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Preliminary Comments on a Proposal for an Aircraft Powered by a Radial-Flow Jet Turbine Engine

by

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SUMMARY

This memorandum gives the results of a preliminary assessment of proposals made by Messrs. Avro Canada relating to an aircraft of low aspect ratio, powered by a novel form of jet engine with radial flow through both compressor and turbine.

The engine layout offers no advantage in improved component efficiencies and has a high specific fuel consumption. However its shape makes possible an aircraft with high thrust/weight and thrust/frontal area ratios, capable of vertical take-off and landing and a top speed Mach number in excess of 2.0 in level flight. Endurance will be low.

Response and manoeuvrability characteristics are poor because of the gyroscopic inertia of the rotating engine mass.

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1 Introduction

In a brochure dated July 1952, Messrs Avro Canada have proposed an aircraft powered by a radial-flow gas turbine. They claim that this type of engine, closely integrated with an airframe of suitable shape enables thrust/weight ratios appreciably greater than unity to be achieved, giving a high top speed and the possibility of vertical take-off and landing.

The evaluation of this design presents unusual difficulties inasmuch as both engine and airframe are radical departures from standard practice. The object of this note is to assess the scheme as a whole in a very general way, and make relevant comparisons with more conventional designs. For this purpose, the following important assumption has been made, viz. that the engineering difficulties involved in its construction can be overcome, and that the engine and airframe can be made adequately stiff without any increase in the component weights given by Messrs Avro Canada in their brochure. The firm's weight analysis seems to have been carefully done, but with such a novel design, it is obviously impossible to forsee all the difficulties which may arise.

2 General Description

Fig.I shows the general arrangement proposed at the time of writing. The power plant is roughly in the shape of a circular disc with a central circular cut-out, the plane of the disc being horizontal in straight and level flight. The external diameter of the disc is about 20 feet and the diameter of the cut-out about 8 feet. The airframe is closely fitted around the power plant, its span being only a little greater than the engine diameter and its aspect ratio about unity. The cockpit is situated in the centre of the disc. All vertical sections of the engine which pass through the centre appear similar. Air flows radially outwards through compressor, combustion and turbine stages. The rotor stages rotate about the vertical axis through the centre of the disc. On leaving the turbine, the exhaust gas is still flowing radially and therefore most of this has to be turned to leave the aircraft in a rearward direction. This is achieved by means of guide vanes and ducts round the periphery of the engine, which direct about three-fifths of the exhaust gas through manifolds at the wing 'tips', and about two-fifths through split control surfaces at the trailing edge. The exhaust gas leaving the manifolds nearest the front of the aircraft may have a spanwise velocity component. The consequent loss of gross thrust should be less than 10% of the whole, making no allowance for the tendency these gases will have to follow the direction of the tip, (Coanda effect).

It is proposed to take-off and land vertically, a large, forwards-retracting undercarriage being required for this purpose.

Two flap-type semispan control surfaces are provided at the trailing edge, which can be deflected either together or differentially. The aerodynamic forces resulting from a control deflection are augmented by the deflection of that part of the exhaust gas flowing between the two parallel surfaces which constitute the control. It will be possible to bring about small spanwise deflections of the exhaust gas leaving the trailing edge, by means of rudders mounted internally in the jet stream.

The stability and control of this aircraft presents a novel problem by virtue of the coupling between the pitching and rolling planes resulting from the considerable gyroscopic inertia of the engine rotor. Thus, the response of the aircraft to an applied rolling moment will consist mainly of a rotation about the pitching axis and vice-versa. The C.G. of the aircraft will be somewhere near the rotor axis. This is about 25% of mean chord behind the theoretical subsonic aerodynamic centre, so in order to maintain stable flight at low speeds the aircraft relies on the large gyroscopic damping effect of the rotor. At supersonic speeds, the aerodynamic centre will be near the C.G. and hence the control angles to trim will be small with a consequent saving in drag. At very low speeds, on the other hand, the large downward control deflection required to trim will improve the lifting characteristics.

The aircraft employs large, forward facing intakes on top and bottom wing surfaces. The designers recognise the necessity of providing an efficient method of removing the boundary layer air at the entry. To alleviate the difficulty of designing for low spillage at supersonic speeds while avoiding choking at low speeds, it is proposed to provide a variable entry area by means of a simple 'nose-flap' on the top and bottom edges of the upper and lower intakes respectively.

