AIRCRAFT SYSTEMS

(R15A2115)

COURSE FILE

III B. Tech II Semester

(2018-2019)

Prepared By

Ms. L Sushma, Assoc. Prof

Department of Aeronautical Engineering

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

(Autonomous Institution – UGC, Govt. of India)

Affiliated to JNTU, Hyderabad, Approved by AICTE - Accredited by NBA & NAAC – 'A' Grade - ISO 9001:2015 Certified)
Maisammaguda, Dhulapally (Post Via. Kompally), Secunderabad – 500100, Telangana State, India.

MRCET VISION

- To become a model institution in the fields of Engineering, Technology and Management.
- To have a perfect synchronization of the ideologies of MRCET with challenging demands of International Pioneering Organizations.

MRCET MISSION

To establish a pedestal for the integral innovation, team spirit, originality and competence in the students, expose them to face the global challenges and become pioneers of Indian vision of modern society.

MRCET QUALITY POLICY.

- To pursue continual improvement of teaching learning process of Undergraduate and Post Graduate programs in Engineering & Management vigorously.
- To provide state of art infrastructure and expertise to impart the quality education.

PROGRAM OUTCOMES
(PO's)

Engineering Graduates will be able to:

1. **Engineering knowledge**: Apply the knowledge of mathematics, science, engineering fundamentals, and an engineering specialization to the solution of complex engineering problems.
2. **Problem analysis**: Identify, formulate, review research literature, and analyze complex engineering problems reaching substantiated conclusions using first principles of mathematics, natural sciences, and engineering sciences.
3. **Design / development of solutions**: Design solutions for complex engineering problems and design system components or processes that meet the specified needs with appropriate consideration for the public health and safety, and the cultural, societal, and environmental considerations.
4. **Conduct investigations of complex problems**: Use research-based knowledge and research methods including design of experiments, analysis and interpretation of data, and synthesis of the information to provide valid conclusions.
5. **Modern tool usage**: Create, select, and apply appropriate techniques, resources, and modern engineering and IT tools including prediction and modeling to complex engineering activities with an understanding of the limitations.
6. **The engineer and society**: Apply reasoning informed by the contextual knowledge to assess societal, health, safety, legal and cultural issues and the consequent responsibilities relevant to the professional engineering practice.
7. **Environment and sustainability**: Understand the impact of the professional engineering solutions in societal and environmental contexts, and demonstrate the knowledge of, and need for sustainable development.
8. **Ethics**: Apply ethical principles and commit to professional ethics and responsibilities and norms of the engineering practice.
9. **Individual and team work**: Function effectively as an individual, and as a member or leader in diverse teams, and in multidisciplinary settings.
10. **Communication**: Communicate effectively on complex engineering activities with the engineering community and with society at large, such as, being able to comprehend and write effective reports and design documentation, make effective presentations, and give and receive clear instructions.
11. **Project management and finance**: Demonstrate knowledge and understanding of the engineering and management principles and apply these to one's own work, as a member and leader in a team, to manage projects and in multi disciplinary environments.
12. **Life- long learning**: Recognize the need for, and have the preparation and ability to engage in independent and life-long learning in the broadest context of technological change.

DEPARTMENT OF AERONAUTICAL ENGINEERING

VISION

Department of Aeronautical Engineering aims to be indispensable source in Aeronautical Engineering which has a zeal to provide the value driven platform for the students to acquire knowledge and empower themselves to shoulder higher responsibility in building a strong nation.

MISSION

The primary mission of the department is to promote engineering education and research. To strive consistently to provide quality education, keeping in pace with time and technology. Department passions to integrate the intellectual, spiritual, ethical and social development of the students for shaping them into dynamic engineers.

QUALITY POLICY STATEMENT

Impart up-to-date knowledge to the students in Aeronautical area to make them quality engineers. Make the students experience the applications on quality equipment and tools. Provide systems, resources and training opportunities to achieve continuous improvement. Maintain global standards in education, training and services.

PROGRAM EDUCATIONAL OBJECTIVES – Aeronautical Engineering

1. **PEO1 (PROFESSIONALISM & CITIZENSHIP):** To create and sustain a community of learning in which students acquire knowledge and learn to apply it professionally with due consideration for ethical, ecological and economic issues.
2. **PEO2 (TECHNICAL ACCOMPLISHMENTS):** To provide knowledge based services to satisfy the needs of society and the industry by providing hands on experience in various technologies in core field.
3. **PEO3 (INVENTION, INNOVATION AND CREATIVITY):** To make the students to design, experiment, analyze, and interpret in the core field with the help of other multi disciplinary concepts wherever applicable.
4. **PEO4 (PROFESSIONAL DEVELOPMENT):** To educate the students to disseminate research findings with good soft skills and become a successful entrepreneur.
5. **PEO5 (HUMAN RESOURCE DEVELOPMENT):** To graduate the students in building national capabilities in technology, education and research

PROGRAM SPECIFIC OUTCOMES – Aeronautical Engineering

1. To mould students to become a professional with all necessary skills, personality and sound knowledge in basic and advance technological areas.
2. To promote understanding of concepts and develop ability in design manufacture and maintenance of aircraft, aerospace vehicles and associated equipment and develop application capability of the concepts sciences to engineering design and processes.
3. Understanding the current scenario in the field of aeronautics and acquire ability to apply knowledge of engineering, science and mathematics to design and conduct experiments in the field of Aeronautical Engineering.
4. To develop leadership skills in our students necessary to shape the social, intellectual, business and technical worlds.

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

III Year B. Tech, ANE-II Sem

L	T/P/D	C
4	1/-/-	3

(R15A2114) AIRCRAFT SYSTEMS

OBJECTIVES:

- To impart knowledge of the hydraulic and pneumatic systems components and types of instruments and its operation to the students

UNIT - I

HYDRAULIC & PNEUMATIC AND LANDING GEAR SYSTEMS

Study of typical workable system – Components – Hydraulic system controllers –Modes of operation Pneumatic systems – Advantages – Working principles – Typical air pressure system – Brake system – Typical pneumatic power system – Components Landing gear systems – Classification – Shock absorbers – Retractive mechanism.

UNIT – II

AIRPLANE CONTROL & MODERN CONTROL SYSTEMS
SYSTEMS

Conventional systems – Power assisted and fully powered flight controls – Power actuated systems – Engine control systems – Push pull rod system, Flexible push full rod system – Components. Digital fly by wire systems – Auto pilot system active control technology, Communication and Navigation systems – Instrument landing systems, VOR – CCV, Case studies.

UNIT - III
ENGINE SYSTEMS STARTING & IGNITION SYSTEMS

Fuel system for piston and jet engines - Components of multi engines – Lubricating systems for piston and jet engines. Starting and Ignition systems – Typical examples for piston and jet engines

UNIT - IV
AIR CONDITIONING AND PRESSURIZING SYSTEMS

Basic air cycle systems – Vapor cycle systems, Boost – strap air cycle system – Evaporative vapor cycle systems – Evaporative air cycle systems – Oxygen systems – Fire protection systems, De-icing and anti-icing systems.

UNIT - V
ELECTRICAL SYSTEMS

Electrical loads in aircraft. Electrical power generation and control of AC and DC. Bus bars, power distribution of different voltages AC & DC. over/under load protection devices-speed and frequency protection devices. Electrical load measurement systems.

OUTCOMESSS

- The student should able to Know the operation of airplane control system, Engine system, Air conditioning and pressing system.
- Know the operation of air data Instruments system

TEXT BOOKS

1. McKinley, J.L., and Bent, R.D., Aircraft Maintenance & Repair, McGraw Hill, 1993.
Transportation, Federal Aviation Administration The English Book Store, New Delhi, 1995
2. General Handbooks of Airframe and Power Plant Mechanics, U.S.Dept. of

MALLA REDDY COLLEGE OF ENGINEERING AND TECHNOLOGY
DEPARTMENT OF AERONAUTICAL ENGINEERING
SESSION PLANNER

SUB: Avionics and Instrumentation **YEAR: IV YR** **SEMESTER: II SEM**

UNIT NO	TOPIC	No of Classes Planned
UNIT-1	**HYDRAULIC & PNEUMATIC AND LANDING GEAR SYSTEMS** Study of typical workable system – Components – Hydraulic system controllers –Modes of operation	2
	Pneumatic systems – Advantages – Working principles – Typical air pressure system – Brake system – Typical pneumatic power system – Components	3
	Landing gear systems – Classification – Shock absorbers – Retractive mechanism.	3
UNIT- II	Conventional systems – Power assisted and fully powered flight controls	3
	Power actuated systems	2
	Engine control systems – Push pull rod system, Flexible push full rod system – Components. Digital fly by wire systems	2
	Auto pilot system active control technology,	3
UNIT III	Fuel system for piston and jet engines - Components of multi engines	3
	Lubricating systems for piston and jet engines. Starting and Ignition systems	3
	Typical examples for piston and jet engines	3
UNIT-IV	Basic air cycle systems – Vapor cycle systems, Boost – strap air cycle system	3
	Evaporative vapor cycle systems ––	2
	Evaporative air cycle systems – Oxygen systems	3
	Fire protection systems, De-icing and anti-icing systems.	2
UNIT-V	Electrical loads in aircraft.	3
	Electrical power generation and control of AC and DC. Bus bars,	2
	power distribution of different voltages AC & DC. over/under	2
	load protection devices-speed and frequency protection devices. Electrical load measurement systems.	2
	Total	**57**

TEXT BOOKS:
1. Moir, I. and sea bridge, Aircraft Systems, AIAA Education series
2. 3. Moir, I. Sea bridge A. Military avionics systems, (Aerospace)

PREPARED BY: L.SUSHMA
Associate Professor **HOD**

AIRCRAFT SYSTEMS
MODEL PAPERS-1

Answer all the questions: 25M

1. Enumerate different types of aircraft systems. [3]
2. Explain about any one Mission system of transport aircraft. [2]
3. Explain about reverse bootstrap system. [2]
4. Explain about g-tolerance. [3]
5. Explain electrical loads of an aircraft. [2]
6. Explain about DC-link. [3]
7. Brief about accumulator working principle. [3]
8. Brief about Landing gear operation. [2]
9. List out the typical start sequence. [3]
10. List out the mode of operation of fuel systems. [2]

Answer any one from each section: 50M

1. Classify and discuss about the a/c systems [10]

   (or)

   Explain the integration of a/c system on process , information, and function levels with required sketches and tables. [10]
2. Explain in detail the electrical system components and working with a schematic diagram [10]

   (or)

   Explain in detail FBW system its components nomenclatures by taking any example.
3. Explain the components and working principle of a hydraulic system. [10]

   (or)

   Explain the brake management system with neat sketches. [10]
4. Explain about refrigeration systems and types with neat sketches [10]

   (or)

   Explain the process and types of passenger & crew evacuation/ejection system .
5. Explain in detail various modes of fuel system [10]

   (or)

   Explain the engine control evolution, fuel flow control, control system parameter with some examples. [10]

AIRCRAFT SYSTEMS
MODEL PAPERS-2

Answer all the questions: 25M

1. Enumerate levels of integration of aircraft systems at component levels. [3]
2. Explain about any one Mission system of fighter aircraft. [2]
3. Brief about the bleed air [3]
4. Brief about air cycle refrigeration. [2]
5. Brief about primary and secondary control systems of a/c. [3]
6. Brief about a cyclo converter working priciple. [2]
7. Enumerate the basic components of hydraulic system. [2]
8. Discuss about reservoir, pump and its working principle. [3]
9. List out the main components of fuel system. [2]
10. Explain about functions of sensors in engine control system. [3]

Answer any one from each section: 50M

1. Explain the purpose, location brief description, aspects/safety of navigation & cockpit displays systems with sketches and tables showing 3 examples each. [10]

   (or)

   Explain the drivers in the product of environment operating environment. [10]
2. Explain the working of a pneumatic system with a neat sketch [10]

   (or)

   Explain about ram air cooling and fuel cooling with sketches. [10]
3. Explain the flight primary and secondary flight controls for civil and fighter a/c. [10]

   (or)

   Explain the terms

   a. Cyclo-converter b. DC-link c. Generator d. VSCF [3,2,3,2]

4. Explain the terms with sketches

   a. Accumulator b. Relief Valve c. Pump d. reservoir [3,2,3,2]

   (or)

   Explain what do you mean by redundancy, types and applications. [10]
5. Explain the factors involved in engine starting control. [10]

   (or)

   Explain the terms

   a. Booster pumps b. Transfer valves c.NRV d. fuel gauging probes [3,2,3,2]

AIRCRAFT SYSTEMS
MODEL PAPERS-3

Answer all the questions: 25M

1. Explain about ADF working with a sketch. [3]
2. Explain about ILS & MLS [2]
3. Explain about Cyclo converter. [2]
4. Explain about Load protection. [3]
5. Explain anti-ice technique. [2]
6. Explain molecular sieve oxygen concentrator. [3]
7. Explain the functions of reservoir. [3]
8. Brief about importance of redundancy [2]
9. List out the typical start sequence. [3]
10. List out the mode of operation of fuel systems. [2]

Answer any one from each section: 50M

1. Discuss on:
   Airframe Systems   Vehicle Systems   Avionics Systems   Mission Systems [2, 3 ,2 ,3]
   (or)
   What are the level of integration involved in aircraft production. [10]
2. Explain the following
   Primary & Secondary flight control [6]
   Control linkages [4]
   (or)
3. Explain the terms:
   Power conversion [4]
   VSCS [3]
   DC link. [3]
4. Explain the basic principle on which hydraulic system works? What are the main components of hydraulic system? [10]
   (or)
   Explain the landing gear system operation with neat sketches. [10]
5. Explain about air refrigeration systems and types with neat sketches [10]
   (or)
   Explain the process and types of passenger & crew evacuation/ejection system .
6. Explain in detail control of fuel flow and air flow [10]
   (or)

Explain the engine control evolution, fuel flow control, control system parameter with some examples. [10]

AIRCRAFT SYSTEMS
MODEL PAPERS-4

Answer all the questions: 25M

1. Enumerate levels of integration of aircraft systems at system levels. [3]
2. Explain about any one avionics system of fighter aircraft. [2]
3. Brief about the fuel cooling [3]
4. Brief about Pitot static system. [2]
5. Brief about how actuation is important in control surface. [3]
6. Brief about a cyclo converter working priciple. [2]
7. Explain the types of pumps used in a/c. [2]
8. Discuss about reservoir, pump and its working principle. [3]
9. List out the main components of fuel system. [2]
10. Explain about functions of sensors in engine control system. [3]

Answer any one from each section: 50M

1. Explain how the operating environment conditions of an aircraft are maintained. [10]

   (or)

   Discuss the aspects of safety and integrity of aircraft for any 4 systems. [10]
2. Explain the working of a cabin pressurization system with a neat sketch [10]

   (or)

   a. Explain about g-tolerance and protection with sketches. [3]

   b. Explain working and principle of boots trap and reverse bootstrap [7]
3. Explain the electrical load management systems. [10]

   (or)

   Explain the terms

   a. Load protection b. DC-link c. Generator d. Redundancy [3,2,3,2]

4. Explain in detail the flight control actuation importance and need

   (or)

   Explain what do you mean by redundancy, types and applications. [10]
5. Explain the full authority and limited authority control systems. [10]

   (or)

   Explain the terms

   a. Exhaust gas flow b. Fuel flow control [5,5]

AIRCRAFT SYSTEMS
MODEL PAPERS-5

Answer all the questions: 25M

1. Explain about integration. [3]
2. Explain about any one vehicle system of transport aircraft. [2]
3. Explain about VSCS. [2]
4. Explain about spoiler and flap. [3]
5. Explain reservoir and pump functions. [2]
6. Explain the importance of redundancy. [3]
7. Brief about fuel cooling working principle. [3]
8. Brief about cabin pressurization operation. [2]
9. Explain the control of fuel. [3]
10. Explain mode of operations of fuel systems. [2]

Answer any one from each section: 50M

1. Explain about the operating environment conditions [10]

   (or)

   Explain the integration of a/c system on system and function levels with required sketches and tables. [10]
2. Describe electrical system components and working principle with a schematic diagram [10]

   (or)

   Explain in detail FBW system its components nomenclatures by taking any example.
3. Discuss the working principle of a hydraulic system with a neat sketch. [10]

   (or)

   Explain the brake management system with neat sketches. [10]
4. Explain about engine as a source of high pressure operating techniques [10]

   (or)

   Explain the process and types of passenger & crew evacuation/ejection system .
5. Explain in detail various modes of fuel system [10]

   (or)

   Explain the engine control evolution, fuel flow control, control system parameter with some examples. [10]

UNIT 1: Hydraulic Systems Pneumatic and Landing Gear Systems

Hydraulics is a division of the science of fluid mechanics that includes the study of liquids and their physical characteristics, both at rest and in motion. The type of hydraulics applied to aircraft and other aerospace-vehicle systems is called power hydraulics because it involves the application of power through the medium of hydraulics.

Hydraulic Terms It is necessary to understand the exact meaning of hydraulic terms in order to understand hydraulic principles and their application to hydraulic systems. These terms are defined as follows.

- Area: is a measurement of a surface. In aircraft hydraulics the technician is concerned with the areas of piston heads. Knowing this area, the amount of force required to actuate a mechanism can be determined.
- Force: Is the amount of push, pull, or twist on an object. The force in a hydraulic system is derived from the pressure acting on the area of a piston head. To measure the force of hydraulics, we must be able to determine force per unit area. This is called pressure and is measured in pounds per square inch (psi) or kilopascals (kPa).
- Stroke: Stroke (length) is a measurement of distance & It represents the distance a piston moves in a cylinder.
- Volume: Volume (displacement) is a mesure of quantity, which represents the amount of fluid contained in a reservoir ordisplaced by a pump or actuating cylinder.
- Fluid: is any substance that is liquid or gaseous in form. A liquid is a fluid whose particles form a definite volume.The term
- Hydraulic fluid: is used in this text as the common name for the fluid used in aircraft hydraulic systems and devices.

Liquids are regarded as being incompressible. This means that the volume of a given quantity of a liquid will remain constant even though it is subjected to high pressure. Because of this characteristic, it is easy to determine the volume of hydraulic fluid required to move a piston through its operating range. The volume of the cylinder through which the piston moves is equal to the area of the piston head multiplied by the length of the cylinder. The area of the piston head is determined by the formula

Hydraulic fluids and other liquids expand as temperature increases; therefore, safeguards must be provided in hydraulic systems to allow for the expansion and contraction of fluid as temperature changes. Liquid seeks its own level

A basic principle of hydraulics is expressed in Pascal's law.

Pascal’s Law: Pascal law states that pressure applied to any part of a confined liquid is transmitted with undiminished intensity to every other part.

UNIT 1: Hydraulic Systems Pneumatic and Landing Gear Systems

Viscosity: When liquids are in motion, certain dynamic characteristics must be taken into consideration. One of the principal factors in liquid motion is friction.

Friction exists between the molecules of the liquid and between the liquid and the pipe through which it is flowing. The effects of friction increase as the velocity of liquid flow increases.

One of the most important properties of any hyd. Fluid is its viscosity .Viscosity is internal resistance to flow. A Viscosity increases with temperature decreases. A satisfactory liquid for a given hyd. Sys must have enough body to give a good seal at pumps, valves, & piston But it must not be so thick that it offers resistance to flow, landing to power loss & higher operating temperature. Viscosity of liquid measured by viscoimeter & viscometer .

Types of Hydraulic Fluids:

There are 3 types of Hydraulic Fluids used in civil aircraft.

1]. Vegetable Base hyd. Fluid

2]. Mineral Base hyd. Fluid

3]. Phosphate ester -Base hyd. Fluid

1]. Vegetable Base hyd. Fluid- (MIL-H-7644):- Mixture of castor oil & alcohol .It is highly inflammable .It is dyed Blue in color ,Natural rubber seals are used with veg. base fluid . it is used in older a/c.

2]. Mineral Base hyd. Fluid- (MIL-H-5606) :-consist of a high quality petroleum oil.

They are used in many systems, especially where the fire hazard is comparatively low. Small aircraft that have hydraulic power systems for operating wheel brakes, flaps, and landing gear usually use mineral fluid .Mineral base fluids are less corrosive and less damaging to certain parts than other types of fluid.

3]. Phosphate ester-Base hyd. Fluid:- utilized in most transport category aircraft are very fire resistant. Although phosphate ester fluids are extremely fire resistant, they are not fireproof. Under certain conditions phosphate ester fluids will burn.

It is classified in 2 classes:

Class 1-Skydrol L-D: A clear purple low weight fluid It is used in large Jumbojet transport a/c whereweight has prime factor.

UNIT 1: Hydraulic Systems Pneumatic and Landing Gear Systems

Class 2- Skydrol 500-B: a clear purple lquid having good low temp. operating characteristics & lowcorrosive side effects .Skydrol is compatible with natural fibers & synthetic including nylon & polyester .

Hydraulic power is used to transferring power from small low energy movements in the cockpit to high energy demands in the aircraft. Hydraulic systems now have an important role to play in all modern aircraft, both military and civil aircraft.

The introduction of powered flying controls by using hydraulic power by which the pilot was able to move the control surfaces with every increasing speeds and demands with less effort.

The system consists of multiple pumps, accumulators to store energy, reservoir, pipelines and actuators and solenoid valves. The hydraulic system today remains a most effective source of power for both primary and secondary flying controls, and for undercarriage, braking and anti-skid systems.

Design Requirements

- The principal requirements are low weight, low volume, low initial cost, high reliability and low maintenance.
- The pipe diameters should be less and able to bend themselves to flexibility of installation and end fittings should not give any leakage,
- The use of oil as the working fluid provides a high degree of lubrication, and the system overloads can be withstood without damage.
- Within the limits of their structural strength, actuators can stall and in some cases actually reverse direction.
- The working Fluid should return to reservoir on removal of the overload

In addition the following are the some parameters needs to understand before designing hydraulic system.

• ***Pressure*** – What will be the primary pressure of the system? This will be determined by the appropriate standards and the technology of the system .

• ***Integrity*** – Is the system critical towards flight safety or can its loss or degradation be tolerated? This determines the number of independent sources of hydraulic power that must be provided, and determines the need for a reversionary source of power.

• ***Flow rate*** – What is the rate of the demand, in angular or linear motion per second, or in liters per second in order to achieve the desired action?

• ***Duty cycle*** – What is the ratio of demand for energy compared to quiescent conditions. This will be high for continuously variable demands such as primary flight control actuation on an unstable aircraft (throughout the flight), whereas it will be low for use as a source of energy for undercarriage lowering and retraction (twice per flight)

• ***Emergency or reversionary use*** – Are there any elements of the system that are intended to provide a source of power under emergency conditions for other power generation systems? An example of this is a hydraulic powered electrical generator. Is there a need for a source of power in the event of main engine loss to provide hydraulic power which will demand the use of reversionary devices?

• ***Heat load and dissipation*** – The amount of energy or heat load that the components of the system contribute to hydraulic fluid temperature .

Types of components required for designing hydraulic system

• A source of energy – engine, auxiliary power unit or ram air turbine

• A reservoir- Tank

• A filter to maintain clean hydraulic fluid

• A multiple redundant distribution system – pipes, valves, shut-off cocks

• Pressure and temperature sensors

• A mechanism for hydraulic oil cooling

• A means of exercising demand – actuators, motors, pumps

• A means of storing energy such as an accumulator

These requirements together with the type of aircraft, determine the design of a hydraulic system. When starting the design of any new hydraulic system the engineer must first determine the functions to be performed, and secondly he must assess their importance to flight safety. Thus lists of functions are Based on functional severity primary flight controls are critical to flight safety and no single failures can be allowed to prevent, or even momentarily interrupt their operation.

The secondary flight controls, are, flaps and slats, spoilers, airbrakes and stabilizer trim tabs. Other functions, commonly known as 'services' or 'utilities', may be considered expendable after a failure, or may needed to operate in just one direction after a positive emergency selection by the pilot.

Hydraulic Fluid Properties

The working fluid will be considered as a physical medium for transmitting power, and the conditions under which it is expected to work, for example maximum temperature and maximum flow rate are described. Safety regulations bring about some differences between military and civil aircraft fluids. The popular mineral based fluid in use are:

UNIT 1: Hydraulic Systems Pneumatic and Landing Gear Systems

• DTD 585 in the UK

• MIL-H-5606 in the USA

• AIR 320 in France

• H 515 NATO

This fluid has many advantages.

It is freely available throughout the world, reasonably priced, and has a low rate of change of viscosity with respect to temperature compared to other fluids. These fluids are not fireproof – there are certain combinations of fluid spray and hot surfaces which will allow them to ignite and burn. Industry standard tests are conducted to demonstrate a level of confidence that ignition or fire will not occur and the hydraulic system design is influenced by these test results.

Operating Fluid Pressure

Systems should maintain standard pressure of 3000 psi or 4000 psi. These have been chosen to keep weight to a minimum. Many studies have been undertaken by industry to raise the standard working pressure. Pressure targets have varied from 5000 psi to 8000 psi, and all resulting systems studies claim to show reduced system component mass and volume. Interestingly DTD 585 cannot be used above 5000 psi because of shear breakdown within the fluid. A detailed study would show that the optimum pressure will differ for every aircraft design. This is obviously impractical and would preclude the common use of well-proven components and test equipment.

Operating Fluid Temperature

With fast jet aircraft capable of sustained operation above Mach 1, there are advantages in operating the system at high temperatures, but this is limited by the fluid used. For many years the use of DTD 585 has limited temperatures to about 130°C, and components and seals have been qualified accordingly. The use of MIL-H-83282 has raised this limit to 200°C and many other fluids have been used from time to time.

A disadvantage to operating at high temperatures is that phosphate ester based fluids can degrade as a result of hydrolysis and oxidation. As temperature increases, so the viscosity of the fluid falls. At some point lubricity will be reduced to the extent that connected actuators and motors may be damaged

Functions as illustrated in Figure 1.1 may appear as:

- **Primary flight controls:**
    - ✓ Elevators
    - ✓ Rudders
    - ✓ Ailerons
    - ✓ Canards

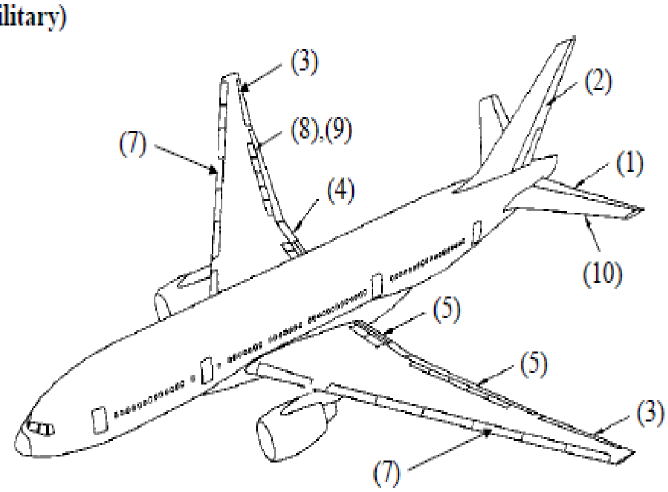
UNIT 1: Hydraulic Systems Pneumatic and Landing Gear Systems

- **Secondary flight controls:** Flaps
  - ✓ Slats
  - ✓ Spoilers
  - ✓ Airbrakes
- **Utility systems:**
  - ✓ Undercarriage – gear and doors
  - ✓ Wheelbrakes and anti-skid
  - ✓ Parking brake
  - ✓ Nosewheel steering
  - ✓ In-flight refuelling probe
  - ✓ Cargo doors
  - ✓ Loading ramp
  - ✓ Passenger stairs
  - ✓ Bomb bay doors
  - ✓ Gun purging scoop
  - ✓ Canopy Actuation

❑ **Primary Flight Controls:**
- -Elevators – (1)
- -All-moving tail surfaces (military)
- -Rudders – (2)
- -Ailerons – (3)
- -Flaperons – (4)
- -Canards

❑ **Secondary Flight Controls**
- -Flaps – (5)
- -Slats – (7)
- -Spoilers – (8)
- -Airbrakes – (9)
- -Stabilizer trim – (10)

❑ **Utilities**
- -Landing gear
- -Brakes
- -Gear steering
- -Aerial refueling probes (military)
- -Cargo doors
- -Loading ramp (military)
- -Passenger stairs



UNIT 1: Hydraulic Systems Pneumatic and Landing Gear Systems

Basic Hydraulic System: The main parts of the basic hydraulic system are as given as follows:

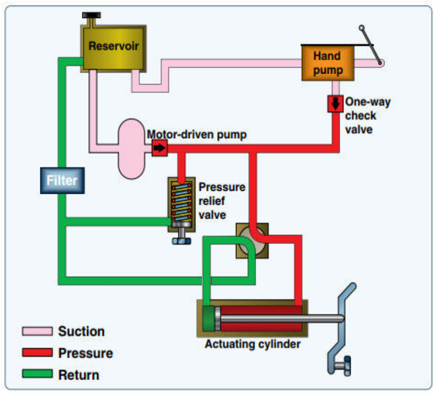


Figure 1.4: Basic Hydraulic Systems

Components

I. Accumulator:

The accumulator is a steel sphere divided into two chambers by a synthetic rubber diaphragm. The upper chamber contains fluid at system pressure, while the lower chamber is charged with nitrogen or air. Cylindrical types are also used in high-pressure hydraulic systems. Many aircraft have several accumulators in the hydraulic system. There may be a main system accumulator and an emergency system accumulator. There may also be auxiliary accumulators located in various sub-systems.

It may be found that the absolute maximum flow demand is of very short duration, involving very small volumes of oil at very high velocities. In this case sizing a pump to meet this demand may not be justified. An accumulator can be used to augment the flow available, but care must be taken. An accumulator contains a compressed gas cylinder, and the gas is used to provide energy to augment system pressure. Therefore, the fluid volume and pressure available will depend on the gas temperature. In a situation where the flow demanded will exceed the pump capabilities the system pressure is controlled by the accumulator, not the pump.

The function of an accumulator is to:

- Dampen pressure surges in the hydraulic system caused by actuation of a unit and the effort of the pump to maintain pressure at a preset level.
- Aid or supplement the power pump when several units are operating at once by supplying extra power from its accumulated, or stored, power.

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- Store power for the limited operation of a hydraulic unit when the pump is not operating.
- Supply fluid under pressure to compensate for small internal or external (not desired) leaks that would cause the system to cycle continuously by action of the pressure switches continually kicking in.

Types of Accumulators

There are two general types of accumulators used in aircraft hydraulic systems:

1. Cylindrical: Cylindrical accumulators consist of a cylinder and piston assembly. End caps are attached to both ends of the cylinder. The internal piston separates the fluid and air/nitrogen chambers. The end caps and piston are sealed with gaskets and packings to prevent external leakage around the end caps and internal leakage between the chambers. In one end cap, a hydraulic fitting is used to attach the fluid chamber to the hydraulic system. In the other end cap, a filler valve is installed to perform the same function as the filler valve installed in the spherical accumulator. [Figure 2]

Figure 2. Cylindrical accumulator

2. Spherical: The spherical-type accumulator is constructed in two halves that are fastened and threaded, or welded, together. Two threaded openings exist. The top port accepts fittings to connect to the pressurized hydraulic system to the accumulator. The bottom port is fitted with a gas servicing valve, such as a Schrader valve.

A synthetic rubber diaphragm, or bladder, is installed in the sphere to create two chambers. Pressurized hydraulic fluid occupies the upper chamber and nitrogen or air charges the lower chamber. A screen at the fluid pressure port keeps the diaphragm, or bladder, from extruding through the port when the lower chamber is charged and hydraulic fluid pressure is zero.

A rigid button or disc may also be attached to the diaphragm, or bladder, for this purpose. [Figure 1] The bladder is installed through a large opening in the bottom of the sphere and is secured with a threaded retainer plug. The gas servicing valve mounts into the retainer plug.

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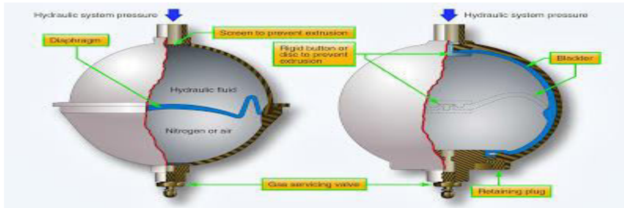

Figure 1. A spherical accumulator with diaphragm (left) and bladder (right).

Operation:

In operation, the compressed-air chamber is charged to a predetermined pressure that is somewhat lower than the system operating pressure. This initial charge is referred to as the accumulator preload. As an example of accumulator operation, let us assume that the cylindrical accumulator is designed for a preload of 1,300 psi in a 3,000-psi system.

When the initial charge of 1,300 psi is introduced into the unit, hydraulic system pressure is zero. As air pressure is applied through a gas servicing valve, it moves the piston toward the opposite end until it bottoms. If the air behind the piston has a pressure of 1,300 psi, the hydraulic system pump has to create a pressure within the system greater than 1,300 psi before the hydraulic fluid can actuate the piston. At 1,301 psi the piston starts to move within the cylinder, compressing the air as it moves. At 2,000 psi, it has backed up several inches. At 3,000 psi, the piston has backed up to its normal operating position, compressing the air until it occupies a space less than one-half the length of the cylinder. When actuation of hydraulic units lowers the system pressure, the compressed air expands against the piston, forcing fluid from the accumulator. This supplies an instantaneous supply of fluid to the hydraulic system component. The charged accumulator may also supply fluid pressure to actuate a component(s) briefly in case of pump failure.

Maintenance of Accumulators

Maintenance consists of inspections, minor repairs, replacement of component parts, and testing. There is an element of danger in maintaining accumulators. Therefore, proper precautions must be strictly observed to prevent injury and damage.

Before disassembling any accumulator, ensure that all preload air (or nitrogen) pressure has been discharged. Failure to release the preload could result in serious injury to the technician. Before making this check, be certain you know the type of high-pressure air valve used. When you know that all air pressure has been removed, you can take the unit apart. Be sure to follow manufacturer’s instructions for the specific unit you have.

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II. Hydraulic Pipelines:

When the system architecture is defined for all aircraft systems using hydraulic power, then it is possible to design the pipe layout in the aircraft. This layout will take into account the need to separate pipes to avoid common mode failures as a result of accidental damage or the effect of battle damage in a military aircraft. Once this layout has been set, it is possible to measure the lengths of pipe and to calculate the flow rate in each section and branch of pipe.

It is advisable that the first attempts to define a layout will result in straight lines only, but this is adequate for a reasonably accurate initial calculation. If an allowable pressure drop of 25% has been selected throughout the system, this may now be further divided between pressure pipes, return pipes and components. The designer will eventually control the specifications for the components, and in this sense he can allocate any value he chooses for pressure drop across each component. Once pipe lengths, flow rates and permissible pressure drops are known, pipe diameters can be calculated using the normal expression governing friction flow in pipes. It is normal to assume a fluid temperature of 0° C for calculations, and in most cases flow in aircraft hydraulic systems is turbulent.

III. Hydraulic Pumps:

A system will contain one or more hydraulic pumps depending on the type of aircraft .The pump is normally mounted on an engine-driven gearbox. In civil applications the pump is mounted on an accessory gearbox mounted on the engine casing. For military applications the pump is mounted on an Aircraft Mounted Accessory Drive (AMAD) mounted on the airframe.

The pump speed is therefore directly related to engine speed, and must therefore be capable of working over a wide speed range. The degree of gearing between the pump and the engine varies between engine types.

The universally used pump type is known as variable delivery, constant pressure. Demand on the pump tends to be continuous throughout a flight, but frequently varying in magnitude. This type of pump makes it possible to meet this sort of demand pattern without too much wastage of power. Within the flow capabilities of these pumps the pressure can be maintained within 5% of nominal except during the short transitional stages from low flow to high flow. This also helps to optimise the overall efficiency of the system.

The pumps are designed to sense outlet pressure and feed back this signal to a plate carrying the reciprocating pistons. The plate is free to move at an angle to the longitudinal axis of the rotating drive shaft. There are normally nine pistons arranged diametrically around the plate. The position of the plate therefore varies the amount of reciprocating movement of each piston. Examples of different types of hydraulic pump are shown in Figures 4.8 together with their salient characteristics.

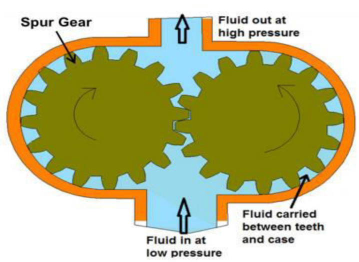
Classification of Hydraulic Pumps: These are mainly classified into two categories according to the displacement:

A. Non-positive displacement pumps B. Positive displacement pumps.

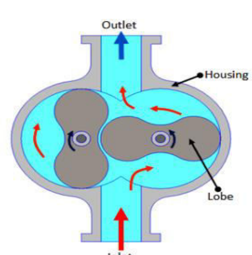
- ✓ **Non-Positive Displacement Pumps:** These pumps are also known as hydro-dynamic pumps. In these pumps the fluid is pressurized by the rotation of the propeller and the fluid pressure is proportional to the rotor speed. These pumps can not withstanding high pressures and generally used for low-pressure and high-volume flow applications. The fluid pressure and flow generated due to inertia effect of the fluid. The fluid motion is generated due to rotating propeller. These pumps provide a smooth and continuous flow but the flow output decreases with increase in system resistance (load). The flow output decreases because some of the fluid slip back at higher resistance. The fluid flow is completely stopped at very large system resistance and thus the volumetric efficiency will become zero. Therefore, the flow rate not only depends on the rotational speed but also on the resistance provided by the system. The important advantages of non-positive displacement pumps are lower initial cost, less operating maintenance because of less moving parts, simplicity of operation, higher reliability and suitability with wide range of fluid etc
- ✓ **Positive displacement pump:** These pumps deliver a constant volume of fluid in a cycle. The discharge quantity per revolution is fixed in these pumps and they produce fluid flow proportional to their displacement and rotor speed. These pumps are used in most of the industrial fluid power applications. The output fluid flow is constant and is independent of the system pressure (load). The important advantage associated with these pumps is that the high-pressure and low-pressure areas (means input and output region) are separated and hence the fluid cannot leak back due to higher pressure at the outlets. These features make the positive displacement pump most suited and universally accepted for hydraulic systems. The important advantages of positive displacement pumps over non-positive displacement pumps include capability to generate high pressures, high volumetric efficiency, high NPTEL – Mechanical – Mechatronics and Manufacturing Automation Joint initiative of IITs and IISc – Funded by MHRD Page 10 of 63 power to weight ratio, change in efficiency throughout the pressure range is small and wider operating range pressure and speed.
- ✓ **Gear Pump:** Gear pump is a robust and simple positive displacement pump. It has two meshed gears revolving about their respective axes. These gears are the only moving parts in the pump. They are compact, relatively inexpensive and have few moving parts. The rigid design of the gears and houses allow for very high pressures and the ability to pump highly viscous fluids. They are suitable for a wide range of fluids and offer self-priming performance. Sometimes gear pumps are designed to function as

either a motor or a pump. These pump includes helical and herringbone gear sets (instead of spur gears), lobe shaped rotors similar to Roots blowers (commonly used as superchargers), and mechanical designs that allow the stacking of pumps. Based upon the design, the gear pumps are classified as:

• **External gear pumps :** External gear pump The external gear pump consists of externally meshed two gears housed in a pump case as shown in figure 5.2.1. One of the gears is coupled with a prime mover and is called as driving gear and another is called as driven gear. The rotating gear carries the fluid from the tank to the outlet pipe. The suction side is towards the portion whereas the gear teeth come out of the mesh. When the gears rotate, volume of the chamber expands leading to pressure drop below atmospheric value. Therefore the vacuum is created and the fluid is pushed into the void due to atmospheric pressure. The fluid is trapped between housing and rotating teeth of the gears. The discharge side of pump is towards the portion where the gear teeth run into the mesh and the volume decreases between meshing teeth. The pump has a positive internal seal against leakage; therefore, the fluid is forced into the outlet port.



• **Lobe pumps:** Lobe pumps work on the similar principle of working as that of external gear pumps. However in Lobe pumps, the lobes do not make any contact like external gear pump (see Figure 5.2.3). Lobe contact is prevented by external timing gears located in the gearbox. Similar to the external gear pump, the lobes rotate to create expanding volume at the inlet. Now, the fluid flows into the cavity and is trapped by the lobes. Fluid travels around the interior of casing in the pockets between the lobes and the casing. Finally, the meshing of the lobes forces liquid to pass through the outlet port. The bearings are placed out of the pumped liquid. Therefore the pressure is limited by the bearing location and shaft deflection.



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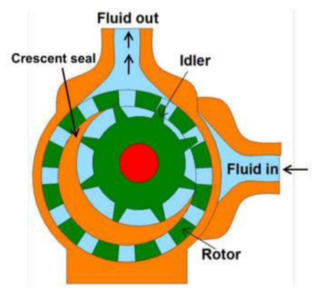
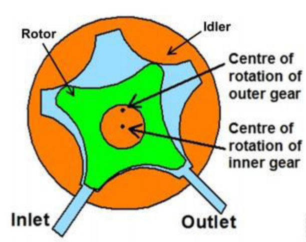
• **Internal gear pumps:** Internal gear pumps are exceptionally versatile. They are often used for low or medium viscosity fluids such as solvents and fuel oil and wide range of temperature. This is nonpulsing, self-priming and can run dry for short periods. It is a variation of the basic gear pump. It comprises of an internal gear, a regular spur gear, a crescent-shaped seal and an external housing. The schematic of internal gear pump is shown in figure 5.2.4. Liquid enters the suction port between the rotor (large exterior gear) and idler (small interior gear) teeth. Liquid travels through the pump between the teeth and crescent. Crescent divides the liquid and acts as a seal between the suction and discharge ports. When the teeth mesh on the side opposite to the crescent seal, the fluid is forced out through the discharge port of the pump. This clearance between gears can be adjusted to accommodate high temperature, to handle high viscosity fluids and to accommodate the wear. These pumps are bi-rotational so that they can be used to load and unload the vessels.



- **Gerotor pumps:** Gerotor is a positive displacement pump. The name Gerotor is derived from "Generated Rotor". At the most basic level, a Gerotor is essentially one that is moved via fluid power. Originally this fluid was water, today the wider use is in hydraulic devices. The schematic of Gerotor pump is shown in figure 5.2.5. Gerotor pump is an internal gear pump without the crescent. It consists of two rotors viz. inner and outer rotor. The inner rotor has N teeth, and the outer rotor has N+1 teeth. The inner rotor is located off-center and both rotors rotate. The geometry of the two rotors partitions the volume between them into N different dynamically-changing volumes. During the rotation, volume of each partition changes continuously.



- **Vane pumps:** Vane pumps generate a pumping action by tracking of vanes along the casing wall. The vane pumps generally consist of a rotor, vanes, ring and a port plate with inlet and outlet ports. The rotor in a vane pump is connected to the prime mover

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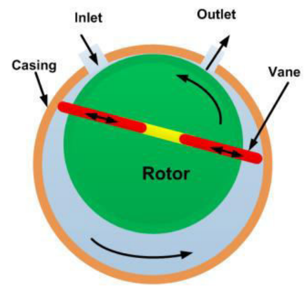
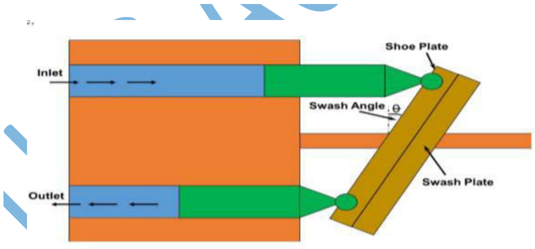
through a shaft. The vanes are NPTEL – Mechanical – Mechatronics and Manufacturing Automation Joint initiative of IITs and IISc – Funded by MHRD Page 17 of 63 located on the slotted rotor.



- **Piston pump:** Piston pumps are meant for the high-pressure applications. These pumps have highefficiency and simple design and needs lower maintenance. These pumps convert the rotary motion of the input shaft to the reciprocating motion of the piston. These pumps work similar to the four stroke engines. They work on the principle that a reciprocating piston draws fluid inside the cylinder when the piston retracts in a cylinder bore and discharge the fluid when it extends. Generally, these pumps have fixed inclined plate or variable degree of angle plate known as swash plate (shown in Figure 5.3.5 and Figure 5.3.6). When the piston barrel assembly rotates, the swash plate in contact with the piston slippers slides along its surface. The stroke length (axial displacement) depends on the inclination angle of the swash plate. They are of two types axial and radial



IV. Hydraulic Reservoir (tank):

A hydraulic reservoir is a tank or container designed to store sufficient hydraulic fluid for all conditions of operation. Usually the hydraulic reservoir must have the capability of containing extra fluid not being circulated in the system during certain modes of operation. When accumulators, actuating cylinders, and other units do not contain their maximum quantities of fluid, the unused fluid must be stored in the reservoir. On the other hand, when a maximum amount of fluid is being used in the system, the reservoir must still have a reserve adequate to meet all requirements. Reservoirs in hydraulic systems that require a reserve of fluid for the emergency operation of landing gear, flaps, etc., are equipped with standpipes. During normal operation, fluid is drawn through the stand pipe. When system fluid is lost, Emergency fluid is drawn from the bottom of the tank. Reservoirs are not designed to be completely filled; they must allow for an air space above the fluid level to allow for expansion of the fluid when it is headed during system operation. Reservoir will provide some means of checking the fluid level and of being replenished. The quantity-indicating method may be nothing more than a dipstick on the filler cap, or it may consist of a remote indicating system that displays the quantity on the aircraft flight deck.

