C.P. No. 562

C.P. No. 562

MINISTRY OF AVIATION

AERONAUTICAL RESEARCH COUNCIL

CURRENT PAPERS

Calculations of the Lift Slope and Aerodynamic Centre of Cropped Delta Wings at Supersonic Speeds

by

J. H. B. Smith, J. A. Beasley and A. Stevens

LONDON: HER MAJESTY'S STATIONERY OFFICE

1961

THREE SHILLINGS NET

U.D.C. No. 533.693.3

C.P. No.562

July, 1960

CALCULATIONS OF THE LIFT SLOPE AND AERODYNAMIC CENTRE OF CROPPED DELTA WINGS AT SUPERSONIC SPEEDS

by

J. H. B. Smith, J. A. Beasley and A. Stevens

SUMMARY

The linearised theory of supersonic flow is used to calculate the lift slope and the aerodynamic centre of cropped delta wings which have subsonic leading edges. The cropped tip must be short enough for each tip to lie entirely upstream of the disturbances produced by the other. The results are presented graphically and in brief tables.

Previously issued as R.A.E. Tech. Note No. Aero 2697 - A.R.C. 22,200.

LIST OF CONTENTS

LIST OF TABLES

LIST OF ILLUSTRATIONS

1 INTRODUCTION

The results of recent experiments[1,2] at supersonic speeds on wings of small aspect ratio, with unswept trailing edges and streamwise tips, have shown that slender thin-wing theory* may be considerably in error in its predictions of lift and pitching moment. The present calculations of the lift slope and aerodynamic centre of cropped delta planforms according to supersonic linearised theory were undertaken to illustrate one source of the discrepancies.

A number of authors[3,4] have used supersonic linearised theory to study the properties of cropped delta wings when the influence of one tip is not felt by the other. We accept the same restriction and make use of their work. Only the case of subsonic leading edges is considered here, in view of the intended comparison with slender theory. Thus we obtain the properties of any given wing for a range of Mach numbers between that for which the tips no longer interfere and that for which the leading edge becomes sonic. The results of the calculations are presented in tables and charts. Charts of lift-slope and aerodynamic centre position for a 3-parameter family of wings have previously been prepared by Stanbrook[5]. The wide intervals between successive curves necessitated by the range of planforms considered make it impossible to extract the information now required from these charts.

The theory used makes the usual assumptions of small disturbances, negligible viscous effects and flow separation from the trailing edge only. It is known that the flow past a small aspect ratio wing at incidence usually separates from a highly swept leading edge or from a side edge. For an unwarped wing there is a small range of incidence about zero in which the flow is little affected by leading edge separation; this range is very small if the wing is thin, the leading edge sharp and the Mach number of the flow normal to the leading edge is small; it increases with thickness, edge angle and Mach number normal to the leading edge; and, in general terms, it is greater for a wing with a round leading edge. Thus, although the present results are less widely applicable than those obtained from linearised theory for planforms of higher aspect ratio, they provide reasonable estimates for the lift and the position of the centre of pressure near zero incidence for a thin flat wing. Similarly, if a sharp-edged wing is so warped that at some incidence its leading edge is an attachment line and separation ahead of the trailing edge is avoided, there is again a small range of incidence about this "attachment incidence" in which leading edge separation has little effect. The size of the range then depends on the warp distribution as well. The present results can also be used for the slope of the lift curve and the position of the aerodynamic centre in this range for a warped wing. A simple treatment by slender body theory of the separated flow past planforms with continuously curved leading edges is given in Ref. 6: no theoretical treatment of the flow with leading edge separation is yet known which would enable us to calculate the effect of varying Mach number.

The lift and pitching moment of a wing are affected also by its finite thickness. This effect cannot be calculated by a thin-wing theory of attached flow, since such a theory excludes any interaction between lifting and thickness effects. Slender-body theory allows an estimate to be made of the shift of aerodynamic centre due to thickness and the loss of lift due to a non-zero base area for a slender, pointed wing with unswept trailing edge.

---

*Slender thin-wing theory uses simultaneously the assumptions of slender body theory and of thin wing theory. It is the limiting form of slender body theory for vanishing thickness and camber and the limiting form of supersonic linearised theory for vanishing βA.

For wings with diamond cross-sections these effects have been calculated by E.C. Maskell (unpublished). He finds a 10% loss of lift when the trailing edge thickness is 30% of the span. For a delta wing of aspect ratio one, 12% biconvex centre section, (the subject of low-speed tests[7]) the aerodynamic centre is calculated to be 2.6% of the root chord further aft than it would be on a thin wing. The experimental results at small incidence confirm this difference.

Having briefly indicated the deficiencies of the proposed model of the flow, we go on to outline the theoretical treatment of it and to discuss the results.

2 THEORY

The simplest approach to the calculation of the lifting properties of a cropped delta wing by supersonic linearised theory is through the superposition of conical velocity fields. The basic solution is that for the uncropped flat delta wing at incidence and this applies forward of the Mach cones from the leading edge tips (ABDC in Fig.1). This solution implies a load distribution, constant along each ray through the wing apex, which extends beyond the wing tip and must therefore be cancelled. The cancellation is carried out by introducing an elementary solution which produces constant load over the region between a ray from the apex and the tip edge, on the outboard side of the tip; and produces zero load forward of the ray through the apex and zero downwash inboard of the tip (see Fig.1). This elementary solution is conical: its vertex is the intersection of the ray through the apex and the tip edge. By integration of these elementary conical solutions, a solution is formed which cancels by superposition the load produced by the basic solution at points off the planform but leaves the downwash on the wing unchanged. The load distribution induced on the wing by these elementary solutions modifies the basic solution everywhere behind the Mach lines BD and CD from the leading edge tips, the effects of the two tips being additive in the region behind both Mach lines. The construction of the cancellation solution in this way is no longer possible as soon as the influence of the port tip is felt by the starboard tip; since then the load distribution off the wing which is to be cancelled is no longer constant along rays through the wing apex. Although there is no difficulty of principle involved in carrying the solution past this point by using higher order conical fields the work involved would be extensive and the solution along these lines has not been constructed.