3 Aircraft Performance

Leading geometrical details of the proposed aircraft are given below:-

Wing area	500 feet^2
Aspect ratio	1.0
Span	23 feet
Section through rotor axis	10% biconvex

The firm's T.O. weight estimate is 26,500 lb. This is made up as follows:-

Structure	Power Plant	Fuel	Fuel Supply	Power Services	Equipment
30% (35.2)	30.9% (32.6)	33.3% (20.2)	1.4% (2.0)	1.7% (4.0)	2.7% (6.0)

The figures in brackets are those estimated[1] for an aircraft of conventional configuration, designed for a Mach number of about 2, and having a similar weight. The lower percentage structure weight of Project 'Y' compared with the conventional design seems reasonable in view of its compact layout and low aspect ratio. Low values for the services and equipment items are presumably a result of this particular design being considered only as a research vehicle. It will be assumed that the power plant weight estimate is a realistic one. It does not include reheat equipment.

The following engine performance figures have been estimated by N.G.T.E. for maximum r.p.m. (800) and no reheat. Corresponding estimates made by the firm are shown in brackets.

Conditions	Nett Thrust lb	Specific Consumption lb/lb nett thrust.hr.
Sea level, static	34,400	1.59
36,090 feet, M = 0.9	15,110 (14,500)	1.70 (1.70)
36,090 feet, M = 1.5	19,800 (22,000)	1.96 (1.75)
36,090 feet, M = 2.0	18,650 (23,500)	2.57 (2.1)

An intake total head efficiency of 94% with normal shock loss has been assumed at all speeds except the static case, for which a value of 90% has been used. These efficiencies should be obtainable with effective boundary layer removal by suction and include an allowance for losses in the bleed system. It should be noted that the mass flow is very large, about 800 lb/sec under static, sea level conditions, and the nett thrust estimates are thus particularly sensitive to the assumptions made for component efficiencies, and in particular to the losses resulting from sideways velocity components in the jet (see para.2). This has been assumed to result in 8% loss of gross thrust, but if this were zero (due, say, to Coanda effect) the nett thrust would be increased by some 32% at M = 2.0. The high specific fuel consumption results from the low compression ratio of this engine, and is comparable with that of a conventional axial flow engine with reheat. In the following table, nett thrust/$\frac{1}{2}\rho V^2$ is compared with corresponding values for the supersonic aircraft design of Ref.1, for tropopause conditions. These are shown in brackets and are obtained with reheat.

Mach number	0.9	1.5	2.0	2.25
Nett thrust/$\frac{1}{2}\rho V^2$	56.0 (34.0)	26.5 (19.0)	14.0 (11.5)	10.0 (9.0)

The values for Project Y with reheat have not been calculated at the time of writing.

The wing planform is roughly that of a cropped delta having a taper ratio of about 0.5. Faced with the problem of building an aircraft round the radial flow engine, the planform chosen seems a logical one for the following reasons.

(a) Assuming all-up weight, root chord, root thickness and taper ratio to be fixed by other considerations, but the span to be variable an analysis shows that A = 1 gives minimum drag for $M \simeq 2$ at 60,000 feet and for $M \simeq 1.2$ at 45,000 feet. An increase in aspect ratio would therefore be a disadvantage by increasing drag, weight and probably duct losses.

(b) A gentle stall at very high incidences accompanied by high drag, which is a characteristic of low aspect ratio wings with sharp leading edges, is necessary in order that a smooth transition may be effected between normal flight and the vertical landing attitude.

The gross mean thickness/chord is about $9\frac{1}{2}$%. Allowing for the intake, the nett mean thickness/chord (nett frontal area/plan area) is about $6\frac{1}{2}$%. In two dimensions, and in the absence of boundary layer effects, the pressure distribution and hence the drag would be determined by the nett thickness/chord. To make full allowance for this in a practical case seems unduly optimistic. Assuming a mean effective thickness/chord of 8%, an analysis of rocket and transonic-bump tests suggests a drag coefficient for this aircraft at zero lift of about 0.027 at $M \simeq 1.1$ and 0.020 at $M \simeq 2$. The drag rise will start at about M = 0.95. The drag due to lift, will of course, be high. Assuming a subsonic induced drag factor of 1.5 and for supersonic flow, $C_D = C_{D_0} + \alpha C_L$, gives the following rough drag summary.

$$M < 0.95:- \quad C_D = 0.007 + 0.48\,C_L^2$$

$$M \simeq 1.1:- \quad C_D = 0.027 + 0.6\,C_L^2$$

$$M \simeq 2.0:- \quad C_D = 0.020 + 0.6\,C_L^2$$

In the following table, values of Drag/$\frac{1}{2}\rho V^2$ for the present design are compared with those for the conventional design of Ref.1 for various altitudes and a weight of 25,000 lb. Values for the aircraft of Ref.1 are shown in brackets.

