AP3456
The Central Flying School (CFS) Manual of Flying

Volume 4 Aircraft Systems

CHAPTER 1 - HYDRAULIC SYSTEMS

Introduction

1. Hydraulic power has unique characteristics which influence its selection to power aircraft systems instead of electrics and pneumatics, the other available secondary power systems. The advantages of hydraulic power are that:

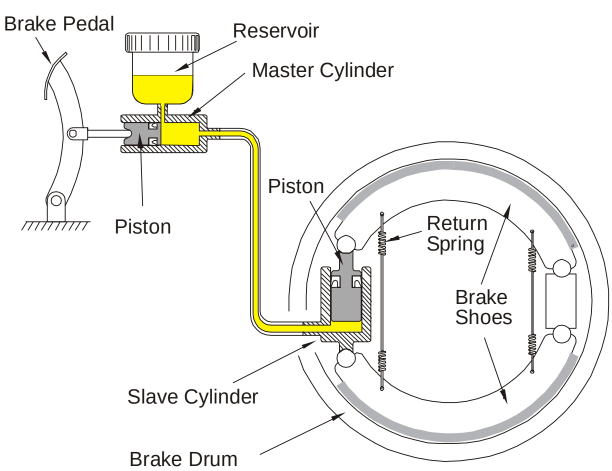
a. It is capable of transmitting very high forces.
b. It has rapid and precise response to input signals.
c. It has good power to weight ratio.
d. It is simple and reliable.
e. It is not affected by electro-magnetic interference.

Although it is less versatile than present generation electric/electronic systems, hydraulic power is the normal secondary power source used in aircraft for operation of those aircraft systems which require large power inputs and precise and rapid movement. These include flying controls, flaps, retractable undercarriages and wheel brakes.

Principles

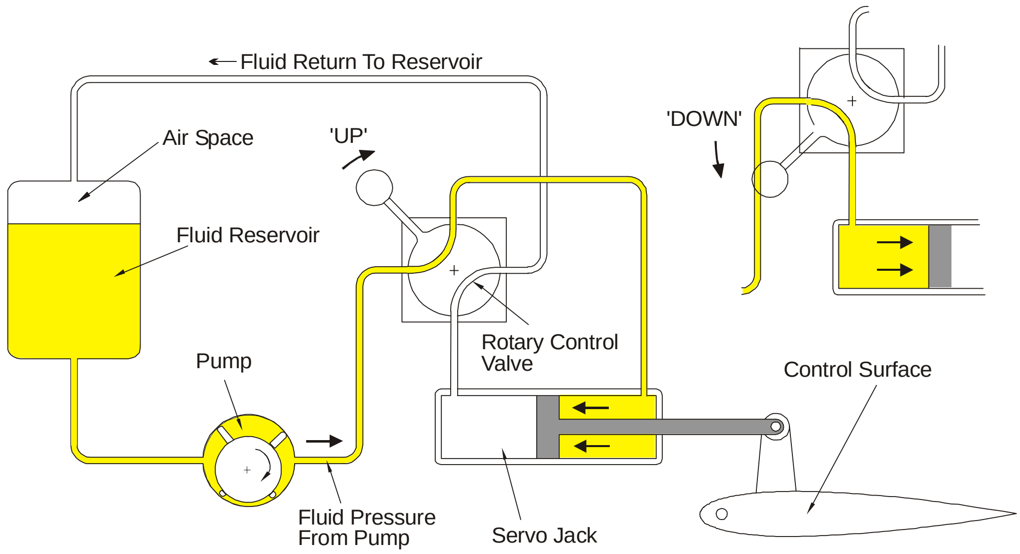
2. **Basic Power Transmission**. A simple practical application of hydraulic power is shown in Fig 1 which depicts a closed system typical of that used to operate light aircraft wheel brakes. When the force on the master cylinder piston is increased slightly by light operation of the brake pedals, the slave piston will extend until the brake shoe contacts the brake drum. This restriction will prevent further movement of the slave and the master cylinder. However, any increase in force on the master cylinder will increase pressure in the fluid, and it will therefore increase the braking force acting on the shoes. When braking is complete, removal of the load from the master cylinder will reduce hydraulic pressure, and the brake shoe will retract under spring tension. The system is limited both by the relatively small driving force which in practice can be applied to the master cylinder and the small distance which it can be moved.

4-1 Fig 1 Simple Closed Hydraulic System



3. **Pump-powered Systems.** These limitations can be overcome by the introduction of a hydraulic pump. Fig 2 shows a simplified pump-powered system in which a control valve transmits pressure from the pump to the hydraulic jack.

4-1 Fig 2 Simplified Pump-powered Hydraulic System

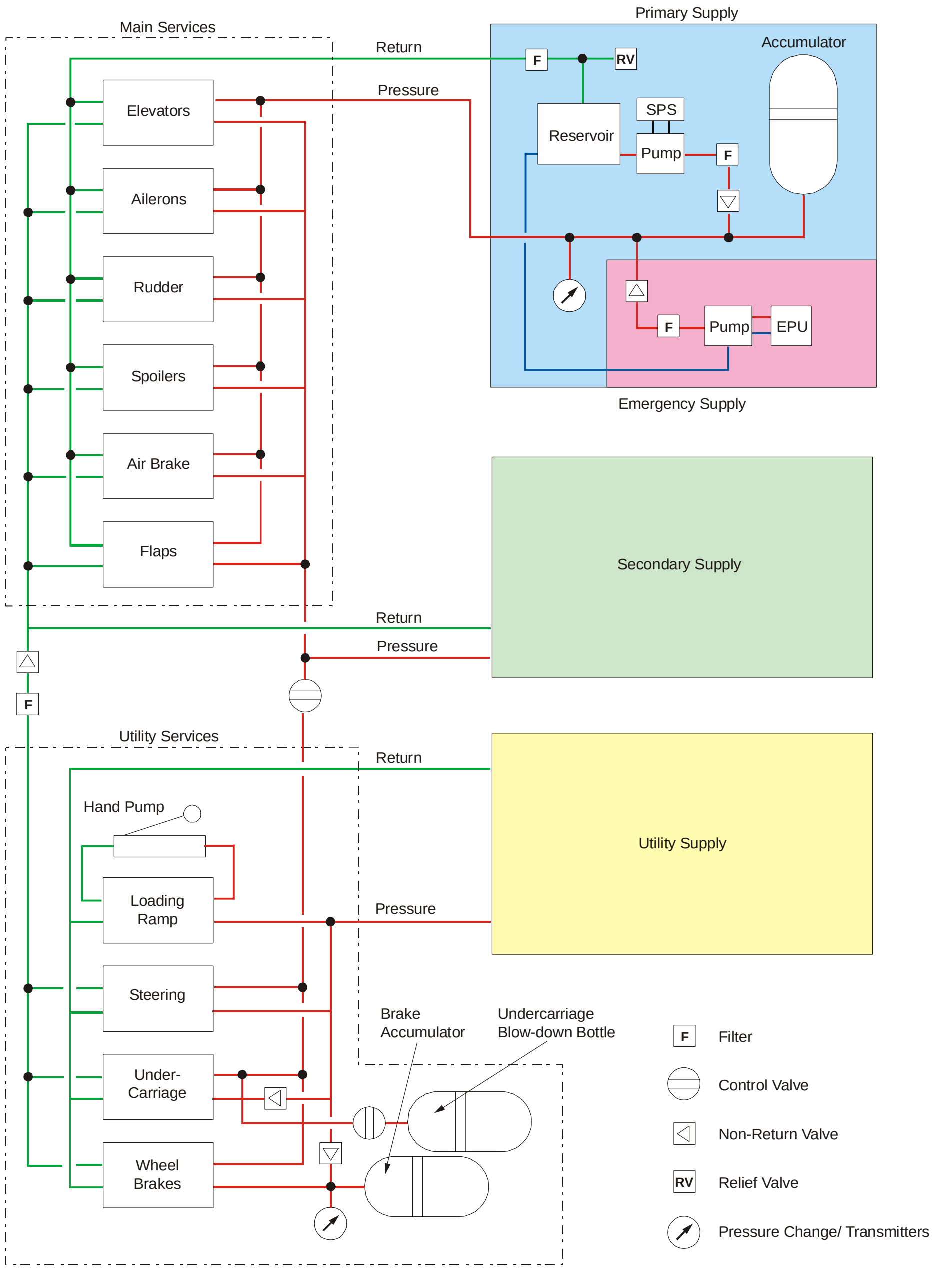


A 'down' selection at the valve causes hydraulic pressure to be fed to the 'down' side of the jack and the pump will work to maintain system pressure during and after jack travel. Fluid from the unpressurized side of the jack will be pushed through the return part of the system circuit back into a reservoir. When 'up' is selected, hydraulic pressure is removed from the jack 'down' side and applied to the 'up' side; fluid displaced by the subsequent retraction of the jack is returned to the reservoir. Within the strength and size limitations of its components, the force transmitted by the system is now effectively limited only by the pressure which the pump can generate; the distance over which the jack can expand or contract is limited by the volume of the fluid.

Typical System

4. To maintain the integrity and reliability of hydraulic systems which power ancillary services fundamental to aircraft airworthiness, power sources for the primary flying controls are duplicated. A typical arrangement is for one of the sources to be dedicated to the primary flying controls and the other to a wider range of services. In transport aircraft, a third hydraulic source is sometimes provided to operate those systems not essential to flight such as undercarriages, brakes and doors. The provision of a fourth power source for emergency use, and the cross-coupling between sources, maintain power to essential services even in the event of two power sources failing. Terminology for this arrangement of systems varies from aircraft type to type; however, the source dedicated solely to powering the flight control units is usually termed the 'Primary System', whilst 'Secondary System' is used to describe the system providing flight control back-up and powering other services. 'Utility' or 'Auxiliary' is applied to the third system whilst the fourth is known as the 'Emergency' or 'Back-up System'. A schematic diagram for a transport aircraft hydraulic system is shown at Fig 3. The function of components typical to most systems is described in the following paragraphs.

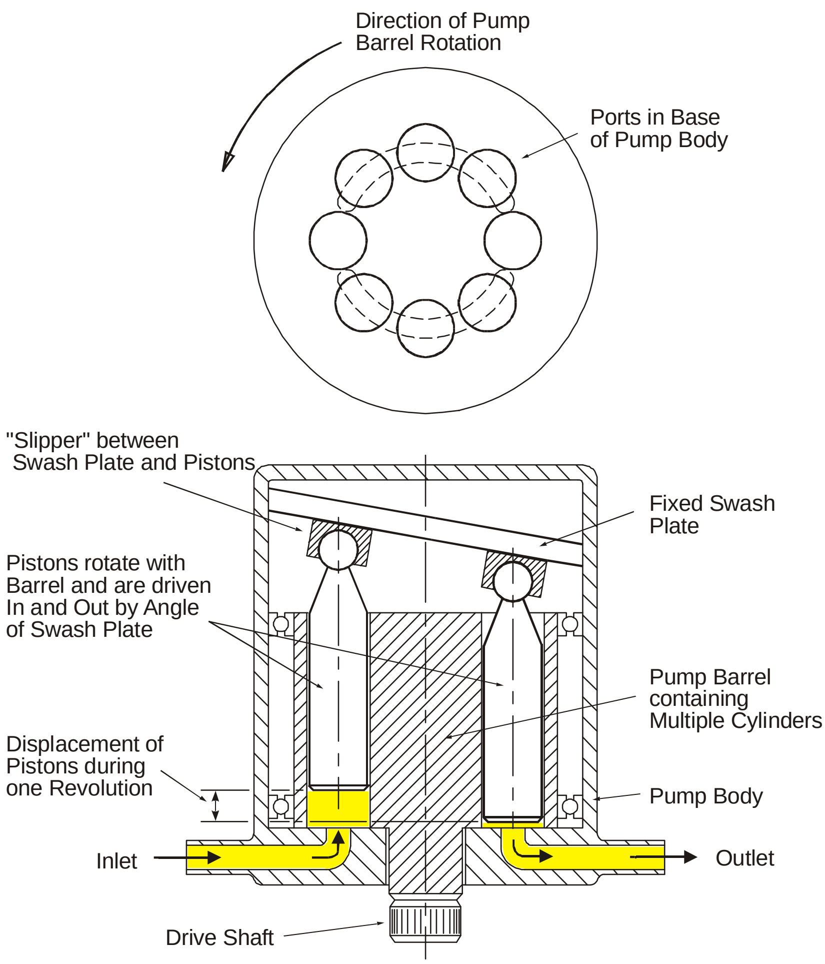
4-1 Fig 3 Typical Hydraulic Power System



System Components

5. **Pumps**. The majority of engine or motor driven pumps are positive displacement, rotary swash plate types, having up to 10 axial pistons and cylinders contained in a barrel which is splined to the drive shaft. Each piston terminates with a ball and slipper (or shoe). The slipper bears against the swash plate surface, the angle of which determines displacement and direction of the flow relative to rotation. As the barrel rotates, the distance between the swash plate and pump body increases and decreases throughout each revolution. The pistons are driven in and out of the cylinders, drawing in fluid at low pressure at the open end of the stroke and expelling it at high pressure at the closed end, as shown in Fig 4.

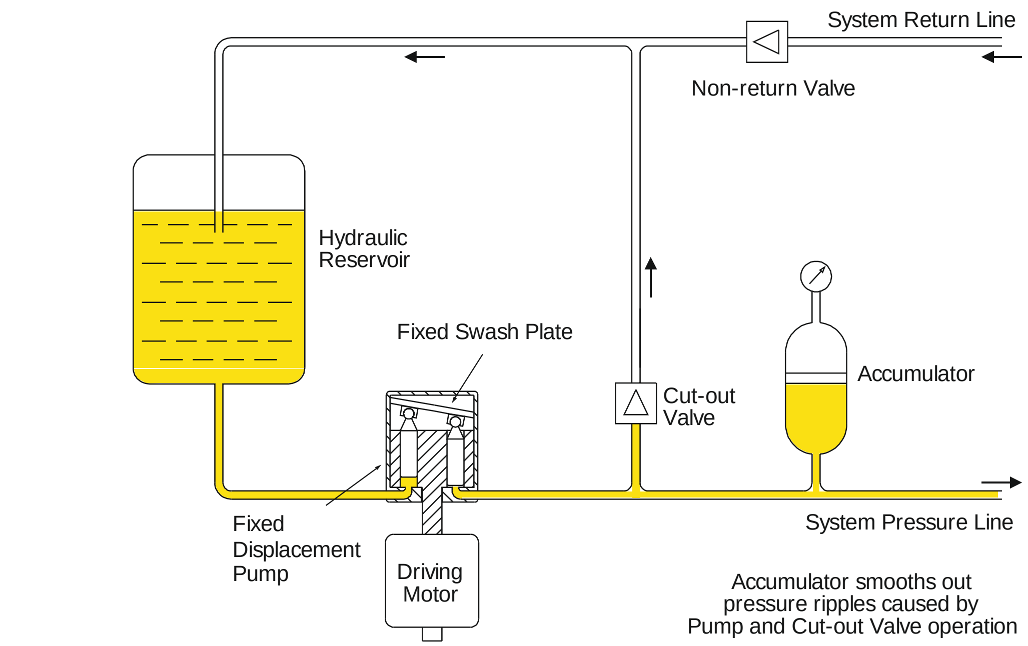
4-1 Fig 4 Principle of a Swash Plate Pump



Two basic variations of this type of pump are commonly used in aircraft systems; one produces a constant volume output, relying upon other components in the system to control both pressure and volume, whilst the other is self-regulating, automatically varying its output to meet system demands.

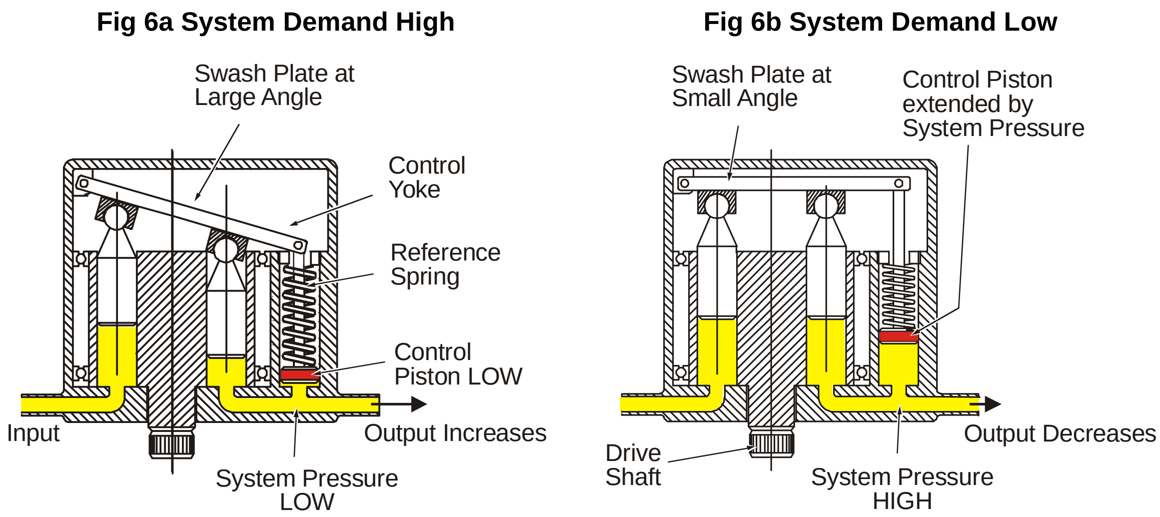
6. **Fixed Displacement Pumps**. Fig 5 shows a fixed displacement pump and the associated components needed to control system conditions. Fixed displacement pumps absorb constant driving power whatever the output demand; when pressure in the system reaches an upper limit, a cut-out valve allows fluid to bypass the pressure line and flow back to the reservoir. Because large volumes of high-pressure hydraulic fluid are therefore constantly being circulated, greater attention must be paid in system design to cooling the fluid to maintain it within design temperature limitations.

4-1 Fig 5 Fixed Displacement Pump and Control System



7. **Variable Displacement Pumps**. Although variable displacement pumps are more expensive, and cost more to maintain, they allow simplification of the total system and they are therefore more usually chosen for Primary and Secondary systems. Fig 6 shows such a pump.

4-1 Fig 6 Variable Displacement Hydraulic Pump

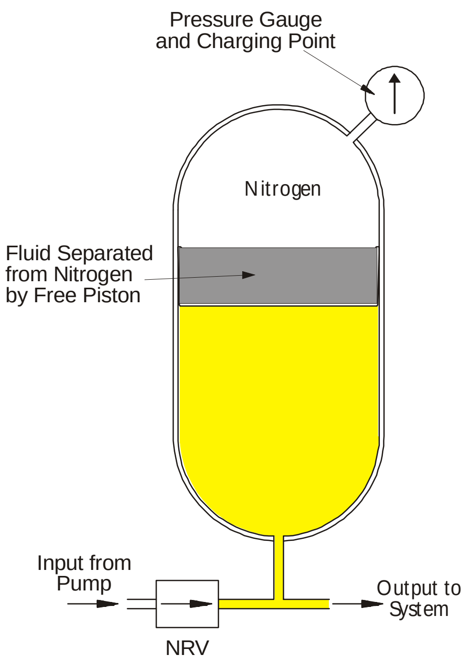


Its operation is similar to that of the fixed displacement pump, but the angle of the swash plate is variable and is changed automatically during operation by a device sensitive to system pressure. As the swash plate angle varies, so does the stroke of the pistons and the output of the pump. Thus, when system pressure drops as power demands on the pump are increased, the output of the pump is increased to match the new demand. When system pressure increases, as all demands are satisfied, the pump output is reduced, and the pump absorbs less power.

8. **Hand Pumps**. Some aircraft are fitted with a hand operated, positive displacement, linear pump for use on the ground. Its operation is usually restricted to pressurizing systems sufficiently for opening and closing doors and canopies, and for lowering and raising ramps. The aircraft Auxiliary Power Unit or a Ground Power Unit is used if more extensive use of the hydraulic system must be made on the ground.

9. **Accumulators**. As illustrated in Fig 5, hydraulic systems include an accumulator, the purpose of which is to absorb shocks and sudden changes in system pressure. A typical nitrogen filled hydraulic accumulator is shown in Fig 7.

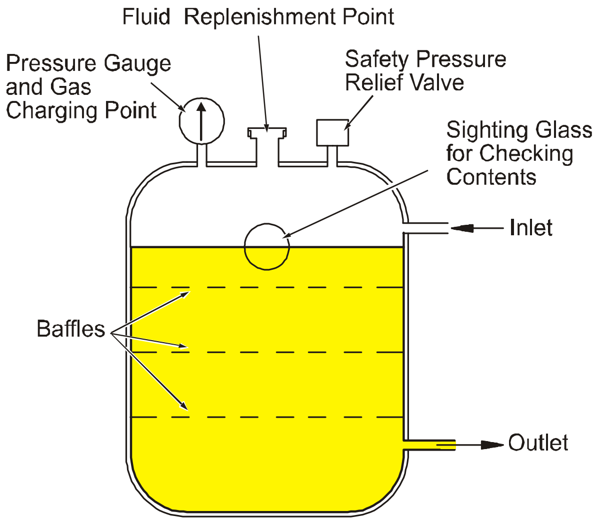
4-1 Fig 7 Typical Hydraulic Accumulator



Compressibility of the nitrogen allows the accumulator to absorb and smooth out the pressure ripples caused by pump operation and also the sudden changes in pressure caused by operation of components such as jacks and valves. It also acts to maintain pressure, to the limit of its piston movement, when the pump ceases to operate. This facility is used, for example, to maintain aircraft parking brake pressure for long periods. When the hydraulic system is not pressurized by the pumps, the gas pressure is typically 70 Bar (1,000 psi).

10. **Reservoir**. Hydraulic systems require a reservoir in which the fluid displaced when the servo jacks are retracted is stored until required again. Obviously, the capacity must be designed to accommodate fluid displaced when all the system jacks are retracted simultaneously. The reservoir performs the secondary functions of cooling the fluid and allowing any air absorbed to separate out. The construction of a typical reservoir is shown at Fig 8.

4-1 Fig 8 Construction of a Reservoir



Reservoirs are usually pressurized either with nitrogen or by system hydraulic pressure acting on a piston. This pressure, of between 3 and 7 Bar, prevents the fluid foaming and provides a boost pressure at the pump inlet. A relief valve is fitted to prevent excessive pressure build up due to heating or system malfunction. The body of the reservoir may contain horizontal baffles both to prevent fluid surging during aircraft manoeuvre and to promote de-aeration of returning fluid.

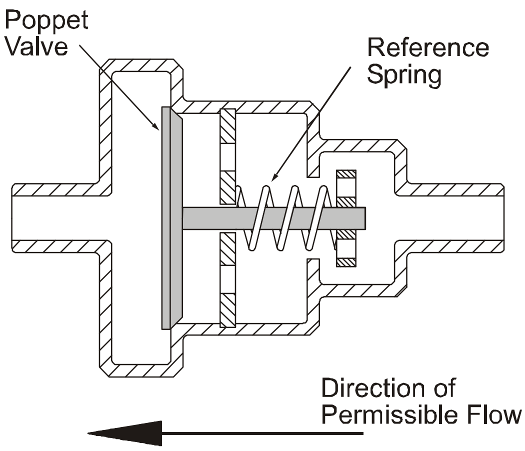
11. **Heat Exchangers and Temperature Warning Systems**. As described in para 22, hydraulic system performance is adversely affected by the presence of either air or vapour absorbed in or mixed with the fluid, and additional heat exchangers are usually included in high performance systems to keep the fluid well below its vaporization point. Such systems also include temperature sensors and warning systems to alert the crew if excessive temperature excursions do occur. For normal fluids, such warning systems are activated at temperatures of about 100 ºC.

12. **Filtration**. To prevent fluid leakage and loss of pressure, the clearances between the moving parts of a hydraulic component are minute, and the inclusion of even the smallest particles in the fluid would cause damage to its precise surfaces. High levels of filtration are therefore applied to the fluid. Several filters are included in most systems, so that each major component can be protected from debris generated upstream of it.

13. **Pressure and Thermal Relief Valves**. The use of a cut-out valve to regulate the output pressure of a constant displacement hydraulic pump was discussed in para 6. Because hydraulic fluid is incompressible and mechanical damage can be caused to components if over-pressurization occurs, further pressure relief valves are situated at critical points in the system. They are frequently termed 'fuses' because of this protective role, and they operate by balancing system pressure against an internal reference spring. If system pressure rises above spring pressure, the valve opens allowing fluid to escape into the system return pipes thus reducing pressure. The valve re-seals automatically once system pressure returns to below the reference level.

14. **Non-return Valves**. There are areas in most hydraulic systems in which it is necessary to allow fluid to flow to a component but to prevent that fluid returning along the same pipe. Non-return valves are used for this purpose, and several are included in the system in Fig 3. Such valves are similar in construction to relief valves, and the principle of operation is shown at Fig 9. The valve poppet is held closed by a weak internal reference spring. Pressure of fluid flowing in the desired direction can readily overcome spring force, and fluid can therefore flow through the valve almost without restriction. If fluid pressure upstream of the valve is reduced, the poppet snaps closed to prevent a fluid flow reversal.

4-1 Fig 9 Principle of a Non-return Valve



15. **Control Valves.** Both rotary and linear action control valves are used in hydraulic systems, and each type is shown diagrammatically in Fig 10. Valve movement may be achieved by mechanical, hydraulic or electrical means depending upon the application. The valves are invariably of an 'on/off' rather than a variable throttle type.

4-1 Fig 10 Control Valves

Fig 10a Linear Control Valve

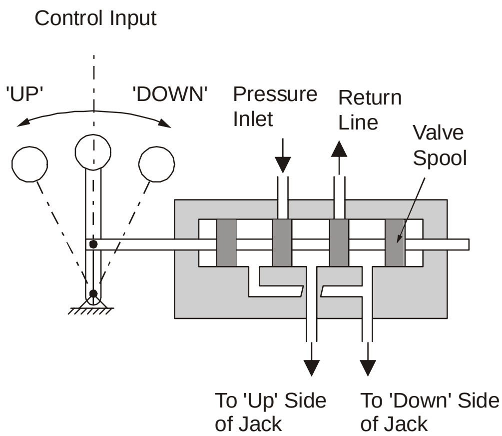
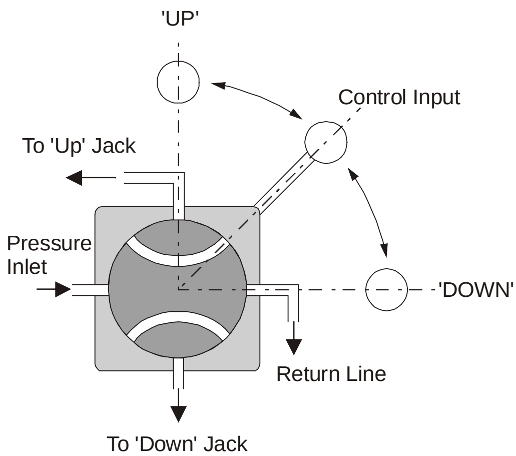


Fig 10b Rotary Control Valve



16. **Jacks and Motors**. Jacks translate hydraulic fluid pressure into linear mechanical movement, as in the example illustrated in Fig 2. Part rotary motion is often achieved by causing the jack to drive a

connected crank in an arc; however, full rotary motion is achieved by using a hydraulic motor. This operates on the reverse principle of the swashplate pump shown in Fig 4. Hydraulic pressure is fed sequentially to the pistons arranged around the motor body, and these react against the swash plate forcing it to rotate.

17. **Instrumentation and Control**. Compared to electrical systems, the instrumentation and control of hydraulic systems are very simple. Cockpit instrumentation monitors system pressure, and the aircraft central warning system usually provides warning of system pressure failure and system over-heating. The crew are able to manually select an alternative system if one fails, although this reversion can be automatic by operation of cross-system control valves sensitive to system pressure. Sight glasses and gauges are provided in most reservoirs and accumulators so that fluid levels and nitrogen pressures can be checked on the ground, whilst remote gauging systems are installed in cases where these components are not readily accessible.

System Safety Features

18. Hydraulic systems and their components reach very high statistical levels of reliability. Nevertheless, both military and civil aircraft design standards require that aircraft hydraulically powered primary flying control systems must have a back-up with the capacity to provide continued control for an indefinite period after failure of the primary system. They also require that secondary systems, such as undercarriages and brakes, have back-up with capacity to operate them for one landing. The provision of alternative power sources, system redundancy and emergency power is made to meet these requirements.

19. **System Redundancy**. Alternative sources may include provision for the powered flying control units of a control system to revert to manual control, or for other hydraulic sources to be connected to the failed power system. For this purpose, hydraulically powered primary control systems are powered by at least two hydraulic systems. The power systems are configured to be totally independent of each other so that the failure of one, for whatever reason, does not jeopardize operation of the other.

20. **Emergency Power**. Assurance that system operation can be continued for indefinite periods, after failure of one hydraulic pump, requires that two other pumps' sources are provided. One is usually a pump driven from the aircraft normal secondary power system. The other may be a pump powered by an emergency source such as an Emergency Power Unit or a Ram Air Turbine (see Volume 4, Chapter 11). For systems requiring only a limited duration of operation under emergency power, such as wheel brakes and undercarriages, the stored energy of accumulators or 'blow down' nitrogen cylinders (see Volume 4, Chapter 2) situated in the system is used.

Limiting Factors

21. Several factors influence the effectiveness of hydraulic systems, and some of these are expanded upon below. The adverse influence of such factors is minimized by careful design and maintenance of the

systems and selection of the most appropriate fluids. There is no ideal solution in these cases, and the chosen solution is invariably a compromise between performance and the other factors.

22. **Temperature and Aeration.** As hydraulic fluid nears its boiling point, fluid vapour and absorbed air are given off and carried in the fluid. The presence of gas from this or any other source introduces an unacceptable degree of compressibility into the columns of fluid in the system, causing operation to become sluggish and erratic. In high performance systems, preventive design features, such as reservoirs to prompt and contain the separation of gases from the fluid, and the provision of adequate cooling, are backed by careful system maintenance to minimize the likelihood of air entering the system.

23. **Contamination**. As discussed in para 12, contamination of fluid with even minute particles will damage and degrade systems performance. Careful systems replenishment avoids this problem, and adequate system filtration ensures that particles introduced into or generated by the system are removed before they can be carried through the system into components where they will cause mechanical damage. Many hydraulic fluids are also hygroscopic to a small degree. Again, careful system replenishment and routine monitoring of the fluid will minimize the possibility of water absorption.

24. **Flammability**. Certain hydraulic fluids are highly flammable, and leaks or spillage present a significant fire risk, although appropriate husbandry precautions can minimize this. Non-flammable fluids are used almost universally in the systems of passenger-carrying aircraft, despite them being highly corrosive.

25. **Hazardous Liquids**. All hydraulic fluids are active solvents and many are also corrosive. They are therefore hazardous to both aircraft surfaces and materials and to human beings. Non-flammable fluids are particularly hazardous. Careful handling during maintenance is necessary to avoid this problem.

System Health Monitoring and Maintenance

26. The maintenance activities carried out on hydraulic systems include first aid action to disclose, contain and rectify component failure, and fluid monitoring used to observe overall system health trends and to detect component degradation.

27. **Filter Checks**. As shown in Fig 4, filters are strategically placed throughout an aircraft hydraulic system. A component failure may not immediately manifest itself as a system malfunction, but routine inspection of the filter tell-tale devices will reveal that a failure has occurred. The filter will also prevent debris migrating around the system to cause secondary failures. Maintenance action can then be taken to restore and safeguard system integrity.

28. **Fluid Monitoring**. A systematic sampling programme of fluid contamination is carried out on the majority of aircraft. The periodic chemical and spectral analysis of fluids serves to indicate failure trends in particular components and the contamination and degradation of system fluid. Based on these trends, timely component replacement can be taken, thus preventing eventual failure occurring in the air, and reducing repair costs.

CHAPTER 2 - PNEUMATIC SYSTEMS

Introduction

1. The use of air as a medium to transmit energy and to do work offers many advantages to the aircraft designer. Although some early applications of pneumatics have been superseded by hydraulics or electrics, as technological advance has overcome the initial disadvantages of these alternative media, the inherent and unique advantages offered by the use of air and its main constituent gases ensure that pneumatics will remain one of these three fundamental power transmission media for aviation use into the foreseeable future. Unlike hydraulics and electrics, pneumatic power is generated and stored in a number of different ways each relevant to the specific end use, and it is therefore not appropriate to consider pneumatic power generation as a specific topic. Instead, the principle characteristics of the medium, and the techniques and equipment configurations used to exploit those characteristics for specific applications, are discussed in the following paragraphs.

Unique Characteristics

2. The ready availability of high temperature, high pressure air as a by product of the propulsion system, or even of aircraft forward motion, provides an extremely cost effective source of heat or pressure energy. Systems which utilize such energy sources include cabin and cockpit pressurization and heating, airframe and engine de-icing and the augmentation of flying controls. Similarly, air can be cycled through a system and exhausted overboard after use, without penalty. Such 'total loss' systems are extremely space and weight efficient, and this factor influences the choice of air above other energy transmission media which usually require to be contained in a closed circuit system for technical or environmental reasons. Such 'total loss' air systems include engine starting and cabin and equipment conditioning. Again, although air will support combustion, its properties are not affected by temperature extremes, and it can therefore be used in power transmission applications where high temperatures, fire risks or chemical reaction rule out the use of normal hydraulic fluids. Pneumatic systems are therefore often used in engine nozzle and thrust reverser operating systems.

3. The ready compressibility of air offers both advantages and disadvantages for its use. The advantages are that air can be compressed and used to store the resultant pressure energy either long term for subsequent use or short term to absorb shocks or sudden changes in pressure levels. However, because of this same compressibility, pneumatics are not suitable for use in control systems requiring precise, rapid response movements.

Typical Applications

4. The applications of pneumatics can be categorized under four main headings. These are:

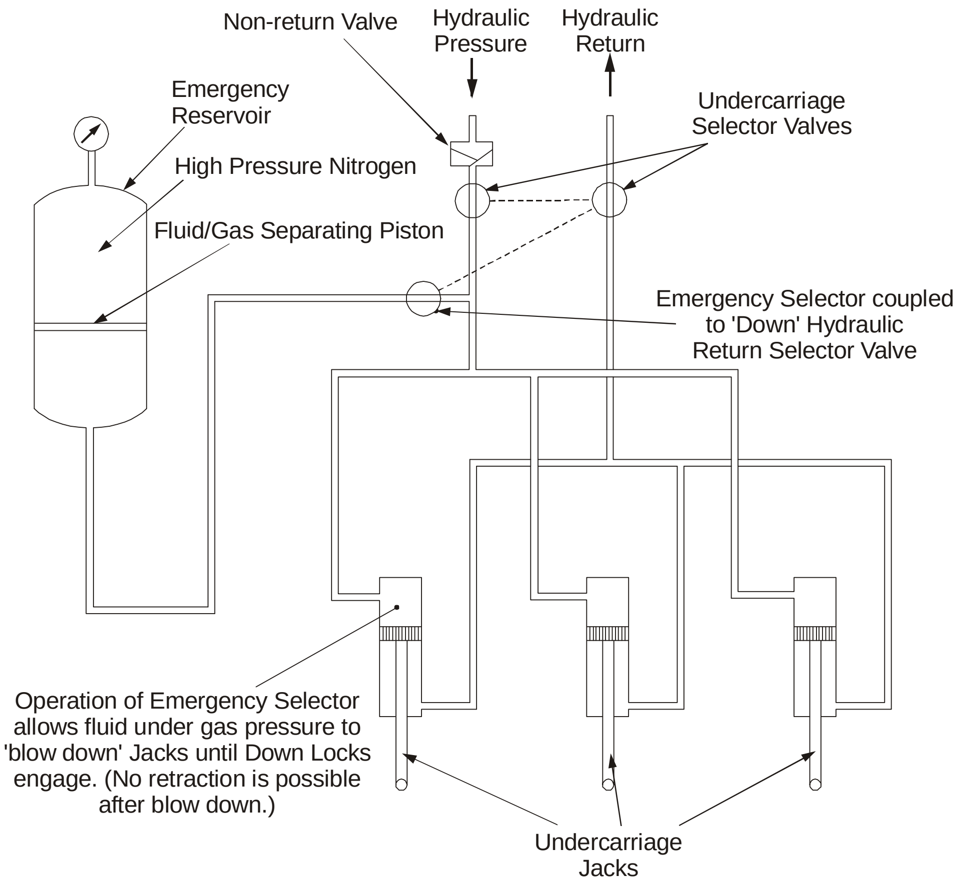
a. Pressure energy storage.
b. Compression.
c. Pressure energy transfer.
d. Heat energy transfer.

Details of several of the major systems which utilize pneumatics in these ways are discussed in the relevant chapters of this Volume, whilst other specific examples are given below.

Pressure Energy Storage

5. **Undercarriage Blow-down Systems**. Because hydraulic fluids cannot normally be compressed, energy cannot be stored within simple hydraulic systems. However, this disadvantage can be overcome by the integration of pneumatics into hydraulic systems. An application of such hydro-pneumatics is the undercarriage emergency blow-down system. A schematic diagram of such a system is at Fig 1. In this particular example, release of high pressure air (nitrogen is normally used to reduce the risk of a hydraulic oil fire) from the blow-down bottle enables the undercarriage lowering system to be pressurized sufficiently to lower the undercarriage in the event of hydraulic malfunction or failure. Other similar systems feed nitrogen directly into the hydraulic fluid. Although such an arrangement avoids the disadvantage of pressurizing only a set volume of fluid (that displaced within the accumulator by the gas), it does impose a considerable penalty during subsequent fault rectification, because the gas is absorbed into the fluid necessitating replacement of all of the fluid within the hydraulic system.

4-2 Fig 1 Simplified Undercarriage Blow-down System



6. **Fire Extinguishers and Liferafts**. There are many other applications in which compressed gases are stored for eventual use as an emergency or occasional energy source. Amongst the most relevant are the use of nitrogen to pressurize engine fire extinguisher bottles for eventual use in propelling

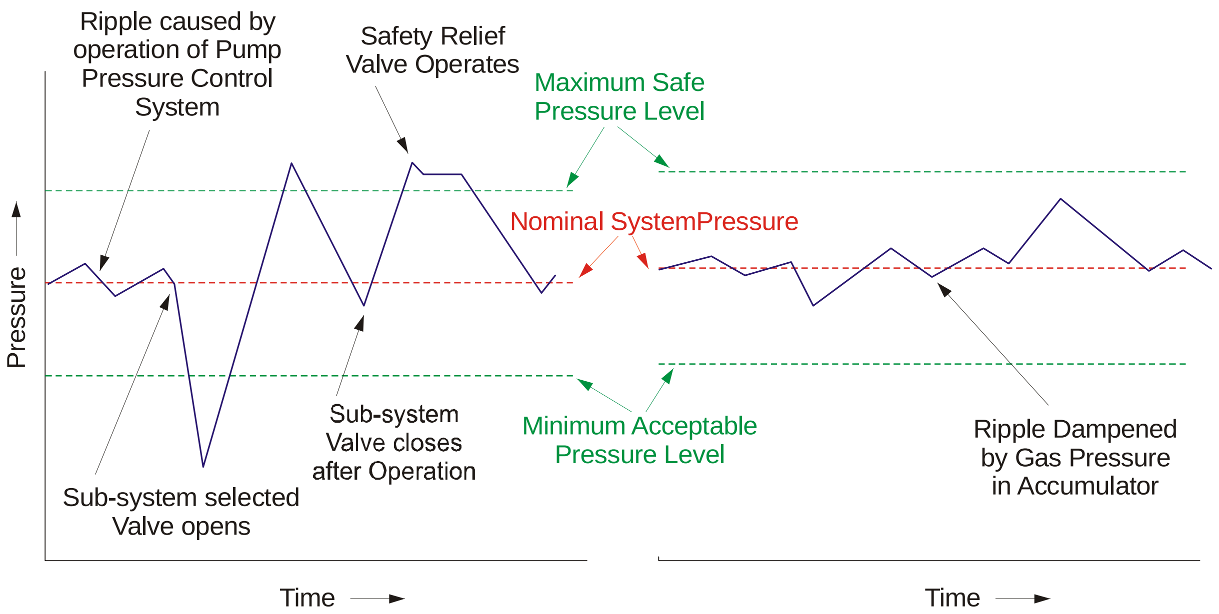
extinguishant on to a fire, and the use of carbon dioxide stored with liferafts and life jackets for subsequent release to inflate these items when they are required.

Compression

7. **Shock Absorbers**. Hydraulic systems are frequently configured to use the compressibility of air to absorb shocks and sudden changes in system pressure. The system shown in Fig 1 includes a nitrogen filled hydraulic accumulator. The functions of the accumulator are to smooth out any sudden changes in systems pressure caused by operation of components such as jacks and to protect the system from sudden peaks in pressure which occur when system valves close. The graph at Fig 2a shows the typical pressure variation in a system without an accumulator, whilst Fig 2b shows the comparable variation when an accumulator is used. Hydro-pneumatic shock absorbers, based on a similar principle, are widely used in many undercarriages, and these are discussed in detail in the relevant chapter of this Volume.

4-2 Fig 2 Hydraulic Accumulator Performance

a System Pressure Fluctuations without Accumulator | **b System Pressure Fluctuations with Accumulator**



8. **Seal Inflation**. The doors and canopies of pressurized aircraft require to be sealed effectively, to maintain pressurization within the fuselage and to prevent the escape of unacceptable volumes of conditioning air. The sealing of the irregular gaps between such doors and hatches and their frames imposes a significant problem, and seals inflated by compressed air are often used in such situations. The omni-directional force applied to such seals by low pressure air is ideal for such applications, and the air can readily be tapped from the aircraft pressurization system.

Pressure Energy Transfer

9. **Augmented (Blown) Lift Devices and Flying Controls**. Within the restrictions of current aerodynamic knowledge and technology, all VSTOL aircraft must be provided with devices which impart energy to the surrounding air stream to provide lift and control forces in the absence of adequate forward air speed. Purpose-designed VSTOL aircraft usually use vectored lift/thrust systems which also provide flight control forces. However, the use of conventional aerodynamic devices enhanced for STOL operation by ducting high energy air streams over them is an effective alternative solution. Fig 3 shows such a system in schematic form.

4-2 Fig 3 Augmented Lift Devices

Fig 3a Wing in Normal Flight

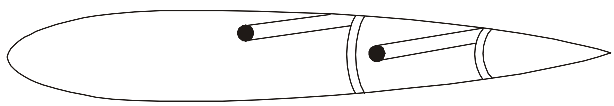
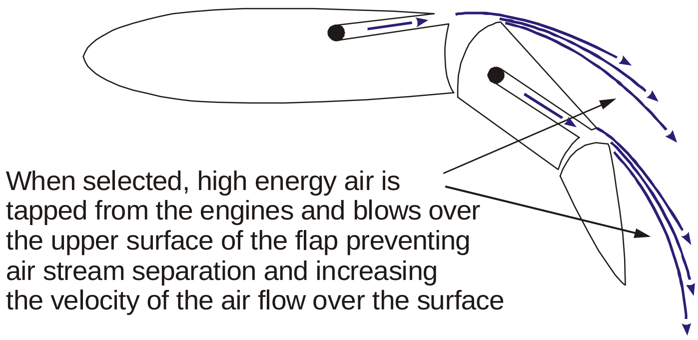


Fig 3b Wing with Blown Flap Extended



10. **Starters**. The abundant availability of high pressure air from gas turbine engines and APUs allows its use for engine starting. This is achieved either by impinging upon the turbine directly, and thus spinning up the engine, or more usually by driving a small turbine which is connected to the main engine through suitable gearing. Both of these applications are dealt with in the appropriate Chapter of this Volume (see Volume 4, Chapter 12).

