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Balance Measurements (Excluding Drag) on a Delta Wing Aircraft at Transonic Speeds

by

M. M. Shaw

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BALANCE MEASUREMENTS (EXCLUDING DRAG) ON A DELTA WING AIRCRAFT AT TRANSONIC SPEEDS

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SUMMARY

Force measurements (excluding drag) have been made on a delta wing aircraft at transonic Mach numbers. The effects of removing the nacelles and fin in turn have been measured, and the longitudinal and lateral derivatives are plotted against Mach number.

A large contribution by the nacelles to lift and pitching moment was found, and this has been successfully accounted for using existing methods.

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CONTENTS

Page

1 INTRODUCTION 3

2 RANGE OF EXPERIMENTS 3

3 ACCURACY 3

4 RESULTS AND DISCUSSION 4

4.1 Longitudinal measurements 4

4.1.1 Normal force 4

4.1.2 Pitching moment 5

4.1.3 Effect of nacelles 6

4.2 Lateral measurements 7

5 CONCLUSIONS 8

Table 1 Model details 9

Table 2 Nacelle co-ordinates 10

References 11

Illustrations Figures 1-18

Detachable abstract cards -

1 INTRODUCTION

The model described in this Report is one of those tested to give information about the behaviour of a supersonic transport aircraft designed to cruise around M = 2.

Earlier work[1] had established the suitability of the delta shape for a supersonic transport, and at the time of the present tests (June 1962) a stage had been reached when extra detail could be introduced and work done at off-design conditions.

The model tested in the 3 ft × 3 ft wind tunnel at R.A.E. (Bedford) was a 1/85 scale representation of the current shape of the aircraft. The engine nacelles and the fin were removable in order to isolate the effects of each. Model details are given in Table 1 and Figs.1-4.

2 RANGE OF EXPERIMENTS

Five-component balance measurements were taken at five transonic Mach numbers. The tests were repeated to include three model configurations. The complete aircraft was tested first, then the nacelles were removed, and finally the nacelles were replaced and the fin removed.

Measurements on each configuration included a range of incidence at zero sideslip and a range of sideslip at about $6\frac{1}{2}^\circ$ incidence.

Normal force and pitching moment are presented for the symmetrical incidence variations. Sideforce, yawing moment and rolling moment derivatives have been obtained from measurements with varying sideslip.

Forces and moments are referred to body axes, and the reference length for longitudinal and lateral coefficients is c_o, the centre line chord of the basic wing.

The Reynolds number based on c_o is $2 \cdot 1 \times 10^6$.

3 ACCURACY

The accuracies quoted below have been obtained from known instrument accuracies and from examination of the curves, with particular reference to repeat points.

Quantity	Definition	Accuracy
C_m	Pitching moment about $0{\cdot}5\ c_o/q\,S\,c_o$	$\pm 0{\cdot}0015$
C_z	Normal force/$q\,S$	$\pm 0{\cdot}004$
α	Sting incidence	$\pm 0{\cdot}05^\circ$
M	Mach number	$\pm 0{\cdot}003$
β	Angle of sideslip	$\pm 0{\cdot}05^\circ$
C_ℓ	Rolling moment about sting axis/$q\,S\,c_o$	$\pm 0{\cdot}0003$
C_n	Yawing-moment about $0{\cdot}5\ c_o/q\,S\,c_o$	$\pm 0{\cdot}0005$
C_y	Sideforce/$q\,S$	$\pm 0{\cdot}003$
ℓ_v	$\partial C_\ell/\partial\beta$	$\pm 0{\cdot}002$
n_v	$\partial C_n/\partial\beta$	$\pm 0{\cdot}005$
y_v	$\partial C_y/\partial\beta$	$\pm 0{\cdot}015$

q is the kinetic pressure, S and c_o are defined in Table 1 giving model details.

The model was tested both erect and inverted in order to discover and correct for any irregularities in the tunnel flow. The flow curvature was found to be negligible, and an upwash of 0·07 degrees has been allowed for in the plotted results.

4 RESULTS AND DISCUSSION

4.1 Longitudinal measurements

4.1.1 Normal force

Fig.5 gives curves of normal force coefficient versus sting incidence with nacelles on and off.

The basic wing planform of the model described in this Report is a delta having about $63\frac{1}{2}^\circ$ sweepback. The actual planform is modified from this by the addition of leading edge fillets and by cutting off the tips. It is useful to relate the work of these tests to that done by Squire, Jones and Stanbrook[2] on a delta wing body combination having 65° sweep, and in fact Fig.5 shows that the development of lift as incidence increases follows a similar pattern to the development shown in Fig.9 of Ref.2. A minimum lift curve slope occurs at approximately 2 degrees incidence and as incidence either increases or decreases from this value the slope increases. This effect is caused by the flow separating from the leading edge over the upper or lower surface respectively and rolling up into vortices giving rise to non-linear lift. The amount of non-linear lift first increases and subsequently decreases as Mach number is increased up to and beyond unity, as also found in Ref.2.