The requirements for this tank vary depending on the type of aircraft involved. For most military aircraft the reservoir must be fully aerobatic. This means that the fluid must be fully contained, with no air/fluid interfaces, and a supply of fluid must be maintained in all aircraft attitudes and g conditions. In order to achieve a good volumetric efficiency from the pump, reservoir pressure must be sufficient to accelerate a full charge of fluid into each cylinder while it is open to the inlet port.

The volume of the reservoir is controlled by national specifications and includes all differential volumes in the system, allowance for thermal expansion and a generous emergency margin.

It is common practice to isolate certain parts of the system when the reservoir level falls below a predetermined point. This is an attempt to isolate leaks within the system and to provide further protection for flight safety critical subsystems. The cut-off point must ensure sufficient volume for the remaining systems under all conditions. The reservoir will be protected by a pressure relief valve which can dump fluid overboard.

Reservoirs can be broken down into two basic types,

- ✓ In-line: In-line reservoirs are those that are separate components in the hydraulic system. This is the most common type of reservoir. These can be pressurized or unpressurized. Unpressurized reservoirs are normally used in aircraft flying at lower altitudes, such as below 15000 ft [4583 m], UI whose hydraulic systems are limited to those associated with ground operations, such as brakes. Pressurized reservoirs are commonly found in aircraft designed for high-altitude flight where atmospheric pressure is low. The most basic rule of hydraulics states that fluid cannot be pulled; it can only be pushed. At sea level the 14.7 psi of atmosphere provides the force to push the fluid from the reservoir to the pump. As altitude increases, atmospheric pressure decreases. With little or no pressure on the fluid, it tends to foam, causing air bubbles to form in the low part of the system. When an aircraft is operating at high altitudes, the pump will be starved for fluid unless some means of pressurization is used. Therefore, to provide a continuous supply of fluid to the pumps, the reservoir is pressurized.
- ✓ Integral: Integral reservoirs are combined with the hydraulic pump. These types of reservoirs are often found in small aircraft, where the compact arrangement of this type of mechanism is desirable. An example of this is the brake master cylinder used

with many light-aircraft systems. the upper portion of the assembly serves as the reservoir and the lower portion serves as the pump to operate the brake.

and these can be further classified as

- ✓ pressurized
- ✓ Unpressurized.
- ✓

V. Control Valves: In a hydraulic system, the hydraulic energy available from a pump is converted into motion and force by means of an actuator. The control of these mechanical outputs (motion and force) is one of the most important functions in a hydraulic system. The proper selection of control selection ensures the desired output and safe function of the system. In order to control the hydraulic outputs, different types of control valves are required. It is important to know various types of control valves and their functions. This not only helps to design a proper hydraulic system but also helps to discover the innovative ways to improve the existing systems.

There are basically three types of valves employed in hydraulic systems:

1. Directional control valves: Directional control valves are used to control the distribution of energy in a fluid power system. They provide the direction to the fluid and allow the flow in a particular direction. These valves are used to control the start, stop and change in direction of the fluid flow. These valves regulate the flow direction in the hydraulic circuit. These control valves contain ports that are external openings for the fluid to enter and leave. The number of ports is usually identified by the term 'way'. For example, a valve with four ports is named as four-way valve. The fluid flow rate is responsible for the speed of actuator (motion of the output) and should controlled in a hydraulic system. This operation can be performed by using flow control valves. The pressure may increase gradually when the system is under operation. The pressure control valves protect the system by maintaining the system pressure within the desired range. Also, the output force is directly proportional to the pressure and hence, the pressure control valves ensure the desired force output at the actuator.

Directional control valves can be classified in the following manner:

1. Type of construction:
   - Poppet valves
   - Spool valves
2. Number of ports:
   - Two- way valves
   - Three – way valves
   - Four- way valves.
3. Number of switching position:
   - Two – position
   - Three - position

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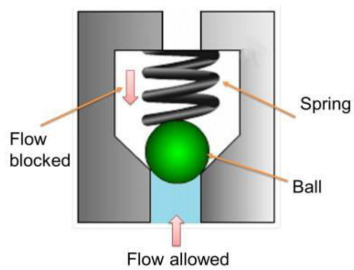
4. Actuating mechanism:
    - Manual actuation
    - Mechanical actuation
    - Solenoid actuation
    - Hydraulic actuation
    - Pneumatic actuation
    - Indirect actuation

Check Valves



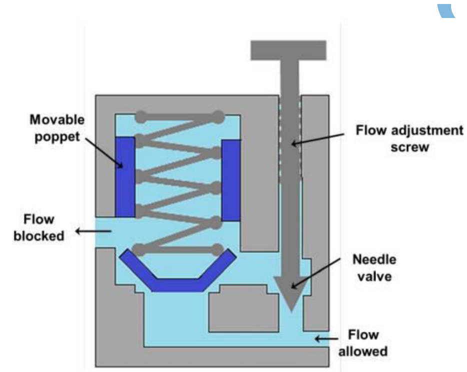


Figure: Inline check valve

Figure: Restriction check valve

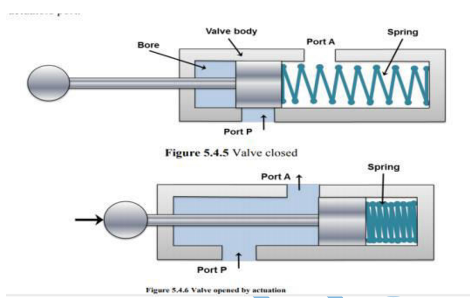


Figure: Valve Pump

2. Flow control valves:

3. Pressure control valves:

VI. Warnings and Indication:

Several instruments are normally situated in the hydraulic power generation system to monitor continuously its performance. Pressure transducers monitor system pressure and transmit this signal to gauges in the cockpit. Pressure switches are also incorporated to provide a warning of low pressure in the system on the central warning panel. Filter blockage indicators show the condition of the filter elements to ground servicing personnel, and a fluid temperature warning may be given to the aircrew. With increasing use of microprocessor based system management units, more in-depth health monitoring of all major components is possible with data displayed to ground crews on a maintenance data panel.

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More-Electric Hydraulic System

The effects on the hydraulic system of adopting more-electric concepts may be seen by comparing the hydraulic system configurations for the Boeing 767 (conventional Boeing wide body) versus the more-electric Boeing 787 as shown

in Figure 10.8. Boeing have been more conservative regarding the use of centralized aircraft hydraulic systems on the Boeing 787 as opposed to the use of more de-centralized systems on the Airbus A380 and certainly Lockheed Martin F-35/JSF. Boeing also use conventional hydraulic actuation in general whereas the Airbus A380 makes considerable use of Electro-Hydrostatic Actuators (EHA), and Electric Backup Hydrostatic Actuators (EBHAs) for primary flight control as described in Chapter 1. Nevertheless Figure 10.8 presents a valid comparison as it effectively contrasts conventional and more electric hydraulic system architectures with one another. Furthermore, it is also a valid size comparison as the 787 family is the direct market successor to the 767. Both 767 and 787 architectures use the Boeing Left (L), Centre (C), Right (R) hydraulic channel philosophy. The key differences are:

• Engine bleed air is removed with deletion of the Air Driven Pump (ADP)

• The use of 5000 psi rather than 3000 psi hydraulics system

• The adoption of 230 VAC, three-phase, VF primary power rather than 115 VAC, three-phase 400 Hz CF

• The use of starter/generators versus generators to facilitate electric engine start

• Use of larger Electric Motor Pumps (EMPs), around four times that of previous units Generally there are also increased levels of electrical power with the Primary channels increasing from 120 KVA to 500 kVA. The levels of power for the Ram Air Turbine (RAT) and the Electric Motor Pumps (EMPs) have also increased dramatically The three channel hydraulic system philosophy is more conservative than the '2H+2E' philosophy adopted on the Airbus A380. On the A380 the blue hydraulic channel has effectively been replaced by a channel using distributed electrically powered actuation using EHAs and EBHAs. Both aircraft utilize 5000 psi hydraulic systems. More-Electric Environmental Control System The abolition of bleed air means that electrically driven compressors must be used to pressurize the cabin and provide a source of air for the environmental control system. See Figure In common with most aircraft of this size, the B787 is fitted with two air-conditioning packs, the difference being that they are electrically powered. Each pack has two electrically driven motor compressors each controlled by a motor controller located in the aft EE bays. Each permanent magnet motor requires 125kVA of electric power to drive it. The outputs from these compressors enter a common manifold before being fed through primary and secondary heat exchangers, cooled by external ram air as would be the case in a conventionally driven Air Cycle Machine (ACM). The resulting cold air is mixed with recirculation air to maintain the desired cabin temperature. Although the power required by the electric ECS is considerable, the key advantage is that air is not being extracted from the engine's central core. More importantly, the temperature and pressure of the delivered air is considerably lower.

Example of Hydraulic system of Airbus A320:

The aircraft is equipped with three continuously operating hydraulic systems called Blue, Green and Yellow. Each system has its own hydraulic reservoir as a source of hydraulic fluid.

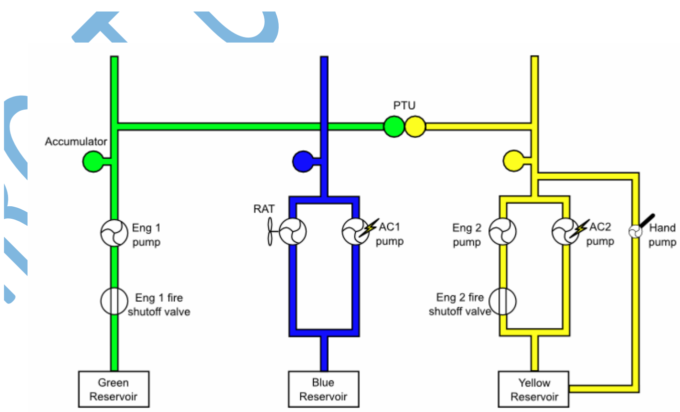
• The Green system (System 1) is pressurised by an Engine Driven Pump (EDP) located on No. 1 engine which may deliver 37 gallon per minute (US gpm) or 140 L/min

• The Blue system (System 2) is pressurised by an electric motor-driven pump capable of delivering 6.1 gpm or 23 L/min. A Ram Air Turbine (RAT) can provide up to 20.6 gpm or 78 L/min at 2175 psi in emergency conditions

• The Yellow system (System 3) is pressurised by an EDP driven by No. 2 Engine. An electric motor driven pump is provided which is capable of delivering 6.1 gpm or 23 L/min for ground servicing operations. This system also has a hand pump to pressurise the system for cargo door operation when the aircraft is on the ground with electrical power unavailable. Each channel has the provision for the supply of ground-based hydraulic pressure during maintenance operations. Each main system has a hydraulic accumulator to maintain system pressure in the event of transients.

Each system includes a leak measurement valve (shown as L in a square on the diagram), and a priority valve (shown as P in a square).

• The leak measurement valve is positioned upstream of the primary flight controls and is used for the measurement of leakage in each flight control system circuit. They are operated from the ground maintenance panel



In the event of a low hydraulic pressure, the priority valve maintains pressure supply to essential systems by cutting of the supply to heavy load users The bi-directional Power Transfer Unit (PTU) enables the Green or the Yellow systems to power each other without the transfer of fluid. In flight in

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the event that only one engine is running, the PTU will automatically operate when Civil Transport Comparison 165 Figure 4.20 Simplified A320 family hydraulic system the differential pressure between the systems is greater than 500 psi. On the ground, while operating the yellow system using the electric motor driven pump, the PTU will also allow the Green system to be pressurized.

The RAT extends automatically in flight in the event of failure of both engines and the APU. In the event of an engine fire, a fire valve in the suction line between the EDP and the appropriate hydraulic reservoir made be closed, isolating the supply of hydraulic fluid to the engine.

Pressure and status readings are taken at various points around the systems which allows the composition of a hydraulic system display to be shown on the Electronic Crew Alerting and Monitoring (ECAM).

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PNEUMATIC SYSTEMS

Introduction

The modern turbofan engine is effectively a very effective gas generator and this has led to the use of engine bleed air for a number of aircraft systems, either for reasons of heating, provision of motive power or as a source of air for cabin conditioning and pressurization systems. Bleed air is extracted from the engine compressor and after cooling and pressure reduction/regulation it is used for a variety of functions.

In the engine, high pressure bleed air is used as the motive power – sometimes called 'muscle power' – for many of the valves associated with the bleed air extraction function. Medium-pressure bleed air is used to start the engine in many cases, either using air from a ground power unit, APU or cross bleed from another engine on the aircraft which is already running. Bleed air is also used to provide anti-ice protection by heating the engine intake cowling and it is also used as the motive power for the engine thrust reversers.

On the aircraft, bleed air tapped from the engine is used to provide air to pressurize the cabin and provide the source of air to the cabin conditioning environmental control system. A proportion of bleed air is fed into air conditioning packs which cool the air dumping excess heat overboard; this cool air is mixed with the remaining warm air by the cabin temperature control system such that the passengers are kept in a comfortable environment. Bleed air is also used to provide main wing anti-ice protection.

Bleed air is also used for a number of ancillary functions around the aircraft: pressurizing hydraulic reservoirs, providing hot air for rain dispersal from the aircraft windscreen, pressurizing the water and waste system and so on. In some aircraft Air Driven Pumps (ADPs) are used as additional means of providing aircraft hydraulic power.

Pitot static systems are also addressed in the pneumatic chapter, as although this is a sensing system associated with measuring and providing essential air data parameters for safe aircraft flight, it nonetheless operates on pneumatic principles. Pitot systems have been used since the earliest days of flight using pneumatic, capsule based mechanical flight instruments. The advent of avionics technology led first to centralized Air Data Computers (ADCs) and eventually on to the more integrated solutions of today such as Air Data & Inertial Reference System (ADIRS).

Pneumatic power is the use of medium pressure air to perform certain functions within the aircraft. While the use of pneumatic power has been ever present since aircraft became more complex, the evolution of the modern turbojet engine has lent itself to the use of pneumatic power, particularly on the civil airliner.

The easy availability of high pressure air from the modern engine is key to the use of pneumatic power as a means of transferring energy or providing motive power on the aircraft. The turbojet engine is in effect a gas generator where the primary aim is to provide thrust to keep the aircraft in the air. As part of the turbojet combustion cycle, air is compressed in two or three stage compressor sections before fuel is injected in an atomized form and then ignited to perform the combustion process. The resulting expanding hot gases are passed over turbine blades at the rear of the engine to rotate the turbines and provide shaft power to drive the LP fan and compressor sections. When the engine reaches self-sustaining speed the turbine is producing sufficient shaft power to equal the LP fan/compressor requirements and the engine achieves a stable condition – on the ground this equates to the ground idle condition. The availability of high pressure, high temperature air bled from the compressor section of the engine lends itself readily to the ability to provide pneumatic power for actuation, air conditioning or heating functions for other aircraft subsystems.

Other areas of the aircraft use pneumatic principles for sensing the atmosphere surrounding the aircraft for instrumentation purposes. The sensing of air data is crucial to ensuring the safe passage of the aircraft in flight.

Use of Bleed Air: The use of the aircraft engines as a source of high-pressure, high-temperature air can be understood by examining the characteristics of the turbojet, or turbofan engine as it should more correctly be described. Modern engines 'bypass' a significant portion of the mass flow past the engine and increasingly a small portion of the mass flow passes through the engine core or gas generation section. The ratio of bypass air to engine core air is called the bypass ratio and this can easily exceed 10:1 for the very latest civil engines; much higher than the 4 or 5:1 ratio for the previous generation.

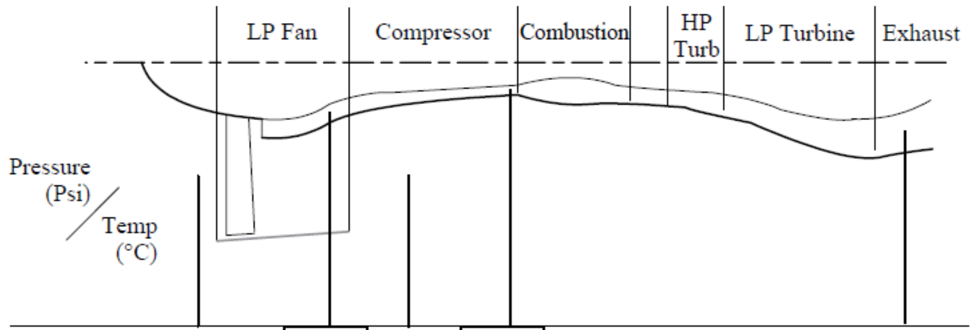
The characteristics of a modern turbofan engine are shown in Figure 6.1. This figure shows the pressure (in psi) and the temperature (in degrees centigrade) at various points throughout the engine for three engine conditions:

1. ground idle,
2. take-off power
3. cruise condition.

It can be seen that in the least stressful condition – ground idle – the engine is in a state of equilibrium but that even at this low level the compressor air pressure is 50 psi and the temperature 180 °C. At take-off conditions the compressor air soars to 410 psi/540 °C. In the cruise condition the compressor air is at 150 psi/400 °C. The engine is therefore a source of high pressure and high temperature air that can be 'bled' from the engine to perform various functions around the aircraft.

The fact that there are such considerable variations in air pressure and temperature for various engine conditions places an imposing control task upon the pneumatic system. Also the variations in engine characteristics between similarly rated engines of different manufactures poses additional design constraints. Some aircraft such as the Boeing 777 offer three engine choices, Pratt & Whitney, General Electric and Rolls-Royce, and each of these engines has to be separately matched to the aircraft systems, the loads of which may differ as a result of operator specified configurations.

UNIT 1: Hydraulic Systems Pneumatic and Landing Gear Systems



Idle	14.7/15	16/20	15/16	50/180	14.8/340
Take-Off	14.7/15	39/115	25/65	410/540	23.0/510
Cruise	5.3/–25	14/60	8.5/15	150/400	6.8/340

Figure 6.1 Characteristics of a modern turbofan engine

As well as the main aircraft engines the Auxiliary Power Unit (APU) is also a source of high pressure bleed air. The APU is in itself a small turbojet engine, designed more from the viewpoint of an energy and power generator than a thrust provider which is the case for the main engines.

The APU is primarily designed to provide electrical and pneumatic power by a shaft driven generator and compressor. The APU is therefore able to provide an independent source of electrical power and compressed air while the aircraft is on the ground, although it can be used as a backup provider of power while airborne. Some aircraft designs are actively considering the use of in-flight operable APUs to assist in in-flight engine re-lighting and to relieve the engines of off take load in certain areas of the flight envelope.

It is also usual for the aircraft to be designed to accept high pressure air from a ground power cart, for aircraft engine starting.

These three sources of pneumatic power provide the muscle or means by which the pneumatic system is able to satisfy the aircraft demands. In a simplified form the pneumatic system may be represented by the interrelationships shown in Figure below.

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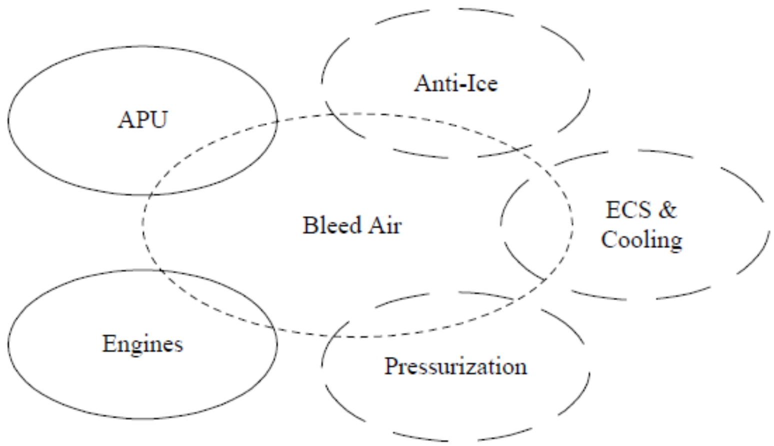

Figure 6.2 Relationship of HP air with major aircraft systems

This simplified drawing – the ground air power source is omitted – shows how the aircraft High Pressure (HP) air sources provide bleed air which forms the primary source for the three major aircraft air related systems:

· Ice protection: the provision of hot air to provide anti icing of engine nacelles and the wing, tailplane or fin leading edges; or to dislodge ice that has formed on the surfaces

· ECS and cooling: the provision of the main air source for environmental temperature control and cooling

· Pressurisation: the provision of a means by which the aircraft may be pressurised, giving the crew and passengers a more comfortable operating environment

A simplified representation of this relationship is shown in Figure. This example shows a twin-engine configuration typical of many business jets and regional jet transport aircraft.

Bleed air from the engines is passed through a Pressure-Reducing Shut-Off Valve (PRSOV) which serves the function of controlling and, when required, shutting off the engine bleed air supply. Air downstream of the PRSOV may be used in a number of ways:

· By means of a cross flow Shut-Off Valve (SOV) the system may supply air to the opposite side of the aircraft during engine start or if the opposite engine is inoperative for any reason

· A SOV from the APU may be used to isolate the APU air supply

· SOVs provide isolation as appropriate to the left and right air conditioning packs and pressurisation systems

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· Additional SOVs provide the means by which the supply to left and right wing anti-icing systems may be shut off in the event that these functions are not required

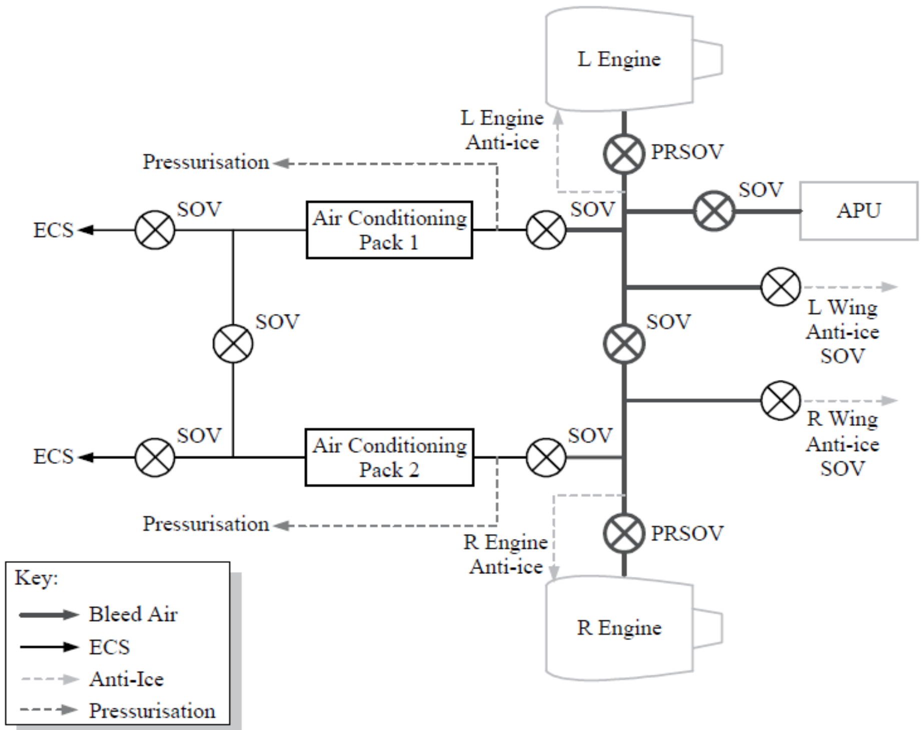

Figure 6.3 Simplified bleed air system and associated aircraft systems

This is a simplified model of the use of engine bleed air in pneumatic systems. A more comprehensive list of those aircraft systems with which bleed air is associated are listed as follows with the accompanying civil ATA chapter classification:

· Air conditioning (ATA Chapter 21)

· Cargo compartment heating (ATA Chapter 21)

· Wing and engine anti-icing (ATA Chapter 30)

· Engine start (ATA Chapter 80)

· Thrust reverser (ATA Chapter 78)

· Hydraulic reservoir pressurisation (ATA Chapter 29)

· Rain repellent nozzles – aircraft windscreen (ATA Chapter 30)

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· Water tank pressurisation and toilet waste (ATA Chapter 38)

· Air driven hydraulic pump (ADP) (ATA Chapter 29)

Several examples will be examined within this pneumatic systems chapter. However, before describing the pneumatically activated systems it is necessary to examine the extraction of bleed air from the engine in more detail.

Engine Bleed Air Control:

Figure gives a more detailed portrayal of the left-hand side of the aircraft bleed air system, the right side being an identical mirror image of the left-hand side.

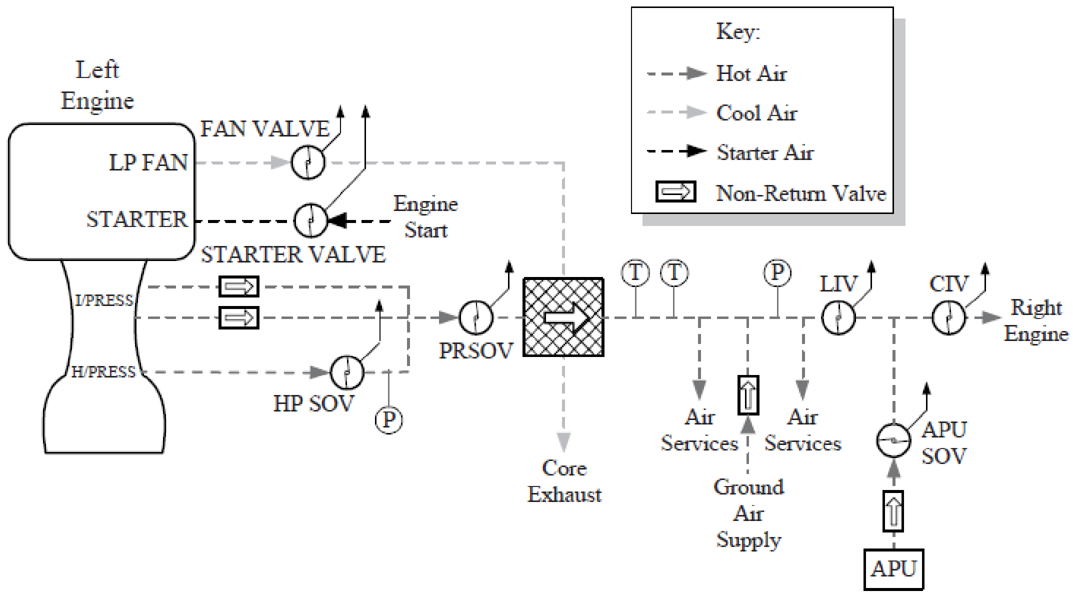

Figure 6.4 Typical aircraft bleed air system – left hand side

Air is taken from an intermediate stage or high pressure stage of the engine compressor depending upon the engine power setting. At lower power settings, air is extracted from the high pressure section of the compressor while at higher power settings the air is extracted from the intermediate compressor stage. This ameliorates to some degree the large variations in engine compressor air pressure and temperature for differing throttle settings as already shown in Figure 6.1. A pneumatically controlled High Pressure Shut-Off Valve (HP SOV) regulates the pressure of air in the engine manifold system to around 100 psi and also controls the supply of bleed air from the engine.

The Pressure-Reducing Shut-Off Valve (PRSOV) regulates the supply of the outlet air to around 40 psi before entry into the pre-cooler. Flow of cooling air through the pre-cooler is regulated by the fan valve which controls the temperature of the LP fan air and therefore of the bleed air entering the aircraft system. Appropriately located pressure and temperature sensors allow the engine bleed air temperature and pressure to be monitored and controlled within specified limits.

A typical PRSOV is shown in Figure 6.5a; an example of a Harrier II valve which is solenoid controlled and pneumatically operated and which controls temperature, flow and pressure is shown in Figure 6.5b.

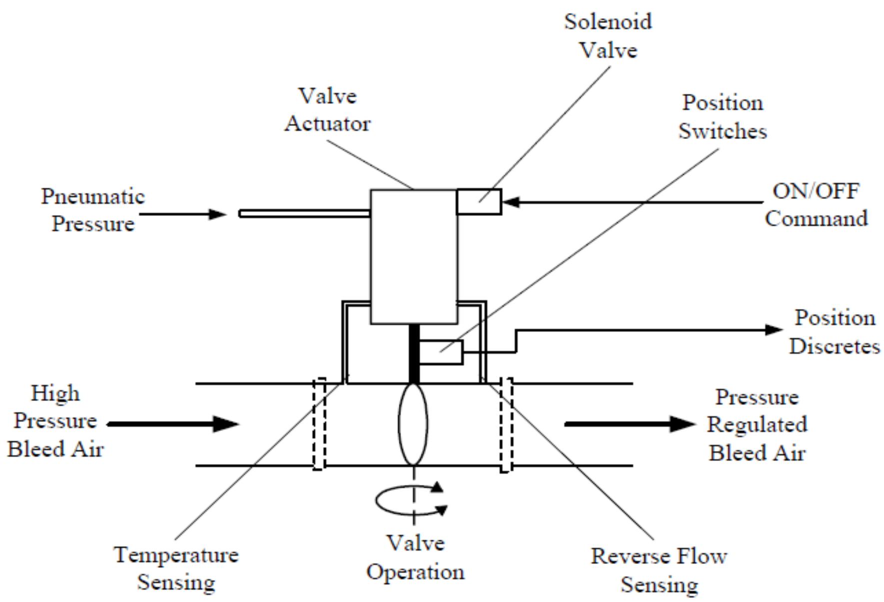

Figure 6.5a Typical Pressure-Reducing Shut-Off Valve (PRSOV)

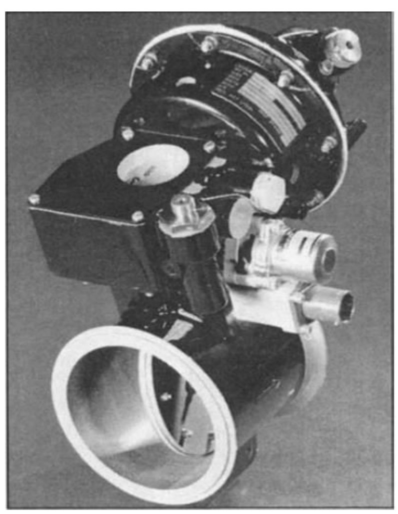

Figure 1p.5b Harrier II pneumatic valve (Courtesy of Honeywell Normalair-Garret Ltd)

The PRSOV performs the following functions:

· On/off control of the engine bleed system

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· Pressure regulation of the engine supply air by means of a butterfly valve actuated by pneumatic pressure

· Engine bleed air temperature protection and reverse flow protection

· Ability to be selected during maintenance operations in order to test reverse thrust operation

The PRSOV is pneumatically operated and electrically controlled. Operation of the solenoid valve from the appropriate controller enables the valve to control the downstream pressure pneumatically to ~40 psi within predetermined limits. The valve position is signalled by means of discrete signals to the bleed air controller and pressure switches provide over and under-pressure warnings. The various pressure, flow and discrete signals enable the bleed air controller Built-In Test (BIT) to confirm the correct operation of the PRSOV and fan control valve combination. This ensures that medium pressure air (~40 psi) of the correct pressure and temperature is delivered to the pre-cooler and thence downstream to the pneumatic and air distribution system. Downstream of the PRSOV and pre-cooler, the air is available for the user subsystems, a number of which are described below.

A number of isolation valves or SOVs are located in the bleed air distribution system. These valves are usually electrically initiated, pneumatically operated solenoid valves taking 28 VDC electrical power for ON/OFF commands and indication. A typical isolation valve is shown in Figure 6.6. The valve shaft runs almost vertically across the duct as shown in the diagram and the valve mechanism and solenoid valve is located on the top of the valve.

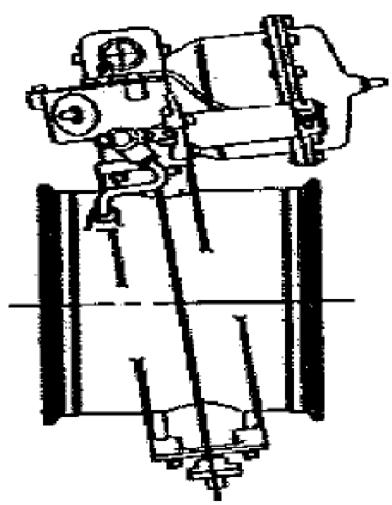

Figure: bleed air system isolation valve

Bleed Air System Indications: It is common philosophy in civil aircraft bleed air systems, in common with other major aircraft subsystems, to display system synoptic and status data to the flight crew on the Electronic Flight Instrument System (EFIS) displays. In the case of Boeing aircraft the synoptic are shown on the Engine Indication and Crew Alerting System (EICAS) display whereas for Airbus aircraft the Electronic Crew Alerting and Monitoring (ECAM) displays are used. Both philosophies display system data on the colour displays located on the central display console where they may be easily viewed by both Captain and First Officer. A typical bleed air system synoptic is shown in Figure.

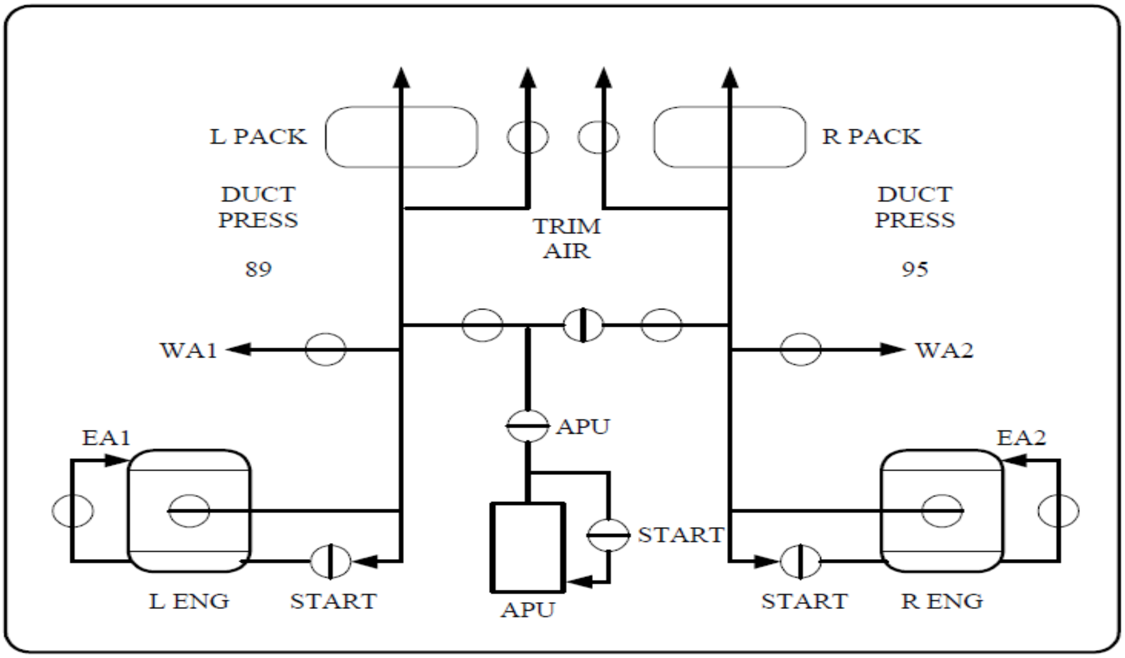

Figure: typical bleed air system synoptic display

The synoptic display as shown portrays sufficient information in a pictorial form to graphically show the flight crew the operating status of the system. In the example, both main engines are supplying bleed air normally but the APU is isolated. The cross-flow valve is shut, as are both engine start valves. The wing and engine anti-ice valves are open, allowing hot bleed air to be fed to the engines and wing leading edge to prevent any ice accretion.

Bleed Air System Users: The largest subsystem user of bleed air is the air system. Bleed air is used as the primary source of air into the cabin and fulfils the following functions:

· Cabin environmental control – cooling and heating

· Cabin pressurisation

· Cargo bay heating

· Fuel system pressurisation in closed vent fuel system used in some military aircraft

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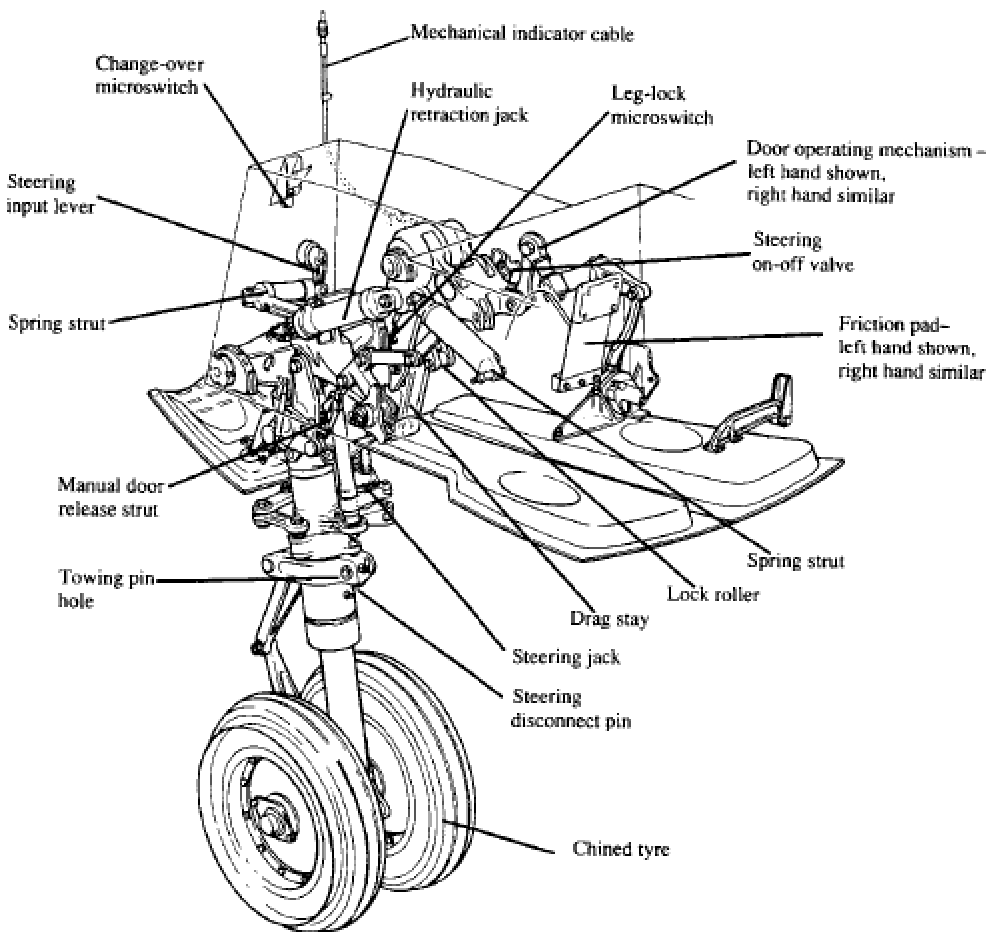
LANDING GEAR SYSTEMS

The Raytheon/BAE 1000 is representative of many modern aircraft; its landing gear is shown in Figures 1.22 and 1.23. It consists of the undercarriage legs and doors, steering and wheels and brakes and anti-skid system. All of these functions can be operated hydraulically in response to pilot demands at cockpit mounted controls.

Nose Gear The tricycle landing gear has dual wheels on each leg. The hydraulically operated nose gear retracts forward into a well beneath the forward equipment 168 Hydraulic Systems Figure 4.22 The Raytheon 1000 nose landing-gear (Courtesy of Raytheon) bay. Hinged nose-wheel doors, normally closed, are sequenced to open when lowering or retracting the nose gear. The advantage of the doors being normally closed is twofold. First, the undercarriage bay is protected from spray on takeoff and landing, and secondly there is a reduction in drag. A small panel on the leg completes enclosure on retraction and a mechanical indicator on the flight deck shows locking of the gear.



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Main Gear: The main gear is also hydraulically operated and retracts inwards into wheel bays. Once retracted the main units are fully enclosed by means of fairings attached to the legs and by hydraulically operated doors. Each unit is operated by a single jack and a mechanical linkage maintains the gear in the locked position without hydraulic assistance.

The main wheel doors jacks are controlled by a sequencing mechanism that closes the doors when the gear is fully extended or retracted. Figure 4.24 shows the landing gear sequence for the BAE 146 and also shows the clean lines of the nose wheel bay with the doors shut.

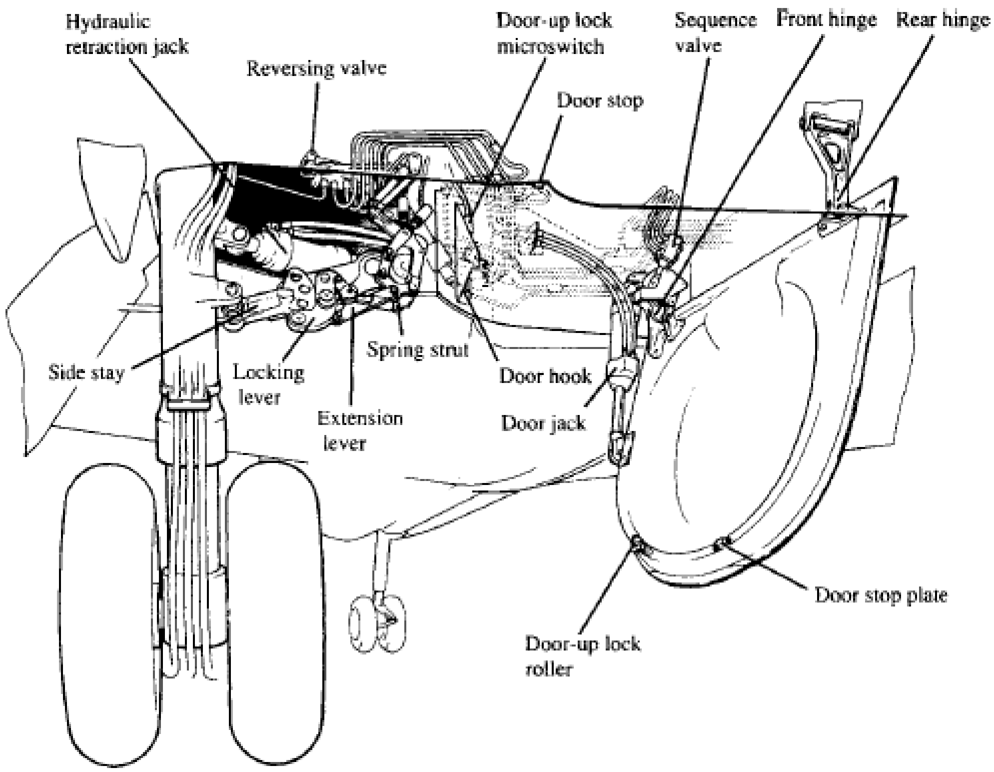

Figure 1.23 The Raytheon 1000 main landing-gear (Courtesy ofRaytheon)

Braking Anti-Skid and Steering: Stopping an aircraft safely at high landing speeds on a variety of runway surfaces and temperatures, and under all weather conditions demands an effective braking system. Its design must take into account tire to ground and brake friction, the brake pressure/volume characteristics, and the response of the aircraft hydraulic system and the aircraft structural and dynamic characteristics.

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Simple systems are available which provide reasonable performance at appropriate initial and maintenance costs. More complex systems are available to provide minimum stopping distance performance with features such as auto-braking during landing and rejected take-off, additional redundancy and self test. Some of the functional aspects of brakes and steering are illustrated in Figure.

The normal functions of landing, deceleration and taxying to dispersal or the airport gate require large amounts of energy to be applied to the brakes. Wherever possible, lift dump and reverse thrust will used to assist braking.

However it is usual for a large amount of heat to be dissipated in the brake pack. This results from the application of brakes during the initial landing deceleration, the use of brakes during taxying, and the need to hold the aircraft on brakes for periods of time at runway or taxiway intersections.

When the aircraft arrives at the gate the brakes, and the wheel assembly will be very hot. This poses a health and safety risk to ground crew working in the vicinity of the wheels during the turnaround. This is usually dealt with by training.

