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by

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WIND TUNNEL TESTS AT MACH NUMBERS UP TO 2.80 TO DETERMINE THE EFFECTS OF CHANGING SPANWISE VOLUME DISTRIBUTION ON A SLENDER, CAMBERED OGEE WING

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T. A. Cook

SUMMARY

Tests have been made at M = 0.31, 1.40 (0.20) 2.40, and 2.80 to examine the effects of concentrating volume near the centreline of an integrated, slender configuration. The results show that the effects of changing spanwise volume distribution are generally small, though skin friction drag is affected by changes in wetted surface area.

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CONTENTS

ILLUSTRATIONS

SYMBOLS

M Mach number

β $\sqrt{M^2 - 1}$

q free-stream dynamic pressure

S wing planform area

S_c area of cross-section

A wing aspect ratio

$\bar{\bar{c}}$ aerodynamic mean chord

c_o model centreline chord

s local semi-span

s_T semi-span at trailing edge

p planform parameter $= S/(2\ c_o\ s_T)$

x,y longitudinal and spanwise coordinates with origin at model nose

α angle of incidence with respect to theoretical zero lift attitude

C_L lift coefficient = lift force/qS

C_m pitching moment coefficient = pitching moment/$qS\bar{\bar{c}}$

C_D drag coefficient = drag force/qS

V volume of wing

1 INTRODUCTION

The test described in this paper was one of a series made in the 3 ft × 3 ft and 8 ft × 8 ft wind tunnels at the Royal Aircraft Establishment, Bedford, in connection with the proposed M = 2.2 supersonic transport. The full programme was outlined in an unpublished memorandum by Evans and Squire, the model numbering system of which is retained here. Part of the programme conducted in the 8 ft × 8 ft wind tunnel has been reported by Taylor[1], who investigated the effects of various camber designs on an integrated configuration of 'ogee' planform. While Taylor's models represented feasible aircraft shapes, insofar as passengers, fuel, systems, etc, could be accommodated in a full-scale version, it was further proposed to investigate the effects of greater concentration of volume towards the centreline of one particular model (viz. Wing 16 of the series). The new configuration (called Wing 19) thus achieves greater distinction between wing and fuselage, which is structurally more desirable and according to slender body theory[2] has no significant drag penalty. The present test was made as a check on the drag and longitudinal characteristics of Wing 19 over a wide Mach number range.

2 DESCRIPTION OF THE MODEL

The model shape is shown in Fig.1 and principal dimensions, etc, are listed in Table 1. Wing 19 was manufactured from glass cloth and araldite, (Wing 16 was all-steel), and sting-mounted through a strain-gauge balance. Forward of the trailing edge the sting was enclosed in a 2.60 inch diameter shroud forming part of the model (this is not shown in Fig.1). The shroud blended into the model design 10 inches ahead of the trailing edge.

The model planform was defined according to the equation:-

$$\frac{s(x)}{s_T} = \frac{x}{c_o}\left\{1.2 - 2.4\frac{x}{c_o} + 2.2\frac{x^2}{c_o^2} + 3\frac{x^3}{c_o^3} - 3\frac{x^4}{c_o^4}\right\} ,$$

where $s_T/c_o = 0.208$, $s(x)$ is local semi-span, s_T is trailing-edge semi-span and c_o is centreline chord length. This shape has a planform parameter p = 0.45. The camber was designed using slender body theory[3] to the following design conditions:-

$$C_{L_d} = 0 \quad \text{and} \quad C_{m_d} = 0.00853 \quad .$$

These details are identical to those of Wing 16.

The lengthwise distributions of cross-sectional area were the same for both Wings 19 and 16 (Fig.2): the spanwise distributions are compared in Fig.3. As shown in Fig.1 the change in distribution has been effected in the case of Wing 19 by redistributing volume so that each spanwise section consists of a circular body faired into the outboard section of Wing 16 in such a way as to leave the area of the section unchanged.

3 TEST DETAILS

The test was made at Mach numbers of 0.31, 1.40 (0.20) 2.40 and 2.80, and at a constant Reynolds' number of 10^7 based on model length. Normal force, pitching moment and axial force were measured over a range of incidences at zero angle of sideslip. Zero incidence has been defined as the model attitude at which there is zero lift according to slender wing theory. Angles of incidence were corrected for sting and balance deflections under load: some allowance for tunnel flow deflections was made by testing the model both right way up and inverted in the tunnel and taking the mean of the results. The model moment reference point was at $\bar{\bar{c}}/2$ and pitching moment coefficients have been based on $\bar{\bar{c}}$.

Corrections to axial force were made for the difference between the measured pressure in the sting/balance cavity and free-stream static pressure. Drag measurements have not otherwise been corrected for the presence of the sting shroud, though corrections have been applied at supersonic speeds to lift and pitching moment for the effective distortion of the model surface due to the shroud. These corrections were evaluated by the method described in the Appendix to Ref.1 and were found to be:-

$$\Delta C_L = -\frac{0.0052}{\beta} ; \qquad \Delta C_m = +\frac{0.0018}{\beta} .$$

Corrections to incidence and pitching moment were also made at supersonic speeds for distortion of Wing 19 under load (as a result of the different material used Wing 19 was more flexible than Wing 16). The corrections were assumed to be the same as those derived from the tests of Ref.1 for Wings 17 and 18, and were:-

$$\frac{\Delta\alpha}{C_L} = -1.5^\circ ; \qquad \frac{\Delta C_m}{C_L} = -0.010 .$$

Corrections were made at the subsonic Mach number for tunnel constraint and blockage effects. Constraint corrections, obtained as in Ref.1, were:-

$$\frac{\Delta\alpha}{C_L} = 0.73^\circ ; \qquad \frac{\Delta C_m}{C_L^2}\,\alpha = 0.057 , \qquad \frac{\Delta C_D}{C_L^2} = 0.010 .$$

The Mach number quoted, i.e. 0.31, is that obtained after applying blockage corrections.

Boundary layer transition was fixed near the leading edges of the model by means of bands of 60 grade carborundum particles on an araldite base. Each band was 0.5 inches wide and started at a distance 0.1 inches from, and normal to, the leading edge.

