# The development of an expendable gas turbine engine

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**Summary**
With the increasing use of small, unmanned target aircraft for ground to air gunnery practice, there is a growing world-wide market for small, cheap jet engines to propel them. During the last five years research work has been carried out at the Mechanical Engineering Department, Queen's University of Belfast, aimed at the production of a simple jet engine to meet this need.

T o keep production cost to a minimum, traditional aerospace practices have been avoided and instead much use has been made of the design and manufacturing methods found in the production of motor vehicle turbochargers. A virtually standard turbocharger core provided the compressor, turbine and bearing assembly. The combustion chamber is a single can reverse flow type burning kerosene injected through an atomiser in the flame tube head. The design and development of the complete combustion system has been carried out at the University. The engine is designed to produce a thrust of 550N with a specific fuel consumption of 0.14 Kg/N hr.

The first test run took place in June 1982, when a thrust of 450N was obtained. Since then development work has concentrated on improving component efficiencies to allow the design thrust to be achieved by April 1984.

## 1.0 INTRODUCTION

Since World War II the world's armed services have acquired ever more sophisticated weapons for defence against attack by aircraft.  training of personnel in the operation of these weapon systems has involved the use of simulators but more realistic situations target aircraft have been sued. These aircraft have normally been refurbished, out-of-date fighters concerted to remote control operation. The huge cost of using these target aircraft or drones has meant only limited use of this, the most realistic training practice.

In an effort to reduce the cost, attention has been turned to manufacturing small specially designed drones. The ideal form of propulsion is obviously the jet engine but small units for this purpose have usually been specially designed to aerospace production standards. This has inevitably made them very expensive and designers have turned to the sue of solid propellant rocket motors or small two-stroke engines. These cover the high and low sped ranges respectively  but have their own disadvantages. The rocket powered drones suffer from the fact that the only way to alter speed is by using expensive staging of rockets, whilst two-stroke versions are propeller driven and cannot reach high speeds. They also suffer from vibration problems.

The market would seem to be waiting for an inexpensive (i.e. expendable) jet engine in the ).5KN thrust class. Although engines are produced at present in this class the title "inexpensive" is not really applicable, cost being in excess of £20,000 per unit.

This point was made in a paper presented at the Gas Turbine Conference and Products Show, London in 1978 by Jonston, Smith and Marstiller, (Ref 1). The paper descried the development of 20HP two-stroke engines for use in RPV's for the United States Army. During the course of the presentation a challenge was given to the gas turbine industry to produce an engine in this power range to replace the two-stroke engines.

This paper describes how the Mechanical Engineering Department of Queen's University, Belfast reacted to the challenge and have produced a cheap jet engine based on turbocharger components.

**2.0 INITIAL CONCEPT**

A simple turbojet engine consists of these main components; a compressor, a combustion chamber, and a turbine to drive the compressor. The remainder of the engine comprises relatively simple casings and various sub-systems e.g. fuel, oil and ignition. Of all these components, the most difficult and expensive to develop and produce are the compressor and turbine wheels. However, modern diesel engine turbochargers contain high p[performance compressor and turbine wheels which are remanufactured in large numbers and are therefore relatively cheap. Thus, using a turbocharger core with a combustion chamber to replace the diesel engine gives the basis for a cheap gas turbine.

The project began in 1979 as a feasibility study using a turbocharger with 75mm diameter compressor wheel, delivering 0.25kg/s of air at a maximum pressure ratio of 3.0:1 with a rotational speed of 110,000 rev/min.

In order to simplify combustion chamber development, a single reverse-flow can type was used (Fig 1). To minimise the size of the combustor, a pre-mix design was first used to achieve the maximum combustion intensity. At this stage the fuel was propane gas, which was premixed with air and then burned, the flame being stabilised on a multi-hole baffle plate (Fig 2). Although a higher than average value of combustion intensity was achieved in this chamber, the stability limits were narrow because of the high degree of mixing between the fuel and air. Good stability as high values of air/fuel ratio is usually helped by having regions of rich mixture in the primary burning zone which keep the flame from being extinguished. Although this combustion chamber performed well both on propane and paraffin, it was felt that the potential development problems were too great compared to more conventional design. Therefore it was decided that the premix chamber should be abandoned in favour of direct fuel injection into the primary zone. propane was used as the fuel because at atomising nozzle which would be used with paraffin was not available for such a small fuel flow rate. The hole pattern used for the flame tube of the chamber is shown in Fig 3. Ignition was achieved by means of an automotive spark plug powered by a capacitive discharge system. After some adjustment of the hole sizes and positions, good burning was obtained with reliable ignition. An overall pressure loss of 12% of inlet total pressure was measured, which was rather high. However, at this stage reliable burning was considered to be the most important factor. The combustion chamber was mounted on the turbo-charger, which was successfully run as a jet engine, giving about 100N thrust at a turbine entry temperature of 925C. The unit is shown in Fig 4.

The success of this first demonstrator engine persuaded Normalair Garrett LTD to provide funding to build a larger engine; based on a 125mm diameter compressor wheel, with the object of obtaining 600N thrust. A market survey by NGL had shown that this  was the size of engine required for RPV applications.

