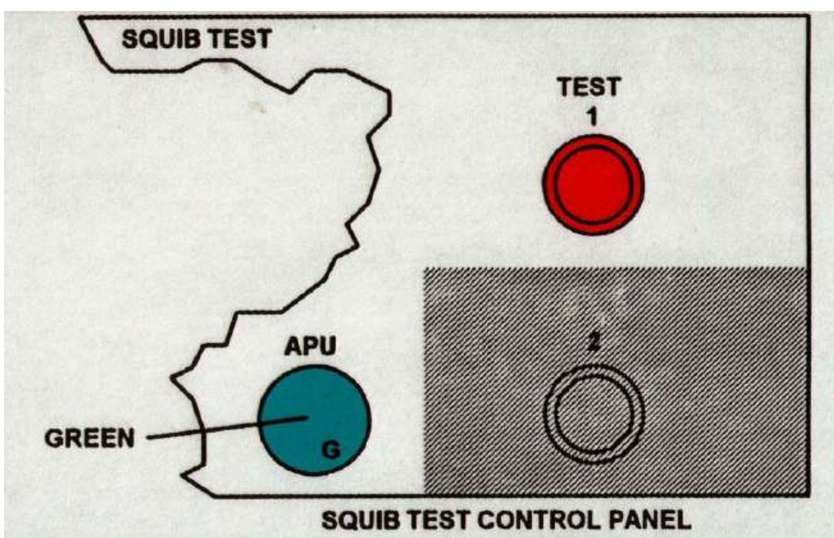


Diagram 14.25 Squib Test Panel

FIXED FIRE EXTINGUISHERS — CYLINDRICAL

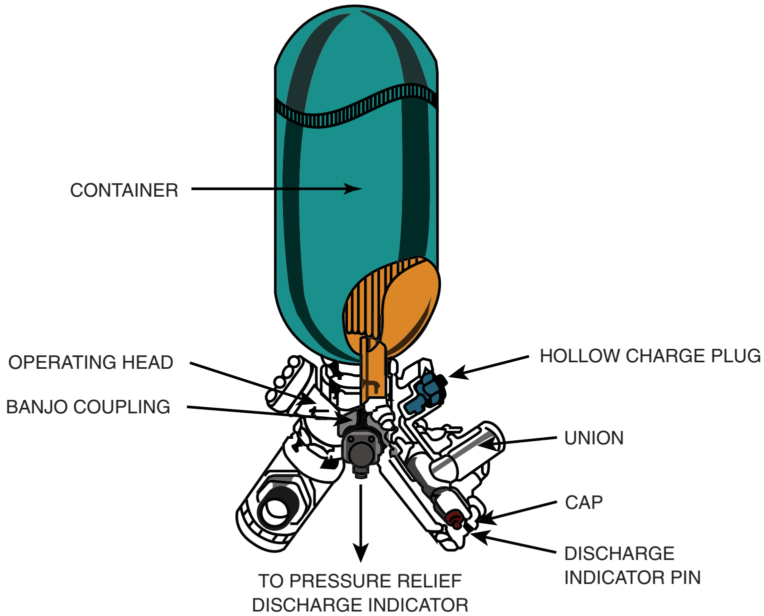


Diagram 14.26 Dual Operating Head Cylindrical Fixed Fire Extinguisher

Diagram 14.26 shows a cylindrical fixed fire extinguisher with two outlet branches, referred to as a **Dual Operating Head**, connected to the neck of the cylinder. Each leg of the dual operating head, which takes the form of an upside down Y, is connected to a separate compartment or engine and has its own squib. See cross feed later in these notes.

THERMAL OVER-PRESSURE VENT VALVE AND THERMAL DISCHARGE INDICATOR

To prevent the containers from bursting in the event of excessive temperature acting on the container, a pressure relief valve opens and vents the entire contents to atmosphere. This operates the external Thermal Discharge Indicator. To ensure that the crew are warned of a discharge, a pressure switch is fitted. This is linked to the fire-warning panel and activates the discharge light.

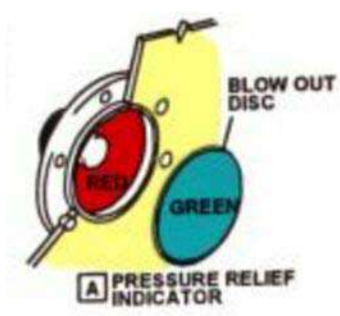


Diagram 14.27 Thermal Discharge Indicators

To indicate that a bottle has been discharged due to thermal over pressurisation, a thermal discharge indicator is fitted to the aircraft's skin, see diagram 14.27. The discharge indicator is connected to the vent line. When the contents of the container are relieved overboard, it ejects a disc to show a red disc. This warns the ground crew and the pilot on a pre-flight walk round that the fire extinguisher bottle is empty.

DISCHARGE INDICATORS

Two methods exist to indicate discharge of a fire extinguisher without applying electrical power and checking the fire panel:

- Discharge pin
- Discharge piston

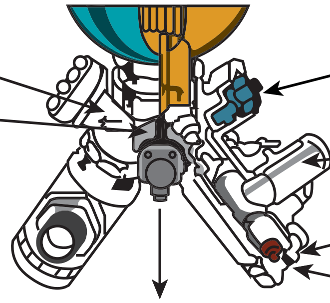


Diagram 14.28 Fixed Fire Extinguisher Discharge Indicator Pin

Where the operating head is accessible, a discharge pin can be used. When the squib is detonated, it drives a piston down that causes a small pin to protrude from the centre of the end cap. This can be felt or seen, see diagram 14.28.

Where the operating head is inaccessible, a discharge piston can be used. This takes the form of a skin fitting that is plumbed into the line from the container to the compartment. When the squib is detonated, gas pressure from the released agent drives a small piston outward. This ejects a yellow disc, and then blocks the flow of the agent, allowing all of the agent to discharge into the hazard zone.

CROSS FEED SYSTEM

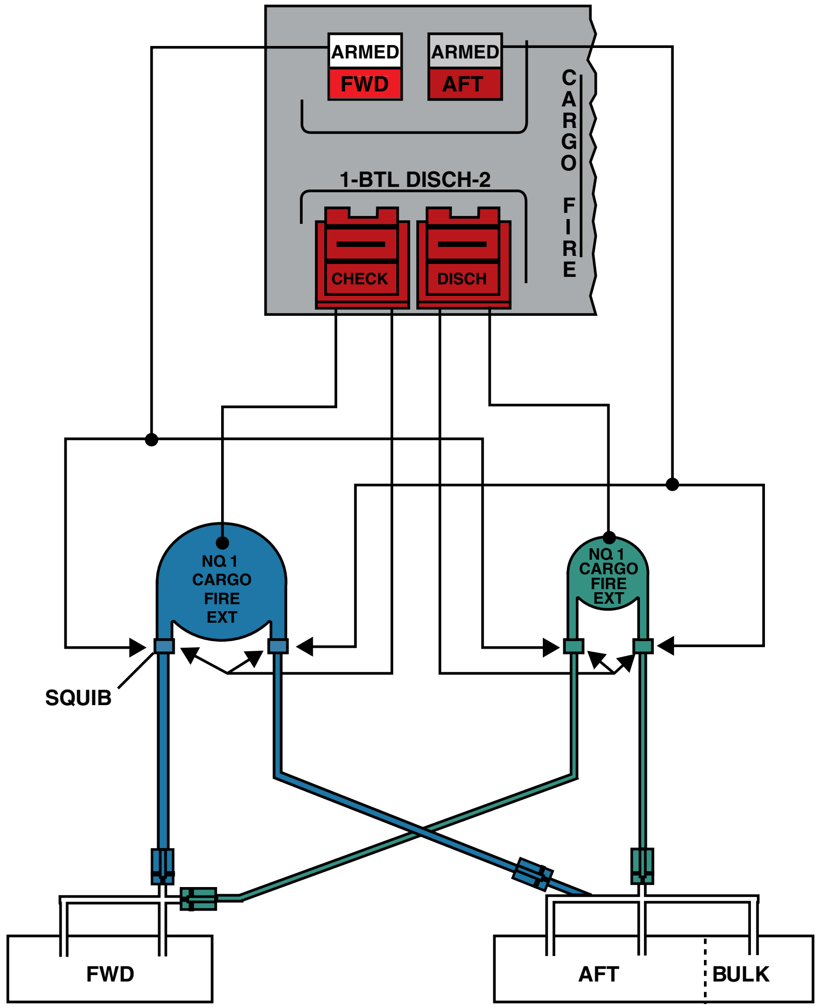


Diagram 14.29 Cross Feed System for a Cargo Compartment

Diagram 14.29 shows a schematic of a cross feed system for a medium-range twin-jet aircraft. In this system, there are two fixed fire extinguisher bottles, both fitted with dual operating heads. One head from each bottle is plumbed to the fwd and aft cargo bays.