Possible errors in the results at supersonic speeds were estimated to be:-

$$C_L \; : \; \pm 0.002 \; \pm 0.006 \; C_L$$

$$C_m \; : \; \pm 0.0004 \; \pm 0.005 \; C_m$$

$$C_D \; : \; \pm 0.0004 \; \pm 0.008 \; C_L^2 \quad .$$

At M = 0.31, the errors were estimated to be:-

$$C_L \; = \; \pm 0.003 \; \pm 0.006 \; C_L$$

$$C_m \; = \; \pm 0.0006 \; \pm 0.005 \; C_m$$

$$C_D \; = \; \pm 0.0009 \; \pm 0.008 \; C_L^2 \quad .$$

4 DISCUSSION OF RESULTS

Results for Wing 19 are compared with those for Wing 16 in Figs.4 to 11. Only the effects of the differences in geometry between the two wings are of interest here since the efficiency of the camber design and other basic aerodynamic properties of Wing 16 are fully discussed in Ref.1.

It is immediately apparent that the differences between Wings 16 and 19 are small. At M = 0.31, the differences between C_L v. α and C_m v. C_L curves (Figs.4 and 5 respectively) are generally less than the possible errors in the results, though there is some indication that Wing 19 has, compared with Wing 16, (a) a slightly-reduced lift-curve slope and (b) a more rapid pitch-up above $C_L \simeq 0.1$. Fig.6 shows that Wing 19 has rather more drag due to lift than Wing 16, while the small increase in $C_{D_{min}}$ is compatible with an estimated increase in skin friction drag of 0.0004 due to additional wetted surface area.

At supersonic speeds there is a small increase in lift-curve slope compared with Wing 16 (Fig.7) the average gain being about 6%. However the only significant effect on pitching moment curves (Fig.8) is above about M = 2.20, where there are small rearward movements of both aerodynamic centre and centre of pressure positions.

Drag polars are compared in Fig.9. As is demonstrated in Fig.10 the difference in minimum drag coefficient is explicable as being the change in skin friction drag due to the greater wetted surface area of Wing 19. (Skin friction estimates have been made by deriving those for a flat plate of the same planform as the model and scaling them by the ratio of the model wetted area to that of the flat plate. This ratio was 1.10 for Wing 19 and 1.05 for Wing 16.) It follows that both wings make very nearly the same wave drag contribution to minimum drag, i.e. wave drag has been unaffected by the redistribution of volume. Fig.11 shows that Wing 19 has greater lift-dependent drag than Wing 16. This is surprising in view of the increased lift-curve slope, though the effect may be due to a loss of leading-edge suction.

5 CONCLUSIONS

A comparison has been made between results of tests on two slender, cambered, ogee wings differing only in their spanwise distributions of volume. The comparison showed the effects of greater concentration of volume near the centreline to be generally small, some increase in minimum drag being due to increased skin friction drag resulting from additional wetted area. A small increase in lift-dependent drag was measured at both subsonic and supersonic speeds.

REFERENCES

No.	Author	Title, etc
1	Taylor, C.R.	Measurements, at Mach numbers up to 2.8, of the longitudinal characteristics of one plane and three cambered slender 'ogee' wings. A.R.C. R&M 3328 December 1961.
2	Lighthill, M.J.	The wave drag at zero lift of slender delta wings and similar configurations. Journal of Fluid Mechanics, Vol.1, Part 3, p.337. September 1956.
3	Weber, J.	Design of warped slender wings with the attachment line along the leading edge. A.R.C. R&M 3406 September 1957.

TABLE 1

Principal model details

Overall model length,	c_o	=	60 inches
Trailing edge semi-span,	s_T	=	12.48 inches
Aerodynamic mean chord,	$\bar{\bar{c}}$	=	36.96 inches
Plan area,	S	=	674 inches^2
Ratio of total wetted area to 2 S		=	1.10 (including sting shroud)
Aspect ratio,	A	=	0.924
Planform parameter,	p	=	0.45
Volume,	V	=	726 inches^3 (excluding sting shroud)

