The JET ENGINE

the Jet engine

Acknowledgements

Contents

Rolls-Royce Trent 800

Developed from the RB211, the Trent covers a thrust range of 71,000 lb to 92,000 lb thrust, with the capability to grow beyond 100,000 lb. The Trent 800 features a 110 inch diameter wide-chord fan, high flow compressors and Full Authority Digital Engine Control (FADEC).

Detailed engineering design began in 1988 to meet the propulsion requirements of the Airbus A330 (Trent 700) and Boeing 777 (Trent 800). The Trent first ran in August 1990, and in January 1994 a Trent 800 demonstrated a world record thrust of 106,087 lb.

The engine entered service in March 1995 in the Airbus A330.

Introduction

This book has been written to provide a simple and self-contained description of the working and underlying principles of the aero gas turbine engine. The use of complex formulae and the language of the specialist have been avoided to allow for a clear and concise presentation of the essential facts. Only such description and formulae, therefore, as are necessary to the understanding of the function and the theory of the engine are included.

It will be noted that the emphasis in this book is on the turbo-jet engine and that no special part deals with the propeller-turbine engine. This is because the working principles of both engine types are essentially the same. However where differences in function or application do exist, these are described. The aero gas turbine is being continually developed to provide improved performance for each new generation of aircraft; the fourth edition of this book has been revised and expanded to include the latest aero gas engine technology.

Rolls-Royce RB183 Mk 555
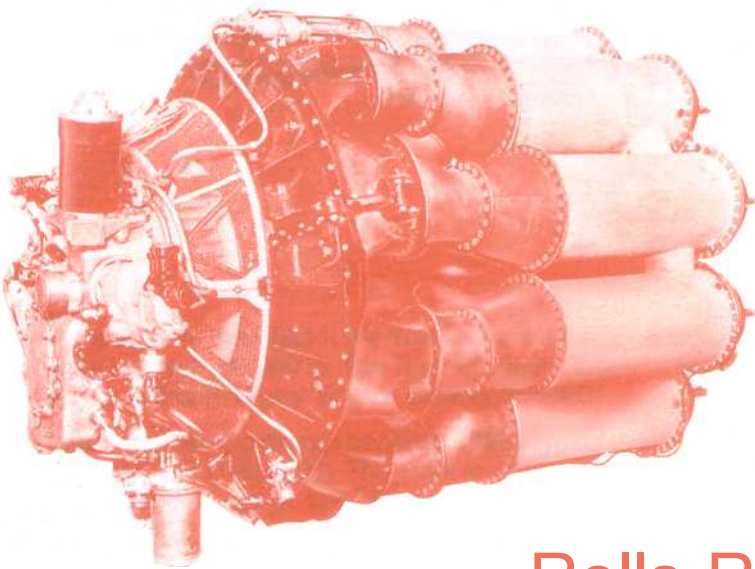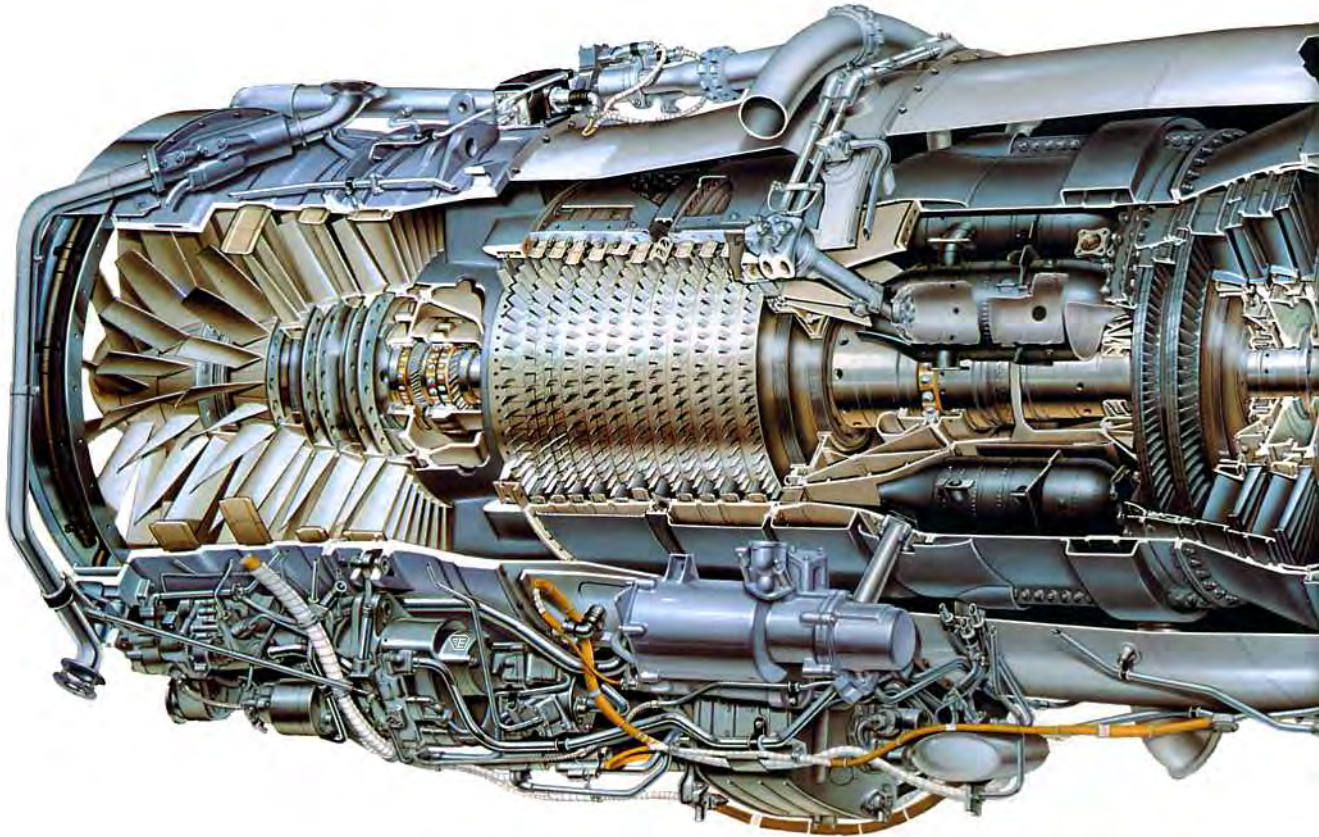
On 1 April, 1943, Rolls-Royce assumed responsibility for the Power Jets W2B which, a month earlier, had made its first flight in the Gloster E28/39 at 1200lb thrust. Later known as the B23 Welland it was, during April, put through a 100 hr test at the design rating of 1600 lb thrust. In June, 1943, it flew in a Gloster Meteor at 1400lb thrust. Production Welland-Meteors were in action against V-1 flying bombs in August 1944.

Rolls-Royce B23 Welland

1: Basic mechanics

Contents

INTRODUCTION

1.   The development of the gas turbine engine as an aircraft power plant has been so rapid that it is difficult to appreciate that prior to the 1950s very few people had heard of this method of aircraft propulsion. The possibility of using a reaction jet had interested aircraft designers for a long time, but initially the low speeds of early aircraft and the unsuitaly of a piston engine for producing the large high velocity airflow necessary for the 'jet' presented many obstacles.

2.   A French engineer, René Lorin, patented a jet propulsion engine (fig. 1-1) in 1913, but this was an athodyd (para. 11) and was at that period impossible to manufacture or use, since suitable heat resisting materials had not then been developed and, in the second place, jet propulsion would have been extremely inefficient at the low speeds of the aircraft of those days. However, today the modern ram jet is very similar to Lorin's conception.

3.   In 1930 Frank Whittle was granted his first patent for using a gas turbine to produce a propulsive jet,
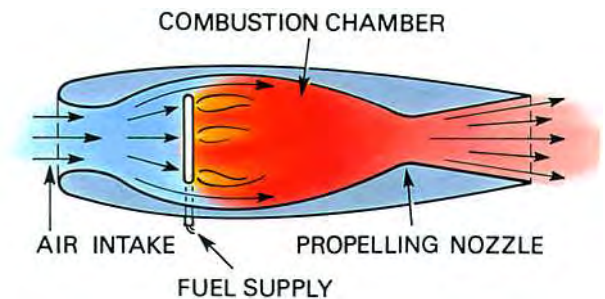
Fig. 1-1   Lorin's jet engine.

but it was eleven years before his engine completed its first flight. The Whittle engine formed the basis of the modern gas turbine engine, and from it was developed the Rolls-Royce Welland, Derwent, Nene and Dart engines. The Derwent and Nene turbo-jet engines had world-wide military applications; the Dart turbo-propeller engine became world famous as the power plant for the Vickers Viscount aircraft. Although other aircraft may be fitted; with later engines termed twin-spool, triple-spool, by-pass, ducted fan, unducted fan and propfan, these are inevitable developments of Whittle's early engine.
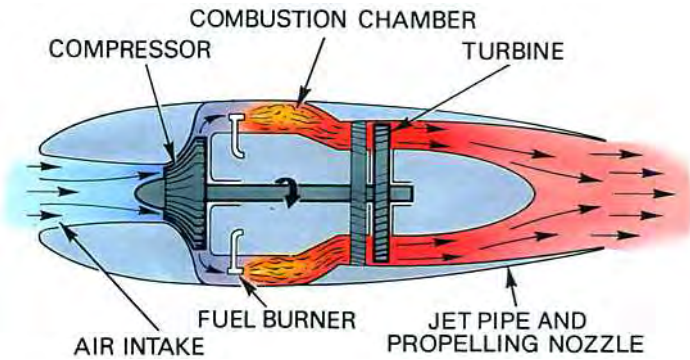
Fig. 1-2   A Whittle-type turbo-jet engine.

4. The jet engine (fig. 1-2), although appearing so different from the piston engine-propeller combination, applies the same basic principles to effect propulsion. As shown in fig. 1-3, both propel their aircraft solely by thrusting a large weight of air backwards.

5. Although today jet propulsion is popularly linked with the gas turbine engine, there are other types of jet propelled engines, such as the ram jet, the pulse jet, the rocket, the turbo/ram jet, and the turbo-rocket.

PRINCIPLES OF JET PROPULSION

6. Jet propulsion is a practical application of Sir Isaac Newton's third law of motion which states that, 'for every force acting on a body there is an opposite and equal reaction'. For aircraft propulsion, the 'body' is atmospheric air that is caused to accelerate as it passes through the engine. 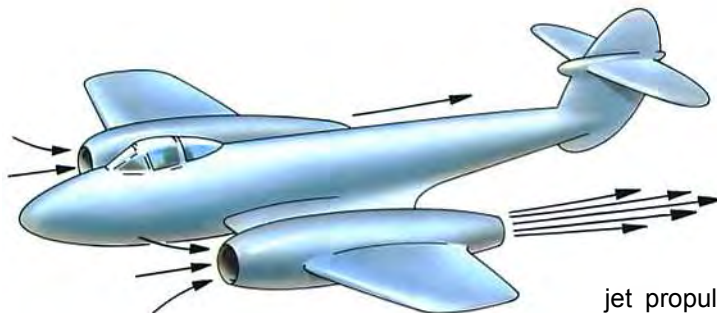The force required to give this acceleration has an equal effect in the opposite direction acting on the apparatus producing the acceleration. A jet engine produces thrust in a

similar way to the engine/propeller combination. Both propel the aircraft by thrusting a large weight of air backwards (fig. 1-3), one in the form of a large air slipstream at comparatively low speed and the other in the form of a jet of gas at very high speed.

7. This same principle of reaction occurs in all forms of movement and has been usefully applied in many ways. The earliest known example of jet reaction is that of Hero's engine (fig. 1-4) produced as a toy in 120 B.C. This toy showed how the momentum of steam issuing from a number of jets could impart an equal and opposite reaction to the jets themselves, thus causing the engine to revolve.

8. The familiar whirling garden sprinkler (fig. 1-5) is a more practical example of this principle, for the mechanism rotates by virtue of the reaction to the water jets. The high pressure jets of modern fire-fighting equipment are an example of 'jet reaction', for often, due to the reaction of the water jet, the hose cannot be held or controlled by one fireman. Perhaps the simplest illustration of this principle is afforded by the carnival balloon which, when the air or gas is released, rushes rapidly away in the direction opposite to the jet.

9. Jet reaction is definitely an internal phenomenon and does not, as is frequently assumed, result from the pressure of the jet on the atmosphere. In fact, the

jet propulsion engine, whether rocket, athodyd, or turbo-jet, is a piece of apparatus designed to accelerate a stream of air or gas and to expel it at high velocity. There are, of course, a number of ways
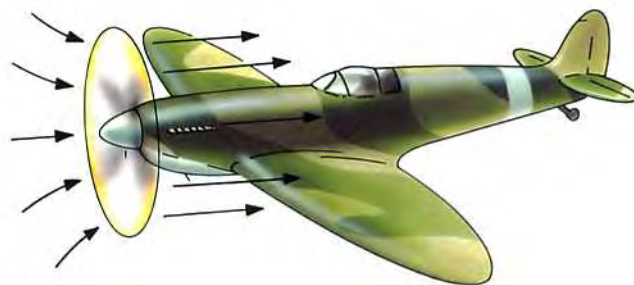
Fig. 1-3   Propeller and jet propulsion.

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velocity. In practice the former is preferred, since by lowering the jet velocity relative to the atmosphere a higher propulsive efficiency is obtained.

METHODS OF JET PROPULSION

10.   The types of jet engine, whether ram jet, pulse jet, rocket, gas turbine, turbo/ram jet or turbo-rocket, differ only in the way in which the 'thrust provider', or engine, supplies and converts the energy into power for flight.

11.   The ram jet engine (fig. 1-6) is an athodyd, or 'aero-thermodynamic-duct to give it its full name. It has no major rotating parts and consists of a duct with a divergent entry and a convergent or

FUEL BURNERS   COMBUSTION CHAMBER

AIR INTAKE   PROPELLING NOZZLE

Fig. 1-6   A ram Jet engine.

convergent-divergent exit. When forward motion is imparted to it from an external source, air is forced into the air intake where it loses velocity or kinetic energy and increases its pressure energy as it passes through the diverging duct. The total energy is then increased by the combustion of fuel, and the expanding gases accelerate to atmosphere through the outlet duct. A ram jet is often the power plant for missiles and .target vehicles; but is unsuitable as an aircraft power plant "because it requires forward motion imparting to it before any thrust is produced.
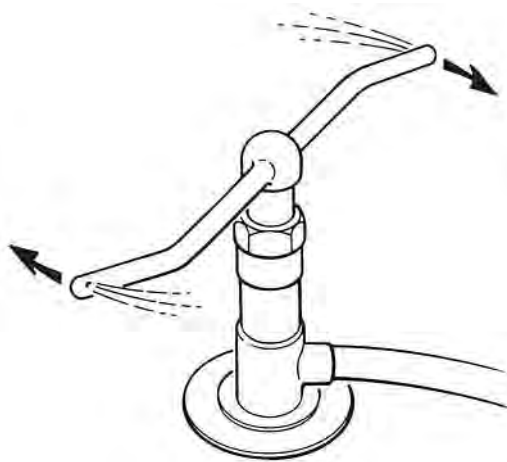
12.   The pulse jet engine (fig. 1-7) uses the principle of intermittent combustion and unlike the ram jet it can be run at a static condition. The engine is formed by an aerodynamic duct similar to the ram jet but, due to the higher pressures involved, it is of more robust construction. The duct inlet has a series of inlet 'valves' that are spring-loaded into the open position. Air drawn through the open valves passes into the combustion chamber and is heated by the burning of fuel injected into the chamber. The resulting expansion causes a rise in pressure, forcing

Fig. 1-4   Hero's engine - probably the earliest form of jet reaction.

of doing this, as described in Part 2, but in all instances the resultant reaction or thrust exerted on the engine is proportional to the mass or weight of air expelled by the engine and to the velocity change imparted to it. In other words, the same thrust can be provided either by giving a large mass of air a little extra velocity or a small mass of air a large extra
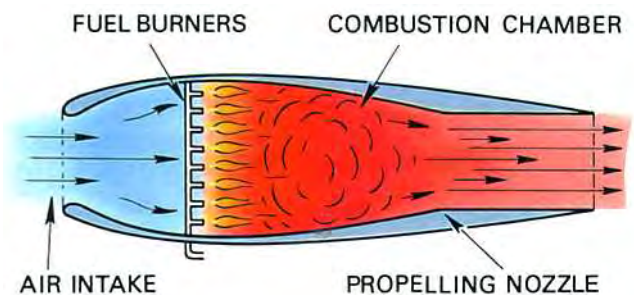

Fig. 1-5   A garden sprinkler rotated by the reaction of the water jets.

Fig. 1-7   A pulse jet engine.

the valves to close, and the expanding gases are then ejected rearwards. A depression created by the exhausting gases allows the valves to open and repeat the cycle. Pulse jets have been designed for helicopter rotor propulsion and some dispense with inlet valves by careful design of the ducting to control the changing pressures of the resonating cycle. The pulse jet is unsuitable as an aircraft power plant because it has a high fuel consumption and is unable to equal the performance of the modern gas turbine engine.

13. Although a rocket engine (fig. 1-8) is a jet engine, it has one major difference in that it does not use atmospheric air as the propulsive fluid stream. Instead, it produces its own propelling fluid by the combustion of liquid or chemically decomposed fuel with oxygen, which it carries, thus enabling it to operate outside the earth's atmosphere. It is, therefore, only suitable for operation over short periods.

14. The application of the gas turbine to jet propulsion has avoided the inherent weakness of the rocket and the athodyd, for by the introduction of a turbine-driven compressor a means of producing thrust at low speeds is provided. The turbo-jet engine operates on the 'working cycle' as described in Part 2. It draws air from the atmosphere and after compressing and heating it, a process that occurs in all heat engines, the energy and momentum given to the air forces It out of the propelling nozzle at a velocity of up to 2,000 feet per second or about 1,400 miles per hour. On its way through the engine, the air gives up some of its energy and momentum to drive the turbine that powers the compressor.

15. The mechanical arrangement of the gas turbine engine is simple, for it consists of only two main rotating parts, a compressor (Part 3) and a turbine (Part 5), and one or a number of combustion chambers (Part 4). The mechanical arrangement of various gas turbine engines is shown in fig. 1 -9. This simplicity, however, does not apply to all aspects of the engine, for as described in subsequent Parts the thermo and aerodynamic problems are somewhat complex. They result from the high operating temperatures of the combustion chamber and turbine, the effects of varying flows across the compressor

Fig. 1-8   A rocket engine.

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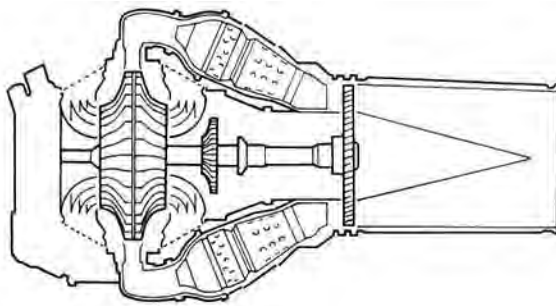
DOUBLE-ENTRY SINGLE-STAGE
CENTRIFUGAL TURBO-JET
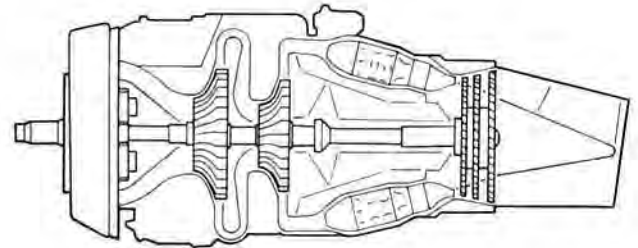
SINGLE-ENTRY TWO-STAGE
CENTRIFUGAL TURBO-PROPELLER
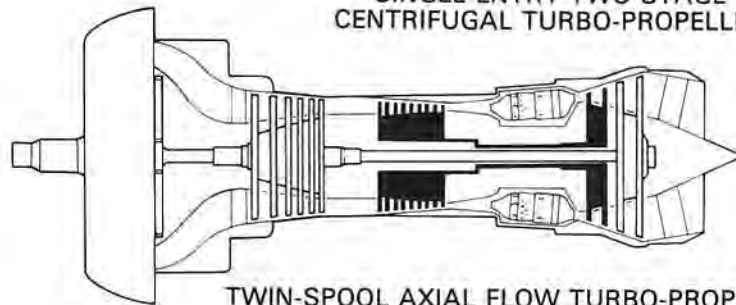
TWIN-SPOOL AXIAL FLOW TURBO-PROPELLER
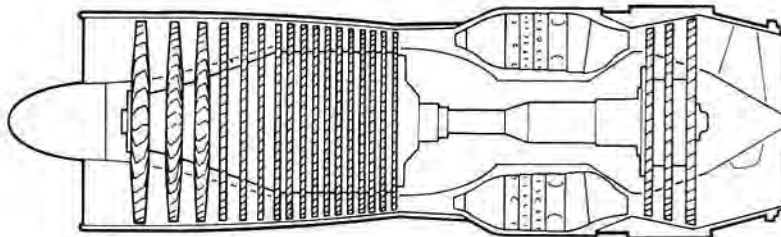
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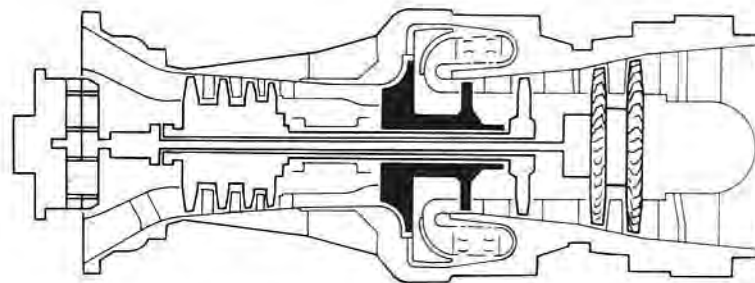
TWIN-SPOOL TURBO-SHAFT (with free-power turbine)

Fig. 1-9-1 Mechanical arrangement of gas turbine engines.

TWIN-SPOOL BY-PASS TURBO-JET
(low by-pass ratio)

TRIPLE-SPOOL FRONT FAN TURBO-JET
(high by-pass ratio)

PROP-FAN-CONCEPT

CONTRA-ROTATING FAN-CONCEPT (high by-pass ratio)
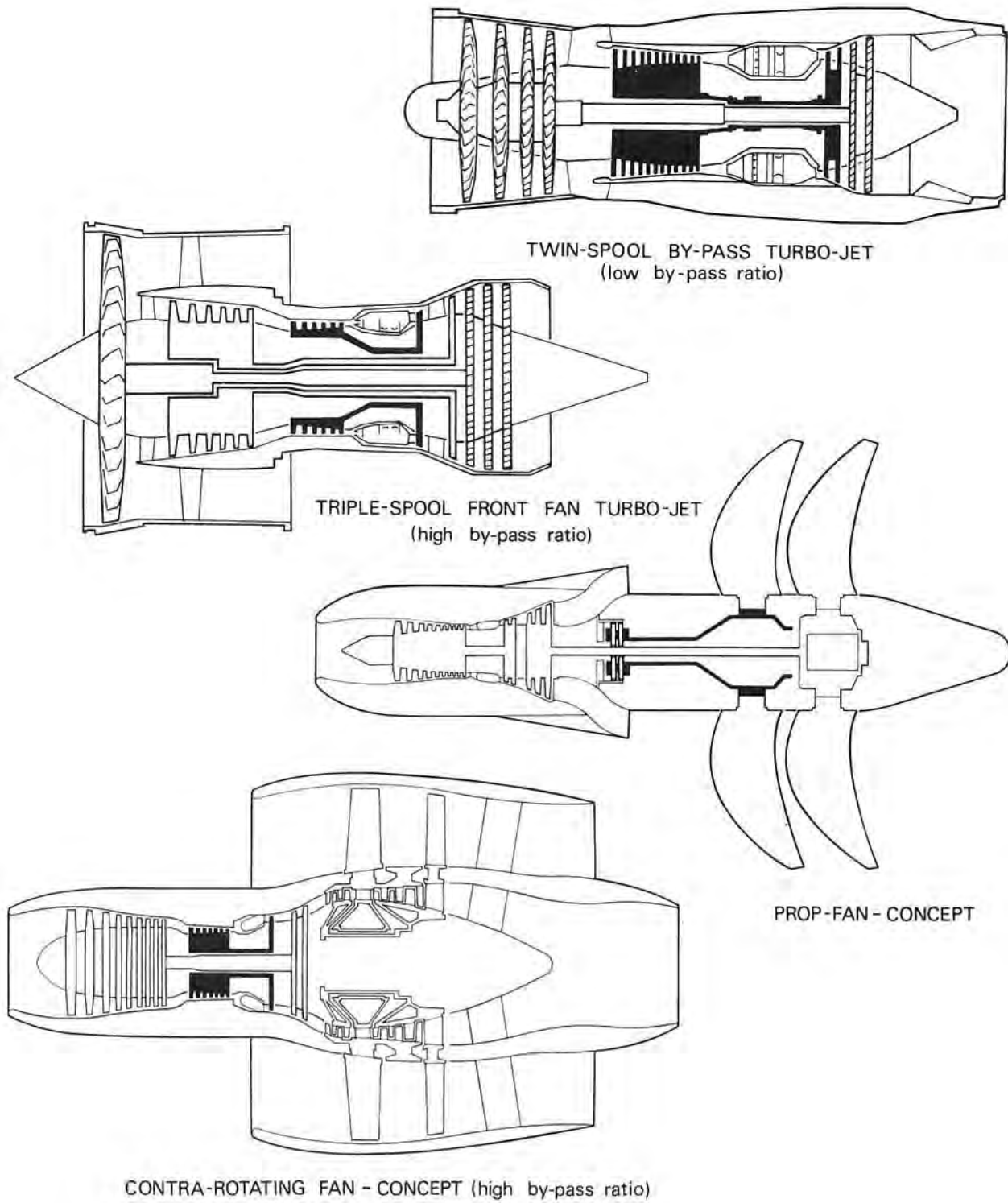
Fig. 1-9-2 Mechanical arrangement of gas turbine engines.

and turbine blades, and the design of the exhaust system through which the gases are ejected to form the propulsive jet.

16. At aircraft speeds below approximately 450 miles per hour, the pure jet engine is less efficient than a propeller-type engine, since its propulsive efficiency depends largely on its forward speed; the pure turbo-jet engine is, therefore, most suitable for high forward speeds. The propeller efficiency does, however, decrease rapidly above 350 miles per hour due to the disturbance of the airflow caused by the high blade-tip speeds of the propeller. These charac-teristics have led to some departure from the use of pure turbo-jet propulsion where aircraft operate at medium speeds by the introduction of a combination of propeller and gas turbine engine.

17. The advantages of the propeller/turbine combination have to some extent been offset by the introduction of the by-pass, ducted fan and propfan engines. These engines deal with larger comparative airflows and lower jet velocities than the pure jet engine, thus giving a propulsive efficiency (Part 21) which is comparable to that of the turbo-prop and exceeds that of the pure jet engine (fig. 1-10).
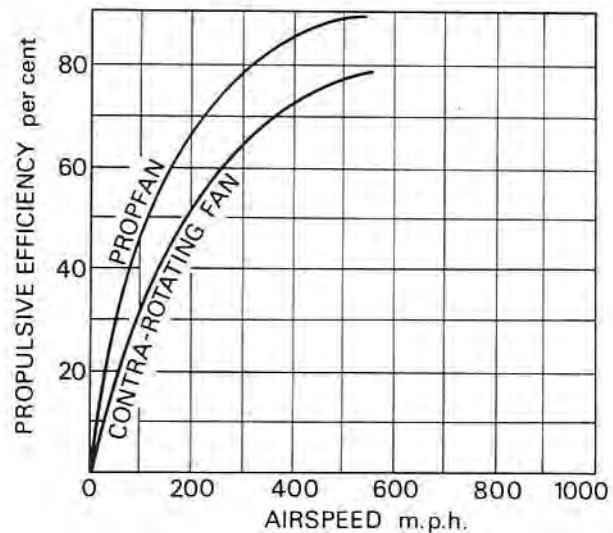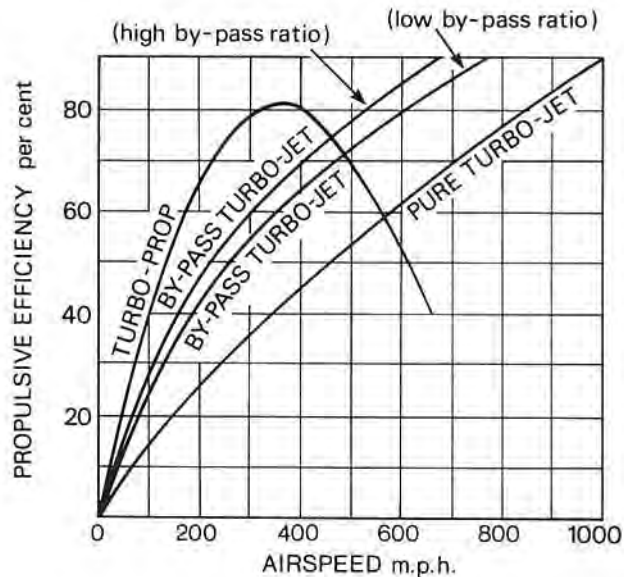
Fig. 1-10   Comparative propulsive efficiencies.

VARIABLE INTAKE
(large area)

INTAKE GUIDE VANES
(open)

VARIABLE NOZZLE
(large area)

LOW MACH NUMBER

VARIABLE INTAKE
(small area)

INTAKE GUIDE VANES
(shut)

VARIABLE NOZZLE
(small area)
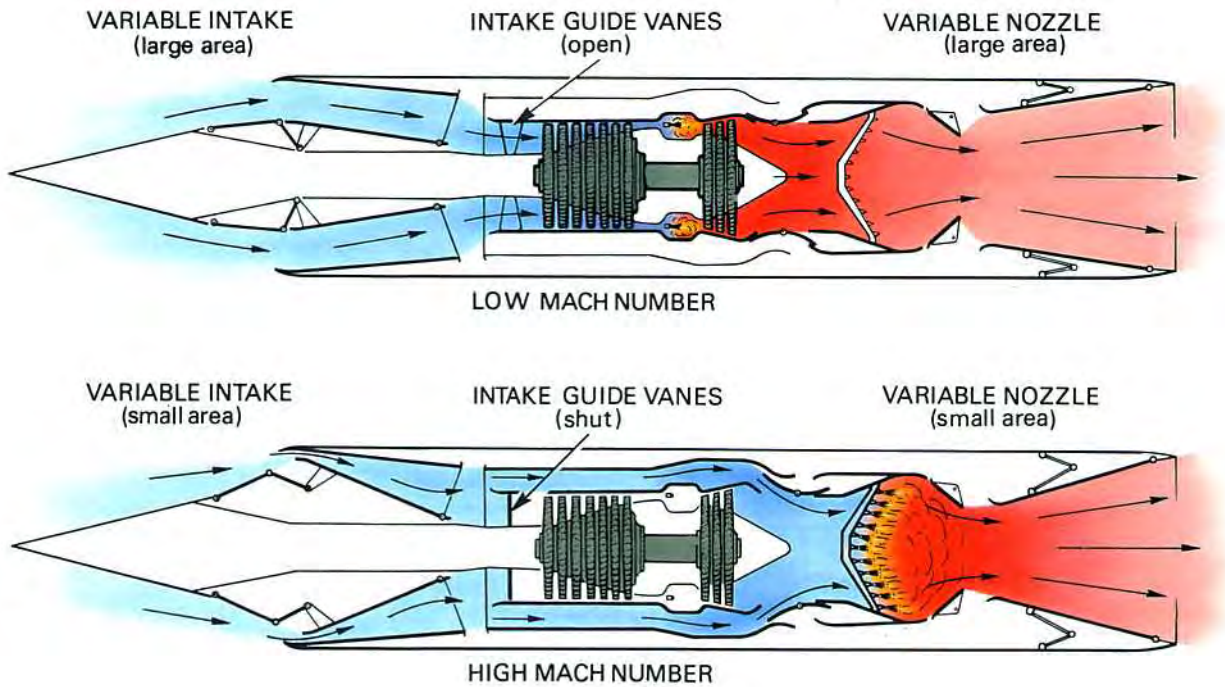
HIGH MACH NUMBER

Fig. 1-11   A turbo/ram jet engine.

18. The turbo/ram jet engine (fig. 1-11) combines the turbo-jet engine (which is used for speeds up to Mach 3) with the ram jet engine, which has good performance at high Mach numbers.

19. The engine is surrounded by a duct that has a variable intake at the front and an afterburning jet pipe with a variable nozzle at the rear. During take-off and acceleration, the engine functions as a con-

ventional turbo-jet with the afterburner lit; at other flight conditions up to Mach 3, the afterburner is inoperative. As the aircraft accelerates through Mach 3, the turbo-jet is shut down and the intake air is diverted from the compressor, by guide vanes, and ducted straight into the afterburning jet pipe, which becomes a ram jet combustion chamber. This engine is suitable for an aircraft requiring high speed and

VARIABLE INTAKE

AFTERBURNING
FUEL BURNERS

COMBUSTION CHAMBER

OXYGEN AND FUEL SUPPLY

VARIABLE NOZZLE
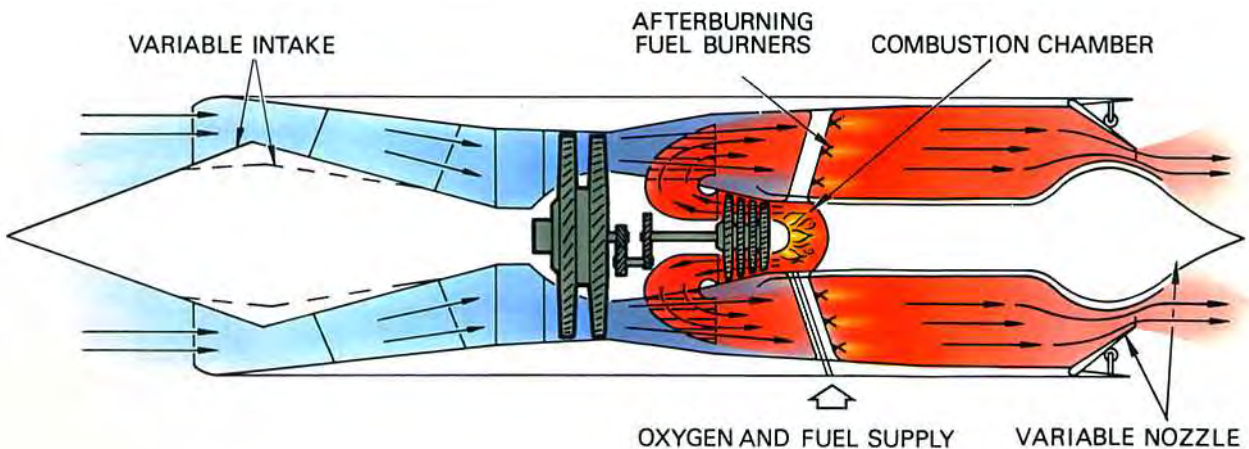
Fig. 1-12   A turbo-rocket engine.

sustained high Mach number cruise conditions where the engine operates in the ram jet mode.

20. The turbo-rocket engine (fig. 1-12) could be considered as an alternative engine to the turbo/ram jet; however, it has one major difference in that it carries its own oxygen to provide combustion,

21. The engine has a low pressure compressor driven by a multi-stage turbine; the power to drive the turbine is derived from combustion of kerosine and liquid oxygen in a rocket-type combustion chamber. Since the gas temperature will be in the order of 3,500 deg. C, additional fuel is sprayed into the combustion chamber for cooling purposes before the gas enters the turbine. This fuel-rich mixture (gas) is then diluted with air from the compressor and the surplus fuel burnt in a conventional afterburning system.

22. Although the engine is smaller and lighter than the turbo/ram jet, it has a higher fuel consumption. This tends to make it more suitable for an interceptor or space-launcher type of aircraft that requires high speed, high altitude performance and normally has a flight plan that is entirely accelerative and of short duration.

Rolls-Royce/Snecma Olympus
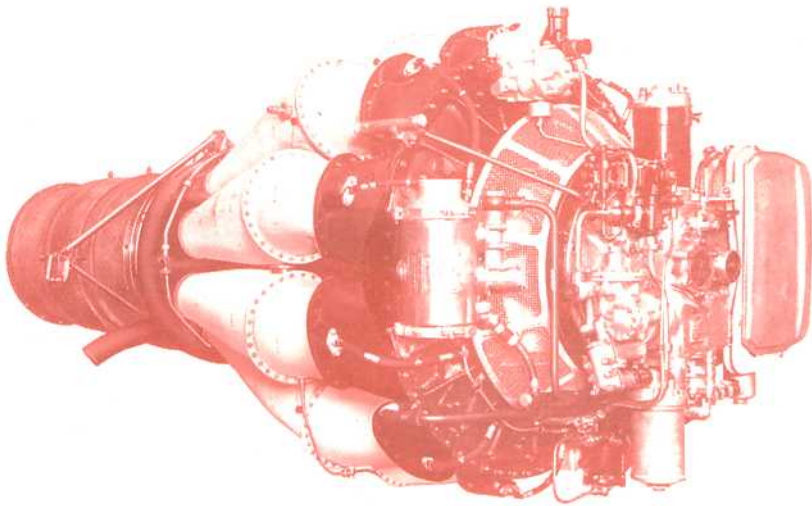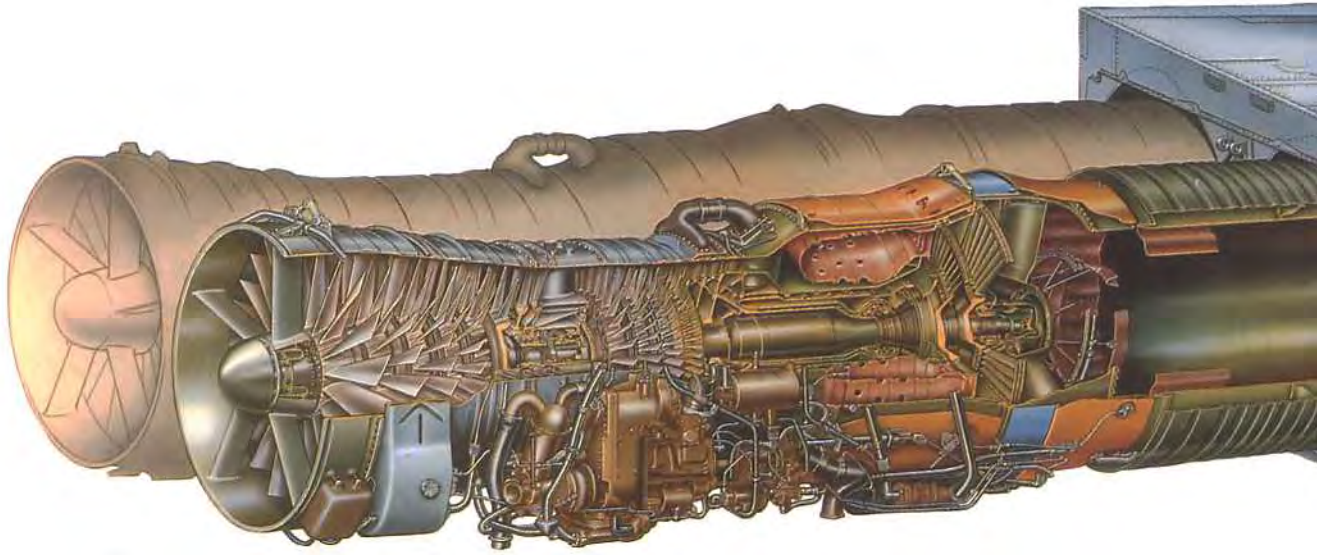
Rolls-Royce RB37 Derwent 1

A straight-through version of the reverse-flow Power Jets W2B, known as the W2B/26, was developed by the Rover Company from 1941 to 1943. Taken over by Rolls-Royce in April 1943 and renamed the Derwent, it passed a 100hr. test at 2000 lb thrust in November 1943 and was flown at that rating in April 1944. The engine powered the Gloster Meteor III which entered service in 1945.

2: Working cycle and airflow

Contents

INTRODUCTION

1. The gas turbine engine is essentially a heat engine using air as a working fluid to provide thrust. To achieve this, the air passing through the engine has to be accelerated; this means that the velocity or kinetic energy of the air is increased. To obtain this increase, the pressure energy is first of all increased, followed by the addition of heat energy, before final conversion back to kinetic Energy in the form of a high velocity jet efflux.

WORKING CYCLE

2. The working cycle of the gas turbine engine is similar to that of the four-stroke piston engine. However, in the gas turbine engine, combustion occurs at a constant pressure, whereas in the piston engine it occurs at a constant volume. Both engine cycles (fig. 2-1) show that in each instance there is induction, compression, combustion and exhaust. These processes are intermittent in the case of the piston engine whilst they occur continuously in the gas turbine. In the piston engine only one stroke is utilized in the production of power, the others being involved in the charging, compressing and exhausting of the working fluid. In contrast, the turbine engine eliminates the three 'idle' strokes, thus enabling more fuel to be burnt in a shorter time; hence it produces a greater power output for a given size of engine.

3. Due to the continuous action of the turbine engine and the fact that the combustion chamber is not an enclosed space, the pressure of the air does not rise, like that of the piston engine, during combustion but its volume does increase. This process is known as heating at constant pressure. Under these conditions there are no peak or fluctuating pressures to be withstood, as is the case with the piston engine with its peak pressures in excess of 1,000 lb. per sq. in. It is these peak pressures which make it necessary for the piston engine to employ cylinders of heavy construction and
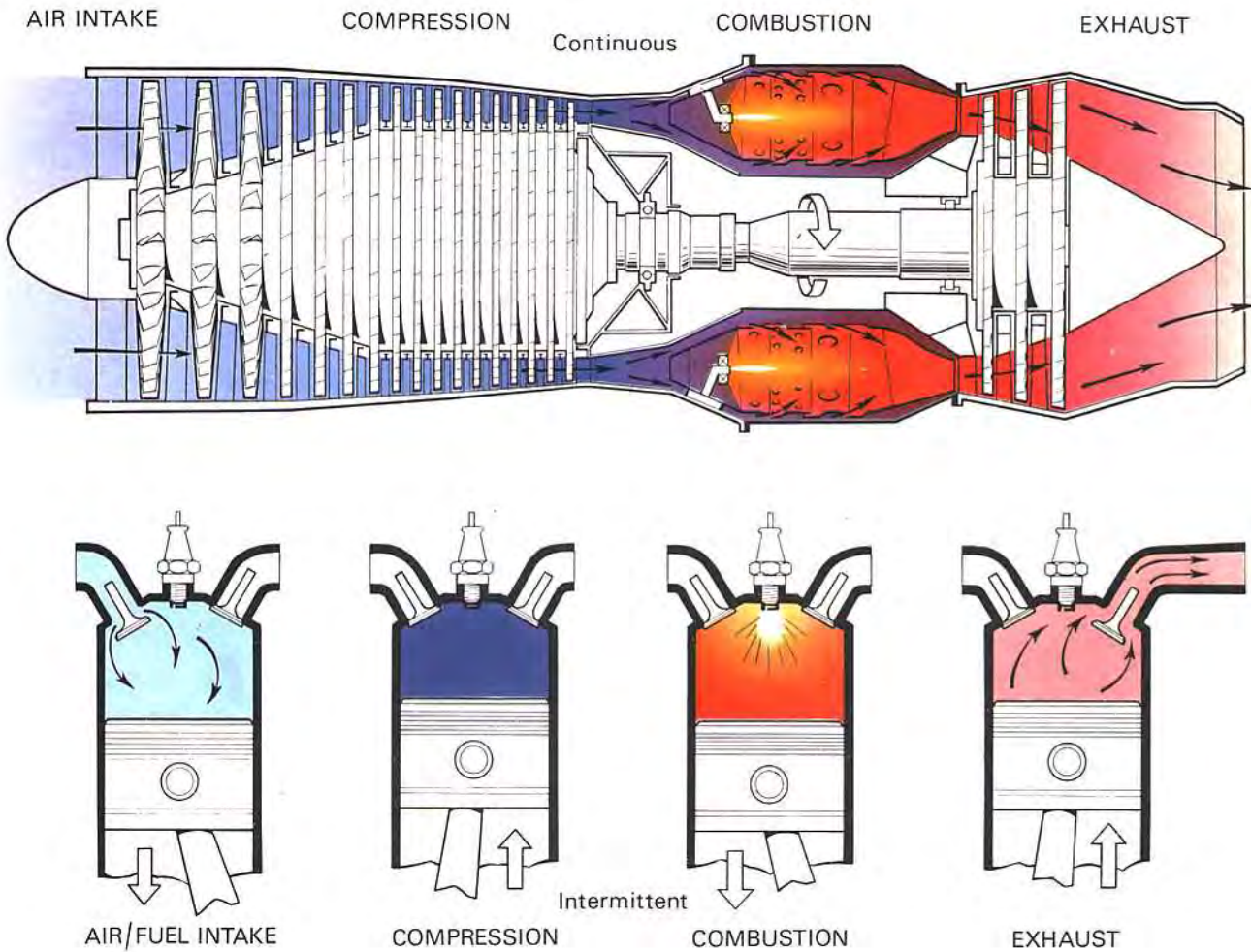
Fig. 2-1   A comparison between the working cycle of a turbo-jet engine and a piston engine.

to use high octane fuels, in contrast to the low octane fuels and the light fabricated combustion chambers used on the turbine engine.

4. The working cycle upon which the gas turbine engine functions is, in its simplest form, represented by the cycle shown on the pressure volume diagram in fig. 2-2. Point A represents air at atmospheric pressure that is compressed along the line AB. From B to C heat is added to the air by introducing and burning fuel at constant pressure, thereby considerably increasing the volume of air. Pressure losses in the combustion chambers (Part 4) are indicated by the drop between B and C. From C to D the gases resulting from combustion expand through the turbine and jet pipe back to atmosphere. During this part of the cycle, some of the energy in the expanding gases is turned into mechanical power by
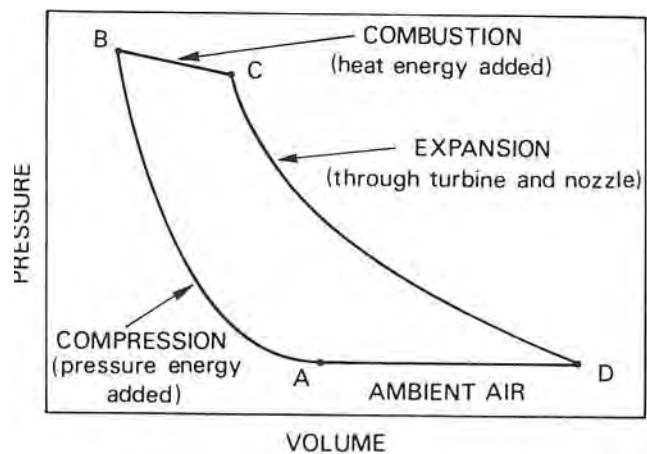
Fig. 2-2   The working cycle on a pressure-volume diagram.

the turbine; the remainder, on its discharge to atmosphere, provides a propulsive jet.

5. Because the turbo-jet engine is a heat engine, the higher the temperature of combustion the greater is the expansion of the gases. The combustion temperature, however, must not exceed a value that gives a turbine gas entry temperature suitable for the design and materials of the turbine assembly.

6. The use of air-cooled blades in the turbine assembly permits a higher gas temperature and a consequently higher thermal efficiency.

THE RELATIONS BETWEEN PRESSURE, VOLUME AND TEMPERATURE

7. During the working cycle of the turbine engine, the airflow or 'working fluid' receives and gives up heat, so producing changes in its pressure, volume and temperature. These changes as they occur are closely related, for they follow a common principle that is embodied in a combination of the laws of Boyle and Charles. Briefly, this means that the product of the pressure and the volume of the air at the various stages in the working cycle is proportional to the absolute temperature of the air at those

Velocity – decreasing
Pressure – increasing
Temperature – increasing

PRINCIPLE

EXAMPLE-TYPICAL AXIAL FLOW COMPRESSOR OUTLET CASING

Velocity – increasing
Pressure – decreasing
Temperature – decreasing

PRINCIPLE

DIRECTION OF ROTATION

EXAMPLE- FLOW THROUGH TURBINE NOZZLE GUIDE VANES

Fig. 2-3   An airflow through divergent and convergent ducts.

stages. This relationship applies for whatever means are used to change the state of the air. For example, whether energy is added by combustion or by compression, or is extracted by the turbine, the heat change is directly proportional to the work added or taken from the gas.

8. There are three main conditions in the engine working cycle during which these changes occur. During compression, when work is done to increase the pressure and decrease the volume of the air, there is a corresponding rise in the temperature. During combustion, when fuel is added to the air and burnt to increase the temperature, there is a corresponding increase in volume whilst the pressure remains almost constant. During expansion, when work is taken from the gas stream by the turbine assembly, there is a decrease in temperature and pressure with a corresponding increase in volume.

9. Changes in the temperature and pressure of the air can be traced through an engine by using the airflow diagram in fig. 2-5. With the airflow being continuous, volume changes are shown up as changes in velocity.
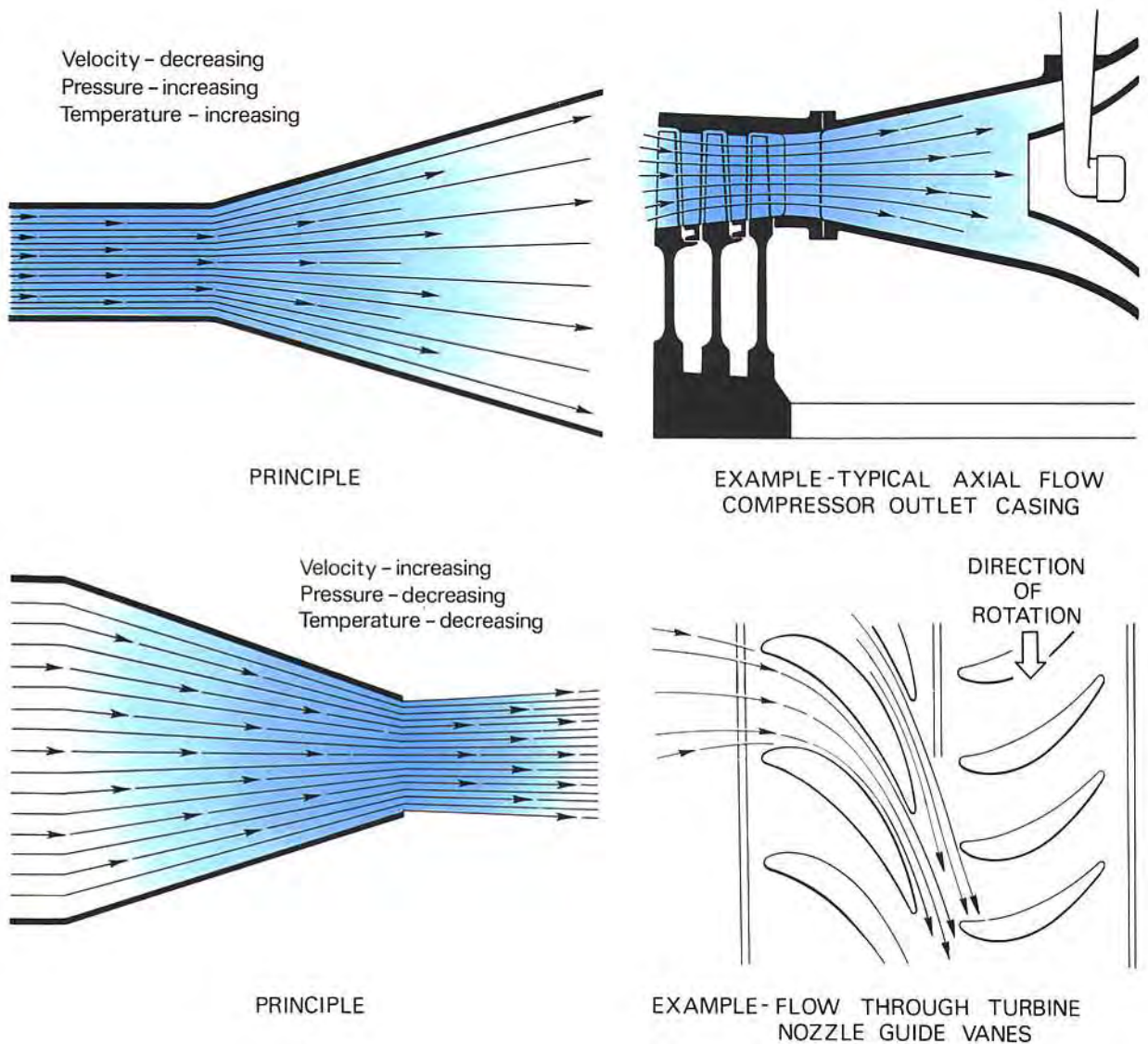
10. The efficiency with which these changes are made will determine to what extent the desired relations between the pressure, volume and temperature are attained. For the more efficient the compressor, the higher the pressure generated for a given work input; that is, for a given temperature rise of the air. Conversely, the more efficiently the turbine uses the expanding gas, the greater the output of work for a given pressure drop in the gas.

11. When the air is compressed or expanded at 100 per cent efficiency, the process is said to be adiabatic. Since such a change means there is no energy losses in the process, either by friction, conduction or turbulence, it is obviously impossible to achieve in practice; 90 per cent is a good adiabatic efficiency for the compressor and turbine.

CHANGES IN VELOCITY AND PRESSURE

12. During the passage of the air through the engine, aerodynamic and energy requirements demand changes in its velocity and pressure. For instance: during compression, a rise in the pressure of the air is required and not an increase in its velocity. After the air has been heated and its internal energy increased by combustion, an increase in the velocity of the gases is necessary to force the turbine to rotate. At the propelling nozzle a high exit velocity is required, for it is the change in the momentum of

the air that provides the thrust on the aircraft. Local decelerations of airflow are also required, as for instance, in the combustion chambers to provide a low velocity zone for the flame to burn.

13. These various changes are effected by means of the size and shape of the ducts through which the air passes on its way through the engine. Where a conversion from velocity (kinetic) energy to pressure is required, the passages are divergent in shape. Conversely, where it is required to convert the energy stored in the combustion gases to velocity energy, a convergent passage or nozzle (fig. 2-3) is used. These shapes apply to the gas turbine engine where the airflow velocity is subsonic or sonic, i.e. at the local speed of sound. Where supersonic speeds are encountered, such as in the propelling nozzle of the rocket, athodyd and some jet engines (Part 6), a convergent-divergent nozzle or venturi (fig. 2-4) is used to obtain the maximum conversion of the energy in the combustion gases to kinetic energy.

14. The design of the passages and nozzles is of great importance, for upon their good design will depend the efficiency with which the energy changes are effected. Any interference with the smooth airflow creates a loss in efficiency and could result in component failure due to vibration caused by eddies or turbulence of the airflow.
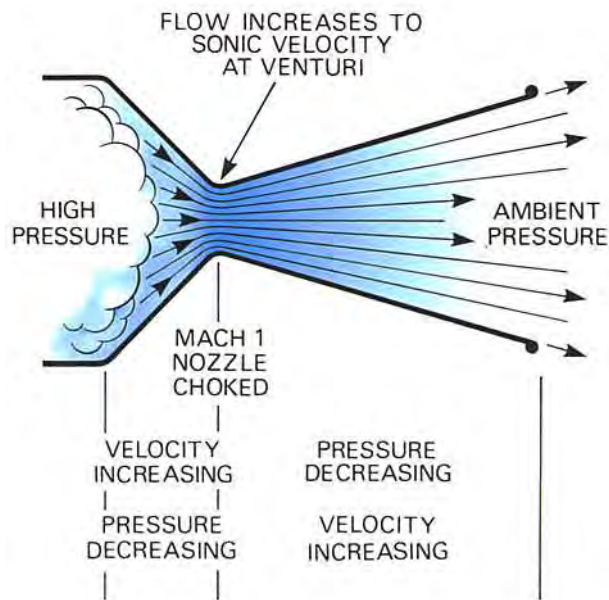
Fig. 2-4   Supersonic airflow through a convergent-divergent nozzle or venturi.

AIR INTAKE

PROPELLING NOZZLE

COMPRESSION   COMBUSTION   EXPANSION   EXHAUST

Deg C.	Ft/sec.	Lb /sq.in.
3000	3000	150
2500	2500	125
2000	2000	100
1500	1500	75
1000	1000	50
500	500	25
0	0	0

TOTAL PRESSURE

Flame temperature

AXIAL VELOCITY   TEMPERATURE

TYPICAL SINGLE-SPOOL AXIAL FLOW TURBO-JET ENGINE

LOW PRESSURE COMPRESSOR   HIGH PRESSURE COMPRESSOR

TWO-STAGE CENTRIFUGAL FLOW TURBO-PROPELLER ENGINE

LOW PRESSURE COMPRESSOR   HIGH PRESSURE COMPRESSOR

TWIN-SPOOL AXIAL FLOW TURBO-PROPELLER ENGINE

LOW PRESSURE COMPRESSOR   REVERSE FLOW COMBUSTION SYSTEM   FREE POWER TURBINE

HIGH PRESSURE COMPRESSOR

TWIN-SPOOL TURBO-SHAFT ENGINE (with free power turbine)

Fig. 2-5-1 Airflow systems.

LOW PRESSURE COMPRESSOR HIGH PRESSURE COMPRESSOR BY-PASS FLOW By-pass air mixing with the exhaust gas stream

TWIN-SPOOL AXIAL FLOW BY-PASS TURBO JET ENGINE(low by-pass ratio)

LOW PRESSURE COMPRESSOR INTERMEDIATE PRESSURE COMPRESSOR

HIGH PRESSURE COMPRESSOR

TRIPLE-SPOOL AXIAL FLOW FRONT FAN TURBO JET ENGINE(high by-pass ratio)

CONTRA-ROTATING PROP-FAN

COMPRESSOR

AXIAL FLOW CONTRA-ROTATING PROP-FAN(with free power turbine)

CONTRA-ROTATING FAN

LOW PRESSURE COMPRESSOR

HIGH PRESSURE COMPRESSOR

TWIN-SPOOL AXIAL FLOW CONTRA-ROTATING REAR FAN(with free power turbine)

Fig, 2-5-2 Airflow systems.

AIRFLOW

15.   The path of the air through a gas turbine engine varies according to the design of the engine. A straight-through flow system (fig. 2-5) is the basic design, as it provides for an engine with a relatively small frontal area and is also suitable for use of the by-pass principle. In contrast, the reverse flow system gives an engine with greater frontal area, but with a reduced overall length. The operation, however, of all engines is similar. The variations due to the different designs are described in the subsequent paragraphs.

16.   The major difference of a turbo-propeller engine is the conversion of gas energy into mechanical power to drive the propeller. Only a small amount of 'jet thrust' is available from the exhaust system. The majority of the energy in the gas stream is absorbed by additional turbine stages, which drive the propeller through internal shafts (Part 5).

17.   As can be seen in fig. 2-5, the by-pass principle involves a division of the airflow. Conventionally, all the air taken in is given an initial low compression and a percentage is then ducted to by-pass, the remainder being delivered to the combustion system in the usual manner. As described in Part 21, this

principle is conducive to improved propulsive efficiency and specific fuel consumption.

18.   An important design feature of the by-pass engine is the by-pass ratio; that is, the ratio of cool air by-passed through the duct to the flow of air passed through the high pressure system. With low by-pass ratios, i.e. in the order of 1:1, the two streams are usually mixed before being exhausted from the engine. The fan engine may be regarded as an extension of the by-pass principle, and the requirement for high by-pass ratios of up to 5:1 is largely met by using the front fan in a twin or triple-spool configuration (on which the fan is, in fact, the low pressure compressor) both with and without mixing of the airflows. Very high by-pass ratios, in the order of 15:1, are achieved using propfans. These are a variation on the turbo-propeller theme but with advanced technology propellers capable of operating with high efficiency at high aircraft speeds.

19.   On some front fan engines, the by-pass airstream is ducted overboard either directly behind the fan through short ducts or at the rear of the engine through longer ducts; hence the term 'ducted fan'. Another, though seldom used, variation is that of the aft fan.

Rolls-Royce RB211-22B

De Havilland H1 Goblin
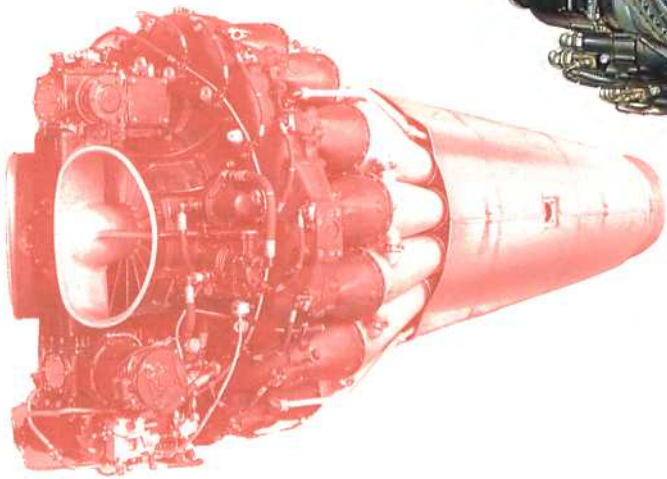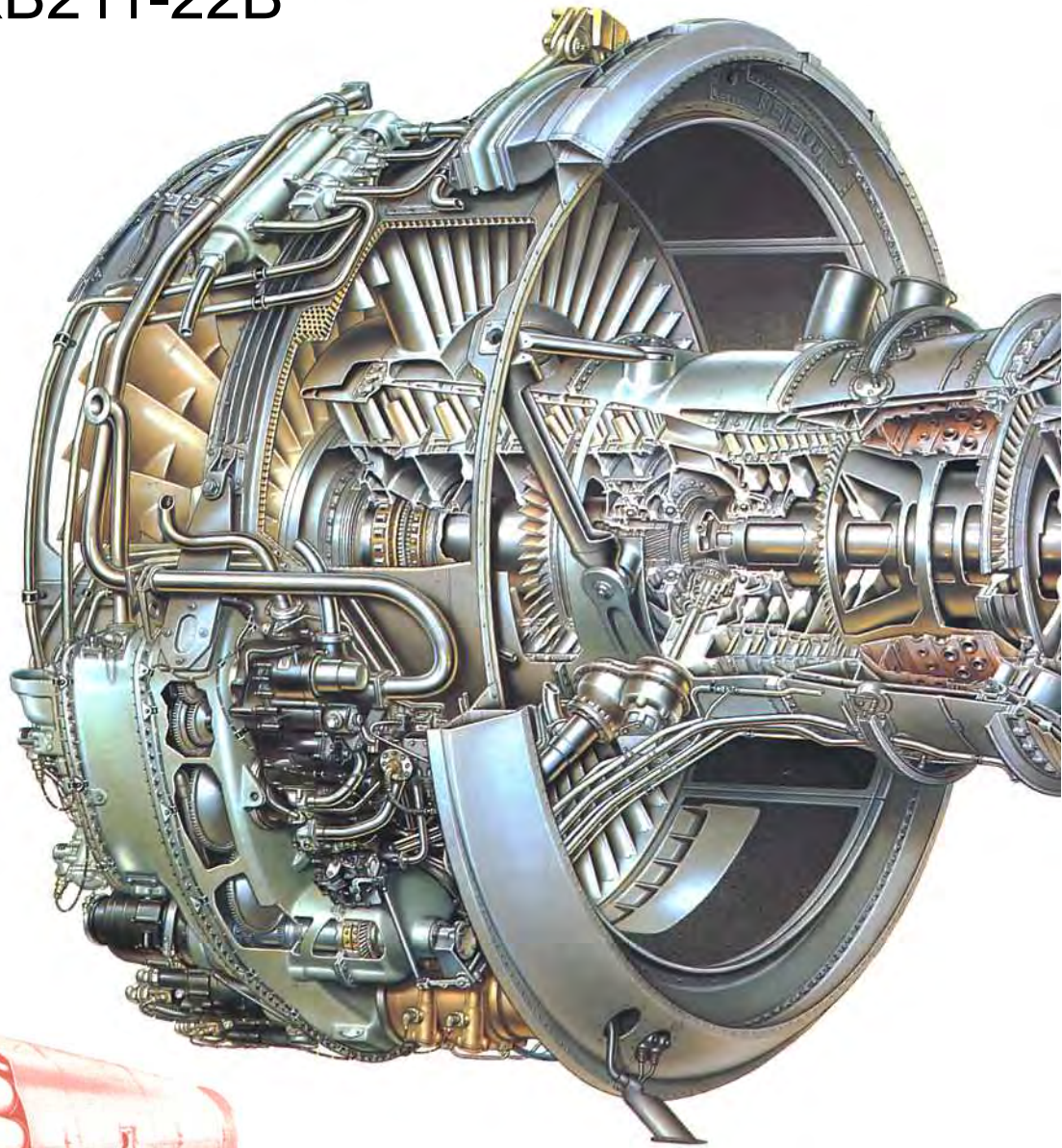
Development of the de Havilland Goblin began in 1941 with the Halford H1 with a design thrust of 3000 lb. The engine passed a 25 hr special category test in September 1942 and was cleared for flight at 2000 lb thrust. This took place in a Gloster Meteor on 5 March 1943 and was also the first flight of that aircraft type. In September 1943 the first flight of a de Havilland DH100 Vampire was made with a Goblin of 2300 lb thrust.

3: Compressors

Contents

INTRODUCTION

1. In the gas turbine engine, compression of the air before expansion through the turbine is effected by one of two basic types of compressor, one giving centrifugal flow and the other axial flow. Both types are driven by the engine turbine and are usually coupled direct to the turbine shaft.

2. The centrifugal flow compressor (fig. 3-1) is a single or two stage unit employing an impeller to accelerate the air and a diffuser to produce the required pressure rise. The axial flow compressor (fig. 3-7 and fig. 3-8) is a multi-stage unit employing alternate .rows of rotating (rotor) blades and stationary (stator) vanes, to accelerate and diffuse the air until the required pressure rise is obtained. In some cases, particularly on small engines, an axial compressor is used to boost the inlet pressure to the centrifugal.

3. With regard to the advantages and disadvantages of the two types, the centrifugal compressor is usually more robust than the axial compressor and is also easier to develop and manufacture. The axial compressor however consumes far more air than a

FRONT AIR INTAKE CASING

COMPRESSOR AIR OUTLET CASING

ROTATING GUIDE VANE

IMPELLER

IMPELLER SHAFT COUPLED DIRECT TO TURBINE

INTAKE CHUTES

SWIRL VANES

REAR AIR INTAKE CASING

DIFFUSER
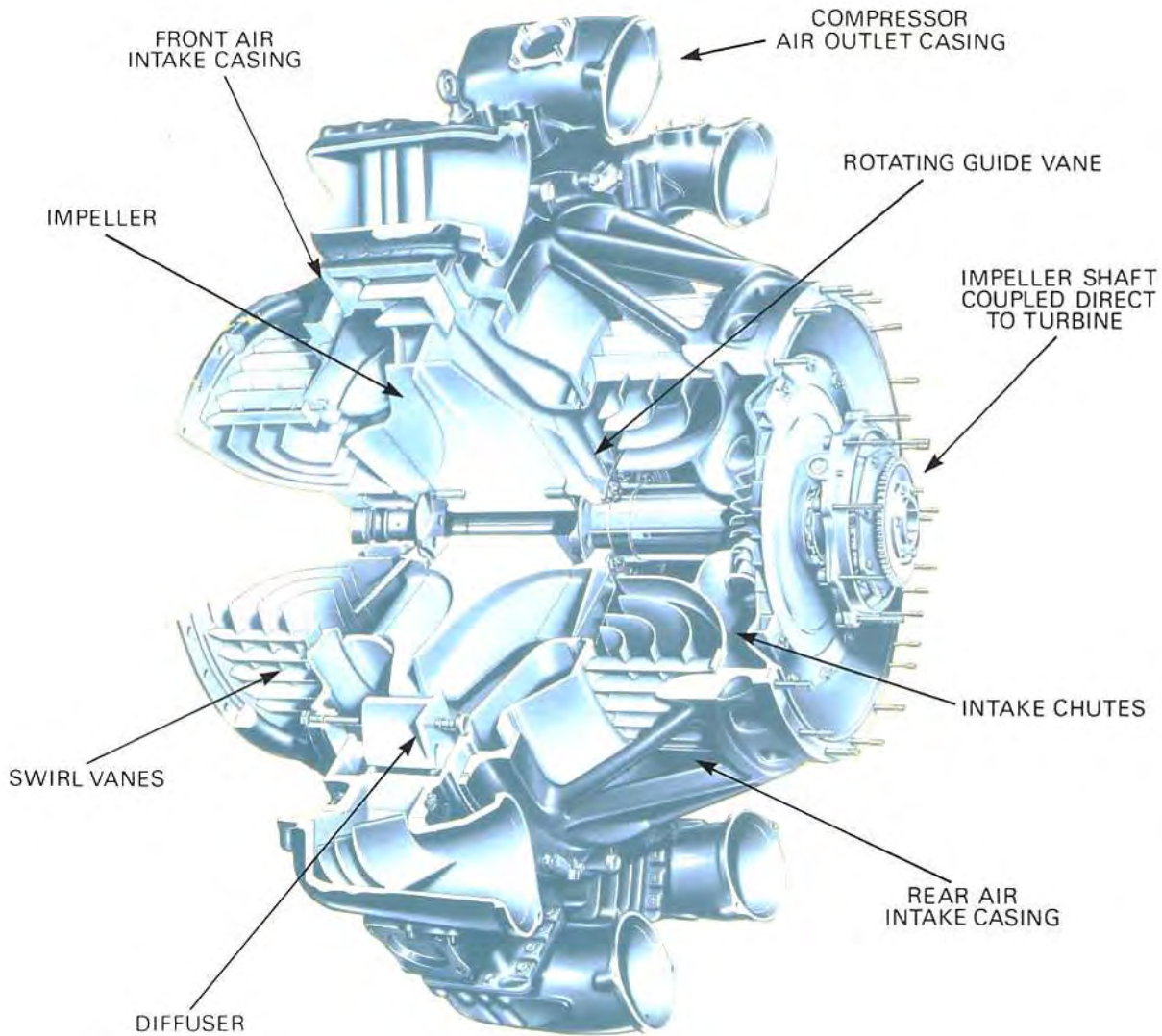
Fig. 3-1   A typical centrifugal flow compressor.

centrifugal compressor of the same frontal area and can be designed to attain much higher pressure ratios. Since the air flow is an important factor in determining the amount of thrust, this means the axial compressor engine will also give more thrust for the same frontal area. This, plus the ability to increase the pressure ratio by addition of extra stages, has led to the adoption of axial compressors in most engine designs. However, the centrifugal compressor is still favoured for smaller engines where its simplicity and ruggedness outweigh any other disadvantages.

4. The trend to high pressure ratios which has favoured the adoption of axial compressors is because of the improved efficiency that results,

which in turn leads to improved specific fuel consumption for a given thrust, ref. fig. 3-2.
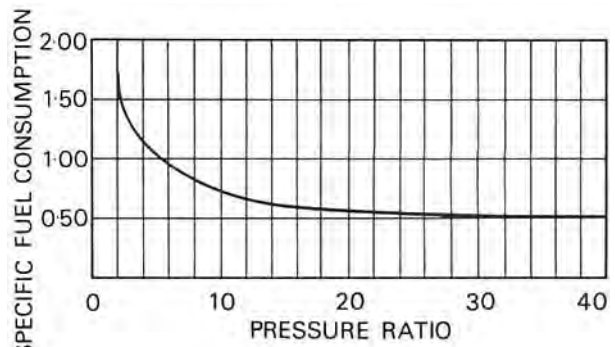
Fig. 3-2   Specific fuel consumption and pressure ratio.

THE CENTRIFUGAL FLOW COMPRESSOR

5. Centrifugal flow compressors have a single or double-sided impeller and occasionally a two-stage, single sided impeller is used, as on the Rolls-Royce Dart. The impeller is supported in a casing that also contains a ring of diffuser vanes. If a double-entry impeller is used, the airflow to the _rear side is reversed in direction and a plenum chamber is required.

Principles of operation
6. The impeller is rotated at high speed by the turbine and air is continuously induced into the centre of the impeller. Centrifugal action causes it to flow radially outwards along the vanes to the impeller tip, thus accelerating the air and also causing a rise in pressure to occur. The engine intake duct may contain vanes that provide an initial swirl to the air entering the compressor.

7. The air, on leaving the impeller, passes into the diffuser section where the passages form divergent nozzles that convert most of the kinetic energy into pressure energy, as illustrated in fig. 3-3. In practice, it is usual to design the compressor so that about half of the pressure rise occurs in the impeller and half in the diffuser.

8. To maximize the airflow and pressure rise through the compressor requires the impeller to be rotated at high speed, therefore impellers are designed to operate at tip speeds of up to 1,600 ft.

per sec. By operating at such high tip speeds the air velocity from the impeller is increased so that greater energy is available for conversion to pressure.

9. To maintain the efficiency of the compressor, it is necessary to prevent excessive air leakage between the impeller and the casing; this is achieved by keeping their clearances as small as possible (fig. 3-4).
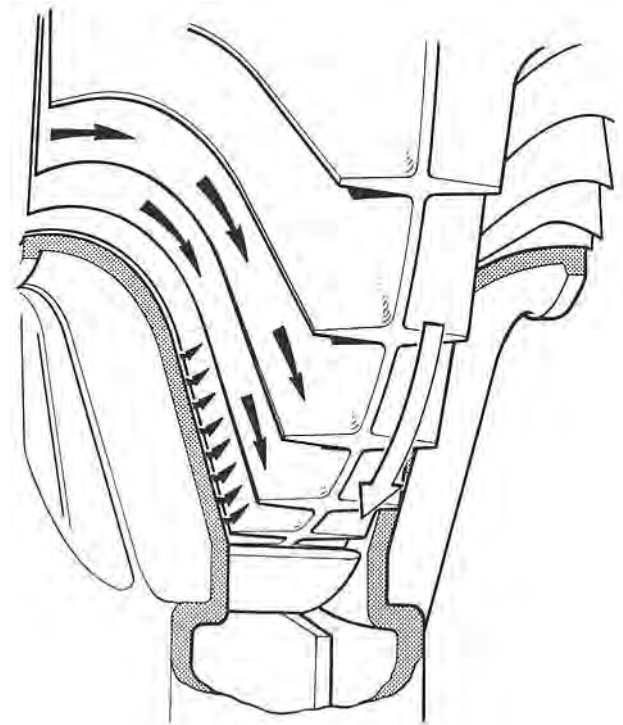
Fig. 3-4   Impeller working clearance and air leakage.

Construction
10. The construction of the compressor centres around the impeller, diffuser and air intake system. The impeller shaft rotates in ball and roller bearings and is either common to the turbine shaft or split in the centre and connected by a coupling, which is usually designed for ease of detachment.

Impellers
11. The impeller consists of a .forged, disc with integral, radially disposed vanes on one or both sides (fig. 3-5) forming convergent passages in conjunction with the compressor casing. The vanes may be swept back, but for ease of manufacture straight

Fig. 3-3   Pressure and velocity changes through a centrifugal compressor.

vanes are in line with the direction of the resultant airflow from the impeller (fig. 3-6). The clearance between the impeller and the diffuser is an important factor, as too small a clearance will set up aerodynamic buffeting impulses that could be transferred to the impeller and create an unsteady airflow and vibration.
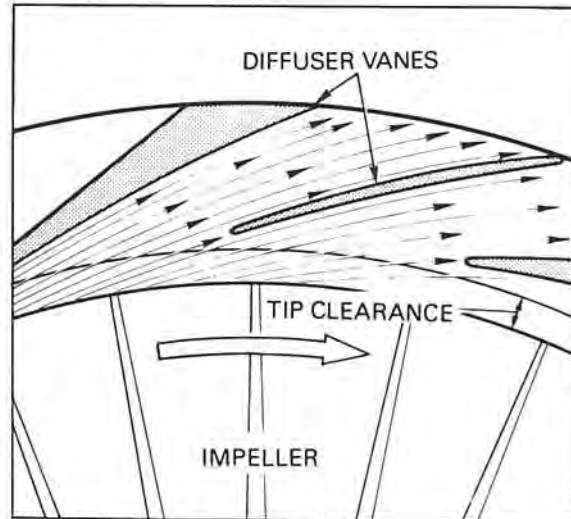
Fig. 3-6   Airflow at entry to diffuser.

Fig. 3-5   Typical impellers for centrifugal compressors.

radial vanes are usually employed. To ease the air from axial flow in the entry duct on to the rotating impeller, the vanes in the centre of the impeller are curved in the direction of rotation. The curved sections may be integral with the radial vanes or formed separately for easier and more accurate manufacture.

Diffusers
12.   The diffuser assembly may be an integral part of the compressor casing or a separately attached assembly. In each instance it consists of a number of vanes formed tangential to the impeller. The vane passages are divergent to convert the kinetic energy into pressure energy and the inner edges of the

THE AXIAL FLOW COMPRESSOR

13.   An axial flow compressor (fig. 3-7 and fig. 3-8) consists of one or more rotor assemblies that carry blades of airfoil section. These assemblies are mounted between bearings in the casings which incorporate the stator vanes. The compressor is a multi-stage unit as the amount of pressure increase by each stage is small; a stage consists of a row of rotating blades followed by a row of stator vanes. Where several stages of compression operate in series on one shaft it becomes necessary to vary the stator vane angle to enable the compressor to operate effectively at speeds below the design condition. As the pressure ratio is increased the incorporation of variable stator vanes ensures that the airflow is directed onto the succeeding stage of rotor blades at an acceptable angle, ref. para. 30, Airflow Control.

14.   From the front to the rear of the compressor, i.e. from the low to the high pressure end, there is a gradual reduction of the air annulus area between

INTAKE CASING   STATOR VANE   ROTOR BLADE   MAIN SHAFT DRIVE FROM TURBINE

ACCESSORY DRIVE

COMBUSTION SYSTEM MOUNTING FLANGE

SINGLE-SPOOL COMPRESSOR

L.P. COMPRESSOR   H.P. COMPRESSOR   H.P. SHAFT DRIVE FROM TURBINE

L.P. SHAFT DRIVE FROM TURBINE

ACCESSORY DRIVE   COMBUSTION SYSTEM MOUNTING FLANGE

TWIN-SPOOL COMPRESSOR

Fig. 3-7   Typical axial flow compressors.

I.P. COMPRESSOR

H.P. COMPRESSOR

COMBUSTION CASE
MOUNTING FLANGE

I.P. SHAFT DRIVE
FROM TURBINE

L.P. SHAFT DRIVE
FROM TURBINE

H.P. DRIVE FROM TURBINE

L.P. COMPRESSOR

Fig. 3-8   Typical triple spool compressor.

the rotor shaft and the stator casing. This is necessary to maintain a near constant air axial velocity as the density increases through the length of the compressor. The convergence of the air annulus is achieved by the tapering of the casing or rotor. A combination of both is also possible, with the arrangement being influenced by manufacturing problems and other mechanical design factors.

15.   A single-spool compressor (fig. 3-7) consists of one rotor assembly and stators with as many stages as necessary to achieve the desired pressure ratio and all the airflow from the intake passes through the compressor.

16.   The multi-spool compressor consists of two or more rotor assemblies, each driven by their own

turbine at an optimum speed to achieve higher pressure ratios and to give greater operating flexibility.

17.   Although a twin-spool compressor (fig. 3-7) can be used for a pure jet engine, it is most suitable for the by-pass type of engine where the front or low pressure compressor is designed to handle a larger airflow than the high pressure compressor. Only a percentage of the air from the low pressure compressor passes into the high pressure compressor; the remainder of the air, the by-pass flow, is ducted around the high pressure compressor. Both flows mix in the exhaust system before passing to the propelling nozzle (Part 6). This arrangement matches the velocity of the jet nearer to the optimum requirements of the aircraft and results in higher

propulsive efficiency, hence lower fuel consumption. For this reason the pure jet engine where all the airflow passes through the full compression cycle is now obsolete for all but the highest speed aircraft.

18. With the high by-pass ratio turbo-fan this trend is taken a stage further. The intake air undergoes only one stage of compression in the fan before being split between the core or gas generator system and the by-pass duct in the ratio of approximately one to five (fig. 3-8). This results in the optimum arrangement for passenger and/or transport aircraft flying at just below the speed of sound. The fan may be coupled to the front of a number of core compression stages (two shaft engine) or a separate shaft driven by its own turbine (three shaft engine).
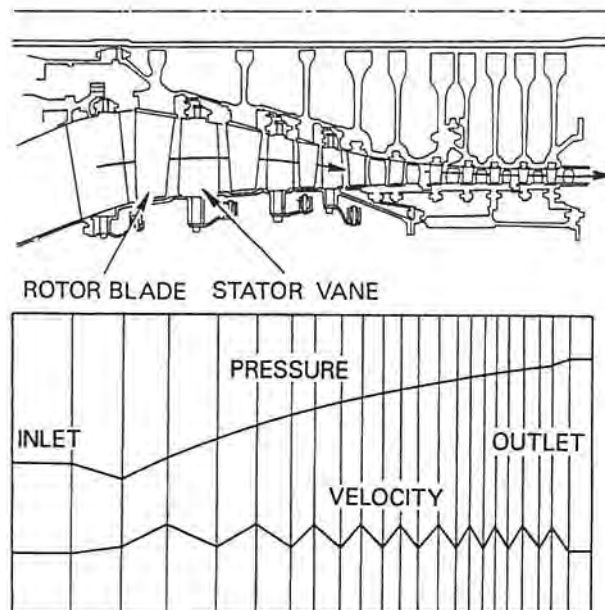
Principles of operation

19. During operation the rotor is turned at high speed by the turbine so that air is continuously induced into the compressor, which is then accelerated by the rotating blades and swept rearwards onto the adjacent row of stator vanes. The pressure rise results from the energy imparted to the air in the rotor which increases the air velocity. The air is then decelerated (diffused) in the following

Fig. 3-9   Pressure and velocity changes through an axial compressor.

stator passage and the kinetic energy translated into pressure. Stator vanes also serve to correct the deflection given to the air by the rotor blades and to present the air at the correct angle to the next stage of rotor blades. The last row of stator vanes usually act as air straighteners to remove swirl from the air prior to entry into the combustion system at a reasonably uniform axial velocity. Changes in pressure and velocity that occur in the airflow through the compressor are shown diagrammatically in fig. 3-9. The changes are accompanied by a progressive increase in air temperature as the pressure increases.

20. Across each stage the ratio of total pressures of outgoing air and inlet air is quite small, being between 1:1 and 1:2. The reason for the small pressure increase through each stage is that the rate of diffusion and the deflection angle of the .blades must be limited if losses due to air breakaway at the blades and subsequent blade stall are to be avoided. Although the pressure ratio of each stage is small, every stage increases the exit pressure of the stage that precedes it. So whilst this first stage of a compressor may only increase the pressure by 3 to 4 lb. per sq. in., at the rear of a thirty to one compression system the stage pressure rise can be up to 80 lb, per sq. in, The ability to design multi-stage axial compressors with controlled air velocities and straight through flow, minimizes losses and results in a high efficiency and hence low fuel consumption. This gives it a further advantage over the centrifugal compressor where these conditions are fundamentally not so easily achieved.

21. The more the pressure ratio of a compressor is increased the more difficult it becomes to ensure that it will operate efficiently over the full speed range. This is because the requirement for the ratio of inlet area to exit area, at the high speed case, results in an inlet area that becomes progressively too large relative to the exit area as the compressor speed and hence pressure ratio is reduced. The axial velocity of the inlet air in the front stages thus becomes low relative to the blade speed, this changes the incidence of the air onto the blades and a condition is reached where the flow separates and the compressor flow breaks down. Where high pressure ratios are required from a single compressor this problem can be overcome by introducing variable stator vanes in the front stages of the system. This corrects the incidence of air onto the rotor blades to angles which they can tolerate. An alternative is the incorporation of interstage bleeds, where a proportion of air after entering the compressor is

removed at an intermediate stage and .dumped into the bypass flow. While this method corrects the axial velocity through the preceding stages, energy is wasted and incorporation of variable stators is preferred.

22. The fan of the high by-pass ratio turbo-fan is an example of an axial compressor which has been optimized to meet the specific requirements of this cycle. While similar in principle to the core compressor stage, the proportions of design are such that the inner gas path is similar to that of the core compressor that follows it, while the tip diameter is considerably larger. The mass flow passed by the fan is typically six times that required by the core, the remaining five sixths by-pass the core and is expanded through its own coaxial nozzle, or may be mixed with the flow at exit from the core in a common nozzle. To optimize the cycle the by-pass flow has to be raised to a pressure of approximately 1.6 times the inlet pressure. This is achieved in the fan by utilizing very high tip speeds (1500 ft. per sec.) and airflow such that the by-pass section of the blades operate with a supersonic inlet air velocity of up to Mach 1.5 at the tip. The pressure that results is graded from a high value at the tip where relative velocities are highest to the more normal values of 1.3 to 1.4 at the inner radius which supercharges the core where aerodynamic design is more akin to that of a conventional compressor stage. The capability of this type of compressor stage achieves the cycle requirement of high flow per unit of frontal area, high efficiency and high pressure ratio in a single rotating blade row without inlet guide vanes within an acceptable engine diameter. Thus keeping weight and mechanical complexity at an acceptable level.

Construction
23. The construction of the compressor centres around the rotor assembly and casings. The rotor shaft is supported in ball and roller bearings and coupled to the turbine shaft in a manner that allows for any slight variation of alignment. The cylindrical casing assembly may consist of a number of cylindrical casings with a bolted axial joint between each stage or the casing may be in two halves with a bolted centre line joint. One or other of these construction methods is required in order that the casing can be assembled around the rotor.

Rotors
24. In compressor designs (fig. 3-10) the rotational speed is such that a disc is required to support the centrifugal blade load. Where a number of discs are fitted onto one shaft they may be coupled and secured together by a mechanical fixing but
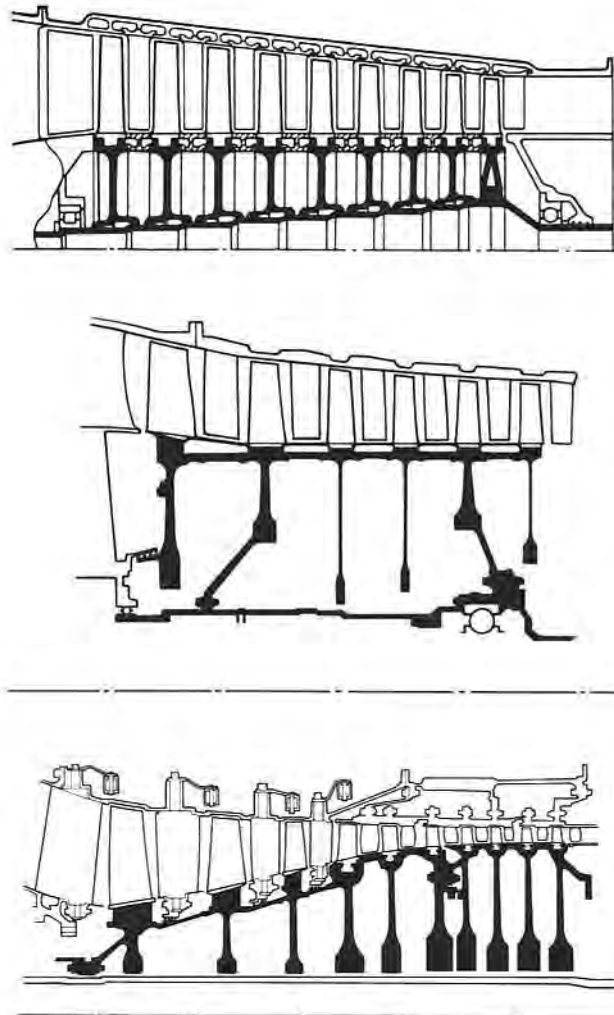
Fig. 3-10   Rotors of drum and disc construction.

generally the discs are assembled and welded together, close to their periphery, thus forming an integral drum.

25. Typical methods of securing rotor blades to the disc are shown in fig. 3-11, fixing may be circumferential or axial to suit special requirements of the stage. In general the aim is to design a securing feature that imparts the lightest possible load on the supporting disc thus minimizing disc weight. Whilst most compressor designs have separate blades for manufacturing and maintainability requirements, it becomes more difficult on the smallest engines to design a practical fixing. However this may be overcome by producing blades integral with the disc; the so called 'blisk'.
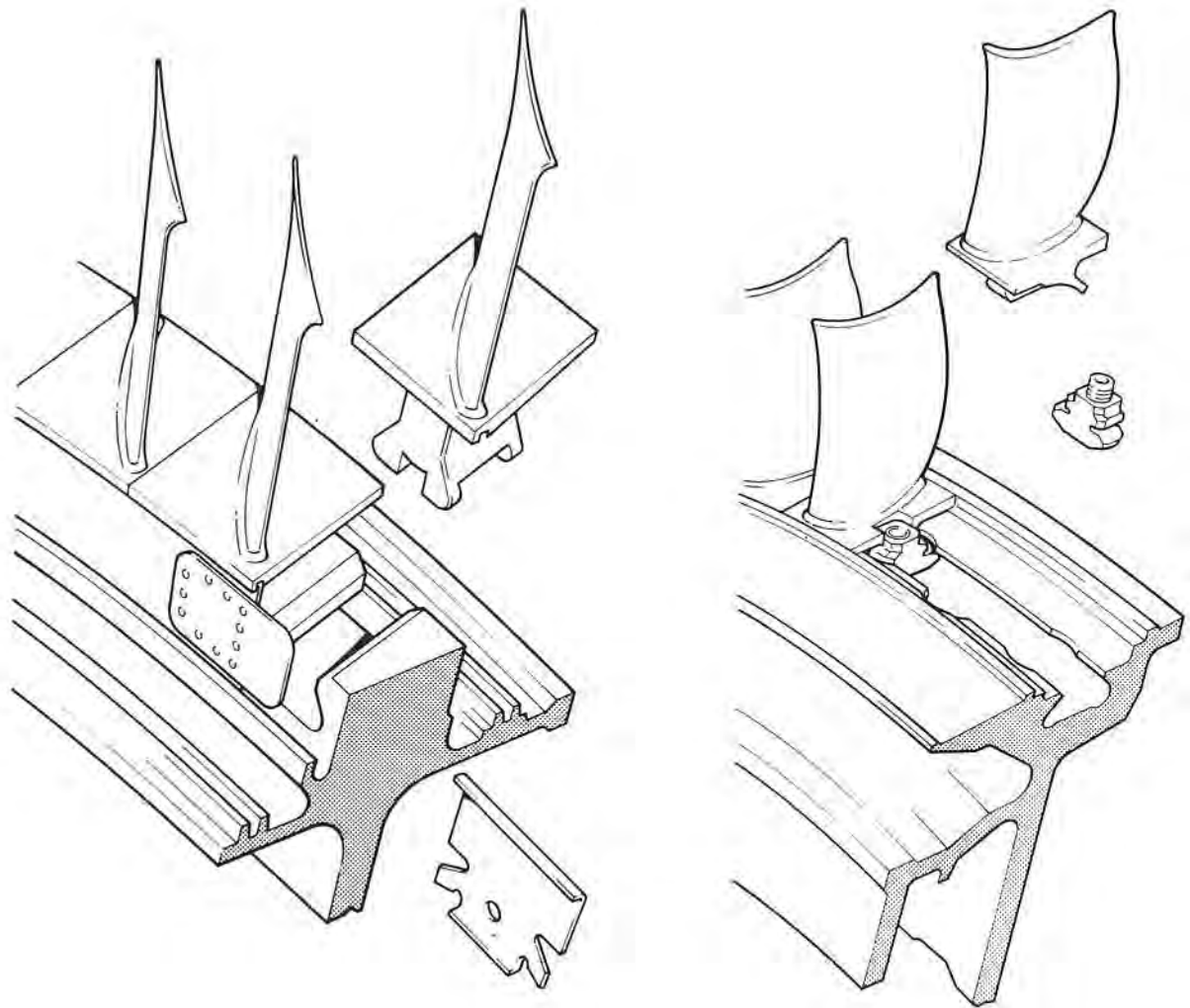
Fig. 3-11   Methods of securing blades to disc.

Rotor blades

26.   The rotor blades are of airfoil section (fig. 3-12) and usually designed to give a pressure gradient along their length to ensure that the air maintains a reasonably uniform axial velocity. The higher pressure towards the tip balances out the centrifugal action of the rotor on the airstream. To obtain these conditions, it is necessary to 'twist' the blade from root to tip to give the correct angle of incidence at each point. Air flowing through a compressor creates two boundary layers of slow to stagnant air on the inner and outer walls. In order to compensate for the slow air in the boundary layer a localized increase in blade camber both at the blade tip and root has been

introduced. The blade extremities appear as if formed by bending over each corner, hence the term 'end-bend'.

Stator vanes

27.   The stator vanes are again of airfoil section and are secured into the compressor casing or into stator vane retaining rings, which are themselves secured to the casing (fig. 3-13). The vanes are often assembled in segments in the front stages and may be shrouded at their inner ends to minimize the vibrational effect of flow variations on the longer vanes. It is also necessary to lock the stator vanes in such a manner that they will not rotate around the casing.

Fig. 3-12   A typical rotor blade showing twisted contour.

OPERATING CONDITIONS

28. Each stage of a multi-stage compressor possesses certain airflow characteristics that are dissimilar from those of its neighbour; thus to design a workable and efficient compressor, the characteristics of each stage must be carefully matched. This is a relatively simple process to implement for one set of conditions (design mass flow, pressure ratio and rotational speed), but is much more difficult when reasonable matching is to be retained with the compressor operating over a wide range of conditions such as an aircraft engine encounters.

29. If the operating conditions imposed upon the compressor blade departs too far from the design intention, breakdown of airflow and/or aerodynamically induced vibration will occur. These phenomena may take one of two forms; the blades may stall because the angle of incidence of the air relative to the blade is too high (positive incidence stall) or too low (negative incidence stall). The former is a front stage problem at low speeds and the latter usually affects the rear stages at high speed, either can lead to blade vibration which can induce rapid destruction. If the engine demands a pressure rise from the compressor, which is higher than the blading can sustain, 'surge' occurs. In this case there is an instantaneous breakdown of flow through the machine and the high pressure air in the combustion system is expelled forward through the compressor with a loud 'bang' and a resultant loss of engine thrust.

Fig. 3-13   Methods of securing vanes to compressor casing.
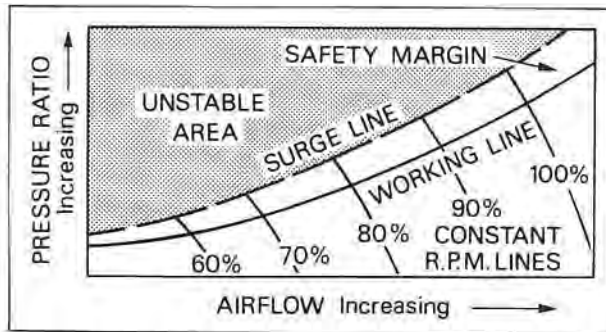
Fig. 3-14   Limits of stable airflow.

Compressors are designed with adequate margin to ensure that this area of instability (fig. 3-14) is avoided.

AIRFLOW CONTROL

30.   Where high pressure ratios on a single shaft are required it becomes necessary to introduce airflow control into the compressor design. This may take the form of variable inlet guide vanes for the first stage plus a number of stages incorporating variable stator vanes for the succeeding stages as the shaft pressure ratio is increased (fig. 3-15). As the compressor speed is reduced from its design value these static vanes are progressively closed in order to maintain an acceptable air angle value onto the following rotor blades. Additionally interstage bleed may be provided but its use in design is now usually limited to the provision of extra margin while the engine is being accelerated, because use at steady operating conditions is inefficient and wasteful of fuel. Three types of air bleed systems are illustrated as follows: fig. 3-16 hydraulic, fig. 3-17 pneumatic and fig. 3-18 electronic.

MATERIALS

31.   Materials are chosen to achieve the most cost effective design for the components in question, in practice for aero engine design this need is usually best satisfied by the lightest design that technology allows for the given loads and temperatures prevailing.
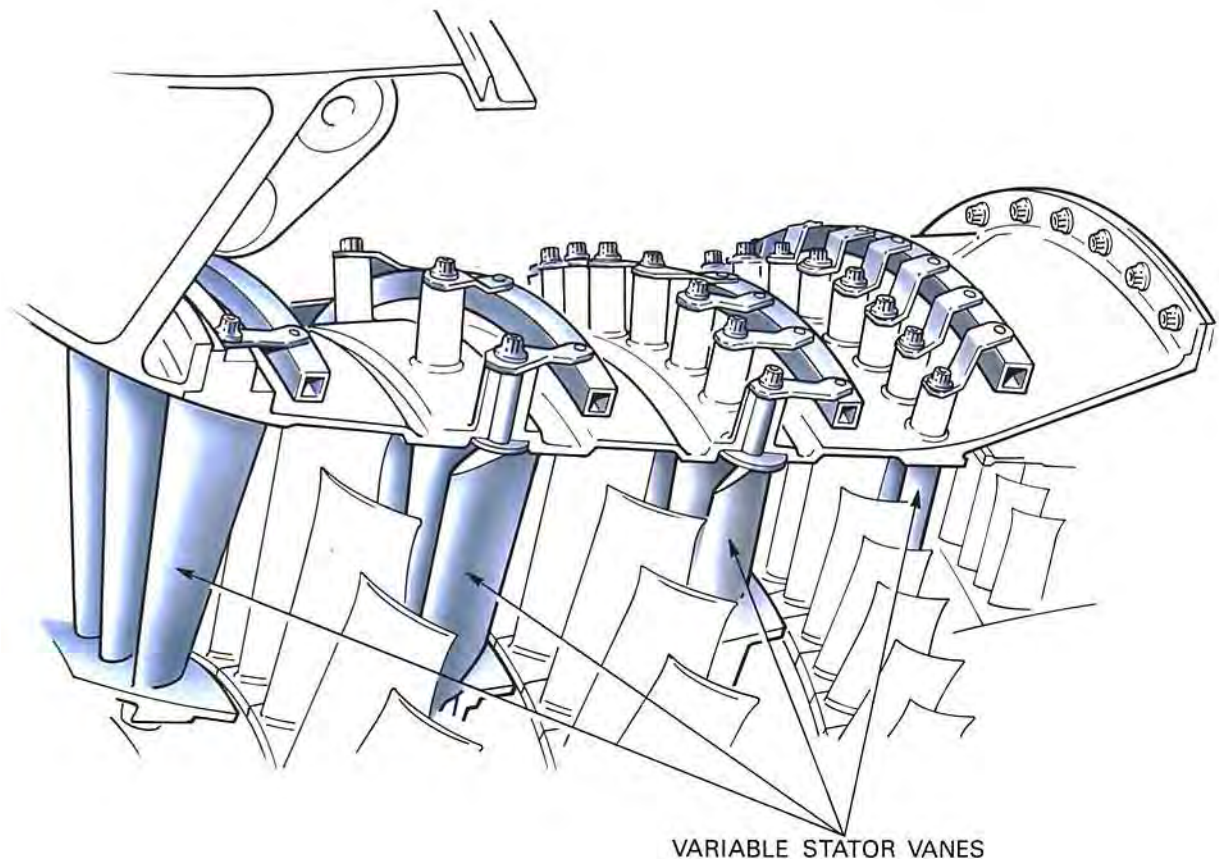
Fig. 3-15   Typical variable stator vanes.

FROM TEMPERATURE PHIAL

INLET GUIDE VANE CLOSED
(Minimum airflow position)

BLEED VALVE (Open)

REFERENCE PRESSURE
RELIEF VALVE

VARIABLE ORIFICE

WEIGHTED
VALVE

CONTINUOUS
BLEED

FROM H.P.
FUEL PUMP

AIRFLOW CONTROL
REGULATOR AND ACTUATOR

AIRFLOW CONTROL RPM
SIGNAL TRANSMITTER

L.P. fuel	Servo pressure	Reference pressure		Compressor bleed air
H.P. fuel	H.P. rev/min pressure signal	Sensing fluid		

INLET GUIDE VANE OPEN
(Maximum airflow position)

BLEED VALVE (Closed)

DIAPHRAGM

PUSH ROD

DIAPHRAGM

PISTON

SERVO
METERING VALVE

Fig. 3-16 A hydraulically operated bleed valve and inlet guide vane airflow control system.

Fig. 3-17   A pneumatically operated bleed valve system.

32. For casing designs the need is for a light but rigid construction enabling blade tip clearances to be accurately maintained ensuring the highest possible efficie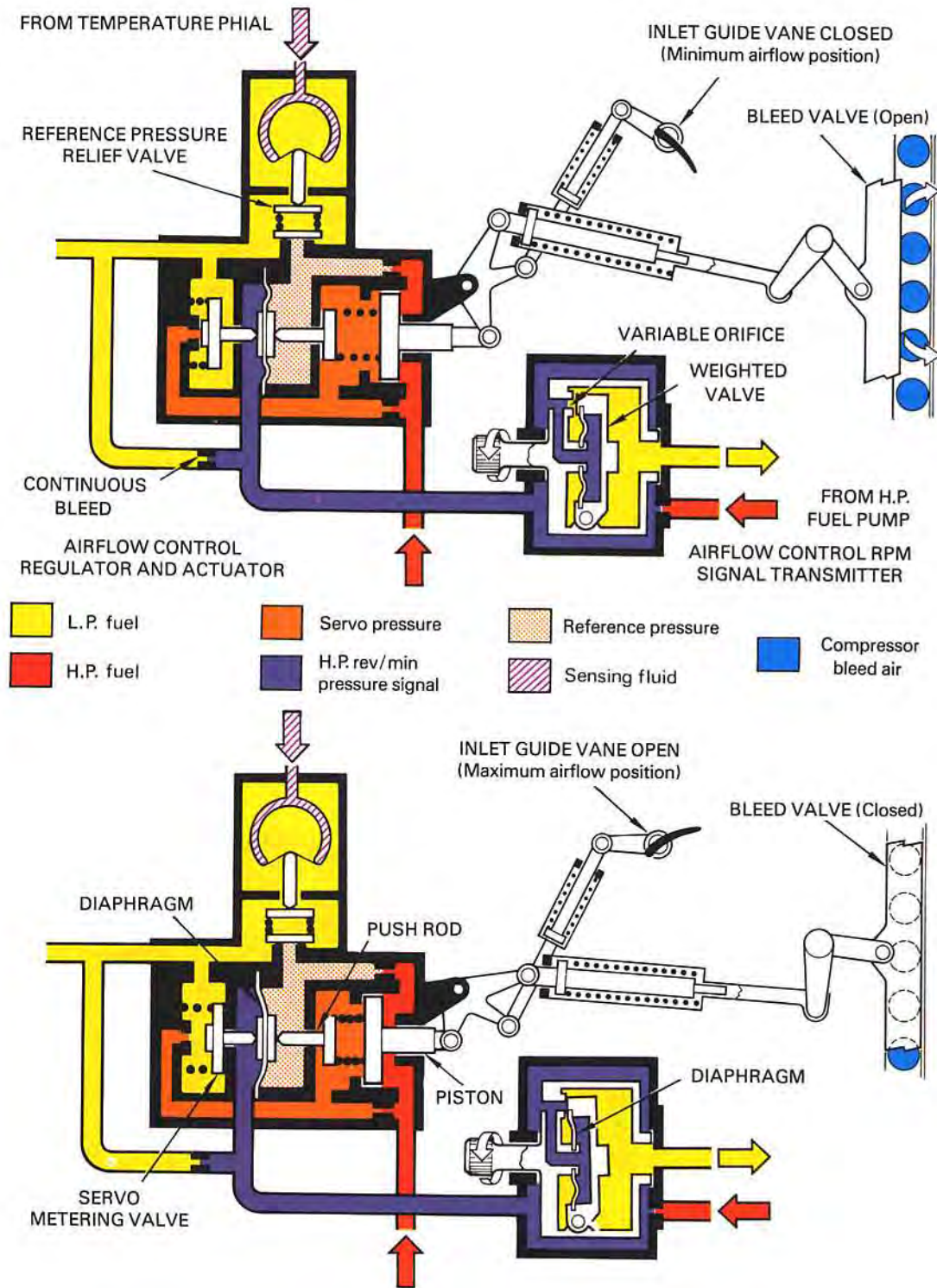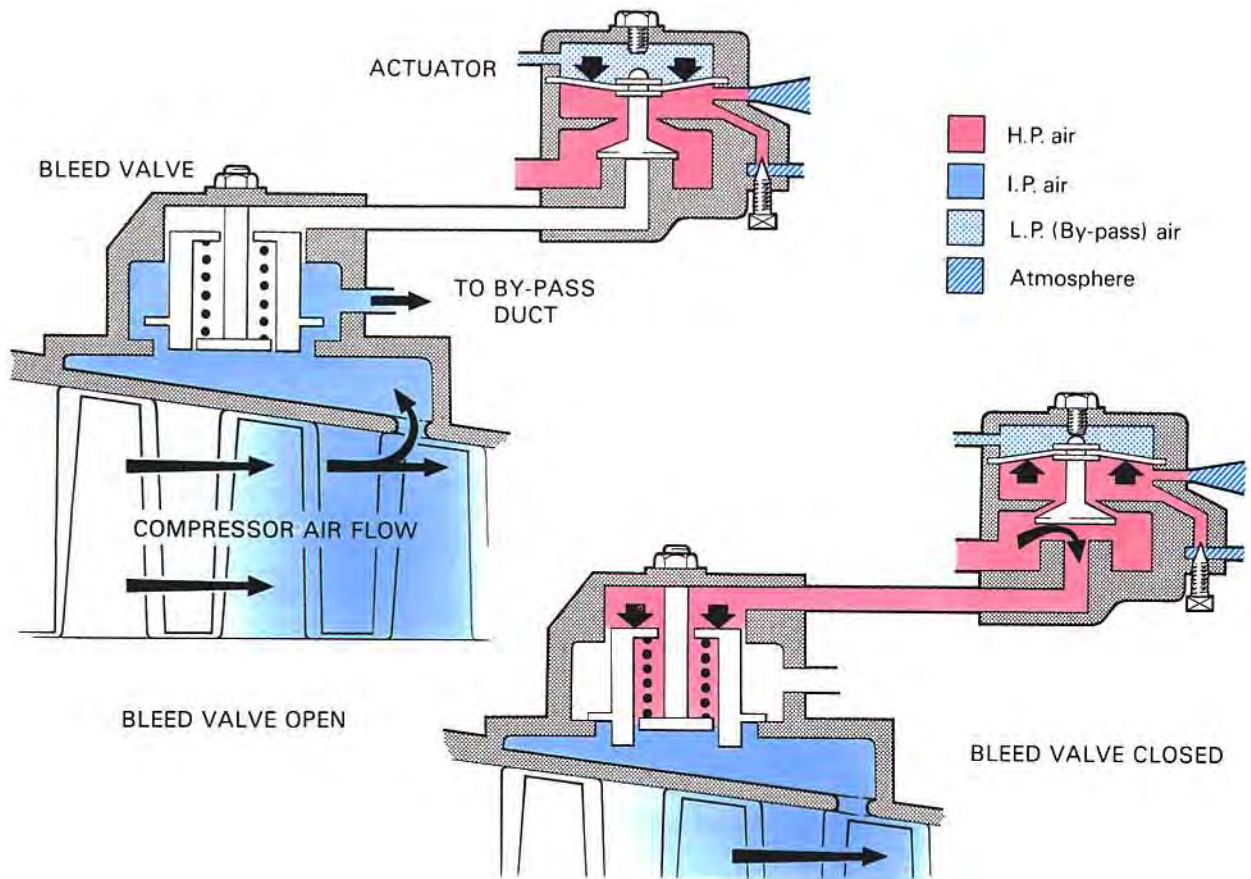ncy. These needs are achieved by using aluminium at the front of the compression system followed by .alloy steel as compression temperature increases. Whilst for the final stages of the compression system, where temperature require-ments possibly exceed the capability of the best steel, nickel based alloys may be required. The use of titanium in .preference to aluminium and steel is now more common; particularly in military engines where its high rigidity to density ratio can result in significant weight reduction. With the development of new manufacturing methods component costs can now be maintained at a more acceptable level in spite of high initial material costs.

33. Stator vanes are normally produced from steel or nickel based alloys, a prime requirement being a high fatigue strength when "notched" by ingestion

damage. Earlier designs specified aluminium alloys but because of its inferior ability to withstand damage its use has declined. Titanium may be used for stator vanes in the low pressure area but is unsuitable for the smaller stator vanes further rearwards in the compression system because of the higher pressures and temperatures encountered. Any excessive rub which may occur between rotating and static components as a result of other mechanical failures, can generate sufficient heat from friction to ignite the titanium. This in turn can lead to expensive repair costs and a possible airworthiness hazard.

34. In the design of rotor discs, drums and blades, centrifugal forces dominate and the requirement is for metal with the highest ratio of strength to density. This results in the lightest possible rotor assembly which in turn reduces the forces on the engine structure enabling a further reduction in weight to be obtained. For this reason, titanium even with its high initial cost is the preferred material and has replaced

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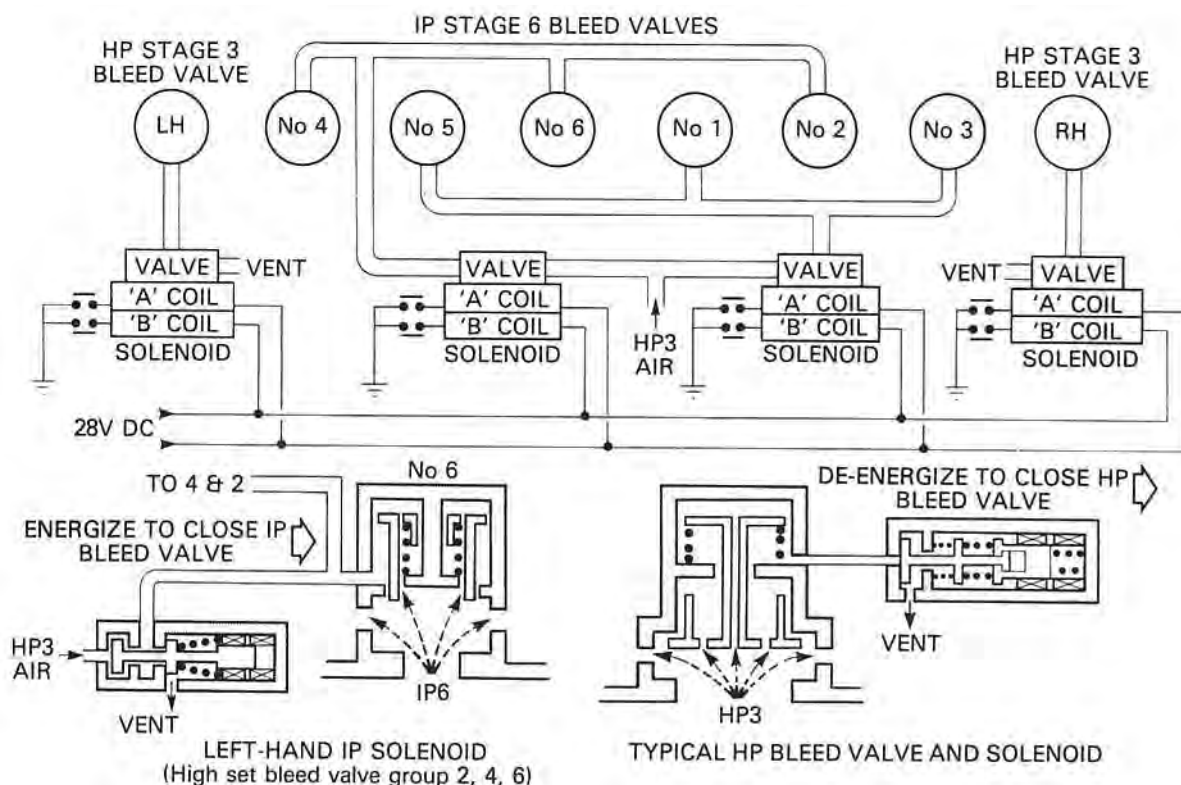
Fig. 3-18   An electronically operated bleed valve system.

the steel alloys that were favoured in earlier designs. As higher temperature titanium alloys are developed and produced they are progressively displacing the nickel alloys for the disc and blades at the rear of the system.

35.   The high by-pass ratio fan blade (fig. 3-19) only became a design possibility with the availability of titanium, conventional designs being machined from solid forgings. A low weight fan blade is necessary because the front structure of the engine must be able to withstand the large out of balance forces that would result from a fan blade failure. To achieve a sufficiently light solid fan blade, even with titanium, requires a short axial length (or chord). However, with this design, the special feature of a mid-span support ('snubber' or 'clapper') is required to prevent aerodynamic instability. This design concept has the disadvantage of the snubber being situated in the supersonic flow where pressure losses are greatest, resulting in inefficiency and a reduction in airflow. This disadvantage has been overcome with the introduction of the Rolls-Royce designed wide chord fan blade; stability is provided by the increased chord of the blade thus avoiding the need for snubbers. The weight is maintained at a low level by fabricating

Fig. 3-19   Typical types of fan blades.

the blade from skins of titanium incorporating a honeycomb core.

36. Centrifugal impeller material requirements are similar to those for the axial compressor rotors. Titanium is thus normally specified though aluminium may still be employed on the largest low pressure ratio designs where robust sections give adequate ingestion capability and temperatures are acceptably low.

BALANCING

37. The balancing of a compressor rotor or impeller is an extremely important operation in its manufacture. In view of the high rotational speeds and the mass of materials any unbalance would affect the rotating assembly bearings and engine operation. Balancing on these parts is effected on a special balancing machine, the principles of which are briefly described in Part 25.

Rolls-Royce RB211 Trent
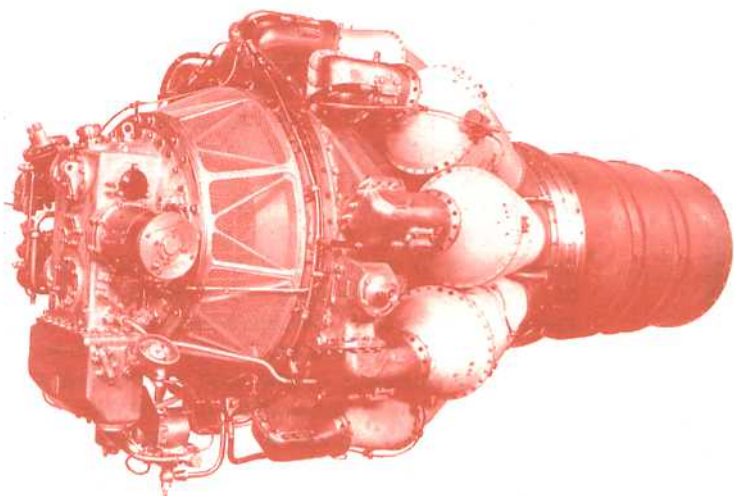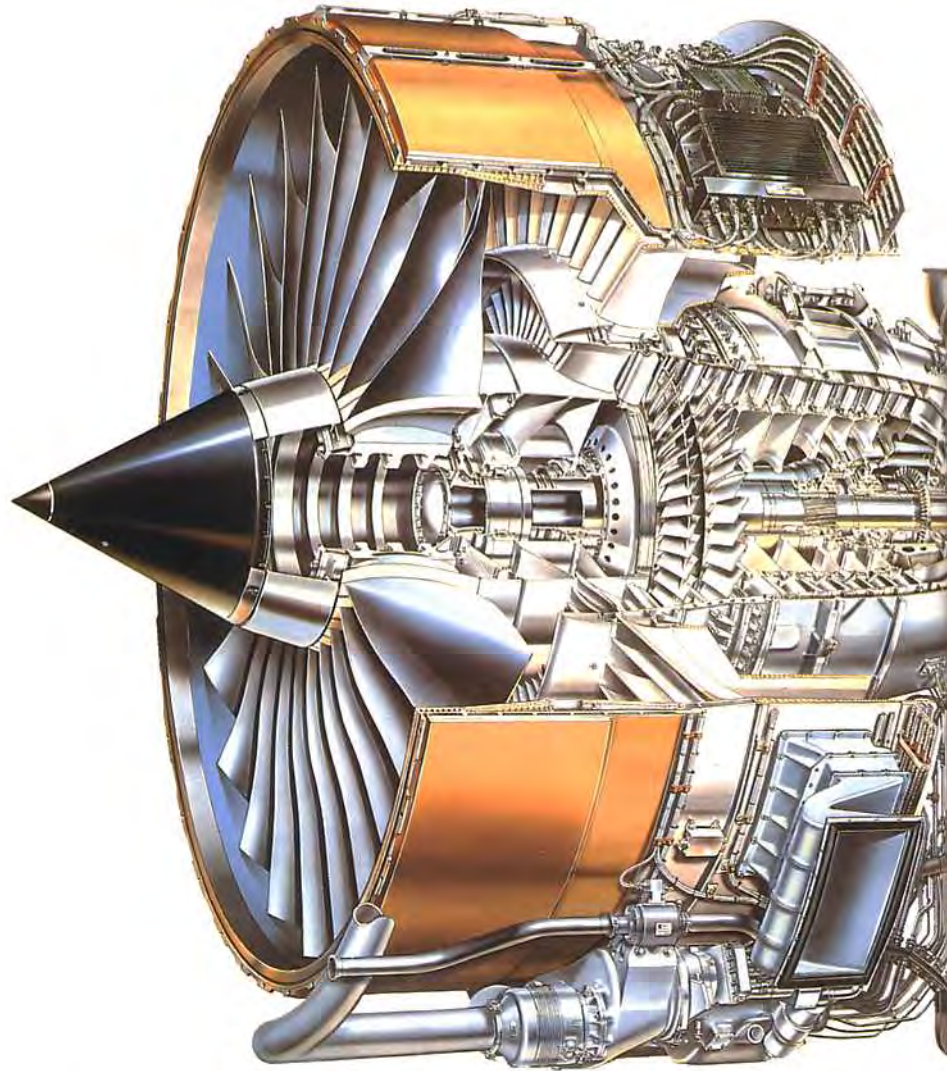
Rolls-Royce RB41 Nene

On 17 March 1944 Rolls-Royce commenced work on the RB40 as the result of a Government request for a turbo-jet of 4200 lb thrust. After discussions with Supermarine, the airframe designers, the engine was scaled down to produce 3400 lb. The resulting Nene was eventually rated at 5000 lb and powered the Hawker Sea Hawk and Supermarine Attacker.

4: Combustion chambers

Contents Page

INTRODUCTION

1. The combustion chamber (fig. 4-1) has the difficult task of burning large quantities of fuel, supplied through the fuel spray nozzles (Part 10), with extensive volumes of air, supplied by the compressor (Part 3), and releasing the heat in such a manner that the air is expanded and accelerated to give a smooth stream of uniformly heated gas at all conditions required by the turbine (Part 5). This task must be accomplished with the minimum loss in pressure and with the maximum heat release for the limited space available.

2. The amount of fuel added to the air will depend upon the temperature rise required. However, the maximum temperature is limited to within the range of 850 to 1700 deg. C. by the materials from which the turbine blades and nozzles are made. The air has already been heated to between 200 and 550 deg. C. by the work done during compression, giving a temperature rise requirement of 650 to 1150 deg. C. from the combustion process. Since the gas temperature required at the turbine varies with engine thrust, and in the case of the turbo-propeller engine upon the power required, the combustion chamber must also be capable of maintaining stable and efficient combustion over a wide range of engine operating conditions.

3. Efficient combustion has become increasingly important because of the rapid rise in commercial aircraft traffic and the consequent increase in atmospheric pollution, which is seen by the general public as exhaust smoke.
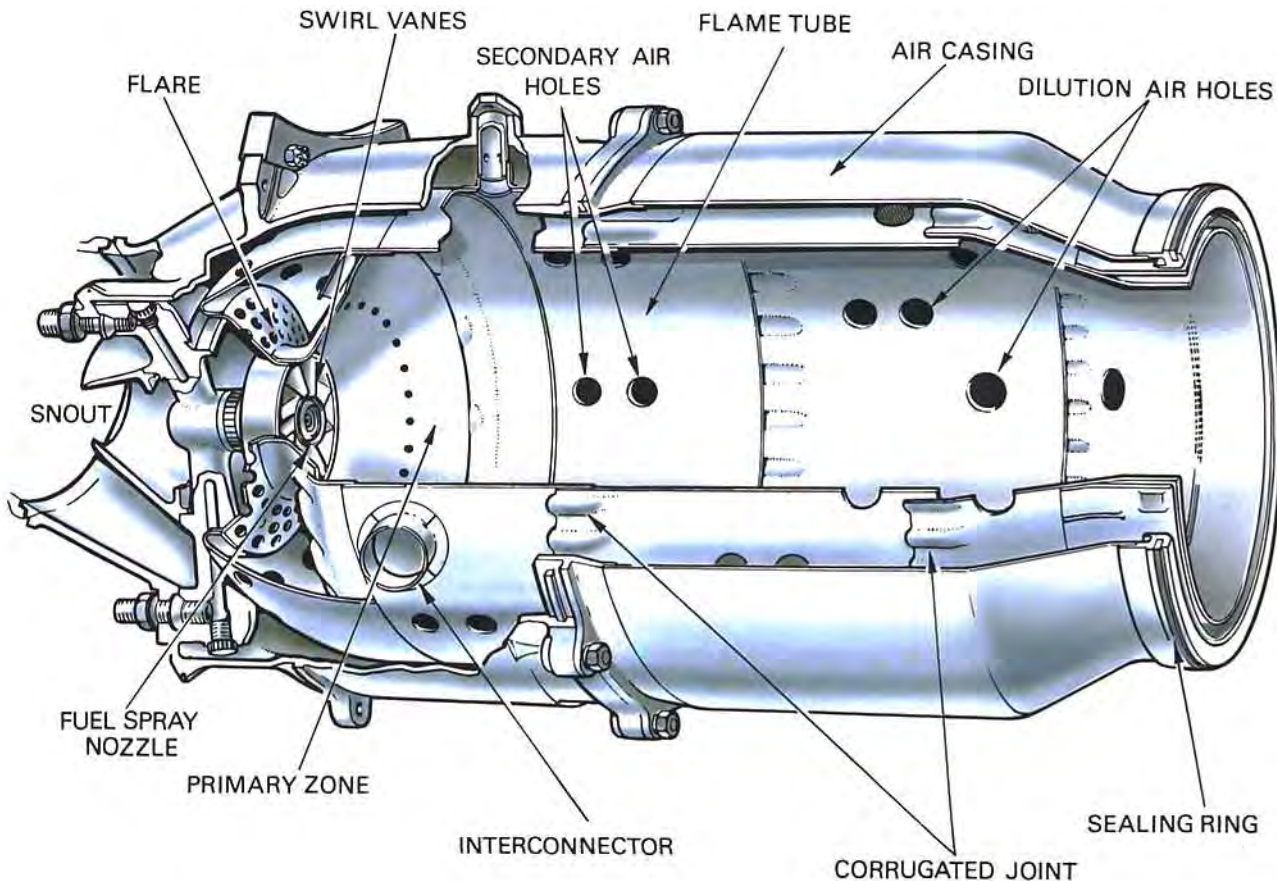
Fig. 4-1   An early combustion chamber.

COMBUSTION PROCESS

4. Air from the engine compressor enters the combustion chamber at a velocity up to 500 feet per second, but because at this velocity the air speed is far too high for combustion, the first thing that the chamber must do is to diffuse it, i.e. decelerate it and raise its static pressure. Since the speed of burning kerosine at normal mixture ratios is only a few feet per second, any fuel lit even in the diffused air stream, which now has a velocity of about 80 feet per second, would be blown away. A region of low axial velocity has therefore to be created in the chamber, so that the flame will remain alight throughout the range of engine operating conditions.

5. In normal operation, the overall air/fuel ratio of a combustion chamber can vary between 45:1 and 130:1, However, kerosine will only burn efficiently at, or close to, a ratio of 15:1, so the fuel must be burned with only part of the air entering the chamber, in what is called a primary combustion zone. This is achieved by means of a flame tube (combustion liner) that has

various devices for metering the airflow distribution along the chamber.

6. Approximately 20 per cent of the air mass flow is taken in by the snout or entry section (fig. 4-2). Immediately downstream of the snout are swirl vanes and a perforated flare, through which air passes into the primary combustion zone. The swirling air induces a flow upstream of the centre of the flame tube and promotes the desired recirculation. The air not picked up by the snout flows into the annular space between the flame tube and the air casing.

7. Through the wall of the flame tube body, adjacent to the combustion zone, are a selected number of secondary holes through which a further 20 per cent of the main flow of air passes into the primary zone. The air from the swirl vanes and that from the secondary air holes interacts and creates a region of low velocity recirculation. This takes the form of a toroidal vortex, similar to a smoke ring, which has the effect of stabilizing and anchoring the flame (fig, 4-3). The recirculating gases hasten the burning of freshly
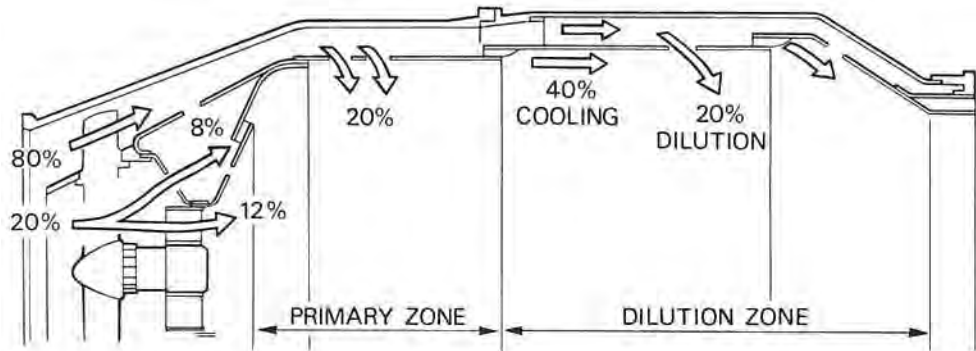
Fig. 4-2   Apportioning the airflow.

injected fuel droplets by rapidly bringing them to ignition temperature.

8.   It is arranged that the conical fuel spray from the nozzle intersects the recirculation vortex at its centre. This action, together with the general turbulence in the primary zone, greatly assists in breaking up the fuel and mixing it with the incoming air.

9.   The temperature of the gases released by combustion is about 1,800 to 2,000 deg. C., which is far too hot for entry to the nozzle guide vanes of the turbine. The air not used for combustion, which amounts to about 60 per cent of the total airflow, is therefore introduced progressively into the flame tube. Approximately a third of this is used to lower the gas temperature in the dilution zone before it enters

the turbine and the remainder is used for cooling the walls of the flame tube. This is achieved by a film of cooling air flowing along the inside surface of the flame tube wall, insulating it from the hot combustion gases (fig. 4-4). A recent development allows cooling air to enter a network of passages within the flame tube wall before exiting to form an insulating film of air, this can reduce the required wall cooling airflow by up to 50 per cent. Combustion should be completed before the dilution air enters the flame tube, otherwise the incoming air will cool the flame and incomplete combustion will result.

10.   An electric spark from an igniter plug (Part 11) initiates combustion and the flame is then self-sustained.
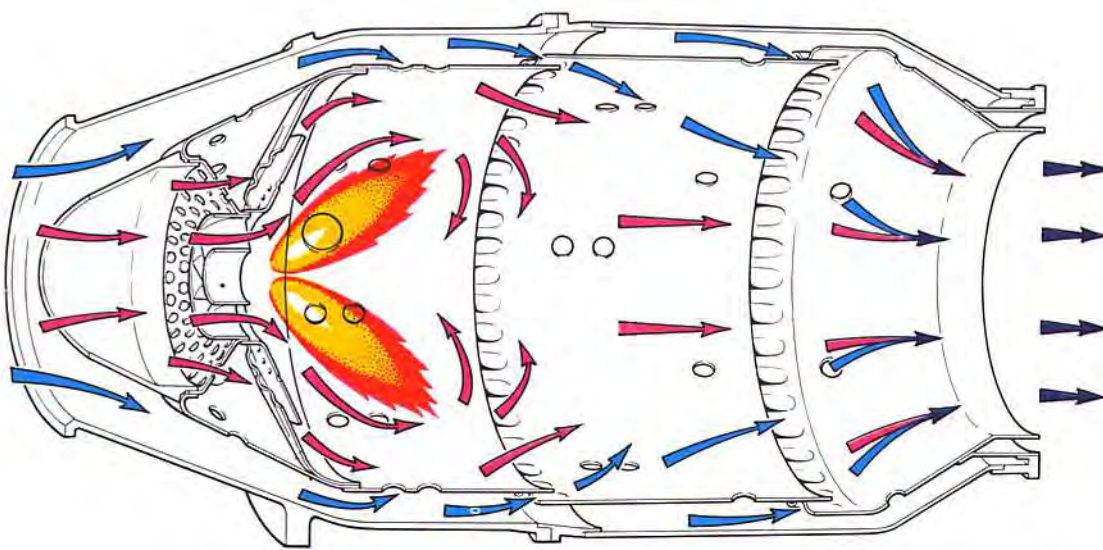
Fig. 4-3   Flame stabilizing and general airflow pattern.

CORRUGATED STRIP COOLING

MACHINED COOLING RING

SPLASH COOLING STRIP

TRANSPIRATION COOLING

Fig. 4-4   Flame tube cooling methods.

11. The design of a combustion chamber and the method of adding the fuel may vary considerably, but the airflow distribution used to effect and maintain combustion is always very similar to that described.

