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PROJECT Y

AERODYNAMIC DESIGN AND PERFORMANCE PREDICTION OF THE ENGINE

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PROJECT "Y"

May 12, 1953.
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To - Prof. D. L. Mordell

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April 27th 1953

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Aerodynamic Design and Performance Prediction of the Engine

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INTRODUCTION

The engine for Project 'Y' is an unconventional turbo-jet having a double sided radial flow compressor, double combustion system and double sided single stage turbine. The exhaust from the turbine emerging at the periphery of the engine is collected and expanded in nozzles facing towards the rear of the aircraft. The engine is unconventional only in its geometry and arrangement in the aircraft.

In this report the main features of the aerodynamic design are discussed and aerodynamic specifications given. The results of performance predictions relating to steady and transient conditions are presented graphically.

BASIC DATA AND ASSUMPTIONS

Air Intake - For all flight conditions a loss in total head equal to that corresponding to a plane shock wave together with a subsonic diffusion at 90% isentropic efficiency has been assumed.

Compressor - At sea level static conditions a compression ratio of 3 to 1 with isentropic efficiency 84% has been taken. With 288°K inlet total temperature, a rise of 126°K is required.

Combustion Chamber - A loss of 5% of the total pressure at entry is assumed. An effective heat release of 10,300 CHU's/lb. has been used to calculate fuel consumptions. With a wide range fuel of 10,500 CHU's/lb., this allows for 2% combustion losses. Figures for fuel flow or specific consumption may be directly corrected for any other fuel calorific values or combustion efficiency values.

Turbine - Inlet temperature 1100°K. Isentropic efficiency 85%.

Exhaust System - An expansion efficiency of 96% was assumed. In the calculations no allowance has been made for any losses due to the angle effects in the installation because of uncertainty as to their probable magnitude and to the benefits which might accrue from the Coanda effect and the Prandtl-Meyer deflection. The unit has been sized, however, to allow for an 8% loss of thrust at static sea level conditions.

Design Mass Flow - With the data assumed above the specific thrust is 52.8 seconds. The unit has been designed for a Mass flow, at take-off, of 823 lb./sec., giving 43,500 lb. thrust without any angle effect, or 40,000 lb. with 8% loss.

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SYMBOLS

M	Mass flow	lb./sec.
f	Fuel flow	lb./sec.
P	Total Pressure	p.s.i.a.
T	Total temperature	°K
F	Thrust	lb.
N	R.P.M.	
SFC	Specific fuel consumption	lb./lb.thrust.hr.

SUFFICES

a	Ambient
1	Compressor inlet
2	Compressor outlet
3	Turbine inlet
4	Turbine outlet
v	Velocity component of thrust
p	Pressure component of thrust

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PART I - DESIGN

Compressor

General - For mechanical reasons the lowest possible blade speed and smallest number of stages are desirable. For this reason a low stagger design with air leaving angles of almost zero has been used. It is recognised that this entails some sacrifice of peak efficiency, but trial calculation showed that the assumed efficiency should be attainable. Other major considerations taken into account were:-

a) The desirability of using a modest Mach number at the first rotor.

b) The desirability of having the final radial discharge velocity into the combustion system diffuser as low as possible.

c) The advantage of using a common blade profile for all stages.

Choice of Triangles - After several preliminary designs it was found that a good compromise could be made by using a 45° camber blade, 9C7.45.C50 of 0.3" chord. Cascade data on this profile was available from work of the Gas Turbine Engineering Division at A.V.Roe(Canada)Ltd. In view of the low angle design chosen there was little virtue in using an inlet stator. On the other hand it diminished capacity, increased losses, and presumably would be a potential source of icing danger. Accordingly the stator has been dispensed with and the air enters the first rotor radially.

Using a relative Mach number onto the first rotor of 0.72 it was found possible to design for the required temperature rise (with a 1% margin) and at the same time to reduce the radial velocity from 633 ft./sec. at entry to 426 ft./sec.at the end of the final stator. Due to the diminishing work done factor the stage rises are just about constant, in spite of the increasing rotor speed.

