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# CYCLE ANALYSIS FOR HELICOPTER GAS TURBINE ENGINES

by

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Procurement Executive, Ministry of Defence
Farnborough, Hants



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## CYCLE ANALYSIS FOR HELICOPTER GAS TURBINE ENGINES

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SUMMARY

The performance potential of a 1000 kW gas turbine engine is determined in terms of specific fuel consumption and specific power. Compressor and turbine efficiencies are assumed size dependent and the cycle temperature is determined from the material capability and cooling technology available. Simple cycle and heat exchanger cycle engines for helicopters are considered.

In the near term, engines with a two-stage gas generator turbine and uncooled power turbine offer an attractive simple cycle solution. However, as cycle temperatures are increased, a cooled power turbine becomes necessary and a lower pressure ratio engine with a single-stage gas generator turbine provides the most cost effective solution.

The heat exchanger cycle is attractive only for those helicopter missions where endurance, or fuel conservation, is the dominating requirement. The benefits of variable power turbine geometry are considered marginal.

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# CYCLE ANALYSIS FOR HELICOPTER GAS TURBINE ENGINES

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## 1 INTRODUCTION

Over the years there have been steady improvements in helicopter engine performance - higher specific powers (power/unit air flow), lower specific fuel consumptions and improved power/weight ratios. The gains have been the result of improving materials and cooling technologies and better aerodynamic design methods. These advances have allowed higher combustor temperatures and component efficiencies, with direct benefit to both specific power and specific fuel consumption (sfc), and higher turbomachinery stage loadings. These higher loadings have mainly been exploited as a means of reducing stage numbers, leading to improved reliability and maintainability and lower acquisition cost - always particularly strong goals in the helicopter field.

In looking forward over the next twenty years, while cost of ownership will clearly continue to occupy a prominent position, performance aspects seem likely to be given renewed attention. This trend is driven by the need for better payload fractions and superior helicopter performance, which means reducing, not just engine weight but also engine plus fuel weight. Moreover high radius of action is becoming increasingly required for many applications, both military and civil, which focuses attention particularly on fuel consumption.

The aim of this paper is to examine the prospects for future engine performance, taking account of technology developments now in progress, and nearing fruition, and also the longer term aspirations for materials, aerodynamics, and cooling technology. For a simple cycle engine, low sfc demands high temperatures, high pressure ratios and high component efficiencies, all of which become increasingly difficult to achieve as the engine becomes smaller.

One possible way of resolving this problem is to use the heat exchanger cycle. Heat exchanger engines for aircraft applications have been considered from time to time in the past but have not appeared sufficiently attractive. However with more demanding mission requirements and a different technology baseline renewed assessment of such cycles is appropriate and the scope of the paper has accordingly been extended to cover the heat exchanger option.

## 2 APPROACH

The present study has concentrated on performance potential for a 1000kW engine, which is in the power bracket typically required for many combat or medium lift helicopters. The major cycle variable used in the analysis is pressure ratio, with the technology level defined by the allowable material temperatures and the cooling effectiveness for the gas generator first stage turbine rotor. This translates into an achievable cycle temperature (*ie* stator outlet temperature, SOT) that depends considerably on the temperature of the cooling air bled from compressor outlet and hence is also pressure ratio dependent. Compressor and turbine efficiency trends, similar to those indicated by Niedzwiecki and Meitner (1987) are assumed.

The analysis is purely thermodynamic, values of specific power and specific fuel consumption being calculated, over a range of pressure ratios. From these data it is possible to deduce the design options to give minimum engine plus fuel weight or minimum engine weight. The minimum engine plus fuel weight is likely to be the choice of the helicopter designer who wishes to design to minimum take-off gross weight (TOGW). Minimum engine weight, particularly when allied to a low component count should provide the minimum first cost option.

Two technology levels have been considered:

*Near Term Technology Engines* - where each component is assumed to have attained the projected performance of current development programmes. Engines at this performance level could be in service by the turn of the century. Component efficiencies are 1-2% higher than for engines currently entering service. Higher values of SOT (stator outlet temperature), relative to today's engines, are obtained mainly by the assumption of improved turbine cooling.

Advanced Technology Engines - the performance that might be achieved due to future improvements in component design techniques, increases in blade speeds and materials developments. Initially 'conventional' metal engines are assumed, but with increased component efficiencies, higher metal temperatures and advanced turbine cooling technology. Subsequently the potential benefits of ceramics are considered, in the form of thermal barrier coatings and as complete rotors.