**3.0 DEVELOPMENT OF THE 600N ENGINE**

The first step in this process was to determine the basic cycle parameters, namely the desired values of compressor mass flow and pressure ratio to give the required thrust. Such calculations, using typical values of compressor and turbine efficiency, gave the results shown in Fig 5, where specific fuel consumption is plotted against specific thrust for various values of pressure ratio and turbine entry temperature(TET). The maximum allowable TET was assumed to be about 1200K, initially. This value would obviously depend on the required engine life. Fig 5 shows that for 1200K, maximum thrust is obtained at a pressure ratio of 5:1, although only a small reduction is indicated by running at 4:1, which is then normal maximum value for a turbocharger compressor. Having decided on the required compressor and matching turbine performance, suitable units were obtained from Holset

Engineering Company LTD., built up as special turbocharger. In order to demonstrate the new turbocharger operating as a jet engine, the next step was to design and develop a suitable combustion chamber.

**3.1 PROTOTYPE ENGINE COMBUSTION CHAMBER**

The design requirements for this combustor were small; small diameter, wide stability limits and simple construction. Outer casing maximum diameter had to be 127mm to keep the engine frontal area within limits and this required the use of a flame tube of around 100mm in diameter. The combustion intensity was now reaching the limits for this type of chamber although a reasonable combustion efficiency of 90% was still possible at high altitude.

The wide stability characteristics of the swirl atomiser ensured that there would be no problem in this area. A spray angle of 90 degrees proved to be the optimum for the configuration chosen.

Following the encouraging results achieved with smaller units it was decided to retain the simple cylindrical flame tube of stainless steel. No primary zone swirler was used because of the complexity and hence costs. Adequate primary zone recirculation was obtained using radial jets. No cooling slots were included although a ring of holes around the periphery of the flame tube head provided cooling along the wall of the primary zone. The dilution zone also included a row of small holes providing a film of cooling air along the wall of the turbine duct.

A surface discharge igniter was positioned adjacent to the atomiser in the flame tube head.

The unique design of the compressor casing presented problems in achieving an evenly distributed airflow to the combustor. The air entered the outer casing through a ring of holes and then flowed along the annular space between casing and flame tube in normal reverse flow fashion. The main problem was the high tangential component of velocity of the air through the outer casing holes. This caused a high degree of swirl in the annular space and hence in the flame tube.

In the absence of a water flow rig the technique using titanium dioxide and oil to trace surface flow directions was used. This provided valuable information and led to a solution. This involved using straight axial splitters in the annular space with each outer casing hole supplying air to each passage so formed. The result was evenly distributed axial flow, with the advantage of a low pressure loss across the outer casing holes.

Although the flame tube hole pattern was based mainly on experience obtained with the earlier combustors, an analytical method was used to optimise the design. This method used empirical relationships for discharge coefficients and jet penetration data.

The resulting combustion chamber has proved to be very satisfactory. Overall pressure losses 8% of compressor delivery pressure. It has provided reliable ignition for the engine and its very wide stability limits can cope with sudden acceleration and deceleration of the engine.

This combustion chamber was mounted on the new turbocharger unit to produce a demonstrator engine.

The demonstrator performed satisfactorily, although it indicated the existence of some matching problems between the compressor and turbine.

## 3.2 FUEL AND LUBRICATION SYSTEMS

It must always be remembered that the objective of this work was to produce a cheap jet engine. Therefore when examining the possible solutions for the various systems, simplicity and ease of manufacture had to be the chief consideration.

The type of fuel injector chosen for kerosene burning in the engine was the swirl spray atomizer, which consists of a circular cylinder into which fluid is forced through tangential openings in the cylinder wall. These openings are either holes or slots. The fluid then passes through a converging section where the tangential velocity increases until it reaches the end of this swirl chamber where it discharges through a central orifice. The vortex created leaves a core which is concentric with the atomizer axis. As it is continuous through the orifice it is at the same pressure as the external atmosphere. The fluid is thus in the form of an annulus as it emerges from the orifice and this spreads out into a conical sheet of fuel which subsequently breaks up into fine droplets. The flow rate and cone angle of the spray can be altered by changing the geometric design of the atomizer.

For any particular design the flow rate and angle of spray can also be changed by altering the supply pressure and herein lies the problem of any atomizer. On an engine the maximum/minimum ratio of required fuel flow might be in excess of 10:1. This in turn requires the maximum/minimum pressure ratio to be 100:1. The minimum acceptable supply pressure for kerosene is around 1 bar and therefore a 100 bar pump is needed. Spray drop size varies inversely with pressure and this places a limit on the lowest pressure which can be used.

Many designs have used different methods for overcoming this problem, one of the most widely used being the Duplex. In this type two concentric orifices are used, one for low flow rates and one for high, with fuel pressure being used to operate a valve between the two. Another type is the spill-return atomizer. Instead of a plain back wall in the swirl chamber it has one or more orifices which "spills" flow back to the pump, flow being controlled by a valve. The advantage of this type is the fact that full fuel supply pressure can be used at all times. The flow is changed by spilling fuel from the swirl chamber but this does not effect the swirl velocity of the remaining fuel, only the axial velocity. As a result atomization improves with smaller drop size but cone angle increases. The latter characteristic is a problem in a combustion chamber as there is usually only one optimum spray angle. However, the system is simple, all orifices are sized for the full flow rate and hence blocking is reduced. The fine spray and wide angle at low flow rates can aid ignition.