Table I gives the essential dimensions and details of the air angles and velocities taken from the design triangles. To compute the blade heights a stage efficiency of 89% was assumed and lines 15 and 16 in Table I give the values of total and static pressure through the compressor, and the radial area required for 323 lb./sec. is shown in line 17. It will be noted that a loss of total pressure of 0.46 p.s.i. was allowed for the intake ($\Delta P/q = 0.13$). If relieving doors are fitted this loss may be less, but will then serve as a margin for developement. The compressor performance is summarised below in different ways:

a) Delivery total pressure from intake total pressure:-

Ratio 3.19
Isentropic Efficiency ... 87.8%

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b) Delivery total pressure from Atmosphere:-

Ratio 3.09
Isentropic Efficiency ... 85.3%

c) Delivery static plus 60% dynamic pressure from Atmosphere:-

Ratio 3.00
Isentropic Efficiency ... 83.2%

No allowance has been made for the centrifugal compression effect, which will be about 4% of the temperature rise used. The slight extra pressure ratio will improve performance, particularly at the higher speeds.

Blading - Using Howell's methods the cascade performance of the blades has been estimated and representative examples are given in Table II. The first is for a cascade set at S/C = 0.7, A.R. = 3.0, α_{ib} = 37.4 and the second for S/C = 0.5, A.R. = 2.0, α_{ib} = 33.9, representing conditions at the first and last stages. In calculating the stage efficiencies Howell's data has been used. It is considered that in so far as secondary and "annulus" drag is concerned this is conservative. The untwisted blade with constant lift coefficients span-wise should diminish the secondary loss appreciably while the centrifugal force on the rotor blade shrouds should diminish the "annulus" drag. On the rotor blades, centrifugal force will act on the boundary layer on the convex side and should help to keep it thin. On the other hand the radial diffusion in the blading results in high pressure rise loadings, although the small space/chord ratio reduces the pressure gradient and the lift coefficients are quite modest.

In the later stages, conditions up to about 92% of the maximum have been used. This is slightly higher than usual, although tests on the Orenda engine have demonstrated the improvements possible by using values greater than the more usual 80% of maximum. The stage efficiency analysis in Table II shows that the best efficiency is reached at an incidence just over the design value. At the design Mass flow it is considered that the assumed 89% stage efficiency is a shade conservative so there should be no difficulty in realising the designed conditions. It should be noted that no allowance for compressibility is made since the maximum Mach number is 0.72 which is the estimated critical Mach number for the first stage. At altitude and high speed the operating Mach number is well below this.

In considering off-design conditions, the use of the low stagger blading gives relatively flat characteristics, while we have adopted a design with increasing incidence through the compressor. This will help to ensure high efficiency at part load and when starting. At the last stage there is a comfortable margin before the stall, and since increasing incidence increases the deviation the real margin is still greater. Table III gives the settings and blade space/chord ratios recommended.

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Combustion Chamber

The layout of the engine lends itself admirably to some form of "annular" combustion system rather than a multiple can type. However, it must be recognised that such a system, as shown on Lucas drawing CHD 23175, presents a combustion problem and a mechanical problem of great magnitude, which will require a considerable developement programme. In order to avoid any possibility of delaying engine developement as a result of combustion troubles it is believed that it is best to design the prototype machine to use a number of conventional flame tubes, whose combustion performance can easily be verified on a low power rig. A rather unconventional air intake system is needed to provide good distribution to the flame tubes. The inlet velocity will be 425 ft./sec. and it is hoped that the profile will not be too distorted. A suitable scheme is shown on Lucas drawing RL 3154, using conventional loadings on the flame tubes, but developement of the intake ducts on an airflow rig should be put in hand as soon as possible. The flame tube design should be regarded as an interim measure to give time for developement of some form of fully "annular" system.

Turbine

The turbine mean blade speed is 733 ft./sec. Assuming 2% air and gas leakage and 1.5% fuel addition a heat drop of 30.4 CHU'S/lb/sec. is demanded. This necessitates a whirl change of 1750 ft./sec., or 2.24 times the blade speed, which is a reasonable loading. Due to the radial arrangement of the turbine, the radial flow area increases (with blades of constant span) and this, combined with the relatively low pressure ratio across the turbine necessitates a slight amount of exit whirl into the exhaust system to avoid recompression. 10° whirl has been used to give a slight degree of reaction, in the interests of high efficiency.

With a mean radius of 112", 1" radial chord, and 2.30" blade height, the following figures specify the turbine triangles and bladings:

Nozzle leaving velocity 1835 ft./sec.
" " " (radial component) 980 ft./sec.
" " " (whirl component) 1550 ft./sec.
Nozzle leaving angle 57.7°

Rotor entry velocity 1245 ft./sec.
Rotor entry angle 38.1°

Rotor leaving velocity 1500 ft./sec.
Rotor leaving angle 40.5°

Absolute leaving velocity 1152 ft./sec.
Absolute leaving angle 10°

It is assumed that T_4 turbine profiles are to be used, and the recommendations of N.A.E. report R17 were followed:

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Rotor Blades	S/C	=	0.75
	Camber	=	91°
	Incidence	=	0°
Stator Blades	S/C	=	0.70
	Camber	=	67°
	Incidence	=	0°
	Throat Area	=	1670 sq.in.

