ISN 13200

A STUDY OF PROPOSALS TO IMPROVE THE
AERODYNAMIC CHARACTERISTICS OF THE
A.V. ROE CANADA LIMITED
C-104 SUPERSONIC FIGHTER PROJECTS

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A.V. ROE CANADA LIMITED.
OCTOBER 25th, 1952.

CONTENTS

SECTION 1 INTRODUCTION

UNCLASSIFIED

SUMMARY

Two Brochures have been submitted by A. V. Roe Canada Limited to the R.C.A.F. describing a single and a twin engine aircraft which have been designed to meet a draft specification proposed by the R.C.A.F. Although, both these aircraft meet this specification in most respects and in some cases exceed it by a handsome margin, they are unable to maintain level flight at a Mach number of 1.5 at 50,000 feet, while executing a 2g turn as required. Since great weight has been attached to this requirement, a detailed discussion of some of the factors involved is given in this report.

DISCUSSION

There are two main lines on which this requirement may be discussed:

(1) If the requirement is based on the operational necessity of the fighter changing its direction in a minimum of time and space so as to put it into a favourable position for attack, then alternative ways of doing this may be considered.

(2) If the requirement is essentially a manoeuvrability criterion for which no rational description of the manoeuvres involved is possible, then the discussion must take the form of a consideration of ways of reducing the drag or of increasing the engine thrusts under these conditions.

In Section (2) of this report, an example is given showing how the aircraft described in the original Brochure can turn after a first pass at the target and make a second pass in very close to the same time as could be achieved by making a flat 2 g turn, were that possible, by using a high g sinking turn and then climbing back up again. This data is presented in accordance with the philosophy of paragraph (1) above. The high values of the normal acceleration, that are permitted by the aerodynamic limitations, as opposed to the power limitations are exploited to give a favourable result provided an optimum procedure is used. It is believed that by similarly utilizing the potential of the aircraft, optimum ways of achieving other turning manoeuvres of operational significance could be devised which would be equivalent to or better than a flat turn.

These optimum procedures can be set up as instructions in the memory of the flight path computer which is the heart of the MX1179 system.

In accordance with the point of view set forth in para. 2, following Section 1 of this report, are several sections which deal with the drag of the aircraft and proposals which have been studied for reducing this drag. These proposals take two forms.

(a) The reduction of the elevator drag by the use of negative wing camber.

(b) The reduction of the drag at relatively high design lift coefficients at the expense of an increase in the drag at lower lift coefficients by the use of twist and camber of the wing.

Elevator drag is not usually included in performance work. Because of this, it was felt appropriate to devote Section 3 of this report to a discussion of the method that has been evolved for estimating the elevator drag at supersonic speeds. It has been found that this drag is very high at high altitudes, as can be seen from the figures given in Table I of Section 6.

The large drags are due to the high elevator angles required, and can be minimized under a reasonable range of conditions by applying a negative C_{M_0} to the wing such that the high elevator angles are reduced. This is discussed in detail in Section 4.

In Section 6 are given some figures relating to the potential improvements in drag and performance that might be achieved. Although these are substantial, they do not approach that required to meet the specified turning performance at 50,000 ft. As discussed in Section 4, there is a very low probability of obtaining suitable values of C_{M_0} throughout the Mach range. It is, however, conceivable that this favourable result could be achieved by a very extensive, lengthy, and expensive testing program, which would delay the finalizing of the configuration from one to two years. Even then there is a chance that the required results might not be achieved full scale, and the aircraft would be prohibitively penalized under certain conditions as discussed in Section 4. It then appears that the risk involved in attempting to utilize negative camber is far too great for anything but a research project.

This applies even more forcibly to the case which has been considered of camber and twist, so combined as to give both a reduction in elevator drag and induced drag. It is not even certain that it is theoretically possible to superimpose these effects, let alone achieve this result in practice. Accordingly, the figures given in Table I of Section 6 for this case should be viewed as a theoretical target only.

The N.A.C.A. have shown experimentally that the drag of highly-swept wings can be reduced at relatively high design lift coefficients at the expense of increased drag at lower lift coefficients. The evidence on this is discussed in a general way in Section 5, and the specific effects on the peformance are given in Section 6. The data of these Sections leads to the conclusion that -

(1) There is an element of uncertainty about the mechanism of the improvements recorded, and that further work at higher Reynolds number would be necessary to define what the full scale performance would be.

(2) The possible improvement in drag at 2g at 50,000 ft. is relatively small, and there would not be any change in the number of g that can be sustained in a level turn with the present power. The performance in other respects undergoes a considerable deterioration.

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Since it appeared in the initial phases of the investigations of supersonic fighters, that the use of camber and twist offered considerable potential advantages, a visit was made to Convair in March, 1952, to discuss these and other similar problems. Since these discussions lasted three days, it was possible to consider these matters in considerable detail with the various individuals concerned, all of whom were most co-operative.

It was concluded from these discussions that -

(1) Convair had not delved into the subject of camber and twist in very great detail.

(2) They had reached virtually the same conclusions that we had obtained, namely, -

(a) That camber and twist do not offer sufficient advantage to warrant their use.

(b) That the application of a negative C_{M_O} was desirable, but involved much too much risk of producing an airplane of completely unsatisfactory characteristics to be entertained. Accordingly, Convair went to considerable lengths to minimize C_{M_O}, so that unacceptable transonic variations would be improbable.

SECTION 2

TURNING PERFORMANCE

It is the object of this section to examine the turning performance of the C-104/2 in relationship to the requirement that it be able to make a 2g level turn at 50,000 ft.

The turning performance of the aircraft may be limited by any one of the following:-

(1) High speed buffet,

(2) Elevator effectiveness,

(3) Elevator hinge movement,

(4) Strength,

(5) Available engine thrust.

It can be seen from Fig. 19 of the C-104/2 Brochure that a level turn is limited to 1.35g at a Mach number of 1.5 at 50,000 ft. by the available thrust, but there are no aerodynamic or other limitations below 3.7g. Hence, it is evident that it is possible to make a much tighter sinking turn than a level turn. It then becomes appropriate to compare the relative merits of these two types of turns in an assumed tactical case, to see whether the aircraft can actually come near the result obtained by a 2g level turn, by using a more favourable procedure.

To evaluate the turning performance of the aircraft, three examples of a typical interception have been prepared. In each example, the fighter is assumed to intercept an enemy bomber cruising at 500 knots at 50,000 ft. The fighter makes the first attack at a Mach number of 1.5 at 120° to the bomber flight path, after which it turns without loss of speed and makes a second attack.

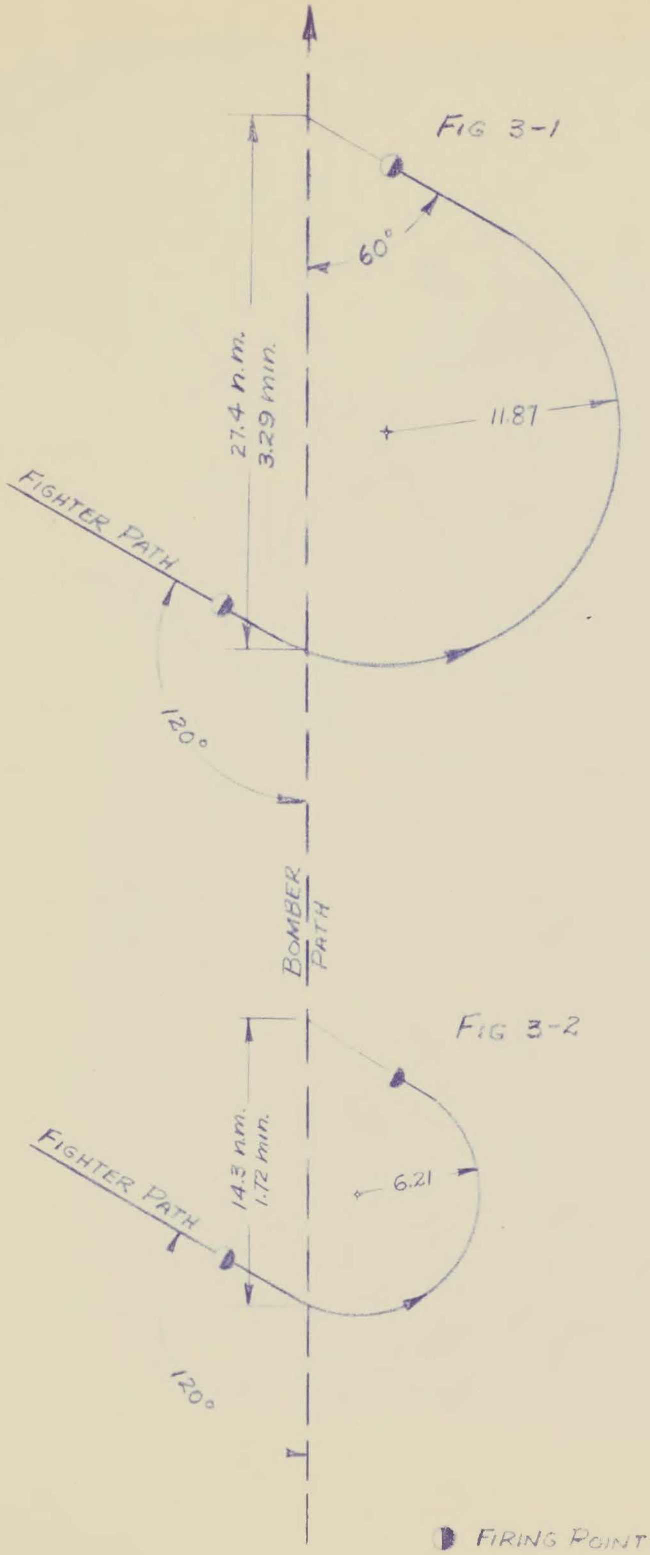
In Example 1, the fighter executes a level turn at a constant Mach number of 1.5, using maximum available power. Example 2 has been prepared for comparison. It assumes that there is sufficient thrust available to permit a 2g level turn at a constant Mach number of 1.5. In Example 3, the fighter executes a sinking turn at a constant Mach number of 1.5 and climbs back up again for the second attack.

