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AD NUMBER
AD091753
CLASSIFICATION CHANGES
TO: unclassified
FROM: confidential
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FROM: Controlling Organization: British Embassy, 3100 Massachusetts Avenue, NW, Washington, DC 20008.
AUTHORITY
DSTL, DSIR 23/24149, 5 Aug 2008; DSTL, DSIR 23/24149, 5 Aug 2008

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TECH. NOTE AERO.2401

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TECH. NOTE AERO.2401

ROYAL AIRCRAFT ESTABLISHMENT

FARNBOROUGH, HANTS

TECHNICAL NOTE No: AERO.2401

SOME EFFECTS OF FLOW SEPARATION ON THE OVERALL FORCE CHARACTERISTICS OF THIN WINGS OF ASPECT RATIO 2 to 4

by

A.L. COURTNEY, B.Sc.

OCTOBER, 1955

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U.D.C. No. 533.691.11 : 532.526.5

Technical Note No. Aero 2401

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Some effects of flow separation on the overall force characteristics of thin wings of aspect ratio 2 to 4

by

A.L. Courtney, B.Sc.

R.A.E. Ref: Aero H/ALC

SUMMARY

This note has been prepared for the Designers' Conference on M = 2 aircraft to be held at the R.A.E. It gives a brief review of some of the available American wind-tunnel data on wings of aspect ratio 2 to 4 and thickness-chord ratio 0.03 to 0.06 (mainly the lower parts of these ranges), with the object of illustrating the type of overall force behaviour to be expected on wings of this kind, particularly with regard to the lift and pitching-moment characteristics and the drag due to lift.

The main conclusions (for others see para.5) are:

(i) For aspect ratios of the order of 2 and thickness-chord ratios of about 0.03 the delta type of planform has the most favourable pitching-moment characteristics and the unswept wing the least favourable.

(ii) At supersonic speeds the drag due to lift corresponds fairly closely to a complete loss of the equivalent linear-theory leading-edge suction force, regardless of leading-edge sweepback and section shape.

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1 Introduction

This paper contains a brief review of some of the NACA wind-tunnel data on wings of moderately low aspect ratio having thickness-chord ratios of 0.03 to 0.06, such as might be used for aircraft intended for operation near $M = 2$. The object is to illustrate the broad effects on overall force characteristics rather than to discuss the nature of the flow in any detail or to attempt comparisons with theoretical predictions; some discussion of the change in the general nature of the flow behaviour compared with that found on current subsonic wing designs is, however, included. Attention is concentrated mainly on the subsonic and transonic force characteristics, but some consideration is given to the supersonic characteristics also, particularly as regards drag due to lift.

Para.2 contains some general remarks on the character of the flow separation to be expected at subsonic speeds on wings of the kind under review, together with its qualitative effects on lift curve slope. Para.3 deals with the pitching-moment behaviour, including the effects of planform, section shape, aspect ratio, camber and twist, leading edge extensions, and wing-mounted fins, by reference to a selected number of typical C_m - C_L curves (Figs.3-12). The main geometrical characteristics of the wings tested are summarized alongside the curves, together with the Mach number and Reynolds number of the test and the number of the appropriate reference, from which further details can be obtained if required. Most of the wings were tested in combination with a body, the fineness ratio being generally of the order of 10:1. Para.4 deals with the drag due to lift at small incidences, the object being to show how the measured value of dC_D/dC_L^2 compares with the theoretical value for attached flow on the one hand and that for complete loss of the equivalent leading-edge suction force on the other. Finally, para.5 indicates what general conclusions may be drawn from the data considered.

2 Character of flow separation at subsonic speeds, and its effects on lift

At low subsonic speeds, on the thin wings under consideration here, leading-edge separation takes place at a relatively low incidence. Compared with current wings designed for high subsonic speeds, having thicknesses of the order of 0.10c and somewhat higher aspect ratio, there are two important differences in the separation characteristics, namely:-

(i) the separation extends rapidly over the whole leading edge, instead of being confined to the outer part of the span over an appreciable incidence range, as on many current designs. This has an important bearing on the pitching-moment behaviour (para.3),

and (ii) the effects of the vortex sheets associated with the separation are greater, and the two-dimensional "bubble" effects are less than on many current designs. In the case of thin low-aspect-ratio delta and sweptback wings, the flow behaviour tends towards that found on the slender delta, in which the two-dimensional type of separation bubble is practically absent and the flow is dominated by the spiral vortex sheets discussed in Ref.1. In the case of unswept wings the vortex sheets associated with the separation from the tip edges become of greater importance at low aspect ratio.

Because of these effects it is found that on wings of the thickness and aspect ratio under consideration the effect of leading-edge separation is generally to increase the lift compared with what would be expected for attached flow (e.g. using the Helmbold-Diederich relation, Ref.2). A typical example is shown in Fig.1, which contrasts the effect of leading-edge separation on a sweptback wing of aspect ratio 6 with that on a delta wing of aspect ratio 2. At the higher aspect ratio the loss of sectional lift slope due to the two-dimensional effect of the separation bubble* predominates, and the lift falls beneath the linear-theory value for attached flow as incidence is increased. On the delta wing, however, the effects of the vortex sheets predominate, giving an appreciable non-linear lift increment, with the result that the lift with separated flow is higher than it would be in unseparated flow. It can be seen that the lift increase persists up to high incidences; this also is typical. In Fig.1, some indication of how far the flow over the delta wing has progressed towards the spiral vortex flow found on narrow delta wings is provided by the comparison of the measured $C_L - \alpha$ curve with that estimated using the narrow delta theory of Ref.1; the measured lift lies about half-way between the estimated curves using the simple Helmbold-Diederich relation and narrow-delta theory respectively.

The amount of the non-linear lift increment due to separation varies, of course, with planform and thickness. Numerous experimental results are available from which the lift characteristics of a particular design can be obtained. Since lift slope, as such, is of less interest to the designer than, for instance, the pitching-moment and drag characteristics, no analysis of lift data is undertaken here.

Increase in Mach number will in general lead to changes in the nature of the flow separation, from the low-speed type discussed above to more complicated types involving shock waves and shock-induced separations in addition to bubbles and vortex sheets. For instance, when the Mach number component normal to the leading edge is sufficiently high, the low-speed leading-edge type of separation may be replaced by a supersonic expansion round the nose, followed by a shock wave and perhaps shock-induced separation further aft. In two-dimensional flow, such changes can give rise to abrupt variations of force characteristics with Mach number, and the possibility therefore exists of such undesirable variations occurring on the wings considered here. However, examination of a considerable number of experimental results indicates that, at any rate so far as lift is concerned, the flow changes that occur on thin wings of fairly low aspect ratio do not in general give rise to serious irregularities in the curves. The shape and slope of the lift curve generally appear to change quite gradually with Mach number, the curves forming a fairly regular family. Fig.2(a) gives some typical results for 3% thick wings of aspect ratio between 2 and 4, from the wind-tunnel tests of Ref.4. It is not practicable to present the complete lift carpet for each wing; instead the values of C_L/α at an incidence of 8° are plotted against Mach number. This incidence is well within the separated-flow regime at low Mach numbers, and any irregularities arising from changes in the nature of the flow with increase of Mach number should thus be noticeable on the curves plotted. No such irregularities can be seen for any of the wings considered; the trends in C_L/α are smooth and closely similar to the trends in $dC_L/d\alpha$ at $\alpha = 0^\circ$ (shown dotted) apart from a tendency towards somewhat higher transonic peaks, more particularly for the delta wing of aspect ratio 4 and the unswept wing of aspect ratio 3.

For Mach numbers between 0.9 and 1.2 most of the available data for thin low-aspect-ratio wings at moderately high incidence consists of transonic bump results, and a selection of these is given in Fig.2(b) for unswept and delta planforms; some transonic tunnel results are also included. It can be seen that the smooth variation of C_L with Mach number noted in Fig.2(a) is continued through the transonic speed range, even on the unswept wings.

Figs.2(a) and 2(b) should not be taken as meaning that in every application the transition from one type of flow to another as Mach number increases will necessarily take place smoothly, but only that there seems to be a good chance of this being so in many cases. It should also be remembered that the

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* and to the effect of the spanwise boundary-layer drift.

above refers only to the effects on lift; the more important question of pitching-moment behaviour is dealt with below.

3 Pitching-moment characteristics

3.1 General trends with aspect ratio and planform

Fig.3 illustrates the effects of aspect ratio on the C_m - C_L curves at $M = 0.6$ for $45°$ swept wings and for delta and cropped delta wings. At high aspect ratio the loss of lift in the leading-edge separation bubble, greatest on the outboard sections, causes an inward and forward movement of the overall centre of pressure at high incidence and so produces the severe "pitch-up" which is characteristic of many current subsonic designs. As the aspect ratio is decreased, however, the vortex sheets associated with the separation become a more important feature of the flow and the lift associated with these reduces the stability loss. When the aspect ratio has fallen to 2 the flow over the whole of the upper surface is dominated by the vortex sheets, which have become of the conical or spiral type found on the slender delta (Ref.1). With fully-developed conical flow, as noted in Ref.1, the theoretical aerodynamic centre remains constant at two-thirds of the root chord. The experimental results for thin delta wings of aspect ratio 2 indicate that, although such wings cannot be regarded as slender, nevertheless the force characteristics may be similar to those for a slender delta inasmuch as there is a considerable non-linear lift increment (above) and, more important the C_m - C_L curves are more or less linear, as in the examples of Figs. 3(b), 4(a), 7 and 12. Thus, it is not necessary to go to the extreme of a slender delta planform in order to achieve one of the main aerodynamic advantages of this planform, namely the linear C_m - C_L characteristics associated with the separated conical vortex flow at subcritical speeds. Even with an aspect-ratio-3 delta (Figs.3(b), 4(a) and (b)) the pitching-moment characteristics, with the thin sections under consideration, are free from large changes in stability at any rate at low Mach numbers, but there is an obvious deterioration compared with the results for aspect ratio 2.

It thus appears that on thin low-aspect-ratio wings the effects of flow separation on the pitching-moment characteristics at low Mach number are much less severe than for current subsonic designs, and for delta wings of aspect ratio about 2 and thickness of about 0.03c in particular, significant stability changes may be absent altogether. It must be emphasized, however, that these conclusions refer to wing-body combinations in the absence of a tailplane. With a tailplane behind the wing severe instability may still occur as a result of the large changes of downwash at the tail with incidence, caused by the wing vortices. An example of this is shown in Fig.5 for a 5% thick delta wing of aspect ratio 2. Without the tail the C_m - C_L curve is practically straight up to a lift coefficient of 1.1, despite the occurrence of separation at a lower lift coefficient, but with the tail in the position tested there is a serious loss of stability beyond $C_L = 0.4$, at which point the tail becomes de-stabilizing. The tail in this example is 0.28 of the wing semispan above the wing chord plane. Past experience has shown that the loss of stability is greatest with the tail in such a moderately high position, due to the fact that it moves into the region of most rapidly increasing downwash as incidence is increased. By placing the tail beneath the wing chord plane the loss of stability due to downwash changes can be minimized and often avoided altogether.

With a canard layout this particular difficulty is absent, but the possibility then arises of the foreplane wake, and in particular the foreplane tip vortices, affecting the flow over the mainplane. It is not yet known how serious such effects may be and how they can best be avoided, and further work is to be done on this question. One obvious possibility is to use a high foreplane and a low mainplane (rather than the other way round as in some recent canard designs) but this may involve some difficulties in internal layout (e.g. aerials) and some increase in wing-body interference drag which is in general least for a mid-wing position.

Fig.4(a) shows some effects of planform (i.e. unswept, sweptback and delta) on pitching-moment characteristics for 3% thick wings of aspect ratio 2 and 3 at subsonic speeds. As already noted, the C_m - C_L curves for the sweptback and delta wings at these aspect ratios are relatively free from large changes of stability with lift coefficient. For the unswept wings, however, there is a large increase of stability beyond a lift coefficient of 0.2 to 0.4 depending on aspect ratio and Mach number. This is due to the fact that on unswept wings the separation is essentially two-dimensional in character. For the aspect ratios under consideration the characteristics are little affected by the tip vortex sheets, and the favourable effect of the leading-edge spiral vortex sheets noted above are of course absent in the absence of appreciable sweepback. At low and moderate subsonic speeds on the unswept wings, therefore, the centre of pressure moves rearwards with incidence because of the rearward expansion of the long separation bubble, and at high subsonic speeds it moves rearwards probably as a result of shock-induced separation effects either near the leading edge or further aft. The unswept-wing results in Fig.4(a) are for 3% thick round-nosed sections (elliptical ahead of 0.5c, biconvex behind) at a Reynolds number of $2\frac{1}{2} \times 10^6$. Sharpening the nose or increasing the Reynolds number to $8\frac{1}{2} \times 10^6$ (see Ref.4) had little effect, and it is likely that behaviour of a similar kind to that shown will be found in full-scale conditions, although the C_L at which the stability begins to increase, and the severity of the increase, will not necessarily be the same. For instance, whereas Fig.4(a) shows that at $M = 0.9$ the stability increase on the aspect-ratio-3 unswept wing begins at $C_L = 0.2$, tests in Ref.16 on a rectangular wing of the same aspect ratio with a 4% thick biconvex section showed no increase up to $C_L = 0.4$. Such differences may be due partly to experimental differences (e.g. the present results refer to tunnel tests on a complete wing with body while those of Ref.16 are for a half-model wing without body, mounted on a transonic bump) but genuine variations, due to differences in section shape, taper etc. (see below) can also be expected. However, it seems clear from the evidence available, that despite uncertainty as to the precise value of the C_L for increase of stability in any particular application, unswept thin wings in general will require some form of nose flap or other modification in order to retard the development of the separation at subsonic Mach numbers and so allow high lift coefficients to be reached without such unacceptable increases in stability. In this respect the unswept wing is at some disadvantage compared with, say, a delta wing of equal aspect ratio for which the C_m - C_L curves are more nearly linear, particularly for aspect ratios of the order of 2.