At all Mach numbers tested, adding the nacelles decreases the lift at constant incidence, moving the lift-versus-incidence curves bodily sideways, so that at all incidences the decrement in lift due to nacelles remains constant, although the amount of displacement varies with Mach number as can be seen in Figs.5 and 7. The explanation of the constancy of lift decrement with incidence lies in the fact that the nacelles are placed well back on the wing, so the wing acts as a turning plate to the flow, and thus the local incidence of the nacelle surfaces remains roughly constant. It is interesting to note that the widest divergence between nacelles on and nacelles off occurs at a Mach number of 0·94 where the lift at constant incidence (Fig.7) has fallen appreciably on the complete model, but has not yet done so for the model without nacelles. This type of variation will be seen again, and comes from the fact that the presence of the nacelles decreases the effective slenderness of the wing-body combination, thus leading to earlier and more exaggerated transonic force variations.

4.1.2 Pitching moment

Pitching moments with and without nacelles are plotted in Fig.8.

The camber surface of this wing was designed[4] to give a small (0·002) positive C_{m_0} at a Mach number of 2·2. Fig.9 shows that at transonic speeds the value of C_{m_0} (nacelles removed) increases to about 0·007.

The similarity of flow development with incidence as between Ref.2 and the present model is again brought out by comparing Fig.8 of this Report with Fig.10 of Ref.2. It should be mentioned that the camber of the wing described in Ref.2 was applied to reduce the subsonic and transonic lift-dependent drag, whereas the camber on the present model was applied to reduce the trim drag at cruise Mach number. Nevertheless the main features of both wings, namely the sharp edges and the sweep, are sufficiently alike, as can be seen, to make a valid comparison.

Addition of the nacelles, to a first rough approximation, increases the pitching moment by a constant amount at all normal force coefficients. This effect however is modified by a relatively large change in shape of the pitching moment curves as the transonic Mach number range is traversed. Again the earlier and more exaggerated variations with nacelles on are evidenced.

The magnitude of the pitching moment increment due to nacelles is surprisingly large. Whereas C_{m_0} due to camber is about 0·007, the maximum increment due to nacelles is 0·017, two and a half times as great.

The aerodynamic centre movement with Mach number (Fig.10) is not affected greatly by the addition of nacelles and is similar to that described in Ref.2.

4.1.3 Effect of nacelles

The discussion of the previous two paragraphs indicates that apart from special variations of pitching moment in the transonic range, the addition of nacelles changes the lift and pitching moment by amounts which are constant with incidence at a given Mach number and increase with subsonic Mach number approaching unity. If, as has been suggested, this result arises because the wings acts as a tilted flat plate, thus allowing the local incidence of the nacelles to remain unchanged and near zero, it seems likely that the effect can be accounted for by a relatively simple non-lifting hypothesis.

In order to apply this hypothesis we need to assess the effect of internal flow of the nacelles. In the present tests the air passed through a parallel passage formed by the wing lower surface and the nacelle internal surface. The assumption is made therefore that the nacelles were running full and further that ahead of the nacelles the local flow remained parallel to the wing out to the depth of the cowl lip. Then the flow velocities and directions over the nacelle external surface may be considered the same as over a bump of the same shape placed directly on the wing surface. The pressures on a bump of this type have been calculated for incompressible flow by the method of Ref.5 as if the bump were one surface of an aerofoil section having a half-thickness distribution equal to the ordinates of the bump relative to the main wing lower surface. Similar calculations have been made of the pressure distribution on the nacelle sidewalls, in order to find the magnitude of the pressures induced thereby on the wing surfaces adjoining the nacelles. The contributions to lift and pitch from this source were small compared with the contribution from the lower surfaces, and have been ignored.

The calculated pressures have been modified by the Glauert factor of $1/\sqrt{1 - M^2}$ and the resulting distribution is shown in Fig.11(a) for a Mach number of 0·6. Estimated and measured incremental lift and moments are written on the same figure. The calculated effect of Mach number on pitching moment increment is compared with experiment in Fig.9. The agreement is seen to be good.

The calculation of pressure distribution on the nacelles was extended to supersonic speeds in order to check whether the effect of the nacelles on C_{m_0} was comparable with that calculated for the wing camber. These results are presented in Fig.11(b). The method used was to find the pressure immediately behind an oblique shock of the strength required to deflect the flow through the lip angle of the nacelle and then to assume a two-dimensional Prandtl-Meyer expansion round the convex surface of the nacelle. Again it can be seen the agreement between theory and experiment[4] is good. Furthermore, the increment in C_{m_0} is now small. In practice the full mass flow condition assumed in the foregoing calculations is not necessarily typical, particularly for transonic speeds. The hypothesis is less easily applied when flow spillage is present.

Tests elsewhere[4] on this model had shown a high supersonic drag due to the nacelles. Accordingly, a nacelle was designed having a more slender external shape (Table 2). The effect of Mach number on the surface pressures of the new (Mark II) nacelle has been calculated by the methods just described. Typical results are given in Fig.12. Owing to the increased slenderness the incremental forces and moments are much smaller, and the supersonic value of C_{m_0} remains small. This last has been confirmed by unpublished results[6] from tests in the 8 ft × 8 ft tunnel at Bedford.