The data given on figures 3-1, 3-2, and 3-3, may be summarized as follows:-

Example	Conditions	Distance Travelled by Bomber between First and Second Interceptions
1.	Level turn-max. thrust as per Brochure.	27.2 n.mi.
2.	Sinking turn-max. thrust as per Brochure.	17.9 n.mi.
3.	Level 2g turn-thrust required assumed available.	14.3 n.mi.

It can be seen that with the design described in the Brochure, a second interception can be made only 3.6 nautical miles farther from the first interception than could be achieved by using the 2g level turn called for in the Specification.

Great pains have not been gone to in selecting the optimum sinking turn. It is probable that the difference could be further narrowed by a more refined procedure. Other tactical problems could be dealt with in a similar way. Since the flight path is governed by a computer, the use of precalculated optimum procedures is quite practical.



<u>EXAMPLE 1</u> - 180° Level Turn, with maximum available power.

M = 1.5

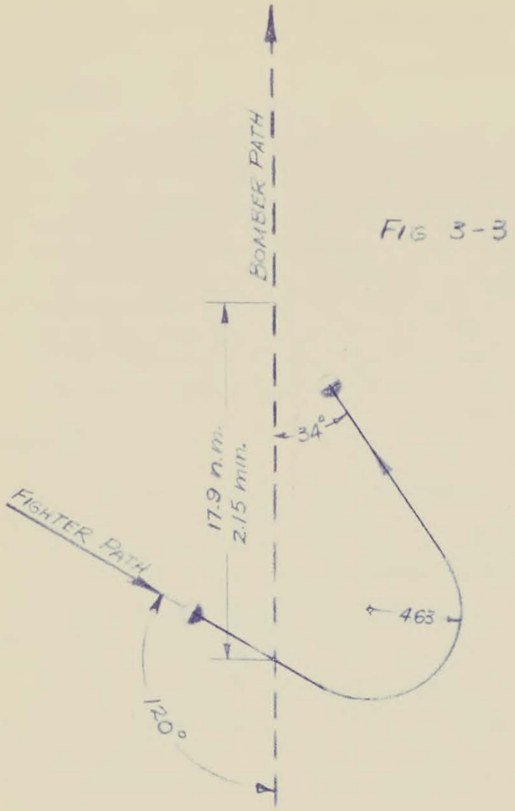
R = 11.87 n.mi.

n = 1.35g

<u>EXAMPLE 2</u> - 180° Level Turn at 2g.

M = 1.5

R = 6.21



EXAMPLE 3 - Sinking Turn

M = 1.5

R = 4.63 n.mi.

n = 2.5g

SECTION 3

ESTIMATION OF ELEVATOR DRAG

INTRODUCTION

It is evident from the results of recent experimental investigations, that the drag of a deflected elevator on a delta wing is considerable. In the case of a supersonic aircraft manoeuvring at high altitude it appears that the drag due to the deflected elevator may be of the order of the drag of the rest of the aircraft. This makes the estimation of elevator drag a matter of importance. The best experimental data available apply to moderate elevator angles and to one particular elevator geometry. These data are used below to predict the drag of other elevator geometries that are of interest, at angles up to maximum deflection. In particular the effect of a part-span inboard elevator is investigated.

DISCUSSION

The most applicable experimental results were those from rocket tests on the Convair Delta reported in Ref. (1). Fig. 18 of this report, presents the drag data in the most convenient form, but subject to two assumptions: (a) that $\Delta C_{D_{MIN}}$ varies as δ^2, and (b) that $\Delta\ (dC_D/dC_L 2)$ is proportional to δ. These assumptions were checked using the subsonic data of Refs. (2) - (5).

It was found that (a) was reasonably true. $\Delta C_{D_{MIN}}$ varied as a power of δ between about 1.5 and 2. Since supersonic drag of an aerofoil varies as $(\text{thickness})^2$, and flap deflection may be looked on as a kind of aerofoil thickness, it was accepted that (a) is valid.

Assumption (b) was not found to hold very well in the cases investigated. Reference (2) which is again the Convair model indicated good linearity in the range $-20^\circ \leqslant \delta < 20^\circ$. Refs. (3) and (4), which were for delta wings, were quite non linear, Ref. (4) in fact being very irregular. Ref. (5) again showed approximate linearity. Since Ref. (3) applies to a circular arc section and Ref. (4) to a double wedge, while the others are orthodox sections, it was decided not to weight these results. Further, since the increment in drag represented by $\Delta\ (dC_D/dC_L^2)$ is smaller than the $\Delta C_{D_{MIN}}$ term, it was decided that in the absence of better information, $\Delta (dC_D/dC_L 2)$ may be taken as linear with δ.

TO FACTOR FIGURE 18 (A) REFERENCE (1) $\left(\dfrac{\Delta C_{D_{MIN}}}{\Delta \delta^2} \right)$

The assumption made is that models of different flap geometries but the same wing area, will, for the same 'q' and flap angle, have drag increments (ΔD_{MIN}) proportional to the flap areas.

This gives: $(\Delta C_{D_{MIN}})$ Test Model $\times S_{T.M.} = \frac{\Delta D_{T.M.}}{q}$

$$(\Delta C_{D_{MIN}})\ C104 \times S\ C104 = \left[\frac{\Delta D_{T.M.}}{q}\right] \times \frac{S_{eC104}}{S_{eT.M.}}$$

$$(\Delta C_{D_{MIN}})\ C104 = (\Delta C_{D_{MIN}})\ T.M. \left[\frac{(\frac{S_e}{S})\ C104}{(\frac{S_e}{S})\ T.M.}\right]$$

A check was made on this assumption as follows:

Diehl Page 150 gives values of $\Delta C_{D_{MIN}} \times \delta$ for various $\frac{Cf}{C}$ ratios. Mean values of $\frac{C_f}{C}$ were calculated for the test model and for the C104, using the value on the chord passing through the centroid of the exposed wing area (i.e. that area not covered by fuselage). The ratio:-

$\frac{(\Delta C_{D_{MIN}})\ C104}{(\Delta C_{D\ MIN})\ T.M.}$ for a given elevator deflection was taken as an alternative factor. It was found to agree reasonably with the factor calculated by the other method.

TO FACTOR FIGURE 18 (C) REFERENCE (1) $\left[\frac{\Delta\ dC_D/dC_L^2}{\Delta\ \delta}\right]$

In the absence of any information to the contrary, it was assumed that this quantity varies with flap span only. See below.

PART SPAN EFFECT

The only drag data on full and part-span inboard flaps on a 60° delta wing are those of Ref. (3) figs. 6 and 7. The value of the results is reduced by the fact that it is necessary to read differences in small quantities. Full span and part-span flaps were compared at the same C_L's and flap angles such as to give approximately the same pitching moment. It was concluded that the main part of the difference in drag is attributable to profile drag, and that this can be allowed for by the same factor as for full-span flaps $\left[(\frac{S_e}{S})_{C104} / (\frac{S_e}{S})_{\text{Test Model}}\right]$

EFFECT OF PART-SPAN FLAP DEFLECTION ON $\left(\frac{dC_D}{dC_L^2}\right)$

Theoretically the span loading on 60° delta wing is close to elliptical. That due to deflecting a full span, constant percent chord flap is therefore close to elliptical also. The loading due to deflecting a part-span inboard flap will be far from elliptical and it is required to estimate the amount of change in (dC_D/dC_L^2) to be expected in this case on theoretical grounds.

Consider a wing $\Lambda_{\frac{c}{4}} = 60^{\circ}$, $\lambda = 0$, $AR = 2.0$ with a flap $\frac{c_f}{c} = .20$ and $\eta_{f_i} = 0$, $\eta_{f_o} = .556$. Loading coefficients were found from Ref. (6) fig. 4b. C_L and C_{D_i} were calculated from Ref. (7). Hence "e" was calculated from its definition: $C_{D_i} = \frac{C_L{}^2}{\pi AR.e}$.

The shape of the span loading curve depends on the ratio $\frac{\alpha}{\delta}$. The value of "e" and $\frac{dC_D}{dC_L{}^2}$ were calculated for $0 \leq \frac{\alpha}{\delta} \leq \infty$ and graphs prepared. These showed that the theoretical $\left[\frac{\Delta\left(\frac{dC_D}{dC_L{}^2}\right)}{\Delta\delta}\right]$ is small compared to the values obtained experimentally. The effect is therefore mainly viscous. In a practical drag calculation on a part-span inboard flap the values for the above flap are used. This is valid since the effect is small (about 3% of the total drag in a typical level flight case).