Mach number	0.9	1.1	1.5	2.0	2.5
Drag/$\frac{1}{2}\rho V^2$ (S.L.)	3.9 (5.5)	13.5 (13.5)	11.5 (10.0)	10.0 (8.5)	9.5 (8.0)
" " (36,000 ft)	12.0 (9.0)	18.5 (16.0)	13.0 (11.0)	10.5 (9.0)	9.5 (8.0)
" " (45,000 ft)	28.5 (14.0)	24.5 (19.5)	15.0 (12.5)	11.0 (10.0)	9.5 (8.5)
" " (60,000 ft)	-	60 (40)	26.0 (21.0)	14.5 (13.5)	11.5 (11.0)

Thus within the limits of accuracy of this rough analysis, the drag characteristics of the two designs are similar except at extreme altitudes and comparatively low speeds, under which conditions the drag of Project 'Y' is considerably greater owing to its low aspect ratio.

Using thrust, drag and component weight values given above, the following performance has been estimated for this project, without reheat.

Maximum rate of climb at sea level	60,000 feet/min
Time to 45,000 feet at $M \simeq 0.9$	1.5 minutes
Top speed Mach number	2.25
Ceiling (obtained at $M \simeq 1.75$)	60,000 feet
Cruising endurance at 45,000 feet, $M = 0.9$	20 minutes
Maximum 'g' for sustained turn without height loss, $M = 1.75$, 45,000 feet	2.5
Maximum 'g' for sustained turn without height loss, $M = 0.9$, 45,000 feet	1.9

Cruising endurance allows for take-off, climb to 45,000 feet and 10 minutes at maximum r.p.m. at the same altitude. The corresponding endurance of the aircraft described in Ref.1 is 30 minutes, carrying a greater military load. At the particular altitude and cruising weights assumed, both aircraft are cruising near optimum conditions and the lower endurance of Project Y is mainly due to the higher specific fuel consumption of the radial flow engine which has a low compression ratio. An increase in cruising weight or altitude would reduce the endurance of Project Y more than that of the more conventional aircraft on account of its low aspect ratio. Thus, if the cruise is carried out before instead of after the 10 minute period at maximum r.p.m., the greater fuel weight carried reduces the endurance by 30%. The design considered here carries very little disposable load so that a larger, military version, equipped for reheat would have a higher wing loading and/or a lower percentage fuel weight. Either of these changes, and of course the use of reheat itself will reduce the endurance still further. To offset the drawback of low endurance and poor manoeuvrability at subsonic speeds, the project will have a high top speed, a high rate of climb and the ability to maintain a comparatively high sustained 'g' at supersonic speeds. The latter quality is a result of the high thrust and low wing loading. On the assumptions outlined above in the drag estimates, the low aspect ratio has little adverse effect at supersonic speeds; these assumptions are subject to doubt in the absence of experimental data on this type of wing and intake.

4 Stability and Control

4.1 Introduction

We have made an attempt to assess some of the stability and control problems which will occur on this type of aeroplane. Some numerical examples are included to show the order of magnitude which these problems will have. We must emphasise however, that the unusual shape of this aeroplane and the flow disturbance which must result from the engine exhaust issuing round a large amount of its perimeter make reliable estimation of its stability and control properties impossible. The numerical results quoted must be treated as illustrative only. Much experimental work would need to be done before a reliable assessment could be made.

4.2 Stability

This aircraft as proposed has a negative manoeuvre margin of the order of -0.2 at subsonic speeds and its inventors claim that this aerodynamic instability will be counteracted by the gyroscopic effect of the rotor. This stabilizing effect acts by coupling the rolling and pitching motions so that a complete analysis of the stability would be a very laborious task and could not be completed in the time available for this survey. In any case it would be of doubtful value because of the uncertainty of the aerodynamic assumptions we should have to make.