FUEL SUPPLY

12. Fuel is supplied to the airstream by one of two distinct methods. The most common is the injection of a fine atomized spray into the recirculating airstream through spray nozzles (Part 10). The second method is based on the pre-vaporization of the fuel before it enters the combustion zone.
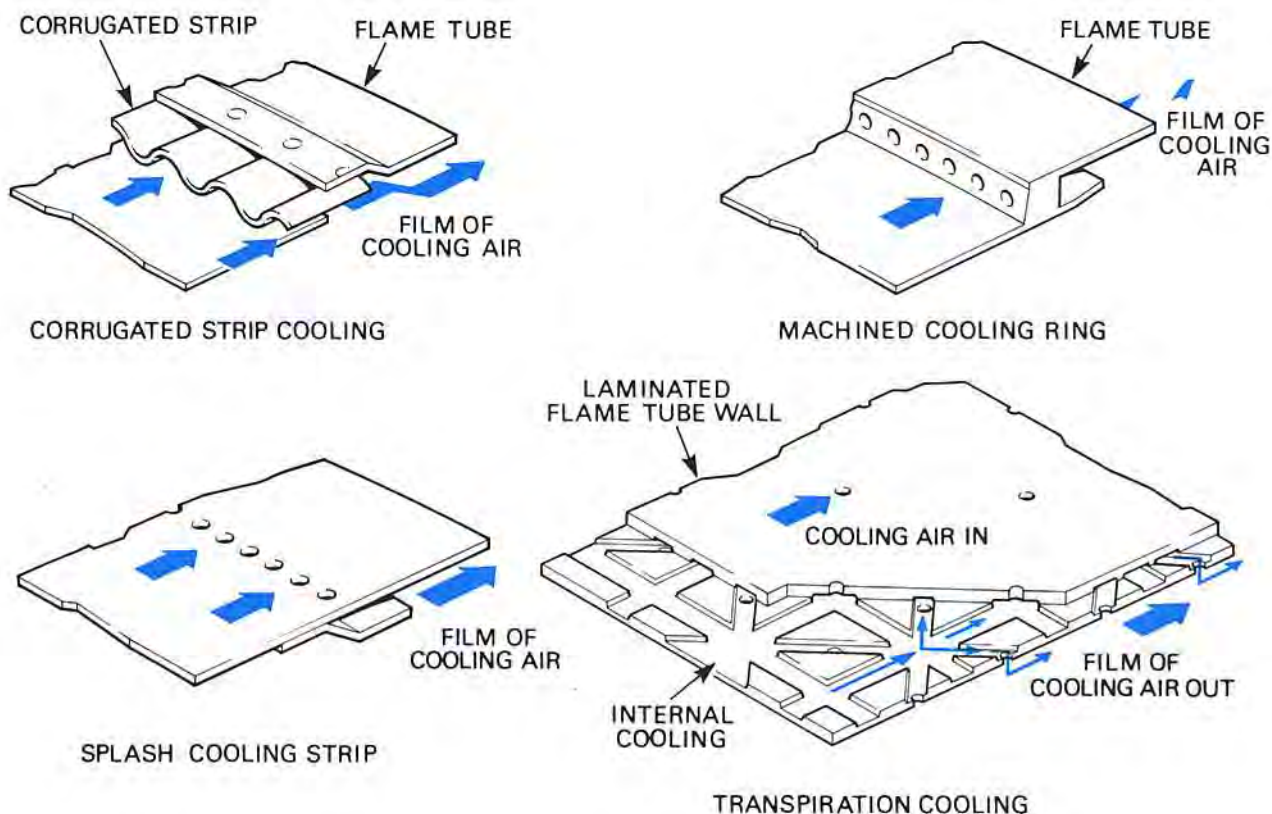
13. In the vaporizing method (fig.4-5) the fuel is sprayed from feed tubes into vaporizing tubes which are positioned inside the flame tube. These tubes turn the fuel through 180 degrees and, as they are heated by combustion, the fuel vaporizes before passing into the flame tube. The primary airflow passes down the vaporizing tubes with the fuel and also through holes in the flame tube entry section which provide 'fans' of air to sweep the flame rearwards. Cooling and dilution air is metered into

the flame tube in a manner similar to the atomizer flame tube.

TYPES OF COMBUSTION CHAMBER

14. There are three main types of combustion chamber in use for gas turbine engines. These are the multiple chamber, the tubo-annular chamber and the annular chamber.

Multiple combustion chamber
15. This type of combustion chamber is used on centrifugal compressor engines and the earlier types of axial flow compressor engines. It is a direct development of the early type of Whittle combustion chamber. The major difference is that the Whittle chamber had a reverse flow as illustrated in fig. 4-6 but, as this created a considerable pressure loss, the straight-through multiple chamber was developed by Joseph Lucas Limited.

16. The chambers are disposed around the engine (fig. 4-7) and compressor delivery air is directed by ducts to pass into the individual chambers. Each
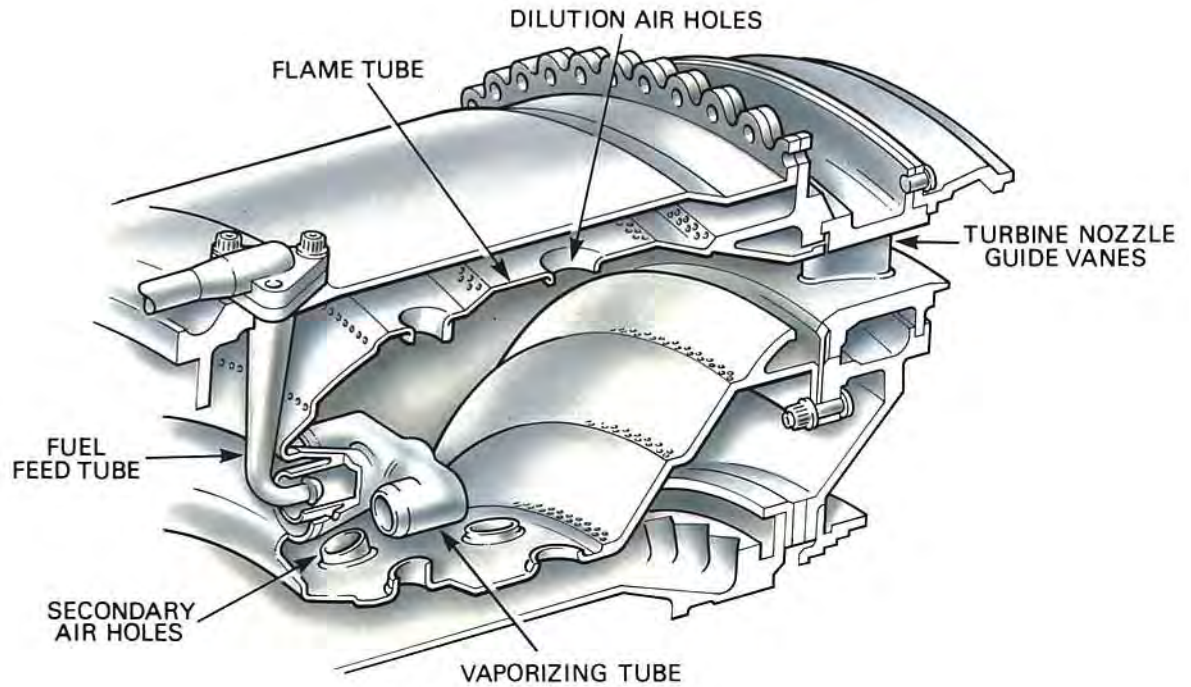
Fig. 4-5   A vaporizer combustion chamber.

chamber has an inner flame tube around which there is an air casing. The air passes through the flame tube snout and also between the tube and the outer casing as already described in para. 6.

17.   The separate flame tubes are all interconnected. This allows each tube to operate at the same pressure and also allows combustion to propagate around the flame tubes during engine starting.
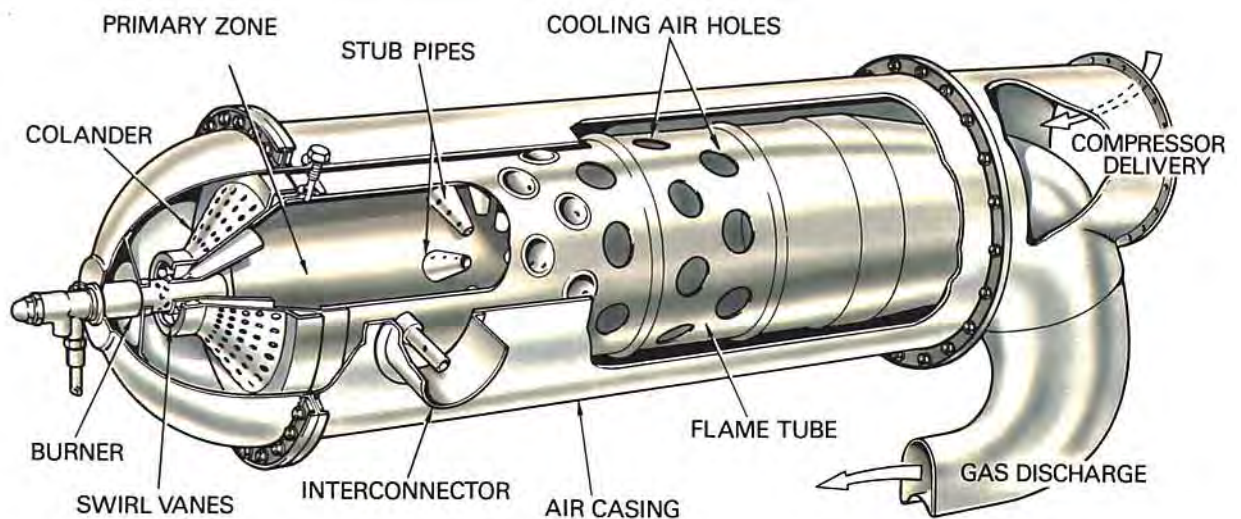
Fig. 4-6   An early Whittle combustion chamber.

COMPRESSOR OUTLET ELBOW FLANGE JOINT

MAIN FUEL MANIFOLD

ENGINE FIRESEAL

COMBUSTION CHAMBER

AIR CASING

DRAIN TUBE

INTERCONNECTOR

PRIMARY FUEL MANIFOLD

PRIMARY AIR SCOOP
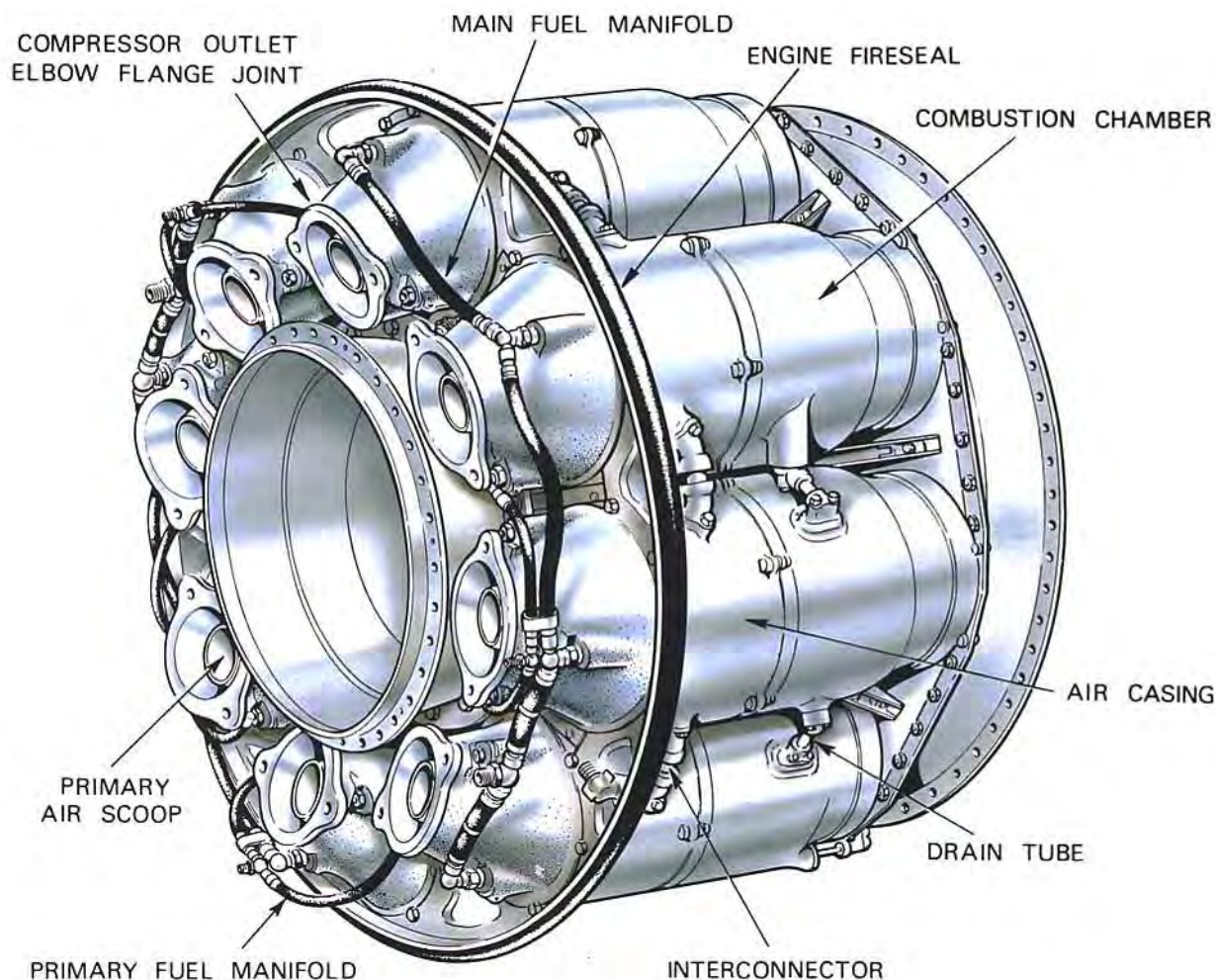
Fig. 4-7   Multiple combustion chambers.

Tubo-annular combustion chamber

18.   The tubo-annular combustion chamber bridges the evolutionary gap between the multiple and annular types. A number of flame tubes are fitted inside a common air casing (fig. 4-8). The airflow is similar to that already described. This arrangement combines the ease of overhaul and testing of the multiple system with the compactness of the annular system.

Annular combustion chamber

19.   This type of combustion chamber consists of a single flame tube, completely annular in form, which is contained in an inner and outer casing (fig. 4-9). The airflow through the flame tube is similar to that already described, the chamber being open at the front to the compressor and at the rear to the turbine nozzles.

20.   The main advantage of the annular chamber is that, for the same power output, the length of the chamber is only 75 per cent of that of a tubo-annular system of the same diameter, resulting in considerable saving of weight and production cost. Another advantage is the elimination of combustion propagation problems from chamber to chamber.

21.   In comparison with a tubo-annular combustion system, the wall area of a comparable annular chamber is much less; consequently the amount of cooling air required to prevent the burning of the flame tube wall is less, by approximately 15 per cent, This reduction in cooling air raises the combustion efficiency (para. 27) to virtually eliminate unburnt fuel, and oxidizes the carbon monoxide to non-toxic carbon dioxide, thus reducing air pollution.

22.   The introduction of the air spray type fuel spray nozzle (Part 10) to this type of combustion chamber
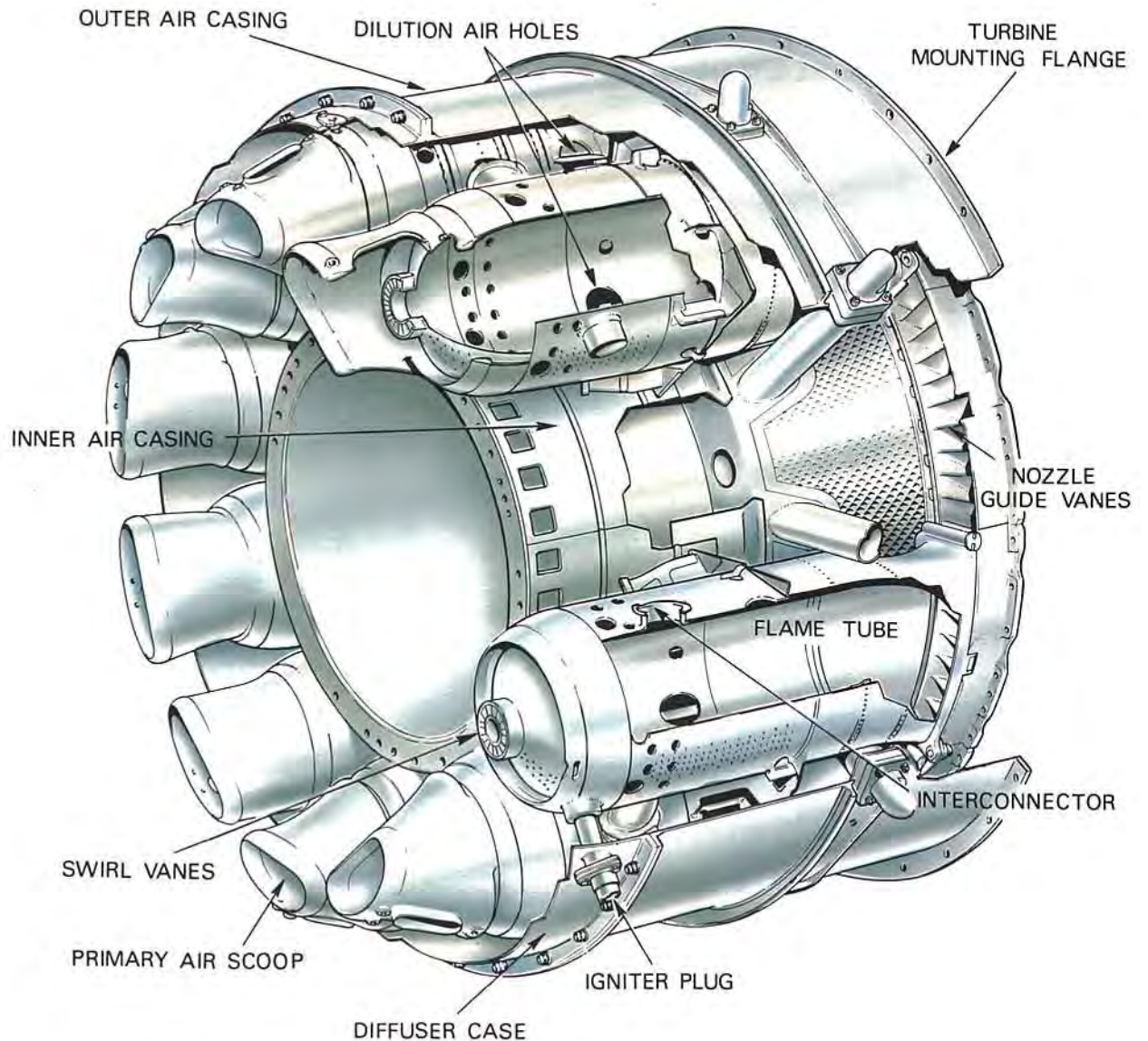
Fig. 4-8   Tubo-annular combustion chamber.

also greatly improves the preparation of fuel for combustion by aerating the over-rich pockets of fuel vapours close to the spray nozzle; this results in a large reduction in initial carbon formation.

COMBUSTION CHAMBER PERFORMANCE

23. A combustion chamber must be capable of allowing fuel to burn efficiently over a wide range of operating conditions without incurring a large pressure loss. In addition, if flame extinction occurs, then it must be possible to relight. In performing these functions, the flame tube and spray nozzle atomizer components must be mechanically reliable.

24. The gas turbine engine operates on a constant pressure cycle, therefore any loss of pressure during the process of combustion must be kept to a minimum. In providing adequate turbulence and mixing, a total pressure loss varying from about 3 to 8 per cent of the air pressure at entry to the chamber is incurred.

Combustion intensity

25. The heat released by a combustion chamber or any other heat generating unit is dependent on the volume of the combustion area. Thus, to obtain the required high power output, a comparatively small

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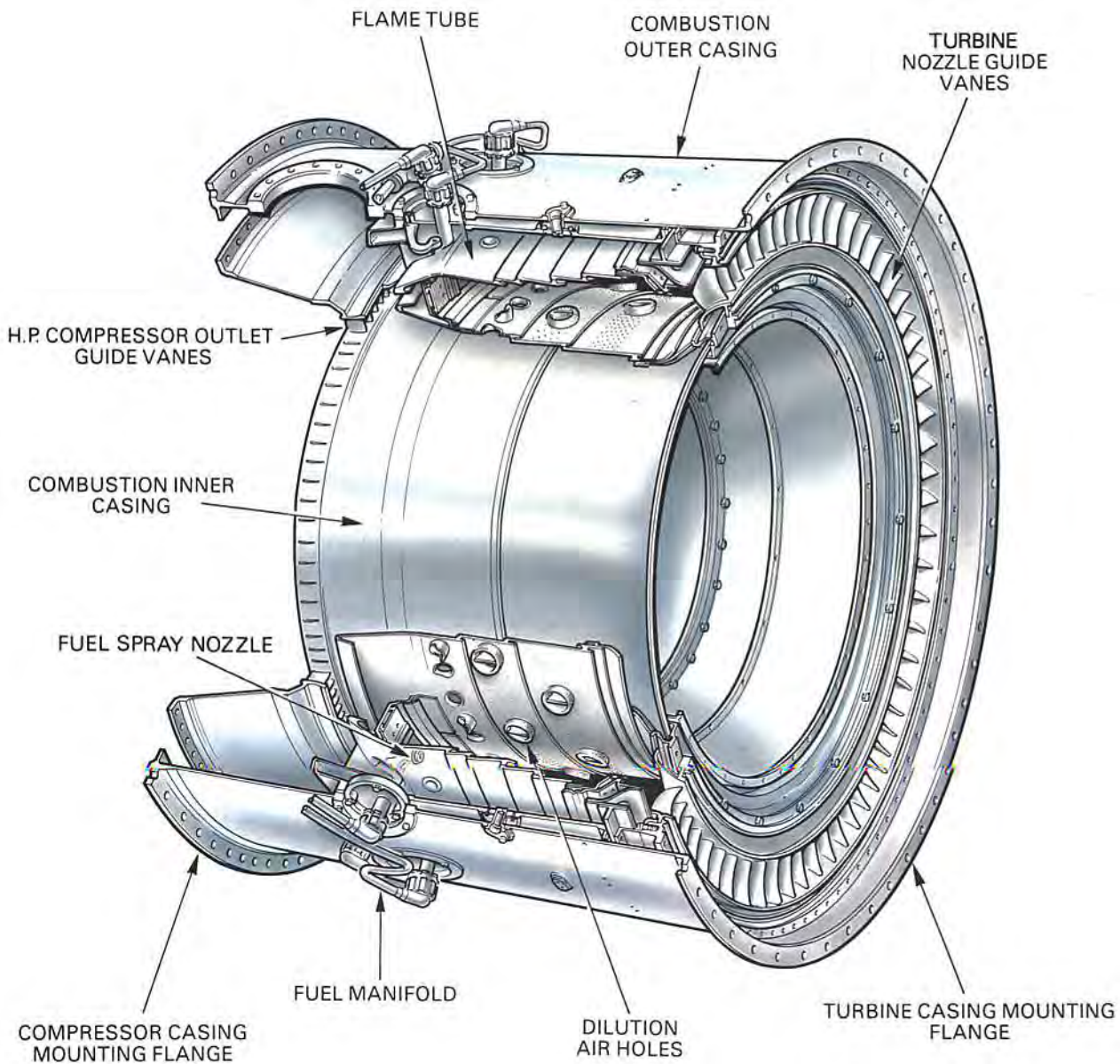
Fig. 4-9   Annular combustion chamber.

and compact gas turbine combustion chamber must release heat at exceptionally high rates.

26. For example, at take-off conditions a Rolls-Royce RB211-524 engine will consume 20,635 lb. of fuel per hour. The fuel has a calorific value of approximately 18,550 British thermal units per lb., therefore the combustion chamber releases nearly 106,300 British thermal units per second. Expressed in another way this is an expenditure of potential heat at a rate equivalent to approximately 150,000 horsepower.

Combustion efficiency

27. The combustion efficiency of most gas turbine engines at sea-level take-off conditions is almost 100 per cent, reducing to 98 per cent at altitude cruise conditions, as shown in fig. 4-10.
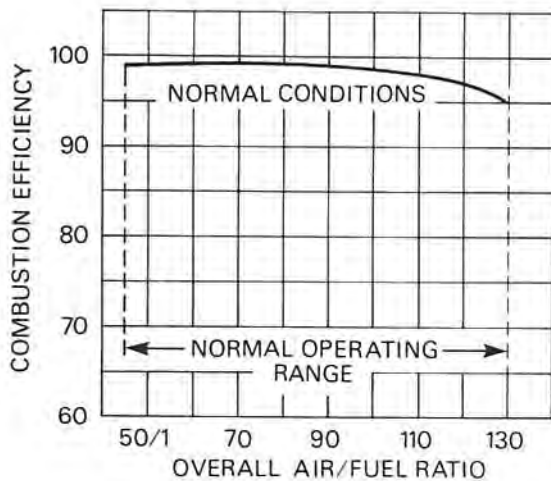
Fig. 4-10 Combustion efficiency and air/fuel ratio.

Combustion stability

28. Combustion stability means smooth burning and the ability of the flame to remain alight over a wide operating range.

29. For any particular type of combustion chamber there is both a rich and weak limit to the air/fuel ratio, beyond which the flame is extinguished. An extinction is most likely to occur in flight during a glide or dive with the engine idling, when there is a high airflow and only a small fuel flow, i.e. a very weak mixture strength.

30. The range of air/fuel ratio between the rich and weak limits is reduced with an increase of air velocity, and if the air mass flow is increased beyond a certain value, flame extinction occurs. A typical stability loop is illustrated in fig. 4-11. The operating range defined by the stability loop must obviously cover the air/fuel ratios and mass flow of the combustion chamber.

31. The ignition process has weak and rich limits similar to those shown for stability in fig. 4-11. The ignition loop, however, lies within the stability loop since it is more difficult to establish combustion under 'cold' conditions than to maintain normal burning.

Emissions

32. The unwanted pollutants which are found in the exhaust gases are created within the combustion chamber. There are four main pollutants which are legislatively controlled; unburnt hydrocarbons (unburnt fuel), smoke (carbon particles), carbon monoxide and oxides of nitrogen. The principal conditions which affect the formation of pollutants are pressure, temperature and time.

33. In the fuel rich regions of the primary zone, the hydrocarbons are converted into carbon monoxide and smoke, Fresh dilution air can be used to oxidize the carbon monoxide and smoke into non-toxic carbon dioxide within the dilution zone. Unburnt hydrocarbons can also be reduced in this zone by continuing the combustion process to ensure complete combustion.

34. Oxides of nitrogen are formed under the same conditions as those required for the suppression of the other pollutants, Therefore it is desirable to cool the flame as quickly as possible and to reduce the time available for combustion. This conflict of conditions requires a compromise to be made, but continuing improvements in combustor design and performance has led to a substantially 'cleaner' combustion process.
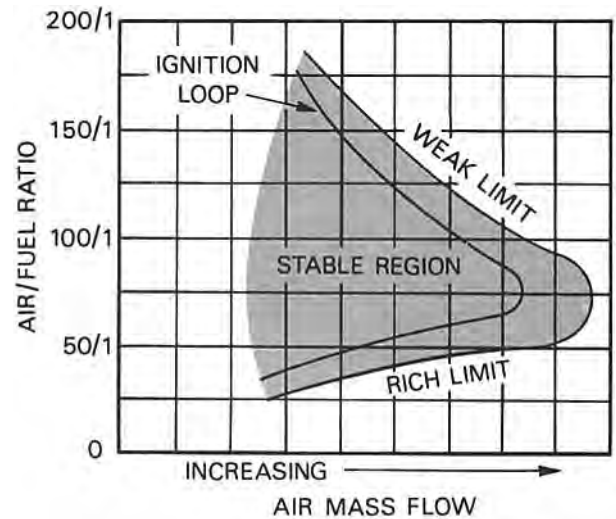
Fig. 4-11 Combustion stability limits.

MATERIALS

35. The containing walls and internal parts of the combustion chamber must be capable of resisting the very high gas temperature in the primary zone. In practice, this is achieved by using the best heat-resisting materials available, the use of high heat resistant coatings and by cooling the inner wall of the flame tube as an insulation from the flame.

36. The combustion chamber must also withstand corrosion due to the products of the combustion, creep failure due to temperature gradients and fatigue due to vibrational stresses.

Rolls-Royce Turbomeca Adour Mk102
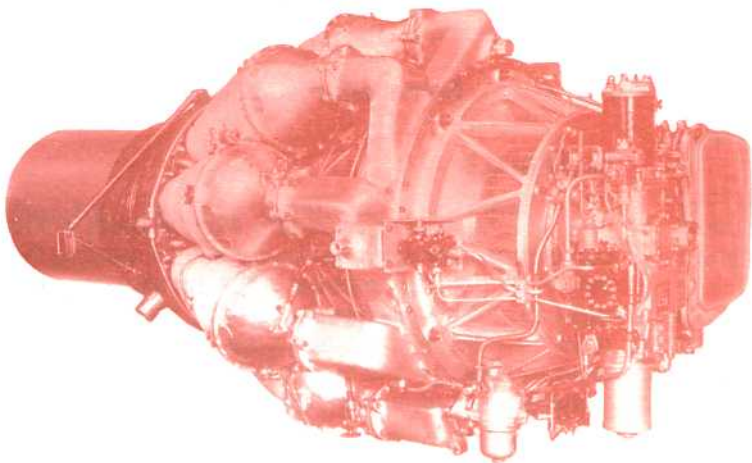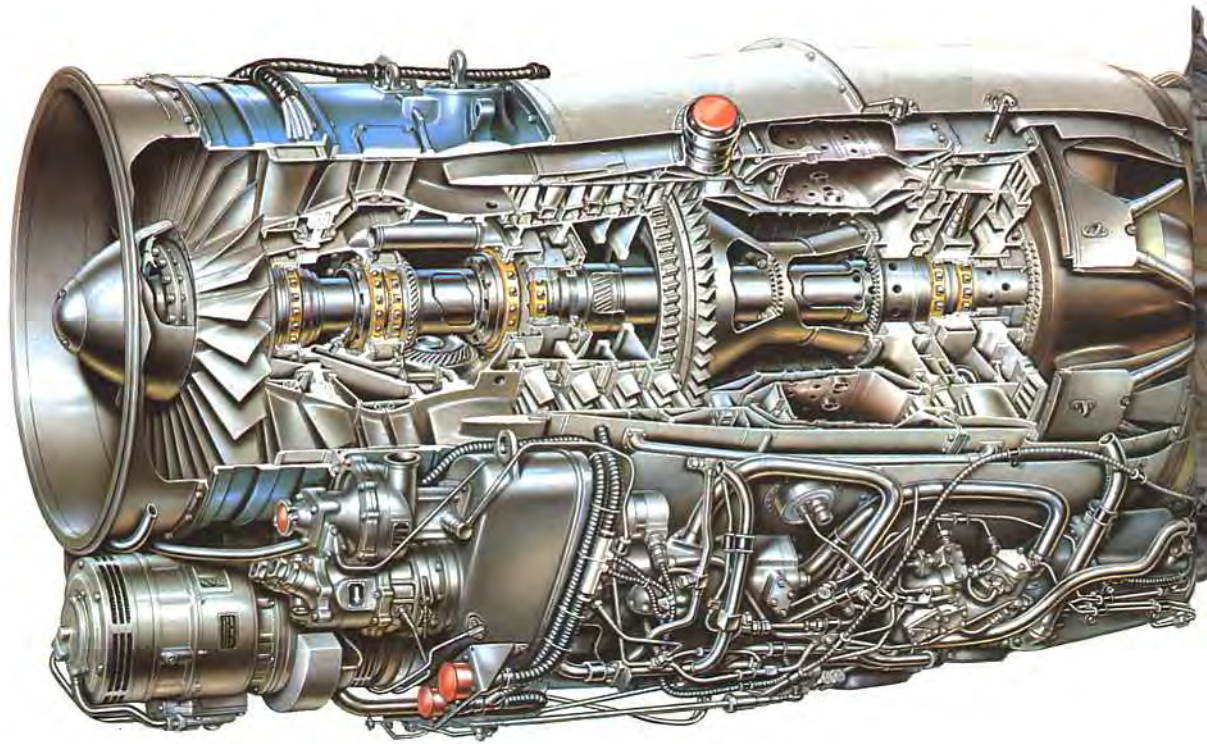
Work commenced in January 1945 on a 0.855 scale Nene, reduced to fit the engine nacelle of a Gloster Meteor. Known as the Derwent V the engine passed a 100 hr test at 2600 lb thrust in June 1945 and in September went into production with a service rating of 3500 lb. Two world speed records were set by Meteor IV's powered by special Derwent V's in November 1945 and September 1946.

Rolls-Royce RB37 Derwent V

5: Turbines

Contents

INTRODUCTION

1.   The turbine has the task of providing the power to drive the compressor and accessories and, in the case of engines which do not make use solely of a jet for propulsion, of providing shaft power for a propeller or rotor. It does this by extracting energy from the hot gases released from the combustion system and expanding them to a lower pressure and temperature. High stresses are involved in this process, and for efficient operation, the turbine blade tips may rotate at speeds over 1,500 feet per second, The continuous flow of gas to which the turbine is exposed may have an entry temperature between 850 and 1,700 deg. C. and may reach a velocity of over 2,500 feet per second in parts of the turbine.

2.   To produce the driving torque, the turbine may consist of several stages each employing one row of stationary nozzle guide vanes and one row of moving blades (fig. 5-1). The number of stages depends upon the relationship between the power required
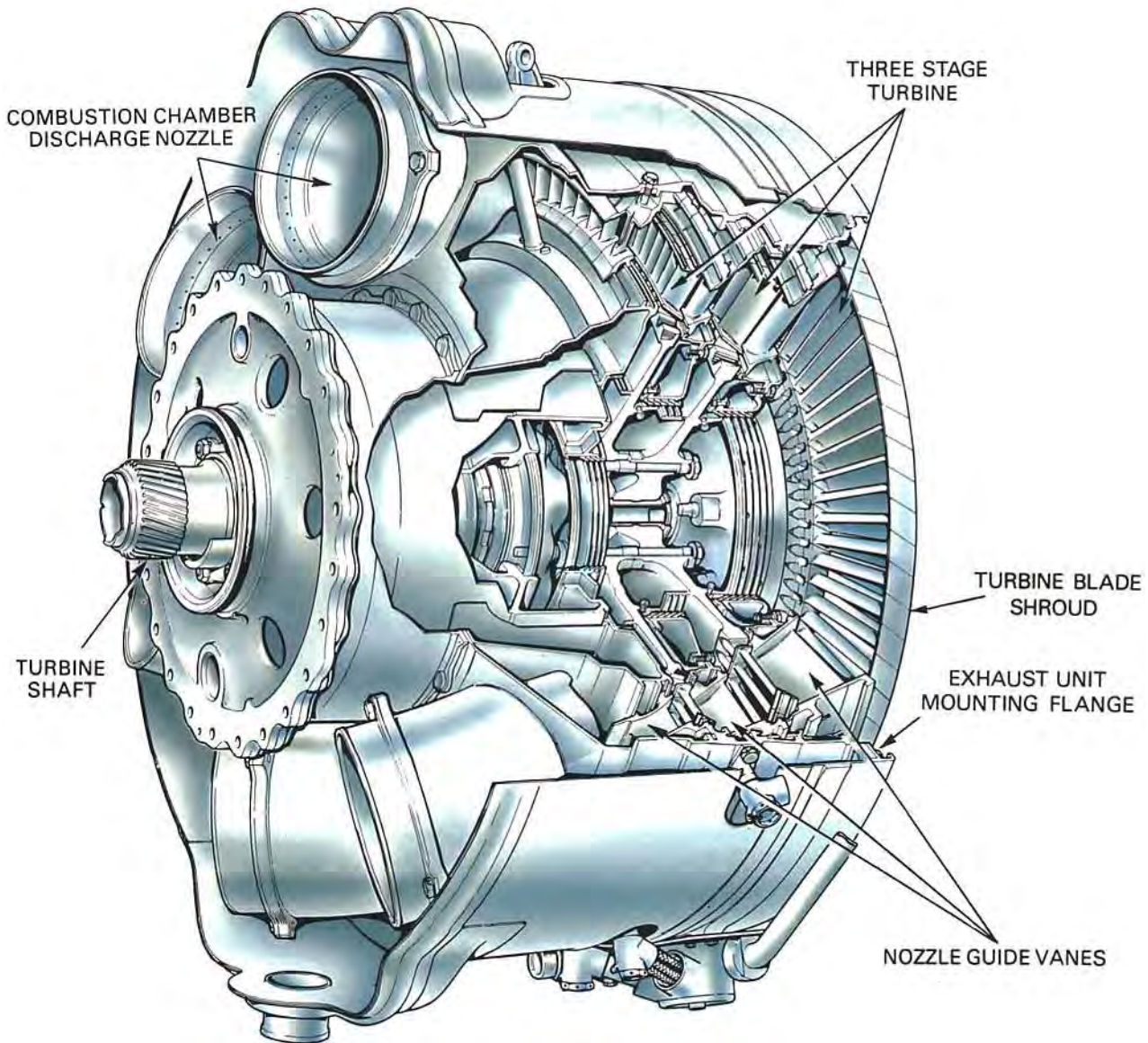
Fig. 5-1   A triple-stage turbine with single shaft system.

from the gas flow, the rotational speed at which it must be produced and the diameter of turbine permitted.

3. The number of shafts, and therefore turbines, varies with the type of engine; high compression ratio engines usually have two shafts, driving high and low pressure compressors (fig, 5-2). On high by-pass ratio fan engines that feature an intermediate pressure system, another turbine may be interposed between the high and low pressure turbines, thus forming a triple-spool system (fig, 5-3). On some engines, driving torque is derived from a free-power turbine (fig. 5-4). This method allows the turbine to run at its optimum speed because it is mechanically independent of other turbine and compressor shafts.

4. The mean blade speed of a turbine has considerable effect on the maximum efficiency possible for a given stage output. For a given output the gas velocities, deflections, and hence losses, are reduced in proportion to the square of higher mean blade speeds. Stress in the turbine disc increases as the square of the speed, therefore to maintain the same stress level at higher speed the sectional thickness, hence the weight, must be increased disproportionately. For this reason, the final design is a compromise between efficiency and weight. Engines operating at higher turbine inlet temperatures are thermally more efficient and have an improved power to weight ratio. By-pass engines have a better propulsive efficiency and thus can have a smaller turbine for a given thrust.

5. The design of the nozzle guide vane and turbine blade passages is based broadly on aerodynamic considerations, and to obtain optimum efficiency, compatible with compressor and combustion design, the nozzle guide vanes and turbine blades are of a
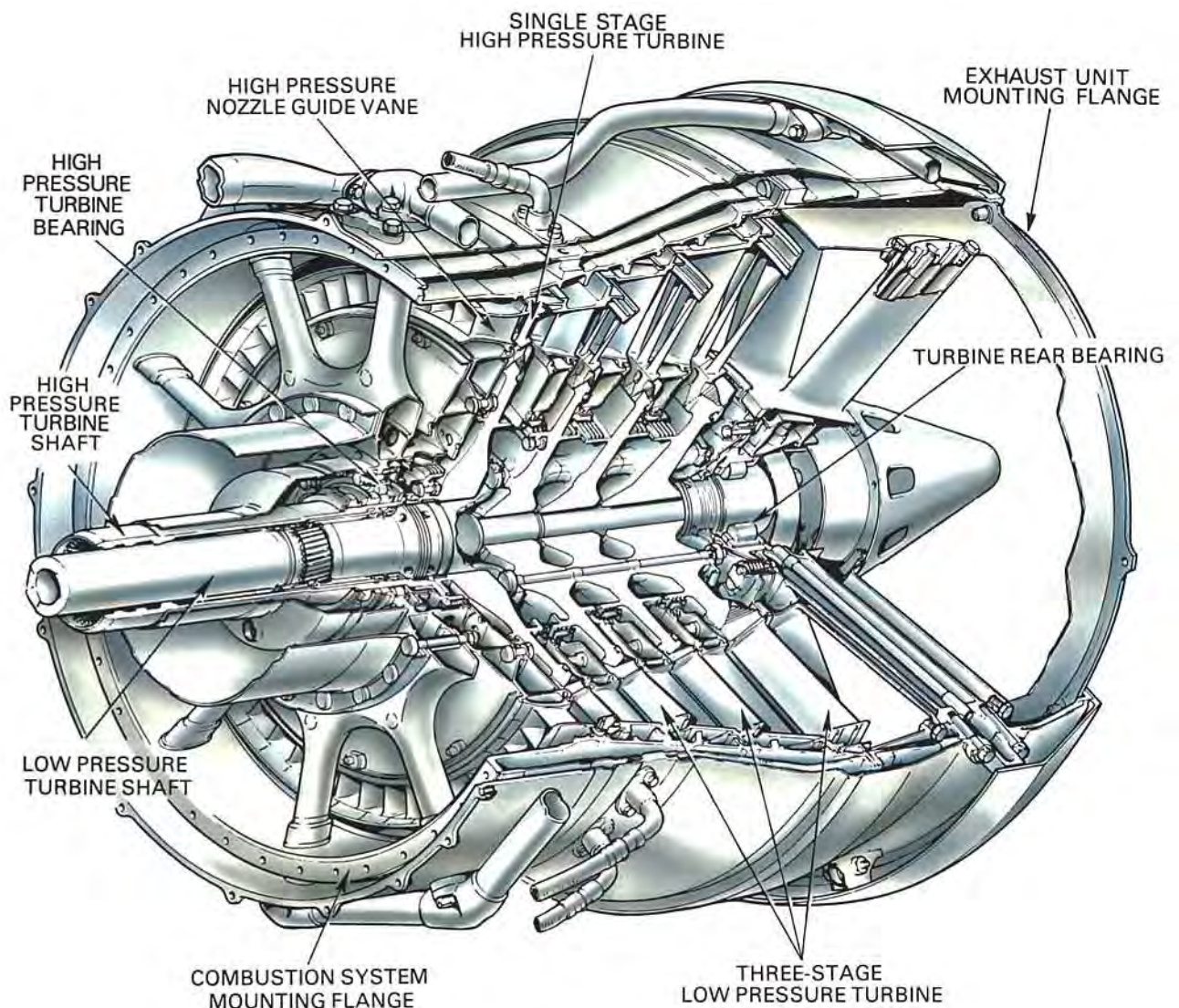
Fig. 5-2   A twin turbine and shaft arrangement.

LOW PRESSURE
TURBINE BEARING

INTERMEDIATE PRESSURE TURBINE BEARING

EXHAUST UNIT
MOUNTING FLANGE

COMBUSTION SYSTEM MOUNTING FLANGE

INTERMEDIATE PRESSURE/HIGH PRESSURE
INTERSHAFT BEARING

LOW PRESSURE
TURBINE SHAFT

INTERMEDIATE PRESSURE
TURBINE SHAFT

HIGH PRESSURE TURBINE SHAFT

HIGH PRESSURE NOZZLE GUIDE VANES

SINGLE STAGE
HIGH PRESSURE TURBINE

INTERMEDIATE PRESSURE
NOZZLE GUIDE VANES

SINGLE STAGE
INTERMEDIATE
PRESSURE TURBINE

TWO-STAGE LOW
PRESSURE TURBINE

LOW PRESSURE
NOZZLE GUIDE VANES

Fig. 5-3   A triple turbine and shaft arrangement.

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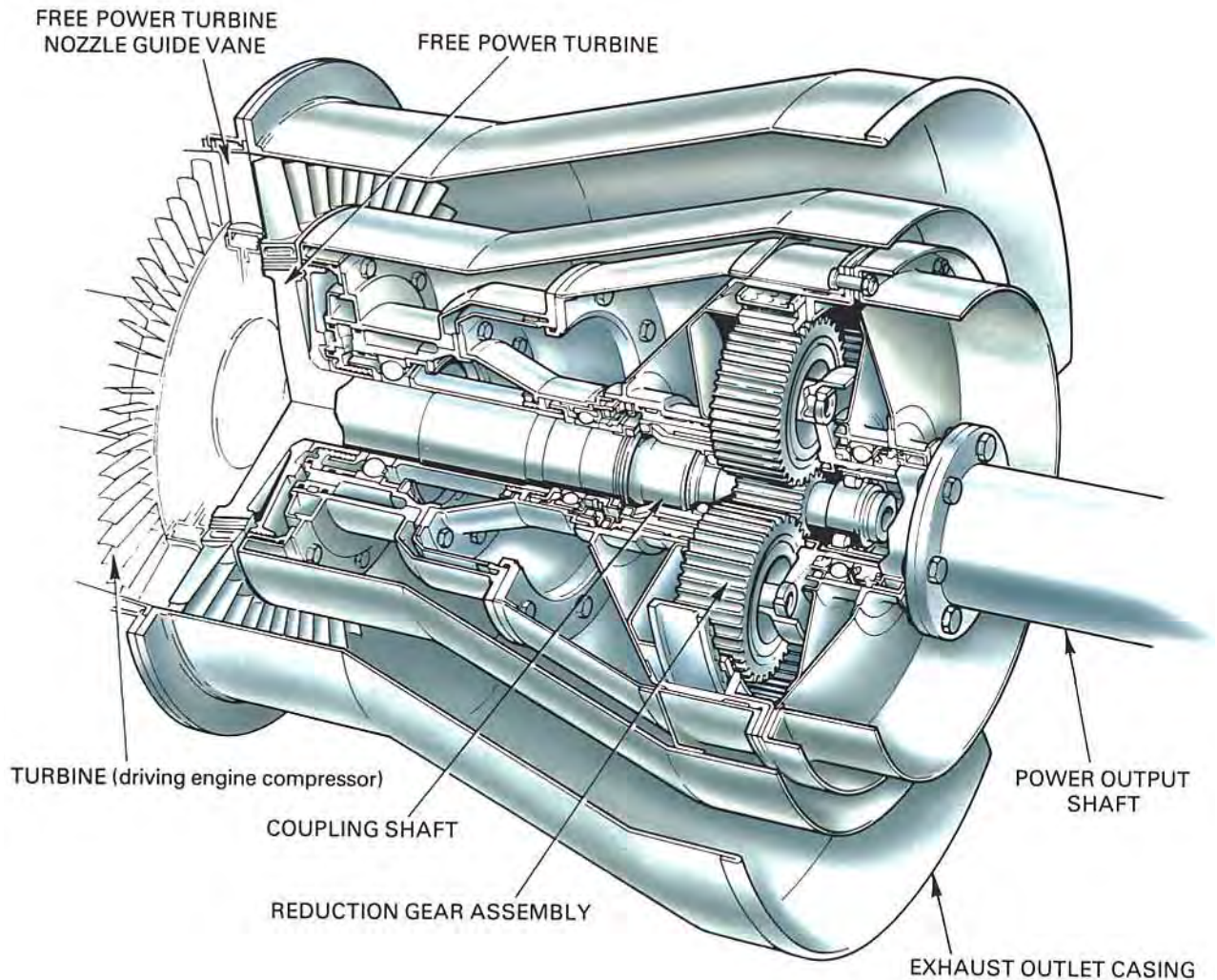
Fig. 5-4   A typical free power turbine.

basic aerofoil shape. There are three types of turbine; impulse, reaction and a combination of the two known as impulse-reaction. In the impulse type the total pressure drop across each stage occurs in the fixed nozzle guide vanes which, because of their convergent shape, increase the gas velocity whilst reducing the pressure. The gas is directed onto the turbine blades which experience an impulse force caused by the impact of the gas on the blades. In the reaction type the fixed nozzle guide vanes are designed to alter the gas flow direction without changing the pressure. The converging blade passages experience a reaction force resulting from the expansion and acceleration of the gas. Normally gas turbine engines do not use pure impulse or pure

reaction turbine blades but the impulse-reaction combination (fig. 5-5). The proportion of each principle incorporated in the design of a turbine is largely dependent on the type of engine in which the turbine is to operate, but in general it is about 50 per cent impulse and 50 per cent reaction. Impulse-type turbines are used for cartridge and air starters (Part 11).

ENERGY TRANSFER FROM GAS FLOW TO TURBINE

6.   From the description contained in para. 1, it will be seen that the turbine depends for its operation on the transfer of energy between the combustion

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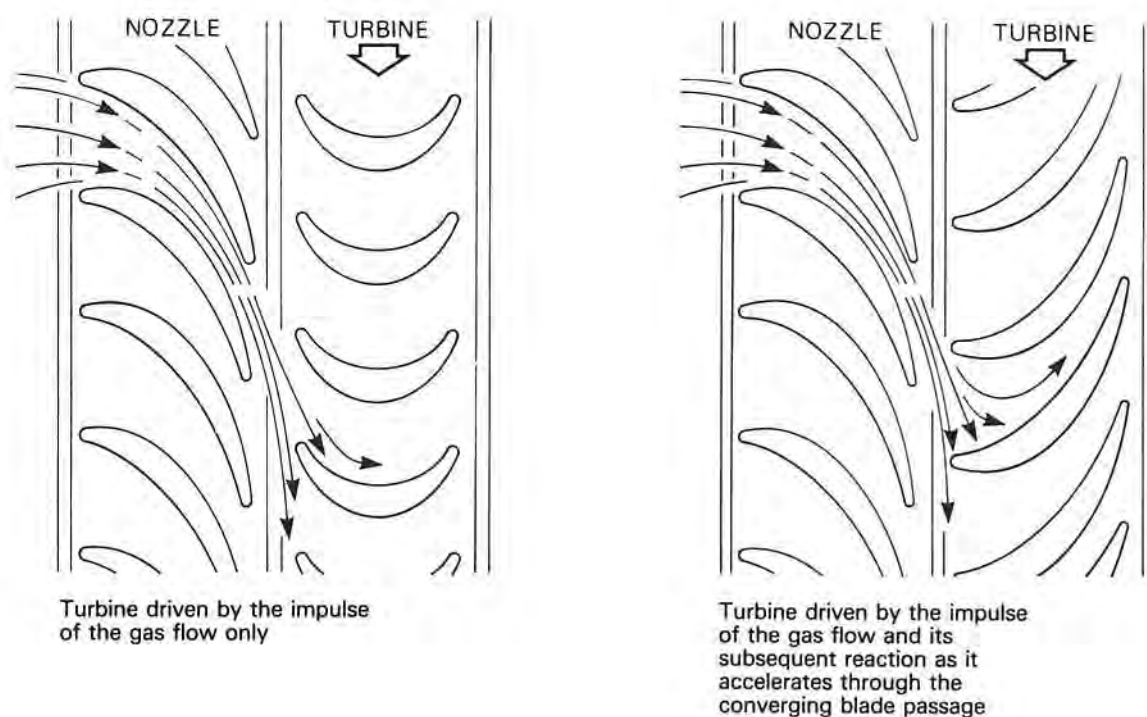
Fig. 5-5   Comparison between a pure Impulse turbine and an impulse/reaction turbine.

gases and the turbine. This transfer is never 100 per cent because of thermodynamic and mechanical losses, (para. 11).

7.when the gas is expanded by the combustion process (Part 4), it forces its way into the discharge nozzles of the turbine where, because of their convergent shape, it is accelerated to about the speed of sound which, at the gas temperature, is about 2,500 feet per second. At the same time the gas flow is given a 'spin' or 'whirl' in the direction of rotation of the turbine blades by the nozzle guide vanes. On impact with the blades and during the subsequent reaction through the blades, energy is absorbed, causing the turbine to rotate at high speed and so provide the power for driving the turbine shaft and compressor.

8. The torque or turning power applied to the turbine is governed by the rate of gas flow and the energy change of the gas between the inlet and the outlet of the turbine blades, The design of the turbine is such that the whirl will be removed from the gas stream so that the flow at exit from the turbine will be substantially 'straightened out' to give an axial flow into the exhaust system (Part 6). Excessive residual whirl reduces the efficiency of the exhaust system and also tends to produce jet pipe vibration which has a detrimental effect on the exhaust cone supports and struts.

9. It will be seen that the nozzle guide vanes and blades of the turbine are 'twisted', the blades having a stagger angle that is greater at the tip than at the root (fig. 5-6). The reason for the twist is to make the gas flow from the combustion system do equal work at all positions along the length of the blade and to ensure that the flow enters the exhaust system with a uniform axial velocity. This results in certain changes in velocity, pressure and temperature occurring through the turbine, as shown diagrammatically in fig. 5-7.

10. The 'degree of reaction' varies from root to tip, being least at the root and highest at the tip, with the mean section having the chosen value of about 50 per cent.

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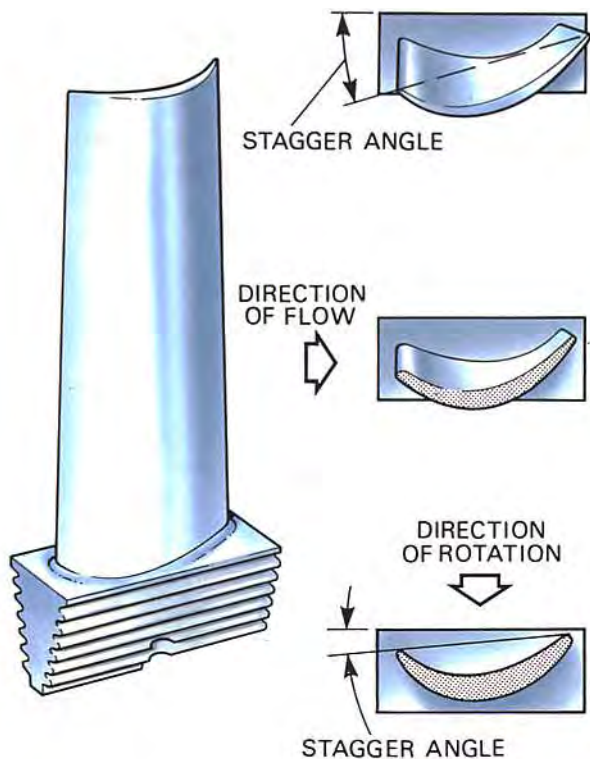
Fig. 5-6   A typical turbine blade showing twisted contour.

11.   The losses which prevent the turbine from being 100 per cent efficient are due to a number of reasons. A typical uncooled three-stage turbine would suffer a 3.5 per cent loss because of aerodynamic losses in the turbine blades. A further 4.5 per cent loss would be incurred by aerodynamic losses in the nozzle guide vanes, gas leakage over the turbine blade tips and exhaust system losses; these losses are of approximately equal proportions. The total losses result in an overall efficiency of approximately 92 per cent.

CONSTRUCTION

12.   The basic components of the turbine are the combustion discharge nozzles, the nozzle guide vanes, the turbine discs and the turbine blades. The rotating assembly is carried on bearings mounted in the turbine casing and the turbine shaft may be common to the compressor shaft or connected to it by a self-aligning coupling.

Nozzle guide vanes

13.   The nozzle guide vanes are of an aerofoil shape with the passage between adjacent vanes forming a convergent duct. The vanes are located (fig. 5-8) in the turbine casing in a manner that allows for expansion.
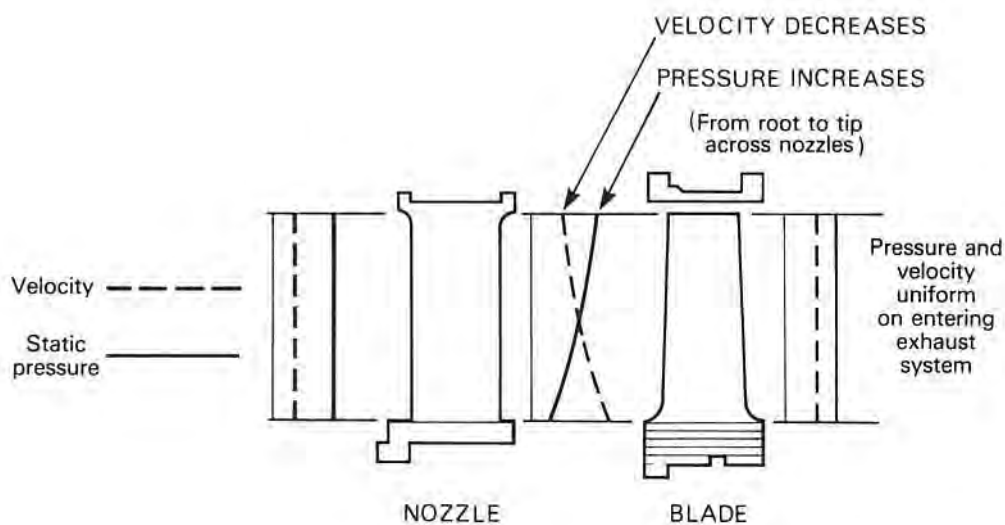
Fig. 5-7   Gas flow pattern through nozzle and blade.
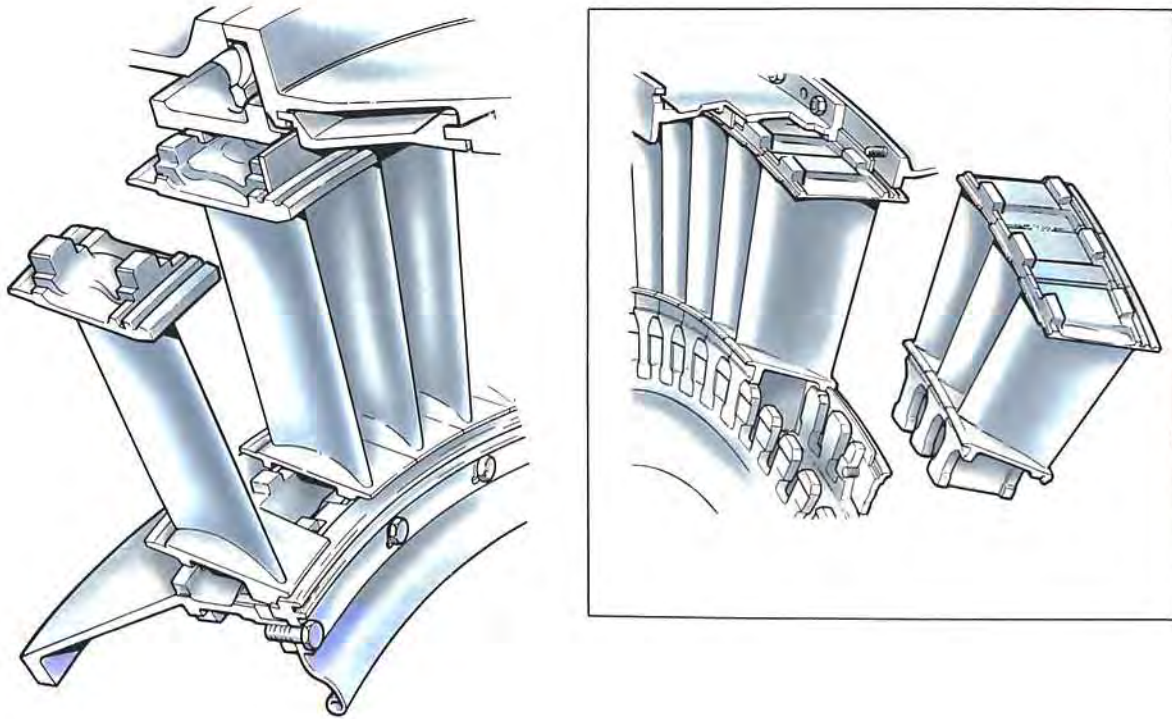
Fig. 5-8   Typical nozzle guide vanes showing their shape and location.

14.   The nozzle guide vanes are usually of hollow form and may be cooled by passing compressor delivery air through them to reduce the effects of high thermal stresses and gas loads. For details of turbine cooling, reference should be made to Part 9.

15.   Turbine discs are usually manufactured from a machined forging with an integral shaft or with a flange onto which the shaft may be bolted. The disc also has, around its perimeter, provision for the attachment of the turbine blades.

16.   To limit the effect of heat conduction from the turbine blades to the disc a flow of cooling air is passed across both sides of each disc (Part 9).

Turbine blades

17.   The turbine blades are of an aerofoil shape, designed to provide passages between adjacent blades that give a steady acceleration of the flow up to the 'throat', where the area is smallest and the velocity reaches that required at exit to produce the required degree of reaction (para. 5).

18.   The actual area of each blade cross-section is fixed by the permitted stress in the material used and by the size of any holes which may be required for cooling purposes (Part 9). High efficiency demands thin trailing edges to the sections, but a compromise has to be made so as to prevent the blades cracking due to the temperature changes during engine operation.

19.   The method of attaching the blades to the turbine disc is of considerable importance, since the stress in the disc around the fixing or in the blade root has an important bearing on the limiting rim speed. The blades on the early Whittle engine were attached by the de Laval bulb root fixing, but this design was soon superseded by the 'fir-tree' fixing that is now used in the majority of gas turbine engines. This type of fixing involves very accurate machining to ensure that the loading is shared by all

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the serrations. The blade is free in the serrations when the turbine is stationary and is stiffened in the root by centrifugal loading when the turbine is rotating. Various methods of blade attachment are shown in fig. 5-9; however, the B.M.W. hollow blade and the de Laval bulb root types are not now generally used on gas turbine engines.

20. A gap exists between the blade tips and casing, which varies in size due to the different rates of expansion and contraction. To reduce the loss of efficiency through gas leakage across the blade tips, a shroud is often fitted as shown in fig. 5-1. This is made up by a small segment at the tip of each blade which forms a peripheral ring around the blade tips. An abradable lining in the casing may also be used to reduce gas leakage as discussed in Part 9. Active Clearance Control (A.C.C.) is a more effective method of maintaining minimum tip clearance throughout the flight cycle. Air from the compressor is used to cool the turbine casing and when used with shroudless turbine blades, enables higher tempera-tures and speeds to be used.

Contra-rotating turbine
21. Fig. 5-10 shows a twelve stage contra-rotating free power turbine driving a contra-rotating rear fan. This design has only one row of static nozzle guide vanes. The remaining nozzle guide vanes are, in effect, turbine blades attached to a rotating casing which revolves in the opposite direction to a rotating drum. Since all but one aerofoil row extracts energy from the gas stream, contra-rotating turbines are capable of operating at much higher stage loadings than conventional turbines, making them attractive for direct drive applications.

Dual alloy discs
22. Very high stresses are imposed on the blade root fixing of high work rate turbines, which make conventional methods of blade attachment impractical. A dual alloy disc, or 'blisk' as shown in fig. 5-11, has a ring of cast turbine blades bonded to the disc. This type of turbine is suitable for small high power helicopter engines.

COMPRESSOR-TURBINE MATCHING

23. The flow characteristics of the turbine must be very carefully matched with those of the compressor to obtain the maximum efficiency and performance of the engine. If, for example, the nozzle guide vanes allowed too low a maximum flow, then a back pressure would build up causing the compressor to surge (Part 3); too high a flow would cause the compressor to choke. In either condition a loss of efficiency would very rapidly occur.

MATERIALS

24. Among the obstacles in the way of using higher turbine entry temperatures have always been the effects of these temperatures on the nozzle guide vanes and turbine blades, The high speed of rotation which imparts tensile stress to the turbine disc and blades is also a limiting factor.

FIR TREE ROOT
(with locking plate)

FIR TREE ROOT
(with shank seals)

DE LAVAL BULB ROOT
(with locking screw)

B.M.W. HOLLOW BLADE
(with retaining pins)
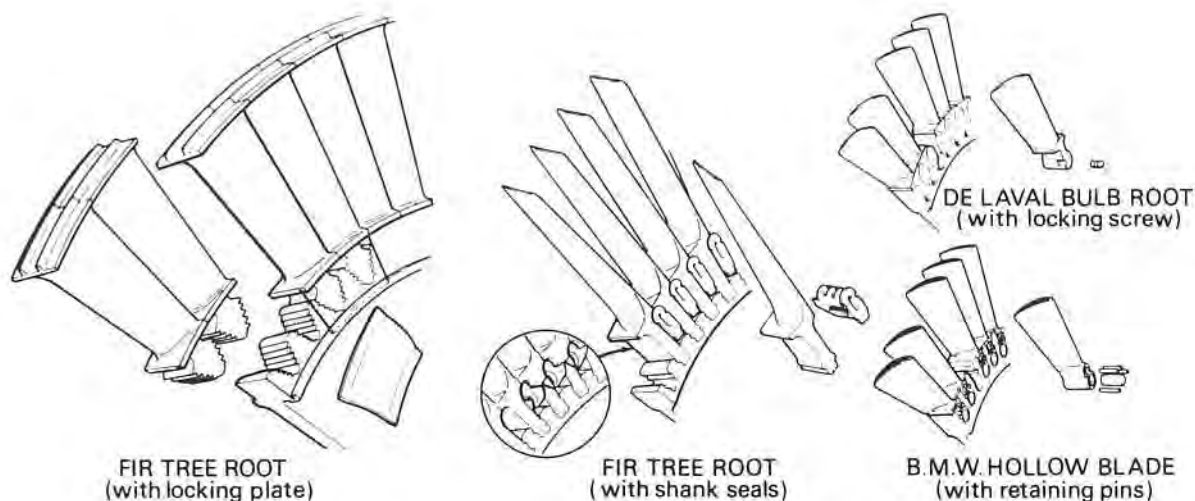
Fig. 5-9   Various methods of attaching blades to turbine discs.
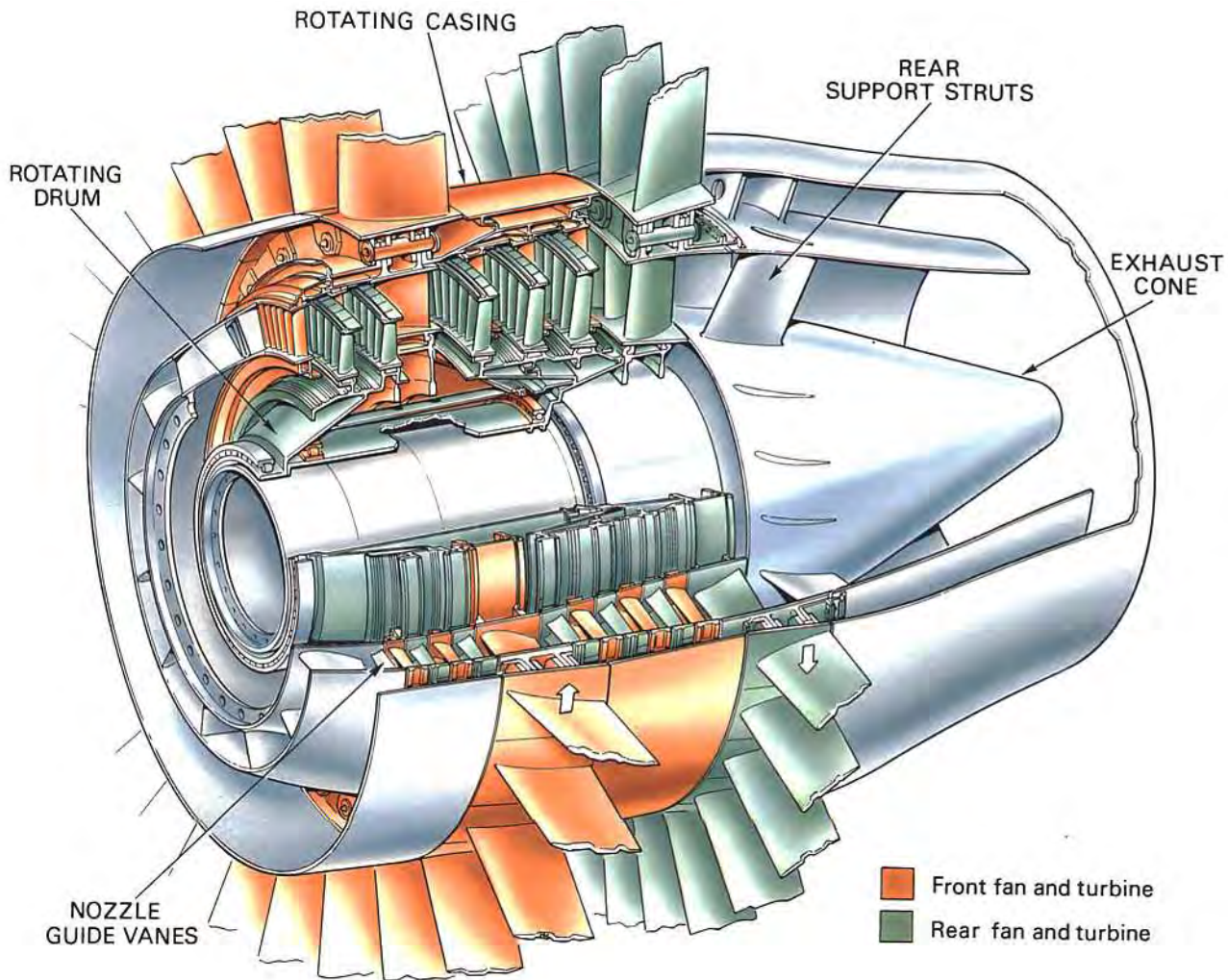
Fig. 5-10   Free power contra-rotating turbine.

CAST BLADE RING
DIFFUSION BOND
POWDER DISC
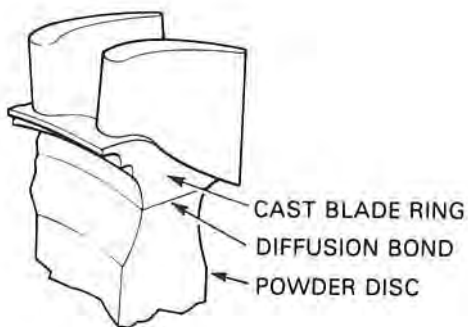
Fig. 5-11   Section through a dual alloy disc.

Nozzle guide vanes

25.   Due to their static condition. the nozzle guide vanes do not endure the same rotational stresses as the turbine blades. Therefore, heat resistance is the property most required. Nickel alloys are used, although cooling is required to prevent melting. Ceramic coatings can enhance the heat resisting properties and, for the same set of conditions, reduce the amount of cooling air required, thus improving engine efficiency.

Turbine discs

26.   A turbine disc has to rotate at high speed in a relatively cool environment and is subjected to large rotational stresses. The limiting factor which affects the useful disc life is its resistance to fatigue cracking.

CONVENTIONALLY CAST TURBINE BLADE

Good mechanical properties
in all directions

Equi-axed
crystal structure

Excellent mechanical properties
in longitudinal axis and
improved heat resistance

Improved mechanical
properties in
longitudinal axis

Columnar
crystal
structure

DIRECTIONALLY SOLIDIFIED TURBINE BLADE
(D.S. blade)

SINGLE CRYSTAL TURBINE BLADE

Fig. 5-11   Section through a dual alloy disc.

27. In the past, turbine discs have been made in ferritic and austenitic steels but nickel based alloys are currently used. Increasing the alloying elements in nickel extend the life limits of a disc by increasing fatigue resistance. Alternatively, expensive powder metallurgy discs, which offer an additional 10% in strength, allow faster rotational speeds to be achieved.

Turbine blades

28. A brief mention of some of the points to be considered in connection with turbine blade design will give an idea of the importance of the correct choice of blade material. The blades, while glowing red-hot, must be strong enough to carry the centrifugal loads due to rotation at high speed. A small turbine blade weighing only two ounces may exert a load of over two tons at top speed and it must withstand the high bending loads applied by the gas to produce the many thousands of turbine horse-power necessary to drive the compressor. Turbine blades must also be resistant to fatigue and thermal shock, so that they will not fail under the influence of high frequency fluctuations in the gas conditions, and they must also be resistant to corrosion and oxidization. In spite of all these demands, the blades must be made in a material that can be accurately formed and machined by current manufacturing methods.

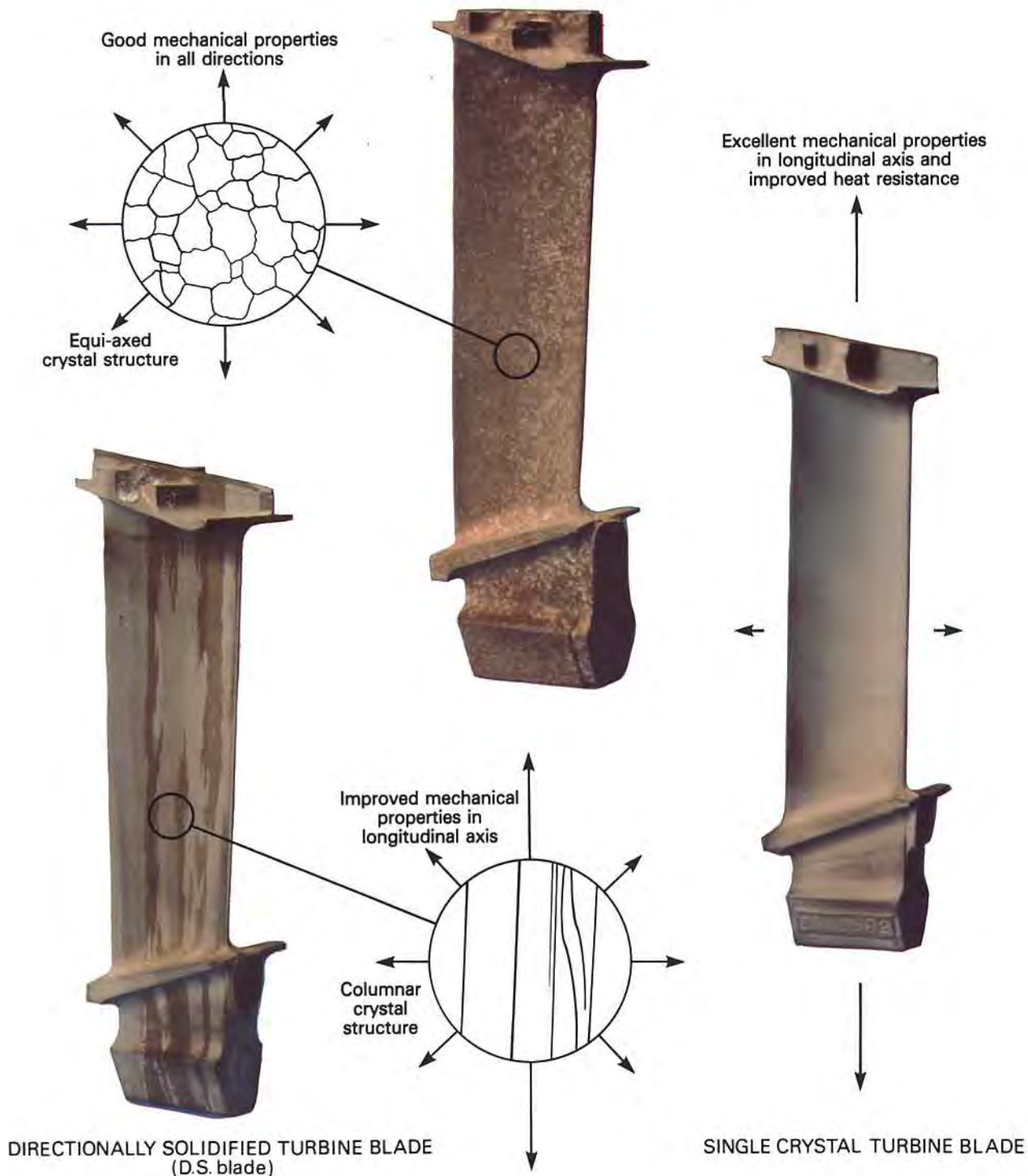29. From the foregoing, it follows that for a particular blade material and an acceptable safe life there is an associated maximum permissible turbine entry temperature and a corresponding maximum engine power. It is not surprising, therefore, that met-allurgists and designers are constantly searching for better turbine blade materials and improved methods of blade cooling.

30. Over a period of operational time the turbine blades slowly grow in length. This phenomenon is known as 'creep' and there is a finite useful life limit before failure occurs.

31. The early materials used were high temperature steel forgings, but these were rapidly replaced by cast nickel base alloys which give better creep and fatigue properties.

32. Close examination of a conventional turbine blade reveals a myriad of crystals that lie in all directions (equi-axed). Improved service life can be obtained by aligning the crystals to form columns along the blade length, produced by a method known as 'Directional Solidification'. A further advance of this technique is to make the blade out of a single
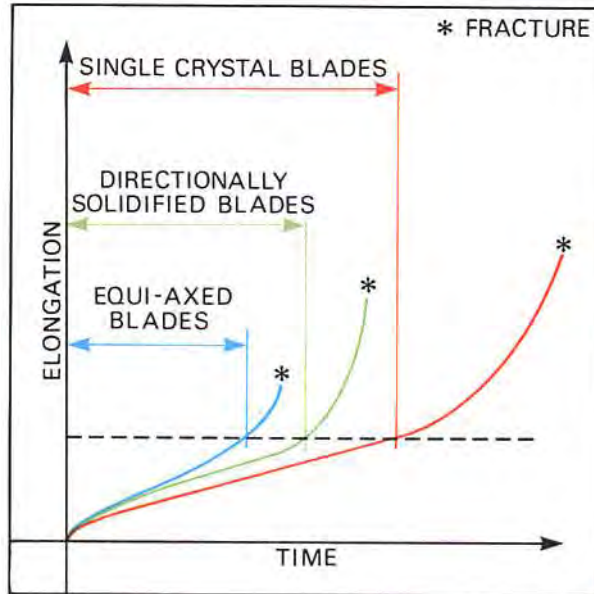
Fig. 5-13 Comparison of turbine blade life properties.
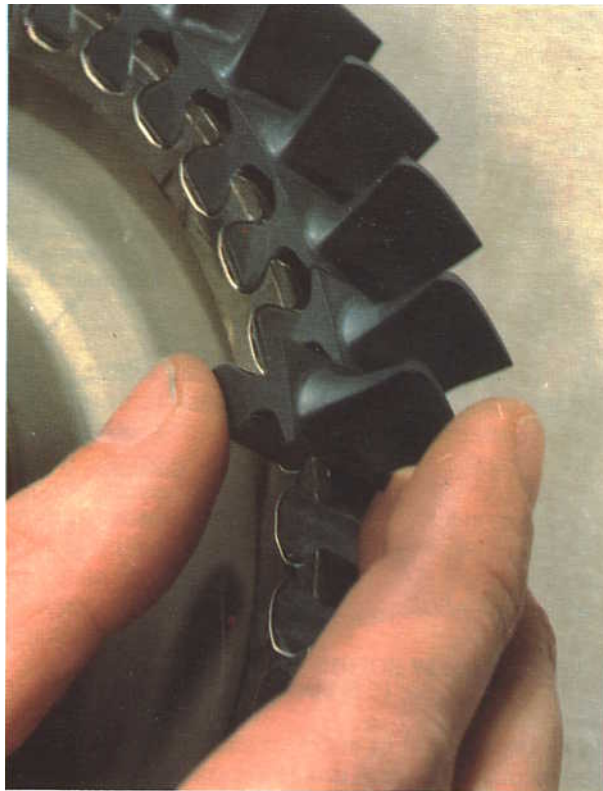
Fig. 5-14 Ceramic turbine blades.

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crystal, Examples of these structures are shown in fig. 5-12. Each method extends the useful creep life of the blade (fig. 5-13) and in the case of the single crystal blade, the operating temperature can be substantially increased.

33.   A non-metal based turbine blade can be manufactured from reinforced ceramics. Their initial production application is likely to be for small high speed turbines which have very high turbine entry temperatures. An example of a ceramic blade is shown in fig. 5-14.

BALANCING

34.   The balancing of a turbine is an extremely important operation in its assembly. In view of the high rotational speeds and the mass of materials, any unbalance could seriously affect the rotating assembly bearings and engine operation. Balancing is effected on a special balancing machine, the principles of which are briefly described in Part 25.

Rolls-Royce RB211-535E4
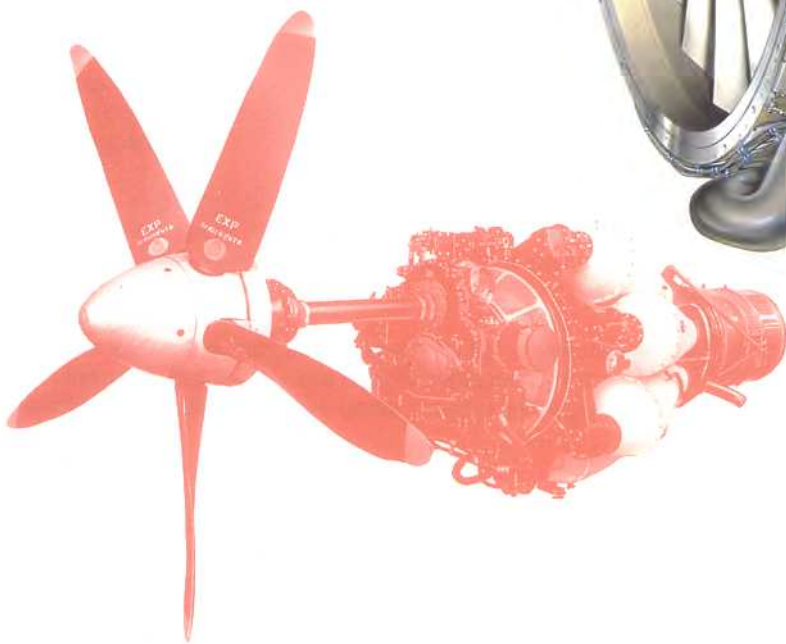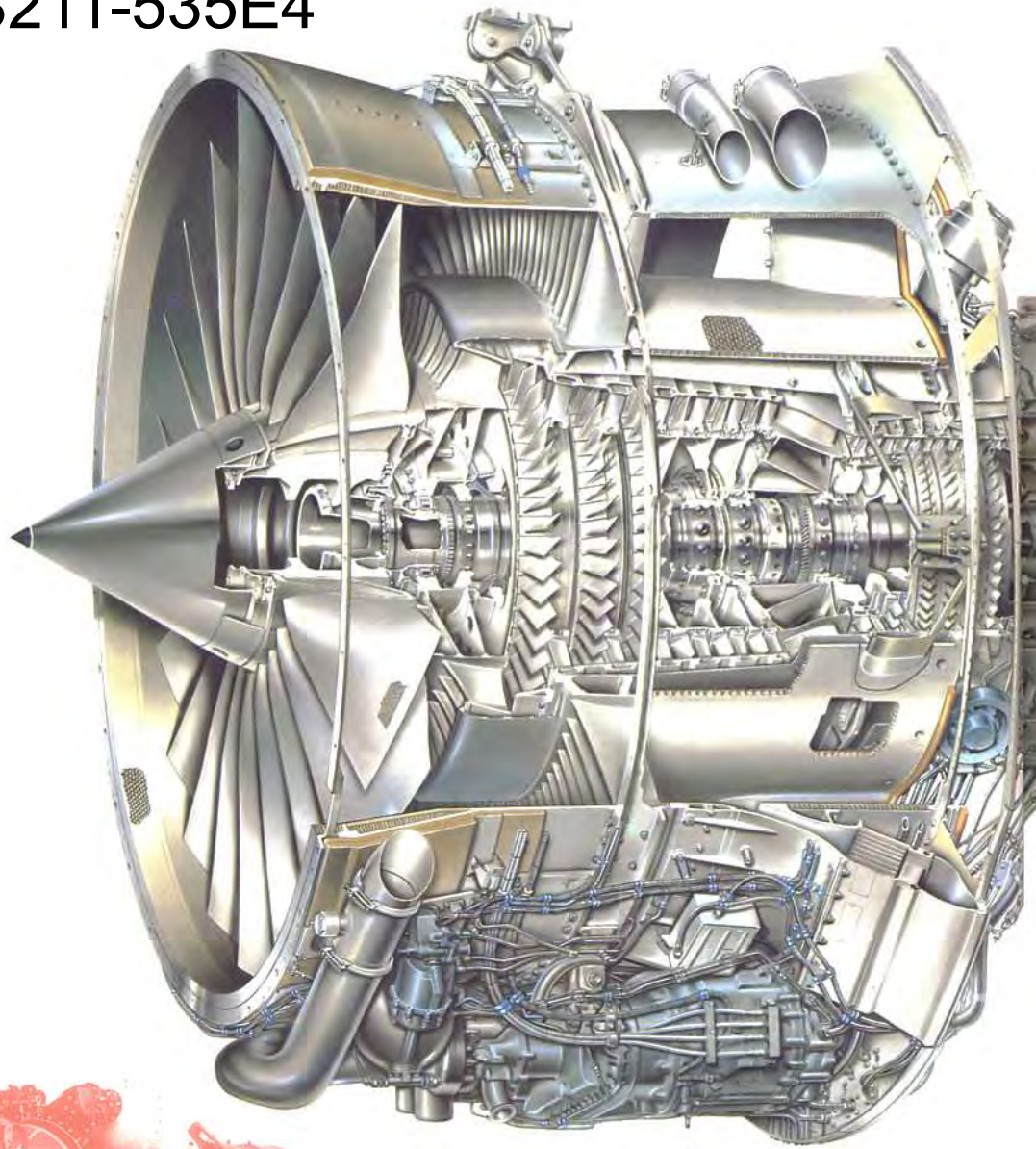
Rolls-Royce RB50 Trent

Late in 1943 the decision was taken at Rolls-Royce to build a turbo-prop for aircraft speeds of around 400 mph. The resulting engine, known as the RB50 Trent, was basically a Derwent II with a flexible quillshaft to reduction gear and propeller. On 20 September 1945 a Gloster Meteor, fitted with two Trents, became the world's first turbo-prop powered aircraft to fly.

6: Exhaust system
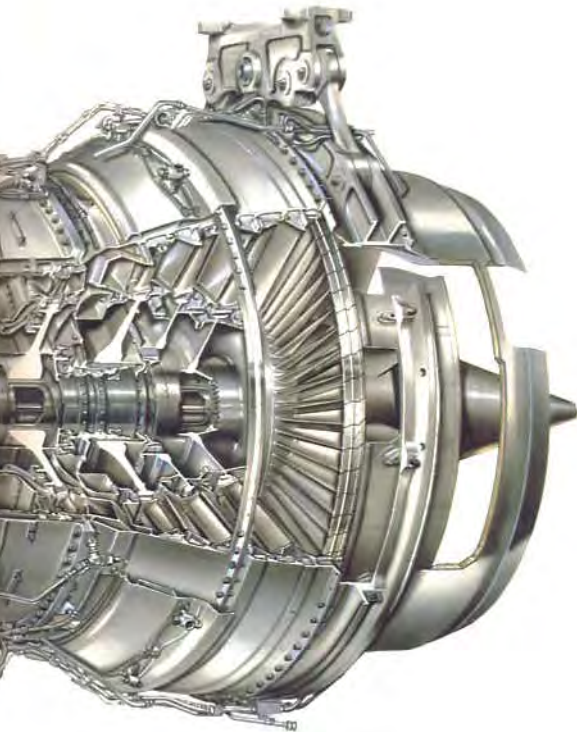
Contents

INTRODUCTION

1. Aero gas turbine engines have an exhaust system which passes the turbine discharge gases to atmosphere at a velocity, and in the required direction, to provide the resultant thrust. The velocity and pressure of the exhaust gases create the thrust in the turbo-jet engine (para. 5) but in the turbo-propeller engine only a small amount of thrust is contributed by the exhaust gases, because most of the energy has been absorbed by the turbine for driving the propeller. The design of the exhaust system therefore, exerts a considerable influence on the performance of the engine. The areas of the jet pipe and propelling or outlet nozzle affect the turbine entry temperature, the mass airflow and the velocity and pressure of the exhaust jet.

2. The temperature of the gas entering the exhaust system is between 550 and 850 deg. C. according to the type of engine and with the use of afterburning (Part 16) can be 1,500 deg. C. or higher. Therefore, it is necessary to use materials and a form of construction that will resist distortion and cracking, and prevent heat conduction to the aircraft structure.

3. A basic exhaust system is shown in fig. 6-1. The use of a thrust reverser (Part 15), noise suppressor (Part 19) and a two position propelling nozzle entails a more complicated system as shown in fig. 6-2. The low by-pass engine may also include a mixer unit (fig. 6-4) to encourage a thorough mixing of the hot and cold gas streams.
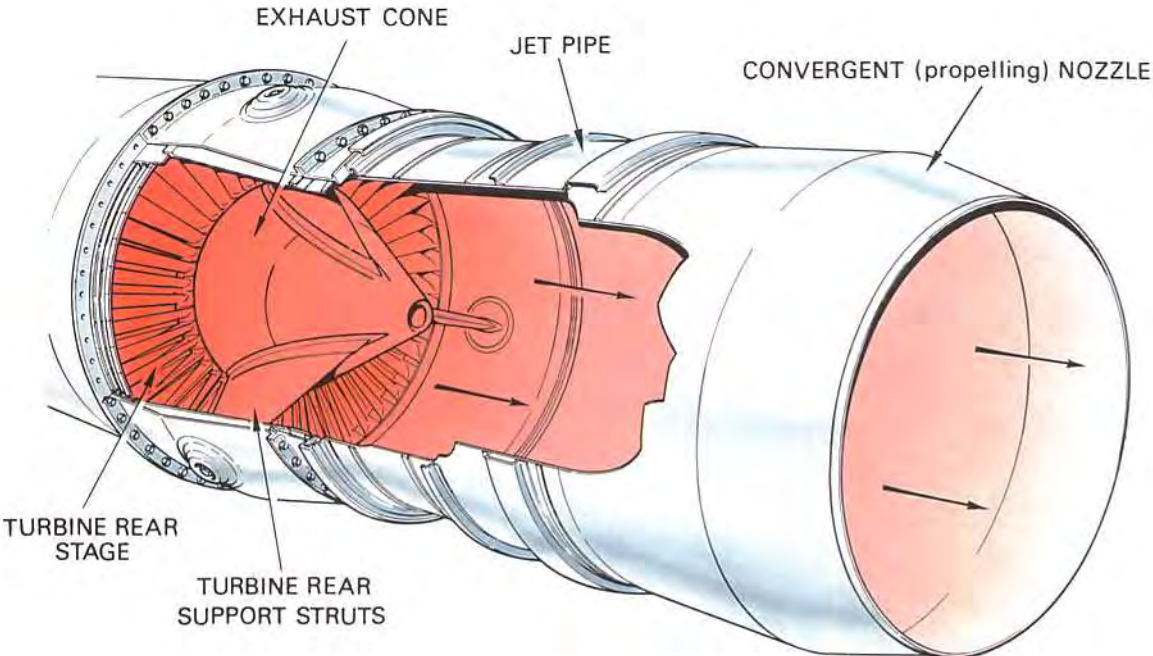
Fig. 6-1   A basic exhaust system.
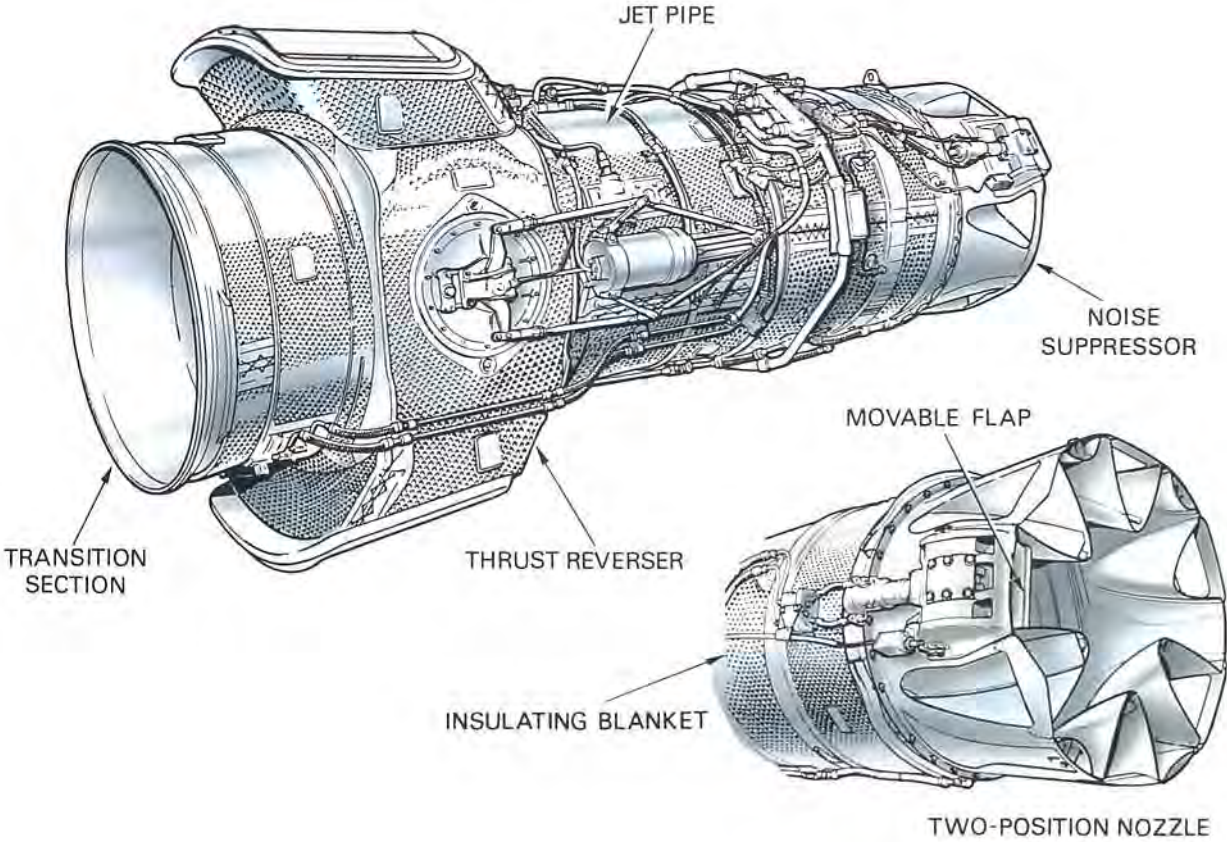
Fig. 6-2   Exhaust system with thrust reverser, noise suppressor and two position propelling nozzle.

EXHAUST GAS FLOW

4. Gas from the engine turbine enters the exhaust system at velocities from 750 to 1,200 feet per second, but, because velocities of this order produce high friction losses, the speed of flow is decreased by diffusion. This is accomplished by having an increasing passage area between the exhaust cone and the outer wall as shown in fig. 6-1. The cone also prevents the exhaust gases from flowing across the rear face of the turbine disc. It is usual to hold the velocity at the exhaust unit outlet to a Mach number of about 0.5, i.e. approximately 950 feet per second. Additional losses occur due to the residual whirl velocity in the gas stream from the turbine. To reduce these losses, the turbine rear struts in the exhaust unit are designed to straighten out the flow before the gases pass into the jet pipe.

5. The exhaust gases pass to atmosphere through the propelling nozzle, which is a convergent duct, thus increasing the gas velocity (Part 2). In a turbo-jet engine, the exit velocity of the exhaust gases is subsonic at low thrust conditions only. During most operating conditions, the exit velocity reaches the speed of sound in relation to the exhaust gas

Fig. 6-4   A low by-pass air mixer unit.

temperature and the propelling nozzle is then said to be 'choked'; that is, no further increase in velocity can be obtained unless the temperature is increased. As the upstream total pressure is increased above the value at which the propelling nozzle becomes 'choked', the static pressure of the gases at exit increases above atmospheric pressure. This pressure difference across the propelling nozzle gives what is known as 'pressure thrust' and is effective over the nozzle exit area. This is additional thrust to that obtained due to the momentum change of the gas stream (Part 20).

Fig. 6-3   Gas flow through a convergent-divergent nozzle.

6. With the convergent type of nozzle a wastage of energy occurs, since the gases leaving the exit do not expand rapidly enough to immediately achieve outside air pressure. Depending on the aircraft flight plan, some high pressure ratio engines can with advantage use a convergent-divergent nozzle to recover some of the wasted energy This nozzle utilizes the pressure energy to obtain a further increase in gas velocity and, consequently, an increase in thrust.

7. From the illustration (fig. 6-3), it will be seen that the convergent section exit now becomes the throat, with the exit proper now being at the end of the flared divergent section. When the gas enters the convergent section of the nozzle, the gas velocity increases with a corresponding fall in static pressure. The gas velocity at the throat corresponds to the local sonic velocity. As the gas leaves the restriction of the throat and flows into the divergent section, it progressively increases in velocity towards the exit. The reaction to this further increase in momentum is a pressure force acting on the inner wall of the nozzle. A component of this force acting parallel to the longitudinal axis of the nozzle produces the further increase in thrust.

EXTERNAL MIXING OF GAS STREAMS

■ Cold by-pass (fan) airflow

■ Hot exhaust gases

COMMON OR INTEGRATED EXHAUST NOZZLE

PARTIAL INTERNAL MIXING OF GAS STREAMS

Fig. 6-5   High by-pass ratio engine exhaust systems.
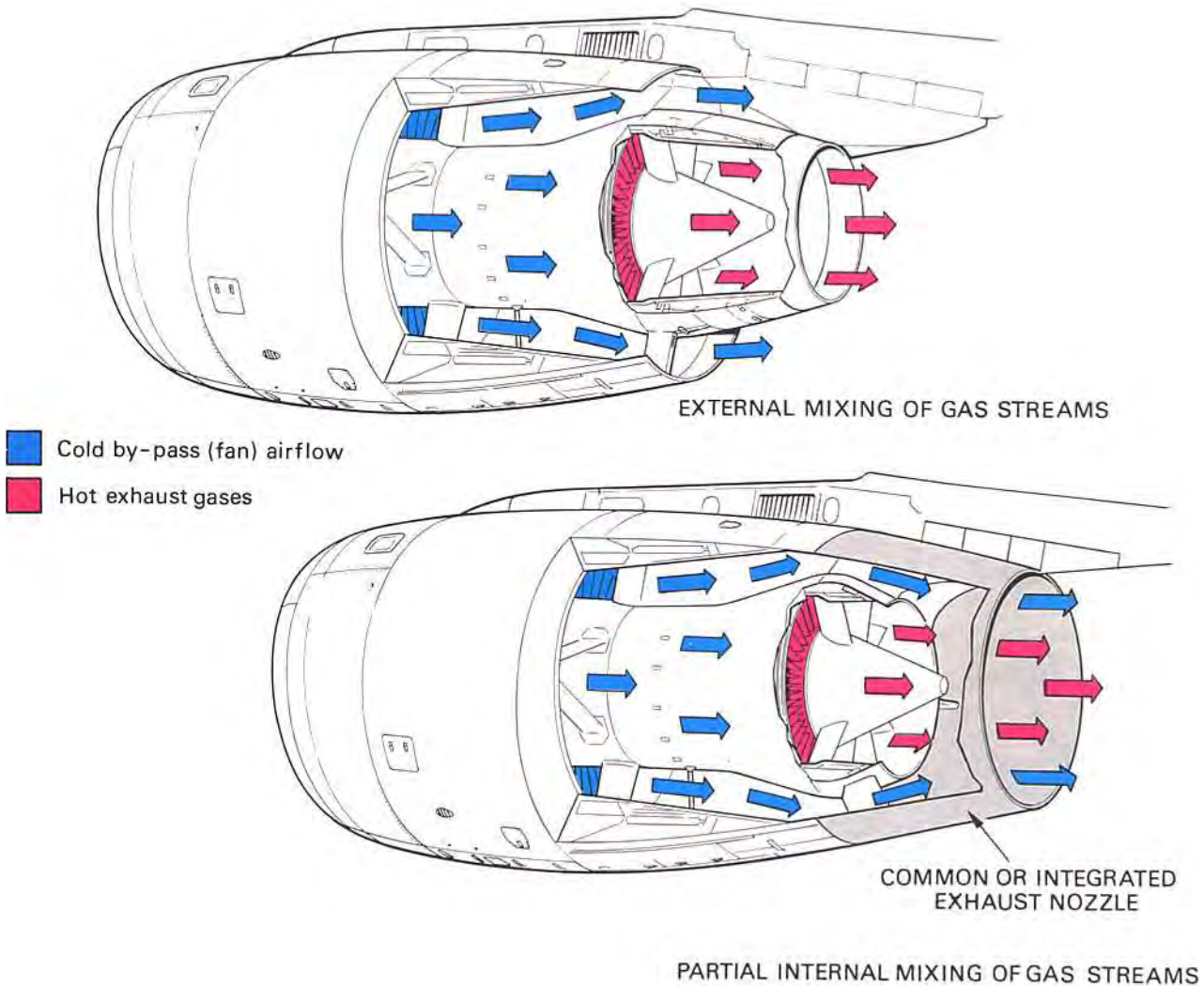
8. The propelling nozzle size is extremely important and must be designed to obtain the correct balance of pressure, temperature and thrust. With a small nozzle these values increase, but there is a possibility of the engine surging (Part 3), whereas with a large nozzle the values obtained are too low,

9. A fixed area propelling nozzle is only efficient over a narrow range of engine operating conditions. To increase this range, a variable area nozzle may be used. This type of nozzle is usually automatically controlled and is designed to maintain the correct balance of pressure and temperature at all operating conditions. In practice, this system is seldom used as the performance gain is offset by the increase in weight. However, with afterburning a variable area nozzle is necessary and is described in Part 16.

10. The by-pass engine has two gas streams to eject to atmosphere, the cool by-pass airflow and the hot turbine discharge gases.

11. In a low by-pass ratio engine, the two flows are combined by a mixer unit (fig. 6-4) which allows the by-pass air to flow into the turbine exhaust gas flow in a manner that ensures thorough mixing of the two streams.

12. In high by-pass ratio engines, the two streams are usually exhausted separately. The hot and cold nozzles are co-axial and the area of each nozzle is designed to obtain maximum efficiency. However, an improvement can be made by combining the two gas flows within a common, or integrated, nozzle assembly. This partially mixes the gas flows prior to ejection to atmosphere. An example of both types of high by-pass exhaust system is shown in fig, 6-5.

CONSTRUCTION AND MATERIALS

13. The exhaust system must be capable of withstanding the high gas temperatures and is therefore manufactured from nickel or titanium. It is also necessary to prevent any heat being transferred to the surrounding aircraft structure. This is achieved by passing ventilating air around the jet pipe, or by lagging the section of the exhaust system with an insulating blanket (fig. 6-6). Each blanket has an inner layer of fibrous insulating material contained by
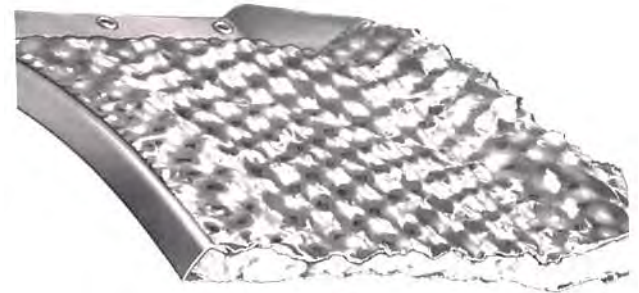
Fig. 6-6   An insulating blanket.

an outer skin of thin stainless steel, which is dimpled to increase its strength. In addition, acoustically absorbent materials are sometimes applied to the exhaust system to reduce engine noise (Part 19).

14. When the gas temperature is very high (for example, when afterburning is employed), the complete jet pipe is usually of double-wall construction (Part 16) with an annular space between the two walls. The hot gases leaving the propelling nozzle induce, by ejector action, a flow of air through the annular space of the engine nacelle. This flow of air cools the inner wall of the jet pipe and acts as an insulating blanket by reducing the transfer of heat from the inner to the outer wall.

15. The cone and streamline fairings in the exhaust unit are subjected to the pressure of the exhaust gases; therefore, to prevent any distortion, vent holes are provided to obtain a pressure balance.

16. The mixer unit used in low by-pass ratio engines consists of a number of chutes through which the bypass air flows into the exhaust gases. A bonded honeycomb structure is used for the integrated nozzle assembly of high by-pass ratio engines to give lightweight strength to this large component.

17. Due to the wide variations of temperature to which the exhaust system is subjected, it must be mounted and have its sections joined together in such a manner as to allow for expansion and contraction without distortion or damage.

Rolls-Royce Gnome
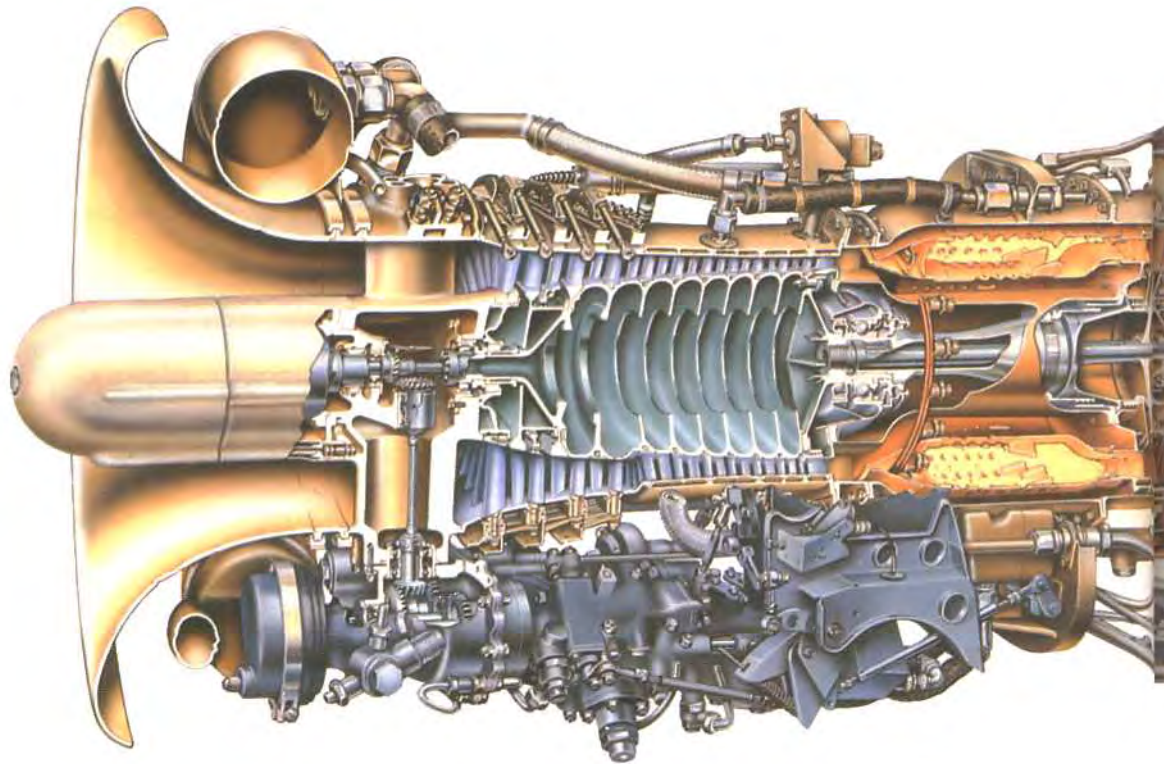
De Havilland H2 Ghost

The Ghost was designed as a larger and more powerful version of the Goblin. After running for the first time on 2 September 1945 the engine was cleared for flight in the outer nacelles of an Avro Lancastrian at 4000 lb thrust. The Ghost later went into production at 5000 lb thrust to power the de Havilland Comet 1 airliner and Venom fighter.

7: Accessory drives
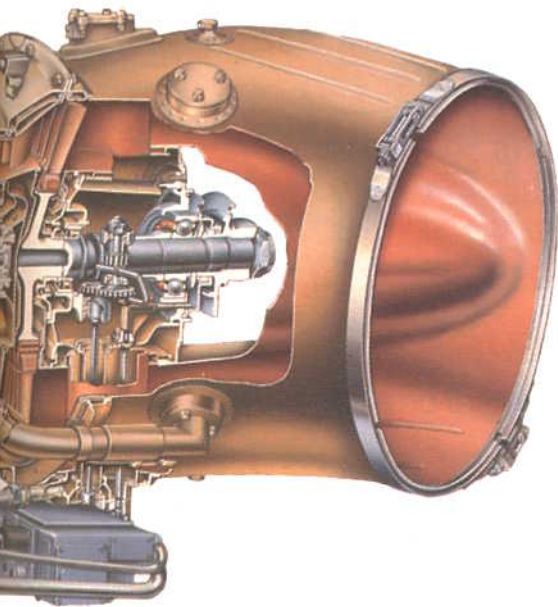
Contents

INTRODUCTION

1. Accessory units provide the power for aircraft hydraulic, pneumatic and electrical systems in addition to providing various pumps and control systems for efficient engine operation. The high level of dependence upon these units requires an extremely reliable drive system.

2. The drive for the accessory units is typically taken from a rotating engine shaft, via an internal gearbox, to an external gearbox which provides a mount for the accessories and distributes the appropriate geared drive to each accessory unit. A starter may also be fitted to provide an input torque to the engine. An accessory drive system on a high by-pass engine takes between 400 and 500 horsepower from the engine.

GEARBOXES AND DRIVES

Internal gearbox
3. The location of the internal gearbox within the core of an engine is dictated by the difficulties of bringing a driveshaft radially outwards and the space available within the engine core.

4. Thermal fatigue and a reduction in engine performance, due to the radial driveshaft disturbing the gasflow, create greater problems within the turbine area than the compressor area. For any given engine, which incorporates an axial-flow compressor, the turbine area is smaller than that containing the compressor and therefore makes it physically easier to mount the gearbox within the compressor section. Centrifugal compressor engines can have limited available space, so the internal gearbox may be located within a static nose cone or, in the case of a turbo-propeller engine, behind the propeller reduction gear as shown in fig. 7-1.

5. On multi-shaft engines, the choice of which compressor shaft is used to drive the internal gearbox is primarily dependent upon the ease of

STATIC NOSE CONE
INTERNAL GEARBOX
HOUSING

EXTERNAL GEARBOX

INTERNAL GEARBOX LOCATED
WITHIN THE NOSE CONE HOUSING

GEAR TRAIN

ACCESSORY
UNIT

EXTERNAL GEARBOX

ACCESSORY
UNIT

DIRECT DRIVE TO ACCESSORY
UNITS AND GEAR TRAIN
DRIVE TO EXTERNAL GEARBOX

FAIRING

INTERNAL GEARBOX

INTERMEDIATE GEARBOX

EXTERNAL GEARBOX

SINGLE DRIVE WITH
INTERMEDIATE GEARBOX

LOW SPEED
EXTERNAL GEARBOX

HIGH SPEED
EXTERNAL GEARBOX

AUXILIARY
GEARBOX
DRIVE

AUXILIARY GEARBOX

TWO-SHAFT DRIVE
WITH AUXILIARY GEARBOX

Fig. 7-1   Mechanical arrangement of accessory drives.

DIRECT DRIVE

STUB SHAFT DRIVE

IDLER GEAR DRIVE
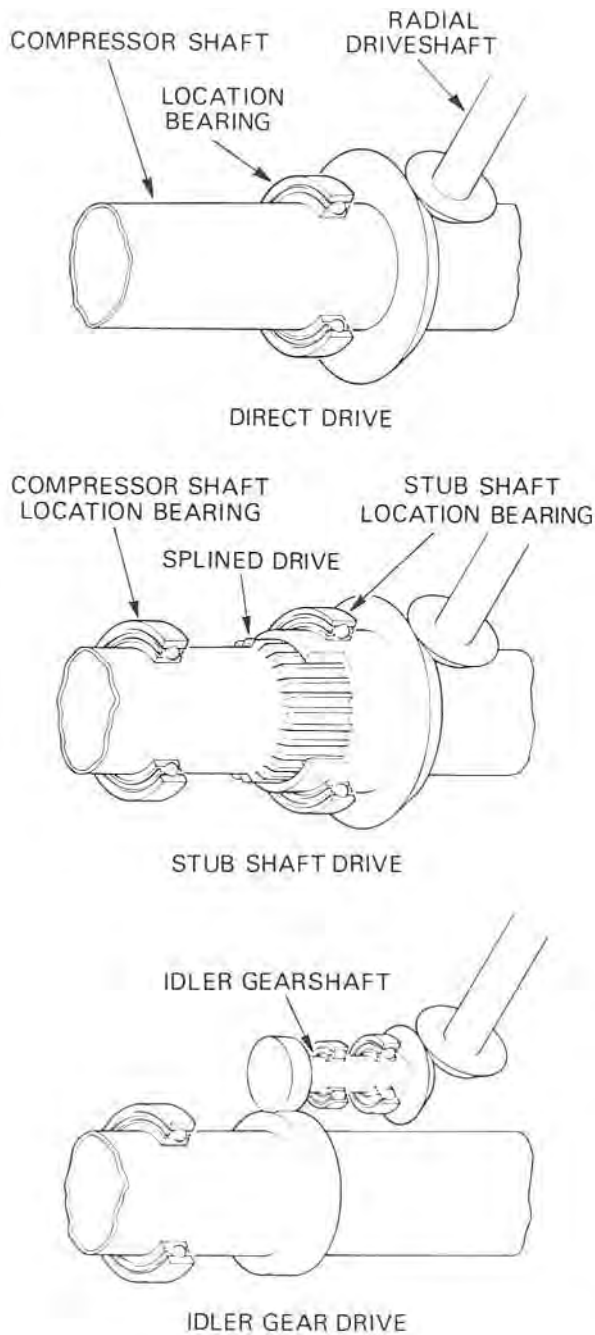
Fig. 7-2   Mechanical arrangement of internal gearboxes.

engine starting. This is achieved by rotating the compressor shaft, usually via an input torque from the external gearbox (Part 11). In practice the high pressure system is invariably rotated in order to generate an airflow through the engine and the high pressure compressor shaft is therefore coupled to the internal gearbox.

6. To minimize unwanted movement between the compressor shaft bevel gear and radial driveshaft bevel gear, caused by axial movement of the compressor shaft, the drive is taken by one of three basic methods (fig. 7-2). The least number of components is used when the compressor shaft bevel gear is mounted as close to the compressor shaft location bearing as possible, but a small amount of movement has to be accommodated within the meshing of the bevel gears. Alternatively, the compressor shaft bevel gear may be mounted on a stub shaft which has its own location bearing. The stub shaft is splined onto the compressor shaft which allows axial movement without affecting the bevel gear mesh. A more complex system utilizes an idler gear which meshes with the compressor shaft via straight spur gears, accommodating the axial movement, and drives the radial driveshaft via a bevel gear arrangement. The latter method was widely employed on early engines to overcome gear engagement difficulties at high speed.

7. To spread the load of driving accessory units, some engines take a second drive from the slower rotating low pressure shaft to a second external gearbox (fig. 7-1). This also has the advantage of locating the accessory units in two groups, thus overcoming the possibility of limited external space on the engine. When this method is used, an attempt is made to group the accessory units specific to the engine onto the high pressure system, since that is the first shaft to rotate, and the aircraft accessory units are driven by the low pressure system. A typical internal gearbox showing how both drives are taken is shown in fig. 7-3.

Radial driveshaft
8. The purpose of a radial driveshaft is to transmit the drive from the internal gearbox to an accessory unit or the external gearbox. It also serves to transmit the high torque from the starter to rotate the high pressure system for engine starting purposes. The driveshaft may be direct drive or via an intermediate gearbox (para. 14).

9. To minimize the effect of the driveshaft passing through the compressor duct and disrupting the airflow, it is housed within the compressor support structure. On by-pass engines, the driveshaft is either housed in the outlet guide vanes or in a hollow streamlined radial fairing across the low pressure compressor duct.

10. To reduce airflow disruption it is desirable to have the smallest driveshaft diameter as possible. The smaller the diameter, the faster the shaft must

HIGH SPEED GEARBOX
DRIVING GEARSHAFT

H.P. COMPRESSOR SHAFT

EXTERNAL (low speed)
GEARBOX
RADIAL DRIVE SHAFT

IDLER
GEARSHAFT

LOW PRESSURE
COMPRESSOR
SHAFT

LOW SPEED GEARBOX
MASTER DRIVING GEAR

EXTERNAL (high speed)
GEARBOX
RADIAL DRIVE SHAFT
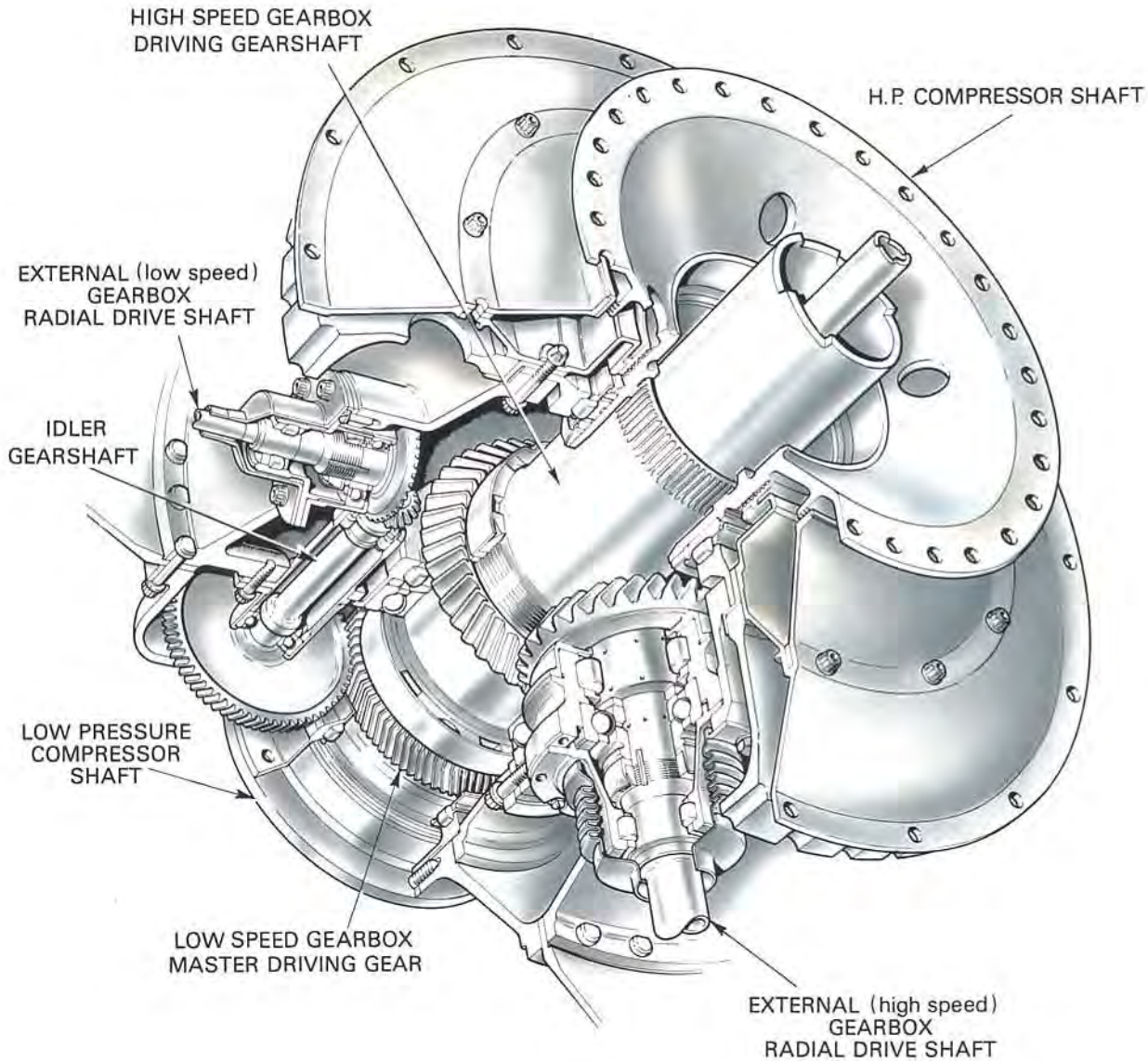
Fig. 7-3   An internal gearbox.

rotate to provide the same power. However, this raises the internal stress and gives greater dynamic problems which result in vibration. A long radial driveshaft usually requires a roller bearing situated halfway along its length to give smooth running. This allows a rotational speed of approximately 25,000 r.p.m. to be achieved with a shaft diameter of less than 1.5 inch without encountering serious vibration problems.

Direct drive
11. In some early engines, a radial driveshaft was used to drive each, or in some instances a pair, of accessory units. Although this allowed each accessory unit to be located in any desirable location around the engine and decreased the power transmitted through individual gears, it necessitated a large internal gearbox. Additionally, numerous radial driveshafts had to be incorporated within the design. This led to an excessive amount of time required for disassembly and assembly of the engine for maintenance purposes.

12. In some instances the direct drive method may be used in conjunction with the external gearbox system when it is impractical to take a drive from a particular area of the engine to the external gearbox. For example, fig. 7-1 shows a turbo-propeller engine

which requires accessories specific to the propeller reduction drive, but has the external gearbox located away from this area to receive the drive from the compressor shaft.

Gear train drive

13.   When space permits, the drive may be taken to the external gearbox via a gear train (fig. 7-1). This involves the use of spur gears, sometimes incorporating a centrifugal breather (Part 8). However, it is rare to find this type of drive system in current use.

Intermediate gearbox

14.   Intermediate gearboxes are employed when it is not possible to directly align the radial driveshaft with the external gearbox. To overcome this problem an intermediate gearbox is mounted on the high pressure compressor case and re-directs the drive, through bevel gears, to the external gearbox. An example of this layout is shown in fig. 7-1.

External gearbox

15.   The external gearbox contains the drives for the accessories, the drive from the starter and provides a mounting face for each accessory unit. Provision is also made for hand turning the engine, via the gearbox, for maintenance purposes. Fig. 7-4 shows the accessory units that are typically found on an external gearbox.

16.   The overall layout of an external gearbox is dictated by a number of factors. To reduce drag whilst the aircraft is flying it is important to present a low frontal area to the airflow. Therefore the gearbox is 'wrapped' around the engine and may look, from the front, similar to a banana in shape. For maintenance purposes the gearbox is generally located on the underside of the engine to allow ground crew to gain access. However, helicopter installation design usually requires the gearbox to be located on the top of the engine for ease of access.

17.   The starter/driven gearshaft (fig. 7-4) roughly divides the external gearbox into two sections. One section provides the drive for the accessories which require low power whilst the other drives the high power accessories. This allows the small and large gears to be grouped together independently and is an efficient method of distributing the drive for the minimum weight.

18.   If any accessory unit fails, and is prevented from rotating, it could cause further failure in the external gearbox by shearing the teeth of the gear train. To prevent secondary failure occurring a weak section is machined into the driveshafts, known as a 'shear-neck', which is designed to fail and thus protect the other drives. This feature is not included for primary engine accessory units, such as the oil pumps, because these units are vital to the running of the engine and any failure would necessitate immediate shutdown of the engine.

19.   Since the starter provides the highest torque that the drive system encounters, it is the basis of design. The starter is usually positioned to give the shortest drive line to the engine core. This eliminates the necessity of strengthening the entire gear train which would increase the gearbox weight. However, when an auxiliary gearbox is fitted (para, 21) the starter is moved along the gear train to allow the heavily loaded auxiliary gearbox drive to pass through the external gearbox. This requires the spur gears between the starter and starter/driven gearshaft to have a larger face width to carry the load applied by the starter (fig. 7-5).

20.   When a drive is taken from two compressor shafts, as discussed in para. 7, two separate gearboxes are required. These are mounted either side of the compressor case and are generally known as the 'low speed' and 'high speed' external gearboxes.

Auxiliary gearbox

21.   An auxiliary gearbox is a convenient method of providing additional accessory drives when the configuration of an engine and airframe does not allow enough space to mount all of the accessory units on a single external gearbox.

22.   A drive is taken from the external gearbox (fig. 7-5) to power the auxiliary gearbox which distributes the appropriate gear ratio drive to the accessories in the same manner as the external gearbox.

CONSTRUCTION AND MATERIALS

Gears

23.   The spur gears of the external or auxiliary gearbox gear train (fig. 7-4 and 7-5) are mounted between bearings supported by the front and rear casings which are bolted together. They transmit the drive to each accessory unit, which is normally between 5000 and 6000 r.p.m. for the accessory units and approximately 20,000 r.p.m. for the centrifugal breather,

24.   All gear meshes are designed with 'hunting tooth' ratios which ensure that each tooth of a gear does not engage between the same set of opposing teeth on each revolution. This spreads any wear evenly across all teeth.

Accessory drives
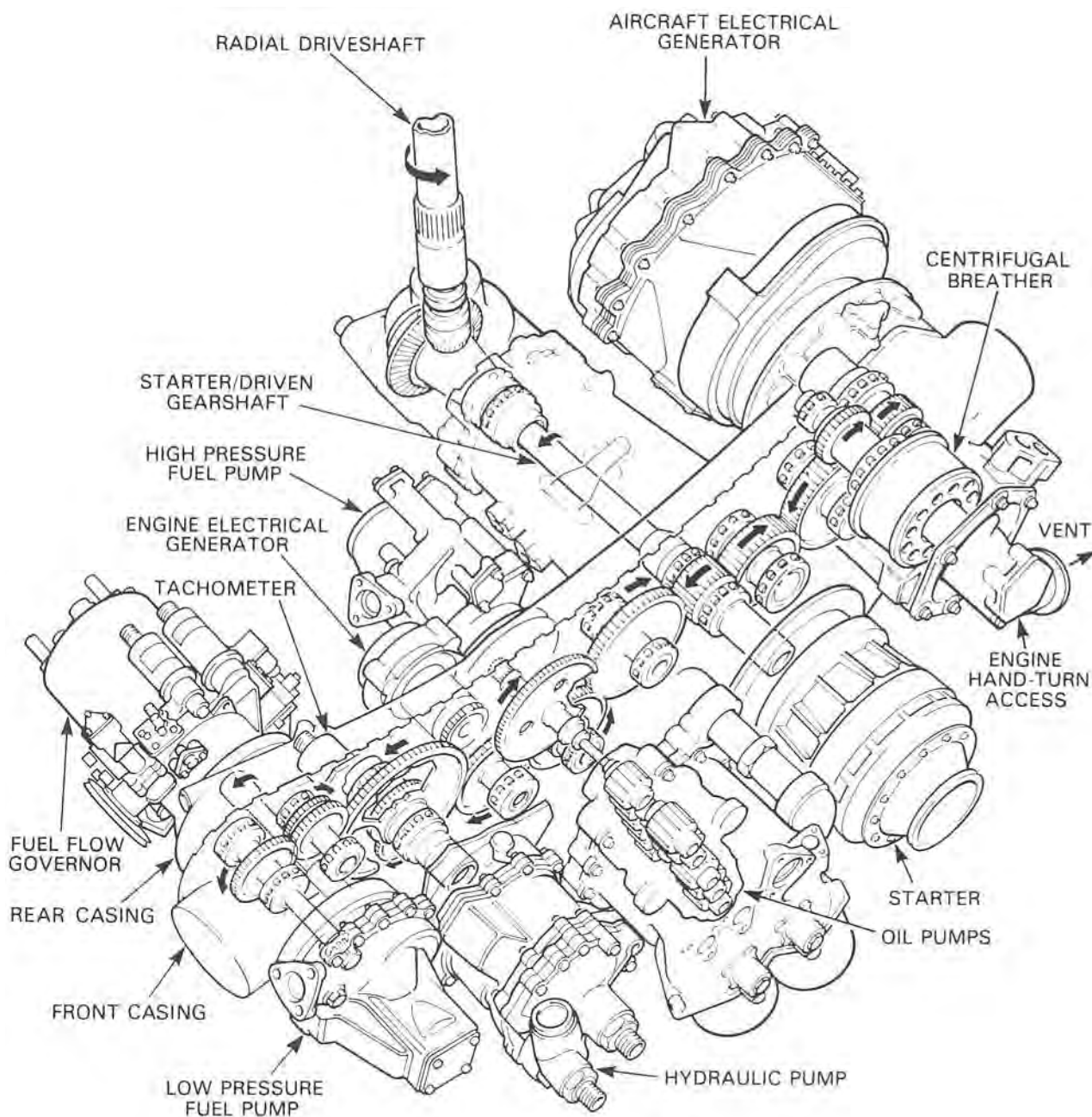
Fig. 7-4   An external gearbox and accessory units.

25. Spiral bevel gears are used for the connection of shafts whose axes are at an angle to one another but in the same plane. The majority of gears within a gear train are of the straight spur gear type, those with the widest face carry the greatest loads. For smoother running, helical gears are used but the resultant end thrust caused by this gear tooth pattern must be catered for within the mounting of the gear.

