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On Three-Dimensional Bodies of Delta Planform which can Support Plane Attached Shock Waves

by

D. H. Peckham

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ON THREE-DIMENSIONAL BODIES OF DELTA PLANFORM WHICH CAN SUPPORT PLANE ATTACHED SHOCK WAVES

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SUMMARY

This Note collects together in one report available theoretical work on bodies which can support attached plane shock waves, discusses some of the possible merits of such shapes, and includes some calculations illustrating their properties. Also, some preliminary results from wind tunnel tests are given, together with details of proposed future tests.

Replaces R.A.E. Tech. Note No. Aero 2812 - A.R.C.23,887.

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1 INTRODUCTION

For many years, the circular cone at zero incidence has been the only three-dimensional body shape for which a complete inviscid flow solution has been available at supersonic speeds[1]. However, for any combination of Mach number and shock wave angle, it is possible to construct three-dimensional bodies which support one or more attached plane shock waves; in this way, shapes are obtained which are amenable to exact oblique shock wave theory. This possibility appears to have remained unrecognised until recently, when Maikapar[2] investigated bodies of polygonal cross-section with re-entrant corners*, and Nonweiler[3,4] delta wings of "inverted-V" and "inverted-W" cross-section**. Examples of such body shapes are illustrated in Fig.1. The upper and lower surface flows on Nonweiler-wings are independent; there is, therefore, no need to define upper surface shapes for the time being. For the sake of simplicity, the upper surfaces of the wings in Fig.1 are shown as being generated by lines parallel to the free stream.

The principle of fitting a three-dimensional shape to a two-dimensional flow is not limited, in the case of a single body segment like the Nonweiler inverted-V wing, to the delta planform with straight leading edges. So long as the leading edges lie in one plane, any leading-edge planform shape is possible. For example, a parabolic planform shape would give an under-surface shape curved in cross-section, like an "inverted-U". Such shapes might offer certain advantages, e.g. structurally, but would probably give more complicated flows at off-design conditions than the delta planform. This Note is limited to discussion of the simplest case, the delta planform, but the possibilities of curved-planform shapes are also being investigated. Generally, any surface with straight generators formed by the parallel streamlines of the flow past a two-dimensional wedge falls in this category. In all cases, the shock is "contained" between the edges of a concave surface.

Other cases exist of flows where a geometrically simple shock shape is produced by a body of comparable simplicity. For example, Mangler[5] has investigated various shock shapes, which he showed to be produced by bodies generated from simple conic sections. Also, power-law bodies produce power-law shocks of similar shape[6]. But with such shapes, the shock is detached from the body and the flow is more complicated than in the case of bodies supporting plane attached shocks, since the curved shocks cause shear and entropy gradients and, with some bodies, regions of subsonic flow occur.

2 GENERAL THEORY

Let us consider a surface consisting of two flat triangular planes AOM and A'OM (Fig.2(a)), having an included angle, ε, such that $0^\circ < \varepsilon < 180^\circ$ (i.e. with no restriction on span, the value $\varepsilon = 180^\circ$ corresponding to an infinite-span wedge). Such a surface could be either one segment of a Maikapar-body, or a Nonweiler "inverted-V" wing. The shock is bounded by the two lines OA and OA', originating from the apex O. If OB is the bisector of the angle AOA' in the plane of the shock wave, and OF is the free stream direction, the angle BOF measures the inclination, ζ, of the shock wave to the free stream direction. The angle FOM measures the angle, δ, through which the flow is turned by passage through the shock.

For an oblique shock wave in continuum flow:-

$$\tan\delta = \frac{2\cot\zeta(\sin^2\zeta - 1/M_\infty^2)}{(\gamma+1) - 2(\sin^2\zeta - 1/M_\infty^2)} \qquad (1)$$

and

$$C_p = \frac{4}{\gamma+1}(\sin^2\zeta - 1/M_\infty^2), \qquad (2)$$

where M_∞ is the Mach number of the free stream, and $C_p = \frac{p - p_\infty}{q_\infty}$, is the pressure coefficient on the surface.

For $\gamma = 1.4$, the above equations become:-

$$\tan\delta = \frac{5\cot\zeta(\sin^2\zeta - 1/M_\infty^2)}{6 - 5(\sin^2\zeta - 1/M_\infty^2)} \qquad (3)$$

and

$$C_p = \frac{5}{3}(\sin^2\zeta - 1/M_\infty^2). \qquad (4)$$

The variation of pressure coefficient with shock wave angle, flow turning angle and Mach number is illustrated in Fig.3.

The case of an asymmetrical segment (Fig.2(b)) is also of interest. If the leading-edges of the segment are inclined at angles α and β to the free-stream direction, the shock wave angle is given by:-

$$\cot^2\zeta = \cot^2\alpha + \cot^2\beta \qquad (5)$$

or, in a form suitable for substitution in equations (1) to (4) inclusive:-

$$\sin\zeta = (1 + \cot^2\alpha + \cot^2\beta)^{-\frac{1}{2}}. \qquad (6)$$

For a body of unit length, the area of the base cross-section of such a segment is:-

$$S = \frac{1}{2}\cdot\frac{\tan\alpha\tan\beta\tan\delta}{\tan\zeta}. \qquad (7)$$

Let us now consider the Maikapar-body of n identical segments (Figs.1(a) and 2(c)), where n is an integer such that $n \geqslant 3$ ($n = 2$ being the infinite wedge).

For a body of unit length, the area of the base cross-section is:-

$$S = n \tan\delta \tan\zeta \tan\frac{\pi}{n} . \qquad (8)$$

Using equation (3) we get:-

$$S = n \tan\frac{\pi}{n} . \frac{5(\sin^2\zeta - 1/M_\infty^2)}{6 - 5(\sin^2\zeta - 1/M_\infty^2)} . \qquad (9)$$

An "equivalent" circular cone of the same length and base area and hence the same volume, has a semi-angle, θ, such that:-

$$\tan\theta = \left\{\frac{n}{\pi}\tan\frac{\pi}{n} . \frac{5(\sin^2\zeta - 1/M_\infty^2)}{6 - 5(\sin^2\zeta - 1/M_\infty^2)}\right\}^{\frac{1}{2}} . \qquad (10)$$

From equations (4) and (10) follows a simple relationship between the semi-angle of the equivalent cone and the pressure coefficient on the Maikapar-body, which is independent of Mach number:-

$$\tan\theta = \left(\frac{n}{\pi}\tan\frac{\pi}{n} . \frac{C_p}{2 - C_p}\right)^{\frac{1}{2}} \qquad (11)$$

or

$$C_p = \frac{2\tan^2\theta}{\tan^2\theta + \frac{n}{\pi}\tan\frac{\pi}{n}} . \qquad (12)$$

Equation (12) enables a direct comparison to be made between the drag of a Maikapar-body and an equivalent cone, but it must be remembered that for any Mach number, and a given number of identical body segments, there is only one cross-section shape which will support attached plane shock waves.

In equation (12), $\frac{n}{\pi}\tan\frac{\pi}{n} > 1$, but as n tends to infinity, $\frac{n}{\pi}\tan\frac{\pi}{n}$ tends to unity, and we get:-

$$(C_p)_{n\to\infty} = \frac{2\tan^2\theta}{\tan^2\theta + 1} = 2\sin^2\theta \qquad (13)$$

which is the Newtonian value for the pressure coefficient on a cone.

Thus for finite values of n, the drag of Maikapar-bodies is less than that of equivalent cones, as given by Newtonian theory; this is illustrated in Fig.4(a). However, Newtonian theory underestimates the pressure coefficient on a cone, and in Fig.4(b) the drag of Maikapar-bodies is compared with values calculated from the Taylor-Maccoll[1] theory for a Mach number of infinity; at finite Mach numbers, the drag of cones is greater, and the comparison would be even more favourable to the Maikapar-bodies. It should be remembered though, that for a given length/diameter ratio, the pointed cone is not a minimum-drag shape; blunted-cones[7] and some power-law bodies[8] can have less drag than pointed cones, but not to the extent that a three-segment Maikapar-body can.

