Paper No. 34

THE DESIGN AND DEVELOPMENT OF THE ROLLS-ROYCE GEM ENGINE

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INTRODUCTION

The Rolls-Royce Small Engine Division began to study the design of a new generation of gas turbine engine for helicopter application, in 1966. The size of helicopter envisaged was one requiring a twin-engined installation - each engine of about 1,000 HP - likely to go into service in the mid 1970's. The eventual outcome of the study was the Gem engine and its first application that of the power plant for the Westland Lynx helicopter, an aircraft that formed part of the Anglo-French helicopter agreement of 1967.

Fig. 1 Westland Sea Lynx

This helicopter is to be used by the armed services of France, the United Kingdom and other countries. A Civil variant, the WHL 606 has now made its appearance with every prospect of substantial sales.

The preliminary study of the form the engine should take was quite exhaustive and included the examination of the possible use of a recuperator or of inter-turbine re-heat. Considerations of weight and technical risk brought an early decision to maintain the traditional thermodynamic cycle.

The basic cycle parameters of pressure ratio and turbine entry temperature were chosen making assumptions regarding component efficiencies, mechanical losses and cooling air flows which produced the relationship between specific fuel consumption and specific power depicted in Figure 2.

It was argued that in order to allow for eventual growth of the engine, the basic design should exclude turbine blade cooling and that the turbine inlet temperature should be conservative. Within this study of alternative

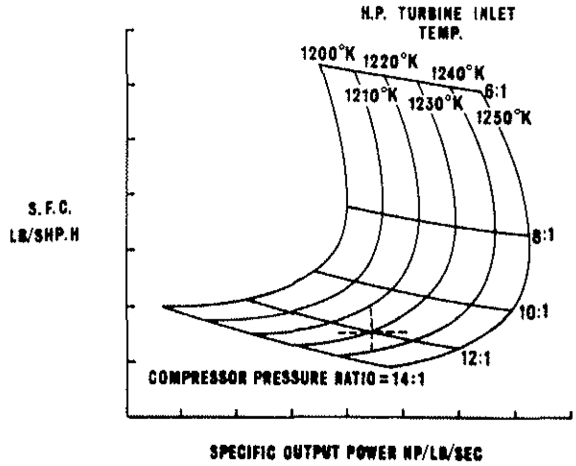


Fig. 2 Parameter Variations in BS.360 Design Cycles

cycles a number of possible design configurations were generated. Some of these are worth showing since they illustrate the change in engine lay-out associated with variations of the cycle parameters - notably the engine pressure ratio. The simplest cycle (6:1

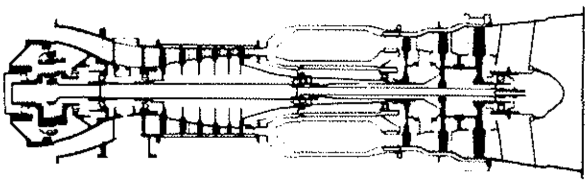


Fig. 3 BS.360 - 03 Engine

pressure ratio) is satisfied by the configuration shown in Figure 3. For a reasonable compressor efficiency and because the pressure ratio is low, a wholly axial compressor system is used and can be driven by a single stage gas generator turbine on a single spool without variables. Compared with engines of higher pressure ratio the BS.360 -03 is cheap

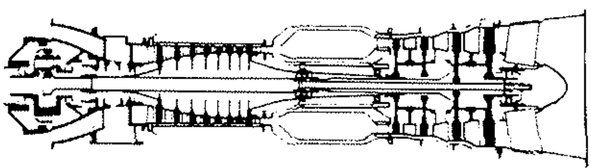


Fig. 4 BS.360 - 05 Engine

and light, but of course, suffers on fuel consumption. This high fuel consumption robs the aircraft of pay load over ranges exceeding about 200 n.m. However, the strongest case for going to a more sophisticated cycle and engine is that the potential improvements in efficiency associable with development in tech-

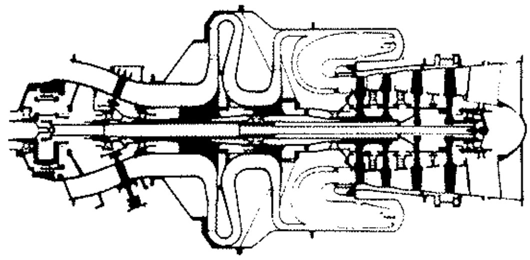


Fig. 5 BS.360 - 06 Engine

nology are very much greater. (Figure 2). An important factor influencing our selection of the RS.360 -07 configuration was the prospect it offered of being the corner stone of a family of improving engines.

The selection of a higher cycle pressure ratio gives rise to the alternative design arrangements shown on Figure 4. The BS.360 -05 is an extension of the -03 to higher pressure ratio maintaining a single spool arrangement. It is therefore restricted as regards pressure ratio (and therefore overall efficiency) by the degree of complexity of the variable geometry one is prepared to introduce into the compressor system. At the time, any higher pressure ratio than that postulated for the BS.360 -03 involved the introduction of a second gas generator turbine stage. By adopting a wholly centrifugal compressor system it is possible to achieve higher pressure ratios and retain a single spool gas generator.

The BS.360 -06 (shown in Figure 5) is such an arrangement and the increased diameter generated by the centrifugal compressors allows the use of a reverse flow combustion system wrapped around the gas generator turbines. This arrangement would be at least as efficient as the BS.360 -05 (the increased pressure ratio out-weighing the reduced compressor efficiency) and would probably be cheaper to manufacture.