4.2 Lateral measurements

Variations of sideforce, yawing moment and rolling moment with sideslip (present model, Mark I nacelles) are plotted in Figs.13, 14 and 15. All these results were taken at an incidence of about $6\frac{1}{2}°$ corresponding to a lift coefficient of 0·2. In Figs.16-18 the derivatives are presented as functions of Mach number. Fig.16 shows that the contributions of the nacelles and fin[3] to the rate of change of sideforce with sideslip are roughly proportional to their projected areas in side elevation. Fig.17 shows that the effect of nacelles on directional stability is not very great compared with that of the fin; this is as might be surmised from the positions of the nacelles and fin relative to the moment centre taken with the values of sideforce derivative for each.

Rolling moment due to sideslip (Fig.18) is more strongly dependent on Mach number than either of the other two derivatives measured, and shows also a marked transonic irregularity with nacelles present. Removal of the nacelles causes a reduction in stability similar in magnitude and sense to that caused by removing the fin. This indicates that the pressures induced by the nacelles on the wing surface have a greater effect on rolling moment due to sideslip than the direct sideforce on the nacelles themselves.

5 CONCLUSIONS

(1) Characteristics of the model of a supersonic aircraft design have been measured in the transonic range, and derivatives have been found.

(2) The nacelles make a large contribution to C_{m_0} in the speed range tested.

(3) Successful predictions of the amount of this contribution have been made up to high subsonic speed and also at supersonic speeds.

(4) The contributions of nacelles to the lateral derivatives y_v and ℓ_v are significant.

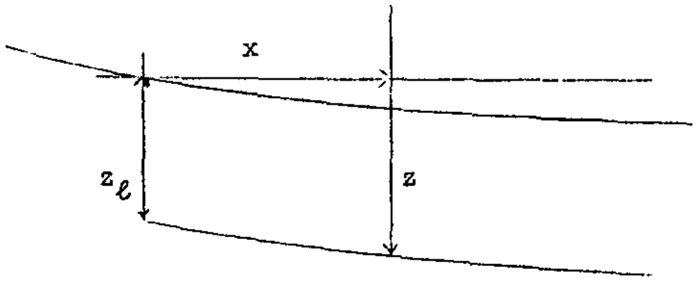
Table 1

REFERENCE DIMENSIONS AND MODEL DETAILS

Wing area S	$66 \cdot 528$ in^2
Wing span b	$9 \cdot 93$ in
Basic wing ℄ chord c_o	$11 \cdot 917$ in
Aspect ratio A	$1 \cdot 483$
Basic wing L.E. sweep	$63 \cdot 43^\circ$
Wing fillet sweep	$75 \cdot 96^\circ$
Trailing edge sweep	-5°
Inclination of wing datum to body datum	$-0 \cdot 5^\circ$
Camber design	Sud. No.2
Body length	$22 \cdot 872$ in
Max depth	$1 \cdot 464$ in
Max width	$1 \cdot 320$ in
Inclination of sting to body datum OH	0°
Fin exposed area	$7 \cdot 488$ in^2
Tip chord	$0 \cdot 98$ in
Height above body datum	$3 \cdot 348$ in
Leading edge sweep	$61 \cdot 5^\circ$
Fin arm	$5 \cdot 988$ in
Fin volume coefficient	$0 \cdot 068$ in
Nacelle length	$5 \cdot 592$ in
Nacelle width	$1 \cdot 272$
Duct cross-sectional area	$0 \cdot 461$ in^2
Base area	$0 \cdot 085$ in^2
Lip angle	14°
Sidewall sweep angle	$32 \cdot 8^\circ$
Moment reference point	$0 \cdot 5\ c_o$

Table 2

NACELLE ORDINATES RELATIVE TO THE WING LOWER SURFACE



$z_t = \dfrac{z - z_\ell}{\ell}$ $\quad x_n = x/\ell \quad$ ℓ = nacelle length

n	x_n	z_t	z_t Mk. II
16	0	0	0
15	0·0096	0·0023	0·0021
14	0·0381	0·0092	0·0061
13	0·0843	0·0178	0·0102
12	0·1464	0·0238	0·0132
11	0·2222	0·0280	0·0170
10	0·3087	0·0310	0·0215
9	0·4025	0·0326	0·0265
8	0·5000	0·0331	0·0303
7	0·5975	0·0330	0·0300
6	0·6913	0·0317	0·0289
5	0·7778	0·0265	0·0280
4	0·8536	0·0192	0·0272
3	0·9157	0·0118	0·0264
2	0·9619	0·0057	0·0258
1	0·9904	0·0014	0·0255
0	1·000	0	0·0253

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2	L. C. Squire J. G. Jones A. Stanbrook	An experimental investigation of the characteristics of some plane and cambered 65° delta wings at Mach numbers from 0·7 to 2·0. A.R.C. R & M 3305 July 1961
3	D. Isaacs	Measurements at subsonic and supersonic speeds of the longitudinal and lateral stability of a slender cambered ogee wing including the effects of a fin, canopy nose and trailing edge controls. A.R.C. R & M 3390 Sept. 1963
4	Bristol Aircraft Ltd	S.S.T. Model 122 Books 1, 4 and 5. Report W.T. 414
5	J. Weber	The calculation of the pressure distribution over the surface of two-dimensional and swept wings with symmetrical aerofoil sections. A.R.C. R & M 2918 July 1953
6	W. J. Britton	Private communication