TO FACTOR FIGURE 18 (C) REFERENCE (1) (PART-SPAN FLAPS)

The data of Ref. (3) do not give a decisive answer to the question of how to factor the test results on $\left[\frac{\Delta\left(\frac{dC_D}{dC_L{}^2}\right)}{\Delta\delta}\right]$ to apply to a part-span flap. No other experimental data of any value are known. It is known from the work above that this is not a true induced drag, but a viscous effect. Since, supersonically, there is no effect ahead of the hinge line, a factor based on the relative flap sizes of the C104 and the test model is suggested by strip theory. This is the same factor used before:

$$\left[\frac{\left(\frac{S_e}{S}\right)_{C104}}{\left(\frac{S_e}{S}\right)_{T.M.}}\right]$$

The drag of the aeroplane with elevators deflected is to be calculated by:

$$C_D = C_{D_o} + \left(\frac{dC_D}{dC_L^2}\right)_{\delta=0} . C_L^2 + \left(\frac{\Delta C_D}{\Delta\delta^2}\right) . \delta^2 . F + \frac{\Delta}{\Delta\delta}\left(\frac{dC_D}{dC_L^2}\right) . \delta . C_L^2 . F + \Delta\frac{dC_D}{dC_L^2} . C_L^2$$

$$\quad (1) \qquad\qquad (2) \qquad\qquad\qquad (3) \qquad\qquad\qquad (4) \qquad\qquad\qquad (5)$$

(1) and (2) are the normal drag terms for the aeroplane ($\delta = 0$)

(3): $\left(\frac{\Delta C_D}{\Delta\delta^2}\right)$ is read from Reference (1) figure 18(a).

δ^2 is read from δ trim and δ/g curves for the aeroplane.

$$F = \left[\left(\frac{S_e}{S}\right)_{C104} \Big/ \left(\frac{S_e}{S}\right)_{TM}\right]$$

(4): $\frac{\Delta}{\Delta\delta}\left(\frac{dC_D}{dC_L^2}\right)$ is read from Reference (1) figure 18(c).

(5): This term is to cover the true induced drag due to a part span inboard flap. The value of α/δ is first calculated and the value of $\Delta\left(\frac{dC_D}{dC_L^2}\right)$ is read from the curve F

ALTERNATIVE METHOD

In order to provide a check on the drag values calculated by the method of Ref. (1). Another method of analysis was attempted, using data of Ref. (2) which apply to the same aircraft.

The quantity $\left[\dfrac{C_D}{C_D = 0} \right]$ was calculated for fixed C_L and varying δ, and plotted versus δ. For up-elevator these curves for various C_L's coincided fairly well; that for $C_L = 0.25$ was taken as applying to all C_L's.

This curve, obtained from subsonic data, was now applied to supersonic drag estimation for the same aircraft. The results could be checked at $= -9^\circ$ by comparison with the data of Ref. (1). It was found that the agreement was very good. The curve was therefore used to estimate the drag of the C 104 at supersonic speeds.

The drag of the C 104 was calculated by both methods for two supersonic flight conditions using full elevator. The method of Ref. (2) gave higher drags on both cases, about 5% and 10% of the Ref. (1) value. Since these are cases where total drag is about twice the clean A/C drag, this comparison is taken to be a justification of the method of Ref. (1).

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ESTIMATION OF ELEVATOR DRAG : DATA FROM RM L50G17

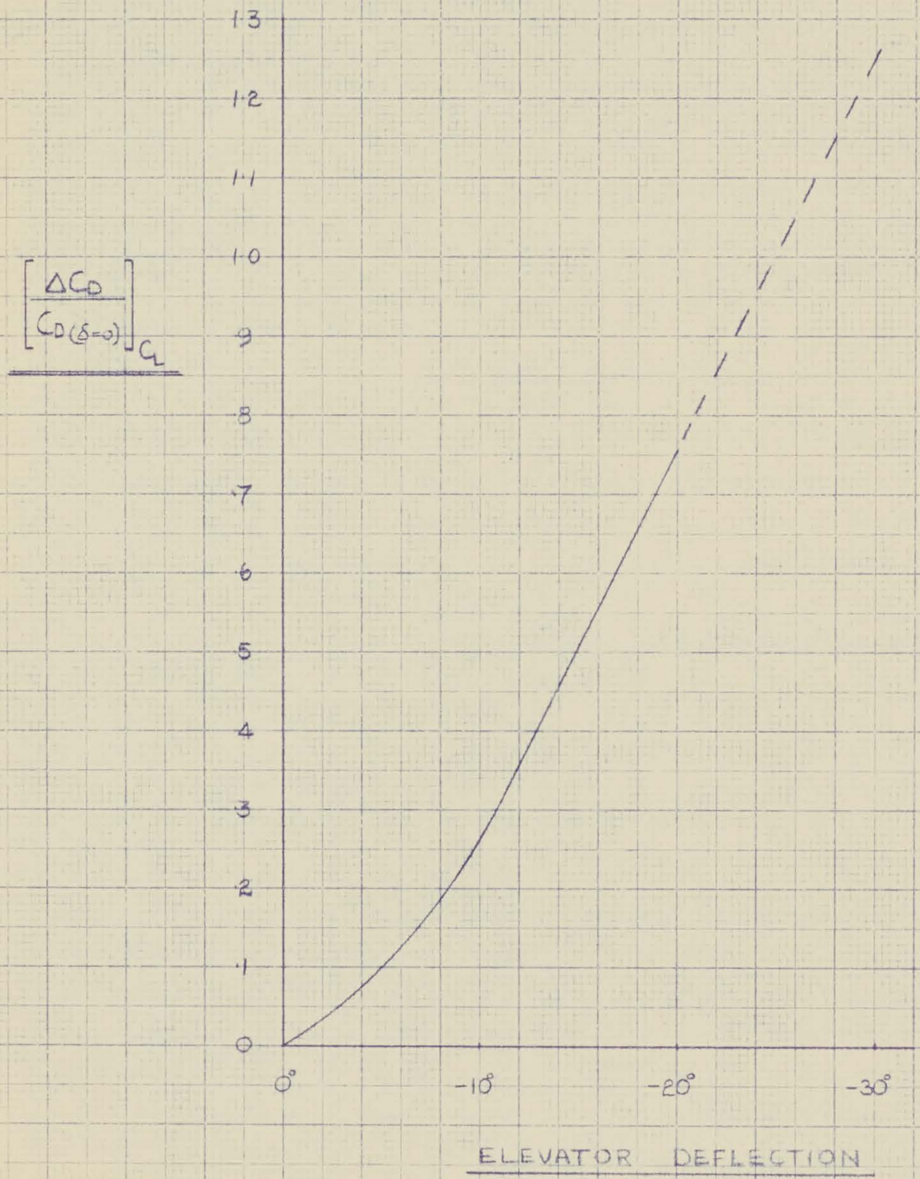

FIG. 3.3

SECTION 4

THE USE OF CAMBER FOR THE REDUCTION OF CONTROL SURFACE DRAG

One of the inherent difficulties associated with the control and manoeuvrability of supersonic aircraft is a large variation, over the speed range, in control deflection required for trim and manoeuvre. This is due to several factors, most important of which are:

(1) Movement of aerodynamic centre from about 35% MAC at subsonic speeds to 50% at supersonic speeds.
(2) Deterioration of control effectiveness at transonic and supersonic speeds.
(3) Short control arm of swept wing aircraft as compared to orthodox, straight wing, subsonic aircraft.
(4) High operational altitudes and consequently low dynamic pressure.

Some measure of positive static stability must be provided at low speed (landing and take-off) and in consequence, due to (1) very large positive static margins will be obtained at transonic and supersonic speeds. For balancing of such high nose down pitching moments large control deflections are required. The situation is further aggravated by (2), (3) and (4).

The objections to large control surfaces deflections are:

(1) Drag increase and consequent deterioration of performance of the aircraft.
(2) Critical limitation of manoeuvrability at transonic speeds and high altitude.

As can be seen from figure 4.1, large control deflections are required at that condition for both trim and manoeuvre.

Efficiency of controls falls off rapidly for deflections over 25° and simultaneously control hinge moments and drag increase out of all proportions. Therefore, total deflection must be limited to some reasonable figure such as say 25°. If the controls are of the elevon type some 10° will be required to maintain rolling performance and thus there is only 15° left for longitudinal control. A glance at figure 4.1, will show that with these limitations there is very little control left for manoeuvring.

In view of the above situation, it is only natural that some novel means of improvement have been investigated. One of these is the use of a negatively cambered wing. The amount of camber built into the wing can be so chosen as to completely trim the aircraft with controls neutral at its most likely operational speed and height. To illustrate the present discussion such a design point was taken at an altitude of 50,000 feet and Mach Number 1.4.

This would be an ideal solution to the problem if the C_{M_o} built into the wing did not vary too much over the Mach range and if it did so in a predictable manner. A large amount of available experimental evidence was investigated to test this. Results of this investigation are presented in figure 4.2. Details pertaining to the geometry of the models tested are given on sheet . All of these models had symmetrical wing sections and those tested with fuselages were symmetrically mounted. Theoretically, no C_{M_o} is expected for such configurations. Yet each one of them showed the existence of C_{M_o} and, what is worse, most of them showed quite irregular variation with Mach Number. The known evidence on conventional wings is that C_{M_o} becomes much more erratic in transonic range for cambered wings than for symmetrical wings. Hence, it may also be expected that on delta wings with camber, the variation of C_{M_o} will become much more irregular than the symmetrical cases investigated.