Gearbox sealing

26. Sealing of the accessory drive system is primarily concerned with preventing oil loss. The internal gearbox has labyrinth seals where the static casing mates with the rotating compressor shaft. For some o! the accessories mounted on the external gearbox, an air blown pressurized labyrinth seal is

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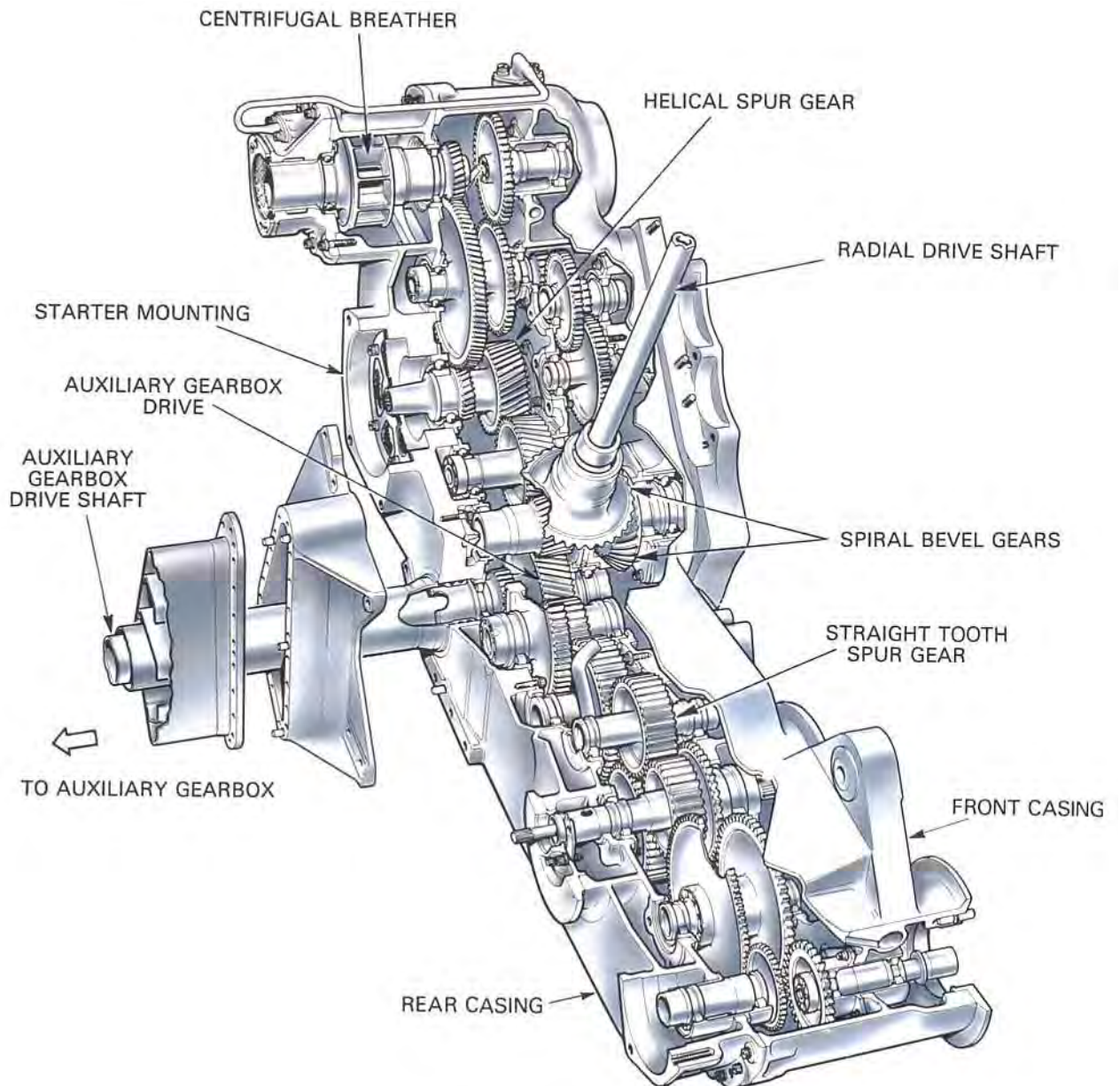
Fig. 7-5   An external gearbox with auxiliary gearbox drive.

employed. This prevents oil from the gearbox entering the accessory unit and also prevents contamination of the gearbox, and hence engine, in the event of an accessory failure. The use of an air blown seal results in a gearbox pressure of about 3 lbs. per sq. in. above atmospheric pressure. To supplement a labyrinth seal, an 'oil thrower ring' may be used. This involves the leakage oil running down the driving shaft and being flung outwards by a flange on the rotating shaft. The oil is then collected and returned to the gearbox.

Materials

27. To reduce weight, the lightest materials possible are used. The internal gearbox casing is cast from aluminium but the low environmental temperatures that an external gearbox is subjected to allows the use of magnesium castings which are lighter still. The gears are manufactured from non-corrosion resistant steels for strength and toughness. They are case hardened to give a very hard wear resistant skin and feature accurately ground teeth for smooth gear meshing.

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Rolls-Royce Tay
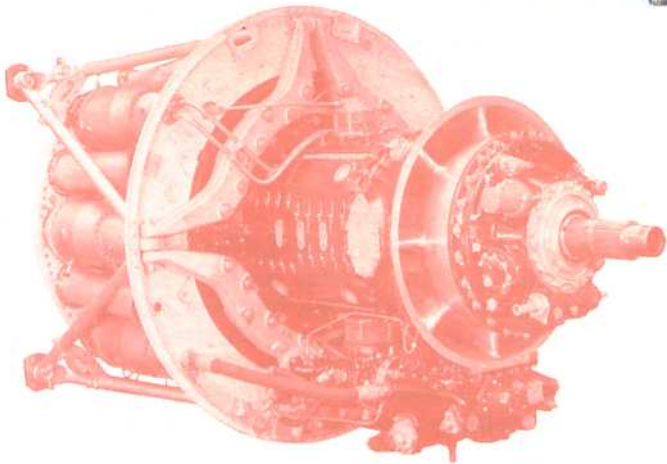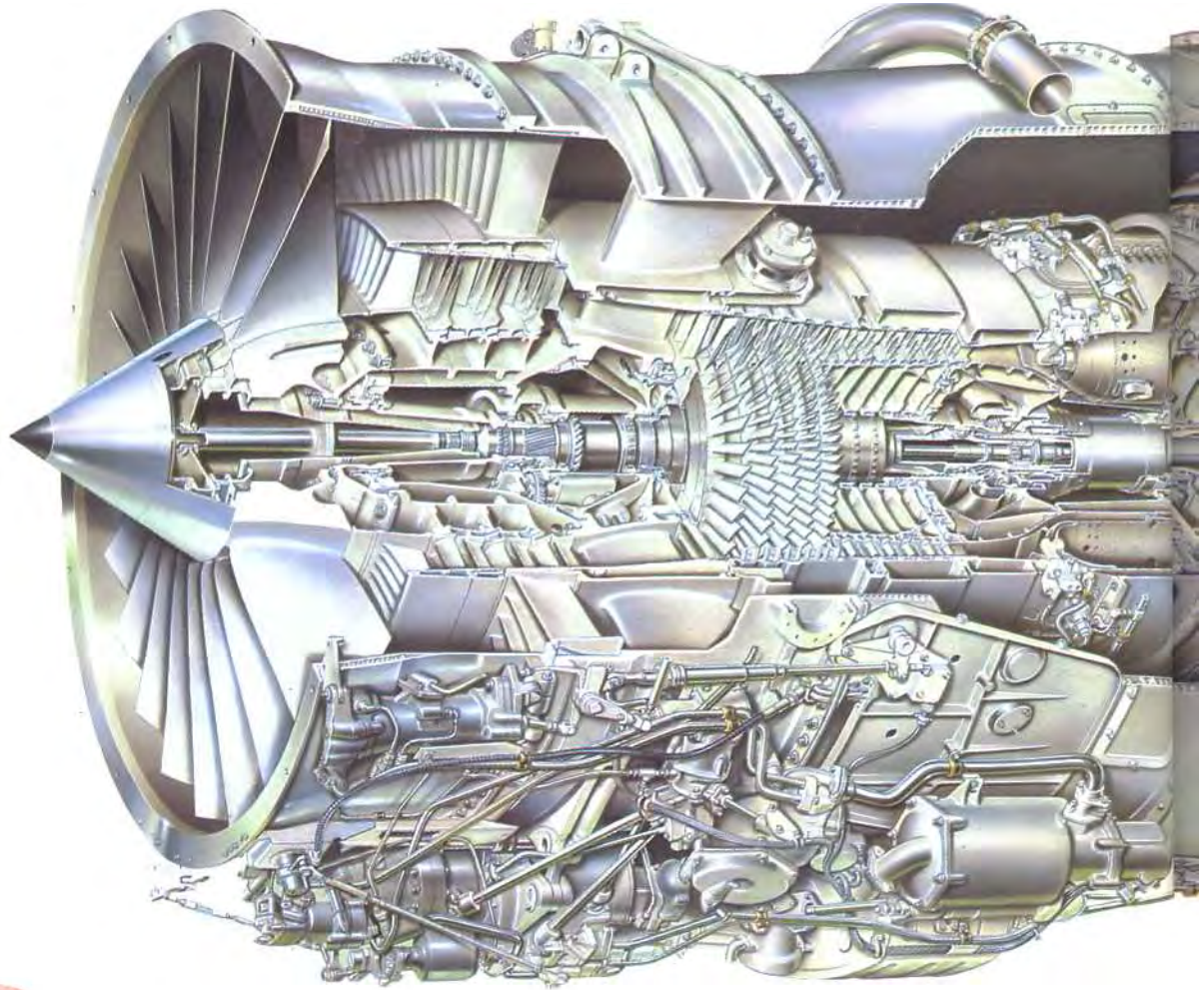
Bristol Theseus

This engine was conceived in 1940 as a 4000 hp turbo-prop but was later scaled down to 2000 hp. Named the Theseus the engine was type tested in December 1946. the world's first turbo-prop to reach this stage of development. After extensive flight testing in an Avro Lincoln, four Theseus engines were installed in a Handley-Page Hermes 5.

8: Lubrication
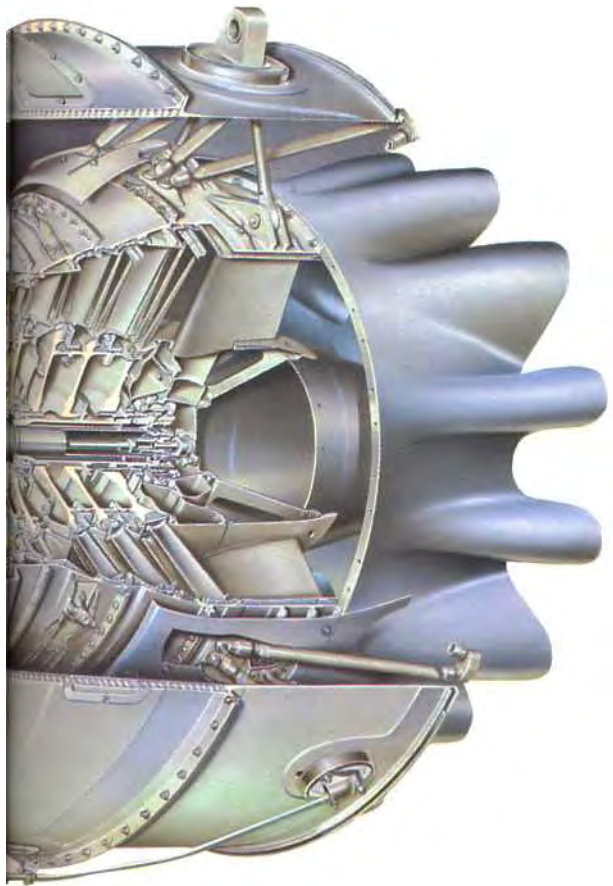
Contents

INTRODUCTION

1. The lubrication system is required to provide lubrication and cooling for all gears, bearings and splines. It must also be capable of collecting foreign matter which, if left in a bearing housing or gearbox, can cause rapid failure. Additionally, the oil must protect the lubricated components which are manufactured from non-corrosion resistant materials. The oil must accomplish these tasks without significant deterioration.

2. The requirements of a turbo-propeller engine are somewhat different to any other types of aero gas turbine. This is due to the additional lubrication of the heavily loaded propeller reduction gears and the need for a high pressure oil supply to operate the propeller pitch control mechanism.

3. Most gas turbine engines use a self-contained recirculatory lubrication system in which the oil is distributed around the engine and returned to the oil tank by pumps. However, some engines use a system known as the total loss or expendable system in which the oil is spilled overboard after the engine has been lubricated.

LUBRICATING SYSTEMS

4. There are two basic recirculatory systems, known as the 'pressure relief valve1 system and the 'full flow' system. The major difference between them is in the control of the oil flow to the bearings. In both systems the temperature and pressure of the oil are critical to the correct and safe running of the engine. Provision is therefore made for these parameters to be indicated in the cockpit.

Pressure relief valve system

5. In the pressure relief valve system the oil flow to the bearing chambers is controlled by limiting the pressure in the feed line to a given design value. This is accomplished by the use of a spring loaded valve which allows oil to be directly returned from the pressure pump outlet to the oil tank, or pressure pump inlet, when the design value is exceeded. The valve opens at a pressure which corresponds to the idling speed of the engine, thus giving a constant feed pressure over normal engine operating speeds. However, increasing engine speed causes the bearing chamber pressure to rise sharply. This reduces the pressure difference between the bearing chamber and feed jet, thus decreasing the oil flow rate to the bearings as engine speed increases. To alleviate this problem, some pressure relief valve systems use the increasing bearing chamber pressure to augment the relief valve spring load, This maintains a constant flow rate at the higher engine speeds by increasing the pressure in the feed line as the bearing chamber pressure increases.

6   Fig. 8-1 shows the pressure relief valve system for a turbo-propeller engine and indicates the basic components that comprise an engine lubrication system. The oil pressure pump draws oil from the tank through a strainer which protects the pump gears from debris which may have entered the tank, Oil is then delivered through a pressure filter to the pressure relief valve which maintains a constant oil delivery pressure to the feed jets in the bearing chambers. Some engines may have an additional relief valve (pressure limiting valve) which is fitted at the oil pressure pump outlet. This valve is set to open at a much higher value than the pressure relief valve to return the oil to the inlet side of the oil pressure pump in the event of the system becoming blocked. A similar valve may also be fitted across the pressure filter to prevent oil starvation of the bearing chambers should the filter become partially blocked or the oil having a high viscosity under cold starting conditions preventing sufficient flow through the filter. Provision is also made to supply oil to the propeller pitch control system, reduction gear and torquemeter
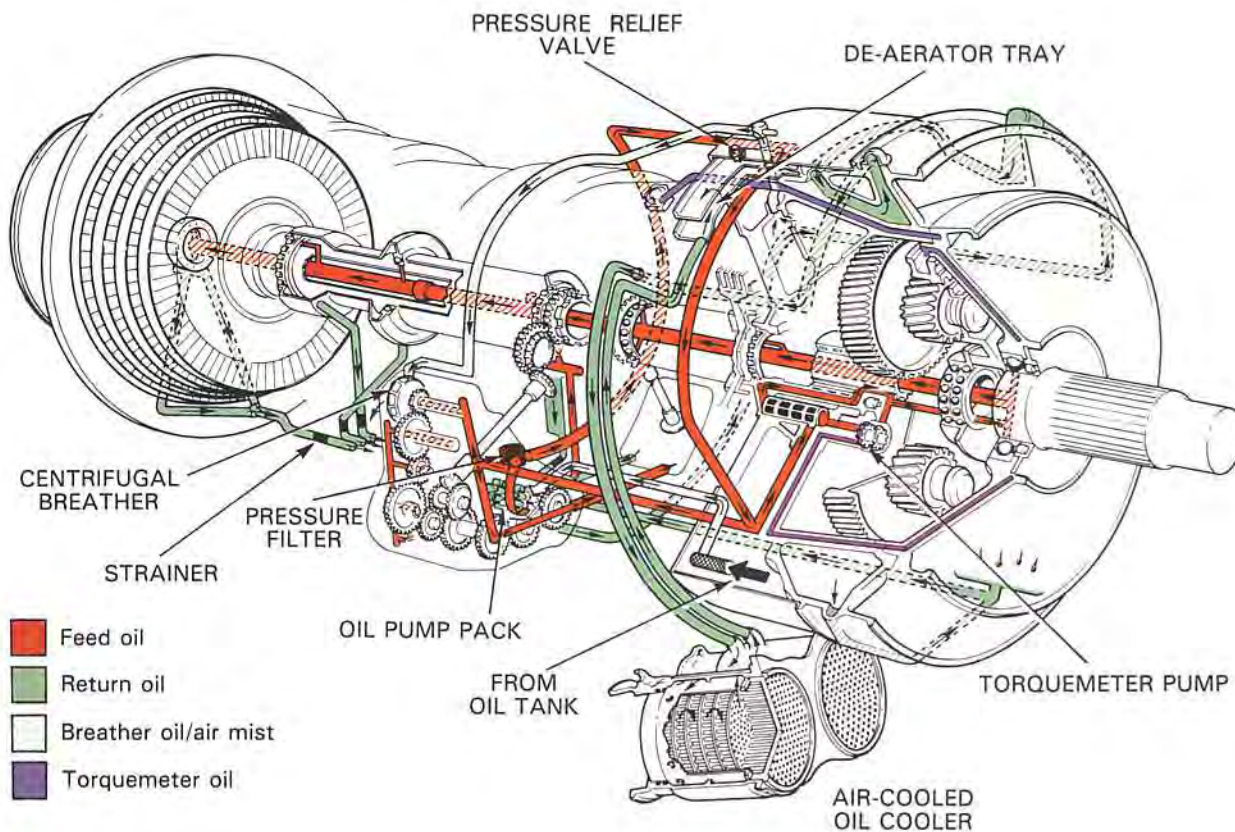
Fig. 8-1   A pressure relief valve type oil system.

system. Scavenge pumps return the oil to the tank via the oil cooler. On entering the tank, the oil is de-aerated ready for recirculation.

Full flow system

7. Although the pressure relief valve system operates satisfactorily for engines which have a low bearing chamber pressure, which does not unduly increase with engine speed, it becomes an undesirable system for engines which have high chamber pressures. For example, if a bearing chamber has a maximum pressure of 90 lb. per sq. in. It would require a pressure relief valve setting of 130 lb. per sq. in. to produce a pressure drop of 40 lb. per sq. in. at the oil feed jet. This results in the

need for large pumps and difficulty in matching the required oil flow at slower speeds.

8. The full flow system achieves the desired oil flow rates throughout the complete engine speed range by dispensing with the pressure relief valve and allowing the pressure pump delivery pressure to supply directly the oil feed jets. Fig. 8-2 shows an example of this system which may be found on a turbo-fan engine. The pressure pump size is determined by the flow required at maximum engine speed. The use of this system allows smaller pressure and scavenge pumps to be used since the large volume of oil which is spilled by the pressure relief valve system at maximum engine speed is obviated.
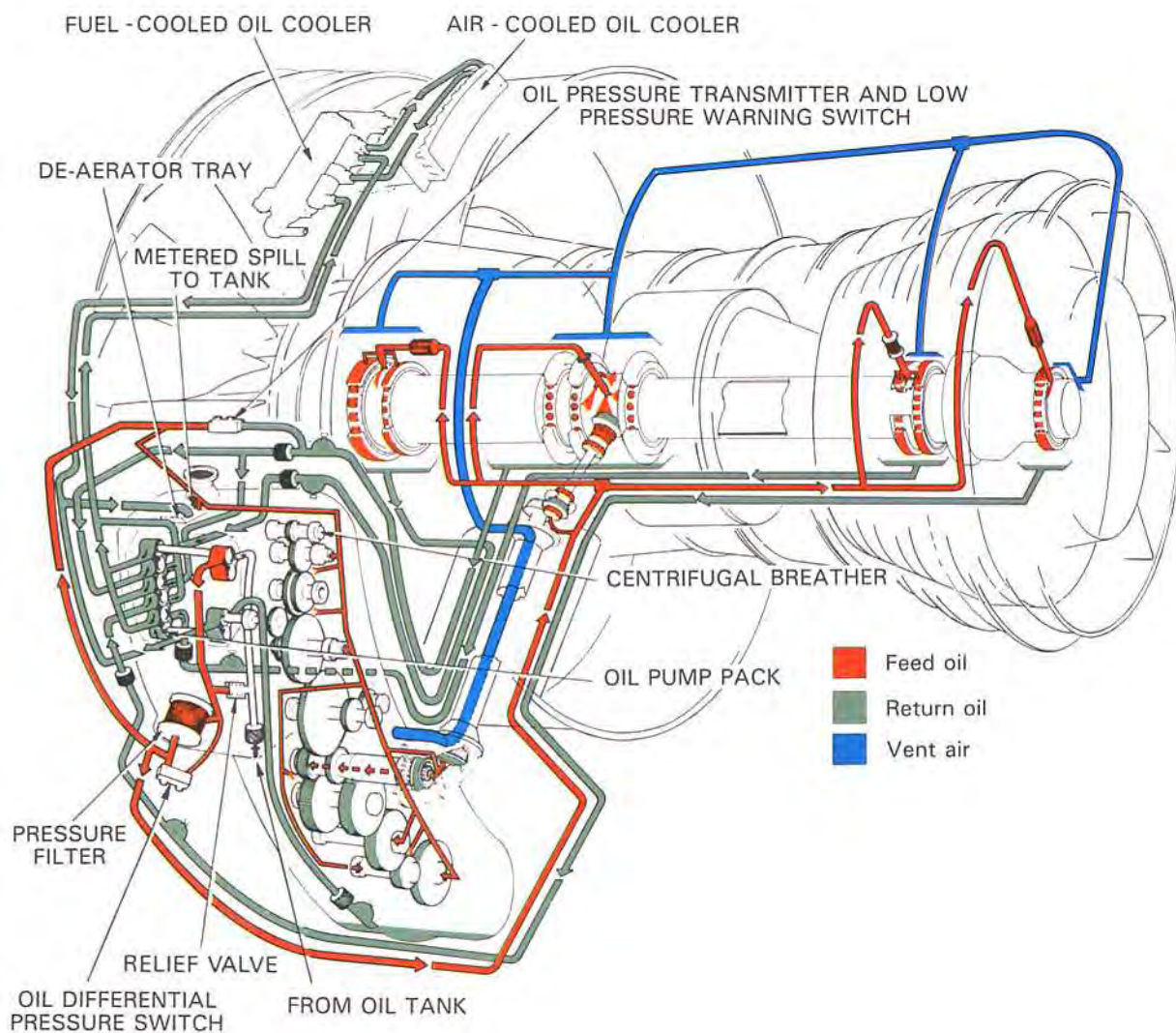
Fig. 8-2   A full flow type oil system.

9. To prevent high oil pressures from damaging filters or coolers, pressure limiting valves are fitted to by-pass these units. These valves normally only operate under cold starting conditions or in the event of a blockage. Advance warning of a blocked filter may be indicated in the cockpit by a differential pressure switch which senses an increase in the pressure difference between the inlet and outlet of the filter.

Total loss (expendable) system
10. For engines which run for periods of short duration, such as booster and vertical lift engines, the total loss oil system is generally used. The system is simple and incurs low weight penalties because it requires no oil cooler, scavenge pump or filters. On some engines oil is delivered in a continuous flow to the bearings by a plunger-type pump, indirectly driven from the compressor shaft; on others it is delivered by a piston-type pump operated by fuel pressure (fig. 8-3). In the latter, the oil supply is automatically selected by the high pressure fuel shut-off valve (cock) during engine starting and is delivered as a single shot to the front and rear bearings. On some engines provision is made for a
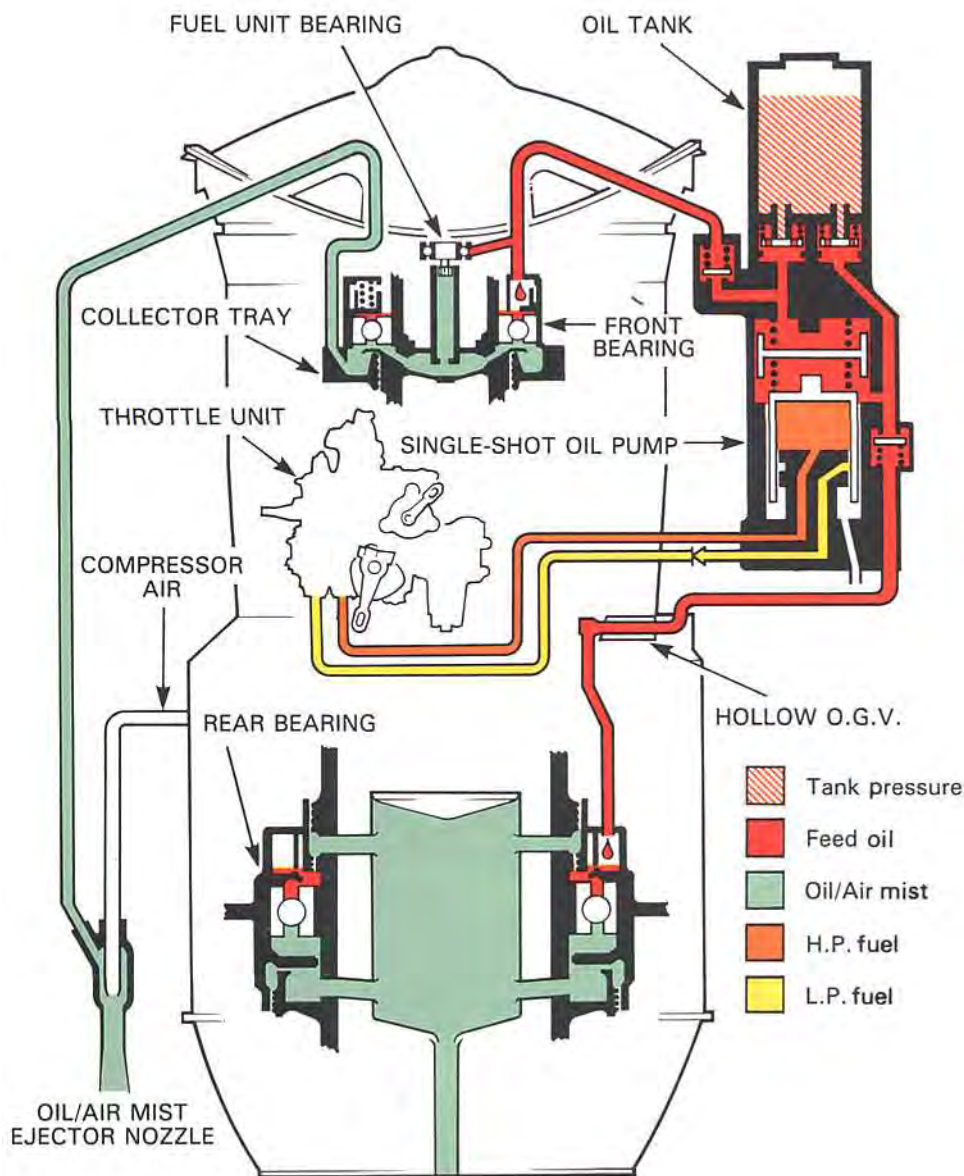
Fig. 8-3   A total loss (expendable) oil system.

second shot to be delivered to the rear bearing only, after a predetermined period.

11.   After lubricating the fuel unit and front bearings, the oil from the front bearing drains into a collector tray and is then ejected into the main gas stream through an ejector nozzle. The oil that has passed through the rear bearing, drains into a reservoir at the rear of the bearing where it is retained by centrifugal force until the engine is shut down. This

oil then drains overboard through a central tube in the exhaust unit inner cone.

OIL SYSTEM COMPONENTS

12.   The oil tank (fig. 8-4) is usually mounted on the engine and is normally a separate unit although it may also be an integral part of the external gearbox. It must have provision to allow the lubrication system to be drained and replenished. A sight glass or a dipstick must also be incorporated to allow the oil
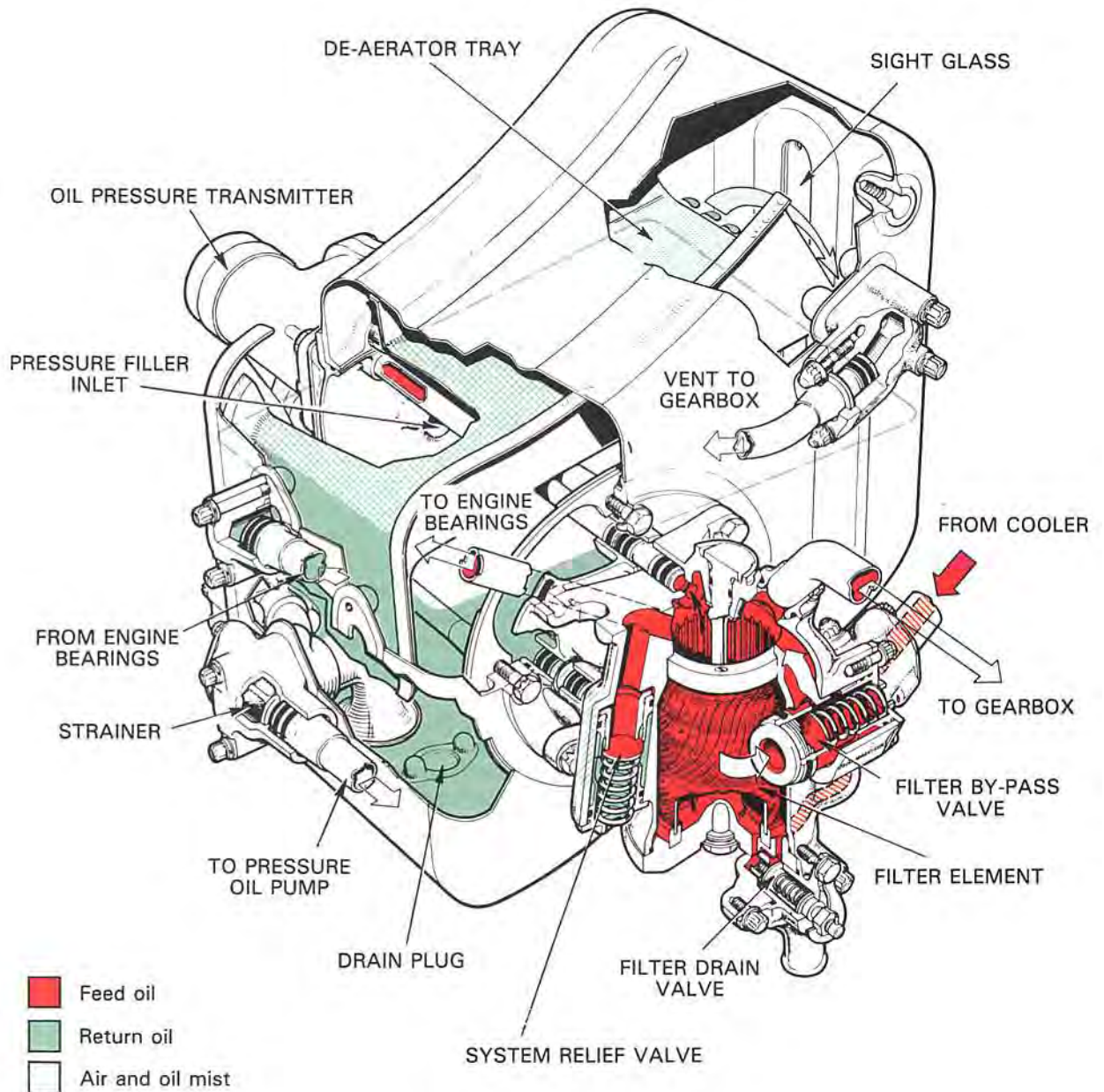
Fig. 8-4   An oil tank.

system contents to be checked. The filler can be either the gravity or pressure filling type; on some engines both types are fitted. Provision is also made for a continuous supply of oil to be made available in aircraft which are designed to operate during inverted flight conditions. Since air is mixed with the oil in the bearing chambers, a de-aerating device is incorporated within the oil tank which removes the air from the returning oil.

13.   The oil pumps are vital to the efficient operation of the engine. Failure of the pumps will necessitate a rapid shutdown of the engine. For this reason, the oil pump driveshafts do not incorporate a weak shear-neck (Part 7) because they must continue to supply oil for as long as possible, regardless of damage.

14.   As the feed oil is distributed to all the lubricated parts of the engine a substantial amount of sealing air (Part 9) mixes with it and increases its volume. Additionally the bearing chambers operate under differing pressures. Therefore, to prevent flooding it is usually necessary to have $. scavenge pump for each chamber.

15.   Gear type pumps are normally used in recirculatory oil systems but vane and gerotor pumps are employed in some engines. The simplicity of single-shot pumps (para. 19) make them ideal for engines which run for a short duration and use the total loss type of oil system.

16.   Gear pumps (fig. 8-5) consist of a pair of inter-meshing steel gears which are housed in a close fitting aluminium casing. When the gears are rotated, oil is drawn into the pump, carried round between the teeth and casing and delivered at the outlet.

17.   Since a small quantity of incompressible oil becomes trapped in the gear mesh, which can cause a hydraulic lock and possible pump damage, a relief slot is machined into the end faces of the casing to provide an escape route for the oil.

18.   Gear pumps are used both as pressure (feed) pumps and scavenge (return) pumps and are incorporated within a common casing. The oil pumps pack is driven by the accessory drive system (Part 7).

19.   Single-shot pumps (fig. 8-6) have a quantity of oil contained within a cylinder. When the piston is forced up the cylinder bore, under the control of the throttle unit, the oil forces the outlet valves to open allowing a flow of oil to the parts required to be lubricated. When the piston reaches the top of the cylinder bore the outlet valves close due to the reduced oil pressure. Recharging of the oil pump
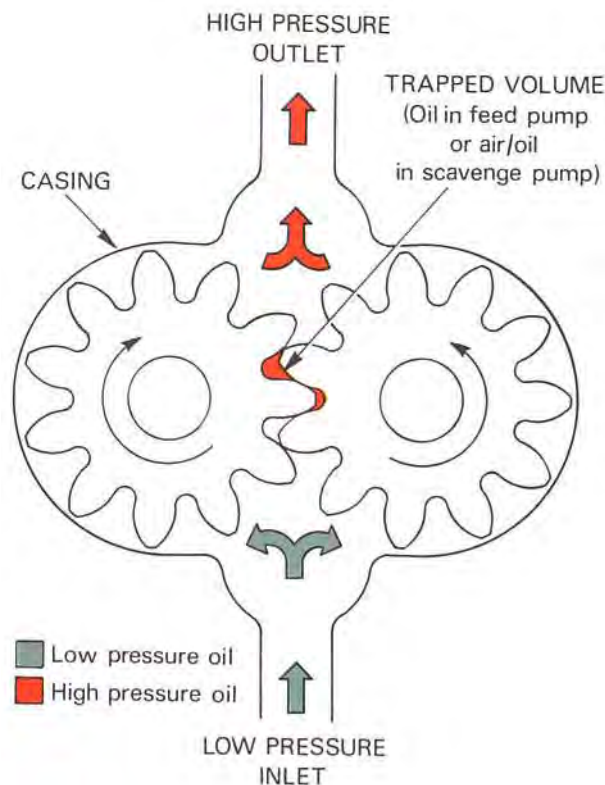
Fig. 8-5   **Principle of a gear pump.**

cylinder is achieved by a spring forcing the piston to its original position. This reduces the pressure between the cylinder and the oil tank which allows the oil replenshing valves to open until the cylinder is recharged.

20.   The most common type of oil distribution device is a simple orifice which directs a metered amount of oil onto its target. These jet orifices are positioned as close to the target area as possible to overcome the possibility of the local turbulent environment deflecting the jet of oil. The smallest diameter of a jet orifice is 0.04 inch which allows a flow of 12 gallons per hour when operating at a pressure of 40 lb. per sq. in. The use of restrictors upstream can reduce the flow rate if required.

21.   All engines transfer heat to the oil by friction, churning and windage within a bearing chamber or gearbox. It is therefore common practice to fit an oil cooler in recirculatory oil systems. The cooling medium may be fuel or air and, in some instances, both fuel-cooled and air-cooled coolers are used.
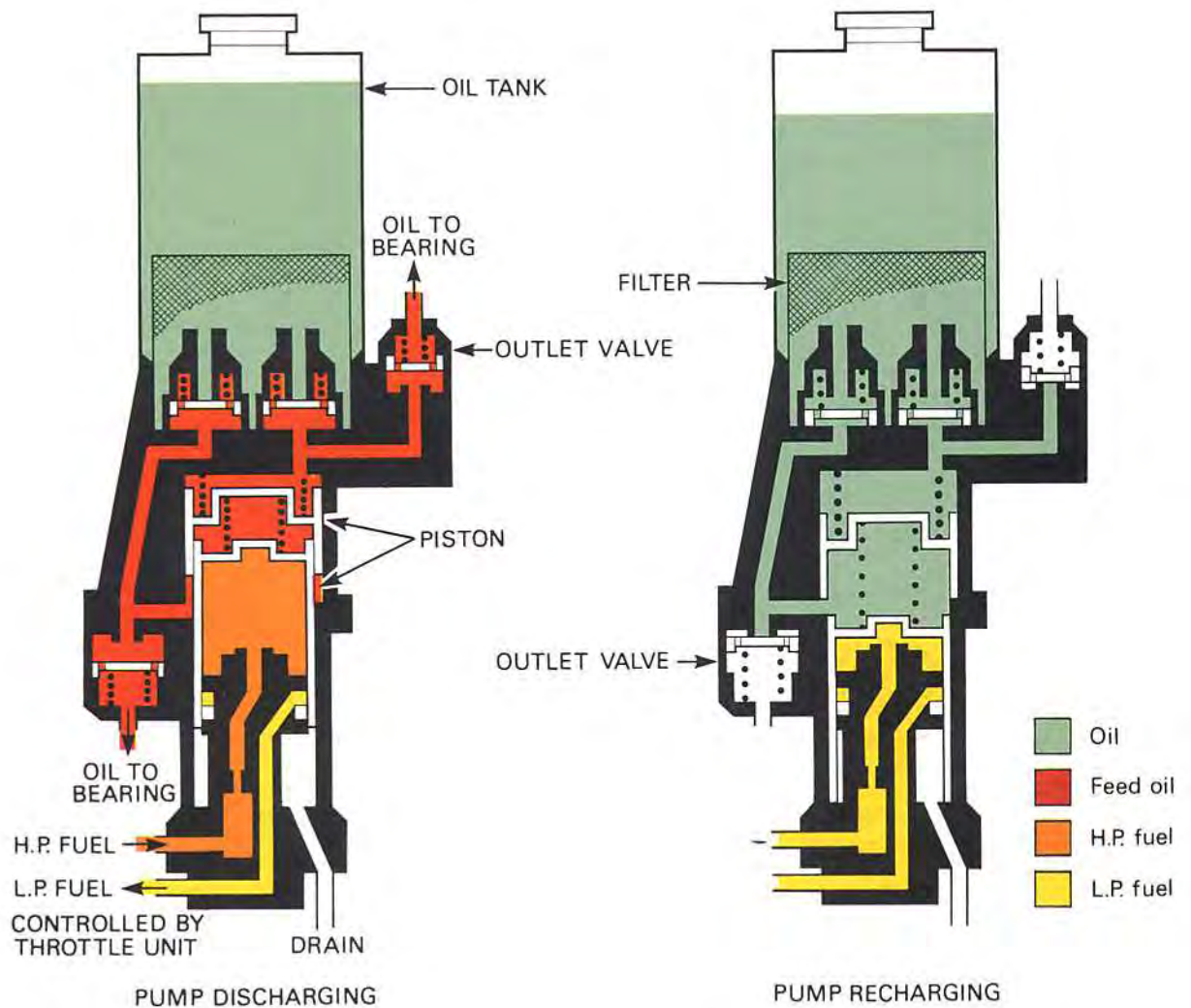
Fig. 8-6   A single-shot oil pump.

22.   Some engines which utilize both types of cooler may incorporate an electronic monitoring system which switches in the air-cooled cooler only when it is necessary. This maintains the ideal oil temperature and improves the overall thermal efficiency.

23.   The fuel-cooled oil cooler (fig. 8-7) has a matrix which is divided into sections by baffle plates. A large number of tubes convey the fuel through the matrix, the oil being directed by the baffle plates in a series of passes across the tubes. Heat is transferred from the oil to the fuel, thus lowering the oil temperature.

24.   The fuel-cooled oil cooler incorporates a bypass valve fitted across the oil inlet and outlet. The valve operates at a pre-set pressure difference across the

cooler and thus prevents engine oil starvation in the event of a blockage. A pressure maintaining valve is usually located in the feed line of the cooler which ensures that the oil pressure is always higher than the fuel pressure. In the event of a cooler internal fault developing, the oil will leak into the fuel system rather than the potentially dangerous leakage of fuel into the oil system.

25.   The air-cooled oil cooler is similar to the fuel-cooled type in both construction and operation; the main difference is that air is used as the cooling medium.

26.   Magnetic plugs, or chip detectors (fig. 8-8), are fitted on the scavenge (return) side to collect ferritic

FUEL OUTLET

L.P. fuel

Feed oil

BAFFLE PLATE

MATRIX ASSEMBLY

FUEL INLET

OIL BY-PASS VALVE

OIL INLET

OIL TEMPERATURE
TRANSMITTER

OIL OUTLET
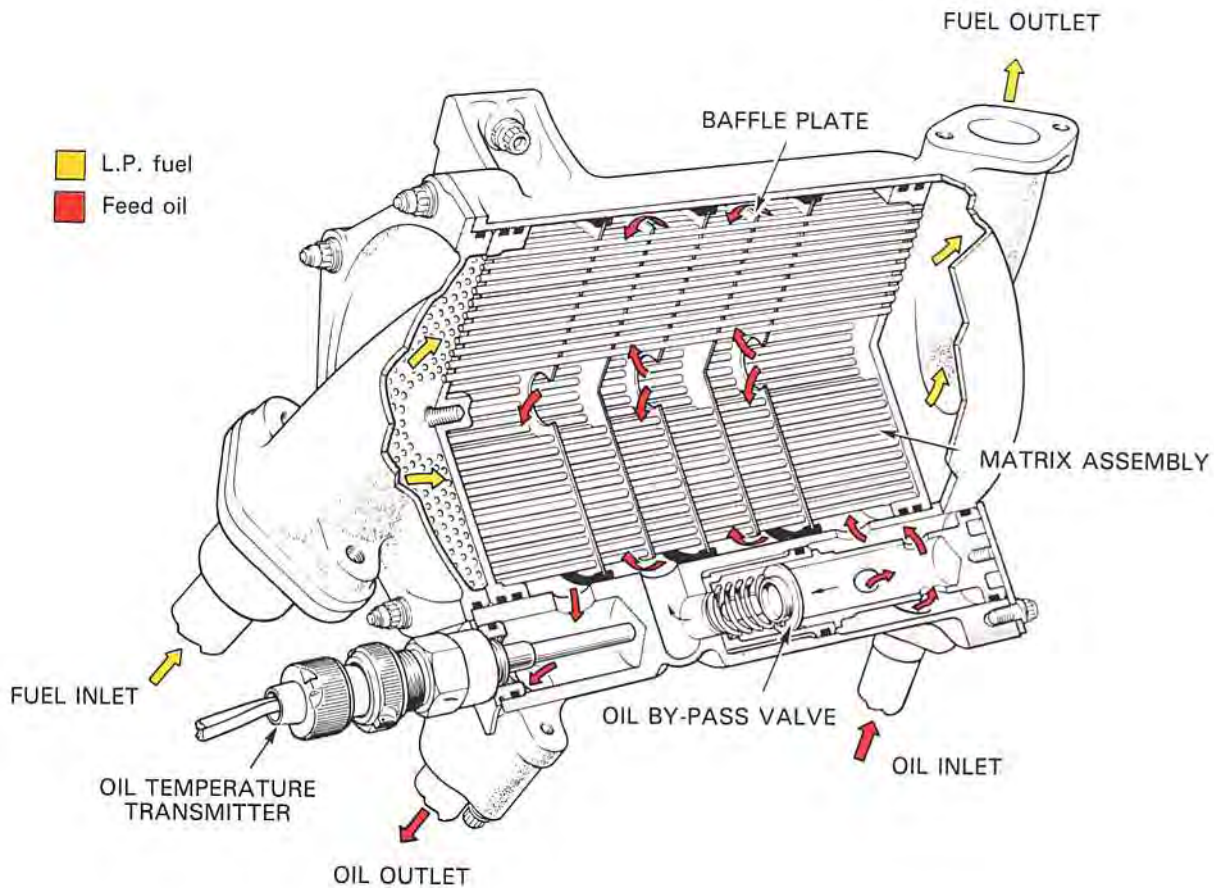
Fig. 8-7   A low pressure fuel-cooled oil cooler.

RETURN
OIL

CHIP DETECTOR

SELF-SEALING HOUSING
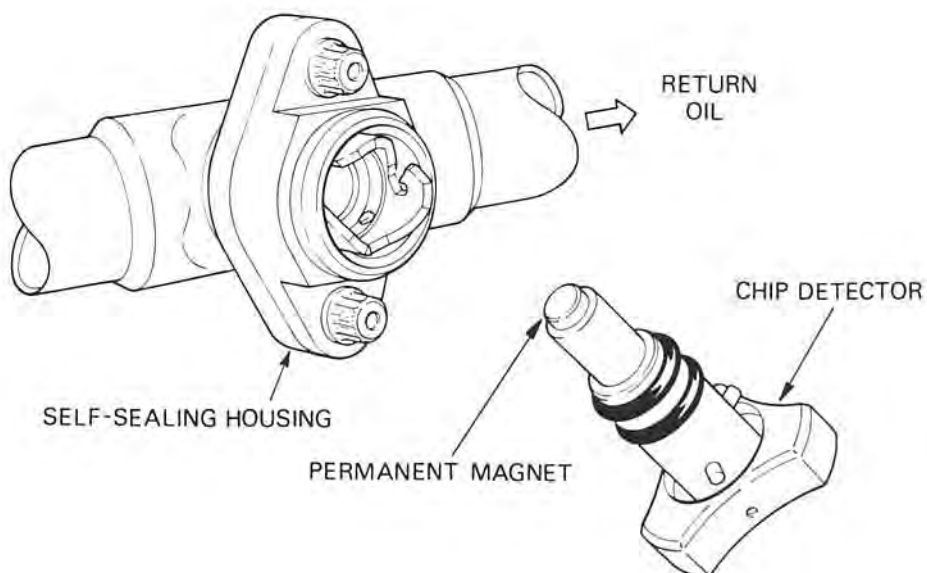
PERMANENT MAGNET

Fig. 8-8   A magnetic chip detector.

debris from each bearing chamber. They are basically permanent magnets inserted in the oil flow and are retained in self-sealing valve housings. Safety features incorporated in the design ensure correct retention within the housing. Upon examination they can provide a warning of impending failure without having to remove and inspect the filters. They are designed to be removed during maintenance inspection, for condition, monitoring purposes (Part 24), without oil loss occurring. Additionally they may be connected to a cockpit warning system to give an in-flight indication.
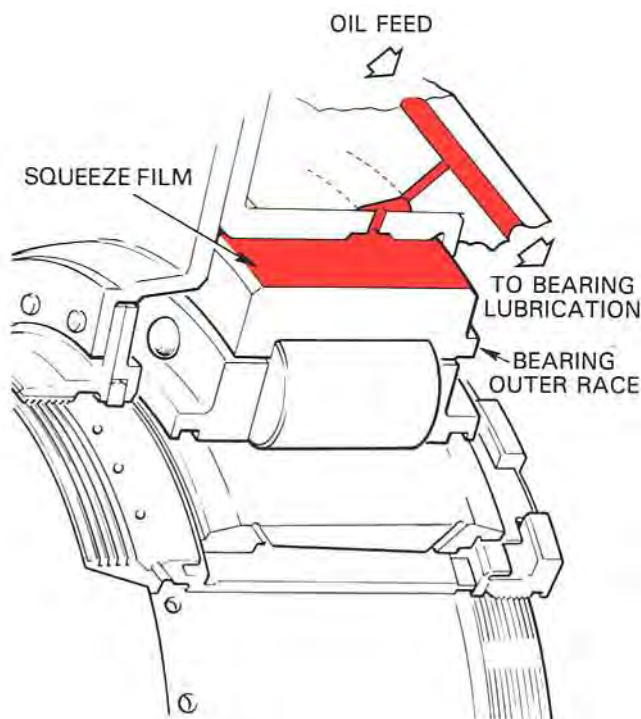
Fig. 8-9   A squeeze film bearing.

27.   In some engines, to minimize the effect of the dynamic loads transmitted from the rotating assemblies to the bearing housings, a 'squeeze film' type of bearing is used (fig. 8-9). They have a small clearance between the outer race of the bearing and housing with the clearance being filled with oil. The oil film dampens the radial motion of the rotating assembly and the dynamic loads transmitted to the bearing housing thus reducing the vibration level of the engine and the possibility of damage by fatigue.

28.   To prevent excessive air pressure within the oil tank, gearboxes and bearing chambers, a vent to atmosphere is incorporated within the lubrication
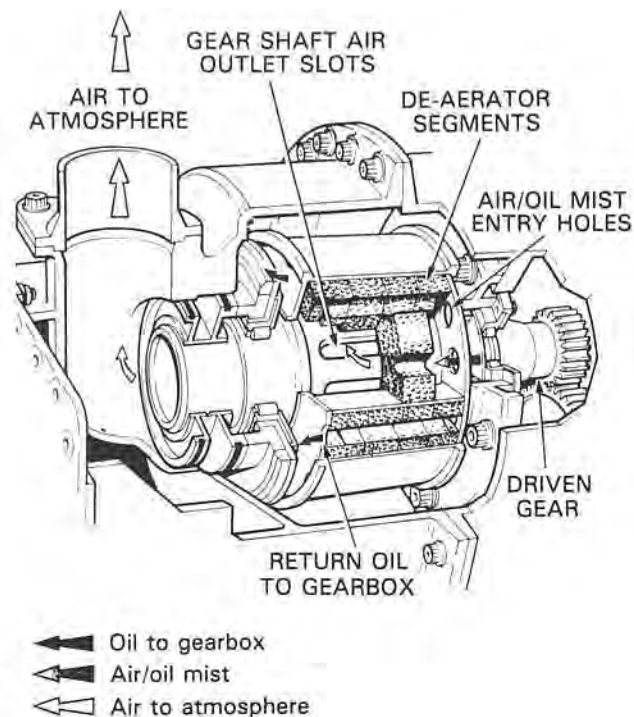
Fig. 8-10   A centrifugal breather.

system. Any oil droplets in the air are separated out by a centrifugal breather prior to the air being vented overboard. Some breathers may incorporate a porous media, forming de-aerator segments, which improves the efficiency of the oil separation (fig, 8-10).
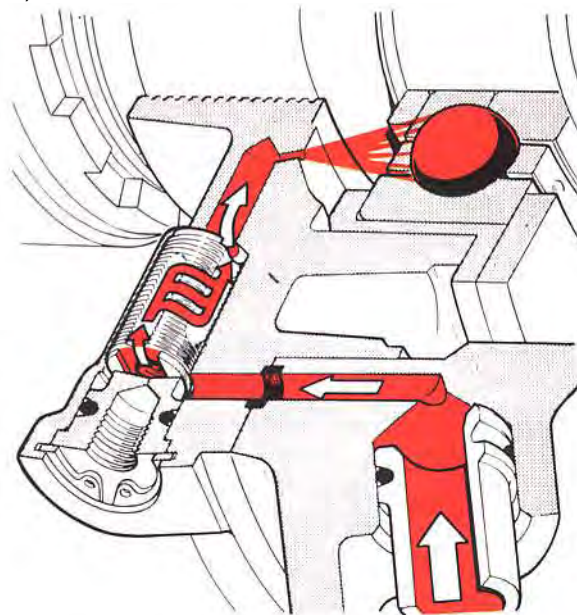
Fig. 8-11   A thread-type oil filter.
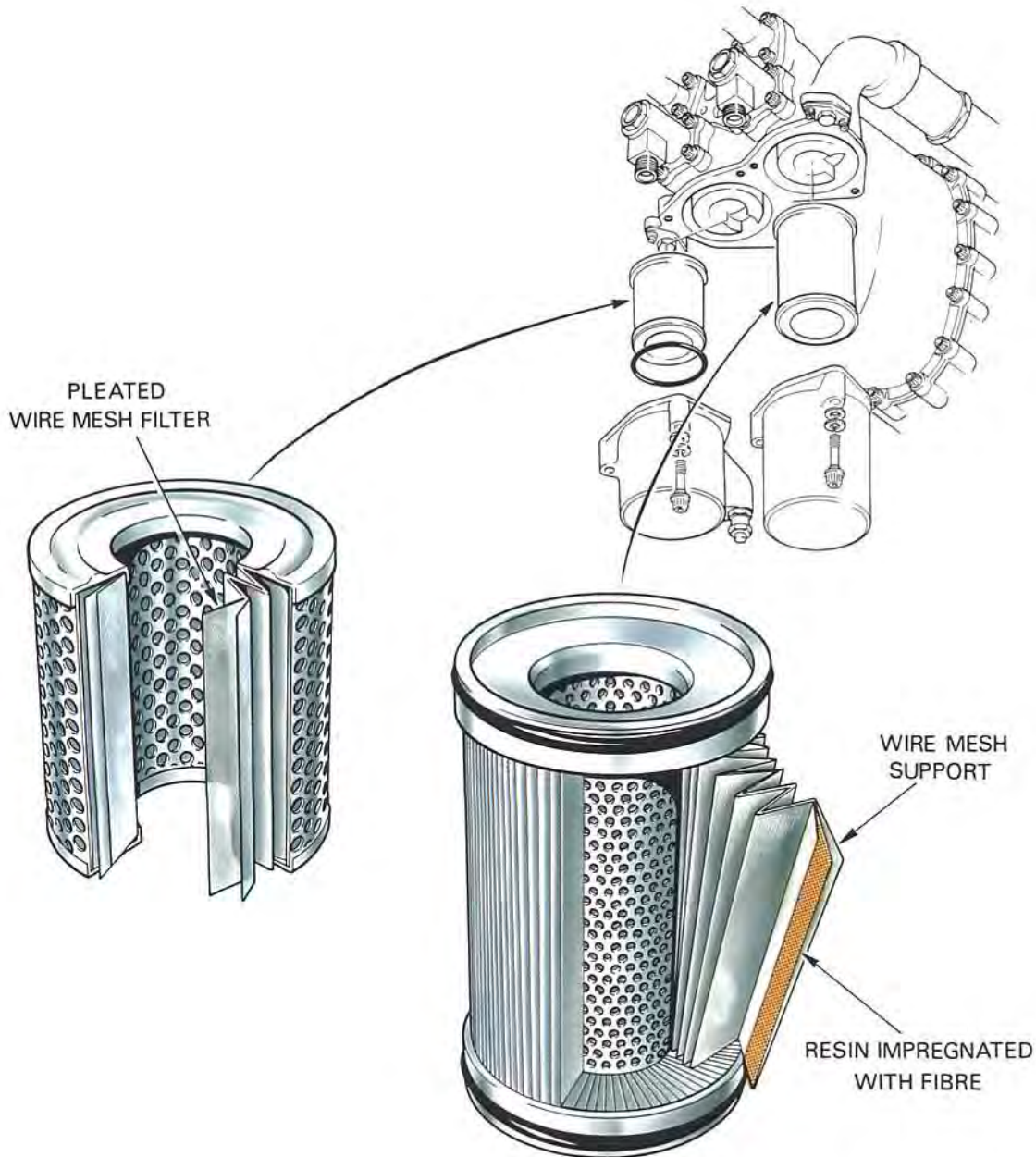
Fig. 8-12   A typical pressure and scavenge filter.

29. To prevent foreign matter from continuously circulating around the lubricating system, a number of filters and strainers are positioned within the system.

30. Coarse strainers are usually fitted at the outlet of the oil tank or immediately prior to the inlet of the oil pumps to prevent debris from damaging the pumps. A fine pressure filter is fitted at the pressure pump outlet which retains any small particles which could block the oil feed jets. Thread-type filters (fig. 8-11) are often fitted as a 'last chance' filter immediately upstream of the oil jets. Sometimes perforated plates or gauze filters are used for this application, Scavenge filters are fitted in each oil return line to collect any debris from the lubricated components. An example of a pressure and scavenge filter is shown in fig. 8-12. They are

invariably of tubular construction with a pleated woven wire cloth, or a resin impregnated with fibres, as the filtering medium. Some filters comprise one or more wire wound elements but these tend to be insufficient for fine filtration. A 'pop up indicator' may be fitted to the filter housing to give a visual warning of a partially blocked filter.

LUBRICATING OILS

31.   Early gas turbines used thinner oils than those used in piston engines but were produced from the same mineral crude oil. As gas turbines were developed to operate at higher speeds and temperatures these mineral oils oxidized and blocked the filters and oilways. The development of low viscosity (thin) synthetic oils overcame the major problems encountered with the early mineral oils.

32.   The choice of a lubricating oil is initially decided by the need to start the engine at very low temperatures, when the viscosity of the oil is high, whilst being able to survive in an engine environment which exhibits very high temperatures. Having met these fundamental requirements, the need to provide improved lubrication characteristics using additives must also be investigated. Special laboratory and engine tests are done to prove the suitability of a particular oil for a specific type of engine. Assessments are made as the extent to which it deteriorates and the corrosive effects it may have on the engine.

33.   Most gas turbines use a low viscosity oil due to the absence of reciprocating parts and heavy duty gearing. This reduces the power required for starting, particularly at low temperatures. In fact normal starts can be made in temperatures as low as -40 deg. C. without having to pre-heat the oil.

34.   Turbo-propeller engines use a slightly higher viscosity oil due to the additional requirements of the reduction gear and propeller pitch change mechanism.

Rolls-Royce RB162-86
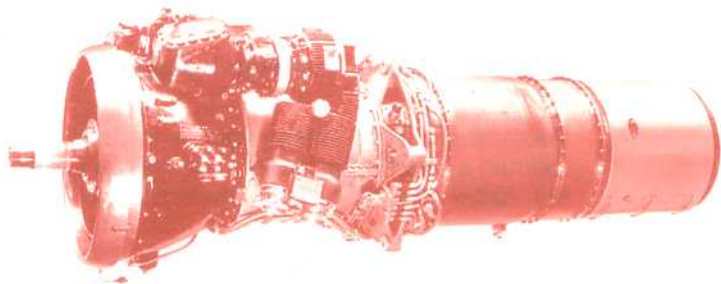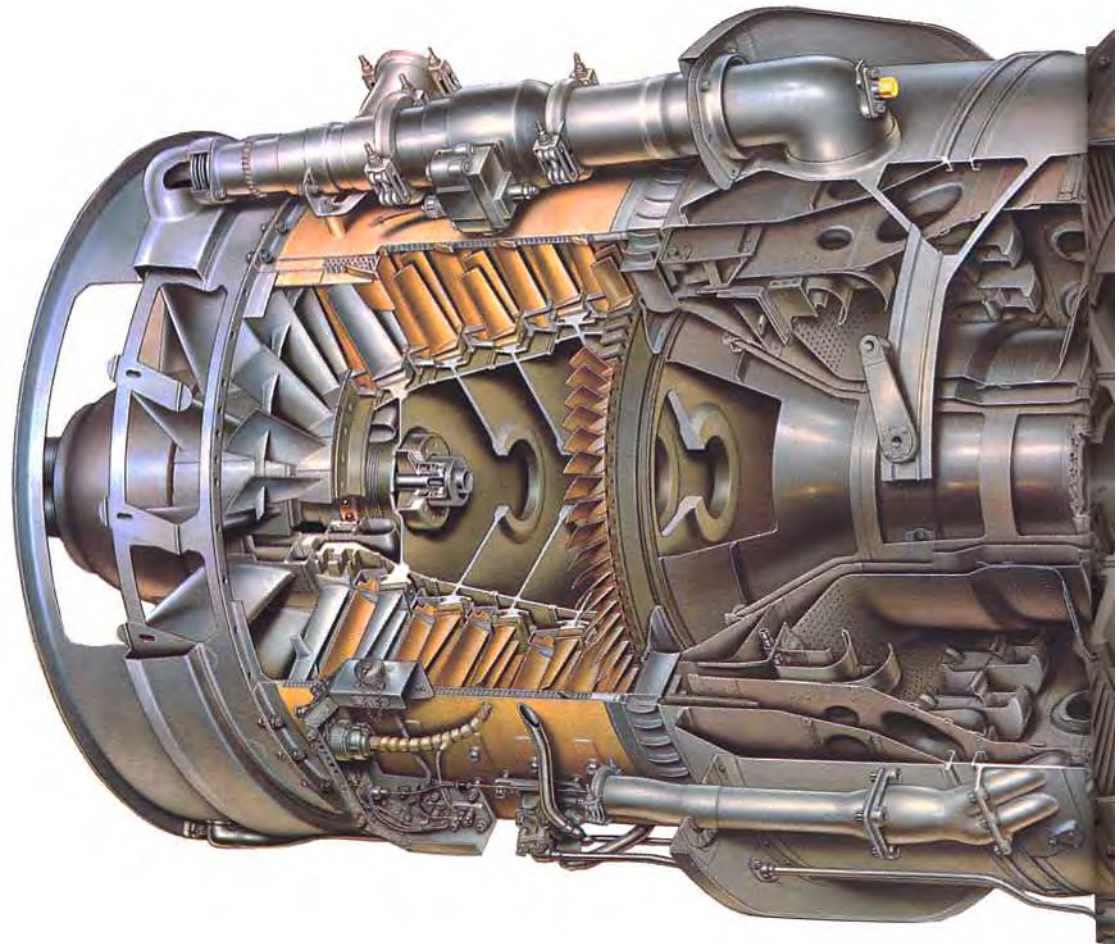
Armstrong Siddeley Mamba

The Mamba axial-flow turbo-prop was conceived in 1945 as a 1000 hp engine. First run in April 1946, the single Mamba eventually went into service with the Short Seamew at 1770 ehp. A further development was the Double Mamba, a combination of two single Mambas in one power unit. Providing up to 3875 ehp, the Double Mamba saw service with the Fairey Gannet.

9: Internal air system

Contents

INTRODUCTION

1. The engine internal air system is defined as those airflows which do not directly contribute to the engine thrust. The system has several important functions to perform for the safe and efficient operation of the engine. These functions include internal engine and accessory unit cooling, bearing chamber sealing prevention of hot gas ingestion into the turbine disc cavities, control of bearing axial loads, control of turbine blade tip clearances (Part 5) and engine anti-icing (Part 13). The system also supplies air for the aircraft services. Up to one fifth of the total engine core mass airflow may be used for these various functions.

2. An increasing amount of work is done on the air, as it progresses through the compressor, to raise its

pressure and temperature. Therefore, to reduce engine performance losses, the air is taken as early as possible from the compressor commensurate with the requirement of each particular function. The cooling air is expelled overboard via a vent system or into the engine main gas stream, at the highest possible pressure, where a small performance recovery is achieved.

COOLING

3. An important consideration at the design stage of a gas turbine engine is the need to ensure that certain parts of the engine, and in some instances certain accessories, do not absorb heat to the extent that is detrimental to their safe operation. The principal areas which require air cooling are the combustor and turbine. Refer to Part 4 for combustor cooling techniques.

4. Cooling air is used to control the temperature of the compressor shafts and discs by either cooling or heating them. This ensures an even temperature distribution and therefore improves engine efficiency by controlling thermal growth and thus maintaining minimum blade tip and seal clearances. Typical cooling and sealing airflows are shown in fig. 9-1.

Turbine cooling

5. High thermal efficiency is dependent upon high turbine entry temperature, which is limited by the turbine blade and nozzle guide vane materials. Continuous cooling of these components allows their environmental operating temperature to exceed the material's melting point without affecting the blade and vane integrity. Heat conduction from the turbine blades to the turbine disc requires the discs to be cooled and thus prevent thermal fatigue and uncontrolled expansion and contraction rates.
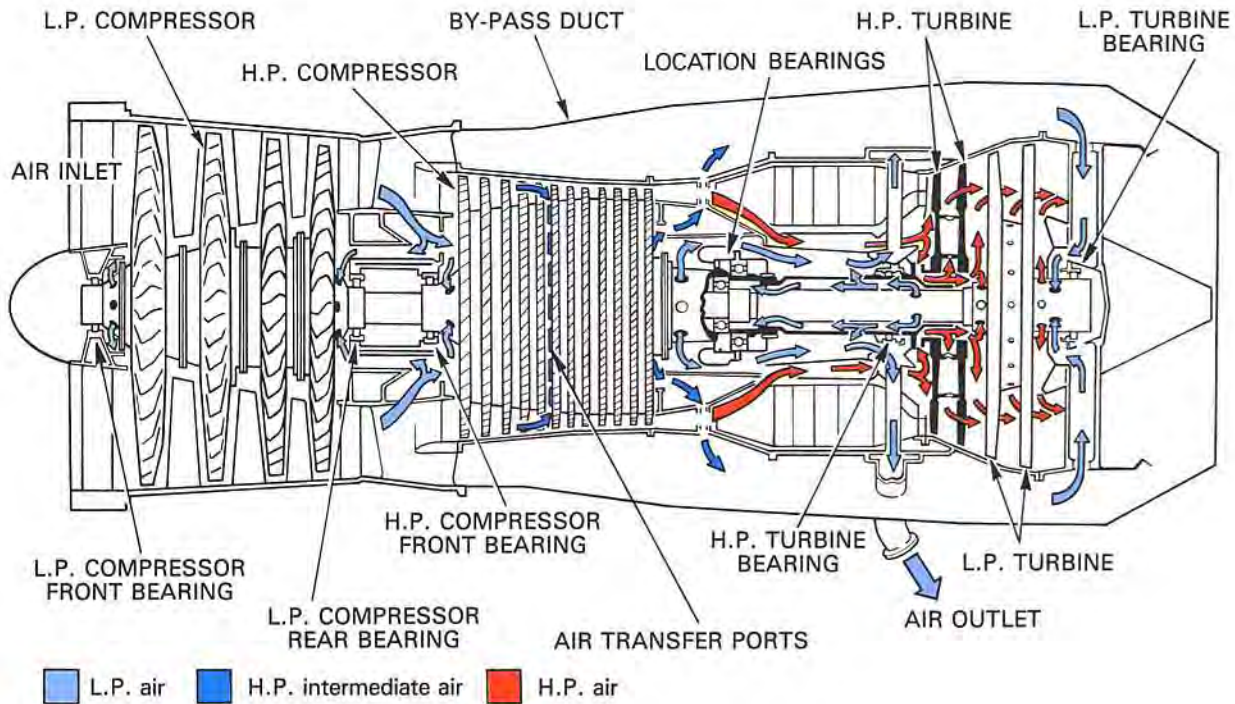
Fig. 9-1   General internal airflow pattern.
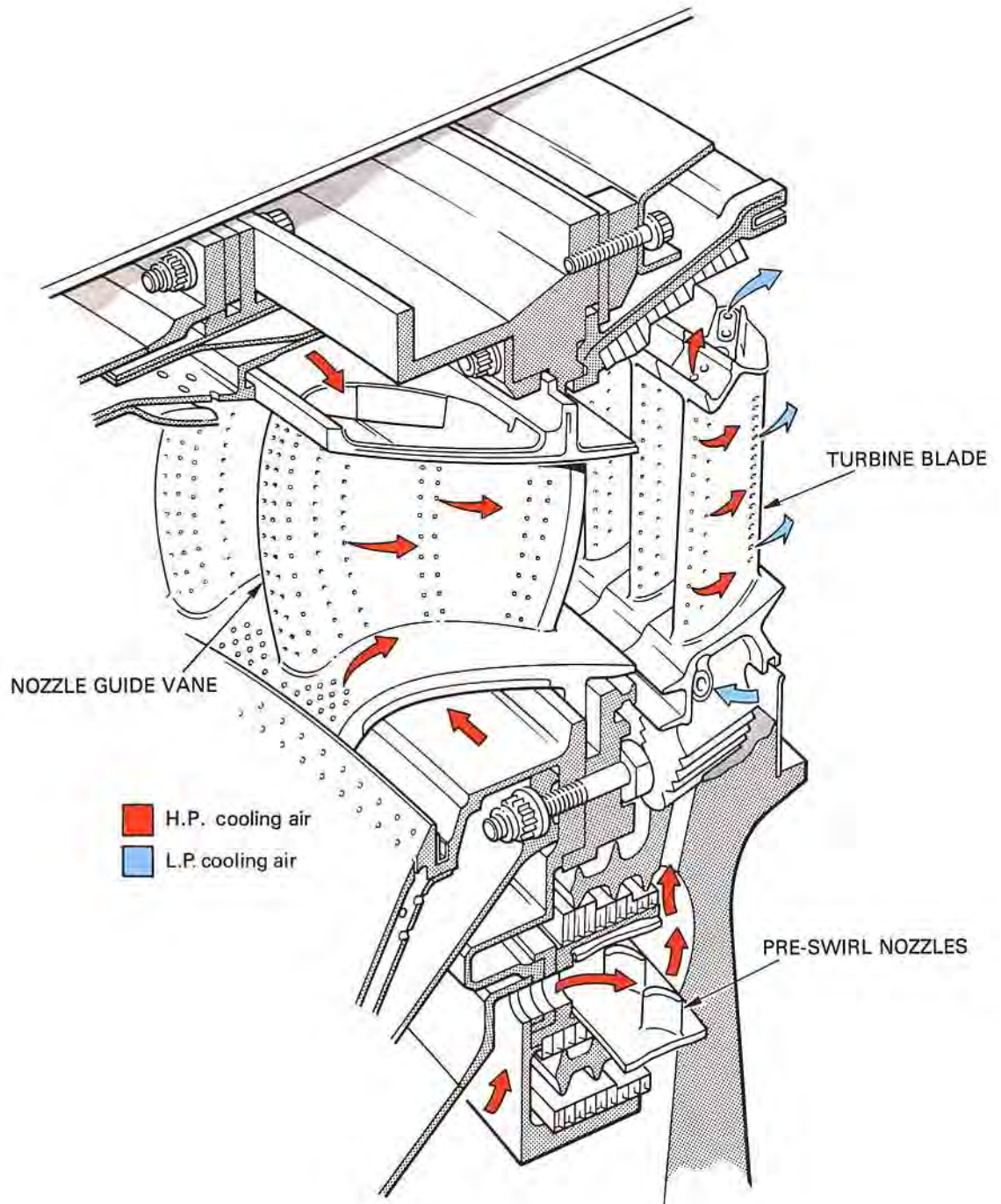
Fig. 9-2   Nozzle guide vane and turbine blade cooling arrangement.

6. An air cooled high pressure nozzle guide vane and turbine blade arrangement illustrating the cooling airflow is shown in fig. 9-2. Turbine vane and turbine blade life depends not only on their form but also on the method of cooling, therefore the flow design of the internal passages is important. There have been numerous methods of turbine vane and turbine blade cooling which have been used throughout the history of gas turbines. Generally, single pass internal (convection) cooling was of great practical benefit but development has lead to multi-pass internal cooling of blades, impingement cooling of vanes with external air film cooling of both vanes and blades, these are shown in fig. 9-3. and fig. 9-4.

7. The 'pre-swirl nozzles' (fig. 9-2) reduce the temperature and pressure of the cooling air fed to the disc for blade cooling. The nozzles also impart a substantial whirl velocity to assist efficient entry of the air into the rotating cooling passages.

8. Cooling air for the turbine discs enters the annular spaces between the discs and flows outwards over the disc faces. Flow is controlled by interstage seals and, on completion of the cooling function, the air is expelled into the main gas stream (fig. 9-5); see para. 23., Hot gas ingestion.

Bearing chamber cooling
9. Air cooling of the engine bearing chambers is not normally necessary since the lubrication system (Part 8) is adequate for cooling purposes. Additionally, bearing chambers are located, where possible, in the cooler regions of the engine. In instances where additional cooling is required, it is good practice to have a double skinned bearing housing with cooling air fed into the intermediate space.

□ L.P. cooling air   ■ H.P. cooling air

SINGLE PASS,
INTERNAL COOLING
(1960's)

SINGLE PASS,
MULTI-FEED
INTERNAL COOLING
WITH FILM COOLING
(1970's)

QUINTUPLE PASS,
MULTI-FEED
INTERNAL COOLING
WITH EXTENSIVE
FILM COOLING
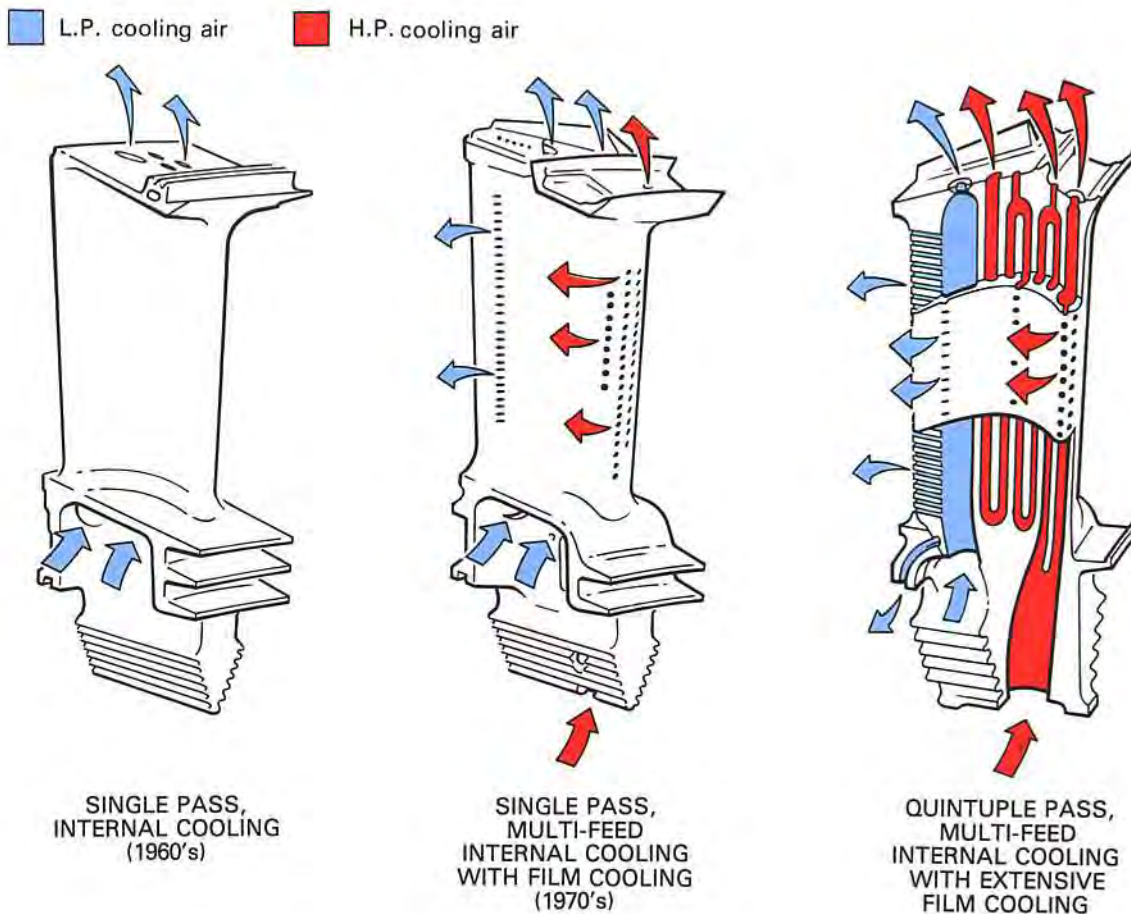
Fig. 9-3   Development of high pressure turbine blade cooling.
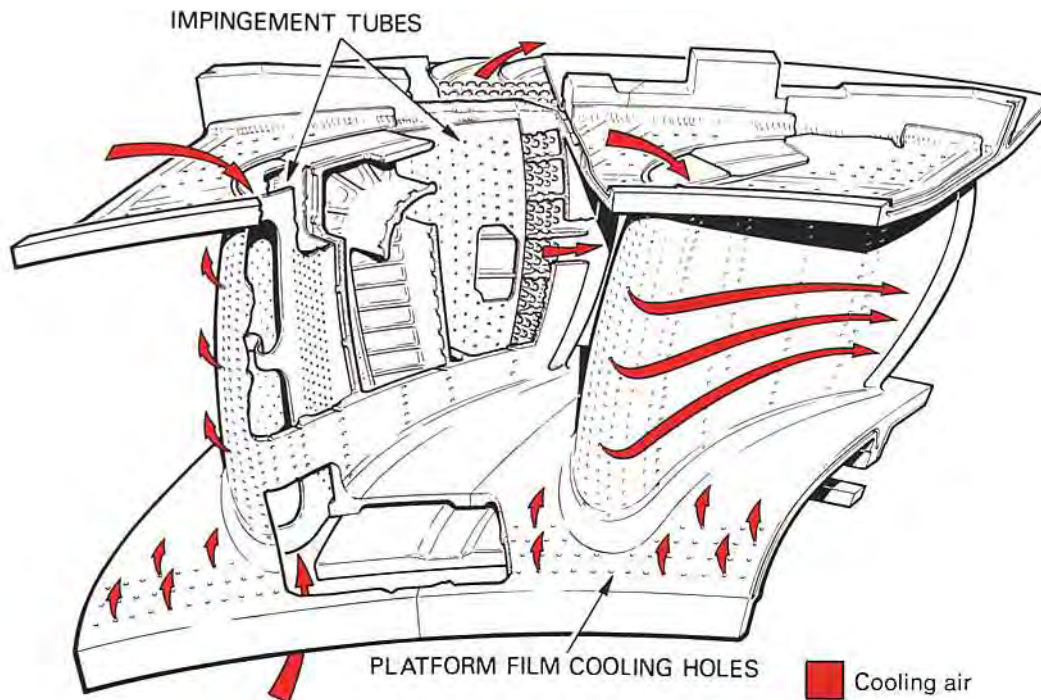
Fig. 9-4   High pressure nozzle guide vane construction and cooling.

Accessory cooling

10.   A considerable amount of heat is produced by some of the engine accessories, of which the electrical generator is an example, and these may often require their own cooling circuit. When air is used for cooling, the source may be the compressor or atmospheric air ducted from intake louvres in the engine cowlings.

11.   When an accessory unit is cooled during flight by atmospheric air it is usually necessary to provide an induced circuit for use during static ground running when there would be no external airflow. This is achieved by allowing compressor delivery air to pass through nozzles situated in the cooling air outlet duct of the accessory. The air velocity through the nozzles create a low pressure area which forms an ejector, so inducing a flow of atmospheric air through the intake louvres. To ensure that the ejector system only operates during ground running, the flow of air from the compressor is controlled by a valve. A generator cooling system with an ejector is shown in fig. 9-6.

SEALING

12.   Seals are used to prevent oil leakage from the engine bearing chambers, to control cooling airflows

and to prevent ingress of the mainstream gas into the turbine disc cavities.

13.   Various sealing methods are used on gas turbine engines. The choice of which method is dependent upon the surrounding temperature and pressure, wearability, heat generation, weight, space available, ease of manufacture and ease of installation and removal. Some of the sealing methods are described in the following paragraphs. A hypothetical turbine showing the usage of these seals is shown in fig. 9-5.

Labyrinth seals

14.   This type of seal is widely used to retain oil in bearing chambers and as a metering device to control internal airflows. Several variations of labyrinth seal design are shown in fig. 9-7.

15.   A labyrinth seal comprises a finned rotating member with a static bore which is lined with a soft abradable material, or a high temperature honeycomb structure. On initial running of the engine the fins lightly rub against the lining, cutting into it to give a minimum clearance. The clearance varies throughout the flight cycle, dependent upon the thermal growth of the parts and the natural flexing of the rotating members. Across each seal fin there is a pressure drop which results in a restricted flow of

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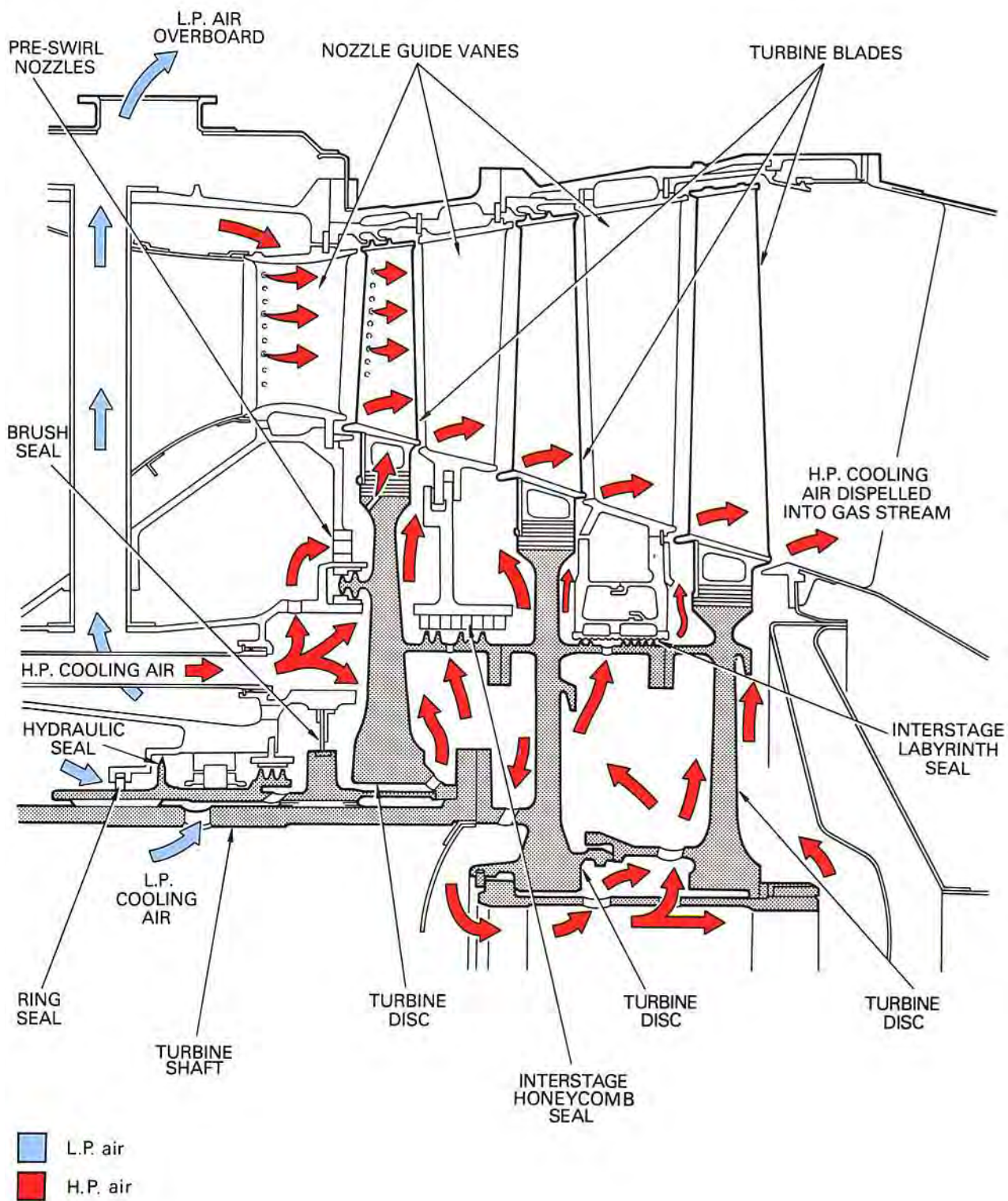
Fig. 9-5   A hypothetical turbine cooling and sealing arrangement.
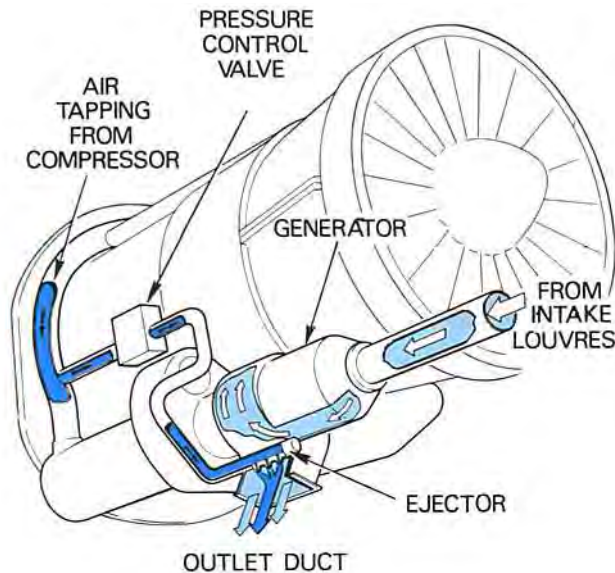
Fig. 9-6   A generator cooling system.

sealing air from one side of the seal to the other. When this seal is used for bearing chamber sealing, it prevents oil leakage by allowing the air to flow from the outside to the inside of the chamber. This flow also induces a positive pressure which assists the oil return system.

16. Seals between two rotating shafts are more likely to be subject to rubs between the fins and abradable material due to the two shafts deflecting simultaneously. This will create excessive heat which may result in shaft failure. To prevent this, a non-heat producing seal is used where the abradable lining is replaced by a rotating annulus of oil. When the shafts deflect, the fins enter the oil and maintain the seal without generating heat (fig. 9-7).

Ring seals
17. A ring seal (fig. 9-7) comprises a metal ring which is housed in a close fitting groove in the static housing. The normal running clearance between the ring and rotating shaft is smaller than that which can be obtained with the labyrinth seal. This is because the ring is allowed to move in its housing whenever the shaft comes into contact with it.

18. Ring seals are used for bearing chamber sealing, except in the hot areas where oil degradation due to heat would lead to ring seizure within its housing.

Hydraulic seals
19. This method of sealing is often used between two rotating members to sea a bearing chamber. Unlike the labyrinth or ring seal, it does not allow a controlled flow of air to traverse across the seal,

20. Hydraulic seals (fig. 9-7) are formed by a seal fin immersed in an annulus of oil which has been created by centrifugal forces. Any difference in air pressure inside and outside of the bearing chamber is compensated by a difference in oil level either side of the fin.

Carbon seals
21. Carbon seals (fig. 9-7) consist of a static ring of carbon which constantly rubs against a collar on a rotating shaft. Several springs are used to maintain contact between the carbon and the collar. This type of seal relies upon a high degree of contact and does not allow oil or air leakage across it. The heat caused by friction is dissipated by the oil system.

Brush seals
22. Brush seals (fig. 9-7) comprise a static ring of fine wire bristles. They are in continuous contact with a rotating shaft, rubbing against a hard ceramic coating. This type of seal has the advantage of withstanding radial rubs without increasing leakage.

Hot gas ingestion
23. It is important to prevent the ingestion of hot mainstream gas into the turbine disc cavities as this would cause overheating and result in unwanted thermal expansion and fatigue. The pressure in the turbine annulus forces the hot gas, between the rotating discs and the adjacent static parts, into the turbine disc rim spaces. In addition, air near the face of the rotating discs is accelerated by friction causing it to be pumped outwards. This induces a complementary inward flow of hot gas.

24. Prevention of hot gas ingestion is achieved by continuously supplying the required quantity of cooling and sealing air into the disc cavities to oppose the inward flow of hot gas. The flow and pressure of the cooling and sealing air is controlled by interstage seals (fig. 9-5),

CONTROL OF BEARING LOADS

25. Engine shafts experience varying axial gas loads (Part 20) which act in a forward direction on the compressor and in a rearward direction on the turbine. The shaft between them is therefore always under tension and the difference between the loads is carried by the location bearing which is fixed in a static casing (fig. 9-8). The internal air pressure acts

ABRADABLE LINING

ROTATING ANNULUS OF OIL

FLUID AND ABRADABLE LINED LABYRINTH SEAL

CONTINUOUS GROOVE INTERSTAGE (labyrinth) AIR SEAL

THREAD TYPE (labyrinth) OIL SEAL

RING TYPE OIL SEAL

INTERSHAFT HYDRAULIC SEAL

SPRING

CARBON

CARBON SEAL

CERAMIC COATING

BRUSH SEAL
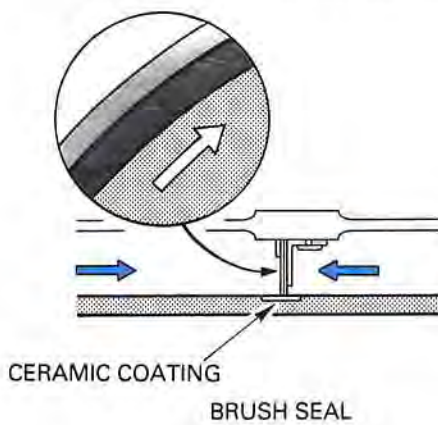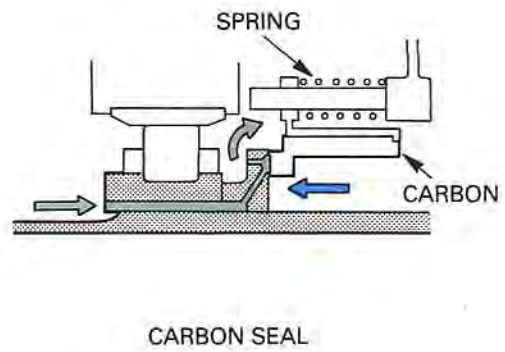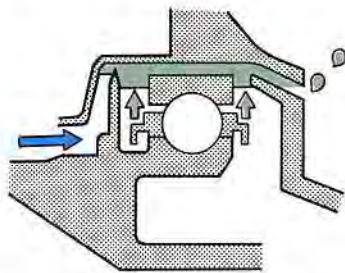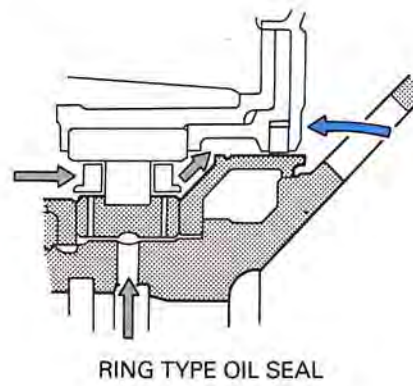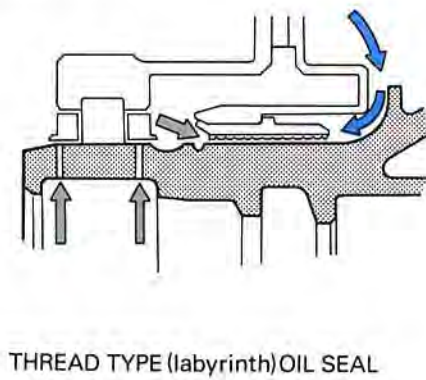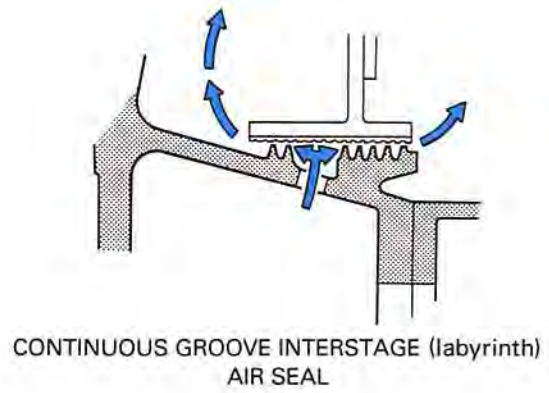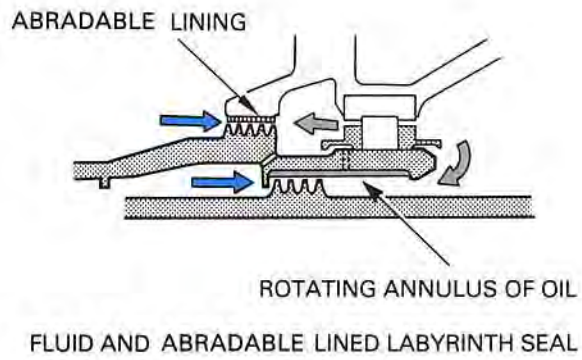
Sealing air

Oil

Rotating assemblies

Fig. 9-7   Typical seals.
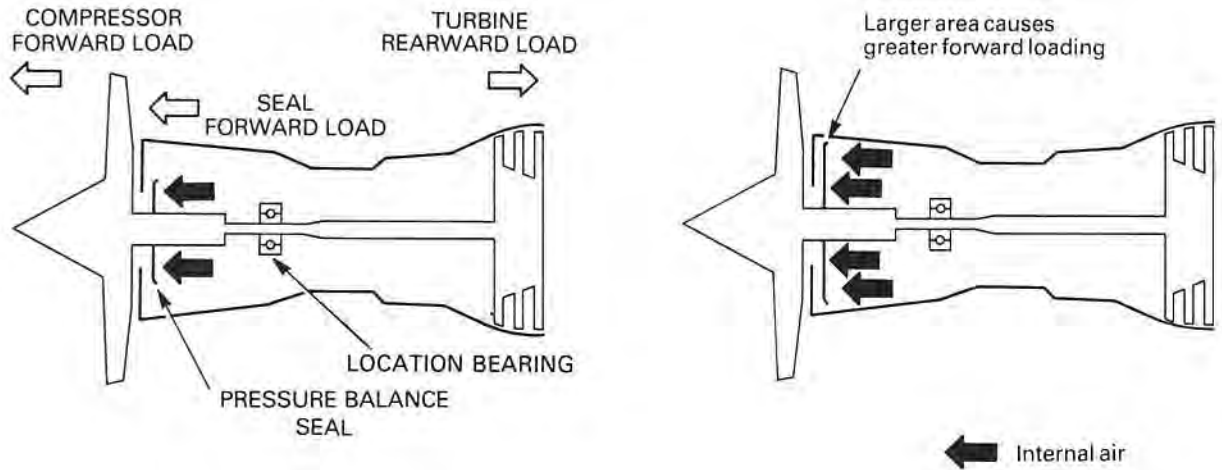
Fig. 9-8   Control of axial bearing load.

upon a fixed diameter pressure balance seal to ensure the location bearing is adequately loaded throughout the engine thrust range.

AIRCRAFT SERVICES

26. To provide cabin pressurization, airframe anti-icing and cabin heat, substantial quantities of air are bled from the compressor. It is desirable to bleed the air as early as possible from the compressor to minimize the effect on engine performance. However, during some phases of the flight cycle it may be necessary to switch the bleed source to a later compressor stage to maintain adequate pressure and temperature.

Rolls-Royce Gem 60
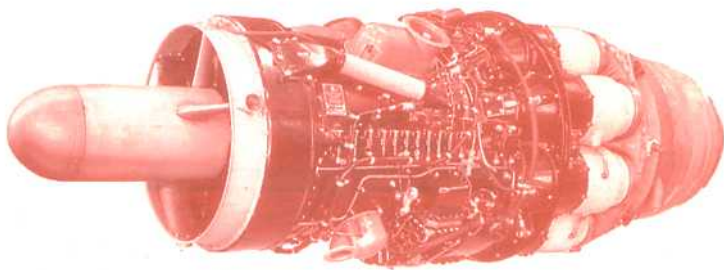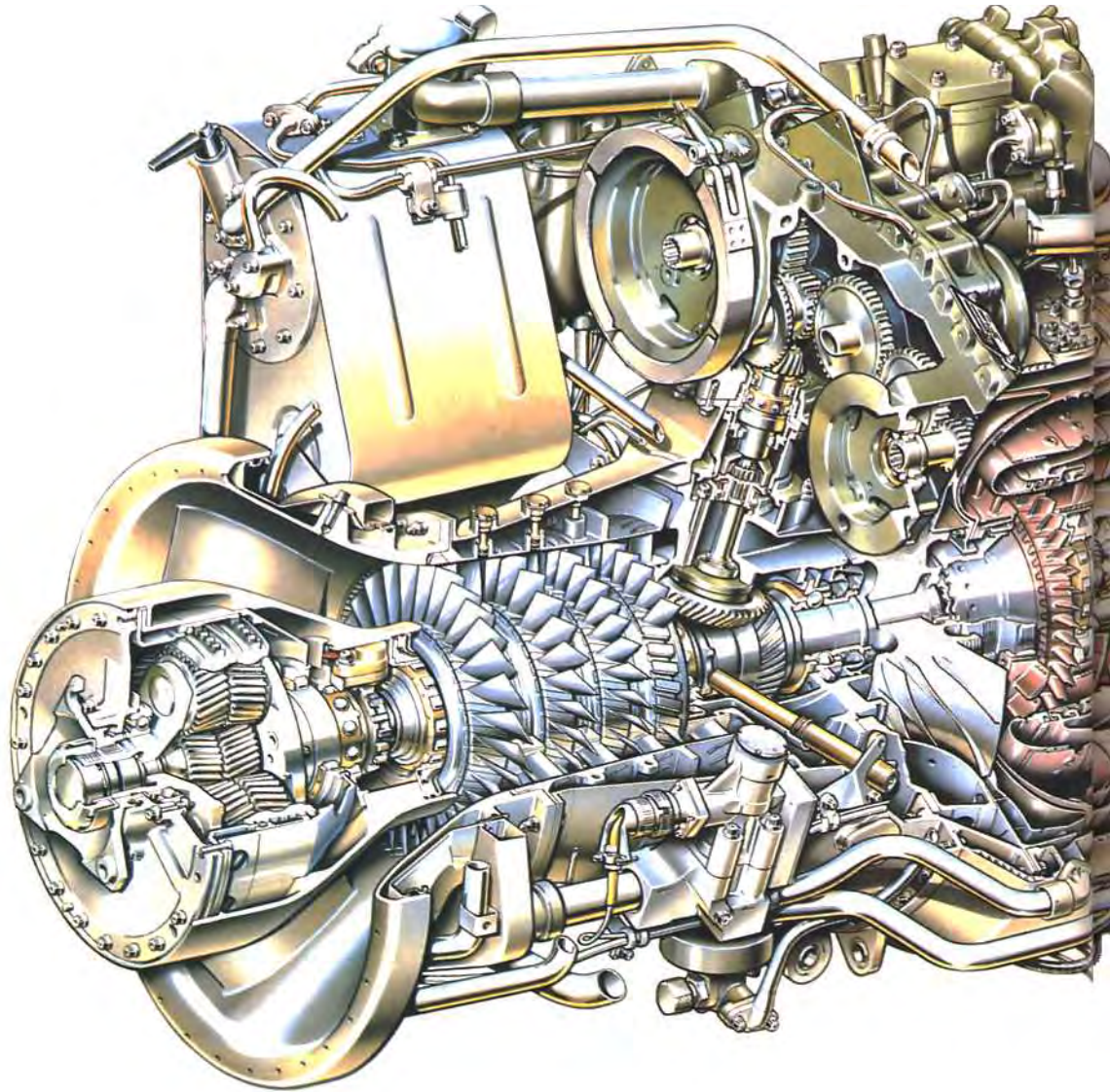
Rolls-Royce AJ65 Avon

Work commenced early in 1945 on the AJ65 axial flow turbo-jet with a design thrust of 6500 lb. This figure was reached in 1951 with the 100 series RA3. In 1953 the considerably redesigned 200 series RA14 was type tested at 9500 lb thrust. Development culminated in the 300 series RB146 which produced 17.110 lb thrust with afterburning.

10: Fuel system

Contents Page

INTRODUCTION

1. The functions of the fuel system are to provide the engine with fuel in a form suitable for combustion and to control the flow to the required quantity necessary for easy starting, acceleration and stable running, at all engine operating conditions. To do this, one or more fuel pumps are used to deliver the fuel to the fuel spray nozzles, which inject it into the combustion system (Part 4) in the form of an atomized spray. Because the flow rate must vary according to the amount of air passing through the engine to maintain a constant selected engine speed or pressure ratio, the controlling devices are fully automatic with the exception of engine power selection, which is achieved by a manual throttle or

power lever. A fuel shut-off valve (cock) control lever is also used to stop the engine, although in some instances these two manual controls are combined for single-lever operation.

2. It is also necessary to have automatic safety controls that prevent the engine gas temperature, compressor delivery pressure, and the rotating assembly speed, from exceeding their maximum limitations.

3. With the turbo-propeller engine, changes in propeller speed and pitch have to be taken into account due to their effect on the power output of the engine. Thus, it is usual to interconnect the throttle lever and propeller controller unit, for by so doing the correct relationship between fuel flow and airflow is maintained at all engine speeds and the pilot is given single-lever control of the engine. Although the maximum speed of the engine is normally determined by the propeller speed controller, over-speeding is ultimately prevented by a governor in the fuel system.

4. The fuel system often provides for ancillary functions, such as oil cooling (Part 8) and the hydraulic control of various engine control systems; for example, compressor airflow control (Part 3).

MANUAL AND AUTOMATIC CONTROL

5. The control of power or thrust of the gas turbine engine is effected by regulating the quantity of fuel injected into the combustion system. When a higher thrust is required, the throttle is opened and the pressure to the fuel spray nozzles increases due to the greater 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 a correspondingly greater airflow, consequently producing an increase in engine thrust.

6. This relationship between the airflow induced through the engine and the fuel supplied is, however, complicated by changes in altitude, air temperature and aircraft speed. These variables change the density of the air at the engine intake and consequently the mass of air induced through the engine. A typical change of airflow with altitude is shown in fig. 10-1. To meet this change in airflow a similar change in fuel flow (fig. 10-2) must occur, otherwise the ratio of airflow to fuel flow will change and will increase or decrease the engine speed from that originally selected by the throttle lever position.
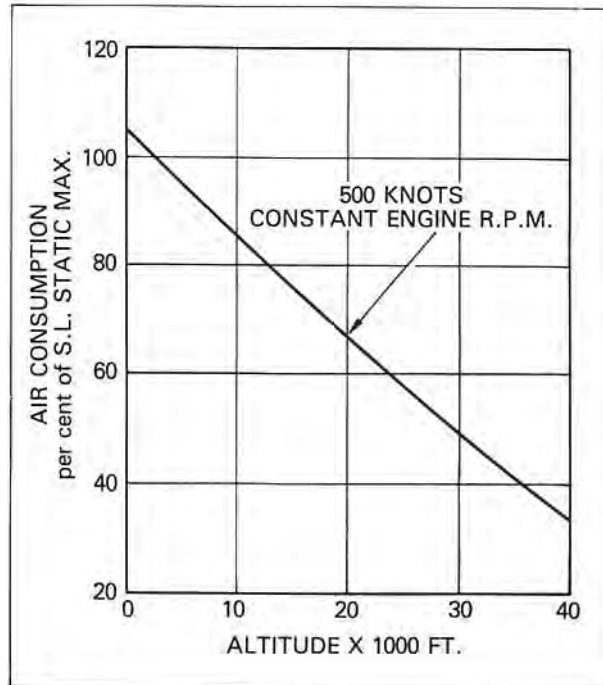
Fig. 10-1   Airflow changing with altitude.
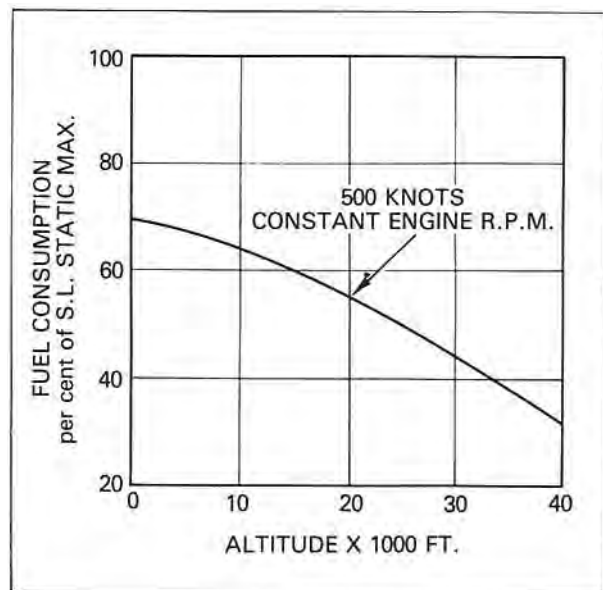
Fig. 10-2   Fuel flow changing with altitude.

7. Described in this Part are five representative systems of automatic fuel control; these are the pressure control and flow control systems, which are
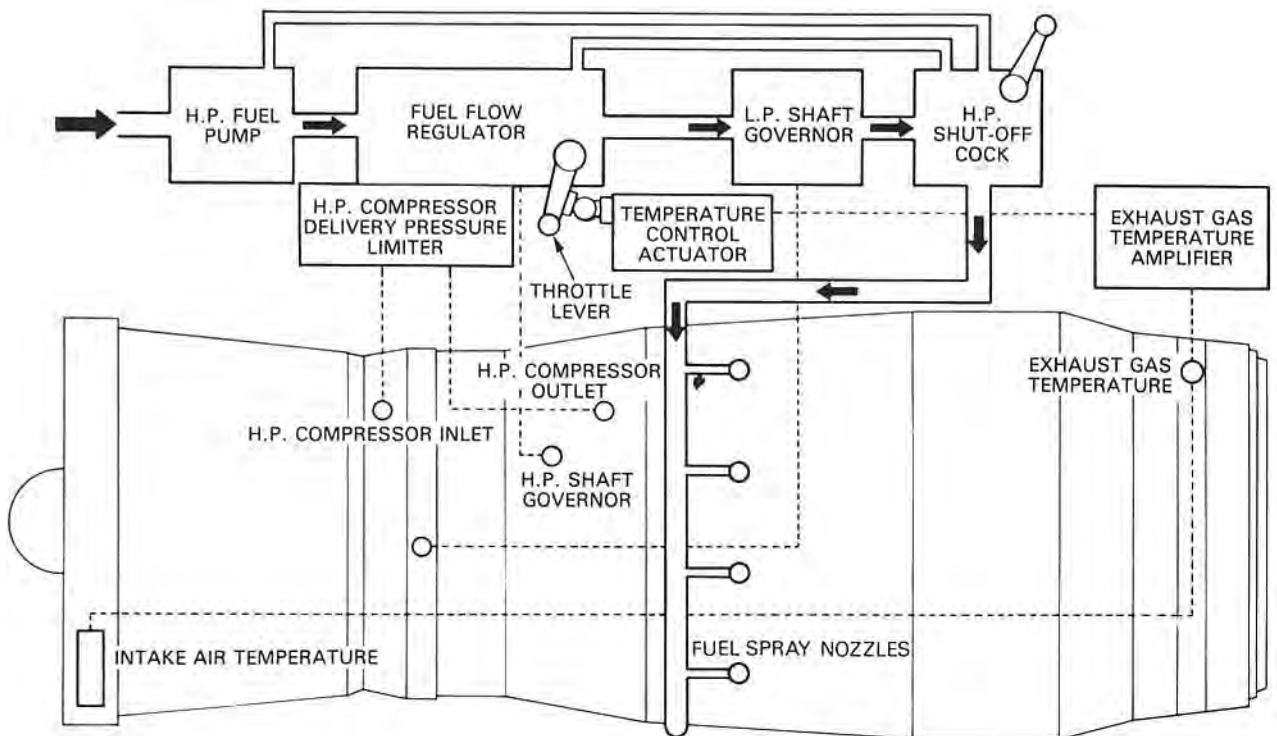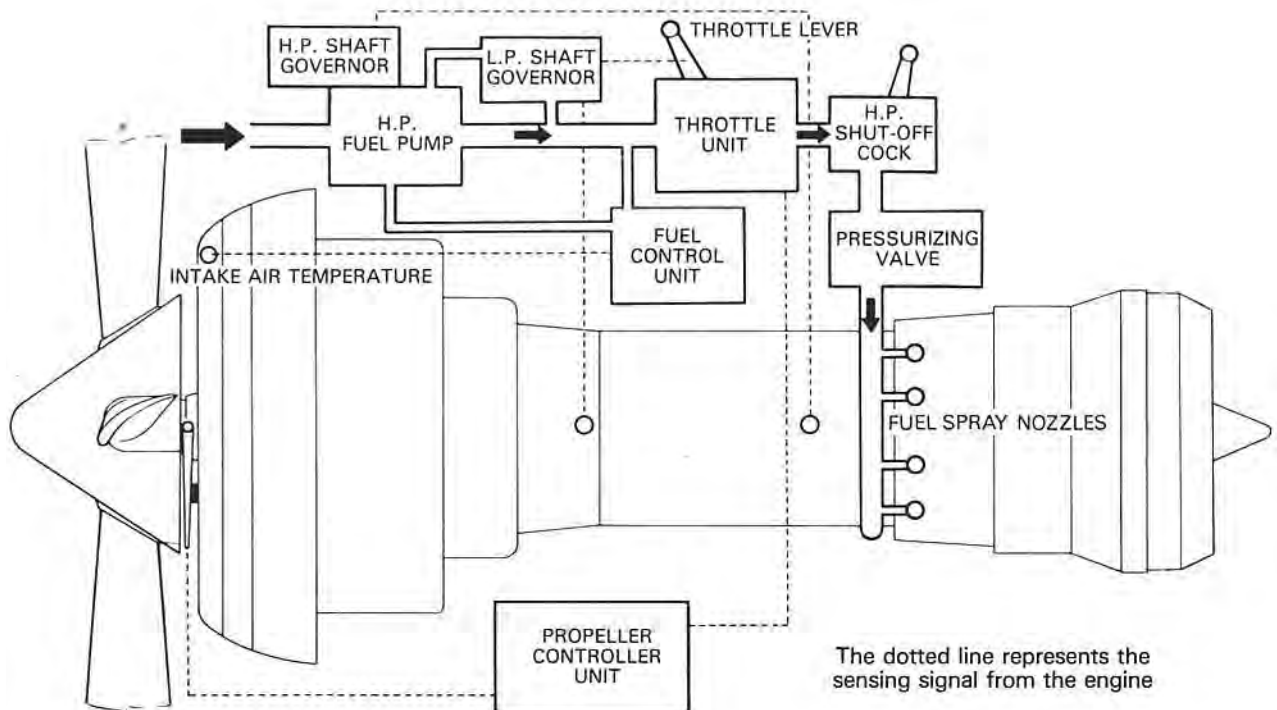
Fig. 10-3   Simplified fuel systems for turbo-propeller and turbo-jet engines.

FLOW CONTROL UNIT — AIR INTAKE PRESSURE

LOW PRESSURE COCK

SPILL VALVE

LOW PRESSURE FILTER

CONTROL PISTON

THROTTLE VALVE

THROTTLE BY-PASS ADJUSTER

HIGH PRESSURE COCK

SERVO PISTON

BACK PRESSURE VALVE

TO DIFFERENTIAL PRESSURE SWITCH

FUEL PUMP

SPILL VALVE

FUEL MANIFOLD

ENGINE SPEED GOVERNOR

L.P. fuel

Pump delivery (H.P. fuel)

Throttle outlet pressure

Spray nozzle pressure

Servo pressure

Governor pressure

Air intake pressure

Combustion chamber pressure

FUEL SPRAY NOZZLES

Fig. 10-4   A pressure control system (turbo-propeller engine).

hydro-mechanical, and the acceleration and speed control and pressure ratio control systems, which are mechanical. With the exception of the pressure ratio control system, which uses a gear-type pump, all the systems use a variable-stroke, multi-plunger type fuel pump to supply the fuel to the spray nozzles.

8. Some engines are fitted with an electronic system of control and this generally involves the use of electronic circuits to measure and translate changing engine conditions to automatically adjust the fuel pump output. On helicopters powered by gas turbine engines using the free-power turbine principle (Part 5), additional manual and automatic controls on the engine govern the free-power turbine and, consequently, aircraft rotor speed.

FUEL CONTROL SYSTEMS

9. Typical high pressure (H.P.) fuel control systems for a turbo-propeller engine and a turbo-jet engine are shown in simplified form in fig. 10-3, each basically consisting of an H.P. pump, a throttle control and a number of fuel spray nozzles. In addition, certain sensing devices are incorporated to provide automatic control of the fuel flow in response to engine requirements. On the turbo-propeller engine, the fuel and propeller systems are co-ordinated to produce the appropriate fuel/r.p.m. combination.

10. The usual method of varying the fuel flow to the spray nozzles is by adjusting the output of the H.P. fuel pump. This is effected through a servo system in response to some or all of the following:

 (1) Throttle movement.
 (2) Air temperature and pressure.
 (3) Rapid acceleration and deceleration.
 (4) Signals of engine speed, engine gas temperature and compressor delivery pressure.

Pressure control (turbo-propeller engine)
11. The pressure control system (fig. 10-4) is a typical system as fitted to a turbo-propeller engine where the rate of engine acceleration is restricted by a propeller speed controller. The fuel pump output is automatically controlled by spill valves in the flow control unit (F.C.U.) and the engine speed governor. These valves, by varying the fuel pump servo pressure, adjust the pump stroke to give the correct fuel flow to the engine.

12. At steady running conditions, at a given air intake pressure and below governed speed, the spill valve in the F.C.U. is in a sensitive position, creating

a balance of forces across the fuel pump servo piston and ensuring a steady pressure to the throttle valve.

13. When the throttle is slowly opened, the pressure to the throttle valve falls and allows the F.C.U. spill valve to close, so increasing the servo pressure and pump delivery. As the pressure to the throttle is restored, the spill valve returns to its sensitive or controlling position, and the fuel pump stabilizes its output to give the engine speed for the selected throttle position. The reverse sequence occurs as the throttle is closed.

14. A reduction of air intake pressure, due to a reduction of aircraft forward speed or increase in altitude, causes the F.C.U. capsule to expand, thus increasing the bleed from the F.C.U. spill valve. This reduces fuel pump delivery until the fuel flow matches the airflow and the reduced H.P. pump delivery (throttle inlet pressure), allows the spill valve to return to its sensitive position. Conversely, an increase in air intake pressure reduces the bleed from the spill valve and increases the fuel flow. The compensation for changes in air intake pressure is such that fuel flow cannot be increased beyond the pre-determined maximum permissible for static International Standard Atmosphere (I.S.A.) sea-level conditions.

15. The engine speed governor prevents the engine from exceeding its maximum speed limitation. With increasing engine speed, the centrifugal pressure from the fuel pump rotor radial drillings increases and this is sensed by the engine speed governor diaphragm. When the engine reaches its speed limitation, the diaphragm is deflected to open the governor spill valve, thus overriding the F.C.U. and preventing any further increase in fuel flow. Some pressure control systems employ a hydro-mechanical governor (para. 23).

16. The governor spill valve also acts as a safety relief valve. If the fuel pump delivery pressure exceeds its maximum controlling value, the servo pressure acting on the orifice area of the spill valve forces the valve open regardless of the engine speed, so preventing any further increase in fuel delivery pressure.

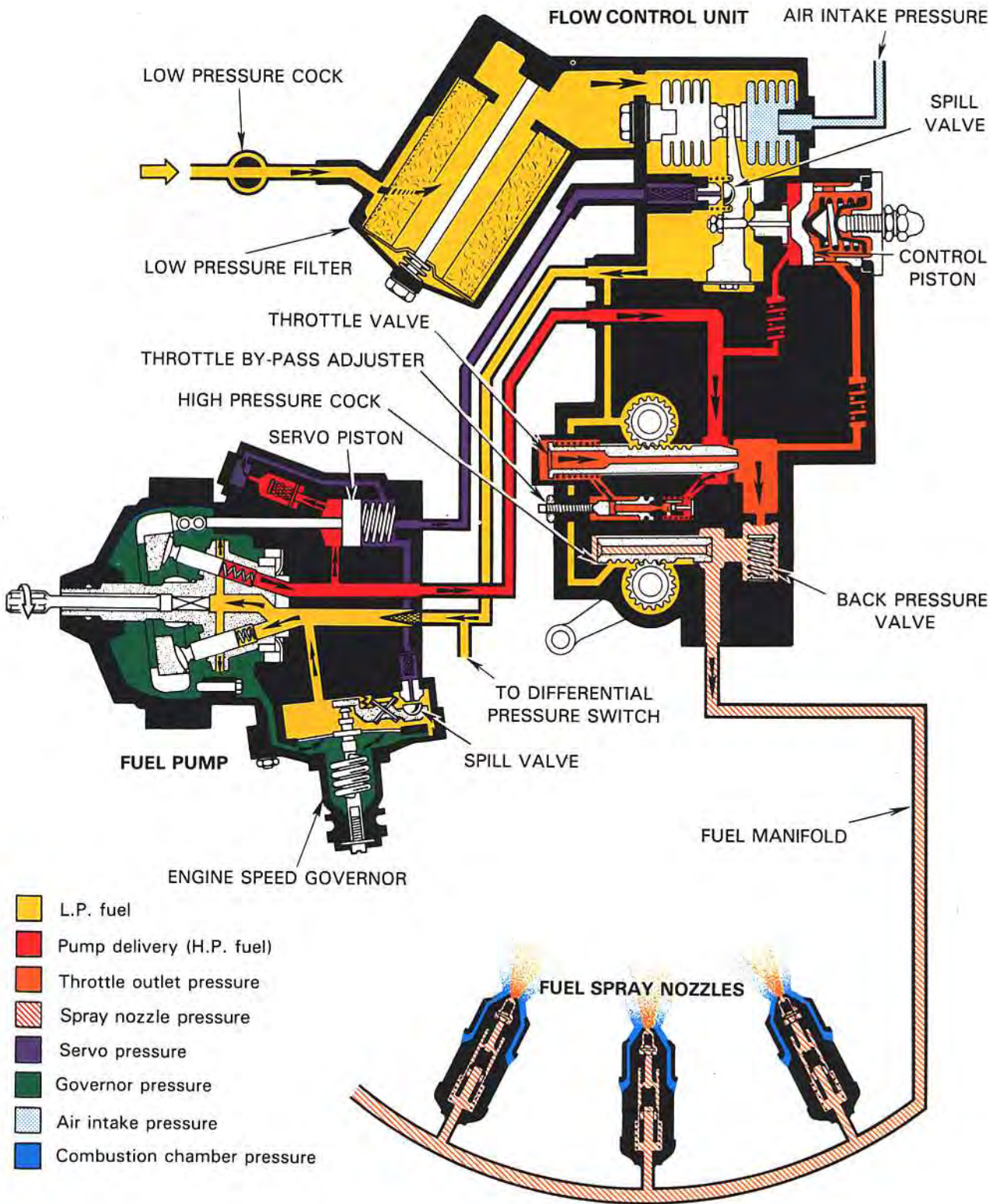### Pressure control (turbo-jet engine)
17. In the pressure control system illustrated in fig. 10-5, the rate of engine acceleration is controlled by a dashpot throttle unit. The unit forms part of the fuel control unit and consists of a servo-operated throttle, which moves in a ported sleeve, and a control valve.
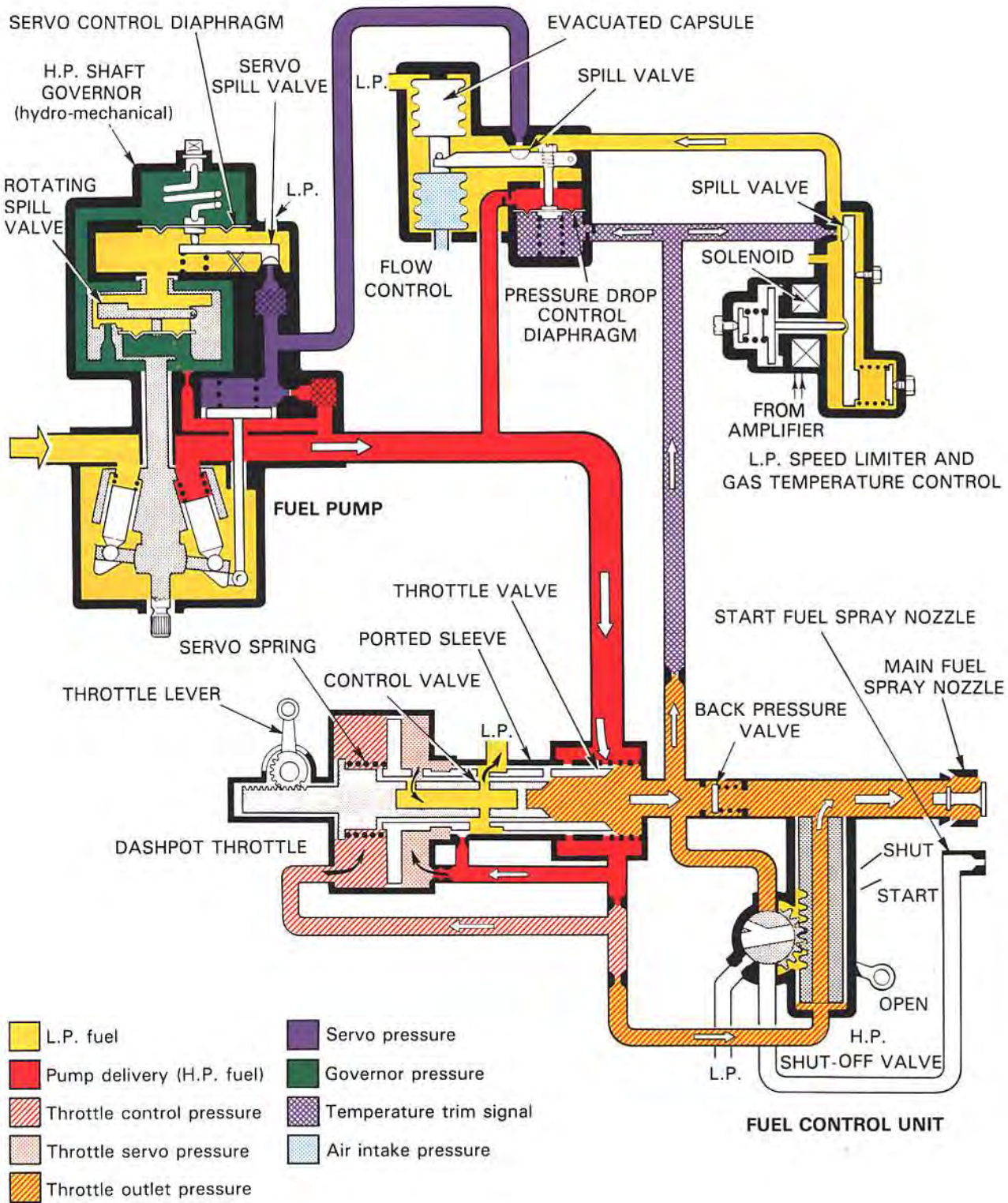
Fig. 10-5   A pressure control system (turbo-jet engine).

The control valve slides freely within the bore of the throttle valve and is linked to the pilot's throttle by a rack and pinion mechanism. Movement of the throttle lever causes the throttle valve to progressively uncover ports in the sleeve and thus increase the fuel flow. Fig. 10-6 shows the throttle valve and control valve in their various controlling positions.

18. At steady running conditions, the dashpot throttle valve is held in equilibrium by throttle servo pressure opposed by throttle control pressure plus spring force. The pressures across the pressure drop control diaphragm are in balance and the pump servo pressure adjusts the fuel pump to give a constant fuel flow.

19. When the throttle is opened, the control valve closes the low pressure (L.P.) fuel port in the sleeve and the throttle servo pressure increases. The throttle valve moves towards the selected throttle position until the L.P. port opens and the pressure balance across the throttle valve is restored. The decreasing fuel pressure difference across the throttle valve is sensed by the pressure drop control diaphragm, which closes the spill valve to increase the pump servo pressure and therefore the pump output. The spill valve moves into the sensitive position, controlling the pump servo mechanism so that the correct fuel flow is maintained for the selected throttle position.

20. During initial acceleration, fuel control is as described in para. 19; however, at a predetermined throttle position the engine can accept more fuel and at this point the throttle valve uncovers an annulus, so introducing extra fuel at a higher pressure (pump delivery through one restrictor). This extra fuel further increases the throttle servo pressure, which increases the speed of throttle valve travel and the rate of fuel supply to the spray nozzle.

21. On deceleration, movement of the control valve acts directly on the throttle valve through the servo spring. Control valve movement opens the flow ports through the control valve and throttle valve, to bleed servo fuel through the L.P. port. Throttle control pressure then moves the throttle valve towards the closed position, thus reducing the fuel flow to the spray nozzles.

22. Changes in air intake pressure, due to a change in aircraft altitude or forward speed, are sensed by the capsule assembly in the fuel control unit. With increased altitude and a corresponding decrease in air intake pressure, the evacuated capsule opens the spill valve, so causing a reduction in pump stroke
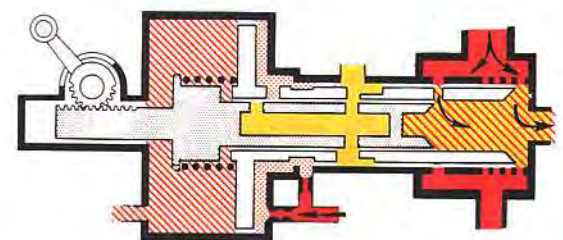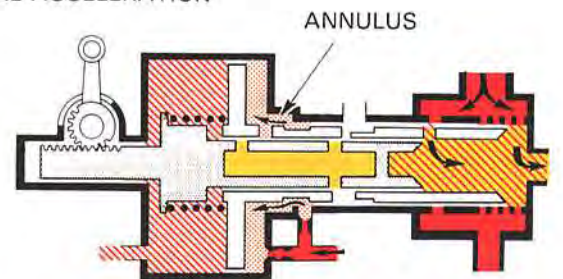
Fig. 10-6 Acceleration control by dashpot throttle.
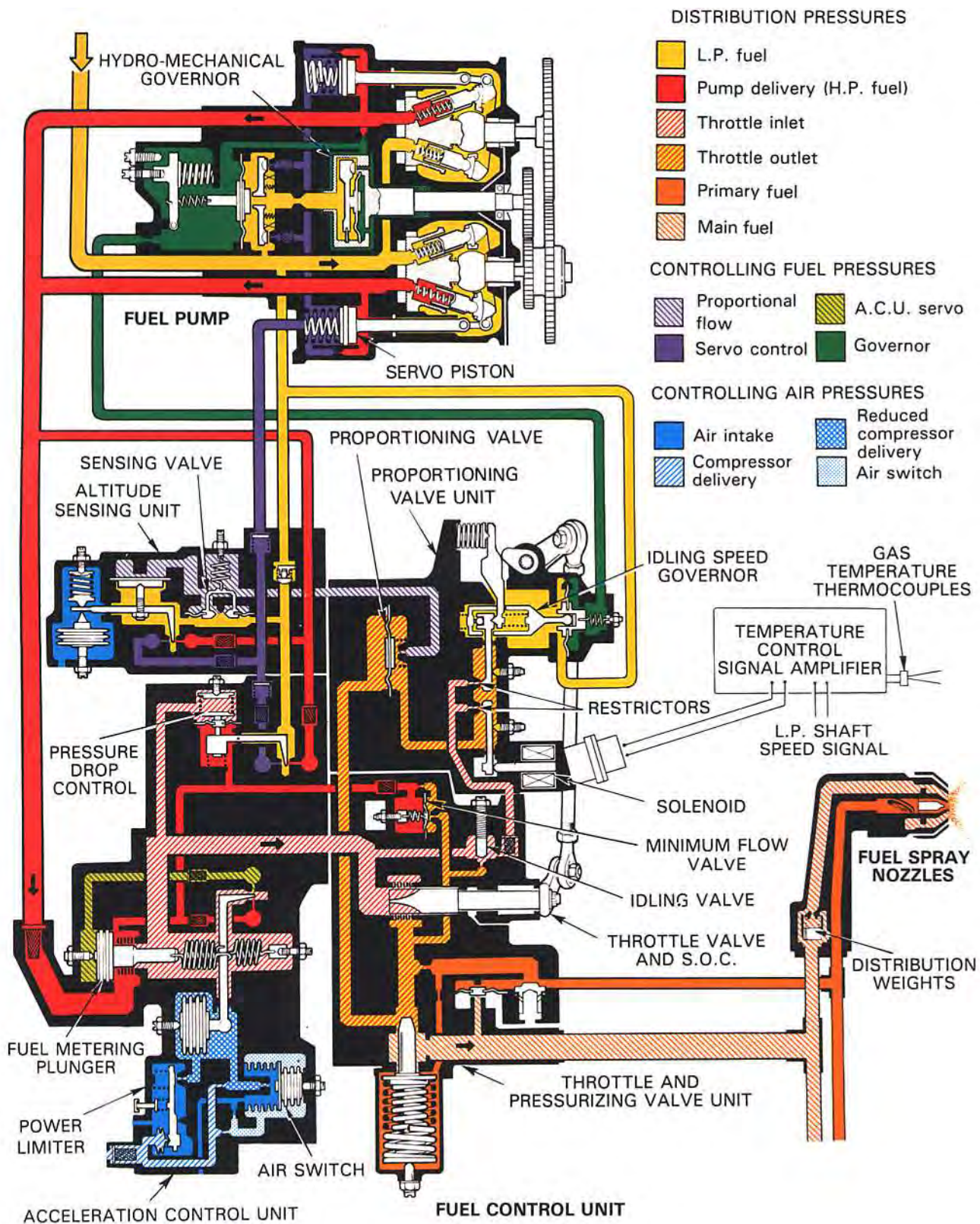
Fig. 10-7 A proportional flow control system.

until the fuel flow matches the airflow. Conversely, an increase in air intake pressure closes the spill valve to increase the fuel flow.

23. H.P. compressor shaft r.p.m. is governed by a hydro-mechanical governor which uses hydraulic pressure proportional to engine speed as its controlling parameter. A rotating spill valve senses the engine speed and the controlling pressure is used to limit the pump stroke and so prevent over-speeding of the H.P. shaft rotating assembly. The controlling pressure is unaffected by changes in fuel specific gravity.

24. At low H.P. shaft speeds, the rotating spill valve is held open, but as engine speed increases, centrifugal loading moves the valve towards the closed position against the diaphragm loads. This restricts the bleed of fuel to the L.P. side of the valve until, at governed speed, the governor pressure deflects the servo control diaphragm and opens the servo spill valve to control the fuel flow and thereby the H.P. shaft speed.

25. If the engine gas temperature attempts to exceed the maximum limitation, the current in the L.P. speed limiter and temperature control solenoid is reduced. This opens the spill valve to reduce the pressure on the pressure drop control diaphragm. The flow control spill valve then opens to reduce the pump servo pressure and fuel pump output.

26. To prevent the L.P. compressor from over-speeding, multi-spool engines usually have an L.P. compressor shaft speed governor. A signal of L.P. shaft speed and intake temperature is fed to an amplifier and solenoid valve, the valve limiting the fuel flow in the same way as the gas temperature control (para. 25).

27. The system described uses main and starting spray nozzles under the control of an H.P. shut-off valve. Two starting nozzles are fitted in the combustion chamber, each being forward of an igniter plug. When the engine has started, the fuel flow to these nozzles is cut off by the H.P. shut-off valve.

28. To ensure that a satisfactory fuel pressure to the spray nozzles is maintained at high altitudes, a back pressure valve, located downstream of the throttle valve, raises the pressure levels sufficiently to ensure satisfactory operation of the fuel pump servo system.

Flow control
29. A flow control fuel system is generally more compact than a pressure control system and is not sensitive to flow effect of variations downstream of the throttle. The fuel pump delivery pressure is related to engine speed; thus, at low engine speeds pump delivery pressure is quite low. The fuel pump output is controlled to give a constant pressure difference across the throttle valve at a constant air intake condition. Various devices are also used to adjust the fuel flow for air intake pressure variations, idling and acceleration control, gas temperature and compressor delivery pressure control.