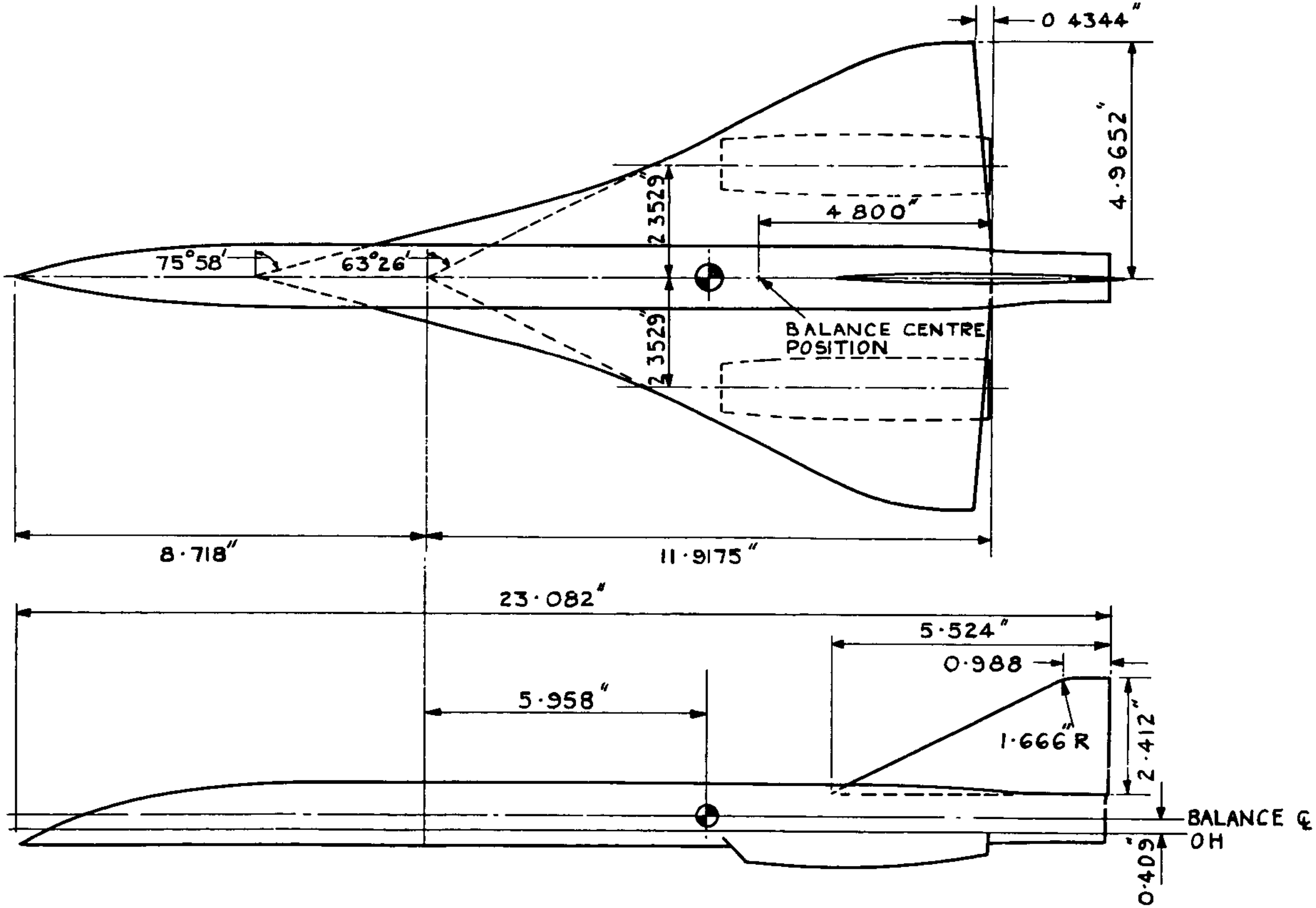

FIG.1 GENERAL ARRANGEMENT OF MODEL 1/85 FULL SCALE