Heat Energy Transfer

11. **Air Conditioning and Ice Protection**. The compressors of most high performance gas turbine engines are designed to produce volumes of air in excess of engine requirements. Such air at high pressure and at temperatures up to 300 ºC is available through engine compressor bleeds, and, as well as being used in cabin and cockpit pressurization systems, the air provides an effective source of heat for air conditioning and for the ice protection of aerofoils and engine intakes. Both applications are dealt with in Volume 4, Chapter 5.

CHAPTER 3 - ELECTRICAL SYSTEMS

Introduction

1. Early aircraft had no electrical equipment other than the engine ignition system. Power for this was provided by an engine driven magneto. The introduction of lighting and communications equipment necessitated this source to be augmented, first by a pre-charged battery, and subsequently by on-board generation systems using wind driven direct current (DC) generators fitted with crude regulators to maintain a constant 12 volts output irrespective of the flying speed. As soon as developments in engine power permitted, these systems were replaced by engine driven generators rotating at relatively constant speed and controlled by more effective regulators. Ever-increasing on-board electrical loads necessitated the use of bigger diameter, and therefore heavier, cables.

2. The power output of an electrical generation system is a product of voltage and current. However, electrical cable diameter is dictated by current and the resistance of the cable material, not by voltage. Therefore, within the practical limitations of cable insulation, the higher the system voltage the higher the power capacity for the same physical cable size. The need to control weight led to the use of higher DC voltage systems. Although DC high voltage systems have been tried, the problems of arcing at altitude, and the size of batteries required, rendered such developments impractical. All military aircraft systems now conform to the present 24-volt international standard for aircraft DC systems (note: DC generators and systems are normally rated at 28 V, to maintain a positive charge state to the 24 V batteries). However, alternating current (AC) generation systems are not constrained by the same disadvantages as DC systems; consequently, AC systems were introduced into aircraft to meet the higher on-board power requirements common in the 1960s. The evolution of these systems has led to the 200 V, three-phase, 400 Hz generator systems, now standard in both military and civil aircraft. Components using the AC output include three-phase devices (e.g. motors), single-phase 115 V devices (e.g. radio equipment) and secondary supplies such as transformers.

3. The introduction of solid-state technology to avionic equipment has significantly reduced the power requirements in those aircraft not equipped with high-powered radars, and this has reversed the earlier trend towards ever-larger electrical generation systems. Aircraft now tend to fall into two categories; those with low electrical demands, the electrical systems of which are primarily DC based, and those with high demands, which are primarily AC based.

Sources of Electrical Power

4. Fig 1 shows the primary sources of aircraft electrical power:

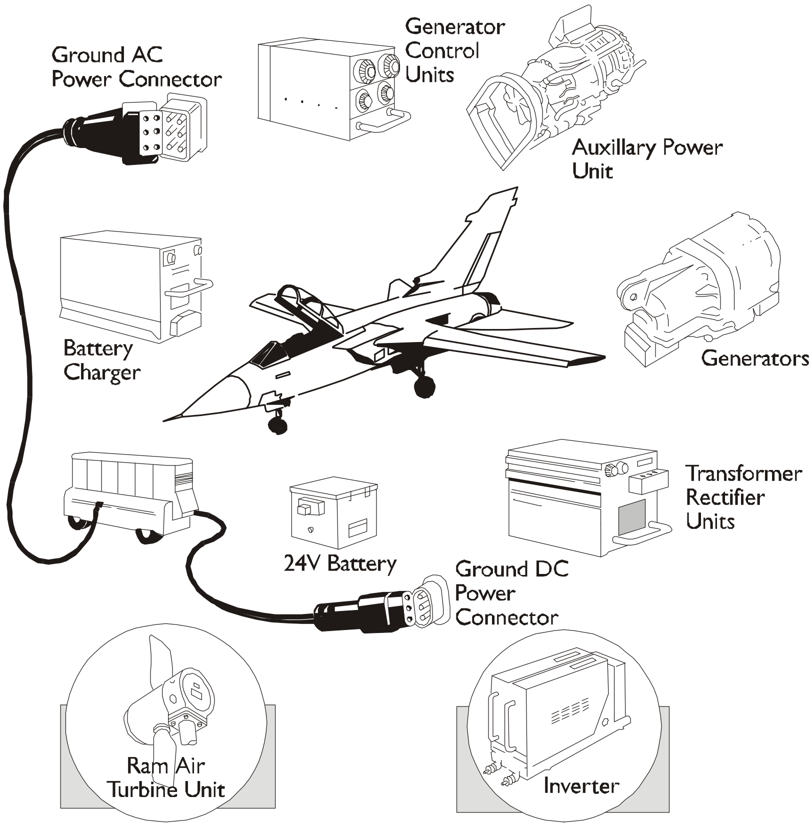
a. Primary electrical power is provided from a combination of batteries providing DC, and generators providing AC or DC.

b. Conversion of DC to AC or AC to DC, at similar or different voltages, is achieved by the use of inverters, converters, rectifiers and transformer/rectifier units. These equipments are described in para 7.

c. Auxiliary electrical power may be provided from either an Auxiliary Power Unit (APU) or a Ground Power Unit (GPU).

d. Emergency electrical power is provided by the use of batteries, a Ram Air Turbine (RAT) Unit or a rapid response Emergency Power Unit (EPU).

4-3 Fig 1 Aircraft Electrical Power Source

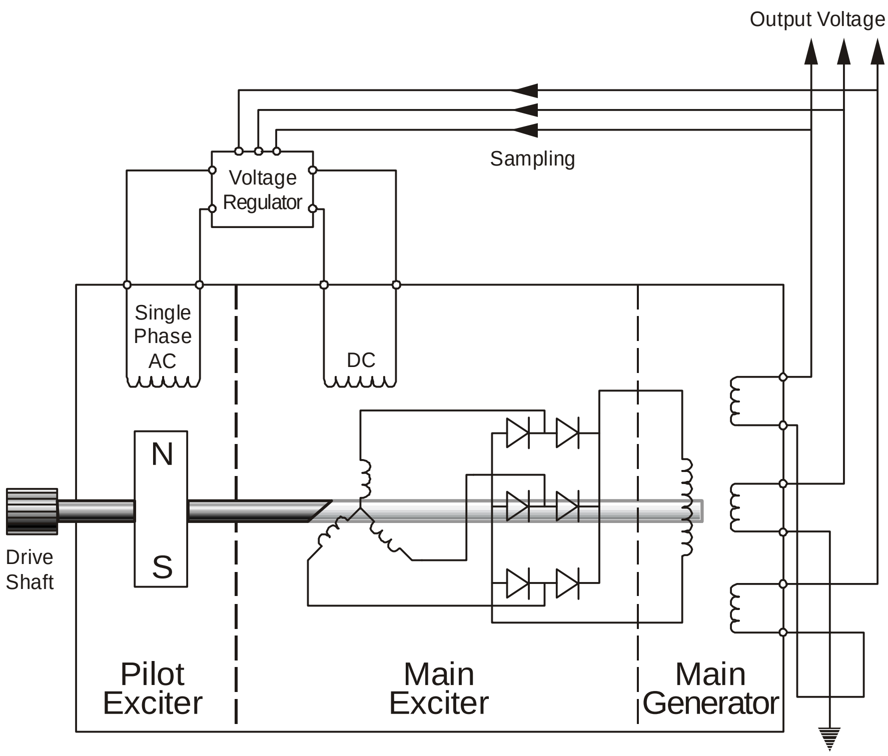


5. **Batteries**. Batteries produce DC electrical power by chemical reaction. Although certain types of battery generate electricity by an irreversible chemical action (these are termed primary cell batteries, typical of which is the dry cell battery), other types can be recharged (these are termed secondary cell batteries). The process of recharging and discharging on demand can be repeated for many hundreds of cycles. Both types of battery produce electricity at voltage values dependent upon their construction - most aircraft batteries are configured to produce 24 V. Such batteries have a fixed capacity but can release their charge over a wide range of current flows. They are thus able to provide short peaks of current in excess of 100 A, adequate for engine starting, whilst also being able to provide long-term, low current requirements of considerably less than 10 A. Batteries are an essential part of the aircraft electrical supply system, providing power before and during engine start, being recharged when the main generators come on line, and providing power again during emergencies or after the main engines have been closed down. Small individual rechargeable or single cycle batteries are used in instruments and avionics equipment to provide memory retention, and in lighting systems and safety equipment to provide emergency lighting and communications.

6. **Generators**. Generators convert mechanical energy derived from the aircraft engines into electrical energy by electro-magnetic induction. Fig 2 shows the principle of operation of a brushless AC generator.

Three windings are mounted on a common shaft which is driven through a suitable power take-off from a main engine. The windings rotate within three associated stator windings mounted in the frame of the machine. The permanent magnet induces a single-phase AC output in the pilot exciter coil, which is rectified and controlled before being fed back to the main exciter field coil. The induced output is rectified by the integral diodes and fed to the main field windings. The main generator produces the output which is fed into the aircraft power distribution circuit. The principle of brushless DC generators is similar, although the power is taken from the machine after the second phase of generation and rectification. Most generators are of the brushless variety which avoids the problems of wear in the brushes (DC) or slip rings (AC) and arcing inherent in these simple but less reliable machines. The power rating for typical DC generators is 6 kW to 9 kW (about 200 A to 300 A at 28 V), whilst AC generators produce output levels up to 60 kVA (200 V and 300 A at a 0.8 power factor = 48 kW) (for an explanation of the term, power factor.

4-3 Fig 2 Principles of a Brushless AC Generator



7. **Power Conversion Equipment**. Many aircraft electrical systems and components operate at voltages which are different to the primary generation source. For example, aircraft having a 200 V, three-phase AC generation system usually require a 115 V AC supply to power instruments and a 28 V DC supply to power the main DC components and to charge the aircraft batteries. Similarly, aircraft having 28 V DC generation systems require an AC supply to power certain avionic and instrument systems. The following types of power conversion equipment are available to achieve these tasks:

a. **Inverters**. Inverters change the primary DC supply to a secondary AC supply. Inverters may be either rotary or solid state. A rotary inverter consists of a DC motor driving an AC generator. A solid-state inverter (also known as a static inverter) uses a transistorized switching unit to do the same job but is often more efficient and more flexible in output.

b. **Converters**. Converters change the frequency of the primary AC supply to a different secondary frequency. They too may be solid state or rotary devices.

c. **Transformers**. In the main, transformers are used to change the voltage of a primary AC supply to a higher or lower secondary AC voltage.

d. **Transformer/Rectifier Units**. A Transformer Rectifier Unit (TRU) is a combination of static transformer and rectifier, for converting an AC input of one voltage into DC outputs of other voltages.

8. **Ground, Auxiliary and Emergency Power Units**.

a. **Ground Power Units**. Because of the finite capacity of aircraft batteries, and the varying requirements for electrical power on the ground between flights, most permanent operating bases are equipped with mobile or fixed AC and DC electrical supply units, called Ground Power Units (GPUs). These can be connected directly into the aircraft electrical system, to provide power for aircraft servicing and for aircraft systems and engine starting. During engine start procedures, as generators are brought on line, the GPU supply is normally isolated automatically.

b. **Auxiliary Power Units**. To avoid dependence upon availability of a GPU, many aircraft are fitted with an Auxiliary Power Unit (APU) capable of providing both electrical and hydraulic power for aircraft starting. Such units each consist of a generator powered by a self contained, small gas turbine 2, engine.

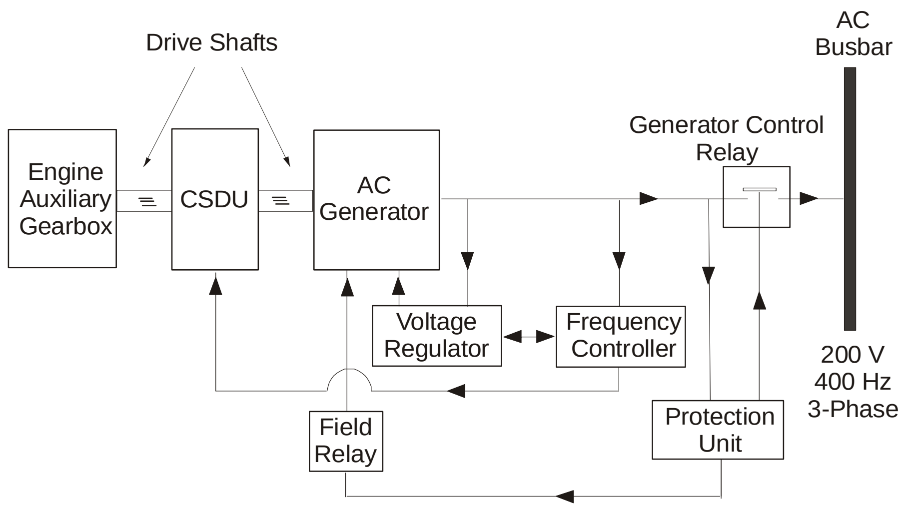
c. **Emergency Power Units**. APUs which can operate during flight are called Airborne Auxiliary Power Units (AAPUs). In addition to their auxiliary uses, they can provide power during emergencies to augment or replace the aircraft primary power generation system. Inherently unstable aircraft, the safety of which is dependent upon the continuous operation of automatic flight control systems, require emergency power supplies which can be brought into full operation within seconds of a primary system failure occurring. Ram Air Turbines (RAT) (propeller-driven generators, which automatically extend into the airstream in the event of a system failure) or turbine powered, rapid response Emergency Power Units (EPUs), are able to fulfil this requirement.

Voltage and Frequency Regulation, Power Output Balancing and Fault Protection

9. **Voltage Regulation**. Aircraft electrical equipment is designed to operate within closely defined voltage limits. To ensure satisfactory operation, the aircraft system voltage must be maintained within a set tolerance over a wide range of engine speeds and electrical loads. This requirement is achieved by the use of automatic voltage regulators, such as that included in Fig 2. These act to adjust the current fed into the generator's main exciter field coils in an inverse relationship to changes in system voltage. Thus, if system voltage drops because of an increase in the load, the generator exciter current is automatically increased, and, therefore, the generator output increases until the balance is restored. Adjustment of the current in the exciter coil is achieved in modern generators either by pulse or frequency modulation of the supply. However, older machines used a technique which controlled the current in the exciter coil by varying the resistance of a carbon pile placed in series with it.

10. **Frequency Regulation**. The output frequencies of AC generators are dependent upon their speed of rotation. For satisfactory equipment operation, it is imperative that the electrical system frequency is controlled precisely. As the initial drive will originate from an engine auxiliary gearbox, it must remain steady irrespective of variations in engine power settings. The drive shaft will, therefore, go through an intermediate device termed a Constant Speed Drive Unit (CSDU). This will maintain the drive to the generator at a constant rpm (a CSDU would not be required on aircraft fitted with constant speed engines). Fig 3 shows a schematic layout of a simple AC generator supply.

4-3 Fig 3 Schematic of a Simple AC Generator Supply



Where a CSDU and generator are designed and built as a single unit, it is termed an Integrated Drive Generator (IDG). CSDU and IDG systems utilize electro-mechanical or electro-hydraulic couplings, which work on the principle of sensing variation in system frequency and adjusting generator speed to maintain a constant output irrespective of the input drive speed (within system parameters). The CSDU and IDG systems are able to control frequency within 1%.

11. **Balancing of DC Generators**. In systems utilizing two or more generators, it is essential that each generator produces an equal output. This is achieved by interconnecting their respective voltage regulators so that the output of each generator is adjusted to balance with those of the others.

12. **Parallel Operation of AC Generators**. The balancing of AC generators requires not only that the load should be shared, but also that voltage, frequency and phase angles be synchronized. Load sharing is achieved automatically by comparing the level of current flowing from each generator and increasing the output of the more lightly loaded machine until a balance is achieved. Paralleling of generators is achieved automatically by control circuits which sense the frequency and phase angle of each. When the frequencies and phase angles of two generators are matched, bus-tie contactors between them close, thus inter-connecting their frequency control circuits. The control devices for each generator are usually located in dedicated Control and Protection Units (CPUs).

13. **Fault Protection**. The distribution circuits of both AC and DC generation systems require the addition of protection devices to prevent generator or consumer unit malfunctions from damaging equipment and

endangering the aircraft. Typical examples of the malfunctions for which protection is provided include over and under-voltage, over and under-frequency, short circuits (line to line and line to earth faults), phase sequence (three-phase AC only), and reverse current (DC only). The protection devices act to disconnect the relevant generator from the distribution busbar and also to de-excite the generator.

Power Distribution Systems

14. **General**. In order to enable generated power to be made available at the power-consuming equipment, an organized form of distribution throughout the aircraft is essential. The precise manner will vary dependent upon aircraft type, and the location of consumer components. Aircraft power distribution systems are configured to allow the maximum flexibility in their management if a component or systems failure occurs.

15. **Busbars**. In most types of aircraft, the output from the generating sources is coupled to one or more low impedance conductors, referred to as busbars. Busbars are usually located at central points in an aircraft, in junction boxes or distribution panels, and provide a convenient point from which supplies can be taken to the consumer unit. Busbars vary in form. In a simple system, a busbar may be a strip of interlinked terminals. In a more complex system, main busbars might be thick metal strips (usually copper) to which input, and output supply connections can be made; subsidiary busbars might be flexible copper wire. Busbars are insulated from the main structure and provided with protective covering.

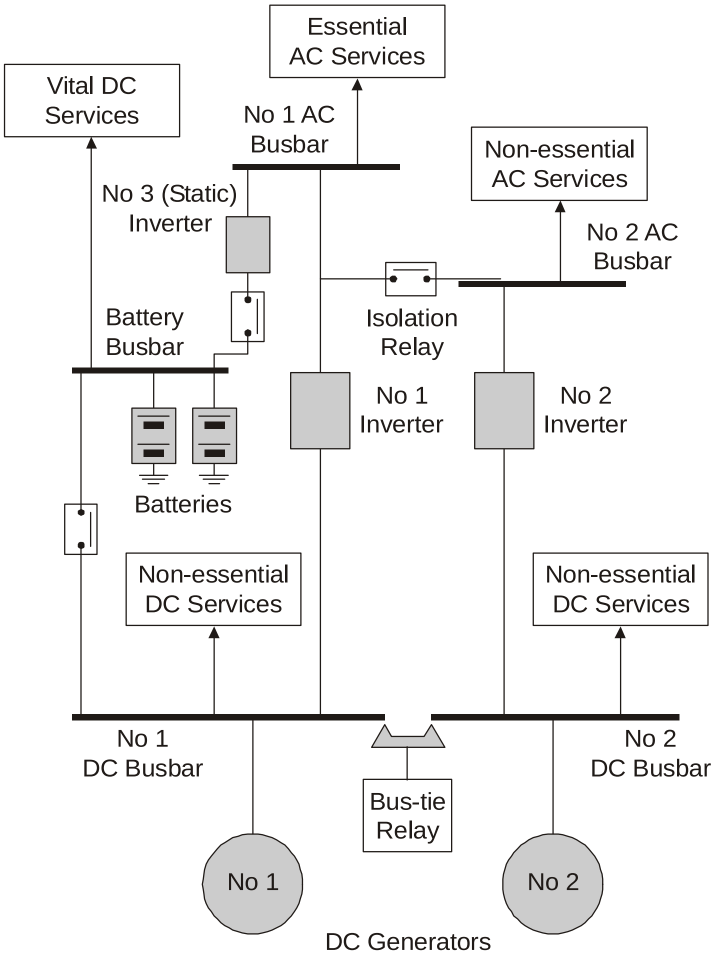
16. **Split-busbar Systems**. The function of a distribution system is primarily a simple one, but it must also work under abnormal conditions. Power to equipment should be maintained, if possible, during primary power source failures, and faults on the distribution system should have minimum effect on system functioning. These requirements are met in a combined manner by paralleling generators, where appropriate, by providing adequate circuit protection devices, and by arranging for faulty components to be isolated from the distribution system. In addition, it is usual to split busbars and distribution circuits into sections in order to power particular consumer components. The principle of the split-busbar system (see Fig 4) is that consumer services are divided into three categories of importance. If a generation system failure occurs, the distribution system can be progressively modified (manually or automatically) to maintain power supplies to essential consumer loads whilst shedding non-essential loads. The three categories of load are defined as follows:

a. **Vital Services**. Vital services are those services which are needed after an emergency landing or crash. These might include inertia switch operated fire extinguishers and emergency lighting. These services are fed directly from the main and emergency batteries.

b. **Essential Services**. Essential services are those services which are required to ensure safe flight during in-flight emergency situations, such as radio and instrument supplies. They are connected to busbars in such a way that they can always be supplied from a generator or from batteries.

c. **Non-essential Services**. Non-essential services are those which are not essential to flight and may be isolated during an in-flight emergency, either by manual or automatic action.

4-3 Fig 4 Split-busbar System (Primary DC Power Source)



System Control and Protection Devices

17. **Control Devices**. In aircraft electrical installations, the function of initiating and subsequently controlling the operating sequences of the circuitry is performed principally by switches and relays. A switch is a device designed to complete or interrupt an electrical circuit safely and efficiently as and whenever required. Switches exist to meet a wide range of applications. They may be operated manually or automatically by mechanical means or at predetermined values of pressure, temperature, time or force. Relays are remotely controlled electrical devices capable of switching one or more circuits. Used extensively in electrical and avionic systems, relays are available in a wide range of physical configurations to meet an equally wide range of performance criteria.

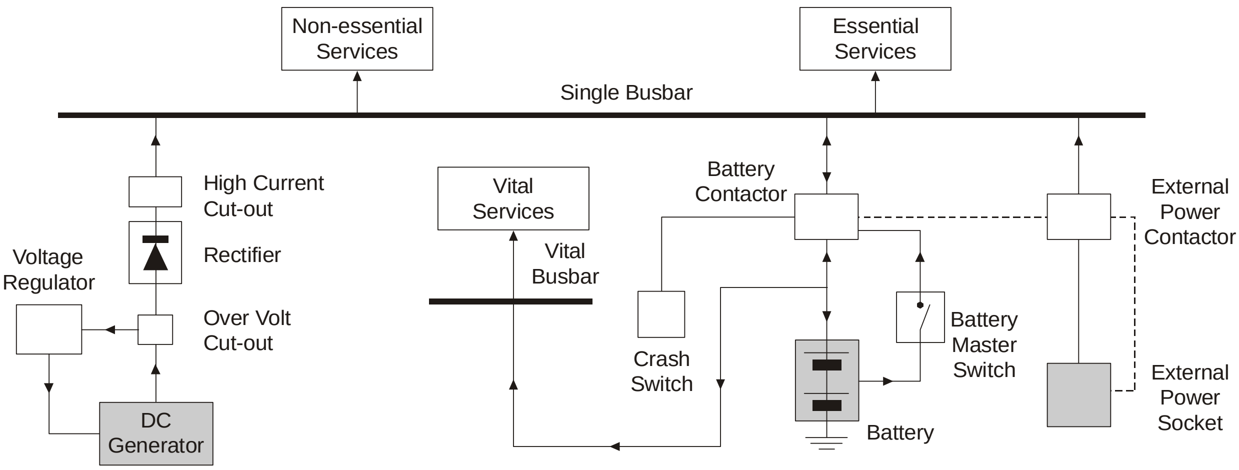
18. **Protection Devices**. An abnormal condition, or fault, may arise in an electrical circuit for a variety of reasons. If allowed to persist, the fault may cause damage to equipment, failure of essential power supplies, fire, or loss of life. It is, therefore, essential to include protection devices in electrical circuits to minimize damage, and safeguard essential supplies, under such fault conditions as over-voltage, over-current or reverse current. The protection devices used include fuses, circuit breakers and reverse current cut-outs (RCCOs). A fuse is a thermal device, designed to protect cables and components against short circuits and overload currents by providing a weak link in the circuit. Rupture of the fuse gives evidence of a system's malfunction, and, after correction of the fault, the fuse can be replaced.

Circuit breakers isolate faulty circuits by means of a mechanical trip, operated by thermal or electro-mechanical means. They can be readily reset in flight, if accessible, after clearance or isolation of the fault. An RCCO senses the difference in voltage between the generator and its busbar. Its contacts remain closed whilst the voltage of the generator is higher than that of the busbar, but open if this situation is reversed.

Typical Generating Systems

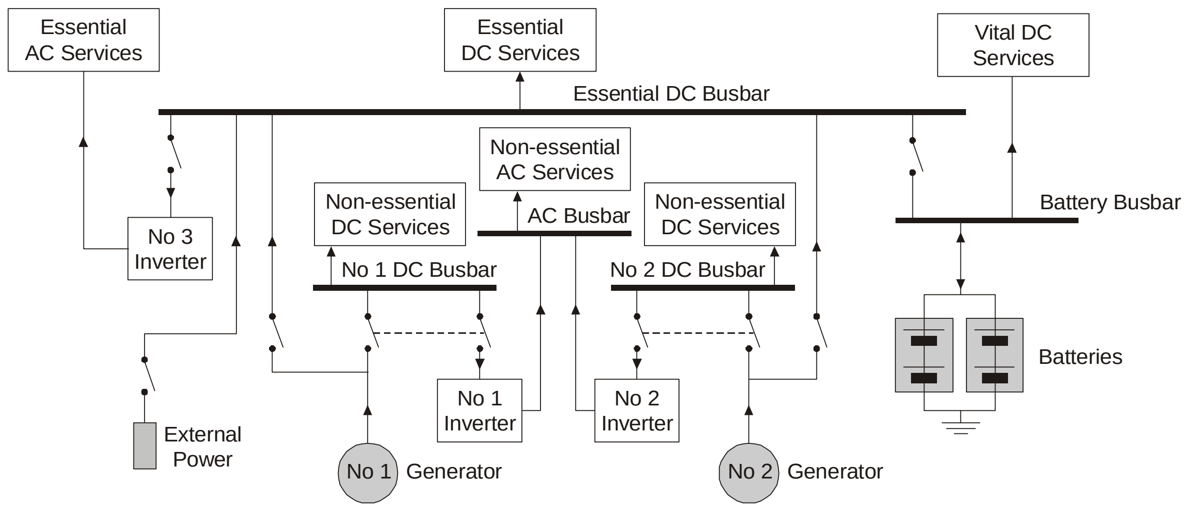
19. **Single Channel DC System**. The simplified schematic diagram at Fig 5 shows a single channel system typical of that used in a single-engine training aircraft. It is a simple system, with many automatic features designed to reduce the workload of an inexperienced pilot. The system comprises a brushed DC generator feeding the single busbar through a diode rectifier. The generator is controlled by a carbon pile voltage regulator and protected by high-current and over-volt relays. The aircraft battery is connected to the busbar through a contactor operated by the battery master switch. This is a two-way feed, allowing the battery to charge when the busbar is energized by the DC generator. The battery contactor is deactivated automatically when an external power supply is connected to the aircraft, or in the event of a crash.

4-3 Fig 5 Single Channel DC System



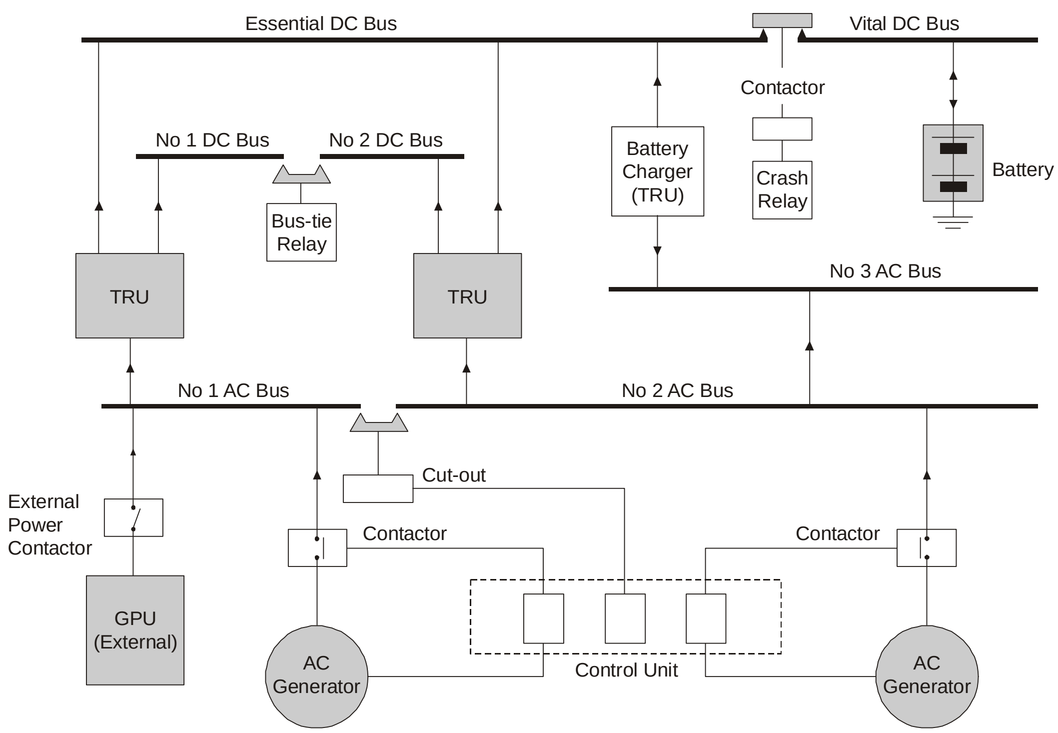
20. **Twin Channel, Split-busbar DC System**. The diagram at Fig 6 shows a twin channel, split-busbar system, typical of that used in a twin-engine aircraft with more than one crew member. The diagram has been simplified by excluding the control and protection devices which are present in such a system. The DC generators each feed a discrete busbar connecting non-essential consumer units. Both also feed the essential busbar, as do the batteries. Thus, in the event of a malfunction, power to essential DC consumers can be maintained, in preference to non-essential services. The generators also supply two inverters, which provide AC power to the non-essential AC services. Power from the essential busbar is converted to AC by a third inverter, to feed essential AC consumers. Vital services are powered direct from the battery busbar.

4-3 Fig 6 DC Twin Channel Split-busbar System



21. **AC Twin Channel, Split-busbar System**. The simplified diagram at Fig 7 shows a twin channel, split-busbar system, typical of that used in a twin-engine combat aircraft with more than one crew member. The two AC generators are regulated by separate, but cross-related, control units. Each AC generator supplies power to an AC busbar. The busbars can be interconnected (if the frequency and phase of each supply are synchronized). Each AC busbar supplies a TRU, which then feeds its respective DC busbars. These can be interconnected, if necessary, and they feed the non-essential services. The TRUs also feed the essential DC busbar which can be interconnected with the battery busbar if necessary. Thus, all similar busbars can be interconnected, but both AC and DC non-essential consumers can be disconnected if a system malfunction significantly reduces generator capacity. Note that the battery charger is supplied from an AC busbar and, in effect, acts as a TRU.

4-3 Fig 7 AC Twin Channel Split-busbar System



CHAPTER 4 - POWERED FLYING CONTROLS

Requirement

1. **Introduction.** The level of aerodynamic forces needed to control the attitude of an aircraft is proportional to the inherent stability of that aircraft and to the square of its speed. Thus, whilst the forces required to control a low-speed, well-balanced aircraft may well be within the physical capabilities of the pilot, those for a high-speed or high-stability aircraft will certainly not be. In such aircraft, a system of assisted flying controls is required. Where possible, the system merely augments the pilot's control inputs by the use of aerodynamic devices or power-assisted controls, but more highly loaded aircraft must utilize fully powered systems in which the pilot provides the basic command signal and the system implements that command. Fig 1 shows four levels of control, from a fully manual system, through aerodynamically assisted (servo-tab) and a power-assisted system, to a fully-powered control system.

Inputs are shown thus: Manual Input ←—, Power Input ⇦.

4-4 Fig 1 Stages of Control Power Augmentation

Fig 1a – Simple Manual Control

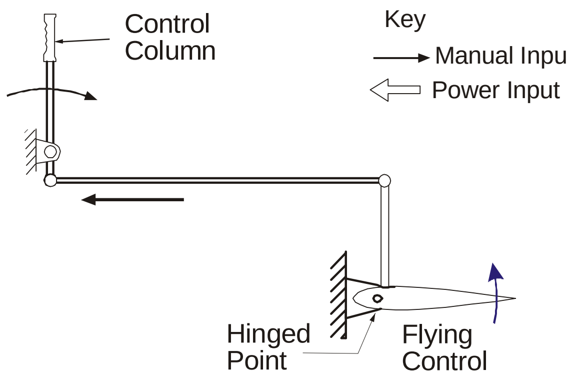


Fig 1b – Servo-tab Assisted Control

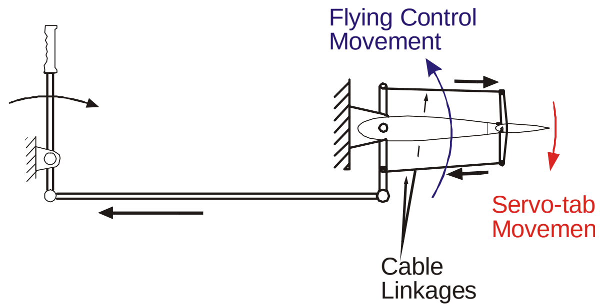


Fig 1c – Power-assisted Control

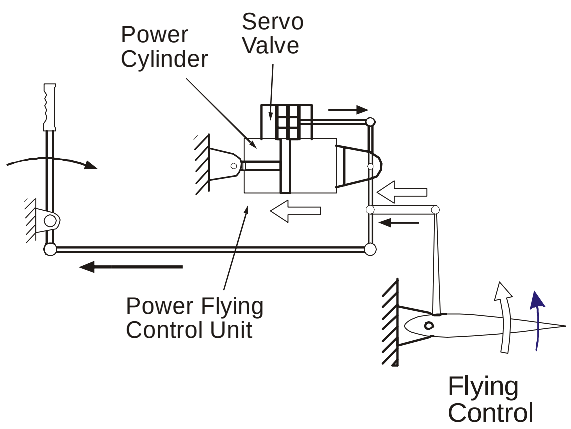
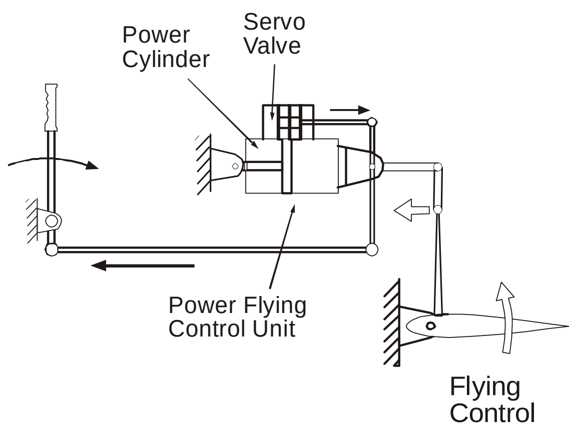


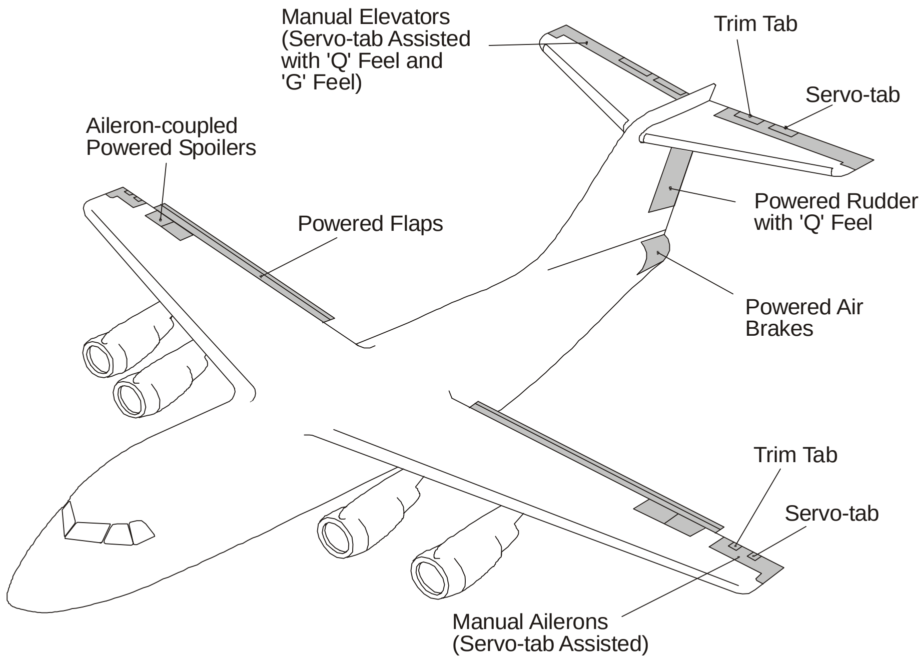
Fig 1d – Fully-powered Control



2. **Application**. The cost and complexity of powered flying controls serve to ensure the retention of manual or assisted control systems wherever possible. Thus, many current aircraft utilize manual or assisted systems for the more lightly-loaded controls such as ailerons and elevators, whilst assisted or fully-powered systems are used for those more heavily-loaded controls such as rudders and roll spoilers.

Fig 2 depicts a multi-engine transport aircraft, with high-speed cruise performance, but also with good low-speed handling, which has a mixture of assisted and fully-powered flying controls.

4-4 Fig 2 Typical Mixture of Flying Controls



3. **Additional Features**. The introduction of many advanced flight control concepts and systems has been made possible by the use of powered flying controls in aircraft. Whilst such features as stall warning (stick shakers) and stall prevention (stick pusher) devices can be integrated into manual systems, they are more effectively installed in aircraft which are fitted with powered systems. The use of such systems as auto-pilot, auto-land, fly-by-wire and fly-by-light, and application of active control technology to neutrally stable, fixed and rotary wing aircraft, is totally dependent upon the use of powered flying control systems.

Basic Requirements of Powered Flying Control Systems

4. ***Performance.*** A powered flying control system must perform to produce satisfactory handling characteristics throughout the aircraft flight envelope. The system must, therefore, have the appropriate power, and range of movement, needed to perform that task, whilst also being designed to achieve a good power-to-weight performance. Powered control systems occasionally have inadequate power for the task in hand, and this causes a condition known as 'jack stall', in which the servo jack cannot overcome the aerodynamic forces acting against it. The probability of such a condition occurring on particular aircraft types is well known, and identification of the areas of their flight envelopes where the phenomenon is likely to occur are clearly documented in the relevant Aircrew Manuals.

5. ***Feedback.*** An essential feature of all powered flying control systems is that of a feedback loop, capable of comparing the response of the system to that demanded by the pilot. In Fig 1, feedback to the pilot in the manual and servo-tab systems is accomplished automatically, because there is a direct,

fixed linkage between the pilot and the control surface. As the pilot moves his control, the corresponding control surface moves by a similar amount. The pilot can then use visual or instrument references to check that the aircraft has responded in the required manner to his control input. Thus, a complete feedback loop is established, and Fig 3a shows such a loop in diagrammatic form. Power-assisted and fully-powered systems require a similar feedback loop. This is usually achieved by a mechanical linkage which causes the powered flying control unit to drive until it reaches a position relative to the pilot's input signal.

4-4 Fig 3 Feedback

Fig 3a Positional and Rate Feedback Loop

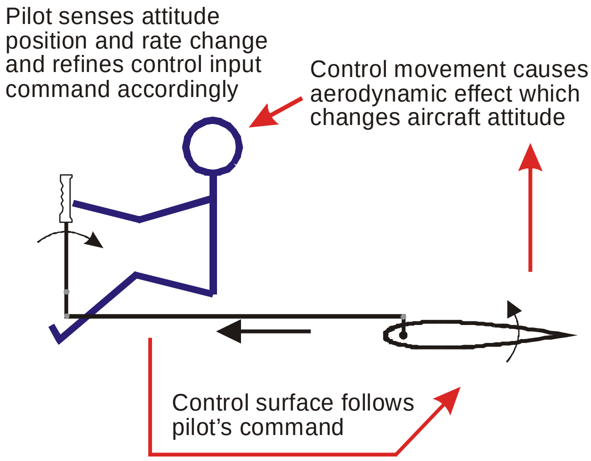
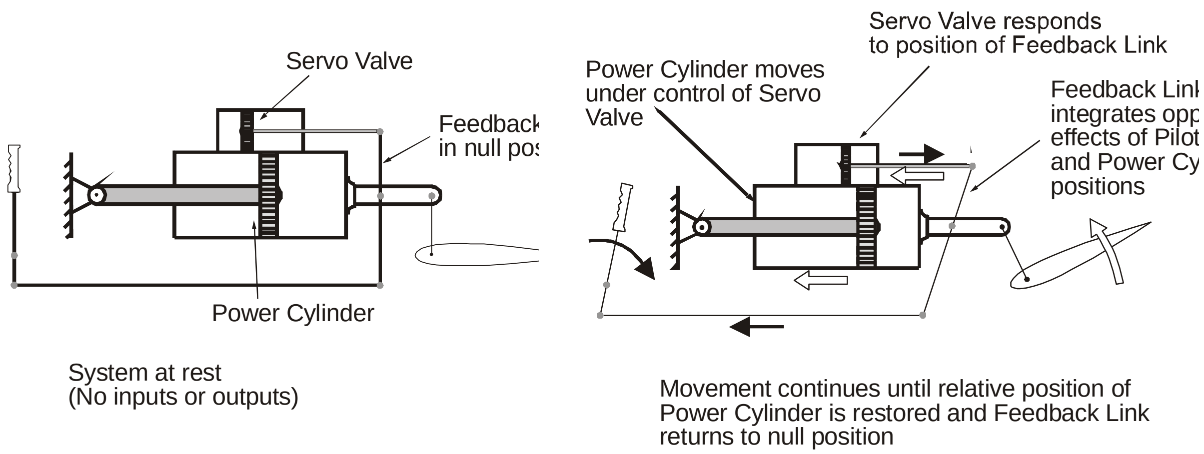


Fig 3b highlights the feedback linkage used in the simple powered unit shown in Fig 1. Feedback in automatic flight control systems is discussed in Volume 4, Chapter 7.

Fig 3b Mechanical Positional Feedback



In addition to positional feedback, a pilot also requires to receive a degree of feedback of flight forces. Such forces are essential to provide the pilot with tactile cues of the performance of the aircraft during flight. In a manually controlled aircraft, stick forces, increasing as the square of airspeed, give essential references to the pilot. Such references are not fed back to the pilot through a powered flying control, and methods of synthesizing feel are therefore incorporated.

6. **Accuracy**. The powered flying control system must respond accurately to both the amplitude and the rate of control input under all conditions of flight. Otherwise, the aircraft may be endangered, either by divergent oscillations being set up through the pilot over-compensating for system inaccuracies, or by overstressing caused through too rapid a response rate. The accuracy of response is partly inherent in the power source used in the system, partly in the effectiveness of feedback in the system, and partly by the precision with which components of the system are manufactured and installed.

7. **Stability.** Not only must the system respond accurately to the control input, but also it must hold the control position, and not deviate through spurious inputs caused by system errors. The stability of a system is largely ensured by initially designing sufficient tolerance into the components, although subsequent component maintenance of the highest order is required to ensure adequate margins of continued stability. Deterioration in the condition of both mechanical and electrical components, and the inclusion of air in hydraulic systems, are typical causes of degraded stability.