30. A variation of the flow control system is the proportional flow control system (fig 10-7), which is more suitable for engines requiring large fuel flows and which also enables the fuel trimming devices to adjust the fuel flow more accurately. A small controlling flow is created that has the same characteristics as the main flow, and this controlling or proportional flow is used to adjust the main flow.

31. A different type of spill valve, referred to as a kinetic valve, is used in this system. This valve consists of two opposing jets, one subjected to pump delivery pressure and the other to pump servo pressure, and an interrupter blade that can be moved between the jets (fig. 10-8). When the blade is clear of the jets, the kinetic force of the H.P. fuel jet causes the servo pressure to rise (spill valve closed) and the fuel pump moves to maximum stroke to increase the fuel flow. When the blade is lowered between the jets, the pressure jet is deflected and the servo pressure falls, so reducing the pump stroke and the fuel flow, When the engine is steadily running, the blade is in an intermediate position allowing a slow bleed from servo and thus balancing the fuel pump output.

32. All the controlling devices, except for the engine speed governor, are contained in one combined fuel control unit. The main parts of the control unit are the altitude sensing unit (A.S.U.), the acceleration control unit (A.C.U.), the throttle and pressurizing valve unit, and the proportioning valve unit.

33. At any steady running condition below governed speed, the fuel pump delivery is controlled to a fixed value by the A.S.U. The spill valve in this unit is held in the controlling position by a balance of forces, spring force and the piston force. The piston is sensitive to the pressure difference across the sensing valve, the pressure difference being created by fuel flowing from the proportioning valve back to the fuel pump inlet.

VALVE OPEN
(Pump output decreasing)

VALVE CLOSED
(Pump output increasing)

VALVE INTERMEDIATE
(Pump output constant)

H.P. fuel   Servo
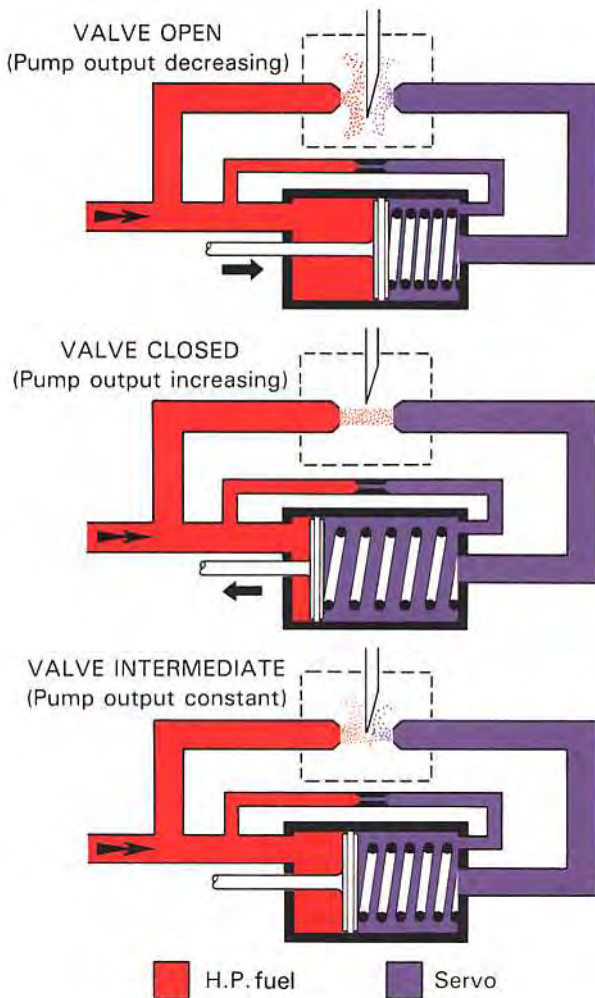
Fig. 10-8 Servo pressure control by kinetic valve.

34. The proportioning valve diaphragm is held open in a balanced condition allowing fuel to pass to the A.S.U. This means that the restrictor outlet pressure is equal to the throttle outlet pressure and, as their inlet pressures are equal, it follows that the pressure difference across the restrictors and the throttle are equal; therefore, a constant fuel flow is obtained.

35. When the throttle is slowly opened, the pressure difference across the throttle valve and the proportioning flow restrictors decreases and the pro-portioning valve diaphragm adjusts its position. This reduces the proportional flow, which closes the A.S.U. spill valve and increases the servo pressure. The fuel pump increases its delivery and this restores the pressure difference across the throttle valve and

equalizes the pressure difference across the restrictors. The proportional flow is restored to its original value and the balance of forces in the A.S.U. returns the spill valve to the controlling position.

36. A variation of air intake pressure, due to a change of aircraft forward speed or altitude, is sensed by the capsule in the A.S.U. A pressure reduction causes the A.S.U. capsule to expand, thus increasing the bleed from the spill valve. This reduces fuel pump delivery until the fuel flow matches the airflow and results in a lower pressure difference across the throttle valve and the propor-tioning valve restrictors. The reduced proportional flow restores the balance in the A.S.U. which returns the spill valve to its controlling position. Conversely, an increase in aircraft forward speed increases the air Intake pressure, which reduces the bleed from the spill valve and increases the fuel flow.

37. During a rapid acceleration, the sudden decrease in throttle pressure difference is sensed by the A.S.U., causing the spill valve to close, Such a rapid increase in fuel supply would, however, create an excessive gas temperature and also cause the compressor to surge (Part 3). This occurs because the inertia of the rotating assembly results in an appreciable time lag in the rate of airflow increase. It is essential therefore, to have an acceleration control to override the A.S.U. to give a corresponding lag in the rate of fuel flow increase.

38. The rapid initial increase of fuel flow causes a rise in the pressure difference across the fuel metering plunger and this is sensed by a diaphragm in the pressure drop control section. At a fixed value of over fuelling, the pressure drop control diaphragm opens its servo spill valve to override the A.S.U, and maintains a constant pressure difference across the metering plunger.

39. The increased fuel supply causes the engine to accelerate and the fuel metering plunger gives the maximum permissible fuel flow to match the increasing compressor delivery pressure. This it achieves through the A.C.U. servo system, which is under the control of a spill valve operated by compressor delivery air pressure acting on a capsule.

40. As the compressor delivery pressure continues to rise, the capsule is compressed to open the spill valve and to bleed pressure from above the metering plunger. Pump delivery pressure acting underneath the plunger causes it to lift, this increases the area of the main fuel flow passage.

41. The pressure drop control spill valve closes to increase the fuel pump delivery and maintains the controlling pressure difference across the plunger. The fuel flow, therefore, progressively rises as airflow through the compressor increases. The degree of overfuelling can be automatically changed by the air switch, which increases the pressure signal on to the capsule. The full value of compressor delivery pressure is now passed on to the A.C.U. capsule assembly, thus increasing the opening rate of the metering plunger.

42. As the controlled overfuelling continues, the pressure difference across the throttle valve increases. When it reaches the controlling value, the A.S.U. takes over due to the increasing proportional flow and again gives a steady fuel flow to the spray nozzles.

43. The engine speed governor can be of the pressure control type described in para. 15, or a hydro-mechanical governor as described in para. 23.

44. The control of servo pressure by the hydro-mechanical governor is very similar to that of the pressure control governor, except that the governor pressure is obtained from pump delivery fuel passing through a restrictor and the restricted pressure is controlled by a rotating spill valve; this type of governor is unaffected by changes in fuel specific gravity.

45. At low engine speeds, the rotating spill valve is held open; however, as engine speed increases, centrifugal loading moves the valve towards the closed position against the diaphragm loads. This restricts the bleed of H.P. fuel to the L.P. side of the drum until, at governed speed, the governor pressure deflects the diaphragm and opens the fuel pump servo pressure spill valve to control the maximum fuel flow and engine speed.

46. If the engine gas temperature exceeds its maximum limitation, the solenoid on the proportioning valve unit is progressively energized. This causes a movement of the rocker arm to increase the effective flow area of one restrictor, thus increasing the proportional flow and opening the A.S.U. spill valve to reduce servo pressure. The fuel flow is thus reduced and any further increase of gas temperature is prevented.

47. To prevent the L.P. compressor from over-speeding, some twin-spool engines have an L.P. shaft r.p.m. governor. A signal of L.P. shaft speed is fed to an amplifier and solenoid valve, which limits

the fuel output in the same way as the gas temperature control.

48. An idling speed governor is often fitted to ensure that the idling r.p.m. does not vary with changing engine loads. A variation of idling r.p.m. causes the rocker arm to move and alter the proportional flow, and the A.S.U. adjusts the pump delivery until the correct idling r.p.m. is restored.

49. On some engines, a power limiter is used to prevent overstressing of the engine. To achieve this, compressor delivery pressure acts on the power limiter capsule. Excess pressure opens the power limiter atmospheric bleed to limit the pressure on the A.C.U. capsule and this controls the fuel flow through the metering plunger.

50. To enable the engine to be relit and to prevent flame-out at altitude, the engine idling r.p.m. is made to increase with altitude. To achieve this, some engines incorporate a minimum flow valve that adds a constant minimum fuel flow to that passing through the throttle valve.

Combined acceleration and speed control

51. The combined acceleration and speed control system (fig. 10-9) is a mechanical system without small restrictors or spill valves. It is also an all-speed governor system and therefore needs no separate governor unit for controlling the maximum r.p.m. The controlling mechanism is contained in one unit, usually referred to as the fuel flow regulator (F.F.R.). An H.P. fuel pump (para. 85) is used and the fuel pump servo piston is operated by H.P. fuel on one side and main spray nozzle (servo) pressure on the spring side.

52. The F.F.R. is driven by the engine through a gear train and has two centrifugal governors, known as the speed control governor and the pressure drop control governor. Two sliding valves are also rotated by the gear train. One valve, known as the variable metering sleeve, has a triangular orifice, known as the variable metering orifice (V.M.O.), and this sleeve is given axial movement by a capsule assembly. The V.M.O. sleeve moves inside a non-rotating governor sleeve that is moved axially by the speed control governor. The other valve, known as the pressure drop control valve, is provided with axial movement by the pressure drop control governor and has a triangular orifice, known as the pressure drop control orifice, and a fixed-area rectangular orifice. The speed control governor is set by the throttle lever through a cam, a spring and a stirrup arm inside the regulator.
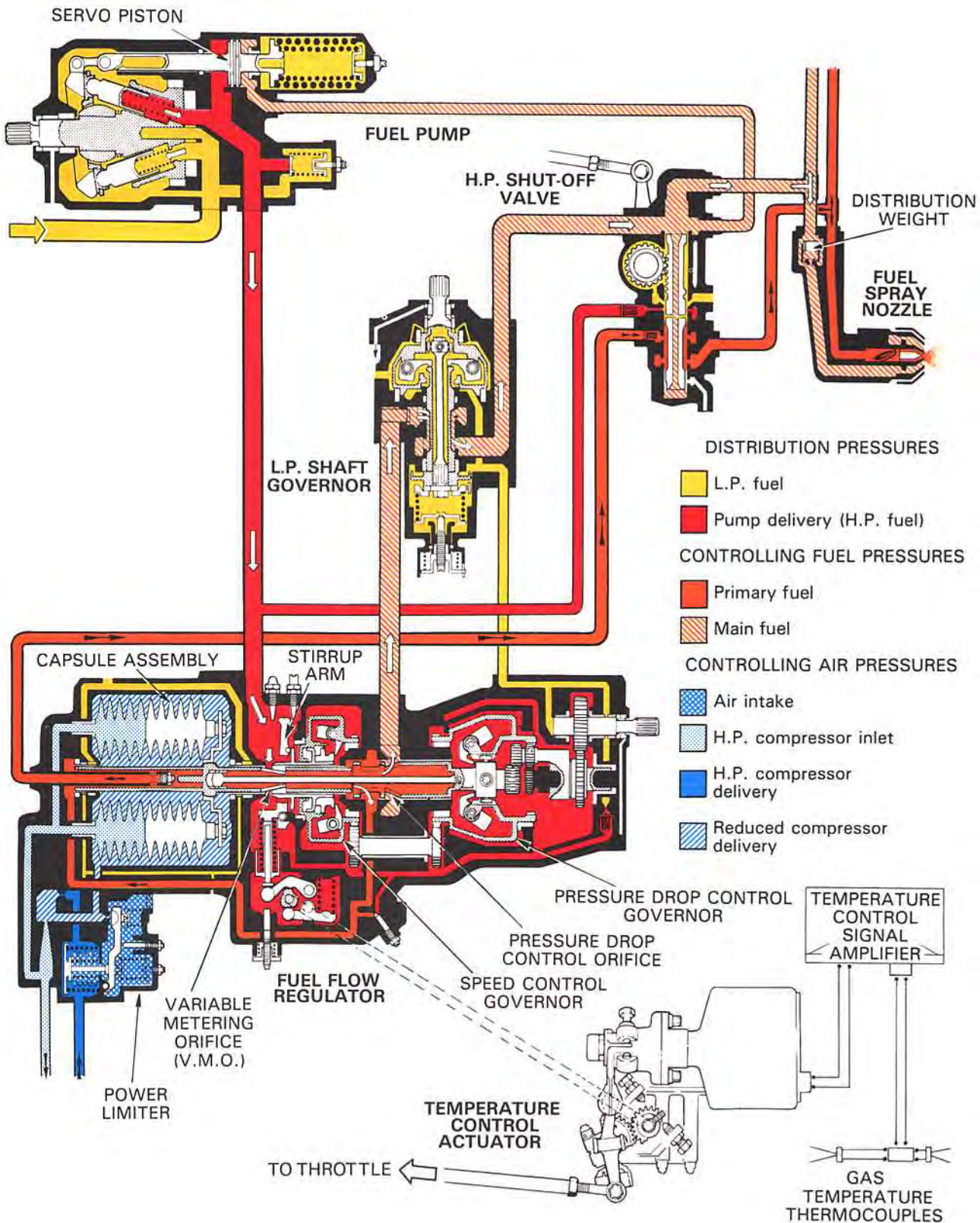
Fig. 10-9   A combined acceleration and speed control system.

SERVO PISTON

FUEL PUMP

H.P. SHUT-OFF VALVE

DISTRIBUTION WEIGHT

FUEL SPRAY NOZZLE

L.P. SHAFT GOVERNOR

DISTRIBUTION PRESSURES

L.P. fuel

Pump delivery (H.P. fuel)

CONTROLLING FUEL PRESSURES

Primary fuel

Main fuel

CONTROLLING AIR PRESSURES

Air intake

H.P. compressor inlet

H.P. compressor delivery

Reduced compressor delivery

CAPSULE ASSEMBLY

STIRRUP ARM

PRESSURE DROP CONTROL GOVERNOR

TEMPERATURE CONTROL SIGNAL AMPLIFIER

PRESSURE DROP CONTROL ORIFICE

FUEL FLOW REGULATOR

SPEED CONTROL GOVERNOR

VARIABLE METERING ORIFICE (V.M.O.)

POWER LIMITER

TEMPERATURE CONTROL ACTUATOR

TO THROTTLE

GAS TEMPERATURE THERMOCOUPLES

53. At any steady running condition, the engine speed is governed by the regulator controlling the fuel flow. The fuel pump delivery is fixed at a constant value by applying the system pressure difference to the fuel pump servo piston. This is arranged to balance the servo piston spring forces.

54. When the air intake pressure is at a constant value, the rotating V.M.O. sleeve is held in a fixed axial position by the capsule loading. The fixed throttle setting maintains a set load on the speed control governor and, as the r.p.m. is constant, the governor sleeve is held in a fixed position.

55. The fuel pump delivery is passed to the annulus surrounding the V.M.O.; the annulus area is controlled by the governor sleeve, and the exposed area of the orifice is set by the axial position of the V.M.O. sleeve. Consequently, fuel passes to the inside of the sleeve at a constant flow and therefore at a constant pressure difference.

56. The pressure drop control valve, which also forms a piston, senses the pressure difference across the V.M.O. and maintains the fuel flow at a fixed value in relation to a function of engine speed, by controlling the exposed area of the pressure drop control orifice.

57. When the throttle is slowly opened, the load on the speed control governor is increased, so moving the governor sleeve to increase the V.M.O. annulus area. The effect of opening the V.M.O. is to reduce the pressure difference and this is sensed by the pressure drop control governor, which opens the pressure drop valve. The reduced system pressure difference is immediately sensed by the fuel pump servo piston, which increases the pump stroke and consequently the fuel output. The increased compressor delivery pressure acts on the capsule assembly, which gradually opens the V.M.O. so that the fuel flow and engine speed continue to increase. At the speed selected, centrifugal forces acting on the speed control governor move the governor sleeve to reduce the V.M.O. annulus area. The resultant increased pressure difference is sensed by the pressure drop control governor, which adjusts the pressure drop valve to a point at which the pump servo system gives an output to match the engine requirements. The function of the governors and the control of the fuel flow is shown diagrammatically in fig. 10-10.

58. During a rapid acceleration, the initial degree of overselling is mechanically controlled by a stop that limits the opening movement of the speed control governor sleeve. A similar stop also prevents the fuel supply from being completely cut off by the governor sleeve during a rapid deceleration.

59. Changes in altitude or forward speed of the aircraft vary the fuel flow required to maintain a constant engine speed. To provide this control, the capsule assembly senses changes in H.P. compressor inlet and delivery pressures and adjusts the V.M.O. accordingly. For instance, as the aircraft altitude increases, the compressor delivery pressure falls and the capsule assembly expands to reduce the V.M.O. The increased system pressure drop is sensed by the fuel pump servo piston, which adjusts the pump output to match the reduced airflow and so maintain a constant engine speed. Conversely, an increase in aircraft forward speed causes the capsule assembly to be compressed and increase the V.M.O. The reduced system pressure drop causes the fuel pump to increase its output to match the increased airflow.

60. To prevent the maximum gas temperature from being exceeded, fuel flow is reduced in response to signals from thermocouples sensing the temperature (Part 12). When the maximum temperature is reached, the signals are amplified and passed to a rotary actuator which adjusts the throttle mechanism. This movement has the same effect on fuel flow as manual operation of the throttle.

61. To ensure that the engine is not overstressed, the H.P. compressor delivery pressure is controlled to a predetermined value. At this value, a pressure limiting device, known as a power limiter, reduces the pressure in the capsule chamber, thus allowing the capsule assembly to expand and reduce the V.M.O. so preventing any further increase in fuel flow.

62. A governor prevents the L.P. compressor shaft from exceeding its operating limitations and also acts as a maximum speed governor in an event of a failure of the F.F.R. The governor provides a variable restrictor between the regulator and the main fuel spray nozzle manifold. Should the L.P. compressor reach its speed limitation, flyweights in the governor move a sleeve valve to reduce the flow area, The increased system pressure drop is sensed by the fuel pump servo piston, which reduces the fuel flow to the spray nozzles.

63. This fuel system has no pressurizing valve to divide the flow from the fuel pump into main and primary fuel flows. Primary fuel pressure is taken from the fixed-area orifice of the pressure drop control valve. This pressure is always higher than the

THROTTLE

SPEED CONTROL GOVERNOR

OPEN CLOSE

PRESSURE DROP CONTROL GOVERNOR

VARIABLE METERING ORIFICE (V.M.O.)

PRESSURE DROP CONTROL ORIFICE

REGULATOR

CAPSULE ASSEMBLY

GOVERNOR SLEEVE

PRESSURE DROP CONTROL VALVE

TO FUEL SPRAY NOZZLES

FUEL PUMP

SERVO PISTON
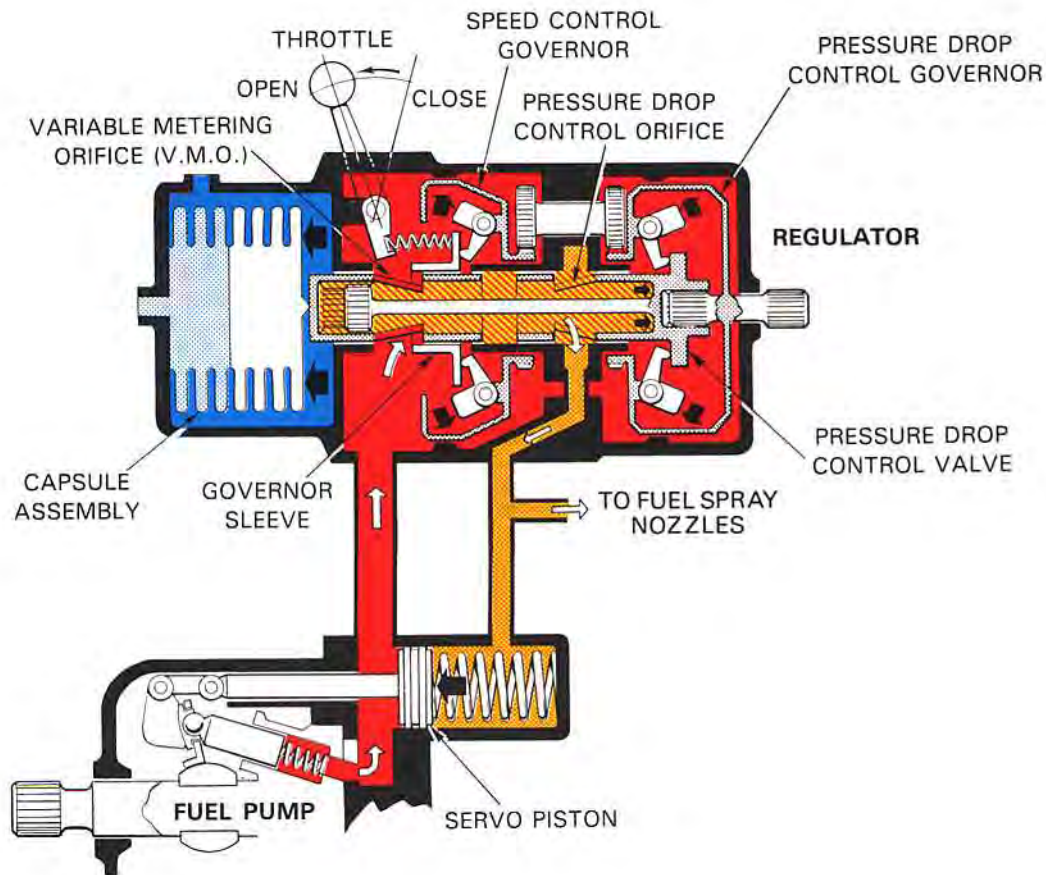
DISTRIBUTION AND CONTROLLING PRESSURES

Pump delivery (H.P. fuel)

Main fuel

Primary fuel

Air intake

Compressor delivery

THROTTLE OPENED

Governor sleeve moves to increase V.M.O. area

Pressure difference sensed by pressure drop control which opens pressure drop valve

Reduced pressure difference across regulator acts on servo piston to increase pump delivery and engine speed

Increased compressor delivery pressure acts on capsule to further increase fuel flow

At engine speed selected, governors control V.M.O. and pressure drop control orifice areas, thus stabilizing pressure difference across regulator

Pump gives fuel output to match engine requirements

Fig. 10-10 Governor movement and fuel flow control.

main fuel pressure and it is not shut off by the pressure drop control piston. It therefore gives a satisfactory idling fuel flow at all altitudes.

64.   On engines featuring water injection (Part 17), a reset device (fig. 10-11), operated by a piston and reset cam, increases the loading on the throttle control spring and stirrup arm, thus selecting a higher engine speed during water injection. To prevent the power limiter (fig, 10-9) cancelling the effect of water injection, a capsule in the limiter is subjected to water pressure to raise the compressor delivery pressure at which the power limiter operates.

Pressure ratio control
65.   The pressure ratio control (fig. 10-12) is a mechanical system similar to the combined acceleration and speed control system, but uses the ratio of H.P. compressor delivery pressure to air intake pressure (P4/P1) as the main controlling parameter. It needs no separate governor unit for controlling the maximum r.p.m. The controlling mechanism is contained in one unit, which is usually referred to as a fuel flow regulator (F.F.R.). A gear-type pump is used, as described in para. 88, and the pump output to the F.F.R. is controlled by a pressure drop spill valve.

66.   The F.F.R. is driven by the engine through a gear train and has two rotating valves. One valve,

known as a variable metering sleeve, has a triangular orifice, known as the variable metering orifice (V.M.O.), and this sleeve is given axial movement by a capsule assembly. The other valve, known as the pressure drop control valve, is provided with axial movement by a centrifugal governor, known as a pressure drop control governor, Both valves form variable restrictors which control the fuel flow to the spray nozzles.

67.   Control of the V.M.O. area is a function of a pressure ratio control unit housed in the F.F.R. A pressure ratio control valve, subjected to P4 and P1, pressures, regulates the movement of the F.F.R. capsule and thus controls the V.M.O. area to produce the pressure ratio dictated by the throttle or power lever.

68.   At any steady running condition, the output of the fuel pump is greater than the engine requirement. The pressure drop spill valve is open to allow surplus fuel to return to the inlet side of the pump. This action controls the fuel delivery to that demanded by the F.F.R.

69.   When the throttle is slowly opened, the throttle-controlled orifice is increased and the control pressure falls, thus allowing the pressure ratio control valve to move towards the closed position (acceleration stop). F.F.R. capsule chamber pressure
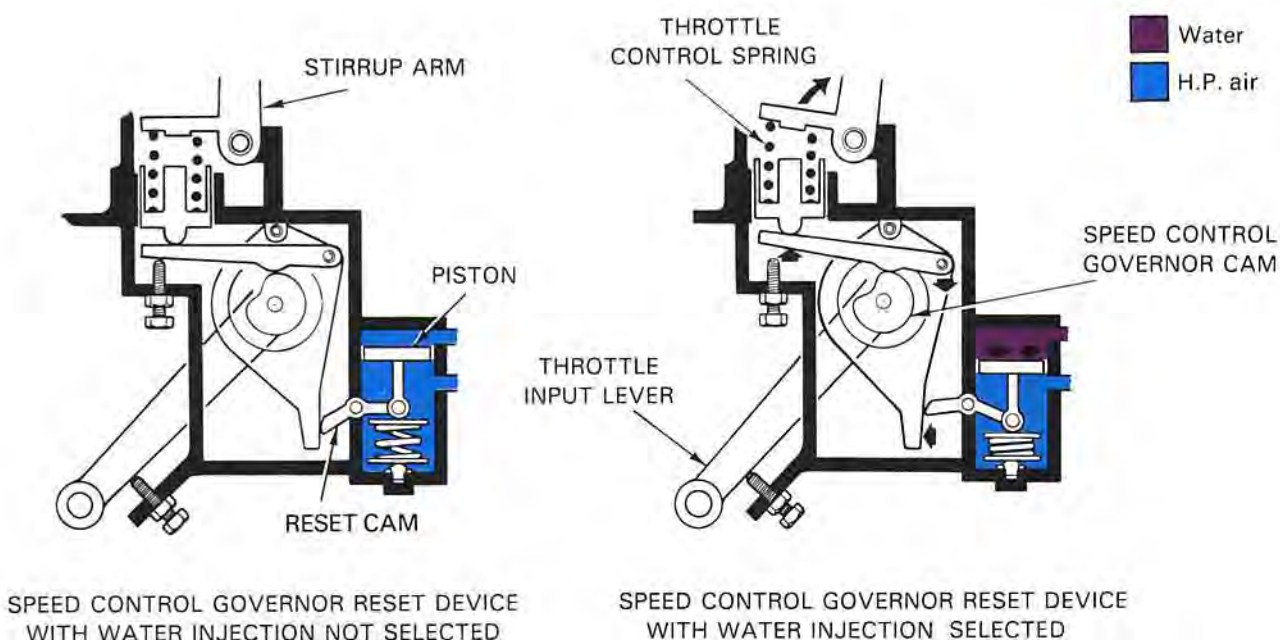
Fig. 10-11 Effect of water reset on speed control governor.

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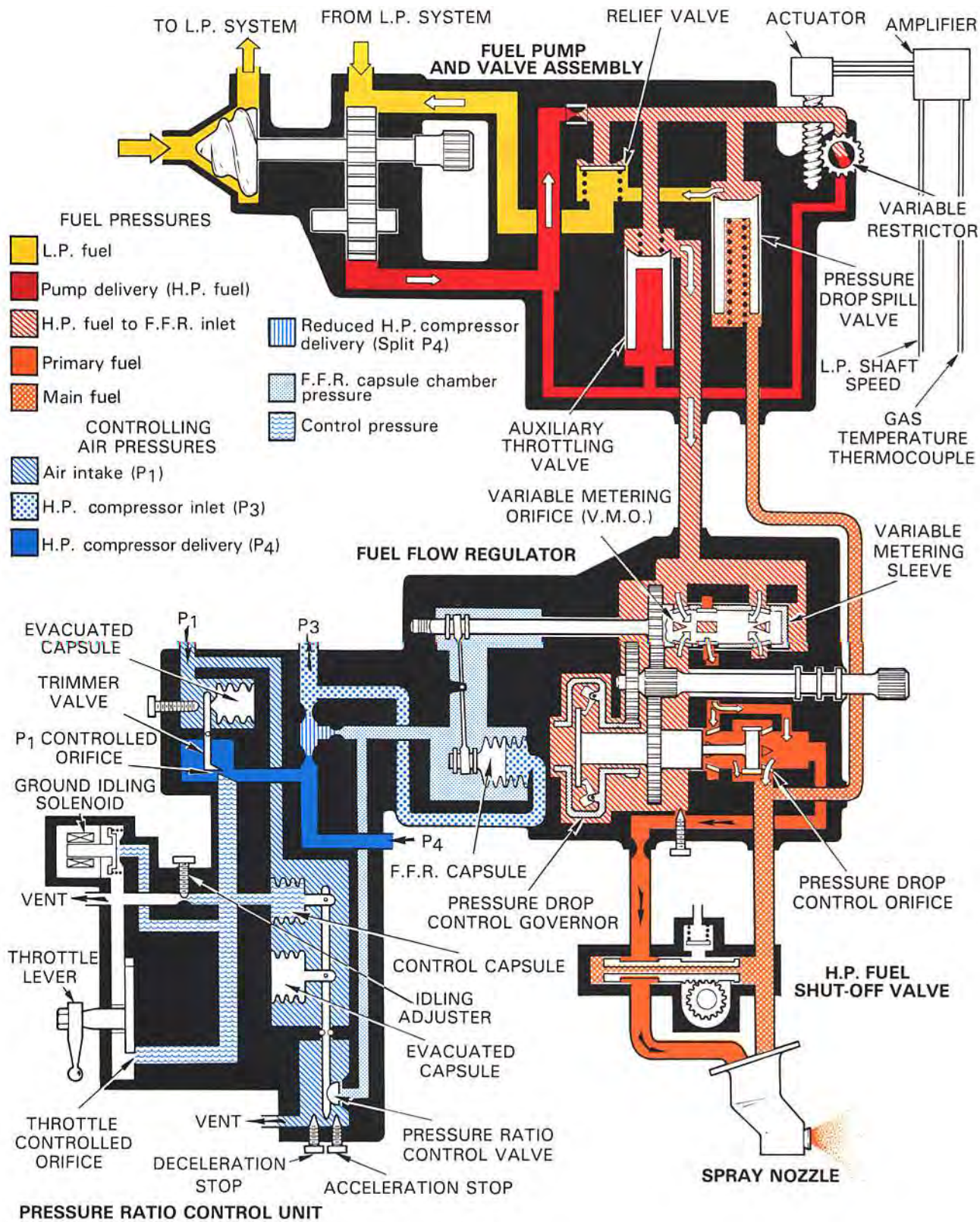
Fig. 10-12 A pressure ratio control system.

increases and the capsule moves the metering sleeve to increase the V.M.O. area. The effect of opening the V.M.O. is to reduce the pressure difference and this is sensed by the pressure drop governor, which opens the pressure drop control orifice. The reduced system pressure difference is immediately sensed by the pressure drop spill valve, which moves towards the closed position and consequently increases the fuel output. The increased fuel flow accelerates the engine with a subsequent increase in pressure ratio (P4/P1). When the required pressure ratio is reached, the pressure ratio control valve opens and the F.F.R. capsule chamber pressure reduces. The capsule assembly expands, moving the V.M.O. sleeve to reduce the orifice area. The resultant increased pressure difference is sensed by the pressure drop control governor, which adjusts the pressure drop control orifice to a point at which the pressure drop spill valve gives a fuel output consistent with steady running requirements.

70. During a rapid acceleration, the degree of overselling is mechanically controlled by the acceleration stop, which limits the movement of the pressure ratio control valve. A similar stop prevents the fuel supply from being completely cut off during a rapid deceleration.

71. When accelerating to a higher P4/P1 ratio, the throttle control orifice is increased. The reduced pressure allows the pressure ratio control capsule to contract so that the valve contacts the acceleration stop. F.F.R. capsule chamber pressure increases and the capsule moves to increase the V.M.O. area. This action continues until the required P4/P1 ratio is reached. The increased P4 pressure allows the pressure ratio control capsule to re-expand and the valve to return to the steady running position.

72. A change in altitude of the aircraft requires a variation in fuel flow to match the engine thrust and aircraft climb requirement. The normal effect of an altitude increase is to decrease the P1 and P4 pressures, thus opening the pressure ratio control valve and allowing the F.F.R. capsule to expand to reduce the V.M.O. area and, in consequence, the fuel flow. However, to match the engine thrust and aircraft climb requirement it is necessary to increase the P4/P1 ratio with increasing altitude. This is done by a trimmer valve and a capsule that is subjected to P1 pressure. As P1 pressure decreases, the trimmer valve moves across the P1 controlled orifice to reduce the control pressure. This is sensed by the control capsule, which, by acting on the pressure ratio control valve, slows the closure of the V.M.O. as

altitude is increased. This maintains the thrust requirement with the throttle at a fixed position.

73. To prevent the maximum L.P. compressor r.p.m. and engine gas temperature from being exceeded, a valve, known as the auxiliary throttling valve, is fitted in the outlet from the fuel pump, Under steady running conditions, the valve is held open by spring force, When limiting conditions are reached, the fuel flow is reduced in response to speed and temperature signals from the engine. The signals are amplified and passed to a rotary actuator that reduces the area of a variable restrictor. The effect of this is to increase the fuel pressure, which partially closes the throttling valve. H.P. fuel pressure acting on the face of the pressure drop spill valve is increased and the spill valve opens to reduce the fuel flow to the spray nozzles.

74. H.P. shaft speed is also governed by the auxiliary throttling valve. Should other controlling devices fail and pump speed increases, the fuel pressure closes the throttling valve and opens the pressure drop spill valve to reduce the fuel flow.

75. With the throttle closed, idling condition is determined by controlling the amount of air being vented through the idling adjuster and the ground idling solenoid valve, With both bleeds in operation, satisfactory flight idling for the air off-takes is ensured. By closing the solenoid valve a lower power condition for ground idling is obtained.

76. This fuel system, like the combined acceleration and speed control system, has no pressurizing valve to divide the flow from the fuel pump into main and primary flows.

ELECTRONIC ENGINE CONTROL

77. As stated in para. 8, some engines utilize a system of electronic control to monitor engine performance and make necessary control inputs to maintain certain engine parameters within predetermined limits. The main areas of control are engine shaft speeds and exhaust gas temperature (E.G.T.) which are continuously monitored during engine operation. Some types of electronic control function as a limiter only, that is, should engine shaft speed or E.G.T. approach the limits of safe operation, then an input is made to the fuel flow regulator (F.F.R.) to reduce the fuel flow thus maintaining shaft speed or E.G.T. at a safe level. Supervisory control systems may contain a limiter function but, basically, by using aircraft generated data, the system enables a more appropriate thrust setting to be selected quickly and

accurately by the pilot. The control system then makes small control adjustments to maintain engine thrust consistent with that pre-set by the pilot, regardless of changing atmospheric conditions. Full authority digital engine control (FAD.E.G.) takes over virtually all of the steady state and transient control intelligence and replaces most of the hydromechanical and pneumatic elements of the fuel system. The fuel system is thus reduced to a pump and control valve, an independent shut-off cock and a minimum of additional features necessary to keep the engine safe in the event of extensive electronic failure.

78.   Full authority fuel control (F.A.F.C.) provides full electronic control of the engine fuel system in the same way as F.A.D.E.C., but has none of the transient control intelligence capability used to control the compressor airflow system as the existing engine control system is used for these.

Speed and temperature control amplifiers
79.   The speed and temperature control amplifier receives signals from thermocouples measuring E.G.T. and from speed probes sensing L.P. and in some cases, L.P. shaft speeds (N1 and N2). The amplifier basically comprises speed and temperature channels which monitor the signals sensed. If either N1, N2 or E.G.T. exceed pre-set datums, the amplifier output stage is triggered to connect an electrical supply to a solenoid valve (para. 47) or a variable restrictor (para. 73) which override the F.F.R. and cause a reduction in fuel flow. The limiter will only relinquish control back to the F.F.R. if the input conditions are altered (altitude, speed, ambient temperature or throttle lever position). The limiter system is designed to protect against parameters exceeding their design values under normal operation and basic fuel system failures.

Engine supervisory control
80.   The engine supervisory control (E.S.C.) system performs a supervisory function by trimming the fuel flow scheduled by the fuel flow governor (F.F.G.) to match the actual engine power with a calculated engine power for a given throttle angle. The E.S.C. provides supervisory and limiting functions by means of a single control output signal to a torque motor in the F.F.G. In order to perform its supervisory function the E.S.C. monitors inputs of throttle angle, engine bleed state, engine pressure ratio (E.P.R.) and air data computer information (altitude, Mach number and temperatures). From this data the supervisory channel predicts the value of N1 required to achieve the command E.P.R. calculated for the throttle angle set by the pilot. Simultaneously a comparison is made between the command E.P.R. and the actual

E.P.R. and the difference is compared with a programmed datum.

81.   During acceleration the comparitor connects the predicted value of N1 to the limiter channel until the difference between the command and actual E.P.R. is approximately 0.03 E.P.R. At this point the predicted L.P. shaft speed is disconnected and the E.P.R. difference signal is connected to the limiter channel.

82.   The final output from the supervisory channel, in the form of an error signal, is supplied to a 'lowest wins' circuit along with the error signals from the limiter channel. While the three error signals remain positive (N1 and E.G.T. below datum level and actual E.P.R. below command E.P.R.) no output is signalled to the torque motor. If, however, the output stage of the E.S.C. predicts that E.G.T. will exceed datum or that N1 will either exceed its datum or the predicted level for the command E.P.R., then a signal is passed to the torque motor to trim the fuel flow.

LOW PRESSURE FUEL SYSTEM

83.   An L.P. system (fig.10-13) must be provided to supply the fuel to the engine at a suitable pressure, rate of flow and temperature, to ensure satisfactory engine operation. This system may include an L.P. pump to prevent vapour locking and cavitation of the fuel, and a fuel heater to prevent ice crystals forming. A fuel filter is always used in the system and in some instances the flow passes through an oil cooler (Part 8). Transmitters may also be used to signal fuel pressure, flow and temperature (Part 12).

FUEL PUMPS

84.   There are two basic types of fuel pump, the plunger-type pump and the constant-delivery gear-type pump; both of these are positive displacement pumps. Where low pressures are required at the fuel spray nozzles, the gear-type pump is preferred because of its lightness.

Plunger-type fuel pump
85.   The pump shown in fig. 10-14 is of the single-unit, variable-stroke, plunger-type; similar pumps may be used as double units depending upon the engine fuel flow requirements.

86.   The fuel pump is driven by the engine gear train and its output depends upon its rotational speed and the stroke of the plungers. A single-unit fuel pump can deliver fuel at the rate of 100 to 2,000 gallons per hour at a maximum pressure of about 2,000 lb. per

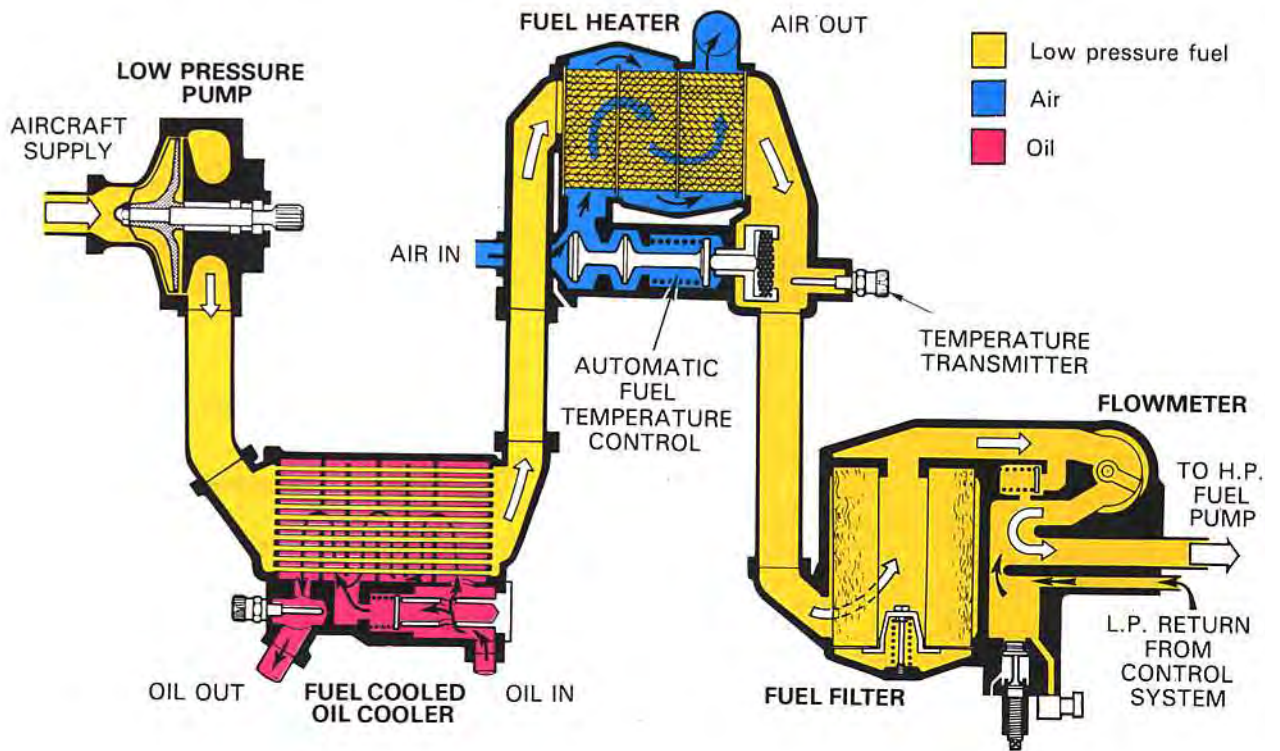**Fig. 10-14 A low pressure system.**

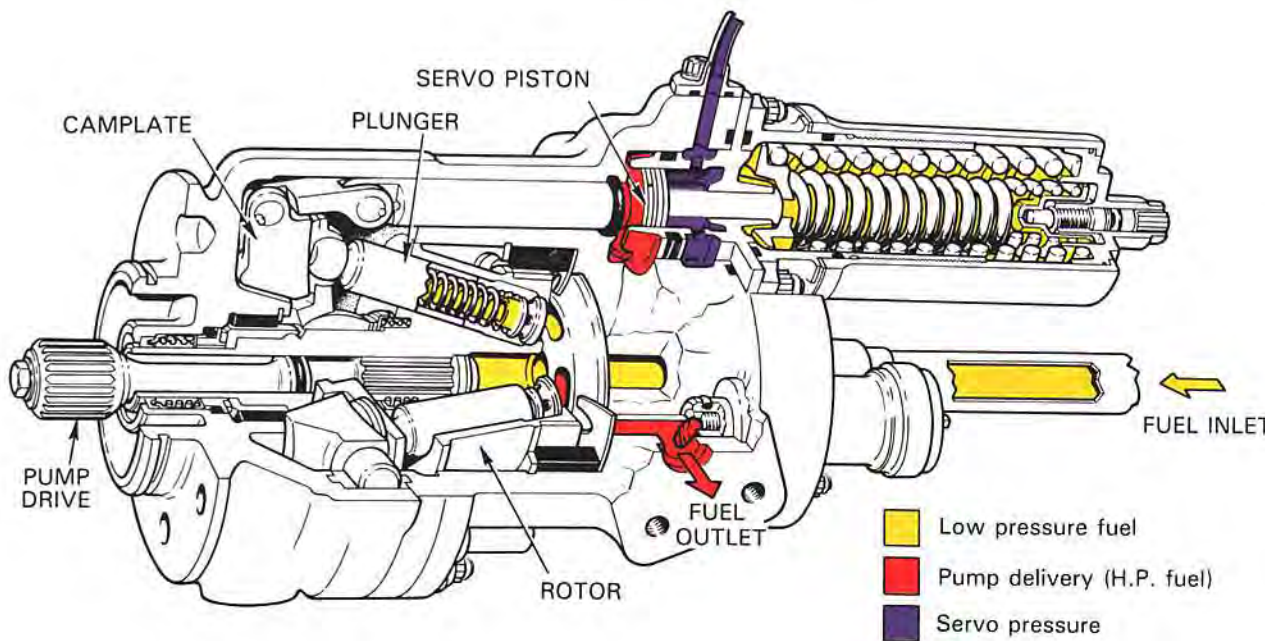**Fig. 10-14 A plunger-type fuel pump.**

square inch. To drive this pump, as much as 60 horsepower may be required.

87. The fuel pump consists of a rotor assembly fitted with several plungers, the ends of which project from their bores and bear on to a non-rotating camplate. Due to the inclination of the camplate, movement of the rotor imparts a reciprocating motion to the plungers, thus producing a pumping action. The stroke of the plungers is determined by the angle of inclination of the camplate. The degree of inclination is varied by the movement of a servo piston that is mechanically linked to the camplate and is biased by springs to give the full stroke position of the plungers. The piston is subjected to servo pressure on the spring side and on the other side to pump delivery pressure; thus variations in the pressure difference across the servo piston cause it to move with corresponding variations of the camplate angle and, therefore, pump stroke.
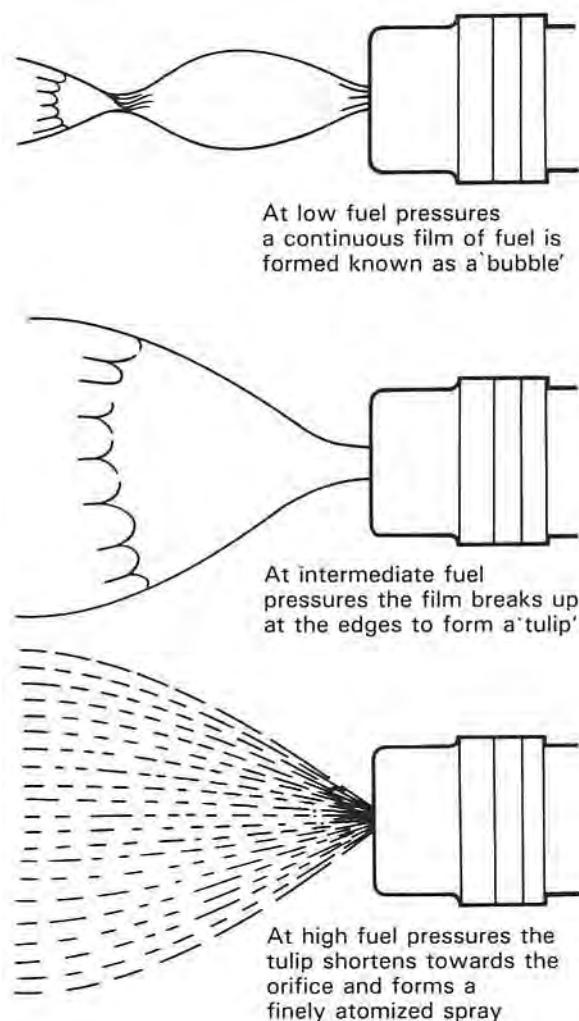
Gear-type fuel pump

88. The gear-type fuel pump (fig. 10-12) is driven from the engine and its output is directly proportional to its speed. The fuel flow to the spray nozzles is controlled by recirculating excess fuel delivery back to inlet. A spill valve, sensitive to the pressure drop across the controlling units in the system, opens and closes as necessary to increase or decrease the spill.

FUEL SPRAY NOZZLES

89. The final components of the fuel system are the fuel spray nozzles, which have as their essential function the task of atomizing or vaporizing the fuel to ensure its rapid burning. The difficulties involved in this process can be readily appreciated when one considers the velocity of the air stream from the compressor and the short length of combustion system (Part 4) in which the burning must be completed.

90. An early method of atomizing the fuel is to pass it through a swirl chamber where tangentially disposed holes or slots imparted swirl to the fuel by converting its pressure energy to kinetic energy. In this state, the fuel is passed through the discharge orifice which removes the swirl motion as the fuel is atomized to form a cone-shaped spray. This is called 'pressure jet atomization'. The rate of swirl and pressure of the fuel at the fuel spray nozzle are important factors in good atomization. The shape of the spray is an indication of the degree of atomization as shown in fig. 10-15. Later fuel spray nozzles utilize the airspray principle which employs

At low fuel pressures a continuous film of fuel is formed known as a 'bubble'

At intermediate fuel pressures the film breaks up at the edges to form a 'tulip'

At high fuel pressures the tulip shortens towards the orifice and forms a finely atomized spray

Fig. 10-15 Various stages of fuel atomization.

high velocity air instead of high velocity fuel to cause atomization. This method allows atomization at low fuel flow rates (provided sufficient air velocity exists) thus providing an advantage over the pressure jet atomizer by allowing fuel pumps of a lighter construction to be used.

91. The atomizing spray nozzle, as distinct from the vaporizing burner (Part 4), has been developed in five fairly distinct types; the Simplex, the variable port (Lubbock), the Duplex or Duple, the spill type and the airspray nozzle.

92. The Simplex spray nozzle shown in fig. 10-16 was first used on early jet engines. It consists of a chamber, which induces a swirl into the fuel, and a fixed-area atomizing orifice. This fuel spray nozzle gave good atomization at the higher fuel flows, that

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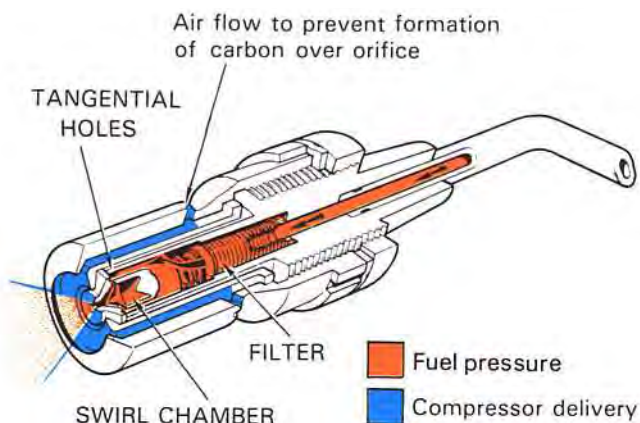
Fig. 10-16 A Simplex fuel spray nozzle.

is, at the higher fuel pressures, but was very unsatisfactory at the low pressures required at low engine speeds and especially at high altitudes. The reason for this is that the Simplex was, by the nature of its design, a 'square law' spray nozzle; that is, the flow through the nozzle is proportional to the square root of the pressure drop across it. This meant that if the minimum pressure for effective atomization was 30 lb. per square inch, the pressure needed to give maximum flow would be about 3,000 lb. per square inch. The fuel pumps available at that time were unable to cope with such high pressures so the variable port spray nozzle was developed in an effort to overcome the square law effect.

93. Although now only of historical value, the variable port or Lubbock fuel spray nozzle (fig. 10-17) made use of a spring-loaded piston to control the area of the inlet ports to the swirl chamber. At low fuel flows, the ports were partly uncovered by the movement of the piston; at high flows, they were fully open. By this method, the square law pressure relationship was mainly overcome and good atomization was maintained over a wide range of fuel flows. The matching of sets of spray nozzles and the sticking of the sliding piston due to dirt particles were, however, difficulties inherent in the design, and this type was eventually superseded by the Duplex and the Duple fuel spray nozzles.

94. The Duplex and the Duple spray nozzles require a primary and a main fuel manifold and have two independent orifices, one much smaller than the other. The smaller orifice handles the lower flows and the larger orifice deals with the higher flows as the fuel pressure increases. A pressurizing valve may be employed with this type of spray nozzle to apportion the fuel to the manifolds (fig. 10-18). As the fuel flow

and pressure increases, the pressurizing valve moves to progressively admit fuel to the main manifold and the main orifices. This gives a combined flow down both manifolds. In this way, the Duplex and Duple nozzles are able to give effective atomization over a wider flow range than the Simplex spray nozzle for the same maximum fuel pressure. Also, efficient atomization is obtained at the low flows that may be required at high altitude. In the combined acceleration and speed control system (para. 51), the fuel flow to the spray nozzles is apportioned in the F.F.R.

95. The spill type fuel spray nozzle can be described as being a Simplex spray nozzle with a passage from the swirl chamber for spilling fuel away. With this arrangement it is possible to supply fuel to the swirl chamber at a high pressure all the time, As the fuel demand decreases with altitude or reduction in engine speed, more fuel is spilled away from the swirl Chamber, leaving less to pass through the atomizing orifice. The spill spray nozzles' constant use of a relatively high pressure means that even at the extremely low fuel flows that occur at high altitude there is adequate swirl to provide constant and efficient atomization of the fuel.

96. The spill spray nozzle system, however, involves a somewhat modified type of fuel supply and control system from that used with the previous types. A means has to be provided for removing the
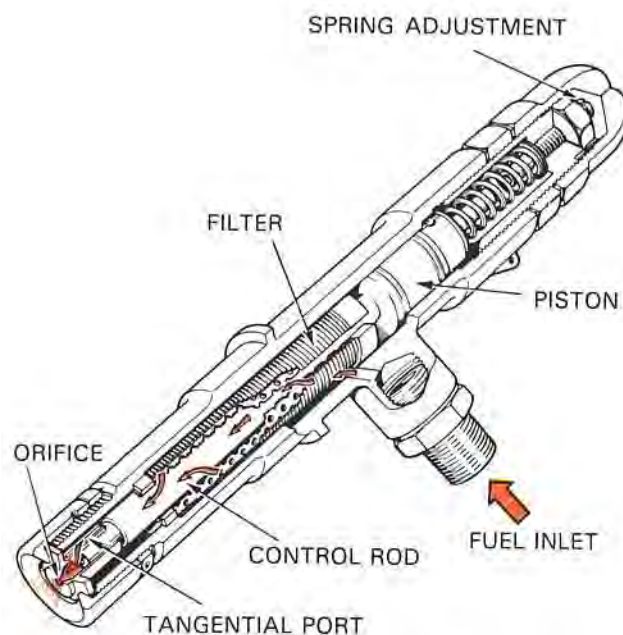
Fig. 10-17 A variable port or Lubbock fuel spray nozzle.
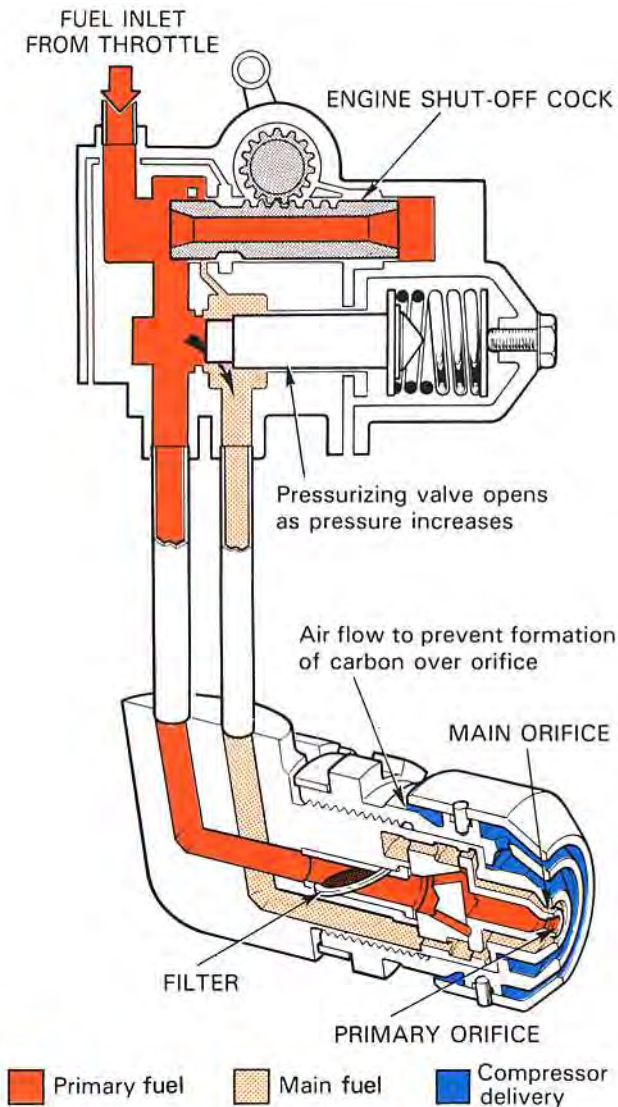
**Fig. 10-18 A Duple fuel spray nozzle and
pressurizing valve.**

spill and also for controlling the amount of spill flow at various engine operating conditions. A disadvatage of this system is that excess heat may be generated when a large volume of fuel is being recirculated to inlet. Such heat may eventually lead to a deterioration of the fuel.

97. The airspray nozzle (fig. 10-19), carries a proportion of the primary combustion air (Part 4) with the injected fuel. By aerating the spray, the local fuel-rich concentrations produced by other types of spray nozzle are avoided, thus giving a reduction in both carbon formation and exhaust smoke. An additional advantage of the airspray nozzle is that the low

pressures required for atomization of the fuel permits the use of the comparatively lighter gear-type pump.

98. A flow distributor (fig. 10-20) is often required to compensate for the gravity head across the manifold at low fuel pressures to ensure that all spray nozzles pass equal quantities of fuel.

99. Some combustion systems vaporize the fuel (Part 4) as it enters the combustion zone.

FUEL HEATING

100. On many engines, a fuel-cooled oil cooler (Part 8) is located between the L.P. fuel pump and the inlet to the fuel filter (fig. 10-13), and advantage is taken of this to transfer the heat from the oil to the fuel and thus prevent blockage of the filter element by ice particles. When heat transference by this means is insufficient, the fuel is passed through a second heat exchanger where it absorbs heat from a thermostatically controlled airflow taken from the compressor.

EFFECT OF A CHANGE OF FUEL

101. The main effect on the engine of a change from one grade of fuel to another arises from the variation of specific gravity and the number of heat units obtainable from a gallon of fuel. As the number of heat units per pound is practically the same for all fuels approved for gas turbine engines, a comparison of heat values per gallon can be obtained by comparing specific gravities.

102. Changes in specific gravity have a definite effect on the centrifugal pressure type of engine speed governor (para. 15), for with an increase in specific gravity the centrifugal pressure acting on the governor diaphragm is greater. Thus the speed at which the governor controls is reduced, and in consequence the governor must be reset.

103. With a decrease in specific gravity, the centrifugal pressure on the diaphragm is less and the speed at which the governor controls is increased; in consequence, the pilot must control the maximum r.p.m. by manual operation of the throttle to prevent overspeeding the engine until the governor can be reset. The hydro-mechanical governor (para. 23) is less sensitive to changes of specific gravity than the centrifugal governor and is therefore preferred on many fuel systems.

104. The pressure drop governor in the combined acceleration and speed control system (para. 51) is density compensated, by using a buoyant material

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Fig. 10-19 An airspray nozzle.

for the governor weights, resulting in fuel being metered on mass flow rather than volume flow.

105. Changes to a lower grade of fuel can lead to production of carbon, giving a greater flame luminosity and temperature, leading to higher combustor metal temperatures and reduced combustor and turbine life.

GAS TURBINE FUELS

106. Fuels for aircraft gas turbine engines must conform to strict requirements to give optimum engine performance, economy, safety and overhaul life. Fuels are classed under two headings, kerosine-type fuel and wide-cut gasoline-type fuel.
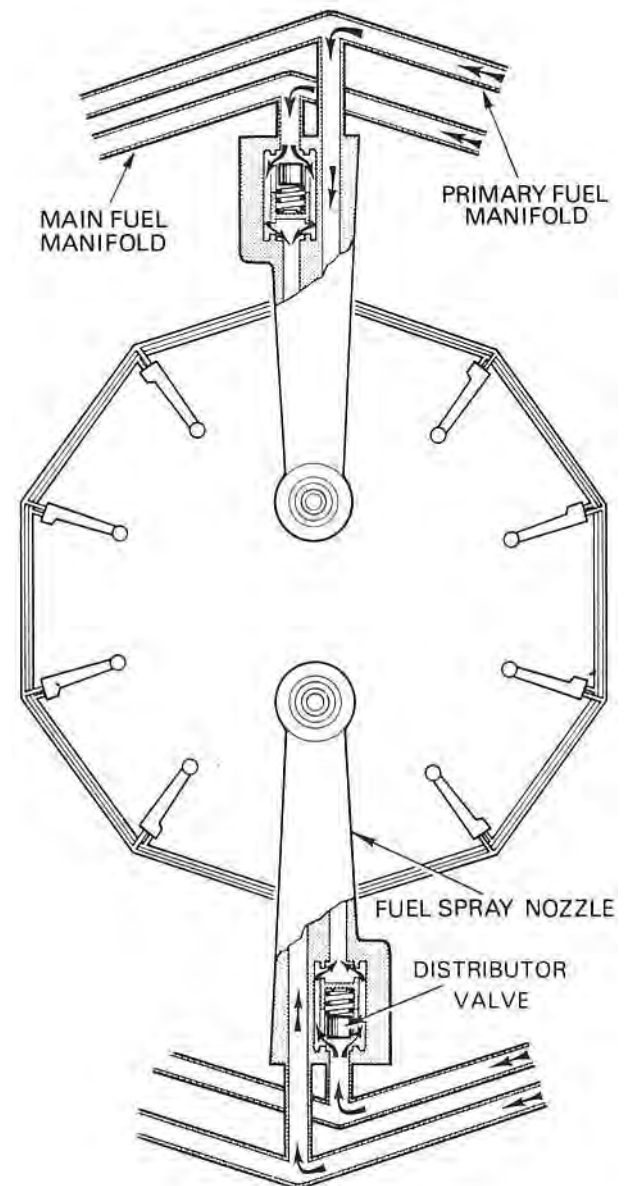
Fig. 10-20 Fuel flow distributor.

Fuel requirements

107.   In general, a gas turbine fuel should have the following qualities:

(1)   Be 'pumpable' and flow easily under all operating conditions.

(2)   Permit engine starting at all ground conditions and give satisfactory flight relighting characteristics.

(3)   Give efficient combustion at all conditions.

(4)   Have as high a calorific value as possible.

(5)   Produce minimal harmful effects on the combustion system or the turbine blades.

(6)   Produce minimal corrosive effects on the fuel system components.

(7)   Provide adequate lubrication for the moving parts of the fuel system.

(8)   Reduce fire hazards to a minimum.

108.   The pumping qualities of the fuel depend upon its viscosity or thickness, which is related to fuel temperature, Fuel must be satisfactory down to approximately -50 deg. C. As the fuel temperature falls, ice crystals may form to cause blockage of the fuel filter or the orifices in the fuel system. Fuel heating and anti-icing additives are available to alleviate this problem.

109. For easy starting, the gas turbine engine depends upon the satisfactory ignition of the atomized spray of fuel from the fuel spray nozzles, assuming that the engine is being motored at the required speed. Satisfactory ignition depends upon the quality of fuel in two ways:

(1)   The volatility of the fuel; that is, its ability to vaporize easily, especially at low temperatures.

(2)   The degree of atomization, which depends upon the viscosity of the fuel, the fuel pressure applied, and the design of the atomizer.

110.   The calorific value (fig. 10-21) of a fuel is an expression of the heat or energy content per pound or gallon that is released during combustion. This value, which is usually expressed in British thermal units, influences the range of an aircraft. Where the limiting factor is the capacity of the aircraft tanks, the calorific value per unit volume should be as high as possible, thus enabling more energy, and hence more aircraft range, to be obtained from a given volume of fuel. When the useful payload is the limiting factor, the calorific value per unit of weight should be as high as possible, because more energy can then be obtained from a minimum weight of fuel.

Other factors which affect the choice of heat per unit of volume or weight, must also be taken into consideration; these include the type of aircraft, the duration of flight, and the required balance between fuel weight and payload.
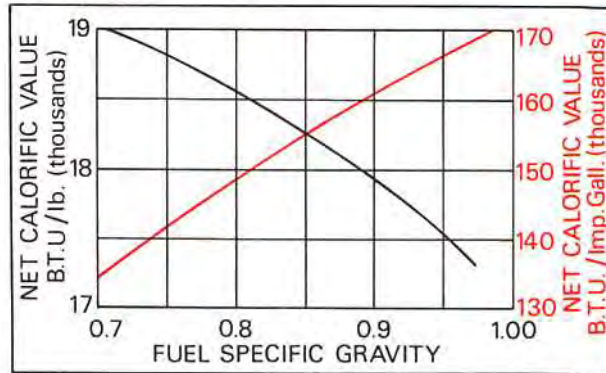
Fig. 10-21 Relationship between calorific value and specific gravity.

111.   Turbine fuels tend to corrode the components of the fuel and combustion systems mainly as a result of the sulphur and water content of the fuel. Sulphur, when burnt in air, forms sulphur dioxide; when mixed with water this forms sulphurous acid and is very corrosive, particularly on copper and lead. Because it is impracticable to completely remove the sulphur content, it is essential that the sulphur be kept to a controlled minimum. Although free water is removed prior to use, dissolved water, i.e. water in solution, cannot be effectively removed, as the fuel would re-absorb moisture from the atmosphere when stored in a vented aircraft or storage tank (para. 118).

112.   All gas turbine fuels are potentially dangerous and therefore handling and storage precautions should be strictly observed.

Vapour locking and boiling

113.   The main physical difference between kerosine and wide-cut fuels is their degree of volatility, the latter type of fuel having a higher volatility, thus increasing the problem of vapour locking and boiling. With kerosine-type fuels, the volatility is controlled by distillation and flash point, but with the wide-cut fuels it is controlled by distillation and the Reid Vapour Pressure (R.V.P.) test. In this test, the absolute pressure of the fuel is recorded by special apparatus with the fuel temperature at 37.8 deg. C. (100 deg. F.).

114.   Kerosine has a low vapour pressure and will boil only at extremely high altitudes or high tempera-

tures, whereas a wide-cut fuel wilt boil at a much lower altitude.

115. The fuel temperature during flight depends upon altitude, rate of climb, duration at altitude and kinetic heating due to forward speed. When boiling does occur, the vapour loss can be very high, especially with wide-cut fuels, and this may cause vapour locking with consequent malfunctions of the engine fuel system and fuel metering equipment.

116. To obviate or reduce the risk of boiling, it is usual to pressurize the fuel tanks. This involves maintaining an absolute pressure above the fuel in excess of its vapour pressure at any specific temperature. This may be accomplished by using an inert gas or by using the fuel vapour pressure with a controlled venting system.

117. For sustained supersonic flight, some measure of tank insulation is necessary to reduce kinetic heating effects, even when lower volatility fuels are used.

Fuel contamination control

118. Fuel can be maintained in good condition by well planned storage and by making routine aircraft tank drain checks. The use of suitable filters, fuel/water separators and selected additives will restrict the contamination level, e.g. free water and solid matter, to a practical minimum. Keeping the fuel free of undissolved water will prevent serious icing problems, reduce the microbiological growth and minimize corrosion. Reducing the solid matter will prevent undue wear in the fuel pumps, reduce corrosion and lessen the possibility of blockage occurring within the fuel system.

Rolls-Royce RB211-535C
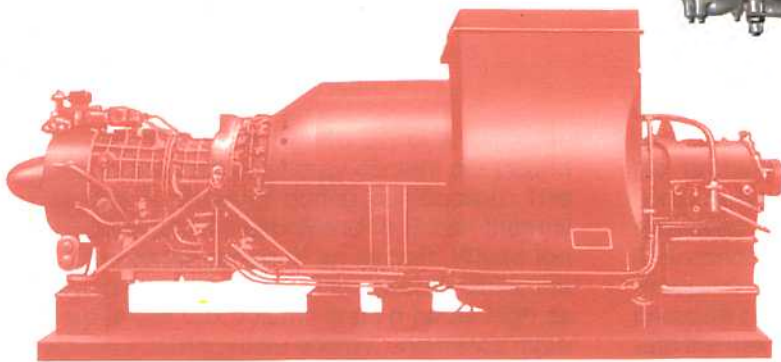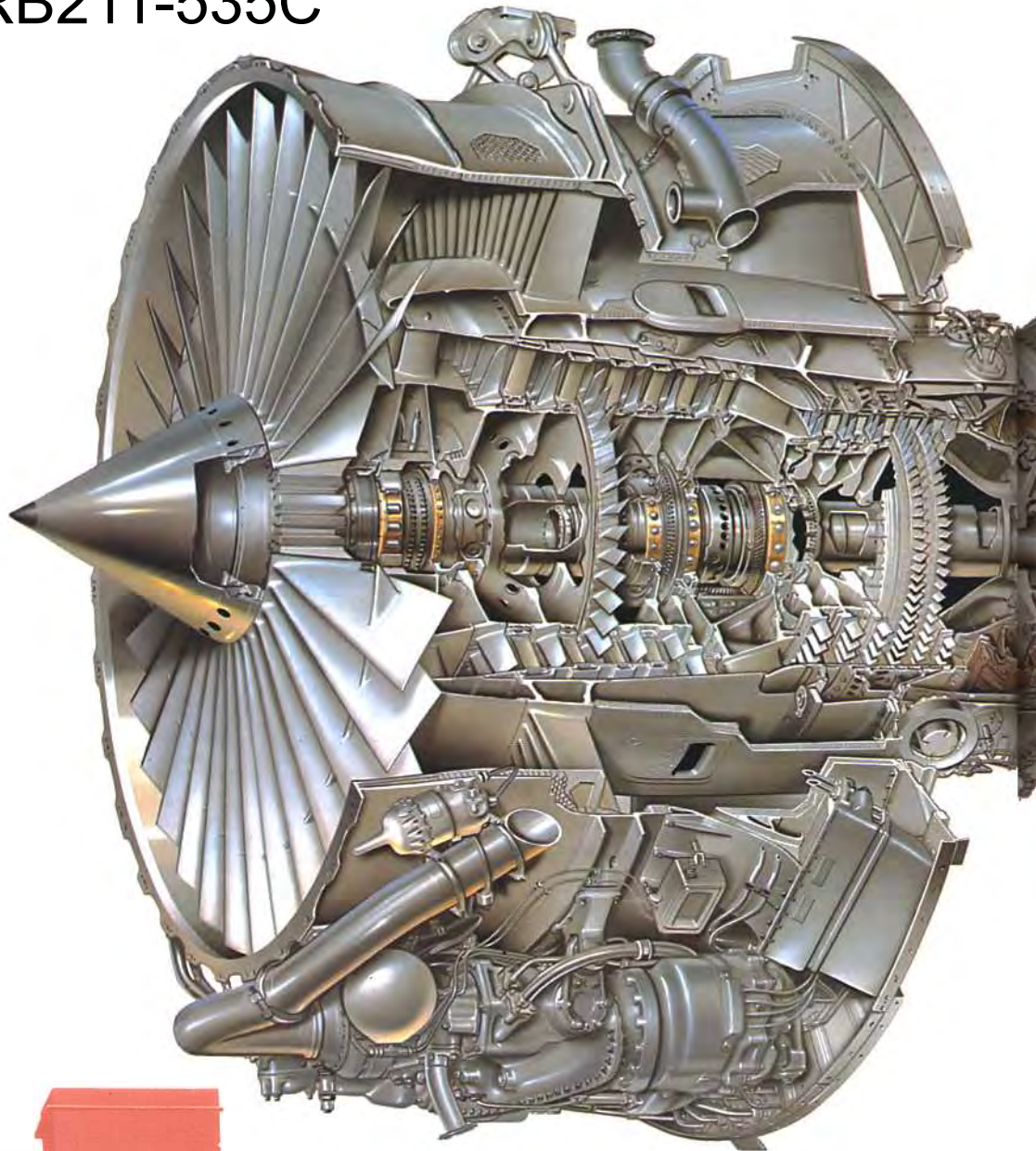
Metrovick G2

Following the successful operation at sea of the Metrovick F2-based 2500 hp Gatric marine gas turbine, the Royal Navy ordered four larger sets with a maximum operational rating of 4500 shp. Developed from the Metrovick F2/4 Beryl axial-flow aircraft engine; the G2s were installed in the Motor Gunboats 'Bold Pioneer1 and 'Bold Pathfinder; the former going to sea in 1951.

11: Starting and ignition

Contents
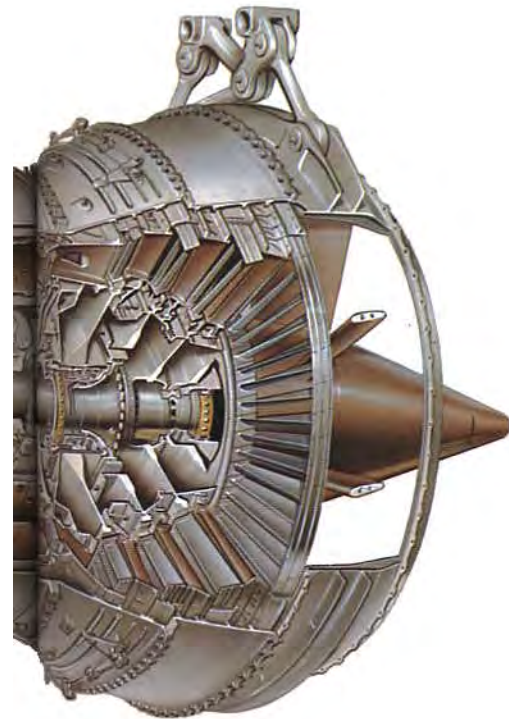
INTRODUCTION

1. Two separate systems are required to ensure that a gas turbine engine will start satisfactorily. Firstly, provision must be made for the compressor and turbine to be rotated up to a speed at which adequate air passes into the combustion system to mix with fuel from the fuel spray nozzles (Part 10). Secondly, provision must be made for ignition of the air/fuel mixture in the combustion system. During engine starting the two systems must operate simultaneously, yet it must also be possible to motor the engine over without ignition for maintenance checks and to operate only the ignition system for relighting during flight (para. 28).

2. The functioning of both systems is co-ordinated during a starting cycle and their operation is automatically controlled after the initiation of the cycle by an electrical circuit. A typical sequence of events during the start of a turbo-jet engine is shown in fig. 11-1.
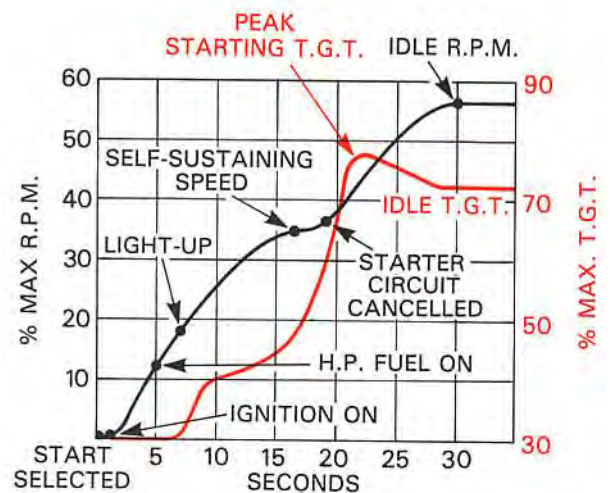
Fig. 11-1   **A typical starting sequence of a turbo-jet engine.**

METHODS OF STARTING

3. The starting procedure for all jet engines is basically the same, but can be achieved by various methods. The type and power source for the starter varies in accordance with engine and aircraft requirements. Some use electrical power, others use gas, air or hydraulic pressure, and each has its own merits. For example, a military aircraft requires the engine to be started in the minimum time and, when possible, to be completely independent of external equipment. A commercial aircraft, however, requires the engine to be started with the minimum disturbance to the passengers and by the most economical means. Whichever system is used, reliability is of prime importance.

4. The starter motor must produce a high torque and transmit it to the engine rotating assembly in a manner that provides smooth acceleration from rest up to a speed at which the gas flow through the engine provides sufficient power for the engine turbine to take over.

Electric

5. The electric starter is usually a direct current (D.C.) electric motor coupled to the engine through a reduction gear and ratchet mechanism, or clutch, which automatically disengages after the engine has reached a self-sustaining speed (fig. 11-2).

6. The electrical supply may be of a high or low voltage and is passed through a system of relays and resistances to allow the full voltage to be progressively built up as the starter gains speed. It also provides the power for the operation of the ignition system. The electrical supply is automatically cancelled when the starter load is reduced after the engine has satisfactorily started or when the time cycle is completed. A typical electrical starting system is shown in fig. 11-3.
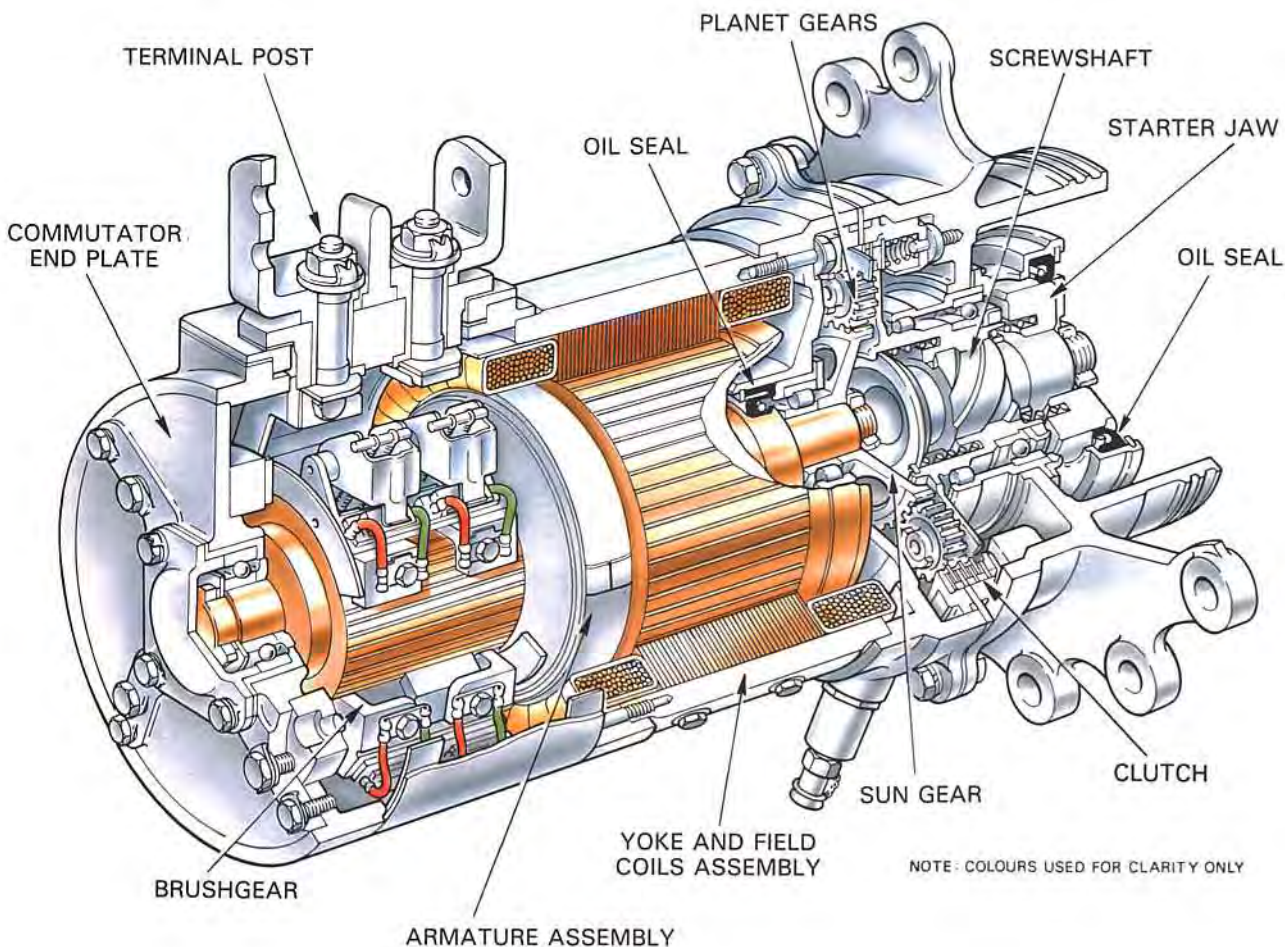
Fig. 11-2   An electric starter.

28 VOLT D.C. SUPPLY

STARTER MASTER SWITCH

START   BLOWOUT   START   RELIGHT

START/RELIGHT SELECTOR SWITCH

INDICATOR LIGHT 'ON'

Start initiation

OVERSPEED RELAY

IGNITION SWITCH ISOLATING RELAY

CUT-OFF TIME SWITCH

IGNITION RELAY

MAIN RELAY

SLOW START RESISTOR

FULL CURRENT TIME SWITCH

REDUCED CURRENT TIME SWITCH

HIGH ENERGY IGNITION UNITS

IGNITER PLUG

STARTER MOTOR

Start circuit
Relight circuit
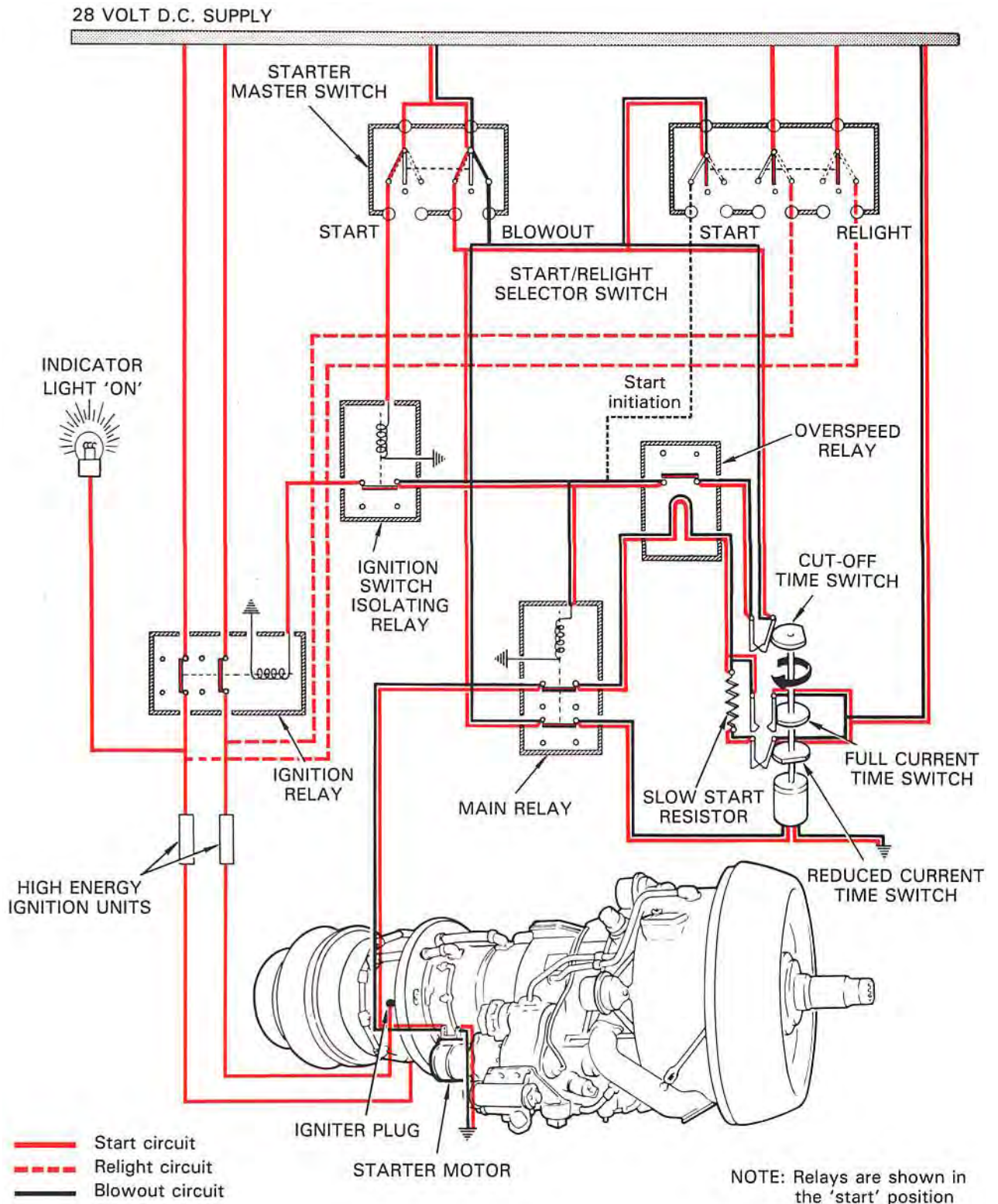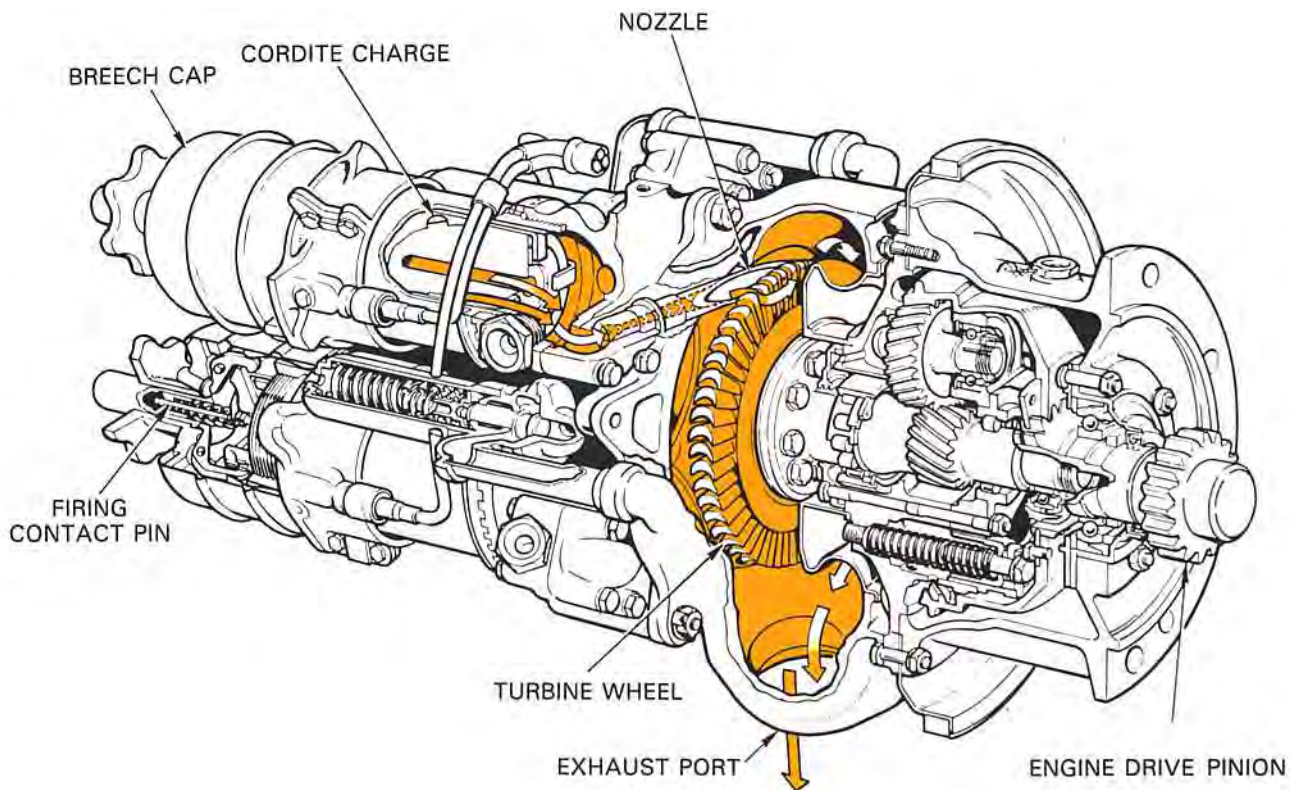Blowout circuit

NOTE: Relays are shown in the 'start' position

Fig. 11-3   A low voltage electrical starting system.

Fig. 11-4   A triple-breech cartridge starter.

Cartridge

7.   Cartridge starting is sometimes used on military engines and provides a quick independent method of starting. The starter motor is basically a small impulse-type turbine that is driven by high velocity gases from a burning cartridge. The power output of the turbine is passed through a reduction gear and an automatic disconnect mechanism to rotate the engine. An electrically fired detonator initiates the burning of the cartridge charge. As a cordite charge provides the power supply for this type of starter, the size of the charge required may well limit the use of the cartridge starters. A triple-breech starter is illustrated in fig. 11-4.

Iso-propyl-nitrate

8.   This type of starter provides a high power output and gives rapid starting characteristics. It has a turbine that transmits power through a reduction gear to the engine. In this instance, the turbine is rotated by high pressure gases resulting from the combustion of iso-propyl-nitrate. This fuel is sprayed into a combustion chamber, which forms part of the starter, where it is electrically ignited by a high-energy ignition system. A pump supplies the fuel to the combustion chamber from a storage tank and an

air pump scavenges the starter combustion chamber of fumes before each start. Operation of the fuel and air pumps, ignition systems, and cycle cancellation, is electrically controlled by relays and time switches. An iso-propyl-nitrate starting system is shown in fig. 11-5.

Air

9.   Air starting is used on most commercial and some military jet engines. It has many advantages over other starting systems, and is comparatively light, simple and economical to operate.

10.   An air starter motor transmits power through a reduction gear and clutch to the starter output shaft which is connected to the engine. A typical air starter motor is shown in fig. 11-6.

11.   The starter turbine is rotated by air taken from an external ground supply, an auxiliary power unit (A.P.U.) or as a cross-feed from a running engine. The air supply to the starter is controlled by an electrically operated control and pressure reducing valve that is opened when an engine start is selected and is automatically closed at a predetermined starter speed. The clutch also automatically disengages as the engine accelerates up to idling r.p.m. and the

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AIRCRAFT ELECTRICAL SUPPLY

STARTER BUTTON

FUEL TANK

AIR BY-PASS

CONTROL BOX

FUEL AND AIR SCAVENGE PUMP

FUEL BY-PASS

HIGH PRESSURE SWITCH

IGNITION UNIT

STARTER MOTOR

IGNITION SWITCH

IGNITER

SPEED CONTROL SWITCH

EXHAUST

Fuel

Air
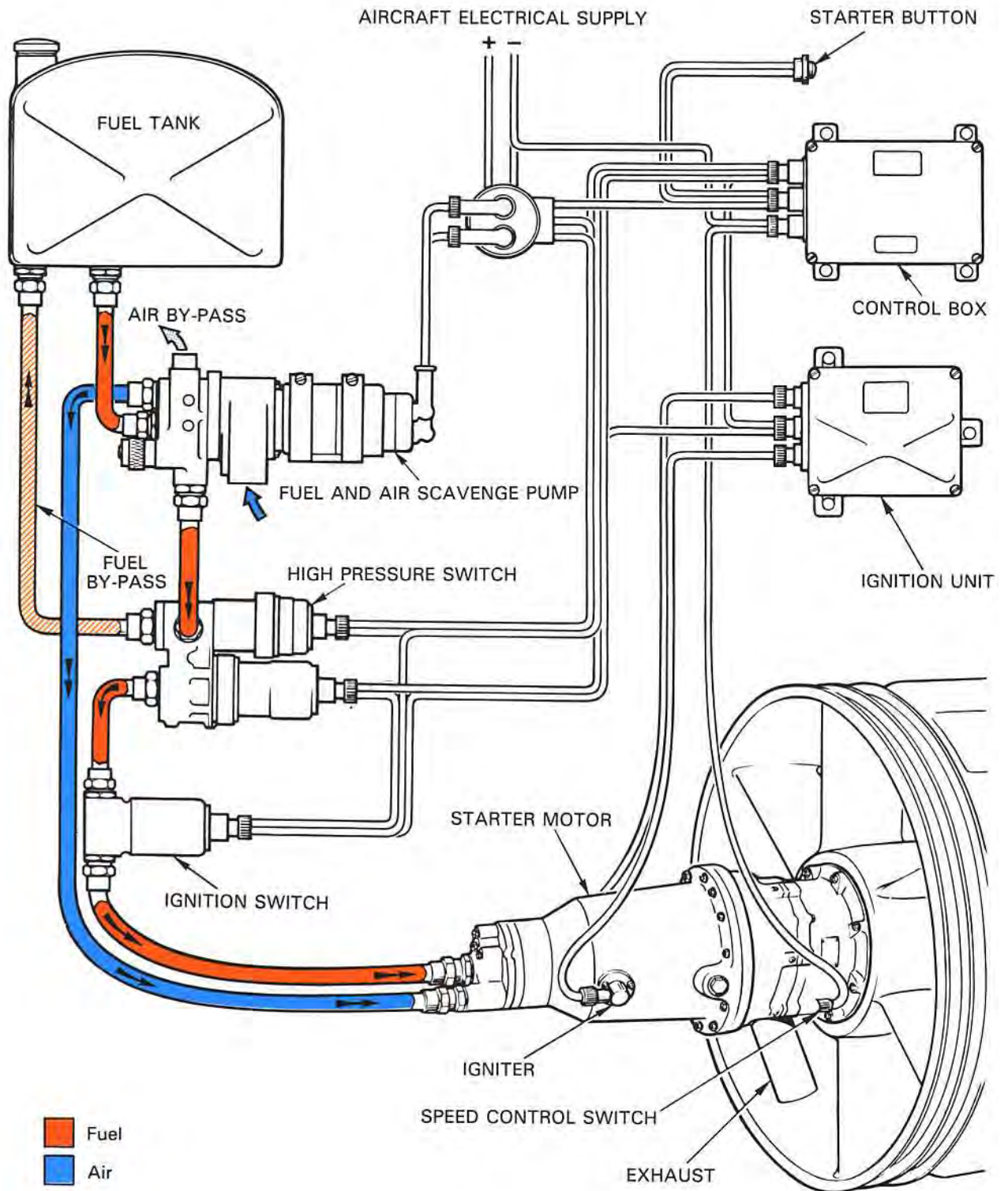
Fig. 11-5   An iso-propyl-nitrate starting system.

AIR OUTLET

ENGINE DRIVE SHAFT

CLUTCH

AIR INLET

TURBINE ROTOR

REDUCTION GEAR
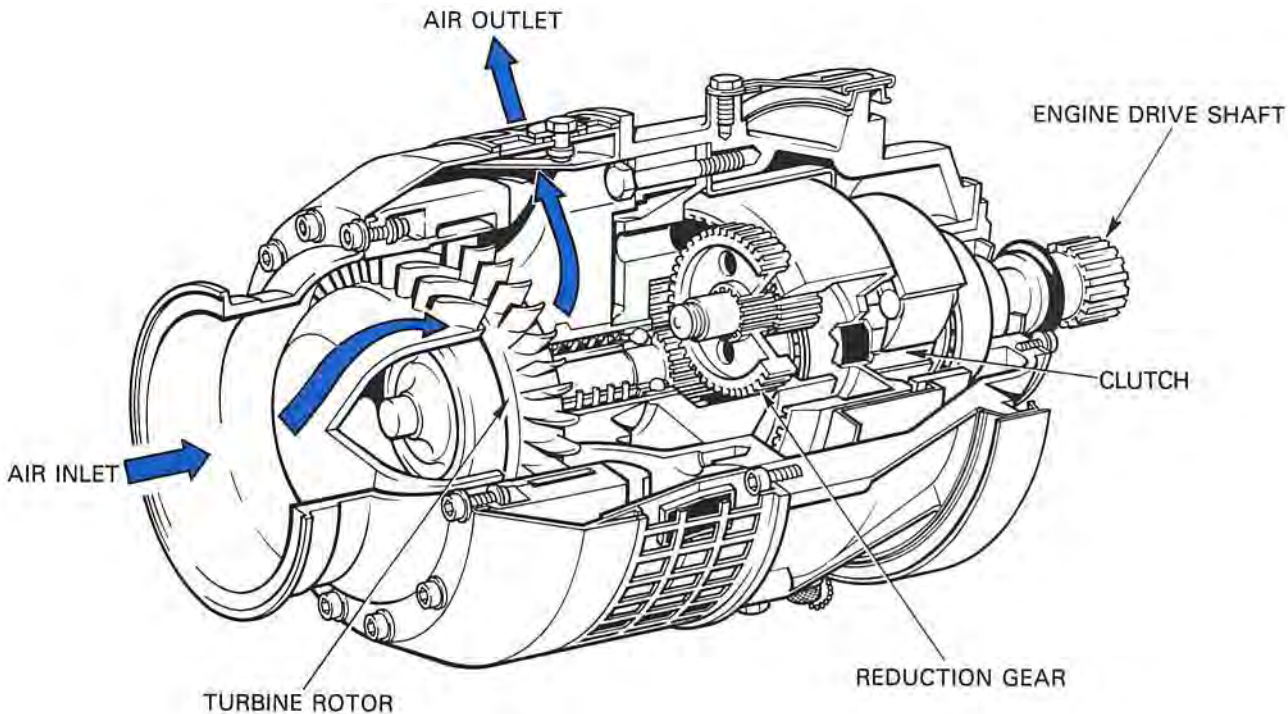
Fig. 11-6   An air starter motor.

rotation of the starter ceases. A typical air starting system is shown in fig. 11-7.

12.   A combustor starter is sometimes fitted to an engine incorporating an air starter and is used to supply power to the starter when an external supply of air is not available. The starter unit has a small combustion chamber into which high pressure air, from an aircraft-mounted storage bottle, and fuel, from the engine fuel system, are introduced. Control valves regulate the air supply which pressurizes a fuel accumulator to give sufficient fuel pressure for atomization and also activates the continuous ignition system. The fuel/air mixture is ignited in the combustion chamber and the resultant gas is directed onto the turbine of the air starter. An electrical circuit is provided to shut off the air supply which in turn terminates the fuel and ignition systems on completion of the starting cycle.

13.   Some turbo-jet engines are not fitted with starter motors, but use air impingement onto the turbine blades as a means of rotating the engine. The air is obtained from an external source, or from an engine that is running, and is directed through non-return valves and nozzles onto the turbine blades. A typical method of air impingement starting is shown in fig. 11-8.

Gas turbine

14.   A gas turbine starter is used for some jet engines and is completely self-contained. It has its own fuel and ignition system, starting system (usually electric or hydraulic) and self-contained oil system. This type of starter is economical to operate and provides a high power output for a comparatively low weight.

15.   The starter consists of a small, compact gas turbine engine, usually featuring a turbine-driven centrifugal compressor, a reverse flow combustion system and a mechanically independent |free-power turbine. The free-power turbine is connected to the main engine via a two-stage epicyclic reduction gear, automatic clutch and output shaft. A typical gas turbine starter is shown in fig. 11-9.

16.   On initiation of the starting cycle, the gas turbine starter is rotated by its own starter motor until it reaches self-sustaining speed, when the starting and ignition systems are automatically switched off. Acceleration then continues up to a controlled speed of approximately 60,000 r.p.m. At the same time as the gas turbine starter engine is accelerating, the exhaust gas is being directed, via nozzle guide vanes, onto the free-power turbine to provide the drive to the main engine. Once the main engine reaches self-sustaining speed, a cut-out switch

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CROSS FEED FROM
RUNNING ENGINE

AUXILIARY
POWER UNIT (A.P.U.)

GROUND
START SUPPLY

AIRFRAME PYLON

High pressure air

AIR CONTROL VALVE

EXHAUST
AIR

EXTERNAL GEARBOX
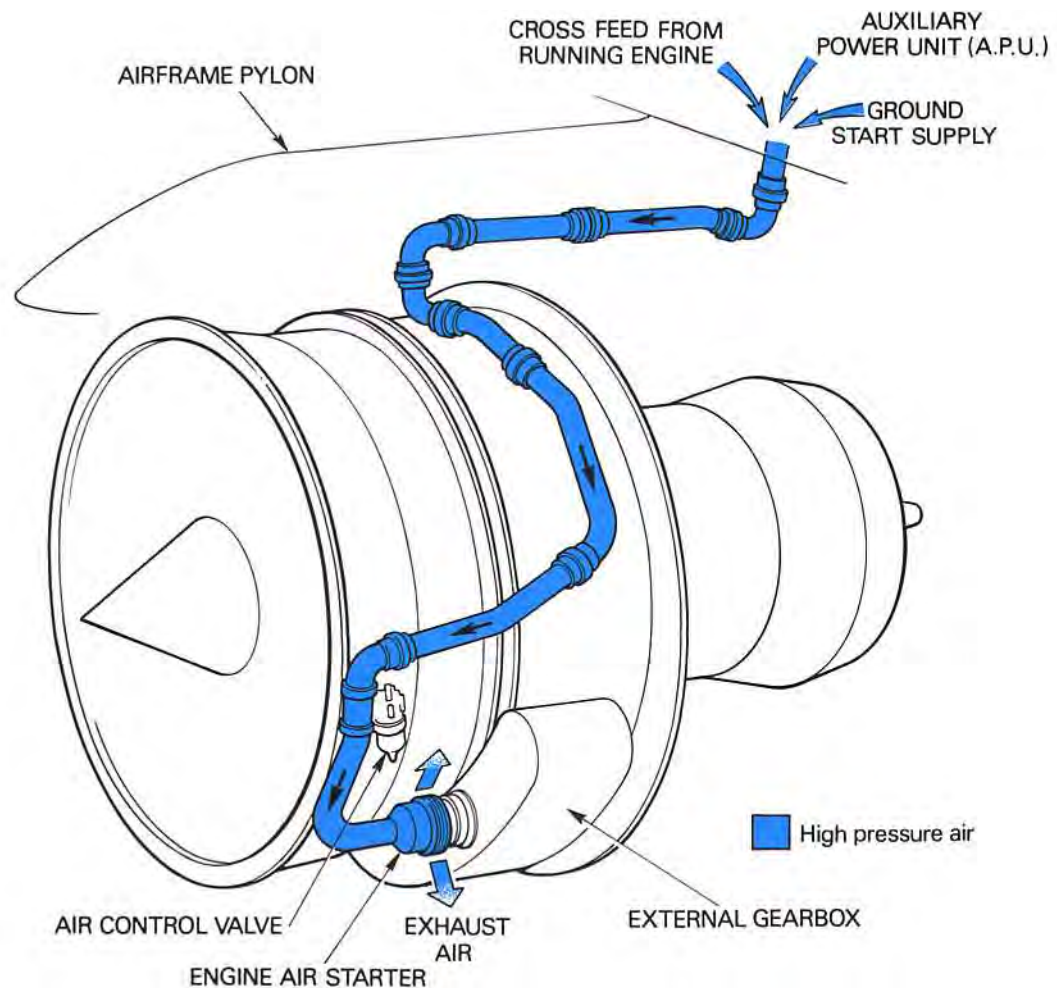
ENGINE AIR STARTER

Fig. 11-7   An air starting system.

operates and shuts down the gas turbine starter. As the starter runs down, the clutch automatically disengages from the output shaft and the main engine accelerates up to idling r.p.m. under its own power.

Hydraulic

17. Hydraulic starting is used for starling some small jet engines. In most applications, one of the engine-mounted hydraulic pumps is utilized and is known as a pump/starter, although other applications may use a separate hydraulic motor. Methods of transmitting the torque to the engine may vary, but a typical system would include a reduction gear and clutch assembly. Power to rotate the pump/starter is provided by hydraulic pressure from a ground supply unit and is transmitted to the engine through the reduction gear and clutch. The starting system is controlled by an electrical circuit that also operates hydraulic valves so that on completion of the starting

cycle the pump /starter functions as a normal hydraulic pump.

IGNITION

18. High-energy (H.E.) ignition is used for starting all jet engines and a dual system is always fitted. Each system has an ignition unit connected to its own igniter plug, the two plugs being situated in different positions in the combustion system.

19. Each H.E. ignition unit receives a low voltage supply, controlled by the starting system electrical circuit, from the aircraft electrical system. The electrical energy is stored in the unit until, at a pre-determined value, the energy is dissipated as a high voltage, high amperage discharge across the igniter plug.

20. Ignition units are rated in 'joules' (one joule equals one watt per second). They are designed to

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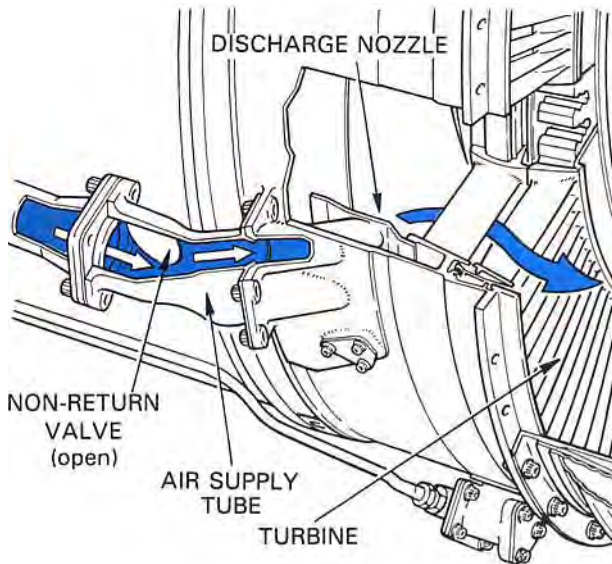
Fig. 11-8   Air impingement starting.

give outputs which may vary according to requirements. A high value output (e.g. twelve joule) is necessary to ensure that the engine will obtain a satisfactory relight at high altitudes and is sometimes necessary for starting. However, under certain flight conditions, such as icing or take-off in heavy rain or snow, it may be necessary to have the ignition system continuously operating to give an automatic relight should flame extinction occur. For this condition, a low value output (e.g. three to six joule) is preferred because it results in a longer life of the igniter plug and ignition unit. Consequently, to suit all engine operating conditions, a combined system giving a high and low value output is favoured. Such a system would consist of one unit emitting a high output to one igniter plug, and a second unit giving a low output to a second igniter plug. However, some ignition units are capable o! supplying both high and low outputs, the value being pre-selected as required.
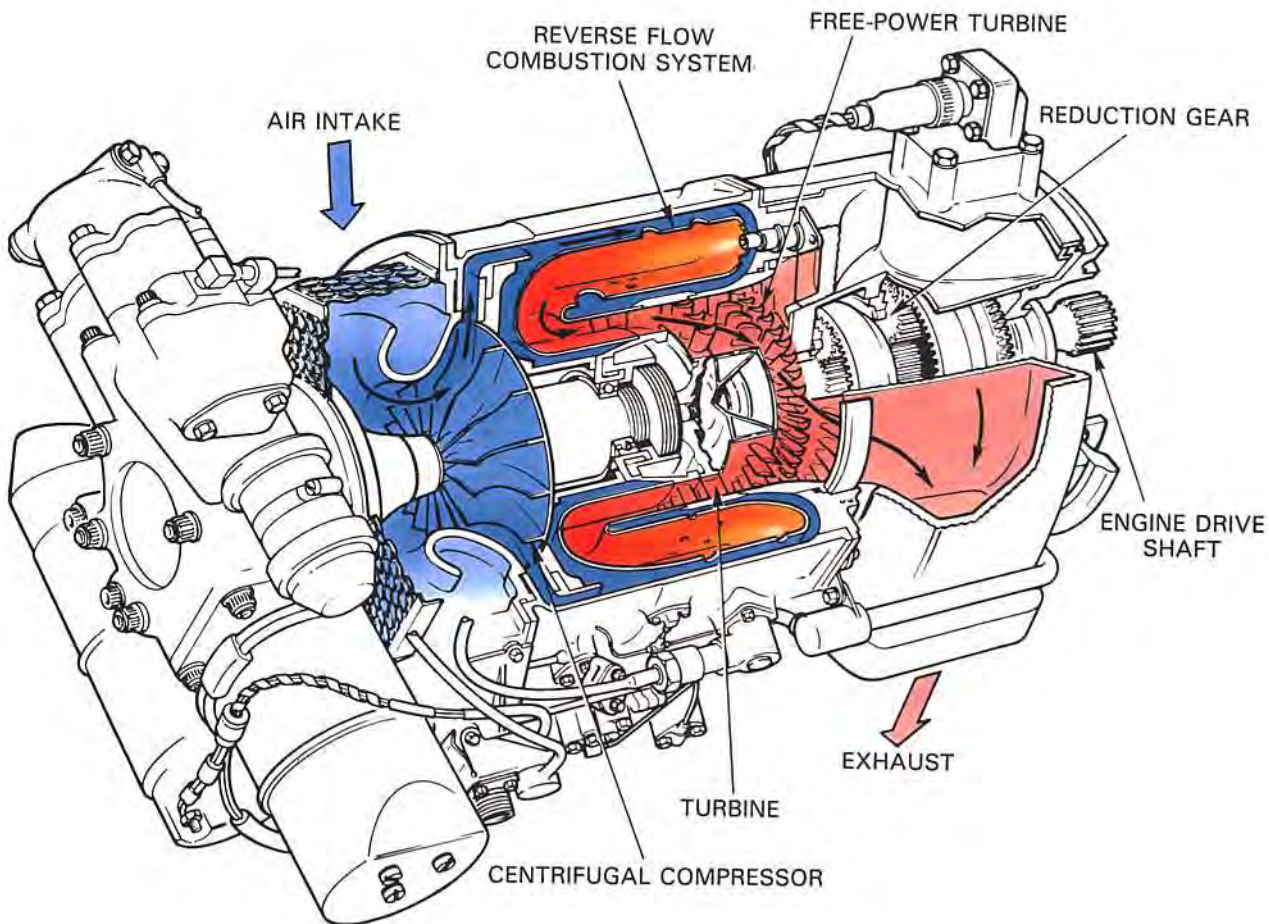
Fig. 11-9   A gas turbine starter.

21. An ignition unit may be supplied with direct current (D.C.) and operated by a trembler mechanism or a transistor chopper circuit, or supplied with alternating current (A.C.) and operated by a transformer. The operation of each type of unit is described in the subsequent paragraphs.

22. The ignition unit shown in fig. 11-10 is a typical D.C. trembler-operated unit. An induction coil, operated by the trembler mechanism, charges the reservoir capacitor (condenser) through a high voltage rectifier. When the voltage in the capacitor is equal to the breakdown value of a sealed discharge gap, the energy is discharged across the face of the igniter plug. A choke is fitted to extend the duration of the discharge and a discharge resistor is fitted to

ensure that any residual stored energy in the capacitor is dissipated within one minute of the system being switched off. A safety resistor is fitted to enable the unit to operate safely, even when the high tension lead is disconnected and isolated.

23. Operation of the transistorized ignition unit is similar to that of the D.C. trembler-operated unit, except that the trembler-unit is replaced by a transistor chopper circuit. A typical transistorized unit is shown in fig. 11-11; such a unit has many advantages over the trembler-operated unit because it has no moving parts and gives a much longer operating life. The size of the transistorized unit is reduced and its weight is less than that of the trembler-operated unit.
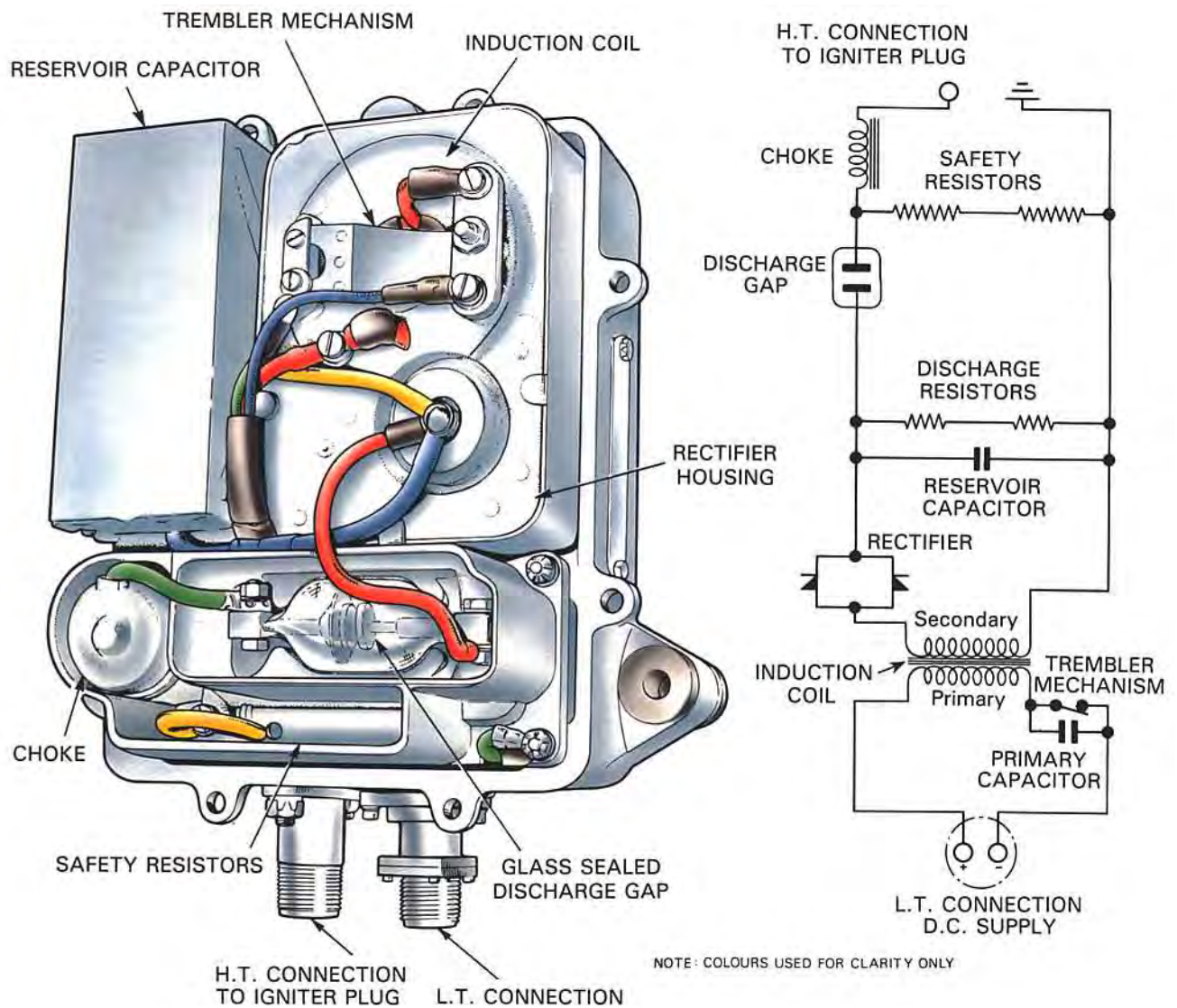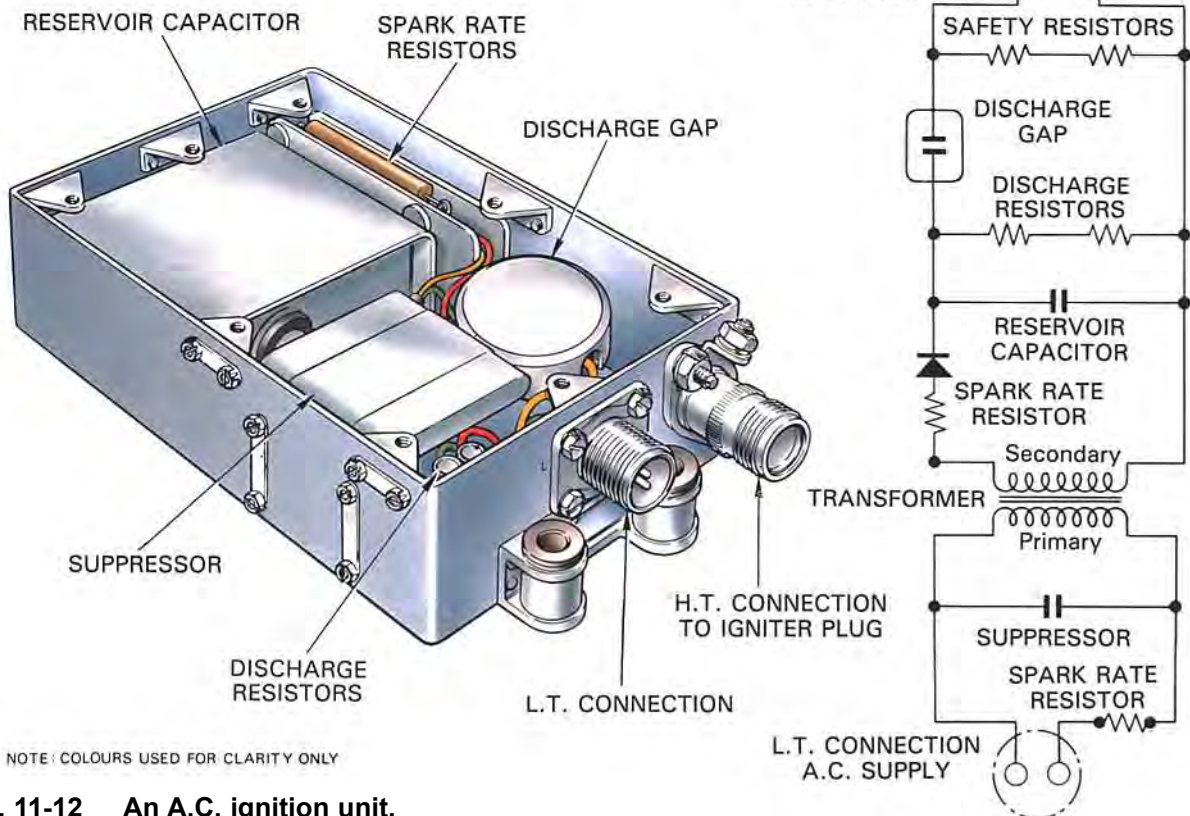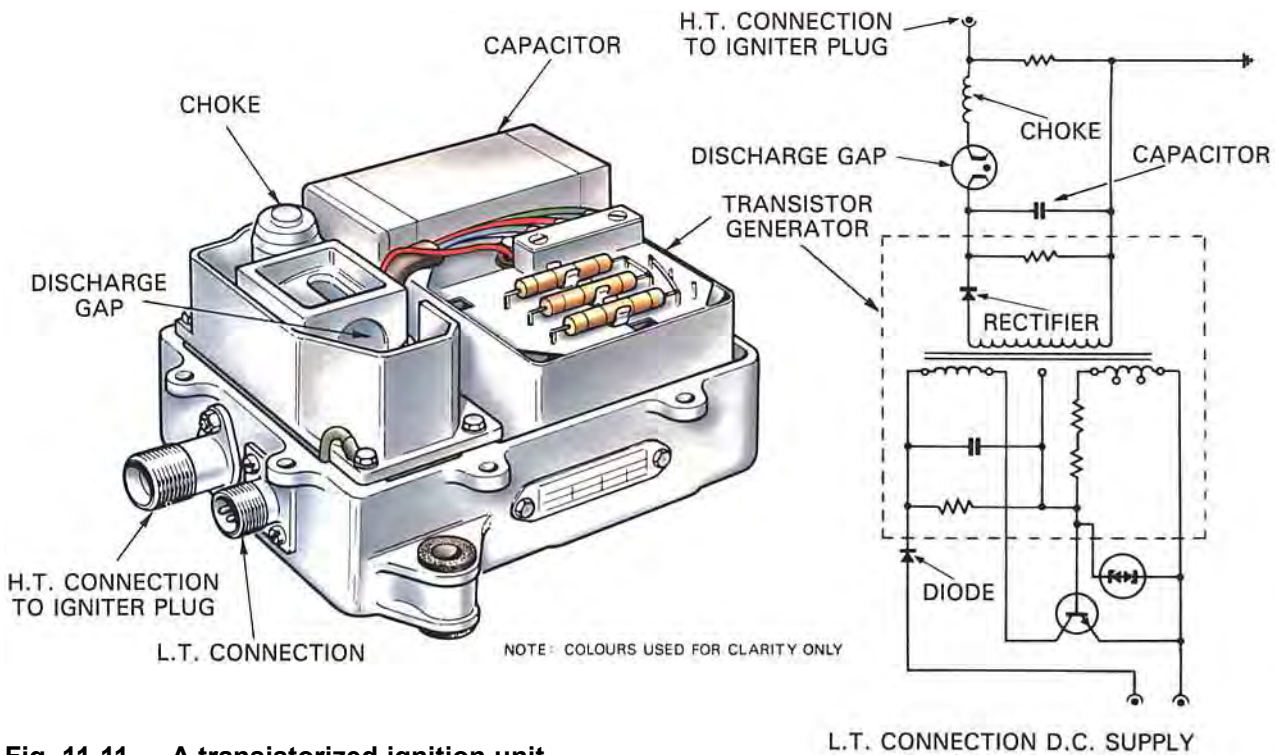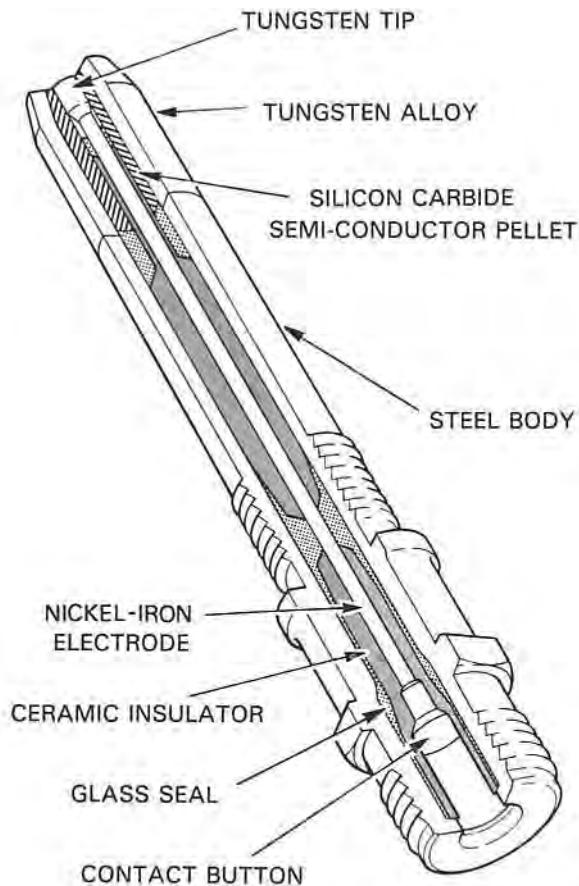

Fig. 11-10   A D.C. trembler-operated ignition unit.

Fig. 11-11   A transistorized ignition unit.

Fig. 11-12   An A.C. ignition unit.

Fig. 11-13   An igniter plug.

Fig. 11-14   A typical flight relight envelope.

24. The A.C. ignition unit, shown in fig, 11-12, receives an alternating current which is passed through a transformer and rectifier to charge a capacitor. When the voltage in the capacitor is equal to the breakdown value of a sealed discharge gap, the capacitor discharges the energy across the face of the igniter plug. Safety and discharge resistors are fitted as in the trembler-operated unit.

25. There are two basic types of igniter plug; the constricted or constrained air gap type and the shunted surface discharge type. The air gap type is similar in operation to the conventional reciprocating engine spark plug, but has a larger air gap between the electrode and body for the spark to cross. A potential difference of approximately 25,000 volts is required to ionize the gap before a spark will occur. This high voltage requires very good insulation throughout the circuit. The surface discharge igniter plug (fig. 11-13) has the end of the insulator formed by a semi-conducting pellet which permits an electrical leakage from the central high tension electrode to the body. This ionizes the surface of the pellet to provide a low resistance path for the energy stored in the capacitor. The discharge takes the form of a high intensity flashover from the electrode to the body and only requires a potential difference of approximately 2000 volts for operation.

26. The normal spark rate of a typical ignition system is between 60 and 100 sparks per minute. Periodic replacement of the igniter plug is necessary due to the progressive erosion of the igniter electrodes caused by each discharge.

27. The igniter plug tip protrudes approximately 0.1 inch into the flame tube. During operation the spark penetrates a further 0.75 inch. The fuel mixture is ignited in the relatively stable boundary layer which then propagates throughout the combustion system.

RELIGHTING

28. The jet engine requires facilities for relighting should the flame in the combustion system be extinguished during flight. However, the ability of the engine to relight will vary according to the altitude and forward speed of the aircraft. A typical relight envelope, showing the flight conditions under which an engine will obtain a satisfactory relight, is shown in fig. 11-14. Within the limits of the envelope, the airflow through the engine will rotate the compressor at a speed satisfactory for relighting; all that is required therefore, provided that a fuel supply is available, is the operation of the ignition system. This is provided for by a separate switch that operates only the ignition system.

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Rolls-Royce contra-rotating fan (concept)
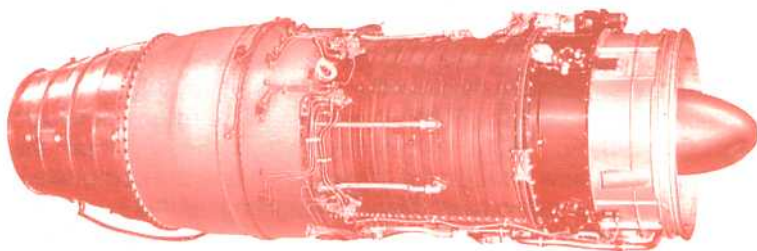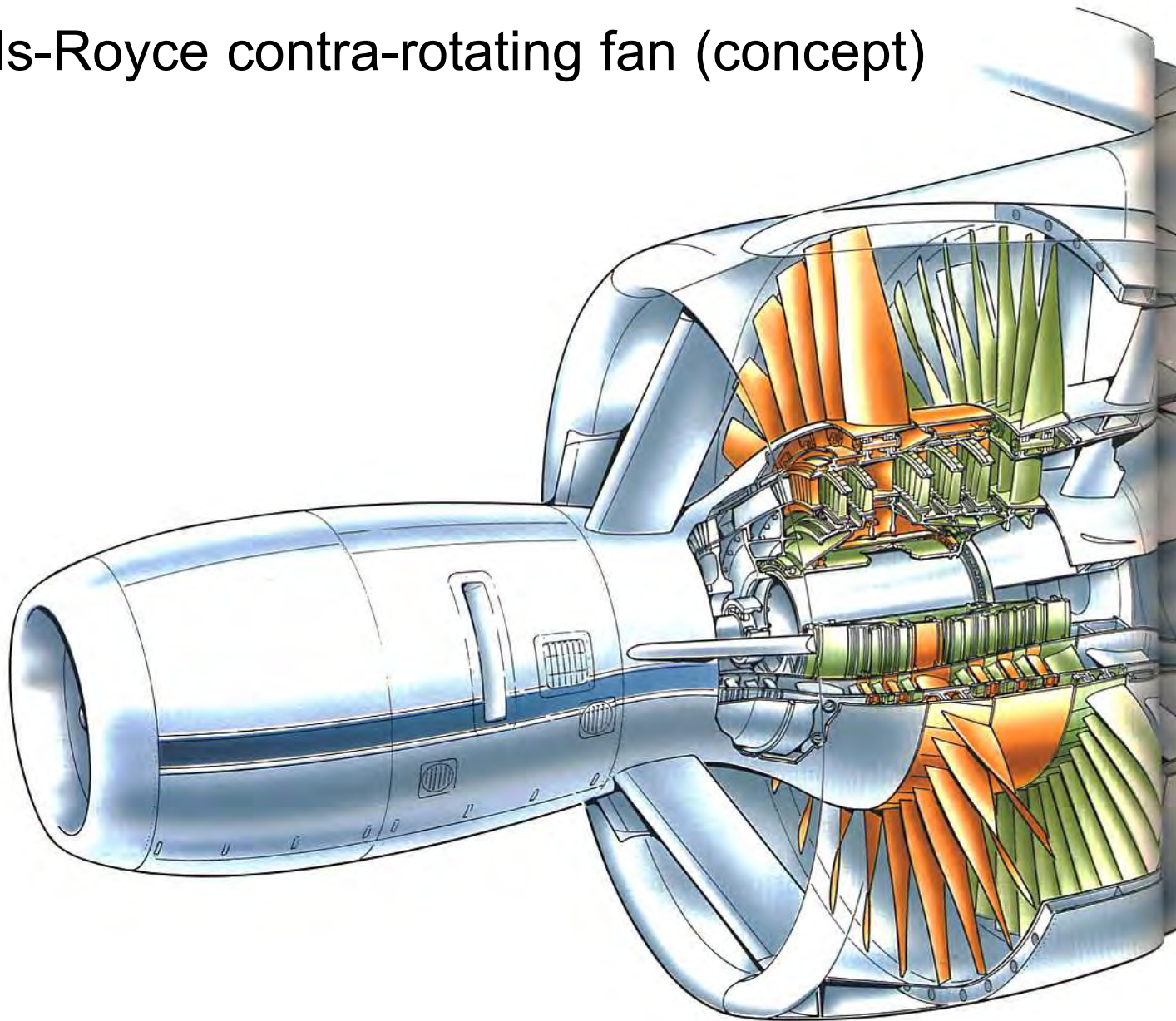
The Sapphire originated in 1946 with the Metrovick F9, which was handed over to Armstrong-Siddeley when Metropolitan-Vickers withdrew from aviation in 1947. The Sapphire first ran in October 1948 and the engine was flight tested in Meteor, Hastings and Canberra aircraft; before going into production for the Gloster Javelin and Hawker Hunter F2.