For these reasons the spill type was chosen for this projects where simplicity was important from cost and reliability viewpoints. The only trouble was the cost of buying one which would have to be purpose designed. It was decided to attempt to design and manufacture atomizers suitable for the work as this would allow the flexibility of making changes during development.

Radcliffe (Ref. 2) was among the first to present results from a comprehensive test program to show the effect of fluid properties and geometry upon atomizer performance. This work formed the basis for a design method which was developed at QUB as part of the combustion chamber design program. The spill nozzle which resulted is shown in Fig 6. This was manufactured in the Departmental workshop and on test was found to meet its required fuel flow performance, combined with good atomization at all flow rates.

At the beginning of this project, much thought was given to the bearings and their lubrication. For a short life engine such as this, it would be possible to run with ball bearings, lubricated by a total loss drip feed oil supply. While this approach held many attractions, it meant a complete redesign of the turbocharger bearing housing and the possibility of many unforeseen development problems in changing to ball bearings. This could have caused an unacceptable delay in the engine program. On the other hand the standard turbocharger bearing assembly was a robust, well proven design, which, if used without modification, would greatly reduce the engine development work. However, the standard turbocharger uses pressure fed plain bearings for both the journals and thrust bearing, so a recirculating oil system had to be used. Because of the large heat pickup by the oil on its passage through the bearings and out of the housing, some means of cooling the oil would have to be provided. An air cooler was undesirable because of increased engine drag. The only other possibility was a fuel-cooled heat exchanger, allowing heat from the oil to be dumped into the fuel on its way to the spill burner. This system had the dual purpose of cooling the oil and raising the fuel temperature, improving vaporization and giving better combustion. The heat exchanger was designed to fit around the engine air intake so that a small amount of heat could also be lost to the incoming air-stream.

Oil and Fuel Pumps - From the point of view of manufacturing cost and reliability it was essential that the standard turbocharger core assembly should be left unmodified This meant that there was no means of obtaining direct shaft power to drive the ancillary equipment such as the oil and fuel pumps.

Another possibility was to use compressor bleed air to drive an air motor or air turbine which could drive the pumps. This still remains a viable option, the only difficulty being at start up when compressor delivery pressure is very small.

it was finally decided to opt for an electric motor to power the pumps, since almost all RPVs have DC electric supply on board to operate the control system. Provision was also made to allow a wild frequency alternator to be incorporated in a bullet in the air intake and to be driven by an extension of the main shaft, if required.

**4.0 PROTOTYPE 600N ENGINE DESIGN**

Having obtained a suitable turbocharger to give the required flow rate and pressure ratio and having developed combustion chamber and sub-systems, the next step was to produce a suitable engine package which could be sensibly used as a flight vehicle power unit. The initial demonstrator unit had used a standard turbocharger complete with compressor and turbine torus covers, which are large and very heavy. The layout of the demonstrator is shown in Fig 7 which indicates that the unit has a large frontal area. For a prototype engine a replacement compressor casing was designed by Normalair-Garrett LTD and cast, as shown in Fig 8. This was actually a plenum in which the combustion chamber was mounted, the air entering through holes in the outer can, thus producing a very compact engine layout. The very heavy turbine volute was replaced by a torus of two stainless steel spinnings welded together. These two new parts were bolted to the turbocharger centre casing, the only modification being some machining to accept the new compressor casing.

The combustor outer casing was seam welded stainless steel tube with a spun end and flanges at either end to accept the atomizer and a v-clamp for the turbine duct. This duct was a complex part, turning the gas through 90 degrees while changing diameter along its length before joining the turbine torus tangentially.

The compressor casing incorporated a one piece impeller shroud and vanned diffuser ring. It was hoped that this would be an improvement on the vaneless diffuser used in the standard turbo-charger. It was anticipated that there would be a significant increase in compressor pressure ratio and efficiency and that the higher efficiency would be maintained at the top speed.

At the turbine end the shroud and nozzle guide vanes were a single machined component of stainless steel.

The intake of the engine is formed by the fuel/oil heat exchanger. This consists of a long coil of aluminum tubing sandwiched between two concentric skins. The space between the skins and the tubing has one fluid flowing while the other counter-flow in the tubing itself. The internal diameter of the heat exchanger is large enough to include a shaft driven alternator package.

Oil and fuel are pumped by two "ganged" pumps driven by one 24V, 100W motor. The oiltank is welded aluminum which in production would be molded plastic. It is wrapped aroundthe front of the heat exchanger.

The compressor shroud has tangential holes drilled in it close to the impeller tip area. Through these holes compressed air is injected from the region between the shroud and compressor casing for starting the engine. The air enters the casing through quick-release valve onto which an airline can be connected.