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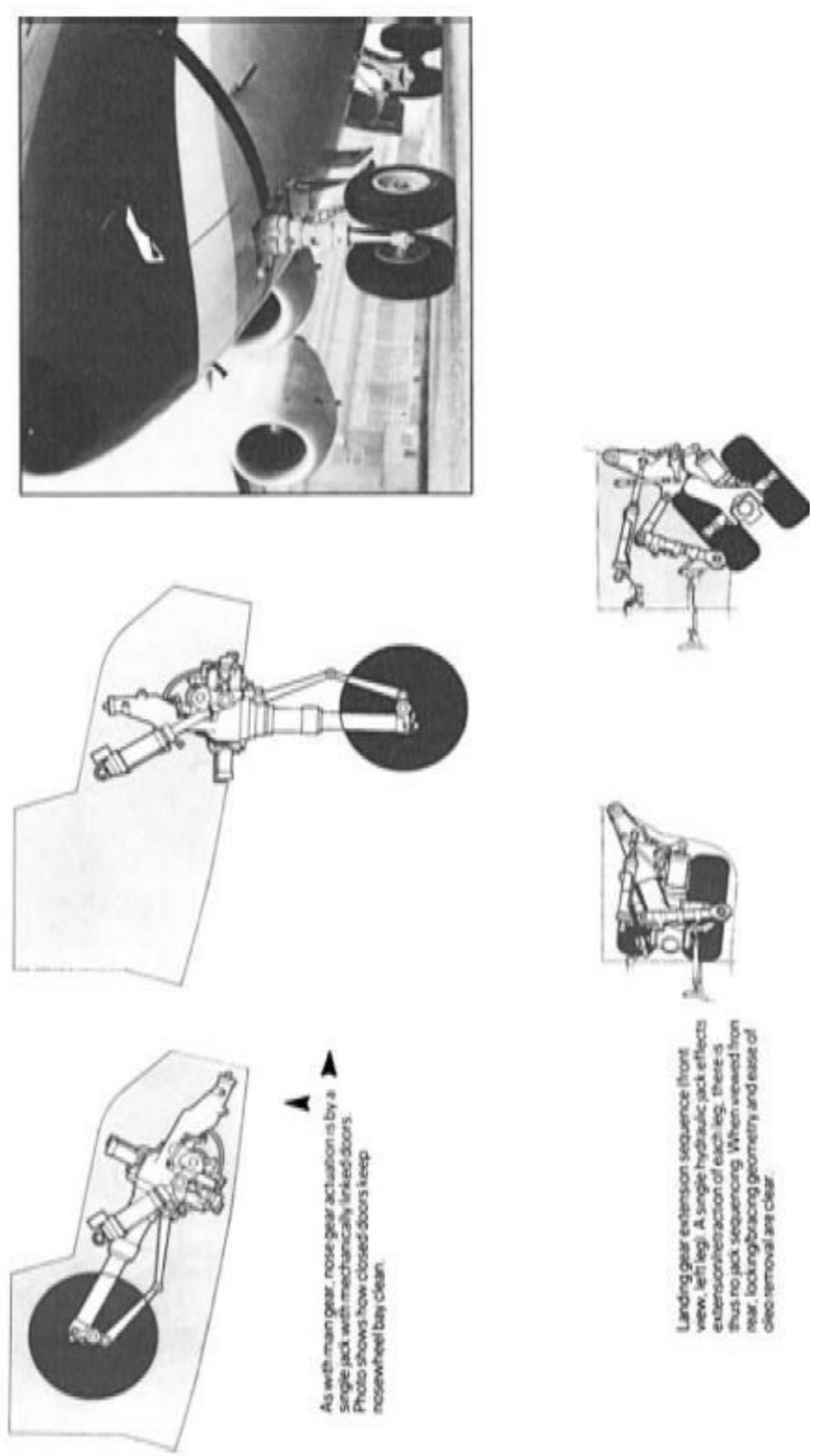

figure

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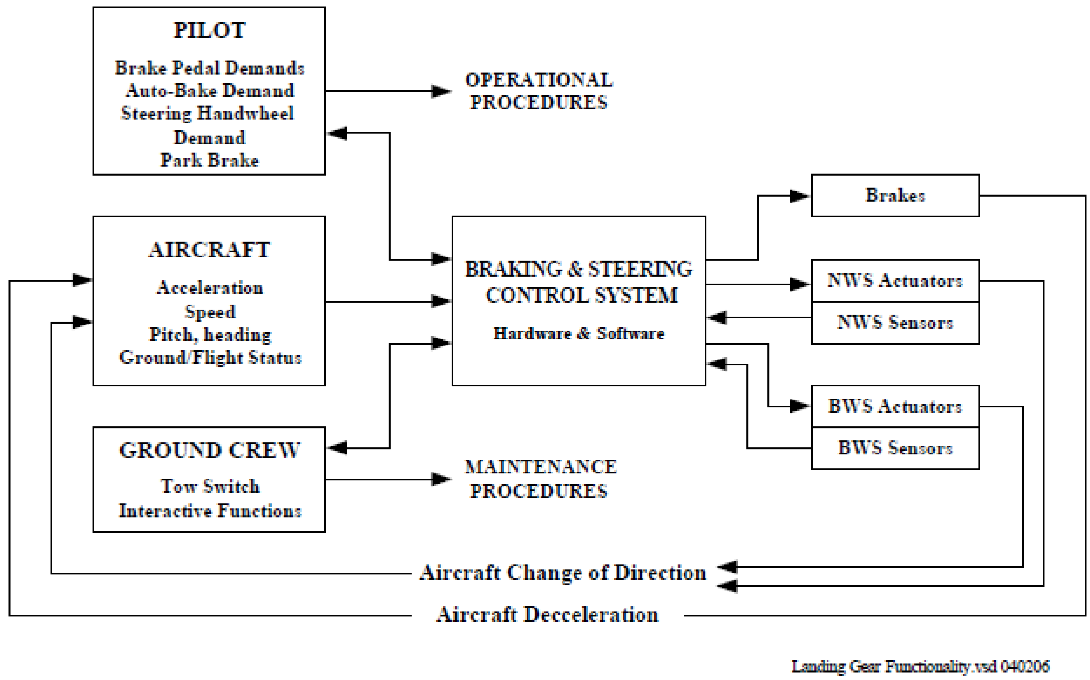

figure

A more serious operational issue is that the aircraft cannot depart the gate until the brake and wheel assembly temperature cools to a value that will not support ignition of hydraulic fluid. This is to ensure that, during the taxi back to the take-off runway, further brake applications will not raise the temperature of the brake pack to a level that will support ignition if a leak of fluid occurs during retraction. Departure from the gate, therefore, may be determined by brake temperature as indicated by a sensor in the brake pack rather than by time taken to disembark and embark passengers. Some aircraft address this issue by installing brake cooling fans in the wheel assembly to ventilate the brakes. An alternative method is to install fire detection and suppression systems in the wheel bays.

There are events that can raise the temperature of the brakes to the extent that a fire may occur and the tyres can burst. Examples of this are an aborted take-off (maximum rejected take-off) or an immediate go around and heavy landing. In both circumstances the aircraft will be fully laden with passengers and fuel. Thermal plugs will operate to deflate the tyres and fire crews will attend the aircraft to extinguish the fire while the passengers disembark.

One of the simplest and most widely known anti-skid system is the Dunlop Maxaret unit which consists of a hydraulic valve assembly regulated by the dynamics of a spring loaded g sensitive flywheel. Figure shows an axle mounted Maxaret together with a modulator.

Rotation of the flywheel is by means of a self-aligning drive from the hub of the wheel, allowing the entire unit to be housed within the axle and protecting the unit from the effects of weather and stones thrown up by the aircraft wheels. Skid conditions are detected by the overrun of the flywheel which

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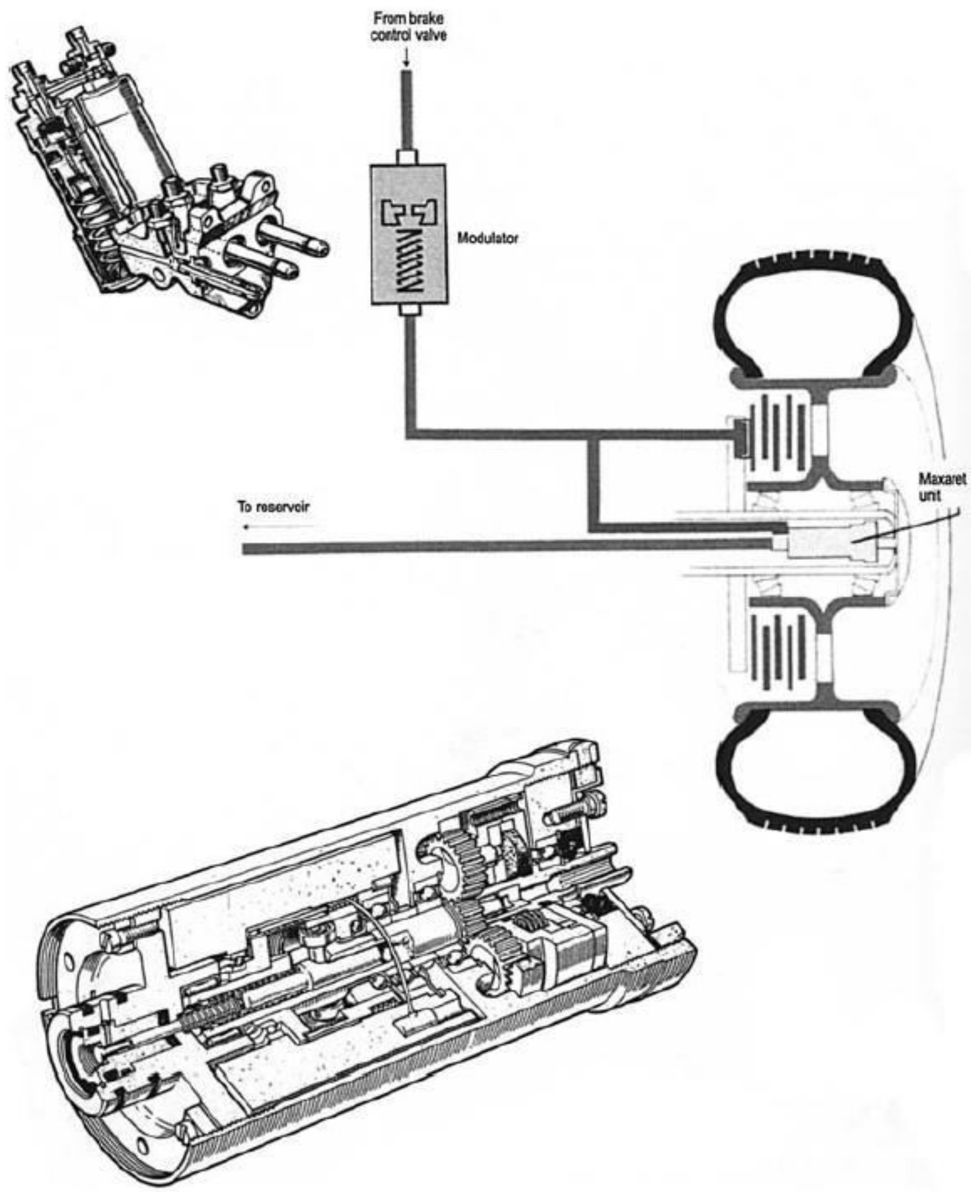
opens the Maxaret valve to allow hydraulic pressure to dissipate. A combination of flow sensitive hydraulic units and switches in the oleo leg provide modulation of pressure for optimum braking force and protection against inadvertent application of the brakes prior to touchdown. This ensures that the aircraft does not land with the brakes applied by only allowing the braking system to become active after the oleo switches have sensed that the oleo is compressed. This condition is known as 'weight-on-wheels'. Without this protection the effect of landing with full braking applied could lead to loss of control of the aircraft; at a minimum a set of burst tyres.



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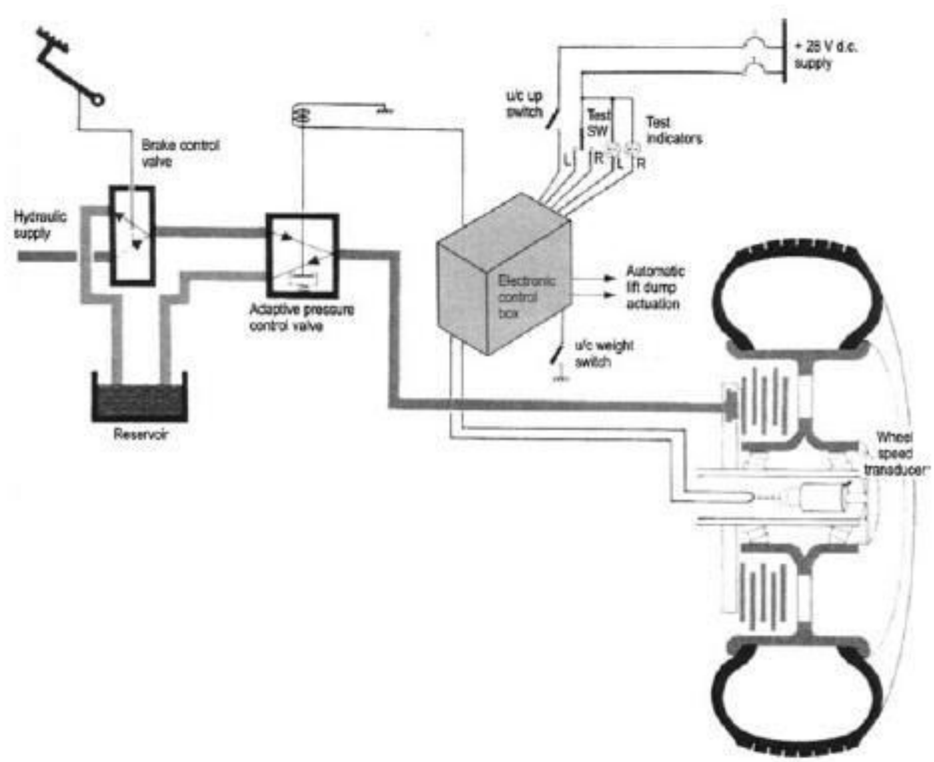
Electronic Control: Electronic control of braking and anti-skid systems has been introduced in various forms to provide different features. An electronic anti-skid system with adaptive pressure control is shown in Figure.



In this system the electronic control box contains individual wheel deceleration rate skid detection circuits with cross reference between wheels and changeover circuits to couple the control valve across the aircraft should the loss of a wheel speed signal occur.

If a skid develops the system disconnects braking momentarily and the adaptive pressure coordination valve ensures that brake pressure is re-applied at a lower pressure after the skid than the level which allowed the skid to occur. A progressive increase in brake pressure between skids attempts to maintain a high level of pressure and braking efficiency.

The adaptive pressure control valve dumps hydraulic pressure from the brake when its first stage solenoid valve is energized by the commencement of a skid signal. On wheel speed recovery the solenoid is de-energized and the brake pressure re-applied at a reduced pressure level, depending on the time interval of the skid. Brake pressure then rises at a controlled rate in search of the maximum braking level, until the next incipient skid signal occurs.

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Automatic Braking: A more comprehensive system is the Dunlop automatic brake control system illustrated in Figure 4.28, which allows an aircraft to be landed and stopped without pilot braking intervention. During automatic braking a two-position three-way solenoid valve is energised following wheel spin-up to feed system pressure via shuttle valves directly to the anti-skid valves where it is modulated and passed to the brakes. Signals from the auto-braking circuit are responsible for this modulation of pressure at the brake to match a preselected deceleration. However, pilot intervention in the anti-skid control circuit or anti-skid operation will override auto-brake at all times to cater for variations in runway conditions.

In the interest of safety a number of prerequisites must be satisfied before auto-braking is initiated:

- Auto-brake switch must be on and required deceleration selected
- Anti-skid switch must be on and operative
- Throttle must be correctly positioned
- Hydraulic pressure must be available
- Brake pedals must not be depressed
- Wheels must be spun up

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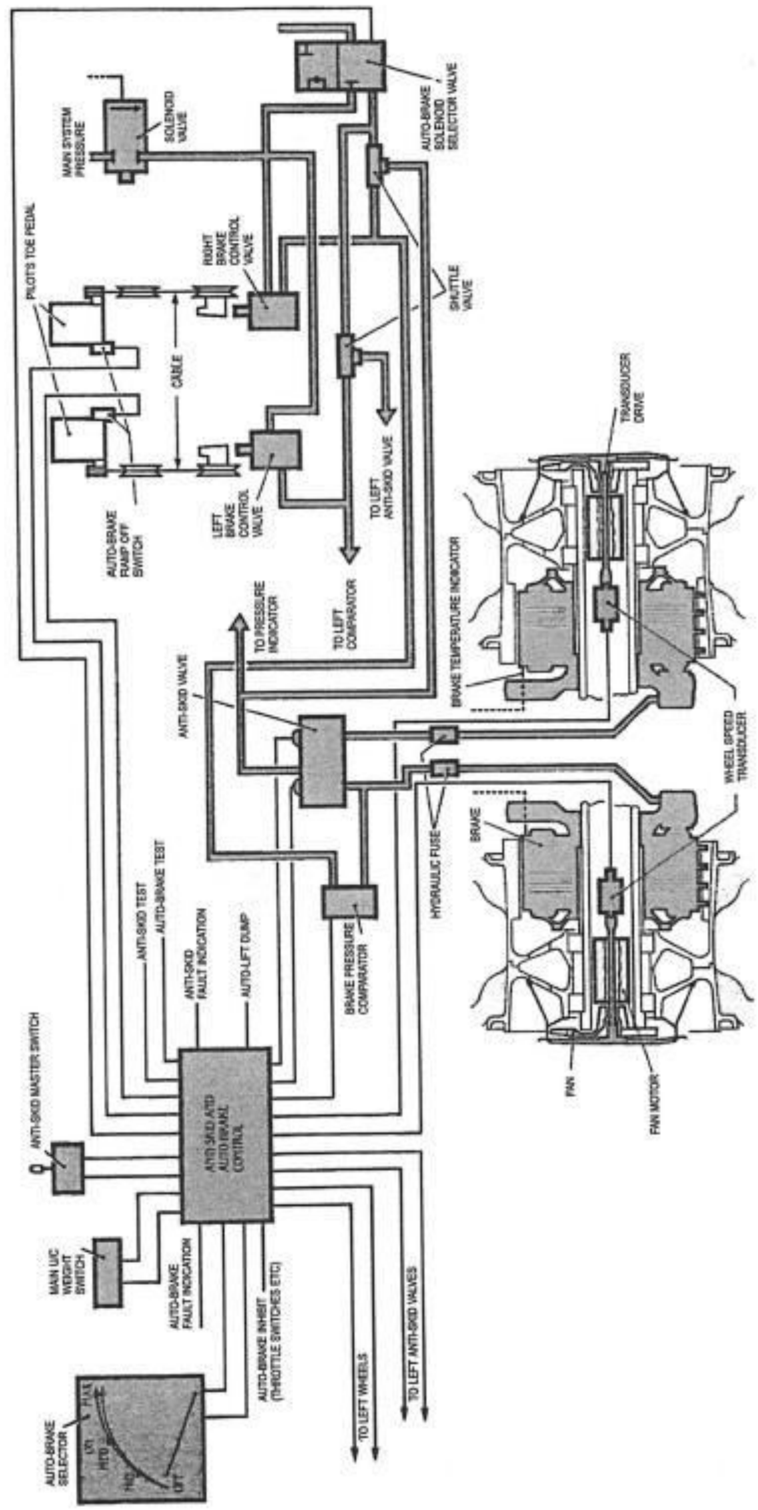

Figure 1.28 An automatic brake control system (Courtesy of Dunlop Aerospace International)

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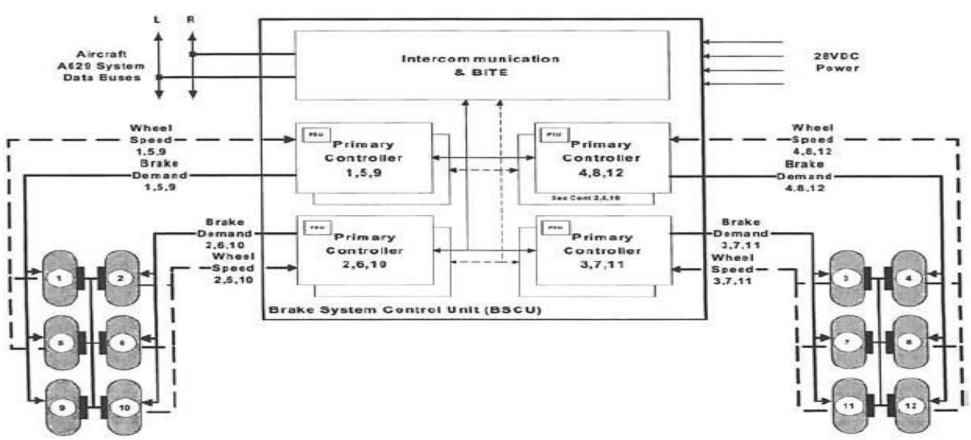
With all these conditions satisfied auto-braking will be operational and will retard the aircraft at a predetermined rate unless overridden by anti-skid activity. At any time during the landing roll the auto-braking may be overridden by the pilot by advancing the throttle levers for go-around, or by normal application of the brakes.

Multi-Wheel Systems: The systems described thus far apply to most aircraft braking systems. However, large aircraft have multi-wheel bogies and sometimes more than two main gears. The B747-400 has four main oleos, each with a total of four wheels each. The B777 has two main bogies with six wheels each. These systems tend to be more complex and utilise multi-lane dual redundant control. The B777 main gear shown in Figure 4.29 is an example.

For control purposes the wheels are grouped in four lines of three wheels, each corresponding to an independent control channels as shown in the figure. Each of the lines of three wheels – 1, 5, 9; 2, 6, 10 and so on – is controlled by a dual redundant controller located in the Brake System Control Unit (BSCU). Brake demands and wheel speed sensor readings are grouped by each channel and interfaced with the respective channel control. Control channels have individual power supplies to maintain channel segregation and integrity. The BSCU interfaces with the rest of the aircraft by means of left and right A629 aircraft systems data buses. This system is supplied by the Hydro-Aire division, part of Crane Aerospace, and is indicative of the sophistication which modern brake systems offer for larger systems.

The landing gear configuration for the Airbus A380 is shown in Figure 4.30. Goodrich provide two six-wheel under-fuselage landing gear and the two four-wheeled wing-mounted landing gear. The wing-mounted landing gear is slightly forward of the fuselage-mounted gear. The wheels on the main landing gear are fitted with carbon brakes.



Figure

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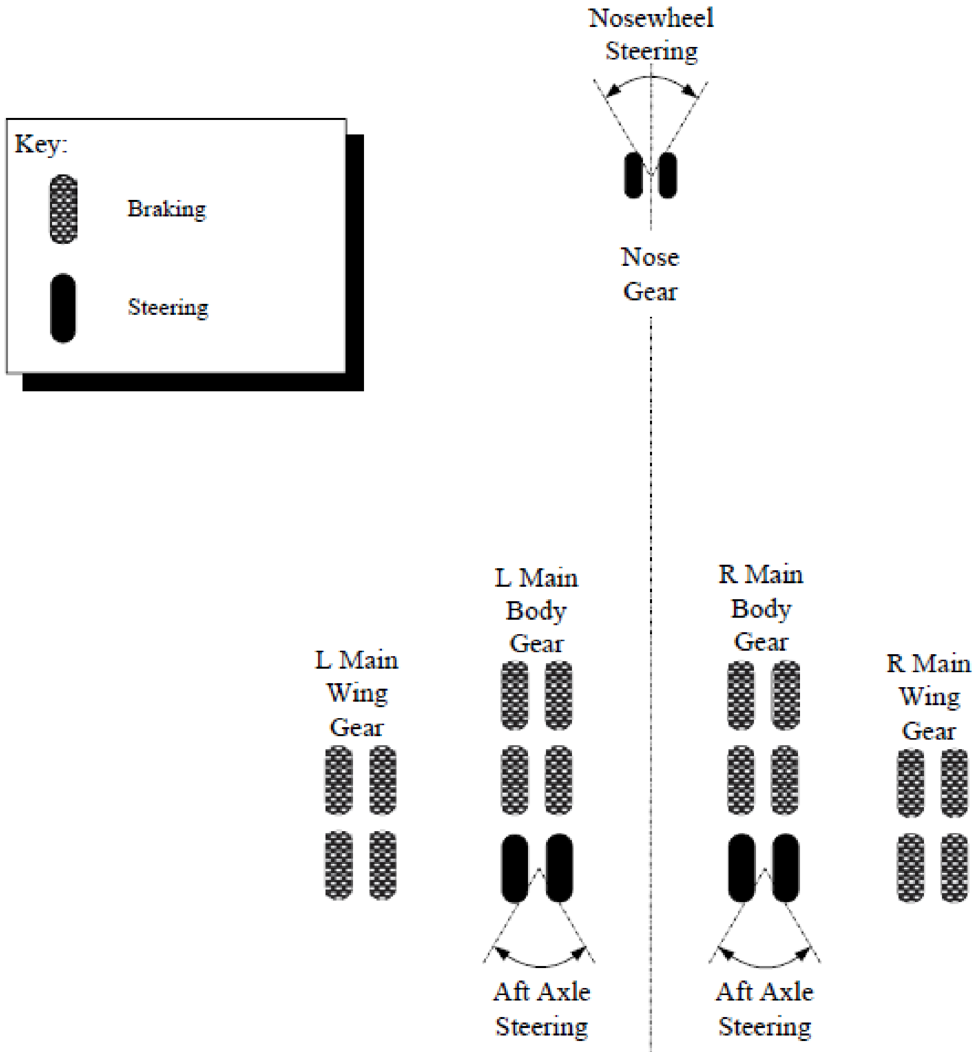

Figure 1.30 The Airbus A380 landing gear configuration

The twin wheel nose landing gear is supplied by Messier-Dowty. The steering control is via the nose gear and via the rear axle of the fuselage landing gear. The gear allows U-turn maneuvers on a 60 m-wide runway. Maneuverability is improved by having a hydraulically steerable aft axle which helps the aircraft attain tight turns without applying unacceptable torsion loads to the main oleo.

The aircraft can maneuver on 23 m-wide taxiways and 45 m-wide runways. The French aerospace company Latecoere, based in Toulouse, developed the External and Taxi Aid Camera System (ETACS). The ETACS consists of five video cameras and an onboard computer. The cameras are installed on the top of the tailfin and under the fuselage and the image data is relayed to cockpit displays to assist the crew in ground maneuvers. The Honeywell terrain guidance and on-ground navigation systems are integrated to the aircraft's flight management system.

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The braking system for the A380 is shown in Figure 4.31. This system is provided by Messier-Bugatti. This system is based on self-adapting braking algorithms that were successfully introduced on the A340-500/600. These allow optimised braking by managing the braking function wheel by wheel and landing by landing based on the prevailing conditions of runway, tyres and brakes. Each wheel is thus continuously and independently controlled in real time, taking account of its individual parameters and its particular environment. Both the number of wheels and the rapidity required in the feedback loop for controlling the wheel speed, necessitated introducing three dedicated computers, called RDCs (Remote Data Concentrator), each equipped with specific operating software. They are connected to the IMA by a digital bus.

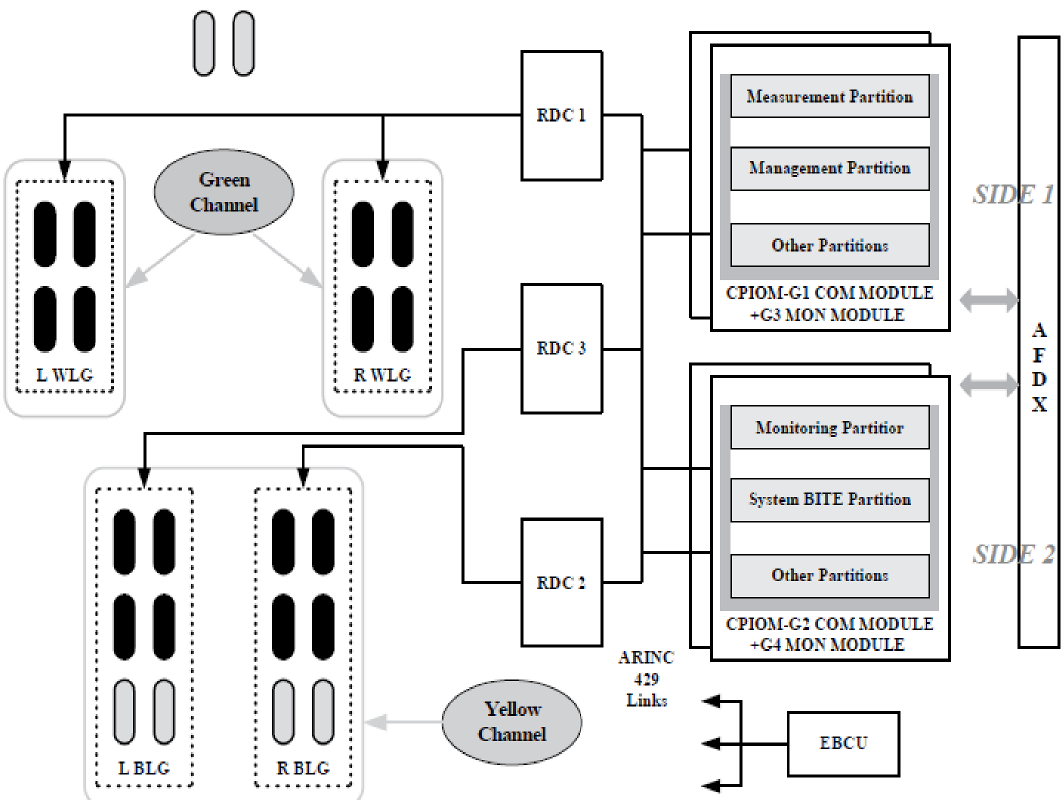

Figure A380 brake control system

Brake Parachute: Military aircraft often require assistance to achieve a high-speed landing on short runways. A brake parachute can be used to provide this facility. The system can be armed in flight and commanded by a weight on wheels switch when the main wheels touchdown. Figure 4.32 shows an F-117 with brake parachute deployed. The chute is jettisoned on to the runway and must be collected before the next aircraft attempts a landing.

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Figure F117 deploying brake parachute (Courtesy of US Air Force/ Senior Airman Darnell Cannady)

Steering: Nose wheel steering is normally not engaged for landing – the rudder can be used until forward speed makes it ineffective. At this point steering is engaged manually or automatically. Steering motors respond to demands from the rudder pedals when nose wheel steering is selected.

The angular range of the wheels, and the rate of change of steering angle are selected to enable the aircraft to steer on runways and taxi-ways with no risk of the aircraft over-steering or scrubbing the tires.

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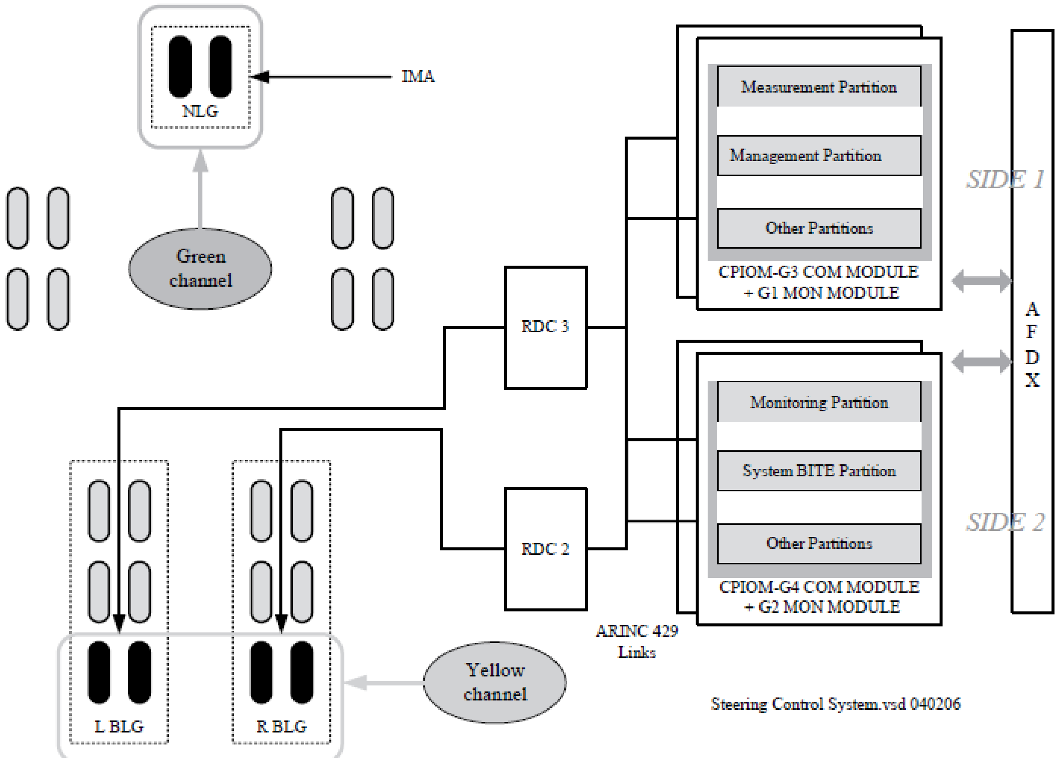

Figure 1.33 A380 steering control system

An example of the Airbus A380 steering system is shown in Figure 4.33. The A380 steers with the nose-wheels, and also the after wheels of the main gear. This enables the aircraft to complete a 180 _ turn within 56.5 m, safely within the standard 60 m runway width.

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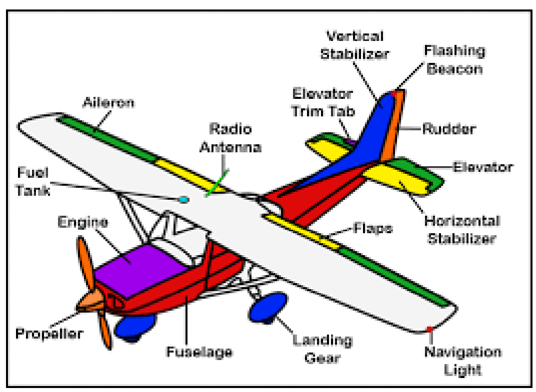
PPT

UNIT 2: AIRPLANE CONTROL & MODERN CONTROL SYSTEMS

FLIGHT CONTROL SYSTEMS



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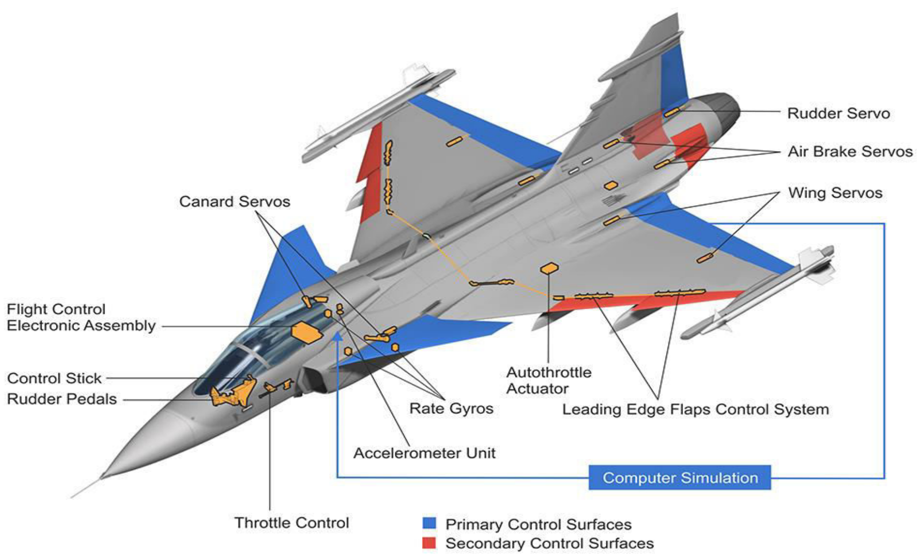
Introduction: The flight controls are the devices and systems that govern the attitude of an aircraft and, as a result, the flight path followed by the aircraft. In the case of many conventional airplanes, the primary flight controls utilize hinged, trailing edge surfaces called elevators for pitch, ailerons for roll, and the rudder for yaw. These surfaces are operated by the pilot in the flight deck or by an automatic pilot the inputs necessary to manipulate the aircraft by the pilot. The flight control systems will vary from aircraft to aircraft but always consists of some combination of primary and secondary flight controls

Flight control systems used by pilot to control the forces of flight and the aircraft's direction and attitude. It should be noted that flight control systems and characteristics can vary greatly depending on the type of aircraft flown. The most basic flight control system designs are mechanical and date back to early aircraft. They operate with a collection of mechanical parts, such as rods, cables, pulleys, and sometimes chains to transmit the forces of the flight deck controls to the control surfaces. Mechanical flight control systems are still used today in small general and sport category aircraft where the aerodynamic forces are not excessive

FLIGHT CONTROL SYSTEMS: Are classified according to the type of Actuation power as follows:

1. Mechanical / Conventional.
2. Hydro mechanical.
3. Fly By Wire.
4. Fly-by-optics – under research
5. Fly-by-wireless – under research
6. Intelligent flight control system – under research



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According to the function they utilized they are classified as:

1. Primary.
2. Secondary.

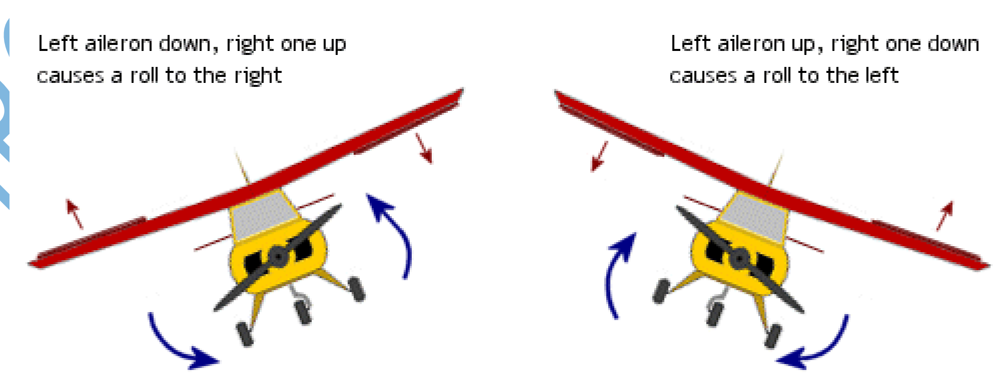
I. **Primary Flight Controls:** Deflection of trailing edge control surfaces, such as the aileron, alters both lift and drag.

1. Ailerons:

i. Ailerons (French for "little wing") are control surfaces on each wing which control the aircraft about its longitudinal axis allowing the aircraft to "roll" or "bank"
   - This action results in the airplane turning in the direction of the roll/bank
   - With aileron deflection, there is asymmetrical lift (rolling moment) about the longitudinal axis and drag (adverse yaw)

ii. They are located on the trailing (rear) edge of each wing near the outer tips
   - They extend from about the midpoint of each wing outward toward the tip, and move in opposite directions to create aerodynamic forces that cause the airplane to roll

iii. The yoke manipulates the airfoil through a system of cables and pulleys and act in an opposing manor
   - Yoke "turns" left: left aileron rises, decreasing camber and angle of attack on the right wing which creates downward lift
     - At the same time, the right aileron lowers, increasing camber (curvature) and angle of attack which increases upward lift and causes the aircraft to turn left
   - Yoke "turns" right: right aileron rises decreasing camber and angle of attack on the right wing which creates downward lift
     - At the same time, the left aileron lowers, increasing camber and angle of attack on the left wing which creates upward lift and causes the aircraft to turn right



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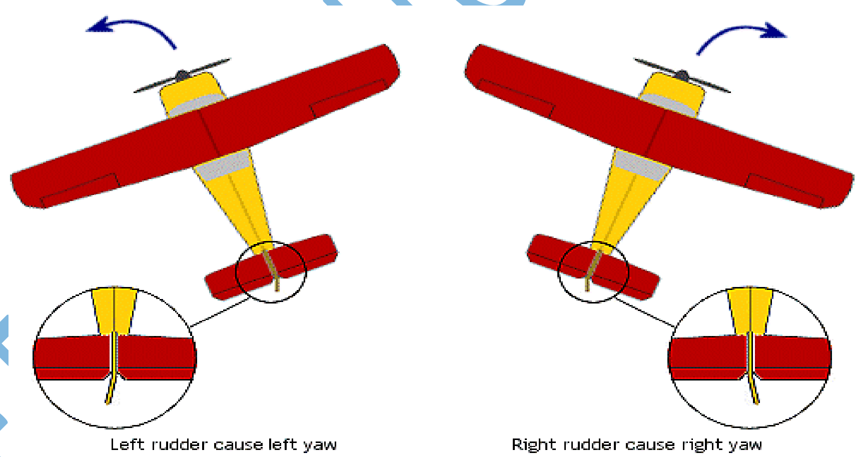
2. Rudder:

i. Rudders are used to control the direction (left or right) of "yaw" about an airplane's vertical axis

ii. Like the other primary control surfaces, the rudder is a movable surface hinged to a fixed surface that, in this case, is the vertical stabilizer, or fin

iii. Its action is very much like that of the elevators, except that it swings in a different plane - from side to side instead of up and down
   - It is not used to make the airplane turn, as is often erroneously believed
   - In practice, both aileron and rudder control input are used together to turn an aircraft, the ailerons imparting roll
     - This relationship is critical in maintaining coordination or creating a slip
     - Improperly ruddered turns at low speed can precipitate a spin

iv. Rudders are controlled by the pilot with his/her feet through a system of cables and pulleys:
   - "Step" on the right rudder pedal: rudder moves right creating a yaw to the right
   - "Step" on the left rudder pedal: rudder moves left creating a yaw to the left



3. Elevators/Stabilators:

i. Elevators and stabilators are both control surfaces which control the aircraft about its lateral axis allowing the aircraft to pitch

ii. Elevators are attached to the trailing edge of the horizontal stabilizer

iii. A stabilator is a combination of both the horizontal stabilizer and the elevator (the entire surface moves)

iv. Used to pitch the aircraft up and down by creating a load on the tail

v. The elevators control the angle of attack of the wings

vi. The yoke manipulates the airfoil through a system of cables and pulleys:
   - Yoke "pulls" back: elevator raises, creating downward lift, raising the nose, increasing the wing's angle of attack

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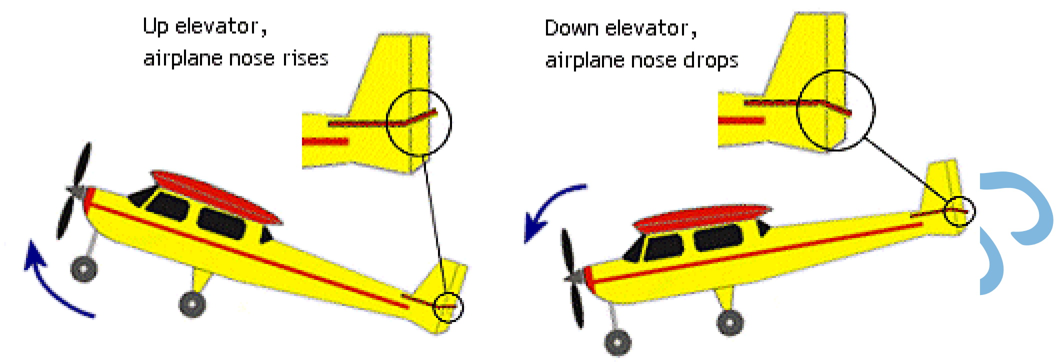
- Yoke "pushes" forward: elevator lowers creating upward lift, lowering the nose, decreasing the wing's angle of attack



II. **Secondary Flight Controls:** Secondary Flight Controls consist of:

i. Flaps:
   - Trailing Edge Flaps
   - Leading Edge Flaps

ii. Trim surfaces

iii. Spoilers/Speed brakes

1. **Flaps:** Flaps allow for the varying of an airfoil's camber. The term, "clean configuration" refers to flaps and gear up. The term, "dirty configuration" refers to flaps and gear down Many attempts have been made to compromise the conflicting requirement of high speed cruise and slow landing speeds
   - High speed requires thin, moderately cambered airfoils with a small wing area
   - The high lift needed for low speeds is obtained with thicker highly cambered airfoils with a larger wing area

   Since an airfoil cannot have two different cambers at the same time, one of two things must be done
   - The airfoil can be a compromise
   - A cruise airfoil can be combined with devices for increasing the camber of the airfoil for low-speed flight (i.e., flaps)