## 3 ASSUMPTIONS ADOPTED IN THE STUDY

### 3.1 Component Performance

For most applications a certain design power is specified, so technology improvements, such as increased temperature levels, that lead to significant increases in specific power, also lead to smaller engines. These will tend to have inferior component performance due to the effects of Reynolds Number, clearances, surface roughness, manufacturing constraints and the limitations of current design methods. Historically the benefits of the higher temperature to the cycle have outweighed this disadvantage, and each new generation of engines has had lower sfc as well as smaller size. However, at very high cycle temperatures the effects of dissociation will eventually negate that trend even if the designer manages to maintain component efficiencies.

In this study, appropriate trends of efficiency versus "corrected flow rate" for compressors and turbines, based on the data presented by Niedzwiecki and Meitner (1987) have been adopted. The curves derived for the near term and advanced technology compressors are shown graphically in Fig. 1. It can be seen that, for the power range of interest, the advanced technology level engines have a compressor efficiency 2-5% higher than the near term development technology engines. These improvements result from the benefits of advanced computational fluid dynamics, reduced clearances etc. The smaller engines in the range are expected to benefit more from the three-dimensional fully viscous codes being developed, because boundary effects and blade thickness are more critical. Similar trends have been used for the turbine stages.

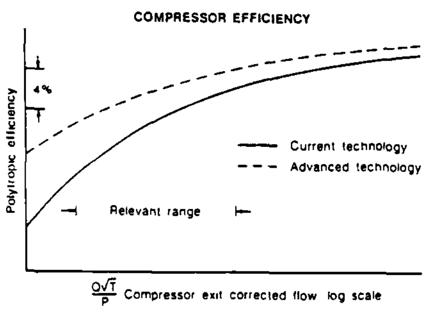


*Fig 1 Compressor Efficiency*

The efficiencies are also dependent upon loading. For the compressor, the cost and maintenance implications of an additional stage are modest. With the gas generator turbine the issue is more critical. Two expensive cooled stages are normally required to achieve peak efficiency in all except the lowest pressure ratio engines. On the other hand, the attractive features of the single stage gas generator turbine, in lower first cost, reduced maintenance cost and a smaller cooling bleed flow requirement, may often outweigh the performance loss due to reduced turbine efficiency. The decrement in efficiency, with pressure ratio, of using a single stage can be seen in Fig 2, where efficiency is plotted against work rate, $\Delta h$. The difference shown between near term and advanced technology levels is commensurate with a 1% increase in efficiency and a 10% increase in blade speed.

Turbine efficiencies are also affected by cooling flows, particularly to the rotor blades. At the near term technology level a reduction of 2% in turbine efficiency is assumed but for the advanced technology level it is projected that better design methods will reduce this decrement to 1%.

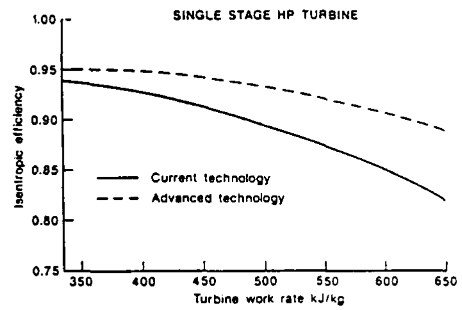


*Fig 2 Single Stage Gas Generator Turbine Efficiency*

### 3.2 Material Temperature, Blade Cooling Effectiveness and Pattern Factor

Thermodynamic analysis of heat engines normally focuses on cycle temperature and pressure ratio as the major variables. For modern engines, where components are cooled, blade metal temperatures represent more practical design parameters than gas temperatures. The highly stressed gas generator rotor blade is often the most critical component and rotor blade temperature has therefore been taken as the representative parameter here. At the near term component technology level, the mean rotor temperature, cooling effectiveness ($\epsilon_R$) and pattern factor (OTDF, outlet temperature distribution factor and RTDF, radial temperature distribution factor) are given in Table 1. Also shown, for the advanced component technology level, are three material options: a conventional metal turbine, with a 50K material temperature increase over the near term technology level, and improved rotor cooling effectiveness; a thermal barrier coating applied to allow the cooled enhanced material a further increase of 100K, in surface temperature; a ceramic turbine with 1800K material capability.