The solution for the case of non-interfering tips has been found, by Cohen[3], for both supersonic and subsonic leading edges. She uses the method outlined above and obtains expressions for overall lift and pitching moment as integrals of algebraic functions. These can be expressed in terms of incomplete elliptic integrals of the third kind, but it is more convenient for calculation purposes to evaluate them directly by numerical integration. In this way the results of the present paper, which are for subsonic leading edges only, have been found. Details of the calculation are given in the Appendix. The results appear in Tables 1 and 2. The independent variables chosen for this tabulation are such that linear interpolation in both directions introduces errors not exceeding 1% of the tabulated function. However, they do not show the changes in lift slope and aerodynamic centre which occur as a given wing is taken through a range of Mach number. These are shown in Figs.2 and 3, in the form of plots against $\beta \cot \Lambda$ for each of a range of values of the taper ratio, λ.

A similar derivation from conical field theory has been used by Gilles[4] to calculate the load distribution over a cropped delta wing at five Mach numbers between that at which the tips last interfere and that for which the leading edge is sonic. These calculations have now been slightly extended so that at each Mach number the load has been calculated on a wing continued rearward until its tips interfere. The load distribution on a shorter wing is obviously the same as that on the forward part of the longer wing. The isobars of the load distribution are sketched in Fig.4 for four values of $\beta \cot \Lambda$. In each case the pressure has been scaled to be unity along the centre-line of the fore part of the wing. These load distributions have been integrated graphically to obtain the total lift and moment in a number of cases. The results agree always to within 2% with the values found from Cohen's formulae. This is the greatest accuracy to be expected from the graphical integration, so the two methods can be taken to be in agreement.

The same case of non-interfering tips can also be dealt with by Evvard's method[8]; an extension[9] of this method seems to permit the calculation of the properties of some wings with interfering tips. This has not been attempted for the present paper as the complications introduced resemble those found if we proceed to higher order conical fields.

Since this work was carried out, the authors have learnt of detailed charts[10] for the lift coefficients of cropped delta and other planforms with supersonic trailing edges. At the points of exact comparison available, the results agree with the present values to within 1%. Ref.10 does not provide aerodynamic centre data.

3 DISCUSSION

Since these calculations were undertaken to throw light on the differences between experimental results and slender thin-wing theory for the gothic* planform, it is appropriate to discuss briefly how far they do so.

Inspection of Figs.2 and 3 shows that, in general terms, the lifting properties of cropped delta wings with subsonic leading edges do vary considerably with Mach number. Thus there is only a small range of Mach number within which slender thin-wing theory is reliable. This is obviously explicable in terms of the distributions of lifting pressure illustrated in Fig.4. These are calculated by supersonic thin-wing theory; according to slender theory the same type of distribution would be found ahead of the kink in the leading edge but with no lift behind it. Such a disparity between the results of the two theories might well be expected on a planform which is so obviously not "smooth" in the sense required by slender body theory. Again in general terms, it would be expected that the properties of a planform like the gothic with a continuously curved leading edge would change less rapidly with Mach number than those of a similar cropped delta. However, the experimental results of Ref.1 show that the lift slope and aerodynamic centre position at $\beta s/c = 0.252$ and above are very different from those calculated for a thin wing in attached flow at $\beta s/c = 0$. It is this discrepancy which we wish to explain.

Since at very small lift coefficients it is impossible to determine C_L/α and C_m/C_L with any accuracy from the experimental points and at large lift coefficients there is much uncertainty about the effects of leading edge separation, we shall compare theory and experiment for $C_L = 0.1$. As indicated above, it is sufficient to consider the results for $\beta s/c = 0.252$, corresponding to $M = 1.42$ and $s/c = 0.25$. We consider the "transition free" values, a quite arbitrary choice since the effects of fixing boundary layer transition on lift and moment were found to be small and not systematic with Mach number.

---

*The gothic planform has its leading edges formed by parabolic arcs. The vertices of the parabolas are at the wing tips, which are streamwise, and

The experimental value of C_L/α is 1.535 and the slender thin-wing theory calculation gives $\pi A/2$ or 1.178, so that the experimental value is 30% above the calculated one. The cropped delta wing with the same ratio of span to length and the same aspect ratio has a taper ratio of $1/3$. Its calculated lift is the same by slender thin-wing theory, but, by the present calculations, at M = 1.42 it has a C_L/α of 1.375, i.e. 17% above the slender thin-wing theory value. Since the effects of Mach number on the gothic wing are likely to be less than on the cropped delta, this leaves at least 13% to be accounted for. In Ref.6 it is suggested that the expression

$$C_L = \pi A\,\alpha/2 + 4\,\alpha^2$$

includes the non-linear lift due to leading edge separation, according to slender theory. The experimental evidence of Ref.2 fits this expression at M = 1. For the present planform it predicts an increment of 0.275 (i.e. 23% of the slender thin-wing attached-flow value) in C_L/α for $C_L = 0.1$. Increase of Mach number above M = 1 reduces non-linear lift rapidly (see Refs.2), so that a contribution to C_L/α from leading edge separation of 15-20% of the slender thin-wing theory value may be expected. It is thus possible to see how the value of C_L/α calculated for a limitingly slender wing in attached flow can be augmented to the experimental value; or, indeed, sufficiently above it to allow for the loss of lift due to boundary layer thickness.

