THE DESIGN, MANUFACTURE AND SUCCESSFUL OPERATION OF A VERY SMALL TURBOJET ENGINE

Franklin van den Hout
PulseJet & TurboJet Engineering
The Netherlands

and

Jo Koullen
PulseJet & TurboJet Engineering
The Netherlands

ABSTRACT

This paper documents the design, manufacture and successful operation of a very small turbojet engine for particular use in radio controlled model jet aircraft and other small and unmanned drones.

This work commenced 6 years ago and basically this engine evolved around one off-the-shelf item: a radial low pressure compressor-turbine rotor from a modern automobile turbocharger system. All other major components such as the single stage compressor diffuser, reverse flow combustor, turbine first stage nozzles and inlet and exhaust ducts are newly manufactured items.

With an overall length of 310 mm, a diameter of 110 mm and a weight of 2.2 kg, this engine formed a great challenge upon both authors with respect to the development of each component. It was both a mixture of art and science. The three challenges were:

a. The reverse flow combustor with a length of 110 mm.
b. The rotor bearing module and vibration damping (balancing).
c. The single stage compressor diffuser.

The design point performance data at ISO conditions is:

Gross Thrust:	40	Newtons
Specific Fuel Consumption:	0.225	kg/N.h
Compressor Air Flow:	0.150	kg/s
Compressor Pressure Ratio:	1.90	
Turbine Inlet Temperature:	680	°C
Rotational Speed:	100000	rpm
Turbine Fuel:	JP3/JP4	kerosene+gas fuel
Maximum Sound Pressure Level:	90	dBA at 1 mtr

This paper will discuss the manufacturing and design techniques used and the performance and handling characteristics of the engine.

NOMENCLATURE

CPR	= Compressor Pressure Ratio
DP	= Design Point
ETAC	= Compressor Isentropic Efficiency, [decimal]
M	= Mach Number
N	= Rotational Speed, [rpm]
NDM	= Non-Dimensional Mass Flow
P	= Absolute Pressure
SFC	= Specific Fuel Consumption, [kg/N.hr]
SFN	= Specific Thrust, [N.s/kg]
T	= Absolute Temperature
TIT	= Turbine Inlet Temperature
ISO	= International Standard Organisation

SUBSCRIPTS

01	= compressor inlet
02	= compressor outlet
03	= turbine inlet
04	= turbine outlet

INTRODUCTION

Since the introduction of the first operational turbojet engine, some 50 years ago by Frank Whittle and his team, these engines have become only bigger and bigger in order to meet the demands of high fuel efficiency and specific thrust. 6 years ago mentioned authors joined forces and attempted the opposite by starting the design and manufacture of a very small turbojet engine.

The availibility of readily made rotor systems, high strength and temperature resistant materials and their maching tools have made the initial step for such a project easier. After many hours of static runs and tests it is near completion and will soon be flight tested in a

Presented at the International Gas Turbine and Aeroengine Congress & Exhibition
Birmingham, UK — June 10-13, 1996

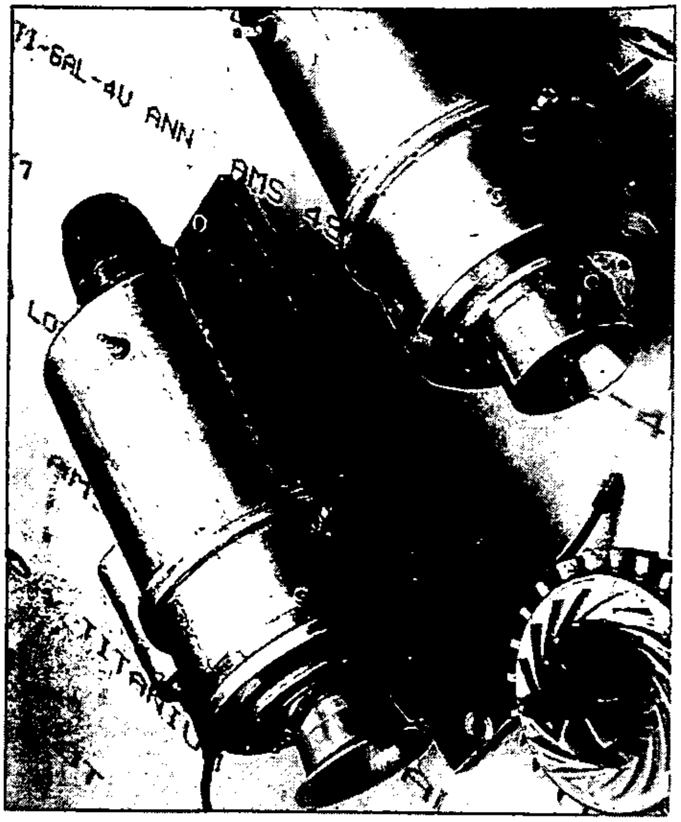

Figure 1: Turbojet Engine and Compressor Diffuser Module

deHavilland Vampire III model aircraft. Figure 1 shows a picture of this engine in its current state.

Due to its size, the thermodynamic cycle of this engine is characterised by relatively high operating temperatures, very low component pressure ratios and efficiencies and a high rotational speed of the rotor. The low Reynolds numbers encountered (1.5E5) indicate the dominance of frictional forces and losses and its influence on boundary layer behaviour. The result is a very sensitive engine in terms of operation and off-design behaviour.

PRELIMINARY DESIGN CONSIDERATIONS

Rotor Selection

The off-the-shelf rotor assembly that has been selected is an unmodified Garrett Airesearch model T3/T31 60-TRIM. This assembly, with its overall diameter of 60 mm and a weight of 300 grams has proven to be an excellent choice. The high peak efficiency, backswept vaned compressor has a wide stable operating range of air flow for a given rotational speed and is already of excellent aerodynamic design. The radial inflow turbine is made of a nickel steel, Inconel 713 alloy and is capable of handling operating temperatures of up to approximately 900°C.