Armstrong Siddeley Sapphire

12: Controls and instrumentation

Contents Page

INTRODUCTION

1. The controls of the gas turbine engine are designed to remove, as far as possible, work load from the pilot while still allowing him ultimate control of the engine. To achieve this, the fuel flow is automatically controlled after the pilot has made the initial power selection (Part 10).

2. All engine parameters require monitoring and instrumentation is provided to inform the pilot of the correct functioning of the various engine systems and to warn of any impending failure. Should any of the automatic governors fail, the engine can be manually controlled by the pilot selecting the desired thrust setting and monitoring the instruments to maintain the engine within the relevant operating limitations.

3. The multitude of dials and gauges on the pilot's instrument panel may be replaced by one or a number of cathode ray tubes to display engine parameters. These are small screens capable of displaying all of the information necessary to operate the engine safely.

CONTROLS

4. The control of a gas turbine engine generally requires the use of only one control lever and the monitoring of certain indicators located on the pilot's instrument panel (fig. 12-1). Operation of the control (throttle/power) lever selects a thrust level which is then maintained automatically by the fuel system (Part 10).

Fig. 12-1   Pilot's instrument panel - turbo-jet engines.

5. On engines fitted with afterburning, single lever control is maintained, although a further fuel system is required to supply and control the fuel to the afterburner (Part 16).

6. On a turbo-propeller engine, the throttle lever is interconnected with the propeller control unit (P.C.U.), thus maintaining single lever operation of the engine. Similarly, the throttle control lever of a helicopter is interconnected with the collective pitch lever, so ensuring that an increase in pitch is accompanied by an increase in engine power,

7. The fuel system (Part 10) incorporates a high pressure fuel shut-off cock to provide a means of stopping the engine. This may be operated by a separate lever, interconnected with the throttle lever, or electrically actuated and controlled by a switch on the pilot's instrument panel.

8. A turbo-jet engine fitted with a thrust reverser usually has an additional control lever that allows reverse thrust to be selected (Part 15). On a turbo-propeller engine, a separate control lever is not required because the interconnected throttle and P.C.U. lever is operated to reverse the pitch of the propeller.

INSTRUMENTATION

9. The performance of the engine and the operation of the engine systems are shown on gauges or by the operation of flag or dolls-eye type indicators. A diagrammatic arrangement of the control and instrumentation for a turbo-jet engine is shown in fig. 12-2.
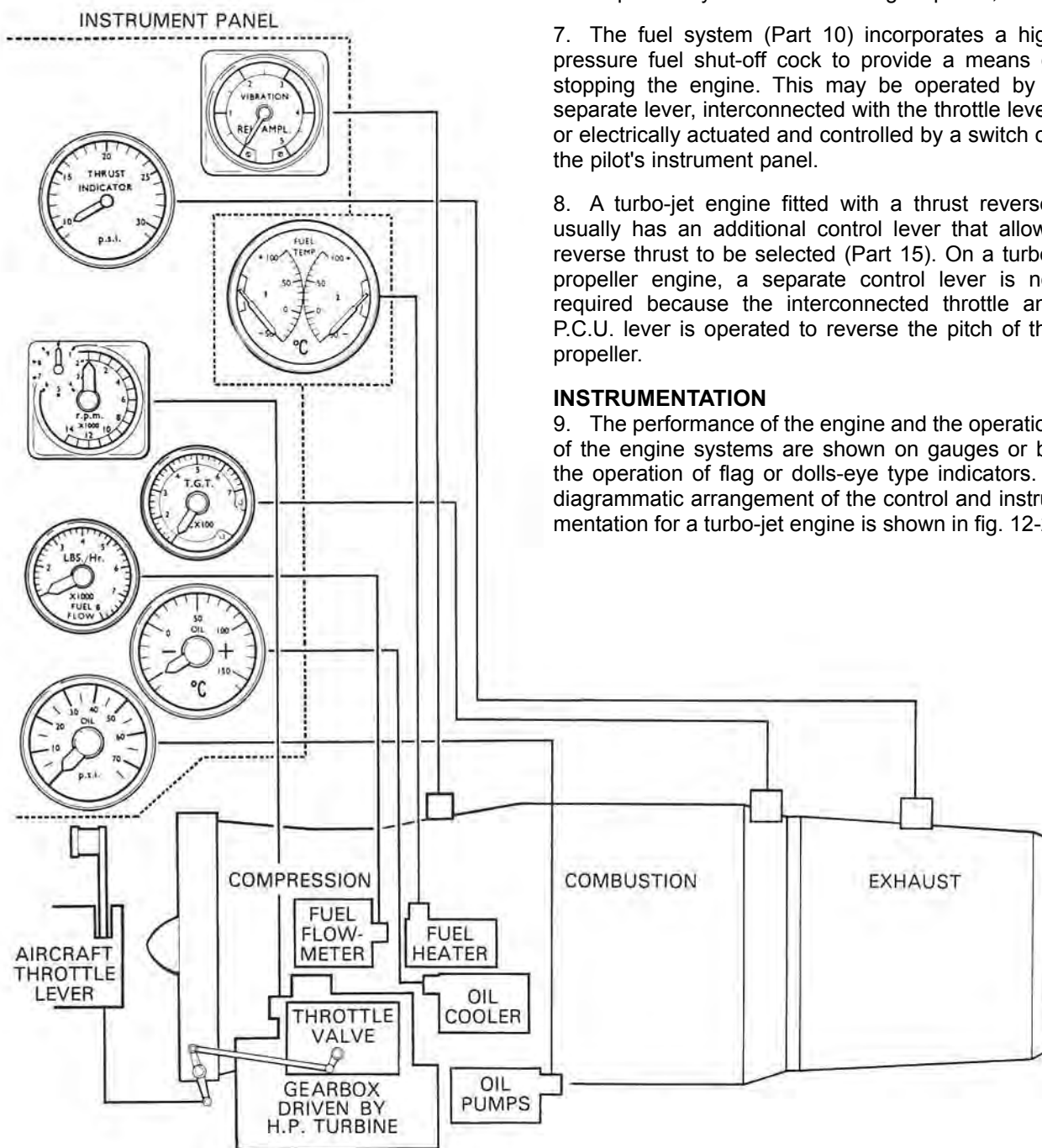
Fig. 12-2   Diagrammatic arrangement of engine control and instrumentation.
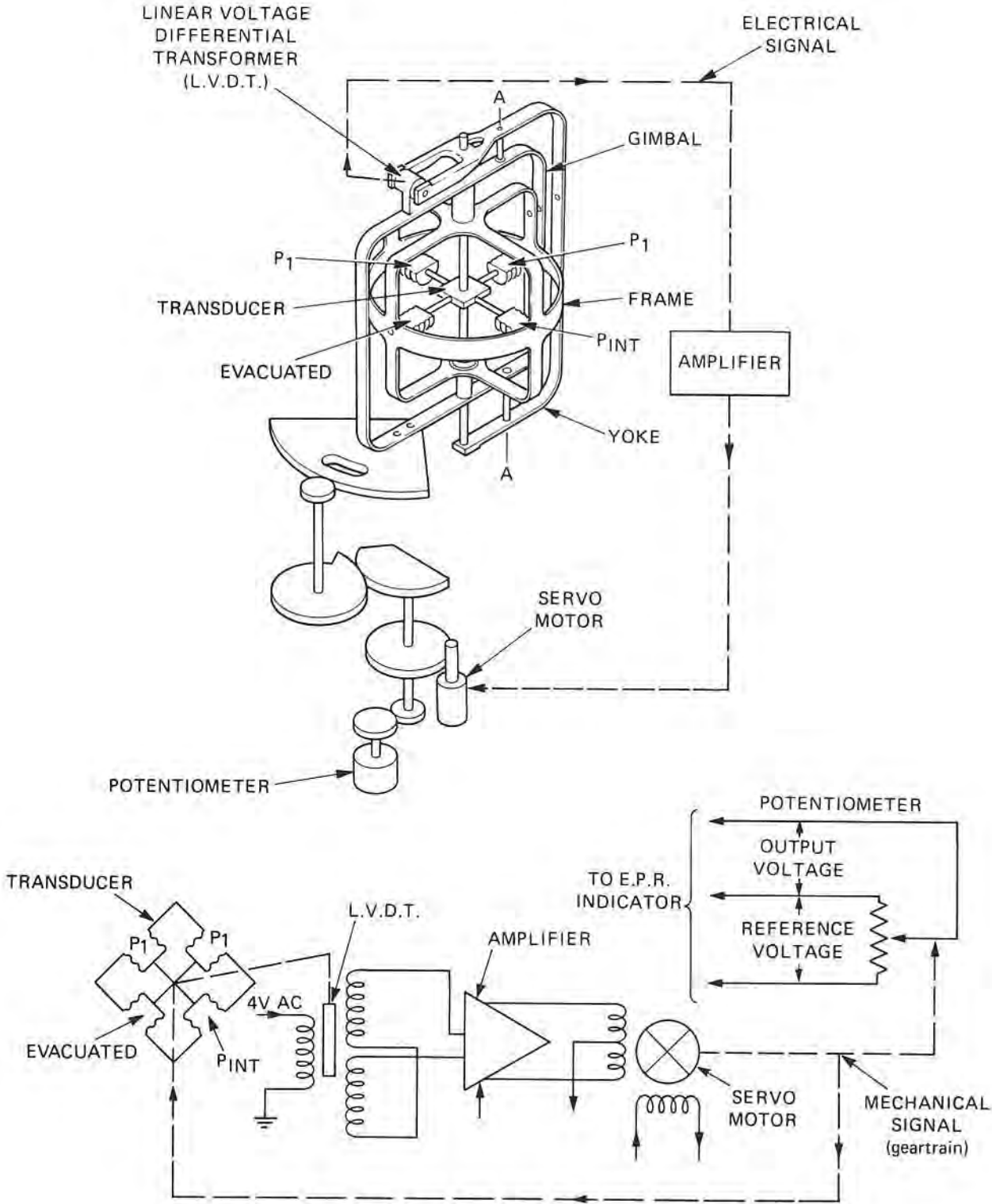
Fig. 12-3   Electro-mechanical E.P.R. transmitter.

Engine thrust

10. The thrust of an engine is shown on a thrust-meter, which will be one of two basic types; the first measures turbine discharge or jet pipe pressure, and the second, known as an engine pressure ratio (E.P.R.) gauge, measures the ratio of two or three parameters. When E.P.R. is measured, the ratio is usually that of jet pipe pressure to compressor inlet pressure. However, on a fan engine the ratio may be that of integrated turbine discharge and fan outlet pressures to compressor inlet pressure.

11. In each instance, an indication of thrust output is given, although when only the turbine discharge pressure is measured, correction is necessary for variation of inlet pressure; however, both types may require correction for variation of ambient air temperature. To compensate for ambient atmospheric conditions, it is possible to set a correction figure to a sub-scale on the gauge; thus,

the minimum thrust output can be checked under all operating conditions.

12. Suitably positioned pilot tubes sense the pressure or pressures appropriate to the type of indication being taken from the engine. The pilot tubes are either directly connected to the indicator or to a pressure transmitter that is electrically connected to the indicator.

13. An indicator that shows only the turbine discharge pressure is basically a gauge, the dial of which may be marked in pounds per square inch (p.s.i.), inches of mercury (in. Hg.), or a percentage of the maximum thrust.

14. E.P.R. can be indicated by either electro-mechanical or electronic transmitters. In both cases the inputs to the transmitter are engine inlet pressure (P1) and an integrated pressure (PINT) comprised of

The torquemeter measures hydraulically the axial load produced by the helical gears when transmitting a driving torque to the propeller

HELICAL GEAR

TORQUEMETER PISTON

PROPELLER SHAFT

Axial thrust

Engine oil pressure

Torquemeter oil pressure

Fig. 12-4   A simple torquemeter system.

fan outlet and turbine exhaust pressures. In some cases either fan outlet pressure or turbine exhaust pressure are used alone in place of PINT.

15. The electro-mechanical system indicates a change in pressure by using transducer capsules (fig. 12-3) to deflect the centre shaft of the pressure transducer causing the yoke to pivot about the axis A.A. This movement is sensed by the linear variable differential transformer (L.V.D.T.) and converted to an a.c. electrical signal which is amplified and applied to the control winding of the servo motor.
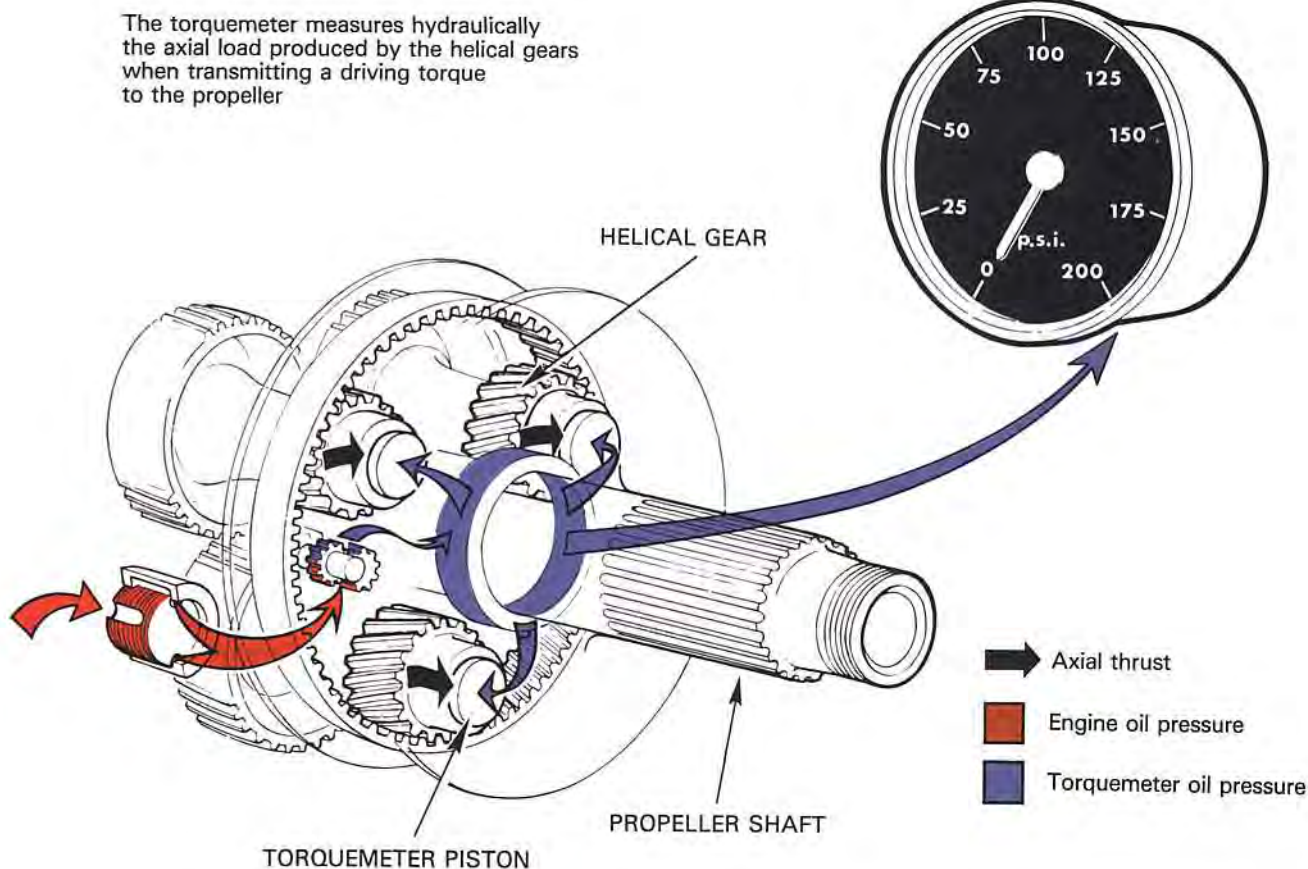
16. The servo motor, through the gears, alters the potentiometer output voltage signal to the E.P.R. indicator and simultaneously drives the gimbal in the same direction as the initial yoke movement until the L.V.D.T. signal to the motor is cancelled and the system stabilizes at the new setting.

17. The electronic E.P.R. system utilizes two vibrating cylinder pressure transducers which sense the engine air pressures and vibrate at frequencies relative to these pressures. From these vibration frequencies electrical signals of E.P.R. are computed and are supplied to the E.P.R. gauge and electronic engine control system (Part 10).

Engine torque

18. Engine torque is used to indicate the power that is developed by a turbo-propeller engine, and the indicator is known as a torquemeter. The engine torque or turning moment is proportional to the horse-power and is transmitted through the propeller reduction gear.

19. A torquemeter system is shown in fig. 12-4. In this system, the axial thrust produced by the helical gears is opposed by oil pressure acting on a number of pistons; the pressure required to resist the axial thrust is transmitted to the indicator.

20. In addition to providing an indication of engine power; the torquemeter system may also be used to automatically operate the propeller feathering system if the torquemeter oil pressure falls due to a power failure. It is also used, on some installations, to assist in the automatic operation of the water injection system to restore or boost the take-off power at high ambient temperatures or at high altitude airports (Part 17).

Engine speed

21. All engines have their rotational speed (r.p.m.) indicated. On a twin or triple-spool engine, the high pressure assembly speed is always indicated; in most instances, additional indicators show the speed
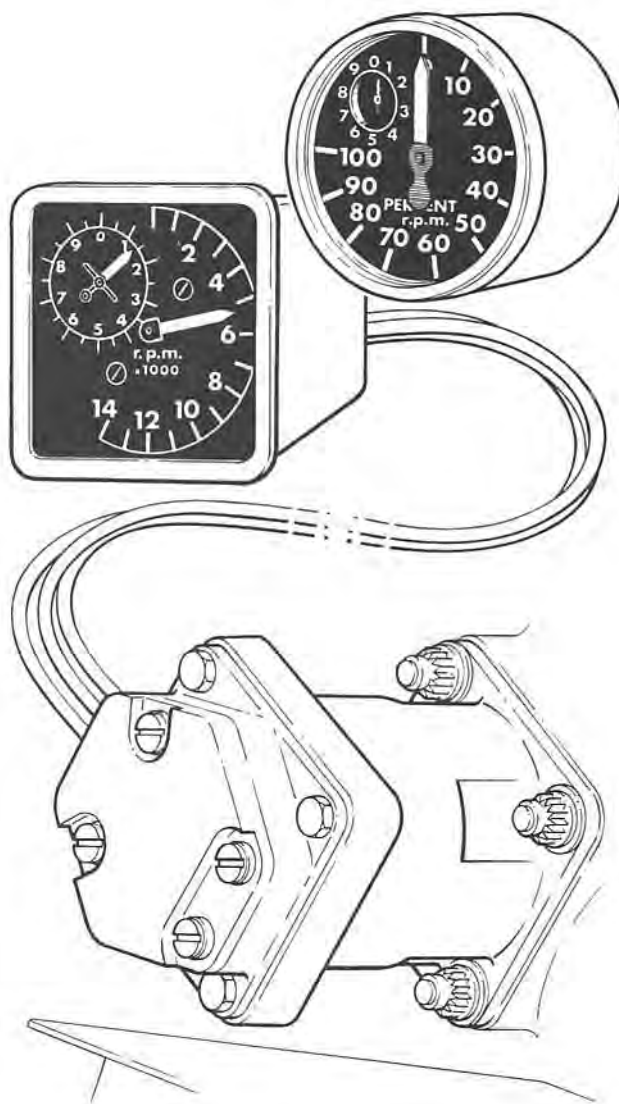
Fig. 12-5   **Engine speed indicators and generator.**

of the low pressure and intermediate pressure assemblies.

22. Engine speed indication is electrically transmitted from a small generator, driven by the engine, to an indicator that shows the actual revolutions per minute (r.p.m.), or a percentage of the maximum engine speed (fig. 12-5). The engine speed is often used to assess engine thrust, but it
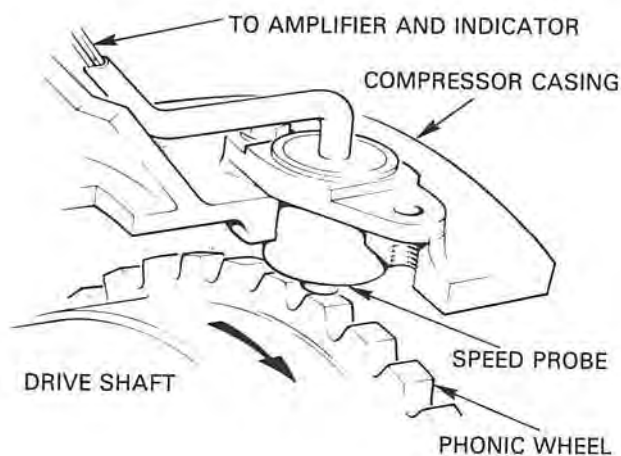
Fig. 12-6   Variable-reluctance speed probe and phonic wheel.

does not give an absolute indication of the thrust being produced because inlet temperature and pressure conditions affect the thrust at a given engine speed.

23.   The engine speed generator supplies a three-phase alternating current, the frequency of which is dependent upon engine speed. The generator output frequency controls the speed of a synchronous motor in the indicator, and rotation of a magnet assembly housed in a drum or drag cup induces movement of the drum and consequent movement of the indicator pointer,

24.   Where there is no provision for driving a generator, a variable-reluctance speed probe, in conjunction with a phonic wheel, may be used to induce an electric current that is amplified and then transmitted to an indicator (fig. 12-6). This method can be used to provide an indication of r.p.m. without the need for a separately driven generator, with its associated drives, thus reducing the number of components and moving parts in the engine.

25.   The speed probe is positioned on the compressor casing in line with the phonic wheel, which is a machined part of the compressor shaft. The teeth on the periphery of the wheel pass the probe once each revolution and induce an electric current by varying the magnetic flux across a coil in the probe. The magnitude of the current is governed by the rate of change of the magnetic flux and is thus directly related to engine speed.

26.   In addition to providing an indication of rotor speed, the current induced at the speed probe can be used to illuminate a warning lamp on the instrument panel to indicate to the pilot that a rotor assembly is turning. This is particularly important at engine start, because it informs the pilot when to open the fuel cock to allow fuel to the engine. The lamp is connected into the slatting circuit and is illuminated during the starting cycle.

Turbine gas temperature

27.   The temperature of the exhaust gases is always indicated to ensure that the temperature of the turbine assembly can be checked at any specific operating condition. In addition, an automatic gas temperature control system is usually provided, to ensure that the maximum gas temperature is not exceeded (Part 10).

28.   Turbine gas temperature (T.G.T.) sometimes referred to as exhaust gas temperature (E.G.T.) or jet pipe temperature (J.P.T.), is a critical variable of engine operation and it is essential to provide an indication of this temperature. Ideally, turbine entry temperature (T.E.T.) should be measured; however, because of the high temperatures involved this is not practical, but, as the temperature drop across the turbine varies in a known manner, the temperature at the outlet from the turbine is usually measured by suitably positioned thermocouples. The temperature may alternatively be measured at an intermediate stage of the turbine assembly, as shown in fig. 12-7.

29.   The thermocouple probes used to transmit the temperature signal to the indicator consist of two wires of dissimilar metals that are joined together inside a metal guard tube. Transfer holes in the tube allow the exhaust gas to flow across the junction. The materials from which the thermocouples wires are made are usually nickel-chromium and nickel-aluminium alloys.

30.   The probes are positioned in the gas stream so as to obtain a good average temperature reading and are normally connected to form a parallel circuit. An indicator, which is basically a millivoltmeter calibrated to read in degrees centigrade, is connected into the circuit (fig. 12-8).

31.   The junction of the two wires at the thermocouple probe is known as the 'hot' or 'measuring' junction and that at the indicator as the 'cold' or 'reference' junction. If the cold junction is at a constant temperature and the hot junction is sensing the exhaust gas temperature, an electromotive force (E.M.F.), proportional to the temperature difference
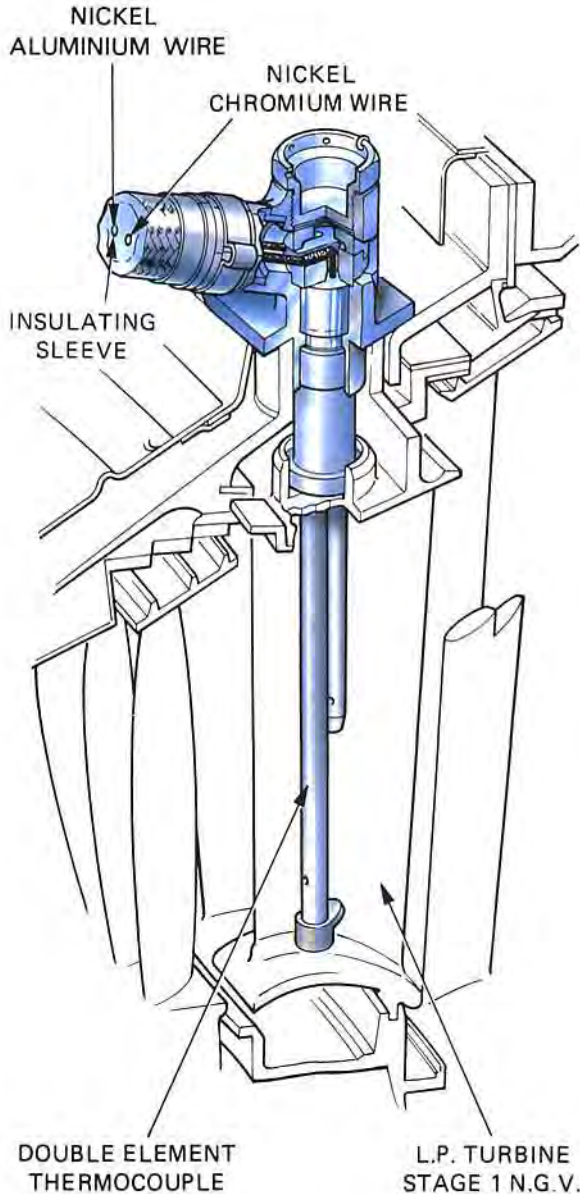
Fig. 12-7   Turbine thermocouple installation.

of the two junctions is created in the circuit and this causes the indicator pointer to move. To prevent variations of cold junction temperature affecting the indicated temperature, an automatic temperature compensating device is incorporated in the indicator or in the circuit.

32.   The thermocouple probes may be of single, double or triple element construction. Where multiple probes are used they are of differing lengths in order to obtain a temperature reading from different points in the gas stream to provide a better average reading than can be obtained from a single probe (fig. 12-7).

33.   The output to the temperature control system can also be used to provide a signal, in the form of short pulses, which, when coupled to an indicator, will digitally record the life of the engine. During engine operation in the higher temperature ranges, the pulse frequency increases progressively causing the cyclic-type indicator to record at a higher rate, thus relating engine or unit life directly to operating temperatures.

34.   Thermocouples may also be positioned to transmit a signal of air intake temperature into the exhaust gas temperature indicating and control systems, thus giving a reading of gas temperature that is compensated for variations of intake temperature. A typical double-element thermocouple system with air intake probes is shown in fig. 12-8.

Oil temperature and pressure
35.   It is essential for correct and safe operation of the engine that accurate indication is obtained of both the temperature and pressure of the oil. Temperature and pressure transmitters and indicators are illustrated in fig 12-9.

36.   Oil temperature is sensed by a temperature-sensitive element fitted in the oil system. A change in temperature causes a change in the resistance value and, consequently, a corresponding change in the current flow at the indicator. The indicator pointer is deflected by an amount equivalent to the temperature change and this is recorded on the gauge in degrees centigrade.

37.   Oil pressure is electrically transmitted to an indicator on the instrument panel. Some installations use a flag-type indicator, which indicates if the pressure is high, normal or low; others use a dial-type gauge calibrated in pounds per square inch (p.s.i.).

38.   Electrical operation of each type is similar; oil pressure, acting on the transmitter, causes a change in the electric current supplied to the indicator. The amount of change is proportional to the pressure applied at the transmitter.

39.   The transmitter may be of either the direct or the differential pressure type. The latter senses the pressure difference between engine feed and return

oil pressures, the return oil being pressurized by cooling and sealing air (Part 9) from the bearings.

40. In addition to a pressure gauge operated by a transmitter, an oil low pressure warning switch may be provided to indicate that a minimum pressure is available for continued safe running of the engine. The switch is connected to a warning lamp in the flight compartment and the lamp illuminates if the pressure falls below an acceptable minimum.

Fuel temperature and pressure

41. The temperature and pressure of the low pressure fuel supply are electrically transmitted to their respective indicators and these show if the low pressure system is providing an adequate supply of fuel without cavitation and at a temperature to suit the operating conditions. The fuel temperature and

pressure indicators are similar to those for temperature and pressure indication.

42. On some engines, a fuel differential pressure switch, fitted to the low pressure fuel filter, senses the pressure difference across the filter element. The switch is connected to a warning lamp that provides indication of partial filter blockage, with the possibility of fuel starvation.

Fuel flow

43. Although the amount of fuel consumed during a given flight may vary slightly between engines of the same type, fuel flow does provide a useful indication of the satisfactory operation of the engine and of the amount of fuel being consumed during the flight. A typical system consists of a fuel flow transmitter, which is fitted into the low pressure fuel system, and an indicator, which shows the rate of fuel flow and the
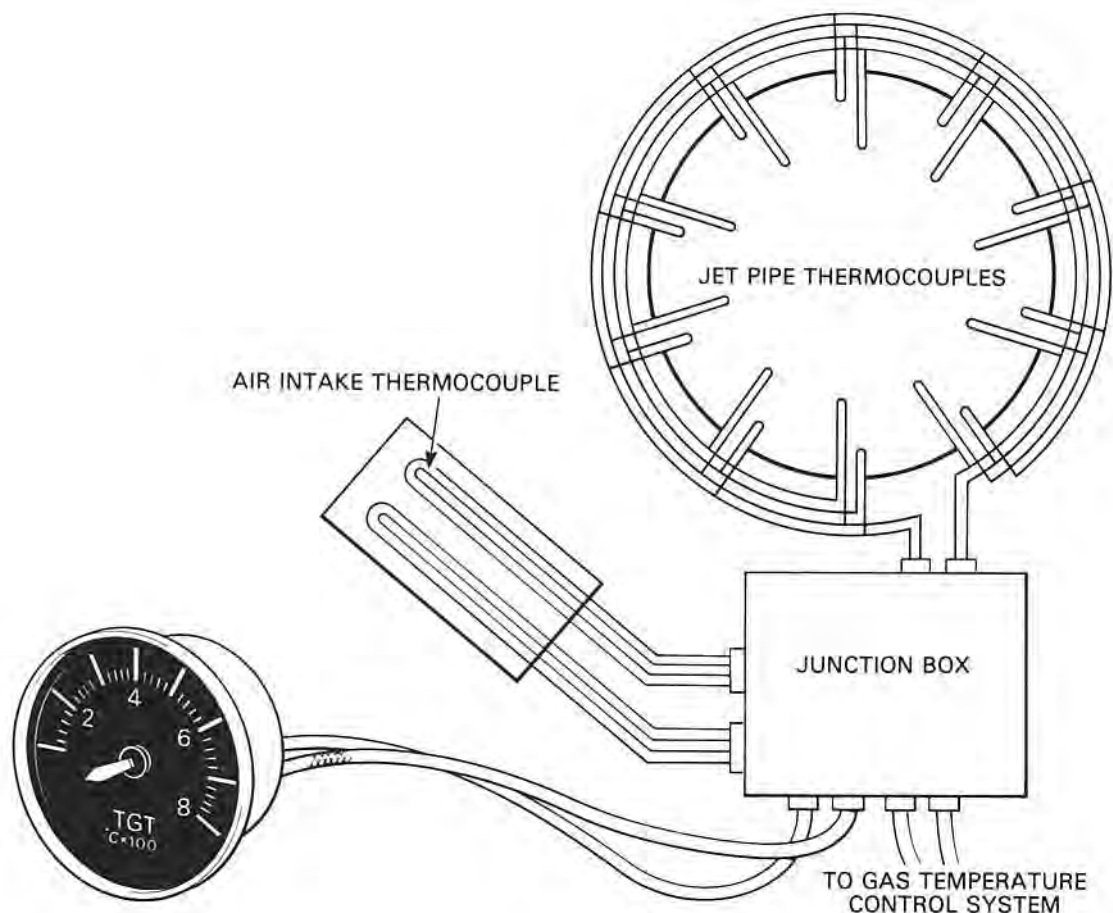
Fig. 12-8   A typical double element thermocouple system.
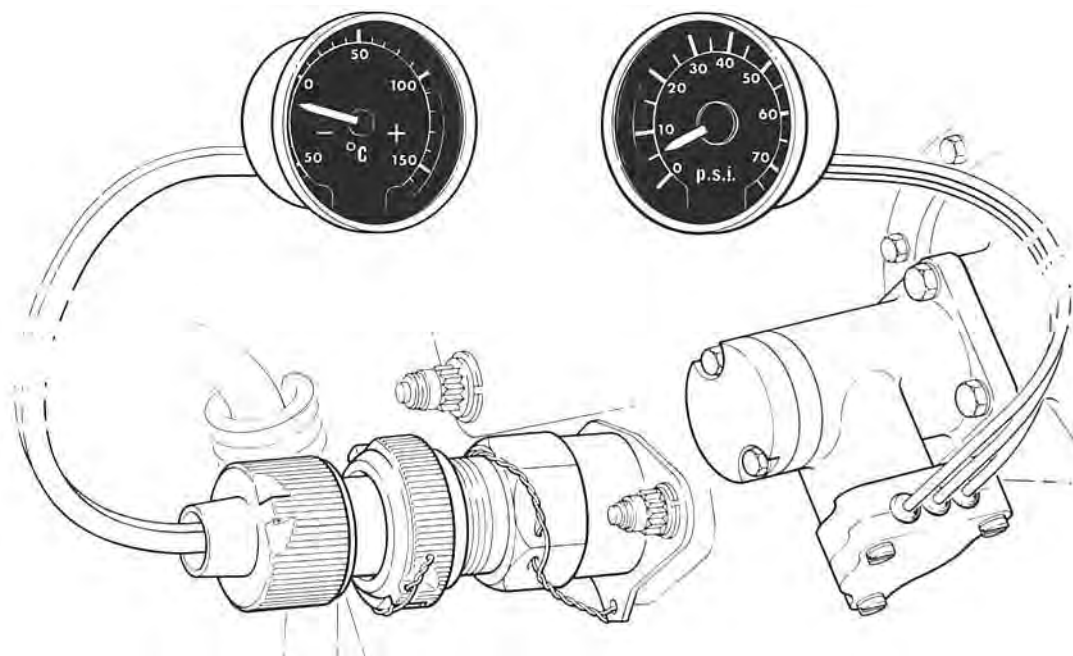
Fig. 12-9   Oil temperature and pressure transmitters and indicators.

total fuel used in gallons, pounds or kilogrammes per hour (fig. 12-10). The transmitter measures the fuel flow electrically and an associated electronic unit gives a signal to the indicator proportional to the fuel flow.

Vibration

44. A turbo-jet engine has an extremely low vibration level and a change of vibration, due to an impending or partial failure, may pass without being noticed. Many engines are therefore fitted with vibration indicators that continually monitor the vibration level of the engine. The indicator is usually a milliammeter that receives signals through an amplifier from engine mounted transmitters (fig. 12-11).

45. A vibration transmitter is mounted on the engine casing and electrically connected to the amplifier and indicator. The vibration sensing element is usually an electro-magnetic transducer that converts the rate of vibration into electrical signals and these cause the indicator pointer to move proportional to the vibration level. A warning lamp on the instrument panel is incorporated in the system to warn the pilot if an unacceptable level of vibration is approached, enabling the engine to be shut down and so reduce the risk of damage.

Fig. 12-10   Fuel flow transmitter and indicator.
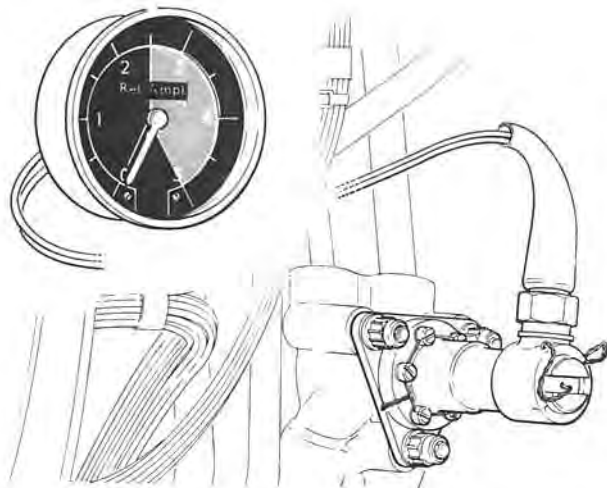
Fig. 12-11   Vibration transmitter and indicator.

46.   The vibration level recorded on the gauge is the sum total of vibration felt at the pick-up. A more accurate method differentiates between the frequency ranges of each rotating assembly and so enables the source of vibration to be isolated. This is particularly important on multi-spool engines.

47.   A crystal-type vibration transmitter, giving a more reliable indication of vibration, has been developed for use on multi-spool engines. A system of filters in the electrical circuit to the gauge makes it possible to compare the vibration obtained against a known frequency range and so locate the vibration source. A multiple-selector switch enables the pilot to select a specific area to obtain a reading of the level of vibration.

Warning systems
48.   Warning systems are provided to give an indication of a possible failure or the existence of a dangerous condition, so that action can be taken to safeguard the engine or aircraft. Although the various systems of an aircraft engine are designed wherever possible to 'fail safe1, additional safety devices are sometimes fitted. Automatic propeller feathering should a power loss occur, and automatic closing of the high pressure fuel shut-off cock should a turbine

shaft failure occur, are but two examples. On some engine types, the fuel system is fitted with a control to enable the engine to be operated by manual throttling should a main fuel system failure occur.

49.   In addition to a fire warning system (Part 14), a number of other audible or visual warning systems can be fitted to a gas turbine engine. These may be for low oil or fuel pressure, excessive vibration or overheating. Indication of these may be by warning light, bell or horn. A flashing light is used to attract the pilot's attention to a central warning panel (C.W.P.) where the actual fault is indicated.

50.   Other instruments and lights warn the pilot of the selected position of the thrust reverser, the fan reverser or the afterburner variable nozzle, when applicable. Gauges also inform the pilot of such things as hydraulic pressure and flow and generator output, which are vital to the correct operation of the aircraft systems.

Aircraft integrated data system
51.   The aircraft integrated data system (A.I.D.S.) is an extension of the 'black box' aircraft accident data recorder. By monitoring and recording various engine parameters, either manually or automatically, it is possible to detect an incipient failure and thus prevent a hazardous situation arising.

52.   Selected performance parameters may be recorded for trend analysis or fault detection (Part 24). Existing instruments are used, wherever possible, to provide the signals to a magnetic tape. Further instrumentation, recording air pressure from points throughout the engine, oil contamination, tank contents and scavenge oil temperature, may be provided as required for flight recording,

53.   After each flight the magnetic tape is processed by computer and the results are analyzed. Any deviation from the normal condition will enable a fault to be identified and the necessary remedial action to be taken.

Electronic indicating systems
54.   Electronic indicating systems consolidate engine indications, systems monitoring, and crew alerting functions onto one or more cathode ray tubes (C.R.T.'s) mounted in the instrument panel. The information is displayed on the screen in the form of dials with digital readout and warnings, cautions and advisory messages shown as text.

Fig. 12-12   Typical electronic indicating display.

55. Only those parameters required by the crew to set and monitor engine thrust are permanently displayed on the screen. The system monitors the remaining parameters and displays them only if one or more exceed safe limitations. The pilot can, however, override the system and elect to have all main parameters in view at any time (fig. 12-12).

56. Warnings, cautions and advisory messages are displayed only when necessary and are colour coded to communicate the urgency of the fault to the flight crew. Provision is made to record any event or out of tolerance parameter in a non-volatile memory for later evaluation by ground maintenance crews.

57. Electronic indicating systems offer improved flight operations by reducing the pilot workload through automatic monitoring of engine operation and a centralized caution and warning system. Reduced flight deck clutter is another feature as the

multitude of instruments traditionally present are replaced by the C.R.T.'s.

SYNCHRONIZING AND SYNCHROPHASING

58. Synchronizing and synchrophasing systems are sometimes used on turbo-propeller engined aircraft to achieve a reduction of noise during flight.

59. On a multi-engined aircraft, a synchronizing system ensures the propeller speeds are all the same. This is achieved by an electrical system that compares speed signals from engine-mounted generators. Out-of-balance signals, using one engine as a master signal, are automatically corrected by electrically trimming the engine speeds until all signals are equal.

60. A synchrophasing system ensures that any given blade of an engine propeller is in the same

relative position as the corresponding, blade of the propeller on the master engine. This again is automatically achieved by very fine trimming of engine speeds resulting from phase signals from the synchrophasing generators.

61. On turbo-jet engines, synchronization can be achieved in a similar manner to that used for a turbo-propeller engine. On multi-spool engines, only one spool is synchronized. Manual trimming of engine or shaft speed can be done with the assistance of a synchroscope. This visually indicates, in comparison with a master engine, if the other engines are running at exactly the same speed; the normal engine speed indicator is, of course, not sufficiently sensitive to use for synchronizing.

Rolls-Royce advanced turbo-propeller
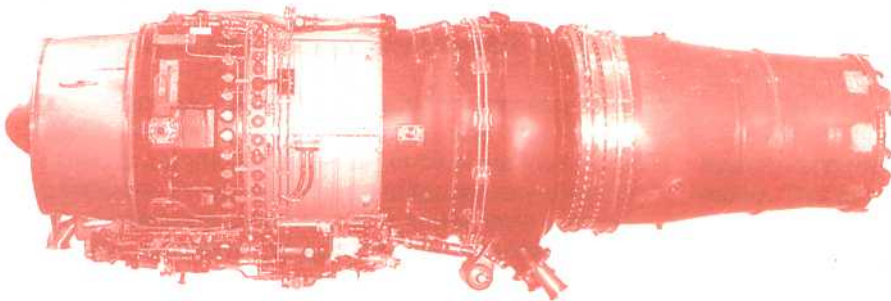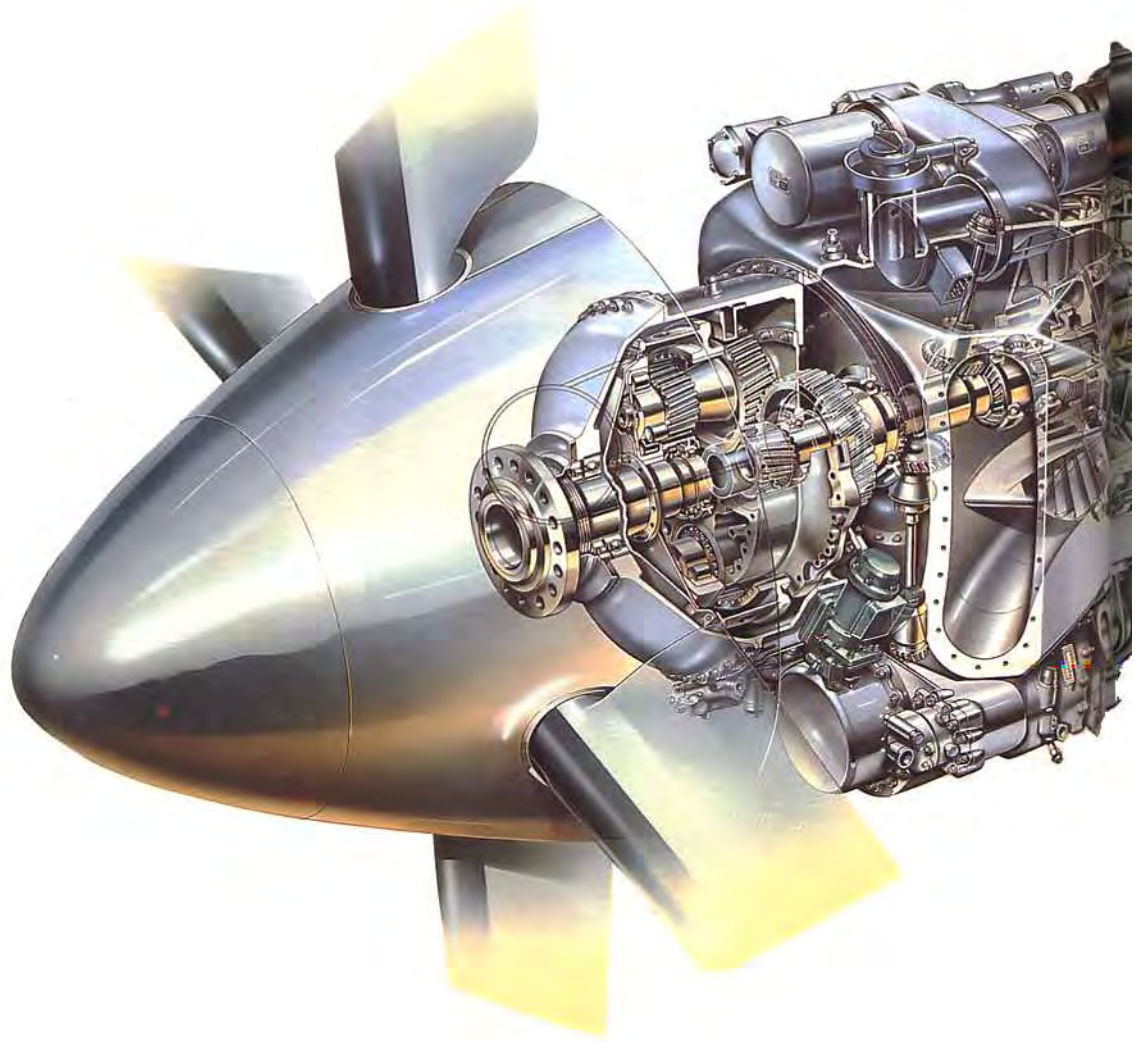
De Havilland H6 Gyron Junior

13: Ice protection

Contents

INTRODUCTION

1. Icing of the engine and the leading edges of the intake duct can occur during flight through clouds containing supercooled water droplets or during ground operation in freezing fog. Protection against ice formation may be required since icing of these regions can considerably restrict the airflow through the engine, causing a loss in performance and possible malfunction of the engine. Additionally, damage may result from ice breaking away and being ingested into the engine or hitting the acoustic material lining the intake duct.

2. An ice protection system must effectively prevent ice formation within the operational requirements of the particular aircraft. The system must be reliable, easy to maintain, present no excessive weight penalty and cause no serious loss in engine performance when in operation.

3. Analyses are carried out to determine whether ice protection is required and, if so, the heat input required to limit ice build up to acceptable levels. Fig. 13-1 illustrates the areas of a turbo-fan engine typically considered for ice protection.

4. There are two basic systems of ice protection; turbo-jet engines generally use a hot air supply (fig. 13-2), and turbo-propeller engines use electrical power or a combination of electrical power and hot

Confirm anti-icing adequate

Ensure icing will not affect performance

Determine heat to prevent icing and ensure anti-icing adequate

Ensure adequate anti-icing (instrumentation)

Determine heat for anti-icing or no heating, icing acceptable

Confirm icing acceptable

Confirm icing acceptable
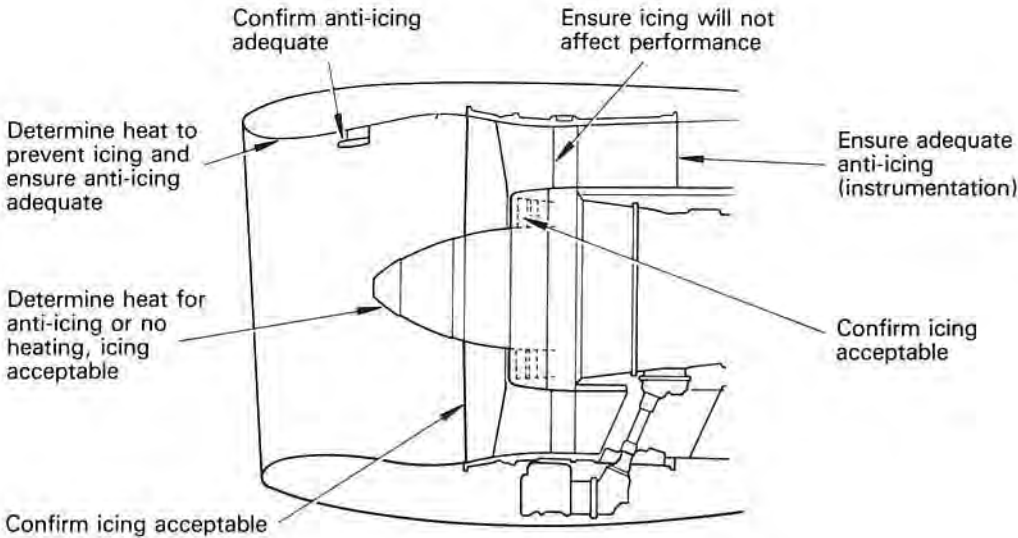
Fig. 13-1   Areas typically considered for ice protection.

NOSE COWL

INTAKE GUIDE VANES

NOSE CONE

PRESSURE REGULATING VALVE

AIR INTAKE MANIFOLD

OUTLET TO NOSE COWL
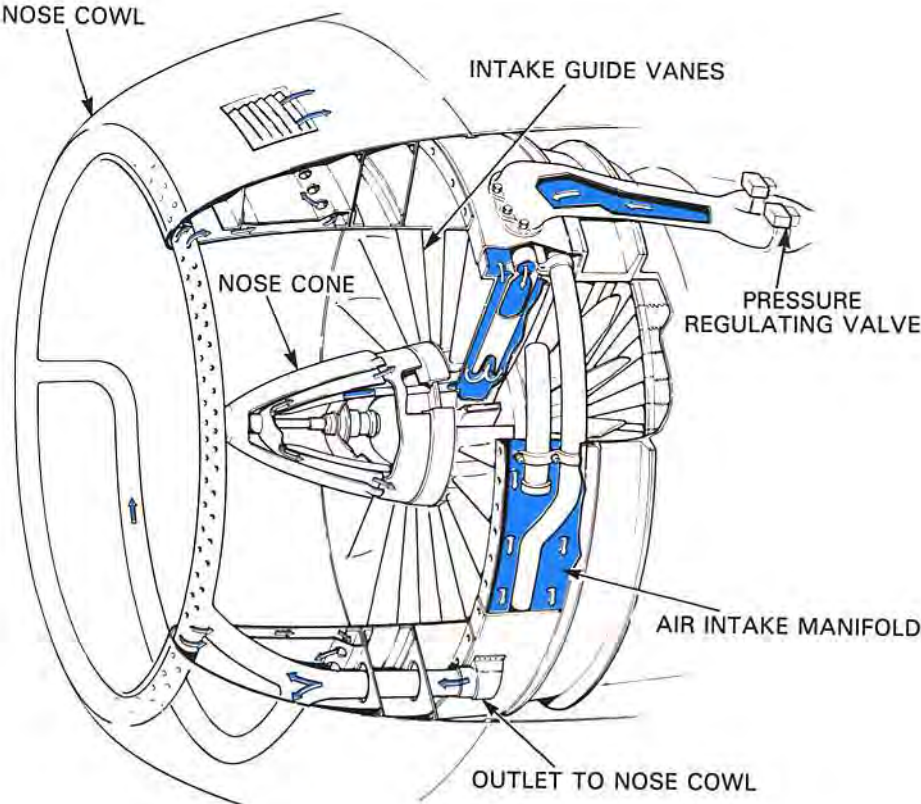
Fig. 13-2   Hot air ice protection.

air. Protection may be supplemented by the circulation of hot oil around the air intake as shown in fig. 13-3. The hot air system is generally used to prevent the formation of ice and is known as an anti-icing system. The electrical power system is used to break up ice that has formed on surfaces and is known as a de-icing system.

HOT AIR SYSTEM

5. The hot air system provides surface heating of the engine and/or powerplant where ice is likely to form. The protection of rotor blades is rarely necessary, because any ice accretions are dispersed by centrifugal action. If stators are fitted upstream of the first rotating compressor stage these may require

protection. If the nose cone rotates it may not need anti-icing if its shape, construction and rotational characteristics are such that likely icing is acceptable.

6. The hot air for the anti-icing system is usually taken from the high pressure compressor stages. It is ducted through pressure regulating valves, to the parts requiring anti-icing. Spent air from the nose cowl anti-icing system may be exhausted into the compressor intake or vented overboard.

7. If the nose cone is anti-iced its hot air supply may be independent or integral with that of the nose cowl and compressor stators. For an independent system, the nose cone is usually anti-iced by a continuous
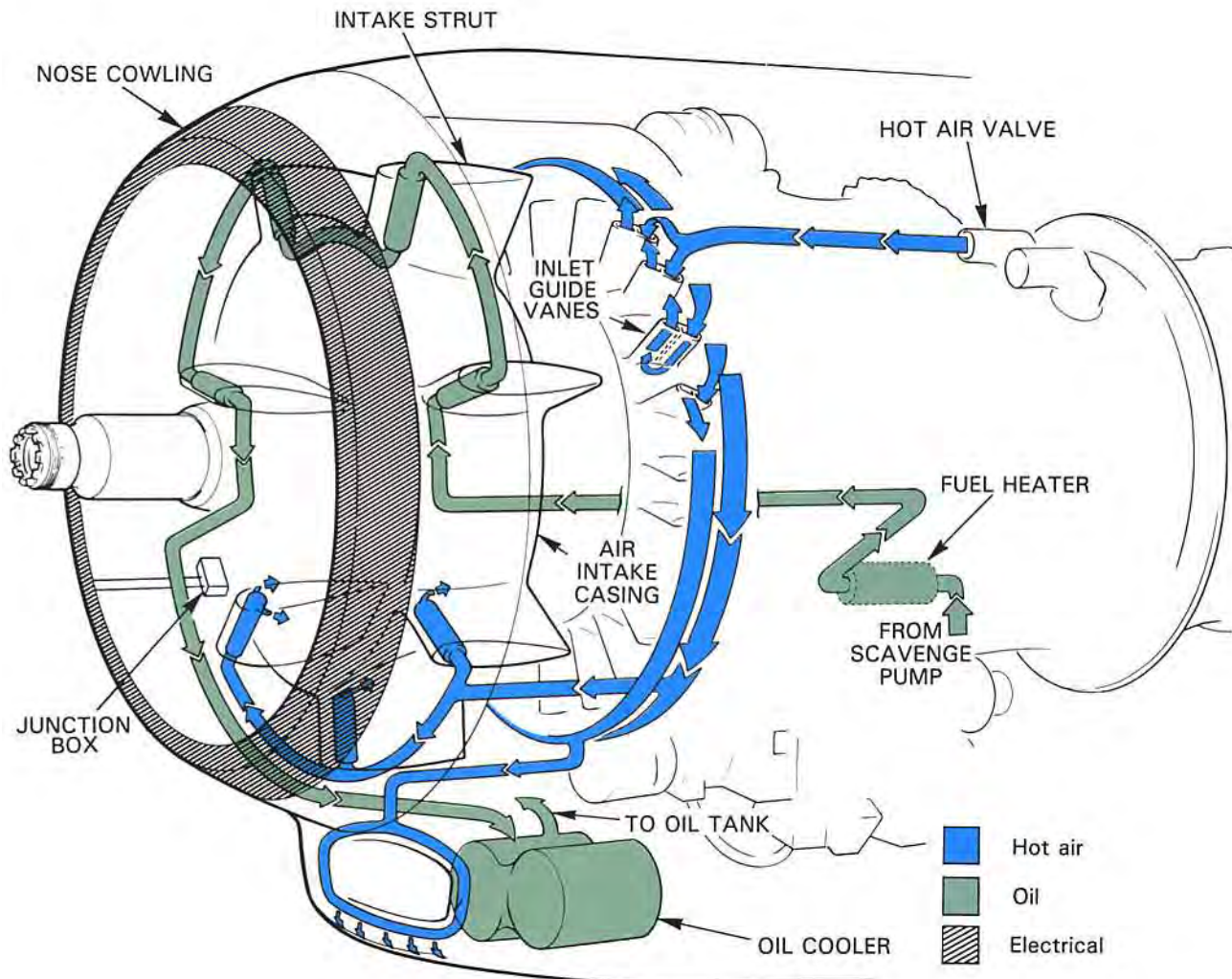
Fig. 13-3   Combination of hot air, oil and electrical ice protection.

unregulated supply of hot air via internal ducting from the compressor.

8. The pressure regulating valves are electrically actuated by manual selection, or automatically by signals from the aircraft ice detection system. The valves prevent excessive pressures being developed in the system, and act also as an economy device at the higher engine speeds by limiting the air offtake from the compressor, thus preventing an excessive loss in performance. The main valve may be manually locked in a pre-selected position prior to take-off in the event of a valve malfunction, prior to replacement.

ELECTRICAL SYSTEM

9. The electrical system of ice protection is generally used for turbo-propeller engine installations, as this form of protection is necessary for the propellers. The surfaces that require electrical heating are the air intake cowling of the engine, the propeller blades and spinner and, when applicable, the oil cooler air intake cowling.

10. Electrical heating pads are bonded to the outer skin of the cowlings. They consist of strip conductors sandwiched between layers of neoprene, or glass cloth impregnated with epoxy resin. To protect the pads against rain erosion, they are coated with a special, polyurethane-based paint. When the de-icing system is operating, some of the areas are continuously heated to prevent an ice cap forming on the leading edges and also to limit the size of the ice that forms on the areas that are intermittently heated (fig. 13-4).

11. Electrical power is supplied by a generator and, to keep the size and weight of the generator to a minimum, the de-icing electrical loads are cycled between the engine, propeller and, sometimes, the airframe.

12. When the ice protection system is in operation, the continuously heated areas prevent any ice
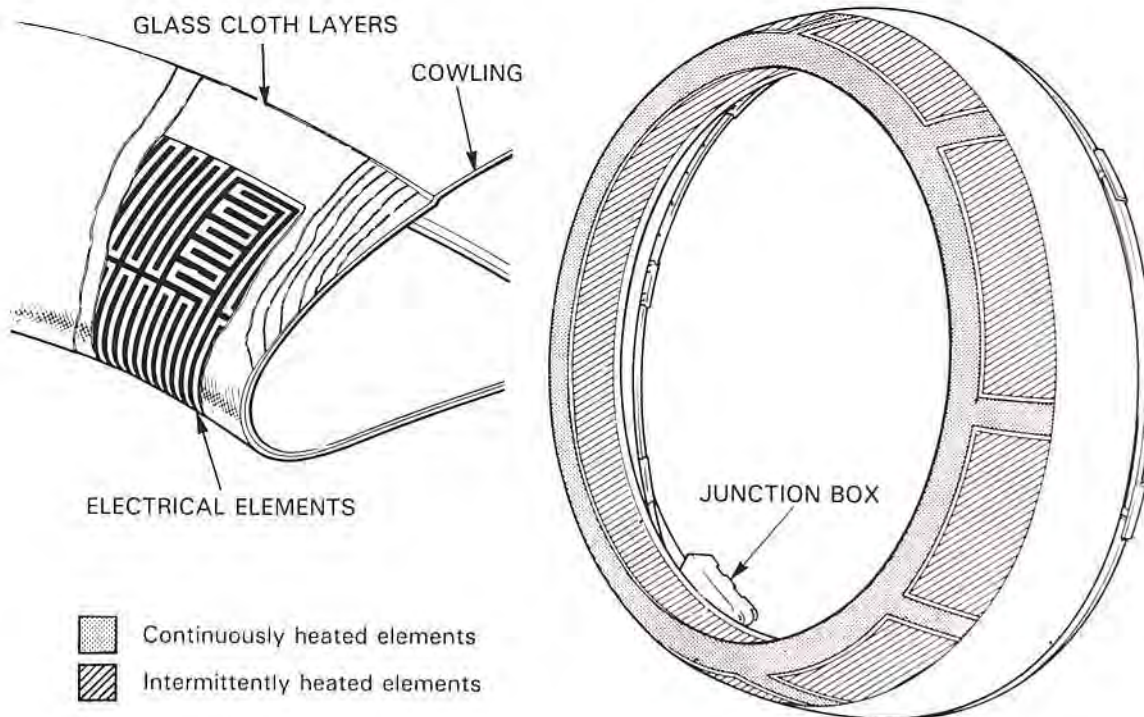
Fig. 13-4   Electrical ice protection.

150
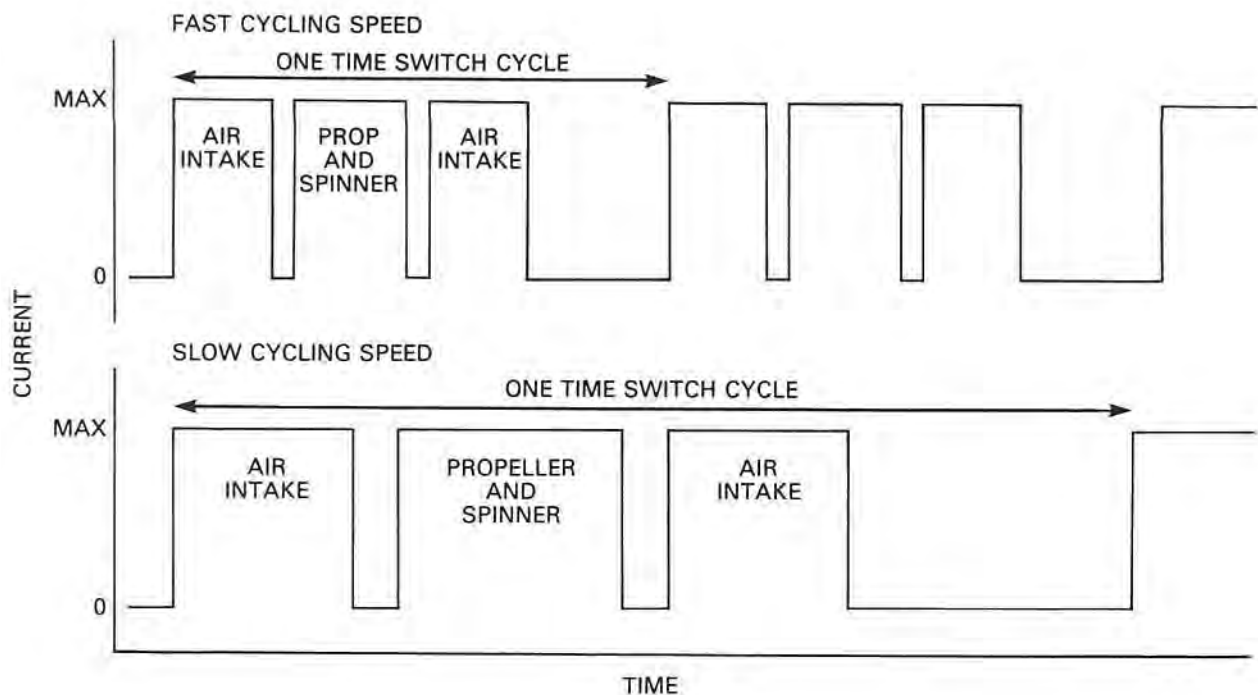
Fig. 13-5   Typical ice protection cyclic sequence.

forming, but the intermittently heated areas allow ice to form, during their 'heat-off period. During the 'heat-on' period, adhesion of the ice is broken and it is then removed by aerodynamic forces.

13.   The cycling time of the intermittently heated elements is arranged to ensure that the engine can accept the amount of ice that collects during the 'heat-off' period and yet ensure that the 'heat-on1 period is long enough to give adequate shedding, without causing any run-back icing to occur behind the heated areas.

14.   A two-speed cycling system is often used to accommodate the propeller and spinner requirements; a 'fast' cycle at the high air temperatures when the water concentration is usually greater and a 'slow' cycle in the lower temperature range. A typical cycling sequence chart is shown in fig, 13-5.

Rolls-Royce RB211-524D4D
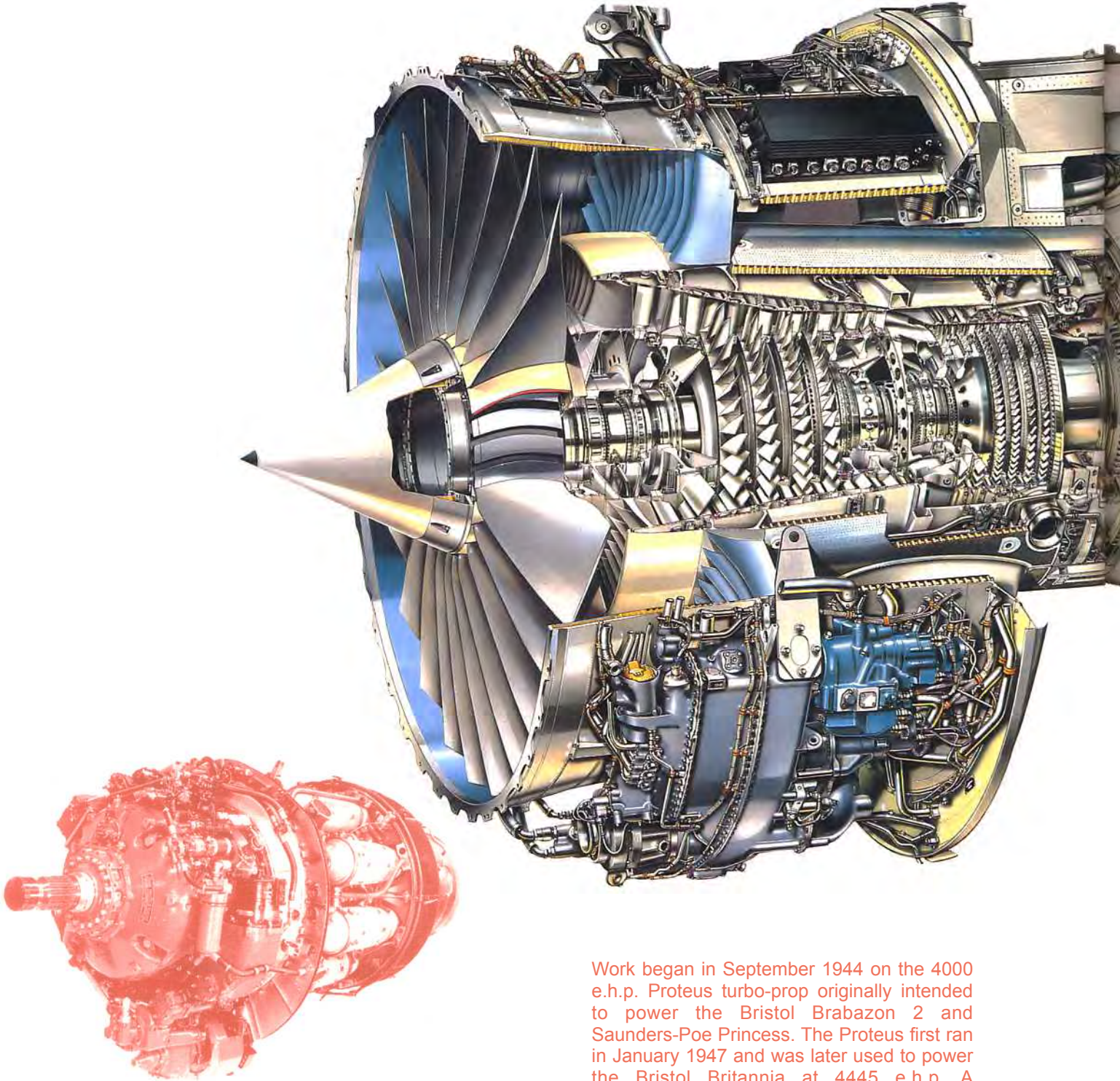
Bristol Proteus

Work began in September 1944 on the 4000 e.h.p. Proteus turbo-prop originally intended to power the Bristol Brabazon 2 and Saunders-Poe Princess. The Proteus first ran in January 1947 and was later used to power the Bristol Britannia at 4445 e.h.p. A development of this engine, the Marine Proteus, is used to power various patrol boats, hovercraft and hydrofoils.

14: Fire protection

Contents

INTRODUCTION

1. All gas turbine engines and their associated installation systems incorporate features that minimize the possibility of an engine fire. It is essential, however, that if a failure does take place and results in a fire, there is provision for the immediate detection and rapid extinction of the fire, and for the prevention of it spreading. The detection and extinguishing systems must add as little weight to the installation as possible.

PREVENTION OF ENGINE FIRE IGNITION

2. An engine/powerplant is designed to ensure that the prevention of engine fire ignition is achieved as far as possible. In most instances a dual failure is necessary before a fire can occur.

3. Most of the potential sources of flammable fluids are isolated from the 'hot end' of the engine. External fuel and oil system components and their associated pipes are usually located around the compressor casings, in a 'cool' zone, and are separated by a fireproof bulkhead from the combustion, turbine and jet pipe area, or 'hot' zone. The zones may be ventilated, as described in para 8, to prevent the accumulation of flammable vapours.

4. All pipes that carry fuel, oil or hydraulic fluid, are made fire resistant/proof to comply with fire regulations, and all electrical components and connections are made explosion-proof. Sparking caused by discharge of static electricity is prevented by bonding all aircraft and engine components. This gives electrical continuity between all the components and makes them incapable of igniting flammable vapour.

5. On some engines, tubes carrying flammable fluids in 'hot areas' of the engine are constructed with a double skin. Should a fracture of the main fluid carrying tube occur the outer skin will contain any leakage, so preventing any possible fire ignition.

6. The power plant cowlings are provided with an adequate drainage system to remove flammable fluids from the nacelle, bay, or pod, and all seal leakages from components are drained overboard at a position such that fluid cannot re-enter the pod and create a fire hazard.

7. Spontaneous ignition can be minimized on aircraft flying at high Mach numbers by ducting boundary layer bleed air around the engine.

However, if ignition should occur, this high velocity air stream may have to be shut off, otherwise it would increase the flame intensity and reduce the effectiveness of the extinguishing system by rapid dispersal of the extinguishant.

External cooling and ventilation

8. The engine bay or pod is usually cooled and ventilated by atmospheric air being passed around the engine and then vented overboard (fig. 14-1). Convection cooling during ground running may be provided by using an internal cooling outlet vent as an ejector system. An important function of the airflow is to purge any flammable vapours from the engine compartment. By keeping the airflow minimal, the power plant drag is minimized and, as the required quantity of fire extinguishant is in proportion to the zonal airflow, any fire outbreak would be of low intensity.

9. On some engines a fireproof bulkhead is also provided to separate the 'cool' area or zone of the engine, which contains the fuel, oil, hydraulic and electrical systems, from the 'hot' area surrounding the combustion, turbine and exhaust sections of the engine. Differential pressures can be created in the two zones by calibration of the inlet and outlet apertures to prevent the spread of fire from the hot zone.

10. Fig. 14-2 shows a more complex cooling and ventilation system used on a turbo-fan engine. Air is induced from the intake duct and also delivered from the fan to provide multi-zone cooling, each zone having its own calibrated cooling flow.

FIRE DETECTION

11. The rapid detection of a fire is essential to minimize the fire period before engine shut-down drill and release of extinguishant is effected. It is also extremely important that a fire detection system will not give a false fire warning resulting from short circuiting caused by chafing or the ingress of moisture in the case of electrically operated systems and chafes of the capillary resulting in loss of the contained gas in the case of the gas filled continuous element sensing type,

12. A detection system may consist of a number of strategically located detector units, or be of the continuous element (gas filled or electrical) sensing type that can be shaped and attached to pre-formed tubes. The sensing element can be routed across outlet orifices, such as a zone extractor ventilation duct, to give early detection of a fire (fig. 14-3).

13. In the case of electrical systems the presence of a fire is signalled by a change in the electrical characteristics of the detector circuit, according to the type of detector, be it thermistor, thermocouple or electrical continuous element. In these cases the change in temperature creates the signal which, through an amplifier, operates the warning indicator.

14. Both the thermocouple and thermistor detectors have properties making them ideally suited to this application. The thermocouple comprises two dissimilar metals which are joined together to form two junctions. As the temperature difference between the two junctions increases an E.M.F. is produced in the circuit and it is this E.M.F. that triggers the fire
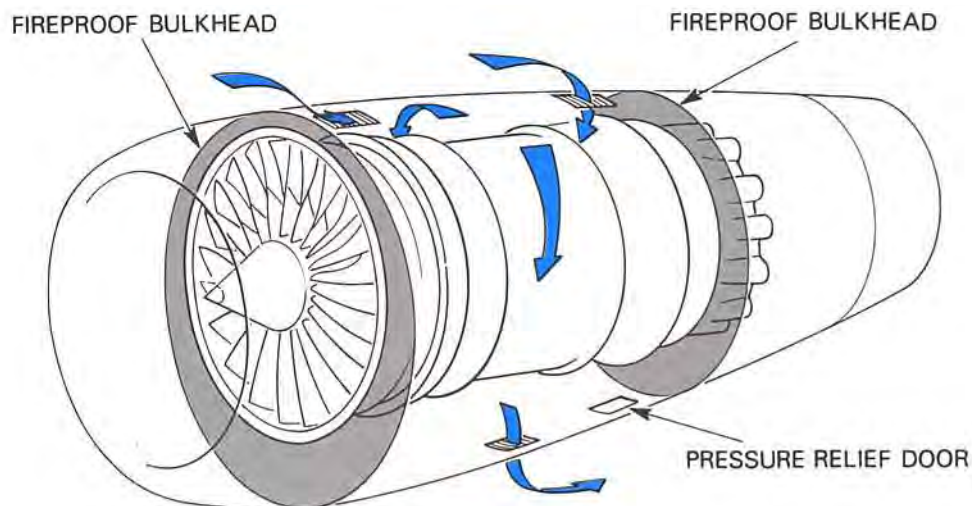
Fig. 14-1   A typical cooling and ventilation system.

ZONE 2
AIRFLOW THROUGH
UNSEALED
COWLING

ZONE 1
AIR INLET

PYLON
SPLITTER FAIRING

ZONE 3
AIR INLET SLOTS

ZONE 3
AIR INLET SLOTS

RADIAL
DRIVE FAIRING

TURBINE CASING
COOLING MANIFOLD

FRONT
FIREWALL

ZONE 1
AIR OUTLET

ZONE 3
AIR OUTLET

PRESSURE
RELIEF DOOR

REAR
FIREWALL

FIREPROOF
ZONE 1
ZONE 2
ZONE 3
VENTILATION/
COOLING AIR

ZONE 3

ZONE 2

ZONE 3

ZONE 1
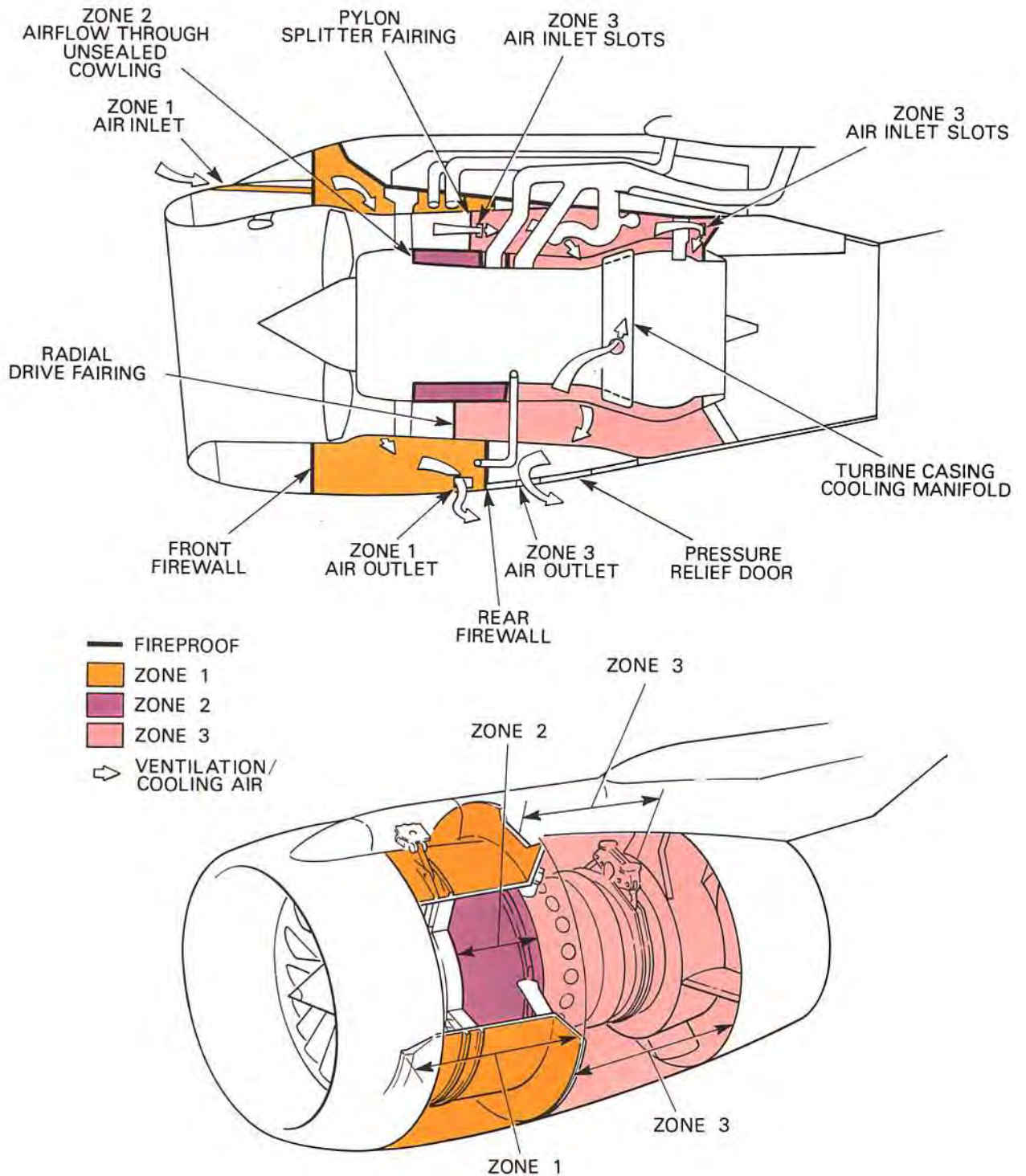
Fig. 14-2   Cooling and ventilation - turbo-fan engine.

warning displays. The thermistor consists of a semi-conductor material whose resistance changes as temperature increases, with a corresponding change in the current flowing in the circuit. It is this change in

the current that operates the warning indicators. A thermistor may be used as a single point detector or as a continuous element sensor.

15. Another form of continuous element sensor takes the form of a capacitor consisting of a tube containing a dielectric material with a conductor running through the centre. A voltage difference is applied between the tube and the centre conductor. As the temperature increases then the properties of the dielectric change with a corresponding change in the value of capacitance. This change of capacitance is displayed as a fire warning.

16. The gas filled detector consists of stainless steel tubing filled with gas absorbent material and in the event of a fire or overheat condition the temperature rise will cause the core of the sensing loop to expel the absorbed active gas into the sealed tube causing a rapid increase in pressure. This build up of pressure is sensed by the detector alarm switch. Should the sensing loop become damaged causing a loss of the pressurized gas, an integrity switch will indicate a detection loop fault on the appropriate engine. Fire indication is given by a warning light and bell.

17. At high Mach numbers, the considerably higher temperature levels may be such as to render the thermistor or thermocouple fire detection system unsatisfactory. Thermal detectors that sense either a temperature rise, or a rate of temperature rise, may therefore prove most suitable.

18. Alternatives to the above types are surveillance detectors that respond to light radiation from a fire. These may be made so sensitive that they respond only to the ultra-violet and infra-red rays emitted from a kerosine fire.

FIRE CONTAINMENT

19. An engine fire must be contained within the power plant and not be allowed to spread to other parts of the aircraft. The cowlings that surround the engine are usually made of aluminium alloys, which would be unable to contain a fire when the aircraft is static. During flight, however, the airflow around the cowlings provides sufficient cooling to render them fireproof. Fireproof bulkheads and any cowlings that are not affected by a cooling airflow, and sections of cowlings around certain outlets that may act as 'flame-holders', are usually manufactured from steel or titanium.
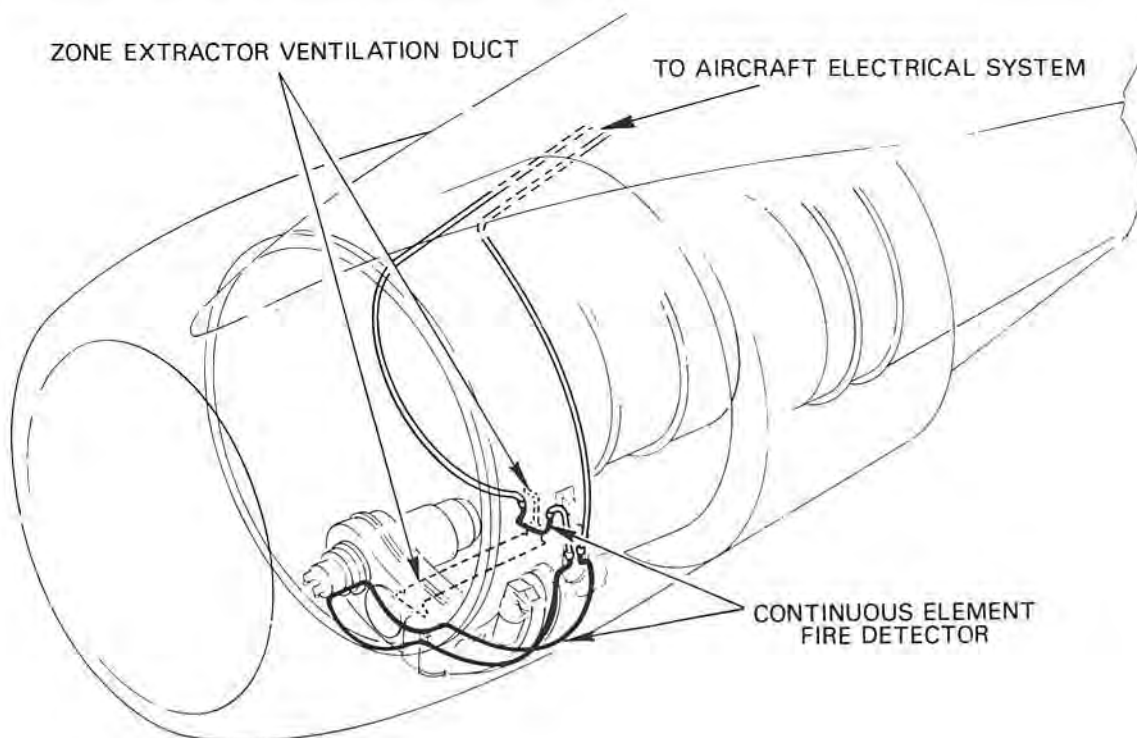
Fig. 14-3   A continuous element fire detecting system.
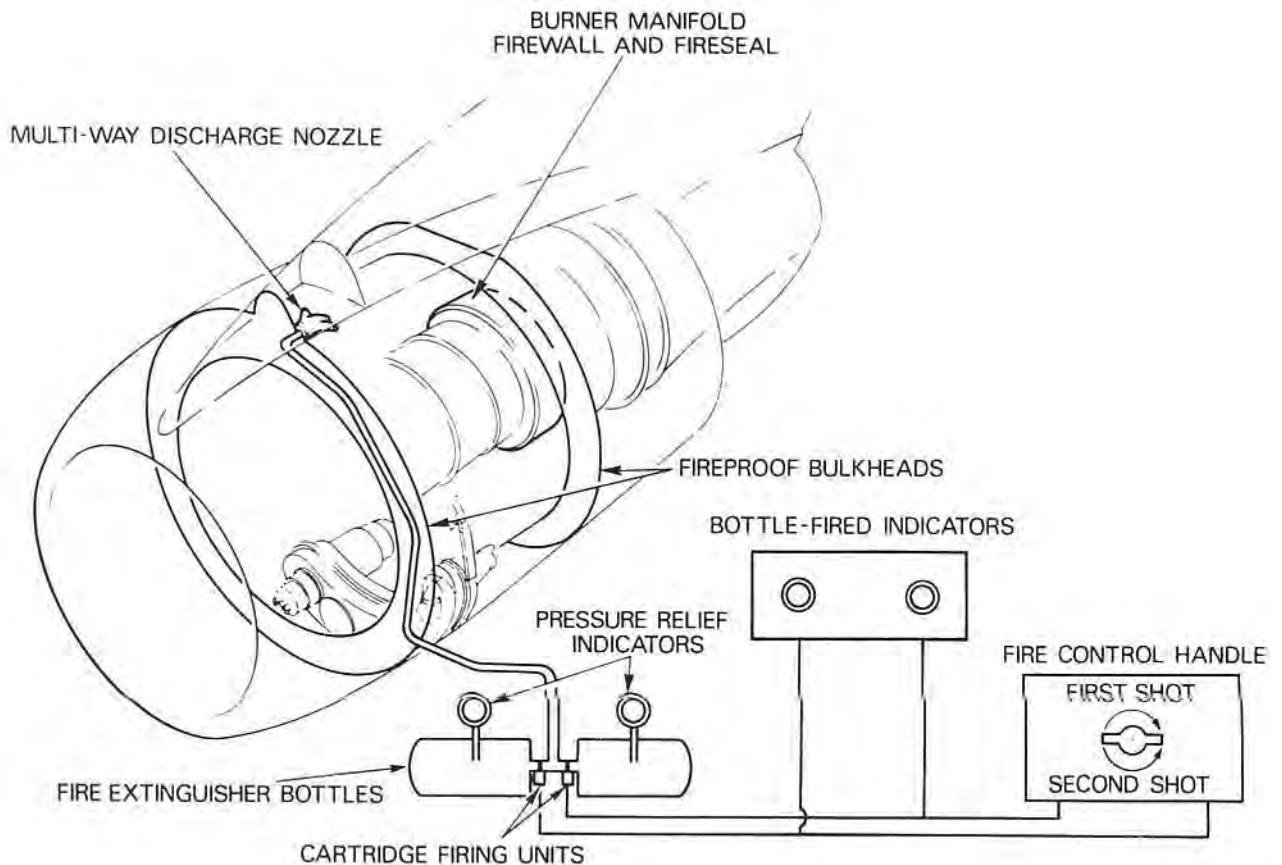
Fig. 14-4   A typical fire extinguishing system.

FIRE EXTINGUISHING

20. Before a fire extinguishing system is operated, the engine must be stopped to reduce the discharge of flammable fluids and air into the fire area. Any valves, such as the low pressure fuel cock, that control the flow of flammable fluid must be situated outside the 'hot' zone to prevent fire damage rendering them inoperative.

21. After a fire has been extinguished, no attempt must be made to start the engine again as this would probably re-establish the fluid leak and the ignition source that were the original causes of the fire. Furthermore, the extinguishing system may be exhausted.

22. The extinguishant that is used for engine fires is usually one of the Freon compounds. Pressurized containers are provided for the extinguishant and these are located outside the fire risk zone. When the relevant electrical circuit is manually operated, the extinguishant is discharged from the containers through a series of perforated spray pipes or nozzles into the fire (fig. 14-4). The discharge must be sufficient to give a predetermined concentration of extinguishant for a period that may vary between 0.5 seconds and 2 seconds. The system is generally one that enables two separate discharges to be made.

ENGINE OVERHEAT DETECTION

23. Turbine overheat does not constitute a serious fire risk. Detection of an overheat condition, however, is essential to enable the pilot to stop the engine before mechanical or material damage results.

24. A warning system of a similar type to the fire detection system, or thermocouples suitably positioned in the cooling airflow, may be used to detect excessive temperatures. Thermal switches positioned in the engine overboard air vents, such as the cooling air outlets, may also be included to give an additional warning.

Rolls - Royce Gem 2

Armstrong Siddeley Python
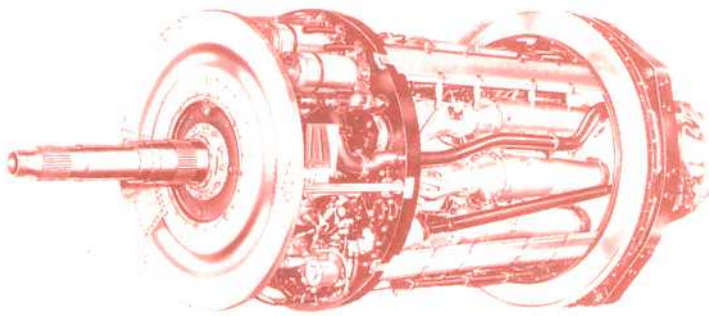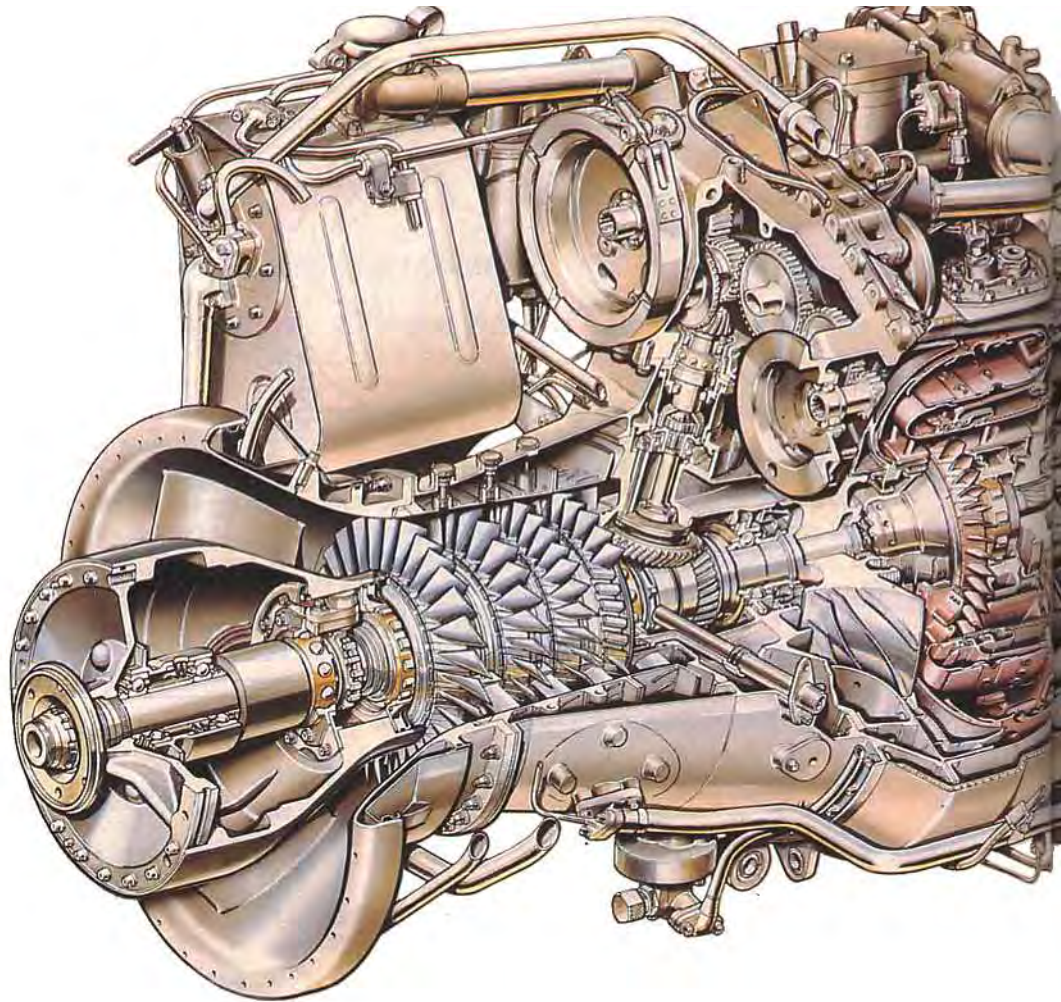
The Python was developed from the ASX axial-flow turbo-jet which first ran in April 1943 and was producing 2800 lb thrust by 1944. With the addition of a propeller gearbox the engine produced 3600 shp plus 1100 lb thrust and was known as the ASP. Renamed the Python it entered service as the power plant for the Westland Wyvern S4 turbo-prop fighter.

15: Thrust reversal

Contents Page

INTRODUCTION

1. Modern aircraft brakes are very efficient but on wet, icy or snow covered runways this efficiency may be reduced by the loss of adhesion between the aircraft tyre and the runway thus creating a need for an additional method of bringing the aircraft to rest within the required distance.

2. A simple and effective way to reduce the aircraft landing run on both dry and slippery runways is to reverse the direction of the exhaust gas stream, thus using engine power as a deceleration force. Thrust reversal has been used to reduce airspeed in flight but it is not commonly used on modern aircraft. The difference in landing distances between an aircraft without reverse thrust and one using reverse thrust is illustrated in fig. 15-1.

3. On high by-pass ratio (fan) engines, reverse thrust action is achieved by reversing the fan (cold stream) airflow. It is not necessary to reverse the exhaust gas flow (hot stream) as the majority of the engine thrust is derived from the fan.

OPERATING CONDITIONS
I.S.A.
SEA LEVEL WET/ICY RUNWAY
LANDING WEIGHT — 60000 LB

TOUCH DOWN AT
TRUE GROUND SPEED
103 KNOTS

0
500
1000
1500

DISTANCE
IN FEET
2000

THRUST REVERSER
OPERATION

BRAKES ONLY

2500

3000
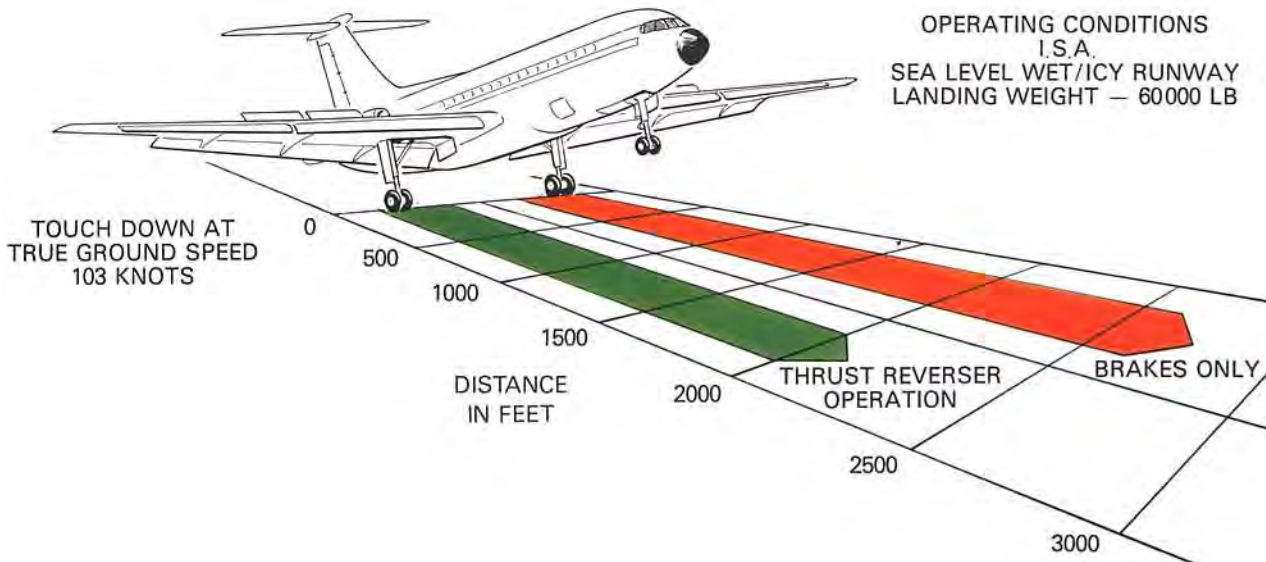
Fig. 15-1   Comparative landing runs with and without thrust reversal.

4. On propeller-powered aircraft, reverse thrust action is obtained by changing the pitch of the propeller blades. This is usually achieved by a hydro-mechanical system, which changes the blade angle to give the braking action under the response of the power or throttle lever in the aircraft.

5. Ideally, the gas should be directed in a completely forward direction. It is not possible, however, to achieve this, mainly for aerodynamic reasons, and a discharge angle of approximately 45 degrees is chosen. Therefore, the effective power in reverse thrust is proportionately less than the power in forward thrust for the same throttle angle.

PRINCIPLES OF OPERATION

6. There are several methods of obtaining reverse thrust on turbo-jet engines; three of these are shown in fig. 15-2 and explained in the following paragraphs.

7. One method uses clamshell-type deflector doors to reverse the exhaust gas stream and a second uses a target system with external type doors to do the same thing. The third method used on fan engines utilizes blocker doors to reverse the cold stream airflow.

8. Methods of reverse thrust selection and the safety features incorporated in each system described are basically the same. A reverse thrust lever in the crew compartment is used to select reverse thrust; the lever cannot be moved to the reverse thrust position unless the engine is running at a low power setting, and the engine cannot be opened up to a high power setting if the reverser fails to move into the full reverse thrust position. Should the operating pressure fall or fail, a mechanical lock holds the reverser in the forward thrust position; this lock cannot be removed until the pressure is restored. Operation of the thrust reverser system is indicated in the crew compartment by a series of lights.

Clamshell door system
9. The clamshell door system is a pneumatically operated system, as shown in detail in fig. 15-3. Normal engine operation is not affected by the system, because the ducts through which the exhaust gases are deflected remain closed by the doors until reverse thrust is selected by the pilot.

10. On the selection of reverse thrust, the doors rotate to uncover the ducts and close the normal gas stream exit. Cascade vanes then direct the gas stream in a forward direction so that the jet thrust opposes the aircraft motion.

CLAMSHELL DOORS IN FORWARD
THRUST POSITION

CLAMSHELL DOORS IN REVERSE
THRUST POSITION

ACTUATOR EXTENDED AND BUCKET DOORS IN
FORWARD THRUST POSITION

ACTUATOR AND BUCKET DOORS IN REVERSE
THRUST POSITION

COLD STREAM REVERSER IN
FORWARD THRUST POSITION

COLD STREAM REVERSER IN
REVERSE THRUST POSITION
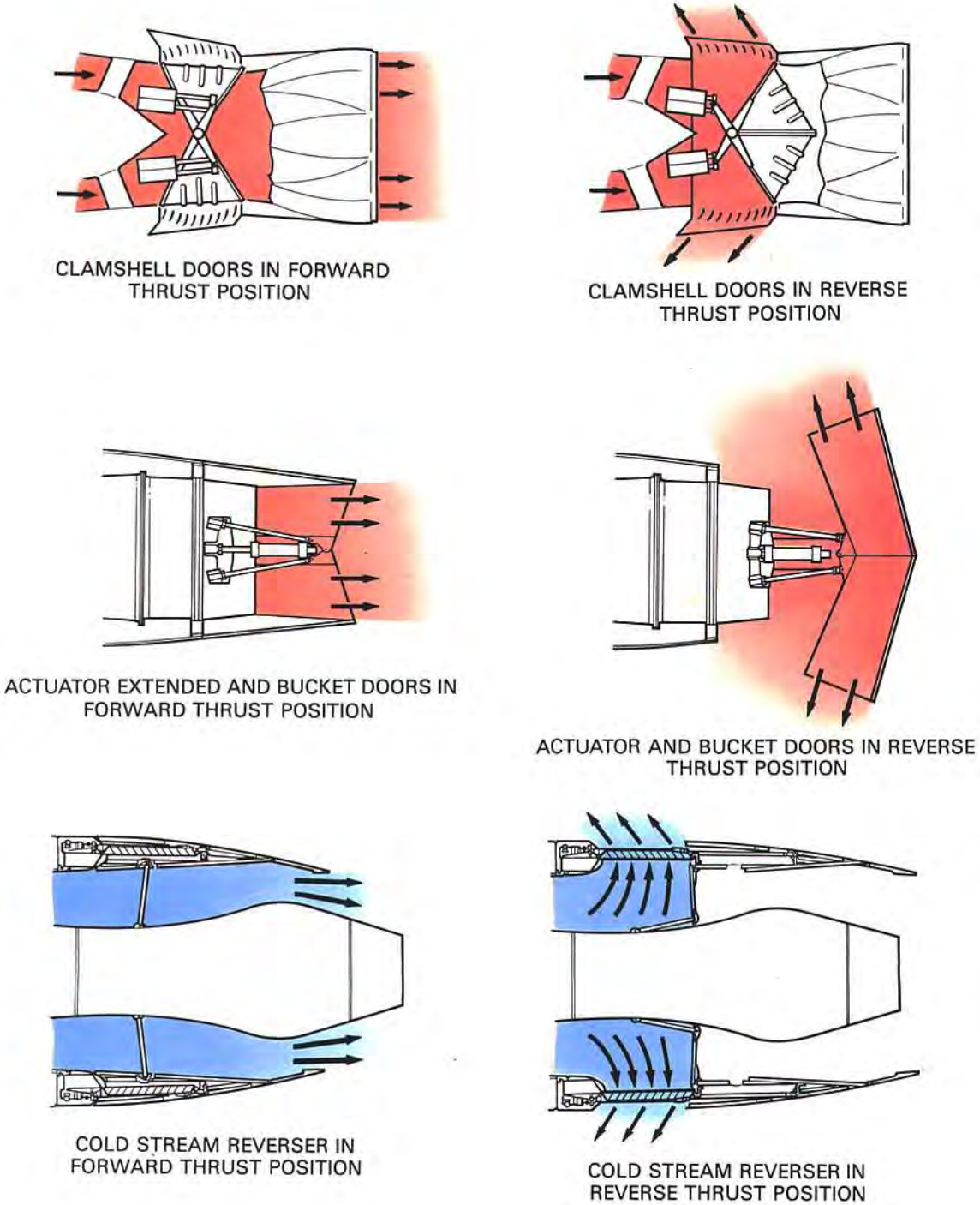
Fig. 15-2   Methods of thrust reversal.

REVERSE THRUST
SELECT LEVER

PNEUMATIC RAM

CASCADE VANES

Low

Low

THROTTLE
LEVER

ENGINE
THROTTLE

CLAMSHELL
DOORS

LOCK MECHANISM
(Engaged)

TOUCH DOWN

Both levers
at low power setting

CONTROL VALVE

REVERSE FLOW
OUTLET DUCTS

Operating air pressur

Vent

Gas stream

High

LOCK MECHANISM
(Released)

FULL BRAKING

Reverse thrust select lever
at high power setting
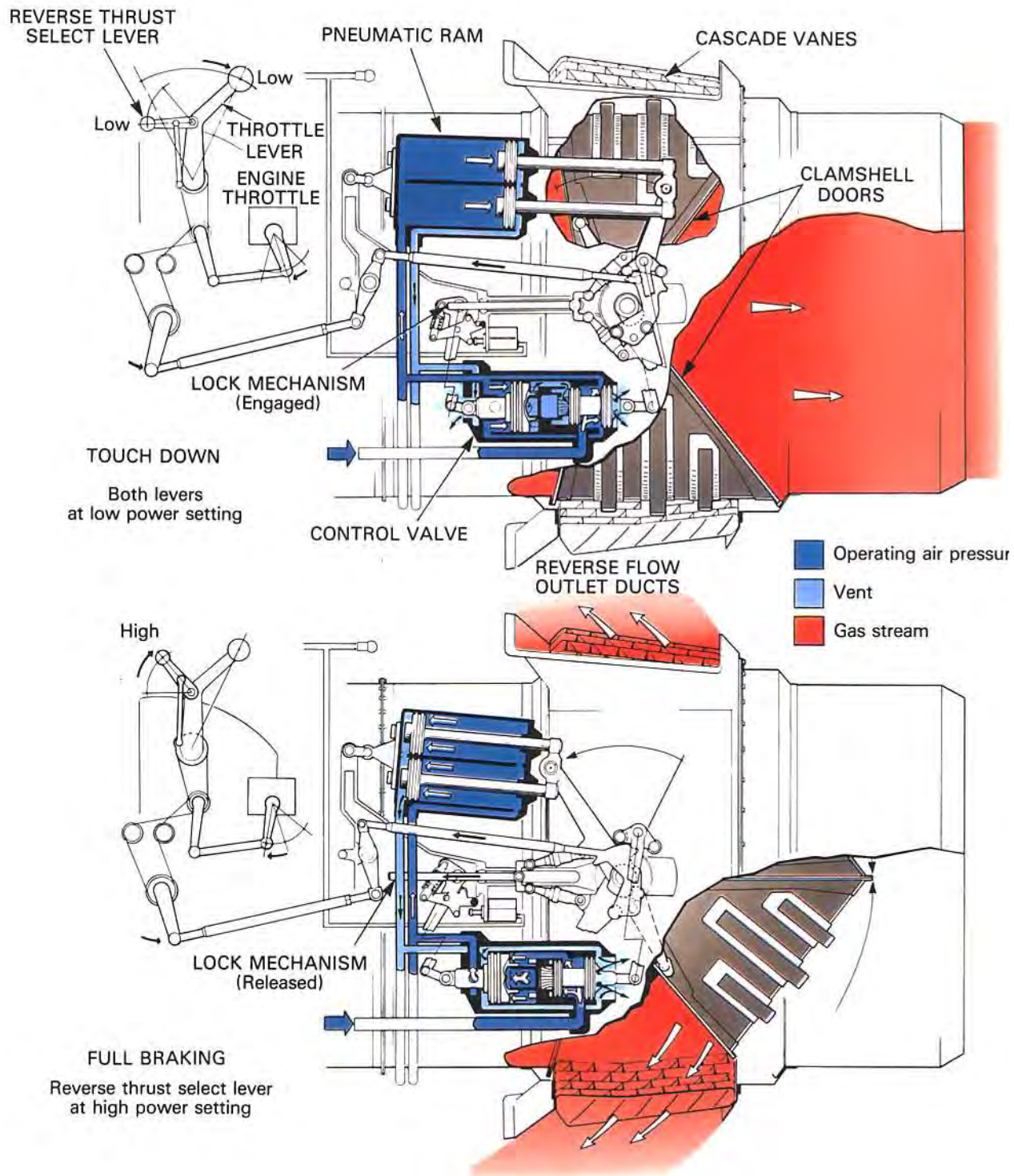
Fig. 15-3   A typical thrust reverser system using clamshell doors.

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REVERSE THRUST SELECT LEVER
(forward thrust)

GEARBOX

POWER LEVER

TRANSLATING COWL

CASCADE VANES

BLOCKER DOORS

H.P. operating air

Cold stream air flow

LOCK INDICATOR LIGHT SWITCH

DIRECTION AND SPEED CONTROL VALVE

Vent

FLEXIBLE DRIVES

GEARBOX

AIR MOTOR

SCREWJACK

FORWARD THRUST POSITION

REVERSE THRUST SELECT LEVER (reverse thrust)

LOCK AND SEQUENCE VALVE

SELECTOR VALVE

EXHAUST

FUEL REGULATOR

AIR MOTOR UNIT

FEEDBACK GEARBOX

PRESSURE REGULATOR AND S.O.V.

BLOCKER DOORS (folded)

REVERSE THRUST POSITION

Fig. 15-4   A typical fan cold stream thrust reversal system.

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11. The clamshell doors are operated by pneumatic rams through levers that give the maximum load to the doors in the forward thrust position; this ensures effective sealing at the door edges, so preventing gas leakage. The door bearings and operating linkage operate without lubrication at temperatures of up to 600 deg. C.

Bucket target system
12. The bucket target system is hydraulically actuated and uses bucket-type doors to reverse the hot gas stream. The thrust reverser doors are actuated by means of a conventional pushrod system. A single hydraulic powered actuator is connected to a drive idler, actuating the doors through a pair of pushrods (one for each door).

13. The reverser doors are kept in synchronization through the drive idler. The hydraulic actuator incorporates a mechanical lock in the stowed (actuator extended) position.
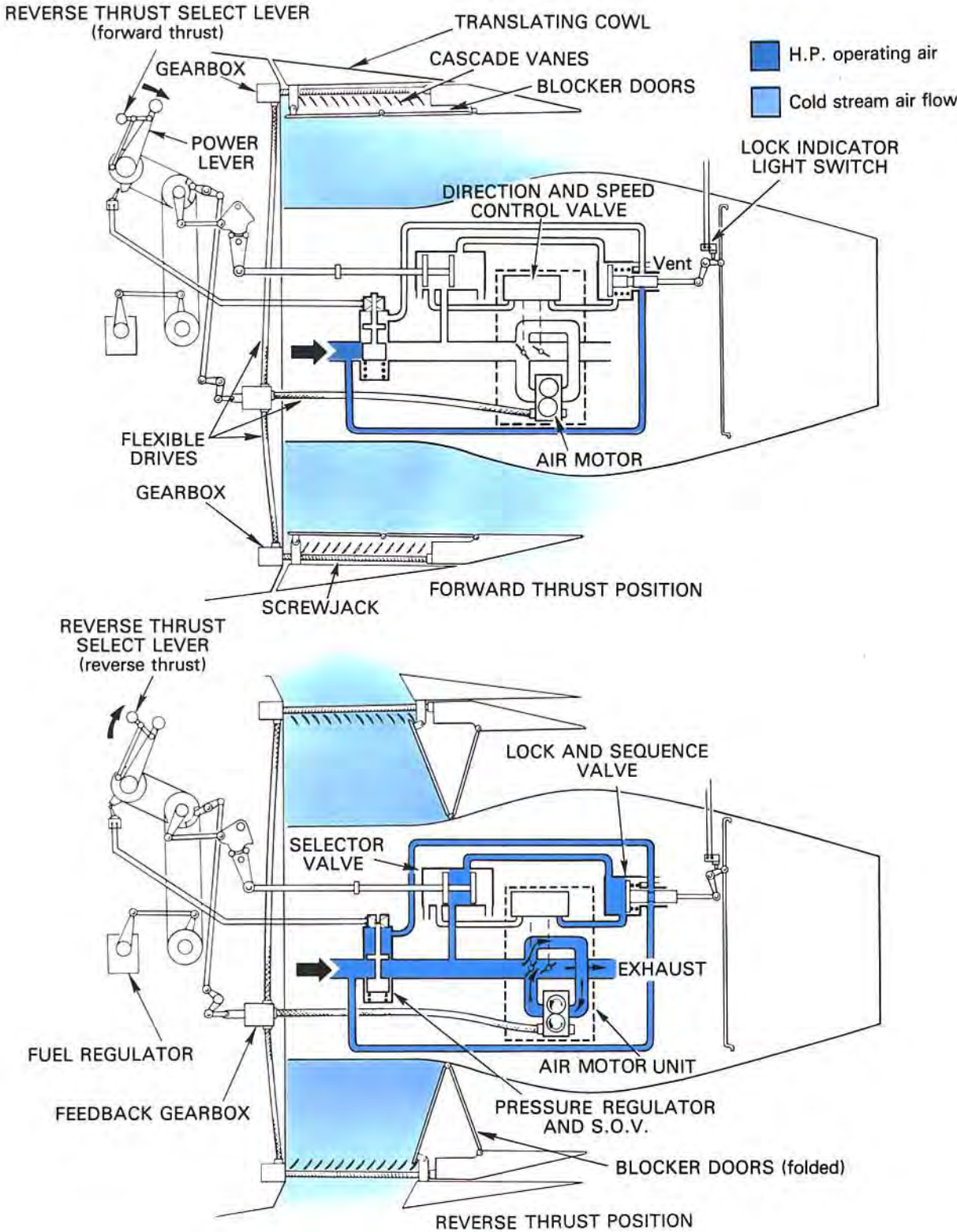
14. In the forward thrust mode (stowed) the thrust reverser doors form the convergent-divergent final nozzle for the engine.

Cold stream reverser system
15. The cold stream reverser system (fig. 15-4) can be actuated by an air motor, the output of which is converted to mechanical movement by a series of flexible drives, gearboxes and screwjacks, or by a system incorporating hydraulic rams.

16. When the engine is operating in forward thrust, the cold stream final nozzle is 'open' because the cascade vanes are internally covered by the blocker doors (flaps) and externally by the movable (translating) cowl; the latter item also serves to reduce drag.

17. On selection of reverse thrust, the actuation system moves the translating cowl rearwards and at the same time folds the blocker doors to blank off the cold stream final nozzle, thus diverting the airflow through the cascade vanes.

Turbo-propeller reverse pitch system
18. As mentioned in para. A, reverse thrust action is affected on turbo-propeller powered aircraft by changing the pitch of the propeller blades through a hydro-mechanical pitch control system (fig. 15-5). Movement of the throttle or power control lever
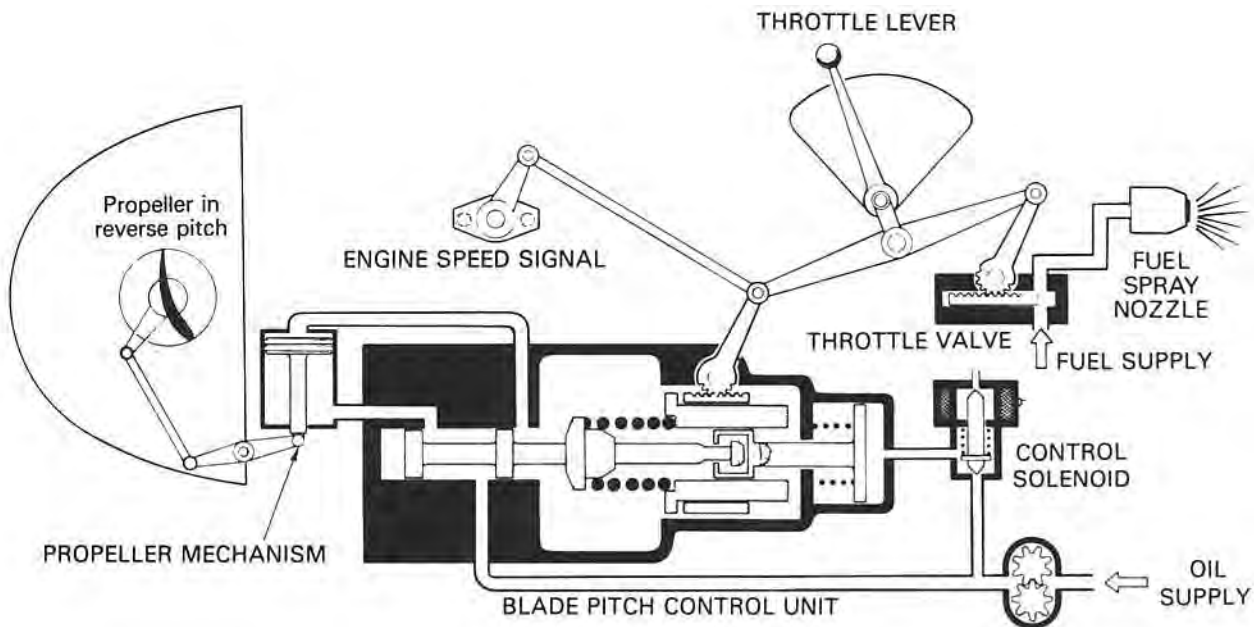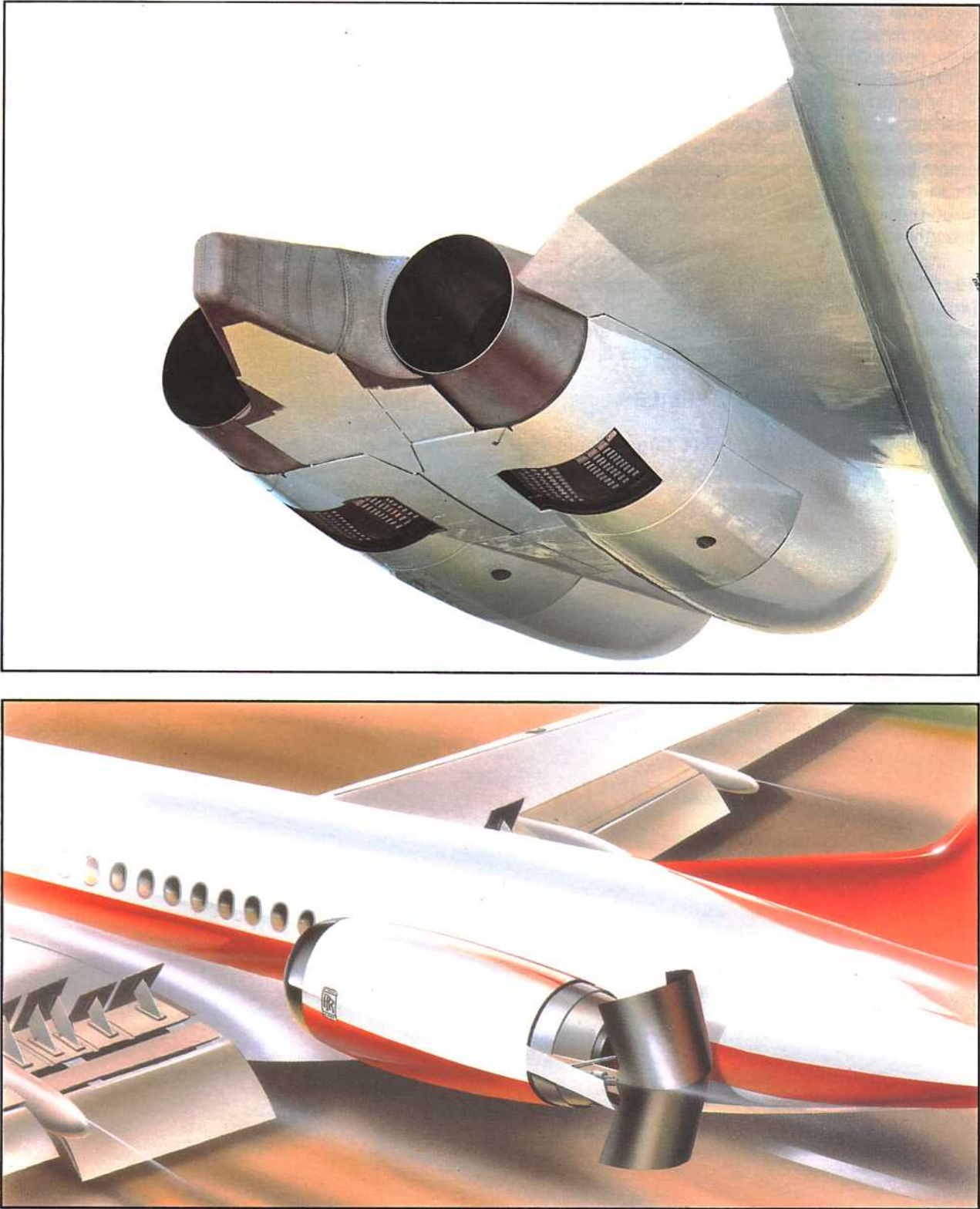
Fig. 15-5   A propeller pitch control system.

Fig. 15-6   Hot stream thrust reverser installations.

directs oil from the control system to the propeller mechanism to reduce the blade angle to zero, and then through to negative (reverse) pitch. During throttle lever movement, the fuel to the engine is trimmed by the throttle valve, which is interconnected to the pitch control unit, so that engine power and blade angle are co-ordinated to obtain the desired amount of reverse thrust. Reverse thrust action may also be used to manoeuvre a turbo-propeller aircraft backwards after it has been brought to rest.

19. Several safety factors are incorporated in the propeller control system for use in the event of propeller malfunction, and these devices are usually hydro-mechanical pitch locking devices or stops.

CONSTRUCTION AND MATERIALS

20. The clamshell and bucket target doors (fig. 15-6) described in paras. 9 and 12 form part of the jet pipe. The reverser casing is connected to the aircraft structure or directly to the engine. The casing supports the two reverser doors, the operating mechanism and, in the case of the clamshell door system, the outlet ducts that contain the cascade vanes. The angle and area of the gas stream are controlled by the number of vanes in each outlet duct.

21. The clamshell and bucket target doors lie flush with the casing during forward thrust operation and are hinged along the centre line of the jet pipe. They are, therefore, in line with the main gas load and this ensures that the minimum force is required to move the doors.

22. Both the clamshell door system and the bucket target system are subjected to high temperatures and to high gas loads. The components of both systems, especially the doors, are therefore
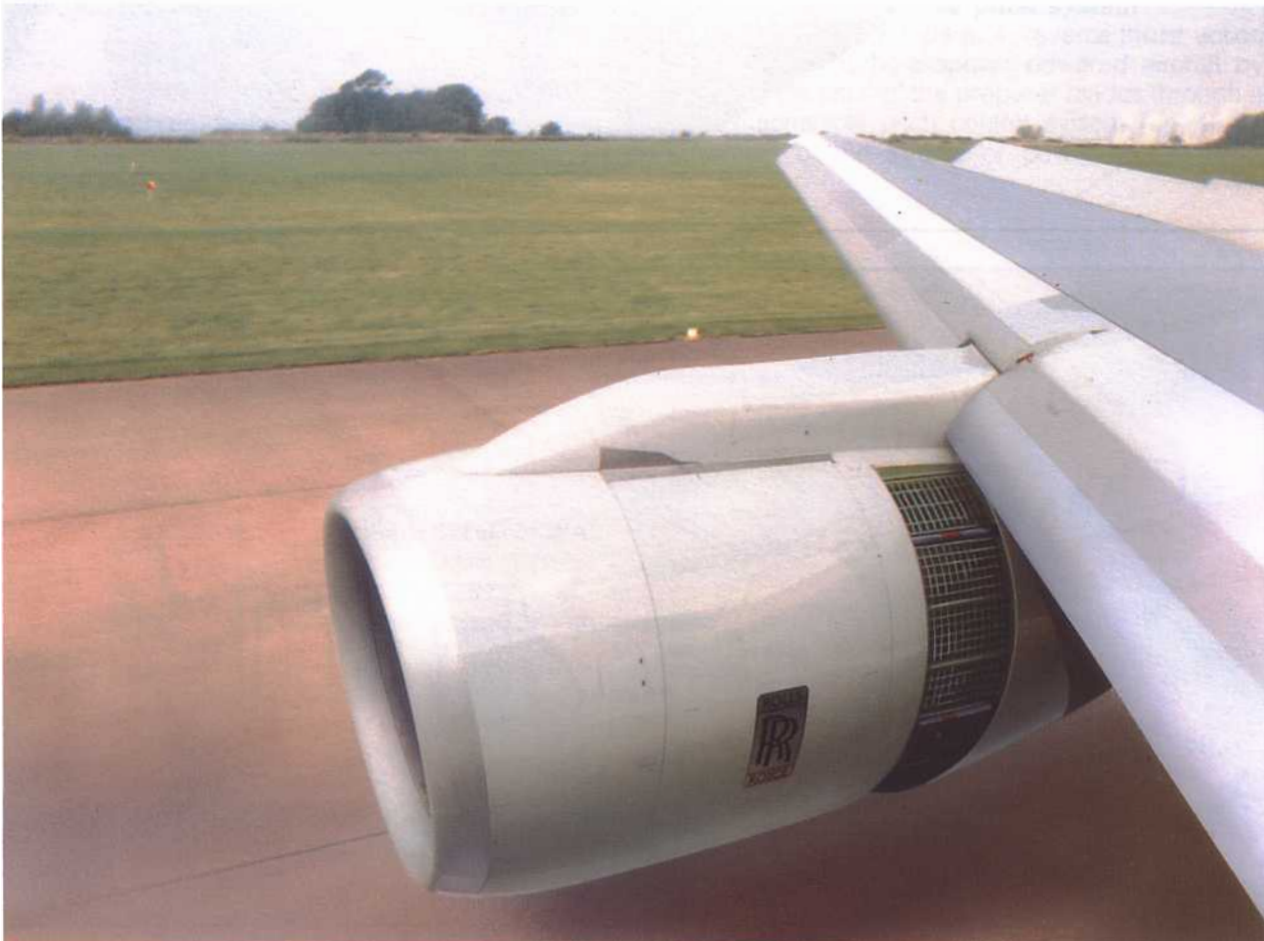
Fig. 15-7   A cold stream thrust reverser installation.

constructed from heat-resisting materials and are of particularly robust construction.

23.   The cold stream thrust reverser casing (fig. 15-7) is fitted between the low pressure compressor casing and the cold stream final nozzle. Cascade vane assemblies are arranged in segments around the circumference of the thrust reverser casing. Blocker doors are internally mounted and are connected by linkages to the external movable (translating) cowl, which is mounted on rollers and tracks. Because the thrust reverser is not subjected to high temperatures, the casing, blocker doors and cowl are constructed mainly of aluminium alloys or composite materials. The cowl is double-skinned, with the space between the skins containing noise absorbent material (Part 19).

Turbo-Union RB199
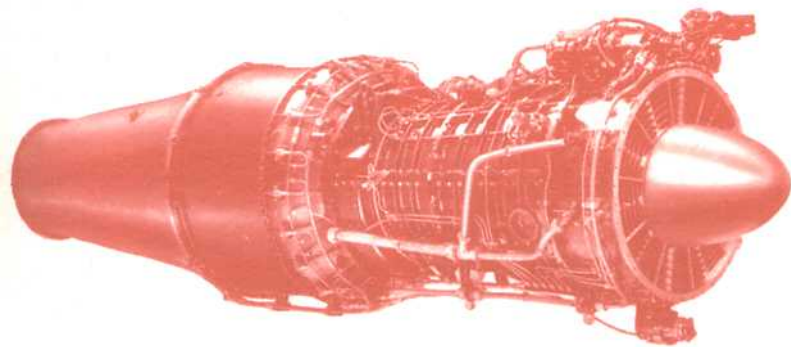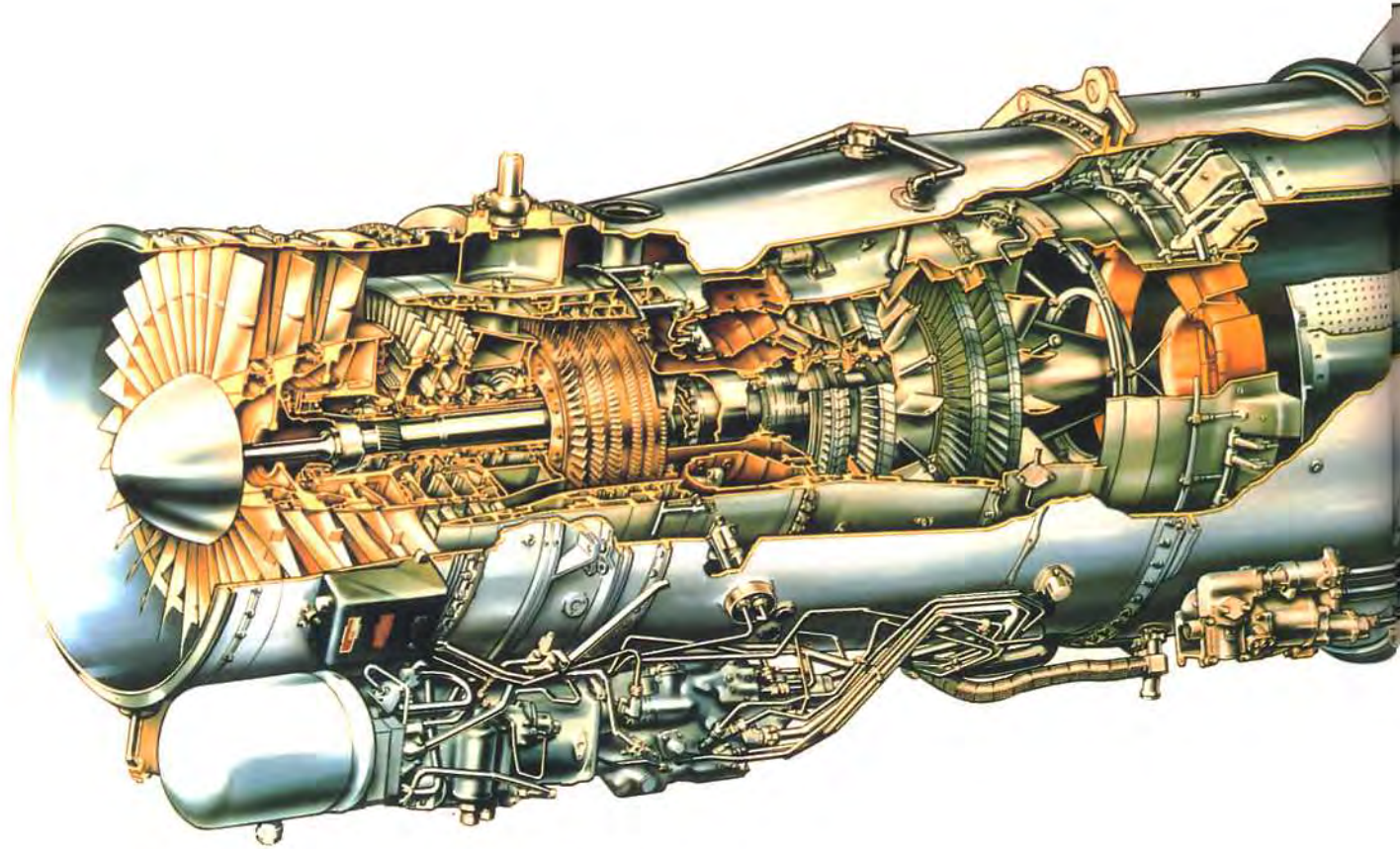
Metrovick F2/4 Beryl

Development of the F2, the first British axial flow turbo-jet, began in f 940. After initial flight trials in the tail of an Avro Lancaster, two F2s were installed in a Gloster Meteor and first flew on 13 November 1943. After early problems the F2/4 Beryl was developed which gave up to 4000 lb thrust and was used to power the Saunders Roe SR/A1 flying boat fighter.