Following the report cited, the total losses were estimated at 14° giving a stage efficiency of 88.9%.

Reheat Combustion Chamber

In making performance calculations, the effect of a mild degree of reheat to 1200 and 1500°K has been considered. The velocity leaving the turbines will be 1152 ft./sec. If this be diffused to 500 ft./sec., an area of 6400 sq.in. = 44.5 sq.ft. will be needed. For the static sea level cases the heat release in the reheat system will be about 100 or 240 x 10^6 CHU's/hr/atm depending upon the reheat temperature. Using a figure of 6 x 10^6 CHU's/hr/atm/cu.ft. for the heat release we arrive at a volume of 16.65 cu.ft. or 40 cu.ft.

Bearing in mind the geometrical arrangement of the engine it is evident that if good diffusion is obtained - which is desirable for low jet pipe losses anyway - it will be easy to provide for the modest degree of reheat contemplated as a possible developement.

General Discussion of Design

Throughout the design, every attempt has been made to be conservative. The value of 85% for turbine efficiency used in all design and performance work compares with a predicted value of 88.9%. This is considered to be a reasonable value and leaves a little margin to take care of any unexpected additional losses in the exhaust or combustion systems.

In the compressor attention should be drawn to two points. The high values of $\Delta p/q$ used in the last stages are not considered excessive. The close spacing, absence of twist and favourable centrifugal effects will, it is considered, make their attainment possible. It is possible to lower them to values well within usual practice by dispensing with the reduction of radial velocity in the last two or three stages. However, the diffusion must be performed somewhere, and the combustion performance may suffer if we put up the velocity at entrance to the chambers.

The other noteworthy point is that the use of a common blade, although obviously desirable from the manufacturing viewpoint, means that the first stage blade chord is 0.8". The Reynolds' number at sea level take-off will be 3.2 x 10^5, while at 50,000 ft., at Mn = 1.0, it will only be 0.77 x 10^5, and at Mn = 2.0, 1.2 x 10^5. This will result in a loss of work capacity of the first one or two stages which will in turn lessen the working incidence in the following stages so that the compressor will lose pressure-ratio.

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General Discussion of Design (Continued)

Little direct evidence on the importance of the Reynolds' number under these conditions is available to the writer. Increase of the blade chords of the earlier stages is possible, and has no serious aerodynamic consequences.

PART II - PERFORMANCE PREDICTION

Steady State Performance

The basis for all performance calculations is the compressor characteristic diagram. The measured characteristics of a six stage axial flow compressor were used. At a measured compression ratio of 3 to 1 the mass flow was suitably scaled and the efficiencies corrected to give 84% at the design point. These characteristics are plotted on Fig.1. The working lines for zero flight Mach number, Mach number 0.5, and Mach numbers greater than 1.0 are shown. For the turbine, in the steady state, at which there is only a very slight shift in incidence over the whole working range, a constant efficiency of 85% was assumed. The nozzle guide vanes were assumed to hold $M\sqrt{T_3}/P_3$ to 652 at all conditions. The calculations were made by the usual methods of graphical superposition. The basic characteristics are shown on Figs. 2, 3, 4, and 5 giving plots of $M\sqrt{T_1}/P_1$, $f/P_1\sqrt{T_1}$, P_4/p_1 and T_4/T_1 against $N/\sqrt{T_1}$. From these for given entry conditions the complete performance is easily worked out. Figs. 6 and 7 plot thrust and specific consumption for various flight conditions.

These curves are plotted from data tabulated on sheets i - vi in the appendix, on which other relevant data may be found, including values of the velocity and pressure components of the gross thrust, and the intake momentum drag. At high speeds the thrust is very sensitive to slight errors as it is a small difference between the gross thrust and the intake drag. The accuracy at speeds above Mn = 2.0 is 5% and about 2.5% at lower speeds. It should be emphasised that at speeds greater than Mn = 2.0 the thrust is very sensitive to the shape of the compressor characteristics - which cannot be exactly estimated at this stage. Those shown in Fig.1 are believed to be slightly worse, from the viewpoint of high speed performance, than those to be expected, as we have designed the compressor for a smaller leaving angle than the compressor whose characteristics were employed.