However, the engine weight is prohibitive - 40% heavier than the -05. These considerations finally led to the BS.360 -07, - the Gem Mk100 as we know it today. The adoption of a two spool gas generator allows elevation of the cycle pressure ratio, and hence the engine's efficiency, without the need to introduce variables in order to maintain satisfactory handling. The use of a centrifugal compressor for the HP stage equates well with the use of a reverse-flow type of combustion system and avoids the use of very short axial compressor blades. The axial LP compressor, being of modest pressure ratio, and reasonable blade span, contributes to a reasonably maintained overall compressor efficiency. The pressure ratio chosen provides a high thermodynamic cycle efficiency and can be achieved within an engine which is both short and light.

In optimising the cycle on which the engine was based, exclusive consideration was given to the "design point" performance - i.e. the performance at maximum power. However, satisfactory as this is from a comparative point of view, it is to be remembered that the important aspects of performance in the field are low s.f.c. when cruising (i.e. around half power) and the ability to provide high contingency power from one engine should the other fail.

Although performance considerations largely determine the basic form of the engine, they were not paramount in the list of objectives presented to the design team. The virtues sought, in order of priority, were:-

1) Reliability and safety.

2) Ease of maintenance.

3) Good mission performance.

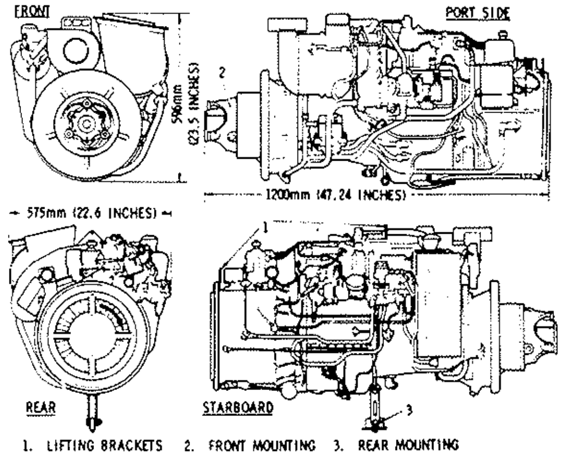


Fig. 6 Gem Installation Diagram

With all these factors in mind, the project was finally evolved including a number of basic design features:-

a) A 2-spool gas generator, at once providing adequate surge characteristics and hence good handling qualities and at the same time avoiding the use of variable geometry or blow off and their associated complication of the engine control system.

b) Allowance for subsequent growth of the engine beyond the original performance standard. To this end, blade cooling was excluded in the first instance and a relatively lightly-loaded 2-stage power turbine was adopted.

c) A reverse flow combustion system - reducing shaft lengths and avoiding possible whirling problems.

d) Forward drive from the engine allowing its installation behind the rotor and so providing a good field of view for the crew of the aircraft.

e) Modular construction - making possible repair in the field by replacement of deficient modules and the direct substitution of factory tested units.

f) Interchangeability of port and starboard engines.

g) The ability to operate on a wide range of fuels without loss of performance or reliability.

h) An overhaul life of not less than 600 hours on entering service, improving to better than 1200 hours over 3 years.

ENGINE GENERAL DESIGN DESCRIPTION

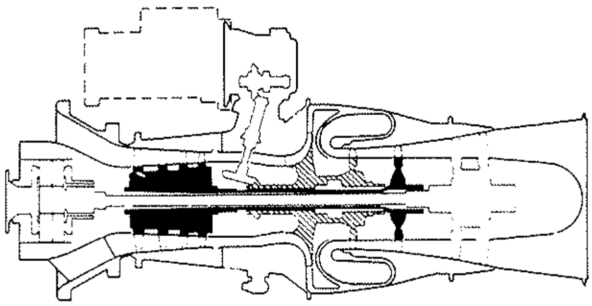


Fig. 7 Gem Two-Spool Turboshaft

The Gem engine comprises a two-spool gas generator and a two-stage free power turbine, with a through shaft to a front mounted integral reduction gear, providing a forward drive at 6000 RPM.

The two-spool gas generator has a four stage axial flow low pressure compressor driven by a single stage axial turbine and a single stage centrifugal HP compressor also driven by a single stage axial turbine.

The combustion chamber is of the reverse flow type and employs a low pressure vaporising fuel injection system.

The engine is constructed from seven self contained modules and is designed to facilitate repair in the field by direct substitution of module assemblies

Accessories are grouped on the top of the engine and are arranged for good accessibility and ease of replacement.

In order to achieve the required standards of reliability, safety and ease of maintenance the engine features all the modern diagnostic and health monitoring facilities. Vibration monitoring points are provided. Most of the engine can be inspected internally through borescope ports. Magnetic chip detectors are fitted in each oil scavenge line. Spectrographic oil analysis is a recommended health monitoring technique.

The engine is completely self contained and carries with it the whole of the oil system, including the tank and oil cooler, the whole of the fuel management and control systems, and the ignition system.

The basic net dry weight of the engine is 300 pounds. The weight of the engine change unit is 396 pounds.

TABLE 1. PERFORMANCE DESCRIPTION OF THE GEM Mk.100 ENGINE

	Engine Gearbox Output (Minimum)		Specific Fuel Consumption (Max)		Output Shaft Speed
Rating	kW	(SHP)	Mg/J	(lb/SHP.h)	RPM
Maximum Contingency (2½ minutes)	671	(900)			6000
Intermediate Contingency and Maximum (5 minutes)	619	(830)			6000
Maximum Continuous	559	(750)			6000
Typical Cruise Power	310	(415)	112.0	(0.66)	6000

MECHANICAL DETAILS

The modules are self contained. Rotating assemblies within them are carried on their own bearings and individually balanced, so that module changes do not involve balance adjustments. The mechanical interchangeability of all the modules has been demonstrated. A test programme is in hand to establish the