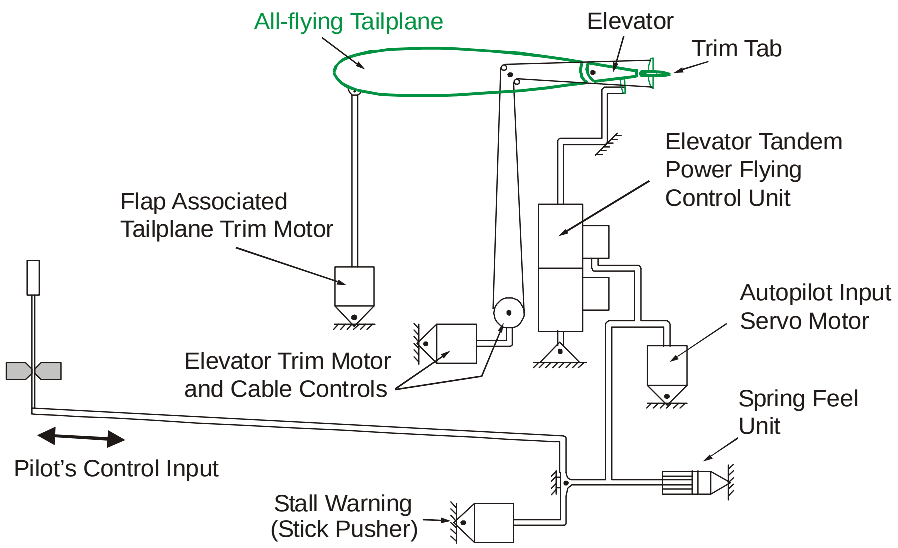
8. **Irreversibility**. The main objective of using powered flying controls is to off-load aerodynamic forces from the pilot's flying controls. Similarly, it is essential that the effects of buffeting, flutter, and turbulence are also off-loaded. The inherent irreversibility of hydraulic and electrical powered flying control units automatically ensures that this is accomplished. The likelihood of control surface fluctuation must, of course, still be minimized by good design plus aerodynamic and dynamic control balancing.

9. **Safety and Reliability**. Obviously, the reliability of its powered flying controls is paramount to the safety of an aircraft. To provide the necessary real and statistical degree of reliability of such controls, they are normally duplicated. In aircraft where flight loads would be within the physical capability of the pilot, reversion to manual control in the event of system failure may be permissible.

Typical Installation

10. Fig 4 shows the essential features of a typical powered flying control installation, in schematic form. Descriptions of its main components are included in the following paragraph. The system is based upon that used for the longitudinal control of a medium-size aircraft. It features an autopilot pitch channel servo and an all-flying tailplane (used for trimming out the pitching moment caused by use of the large-span flaps, often fitted to such aircraft). Elevator trim is provided conventionally by an aerodynamic trim tab, although both this component and alternative integrated trim devices are discussed in the following paragraph. Lastly, the system utilizes conventional cables and push rods to transmit commands between the pilot's controls and the servo units. In aircraft equipped for fly-by-wire or fly-by-light control systems, these items would be replaced by electrical or fibre optic cables.

4-4 Fig 4 Essential Features of a Powered Flying Control System

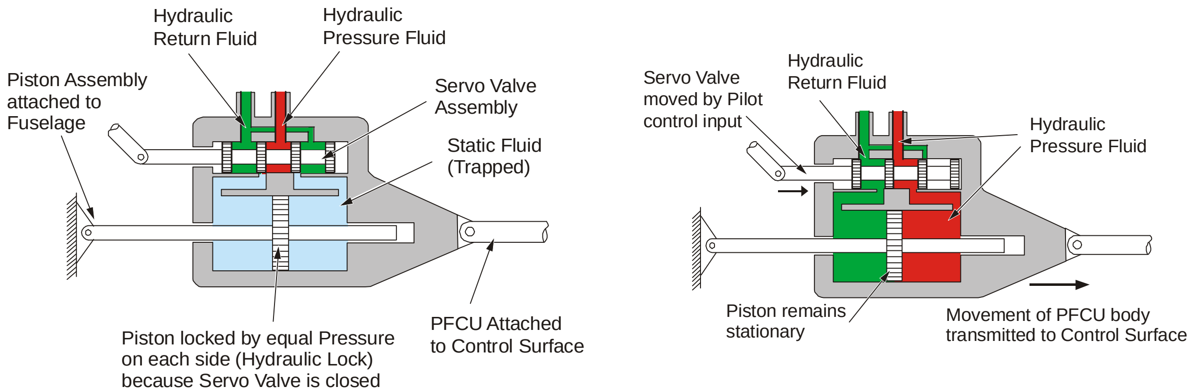


System Components

11. **Powered Flying Control Unit**. The basic features and operation of a hydraulic powered flying control unit are shown at Fig 5. The unit is shown both at rest and in mid-travel. Movement of the servo valve away from its mid-position occurs when the pilot moves the controls. The servo valve allows high-pressure fluid to enter and act upon the appropriate chamber of the unit. The main piston remains stationary, and the whole body of the unit moves under the fluid pressure, and its movement is transferred to the control surface. As the surface reaches its desired position, the movement of the body in relationship to the stationary servo valve restores the valve to its central position. The flow of hydraulic oil then ceases, and the unit is locked in its new position by incompressible fluid trapped on both sides of the piston. This situation remains until a further control signal, from either the pilot or the autopilot, causes the cycle to be repeated. By fixing the piston to the aircraft structure, and the unit body to the control surface, an automatic positional feedback is achieved. If the roles of the two components were reversed, an additional linkage would be needed to act as a feedback. Otherwise, the piston would travel to its extreme position whenever the servo valve was moved.

4-4 Fig 5 Hydraulic Powered Flying Control

Fig 5a – Unit at Rest **Fig 5b – Unit Moving**



12. **Artificial Feel Devices**. Typical feel devices are described in subsequent paragraphs. The essential features of an artificial feel system are as follows:

a. Forces should increase as stick displacement is increased.

b. The forces should be proportional to airspeed but, ideally, should reduce at high subsonic speeds, where the effect of turbulence is to reduce control effectiveness.

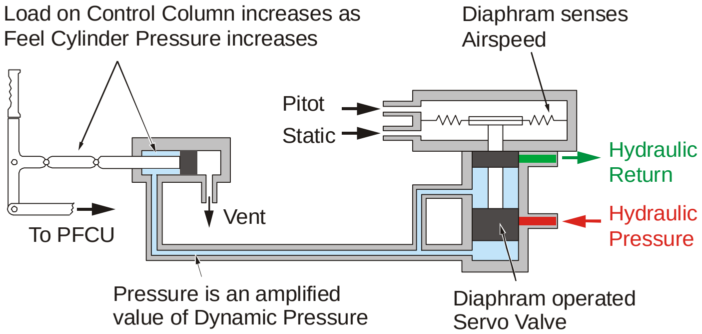
c. To prevent overstressing in the longitudinal plane, feel forces proportional to 'g' forces should be applied to the longitudinal controls.

13. **Spring Feel.** The most common form of feel device is a spring imposed in the pilot's controls. The system in Fig 4 includes a spring feel device in the elevator control run.

14. **'Q' Feel**. A major disadvantage of the simple spring feel is its inability to simulate the increase in force caused by an increase in speed. The 'q' feel system, which incorporates a pitot-static, speed-sensing device, varies its synthesized feel load as the square of the speed. A simple hydraulic 'q' feel unit is shown in Fig 6a.

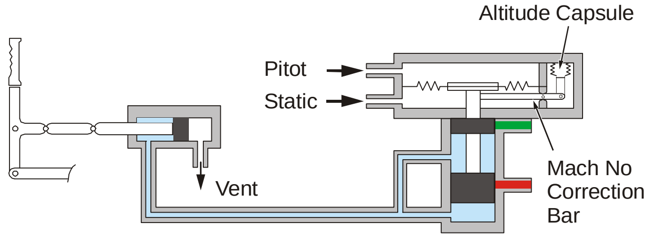
4-4 Fig 6 Speed Sensitive Feel Devices

Fig 6a Simple 'Q' Feel



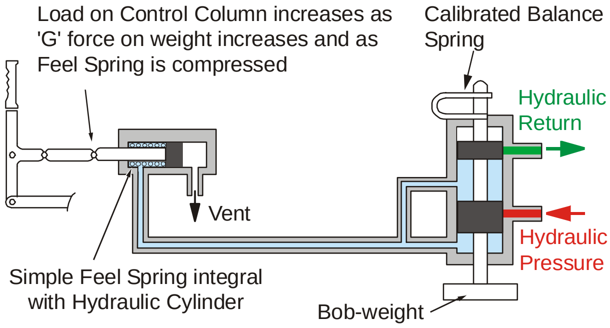
At high subsonic speeds, control surfaces tend to lose power because of compressibility effects. It is, therefore, important to limit or reduce feel forces at these speeds. This is achieved by using a refinement of the 'q' feel system, referred to as the 'Mach Number Corrected 'q' Feel Unit'. This unit has the addition of an altitude capsule in its sensing module. The feel forces generated by the unit are, therefore, proportional to Mach number rather than simply to speed. This arrangement is shown at Fig 6b.

Fig 6b Mach Number Corrected



15. **'G' Feel.** Aircraft stress limitations often necessitate limitation of longitudinal control forces when the aircraft is flying at high 'g' loadings. This is accomplished by fitment of a 'g' sensitive device which increases feel forces in proportion to the longitudinal 'g' forces present. The device is usually in the form of a bob-weight. It is often combined with a normal spring feel unit, which tends to reduce the undesirable effects of turbulence and inertia acting on the 'g' feel unit. Fig 7 shows such an arrangement.

4-4 Fig 7 'G' Feel Unit



16. **Additional Control Inputs.** The use of powered flying control systems allows the integration of additional control inputs such as trim adjustment, stall warning and automatic flight control. Automatic flight control systems are discussed in Volume 4, Chapter 7.

17. **Trim Adjustment.** Manual control systems utilize fine adjustment of the main or ancillary control surfaces to trim the aircraft flight attitude. This compensates for centre of gravity displacement, or attitude variation, at particular speeds or flap settings. The trim systems allow the main cockpit control forces to be minimized and the control positions to be centralized. Although some applications of powered flying control systems retain the use of ancillary control surfaces for trimming, many systems trim the aircraft by small adjustments of the main control system. However, adjustments to reduce feel forces on the cockpit controls, and to centralize them, remain necessary if the pilot is to retain references and cues during flight. A typical feel trim system is shown at Fig 8a. Its principle of operation is to zero the synthesized feel forces when the aircraft is correctly trimmed. A typical position (datum) trim system is at Fig 8b. Its principle is to apply trim adjustments to the aircraft controls without the pilot's controls being moved away from their neutral position.

4-4 Fig 8 Trim Systems

Fig 8a - Feel Trim

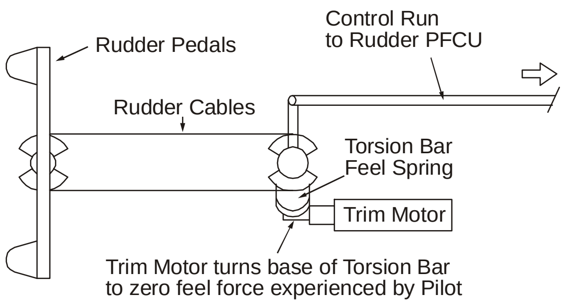
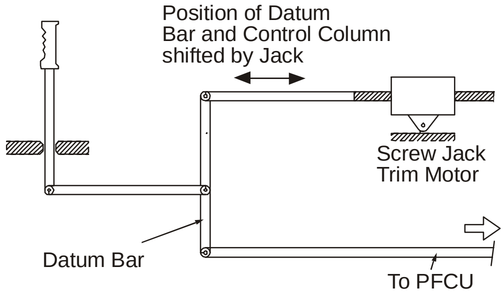


Fig 8b – Datum Shift Trim



18. **Non-linear Response**. Most manual control systems contain a degree of non-linearity in their operation. Small movements of the cockpit controls, when near to the extremes of their travel, produce larger control surface movements than will occur at the centre of their travel. This is a desirable situation, allowing small precise control movements to be made in the critical centre of the control span, and large coarser movements to be made at the extremes. The use of powered flying controls offers the opportunity to increase this non-linearity and thus increase the effectiveness of the control system. The use of non-linear levers and cams, not acceptable in a manual system because of the variation in control loads which they would impose, are permissible in powered systems to which variation in control forces are less significant. A typical non-linear control system is shown at Fig 9.

-4 Fig 9 Typical Non-linear Flying Control Mechanism

Fig 9a – Differential Aileron Control

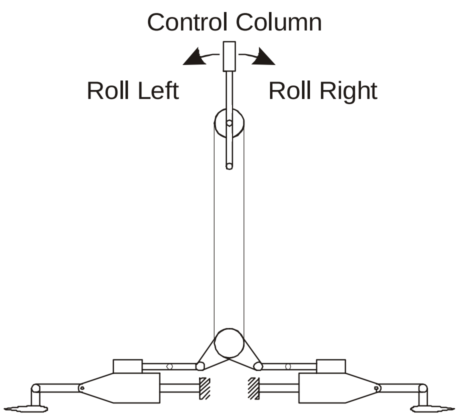
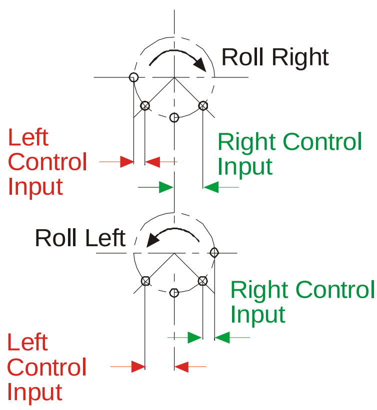


Fig 9b – Differential Effect

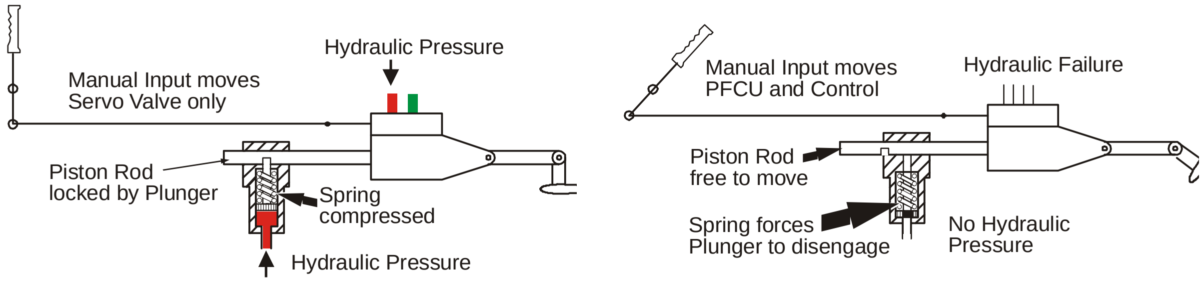


19. **Stall Warning and Prevention**. The dangers inherent in stalling a high-performance aircraft have led to stall warning systems being fitted to most relevant aircraft. In their simplest form, they consist of an electrical device which shakes the control column so that the pilot experiences cues similar to a stall buffet. The pilot is thus alerted to take the necessary corrective action. However, this is not adequate for many transport aircraft, particularly those with a high tailplane configuration, and stall prevention systems are often fitted to these aircraft. Typical systems, such as that in Fig 4, include a pneumatic jack which gradually imparts a nose-down control input to the aircraft. The pilot can then either accept and supplement the input, or consciously over-ride it, and take alternative measures to avoid the stall.

20. **Manual Reversion**. As mentioned above, reversion to manual control in the event of failure of a powered flying control system, is an acceptable option for lightly-loaded aircraft. The system of reversion is usually a very simple one and is often engineered to occur automatically in the event of a hydraulic failure. Fig 10 shows a typical system.

4-4 Fig 10 Hydraulic System with Manual Reversion

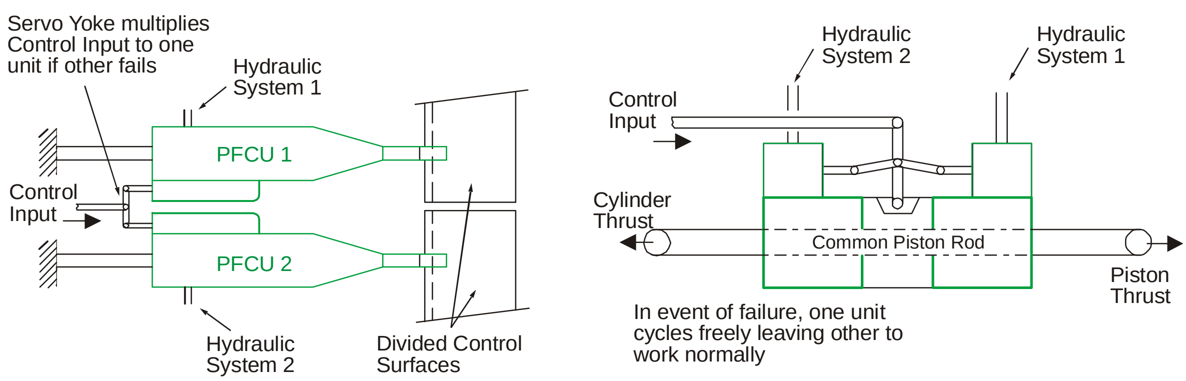
Fig 10a – Normal Operation **Fig 10b – Reversionary Mode**



21. **Multiplication**. A more normal arrangement for retaining adequate control, in the event of failure of a powered flying control system, is achieved by multiplication of critical components of that system. The most usual methods of achieving this redundancy of critical components are shown at Fig 11. These include split control surfaces (typical in an aircraft of the type depicted at Fig 2), each section having an independent powered control unit (Fig 11a), and use of tandem jacks (Fig 11b). All of these systems require an interlock within their respective hydraulic jack servo valves to prevent a hydraulic lock occurring in the redundant jack, and, thus, locking the relevant control surface. The use of parallel jacks is widespread in aircraft in which there is room to site such units adjacent to the control surfaces. However, for aircraft in which this is not possible, particularly combat aircraft and helicopters, the tandem jack arrangement is normally fitted.

4-4 Fig 11 Redundant Flying Control Components

Fig 11a – Parallel Units **Fig 11b – Tandem Units**



CHAPTER 5- CABIN PRESSURIZATION AND AIR CONDITIONING SYSTEMS

Introduction

1. The adverse physiological effects of altitude, and the associated low temperatures, are discussed in Volume 6, Chapter 4. The crew and passengers of aircraft operating at moderate and high altitudes are normally protected against these effects by pressurization of the cabin compartment. Air is fed into the cabin and allowed to build up to the required pressure. An escape of air, through discharge valves, is controlled such that the desired pressure difference is created between the interior of the cabin and the external environment of the aircraft. An air conditioning system usually forms an integral part of the pressurization system, to control the cabin atmosphere in respect of temperature and humidity. Although these two systems are closely interlinked, it is convenient to examine them separately.

Pressurization Systems

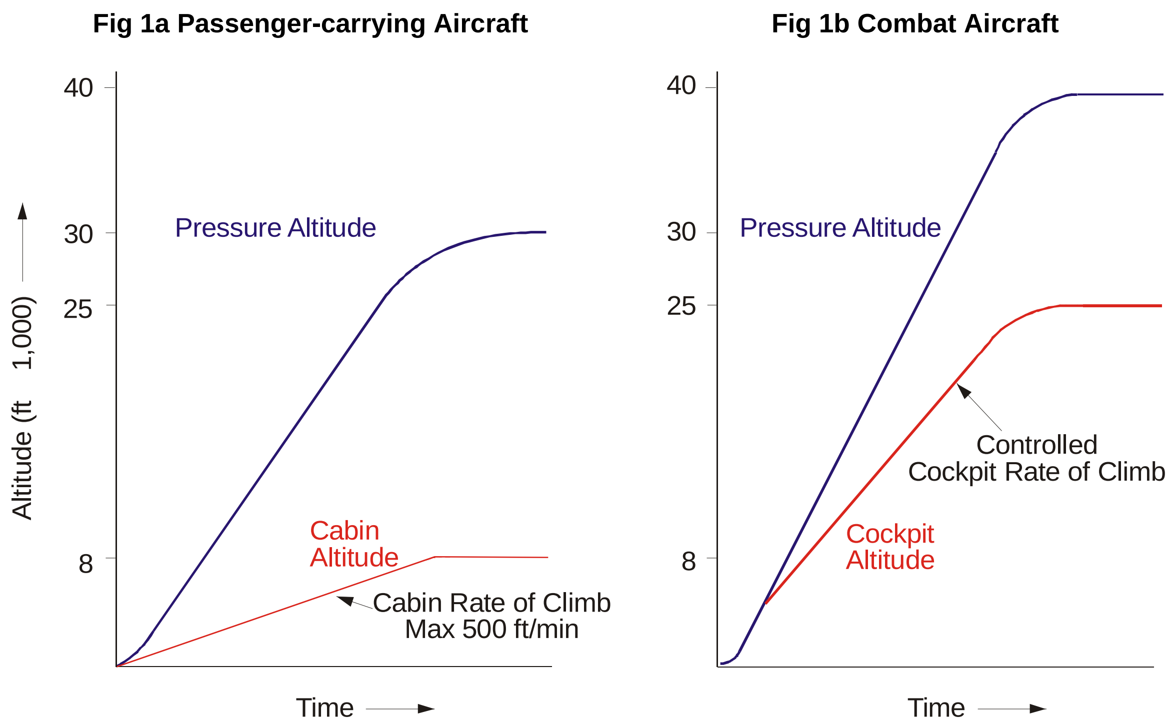
2. The aim of a cabin pressurization system is to maintain the cabin altitude at an optimum pressure, irrespective of the height at which the aircraft is flying. The two major considerations when utilizing a cabin pressurization system are the selection of a suitable cabin altitude, and the rate of change of cabin altitude.

3. **Cabin Altitude.** The required cabin altitude depends upon the role of the aircraft:

   a. **Passenger-carrying Aircraft.** For aircraft with passenger cabins, the maximum cabin altitude of an aircraft must be limited to between 6,000 and 8,000 ft (see Fig 1a). At this altitude, the air will provide sufficient oxygen for normal use.

   b. **Combat Aircraft.** Combat aircraft, with an oxygen supply for the crew, may use a cabin altitude up to a maximum of 25,000 ft, without the need for pressure suits. However, they also frequently use 8,000 ft as a cabin altitude (see Fig 1b).

4-5 Fig 1 Pressurization Profile

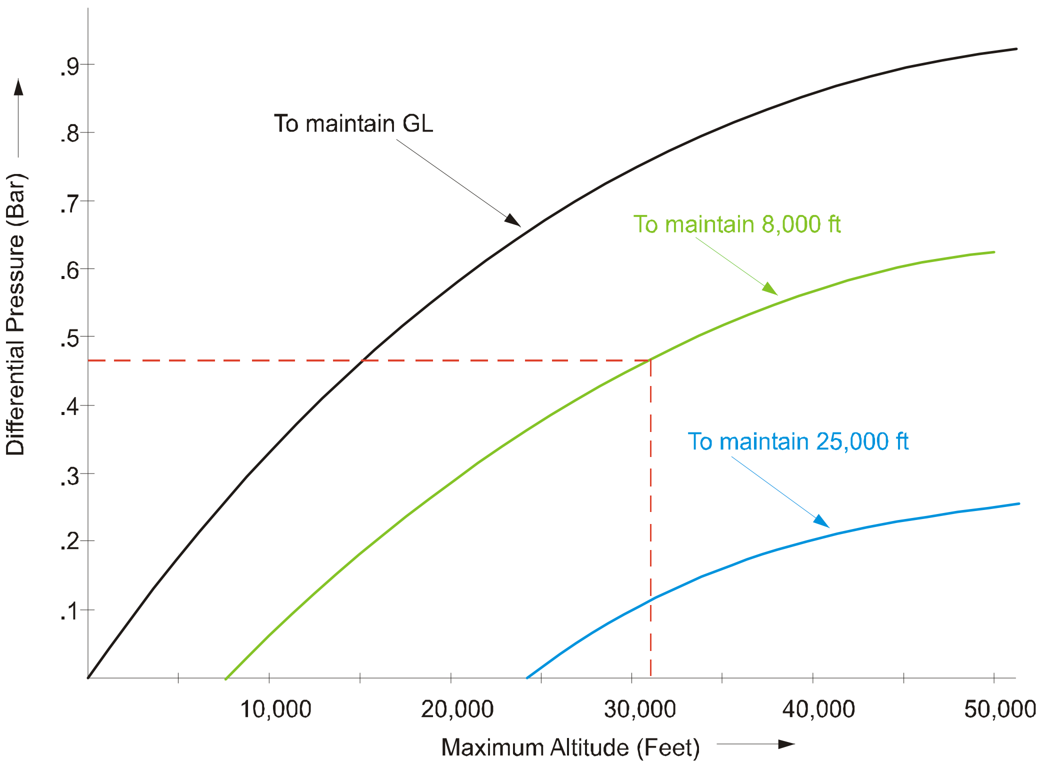


4. **Rate of Change of Cabin Altitude.** The permissible rate of change of cabin altitude is dependent upon the general fitness and health of the passengers/crew. The normal limit for passenger-carrying aircraft is a maximum climb rate of 500 ft/min and a maximum descent rate of 300 ft/min. These rates of change are higher for combat aircraft (see Fig 1).

5. **Cabin Differential Pressure**. Flying an aircraft at high altitude, with the cabin pressure set to a lower altitude, results in a pressure difference acting on the structure of the aircraft fuselage. This pressure difference is known as the 'cabin differential pressure'. The maximum permissible cabin differential pressure is based on the fuselage design strength, and will limit the aircraft's operational ceiling and its associated maximum cabin altitude. The differential pressures relating to aircraft and cabin altitudes are depicted graphically at Fig 2. The example annotated shows that, at a pressure altitude of 31,000 ft and a cabin altitude of 8,000 ft, a differential pressure of 0.46 bar is imposed upon the cabin structure. As mentioned earlier, most passenger-carrying aircraft, and many combat aircraft, use a cabin altitude of about 8,000 ft. However, combat aircraft may be required to operate with a cockpit altitude as high as 25,000 ft when the possibility of battle damage could cause rapid depressurization.

6. **Air Supply**. The supply of air for use in pressurizing and conditioning the cabin or cockpit is normally obtained from a late compressor stage of a gas turbine engine. Older aircraft types may use separate, engine-driven compressors. The high-pressure, high-temperature air supply is regulated and conditioned before being fed into the cabin. The air supply must be sufficient to maintain required cabin pressures, notwithstanding the normal small leakage of air from the cabin and the deliberate dumping of air as part of the air conditioning cycle. Aircraft equipped with an Auxiliary Power Unit (APU) are usually configured so that the air conditioning system can also operate using air supplied by the APU during periods when the aircraft is on the ground.

4-5 Fig 2 Cabin Differential Pressures at Altitude



7. **Control.** Pressurization is achieved by making the cabin a sealed container, into which a pressurized supply of air is fed. The air supply can be selected on/off, by means of a controllable bleed valve (it should be noted that there is a slight loss of engine thrust when the pressurization bleed valve is open). The cabin pressure and its rate of change are controlled by the regulated release of air to atmosphere through a discharge valve in the aircraft skin (Fig 3). In some aircraft, the crew can select the required cabin altitude and the rate of pressure change; in others these parameters are preset. In flight, the cabin altitude is automatically monitored by a control module, which sends a control signal to the discharge valve unit. The discharge valve compares the actual cabin pressure with the control signal, and opens or closes the outlet valve accordingly.

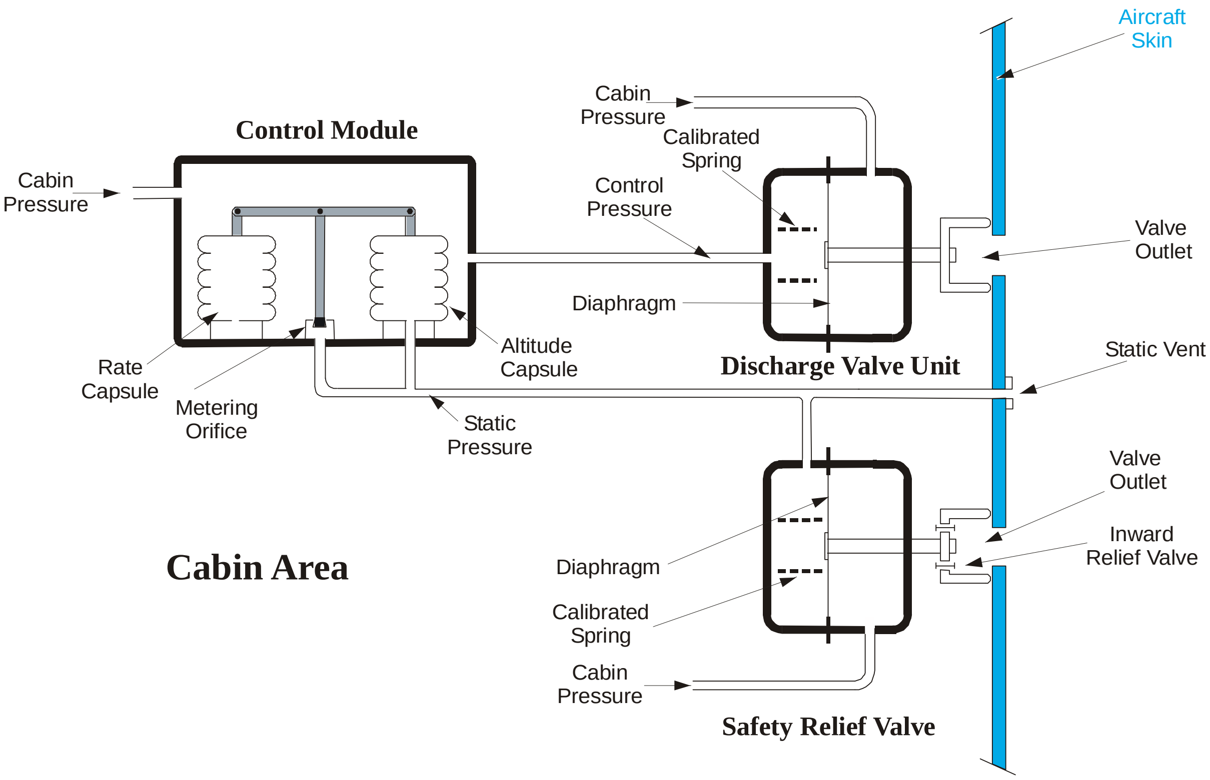
8. **Safety Devices.** Most pressurization systems use multiple discharge valves. In addition, at least two safety devices are incorporated, to provide for cabin pressurization malfunction:

a. **Safety Relief Valve.** If the cabin differential pressure approaches the maximum permitted, it will be sensed by a safety relief valve, which is independent of the normal control system. The safety relief valve outlet will then automatically open to dump air outwards, thereby reducing the interior cabin pressure.

b. **Inward Relief Valve**. An inward relief valve is necessary in case the outside air pressure becomes greater than the cabin pressure (e.g. in a very rapid descent). The inward relief valve is sometimes combined in the same unit as the safety relief valve.

9. **Pressurization Failure**. Passenger-carrying aircraft have a limited oxygen supply available for crew and passengers. This will be used in the event of pressurization failure at altitude. If the pressurization system fails, the aircraft must immediately descend to an altitude below the normal safe cabin altitude.

4-5 Fig 3 Pressurization Control System – Schematic

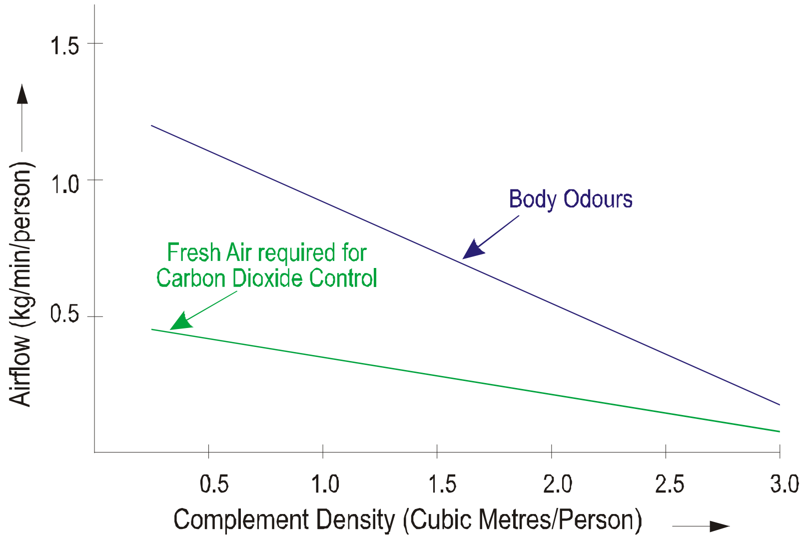


Air Conditioning Systems

10. The air conditioning system must be able to provide a supply of air sufficient to satisfy ventilation and pressurization requirements, at a temperature and humidity necessary to maintain cabin and cockpit conditions at a comfortable level.

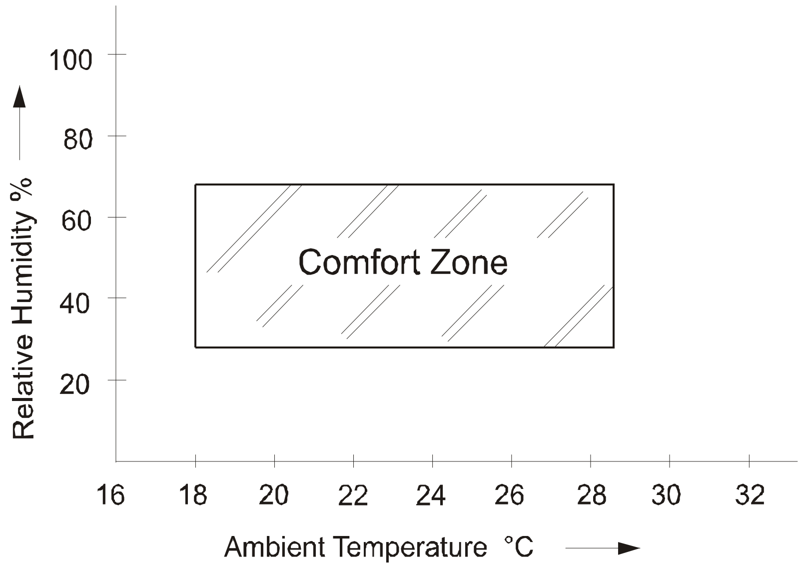
11. **Composition of the Cabin Atmosphere.** To prevent the build up of carbon dioxide, water vapour, dust, fumes and odours, cabin atmosphere must be changed continuously by the ventilation system. The rate of ventilating airflow is dependent upon the volume of cabin space per occupant (the space per occupant, in cubic metres, is termed 'complement density'). The smaller the complement density, the higher must be the airflow. This is illustrated at Fig 4. In passenger aircraft, an airflow of approximately 1.5 kg/min is normally provided. This usually comprises 50% fresh air and 50% recirculated air. The air is discharged into the cabin to create a general circulatory flow, although higher speed airflows are generally provided for each passenger and crew member through individually controlled facilities. In the more restricted volume of a combat aircraft cockpit, a flow of up to 5 kg/min is usually provided. In normal flight conditions, 80% of this air is arranged to circulate directly around the crew, whilst the remainder is used for demisting and general cockpit ventilation.

4-5 Fig 4 Ventilation Flow Requirements



12. **Temperature and Humidity**. The range of ambient temperatures and humidity in which a crew can operate comfortably without rapidly becoming fatigued is known as the 'Comfort Zone' (see Fig 5). Air conditioning systems control both temperature and humidity within the cabin to remain within this zone.

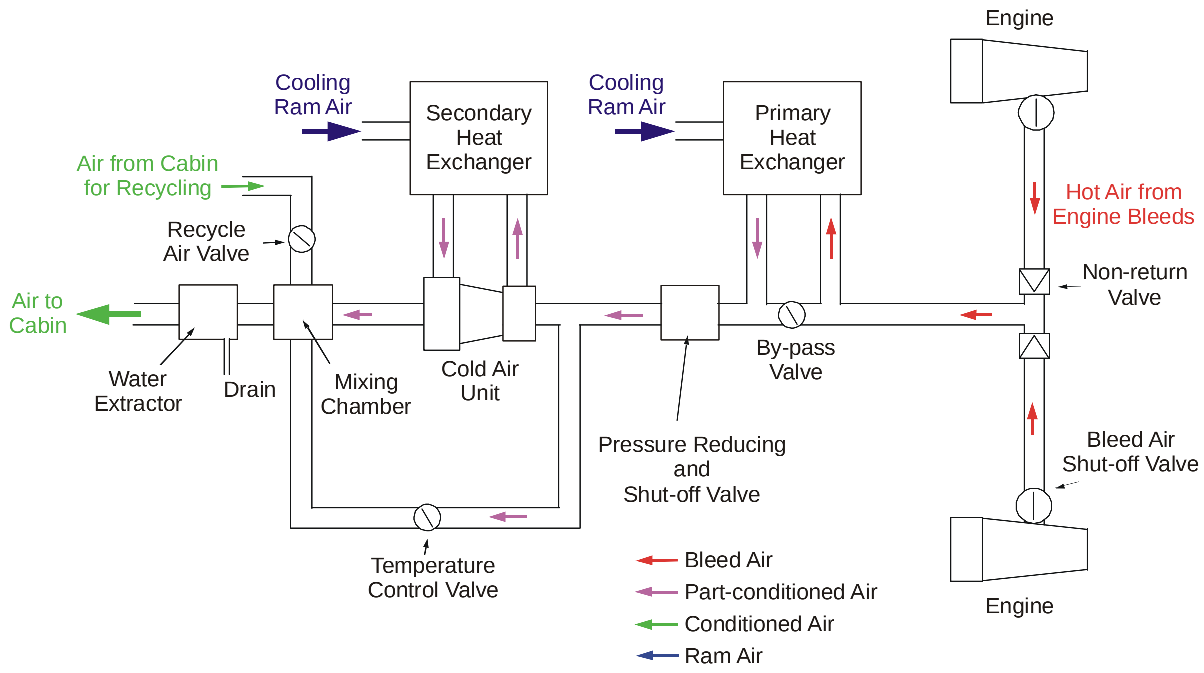
4-5 Fig 5 The Comfort Zone



13. **Conditioning Systems**. Fig 6 shows a typical air conditioning system. The hot air from the engine compressor is cooled by routing it through primary and secondary heat exchangers, and a cold air unit, as necessary. Within the heat exchangers, the hot air is cooled by indirect contact with cold ram air. The cold air unit uses principles of expansion and energy conversion to reduce the air temperature. To provide the final airstream at the temperature required for cabin conditioning, the cooled air passes through a mixture chamber, where it is combined with hot moist by-pass air and recycled cabin air. Any water resulting from the cooling process is separated from the air before it enters the cabin. Most water separators utilize momentum separation techniques to remove the majority of water from the airstream.

This type of separator comprises a bank of swirl vanes, or louvres, and a coarse mesh coalescent filter. As the air passes through the unit, its velocity and momentum are changed and any water held within the air coalesces into droplets. These droplets are then separated from the main airstream and are ducted overboard. Most air conditioning systems provide control and adjustment of air temperature and control of air humidity. Some provide positive filtration of the incoming air, although the majority achieve a degree of filtration only as a secondary function of the water extraction devices used for humidity control.

4-5 Fig 6 Typical Air Conditioning System

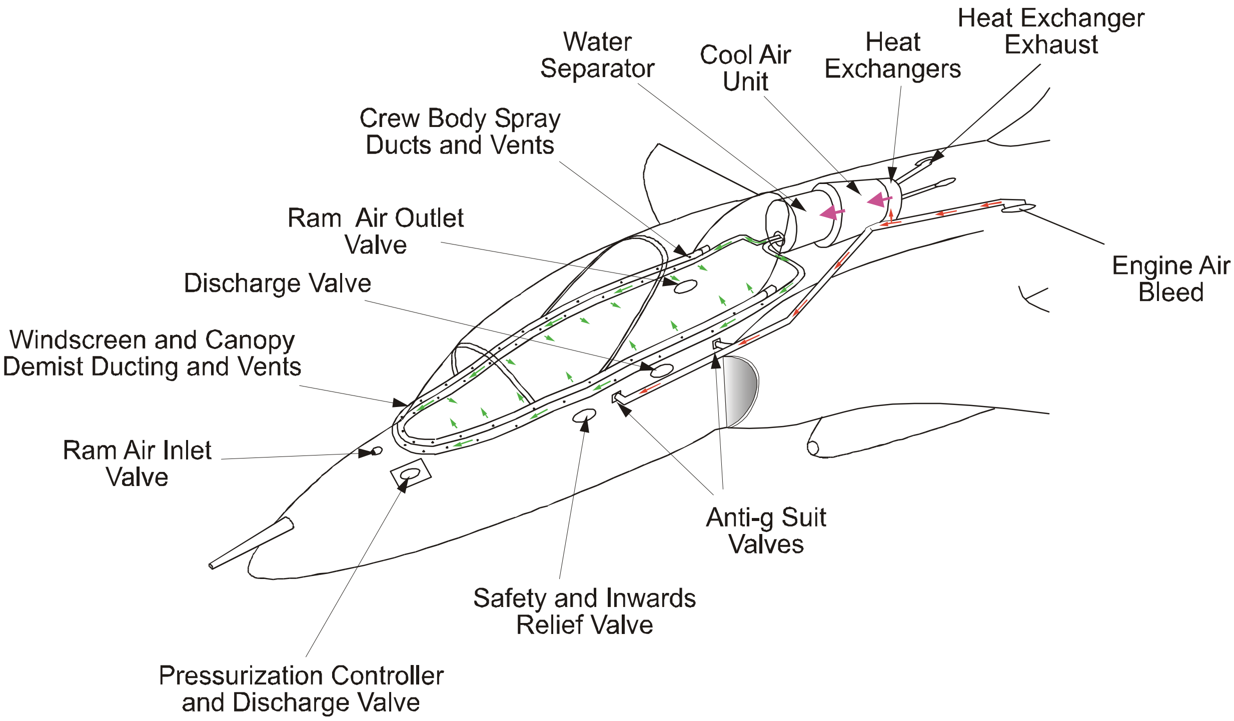


14. **Conditioning Failures**. Multiple air conditioning systems are usually employed to provide an element of redundancy. However, if the pressurization air supply is suspended (during an airborne emergency, for instance), ambient ram air will normally be used as an alternative supply for conditioning. The temperature in the cabin can therefore by expected to fall rapidly, depending upon the ambient air temperature.