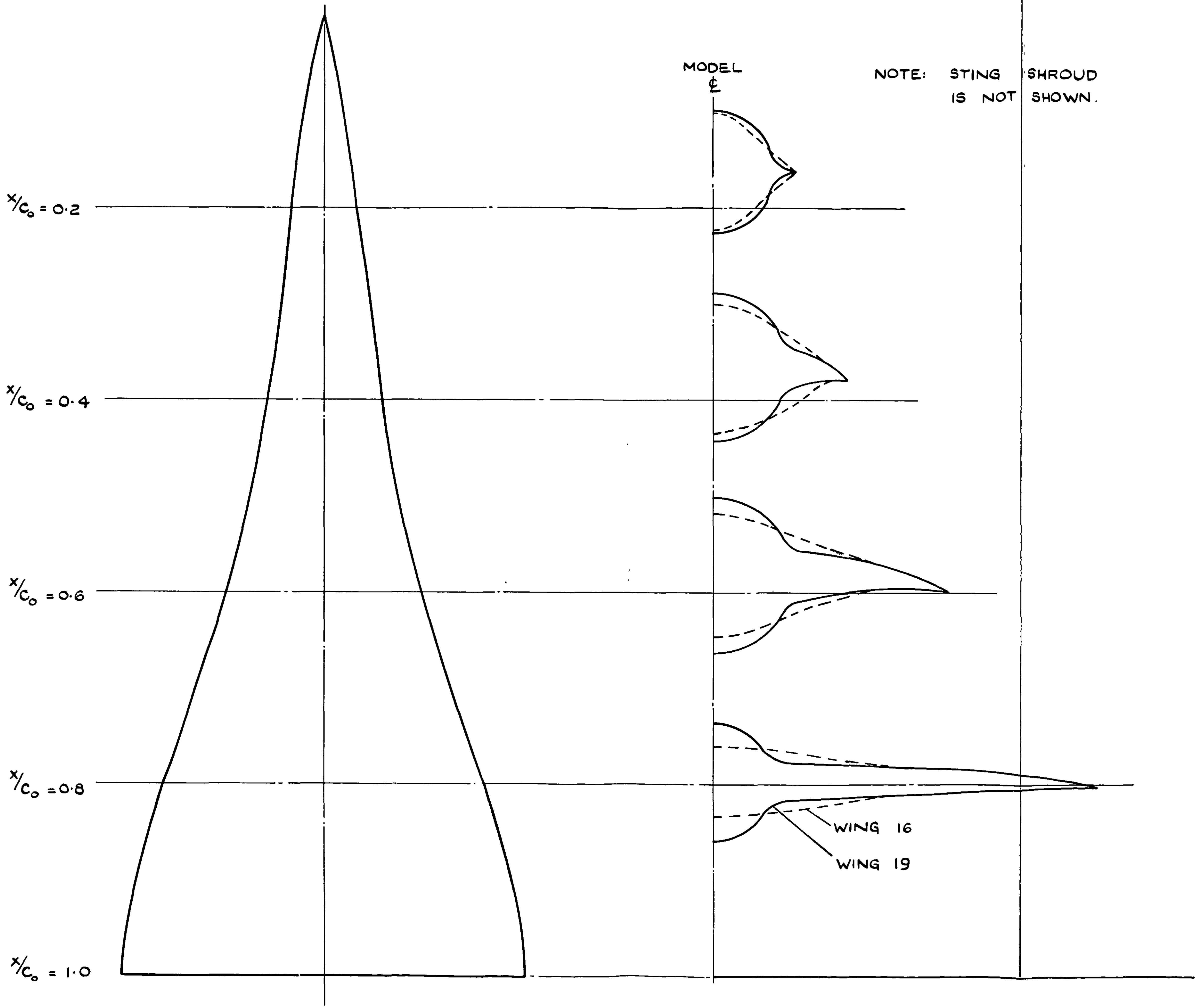

FIG. 1. MODEL DETAILS.

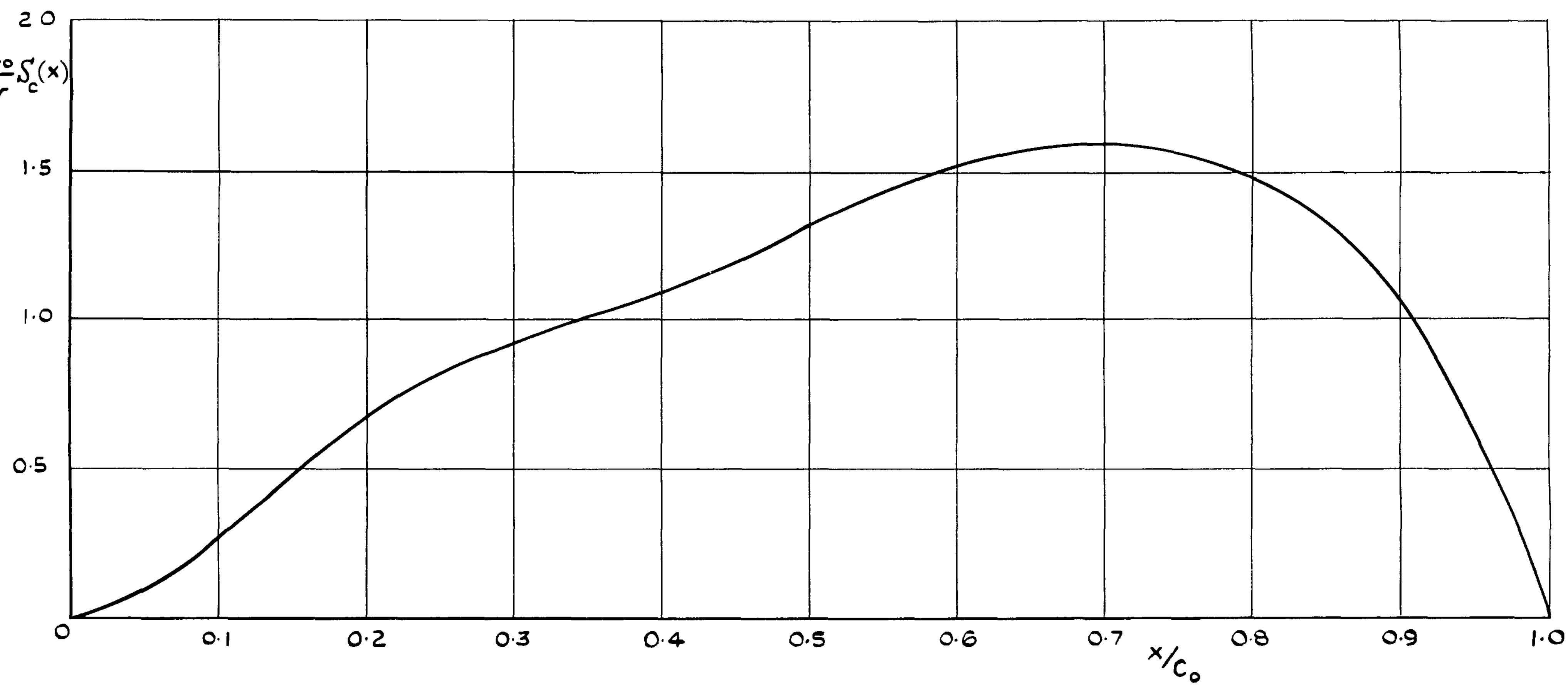

FIG. 2. CHORDWISE AREA DISTRIBUTION OF WINGS 16 AND 19.

