Some Applications of Gas Turbine to Helicopter Propulsion*

By JOHN BROWN, M A **

A Paper presented to The Helicopter Association of Great Britain in the Library of the Royal Aeronautical Society, on Friday, October 15th, 1954

DR G S HISLOP (*Chairman of the Executive Council*) occupying the Chair

The CHAIRMAN, in introducing the lecturer, said that they were fortunate in having an extremely topical lecture The subject of gas turbines and their application to helicopters was currently taking a great deal of attention from designers and industry, and the Association were happy to have it in the programme so early in the session

Mr BROWN had been educated at the City Boys' Secondary Grammar School, Leicester, and was a State Scholar and a College Exhibitioner at Jesus College, Cambridge He had taken an Honours degree in the Mechanical Sciences Tripos, and in 1944 joined Power Jets (Research and Development) Ltd , remaining when it became the National Gas Turbine Establishment He had been engaged as a research and development engineer on combustion processes and was now employed as a Senior Scientific Officer at the N G T E , being engaged on the examination of gas turbine projects and their thermodynamic performance

MR JOHN BROWN

SUMMARY

A detailed comparison is presented between three types of gas turbine power plant suitable for helicopter propulsion The study is limited to aircraft in which the rotor system provides the total lift and forward thrust The types of gas turbine engine-rotor systems examined include a conventional shaft drive scheme, and others in which the power plant provides pressure air or gas feeding hollow rotor blades and discharging as jets at

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the blade tips for driving the rotor , performance estimates are made of the schemes A brief consideration is given to the possibility of mounting gas turbine engines at the tips of the rotor blades An indication of the probable engine weights to power a chosen size of helicopter is given for each type of propulsion

As civil aviation develops, a need is arising for aircraft operation on so-called "feeder" routes, that is, journeys of short length of 200 to 300 miles, each aircraft carrying about 50 people Again, main airports being generally situated away from the centres of large cities, a case can be put for an aircraft capable of carrying passengers between the two In time of war, it might be an advantage to be able to move platoons of soldiers, with their equipment, rapidly from place to place where there are no landing strips These requirements, and many more, can be successfully met by the employment of helicopters, large or small, and for some time now, gas turbine engines have been considered for powering such aircraft The purpose of this paper is to present a comparison between various gas turbine engine-helicopter rotor systems that could be used

Only a limited field is covered dealing with the application of gas turbine engines to helicopters in which the rotor is used to provide the total propulsive thrust in addition to the lift Thus the work does not cover the wider application of gas turbine engines to aircraft in which lift and propulsive power is obtained from a combination of fixed and rotating wing surfaces and orthodox propellers

Advantages and disadvantages of gas turbines compared with piston engines are obvious from experience and have been listed from time to time in a wide variety of publications, but I feel that one or two of the more important ones can bear repetition Among advantages can be included a lower weight to power ratio, little vibration, low frontal area, and the ability to use fuels other than petrol Against these must be set a higher fuel consumption, which could be vital if the fuel tank capacity had to be greatly increased, and possible engine installation difficulties due to their length The size of gearbox might also be troublesome for a direct drive scheme Thus, from a general superficial examination, one can see sound reasons for examining a case for schemes employing gas turbine engines

The main investigation is limited to an analysis of three engine-rotor systems, which will be described as the shaft drive, the ducted gas, and the ducted air schemes, and brief mention will be made of engines mounted at the rotor blade tips For every scheme considered, performance estimates will be presented for three rotor tip Mach numbers of 0 4, 0 5 and 0 6, for hovering and for cruising

Description of Schemes

All the engine-rotor schemes have a common basic unit, the schemes being derived from variations on, and additions to, this basic engine, which is shown diagrammatically in Fig 1 Air is taken into the axial flow compressor at station 1 , between stations 1 and 2 it is compressed, and is then delivered to an annular type of combustion chamber where the requisite quantity of fuel, which is kerosine in these schemes, is injected to raise the

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air temperature to its chosen maximum value at station 3 The hot pressurised gas expands to station 4 through an axial flow turbine which drives the compressor by way of the shaft shown The turbine takes from the gases only sufficient work to drive its own compressor, so that there is at station 4 further energy remaining which the various schemes to be described utilise in different ways

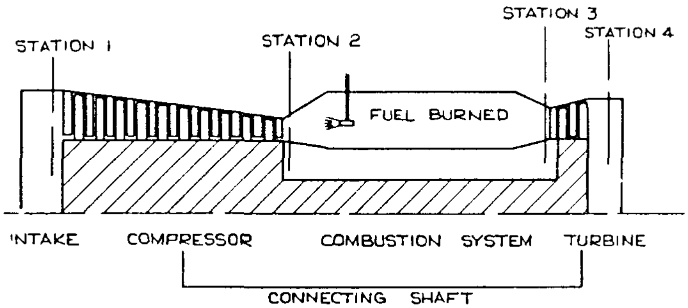
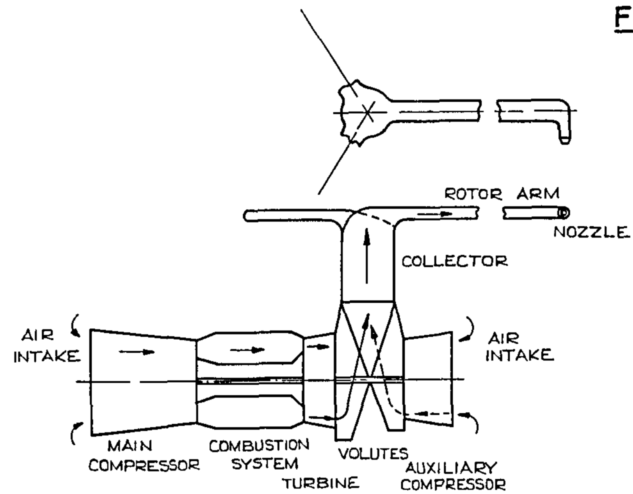