TABLE 1

TURBINE MATERIALS AND COOLING ASSUMPTIONS

| Technology | Rotor | Stator | $\epsilon_R$ | OTDF |
|---|---|---|---|---|
| Near Term | 1200K | 1250K | 0.5 | 0.3 |
| Advanced | 1250K | 1300K | 0.6 | 0.25 |
| Advanced TBC | 1350K | 1400K | 0.6 | 0.25 |
| Ceramic | 1800K | 1800K | - | 0.25 |

RTDF = 0.1 for all cases. Uncooled rotor blades and stator vanes are assumed able to operate at base material temperatures 50K higher, because the stresses due to thermal gradients are much lower. An additional 3% bleed loss to the cycle, is assumed for other component cooling.

In the near term technology engines the design assumptions require a 3% bleed flow rate to satisfy the rotor design effectiveness of 0.5. Coolant flows to the other rotors and stators are adjusted to give the design temperatures of 1200K and 1250K respectively using curves for effectiveness against cooling flow similar to those of Hiroki and Katsumata (1974). This cooling performance, at near term technology levels, has been demonstrated in larger engines but there remains a significant challenge in the manufacture of complex cooling geometries for small blades. At the advanced technology level rotor effectiveness is increased to 0.6 with a 4% bleed flow.

At some stage in advanced component development it will become possible to operate in a gas at the stoichiometric temperature. However, a practical combustor design must take account of cooling to the combustion chamber walls and avoid smoke formation. In small engines this must be achieved in a constrained space envelope. For these reasons it is assumed here that up to 80% of stoichiometric (*ie* F/A = 0.055) might ultimately be achieved.

### 3.3 Losses

All engines have pressure losses, associated with the intake, combustor, inter-ducts and the exhaust duct and nozzle. In a heat exchanger engine there are additional losses through the airside of the heat exchanger, between compressor delivery and the combustor, and the gas side of the heat exchanger, at the outlet from the power turbine. The assumed losses, as a percentage of the available pressure, ($\Delta P/P$), are listed in Table 2:

TABLE 2

LOSSES ($\Delta P/P$%)

| | Simple Cycle | Heat Exchanger Cycle |
|---|---|---|
| Inlet Particle Separator | 1.3 | 1.3 |
| Air Side Heat Exchanger | 0 | 3.0-7.0 |
| Combustor | 3.5 | 3.5 |
| Interduct | 3.0 | 3.0 |
| Gas Side Heat Exchanger | 0 | 5.0-9.0 |
| Exhaust Duct | 1.6 | 1.6 |
| Nozzle | 2.0 | 2.0 |

The losses in the heat exchanger cycle are assumed to vary linearly, over the range of design point effectiveness considered, from 0.5 to 0.9.

No changes in pressure drop have been assumed between near term and advanced technology engines.

## 4 PARAMETRIC ANALYSIS - SIMPLE CYCLE

### 4.1 Near Term Technology

At the 1000kW design power level, three curves define the envelope of best attainable performance as pressure ratio is varied. These can be seen in Fig 3 where sfc is plotted against specific power.

NEAR TERM TECHNOLOGY - SIMPLE CYCLE

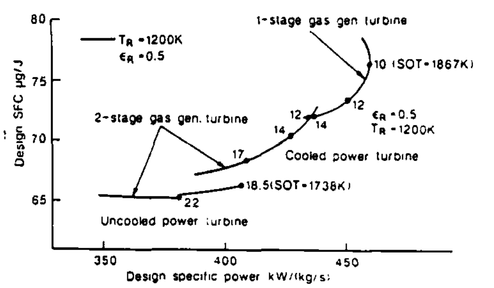


*Fig 3 Design Performance at Near Term Technology Level*

At low pressure ratio an engine with the single stage gas generator turbine offers the highest attainable specific power, at a pressure ratio of 10:1. As pressure ratio is increased the specific fuel consumption is improved, but this improvement is modest because of the effect of decreasing gas generator turbine efficiency. Engines with a 2-stage gas generator turbine give improved thermal efficiency at a given pressure ratio. Above 12:1 pressure ratio the 2-stage gas generator turbine option gives a lower sfc for the attainable specific power level. At pressure ratios above 18.5:1 the gas temperature entering the power turbine rotor has dropped to a sufficiently low level for it to operate uncooled, with a consequent step change in efficiency. Any further increase in pressure ratio produces only a tiny reduction in sfc, the minimum being obtained at 22:1, when the power turbine stator is also uncooled.

Using the material temperature of the rotor as the temperature parameter and exploiting the greater cooling capability presented by the lower pressure ratio options considerably enhances the specific power obtainable at low pressure ratios by allowing a large increase in SOT. The 130K difference in SOT, between the 10:1 and 18.5:1 pressure ratio points, accounts for most of the increase in specific power.