We have, however, considered the effect of the rotor inertia on the stability assuming that the aircraft is free to move in roll, in pitch and with a vertical motion only. This assumption allows us to calculate approximately the most rapid of the longitudinal and lateral motions and, since it can be shown that the gyroscopic effect of the rotor would in the absence of aerodynamic forces lead to a fast undamped oscillation in pitch and roll, we feel that this will give a first approximation to its effect on the stability of the aircraft.

As the angular momentum of the rotor is increased from zero the damping of the root which represents the rolling subsidence decreases and the damping of the (initially unstable) longitudinal motion increases. These two motions coalesce to form an unstable oscillation which with further increase of the angular momentum becomes stable. The angular momentum of the rotor in this design should be sufficient to make the aircraft stable at all subsonic speeds although there is no great margin of stability and if the rotor inertia were halved instability would occur at high subsonic speeds. The aircraft will be stable at supersonic speed because of the rearward movement of aerodynamic centre.

We say then that the aircraft is likely to be dynamically stable at all speeds but that there is some uncertainty about it because the margin of stability is not large and the aerodynamic assumptions we have made are questionable.

4.3 Control

In order to execute manoeuvres with this aeroplane it will be necessary to apply not only moment to overcome the air resistance as in a conventional type of aeroplane but also moments to give the right rate of precession to the rotor. To precess the rotor in pitch we need a rolling moment and to precess it in roll we need a pitching moment. In a pitching manoeuvre, therefore we shall need to apply both elevator and aileron, and similarly in a rolling manoeuvre. The combinations of control that are necessary will vary with speed and the problem of coordinating the control movements may be serious.

The control on this design is rather unusual. Two-fifths of the flow through the engine exhausts between the two sides of the control so that in addition to the aerodynamic effect of deflecting the control there is also a moment due to the deflection of part of the jet thrust. Very rough estimates made of the effectiveness of the control indicate that at supersonic speeds the aerodynamic effect of the control predominates but at 300 ft/sec at sea level the contributors of the thrust and aerodynamic components are about equal.

We have estimated the control deflections necessary to produce a rate of roll of 10 degrees/sec and a normal acceleration of 1g. The damping in roll of this shape of wing is small and the <u>aileron</u> deflection for 10 degrees/sec rate of roll is always less than one degree. Large elevator movements are however necessary, particularly at low speeds. At an indicated airspeed of 300 ft/sec 14 degrees of elevator are required at sea level and 21 degrees at the tropopause. At a Mach number of 0.75 the corresponding figures are 5 degrees at sea level and 14 degrees at the tropopause.

The control angles to produce an extra acceleration of 1g are both large at low speeds. They are of the order of 15 to 20 degrees at an indicated airspeed of 300 ft/sec ($C_L = 0.5$). At higher indicated air-speeds they fall quite rapidly so that at an indicated airspeed of 840 ft/sec at sea level they are of the order 3 to 5 degrees. We must remember that in addition to these elevator angles required for manoeuvring similar elevator angles will be required for trim in straight flight.

The rolling performance will thus be poor at all speeds and in any case can never exceed 1 radian per second which is the rate of roll for which the rotor is stressed. The longitudinal manoeuvrability will be moderate.

5 Conclusions

Any conclusions drawn from the above considerations must be treated as purely tentative. Much experimental work would need to be done to establish the aerodynamic, structural and mechanical properties of the aircraft and power plant before it is established with any certainty whether it can be made acceptable from the performance, stability and control points of view. Drag estimates made in this note are subject to considerable doubt by virtue of the unusual layout and mutual interference between the wing and the jet stream. The indications from our present rough assessment are:-

(1) The aircraft will have a thrust/weight ratio of about 1.3 at sea level, without reheat, and vertical take-off and landing should be possible.

(2) The rate of climb at sea level will be about 60,000 ft/minute and the time to 45,000 feet about 1½ minutes.

(3) The estimated top speed Mach number without using reheat is about 2.25. This speed is theoretically obtainable with a more conventional aircraft layout, using existing engines with reheat.

(4) Owing chiefly to the high specific fuel consumption of the engine, endurance will be poor, even if reheat is not used.

(5) It seems likely that the longitudinal rapid motions will be stabilised by the gyroscopic effect of the rotor. We have been unable to determine the effect of the rotor on the lateral oscillation.

(6) The fact that a combination of lateral and longitudinal controls is necessary to produce a pure longitudinal or pure lateral manoeuvre and that the combinations required vary with speed and height and may make coordination of the control difficult.

(7) The rate of roll will be poor at all speeds and worst at low speeds.