Combustor Configuration

Todays low weight turbocharger rotors also have very short shafts. This leaves little space between compressor and turbine rotor (approx. 90 mm) to add other components such as the combustor, compressor diffuser and turbine nozzles. It was already concluded that positioning all of above mentioned components within this limited space would be impossible. It was unknown at the initial design stage how long the combustion chamber had to be for good combustion characteristics!

Instead of an annular straight flow combustor arrangement we therefore opted for a reverse flow annular configuration situated aft of the turbine. Although the length of the turbojet engine increased, the overall diameter was kept very small. For applications in model jet aircraft overall body diameter is more critical than length. It also allowed for unlimited axial enlargements of the combustor and eliminated the complex integration of for example a straight flow combustor with the bearing module.

This configuration introduced probably one of the most important advantages of all. Placing the combustion chamber behind the turbine isolated the much colder bearing module and compressor arrangement from the high temperatures encountered in combustor and turbine. Measurements have shown that indeed, bearing temperatures are kept at a surprisingly low 80°C throughout the stable speed operating range of the engine! It eliminated the use of special bearings and other complex cooling arrangements.

Fuel Selection

Our wide experience with lpg fueled pulsejets has led us to select kerosene fuel for this turbojet engine. Being the most widely used aviation fuel it already has good combustion characteristics when properly atomised. It is a much safer fuel to handle than gas, gasoline or lpg. Although very flammable when atomised it has a lower tendency to quickly form explosive mixtures when for example a leakage occurs in pumps or feed lines. Safety in fuel handling and engine operation has been a serious design issue.

The use of kerosene does require additional items such as a booster pump for fuel pressurisation and maybe an additional atomizing air system. The main difficulty with kerosene fuel was expected to be that of proper atomisation or evaporation within the very limited space of the combustion chamber.

Modular Design

A modular design concept has been maintained throughout the entire engine. Integration of two or more components not only makes manufacturing difficult but could make any future modifications to a specific item a tedious job. All components have been designed in such a way that they can easily be manufactured and put into series production. The turbojet engine has been devided into several modules that are combined together by a combination of bolts, press-fit connections and special clamps, while all components are aligned with each other using male-female connections.

Design Life

Ample margins in temperature (thermal stress) and rotor speed (component loadings) have been introduced to maintain long life of especially the hot section components. The maximum operating TIT of 750°C is well below the turbines capability. The maximum rotational speed of the rotor during, for example, takeoff is approximately 105.000 rpm and is also well below its design limits.

Containment after Rotor or Bearing Failure

Although the rotor is very light in weight its rotational speed is high and the risk of non-containment should not be ignored. The turbine is surrounded by other components made of high strength stainless steel 316 and Inconel 713. While normal turbojet operation is well below its mechanical and thermal limits, turbine failure has never occured throughout the years of testing. During the first series of tests of the preliminary design an unexpected axial movement of the compressor has once been experienced at a rotational speed of around 40.000 rpm. It was seen that the rubbing of the aluminum rotor against its aluminum casing caused it to decelerate very quickly to rest.

No severe damage to the other components had appeared and any broken pieces remained within the engine. The selection of good rotors without defects or cracks has also been very important.

COMPONENT DESIGN AND MANUFACTURE

General Layout

This turbojet engine has been built up from several individual modules. These are:

1) Engine intake
2) Compressor casing
3) Compressor diffuser
4) Combustion chamber
5) Turbine nozzle guide vanes
6) Lube oil system
7) Rotor bearing module
8) Compressor-turbine rotor
9) Exhaust oozzle assembly
10) Combustor/turbine casing
11) Fuel control system

Figure 2 shows a cut-away view of this engine and the layout of the individual components.

Intake and Compressor Casing

Until now only static tests have been conducted on a testbed and fig. 1 shows a typical bellmouth shaped intake that was used for good inlet flow characteristics. Aircraft speeds of around 280 km/hr have been anticipated. A new divergent intake is therefore being considered for the in-flight purpose of reducing the inlet air velocity to acceptable levels to prevent negative incidence stall at the inducer vanes of the impeller. The intake and compressor casing are two individual items made of an aluminum alloy. The compressor casing also includes a messing nozzle that is used to feed compressed air to the compressor rotor to accelerate it to ignition speed.

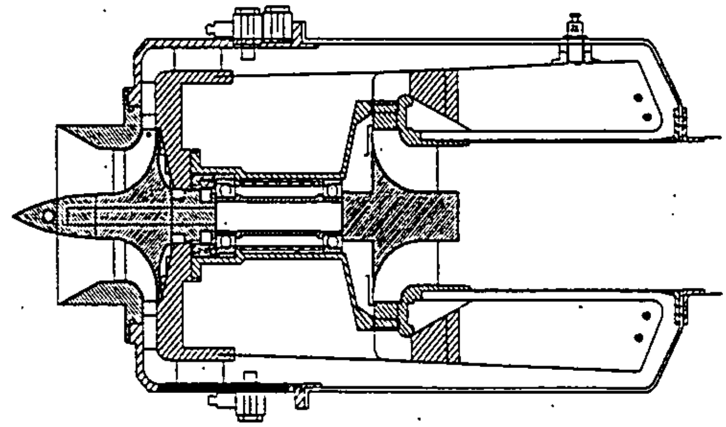

Figure 2: Cut-Away View of the Turbojet Engine

The machining of the rotor casing required high accuracy. Large tip clearances are detrimental to overall engine performance and compressor efficiency, especially on an inlet diameter of only 46 mm. Tip clearances have been kept between 0,2 and 0,3 of a mm. It was found that a tip clearance below 0,2 mm required too much effort into radial and axial rotor displacement control and alignment and was not persued any further.

Compressor Diffuser

The proposed layout of the engine and the selected radial compressor required the compressed air to be redirected to an axial flow. The rotor diameter of 60 mm was small enough to include a radial first stage vaned diffuser without exceeding our target overall diameter of 120 mm. A 90 degree axial redirector channel and a second stage flow straightener are an integral part of the diffuser and were designed to provide an axial flow of air without too much swirl. The entire diffuser has been manufactured from a single piece of aluminum alloy using CadCam technology.

