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SOME RESEARCH ON THE LIFT AND STABILITY OF WING-BODY COMBINATIONS

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

SOME RESEARCH ON THE LIFT AND STABILITY OF

WING-BODY COMBINATIONS[1]

By Paul E. Purser and E. M. Fields

SUMMARY

The present paper summarizes and correlates broadly some of the research results applicable to fin-stabilized ammunition. The discussion and correlation are intended to be comprehensive, rather than detailed, in order to show general trends over the Mach number range up to 7.0. Some discussion of wings, bodies, and wing-body interference is presented, and a list of 179 papers containing further information is included. The present paper is intended to serve more as a bibliography and source of reference material than as a direct source of design information.

INTRODUCTION

A large part of the research conducted by the National Advisory Committee for Aeronautics on the lift and stability of body-wing combinations has been aimed primarily at the problems of airplanes and missiles. Many of the programs, however, have been broad enough to encompass configurations of interest to designers of fin-stabilized ammunition. It is the purpose of the present paper to summarize and correlate broadly some of the research information obtained by the NACA and other research organizations. The discussion and correlation are intended to be comprehensive, rather than detailed, in order to show general trends over the Mach number range up to $M = 7.0$. The paper is thus intended to serve more as a bibliography and source of reference material than as a direct source of design information.

References 1 to 69 and a bibliography listing 110 additional papers present information on the subject of wings and bodies and interference effects. The type of information to be found in each paper is indicated in table I.

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[1]The information presented herein was previously made available to the Fin-Stabilized Ammunition Committee, Picatinny Arsenal, Dover, N. J., July 20, 1955.

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Briefly discussed in the present paper are some interference effects, some effects of geometric changes in isolated wings, and some isolated body effects. A comparison between theory and experiment is presented for some complete configurations wherein the interference effects are combined with isolated wing and body effects.

SYMBOLS

A	aspect ratio, $\frac{4s^2}{S_w}$
a	body radius, ft
c	wing chord, ft
c_r	wing root chord, ft
$\bar{c}$	wing mean aerodynamic chord, $\frac{\int_0^s c^2 dy}{\int_0^s c\ dy}$, ft
C_L	lift coefficient, $\frac{L}{qS_W}$ or $\frac{L}{qS_B}$
ΔC_L	increment in C_L
C_{L_α}	lift-curve slope per degree, $\frac{\partial C_L}{\partial \alpha}$
d	body diameter, ft
K	distance from configuration center of gravity to trailing edge of basic wing, ft
L	lift, lb
L_{α_B}	lift of isolated body due to angle of attack α, lb
L_{α_W}	lift of isolated wing due to angle of attack α, lb
$L_{\alpha_{W(B)}}$	lift of wing in presence of body, lb
$L_{\alpha_{B(W)}}$	lift of body in presence of wing, lb

$\Delta L_{\alpha_{B(W)}}$	interference lift on body in presence of wing, lb
$\Delta L_{\alpha_{W(B)}}$	interference lift on wing in presence of body, lb
l	length of fuselage, ft
l_c	length of cylindrical part of body behind nose, ft
l_n	length of nose, ft
M	Mach number, $\dfrac{V}{V_c}$
q	dynamic pressure, $\frac{1}{2}\rho V^2$, lb/sq ft
s	wing semispan, ft
S_W	wing plan-form area, sq ft
S_B	body cross-sectional area, sq ft
t	wing maximum thickness, ft
V	velocity, ft/sec
V_c	velocity of sound, ft/sec
x_{cp}	when used alone, distance to wing center of pressure measured from wing leading edge or apex, ft
$\dfrac{x_{cp}}{l}$	configuration center of pressure measured from nose, expressed in body lengths
$\dfrac{\Delta x_{cp}}{d}$	body center of pressure rearward movement due to addition of cylinder, expressed in body diameters (fig. 15)
$x_{cp}-x_{cg}$	distance from configuration center of gravity to center of pressure, ft
y	spanwise distance, ft
α	angle of attack, deg
ρ	density of air, slugs/cu ft

DISCUSSION

STATEMENT OF PROBLEM

In order to study the lift and stability of wing-body combinations, the complete configuration must be broken down into its component parts. Figure 1 shows such a breakdown for the lift. First, there is the isolated lift of the wing or fin L_{α_W} and that of the isolated body L_{α_B}. When the wing is in the presence of a body which is at an angle of attack, the upflow around the body induces an "interference" lift $\Delta L_{\alpha_{W(B)}}$ on the wing such that the total lift on the wing (interference plus angle of attack) is $L_{\alpha_{W(B)}}$; similarly, the lift on a wing at angle of attack induces a lift $\Delta L_{\alpha_{B(W)}}$ on the body such that the total body lift is $L_{\alpha_{B(W)}}$. The total lift is then the sum of the components $L_{\alpha_{W(B)}} + L_{\alpha_{B(W)}}$.

Each component lift has its center of pressure, and the center of pressure of the total configuration is found by proper summation of the moments of the component lifts about some reference point such as the center of gravity. This center of pressure of the total configuration is rearward of the center of gravity for a stable configuration, and the greater this distance, the greater the stability for a given lift.

The remainder of this discussion considers the various component lifts and centers of pressure and shows the degree of success achieved by some investigators in using such component data to calculate the lift and stability of wing-body combinations.

INTERFERENCE LIFT

The "interference" lift components are the interference wing lift in the presence of the body $\Delta L_{\alpha_{W(B)}}$ and the interference body lift in the presence of the wing $\Delta L_{\alpha_{B(W)}}$. Figure 2 presents values of these lift components calculated by linear theory and expressed as ratios of the total wing or body lift (that is, the lift due to both interference and angle of attack) to the isolated wing lift L_{α_W}. The horizontal scale is the ratio of body radius to wing semispan, defined as shown in the sketch on the right. As the relative body radius is increased both lift ratios increase.

The increase in $L_{\alpha_{W(B)}}$ is due to the upwash at the surface of the body. This upwash has a value equal to twice the angle of attack at the body surface but decreases in the spanwise direction. Thus, as the body is made larger relative to the wing, a larger proportion of the wing is immersed in regions of large upflow angles and the value of $L_{\alpha_{W(B)}}/L_{\alpha_W}$ approaches 2.0 as the body radius approaches the wing semispan.

The increase in $L_{\alpha_{B(W)}}$ is due to the increase in body area on which the lifting-pressure carryover from the wing to the body can act as the relative body size is increased. Of course, the limiting values of 2.0 for $L_{\alpha_{W(B)}}/L_{\alpha_W}$ and $L_{\alpha_{B(W)}}/L_{\alpha_W}$ are in themselves meaningless, since at a relative body radius of 1.0 there is no wing and thus no value for L_{α_W}. The optimum value of relative body radius exists when the increase in interference lift is balanced against the decrease in isolated wing lift. For rectangular wings this radius is approximately 0.4 semispan and the total lift is approximately 1.2 times the isolated wing lift, if the effects of decreased fin aspect ratio that occur as the body size is increased are not considered. For the more practical case in which aspect ratio must be considered, the optimum body radius would approach zero for the case of fins having aspect ratios of about 2 or less, where C_{L_α} varies almost directly with aspect ratio. For very high aspect ratios or very high Mach numbers, where C_{L_α} does not vary rapidly with aspect ratio, the optimum body radius might approach the value of 0.4 semispan.

See table I for papers containing additional information on wing-body interference.

ISOLATED COMPONENTS

Wings

Wings or fins may have an almost infinite variety of both plan-form and cross-sectional shapes. Three simple plan forms are used for illustration: the rectangular, the untapered sweptback, and the delta (or triangular) plan forms. Some effects on C_{L_α} of aspect ratio, cross-sectional shape, and end plates are discussed. The static stability of the complete configuration, as affected by the addition of wing chord, is discussed briefly. See table I for papers containing information on wings.

Effect of aspect ratio and plan form.- Figures 3 to 5 present plots of the variation with Mach number of the lift and center of pressure of rectangular, swept, and delta wings for aspect ratios of 1/2 to 4. The lift-curve slope is per degree, and the center-of-pressure locations are expressed as distance in fractions of the root-chord length behind the apex of the wings. The curves represent linear-theory (zero-thickness) values of C_{L_α} and x_{cp}/c_r, and the test points represent various experimental data presented to show the general level of agreement with the linear theory. Because lift curves for low aspect ratios are generally somewhat nonlinear, the experimental slopes have been taken over an angle-of-attack range of $\pm 4^{\circ}$. The experimental data are generally for wings having curved airfoil-section profiles and low values of thickness ratio (t/c between 0.03 and 0.10). Some of the experimental data in figures 3 to 5 represent an average of a number of test points, and some of the experimental data in figure 4 are for a wing having a taper ratio of 0.6.

The most important points to be noted on the theoretical curves for all three plan forms are the peak in lift-curve slope C_{L_α} at Mach numbers near 1.0, the marked decrease in C_{L_α} as the aspect ratio is reduced at the lower speeds, the decrease in C_{L_α} as the Mach number is increased above 1.0, and the general rearward movement of the center of pressure at supersonic speeds as compared with subsonic speeds. For the two untapered plan forms the center of pressure tends to move forward at Mach numbers near 1.0, and decreases in aspect ratio tend to move the center of pressure forward at all Mach numbers. For the delta wings, decreases in aspect ratio tend to move the center of pressure rearward at subsonic speeds and have no effect on center of pressure at supersonic speeds.

The chief differences between experiment and linear theory are attributable to the effects of finite thickness, and the use of more exact theory would reduce the differences considerably. At transonic speeds the finite thickness acts somewhat like an increase in Mach number - in figure 3, for example, the most forward location of the center of pressure occurs at a lower Mach number than theory predicts for a thin plate; the same is true for the lift-curve slope peak. The experimental center of pressure is generally ahead of the location predicted by linear theory at supersonic speeds. Except for the region just above $M = 1.0$, the experimental lift-curve slope tends to agree fairly well with the linear-theory value.

Effect of adding chord on configuration static stability.- When the available wing or fin span is limited, as in the case of ammunition with fixed fins, there is often a natural desire to increase the wing lift and thus the stability by increasing the wing area through increases in the wing chord. Figures 6 to 8 illustrate the effects of such wing-chord increases for the three wing plan forms considered, with the assumption that the trailing edge of the basic aspect-ratio-4 wing is at the rear of the body so that wing-chord increases move the leading edge forward. For simplicity, an imaginary body having no lift, no moment, and no interference is assumed so that only the wing lift and center-of-pressure location relative to the center of gravity affects the stability. Figure 6 shows the results of the analysis for rectangular wings. The distance between the imaginary center of gravity and wing center-of-pressure location is represented by $\left(x_{cp} - x_{cg}\right)/s$, the wing lift is represented by $S_w C_{L_\alpha}$, and the wing moment is represented by $S_w C_{L_\alpha}\left(x_{cp} - x_{cg}\right)/s$, all plotted against wing aspect ratio. Lengths and areas are all referenced to the semispan and area of the basic aspect-ratio-4 wing. Figure 6 shows that as the wing chord is increased, the moment arm decreases, the lift generally increases, and the resulting moment first increases and then decreases but peaks at different aspect ratios for different Mach numbers. At $M = 0.8$ the wing stability contribution peaks at $A \approx 1.5$ and as the Mach number is increased the optimum aspect ratio is decreased until at $M \approx 6$ it appears that $A < 1/2$ is the optimum. Figure 7 presents similar data for sweptback untapered wings and the general results are similar to those shown for rectangular wings. For delta wings (fig. 8) the results are again similar except that the optimum aspect ratio tends to be slightly higher than for the untapered wings for a given Mach number.