A further, associated, disadvantage of thin unswept wings of aspect ratio 2 or more is that the variation of the pitching-moment characteristics with Mach number is far less smooth than for thin sweptback and delta wings. Some indication of this is given in Fig.4(a) where the curves for $M = 1.3$ have been added for the unswept and delta wings of aspect ratio 2. For the delta wing the C_m - C_L curves are more or less linear at all three Mach numbers (0.6, 0.9 and 1.3), the transition from subsonic to supersonic free-stream Mach numbers taking the form of a gradual increase in stability at all lift coefficients. This is a general feature of thin sweptback and delta wings of fairly low aspect ratio, apart from occasional isolated kinks or moderate losses of stability which are dealt with below. On the unswept wing, however, there is a marked change in the shape of the C_m - C_L curve with Mach number. At low lift coefficients the stability first decreases between $M = 0.6$ and $M = 0.9$ and then increases considerably between $M = 0.9$ and $M = 1.3$; at lift coefficients above about 0.4, on the other hand, the stability tends to decrease throughout the Mach number range. A further comparison between the unswept and the delta planform is given in Fig.4(b). Because of the higher aspect ratio (3) in

Fig.4(b) the delta-wing curves are less straight than in Fig.4(a), but they nevertheless form a much more regular series, as Mach number is varied, than those for the unswept wing. On the unswept wing the pitching-moment characteristics vary with Mach number as a result of flow changes of the type which occur in two-dimensional conditions as Mach number increases. The advantage of the low-aspect-ratio delta planform probably lies at any rate partly in the fact that, because of its high leading-edge sweepback ($63\frac{1}{2}^{\circ}$ for $A = 2$), which reduces the free-stream Mach number component normal to the leading edge, the character of the separation remains basically unchanged (i.e. subsonic with spiral vortex sheets) up to moderate supersonic Mach numbers.

The unswept-wing results of Figs.4(a) and 4(b) are for taper ratios of 1.0 and 0.4. The irregularities in pitching-moment characteristics can be reduced by decreasing the taper ratio (i.e. increasing the taper). An example of this is shown in Fig.4(c) where results are compared for two wings of aspect ratio 3 having NACA 64A003 sections and unswept half-chord lines. One wing has a taper ratio of 0.5 and the other a taper ratio of zero, i.e. it is diamond-shaped with 34° sweepback on the leading edge and 34° sweep-forward on the trailing edge. The diamond-shaped wing is appreciably better than the wing of taper ratio 0.5, but although part of the improvement is no doubt due to the increased taper as such, it is probable that an appreciable part comes from the increased leading-edge sweepback. Whether such a wing should strictly be classified as unswept is open to argument. Certainly the leading-edge sweepback will make it more difficult to improve the low-speed behaviour by droop etc. than on the less tapered wing, and the characteristics of flap-type trailing-edge controls and of high-lift flaps will be affected by the sweep-forward on the trailing edge.

The characteristics of diamond-shaped "unswept" wings can be expected to improve as aspect ratio is reduced below 3, if only because of the greater leading-edge sweepback entailed. The characteristics of the more conventional type of unswept wing may also be less adverse at lower aspect ratio, due to the increasing three-dimensional effects of the tip vortex sheet, although to derive appreciable benefit from this the aspect ratio may have to fall to 1 or less. However, the characteristics of sweptback and delta wings also improve with decrease of aspect ratio, as already noted, and it has been seen that for a thin aspect-ratio-2 delta in particular the $C_m - C_L$ curves are practically linear at all Mach numbers. It is most unlikely that this would be achieved with an unswept wing at this aspect ratio, even with a taper ratio of zero, and so in this respect there seems to be a clear advantage for the low-aspect-ratio delta planform.

In the preceding discussion on general trends little attempt has been made to differentiate between the delta planform and what has been termed the sweptback planform, involving a sweptback trailing edge. It should be emphasized at this point, however, that although many of the remarks on general trends apply more or less equally to both types of wing, the particular advantage of relatively straight $C_m - C_L$ characteristics at all Mach numbers seems to be associated largely with the low-aspect-ratio delta planform. Certainly at an aspect ratio of 3 (Fig.4(d)) and probably also at an aspect ratio of 2, there is a significant deterioration in pitching-moment behaviour as the trailing-edge sweepback is increased and the taper decreased at constant aspect ratio and leading-edge sweepback. Fig.4(d) shows this for wings having a leading-edge sweepback of 53° and aspect ratio of 3, with NACA 0003 sections, the delta planform being compared with a sweptback planform having a taper ratio of 0.4 and trailing-edge sweepback of 36°.

Most of the above has been concerned with the behaviour at subsonic and transonic free-stream Mach numbers. At supersonic Mach numbers up to

1.7-1.8, however, (the limiting Mach number in most of the tests considered here) examination of a large number of experimental results on thin wings of aspect ratio 2 to 3, whether unswept, sweptback or of delta type, has shown no appreciable irregularities in the pitching-moment characteristics up to lift coefficients as high as 0.6. This is not altogether surprising, since on the unswept wings the flow conditions are fully supersonic throughout this range, and on the sweptback and delta wings the leading-edge sweepback is in most cases sufficiently high to give a subsonic leading-edge, and therefore perhaps an essentially subsonic flow behaviour, up to these Mach numbers. It does not necessarily follow, however, that this will be true for all supersonic Mach numbers. The possibility exists, on the highly-swept wings, of irregularities being found in some cases near the Mach number at which the component of free-stream velocity normal to the leading edge becomes supersonic.

3.2 Other effects

3.21 Irregularities in pitching-moment characteristics, and the effects of camber and twist

Although it has been seen that on thin low-aspect-ratio sweptback and delta wings, and especially the latter, the effects of flow separation are less severe than for current subsonic designs, there are nevertheless some irregularities at subsonic speeds, more particularly at the higher aspect ratios (e.g. 3). On the delta wing of aspect ratio 3 in Fig.4(a), for example, there is a local kink in the C_m - C_L curve near C_L = 0.4 at M = 0.6, where there is a temporary reduction in stability followed by recovery beyond C_L = 0.5. This kind of kink has been found in several tests. For the aspect ratios and wing thicknesses under review it usually occurs only at Mach numbers between about 0.6 and 0.9. At lower Mach numbers any irregularity usually takes the form of a small or moderate loss of stability persisting over a larger C_L range, as shown in Fig.4(a) for the sweptback wing of aspect ratio 3 at M = 0.6.

The local kinks at moderate subsonic Mach numbers may not be of much importance operationally in a fighter or bomber designed to operate mainly at a Mach number of 2, since they generally occur at a lift coefficient of 0.4 or more, which is outside the range for steady level or climbing flight on such aircraft, and will therefore affect only the behaviour in manoeuvres, e.g. turns. They will not in general constitute an operational danger, as does the pitch-up found on some subsonic designs, unless of course bad tail positioning causes a simultaneous loss of tail effectiveness due to downwash variations. Their main effect will be on the ease and accuracy with which a given 'g' can be applied and held by the pilot at subsonic speeds. It can be argued, however, that most combat manoeuvres will be performed, or at any rate initiated, at supersonic speeds, and if the manoeuvre is sufficiently severe to reduce the speed to M = 0.9 or below then the trim changes associated with kinks in the subsonic C_m - C_L curves may be of little significance compared with the large changes which take place during the deceleration through the transonic speed range. It is thus possible that the existence of such kinks may not automatically rule out an otherwise attractive planform for an M = 2 aircraft, or make necessary the use of devices to eliminate them which may be costly in other directions, e.g. drag. Before considering such courses, therefore, it is suggested that further discussion is needed to assess the operational importance of these irregularities at subsonic speeds.

The type of C_m - C_L characteristic in which the loss of stability persists over an appreciable C_L-range at low Mach number, noted above in the case of the sweptback wing of Fig.4(a), is undoubtedly of importance. Occurring at take-off and landing, where accurate handling is essential, it would necessitate a further-forward c.g. position than would otherwise be used, and consequently the stability margin at lower lift coefficients, and at supersonic

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The local kinks at moderate subsonic Mach numbers may not be of much importance operationally in a fighter or bomber designed to operate mainly at a Mach number of 2, since they generally occur at a lift coefficient of 0.4 or more, which is outside the range for steady level or climbing flight on such aircraft, and will therefore affect only the behaviour in manoeuvres, e.g. turns. They will not in general constitute an operational danger, as does the pitch-up found on some subsonic designs, unless of course bad tail positioning causes a simultaneous loss of tail effectiveness due to downwash variations. Their main effect will be on the ease and accuracy with which a given 'g' can be applied and held by the pilot at subsonic speeds. It can be argued, however, that most combat manoeuvres will be performed, or at any rate initiated, at supersonic speeds, and if the manoeuvre is sufficiently severe to reduce the speed to $M = 0.9$ or below then the trim changes associated with kinks in the subsonic C_m - C_L curves may be of little significance compared with the large changes which take place during the deceleration through the transonic speed range. It is thus possible that the existence of such kinks may not automatically rule out an otherwise attractive planform for an $M = 2$ aircraft, or make necessary the use of devices to eliminate them which may be costly in other directions, e.g. drag. Before considering such courses, therefore, it is suggested that further discussion is needed to assess the operational importance of these irregularities at subsonic speeds.

The type of C_m - C_L characteristic in which the loss of stability persists over an appreciable C_L-range at low Mach number, noted above in the case of the sweptback wing of Fig.4(a), is undoubtedly of importance. Occurring at take-off and landing, where accurate handling is essential, it would necessitate a further-forward c.g. position than would otherwise be used, and consequently the stability margin at lower lift coefficients, and at supersonic speeds particularly, would be undesirably large. The available data indicate that this type of C_m - C_L curve occurs mainly on wings having aspect ratios of the order of 3 or more (compare for instance the curves for delta wings of aspect ratios 2 and 3 in Fig.4(a)) and it is less pronounced on delta and cropped-delta wings than on less highly-tapered sweptback wings. Since the optimum aspect ratio on performance and structural grounds is fairly low for a thin-winged M = 2 aircraft, it may be that the difficulty will not arise in a practical case in existing design conditions. If, however, the optimum aspect ratio were to increase to 3 or more (e.g. as a result of improvements in materials or structures) then the difficulty would be a real one. It might be possible to reduce or delay the loss of stability by such devices as leading-edge droop and extension, or camber, but there are few experimental results in this connection for very thin wings, most of the data being for wing thicknesses of 6% or more. Data are available (Fig.6) for a delta wing of aspect ratio 3 and NACA 0003 sections, with and without a 0.135c chord extension over the outer half of the span, showing that at M = 0.6 the extension delayed the loss of stability from C_L = 0.4 to C_L = 0.8. In this test the extension was drooped 3°. The effect on maximum L/D was slightly favourable at subsonic and transonic speeds but there was a 5% loss in maximum L/D at M = 1.9. It should not be concluded from this single result that the problem on thin wings of aspect ratio 3 or more is likely to be easily solved; in general it might be expected that the achievement of good low-speed high-lift characteristics on the thin wings needed for an M = 2 design will be even more difficult than for the subsonic designs tested hitherto, and the field will be more restricted by the need to avoid large drag penalties at supersonic speeds.* If aspect ratios of 3 or more are to be seriously considered for M = 2 designs, more experimental work will be required on these points.

Figs.7 and 8 are concerned with the local pitching-moment kinks at moderate subsonic speeds discussed earlier. Fig.7 shows the effect of thickness-chord ratio and camber on a delta wing of aspect ratio 2. With an NACA 0003 - 63 section the pitching-moment characteristics are free from kinks. With an NACA 0005 - 63 section, however, there are kinks at M = 0.6 and M = 0.8 near C_L = 0.5. The use of camber and twist practically eliminates these kinks. The camber and twist of this example are of the NACA "conical" type, in which the mean surface is generated by straight lines through the apex, designed to give elliptical loading at C_L = 0.25, M = 1.5 in attached flow. Details are given in Ref.4. A notable feature of the results in that the improvement in pitching-moment characteristics is obtained at no cost in maximum L/D, at any rate up to M = 1.7 (the highest test speed). Indeed, at subsonic and low supersonic speeds the camber increases the maximum L/D - from 11.7 to 13.0 at M = 0.9, and from 7.5 to 7.9 at M = 1.3. In addition, it gives large reductions in drag due to lift at high subsonic speeds; at M = 0.9 the value of C_D at a lift coefficient of 0.3 falls from 0.034 for the plane wing to 0.0265 for the cambered and twisted wing. A similar combination of camber and twist was also tested on the 3% thick wing and gave even larger drag improvements, maximum L/D being increased from 10.3 to 12.5 at M = 0.9 and from 8.1 to 9.1 at M = 1.3. There was little scale effect on either wing up to R = 7.5 x 10^6 (the highest Reynolds number tested).

Fig.8 gives an example of a favourable effect of conventional camber on a 3% thick delta wing of aspect ratio 3. In this example there is no twist, and the camber is applied by drooping the forward 25% of the chord to give a mean line having ordinates two-thirds of those of the NACA 230 mean line, giving a maximum camber of 0.012c at 0.15c. This practically eliminates sharp kinks found in the plane wing C_m - C_L curves at lift coefficients of 0.5 - 0.6 for Mach numbers between 0.76 and 0.96. It also

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* But see below on the effects of moderate amounts of camber, which can have a beneficial effect both on the subsonic stability characteristics and on max. L/D at supersonic speeds.

increases maximum L/D at all Mach numbers between 0.76 and 1.36 (the complete test range), the increase being from 12.9 to 14.0 at M = 0.9 and from 9.4 to 9.6 at M = 1.3. The beneficial effect of camber was present throughout the test Reynolds number range of $2\frac{1}{2} \times 10^6$ to $5\frac{1}{2} \times 10^6$.

It appears from these results, and others, that for aspect ratios of the order of 2 to 3 it may be possible to eliminate local kinks in the pitching-moment characteristics, if found, by the use of camber and twist, at the same time increasing maximum L/D at subsonic and moderate supersonic Mach numbers. It should be noted, however, that the data do not extend to M = 2, and the gain in L/D is decreasing with Mach number in the examples considered, so that it is possible that at M = 2 there might be some loss in maximum L/D, at any rate on the configurations tested.

Camber and twist can also have a beneficial effect on the more persistent loss of stability discussed above, although in the few examples available the effect has not been completely eliminated by these means. Fig.9 shows the effect on a 60° swept wing, of aspect ratio 1.9 and small taper, of twist and camber for uniform loading at $C_L = 0.25$, M = 1.1. The twist varies from $+3^\circ$ at the root to -1.3° at the tip, and the maximum camber is 0.8% at the root, 1.2% at the tip, and occurs at about 40%c. The wing has NACA 64A sections normal to the quarter-chord line, the streamwise thickness varying from 0.06c at the root to 0.045c at the tip. The applied camber and twist increase the C_L at which loss of stability occurs by 0.2, from $C_L = 0.6$ to $C_L = 0.8$, but do not eliminate it. These tests were made at $R = 2 \times 10^6$, M = 0.15.