The first two references have provided us with excellent tools and information for the design of our radially vaned diffuser. For our turbojet engine, impeller exit velocities were estimated to be around M=0.55. It was shown by Adkins (1983/1990) that at and above M=0.5 compressibility effects have to be taken into account when designing a diffuser and above M=0.6 these effects are considerable. These severe pressure gradients at inlet to the diffuser are caused by the large changes in air density and could easily result in early boundary layer separation and hence poor diffuser performance. The divergence angle or area ratio at inlet of the diffuser should therefore be small (conservative) but may increase progressively towards its exit. These "trumpet shape" diffuser designs are difficult to manufacture and another approach would be longer diffusers with a lower but constant angle of divergence.

For our design easy to manufacture "trumpet shape" diffuser channels were possible with socalled symmetrical "half moon" shaped diffuser vanes as shown in figure 3.

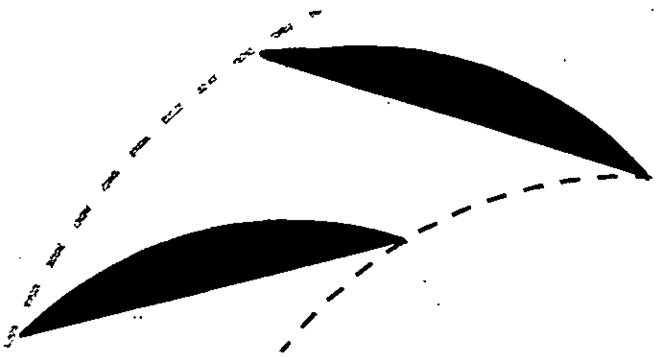

Figure 3: First Stage Compressor Diffuser Vane Geometry

Similar designs have been found in many other small gas turbine engines and auxiliary power units.

For this small diffuser the method of calculation as described by Adkins (1983/1990) can only be approximate due to other, very difficult to quantify, factors such as boundary layer behaviour and the non-uniform inlet conditions of the air entering each diffuser passage. The design and manufacture of it has therefore been much a mixture of art (a good eye for shape) and science.

Splitting a flow of air such that each diffuser passage has the same air mass flow and pressure is difficult. It is this inbalance between adjacent diffuser passages that increases the tendency to surge the compressor and experiments have shown that it is likely to increase with an increased number of vanes. Increased surface area, friction and diffuser blockage are side effects that must be taken into account when a large number of vanes is selected.

Experiments have been done with diffusers with 15, 16 and 17 radial vanes. The final configuration chosen was 17 vanes since it had the best performance within the part speed operating range of the engine. An important aspect with respect to the number of vanes is the aerodynamically induced blade vibration caused by the blade passing frequencies of rotor and stator vanes. An "unbalanced" ratio of the number of diffuser vanes to the number of compressor rotor blades is preferred. To damp aerodynamically induced blade vibration and to even out any imbalances in air flow and pressure to each passage a vaneless space of 5 mm between rotor exit and diffuser inlet was introduced.

During the tests of early designs low frequency rumbles could be heard very well at certain operating points of the engine. Pressure fluctuations also indicated such flow instabilities and the sensitive nature of diffuser design on overall engine performance.

Combustion Chamber

The design of the reverse flow combustor has greatly been a process of trial and error. With its length of 110 mm and its overall diameter of 90 mm it has reached an absolute limit with respect to the available volume for reasonably good combustion characteristics on kerosene fuel. Much effort has been put into the testing of specific cooling concepts (film, impingement) described in the literature. After the many test runs, each up to 10 minutes, visual inspections were performed. Colour differences on the liner and the appearance of soot, smoke and carbon deposits provided feedback on the quality of combustion with respect to uniform temperature distribution and local overheating (hot spots). Feedback was also obtained on the proper layout and size of these cooling holes. It was experienced that for these small combustors a correct balance of several of these cooling techniques is of most importance.

Impingement cooling of the hot gases and the combustion chamber inner walls by means of air jets has been a predominant technique in the primary and secondary zone of the combustor. Film cooling techniques have been used extensively in the tertiary zone and around the transition piece near the turbine first stage nozzle assembly. Due to the changing direction of the impinging hot gases this transition piece required extra film cooling. Without this thermal barrier local overheating and even melt-down could occur very easily.

The other very important aspect of combustor design has been that of primary zone recirculation and flame stabilisation. For this purpose the use of socalled swirl vanes at entry to the combustor has been dispensed with due to difficulties in manufacturing and the fact that kerosene fuel was not intended to be injected by means of individual orifice nozzles. Instead, the adoption of large scale primary zone recirculation using a small number of large air jets has finally led to the successful design of this combustion chamber. The recirculation holes have been introduced at a certain distance from the combustor end cover.

These recirculation holes, when correctly placed, significantly improved mixing and combustion stability and after test runs resulted in a spotless combustion chamber. When placed badly and with the wrong dimensions, it chilled the combustion process and resulted in unstable combustion with measurable instabilities in exhaust temperature and overall engine behaviour. Visible smoke emissions, extreme hot spots, non-uniform temperature distributions and even carbon deposits on the liner inner wall were then detected.

It was thought that the performance of such a small and fixed geometry combustor would deteriorate very quickly at part speed operation. The already low operating pressures of this combustor decrease dramatically at part speed and this is detrimental to flame stability. The use of the large scale recirculation principles as described above prevented this and within the stable operating range of the engine reasonably good combustor performance was maintained.

Although many correlations exist that provide a good starting point for a combustor design it still remains a true fact that the development of any good combustor, no matter how big or small, very much depends on the time spent on testing.

First Stage Turbine Nozzle

The use of a radial inflow turbine required that the hot gases be redirected from an axial reverse flow to a radially inward one. The first stage nozzle assembly is constructed from one single piece of alloy and includes not only a curved passage (transition piece) to redirect this flow of hot gases but also a set of fixed geometry radial vanes or nozzles. These vanes are used to simultaneously accelerate the hot gases and generate the correct velocity triangles at turbine inlet, such that a smooth inward flow through the turbine impeller passages is accomplished at its design point.