The experimental centre of pressure at M = 1.42, $C_L = 0.1$, transition free, is at 56.7% of the centre-line chord from the apex. According to slender thin-wing theory it should be 10%c further forward at 46.7%. It is less easy to choose an "equivalent" cropped delta wing than it was for the lift alone. This is because the A = 0.75, s/c = 0.25 cropped delta considered above has its centre of pressure at M = 1.0 at 44.4%, while it is a wing with $\lambda = 0.3$ which has the same centre of pressure at M = 1.0 as the gothic. However, all these possible "equivalent" cropped delta wings have centres of pressure about 8%c further forward at M = 1.0 than at M = 1.42, so we can expect a difference of not more than 8%c on the gothic. Calculations supplementing those of Ref.6 for the effect of leading edge separation on centre of pressure position show a rearward shift of about 4%c between $C_L = 0$ and $C_L = 0.1$, according to slender theory. The effect of Mach number must again be to reduce this. Since the wing tested has a sharp trailing edge, its finite thickness has no effect on the lift, according to slender body theory. However, it does affect the centre of pressure by an amount calculable by the method of Maskell mentioned in the Introduction. For the wing tested the centre of pressure is calculated to be 2.5% further back than for the flat wing of the same planform. This figure from slender body theory is also likely to be reduced by the effects of Mach number since for large enough Mach numbers strip-theory becomes applicable and this predicts a forward shift of centre of pressure due to thickness. Thus we have once again produced corrections to the slender thin-wing calculation, which are known to be individually over-estimated and which together more than account for the discrepancy between the calculated and experimental values.

The lifting properties of the gothic planform have been calculated by Squire[11] using the not-so-slender theory of Adams and Sears[12]. It is of interest that his calculated results for the lift-slope and aerodynamic

The above discussion makes it clear that the problem of predicting the lift and moment of slender wings at supersonic speeds is not likely to be solved easily: attached flow theory must be modified to account for leading edge separation, thin wing theory must be modified to account for finite wing thickness and slender theories must all be modified to account for Mach number variations. The present paper shows the importance of the last of these for the attached flow past a particular family of planforms without thickness.

LIST OF SYMBOLS

A	aspect ratio
c	length of wing
C_L	lift coefficient
C_p	pressure coefficient
E	complete elliptic integral of second kind (Legendre)
h	distance of aerodynamic centre from apex
L_o	basic lift
ΔL	additional lift
M	Mach number
M_o	basic moment about apex
ΔM	additional moment about apex
m	$\beta \cot \Lambda$
q	dynamic pressure $(\frac{1}{2}\rho V^2)$
s	semi-span
α	incidence
β	$\sqrt{M^2 - 1}$
λ	taper ratio
Λ	leading edge sweep angle

LIST OF REFERENCES

No.	Author(s)	Title, etc.
1	Squire, L.C.	An experimental investigation at supersonic speeds of the characteristics of two gothic wings; one plane and one cambered. A.R.C. R. & M. 3211. May 1959.
2	Squire, L.C. Capps, D.S.	An experimental investigation of the characteristics of an ogee wing from M = 0.4 to M = 1.8. A.R.C. 21,798. August 1959.
3	Cohen, D.	The theoretical lift of flat swept-back wings at supersonic speeds. N.A.C.A. Tech. Note No. 1555. March 1948.
4	Gilles, A.	Calcul des répartitions de pressions sur certaines plaques minces portantes. La Recherche Aéronautique No. 19. Jan-Feb. 1951.
5	Stanbrook, A.	The lift-curve slope and aerodynamic centre position of wings at subsonic and supersonic speeds. A.R.C. 17615. November, 1954.
6	Smith, J.H.B.	A theory of the separated flow from the curved leading edge of a slender wing. A.R.C. R. & M. No. 3116. November 1957.
7	Peckham, D.H.	Low-speed wind-tunnel tests on a series of uncambered slender pointed wings with sharp edges. A.R.C. R. & M. 3186.
8	Evvard, J.C.	Distribution of wave drag and lift in the vicinity of wing tips at supersonic speeds. N.A.C.A. Tech. Note No. 1382. July 1947.
9	Hancock, G.J.	Note on the extension of Evvard's method to wings with subsonic leading edges at supersonic speeds. Aeronautical Quarterly, Vol. VIII, Part 1. Feb. 1957.
10	Lupin, E.	Charts for the computation of lift and drag of finite wings at supersonic speeds. Douglas Aircraft Co. Report No. SM-13480. Oct. 1949.

LIST OF REFERENCES (Contd.)

No.	Author(s)	Title, etc.
11	Squire, L.C.	Some applications of not-so-slender wing theory to wings with curved leading edges. A.R.C. 22,437. July, 1960.
12	Adams, MacC. Sears, W.R.	Slender-body theory - Review and extension. Journal of the Aeronautical Sciences, Vol. 18, p.75. February, 1951.