The specific values resulting from such an analysis will be changed for other plan forms and when wing-body interference and wing weight are taken into account. Another factor which would change the specific values is K/s, the distance from the configuration center of gravity to the trailing edge of the basic wing expressed in wing semispans. The curves of figures 6 to 8 are based on $K/s = 7.5$, and the optimum aspect ratio would be different for other values of K/s; for example, for the rectangular wing at $M = 6.4$ (fig. 6) the optimum aspect ratios would be approximately 0.3, 1.1, and 2.1 for K/s values of 10, 2, and 1, respectively.

The general trends, based on the preceding simplified analysis, appear to be as follows:

(1) There seems to exist an optimum aspect ratio for any fin plan form below which the addition of area by increased chord will reduce the stability contribution of the fin, even though the added area is behind the center of gravity.

(2) This optimum aspect ratio for the untapered fins seems to be slightly lower than for the delta fins and also decreases as the Mach number increases.

(3) Changing the aspect ratio by changing the fin chord seemed to have the greatest effect on the fin stability contribution at the lower Mach numbers.

(4) At the higher Mach numbers the penalty suffered by not choosing the optimum aspect ratio appears to be reduced.

Since the original presentation of the present paper, a study was made (see ref. 1) of the effects of adding fin chord for rectangular and delta fins on a body of fineness ratio 14. In reference 1, body lift and moment and interference effects were considered, and the results are in general agreement with the trends previously mentioned except that the optimum aspect ratio tended to be slightly higher for the untapered fin compared with that for the delta fin.

Effect of airfoil section.- Some information on the effects of airfoil-section thickness distribution is shown in figures 9 to 11. The data of figure 9 show a comparison of the lift and center-of-pressure location at transonic speeds for tapered wings with NACA 0003-63 and 3-percent-thick circular-arc airfoils. The circular-arc airfoil has less lift and a more forward center-of-pressure location. Although the most obvious geometric difference between the two sections is the nose shape, the difference in aerodynamic chracteristics is principally due to the less obvious difference in trailing-edge angle. The angle included between the upper and lower surfaces at the trailing edge is about twice as large for the circular-arc airfoil.

The effects of a more extreme difference in the trailing-edge angle are shown in figure 10 for subsonic and low supersonic Mach numbers. The effects on lift are smaller at supersonic speeds than at subsonic speeds. The effects on center-of-pressure location are somewhat less than shown in figure 9, but this may be because the critical transonic Mach number range is not covered by the data of figure 10.

The data presented in figure 11 are for a high supersonic Mach number ($M = 6.86$) and show the effects of changing from a double-wedge or diamond airfoil to a single wedge. The effects on both lift and center-of-pressure location are appreciable but are reasonably well calculated by shock-expansion theory. The agreement between experiment and linear theory is best, of course, at low angles of attack.

Effect of end plates.- Wings or fins have often been equipped with end plates in order to reduce the tip losses, or increase the effective aspect ratio, and thus increase the lift-curve slope of the wing or fin. Figure 12 presents some data showing the effect of an end plate on the lift of a tapered wing swept back 20° at the quarter chord. The supersonic theory for wings with end plates corresponds to the infinite-aspect-ratio value at and above the Mach number at which the particular end plate considered completely covers the tip Mach cone.

Although an end plate of reasonable size provides a considerable increase in C_{L_α}, the increase is still not a large part of the potential gain at subsonic speeds. At supersonic speeds the same end plate will provide a greater percentage of the potential gain but the potential becomes smaller as Mach number is increased and it would appear not very worthwhile to consider end plates at the higher supersonic speeds.

Bodies

It is in the field of bodies that the configurations used in basic research probably bear the least resemblance to practical fin-stabilized ammunition. Data do exist, however, on certain basic shapes and these data are roughly correlated herein. Because of the scatter involved in the various experimental data, the body-characteristics curves to be presented should be considered to be illustrative material showing general trends rather than design charts. See table I for papers containing information on bodies.

Lift of cones and cone-cylinders.- The upper part of figure 13 shows C_L (based on body frontal area) at an angle of attack of 4° plotted against Mach number for pure cones having fineness ratios from 3 to 7. The data showed a scatter of approximately 0.02 in C_L, and the faired line is slightly lower (about 7 percent) than the linear-theory value. The angle of attack of 4° was chosen because of the basic nonlinearity of body lift data and because a considerable portion of the center-of-pressure data did not extend to lower angles of attack. The lower part of figure 13 shows the increment in C_L at an angle of attack of 4° due to the addition of cylinders of various fineness ratios to the cones. In a number of cases the curves in the lower part of figure 13 were obtained by subtracting the average value of the cone lift from the lift of the cone-cylinder combination, and this would account for part of the scatter of 0.03 to 0.04 in C_L.

For the pure cones, no consistent effects of cone fineness ratio or Mach number could be detected in the experimental data for $\alpha = 4^\circ$; at relatively high angles of attack, this would not necessarily be true. For the incremental lift due to the addition of a cylinder, the effects of fineness ratio and Mach number were more apparent. Increasing the length of the cylinder increased the lift, as might be expected. Increasing the Mach number increased the lift, except above $M \approx 5$, where the lift began to decrease somewhat.

Lift of ogives and ogive-cylinders.- Figure 14 is the ogive and ogive-cylinder counterpart of figure 13. The data cover about the same ranges of Mach number and cylinder fineness ratio and have about the same scatter in C_L as noted for the cone-nose bodies.

The lift of the pure ogives is about 20 percent greater than for the pure cones and is about 15 percent greater than the linear-theory value. The incremental lift of cylinders behind ogives is slightly less than the incremental lift of cylinders behind cones shown in figure 13, and the data do not cover a sufficient Mach number range to show whether there is a drop in ΔC_L above $M \approx 5$ as was shown for the cylinders behind cones.

Center of pressure.- The upper part of figure 15 shows the center-of-pressure locations, at $\alpha = 4^{\circ}$, for pure cones and pure ogives. The lower part of figure 15 shows the increment in center-of-pressure location resulting from the addition of cylinders behind cones or ogives. Center of pressure is difficult to measure accurately, and there was considerable scatter in the data from which figure 15 was prepared. The data for the pure cones and pure ogives showed a satisfactorily small scatter of less than 10 percent of the length (less than 1/2 diameter) for center-of-pressure location. For the nose-cylinder combinations, the scatter was generally less than 3/4 diameter although several data points showed a considerably larger scatter. The scatter of the data effectively masked any effects of nose shape and small changes in Mach number, so that single curves are presented for rather large speed ranges and for both nose shapes. The curves shown in the lower part of figure 15 should be considered to be qualitative only.

The upper part of figure 15 shows that pure cones have more rearward center-of-pressure locations than do ogives and that neither nose fineness ratio nor Mach number has a consistent effect on the center-of-pressure location. Adding cylinders behind cones or ogives moves the center of pressure rearward; the increment increases with cylinder length and Mach number.

Rear end modification.- The following general trends have been noted for boattails and flares:

(1) Boattailing the rear of the body reduces the lift and moves the center of pressure forward.

(2) Flaring the rear of the body increases the lift and moves the center of pressure rearward. A flared rear end has successfully been used to provide static and dynamic stability in free flight at Mach numbers up to 10 (ref. 2).

(3) Increasing the length of the flare by including more of the cylinder length without changing the base diameter may reduce the stability contribution (ref. 3) by moving the center of pressure forward while increasing the lift slightly.

CHARACTERISTICS OF COMPLETE CONFIGURATIONS

Having disposed, however roughly, of the various components of the lift and stability of body-wing combinations, the complete configuration may now be considered with particular attention paid to the general degree of success attained in calculating the characteristics of simple body-wing combinations. The general procedures used in the calculations were

(1) Use of the theories of Spreiter, Nielsen, Tucker, et al., to calculate the interference effects

(2) Use of the measured component C_L and center-of-pressure values for the body and fins, or use of interpolation or extrapolation of experimental data (guided by theory), or use of fairly exact theory to calculate component values

(3) Summation of the component values and comparison with experimental results

Lift

Figure 16 shows a comparison of calculated and experimental values of C_{L_α} for simple wing-body combinations. Data for unswept wings (tapered and rectangular) and delta wings are shown for subsonic (solid symbols) and supersonic speeds (open symbols). The agreement between calculation and experiment is good, with most of the points lying within 10 percent of the line of perfect correlation and there is no variation in quality of agreement with plan form or Mach number.

Center of Pressure

Figure 17 is the center-of-pressure counterpart of figure 16 and the data cover the same plan form and Mach number range. The agreement between calculation and experiment is good, all points lying within 10 percent of the line of perfect correlation, and again no Mach number or plan-form trends are evident.

CONCLUDING REMARKS

The present paper summarizes and correlates broadly some of the research results applicable to fin-stabilized ammunition. The discussion and correlation are intended to be comprehensive, rather than detailed, in order to show general trends over the Mach number range up to 7.0. The present paper is intended to serve more as a bibliography and source of reference material than as a direct source of design information.

The foregoing discussion represents a brief digest of a large amount of data. The summary figures presented are considered to be suitable only for trend studies or first-order calculations of lift and stability of particular wing-body combinations. Most of the comparisons of experiment with theory have been based on linear theory for simplicity; improved agreement will generally be obtained by use of more exact theories.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., July 6, 1955.

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TABLE I.- CLASSIFICATION OF INFORMATION CONTAINED IN REFERENCES AND BIBLIOGRAPHY

[T, theoretical; X, experimental]

Wing	Body	Wing-body combination	Wing-body interference	Reference or bibliography number
Subsonic speeds				
T				Ref. 5
X				Ref. 13
X				Ref. 15
T				Ref. 17
X				Ref. 18
	X	T,X	T,X	Ref. 46
	X	X		Ref. 52
X				Bib. 1
T				Bib. 2
T				Bib. 3
T,X				Bib. 4
T	T	T		Bib. 5
X				Bib. 6
	T,X			Bib. 7
T,X	T,X	T,X	T,X	Bib. 8
		X		Bib. 9
	T,X			Bib. 10
	X	X		Bib. 11
	T,X			Bib. 12
X		X		Bib. 13
	X	X		Bib. 14
	X	X		Bib. 15
			T,X	Bib. 16
T,X				Bib. 17
	X	X		Bib. 18
T,X	T,X	T,X		Bib. 19
Subsonic and transonic speeds				
X				Ref. 7
T,X				Bib. 20
	X	X		Bib. 21
		X		Bib. 22
		X		Bib. 23
X				Bib. 24
Subsonic, transonic, and supersonic speeds				
		T,X		Ref. 1
	X			Ref. 2
T,X		T,X		Ref. 9
		X		Ref. 23
	T,X			Ref. 37
		T,X	T,X	Ref. 68
		T		Bib. 25
X		T,X	X	Bib. 26
	X			Bib. 27
		T,X		Bib. 28
		T,X		Bib. 29
T,X	T,X	T,X	T,X	Bib. 30