   Flap deflection does not increase the critical (stall) angle of attack, and in some cases flap deflection actually decreases the critical angle of attack. The aircraft stalling speed however (different from angle of attack), will lower. Wing flaps should not induce a roll or yaw effect, and pitch changes depend on the airplane design. Un-commanded roll/yaw with flaps alone could indicate a split flap condition Pitch behavior depends on the aircraft's flap type, wing position, and horizontal tail location This produces a nose-down pitching moment; however, the change in tail load from the down-wash deflected by the flaps over the horizontal tail has a significant influence on the pitching moment Flap deflection of up to 15° primarily produces lift with minimal drag Deflection beyond 15° produces a large increase in drag Drag from flap deflection is parasite drag, and as such is proportional to the square of the speed Also, deflection beyond 15° produces a significant nose-up pitching moment in most high-wing airplanes because the resulting down-wash increases the airflow over the horizontal tail

i. ***Trailing Edge Flaps:***

- Flap operation is used for landings and takeoffs, during which the airplane is in close proximity to the ground where the margin for error is small
  - Since the recommendations given in the AFM/POH are based on the airplane and the flap design combination, the pilot must relate the manufacturer's recommendation to aerodynamic effects of flaps
  - The increased camber from flap deflection produces lift primarily on the rear portion of the wing allowing for decreased approach speed and steeper approach paths
- With this information, the pilot must make a decision as to the degree of flap deflection and time of deflection based on runway and approach conditions relative to the wind conditions
- The time of flap extension and degree of deflection are related and affect the stability of an approach
  - Large flap deflections at one single point in the landing pattern produce large lift changes that require significant pitch and power changes in order to maintain airspeed and glide slope
  - Incremental deflection of flaps on downwind, base, and final approach allow smaller adjustment of pitch and power compared to extension of full flaps all at one time
- The tendency to balloon up with initial flap deflection is because of lift increase, but the nose-down pitching moment tends to offset the balloon
- A soft- or short-field landing requires minimal speed at touchdown
- The flap deflection that results in minimal ground-speed, therefore, should be used
- If obstacle clearance is a factor, the flap deflection that results in the steepest angle of approach should be used
- It should be noted, however, that the flap setting that gives the minimal speed at touchdown does not necessarily give the steepest angle of approach; however, maximum flap extension gives the steepest angle of approach and minimum speed at touchdown
- Maximum flap extension, particularly beyond 30 to 35°, results in a large amount of drag
- This requires higher power settings than used with partial flaps
- Because of the steep approach angle combined with power to offset drag, the flare with full flaps becomes critical
- The drag produces a high sink rate that must be controlled with power, yet failure to reduce power at a rate so that the power is idle at touchdown allows the airplane to float down the runway
- A reduction in power too early results in a hard landing
- Crosswind Considerations:
  - Crosswind component must be considered with the degree of flap extension because the deflected flap presents a surface area for the wind to act on

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- In a crosswind, the "flapped" wing on the upwind side is more affected than the downwind wing
- This is, however, eliminated to a slight extent in the crabbed approach since the airplane is more nearly aligned with the wind
- When using a wing low approach, however, the lowered wing partially blankets the upwind flap, but the dihedral of the wing combined with the flap and wind make lateral control more difficult
- Lateral control becomes more difficult as flap extension reaches maximum and the crosswind becomes perpendicular to the runway
- Crosswind effects on the "flapped" wing become more pronounced as the airplane comes closer to the ground
- The wing, flap, and ground form a "container" that is filled with air by the crosswind
- With the wind striking the deflected flap and fuselage side and with the flap located behind the main gear, the upwind wing will tend to rise and the airplane will tend to turn into the wind
- Proper control position, therefore, is essential for maintaining runway alignment
- Also, it may be necessary to retract the flaps upon positive ground contact
- The go-around is another factor to consider when making a decision about degree of flap deflection and about where in the landing pattern to extend flaps
- Because of the nose-down pitching moment produced with flap extension, trim is used to offset this pitching moment
- Application of full power in the go-around increases the airflow over the "flapped" wing
- This produces additional lift causing the nose to pitch up
- The pitch-up tendency does not diminish completely with flap retraction because of the trim setting
- Expedient retraction of flaps is desirable to eliminate drag, thereby allowing rapid increase in airspeed; however, flap retraction also decreases lift so that the airplane sinks rapidly
- The degree of flap deflection combined with design configuration of the horizontal tail relative to the wing requires that the pilot carefully monitor pitch and airspeed, carefully control flap retraction to minimize altitude loss, and properly use the rudder for coordination
- Considering these factors, the pilot should extend the same degree of deflection at the same point in the landing pattern
- This requires that a consistent traffic pattern be used
- Therefore, the pilot can have a preplanned go-around sequence based on the airplane's position in the landing pattern
- There is no single formula to determine the degree of flap deflection to be used on landing, because a landing involves variables that are dependent on each other

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- The manufacturer's requirements are based on the climb performance produced by a given flap design
- Under no circumstances should a flap setting given in the AFM/POH be exceeded for takeoff

a) Plain Flaps:

- Plain flaps are the most common, but least efficient flap system [***Figure 1***]
- Attached on a hinged pivot, which allows the flap to the moved downward
- The structure and function are comparable to the other control surfaces-ailerons, rudder, and elevator
- When extended, it increases the chord line, angle of attack, and camber of the wing, which results in an increase in both lift and drag
- It is important to remember that control surfaces are nothing more than plain flaps themselves
- They they call same as a wing except it will only stall one wing at a time

b) Split Flap:

- Similar to the plain flap, but more complex [***Figure 1***]
- It is only the lower or underside portion of the wing
- The deflection of the flap leaves the trailing edge of the wing undisturbed
- It is more effective than the hinge flap because of greater lift and less pitching moment, but there is more drag
- More useful for landing, but the partially deflected hinge flaps have the advantage in takeoff
- The split flap has significant drag at small deflections, whereas the hinge flap does not because airflow remains "attached" to the flap

c) Slotted Flap:

- The slotted flap has greater lift than the hinge flap but less than the split flap; but, because of a higher lift-drag ratio, it gives better takeoff and climb performance [***Figure 1***]
- Small deflections of the slotted flap give a higher drag than the hinge flap but less than the split
- This allows the slotted flap to be used for takeoff
- A slotted flap will produce proportionally more lift than drag
- Its design allows high-pressure air below the wing to be directed through a slot to flow over the upper surface of the flap delaying the airflow separation at higher angles of attack
- This design lowers the stall speed significantly

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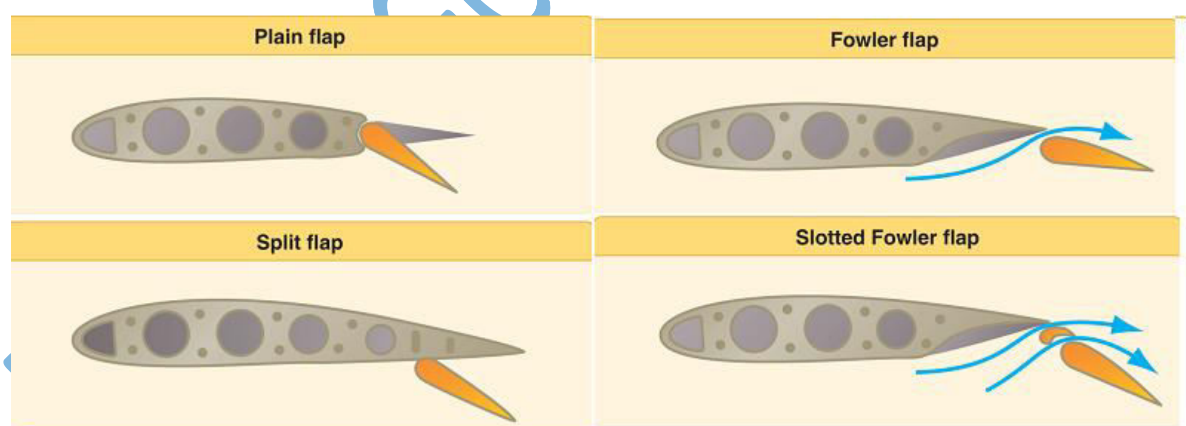
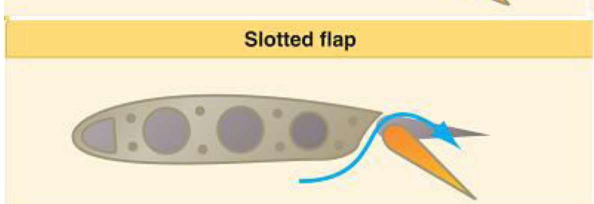
d) Fowler Flap:

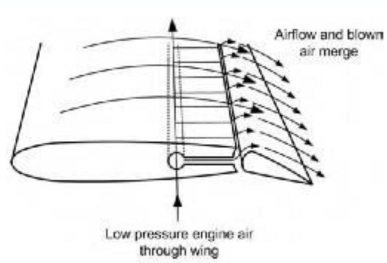
- Most efficient design [***Figure 1***]
- Moves backward on first part of extension increasing lift with little drag; also utilizes a slotted design resulting in lower stall speeds and increased wing area
- Creates the greatest change in pitching moment
- Provides greatest increase in lift coefficient with the least change in drag
- This flap can be multi-slotted making it the most complex of the trailing edge systems
- Drag characteristics at small deflections are much like the slotted flap
- Because of structural complexity and difficulty in sealing the slots, Fowler flaps are most commonly used on larger airplanes

e) Blown Flap:

- An aircraft with wing-mounted propellers, exhibits a blown flap effect
- Provides extra airflow for wings by blowing air over the surfaces
- Prevents boundary layer from stagnating, improving lift
- At low speeds this system can "fool" the airplane into thinking it is flying faster
- Can improve lift 2 or 3 times; however, the bleed air off the engine causes a decrease in thrust for phases of flight such as take off







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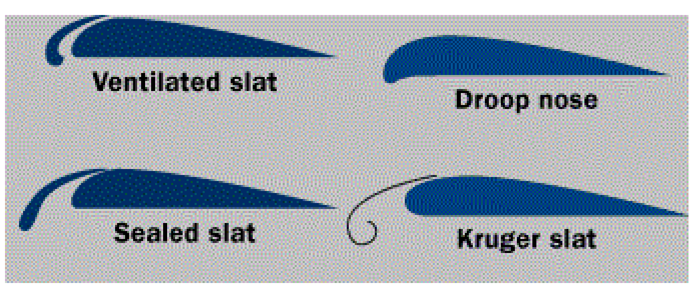
ii. ***Leading Edge Flaps:*** *Leading edge flaps increase stall margin There are several types:*

a) Slats:

- Aerodynamic surfaces on the leading edge of the wings
- When deployed, they allow the wing to operate at a higher angle of attack, so it can fly slower or take off and land in a shorter distance
- Usually used while landing or performing maneuvers, which take the aircraft close to the stall, but are usually retracted in normal flight to minimize drag
- Slats work by increasing the camber of the wing, and also by opening a small gap (the slot) between the slat and the wing leading edge, allowing a small amount of high-pressure air from the lower surface to reach the upper surface, where it helps postpone the stall
- The chord of the slat is typically only a few percent of the wing chord
- They may extend over the outer third of the wing or may cover the entire leading edge
- The slat has a counterpart found in the wings of some birds, the Alula, a feather or group of feathers which the bird can extend under control of its "thumb"
- ***Types of Slat Systems:***
  - Automatic:
    - The slat lies flush with the wing leading edge until reduced aerodynamic forces allow it to extend by way of springs when needed
    - This type is typical on light aircraft
  - Fixed:
    - This slat is permanently extended
    - This is rarely used, except on special low-speed aircraft (these are referred to as slots)
  - Powered:
    - The slat extension can be controlled by the pilot
    - This is commonly used on airliners



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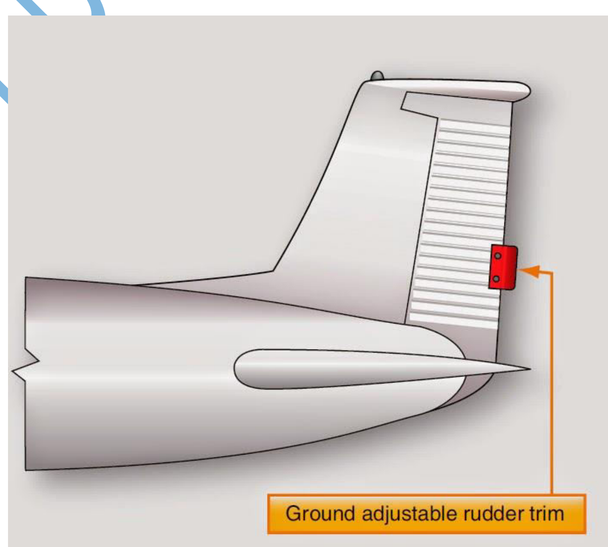
2. Control Surface Tabs:

- Tabs are small, adjustable aerodynamic devices on the trailing edge of the control surface
- These movable surfaces reduce pressures on the controls
- Trim controls a neutral point, like balancing the aircraft on a pin with unsymmetrical weights
-
- This is done either by trim tabs (small movable surfaces on the control surface) or by moving the neutral position of the entire control surface all together
- These tabs may be installed on the ailerons, the rudder, and/or the elevator

a) Trim Tabs:

- The force of the airflow striking the tab causes the main control surface to be deflected to a position that corrects the unbalanced condition of the aircraft
- An aircraft properly trimmed will, when disturbed, try to return to its previous state due to aircraft stability
- Trimming is a constant task required after any power setting, airspeed, altitude, or configuration change
- Proper trimming decreases pilot workload allowing for attention to be diverted elsewhere, especially important for instrument flying
- Trim tabs are controlled through a system of cables and pulleys
    - Trim tab adjusted up: trim tab lowers creating positive lift, lowering the nose
        - This movement is very slight
    - Trim tab adjusted down: trim tab raises creating positive lift, raising the nose
        - This movement is very slight



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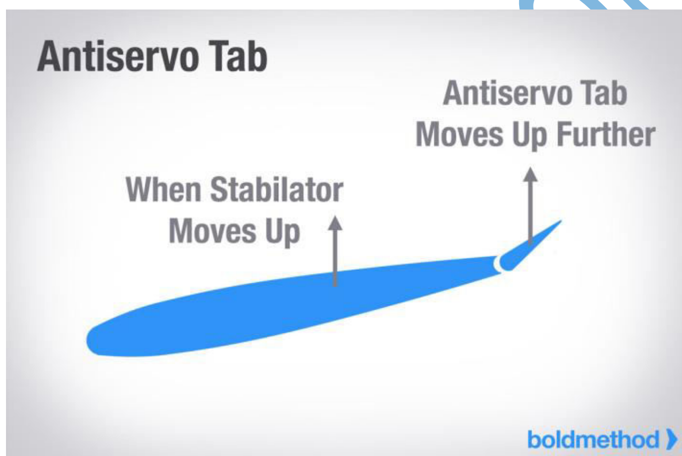
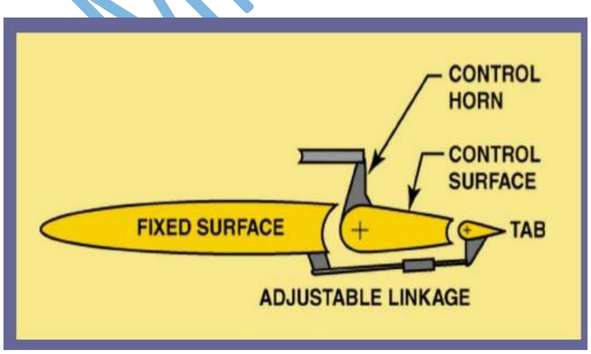
b) Servo Tabs:

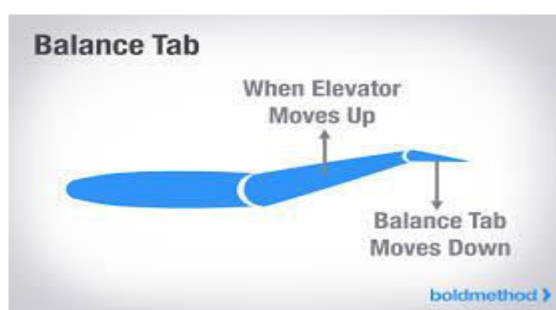
- Servo tabs are similar to trim tabs in that they are small secondary controls which help reduce pilot workload by reducing forces
- The defining difference however, is that these tabs operate automatically, independent of the pilot

  Servo Tab Designs:
  - *Anti-servo:*
    - Also called an anti-balance tab, are tabs that move in the same direction as the control surface
  - *Servo:*
    - Tabs that move in the opposite direction as the control surface



c) Balance Tabs The control forces may be excessively high in some aircraft, and, in order to decrease them, the manufacturer may use balance tabs. They look like trim tabs and are hinged in approximately the same places as trim tabs. The essential difference between the two is that the balancing tab is coupled to the control surface rod so that when the primary control surface is moved in any direction, the tab automatically moves in the opposite direction. The airflow striking the tab counterbalances some of the air pressure against the primary control surface and enables the pilot to move the control more easily and hold the control surface in position.





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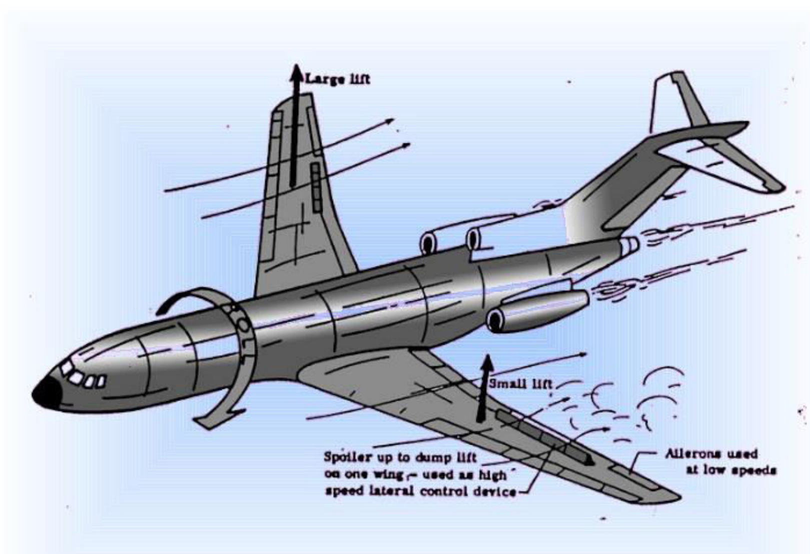
3. **Spoilers:** Found on some fixed-wing aircraft, high drag devices called spoilers are deployed from the wings to spoil the smooth airflow, reducing lift and increasing drag. On gliders, spoilers are most often used to control rate of descent for accurate landings. On other aircraft, spoilers are often used for roll control, an advantage of which is the elimination of adverse yaw. To turn right, for example, the spoiler on the right wing is raised, destroying some of the lift and creating more drag on the right. The right wing drops, and the aircraft banks and yaws to the right. Deploying spoilers on both wings at the same time allows the aircraft to descend without gaining speed. Spoilers are also deployed to help reduce ground roll after landing. By destroying lift, they transfer weight to the wheels, improving braking effectiveness.



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I. **Mechanical / Conventional Control Systems:** The conventional older aircrafts used to employ Push pull rod for moving the control surfaces. This was making the aircraft heavier because the rods used to run along the fuselage to connect the control surfaces. The linkage from cabin to control surface can be fully mechanical if the aircraft size and its flight envelop allow. In this case the hinge moment generated by the surface deflection is low enough to be easily contrasted by the muscular effort of the pilot.

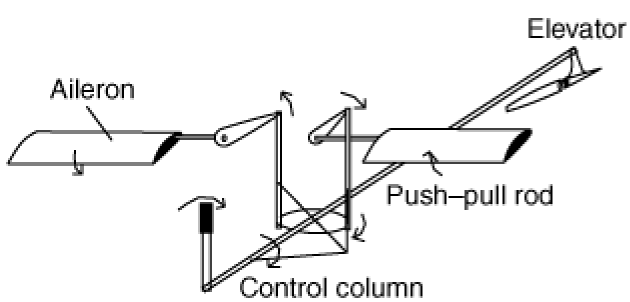
Two types of mechanical systems are used:

1. PUSH-PULL rods and: In this case a sequence of rods links the control surface to the cabin input. Push-pull rod system for elevator control
   Fig. sketches the push-pull control rod system between the elevator and the cabin control column.
2. Cable Pulley:

a) Push-Pull rod:

A stiff rod or hollow tube in an aircraft control system that moves a control surface by either pushing it or pulling it.

- In a push-pull tube system, metal push-pull tubes (or rods) are used as a substitute for the cables .
- Push-pull tubes get their name from the way they transmit force. In the torque tube system, metal tubes (rods) with gears at the ends of the tubes are used. Motion is transmitted by rotating the tubes and gears. On all high-performance aircraft, the control surfaces have great pressure exerted on them.
- At high airspeed, it is physically impossible for the pilot to move the controls manually. As a result, power-operated control mechanisms are used. In a power-operated system, a hydraulic actuator (cylinder)is located within the linkage to assist the pilot in moving the control surface. A typical flight control mechanism is shown in figure 4-12. This is the elevator control of a light weight trainer-type aircraft. It consists of a combination of push-pull tubes and cables.
- The control sticks in the system shown in figure4-12 are connected to the forward sector by push-pull tubes. The forward sector is connected to the aft (rear) sector by means of cable assemblies. The aft sector is connected to the flight control by another push-pull tube assembly.



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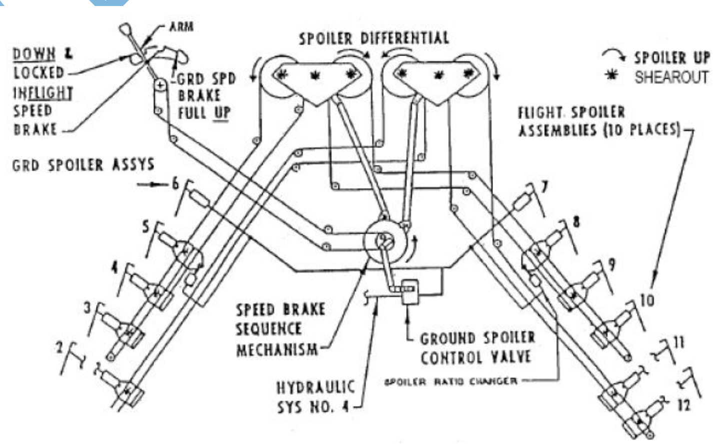
b) **Cable and Pulley system**: The cable and pulley system is widely used for commercial aircraft; sometimes used in conjunction with push–pull control rods.

- Manual control inputs are routed via cables and a set of pulleys from both captain's and first officer's control yokes to a consolidation area in the centre section of the aircraft.
- At this point aileron and spoiler runs are split both left/right and into separate aileron/spoiler control runs. Both control column/control yokes are synchronized.
- A breakout device is included which operates at a predetermined force in the event that one of the cable runs fails or becomes jammed. Control cable runs are fed through the aircraft by a series of pulleys, idler pulleys, quadrants and control linkages in a similar fashion to the push–pull rod system already described.
- Tensiometer / lost motion devices situated throughout the control system ensure that cable tensions are correctly maintained and lost motion eliminated. Differing sized pulleys and pivot/lever arrangements allow for the necessary gearing changes throughout the control runs. Figure 4 shows a typical arrangement for interconnecting wing spoiler and speed brake controls. Trim units, feel units and PCUs are connected at strategic points throughout the control runs as for the push–pull rod system.

Advantage of cable-pulley system

• The same operation described can be done by a cable-pulley system, where couples of cables are used in place of the rods.
• In this case pulleys are used to alter the direction of the lines, equipped with idlers to reduce any slack due to structure elasticity, cable strands relaxation or thermal expansion.
• Often the cable-pulley solution is preferred, because it is more flexible and allows reaching more remote areas of the airplane.
• An example is shown in Fig., where the cabin column is linked via a rod to a quadrant, which the cables are connected to.
• For this reason the actuation system in charge of primary control has a high redundancy and reliability, and is capable of operating close to full performance even after one or more failures.



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II. Hydro Mechanical Control systems: The Complexity and Weight of the system (Mechanical) increased with Size and Performance of the aircraft.

- When the pilot's action is not directly sufficient for the control, the main option is a powered system that assists the pilot.
- The hydraulic system has demonstrated to be a more suitable solution for actuation in terms of reliability, safety, weight per unit power and flexibility, with respect to the electrical system Powered Assisted Control System
- The pilot, via the cabin components, sends a signal, or demand, to a valve that opens ports through which high pressure hydraulic fluid flows and operates one or more actuators.
- The valve, that is located near the actuators, can be signaled in two different ways: mechanically or electrically
- Mechanical signaling is obtained by push-pull rods, or more commonly by cables and pulleys
- Electrical signaling is a solution of more modern and sophisticated vehicles
- The basic principle of the hydraulic control is simple, but two aspects must be noticed when a powered control is introduced:
- The system must control the surface in a proportional way, i.e. the surface response (deflection) must be function to the pilot's demand (stick deflection, for instance)
- The pilot that with little effort acts on a control valve must have a feedback on the maneuver intensity.
- The first problem is solved by using (hydraulic) servo-mechanisms, where the components are linked in such a way to introduce an actuator stroke proportional to the pilot's demand

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- The pilot, in normal hydraulic operating conditions, is requested for a very low effort, necessary to contrast the mechanical frictions of the linkage and the movement of the control valve
- The pilot is then no more aware of the load condition being imposed to the aircraft.
- An artificial feel is introduced in powered systems, acting directly on the cabin control stick or pedals.
- The simplest solution is a spring system, then responding to the pilot's demand with a force proportional to the stick deflection; this solution has of course the limit to be not sensitive to the actual flight conditions.
- A more sophisticated artificial feel is the socalled Q feel. This system receives data from the pitot-static probes, reading the dynamic pressure, or the difference between total (pt) and static (ps) pressure, that is proportional to the aircraft speed v through the air density ρ: This signal is used to modulate a hydraulic cylinder that increases the stiffness in the artificial feel system, in such a way that the pilot is given a contrast force in the pedals or stick that is also proportional to the aircraft speed.

Disadvantages of Mechanical and Hydro-Mechanical Systems

- Heavy and require careful routing of flight control cables through the aircraft using pulleys, cranks, tension cables and hydraulic pipes.
- They require redundant backup to deal with failures, which again increases weight.
- Limited ability to compensate for changing aerodynamic conditions Disadvantages of Mechanical and Hydro-Mechanical Systems
- Dangerous characteristics such as stalling, spinning and pilot-induced oscillation (PIO), which depend mainly on the stability and structure of the aircraft concerned rather than the control system itself, can still occur with these systems
- By using electrical control circuits combined with computers, designers can save weight, improve reliability, and use the computers to mitigate the undesirable characteristics mentioned above. Modern advanced fly-by-wire systems are also used to control unstable fighter aircraft

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III. **FLY BY WIRE:** The fly-by-wire means that in the aircraft, the (pilot or autopilot) control inputs are fed to the (flight) computer, which processes them and determines the required control surface movements and transmits this by electrical signals (through wires, hence the term fly-by-wire) to the appropriate actuators.

The first aircraft to use this mode of control was the Avro Arrow, which used a dual-channel fly-by-wire system made of analog circuits.

Fly-by-wire means that the control commands travel by wire to the control surfaces. There, either electric motors or hydraulic actuators do the physical work of moving the surfaces. The pilot has either a conventional control column or a side stick with transducers which will encode the commands of the pilot into electric signals. Optionally, a computer can be employed to modify the signals such that overload conditions or dangerous maneuvers are avoided.

Actuation:

The conventional linear actuator used in powered flight controls would be of the type show in Figure 2.12. This type of actuator would usually be powered by one of the aircraft hydraulic systems – in this case the blue channel is shown. In functionally critical applications a dual hydraulic supply from another aircraft hydraulic system may be used. A mechanically operated Servo Valve (SV) directs the hydraulic supply to the appropriate side of the piston ram.

As the pilot feeds a mechanical input to the flight control actuator, the summing link will rotate about the bottom pivot, thus applying an input to 20 Flight Control Systems the servo valve. Hydraulic fluid will then flow into one side of the ram while exiting the opposite side resulting in movement of the ram in a direction dependent upon the direction of the pilot's command. As the ram moves, the feedback link will rotate the summing link about the upper pivot returning the servo valve input to the null position as the commanded position is achieved. The attributes of mechanical actuation are straightforward; the system demands a control movement and the actuator satisfies that demand with a power assisted mechanical response. The BAE Hawk 200 is a good example of a system where straightforward mechanical actuation is used for most of the flight control surfaces.

For most applications the mechanical actuator is able to accept hydraulic power from two identical/redundant hydraulic systems. The obvious benefit of this arrangement is that full control is retained following loss of fluid or a failure in either hydraulic system. This is important even in a simple system as the loss of one or more actuators and associated control surfaces can severely affect aircraft handling. The actuators themselves have a simple reversion mode following failure,that is to centre automatically under the influence of aerodynamic forces.

This reversion mode is called aerodynamic centre and is generally preferred for obvious reasons over a control surface freezing or locking at some intermediate point in its travel. In some systems 'freezing'

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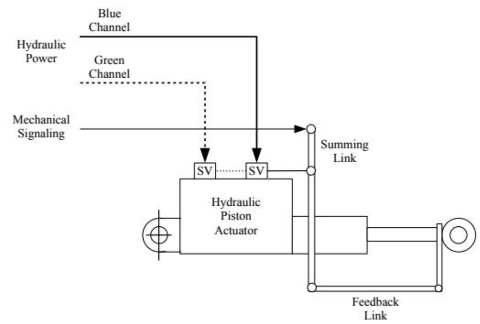

Figure: Conventional linear actuator

the flight control system may be an acceptable solution depending upon control authority and reversionary modes that the flight control system possesses. The decision to implement either of these philosophies will be a design decision based upon the system safety analysis.

Mechanical actuation may also be used for spoilers where these are mechanically rather than electrically controlled. In this case the failure mode is aerodynamic closure, that is the airflow forces the control surface to the closed position where it can subsequently have no adverse effect upon aircraft handling. Figure 1.13 illustrates the mechanical spoiler actuator supplied by

This unit is simplex in operation. It produces thrust of 59.9 kN (13 460 lb) over a working stroke of 15 mm (0.6 inch). It has a length of 22.4 mm (8.8 inch) and weighs 8.3 kg (18.2 lb). The unit accepts hydraulic pressure at 20.7 MN/sqm (3000 psi).

Mechanical Actuation with Electrical Signaling

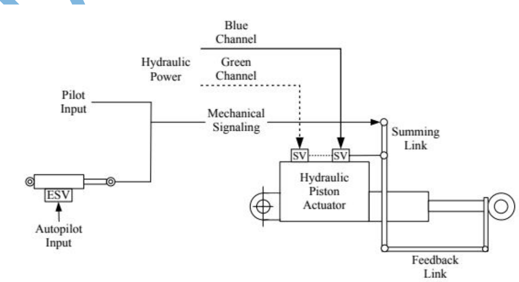

Figure: Conventional linear actuator with autopilot interface

The use of mechanical actuation has already been described and is appropriate for a wide range of applications. However the majority of modern aircraft use electrical signaling and hydraulically powered (electro-hydraulic) actuators for a wide range of applications with varying degrees of redundancy.

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The demands for electro-hydraulic actuators fall into two categories: simple demand signals or autostabilisation inputs.

As aircraft acquired autopilots to reduce pilot work load then it became necessary to couple electrical as well as mechanical inputs to the actuator as shown in Figure 1.14. The manual (pilot) input to the actuator acts as before when the pilot is exercising manual control. When the autopilot is engaged electrical demands from the autopilot computer drive an electrical input which takes precedence over the pilot's demand. The actuator itself operates in an identical fashion as before with the mechanical inputs to the summing link causing the Servo-Valve (SV) to move. When the pilot retrieves control by disengaging the autopilot the normal mechanical link to the pilot through the aircraft control run is restored. Simple electrical demand signals are inputs from the pilots that are signaled by electrical means. For certain noncritical flight control surfaces it may be easier, cheaper and lighter to utilise an electrical link. An example of this is

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The airbrake actuator used on the BAE 146; simplex electrical signaling is used and in the case of failure the reversion mode is aerodynamic closure. In most cases where electrical signaling is used this will at least be duplex in implementation and for fly-by-wire systems signaling is likely to be quadruplex; these more complex actuators will be addressed later. An example of duplex electrical signaling with a simplex hydraulic supply is the spoiler actuators on Tornado. There are four actuators fitted on the aircraft, two per wing, which are used for roll augmentation.

In general, those systems which extensively use simplex electrical signaling do so for autostabilisation. In these systems the electrical demand is a stabilization signal derived within a computer unit. The simplest form of autostabilisation is the yaw damper which damps out the cyclic cross-coupled oscillations which occur in roll and yaw known as 'Dutch roll'. The Hawk 200 illustrated this implementation. Aircraft which require a stable platform for weapon aiming may have simplex autostabilisation in pitch, roll and yaw; an example of this type of system is the Harrier/AV-8A. A similar system on the Jaguar uses simplex autostabilisation in pitch and roll.

1.10.3 Multiple Redundancy Actuation:

Modern flight control systems are increasingly adopting fly-by-wire solutions as the benefits to be realized by using such a system are considerable.

These benefits include a reduction in weight, improvement in handling performance and crew/passenger comfort. Concorde was the first aircraft to pioneer these techniques in the civil field using a flight control system jointly developed by GEC (now Finmeccanica) and SFENA.[3] The Tornado, fly-by-wire Jaguar and EAP have extended the use of these techniques; the latter two were development programmes into the regime of the totally unstable aircraft. In the civil field the Airbus A320 and the Boeing 777 introduced modern state-of-the-art systems into service. For obvious reasons, a great deal of care is taken during the definition, specification, design, development and certification of these systems.

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Multiple redundant architectures for the aircraft hydraulic and electrical systems must be considered as well as multiple redundant lanes or channels of computing and actuation for control purposes. The implications of the redundancy and integrity of the other aircraft systems will be addressed.

For the present, attention will be confined to the issues affecting multiple redundant electro-hydraulic actuations.

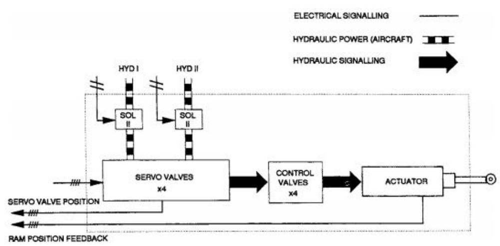

Figure: Simplified block schematic diagram of a multiple redundant electrically signalled hydraulic actuator

A simplified block schematic diagram of a multiple redundant electro-hydraulic actuator is shown in Figure 1.15. For reasons of simplicity only one lane or channel is shown; in practice the implementation is likely to be quadruplex, i.e. four identical lanes. The solenoid valve is energized to supply hydraulic power to the actuator, often from two of the aircraft hydraulic systems. Control demands from the flight control computers are fed to the servo valves. The servo valves control the position of the first-stage valves that are mechanically summed before applying demands to the control valves. The control valves modulate the position of the control ram. Linear variable electrically signaled hydraulic actuator differential transformers (LVDTs) measure the position of the first-stage actuator and output ram positions of each lane and these signals are fed back to the flight control computers, thereby closing the loop. Two examples of this quadruplex actuation system are given below: the Tornado quadruplex taileron and rudder actuators associated with the Control Stability Augmentation System (CSAS) and the EAP flight control system. Both of these systems are outlined at system level in reference [1]. The description given here will be confined to that part of the flight control system directly relevant to the actuator drives. The Tornado CSAS flight control computation is provided by pitch and lateral computers supplied by GEC (now part of Finmeccanica) and Bodenseewerk (now Thales). The pitch computer predominantly handles pitch control computations and the lateral computer roll and yaw computations though there are interconnections between the two (see Figure 1.16a). There are three computing lanes; computing is analogue in nature and there are a number of voter-monitors within the system to vote out lanes operating outside specification. The combined pitch/roll output to the taileron actuators is consolidated from three lanes to four within the pitch computer so the feed to the taileron actuators is quadruplex. The quadruplex taileron actuator is provided by Fairey Hydraulics (now Hamilton Sundstrand) and is shown in Figure 1.16b. This actuator provides a thrust of 339.3 kN (76 291 lb) over a working stroke of 178 mm. The actuator is 940 mm (37.0 in) long and weighs 51.0 kg and operates with the two aircraft 4000 psi hydraulic systems. The rudder actuator similarly receives a quadruplex rudder demand from the lateral computer, also shown in Figure 1.14b. The rudder actuator is somewhat

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smaller than the taileron actuator delivering a thrust of 80.1 kN. The CSAS is designed so that following a second critical failure it is possible to revert to a mechanical link for pitch and roll. In these circumstances the rudder is locked in the central position.

The Tornado example given relates to the analogue system that comprises the CSAS. The EAP flight control system (FCS) is a quadruplex digital computing

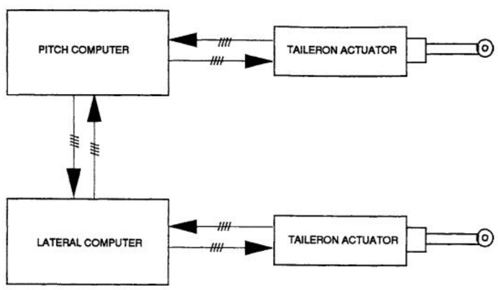

Figure 2.16a: Tornado Taileron/Rudder CSAS drive interface

Figure 2.16b Tornado taileron and rudder actuators (Courtesy of Claverham/Hamilton Sundstrand) system in which control computations are undertaken in all four computing lanes. The system is quadruplex rather than triplex as a much higher level of integrity is required. As has been mentioned earlier the EAP was an unstable aircraft and the FCS has to be able to survive two critical failures. Figure 2.17a shows the relationship between the flight control computers (FCCs), Actuator Drive Units (ADUs) and the actuators. The foreplane actuators are fed quadruplex analogue demands from the quadruplex digital FCCs. Demands for the left and right, inboard and outboard flaperons and the rudder are fed in quadruplex analogue form from the four ADUs. The ADUs receive the pitch, roll and yaw demands from the FCCs via dedicated serial digital links and the digital to analogue conversion is carried out within the ADUs.

The total complement of actuators supplied by Dowty (now GE Aviation) for the EAP is as follows:

- Quadruplex electrohydraulic foreplane actuators: 2
- Quadruplex electrohydraulic flaperon actuators:
  - – outboard flaperons – 100 mm working stroke: 2
  - – inboard flaperons – 165 mm working stroke: 2
- Quadruplex electro hydraulic rudder actuators – 100 mm working stroke: 1

All seven actuators are fed from two independent hydraulic systems. The EAP flight control system represented the forefront of such technology of its time and the aircraft continued to exceed expectations following the first flight in August 1986 until the completion of the programme. Further detail regarding the EAP system and the preceding Jaguar fly-by-wire programme may be found in a number of technical papers which have been given in recent years references [3–8]. Most of these papers are presented from an engineering perspective. The paper by Chris Yeo, Deputy Chief Test Pilot at British Aerospace at the time of the fly-by-wire programme, includes an overview of the aircraft control laws.

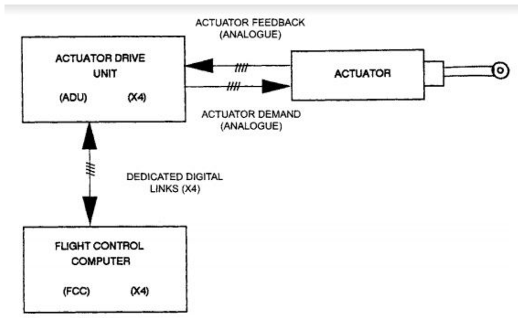

Figure 2.17a: EAP actuator drive configuration

Mechanical Screw jack Actuator

The linear actuators described so far are commonly used to power aileron, elevator and rudder control surfaces where a rapid response is required but the aerodynamic loads are reasonably light.

There are other applications where a relatively low speed of response may be tolerated but the ability to apply or withstand large loads is paramount. In these situations a mechanical screw jack is used to provide a slow response with a large mechanical advantage. This is employed to drive the Tailplane Horizontal Stabilator or Stabilizer (THS), otherwise known years ago as a 'moving tailplane'.

The THS is used to trim an aircraft in pitch as airspeed varies; being a large surface it moves slowly over small angular movements but has to withstand huge loads. The mechanical screw jack shown in Figure 2.18 often has one or two aircraft hydraulic system supplies and a summing link that causes SVs to move in response to the mechanical inputs. In this case the SVs moderate the pressure to hydraulic motor(s) which in turn drive the screw jack through a mechanical gearbox.

As before the left-hand portion of the jack is fixed to aircraft structure and movement of the screw jack ram satisfies the pilot's demands, causing the tailplane to move, altering tailplane lift and trimming the aircraft in pitch.

As in previous descriptions, movement of the ram causes the feedback link to null the original demand, whereupon the actuator reaches the demanded position.

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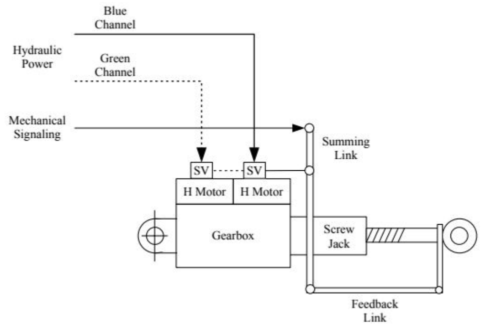

Figure: Mechanical screw jack actuator

Integrated Actuator Package (IAP)

In the UK, the introduction of powerful new AC electrical systems paved the way for the introduction of electrically powered power flying controls. Four channel AC electrical systems utilized on the Avro Vulcan B2 and Handley Page Victor V-Bombers and the Vickers VC10 transport aircraft utilized flight control actuators powered by the aircraft AC electrical system rather than centralized aircraft hydraulic systems.

Figure 2.19 shows the concept of operation of this form of actuator known as an Integrated Actuator Package (IAP). The operation of demand, summing and feedback linkage is similar to the conventional linear actuator already described.

The actuator power or 'muscle' is provided by a three-phase constant speed electrical motor driving a variable displacement hydraulic pump. The hydraulic pump and associated system provides hydraulic pressure to power the actuator ram. The variable displacement hydraulic pump is the hydraulic pressure source for the actuator.

A bi-directional displacement mechanism which is controlled via a servo valve determines the pumps flow and hence actuator velocity. As with the linear actuator, a feedback mechanism nulls off the input to the servo valve as the desired output position is achieved.

Therefore when the actuator is in steady state, the pump displacement is set to the null position but the pump continues to rotate at a constant speed imposing a significant 'windage' power loss which is a significant disadvantage with this design. The more modern integrated actuator designs, specifically the Electro-Hydrostatic Actuator (described later) eliminates this problem.

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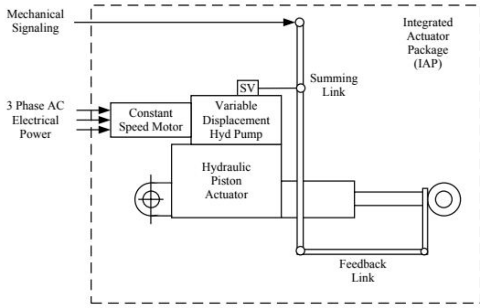

Figure 1.20 depicts an overview of a typical IAP used on the Vickers VC-10 flight control system. A total of 11 such units were used in the VC-10 system to power each of the following flight control surfaces:

• Ailerons: 4 sections

• Elevators: 4 sections

• Rudder: 3 sections

The power consumption of each of the IAPs is in the region of 2.75 kVA and are still flying today in the Royal Air Force's VC-10 Tanker fleet. The units are powered by a constant frequency, split-parallel, 115 VAC three-phase electrical system.

The Avro Vulcan B-2 also used IAP to power the primary flight control surfaces. Being a large delta aircraft this system had an unusual configuration comprising eight elevons powered by IAPs located on the trailing edge of the delta wing plus two on the aircraft rudder. The elevons provided a combined elevator and aileron function to control the aircraft in pitch and roll.

Figure 1.21 Avro Vulcan B-2 FCS architecture using IAPs illustrates how the total complement of ten power flight control units were powered by the four aircraft AC buses. 1.10.6 Advanced Actuation Implementations The actuation implementations described so far have all been mechanical or electro-hydraulic in function using servo valves. There are a number of recent developments that may supplant the existing electro-hydraulic actuator. These newer types of actuation are listed below and have found application in aircraft over the past 10–15 years:

• Direct drive actuation

• Fly-by-Wire (FBW) actuation

• Electro-Hydrostatic Actuator (EHA)

• Electro-Mechanical Actuator (EMA)

Direct Drive Actuation

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In the electro-hydraulic actuator a servo valve requires a relatively small electrical drive signal, typically in the order of 10–15 mA. The reason such low drive currents are possible is that the control signal is effectively amplified within the hydraulic section of the actuator. In the direct drive actuator the aim is to use an electrical drive with sufficient power to obviate the need for the servo valve/1st stage valve. The main power spool is directly driven by torque motors requiring a higher signal current, hence the term 'direct drive'. Development work relating to the direct drive concept including comparison with Tornado requirements and operation with 8 000psi hydraulic systems has been investigated by Fairey Hydraulics .

Fly-By-Wire Actuator

The advent of Fly-By-Wire (FBW) flight control systems in civil aircraft commencing with the Airbus A320 introduced the need for a more sophisticated interface between the FCS and actuation. Most first generation FBW aircraft may operate in three distinct modes that may be summarised in general terms as follows:

• Full FBW Mode. This mode encompasses the full FBW algorithms and protection and is the normal mode of operation

• Direct Electrical Link Mode. This mode will usually provide rudimentary algorithms or possibly only a direct electrical signaling capability in the event that the primary FBW mode is not available

• Mechanical Reversion Mode. This provides a crude means of flying the aircraft probably using a limited number of flight control surface following the failure of FBW and direct electrical link modes.