The larger bottle is the master cylinder and will be operated in the event of a fire into either of the two bays. The contents of this cylinder are sufficient to knock down a fire in either bay and maintain an atmosphere that does not support combustion for sufficient time to allow the structure/source of ignition to cool so as not to re-ignite the fire.

The second bottle allows the concentration of extinguishing agent to be increased if the fire warning remains on to cover for loss via ventilation rate and to provide cover for the other bay for the remainder of the flight. This system allows the operator to save weight. Otherwise, the aircraft would have to be equipped with two fire bottles per bay.

ELECTRICAL ACTUATION OF A SINGLE SHOT SYSTEM

In the case of single fire bottles, each firing circuit consists of an arming switch and a guarded discharge switch. These switches normally take the form of the push and lock type buttons.

The continuous red fire warning is given by illuminating the warning legend. This takes the form of "FWD/AFT" or "FIRE/SMOKE" in the arming button. Pushing this switch arms the firing circuit of the squib in the discharge outlet of the fire extinguisher.

To indicate that the arming switch has been depressed and that the firing circuit is armed, a white light in the button illuminates the legend "ARMED". Now both legends in the arming button are illuminated.

To discharge the fire extinguisher, the pilot has to lift the guard and depress the firing button. This completes the firing circuit, causing the squib to detonate and illuminates the white discharge legend. When the fire has been extinguished, and the sensor detects a return to normal conditions, the red warning legend goes out.

ELECTRICAL ACTUATION OF A CROSS FEED SYSTEM

To discharge either fire extinguisher, the pilot has to lift the appropriate switch guard and depress the firing button. This completes the firing circuit, causing the squib in the selected fire extinguisher to detonate and illuminates the appropriate white discharge legend.

When the fire has been extinguished, and the sensor detects a return to normal conditions, the red warning legend goes out. If the fire condition continues, the pilot operates the second extinguisher, which is already armed.

ENGINE FIRE INDICATIONS

For large air transport aircraft, the continuous fire indication is normally given by illuminating the appropriate engine's fire handle. In the event of an engine fire, to meet the requirements of isolating the engine from flammable fluids, removing possible sources of ignition, closing the engine down, arming and operating the fire extinguisher quickly, these functions have been combined into two operations of the fire handle.

FIRE HANDLES

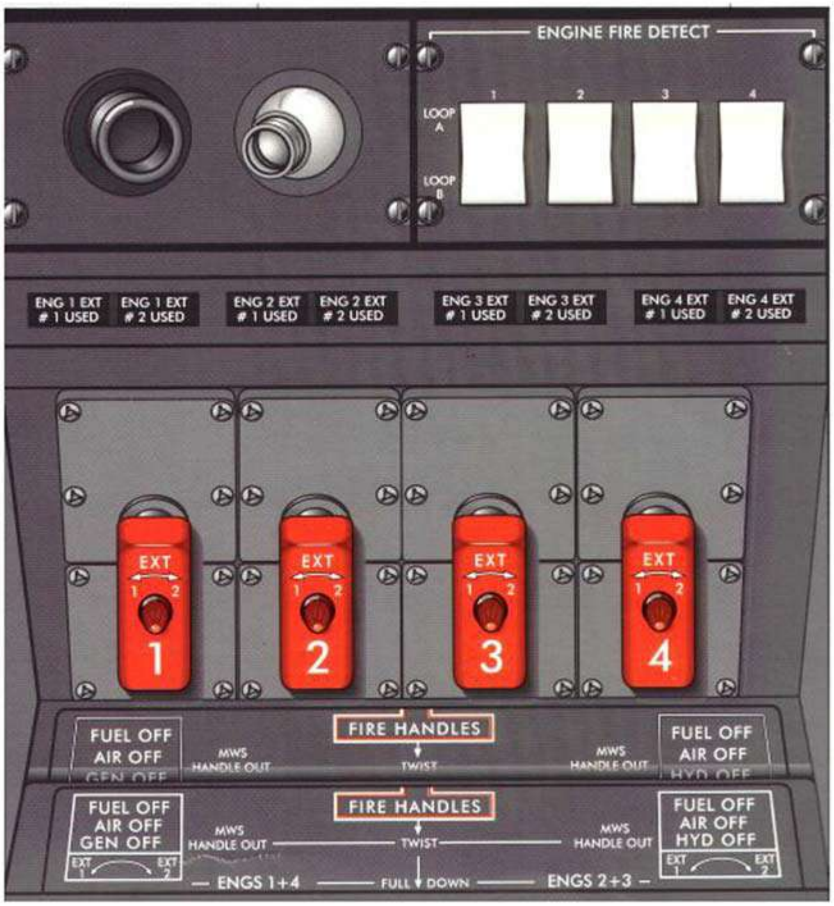


Diagram 14.30 Engine Fire Warning Panel

Refer to diagram 14.30. The first action is to pull the appropriate fire handle. This action simultaneously:

- Disconnects the engine's alternator from the aircraft's electrical system
- Closes the fuel, hydraulic, and any other flammable fluid FSOVs
- Isolates the engine's ignition circuit
- Arms the firing circuits

In the case of a large fanjet, the crew must wait for the engine's RPM to drop before firing the extinguishers, to reduce the ventilation rate through the fire zone.

The second action is to turn the fire handle in the direction of the master cylinder No. 1, hold it to actuate the appropriate squib, and then carefully return it to the central position. Turning the handle in the opposite direction fires the other extinguisher. Take care as the handles are spring-loaded to centre, and it is feasible for the handle to bounce when released.

The fire handle remains illuminated until the fire has gone out, and the firewire has reset itself. Operation of fire handle in discharging a bottle illuminates the appropriate white discharged light. This remains on for the remainder of the flight. It is standard practice for the fire handle to remain pulled out as this ensures that the engine bay remains isolated from fuel, etc.

After a fire has been successfully put out, many operational procedure manuals instruct the crew to operate the second engine/cargo bay bottles prior to landing as a precaution against re-ignition during the landing phase.

For turboprops, the propeller must be feathered to prevent it from windmilling, not only creating drag, but also driving the engine and increasing the ventilation rate.

AUXILIARY POWER UNITS FIRE PROTECTION

Modern APUs have two operating conditions:

- Ground
- Flight

On the ground, it is common practice for APUs to be left running when the flight deck is unattended. To ensure that there is fire protection, the fire detection and suppression system can be selected to automatic. If either loop of a twin loop system detects a fire condition, it automatically shuts down, isolates the APU, and operates the fire extinguisher.

In flight, regulations require the crew to monitor the APU, so the system is set to manual. Where any fire warning is given via the attention getters, horn, and dedicated continuous red light, the crew activates the APU's fire bottle. Modern solid-state systems use the aircraft's weight-on-wheels ground logic to automatically change the fire detector from in-flight AND logic to ground OR logic, see diagram 14.31.

Diagram 14.31 APU Control Panel

WHEEL BAYS

Main wheel wells are monitored for fire and overheat, as there is a possibility of a fire starting alongside the wing fuel tanks due to overheated brake units. The corrective action in the event that an overheat warning is given, is for the pilot to slow down to V_{LE} and lower the gear to cool or blow out the flames.

HAND HELD FIRE EXTINGUISHERS

The regulations state that:

- Each hand fire extinguisher must be approved.
- The type and quantity of extinguishing agent must be suitable for the kinds of fires likely to occur in the compartment where the extinguisher is intended to be used and must minimise the hazard of toxic gas concentration for personnel compartments.
- Each extinguisher should be readily accessible and mounted to facilitate quick removal from its mounting bracket.
- Unless an extinguisher is clearly visible, its location should be indicated by a placard or sign.

FLIGHT DECK

There should be at least one fire extinguisher, suitable for both flammable fluid and electrical equipment fires, installed on the flight deck. This fire extinguisher must contain Halon 1211 (Bromochlorodifluromethane, CBrCIF2) or equivalent and must be conveniently located on the flight deck for use by the flight crew. This is normally taken as being Halon 1211.

Dry chemical fire extinguishers should not be used on the flight deck or in any compartment not separated by a partition from the flight deck because of the adverse effect on vision during discharge and, if non-conductive, interference with electrical contacts by the chemical residues.

CARGO COMPARTMENTS A, B, AND E

At least one readily accessible hand fire extinguisher must be available for use in each Class A or Class B cargo or baggage compartment and in each Class E cargo or baggage compartment that is accessible to crewmembers in flight.