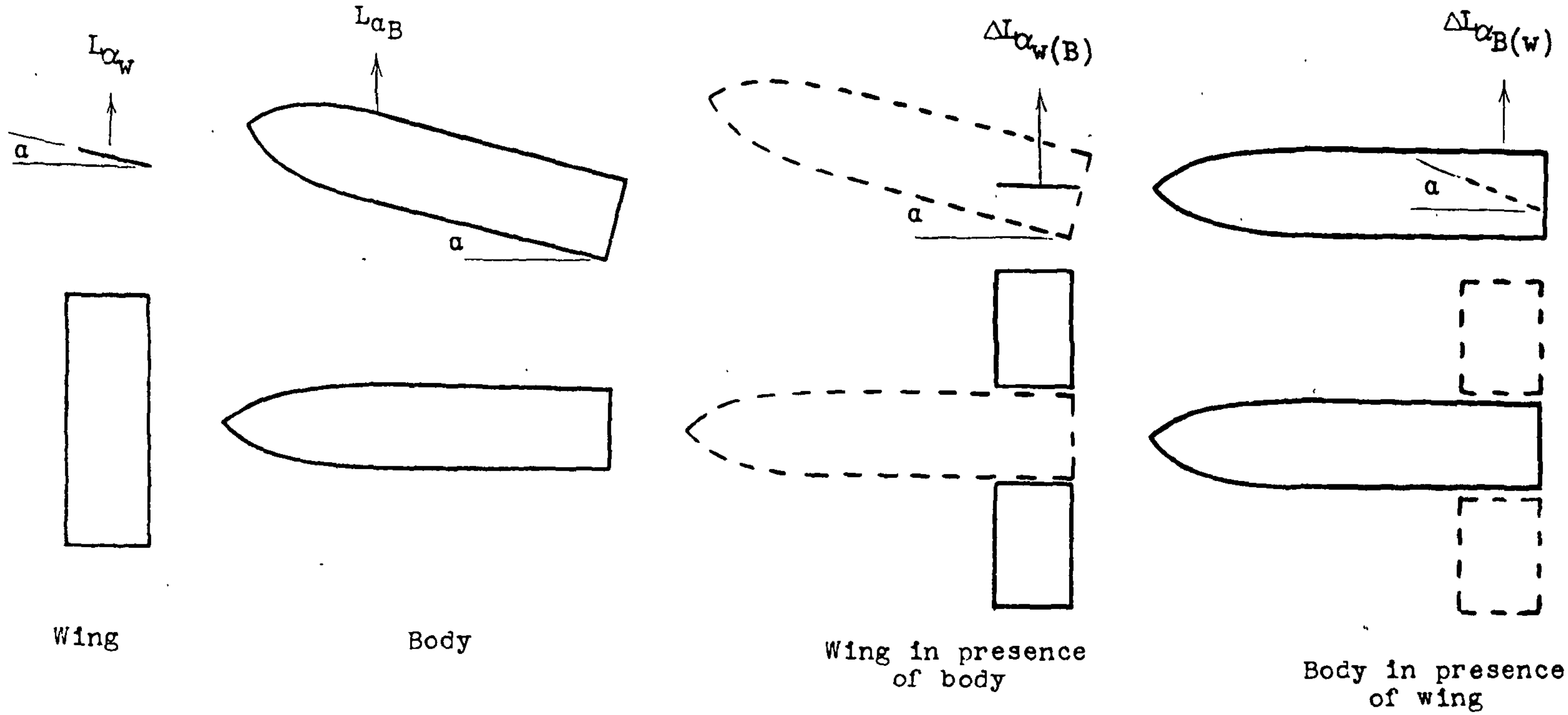
Wing	Body	Wing-body combination	Wing-body interference	Reference or bibliography number
Subsonic and supersonic speeds				
T,X				Ref. 12
	X	X		Ref. 56
		X		Bib. 31
Transonic speeds				
X				Ref. 14
X		X		Ref. 22
T,X				Ref. 24
	T,X			Ref. 47
	X	T,X		Ref. 49
	X	X		Ref. 59
		T		Bib. 32
X				Bib. 33
X				Bib. 34
	X	X		Bib. 35
		X		Bib. 36
		X	X	Bib. 37
		X		Bib. 38
	X	X		Bib. 39
		T,X		Bib. 40
	X			Bib. 41
	X			Bib. 42
Transonic and supersonic speeds				
		X		Bib. 43
		X		Bib. 44
	X			Bib. 45
Supersonic speeds				
	T			Ref. 3
		T,X	T,X	Ref. 4
T				Ref. 6
T,X				Ref. 8
T,X				Ref. 10
T				Ref. 11
T,X		X		Ref. 16
T,X				Ref. 19
X				Ref. 20
T,X				Ref. 21
	T,X			Ref. 25
	T,X			Ref. 26
	T,X			Ref. 27
	X	X		Ref. 28
	T,X			Ref. 29
	T,X	T,X		Ref. 30
	T,X			Ref. 31
	T,X			Ref. 32

TABLE I.- CLASSIFICATION OF INFORMATION CONTAINED IN REFERENCES AND BIBLIOGRAPHY - Concluded

[T, theoretical; X, experimental]

Wing	Body	Wing-body combination	Wing-body interference	Reference or bibliography number	Wing	Body	Wing-body combination	Wing-body interference	Reference or bibliography number
Supersonic speeds - Continued					Supersonic speeds - Concluded				
	X			Ref. 33		X	T,X	X	Bib. 64
	X	X		Ref. 34			T,X		Bib. 65
	X	X		Ref. 35		X	X		Bib. 66
	X			Ref. 36			T,X		Bib. 67
	X			Ref. 38		X			Bib. 68
	T,X			Ref. 39		T,X	T,X		Bib. 69
	T,X			Ref. 40			X		Bib. 70
	T,X	T,X		Ref. 41	T,X		T,X		Bib. 71
	T,X	T,X		Ref. 42			T,X		Bib. 72
	T,X	X		Ref. 43		T,X			Bib. 73
	T,X			Ref. 44		X			Bib. 74
	X	X		Ref. 45		T,X			Bib. 75
	T,X			Ref. 48		X	X		Bib. 76
	X	X		Ref. 50			T,X		Bib. 77
	X	X		Ref. 51		T,X	T,X		Bib. 78
	X	X		Ref. 53		T,X			Bib. 79
	X			Ref. 54	X	X	X	T,X	Bib. 80
	X	X		Ref. 55		T,X	T,X		Bib. 81
	X	X		Ref. 57			X		Bib. 82
	X	X		Ref. 58			X		Bib. 83
	X	X		Ref. 60			T,X		Bib. 84
	X	X		Ref. 61			T,X		Bib. 85
		X		Ref. 62			T		Bib. 86
	X			Ref. 63		X	X		Bib. 87
	X	X		Ref. 64			T,X		Bib. 88
T,X	T,X	T,X	T,X	Ref. 65			X		Bib. 89
			T,X	Ref. 66	T				Bib. 90
T		T,X	T,X	Ref. 67			T		Bib. 91
		T,X	T,X	Ref. 69			T		Bib. 92
	T,X			Bib. 46		X	X		Bib. 93
T				Bib. 47		X	X		Bib. 94
	T		T	Bib. 48		T,X			Bib. 95
T,X	T,X	X	X	Bib. 49		X			Bib. 96
T,X				Bib. 50		X	X		Bib. 97
	T,X			Bib. 51			X		Bib. 98
	T,X			Bib. 52	T				Bib. 99
	T,X			Bib. 53			T	T	Bib. 100
	X	X	X	Bib. 54			X		Bib. 101
	X	X	T	Bib. 55			X		Bib. 102
	X	X	X	Bib. 56		X	X		Bib. 103
	T,X	T,X		Bib. 57			X		Bib. 104
	T,X			Bib. 58				T	Bib. 105
	T,X			Bib. 59		T	T		Bib. 106
		X		Bib. 60		T,X			Bib. 107
		T,X		Bib. 61		T,X			Bib. 108
T,X				Bib. 62		T,X			Bib. 109
	T,X			Bib. 63		T,X			Bib. 110

[redacted]



$$C_{L_\alpha} = \frac{L_{\alpha_{W(B)}} + L_{\alpha_{B(W)}}}{qS_W} \quad \text{where} \quad L_{\alpha_{W(B)}} = L_{\alpha_W} + \Delta L_{\alpha_{W(B)}} \quad \text{and}$$

$$L_{\alpha_{B(W)}} = L_{\alpha_B} + \Delta L_{\alpha_{B(W)}}$$

Figure 1.- Lift components on wing-body combinations.

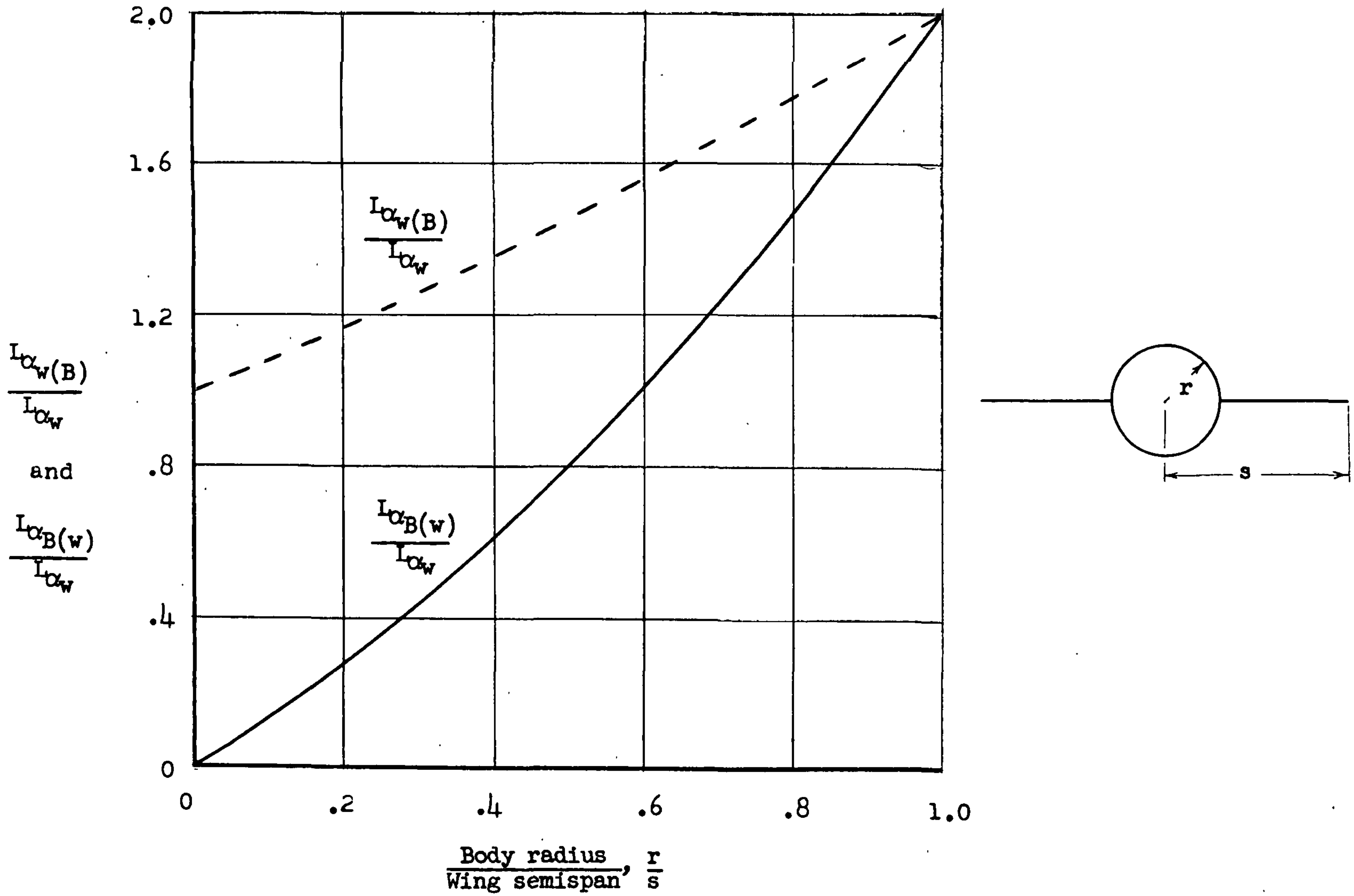


Figure 2.- Linear-theory values of wing-body lift interference. (See ref. 4.)