Fig 1 Diagram of Basic Engine

The shaft drive scheme simply entails the addition to the basic unit of a free power turbine which is placed in series with, and downstream of, the turbine driving the compressor By " free " power turbine is meant, of course, a turbine whose output shaft is not mechanically linked in any way to the basic turbo-compressor unit , referring again to Fig 1, the gas at station 4 is now expanded through this second turbine to a pressure which is only slightly above ambient static pressure, the work taken from the gases being used to drive the helicopter rotor through gearing

An alternative scheme would be to couple the power turbine mechanically to that driving the compressor, such that the combined turbine drives both compressor and rotor Such a scheme, however, would give no advantage as far as engine performance is concerned, especially when the power speed characteristics of helicopter rotor operation are considered, for the coupled turbine scheme is known to give low powers at low speeds Moreover, it would require a clutch between the engine and the rotor to allow for starting Almost all arguments, therefore, lead to the conclusion that the free power turbine is by far the more suitable for the shaft drive application

The second scheme to be described has been termed the ducted gas scheme This returns to the principle of pure jet propulsion used, however, in the rotor itself The scheme is shown in block-diagram form in Fig 2 The basic engine components are shown, and it will be seen that the turbine of the basic unit, in addition to driving its own compressor, is extended to provide power for an auxiliary compressor So, returning to the basic concepts shown in Fig 1, it can be seen that the energy remaining in the exhaust gases at station 4 is partly used by an additional turbine stage which supplies power to drive the auxiliary compressor The exhaust gas from the turbine mixes with the air from the auxiliary compressor at a common static pressure, so that the mass of gas flowing into the collector box is increased, but its temperature reduced The gas is ducted along the rotor



DUCTED GAS SCHEME

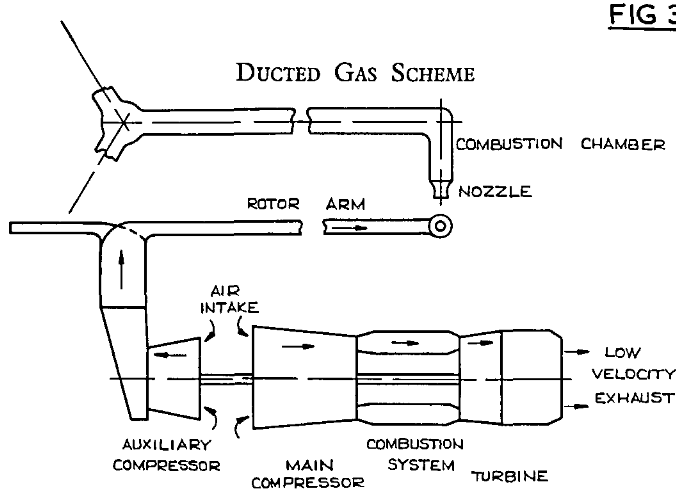
blades, finally being turned to the tangential direction where it expands to ambient pressure through a convergent nozzle, providing thrust for driving the rotor

The third scheme mentioned is the ducted air one This, again, is a jet propulsion scheme applied to the helicopter blade tips, and a block diagram is shown in Fig 3 As before, the turbine of the basic unit is extended to drive an auxiliary compressor, but in this case the turbine exhausts to atmosphere This scheme is, therefore, seen to be analogous to the shaft drive scheme with a coupled power turbine, for the energy remaining in the exhaust gas at station 4, Fig 1, has again been converted, as completely as practicable, into work by a turbine , but instead of driving the rotor directly through gears, the work from this turbine is used to drive an auxiliary compressor, the air from which is led to the rotor head and thence distributed to the blades, along which it is ducted A right-angled bend turns the air into the tangential direction, where it expands through a suitable nozzle to drive the rotor

An extension of the scheme described above, in order to increase the rotor driving power using a given basic engine, consists of adding a low pressure loss combustion system at the rotor tip, the hot gases then being expanded through a suitable nozzle This is also shown in Fig 3 Separate control of the fuel to the main and tip combustion systems has, of course, to be made

It was remarked earlier that brief mention would be made of another scheme , this is the simple one in which a gas turbine engine is mounted at the tip of each blade The engine is, of course, the basic one, but in this case the gases at station 4 of Fig 1 are expanded down to sonic ve ocity, that is, the final nozzle is choking, so that we have changed the energy