The lift of Nonweiler-wings can be compared with that of other lifting shapes by a procedure similar to the one followed above. For example, dividing a four-segment Maikapar-body into two equal parts gives a Nonweiler-wing of "inverted-W" section, which can be compared with a semi-cone body of the same base area and volume. However, with this latter shape, a thin delta wing can be profitably mounted on top of the body to contain the semi-conical shock and support the same pressure coefficient as the body, to produce an increased lift without adding base area. A comparison of the lift of this shape (calculated from tables in Ref.9), with that of Nonweiler-wings consisting of two 90°-segments, is given in Fig.5. Since both shapes have constant pressure on their lower surfaces, the lift/drag ratio is simply the ratio of projected plan area to base area, and this ratio is, in general, somewhat higher for the Nonweiler shape. However, the pressure on a Nonweiler-wing is lower than that on an equivalent semi-cone/delta body. The net result is that the lift of this type of Nonweiler-wing is slightly less than that of the equivalent semi-cone/delta body of the same base area, the difference being greatest at the lower Mach numbers; for the same lift coefficient, the lift/drag ratio of the Nonweiler-wing is the higher, except at low values of lift coefficient at the lower Mach numbers. As a guide to the relative values of skin-friction drag on these shapes, the surface area of this type of Nonweiler-wing is some 20-25% greater than that of the semi-cone/delta body of the same base area.

It must be emphasised that the Nonweiler shape chosen for the above comparison is not an optimum shape. Higher values of lift/drag ratio would be obtained from a shape consisting of two asymmetrical segments of the type illustrated in Fig.2(b), which would have a higher ratio of plan area to base area; with this shape, as the span becomes large relative to its depth, two-dimensional wedge conditions are approached and the lift/drag ratio approaches a maximum of cot δ. It is noted that the values of L/D quoted are, of course, not the maximum values obtainable when the top surface is allowed to be inclined to the mainstream. The above values give an indication of the lift produced by a given volume and of the associated drag force.

3 COMPARISONS WITH LESS-EXACT THEORIES

With most lifting shapes, it is not possible to predict pressure distributions by exact shock wave theory, and one has to rely, for example, on linear theory at supersonic speeds and Newtonian theory at hypersonic speeds. It is interesting, therefore, to see whether these theories provide reasonable estimates for shapes which support attached plane shock waves.

In Fig.6, taken from Ref.10, the lift developed by the lower surface of Nonweiler-wings of inverted-V section is shown for various Mach numbers, and a design incidence of 10°; the assumption was made that at a given Mach number exact oblique shock wave theory applied provided that the value of the angle BOM at the design condition (Fig. 2(a)), remained within 0.2° of the theoretical value of the angle between the shock and the ridge-line OM. In linear theory, for a delta wing with supersonic leading edges, the total lift coefficient is $4\delta/(M_\infty^2 - 1)^{\frac{1}{2}}$, where δ is the incidence. This lift is divided equally between the upper and lower surfaces, i.e. the lower surface lift coefficient is $2\delta/(M_\infty^2 - 1)^{\frac{1}{2}}$. Fig. 6 shows that a body supporting a plane attached shock develops more lift than linear theory predicts, but this should not be taken as meaning that a Nonweiler configuration necessarily produces more lift than a plane delta wing. For example, tests at a Mach number of 4 on delta wings[10], have given lift coefficients in reasonable agreement with linear theory up to at least 15° incidence; at this incidence the upper surface pressure predicted by linear theory exceeds the limit of absolute vacuum. Thus a loss of suction on the upper surface, relative to linear theory, appears to have been compensated by a corresponding increase in pressure on the lower surface.

A comparison of Newtonian theory with oblique shock theory can most conveniently be made by correlating pressures on a wedge in the form of the Newtonian impact theory equation. If the pressure coefficient on a wedge is defined as $C_p = K \sin^2\delta$, where δ is the wedge semi-angle, a unique curve for K is obtained when correlated on the basis of $1/(M_\infty^2 - 1)^{\frac{1}{2}} \sin\delta \cos\delta$ (Fig. 7). This correlation applies for $M_\infty \delta >> 1$, and the minimum value for K is found to be 2.4; this value can also be derived by applying the strong shock approximation to the oblique shock wave equations, when it is found that $C_p = (\gamma + 1) \sin^2\delta$ as M_∞ tends to infinity[11]. Thus, the usual impact theory equation, $C_p = 2 \sin^2\delta$, underestimates pressures on wedges by at least one sixth (for $\gamma = 1.4$); physically, this is to be expected since the basic assumption of Newtonian theory, that the shock lies close to the body surface, is violated.

4 SOME PROPERTIES OF MAIKAPAR-BODIES AND NONWEILER-WINGS

4.1 Off-design performance

In section 2, the performance of these shapes at conditions for shock attachment only was considered, but their performance at off-design incidences and Mach numbers is also of interest. There seems to be no certain method for tackling this question directly, but from examination of the requirements for shock attachment, one can infer the degree to which off-design conditions would affect the two-dimensional flow of the attached-shock state. In this context, a distinction must be drawn between the Maikapar-body and the Nonweiler-wing; with the former, the flow-turning angle δ, and the shock-angle ζ, are fixed by the geometry of the body, so for each body there is only one Mach number for shock attachment; with the latter there is the freedom to change incidence, so that δ and ζ can be varied, and the attached-shock condition obtained over a range of Mach numbers.

For a single body segment of the type illustrated in Fig. 2(a), the angle between the plane of the leading-edges and the ridge-line OM $(\zeta - \delta)$ is these bodies could operate only over a limited range of Mach number while still maintaining approximately two-dimensional flow conditions.

(iii) At a constant Mach number, the required value of $\zeta - \delta$ for shock attachment varies little with shock wave angle. This has been pointed out by Squire[10], and Fig.2 of his report is reproduced in this Note (Fig.9). In this case, therefore, a body should be able to cover a wide range of attitudes without departing significantly from the attached-shock condition. It should be noted also in Fig.8, that at a given value of $\zeta - \delta$, and a given Mach number close to the minimum Mach number for shock attachment, two shock wave angles exist for the condition of shock attachment. Thus, at a constant Mach number, a body can be at the attached-shock condition at a low incidence, depart from this condition as the incidence is increased, but return to a second attached-shock condition at a still higher incidence.

4.2 Possible flight trajectories

A Nonweiler-wing of given geometry is not necessarily limited to one particular design condition, and Fig.8 illustrates that there is a range of Mach numbers and shock wave angles over which a wing of fixed $\zeta - \delta$ can support an attached plane shock by allowing its attitude to vary; it is interesting to investigate whether this range can be linked with useful flight trajectories.

For example, referring again to Fig.8, a wing could follow a gliding re-entry trajectory starting at a high altitude and Mach number, with a high incidence and lift coefficient; as it descended and decelerated the incidence could be appropriately decreased so as to maintain the shock attached to its leading edges and give a decreasing lift coefficient. This process could continue until the minimum Mach number for shock attachment is reached. It is not suggested that such a trajectory would be an optimum. However, the variation of lift/drag ratio with Mach number is similar to that derived by Plascott[12] for a constant dynamic pressure, and a constant path angle, lifting re-entry.

4.3 Propulsion

The region of uniform two-dimensional flow between the shock and wing surface should be particularly suitable for an engine intake, with the added advantage that the flow direction in this region is apparently not altered greatly by changes of incidence. Furthermore, a plane shock would cause pre-compression of the intake air, giving the advantage of greater mass-flow for a given intake area as compared with an intake in the free stream, and an intake Mach number less than the Mach number of the vehicle. At high Mach numbers, say 5 and above, the dimensions of the propulsion and lifting systems of a vehicle would be comparable, and integration of the two would be of great advantage[13].

Another possibility is external combustion; fuel could be injected through the surfaces of the wing, or out of its leading edges, and ignited at the rear of the vehicle which would be shaped to have surfaces on which lift and thrust forces could be sustained.

4.4 Stability and control

At the present stage, little can be said on this subject, but since the segments of Maikapar-bodies and Nonweiler-wings naturally form "wings" and "fins", simple trailing-edge controls might prove adequate. However, problems can arise with two-dimensional compression corners[14]. With Nonweiler-wings, there is no restriction on aspect ratio in achieving the attached-shock condition at supersonic speeds, so an aspect ratio could be chosen which gave adequate lift, and longitudinal as well as lateral stability at low speeds.

The effect of yaw on the flow over such bodies at supersonic speeds is not yet clear, and this question will be investigated experimentally, but some preliminary theoretical estimates have been made by Bagley[15].