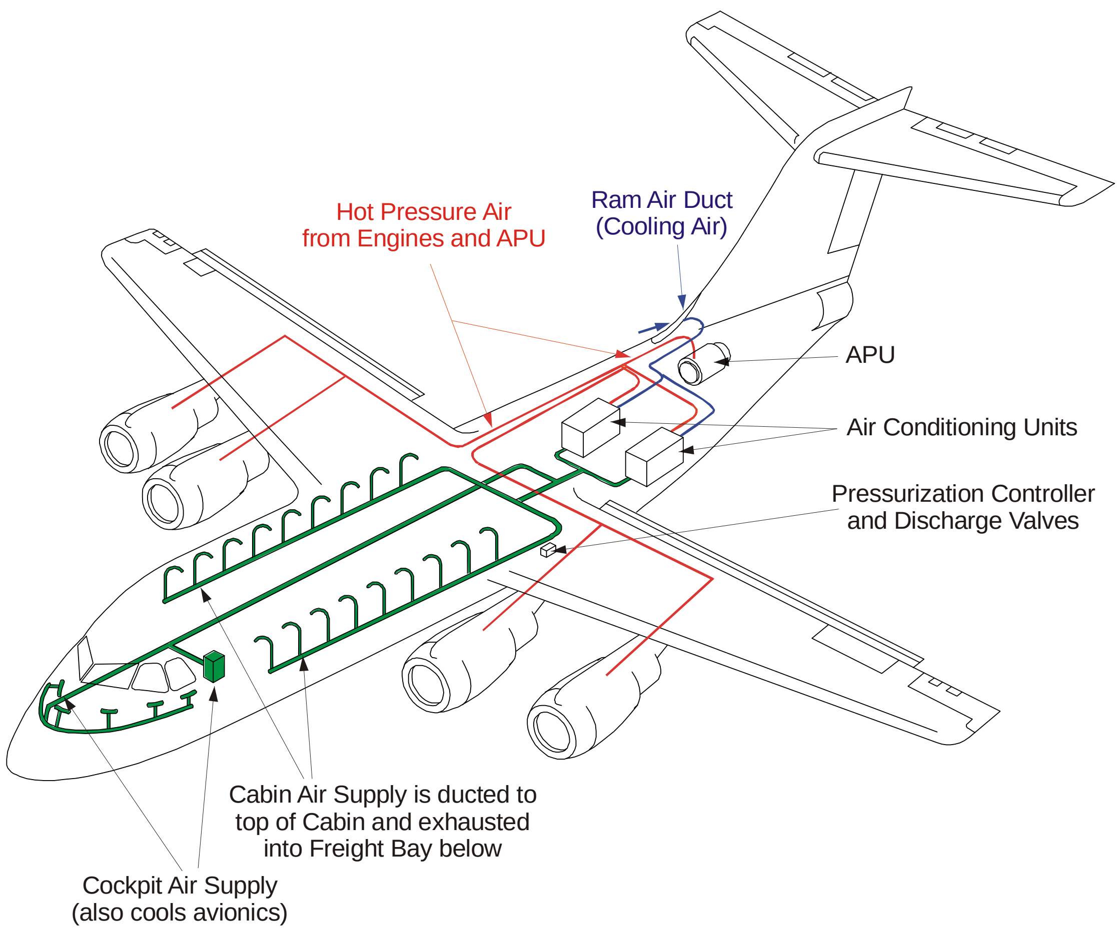
Aircraft Configuration

15. Pressurization and air conditioning systems need to be self-contained and, ideally, duplicated to provide an acceptable degree of conditioning in the event of one failure. For combat aircraft, they need to be simple, compact and automatic (or semi-automatic). For transport aircraft the systems must have large capacity and a flexible control system, to cope with widely varying conditions. Ventilation systems need to be capable of operating on the ground. It should be noted that supplies of pressurized and conditioned air are used for other tasks, including pressurization and cooling of avionics modules, sealing of canopies, and supply of pressure for aircrew anti-g clothing. Fig 7 shows a representative system installation for a two seat, single engine combat/training aircraft, whilst Fig 8 shows a typical system for a multi-engine transport aircraft.

4-5 Fig 7 Combat Aircraft System Configuration



4-5 Fig 8 Transport Aircraft System Configuration



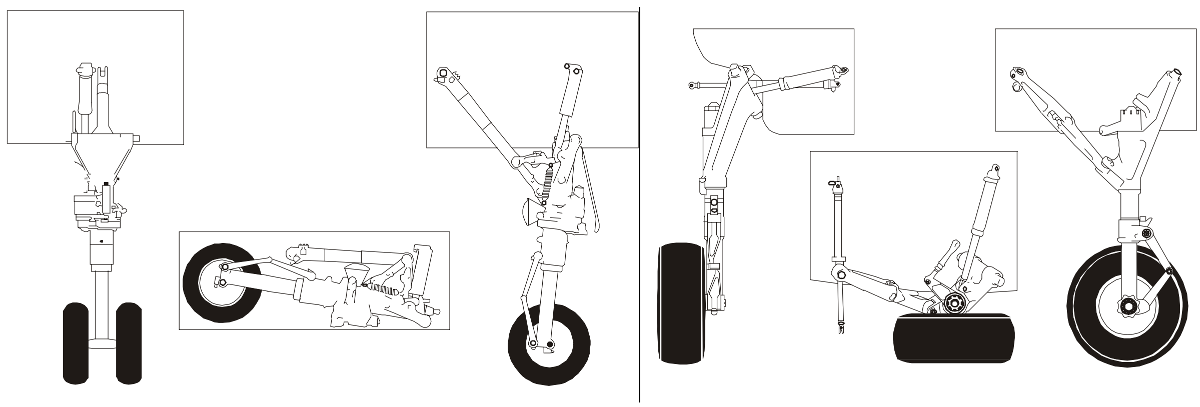
CHAPTER 6 - UNDERCARRIAGES

Introduction

1. The undercarriage of an aircraft includes the wheels, tyres and brakes as well as the main undercarriage leg components. It performs the essential function of providing an interface between aircraft and ground during landing, take off, ground manoeuvring and whilst at rest. However, it is completely redundant during flight, and therefore the design of an undercarriage is usually a critical compromise between optimising performance on the ground and minimizing weight and drag penalties in the air. Examples of the extremes of this compromise range between the provision of the minimum for a Remotely Piloted Vehicle - a detachable wheeled dolly for the aircraft to take off from and a parachute to lower it safely after flight - to the more generally serviceable - such as the undercarriage of the Tornado, shown at Fig 1 - which allows the aircraft to be landed at high weights and on a wide variety of surfaces, and to be manoeuvred rapidly and precisely between the runway and its dispersal area for replenishment, prior to dispatch on further sorties.

4-6 Fig 1 Undercarriage of the Tornado Aircraft

Fig 1a Tornado Nose Landing Gear **Fig 1b Tornado Main Landing Gear**



Runway Pavements

2. The increase in aircraft performance has led to the need for ever-higher landing speeds and weights. Eventually, the practical and tactical limitations of stronger and longer runways were reached, and research and development were then concentrated on improving the aircraft rather than the runways. This gave rise to the introduction of STOL and V/STOL technology, a trend which continues for military and many civil transport aircraft. As discussed at Volume 2, Chapter 21, standard systems are now available for classifying and matching the landing requirements of aircraft and the strength (load bearing capabilities) of runways.

Design Considerations

3. Principal factors which govern the design configuration of a particular undercarriage are:

a. The aircraft's role and its intended theatre of operation - for example, the requirements for strategic aircraft operating from well founded airfields are considerably different from those of tactical STOL aircraft intended to operate from semi-prepared strips.

b. The configuration of the aircraft and its intended performance/cruise speed - for example, high wing, high-speed aircraft impose greater design problems than do low wing, low speed aeroplanes and helicopters.

c. The numerical factors - for example, landing speeds and weights, permissible length of landing run and cross wind landing/take off capability - all have considerable influence upon undercarriage design.

Typical Configurations

4. The general design configuration for an undercarriage emerges from consideration of the above factors:

a. Physical strength of the components necessary to withstand landing, braking, and crosswind loads. The strength parameters are set out in defined design standards.

b. Shock absorber performance capable of accepting the maximum intended sink rate of the aircraft onto the ground, the type of surface over which the aircraft will taxi and the speed of turning during taxi.

c. Fixed (stronger, simpler and lighter) undercarriage or a retractable (less drag) undercarriage.

d. Streamlining and provision of undercarriage doors necessary to reduce drag during flight.

e. Dimensions of the ground track needed to provide stability during landing and taxi.

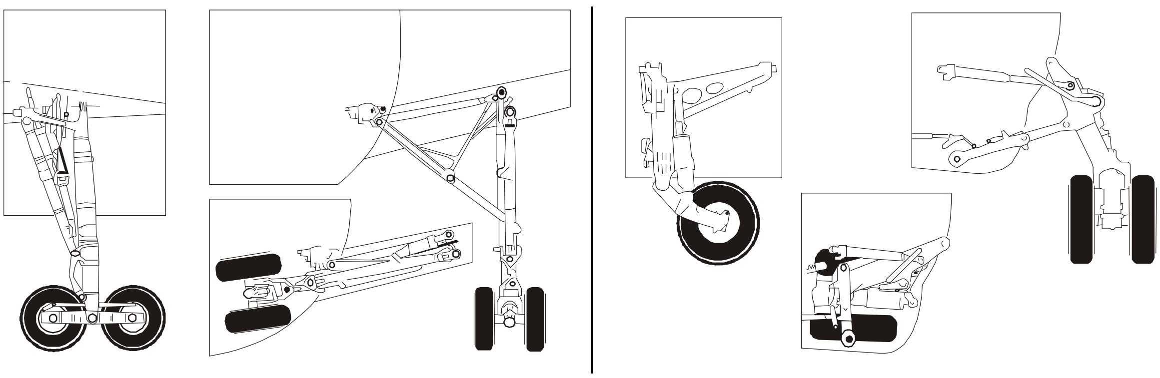
f. Fuselage or wing space available for stowing and attaching the gear.

g. Basic configuration. For instance, the standard tricycle for good ground manoeuvre and stability, bicycle (with outriggers) for strength and relative ease of stowage or tail wheel for simplicity and low cost. The most appropriate undercarriage for a small helicopter may be a pair of skids, despite the complications which these impose upon ground handling.

The retractable main undercarriages of the Airbus A310 and the BAe 146 are shown in Fig 2. The A310 undercarriage is a very simple assembly which retracts into a space in the wing root and fuselage. However, that for the 146 (a high wing aircraft) retracts into the bottom of the fuselage (to minimize its length and thus maximize its strength) and extends sideways (to give a good wheel track for stability). This has resulted in the complex, multi-pivoting mechanism shown.

4-6 Fig 2 Examples of Retractable Main Undercarriages

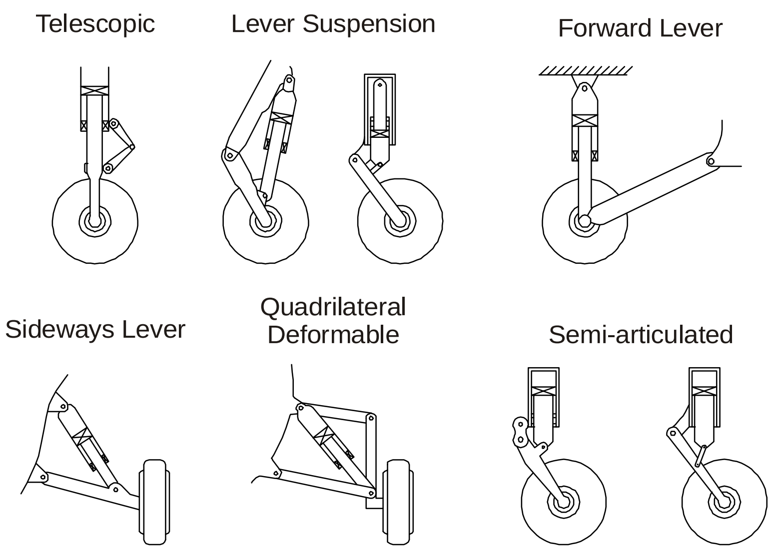
Fig 2a A310 Main Landing Gear **Fig 2b BAe 146-200 Main Landing Gear**



Undercarriages

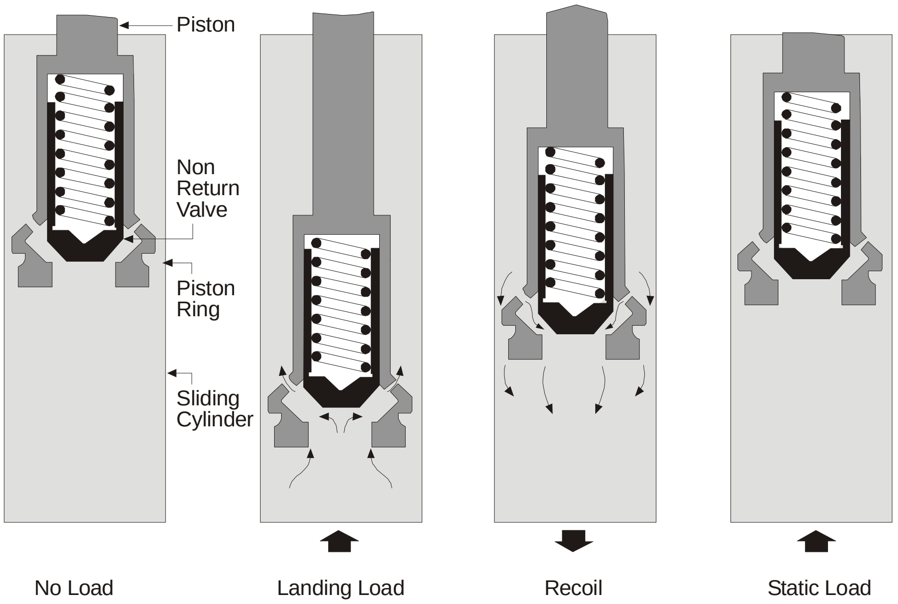
5. **Undercarriage Legs**. The undercarriage leg performs the functions of absorbing the forward, aft and side loads of landing and braking. Nose and tail undercarriages also require to swivel to allow the aircraft to be steered. These functions must be performed by as few components as possible, and Fig 3 shows several undercarriage legs (in schematic form) designed to achieve these objectives.

4-6 Fig 3 Basic Undercarriage Leg Configurations



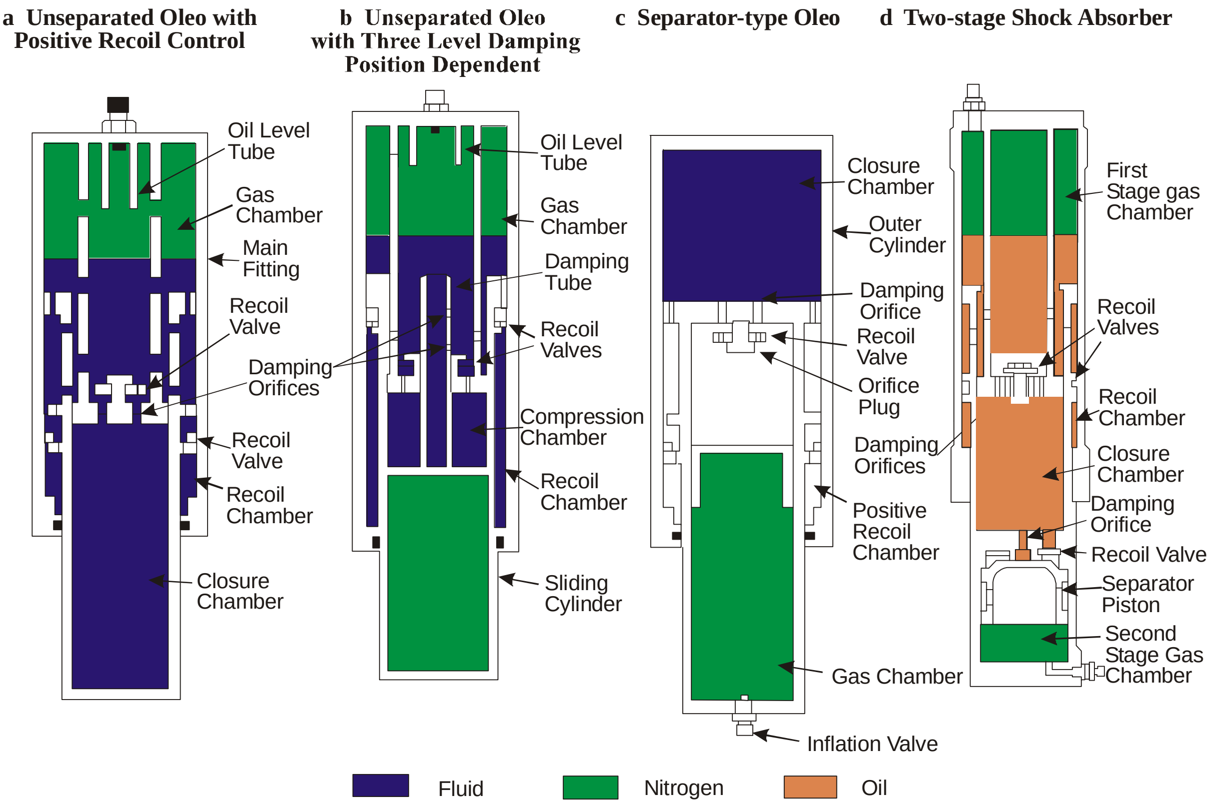
6. **Shock Absorbers.** The shock absorber is the most complex component of the undercarriage. Its role is to dampen the shocks of landing and taxiing and of movement over uneven runway pavements. Two basic types of shock absorber are available, one utilizes the compressibility of oil at pressures above 700 bar to damp out shocks, whilst the other utilizes various combinations of oil and nitrogen under pressure to provide damping. The principle of the oil filled (liquid spring) absorber is shown at Fig 4.

4-6 Fig 4 Principle of the Liquid Spring



On landing, movement of the leg is restricted by the slow rate at which oil is able to pass through the damping orifices into the upper chamber of the liquid spring. If large shocks are experienced, the oil remaining beneath the piston is compressed until its pressure exceeds the loading of the piston non-return valve spring. At this point, a larger volume of oil is released round the valve, thus damping out the larger landing shocks. On the recoil, oil is forced back below the piston through the small damping orifices. Variations on the oil/gas (oleo-pneumatic) absorber are shown at Fig 5.

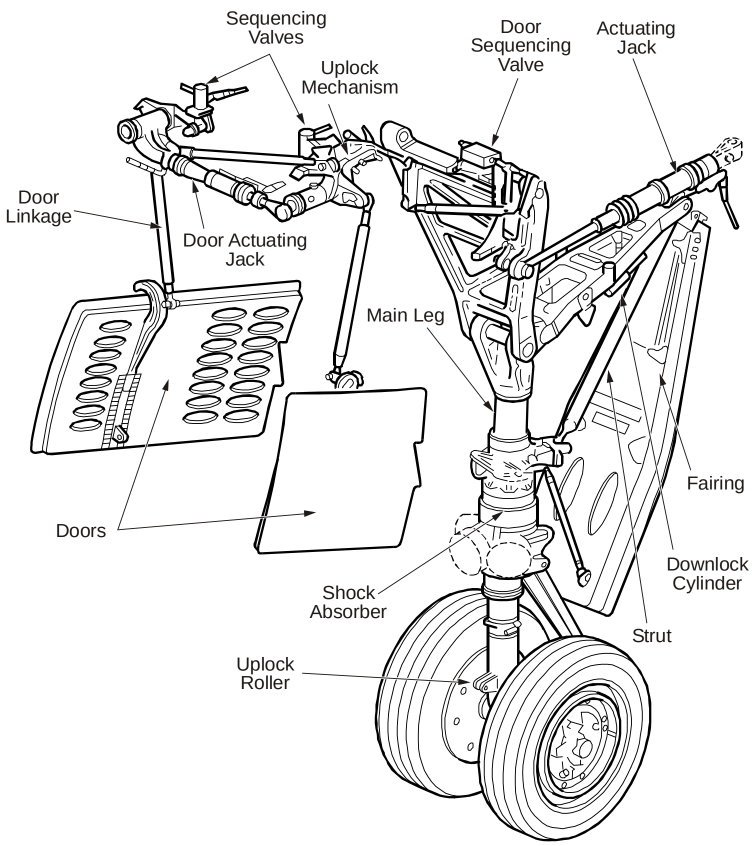
4-6 Fig 5 Oleo-pneumatic Absorbers



The combination of oil and gas provides a more effective method of shock absorption, enabling a reduction in component size and weight to be made for the same performance. Shock absorbers must be designed so that they never reach full extension or closure under any operational load condition, otherwise the undercarriage will momentarily become rigid passing very high peak loads into the aircraft structure. The nose undercarriage is subjected to a wider range of conditions than is the main undercarriage, because of centre of gravity movement and pitching moments caused by braking reactions. For this reason, two stage shock absorbers similar to that shown at Fig 5d are often fitted to the nose to provide the greater required range of operation.

7. **Retraction Mechanisms**. Although retracting undercarriages are usually configured specifically for a particular aircraft type, all have similar basic features. The more significant of these are highlighted in Fig 6 and are described in the following sub-paragraphs.

4-6 Fig 6 Undercarriage Retraction Mechanisms



a. **Doors and Fairings**. To avoid undesirable aerodynamic effects, the receptacles or wells in which retracting undercarriages are housed require to be faired over after the gear has been retracted and, as far as possible, after the gear has been extended. This is achieved by the fitment of doors which are either mechanically attached to the undercarriage legs or are sequenced to open and close at appropriate times during the retraction or extension cycle.

b. **Jacks and Linkages**. Because of the extremely high power to weight and power to volume ratios which their use offers, hydraulics are used to power all conventional retraction mechanisms.

Typically, 3 or 4 hydraulic jacks operating in a controlled sequence will raise or lower the undercarriage leg, open and close the doors and lock the undercarriage in the fully up or down position. A series of mechanical linkages transfer jack forces to separate areas of the mechanism.

c. **Up and Down Locks**. When fully retracted, the undercarriage must be positively restrained against 'g' forces in flight. Equally, when fully extended it must lock solidly to absorb landing loads. Mechanical locks are provided to achieve these requirements. A typical 'up' lock is shown at Fig 7a. It comprises a simple rotating jaw which turns to lock round a pin on the undercarriage. The lock is turned into position by engagement with the pin, as the undercarriage moves to its fully retracted position. When the undercarriage is lowered, the lock is opened hydraulically to release the pin. Because it is critically important that 'up' locks release, even in the event of total hydraulics failure, secondary and sometimes tertiary opening methods are provided. These usually employ an electrical solenoid, although light aircraft are sometimes fitted with manual release mechanisms operated by cables from the cockpit. Whilst the 'up' lock must be capable of supporting the full weight of the undercarriage, it can be arranged for the 'down' lock to take none of the landing forces.

4-6 Fig 7 Up and Down Locks

Fig 7a Up Lock

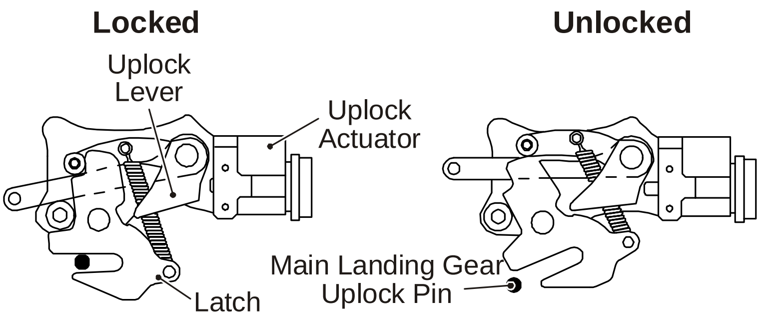
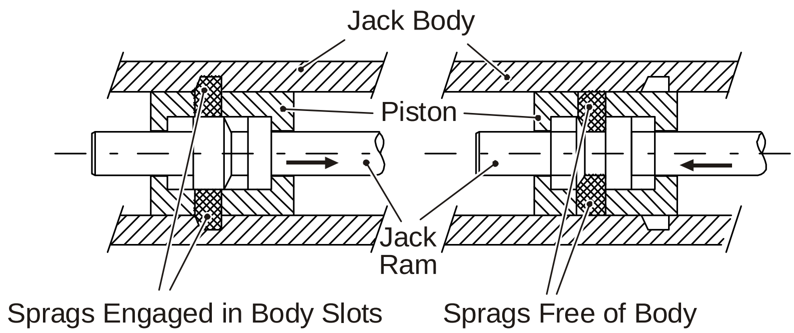


Fig 7b Down Lock



Two types of lock are in common use. The one shown in Fig 6 is a small hydraulic 'bolt' which geometrically locks a hinged lever when the undercarriage is fully lowered. Thus, the lever is locked in its fully unhinged position, taking all landing forces and imposing none on the bolt. The other is integral with the extension jack and mechanically locks the jack in its fully extended position. Fig 7b includes a simplified diagram of the device. An integral lock offers the many advantages of simplicity.

d. **Sequencing.** Complex folding and unfolding movements of the undercarriage and opening and closing of the doors must all be sequenced precisely to prevent damage and failure occurring. This is achieved by fitment of hydraulic valves or electrical switches in the system which do not permit one part of the sequence to commence until the preceding part has been completed.

8. **Controls and Indications.** Retraction or extension is initiated by operation of simple cockpit control, usually in the form of a single lever or switch. The international standard indications provided in the cockpit, and usually integrated in the switch unit, consist of 3 green lights to show when each of the undercarriages are locked down and 3 red lights to show that the undercarriages are unlocked - that is moving between their up and down positions. In most aircraft, a series of interlocks and safeguards are incorporated in the control system to prevent inadvertent operation on the ground or at too high an air speed, and to reduce crew workload during landing and take off. 'Weight on wheels' or 'nutcracker' switches, activated by deflection of the undercarriage on the ground, are used to prevent operation of the retraction mechanism and to unlock operation of the steering, braking and thrust reverser systems. The signals from these switches are used in other aircraft systems to prevent their operation on the ground or to initiate their operation immediately after the aircraft has taken off.

9. **Steering.** Whilst taxiing and during initial stages of take off and final stages of landing, airspeeds are too low for rudder authority to be maintained. A system of differential operation of the main wheel brakes, achieved by manipulation of brake pedals attached to the rudder bar, provides a steering force in most aircraft at these lower speeds. However, precise ground manoeuvring is required in crowded aircraft dispersal areas, and a system of positively steering the aircraft wheels is therefore necessary. In light aircraft, such steering is often provided through direct mechanical linkage of the rudder pedals to a steerable nose or tail wheel. The majority of high performance aircraft utilize steering systems in which the nose wheel can be controlled hydraulically during taxi, through a small tiller or wheel in the cockpit. To avoid such a steering system providing an unwanted input at the points of lift off and touch down, the steerable nose wheel is automatically aligned centrally whenever aircraft weight is lifted off the wheels.

10. **Emergency Extension.** To avoid the inevitable consequences of a 'wheels up' landing, design standards require that all aircraft fitted with retractable undercarriages are equipped with at least one alternative driving force for extending the undercarriage. In the majority of aircraft, this is achieved by pressurizing the undercarriage hydraulic extension system either directly by the release of compressed nitrogen into the system or indirectly by release of nitrogen into an associated booster system. Operation of the emergency extension control lever releases the nitrogen and affects any necessary changes in hydraulic valve settings. Many aircraft are equipped with a secondary emergency system which releases the undercarriage up lock allowing the undercarriage to extend under gravitational and aerodynamic forces. Use of the emergency systems prevents subsequent retraction of the undercarriage, until necessary engineering actions have been carried out on the ground.

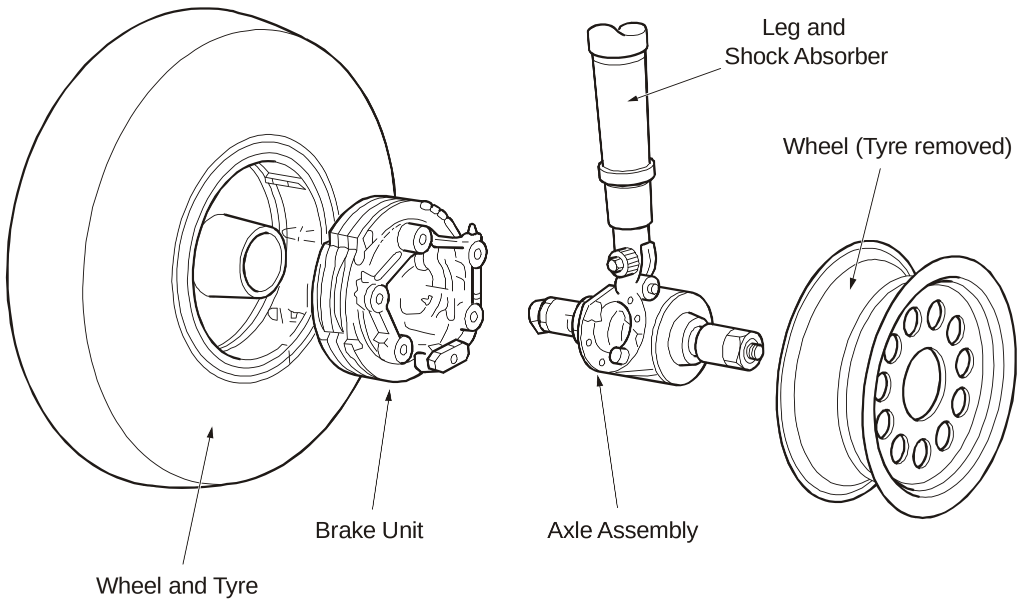
Wheels

11. The design criteria for aircraft wheels are:

a. Light weight.
b. Minimum size.
c. Easy tyre replacement.
d. Accommodation for the brake unit and dissipation of the heat generated during braking.
e. Good fatigue resistance.

Aircraft wheels differ in many ways from those fitted to road vehicles. For instance, aircraft wheels are made in 2 halves which unbolt to allow the fitment of tyres without stretching their beading over the wheel rims. Also, the wheels house the wheel bearings, unlike automobile practice in which a separate axle houses the bearings and the wheels bolt to this axle. A typical wheel and axle arrangement is shown at Fig 8.

4-6 Fig 8 Aircraft Twin Main Wheel and Axle

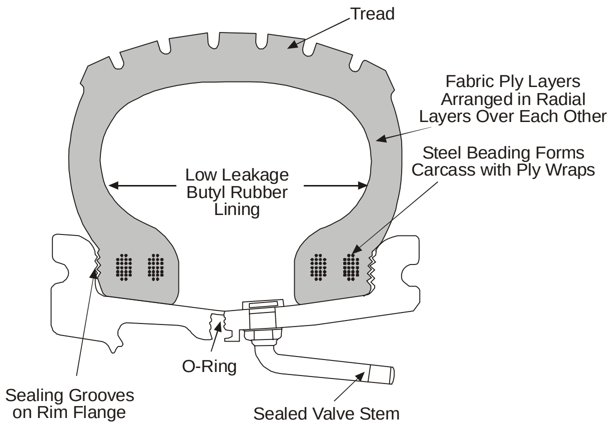


Tyres

12. Aircraft tyres must be able to withstand higher loads than road tyres, but they are not required to be capable of continuous running over great distances. However, the general structure of aircraft and road tyres is similar, and radial ply tubeless tyres are now used almost exclusively in both applications. Radial ply tubeless tyres offer higher strength, lower weight, cooler running, and better overload capabilities than the earliercross ply tyres fitted with inner tubes. The construction of the radial tyre is shown at Fig 9. Because the load bearing carcass and the tread are effectively two separate components of the tyre and the tread tends to wear out before the carcass, aircraft tyres are retreaded as a matter of course to extend their life. Most tyres used on fixed wing aircraft are retreaded several times before their carcasses require to be scrapped. Under the high impact loads experienced during landing, tyres tend to creep by small

distances around the wheels. This presents no problem with tubeless tyres, but if tubed tyres creep, the valve stem of the inner tube which is firmly attached to the wheel is stretched and will eventually fracture. For this reason, white 'creep' witness marks are painted on tubed tyres at fitment, so that the degree of creep can be monitored. Aircraft nose and tail wheel tyres are constructed to be electrically conductive, by the addition of carbon in the rubber mix. This enables the static charges built up in an aircraft during flight to be discharged automatically on landing.

4-6 Fig 9 Construction of a Radial Ply Tyre



Braking Systems

13. **Principles.** Stopping an aircraft requires the rapid dissipation of large amounts of kinetic energy. The energy is dissipated by conversion to heat energy in the wheel braking system and by being used to do work against applied loads. Such loads include drag (from aerodynamic devices such as flaps and spoilers) and opposing forces provided by reverse thrust devices or propeller reverse pitch. In extreme cases, brake parachutes or external retardation devices such as arrester wires are also used to absorb the kinetic energy. Typically, wheel brakes, aerodynamic devices, and thrust reversers absorb equal amounts of energy during a normal landing.

14. **Heat Dissipation.** Temperatures of up to 1400 ºC are reached in high performance braking systems. To prevent damage to the tyres and undercarriage structure, the heat energy must be dissipated rapidly into the surrounding air. If this does not happen, as can be the case after an aborted take off and subsequent long taxi back to a dispersal, the tyres can overheat and burst, and brake fires are likely to occur. To prevent tyres bursting due to overheating, wheels are fitted with fusible plugs which melt at a preset temperature. This allows the tyre to deflate at a steady controlled rate.

15. **Design Objectives.** The following general specification is typical of the design objectives for combat aircraft braking systems:

a. Absorb the energy of a normal landing or a rejected take off, due allowance being made for aerodynamic and rolling drag and for engine thrust decay time and idling thrust developed during the landing run.

b. Provide a deceleration of 0.3g.

c. Dissipate the heat generated during a normal landing sufficiently quickly to allow operational turn round of the aircraft.

d. Have minimum friction material wear to give a long life.

e. Have automatic adjustment and visible wear rate indication.

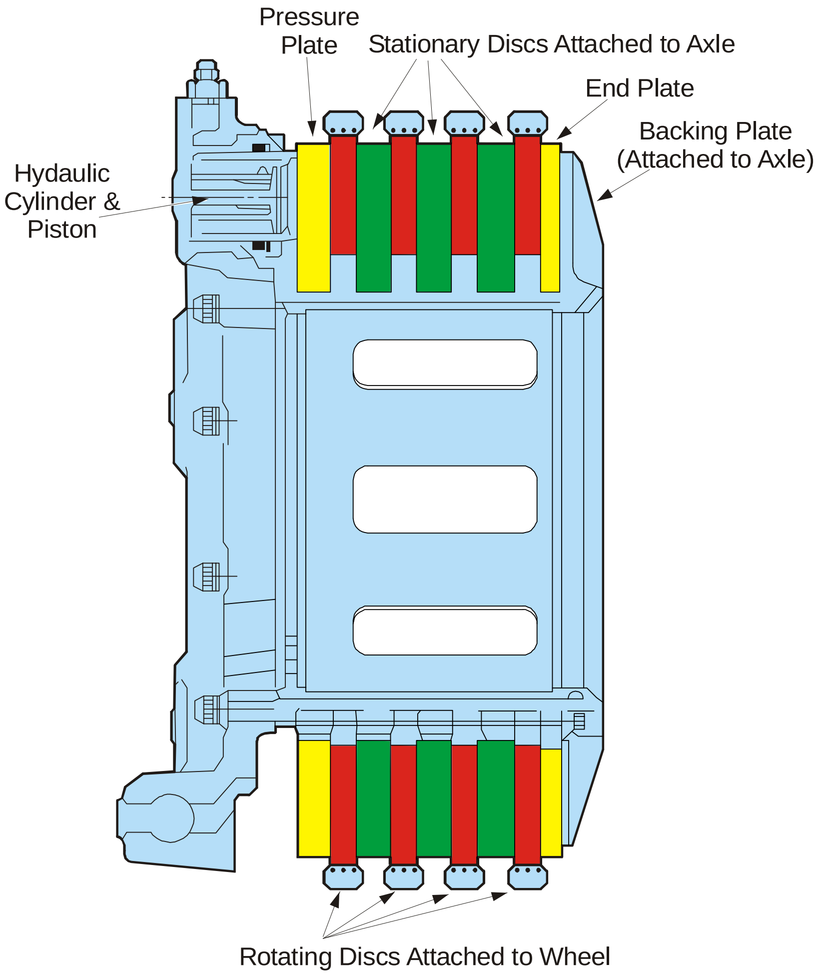
f. Provide a static drag force sufficient to enable engines to be run up to full dry power without wheel rotation.

g. Permit ground manoeuvring without the use of excessive brake pedal pressures and without snatching.

h. Provide a completely independent method of hydraulic brake application capable of meeting all of the above criteria.

16. **Configuration.** Most aircraft are equipped with hydraulically operated disc brakes, although drum brakes are sufficiently effective for light aircraft. Disc brakes offer the advantages of higher surface area for contact between the brake material and the rotating surfaces and larger capacity heat sinks to absorb the heat generated during braking. High performance disc brakes are constructed as multiple stacks of discs made from carbon composites which are able to operate at the necessary temperatures. A typical multiple disc unit consists of four or more rotors keyed to the inside of each main wheel, and five or more stators assembled on to splines of each main undercarriage axle assembly. Fig 10 shows such a brake assembly in situ. Operation of the brakes is usually through a single selection lever. Pedals attached to the pilot's rudder bar direct differential hydraulic pressure to the main wheel brake units to provide steering. Hydraulic pressure operates either directly or through a servo system upon the brake units. The pressure causes rotor and stator discs to be pressed together, and the resulting friction provides a retarding force to the main wheels generating heat in the process. Rotor discs are usually constructed in segments which allow a small amount of deflection to take place. This reduces stresses and prevents the discs cracking.

4-6 Fig 10 Brake Assembly



(Note: Colours used for clarity only)

17. **Emergency Braking Systems**. Hydraulic braking systems are normally configured to operate from two different hydraulic power sources. Thus, in the event of one power source failing, the other can be selected either automatically or manually. In addition, most electronic systems have fail-safe characteristics which allow acceptable standards or braking to be achieved through simple pulse modulation of hydraulic pressure in the event of failure of the electronic control system.

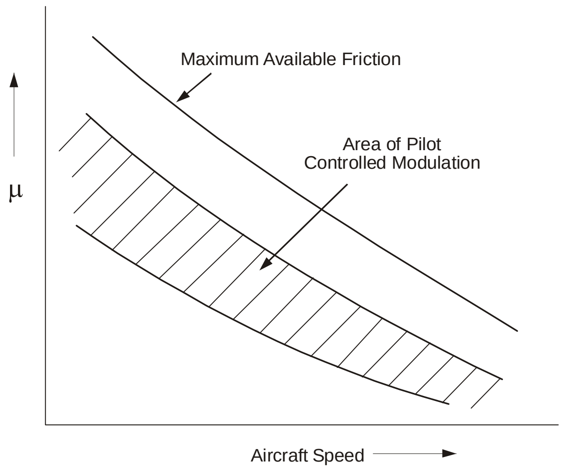
18. **Parking Brakes.** Braking during periods when the aircraft is parked is provided by permanent pressurization of the brake hydraulic circuits. This is normally achieved by utilizing a hydraulic accumulator pressurized by nitrogen. In some simple braking systems, the accumulator also provides the source of emergency braking system pressure, albeit for only a limited number of brake applications after which pressure in the accumulator becomes exhausted.

Braking Control and Anti-skid Systems

19. **Braking Dynamics**. To minimize the landing run, it is imperative that brakes apply maximum retardation force without causing the adhesion between tyre and pavement to be exceeded thus causing the aircraft wheels to skid. The point of skidding is dependent upon the condition of the runway surface,

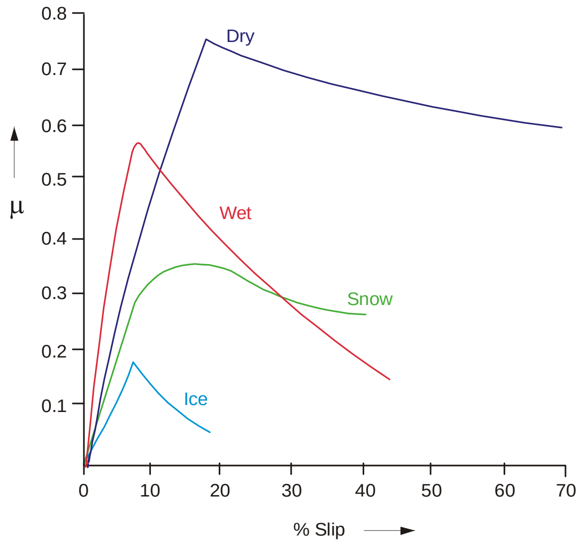
the vertical load which the aircraft tyres exert on the ground and the retarding force applied by the brakes. The mathematical relationship between vertical load and retarding force is termed 'μ'. Because vertical load is inversely proportional to the aerodynamic lift acting on the aircraft, it follows that μ will increase as aircraft speed decreases. Fig 11 shows this relationship for constant runway conditions.

4-6 Fig 11 Relationship Between μ and Vertical Load on the Tyres



Whenever braking force is applied to the tyre, a degree of slip occurs between the tyre and the surface of the pavement. This is defined in terms of the difference between the rotational speed of a braked wheel and the rotational speed of a similar free rolling wheel. It is expressed as a percentage. The value of μ varies with wheel slip, and Fig 12 shows the relationship between maximum available μ and wheel slip for varying conditions on the same runway.

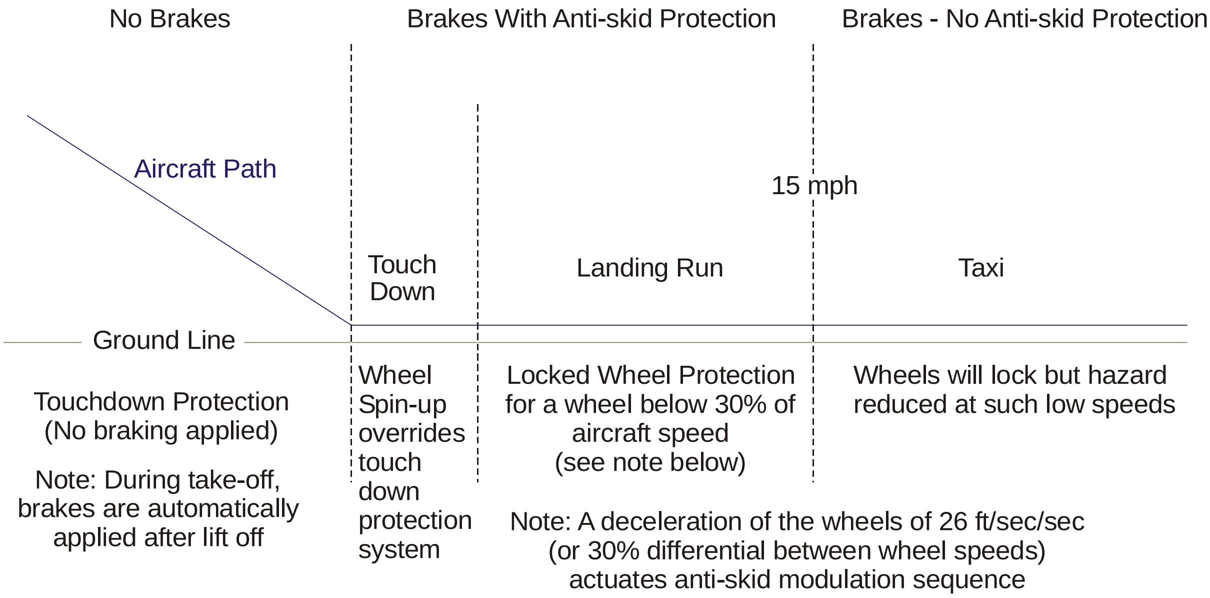
4-6 Fig 12 Relationship Between μ and Wheel Slip for Varying Runway Conditions



20. **Control Systems.** As can be deduced from the above considerations, the application of maximum braking effort to minimize the landing run requires the solution of complex dynamic equations, balancing braking forces with speed, weight and runway conditions. Electronic sensing has permitted all phases

of the braking process to be inter-related and fail safe over-rides to be employed. A typical brake system operation profile is at Fig 13.