FIG 3



Ducted Air Scheme with Burning at Rotor Tips

remaining in the gas at station 4 into velocity energy, giving the normal jet propulsion thrust to drive the rotor There are three possible installation positions , firstly, across the blades, that is, with the whole engine in a line tangential to the circle which the rotor tip describes , secondly radially along the blade, with the intake at an inner radius and the exhaust jet at the tip , or lastly, along the blade with the intake at the tip and the jet at an inner radius All three installations have the advantage that they neither use gears, nor require ducting along the blades, but the major disadvantages appear to be mechanical ones for the engine designer , for either there are large thrust loads on bearings for an engine mounted radially along a blade, or there are large gyroscopic forces on the compressor or turbine discs for an engine across the blade, each effect consequent on the high centrifugal force acting on the engine

Performance Estimates of Engine-rotor Schemes

The design point for all the schemes is taken at maximum power, that is, maximum main engine r p m , at sea level, in order to meet the climb and hovering duty The performance of all the systems benefits from the choice of a high pressure ratio for the basic unit, and a design value of 7 is selected as appropriate to the single-shaft turbo-compressor already described earlier A design maximum cycle temperature, or turbine inlet temperature, of 1050°K is chosen , this is, of course, 777°C or 1430°F This value is regarded as a safe limit for long life for turbine blades which are not internally cooled by air bled from the compressor, but calculations have been made to find the effect of increasing this maximum cycle temperature, and it is found that the shaft drive and ducted gas schemes benefit more than the ducted air scheme with tip combustion chambers Engine-rotor performance

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results are presented for both hover and cruise conditions, it must be pointed out that hover and cruise rotor tip speeds have been kept the same for a given case, for instance, a ducted gas scheme may have been designed to hover at a rotor tip Mach No of 0 4, cruise calculations on this particular design will then have been made still at a tip Mach No of 0 4 The maximum cruise conditions in the engine have been arbitrarily fixed at 1020°K (747°C or 1345°F) and 97% of the maximum engine r p m, and any scheme which, from calculations of helicopter performance, outsteps these imposed limits, would be regarded as having a disadvantage in any final comparison of schemes The maximum cruise limits described are merely typical values which would lower stresses at the given temperature to value acceptable for continuous cruise operation

Typical efficiencies have been chosen for each of the components The intake has been assumed to have a 95% pressure recovery that is, the pressure at inlet to the compressor is 0 95 times ambient pressure The compressor is taken to have a polytropic, or stage efficiency of 86%, fuel is burnt in the combustion system at 98% efficiency The hot gas expands through the turbine with an adiabatic efficiency of 88%, and suitable pressure losses are taken in the ducting, turbine exhaust system, and combustion chambers When estimating cruise performance, compressor characteristics obtained from tests are used, so that the variations in pressure ratio, mass flow and efficiency are realistic, but turbine and combustion efficiencies are assumed constant during cruise, and equal to their design values

The results of the performance calculations for the shaft-drive scheme are shown in Fig 4 Here, the main engine r p m relative to its design maximum value, the maximum cycle temperature, and the specific fuel consumption are plotted against power turbine output expressed as a percentage of the design maximum power used at hovering Under design conditions, the output is 67 h p for each lb per second of air mass flow passing through the engine, and the specific fuel consumption is 0 728 lb /h p -hr The two turbines, that is, the one driving the engine compressor and the other driving the helicopter rotor, are assumed each to be of two-stage design, and appropriate single-line characteristics are used in deriving the curves of Fig 4 From these, the engine conditions and specific fuel consumption can be obtained for any cruising power

The ducted gas scheme will next be considered At design conditions, with the pressure ratio and the maximum cycle temperature of the main engine fixed, the chief independent variable in this scheme is the pressure ratio of the auxiliary compressor It seems clear that there is an optimum pressure ratio, for, since the turbine exhausts to the back pressure given by the auxiliary compressor, then if this back pressure is too high the work available for use in the auxiliary compressor is small On the other hand, if the compressor pressure ratio is too small, the expansion ratio across the tip nozzle is low, resulting in low exit velocity and low power For the given engine conditions, and depending on nozzle forward speed, the curves of Fig 5 show that there is an optimum value of the auxiliary compressor pressure ratio Here, for our chosen design conditions, specific rotor power, which is defined as rotor power per unit air mass passing through the main compressor, is plotted against auxiliary compressor pressure ratio for three values of rotor blade tip Mach number With the design pressure ratio