4.5 Boundary layers

The isobaric surfaces of these shapes when supporting attached plane shocks should favour the maintenance of a laminar boundary layer, since the presence of spanwise pressure gradients which produce secondary flows and early transition to a turbulent boundary layer, is avoided. It can be expected that the displacement effect of the boundary layer will tend to oppose shock attachment, but allowance for this could probably be made, except in the case of very thick boundary layers. Results of some calculations on boundary layer properties are reported in Ref.16.

5 PRELIMINARY EXPERIMENTAL RESULTS

Exploratory tests on two "inverted-V" models have been made by Treadgold[17] at a Mach number of 4.3 in the R.A.E. No.8 (9 in. × 9 in.) wind tunnel to check whether the predicted flow was obtained, and as a guide to planning future experiments; one model was designed to have a subsonic component of the flow behind the shock normal to its leading-edges, the other model a supersonic component normal to its leading-edges. Pressures were measured at one point on the undersurface of each model, and shadowgraph pictures taken, over a 12-degree range of incidence on either side of the design incidence. The experimental results are plotted in Fig.10, and it can be seen that theory and experiment are in close agreement for the design incidence; away from the design attitude, pressure coefficient and shock wave angle do not depart far from that predicted by two-dimensional theory, nor is there any significant difference in the results for the two models. However, firm conclusions cannot be drawn from pressures measured at one station only. The shock wave angle is on average about 2° greater than the theoretical value, this effect probably being due to the displacement effect of the boundary layer, and some lack of sharpness of the leading-edges of the wings.

6 CONCLUSIONS

This Note has discussed the theory of a family of bodies of delta planform, which have the particular attraction to the aerodynamicist of producing a simple flow amenable to exact theory. Results of preliminary experiments show sufficient confirmation of the theory to justify more detailed investigations.

LIST OF SYMBOLS

α, β	inclinations of leading-edges of segment of a body to the free-stream direction
γ	ratio of specific heats for air (= 7/5)
δ	flow-turning angle through plane shock (or incidence of flat delta wing)
ζ	angle of plane shock to free stream direction
ε	included angle between the two triangular planes which make up one segment of a body
θ	semi-angle of "equivalent" cone
n	number of body segments
p	pressure on surface of a body
p_∞	free-stream static pressure
q_∞	free-stream dynamic pressure $\left(= \frac{\gamma}{2} p_\infty M_\infty^2\right)$
x, y, z	rectangular co-ordinates, origin at body apex, Ox streamwise
M_∞	Mach number
S	body base area
C_p	pressure coefficient $\left(= \frac{p - p_\infty}{q_\infty}\right)$
$(C_L)_B$	lift coefficient $\left(= \frac{L}{q_\infty S}\right)$
$(C_D)_B$	drag coefficient $\left(= \frac{D}{q_\infty S}\right)$

Suffix

B	relative to base area

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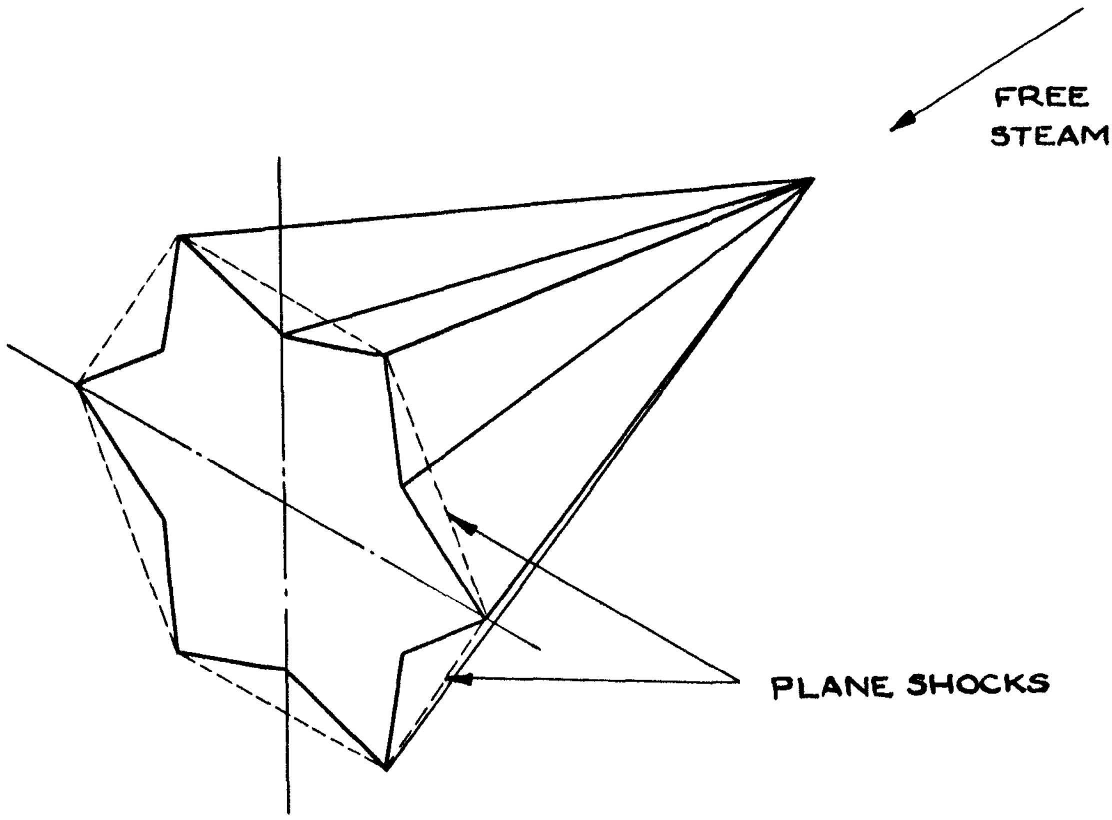
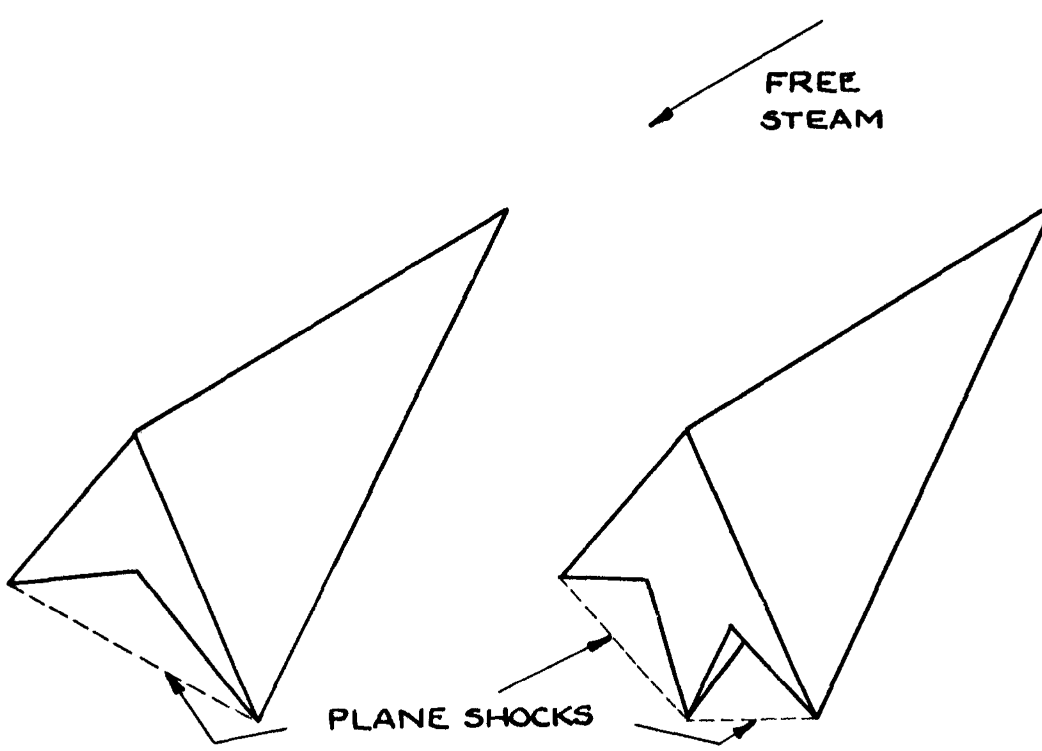
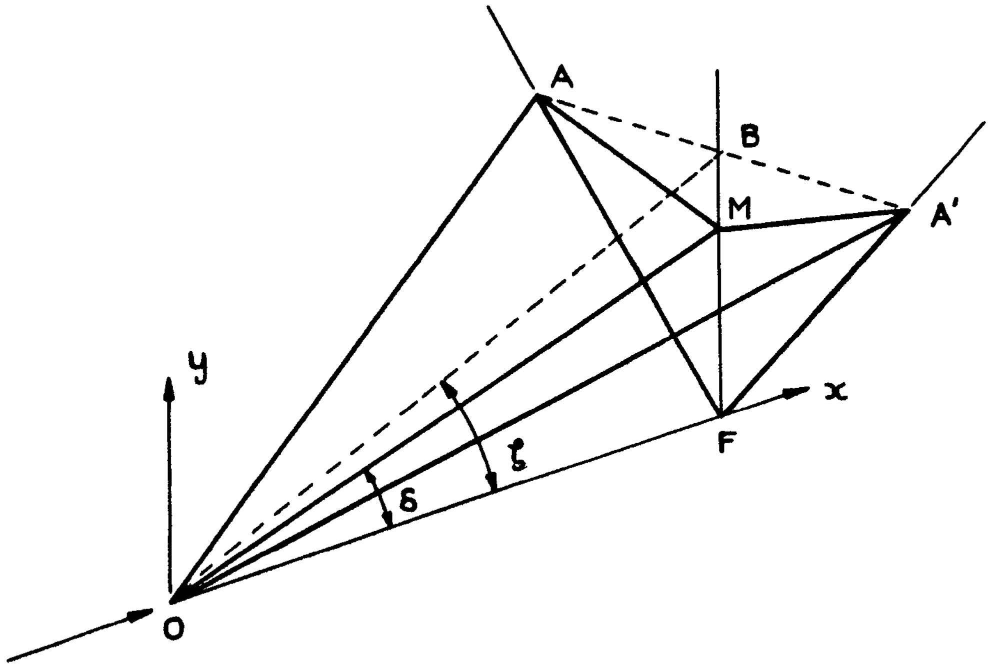