4-6 Fig 13 Typical Brake System Operation Profile



21. **Anti-skid Systems**. Early mechanical anti-skid systems utilized the inertia of a small flywheel to sense rapid changes of main wheel rotational speed such as occurs during a skid. On sensing a skid, the systems reduced hydraulic pressure - thereby reducing braking effort and stopping the skid. They reinstated pressure when skidding had reduced. The resultant cycling between skid/no skid conditions caused the braking pressure to continuously pulse or modulate, and the technique became known as brake modulation. Subsequent electrical systems used sensors to measure wheel speed and compared the speed to a datum. The use of simple electronic processing allowed a controlled profile of modulation to be achieved instead of the on/off characteristics of the earlier mechanical systems, and considerable improvements in braking efficiency were achieved. Modern anti-skid systems utilize control technology to vary not only the frequency of modulated braking pulses but also their amplitude (pressure). Thus, the systems can maintain braking forces at a level immediately below that which would cause skidding for all speeds and surface conditions. The systems also hold the brakes off until after touch down and wheel spin up has occurred, and similarly apply braking to spin down the wheels after take off and undercarriage retraction has taken place. Thus, the systems can relieve the crew of much of the workload of brake management at the critical periods of landing and take off.

CHAPTER 7 - AUTOMATIC FLIGHT CONTROL SYSTEMS

Introduction

1. **The Problem.** Since the first days of flight, the need to compromise between aircraft performance and controllability of the aircraft has formed a central factor influencing specification and design. The pilot has to contend with a demanding workload, whilst maintaining a span of concentration throughout the whole flight, sometimes of long duration. At the same time, a rapid response is necessary to counter any adverse changes in aircraft attitude. Historically, therefore, major compromises in performance have been necessary in order to obtain an acceptable balance between stability and controllability.

2. **The Solution**. The problems posed by workload, speed of reaction and fatigue have been solved gradually, by the development and subsequent evolution of automated flight control systems. Such systems augment the control applied directly by a pilot, whilst ensuring that full command of the aircraft is retained. The earliest systems consisted of automatic stabilization devices to counter gross changes in aircraft trim. Later systems, known as 'automatic pilots', provided stability in three axes and ensured that a selected heading and altitude were maintained.

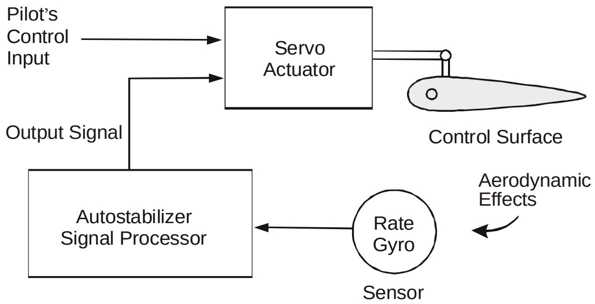
3. **Progression towards Maximum Performance.** The continuing advances in technology, particularly the development of computing and fly-by-wire controls, have made it possible to combine the outputs of individual avionics systems. This has resulted in the introduction of the fully integrated automatic flight control system (AFCS), and its successor, the flight management system. The latter has permitted advances to be made towards achieving maximum theoretical performance during a flight.

AUTO STABILIZERS AND BASIC AUTOPILOTS

Auto stabilizers

4. An auto stabilizer will maintain the aircraft in an attitude as initially set up by the pilot. This is achieved by sensing any variation from the prescribed attitude, and employing a feedback control circuit to eliminate the unwanted change. Auto stabilizers are sometimes known as stabilization augmentation systems (SAS). Fig 1 illustrates a single-channel auto stabilizer system.

4-7 Fig 1 Auto stabilizer Feedback System



5. The main components of the simple auto stabilizer illustrated in Fig 1 are:

a. **Sensor.** Aircraft movement in the pitch, roll or yaw axes is sensed by an associated rate gyroscope.

b. **Signal Processor**. The signal processor is usually electronic, but some mechanical systems do exist. Typical functions within an auto stabilizer processor include:

(1) **Amplification.** The signal from the sensor will be amplified.

(2) **Phase Advance.** In any practical control loop, there will always be a time delay between the detection of a disturbance and the application of corrective action. Since disturbances in the aircraft flight path will result in oscillatory motions, it is easy to use a phase advance network to ensure that the corrective action applied at the control surface occurs in exact antiphase to the disturbing oscillation.

(3) **Band Pass Filtering.** Aircraft manoeuvres initiated by the pilot will also be detected by the rate gyro, and would therefore be opposed by the auto stabilizer. This occurrence is prevented by the use of band pass filters, which detect the oscillation frequency, and, with preset values to suit the axis plane, can differentiate between pilot input and other disturbances.

(4) **Limiting.** A limiter circuit will ensure that certain parameter changes are kept within prescribed limits.

(5) **Shaping or Scheduling**. A shaping circuit will adapt the system response to suit the handling qualities or flight path of the aircraft.

c. **Servo Actuator.** The correction signal is fed from the signal processor to the servo actuator, to move the control surface.

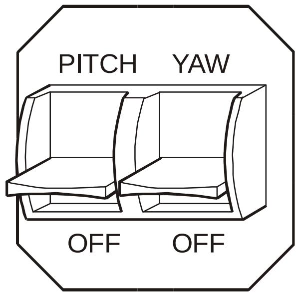
(1) The error signal moves the control surface, without moving the pilot’s controls.
(2) The actuator reverts to a rigid link when not operative.
(3) The authority of an actuator is normally limited (usually 10 to 15% of the total movement available) as a safety precaution in the event of a failure with subsequent active runaway.

6. Auto stabilizers may be able to augment control in all three axes, but they do not usually include the facility to implement changes in attitude. Single or dual-axis auto stabilizers are installed in most aircraft which have insufficient natural stability. In VSTOL aircraft, they counter the problems of maintaining stable flight at low forward speeds. In helicopters, they compensate for the marked changes in dynamic stability that occur at different airspeeds, and counter the low values of longitudinal stability and manoeuvre stability.

7. **Yaw Autostabilizers**. Yaw auto stabilizers are required in most jet aircraft to suppress the lightly damped, short period motion and oscillatory rolling motion, known as Dutch Roll. A yaw auto stabilizer is essential to produce the steady air platform necessary for weapon aiming.

8. **Pilot Over-ride.** The pilot may select or disengage the auto stabilizer channels by means of a control panel (Fig 2). In the event of auto stabilizer failure, an override button is normally located on the pilot's control column to enable rapid disconnection of all engaged channels.

4-7 Fig 2 Typical Auto stabilizer Control Panel

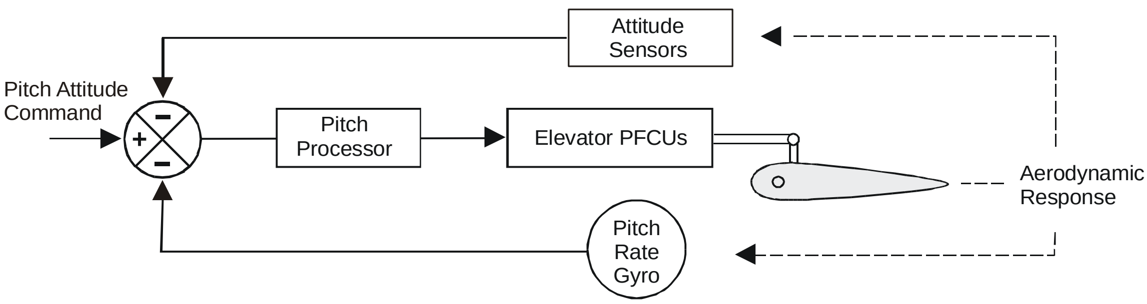


Basic Autopilot Systems

9. A basic autopilot system will hold the aircraft on a flight path selected by the pilot. When the autopilot is selected, it will initially hold the aircraft attitude at that moment. This function is carried out by an 'attitude store', which is a memory unit within the processor. When the attitude hold is engaged, the input to the memory unit is disconnected so that the recorded attitude becomes a fixed datum, against which the actual flight attitude can be measured.

10. **Principles of Operation.** The autopilot will normally work in all three axes, with attitude hold loops for pitch, roll and yaw. A typical pitch loop is represented in Fig 3.

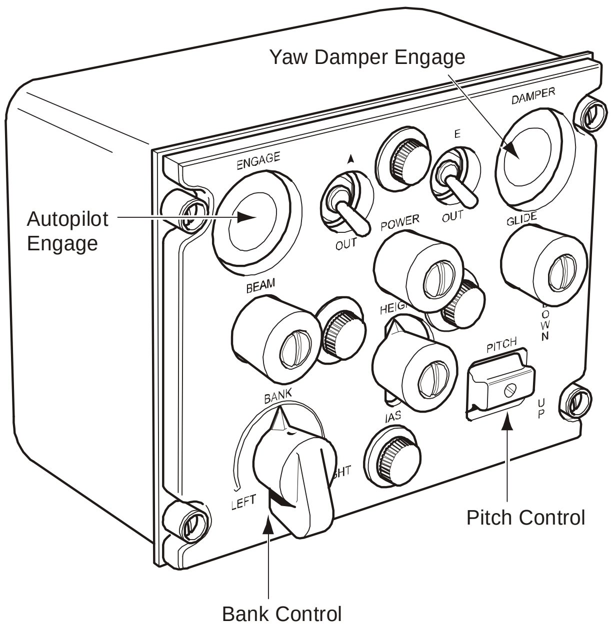
4-7 Fig 3 Autopilot Loop (Pitch Axis only)



The autopilot will detect disturbances to the aircraft flight path by means of three rate gyros, one for each axis. Other sensors may also be used; these would include attitude gyros, lateral accelerometers, and some form of heading reference such as a gyro-magnetic compass. Amplified signals from the gyros and sensors will be summated with the datum attitude, and fed back to the processor, which will then send correcting signals to the servos driving the control surfaces. In the yaw axis loop, a cross-feed system is incorporated which enables correction signals to be fed to both aileron and rudder circuits; corrections to heading are thus made with both of these circuits.

11. **Manoeuvring the Aircraft**. Whilst the autopilot is engaged, the pilot may enter attitude demands manually, by means of switches or knobs located on the autopilot control panel. These controls will produce electrical signals which are fed directly to the autopilot as pitch, roll and yaw demands. Fig 4 shows a typical autopilot control unit. The unit illustrated also controls the yaw damper circuit, and can be linked to signals from ILS azimuth and glideslope beams.

4-7 Fig 4 Autopilot Control Panel



12. **Automatic Control Facilities**. The outputs of other aircraft systems can be fed into the autopilot manoeuvring facility by selection on the control panel. Typically, signals may be derived from:

a. Heading or track demand, set by moving an index marker on the horizontal situation indicator.
b. Radials derived from TACAN or VOR.
c. ILS glidepath and localizer signals.
d. Datum speed or barometric altitude from air data systems.
e. Steer signals from navigation computers.

13. **Advanced Autopilot Systems**. The autopilot systems described thus far are simple, largely self-contained and inexpensive. They therefore provide an extremely cost-effective method of reducing pilot workload by the augmentation of control during stable periods of flight. By introducing steering commands to the autopilot, from external avionics systems, the design becomes progressively more complex, and generally requires computer processing. Safety features and system integrity become of paramount importance as more active control is assumed. An autopilot that is fully integrated with the aircraft's avionics is usually referred to as an Automatic Flight Control System (AFCS). However, the precise point of division between an autopilot and an automatic flight control system is difficult to

quantify. For this reason, advanced autopilot design is covered in the following section, under the generic heading of automatic flight control systems.

AUTOMATIC FLIGHT CONTROL SYSTEMS

Principles of AFCS Operation

14. **Basic Mode of Operation**. The automatic stabilization of an aircraft in roll, pitch and yaw is a basic function of an AFCS. All AFCSs carry out this task by following the same basic principles:

a. Compare the actual response of the aircraft with that demanded by the pilot.

b. Process any error between actual and required performance, in order to generate a correcting control command.

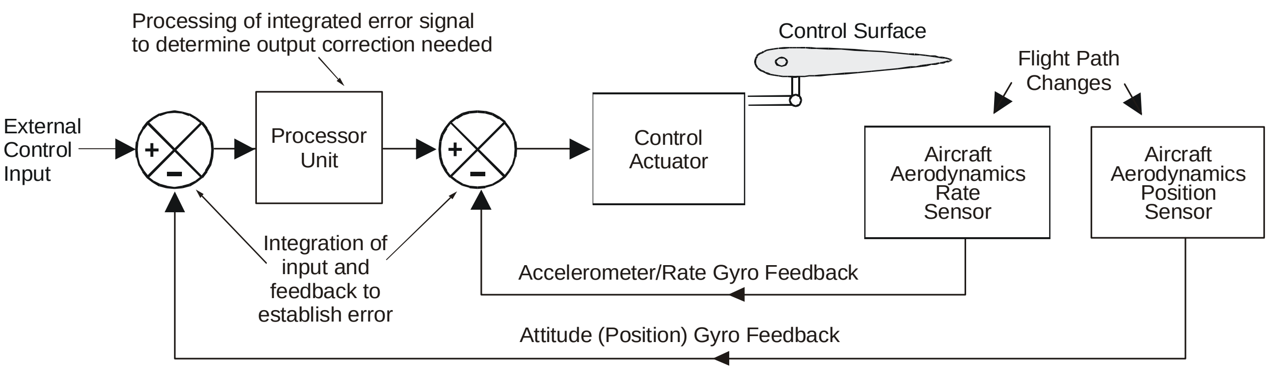
c. Communicate the correcting control command to the relevant aircraft control components.

d. Implement the corrections by moving the relevant control surfaces.

e. Monitor compliance with the original command by feeding back the actual effect of the control input to comparator circuits.

The basic mode of operation (in one axis only) is shown in Fig 5, with appropriate input and feedback loops.

4-7 Fig 5 Basic Flight Control System Operation



15. **System Integrity.** Whether a system is a fully integrated AFCS, or a simple part-system (ie autopilot or auto stabilizer), it must:

a. Be reliable.

b. Be accurate.

c. Provide a stable output.

d. Offer a fail-safe solution.

AFCS Components

16. Although the precise configuration of an AFCS will vary with aircraft type, each will utilize the same basic components, as illustrated in Fig 5.

17. **External Control Input**. The external control input to an AFCS will originate from three sources:

a. The initial flight profile demanded by the pilot.

b. Changes to attitude, course and altitude needed for operational or air traffic reasons. These are interpreted and fed into the system by the pilot.

c. Basic navigational information fed directly into the AFCS from ILS/MLS, VOR, GPS, INS and Flight Director systems.

18. **Sensors**. To evaluate the difference between the performance demanded and that achieved, the AFCS processing unit requires datum information for all relevant parameters. This data may be obtained from:

a. Sensors provided specifically for this purpose, or, more usually, outputs from sensors forming part of other discrete systems.

b. Standard model parameter profiles, usually stored within the AFCS processor, against which the flight conditions may be compared. These would include the performance data for optimum flight profiles.

19. **Processor Unit.** The processor unit performs the basic judgemental process which would be provided by the pilot in manual systems. Its functions include:

a. Manipulating sensor information into useable and comparable signals.

b. Comparing rate and positional sensors and feedback inputs by using differentiation and integration computing techniques.

c. Establishing what degree of error exists between parameters demanded and achieved.

d. Calculating the amount of control response needed to correct any error. Any solution would remain within defined limits and suit the handling qualities or scheduled flight path of the aircraft.

e. Initiating control response by signalling movement commands to the appropriate control surfaces.

The relatively simple processing needed for the operation of part-systems can be provided by mechanical levers and linkages, or by simple electrical bridge balance networks. However, a full AFCS requires the more powerful and versatile electronic processing capabilities of microchip devices.

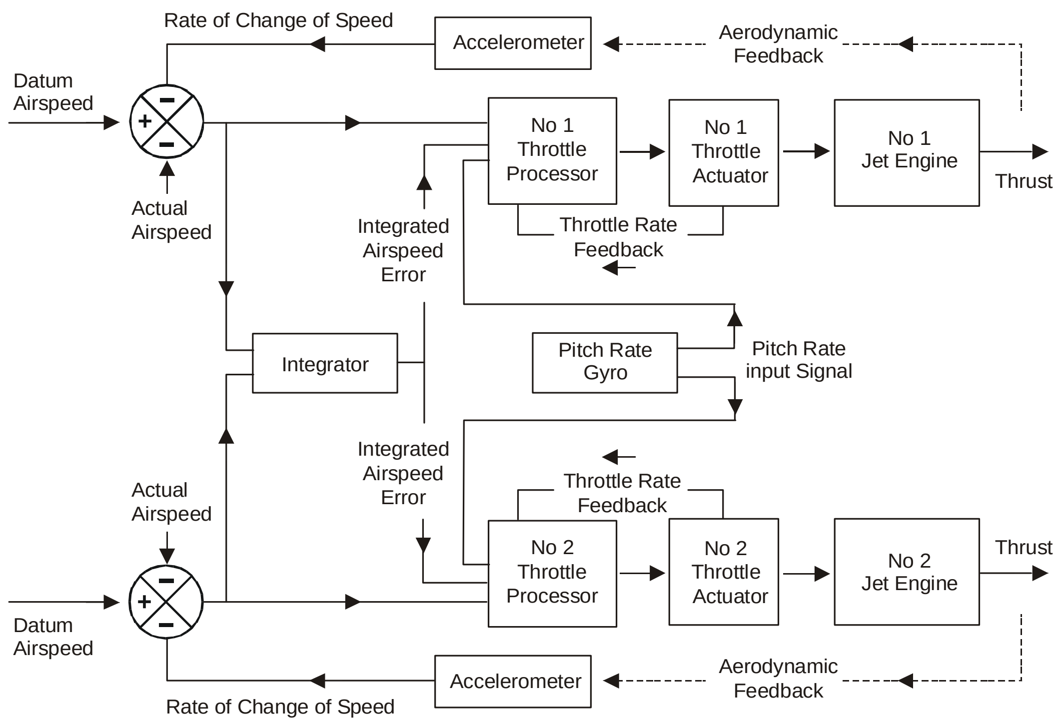
20. **Actuators**. AFCS actuators are powered flying controls, and were dealt with in Volume 4, Chapter 4. The need for actuators to respond rapidly and accurately to signal inputs has resulted in the elimination of all types other than those powered by hydraulics or electrics.

AFCS Functions

21. **Altitude and Heading Control.** Each AFCS is based on a sophisticated autopilot system, which will be used to provide altitude and directional command and control of the aircraft.

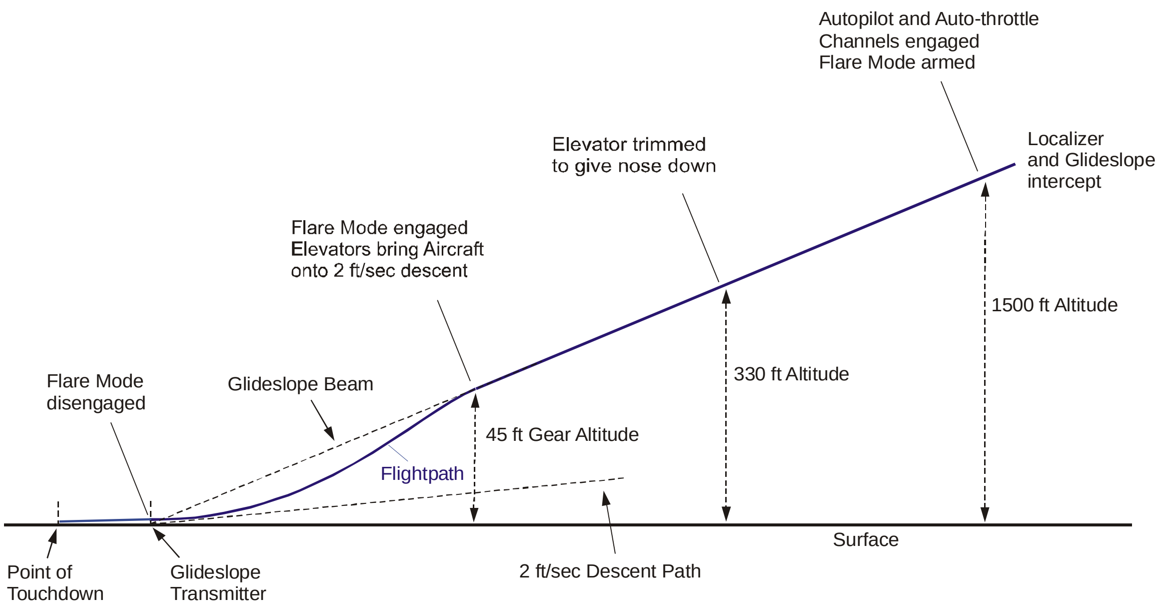
22. **Automatic Throttle Control**. Many of the tasks related to AFCS control of the aircraft's flight path and profile require associated throttle adjustment. The auto-throttle facility provides such control. It can be employed for cruise conditions, such as maximum range or endurance, and also to provide for automatic landings. Fig 6 shows an auto-throttle system in schematic form. The system monitors airspeed and pitch rate against datum parameters set either by the pilot or as a product of an associated auto-land system. Any airspeed error will be resolved by a closed loop control system. By this means, the error signal is processed and controls the throttle actuators, thereby increasing or decreasing the thrust.

4-7 Fig 6 Auto-throttle Control System



23. **Automatic Landing.** An AFCS with auto-land facility will process the signals received from external ILS or MLS facilities. Following the closed loop principle, similar to the one depicted in Fig 5, the auto-land system compares the actual aircraft landing profile, detected from on-board sensors and ILS/MLS signals, with a programmed profile. It then makes appropriate corrections in attitude, direction and engine power settings. Fig 7 shows the profile of a typical automatic landing, and includes reference to the associated ILS signals used.

4-7 Fig 7 Typical Auto-land Sequence



24. **Automatic Compliance with a Defined Flight Profile**. Micro-processors and associated memory storage devices provide the capability for an AFCS to be programmed with details of required flight profiles. By integrating the sub-routines of auto-land, auto-cruise, autopilot and autostabilization as necessary, fully automatic flight control can be achieved. With much of the routine workload of mission profiles now automated, the overall crew workload is reduced, permitting additional time to be allocated to the more important, non-routine work of combat or transport missions. Similarly, the facility for auto-hover in SAR and ASW helicopters greatly increases mission effectiveness. The integration of the sub-routines of auto-control with an easily managed, user interface is developed still further within the concept of a Flight Management System (see para 36).

Influence of AFCS on Aircraft Design

25. The description of AFCS functions assumes that such systems are fitted to conventional aircraft in order to improve handling or operational effectiveness. However, the full integration of AFCS technology into purpose-designed aircraft enables many of the design compromises previously necessary in aircraft performance to be avoided. Thus, use of an AFCS allows the building and operation of much higher performance aircraft, in which the AFCS performs the core function of aircraft control, albeit at the direction of the crew.

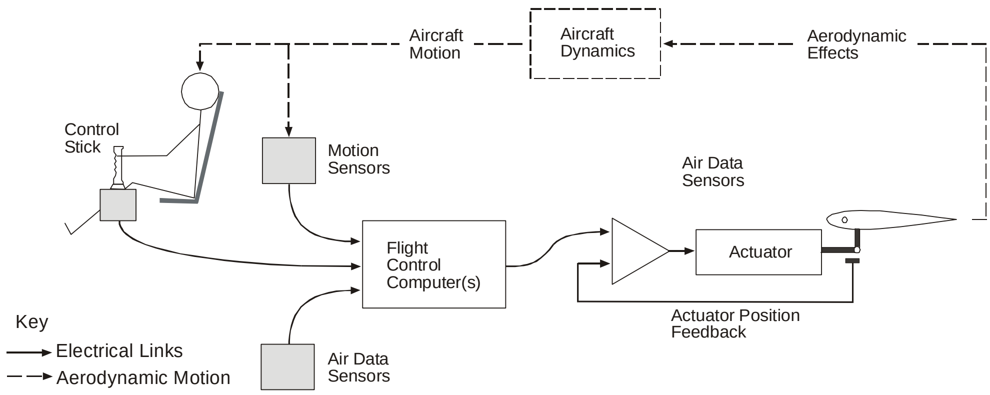
26. The size, and hence the structural weight, of the control surfaces fitted to conventional, inherently stable, aircraft is dictated by the need to achieve manoeuvrability. Inherent stability in an aircraft results in a balance between lift forces and aircraft weight such that tail plane forces act downwards. This reduction of lift requires the wing to be larger, or at a greater angle of attack, which leads to reduced aerodynamic performance. The use of Fly-by-wire and Fly-by-light systems (see para 27), and Active Control Technology (ACT) (see para 44) enables the size of the tail plane balancing force to be reduced by allowing the aircraft's centre of gravity and centre of lift to be placed closer together. Sensors and

computer processors then balance the moments generated by the wing lift and tail lift, to provide the pilot an artificially stable aircraft with excellent manoeuvrability.

Fly-by-wire and Fly-by-light Systems

27. **Fly-by-wire Systems.** The term 'fly-by-wire' (FBW) was first coined to describe the control of an aircraft, by the pilot, through electrical signals generated by movements of the pilot's controls, and transmitted along twisted-pair cable or coaxial cable. Such systems were initially introduced purely to obtain the advantages of electrical signalling over the bulk and mechanical complexity of control rods and linkages. FBW also permits duplication of signalling paths without incurring significant weight and space penalties. The introduction of FBW also resulted in easier integration of pilot control inputs with other autopilot functions; this has been a major contributor to the development of the fully integrated AFCS. The term FBW is now used to denote systems in which electrical signals generated by pilot control inputs are integrated with sensor signals within the flight control computer, before being fed to the control surfaces (see Fig 8).

4-7 Fig 8 Fly-by-wire Flight Control System



28. **Fly-by-light Systems.** Fly-by-light (FBL) systems operate in the same manner as FBW systems, but the electrical signals are transmitted via fibre optic cable. Fibre optic cable is superior to coaxial or twisted pair cable, in that it is lighter in weight and easier to maintain. Also, the optical power source has a low power requirement. Fibre optic cable has advantages associated with electromagnetic interference (EMI), and provides greater immunity to:

a. Lightning strikes.

b. Failures caused by flying close to sources of high intensity radiation transmissions.

c. Failures in the aircraft's electromagnetic screening system.

d. Electromagnetic emissions from nuclear explosions.

As far as flight control is concerned, the use of FBW and FBL is the same, so the term FBW will be used as a generic term for both, in this chapter.

29. **Optimizing Flight Performance**. FBW makes it relatively easy for computers to modify the signals that are fed to the control surfaces. No direct link remains between pilot and control surfaces. The severing of such a direct link allows the AFCS to optimize aircraft performance in all flight conditions. FBW provides a basis for ACT (see para 44) to be readily incorporated into aircraft such as Typhoon. The use of FBW enables computers to monitor the flight regime for divergence, leaving the pilot free to concentrate on the mission in hand.

30. **FBW Control Loop.** Fig 8 illustrates the essential features of a FBW flight control loop. Electrical links have replaced the mechanical links of a conventional flight control system. In early FBW aircraft, the flight control computer was analogue, but now digital systems are used, enabling complex control law algorithms to be implemented.

31. **Benefits of FBW Control**. The major benefit of FBW is the ability to tailor the system's characteristics at each point in the aircraft's flight envelope. The performance benefits from FBW are often quoted in terms of manoeuvrability, but it is often the avoidance of a manoeuvre, which would stall or over-stress the aircraft, that is a more important benefit. The other benefits of FBW include:

a. Carefree handling, provided by automatic protection against stall and departure (using angle-of-attack control and angle-of-sideslip suppression). In addition, overstressing of the airframe is prevented by automatic limiting of normal acceleration and roll rates.

b. Handling qualities which are optimized across the flight envelope, providing for a wide range of aircraft stores, asymmetric configurations and in-flight changes, such as those encountered when ordnance is released.

c. Improved agility for fighter aircraft. Aircraft configurations with negative stability assist rapid changes in fuselage aiming and/or velocity vector. This greatly enhances offensive and defensive manoeuvrability.

d. Improved aircraft performance, due to increased lift/drag ratio. FBW is lighter than mechanical linkages, and also permits the use of a smaller tail plane, fin and rudder. Drag is also reduced due to the optimized trim setting of controls.

e. An extended flight envelope, provided by the use of thrust vectoring to augment or replace aerodynamic control surfaces.

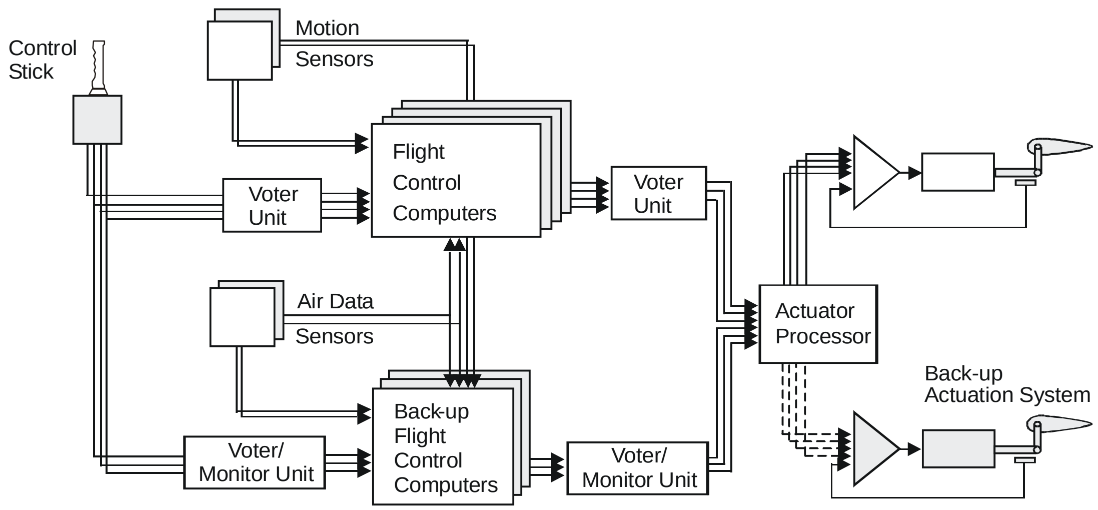
f. The ability to reconfigure systems easily following failures or battle damage. This enables missions to be completed, or safe recoveries made.

g. Reduced maintenance costs, resulting from a reduction in mechanical complexity and the introduction of built-in test facilities.

32. **FBW Integrity.** The overall system integrity of FBW must be as high as the mechanical control system it replaces. The probability of a catastrophic failure must not exceed 10^{-9}/hour for civil aircraft and 10^{-7}/hour in military aircraft. In order to achieve this reliability, multiple signal sources and several lanes of computing are necessary to provide redundancy. A system of cross-monitoring is included in order to isolate any failed equipment, thereby ensuring safe operation. The current trend is towards mixed triplex and quadruplex redundancy, as illustrated in Fig 9. A comprehensive built-in test capability is used to identify and locate

failures, and to ensure that the system is safe prior to each flight. The back-up systems in FBW usually provide limited flight control capability, although the trend is towards a full capability.

4-7 Fig 9 Typical Redundancy in a FBW Control System

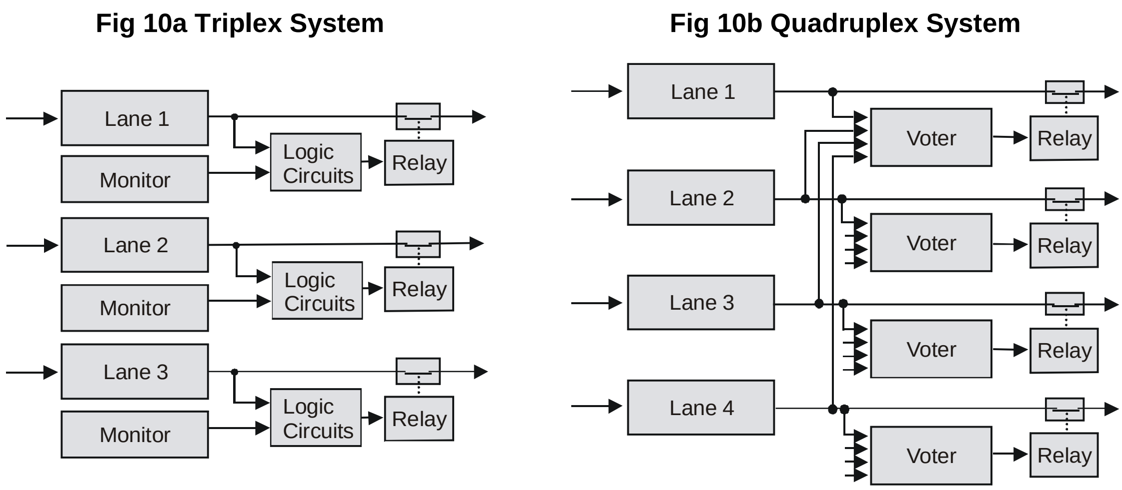


33. **Use of Dissimilar Systems.** In many early FBW systems, the back-up computer was analogue but digital computers are now employed. Dissimilar hardware is used to avoid failures being repeated in duplicated systems; this may occur through common design errors. System integrity requires back-up digital computers to be procured through alternative requirement documents, using different software, independent programming teams, and utilizing dissimilar operating systems. Fig 9 illustrates a back-up control surface actuation system. This is more common in civil airliners but has been partially used on military jets. Early FBW aircraft, such as the Tornado, reverted to mechanical flight control but only with a 'get you home' capability. FBW is being developed for helicopter use, but the development of full authority flight control is slow because of the complex nature of helicopter flight controls. The only operational FBW in a military helicopter is a simplified flight control system, used as a back-up in event of failure of the mechanical system.

34. **Voting and Monitoring**. Examples of triplex and quadruplex systems employed in a FBW flight control system are shown in Fig 10. Provided the monitoring is to a high degree of integrity and confidence level, such systems provide sufficient redundancy to survive any two failures, from whatever cause.

a. **Monitored Triplex Redundancy**. Fig 10a illustrates a monitored triplex redundancy system, consisting of three independent and parallel channels. Each channel incorporates its own monitoring system, to check the channel's functioning to a high confidence level. If any channel fails, its associated monitoring system will identify the failure, and isolate the output by means of relays. The system illustrated could survive two lane failures, leaving the third to run the service.

4-7 Fig 10 Triplex and Quadruplex Voter/Monitor Systems



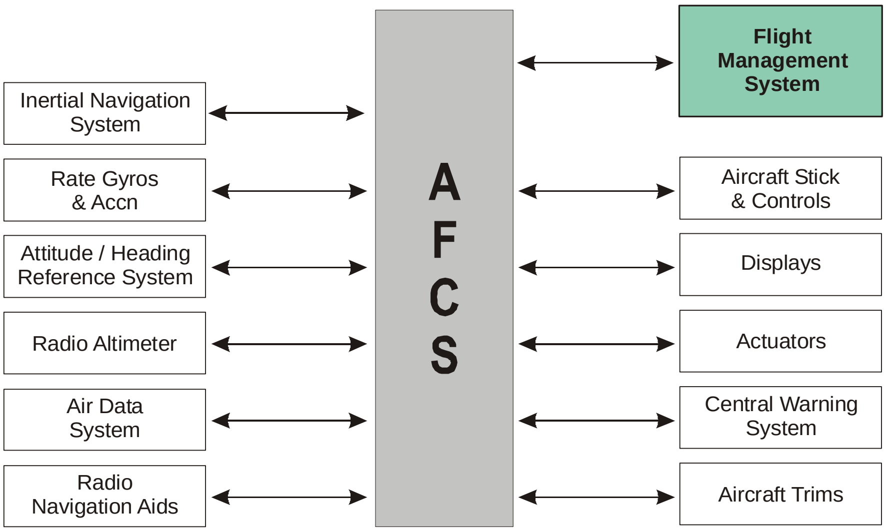
b. **Quadruplex Monitoring**. Fig 10b shows a quadruplex system, which detects failures by cross-comparison of the parallel channels, and uses majority voting to determine the 'odd man out'. Once a failure is identified, that channel is disconnected. This system is able to survive two failures on majority voting. Depending upon the nature of a third channel failure, the system may be able to survive on a single channel.

FLIGHT MANAGEMENT SYSTEMS

Introduction

35. Computerized systems, known as Navigation Management Systems (NMS), were introduced in the early 1980s, to simplify the navigation task and to optimize the use of navigation aids. With the introduction of computer processors in aircraft, along with advanced AFCS and FBW technology, it has been possible to integrate the outputs from multiple aircraft systems, and to correlate aircraft flight conditions with a database containing performance values. By these means, achievements have been made towards attaining the theoretical maximum performance from an aircraft. A Flight Management System (FMS) is a computer-controlled AFCS, which allows the pilot to select specific modes of operation. These might include standard instrument departures and auto-landings. In large aircraft, the FMS has become one of the key avionics systems because of the reductions it can make to the pilot's workload. In military aircraft, the FMS has enabled single-crew operation of advanced combat aircraft. Fig 11 illustrates, in schematic format, the relationship between an FMS and the AFCS.

4-7 Fig 11 Relationship between an FMS and the AFCS



36. The FMS combines navigational and performance data with flight-derived data to determine an automatic flight profile that is normally optimized for specific operational parameters, such as maximum endurance or minimum fuel use. In its most comprehensive form, the FMS is directed to the auto-throttle system to optimize the power controls. An FMS can lead the pilot through the complete profile of the flight: take off, climb out, cruise climb, initial level off, step climb, cruise, top-of-descent, descent, approach and landing.

Flight Planning

37. **Flight Planning Database.** The FMS plays a major role in the flight planning task. It will hold a readily available database of air traffic significant points, runway information and navigation beacons. The navigation database is updated in accordance with the AIRAC dates (see Volume 9, Chapter 13). A typical FMS flight plan may contain up to 100 waypoints and the system can store a library of prepared flight plans for future use. As the FMS is essentially just a computer, there are many types and variations. The applications and the extent of the integration and automation vary greatly, as do other criteria such as accuracy.

38. **Navigation Aids.** The FMS will select and tune navigation aids in accordance with the planned profile. It will then determine the best estimate of aircraft position from all the navigation sources, through a Kalman filter process. Navigation sources will include:

a. Multiple INS.
b. GPS.
c. Air Data.
d. Radio navigation aids such as VOR and DME.
e. ILS/MLS.

The FMS will compute and display groundspeed, track and wind velocity.

39. **Flight Profile.** The FMS will provide both lateral and vertical guidance signals to the AFCS. In the lateral mode, the FMS computes the aircraft's position relative to the planned route, and gives guidance signals to the AFCS to capture and follow the track specified by the flight plan.

Optimized Flight Performance

40. The FMS will continually monitor the aircraft flight envelope. It can ensure that speed restrictions are not exceeded. It can also compute the optimum speed and altitude for each phase of the flight profile. To do this, the FMS will monitor:

a. Aircraft mass.
b. The position of the centre of gravity.
c. Constraints imposed by the route flight plan and air traffic regulations.
d. Wind velocity and outside air temperature.

It can therefore compute the recommended cruise altitude, and maximum altitude possible.

FMS Operations

41. The FMS can be programmed for a multiplicity of operational modes, to suit all different stages of the flight. Examples are:

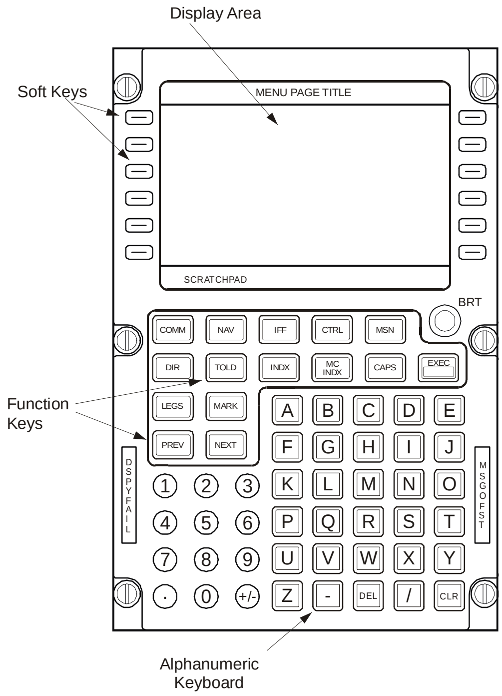
a. **Standard Instrument Departures**. The FMS can fly the aircraft along a complicated Standard Instrument Departure, controlling engine power, altitude restrictions and route navigational aspects.

b. **Holding Patterns**. The FMS can fly the aircraft through a precise holding pattern, based on a selected datum, using ICAO procedures.

c. **Time Control.** The FMS can calculate ETAs, or, if required, produce aircraft performance to meet a specified arrival time.

42. **Crew/FMS Interface.** The crew must retain ultimate control of the aircraft at all times. They will therefore operate the FMS by means of multi-function control and display units. Fig 12 shows a typical pilot/FMS interface unit.

4-7 Fig 12 A Typical Pilot/FMS Interface



Multiple units are provided to allow for redundancy. However, by necessity, the units are normally positioned in a 'head-down' location. The display area may, therefore, be reproduced on one of the larger multi-function screens in front of the pilots.

ACTIVE CONTROL TECHNOLOGY (ACT)

Introduction

43. **Passive Technology.** Conventional design of airframes has brought aircraft technology to its present high level, but in many cases has reached its limits. Aircraft lifting surfaces have been designed to be strong enough to meet loading requirements, with material added to provide stiffness adequate to keep them free from flutter, divergence and buckling. However, this added stiffness usually means adding structural weight. For a given set of aerodynamic requirements, aircraft design has therefore been a compromise between weight and aerodynamic performance. This legacy of normally stable aircraft with conventional control surfaces is sometimes referred to as 'passive control'.

44. **Active Control Systems**. The trend in aircraft development is towards high manoeuvrability, lower specific fuel consumption, higher power-to-weight ratios and lower life-cycle cost. An active control system (ACS) may be likened to an AFCS, but designed to provide several special features including:

a. Activation of flight control surfaces to minimize gust loads and bending stresses in the wing. This is done by detection and response to normal accelerations.

b. Provision of stability to a naturally unstable aircraft.

c. Implementation of pilot manoeuvre demands by more active means than conventional control surfaces.

An ACS requires extensive integration between aerodynamics, structure and electronic system design to achieve these advantages with reliability and safety.

Employment of ACT

45. The employment of ACT has become one of the most important aspects of aircraft design and operation, and, in some cases, there is potential for retrofit. ACT is linked to the development of computer technology, sensors and actuators, micro electro-mechanical technology, smart materials and improved knowledge of process laws. Specific uses for ACT include:

a. ***Gas Turbine Engines.*** Active control can be used within gas turbine engines, in areas of compression, combustion and airflow. ACT can produce higher pressure ratios, which in turn leads to smaller engines. Better performance and advanced diagnostics can lead to reduction on overall life cycle costs and savings in maintenance.

b. **Fluid Aerodynamics**. Boundary layer control and vortex flow can be influenced, and used to control flight attitude, thus avoiding the use of control surfaces, which are heavy and energy consuming.

c. ***Vertical Take-off and Landing Aircraft (VTOL).*** ACT has proven to be advantageous in controlling VTOL aircraft, particularly when manoeuvring in the hover. The development of the Joint Strike Fighter utilizes this concept, whereas, by contrast, the Harrier still requires much of the hover control to be a manual input from the pilot.

d. **Helicopters**. The introduction of FBW and ACT to helicopter design would revise control-system architecture through revised crew/machine interface and pilot-assistance systems. Such developments offer potential 'carefree handling' qualities, and may introduce new rotorcraft configurations.

e. ***Active Aeroelastic Wings.*** There is high potential for use of ACT in structural applications. The concept of the active aeroelastic wing (AAW) makes use of multiple leading edge and trailing edge control surfaces, each activated by a digital flight control system to reshape the wing cross-section. This reshaping of the wing (sometimes referred to as 'wing twisting') provides roll manoeuvre, in place of conventional ailerons. Fig 13 demonstrates the AAW principle, based on a port wing, with right roll demand input.