The fixed geometry vanes of our assembly have blunt leading edges to allow the hot gases to enter the nozzle passages smoothly from any direction. The disadvantage of fixed geometry vanes is that the flow at exit would maintain a constant angle relative to it. At part speed the change in air flow and rotational speed will rearrange the required velocity triangles at turbine rotor inlet. The situation might occur when hot gas impinges onto the turbine blades either on the driving side of the blades or in opposite direction. In general the radial turbine is seen to be less sensitive towards this reaction and the loss in stagnation pressure associated with it (shock losses) does not increase much over a wide range of incidence angles relative to the rotor. The true nozzle and turbine characteristics of this engine are yet unknown with respect to the above.

Fuel Injection, Ignition and Control

While stability limits for sustained combustion with respect to fuel to air ratio are wide, for ignition these limits are narrower. Therefore, good ignition characteristics depend very much on the fuel injector design and the achievable atomisation quality. For good ignition performance a well atomised or evaporated fuel (preferably close to stoichiometric fuel to air ratio) is required in the primary zone. The air conditions at inlet to our combustion chamber (temperature and pressure) at the very low rotational speeds are just about ambient. This is especially detrimental to ignition performance due to large ignition heat loss and the very poor fuel atomisation quality that can actually be achieved.

Two methods have been investigated to achieve above goals:

1) High quality fuel atomisation using plain orifice nozzles.
2) Fuel pre-evaporation with an evaporator tube or manifold.

The first method has been dispensed with very quickly. Small high performance orifice nozzles require high fuel pressures and heavy on board boost pumps to achieve the fine fuel sprays and are also difficult to manufacture. If multiple injectors were proposed then the difficulty of flow division would be the next problem to deal with. The most important disadvantage is that such orifices tend to create large spray cone angles and the finer the fuel spray the larger the cone angle will be. The resulting radiation due to impingement of burning droplets onto the liner inner wall is high, especially in a very small combustion chamber. Fuel control however is more accurate and the engine response time due to changes in fuel flow (for example transient operation) is very short.

It was concluded that fuel pre-evaporation would provide the best solution. Our design consists of a fuel pre-evaporator manifold located within the combustion chamber. Since the fuel and the combustion chamber are cold at startup fuel pre-evaporation can not be achieved unless the fuel is preheated to the high evaporation temperatures just before ignition. Fuel preheating makes startup procedures cumbersome and the solution was the use of a gas fuel for startup and ignition.

The advantage of an evaporated fuel has been less emissions of smoke and unburned hydrocarbons. A more uniform combustion process with uniform combustion temperature patterns and fewer hot spots could be achieved than with the first method. It can also operate at much lower fuel pressures. The engines response time to changes in liquid fuel flow is longer and actual flow control is less accurate. This is primarily dictated by the evaporation process. One negative side of elevated fuel temperatures is that of fuel cracking. This will result in carbon formation and clogging of the small holes and fuel lines. This process did not appear to be very fast and was acceptable within the already short running times of the engine.

Not surprisingly, the use of a gas fuel for ignition and engine startup has proven to be excellent. After startup to its minimum idle speed transfer to kerosene fuel is initiated through the same manifold using synchronised valves. The already hot gases in the combustion chamber then preheats the fuel in the manifold to the high evaporation levels before it enters the combustion chamber. Fuel flow is controlled by a variable speed electric motor driven miniature gear pump.

Ignition is accomplished by means of a specially in-house developed 5kV Electronic Condensor Discharge Spark Ignition Unit. The position of the spark igniter very much influenced the startup and ignition sequence of the engine. By changing its position relative to the evaporator manifold and the recirculation or cooling holes, the rotational speed at which ignition could actually occur changed significantly. Successful light-up speeds were observed to change from 10000 rpm to as low as 3000 rpm!

A trial and error process was required to find the right position. Ignition at reasonably high rotational speeds is preferred to prevent flash back due to the low pressures and flows throughout the engine. Ignition speed has finally been set to approximately 10000 rpm.

Rotor Bearing Module

The bearing module is not only the heaviest module of the entire engine but construction wise it is also the most important one. It is entirely made of stainless steel and in combination with the compressor diffuser and turbine nozzle assembly it is basically the mainframe (chassis) of the engine. The alignment of this bearing module with all other components is of most importance since tip clearances of both compressor and turbine are controlled by it.

This construction allowed the hot section components to expand freely to the rear of the engine while the compressor diffuser-bearing module combination acts as a fixed pivot. The bearing arrangement within this module is a double overhung arrangement comprising two special ball bearings that are capable of withstanding the axial loadings at high rotational speeds. These bearings require pre-loading in axial direction. This has been accomplished with a spring.

Both bearings are lubricated and cooled with a special turbine oil which is fed from the externally mounted tank to the bearing module by means of a tube through the compressor casing as shown in fig. 2. This lube oil system is of the total loss system. Only little oil is required during normal operation and a closed loop lube oil system would be too difficult to engineer and would also require a too complex sealing arrangement. This would increase the overall weight of the engine and was not persued any further. The lube oil flow, or more appropriate, the lube oil "droplet flow" is controlled by a very small orifice and the oil is fed to both bearings using compressor discharge air. Before startup, when no pressure is available, a few drops are fed manually. The oil is finally lost in the exhaust duct where it is entrained in the hot gases.

The biggest problem of all was that of rotor balancing and vibration damping. The use of the in-house capability of rotor balancing by means of a two-plane dynamical balancing machine proved to be worthwhile. It has G1.5 balancing quality capabilities and is also used to balance ducted fan units for radio controlled model aircraft.

COMPUTATIONAL STUDIES

Design Point Performance

To aid in the design process and the understanding of the turbojet engine cycle a design point performance program was developed. A simple cycle parameter study was carried out and figure 4 graphically shows one of the most important results.