Fig.10 gives another example of partial improvement in the low-speed behaviour due to camber and twist. The wing in this case is a 5% thick delta of aspect ratio 4. The camber and twist, of the NACA conical type, were designed to give a trapezoidal span loading at $C_L = 0.35$, M = 1.15, in attached flow, and were rather large in amount (see Ref.4 for details; it is difficult to express the mean surface in terms of conventional chordwise camber and twist). At M = 0.25, $R = 8 \times 10^6$, the camber and twist do not prevent a loss of stability taking place near $C_L = 0.5$, but they considerably reduce its magnitude.* The effects on maximum L/D are less favourable than in the examples of Figs.7 and 8. At Mach numbers between M = 0.85 and M = 1.7 ($R = 3 \times 10^6$) the effect is adverse, maximum L/D being reduced from 16 to 15 at M = 0.9 and from 5.7 to 5.5 at M = 1.7. There is an increase of the order of 2 in maximum L/D below M = 0.8. As discussed in Ref.4, and as might be expected on general grounds, the conical camber and twist giving an elliptical span loading, as in the example of Fig.7, is to be preferred to that of Fig.10 giving a trapezoidal span loading.

Several examples have been quoted where camber and twist have had a favourable effect on the pitching-moment behaviour in separated flow conditions. In these examples the camber and twist were designed assuming unseparated flow and with the object of improving the characteristics at low incidence. It might be expected that if they had been chosen specifically to improve the behaviour in separated flow at high incidence more favourable results would be obtained for this condition, perhaps at some penalty in drag at low incidence, but at present no methods exist

* Tests at a Reynolds number of only 1.5×10^6 on the same wing showed hardly any gain due to camber and twist, indicating the need for caution in interpreting tunnel tests on this question at low Reynolds numbers. It is possible that the improvement due to camber and twist on the swept wing of Fig.9 would be greater at higher Reynolds numbers.

whereby the required camber and twist distribution can be estimated for such a case, nor is it clear what the basis of design should be. In view of the improvements obtained from camber and twist not so designed, further work on this aspect might be profitable.

3.22 Buffeting

It has been seen that with a suitable wing design it is possible to achieve satisfactory pitching-moment characteristics despite the existence of flow separation. There remains, however, the question of buffeting, which may give rise to some problems on wings of the kind under consideration. Little has so far been done on this question. Direct measurements of pressure fluctuations on two-dimensional wings and behind shock waves have been made, and correlation has been found to exist between the rise in trailing-edge pressure on wind-tunnel models and the onset of shock-induced buffeting in flight, but on the thin low-aspect-ratio delta wing the flow conditions are of a different kind which will necessitate separate investigation. Indirect methods, such as the measurement of mean trailing-edge pressures, or in fact of any mean pressures, can hardly be expected to yield much information on the likelihood of buffeting occurring as a result of the complex flow found on thin highly-swept wings, involving separation from the leading-edge and reattachment further aft over the top of a spiral vortex sheet. Direct measurements of fluctuating pressures at various points on the wing surface will be essential in order to determine whether this type of flow gives rise to serious buffeting, and if it does, whether the buffeting can be avoided by suitable design changes. In the absence of such tests there can be no assurance that thin sweptback and delta wings will be capable of flying satisfactorily in the separated-flow conditions which exist at quite small lift coefficients.

3.23 Wing-mounted fins

On canard layouts, which have been suggested for some $M = 2$ designs, the possibility arises of the fins being mounted at or near the wing tips. There is some danger of adverse interference on lift and pitching-moment in this position, due to interference between the fin and the wing leading-edge flow separation and vortex sheet. Little information is available, but what there is indicates the need for an experimental check of the effects of fins near the wing tip for any design on which such an arrangement is proposed. Two examples of what can happen are shown in Figs.11 and 12. On a delta wing of aspect ratio 2.31 with a 10% thick biconvex section, Fig.11 shows that the addition of fins at 0.75, 0.60 or 0.45 of the semispan caused earlier and/or more severe loss of stability and reduced $C_{L_{max}}$ compared with the wing without fins. On a cropped delta wing of aspect ratio 2 with a 3% thick round-nosed biconvex section, Fig.12 shows how the addition of tip fins introduced kinks in the C_m - C_L curves near $C_L = 0.3$ at Mach numbers between 0.6 and 0.9 where none occurred on the plain wing.

4 Drag due to lift

This paragraph deals with the drag due to lift, and its variation with Reynolds number and Mach number, over the range of C_L for which dC_D/dC_L^2 is approximately constant. In most cases examined it has been found that this range extends up to lift coefficients of the order of 0.20 - 0.25 at low Mach number and up to about 0.5 at supersonic Mach numbers, despite the fact that flow separations probably occur at lower lift coefficients particularly for low Mach number and for sharp-nosed sections. These ranges generally include the lift coefficient for maximum L/D and the results are therefore of direct practical interest. At higher lift coefficients analysis is difficult due to the absence of a simple parameter such as dC_D/dC_L^2 describing the drag due to lift.

On the type of wing under consideration the drag due to lift is of course generally greater than the minimum value of the induced drag, C_L^2/A, for attached flow and elliptic loading, partly because the span loading may be far from elliptical but also because of the reduced values of tangential force ("leading-edge suction force" in linear theory) associated with sharp leading edges, supersonic* leading edges, or subsonic rounded leading edges with flow separation either at the leading edge or further aft. Because of the importance of the latter effects the drag due to lift is often discussed in terms of the proportion of the linear-theory leading-edge suction force realized in practice, and an approach on these lines is used here. The concept of leading-edge suction force is of course entirely abstract when applied to a wing of finite thickness, the real force in such a case being the tangential force obtained by integrating the chordwise components of pressure over the whole surface and not simply at the leading edge. However, whether the force in question is regarded as the idealized leading-edge force of linear theory or the real force distributed over the surface, its effect on the resultant overall force vector is the same, namely to incline it forward of the normal to the chord. Thus the angle θ, say, between the resultant force and the normal to the chord, excluding the effects of skin friction, represents a variable which is equally valid both in linear theory and in practice, and which gives a direct indication of how much leading-edge suction force or tangential force has been achieved. When θ is zero the tangential force (in practice) or the leading-edge suction force (in linear theory) is zero, and the greater the value of θ at a given value of C_L, the greater is the equivalent leading-edge suction force achieved.

Hall, in Ref.4, uses the parameter θ/α (α = incidence) as a measure of the equivalent leading-edge suction force realized experimentally, and a somewhat similar parameter is suggested by Relf in Ref.20 for thin wings. The parameter adopted below is $(1-\theta/\alpha)$. The relation of this parameter to the measured value of dC_D/dC_L^2, or to the "induced drag factor", K, used in the past on thicker unswept wings, is simple. Thus, if ΔC_D is the increase in drag coefficient over its value at zero lift, then for plane wings at small incidences we have

$$\Delta C_D = C_L(\alpha-\theta) \quad \text{by geometry}$$

i.e. near zero lift

$$\underline{\frac{dC_D}{dC_L^2} = \frac{(1-\theta/\alpha)}{a}} \qquad (1)$$

where $a = \dfrac{dC_L}{d\alpha}$ near zero lift.

The "induced drag factor", K, is defined by

$$K = \pi A \frac{\partial C_D}{\partial C_L^2}$$

so that, using (1),

$$\underline{K = \frac{\pi A}{a}(1-\theta/\alpha)} \qquad (2)$$

---

* A leading-edge is described as supersonic or subsonic according to whether it lies ahead of or behind the Mach cone from its most forward point.

Now, if K' is the value of the "induced drag factor" corresponding to complete loss of the equivalent linear-theory leading-edge suction force, i.e. to $\theta/\alpha = 0$, then

$$K' = \frac{\pi A}{a} \quad \text{from eqn. (2)}$$

so that

$$\frac{K}{K'} = (1-\theta/\alpha) = a \cdot \frac{dC_D}{dC_L^2} \qquad (3)$$

Thus, the parameter $(1-\theta/\alpha)$ gives the ratio of the measured "induced drag factor" to that which would be found if there were a complete loss of equivalent leading-edge suction force, and in the following discussion the parameter will in fact be referred to simply as K/K'.

It should be noted that the value of this parameter takes account of all effects on the chordwise pressure force, and not simply those effects arising from flow separations, leading-edge shape etc. That is to say, it includes the classical induced effect whereby θ is reduced from its value of α in two dimensions to $(\alpha - C_L/\pi A)$ for finite aspect ratio, the effects of non-elliptic span loading and non-linear lift in causing K to depart from unity, and also any variation of form drag with lift.

In deriving the values of K/K' or $(1-\theta/\alpha)$ from the experimental results, the measured values of $dC_L/d\alpha$ (near zero lift) have been used. In most cases these differ only slightly from what would be estimated for low incidences using the simple Helmbold-Diederich relation (Ref.2).

Figs.13 and 14 give results from various sources plotted on the above basis. Fig.13 shows the low-speed results ($M = 0.25$) plotted against Reynolds number and Fig.14 shows the effect of Mach number up to $M = 1.7$.

In Fig.13 the experimental points are somewhat scattered as must be expected in view of the limited nature of the analysis and the wide range of wings and test conditions covered, but certain general trends can be seen, as follows:-

(i) At any rate up to $R = 10 \times 10^6$, there is little difference between round-nosed sections having thicknesses less than or equal to 0.03c, and sharp-nosed sections. For both classes of section the drag due to lift is relatively large but except in one case it is always less than that corresponding to a complete loss of equivalent leading-edge suction ($K/K' = 1$).

(ii) Round-nosed sections having thicknesses of 0.05c and more form a separate group having appreciably smaller values of K/K', indicating less severe separation and loss of leading-edge suction than for the thinner or sharp-nosed sections, at any rate up to $R = 10 \times 10^6$.

(iii) There is little consistent effect of wing planform or aspect ratio on K/K'.

(iv) There is a significant reduction in K/K', i.e. increase in leading-edge suction force, as Reynolds number is increased. This is more marked for the thinner sections, for which favourable scale effect exists up to a Reynolds number of at least 16×10^6. The points for the 3% thick round-nosed delta wings of aspect ratio 2, 3 and 4, for instance, lie on a single curve with K/K' falling from 0.78 at $R = 2.7 \times 10^6$ to 0.58 at $R = 16.6 \times 10^6$. The mean line for all the 3% thick and sharp-nosed sections follows a similar trend, at any rate up to $R = 10 \times 10^6$.

Fig.14 gives some theoretical values of K/K' taken from Ref.4, based on linear theory in the absence of separations or sharp leading edges. At M = 0.25 the theoretical value of K/K' is of the order of 0.3 to 0.4 depending on planform and aspect ratio. The measured values of drag due to lift for the thin and sharp-nosed sections at $R = 10 \times 10^6$ are thus of the order of twice the linear-theory values and about three-quarters of that for complete loss of leading-edge suction. For the thicker sections, at high Reynolds numbers, the drag due to lift is only about 30% greater than the linear-theory value. For reference, the actual measured values of the "induced drag factor", K, corresponding to the points of Fig.13, are given in Fig.15, where it can be seen that at $R = 10 \times 10^6$, K is of the order of 1.3 for the thicker sections and between 2 and 3 for the thinner and sharp-nosed sections. In view of the persistence of favourable scale effect up to higher Reynolds numbers on the thinner sections, it can be expected that at full-scale Reynolds numbers the difference between the thicker sections and the thinner or sharp-nosed sections will be less than that found here, but the available data cannot reliably be extrapolated to higher Reynolds numbers. However, the uncertainty as to the full-scale behaviour is not of practical importance, since at a Mach number of 0.25 practical interest is centred on the drag at high lift for take-off and landing rather than on dC_D/dC_L^2 at low incidence; the results in Figs.13 and 15 are presented mainly out of academic interest.

Results of more practical significance, showing K/K' at supersonic and high subsonic Mach numbers over a range of C_L up to about 0.5, and thus including both maximum L/D and manoeuvring conditions, are presented in Fig.14. Except at M = 0.25, the Reynolds number for these data is of the order of 5×10^6, and at supersonic speeds particularly, variation of Reynolds number up to this value had only a small favourable effect on K/K' in all cases.

The main trend apparent in Fig.14 is the expected one of K/K' approaching more closely the value (unity) for complete loss of leading-edge suction as Mach number increases. At M = 1.2 the measured values of K/K' lie between about 0.85 and 1.0, and at M = 1.7 they lie between about 0.90 and 1.05, for all planforms and wing sections tested. In other words, over a fairly wide range of wing planform and section shapes, the equivalent leading-edge suction force realized is relatively small. For an unswept or sharp leading edge at supersonic speeds a complete loss of equivalent leading-edge suction is of course predicted by linear theory. For a subsonic round-nosed leading edge, however, as for the delta of aspect ratio 2 having 63° of sweepback on the leading edge, the results at low supersonic speeds are less favourable than has often been assumed, based partly on linear theory. The experimental results are, it is true, more favourable for swept than for unswept leading edges, but the amount of improvement due to sweepback is considerably less than would be expected theoretically. Moreover, the improvement seems to be present for sharp-nosed sections as well as round-nosed ones.

The theoretical curves for round-nosed delta wings of aspect ratios 2, 3 and 4 are shown in Fig.14, and it can be seen that these bear no resemblance to the experimental results. On the basis of the data presented, a more accurate answer for wings of 45° sweep or more, whether round-nosed or sharp-nosed, is obtained by assuming that K/K' varies from about 0.90 at M = 1.2 to about 0.95 at M = 1.7 and probably about 1.0 near M = 2.0. A similar conclusion is reached by Hall in Ref.4, where it is stated that it is more accurate to assume complete loss of the equivalent leading-edge suction force than to assume the value given by linear theory. There may be some favourable scale effect between $R = 5 \times 10^6$ and full-scale Reynolds numbers, but judging from the results up to $R = 5 \times 10^6$ this should not be large.

It will be noted that for the unswept wings the experimental results correspond to a drag due to lift somewhat larger than the maximum usually assumed ($K/K' = 1$, complete loss of equivalent leading-edge suction).