The theoretically expected variation of C_{M_o} is shown as curve A in figure 4.3. Although models tested were symmetrical yet some of them displayed a variation of C_{M_o} similar qualitatively to that expected from a C_{M_o} originated by a built-in camber. These were used to estimate the experimentally expected variation of C_{M_o} and are shown as curves B and C in figure 4.3. Curve C is still not the worst possible for no sudden erratic change of sign was assumed. That such a change can take place is evidenced by the curve marked with full line (RM L9L07) on figure 4.2. Trim curves were calculated using these different estimates (A, B and C) of C_{M_o} and are presented in figure 4.4.

The estimate A produces a very desirable trim curve at 50,000 ft. But the trim curve at sea level is inadmissible for the hinge moment required to keep the aircraft trimmed at sea level at M = 1.05 exceeds by such a large margin available maximum design hinge moment (54,000 lbs. ft) that the aircraft would be subjected to 5.7 g nose up acceleration, if flight was attempted under those conditions.

Trim curves resulting from C_{M_o} estimates B and C indicate still worse situation . Corresponding figures are given below-

Estimate of C_{M_o}	M	Out of balance acceleration
A	1.05	5.7
B	1.10	7.3
C	1.10	13.3

It is believed that these considerations rule out completely the possibility of application of negatively cambered wing as a design proposition.

REFERENCES

Report	A. R.	$\Lambda_{L.E.}$	t/c %	Section	Body	R × 10^{-6} Average	Remarks
RM A50K20	2.00	63.4°	8.0	0008-63	Yes	3.0	
L9L07	2.31	60.0°	6.5	65_{06}-006.5	Yes	15.0	Flight Test
A7K05	2.00	63.0	5.0	D.W. (20)	Yes	5.3	
A7K05	2.00	63.0	5.0	D.W. (20)	No	5.3	
A8E21	2.00	63.0°	4.8	D.W. (20)	No	3.2	
A8E03	2.00	63.4°	5.0	D.W. (20)	No	15.0	
A8E03	2.00	63.4°	5.0	D.W. (20)	Yes	15.0	
A50E10	2.00	63.4°	5.0	D.W. (20)	No	.92	
A8I16	2.00	63.4°	5.0	D.W. (20)	Yes	.85	
A8I20	2.00	63.4°	5.0	D.W. (50)	Yes	.85	
A50K21	2.00	63.4°	5.0	0005-63	Yes	3.0	
A50K24	4.00	45.0°	5.0	0005-63	Yes	1.5	
A50K24a	2.00	63.4°	3.0	0003-63	Yes	5.0	
RAE 2022	3.00	45.0°	7.5	RAE 101	No	.6	
RM A7J10	2.00	63.4°	5.0	D.W. (20)	Yes	.75	
A7K03	2.00	63.4°	5.0	D.W. (20)	No	1.8	
A8C03	2.00	63.4°	5.0	D.W. (20)	No	1.8	
L9E18	2.31	60.0°	6.5	Sym.	Yes	1.6	
L9F10	2.31	60.0°	-	Flat	No	3.0	
L9J07	2.31	60.0°	10.0	B.C.	No	.85	
L51E11	1.57	68.6°	2.0-3.7	C.A.	Yes	1.45	

D.W. = Double Wedge
C.A. = Circular Arc
B.C. = Biconvex

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C104 - SINGLE ENGINE

TRIM AND MANOEUVRABILITY CHARACTERISTICS VS M.N.

NO CAMBER

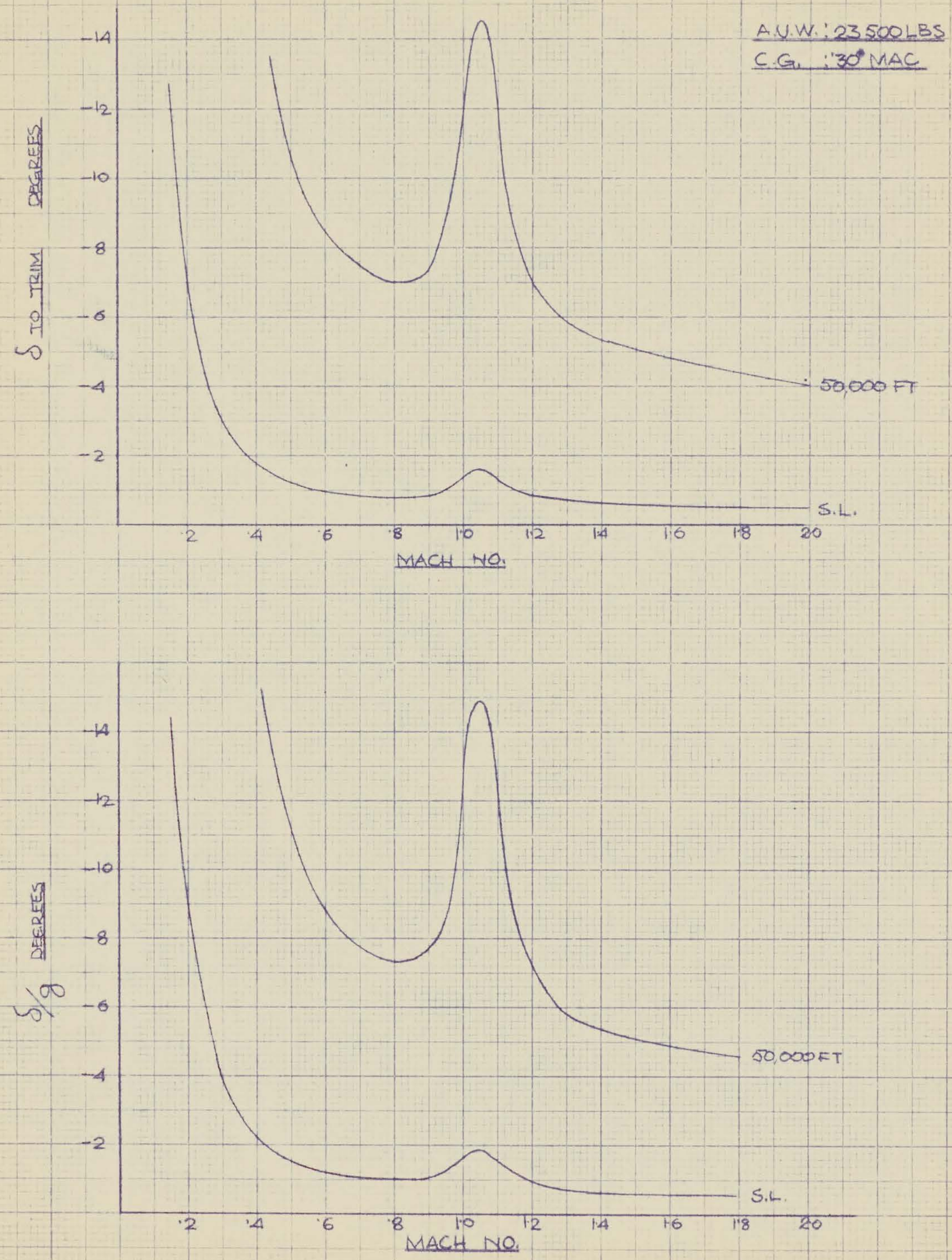

FIG. 4.1

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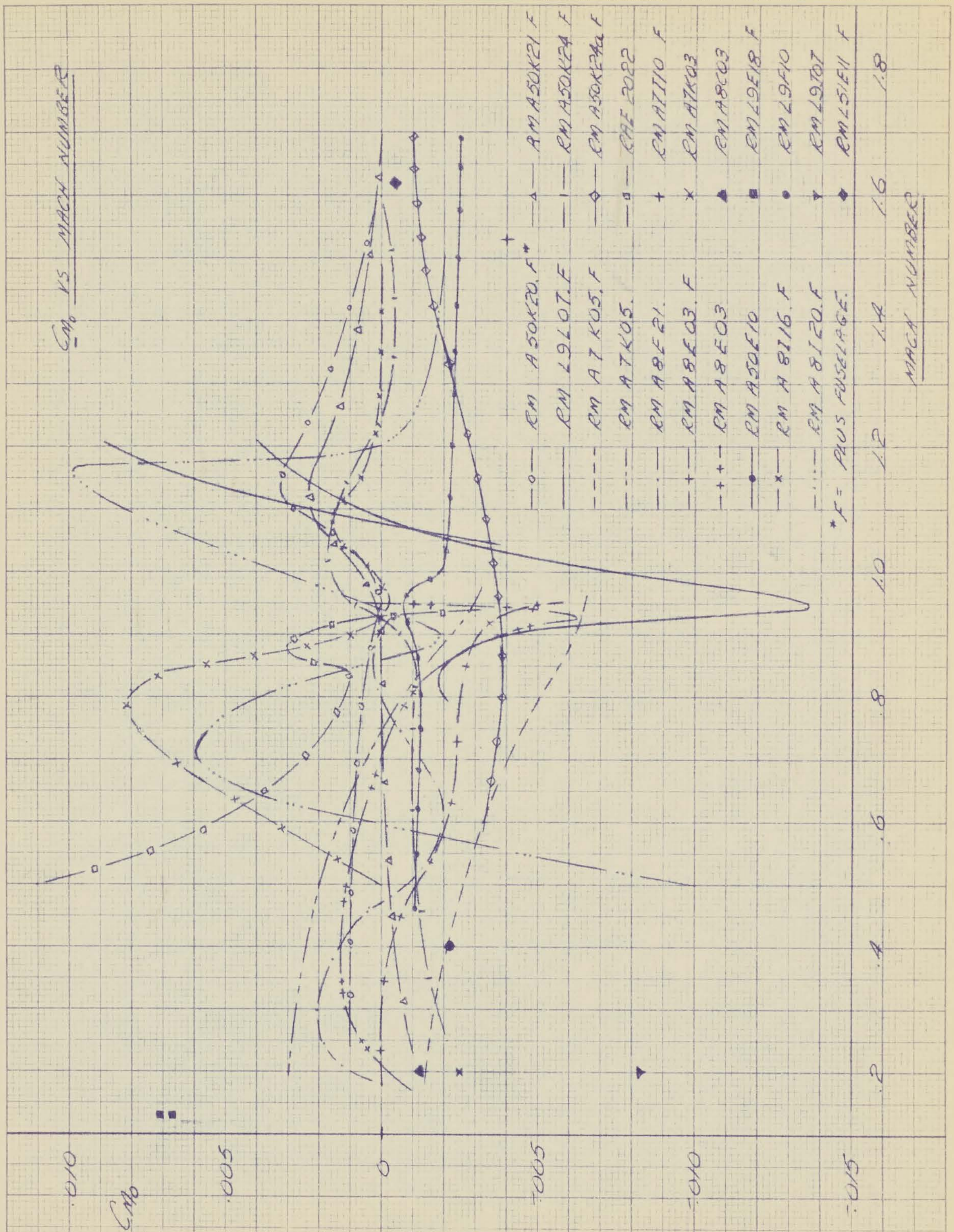