(a) MAIKAPAR BODY.

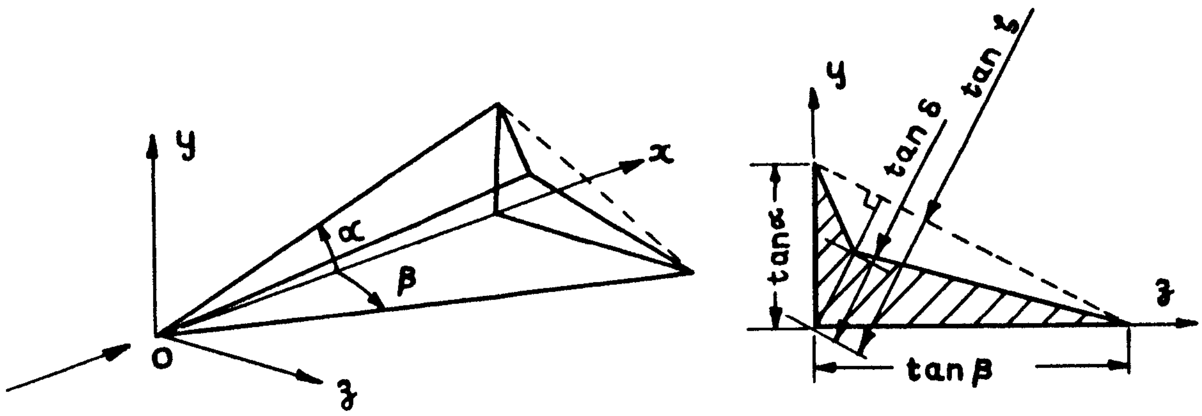


(b) NONWEILER WINGS.

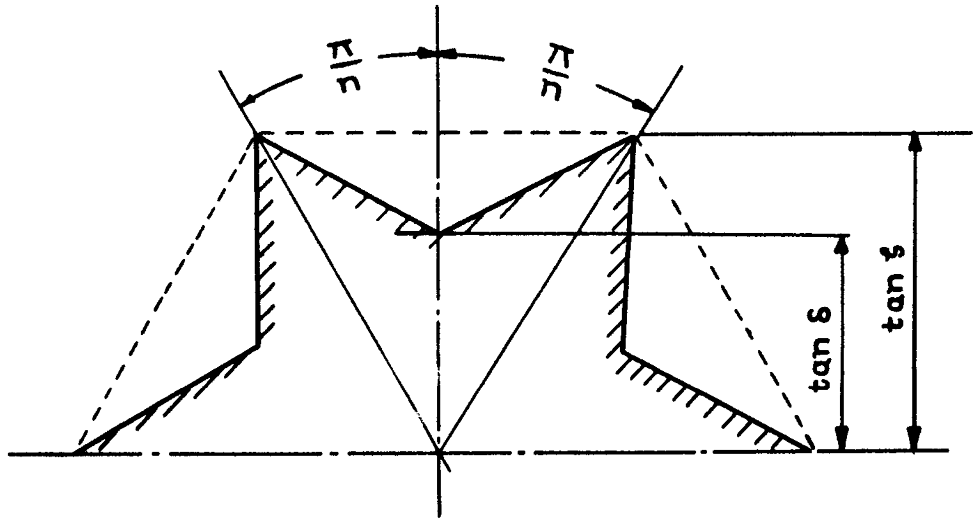
FIG.1. BODY SHAPES WHICH CAN SUPPORT PLANE SHOCK WAVES.



(a) SYMMETRICAL SEGMENT.