GALLEYS

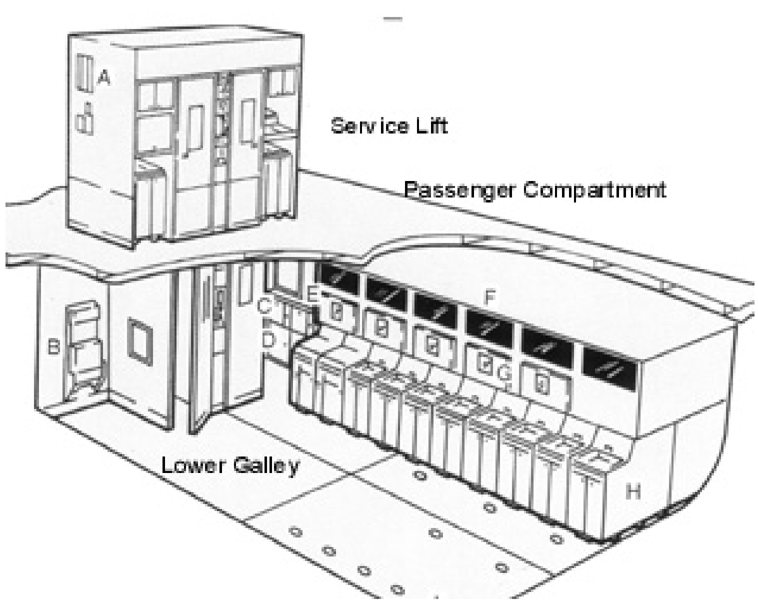


Diagram 14.32 Lower Galley of a Large Air Transport Aircraft

At least one hand fire extinguisher must be located in or readily accessible for use in each galley located above or below the passenger compartment.

HAND-HELD FIRE EXTINGUISHER FOR PASSENGER COMPARTMENTS

The **minimum** number of hand fire extinguishers and their types appear in the table below.

Passenger capacity	Number of extinguishers	Type
7 to 30	1	Halon 1211 or equivalent
31 to 60	2	One Halon 1211 or equivalent
61 to 200	3	Two Halon 1211 or equivalent
201 to 300	4	Two Halon 1211 or equivalent
301 to 400	5	Two Halon 1211 or equivalent
401 to 500	6	Two Halon 1211 or equivalent
501 to 600	7	Two Halon 1211 or equivalent
601 to 700	8	Two Halon 1211 or equivalent

- Where more than one fire extinguisher is required, the other extinguisher(s) must be appropriate for the kinds of fires likely to occur where used.
- Where only one hand fire extinguisher is required in the passenger compartments, it should be located near the cabin crewmember's station, where provided.

In situations where there is no cabin crewmember, the fire extinguisher is located adjacent to the entry door.

- Where two hand fire extinguishers are required in the passenger compartments and their location is not otherwise dictated by consideration of regulations, the fire extinguishers should be located near each end of the cabin.

  In the situation where there is a cabin crewmember, the second fire extinguisher would be located adjacent to the entry door or the opposite end of the cabin.

- Where more than two hand fire extinguishers are required in the passenger compartments, the additional extinguisher(s) should be located near each end of the cabin, with the remainder distributed throughout the cabin as evenly as is practicable.

In reality, taking into account the number and size of the passenger compartments, the need to minimise the hazard of toxic gas concentrations and the location of lavatories, galleys, etc., may result in the number being greater than the minimum prescribed.

LAVATORY FIRE PROTECTION

There is a requirement for aircraft with 20 passenger seats or more to have a built-in fire extinguisher that discharges automatically into the waste bin if a fire occurs. This is because some passengers on long-haul non-smoking flights try to use the aircraft's lavatories as smoking booths.

They block off the smoke detectors, then dispose of their cigarette butts in the waste paper bins. In some cases, they dispose of them before they are fully extinguished, and there is a very real risk of the contents of the waste bin igniting and setting fire to the surrounding trim. The chemicals in the lavatory fluid can also act as a fuel. Therefore, there is a need for an automatic means of extinguishing any fire in the waste bins.

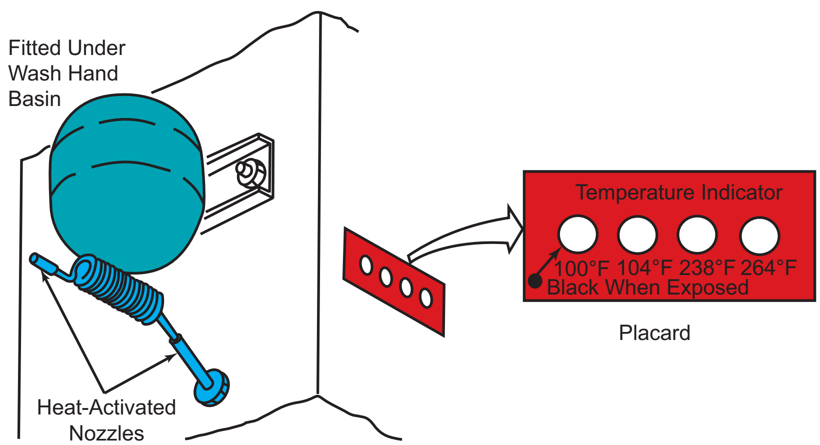


Diagram 14.33 Potty Bottle For Lavatory Fire Suppression

Refer to diagram 14.33. A small capacity Halon fire extinguisher, often termed a **potty bottle**, is fitted behind the waste bin unit in each lavatory. The bottle has a small diameter pipe, which is sealed with a low melting point alloy after it has been charged with the BCF.

In the event of a fire in the waste bin, the heat melts the alloy, and the bottle discharges into the bin. To warn the ground engineers of the temperatures that were obtained before the fire was extinguished, a placard with a series of temperature sensitive silver spots is mounted inside the vanity unit, see right-hand graphic of diagram 14.33. These darken to black as the temperature rises.

WHEEL BRAKE FIRES

For the near future, in answer to questions asked in the JAA ATPL (A) or CPL (A) Aircraft General Knowledge Exam about what medium is used to extinguish a wheel, wheel brake fire, or undercarriage fire on the ground, the best answer would be dry powder/chemical powder.

In reality, the commercial airports do not have these extinguishers available at the stands but mount them on the fire tender. Following the Home Office Fire Service Manual Volume 4, they approach the wheels from directly in front or behind and use a fine spray of water to control, cool, and extinguish the fire. If required, they also pump chemical powder through the spray into the wheel unit. Do not attempt to apply water to a brake unit fire.

Diagram 14.34 After a Wheel Brake Fire

Chapter 15
General Emergency Equipment

ACKNOWLEDGEMENTS

We would like to thank and acknowledge:

For diagrams and assistance with technical information

15.1 15.4 15.5 15.10 15.11 15.12 15.14 15.15 15.16 15.21 15.22 15.23 15.24 15.27	Virgin Atlantic Airways with special thanks to Mr. J. Jasper of Virgin Atlantic's Cabin Crew Training, Horley.
15.17	An unknown American Coastguard on a Lightship in the Pacific Ocean
15.18	Mr. Ashley Gibb
15.19 15.20	Aerzur, 48,BD Gallieni–92137 Issy Les Moulineaux, France
15.25	Mr. Stuart Powney

INTRODUCTION

In the JAA syllabus, Emergency Equipment covers items and regulations governing systems that range from first aid kits to sonar locating beacons. To make the information more manageable, it has been subdivided. As already seen, oxygen has been addressed in Chapter 13 and fire has been addressed in Chapter 14. The remaining items are covered in this chapter.

EMERGENCY LIGHTING

All passenger aircraft must have cabin lighting to enable passengers to embark and disembark. They also require emergency lighting to enable the passengers and crew to evacuate the aircraft in the event of the normal cabin lighting failing after a forced or crash landing.

The emergency lighting can be either completely independent of the aircraft's electrical system or have batteries that are kept charged by the aircraft's system. Whichever system is used, the emergency light must remain illuminated for a minimum of 10 minutes.

If the later system is used, the lights must be capable of functioning in the event of fuselage separation and have a means of preventing the storage batteries from discharging back through the aircraft's system. Whichever system is fitted, the light must be able to be manually operated from either flight deck or cabin.

EMERGENCY LIGHTING SWITCHES

Diagram 15.1 Flight Deck Emergency Light Switch and Indicator Light

As shown in diagram 15.1, the switch on the flight deck must have "on", "armed", and "off" positions with a guard to prevent inadvertent selection of "on" from the "armed" position. A warning light is illuminated when power is applied to the aircraft and the emergency light selector is not in the armed condition. When the flight deck switch is set to "armed", loss of 115 AC power due to the operation of the crash switch caused by abnormal de-acceleration in a crash condition automatically switches the emergency lights on.