FIG. 4.2

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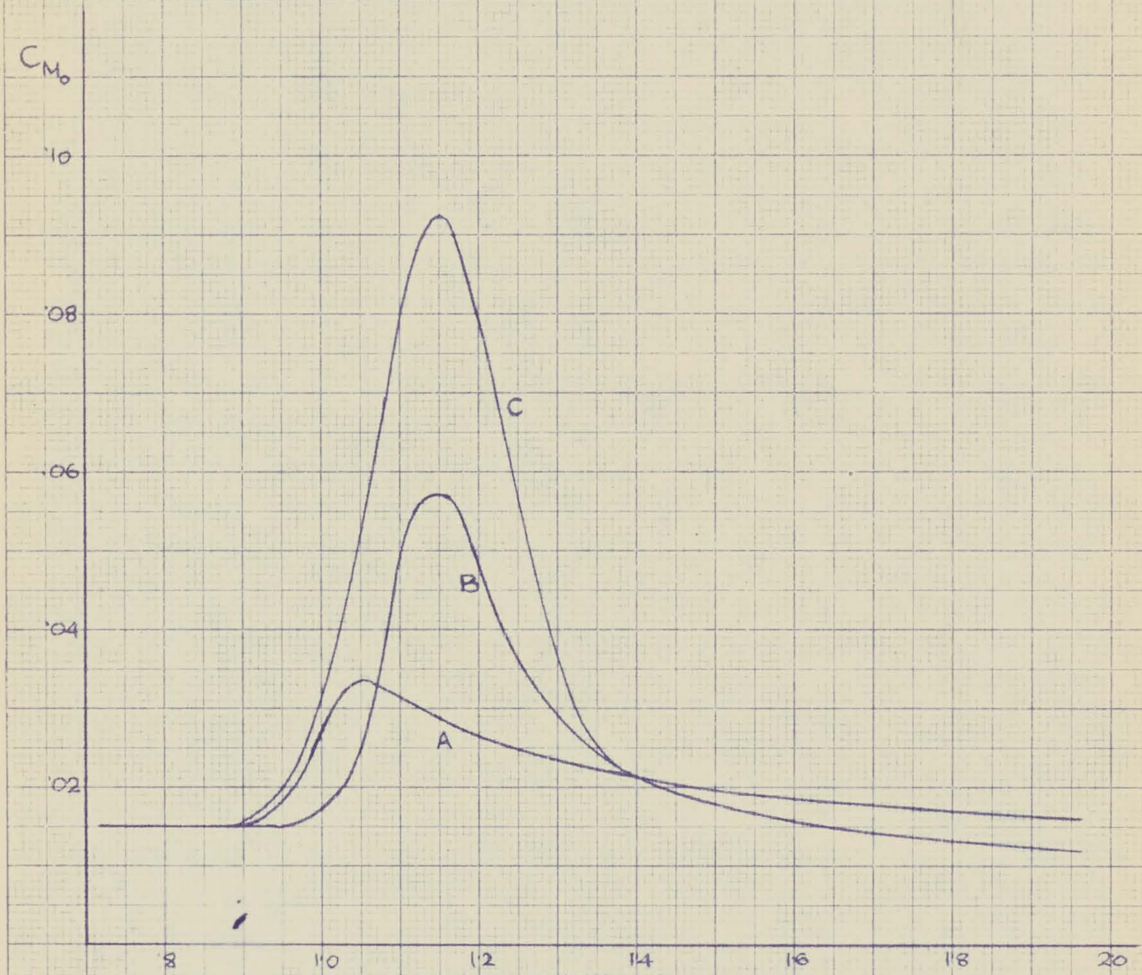
C104 - SINGLE ENGINE.

ESTIMATES OF C_{M_0} DUE TO CAMBER.



A THEORETICAL ESTIMATE

B MEAN EXPERIMENTAL ESTIMATE

C OUTSIDE EXPERIMENTAL ESTIMATE

FIG. 4.3

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C104 - SINGLE ENGINE

INFLUENCE OF ESTIMATED C_{M_0} VARIATION WITH MACH NO. ON TRIM CURVES

A.U.W.: 23 500 LBS
C.G.: ·30 MAC

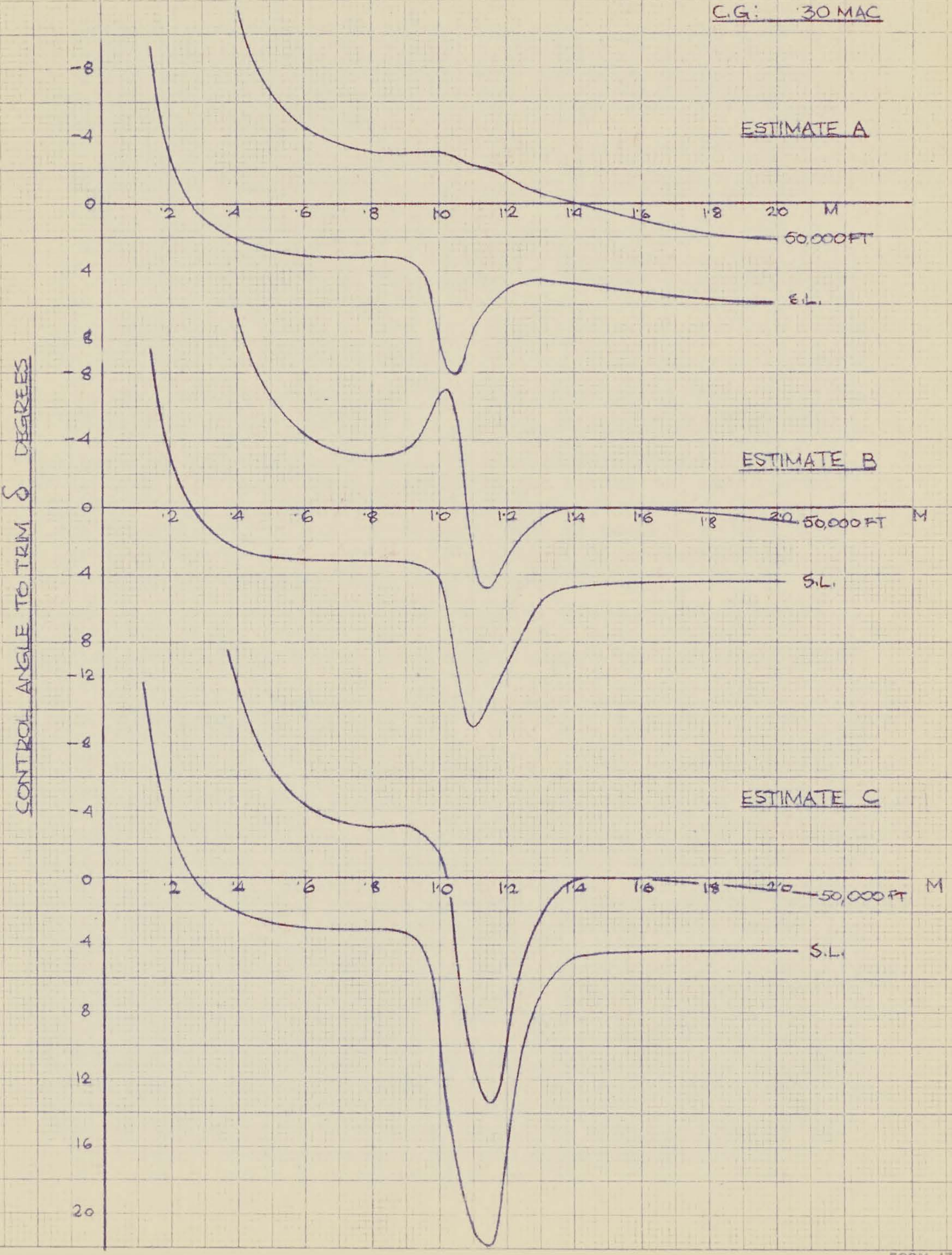

FIG. 4·4

THE USE OF CAMBER AND TWIST TO INCREASE THE RATIO OF LIFT TO DRAG

Experimental evidence shows that camber and twist can be used to increase the maximum L/D of highly swept wings at supersonic speeds (Ref. (4), (5), (6), (7), (8), (9)). In Section 6 of this study the changes in performance caused by such camber and twist applied to the present C104/2 wing are tabulated. In this Section, the following topics are dealt with briefly:

(a) Effect of Reynolds number.
(b) Effect of different airfoil thickness ratio, and different camber and twist.
(c) Theoretical reasons for using camber and twist, and their applications to the present C104/2.
(d) Changes which might enable the benefits of camber and twist to be realized more fully.