It was carried out using constant, best estimate, values for component efficiencies and pressure losses while inlet conditions were held constant at ISO conditions and zero flight speed. It can be concluded that the reason for increasing CPR is clearly to decrease SFC. An increase in TIT however, has a significant positive effect on specific thrust but at the same time results in an increases in SFC. For a given size and weight of the turbojet engine CPR is more or less constrained to the narrow band as indicated by the shaded area in fig. 4 and the only "handle" left is that of TIT. The main design target for these small turbojet engines remains a high thrust to weight ratio, thus a high value for TIT but will always result in an increase in SFC. No true optimums are available to us at all.

The only way SFC can really be improved is by increasing either component efficiencies or by reducing the pressure losses and tip clearances of respectively combustor and rotating components.

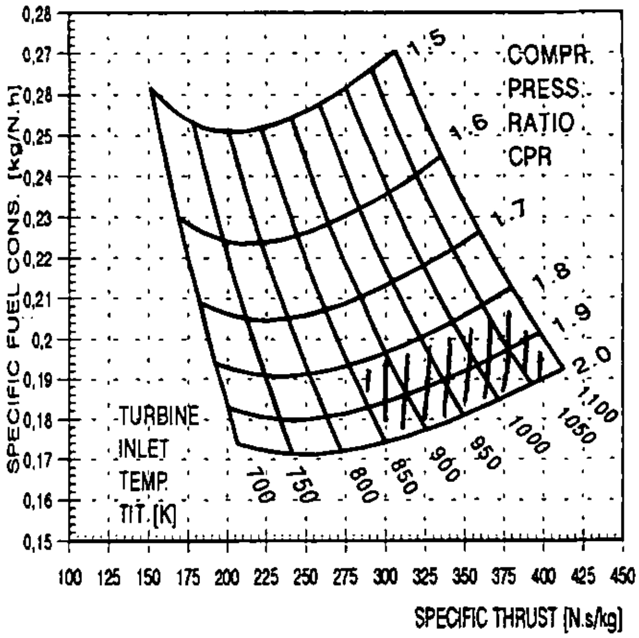

Figure 4: Simple Turbojet Cycle Parameter Study

When component pressure losses are reduced and their efficiencies are increased the entire map in fig. 4 will move towards lower specific fuel consumption and higher specific thrust.

Off-Design Performance

Simulation Program. During the design and testing phase of the engine a second, off-design performance program was developed. To perform off-design simulations the characteristics of all components from inlet to exhaust nozzle are required. The addition of a diffuser and first stage nozzle assembly has greatly influenced the known rotor characteristics of both compressor and turbine and thus their overall characteristics. To obtain these new characteristics a whole set of measurements would have to be undertaken. These measurements are cumbersome and would require a fully instrumented and accurate test rig. At this time no effort has been put into such measurements. The performance simulations were done using best estimates on the shape of these characteristics.

Part Speed Operation. The results of the part speed simulations are shown in figures 5 and 6. These simulations were done at ISO inlet conditions and with zero flight speed.

Figure 5 shows the part speed lines that have been obtained and are typical for these single spool turbojet engines. Line 1 represents the off-design behaviour corresponding to the efficiency lines as shown on the compressor map. Line 2 is the result of constant DP component efficiencies throughout its part speed operating range.

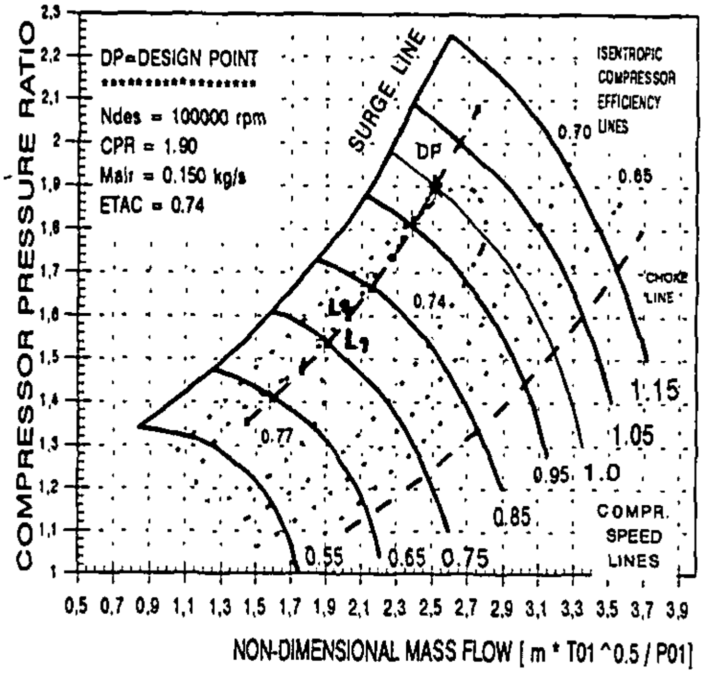

Figure 5: Turbojet Engine Part Speed Operating Lines

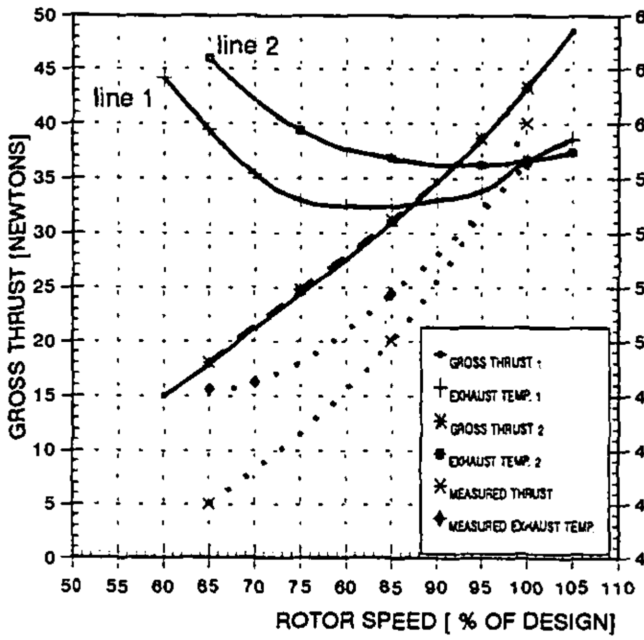

Figure 6: Engine Part Speed Performance Characteristics

The nett effects on engine performance become more clear in fig. 6. Here it can be concluded that component efficiencies have a large effect on exhaust temperature while thrust is more or less uneffected.