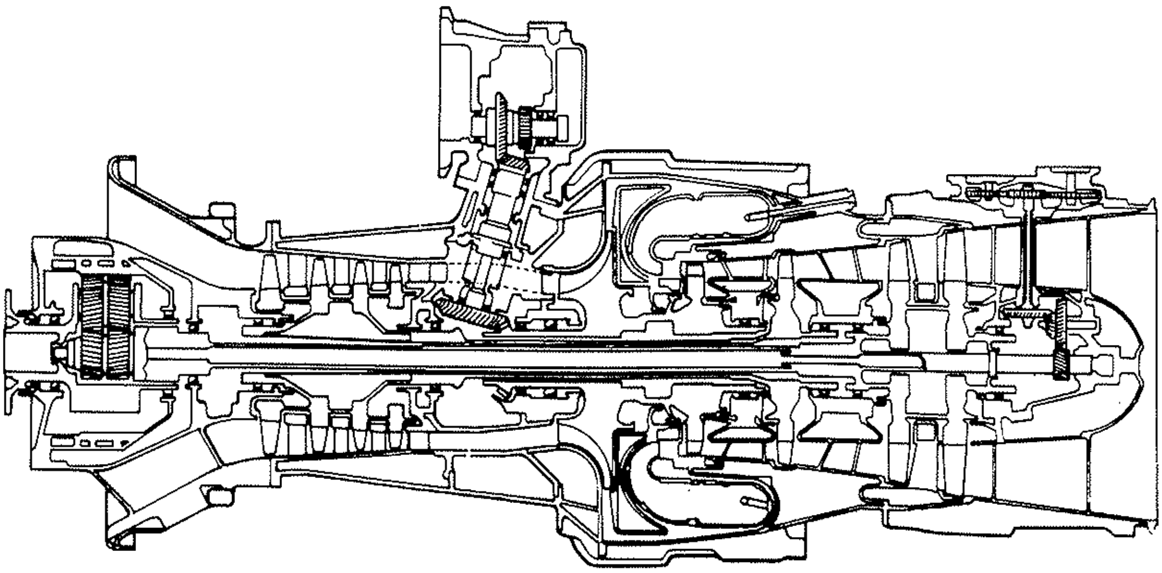


Fig. 8 G.A. of Components

permissible tolerance on the aerodynamic performance of individual modules. This programme is proceeding with high priority.

All main line journal bearings are cushioned on "squeeze films", and this has contributed significantly to the very low levels of vibration which have been a consistent feature of the engine throughout its development. The main bearings transmit their loads directly into the main casings and not through the intermediary of intershaft bearings. There are steadies between the power turbine and L.P. shaft systems but these carry no loads.

Bearing loads are transmitted to the engine mountings via carcass structures which are separate from those forming the gas annulus, and are thus isolated from the stresses due to flight manoeuvre. The problems of rotor tip sealing are thereby reduced. An additional benefit of this form of construction is that it tends to reduce secondary damage occurring in the event of a blade detachment. The released blade is able to escape from the path of the following blades and is itself arrested at a greater diameter but still within the engine carcass.

In designing the engine due attention has been paid to the problems of engines having Naval applications. Materials which are inherently resistant to corrosion have been used rather than relying on protective coatings. Magnesium alloys have not been used anywhere in the engine despite the temptation of their specific weight advantage.

COMPRESSORS

As stated before, the compressor system evolved for the Gem Mk 100 engine has enabled the exacting performance and handling targets to be met without having to embody either variable blade geometry or blow-off systems; neither has it been necessary to complicate the fuel control schedules. This simplicity makes an important contribution towards the achievement of the reliability and maintainability objectives.

The L.P. compressor rotor is a one-piece titanium drum carrying four rows of separate blades. The drum comprises four discs connected just below the blade roots by cylindrical rings carrying the interstage seal fins. The discs are supported at the front and rear by conical stub shafts picking up the inner two discs, both the outer ones being overhung. This arrangement leaves room under the outer discs to accommodate the bearing housings, providing

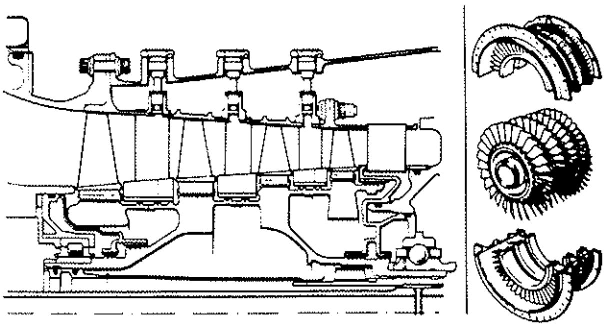


Fig. 10 L.P. Compressor

closely spaced bearings and freedom from shaft whirling problems. The drum is fabricated from four finish machined pieces which are electron beam welded at the cylindrical ring abutment faces. The construction is light and compact and free from the balance maintainability problems associated with bolted assemblies.

The L.P. stator assembly of Stages 1, 2 and 3 is split along its axis and the upper and lower halves are clamped together by bolted joints down each side. The tendency of split casing to go out-of-round has been prevented by careful attention to detail design and by the manufacturing tech-

