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Project 'Y'

An assessment of the power plant*

- by -

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SUMMARY

An assessment has been made of the overall performance of the engine at sea level static conditions and at various forward speeds up to M = 2 at the tropopause. The aerodynamic design of the compressor has been examined and blade stresses in the first and last stages checked. Other engine components have been examined briefly. Some comments on the major mechanical design features are included.

It is concluded that whilst there is no fundamental reason why an engine of the form proposed should not be made, the mechanical design problems are formidable. The chief difficulty lies in the necessity for preventing excessive rotor deflection under the high gyroscopic loads that will exist when the aircraft is manoeuvring.

The proposed engine layout offers no advantage as far as the efficiencies of the compressor, combustion system and turbine are concerned, but the losses in the intake and exhaust will be greater than those in engines of conventional design. In addition there will be a loss of about 8 per cent of gross thrust due to the non-axial discharge of the exhaust.

It has not been possible to make a reliable estimate of the engine specific weight, but the Brochure quotes values in the region 0.2 - 0.3 lb./lb. thrust. These figures are not outstanding when compared with proposed developments of conventional power plants.

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*This paper should be read in conjunction with R.A.E. Technical Memorandum Aero. 316 "Preliminary comments on a proposal for an aircraft powered by a radial-flow jet turbine engine" by J. R. Collingbourne and A. L. Thorpe.

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CONTENTS

ILLUSTRATIONS

1.0 Introduction

This Note presents an overall assessment of the proposed power plant for Project 'Y' and is based on details given in the design brochure(1) and information obtained during discussions with the authors of the scheme, Messrs. J. C. M. Frost and T. D. Earl.

2.0 General description

The power plant consists essentially of a single rotor 19 ft. diameter carrying both the compressor and turbine blades and rotating about a hollow centre body some 11 ft. diameter within which the pilot is housed. The axis of rotation is vertical and the whole engine is symmetrical about a horizontal centre line through the rotor. In what follows only one half of the engine will be described. The terms front and rear relate to the direction of flight.

The intake is divided into four sections each of which feeds one quarter of the compressor inlet annulus. Thus air entering the front of the compressor has to turn through 180° whilst that entering the rear undergoes only a small lateral displacement without changing its general direction of flow. After compression the air passes into the combustion system which consists of 60 flame tubes symmetrically arranged in an annular air casing and thence to the single stage turbine located at the rotor rim. On leaving the turbine the gases are deflected in a generally backwards direction by vanes fixed in the exhaust duct. Part of the exhaust flows from the sides of the aeroplane to produce a forward thrust component and the remainder passes through a rectangular nozzle which extends across the full width of the wing trailing edge.

3.0 The philosophy behind the radial engine

The two main advantages claimed for the proposed engine layout are that it enables a high thrust per sq. ft. of aircraft frontal area to be achieved and that it results in a low overall aircraft weight.

The low frontal area arises from the fact that the main flow area through the engine is not in a plane perpendicular to the line of flight as is the case with conventional installations. The claim for low aircraft weight is based on the fact that main structural members are common to both engine and airframe.

It is clear, therefore, that in view of the close integration of engine and airframe the case for the scheme rests on the overall aircraft performance that can be achieved.

4.0 Overall performance

Except in one or two minor particulars it is not anticipated that the component efficiencies will differ greatly from those achieved in conventional engines. The points of difference arise from the fact that the air passing through the engine has to traverse a more or less tortuous path depending on whether or not it enters and leaves the rotor at the front or back of the engine circumference. Each segment of the engine will therefore operate at slightly different inlet and exhaust conditions due to the circumferential variations of loss in the ducting. Since the overall effect on performance cannot be predicted this effect has been neglected.

Calculations have been made to determine the overall performance of the engine on the basis of the following assumed component efficiencies:

Compressor efficiency	= 85 per cent isentropic
Turbine efficiency	= 85 per cent isentropic
Combustion chamber pressure loss	= 5 per cent compressor exit pressure
Exhaust duct loss	= 7 per cent turbine exit pressure
Maximum gas temperature	= 1100°K.
Combustion efficiency	= 98 per cent
Loss due to non-axial exhaust	= 8 per cent gross thrust
Air mass flow	= 825 lb./sec. (sea level static)
Engine speed	= 800 r.p.m.