It must be emphasized that the curves in Fig 3 represent the boundaries of best performance for the design assumptions made. If any of the limits proposed cannot be met, in a particular design, the resulting performance will be inferior in terms of specific power and specific fuel consumption. Moreover an engine, at the same level of technology, could be designed to operate at a lower temperature level (and hence potentially better life and reliability) and achieve similar specific fuel consumption.

4.2 The Conflicting Constraints of TOGW and Costs

For helicopters, TOGW is often a critical parameter and in such cases the engine that provides minimum engine plus fuel weight is likely to be favoured. Fuel burn will depend on the mission, but typically over half the fuel consumption occurs below half power, as is indicated by Hirschkron and Russo (1986). The fuel consumption at half power can, therefore, be taken as a reasonable average for a mission, say of typical duration 2.5 hours. Engine weight can also be generalised, to a first approximation, as proportional to core flow rate, neglecting any weight savings that will occur with reducing stage numbers. With these simple assumptions lines of constant engine plus fuel weight can be drawn on the sfc versus design specific power plot in Fig 4. The minimum engine plus fuel weight is obtained by the engine with a pressure ratio of 18.5, which is the lowest pressure ratio at which the more efficient, uncooled power turbine rotor can be used.

The other important consideration is cost. For most applications minimum cost of ownership of the whole helicopter would be the preferred optimizing parameter. Changes in engine plus fuel weight for a given requirement will lead to some consequent variation in vehicle weight and cost. If these are considered small, vehicle cost variations between one engine choice and another will, to a first approximation, be equal to the changes in engine and fuel costs. Fuel costs are easily deduced, with fuel at $0.1/lb and assuming 400 flight hours per year at half power. Engine first cost can be taken, to a first approximation, as proportional to the core flow, and to be about $100000 (kg/s). (In terms of engine weight this equates to about $2200/kg). Annual engine costs, including maintenance, depreciation, investment, spares holding etc are assumed to be 20% of first cost. On this basis annual engine costs are about four times the fuel costs.

COST AND WEIGHT IMPLICATIONS

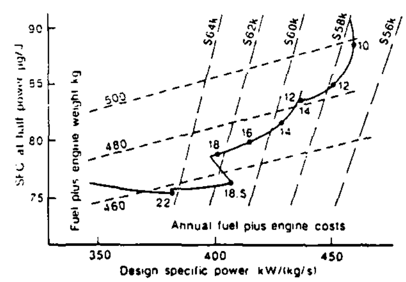


*Fig 4 Trends in weight and costs with design*

With these assumptions lines of constant annual cost can be drawn, as shown in Fig 4. These indicate that minimum fuel plus engine cost would occur at close to maximum specific power. However, any increase in fuel price or higher utilisation would reduce the slope of the cost lines, having the effect of moving the minimum cost design towards a higher pressure ratio. Eventually, when annual fuel costs equate approximately to engine costs, the two stage gas generator turbine, with uncooled power turbine rotor would become the minimum cost option as well as offering the lowest weight penalty. Irrespective of fuel price the results imply that there is little merit on cost or weight grounds in selecting the 2-stage gas generator turbine/cooled power turbine rotor option (ie cycles on the intermediate curve in Figs 3 and 4). Indeed, if the step changes in cost, associated with the deletion of a cooled gas generator stage or the simplification of an uncooled power turbine rotor, were allowed for, the argument would be even stronger.

If allowance were to be made for the consequences of increasing engine plus fuel weight as pressure ratio is reduced (which will force a small growth in airframe weight and in absolute engine size and power needed to meet a given payload-range requirement), the slopes of the constant cost lines would again tend to reduce slightly. This consideration reinforces the general indications emerging from Fig 4:

> The 2-stage gas generator turbine with uncooled power turbine rotor offers the lowest weight solution, with the lowest pressure ratio at which this can be achieved (18.5:1 in the present analysis) appearing to be optimal. There is little merit, on either weight or cost grounds, in selecting cycles demanding 2 gas generator turbine stages and a cooled power turbine.
>
> The single stage gas generator turbine option may offer a competitive engine, in terms of cost of ownership. In this case the expected optimum pressure ratio would lie between 10.5:1, where fuel plus engine costs are a minimum, and 12:1. Beyond this the 2-stage gas generator turbine, at P/P = 18.5, with uncooled power turbine rotors would be expected to become competitive.