5 Conclusions

(i) For sweptback and delta wings having moderately small aspect ratios (i.e. 2 to 3) and thin sections (e.g. 0.03c) the effects of flow separation on the wing-alone pitching-moment behaviour are considerably less severe than for current subsonic designs. "Pitch-up" does not occur, and the transition from subsonic to supersonic speeds is generally accomplished smoothly and without serious irregularities in the C_m - C_L curves. On a 3% thick delta wing of aspect ratio 2, in particular, the C_m - C_L curves were found to be practically linear throughout the Mach number range up to $M = 1.7$. At higher aspect ratios (e.g. 3) or with thicker sections (e.g. 5% round-nosed) local pitching-moment kinks are sometimes found at moderate subsonic Mach numbers. However, some success has been achieved in eliminating such kinks by the use of appropriate camber and twist, which can at the same time improve the lift-drag ratio, at any rate up to $M = 1.7$.

(ii) For unswept thin wings the pitching-moment characteristics are less favourable. At low Mach numbers there is a large increase of stability between low and high incidences, and the transition from subsonic to supersonic speeds is less smooth than for sweptback wings.

(iii) At supersonic speeds up to $M = 1.7 - 1.8$, there appear to be no serious effects of flow separation on pitching moments, the C_m - C_L curves being more or less linear for all the planforms considered.

(iv) Care will be needed in positioning either a rear tail or a canard foreplane, and the placing of vertical fins near the wing tips on sweptback wings will also require attention.

(v) Although the lift and pitching-moment characteristics may appear acceptable despite the existence of flow separation, it is possible that separation may set a limit in the form of buffeting, and direct experimental investigation of this question on the type of wing under consideration is likely to be needed.

(vi) Drag due to lift seems to be little affected by section shape or type of planform at supersonic speeds. For all cases investigated, including those with subsonic rounded leading edges, the results are close to those corresponding to complete loss of the equivalent linear-theory leading-edge suction force.

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FIG. 1.

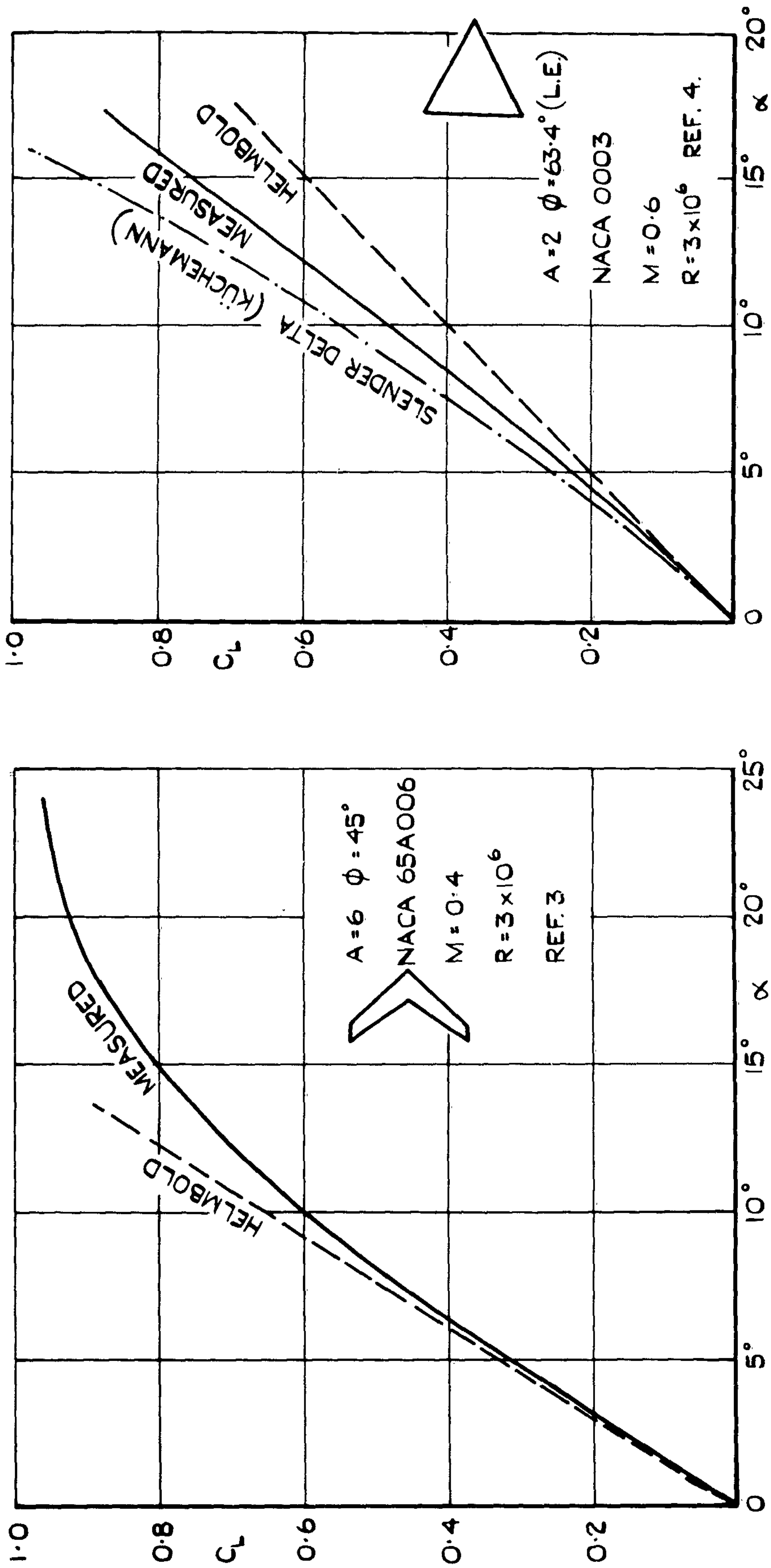


FIG. 1. C_L vs. α FOR TWO TYPES OF PLANFORM.

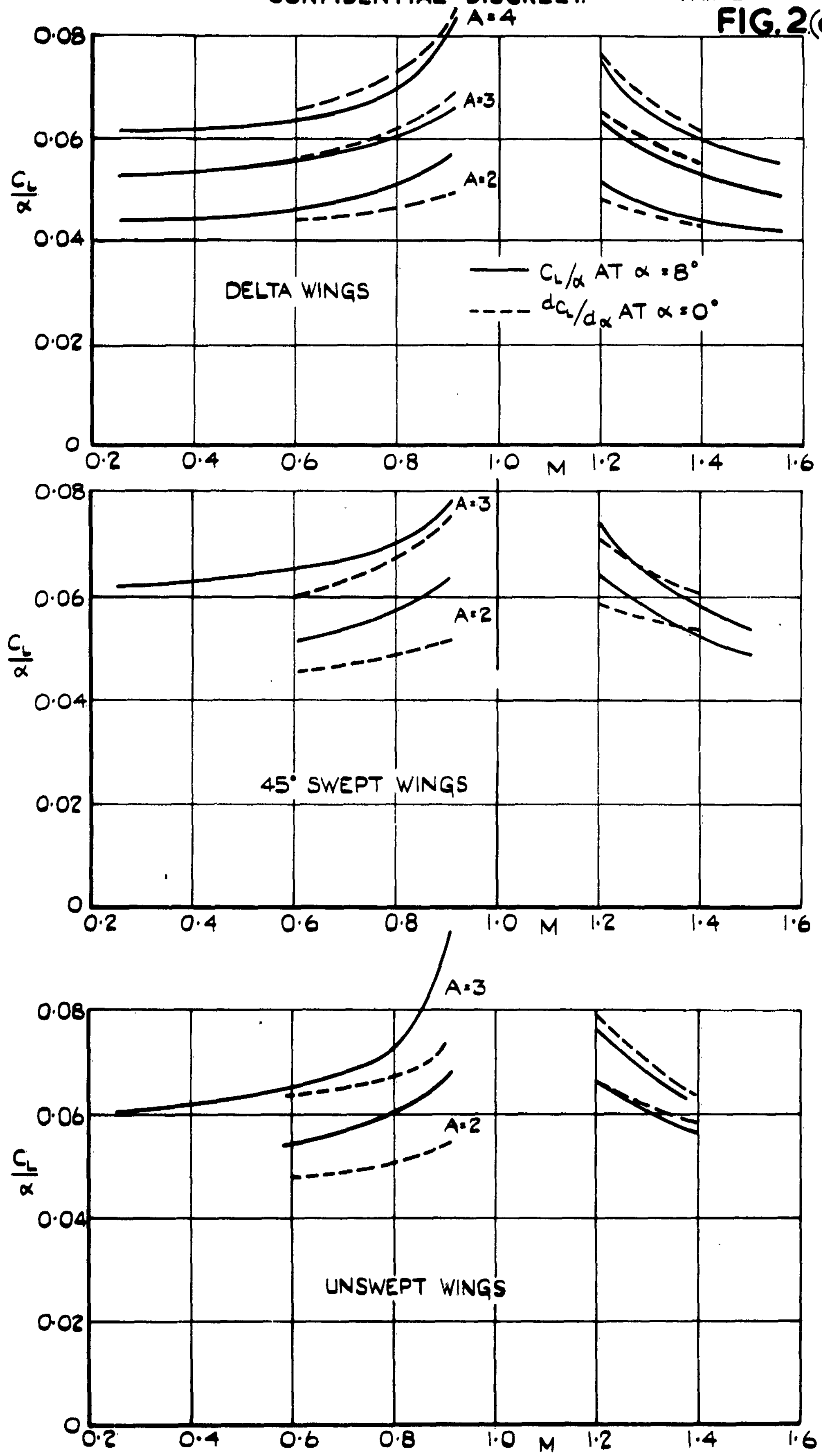
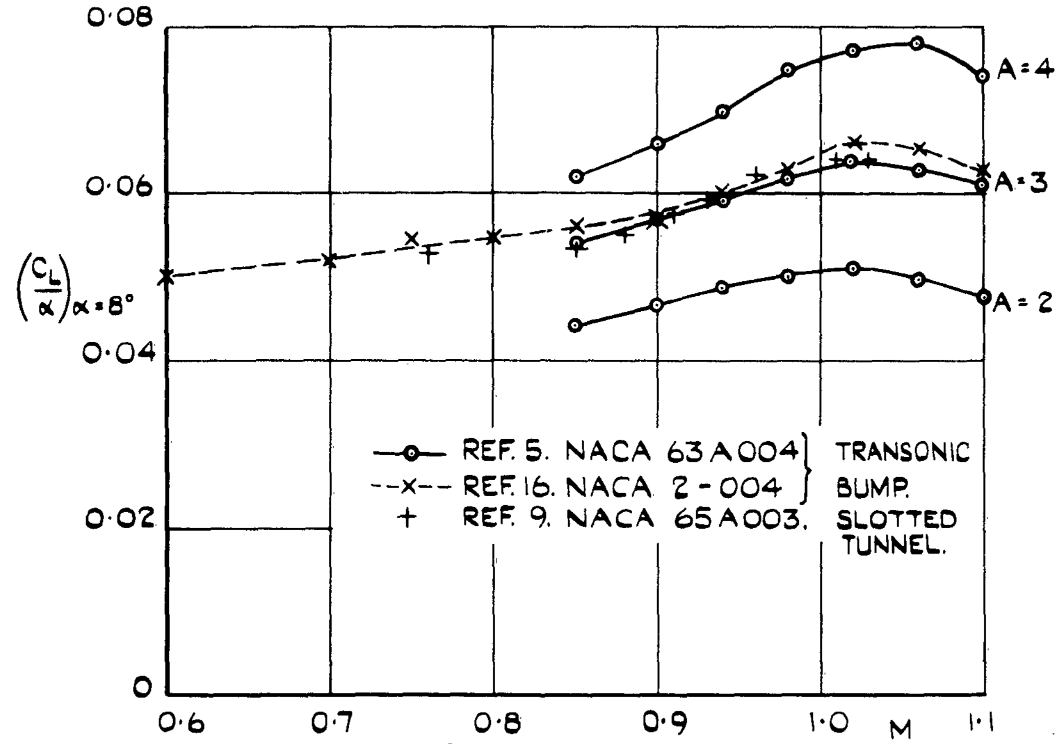
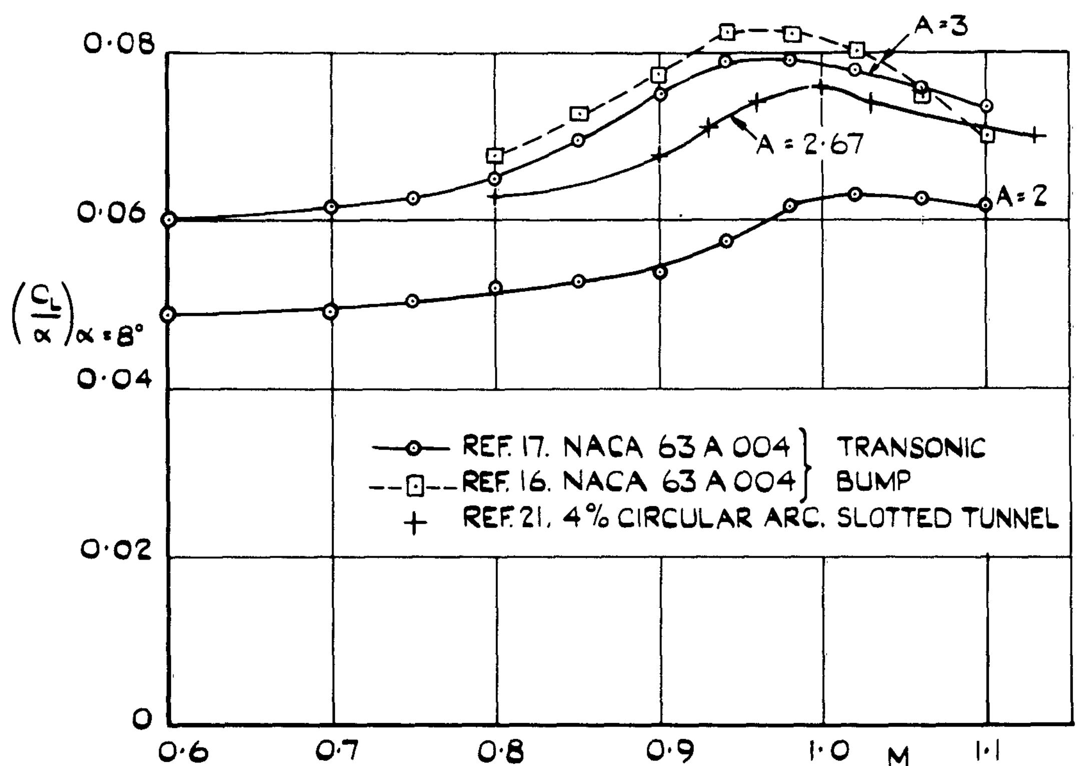


FIG. 2.(a). VARIATION OF C_L/α WITH MACH NUMBER. REF. 4, t/c = 0·03.