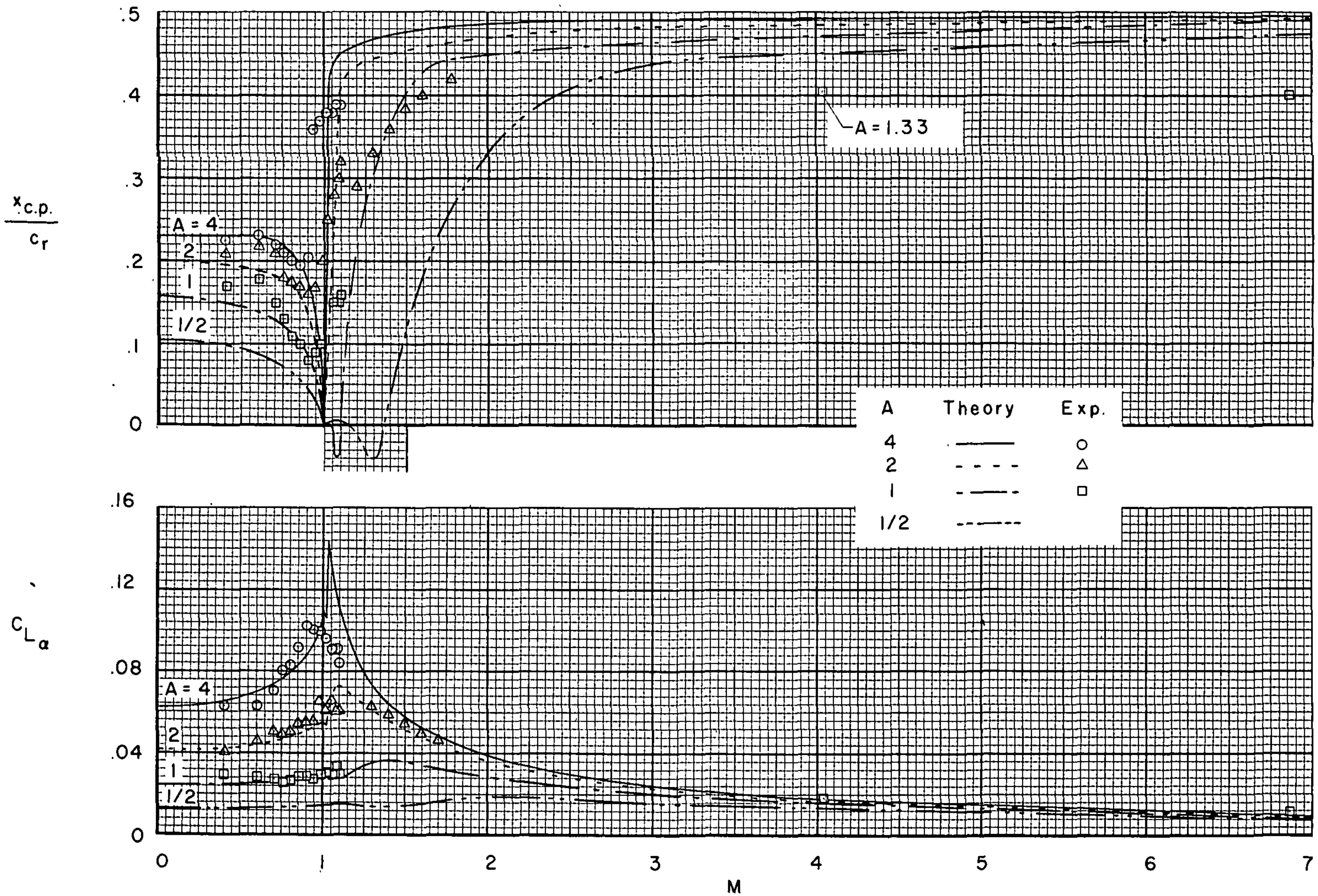


Figure 3.- Lift and center of pressure for rectangular wings. (Theory from refs. 5 and 6; experimental data from refs. 7 to 10.)

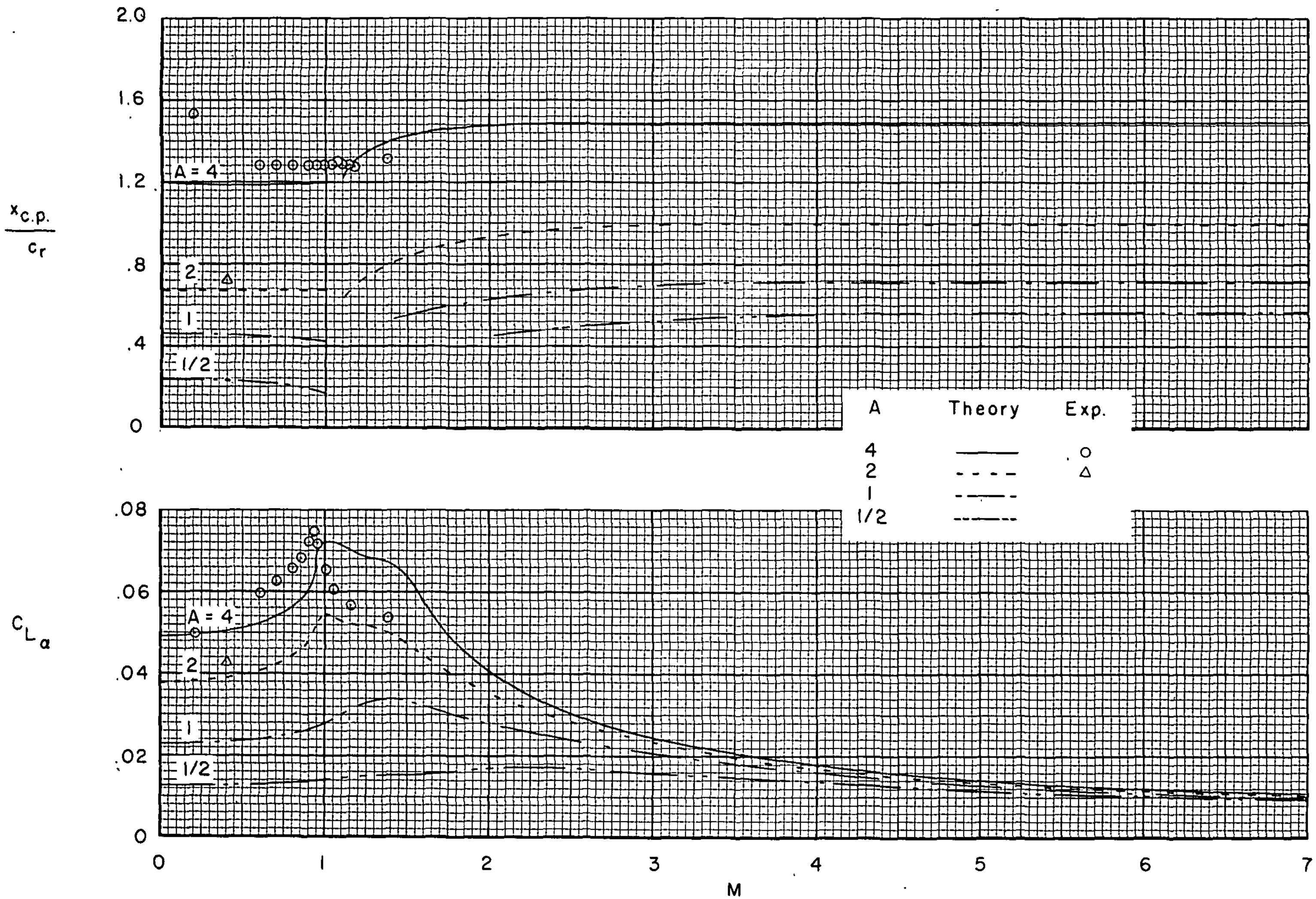


Figure 4.- Lift and center of pressure for untapered 45° sweptback wings. (Theory from refs. 6, 11, and 12; experimental data from refs. 13 to 16.)

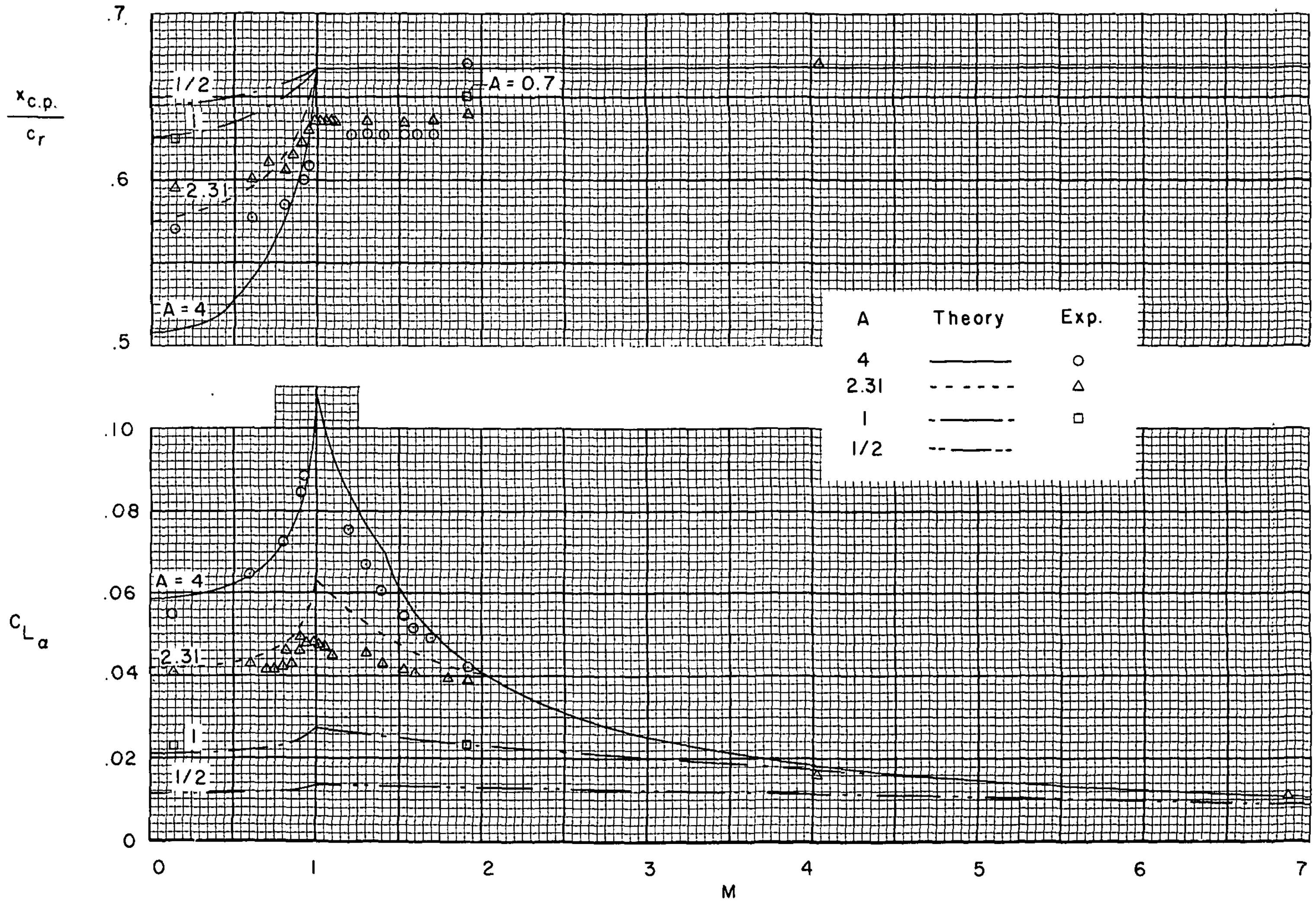


Figure 5.- Lift and center of pressure for delta wings. (Theory from refs. 5, 6, and 17; experimental data from refs. 9 and 18 to 21.)