The plenum castings, compressor shroud/diffuser, turbine shroud, fuel and oil pump housings and numerous other small parts were manufactured by the Queen's University workshop. Extensive use was made of the University's numerically controlled machining facilities.

It took only six months to take the engine from drawing board stage to the first test.

**5.0 PROTYPE ENGINE TESTING**

The starting system for rotating the engine was entirely successful. using an air pressure of around 6 bat the engine could be spun at speeds up to 10,000 revs/min. Light-up and acceleration were then easily achieved at the first attempt.

The combined oil/fuel system was tested by passing the fuel supply from the external pumps through the heat exchanger and then into the atomizer. As the spill fuel is heated in the atomizer there are therefore two sources of heat into the fuel and the spill temperature had to be monitored closely.

The system coped well with the quantities of heat involved. The longest run time has been one hour at 70,000 revs/min where the heat transfer to the oil was around 2.5-2.7KW. The stable condition for the whole system involved an oil temperature (at exit from the bearing housing) of 134 degrees C and a maximum spill fuel temperature of 54 degrees C. This indicates that the system could cope at a speed of 75,000 revs/min as oil temperature can go as far as 200 degrees C and fuel temperature to around 80 degrees C if necessary.

Mechanically, the engine has proved very reliable. The actual core of the prototype was the same one that was used in the previous demonstrator unit. At this time it has had a total running time of over 20 hours and almost 120 starts.

Performance Testing - In order to assess prototype engine performance the following instrumentation was used.

1. Three total pressure probes and two static pressure wall taps plus one mineral insulated thermocouple at one diffuser channel exit:
2. One turbine duct static pressure wall tap plus one thermocouple to measure chamber exit temperature.
3. One thermocouple measuring gas temperature in the turbine torus.
4. Mass flow-measuring intake with thermocouple for intake air temperature.
5. Load cell used previously for thrust measurement.

The maximum thrust obtained to date has been 432N (97LBf) using an 85mm propelling nozzle at 72,600 rpm and 1225K TET. At this condition the compressor pressure ration and isentropic efficiency are both improved relative to the standard turbocharger compressor with its vaneless diffuser. However the total mass flow is reduced, which is a major factor in the lower than predicted thrust. It has also been found that losses inside the compressor plenum chamber are considerably higher than expected, one dynamic head being loss from diffuser outlet to combustor inlet. To improve this situation an additional test program is being carried out on a separate compressor test rig to improve the vaned diffuser and hence reduce the outlet dynamic head.

In addition, the turbine used in the first prototype engine has been shown to have too small a flow capacity and is therefore choking in the exducer passages, with a corresponding loss in efficiency. A new larger capacity turbine has been obtained and is currently being tested to establish its performance characteristics. It is anticipated that this turbine will give a considerably better isentropic efficiency and allow the design thrust to be obtained.

## 6.0 CONCLUSIONS

A simple, cheap turbojet engine has been built using a standard well proven rotating assembly from a production turbocharger, which should result in good mechanical reliability. Design thrust has not yet been demonstrated but rig tests on the individual components indicate that the required thrust will be achieved with correct compressor and turbine flow matching.

## 7.0 REFERENCES

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2. Radcliffe, A, "The Performance of a Type of Swirl Atomiser", Proc. I. mech.E. Vol. 169, P.93, 1955.

## 8.0 ACKNOWLEDGEMENTS

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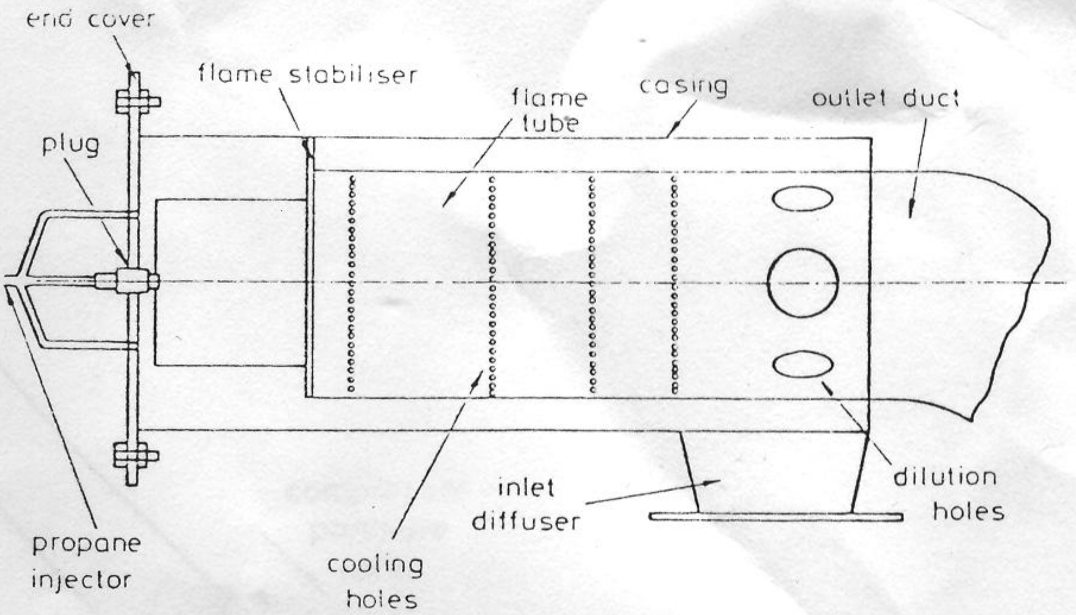