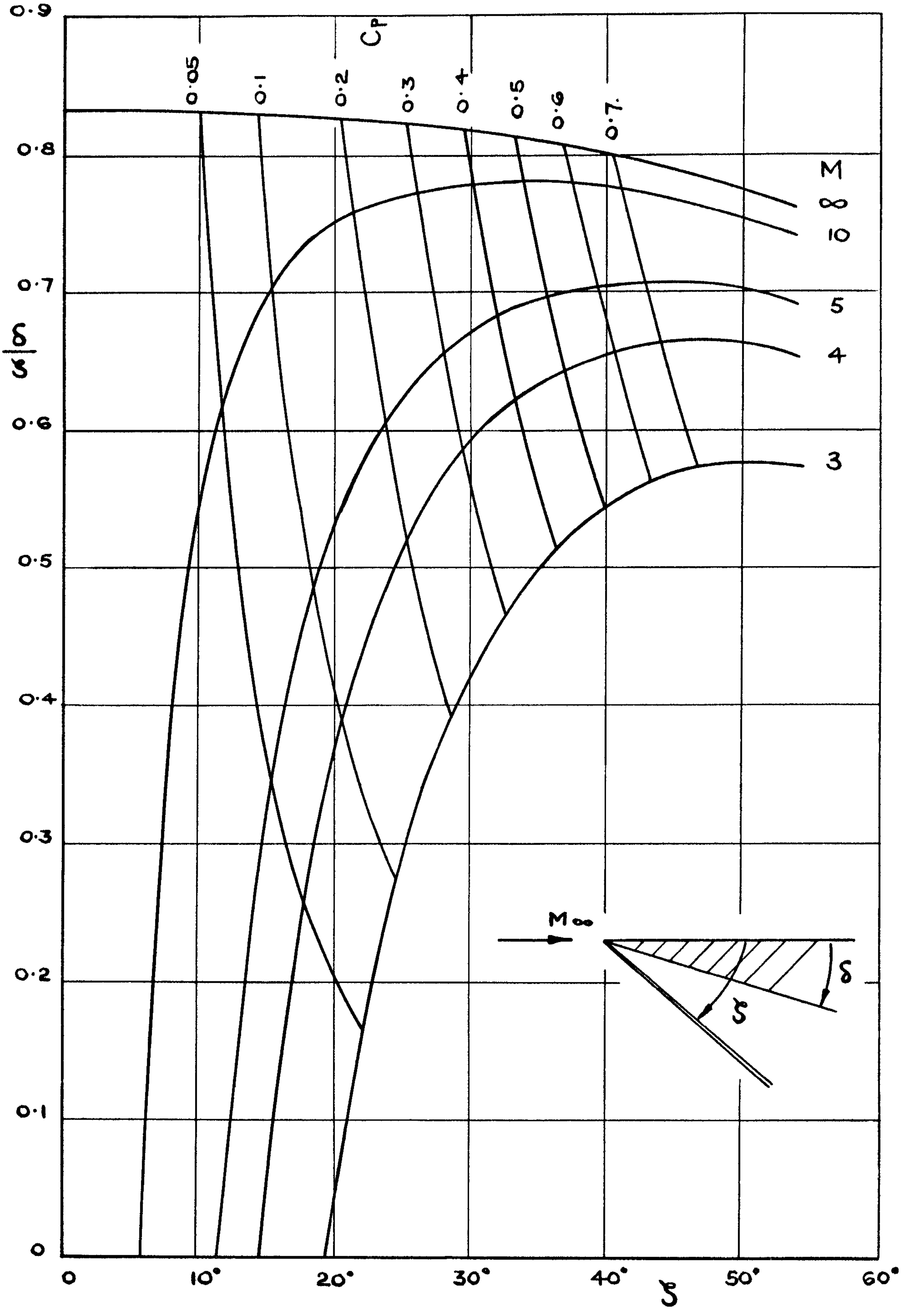


(b) ASYMMETRICAL SEGMENT.

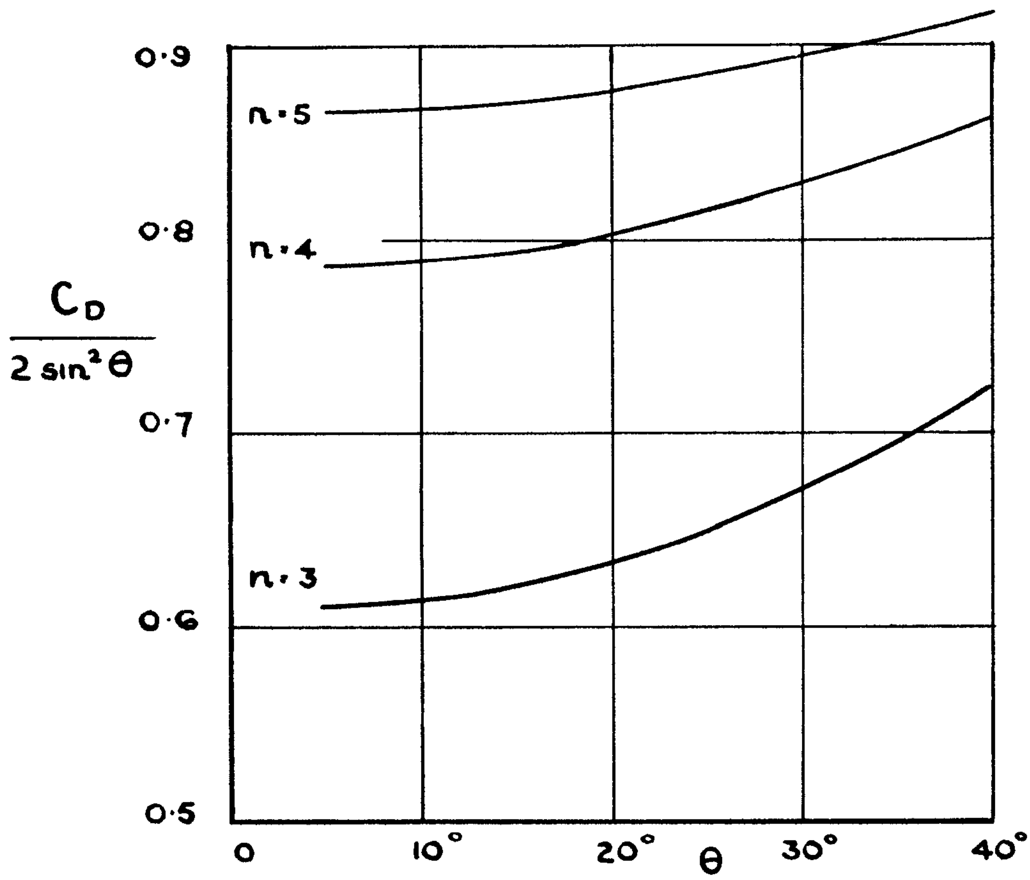


(c) MAIKAPAR-BODY OF n SEGMENTS.

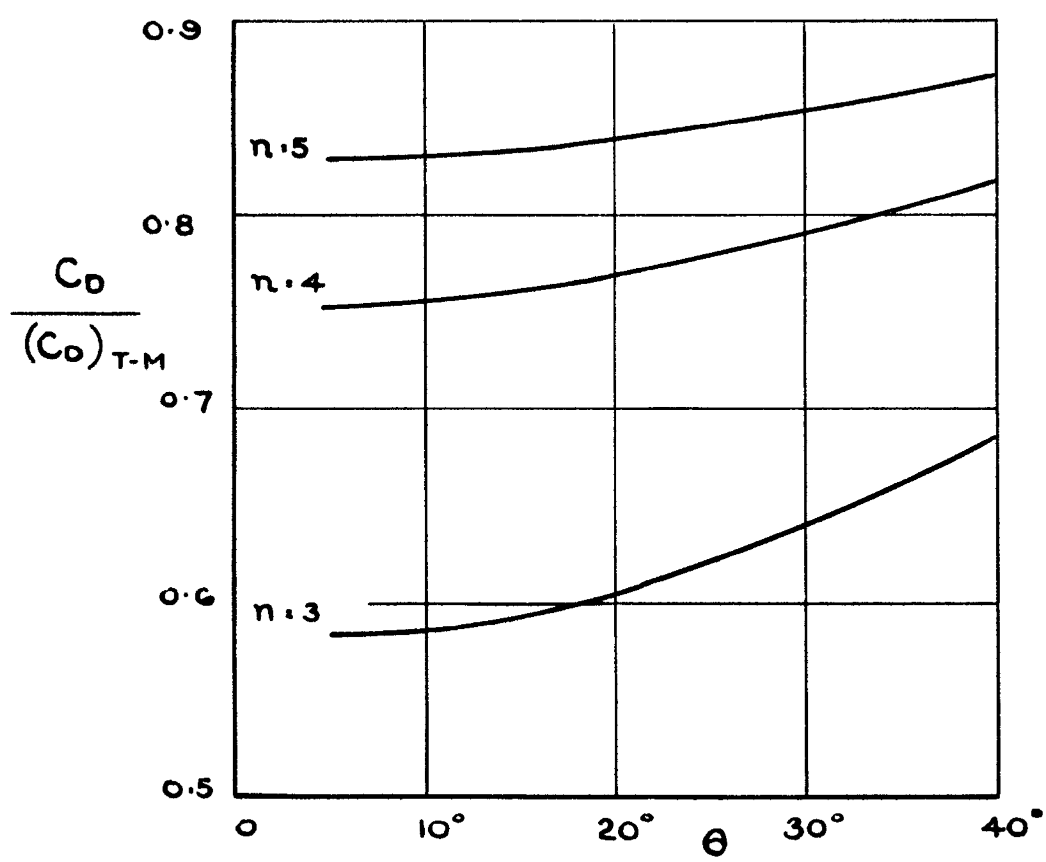
FIG. 2.(a,b&c) GEOMETRIC FEATURES AND NOTATION.