Values for the intake pressure recovery factor were obtained from Dr. J. Seddon of R.A.E. and these are shown plotted in Figure 1. In deriving these figures allowance has been made for the losses due to the changes in direction of the air flow.

The loss of thrust due to the non-axial discharge of that portion of the exhaust which leaves the sides of the aircraft was estimated by measuring the efflux angles off a drawing contained in the Brochure. The net thrust is, of course, very sensitive to this loss value particularly at high forward speeds. Thus at M = 2, 36,000 ft. the thrust could be increased by 32 per cent if all the exhaust could be discharged axially. In the following calculations a constant thrust loss of 8 per cent has been assumed.

On the basis of the above component efficiencies the thrust at sea level static conditions was estimated to be 34,400 lb. and the corresponding specific fuel consumption 1.59 lb./hr./lb. thrust. These values compare with the 38,900 lb. thrust and specific consumption of 1.40 lb./hr./lb. calculated using efficiencies quoted in the Brochure.

The performance at the tropopause at maximum engine speed is given in Table I below and is shown plotted in Figures 2 and 3. When making these calculations it was assumed that the propelling nozzle area was variable so that a constant maximum gas temperature could be maintained.

TABLE I

Performance at 36,090 ft., 800 r.p.m.

Mach Number	-	0	0.4	0.9	1.25	1.50	1.75	2.0
Net thrust	lb.	13590	13318	15110	18330	19800	20000	18650
Air mass flow	lb./sec.	247	280	371	499	606	725	844
Fuel flow	lb./hr.	17740	19575	25730	33200	38700	44000	47900
Specific consumption	lb./hr./lb.	1.31	1.47	1.70	1.81	1.96	2.20	2.57

5.0 The compressor

5.1 Aerodynamic design

The compressor design has been considered in fair detail in order to determine whether the required performance could be obtained within the limited radial space available.

The axial velocity at inlet to the first stage was quoted by the designers as 550 ft./sec., a figure which is considered to be rather high. However, design calculations have been made using both this velocity and also a more conservative value of 500 ft./sec.

In order to avoid excessive losses in the diffuser leading to the combustion system the axial velocity in the last stage was assumed to be limited to 400 ft./sec.

The general method of procedure was as follows:-

(1) Blade angles were derived to give the maximum stage temperature rise.

(2) The maximum permissible t/c ratio that could be used without exceeding the critical Mach Number was deduced.

(3) The chord required to give permissible stresses was determined. In this calculation the bending fatigue factor was assumed to be 3, the conventional value for aircraft engine practice.

It will be noted that this method gives the shortest possible compressor.

Drawings presented in the Brochure indicate that shrouding of all rotor rows is contemplated. No allowance has been made in the following calculations for the effect of the shrouds on the blade root stresses, but this is not thought to involve serious error.

TABLE II

Compressor Blade Details

C4 Aerofoil. 50 per cent reaction. s/c = 0.7
Blade material S80 steel (U.T.S. = 55 tons/sq. in.
Fatigue stress ± 24 tons/sq. in.)

		First Stage		Last Stage
Axial velocity	ft./sec.	500	550	400
Blade Angles β_1		34.6	43.0	51.6
Blade Angles β_2		1.1	-3.4	4.8
Blade Angles θ		33.5	46.4	46.8
Blade Angles ζ		17.8	19.8	28.2
Air Angles α_1		39.7	37.9	48.8
Air Angles α_2		8	6	15
Stage temp. rise*	°C.	15.2	16.3	15.1
Maximum permissible t/c		0.12	0.09	0.15
Blade chord	in.	1.05	1.21	0.42
Blade height	in.	4.37	4.07	1.92
Number of blades		594	516	1770
Centrifugal load	lb.	150	140	15.4
Gas load	lb.	7.3	9.0	2.0

*Using work done factor = 0.98 for first stage and 0.83 for last stage.