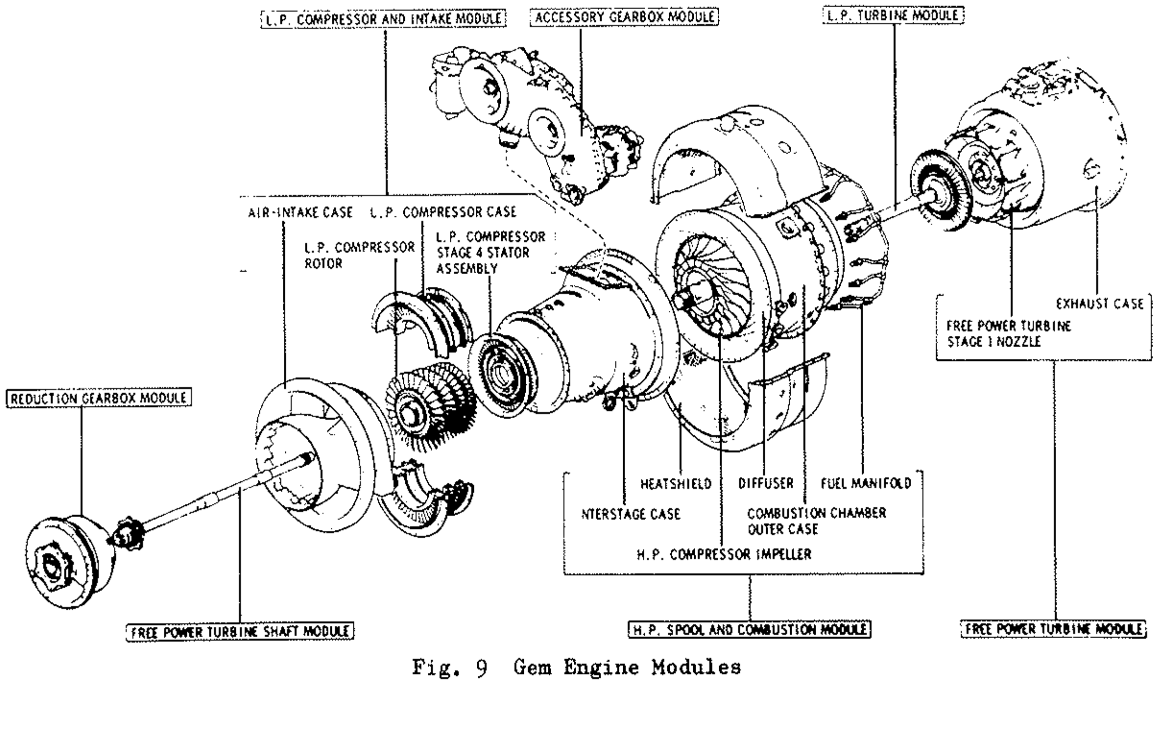


Fig. 9 Gem Engine Modules

niques used. The split casing is dowel located and rigidly bolted to continuous rings at both ends, the intake casing at the front and the Stage 4 stator at the rear. The salient points in the manufacturing technique are to make the casing as a continuous cylinder and to carry out the operations of brazing-in the stator blades and reaming of the split line clamping bolt holes prior to splitting by sawcutting, thereby ensuring true mating. The metal removed by the sawcutting operation is replaced by bonding a shim to one side of each half to form two similar half-casings.

Abradable coatings on the bore and end faces of the plastic inter-stage inner seals, allow close running clearances. The accuracy achievable in the roundness of the stator casing permits close clearances at the rotor tips.

Air entry to the L.P. compressor is through a cast aluminium alloy intake casing having five radial spokes which support a hub housing the L.P. rotor front bearing and the main reduction gearbox.

Air is passed from the L.P. compressor to the H.P. compressor via a cast aluminium alloy annular duct having four radial spokes supporting a hub containing the H.P. compressor front bearing and the inner spiral bevel gears of the accessory drive line. The hub also provides support for the L.P. compressor rear bearing.

The H.P. compressor is of conventional design. The impeller is fully machined from a titanium forging and has nineteen swept back vanes. It is driven from the H.P. turbine via a curvic coupling clamped by sixteen set-bolts which screw into self locking wire thread-inserts in the back of the impeller hub.

The H.P. diffuser is a fabricated assembly, carrying radially disposed diffuser vanes followed, after the bend, by axial straightener vanes.

COMBUSTION SYSTEM

As previously stated, the reverse flow combustion system was chosen mainly for reasons of space and length saving, (the combustion process being one which does not scale down directly as the size of the engine is reduced).

A vaporising combustion system was chosen as being particularly appropriate to the overall conception of the engine. It does not demand high fuel pressure at the burners to atomise the fuel in order to ensure its rapid burning. The use of low fuel pressures in the burner manifold makes it possible to inject the fuel at a large number of points around the circumference of the combustion chamber primary zone, thus ensuring a satisfactorily uniform temperature distribution, without introducing excessively small metering orifices. The vaporising system is undeniably more tolerant of dirty fuel and so makes fewer demands on the fuel filtration arrangements. Moreover, the metering holes are in relatively cool upstream locations where they are not prone to blockage by carbon formation. Vaporising systems also produce very low levels of exhaust smoke.

In addition to the savings in overall length, the reverse flow combustion chamber has several other advantages to offer. The air flow path of this type of chamber renders the combustion system insensitive to compressor exit velocity profile. Furthermore, the large volume conferred by this layout gives light combustion loading.

One undesirable quality of this type of system is its high surface area to volume ratio, which can accentuate the difficulties of wall cooling.

The combustion chamber is mounted from the H.P. compressor diffuser assembly at its forward end. The rear, (upstream) end of the chamber is supported in an arrangement of lugs which are designed to accommodate thermal expansions in the axial and radial directions.

Air flow patterns in the primary zone are generated partly by the air flows through the vaporiser tubes and partly by the interaction between the wall cooling air films and the deep penetrating dilution streams. Wall cooling is by splash strips.

On engine start-up the chamber is lit by four primer (torch) fuel atomisers which are spaced around the annulus of the primary zone. The primers are lit by two high energy ignition plugs. With this system satisfactory engine starts have been demonstrated at air temperatures down to -26°C using AVTUR and AVCAT fuels and 0°C using DIESO.

TURBINES

In the turbines the designer of small engines is faced with a particularly difficult problem in selecting the best compromise between mechanical and aerodynamic requirements. An aerodynamically ideal turbine blade is unacceptably fragile and difficult to manufacture. Lower aspect ratio designs as adopted for the Gem provide robust blading, but tend to incur increased secondary losses.