In later implementations such as the Airbus A380 and Boeing 787 no mechanical reversion is provided. The interface with the actuator is frequently achieved by means of an Actuator Control Electronics (ACE) unit that closes the control loop electrically around the actuator rather than mechanical loop closure as hitherto described (see Figure 1.22). The digital FBW or direct link demands from the flight control system are processed by the ACE which supplies an analogue command to the actuator SV. This allows aircraft systems hydraulic power to be supplied to the appropriate side of the ram piston moving the ram to the desired position. In this implementation the ram position is detected by means of a Linear Variable Differential Transducer (LVDT) which feeds the signal back to the ACE where the loop around the actuator is closed. Therefore ACE performs two functions: conversion of digital flight control demands into analogue signals and analogue loop closure around the actuator.

Electro-Hydrostatic Actuator (EHA) The move towards more-electric aircraft has coincided with another form of electrical actuation – the Electro-Hydrostatic Actuator (EHA) which uses state-of-the-art power electronics and control techniques to provide more efficient flight control actuation. The conventional actuation techniques described so far continually pressurize the actuator whether or not there is any demand.

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In reality for much of the flight, actuator demands are minimal and this represents a wasteful approach as lost energy ultimately results in higher energy offtake from the engine and hence higher fuel consumption. The EHA seeks to provide a more efficient form of actuation where the actuator only draws significant power when a control demand is sought; for the remainder of the flight the actuator is quiescent (see Figure 1.23). The EHA accomplishes this by using the three-phase AC power to feed power drive electronics which in turn drive a variable speed pump together with a constant displacement hydraulic pump. This constitutes a local hydraulic system for the actuator in a similar fashion to the IAP; the difference being that when there is no demand the only power drawn is that to maintain the control electronics. When a demand is received from the ACE the power drive electronics is able to react sufficiently rapidly to drive the variable speed motor and hence pressurize the actuator such that the associated control surface may be moved to satisfy the demand. Once the demand has been satisfied then the power electronics resumes its normal dormant state. Consequently power is only drawn from the aircraft buses bars while the actuator is moving, representing a great saving in energy. The ACE closes the control loop around the actuator electrically as previously described. EHAs are being applied across a range of aircraft and Unmanned Air Vehicle (UAV) developments. The Airbus A380 and Lockheed Martin F-35 Lightning II both use EHAs in the flight control system. For aircraft such as the A380 with a conventional three-phase, 115 VAC electrical system, the actuator uses an in-built matrix converter to convert the aircraft three-phase AC power to 270 VDC to drive a brushless DC motor which in turn drives the fixed displacement pump. The Royal Aeronautical Society Conference, More-Electric Aircraft, 27–28 April 2004, London is an excellent reference for more-electric aircraft and more-electric engine developments where some of these solutions are described.

Aircraft such as the F-35 have an aircraft level 270 VDC electrical system and so the matrix converter may be omitted with further savings in efficiency. Furthermore, electric aircraft/more-electric engine development programmes with civil applications envisage the use of 540 VDC or ±270 VDC systems on the aircraft or engine platform and therefore making similar savings in energy. These developments, including a European Community (EC) funded programme called Power Optimised Aircraft (POA), were described and discussed at the Technologies for Energy Optimised Aircraft Equipment Systems (TEOS) forum in Paris, 28–30 June 2006.

A common feature of all three new actuator concepts outlined above is the use of microprocessors to improve control and performance. The introduction of digital control in the actuator also permits the consideration of direct digital interfacing to digital flight control computers by means of data buses (ARINC 429/ARINC 629/1553B). The direct drive developments described emphasise concentration upon the continued use of aircraft hydraulics as the power source, including the accommodation of system pressures up to 8000 psi. The EMA and EHA developments, on the other hand, lend themselves to a greater use of electrical power deriving from the all-electric aircraft concept, particularly if 270 VDC power is available.

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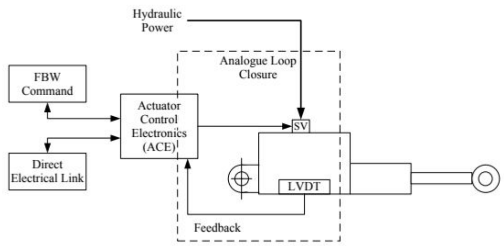

Figure 2.22 : Fly-by-wire actuator

Electro-Mechanical Actuator (EMA)

The electromechanical actuator or EMA replaces the electrical signalling and power actuation of the electro-hydraulic actuator with an electric motor and gearbox assembly applying the motive force to move the ram. EMAs have been used on aircraft for many years for such uses as trim and door actuation; however the power, motive force and response times have been less than that required for flight control actuation. The three main technology advancements that have improved the EMA to the point where it may be viable for flight control applications are: the use of rare earth magnetic materials in 270 VDC motors; high power solid-state switching devices; and microprocessors for lightweight control of the actuator motor.

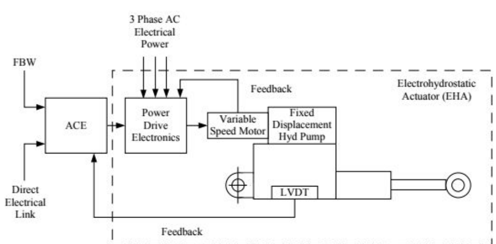

Figure 2.23: Electro-Hydrostatic Actuator (EHA)

As the EHA is the more-electric replacement for linear actuators so the Electro-Mechanical Actuator (EMA) is the more-electric version of the screw jack actuator as shown in Figure 1.24. The concept of the EMA is identical with the exception that the power drive electronics drives a brushless DC motor operating a reduction gear that applies rotary motion allowing the jack ram to extend or retract to satisfy input demands.

EMAs are therefore used to power the THS on civil aircraft and flap and slat drives and also find a use in helicopter flight control systems. A major concern regarding the EMA is the consideration of the actuator jamming case and this has negated their use in primary flight controls on conventional aircraft.

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Actuator Matrix

Most of these actuation types are used in civil aircraft today. Table 1.1 lists how the various actuator types may be used for different actuation tasks on a typical civil airliner.

Table 2.1: Typical applications of flight control actuators

Actuator type	*Power source*	*Primary flight control*	*Spoilers*	*Tailplane horizontal stabilator*	*Flaps and slats*
Conventional Linear Actuator	Aircraft Hydraulic Systems: B/Y/G or L/C/R [1]	X	X		
Conventional Screw-jack Actuator	Aircraft Hydraulic or Electrical Systems [2]			X	X
Integrated Actuator Package (IAP)	Aircraft Electrical System (115VAC)	X	X		
Electrically Signalled Hydraulic Actuator	Aircraft Hydraulic Systems	X	X		
Electro-Hydrostatic Actuator (EHA)	Aircraft Electrical System [3] [4]	X	X		
Electro-Mechanical Actuator (EMA)	Aircraft Electrical System [3]			X	X

Notes: (1) B/Y/G = Blue/Green/Yellow or L/C/R = Left/Centre/Right (Boeing)
(2) For THS and Flaps & Slats both hydraulic and electrical supplies are often used for redundancy
(3) 3-phase VAC to 270 VDC matrix converter used in civil
(4) 270 VDC aircraft electrical system used on F-35/JSF

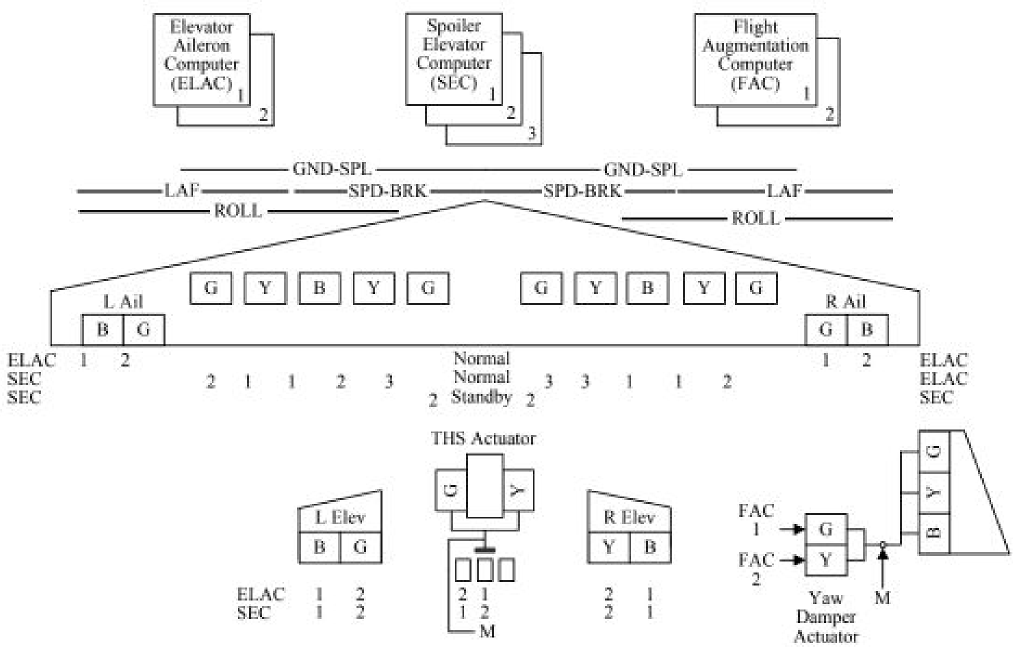

Figure: 2.26: A320 flight control system

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Example: A320 FBW System:

A schematic of the A320 flight control system is shown in Figure 1.26. The flight control surfaces are all hydraulically powered and are tabulated as follows:

• Electrical control:

Elevators 2

Ailerons 2

Roll spoilers 8

Tailplane trim 1

Slats 10

Flaps 4

Speedbrakes 6

Lift dumpers 10

Trims

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• Mechanical control:

Rudder

Tailplane trim (reversionary mode)

The aircraft has three independent hydraulic power systems: blue (B), green (G) and yellow (Y). Figure 1.26 shows how these systems respectively power the hydraulic flight control actuators. A total of seven computers undertake the flight control computation task as follows:

• Two Elevator/Aileron Computers (ELACs). The ELACs control the aileron and elevator actuators according to the notation in the figure

• Three Spoiler/Elevator Computers (SECs). The SECs control all of the spoilers and in addition provide secondary control to the elevator actuators.

The various spoiler sections have different functions as shown namely:

– ground spoiler mode: all spoilers

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– speed brake mode: inboard three spoiler sections

– load alleviation mode: outboard two spoiler sections (plus ailerons); this function has recently been disabled and is no longer embodied in recent models

– roll augmentation: outboard four spoiler sections

• Two Flight Augmentation Computers (FACs). These provide a conventional yaw damper function, interfacing only with the yaw damper actuators

The three aircraft hydraulic systems; blue, green and yellow provide hydraulic power to the flight control actuators according to the notation shown on the diagram. In the very unlikely event of the failure of all computers it is still possible to fly and land the aircraft – this has been demonstrated during certification.

In this case the Tailplane Horizontal Actuator (THS) and rudder sections are controlled directly by mechanical trim inputs – shown as M in the diagram – which allow pitch and lateral control of the aircraft to be maintained.

Another noteworthy feature of the Airbus FBW systems is that they do not use the conventional pitch and roll yoke. The pilot's pitch and roll inputs to the system are by means of a side-stick controller and this has been widely accepted by the international airline community. In common with contemporary civil aircraft, the A320 is not an unstable aircraft like the EAP system briefly described earlier in this chapter. Instead the aircraft operates with a longitudinal stability margin of around 5% of aerodynamic mean chord or around half what would normally be expected for an aircraft of this type. This is sometimes termed relaxed stability. The A320 family can claim to be the widest application of civil FBW with over 3000 examples delivered.

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Trim: The need for trim actuation may be explained by recourse to a simple explanation of the aerodynamic forces which act upon the aircraft in flight. Figure 2.10 shows a simplified diagram of the pitch forces which act upon a stable aircraft trimmed for level flight.

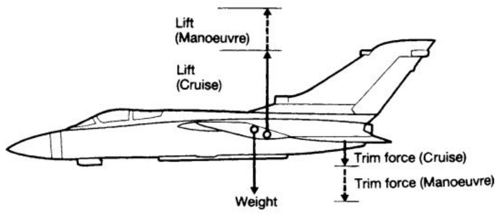

Figure 2.10 Pitch forces acting in level flight

The aircraft weight usually represented by the symbol W, acts downwards at the aircraft centre-of-gravity or CG. As the aircraft is stable the CG is ahead of the 16 centre of pressure where the lift force acts (often denoted by the symbol L) and all aerodynamic perturbations should be naturally damped. The distance between the CG and the centre of pressure is a measure of how stable and also how manoeuvrable the aircraft is in pitch. The closer the CG and centre of pressure, the less stable and more manoeuvrable the aircraft. The converse is true when the CG and centre of pressure are further apart. Examining the forces acting about the aircraft CG it can be seen that there is a counter-clockwise moment exerted by a large lift force acting quite close to the pivot point. If the aircraft is not to pitch nose-down this should be counterbalanced by a clockwise force provided by the tailplane. This will be a relatively small force acting with a large moment. If the relative positions of the aircraft CG and centre of pressure were to remain constant throughout all conditions of flight then the pilot could set up the trim and no further control inputs would be required.

In practice the CG positions may vary due to changes in the aircraft fuel load and the stores or cargo and passengers the aircraft may be carrying. Variations in the position of the aircraft CG position are allowed within carefully prescribed limits. These limits are called the forward and aft CG limits and they determine how nose heavy or tail heavy the aircraft may become and still be capable of safe and controllable flight. The aerodynamic centre of pressure similarly does not remain in a constant position as the aircraft flight conditions vary. If the centre of pressure moves aft then the downward force required of the tailplane will increase and the tailplane angle of incidence will need to be increased. This requires a movement of the pitch control run equivalent to a small nose-up pitch demand. It is inconvenient for the pilot constantly to apply the necessary backward pressure on the control column, so a pitch actuator is provided to alter the pitch control run position and effectively apply this nose-up bias. Forward movement of the centre of pressure relative to the CG would require a corresponding nose-down bias to be applied. These nose-up and nose-down biases are in fact called nose-up and nose-down trim respectively.

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Pitch trim changes may occur for a variety of reasons: increase in engine power, change in airspeed, alteration of the fuel disposition, deployment of flaps or airbrakes and so on. The desired trim demands may be easily input to the flight control system by the pilot. In the case of the Hawk the pilot has a four-way trim button located on the stick top; this allows fore and aft (pitch) and lateral (roll) trim demands to be applied without moving his hand from the control column.

The example described above outlines the operation of the pitch trim system as part of overall pitch control. Roll or aileron trim is accomplished in a very similar way to pitch trim by applying trim biases to the aileron control run by means of an aileron trim actuator. Yaw or rudder trim is introduced by the separate trim actuator provided; in the Hawk this is located in the rear of the aircraft. The three trim systems therefore allow the pilot to offload variations in load forces on the aircraft controls as the conditions of flight vary.

Feel

The provision of artificial 'feel' became necessary when aircraft performance increased to the point where it was no longer physically possible for the pilot to apply the high forces needed to move the flight control surfaces. Initially with servo-boosting systems, and later with powered flying controls, it became necessary to provide powered assistance to attain the high control forces required. This was accentuated as the aircraft wing thickness to chord ratio became much smaller for performance reasons and the hinge moment available was correspondingly reduced. However, a drawback with a pure power assisted system is that the pilot may not be aware of the stresses being imposed on the aircraft. Furthermore, a uniform feel from the control system is not a pleasant characteristic; pilots are not alone in this regard; we are all used to handling machinery where the response and feel are sensibly related. The two types of feel commonly used in aircraft flight control systems are

spring feel

'Q' feel.

Typically the goal is to provide a fairly constant 'Stick force per g' over the full flight envelope. In this regard, the feel system is further complicated with variable geometry aircraft such as the Tornado since aircraft response in pitch and roll varies dramatically with wing sweep. The feel system must therefore take into account both Q and wing sweep.

Spring feel, as the name suggests, is achieved by loading the movement of the flight control run against a spring of a predetermined stiffness. Therefore when the aircraft controls are moved, the pilot encounters an increasing force proportional to the spring stiffness. According to the physical laws spring stiffness is a constant and therefore spring feel is linear unless the physical geometry of the control runs impose any nonlinearities. In the Hawk 200, spring feel units are provided in the tailplane, aileron and rudder control runs. The disadvantage of spring feel units is that they only impose feel proportional to control demand and take no account of the pertaining flight conditions.

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'Q' feel is a little more complicated and is more directly related to the aerodynamics and precise flight conditions that apply at the time of the control demand. As the aircraft speed increases the aerodynamic load increases in a mathematical relationship proportional to the air density and the square of velocity. The air density is relatively unimportant; the squared velocity term has a much greater effect, particularly at high speed. Therefore it is necessary to take account of this aerodynamic equation; that is the purpose of 'Q' feel. A 'Q' feel unit receives air data information from the aircraft pitot-static system. In fact the signal applied is the difference between pitot and static pressure, (known as Pt-Ps) and this signal is used to modulate the control mechanism within the 'Q' feel unit and operate a hydraulic load jack which is connected into the flight control run. In this way the pilot is given feel which is directly related to the aircraft speed and which will greatly increase with increasing airspeed. It is usual to use 'Q' feel in the tailplane or rudder control runs; where this method of feel is used depends upon the aircraft aerodynamics and the desired handling or safety features. The disadvantage of 'Q' feel is that it is more complex and only becomes of real use at high speed.

Figure 2.11 is a photograph of a 'Q' feel unit supplied by Dowty for the BAE Harrier GR5 and McDonnell Douglas AV-8B aircraft.

This unit is fitted with an electrical solenoid so that the active part of the system may be disconnected if required. This unit is designed to operate with an aircraft 20.7 MN/sq m (3000 psi) hydraulic system pressure.

The rudder control run on Hawk 200 shown in Figure 1.6 uses both spring and 'Q' feel. It is likely that these two methods have been designed to complement each other. The spring feel will dominate at low speed and for high deflection control demands. The 'Q' feel will dominate at high speeds and low control deflections.

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Introduction: In early days centrifugal compressor used for controlling fuel to the jet engines engine combustion chamber that used as a fuel pump, a relief valve and a throttle valve.
Later axial compressor and reheat (afterburning) created a demand for more complex methods of controlling airflow, fuel flow and exhaust gas flow.

As engine and materials development continued a need arise to exercise greater control of turbine speeds and temperatures to suit prevailing atmospheric conditions and to achieve surge-free operation. The latter was particularly important in military engines where handling during rapid acceleration tended to place the engine under severe conditions of operation.

In support of the needed improvements, limited authority electronic trimmers so called as **'supervisory controls'** were developed to provide added functions such as **temperature limiting and thrust management** thus relieving the flight crew of this workload. This became important as new aircraft entering service eliminated the flight engineer position on the flight deck. Further developments in engine design led to the need to control more parameters and eventually led to the use of full authority analogue control systems with electrical signaling from the throttle levers.

Engine/Airframe Interfaces

The engine is a major, high value item in any aircraft procurement programme. Often an engine is especially designed for a new military aircraft as per demand specifications. The control of the interfaces between the **engine and the airframe** is essential to allow the airframe contractor and the engine contractor to develop their products independently. The interface may be between the **engine and a nacelle** in the case of a podded, under-wing engine, for commercial aircraft; or between the **engine and the fuselage** for jet military aircraft. Later full authority control systems were introduced in analogue form, semiconductor technology demanded that the electronic control units were mounted on the airframe. This led to a large number of wire harnesses and connectors at the engine–airframe interface. Together with the mechanical, fluid and power offtake interfaces, this was a measure of complexity that had the potential for interface errors that could compromise an aircraft development programme. Although the emergence of rugged electronics, data buses and bleedless engines has simplified this interface, nevertheless it needs to be controlled. What often happens is that an Interface Control Document or ICD is generated that enables the major project contractors to declare and agree their interfaces. The nature of the interfaces and the potential for rework usually means that the ICD becomes an important contractual document.
Typical of the interfaces declared in ICD are the following.

Installation

- **Engine mass, centre of mass and volume**
- **Engine space envelope**
- **Engine clearances under static and dynamic conditions**
- **Attachments**
- **Thrust bearings and fuselage loads**
- **Interface compatibility**
- **Turbine/disc containment measures**
- **Maintenance access points**
- **Drains and vents**

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- **Engine change/winching points**
- **Ground crew intake and exhaust safety clearances**
- **Noise**

System Connections

- **Fuel connections**
- **Control system connections (throttles, reverse thrust command)**
- **Cockpit indications, alerts and warnings**
- **Air start interconnections**
- **Air data requirements**
- **Fire detection and protection**
- **Engine start/relight commands**
- **Engine health monitoring**
- **Ground equipment connections**
- **Inspection access**

Power Off-takes

- **Hydraulic power generation**
- **Electrical power generation**
- **Air bleeds**

Engine Technology and Principles of Operation

The introduction of digital technology and serial data transmission systems, as well as higher performance electronic devices led to the FADEC (Full Authority Digital Electronic Control). This, in turn led to integrate the control systems with the aircraft avionics and flight control systems. When mounting these electronic controls on the engine, great care must be taken to isolate the units from the hostile environment by providing anti-vibration mountings.

Engine technology has advanced considerably with new materials and new manufacturing techniques leading to smaller, lighter and more efficient engines capable of delivering more thrust and more reliable engines. Most of the thrust is generated by the fan since most of the high energy gas from the gas generator section is dissipated in the turbine connected to the fan. In the turbofan engine, thrust is generated by imparting a relatively small increase in velocity to a very large air mass flow through the fan while, in the older turbojet engines, the total air mass flow through the engine is much smaller and therefore, to achieve the same thrust, the velocity of the exhaust gasses must be much greater, i.e.

• Turbofan: Thrust $= \dot{\mathbf{M}} \times \mathbf{v}$ Large mass flow, small velocity change
• Turbojet: Thrust $= \dot{\mathbf{m}} \times \mathbf{V}$ Small mass flow, large velocity change

It is for this reason that today's large fan engines are much quieter than their turbojet or low by-pass ratio Engines.

The Control Problem

The basic control action is to control fuel flow and air to the engine to allow it to operate at its optimum efficiency over a wide range of forward **speeds, altitudes and temperatures** while allowing the pilot to handle the engine without fear of malfunction. The military aircraft is usually specified to operate in worldwide conditions, and is expected to experience a wide range of operating temperatures. To be successful in combat the aircraft must be maneuverable. The pilot, therefore, expects to be able to demand minimum or maximum power with optimum acceleration rates, as well as to make small

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adjustments with equal ease, without fear of surge, stall, flame-out, over-speed or over-temperature. The pilot also needs a fairly linear relationship between throttle lever position and thrust. The civil pilots requires reliable, economical and long-term operation under clearly defined predictable conditions with minimum risk to passengers and schedules. To obtain these objectives, control can be exercised over the following aspects of engine control:

• *Fuel flow* – to allow varying engine speeds to be demanded and to allow the engine to be handled without damage by limiting rotating assembly speeds, rates of acceleration and temperatures.
• *Air flow* – to allow the engine to be operated efficiently throughout the aircraft flight envelope and with adequate safety margins.
• *Exhaust gas flow* – by burning the exhaust gases and varying the nozzle area to provide additional thrust.

Electronic control has been applied in all these above cases with varying degrees of complexity and control authority. Such control can take the form of simple limiter functions through to sophisticated multi-variable, full authority control systems closely integrated with other aircraft systems.

Fuel Flow Control Control of power or thrust is achieved by regulating the fuel flow into the combustor. On turbo jet or turbo fan engines thrust can be controlled by setting an engine pressure ratio or, in the case of the larger commercial fan engines, by controlling fan speed, while on shaft power engines the speed of the gas generator is a measure of the power delivered to the propeller or to the rotor. When changing the thrust or power setting the fuel control system must limit the rate of acceleration and deceleration of the engine rotating assemblies in order to prevent compressor surge or flame out. This control process is further complicated by the change in engine inlet conditions, i.e. inlet temperature, inlet pressure and Mach number that can occur as the aircraft moves around the flight envelope. Airflow modulation through the compressor may also be necessary by the use of variable vanes and/or bleed valves to provide adequate surge margin under all operating conditions.

The control of power or thrust of the gas turbine engine is obtained by regulating the quantity of fuel injected into the combustion system. When a higher thrust is required the throttle is opened and the fuel pressure to the burners increases due to the higher fuel flow. This has the effect of increasing the gas temperature which, in turn, increases the acceleration of the gases through the turbine to give a higher engine speed and correspondingly greater air flow, resulting in an increase in thrust.

The relationship between the air flow induced through the engine and the fuel supplied is, however, complicated by changes in altitude, air temperature and aircraft speed. To meet this change in air flow a similar change in fuel flow must occur, otherwise the ratio of air to fuel will alter and the engine speed will increase or decrease from that originally selected by the pilot in setting the throttle lever position. Fuel flow must, therefore, be monitored to maintain the conditions demanded by the pilot whatever the changes in the outside world.

Failure to do so would mean that the pilot would constantly need to make minor adjustments to throttle lever position, increasing his work load and distracting his attention from other aspects of aircraft operation. The usual method of providing such control is by means of a fuel control unit (FCU) or fuel management units (FMU). The FCU/FMU is a hydro chemical device mounted on the engine. It is a complex engineering mechanism containing valves to direct fuel and to restrict fuel flow, pneumatic

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capsules to modify flows according to prevailing atmospheric conditions. The engine speed must be controlled from idle to maximum rating. Over speed must be avoided to reduce stresses in the rotating assemblies, and over temperature must be avoided to prevent blade damage and to reduce thermal creep. The engine must be allowed to accelerate and decelerate smoothly with no risk of surge.

Such control influences are difficult to achieve manually. Therefore the FCU has, over the generations of jet engines, been designed to accommodate control inputs from external electronic devices. Electrical valves in the FCU can be connected to electronic control units to allow more precise and continuous automatic control of fuel flows in response to throttle demands, using measurements derived from the engine, to achieve steady state and transient control of the engine without fear of malfunction.

A typical fuel control circuit is shown in Figure On some military aircraft the fuel system receives a demand from the weapon release switch or gun trigger to preempt weapon release. This allows fuel flow to the engines to be modified to prevent an engine surge resulting from disturbance of the intake conditions from missile exhaust, shock from the gun muzzle or smoke from the gun breech. This facility is known as '**fuel dip'**.

Air Flow Control

It is sometimes necessary to control the flow of air through to the engine to ensure efficient operation over a wide range of environmental and usage conditions to avoid **engine surge**. Most modern commercial engines have variable compressor vanes and/or bleed valves to provide optimum acceleration without surge. The figure shows air is bled from the engine for various purposes, including engine stability reasons and also to provide a source of air for conditioning systems and bleed air systems such as wing leading edge anti-icing.

The number of variables that affect engine performance is high and the nature of the variables is dynamic, so that the pilot cannot be expected constantly to adjust the throttle lever to compensate for changes, particularly in multiengine aircraft.

A throttle movement causes a change in the fuel flow to the combustion chamber spray nozzles. This, in turn, causes a change in **engine speed and in exhaust gas temperature.** Both of these parameters are measured; engine speed by means of a gearbox mounted speed probe and Exhaust Gas Temperature (EGT), or Turbine Gas Temperature (TGT), by means of thermocouples, and presented to the pilot as analogue readings on cockpit-mounted indicators. The FCU, with its internal capsules, looks after variations due to atmospheric changes. In the dynamic conditions of an aircraft in flight at different altitudes, temperatures and speeds, continual adjustment by the pilot soon becomes impractical. He cannot be expected continuously to monitor the engine conditions safely for a flight of any significant duration. For this reason some form of automatic control is essential.

Control System Parameters

To perform any of the control functions electrically requires devices to sense engine operating conditions and to perform a controlling function. These can usually be conveniently subdivided into input and output devices producing input and output signals to the control system. To put the control problem into perspective the control system can be regarded as a box on a block diagram receiving input signals from the aircraft and the engine and providing outputs to the engine and the aircraft systems. This system is shown diagramatically in Figure 2.5.

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The input signals provide information from the aircraft and the engine to be used in control algorithms, while the output signals provide the ability to perform a control function. Further signals derived from output devices provide feedback to allow loop closure and stable control. Typical inputs and outputs are described below.

Input Signals

• *Throttle position* – A transducer connected to the pilot's throttle lever allows thrust demand to be determined. The transducer may be connected directly to the throttle lever with electrical signalling to the control unit, or connected to the end of control rods to maintain mechanical operation as far as possible. The transducer may be a potentiometer providing a DC signal or a variable transformer to provide an AC signal. To provide suitable integrity of the signal a number of transducers will be used to ensure that a single failure does not lead to an uncommanded change in engine demand.
• *Air data* – Airspeed and altitude can be obtained as electrical signals representing the pressure signals derived from airframe mounted capsule units. These can be obtained from the aircraft systems such as an air data computer (ADC) or from the flight control system air data sensors
• *Total temperature* – A total temperature probe mounted at the engine face provides the ideal signal. Temperature probes mounted on the airframe are usually provided, either in the intakes or on the aircraft structure
• *Engine speed* – The speed of rotation of the shafts of the engine is usually sensed by pulse probes located in such a way as to have their magnetic field interrupted by moving metallic parts of the engine or gearbox. The blades of the turbine or compressor, or gear box teeth, passing in front of a magnetic pole piece induce pulses into a coil or a number of coils wound around a magnet. The resulting pulses are detected and used in the control system as a measure of engine speed
• *Engine temperature* – The operating temperature of the engine cannot be measured directly since the conditions are too severe for any measuring device. The temperature can, however, be inferred from measurements taken elsewhere in the engine. The traditional method is to measure the temperature of the engine exhaust gas using thermocouples protruding into the gas stream. The thermocouples are usually arranged as a ring of parallel connected thermocouples to obtain a measurement of mean gas temperature and are usually of chromel-alumel junctions. A cold junction is provided to obtain a reference voltage. An alternative method is to measure the temperature of the turbine blades with an optical pyrometer. This takes the form of a fiber optic with a lens mounted on the engine casing and a semiconductor sensor mounted in a remote and cooler environment. Both of these temperatures can be used to determine an approximation of turbine entry temperature, which is the parameter on which the temperature control loop should ideally be closed
• *Nozzle position* – For those aircraft fitted with reheat (or afterburning) the position of the reheat nozzle may be measured using position sensors connected to the nozzle actuation mechanism or to the nozzle itself. An inductive pick-off is usually used since such types are relatively insensitive to temperature variations, an important point because of the harsh environment of the reheat exhaust
• *Fuel flow* – Fuel flow is measured by means of a turbine type flow meter installed in the fuel pipe work to obtain a measure of fuel inlet flow as close to the engine as possible. Fuel flow measured by the turbine flow meter is for instrumentation and monitoring purposes and is not used as an input to the engine control system. The dynamic response of this device is much too slow for this function. Instead the position of the fuel metering valve within the FCU is used as a measure of fuel flow

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• *Pressure ratio* – The ratio of selected pressures between different stages of the engine can be measured by feeding pressure to both sides of a diaphragm operated device. The latest technology pressure ratio devices use two high accuracy pressure sensors and electronics to generate pressure ratio

Output Signals

• *Fuel flow control* – The fuel supply to the engine can be varied in a number of ways depending on the type of fuel control unit used. Solenoid operated devices, torque motor or stepper motor devices have all been employed on different engine types. Each device has its own particular failure modes and its own adherents
• *Air flow control* – The control of air flow at different stages of the engine can be applied by the use of guide vanes at the engine inlet, or by the use of bleed valves between engine stages. These are controlled automatically to preserve a controlled flow of air through the engine for varying flight conditions

FADC Systems Example

Using various combinations of input and output devices to obtain information from the engine and the airframe environment, a control system can be designed to maintain the engine conditions stable throughout a range of operating conditions. The input signals and output servo demands can be combined in varying degrees of complexity to suit the type of engine, the type of aircraft, and the manner in which the aircraft is to be operated. Thus the systems of civil airliners, military trainers and high speed combat aircraft will differ significantly.

In a simple control system, such as may be used in a single engine trainer aircraft the primary pilot demand for thrust is made by movements of a throttle lever. Rods and levers connect the throttle lever to a fuel control unit (FCU) so that its position corresponds to a particular engine condition, say rpm or thrust. Under varying conditions of temperature and altitude this condition will not normally stay constant, but will increase or decrease according to air density, fuel temperature or demands for take-off power. To obtain a constant engine condition, the pilot would have continually to adjust the throttle lever, as was the case in the early days of jet engines. Such a system with the pilot in the loop is shown in Figure 2.6.

The flow of fuel to the combustion chambers can be modified by an electrical valve in the FCU that has either an infinitely variable characteristic, or moves in a large number of discrete steps to adjust fuel flow. This valve is situated in the engine fuel feed line so that flow is constricted, or is by-passed and returned to the fuel tanks, so that the amount of fuel entering the engine is different from that selected. This valve forms part of a servo loop in the control system so that continuous small variations of fuel flow stabilize the engine condition around that demanded by the pilot. This will allow the system to compensate for varying atmospheric and barometric conditions, to ensure predictable acceleration and deceleration rates and to prevent over-temperature or over-speed conditions occurring over the available range – acting as a range speed governor; Figure 2.7 illustrates such a control system. It can be seen that the pilot shown in Figure 2.6 now acts in a supervisory role, relying on the control system to maintain basic control conditions while he monitors the indicators for signs of over-speed or over-temperature.

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Even this task can be reduced considerably by incorporating an automatic means of signalling an over-speed or over-temperature. This can be performed in the control unit by setting a datum related to a particular engine type, or by setting a variable 'bug' on the cockpit indicator. If either preset datum is exceeded a signal is sent to the aircraft warning system to warn the pilot by means of a red light and signal tone (see Chapter 9). This principle is illustrated in Figure 2.8 which shows warning systems for both over-temperature and over-speed conditions. In this diagram the over-speed warning is provided by a mechanism in the turbine gas temperature (TGT) indicator.

A knob on the indicator allows the pilot to set a 'bug' to a particular temperature. When the indicator pointer exceeds that setting, a pair of contacts in the indicator closes and provides a signal to the aircraft central warning system. The over-speed warning is provided by a pair of contacts in the engine control unit. In practice either one method or the other is used in one aircraft type, rather than a mix of methods.

In many modern aircraft the simple throttle signaling system is retained, but with the replacement of rods and levers by electrical signaling from the throttle levers. This reduces friction and eliminates the possibility of jamming in the control rod circuit. An example of a system with electrical throttle signaling is illustrated in Figure 2.9. The removal of any mechanical links between the pilot and the engine means that the control unit has full authority control.

There is nothing the pilot can do to correct an engine malfunction other than to shut down the engine. Because of this the throttle signaling circuit (like the rest of the control system) is designed with great care to ensure that all failures are detected and taken care of by the control system. For example, additional windings on the Tornado throttle position transducer enable the control system to detect open circuits and short circuits and to take corrective action.

Engine Starting

To start the engines a sequence of events is required to allow fuel flow, to rotate the engine and to provide ignition energy. For a particular type of aircraft this sequence is unvarying, and can be performed manually with the pilot referring to a manual to ensure correct operation, or automatically by the engine control unit. Before describing a typical sequence of events, an explanation of some of the controls will be given.

Fuel Control

Fuel from the tanks to the engine feed line is interrupted by two shut-off cocks. The first is in the low pressure feed lines, at which fuel pressure is determined by the fuel boost pumps (see Chapter 3 – Fuel Systems). The valve, known as the LP cock or firewall shut-off cock, is situated close to the engine firewall. Its primary purpose is to isolate the engine in the event of a fire. It is usually a motor-driven valve controlled by a switch in the cockpit and, once opened, cannot be shut except by means of the switch. The switch is usually covered by a guard so that two actions are needed to select the switch to either open or close the cock.

This helps to prevent inadvertent actions that may lead to accidental engine shutdown. The second valve is in the high-pressure fuel line, in which the fuel pressure is determined by an engine-driven pump. The function of this valve is to open and close the fuel feed close to the engine inlet at the fuel control unit. It is opened manually by the pilot, or automatically by the engine control unit at an appropriate stage in the engine start cycle. The location of these valves is shown in Figure

Ignition Control

The ignition system consists of a high energy ignitor which is switched on for a period during the start cycle. The ignitors initiate combustion of the fuel vapor in the combustion chamber. An ignitor plug is supplied with electrical energy by an ignition exciter that produces stored energy from 1 to 6 joules depending on the type required. High energy systems are used for starting, and low energy systems can be provided to maintain engine ignition during aircraft operations in heavy rain, slushy runways or icing conditions. A typical ignition circuit is shown in Figure 2.19 and some examples of typical ignition equipment are shown in Figure.

Engine Rotation

During the starting cycle the engine needs to be rotated until the fuel has ignited and the temperature of combustion is sufficient for the engine to rotate without assistance. At this point the engine is said to be self-sustaining. A number of methods are in current use for providing assistance by means of air, electrical energy or chemical energy. The most common method in modern use is to use an external air source or an internal auxiliary power unit to start the first engine and to cross drive starts the remaining engines. Some smaller engines and the More-Electric B787 use electrical engine start. Air at high pressure can be provided by an external air compressor trolley connected to the engine by ground crew, or by air supplied by an onboard

Auxiliary Power Unit (APU). This is a small gas turbine that is started prior to engine start. It has the advantage of making the aircraft independent of ground support and is useful at remote airfields. It is also used to provide electrical and hydraulic energy for other aircraft services. An example APU is shown in Figure 2.21, this has its own intake concealed beneath an opening hatch and its own exhaust positioned so as not to present a hot gas hazard to ground crew.

Throttle Levers

The throttle lever assembly is often designed to incorporate HP cock switches so that the pilot has instinctive control of the fuel supply to the engine. Microswitches are located in the throttle box so that the throttle levers actuate the switches to shut the valves when the levers are at their aft end of travel. Pushing the levers forward automatically operates the switches to open the fuel cocks, which remain open during the normal operating range of the levers. Two distinct actions are required to actuate the switches again. The throttle lever must be pulled back to its aft position and a mechanical latch operated to allow the lever to travel further and shut off the fuel valve.

Starting Sequence

A typical start sequence is:

- Open LP cocks
- Rotate engine
- Supply ignition energy
- Set throttle levers to idle – open HP cocks

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• When self-sustaining – switch off ignition
• Switch off or disconnect rotation power source

Together with status and warning lights to indicate 'start in progress', 'failed start' and 'engine fire' the pilot is provided with information on indicators of engine speeds, temperatures and pressures that he can use to monitor the engine start cycle.

In many modern aircraft the start cycle is automated so that the pilot has only to select START for the complete sequence to be conducted with no further intervention. This may be performed by an aircraft system such as Vehicle Management, or by the FADEC control unit. In future this sequence may be initiated by an automated pre-flight check list.

Engine Indications

Despite the fact that engine control systems have become very comprehensive in maintaining operating conditions at the most economic or highest performance, depending on the application, there is still a need to provide the pilot with an indication of certain engine parameters.
Under normal conditions the pilot is interested in engine condition only at the start and when something goes wrong. The engine control system, with its monitoring and warning capability, should inform the pilot when something untoward does happen. However, there may be circumstances when human intuition wins the day.

During engine start the pilot monitors (and checks with his co-pilot in a multi-crew aircraft) that start progresses satisfactorily with no observed sluggish accelerations, no low oil pressures or over-temperatures. Much of this monitoring involves pilot familiarity with the aircraft type and engine type, incurred over many starts. The crew may accept certain criteria that an automatic system would not. During normal operation the control system should provide sufficient high integrity observation by self-monitoring and by checking certain parameters against preset values. In this way the system can monitor accelerations, rates of change, value exceedance and changes of state and issue the necessary warning.

Control systems are good at detecting sudden changes of level or state. However, slow, gradual but persistent drift and transient or intermittent changes of state are a designer's nightmare. The first may be due to degradation in performance of a component, e.g. a component becoming temperature sensitive, a gradually blocking filter or the partial occlusion of a pipe or duct.

The second may be due to a loose connection some where in the system. The pilot can observe the effects of these circumstances. In a four-engine aircraft, for example, one indicator reading differently to three others can be easily seen because the indicators are grouped with just such a purpose in mind.
Until recently all aircraft had at least one panel dedicated to engine instruments. These were in view at all times and took the form of circular pointer instruments, or occasionally vertical strip scales, reading such parameters as:

• Engine speed – NH and NL
• Engine temperature
• Pressure ratio

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• Engine vibration
• Thrust (or torque)

In modern aircraft cockpits the individual indicator has largely given way to the Multi-Function Display Unit (MFDU). With a MFDU any information can be shown in any format, in full color, at any time. This facility is often exploited to ensure that the pilot is only given the information that is essential for a particular phase of flight. This means that engine displays may occur on a single screen or page that is automatically presented to the pilot at certain times, say starting, take-off and landing, but is hidden at all other times. Provision is made for the pilot to select any page so that he can check from time to time, and an engine warning may automatically trigger the engine page to appear.

Engine indications are obtained from the same type of sensors and transducers that provide the inputs to the control system, as described earlier. However, for integrity reasons at least two sources of signal are required – one (or more) for control, another for the indicator. For example the engine rpm signal will be obtained from two separate coils of a speed sensor. These guards against a common mode failure that would otherwise affect both the control system and the indication system.

Such systems are the Engine Indication and Crew Alerting System (EICAS) used on Boeing and other aircraft and the Electronic Crew Alerting and Monitoring (ECAM) on Airbus aircraft.
Some examples of engine synoptic displays are shown in Figure 2.24 and the Trent 800 indication circuit is shown in Figure 2.25.

Engine Off takes

The engine is the prime mover for the majority of sources of power on the aircraft. An accessory gearbox enables accessories to be connected to the engine HP shaft and allows a starter connection so that the engine can be started from an external supply or from the Auxiliary Power Unit (APU). It is also a convenient place to obtain measurement of engine rotational speed by measuring the speed of rotation of the gearbox using a tachometer or pulse probe. An example accessory gearbox is shown in Figure 2.27.
Typical services, shown in Figure 2.28, include:
• Electrical power from generators
• Hydraulic power from hydraulic pumps
• Cabin and equipment conditioning system air from engine bleed
• Pneumatic power
• Anti and/or de-icing system air
It can be seen that many of the drives off the accessory gearbox are for the use of the engine:
• LP and HP fuel pumps
• Oil scavenge pumps; oil is used to cool the electrical generator as well as lubricate the engine
• PMAs to supply 28 VDC power for the dual channel FADEC
• Oil breather
Interfaces with the aircraft include:
• Supply of three-phase 115 VAC, 400 Hz electrical power – rated in the range from 40 to 90 kVA per channel on most civil transport aircraft; 120 kVA per

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channel on B777 and B767-400
• Supply of 3000 psi hydraulic power
• Engine tachometer and other engine indications

Input of bleed air from a suitable air source to start the engine. This can be a ground power cart, the APU or air from the other engine if that has already been started.

Reverse Thrust

A mechanism is provided on most engines to assist in decelerating the aircraft. On a turbo-prop engine this mechanism is to apply reverse pitch to the propeller blades. On a turbo-fan engine the usual mechanism is to deploy spoilers or buckets into the exhaust gas stream. Both of these methods have the effect of reversing the thrust provided by the engines to assist the brakes and shorten the landing distance.

Reverse thrust is commanded by the crew by a mechanism in the throttle levers, usually by pulling the levers back to idle, selecting reverse thrust and then increasing the throttle lever position towards maximum to achieve the required braking effect. The effect is often combined with lift dumping, in which air brakes and spoilers are deployed at the same time to provide a combined deceleration effect.

The thrust reverser circuit must be designed to prevent inadvertent operation in the air, and usually combined interlocks between throttle position, reverser selection and main wheel

Engine Control on Modern Civil Aircraft

Most commercial aircraft engines are twin shaft engines with LP and HP shafts. Some Rolls-Royce engines such as the RB211 and Trent family are triple shaft engine with LP, IP and HP shafts. A high proportion of air by-passes the engine core on a modern gas turbine engine; the ratio of bypass air to engine core air is called the by-pass ratio. The by-pass ratio for most civil engines is in the ratio of 4:1 to 5:1. The Rolls-Royce Trent engine is shown in Figure 2.30 as an example of a modern high bypass ratio engine for the modern generation of commercial airliners. Further views of the engine are shown in Figures 2.31 and 2.32. Most modern civil engines use a Full Authority Digital Engine Control System (FADEC), mounted on the fan casing to perform all the functions of powerplant management and control.
The key areas of monitoring and control are:
• Various speed probes (N1, N2); temperature and pressure sensors (P2/T2, P2.5/T2.5, and T3); Exhaust Gas Temperature (EGT) and oil temperature and pressure sensors are shown
• The turbine case cooling loops – High Pressure (HP) and Low Pressure (LP)
• Engine start
• Fuel control for control of engine speed and, therefore, thrust

The engine Permanent Magnet Alternators (PMAs) are small dedicated generators that supply primary power on the engine for critical control functions

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• Various turbine blade cooling, Inlet Guide Vanes (IGVs), Variable Stator Vanes (VSVs) and bleed air controls

Characteristics of Fuel Systems

The purpose of an aircraft fuel system is primarily to provide a reliable supply of fuel to the engines. Without the motive power provided by them the aircraft is unable to sustain flight. Therefore the fuel system is an essential element in the overall suite of systems required to assure safe flight. Modern aircraft fuels are hydrocarbon fuels similar to those used in the automobile. Piston engine aircraft use a higher octane fuel called AVGAS in aviation parlance.