APPENDIX

DETAILS OF LIFT AND MOMENT CALCULATION

Cohen in Ref. 3 gives expressions for the lift and moment about the apex of a wing with straight subsonic or supersonic leading edge, streamwise tip and straight supersonic trailing edge. The expressions are obtained as the sum of two terms, one due to the basic delta solution and the other due to the cancelling solution as described in section 2. In all these, we take the limiting case appropriate to zero trailing edge sweep and obtain:

$$\frac{L_o}{q\alpha} = \frac{4s^2}{(1-\lambda)^2 E(\sqrt{1-m^2})}\left(\sin^{-1}(1-\lambda) + (1-\lambda)\sqrt{\lambda(2-\lambda)}\right)$$

$$\frac{M_o}{q\alpha} = -\frac{4s^2 c}{3(1-\lambda)^2 E(\sqrt{1-m^2})}\left(2\sin^{-1}(1-\lambda) + (1-\lambda)\sqrt{\lambda(2-\lambda)} + (1-\lambda)^3 \cosh^{-1}\frac{1}{1-\lambda}\right)$$

for the lift and moment arising from the basic delta solution; and

$$\frac{\Delta L}{q\alpha} = -\frac{4s^2}{(1-\lambda)^2 E(\sqrt{1-m^2})}\int_{1-\lambda}^{1}\frac{z-1+\lambda}{z^2}\left\{\left(1+\frac{1-\lambda}{z}\right)\left(\sqrt{z^2+\frac{z}{m}}-z\right) - \frac{z-1+\lambda}{2m\sqrt{z^2+\frac{z}{m}}}\right\}\frac{dz}{\sqrt{1-z^2}}$$

$$\frac{\Delta M}{q\alpha} = \frac{4s^2 c}{3(1-\lambda)^2 E(\sqrt{1-m^2})}\int_{1-\lambda}^{1}\frac{z-1+\lambda}{z^3}\left\{\frac{2}{z}\left(z^2+(1-\lambda)z+(1-\lambda)^2\right)\left(\sqrt{z^2+\frac{z}{m}}-z\right) - \frac{(z-1+\lambda)(2z+1-\lambda)}{2m\sqrt{z^2+\frac{z}{m}}}\right\}\frac{dz}{\sqrt{1-z^2}}$$

for the additional lift and moment due to the cancelling solution. Here λ is the taper ratio, $m = \beta \cot \Lambda$, Λ is the leading edge sweep, s is the semi-span, c is the length, $E(\sqrt{1-m^2})$ is the complete elliptic integral of the second kind of modulus $\sqrt{1-m^2}$. By the substitution

$$z^2 = 1 - \lambda(2-\lambda)u^2$$

the limits of integration in the expressions for ΔL and ΔM are made 0 and 1 and the integrands are transformed into functions which behave like polynomials at the end points. Standard Gaussian integration formulae can then be used to evaluate ΔL and ΔM.

For the present calculations, the ten-point Gaussian formula was used and the work carried out on DEUCE using the T.I.P. (Bristol Tabular Interpretive Programme).

The Prandtl-Glauert rule for thin wings implies that it is sufficient to calculate the properties of a two-parameter system of cropped delta wings. It tells us that, if the spanwise dimensions of the wing and Mach cone are multiplied by a factor, then the lift coefficient is multiplied by the same factor and the aerodynamic centre is unaltered. Thus the calculated quantities $\frac{1}{A} \frac{\partial C_L}{\partial \alpha}$ and $\frac{h}{c}$ are functions of the parameters $\beta \cot \Lambda$ and $\lambda c/\beta s$ for each example, only.

TABLE 1

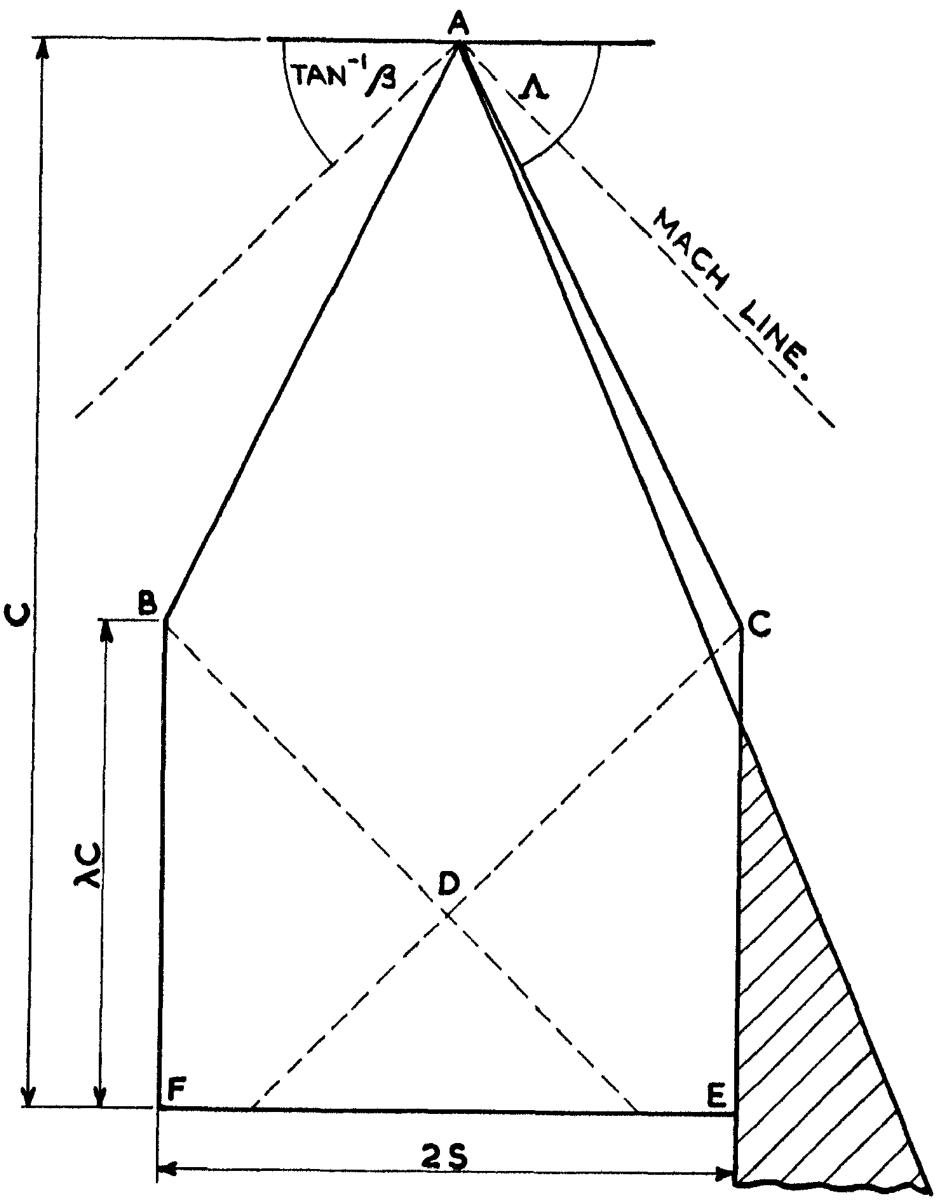
Values of $\frac{1}{A} \frac{\partial C_L}{\partial \alpha}$ for cropped delta wings