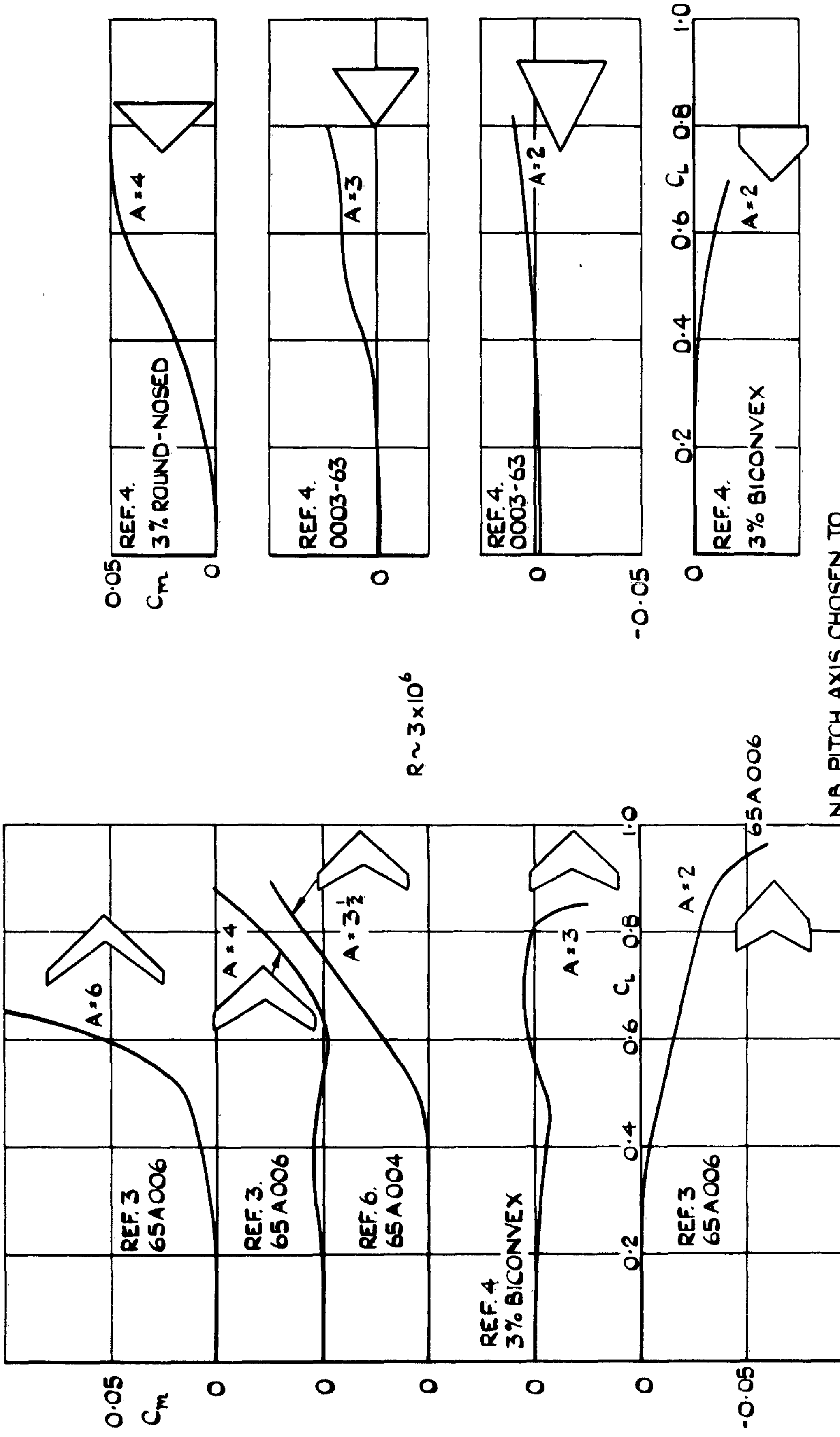


(i) DELTA WINGS.



(ii) UNSWEPT WINGS.

FIG. 2(b). VARIATION OF C_L/α WITH M. TRANSONIC RESULTS.



(a) 45° SWEPT WINGS.

(b) DELTA & CROPPED DELTA WINGS.

N.B. PITCH AXIS CHOSEN TO GIVE NEUTRAL STABILITY AT ZERO C_L IN EACH CASE.

FIG. 3.(a&b). $C_m - C_L$ CURVES AT $M = 0 \cdot 6$. EFFECT OF ASPECT RATIO.

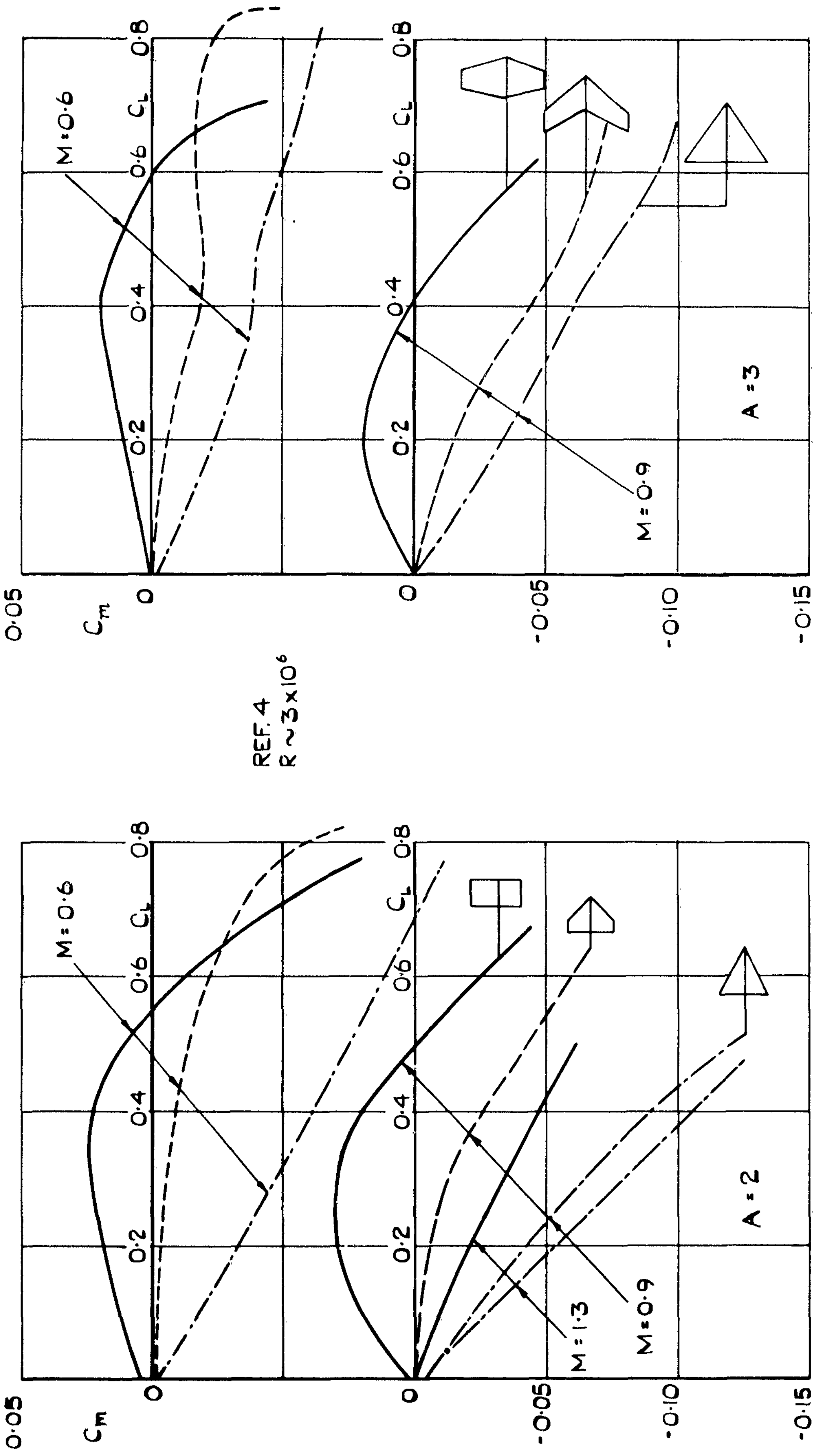


FIG. 4 (a) $C_m - C_L$ CURVES. EFFECT OF PLANFORM SHAPE. 3% THICK WINGS.

FIG. 4 (b).

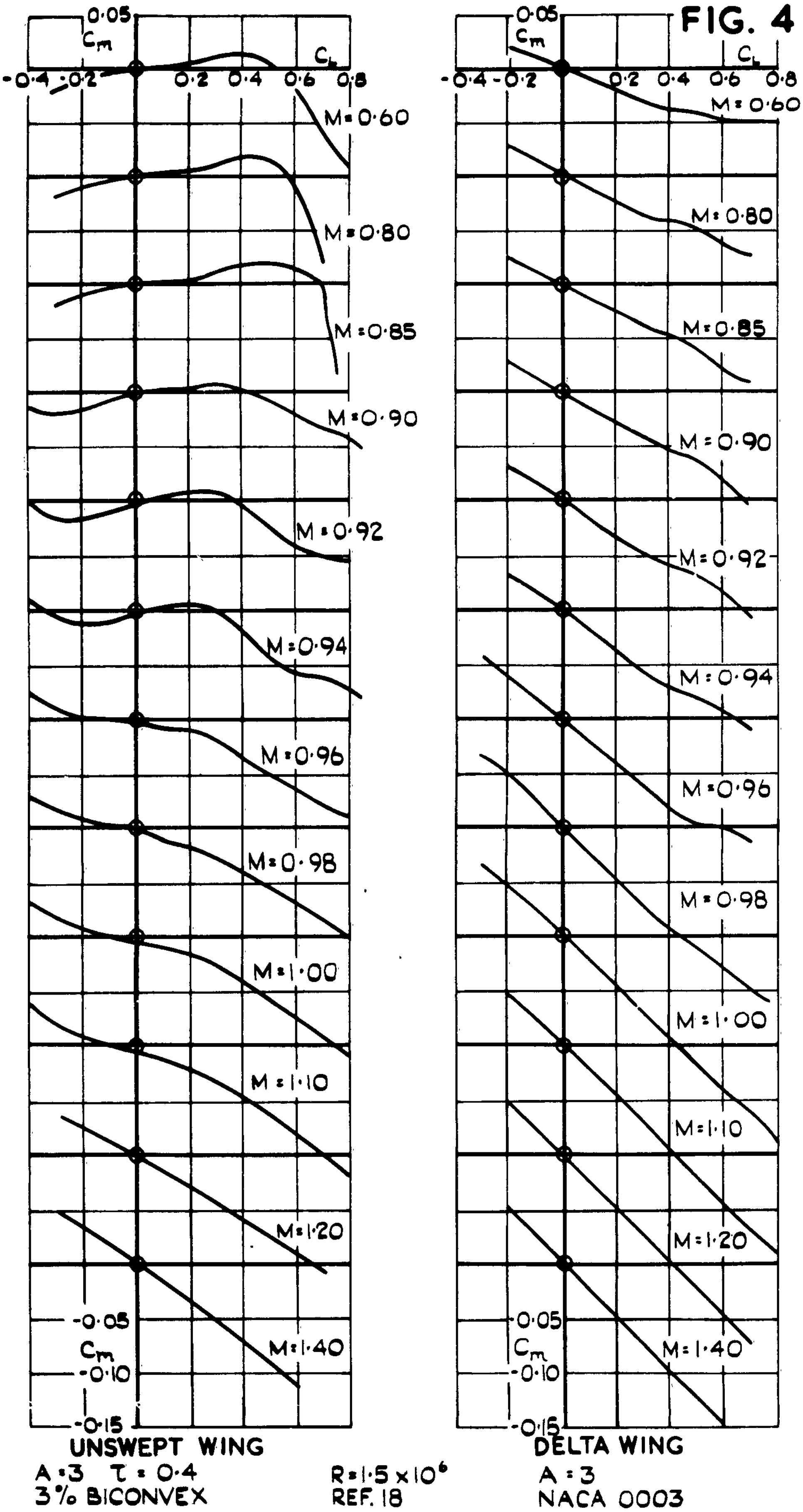
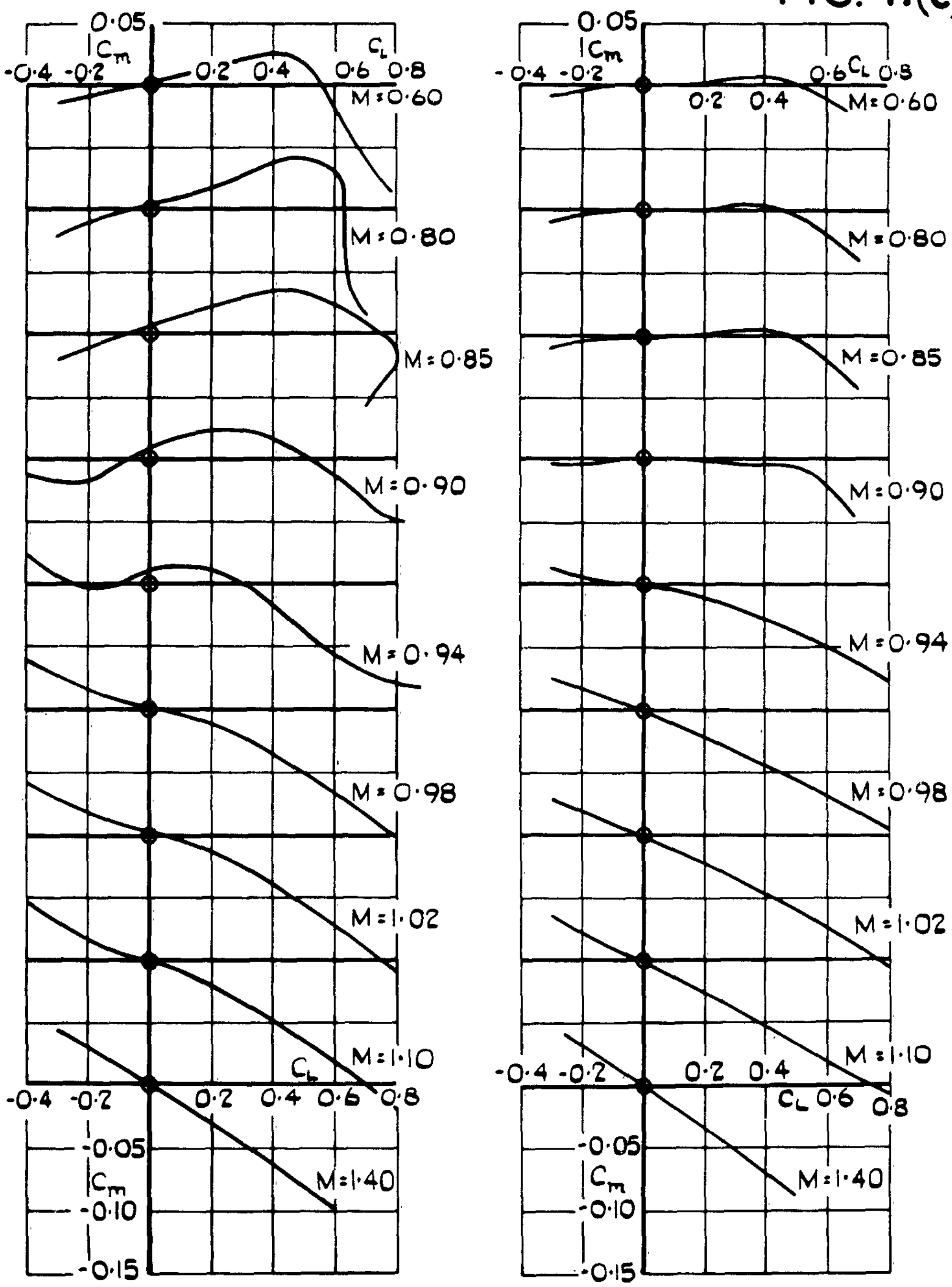