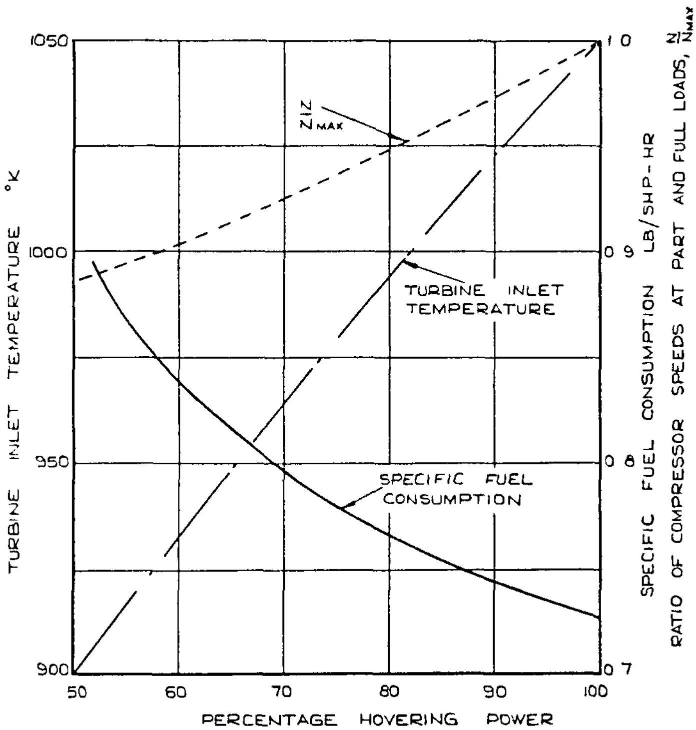
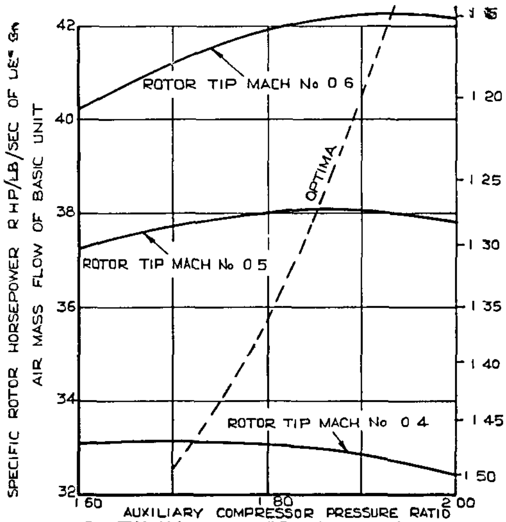


Fig 4 Part-Load Performance of Shaft Drive Scheme

of the main compressor and the design maximum cycle temperature fixed, the specific fuel consumption, lb /r h p /hr, is proportional to the inverse of the specific rotor power, and its scale is shown on the right of Fig 5 In deriving the curves of this figure, due allowance is made for frictional flow losses in the ducting and the centrifugal compression imparted to the air passing up the rotor blades A further independent variable is the Mach number of flow of the gases along the rotor blades, the gases being assumed to flow along tubes within each rotor blade The smaller these tubes are, the lower is the drag of the helicopter blade, but the higher the internal flow friction and hence the lower the power developed Calculations—and here I was aided by others having a knowledge of the performance of helicopters rather than that of their propulsive systems—show that there is an optimum Mach number of 0 25, and this value is used in all succeeding design point calculations for this scheme It will be seen from the curves in Fig 5 that although the optimum value of the auxiliary compressor pressure ratio varies with the rotor blade tip Mach number, there is little loss of performance in using a design value of 1 8 in all cases It is to be noted that the thermodynamic design performance of this system is inferior to that of the shaft drive

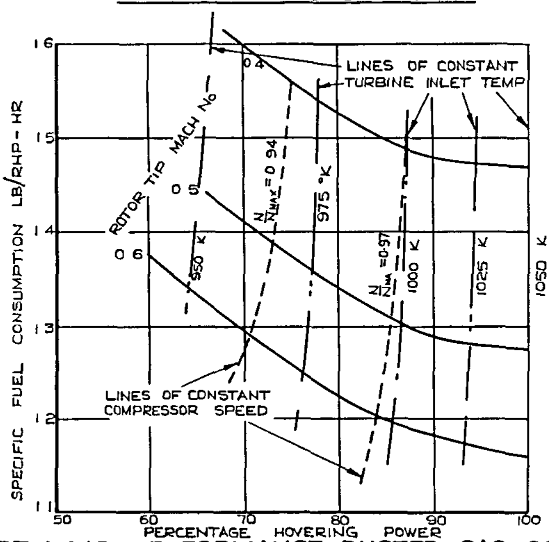
The rotor propelling nozzle areas can be determined from the design point calculations, and part load performance estimates are made for each rotor blade tip Mach number using the design propelling nozzle area appropriate to that speed Fig 6 shows a plot of specific fuel consumption against percentage hovering power, with lines of constant compressor speed and maximum cycle temperature superimposed Thus, for any required cruise power, the specific fuel consumption may be obtained