4-7 Fig 13 Comparison of AAW and Conventional Controls

Fig 13a Conventional Aileron Controls

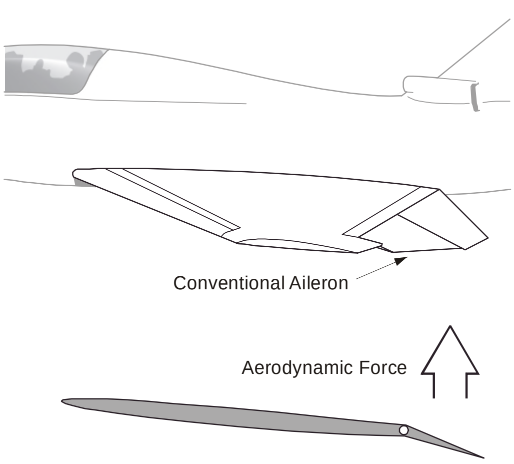
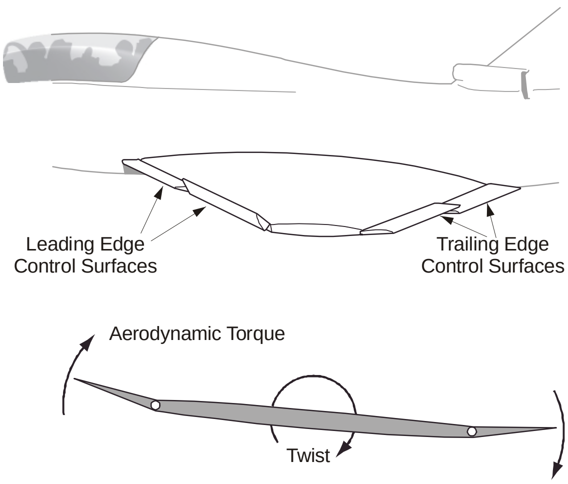


Fig 13b Active Aeroelastic Configuration



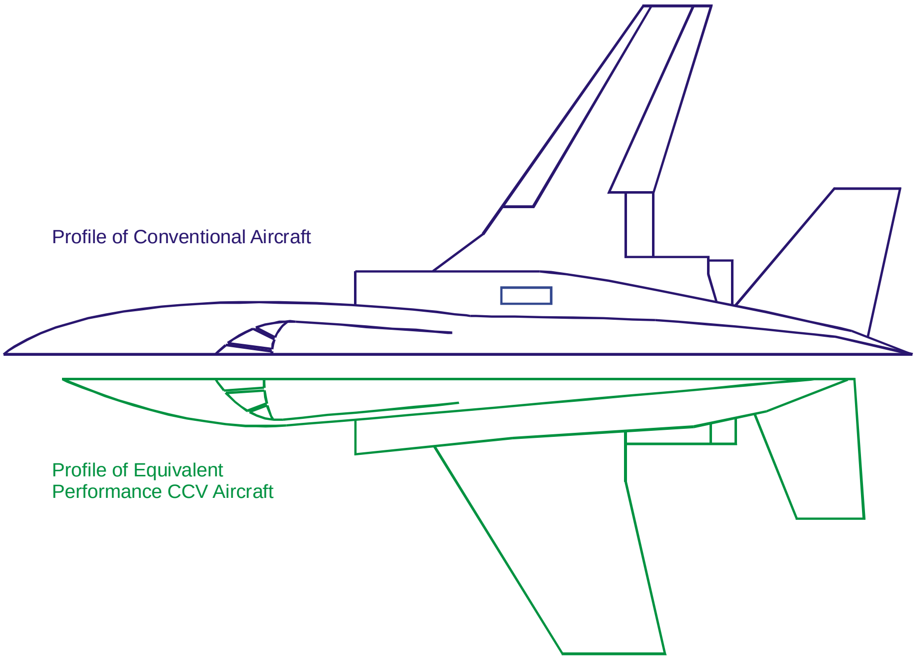
46. **Control Configured Vehicles.** By using FBW control systems and ACT to stabilize the aircraft, it is now possible to reduce the need for conventional aerodynamic stability. The centre of gravity can now be positioned well aft of the centre of pressure. In such instance, reversion to manual control would be difficult, or impossible. However, the advantages gained by reducing longitudinal stability are:

a. The aircraft becomes highly manoeuvrable.

b. Because the tail plane can now contribute positive lift, the weight and size of the airframe can be reduced.

Aircraft designed this way are termed Control Configured Vehicles (CCVs). Fig 14 compares the relative plan view of a conventional aircraft with that of a CCV. By using a canard-delta configuration, with ACT, positive lift is generated by all flying surfaces, and the weight and size of a fighter aircraft can be reduced.

4-7 Fig 14 Comparison of Conventional and Control Configured Aircraft



47. **ACT and Safety.** The fast response to un-demanded flight path divergence, inherent in a CCV, will improve the ride quality in turbulent conditions. Trim alteration, due to configuration changes such as weapon release, can be eliminated by suitable inputs to the pitch computer. In addition, the control system can be programmed to provide manoeuvre envelope protection. However, once the step of utilizing ACT to produce CCVs has been taken, the aircraft's control system must be designed such that there is minimal likelihood of failure, since the pilot may well be unable to control the aircraft without the assistance of AFCS computers.

CHAPTER 8 - FIRE WARNING AND EXTINGUISHER SYSTEMS

CHAPTER 8 - FIRE WARNING AND EXTINGUISHER SYSTEMS

Introduction

1. The design philosophy used in aircraft is first to prevent fire and secondly to provide adequate fire protection. Protection is usually in the form of fire resistant materials used in the construction of strategic systems and structures and fire retardant materials used in aircraft furnishings. However, in those areas where a risk of fire remains, active aircraft fire protection systems are utilized. These perform two basic, and usually independent, functions:

a. Fire and overheat detection.
b. Fire extinguishing.

The fire detection and overheat systems sense the presence of fire or excessive heat. They employ area detectors in large fire zones and spot detectors for individual pieces of equipment. In freight and passenger aircraft, they are often supplemented by smoke detectors positioned in the freight bays, baggage holds, and toilet compartments. In the case of fire, overheat or smoke, the systems provide a visual and aural warning to the crew, identifying the area in which the problem exists. The fire extinguisher system provides a capability for fighting airborne fires in specific major areas, typically the engines and auxiliary power unit (APU). Fire extinguisher systems invariably require crew intervention for their operation in the air. However, in the event of a crash or crash landing, they are activated automatically by switches which close under high retardation forces or through airframe deformation. Aircraft are also equipped with hand-held extinguishers for use against small fires in internal areas and equipment.

2. The most common causes of aircraft fires are:

a. Fuel leaks in the vicinity of hot equipment.
b. Hot gas leaks, from engines or ducting, impinging on inflammable materials.
c. Electrical or mechanical malfunctions in equipment.

The initiating cause is usually equipment failure, although, obviously, damage incurred during combat or a crash landing would provide ample additional cause. It follows, therefore, that the areas in which fire protection systems are deployed should include the engine bays, the APU enclosure, and significant pieces of high-energy equipment. Fig 1 provides a summary of the equipment given protection in a medium-size aircraft. The table includes the temperature at which a fire or overheat warning will be given and the visual warning which the crew will receive. Although the carriage of dangerous cargo in aircraft is adequately legislated for, spontaneous fires can occur in freight and baggage holds. Access to such areas whilst airborne may be possible, allowing the crew to enter them and fight the fires with portable extinguishers.

4-8 Fig 1 Equipment Monitored by a Fire Detection System

Unit/Area Covered	Type of Detector	Indication
Engine fire detection	Graviner continuous wire	L or R Fire lights
Fire/short test	Graviner continuous wire	L or R ENG FIRE DET fault
Engine (cooling air) overheat	Thermal switches 860 ºF	R and L ENGINE HOT
APU fire detection	Four thermal switches Three at 450 ºF One at 600 ºF	APU FIRE light
Radio rack overheat	Twelve thermal switches 200 ºF	FWD RADIO RACK HOT light
Alternator overheat	Thermal switches 250 ºF	L or R ALT HOT light
APU alternator overheat	Thermal switches 300 ºF	APU ALT HOT light
Emergency transformer rectifier overheat	Thermal switch 375 ºF	TRU HOT light
Tail compartment overheat	Two thermal switches 200 ºF	AFT EQUIP HOT light
Auxiliary hydraulic pump overheat	Thermal switch 300 ºF	AUXILIARY HYD HOT light
Pylon overheat	Thermal switches 325 ºF	L or R PYLON HOT light
Bleed-air duct overheat	Thermal switches 550 ºF	L or R BLEED AIR HOT light
Flight system hydraulic reservoir overheat	Thermal switch 220 ºF	FLT HYD HOT light
Combined system hydraulic reservoir overheat	Thermal switch 220 ºF	CMB HYD HOT light
Wing anti-ice overheat	Thermal switches 180 ºF	L or R WING HOT light
Cowl anti-ice overheat	Thermal switches 675 ºF	L or R COWL A/I OVHT
Bootstrap units	Thermal switches 450 ºF	L or R COOL TURB HOT
Forward radio rack	Thermal switches 200 ºF	FWD RADIO RACK HOT

Fire Detection Systems

3. The functions of fire detection systems are to:

a. Monitor designated areas or equipment for a rise in temperature, either at a higher rate, or to a higher level, than predetermined acceptable limits.

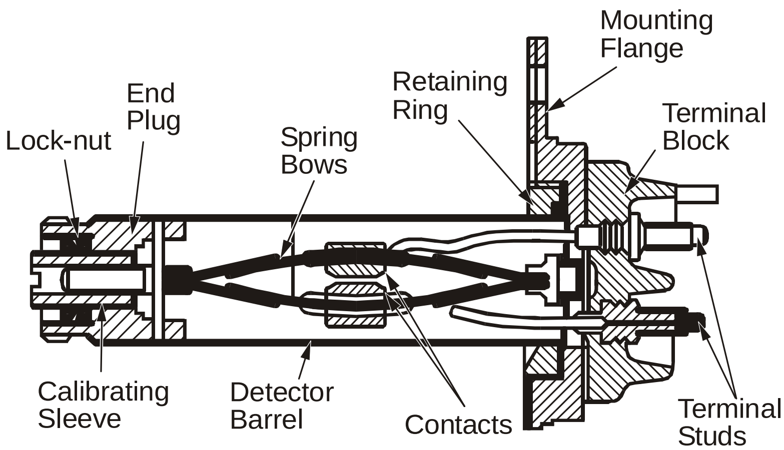
b. Provide a warning to the crew.

c. Complete electrical safety circuits within the fire extinguisher systems, to permit necessary operation by the crew.

The electrical safety circuits are provided to prevent accidental operation of extinguishers, and they are over-ridden either by the detection systems or by deliberate manual selection by the crew. The APU fire detection circuits are usually arranged to automatically close down the equipment as part of their operation. It is important that detection systems reset automatically when conditions return to normal, not only to inform the crew that the problem has receded, but also to be ready to react again if further overheating occurs. Two basic principles of operation are used in detectors, either as simple electrical switches activated by the differential thermal expansion of component metals, or as sensors in which temperature-dependent changes in the electrical resistance or capacitance are used to activate an electronic circuit. Smoke detectors are devices which are sensitive to an increased presence of the chemical products of combustion in the surrounding air. If smoke is detected, an alert is given to the crew, indicating the problem area.

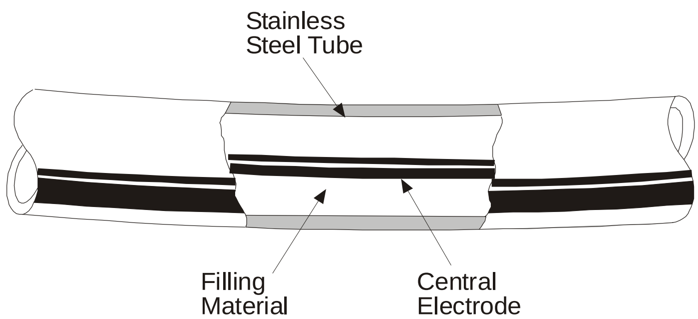
4. Fig 2 shows the construction of a typical heat-sensitive, self-resetting thermal switch. It consists of a stainless steel barrel, which has a high coefficient of thermal expansion, attached to a mounting flange. Inside the barrel is a spring bow assembly, which has a low coefficient of expansion. At normal operating temperatures, the contacts attached to the assembly remain open because the cylinder restricts the bow, forcing its arms apart. As the temperature rises, the cylinder expands more than the bow, thus removing the restraint upon its length, and allowing the contacts to spring together. This process is reversed as the temperature returns to normal. The device can be adjusted to operate precisely at the required temperature.

4-8 Fig 2 Thermal Switch



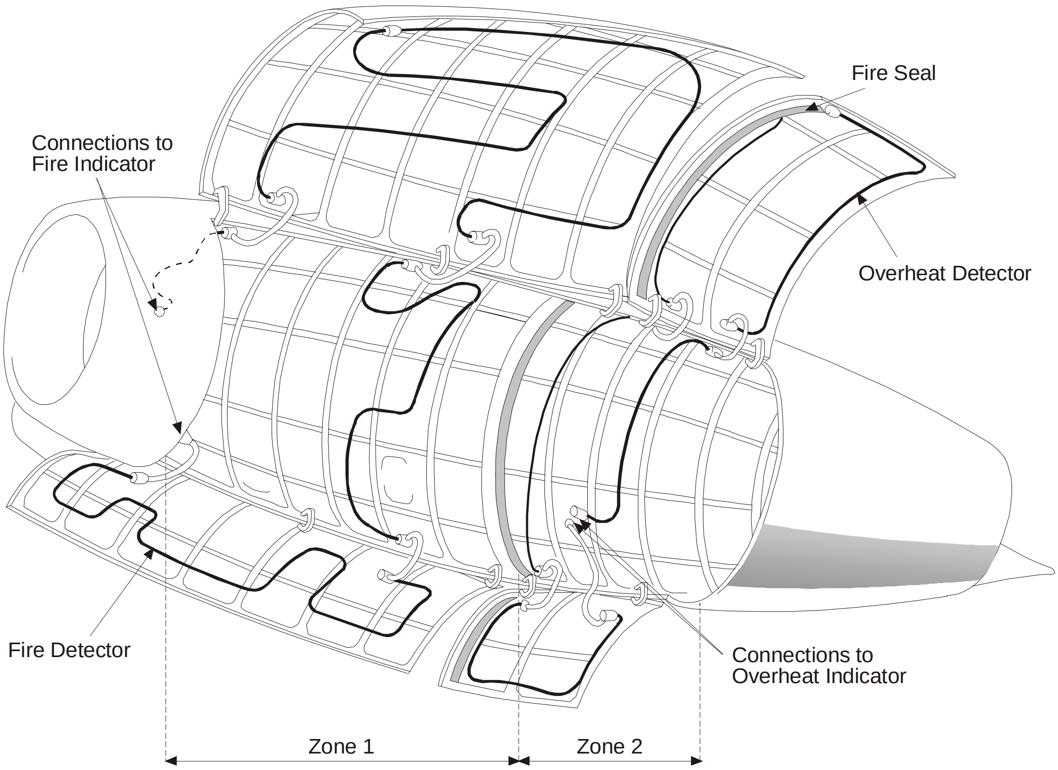
5. Fig 3 shows the sensing element used in the majority of area fire detectors. The sensing element is in the form of a wire about 2 mm in diameter, and is known as the 'continuous wire' detection element (although the term 'Firewire', an early trade name, is still widely used).

4-8 Fig 3 Continuous Wire Detection Element



The element can be relatively easily installed around the areas which require to be monitored. Fig 4 shows such an installation comprising two separate systems. The system in Zone 1 monitors the engine pod for fire, whilst that in Zone 2 monitors the jet pipe area for overheating caused by gas leaks. Similar installations would also be used in an APU enclosure. Such systems are invariably installed in continuous loops, as shown in the figure. The wire is vulnerable to damage caused by vibration, which may result in a reduction in electrical properties or actual fracture. The use of a continuous loop avoids the effect of the resultant open circuit, allowing the wire to operate normally and provide a fire signal even when defective in part. A test device is included in the system to highlight the existence of a fault, and, thus, allow timely rectification to be carried out.

4-8 Fig 4 Installation of a Continuous Element System



6. **Crash Switches**. Crash switches operate either by sensing high retardation forces (typically in excess of 6 g), or by the effect of structural deformation around them. They are usually installed in the undercarriage bays or inside the belly of an aircraft.

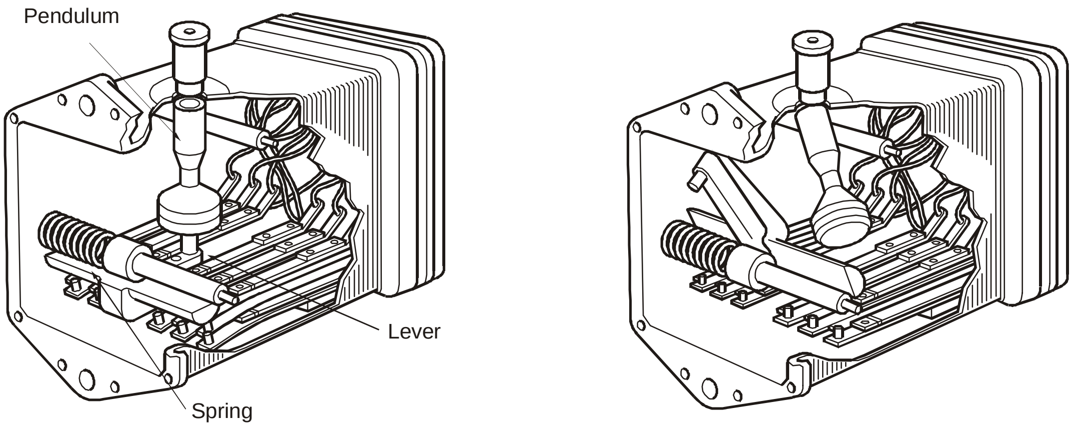
a. **Inertia Switches**. The inertia switch senses excessive 'g' forces, utilizing either electronic or mechanical accelerometers. Fig 5 shows a pendulum inertia switch. It has the advantage of being omni-directional, albeit only in a horizontal plane. The pendulum is suspended on a beam which allows it to swing horizontally in any direction. Normally it is restrained from moving by a spring-loaded lever below it. However, if subjected to excessive horizontal deceleration, the pendulum breaks away from its restraining lever, allowing the lever to rotate and actuate a bank of electrical contacts in the fire extinguisher circuits.

b. **Piston Switches**. The piston type of switch operates on a similar principle. A horizontal piston is restrained in its cylinder by a sprung lever. Under the effect of high horizontal deceleration forces, the piston will overcome its restraint and move along the cylinder, allowing the sprung lever to rotate and make a series of electrical contacts.

c. **Structural Distortion Switches**. Structural distortion switches are positioned inside the belly of the aircraft. They are intended to operate during a crash landing when skin deformation will occur, despite horizontal deceleration forces not being excessive.

4-8 Fig 5 A Pendulum Inertia Switch

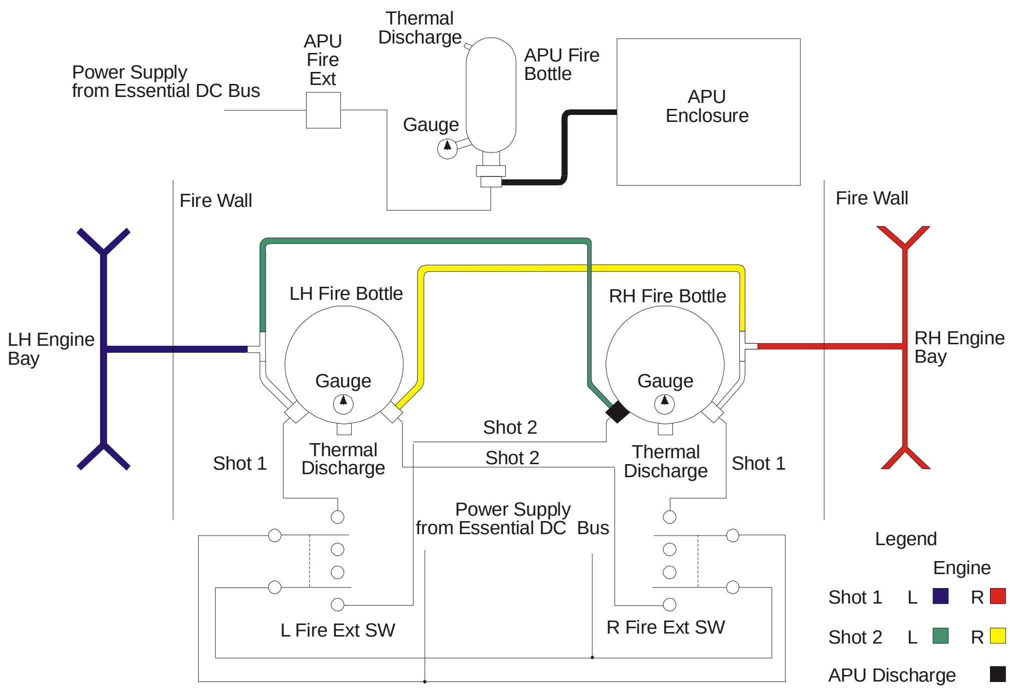
Fig 5a Before Operation **Fig 5b After Operation**



Engine Fire Extinguisher Systems

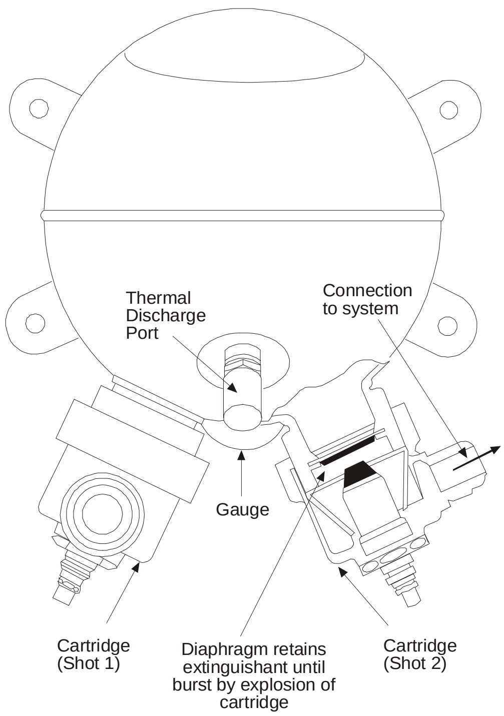
7. Permanently installed fire extinguisher systems are normally provided to suppress fires in the engine nacelles or bays, and the APU and heater enclosures of an aircraft. The fire extinguisher system comprises selection switches, sited with the cockpit engine control levers, which activate the fire bottles adjacent to the engines or APU. Power is provided from the 28 V DC essential bus, to ensure that the systems are always live. The systems are designed to deliver predetermined volumes of extinguishant from the fire bottles to designated areas of the appropriate engine installation. One fire bottle per engine is provided, and the systems of multi-engine aircraft are invariably arranged so that extinguishant from each fire bottle can be fed to one or other engine. Thus a '2-shot' system is provided, allowing the crew two attempts to extinguish a fire. This arrangement is shown in the schematic layout of a typical system at Fig 6. The figure shows the pipe and electrical interconnections necessary to provide the second shot capability. Shot 1 fires the left engine bottle to the left engine or the right engine bottle to the right engine. Shot 2 fires the right engine bottle to the left engine or the left engine bottle to the right engine. The extinguishant used in such systems is likely to be an inert gas or a halocarbon agent. When released from the system, the gas 'blankets' the fire, purging oxygen away from it.

4-8 Fig 6 Typical Engine and APU Systems Diagram



8. The fire bottles comprise metal spheres or cylinders containing the extinguishant, pressurized by nitrogen, typically at 40 bar. At this pressure, the agent is in liquid form. A range of bottle sizes is available to meet different fire threats. A typical engine bottle holds a charge of 2.5 kg, whilst a bottle of l kg would be used for smaller applications such as the APU. Fig 7 shows more detail of the 2-shot bottles used in the above system.

4-8 Fig 7 Detail of a 2-shot Fire Bottle



The bottle has two firing heads, each containing an electrically actuated explosive squib cartridge. When a head is fired, either by crew selection or automatically by operation of the crash switches, the agent is propelled into the relevant pipe gallery. It flows under pressure to the fire area and issues through spray nozzles, vaporizing as it does so. The reduction of pressure, and vaporization of the agent, as it is sprayed from the system nozzles, cools the resultant gas thus enhancing its fire fighting capability by cooling the area of the fire. As a safety device, each bottle is fitted with a safety disc. If excessive pressure builds up in the bottle, the disc ruptures, allowing the agent to vaporize and escape harmlessly overboard. Each bottle has an integral pressure gauge which is read during each flight servicing. This enables any failure or inadvertent discharge of the system to be detected before further flight. Although the explosive squib cartridges have a limited effective life, and need to be replaced routinely, very little maintenance is required by extinguisher systems. Essential servicing includes the annual weighing of the bottles to ensure that they still contain a full charge of agent, and five-yearly pressure testing to validate their integrity.

Portable Fire Extinguishers

9. Portable hand-operated fire extinguishers are fitted in aircraft to combat fires that may occur in crew compartments. For this role, Military aircraft carry a fire extinguisher containing bromochlorodifluoromethane (BCF) extinguishant, pressurized with dry nitrogen gas. BCF is a non-corrosive chemical that, when released, forms a blanketing mist to deprive the fire of oxygen.

a. **Description**. The BCF extinguisher is coloured red, and has a label with the operating instructions on. It is secured in a mounting bracket by a retaining band, which has a quick-release fastener. A sealing cap is fitted over the nozzle to prevent the ingress of dirt (see Fig 8a). As the BCF extinguisher has no safety pin, the mounting bracket incorporates a shaped guard to prevent inadvertent operation of the lever whilst stowed. A modified bracket, with a secondary locking system, is available for high 'g' aircraft.

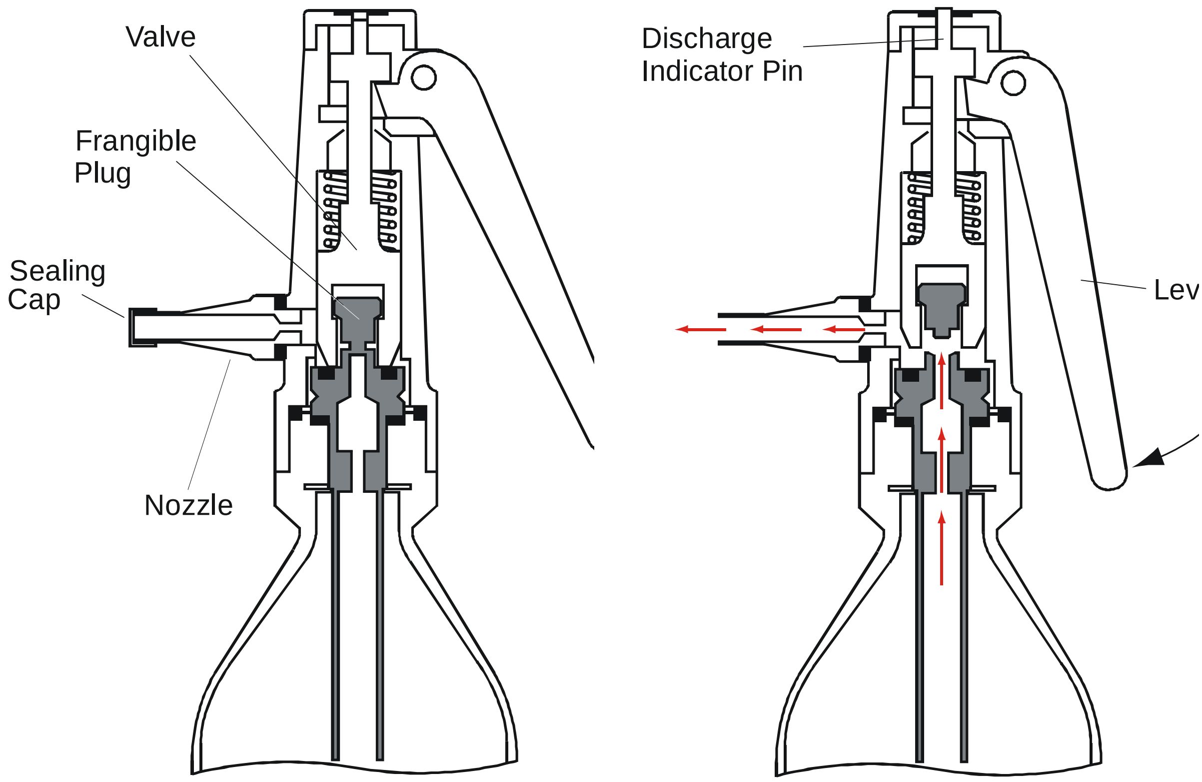
b. **Pre-flight Checks**. Pre-flight, the BCF extinguisher should be checked to confirm that the nozzle cap is present, and that the cap covering the discharge indicator pin is flush with the top of the extinguisher head. If the discharge indicator pin is visible, or its cap distorted, the extinguisher should be treated as unserviceable, as the quantity of the contents cannot be guaranteed. The extinguisher should also be checked for signs of external damage.

c. **Operation**. To operate the extinguisher, the lever must be fully depressed by a sharp action (rather than by gradual squeezing). This action lifts the valve against the compression spring (Fig 8b), thereby breaking the frangible plug and allowing the BCF to flow from the container to the nozzle (forcing off the sealing cap). At the same time, the lever bolt pierces the indicator disc. Releasing the lever allows the spring to push the valve back against a seal and stops the flow of extinguishant. Further operation of the lever allows extinguishant to flow, until the container is empty.

Note: Many aircraft currently in service are equipped with the Kidde Graviner Handheld Fire Extinguisher, NSN 4120-99-1042111. The manufacturer recommends that to ensure optimum fire fighting performance from this extinguisher, it should be used at a range between 4 to 6 feet from the source and held within 60 degrees of the vertical.

4-8 Fig 8 The BCF Portable Fire Extinguisher

Fig 8a Operating Head Schematic **Fig 8b Operation**



Cabin Protection

10. Research and development continue into means of safeguarding passengers and crew in the event of a cabin fire on the ground. Although more relevant to civil passenger aircraft, equipment improvements resulting from such research will be read across to military aircraft in due course. Two main areas of research are being followed. One is the provision of smoke hoods or masks for passengers and crew, to prevent smoke inhalation. The other is the provision of water mist inside the cabin to cool it and to wash away smoke particles. The latter system requires considerable volumes of compressed air and water to be pumped into the cabin through the aircraft air conditioning ducting. Since this process relies on the availability of specialist ground equipment, its use would be restricted to major airports.

CHAPTER 9 - ICE AND RAIN PROTECTION SYSTEMS

Introduction

1. The operation of military aircraft may necessitate flying in adverse weather conditions. Provision must therefore be made to safeguard the aircraft against icing, the effects of which may endanger performance and safety. The areas on an aircraft which are sensitive to ice formation are:

a. Aerofoil surfaces
b. Engine intakes
c. Engine internal surfaces
d. Rotor blades and propellers
e. Windscreens
f. Instrumentation probes and vanes
g. Control hinges and linkages
h. Weapons and weapon carriers.

Principles of Operation

2. Ice protection systems are either active or passive. Active systems operate either by increasing the temperature of local areas of the aircraft to above freezing point, or by chemically reducing the freezing point of precipitation impinging upon the aircraft. Passive methods harness the momentum of the main airstream to separate out precipitation and divert it away. Active systems may be further categorized as either anti-icing or de-icing. Anti-icing systems prevent the formation of ice in critical areas whilst de-icing systems work to remove ice which has already formed.

3. Many active systems will perform both functions, and the type used for each particular application will depend both upon the sensitivity of a specific area to the effects of ice, and upon the overall need to minimize aircraft weight and aircraft power consumption. Most aircraft utilize more than one type of system, because of the wide range of requirements. De-icing systems tend to be lighter and use less energy, but in certain areas the formation of any ice cannot be tolerated and therefore an anti-icing system must be used. Such an area is the engine air intake. Any build-up of ice would dramatically reduce its aerodynamic efficiency - thus affecting engine performance - whilst ice dislodged by a de-icing system would be ingested risking an engine flame out and damage to compressor blades. Both active and passive systems are used for intake anti-icing.

Ice Protection Systems

4. **Thermal (Hot Air)-Airframe**. The majority of airframe structure anti-icing and de-icing systems utilize hot air bled from the engine compressors. Fig 1 shows the airframe areas of a typical medium transport, which may be protected by engine bleed air. Such systems sometimes allow air to be bled from an APU for anti-icing use during critical periods of flight, and this configuration allows anti-icing to be used during an emergency landing, even though maximum power from the main engines may be essential, and therefore no engine bleed air is available. The hot air is fed through a system of selector

valves, pressure regulators, and mixing valves which reduce the pressure and temperature of the air to operating levels. The controlled air is then ducted through galleries to relevant areas.

4-9 Fig 1 Aircraft Structure Anti-icing System

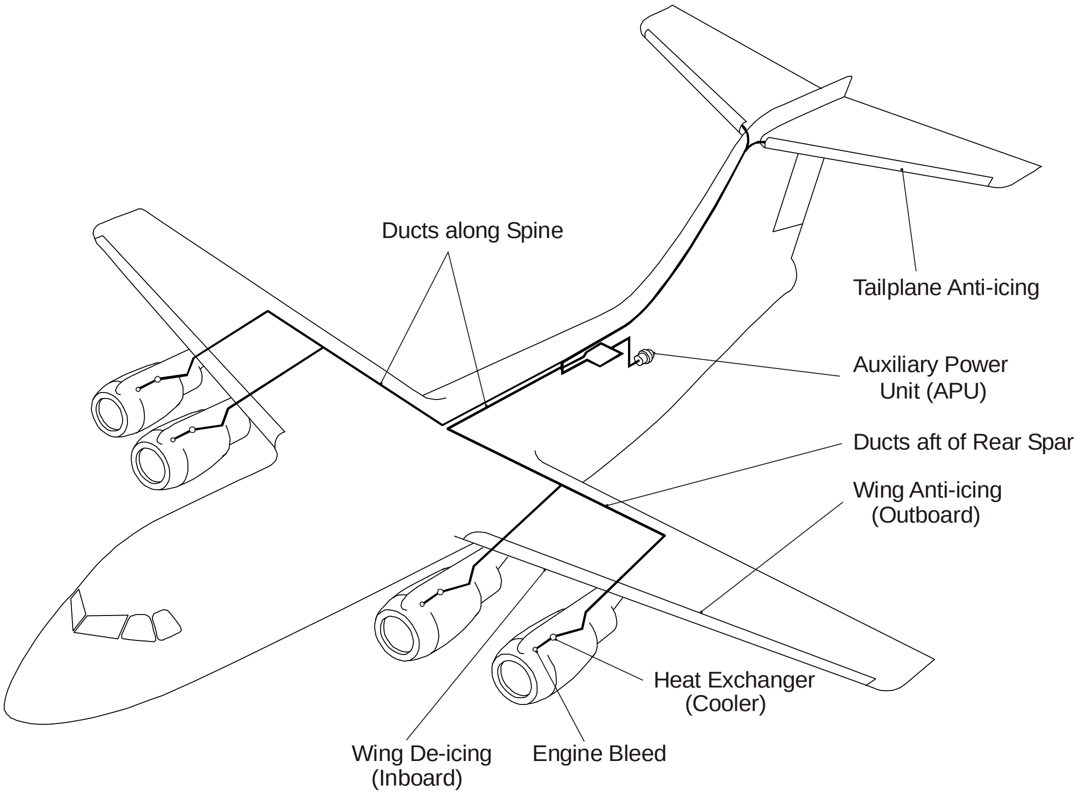
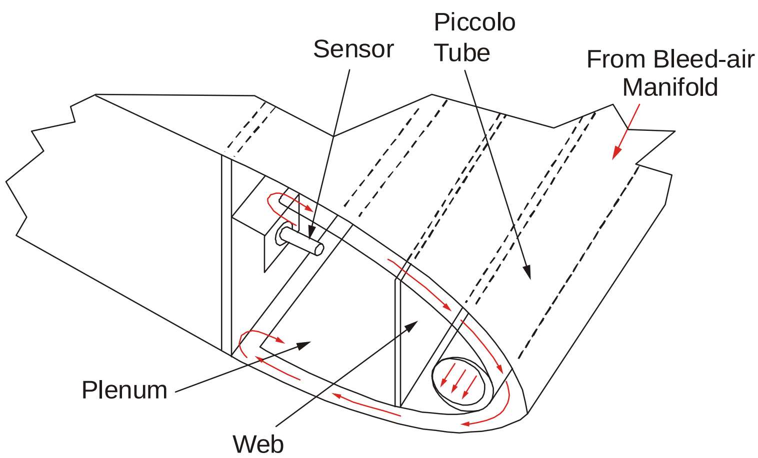


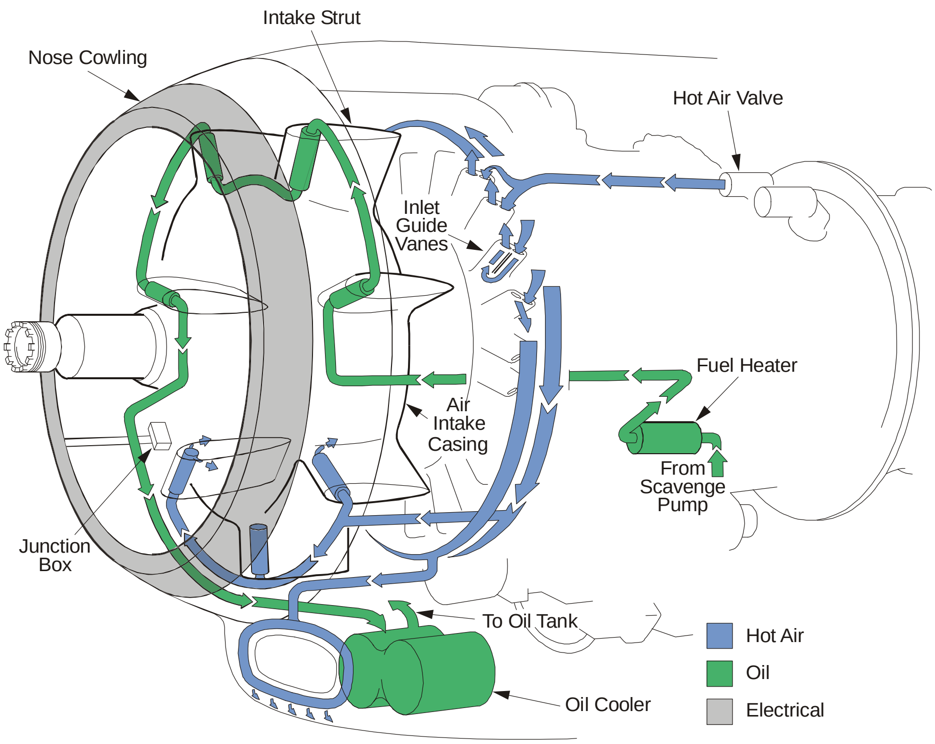
Fig 2 shows the configuration of a typical hot air bleed to a wing leading edge. Included in the illustration is a temperature sensor installed to activate system temperature control valves and to provide a warning to the crew if overheating occurs. Normal cockpit indications include the temperature and pressure of air in the system.

4-9 Fig 2 Typical Wing Leading Edge Anti-icing System



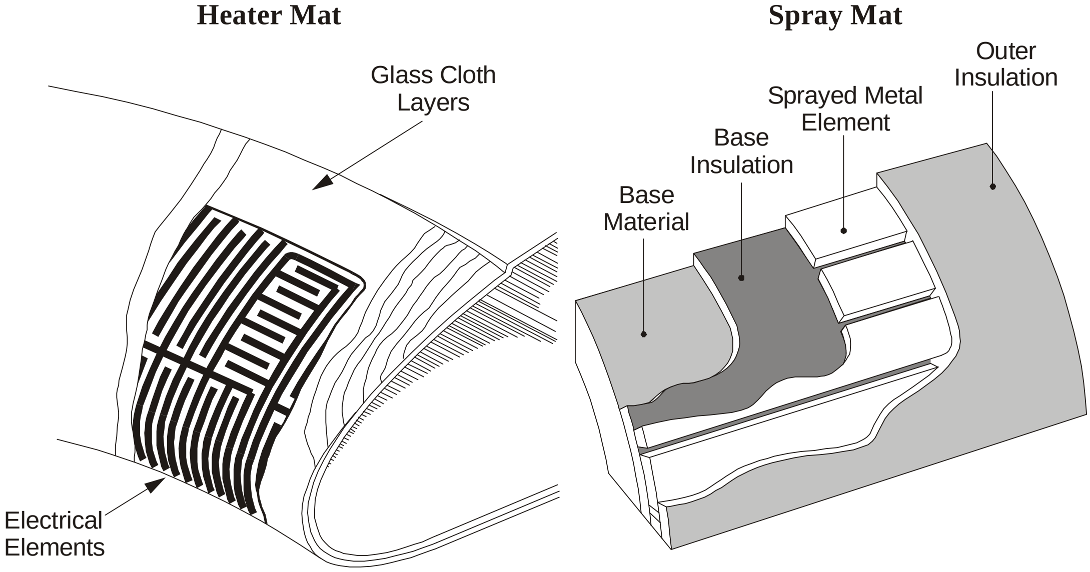
5. **Thermal (Hot Air) - Engine**. Hot air systems are also used for engine anti-icing. The components which require protection include the inlet guide vanes and first stage compressor stator blades plus the nose cone and structural support members within the intake. Engine anti-icing systems are often an integral part of the engine and are independent of the airframe anti-icing system. Fig 3 shows such an arrangement.

4-9 Fig 3 Integral Engine Anti-icing System



6. **Thermal (Electrical)**. Although hot air systems offer advantages of simplicity and robustness, electrical heating is widely used for anti-icing and de-icing systems when complex control arrangements are needed or only small areas require to be heated. Electrical systems usually include heater elements made from copper-manganese alloy either built up onto a backing material or deposited (sprayed) onto the backing. Fig 4 shows both a built up and a deposited system.

4-9 Fig 4 Electrically Heated Mats



The versatility which such manufacturing techniques offer, and the small cross section of the resultant element make them ideal for anti-icing pitot heads, static vents and other probes and vanes such as stall/angle of attack indicators. Heater mats are also used for de-icing helicopter rotor blades and fixed wing propellers. The rotor blade application offers particular problems in system control. A typical rotor blade has a span of 8 to 10 metres and a cord of 0.5 metres. The electrical load required to heat such large areas continuously exceeds the power available in a helicopter, and therefore the blades are de-iced by intermittent heating. However, the aerodynamic and dynamic balance of rotor blades are critical to the controllability and airworthiness of the aircraft. Therefore, blade heating must be programmed so that the ice build up and subsequent break down occur symmetrically, and the control system must protect against asymmetric failure of the heater mats. Because of the resultant complexity and cost of helicopter rotor blade anti-icing, helicopters operating in temperate or tropical areas are not normally equipped with blade anti-icing systems.