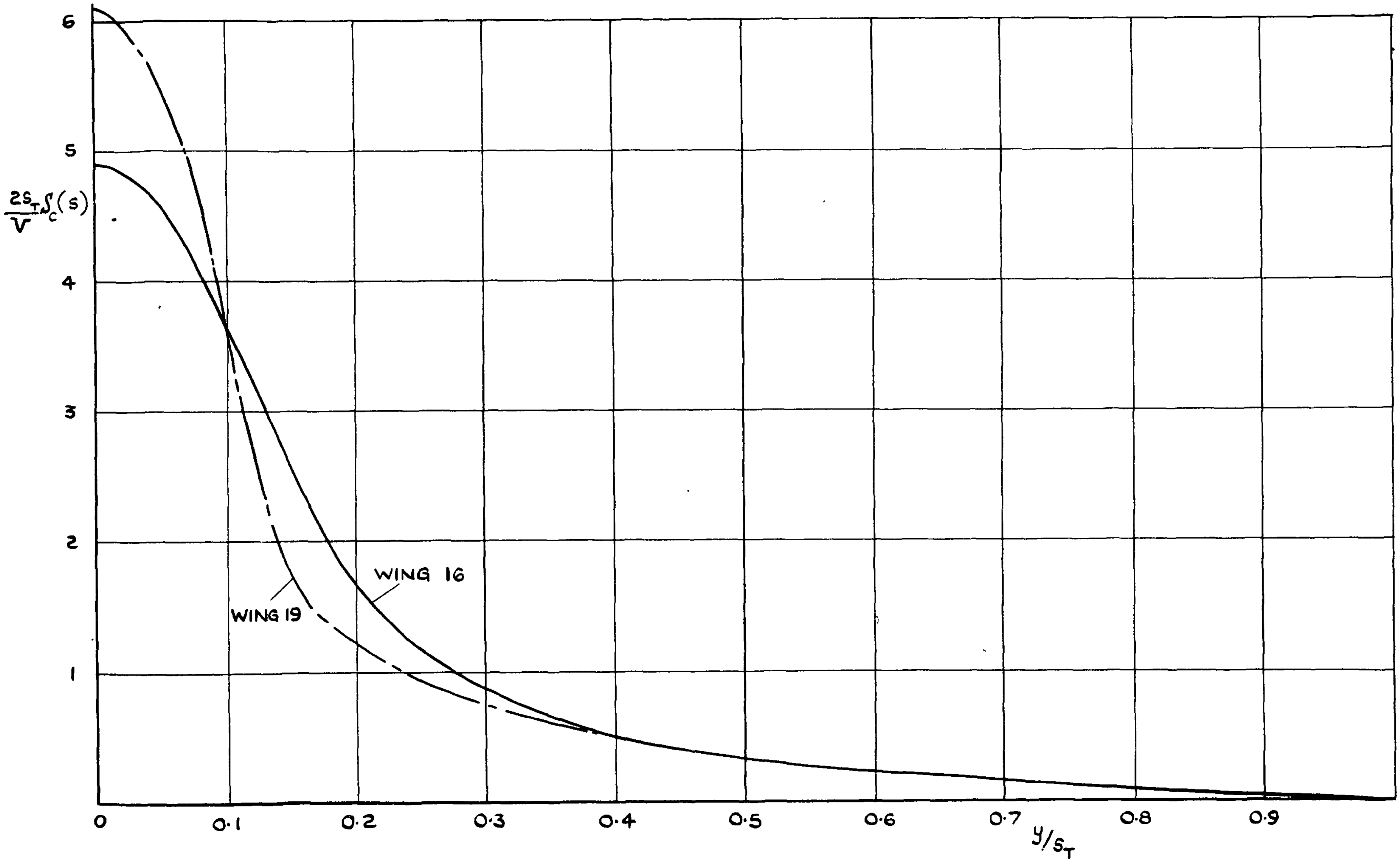

FIG. 3. SPANWISE AREA DISTRIBUTION OF WINGS 16 AND 19.