(a) The Effect of Reynolds number

The Reynolds number of the C104/2 based on the mean aerodynamic chord is at least 50×10^6 at M = 1.5 at 50,000 feet. The zero lift supersonic drag estimates for the C104/2 with a flat wing are based on flight measurements at Reynolds numbers of 50×10^6 at M= 1.5 (Ref. (10)) but the drag estimates for the twisted and cambered wing must be based on wind tunnel tests at Reynolds numbers of at most 7.5×10^6 (Ref. (8)). While these tests show no measurable effect of Reynolds numbers from 3.0×10^6 to 7.5×10^6, it is not certain that there is no scale effect between Reynolds numbers of 7.5×10^6 and 50×10^6. It is said that further reductions in drag may be expected at higher Reynolds numbers. This statement is, however, equally applicable to the drag of flat wings having rounded leading edges.

(b) The Effect of different airfoil thickness ratio, and different camber and twist.

In Figure 1, page 29, L/D has been plotted versus Mach number for several wings. A comparison of Figure 1(a) and 1(b) shows that the L/D of the flat 3% triangular wing is greater than the L/D of the best cambered 5% wing and that at lower supersonic Mach numbers, from 1.3 to 1.5, the increase in L/D which accompanies the camber and twist is greater for the 3% wing than it is for the 5% wing. This comparison certainly provides no reason for using a 5% thick delta wing rather than a 3% one. The 5% wings of Figure 1(c), have larger L/D's but they also have larger aspect ratios, so that no direct comparison with the delta wings can be made. Furthermore, it should be pointed out that the reason for the increased L/D of the cambered and twisted wing of Figure 1(c) is somewhat obscured by the fact that the two wings have different airfoil sections. The flat wing has an NACA 0010 section perpendicular to the leading edge, while the cambered and twisted wing has an NACA 64A005 a = 1.0, section streamwise. If the flat wing were untapered, its streamwise thickness ratio would be .045, but because of the taper it is .056. (The effect of taper on the relative values of thickness ratio streamwise and perpendicular to the leading edge is referred to in Ref. (11) and (12).) Thus part of the gain in L/D is due to decreasing the maximum thickness chord ratio from 0.056 to 0.050 and is not due to camber and twist.

Figure 1(a) shows that the L/D of the wing cambered and twisted for a trapezoidal load distribution has a slightly lower L/D than the flat wing and the L/D of the one cambered for an elliptical load distribution has a slightly higher L/D than the flat wing except for values of M above 1.6. This indicates either that there is some experimental error or that the reasons for using trapezoidal loading are not valid in practice.

(c) The Theoretical reasons for using camber and twist, and their application to the present C104/2.

The theoretical reasons for using camber and twist are not clearly explained and substantiated experimentally in the NACA reports. For example, in Reference (3) one finds the following paragraph:

"It was also shown in Reference (1) that flat wings of the aforementioned plan form developed a high section loading at the tips and that the chordwise distribution of load gave very high peak pressures near the leading edge. Such load concentrations are undesirable from both the aerodynamic and structural points of view. To avoid this condition the solution of the camber and twist of a swept wing designed to support a uniform load was developed in Reference (1)" ("Reference (1)" is also Reference (1) of this report). This may be the reason for designing such a wing, but one would not know it from Reference (1). In fact, in that report it is implied that it was because theoretical solutions for the drag due to lift were available for uniformly loaded wings that such wings were considered. The author wrote:

"The drag due to lift is estimated from theoretical solutions for the supersonic flow over thin lifting surfaces. Theoretical solutions are known for cases in which the lifting surface is curved and twisted in such a way as to support a uniform load and for certain rectangular, triangular or tapered flat surfaces." Later in Reference (1) it does say that it is expected that the leading edge suction predicted by theory will not be realized in practice unless the leading edge is given a finite radius or camber. The fact that the trapezoidally loaded wing of Figure 1(a) has a smaller L/D than have either of the other two wings casts doubt on the reasoning in Reference (3).

However, if it is assumed that the reasoning is correct, it should be pointed out that the C104 wing will not have some of the high loadings referred to in Reference (3). This is because, first of all, a convential subsonic airfoil section with a rounded nose is used. This according to Jones in Reference (1) and according to experiments described in Reference (2), would enable about half of the theoretical leading edge suction to be realized. Secondly, the high section loading at the tip is reduced because the tips are not pointed. Curves showing the spanwise variation of local lift coefficient at M = 1.4 for several wings according to linearized supersonic theory are on page 30. (The loss in lift at the tip was calculated using the methods of Reference (13).) The curves show that because of the finite tip effect the local lift coefficients are actually smaller than those of the cambered and twisted wing of Reference (8) when the local lift coefficients at the root chord are equal.

(d) Changes which might enable the benefits of camber and twist to be realized more fully.

The calculations of the performance shown in Section 6 for the wing cambered and twisted for increased $L/_D$ are based directly on a comparison of the drag polars of Ref. (6) and (8). This means that the design C_L was assumed to be .25. The reason for making this assumption was that the wind tunnel results showed that the greatest improvement in drag occurred at C_L = .20, and coincidently, the C_L for a 2g turn at M = 1.5 at 50,000 ft. is .195, at a combat weight of 44,000 lb. Unfortunately, the figures show that the performance is in general decreased when camber and twist at this theoretical design C_L of .25 is used. This is because the aircraft is flown most of the time at a C_L lower than the design C_L and consequently, the increased C_{D_o} becomes the dominant term in the drag equation. There are two possible ways of correcting this situation. One is make the design C_L smaller. The other is to increase the C_L at which the aircraft usually flies. The theoretical design C_L is already as low as .25, and if it were decreased, the gain in $L/_D$ would also decrease and much of the present advantage would be lost. Therefore, the C_L of the aircraft must be raised. This can be done by increasing the design altitude and/or by increasing the wing loading. However, the design altitude is limited to 50,000 ft. by the Specification and to not much more by the characteristics of turbojet engines, all of which have such a large decrease of thrust with increasing altitude above 50,000 ft., that only an impractical sea level power to weight ratio would provide enough thrust to increase the ceiling appreciably. Thus, the only thing to do is to increase the wing loading. This can be done by decreasing the wing area, but it is not as effective as one might suppose. The wing loading will not increase in the same proportion as the wing area decreases because, unless the fuselage size is increased to accommodate the fuel which can no longer be stored in the wing, the aircraft structure weight and fuel weight will both decrease also. If the fuselage size is increased to store more fuel, the drag will increase, thus defeating the purpose of decreasing the wing area. Since the maximum $L/_D$ is proportional to the square root of C_{D_o}, even if the fuselage size is not increased, the possible gain in $L/_D$ made by increasing the C_L will be counteracted by the increase in C_{D_o} which results when the wing area is reduced.

Calculations have been made of the weights and performance of a version of the twin-engined C-104, designated C-104/X, which has a wing area of 900 sq.ft. A table of comparative figures for the two versions, with flat wings, is presented below:-

	C-104/2	C-104/X
Wing area (Sq.Ft.)	1,184	900
Wing loading at a comparable weight, (with half fuel) (Lb./Sq.Ft.)	36.3	42.5
C_L at 2g; M = 1.5; h = 50,000 ft.	.190	.222
Drag at 2g; M = 1.5; h = 50,000 ft.		
- without elevator drag (Lb.)	12,660	12,530
- with elevator drag (Lb.)	17,710	17,920
Landing distance (Ft.)	5,770	6,730

These figures show that the cruising C_L has increased from .095 to .111, and that the landing distance has increased almost 1,000 ft. The change in C_L is hardly enough to enable advantage to be taken of the greater L/D that can be obtained using camber and twist, and the change in landing distance is unacceptable.

Another change which has been suggested is a change of plan form to that of the wings of Fig. 1(c). Fig. 1(b) and (c) show that there is not much to choose between the flat wings. The cambered wing of Fig. 1(c) has a larger L/D but as it was explained on p. 24, this result is somewhat dubious because of the different airfoil sections used. If it is assumed that the result is correct, there perhaps would be a good case for using a wing of that plan form. However, there are several considerations which make it inadvisable to do so:-

(1) Calculations have shown that to make the aeroelastic properties of such a wing satisfactory, the wing weight would have to be much greater than that of an aerodynamically thinner delta wing.

(2) There is not enough usable space in the large aspect wing and so the fuselage would have to be larger. The available evidence does not show that the fuselage would have better drag characteristics at supersonic speeds than the wing. Therefore, for the same fuel capacity, the drag would increase.