Fig 1 Premix combustion chamber

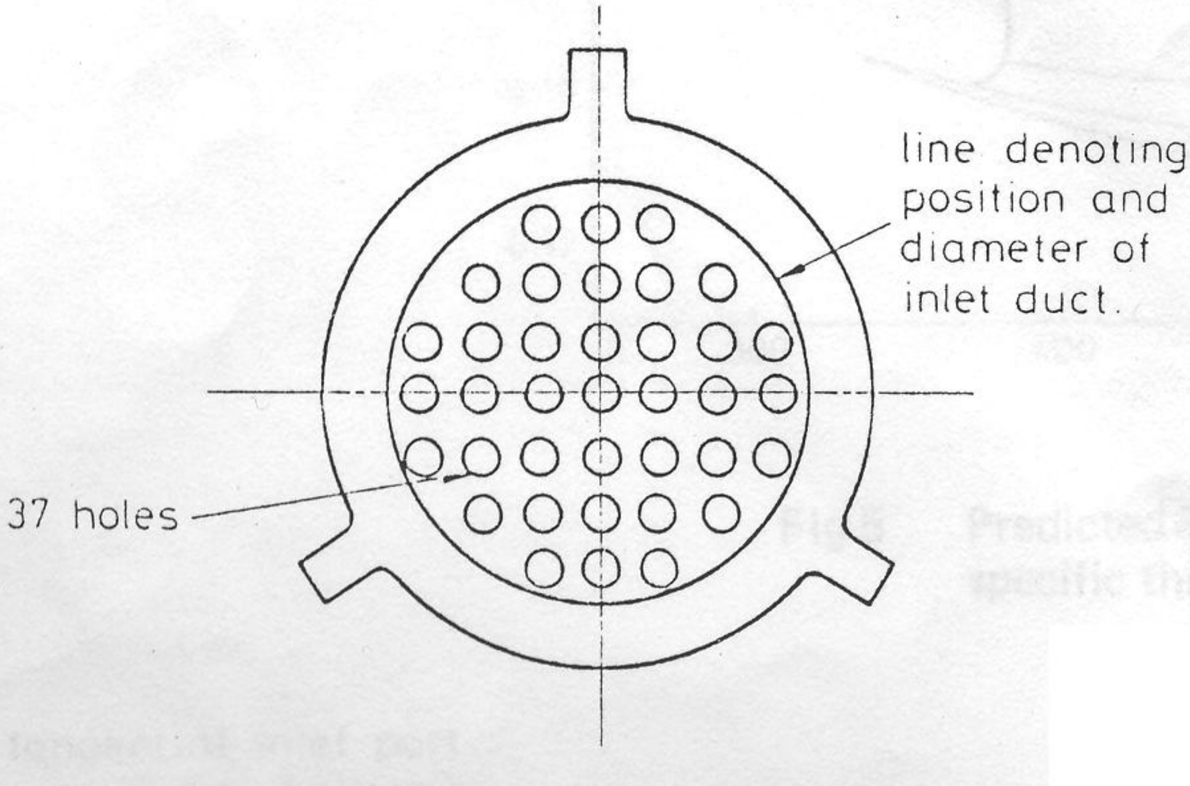


Fig 2 Baffle plate stabiliser

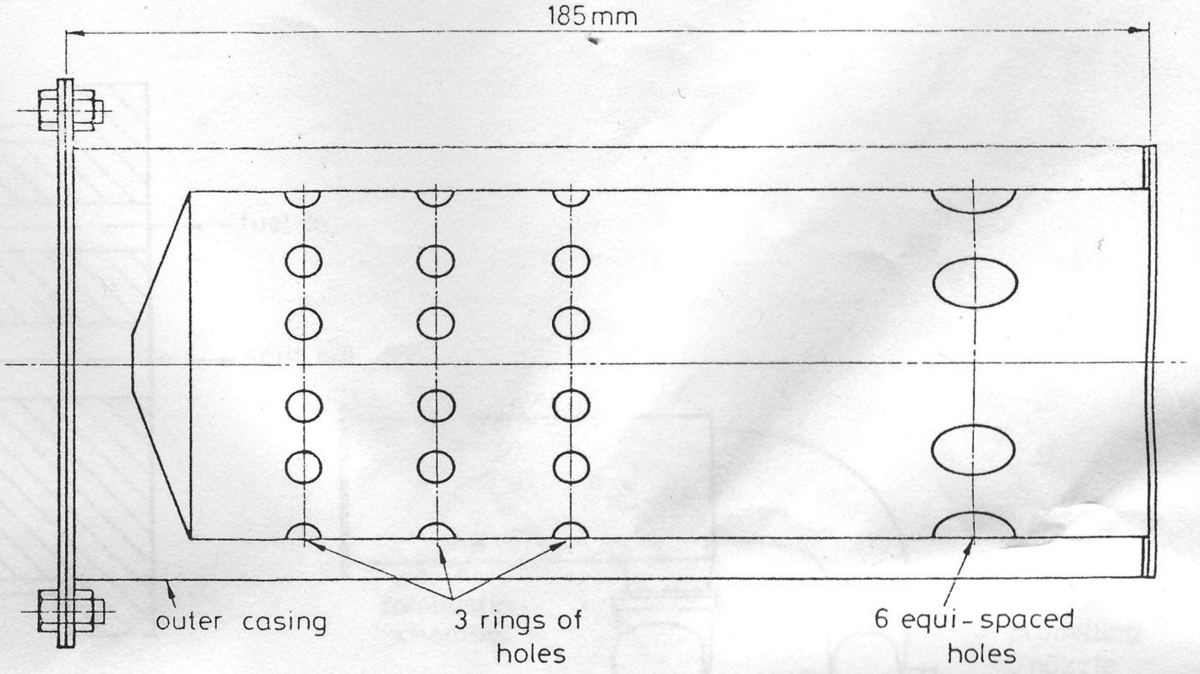