Transient Performance

For the transient calculations the compressor characteristics were extrapolated down to 20% of the design speed. In order to approximate the turbine performance, the mass flow was assumed to be controlled by the guide vane area and the turbine pressure ratio, while the efficiency was assumed to vary only with the value of the ratio $(N/\sqrt{T_3})^2 \div \Delta T^{x}/T_3$ where ΔT^{x} is the isentropic temperature drop. This variation is shown on Fig.8 and was taken from rig tests of single stage turbines of the same work capacity as that proposed. Using assumed jet pipe characteristics the curves shown in Fig.9 were prepared, and then mated with the compressor characteristics. Each point yields a value of thrust, fuel consumption and the net shaft torque. Taking 8000 slugs-ft^2 for the polar moment of inertia, we convert torque to dN/dt, which is plotted on Fig.10 as a function of $N/\sqrt{T_1}$ and T_3/T_1.

To calculate possible acceleration times, values of 1/dN/dt were integrated with respect to N. We assumed a value of T_3/T_1 = 4.16 up to 70% speed falling to 4.0 at 100% speed and on Fig.11 are plotted the speed and thrust curves against time.

The time zero is at 160 R.P.M. and the time to accelerate to 800 R.P.M. is 71 seconds. From 25% speed the time is about 45 seconds. These values must not be taken too exactly, as considerable extrapolation was necessary at the lower speeds, and no allowances were made for Reynolds' number effects. However, it is believed that the figures are a useful indication. The engine should be self running at 100 R.P.M.

Starting

To make a rough estimate of the starting performance, the cold motoring power is found by extrapolation to be about 110 x $(N/100)^3$ H.P. and the starter motor is assumed to deliver 110 H.P. at 100 R.P.M. and to deliver this H.P. over a speed range from 200 R.P.M. down to 33 R.P.M., below which speed a constant torque is assumed. We postulate a starting cycle with ignition at 75 - 80 R.P.M. and by integration find a time of about 25 seconds from 0 to 200 R.P.M. Again this should be taken as a first approximation rather than an exact estimate, as a lot of extrapolation is required.

General Discussion on Performance

It will be seen that a useful performance range is available for the flight speeds considered. At altitude, maximum thrust is achieved at Mn=1.7 but the fall off at Mn=2.5 is not very great. A modest amount of reheat (to 1200°K) gives more than 50% increase of thrust at Mn=2.5, and reduces specific consumption. In calculating consumption, no allowance has been made for any possible fall off in combustion efficiency at extreme altitudes.

The starting and acceleration times are not at all unreasonable, the time of 45 seconds from idling to maximum R.P.M. comparing with about 5 seconds for a conventional 6500 lb.thrust engine. This is a factor of nine and since the ratio of the moment of inertia to mass flow is about seventy, and the R.P.M. about an eighth that for a conventional engine, is just what would be expected.

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PART III - DEVELOPMENT POTENTIAL

In formulating the aerodynamic design, every possible attempt has been made to be conservative. Aerodynamically the engine corresponds to a conventional engine of about 1947. There is consequently a great deal of developement potential built into the design.

The Compressor

An efficiency of 84% for a 3 to 1 compression ratio is very low, and a considerable improvement should be possible. This engine, in order to meet the high speed flight requirement, has a surplus of take-off power. By restricting the sea level performance or accepting some sacrifice in efficiency, the compressor could be designed to have its optimum efficiency at the operating point at high speeds. A target of 88% would then be perfectly feasible.

The compressor mass flow can be increased slightly if need be by simply tolerating a higher first stage Mach number at sea level with again a sacrifice in take-off consumption to get more high speed thrust. On a larger scale by simply lengthening the blades, more mass flow can be handled at some expense in weight.

The Combustion System

Due to the low pressure ratio, the combustion system is liable to restrict the mass flow which can usefully be used. Any reduction in pressure loss factor will be of direct benefit, but if an improvement in component efficiencies is made, some of it may be used to permit a mass flow increase without waiting for a reduction in the loss factor.

The Turbine

Again the assumed efficiency of 85% is low, and in fact our own estimate suggests better than 88%. This will be of great help at high speeds as it will provide a little extra gross thrust, without increasing intake drag or fuel consumption.

The simple untwisted blade lends itself very easily to fabrication with an internal cooling passage for air cooling and any increase in permissible maximum temperature results in a large increase in nett thrust at high speeds.

The effects of the possible improvements in compressor and turbine efficiency will give an extra 15% of thrust at Mn = 2.5 without any other change.

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