7. **Chemical (Fluid) Diffusion**. Chemical fluid systems are limited in use to anti-icing aerofoil surfaces. The advantage of chemical diffusion methods is that they require only a limited power input. The disadvantages are that they require replenishment after use and that they are difficult and expensive to maintain and repair. Their principle of operation is shown in Fig 5. When anti-icing or de-icing is required, de-icing fluid is pumped from a reservoir into porous surfaces which form the aerofoil leading edges. The fluid diffuses through to the surface of the leading edges where it mixes with any moisture lowering its freezing point. This prevents the formation of ice and causes existing ice to break away. The effectiveness of chemical fluid systems is very dependent upon an even distribution of fluid over the aerofoil leading edge. In turn, distribution is sensitive to the aerofoil angle of attack and the resultant air-stream pattern along its top surface. Consequently, most aircraft equipped with chemical fluid systems must be flown within a restricted speed band when the system is in use.

4-9 Fig 5 Typical Fluid Anti-icing System

Fig 5a Schematic

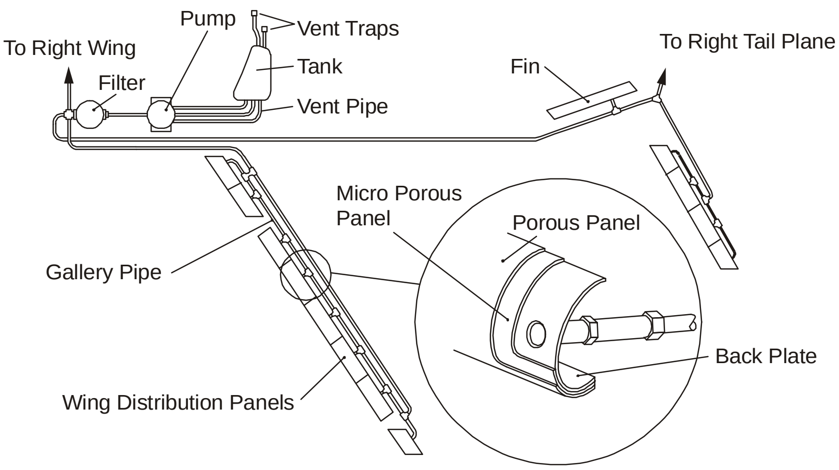
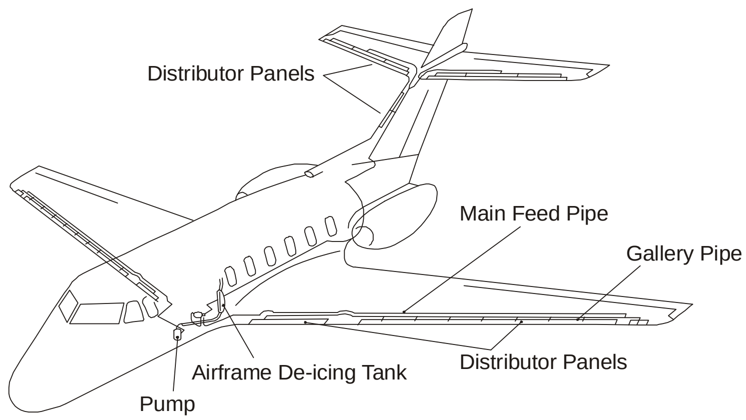
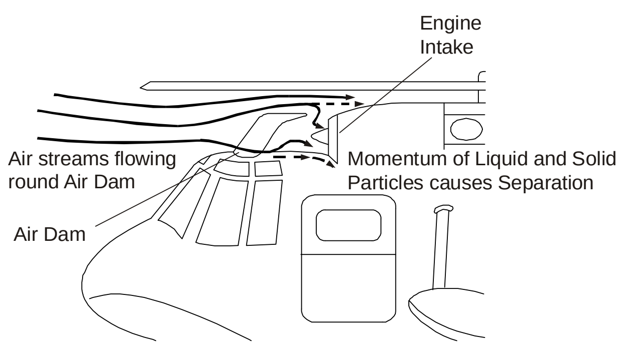


Fig 5b Location of Components



8. **Momentum Separation**. Momentum separation devices are used for anti-icing the engine intake systems of helicopters and some ground attack aircraft and also to protect exposed control system components. They are passive devices, and their principle of operation is to force the air stream to make sharp changes in direction and therefore of velocity. During such changes, the higher momentum of water particles - because of their higher mass - causes them to separate from the main air stream. They can then be deflected away from the intake or other critical area. Fig 6 shows a common form of momentum separation device - the air dam, or 'barn door', and its principle of operation.

4-9 Fig 6 Principle of the Air Dam Separator



9. **Limitations and Effects**. Because momentum separator systems interfere with the air intake ram effect, their use is restricted to helicopters and other slow flying or piston engine aircraft which do not harness the ram effect. There are many different configurations of separator ranging in complexity from the most simple arrangement of positioning the engines with their intakes facing downstream - so that the intake air stream must turn through 180 degrees throwing water and debris clear - to the Aerospatiale Polyvalent (multi-purpose) intake shown at Fig 7a. Figs 7b and 7c show two more commonly used systems. Fig 7b is an intake shield and 7c is the more ingenious wire grill or basket. In non-icing conditions, the grill imposes little resistance to the air stream, but in icing conditions, air passing through the grill speeds up and rapidly cools down causing water particles to freeze and adhere to the mesh. Thus, a shield of ice rapidly accumulates and protects the intake in much the same way as does the conventional air dam. The design of the grill is such that ample surface area along its sides will always remain clear of ice to allow sufficient air to enter the engine. The theoretical disadvantage of the grill intake is that when the aircraft enters warmer air with a frozen grill ice will melt and break away to enter the engine. In practice however, this problem is not significant. First, the mesh size of the grill is selected during design to control the size of ice particles which do break away, and secondly, such large changes in climatic conditions are seldom encountered (or can be avoided) within the normal sortie pattern of a helicopter.

4-9 Fig 7 Momentum Separation Device Configuration

Fig 7a Aerosptiale Polyvalent (Multi-purpose) Intake

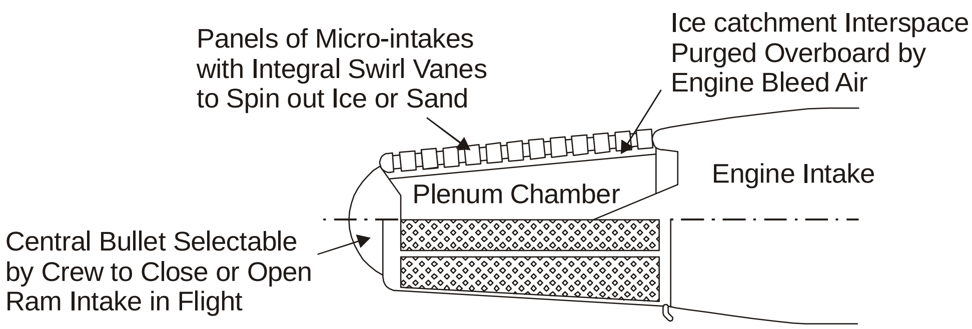


Fig 7b Intake Shield

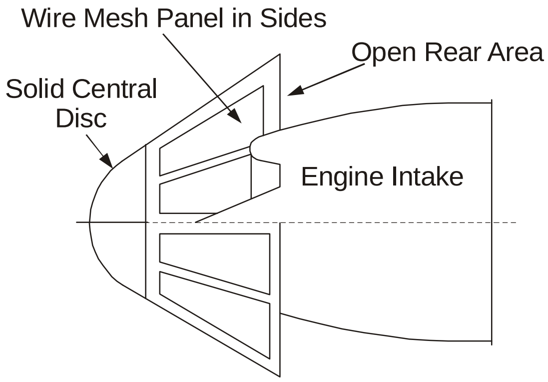
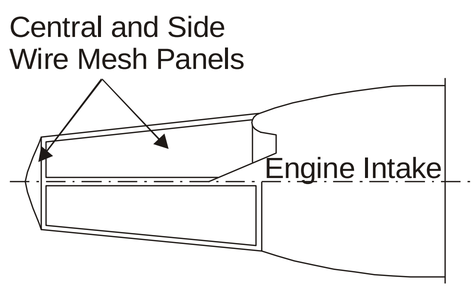


Fig 7c Wire Mesh 'ChipBasket' Intake



Ground De-icing

10. Active and passive anti-icing and de-icing systems are designed to become effective as soon as the aircraft engines are started so that the aircraft can be protected from ice formation during the critical take-off and subsequent climb-out phases of flight. However, if the aircraft has been parked in the open in adverse conditions prior to start-up, significant accretions of ice, snow, or slush may have built up on the aircraft flying surfaces. Such deposits must be removed prior to flight, by the ground crews. After physically removing the majority of such deposits, chemical fluid de-icing is used to complete the task. This is achieved by the application of de-icing fluids in specially prepared thyxotropic paste or gel form by the use of ground-based spray equipment. The effect of applying such a de-icing gel is to melt any ice present and to prevent its reformation until the aircraft is airborne. The principle is also sometimes used for the airborne anti-icing of unheated rotor blades on helicopters which must fly for operational reasons in icing conditions. However, the effectiveness of the gel reduces during flight as it is gradually thrown from the blades by centrifugal forces.

Ice Detection

11. Although significant flight hazards are posed by ice build-up, the fact that most active anti-icing and de-icing systems consume considerable amounts of power preclude their use except when icing conditions are actually encountered. Meteorological forecasts go much of the way to alerting crews to the likelihood of entering icing conditions during a particular flight. However, such forecasts are not always sufficient, and the crew must therefore remain alert to the need to activate the anti-icing and de-icing systems at any time. The formation of ice on external visible projections would normally be the first manifestation of having entered icing conditions. For this reason, the majority of aircraft are equipped with flood lights aligned to illuminate relevant areas of the airframe which are visible from the cockpit. Aircraft in which crew visibility is limited are often equipped with illuminated ice accretion probes as shown in Fig 8a. These are positioned to be visible from the cockpit. In addition, most aircraft are equipped with ice detection devices which either provide a positive alert or automatically activate the anti-icing and de-icing systems.

4-9 Fig 8 Ice Detection Devices

Fig 8a Teddington Visual Ice Detector

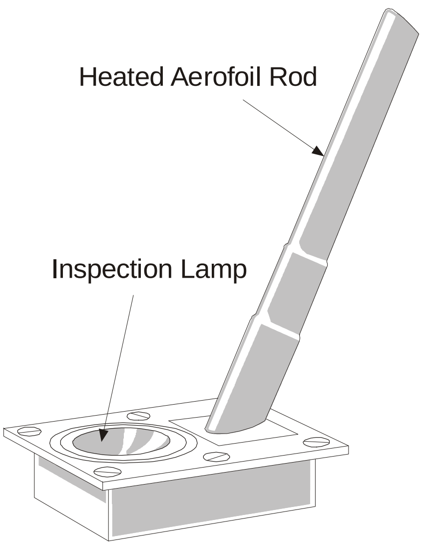


Fig 8b Smiths Differential Static Pressure Ice Detector

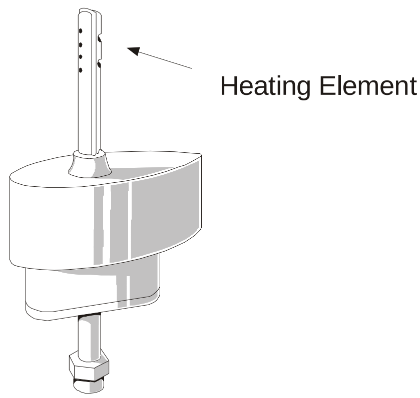


Fig 8c Rosemount Frequency Monitor Ice Detector

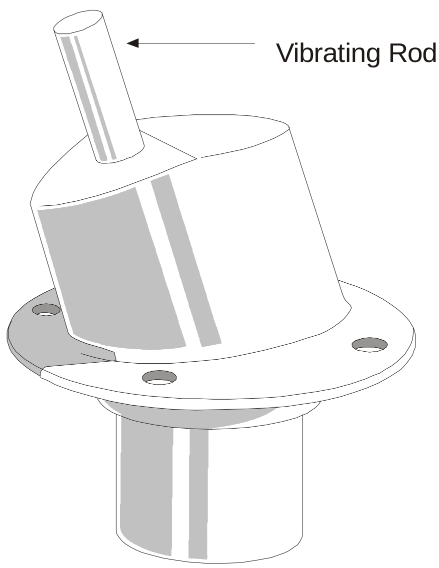
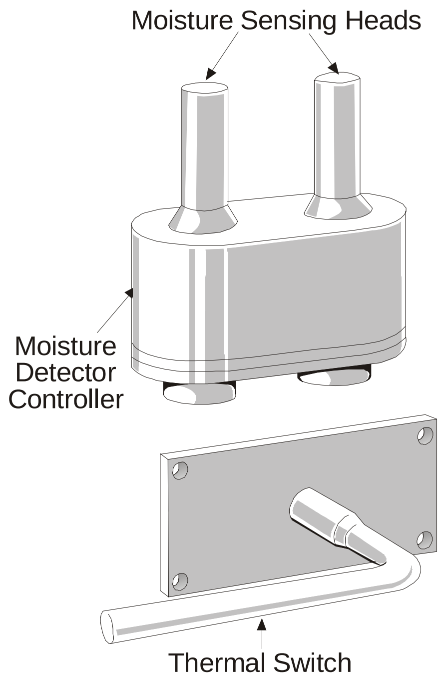


Fig 8d Sangamo Weston Icing Condition Monitor



12. Many different principles of operation are used in ice detection devices, but all either detect the actual build-up of ice or the conditions in which a build-up will occur. Three different devices are shown in Fig 8a to d. The probe shown in Fig 8b contains a series of holes positioned in its leading edge and a separate series in its trailing edge. The detector monitors pressure differential between the two edges. In icing conditions, holes in the leading edge rapidly become blocked by ice. This causes a change in the pressure differential. The change is detected, and a cockpit alert is activated. The device in Fig 8c utilizes the change in resonant frequency of a probe which occurs when ice forms on it. The probe is

vibrated at its clean resonant frequency of about 35 kHz. The mass of any ice which forms on the probe will reduce this resonant frequency, and the detector senses any significant frequency change and activates the cockpit alarm. The device in Fig 8d works on the same principle as a wet and dry bulb hygrometer, and it comprises two heated bulbs, one exposed to the air stream and the other shielded, plus a simple outside air temperature (OAT) probe. The detector monitors the temperature of the bulbs which are heated at a constant rate. When the exposed bulb is in a moist air stream, it loses its heat to the surrounding air at a greater rate than does the dry shielded bulb. The resultant temperature imbalance is detected. If the OAT is detected to be within the icing range, contacts in the probe circuit close, and the alert system is activated.

Windscreen Ice and Rain Protection

13. Although de-icing fluid spray systems or hot air jets were utilized to de-ice the windscreens of older aircraft, all current aircraft are fitted with electrically heated screens. The heating elements and associated temperature control and overheat sensors are sandwiched in the glass laminations of the screen. A thin film of gold is used for the heating element, and it is deposited directly onto glass. Electrical connectors formed on the edges of the panel interface with the system electrical supply and temperature controller. The heater systems serve both to de-ice and de-mist the screens.

14. At normal flying speeds, rain falling onto the screens is rapidly dispersed by the airflow. However, to keep the screen clear during landing or during low speed flight, conventional, high-speed windscreen wipers are fitted to most fixed and rotary wing aircraft. The wipers are electrically activated by the crew as and when needed.

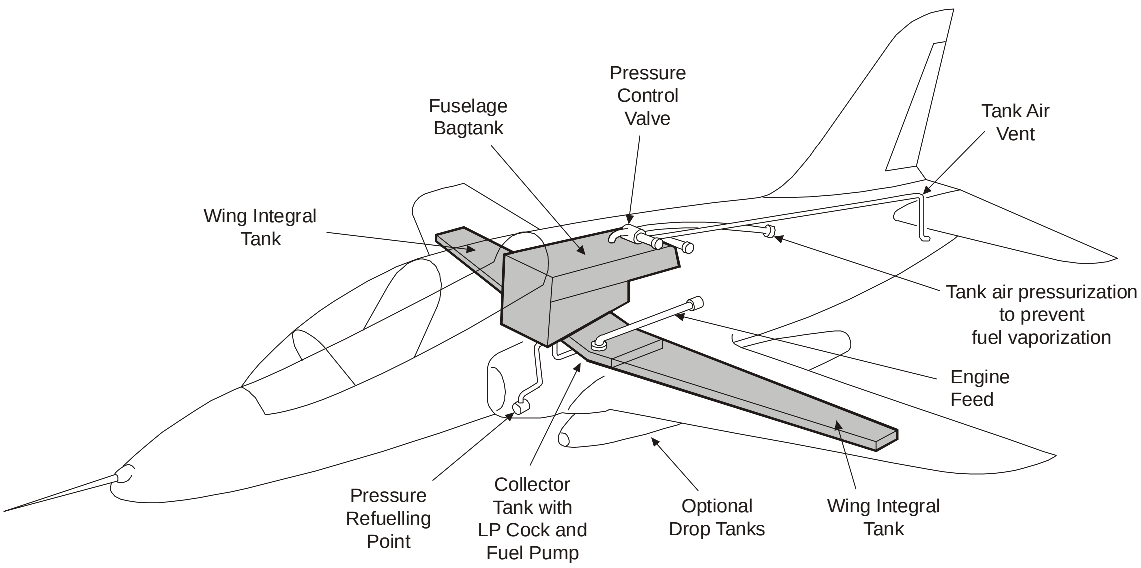
CHAPTER 10 - AIRCRAFT FUEL SYSTEMS

Principles

1. The fuel system of an aircraft consists of two distinct sub-systems. One is integral with the engine or Auxiliary Power Unit (APU) and the other with the airframe. A typical engine system comprises a high pressure (HP) pump, final filtration, a fuel control unit (FCU) and a carburation device which introduces the fuel into the combustion system. Details of engine fuel systems are included at Volume 3, Chapter 11. The functions of the aircraft airframe fuel system are to store the fuel until it is required and then to deliver it in quantities appropriate to the power being demanded of the engines or the APU, to a set pressure and quality.

2. To reduce crew workload, and to minimize the risk of a fuel management error occurring during flight, control of the system requires to be automatic or semi-automatic in operation. Also, to provide for the extensions in aircraft range and endurance imposed by a wide variation in operational requirements, the system configuration requires sufficient flexibility to be extended when operations demand greater range or endurance. In many cases, this is achieved by fitting additional tankage or by providing a capability to pick up additional fuel during flight. A typical single-engine fuel system is illustrated in Fig 1. Fuel is held in integral wing and fuselage tanks, with (in this example) the option of extra fuel in an external tank.

4-10 Fig 1 Typical Combat Aircraft Tank Configuration



Fuel Storage

3. **Tank Position**. Fuel is stored in tanks which are usually an integral part of the aircraft structure or are constructed from flexible fabric membranes or bags. The strength and rigidity of such bags are derived from the aircraft structure. The disposition of the tanks depends upon the role of the aircraft and therefore the priority for space within its airframe. For instance, the tanks of transport aircraft are generally situated in the wings, those of helicopters are beneath the cabin floor and those of combat aircraft in the wings and centre fuselage. The need to maximize the amount of fuel carried in flight has

led to much ingenuity in tank location. Tail fins, flaps and the outer walls of air intakes have all been used as tanks in various aircraft types.

4. **Structure**. The walls of integral fuel tanks are formed by the aircraft structure. Considerable care must be taken during construction to ensure that all joints and inspection hatches in the structure are adequately sealed and that tank walls are treated to prevent corrosion. Such corrosion is usually caused by bacterial action which takes place at the interface between the fuel and any water, which may settle into the bottom of the tank. Fuel additives prevent the formation of such bacteria, but the availability of treated fuel cannot be guaranteed in all operational circumstances. Bag tanks do not suffer the same problems of sealing and corrosion, but their use imposes both a weight penalty, and the need to remove them for periodic maintenance. This necessitates access ports to be provided in the surrounding structure.

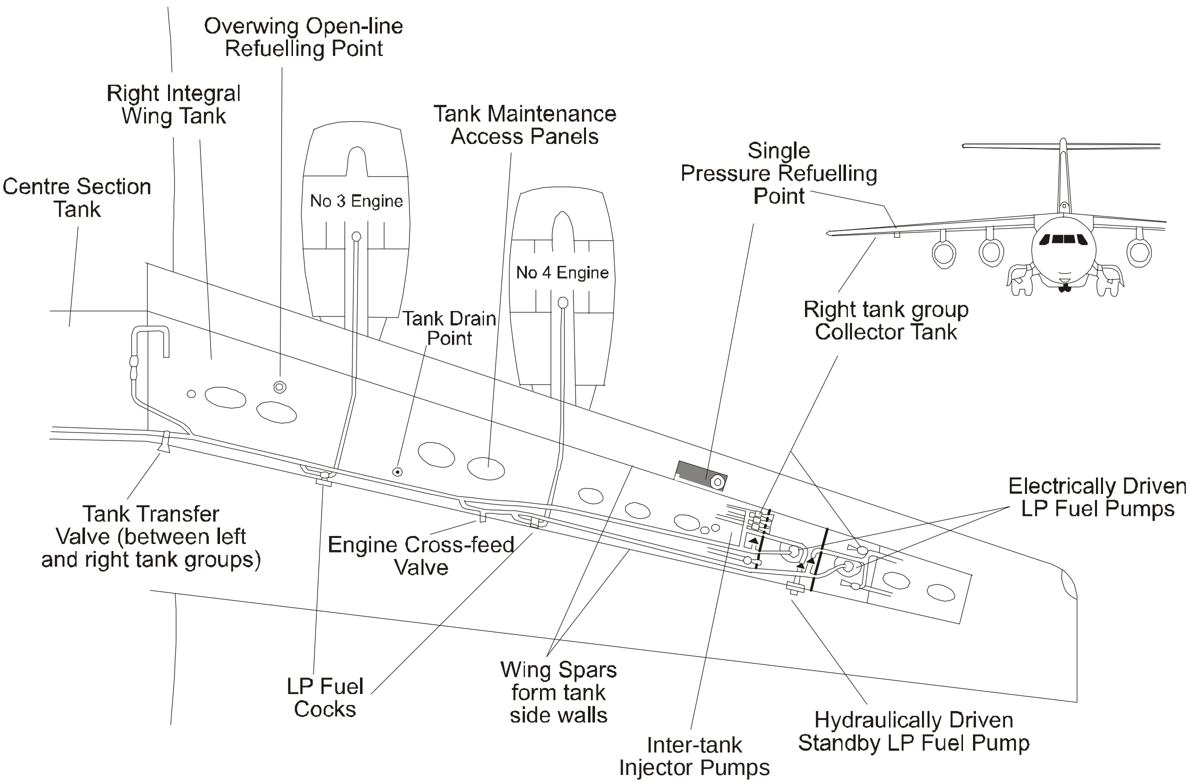
5. **Collector Tank**. For ease of control and system integrity, fuel tanks are usually arranged in groups. On multi-engine aircraft, fuel from each group feeds one specific engine, although the facility to transfer fuel to other engines or tank groups is provided. Each tank in a group feeds fuel through pipes or galleries into a collector tank which is therefore always full of fuel. The collector tank feeds the engine directly, thus, an uninterrupted supply of fuel is ensured to each engine during periods of turbulence or manoeuvre. Devices to ensure the supply of fuel during extreme manoeuvre, such as inverted flight or flight in negative 'g' conditions are discussed in para 22.

6. **Tank Pressurization**. The boiling point of fuel will vary with the temperature of the fuel and the pressure at the fuel surface. If an aircraft is refuelled with warm fuel and then climbed to altitude, the pressure above the fuel is reduced whilst the temperature, because of the large volume involved, remains essentially the same. The fuel will boil and vapour will form which could form vapour locks and engine malfunction. The primary method of preventing this boiling action is to apply a positive air pressure above the fuel. The fuel tanks are therefore pressurized by engine air, regulated by a pressure control valve. The pressure control valve incorporates a non-return valve (NRV), a reducing valve and a relief valve. The NRV prevents reverse airflow to the engine during refuelling and stops fuel entering the air line. The reducing valve controls the air pressure to a specified value and the relief valve prevents overpressure damage by venting excess air if the reducing valve fails to operate.

Delivery System Components

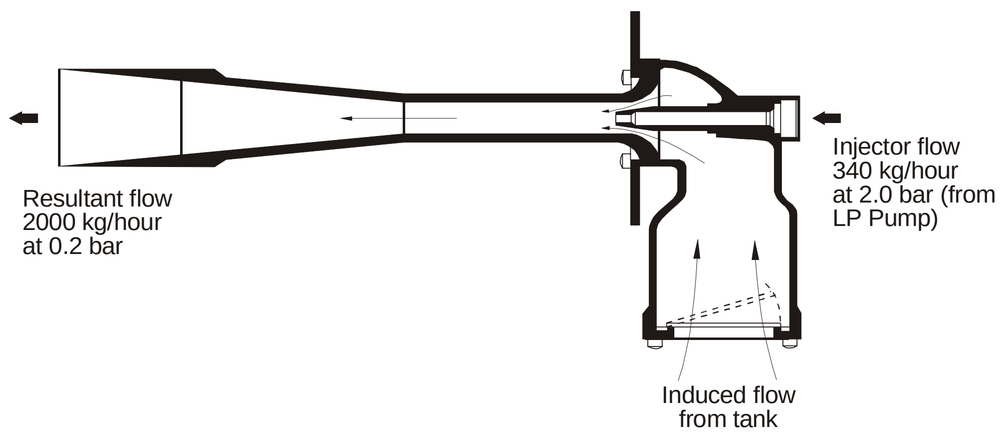
7. **Low Pressure Cock and Pumps**. The boundary between the engine and airframe sub-systems is always defined as the low pressure (LP) cock which is fitted as the final component in the airframe system. The typical arrangement of tank groups in a multi-engine aircraft, including the position of the LP cocks, is shown at Fig 2. When required for the engines or APU, fuel is fed from the collector tanks by low pressure (LP) pumps.

4-10 Fig 2 Arrangement of Tank Groups and Controls



These provide a backing pressure to the engine system HP pumps. LP pumps are often termed booster pumps. The LP pumps are electrically or hydraulically driven, and they run fully submerged in the collector tanks. Unless the aircraft configuration is such that fuel will flow from the collector tanks to the engines by gravity, multiple LP pumps are provided to obviate fuel starvation occurring in the event of pump failure. The intakes of LP pumps incorporate a coarse filter and also a by-pass valve to allow fuel to continue to flow in the event of filter blockage or pump failure. Jet (or injector) pumps are often used to feed fuel from storage to collector tanks. Such pumps work on a venturi principle and the motive force is provided by fuel bled from the LP pumps. Fig 3 shows their principle of operation. The advantages of the jet pump are that it requires no separate power supply or control circuits and that it is extremely reliable.

4-10 Fig 3 Fuel Injector (Jet) Pump



8. **Water Drains**. Hydrocarbon fuels tend to absorb water, and such water will precipitate out of the fuel when its temperature drops. There is therefore a likelihood of some water collecting in aircraft fuel tanks, despite all possible precautions being taken to maintain the quality of fuel up to the point of it being pumped into the aircraft. All tanks are fitted with simple to operate water drain valves, positioned in the tank bottoms, and these are operated during daily servicing to dump any water which may have collected.

9. **Filters**. Although all fuel is filtered to a high standard immediately prior to being dispensed into the aircraft, there remains a likelihood that debris may enter the fuel either through the tank vents, from residual deposits in the tanks or through open line refuelling points (see Para 17). Therefore, as well as there being a very fine filter in the engine fuel sub-system, a relatively coarse filter is usually included in the airframe sub-system. Such filters are often of the paper cartridge type and include either visual tell-tales or electrical warning devices to indicate blockage. The filters also incorporate by-pass systems to ensure that a continuous supply of fuel, albeit unfiltered, reaches the engines in the event of filter blockage.

10. **Venting**. As mentioned in para 6, fuel tanks require to remain either at the pressure altitude of the aircraft or, more usually, at a small positive differential pressure during flight. The tanks therefore require venting systems, which control the entry and exit of air both during flight and on the ground. The requirements of such a system are:

a. To allow air to enter as fuel is consumed, as the aircraft descends or as the fuel cools and contracts.

b. To allow air to exit as it is displaced during refuelling, as the aircraft climbs or as the fuel warms and expands.

c. To maintain the tanks at a controlled positive pressure differential thus reducing vaporization and, in some systems, providing the driving force for fuel transfer.

d. To prevent fuel being lost from the venting system during flight manoeuvre - although the system must allow fuel to vent from full tanks as it warms and expands during diurnal temperature cycles.

The vent system usually comprises one or more pipes positioned in the top of the tanks and which duct air from outside the aircraft into all the tanks in the group. A system of float valves in the pipes allows air to enter the tanks when their pressure is lower than the pipe pressure, and air to vent from the tanks when their pressure is higher. The float valves serve to prevent any significant amounts of fuel from entering the vent pipes, but because small amounts of fuel will be carried into the pipes during manoeuvre or during heavy venting, a fuel surge tank is provided at the entrance to the pipe system, to separate out this fuel. Air enters the system through a ram vent in the lower wing surface, thus ensuring that during flight the system pressure is always a little higher than static pressure. During normal operation, air enters and vents from the system as necessary, and any fuel vented with this air collects in the surge tank. When aircraft fuel contents are sufficiently low for the surge tank drain float valves to open, vented fuel is allowed to drain back into the main tanks. However, if relatively cold fuel is pumped into the tanks during replenishment, its volume will increase as the fuel heats up to tank-soak temperature. Any fuel displaced from the tanks through such expansion will vent into the surge tank, and if this becomes over full, excess fuel will drain from the aircraft through the wing vent giving a misleading impression that a tank fuel leak exists. This situation frequently occurs when aircraft are refuelled early in the morning and remain parked in hot sunlight throughout the day.

Design Objectives and Typical Configuration

11. Considering the foregoing description, fuel system design must take into account the following criteria:

a. Optimum use of the fuselage space available for fuel storage.

b. Delivery of fuel to the engines and APU at required flow rates and to design pressure and quality.

c. Cross flow of fuel between specific tank groups and engines and the transfer of fuel between tanks both to allow manipulation of the aircraft's required centre of gravity position during flight and to counter any feasible system malfunction.

d. Ease of system control and monitoring during flight.

e. Rapid and safe fuel replenishment, and the flexibility to satisfy fuel requirements posed by a wide spectrum of operations.

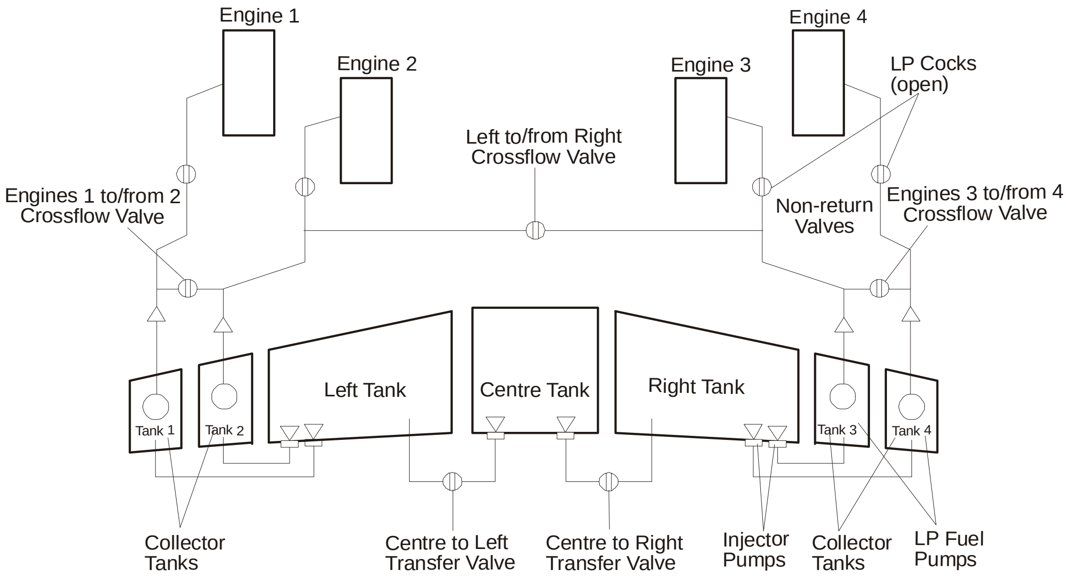
f. Tolerance to aircraft manoeuvre and to damage.

g. Secondary uses of fuel - for example as a coolant.

Transfer, Cross-feed and Jettison

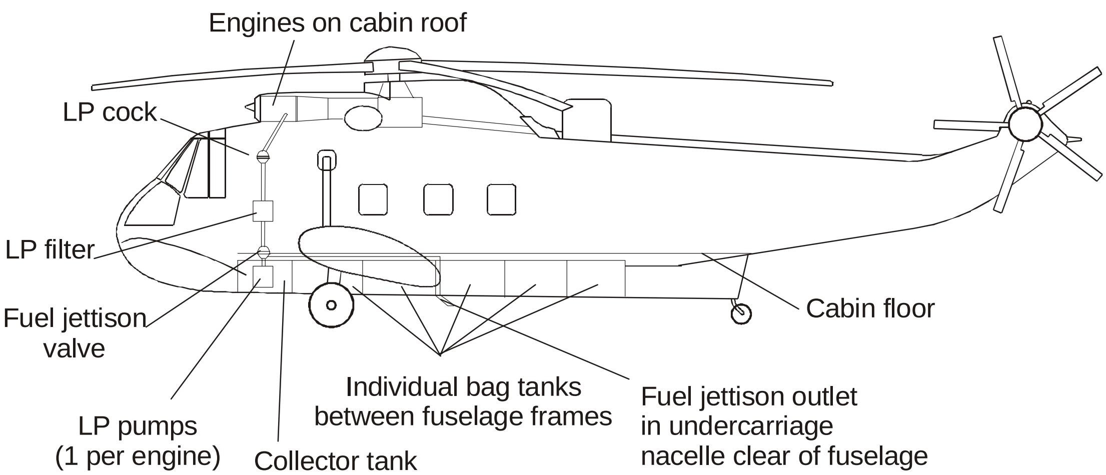
12. **Transfer and Cross-feed**. In the majority of aircraft, distribution of fuel is critical to aircraft balance. Although devices such as fuel proportioners are used wherever possible to meter the flow from individual tanks and thus achieve a degree of automatic control of fuel distribution, system malfunction, uneven rates of fuel burn between engines, or even the gradual reduction of fuel load during flight will necessitate system management action being taken to redistribute fuel from one tank to another. Fuel transfer is achieved either automatically by the control system sensing an imbalance or by crew selection. Similarly, if an engine is closed down in flight or a lesser imbalance of fuel burn between engines occurs, it will become necessary for the crew to cross-feed fuel from one group of tanks to another engine. To maintain fuel system integrity, transfer and cross-feed networks are independent of each other. Fig 4 shows a typical fuel system with transfer and cross-feed facilities. In this case, the configuration of the multi-engine aircraft allows transfer to be accomplished through a simple gravity feed line between the two tank groups. In other cases, such as depicted in Fig 2, fuel must be transferred by pumps.

4-10 Fig 4 Fuel System with Transfer and Cross-feed



13. **Jettison**. Because of undercarriage and structural stress constraints, the maximum permissible take-off weight of an aircraft is often considerably higher than its maximum permissible landing weight. It is therefore possible for the crew of such an aircraft to take off at maximum weight, experience an in-flight emergency or be recalled for operational reasons, and then be unable to land safely until sufficient fuel has been consumed to bring the weight within safe landing limits. To resolve this problem, most systems incorporate a facility for dumping fuel in flight. The system in Fig 5 includes such a facility. Fuel is pumped, or it feeds by gravity, through a system of valves and pipes to be vented overboard well clear of the fuselage and the jet efflux.

4-10 Fig 5 Tank Jettison System



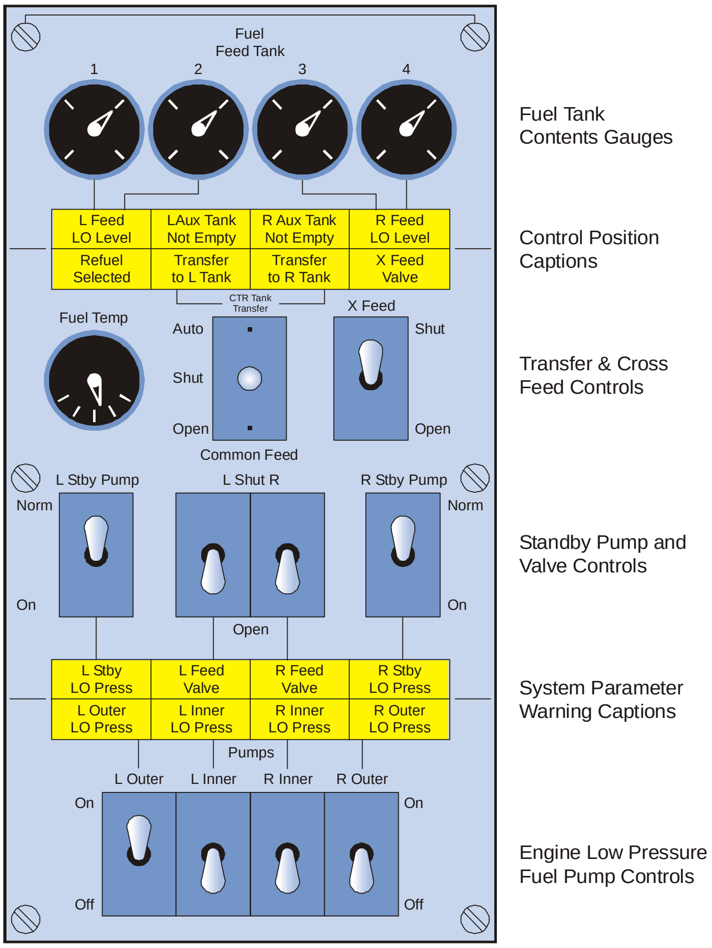
Systems Management

14. **Control**. Operation of the fuel system is often automatic or semi-automatic, and many such systems start to operate as soon as the crew has opened the LP cocks and switched on the LP pumps. Control is maintained by float switches sensing changes in fuel levels and operating appropriate valves and pumps.

However, crew intervention is normally required to initiate transfer of fuel between tank groups, cross-feed fuel from tank groups to different engines and to jettison fuel.

15. **Indications**. Instrumentation within the fuel system includes continuous measurement of fuel contents plus indicators to show system configuration, alerting the crew to low fuel pressure or contents and warning of malfunctions, such as pump failure or filter blockage. Fig 6 shows a simple fuel control panel and indicators within the cockpit of a multi-engine aircraft. The gauging of fuel contents in most aircraft is achieved by integrating the signals received from a network of sensor units positioned throughout the tank systems. Electrical capacitance of the sensors varies proportionally to the depth of fuel surrounding them, and this enables fuel contents to be computed and presented to the crew. In simple aircraft, float-actuated variable resistors are used to sense fuel levels and to effect an appropriate gauge reading.

4-10 Fig 6 Fuel System Control and Instrumentation Panels



Refuelling and Additional Fuel

16. **Pressure Refuelling**. Most aircraft have a fuel capacity measuring thousand or even tens of thousands of litres. To uplift such large volumes rapidly, cleanly and with minimum risk of spillage, pressure refuelling is the normal method used. Ground facilities are used to deliver fuel at a standard pressure of 3.75 bar through hoses and quick-release couplings. The hoses are electrically bonded,

and aircraft are additionally bonded to the installation during refuelling. This ensures that aircraft and installation are at the same electrical potential and that static charges built up in the fuel, because of the high flow rate, are safely dissipated. Many aircraft systems allow refuelling to selected partial fuel loads to be achieved automatically. Such systems usually utilize signals from the gauging system or a series of float switches to close the tank inlet valves as the desired fuel levels for each tank are reached.

17. **Open-line Refuelling**. Some smaller turbine powered aircraft and all piston driven aircraft are refuelled at low pressure through open nozzles feeding into the top of the aircraft tank system. The technique is also known as gravity or over-wing refuelling, and it is identical to the system used for motor vehicles. Because the equipment is so widely available, open-line refuelling is often held as the reserve system at remote airfields or in the battlefield. Therefore, many aircraft which are likely to operate in such areas have an added capability for open-line refuelling. The system depicted in Fig 2 shows such a feature on a medium transport aircraft which can be expected to operate into relatively remote airfields.

18. **De-fuelling**. For operational reasons or during servicing, the requirement often occurs to reduce the fuel load of an aircraft prior to take off. To achieve this, pressure refuelling systems all include a facility for pumping fuel out of the aircraft tanks, after appropriate manipulation of the aircraft system controls. To avoid damage to the aircraft structure, the suction level used for de-fuelling is much lower than the pressure for refuelling.

19. **Additional On-board Fuel**. During the design phase, the fuel system capacity is optimized for the range or endurance normally required for the aircraft. Thus, if a temporary increase in range or endurance becomes necessary, such as a ferry flight, a fuel load greater than that designed for must be carried. This is usually achieved by trading payload for fuel, fitting additional fuel tanks in the cargo space or on weapons stations. For example, the aircraft depicted at Fig 1 is configured to carry ferry fuel in conventional drop tanks attached to the weapons stations. The arrangements for carrying and managing additional fuel are decided upon at the design phase, and the aircraft fuel system is built to accept the additional tankage as role equipment, to be fitted and removed as necessary.

20. **In-flight Refuelling**. When carriage of the full payload is required over an extended range, either more frequent refuelling stops must be made or, if this is not possible, the aircraft must be refuelled in flight. Most relevant fixed wing aircraft and some helicopters are equipped for in-flight refuelling. However, the rotors of most helicopters sweep a path so close to the nose of the aircraft that the use of air-to-air refuelling techniques can be impracticable. Nevertheless, helicopters frequently require to be refuelled in mid-sortie, and this is achieved by hovering the aircraft close to the ground (or with the undercarriage just touching) whilst conventional refuelling is carried out. The technique is termed rotors running refuelling (RRR), and it is commonly used to extend the range or endurance of SAR, ASW and battlefield helicopters.

Tolerance to Manoeuvre and Damage

21. **Manoeuvre**. During aircraft manoeuvre or flight in turbulent conditions, fuel in the tanks will be affected by the resultant g forces. The surges of fuel produced will, if uncontrolled, cause interruptions in flow. Indeed, in extreme cases, the rapid movement of significant masses of fuel will tend to de-stabilise the aircraft and cause structural damage to the tanks. These effects are minimized by positioning baffle plates in the tanks. Usually part of the aircraft structure, they effectively divide the tanks into small sub-compartments reducing and absorbing the energy in the surges.

22. **Negative 'g' Devices**. Although collector tanks (see para 5) ensure a stable fuel flow to the engines in most flight conditions, more positive methods are required for providing fuel during inverted flight or flight in negative 'g' conditions. Many combat or aerobatic training aircraft utilize the fuel recuperator system. The recuperator is a separate container positioned in the engine fuel supply line and always full of fuel. The container is maintained at a pressure of about 0.5 bar, slightly below normal fuel LP pump pressure, by engine bleed air. If LP pump pressure drops, because of tank fuel surging during manoeuvre or inverted flight, fuel from the recuperator is forced into the system and thus maintains the engine fuel supply during the limit of its capacity. Another frequently used and simpler device is the double entry LP pump. Situated in the collector tank, this pump has gravity operated flap valves in its lower entry port. During inverted or negative 'g' flight, the valves close allowing the now inverted pump to draw fuel through its upper port to the limit of the collector tank contents.

23. **Damage**. Many aircraft fires occurring on the ground and in the air have been caused by fuel leaks resulting from damage to the fuel system. Such damage may be caused by enemy action, disintegration of engine components or by ground impact during an otherwise survivable crash landing. Therefore, considerable effort is made to ensure that systems are tolerant to such damage. It is normal for fuel lines which of necessity pass adjacent to engines, to be armoured and fire proofed. One of the more extreme examples of designed damage tolerance is that of the Chinook helicopter. Its fuel tanks are external panniers, and no fuel is carried within the fuselage. The tanks are designed to break away from the fuselage and roll clear of the aircraft on impact.