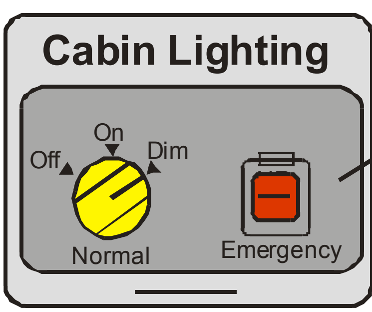


Diagram 15.2 Rear Crew Emergency Light Switch

The lights must also be operable manually from a point in the passenger compartment that is readily accessible to a seated cabin crewmember. This switch is an on–off switch and overrides the flight deck armed position in the event that the cabin crew requires emergency lighting, as diagram 15.2 illustrates.

AREAS OF EMERGENCY LIGHTING

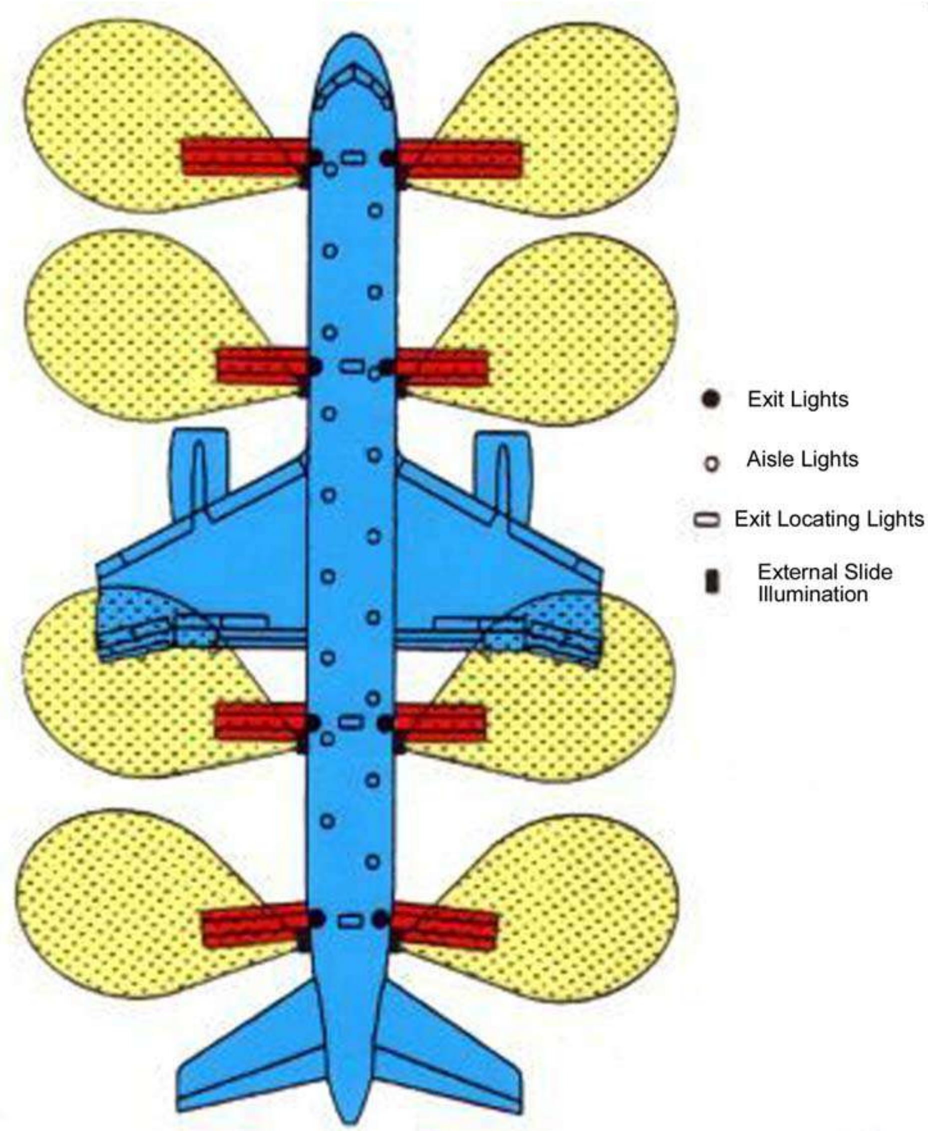


Diagram 15.3 Emergency Lighting

For aircraft certified under JAR 23 and 25 or type certified after 1972, there must be emergency illumination both internally and externally. Diagram 15.3 illustrates this.

INTERNAL EMERGENCY LIGHTING

To meet the JAA requirements for internal emergency lighting, there must be a specific value of lighting at floor level or floor proximity when measured every 40 inches. This is to illuminate the escape route to over wing exits and exit doors, so that on a dark night after leaving a seat, a passenger can visually identify the emergency escape path along the cabin aisle to the first exit or pairs of exits, either forward or aft of the seat.

For electrical floor level or floor proximity lighting, this means that the lights cannot be more than 40 inches apart. Modern aircraft are making use of photo-luminescent strips, which meet the JAA requirements for the lighting levels.

At junctions to exits or pairs of exits, there must be an overhead EXIT sign, so that the passenger can readily identify any exit by its markings and visual features. Again, these can be lit or photo-luminescent signs using the colour code of red letters on a white background or white letters on a red background. Diagram 15.9 illustrates this, as does the section on doors that follows.

EXTERNAL EMERGENCY LIGHTING

To assist passengers evacuating via over wing exits or escape slides, emergency lights are fitted by the exit doors/hatches. These illuminate the general area, as seen in diagram 15.3.

EMERGENCY TORCHES

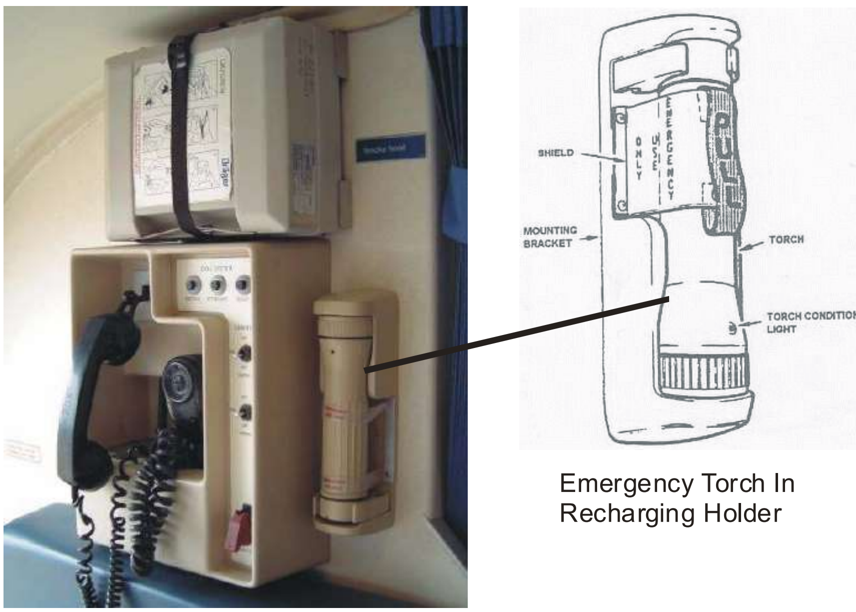


Diagram 15.4 Cabin Crew Station

An electric torch is required for each crewmember and must be readily accessible when seated at their designated stations. Diagram 15.4 shows an emergency torch located adjacent to a cabin crew position. Each torch is held in a recharging holder that is connected to the aircraft’s electrical system and recharges the torch’s battery while the generators are working.

A red light emitting diode (LED) shows the condition of the torch. This flashes red while power to the re-charger is available and the torch is serviceable. A continuous red shows a fault. When the aircraft is shut down, the LED does not illuminate. The torches are fitted with a reed switch and light up on removal from the holder, giving a minimum of 20 minutes of light.

NORMAL/EMERGENCY INTERNAL COMMUNICATIONS

Diagram 15.5 Rear Crew Station Interphone and P.A.

Normal communications between flight crew and cabin crew is via an interphone. Where more than one flight crew is required, they must have a hands-free system. For aircraft with more than 19 passenger seats, an interphone system for the rear crew is required. It takes the form of a telephone hand set and allows communication between cabin crew stations and the flight crew. A public address system that can be heard in all passenger and crew areas is also required for aircraft of more than 19 seats. Diagram 15.5 shows an illustration of an interphone.

MEGAPHONES

To enable the cabin staff of air transport aircraft to control any emergency evacuation (normal public address system failed), there is a requirement for portable battery-powered megaphones to be readily accessible for use by crewmembers during an emergency. The minimum number of megaphones required by JARs is:

Passenger Seating Configuration	Number of Megaphones Required
61 to 99	1
100 or more	2

A passenger aircraft with 61 or more passenger seats, carrying one or more passengers, must not operate unless the megaphones, as per the table above, are carried and serviceable.