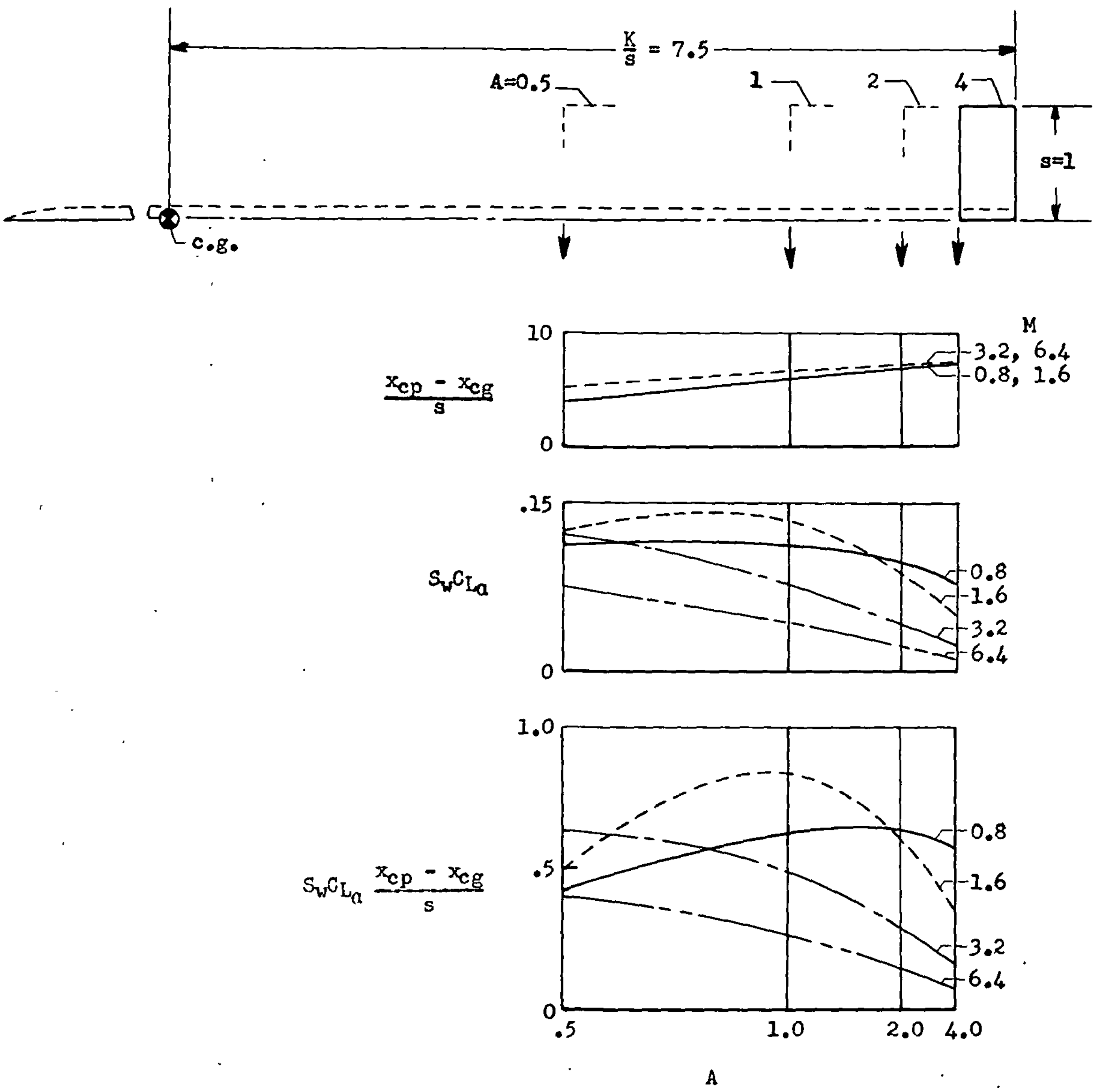


Figure 6.- Effect on static stability of adding wing chord when trailing edge of basic aspect-ratio-4 wing is at rear end of body. Rectangular wings.

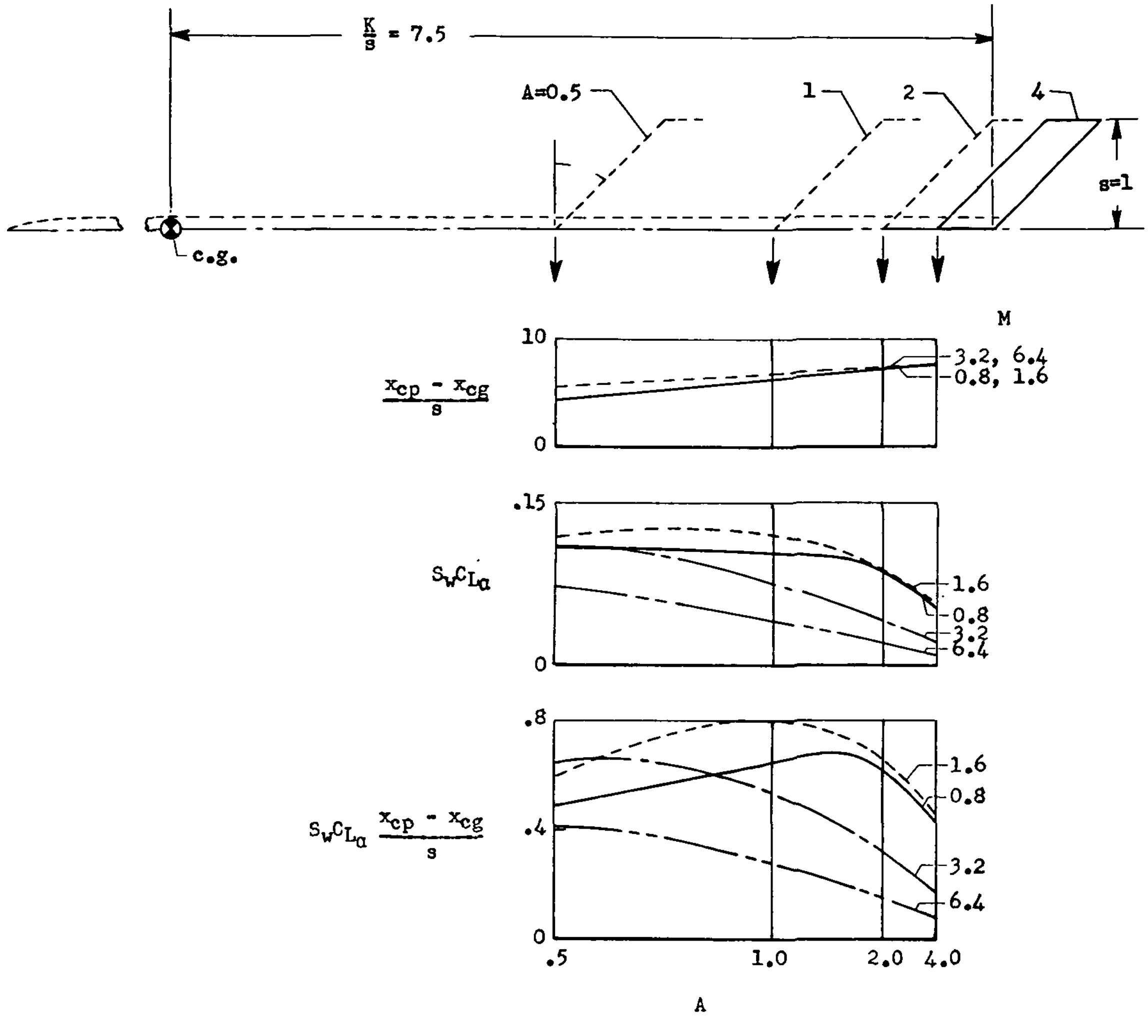


Figure 7.- Effect on static stability of adding wing chord when trailing edge of basic aspect-ratio-4 wing is at rear end of body. 45° swept-back untapered wings.

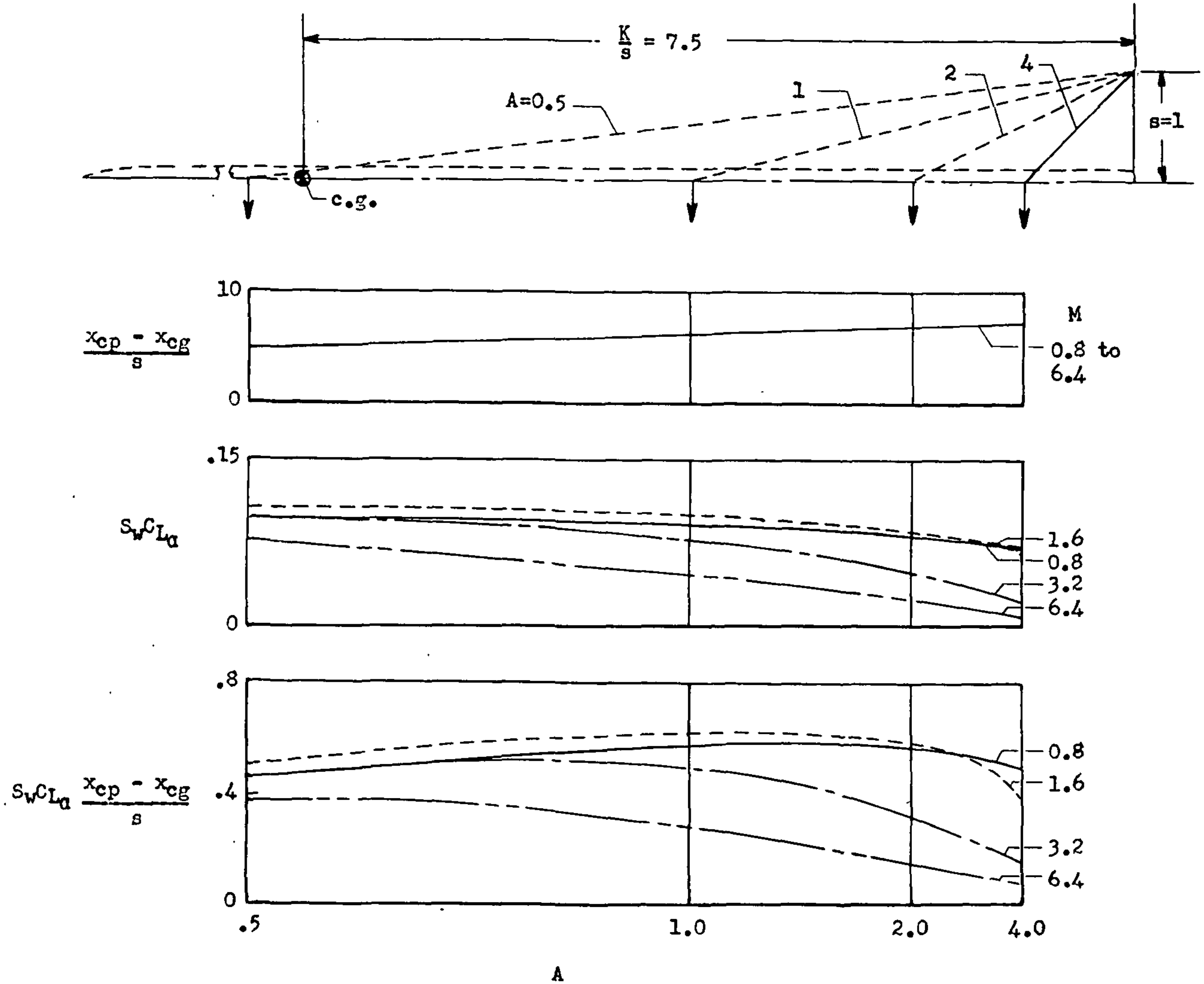


Figure 8.- Effect on static stability of adding wing chord when trailing edge of basic aspect-ratio-4 wing is at rear end of body. Delta wings.