Secondary Uses of Fuel

24. **Coolant**. The most common secondary use for fuel is as a coolant for lubrication oils or hydraulic fluids. Heat absorbed by the fuel from the fluids is dissipated into the tanks and ultimately through the aircraft structure to atmosphere. Within obvious safety limits, turbine engine fuel can be put to this use before it is burnt. However, the low flash point of gasoline tends to preclude the use of piston engine fuels for cooling purposes. Fuel is also used in supersonic aircraft to cool areas of aerodynamic heating. In such cases, cold fuel from the tanks is pumped through heat exchanger galleries in the hot structure prior to reaching the engines.

CHAPTER 11 - SECONDARY POWER SYSTEMS, AUXILIARY AND EMERGENCY POWER UNITS

Introduction

1. **Definitions**. Primary power is defined as the basic propulsive force for an aircraft and is provided by its main engines. Secondary power is defined as the electrical, hydraulic and pneumatic power generated to drive the aircraft ancillary systems. When the aircraft is airborne, secondary power is usually generated via a power take-off from the main engines. However, most aircraft have additional integral secondary power sources to augment that of the main engines, for use particularly when the aircraft is on the ground, or in the event of failure of a main engine or other emergency. These additional sources are termed Auxiliary Power Units (APUs) and Emergency Power Units (EPUs) respectively. Secondary power may also be provided at most aircraft fixed operating bases from Ground Power Units (GPUs). GPUs are used to provide power during prolonged periods of maintenance when the power output of APUs may not be adequate or their operation not practicable.

2. **Configuration**. Although the concept of secondary power and the methods of providing it can be readily defined, the related hardware is less clearly identifiable because many items are constructed to perform dual roles. The schematic diagrams in Fig 1 present typical combinations of equipment and the roles which each item fulfils.

4-11 Fig 1 Secondary Power System Arrangements

Key A Engine Accessories H Hydraulic Pumps E Electrical Generators

Fig 1a Accessory Gearbox with RAT

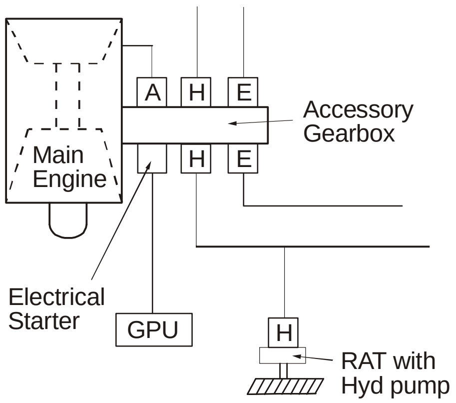


Fig 1b Accessory Gearbox with APU

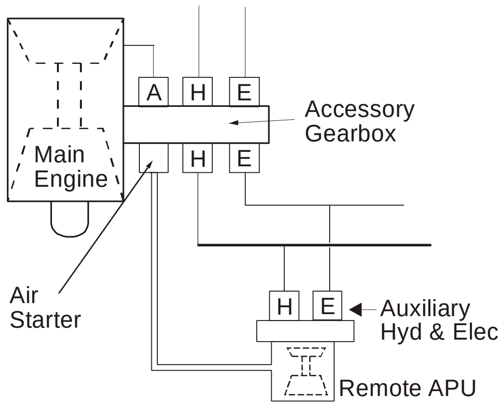


Fig 1c SPS Module with APU and EPU

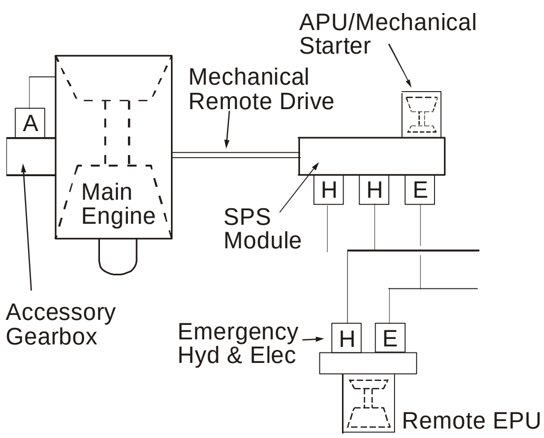
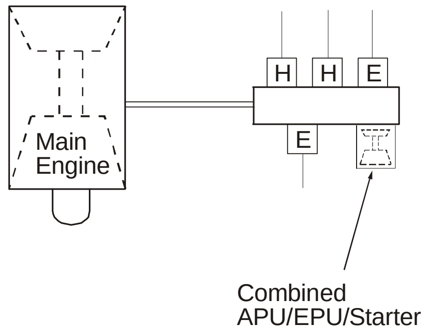


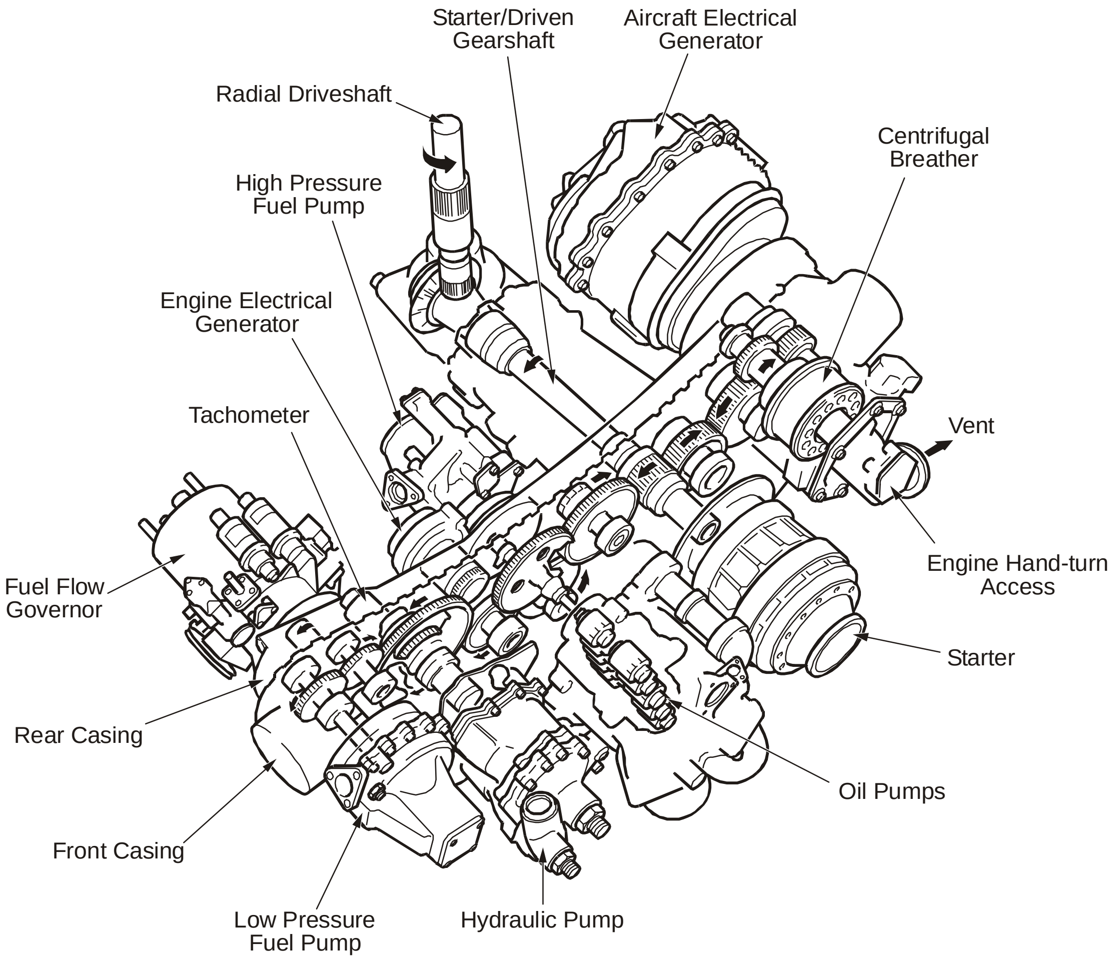
Fig 1d SPS Module with Combined APU/EPU



Modular Secondary Power Systems

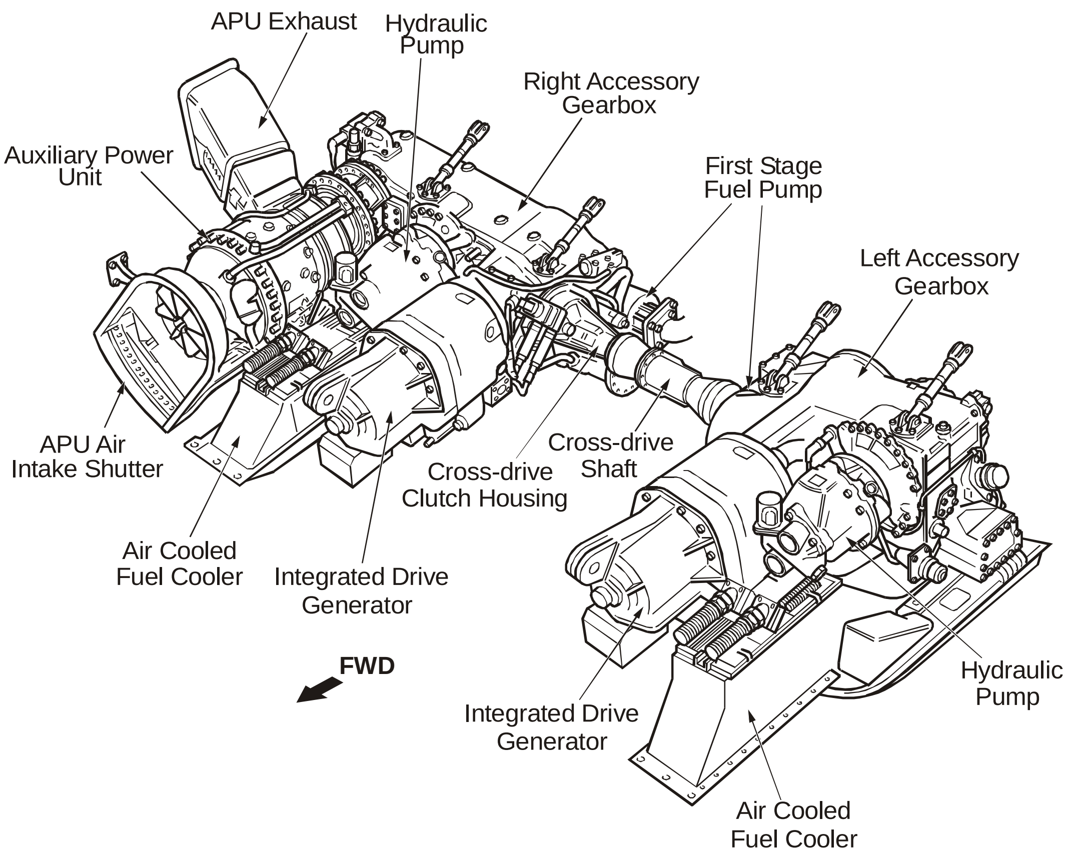
3. The provision of secondary power has evolved from the simple arrangement still common in light aircraft of attaching electrical generators, hydraulic pumps and pneumatic pumps directly to the engine and driving them through belts or drive shafts. This arrangement for generating secondary power has developed into the provision of a discrete module of the engine, termed the accessory or ancillary gearbox, providing a location for all secondary power generators and driven through a power take-off from the main engine. All engine accessories are also attached to this module. Fig 2 shows a typical accessory gearbox and the secondary power generators which it supports.

4-11 Fig 2 Engine Accessory Gearbox and Secondary Power Generators



4. Such an arrangement achieves the desirable goal of locating all secondary power sources in one unit, but it suffers the basic disadvantage of requiring all of these secondary systems to be physically disconnected from the aircraft whenever the host engine is removed for maintenance. An engine change therefore adversely affects the integrity of all secondary power systems. A method of overcoming this major disadvantage, whilst retaining the advantage of centralizing the location of all components, is to locate all secondary power generators on a dedicated gearbox mounted directly onto the airframe remote from the engine. In such an arrangement, the gearbox is still driven by the main engine either through a mechanical (shaft) or hydraulic (pump/motor) coupling, but the arrangement enables removal of the engine without disturbing the secondary power system. Fig 3 shows a typical modular secondary power system.

4-11 Fig 3 Typical Secondary Power System Module



5. A typical Secondary Power System (SPS) Module for a twin engined aircraft comprises two similar accessory drive gearboxes each mechanically driven by an aircraft engine. A freewheel attached to the drive shaft of each accessory drive gearbox effectively disconnects the respective engine when it is not running or in the event of it being closed down during flight. In the event of an engine failure, both accessory drive gearboxes can be driven by the remaining engine through a clutch and cross shaft connecting the two units. On the ground when neither engine is running, an APU attached to one of the accessory drive gearboxes can be operated to provide all secondary power requirements. A clutch disconnects the APU from the drive train, when it is not required. The APU is also used to start the main engines. In the start mode, a torque convertor in the drive train between gearbox and engine transmits power from the APU, controlled to run at full speed, into each stationary engine in turn. As the engine starts and commences to run under its own power, the torque convertor is automatically programmed to cease driving. The secondary power generators fitted to each aircraft system accessory drive gearbox include a hydraulic pump, an electrical generator and a fuel boost pump.

Auxiliary Power Units

6. The need to provide an auxiliary secondary power source on aircraft has long been recognized, and initial arrangements included the use of small piston engines mounted in the fuselage. Most current units are small gas turbine engines. Such small engines develop some 75 to l00 kW. Their design is of a constant speed, variable torque engine using a single stage centrifugal compressor to feed air to a single combustion chamber. This in turn powers a single stage radial turbine driving either an accessory

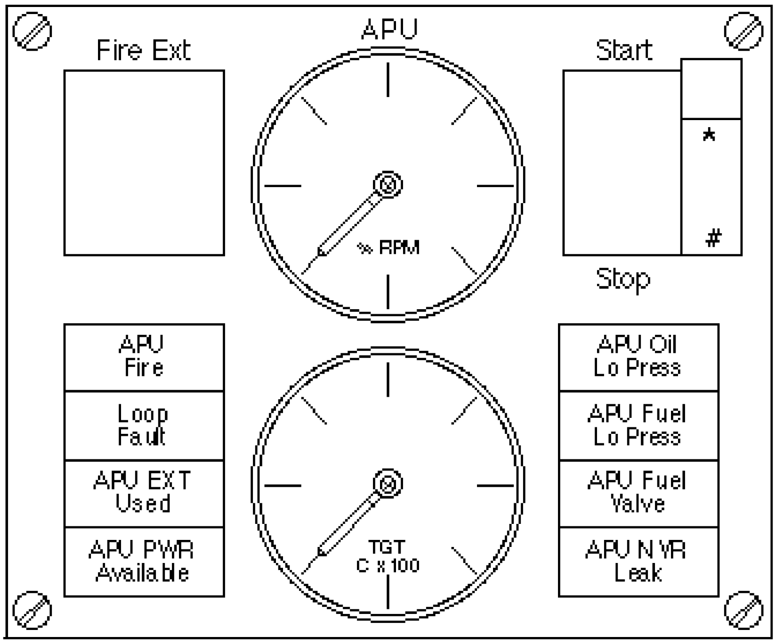
gearbox which is integral with the APU or an adjacent secondary power module. Some higher power APUs use a separate free turbine output drive configuration.

7. **Services Provided**. The APU is able to power all of the ancillary services required whilst the aircraft is on the ground. Air for cockpit and cabin air-conditioning and for main engine starting is bled from the compressor, whilst the required combination of hydraulic and pneumatic pumps and electrical alternators are driven through the APU accessory drive arrangement or, as depicted at Fig 3 above, through the complete secondary power system module. To minimize logistical costs and to provide maximum flexibility, the APU accessory gearbox is normally fitted with the same types of electrical generators and hydraulic and pneumatic pumps as those fitted to the main secondary power system. In most aircraft, the APU is situated in a fireproof enclosure in the tail cone or rear fuselage.

8. **Airborne Auxiliary Power Units**. Certain configurations of APU may be used during flight to augment secondary power sources or to provide power during an emergency. Such equipments are termed Airborne Auxiliary Power Units (AAPUs). The air bleed output available from a typical AAPU falls off rapidly with increasing altitude. Therefore their use for engine re-starting is not usually possible above 25,000 feet or even less, whilst that for cabin air-conditioning and pressurization is often limited to ground level only.

9. **Control of APUs**. The panel shown in Fig 4 is typical of an APU cockpit control. After manual selection of the master switch and the start/stop command button, the APU commences to operate within set parameters, requiring only the aircraft battery electrical supply for starting and fuel drawn from the main aircraft system. All operations are automatically controlled by the APU integrated control system. The unit provides start and close-down sequencing and governs the running RPM at a preset constant figure. It acts to close down the APU in the event of a malfunction such as low power, over-speed, overheating, loss of oil pressure or a fire warning. The APU fire extinguisher system is similar to that used for the main engines.

4-11 Fig 4 Typical APU Cockpit Control Panel



Emergency Power

10. The total dependence of inherently unstable (active control) aircraft on the uninterrupted operation of their Automatic Flying Control Systems (AFCSs) and Powered Flying Control Units (PFCUs) demands that emergency power be available effectively instantaneously in the event of loss of main secondary power sources. EPUs capable of developing full power within 2 seconds of initiation are therefore needed in such aircraft. A similar if less urgent requirement exists for emergency power in more conventional aircraft. A less rigourous specification may be applied to such EPUs or a variety of alternative power sources be used to satisfy the requirement.

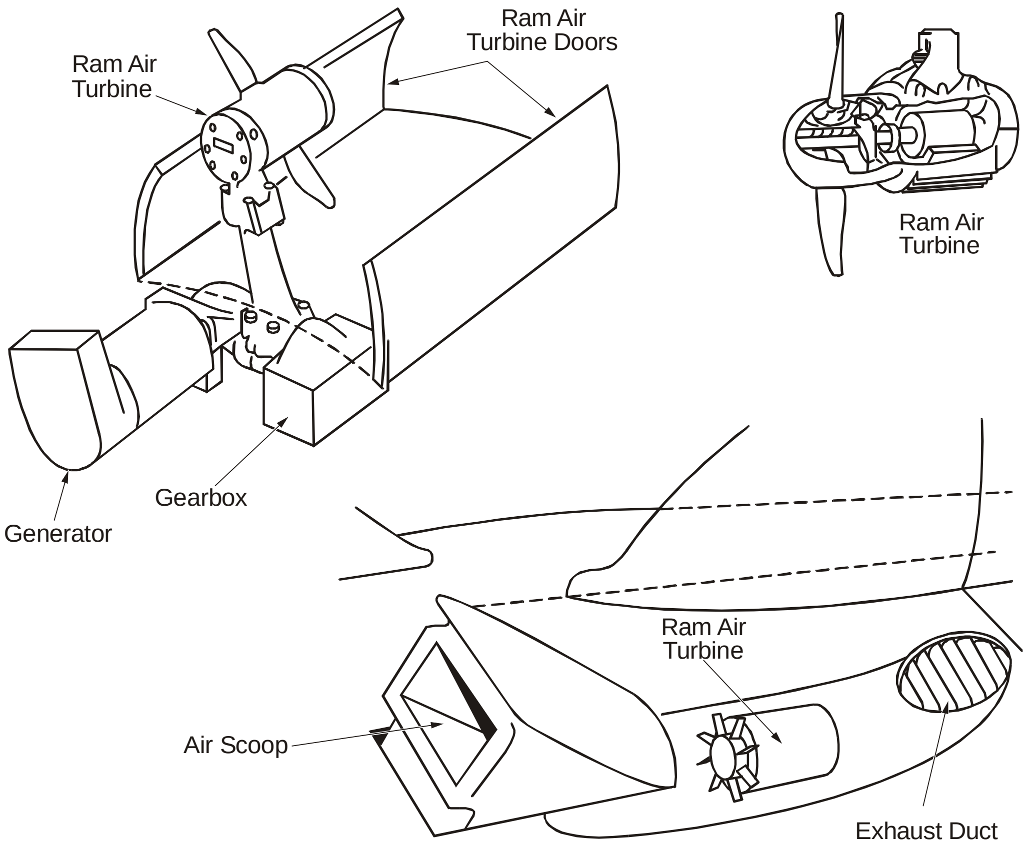
11. **Emergency Power Units**. Current rapid reaction EPUs are self contained modules consisting of a small mono-fuel powered gas turbine driving the necessary essential secondary power generators. Units under development include combined APU/EPU modules powered by gas turbines able to burn a mono-fuel to satisfy the rapid reaction criterion and then to revert to a conventional air/fuel mixture once they are running or when they are used as an APU. Less demanding requirements for emergency power are frequently met by the use of AAPUs.

Alternative Sources of Emergency Power

12. **Aircraft Batteries**. Many smaller aircraft are reliant upon their internal batteries for emergency secondary power. To conserve this finite power source, non-essential electrical loads must usually be shed by deliberate manual selection or by automatic load shedding systems. Such aircraft are normally equipped with manual flying controls or PFCUs with a manual reversion facility, and their undercarriage systems include an alternative system of lowering. Therefore, hydraulic power can be dispensed with for the period of the emergency.

13. **Ram Air Turbines**. The Ram Air Turbine (RAT) is a rapid response emergency secondary power source, the operation of which relies upon aircraft forward speed. Fig 5 shows the configuration of a typical RAT installation. The air turbine assembly is lowered into the air stream either upon manual selection or upon automatic sensing of engine or main secondary power failure. Once exposed to the air stream, the unit will spin up to operating speed within 2 to 4 seconds. However, although it will provide power for as long as the aircraft remains airborne, its power output will fall with airspeed.

4-11 Fig 5 Typical RAT Installation



Ground Power Units

14. The generic term GPU covers a wide range of equipments from the simple, towed trolley-accumulator providing limited DC power, to the multi-purpose unit providing hydraulic and electrical power and engine-start air from a single, self-propelled vehicle. The major advantage of a GPU is that, since it is not part of the aircraft, no weight penalty need be imposed on its construction. It can therefore be built to produce large power outputs, using economical electrical or diesel engine power, and it can be constructed robustly for minimum maintenance. Subject to suitable arrangements for cooling and the extraction of exhaust gases, mobile GPUs can also be used inside hangars or aircraft shelters. Access to an aircraft being prepared for flight or undergoing servicing is always at a premium, and this factor has a major influence on the size and configuration of GPUs, and established maintenance hangars are designed to provide all such services as part of the fixed installations of the building. The services can then be ducted as necessary to the aircraft from permanently mounted power units sited

well clear of the work area. Similar arrangements are also made in hardened aircraft shelters (HASs) and on the aircraft servicing platforms (ASPs) of major fixed operating bases.

CHAPTER 12 - ENGINE STARTER SYSTEMS

Principles

1. All turbine and piston engines require starter systems which are able to accelerate the engine from rest to a speed at which stable (self-sustained) operation is achieved and from which the engine can produce usable power. The basic components of a starter system are:

a. A motor to impart sufficient force to overcome the inertia and friction of the rotating assembly of the engine and its ancillary equipment, and to accelerate it to self sustaining speed within an acceptable operational period.

b. A fuel system able to introduce an initial charge of fuel/air mixture into the engine, appropriately metered for combustion to commence at the ambient temperature of the engine.

c. An ignition system able to provide a means of igniting the initial charge of fuel/air mixture.

d. A control system to programme the start sequence and to prevent design parameters (particularly speed and temperature upper limits) being exceeded during this initial, unstable stage of engine operation.

Basic Components of Starter Systems

2. Although the starting procedure for all engines is similar, many different types of starter system are used. In all cases reliable operation is the prime requirement. However, other factors such as speed of operation, independence of external support equipment, overall cost effectiveness and quietness of operation (particularly in passenger aircraft) must be balanced for each application.

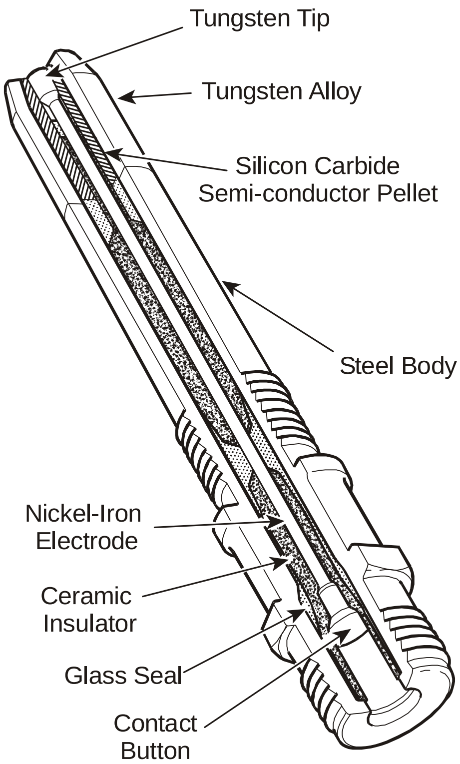
3. **Motive Power**. The motor or engine used for starting the main engine must develop very high power and transmit it to the rotating assembly of the engine in a manner which provides smooth acceleration. Some starter motors convert electrical energy, others use the potential energy of high or low pressure air or hydraulic systems. Starter engines use solid or liquid fuels to produce high pressure gases which are subsequently used to turn the main engine. Normally, power is provided to the starter motor from the aircraft auxiliary power unit (APU) or internal batteries, although external sources may be used as alternatives. Starter engines use on-board fuel supplies. To achieve a net weight saving, commercial aircraft which always operate from large, well equipped airfields are usually equipped with light weight starter systems for which external power sources are essential.

4. **Fuel Control**. The simplest starter systems rely upon manual control of the initial fuel/air mixture. However, in most cases the mixture is programmed automatically by the engine fuel control unit.

5. **Ignition**. The spark ignition systems of piston engines usually require only minor adjustment of the ignition timing to achieve combustion during the start cycle. This is invariably achieved automatically either mechanically or by operation of the electronic engine management system. Gas turbine engines use high energy electrical ignition systems to initiate and establish combustion during the start. Once stable engine running is achieved, combustion of the fuel/air mixture is self perpetuating, and the ignition system is switched off automatically.

6 . **Igniter Units**. Gas turbine high energy ignition systems are always duplicated to ensure reliability. The systems comprise a transistorized ignition unit feeding power to igniter plugs inside the engine combustion chambers. Fig 1 shows the construction of an igniter plug. Each ignition unit receives a low voltage supply controlled by the starter system. The electrical energy is stored in the unit until, at a predetermined value, it dissipates as a high energy discharge across the face of the semi-conductor in the igniter plug.

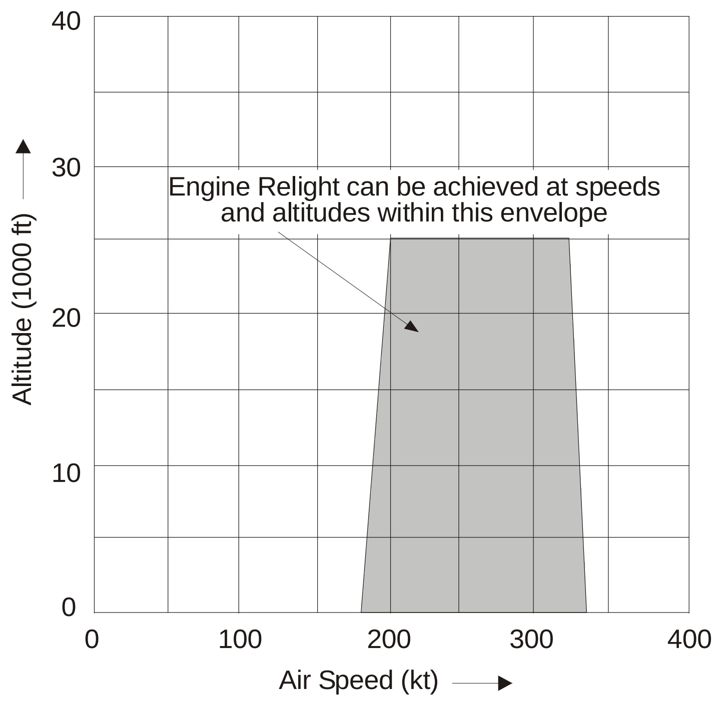
4-12 Fig 1 Igniter Plug



7. **Relight Systems**. In adverse engine operating conditions, it is possible for combustion to be interrupted or to break down completely. For this reason, the electrical igniter system is designed to be used during flight as well as during engine starting, either to act as a precaution against such flame-outs or to achieve relight after a flame-out has occurred. Ignition units are designed to give outputs appropriate to these differing requirements. A high output, typically 12 joules, is necessary during initial starting and to ensure that a satisfactory relight is achieved at high altitude. However, for continuous precautionary operation during flight, a low output of 3 to 6 joules is adequate and ensures longer life and higher reliability of the unit. The starter and relight control circuits automatically ensure that the appropriate level of power is provided.

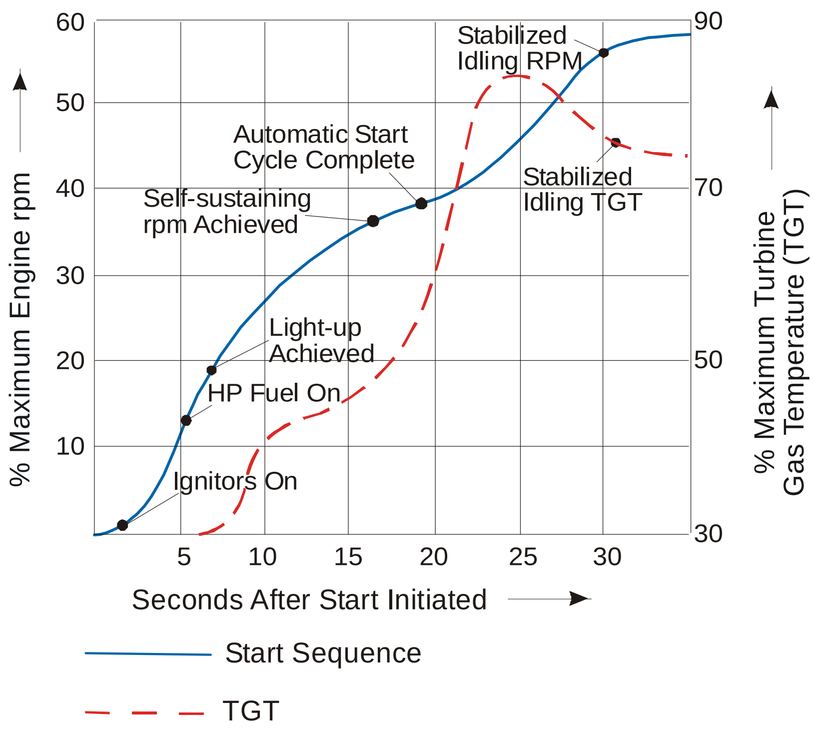
8. **Relight Envelope**. The ability to relight an engine after flame-out will vary according to the altitude and forward speed of the aircraft. A typical relight envelope showing the flight conditions under which a satisfactory relight can be achieved is at Fig 2. Within the limits of the envelope, airflow through the engine will be sufficient to maintain the rotating assembly at a speed satisfactory for combustion to be re-established. All that is required therefore, provided that a fuel supply is available, is operation of the ignition system by selection of the 'Relight' control.

4-12 Fig 2 Flight Relight Envelope



9. **Sequence Controller**. The simplest starter systems rely upon manual control of the start sequence. However, to achieve consistency, and to avoid engine parameters being exceeded, most aircraft are equipped with automatic or semi-automatic start sequence controllers. A typical start sequence is shown at Fig 3. After crew initiation of the start, the controller will automatically run through the start cycle allowing the crew to monitor critical engine conditions and to resume full control of the engine once it has reached stable running.

4-12 Fig 3 Start Sequence for a Gas Turbine Engine

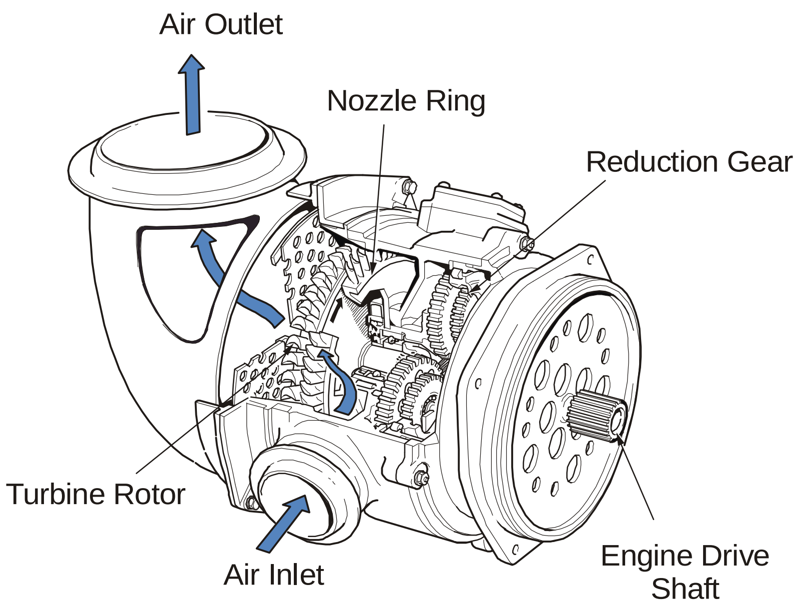


Main Types of Starter

10. **Electrical**. Electric motors developing sufficient power to accelerate the rotating assembly of a turbine engine are large and heavy in comparison with other types of starter. However, such motors are simple and comparatively cheap to produce and maintain, and electrical power is easy to transmit and control. It is available from internal batteries or APUs and from external power sources. For these reasons, electric starters are in wide spread use. If the starter motor can be configured to perform the dual role of driving the engine for starting and subsequently being driven by the engine to generate electrical power for the aircraft systems, significant advantages of low net weight, simplicity and low cost are available. The so called starter/generator is often used for small turbine engines, APUs and piston engines. Its drive shaft is permanently coupled to the engine, whereas single-role electrical starter motors must be connected to the engine through a clutch mechanism which engages at commencement of the start cycle and disengages as the engine reaches self-sustaining speed.

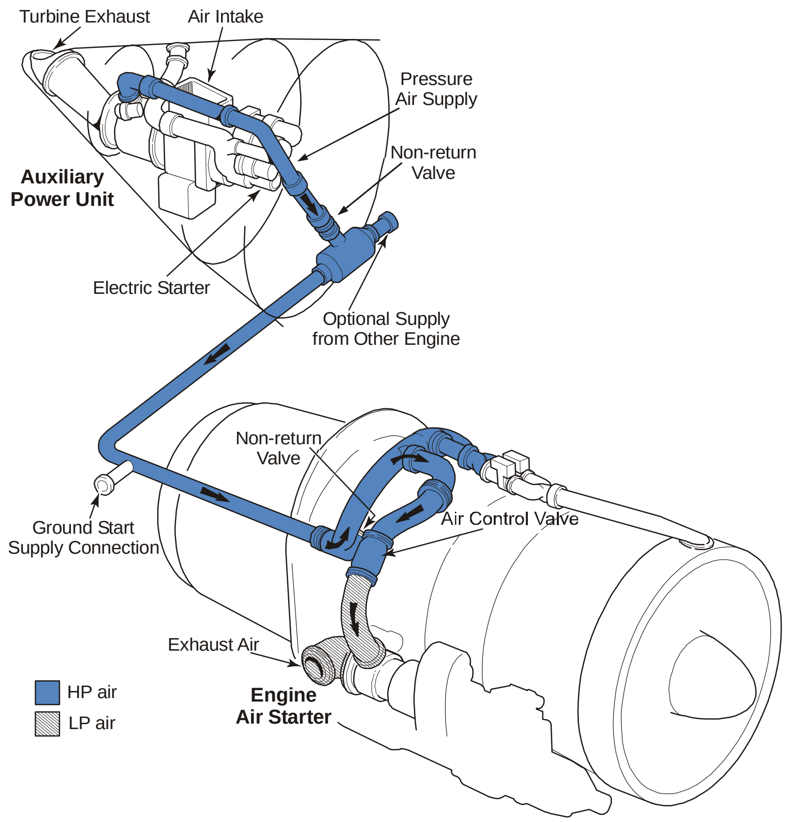
11. **Air Turbine (Low Pressure)**. Low pressure air starting is widely used for both military and commercial aircraft engines. The air turbine starter is simple and has relatively low airborne weight. A typical air turbine starter is shown below in Fig 4.

4-12 Fig 4 Typical Air Turbine Starter Motor



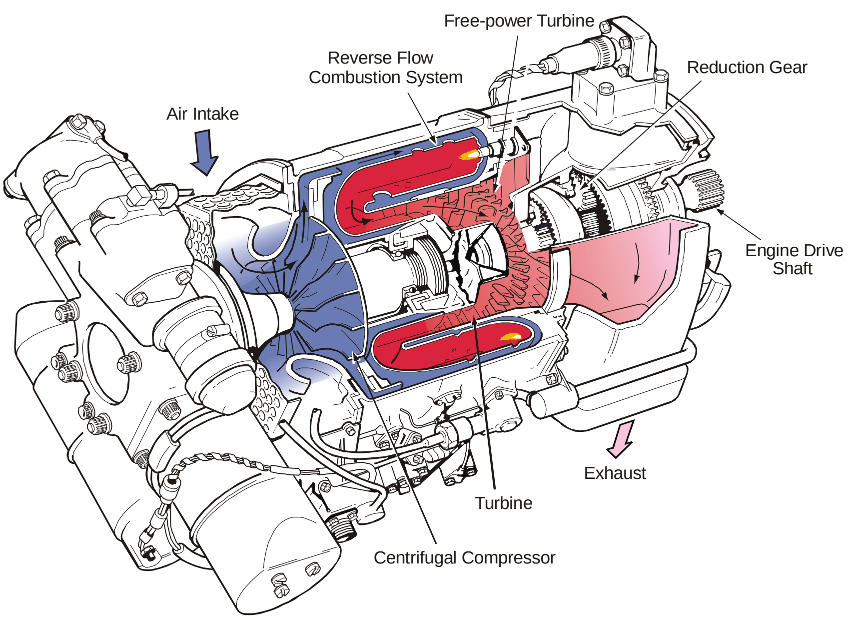
The air starter motor consists of a turbine which transmits power to the engine rotating assembly through reduction gearing and a clutch. The starter turbine is rotated by compressed air from an external ground supply, the aircraft APU or dedicated gas turbine air producer. Many aircraft are configured so that compressed air can be ducted from a running engine and used to start another. The air supply is controlled by electrically actuated valves which open when the engine start cycle is initiated. The air supply is automatically closed and the starter turbine clutch disengaged once the main engine reaches self-sustaining speed. A typical air turbine starter system configured to use air from an APU, another engine or from an external source is shown at Fig 5.

4-12 Fig 5 Typical Air Turbine Starter System



12. **Gas Turbine Starter**. The gas turbine starter is a small, compact engine. It usually comprises a centrifugal compressor driven by an axial power turbine, a reverse flow combustion system (to reduce overall length of the unit) and a free turbine driving the main engine starter drive shaft. The drive shaft is coupled to the main engine through reduction gearing and a clutch. A typical example is shown at Fig 6. The gas turbine starter engine is very similar in power output and size to the APU engine (see Volume 4, Chapter 11), and it is similarly used to provide services other than engine starting.

4-12 Fig 6 Gas Turbine Starter



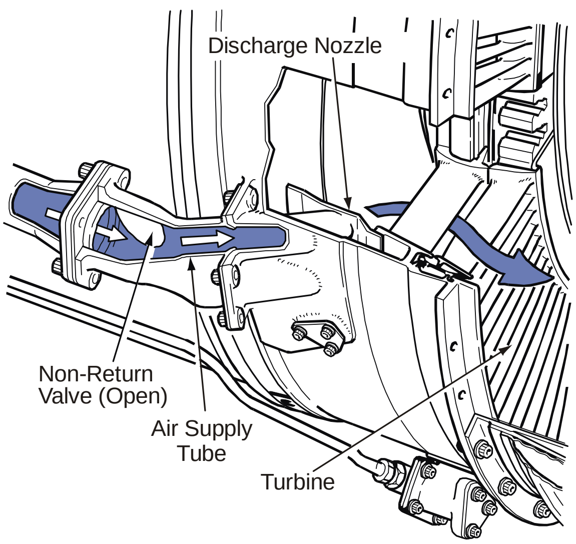
13. **Hydraulic**. Hydraulic power is sometimes used for starting small engines and APUs. In a typical installation, a hydraulic pump which can also be driven by hydraulic power to act as a motor is used. Other applications may use a separate hydraulic motor. Power from the motor is applied to the engine through reduction gearing and, in the case of the pure motor, through a clutch. The start sequence is automatically controlled by an electrical circuit which operateshydraulic valves. In the case of the motor/pump unit, the sequencing valves operate to allow the unit to act as a pump. The motor/pump offers similar advantages to those of an electrical starter/generator. Hydraulic power may be supplied from external sources, the APU or from a hydraulic accumulator in the aircraft system. Pressure is stored in the accumulator during previous engine running or by operation of an internal hand pump. A particular advantage of hydraulic starting for an APU is that the system can function by manual operation of the hand pump and with minimal internal electrical power. Therefore, a tactical aircraft can be started after long periods on the ground remote from support.

Miscellaneous Starters

14. Changes in operational requirements and advances in the appropriate technologies permitted considerable rationalization of aircraft systems to be achieved during the 1970s. However, many 1970 aircraft remain in service equipped with obsolescent systems, and brief descriptions of such starter systems are included in the following paragraphs.

15. **Air Impingement (High Pressure)**. The high pressure air impingement starter system does not use a starter motor as such but relies upon direct impingement of large volumes of high pressure air acting on the engine turbine blades as means of rotating the engine. A typical system is shown at Fig 7. The air may be provided from an external source, an APU or from a running engine. The benefits of the type are low weight and simplicity. However, its use is limited by its requirement for large volumes of high pressure air delivered from complex ground support units.

4-12 Fig 7 Air Impingement Starting



16. **Cartridge (Turbine Engine)**. The turbine cartridge starter provides a simple, light self-contained starter system. The starter is basically a small impulse turbine powered by high pressure gases released by burning cordite in the cartridge. It usually has a magazine of three cartridges each large enough for one engine start. The turbine rotates the main engine through reduction gearing and a clutch. This starter offers the advantage of low weight, but it suffers the disadvantages of providing a limited duration pulse of power per start and a limited number of starts (usually three) before replenishment. Also, its use results in the complication of ground logistic support of transporting and storing cordite filled cartridges.

17. **Liquid Fuel**. The liquid fuel starter is similar in principle to a cartridge starter. However, its gas source is derived from burning a liquid rather than a solid mono-fuel. The fuel is usually iso-propyl-nitrate (AVPIN). The starter develops high power, and this enables rapid engine starts to be achieved. The system is self-contained and consists of a fuel tank, a combustion chamber and the power turbine attached to the main engine through reduction gearing and a clutch. During a start, fuel is pumped from the tank and is ignited in the combustion chamber. The gases generated are then directed into its turbine. Although simple in principle and overcoming the '3 shot' disadvantage of the cartridge starter, the liquid fuel starter was comparatively unreliable and prone to catching fire. Its use was complicated by the extreme caution necessary in handling and storing the highly volatile liquid fuel.