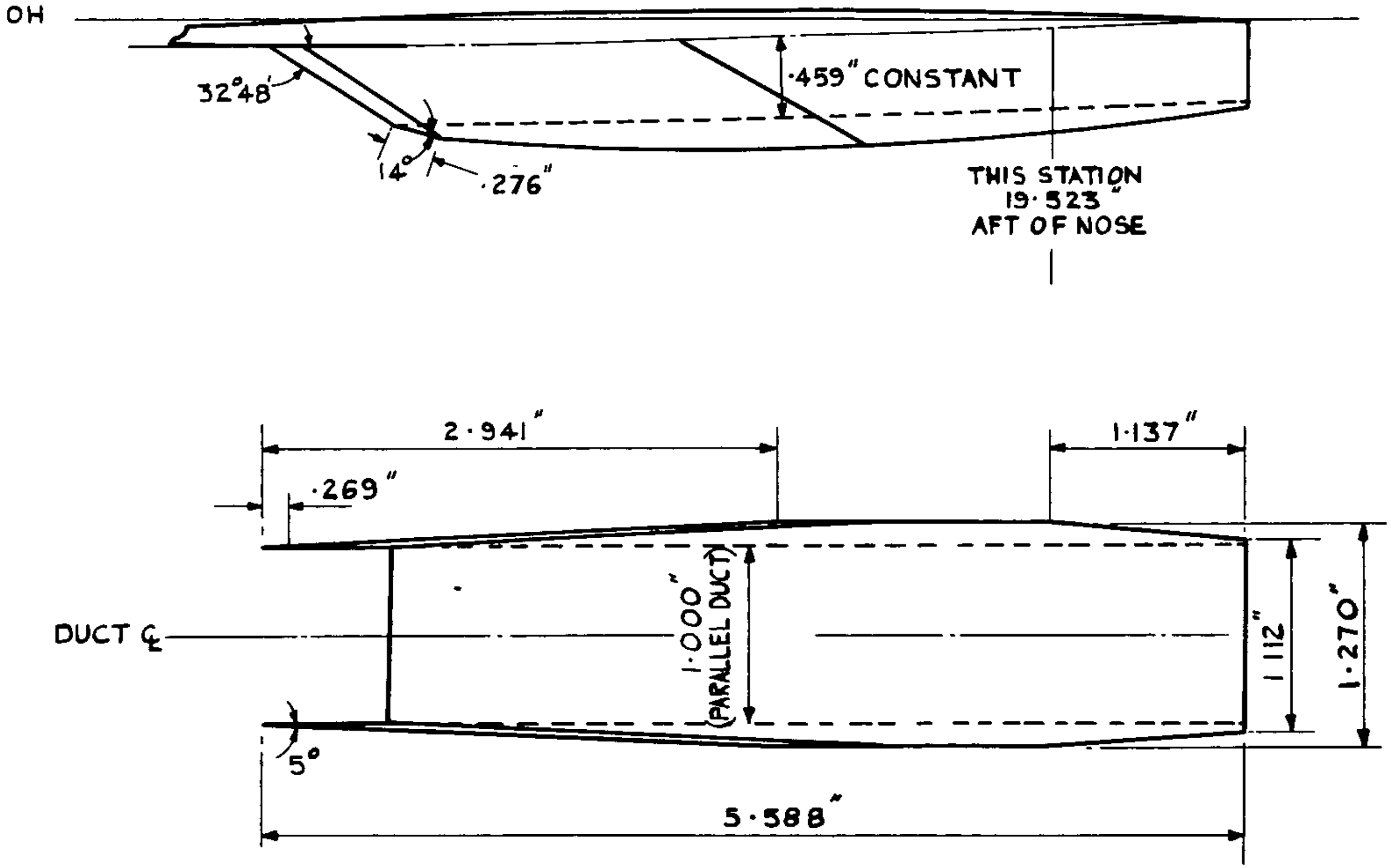

FIG. 2 NACELLE DETAILS

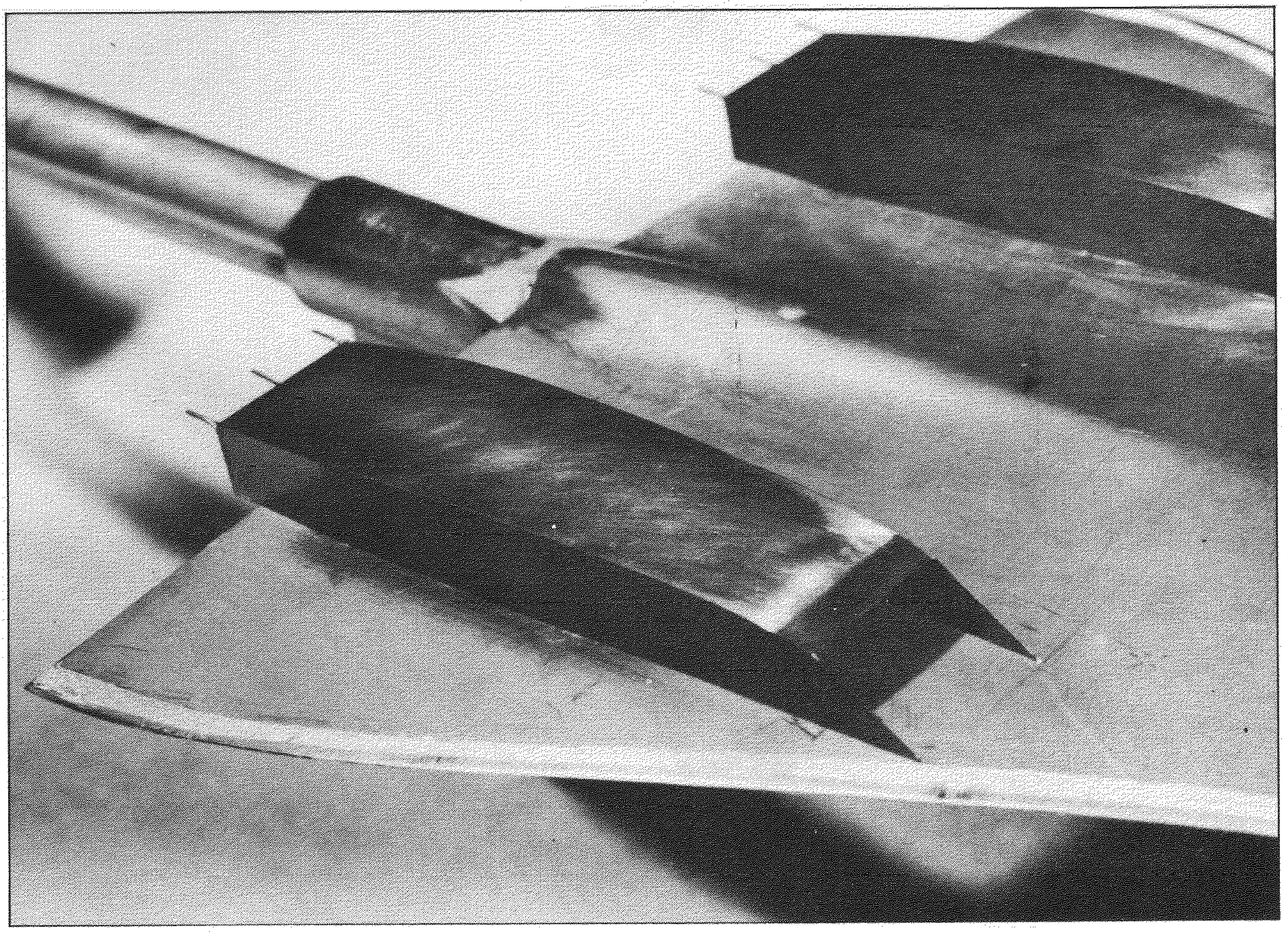