FIG 5



OPTIMUM PRESSURE RATIO FOR DUCTED GAS SCHEME

FIG 6



PART-LOAD PERFORMANCE DUCTED GAS SCHEME

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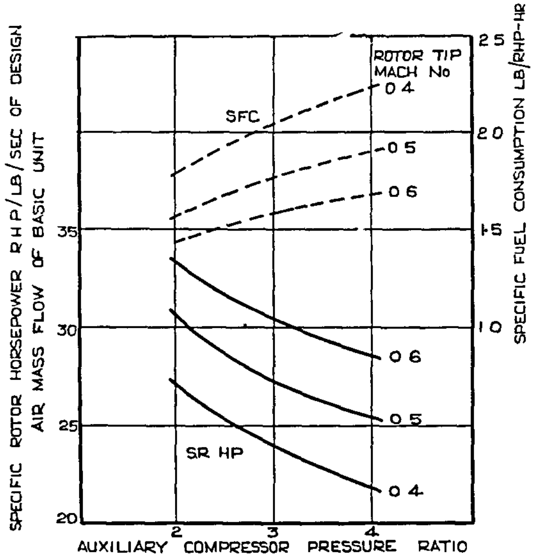
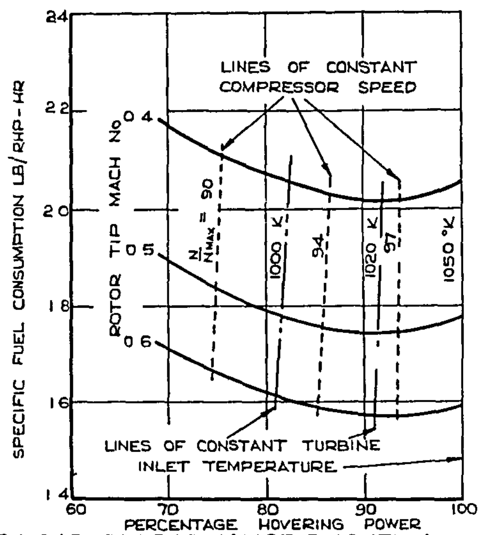
The third main scheme to be considered has already been described as the ducted air scheme As with the ducted gas scheme, the main independent variable, with design conditions fixed for the basic turbo-compressor, is again the pressure ratio of the auxiliary compressor The difference between this scheme and the previous one is that only the auxiliary compressor

FIG 7



FULL LOAD PERFORMANCE DUCTED AIR SCHEME

FIG 8



PART-LOAD PERFORMANCE DUCTED AIR SCHEME
DESIGN PRESSURE RATIO OF AUXILIARY COMPRESSOR=3/1

air is fed to the rotor blades and the full potential output of the main unit is available for driving this auxiliary compressor Once again, it can be seen that, for given design conditions, increasing the mass flow of the auxiliary air is accompanied by a reduction of its pressure ratio and therefore of final jet velocity , hence an optimum pressure ratio will occur The design point performance curves plotted in Fig 7 cover a range of auxiliary compressor pressure ratios from 2 to 4 and indicate that the optimum pressure ratio is lower than 2, its actual value being only slightly lower than 2, after which there is a very rapid fall in power with decreasing pressure ratio The curves of Fig 7 are based on a mean value of flow Mach number along the rotor blade of 0 1, thus giving velocities comparable with those assumed in the previous scheme Calculations involving blade drag estimates, however, show that, all factors considered, the Mach number of air flow along the blade should be raised to achieve the best result, but only small differences in the results would then occur

Schemes are explored with possible design point values for the auxiliary compressor pressure ratio of 2, 3 and 4, and a typical part-load performance curve, for the case of auxiliary pressure ratio equal to 3, is shown in Fig 8 Similar curves could be drawn for other pressure ratios, but they all show a thermodynamic performance inferior to that of the ducted gas scheme Once again, in these cruise calculations, the design propelling nozzle area appropriate to a particular rotor tip Mach number is used

Rotor power may, as remarked earlier, be increased by burning additional fuel in combustion chambers at the rotor tips, and in this case the final temperature prior to expansion in the rotor nozzle becomes important A maximum gas temperature of 1500°K (2240°F) is chosen for the rotor nozzles as a reasonable figure from a metallurgical viewpoint, but results are also presented for a temperature of 1000°K Again, a wide variation of auxiliary compressor pressure ratio is covered for each rotor nozzle gas temperature As in the ducted air scheme just described, a value of 0 1 is chosen for the airflow Mach number along the rotor blades, and this value is also used to fix flow conditions in the rotor tip combustion chamber This combustion chamber is assumed to have a low pressure loss equal to 6 inlet dynamic pressure heads

Fig 9 shows, for a rotor nozzle gas temperature of 1000°K , a plot of specific rotor horsepower against auxiliary compressor pressure ratio within the range of pressure ratio 2 to 7 Power is seen still to be increasing at 2, but by arguments similar to those used previously, maximum specific power could be obtained at a pressure ratio somewhere below 2 However, the curves shown in Fig 10 indicate a point of minimum specific fuel consumption at about $3\frac{1}{2}/1$, a value which, within the range covered, is almost independent of rotor tip Mach number The rapid rise in s f c from the optimum to a pressure ratio of 2 precludes consideration of any values lower than 2 Fig 11 shows curves similar to those of Fig 9, but for a rotor nozzle gas temperature of 1500°K , it can be seen that specific powers are substantially higher for this case than for the lower gas temperature Fig 12 presents a set of curves for specific fuel consumption at the higher gas temperature , the optimum pressure ratio has moved to about $4\frac{1}{2}$, and is again largely independent of rotor tip Mach number

The cruise calculations call for a definite limitation to be placed on one of the many variables since it will be appreciated that for a given rotor power, the lowest specific fuel consumption is obtained when the greatest proportion of the fuel is burned in the main unit Thus, logically, the main unit should be run continuously at 1050°K and full speed, variations in power being obtained by reducing the rotor nozzle gas temperature However, the full speed conditions are regarded as short-time ratings only, such as take-off conditions in engines for conventional aircraft, so that a