0·9
0·8
0·7
0·6
0·5
0·4
0·3
0·2
0·1
0
δ/ζ
Cp
0·05
0·1
0·2
0·3
0·4
0·5
0·6
0·7
M
∞
10
5
4
3
M∞
δ
ζ
0
10°
20°
30°
40°
50°
60°
ζ



(a) COMPARISON WITH NEWTONIAN THEORY



(b) COMPARISON WITH TAYLOR-MACCOLL THEORY M = ∞

FIG 4(a & b) DRAG OF MAIKAPAR-BODY COMPARED WITH THAT OF EQUIVALENT CONE

$$C_D = \frac{2\tan^2\theta}{\tan^2\theta + \frac{n}{\pi}\tan\frac{\pi}{n}}$$

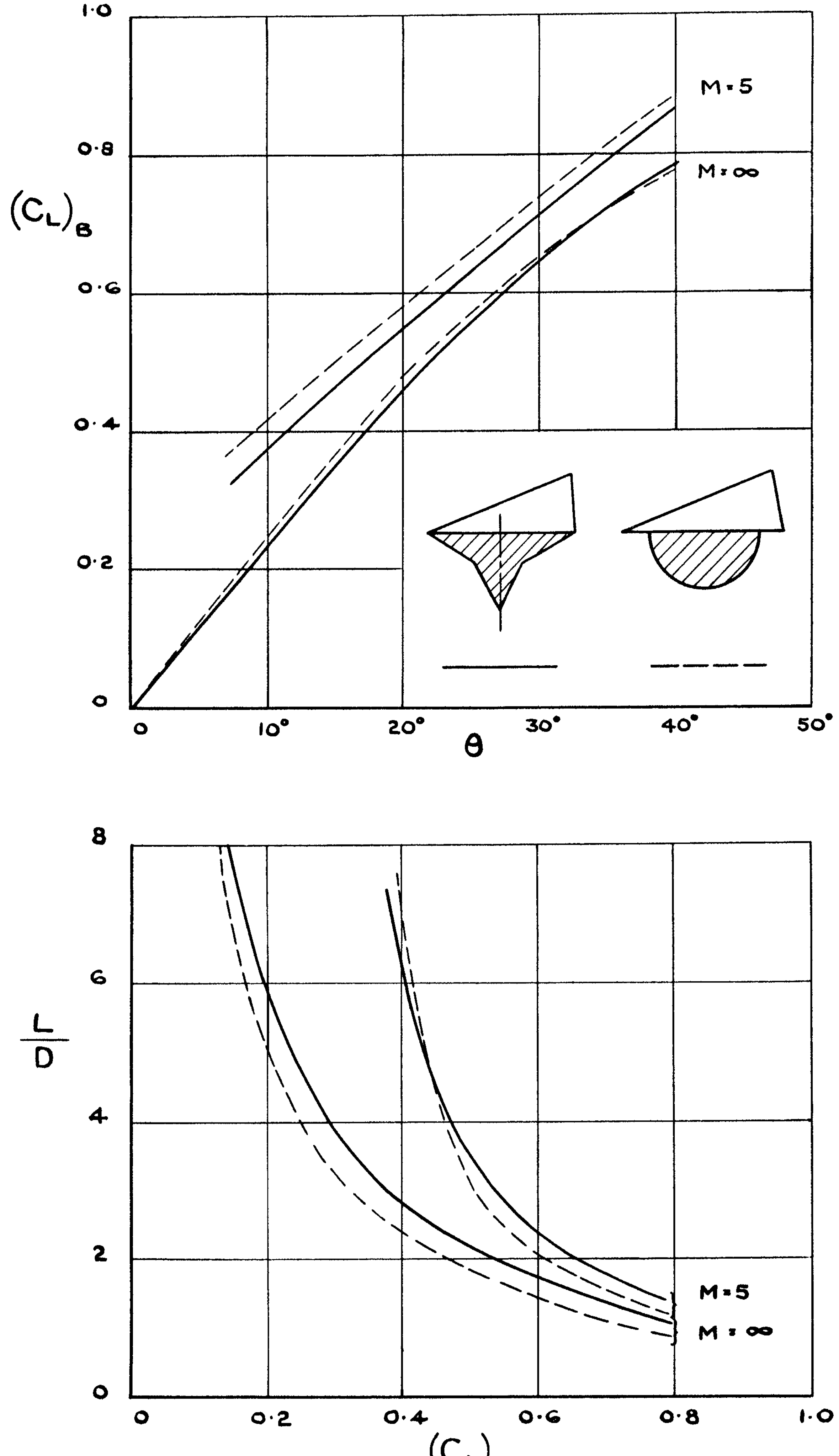


FIG.5. COMPARISON BETWEEN LIFT PROPERTIES OF NONWEILER WING AND SEMI-CONE/DELTA WING OF THE SAME BASE AREA.
(NO ALLOWANCE FOR SKIN FRICTION OR BASE DRAG.)

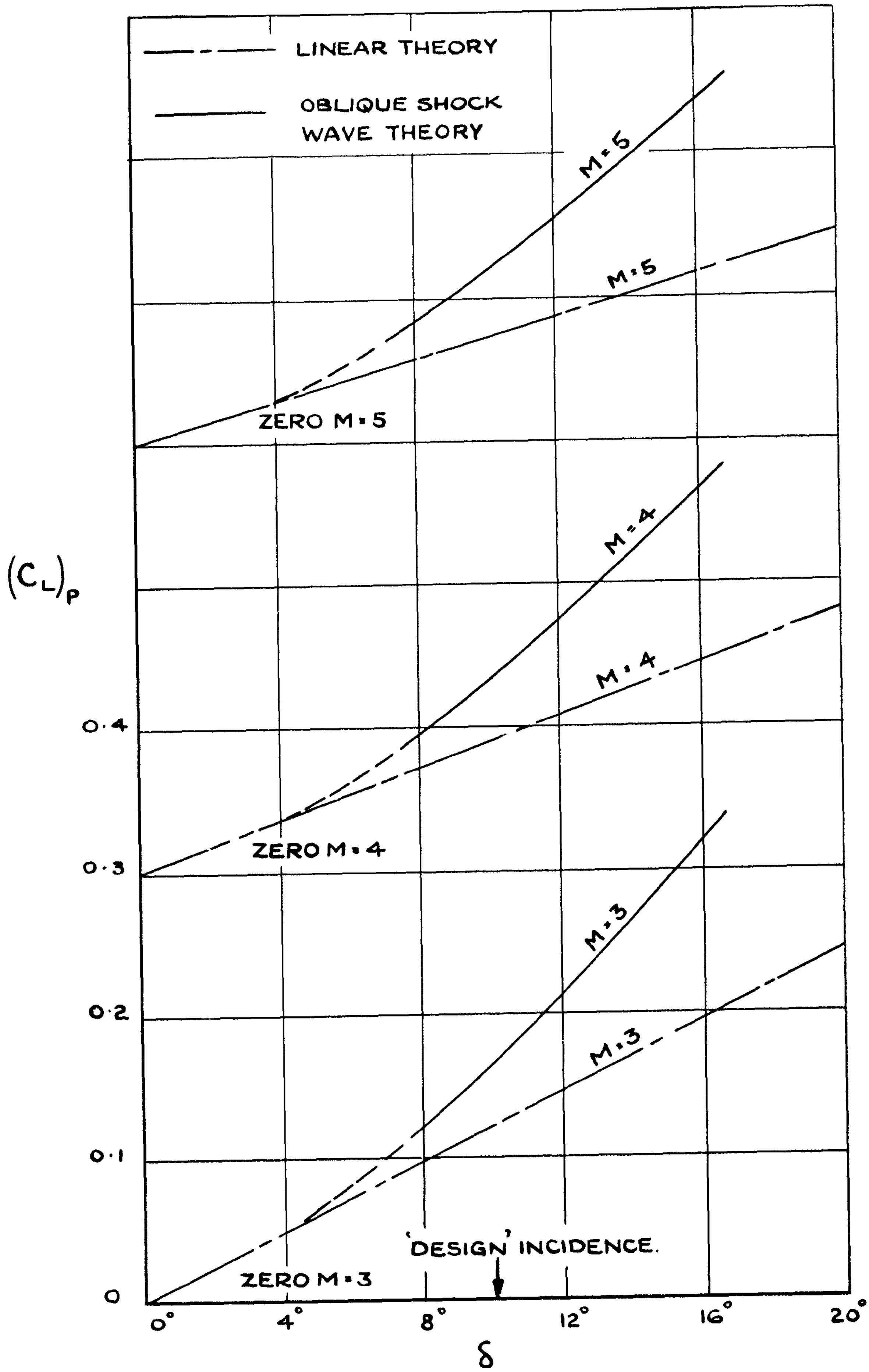
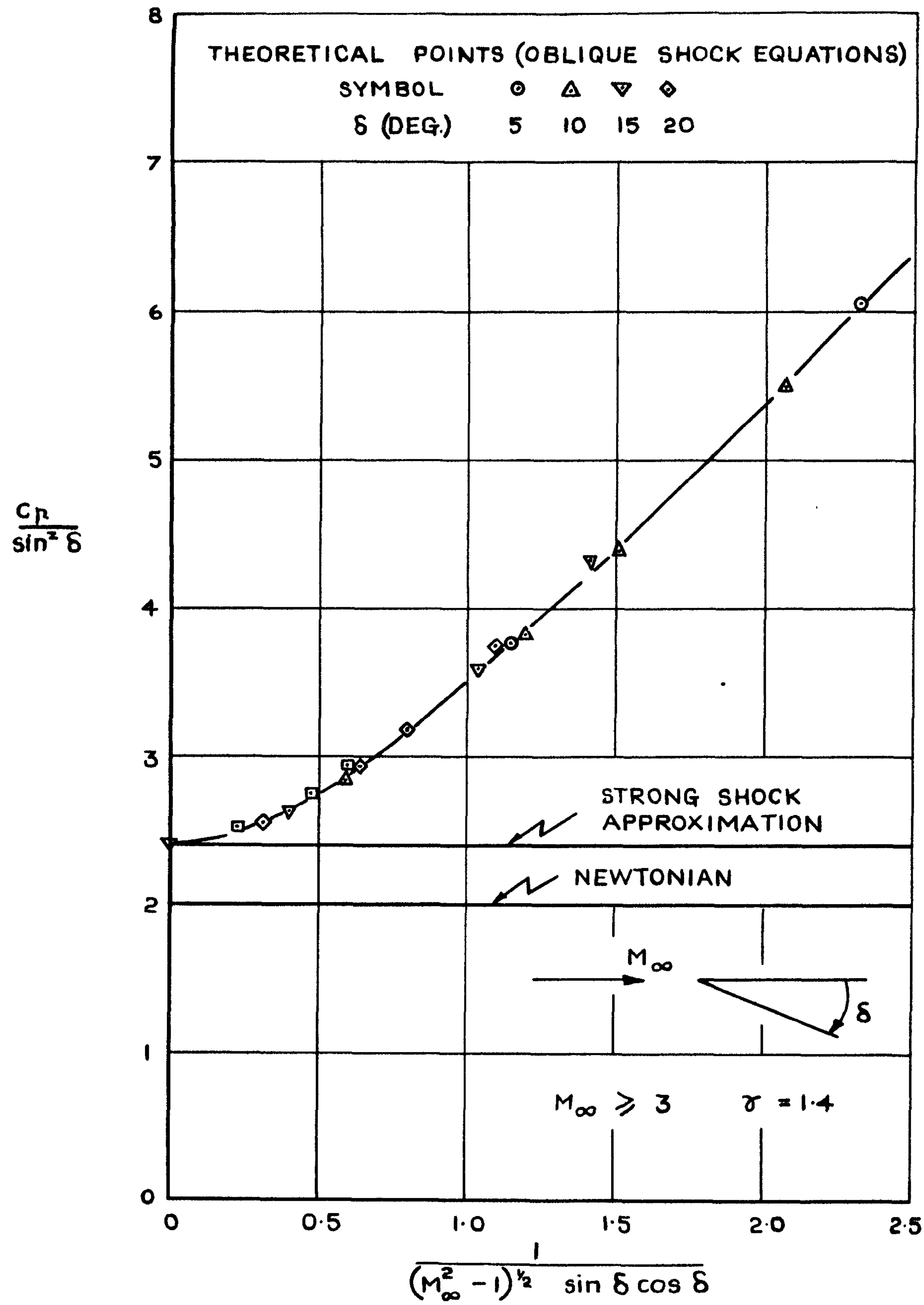