The effect of size will not change these general conclusions, except in detail, smaller engines being more biased to the single stage gas generator turbine solution.

4.3 Advanced Technology (Metal Engines)

Increasing the technology level produces substantial improvements of about 30% in specific power and 10% in specific fuel consumption, for the component performance and materials capability assumed. (See Fig 5).

NEAR TERM TO ADVANCED TECHNOLOGY

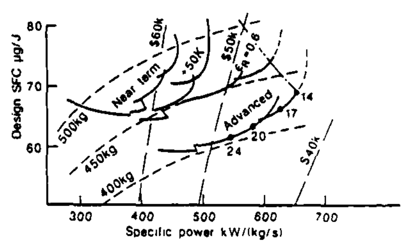


*Fig 5 Influence of Improvements in Technology*

To isolate the major causes of the improvement, the progression from near term to the advanced technology standards is considered in three steps:

The first step shows the benefits of increased rotor temperature (+50K) and a 20% reduction in cooling flow rates with effectiveness maintained at $\epsilon_a = 0.5$ .

In the second step the cooling effectiveness of the rotor blade is increased to $\epsilon_a = 0.6$ .

The final step shows the contribution due to improved component efficiencies.

For the intermediate steps there is little improvement in sfc, any benefits from increases in cycle temperature being eroded by the reducing component efficiencies with decreasing engine size. In the first step, where the rotor temperature is increased by 50K, maximum specific power is increased by 11%. Improving the effectiveness to 0.6 takes the increase to 33%, where the F/A = 0.055 limit is reached. The final step, with improved component efficiencies, provides about an 8% reduction in sfc and a commensurate increase in specific power.

The impact of these advances on the choice of design is clear. As higher temperatures become acceptable the fuel plus engine weight disadvantage of the single stage gas generator turbine is reduced and the potential cost saving markedly increased. Unless the fuel plus engine weight is of paramount importance the single stage gas generator turbine option must be the preferred solution.

It is particularly noticeable that, for the two stage gas generator turbine with uncooled power turbine rotor, advanced technology does not offer a significant increase in specific power. This is because defining power turbine entry temperature is equivalent to fixing the fuel to air ratio.

The cooled power turbine is therefore essential at this advanced technology level and there are then only two regions of design interest. Single stage gas generator turbine engines with a pressure ratio between 14:1 and 17:1 and 2-stage gas generator turbine engines with a pressure ratio between 20:1 and 24:1.

4.4 Advanced Performance with Thermal Barrier Coatings and Ceramics

The performance of the advanced technology engine can only be further improved, at the F/A limit, by increases in pressure ratio, which would result in yet higher cycle temperatures. One way of achieving this might be by exploiting thermal barrier coatings (TBC). Fig 6 shows the performance boundary, with surface temperatures of cooled rotors and stators 100K higher than for the baseline uncoated materials. As would be expected the major difference is the increase in specific power.

At these temperature levels a cooled power turbine rotor is essential and the region of design interest is for a pressure ratio between 20:1 and 28:1. There is no apparent advantage from the single stage gas generator turbine option since the lower pressure ratio range where this would be possible lies almost entirely beyond the F/A limit.

While thermal barrier coatings, in principle, offer an attractive route to improved performance, their successful application to turbine blade aerofoils has yet to be achieved, even in large engines. A major problem is obtaining a satisfactory bond to the base material and providing a smooth surface finish. For small engines, in particular, most research effort is directed more towards making the whole blade out of ceramic material.

Ceramics are particularly attractive provided that simple and relatively cheap manufacturing processes can be developed and combined with adequate component integrity. The specific power of the uncooled ceramic engine is limited, by the 1800K material temperature, to the levels of specific power achieved by near term technology engines. However the sfc levels are very good and because there is no need to compromise the aerofoil design to allow for cooling, particularly high turbine efficiencies should be attainable. If the stator is cooled (and there would be no performance penalty associated with using metal vanes), the uncooled gas generator turbine rotor temperature becomes the constraint. The increased cycle temperature then gives some further improvement in sfc, combined with higher specific power. This level of thermal efficiency is about the best possible with a simple cycle at these component efficiency levels.

It is of interest to note that with cooled ceramic blading, the maximum specific power would occur, for the fuel to air ratio limit of 0.055, at the relatively modest pressure ratio of 34. It might be expected that, once the fuel to air limit was reached, increasing the pressure ratio would result in ever reducing sfc, as would happen with a perfect gas. However, the reducing component efficiencies, due to decreasing component size, together with dissociation, at these high temperatures, combine to give this limit. It is doubtful whether this all cooled ceramic engine can be considered a real design alternative but it provides a benchmark for ultimate performance.