$\beta \cot \Lambda$ \ $\frac{\lambda c}{\beta s}$	0	0.4	0.8	1.2	1.6	2.0
0	1.571	1.571	1.571	1.571	1.571	1.571
0.05	1.563	1.602	1.621	1.628	1.625	1.613
0.10	1.546	1.622	1.661	1.676	1.670	1.648
0.15	1.523	1.635	1.693	1.716	1.710	1.679
0.20	1.495	1.642	1.719	1.750	1.745	1.706
0.30	1.433	1.643	1.755	1.804	1.801	1.752
0.40	1.365	1.632	1.776	1.843	1.845	1.789
0.50	1.297	1.613	1.787	1.871	1.880	1.820
0.60	1.231	1.590	1.790	1.891	1.906	1.845
0.70	1.167	1.564	1.788	1.904	1.927	1.865
0.80	1.108	1.537	1.782	1.913	1.944	1.882
0.90	1.052	1.509	1.774	1.918	1.957	1.897
1.00	1	1.482	1.764	1.921	1.967	1.909

TABLE 2

Distance of aerodynamic centre of cropped delta wing, measured from apex in terms of length h/c

$\beta \cot \Lambda$ \ $\frac{\lambda c}{\beta s}$	0	0.4	0.8	1.2	1.6	2.0
0	0.667	0.667	0.667	0.667	0.667	0.667
0.05	0.667	0.662	0.653	0.642	0.630	0.615
0.10	0.667	0.657	0.641	0.621	0.598	0.573
0.15	0.667	0.652	0.629	0.602	0.572	0.537
0.20	0.667	0.648	0.619	0.586	0.549	0.507
0.30	0.667	0.639	0.601	0.559	0.513	0.460
0.40	0.667	0.632	0.586	0.538	0.485	0.426
0.50	0.667	0.624	0.573	0.520	0.464	0.400
0.60	0.667	0.618	0.562	0.506	0.448	0.381
0.70	0.667	0.611	0.552	0.495	0.435	0.366
0.80	0.667	0.606	0.543	0.486	0.425	0.354
0.90	0.667	0.600	0.536	0.478	0.417	0.345
1.00	0.667	0.595	0.529	0.471	0.410	0.337



ELEMENTARY CANCELLATION SOLUTION PRODUCES CONSTANT LOAD OVER REGION ▨ AND ZERO DOWNWASH TO THE LEFT OF CE.

FIG. 1. NOTATION.