FIG.6. COMPARISON OF LINEAR THEORY WITH OBLIQUE SHOCK-WAVE THEORY (FROM REF. 10.)

$(C_L)_P$ = LIFT COEFF. REFERRED TO PLAN AREA.


THEORETICAL POINTS (OBLIQUE SHOCK EQUATIONS)
SYMBOL ⊙ △ ▽ ◇
δ (DEG.) 5 10 15 20
STRONG SHOCK APPROXIMATION
NEWTONIAN
M_∞
δ
$M_\infty \geqslant 3$ $\gamma = 1{\cdot}4$
$\frac{C_p}{\sin^2\delta}$
$\frac{1}{(M_\infty^2-1)^{1/2}\sin\delta\cos\delta}$
0 1 2 3 4 5 6 7 8
0 0·5 1·0 1·5 2·0 2·5

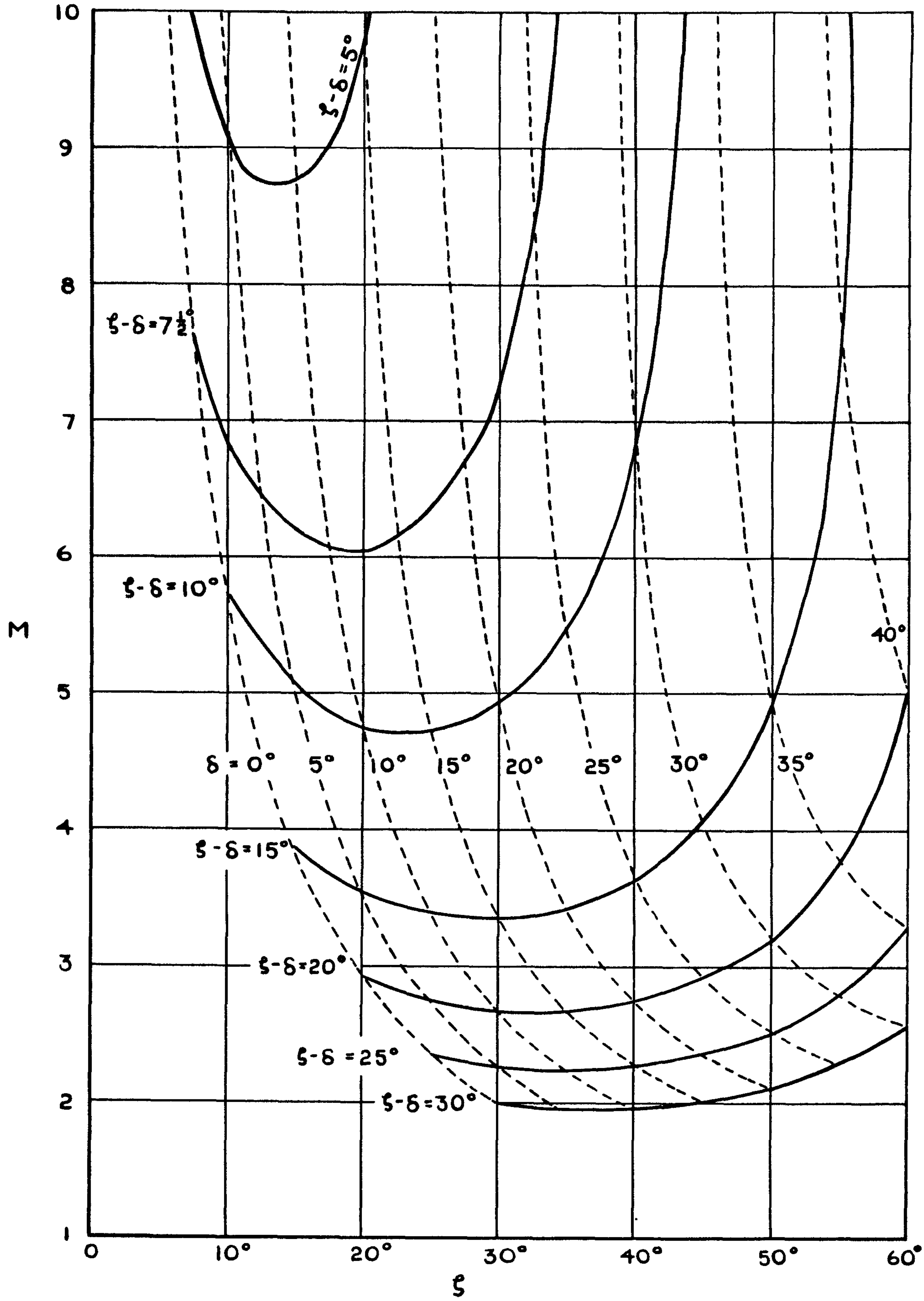


FIG. 8. VARIATION OF SHOCK-WAVE ANGLE WITH MACH NUMBER FOR SHOCK ATTACHMENT.