When lines of constant T_{03}/T_{01} are plotted on the same compressor map they appear to be very steep and run almost parallel to the part speed operating lines of the engine. This behaviour is very much the result of low component pressure ratios and efficiencies in combination with high ratios of T_{03}/T_{01}. The increase in exhaust temperature at part speed is also seen in the much larger turbojet and turbofan engines but occurs at the very low end of its stable operating speed range. With these small turbojet engines this behaviour can occur very early, depending on its design.

In the early phase of turbojet testing, when fine tuning had not yet taken place, measurements showed similar behaviour to that of line 1 in fig. 6. Exhaust temperature would decrease slightly at first but quickly increase again from 550°C to 650°C. It was only after the many fine tuning sessions of each component of the engine that this behaviour was greatly altered. With the current design exhaust temperatures decrease steadily from 580°C at 100.000 rpm to approximately 480°C at 65.000 rpm.

At the same time the stable operating speed range improved significantly from between 80.000 rpm and 100.000 rpm to between 65.000 rpm and 105.000 rpm. The performance simulations also indicate a much larger stable operating speed range than in reality. This is seen to be caused partly by the unknown shock losses at the turbine rotor entry and possibly the negative incidence stall behaviour of the compressor diffuser at its inlet but mostly due to the unknown component characteristics and their interactions.

Ambient Effects. The typical effects of ambient temperature on thrust, SFC and exhaust temperature can be seen in figure 7. This curve is the result of simulations at constant rotational speed of the rotor (100% of DP) and at an ISO absolute humidity of 0.0064 kg of water vapour per kg of dry air. This graph does not entirely represent real engine operation on a cold or hot day. Turbojet engine operation is normally restricted by two main factors: a) maximum allowable rotor speed and b) maximum allowable TIT.

Figure 7 shows that on a hot day exhaust temperature increases. The limitation on TIT requires that at a certain exhaust temperature fuel flow must be reduced in order not to exceed the TIT limit. The effect is however that the resulting decrease in rotor speed and thus air flow reduces thrust further than initially anticipated.

On a cold day exhaust temperature is reduced. This allows the fuel flow to be increased to bring the exhaust temperature and TIT back to their normal operating levels. The result is an increase in rotational speed and air flow and an additional increase in thrust. Thus, turbojet operation on a cold day is in general restricted by its maximum allowable rotational speed of the rotor while on a hot day it is limited by its maximum allowable Turbine Inlet Temperature.

While the thrust setting of this engine is set by the measurement of exhaust temperature care should be taken during extreme conditions.

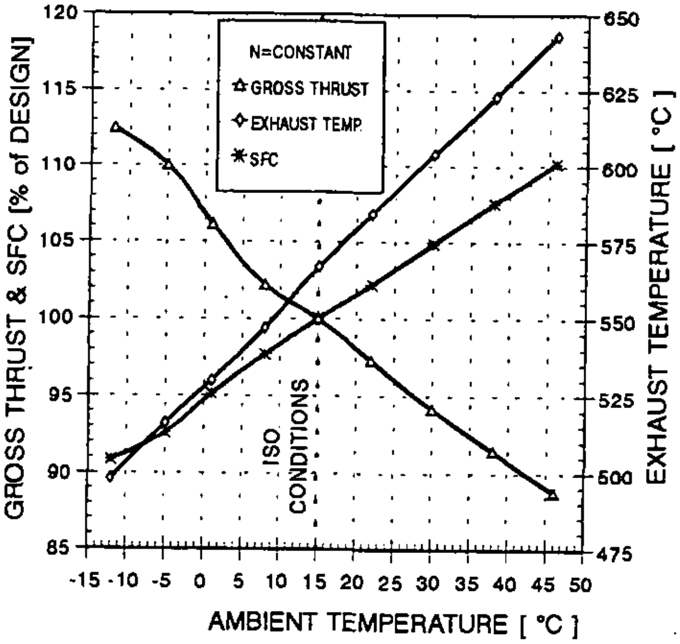

Figure 7: Ambient Temperature Effects on Engine Performance

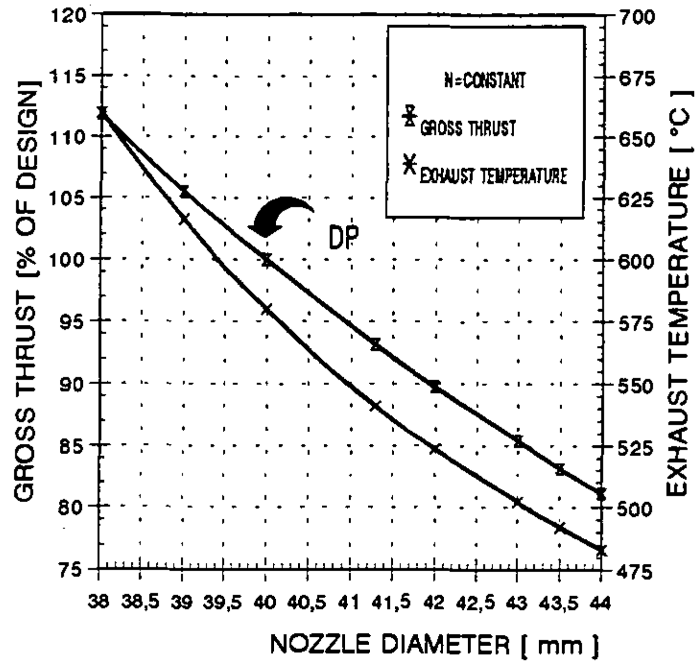

Figure 8: Nozzle Diameter Effects on Engine Performance

Our experience was that exhaust temperature is easier and faster to measure during startup and operation in the field than for example thrust or rotational speed. The negative effect of an increase in absolute humidity on air density, engine air mass flow and hence thrust has not been taken into account. This effect is however small and is at maximum only half of a percent.