Jet engines use a cruder fuel with a wider distillation cut and with a lower flashpoint. AVTAG and AVTUR are typical jet engine fuels. The specific gravity of aviation fuels is around 0.8, that is about eight-tenths of the density of water. Therefore fuel may be quantified by reference to either volume (gallons or liters) or weight (pounds or kilograms). As the density of fuel varies according to temperature both may be used. The volume of an aircraft fuel tank age is fixed and therefore it will not be able to accommodate the same weight of fuel at high temperature when the fuel density is lower. For most practical purposes a gallon of fuel may be assumed to weigh around 8 lb (as opposed to 10 lb for a gallon of water).

The essential characteristics of a modern aircraft fuel management system may embrace some or all of the following modes of operation:

• Fuel pressurization
• Engine feed
• Fuel transfer
• Refuel/defuel

Fuel Systems

• Fuel storage – there are many issues related to the storage and assured supply of fuel during aircraft flight; these issues vary from aircraft to aircraft and form the kernel of the overall aircraft fuel system requirements
• Vent systems
• Use of fuel as heat sink
• Fuel jettison
• In-flight refueling

Before describing the operation of these typical modes of operation it is worth examining one and outlining the primary components that comprise such a system. It should also be stated that this represents the briefest introduction of issues addressed in a companion volume dedicated to aircraft fuel systems.

3.3 Description of Fuel System Components

3.3.1 Fuel Transfer Pumps

Fuel transfer pumps perform the task of transferring fuel between the aircraft fuel tanks to ensure that the engine fuel feed requirement is satisfied. On most aircraft this will require the supply of fuel to collector tanks which carry out the obvious task of collecting or consolidating fuel before engine feed; thereby assuring a guaranteed (short-term) supply to each engine. Transfer pumps may also be required to transfer fuel around the aircraft to maintain pitch or lateral trim. In the case of pitch trim this requirement is becoming more critical for unstable control configured aircraft where the task of active CG control may be placed upon the fuel management system.

On civil aircraft there is a requirement to transfer fuel from the fuselage centre wing tanks to tanks where fuel may typically be consolidated before engine feed. However there are FAR/JAR regulations which require independent engine feed systems. On more recent civil aircraft such as the Airbus A340 the horizontal stabilizer may contain up to 7 tons of fuel which has to be transferred to maintain the aircraft CG within acceptable limits during the cruise phase. Typically this schedule will be invoked when the aircraft has exceeded an altitude of FL250.

Older aircraft such as the Vickers VC10 also contain fuel in the empennage, in this case the fin, to increase fuel capacity. In these cases pumps are also required to transfer fuel forward to a centre tank for consolidation. A typical aircraft system will have a number of transfer pumps for the purposes of redundancy, as will be seen in the examples given later in this chapter.

An example of a fuel transfer pump is shown in Figure 3.3, this particular example being used on the Anglo-French Jaguar fighter. This is a fuel lubricated pump; a feature shared by most aircraft fuel pumps. The pump has the capability of safely running dry in the event that no fuel should remain in the tank for any reason. Thermal protection is also incorporated to prevent over-heating. This particular pump is designed to supply in the region of 400 lb/minute at a pressure of 10 psi.

Fuel Booster Pumps

Fuel booster pumps, sometimes called engine feed pumps, are used to boost the fuel flow from the aircraft fuel system to the engine. One of the reasons for this is to prevent aeration (i.e. air in the fuel lines that could cause an engine 'flameout' with consequent loss of power). Another reason in the case of military aircraft is to prevent 'cavitations' at high altitudes. Cavitations is a process in which the combination of high altitude, relatively high fuel temperature and high engine demand produce a set of circumstances where the fuel is inclined to vaporize. Vaporization is a result of the combination of low fuel vapor pressure and high temperature. The effect is drastically to reduce the flow of fuel to the engine that can cause a flameout in the same way as aeration (as may be caused by air in the fuel). An aircraft system will possess a number of transfer pumps as will be illustrated later in the chapter.

The engine manufacturer usually imposes a requirement that fuel feed pressure must remain at least 5 psi above true vapor pressure at all times. Booster pumps are usually electrically driven; for smaller aircraft such as the BAE Systems Jet Provost and the Harrier the pump is driven from the aircraft 28 VDC system with delivery pressures in the range 10–15 psi and flow rates up to 2.5 kg/sec of fuel. The higher fuel consumption of larger, high performance aircraft booster pumps are powered by three-phase AC motors; in the case of Tornado delivering 5 kg/sec. Booster pumps are cooled and lubricated by the fuel in which they are located in a similar way to transfer pumps, and may be specified to run for several hours in a 'dry' environment. Fuel pumps can also be hydraulically driven or, in certain cases, ram air turbine driven, such as the VC10 tanker in-flight refueling pump. While most of the larger aircraft use electric motor-driven pumps, ejector pumps are in common use for both fuel feed and transfer in some applications.

Fuel Transfer Valves

A variety of fuel valves will typically be utilized in an aircraft fuel system. Shut-off valves perform the obvious function of shutting off fuel flow when required. This might involve stemming the flow of fuel to an engine, or it may involve the prevention of fuel transfer from one tank to another. Refuel/defuel valves are used during aircraft fuel replenishment to allow flow from the refueling gallery to the fuel tanks. These valves will be controlled so that they shut off once the desired fuel load has been taken on board. Similarly, during defueling the valves will be used so that the load may be reduced to the desired level – almost entirely used for maintenance purpose. Cross-feed valves are used when the fuel is required to be fed from one side of the aircraft to the other.

Fuel dump valves perform the critical function of dumping excess fuel from the aircraft tanks in an emergency. These valves are critical in operation in the sense that they are required to operate and dump fuel to reduce the fuel contents to the required levels during an in-flight emergency. Conversely, the valves are not required to operate and inadvertently dump fuel during normal flight.

The majority of the functions described are performed by motorized valves that are driven from position to position by small electric motors. Other valves with a discrete on/off function may be switched by electrically operated solenoids. Figure 3.5 shows an example of a transfer valve driven by a DC

Non-Return Valves (NRVs)

A variety of non-return valves or check valves are required in an aircraft fuel system to preserve the fluid logic of the system. Non-return valves as the name suggests prevent the flow of fuel in the reverse sense. The use of non-return valves together with the various transfer and shut off valves utilised around the system ensure correct system operation in the system modes listed above and which will be described in more detail later in the chapter

Level Sensors

Level sensors measure the fuel level in a particular tank and thereby influence fuel management system decisions. Level sensors are used to prevent fuel tank overfill during refueling. Level sensors are also used for the critical low level sensing and display function to ensure that fuel levels do not drop below flight critical levels where the aircraft has insufficient fuel to return to a suitable airfield. Level sensors may be one of a number of types: Float operated; optical; sound or zener diodes – two of which are described below.

Float Level Sensors

Float level sensors act in a similar way to a domestic toilet cistern connected to the water supply shut-off valve that is closed as the float rises. The refueling valve, operating in the same way, is a simple but effective way of measuring the fuel level but it has the disadvantage that, having moving parts, the float arm may stick or jam.

Zener Diode Level Sensors

By using simple solid state techniques it is possible to determine fluid levels accurately. The principle is based upon a positive temperature coefficient directly heated Zener diode. The response time when sensing from air to liquid is less than 2 seconds (refueling valve) and from liquid to air less than 7 seconds (low level warning). Fluid level may be sensed to an accuracy of about plus/minus 2 mm and the

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power required is around 27 mA per channel at 28 V DC. The sensor operates in conjunction with an amplifier within a control unit and can accommodate multi-channel requirements. A typical fluid sensor of this type is shown in Figure 3.7. The advantage of this method of level sensing is accuracy and the fact that there are no moving parts. In more recent times this technique is disfavored for safety reasons.

Capacitance Sensors

Capacitance sensors were used on A340 and A380 for sensing fuel level. The advantage is that there is a measurable signal from the sensor under both states.

Ultrasonic Sensors

Ultrasonic point sensors are becoming favored as point level sensors within the fuel sensing system.

Fuel Gauging Probes

Many of the aircraft functions relating to fuel are concerned with the measurement of fuel quantity on board the aircraft. For example, the attainment of a particular fuel level could result in a number of differing actions depending upon the circumstances: opening or closing fuel valves or turning on/off fuel pumps in order to achieve the desired system state. Quantity measurement is usually accomplished by a number of probes based upon the principle of fuel capacitance measurement at various locations throughout the tanks.

Air and fuel have different dielectric values and by measuring the capacitance of a probe the fuel level may be inferred. The locations of the fuel probes are carefully chosen such that the effects of aircraft pitch and roll attitude changes are minimized as far as quantity measurement is concerned. Additional probes may cater for differences in fuel density and permittivity when uplifting fuel at differing airfields around the world as well as for fuel at different temperatures.

Fuel gauging, or Fuel Quantity Indication Systems (FQIS) as they are sometimes known, are therefore an essential element in providing the flight and ground crews with adequate information relating to the amount of fuel contained within the aircraft tanks.

Fuel Quantity Measurement Basics

The underlying difficulties in accurately measuring aircraft fuel contents; also referred to as Fuel On Board (FOB) lie in the very nature of the agility and mobility of the air vehicle. The most obvious factors are:

• The difficulty in measuring a fluid level within a body in motion
• The fact that aircraft tanks are virtually never regular shapes
• The fact that aircraft fuel demonstrates diverse properties and has different composition when uplifted in different locations Fuel quantity may be expressed as kilograms (1000 kilogram=a metric tonne), pounds (lb), or gallons – either Imperial or US gallons.

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A US gallon is 0.8×an Imperial gallon (1 Imperial gallon=8×20=160 fluid ounces). The Specific Gravity (SG) of fuel is around 0.8, therefore an Imperial gallon is roughly equivalent to 160/16 or 10 lbs whereas a US gallon equates to around 8 lbs. Since the contents of aircraft tanks are characterised by tank volume the amount of energy contained within a fuel load is therefore determined by the weight (mass) of the FOB; itself a function of fuel density and fuel temperature.

Fluid Motion

Measuring fuel level in flight is analogous to trying to run while carrying a bucket of water; the fluid appears to take on a mind of its own and the 'inertia' of the fluid has to be anticipated both when starting out and when stopping. This fluid can be ameliorated to a degree by natural boundaries such as wing ribs or fuselage frames that may protrude into the tank. The insertion of baffles may also prevent undue 'sloshing' of the fuel. This sloshing action can be modeled using 3D computer aided design tools together with fluid dynamic modeling tools such as Flow master. This enables a simulation of the fuel system, in whole or in part, to be modeled and subjected to aircraft maneuvers to observe the effects on the fuel. Baffles can then be inserted into the model to allow observation of their effect on fuel slosh, and to optimize their location in a tank

Tank Shapes

Aircraft tank shapes vary greatly and are difficult to determine, particularly at an early stage in the aircraft design. Large, regular volumes are at a premium within an aircraft and the volumes available to the fuel system designer are usually those remaining when the structures and propulsion designers have had their day. Therefore not only are the tank shapes irregular but their boundaries may not be fixed until fairly late in the design.

Once the tank boundaries are frozen, the tank designer has to characterize the volumetric shape of the tank to understand what the fluid level means for a variety of tank attitudes. The problem may be better understood by referring to Figure 3.8. This is a representation of a simple rectangular tank that might approximate to the centre tank on many typical civil aircraft. While the shape is regular the tank will be rotating as the aircraft pitch and roll attitude alter.

Aircraft accelerations will also occur as speed changes are made. The fluid contents of this tank, or more correctly, the fluid level may be determined by placing quantity probes in each corner of the tank. This may be acceptable for a basic configuration but to permit necessary levels of accuracy following a probe failure, additional probes may need to be added. In a sophisticated long range aircraft the probes may need to be replicated to provide dual redundant sensing.

Forward

Pitch

Roll

Additional Effects:

–Wing Sweep

–Wing Dihedral

–Wing Flexure

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Fuel System Operating Modes

The modes of operation described in the following paragraphs are typical of many aircraft fuel systems. Each is described as an example in a particular fuel system. Any system may exhibit many but probably not all of these modes. In an aircraft the fuel tanks and components have to compete with other systems, notably structure and engines for the useful volume contained within the aircraft profile. Therefore fuel tanks are irregular shapes and the layman would be surprised by how many tanks there are, particularly within the fuselage where competition for usable volume is more intense.

The proliferation of tanks increases the complexity of the interconnecting pipes and certainly does not ease the task of accurate fuel measurement. As an example of a typical fighter aircraft fuel tank configuration see Figure 3.14 that shows the internal fuel tank configuration for EAP. This is a simplified diagram showing only the main fuel transfer lines; refueling and vent lines have been omitted for clarity. Whereas the wing fuel tanks are fairly straightforward in shape, the fuselage tanks are more numerous and of more complex geometry than might be supposed.

The segregation of fuel tanks into smaller tanks longitudinally (fore and aft) is due to the need to avoid aircraft structural members. The shape of most of the fuselage tanks also shows clearly the impositions caused by the engine intakes. Furthermore as an experimental aircraft EAP was not equipped for in-flight refuelling nor was any external under-wing or ventral tanks fitted. It can be seen that a fully operational fighter would have a correspondingly more complicated fuel system than the one shown.

Pressurization

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

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

In some aircraft such as the F-22, inert gas is used to pressurize the fuel tanks. Inert gas for this purpose can be obtained from an On-Board Inert Gas Generating System (OBIGGS).

Engine Feed

The supply of fuel to the engines is by far the most critical element of the fuel system. Fuel is usually collected or consolidated before being fed into the engine feed lines. The example in Figure 3.15 shows a typical combat aircraft, the fuel is consolidated in two collector tanks; one for each engine.

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Introduction:

The modern turbofan engine is very effective for generating gas use of engine bleed air for a number of aircraft systems, either for heating of various equipments, provision of air for cabin air conditioning and pressurization. Bleed air is extracted from the engine compressor and after cooling and pressure reduction/regulation it is used for a variety of functions. In the engine, high pressure bleed air is used as the motive power – sometimes called 'muscle power' which is used for actuation of many valves associated with the bleed air extraction function. Medium-pressure bleed air is used to start the engine in many cases, either using air from a ground power unit, APU or cross bled from another engine on the aircraft which is already running. Bleed air is also used to provide anti-ice protection by heating the engine intake cowling and it is also used as the motive power for the engine thrust reversers. Bleed air is also used for aircraft pressurizing hydraulic reservoirs, providing hot air for rain dispersal from the aircraft windscreen, pressurizing the water and waste system and so on. In some aircraft Air Driven Pumps (ADPs) are used as additional means of providing aircraft hydraulic power. In Pitot static systems to air is used for air data computer and other parts of this system for safe aircraft flight.

Use of Bleed Air

Modern engines 'bypass' a significant portion of the mass flow past the engine and increasingly a small portion of the mass flow passes through the engine core or gas generation section**. The ratio of bypass air to engine core air is called the bypass ratio** and this can easily exceed 10:1 for transport civil aircraft engines. The characteristics of a modern turbofan engine are shown in Figure . This figure shows the **pressure (in psi) and the temperature** (in degrees centigrade) at various points throughout the engine for three engine conditions: ground idle, take-off power and in the cruise condition.

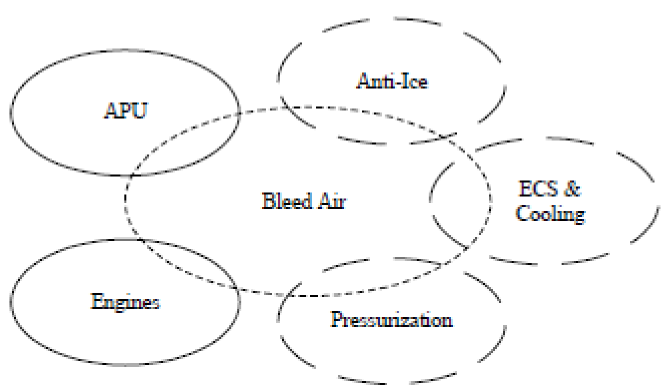

Figure 4.1: Uses of Bleed Air

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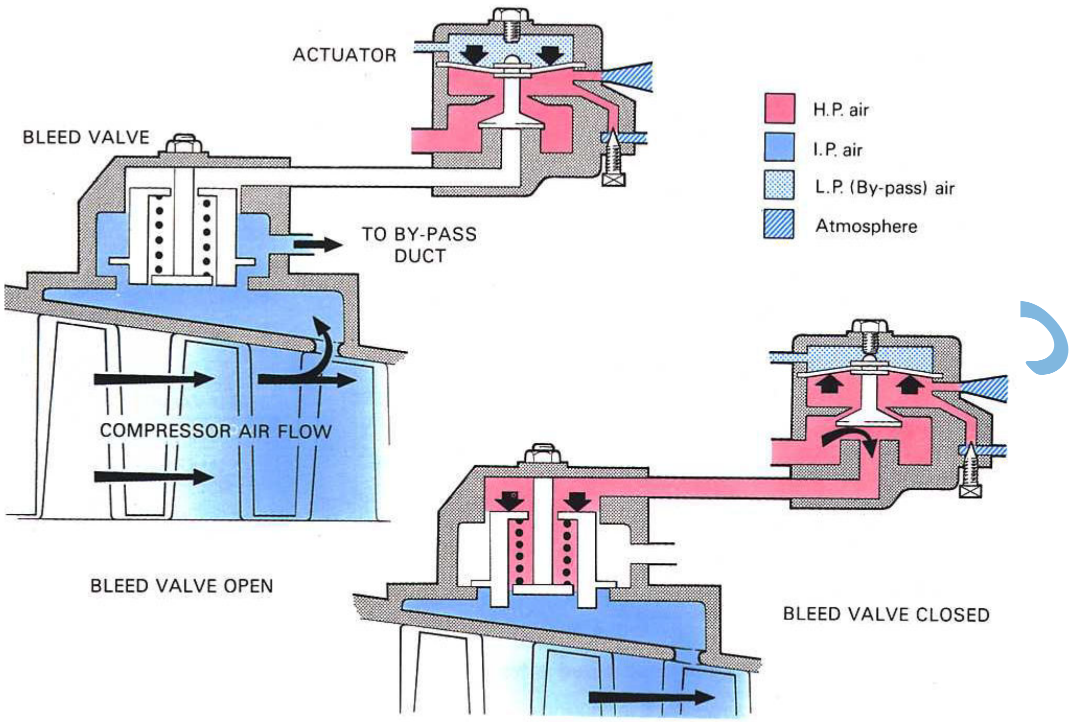

Figure 4.2: Relation between air bleed and other aircraft demanded systems

The main aircraft engines and Auxiliary Power Unit (APU) is also a source of high pressure bleed air. The APU is in itself a small turbojet engine, designed more from the viewpoint of an energy and power generator than a thrust provider like main engines.

APU The APU is primarily designed to provide **electrical and pneumatic power** by a shaft driven generator and compressor. The APU is therefore able to provide an independent source of electrical power and compressed air while the aircraft is on the ground, although it can be used as a backup provider of power while airborne. Some aircraft designs are actively considering the use of in-flight operable APUs to assist in in-flight engine re-lighting. The figure shows supply of air bleed to various services.

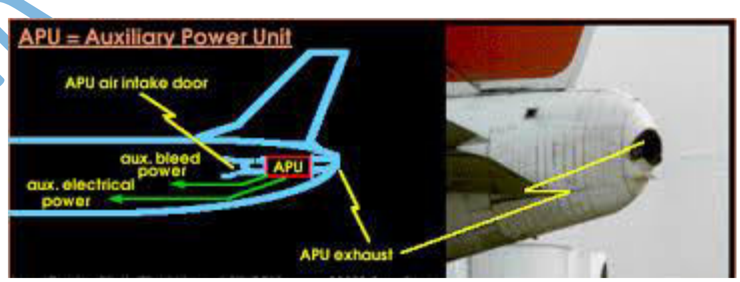

Figure 4.3: APU

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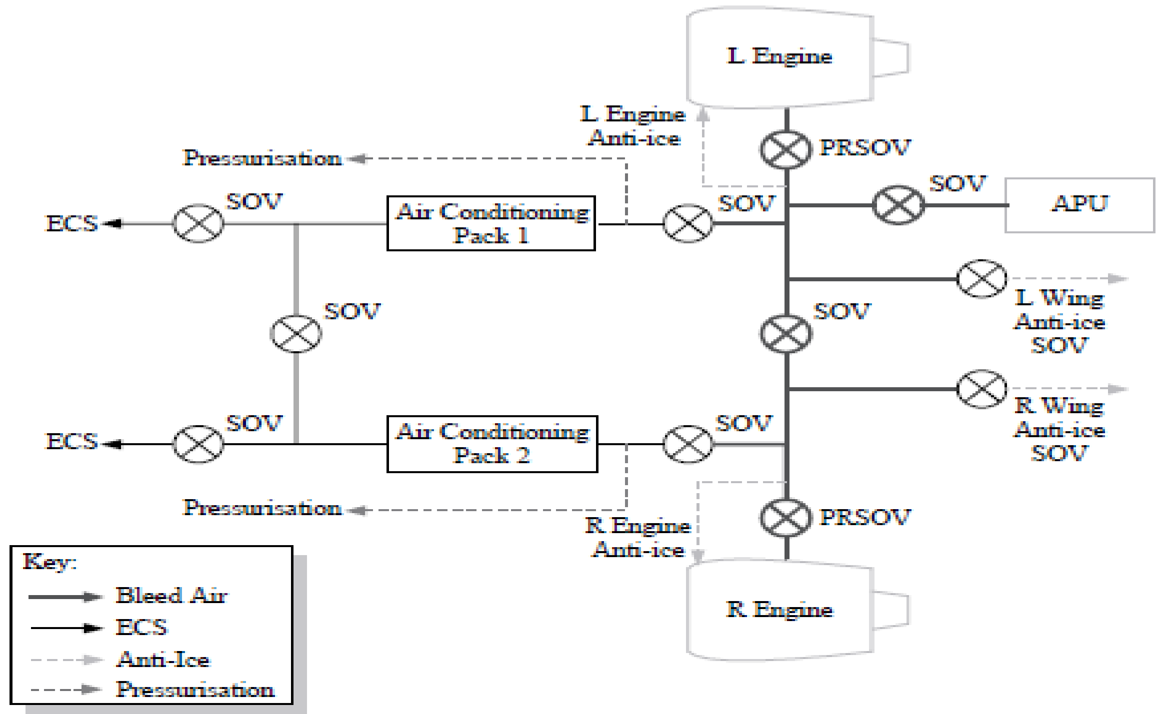

Figure 4.4: Air Bleed Operation

Air bleed Operation

The above figure shows the air bleed from the two engines is passed through a Pressure-Reducing Shut-Off Valve (PRSOV) which serves the function of controlling and, when required, shutting off the engine bleed air supply. Air downstream of the PRSOV may be used in a number of ways:

• By means of a cross flow Shut-Off Valve (SOV) the system may supply air to

the opposite side of the aircraft during engine start or if the opposite engine is inoperative for any reason.

• A SOV from the APU may be used to isolate the APU air supply • SOVs provide isolation as appropriate to the left and right air conditioning packs and pressurization systems.

• Additional SOVs provide the means by which the supply to left and right wing anti-icing systems may be shut off in the event that these functions are not required.

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Engine air bleed users

The following are loads of air bleed:-

1. **Air conditioning**
2. **Cargo compartment heating**
3. **Wing and engine anti-ice protection.**
4. **Engine start**
5. **Thrust reversal actuation**
6. **Hydraulic reservoir pressurization**
7. **Rain repellent nozzles – aircraft windscreen**
8. **Water tank pressurization and toilet waste**
9. **Air driven hydraulic pump (ADP)**
10. **Hydraulic System.**
11. **Fuel system pressurization and vent system**

Engine air bleed components

Pressure Reducing Shut off Valve (PRSOV)

A typical PRSOV is shown in Figure is an example of a solenoid controlled and pneumatically operated and which controls temperature, flow and pressure is shown in Figure.

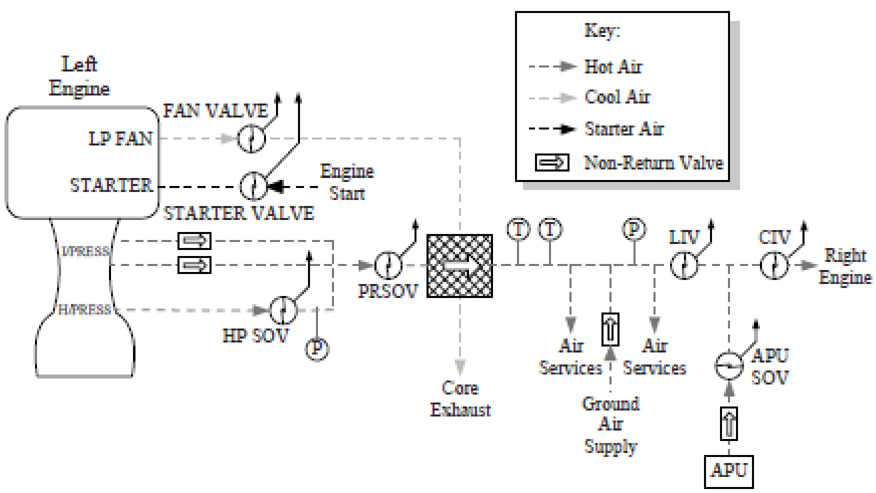

Figure 4.5: Pressure Reducing Shut off Valve

Function of PRSOV

• On/off control of the engine bleed system

• Pressure regulation of the engine supply air by means of a butterfly valve actuated by pneumatic pressure

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• Engine bleed air temperature protection and reverse flow protection.

• Ability to be selected during maintenance operations in order to test reverse thrust operation.

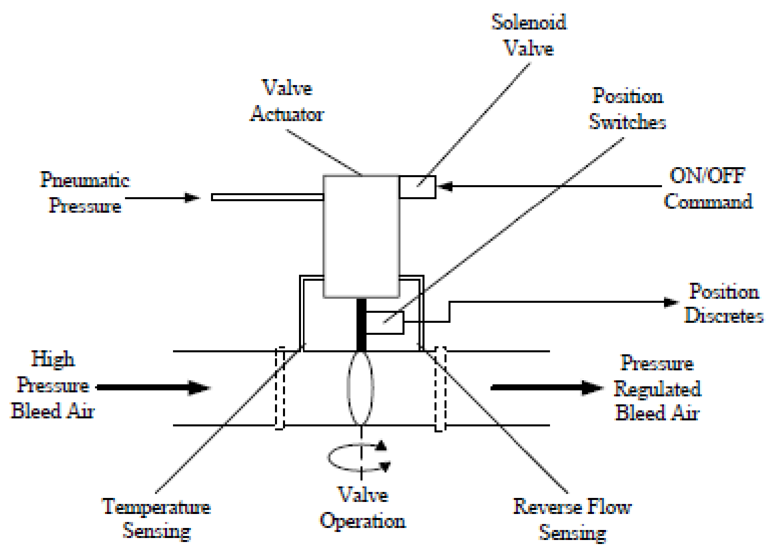

Figure4.6: Working Principle of PSROV

Working Principle:-

The PRSOV is pneumatically operated and electrically controlled. Operation of the solenoid valve from the appropriate controller enables the valve to control the downstream pressure pneumatically to approximately 40 psi within predetermined limits. The valve position is signaled by means of discrete signals to the bleed air controller and pressure switches provide over and under-pressure warnings. The various pressure, flow and discrete signals enable the bleed air controller Built-In Test (BIT) to confirm the correct operation of the PRSOV and fan control valve combination. This ensures that medium pressure air (40 psi) of the correct pressure and temperature is delivered to the pre-cooler and thence downstream to the pneumatic and air distribution system.

Wind Shield Heating

Ice that accumulates on the aircraft windscreens are kept clear of ice by the use of window heating so that the flight crew has an unimpeded view ahead. The prevention of ice build-up on the windscreen is achieved by means of electric heating elements embedded in the glass panels.

Wing, Tail Unit and Engine air intake & IGV Anti-Icing

The protection of the aircraft from the effects of aircraft icing represents one of the greatest and flight critical challenges which confront the aircraft. Wing leading edges and engine intake cowlings need to be kept free of ice accumulation at all times.

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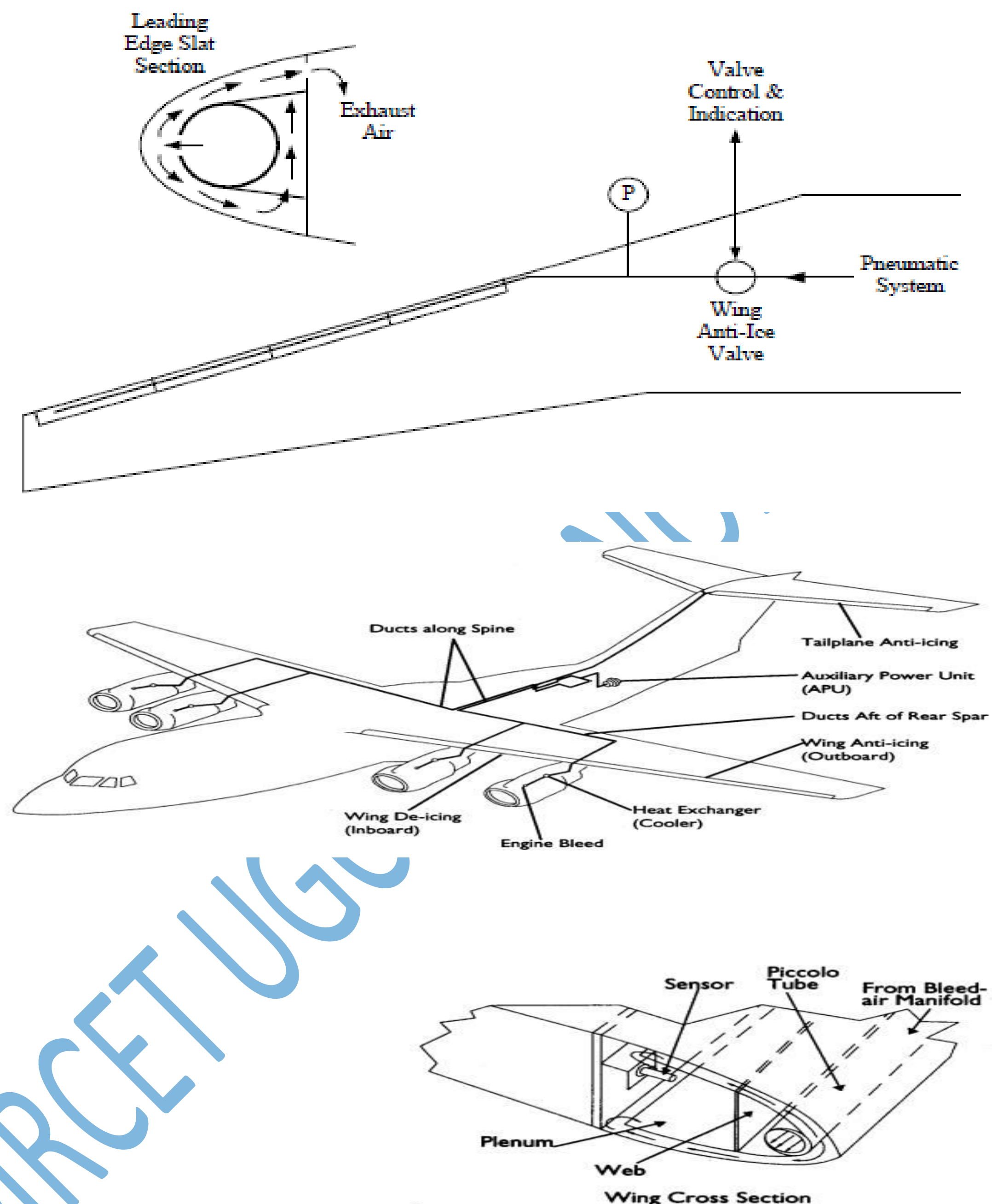

Figure 4.7: Wing, Tail Unit and Engine air intake & IGV Anti-Icing

In the case of the wings, the gathering of ice can degrade the aerodynamic performance of the wing, leading to an increased stalling speed with the accompanying hazard of possible loss of aircraft control.

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Ice that accumulates on the engine intake and then breaks free entering the engine can cause substantial engine damage with similar catastrophic results. Finally, the aircraft air data sensors are heated to ensure that they do not ice up and result in a total loss of air data information that could cause a hazardous situation or the aircraft to crash..

In the case of the wing and engine anti-icing the heating is provided by hot engine bleed air which prevents ice forming while the system is activated. The principles of wing anti-ice control are shown in Figure. The flow **of hot air to the outer wing leading edges is controlled** by the Wing Anti- Ice Valve. The air flow is modulated by the electrically enabled anti-icing controller; this allows air to pass down the leading edge heating duct. This duct can take the form of a **pipe with holes appropriately sized to allow a flow of air onto the inner surface of the leading edge**. The air is bled out into the leading edge slat section to heat the structure before being dumped overboard.

Engine anti-icing

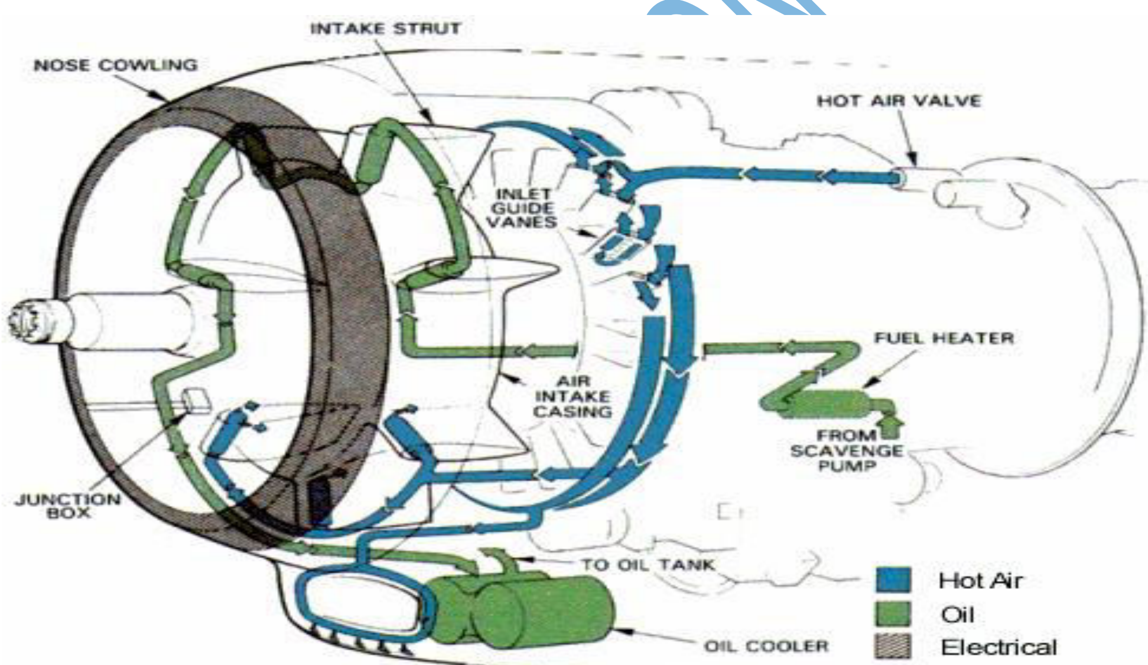

Figure 4.8: Anti Icing

Engine anti-icing is similarly achieved by a valve fitted on the engine and Prevention of forming ice in the engine cowling is done by blowing hot air inside the engine compartment. This activation of the engine anti-icing system is confirmed by the flight crew by means of the closure of a pressure switch that provides an indication to the display system. The presence of hot air ducting throughout the airframe in the engine nacelles and wing leading edges poses an additional problem; that is to safeguard against the possibility of hot air duct leaks causing an overheat hazard. Accordingly, overheat detection

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loops are provided in sensitive areas to provide the crew with a warning in the event of a hot gas leak occurring. An overheat detection system will have elements adjacent to the air conditioning packs, wing leading edge and engine nacelle areas to warn the crew of an overheat.

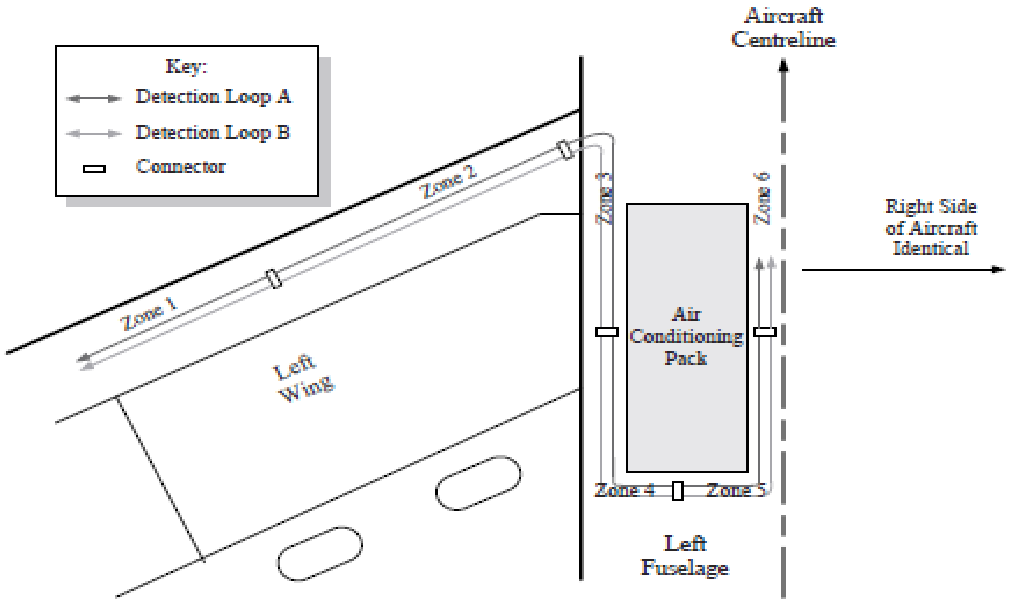
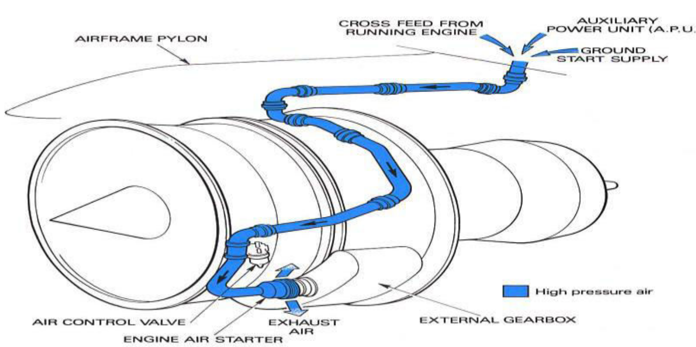


Figure 4.9: Air Conditioning

Engine Start

The availability of high pressure air throughout the bleed air system lends itself readily to the provision of motive power to crank the engine during the engine start cycle. The start valve is activated to supply bleed air to the engine starter. On the ground the engines may be started in a number of ways:

- **By use of a ground air supply cart**
- **By using air from the APU – probably the preferred means**
- **By using air from another engine which is already running.**



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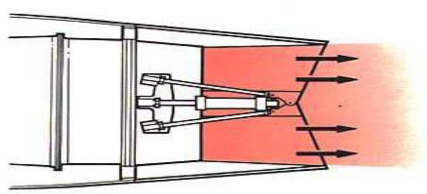
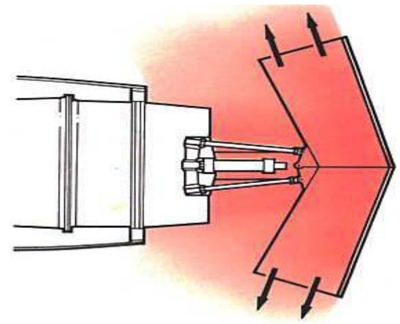
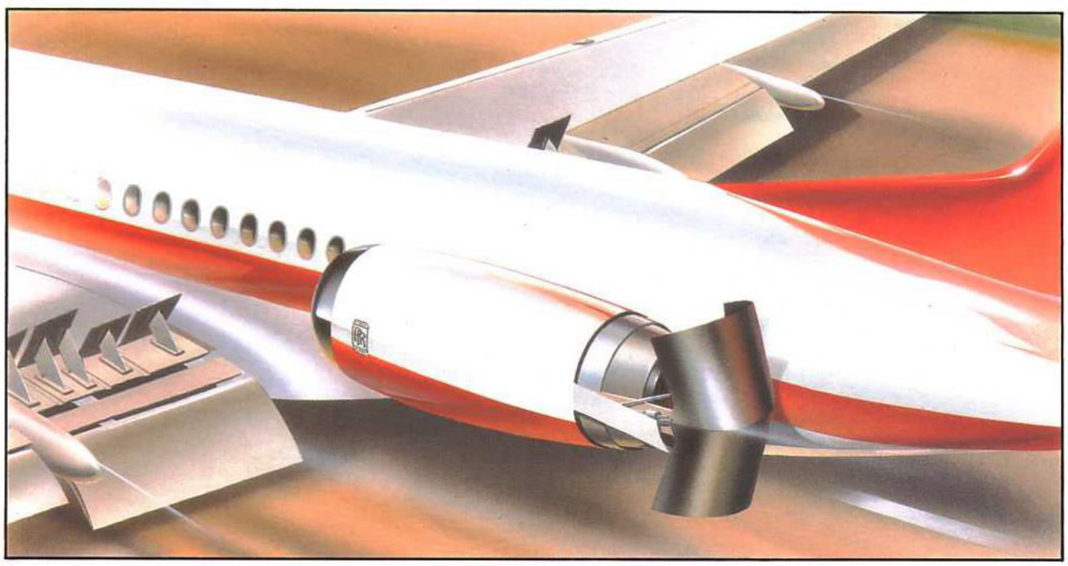
Thrust Reversers



ACTUATOR EXTENDED AND BUCKET DOORS IN FORWARD THRUST POSITION



ACTUATOR AND BUCKET DOORS IN REVERSE THRUST POSITION



Engine thrust reversers are commonly used to deflect engine thrust forward during the landing roll-out to slow the aircraft and **preserve the brakes**. Thrust reversers are commonly used in conjunction with a lift dump function, whereby all the spoilers are simultaneously fully deployed, slowing the aircraft by providing additional aerodynamic drag while also dispensing lift. Thrust reversers deploy two buckets, one on each side of the engine, which are pneumatically operated by means of air turbine motor actuators to deflect the fan flow forward, thereby achieving the necessary braking effect when the aircraft has a 'weight-on-wheels' condition.

Hydraulic Systems

The hydraulic reservoir is pressurized using regulated bleed air from the pneumatic/bleeds air system. Supply hydraulic fluid may be pressurized by the two alternate pumps:

• By means of the ACMP powered by three-phase 115 VAC electrical power.

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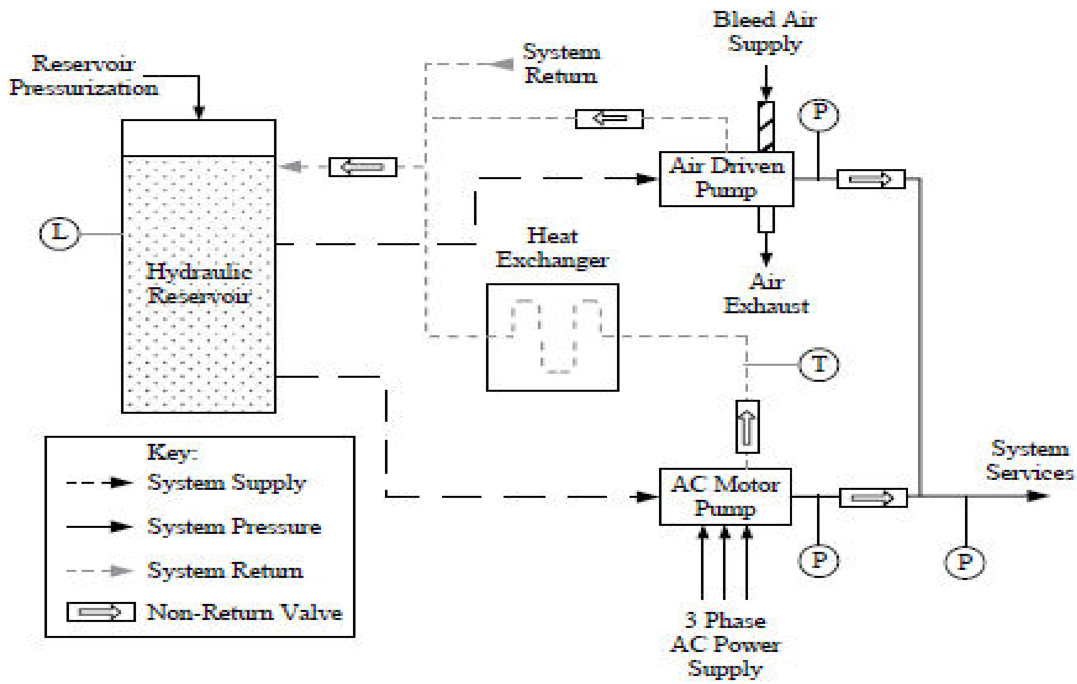
• By means of the Air Driven Pump (ADP) using pneumatic power as the source.