The performance predicted in the Brochure, namely a temperature rise of 125°C. in 6 stages with a mean peripheral speed of 520 ft./sec. is optimistic even for compressors of conventional design. For example, the first 6 stages of the RA 14 give only 110°C. rise with a peripheral speed of 820 ft./sec. From the analysis given in Table II it appears that a temperature rise of only 15°C. per stage can be expected and therefore in order to obtain the desired pressure ratio of 3:1 at least 8 stages will be required.

There will, however, be an additional pressure rise across the rotor due to the centrifugal effect. An estimate of the magnitude of this is given in the following paragraph.

5.2 The centrifugal effect

The temperature rise resulting from the centrifugal effect is given by:-

$$\Delta T = \frac{U_{m2}^{\;2} - U_{m1}^{\;2}}{2gJK_p}$$

where U_{m1} and U_{m2} are the blade mean peripheral velocities for the first and last stages.

Calculations show that an additional overall temperature rise of 4.6°C. for the compressor with a first stage axial velocity of 500 ft./sec. and 5.0°C. for the compressor with 550 ft./sec. velocity will result. Thus the compressor will require a minimum of 8 stages even when the centrifugal effect is taken into account.

5.3 Compressor length

It has been proposed that all the compressor blades should be of the same chord and section and whilst this may offer great manufacturing advantages it results in an excessively heavy engine. The reason for this is that the most highly loaded blades are those in the first stage (by virtue of their height) and if the chord of all blades is decided by the stresses in the first stage all other stages will be under-stressed. Since there is such a premium on axial length, it is desirable to use blades of varying chord and thickness. In this way the stresses in each stage can be taken to the limit.

Although it has been shown that the chords required in the earlier stages exceed those proposed in the Brochure, the chords of later stages can be reduced. Thus it may well transpire that additional stages can be added without serious increase of the overall length occupied by the compressor.

In order to verify this the radial length occupied by the compressor has been calculated assuming a linear variation of blade chord between the first and last stages. The clearances proposed between successive rows have been given as 0.3 in. and 0.2 in. at the leading and trailing edges of the rotor blades respectively, these values being the same for all stages. Although from an aerodynamic point of view these clearances are unnecessarily large in the later stages, they have been adhered to in the present assessment.

The results of these calculations show that an 8 stage compressor could be accommodated in a length of 14.6 in. if the first stage axial velocity were 500 ft./sec. and in 15.7 in. if the velocity were 550 ft./sec. These values compare with a length of 13.0 in. for the 6 stage design shown on a drawing accompanying the Brochure.

5.4 Compressor matching

It is doubtful whether exact matching of the two halves of the compressor could be achieved over the entire operating range. In the proposed design the outer portions of the rotor disc are exposed to the compressor delivery pressure so that any mis-matching would give rise to a bending moment in the rotor. It is suggested that this problem might be overcome by the provision of pressure-balancing holes in the rotor.

6.0 The turbine

The peripheral speed of 782 ft./sec. proposed, is somewhat low but the temperature drop coefficient $\Delta T/\frac{1}{2}U^2$ of 4.45 in. not excessive and should enable the necessary work to be extracted in a single stage with the assumed efficiency.

An estimate of turbine blade stresses has been made assuming constant section straight blading of thickness/chord ratio 12 per cent. The stress in the leading edge at the blade root due to centrifugal loading is 7 tons/in.^{2} To this must be added the stress due to gas loading but from experience it is not considered likely that the combined stress would be prohibitive. It does, however, appear that the turbine blade carrier ring and its attachment to the disc will be highly stressed and as far as the mechanical design of the turbine is concerned it is likely that this will prove the most important problem to be overcome.

7.0 The combustion system

The design proposed, separate flame tubes in a common annulus, represents a conventional form of design and there is no reason why the required performance of 4.0×10^6 C.H.U./Cu. ft./hr./atm., with a pressure drop of 5 per cent of the inlet total pressure should not be achieved.