The effect of ambient pressure on turbojet performance is very straightforward. At a fixed rotational speed a decrease in ambient pressure does not have an effect on the engines non-dimensional operating point on the compressor characteristics. For a constant value of inlet non-dimensional mass flow (NDM_{01}) a change in P_{01} has a proportional effect on the air flow entering the compressor and hence thrust and fuel flow. In theory, SFC would therefore remain constant with change in ambient pressure. In reality however, a reduction in ambient pressure causes a reduction in air density. The resulting decrease in Reynolds number has a negative effect on component efficiencies and thus specific fuel consumption.

<u>Nozzle Area Variations.</u> Another interesting aspect that has been investigated is the effect of nozzle area (diameter) on engine performance and off-design behaviour. Figure 8 shows the results of these simulations. Nozzle area was varied by ±15% around its design value and at a constant rotational speed (100% of DP).

To maintain constant rotational speed of the rotor the simulations revealed that a decrease in nozzle area requires an increase in fuel flow. The engine operating point moves along the constant speed line towards higher values of T_{03}/T_{01}, thereby increasing Turbine Inlet Temperature and exhaust temperature but at the same time decreasing surge margin. A decrease in nozzle area of 15% resulted in a 15% increase in thrust but also in an increase in both Turbine Inlet Temperature and exhaust temperature of more than 100°C! These simulated effects have actually been confirmed on the testbed and show the large sensitivity of nozzle area on engine performance.

Basically, by changing the nozzle area the off-design speed line can be moved further away or nearer to the surge margin. From the above it can also be concluded that the choice of nozzle area is very much a trade-off between surge margin, rotational speed, turbine inlet temperature and component efficiencies for a specific value of thrust. A large nozzle area places a limit on the rotational speed to obtain a certain thrust level while its surge margin is high. A small nozzle area places a limit on Turbine Inlet Temperature to obtain the same thrust as the forementioned scenario while its surge margin is low.

Since the compressor characteristics are not entirely known it is, again, a trial and error process to obtain the best turbojet engine operating point on the map. A reasonable surge margin as well as high component efficiencies and thus pressure ratio must be targeted for while the best balance between Turbine Inlet Temperature and rotational speed must be found for a specific value of thrust. Many experiments have been done with various nozzle sizes and for our turbojet engine design a nozzle diameter of approximately 40 mm was found to give the best results.

It was discovered that changing nozzle area allowed the fine tuning of the internal aerodynamics. By changing nozzle area a new set of unique values for air flow, operating temperature and pressure and rotational speed can be created, thereby changing the velocity triangles within the rotating component and stator assemblies. It should also be mentioned however that this process does not guarantee success. It greatly depends on the design of the engine and its components. A small change in nozzle area may well improve diffuser performance slightly but at the same time could worsen characteristics of other downstream components such as the combustor or the turbine first stage nozzle assembly.

TURBOJET HANDLING AND OPERATION

The Turbojet Engine Testbed

At this time engine startup, shutdown and normal running have only been done on a specially designed testbed. This testbed contains several measuring devices to monitor engine performance and correct turbojet operation. Figure 9 shows a picture of this testbed. The available measurements include:

1) Rotational speed
2) Exhaust temperature
3) Compressor discharge pressure
4) Thrust (static)
5) Bearing + oil temperature
6) Fuel pressure/temperature

Oil droplet flow to the bearing module can be monitored with a sight glass. Rotational speed is measured with an optical sensor while static thrust is measured with a loadcell. The small lube oil tank is mounted on the compressor casing of the engine. The testbed also contains the miniature gear type liquid fuel pump, the condensor discharge spark ignition unit for the igniter and the synchronised control valves for both gas fuel and kerosene.

Engine Startup and Shutdown

As mentioned earlier, engine startup is accomplished on gas fuel and the startup sequence can be done safely by a single person.

Compressed air from a small, electric motor driven, compressor unit is fed to the startup nozzle located on the compressor casing. First of all the rotor is accelerated with compressed air to 10.000 rpm. At this point ignition is turned on, the gas fuel valve is opened and light-up occurs, accelerating the engine further to its minimum idle speed of appproximately 65000 rpm. The air starter unit is shut down shortly afterwards. The thrust then produced is about 5 Newtons while the exhaust temperature is 480°C. Compressor pressure ratio is then approximately 1.3.

Changeover from gas fuel to kerosene is accomplished using the same fuel manifold system. By simultaneously closing the gas fuel valve and opening the liquid fuel valve (synchronised) changeover is initiated. This means that during changeover the turbojet engine runs on a mixture of gas fuel and kerosene for a few seconds. This method has been very successful. Further acceleration to its maximum continuous speed of 100000 rpm can then be initiated.

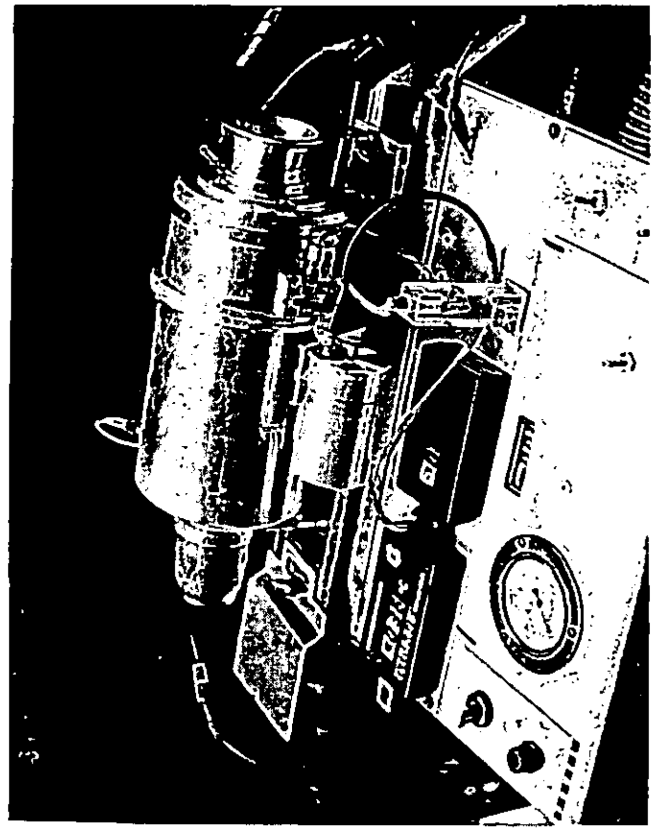

Figure 9: The Turbojet Engine Testbed

Engine shutdown can be accomplished by simply closing the liquid fuel valve after the engine has been decelerated back to its minimum idle speed and maintained at this speed for approx. 1 minute. After rotor runout the temperature within the bearing module increases slightly from 80°C to 87°C due to radiation effects from the turbine and combustor. This is due to the fact that the bearing module is no longer cooled by the relatively cold air passing it during normal engine operation.