An aircraft with less than 60 seats but with two or more passenger decks must also carry a megaphone. If two or more megaphones are required, they should be suitably distributed in the passenger cabin(s) and readily accessible to crewmembers assigned to direct emergency evacuations.

The megaphones should be located where they are readily accessible from the cabin crewmember's assigned seat. However, this requirement does not necessarily require positioning the megaphones such that they are reachable by crewmembers when strapped in. For a JAR 25 certified aircraft, the megaphone storage must have a means of restraining them in the event of a forced or crash landing.

EMERGENCY LOCATION

To assist in locating crashed or ditched aircraft and survivors, the regulations require air transport aircraft to carry emergency locator transmitters, pyrotechnic signal flares, and a sonar-locating beacon. The regulations and the equipment's function appear below.

EMERGENCY LOCATOR TRANSMITTER — ELT

There are different types of ELTs, which appear below. They are designed to alert and aid Search And Rescue (SAR) teams in locating a crash site and/or survivors.

ELT (AF) — AUTOMATIC FIXED ELT

This type of ELT is permanently fixed to the aircraft's structure and is intended to remain with the wreckage after a crash landing.

ELT (AD) — AUTOMATIC DEPLOYABLE ELT

This type of ELT is physically and rigidly attached to the aircraft. In the event of a crash landing or ditching, a crash sensor automatically ejects it and deploys its aerial, so that it starts transmitting immediately. This type of ELT floats.

ELT (AP) — AUTOMATIC PORTABLE ELT

This type of ELT is physically and rigidly attached to the aircraft. In the event of a crash landing or ditching, its crash sensor automatically starts transmitting immediately. This unit must be easily removed from the aircraft and taken with the survivors.

If the unit was using an aircraft mounted aerial, an auxiliary antenna stored in the ELT's case is used. This type of ELT is buoyant for use in ditching situations.

ELT(S) — SURVIVAL ELT

This type of ELT is designed to be removed from an aircraft and activated by the survivors after a crash or ditching, either manually or automatically by immersion in water. It is designed to be attached to the survivor or life raft. This type of ELT floats.

ELT FREQUENCIES AND MINIMUM LEVELS

ICAO sets the frequencies on which ELTs transmit. JAR-OPS 1 dictates the minimum level of ELT equipment, which state:

- Aircraft certified on or after 1 January 2002 cannot operate unless they are equipped with any type of ELT that transmits on both 406 MHz and 121.5 MHz. Aircraft certified on or after this date must have an automatic ELT.
- As of 1 January 2005, aircraft must be equipped with automatic ELTs that transmit on both 406 MHz and 121.5 MHz.
- From 1 January 2005, ELTs must transmit on both 406 MHz and 121.5 MHz.
- For extended flights over water, aircraft must carry two ELT(S**)**.

FLARES

One of the most common types of pyrotechnic carried in aircraft is the combined Day/Night distress signal. This is normally a cylinder with a cap at each end; one for day, the other for night. The day distress signal is an orange smoke. The night signal is a hand-held flare.

SONAR LOCATING BEACONS

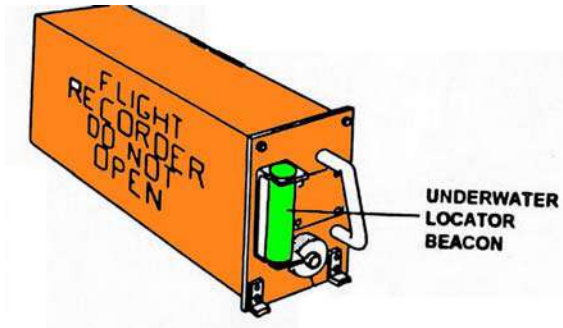


Diagram 15.6 Underwater Locating Beacon

JAR-Ops 1 states that cockpit voice recorders and flight data recorders must have a device to assist in locating them in water. This takes the form of an active sonar-locating beacon, as shown in diagram 15.6. A beacon unit attached to each recorder has a switch activated by water pressure (approx. 50 ft below the surface). This triggers the unit to send out a “ping”, which has a range of approximately 3 miles and lasts for 30 days.

SURVIVAL EQUIPMENT

For aircraft making flights over areas of the globe that would be especially difficult for search and rescue, suitable survival equipment is required. The definition of especially difficult for search and rescue is:

- 120 minutes from a suitable landing site for an aircraft that can fly with a critical power unit inoperative.
- 30 minutes for all others.

EMERGENCY EXITS

The term emergency exits include main entry doors as well as over wing escape hatches and ventral escape hatches. For structural reasons, all these doors and hatches have curved corners. However, these curves (radii) are limited to ease egress. The size, number, and location of these are detailed in JARs and with their different type designation given below. Diagram 15.7 shows them in approximate scale for comparison.

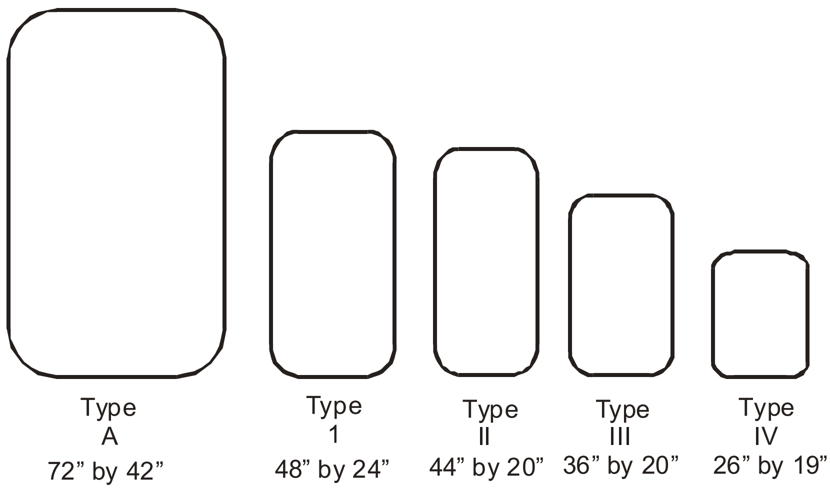


Diagram 15.7 Comparison of Exits

TYPE A

These are the normal entry and exit doors of air transport aircraft, which are a floor level exit with a rectangular opening of not less than 42 inches (1.07 m) wide by 72 inches (1.83 m) high.

TYPE I

These are smaller floor level doors with a rectangular opening of not less than 24 inches (61 cm) wide by 48 inches (1.22 m) high.

TYPE II

These are smaller rectangular hatches. The minimum size is 20 inches (50.8 cm) wide by 44 inches (1.12 m) high. These exits must mount at floor level unless fitted as over wing escape hatches.

If these hatches are used as over-wing escape hatches, the lower lip of the exit must not be more than 10 inches (25.4 cm) from the cabin floor level. This is termed the **step-up** height. The step-down onto the wing or any other horizontal surface must not be more than 17 inches (43.2 cm).

TYPE III

These are smaller rectangular hatches with a minimum size of 20 inches (50.8 cm) wide by 36 inches (91.4 cm) high and a step-up of not more than 20 inches (50.8 cm). If the exit is located over the wing, the step-down outside the aeroplane may not exceed 27 inches (68.6 cm).

TYPE IV

These hatches are for use as wing escape hatches. They are rectangular with a minimum opening of 19 inches (48.3 cm) wide by 26 inches (66 cm) high. The step-up must not be more than 29 inches (73.7 cm) and a step-down not more than 36 inches (91.4 cm).

TYPES AND NUMBERS OF PASSENGER EXITS

The number of passenger seats determines the minimum number of passenger emergency exits and their type. See the tables below for passenger seating configurations of 1 to 299 seats.

Passenger Seating Configuration (Crewmembers seats not included)	Emergency exits for each side of the fuselage			
	Type I	Type II	Type III	Type IV
1 to 9	-	-	-	1
10 to 19	-	-	1	-
20 to 39	-	1	1	-
40 to 79	1	-	1	-
80 to 109	1	-	2	-
110 to 139	2	-	1	-
140 to 179	2	-	2	-

Additional exits are required for passenger seating configurations greater than 179 seats in accordance with the following table:

Additional emergency exit (Each side of fuselage)	Increase in passenger seating configuration allowed
Type A	110
Type I	45
Type II	40
Type III	35

For aircraft with more than 299 passenger seats, each emergency exit in the side of the fuselage must be either a Type A or a Type I. A passenger seating configuration of 110 seats is allowed for each pair of Type A exits, and a passenger seating configuration of 45 seats is allowed for each pair of Type I exits.