Pitot Static Systems

By contrast with the bleed air system already described which provides energy or power for a number of diverse aircraft systems, the pitot static system is an instrumentation system used to sense air data parameters of the air through which the aircraft is flying. Without the reliable provision of air data the aircraft is unable safely continue flight. The pitot static system is therefore a high integrity system with high levels of redundancy. There are two key parameters which the pitot static system senses:

• Total pressure Pt is the sum of local static pressure and the pressure caused by the forward flight of the aircraft (Dynamic pressure). The pressure related to the forward motion of the aircraft by the following formula: Pressure = ½ ρV^2 Where _ is the air density of the surrounding air and V is the velocity

• Static pressure or Ps is the local pressure surrounding the aircraft and varies with altitude

Therefore total pressure, Pt=Ps+ ½ ρV^2

The forward speed of the aircraft is calculated by taking the difference between Pt and Ps An aircraft will have three or more independent pitot and static sensors

Figure 6.11 shows the principle of operation of pitot and sensors.

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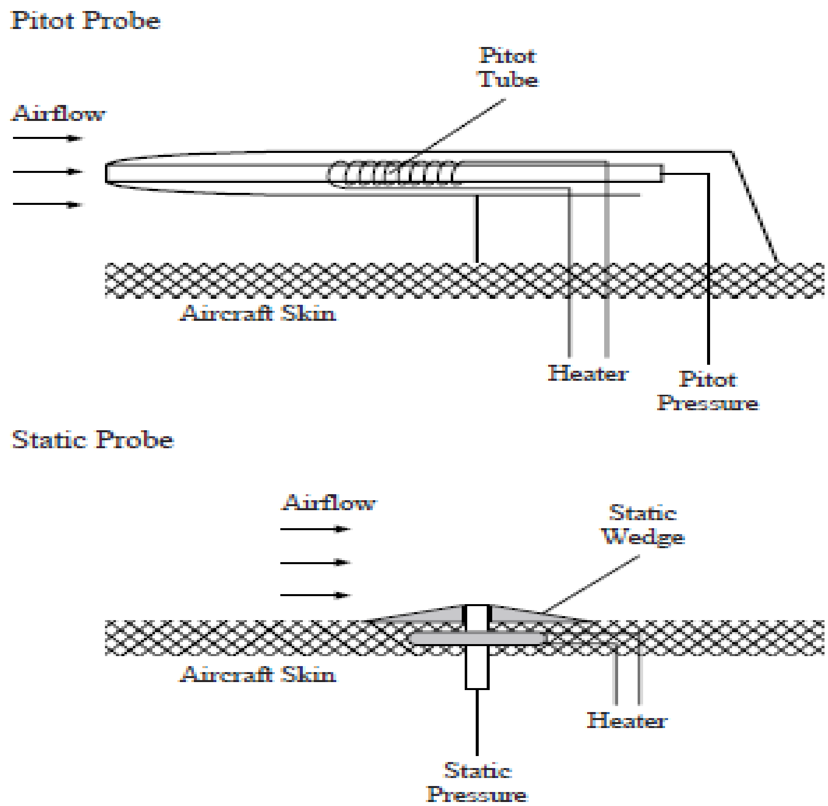
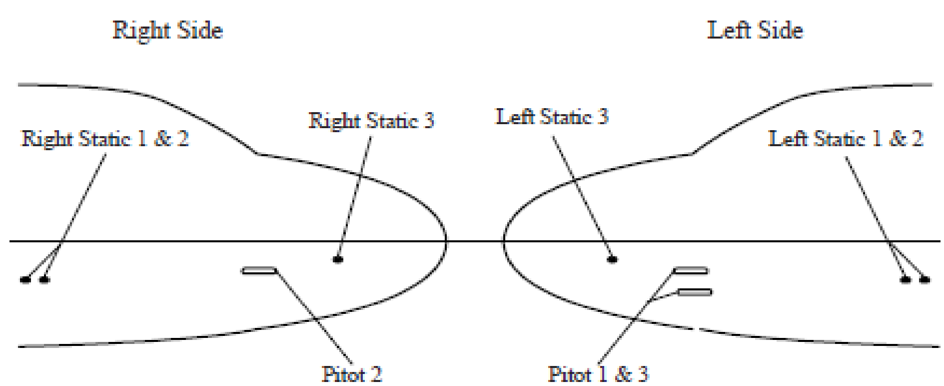


The static probe shown in the lower diagram is located perpendicular to the airflow and so is able to sense the static pressure surrounding the aircraft. Like the pitot probe the static probe is provided with a **heater element** that **continuously heats the sensor and prevents the formation of ice**. On some aircraft the pitot and static sensing functions are combined to give a pitot-static probe capable of measuring both dynamic and static pressures. A typical installation on a civil transport aircraft is depicted in Figure .



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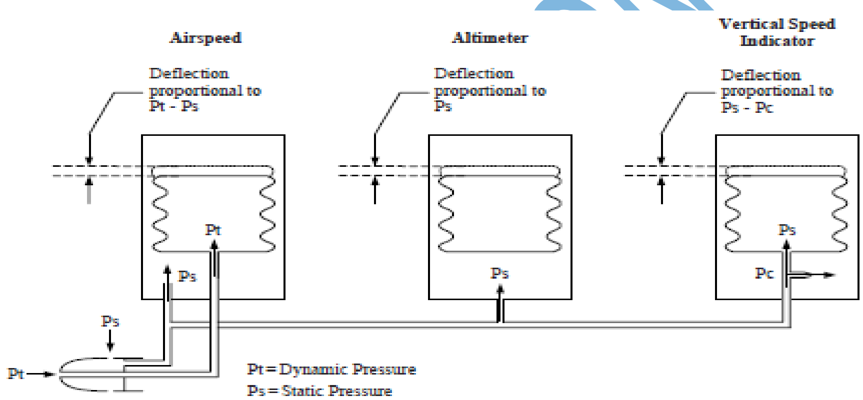
Measurement of Air Speed

Airspeed may be calculated from the deflection of needle in the instrument where Pt and Ps are differentially sensed. Airspeed is proportionate to Pt – Ps and therefore the mechanical deflection may be sensed and airspeed deduced. This may be converted into a meaningful display to the flight crew value in a mechanical instrument by the mechanical gearing between aneroid capsule and instrument dial.

• Altitude may be calculated by the deflection of the static capsule in the centre instrument. Again in a mechanical instrument the instrument linkage provides the mechanical scaling to transform the data into a meaningful display.

• Vertical speed may be deduced in the right hand instrument where the capsule deflection is proportional to the rate of change of static pressure with reference to a case pressure, Pe. Therefore the vertical speed is zero when the carefully sized bleed orifice between capsule inlet and case allows these pressures to equalize.



Need for Controlled Cabin Environment

In the early days of flight, pilots and passengers were prepared to brave the elements for the thrill of flying. However, as aircraft performance has improved and the operational role of both civil and military aircraft has developed, requirements for Environmental Control Systems (ECS) have arisen. They provide a favorable environment for the instruments and equipment to operate accurately and efficiently, to enable the pilot and crew to work comfortably, and to provide safe and comfortable conditions for the fare-paying passengers . Providing sufficient heat for the aircraft air conditioning system is never a problem, since hot air can be bled from the engines to provide the source of conditioning air. The design requirement is to reduce the temperature of the air sufficiently to give adequate conditioning on a hot day.

Principle Heat Sources in the Aircraft which need to be addressed for cooling problem:

(a) ***Kinetic Heating*** **($T_{rec}=T_{amb}(1+0.18\ M^2)$ &$T_{ram}=T_{amb}(1+0.2\ M^2)$)**

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(b) ***Solar Heating*** **(**The combined effect of internal heating and direct solar radiation has an effect on the pilot, especially when wearing survival gear and anti-g trousers and vest which requires considerable cooling air in the cockpit).

(c) ***Airframe System Heat Loads*** (Heat is produced by hydraulic systems, electrical generators, engines and fuel systems components.This produce a radiation effect in the systems such as pumps or motors, or from heat rejected in cooling fluids such as oil. To maintain operating efficiency and to prevent chemical breakdown of fluids with resulting degradation in their performance it is essential to cool these fluids.)

(d) ***Avionics Heat Loads (****The avionic equipment is generally powered continuously from power up to power down and, hence, dissipates heat is continuous. The equipment is kept in the boxes in designated avionic equipment bays in small aircraft, or in equipment cabinets in larger aircraft. Air is ducted to these areas for the specific purpose of cooling equipment and is then re-circulated or dumped overboard. The system must be designed to protect the components of the equipment throughout the aircraft flight envelope, and in whatever climatic conditions the aircraft must operate.)

Need for Cabin Conditioning

Design considerations for providing air conditioning in the cockpit of a high performance fighter are far more demanding than those for a subsonic civil airliner cruising between airports. The cockpit is affected by the sources of heat described above, but a high performance fighter is particularly affected by high skin temperatures and the effects of solar radiation through the large transparency.

However, in designing a cabin conditioning system for the fighter, consideration must also be taken of what the pilot is wearing. If, for example, he is flying on a mission over the sea, he could be wearing a thick rubber immersion suit which grips firmly at the throat and wrists. In addition, the canopy and windscreen will have hot air blown over the inside surfaces to prevent misting which would affect the temperature of the cabin. Another important factor is pilot workload or high stress conditions such as may be caused by a failure, or by exposure to combat. All these factors make it very difficult to cool the pilot efficiently so that his body temperature is kept at a level that he can tolerate without appreciable loss of his functional efficiency.

Methods of cooling

(a) Ram air:-

Ram air cooling is the process of rejecting aircraft heat load to the air flowing round the aircraft. This can be achieved by scooping air from the aircraft boundary layer or close to it. The air is forced through a scoop which faces into the external air flow, through a heat exchanger matrix and then rejected overboard by the forward motion of the aircraft. The heat exchanger works just like the radiator of a car. **This system has the disadvantage that it increases the aircraft drag because the resistance of the scoop**, pipe work and the heat exchanger matrix slows down the ram airflow.

The use of ram air as a cooling medium, at high altitude the air density becomes very low, reducing the ram air mass flow and hence its cooling capacity. In fact, when conditioning is required for systems which require cooling on the ground, then ram air cooling alone is unsuitable. However, this situation

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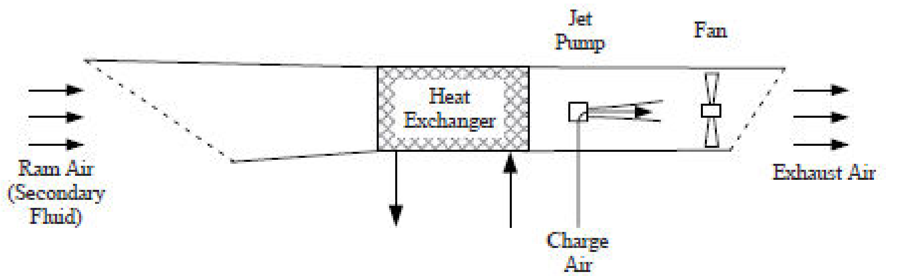
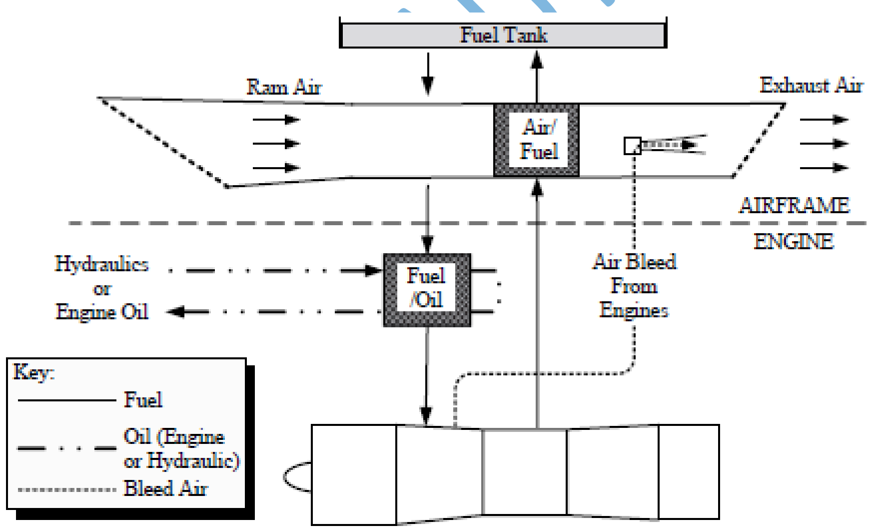
can be improved by the use of a cooling fan, such as used on a civil aircraft, or a jet pump, mainly used on military aircraft, to enhance ram air flow during taxi-ing or low speed flight. The jet pump enhances ram air cooling in the heat exchanger by providing moving jets of primary fluid bled from the engines to entrain a secondary fluid, the ram air, and move it downstream as shown in Figure



(b) Fuel Cooling:-

Fuel cooling systems have limited applications on aircraft for the transfer of heat from a heat source into the aircraft fuel as fuel is much better than air as a cooling medium. Fuel has a higher heat capacity and a higher heat transfer coefficient. Fuel is typically used to cool engine oil, hydraulic oil and gearbox oil. Figure shows a typical fuel and oil cooling system.



(c) Engine Bleed

The main source of conditioning air for both civil and military aircraft is engine bleed from the high pressure compressor. This provides a source whenever the engines are running. The conditioning air is also used to provide cabin pressurization. There are two types of bleed air system**: open loop and closed loop**.

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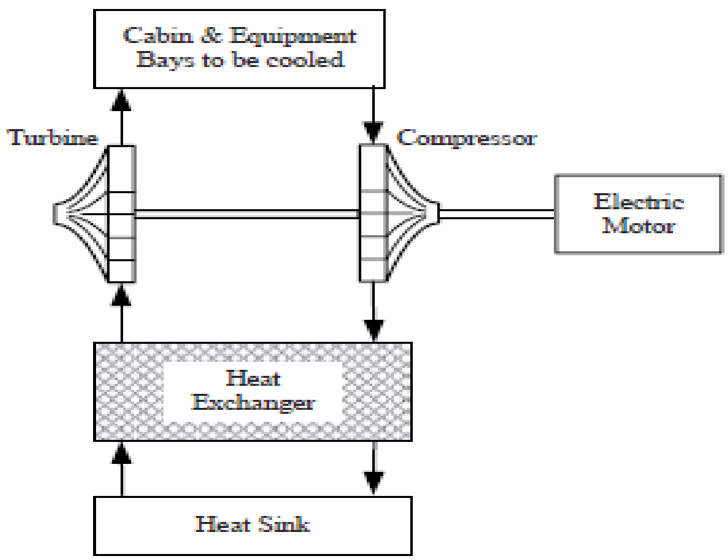
Open loop environmental control systems continually bleed large amounts of air from the engines, refrigerate it and then use it to cool the passengers and crew, as well as equipment, before dumping out in to the overboard.

Closed loop systems, as shown in Figure, collect the air once it has been used for cabin conditioning, refrigerate it and recycle it to be used again. In this way bleed air is used only to provide pressurization, a low venting air supply and sufficient flow to compensate for leaks in the closed loop system. This means that such a system uses considerably less engine bleed air than an open loop system ..



Cooling Systems

There are two main types of refrigeration systems in use:

- Air cycle refrigeration systems
- Vapor cycle refrigeration systems

(a) Air Cycle Refrigeration Systems

Air cycle refrigeration systems are used to cool engine bleed air down to temperatures required for cabin and equipment conditioning. Since engine bleed air is generally available, air cycle refrigeration is used because it is the simplest solution to the cooling problem, fulfilling both cooling and cabin pressurization requirements in an integrated system. This type is lighter and more compact.

Turbofan System

This will typically be used in a low-speed civil aircraft where ram temperatures

will never be very high as shown in figure below.

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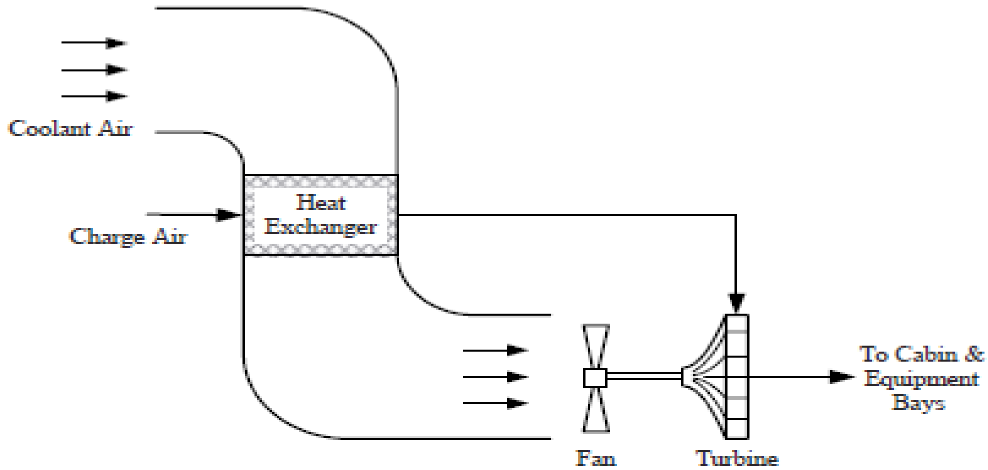

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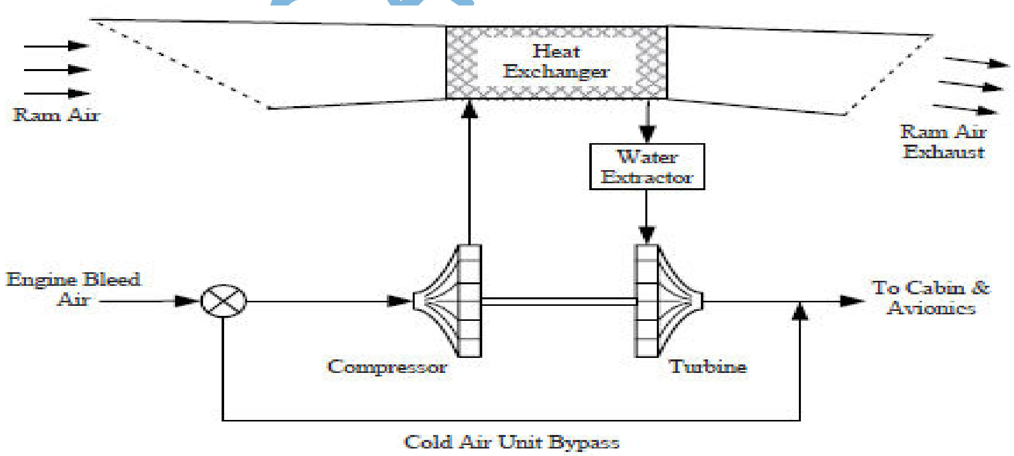
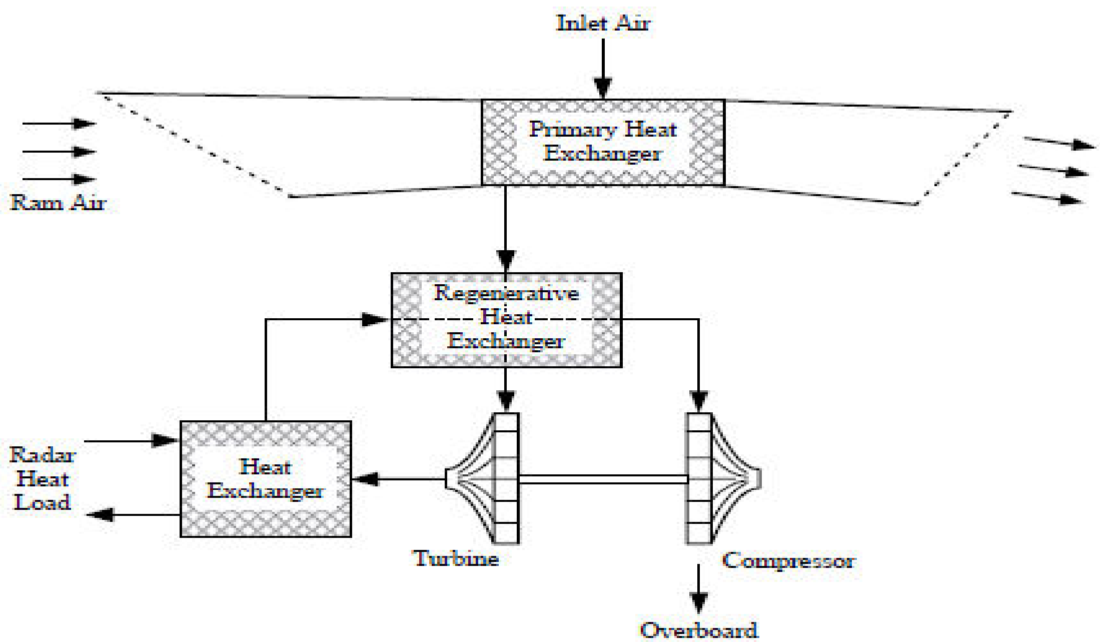
Bootstrap Refrigeration System

Conventional bootstrap refrigeration is generally used to provide adequate cooling for high ram temperature conditions, for example a high performance fighter aircraft. The basic system consists of a **cold air unit** and a **heat exchanger** as shown in Figure .The turbine of the cold air unit drives a compressor. Both are mounted on a common shaft. Three-rotor cold air units or air cycle machines can be found on most recently designed large aircraft, incorporating a heat exchanger coolant fan on the same shaft as the compressor and turbine.



Reversed Bootstrap

The reversed bootstrap system is so named because the charge air passes through the turbine of the cold air unit before the compressor. Following initial ram air cooling from a primary heat exchanger the air is cooled further in a regenerative heat exchanger and is then expanded across the turbine with a corresponding decrease in temperature. This air can then be used to cool an air or liquid closed-loop system, for radar transmitter cooling for example. The air then passes through the coolant side of the regenerative heat exchanger before being compressed by the compressor and dumped overboard.



Vapor Cycle Refrigeration Systems

The vapor cycle system is a closed loop system where the heat load is absorbed by the evaporation of a liquid refrigerant such as Freon gas in an evaporator. The refrigerant then passes through a compressor with a corresponding increase in pressure and temperature, before being cooled in a condenser where the heat is rejected to a heat sink. The refrigerant flows back to the evaporator via an expansion valve as shown in the figure.

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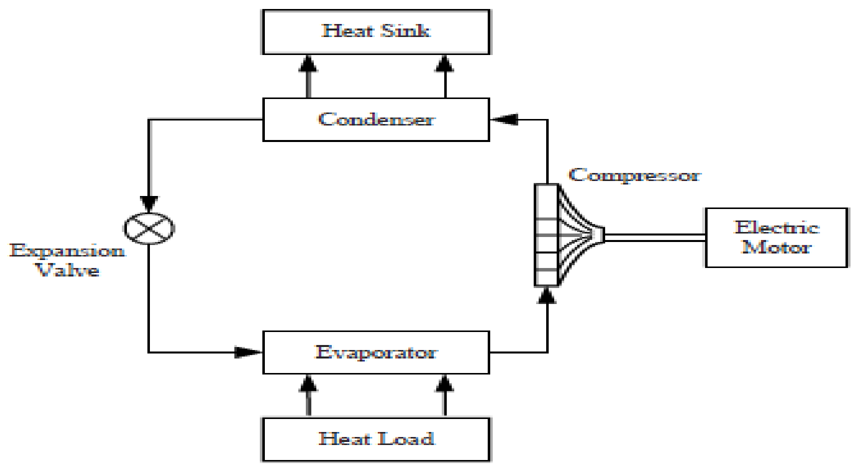

Humidity Control

Passenger comfort is achieved not only by overcoming the problems of cooling and cabin pressurization, but also by controlling humidity in the passenger

cabin. Without good humidity control this can result in a wet mist being supplied to the cabin. In addition to the aim of ensuring passenger comfort, humidity levels must be controlled to prevent damage to electrical and electronic equipment due to excessive condensation. Humidity control also reduces the need for windscreen and window de-misting and anti-misting systems. H***umidity is only a problem on the ground and at low altitudes, since the amount of moisture in the air decreases with increasing altitude***. Two types of water separator are in common use with air cycle refrigeration systems:

a) Centrifugal device: In the centrifugal devices a turbine is commonly used to swirl the moist air. The relatively heavy water droplets are forced to the sides of a tube, where the water and a small amount of air is trapped and drained away, thus reducing the water content of the air downstream of a water separator.

b) Mechanical device: Which consists of a coalesce, a relief valve and a water collector, achieves the same result by forcing the moist air to flow through the coalescer where large droplets are formed and blown onto collector plates. The water runs down the plates and is then drained away The mechanical water separator,

Air Distribution Systems

Avionics Cooling: In civil aircraft the total avionics heat load is low when compared with the military aircraft. In civil aircraft it is often sufficient to draw cabin ambient air over the avionics equipment racks using fans. This will have the effect of increasing the overall cabin temperature but,

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since the total avionics heat load is not massive, the environmental control system has sufficient capacity to maintain cabin temperatures at acceptable levels.

However, on a military aircraft with a high avionics heat load, only a few items of the avionics equipment are located in the cabin. The majority are located in either conditioned or no conditioned equipment bays, an installation decision which is made by taking into consideration such criteria as the effect of temperature on equipment reliability or damage, and the amount of engine bleed available for air conditioning. Since the equipment can operate in ambient temperatures higher than humans can tolerate, the air used to condition it tends to be cabin exhaust air. There is usually very little space in equipment bays as they are tightly packed with equipment. There is little space left for the installation of cooling air ducts. Therefore, the equipment racking and air distribution system must be carefully designed to ensure an even temperature distribution.

Unconditioned Bays

Unconditioned bays may reach temperatures up to recovery temperature. However, air in these bays is not totally stagnant. The aircraft is usually designed to have a continuous venting flow through each equipment bay, only the pressure cabin is sealed. This ensures that there is no build up of differential pressure between bays, particularly during rapid climb and descent. The venting flow tends to be the conditioned bay outlet flow.

Conditioned Bays

Equipment can be cooled by a variety of methods, including the following;

• cooling by convection air blown over the outside walls of the equipment

boxes (external air wash)

• air blown through the boxes and over the printed circuit boards (direct forced air)

• Air blown through a cold wall heat exchanger inside the box (indirect forced air)

• Fans installed in the box to draw a supply of cooling air from the box Surroundings.

The first method of cooling is adequate for equipment with low heat loads. As the heat load increases it tends to become very inefficient, requiring a lot more cooling air than the other three methods to achieve the same degree of cooling. It is very difficult to design an avionics equipment box with a high heat load to enable the efficient dissipation of heat by convection via the box walls. Local 'hot spots' inside the box will lead to component unreliability. The other three methods of cooling are very much more efficient, but the boxes must have a good thermal design to ensure precious conditioning air is not wasted.

Ground Cooling

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For aircraft with separate equipment bays fans are provided which are often located in the undercarriage bays. These are used to provide ambient cooling air flow for the avionics bays when the aircraft is on the ground, and there is only enough bleed air flow from the engines in this case to provide cabin conditioning.

The fans can also be used to cool the equipment if the environmental control system fails.

Cabin Distribution Systems

Cabin distribution systems on both civil and military aircraft are designed to provide as comfortable an environment as possible. The aircrew and passenger's body temperature should be kept to acceptable levels without hot spots, cold spots or draughts. Civil aircraft are designed to maintain good comfort levels throughout the cabin since passengers are free to move about. On some aircraft each passenger has personal control of flow and direction of local air from an air vent above the head (often known as a 'punkah louvre'), although on modern large aircraft total air conditioning is provided. The personal air vent is no longer provided, partly because of the better performance of air conditioning systems, and also because the increased height of passenger cabins means that passengers are no longer able to reach the vent while seated. There are usually additional vents which blow air into the region of the passengers' feet so that there is no temperature gradient between the head and feet. Figure 7.21 shows an example of a Boeing B777 air conditioning pack and an illustration of the way in which air enters at the roof and

is extracted at floor level in a typical cabin. Air flows predominantly down from the roof vents across the front of each passenger, and is extracted at floor level. A proportion of the exhaust air (up to 50%) is re-circulated by being first

Cabin Pressurization

Cabin pressurization is achieved by a cabin pressure control valve which is installed in the cabin wall to control cabin pressure to the required value depending on the aircraft altitude by regulating the flow of air from the cabin. For aircraft where oxygen is not used routinely, and where the crew and passengers are free to move around as in a long range passenger airliner, the cabin will be pressurised so that a cabin altitude of about 8000 ft is never exceeded. This leads to a high differential pressure between the cabin and the external environment. Typically for an airliner cruising at 35 000 ft with a cabin altitude of 8000 ft there will be a differential pressure of about 50 kpa (0.5 atmosphere) across the cabin wall. The crew is able to select a desired cabin altitude from the cockpit and cabin pressurisation will begin when the aircraft reaches this altitude. This will be maintained until the maximum design cabin differential pressure is reached. This is also true for large military aircraft such as surveillance platforms or air-to-air refuelling tankers. For aircraft with the crew in fixed positions, using oxygen routinely as in a military aircraft, the pressurisation system is usually designed so that the cabin altitude does not exceed about 20 000 ft. Figure 7.22a shows a typical fighter aircraft automatic pressurisation

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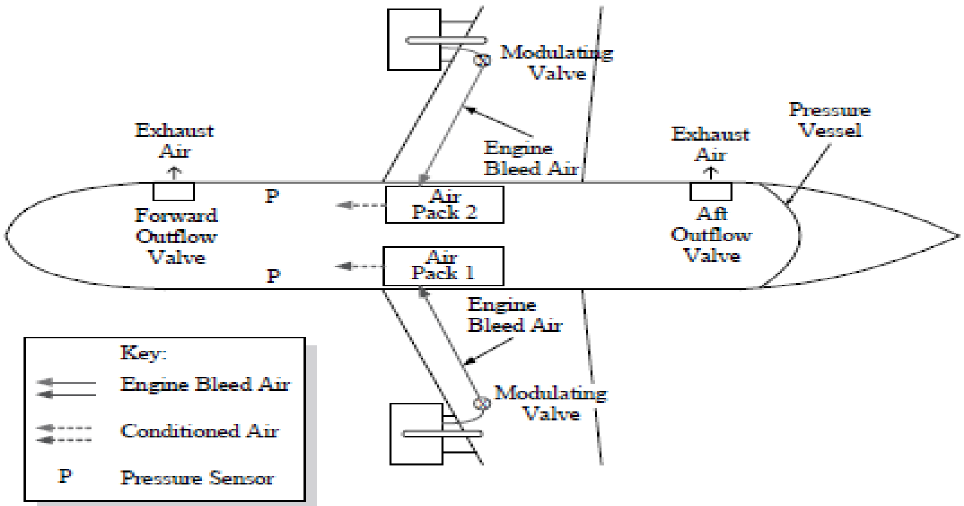
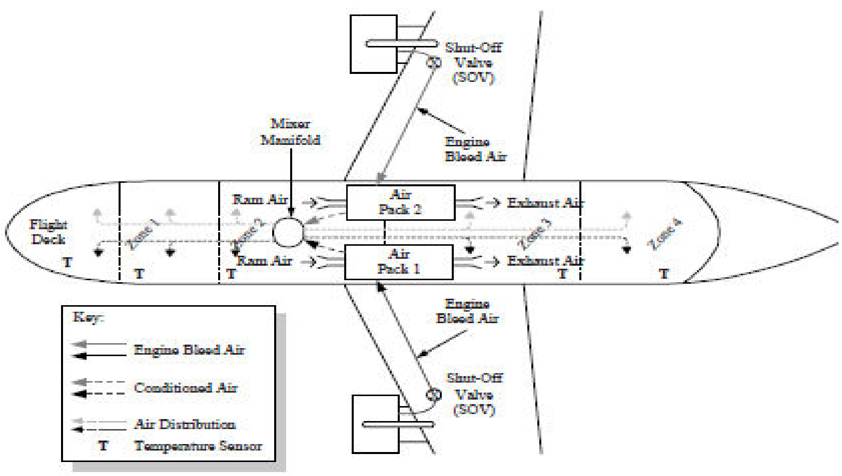
schedule with tolerances plotting Cabin Altitude (y-axis) versus Aircraft Altitude (x-axis). The cabin pressure control





Molecular Sieve Oxygen Concentrators

Until recently the only practical means of supplying oxygen during flight has been from a cylinder or a liquid oxygen bottle. This has several disadvantages, particularly for military aircraft. It limits sortie duration (fuel may not be the limiting factor if in-flight re-fuelling is used), the equipment is heavy and the Molecular Sieve Oxygen Concentrators (MSOC) are currently being developed for military applications. The MSOCs use air taken from the environmental control systems as their source of gas. Most of the gases in air have larger molecules than oxygen. These molecules are sieved out of the air mixture until mostly oxygen remains. This means that a continuous supply of oxygen can be made available without needing to replenish the traditional oxygen storage system after each flight. The

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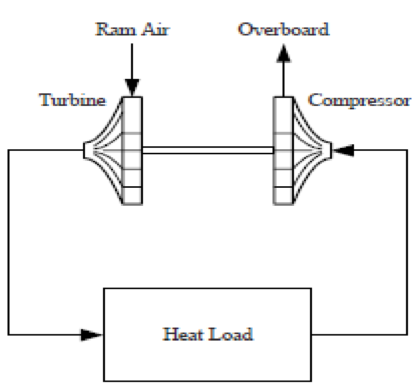
residual inert gases can be used for fuel tank pressurization and inerting. A system designed specifically for the production of inert gases is known as On-Board Inert Gas Generating System (OBIGGS). However, MSOCs have a major disadvantage. If the environmental air supply from the engines stops then so does the supply of oxygen. Therefore, small backup oxygen systems are required for emergency situations to enable the pilot to descend to altitudes where oxygen levels are high enough for breathing. Developments of MSOCs are watched with interest, and further systems may be efficient enough to provide oxygen enriched air for civil aircraft cabins. In military aircraft which are typically designed to fly to altitudes in excess of 50 000 ft, both cabin pressurization and oxygen systems are employed to help alleviate the effects of hypoxia. In cases where aircrew are exposed to altitudes greater than 40 000 ft, either due to cabin de-pressurization or following escape from their aircraft, then additional protection is required. In the event of cabin de-pressurization the pilot would normally initiate an emergency descent to a 'safe' altitude. However, short-term protection against the effects of high altitude is still required.

At altitudes up to 33 000 ft, the alveolar oxygen pressure can be increased up to its value at ground level by increasing the concentration of oxygen in the breathing gas. However, even when 100 per cent oxygen is breathed, the alveolar oxygen pressure begins to fall at altitudes above 33 000 ft. It is possible to overcome this problem by increasing the pressure in the lungs above the surrounding environmental pressure. This is called **positive pressure breathing**. At altitudes above 40 000 ft the rise in pressure in the lungs relative to the pressure external to the body seriously affects blood circulation round the body and makes breathing more difficult. Partial pressure suits are designed to apply pressure to parts of the body to counter the problems of pressure breathing for short durations above 40 000 ft. A partial pressure suit typically includes a pressure helmet and a bladder garment which covers the entire trunk and the upper part of the thighs. The pressure garments are inflated when required by air taken from the environmental control system and are used in conjunction with an inflatable bladder in anti-g trousers which are used primarily to increase the tolerance of the aircrew to the effects of g. Full pressure suits can be used to apply an increase in pressure over the entire surface of the body. This increases duration at altitude. For durations exceeding 10 minutes, however, other problems such as decompression sickness and the effects of exposure to the extremely low temperatures at altitude bottles need replenishing frequently.



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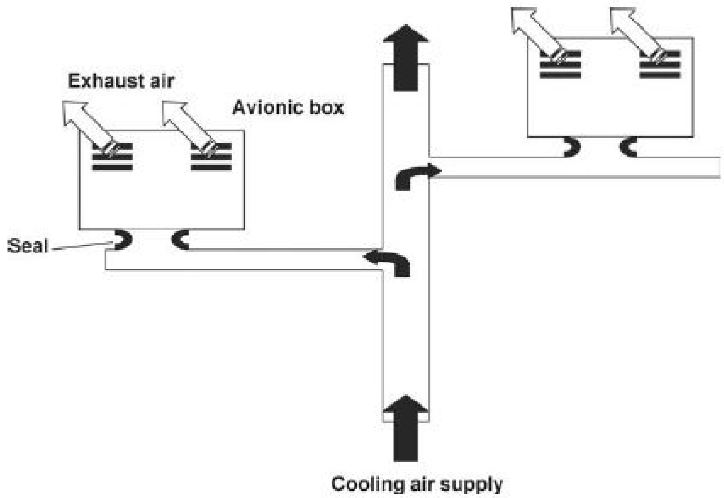

g Tolerance

For aircraft which are likely to perform frequent high g maneuvers such as Typhoon, a ‘relaxed g protection’ system is beneficial.

This consists of

- ***Increased coverage g trousers and***
- ***pressure breathing with g and altitude which requires a breathing gas regulator***
- ***mask capable of providing increased pressure gas, and a pressurized upper body garment to provide external counter pressure (a chest counter pressure garment).***

This enables the pilot to perform repeated high g maneuvers without the need for g straining. It also provides altitude protection in the case of a cabin decompression in a manner similar to a full pressure suit. Engineers strive constantly to improve the agility and combat performance of military aircraft. Indeed technology is such that it is now man who is the limiting factor and not the machine. Accelerations occur whenever there is a change in velocity or a change in direction of a body at uniform velocity. For a centripetal acceleration, towards the centre of rotation, a resultant centrifugal force will act to make the body feel heavier than normal, as illustrated in Figure ..

Forces due to acceleration are measured in g. 1 g is the acceleration due to gravity, i.e. 9.81 m/s. A typical pilot is capable of performing aircraft maneuvers up to 3 or 4 g, i.e. until he feels about three or four times his normal body weight. At g levels above this the heart becomes unable to maintain an adequate supply of oxygenated blood to the brain, which will result in blackout. This is a very dangerous condition, particularly in low-flying aircraft. If the acceleration onset is gradual then the blood supply to the eyes is the first to reduce sufficiently to provide the symptoms of tunneling of vision, before blackout and loss of consciousness occurs. Anti-g trousers are used partially to alleviate the effects of

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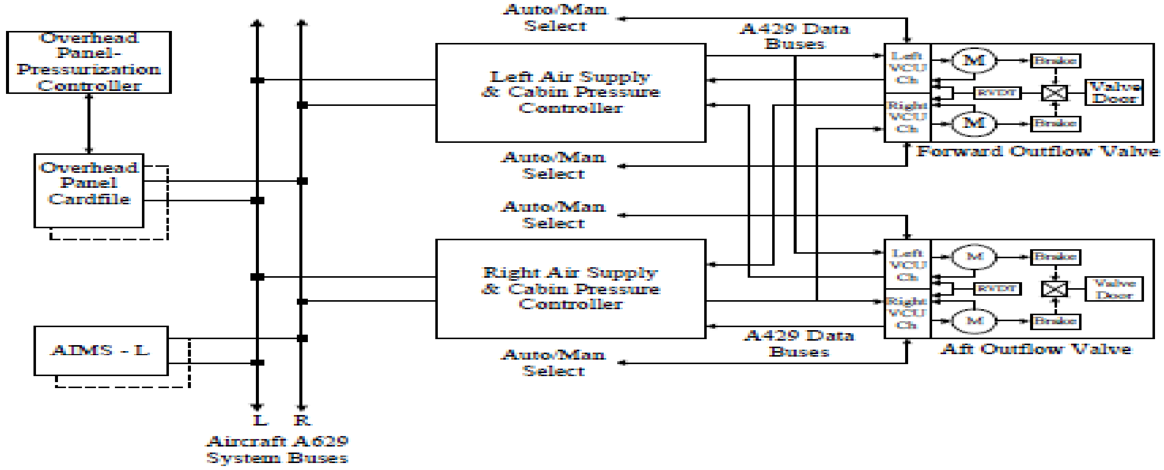
excessive g on the body. The trousers consist of inflatable air bladders retained beneath a non-stretch belt and leggings. The trousers are inflated using air from the

environmental control system. Inflation and deflation of the trousers is typically controlled by an inertial valve. The valve consists of a weight acting on a spring. At the onset of g, as the pilot is pushed down in his seat, the weight compresses the spring which acts to open the valve, thus allowing a supply of air to inflate the bladders in the trousers. The inflation action acts to restrict the flow of blood away from the brain. Using anti-g trousers a typical pilot can perform manoeuvres up to about 8 g. Positive pressure breathing also increases short term resistance to g. Another method of increasing g tolerance is to recline the pilot's seat. This increases the ability of the heart to provide an adequate supply of blood to the brain under high g conditions. However, in practice the seat can only be slightly reclined because of cockpit design problems, pilot visibility and the need to provide a safe ejection pathway to ensure injury free emergency exit from the cockpit.



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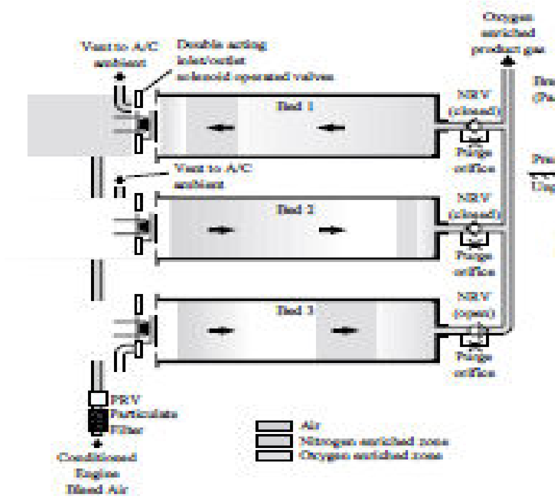

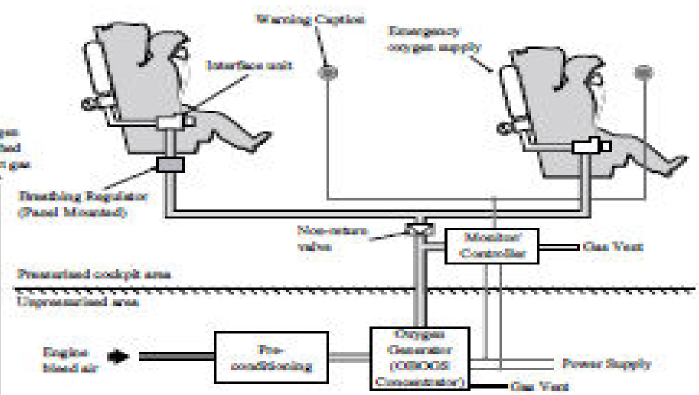

UNIT 5: Electrical Systems

Electrical Loads in Aircraft

Once the aircraft electrical power has been generated and distributed then it is available to the aircraft services. These electrical services cover a range of functions throughout the aircraft depending upon their task. The electrical services may be subdivided into the following categories:

- ***Motors and actuators***
- ***Lighting services***
- ***Heating services***
- ***Subsystem controllers and***
- ***Avionics systems***

Motors and Actuators: Motors are used as a motive force to drive a valve or an actuator from one position to another depending upon the requirements. Typical uses for motors are:

- ***Linear actuation***: Electrical position actuators for engine control; trim actuators for flight control systems.
- ***Rotary actuation***: Electrical position actuators for flap/slat operation
- ***Control valve operation***: Electrical operation of fuel control valves; hydraulic control valves; air control valves; control valves for ancillary systems.
- ***Starter motors***: For starting engines, APU and other systems that require assistance to reach self-sustaining operation.
- ***Pumps*****:** For provision of motive force for fuel pumps, hydraulic pumps; pumping for auxiliary systems.
- ***Gyroscope motors***: For provision of power to run gyroscopes for flight instruments and autopilots; in modern avionics systems gyroscopic sensors are increasingly used for solid state and therefore will not require an AC supply.
- ***Fan motors***: For provision of power to run turbo-coolers and fans for the provision of air to passengers or equipment.

The characteristics of the DC and AC motor types commonly used for aircraft applications are

DC Motors :- DC motors are most likely to be used for **linear and rotary actuation, fuel valve actuation and starter** functions.

AC Motors :- AC motors are most likely to be used for **continuous operation in flight**, i.e. those applications where motors are continuously operating during flight, such as

- Fuel booster pumps,
- Flight instrument gyroscopes and
- Air conditioning cooling fans.

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Lighting: Lighting systems are used mostly during night or low-visibility conditions. The availability of adequate lighting is essential to the safe operation of the aircraft. Lighting systems may be classified as follows:

External Lighting Systems

- ***Navigation lights***
- ***Strobe lights and High Intensity Strobe Lights (HISL)***
- ***Landing/taxi lights***
- ***Formation lights***
- ***Inspection lights (wing/empennage/engine anti-ice)***
- ***Emergency evacuation lights***
- ***Logo lights***
- ***Searchlights (for search and rescue or police aircraft)***

Internal Lighting Systems

- ***Cockpit/flight deck lighting (general, spot, flood and equipment panel)***
- ***Passenger information lighting***
- ***Passenger cabin general and personal lighting***
- ***Emergency/evacuation lighting***
- ***Bay lighting (cargo or equipment bays for servicing)***

Heating: The use of electrical power for heating on aircraft parts like anti/de-icing tail plane and fin leading edges, intake & engine cowlings, propellers, blade and spinners etc,. For windscreen heating heating element and the controlling thermostat are embedded in the windscreen itself.