THE SIMPLE CYCLE RANGE

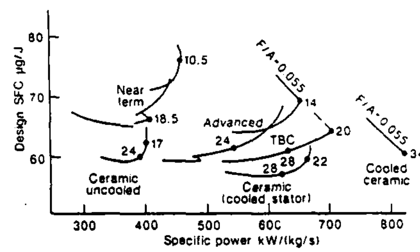


Fig 6 The Simple Cycle Range

5 PARAMETRIC ANALYSIS - HEAT EXCHANGER CYCLE

5.1 Assumptions

The component performance assumptions, for the heat exchanger cycle, are the same as those adopted for the simple cycle. A single stage gas generator turbine is used for all cases, with cooling flow provided from the compressor discharge. There are additional pressure losses across the airside and gas side of the heat exchanger as indicated in Section 3.2. Part load performance is estimated, with design component efficiencies maintained, for both fixed and variable geometry power turbine stators.

Heat exchanger design effectiveness in the range 0.5 to 0.9 are considered with off-design effectiveness estimated assuming typical 2 pass crossflow heat exchanger performance. Fig 7 shows the off-design performance for a fixed geometry machine and for a machine with variable geometry power turbine, controlled to maintain constant heat exchanger entry temperature. The latter approach has been used in a number of applications where part load performance is important. It can be seen that a significant gain in sfc is attainable, at least in principle (Fig 7 assumes no reduction in turbine efficiency at part load, although, as is discussed later, evidence indicates this may be impossible).

The heat exchanger cycle engine must offer reduced TOGW or lower cost of ownership to attract interest from the helicopter designer. Grieb and Klussmann (1981) estimate that the weight of a heat exchanger, of effectiveness 0.6, is 53 kg for a 900kW engine. (At this effectiveness the weight penalty is divided equally between matrix weight and a constant weight penalty due to ducting and headers.) Using this as a datum the weight of heat exchangers of different effectiveness can be calculated assuming matrix weight proportional to surface area. This weight/area relationship has been used to derive a reference "characteristic matrix weight". The effect on design of higher and lower characteristic weights has also been considered.

OFF DESIGN PERFORMANCE

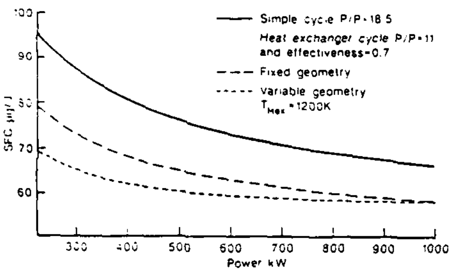


*Fig 7 Off Design Performance*

5.2 Near Term Technology and the Influence of Heat Exchanger Weight

The sfc at half power over a range of design pressure ratios is shown in Fig 8 for the heat exchanger cycle at design effectiveness levels of 0.5, 0.7 and 0.9 with fixed and variable power turbine stators. The pressure ratio range of interest is very small. With fixed or variable geometry, minimum engine plus fuel weight and, minimum fuel plus engine cost, will occur, in a narrow band of specific power, at pressure ratios between 11:1 and 9:1 in all cases. This indicates that for optimal performance the basic engine size is nearly constant irrespective of heat exchanger size. The powerplant plus fuel weight trends can therefore be simplified to a consideration of the heat exchanger plus fuel weight, both of which are related to heat exchanger size. The minimum weight of heat exchanger plus fuel is then a function of the endurance required and the characteristic weight of the matrix.

HEAT EXCHANGER ENGINE PERFORMANCE

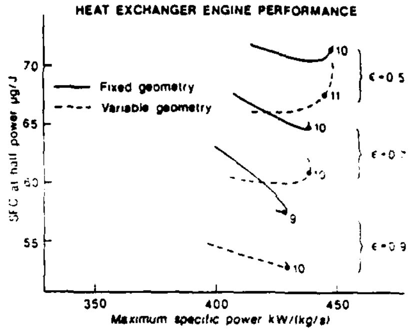


*Fig 8 Specific Fuel Consumption at Half Power*

Fig 9 shows the maximum possible improvement in endurance offered by the heat exchanger engine, compared to the optimal simple cycle engine of the same powerplant plus fuel weight. The endurance is based on average operation at half power, the optimum heat exchanger size increasing as endurance increases. For engines with a fixed turbine geometry and heat exchangers of the reference characteristic weight, there is no endurance advantage over the simple cycle below about two and a half hours. However there is a significant gain as baseline endurance increases, with a 10% extra being available at 4 hours and 20% at 9 hours. With the variable geometry turbine option, these gains are increased by an additional 10%, with the break even endurance at just over one and a half hours. Fig 9 also shows that the gains in endurance are very sensitive to heat exchanger weight, doubling the characteristic weight will double the baseline endurance at which a given increase is obtained.