FIG. 4. (b) COMPARISON OF $C_m - C_L$ CURVES FOR 3% THICK UNSWEPT & DELTA WINGS OF ASPECT RATIO 3.



(i) TAPER RATIO = 0·5.

(ii) TAPER RATIO = 0

NACA 64 A 003 SECTIONS
$R = 1{\cdot}5 \times 10^6$
REF. 19.

FIG. 4.(c). EFFECT OF TAPER RATIO ON C_m-C_L CHARACTERISTICS FOR 3% THICK UNSWEPT WINGS OF ASPECT RATIO 3.

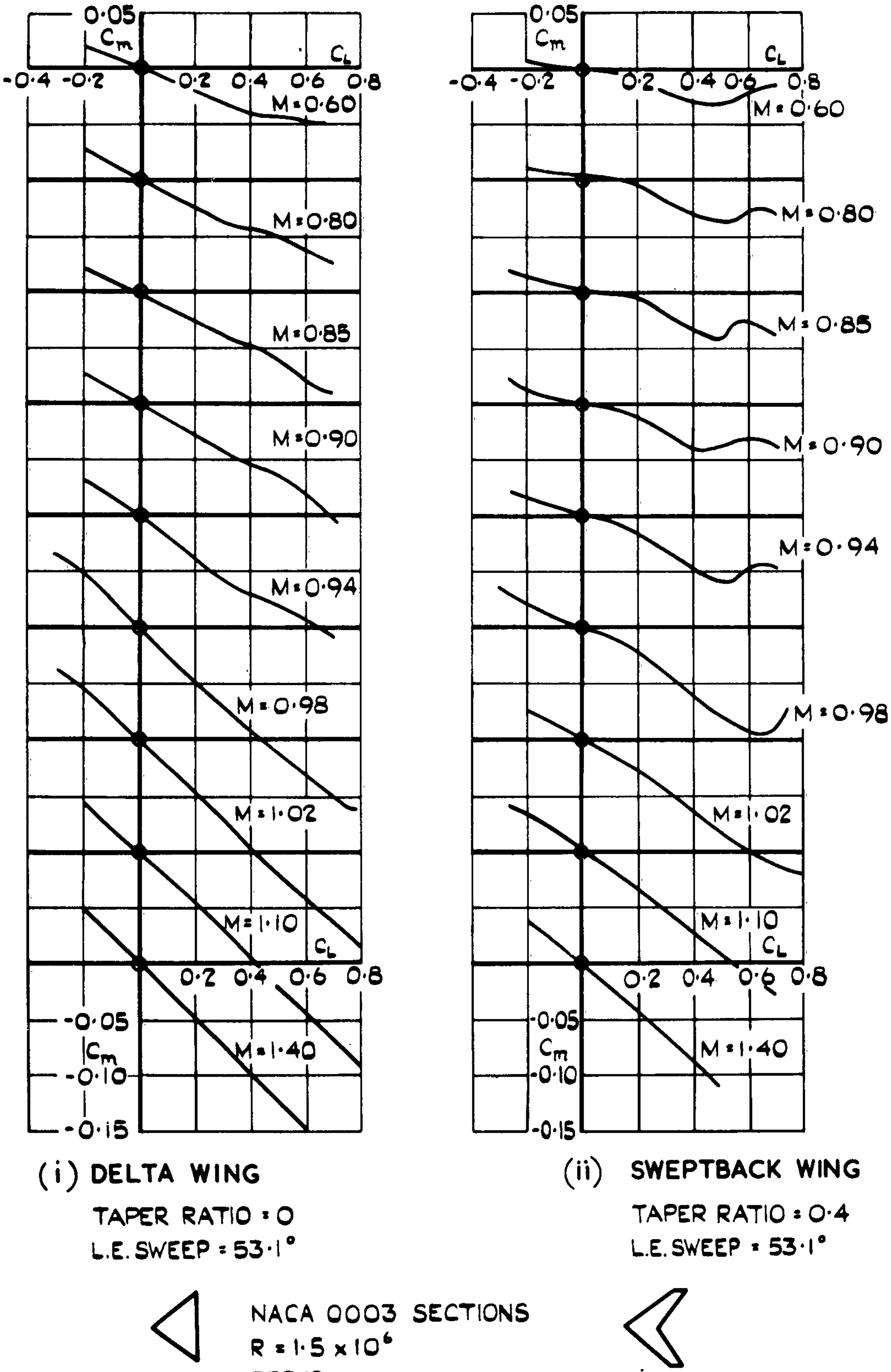


FIG. 4. (d). COMPARISON OF $C_m - C_L$ CURVES FOR 3% THICK DELTA AND SWEPTBACK WINGS OF ASPECT RATIO 3.

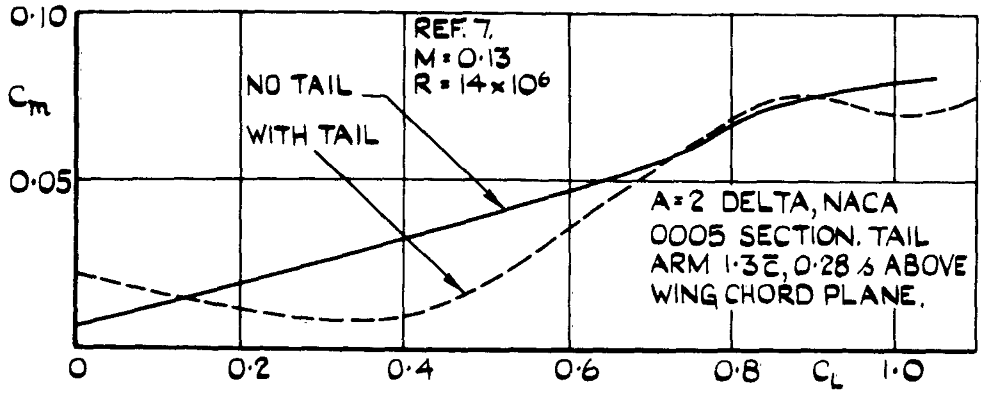


FIG. 5. EFFECT OF BADLY-PLACED TAIL ON C_m-C_L CHARACTERISTICS.

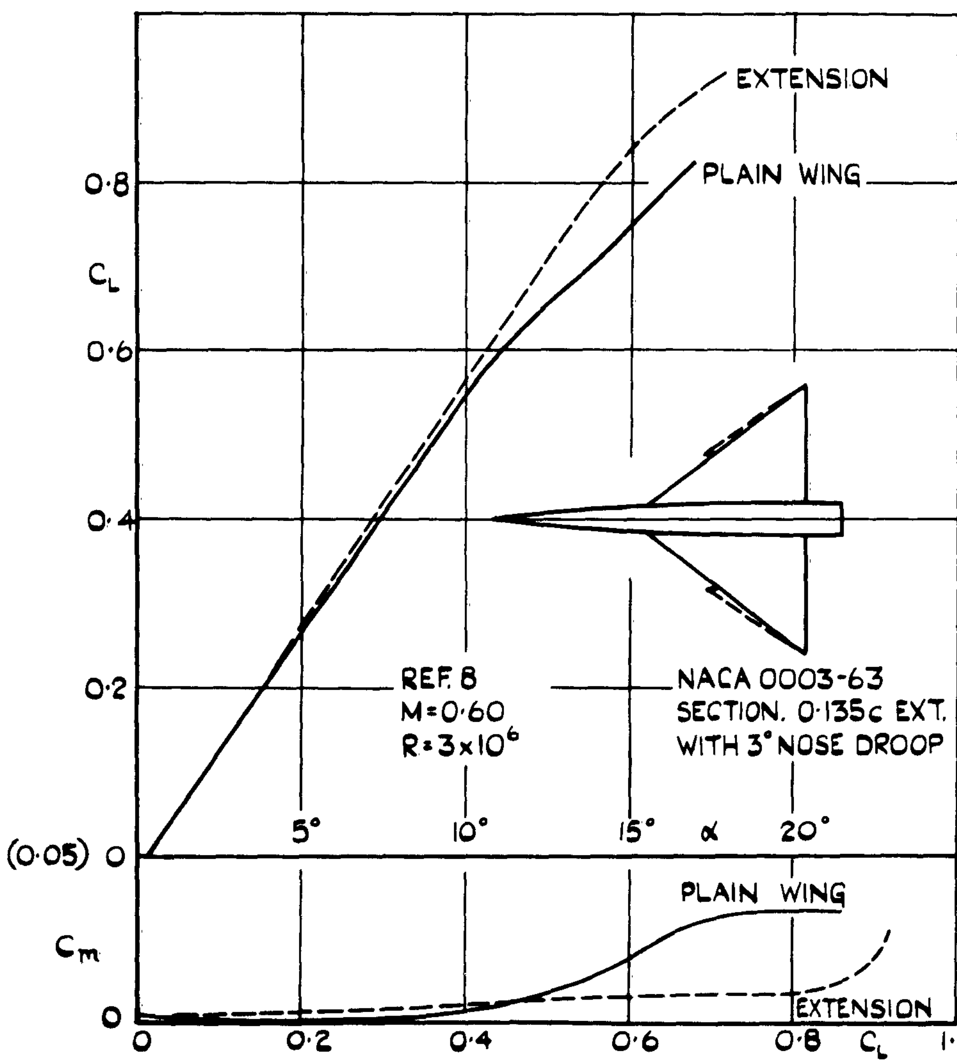


FIG. 6. EFFECT OF CHORD-EXTENSION ON A=3 DELTA.

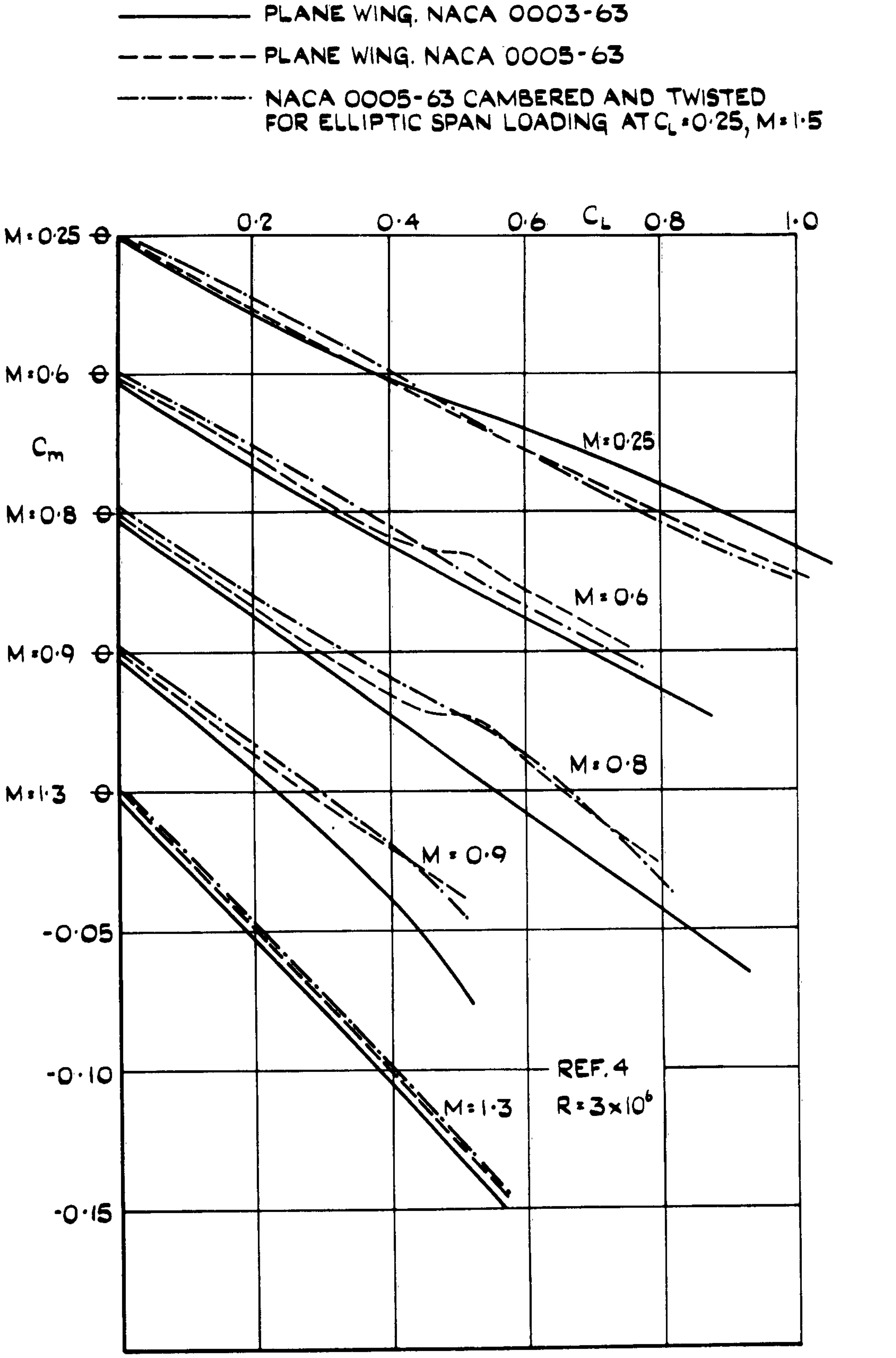


FIG. 7. EFFECT OF CONICAL CAMBER AND TWIST, AND THICKNESS-CHORD RATIO, ON C_m-C_L FOR A DELTA WING OF ASPECT RATIO 2.