Fig 3 Direct fuel injection combustion chamber

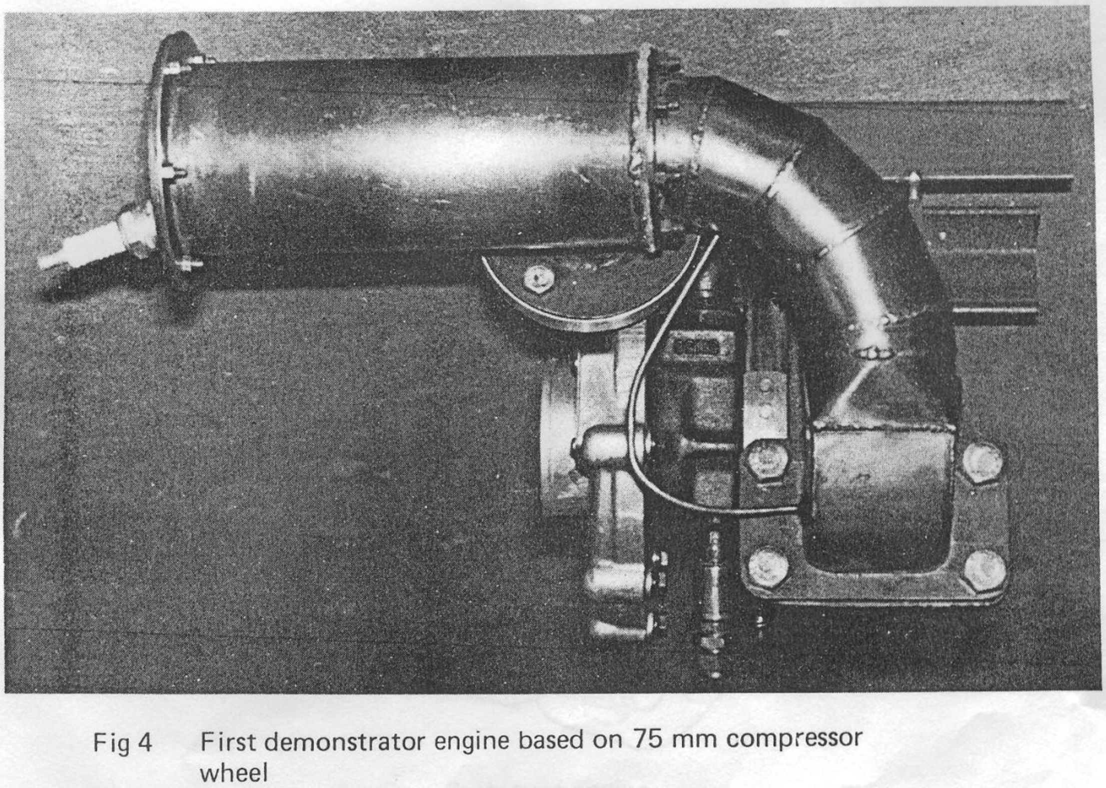

Fig 4 First demonstrator engine based on 75 mm compressor wheel