8.0 Mechanical design

8.1 The rotor disc

It is considered that the prevention of excessive rotor tip deflection under the high gyroscopic loading that will exist during manoeuvring of the aircraft presents the greatest single mechanical design problem. This requirement would be severe in an engine of the size proposed even if it were mounted in the conventional way within a separate airframe, but in the present case, the situation is further aggravated by the fact that all the aerodynamic loads have to be resisted by members which also comprise the engine structure. Since the deflection of complex structures, even when of conventional form, cannot be determined by calculation to the accuracy required in the present instance, the magnitude of the penalty to be paid for providing the necessary stiffness will have to be determined largely by experiment.

8.2 The rotor bearing

It is proposed to use 60 ball bearings staggered alternately above and below the centre line in order to withstand both upward and downward loads. When the aircraft is on the ground or in steady flight the load on these bearings is due primarily to the weight of the rotor but during manoeuvres the load will be greatly increased due to gyroscopic effects.

The bearings proposed have a 1 in. diameter bore and at an engine speed of 800 r.p.m. rotate at 31,600 r.p.m. Whilst the corresponding track speed would be satisfactory, there are two important design features which are not adequately covered in the proposed arrangement. These are:-

(a) Uniform distribution of loading

To ensure a uniform distribution of loading, the centre pins must be eccentrically mounted so that the bearings can be adjusted individually to bring the periphery of the outer race into positive or even pre-loaded contact with the inner ring of the rotor.

(b) Rotor concentricity

In the proposed scheme, rotor concentricity is maintained by point contact only on all bearings and this would require individual adjustment or spring loading of the bearing spindles in a radial plane. This method of location appears unsatisfactory even if it could be achieved. In order to locate the rotor concentrically with the central hub-member, it is suggested that a number of ball races be introduced, rotating about axes at right angles to those of the existing bearings.

8.3 Lubrication

Lubrication of the central bearing components does not appear to offer any very great difficulty from the point of view of oil supply but the need for excluding oil from the compressor inlet presents a considerable sealing problem.

9.0 Conclusions

Although a discussion of the desirability of developing an aircraft of the type proposed is beyond the scope of this paper, it is worth noting that the performance claimed in the Brochure, namely a top speed of M = 2.5 at 36,000 ft., is not markedly different from that which could be achieved by an aircraft of conventional design using two engines each of 10,000 lb. thrust. The main performance advantages of the Project 'Y' aircraft lie in its ability to take off and descend vertically and in its high rate of climb at sea level, but to offset these must be set its poor subsonic manoeuvreability and short endurance. Further, the use of a single engine of the size proposed results in a high vulnerability to attack and would undoubtedly create severe servicing problems.

A general overall assessment of the power plant has not revealed any fundamental difficulty which would prevent its achievement. The claim for high thrust per unit aircraft frontal area seems capable of fulfilment but that for low overall aircraft weight cannot easily be verified without a more detailed examination than has been possible in the present instance.

The greatest single mechanical design problem is undoubtedly the prevention of excessive rotor deflection under the high gyroscopic loading that will exist when the aircraft is manoeuvring. Provision of adequate stiffness is further complicated by the fact that the aerodynamic loads on the airframe have also to be resisted by the structural members which comprise the engine frame. The proposal incorporates many untried mechanical design features and the feasibility of these will have to be decided mainly by experiment.

No estimates of the power plant weight with the low compression engine were included in the Brochure although values are quoted for the high pressure ratio version. These values, namely 0.23 lb./lb. thrust excluding the structural stiffening ribs or 0.32 lb./lb. thrust including these ribs, are not outstanding when compared with conventional figures.