Fig.3 Detail of Nacelles

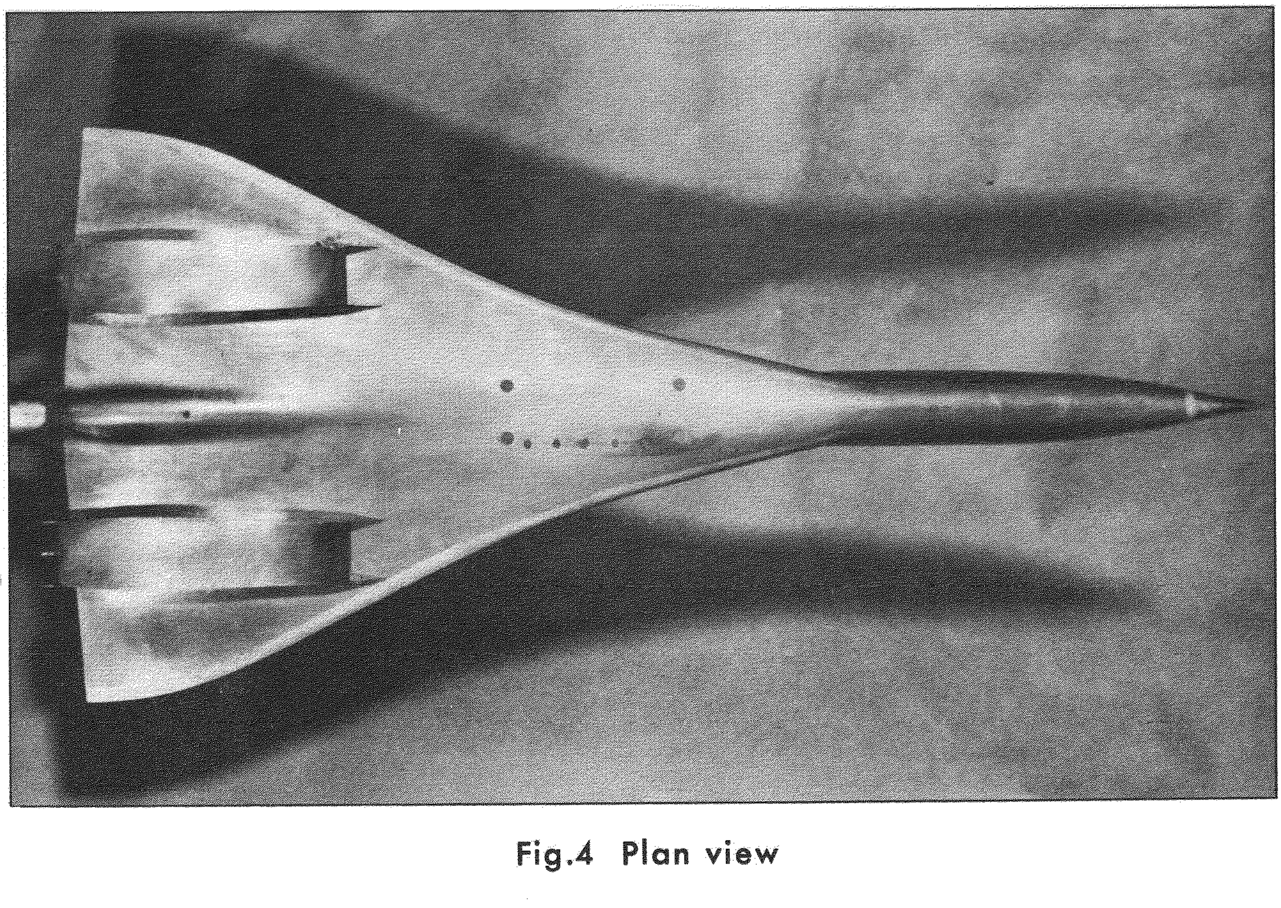

Fig.4 Plan view