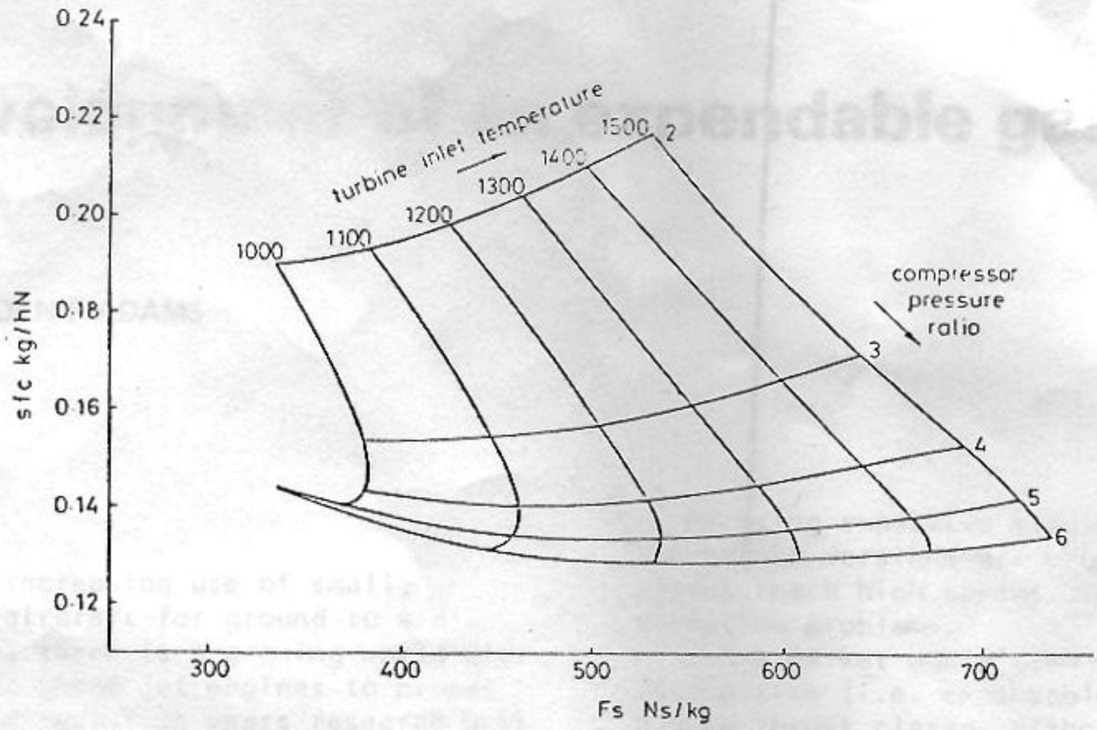


Fig 5 Predicted Turbojet specific fuel consumption and specific thrust

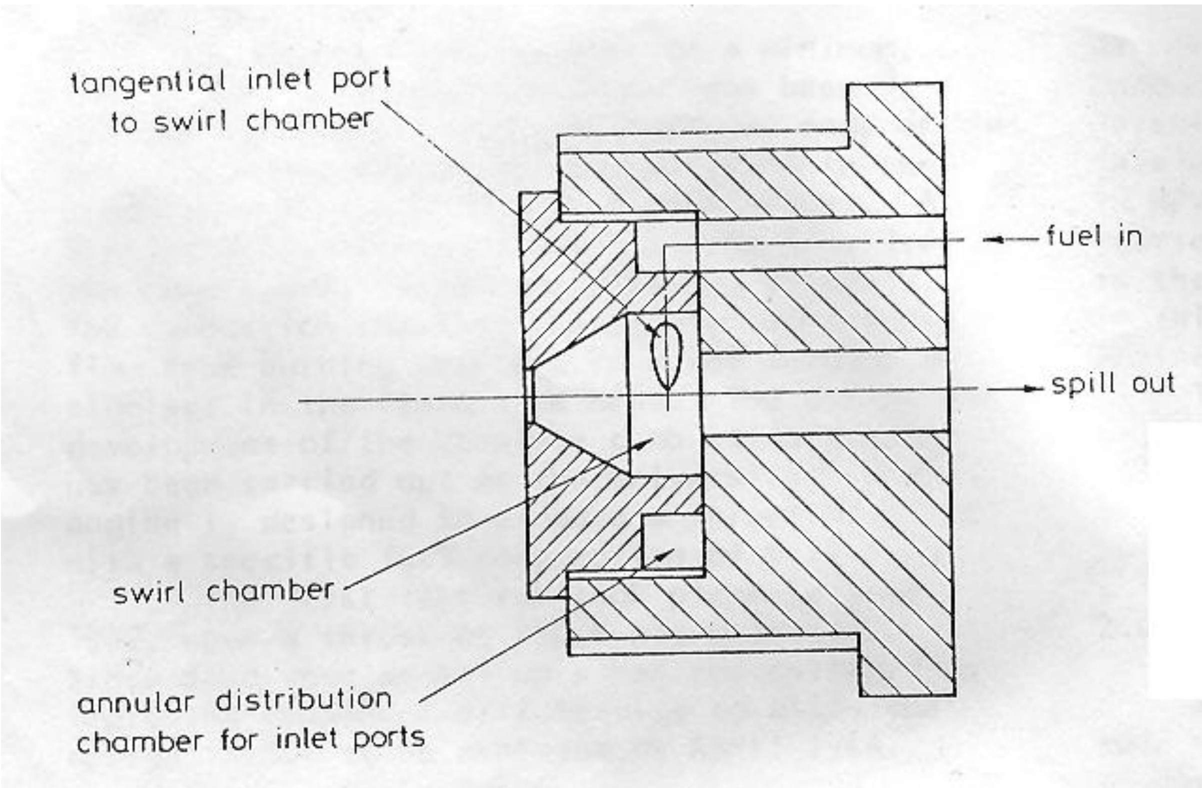


Fig 6 Fuel spray atomiser