ADDITIONAL HEAT EXCHANGER CYCLE ENDURANCE

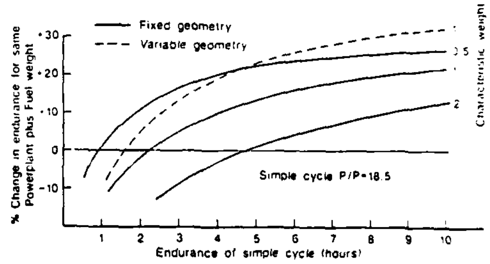


*Fig 9 Increase in Endurance of Heat Exchanger Cycle*

The cost and volume implications of large heat exchangers will bias the design to lower values of effectiveness than that giving minimum powerplant plus fuel weight. Fig. 10 shows the trend for a 5 hour endurance plotted with heat exchanger weight the variable. The effectiveness values used in the calculation of Fig 9 are the minima of each curve but it must be expected that, unless the cost implications of fuel plus powerplant weight are paramount, reducing the weight of heat exchanger (and hence its associated cost) will offer a lower overall cost solution. The cost issues are complex and involve many factors which cannot be addressed here, except in a superficial way. However, it is apparent that as the size of the heat exchanger is reduced, from the value of effectiveness giving minimum weight, there will be reached a trade-off point at which 'costs' are minimised. To be competitive with the best simple cycle engine over this endurance, this minimum must occur when the heat exchanger, whatever the characteristic weight, has an effectiveness greater than about 0.7. (This 'break-even' effectiveness is a consequence of the heat exchanger weight assumptions. If the matrix weight is a higher proportion of heat exchanger weight then the break-even effectiveness would be reduced.) The optimal effectiveness will depend on the cost sensitivities and will increase as characteristic weight is reduced.

EFFECT OF HEAT EXCHANGER SIZE AND WEIGHT

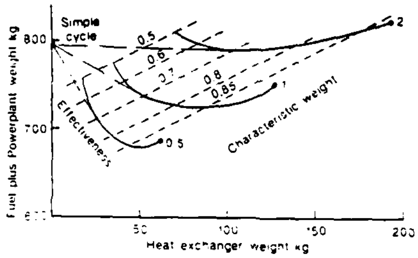


*Fig 10 The Influence of Heat Exchanger Effectiveness and Weight*

5.3 Advanced Technology and the Effect of Thermal Barrier Coatings

The maximum specific power attained by advanced technology heat exchanger engines is comparable with the simple cycle engine at the same technology level. Fig 11 shows the performance projected, for an effectiveness of 0.7, at the advanced technology level compared to the near term technology level. Also shown is the range of performance available from the simple cycle engines. If the comparisons are made with the minimum fuel plus powerplant weight simple cycle the heat exchanger solution is offering 11% lower sfc and 17% higher specific power.

COMPARISON BETWEEN SIMPLE CYCLE AND EXCHANGER

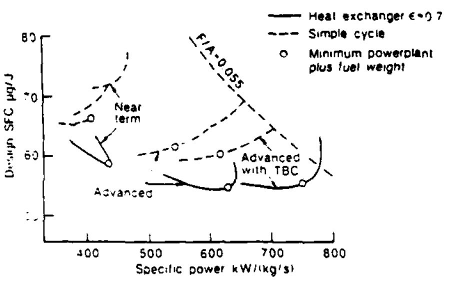


*Fig 11 Comparison between Simple Cycle and Heat Exchanger Engines*

The addition of thermal barrier coatings allows operation at higher specific power than can be achieved by the simple cycle. Again comparing with the minimum fuel plus powerplant weight simple cycle option there is an 8% reduction in sfc and a 23% increase in specific power. At this technology level the heat exchanger cycle powerplant could arguably compete with the simple cycle engine on cost grounds.

5.4 Additional Design Considerations

The results presented here suggest that, for all applications requiring long endurance the heat exchanger cycle is worthy of consideration. However, attention must be given to a number of potential problem areas.