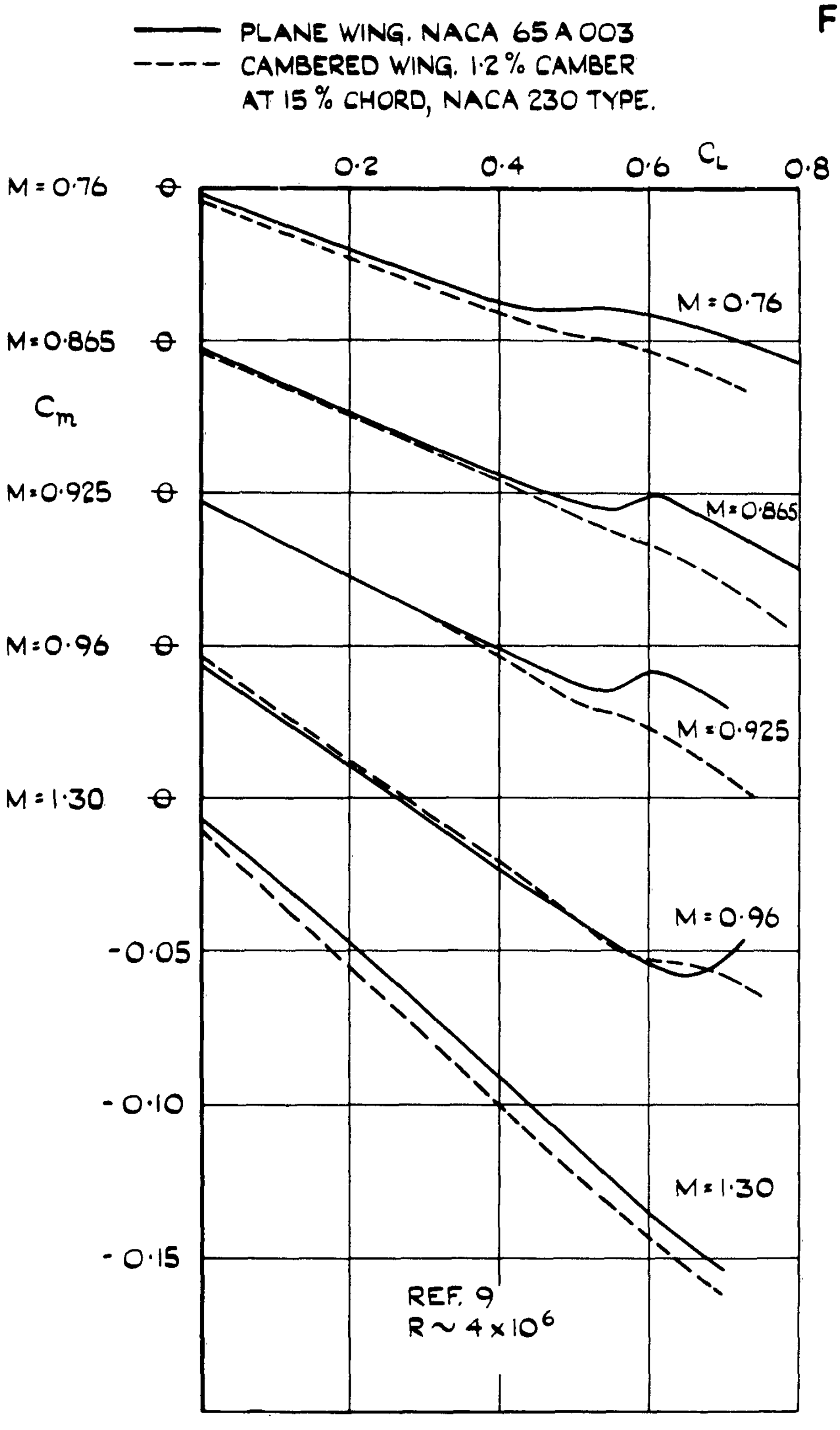


FIG. 8. EFFECT OF 1·2 % CAMBER (NACA 230 TYPE) ON $C_m - C_L$ FOR A 3 % THICK DELTA WING OF ASPECT RATIO 3.

——— PLANE WING.
- - - - CAMBER AND TWIST, CAMBER = 0·8 % AT ROOT, 1·2 % AT TIP. MAX. CAMBER AT 0·40 C. TWIST = +3° AT ROOT, -1·3° AT TIP.

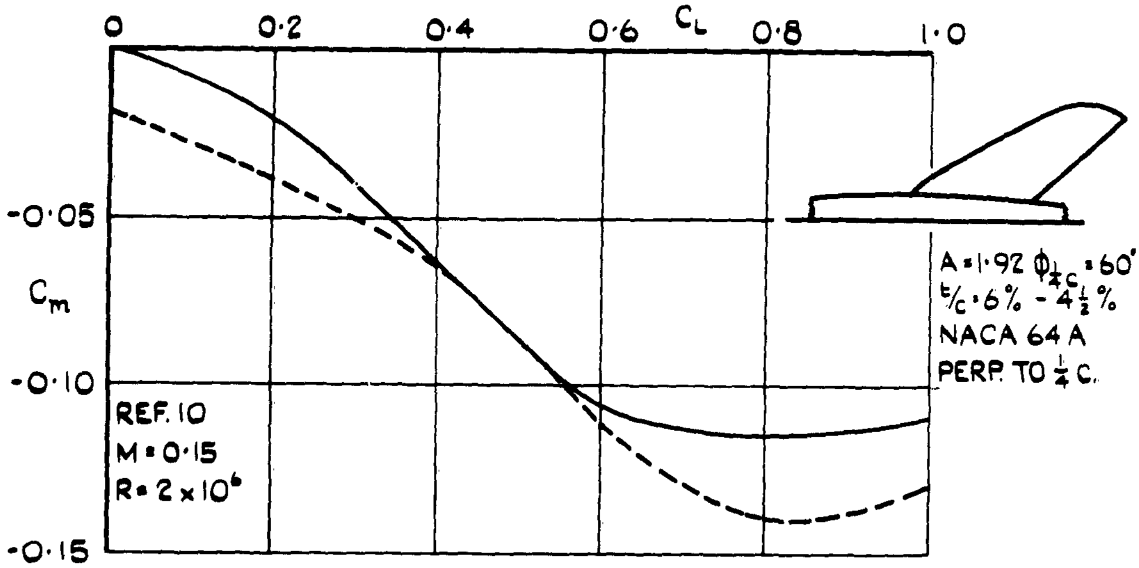


FIG. 9. EFFECT OF CAMBER AND TWIST ON $C_m - C_L$ FOR A 60° SWEPT WING OF ASPECT RATIO 1·92.

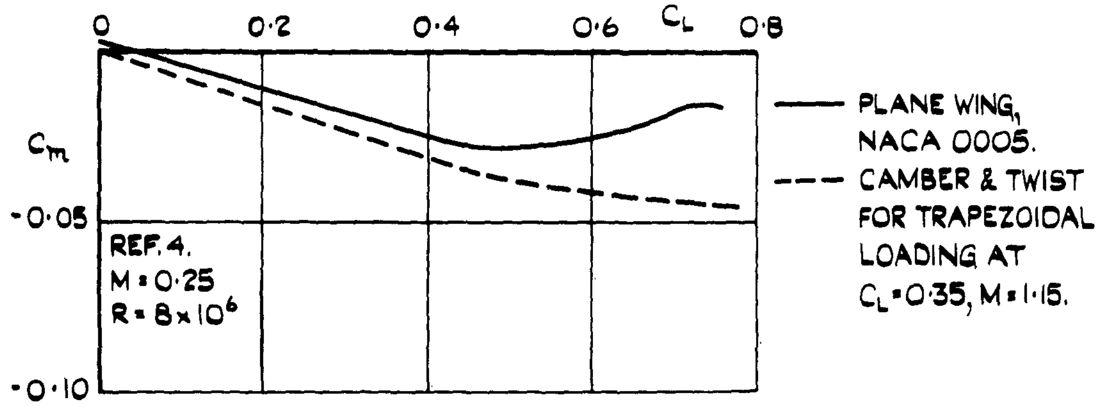


FIG. 10. EFFECT OF CAMBER AND TWIST ON $C_m - C_L$ FOR A DELTA WING OF ASPECT RATIO 4.

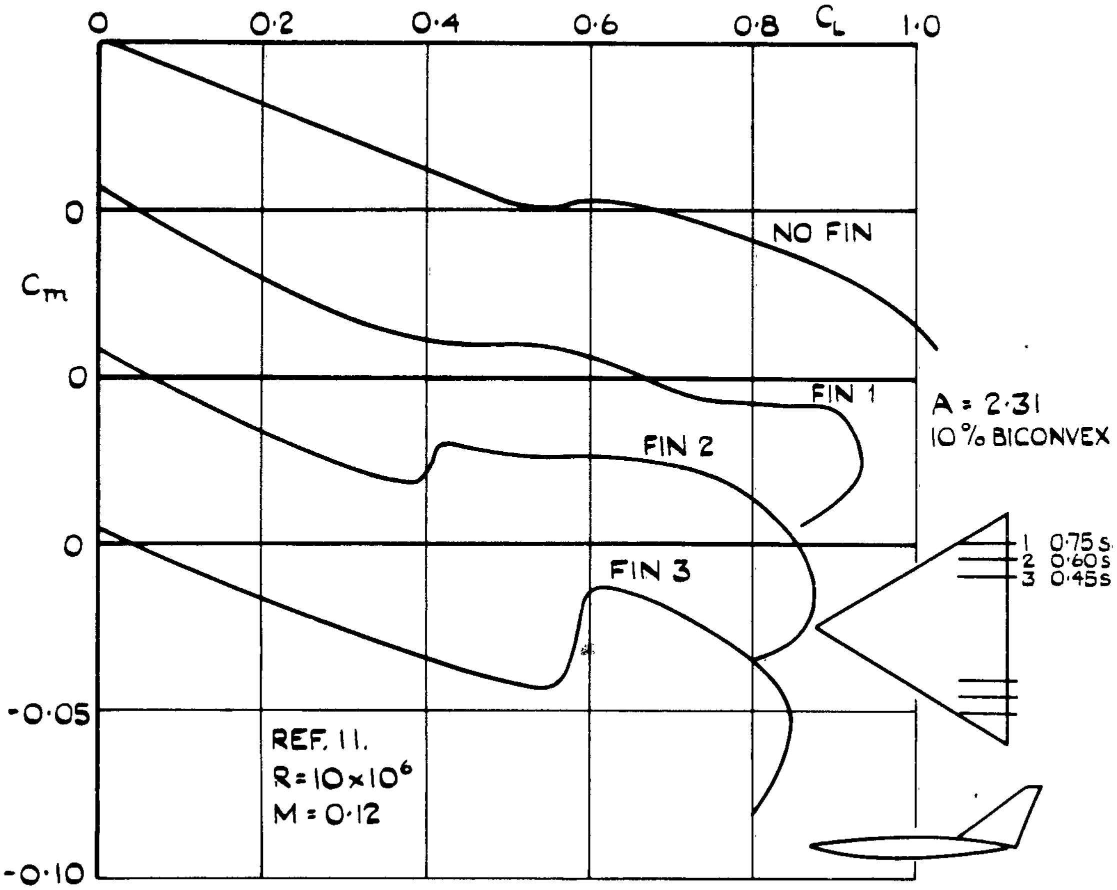


FIG. 11. EFFECT OF WING - MOUNTED FINS ON $C_m - C_L$ FOR A DELTA WING OF ASPECT RATIO 2·31.

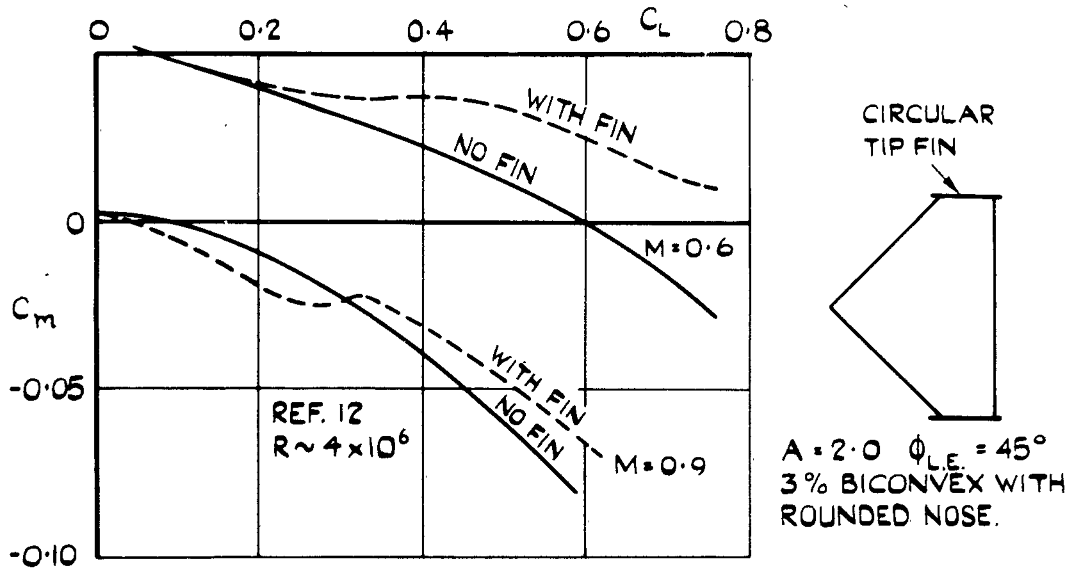
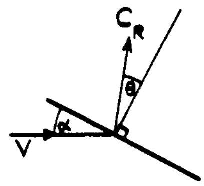


FIG. 12. EFFECT OF CIRCULAR TIP FINS ON $C_m - C_L$ FOR A CROPPED DELTA WING OF ASPECT RATIO 2·0.