The H.P. and L.P. spools are driven by single-stage axial turbines. The power turbine is a two stage design in order to achieve the required efficiency and to minimise the losses due to swirl in the exhaust system.

All turbine blades and nozzle guide vanes are uncooled, with the exception of the H.P. nozzles which use air tapped from the H.P. compressor delivery for a simple form of internal impingement cooling of their leading edges, together with pressure surface film cooling of the trailing edge.

The H.P. turbine blades are unshrouded, the advantage of the higher turbine entry temperatures possible without shrouds more than offsetting the increased tip leakage. All other turbine rotor blades are shrouded to minimise overtip leakage and reduce the risk of vibration mode failures

All blades are manufactured from castings and attached to the discs by conventional fir tree fixings. All discs are manufactured from forgings of nickel based alloys.

The H.P. nozzle, L.P. nozzle and P.T. Stage 1 nozzles are fabricated assemblies in which individual cast vanes are brazed into profiled slots in the inner and outer rings. In the P.T. Stage 2 nozzle the whole vane row and inner and outer rings are cast as one, and subsequently split into upper and lower half rings during manufacture to permit assembly over the one-piece Stages 1 and 2 power turbine discs.

MAIN REDUCTION GEARBOX

Power take-off is through an epicylic reduction gearbox housed in the air intake hub, providing an output shaft speed of 6000 RPM. The gears are of double helical "herring bone" form.

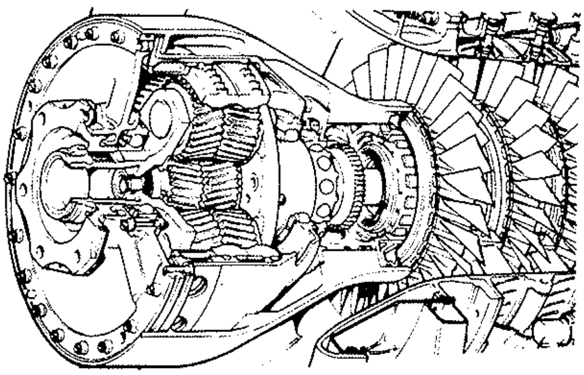


Fig. 11 Main Reduction Gearbox

Input is through a sun-wheel which is fully floating and centred by the three planet gears. Output is through a one-piece planet carrier. The planet gears also mesh with a divided annulus gear which locates the train axially. The annulus is arranged in such a manner as to ensure equal division of the load between the two helices. The helices are closely spaced axially to minimise the axial length of the gear train. The opposite handed halves of each planet gear are finished on their teeth prior to being permanently joined by electron-beam welding at their hub diameter, leaving only the bore to be finish machined. This design was evolved during the development of the gear.

Originally, the planet gear halves were bolted and dowelled together. The electron-beam welded assembly is much lighter and thus reduces the planet gear bearing loads. It is also cheaper to produce. Gear stresses are conservative and are based on the extensive experience gained from the development of the Dart and Tyne engines.

Each of the handed gear meshes is separately lubricated by oil jets.

The whole assembly is extremely compact and can be conveniently handled by one man during assembly. Through the development testing it has exhibited reliability of a very high order.

CONTROLS

The control system is designed and manufactured by the Plessey Company to comply with requirements specified by Rolls-Royce. A high level of control performance, simplicity of

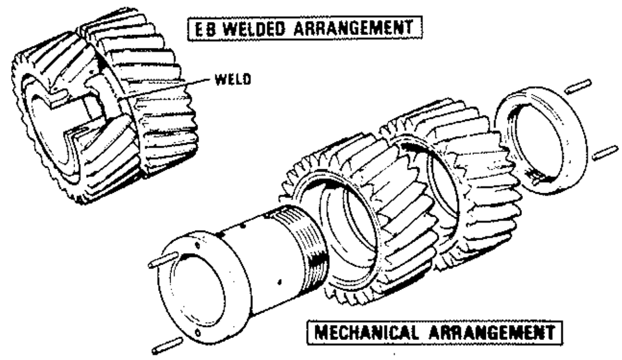


Fig. 12 Planet Pin Construction

concept, safety, reliability and ease of servicing were emphasised as overriding objectives.

The control gives fully automatic power turbine speed governing under all modes of flight, to maintain a substantially constant helicopter rotor-speed, with an inherent capability for load sharing between the two engines by means of closely matching the individual free turbine droop characteristics.

The system is basically hydro-mechanical with pneumatic actuation for the main free turbine speed control function. Electrical overrides provide protection against overspeed of any of the engine shafts and also against engine overheating.

There are two mechanical drives for the system, one geared off the HP spool, driving the fuel pump and the H.P. spool speed governor, the other off the rear side of the free turbine providing an input to the free turbine governor.

Air for the pneumatic actuation is tapped from the high pressure compressor delivery and is filtered before being used for control purposes.

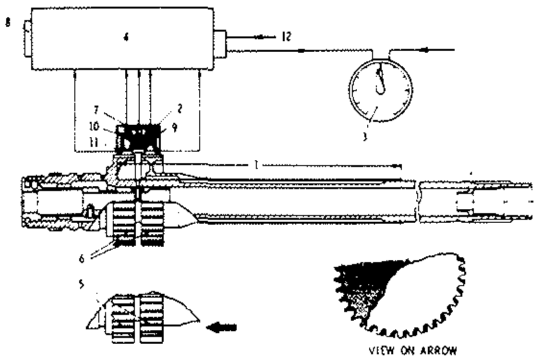
Electrical pick-ups are provided from:-

1) Six chromel-alumel thermocouples around the free turbine inlet annulus for overheat control.