DISTANCE BETWEEN PASSENGER EXITS

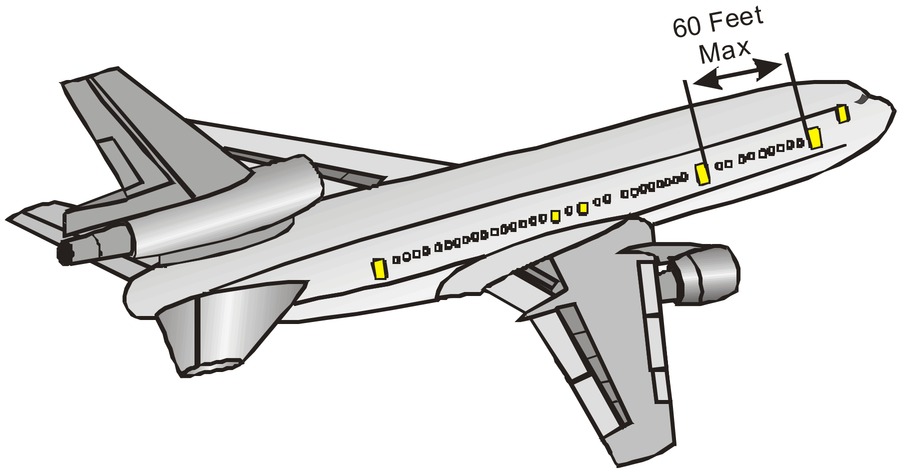


Diagram 15.8 Maximum Distance Between Emergency Exits

Where more than one passenger emergency exit on each side of the fuselage is required, these exits are not to be more than 60 ft (18.29 m) from the next adjacent passenger emergency exit on the same side of the same deck of the fuselage. The distance is measured parallel to the aeroplane's longitudinal axis between the nearest exit edges, as seen in diagram 15.8.

DITCHING EMERGENCY EXITS

As there is a risk that an aircraft may have to ditch, the location of the exits has to take the water line at which the aircraft will float into account.

DOOR OPENING

Diagram 15.9 Type I Door Locking Lever and Cover

Each emergency exit must be able to be opened from both the inside and the outside of the aircraft. When the door is fully open and stowed, it must not obstruct the opening to the outside. Emergency exits must be capable of being opened when there is no fuselage deformation, the aircraft is in the normal ground attitude, and in each of the attitudes corresponding to collapse of one or more legs of the landing gear.

The means of opening emergency exits must be simple and obvious and must not require exceptional effort. The door must be fully open within 10 seconds of the opening mechanism being actuated. If a single power-boost or single power-operated system is the primary system for operating more than one exit in an emergency, each exit must be capable of meeting the requirements above in the event of failure of the primary system. To comply, door opening mechanisms have a manual operation.

Each door and emergency exit must have a means to lock it and safeguard against its opening in flight, either inadvertently by persons or as a result of mechanical failure. For doors that initially open outward, there must be a means for direct visual inspection of the locking mechanism by crewmembers and a window, so that the crew can verify that the area into which the door opens is clear. Diagram 15.9 illustrates this.

Diagram 15.10 Door Indication on ECAM Lower Screen

There must be a visual warning to alert the flight crew of any external passenger or service doors that are not closed and locked. The warning system must be made so that any failure does not result in a false closed and locked indication. Aircraft fitted with ECAM systems use the lower screen, which gives system information to display the condition of the doors, as seen in diagram 15.10. Aircraft not equipped with ECAM use the central warning panel displaying a red light.

FLIGHT CREW EMERGENCY EXITS

Diagram 15.11 Flight Crew Emergency Exits–Escape Hatch and Removable DV Window

Flight crew emergency exits are required on the flight deck of all aircraft that have a passenger seating capacity of 21 or more. To comply with this requirement, there must be an exit each side of the aircraft or a top hatch. The exits must not be less than 19 by 20 in.

For aircraft of less than 21 passenger seats, a flight crew escape hatch is only required if the passenger emergency exits do not offer convenient and readily accessible means of evacuation. For some aircraft, the pilots escape route is via a removable or sliding side window, which also provides the direct vision window that the regulations require. On large aircraft with these exits, the crew are provided with escape ropes. See diagram 15.11 for flight crew escape exits.

AIRCRAFT EVACUATION TIME

To comply with the safety regulations, each aircraft with more than 44 passenger seats must be able to be evacuated in 90 seconds using half of the available emergency exits, internal emergency lighting, and escape slides. Commuter aircraft certified under JAR 23 must also meet the criteria of evacuating all the passengers within 90 seconds from the exits on one side of the aircraft.

ESCAPE SLIDES

These slide types apply for aircraft type certificated on or after 1 April 2000.

Diagram 15.12 Upper Deck Slide of a 747

Escape slides speed evacuation and prevent passengers and crew from being injured by jumping down from an exit, as seen in diagram 15.12. The regulations require escape slides to be fitted to each emergency exit door where the distance between the door-sill (sill height) and the ground is more than 6 ft (1.83 m):

- With the landing gear extended or,
- After the collapse of one or more gear legs would be more than 1.83 m above.

All slides fitted to Type A doors must have at least two parallel exit lanes capable of allowing simultaneous use by two rows of evacuees. All other exits must have slides with at least one lane.

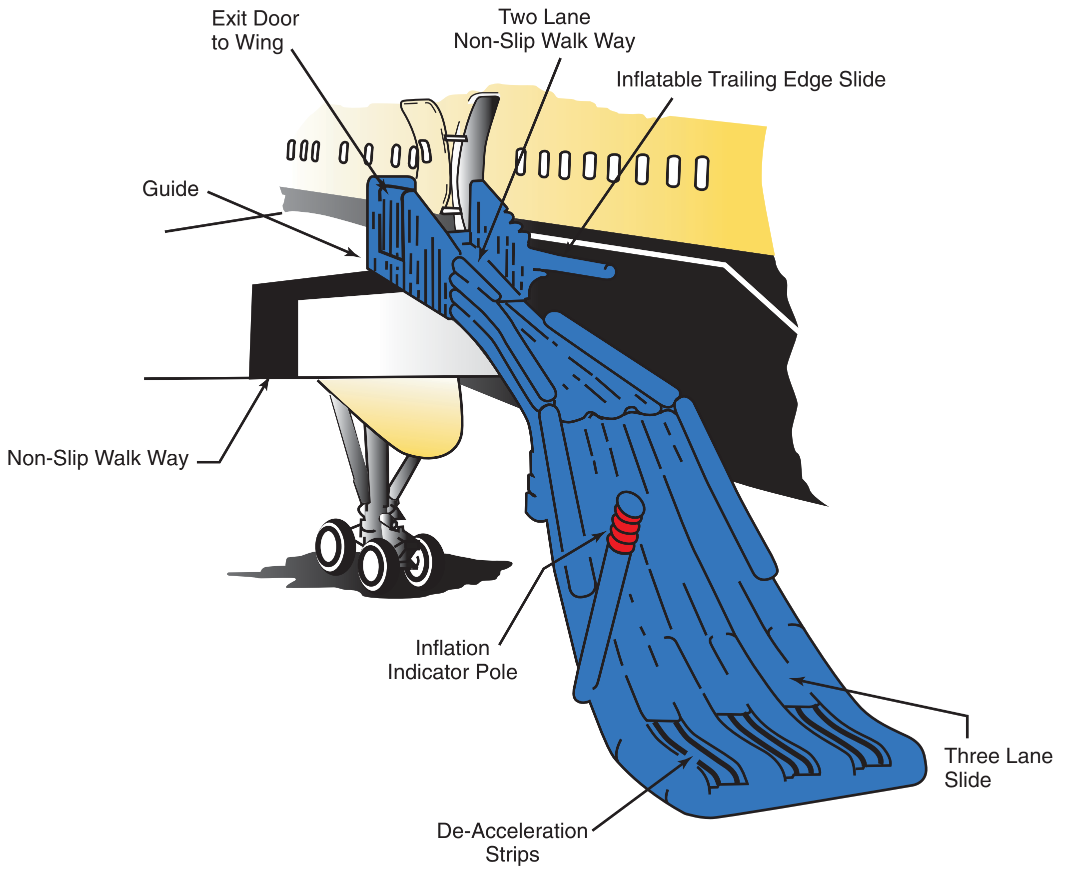


Diagram 15.13 Over Wing Escape Slide

For hatches and doors that exit onto the wing, the escape route is normally via the trailing edge as this is lower than the leading edge. If the distance from the ground to the trailing edge of the flap in the take-off or landing position, whichever is the higher, is more than 6 ft (1.83 m) when the aircraft is on its landing gear, an escape slide must be fitted. See diagram 15.13.