SYMBOL	ASPECT RATIO	PLANFORM	SECTION SHAPE	REF.
⊙	2	DELTA	NACA 0003-63	4
×	2	DELTA	NACA 0005-63	4
+	2	DELTA	NACA 0008-63	4
▽	2	45° SWEPT	3% ROUND-NOSED BICONVEX	4
▼	2	45° SWEPT	3% BICONVEX	4
>	2	45° SWEPT	NACA 65 A 006	3
⊡	2	UNSWEPT	3% ROUND-NOSED BICONVEX	4
■	2	UNSWEPT	3% BICONVEX	4
◬	2·31	DELTA	NACA 65 A 002	13
△	2·31	DELTA	NACA 65 A 003	14
<	2·31	DELTA	NACA 65 A 006	13
▲	2·31	DELTA	10% BICONVEX	11
λ	2·31	DELTA	10% ROUND-NOSED BICONVEX	11
▲'	2·31	DELTA	SHARP-EDGED FLAT PLATE	15
⊙'	3	DELTA	NACA 0003-63	4
▽'	3	45° SWEPT	NACA 0003-63	4
▼'	3	45° SWEPT	3% BICONVEX	4
⊡'	3	UNSWEPT	3% ROUND-NOSED BICONVEX	4
■'	3	UNSWEPT	3% BICONVEX	4
⊙''	4	DELTA	3% ROUND-NOSED BICONVEX	4
●''	4	DELTA	3% BICONVEX	4
v	4	DELTA	NACA 0005-63	4
ʌ	4	45° SWEPT	NACA 65 A 006	3



$K = \pi A \times (\text{MEASURED } dC_D/dC_L{}^2)$.

K' = LINEAR-THEORY VALUE FOR ZERO LEADING-EDGE SUCTION $= \pi A \div dC_L/d\alpha$

$$\frac{K}{K'} = \frac{dC_D}{dC_L^2} \cdot \frac{dC_L}{d\alpha} = \left(1 - \frac{\theta}{\alpha}\right)$$

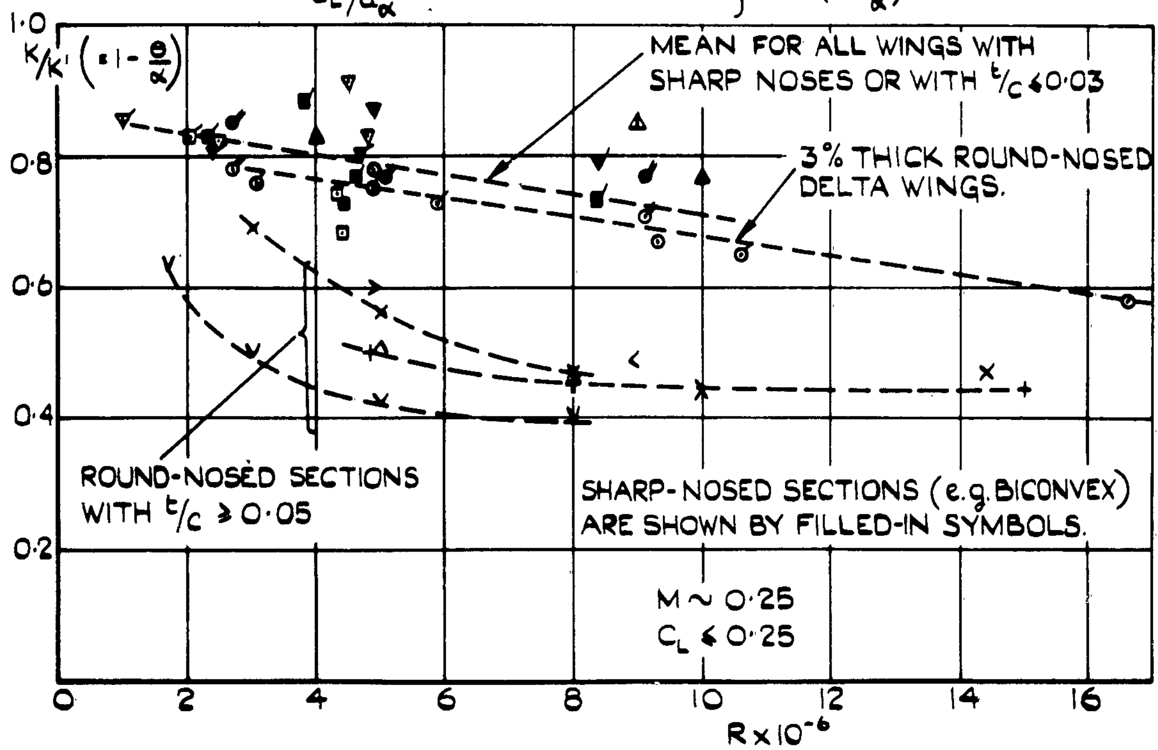


FIG. 13. K/K' vs. REYNOLDS NUMBER FOR VARIOUS WING PLANFORMS & SECTION SHAPES AT LOW MACH NUMBER.

REYNOLDS NUMBER IS ABOUT 5×10^6 FOR MACH NUMBERS OF 0·6 AND ABOVE. AT M=0·25 THE REYNOLDS NUMBER IS THE HIGHEST PLOTTED IN FIG. 13. FOR THE CORRESPONDING POINT.

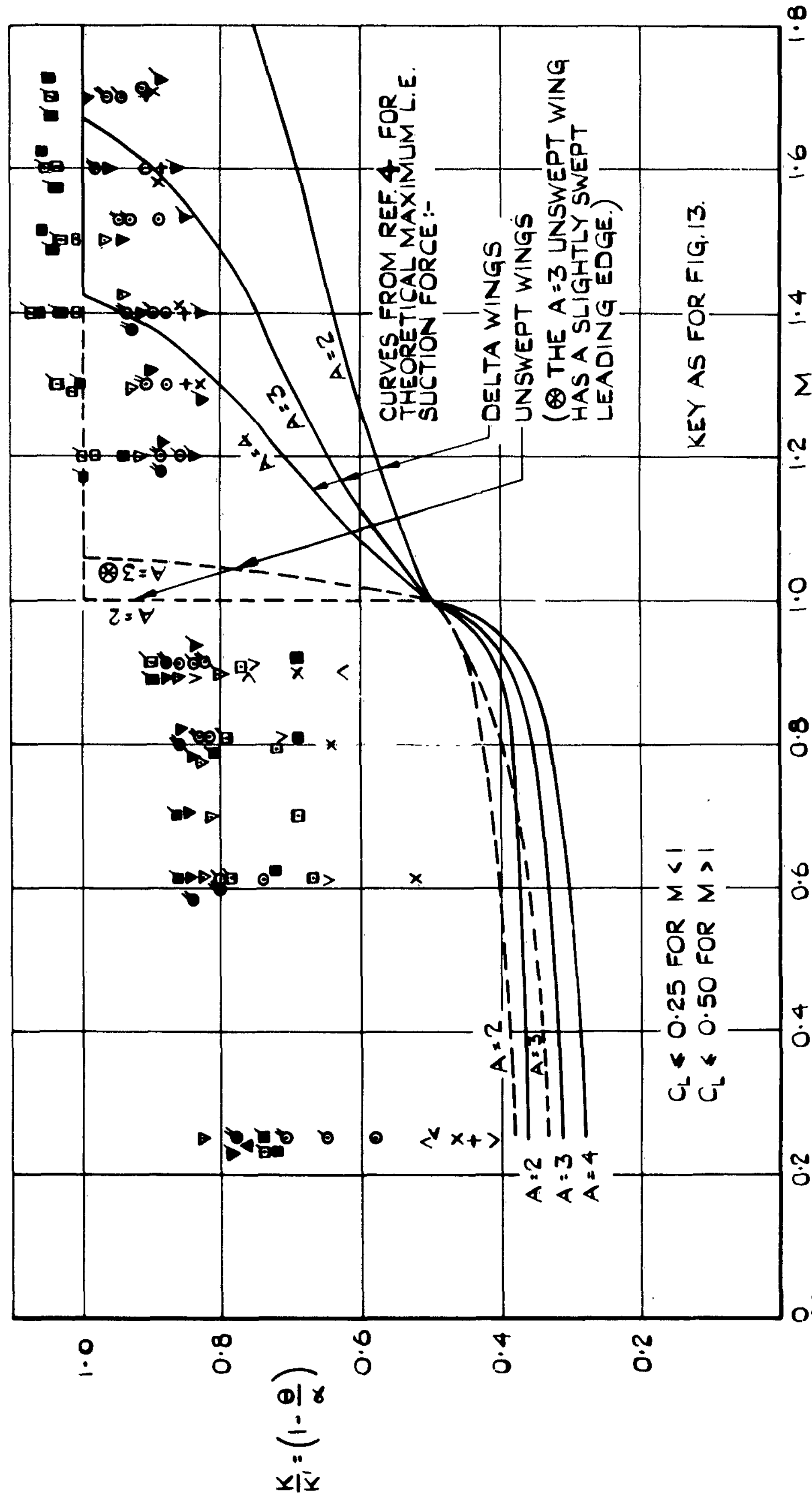


FIG. 14. K/K' vs. MACH NUMBER FOR VARIOUS WING PLANFORMS AND SECTION SHAPES.

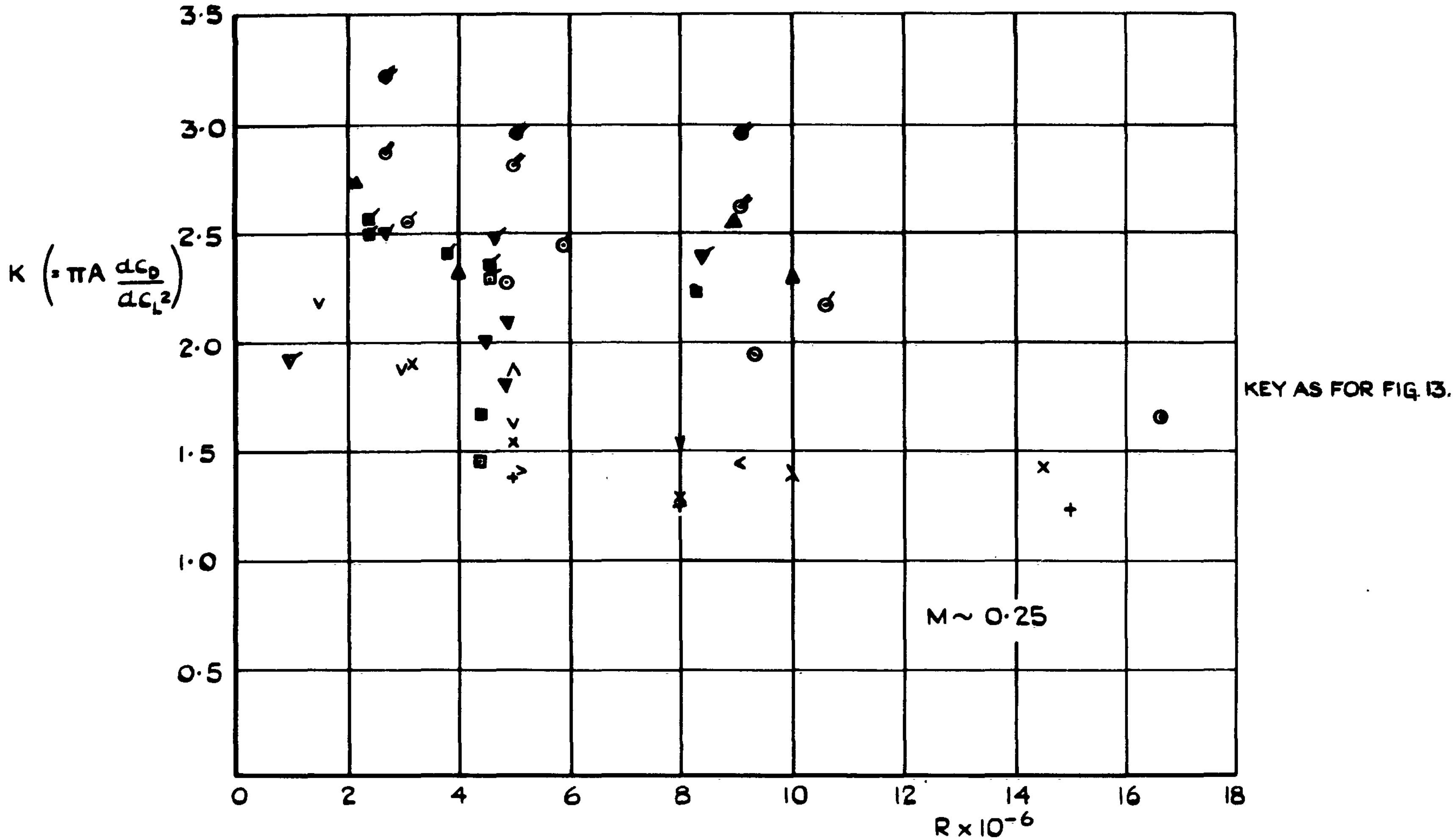


FIG. 15. K vs. REYNOLDS NUMBER FOR VARIOUS WING PLANFORMS & SECTION SHAPES AT LOW MACH NUMBER.

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Date of Search: 5 August 2008

Record Summary: DSIR 23/24149
Title: Effect of flow separation on force characteristics of thin wings (RAE TN AERO 2401)s
Availability Open Document, Open Description, Normal Closure before FOI Act: 30 years
Former reference (Department) ARC 18312
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