2) A pulse probe facing a toothed wheel on the L.P. spool, generating an A.C. signal of frequency proportional to L.P. speed for back-up gas generator overspeed protection.

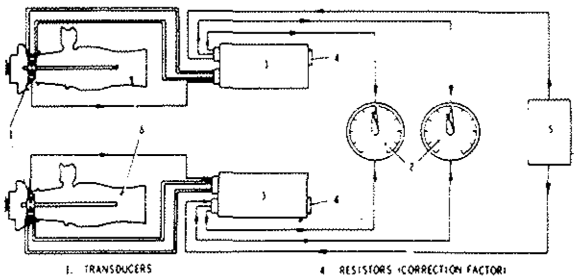
3) A pulse probe facing a toothed ring on the power turbine disc for back-up power turbine overspeed protection even in the event of a total transmission failure. All these electrical pick-ups for control are entirely separate from signals used for cockpit indication.

TORQUEMETER



1. FREE POWER TURBINE SHAFT (monitored length)
2. TRANSDUCER
3. TORQUE INDICATOR
4. PHASE - DISPLACEMENT COMPARATOR
5. WHEELS OUT OF ALIGNMENT
6. WHEELS IN ALIGNMENT
7. TEMPERATURE SENSOR
8. RESISTORS (CORRECTION FACTOR)
9. COIL
10. PERMANENT MAGNET
11. PICK - UP STATOR
12. AIRCRAFT 28v d.c. SUPPLY

Fig. 13 Torquemeter Operation (1 Engine)



1. TRANSDUCERS
2. TORQUE INDICATORS
3. PHASE DISPLACEMENT COMPARATORS
4. RESISTORS (CORRECTION FACTOR)
5. AIRCRAFT 28v d.c. SUPPLY
6. ENGINES (PORT AND STARBOARD)

Fig. 14 Torquemeter System Diagram (2 Engines)

A phase displacement torquemeter system provides the pilot with an accurate indication of power being delivered by each engine.

The system consists of a number of components which together provide an electrical signal directly related to engine torque which is used to drive an instrument in the cockpit graduated in percentage torque.

The engine mounted components are:-

1) The torque tube. This forms part of the engine shaft connecting the free turbine to the main reduction gearbox. The torque tube has slotted discs attached to each end which form the rotors of phase generators.

2) The torquemeter transducer. This consists of the stator parts of the dual electro-magnetic phase generators, and a temperature sensing element for providing compensation for variations in shaft stiffness due to temperature.

The helicopter mounted components are:-

3) The phase displacement comparator, which computes the phase angle between the two phase generators, compensates the resultant signal for changes in torque tube stiffness due to temperature, and provides an output signal related to torque output from the engine.

4) The cockpit indicator, calibrated in percentage torque, and having duel concentric needles for clear presentation of the power matching of the two engines.

INSTALLATION

Port and Starboard engines are interchangeable.

The conventional European two fire-zone approach has been abandoned. Instead, all external surfaces of the carcass which are sufficiently hot to raise the risk of igniting any fuel and oil leaking onto them, are covered with insulation shields. By this precaution, plus forced ventilation of the engine bay, conditions condusive to spontaneous ignition are avoided.

THE DEVELOPMENT PROGRAMME

Although consideration was given to adopting a demonstrator programme for the initial development of the Gem engine this was precluded by the need to simultaneously develop the aircraft and engine combination. Accordingly, when component manufacture began in 1968 the division was committed to producing units for both engine and aircraft programmes, with a minimal time to clear the engine for flight testing. Also, with an 'entry to service' target of 1976, early provisioning of production tooling and materials was essential.

Performance Development

During the conception of the engine it was realised that to obtain the inherent advantages of the multi-shaft arrangement, when applied for the first time to a small engine, a degree of optimisation and careful matching would be required. As a result, a component development programme was planned in which those features expected to present problems were examined in isolation. Rigs were constructed to permit testing of the compressor, combustion chamber and turbines. By this means individual module behaviour was measured and understood, such effects as blade tip clearance, surface finish, nozzle vs rotor capacity, windage etc. were established. Of course, further refinement was necessary in the engine and original performance diagnostic methods were developed. These demanded accurate measurements and a number of instruments were developed to satisfy this requirement including capacitance probes to measure running clearances in both rig and engine. Particularly valuable was the information revealed by using X Ray techniques.

These had previously been used on other Rolls-Royce engines and enabled a study of structural behaviour and running rotor positions with a consequent refinement of running clearances. Also used with success was helium gas trace equipment, which assisted in quantifying source identified leakage levels.

Matching and Handling

A number of performance requirements are special to the helicopter turboshaft engine application. These include very rapid response rates and the need for a good part load fuel economy. The latter is particularly appropriate to the twin engined helicopter where the average power required during normal operation is less than 50%. This is further complicated by the requirement for higher than normal powers to deal with an engine failure case. It is the requirement for good part load economy which demands a high pressure ratio of 12:1 to satisfy this demand. Figures 15 and 16 show Typical engine working lines on the L. P. and H. P. compressor characteristics and indicate the considerable range of operation required. Early development engines tended to display low speed stall which was identified as primarily due to poor L. P. compressor performance. However, development rig testing

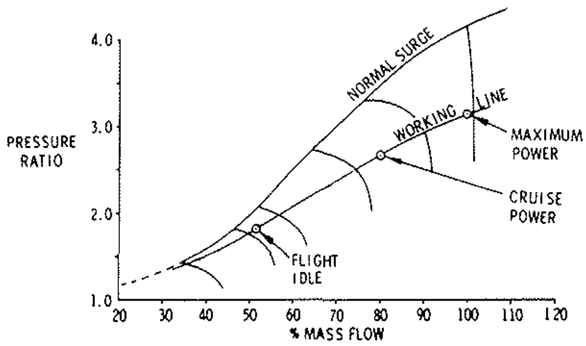


Fig. 15 L.P. Compressor Characteristics and Operating Line

was able to provide a much improved version (Figure 17) and the opportunity was taken to re-match the engine by slightly off-loading the H. P. compressor. It was also found beneficial to make small adjustments to the L. P. and P. T. nozzles. The result is an engine of improved overall performance which meets specification requirements and permits acceleration from