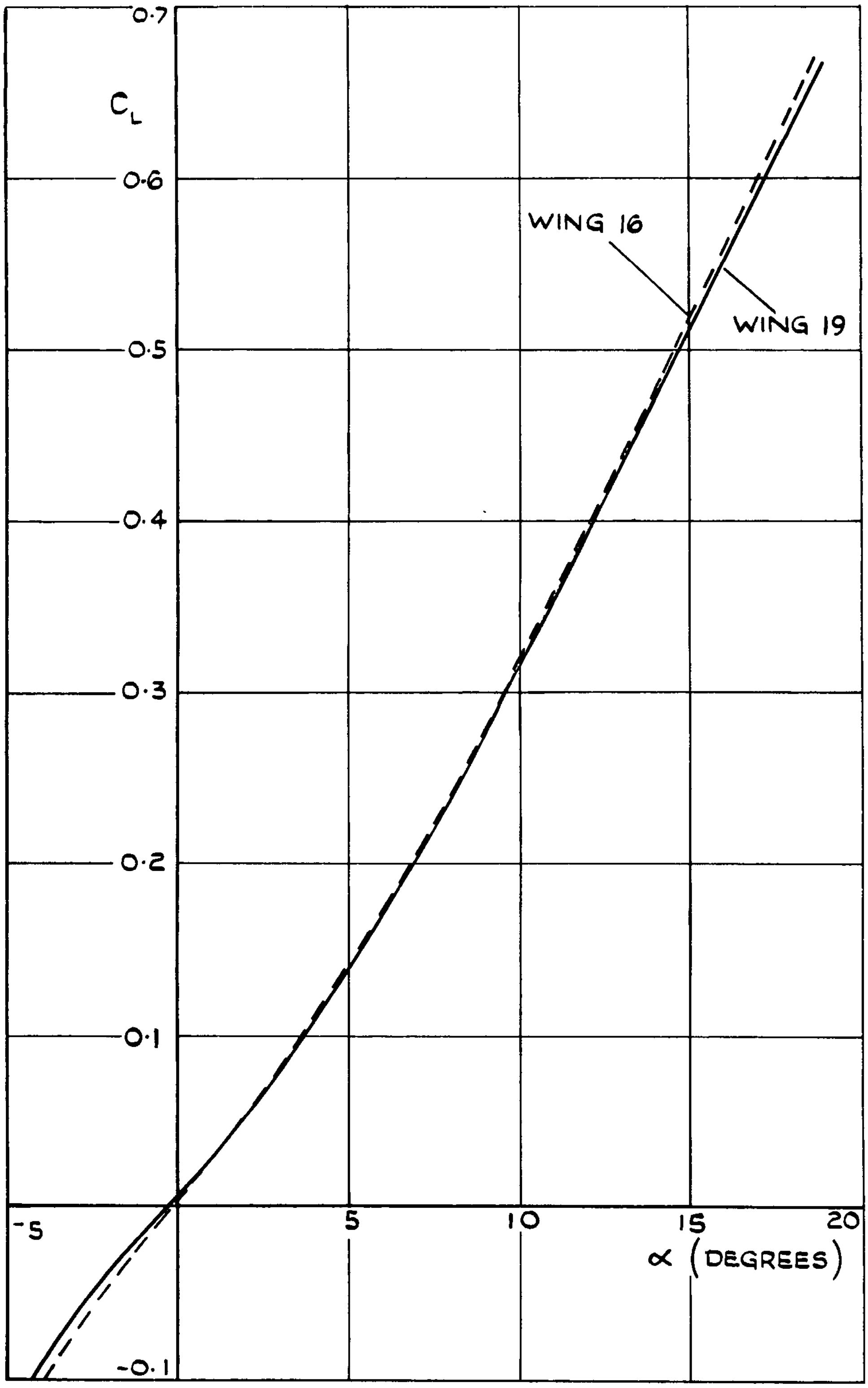

FIG 4. LIFT COEFFICIENTS AT M = 0·31

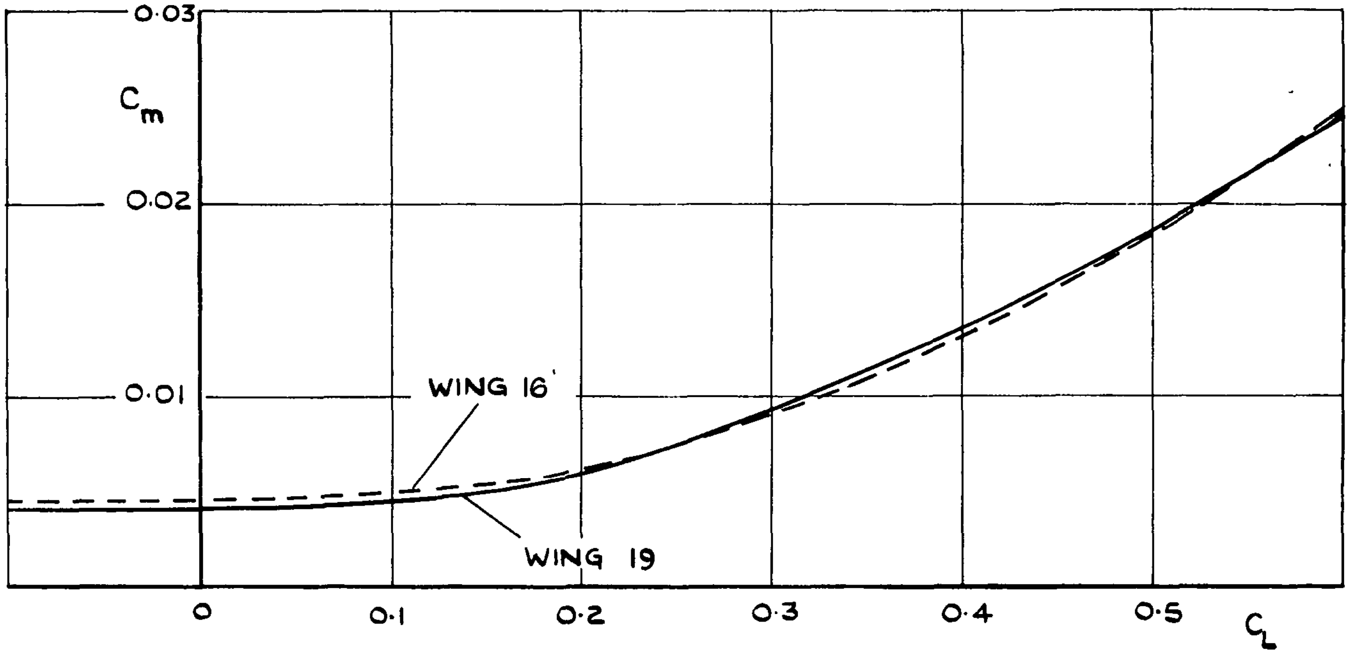

FIG. 5. PITCHING MOMENT COEFFICIENTS AT M = 0·31.