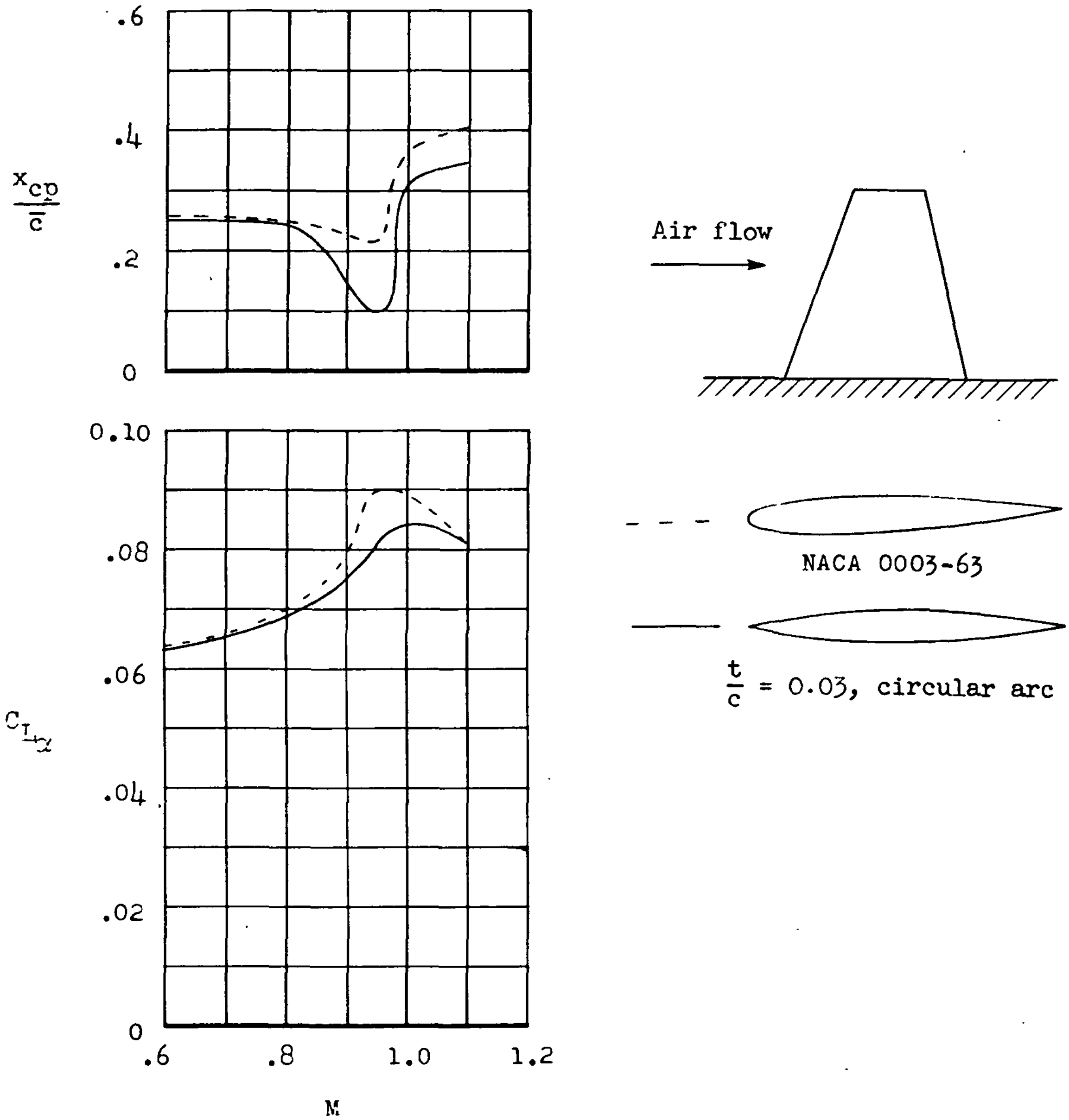


Figure 9.- Effect of airfoil section at transonic speeds. A = 3.1; taper ratio = 0.4; unswept wing alone. (See ref. 22.)

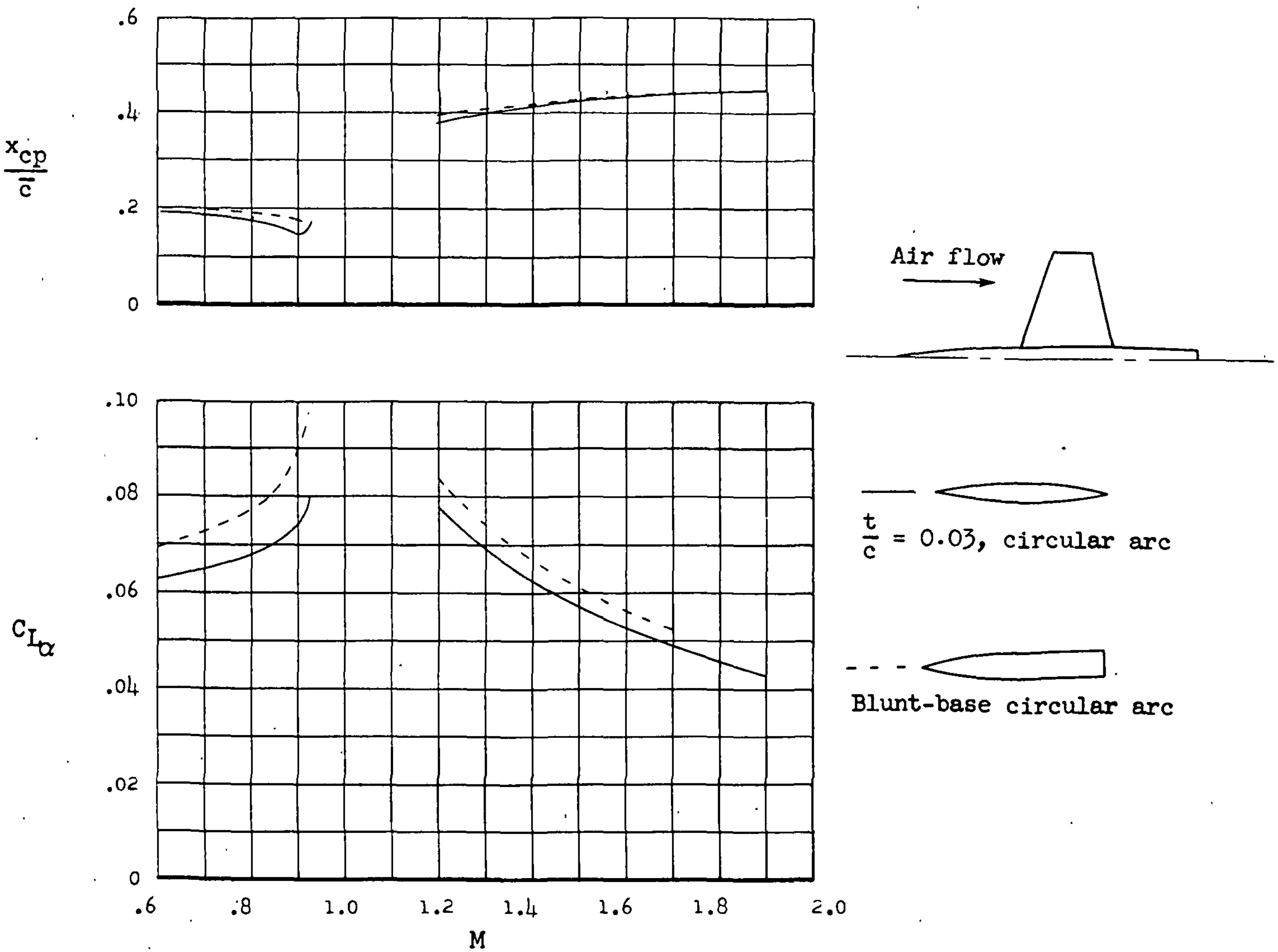


Figure 10.- Effect of airfoil section at subsonic and supersonic speeds. A = 3.1; taper ratio = 0.4; unswept-wing—body combination. (See ref. 23.)

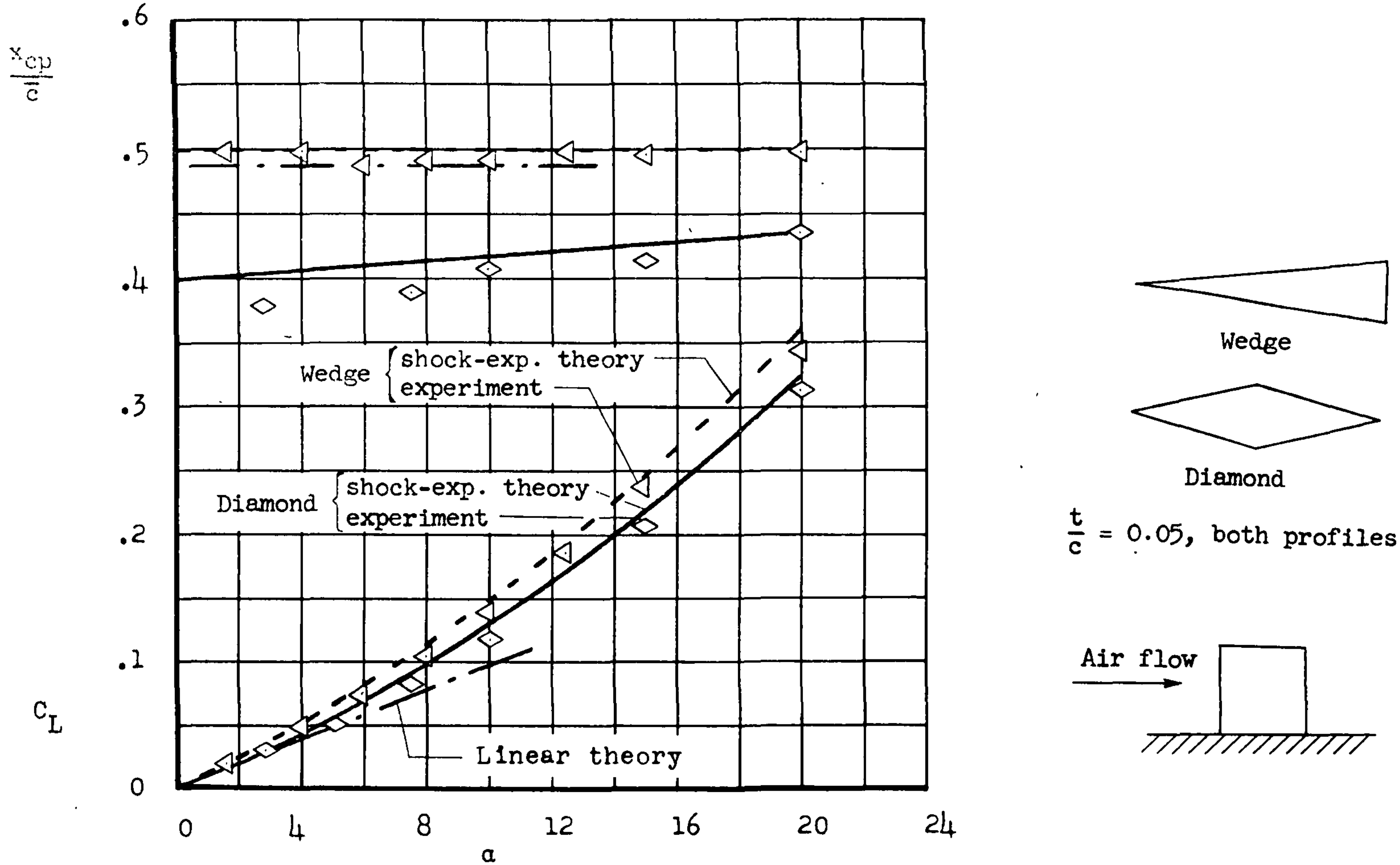


Figure 11.- Effect of airfoil section at M = 6.86. A = 1.0; wing alone. (See ref. 8.)