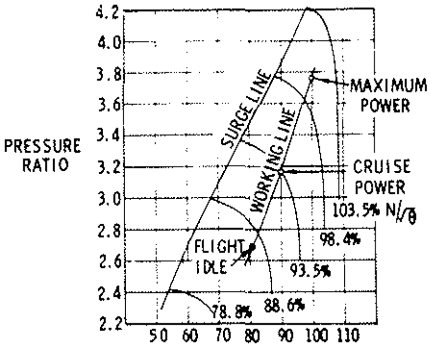


Fig. 16 H.P. Compressor Operating Line Efficiency

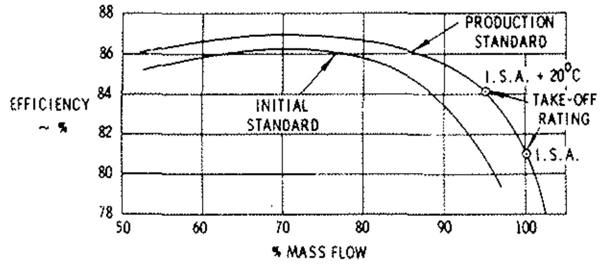


Fig. 17 L.P. Compressor Operating Line Efficiency

flight idle (zero power) to helicopter transmission limit in 2.7 seconds and from 50% power to Maximum Contingency Rating in 1.5 seconds.

Combustion

There have been few performance or mechanical problems with the small annular reverse flow combustion chamber. The low pressure-loss flame tube provides satisfactory efficiency levels over the operating power range with an acceptable delivery temperature distribution. The overall distribution factor is 30% and is achieved with a radial distribution of 6%.

As was stated earlier, an undesirable feature of the reverse flow chamber is the high surface area to volume ratio which can lead to wall cooling difficulties. In view of this a series of rig tests was initiated to study the nature of the problem and following a programme of wall cooling flow optimisation, satisfactory mechanical integrity has now been demonstrated. It is a military requirement for the Gem engine to operate on Dieso fuel and although early engines encountered ignition problems on this fuel the introduction of four torches, two of which are equipped with igniters, eliminated this difficulty.

Mechanical Behaviour

An expected problem was that of shaft whirling which in the event failed to materialise and is probably a tribute to the now 'classical' squeeze film bearings which have featured on most Rolls-Royce engines over the past two decades. A notable feature of the development programme has been the smooth operation of the engine and its tolerance of rotor un-balance.

Air and Oil System

The air and oil systems did however, require some development. Oil flows to the bearings needed optimisation such that the three sources of heat to oil, namely from environment, lubrication and churning could be dealt with. Peripheral speeds of the bearing rolling elements were in excess of previous experience and rigs were constructed to evaluate different bearing types and feed mechanisms. These rigs were particularly useful in producing co-efficients for a mathematical model expressing the behaviour of the engine system. As a result, insulation barriers were devised in order to

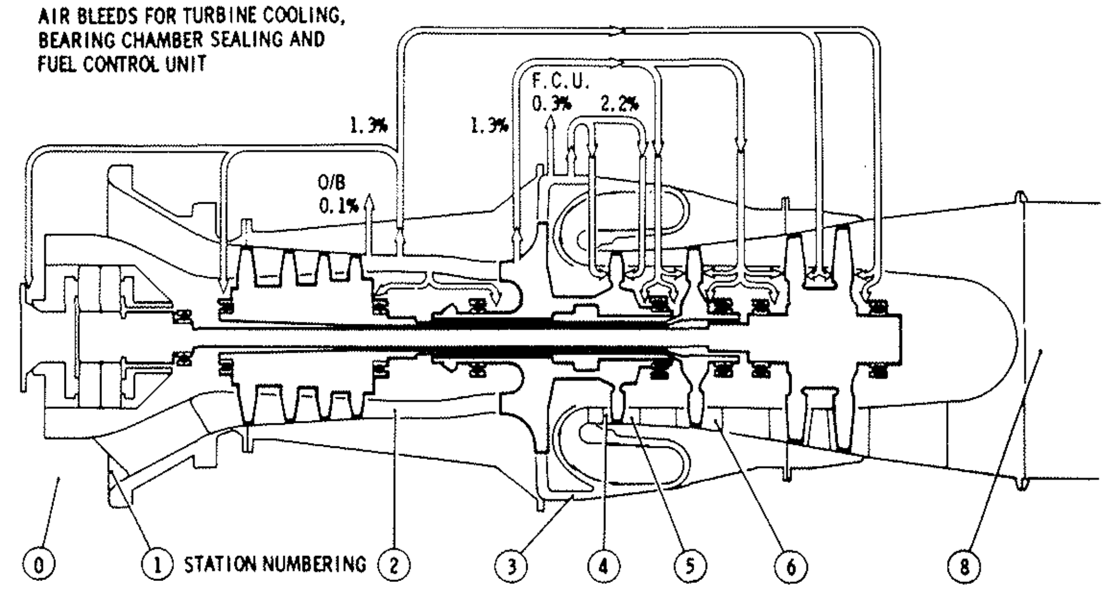


Fig. 18 Bearings and Air System

minimise heat flow to the air and oil systems. Figure 19shows where ducting was added to assist the process of cooling the bearing hubs in the L.P. and P.T.1 nozzles. Initial engines also displayed higher than specification oil consumption and investigations showed two main problems. Firstly, the external pressures to each side of the main line bearing chambers were mis-matched and this can be attributed to the matching problems resulting from the poor L.P. compressor performance mentioned previously. Secondly, where mechanical seals had been used in the L.P.T. nozzle and P.T. nozzle bearing chambers a high rate of wear was experienced.