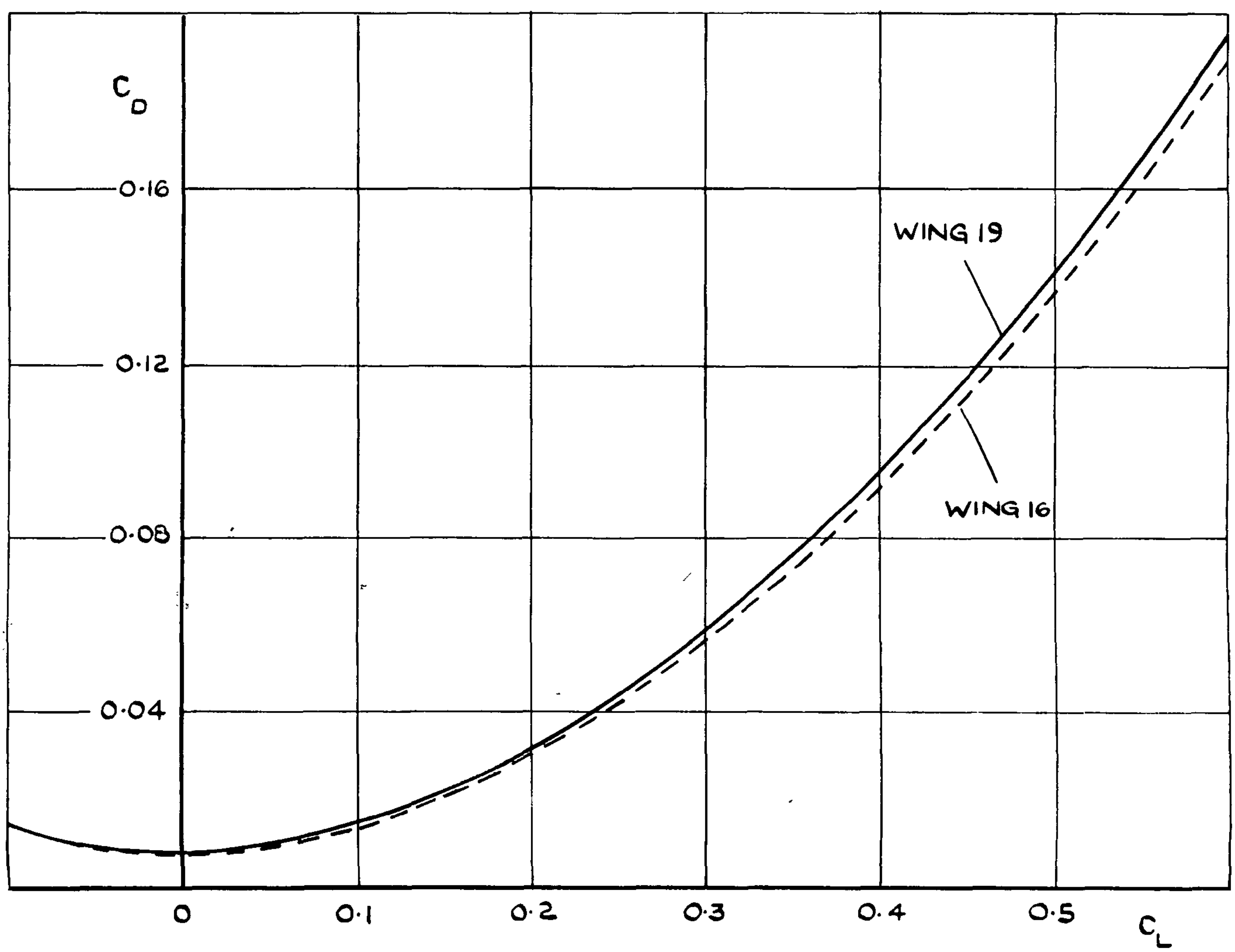

FIG. 6. DRAG COEFFICIENTS AT M = 0·31

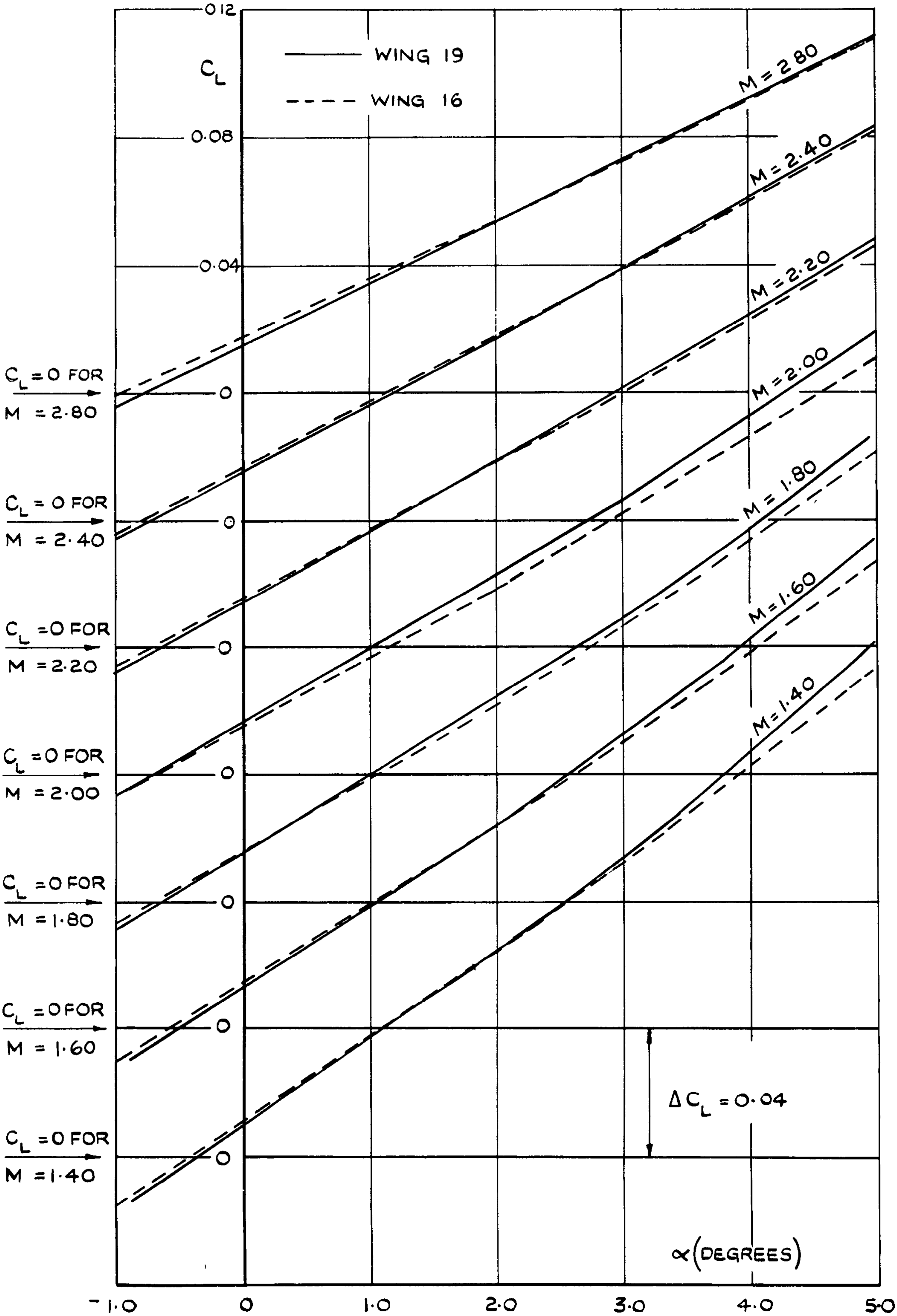

FIG. 7. LIFT CURVES AT SUPERSONIC SPEEDS