Where escape slides are not required by the height of the trailing edge, an escape route must be established and covered with a slip resistant surface. For a Type A exit, this must be 42 inches (1.07 m), and at all other passenger emergency exits at least 2 ft (61 cm) wide.

An escape slide must be automatically deployed when the emergency exit it serves is opened from inside. Each slide must be fully inflated within 10 seconds of its initiation. It must be capable of being correctly deployed into a 25 kt wind directed from the most critical angle to its deployment and, with the assistance of only one person, to remain usable after full deployment to evacuate occupants safely to the ground.

ARMING/DISARMING SLIDES

Where the emergency exit is also a normal passenger entrance/exit or service door, slides fitted to these doors must be able to be disarmed for normal ground use and rearmed for flight before the aircraft taxies out of the stand. The arming/disarming process is carried out by the cabin crew on the command of “Doors to automatic/Doors to manual.”

For small medium-range twinjet aircraft like Boeing 737s, the slides are contained in an upside down pocket which is part of the cabin door trim. On the command of “Doors to automatic”, a member of the cabin staff connects a steel rod, termed a **GIRT BAR** into two catches, one each side of the door at floor level. The inboard end of the slide is attached to the girt bar. If the door is opened with the girt bar in the armed condition, the slide is pulled from its container. The act of falling pulls an inflation lanyard and starts the inflation process. To disarm the slide, the girt bar is removed from the floor fastenings and stowed in clips on the door, allowing the door to be opened without slide activation.

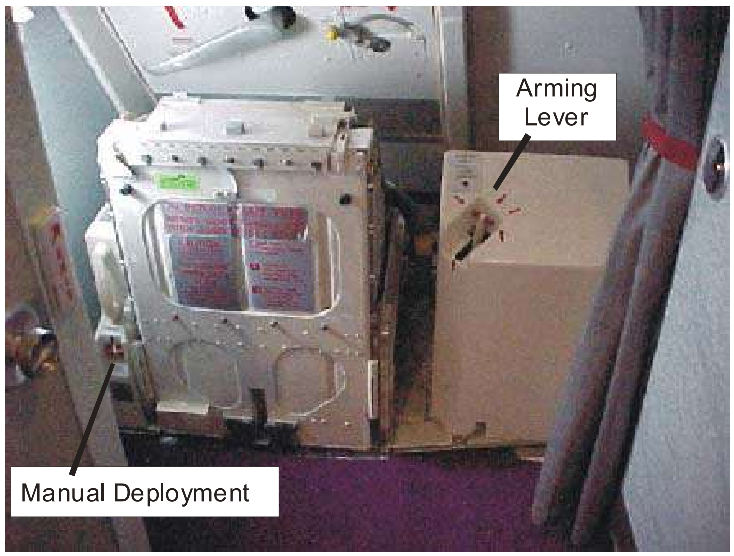


Diagram 15.14 Older Slide

Older designs of slide mechanism for larger wide-bodied aircraft (such as the Boeing 747 Classic) stored the slide in a container that was pulled across the doorway for flight and pushed into a bulkhead stowage space to allow passengers to embark and disembark, as seen in diagram 15.14. The slide had an arming lever. When placed in the armed position, the slide would deploy if the door was opened. In the event of the slide failing to deploy automatically, a manual control allowed a member of the cabin staff to deploy the slide.

Diagram 15.15 Airbus Door

Diagram 15.15 shows how modern slide systems stow the slide in a container that is part of the door. When the doors are set to automatic, the girt bar is locked to the door threshold. See diagram 15.16 for a sequence of a slide's inflation from door opening to full inflation (10 seconds).

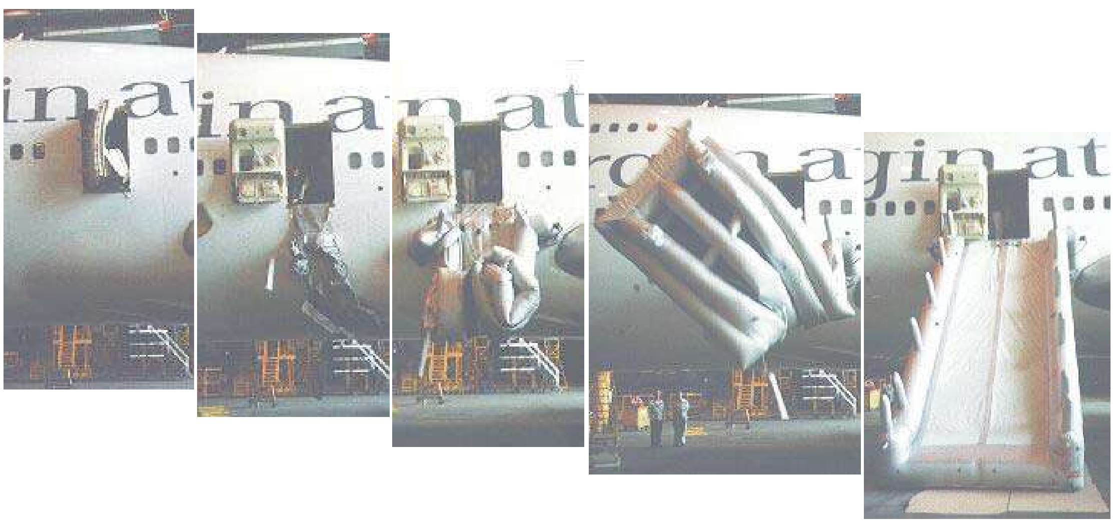


Diagram 15.16 Full Slide Deployment within 10 Seconds of the Door Opening

FLIGHTS OVER WATER

Diagram 15.17 Boeing Stratocruiser Ditching In the Pacific

The level of floatation survival equipment required is determined by:

- The speed of the aircraft
- The distance from the shore
- The distance from a suitable landing site
- Whether approach or departure from an airfield is over water
- The certification of the aircraft

REQUIREMENT FOR LIFE JACKETS

Life jackets are required:

- For all amphibians or float planes operating on water.
- For all land planes when flying over water and at a distance of more than 50 nautical miles from the shore.
- Where in the event of a mishap during an approach or take-off from an airport, there would be a likelihood of a ditching in water.

REQUIREMENT FOR LIFE RAFTS

Life rafts are required for flights over water:

- For all non-ETOPS aircraft, the lower distance of either 100 nm or 30 min at cruising speed from a suitable landing ground.
- For ETOPS certified aircraft, the lower distance of either 400 nm or 120 min flight at cruising speed.

LIFE JACKETS

Diagram 15.18 Pax Life Jacket

To meet JAA and CAA specifications, each person on board must be equipped with an inflatable life jacket that must be stowed in a position easily accessible from the seat or berth of the person for whose use it is provided. Diagram 15.18 illustrates this type of life jacket. The jacket must be inflated by a compressed gas (normally CO2), have a means of oral inflation, and be equipped with a water activated (survivor) light. It is standard to equip each life jacket with a whistle. Seat cushions are not acceptable to the JAA as life preservers.

Diagram 15.19 Baby Float

Life jackets for infants may be substituted by other approved floatation devices such as floating cribs. Diagram 15.19 shows a Baby Float by Aerzera. These must have gas inflation, aural top up, and a survivor locator light. They also include a strap to link the floatation aid to an adult.

LIFE RAFTS

Diagram 15.20 Life Raft

Where life rafts are carried internally, they must be stowed near exits through which they can be launched and inflated in the event of an unplanned ditching. Where life rafts are stowed externally, such as beneath panels in the wing root fairings, they must be able to be released and inflate either automatically or by controls within the aircraft.

Externally mounted life rafts must be attached to the aircraft by a static line that can easily be released. To prevent the raft or the occupants from endangerment, the connection line must break in the event that the aircraft sinks before it is released.

The number of life rafts carried depends on the seating capacity of each raft and the seating capacity of the aircraft. The passenger to life raft ratio must be sufficient that, should a life raft fail, the remaining rafts could accommodate all the passengers. For example, a 100-seater aircraft using rafts with a maximum capacity of 25 persons must carry five rafts.

All life rafts must be inflated with compressed gas (normally CO_2) and have the means of being manually inflated to maintain the gas pressure. Each raft must have a cover that is coloured high visibility orange/red or yellow, separate buoyancy chambers so that a failure in one does not result in the loss of the others, a water activated survivor light, and a length of rope to allow the rafts to be attached to each other, as diagram 15.20 shows.

EQUIPMENT THAT A LIFE RAFT MUST CARRY

Each life raft must carry the following equipment:

- A pump
- Sea anchor
- Lifelines around the raft
- Water resistant torch
- First aid kit
- Food
- Water
- Survivor light

The only alteration to the equipment requirements for life rafts is that it must carry paddles if it has a capacity of six or less.