16: Afterburning

Contents
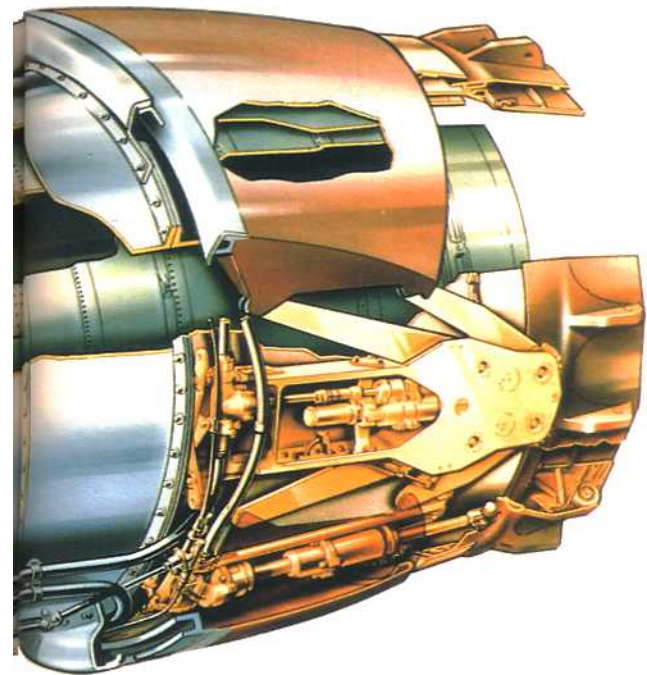
INTRODUCTION

1. Afterburning (or reheat) is a method of augmenting the basic thrust of an engine to improve the aircraft take-off, climb and (for military aircraft) combat performance. The increased power could be obtained by the use of a larger engine, but as this would increase the weight, frontal area and overall fuel consumption, afterburning provides the best method of thrust augmentation for short periods.

2. Afterburning consists of the introduction and burning of fuel between the engine turbine and the jet pipe propelling nozzle, utilizing the unburned oxygen in the exhaust gas to support combustion (fig. 16-1). The resultant increase in the temperature of the exhaust gas gives an increased velocity of the jet leaving the propelling nozzle and therefore increases the engine thrust.

3. As the temperature of the afterburner flame can be in excess of 1,700 deg. C., the burners are usually arranged so that the flame is concentrated around the axis of the jet pipe. This allows a proportion of the turbine discharge gas to flow along the wall of the jet pipe and thus maintain the wall temperature at a safe value.
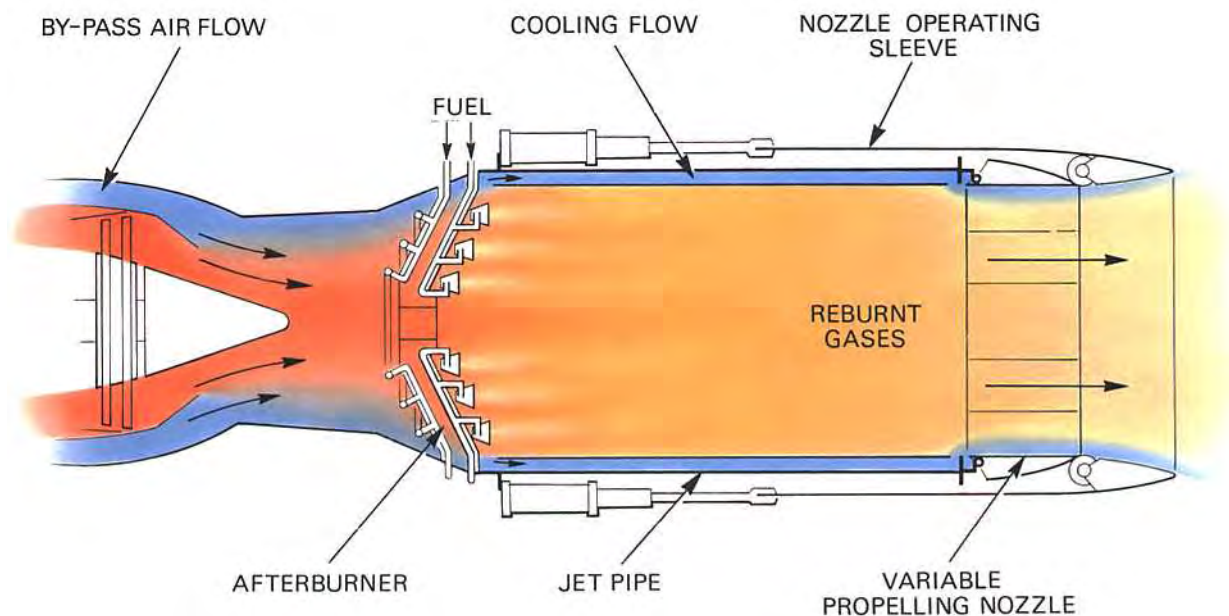
Fig. 16-1   Principle of afterburning

4.   The area of the afterburning jet pipe is larger than a normal jet pipe would be for the same engine, to obtain a reduced velocity gas stream. To provide for operation under all conditions, an afterburning jet pipe is fitted with either a two-position or a variable-area propelling nozzle (fig. 16-2). The nozzle is closed during non-afterburning operation, but when afterburning is selected the gas temperature increases and the nozzle opens to give an exit area suitable for the resultant increase in the volume of the gas stream. This prevents any increase in pressure occurring in the jet pipe which would affect the functioning of the engine and enables afterburning to be used over a wide range of engine speeds.

5.   The thrust of an afterburning engine, without afterburning in operation, is slightly less than that of a similar engine not fitted with afterburning equipment; this is due to the added restrictions in the jet pipe. The overall weight of the power plant is also increased because of the heavier jet pipe and afterburning equipment.

6.   Afterburning is achieved on low by-pass engines by mixing the by-pass and turbine streams before the afterburner fuel injection and stabilizer system is reached so that the combustion takes place in the mixed exhaust stream. An alternative method is to inject the fuel and stabilize the flame in the individual by-pass and turbine streams, burning the available gases up to a common exit temperature at the final nozzle. In this method, the fuel injection is scheduled separately to the individual streams and it is normal to provide some form of interconnection between the flame stabilizers in the hot and cold streams to assist the combustion processes in the cold by-pass air.

OPERATION OF AFTERBURNING

7.   The gas stream from the engine turbine enters the jet pipe at a velocity of 750 to 1,200 feet per second, but as this velocity is far too high for a stable flame to be maintained, the flow is diffused before it enters the afterburner combustion zone, i.e. the flow velocity is reduced and the pressure is increased. However, as the speed of burning kerosine at normal mixture ratios is only a few feet per second, any fuel lit even in the diffused air stream would be blown away. A form of flame stabilizer (vapour gutter) is, therefore, located downstream of the fuel burners to provide a region in which turbulent eddies are formed to assist combustion and where the local gas velocity is further reduced to a figure at which flame stabilization occurs whilst combustion is in operation.

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AFTERBURNER JET PIPE

MOVABLE EYELIDS

EYELID OPERATING RAMS

TWO-POSITION NOZZLE

NOZZLE OPERATING RAMS

AFTERBURNER JET PIPE

ACTUATING SLEEVE

VARIABLE-AREA NOZZLE

INTERLOCKING FLAPS
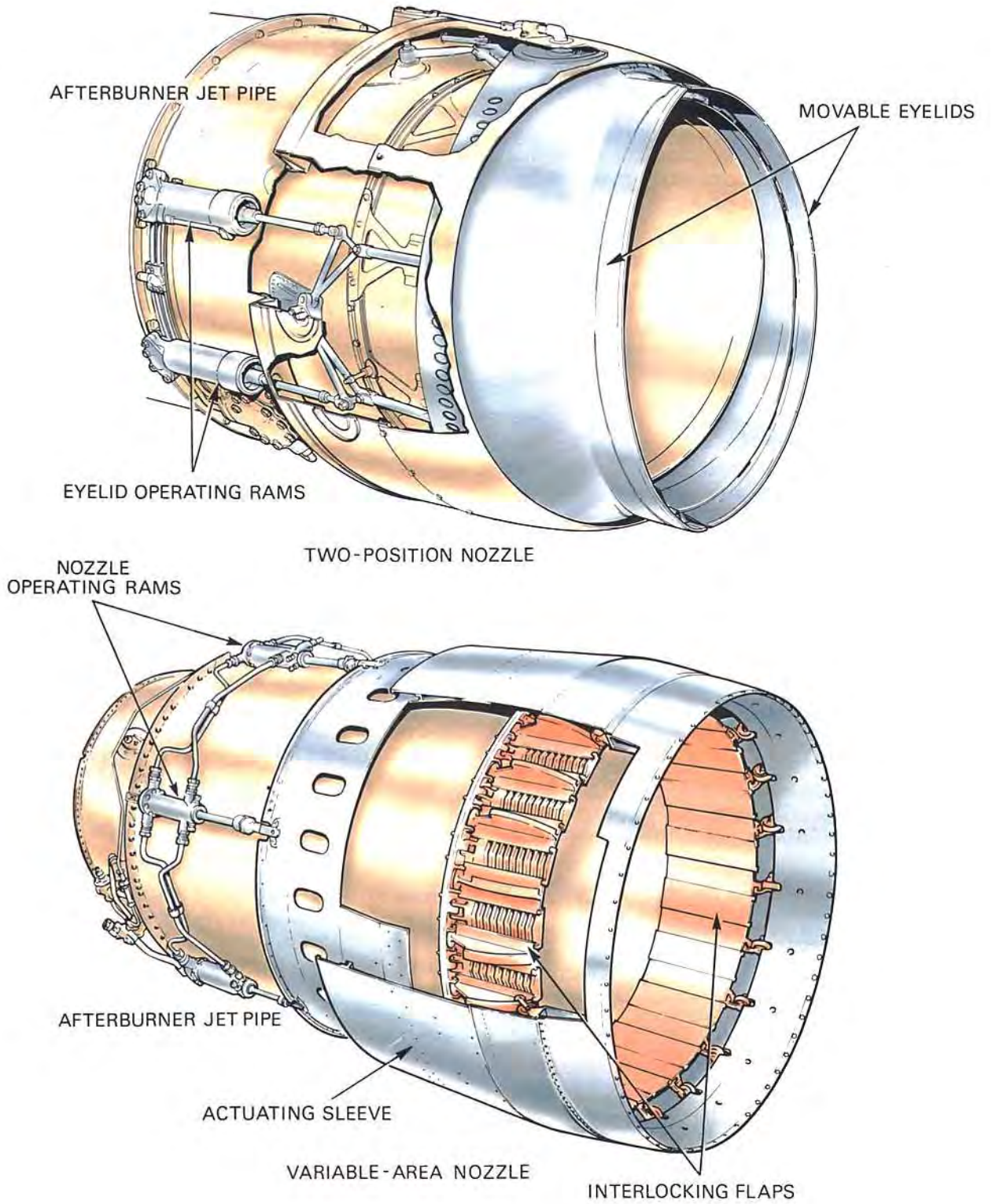
Fig. 16-2   Examples of afterburning jet pipes and propelling nozzles.

8. An atomized fuel spray is fed into the jet pipe through a number of burners, which are so arranged as to distribute the fuel evenly over the flame area. Combustion is then initiated by a catalytic igniter, which creates a flame as a result of the chemical reaction of the fuel/air mixture being sprayed on to a platinum-based element, by an igniter plug adjacent to the burner, or by a hot streak of flame that originates in the engine combustion chamber (fig. 16-3): this latter method is known as 'hot-shot' ignition. Once combustion is initiated, the gas temperature increases and the expanding gases accelerate through the enlarged area propelling nozzle to provide the additional thrust.

9. In view of the high temperature of the gases entering the jet pipe from the turbine, it might be assumed that the mixture would ignite spontaneously. This is not so, for although cool flames form at
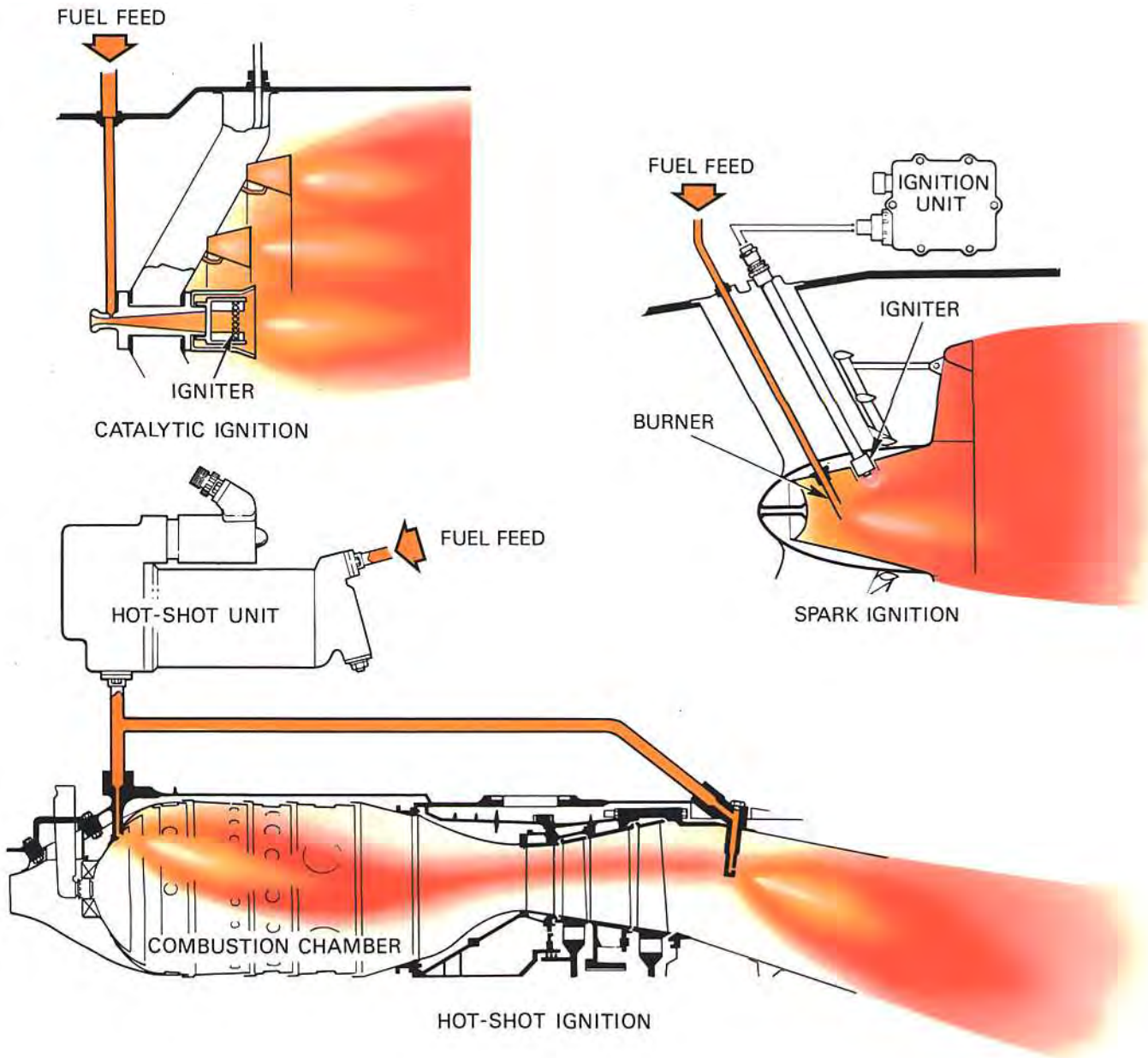

Fig. 16-3   Methods of afterburning ignition.

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temperatures up to 700 deg. C., combustion will not take place below 800 deg. C. If however, the conditions were such that spontaneous ignition could be effected at sea level, it is unlikely that it could be effected at altitude where the atmospheric pressure is low. The spark or flame that initiates combustion must be of such intensity that a light-up can be obtained at considerable altitudes.

10.   For smooth functioning of the system, a stable flame that will burn steadily over a wide range of mixture strengths and gas flows is required. The mixture must also be easy to ignite under all conditions of flight and combustion must be maintained with the minimum loss of pressure.

CONSTRUCTION

Burners

11.   The burner system consists of several circular concentric fuel manifolds supported by struts inside the jet pipe. Fuel is supplied to the manifolds by feed pipes in the support struts and sprayed into the flame area, between the flame stabilizers, from holes in the downstream edge of the manifolds. The flame stabilizers are blunt nosed V-section annular rings located downstream of the fuel burners. An alternative system includes an additional segmented fuel manifold mounted within the flame stabilizers. The typical burner and flame stabilizer shown in fig. 16-4 is based on the latter system.

Jet pipe

12.   The afterburning jet pipe is made from a heat-resistant nickel alloy and requires more insulation than the normal jet pipe to prevent the heat of combustion being transferred to the aircraft structure. The jet pipe may be of a double skin construction with the outer skin carrying the flight loads and the inner skin the thermal stresses; a flow of cooling air is often induced between the inner and outer skins. Provision is also made to accommodate expansion and contraction, and to prevent gas leaks at the jet pipe joints.

13.   A circular heatshield of similar material to the jet pipe is often fitted to the inner wall of the jet pipe to improve cooling at the rear of the burner section. The heatshield comprises a number of bands, linked by cooling corrugations, to form a single skin. The rear of the heatshield is a series of overlapping 'tiles' riveted to the surrounding skin (fig. 16-4). The shield also prevents combustion instability from creating excessive noise and vibration, which in turn would cause rapid physical deterioration of the afterburner equipment.

Propelling nozzle

14.   The propelling nozzle is of similar material and construction as the jet pipe, to which it is secured as a separate assembly. A two-position propelling nozzle has two movable eyelids that are operated by actuators, or pneumatic rams, to give an open or closed position (para. 4.). A variable-area propelling nozzle has a ring of interlocking flaps that are hinged to the outer casing and may be enclosed by an outer shroud. The flaps are actuated by powered rams to the closed position, and by gas loads to the interme-diate or the open positions; control of the flap position is by a control unit and a pump provides the power to the rams (para. 18).

CONTROL SYSTEM

15.   It is apparent that two functions, fuel flow and propelling nozzle area, must be co-ordinated for sat-isfactory operation of the afterburner system, These functions are related by making the nozzle area dependent upon the fuel flow at the burners or vice-versa. The pilot controls the afterburner fuel flow or the nozzle area in conjunction with a compressor delivery/jet pipe pressure sensing device (a pressure ratio control unit). When the afterburner fuel flow is increased, the nozzle area increases; when the afterburner fuel flow decreases, the nozzle area is reduced. The pressure ratio control unit ensures the pressure ratio across the turbine remains unchanged and that the engine is unaffected by the operation of afterburning, regardless of the nozzle area and fuel flow.

16.   Since large fuel flows are required for afterburn-ing, an additional fuel pump is used. This pump is usually of the centrifugal flow or gear type and is energized automatically when afterburning is selected. The system is fully automatic and incorpo-rates 'fail safe' features in the event of an afterburner malfunction. The interconnection between the control system and afterburner jet pipe is shown diagram-matically in fig. 16-5.
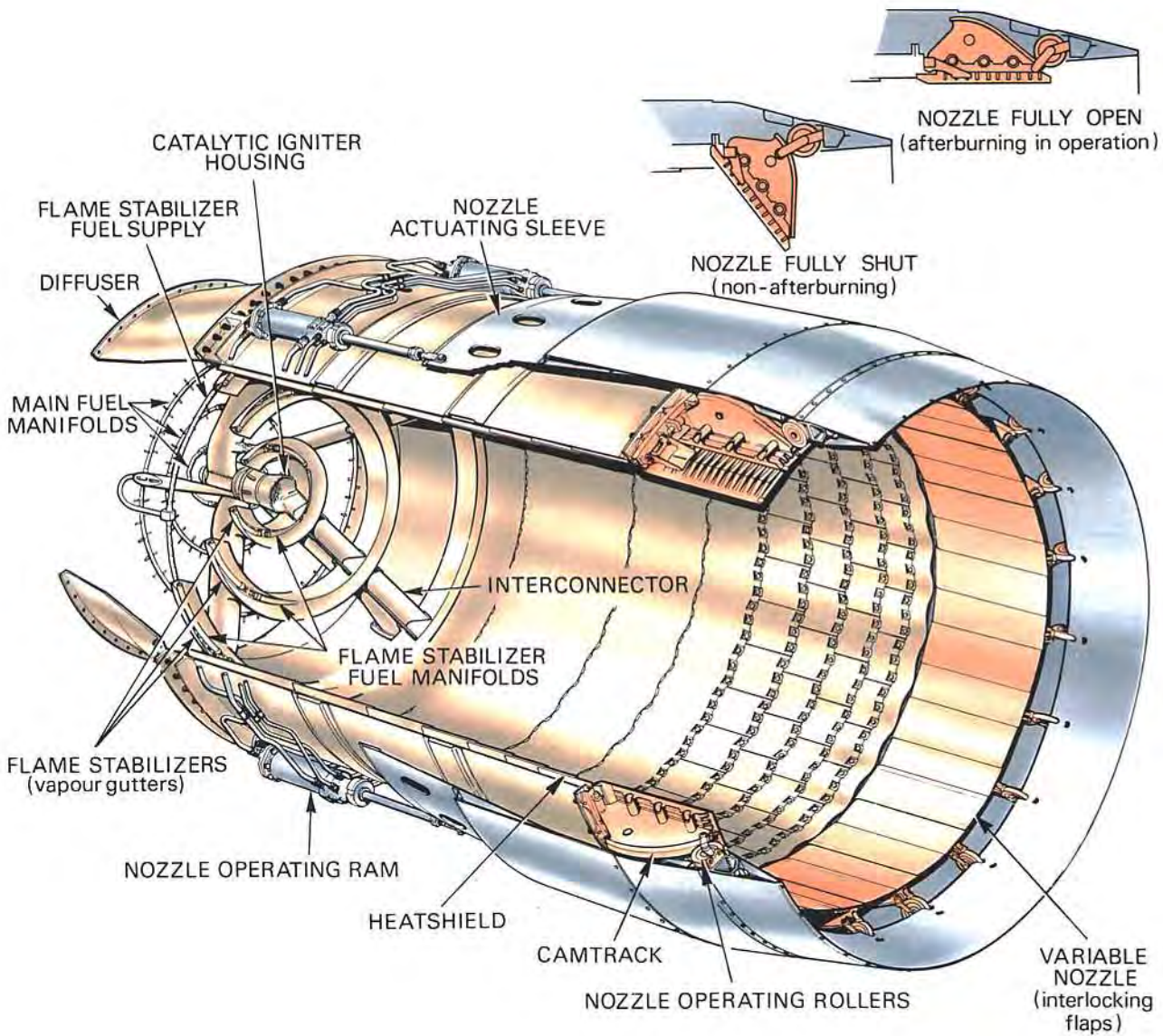
Fig. 16-4   Typical afterburning jet pipe equipment.

17. When afterburning is selected, a signal is relayed to the afterburner fuel control unit. The unit determines the total fuel delivery of the pump and controls the distribution of fuel flow to the burner assembly. Fuel from the burners is ignited, resulting in an increase in jet pipe pressure (P6). This alters the pressure ratio across the turbine (P3/P6), and the exit area of the jet pipe nozzle is automatically increased until the correct PS/PS ratio has been restored. With a further increase in the degree of afterburning, the nozzle area is progressively increased to maintain a satisfactory P3/P6 ratio. Fig. 16-6 illustrates a typical afterburner fuel control system.

18. To operate the propelling nozzle against the large 'drag' loads imposed by the gas stream, a pump and either hydraulically or pneumatically operated rams are incorporated in the control system. The system shown in fig. 16-7 uses oil as the
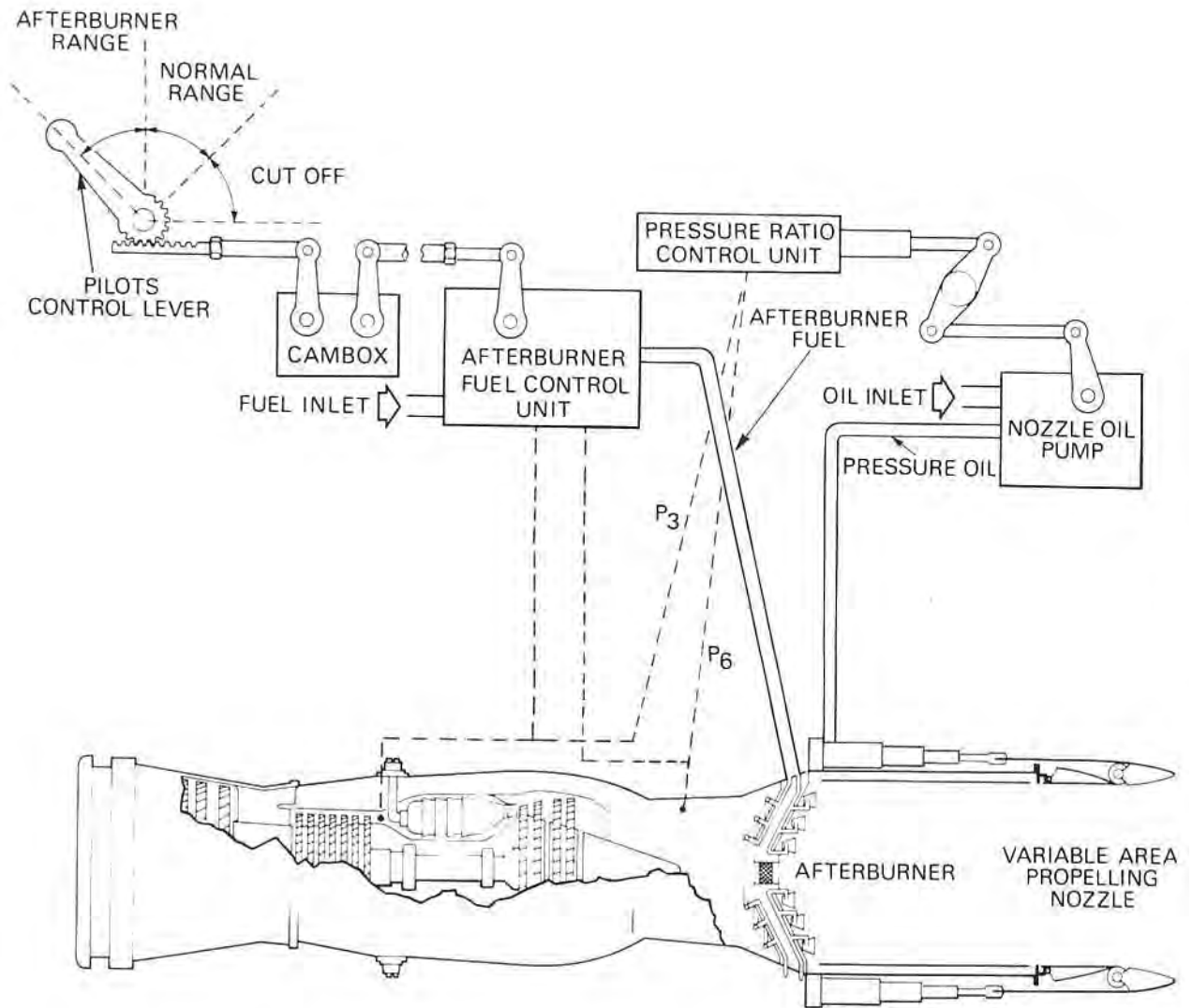
Fig. 16-5   Simplified control system.

hydraulic medium, but some systems use fuel. Nozzle movement is achieved by the hydraulic operating rams which are pressurized by an oil pump, pump output being controlled by a linkage from the pressure ratio control unit. When an increase in afterburning is selected, the afterburner fuel control unit schedules an increase in fuel pump output. The jet pipe pressure (P6) increases, altering the pressure ratio across the turbine (P3/P6). The pressure ratio control unit alters oil pump output, causing an out-of-balance condition between the hydraulic ram load and the gas load on the nozzle flaps. The gas load opens the nozzle to increase its exit area and, as the nozzle opens, the increase in

nozzle area restores the P3/P6 ratio and the pressure ratio control unit alters oil pump output until balance is restored between the hydraulic rams and the gas loading on the nozzle flaps.

THRUST INCREASE

19.   The increase in thrust due to afterburning depends solely upon the ratio of the absolute jet pipe temperatures before and after the extra fuel is burnt. For example, neglecting small losses due to the afterburner equipment and gas flow momentum changes, the thrust increase may be calculated as follows.
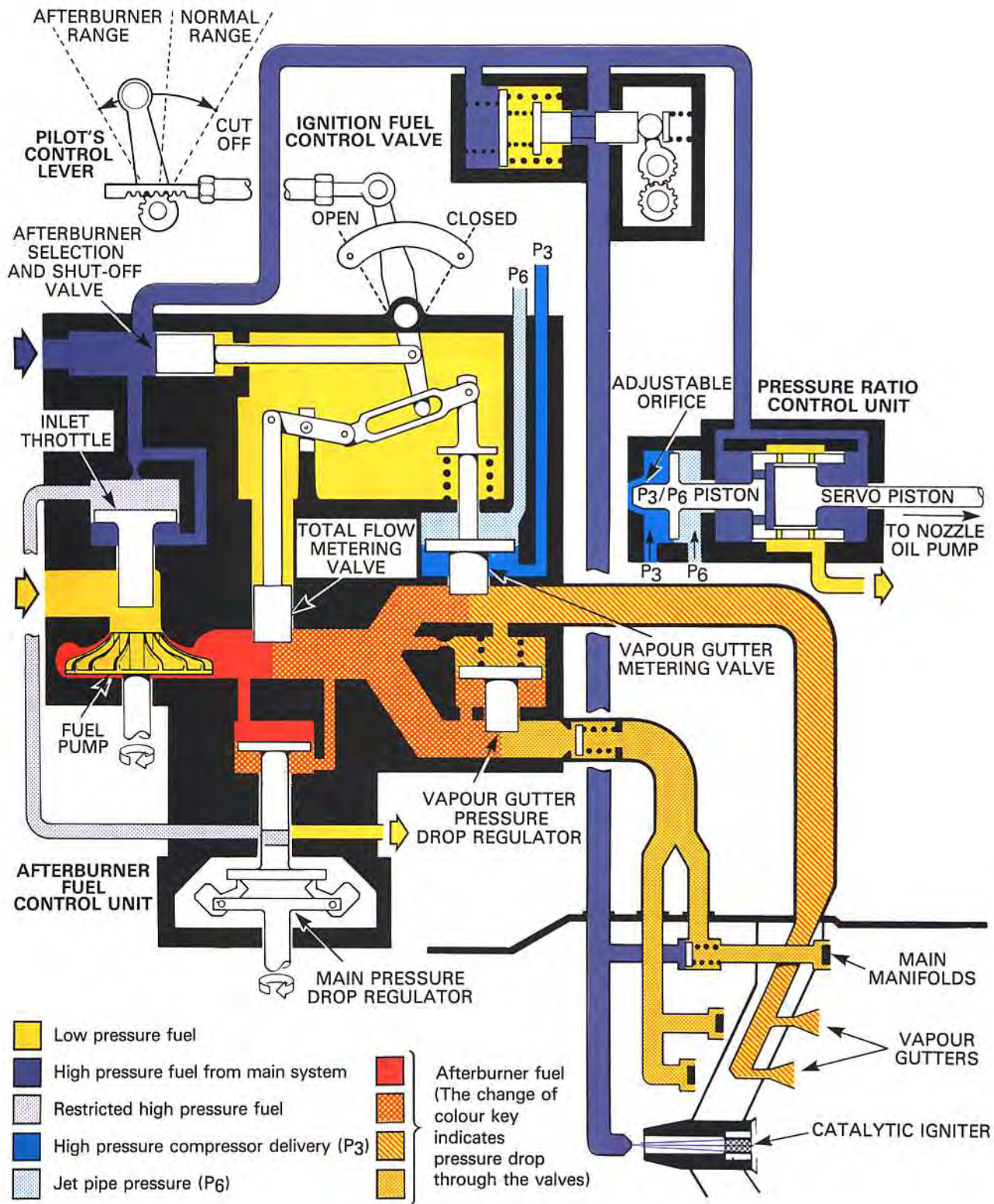
Fig. 16-6   A simplified typical afterburner fuel control system.

AFTERBURNER RANGE | NORMAL RANGE
PILOT'S CONTROL LEVER
CUT OFF
IGNITION FUEL CONTROL VALVE
AFTERBURNER SELECTION AND SHUT-OFF VALVE
OPEN   CLOSED
P3
P6
INLET THROTTLE
ADJUSTABLE ORIFICE
PRESSURE RATIO CONTROL UNIT
P3/P6 PISTON
SERVO PISTON
TO NOZZLE OIL PUMP
P3   P6
TOTAL FLOW METERING VALVE
FUEL PUMP
VAPOUR GUTTER METERING VALVE
VAPOUR GUTTER PRESSURE DROP REGULATOR
AFTERBURNER FUEL CONTROL UNIT
MAIN MANIFOLDS
MAIN PRESSURE DROP REGULATOR
VAPOUR GUTTERS

Low pressure fuel
High pressure fuel from main system
Restricted high pressure fuel
High pressure compressor delivery (P3)
Jet pipe pressure (P6)
Afterburner fuel (The change of colour key indicates pressure drop through the valves)
CATALYTIC IGNITER

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Fig. 16-7   A simplified typical afterburner nozzle control system.

20.   Assuming a gas temperature before afterburning of 640 deg. C. (913 deg. K.) and with afterburning of 1,269 deg. C. (1,542 deg. K.). then the temperature ratio = 1,542 = 1.69. 913

The velocity of the jet stream increases as the square root of the temperature ratio. Therefore, the jet velocity = $\sqrt{T.69}$ = 1.3. Thus, the jet stream velocity is increased by 30 per cent, and the increase in static thrust, in this instance, is also 30 per cent (fig. 16-8).

21.   Static thrust increases of up to 70 per cent are obtainable from low by-pass engines fitted with afterburning equipment and at high forward speeds several times this amount of thrust boost can be obtained. High thrust boosts can be achieved on low by-pass engines because of the large amount of oxygen in the exhaust gas stream and the low initial temperature of the exhaust gases.
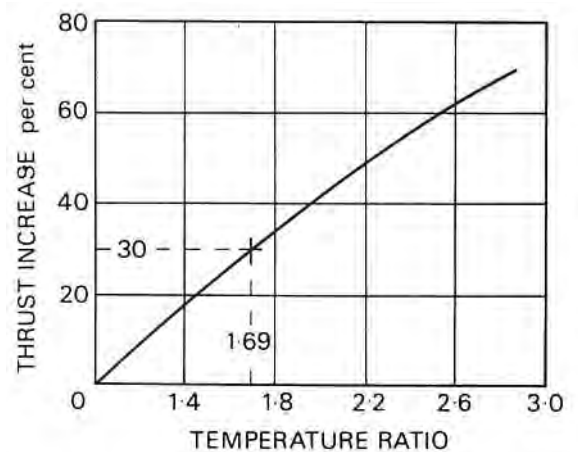
Fig. 16-8   Thrust increase and temperature ratio.

22. It is not possible to go on increasing the amount of fuel that is burnt in the jet pipe so that all the available oxygen is used, because the jet pipe would not withstand the high temperatures that would be incurred and complete combustion cannot be assured.

FUEL CONSUMPTION

23. Afterburning always incurs an increase in specific fuel consumption and is, therefore, generally limited to periods of short duration. Additional fuel must be added to the gas stream to obtain the required temperature ratio (para. 19). Since the temperature rise does not occur at the peak of compression, the fuel is not burnt as efficiently as in the engine combustion chamber and a higher specific fuel consumption must result. For example, assuming a specific fuel consumption without afterburning of 1,15 lb./hr./lb. thrust at sea level and a speed of Mach 0,9 as shown in fig. 16-9. then with 70 per cent afterburning under the same conditions of flight, the consumption will be increased to
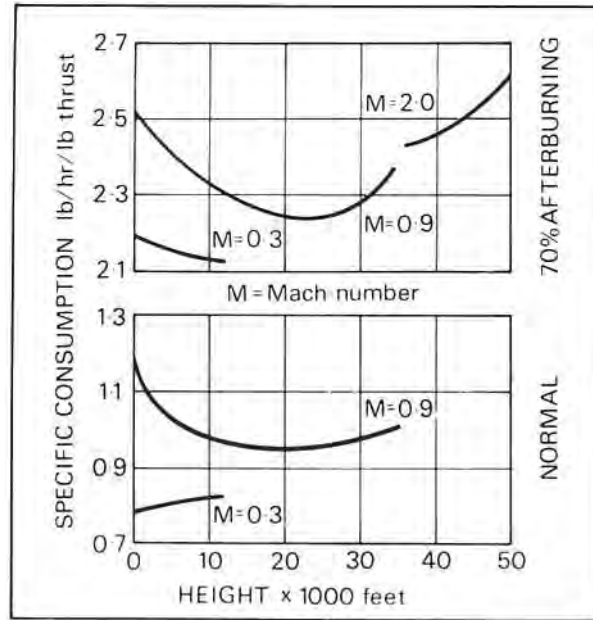
Fig. 16-9   Specific fuel consumption comparison.
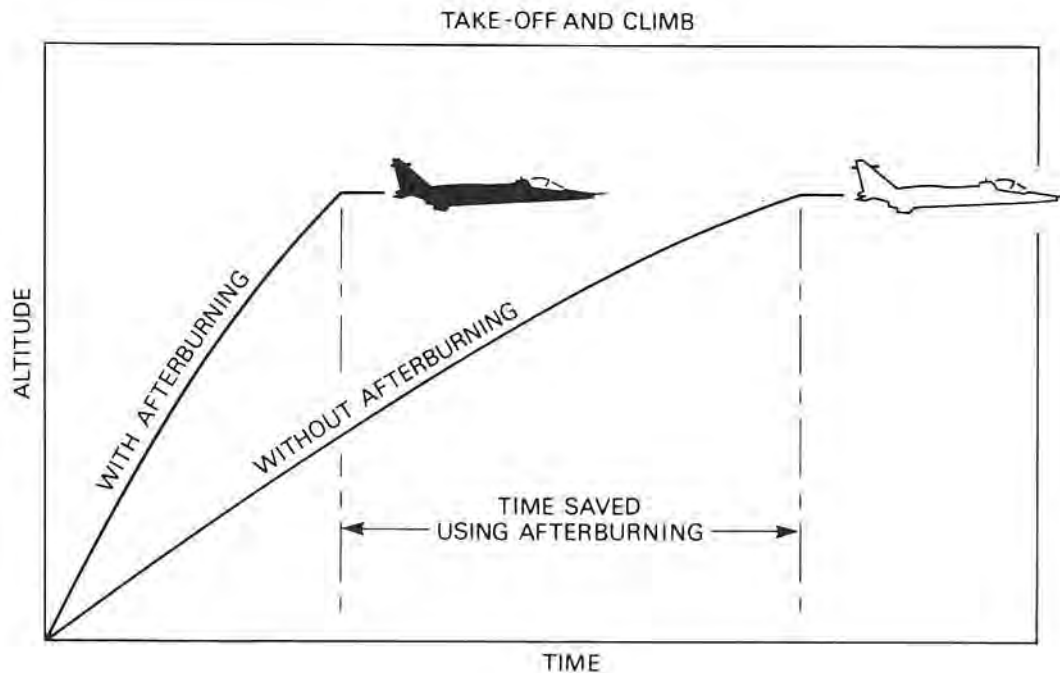
Fig. 16-10   Afterburning and its effect on the rate of climb.

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approximately 2.53 lb./hr./lb. thrust. With an increase in height to 35,000 feet this latter figure of 2.53 lb./hr./lb. thrust will fall slightly to about 2.34 lb./hr./lb. thrust due to the reduced intake temperature. When this additional fuel consumption is combined with the improved rate of take-off and climb (fig. 16-10), it is found that the amount of fuel required to reduce the time taken to reach operation height is not excessive.

Rolls-Royce Dart
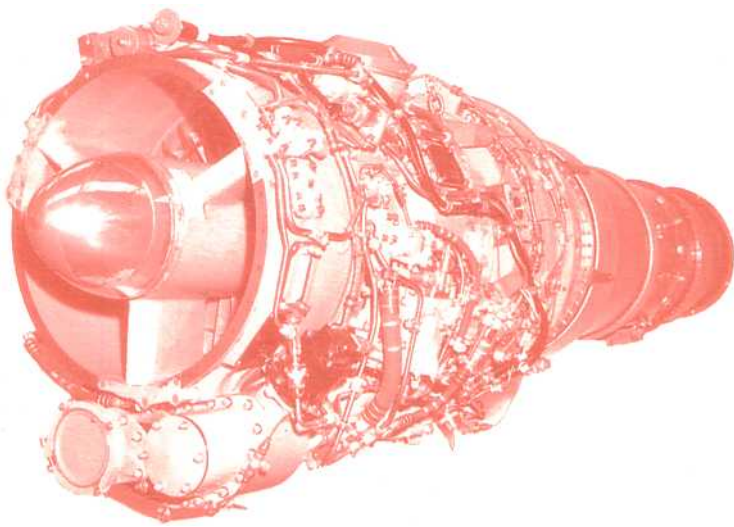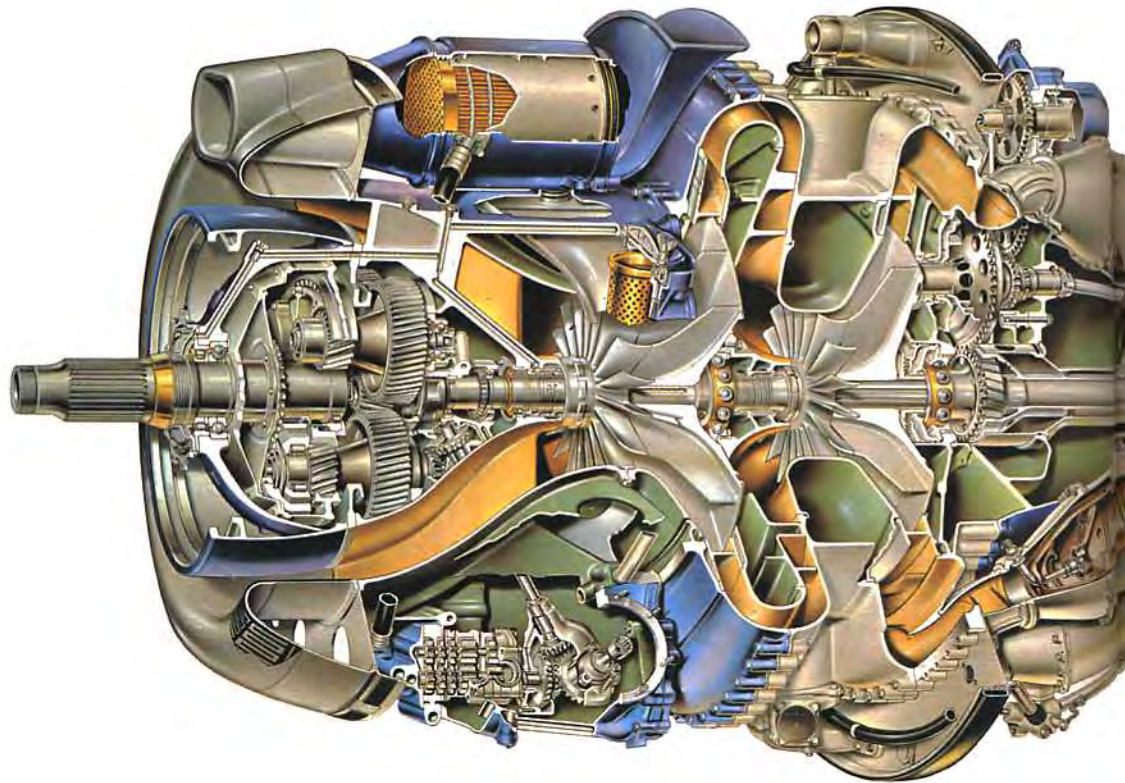
Armstrong Siddeley Viper

The Viper was designed as a result of experience gained with the larger Sapphire turbojet. Originally built as a 1,640 lb thrust short-life engine for target drones, it later emerged as a long life engine for the Jet Provost. Subsequently the engine was developed by Bristol Siddeley as the power plant for civil executive jets, and Rolls-Royce for present generation trainers and light strike aircraft with a maximum thrust of 4,400 lb (5,000 lb with reheat).

17: Water injection

Contents

INTRODUCTION
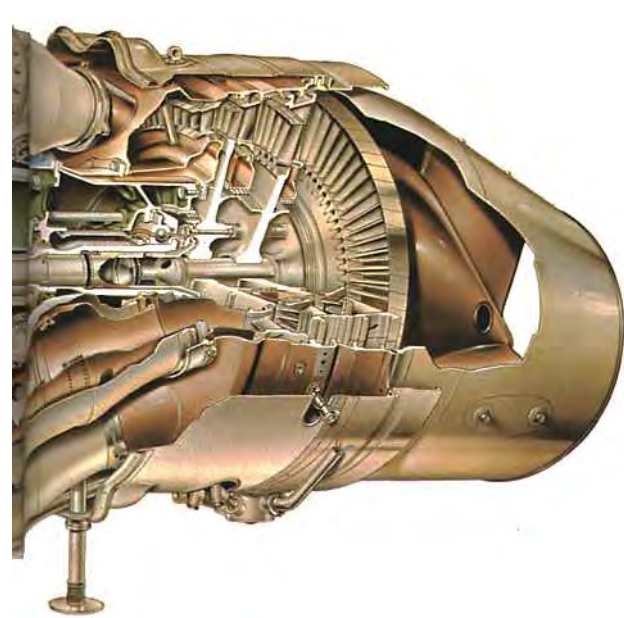
1. The maximum power output of a gas turbine engine depends to a large extent upon the density or weight of the airflow passing through the engine. There is, therefore, a reduction in thrust or shaft horsepower as the atmospheric pressure decreases with altitude, and/or the ambient air temperature increases. Under these conditions, the power output can be restored or, in some instances, boosted for take-off by cooling the airflow with water or water/methanol mixture (coolant). When methanol is added to the water it gives anti-freezing properties and also provides an additional source of fuel. A typical turbo-jet engine thrust restoration curve is shown in fig. 17-1 and a turbo-propeller engine power restoration and boost curve is shown in fig. 17-2.

2. There are two basic methods of injecting the coolant into the airflow. Some engines have the coolant sprayed directly into the compressor inlet, but the injection of coolant into the combustion chamber inlet is usually more suitable for axial flow compressor engines. This is because a more even distribution can be obtained and a greater quantity of coolant can be satisfactorily injected.

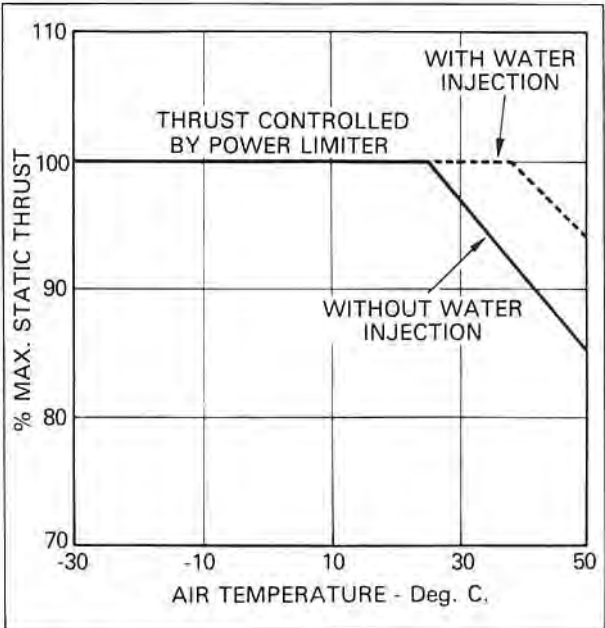3. When water/methanol mixture is sprayed into the compressor inlet, the temperature of the compressor

Fig. 17-1   Turbo-jet thrust restoration.
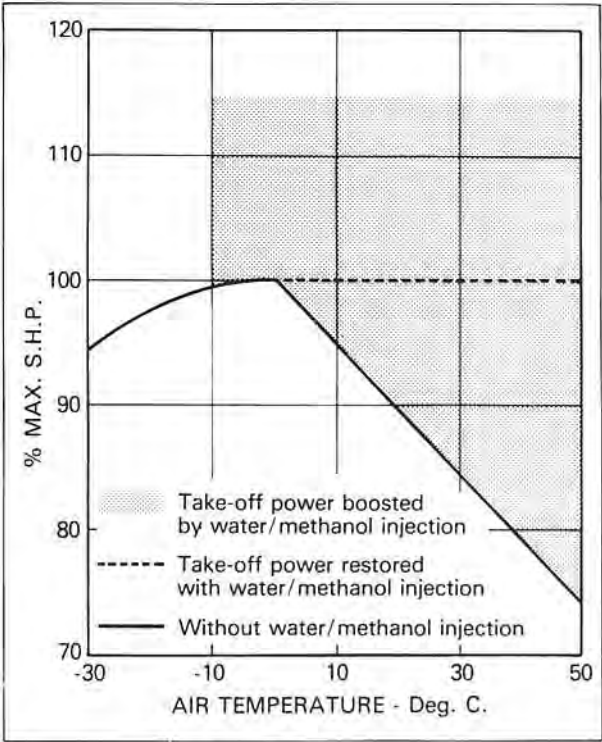
Fig. 17-2   Turbo-propeller power boost.
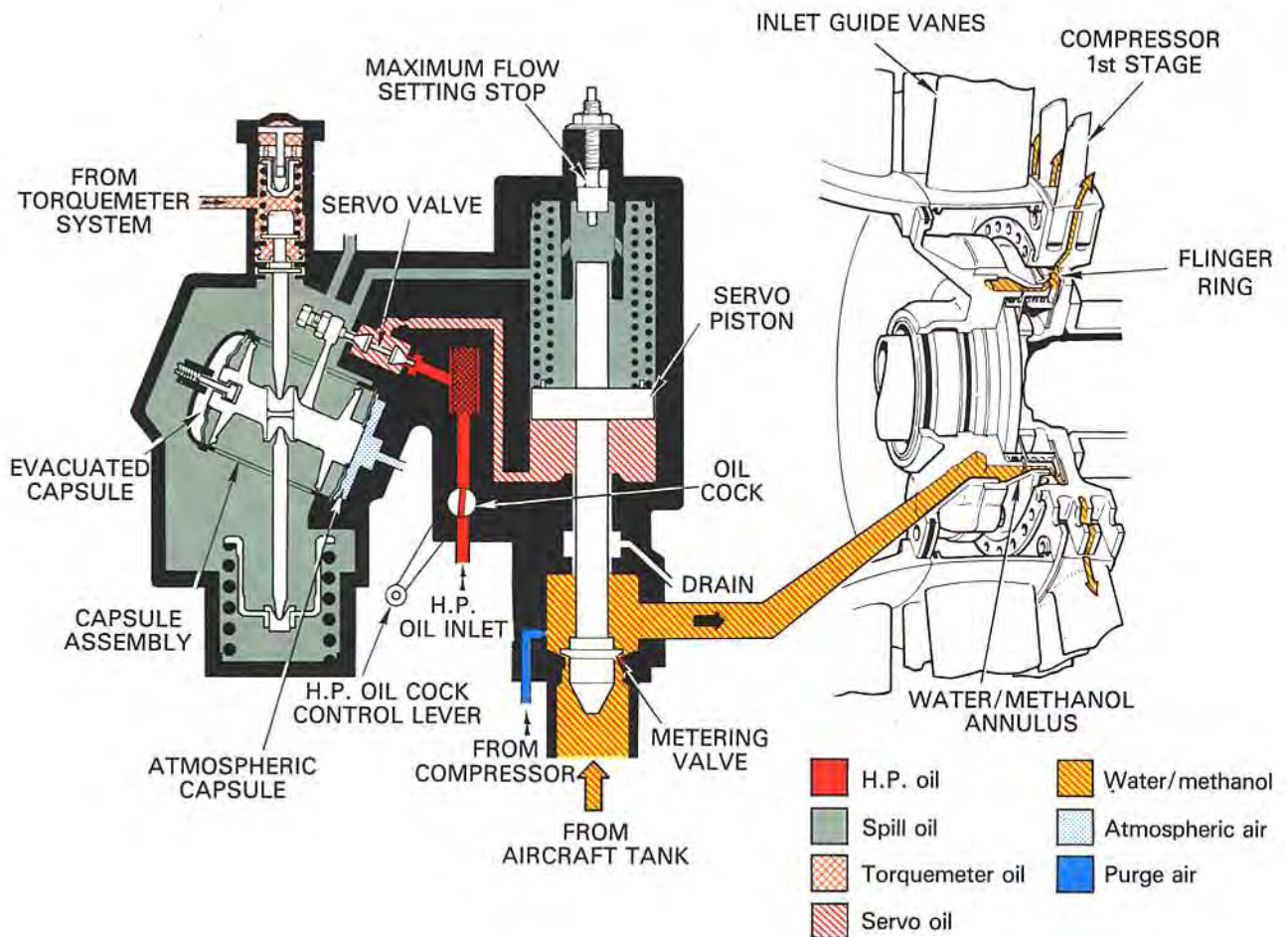
Fig. 17-3   A typical compressor inlet injection system.

inlet air is reduced and consequently the air density and thrust are increased. If water only was injected, it would reduce the turbine inlet temperature, but with the addition of methanol the turbine inlet temperature is restored by the burning of methanol in the combustion chamber. Thus the power is restored without having to adjust the fuel flow.

4. The injection of coolant into the combustion chamber inlet increases the mass flow through the turbine, relative to that through the compressor. The pressure and temperature drop across the turbine is thus reduced, and this results in an increased jet pipe pressure, which in turn gives additional thrust. The consequent reduction in turbine inlet temperature, due to water injection, enables the fuel system to schedule an increase of fuel flow to a value that gives

an increase in the maximum rotational speed of the engine, thus providing further additional thrust, Where methanol is used with the water, the turbine inlet temperature is restored, or partially restored, by the burning of the methanol in the combustion chamber.

COMPRESSOR INLET INJECTION

5. The compressor inlet injection system shown in fig. 17-3 is a typical system for a turbo-propeller engine. When the injection system is switched on, water/methanol mixture is pumped from an aircraft-mounted tank to a control unit. The control unit meters the flow of mixture to the compressor inlet through a metering valve that is operated by a servo piston. The servo system uses engine oil as an operating medium, and a servo valve regulates the
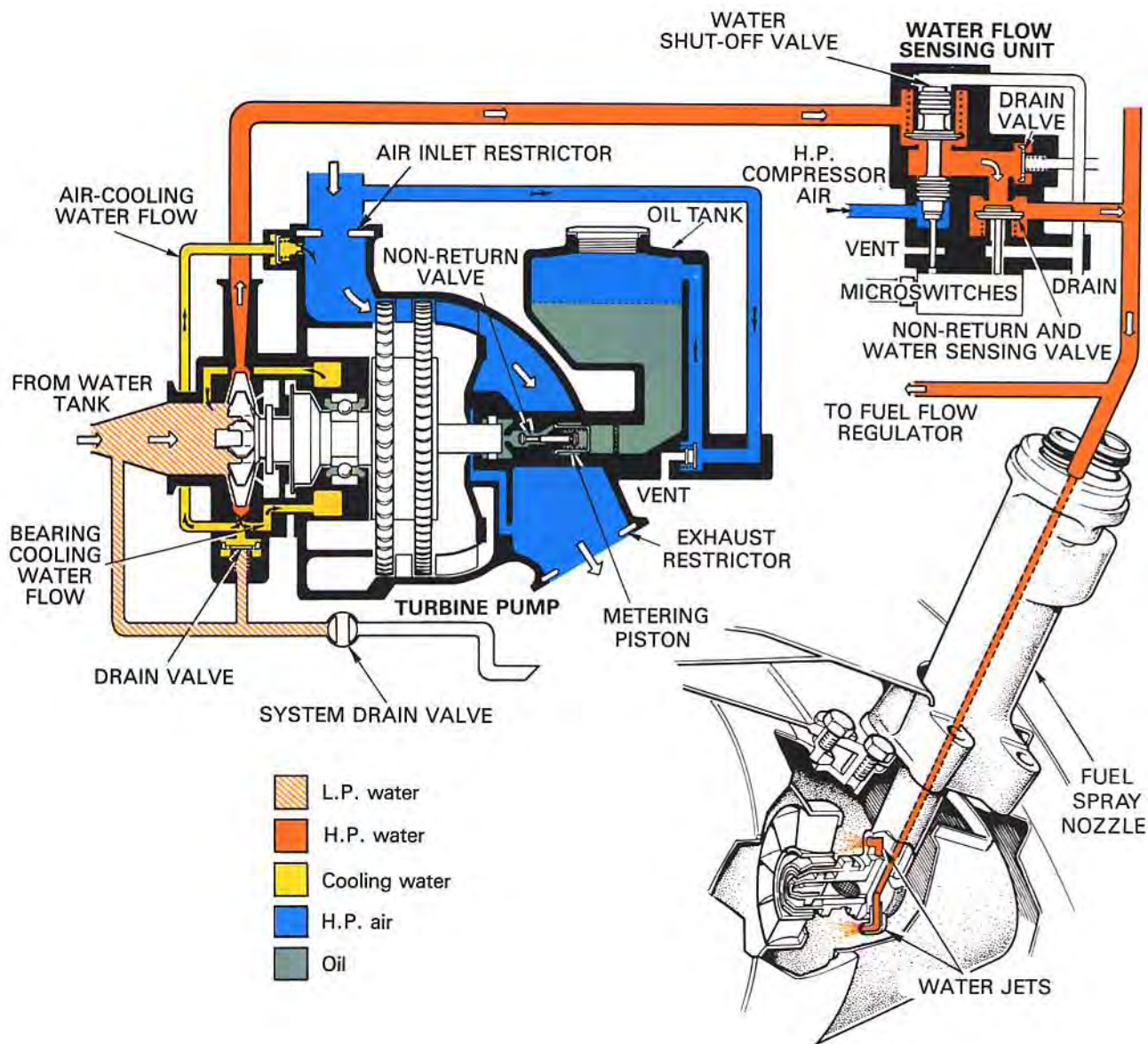
Fig. 17-4   A typical combustion chamber injection system.

supply of oil. The degree of servo valve opening is set by a control system that is sensitive to propeller shaft torque oil pressure and to atmospheric air pressure acting on a capsule assembly.

6. The control unit high pressure oil cock control lever is interconnected to the throttle control system in such a manner that, until the throttle is moved towards the take-off position, the oil cock remains closed, and thus the metering valve remains closed, preventing any mixture flowing to the compressor

inlet Movement of the throttle control to the take-off position opens the oil cock, and the oil pressure passes through the servo valve to open the metering valve by means of the servo piston.

COMBUSTION CHAMBER INJECTION

7. The combustion chamber injection system shown in fig. 17-4 is a typical system for a turbo-jet engine. The coolant flows from an aircraft-mounted tank to an air-driven turbine pump that delivers it to a water flow sensing unit. The water passes from the

sensing unit to each fuel spray nozzle and is sprayed from two jets onto the flame tube swirl vanes, thus cooling the air passing into the combustion zone. The water pressure between the sensing unit and the discharge jets is sensed by the fuel control system, which automatically resets the engine speed governor to give a higher maximum engine speed.

8.   The water flow sensing unit opens only when the correct pressure difference is obtained between compressor delivery air pressure and water pressure. The system is brought into operation when the engine throttle lever is moved to the take-off position, causing microswitches to operate and select the air supply for the turbine pump.

9.   The sensing unit also forms a non-return valve to prevent air pressure feeding back from the discharge jets and provides for the operation of an indicator light to show when water is flowing.

Rolls-Royce Pegasus
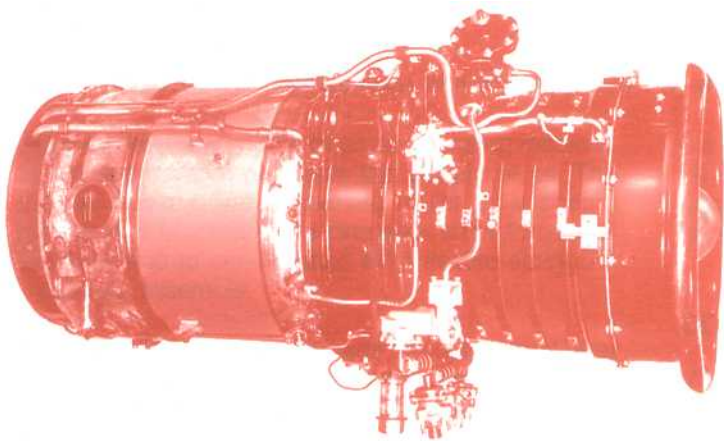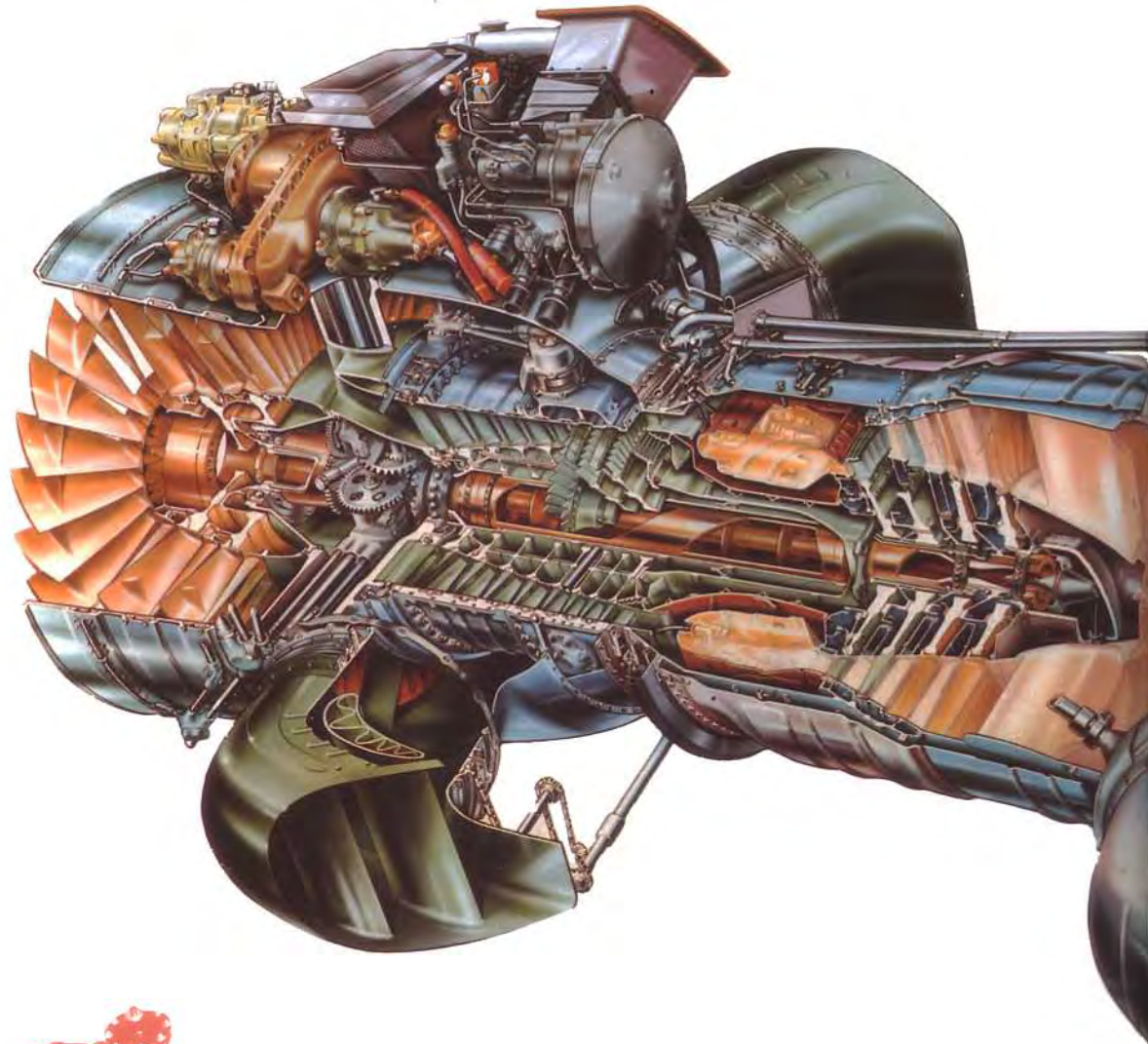
Rolls-Royce RB 108

The RB108 was the first engine to be designed specifically as a direct VTOL engine. First running in July 1955 the engine was subsequently thrust rated at 2340 lb, giving a thrust to weight ratio of 8.7:1. In addition to powering a variety of VTOL test rigs, the RB108 flew in a Gloster Meteor, the Short SC1 and the Marcel Dassault Balzac.

18: Vertical/short take-off and landing

Contents

INTRODUCTION

1. Vertical take-off and landing (VTOL) or short take-off and landing (STOL) are desirable characteristics for any type of aircraft, provided that the normal flight performance characteristics, including payload/range, are not unreasonably impaired. Until the introduction of the gas turbine engine, with its high power/weight ratio, the only powered lift system capable of VTOL was the low disc loading rotor, as on the helicopter.

2. Early in 1941, the late Dr A. A. Griffiths, the then Chief Scientist at Rolls-Royce, envisaged the use of the jet engine as a powered lift system. However, it was not until 1947 that a light weight jet engine, designed by Rolls-Royce for missile propulsion, existed and had a high enough thrust/weight ratio for the first pure lift-jet engine to be developed from it.

3. In 1956 the Bristol Aero-Engine Company was approached by Monsieur Michel Wibault with a proposal to use a turbo-shaft engine and a reduction gearbox to drive four centrifugal compressors which would be situated two on each side of the aircraft. The casing of these compressors could be rotated to change direction of the thrust (fig. 18-1). The concept incorporated two original ideas i.e. the ability to deflect the thrust over the complete range of angles from the position for normal flight to that for vertical lift and a system where the resultant thrust always acted near to the centre of gravity of the aircraft.

Fig. 18-1   Michel Wibault's ground attack gyropter (concept) 1956

4. The principle proposed by M. Wibault was developed by using a pure jet engine with a free power turbine to drive an axial flow fan which exhausted into a pair of swivelling nozzles, one on each side of the aircraft. A further development was to use the fan to supercharge the engine, exhausting the by-pass air through one pair of swivelling nozzles and adding a second pair of swivelling nozzles to the
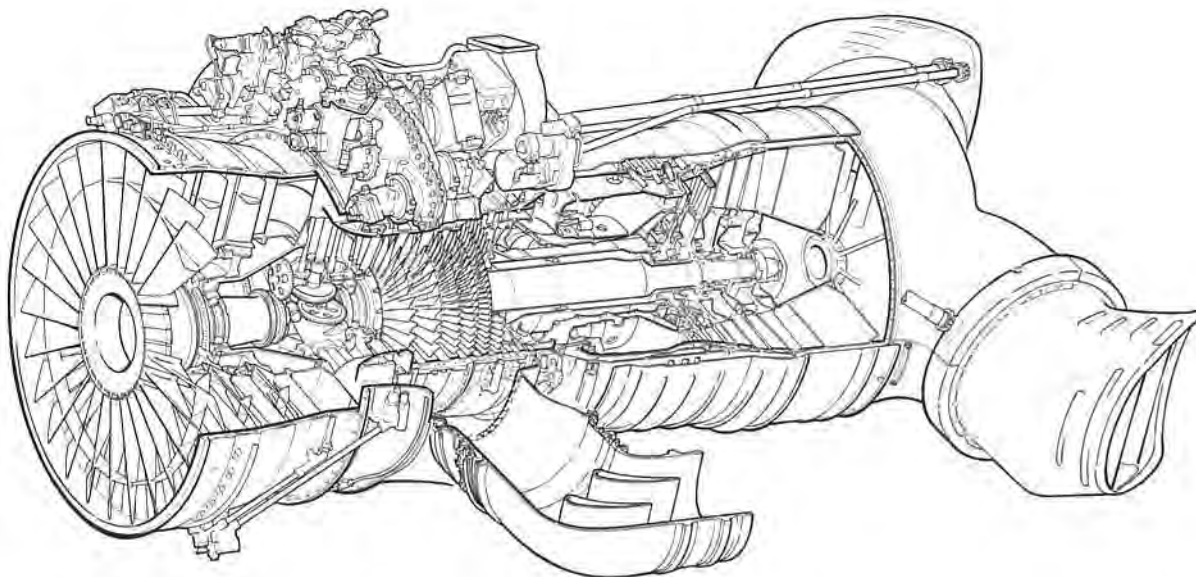
Fig. 18-2   Lift/Propulsion engine.

Fig. 18-3   V/STOL fighter aircraft.

exhaust system from the engine turbine. In this way the first ducted fan lift/propulsion engine (the Pegasus) evolved (fig. 18-2).

5.   Subsequent experience with the Pegasus engine in the Harrier V/STOL fighter aircraft (fig. 18-3), lead to the development of the short take-off and vertical landing (STOVL) operational technique. In this way the additional lift generated by the aircraft wing, even after a short take-off run, provided a large increase in the payload/range capability of the aircraft compared to a pure vertical take-off. Vertical landing had several operational advantages compared to a short landing and so was maintained.

METHODS OF PROVIDING POWERED LIFT

6.   Although the Pegasus engine is the only V/STOL engine in operational service in the Western World there are several possible methods of providing powered lift, such as;

(1)   Deflecting (or vectoring) the exhaust gases and hence the thrust of the engine.

(2)   Using specially designed engines for lift only.

(3)   Driving a lift system, which is remote from the engine, either from the engine or by a separate power unit.

(4)   Swivelling the engines.

(5)   For STOL aircraft, using bleed air from the engines to increase circulation around the wing and hence increase lift.

In several of the projected V/STOL aircraft a combination of two or more of these methods has been used.

Lift/Propulsion engines

7.   The lift/propulsion engine is capable of providing thrust for both normal wing borne flight and for lift. This is achieved by changing the direction of the thrust either by a deflector system consisting of one, two or four swivelling nozzles or by a device known as a switch-in deflector which redirects the exhaust gases from a rearward facing propulsion nozzle to one or two downward facing lift nozzles (fig, 18-4).

8.   Thrust deflection on a single nozzle is accomplished by connecting together sections of the jet

TWIN NOZZLE DEFLECTOR

FOUR NOZZLE DEFLECTOR

SWITCH-IN DEFLECTOR
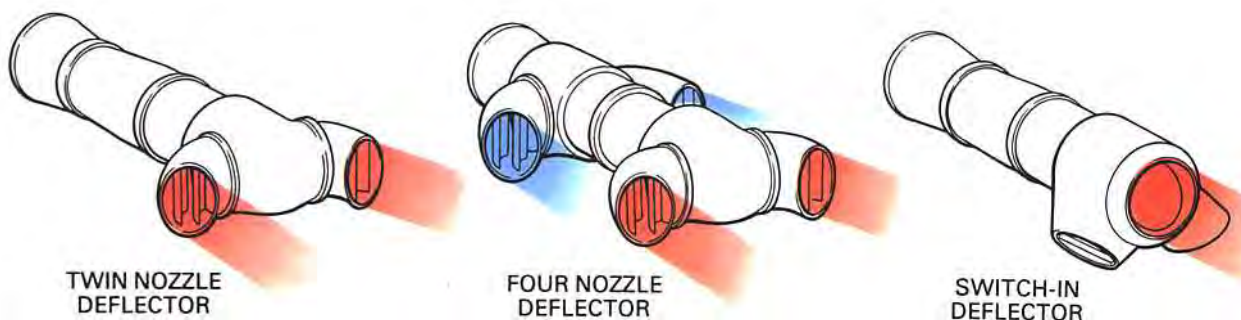
Fig. 18-4   Thrust deflector systems.

CONTRA-ROTATING JET PIPE SECTIONS

VARIABLE NOZZLE

NOZZLE IN PROPULSION POSITION

NOZZLE DEFLECTED FOR
VERTICAL LIFT
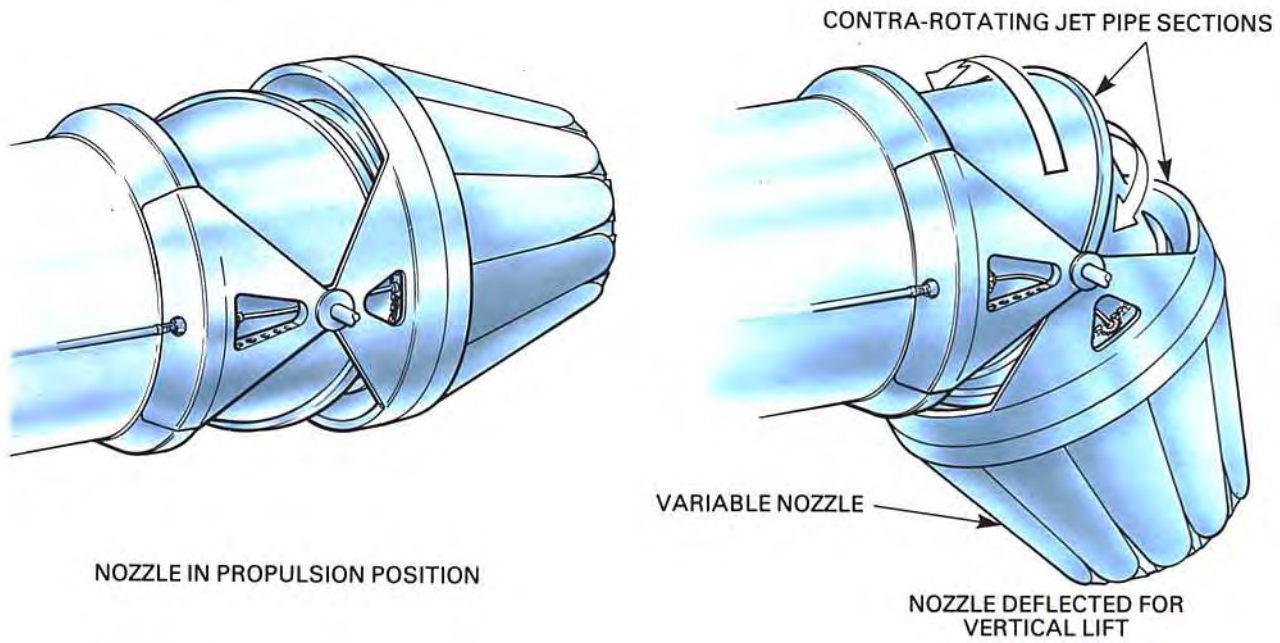
Fig. 18-5   Deflector nozzle.

CHAIN GUARD

NOZZLE

SPACERS

RACE BEARINGS

CHAIN

DRIVING SPROCKET
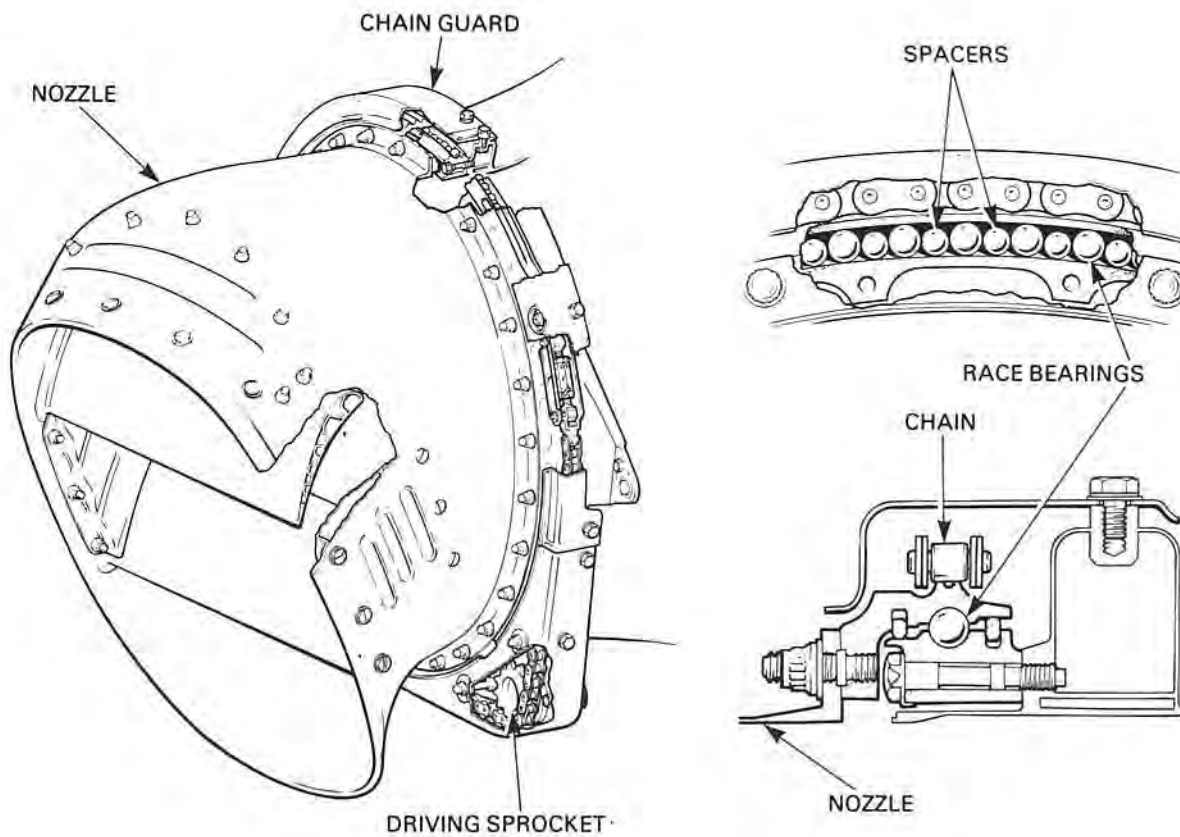
NOZZLE

Fig. 18-6   Side mounted swivelling nozzle.

pipe, the joint faces of which are so angled that, when the sections are counter-rotated, the nozzle moves from the horizontal to the vertical position (fig. 18-5). To avoid either a side component o! thrust or a thrust line offset from the engine axis during the movement of the nozzle it is necessary that the first joint face is perpendicular to the axis of the jet pipe. If it is desired that the nozzle does not rotate, as may be the case if it is a variable area nozzle, a third joint face which is perpendicular to the axis of the nozzle is required.

9.   The two and four nozzle deflector systems use side mounted nozzles (fig. 18-6) which can rotate on simple bearings through an angle of well over 90 degrees so that reverse thrust can be provided if required. A simple drive system, for example, a sprocket and chain, can be used and by mechanical connections all the nozzles can be made to deflect simultaneously. For forward flight, to avoid a high performance loss and consequent increase in fuel consumption, careful design of the exhaust unit and nozzle aerodynamic passages are essential to minimize the pressure losses due to turning the exhaust flow through two close coupled bends (fig. 18-7).
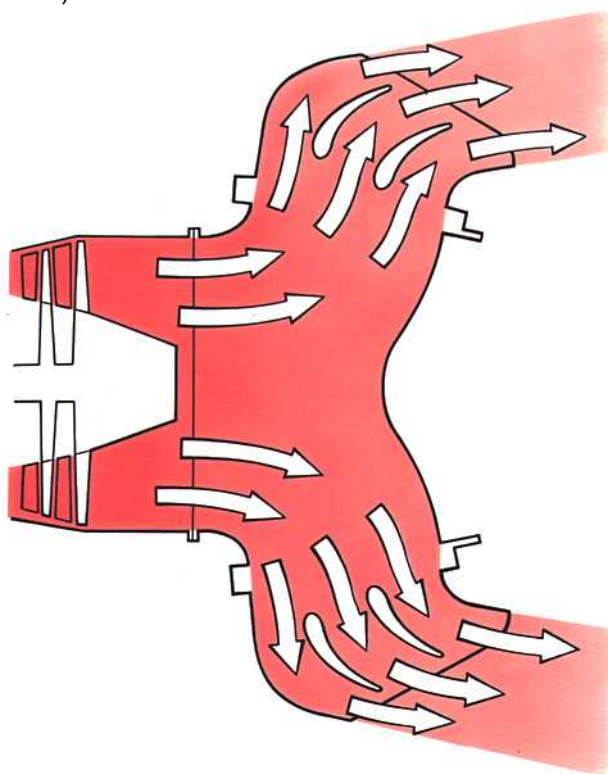
Fig. 18-7   Nozzle duct configuration.

10.   The switch-in deflector consists of one or a pair of heavily reinforced doors which form part of the jet pipe wall when the engine is operating in the forward thrust condition. To select lift thrust, the doors are moved to blank off the conventional propelling nozzle and direct the exhaust flow into a lift nozzle (fig. 18-8). The lift nozzles may be designed so that they can be mechanically rotated to vary the angle of the thrust and permit intermediate lift/thrust positions to be selected.

11.   A second type of switch-in deflector system is used on the tandem fan or hybrid fan vectored thrust engine (fig. 18-9). In this case the deflector system is situated between the stages of the fan of a mixed flow turbo-fan engine. In normal flight the valve is positioned so that the engine operates in the same manner as a mixed flow turbo-fan and for lift thrust the valve is switched so that the exhaust flow from the front part of the fan exhausts through downward facing lift nozzles and a secondary inlet is opened to provide the required airflow to the rear part of the fan and the main engine. On a purely subsonic V/STOL aircraft where fuel consumption is important the valve may be dispensed with and the engine operated permanently in the latter high by-pass mode described above.

12.   Thrust deflecting nozzles will create an upstream pressure distortion which may excite vibration of the fan or low pressure turbine blades if the nozzle system is close to these components. Snubbers (Part 3) may be used on the fan blades to resist vibration. On the low pressure turbine, shrouds at the blade tips (Part 5) or wire lacing may be used to achieve the same result.

Lift engines
13.   The lift engine is designed to produce vertical thrust during the take-off and landing phases of V/STOL aircraft. Because the engine is not used in normal flight it must be light and have a small volume to avoid causing a large penalty on the aircraft. The lift engine may be a turbo-jet which for a given thrust gives the lowest weight and volume. Should a low jet velocity be necessary a lift fan may be employed.

14.   Pure lift-jet engines have been developed with thrust/weight ratios of about 20:1 and still higher values are projected for the future. Weight is reduced by keeping the engine design simple and also by extensive use of composite materials (fig. 18-10). Because the engine is operated for only limited periods during specific flight conditions i.e. during take-off and landing, the fuel system can be simplified and a total loss oil system (Part 8), in which
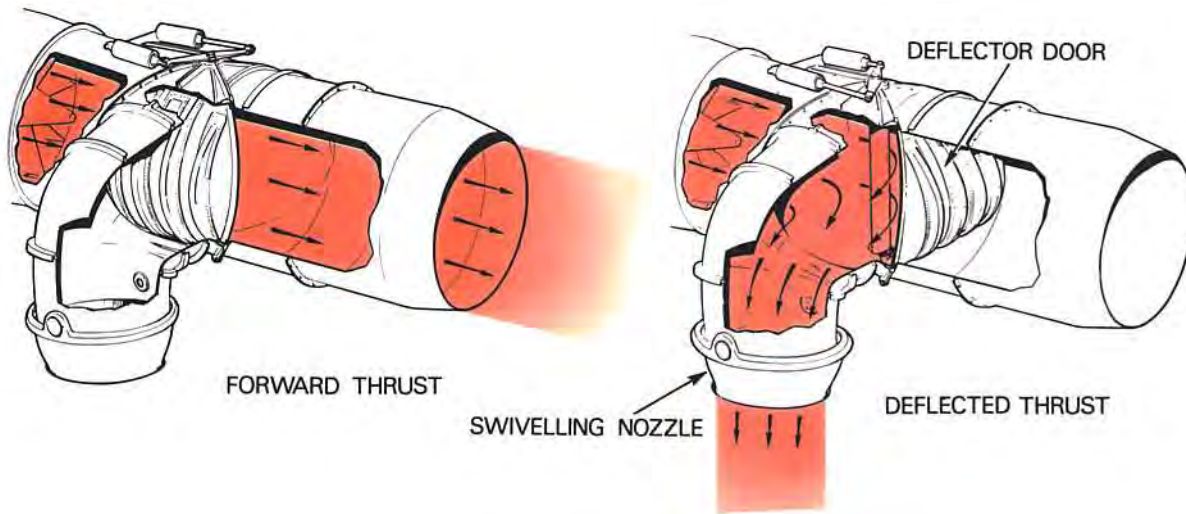
Fig. 18-8   Switch-In deflector system.

the used lubricating oil is ejected overboard, can be used.

15.   Lift engines can be designed to operate in the vertical or horizontal position and a thrust deflecting nozzle fitted to provide some of the advantages of thrust vectoring. Alternatively, the engine may be mounted so that it can swivel through a large angle to provide thrust vectoring. The lift-jet engine will have an extremely hot, high velocity jet exhaust and to reduce ground erosion by the jet the normal

exhaust nozzle may be replaced by a multi-lobe nozzle to increase the rate of mixing with the surrounding air.

16.   The lift-fan engine is designed to reduce the jet exhaust velocity, to reduce ground erosion and allow operation from unprepared ground surfaces. It also reduces the jet noise significantly. A range of design options have been considered for this type of engine and some are shown on fig. 18-11.
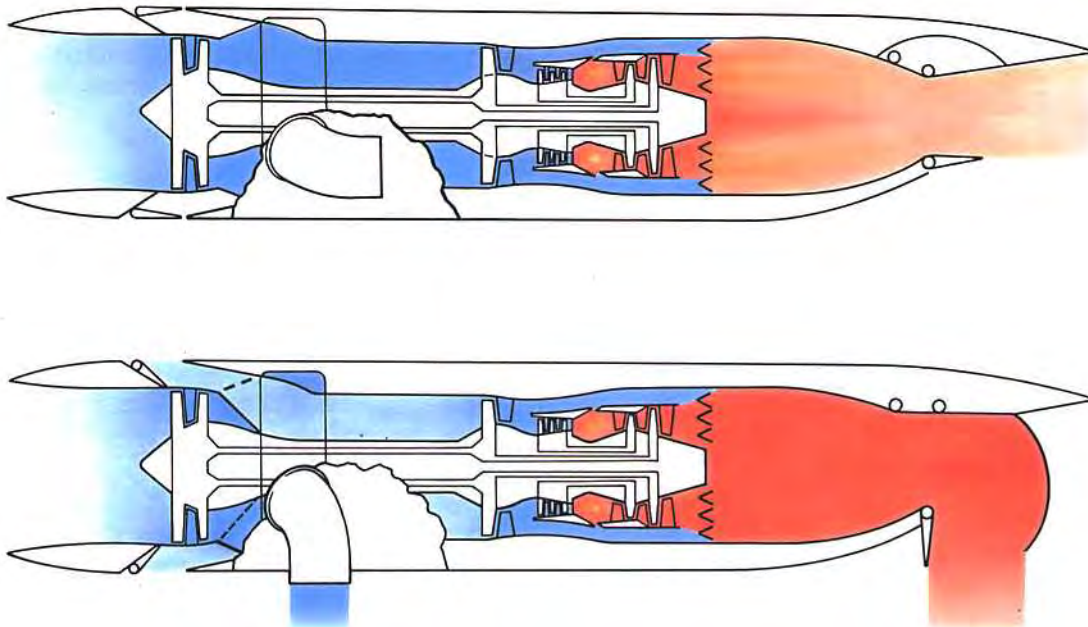
Fig. 18-9   Vectored thrust engine.
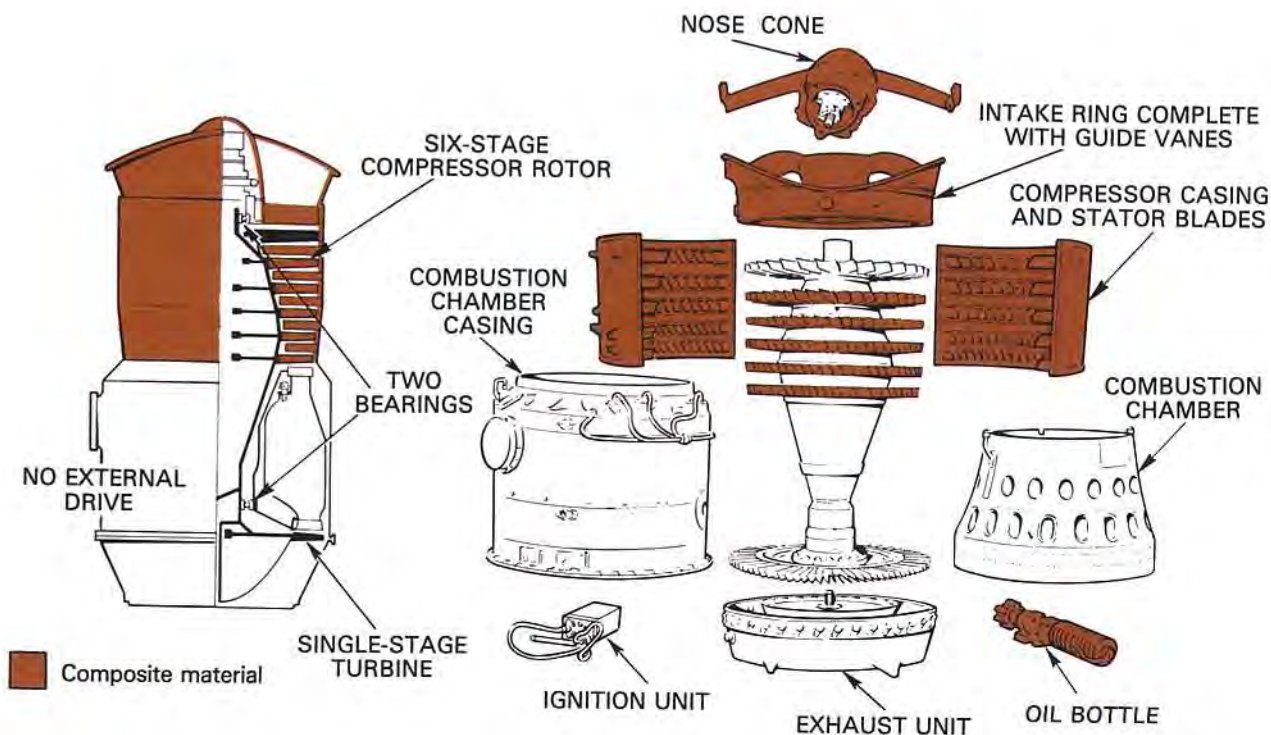
Fig. 18-10   A lift-jet engine.

Remote lift systems

17.   Direct lift remote systems duct the by-pass air or engine exhaust air to downward facing lift nozzles remote from the engine. These nozzles may be in the front fuselage of the aircraft or in the wings. The engine duct is blocked by means of a diverter similar to that described in para. 10.

18.   The remote lift-fan (fig. 18-12) is mounted in the aircraft wing or fuselage, and is driven mechanically or by air or gas ducted into a tip turbine, The drive system is provided by the main propulsion power plant or by a separate engine.

19.   The advantage of the remote lift system is that it gives some freedom to the aircraft to position the
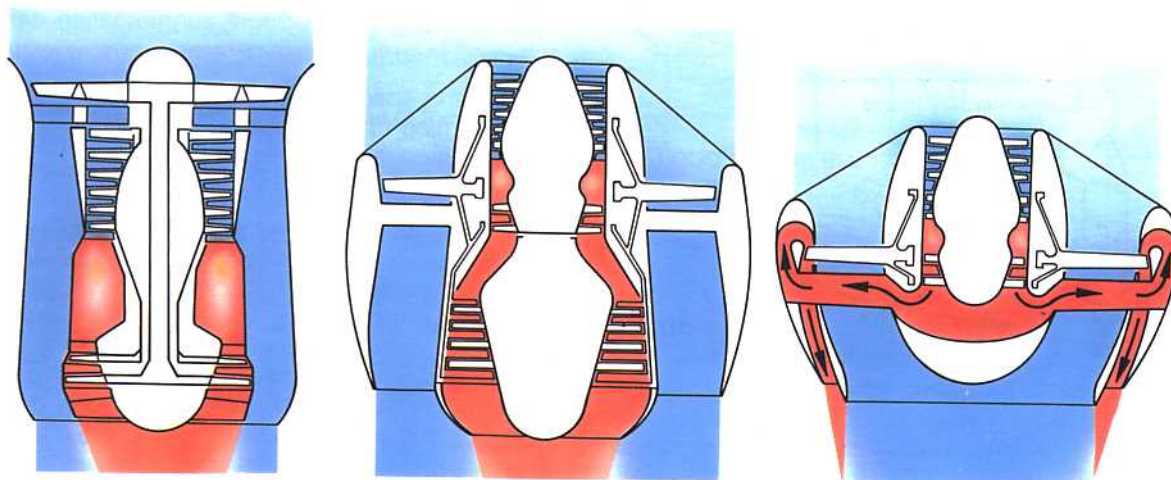
Fig. 18-11   Lift-fan engine configurations.

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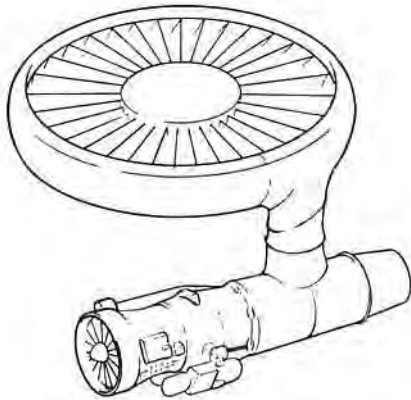
Fig. 18-12   Remote lift fan.

propulsion system to the best advantage whilst still maintaining the resultant thrust near the aircraft centre of gravity in the jet lift mode. This freedom is achieved at a cost of increased volume, particularly with the gas driven systems, due to the size of the ducts to feed the gas to the remote lift system. Although the mechanically driven remote lift-fan eliminates the need for these large gas ducts, it is done at the expense of long shafts and high power gearboxes and clutch systems.

Swivelling engines
20.   This method consists of having propulsion engines which can be mechanically swiveled closed
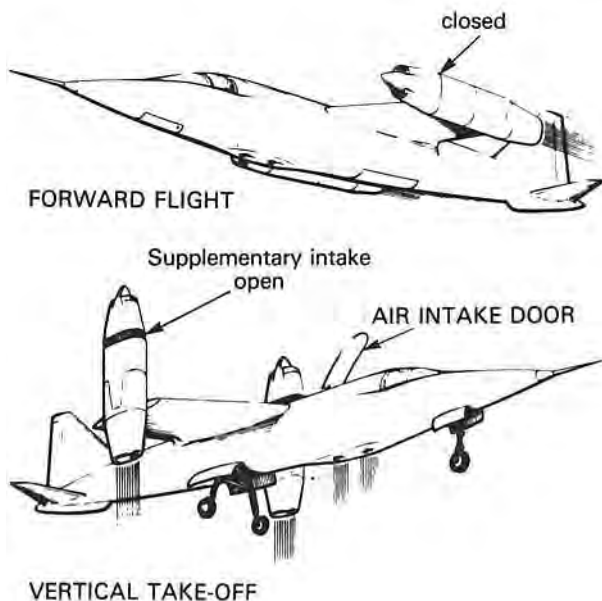
Fig. 18-13   Jet lift with swivelling nozzles.

through at least 90 degrees to provide thrust vectoring (fig. 18-13). In addition to these propulsion engines, one or more lift engines may be installed to provide supplementary lift during the take-off and landing phase of flight.

21.   The swivelling engine system can only be used with two or more engines. This then introduces the problem of safety in the event of an engine failure. So, although there is only a small weight penalty and no increase in fuel consumption, safety considerations tend to offset these advantages compared to some of the other powered lift systems. The normal method of providing aircraft control at low speeds is by differential throttling and vectoring of the engines which simplifies the basic engine design but makes the control system more complex.

Bleed air for STOL
22.   Fig. 18-14 shows one method how STOL can be achieved with a form of 'flap blowing'. The turbofan engine has a geared variable pitch fan and an oversized low pressure (L. P.) compressor from the exit of which air is bled and ducted to the flap system in the wing trailing edge. The variable pitch fan enables high L.P. compressor speed and thus high bleed pressure to be maintained over a wide range of thrusts. This gives excellent control at greatly different aircraft flight conditions.

LIFT THRUST AUGMENTATION

23.   In many cases on V/STOL aircraft augmentation of the lift thrust is necessary to avoid an engine which is oversized for normal flight with the consequent effects of higher engine weight and fuel consumption than would be the case for a conventional aircraft- This lift thrust augmentation can be achieved in a number of different ways:

(1)   Using special engine ratings.

(2)   Burning in the lift nozzle gas flow.

(3)   By means of an ejector system.

Special engine ratings
24.   Experience has shown that an engine rating structure can be devised which provides high thrust levels for short periods of time without reducing engine life. Operation in ground effect and the take-off and landing manoeuvres require maximum thrust for less than 15 seconds so that use of a short lift rating for that time is feasible. Fig. 18-15 shows an example of thrust permissible with a 15 second short lift rating compared to that with a 2.5 minute normal lift rating.
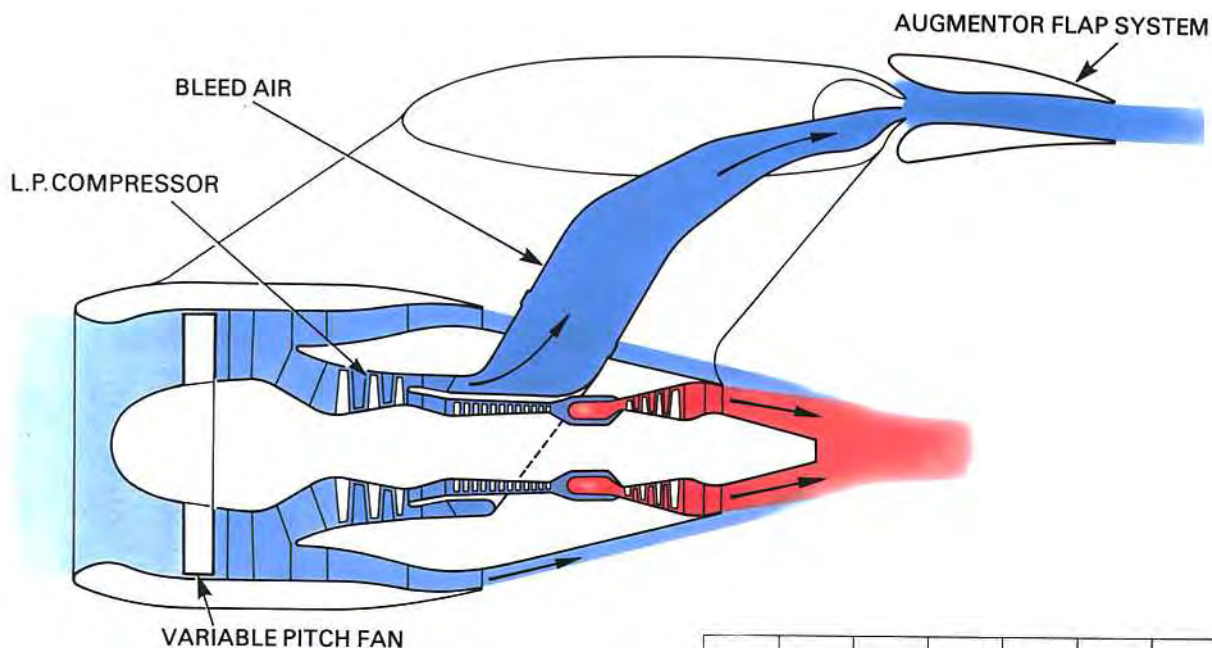
Fig. 18-14   Flap blowing engine.

25.   At high ambient temperatures, the engine may run into a turbine temperature limit before reaching its maximum r.p.m. and suffer a thrust loss as a result. Restoration of the thrust can be achieved by means of water injection into the combustion chamber (Part 17) which allows operation at a higher turbine gas temperature for a given turbine blade temperature. If desired, water injection can also be used to increase the thrust at low ambient temperatures.

Lift burning systems

26.   The thrust of the four nozzle lift/propulsion engine may be boosted by burning fuel in the bypass flow in the duct or plenum chamber supplying the front nozzles. This is called plenum chamber burning (P.C.B.) (fig. 18-16) and thrust of the by-pass air may be doubled by this process. This thrust capability is available for normal flight as well as take-off and landing and so can be used to increase manoeuvrability and give supersonic flight.

27.   The thrust of a remote lift jet can also be augmented by burning fuel in a combustion chamber just upstream of the lift nozzle (fig. 18-17). This system is commonly known as a remote augmented lift system (R.A.L.3.). The thrust boost available from the burner reduces the amount of airflow to be supplied to it and therefore reduces the size of the ducting needed to direct the air from the engine to the remote lift nozzle.
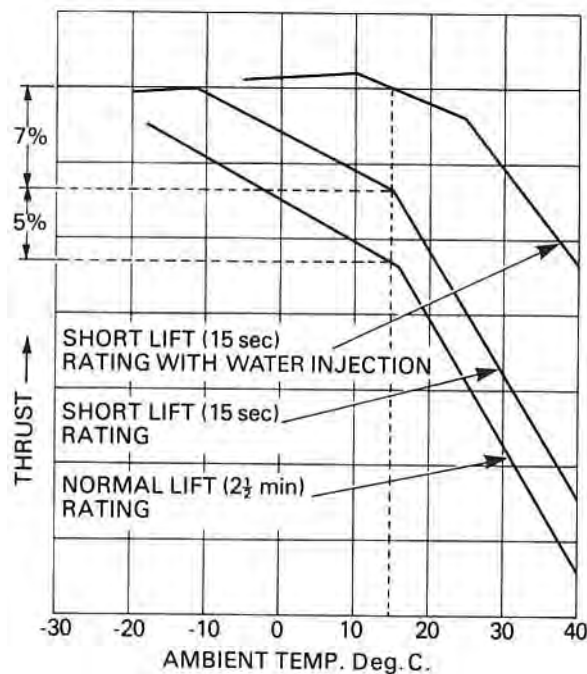
Fig. 18-15   Thrust increases with short lift ratings.

Ejectors

28.   The principle of the ejector is that a small, high energy jet entrains large quantities of ambient air by viscous mixing and an increase in thrust over that of the high energy jet results. A number of projected V/STOL aircraft have incorporated this concept using either all the engine exhaust air or just the bypass flow.
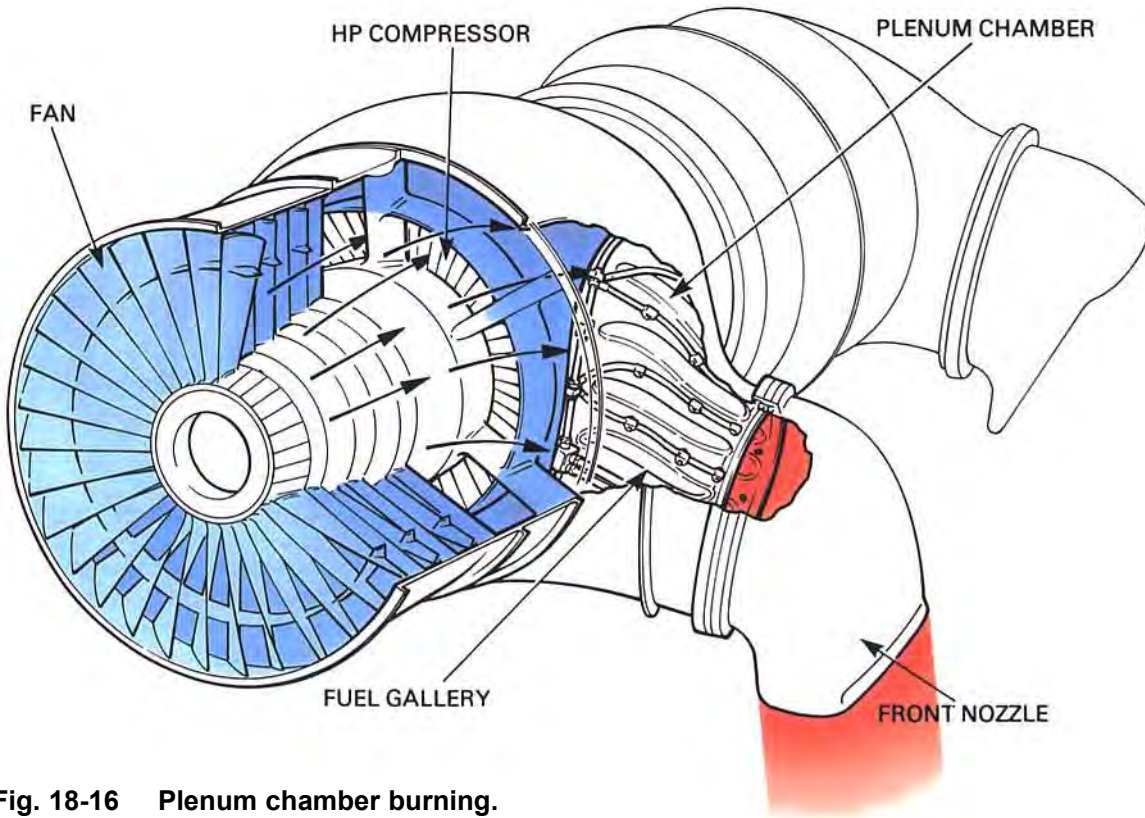
Fig. 18-16   Plenum chamber burning.
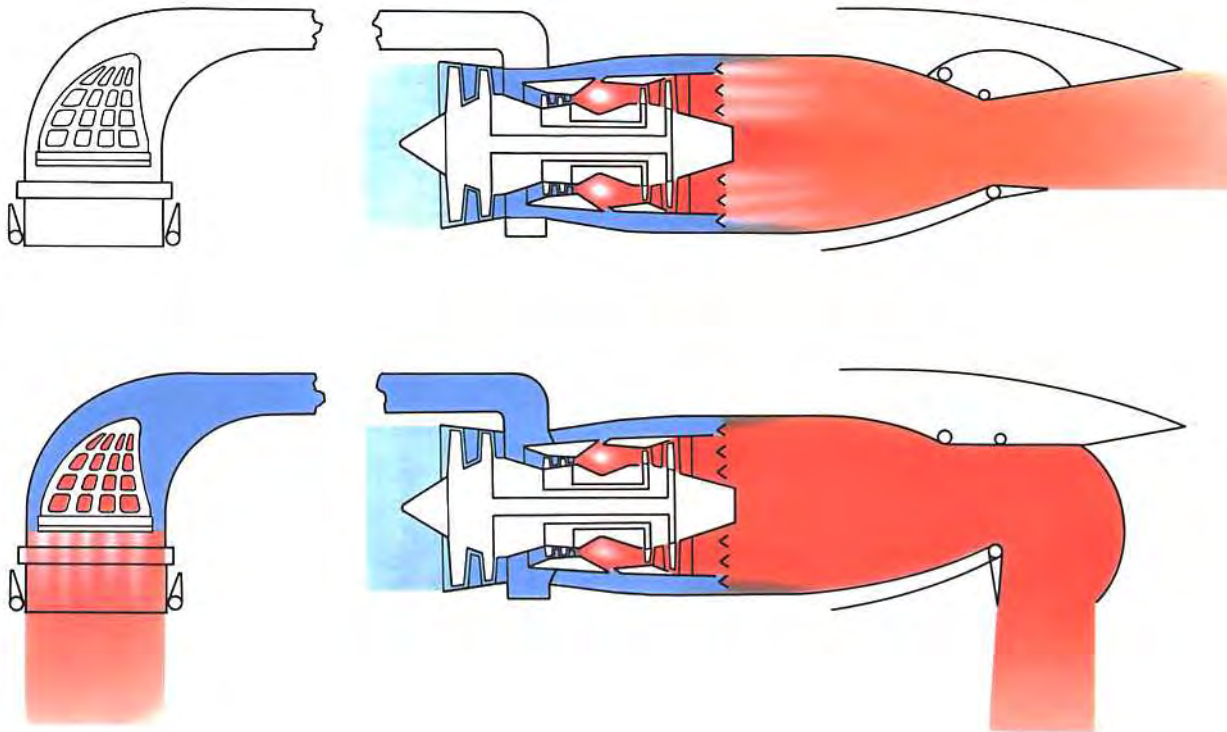
Fig. 18-17   Remote augmented lift system.
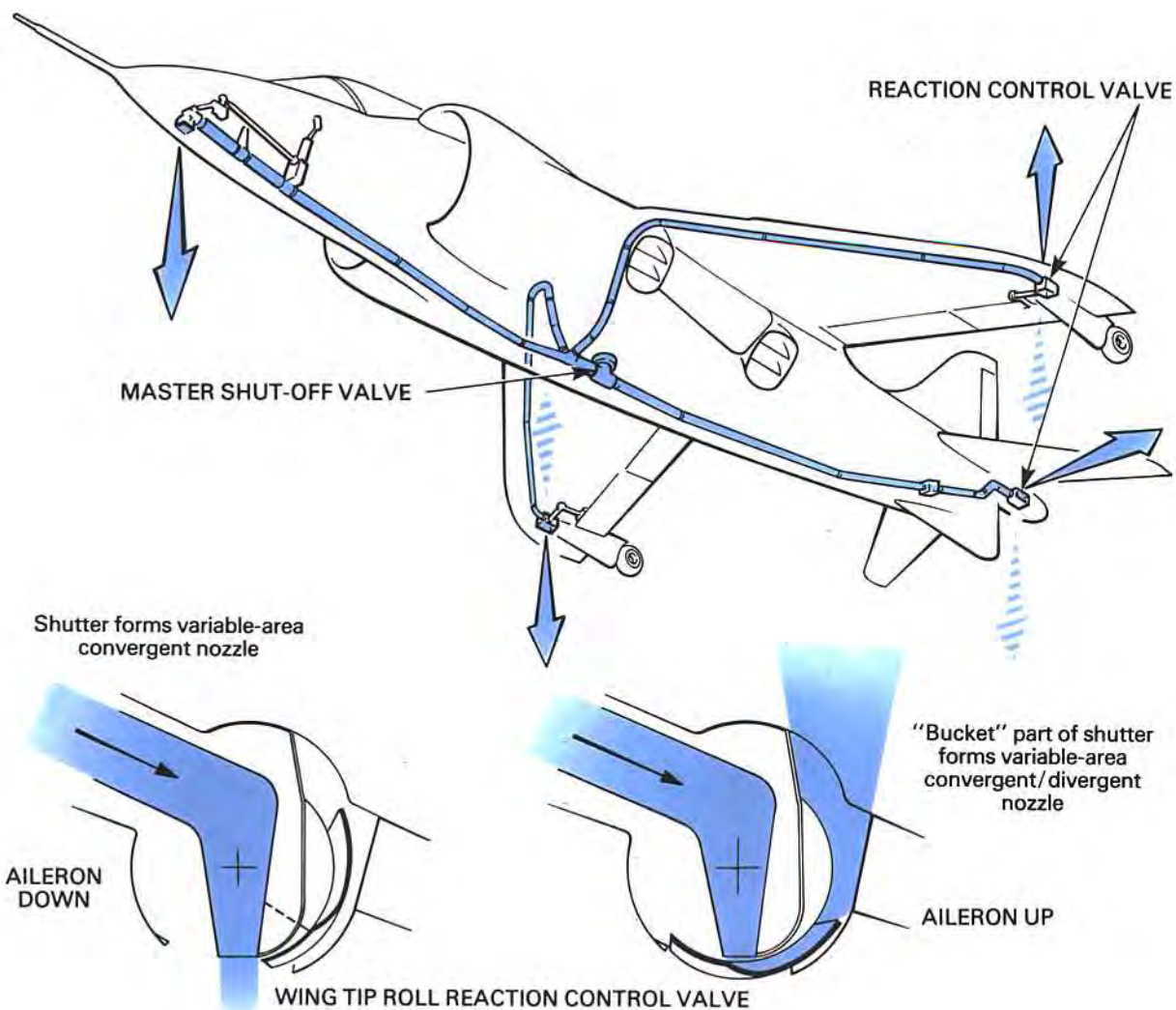
Fig. 18-18   Reaction control system.

AIRCRAFT CONTROL

29. The low forward speeds of V/STOL aircraft during take-off and transition do not permit the generation of adequate aerodynamic forces from the normal flight control surfaces, it is therefore necessary to provide one or more of the following additonal methods of controlling pitch, roll and yaw.

Reaction controls

30. This system bleeds air from the engine and ducts it through nozzles at the four extremities of the aircraft (fig. 18-18), The air supply to the nozzles is automatically cut off when the main engine swivelling propulsion nozzles are turned for normal flight or when the lift engines are shut down. The thrust of the control nozzles is varied by changing their area which varies the amount of airflow passed.

Differential engine throttling

31. This method of control is used on multi-engined aircraft with the engines positioned in a suitable configuration. A rapid response rate is essential to enable the engines to be used for aircraft stability and control. It is usually necessary to combine differential throttling with differential thrust vectoring to give aircraft control in all areas.

Automatic control systems

32. Although it is possible for the pilot to control a V/STOL aircraft manually, some form of automation can be of benefit and in particular will reduce the pilot workload. The pilot's control column is electronically connected to a computer or stabilizer that receives signals from the control column, compares them with signals from the sensors that measure the attitude of the aircraft, and automatically adjusts the reaction controls, differential throttling or thrust vectoring controls to maintain stability.

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Rolls-Royce Turbomeca Adour MK151
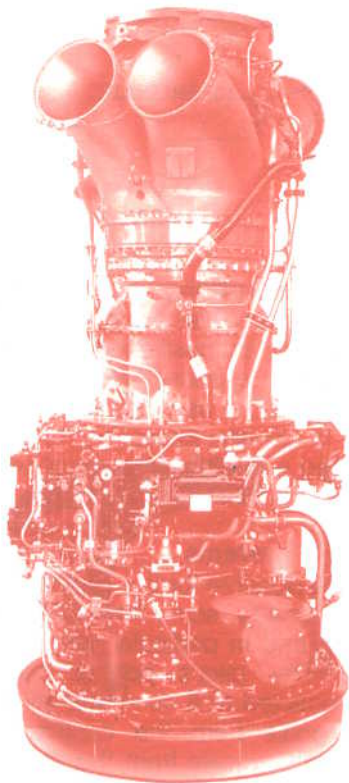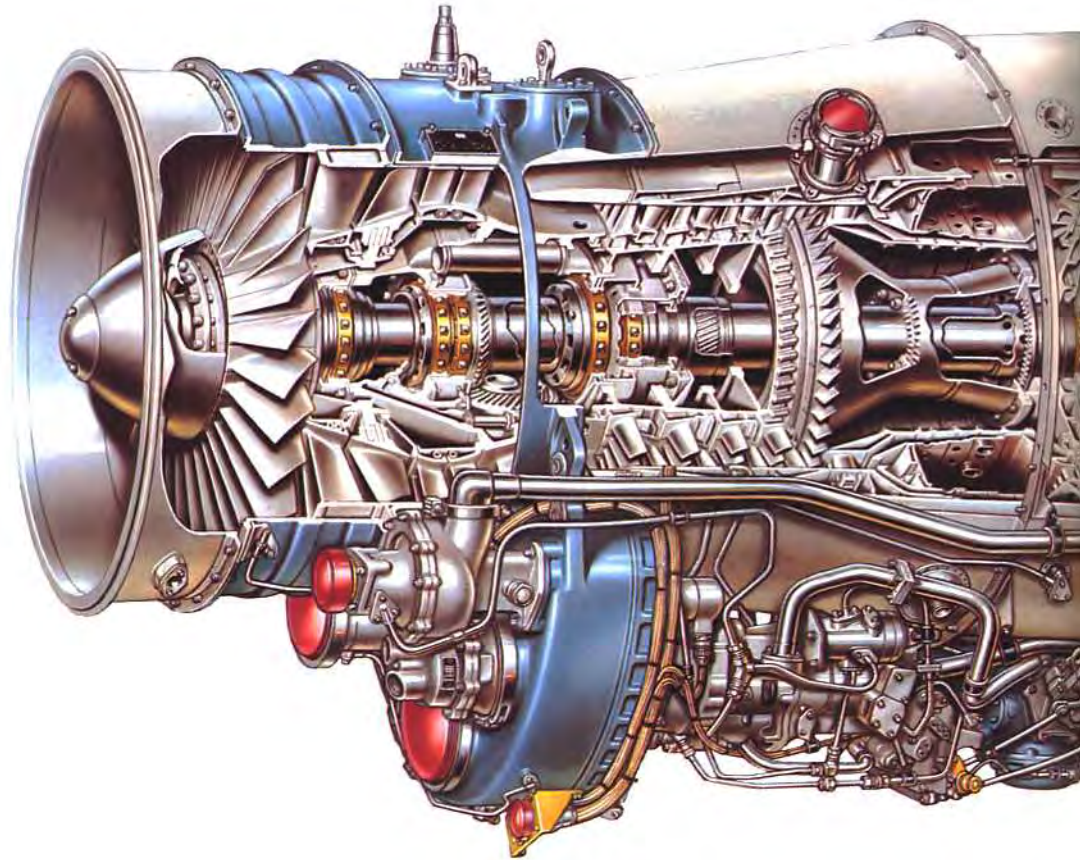
Napier Gazelle

The Gazelle turbo-shaft engine first ran in December 1955 at 1260 shp, a figure later increased to 1610 shp on production engines. Gazelles were used to power Bristol Belvedere and Westland Wessex helicopters. Gazelle production was taken over by Rolls-Royce in 1961.

19: Noise suppression

Contents

3. Airframe self-generated noise is a factor in an aircraft's overall noise signature, but the principal noise source is the engine.

INTRODUCTION

1. Airport regulations and aircraft noise certification requirements, all of which govern the maximum noise level aircraft are permitted to produce, have made jet engine noise suppression one of the most important fields of research.

2. The unit that is commonly used to express noise annoyance is the Effective Perceived Noise deciBel (EPNdB). It takes into account the pitch as well as the sound pressure (deciBel) and makes allowance for the duration of an aircraft flyover. Fig. 19-1 compares the noise levels of various jet engine types.

ENGINE NOISE

4. To understand the problem of engine noise suppression, it is necessary to have a working knowledge of the noise sources and their relative importance. The significant sources originate in the fan or compressor, the turbine and the exhaust jet or jets. These noise sources obey different laws and mechanisms of generation, but all increase, to a varying degree, with greater relative airflow velocity. Exhaust jet noise varies by a larger factor than the compressor or turbine noise, therefore a reduction in exhaust jet velocity has a stronger influence than an equivalent reduction in compressor and turbine blade speeds.
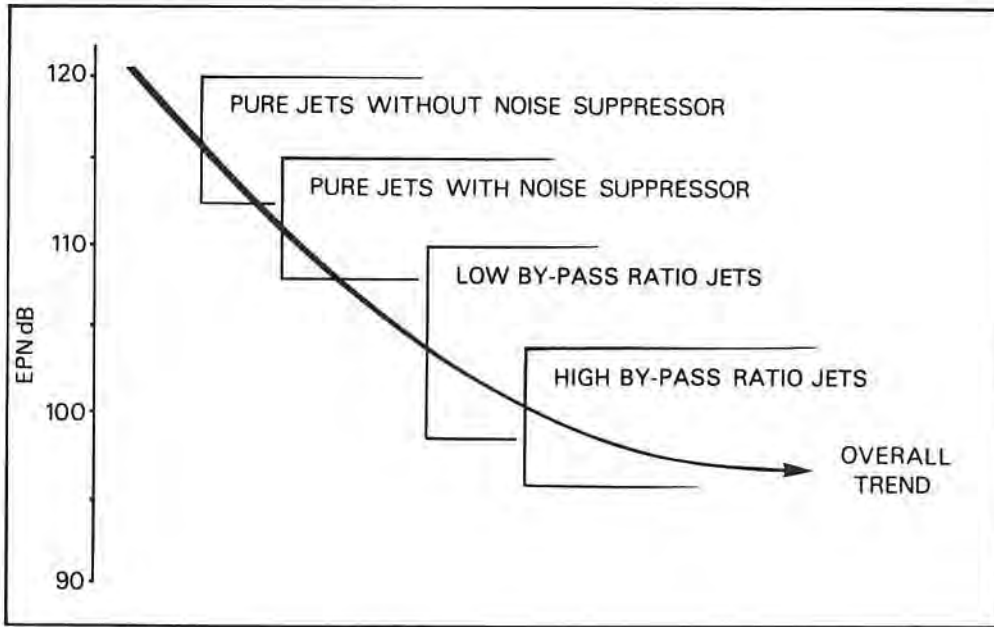
Fig. 19-1   Comparative noise levels of various engine types.
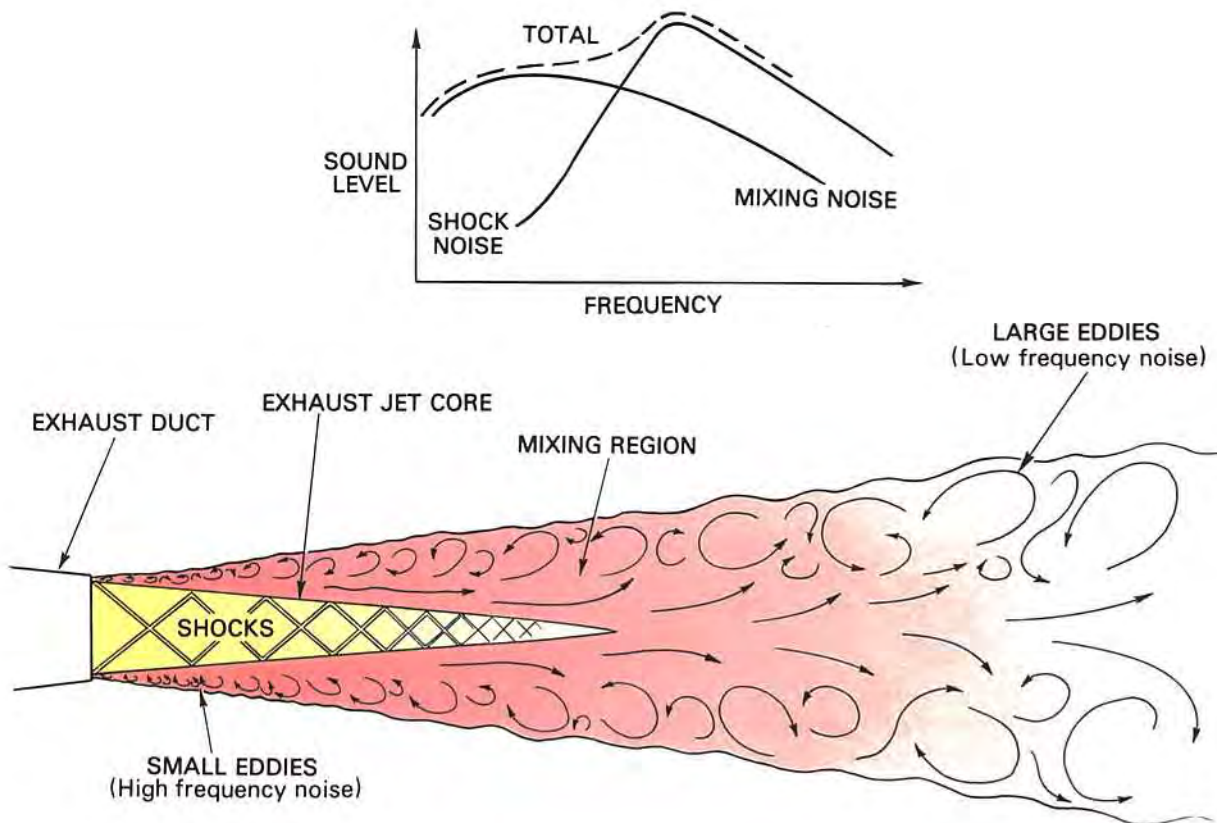
Fig. 19-2   Exhaust mixing and shock structure.

5. Jet exhaust noise is caused by the violent and hence extremely turbulent mixing of the exhaust gases with the atmosphere and is influenced by the shearing action caused by the relative speed between the exhaust jet and the atmosphere. The small eddies created near the exhaust duct cause high frequency noise but downstream of the exhaust jet the larger eddies create low frequency noise. Additionally, when the exhaust jet velocity exceeds the local speed of sound, a regular shock pattern is formed within the exhaust jet core. This produces a discrete (single frequency) tone and selective amplification of the mixing noise, as shown in fig. 19-2. A reduction in noise level occurs if the mixing rate is accelerated or if the velocity of the exhaust jet relative to the atmosphere is reduced. This can be achieved by changing the pattern of the exhaust jet as shown in fig. 19-3.

6. Compressor and turbine noise results from the interaction of pressure fields and turbulent wakes from rotating blades and stationary vanes, and can be defined as two distinct types of noise; discrete tone (single frequency) and broadband (a wide range of frequencies). Discrete tones are produced by the regular passage of blade wakes over the stages downstream causing a series of tones and harmonics from each stage. The wake intensity is largely dependent upon the distance between the rows of blades and vanes. If the distance is short then there is an intense pressure field interaction which results in a strong tone being generated. With the high bypass engine, the low pressure compressor (fan) blade wakes passing over downstream vanes produce such tones, but of a lower intensity due to lower velocities and larger blade/vane separations. Broadband noise is

PLAIN NOZZLE (low mixing rate) HIGH NOISE LEVEL

SUPPRESSOR NOZZLE (high mixing rate) REDUCED NOISE LEVEL
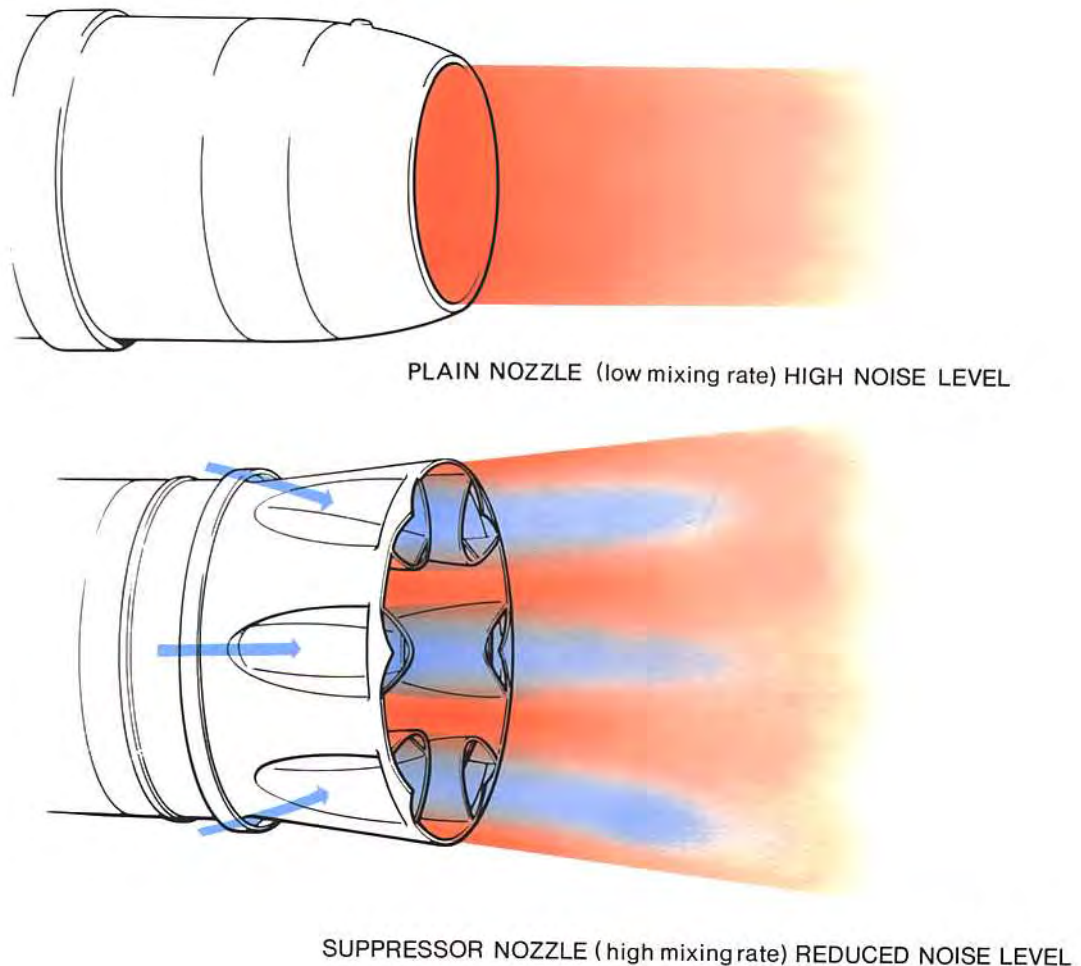
Fig. 19-3   Change of exhaust jet pattern to reduce noise level.

produced by the reaction of each blade to the passage of air over its surface, even with a smooth airstream. Turbulence in the airstream passing over the blades increases the intensity of the broadband noise and can also induce tones.

7. With the pure jet engine the exhaust jet noise is of such a high level that the turbine and compressor noise is insignificant at all operating conditions, except low landing-approach thrusts. With the by-pass principle, the exhaust jet noise drops as the velocity of the exhaust is reduced but the low pressure compressor and turbine noise increases due to the greater internal power handling.

8. The introduction of a single stage low pressure compressor (fan) significantly reduces the compressor noise because the overall turbulence and interaction levels are diminished. When the by-pass ratio is in excess of approximately 5 to 1, the jet exhaust noise has reduced to such a level that the increased internal noise source is predominant. A comparison between low and high by-pass engine noise sources is shown in fig. 19-4.

9. Listed amongst the several other sources of noise within the engine is the combustion chamber. It is a significant but not a predominant source, due in part to the fact that it is 'buried' in the core of the engine. Nevertheless it contributes to the broadband noise, as a result of the violent activities which occur within the combustion chamber.