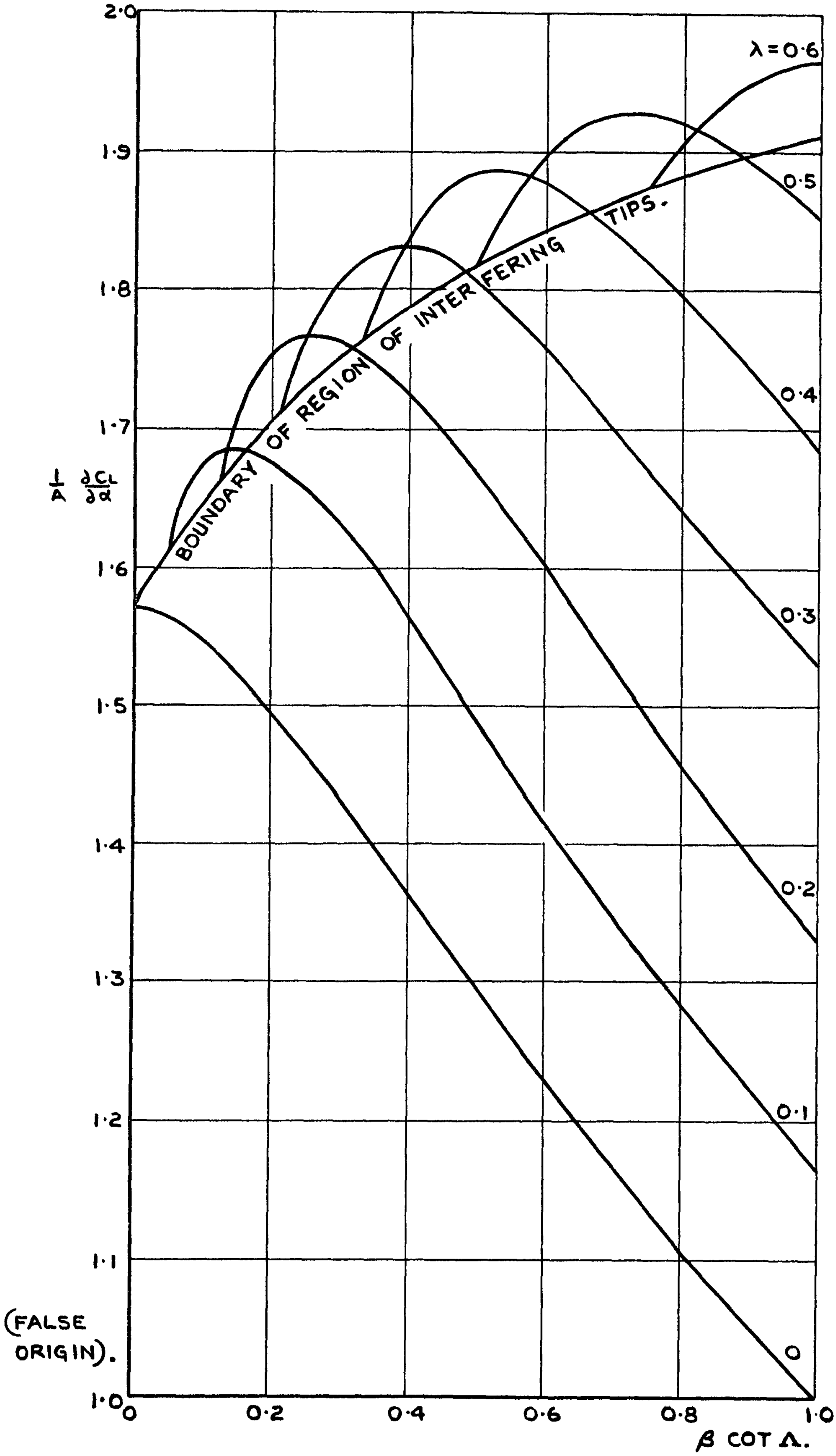
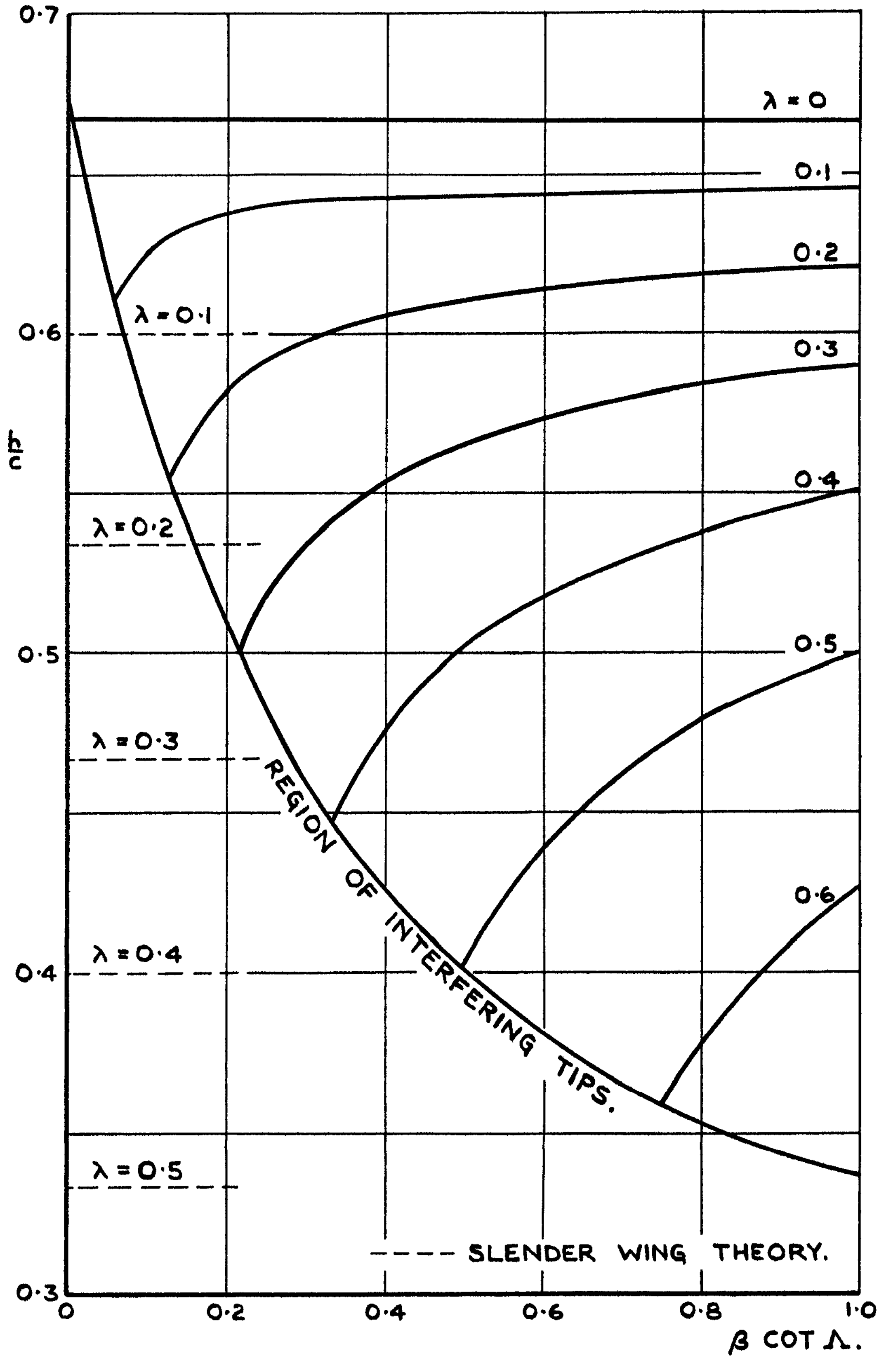


FIG. 2. LIFT CURVE SLOPE.