(8) The low wing loading and high thrust will give comparatively good longitudinal manoeuvrability at supersonic speeds in the stratosphere. At subsonic speeds the rapid increase of drag with lift at high altitude and the large control angles required suggest that the longitudinal manoeuvrability will be poor.

REFERENCE

No.	Author	Title, etc.
1	C.H.E. Warren J. Poole and D.C. Appleyard	An investigation into an aircraft to fly at a Mach number of 2. RAE Report No. Aero 2462, June, 1952.

Attached:

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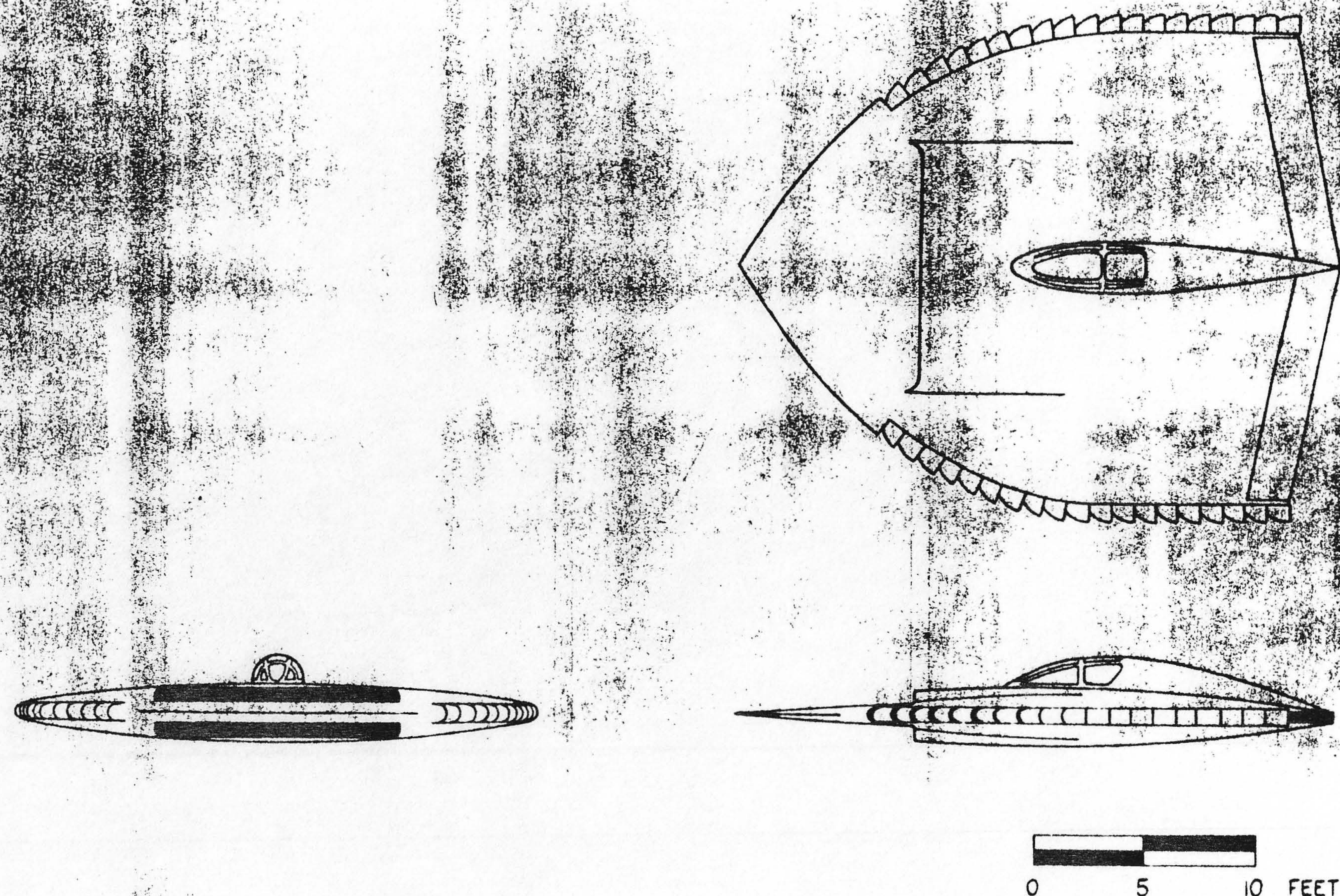

FIG.1. GENERAL ARRANGEMENT OF PROPOSED DESIGN.