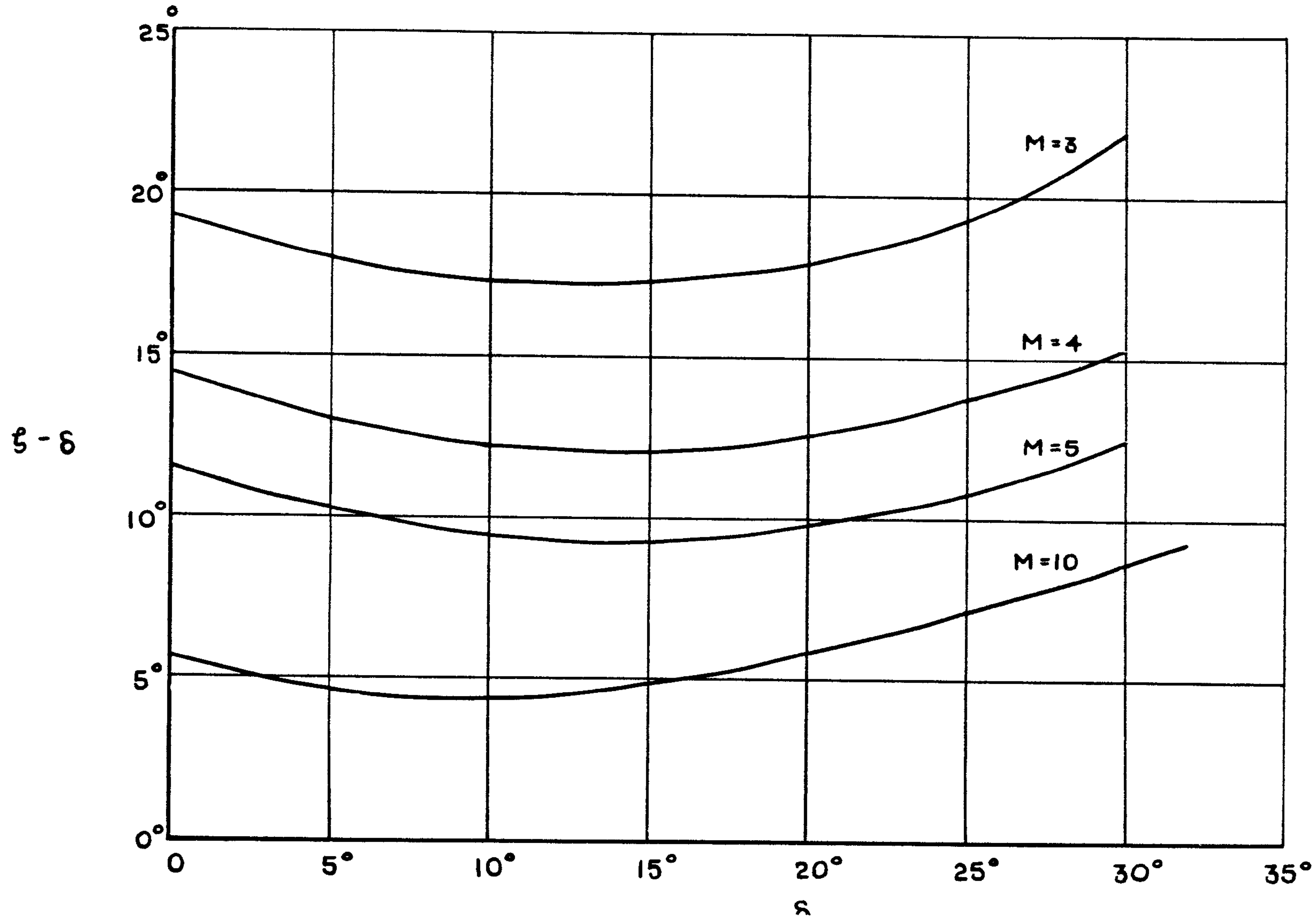


FIG. 9. VARIATION OF ANGLE BETWEEN SHOCK AND WING SURFACE.

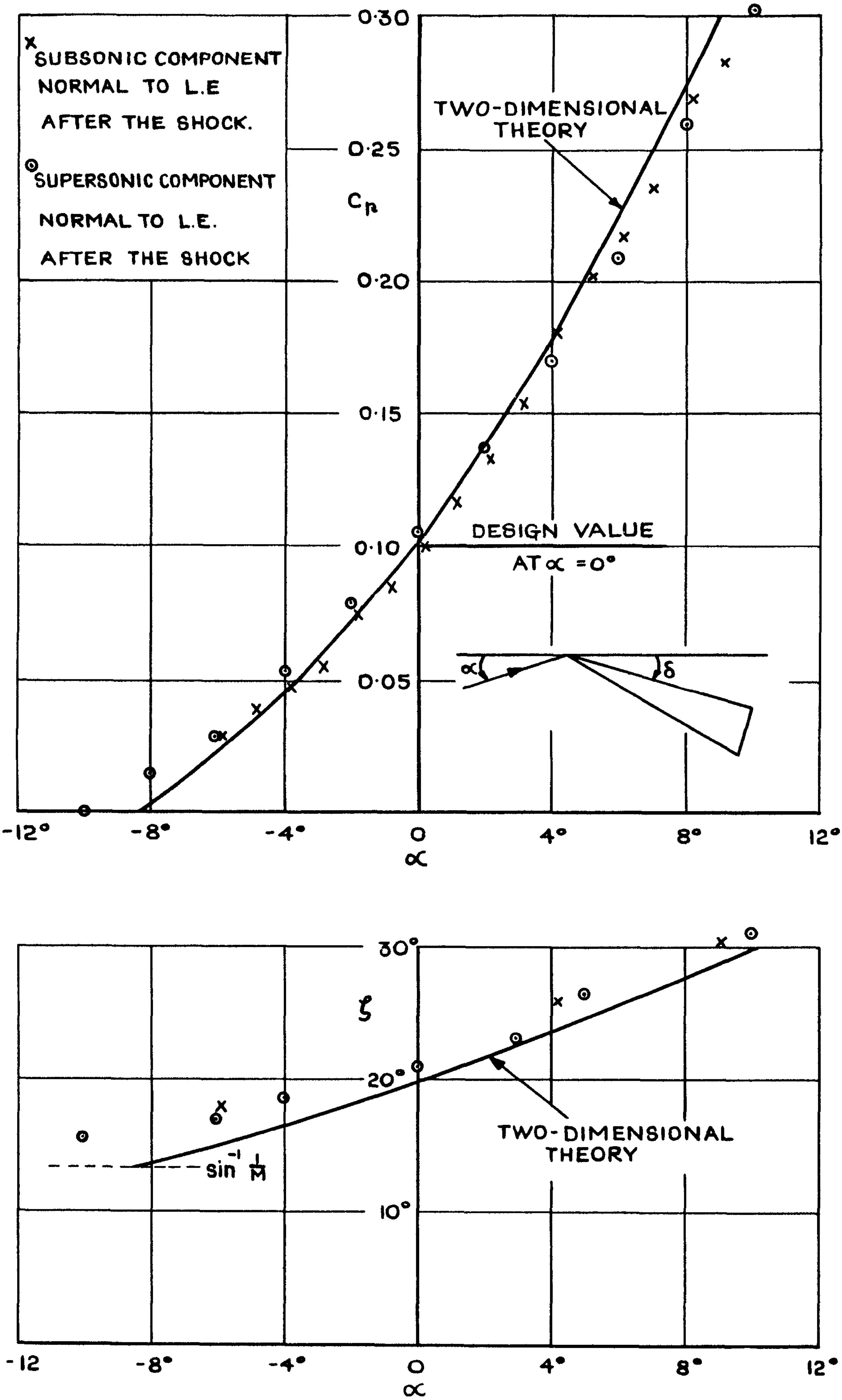


FIG. 10. PRESSURE COEFFICIENT AND SHOCK - WAVE ANGLE FOR NONWEILER WING M=4·3.

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533.6.011.72

ON THREE-DIMENSIONAL BODIES OF DELTA PLANFORM WHICH CAN SUPPORT PLANE ATTACHED SHOCK WAVES. Peckham, D.H. March, 1962.

This Note collects together in one report available theoretical work on bodies which can support attached plane shock waves, discusses some of the possible merits of such shapes, and includes some calculations illustrating their properties. Also, some preliminary results from wind tunnel tests are given, together with details of proposed future tests.

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