The characteristic weight of the matrix is obviously critical, while the packaging of the heat exchanger will pose serious problems. The total volume of a heat exchanger powerplant will probably be more than double that of a simple cycle engine. The consequent effect on airframe size and weight has not been taken into account here but could be significant.

For military applications the increased size of the heat exchanger powerplant must increase vulnerability. Conversely exhaust temperatures are lower and hence less complex mixing will be required to achieve a satisfactory infrared signature.

Material selection for the heat exchanger matrix becomes very difficult as temperatures rise. For the near term technology defined matrix entry temperatures in excess of 1200K are required but material treatments are being developed which enable certain austenitic stainless steels to be used in excess of this. However, there is little likelihood of finding a metal capable of running in excess of the 1400K required by the advanced technology engines. In order to exploit the full potential of the heat exchanger configuration higher temperature materials must be available and, it would seem, that there is little alternative to a ceramic matrix. Past experience with ceramics in this application has not been good and, although there have been great advances in ceramic technology, the complex mechanical structure and thermal gradients of a heat exchanger present a requirement that is, perhaps, even more demanding than for turbine blading.

If a variable geometry turbine is used several factors are likely to penalise performance. Firstly, according to Munzberg and Kurzke (1976), design point efficiency will be about 2% lower due to the radial gap in the stators. Secondly, as the stators are adjusted to give a flow reduction, there will be a drop in efficiency as indicated by Rahnke (1969) and Latimer (1976). Allowing for these effects would erode about half of the theoretical advantage of the variable geometry machine. Also variable turbine geometry would increase engine first costs and maintenance costs and adversely affect reliability. Unless these penalties can be limited, the fixed geometry cycle may be the better overall choice.

Even the fixed geometry heat exchanger cycle raises questions about maintenance and reliability. Thermal cycling, particularly of lightweight matrix structures will in time lead to fatigue failure and the heat exchanger will be a life limited component. On the other hand, in comparison with a competing simple cycle engine optimised for a similar duty, the base engine has fewer compressor and turbine stages, and is smaller and cheaper. The cost and reliability issues are complex and require a detailed study that lies beyond the scope of the present paper. However, the increase in endurance and range offered by the heat exchanger cycle indicates that the performance benefits can be significant.

## 6 CONCLUSIONS

In the near term the simple cycle offers the best options for typical helicopter missions of about 2.5 hours. Moreover, there appears to be little advantage in designing at high pressure ratios, unless minimum sfc is the overriding consideration. However, with advancing technology, the step to cooled power turbine rotors must be made if the benefits of improved material capability and superior blade cooling technology are to be exploited. The single stage gas generator turbine option offers a good cost to weight compromise at near term technology levels and must be the favoured contender at the advanced technology level for short duration missions.

Eventually, as the higher temperature capabilities of ceramics are exploited, high pressure ratio engines with two stage gas generator turbines will be necessary to obtain good sfc and, for the ultimate specific power, all stages must also be cooled. There appears to be only a marginal advantage, in performance terms, for the ceramic engine, unless the material properties of weight and cheapness can be exploited. A compromise solution of cooled stators of conventional design and uncooled ceramic rotors may be worth serious investigation once the technology is proven.

The heat exchanger cycle can be an attractive option for long endurance missions. Although the variable geometry machine offers the best performance, this is dependent upon maintaining high component efficiencies at off design. The fixed geometry machine therefore offers a more secure development path with intrinsically higher reliability. However, the heat exchanger poses many practical problems - size and packaging, vulnerability in military applications, life and reliability issues etc. These will have to be resolved before this form of powerplant can become a serious candidate for project applications. The potential long term rewards are, however, very attractive.

## 7 ACKNOWLEDGMENTS

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17. Abstract
The performance potential of a 1000 kW gas turbine engine is determined in terms of specific fuel consumption and specific power. Compressor and turbine efficiencies are assumed size dependent and the cycle temperature is determined from the material capability and cooling technology available. Simple cycle and heat exchanger cycle engines for helicopters are considered.

In the near term, engines with a two-stage gas generator turbine and uncooled power turbine offer an attractive simple cycle solution. However, as cycle temperatures are increased, a cooled power turbine becomes necessary and a lower pressure ratio engine with a single-stage gas generator turbine provides the most cost effective solution.

The heat exchanger cycle is attractive only for those helicopter missions where endurance, or fuel conservation, is the dominating requirement. The benefits of variable power turbine geometry are considered marginal.

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