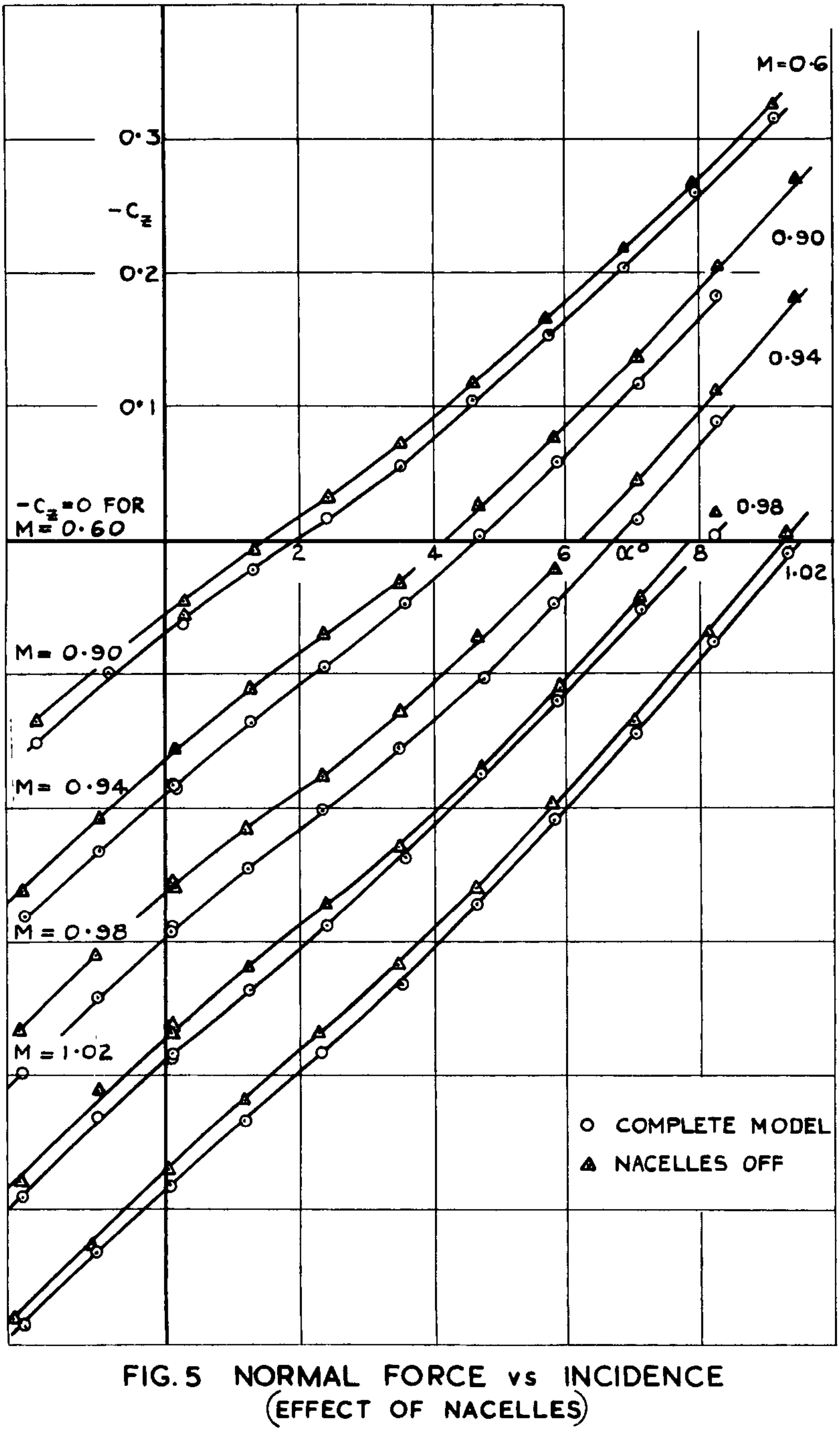

FIG. 5 NORMAL FORCE vs INCIDENCE (EFFECT OF NACELLES)