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DD/RAE(A)
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Mr. A. R. Howell
Mr. S. J. E. Moyes
Library - (2)
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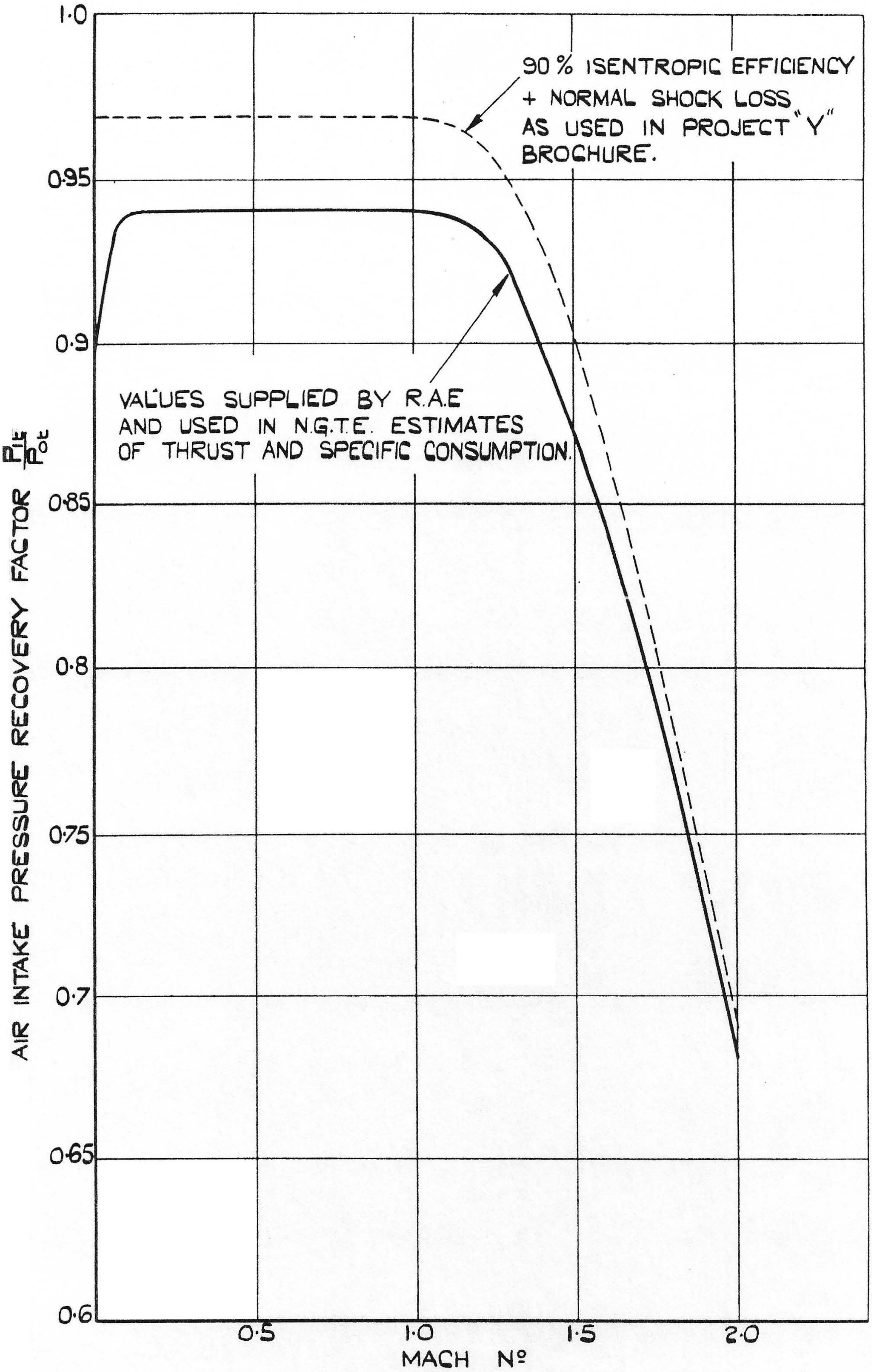
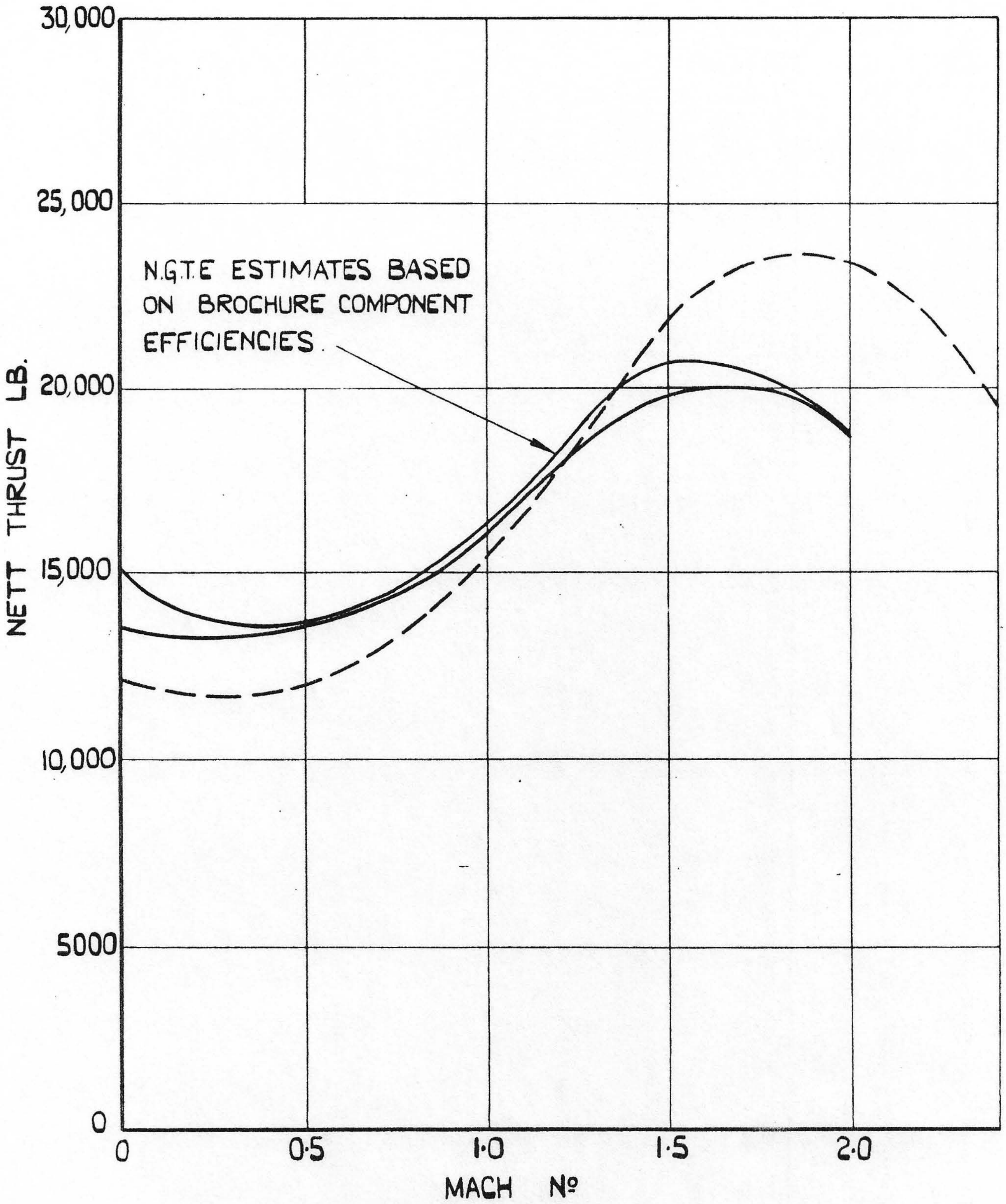
PFA/DGH/BBMP/16/10/3/11.3.53