As a result of the performance programme and some air system optimisation, the former problems were eliminated and precise bearing chamber balance was achieved over the full operation range. However, rig and engine tests showed that although the mechanical seals were capable of improvement they would remain a problem where high temperature operation was concerned. Accordingly, in the case of the L.P. nozzle bearing chamber which has the highest environmental temperature and rubbing speed, air blown labyrinth seals were adopted for the production engine. Consequently oil consumption has been improved to a level where specification levels of less than 0.5 p.p.h. can be achieved consistently.

GEM DEVELOPMENT MILESTONES - Table 2

Design Commenced	January 1968
First engine run on bench	September 1969
First engine run in rotor rig	September 1970
First flight in Lynx	March 1971
Basic type approval test	March 1972
150 hour endurance test at production ratings	January 1973
Application Type Test	July 1974
First Production engines deliveries (target)	September 1975

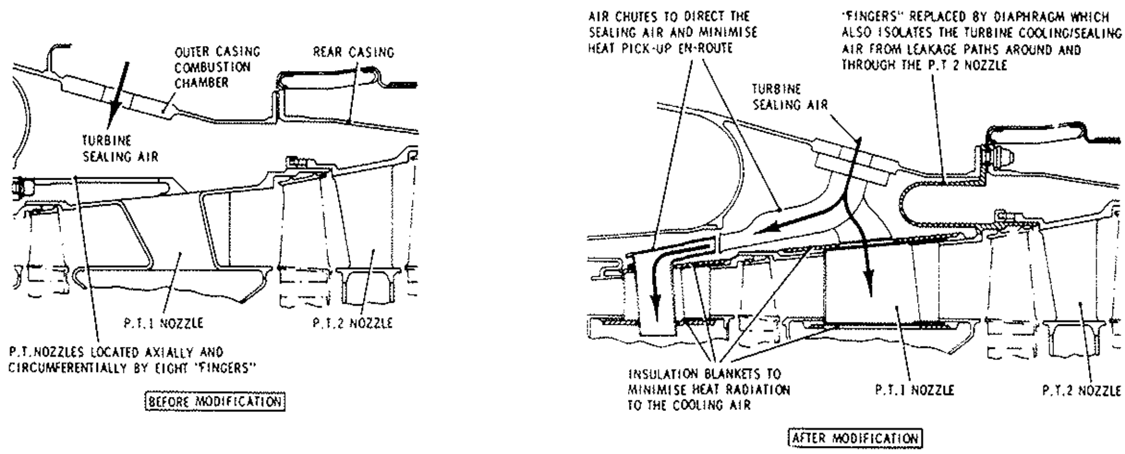


Fig. 19 Gem Turbine Cooling Arrangement

DEVELOPMENT PROGRAMME - Table 3

Numbers of Engines in Programme

Bench programme	10
Rotor Rig and Flight Programmes	41
Engine hours (Feb. 1975)	
Total Project hours	15750
Bench Hours	8700
Rotor Rig hours	1850
Flight hours	5200
Highest total running hours on a single bench engine	1085
Highest installed hours on a single flight engine (on one engine release)	350

BENCH PROGRAMME HIGHLIGHTS

Six 150 hour endurance tests have been successfully completed. The latter two of these tests were completed with engines exceeding the maximum contingency power rating of 900 s.h.p.

Most of the endurance running has been in the form of cyclic tests to schedules agreed with the U.K. services as being representative of typical service operation. One cycle in fifty includes running at the maximum and intermediate contingency ratings.

The programmes of mandatory tests and demonstrations required for type and application approvals are almost completed (February 1975). These include:-

1) Overspeed integrity (All three shafts)

2) Cut shaft integrity (Failed transmission case)

3) Over temperature integrity

4) Over torque integrity

5) Blade containment

6) Bird ingestion

7) Salt water ingestion

8) Corrosion resistance

9) Clearance of a full range of fuels and oils

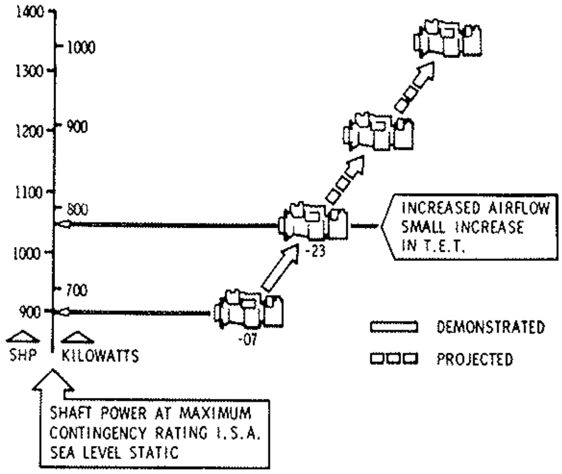


Fig. 20 Uprating Diagram

Having achieved specification performance with the basic RS.360 -07 the Division is now ready to pursue an uprating programme for the Gem. The initial variant to be offered is the RS.360 -23 which has a modest increase of T.E.T. and mass flow. This provides a maximum contingency power of 1050 s.h.p. with similar increases to the intermediate ratings. A demonstration engine is currently (April 1975) being bench tested and is comfortably achieving the increased power requirements.

Further increases in T. E. T. and mass flow will give increases of power up to 1350 s.h.p. These variants will require blade cooling, the technology for which is the subject of current component development programme.

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