3. DISTANCE OF AERODYNAMIC CENTRE FROM APEX, IN TERMS OF LENGTH.

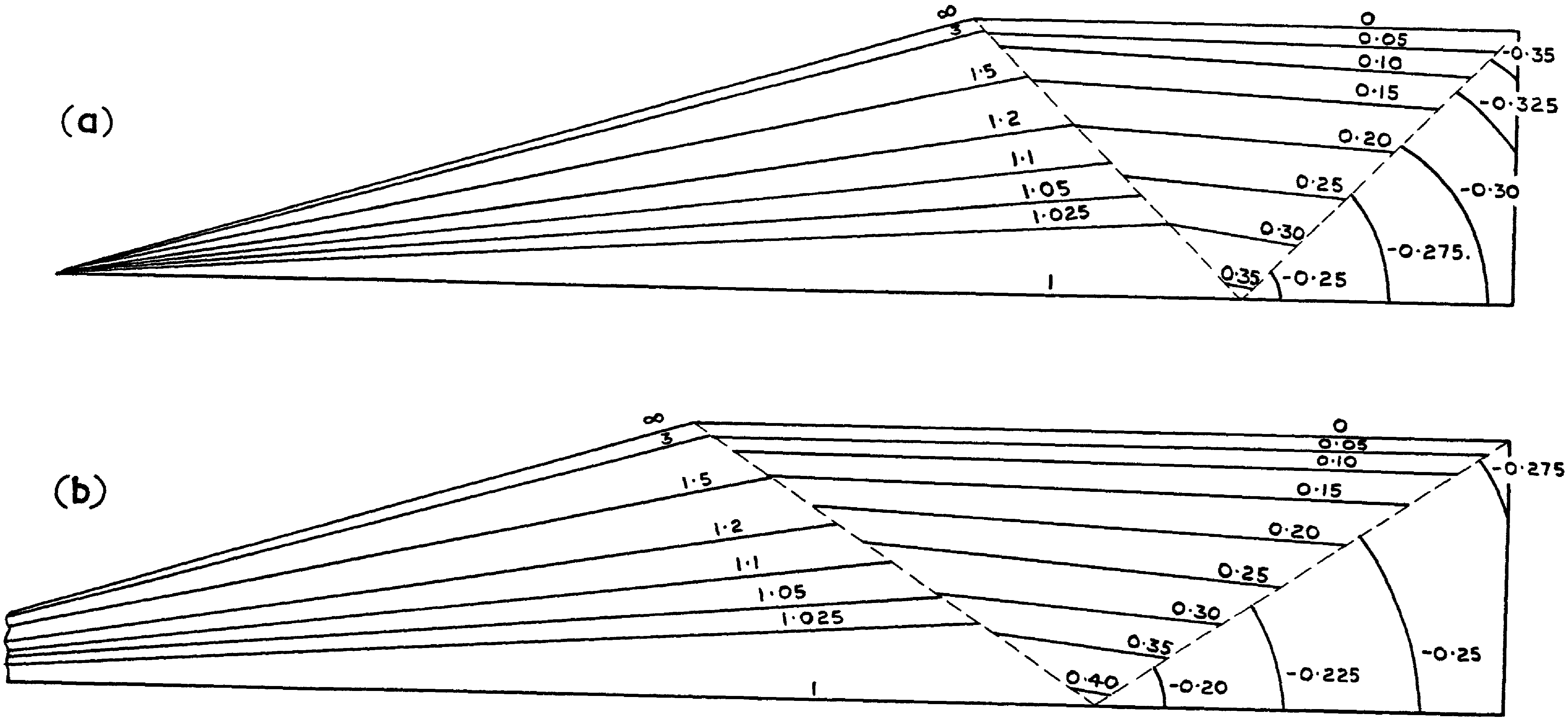


FIG. 4. SKETCHES OF ISOBAR PATTERNS:
(a) $\beta \cot \Lambda = 0{\cdot}3$ (b) $\beta \cot \Lambda = 0{\cdot}45$.
(THE NUMBERS ARE VALUES OF $C_p / C_p |_{\text{CENTRE LINE}}$)