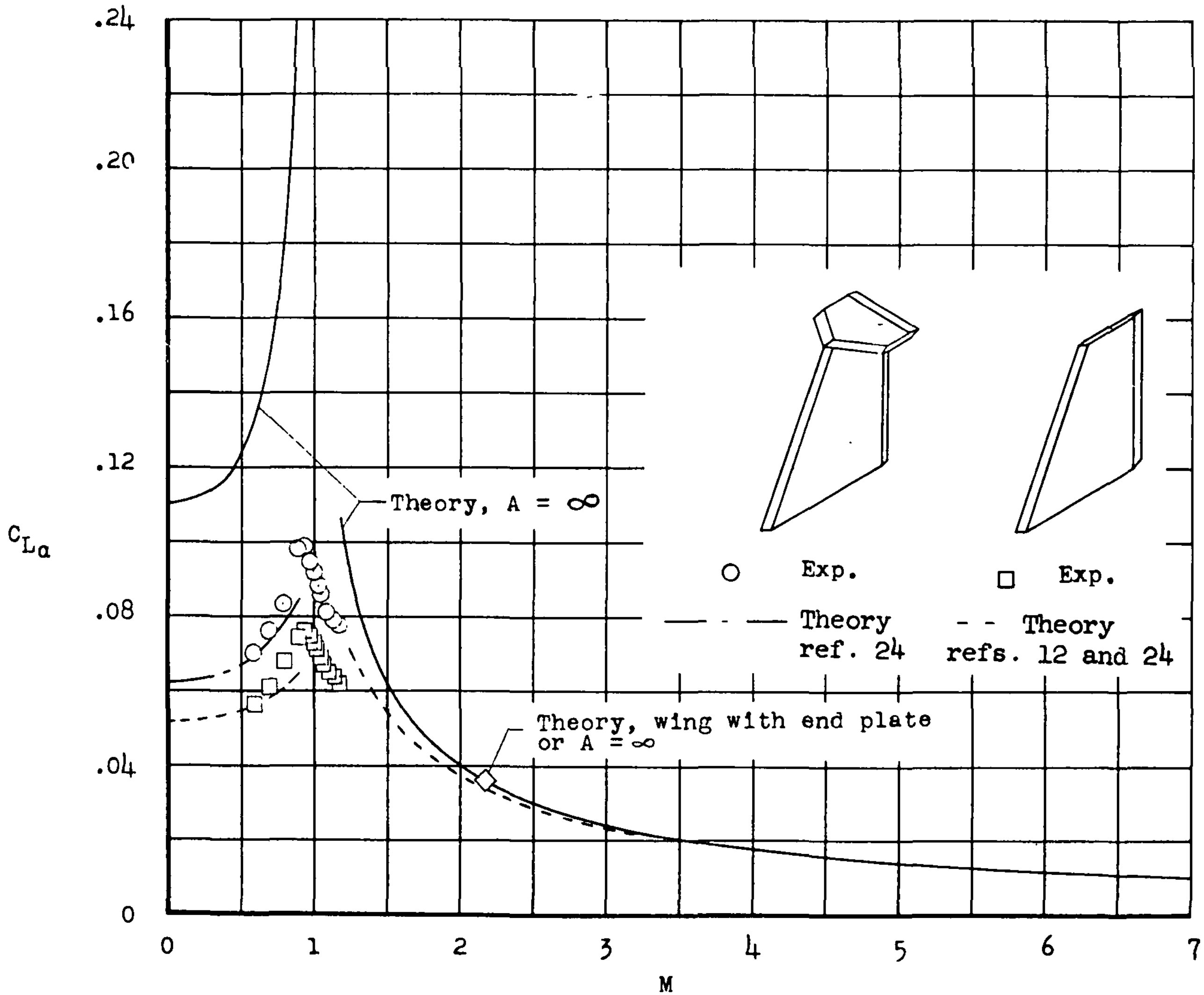


Figure 12.- Effect of end plate on lift. (Experimental data from ref. 24.)

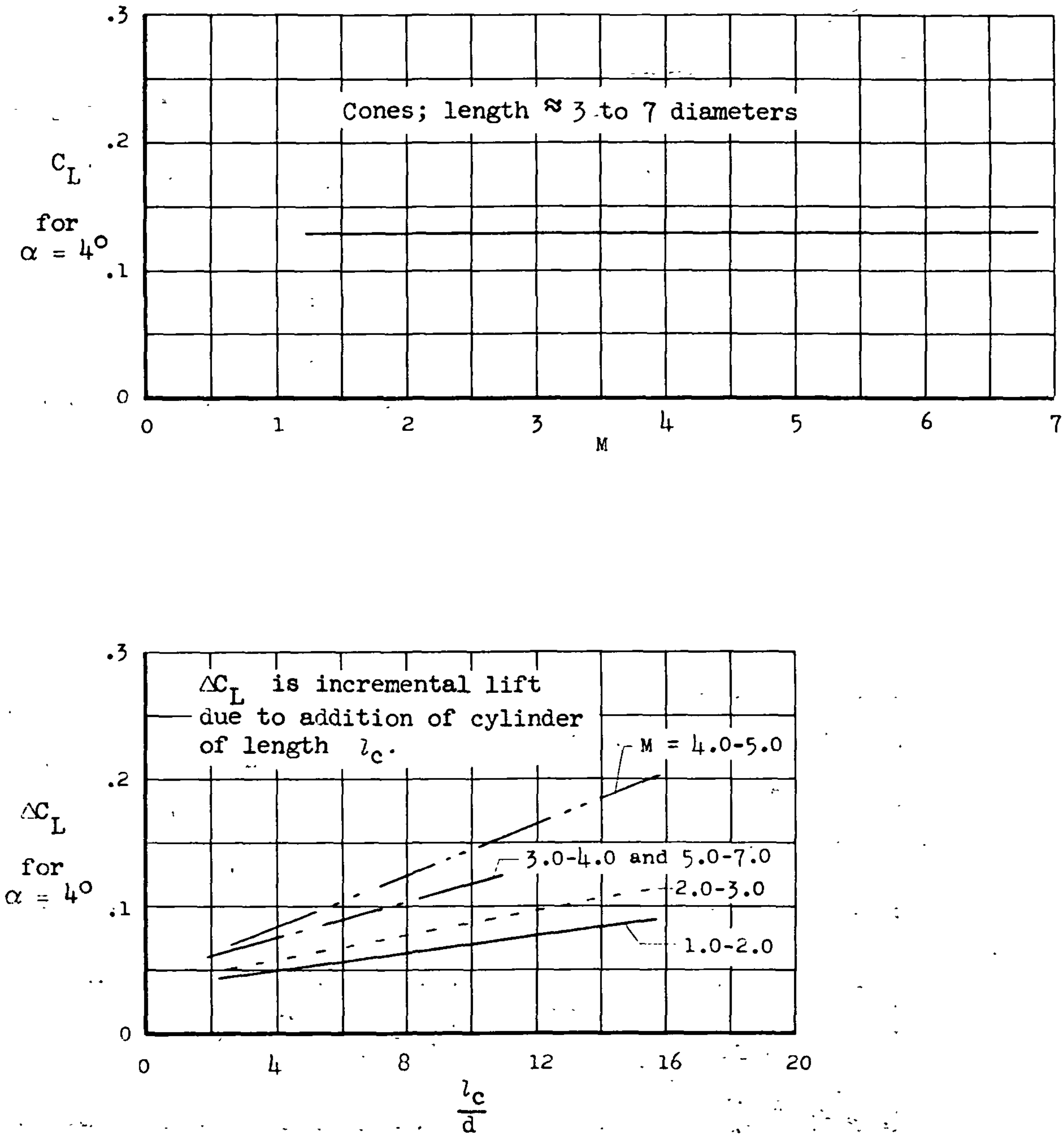
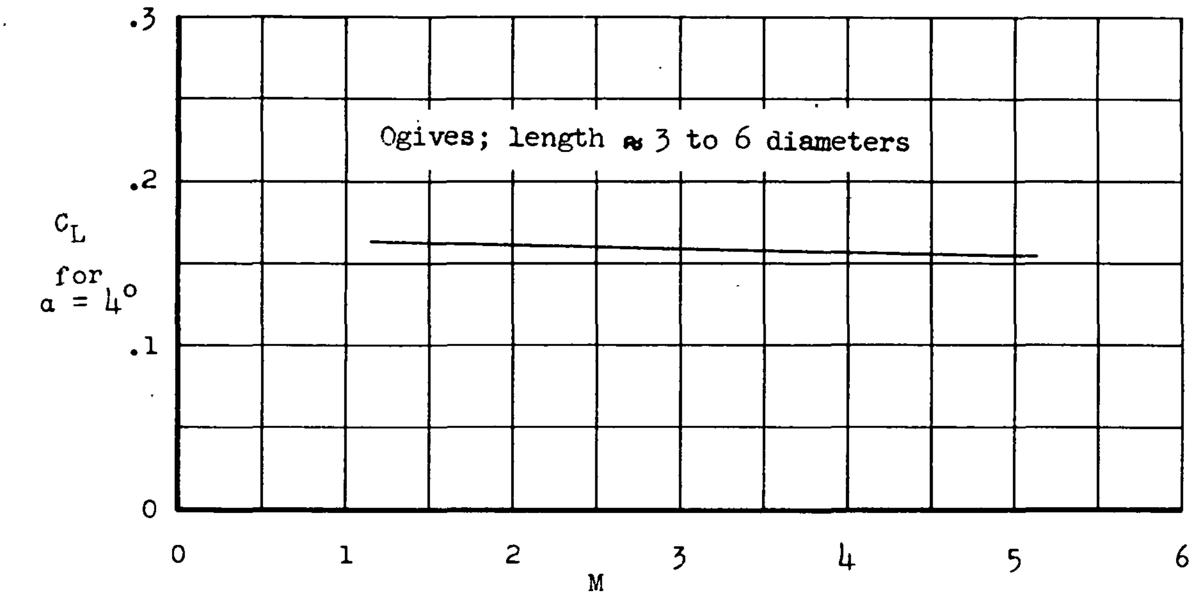


Figure 13.- Cone-cylinder lift. (Basic data for these curves may be found in refs. 25 to 39.)



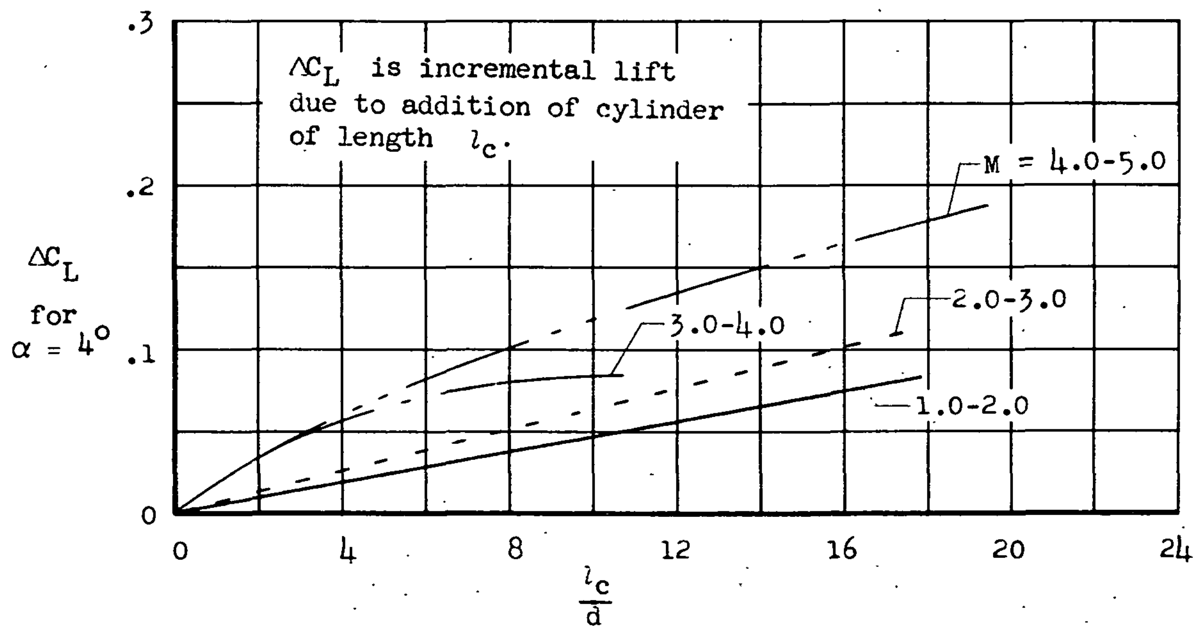


Figure 14.- Ogive-cylinder lift. (Basic data for these curves may be found in refs. 27, 31, 33, 34, 36, 38, 39, and 40 to 63.)