(3) The problem of keeping the C.G. movements small enough when the wing fuel tanks are drained is greater.

(4) It is more difficult to obtain a large enough ground angle with such long wings.

(5) There is not as much control surface area available at large distances from the C.G.

REFERENCES

(1) NACA TN 1350

(2) 2100

(3) NACA RM A9J24

(4) A50A31a

(5) A50K21

(6) A50K24a

(7) A50K27a

(8) A51E01

(9) A52B08

(10) L50I22

(11) A8D02

(12) A9D07

(13) TN 1860

EXPERIMENTAL MAXIMUM LIFT-DRAG RATIOS

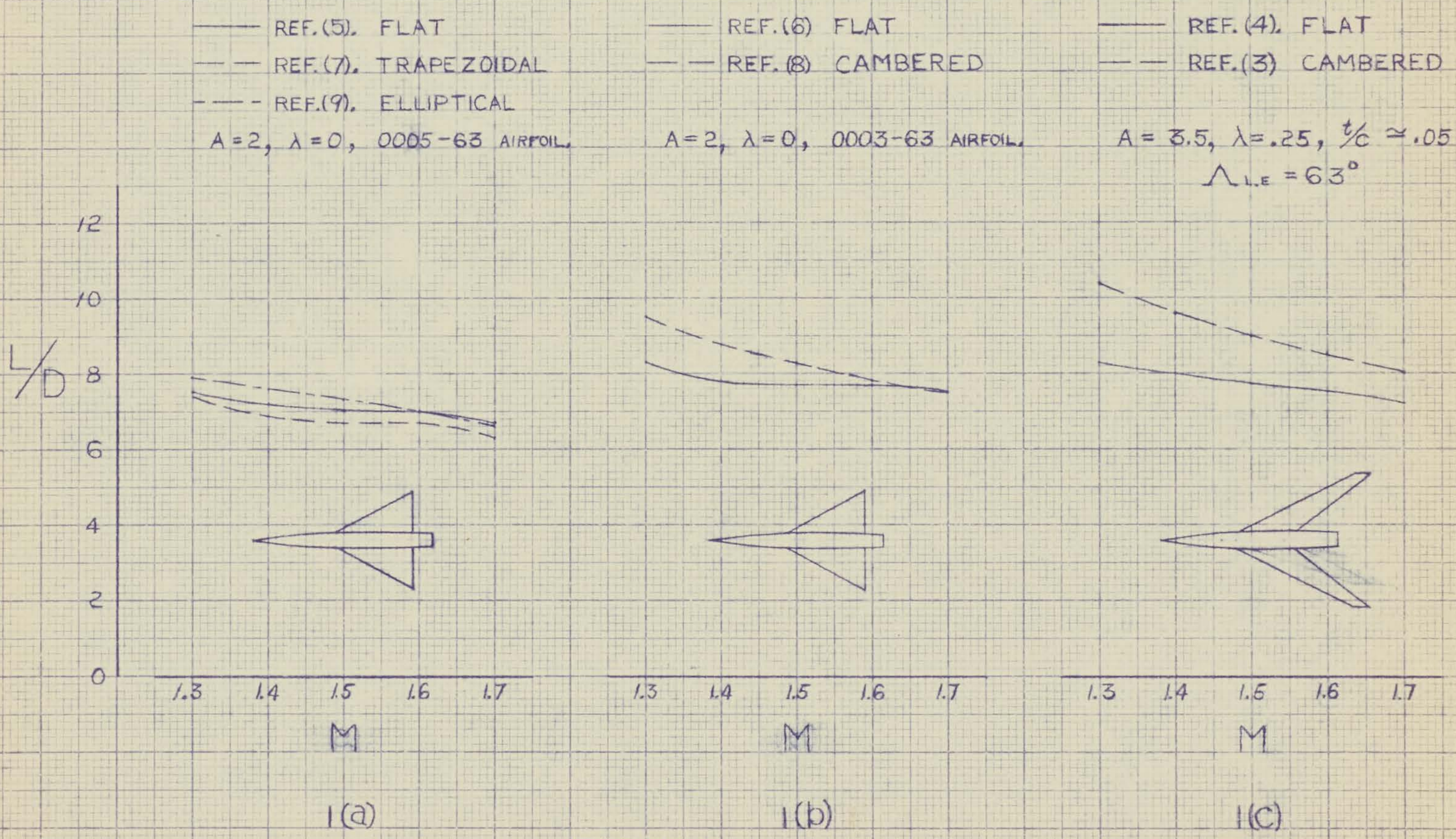

FIG. 5.1

AIRCRAFT C-104 COMPONENT SHEET No. 30 REPORT No.
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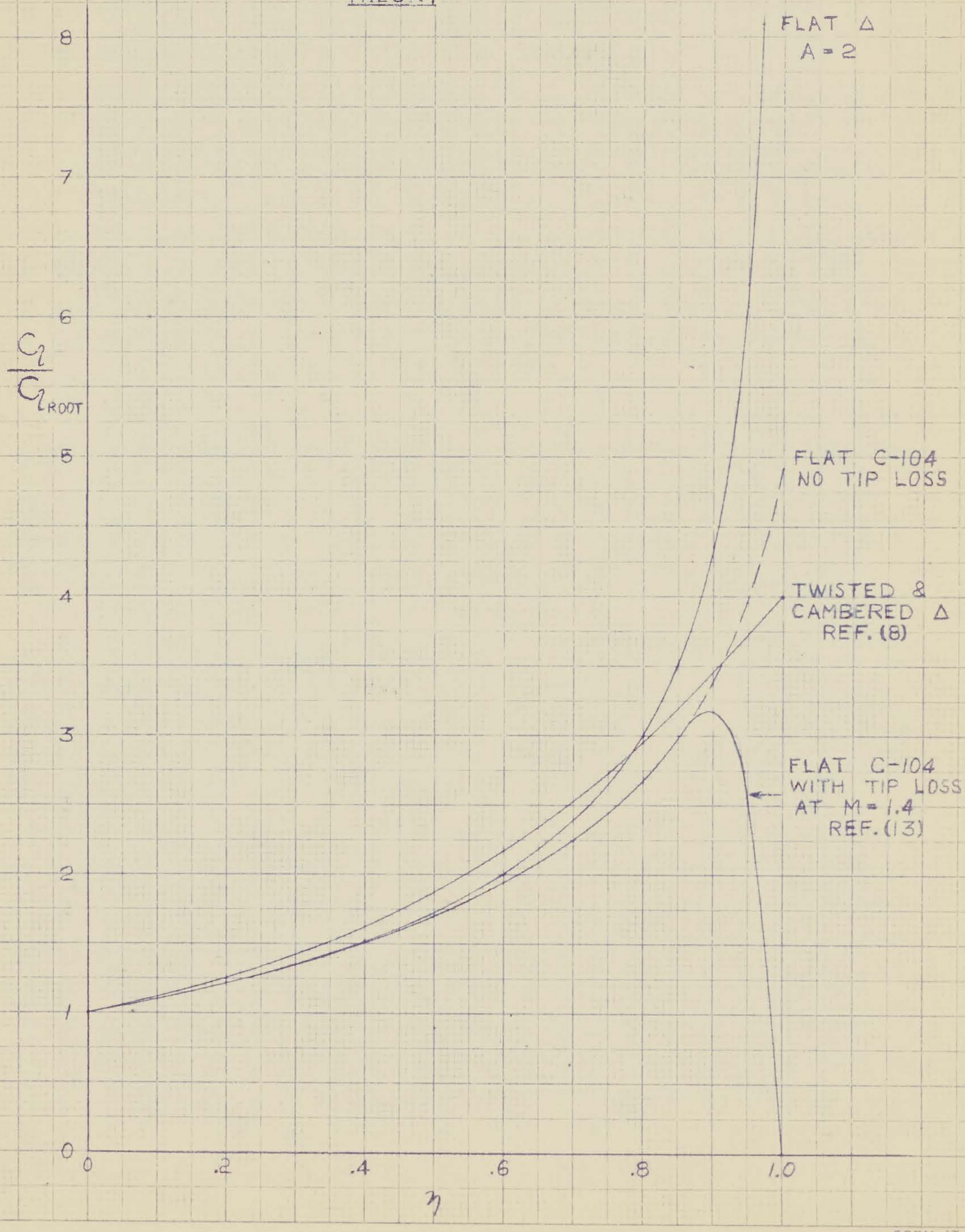
FIG. 5.2

SPANWISE VARIATION OF LOCAL LIFT COEFFICIENT ACCORDING TO LINEARIZED SUPERSONIC THEORY



FORM 1746

SECTION 6

PERFORMANCE WITH CAMBER AND TWIST

In this section, comparative figures for the drag under "g" and the basic performance of the C-104/2 are given in Tables I and II respectively for the plane wing and the modes of camber and twist, which have been discussed in the previous sections.

For the purposes of this section, "twisted" refers to the mode described in Section 5, which is used for reducing the total drag at high design lift coefficients at the expense of a higher drag at lower lift coefficients. Similarly, "cambered" refers to the mode described in Section 4, which is used to reduce the elevator drag.

It should be understood that the figures for cambered and cambered and twisted wings are somewhat theoretical, because of the objections to these modes which have been fully discussed in Sections 1 and 4. When the figures presented are viewed in this light, there does not seem to be much hope for meeting the turning requirements of the Specification at 50,000 ft., by aerodynamic means alone. For this reason, the engine size required to meet the requirement has been given in Table I.