POWER GENERATION APPLICATIONS

Although we have been concentrating on the design of a turbojet engine for model aircraft propulsion, its extraordinary thermodynamic characteristics have provided us with other interesting thoughts in the field of very small Simple Cycle Power or Combined Heat & Power generation applications.

The low combustor inlet temperature of approximately 90°C and the very high exhaust temperature of 580°C make it ideal for a heat-exchanger cycle. As demonstrated by Cohen and Rogers, (1987) the positive effect of the heat-exchanger cycle on SFC increases significantly with a decrease in CPR and an increase in TIT. The optimum CPR for the maximum thermal efficiency however increases slightly with increase in Turbine Inlet Temperature.

Studies in this field were carried out and for the same nett power output of approximately 5.5 KW, the simple cycle thermal efficiency of 5% can be increased by at least a factor of 3 through the addition of such a system. Even with the addition of a heat-exchanger the remaining exhaust gases would still be hot enough to make warm water.

The heat-exchanger cycle introduces other practical design and manufacturing problems, but the reverse flow combustor arrangement would easily allow for the addition of such a system. The use of more sophisticated super alloys or even modern glass-ceramic-matrix-composite materials would allow higher combustor operating temperatures and the results could be very promising.

CONCLUSIONS

Todays availability of high strength, temperature resistant materials is very good. The materials used in this turbojet are common everyday materials and they are not too difficult to handle and to machine with the proper equipment at hand. When the in-house capabilities were insufficient the manufacturing of a particular component was contracted out by us. This combination of activities has proved to be very successful.

It has been concluded earlier that the only way to increase specific thrust is to increase the Turbine Inlet Temperature to higher levels. This would require the use of superalloys such as Hastelloy, perhaps in combination with thermal barrier coatings, or even new technology glass ceramic matrix composites.

The use of such materials however requires a new approach with respect to the manufacturing and design of components. The costs involved are an additional barrier and for model aircraft applications it is not worthwhile at this time. For any industrial power generation application it is certainly worth the consideration.

Any discrepancies between theoretical off-design analysis and actual measurements are largely due to the uncertainties of component behaviour with respect to pressure losses, efficiencies and their characteristics and interactions. The sensitivity towards these component pressure losses and efficiencies increases significantly at very low Compressor Pressure Ratios and high Turbine Inlet Temperatures. As a result the off-design performance simulations can be far away from reality and extremely difficult to approach.

Component interaction on this scale is also extreme. Slight changes in geometry of, for example the diffuser, may have a tremendous effect (positive or negative) on the behaviour of other downstream components as well as overall engine behaviour. The good characteristics of an off-the-shelf item of already excellent design can very much be destroyed by incorrectly matched components.

It can also be concluded that the nozzle area may give the designer a tool for the optimisation and fine tuning of components, their behaviour and interactions as well as overall engine performance.

ACKNOWLEDGEMENTS

The authors wish to thank Dr. Periclis Pilidis of the Cranfield University, UK for his support and advise on turbojet engine performance simulations and providing confidence in the results of our work.

Thanks also to Mr B. Conen for his advise on microwelding procedures and techniques and Mr W. Winkens for his advise and help on Computer Aided Design and Computer Aided Manufacturing procedures. Their valuable cooperation has led to the manufacture of some high quality engine components.

Special thanks to Mrs Janet Kirby for her enthusiasm, wonderful moral support, suggestions and advice on the writing of this paper. The result was the additional research and activities in the field of small industrial power generation and Combined Heat and Power applications and heat-exchanger cycles.

REFERENCES

Adkins, R.C., 1983, "A Simple Method for Designing Optimum Annular Diffusers.", ASME Paper No. 83-GT-42.

Adkins, R.C., Wardle, M.H., 1990, "A Method for the Design of Optimum Annular Diffusers of Canted Configuration.", ASME Paper No. 90-GT-52, Presented at the Gas Turbine and Aero Engine Congress and Exposition-June 11-14, 1990, Brussels.

Ballal, D.R., Lefebvre, A.H., 1978, "Ignition of Liquid Fuel Sprays at Sub-atmospheric Pressures", Combustion and Flame, Vol. 31, No. 2, pp 115-126.

Cohen, H., Rogers, G.F.C., Saravanamutto, H.I.H., 1987, "Gas Turbine Theory", 3rd edition, John Wiley and Sons Inc., New York.

Harman, Richard T.C., 1981, "Gas Turbine Engineering - Applications, Cycles and Characteristics", The macmillan Press Ltd., London and Basingstoke, UK.

Humphries, John D., 1992, "Automotive Supercharging and Turbocharging Manual - A Technical Guide", Haynes Publishing Group, UK.

Lefebvre, A.H., 1988, "Gas Turbine Combustion", MaGraw-Hill, New York, USA.

Rao, K.V.L., Lefebvre, A.H., 1976, "Minimum Ignition Energies in Flowing Kerosene-Air Mixtures", Combustion and Flame, Vol. 27, No. 1, pp 1-20.

Siskind, K.S., 1990, "Glass Ceramic Matrix Composites for Advanced Gas Turbines", AIAA Report 90-2014, 26th Joint Propulsion Conference, July 16-18, 1990, Orlando Florida, USA.