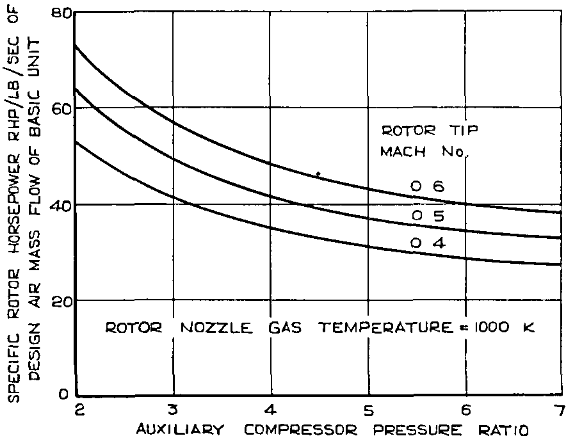


FIG 9

FULL LOAD PERFORMANCE DUCTED AIR SCHEME WITH BURNING AT ROTOR TIPS

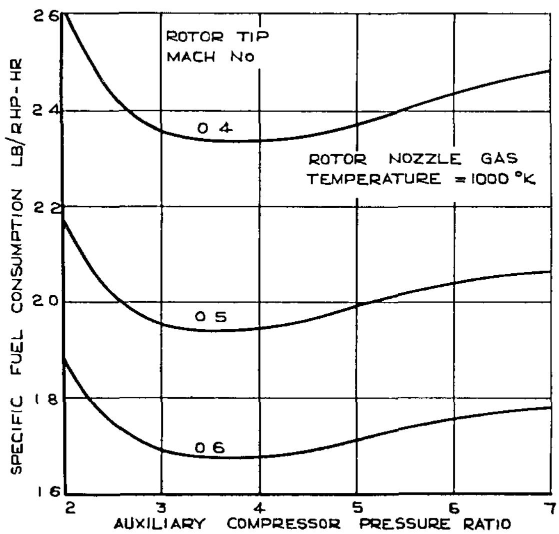


FIG 10

FULL LOAD PERFORMANCE DUCTED AIR SCHE WITH BURNING AT ROTOR TIPS

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definite cruising engine r p m equal to 97% of the maximum is laid down, temperatures then following from thermodynamic calculations If helicopter cruising requirements should demand a turbine inlet temperature in excess of the engine maximum cruising limit of 1020°K, already mentioned, this would be recorded as a disadvantage of the scheme

Fig 13 presents typical cruise performance curves, in this case, the

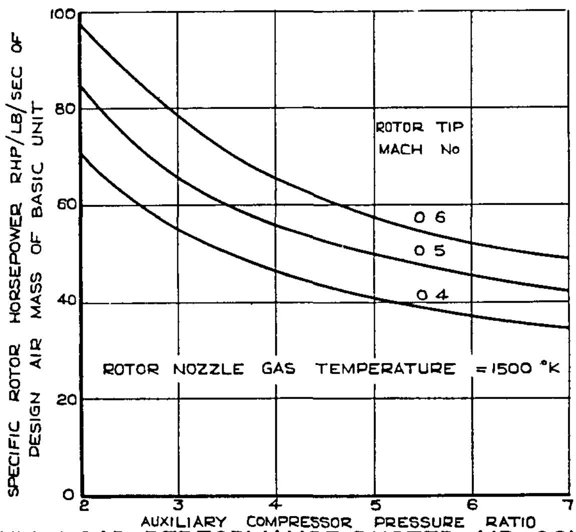


FIG 11

FULL LOAD PERFORMANCE DUCTED AIR SCHEME WITH BURNING AT ROTOR TIPS

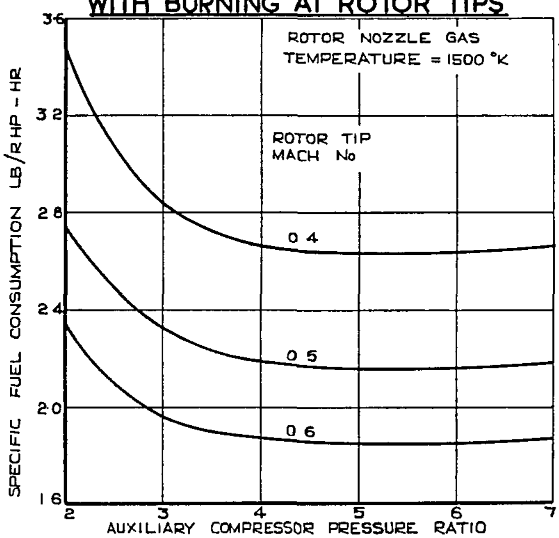
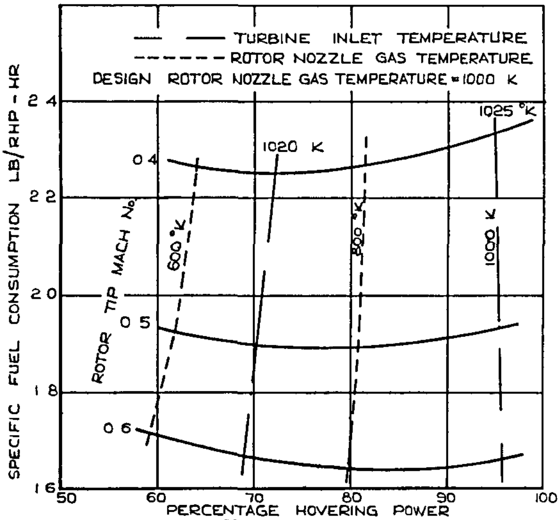