METHODS OF SUPPRESSING NOISE

10. Noise suppression of internal sources is approached in two ways; by basic design to minimize noise originating within or propagating from the engine, and by the use of acoustically absorbent linings. Noise can be minimized by reducing airflow disruption which causes turbulence. This is achieved by using minimal rotational and airflow velocities and reducing the wake intensity by appropriate spacing between the blades and vanes. The ratio between the number of rotating blades and stationary vanes can also be advantageously employed to contain noise within the engine.
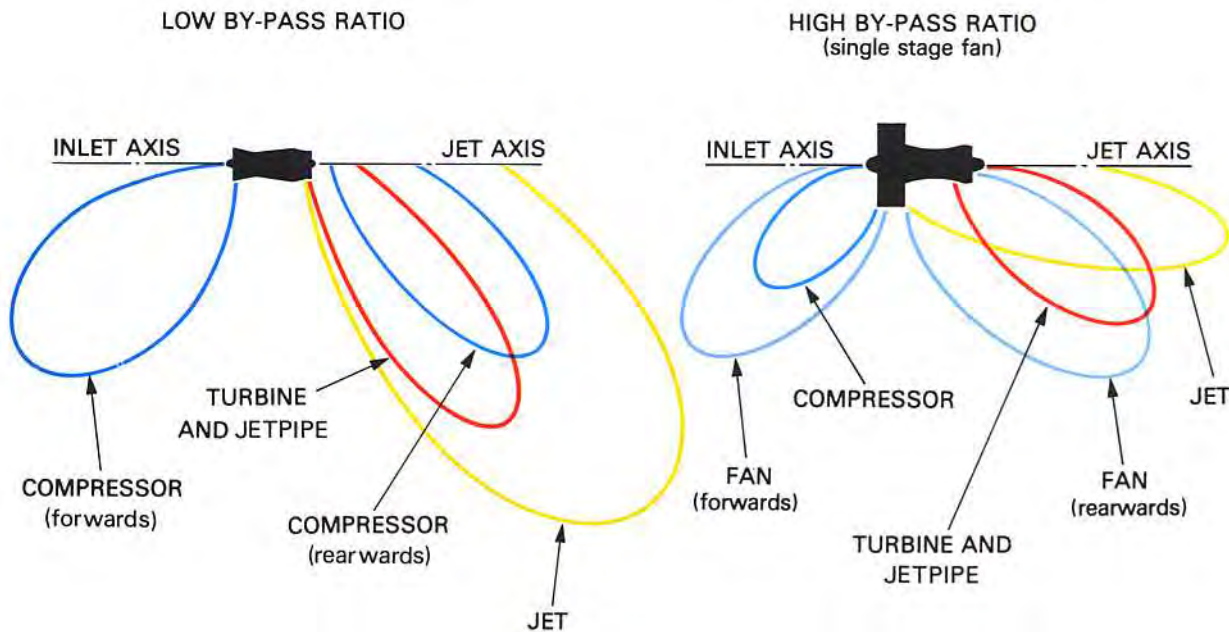
Fig. 19-4   Comparative noise sources of low and high by-pass engines.

11. As previously described, the major source of noise on the pure jet engine and low by-pass engine is the exhaust jet, and this can be reduced by inducing a rapid or shorter mixing region. This reduces the low frequency noise but may increase the high frequency level. Fortunately, high frequencies are quickly absorbed in the atmosphere and some of the noise which does propagate to the listener is beyond the audible range, thus giving the perception of a quieter engine. This is achieved by increasing the contact area of the atmosphere with the exhaust gas stream by using a propelling nozzle incorporating a corrugated or lobe-type noise suppressor (fig. 19-5).

12. In the corrugated nozzle, freestream atmospheric air flows down the outside corrugations and into the exhaust jet to promote rapid mixing. In the lobe-type nozzle, the exhaust gases are divided to flow through the lobes and a small central nozzle. This forms a number of separate exhaust jets that rapidly mix with the air entrained by the suppressor lobes. This principle can be extended by the use of a series of tubes to give the same overall area as the basic circular nozzle.

13. Deep corrugations, lobes, or multi-tubes, give the largest noise reductions, but the performance penalties incurred limit the depth of the corrugations or lobes and the number of tubes. For instance, to achieve the required nozzle area, the overall diameter of the suppressor may have to be increased by so much that excessive drag and weight results. A compromise which gives a noticeable reduction in noise level with the least sacrifice of engine thrust, fuel consumption or addition of weight is therefore the designer's aim.

14. The high by-pass engine has two exhaust streams to eject to atmosphere. However, the principle of jet exhaust noise reduction is the same as for the pure or low by-pass engine, i.e. minimize the exhaust jet velocity within overall performance objectives. High by-pass engines inherently have a lower exhaust jet velocity than any other type of gas turbine, thus leading to a quieter engine, but further noise reduction is often desirable. The most successful method used on by-pass engines is to mix the hot and cold exhaust streams within the confines of the engine (fig. 19-5) and expel the lower velocity exhaust gas flow through a single nozzle (Part 6).

15. In the high by-pass ratio engine the predominant sources governing the overall noise level are the fan and turbine. Research has produced
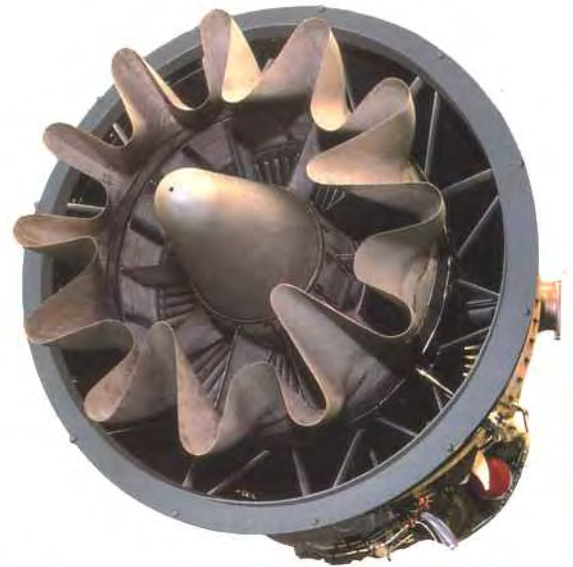
CORRUGATED INTERNAL MIXER
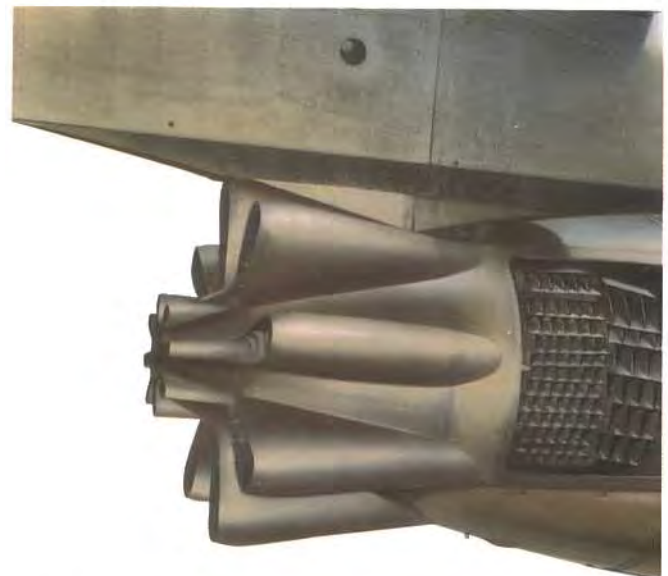
LOBE-TYPE NOZZLE

Fig. 19-5   Types of noise suppressor.

a good understanding of the mechanisms of noise generation and comprehensive noise design rules exist. As previously indicated, these are founded on the need to minimize turbulence levels in the airflow, reduce the strength of interactions between rotating blades and stationary vanes, and the optimum use of acoustically absorbent linings.

PERFORATE FACESHEET

TYPICAL
PERFORATE LINER
(Titanium or
aluminium or
composite)

HONEYCOMB SUPPORT

SOLID BACKING SHEET

CLOSE WOVEN
WIRE CLOTH

LINEAR LINER
(Stainless steel
and aluminium)

DOUBLE
PERFORATE LAYER
(Aluminium)
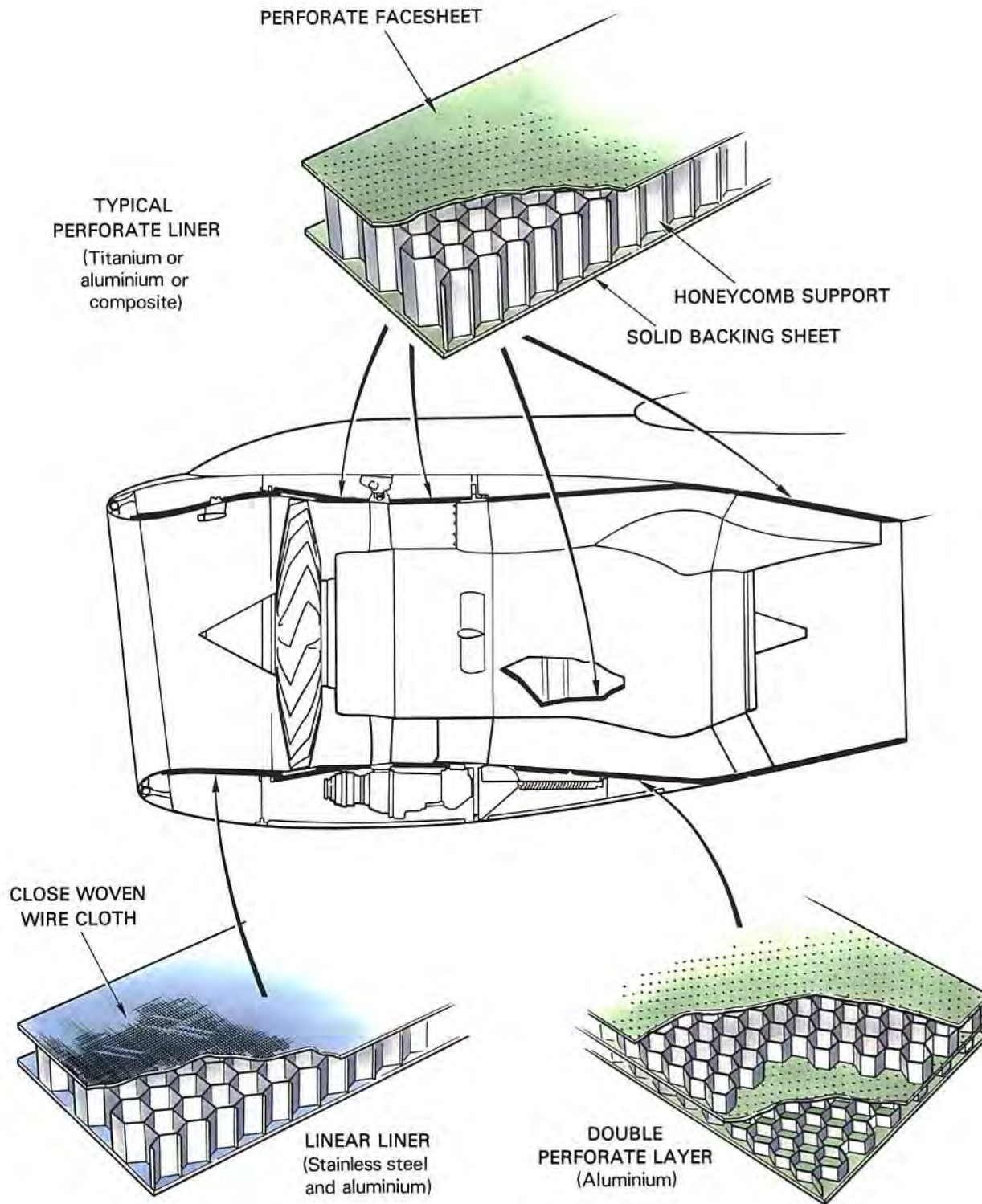
Fig. 19-6   Noise absorbing materials and location.

16. Noise absorbing 'lining' material converts acoustic energy into heat. The absorbent linings (fig. 19-6) normally consist of a porous skin supported by a honeycomb backing, to provide the required separation between the facesheet and the solid engine duct. The acoustic properties of the skin and the liner depth are carefully matched to the character of the noise, for optimum suppression. The disadvantage of liners is the slight increase in weight and skin friction and hence a slight increase in fuel consumption. They do however, provide a very powerful suppression technique.

CONSTRUCTION AND MATERIALS

17. The corrugated or lobe-type noise suppressor forms the exhaust propelling nozzle and is usually a separate assembly bolted to the jet pipe. Provision is usually made to adjust the nozzle area so that it can be accurately calibrated. Guide vanes are fitted to the lobe-type suppressor to prevent excessive losses by guiding the exhaust gas smoothly through the lobes to atmosphere. The suppressor is a fabricated welded structure and is manufactured from heat-resistant alloys.

18. Various noise absorbing lining materials are used on jet engines. They fall mainly within two categories, lightweight composite materials that are used in the lower temperature regions and fibrous-metallic materials that are used in the higher temperature regions. The noise absorbing material consists of a perforate metal or composite facing skin, supported by a honeycomb structure on a solid backing skin which is bonded to the parent metal of the duct or casing. For details of manufacture of these materials refer to Part 22.

Rolls-Royce Conway
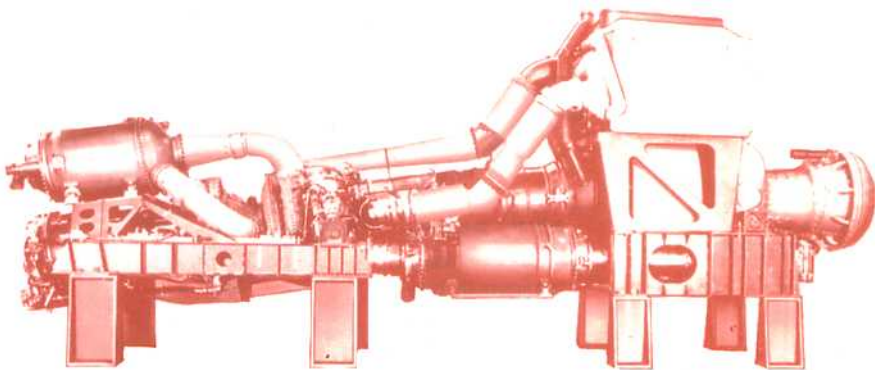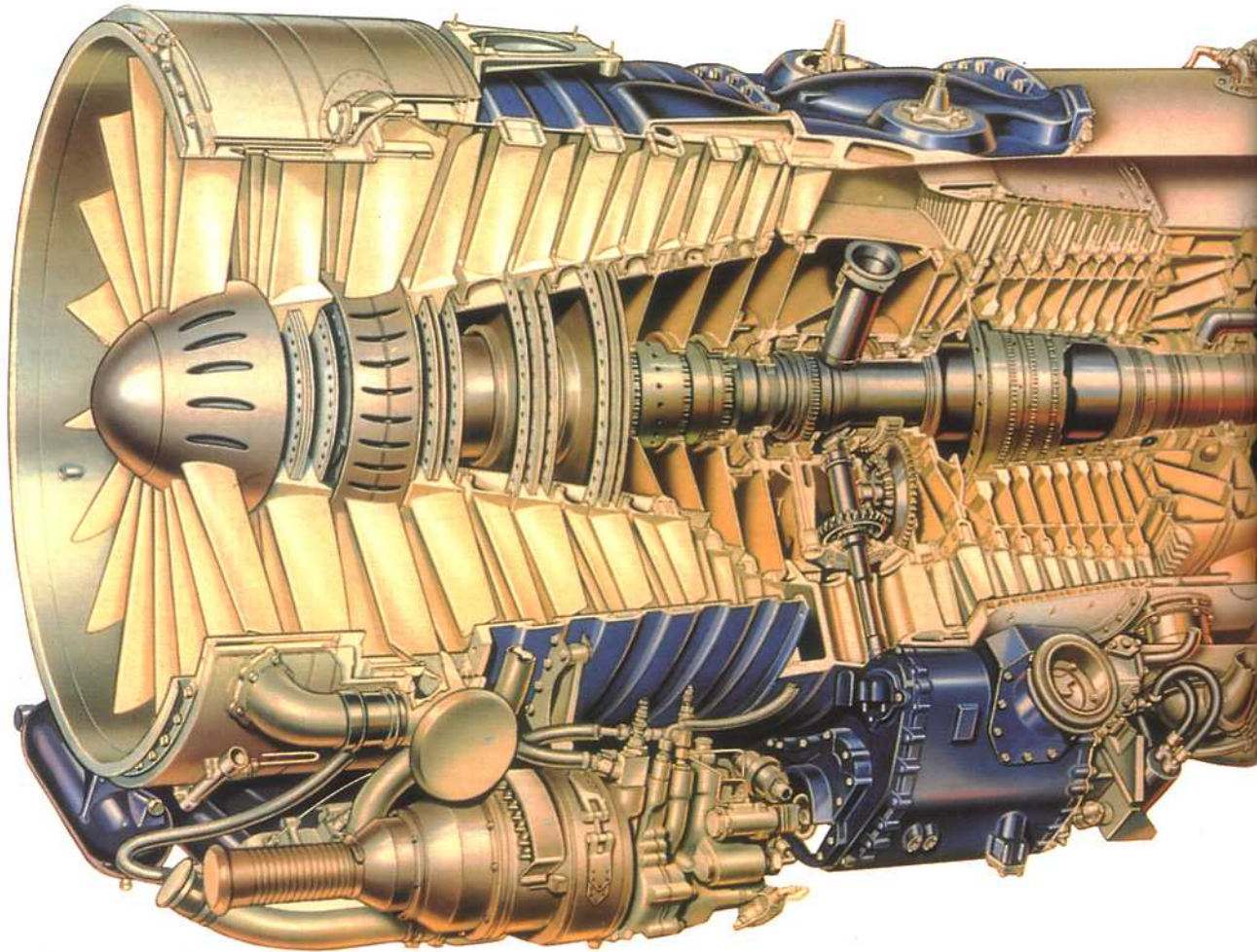
Rolls-Royce RM60

Produced in response to an Admiralty contract for a coastal-craft engine with good cruising economy, the RM60, although based on aeroengine philosophy, was designed from the first as a marine gas turbine. Two RM60s went to sea in 1953 in the former steam gunboat HMS Grey Goose, the world's first warship to be powered solely by gas turbines.

20: Thrust distribution

Contents Page

INTRODUCTION

1.   Although the principles of jet propulsion (see Part 1) will be familiar to the reader, the distribution of the thrust forces within the engine may appear somewhat obscure- These forces are in effect gas loads resulting from the pressure and momentum changes of the gas stream reacting on the engine structure and on the rotating components. They are in some locations forward propelling forces and in others opposing or rearward forces. The amount that the sum of the forward forces exceeds the sum of the rearward forces is normally known as the rated thrust of the engine.

DISTRIBUTION OF THE THRUST FORCES

2.   The diagram in fig. 20-1 is of a typical single-spool axial flow turbo-jet engine and illustrates where the main forward and rearward forces act. The origin of these forces is explained by following the engine working cycle shown in Part 2.

FORWARD GAS LOAD 57,836lb. **REARWARD GAS LOAD 46,678lb.**
TOTAL THRUST 11,158lb.

19,049lb.

2,186lb. 34,182lb. 41091lb. 5,587lb.

2,419lb.

PROPELLING NOZZLE

COMPRESSOR   DIFFUSER   TURBINE   EXHAUST UNIT AND JET PIPE

COMBUSTION CHAMBER
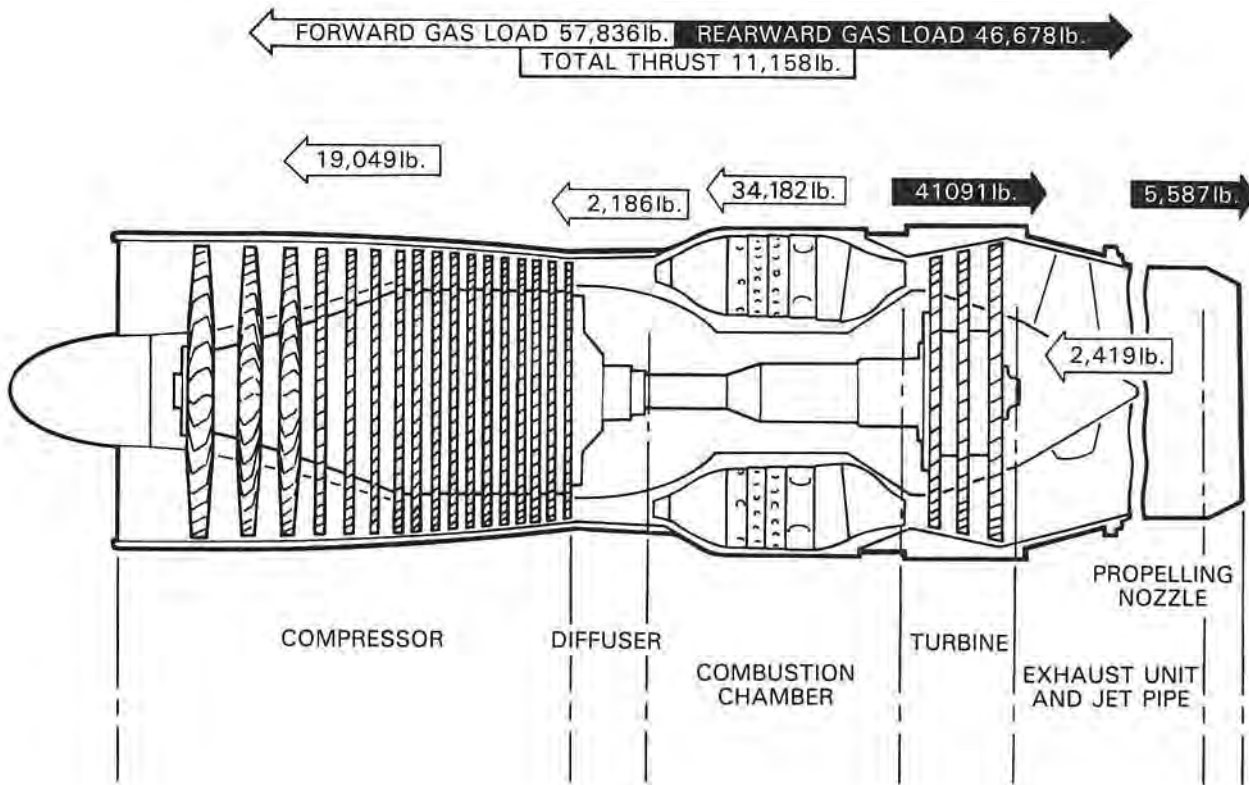
Fig. 20-1   Thrust distribution of a typical single-spool axial flow engine.

3. At the start of the cycle, air is induced into the engine and is compressed. The rearward accelerations through the compressor stages and the resultant pressure rise produces a large reactive force in a forward direction. On the next stage of its journey the air passes through the diffuser where it exerts a small reactive force, also in a forward direction,

4. From the diffuser the air passes into the combustion chambers (Part 4) where it is heated, and in the consequent expansion and acceleration of the gas large forward forces are exerted on the chamber walls.

5. When the expanding gases leave the combustion chambers and flow through the nozzle guide vanes they are accelerated and deflected on to the blades of the turbine (Part 5). Due to the acceleration and deflection, together with the subsequent straightening of the gas flow as it enters the jet pipe, considerable 'drag' results; thus the vanes and blades are subjected to large rearward forces, the magnitude of

which may be seen on the diagram. As the gas flow passes through the exhaust system (Part 6), small forward forces may act on the inner cone or bullet, but generally only rearward forces are produced and these are due to the 'drag' of the gas flow at the propelling nozzle.

6. It will be seen that during the passage of the air through the engine, changes in its velocity and pressure occur (Part 2). For instance, where a conversion from velocity (kinetic) energy to pressure energy is required the passages are divergent in shape, similar to that used in the compressor diffuser. Conversely, where it is required to convert the energy stored in the combustion gases to velocity, a convergent passage or nozzle, similar to that used in the turbine, is employed. Where the conversion is to velocity energy, 'drag' loads or rearward forces are produced; where the conversion is to pressure energy, forward forces are produced. Part 2, fig. 2-3 illustrates velocity and pressure changes at two points on the engine.

METHOD OF CALCULATING THE THRUST FORCES

7. The thrust forces or gas loads can be calculated for the engine, or for any flow section of the engine, provided that the areas, pressures, velocities and mass flow are known for both the inlet and outlet of the particular flow section.

8. The distribution of thrust forces shown in fig. 20-1 can be calculated by considering each component in turn and applying some simple calculations. The thrust produced by the engine is mainly the product of the mass of air passing through the engine and the velocity increase imparted to it (i.e. Newtons Second Law of Motion), however, the pressure difference between the inlet to and the outlet from the particular flow section will have an effect on the overall thrust of the engine and must be included in the calculation.

9. To calculate the resultant thrust for a particular flow section it is necessary to calculate the total thrust at both inlet and outlet, the resultant thrust being the difference between the two values obtained.

10. Calculation of the thrust is achieved using the following formula:

$$\text{Thrust} = (A \times P) + \frac{Mv_J}{g}$$

Where A = Area of flow section in sq.in.
P = Pressure in lb. per sq.in.
W = Mass flow in lb. per sec.
v_J = Velocity of flow in feet per sec.
g = Gravitational constant 32.2 ft. per sec. per sec.

CALCULATING THE THRUST OF THE ENGINE

11. When applying the above method to calculate the individual thrust loads on the various components it is assumed that the engine is static. The effect of aircraft forward speed on the engine thrust will be dealt with in Part 21. In the following calculations 'g' is taken to be 32 for convenience. To assist in these calculations the locations concerned are illustrated by a number of small diagrams.

Compressor casing
12. To obtain the thrust on the compressor casing it is necessary to calculate the conditions at the inlet to

the compressor and the conditions at the outlet from the compressor. Since the pressure and the velocity at the inlet to the compressor are zero, it is only necessary to consider the force at the outlet from the compressor. Therefore, given that the compressor-

OUTLET Area (A) = 182 sq.in.
Pressure (P) = 94 lb. per sq.in. (gauge)
Velocity (v_J) = 406 ft. per sec.
Mass flow (W) = 153 lb. per sec.

The thrust

$$= (A \times P) + \frac{Mv_J}{g} - 0$$

$$= (182 \times 94) + \frac{153 \times 406}{32} - 0$$

= 19,049 lb. of thrust in a forward direction.
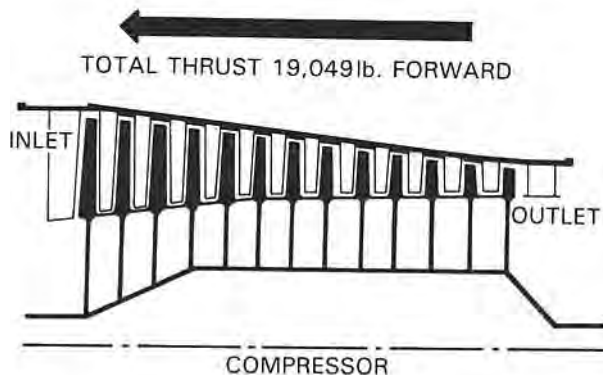
Diffuser duct
13. The conditions at the diffuser duct inlet are the same as the conditions at the compressor outlet, i.e. 19,049 lb.
Therefore, given that the diffuser--
OUTLET Area (A) = 205 sq.in.
Pressure (P) = 95 lb. per sq.in. (gauge)
Velocity (v_J) = 368 ft. per sec.
Mass flow (W) = 153 lb. per sec.

The thrust

$$= (A \times P) + \frac{Wv_J}{g} - 19,049$$

$$= (205 \times 95) + \frac{153 \times 368}{32} - 19,049$$

= 21,235 - 19,049

= 2,186 lb. of thrust in a forward direction.

TOTAL THRUST 2,186 lb. FORWARD

INLET | OUTLET

DIFFUSER

Combustion chambers

14. The conditions at the combustion chamber inlet are the same as the conditions at the diffuser outlet, i.e. 21,235 lb. Therefore, given that the combustion chamber-

OUTLET Area   (A) = 580 sq.in.
        Pressure (P) = 93 lb. per sq.in. (gauge)
        Velocity (v_J) = 309 ft. per sec.
        Mass flow (W) = 153 lb. per sec.

The thrust

$$= (A \quad x \quad P) + \frac{W v_J}{g} - 21{,}235$$

$$= (580 \quad x \quad 93) + \frac{153 \quad x \quad 309}{32} - 21{,}235$$

$$= 55{,}417 - 21{,}235$$

$$= 34{,}182 \text{ lb. of thrust in a forward direction.}$$

TOTAL THRUST 34,182 lb. FORWARD

INLET | OUTLET

COMBUSTION CHAMBER
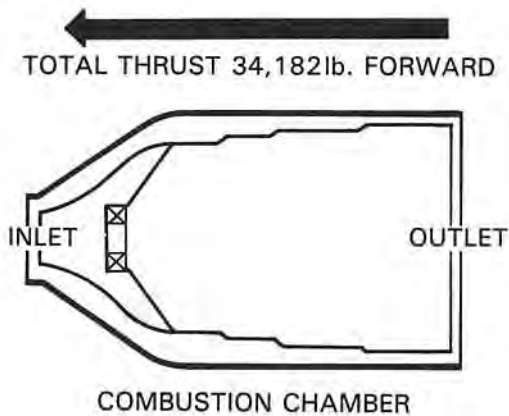
Turbine assembly

15. The conditions at the turbine inlet are the same as the conditions at the combustion chamber outlet, i.e. 55,417 lb.

Therefore given that the turbine--

OUTLET Area   (A) = 480 sq.in.
        Pressure (P) = 21 lb. per sq.in. (gauge)
        Velocity (v_J) = 888 ft. per sec.
        Mass flow (W) = 153 lb. per sec.

The thrust

$$= (A \quad x \quad P) + \frac{W v_J}{g} - 55{,}417$$

$$= (480 \quad x \quad 21) + \frac{153 \quad x \quad 888}{32} - 55{,}417$$

$$= 14{,}326 - 55{,}417$$

$$= -41{,}091$$

This negative value means a force acting in a rearward direction.

TOTAL THRUST 41,091 lb. REARWARD

INLET | OUTLET

NOZZLE GUIDE VANE

TURBINE WHEELS

TURBINE ASSEMBLY
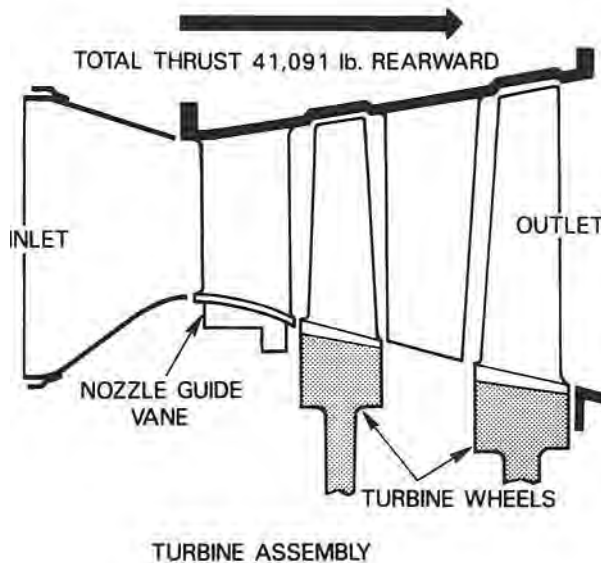
Exhaust unit and jet pipe

16. The conditions at the inlet to the exhaust unit are the same as the conditions at the turbine outlet, i.e. 14,326 lb.

Therefore, given that the jet pipe--

OUTLET Area   (A) = 651 sq.in.
        Pressure (P) = 21 lb. per sq.in. (gauge)
        Velocity (v_J) = 643 ft. per sec.
        Mass flow (W) = 153 lb. per sec.

The thrust

$$= (A \quad x \quad P) + \frac{W_{V_J}}{g} - 14{,}326$$

$$= (651 \quad x \quad 21) + \frac{153 \quad x \quad 643}{32} - 14{,}326$$

$$= 16{,}745 - 14{,}326$$

$$= 2{,}419 \text{ lb. of thrust in a forward direction.}$$

TOTAL THRUST 2,419lb. FORWARD

INLET

OUTLET

EXHAUST UNIT

JET PIPE

EXHAUST UNIT AND JET PIPE
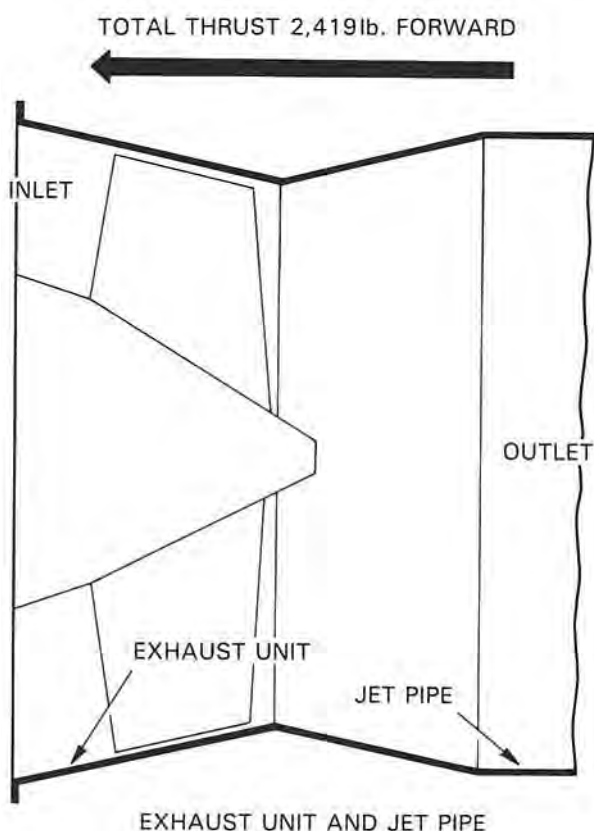
Propelling nozzle

17. The conditions at the inlet to the propelling nozzle are the same as the conditions at the jet pipe outlet, i.e. 16,745 lb.
Therefore, given that the propelling nozzle--

OUTLET   Area   (A)   =   332 sq.in.
            Pressure   (P)   =   6 lb. per sq.in.
                                       (gauge)
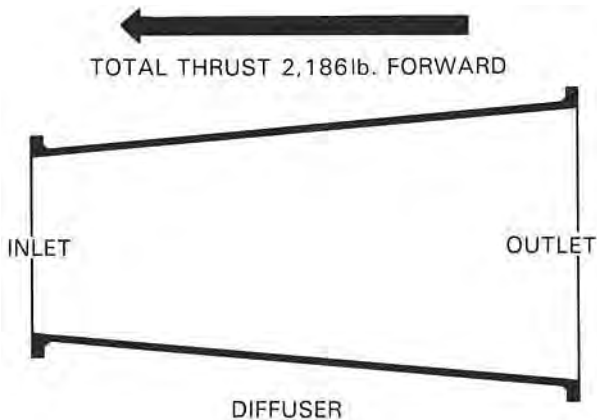   Velocity (v_J) =   1,917 ft. per sec.
            Mass flow   (W)   =   153 lb. per sec.

The thrust

$$= (A \quad x \quad P) + \frac{W_{V_J}}{g} - 16{,}745$$

$$= (332 \quad x \quad 6) + \frac{153 \quad x \quad 1{,}917}{32} - 16{,}745$$

$$= 11{,}158 - 16{,}745$$

$$= 5{,}587 \text{lb. acting in a rearward direction.}$$

TOTAL THRUST 5,587lb. REARWARD

INLET

OUTLET

PROPELLING NOZZLE
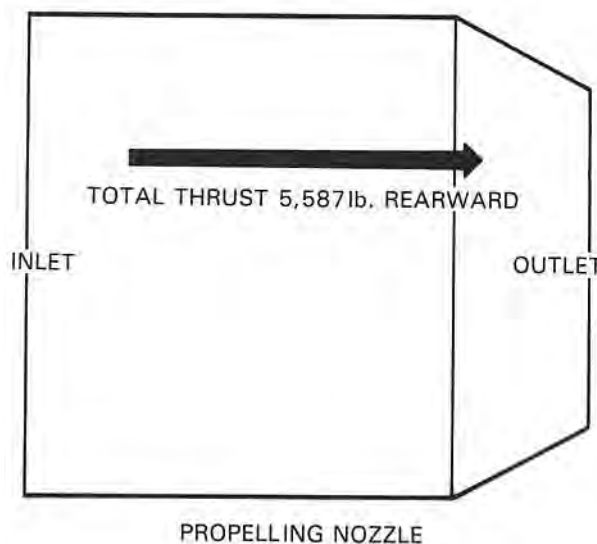
It is emphasized that these are basic calculations and such factors as the effect of air offtakes have been ignored.

18. Based on the individual calculations, the sum of the forward or positive loads is 57,836 lb. and the sum of the rearward or negative loads is 46,678 lb. Thus, the resultant (gross or total) thrust is 11,158 lb.

Engine
19. It will be of interest to calculate the thrust of the engine by considering the engine as a whole, as the resultant thrust should be equal to the sum of the individual gas loads previously calculated.

20. Although the momentum change of the gas stream produces most of the thrust developed by the engine (momentum thrust $= \dfrac{W_{V_J}}{g}$), an additional thrust is produced when the engine operates with the propelling nozzle in a 'choked' condition (Part 6). This thrust results from the aerodynamic forces which are created by the gas stream and exert a pressure

across the exit area of the propelling nozzle (pressure thrust). Algebraically, this force is expressed as $(P-P_0)$ A.

Where A = Area of propelling nozzle in sq.in.
P = Pressure in lb. per sq.in.
P_0 = Atmospheric pressure in lb. per sq.in.

Therefore, assuming values of mass flow, pressure and area to be the same as in the previous calculations i.e.

Area of propelling nozzle (A) = 332 sq.in.
Pressure (P) = 6 lb. per sq.in. (gauge)
Atmospheric Pressure (P) = 0 lb. per sq.in. (gauge)
Mass flow (W) = 153 lb. per sec.
Velocity (v_J) = 1,917 ft. per sec.

The thrust

$$= (P - P_0) \cdot A + \frac{W v_J}{g} - 0$$

$$= (6 - 0) \cdot 332 + \frac{153 \ \times \ 1,917}{32} - 0$$

$$= 1,992 + 9,166$$

$$= 11,158 \text{ lb., the same as previously calculated}$$

by combining the gas loads on the individual engine locations.

21. On engines that operate with a non-choked nozzle, the $(P-P_0)$ A function does not apply and the thrust results only from the gas stream momentum change.

Inclined combustion chambers

22. In the previous example (Para. 14) the flow through the combustion chamber is axial, however, if the combustion chamber is inclined towards the axis of the engine, then the axial thrust will be less than for an axial flow chamber. This thrust can be obtained by multiplying the sum of the outlet thrust by the cosine of the angle (see fig. 20-2). The

$$\text{cosine} = \frac{\text{Base}}{\text{Hypotenuse}} \text{ and for a given angle}$$

is obtained by consulting a table of cosines. It should be emphasized that if the inlet and outlet are at different angles to the engine axis, it is necessary to multiply the inlet and outlet thrusts separately by the cosine of their respective angles.

AFTERBURNING

23. When the engine is fitted with an afterburner (Part 16), the gases passing through the exhaust
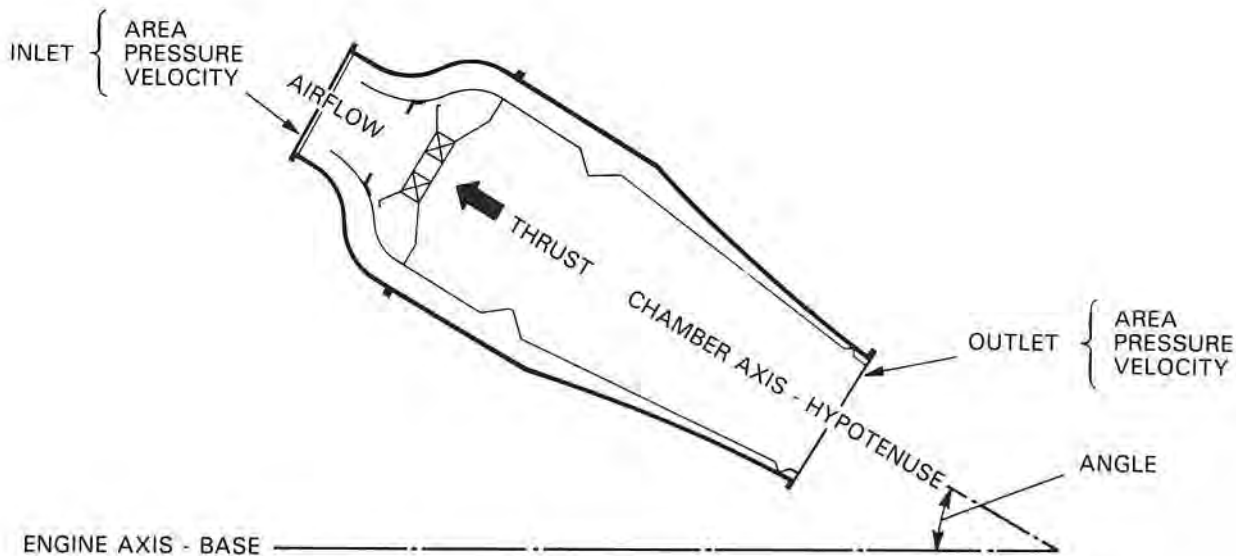
Fig. 20-2   A hypothetical combustion chamber showing values required for calculating thrust.

TOTAL THRUST 2,676lb. REARWARD

INLET

OUTLET

PROPELLING NOZZLE

AFTERBURNER PROPELLING NOZZLE
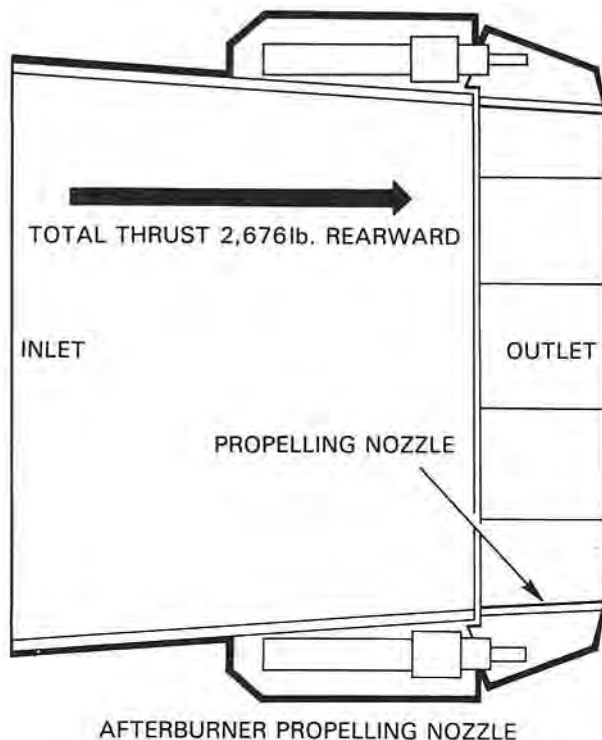
system are reheated to provide additional thrust. The effect of afterburning is to increase the volume of the exhaust gases, thus producing a higher exit velocity at the propelling nozzle.

24. Assuming that an afterburner jet pipe and propelling nozzle are fitted to the engine used in the previous calculations, and the new conditions at the propelling nozzle are as follows-

OUTLET Area   (A) = 455 sq.in.
           Pressure (P) = 5 lb. per sq.in.
                                (gauge)
           Velocity (v_J) = 2,404 ft. per sec.
           Mass flow (W) = 157 lb. per sec.

The thrust

$$= (A \quad x \quad P) + \frac{W v_J}{g} - 16,745$$

$$= (455 \quad x \quad 5) + \frac{157 \quad x \quad 2,404}{32} - 16,745$$

$$= 14,069 - 16,745$$

$$= 2,676 \text{ lb. acting in a rearward direction.}$$

Therefore, compared with the previous calculation in para. 17, it will be seen that the negative thrust is reduced from -5,587 lb. to -2,676 lb.; the overall positive thrust is thus increased by 2,911 lb; which is equivalent to a thrust increase of more than 25 per cent.

25. To arrive at the total thrust of the engine with afterburning the calculations in para. 20 should use the above figures.

Rolls-Royce RB168 MK807
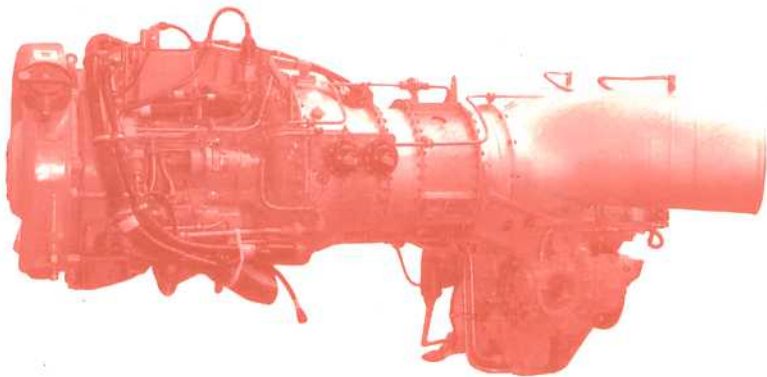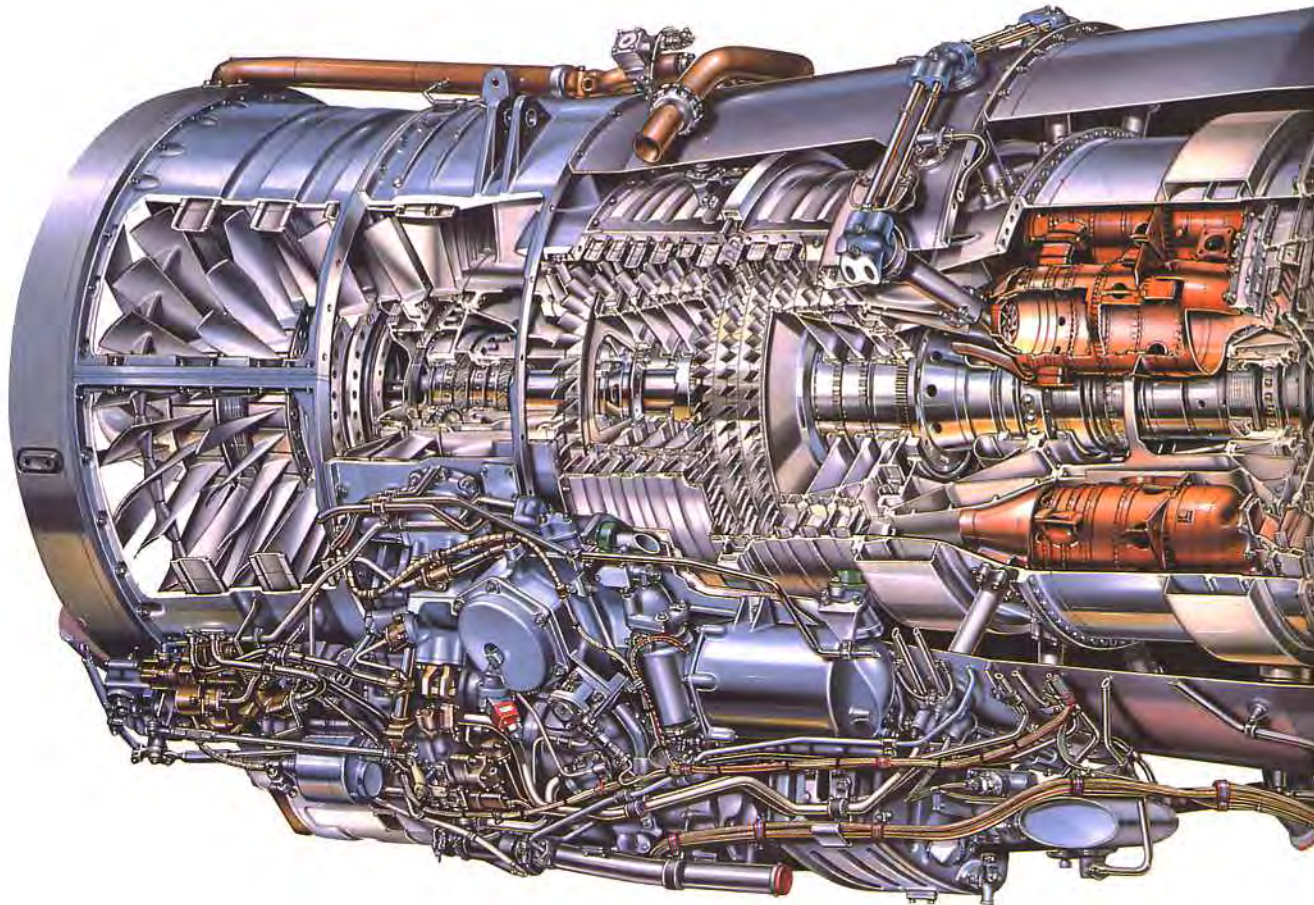
Blackburn Nimbus

The Nimbus was developed from the A129 turbo-shaft which, in its turn, was a modified Turbomeca Artouste built under licence. The Nimbus developed 968 hp, but for helicopter use was flat-rated at 710 hp. The engine was used in Westland Wasp and Scout helicopters and four 700 hp units were used to power the experimental 5RN-2 hovercraft.

21: Performance

Contents

INTRODUCTION

1. The performance requirements of an engine are obviously dictated to a large extent by the type of operation for which the engine is designed. The power of the turbo-jet engine is measured in thrust, produced at the propelling nozzle or nozzles, and that of the turbo-propeller engine is measured in shaft horse-power (s.h.p.) produced at the propeller shaft. However, both types are in the main assessed on the amount of thrust or s.h.p. they develop for a given weight, fuel consumption and frontal area.

2. Since the thrust or s.h.p. developed is dependent on the mass of air entering the engine and the acceleration imparted to it during the engine cycle, it is obviously influenced, as subsequently described, by such variables as the forward speed of the aircraft, altitude and climatic conditions, These variables influence the efficiency of the air intake, the compressor, the turbine and the jet pipe; consequently, the gas energy available for the production of thrust or s.h.p. also varies.

3. In the interest of fuel economy and aircraft range, the ratio of fuel consumption to thrust or s.h.p. should be as low as possible. This ratio, known as the specific fuel consumption (s.f.c.), is expressed in pounds of fuel per hour per pound of net thrust or s.h.p. and is determined by the thermal and propulsive efficiency of the engine. In recent years considerable progress has been made in reducing s.f.c. and weight. These factors are further explained in para. 46.

4. Whereas the thermal efficiency is often referred to as the internal efficiency of the engine, the propulsive efficiency is referred to as the external efficiency. This latter efficiency, described in para. 37, explains why the pure jet engine is less efficient than the turbo-propeller engine at lower aircraft speeds leading to development of the by-pass principle and, more recently, the propfan designs.

5. The thermal and the propulsive efficiency also influence, to a large extent, the size of the compressor and turbine, thus determining the weight and diameter of the engine for a given output.

6. These and other factors are presented in curves and graphs, calculated from the basic gas laws (Part 2), and are proved in practice by bench and flight testing, or by simulating flight conditions in a high altitude test cell. To make these calculations, specific symbols are used to denote the pressures and temperatures at various locations through the engine; for

instance, using the symbols shown in fig. 21-1 the overall compressor pressure ratio is $\dfrac{P_3}{P_1}$. These symbols vary slightly for different types of engine; for instance, with high by-pass ratio engines, and also when afterburning (Part 16) is incorporated, additional symbols are used.

7. To enable the performance of similar engines to be compared, it is necessary to standardize in some conventional form the variations of air temperature and pressure that occur with altitude and climatic conditions. There are in use several different definitions of standard atmospheres, the one in most common use being the International Standard Atmosphere (I.S.A.). This is based on a temperature lapse rate of approximately 1.98 K. degrees per 1,000ft,, resulting in a fall from 288.15 deg.K. (15 deg.C) at sea level to 216.65 deg.K (-56.5 deg.C.) at 36,089 ft. (the tropopause). Above this altitude the
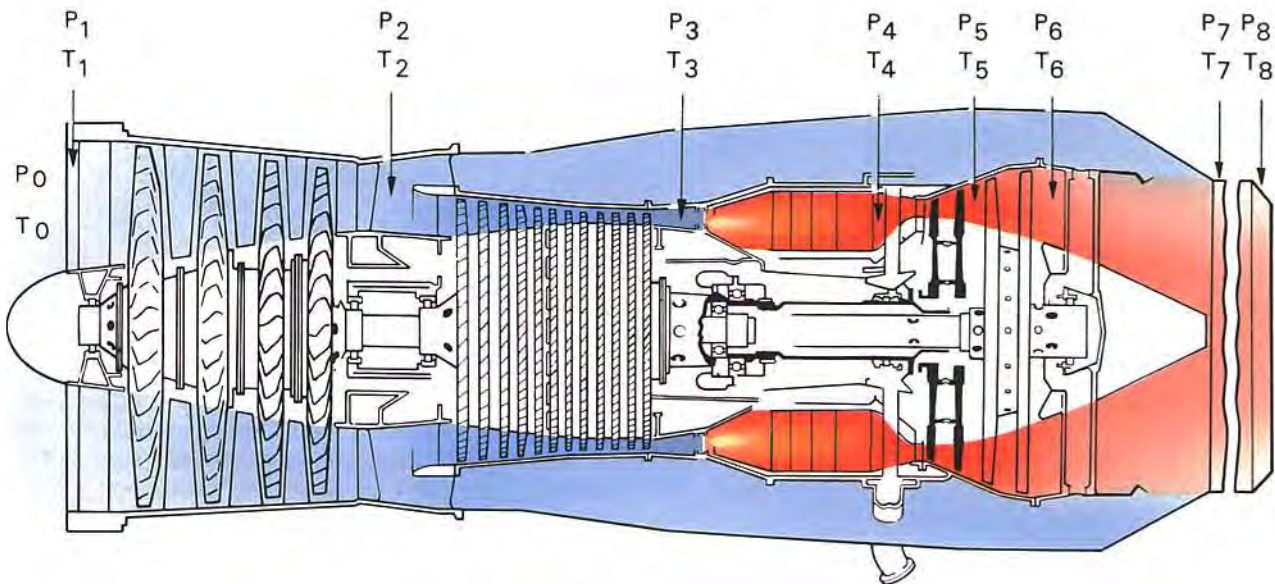
P_0 T_0 Ambient

P_1 T_1 Inlet

P_2 T_2 Low pressure compressor delivery

P_3 T_3 High pressure compressor delivery

P_4 T_4 Turbine entry

P_5 T_5 High pressure turbine exit

P_6 T_6 Low pressure turbine exit

P_7 T_7 Exhaust

P_8 T_8 Propelling nozzle

Fig. 21-1   Temperature and pressure notation of a typical turbo-jet engine.

temperature is constant up to 65.617ft. The I.S.A. standard pressure at sea level is 14.69 pounds per square inch falling to 3.28 pounds per square inch at the tropopause (refer to I.S.A. table fig. 21-10).

ENGINE THRUST ON THE TEST BENCH

8. The thrust of the turbo-jet engine on the test bench differs somewhat from that during flight. Modern test facilities are available to simulate atmospheric conditions at high altitudes thus providing a means of assessing some of the performance capability of a turbo-jet engine in flight without the engine ever leaving the ground. This is important as the changes in ambient temperature and pressure encountered at high altitudes considerably influence the thrust of the engine.

9. Considering the formula derived in Part 20 for engines operating under 'choked' nozzle conditions,

$$\text{Thrust} = (P - P_0) \cdot A + \frac{W_{V_J}}{g}$$

it can be seen that the thrust can be further affected by a change in the mass flow rate of air through the engine and by a change in jet velocity. An increase in mass airflow may be obtained by using water injection (Part 17) and increases in jet velocity by using afterburning (Part 16).

10. As previously mentioned, changes in ambient pressure and temperature considerably influence the thrust of the engine. This is because of the way they affect the air density and hence the mass of air entering the engine for a given engine rotational speed. To enable the performance of similar engines to be compared when operating under different climatic conditions, or at different altitudes, correction factors must be applied to the calculations to return the observed values to those which would be found under I.S.A. conditions. For example, the thrust correction for a turbo-jet engine is:
Thrust (lb.) (corrected) =

$$\text{thrust (lb.) (observed)} \times \frac{30}{P_0}$$

where P_0 = atmospheric pressure in inches of mercury (in. Hg.) (observed)
30 = I.S.A. standard sea level pressure (in.Hg.)

11. The observed performance of the turbo-propeller engine is also corrected to I.S.A. conditions, but due to the rating being in s.h.p. and

not in pounds of thrust the factors are different. For example, the correction for s.h.p. is:
S.h.p. (corrected) =

$$\text{s.h.p. (observed)} \times \frac{30}{P_0} \times \sqrt{\frac{273+15}{273+T_0}}$$

where P_0 = atmospheric pressure (in.Hg.) (observed)
T_0 = atmospheric temperature in deg.C. (observed)
30 = I.S.A. standard sea level pressure (in.Hg.)
273 + 15 = I.S.A. standard sea level temperature in deg.K.
273 + T_0 = Atmospheric temperature in deg.K.

In practice there is always a certain amount of jet thrust in the total output of the turbo-propeller engine and this must be added to the s.h.p. The correction for jet thrust is the same as that in para. 10.

12. To distinguish between these two aspects of the power output, it is usual to refer to them as s.h.p. and thrust horse-power (t.h.p.). The total equivalent horse-power is denoted by t.e.h.p. (sometimes e.h.p.) and is the s.h.p. plus the s.h.p. equivalent to the net jet thrust. For estimation purposes it is taken that, under sea-level static conditions, one s.h.p. is equivalent to approximately 2.6 lb. of jet thrust. Therefore :

$$\text{t.e.h.p.} = \text{s.h.p.} + \frac{\text{jet thrust lb.}}{2.6}$$

13. The ratio of jet thrust to shaft power is influenced by many factors. For instance, the higher the aircraft operating speed the larger may be the required proportion of total output in the form of jet thrust. Alternatively, an extra turbine stage may be required if more than a certain proportion of the total power is to be provided at the shaft. In general, turbo-propeller aircraft provide one pound of thrust for every 3.5 h,p. to 5 h.p.

Comparison between thrust and horse-power
14. Because the turbo-jet engine is rated in thrust and the turbo-propeller engine in s.h.p., no direct comparison between the two can be made without a power conversion factor. However, since the turbo-propeller engine receives its thrust mainly from the propeller, a comparison can be made by converting the horse-power developed by the engine to thrust or the thrust developed by the turbo-jet engine to t.h.p.; that is, by converting work to force or force to work. For this purpose, it is necessary to take into account the speed of the aircraft.

15. The t.h.p. is expressed as $\dfrac{FV}{550 \text{ ft. per sec.}}$

where F = lb. of thrust
V = aircraft speed (ft. per sec.)

Since one horse-power is equal to 550 ft.lb. per sec. and 550 ft. per sec. is equivalent to 375 miles per hour, it can be seen from the above formula that one lb. of thrust equals one t.h.p. at 375 m.p.h. It is also common to quote the speed in knots (nautical miles per hour); one knot is equal to 1.1515 m.p.h, or one pound of thrust is equal to one t.h.p. at 325 knots.

16. Thus if a turbo-jet engine produces 5,000 lb. of net thrust at an aircraft speed of 600 m.p.h. the t.h.p. would be $\dfrac{5,000 \times 600}{375} = 8,000$

However, if the same thrust was being produced by a turbo-propeller engine with a propeller efficiency of 55 per cent at the same flight speed of 600 m.p.h., then the t.h.p. would be

$$8,000 \times \dfrac{100}{55} = 14,545$$

Thus at 600 m.p.h. one lb. of thrust is the equivalent of about 3 t.h.p.

ENGINE THRUST IN FLIGHT

17. Since reference will be made to gross thrust, momentum drag and net thrust, it will be helpful to define these terms:

from Part 20, gross or total thrust is the product of the mass of air passing through the engine and the jet velocity at the propelling nozzle, expressed as:

$$(P - P_0)A + \dfrac{W v_J}{g}$$

The momentum drag is the drag due to the momentum of the air passing into the engine relative to the aircraft velocity, expressed as $\dfrac{WV}{g}$ where

W = Mass flow in lb. per sec.
V = Velocity of aircraft in feet per sec.
g = Gravitational constant 32.2 ft. per sec. per sec.

The net thrust or resultant force acting on the aircraft in flight is the difference between the gross thrust and the momentum drag.

18. From the definitions and formulae stated in para, 17; under flight conditions, the net thrust of the

MOMENTUM DRAG $= \dfrac{WV}{g}$ ➡️⬅️ GROSS THRUST $= (P - P_0)A + \dfrac{Wv_J}{g}$ $\Bigg\{$ ⬅️ MOMENTUM THRUST $= \dfrac{Wv_J}{g}$

⬅️ PRESSURE THRUST $= (P - P_0)A$
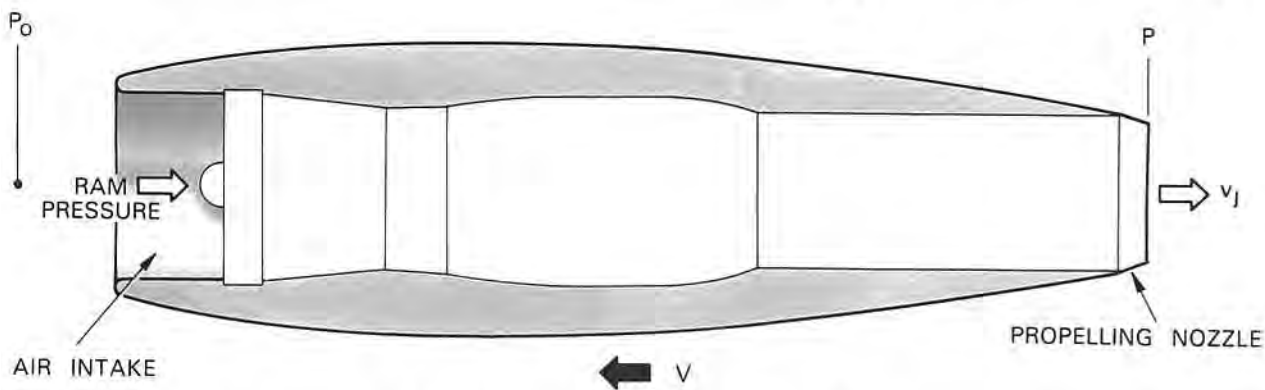
All pressures are total pressures except P which is the static pressure at the propelling nozzle

W = Mass of air passing through engine (lb. per sec.)
v_J = Jet velocity at propelling nozzle (ft. per sec.)
P = Static pressure across propelling nozzle (lb. per sq. in.)
P_0 = Atmospheric pressure (lb. per sq. in.)
A = Propelling nozzle area (sq. in.)
V = Aircraft speed (ft. per sec.)
g = Gravitational constant 32.2

Fig. 21-2   The balance of forces and expression for thrust and momentum drag.

engine, simplifying, can be expressed as:

$$(P - P_0)A + \frac{W(v_J - V)}{g}$$

Fig. 21-2 provides a diagrammatic explanation.

Effect of forward speed

19. Since reference will be made to 'ram ratio' and Mach number, these terms are defined as follows:

Ram ratio is the ratio of the total air pressure at the engine compressor entry to the static air pressure at the air intake entry.

Mach number is an additional means of measuring speed and is defined as the ratio of the speed of a body to the local speed of sound. Mach 1.0 therefore represents a speed equal to the local speed of sound.

20. From the thrust equation in para. 18, it is apparent that if the jet velocity remains constant, independent of aircraft speed, then as the aircraft speed increases the thrust would decrease in direct proportion. However, due to the 'ram ratio' effect from the aircraft forward speed, extra air is taken into the engine so that the mass airflow and also the jet velocity increase with aircraft speed. The effect of this tends to offset the extra intake momentum drag
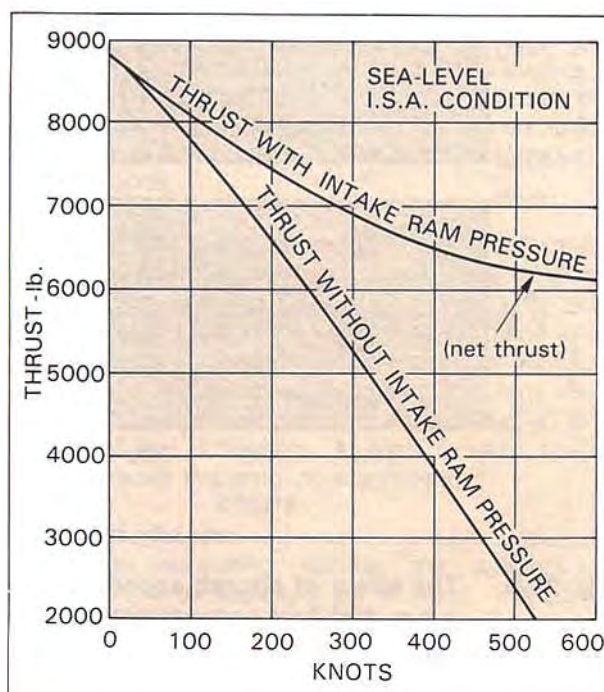
Fig. 21-3   Thrust recovery with aircraft speed.
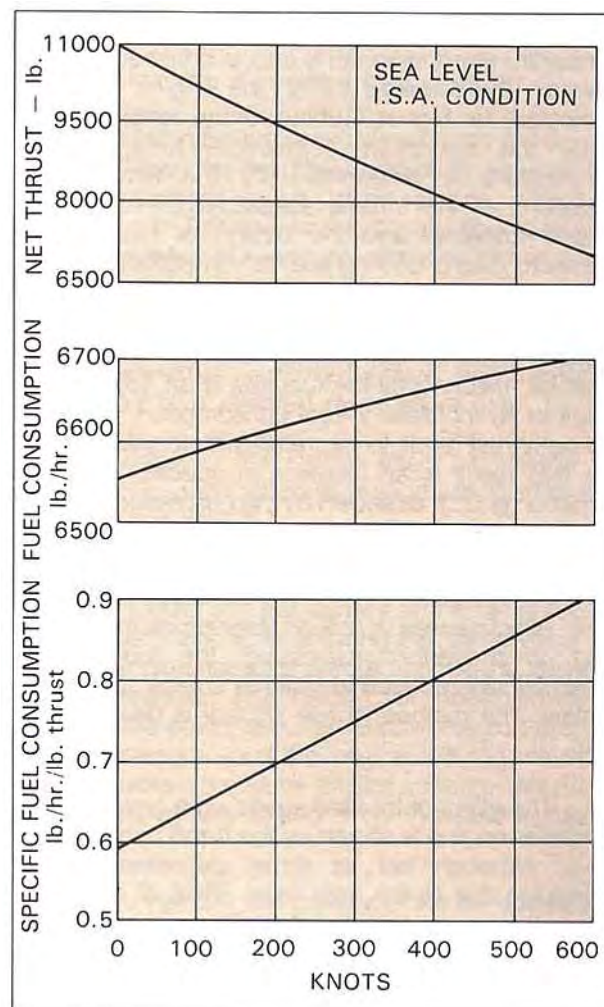
Fig. 21-4   The effect of aircraft speed on thrust and fuel consumption.

due to the forward speed so that the resultant loss of net thrust is partially recovered as the aircraft speed increases. A typical curve illustrating this point is shown in fig. 21-3. Obviously, the 'ram ratio' effect, or the return obtained in terms of pressure rise at entry to the compressor in exchange for the unavoidable intake drag, is of considerable importance to the turbo-jet engine, especially at high speeds. Above speeds of Mach 1.0, as a result of the formation of shock waves at the air intake, this rate of pressure rise will rapidly decrease unless a suitably designed air intake is provided (Part 23); an efficient air intake is necessary to obtain maximum benefit from the ram ratio effect.

21. As aircraft speeds increase into the supersonic region, the ram air temperature rises rapidly consistent with the basic gas laws (Part 2). This

temperature rise affects the compressor delivery air temperature proportionately and, in consequence, to maintain the required thrust, the engine must be subjected to higher turbine entry temperatures. Since the maximum permissible turbine entry temperature is determined by the temperature limitations of the turbine assembly, the choice of turbine materials and the design of blades and stators to permit cooling are very important.

22. With an increase in forward speed, the increased mass airflow due to the 'ram ratio' effect must be matched by the fuel flow (Part 10) and the result is an increase in fuel consumption. Because the net thrust tends to decrease with forward speed the end result is an increase in specific fuel consumption (s.f.c.), as shown by the curves for a typical turbo-jet engine in fig, 21-4.

23. At high forward speeds at low altitudes the 'ram ratio' effect causes very high stresses on the engine and, to prevent overstressing, the fuel flow is automatically reduced to limit the engine speed and airflow. The method of fuel control is described in Part 10.

24. The effect of forward speed on a typical turbo-propeller engine is shown by the trend curves in fig. 21 -5. Although net jet thrust decreases, s.h.p. increases due to the 'ram ratio1 effect of increased mass flow and matching fuel flow. Because it is standard practice to express the s.f.c. of a turbo-propeller engine relative to s.h.p., an improved s.f.c. is exhibited. However, this does not provide a true comparison with the curves shown in fig. 21-4, for a typical turbo-jet engine, as s.h.p, is absorbed by the propeller and converted into thrust and, irrespective of an increase in s.h.p., propeller efficiency and therefore net thrust deteriorates at high subsonic forward speeds. In consequence, the turbo-propeller engine s.f.c, relative to net thrust would, in general comparison with the turbo-jet engine, show an improvement at low forward speeds but a rapid deterioration at high speeds.

Effect of afterburning on engine thrust
25. At take-off conditions, the momentum drag of the airflow through the engine is negligible, so that the gross thrust can be considered to be equal to the net thrust. If afterburning (Part 16) is selected, an increase in take-off thrust in the order of 30 per cent is possible with the pure jet engine and considerably more with the by-pass engine. This augmentation of basic thrust is of greater advantage for certain specific operating requirements.
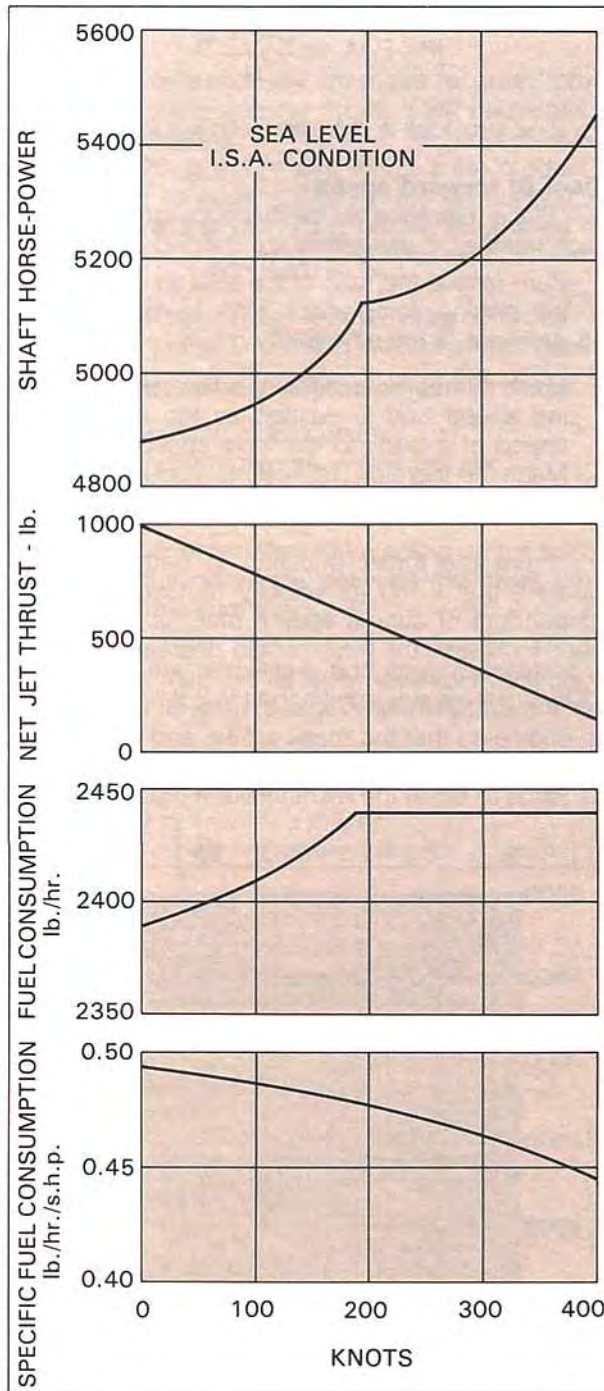
Fig. 21-5   The effect of aircraft speed on s.h.p. and fuel consumption.

26. Under flight conditions, however, this advantage is even greater, since the momentum drag is the same with or without afterburning and, due to the ram effect, better utilization is made of every pound

of air flowing through the engine. The following example, using the static values given in Part 16, illustrates why afterburning thrust improves under flight conditions.

27. Assuming an aircraft speed of 600 m.p.h. (880ft. per sec.), then Momentum drag is:

$$\frac{880}{32} = 27.5 \text{ (approximately)}$$

This means that every pound of air per second flowing through the engine and accelerated up to the speed of the aircraft causes a drag of about 27.5 lb.

28. Suppose each pound of air passed through the engine gives a gross thrust of 77.5 lb. Then the net thrust given by the engine per lb. of air per second is 77.5 - 27.5 = 50 lb.

29. When afterburning is selected, assuming the 30 per cent increase in static thrust given in para. 25, the gross thrust will be 1.3 x 77.5 - 100.75 lb. Thus, under flight condition of 600 m.p.h., the net thrust per pound of air per second will be 100.75 - 27.5 = 73.25 lb. Therefore, the ratio of net thrust due to afterburning is $\frac{73.25}{50}$ = 1.465. In other words, a 30 per cent increase in thrust under static conditions becomes a 46.5 per cent increase in thrust at 600 m.p.h.

30. This larger increase in thrust is invaluable for obtaining higher speeds and higher altitude performances. The total and specific fuel consumptions are high, but not unduly so for such an increase in performance.

31. The limit to the obtainable thrust is determined by the afterburning temperature and the remaining usable oxygen in the exhaust gas stream. Because no previous combustion heating takes place in the duct of a by-pass engine, these engines with their large residual oxygen surplus are particularly suited to afterburning and static thrust increases of up to 70 per cent are obtainable. At high forward speeds several times this amount is achieved.

Effect of altitude
32. With increasing altitude the ambient air pressure and temperature are reduced. This affects the engine in two interrelated ways:

The fall of pressure reduces the air density and hence the mass airflow into the engine for a given engine speed. This causes the thrust or s.h.p. to fall. The fuel control system, as described in Part 10, adjusts the fuel pump

output to match the reduced mass airflow, so maintaining a constant engine speed.

The fall in air temperature increases the density of the air, so that the mass of air entering the compressor for a given engine speed is greater. This causes the mass airflow to reduce at a lower rate and so compensates to some extent for the loss of thrust due to the fall in atmospheric pressure. At altitudes above 36,089 feet and up to 65,617 feet, however, the temperature remains constant, and the thrust or s.h.p. is affected by pressure only.

Graphs showing the typical effect of altitude on thrust, s.h.p, and fuel consumption are illustrated in fig. 21-6 and fig. 21-7.

Effect of temperature
33. On a cold day the density of the air increases so that the mass of air entering the compressor for a given engine speed is greater, hence the thrust or s.h.p, is higher. The denser air does, however, increase the power required to drive the compressor or compressors; thus the engine will require more fuel to maintain the same engine speed or will run at a reduced engine speed if no increase in fuel is available.

34. On a hot day the density of the air decreases, thus reducing the mass of air entering the compressor and, consequently, the thrust of the engine for a given r.p.m. Because less power will be required to drive the compressor, the fuel control system reduces the fuel flow to maintain a constant engine rotational speed or turbine entry temperature, as appropriate; however, because of the decrease in air density, the thrust will be lower. At a temperature of 45 deg.C., depending on the type of engine, a thrust loss of up to 20 per cent may be experienced. This means that some sort of thrust augmentation, such as water injection (Part 17), may be required.

35. The fuel control system (Part 10) controls the fuel flow so that the maximum fuel supply is held practically constant at low air temperature conditions, whereupon the engine speed falls but, because of the increased mass airflow as a result of the increase in air density, the thrust remains the same. For example, the combined acceleration and speed control fuel system (Part 10) schedules fuel flow to maintain a constant engine r.p.m., hence thrust increases as air temperature decreases until, at a predetermined compressor delivery pressure, the fuel flow is automatically controlled to maintain a constant compressor delivery pressure and,
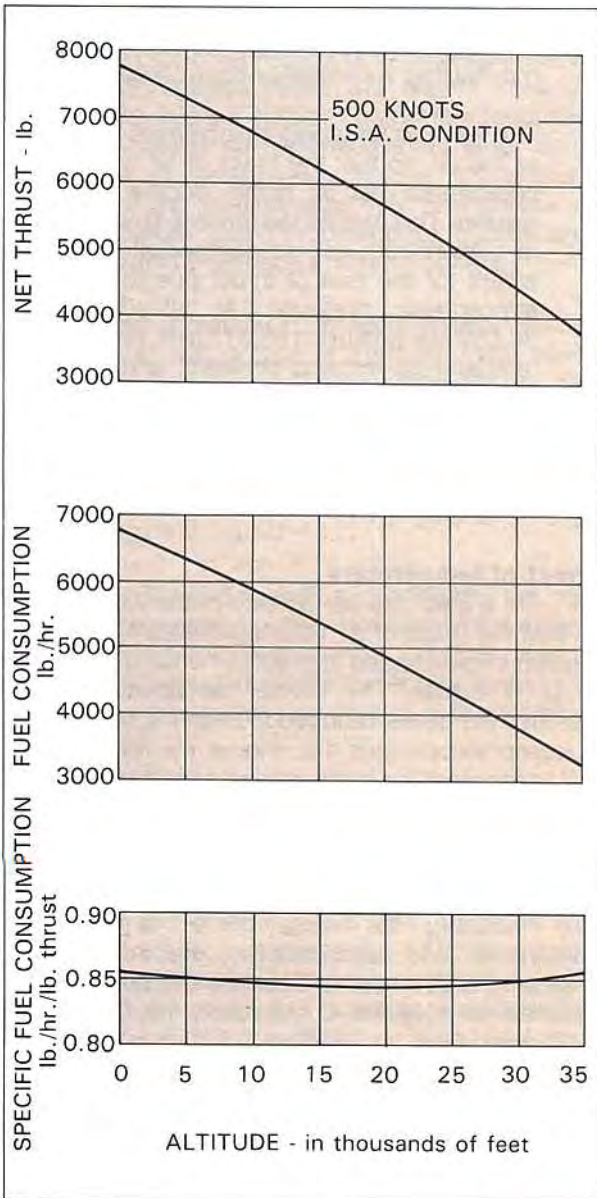
Fig. 21-6   The effects of altitude on thrust and fuel consumption.

therefore, thrust. Fig. 21-8 illustrates this for a twin-spool engine where the controlled engine r.p.m. is high pressure compressor speed and the compressor delivery pressure is expressed as P3. It will also be apparent from this graph that the low pressure compressor speed is always less than its limiting maximum and that the difference in the two speeds is reduced by a decrease in ambient air temperature. To prevent the L.P. compressor over-speeding, fuel flow is also controlled by an L.P. governor which, in this case, takes a passive role.
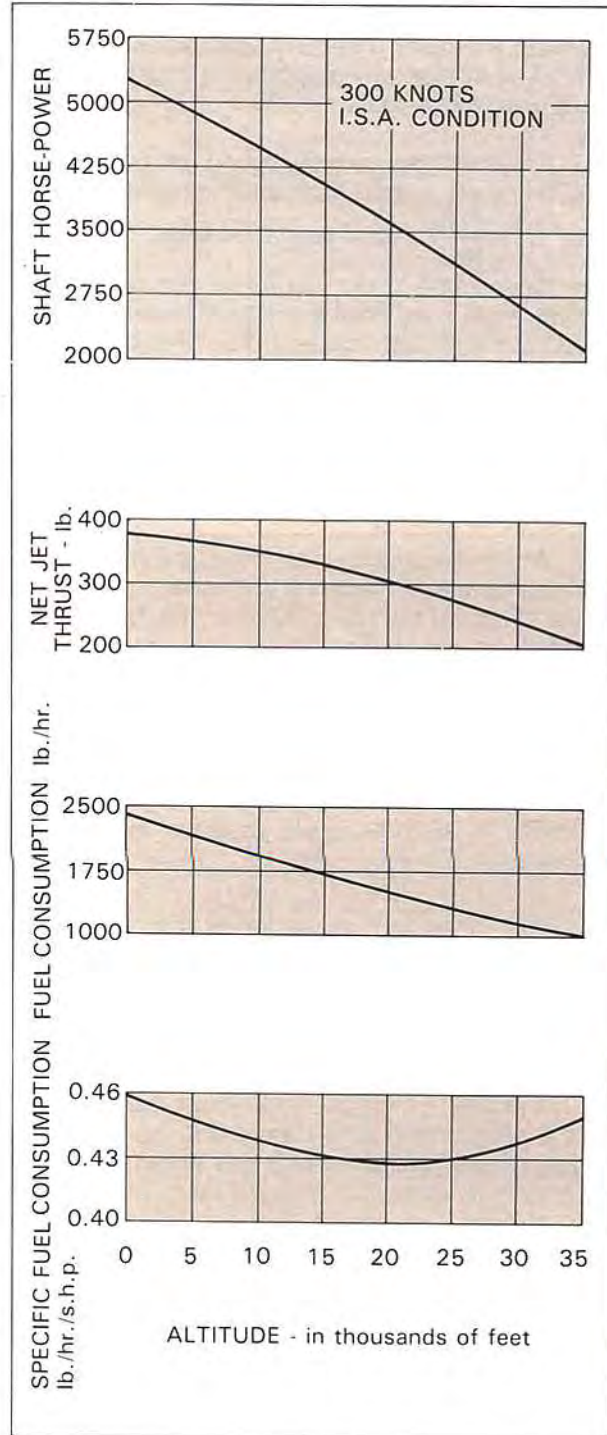
Fig. 21-7   The effect of altitude on s.h.p. and fuel consumption.

36.   The pressure ratio control fuel system (Part 10) schedules fuel flow to maintain a constant engine pressure ratio and, therefore, thrust below a prede-

termined ambient air temperature. Above this temperature the fuel flow is automatically controlled to prevent turbine entry temperature limitations from being exceeded, thus resulting in reduced thrust and, overall, similar curve characteristics to those shown in fig. 21-8. In the instance of a triple-spool engine the pressure ratio is expressed as P_4/P_1. i.e. H.P. compressor delivery pressure/engine inlet pressure.

PROPULSIVE EFFICIENCY

37. Performance of the jet engine is not only concerned with the thrust produced, but also with the efficient conversion of the heat energy of the fuel into kinetic energy, as represented by the jet velocity, and the best use of this velocity to propel the aircraft forward, i.e. the efficiency of the propulsive system.

38. The efficiency of conversion of fuel energy to kinetic energy is termed thermal or internal efficiency and, like all heat engines, is controlled by the cycle pressure ratio and combustion temperature. Unfortunately, this temperature is limited by the thermal and mechanical stresses that can be tolerated by the turbine. The development of new materials and techniques to minimize these limitations is continually being pursued.

39. The efficiency of conversion of kinetic energy to propulsive work is termed the propulsive or external efficiency and this is affected by the amount of kinetic

energy wasted by the propelling mechanism. Waste energy dissipated in the jet wake, which represents a loss, can be expressed as $\dfrac{W(v_J - V)^2}{2g}$ where $(v_J$-V) is the waste velocity. It is therefore apparent that at the aircraft lower speed range the pure jet stream wastes considerably more energy than a propeller system and consequently is less efficient over this range. However, this factor changes as aircraft speed increases, because although the jet stream continues to issue at a high velocity from the engine its velocity relative to the surrounding atmosphere is reduced and, in consequence, the waste energy loss is reduced.

40. Briefly, propulsive efficiency may be expressed as:

$$\dfrac{\text{Work done on the aircraft}}{\text{Energy imparted to engine airflow}} \text{ or simply}$$

$$\dfrac{\text{Work done}}{\text{Work done + work wasted in exhaust}}$$

Work done is the net thrust multiplied by the aircraft speed. Therefore, progressing from the net thrust equation given in para. 18, the following equation is arrived at:

Propulsive efficiency =

$$\dfrac{V\left[(P - P_0)A + \dfrac{W(v_J - V)}{g}\right]}{V\left[(P - P_0)A + \dfrac{W(v_J - V)}{g}\right] + \dfrac{W(v_J - V)^2}{2g}}$$
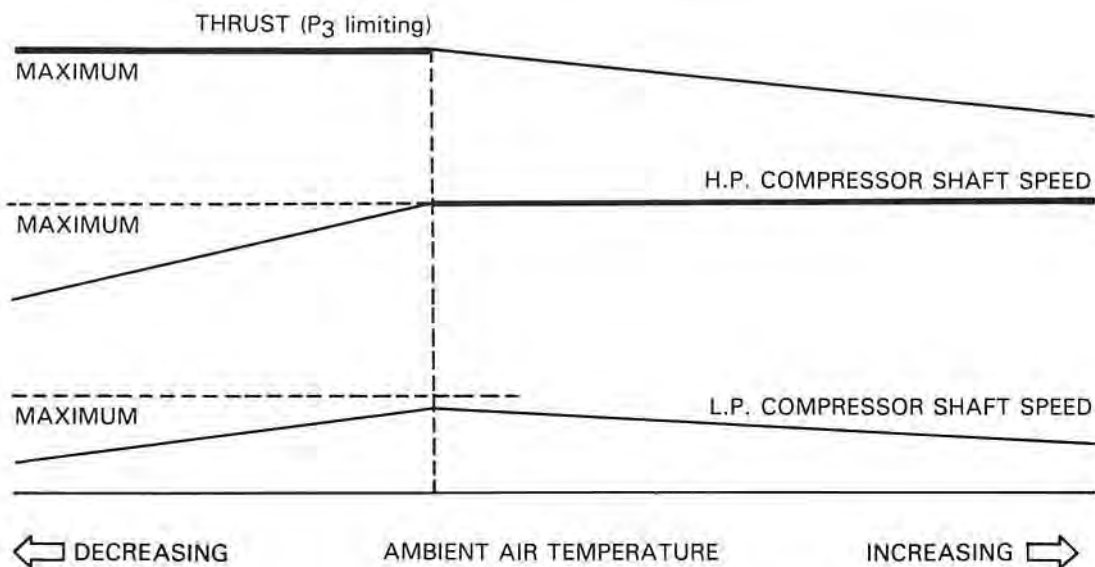
Fig. 21-8   The effect of air temperature on a typical twin-spool engine.

In the instance of an engine operating with a non-choked nozzle (Part 20), the equation becomes:

$$\frac{WV(v_J - V)}{WV(v_J - V) + \frac{1}{2} W(v_J - V)^2}$$

$$\text{Simplified to : } \frac{2V}{V + v_J}$$

41. This latter equation can also be used for the choked nozzle condition by using vj to represent the jet velocity when fully expanded to atmospheric pressure, thereby dispensing with the nozzle pressure term $(P-P_0)A$.

42. Assuming an aircraft speed (V) of 375 m.p.h. and a jet velocity (vj) of 1,230 rn.p.h., the efficiency of a turbo-jet is:

$$\frac{2 \times 375}{375 + 1,230} = \text{approx. 47 per cent}$$

On the other hand, at an aircraft speed of 600 m.p.h. the efficiency is:

$$\frac{2 \times 600}{600 + 1,230} = \text{approx. 66 per cent}$$

Propeller efficiency at these values of V is approximately 82 and 55'per cent, respectively, and from
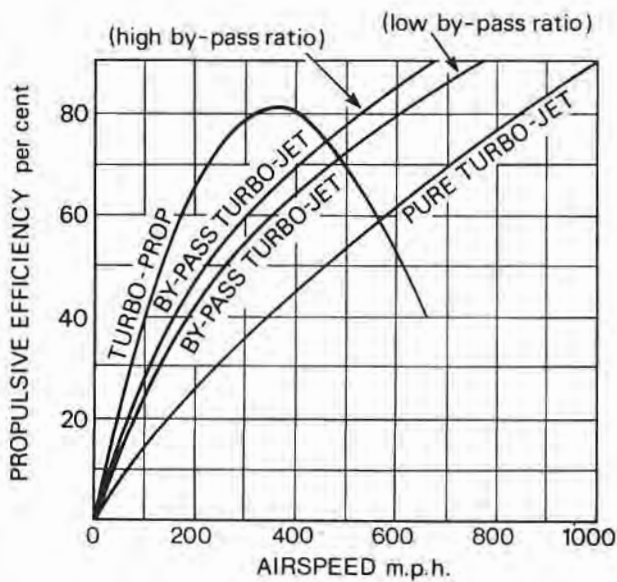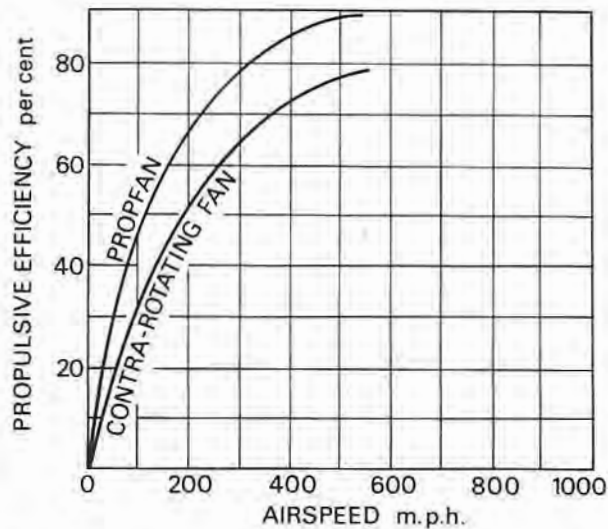
Fig. 21-9   Propulsive efficiencies and aircraft speed.

reference to fig. 21-9 it can be seen that for aircraft designed to operate at sea level speeds below approximately 400 m.p.h. it is more effective to absorb the power developed in the jet engine by gearing it to a propeller instead of using it directly in the form of a pure jet stream. The disadvantage of the propeller at the higher aircraft speeds is its rapid fall off in efficiency, due to shock waves created around the propeller as the blade tip speed approaches Mach 1.0. Advanced propeller technology, however, has produced a multi-bladed, swept back design capable of turning with tip speeds in excess of Mach 1.0 without loss of propeller efficiency. By using this design of propeller in a contra-rotating configuration, thereby reducing swirl losses, a 'prop-fan' engine, with very good propulsive efficiency capable of operating efficiently at aircraft speeds in excess of 500 m.p.h. at sea level, can be produced.

43.   To obtain good propulsive efficiencies without the use of a complex propeller system, the by-pass principle (Part 2) is used in various forms. With this principle, some part of the total output is provided by a jet stream other than that which passes through the engine cycle and this is energized by a fan or a varying number of LP. compressor stages. This bypass air is used to lower the mean jet temperature and velocity either by exhausting through a separate propelling nozzle, or by mixing with the turbine stream to exhaust through a common nozzle.

44.   The propulsive efficiency equation for a high by-pass ratio engine exhausting through separate nozzles is given below, where W_1 and V_{J1} relate to the by-pass function and W_2 and v_{J2} to the engine main function.

Propulsive efficiency =

$$\frac{W_1V(v_{J_1}-V)+W_2V(v_{J_2}-V)}{W_1V(v_{J_1}-V)+W_2V(v_{J_2}-V)+\tfrac{1}{2}W_1V(v_{J_1}-V)^2+\tfrac{1}{2}W_2V(v_{J_2}-V)^2}$$

By calculation, substituting the following values, which will be typical of a high by-pass ratio engine of triple-spool configuration, it will be observed that a propulsive efficiency of approximately 85 per cent results.

$$V \quad = 583 \text{ rn.p.h.}$$
$$W_1 = 492 \text{ lb. per sec.}$$
$$W_2 = 100 \text{ lb. per sec.}$$
$$V_{J1} = 781 \text{ m.p.h.}$$
$$V_{J2} = 812 \text{ m.p.h.}$$

Propulsive efficiency can be further improved by using the rear mounted contra-rotating fan configuration of the by-pass principle. This gives very high by-

pass ratios in the order of 15:1, and reduced 'drag' results due to the engine core being 'washed' by the low velocity aircraft slipstream and not the relatively high velocity fan efflux.

45.   The improved propulsive efficiency of the bypass system bridges the efficiency gap between the turbo-propeller engine and the pure turbo-jet engine. A graph illustrating the various propulsive efficiencies with aircraft speed is shown in fig. 21-9.

FUEL CONSUMPTION AND POWER-TO-WEIGHT RELATIONSHIP

46.   Primary engine design considerations, particularly for commercial transport duty, are those of low specific fuel consumption and weight. Considerable improvement has been achieved by use of the by-pass principle, and by advanced mechanical and aerodynamic features, and the use of improved materials. With the trend towards higher by-pass ratios, in the range of 15:1, the triple-spool and contra-rotating rear fan engines allow the pressure and by-pass ratios to be achieved with short rotors, using fewer compressor stages, resulting in a lighter and more compact engine.

47.   S.f.c. is directly related to the thermal and propulsive efficiencies; that is, the overall efficiency of the engine. Theoretically, high thermal efficiency requires high pressures which in practice also means high turbine entry temperatures. In a pure turbo-jet engine this high temperature would result in a high jet velocity and consequently lower the propulsive efficiency (para. 40). However, by using the by-pass principle, high thermal and propulsive efficiencies can be effectively combined by bypassing a proportion of the L.P. compressor or fan delivery air to lower the mean jet temperature and velocity as referred to in para. 43. With advanced technology engines of high by-pass and overall pressure ratios, a further pronounced improvement in s.f.c. is obtained.

48.   The turbines of pure jet engines are heavy because they deal with the total airflow, whereas the turbines of by-pass engines deal only with part of the flow; thus the H.P. compressor, combustion chambers and turbines, can be scaled down. The increased power per lb. of air at the turbines, to take advantage of their full capacity, is obtained by the increase in pressure ratio and turbine entry temperature. It is clear that the by-pass engine is lighter, because not only has the diameter of the high pressure rotating assemblies been reduced but the engine is shorter for a given power output. With a low

THE INTERNATIONAL
STANDARD ATMOSPHERE (I.S.A.)

ALTITUDE (h)		AMBIENT TEMPERATURE (To)			AMBIENT PRESSURE (Po)		SPEED OF SOUND (ao)		
Feet	Metres	Deg. K.	Deg. C.	Deg. F.	lb./sq. in.	millibars	ft./sec.	knots	m./sec.
−1,000	−304.8	290.13	+16.98	62.6	15.24	1050.4	1120.3	663.3	341.5
0	0	288.15	15.00	59.0	14.69	1013.2	1116.6	661.1	340.3
+1,000	+304.8	286.17	13.02	55.4	14.17	977.1	1112.6	658.8	339.1
2,000	609.6	284.19	11.04	51.9	13.66	942.1	1108.7	656.5	337.9
3,000	914.4	282.21	9.06	48.3	13.17	908.1	1104.9	654.2	336.8
4,000	1219.2	280.23	7.08	44.7	12.69	875.1	1100.9	651.9	335.6
5,000	1524.0	278.24	5.09	41.2	12.23	843.0	1097.1	649.6	334.4
6,000	1828.8	276.26	3.11	37.6	11.78	811.9	1093.2	647.8	333.2
7,000	2133.6	274.28	1.13	34.0	11.34	781.8	1089.3	644.9	332.0
8,000	2438.4	272.30	−0.85	30.5	10.92	752.6	1085.3	642.6	330.8
9,000	2743.2	270.32	−2.83	26.9	10.51	724.3	1081.4	640.3	329.6
10,000	3048.0	268.34	−4.81	23.3	10.11	696.8	1077.4	637.9	328.4
11,000	3352.8	266.36	−6.79	19.8	9.72	670.2	1073.4	635.6	327.2
12,000	3657.6	264.38	−8.77	16.2	9.35	644.4	1069.4	633.2	325.9
13,000	3962.4	262.39	−10.76	12.6	8.98	619.4	1065.4	630.8	324.7
14,000	4267.2	260.41	−12.74	9.1	8.63	595.2	1061.4	628.4	323.5
15,000	4572.0	258.43	−14.72	5.5	8.29	571.7	1057.3	626.0	322.3
16,000	4876.8	256.45	−16.70	1.9	7.97	549.1	1053.3	623.6	321.1
17,000	5181.6	254.47	−18.68	−1.6	7.65	527.2	1049.2	621.2	319.8
18,000	5486.4	252.49	−20.66	−5.2	7.34	505.9	1045.1	618.8	318.5
19,000	5791.2	250.51	−22.64	−8.8	7.04	485.6	1040.9	616.4	317.3
20,000	6096.0	248.53	−24.62	−12.3	6.75	465.6	1036.9	613.9	316.1
21,000	6400.8	246.54	−26.61	−15.9	6.48	446.4	1032.7	611.5	314.8
22,000	6705.6	244.56	−28.59	−19.5	6.21	427.9	1028.6	609.0	313.5
23,000	7010.4	242.58	−30.57	−23.0	5.95	409.9	1024.4	606.5	312.2
24,000	7315.2	240.60	−32.55	−26.6	5.69	392.7	1020.2	604.1	310.9
25,000	7620.0	238.62	−34.53	−30.2	5.45	375.9	1015.9	601.6	309.7
26,000	7924.8	236.64	−36.51	−33.7	5.22	359.9	1011.8	599.1	308.4
27,000	8229.6	234.66	−38.49	−37.3	4.99	344.3	1007.5	596.6	307.1
28,000	8534.4	232.68	−40.47	−40.9	4.78	329.3	1003.2	594.0	305.8
29,000	8839.2	230.69	−42.46	−44.4	4.57	314.8	998.9	591.5	304.5
30,000	9144.0	228.71	−44.44	−48.0	4.36	300.9	994.7	588.9	303.2
31,000	9448.8	226.73	−46.42	−51.6	4.17	287.4	990.3	586.4	301.9
32,000	9753.6	224.75	−48.40	−55.1	3.98	274.5	986.0	583.8	300.5
33,000	10058.4	222.77	−50.38	−58.7	3.80	261.9	981.7	581.2	299.2
34,000	10363.2	220.79	−52.36	−62.3	3.63	249.9	977.3	578.7	297.9
35,000	10668.0	218.81	−54.34	−65.8	3.46	238.4	972.9	576.1	296.5
36,000	10972.8	216.83	−56.32	−69.4	3.29	227.3	968.5	573.4	295.2
36,089	11000.0	216.65	−56.50	−69.7	3.28	226.3	968.1	573.2	295.1
37,000	11277.6				3.14	216.6			
38,000	11582.4	Ambient temperature remains			2.99	206.5	Speed of sound remains		
39,000	11887.2	constant from this point up to			2.85	196.8	constant from this point		
40,000	12192.0	65,617 ft.			2.72	187.5	up to 65,617 ft.		
45,000	13716.0				2.14	147.5			
50,000	15240.0				1.68	115.9			
55,000	16764.0				1.32	91.2			
60,000	18288.0				1.04	71.7			
65,000	19812.0				0.82	56.4			

Fig. 21-10 International Standard Atmosphere.

by-pass ratio engine, the weight reduction compared with a pure jet engine is in the order of 20 per cent for the same air mass flow.

49.   With a high by-pass ratio engine of the triple-spool configuration, a further significant improvement in specific weight is obtained- This is derived mainly from advanced mechanical and aerodynamic design, which in addition to permitting a significant reduction in the total number of parts, enables rotating assemblies to be more effectively matched and to work closer to optimum conditions, thus minimizing the number of compressor and turbine stages for a given duty. The use of higher strength light-weight materials is also a contributory factor.

50.   For a given mass flow less thrust is produced by the by-pass engine due to the lower exit velocity. Thus, to obtain the same thrust, the by-pass engine must be scaled to pass a larger total mass airflow than the pure turbo-jet engine. The weight of the engine, however, is still less because of the reduced size of the H.P. section of the engine. Therefore, in addition to the reduced specific fuel consumption, an improvement in the power-to-weight ratio is obtained.

Rolls-Royce RB168 Mk202/Mk203
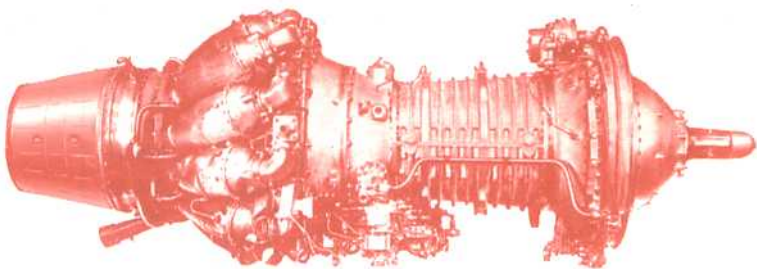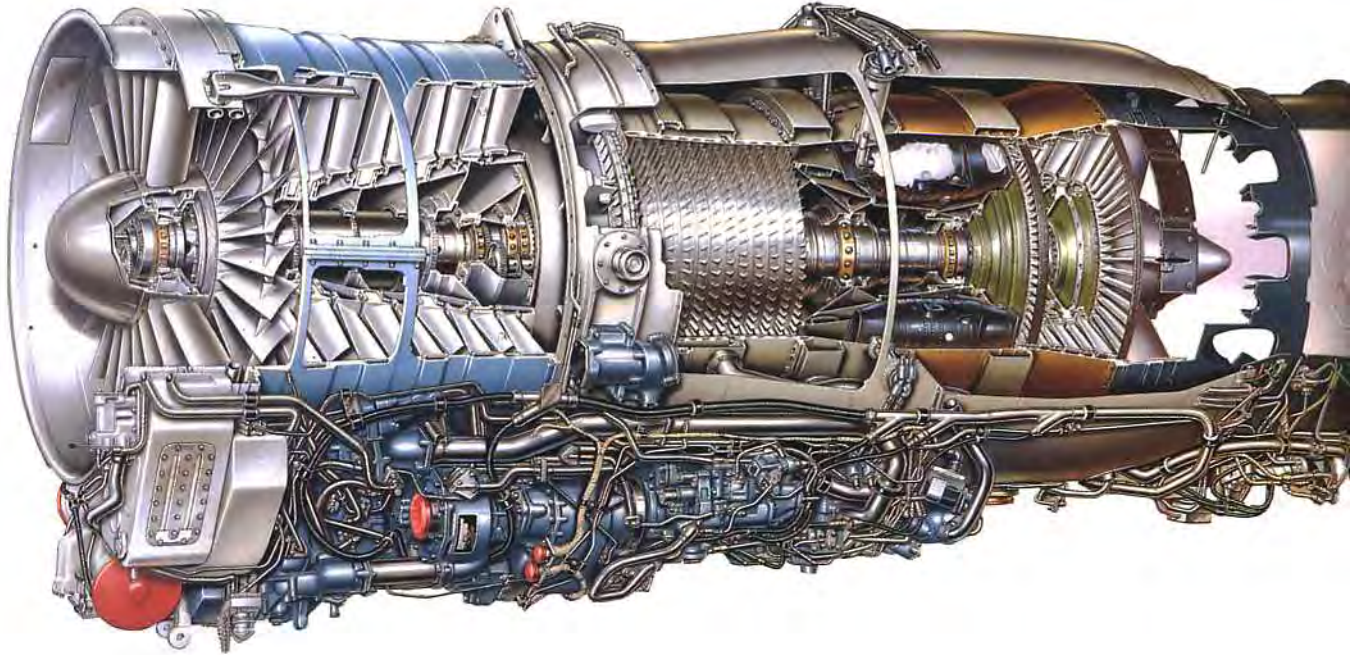
Rolls-Royce RB39 Clyde

Encouraged by results obtained from the Trent, Rolls-Royce decided to go ahead with an engine designed from the start as a turbo-prop. Named the Clyde it utilized the axial compressor from the Metrovick F2 as first stage and a scaled up supercharger impeller from a Merlin as second stage. First running in August 1945 at 2000 shp, later engines produced up to 4200 shp.

22: Manufacture

Contents

INTRODUCTION

1. During the design stages of the aircraft gas turbine engine, close liaison is maintained between design, manufacturing, development and product support to ensure that the final design is a match between the engineering specification and the manufacturing process capability.

2. The functioning of this type of engine, with its high power-to-weight ratio, demands the highest possible performance from each component. Consistent with this requirement, each component must be manufactured at the lowest possible weight and cost and also provide mechanical integrity through a long service life. Consequently, the methods used during manufacture are diverse and are usually determined by the duties each component has to fulfil.

3. No manufacturing technique or process that In any way offers an advantage is ignored and most available engineering methods and processes are employed in the manufacture of these engines, In some instances, the technique or process may appear by some standards to be elaborate, time consuming and expensive, but is only adopted after confirmation that it does produce maximized component lives comparable with rig test achievements.

4. Engine components are produced from a variety of high tensile steel and high temperature nickel and cobalt alloy forgings. A proportion of components are cast using the investment casting process. Whilst fabrications, which form an increasing content, are produced from materials such as stainless steel, titanium and nickel alloys using modern joining
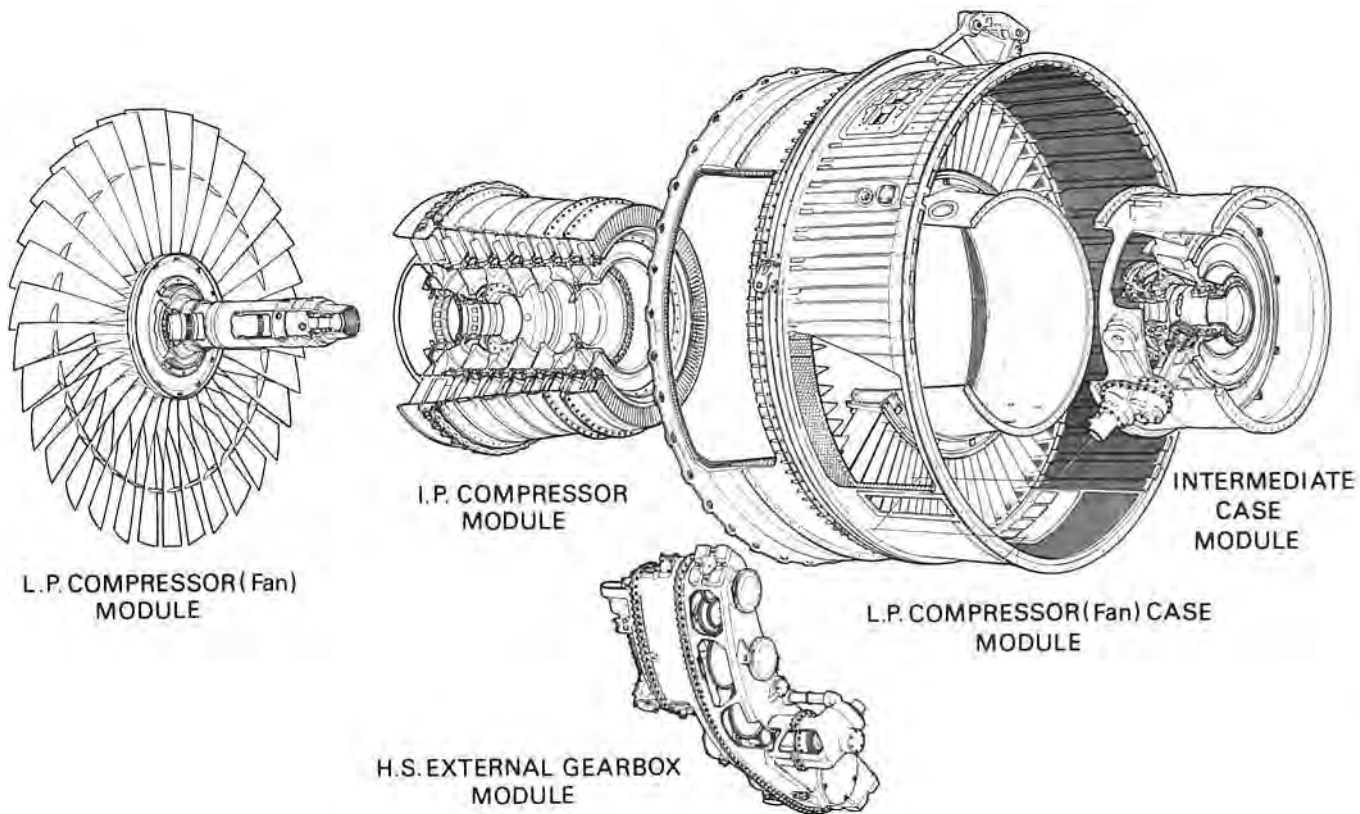
Fig. 22-1   Arrangements of a triple-spool turbo-jet engine.

techniques i.e., tungsten inert gas welding, resistance welding, electron beam welding and high temperature brazing in vacuum furnaces.

5. The methods of machining engine components include grinding, turning, drilling, boring and broaching whenever possible, with the more difficult materials and configurations being machined by electro-discharge, electro-chemical, laser hole drilling and chemical size reduction.

6. Structural components i.e., cold spoiler, location rings and by-pass ducts, benefit by considerable weight saving when using composite materials.

7. In addition to the many manufacturing methods, chemical and thermal processes are used on part finished and finished components. These include heat treatment, electro-plating, chromate sealing, chemical treatments, anodizing to prevent corrosion, chemical and mechanical cleaning, wet and dry abrasive blasting, polishing, plasma spraying, elec-trolytic etching and polishing to reveal metallurgical defects. Also a variety of barrelling techniques for removal o! burrs and surface improvement. Most processes are concerned with surface changes,

some give resistance to corrosion whilst others can be used to release unwanted stress.

8. The main structure of an aero gas turbine engine is formed by a number of circular casings, ref. fig. 22-1, which are assembled and secured together by flanged joints and couplings located with dowels and tenons. These engines use curvic and hurth couplings to enable accurate concentricity of mating assemblies which in turn assist an airline operator when maintenance is required.

MANUFACTURING STRATEGY

9. Manufacturing is changing and will continue to change to meet the increasing demands of aeroengine components for fuel efficiency, cost and weight reductions and being able to process the materials required to meet these demands.

10. With the advent of micro-processors and extending the use of the computer, full automation of components considered for in house manufacture are implemented in line with supply groups manufac-turing strategy, all other components being resourced within the world-wide supplier network.

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H.P. SYSTEM
MODULE

I.P. AND L.P. TURBINE
MODULE
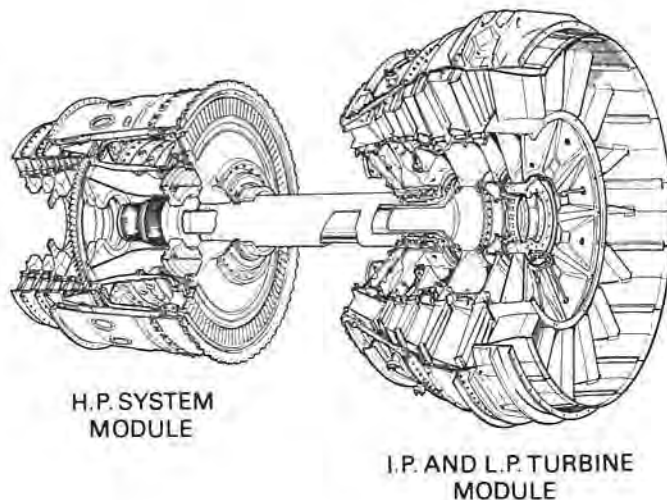
11. This automation is already applied in the manufacture of cast turbine blades with the seven cell and computer numerical controlled (C.N.C.) grinding centres, laser hardfacing and film cooling hole drilling by electro-discharge machining (E.D.M.). Families of turbine and compressor discs are produced in flexible manufacturing cells, employing automated guided vehicles delivering palletized components from computerized storage to C.N.C. machining cells that all use batch of one techniques. The smaller blades, with very thin airfoil sections, are produced by integrated broaching and 360 degree electrochemical machining (E.C.M.) while inspection and processing are being automated using the computer.

12. Tolerances between design and manufacturing are much closer when the design specification is matched by the manufacturing proven capability.

13. Computer Aided Design (C.A.D.) and Computer Aided Manufacture (CAM.) provides an equivalent link when engine components designed by C.A.D. can be used for the preparation of manufacturing drawings, programmes for numerically controlled machines, tool layouts, tool designs, operation

sequence, estimating and scheduling. Computer simulation allows potential cell and flow line manufacture to be proven before physical machine purchase and operation, thus preventing equipment not fulfilling their intended purpose.

14. Each casing is manufactured from the lightest material commensurate with the stress and temperatures to which it is subjected in service. For example, magnesium alloy, composites and materials of sandwich construction are used for air intake casings, fan casings and low pressure compressor casings, since these are the coolest parts of the engine. Alloy si eels are used for the turbine and nozzle casings where the temperatures are high and because these casings usually incorporate the engine rear mounting features. For casings subjected to intermediate temperatures i.e. by-pass duct and combustion outer casings, aluminium alloys and titanium alloys are used.

FORGING

15. The engine drive shafts, compressor discs, turbine discs and gear trains are forged to as near optimum shape as is practicable commensurate with non-destructive testing i.e., ultrasonic, magnetic particle and penetrant inspection. With turbine and compressor blades, the accurately produced thin airfoil sections with varying degrees of camber and twist, in a variety of alloys, entails a high standard of precision forging, ret. fig. 22-2. Nevertheless precision forging of these blades is a recognised practice and enables one to be produced from a shaped die with the minimum of further work.
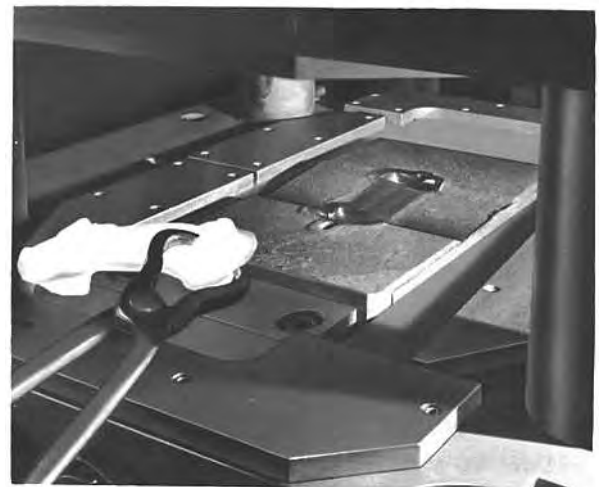
Fig. 22-2   Precision forging.

1. BASE PLATE
DOWELS
RECESS

2. CORE LOCATION MOULDING BOX
CORE LOCATION DOWEL

3. SAND CORES
CORE LOCATION MOULDING BOX
SAND CORES

4. WOOD PATTERN, MAIN MOULDING BOX AND RISER PATTERNS
RISER PATTERNS
WOOD PATTERN
POURER TUBE SPRUE PATTERN

5. MAIN MOULDING BOX
POURING TUBE PASSAGE
RUNNER CHANNEL

6. ASSEMBLING MAIN MOULD TO CORES
MAIN MOULD
CORES
MOULD LOCATION BOX

7. MOULD READY TO CAST
RISERS
POURING TUBE COLLAR

8. FINISHED COMPONENT AFTER MACHINING

Fig. 22-3   Method of producing an engine component by sand casting.
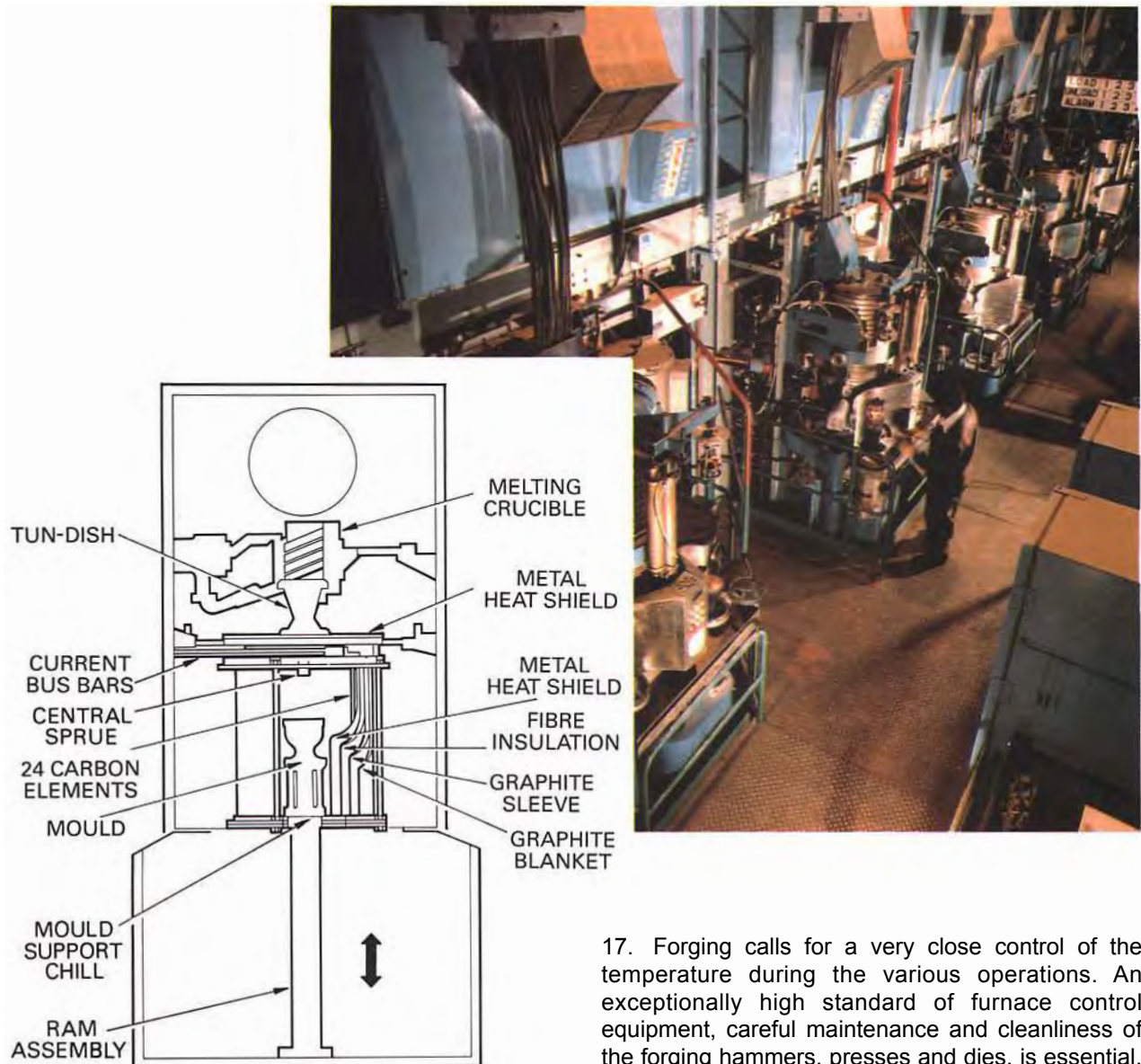
Fig. 22-4   Automatic investment casting.

16. The high operating temperatures at which the turbine discs must operate necessitates the use of nickel base alloys. The compressor discs at the rear end of the compressor are produced from creep-resisting steels, or even nickel base alloys, because of the high temperatures to which they are subjected. The compressor discs at the front end of the compressor are produced from titanium. The higher strength of titanium at the moderate operating temperatures at the front end of the compressor, together with its lower weight provides a considerable advantage over steel.

17. Forging calls for a very close control of the temperature during the various operations. An exceptionally high standard of furnace control equipment, careful maintenance and cleanliness of the forging hammers, presses and dies, is essential.

18. Annular combustion rings can be cold forged to exacting tolerances and surfaces which alleviates the need for further machining before welding together to produce the combustion casing.

19. H.P. compressor casings of the gas turbine engine are forged as rings or half rings which, when assembled together, form the rigid structure of the engine. They are produced in various materials, i.e., stainless steel, titanium and nickel alloys.

CASTING

20. An increasing percentage of the gas turbine engine is produced from cast components using

sand casting, ref. fig. 22-3, die casting and investment casting techniques; the latter becoming the foremost in use because of its capability to produce components with surfaces that require no further machining. It is essential that all castings are defect free by the disciplines of cleanliness during the casting process otherwise they could cause component failure.

21. All casting techniques depend upon care with methods of inspection such as correct chemical composition, test of mechanical properties, radiological and microscopic examination, tensile strength and creep tests.

22. The complexity of configurations together with accurate tolerances in size and surface finish is totally dependent upon close liaison with design, manufacturing, metallurgist, chemist, die maker, furnace operator and final casting.

23. In the pursuit of ever increasing performance, turbine blades are produced from high temperature

nickel alloys that are cast by the investment casting or lost wax' technique. Directionally solidified and single crystal turbine blades are cast using this technique in order to extend their cyclic lives.

24. Figure 22-4 illustrates automatic casting used in the production of equi-axed, directional solidified and single crystal turbine blades. The lost wax process is unparalleled in its ability to provide the highest standards of surface finish, repeatable accuracy and surface detail in a cast component. The increasing demands of the engine has manifested itself in the need to limit grain boundaries and provide complex internal passages. The moulds used for directional solidified and single crystal castings differ from conventional moulds in that they are open at both ends, the base of a mould forms a socketed bayonet fitting into which a chill plate is located during casting. Metal is introduced from the central sprue into the mould cavities via a ceramic filter. These and orientated seed crystals, if required, are assembled with the patterns prior to investment. Extensive automation is possible to ensure the wax patterns are coated with the shell material consistently by using robots. The final casting can also have their rises removed using elastic cut-off wheels driven from robot arms, ref. fig. 22-5.

FABRICATION

25. Major components of the gas turbine engine i.e. bearing housings, combustion and turbine casings, exhaust units, jet pipes, by-pass mixer units and low pressure compressor casings can be produced as fabricated assemblies using sheet materials such as stainless steel titanium and varying types of nickel alloys.

Fig. 22-5   Robot cut-off

26. Other fabrication techniques for the manufacture of the low pressure compressor wide chord fan blade comprise rolled titanium side panels assembled in dies, hot twisted in a furnace and finally hot creep formed to achieve the necessary configuration. Chemical milling is used to recess the centre of each panel which sandwiches a honeycomb core, both panels and the honeycomb are finally joined together using automated furnaces where an activated diffusion bonding process takes place, ref. fig. 22-6.

WELDING

27. Welding processes are used extensively in the fabrication of gas turbine engine components i.e., resistance welding by spot and seam, tungsten inert gas and electron beam are amongst the most widely used today. Care has to be taken to limit the distortion and shrinkage associated with these techniques.

Tungsten inert gas (T.I.G.) welding
28. The most common form of tungsten inert gas welding, fig, 22-7, in use is the direct current straight polarity i.e., electrode negative pole. This is widely used and the most economical method of producing high quality welds for the range of high strength/high temperature materials used in gas turbine engines. For this class of work, high purity argon shielding gas is fed to both sides of the weld and the welding torch nozzle is fitted with a gas lens to ensure maximum efficiency for shielding gas coverage. A consumable

Fig. 22-6 Wide chord fan blade construction.
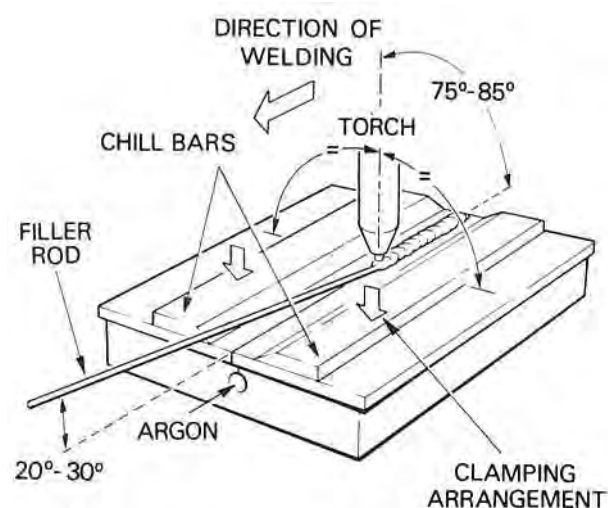
Fig. 22-7 Typical tungsten inert gas welding details.

Fig. 22-8   Tungsten inert gas welding.

four per cent thoriated tungsten electrode, together with a suitable non-contact method o! arc starting is used and the weld current is reduced in a controlled manner at the end of each weld to prevent the formation of finishing cracks. All welds are visually and penetrant inspected and in addition, welds associated with rotating parts i.e., compressor and/or turbine are radiologically examined to Quality Acceptance Standards. During welding operations and to aid in the control of distortion and shrinkage the use of an expanding fixture is recommended and, whenever possible, mechanised welding employed together with the pulsed arc technique is preferred. A typical T.I.G. welding operation is illustrated in fig. 22-8.

Electron beam welding (E.B.W.)
29. This system, which can use either low or high voltage, uses a high power density beam of electrons to join a wide range of different materials and of varying thickness. The welding machine ref. fig. 22-9, comprises an electron gun, optical viewing system, work chamber and handling equipment, vacuum pumping system, high or low voltage power supply and operating controls. Many major rotating assemblies for gas turbine engines are manufactured as single items in steel, titanium and nickel alloys and joined together i.e., intermediate and high pressure compressor drums. This technique allows
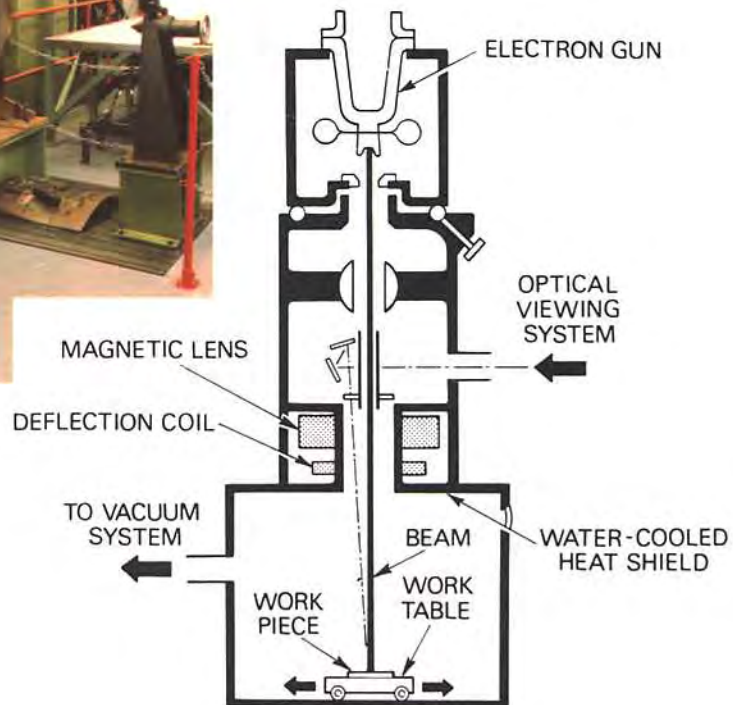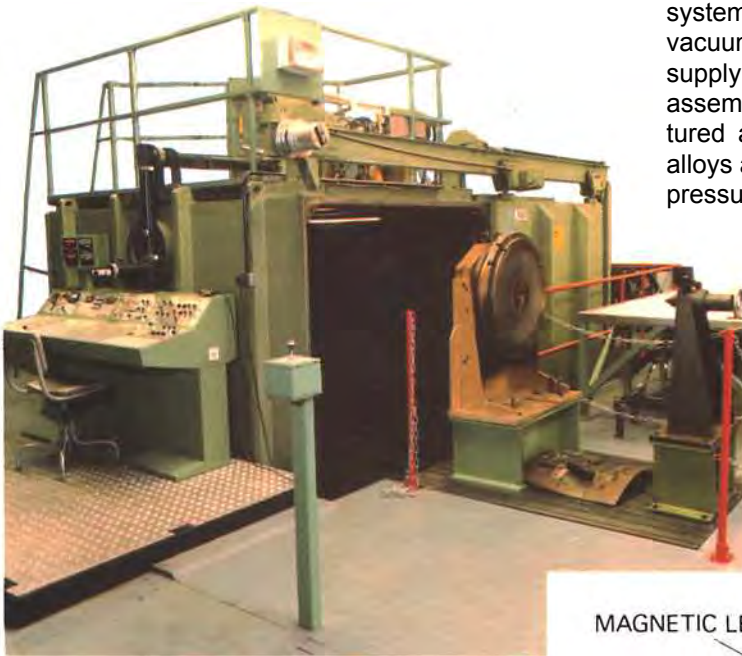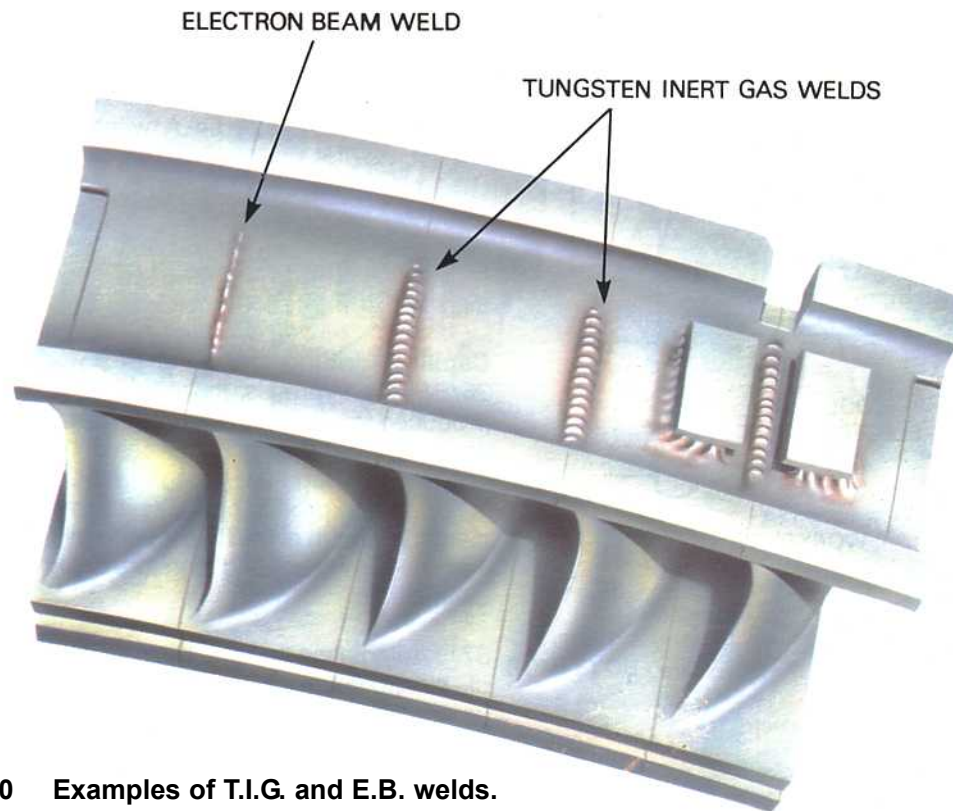

Fig. 22-9   Electron beam welding.