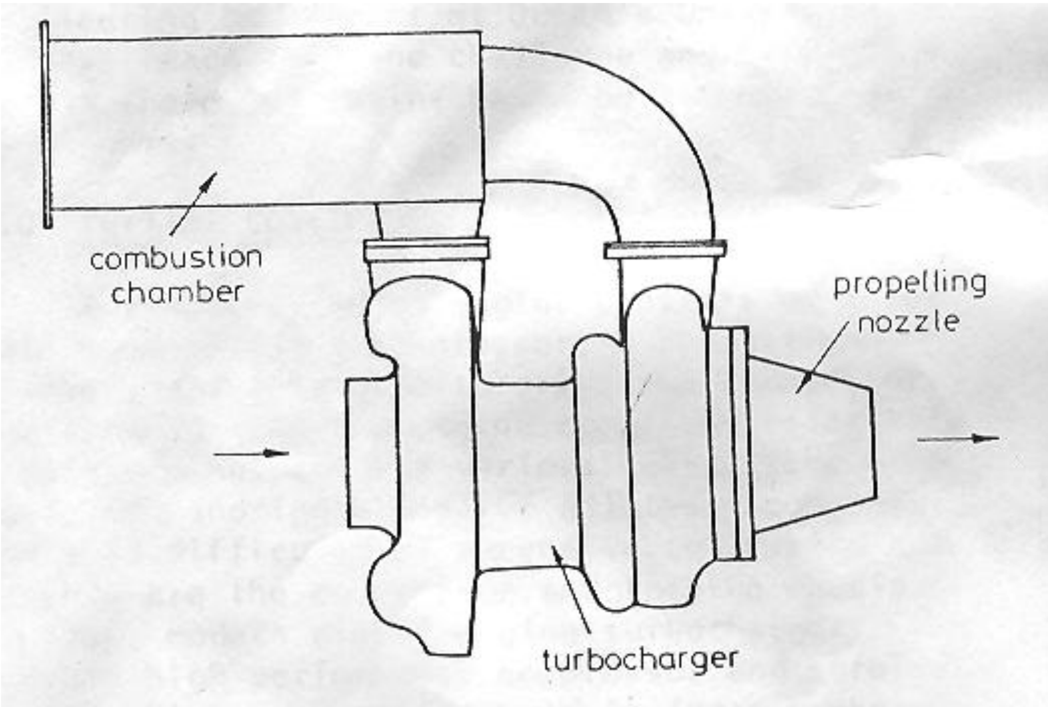


Fig 7 Layout of the demonstrator engine

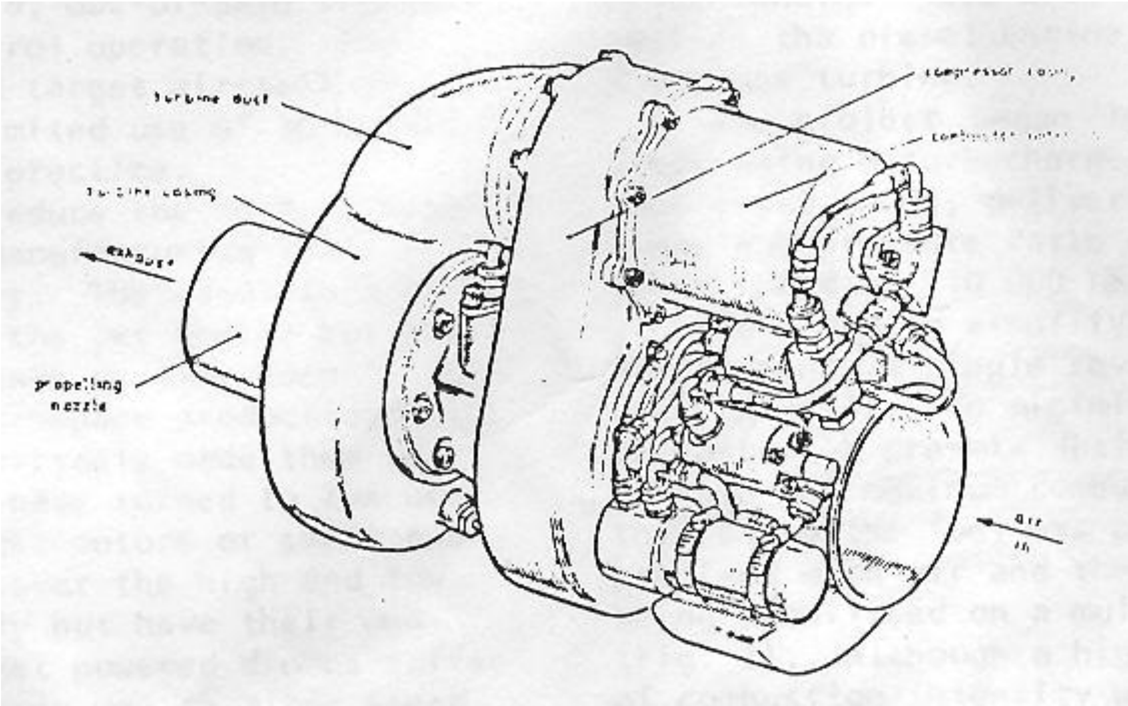


Fig 8 Prototype engine