In this connection, it is of interest to note that recently Dr. Hooker, of the Bristol Company, mentioned that they were contemplating the development of an engine very similar to the Olympus having 20,000 lb. Sea Level static thrust. This engine would come very close to giving the C-104 the necessary turning performance. This is because the Olympus operates at a lower temperature than the TR.9, thus permitting higher afterburner augmentation. Hence, the required S.L. static thrust for an Olympus type of engine would be slightly less than for the TR.9 type of engine, which is used for reference in Table I.

In view of the pessimistic nature of the results of the investigations of ways of improving the drag, it was felt appropriate to find out whether the results were sensitive to the value of the drag efficiency "e" which has been assumed. The effect of errors, in the estimation of "e", is shown on the carpet presented in figure 6.1 of this section. It can be seen that fairly large errors do not alter the position materially. This is of considerable interest, since the experimental data on this subject is not entirely satisfactory.

TABLE I

C-104/2 DRAG VARIATION WITH "G", FOR TWISTED, CAMBERED, AND STANDARD WINGS, AT 1.5 M.N., AT 50,000 FT.

Weight, 44,000 lb.

	"g"	Drag				% Change	S.L. * Static Thrust Req'd.
		Zero Lift	"Induced"	Elevator	Total		
Standard	1.0 1.5 2.0	6,320 6,320 6,320	1,750 3,830 7,000	1,113 2,859 5,294	9,183 13,009 18,614	0 0 0	23,900
Twisted	1.0 1.5 2.0	8,850 8,850 8,850	-50 1,150 3,550	1,113 2,859 5,294	9,913 12,859 17,694	+8.0 -1.9 -5.0	22,600
Cambered	1.0 1.5 2.0	6,320 6,320 6,320	1,750 3,830 7,000	70 637 2,030	8,140 10,787 15,350	-11.4 -17.5 -17.5	19,600
Twisted and Cambered	1.0 1.5 2.0	8,850 8,850 8,850	-50 1,150 3,550	70 637 2,030	8,870 10,637 14,430	-3.4 -18.6 -22.5	18,500

* Based on TR.9 Engine Characteristics, with 1,800°K Reheat.

TABLE II

C-104/2 PERFORMANCE FOR TWISTED, CAMBERED, AND STANDARD WINGS

Weight, 44,000 lb.

		TWISTED	CAMBERED	STANDARD
Maximum Speed - M.N.				
	0	1.28	1.60	1.66
	20,000'	1.59	1.94	1.97
	36,090'	1.85	2.14	2.15
	50,000'	1.72	2.05	1.99
Maximum Rate-of-Climb - Ft./Min.				
	0	38,200	43,500	48,000
	20,000'	27,000	28,200	30,000
	36,090'	15,520	20,400	20,400
	50,000'	4,850	6,840	5,000
Maximum Operational Ceiling - Ft.		55,700	59,500	55,000
Acceleration Time from .95M to 1.5M - Min. (36,090')		1.26	1.11	1.12
High Speed Combat Radius - Naut. Mi.		228	274	242

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COMPONENT

SHEET No. ______ REPORT No. ______

DATE ______ PREP. BY ______ 34.

VARIATION OF DRAG WITH VARIATION OF THE DRAG EFFICIENCY FACTOR "e" AND THE NORMAL ACCELERATION (No. OF "g")

ALT. - 50000 FT
MACH No 1.5
STANDARD "e" - .40

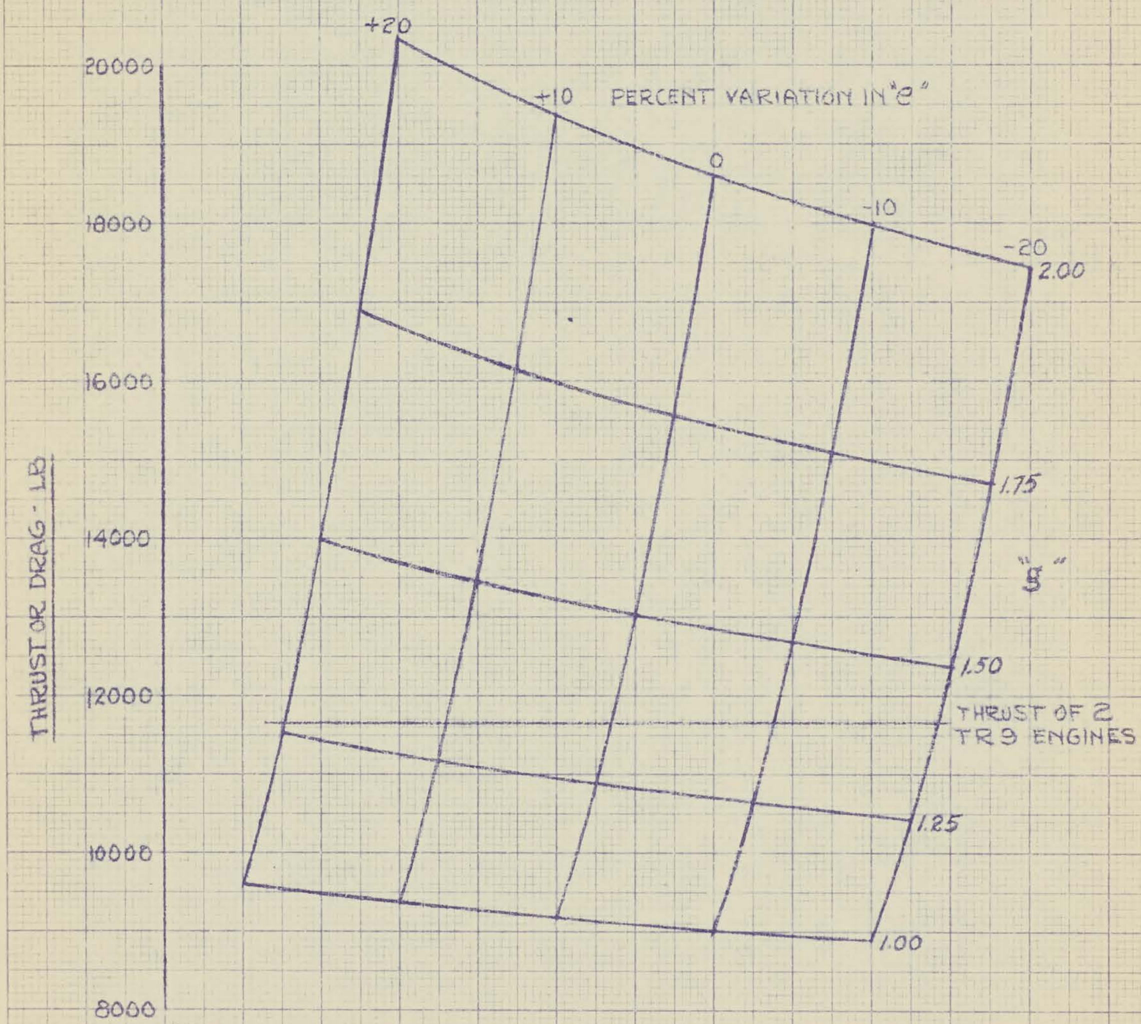

FIG. 6.1

APPENDIX I

"ENGINE INSTALLATION ON C104/1 (SINGLE ENGINE)

The single engine version of the C104 must be tailored very close to the engine and equipment to get the required performance. From the drawings, the impression might be created that there are excessive clearances around the engine, thus permitting great flexibility with respect to the fitting of different engines. While this is true in the early stages of design, it is not true after finalization of the various details, because of the close clearances of a great number of items which cannot be moved without a great amount of redesign. These items are listed below, with an estimate of the clearance allowed at present. It only takes a small pipe or accessory of a new engine to foul these clearances and make a new installation very difficult. Also it should be noted that with this installation the engine must be withdrawn from the rear, so that clearance for any accessories (fuel pumps, pipes, etc.) on the engine must be available for the entire backward movement of the engine.

CLOSE CLEARANCE ITEMS

DESCRIPTION		APPROX. CLEARANCE
Wheels to Engine Shroud	(Shroud outline modified) (locally to clear wheels)	1.0"
Wheels to Fuselage Skin line		1.75"
Wheels to Main Spars		2.5"
Engine and Shroud		1.5"
Engine Shroud and Former Rings		0."
Engine Shroud and Wing Spars		.5"
Engine Shroud and Fin Spars		0."
Engine Shroud and Hydraulic Reservoirs		2.0"
Engine Shroud and Hydraulic Units	(2 Turbines 3 Air Bottles) (2 pumps 2 Accumulators)	2.0"
Engine Shroud and Ram Air Turbine		1.5"
Outside Skin and Ram Air Turbine		1.5"
Engine Shroud and Aft Fuselage Fuel Tanks		2.0"
Afterburner Shroud and Dive Brake Actuators		1.5"
Afterburner Shroud and Elevon Actuators in Fuselage		2.0"
Afterburner Shroud and Elevon Control Valves		2.0"
Afterburner Shroud and Rudder Actuator		2.0"
Afterburner Shroud and Rudder Control Valve		2.0"
Afterburner Shroud and Drag Shute Stowage		3.0"
Afterburner Shroud and Tail Bumper Jack		2.0"

NOTE:

Remaining space under engine and afterburner must be left clear for accessability to these components.