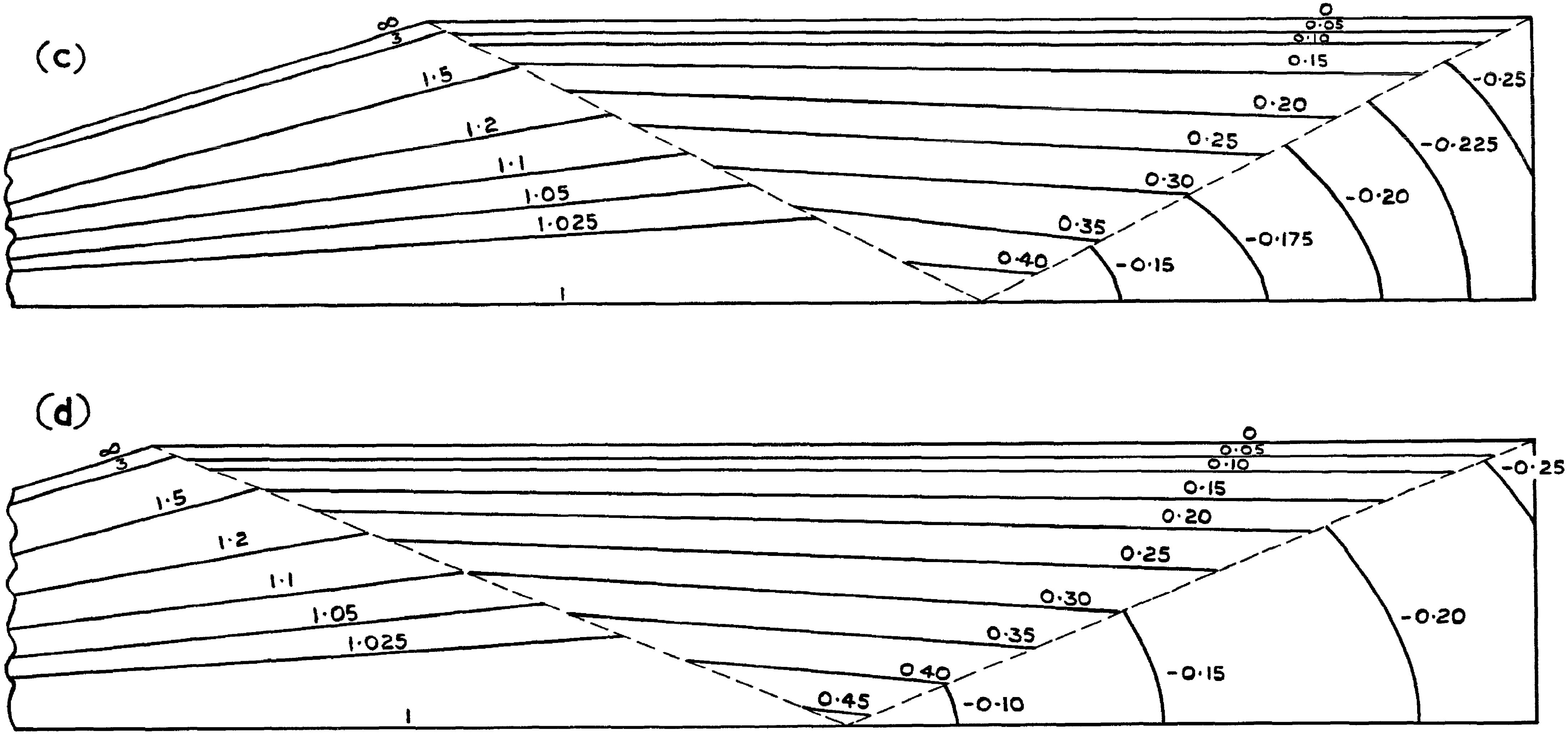


FIG. 4. (CONT.) SKETCHES OF ISOBAR PATTERNS:

(c) $\beta \cot \Lambda = 0{\cdot}6$ (d) $\beta \cot \Lambda = 0{\cdot}75$.

(THE NUMBERS ARE VALUES OF $C_p / C_p |_{\text{CENTRE LINE}}$)

A.R.C. C.P. No.562 533.693.3

CALCULATIONS OF THE LIFT SLOPE AND AERODYNAMIC CENTRE OF CROPPED DELTA WINGS AT SUPERSONIC SPEEDS. Smith, J.H.B., Beasley, J.A. and Stevens, A. July 1960.

The linearised theory of supersonic flow is used to calculate the lift slope and the aerodynamic centre of cropped delta wings which have subsonic leading edges. The cropped tip must be short enough for each tip to lie entirely upstream of the disturbances produced by the other. The results are presented graphically and in brief tables.

A.R.C. C.P. No.562 533.693.3

CALCULATIONS OF THE LIFT SLOPE AND AERODYNAMIC CENTRE OF CROPPED DELTA WINGS AT SUPERSONIC SPEEDS. Smith, J.H.B., Beasley, J.A. and Stevens, A. July 1960.

The linearised theory of supersonic flow is used to calculate the lift slope and the aerodynamic centre of cropped delta wings which have subsonic leading edges. The cropped tip must be short enough for each tip to lie entirely upstream of the disturbances produced by the other. The results are presented graphically and in brief tables.

A.R.C. C.P. No.562 533.693.3

CALCULATIONS OF THE LIFT SLOPE AND AERODYNAMIC CENTRE OF CROPPED DELTA WINGS AT SUPERSONIC SPEEDS. Smith, J.H.B., Beasley, J.A. and Stevens, A. July 1960.

The linearised theory of supersonic flow is used to calculate the lift slope and the aerodynamic centre of cropped delta wings which have subsonic leading edges. The cropped tip must be short enough for each tip to lie entirely upstream of the disturbances produced by the other. The results are presented graphically and in brief tables.

A.R.C. C.P. No.562 533.693.3

CALCULATIONS OF THE LIFT SLOPE AND AERODYNAMIC CENTRE OF CROPPED DELTA WINGS AT SUPERSONIC SPEEDS. Smith, J.H.B., Beasley, J.A. and Stevens, A. July 1960.

The linearised theory of supersonic flow is used to calculate the lift slope and the aerodynamic centre of cropped delta wings which have subsonic leading edges. The cropped tip must be short enough for each tip to lie entirely upstream of the disturbances produced by the other. The results are presented graphically and in brief tables.

Published by
HER MAJESTY'S STATIONERY OFFICE

To be purchased from
York House, Kingsway, London w.c.2
423 Oxford Street, London w.1
13A Castle Street, Edinburgh 2
109 St. Mary Street, Cardiff
39 King Street, Manchester 2
50 Fairfax Street, Bristol 1
2 Edmund Street, Birmingham 3
80 Chichester Street, Belfast 1
or through any bookseller

Printed in England

S.O. CODE No.23-9012-62
C.P. No. 562