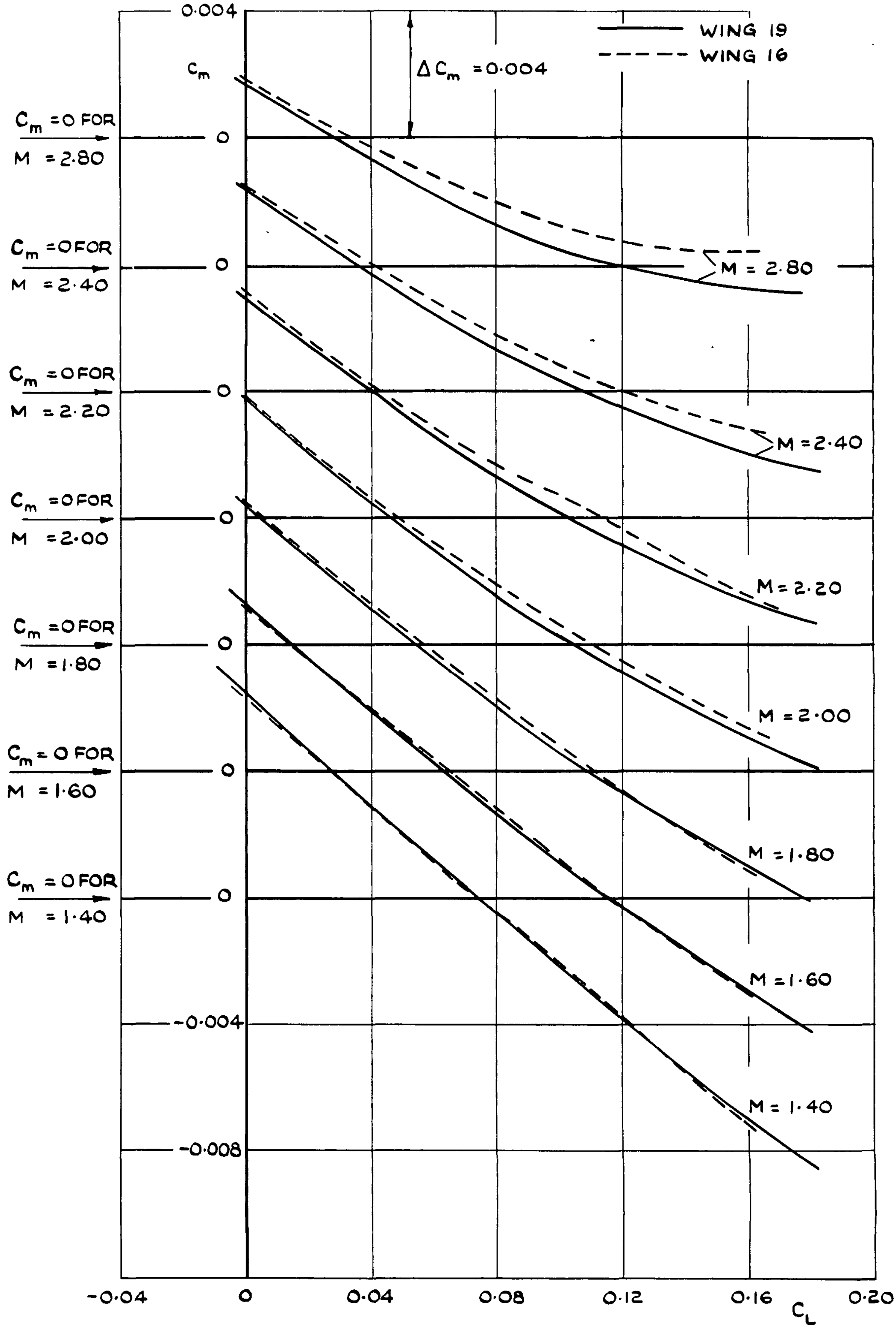

FIG 8. PITCHING MOMENT CURVES AT SUPERSONIC SPEEDS.

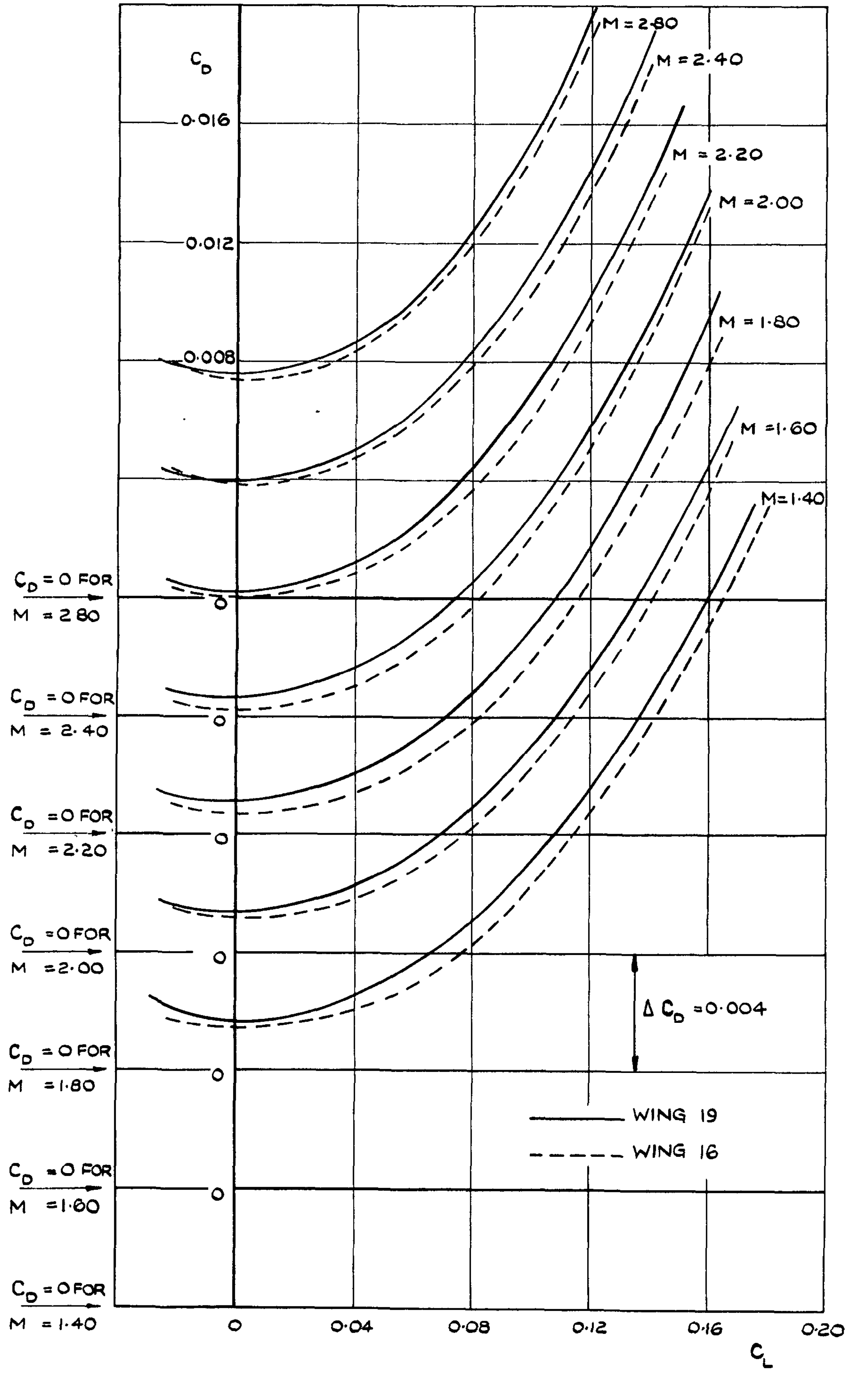

FIG. 9. DRAG POLARS AT SUPERSONIC SPEEDS.