ELECTRON BEAM WELD

TUNGSTEN INERT GAS WELDS

Fig. 22-10   Examples of T.I.G. and E.B. welds.

design flexibility in that distortion and shrinkage are reduced and dissimilar materials, to serve quite different functions, can be homogeneously joined together. For example, the H.P. turbine stub shafts requiring a stable bearing steel welded to a material which can expand with the mating turbine disc. Automation has been enhanced by the application of computer numerical control (C.N.C.) to the work handling and manipulation. Seam tracking to ensure that the joint is accurately followed and close loop under bead control to guarantee the full depth of material thickness is welded. Focus of the beam is controlled by digital voltmeters. See fig. 22-10 for weld examples.

ELECTRO-CHEMICAL MACHINING (E.C.M.)

30.   This type of machining employs both electrical and chemical effects in the removal of metal. Chemical forming, electro-chemical drilling and elec-trolytic grinding are techniques of electro-chemical machining employed in the production of gas turbine engine components.

31.   The principle of the process is that when a current flows between the electrodes immersed in a solution of salts, chemical reactions occur in which metallic ions are transported from one electrode to

another (fig. 22-11). Faraday's law of electrolysis explains that the amount of chemical reaction produced by a current is proportional to the quantity of electricity passed.

32.   In chemical forming, (fig. 22-11), the tool electrode (the cathode) and the workpiece (the anode) are connected into a direct current circuit. Electrolytic solution passes, under pressure, through the tool electrode and metal is removed from the work gap by electrolytic action. A hydraulic ram advances the tool electrodes into the workpiece to form the desired passage.

33.   Electrolytic grinding employs a conductive wheel impregnated with abrasive particles. The wheel is rotated close to the surface of the workpiece, in such a way that the actual metal removal is achieved by electro-chemical means. The by-products, which would inhibit the process, are removed by the sharp particles embodied in the wheel.

34.   Stem drilling and capillary drilling techniques are used principally in the drilling of small holes, usually cooling holes, such as required when producing turbine blades.

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1. Electrolytic solution passes under pressure through tool electrode
2. Tool electrode and workpiece are connected into D.C. circuit
3. Ram advances tool electrode into workpiece
4. Metal is removed in work gap by continuous electrolytic action

Fig. 22-11   Electro-chemical machining.

Stem drilling

35. This process consists of tubes (cathode) produced from titanium and suitably insulated to ensure a reaction at the tip. A twenty per cent solution of nitric acid is fed under pressure onto the blade producing holes generally in the region of 0.026 in. diameter. The process is more speedy in operation than electro-discharge machining and is capable of drilling holes up to a depth two hundred times the diameter of the tube in use.

Capillary drilling

36. Similar in process to stem drilling but using tubes produced from glass incorporating a core of platinum wire (cathode). A twenty per cent nitric acid solution is passed through the tube onto the workpiece and is capable of producing holes as small as 0.009 in. diameter. Depth of the hole is up to forty times greater than the tube in use and therefore determined by tube diameter.

37. Automation has also been added to the process of electro-chemical machining (E.C.M.) with the introduction of 360 degree E.G. machining of small compressor blades, ref. fig. 22-12. For some blades of shorter length airfoil, this technique is more cost effective than the finished shaped airfoil when using precision forging techniques. Blades produced by E.C.M. employ integrated vertical broaching machines which take pre-cut lengths of bar material, produce the blade root feature, such as a fir-tree, and then by using this as the location, fully E.C.M. from both sides to produce the thin airfoil section in one operation.

ELECTRO-DISCHARGE MACHINING (E.D.M.)

38. This type of machining removes metal from the workpiece by converting the kinetic energy of electric sparks into heat as the sparks strike the workpiece.

39. An electric spark results when an electric potential between two conducting surfaces reaches the point at which the accumulation of electrons has acquired sufficient energy to bridge the gap between the two surfaces and complete the circuit. At this point, electrons break through the dielectric medium between the conducting surfaces and, moving from negative (the tool electrode) to positive (the workpiece), strike the latter surface with great energy; fig, 22-13 illustrates a typical spark erosion circuit.

40. When the sparks strike the workpiece, the heat is so intense that the metal to be removed is instantaneously vaporized with explosive results. Away
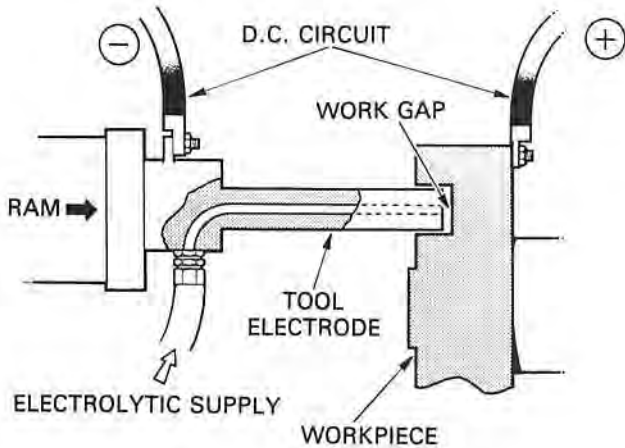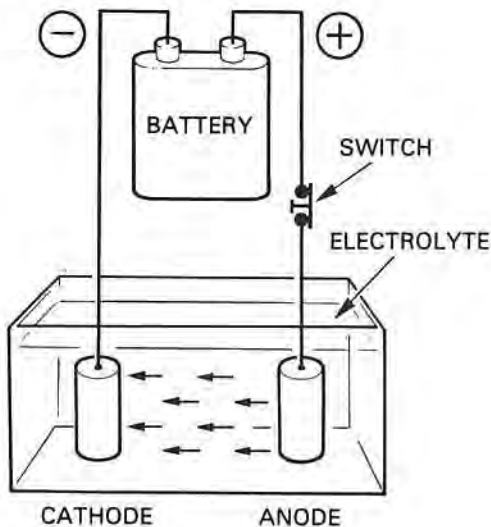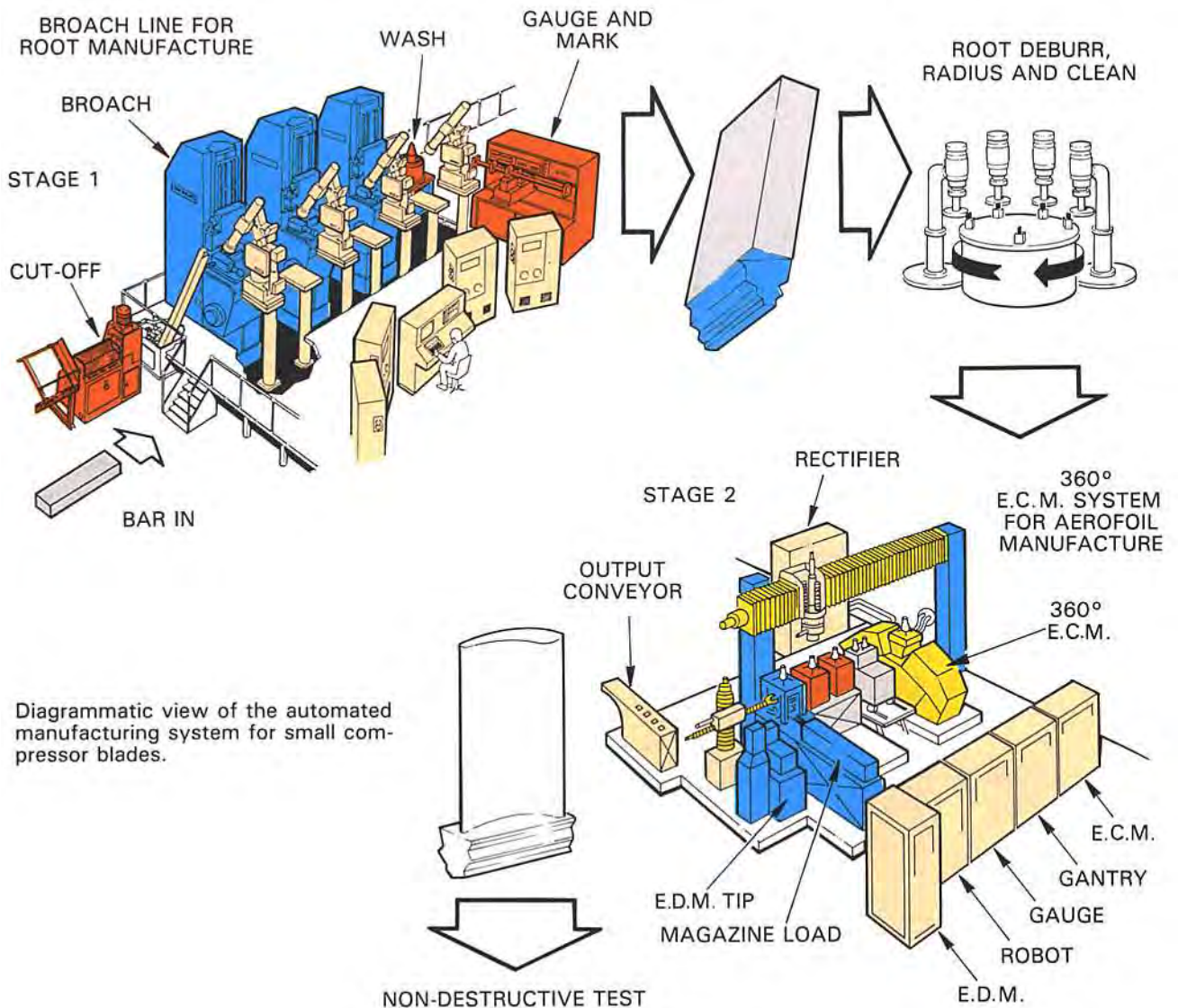
Fig. 22-12   Typical automated manufacture of compressor blades.

from the actual centre of the explosion, the metal is torn into fragments which may themselves be melted by the intense heat. The dielectric medium, usually paraffin oil. pumped into the gap between the tool electrode and the workpiece, has the tendency to quench the explosion and to sweep away metallic vapour and molten particles.

41.   The amount of work that can be effected in the system is a function of the energy of the individual sparks and the frequency at which they occur.

42.   The shape of the tool electrode is a mirror image of the passage to be machined in the workpiece and, to maintain a constant work gap, the electrode is fed into the workpiece as erosion is effected.
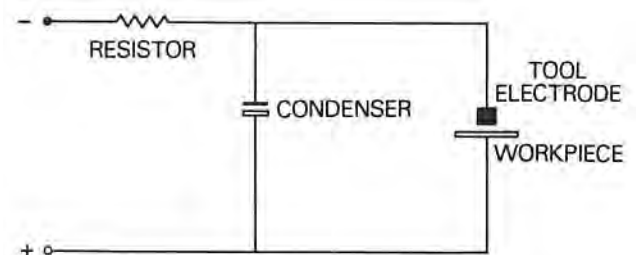
Fig. 22-13   Electro-discharge machining circuit.

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COMPOSITE MATERIALS AND SANDWICH CASINGS

43. High power to weight ratio and low component costs are very important considerations in the design of any aircraft gas turbine engine, but when the function of such an engine is to support a vertical take-off aircraft during transition, or as an auxiliary power unit, then the power to weight ratio becomes extremely critical.

44. In such engines, the advantage of composite materials allows the designer to produce structures in which directional strengths can be varied by directional lay-up of fibres according to the applied loads.

45. Composite materials have and will continue to replace casings which, in previous engines, would have been produced in steels or titanium. By-pass duct assemblies comprising of three casings are currently being produced up to 4ft-7in. in diameter and 2ft-0in in length using pre-cured composite materials for the casing fabric. Flanges and mounting bosses are added during the manufacturing process, which are then drilled for both location and machined

for peripheral feature attachment on C.N.C. machining centres, which at one component load, completely machine all required features. Examples of composite material applications are illustrated in fig. 22-14.

46. Conventional cast and fabricated casings and cowlings are also being replaced by casings of sandwich construction which provide strength allied with lightness and also act as a noise suppression medium. Sandwich construction casings comprise a honeycomb structure of aluminium or stainless steel interposed between layers of dissimilar material. The materials employed depend upon the environment in which they are used.

INSPECTION

47. During the process of manufacture, component parts need to be inspected to ensure defect free engines are produced. Using automated machinery and automated inspection, dimensional accuracy is maintained by using multi-directional applied probes that record sizes and position of features. The C.N.C. inspection machine can inspect families of components at pre-determined allotted intervals

Fig. 22-14   Some composite material applications.

AUTOMATED GUIDED VEHICLE
(A.G.V.)

Legend:
- A.G.V. Track
- Processing
- Surface finishing
- Stores and handling
- Quality control
- Machining
- Computer control
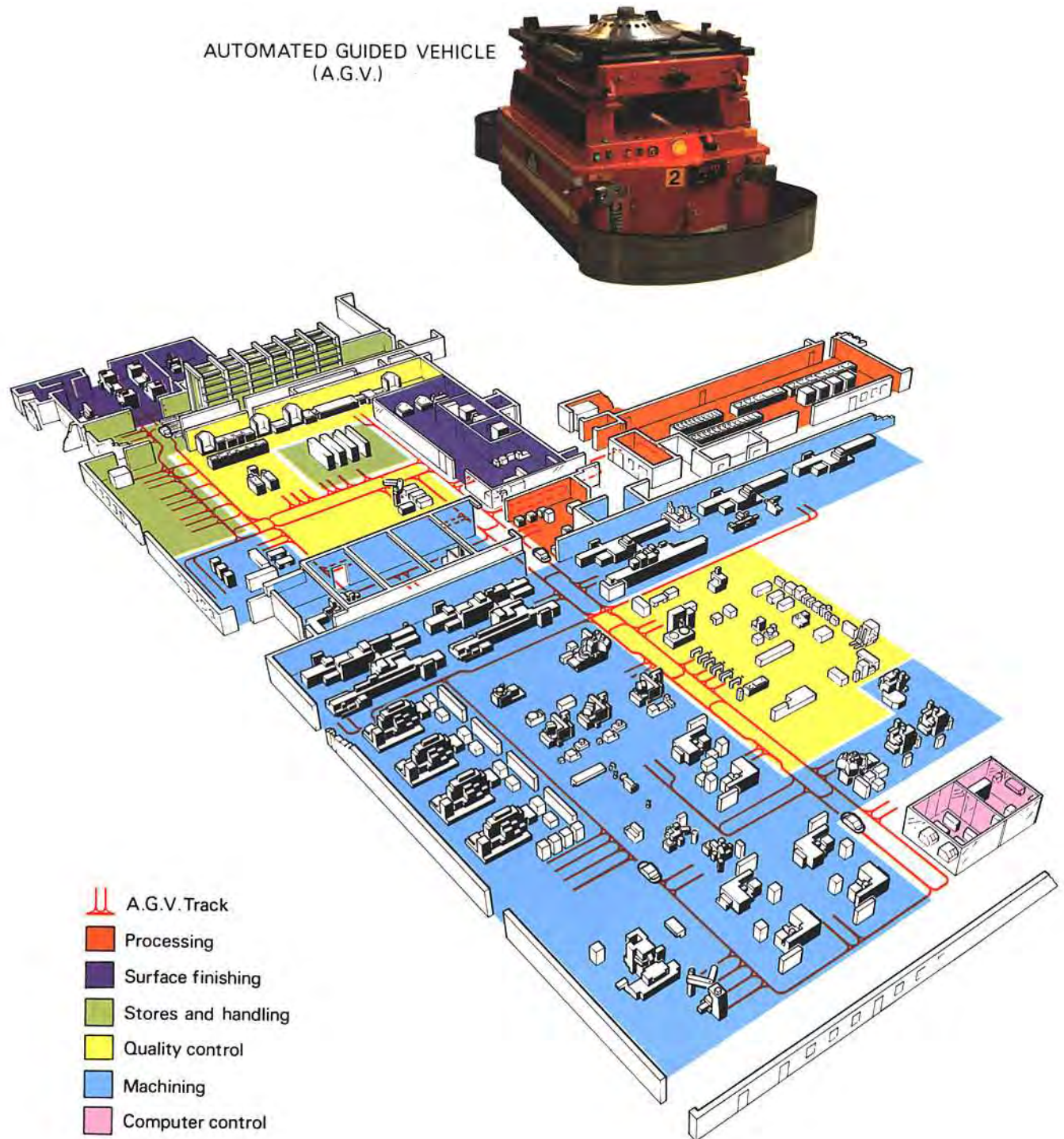
Fig. 22-15   Advanced integrated manufacturing system (A.I.M.S.).

without further operator intervention. In the chip machining (i.e., turning, boring, milling etc.) and metal forming processes C.N.C. machine tools enable consistency of manufacture which can be statistically inspected i.e., one in ten. Component integrity is achieved by use of ultrasonic, radiological, magnetic particle and penetrant inspection techniques, as well as electrolytic and acid etching to ensure all material properties are maintained to both laboratory and quality acceptance standards.

Rolls-Royce Turbomeca RTM 332 Turboshaft
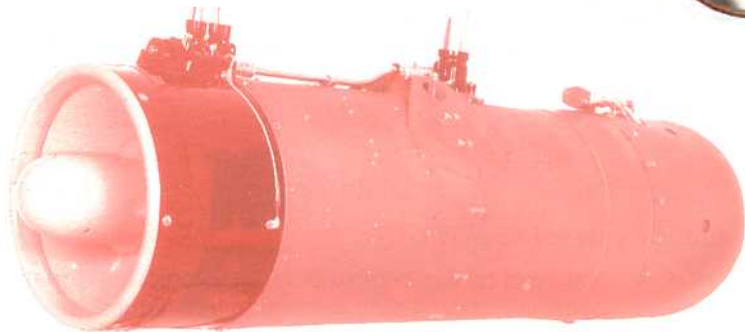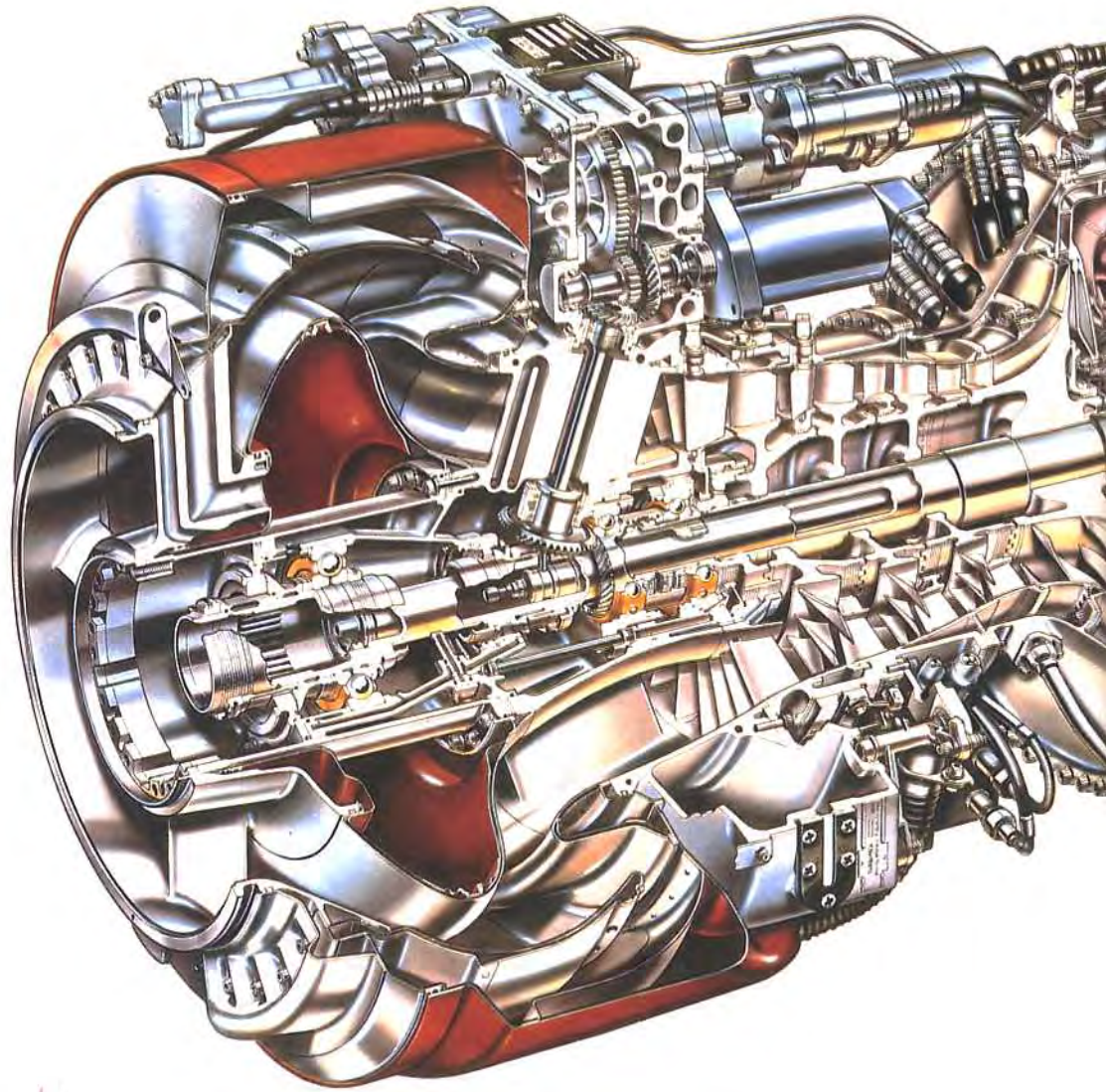
Rolls-Royce RB 93 Soar

Developed as a lightweight expendable engine for winged missiles, the Soar first ran in 1952. At that time it had the best thrust to weight ratio of any gas turbine in the world, producing 1810 lb thrust from only 275 lb total weight. The Soar was flight tested in a Gloster Meteor with one engine on each wingtip. It was also built under licence in the USA as the J81 for the XQ4 supersonic drone.

23: Power plant installation
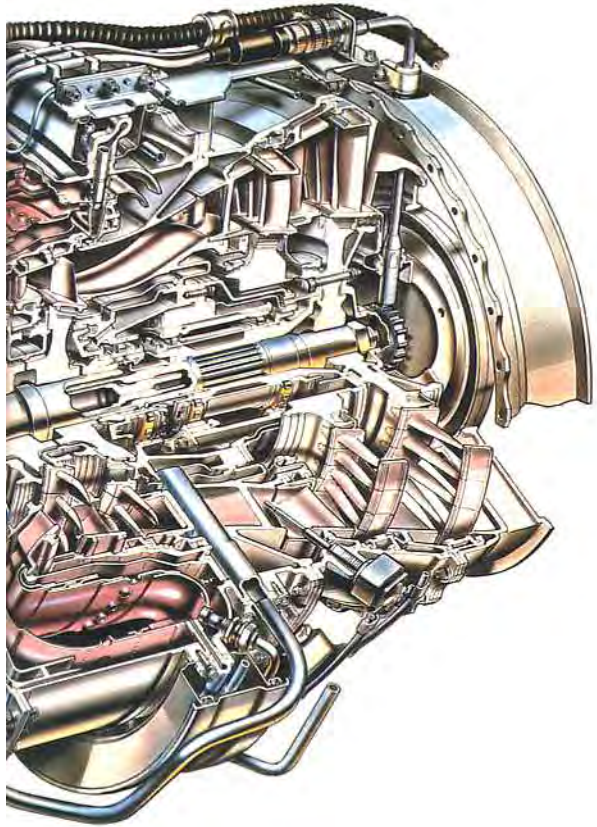
Contents Page

INTRODUCTION

1. When a gas turbine engine is installed in an aircraft it usually requires a number of accessories fitting to it and connections made to various aircraft systems. The engine, jet pipe and accessories, and in some installations a thrust reverser, must be suitably cowled and an air intake must be provided for the compressor, the complete installation forming the aircraft power plant.

POWER PLANT LOCATION

2. The power plant location and aircraft configuration are of an integrated design and this depends upon the duties that the aircraft has to perform. Turbo-jet engine power plants may be in the form of pod installations that are attached to the wings by pylons (fig. 23-1), or attached to the sides of the rear fuselage by short stub wings (fig. 23-2), or they may be buried in the fuselage or wings. Some aircraft have a combination of rear fuselage and tail-mounted power plants, others, as shown in fig. 23-3, have wing-mounted pod installations with a third engine buried in the tail structure. Turbo-propeller engines, however, are normally limited to installation in the wings or nose of an aircraft.
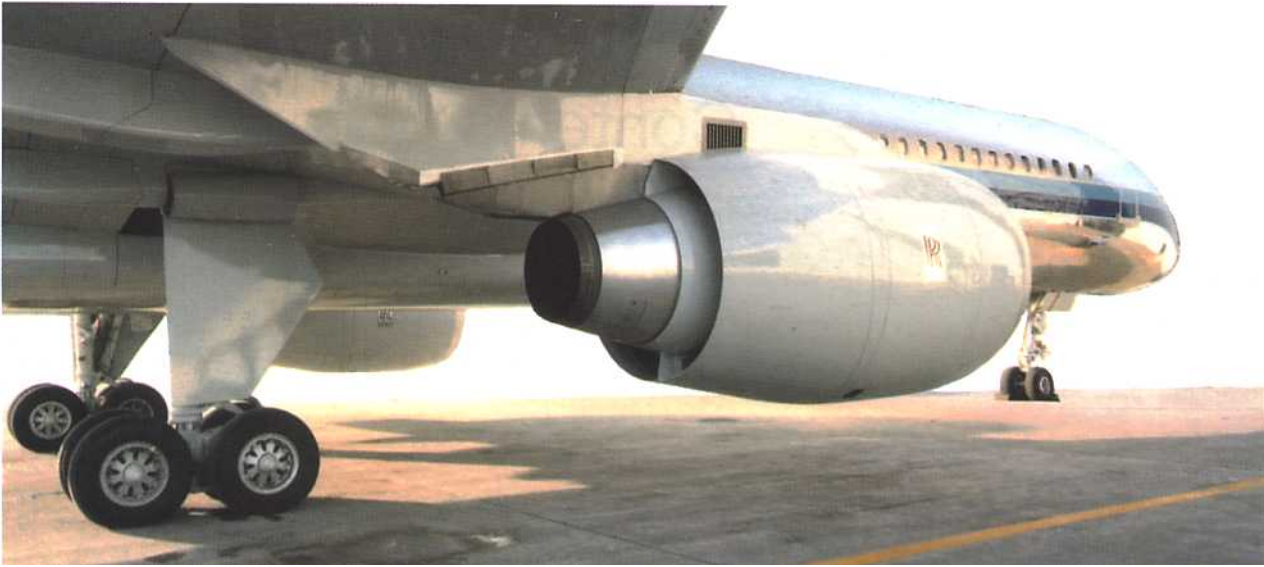
Fig. 23-1   Wing-mounted pod installation.

Fig. 23-2   Fuselage - mounted pod installation.

Fig. 23-3   Tail and wing-mounted pod installation.

3. The position of the power plant must not affect the efficiency of the air intake, and the exhaust gases must be discharged clear of the aircraft and its control surfaces. Any installation must also be such that it produces the minimum drag effect.

4. Power plant installations are numbered from left to right when viewed from the rear of the aircraft.

5. Supersonic aircraft usually have the power plants buried in the aircraft for aerodynamic reasons. Vertical lift aircraft can use either the buried installation or the podded power plant, or in some instances both types may be combined in one aircraft (Part 18).

AIR INTAKES

6. The main requirement of an air intake is that, under all operating conditions, delivery of the air to the engine is achieved with the minimum loss of energy occurring through the duct. To enable the compressor to operate satisfactorily, the air must reach the compressor at a uniform pressure distributed evenly across the whole inlet area.

7. The ideal air intake for a turbo-jet engine fitted to an aircraft flying at subsonic or low supersonic speeds, is a short, pitot-type circular intake (fig. 23-4). This type of intake makes the fullest use of the ram effect on the air due to forward speed, and suffers the minimum loss of ram pressure with changes of aircraft attitude. However, as sonic speed

Fig. 23-4   Pitot-type intake.

is approached, the efficiency of this type of air intake begins to fall because of the formation of a shock wave at the intake lip.

8. The pitot-type intake can be used for engines that are mounted in pods or in the wings, although the latter sometimes require a departure from the circular cross-section because of the wing thickness (fig. 23-5).

Fig. 23-5   Wing leading edge intakes.

Fig. 23-6   Single engined aircraft with fuselage intakes.

9. Single engined aircraft sometimes use a pilot-type intake; however, because this generally involves the use of a long duct ahead of the compressor, a divided type of intake on each side of the fuselage is often used (fig. 23-6).

10. The disadvantage of the divided type of air intake is that when the aircraft yaws, a loss of ram pressure occurs on one side of the intake, as shown in fig. 23-7, causing an uneven distribution of airflow into the compressor.

Large loss due to
separation of
boundary layer air
from fuselage

YAW

Small loss due to build up of
boundary layer air on fuselage
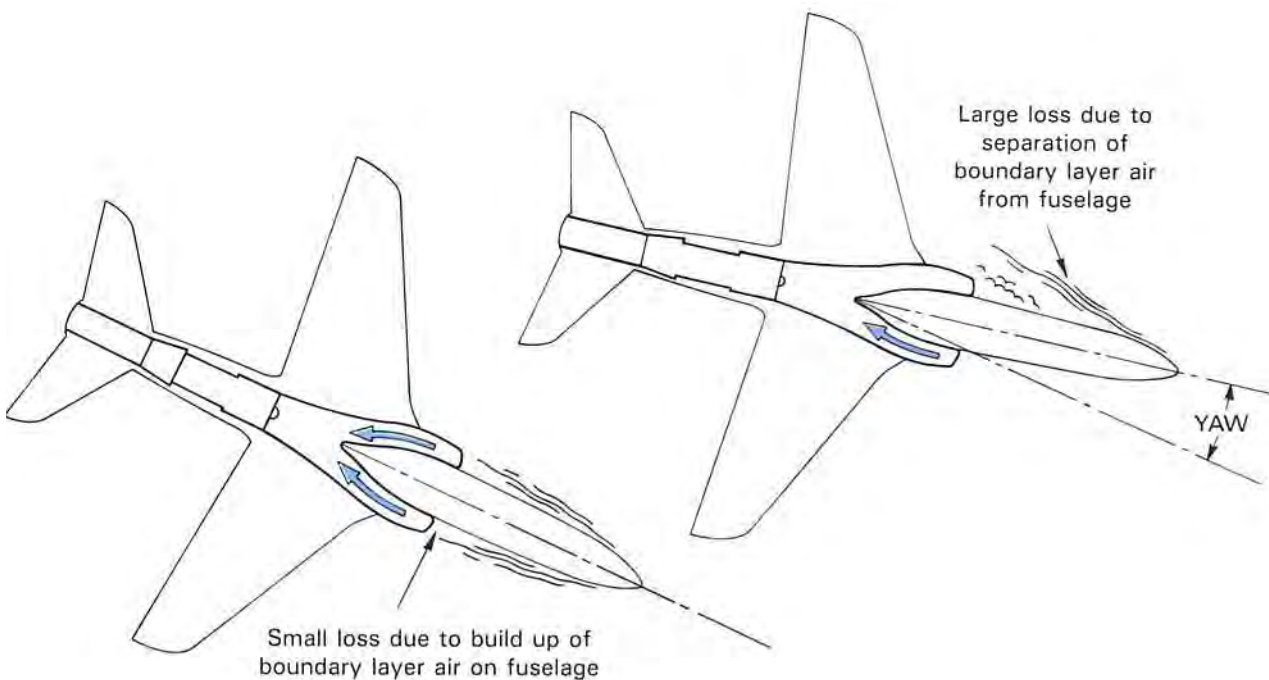
Fig. 23-7   Loss of ram pressure in divided intakes.

11.   At higher supersonic speeds, the pitot type of air intake is unsuitable due to the severity of the shockwave that forms and progressively reduces the intake efficiency as speed increases. A more suitable type of intake for these higher speeds is known as the external/internal compression intake (fig. 23-8). This type of intake produces a series of mild shock waves without excessively reducing the intake efficiency.

12.   As aircraft speed increases still further, so also does the intake compression ratio and, at high Mach numbers, it is necessary to have an air intake that has a variable throat area and spill valves to accommodate

WEDGE LOWERED (Throat area reduced)

SUPERSONIC CONDITION

DUMP VALVE OPEN

SPILL VALVE OPEN (To vent excess air flow)

WEDGE RETRACTED (Throat area increased)

SUBSONIC CONDITION

DUMP VALVE (Used as scoop to increase airflow)

SPILL VALVE OPEN (To prevent turbulence)

Fig. 23-9   Variable throat area intake.

and control the changing volumes of air (fig. 23-9). The airflow velocities encountered in the higher speed range of the aircraft are much higher than the engine can efficiently use; therefore, the air velocity must be decreased between the intake and the engine air inlet. The angle of the variable throat area intake automatically varies with aircraft speed and positions the shock wave to decrease the air velocity at the engine inlet and maintain maximum pressure recovery within the inlet duct. However, continued development enables this to be achieved by careful design of the intake and ducting. This, coupled with auxiliary air doors to permit extra air to be taken in under certain engine operating conditions, allows the airflow to be controlled without the use of variable geometry intakes. The fuselage intakes shown in fig. 23-10 are of the variable throat area type.

Fig. 23-8   External/internal compression intake.

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Fig. 23-10   Fuselage intakes.

ENGINE AND JET PIPE MOUNTINGS

13. The engine is mounted in the aircraft in a manner that allows the thrust forces developed by the engine to be transmitted to the aircraft main structure, in addition to supporting the engine weight and carrying any flight loads. Because of the wide variations in the temperature of the engine casings, the engine is mounted so that the casings can expand freely in both a longitudinal and a radial direction. Types of engine mountings, however, vary to suit the particular installation requirement. Turbo-jet engines are usually either side mounted or underslung as illustrated in fig. 23-11. Turbo-propeller engines are mounted forward on a tubular framework as illustrated in fig. 23-13.
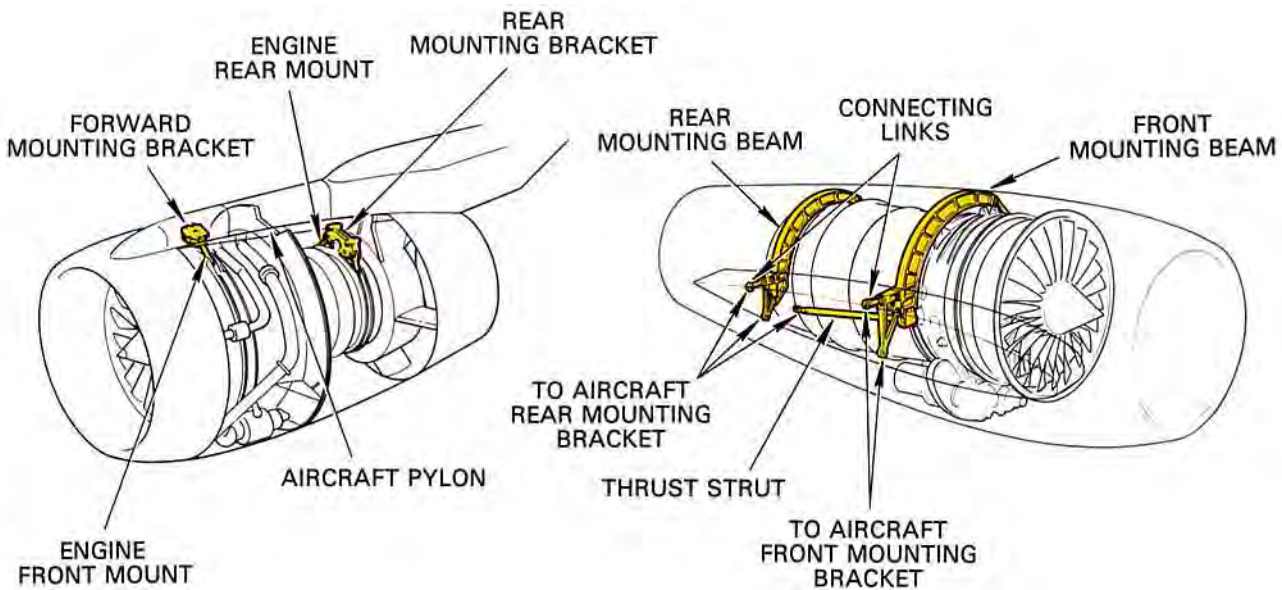

Fig. 23-11   Typical turbo-jet engine mountings.

Fig. 23-12 Engine accessibility, turbo-fan engine.

14. The jet pipe is normally attached to the rear of the engine and supported by the engine mountings. In some installations, particularly where long jet pipes are employed, an additional mounting is provided, usually in the form of small rollers attached to each side of the jet pipe. The rollers locate in airframe-mounted channels and support the weight of the jet pipe, whilst still allowing it to freely expand in a longitudinal direction.

ACCESSORIES

15. An aircraft power plant installation generally includes a number of accessories that are electrically operated, mechanically driven or driven by high pressure air.

16. Electrically operated accessories such as engine control actuators, amplifiers, air control valves and solenoids, are supplied with power from the aircraft electrical system or an engine driven dedicated electrical generator.

17. Mechanically driven units, such as generators, constant speed drive units, hydraulic pumps, low and high pressure fuel pumps, and engine speed signalling, measuring or governing units are driven from the engine through internal and external gearboxes (Part 7).

18. Air-driven accessories, such as the air starter and possibly the thrust reverser, afterburner and

Fig. 23-13 Engine accessibility, turbo-propeller engine.

water injection pumps, are driven by air tapped from the engine compressor. Air conditioning and cabin pressurization units may have a separate air-driven compressor or use air direct from the engine compressor. The amount of air that is taken for all accessories and services must always be a very small percentage of the total airflow, as it represents a thrust or power loss and an increase in specific fuel consumption.

COWLINGS

19. Access to an engine mounted in the wing or fuselage is by hinged doors; on pod and turbo-propeller installations the main cowlings are hinged. Access for minor servicing is by small detachable or hinged panels. All fasteners are of the quick-release type.

20. A turbo-propeller engine, or a turbo-jet engine mounted in a pod, is usually far more accessible than a buried engine because of the larger area of hinged cowling that can be provided. The accessibility of a podded turbo-fan engine is shown in fig, 23-12 and that of a turbo-propeller engine is shown in fig, 23-13.

I.A.E. International Aero Engines V2500
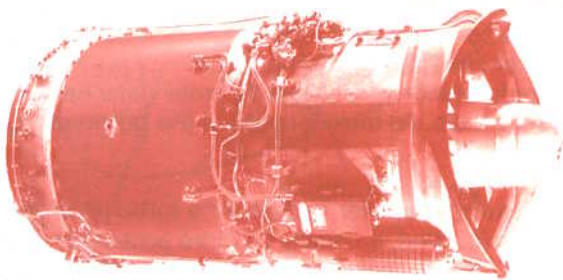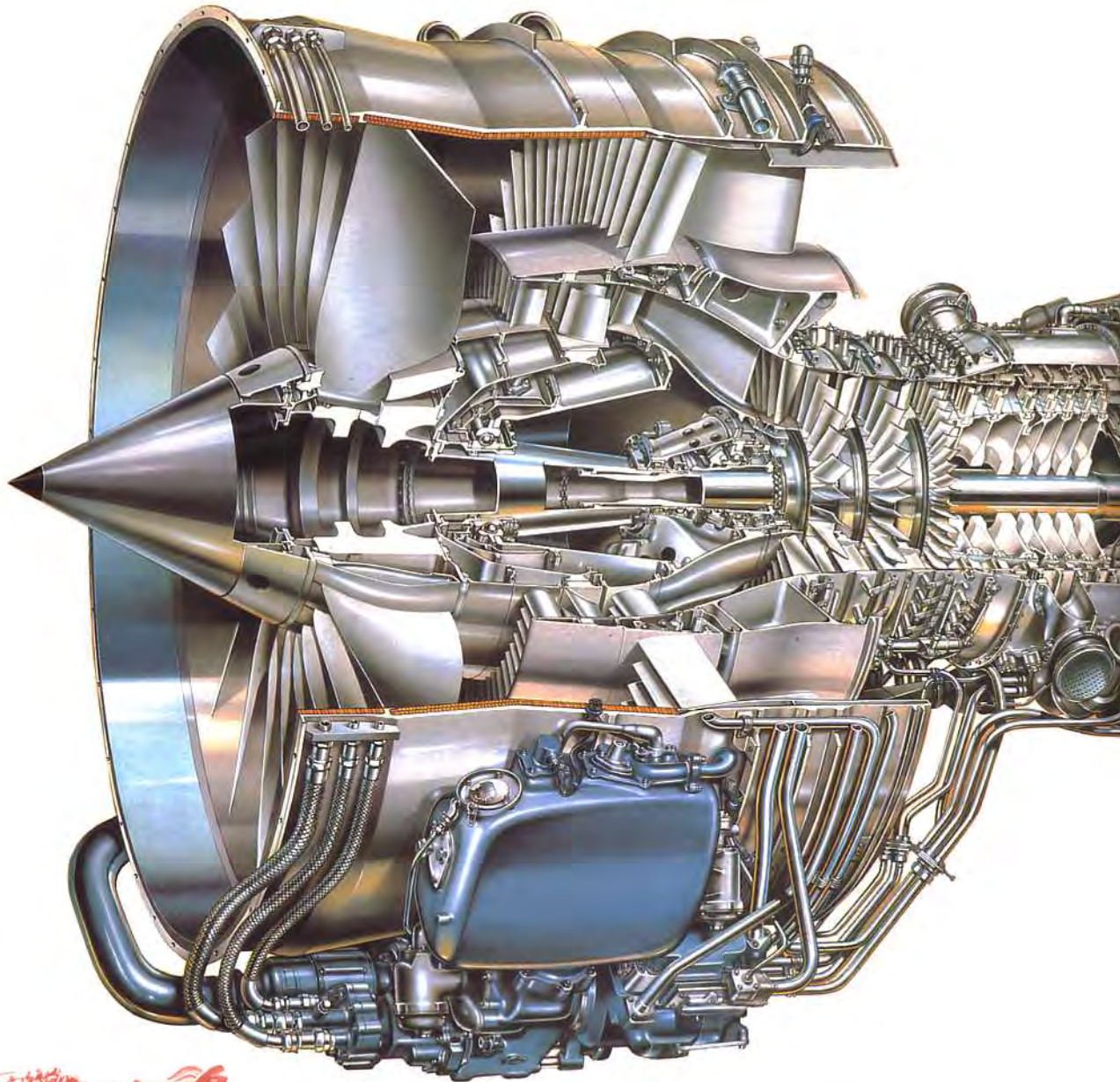
Rolls-RoyceRB 162

Design of the RB162 began in 1959 using experience, gained on the RB108, of simplified lightweight constructions and systems. These measures, combined with lightweight materials, served to keep the engine weight down to 280 lb; giving a thrust to weight ratio of 16:1. First run in December 1961, the RB162 was used to provide lift for a variety of VTOL research aircraft.

24: Maintenance

Contents
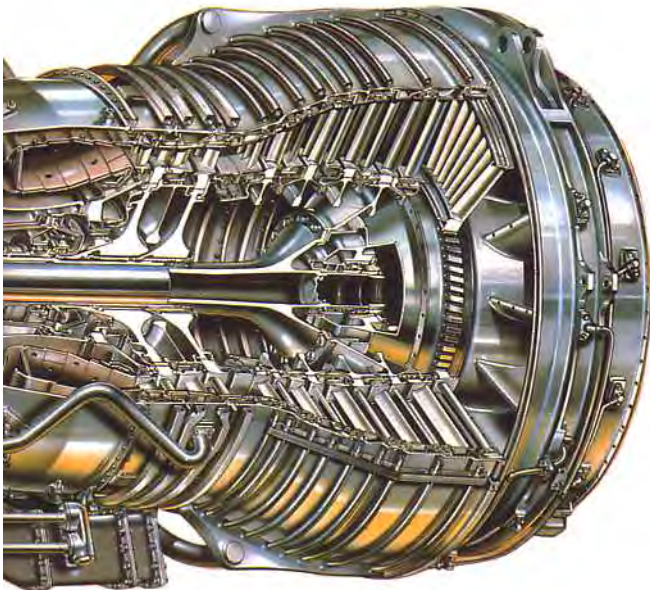
INTRODUCTION

1. Maintenance covers both the work that is required to maintain the engine and its systems in an airworthy condition while installed in an aircraft (on-wing or line maintenance) and the work required to return the engine to airworthy condition when removed from an aircraft (overhaul or shop maintenance). On-wing maintenance is covered in this Part; overhaul is covered in Part 25.

2. Because many aspects of maintenance are subject to the approval of a recognized authority, it should be fully understood that the information given in this Part is of a general nature and is not intended as a substitute for any official instructions.

3. The comprehensive instructions covering the actual work to be done to support scheduled maintenance (para. 8) and unscheduled maintenance (para. 10) are contained in the aircraft maintenance manual. Both this publication, and the aircraft maintenance schedule mentioned in para. 8, are based on manufacturers' recommendations and are approved by the appropriate airworthiness authority.

4. The maximum time an engine can remain installed in an aircraft (engine life) is limited to a fixed period agreed between the engine manufacturer and airworthiness authority. On some engines this period is referred to as the time between overhaul (T.B.O.) and on reaching it the engine is removed for complete overhaul.

5. Because the T.B.O. is actually determined by the life of one or two assemblies within the engine, during overhaul, it is generally found that the other assemblies are mechanically sound and fit to continue in service for a much longer period. Therefore, with the introduction of modular engines and the improved inspection and monitoring techniques available, the T.B.O. method on limiting the engine's life on-wing has been replaced by the 'on-condition' method.

6. Basically this means that a life is not declared for the total engine but only for certain parts of the engine. On reaching their life limit, these parts are replaced and the engine continues in service, the remainder of the engine being overhauled 'on condition', Modular constructed engines are particularly suited to this method, as the module containing a life limited part can be replaced by a similar module and the engine returned to service with minimum delay, The module is then disassembled for life limited part replacement, repair or complete overhaul as required.

ON-WING MAINTENANCE

7. On-wing maintenance falls into two basic categories: scheduled maintenance and unscheduled maintenance.

Scheduled maintenance
8. Scheduled maintenance embraces the periodic and recurring checks that have to be effected in accordance with the engine section of the appropriate aircraft maintenance schedule. These checks range from transit items, which do not normally entail opening cowls, to more elaborate checks within specified time limits, usually calculated in aircraft flying hours and phased with the aircraft check cycle.

9. Continuous 'not-exceed-limit' maintenance, whereby checks are carried out progressively and as convenient within given time limits rather than at specific aircraft check periods, has been widely adopted to supersede the check cycle. With the progressive introduction of condition monitoring devices (para. 11) of increased efficiency and reliability, a number of traditionally accepted scheduled checks may become unnecessary. Extracts from a typical maintenance schedule are shown in fig. 24-1.

Unscheduled maintenance
10. Unscheduled maintenance covers work necessitated by occurrences that are not normally related to time limits, e.g. bird ingestion, a strike by lightning, a crash or heavy landing, Unscheduled work required may also result from malfunction, trouble shooting, scheduled maintenance, and occasionally, manufacturers' specific recommendations. This type of maintenance usually involves rectification adjustment or replacement.

CONDITION MONITORING

11. Condition monitoring devices must give indication of any engine deterioration at the earliest possible stage and also enable the area or module in which deterioration is occurring to be identified. This facilitates quick diagnosis, which can be followed by scheduled monitoring and subsequent programmed rectification at major bases, thereby avoiding in-flight shut-down, with resultant aircraft delay, and minimizing secondary damage. Monitoring devices and facilities can be broadly categorized as flight deck indicators, in-flight recorders and ground indicators.

Flight deck indicators
12. Flight deck indicators are used to monitor engine parameters such as thrust or power, r.p.m., turbine gas temperature, oil pressure and vibration. Most of the indicators used are described in Part 12. Other devices, however, may be used and these include:

Accelerometers for more reliable and precise vibration monitoring.
Radiation pyrometers for direct measurement of turbine blade temperature.
Return oil temperature indicators.
Remote indicators for oil tank content.
Engine surge or stall detectors.
Rub indicators to sense eccentric running of rotating assemblies.

In-flight recorders
13. Selected engine parameters are recorded, either manually or automatically, during flight. The recordings are processed and analyzed for significant trends indicative of the commencement of failure. An in-flight recording device that may be used is the time/temperature cycle recorder. The purpose of this device is to accurately record the engine time spent operating at critical high turbine gas temperatures, thus providing a more realistic measure of 'hot-end' life than that provided by total engine running hours.

14. Automatic systems (Part 12) known as aircraft integrated data systems (A.I.D.S.) are able to record parameters additional to those normally displayed e.g. certain pressures, temperatures and flows.

15. Many of the electronic components used in modern control systems have the ability to monitor

ITEM	NOT EXCEED LIMIT	REQUIREMENT
Engine oil tank	Flight termination	Check oil level. Replenish as necessary. Record amount taken
Cowls	Transit	Check the pod cowls for damage and external evidence of fuel and oil leaks
Caps and access panels	Transit	Check secure
Engine intake	Transit	Check clear. Free from damage and loose objects
Turbine and exhaust collector	Transit	Visually inspect for signs of damage and metal deposits
Engine intake	25 hours	Visually inspect front of engine through air intake for signs of damage paying particular attention to intake guide vanes and leading stage rotor blades
Turbine and exhaust collector	25 hours	Visually inspect L.P.2 turbine blades, nozzle guide vanes and mixer unit for cracking and damage by viewing from rear using a strong spot light
Fuel filter	125 hours	Drain sample and check for water contamination
Magnetic chip detector	200 hours	Remove and inspect
Igniter plugs	200 hours	Audibly check operation
Oil pressure filter	600 hours	Check and clean/renew filter element
Fuel filter	800 hours	Remove filter element, check and renew

Fig. 24-1   A typical maintenance schedule (extracts).

their own and associated component operation. Any fault detected is recorded in its built-in memory for subsequent retrieval and rectification by the ground crew. On aircraft that feature electronic engine parameter flight deck displays (Part 12) certain faults are also automatically brought to the flight crew's attention.

Ground indicators

16.   The devices used or checked on the ground, as distinct from those used or checked in flight, may conveniently be referred to as ground indicators; this title is also taken to embrace instruments used for engine internal inspection.

17.   Internal viewing instruments can be either flexible or rigid, designed either for end or angled viewing and, in some instances adaptable for still or video photography which may be linked to closed circuit television. These instruments are used for examining and assessing the condition of the compressor and turbine assemblies, nozzle guide vanes (fig. 24-2) and combustion system, and can be inserted through access ports located at strategic points in the engine main casings.

18.   The engine condition indicators include magnetic chip detectors, oil filters and certain fuel filters. These indicators are frequently used to substantiate indications of failures shown by flight deck monitoring and in-flight recordings. For instance, inspection of the oil filters and chip detectors can reveal deposits from which experienced personnel can recognize early signs of failure. Some maintenance organizations progressively log oil filter
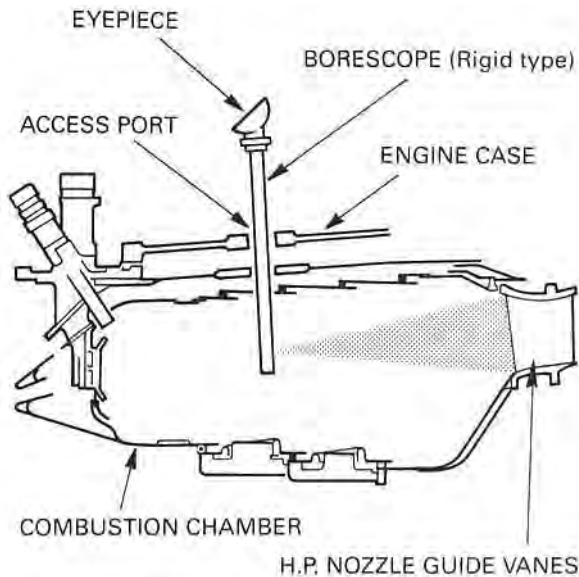
Fig. 24-2   Inspection of H.P. nozzle guide vanes.

and magnetic chip detector history and catalogue the yield of particles. Fuel filters may incorporate a silver strip indicator that detects any abnormal concentration of sulphur in the fuel.

MAINTENANCE PRECAUTIONS

19.   During engine maintenance, it is necessary to observe certain precautions. The ignition system is potentially lethal and, therefore, before any work is done on the high energy ignition units, igniter plugs or harness, the low tension supply to the units must be disconnected and at least one minute allowed to elapse before disconnecting the high tension lead. Similarly, before carrying out work on units connected to the electrical system, the system must be made safe, either by switching off power or by tripping and tagging appropriate circuit breakers. With some installations, the isolation of certain associated systems may be required.

20.   When the oil system is being replenished, care must be taken that no oil is spilt. If any oil is accidentally spilt, it should be cleaned off immediately as it is injurious to paintwork and to certain rubber compounds such as could be found in the electrical harnesses, Oil can also be toxic through absorption if allowed to come into contact with the human skin for prolonged periods. Care should be taken not to overfill the oil system; this may easily occur if the aircraft is not on level ground or if the engine has been stationary for a long period before the oil level is checked.

21.   Before an inspection of the air intake or exhaust system is made it must be ascertained that there Is no possibility of the starter system being operated or the ignition system being energized.

22.   A final inspection of the engine, air intake and exhaust system, must always be made after any repair, adjustment or component change, to ensure that no loose items, no matter how small, have been left inside. Unless specific local instructions rule otherwise, air intake and exhaust blanks or covers should be fitted when engines are not running.

TROUBLE SHOOTING

23.   The procedure for locating a fault is commonly referred to as trouble shooting, and the requirement under this procedure is for quick and accurate diagnosis with the minimum associated work and the prevention of unnecessary unit or engine removals.

24.   The basic principle of effective trouble shooting is to clearly define and interpret the reported symptom and then proceed to a logical and systematic method of diagnosis (fig. 24-3).
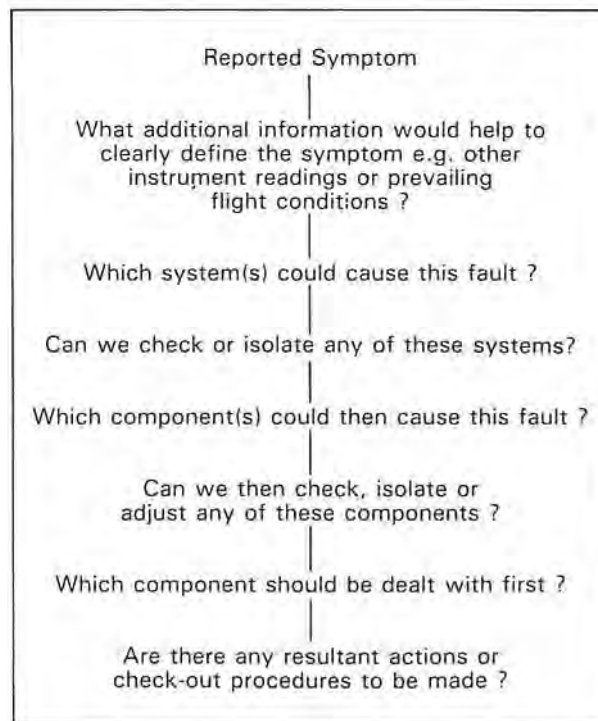
Fig. 24-3   Trouble shooting - logical sequence.

25. The reported symptom will frequently originate from flight deck instrument readings and, unless it is apparent from supporting information that the readings are genuine, instrumentation should be checked before proceeding further. Similarly, quick elimination checks should normally be undertaken before more involved tasks. The manufacturers' maintenance manual contains trouble shooting information, usually in chart form and fig. 24-4 shows a typical example.

LOW POWER — TGT HIGH — OTHER PARAMETERS NORMAL

1 TEST INDICATOR (MM 77-21-03) DOES INDICATOR TEST OK? —NO→ 2 CHANGE TGT INDICATOR (MM 77-21-03)

YES↓

3 BORESCOPE INSPECT IP AND HP COMPRESSOR ROTOR BLADES (MM 72-00-00) INSPECTION OK? —NO→ 4 REJECT ENGINE

YES↓

5 WASH LP AND IP COMPRESSOR BLADES AND LP OGV (MM 72-30-00) AND (MM 72-31-01) IS TGT REDUCED? —YES→ 6 RETURN TO SERVICE AND MONITOR TGT

NO↓

7 PERFORM TEST 4 (MM 71-00-00) AND ADJUST AS NECESSARY IS TGT NOW ACCEPTABLE? —YES→ 8 RETURN TO SERVICE

NO↓

9 TEST RESISTANCE OF BALLAST AND TRIM RESISTORS (MM 77-21-04) TEST OK? —NO→ 10 CHANGE BALLAST OR TRIM RESISTORS AS NECESSARY

YES→ 11 REJECT ENGINE
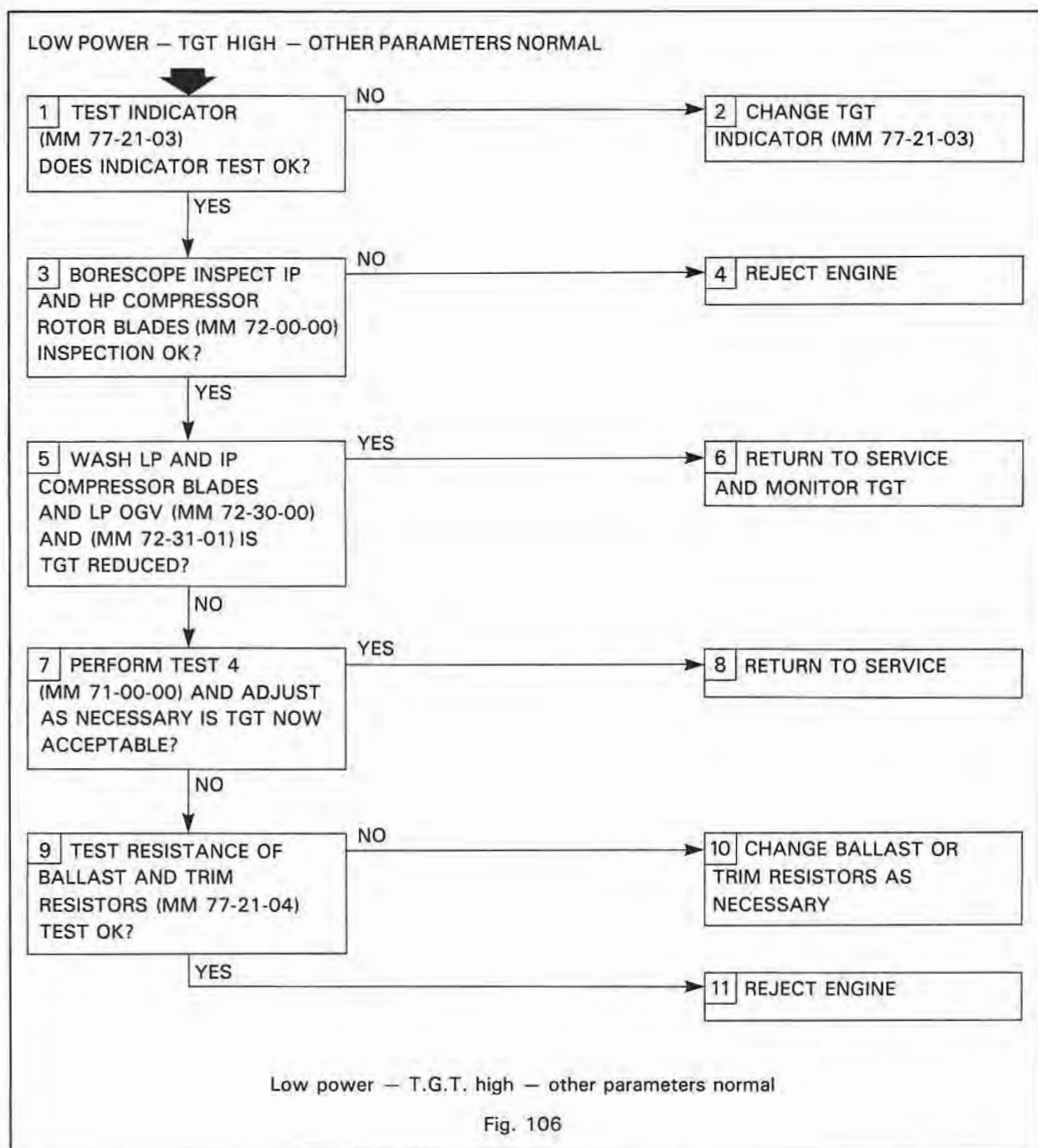
Low power — T.G.T. high — other parameters normal

Fig. 106

Fig. 24-4   A typical trouble shooting chart.

26. The progressive introduction of improved and more reliable condition monitoring devices (para. 11) will have considerable influence on accepted trouble shooting practice, since to a large extent these devices are designed to pin-point, at an early stage, the specific system or assembly at fault. The development of suitable test sets could eventually eliminate the need for engine ground testing after trouble shooting

ADJUSTMENTS

27. There are usually some adjustments that can be made to the engine controlling the fuel trimming devices. Typical functions for which adjustment provision is normally made include idling and maximum r.p.m., acceleration and deceleration times, and compressor air bleed valve operation.

28. Adjustment of an engine should be made only if it is quite certain that no other fault exists that could be responsible for the particular condition, The maintenance manual instructions relative to the adjustment must be closely adhered to at all times. In many instances, subject to local instructions, a ground adjustment can be made with the engine running.

29. Adjusters are usually designed with some form of friction locking (fig. 24-5) that dispenses with locknuts, lockplates and locking wire. On some engines, provision is also made for fitting remote adjustment equipment (fig. 24-6) that permits adjustment to be made during ground test with the cowls closed, the adjustment usually being made from the flight deck.

GROUND TESTING

30. The basic purpose of engine ground testing is to confirm performance and mechanical integrity and to check a fault or prove a rectification during trouble shooting. Ground testing is essential after engine installation, but scheduled ground testing may not normally be called for where satisfactory operation on the last flight is considered to be the authority or acceptance for the subsequent flight. In some instances, this is backed up by specific checks made in cruise or on approach and, of course, by evidence from flight deck indicators and recordings.

31. For economic reasons and because of the noise problem, ground testing is kept to a minimum
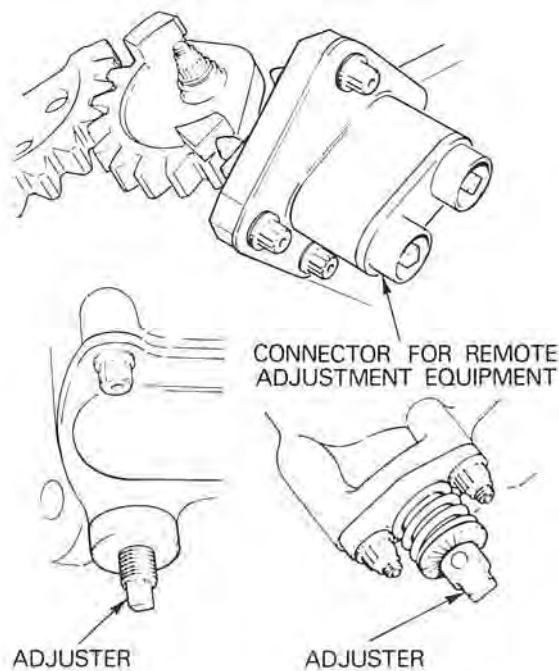
Fig. 24-5   Typical friction locked adjusters.

and is usually only carried out after engine installations, during trouble shooting, or to test an aircraft system. With the improved maintenance methods and introduction of system test sets which simulate running conditions during the checking of a static engine, the need for ground testing, particularly at high power, is becoming virtually unnecessary.

32. Before a ground test is made, certain precautions and procedures must be observed to prevent damage to the engine or aircraft and injury to personnel.

33. Because of the mass of air that will be drawn into the intake and the resultant high velocity and temperature of the exhaust gases during a ground test, danger zones exist at the front and rear of the aircraft. These zones will extend for a considerable distance, and atypical example is shown in fig. 24-7. The jet efflux must be clear o! buildings and other aircraft. Personnel engaged in ground testing must ensure that any easily detachable clothing is securely fastened and should wear acoustic ear muffs.

Fig. 24-6   Remote adjustment equipment fitted to a turbo-propeller engine.
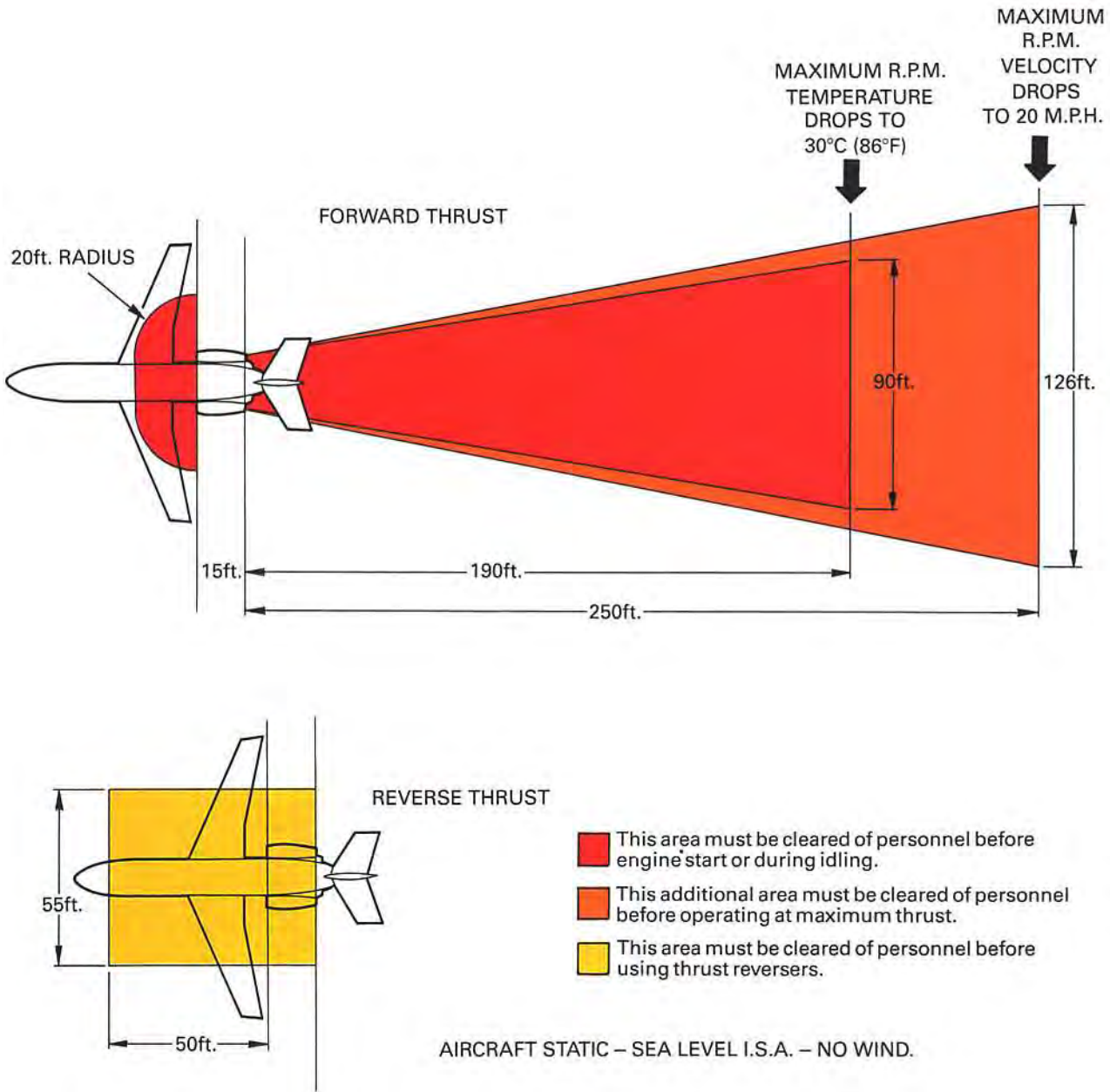
Fig. 24-7   Ground running danger zones.

34. The aircraft should be headed into wind and positioned so that the air intake and exhaust are over firm concrete, or a prepared area that is free from loose material and loose objects, and clear of equipment. Where noise suppression installations are used, the aircraft should be positioned in accordance with local instructions. When vertical take-off aircraft are being tested, protective steel plates and deflectors may be used to prevent ground erosion and engine ingestion of exhaust gases and debris. Aircraft wheels should be securely chocked and braked; with vertical take-off aircraft, anchoring

or restraining devices are also used. Adequate fire fighting equipment must be readily available and local fire regulations must be strictly enforced.

35. Before an engine is started, the air intake and jet pipe must be inspected to ensure that they are free from any debris or obstruction. Each operator will detail his individual pre-start inspection requirements; a typical example of this for a multi-engined aircraft is shown in fig. 24-8.

36. The starting drill varies between different aircraft types and a starting check procedure is normally used. Generally, all non-essential systems are switched or selected off; warning and emergency systems are checked when applicable. Finally, after ensuring that the low pressure fuel supply is selected on, the starting cycle is initiated.

37. At a predetermined point during the starting cycle, the high pressure fuel shut-off valve (cock) is opened to allow fuel to pass to the fuel spray nozzles, this point varying with aircraft and engine type; on some installations the shut-off valve may be opened before the starting cycle is initiated. During the engine light-up period and subsequent acceleration to idling speed, the engine exhaust gas temperature must be carefully monitored to ensure
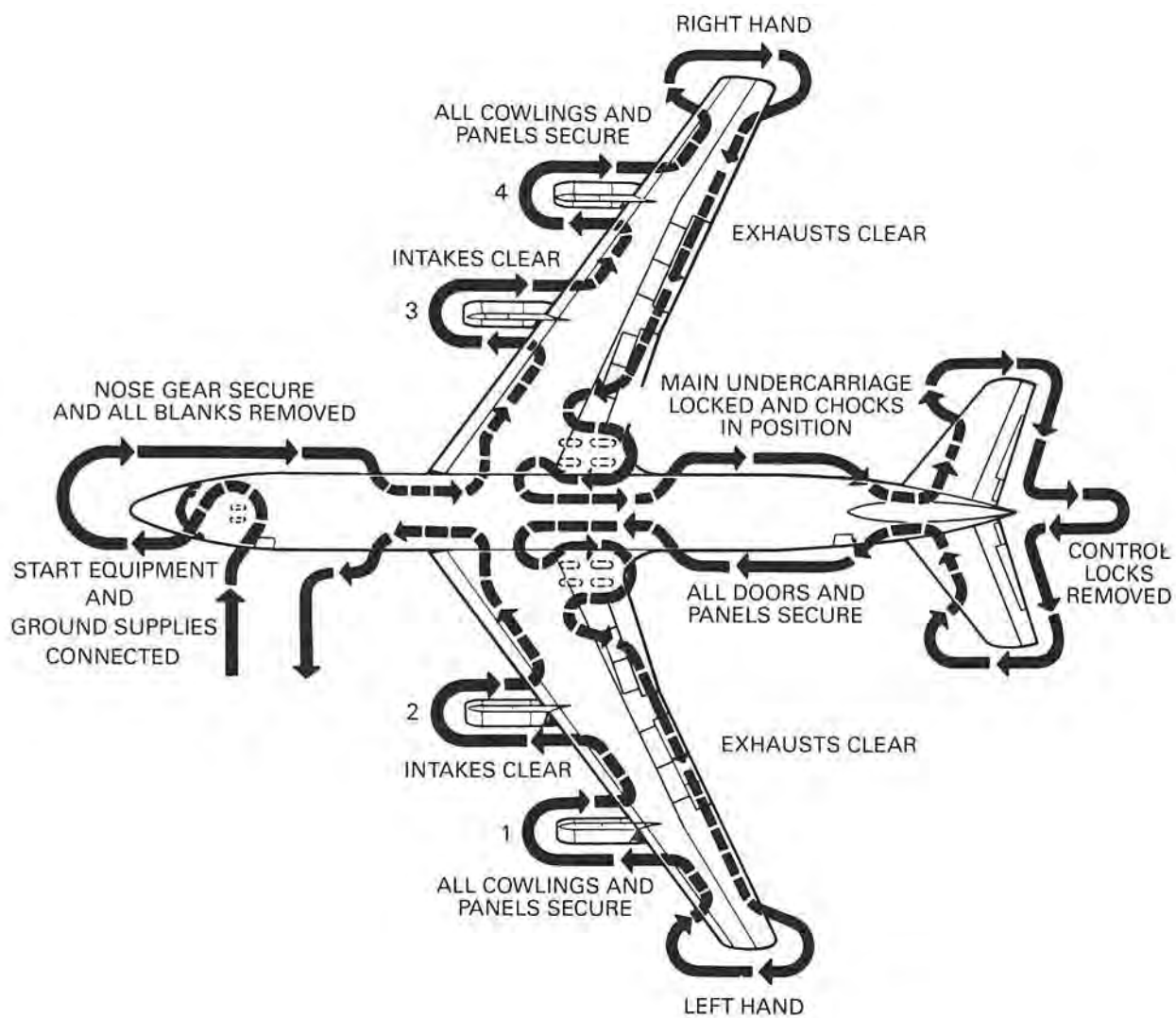
Fig. 24-8   A pre-start inspection sequence.

that the maximum temperature limitation is not exceeded. If the temperature limitation appears likely to be exceeded, the shut-off valve must be closed and the starting cycle cancelled; the cause and possible effect of the high temperature must then be investigated before the engine is again started.

38.   When a turbo-propeller engine is being started, the propeller must be set to the correct starting pitch, as recommended by the engine manufacturer. To provide the minimum resistance to turning and thus prevent an excessive exhaust gas temperature occurring during the starting cycle, some propellers have a special fine pitch setting.

39.   Throttle movements should be kept to a minimum and be smooth and progressive to avoid thermal stresses associated with rapid changes in temperature. Rapid throttle movements to check the acceleration and deceleration capabilities of the engine should be made only after all other major checks have proved satisfactory and after some slower accelerations and decelerations have proved successful.

40.   Before an engine is stopped, it should normally be allowed to run for a short period at idling speed to ensure gradual cooling of the turbine assembly. The only action required to stop the engine is the closing

Fig. 24-9   Overheated turbine blades.

of the shut-off valve. The shut-off valve must not be re-opened during engine rundown, as the resulting supply of fuel can spontaneously ignite with consequent severe overheating of the turbine assembly. An example of turbine blades which have been subjected to overheating is shown in fig. 24-9.

41. The time taken for the engine to come to rest after the shut-off valve is closed is known as the 'rundown time' and this can give an indication of any rubbing inside the engine. However it should be borne in mind that variations in wind velocity and direction may affect the run-down time of an engine.

Rolls-Royce Tyne
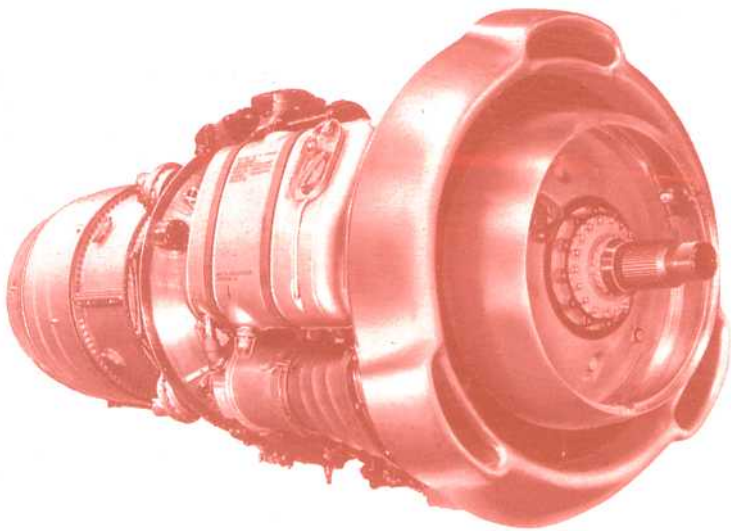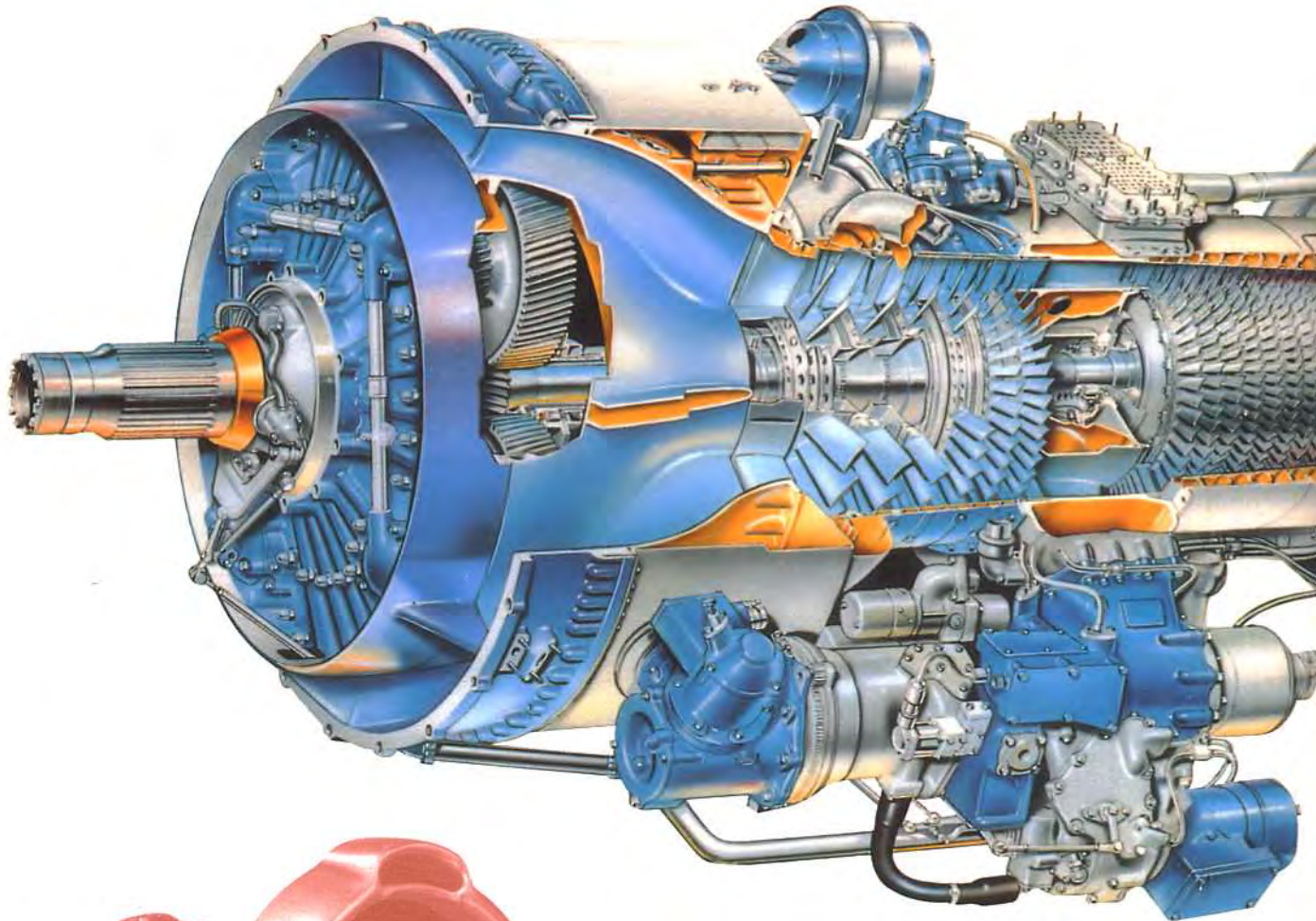
Bristol BE 25 Orion

The Orion was a two-spool turbo-prop designed to operate at its full rated power to 20,000ft, achieved by throttling its sea level power to a maximum of 5150 ehp. Flight testing commenced in August 1956 with the engine installed in the port outer nacelle of the Bristol Britannia. Development was discontinued owing to lack of demand for turbo-props at this time.

25: Overhaul

Contents

INTRODUCTION

1. It is most important that the cost of maintaining an engine in service is considered at the design stage. All aspects of engine repairability are also considered, both to reduce the requirement for overhaul or repair and to avoid, where possible, designs which make repair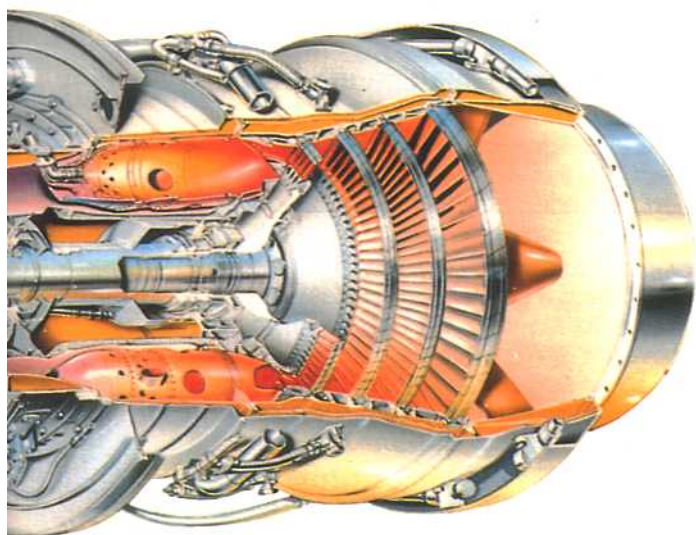s difficult to effect. Engine construction must allow the operator to complete the overhaul or repair work as quickly and cheaply as possible.

2. In service, the engine is inspected at routine periods based on manufacturers' recommendations and agreed between the operator and the relevant airworthiness authority. The engine is removed from the aircraft when it fails, during these inspections, to meet the specified standards. This concept is a form of 'on-condition' monitoring, reference para. 9, however, regardless of condition, some engines are removed when a stipulated number of engine flying

hours have been achieved, this concept is known as time between overhauls (T.B.O.). Operators will often remove engines in order to acquire 'fleet stagger' thus preventing a situation when an unacceptable number of engines require removal at the same period of time.

3. The length of time between overhauls varies considerably between different engine types, being established as a result of discussions between the operator, the airworthiness authority and the manufacturer, when such considerations as the total experience gained with the particular engine series, the type of operation, the utilization, and sometimes climatic conditions, are taken into account. In improving the overhaul period the airworthiness authority may take into consideration the background of the operator, his servicing facilities and the experience of his maintenance personnel.

4. When a new type of engine enters service, sampling (i.e. engine removal, dismantling and inspection) may be called for at a modest life. The sampling will be continued until the life at which the engine should be overhauled is indicated by the condition of the sample engines or by its reliability record in service. In some instances, the ultimate life
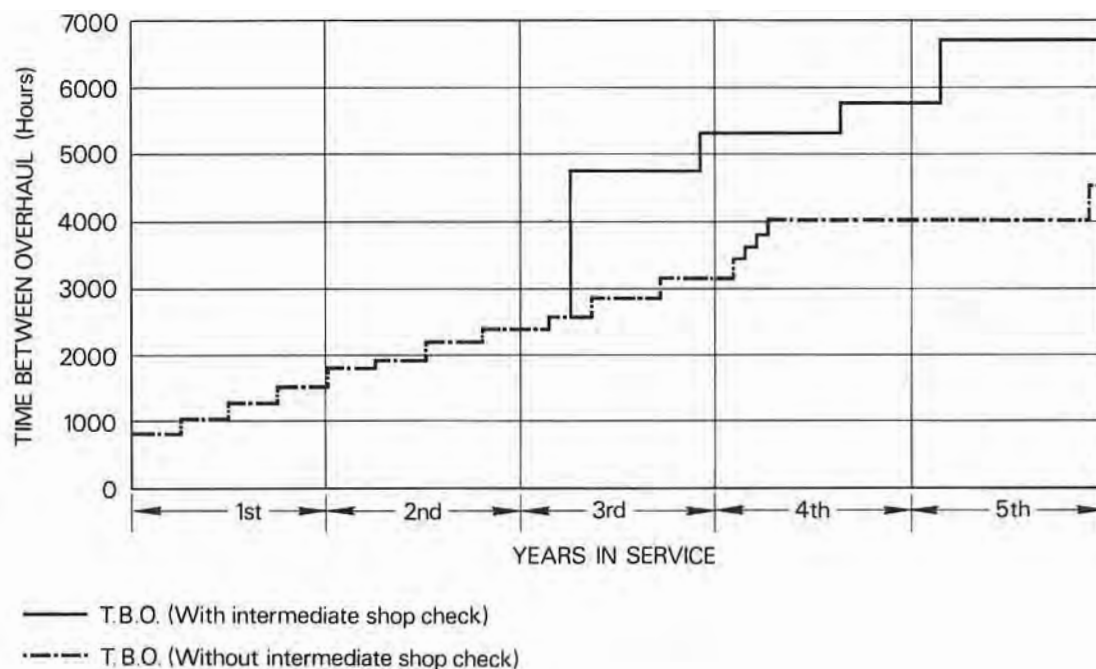
Fig. 25-1   Example of growth of time between overhaul (T.B.O.).

obtained may be two, three, or even four times the original period permitted. The development of the T.B.O, from the introduction of an engine into service, through several years of operation, is shown as an example in fig, 25-1.

5. Among the main factors affecting the overhaul period for an engine is the use to which it is put in service. For example, a military engine will generally have a much lower T.B.O. than its civil counterpart, as performance capability is the operating criterion rather than economics. Due to the effect of rapid temperature changes in the hot parts of the engine, the most arduous treatment is the frequent changing of power output to which short-haul transports and fighter aircraft are subjected.

6. When aircraft are based in areas with exceptionally high humidity or salt content in the atmosphere, there exists the added danger of corrosion, resulting in the need for more frequent overhauls. In peace time, some military aircraft have a very low utilization, this introduces the additional problem of certain materials used in its construction deteriorating before the engine has otherwise reached a condition which would normally require an overhaul. Elapsed time, as well as flying hours, would then influence the overhaul period.

7. In addition to scheduled overhauls, there are problems that arise from damage and defects. A proportion of these, which are uneconomic or impractical to rectify in the aircraft, necessitate unscheduled removals and require the engine to be returned to an engineering base or an overhaul shop for partial or complete overhaul.

8. The purpose of overhaul is to restore an engine enabling it to complete a further life by complying with new engine performance acceptance limitations and maintaining the same reliability. This is achieved by dismantling the engine in order that parts can be inspected for condition and to determine the need for renewal or repair of those parts whose deterioration would reduce the performance, or would not remain in a serviceable condition until the next overhaul.

9. The design of the modular constructed engine (Part 22, fig. 22-1) makes it particularly suited to a different technique of overhaul/repair. This technique is based on 'on-condition' monitoring (Part 24). This means that a life is not declared for the total engine but only certain parts of the engine. On reaching their life limit, these parts are replaced and the engine returned to service, the remainder of the engine being overhauled 'on-condition'.

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10. Modular construction, together with associated tooling, enables the engine to be disassembled into a number of major assemblies (modules). Modules which contain life limited parts can be replaced by similar assemblies and the engine returned to service with minimum delay. The removed modules are disassembled into mini-modules for life limited part replacement, repair or complete overhaul as required.

OVERHAUL/REPAIR

11. The high cost of new engines has a considerable influence on the overhaul/repair arrangements, as the number of spare engines normally bought by the operator is kept to an absolute minimum. This means that an unserviceable engine must be quickly restored to serviceability by changing a module, or a part if the modular construction will permit it, or by careful scheduling of planned removals for overhauls at time expiry. This scheduling, through the workshop, of engines or modules that require the use of specialized equipment for repair is important, both to keep the flow of work even and to stagger removals to avoid aircraft being grounded by shortage of serviceable engines or modules.

12. Because the work that is to be implemented must be planned and subsequently recorded, the engine or module is received in the workshop with documents to show its modification standard and its reason for rejection from service. The planning will include a list of the modifications that can or must be incorporated to improve engine reliability or performance or to reduce operating costs.

13. The layout of the overhaul/repair workshop is designed to facilitate work movement through the complete range of operations, to achieve maximum utilization of floor space and to allow special equipment to be sited in positions that will suit the general flow pattern. All these considerations are aimed at achieving a quick turnround of engines. As an example of how shop layouts may be planned, a typical arrangement is shown in fig, 25-2.

Disassembly

14. The engine can be disassembled in the vertical or horizontal position. When it is disassembled in the vertical position, the engine is mounted, usually front end downwards, on a floor-fixture stool or a ram-top fixture. To enable it to be disassembled horizontally, the engine is mounted in a special turnover stand.

15. When the floor-fixture stool is used, the personnel use a mobile work platform to raise themselves to a reasonable working position around the engine. When the ram-top fixture is used, the ram and engine are retracted into a pit, so enabling the workmen to remain at floor level.

16. The engine is disassembled into main sub-assemblies or modules, which are fitted in transportation stands and despatched to the separate areas where they are further disassembled to individual parts. The individual parts are conveyed in suitable containers to a cleaning area in preparation for inspection.

Cleaning

17. The cleaning agents used during overhaul range from organic solvents to acid and other chemical cleaners, and extend to electrolytic cleaning solutions.

18. Organic solvents include kerosine for washing, trichloroethane for degreasing and paint stripping solutions which can generally be used on the majority of components for carbon and paint removal. The more restricted and sometimes rigidly controlled acid and other chemical cleaners are used for corrosion, heat scale and carbon removal from certain components. To give the highest degree of cleanliness to achieve the integrity of inspection that is considered necessary on certain major rotating parts, such as turbine discs, electrolytic cleaning solutions are often used.

19. Aircraft which operate at high altitudes can become contaminated with radio-active particles held in the atmosphere, this radio-activity is retained in the dirt and carbon deposits in the engine.

20. If contamination is suspected the radio-activity level of the engine must be determined to ensure the limitations agreed by the health authorities are not exceeded, Evidence of contamination will entail additional cleaning in a designated region, separate from the overhaul area, to safeguard the health of personnel in the workshop. Arrangements have to made with the health authorities for disposal of the waste radio-active cleaning material.

Inspection

21. After cleaning, and prior to inspection, the surfaces of some parts, e.g. turbine discs, are etched. This process removes a small amount of material from the surface of the part, leaving an even
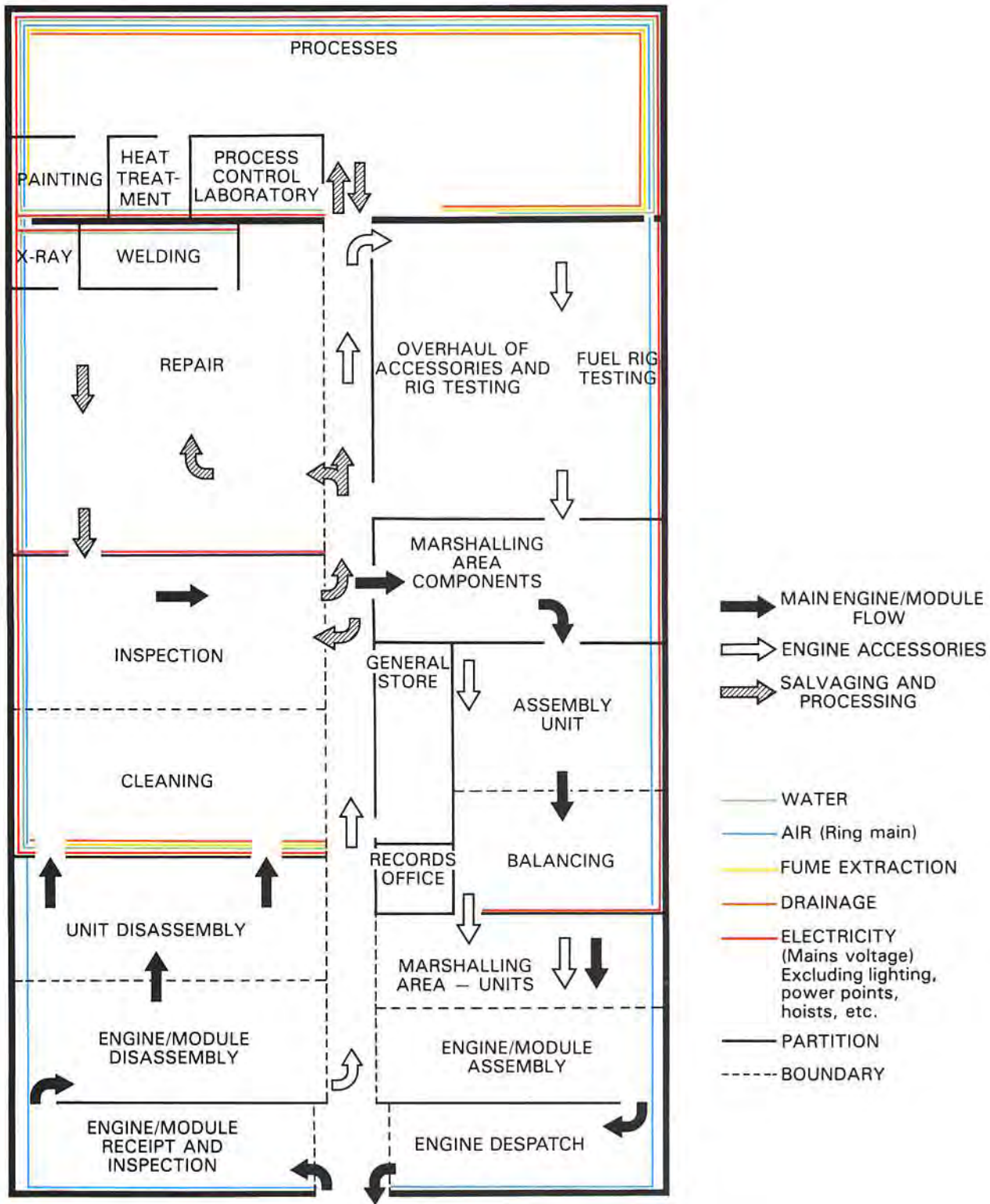
Fig. 25-2   Typical overhaul workshop layout.

matt finish which reveals surface defects that cannot normally be seen with the naked eye. The metal removal is normally achieved either by an electrolytic process in which the part forms the anode, or by immersing the part for a short time in a special acid bath. Both methods must be carefully controlled to avoid the removal of too much material.

22. After the components have been cleaned they are visually and, when necessary, dimensionally inspected to establish general condition and then subjected to crack inspection. This may include binocular and magnetic or penetrant inspection techniques, used either alone or consecutively, depending on the components being inspected and the degree of inspection considered necessary.

23. The non-dimensional inspections can be divided into visual examination for general condition and inspection for cracks. The visual examination depends on the inspector's judgement, based on experience and backed by guidance from the manufacturer. Although the visual examination of many parts of the engine conform to normal engineering practice, for some parts the acceptance standards are specialized, for example, the combustion chambers, which are subjected to very high temperatures and high speed airflows in service.

24. Dimensional inspection consists of measuring specific components to ensure that they are within the limits and tolerances laid down and known as 'Fits and Clearances'. Some of the components are measured at each overhaul, because only a small amount of wear or distortion is permissible or to enable the working clearances with mating components to be calculated. Other components are measured only when the condition found during visual inspection requires dimensional verification. The tolerances laid down for overhaul, supported by service experience, are often wider than those used during original manufacture.

25. The detection of cracks that are not normally visible to the naked eye is most important, particularly on major rotating parts such as turbine discs, since failure to detect them could result in crack propagation during further service and eventually lead to component failure. Various methods of accentuating these are used for inspection, the two principal techniques being penetrant inspection for nonmagnetic materials and electro-magnetic inspection for those parts that can be magnetized.

26. Two forms of penetrant inspection in common use are known as the dye penetrant and the fluorescent test. With the dye test, a penetrating coloured dye is induced to enter any cracks or pores in the surface of the part. The surface is then washed and a developer fluid containing white absorbents is applied. Dye remaining in cracks or other surface defects is drawn to the surface of the developer by capillary action and the resultant stains indicate their locations.
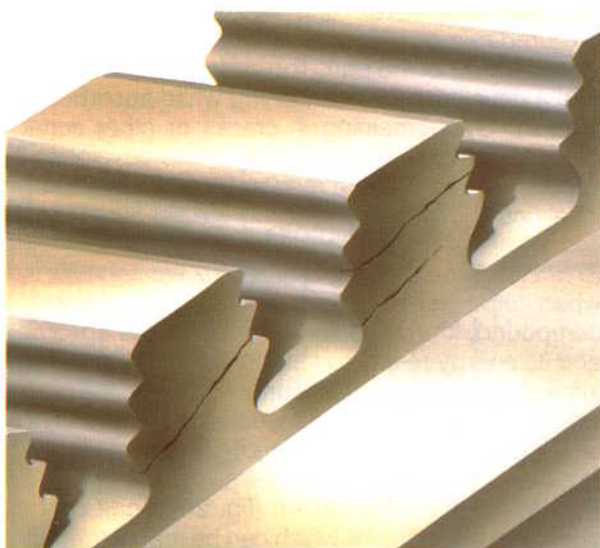
27. Fluorescent testing is based on the principle that when ultra-violet radiation falls on a chemical compound, known as fluorescent ink, it is absorbed and its energy re-emitted as visible light. If a suitable ink is allowed to penetrate surface cavities, the places where it is trapped will be revealed under the rays of an ultra-violet lamp by brilliant light emissions.

28. Magnetic crack testing (fig. 25-3) can only be applied to components which can be magnetized. The part is first magnetized and then sprayed with, or immersed in, a low viscosity fluid which contains ferrous particles and is known as 'ink'. The two walls of a crack in the magnetized part form magnetic poles and the magnetic field between these poles attracts the particles in the ink, so indicating the crack (fig. 25-4). In some instances, the ink may contain fluorescent particles which enable their build-up to be viewed under an ultra-violet lamp, A part that has been magnetically crack tested must be de-magnetized after inspection.

29. Chromic acid anodizing may be used as a means of crack detection on aluminium parts, e.g. compressor blades. This process, in addition to providing an oxide film that protects against corrosion, gives a surface that reveals even the smallest flaws.

Fig. 25-3   Magnetic crack testing.

these alloys, which have to withstand high stress loadings in service, are often welded in a bag or plastic dome that is purged by an inert gas before welding commences.

34. More advanced materials and constructions may have to be welded by electron-beam welding. This method not only enables dissimilar metals to be welded, but also complete sections of the more advanced fabricated constructions, e.g. a section of a fabricated rotor drum, to be replaced at a low percentage cost of a new drum.

35. Some repair methods, such as welding, may affect the properties of the materials and, to restore the materials to a satisfactory condition, it may be necessary to heat treat the parts to remove the stresses, reduce the hardness of the weld area or restore the strength of the material in the heat affected area, Heat treatment techniques are also used for removing distortions after welding. The parts are heated to a temperature sufficient to remove the stresses and, during the heat treatment process, fixtures are often used to ensure the parts maintain their correct configuration.

36. Electro-plating methods are also widely used for repair purposes and these range from chromium plating, which can be used to provide a very hard surface, to thin coatings of copper or silver plating, which can be applied to such areas as bearing locations on a shaft to restore a fitting diameter that is only slightly worn.

37. Many repairs are effected by machining diameters and/or faces to undersize dimensions or bores to oversize dimensions and then fitting shims, liners or metal spraying coatings of wear resistant material. The effected surfaces are then restored to their original dimensions by machining or grinding.

38. The inspection of parts after they have been repaired consists mainly of a penetrant or magnetic inspection. However, further inspection may be required on parts that have been extensively repaired and this may involve pressure testing or X-ray inspection of welded areas.

39. Re-balancing of the main rotating assembly will be necessary during overhaul, even though all the original parts may be refitted, and this is done as described in para 40.

Fig. 25-4   Cracks revealed by magnetic crack detection.

30. When the requirement for a detailed inspection on a component such as a turbine disc is necessary, etching of the disc surfaces would be followed by binocular inspection of the blade retention areas. The whole disc would then be subjected to magnetic crack test, followed by re-inspection of the disc including a further binocular inspection of the blade retention areas.

Repair

31. To ensure that costs are maintained at the lowest possible level, a wide variety of techniques are used to repair engine parts to make them suitable for further service. Welding, the fitting of interference sleeves or liners, machining and electro-plating are some of the techniques employed during repair.

32. The welding techniques detailed in Part 22 are extensively used and range from welding of cracks by inert gas welding to the renewing of sections of flame tubes and jet pipes by electric resistance welding.

33. On some materials now being used for gas turbine engine parts, different techniques may have to be employed. An example of this is the high strength titanium alloys which suffer from brittle welds if they are allowed to become contaminated by oxygen during the cooling period. Parts made in
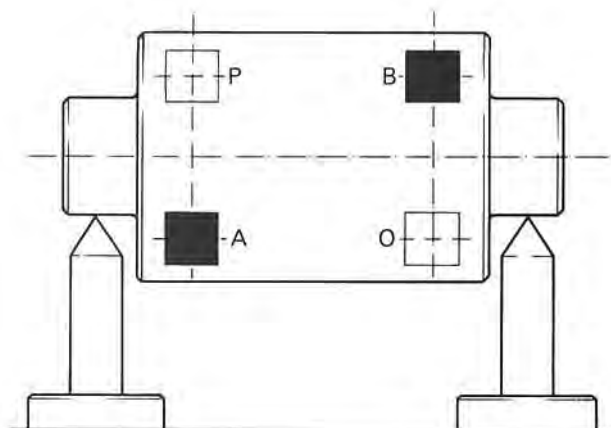
Balancing

40. Because of the high rotational speeds, any unbalance in the main rotating assembly of a gas turbine engine is capable of producing vibration and stresses which increase as the square of the rotational speed. Therefore very accurate balancing of the rotating assembly is necessary.

41. The two main methods of measuring and correcting unbalance are single plane (static) balancing and two plane (dynamic) balancing. With single plane, the unbalance is only in one plane i.e., centrally through the component at 90 degrees to the axis. This is appropriate for components such as individual compressor or turbine discs.

42. For compressor and/or turbine rotor assemblies possessing appreciable axial length, unbalance may be present at many positions along the axis. In general it is not possible to correct this combination of distributed unbalance in a single plane. However, if two correction planes are chosen, usually at axially opposed ends of the assembly, it is always possible to find a combination of two unbalance weights which are equivalent for the unbalances present in the assembled rotor, hence two plane balancing.

43. To illustrate this point refer to fig. 25-5, the distribution of unbalance in the rotor has been reduced to an equivalent system of two unbalances 'A' and 'B'. The rotor is already in static balance because in this example 'A' and 'B' are equal and opposed. However, when the part is rotating, each weight produces its own centrifugal force in opposition to the other causing unbalance couples, with the tendency to turn the part end-over-end. This action is restricted by the bearings, with resultant stresses and vibration. It will be seen, therefore, that to bring the part to a state of dynamic balance, an equal amount of weight must be removed at 'A' and 'B' or added at 'P' and 'O'. When the couples set up by the centrifugal forces are equal, it is said that a part is dynamically balanced. Unbalance is expressed in units of ounce-inches, thus one ounce of excess weight displaced two inches from the axis of a rotor is two ounce inches of unbalance.

44. When balancing assemblies such as L.P. compressor rotors, the readings obtained are inconsistent due to blade scatter. Blade scatter is caused by the platform and root or retaining pin clearances allowing the blades to interlock at the platforms and assume a different radial position during each

Fig. 25-5   Unbalance couples due to centrifugal force.

balancing run. This only occurs at the relatively low r.p.m. used for balancing, because, during engine running, the blades will assume a consistent radial position as they are centrifuged outwards.

45. To obtain authentic balance results when blade scatter is present, it is necessary to record readings from several balance runs, e.g. 8 runs, thereafter determining a vector mean.

46. A typical dynamic balancing machine for indicating the magnitude and angular position of unbalance in each plane is shown in fig. 25-6. Correction of unbalance may be achieved by one or a combination of the following basic methods; redistribution of weight, addition of weight and removal of weight.

47. Redistribution of weight is possible for such assemblies as turbine and compressor discs, when blades of different weight can be interchanged and, on some engines, clamped weights are provided for positioning around the disc.

48. The addition of weight is probably the most common method used, certain parts of the assembly having provision for the fitting of screwed or riveted plugs, heavy wire, balancing plates or nuts.
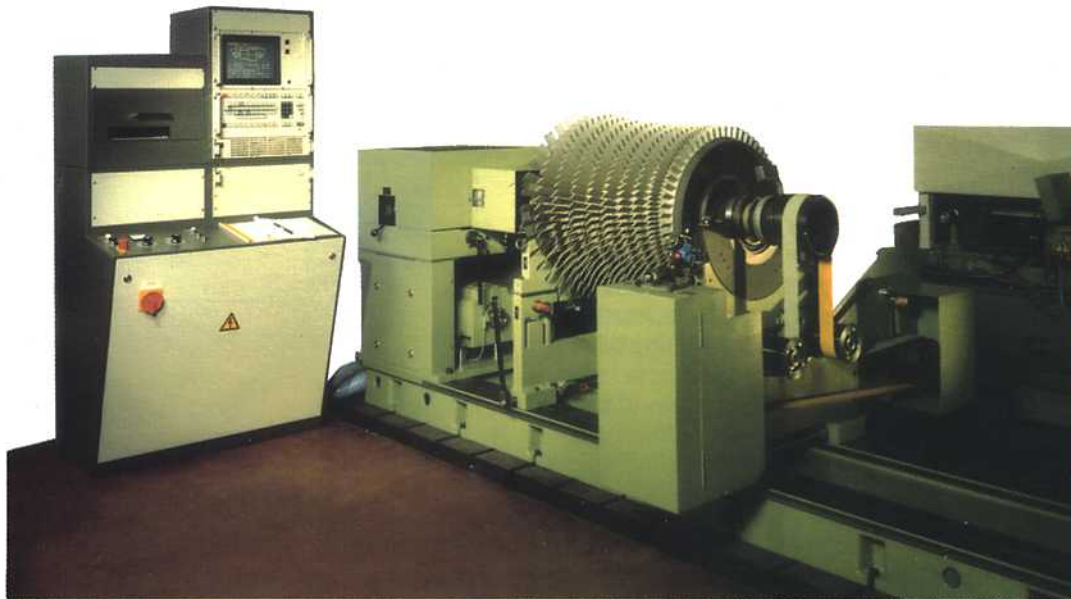
Fig. 25-6   Dynamic balancing machine.

49. Removing weight by machining metal from balancing lands is the third basic method, but normally it is only employed on initial manufacture when balancing a component, e.g. a turbine shaft or a compressor shaft, that is part of a larger assembly.

50. Modular assembled engines demand different balancing methods which include the use of simulated engine rotors. The dummy rotors must reproduce the bearing span, weight, centre of gravity and dynamic characteristics of the sub-assembly it

COMPRESSOR AND TURBINE ASSEMBLY

TRUE COMPRESSOR

DUMMY TURBINE

DUMMY COMPRESSOR

TRUE TURBINE
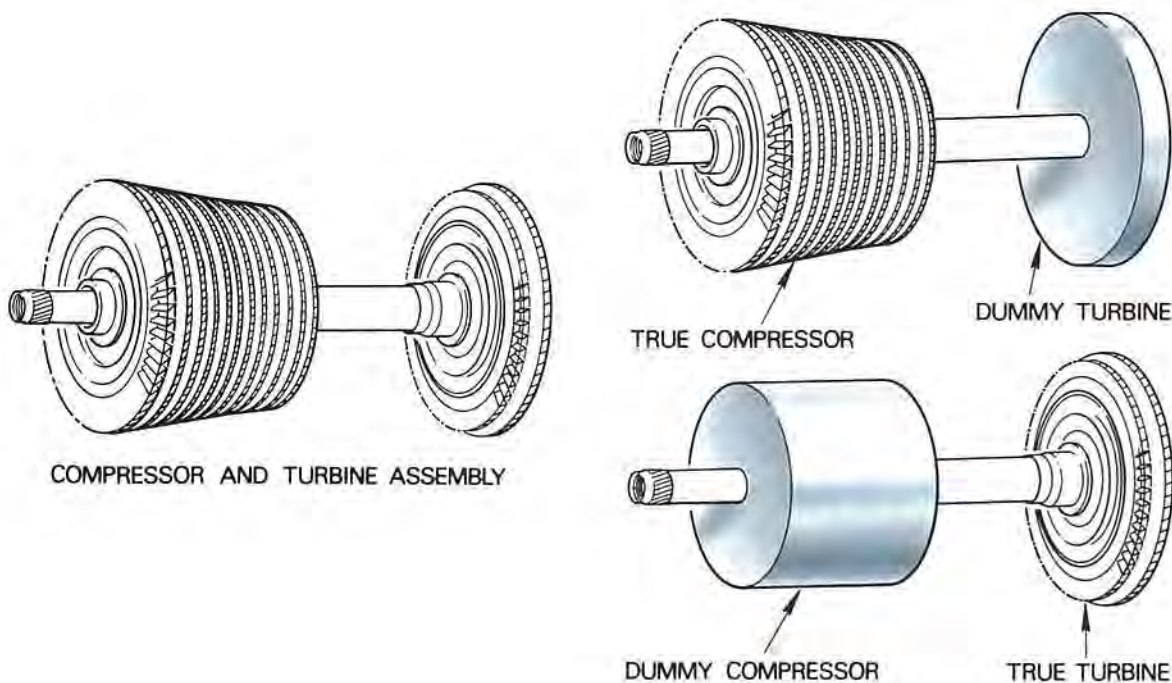
Fig. 25-7   Simulated engine rotor assemblies.

replaces and must be produced and maintained so that their own contribution to the measured unbalance is minimal. In order to obtain the correct dynamic reactions when balancing a compressor and/or turbine rotor assembly on its own, with the intention of making it an independent module, a simulated engine rotor must be used to replace the mating assembly, ref. fig. 25-7. The compressor and/or turbine rotor assembly having then been independently balanced with the appropriate dummy rotor is thus corrected both for its own unbalance and influence due to geometric errors on any other mating assembly.

Moment weighing of blades

51. With the introduction of the large fan blade, moment weighing of blades has assumed a greater significance, ref. fig. 25-8. This operation takes into account the mass of each blade and also the position of its centre of gravity relative to the centre line of the disc into which the blade is assembled. The mechanical system of blade moment weighing may be integrated with a computer, ref. fig, 25-9, which will automatically optimise the blade distribution. The moment weight of a blade in units i.e. g.mm. or oz.in., is identical to the unbalance effect of the blade when installed into a disc. The recorded measurement of blade moment weights enables each blade to be

$$\text{BLADE MOMENT} = (W_b y = W_p x \pm W_c z)$$
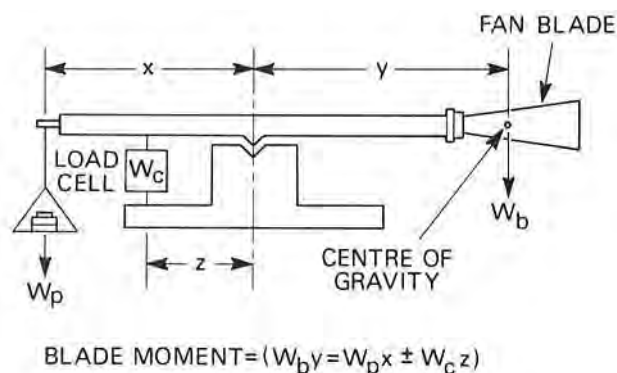
Fig. 25-9   Integrated blade moment weighing.

distributed around the disc in order that these unbalances are cancelled.

Assembling

52. The engine can be built in the vertical or horizontal position, using the ram or stand illustrated in fig. 25-TO and 25-11 respectively. Assembling of the engine sub-assemblies or modules is done in separate areas, thus minimizing the build time on the build rams or stands.
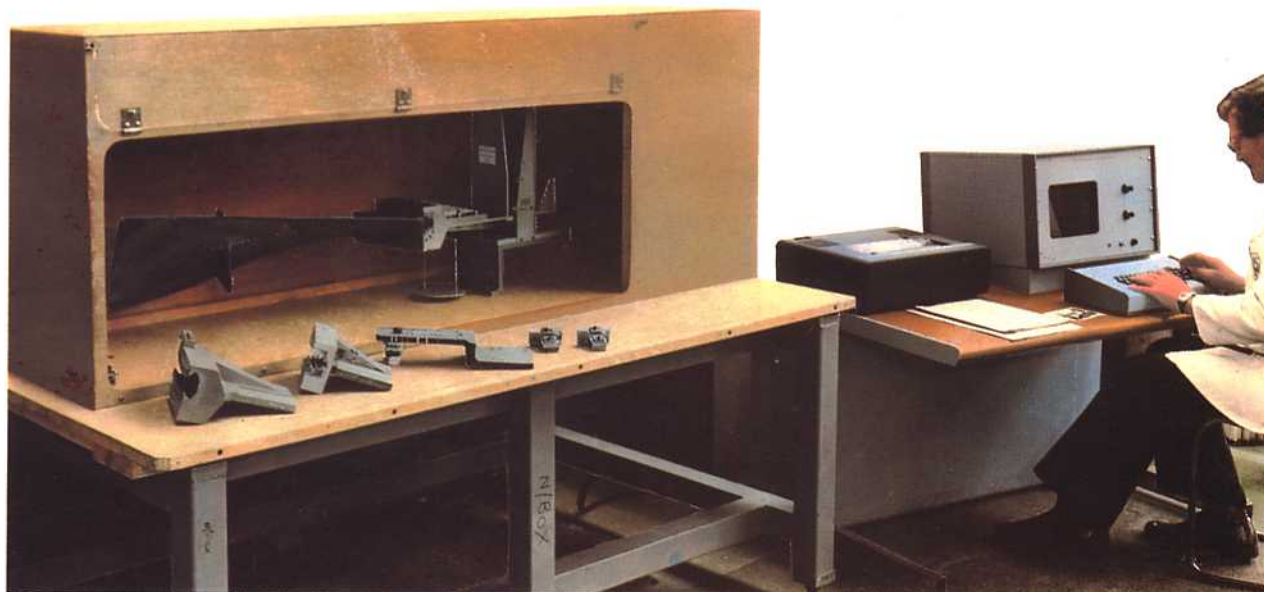
Fig. 25-8   Principle of blade moment weighing.

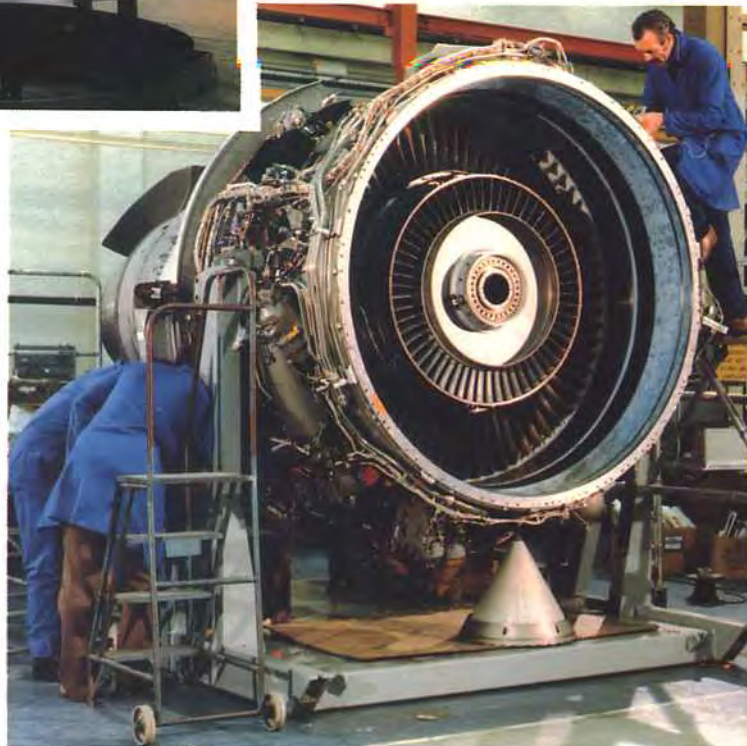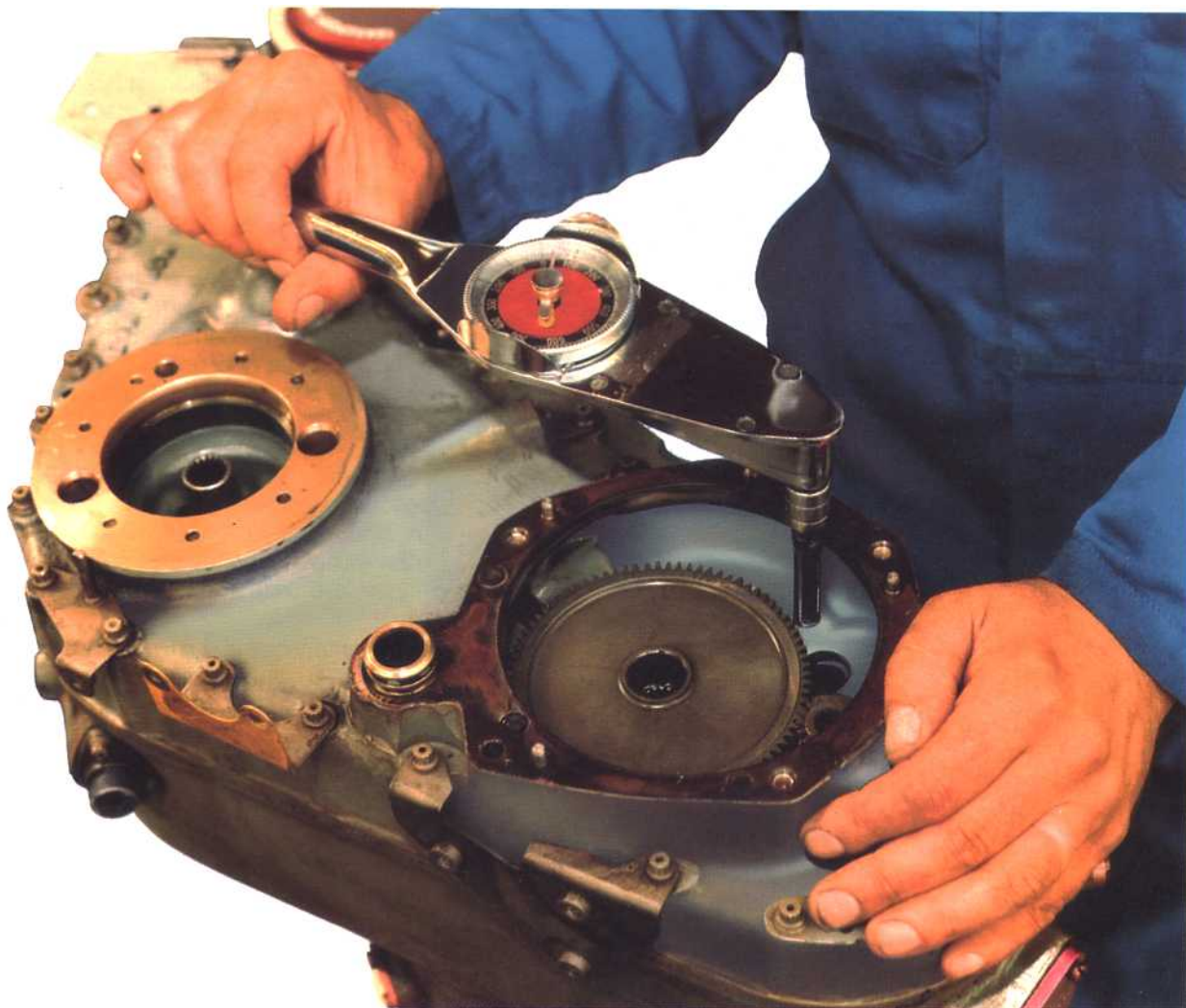**Fig. 25-10   Engine assembly
--- vertical.**

**Fig. 25-11   Engine assembly
--- horizontal.**

Fig. 25-12   Torque tightening.

53. During assembling, inspection checks are made. These checks can establish dimensions to enable axial and radial clearances to be calculated and adjustments to be made, or they can ascertain that vital fitting operations have been correctly effected. Dimensional checks are effected during disassembly to establish datums which must be repeated on subsequent re-assernbly.

54. To ensure that the nuts, bolts and setscrews throughout the engine and its accessories are uniformly tight, controlled torque tightening is applied, fig. 25-12, the torque loading figure is determined by the thread diameter and the differing coefficients of friction allied with thread finish i.e., silver or cadmium plating and the lubricant used.

Testing
55. On completion of assembly, every production and/or overhauled engine must be tested in a 'sea-level' test cell (fig. 25-14), i.e. a test cell in which the engine is run at ambient temperature and pressure conditions, the resultant performance figures being corrected to International Standard Atmosphere (I.S.A.) sea-level conditions (Part 21).

56. To ensure that the engine performance meets that guaranteed to the customer and the requirements of the Government licensing and purchasing authorities, each engine is tested to an acceptance test schedule.

57. In addition to the 'sea-level' tests, sample engines are tested to a flight evaluation test

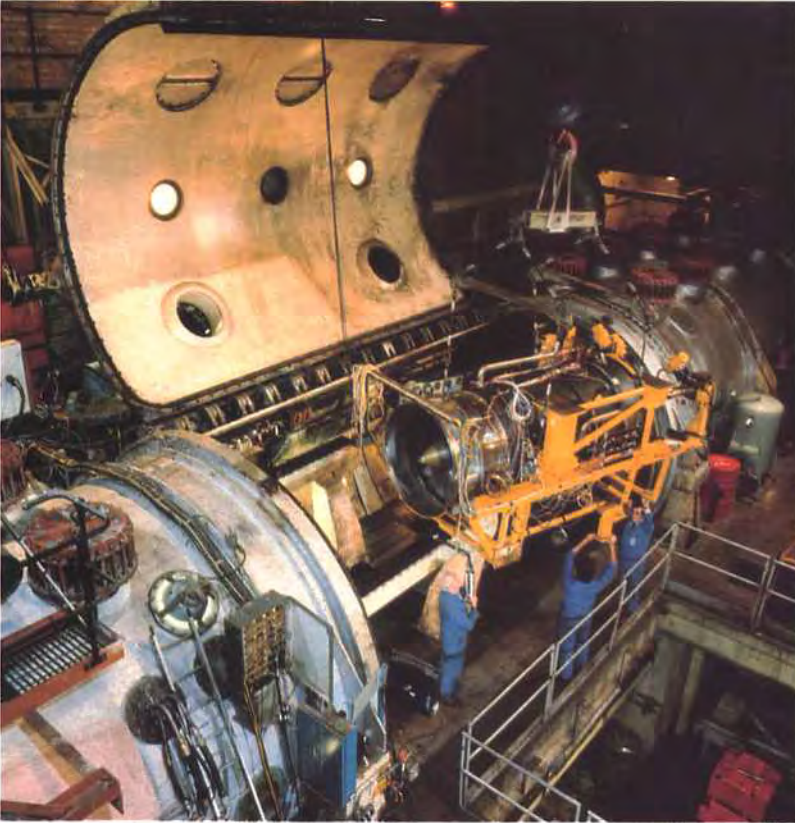Fig. 25-13   A high altitude test cell.

Fig. 25-14   A sea-level test cell.

schedule. These tests cover such characteristics as anti-icing, combustion and reheat efficiencies, performance, mechanical reliability, and oil and fuel consumptions at the variety of conditions to which the engine may be subjected during its operational life. Flight evaluation testing can be effected by installing the engine in an aircraft or in an altitude test cell (fig. 25-13) to test the variations of air humidity, pressure and temperature on the engine, its accessories and the oil and fuel systems. When in an aircraft, the engine is operated at the actual atmospheric conditions specified in the schedule, but in an altitude test cell, the engine is installed in an enclosed cell and tested to the schedule in conditions that are mechanically simulated.

58. Mechanical simulation comprises supplying the engine inlet with an accurately controlled mass airflow at the required temperature and humidity, and adjusting the atmospheric pressure within the exhaust cell to coincide with pressure at varying altitudes.

59. The data which is accumulated from either 'sea-level' or altitude testing is retained for future development, engine life assessment, material capabilities or any aspect of engine history.

60. During the testing of turbo-jet engines there is a need to reduce the exhaust noise to within acceptable limits. This may be achieved by several different means, each involving costly equipment. However, a typical silencer would do this by expansion in the first section, damping by acoustic tubes and final diffusion by a large exit through which
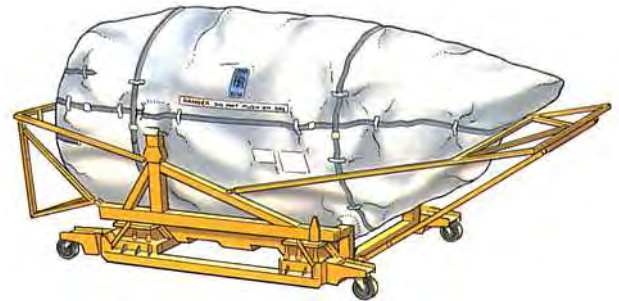
Fig. 25-15   Transportation stand and storage bag.

the hot gas would be directed upwards at a low velocity.

Preparing for storage/despatch
61. The preparation of the engine/module for storage and/or despatch is of major importance, since storage and transportation calls for special treatment to preserve the engine. To resist corrosion during storage, the fuel system is inhibited by special oil and all apertures are sealed off. The external and internal surfaces of the engine are also protected by special inhibiting powders or by paper impregnated with inhibiting powder and the engine is enclosed in a re-usable bag (fig. 25-15) or plastic sheeting into which a specific amount of desiccant is inserted. If transportation by rail or sea is involved, the inhibited and bagged engine may be packed in a wooden crate or metal case.

Appendix 1

Conversion factors

UNIT ABBREVIATIONS

in	=	inch	s	=	second	km	=	kilometre (mx1000)
ft	=	foot	min	=	minute	g	=	gram
yd	=	yard	h	=	hour	kg	=	kilogram
oz	=	ounce	f	=	force	N	=	newton
lb	=	pound	W	=	watt	Pa	=	pascal
cwt	=	hundredweight	kW	=	kilowatt (Wx1000)	kPa	=	kilopascal
Btu	=	British thermal unit	mm	=	millimetre (mx0.001)	J	=	Joule
hp	=	horsepower	m	=	metre	kJ	=	kilojoule (Jx1000)
Hg	=	mercury				MJ	=	megajoule (Jx1 000 000)

CONVERSION FACTORS - Exact values are printed in bold type.

LENGTH	1 in	=	**25.4** mm
	1 ft	=	**0.3048** m
	1 mile	=	1 .60934 km
	1 International nautical mile	=	**1.852** km
AREA	1 in^2	=	**645.16** mm^2
	1 ft^2	=	**92903.04** mm^2
VOLUME	1 UK fluid ounce	=	28413.1 mm^3
	1 US fluid ounce	=	29573.5 mm^3
	1 Imperial pint	=	568261.0 mm^3
	1 US liquid pint	=	473176.0 mm^3
	1 UK gallon	=	4546090.0 mm^3
	1 US gallon	=	3785410.0 mm^3
	1 in^3	=	16387.1 mm^3
	1 ft^3	=	0.0283168 m^3
MASS	1 oz (avoir.)	=	28. 3495 g
	1 lb	=	**0.45359237** kg
	1 UK ton	=	1.01605 tonne
	1 short ton (2000 lb)	=	0.907 tonne

DENSITY	1 lb/in^3	=	27679.9 kg/m^3		
	1 lb/ft^3	=	16.0185 kg/m^3		
VELOCITY	1 in/min	=	0.42333 mm/s		
	1 ft/min	=	**0.00508** m/s		
	1 ft/s	=	**0.3048** m/s		
	1 mile/h	=	1.60934 km/h		
	1 International knot	=	**1.852** km/h		
ACCELERATION	1 ft/s^2	=	**0.3048** m/s^2		
MASS FLOW RATE	1 lb/h	=	1.25998x10^{-4} kg/s		
FORCE	1 lbf	=	4.44822 N		
	1 kgf	=	**9.80665** N		
	1 tonf	=	9964.02 N		
PRESSURE	1 in Hg (0.0338639 bar)	=	3386.39 Pa		
	1 lbf/in^2 (0.0689476 bar)	=	6894.76 Pa		
	1 bar	=	**100.0** kPa		
	1 standard atmosphere	=	**101.325** kPa		
MOMENT (torque)	1 lbf in	=	0.112985 Nm		
	1 lbf ft	=	1.35582 Nm		
ENERGY/ HEAT/ WORK	1 hph	=	2.68452 MJ		
	1 therm	=	105.506 MJ		
	1 Btu	=	1.05506 kJ		
	1 kWh	=	**3.6** MJ		
HEAT FLOW RATE	1 Btu/h	=	0.293071 W		
POWER	1 hp (550 ft lbf/s)	=	0.745700 kW		
KINEMATIC VISCOSITY	1 ft^2/s	=	929.03 stokes = 0.092903 m^2/s		
SPECIFIC ENTHALPY	1 Btu/ft^3	=	37.2589 kJ/rn^2		
	1 Btu/lb	=	**2.326** kJ/kg		
PLANE ANGLE	1 radian (rad)	=	57.2958 degrees		
	1 degree	=	0.0174533 rad	=	1.1111 grade
	1 second	=	4.84814x10-6 rad	=	0.0003 grade
	1 minute	=	2.90888x10-4 rad	=	0.0185 grade
VELOCITY OF ROTATION	1 revolution/min	=	0.104720 rad/s		