FIG 12

FULL LOAD PERFORMANCE DUCTED AIR SCHEME WITH BURNING AT ROTOR TIPS auxiliary compressor design pressure ratio is equal to 3 and the design rotor nozzle gas temperature is 1000°K The main unit r p m is 97% of the maximum The rotor nozzle design areas vary according to the rotor tip Mach number, but for cruising, the nozzle areas are held constant and equal to the appropriate design value Fig 14 shows curves to the same basic specification but for a design rotor nozzle gas temperature of 1500°K

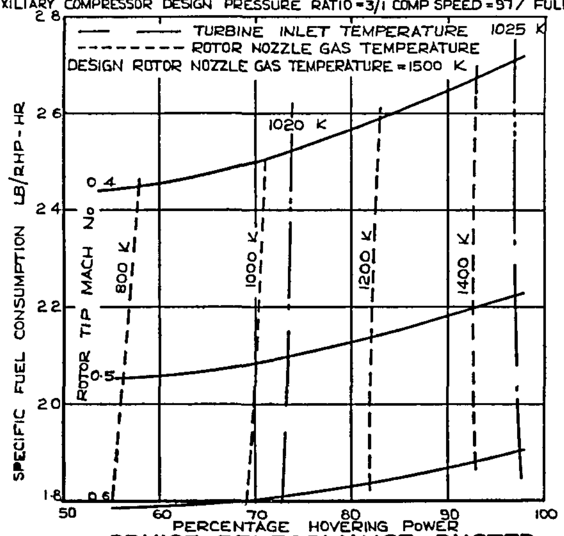
FIG.13



CRUISE PERFORMANCE DUCTED AIR SCHEME WITH TIP BURNING

AUXILIARY COMPRESSOR DESIGN PRESSURE RATIO = 3/1 COMP SPEED = 97/ FULL SPEED

FIG 14



CRUISE PERFORMANCE DUCTED AIR SCHEME WITH TIP BURNING

AUXILIARY COMPRESSOR DESIGN PRESSURE RATIO = 3/1 COMP SPEED = 97% FULL SPEED

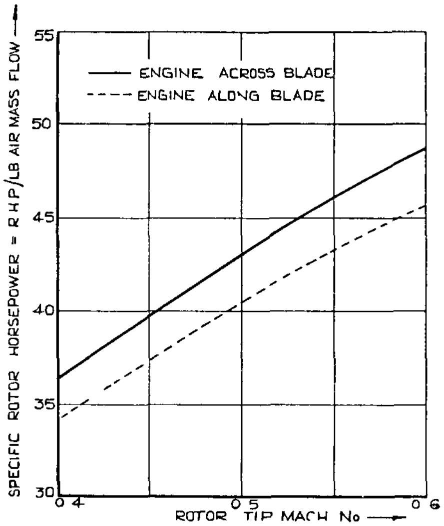


FIG 15

SPECIFIC ROTOR HORSEPOWER

TIP MOUNTED ENGINES

Finally, in this section on performance, I will make brief mention of the tip-mounted engines There are three possible installations, two with engines radially along the rotor blades and one with engines tangentially across Considering first the engines along the blades, there might be an advantage of greater ram pressure rise if the intake is placed at the outer radius of an arm Calculations, taking into account the centrifugal effects on the air in its passage along the length of the engine, show conclusively, however, that the installation having its intake at the inner radius is superior to that with a tip intake Performance estimates of engines mounted across the blades are also made, and results for the two installations are given firstly in Fig 15, which shows the variation of specific power for the two configurations plotted against rotor tip Mach number , and in Fig 16, which shows the variation in specific fuel consumption It should be noted that in arriving at the figures for power, intake momentum has been deducted but engine drag has not No cruise calculations have been made, for, as remarked earlier, the scheme is considered to be mechanically very difficult , these results are presented merely to indicate that, if these difficulties can be overcome, then the specific fuel consumption appears promising in comparison with that of other schemes

I mentioned in the summary of this paper that I would give an indication of engine weight for a chosen size of helicopter I have considered for this purpose a helicopter of 10,000 lb A U W which is powered by two engines of the schemes already described, and have chosen a representative case, with rotor tip Mach number 0 5, for each scheme For the ducted air schemes, I have chosen those with an auxiliary compressor pressure ratio of 3 Now, the full-load design air mass flows to the basic unit of each scheme are some 20 lb /sec or less, and so the most accurate weight estimates can be made by use of a weight breakdown of an existing small, axial-flow gas turbine engine, component weights being scaled for air mass flow according to certain rules Due weight allowance is made for additional turbine stages, and for the number of stages required in schemes employing auxiliary compressors Weights of the pair of engines are estimated as