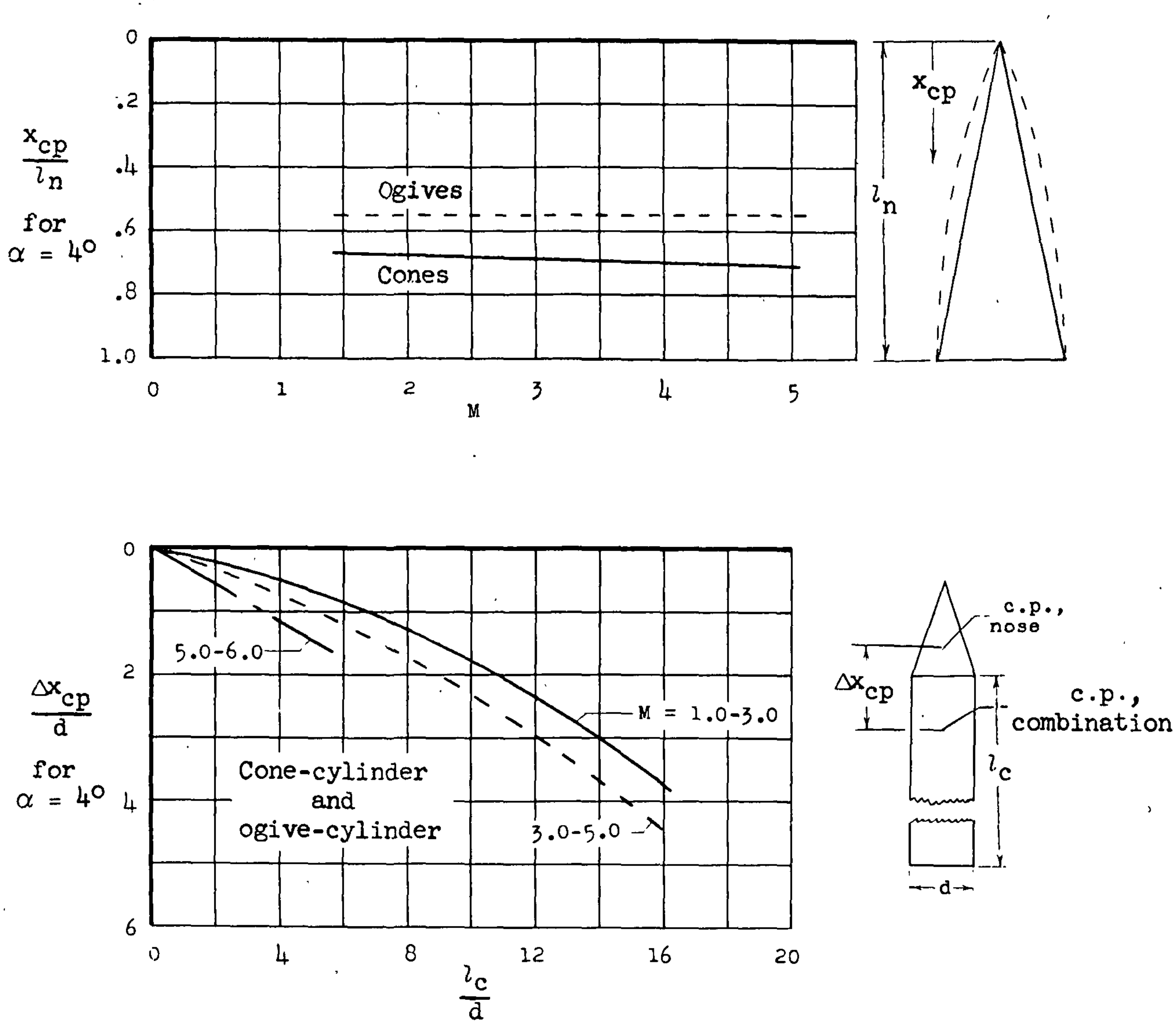


Figure 15.- Center of pressure for cone-cylinders and ogive-cylinders. (Basic data may be found in refs. 25 to 28, 30 to 39, 41 to 48, 50, 56, and 58 to 64.)

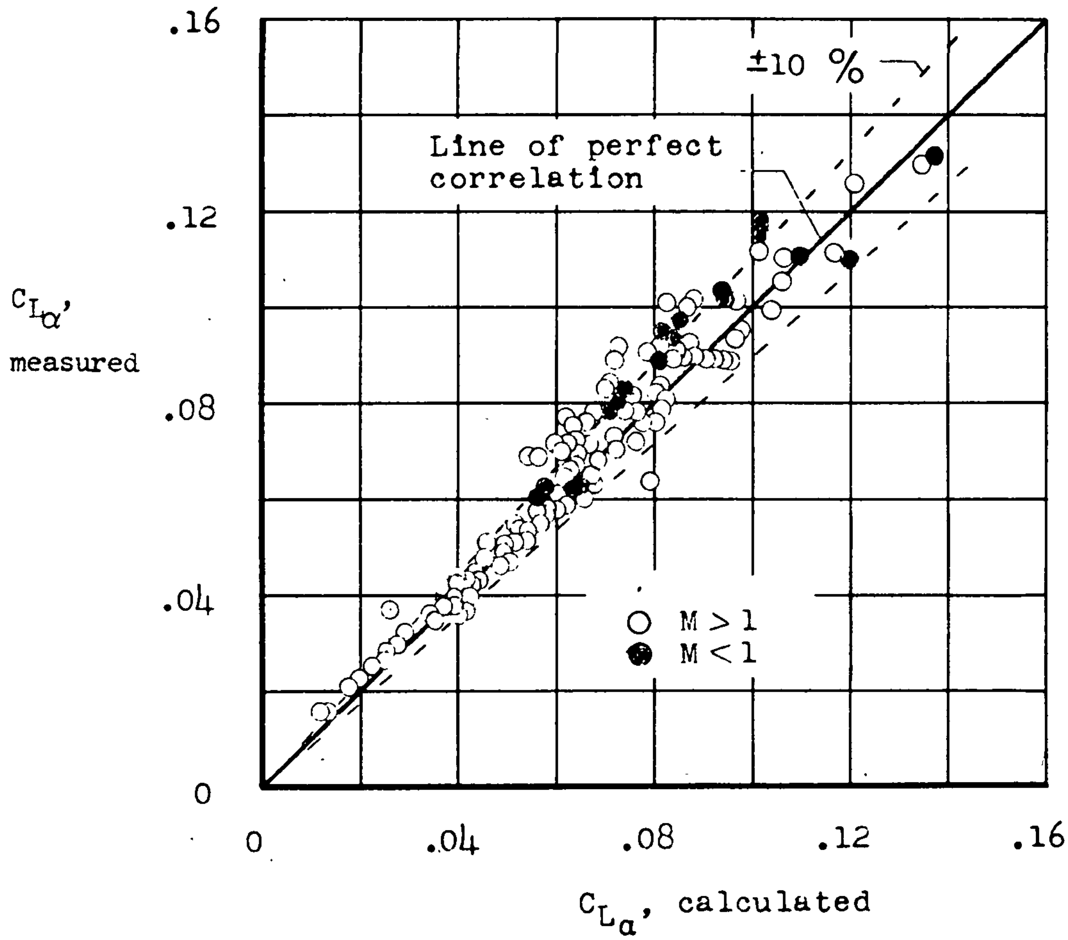


Figure 16.- Comparison between measured and calculated lift for complete wing-body combination. (See refs. 65 to 68.)

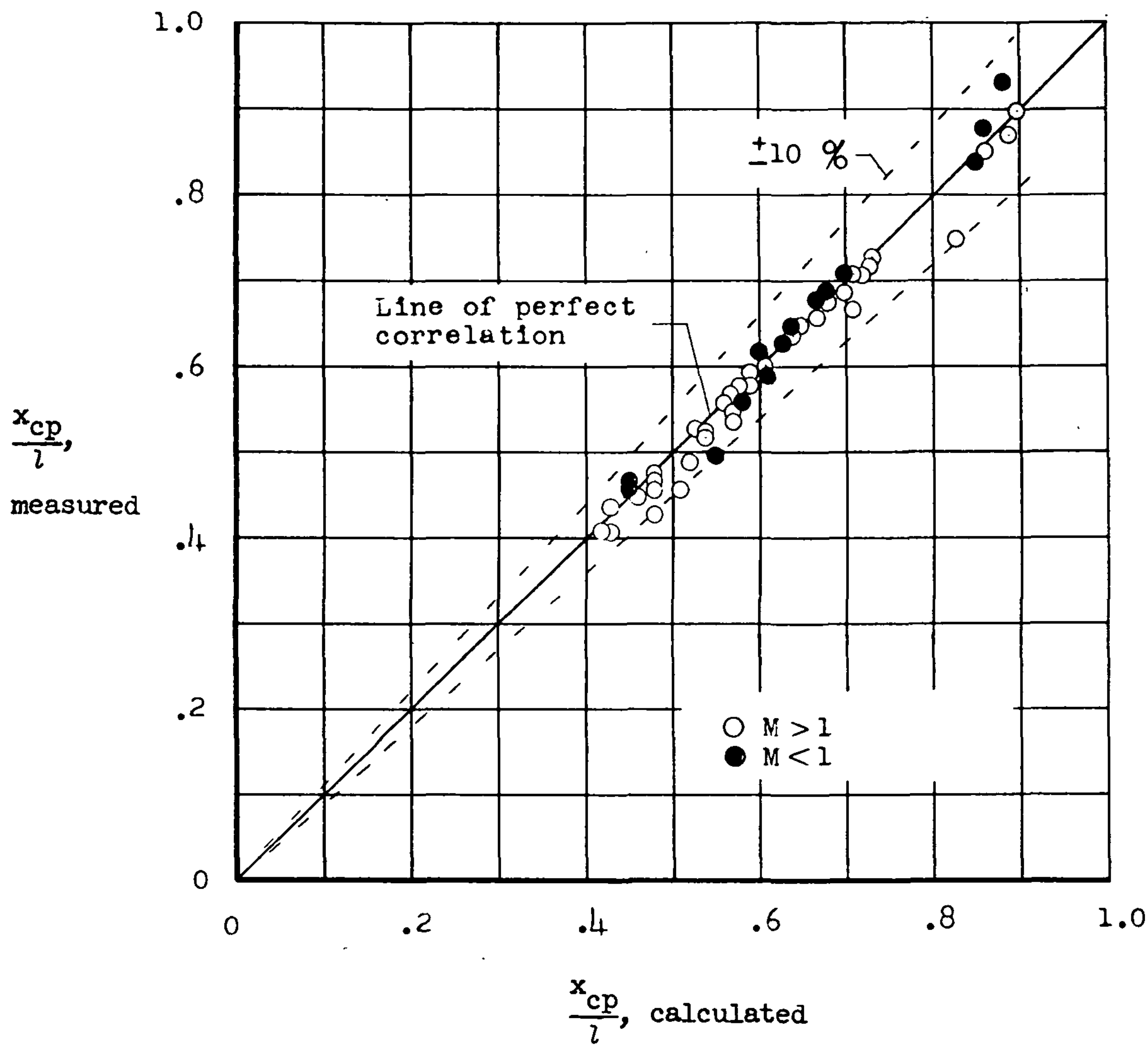


Figure 17.- Comparison between measured and calculated center of pressure for complete wing-body combination. (See refs. 4, 68, and 69.)