SLIDE RAFTS

Diagram 15.21 Slide Rafts

Diagram 15.21 shows how in wide-bodied aircraft (ETOPS aircraft), slide rafts replace the life rafts. Here the escape slides fitted to the main fuselage exit doors (not over wing, ventral, or tail cone exits) double up as the aircraft's life rafts. Slide rafts as per life rafts have to have sufficient capacity that all the passengers can be accommodated on the remaining slide rafts should one fail to function. They must have an orange/red-yellow high visibility cover that protects the occupants when at sea in a 35 kt wind. This cover can be integral or attached by the occupants.

MANUAL DEPLOYMENT

Slide rafts must have both automatic and back up manual inflation controls. Diagram 15.22 shows manual inflation after the automatic system failed. The slide raft must be securely attached to the aircraft's structure, so that it can act as a slide on land. This connection via a girt bar must be easily disconnected, so that the slide raft can float free. However, when the girt bar is released, the slide raft must remain attached to the aircraft via a static line that can be simply disconnected or break in the event of the aircraft sinking.

Diagram 15.22 Manual Inflation of Slide Raft after Automatic System Failed

Diagram 15.23 Slide Raft Inflating

Slide rafts as per slides must be able to deploy into a 25 kt wind and inflate within 10 seconds of the door opening or the manual inflation control being operated. Some slide/slide raft combinations require a transition from slide mode to raft mode. If such a conversion is needed by inflating extension tubes, these must have a separate control and inflate within 10 seconds of its operation.

Slide rafts also carry survival equipment but to a lesser degree than life rafts. This is normally stowed in waterproof containers that are released when the girt bar is disconnected. This ensures that they do not impede normal escape use when on land. Below is the minimum equipment list.

Each slide raft must carry the following equipment:

- A pump
- Sea anchor
- Lifelines around the slide raft
- Water resistant torch
- First aid kit
- Food
- Water
- Survivor light
- Emergency knife

INOPERATIVE EXIT DOOR, SLIDE, OR SLIDE RAFT

The JAA Minimum Equipment Lists (MELs) allow for one emergency exit door per deck to be inoperative provided it is not practical to repair it before the next flight. The regulation limits the time that the operator can use the aircraft in this condition to a maximum of 72 hours and five passenger flights before it is repaired. Where an exit door is placed unserviceable, the operator must consider the need to reduce the number of passengers, as the slide or slide raft associated with that door must also be considered to be unserviceable.

Diagram 15.24 Unserviceable Door/Slide

The aircraft may be operated with one slide unserviceable, limited to the same duration as above, provided that the door is also considered unserviceable. Doors that are unserviceable must be marked both internally and externally. Diagram 15.24 shows how an unserviceable door is marked internally with orange tape and a no exit sign. A no entry sign externally will be stuck to the door adjacent to the handle. The slide's automatic arming lever is placed in the ground position, locked, and flagged.

In the event of the cabin crew being unable to disarm a slide after landing, the door must be placed unserviceable, the ground crew warned, and a member of the cabin staff stationed to prevent anybody from trying to open the exit. The aircraft may continue to be operated in this condition for the limiting period provided these precautions are observed and a no entry sticker is placed adjacent to the external door handle.

RESCUE

EXTERNAL MARKINGS OF DOORS AND HATCHES

Diagram 15.25 External Side Door Markings

External doors that are located in the fuselage sides and are required to be opened from the outside as emergency exits must have a 2-inch colour band outlining the exit marked on the fuselage. This band of paint must be a contrasting colour to the aircraft’s livery to make the door readily discernable from the surrounding fuselage, as seen in diagram 15.25.

All exits and their instructions, other than those in the side of the fuselage, must be conspicuously marked in red or bright chrome yellow if the background colour is such that red is inconspicuous. When the opening means is located on only one side of the fuselage, a conspicuous marking to that effect must be provided on the other side.

BREAK IN MARKINGS

To enable rescuers to cut through the fuselage without endangering themselves with high-pressure pipes or high-voltage cables, break in or cut in areas are marked on the external surface. These show the rescuers where to cut as the area behind them is kept clear of hazards and guide them away from cutting into further supporting structure such as bulkheads. The area marked out is NOT structurally weakened in any way.

The markings have to conform with the regulations of being conspicuously marked in red or bright chrome yellow if the background colour is such that red is inconspicuous. The thickness and length of the lines and the maximum distance apart is given in diagram 15.26.

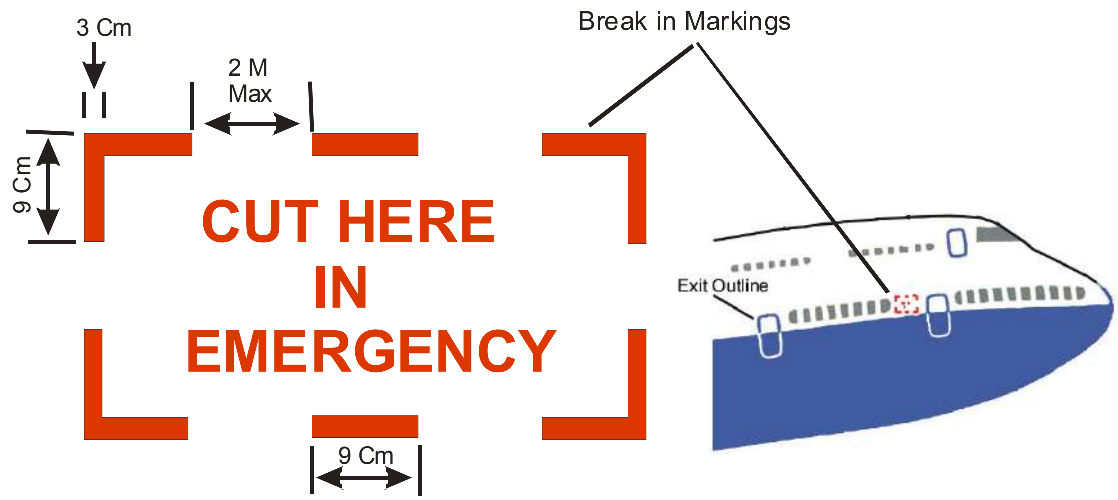


Diagram 15.26 Break In Markings

FIRST AID KITS

Air transport aircraft are required to carry first aid kits. These are for use by the crew and survivors. The minimum number of kits to be carried is as follows:

Number of pax seats	Number of first aid kits required
0 – 99	1
100 – 199	2
200 – 299	3
300 or more	4

Operators are responsible for ensuring that the kits are inspected periodically and replenished as required to ensure that the minimum number of kits are serviceable prior to flight.

The MELs state that any first aid kits in excess of those required may be inoperative, but if more than one first aid kit is required, only one of the required kits may be incomplete for a maximum of two flight days.

EMERGENCY MEDICAL KIT

Emergency medical kits are required for aircraft with 30 or more seats, which are operating at more than 60 minutes (at cruise speed) from an airport where medical assistance could be expected to be available. The Commander of the aircraft is responsible for ensuring that only medically qualified personnel administer drugs, and the emergency medical kit should be kept on the flight deck but must be under secure conditions where practicable.

The kits must be inspected periodically to ensure that their contents are in a serviceable condition. The contents of the kit must be listed in two languages, one in English, and should include the information on the effects and side effects of the drugs being carried.

The operator is responsible for ensuring that a flight does not commence unless the emergency medical kit (doctor's kit) is serviceable where it is required. The MELs state that any first aid kits in excess of those required may be inoperative.

CRASH AXES AND CROWBARS

An operator shall not operate an aeroplane with a maximum certificated take-off mass exceeding 5700 kg or having a maximum approved passenger-seating configuration of more than nine seats unless it is equipped with at least one crash axe or crowbar located on the flight deck. If the maximum approved passenger-seating configuration is more than 200, an additional crash axe or crowbar must be carried and located in or near the most rearward galley area. Crash axes and crowbars located in the passenger compartment must not be visible to passengers.

FIRE-RESISTANT GLOVES

Although not required by JAR-Ops 1, fire-resistant gloves are carried on many aircraft. These are normally manufactured in Kevlar in the form of long gauntlets, which protect the hands and forearms from heat and sharp edges. These gloves are stored with the crash axe/crow bar and fire extinguishers out of sight of the passengers.

SMOKE HOODS

Diagram 15.27 Smoke Hood Location

To enable the crew to organise the evacuation of passengers from a smoke-filled aircraft, smoke hoods are used. These are stored adjacent to the crew stations. Refer to diagram 15.27. Smoke hoods fit over the wearer's head and shoulders, protecting the wearer from the effects of heat and smoke while leaving the arms free. The hoods have a chemical oxygen supply that enables the wearer to operate for 20 minutes in a smoke-filled environment.