(1) Shaft drive scheme — engine weight 820 lb, tail rotor gearing and transmission weight 900 lb

(2) Ducted gas scheme — engine weight 1300 lb

(3) Ducted air scheme — engine weight 2340 lb

(4) Ducted air scheme with combustion at the rotor tips to 1000°K gas temperature — engine weight 1170 lb

(5) Ducted air scheme with combustion at the rotor tips to 1500°K gas temperature — engine weight 840 lb

It must also be pointed out, in fairness to the shaft drive scheme, that a saving in weight of some 200 lb is estimated on other parts of the helicopter Similar weight estimates can be made for each variation of the schemes, and would-be gas turbine-powered helicopter operators could eventually draw pay-load endurance curves for specified hover times and cruising conditions This kind of calculation has been made for me, and the results—which really form the conclusions of this paper—show that, for low endurance, because of its low engine weight, the ducted air scheme with burning at the rotor tips is the best, possibly pointing to a conclusion that this type of drive might be advantageous for take-off purposes provided some other scheme of forward propulsion could be used For endurances of, say, 1 hour cruising with 10 min hover, the order of preference would be shaft drive, ducted gas, then ducted air schemes, with or without tip-burning

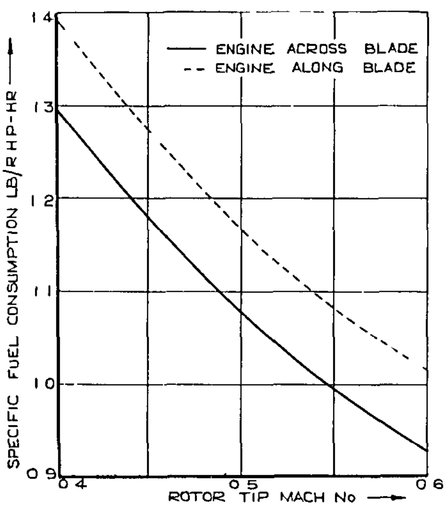


FIG 16

SPECIFIC FUEL CONSUMPTION
TIP MOUNTED ENGINES

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NOTE

Mr BROWN, in presenting his paper, explained that Figure 7 might need correction This scheme and the one with the tip burning had not been shown in their best perspective—in other words, the Mach number ought to have been increased from 1 to something like 2 or 25 , but having done all the calculations on 1, he had not gone to the trouble of doing them again

To stimulate discussion, he added at the end of his lecture that the chief reason for the variation in the engine weight figures (*i e*, 820 lb 1300 lb , 2340 lb , 1170 lb and 840 lb) was not only the specific power produced by the various schemes, but also the actual power required *i e*, the horse power calculated as being required to lift the 10,000 lb helicopter This was something which appeared to be difficult to calculate, and people did not seem able to agree on it It depended upon such things as the drag of blades which carried ducting and upon whether a blade which carried along it a series of ducts which had in them hot gas or air could be made to give drags as low as those of the conventional blade

Discussion

The **Chairman** said that they had listened to a very interesting discourse and he—in another capacity—was relieved that the ducted air scheme with tip burning had come out reasonably well ! Before making any detailed comments on the paper he would like to hear the views from the engine builders and others Accordingly he called upon Mr BELL, Chief Designer (Engines) to Blackburn & General Aircraft Ltd , to open the discussion

Mr F R Bell (*Chief Designer (Engines), Blackburn & General Aircraft, Ltd*), who opened the discussion, said that Mr Brown's paper was extremely useful because it gave a great deal of information which he had always wanted and also because Mr Brown had come to the conclusion which he wanted him to reach, having said the same things himself On most of the points he was in complete agreement

One of the difficulties with helicopters today was the question of safety over cities and the like For civil transport helicopters at any rate, twin engine reliability was needed The author had said that the free turbine was better for periods of over 30 minutes, but from the point of view of safety, even if one wanted to travel for only 30 minutes, one would need a greater margin Were he himself the A R B , he would say that he wanted longer than 30 minutes Therefore, even on Mr Brown's own conclusions, the free turbine type of engine with a mechanical drive would appear to be better

Taking into account the twin engine reliability, however, the picture leaned even more that way As the lecturer had shown, for all schemes other than the free turbine and the hot gas scheme burning at the rotor tip, the engine weight was much greater

It was a characteristic, apparently, of the helicopter that its cruise performance required somewhat more than half its maximum hovering power This meant that with a twin engine unit, and if the helicopter was capable of cruising on one engine —which was necessary for safety—there must be somewhat bigger engines than was demanded by the maximum power conditions The engine weight then became a rather greater percentage than Mr Brown had used In other words, all the calculations being made for a single engine, if one took into account the cruise conditons on one engine and on a twin engine unit, these figures would be emphasised somewhat because the engine weight was a bigger percentage

He admitted that his next remarks might not apply to small helicopters—privately-owned little things, running on one engine—but in order to use two engines with a ducted gas scheme a variable tip jet was necessary, because when one engine was closed down the nozzle areas must be closed down proportionately, otherwise the