VARIATION OF AIR INTAKE PRESSURE RECOVERY FACTOR WITH MACH. No.

N.G.T.E. ESTIMATES OF THRUST AS SHOWN BY THE FULL LINES, ARE BASED ON CONSTANT ENGINE SPEED OF 800 R.P.M. AND AIR INTAKE PRESSURE RECOVERY FACTORS FROM FIG 1.

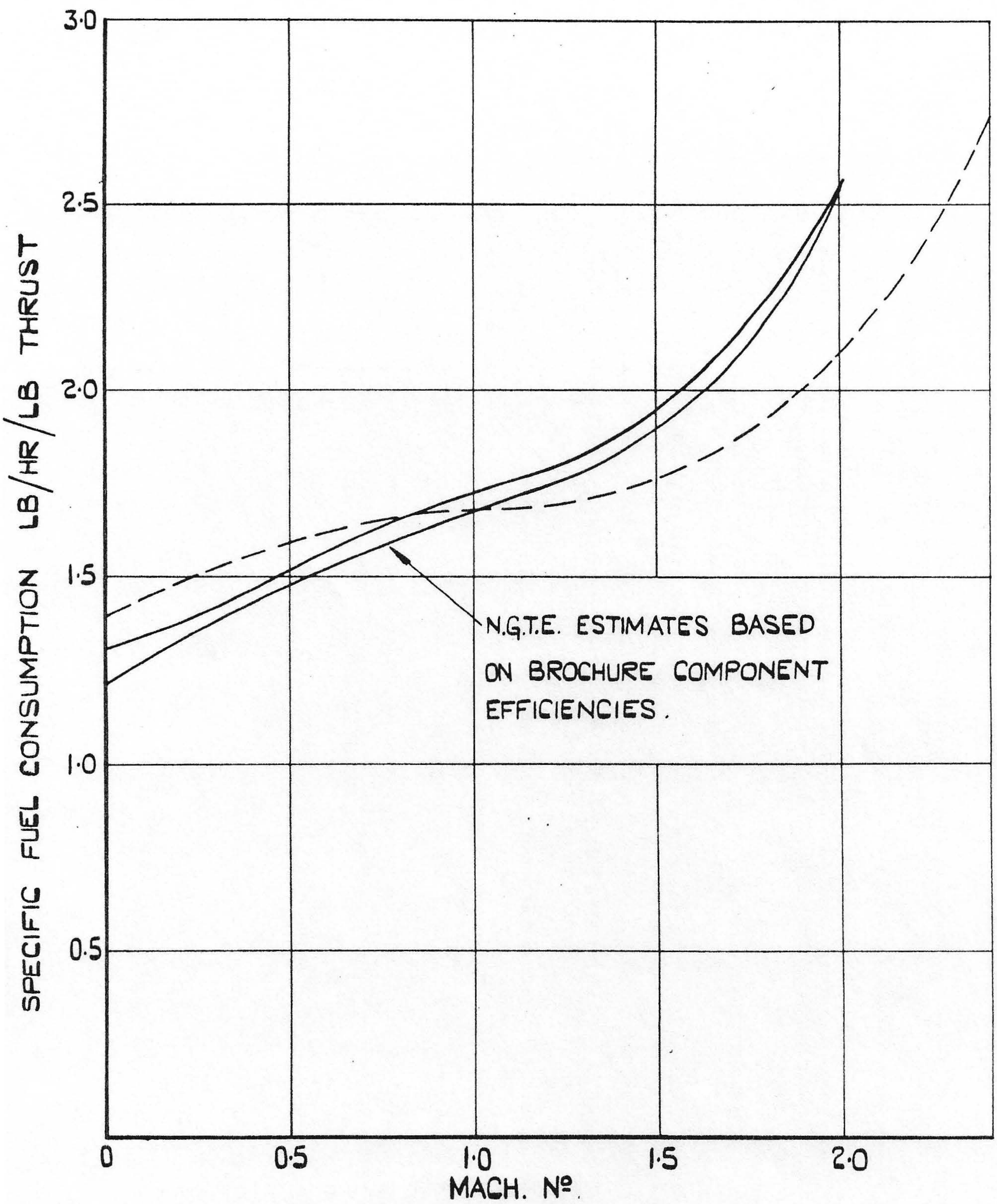
FOR COMPARISON, THE THRUST CURVE FROM THE PROJECT "Y" BROCHURE IS REPRODUCED BY THE DOTTED LINE.



VARIATION OF NET THRUST WITH MACH. No. AT 36,090 FT.

N.G.T.E. ESTIMATES OF SPECIFIC FUEL CONSUMPTION ARE SHOWN BY THE FULL LINES.

FOR COMPARISON THE SPECIFIC CONSUMPTION CURVE FROM THE PROJECT "Y" BROCHURE IS REPRODUCED BY THE DOTTED LINE.



VARIATION OF SPECIFIC FUEL CONSUMPTION WITH MACH Nº AT 36,090 FT.