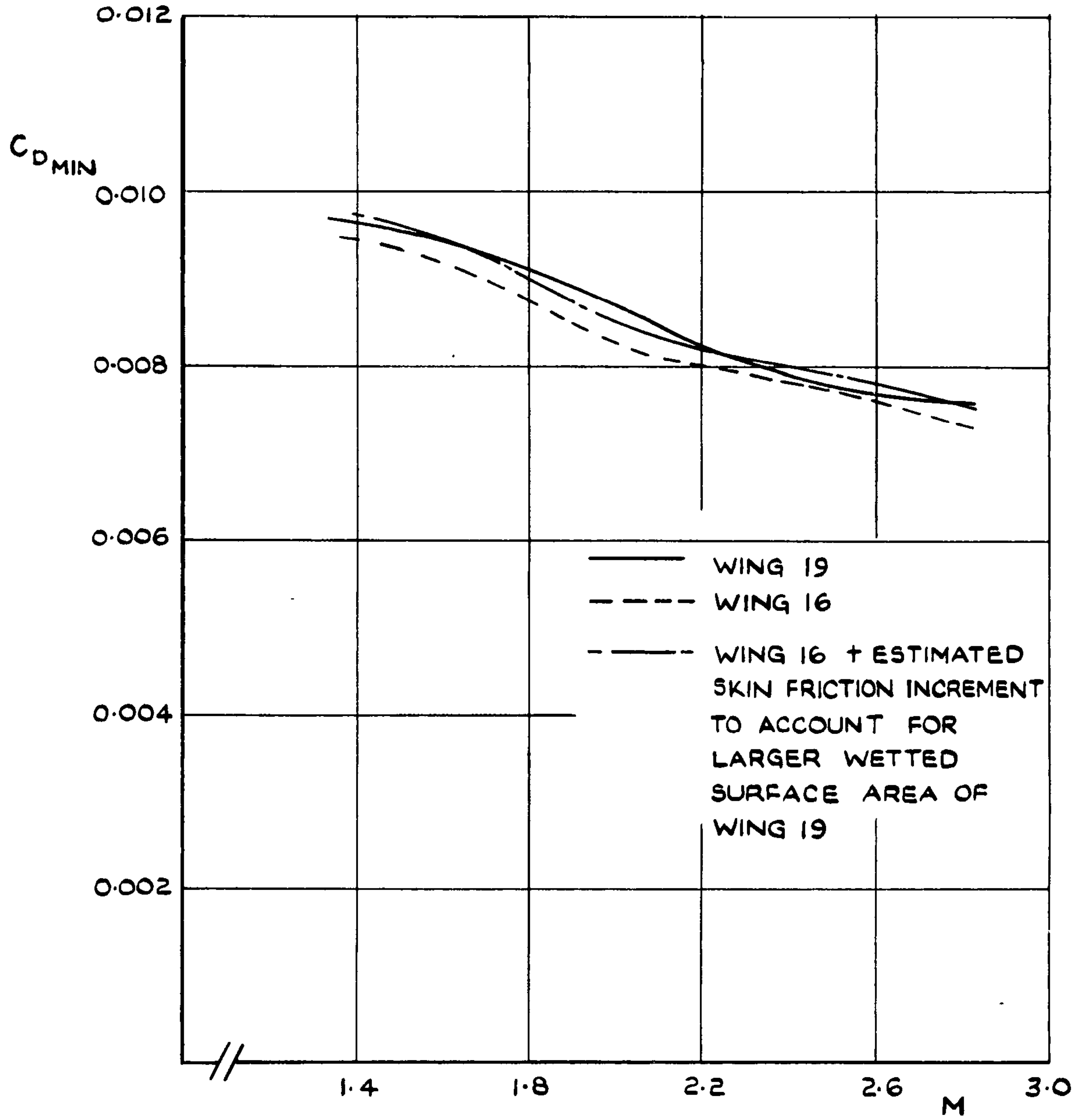

FIG. 10. MINIMUM DRAG COEFFICIENTS.

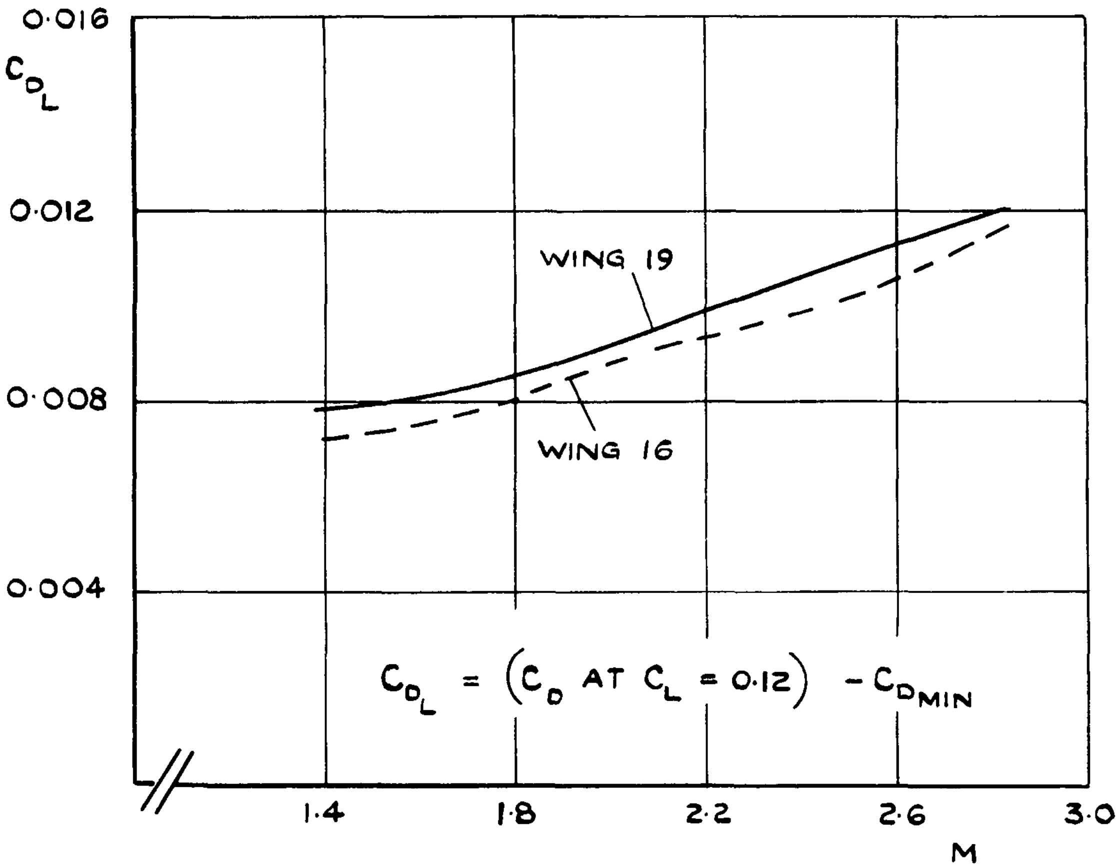

FIG.11. DRAG DUE TO LIFT AT $C_L = 0{\cdot}12$.

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