Avionics Systems :- Radio and Radar equipments like

- **HF1 &HF2**

- **VHF1 &VHF2**

- **CVR**

- **ADF**

- **IFF**

- **DME**

- **GPWS**

- **Wx Radar**

- **GPS etc,.**

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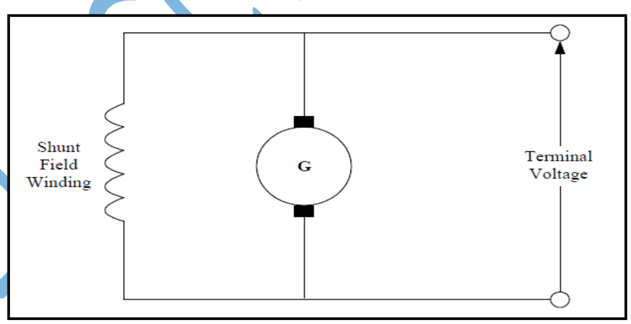
Ground Power:- During servicing period of aircraft on the ground a supply of power is needed. Ground power may be generated by means of a motor-generator set where a prime motor drives a dedicated generator supplying electrical power to the aircraft power **receptacle**. **The usual standard for ground power is 115 VAC, 3-phase 400 Hz**, which is the same as the power supplied by the aircraft AC generators.

Emergency Power Generation : In certain emergency conditions the typical aircraft power generation system may not meet all the airworthiness requirements and additional sources of power generation may need to be used for power the aircraft systems. The aircraft battery may be used on emergency basis for **short time period of 30 minutes**.

Electrical Power Generation

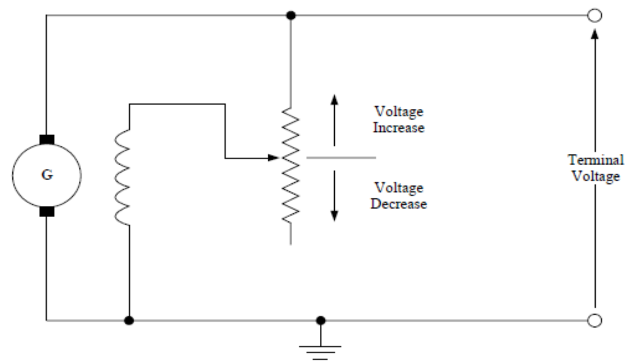
DC Power Generation:-

DC systems use generators to develop a DC voltage to supply aircraft system loads; usually the voltage is 27±2 V DC but there are 270 VDC systems are use in advanced aircraft systems. The generator is regulated to supply 27±2 V DC at all times to the aircraft loads such that any tendencies for the voltage to fluctuate are overcome. DC generators are **self-exciting**, in that they contain rotating electro-magnets that generate the output voltage through commutator in the form of **sine wave**. In aircraft applications the generators are typically shunt-wound in which the high resistance field coils are connected in parallel with the armature as shown in Figure.

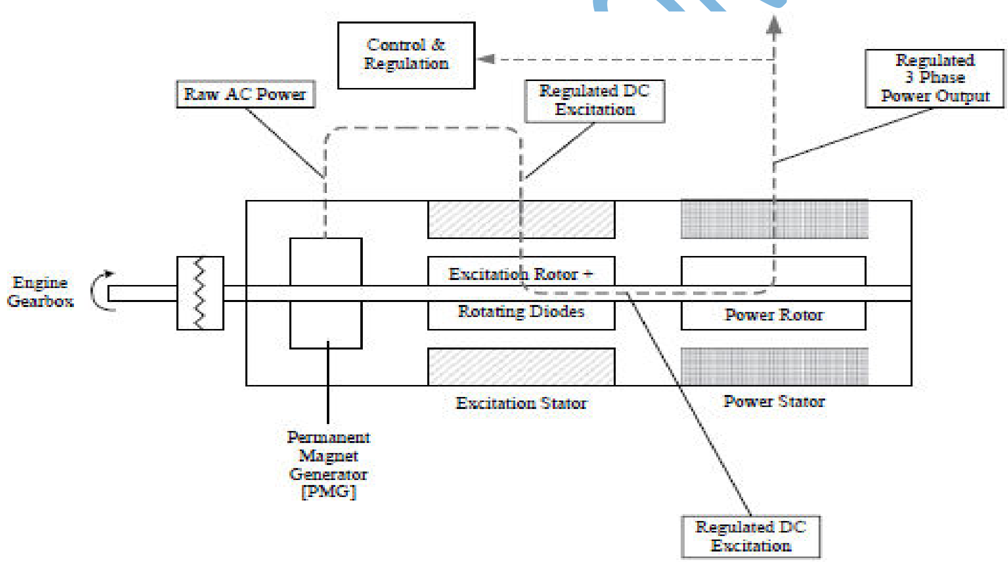


The natural load characteristic of the **shunt-wound generator** is for the voltage to 'drop' with the increasing load current, whereas the desired characteristic is to control the output at a **constant voltage** – nominally 27±2 VDC. For this purpose a voltage regulator is used which modifies the **field current** to ensure that terminal voltage is maintained constant irrespective of engine speed. The principle of operation of the DC voltage regulator is shown in Figure.

UNIT 5: Electrical Systems



AC Power Generation:-An AC system uses a generator to generate a sine wave at constant frequency for a desired voltage. The construction of the alternator is simpler than that of the DC generator in that **no commutator** is required. Modern AC generators work on the principle shown in Figure which is known as a compound generator.



This AC generator may be regarded as several machines sharing the same shaft as shown in above diagram.

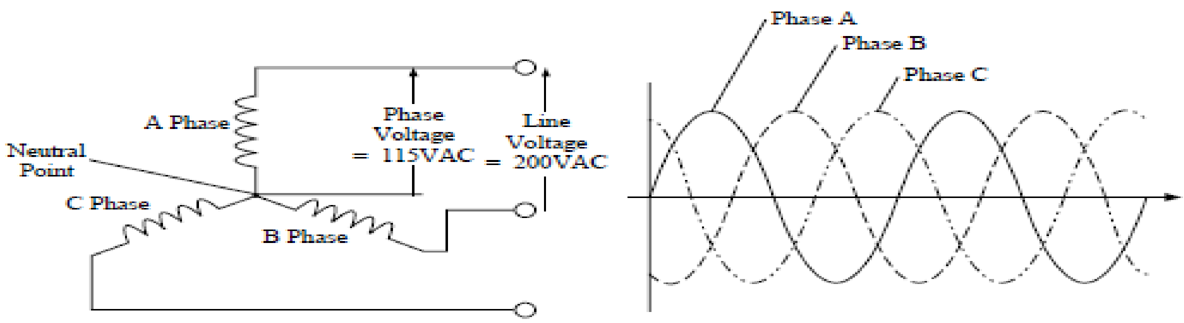
• A Permanent Magnet Generator (PMG)

• An excitation stator surrounding an excitation rotor containing rotating diodes

• A power rotor encompassed by a power stator

UNIT 5: Electrical Systems

The flow of power through this generator is highlighted by the dashed line. The PMG generates 'raw' (variable frequency, variable voltage) power sensed by the control and regulation section that is part of the generator controller. This modulates the flow of DC current into the excitation stator windings and therefore controls the voltage generated by the excitation rotor.



Most AC systems used on aircraft use a **three-phase** system that is the alternator generates three sine waves; each phase positioned 120° out of phase with the others. These phases are most often connected in a star configuration with one end of each of the phases connected to a neutral point as shown in figure. In this layout the phase voltage of a standard aircraft system is 115 VAC, whereas the line voltage is 200 VAC with 400 cycles/sec or 400 Hz frequency.

The main advantage of AC power is that it operates at a higher voltage; 115 V AC rather than 28 V DC for the DC system. The use of a higher voltage is not an advantage as it requires better **standards of insulation**. For lower voltage it drops (proportional to current) and power losses. *To avoid more loses use of 115VAC is generally accepted.*

Power Generation Control : The primary elements of power system control are:

• **DC systems**

– *Voltage regulation*

– *Parallel operation*

– *Protection functions*

• **AC systems**

– *Voltage regulation*

– *Parallel operation*

– *Supervisory functions*

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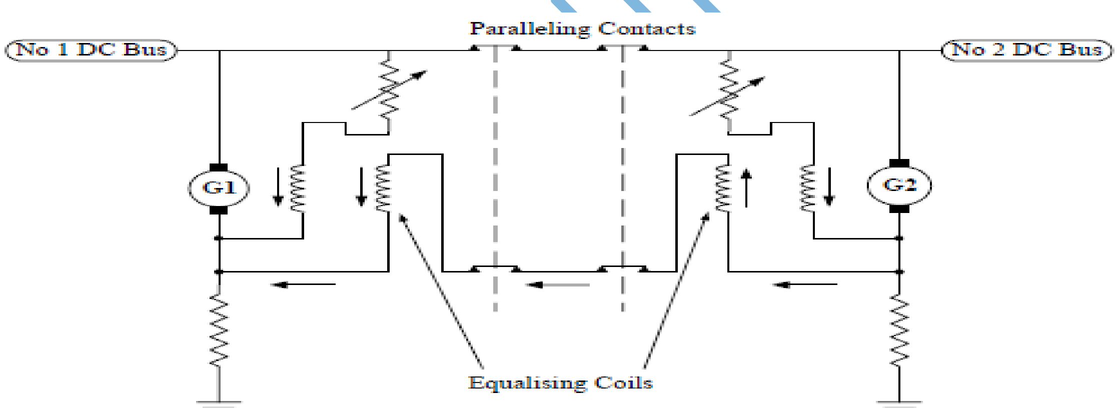
DC System Generation Control

(a) **Voltage Regulation:-**DC generation is done by means of shunt-wound self-exciting field. The principle of voltage regulation is done by connecting a variable resistor in series with the field winding such that variation of the resistor alters the resistance of the field winding; hence the field current and output voltage may be varied. In actual fact the regulation is required to be an automatic function that takes account of load and engine speed.

(b) Parallel operation:-In multi-engine aircraft each engine will be driving its own generator and in this situation it is desirable that 'no-break' or uninterrupted power is to be provided in cases of engine or generator failure. A number of sensitive aircraft instruments and navigation devices which comprise some of the electrical loads may be disturbed and may need to be restarted or re-initialized following a power interruption. In order to satisfy this requirement generators are paralleled to carry an equal proportion of the electrical load between them. Individual generators are controlled by means of voltage regulators that automatically compensate for variations. In the case of parallel generator operation there is a need to interlink the voltage regulators such that any unequal loading of the generators can be adjusted by means of corresponding alterations in field current. This paralleling feature is more often known as an equalizing circuit and therefore provides 'no break' power in the event of a major system failure. A simplified diagram showing the main elements of DC parallel operation is shown in fig.



(c) Load Protection :-The primary conditions for which protection needs to be considered in a DC system are as follows:

• ***Reverse current***. In a DC system it is evident that the current should flow from the generator to the bus-bars and then distribution systems. In a fault situation it is possible for current to flow in the reverse direction and the primary system components need to be protected from this eventuality. This is usually achieved by means of reverse current circuit breakers or relays which prevents of damaging sensitive equipments.

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• ***Overvoltage protection***. Faults in the field excitation circuit can cause the generator to over-excite and thereby regulate the supply voltage to an erroneous **OVERVOLTAGE** condition. This could then result in the electrical loads being subject to conditions that could cause permanent damage. Overvoltage protection senses these failure conditions and opens the line contactor taking the generator offline.

• ***Under voltage protection***. In a single generator system **UNDER VOLTAGE** is a similar fault condition as the reverse current situation already described. However, in a multi-generator configuration with paralleling by means of an equalizing circuit, the situation is different. Here an under voltage protection capability is essential as the equalizing circuit is always trying to raise the output of a lagging generator; in this situation the under voltage protection is an integral part of the parallel load sharing function.

AC Power Generation Control

(a) Voltage Regulation

AC generators differ from DC generators in that they require a separate source of DC excitation for the field windings. The subject of AC generator excitation is a complex topic for which the technical solutions vary according to whether the generator is frequency-wild or constant frequency. Some of these solutions comprise sophisticated control loops with error detectors, pre-amplifiers and power amplifiers.

(b) Parallel Operation

In the same way that DC generators are operated in parallel to provide 'no break' power, AC generators may also be controlled in a similar fashion. This technique only applies to constant frequency AC generation as it is impossible to parallel frequency-wild or Variable Frequency (VF) AC generators. In fact many of the aircraft loads such as anti/de-icing heating elements driven by VF generators are relatively frequency insensitive and irrespective of power interruption frequency is need for smooth operation of VF generators. No break power transfer is also important during start up /shutdown in the transition from/to ground power, and/or APU generated power, to/from aircraft main generator power, to avoid malfunction or resetting of electrically powered equipment. Constant speed or constant frequency AC generation depends upon the tracking accuracy of the constant speed drives of the generators involved.

The generator output voltages depend upon the relevant performance of the voltage regulators and field excitation circuitry. To accomplish reactive load sharing requires the use of special transformers called mutual reactors, error detection circuitry and pre-amplifiers/power amplifiers to adjust the field excitation current. Therefore by a using a combination of trimming the speed of the Constant Speed Drives (CSDs) and balancing the field excitation to the generators, real and reactive load components may be shared equally between the generators.

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(c) Supervisory or Protection Functions

Typical supervisory or protection functions undertaken by a typical AC generator controller or GCU are:

- ***Over voltage***
- ***Under voltage***
- ***Under/over excitation***
- ***Under/over frequency***
- ***Differential current protection***
- ***Correct phase rotation***

Circuit Breakers

Circuit breakers perform the function of protecting a circuit in the event of an electrical overload. Circuit breakers serve the same purpose as fuses or current limiters. A circuit breaker comprises a set of contacts which are closed during normal circuit operation. The device has a mechanical trip mechanism which is activated by means of a bi-metallic element. When an overload current flows, the bi-metallic element causes the trip mechanism to activate, thereby opening the contacts and removing power from the circuit. A push button on the front of the unit protrudes showing that the device has tripped. Pushing in the push button resets the breaker but if the fault condition still exists the breaker will trip again. Physically pulling the button outwards can also allow the circuit breaker to break the circuit, perhaps for equipment isolation or aircraft maintenance reasons. Circuit breakers are rated at different current values for use in differing current carrying circuits. This enables the trip characteristic to be matched to each circuit. The trip characteristic also has to be selected to coordinate with the feeder trip device upstream. Circuit breakers are literally used by the hundred in aircraft distribution systems; it is not unusual to find 500–600 devices throughout a typical aircraft system. Figure shows a circuit breaker and a typical trip characteristic.

Modern Aircraft Electrical Power Generation Types

The different types of electrical power generation currently being considered are shown in Figure. The Constant Frequency (CF) 115 VAC, 3-phase, 400 Hz generation types are modified by the Integrated Drive Generator (IDG), Variable Speed Constant Frequency (VSCF) Cycloconverter and DC Link options. Variable Frequency (VF) 115 VAC, 3-phase power generation – sometimes termed '**frequency wild**' – is a inexpensive form of power generation, the disadvantage with this is that some motor loads may require motor controllers. Military aircraft in the US are inclining towards 270 VDC systems. Permanent Magnet Generators (PMGs) are used to generate 28 VDC emergency electrical power for high integrity systems.

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The following are types of modern power generation methods-

• ***Constant frequency using an IDG***

• ***Variable frequency***

• ***Variable Speed Constant Frequency (VSCF) options***

Constant Frequency/IDG Generation

The main features of CF/IDG power are

- Constant frequency AC power is most commonly used on turbofan aircraft today.
- System is expensive to purchase & maintain; primarily due to complexity of Constant Speed Drive (CSD).
- Single company monopoly on supply of CSD/IDG.
- Alternate methods of power generation are under consideration.

In common with all the other power generation types this gives twice the power at max speed to ground idle speed. The Constant Speed Drive (CSD) in effect acts as an automatic gearbox, maintaining the generator shaft speed at a constant rpm which results in a constant frequency output of 400 Hz,. The drawback of the hydro-mechanical CSD is that it needs to be correctly maintained in terms of oil charge level and oil cleanliness.

Variable Frequency Generation:-The main features of VFG are (Airbus A380 and Boeing 787)

- Simplest form of generating power, cheapest and most reliable
- Variable frequency has impact upon other aircraft subsystems
- Motor controllers may be needed for certain aircraft loads
- Beginning to be adopted for new programmes: gains outweigh disadvantages

In this method no attempt is made to nullify the effects of the 2:1 engine max speed idle speed ratio and the power output, though regulated to 115 VAC, suffers a frequency variation typically from 380 to 720 Hz. This wide band VF power has an effect on frequency sensitive aircraft loads, the most obvious being the effect on AC electric motors that are used in many aircraft systems VF is being widely adopted in the business of jet community as their power requirements take them above the 28 VDC/12 kW limit of twin 28VDC systems.

Variable Speed Constant Frequency Generation:-The mail features of VSCF are

- Conversion of VF electrical power to CF is accomplished by electronic controlled power switching
- DC Link & Cyclo-converter options available
- Not all implementations have proved to be robust/reliable –Cyclo-converter shows most promise

In this technique the variable frequency power produced by the generator is electronically converted power. These are two types :

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• ***DC link***: In the DC link the raw power is converted to an intermediate DC power stage – the DC link – before being electronically converted to 3-phase AC power.

• ***Cyclo-converter***: The cyclo-converter uses a different principle. 6- phases are generated at relatively high frequencies in excess of 3000 Hz and the solid state devices switch between these multiple phases in a predetermined and carefully controlled manner. The effect is to electronically commutate the input and provide three phases of constant frequency 400 Hz power. Though this appears to be a complex technique it is in fact quite elegant and cyclo-converter systems have been successfully used on military aircraft in the US:

Power Distribution

Primary Power Distribution:-The primary power distribution system consolidates the aircraft electrical power inputs from the following sources:

• Main aircraft generator; by means of a Generator Control Breaker (GCB) under the control of the GCU.

• Alternate aircraft generator – in the event of generator failure – by means of a Bus Tie Breaker under the control of a Bus Power Control Unit (BPCU)

• APU generator; by means of an APU GCB under the control of the BPCU

• Ground power; by means of an External Power Contactor (EPC) under the control of the BPCU

• Backup converter, by means of a Converter Control Breaker (CCB) under the control of the VSCF Converter (B777 only)

• RAT generator when deployed by the emergency electrical system

The power switching used in these cases is a power contactor or breaker. These are special high power switches that usually switch power in excess of 20 amps per phase. As well as the power switching contacts auxiliary contacts are included to provide contactor status – 'Open' or 'Closed – to other aircraft systems.

` Higher power aircraft loads are increasingly switched from the primary aircraft bus bars by using **Electronic Load Control Units (ELCUs) or 'smart contactors'** for load protection.

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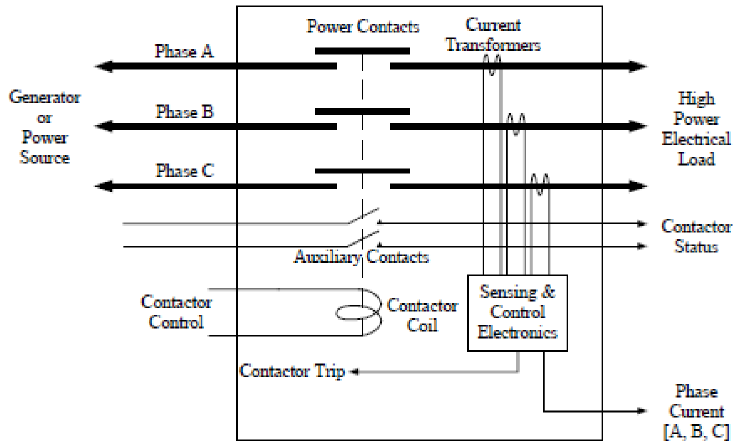
Secondary Power Distribution



Power Switching

In order to reconfigure or to change the state of a system it is necessary to switch power at various levels within the system. At the high power levels that prevail at the primary power part of the system, power switching is accomplished by high power electromagnetic devices called contactors. These devices can switch hundreds of amps and are used to switch generator power on to the primary bus-bars in both DC and AC systems. The devices may be arranged so that they magnetically latch, that is they are magnetically held in a preferred state or position until a signal is applied to change the state.

In other situations a signal may be continuously applied to the contactor to hold the contacts closed and removal of the signal causes the contacts to open. Primary power contactors and ELCUs have been described earlier in the chapter. For switching currents below 20 amps or so relays are generally used. These operate in a similar fashion to contactors but are lighter, simpler and less expensive. Relays may be used at certain places in the primary electrical system. However, relays are more likely to be employed for switching of medium and high power secondary aircraft loads or services.

For lower currents still where the indication of device status is required, simple switches can be employed. These switches may be manually operated by the crew or they may be operated by other physical means as part of the aircraft operation. Such switches are travel limit switches, pressure switches, and temperature switches and so on.

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Power Conversion and Energy Storage

There are many occasions within an aircraft electrical system it is required to convert power from one form to another. Typical examples of power conversion are:

• Conversion from DC to AC power – this conversion uses units called inverters to convert 28 VDC to 115 VAC single phase or three-phase power.

• Conversion from 115 VAC to 28 VDC power – this is a much used conversion

using units called Transformer Rectifier Units (TRUs)

• Conversion from one AC voltage level to another; a typical conversion would

be from 115 VAC to 36 VAC

• Battery charging – as previously outlined it is necessary to maintain the state

of charge of the aircraft battery by converting 115 VAC to a 28 VDC battery

charge voltage.

• In more recent military platforms such as F-22 and F-35 utilising 270 VDC; conversion to 115 VAC, 3 phase, 400 Hz AC and 28 VDC is required to power

legacy equipments originally designed to operate using these voltages

(a) Inverters

Inverters which convert 28VDC power into 115VAC single phase electrical power. This is usually required in a civil application to supply captains or first officer's instruments following an AC failure. Alternatively, under certain specific flight conditions, such as auto-landing, the inverter may be required to provide an alternative source of power to the flight instruments in the event of a power failure occurring during the critical auto landing time.

(b) Transformer Rectifier Units (TRUs)

TRUs are probably the most frequently used method of power conversion on modern aircraft electrical systems. Most aircraft have a significant 115VAC three-phase AC power generation capability inherent within the electrical system and it is usual to convert a significant portion of this to 28 VDC by the use of TRUs. **TRUs comprise star primary and dual star/delta secondary transformer windings together with three-phase full wave rectification and smoothing to provide the desired 115 VAC/28 VDC conversion**. TRUs are usually simple, unregulated units; that is the voltage is not controlled to maintain 28 VDC as load is increased and accordingly the load characteristic tends to 'drop'. In some specialist military applications this feature is not desirable and regulated TRUs are used. TRUs are usually operated in isolation; however, when regulated they may also be configured to operate in parallel in a similar way to the parallel operation of DC generators.

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(c) Auto-Transformers

In certain parts of an electrical system simple auto-transformers may be used to provide a simple voltage step-up or step-down conversion. An example of this is the 115 V/26 VAC transformation used to provide 26 VAC aircraft lighting supplies direct from main 115 VAC bus-bars in the easiest way.

(d) Battery Chargers

Battery chargers share many of the attributes of TRUs and are in fact dedicated units whose function is purely that of charging the aircraft battery. In some systems the charger may also act as a standby TRU providing a boosted source of DC power to the battery in certain system modes of operation. Usually the task of the battery charger is to provide a controlled charge to the battery without overheating and for this reason battery temperature is usually closely monitored.

(e) Batteries

Batteries provide an electrical storage medium independent of the primary generation sources. Its main purposes are:

- To assist in damping transient loads in the DC system
- To provide power in system startup modes when no other power source is
- available
- To provide a short-term high-integrity source during emergency conditions while alternative/backup sources of power are being brought on line.

The capacity of the aircraft battery is limited and is measured in terms of ampere-hours. This parameter effectively describes a current/time capability or storage capacity. Thus a 40 ampere-hour battery when fully charged would have the theoretical capacity of feeding a 1 ampere load for 40 hours or a 40 ampere load for 1 hour. In fact the capacity of the battery depends upon the charge sustained at the beginning of the discharge and this is a notoriously difficult parameter to quantify. Most modern aircraft systems utilize battery chargers to maintain the battery charge at moderately high levels during normal system operation thereby assuring a reasonable state of charge should solo battery usage be required.

The battery most commonly used is the nickel-cadmium (Ni-Cd) type which depends upon the reaction between nickel oxides for the anode, cadmium for the cathode and operating in a potassium hydroxide electrolyte. **Lead-acid batteries are not favoured in modern applications due to corrosive effects.** To preserve battery health it is usual to monitor its temperature which gives a useful indication of over-charging and if thermal runaway is likely to occur.

More-Electric Aircraft

In the early ’80s NASA funded a number of studies addressing the Integrated Digital Electrical Airplane (IDEA). The IDEA concept could improve the efficiency of a 250–300 seater replacement for an aircraft such as the Lockheed L1011 (Tristar). In these studies the following areas are covered:

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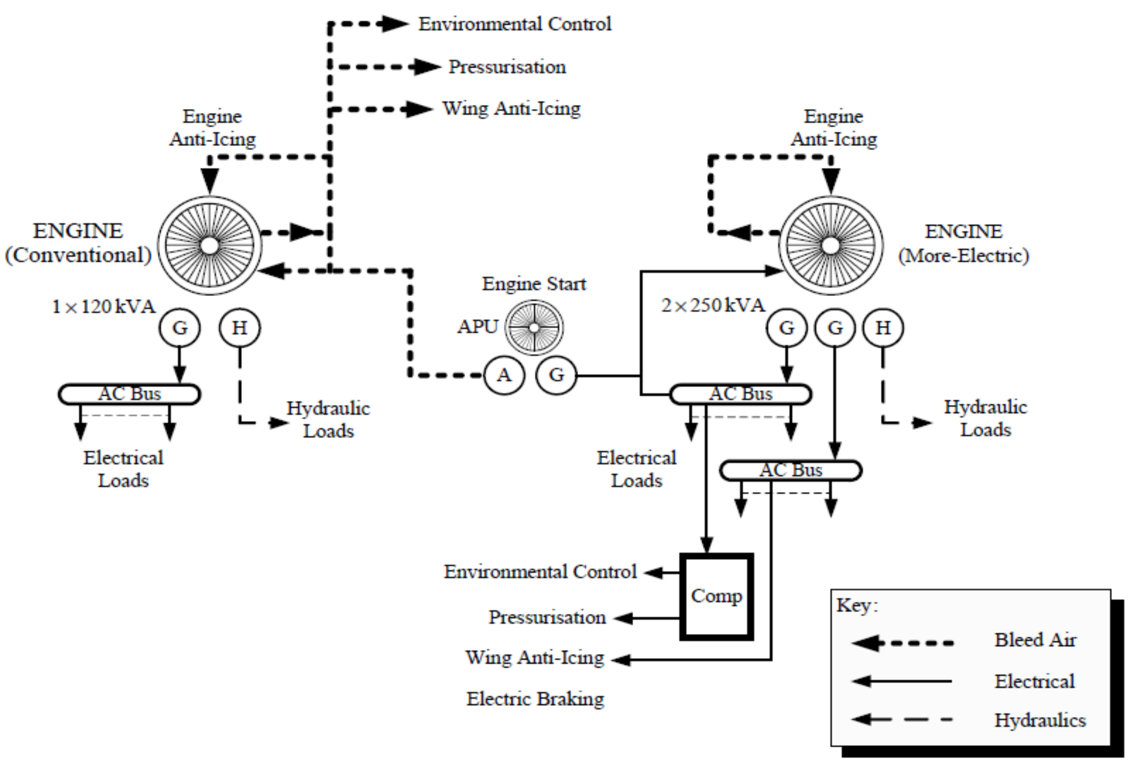
- *Flight Control Technology* – relaxed stability augmentation leading to a reduction in drag as a result reduction in -size of the tail plane and fuel savings.
- *Wing Technology* – use of efficient high aspect ratio wings using gust alleviation modes of the FCS to improve range and fuel consumption and reduce wing bending moments.
- *Engine Power Extraction* – the reduction of engine power extraction losses by minimising the use of high pressure bleed air and hydraulic power and maximising the use of more efficient electrical power extraction techniques.
- *Flight Control Actuation* – the use of electro-mechanical actuation in lieu of hydro-mechanical actuation systems.
- *Advanced Electrical Power Systems* – the development of new systems to generate and distribute electrical power as an adjunct to more efficient engine power extraction.

Flight control system and flight control actuation developments are already underway or are embodied in major civil programmes as evidenced by systems on the A380 and Boeing 787 aircraft. The more-electric features are used in Airbus A380 and B787.



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More-Electric Engine

The engine also benefits from the adoption of more-electric technology to address the following troublesome issues:

• ***Reduction of bleed air offtakes*** – As engine bypass ratios increases; there will be reduction in engine efficiency and increasing fuel consumption. The reduction of engine HP air offtakes and the use of more-electric techniques has a considerable improvement on these adverse effects.

• ***Removal of the accessory gearbox*** – Engine accessory gearboxes are becoming

increasingly complex as the number of drives and power offtakes increase.

• ***Oil-less engine*** – The engine oil system is complex on many engines, usually comprising a number of oil pumps, filter assemblies, coolers etc. The generation/

conversion losses from the aircraft electrical generators reject heat into the engine. Great savings could be made if the ***oil system could be replaced with an alternative form of supporting the rotating engine assemblies***. Electromagnetic bearing technology has been demonstrated on both sides of the Atlantic. However in order to be totally practicable, additional technologies have to be developed which permit the removal of the accessory gearbox and its associated power offtakes from the engine.

• ***IGV/VSV control*** – Many engines use Variable Inlet-Guide Vanes (VIGVs), and Variable Stator Vanes (VSVs), to control the airflow into the engine central core. These may be variously powered by hydraulics, pneumatics (bleed air). Programmes are underway to examine the feasibility of using electrical actuation techniques to replace the fluidic power media.

• ***Distributed engine control*** – Present primary engine control is by means of a Full Authority Digital Engine Control (FADEC) which is normally located on the engine fan casing. However there are many features of engine control which are distributed around the engine – such as reverse thrust, presently pneumatically actuated – which would need to be actuated by alternative means in a more-electric engine. This leads to the possibility of using distributed engine control.

Electrically driven fuel pump – Engine fuel is pressurized by means of a shaft driven High Pressure (HP) pumpin conventional aircraft. The HP fuel pump is typically sized for engine starting and when driven directly by the engine (via a reduction gearbox) operation at high engine rotational speeds produces excess fuel flow that must be spilled back to the pump inlet. This is aggravated at cruise altitudes where the fuel required by the engine is about five times lower resulting in even more wasted pumping energy. ***In an all-electric engine the HP pump would be electrically driven at the optimum speed for the prevailing operating condition***.

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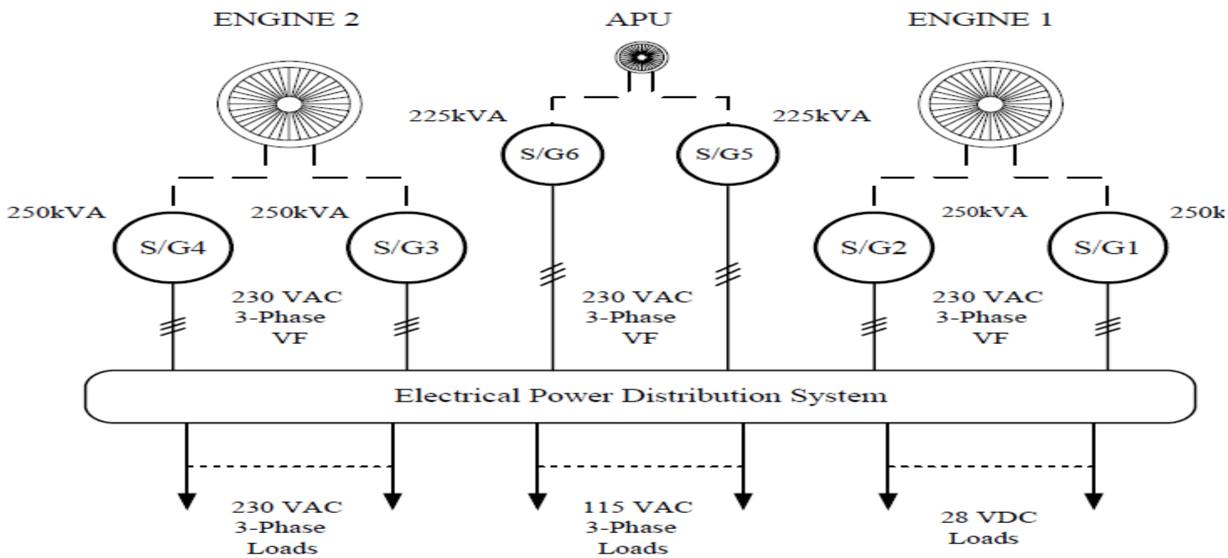
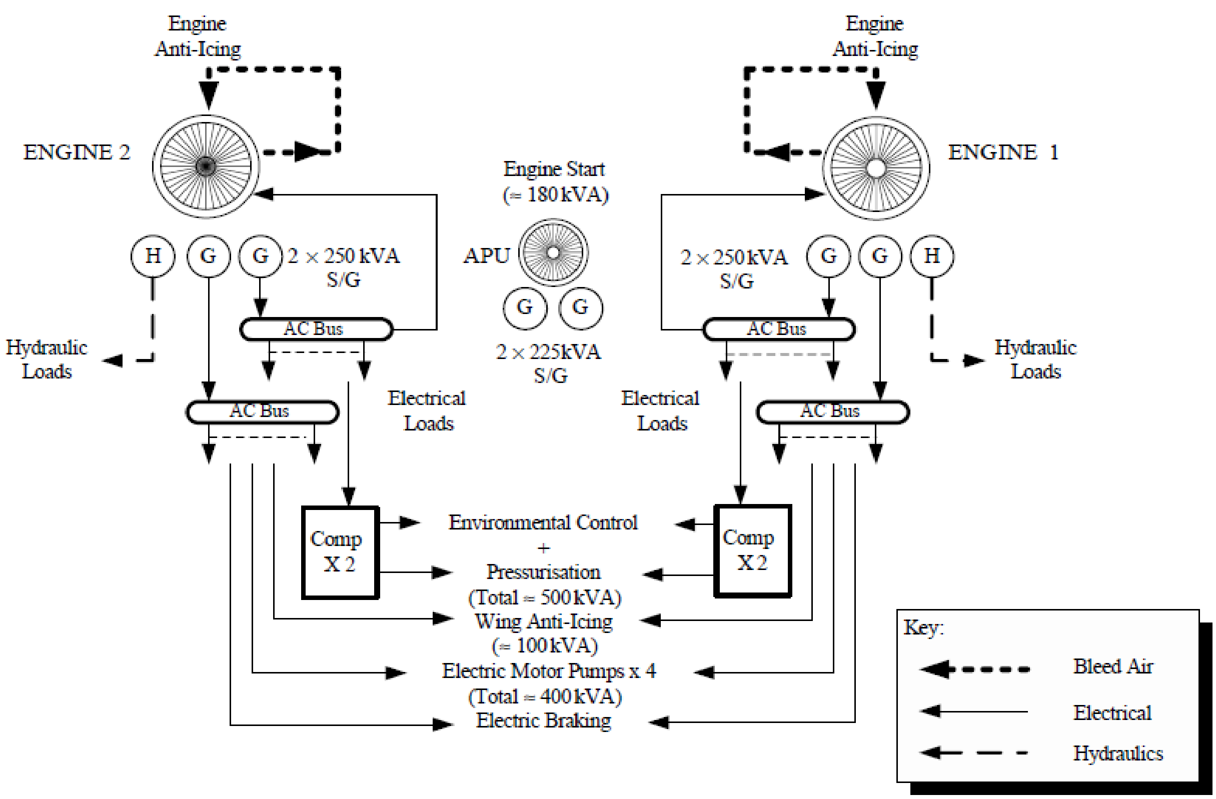


Boing 787 Elecrtic system



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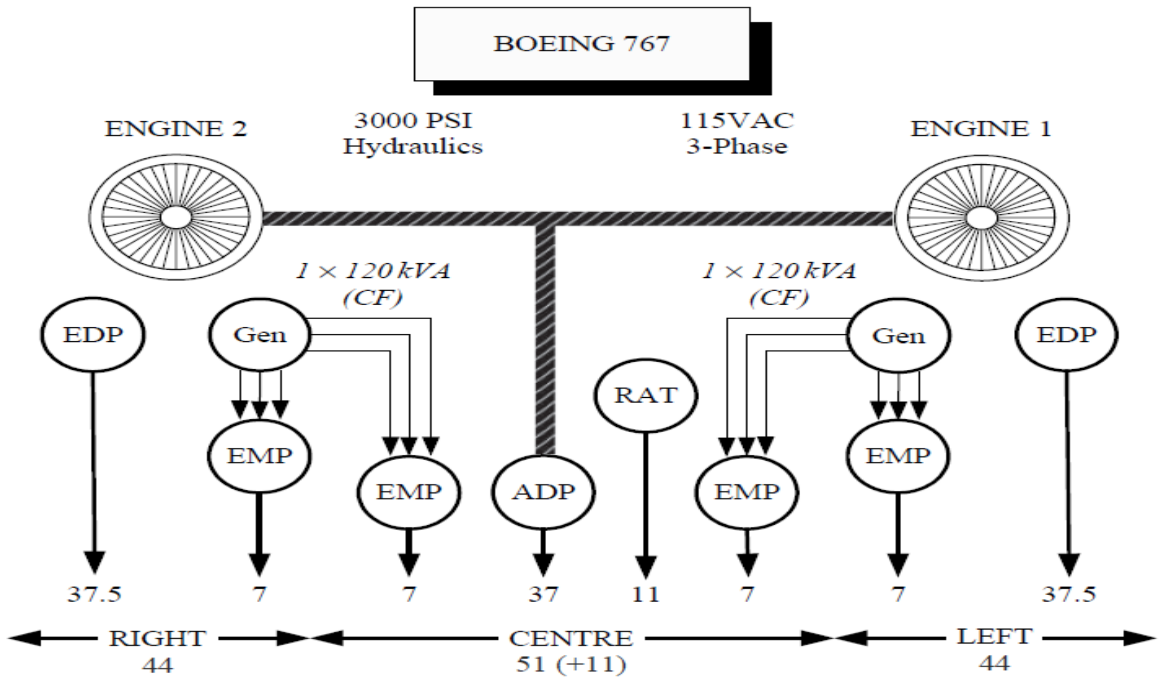
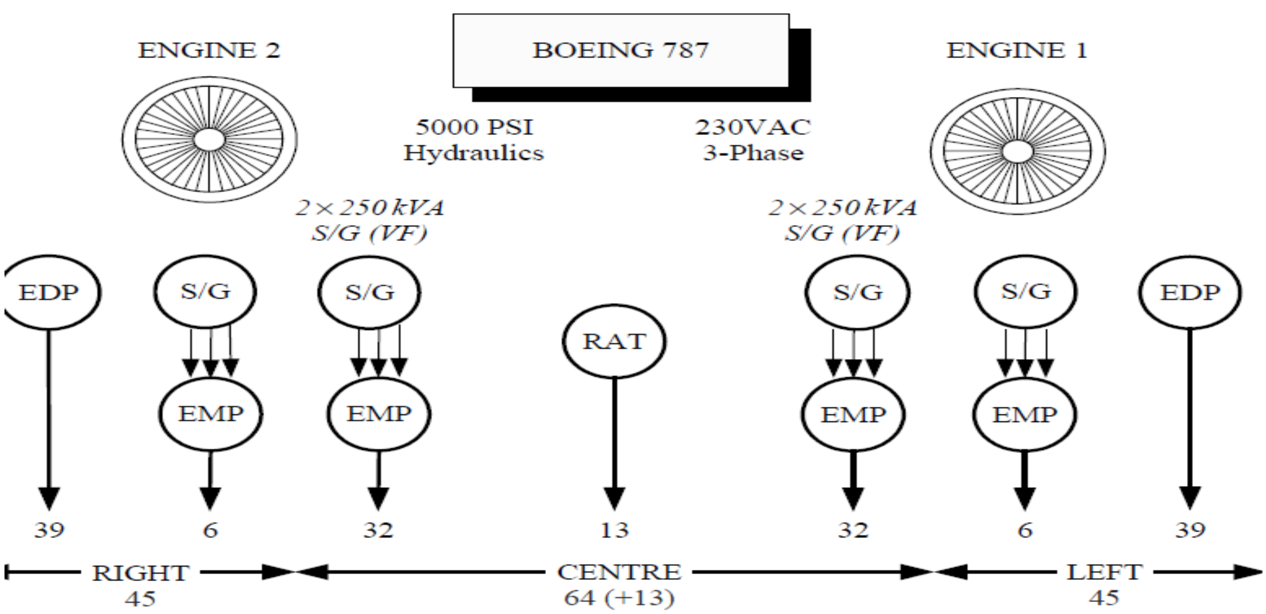
Boing 787 Electrical loads

Comparision of Boing 767 and Boing 787 (Conventional and More Electric)





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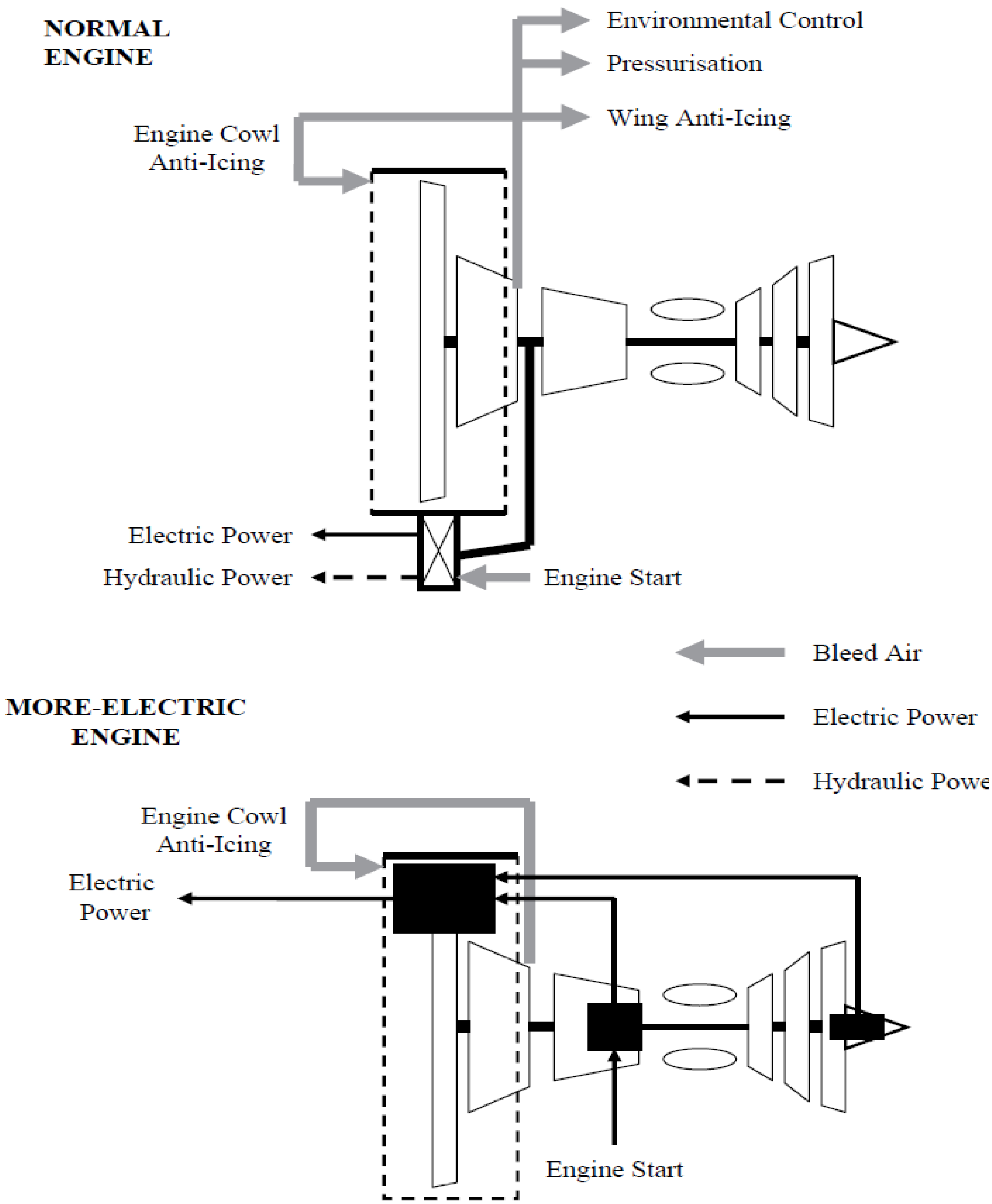
Comparison of Conventional and More electric engine



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More Electric Engine Components