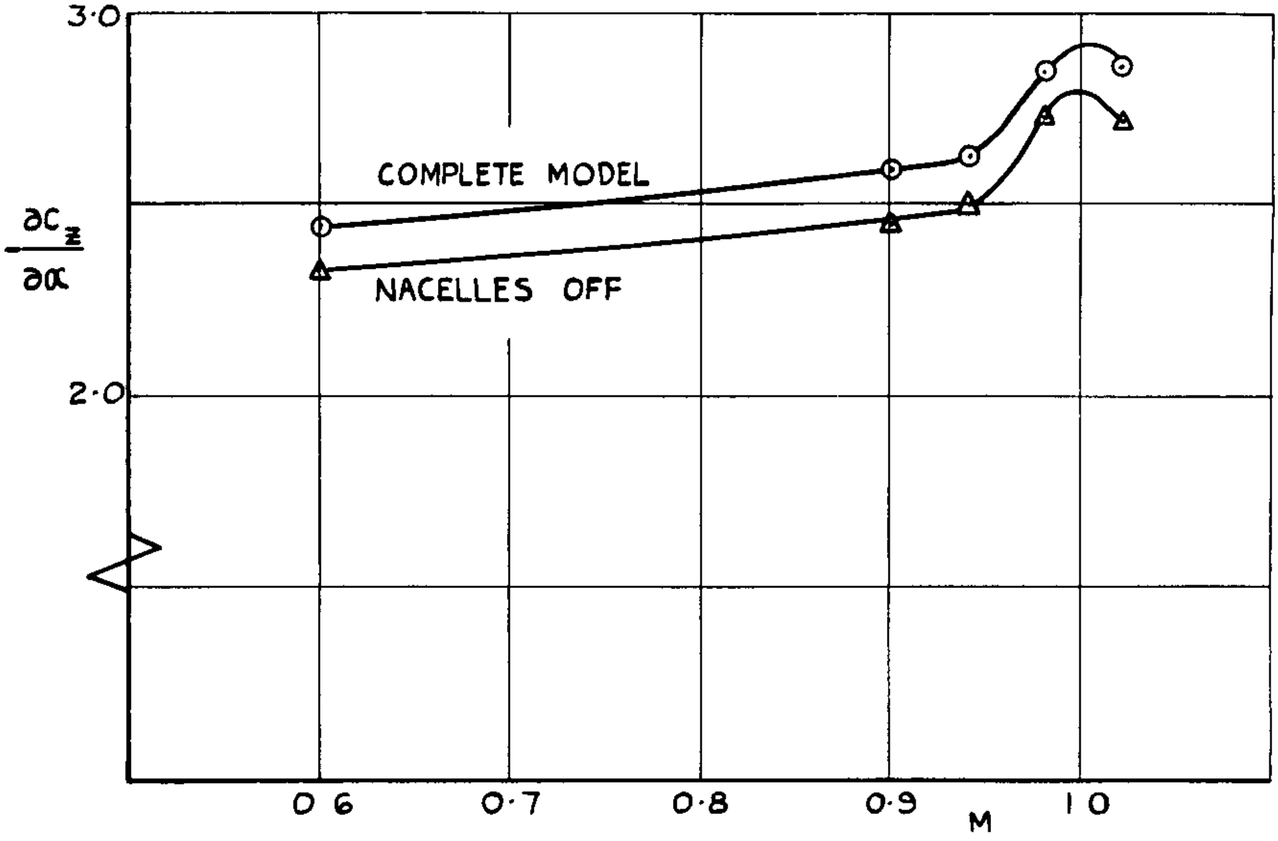

FIG. 6 LIFT CURVE SLOPE VS MACH NUMBER

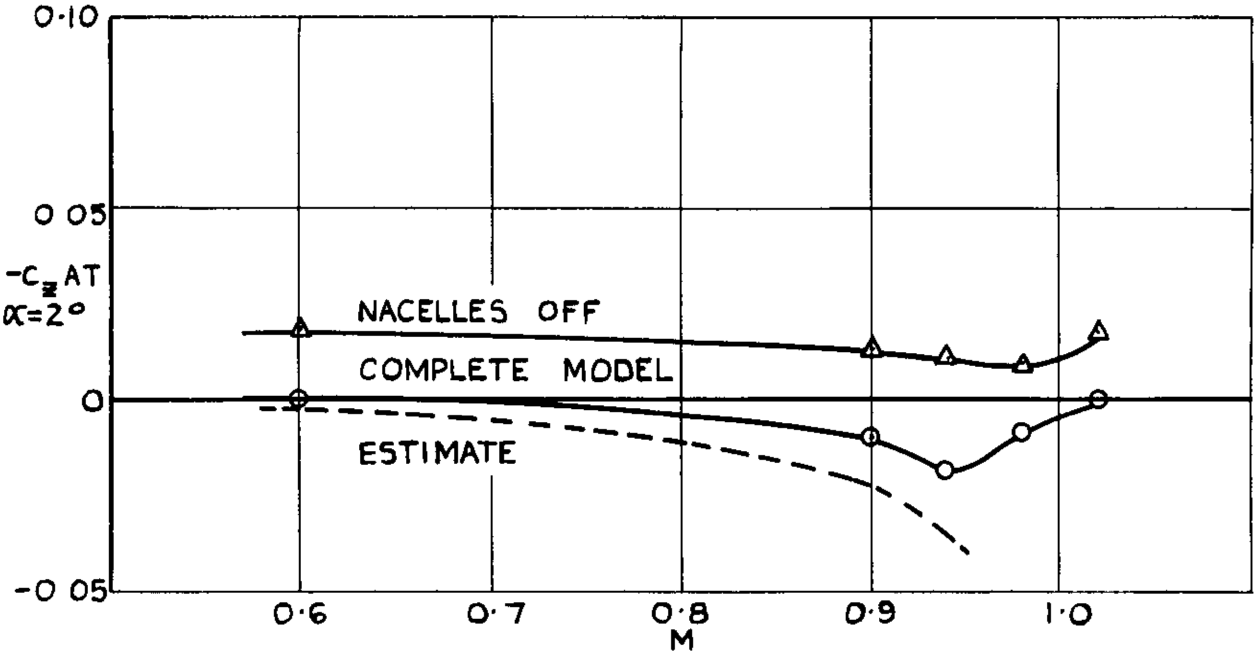

FIG. 7 NORMAL FORCE AT 2° INCIDENCE VS MACH NUMBER

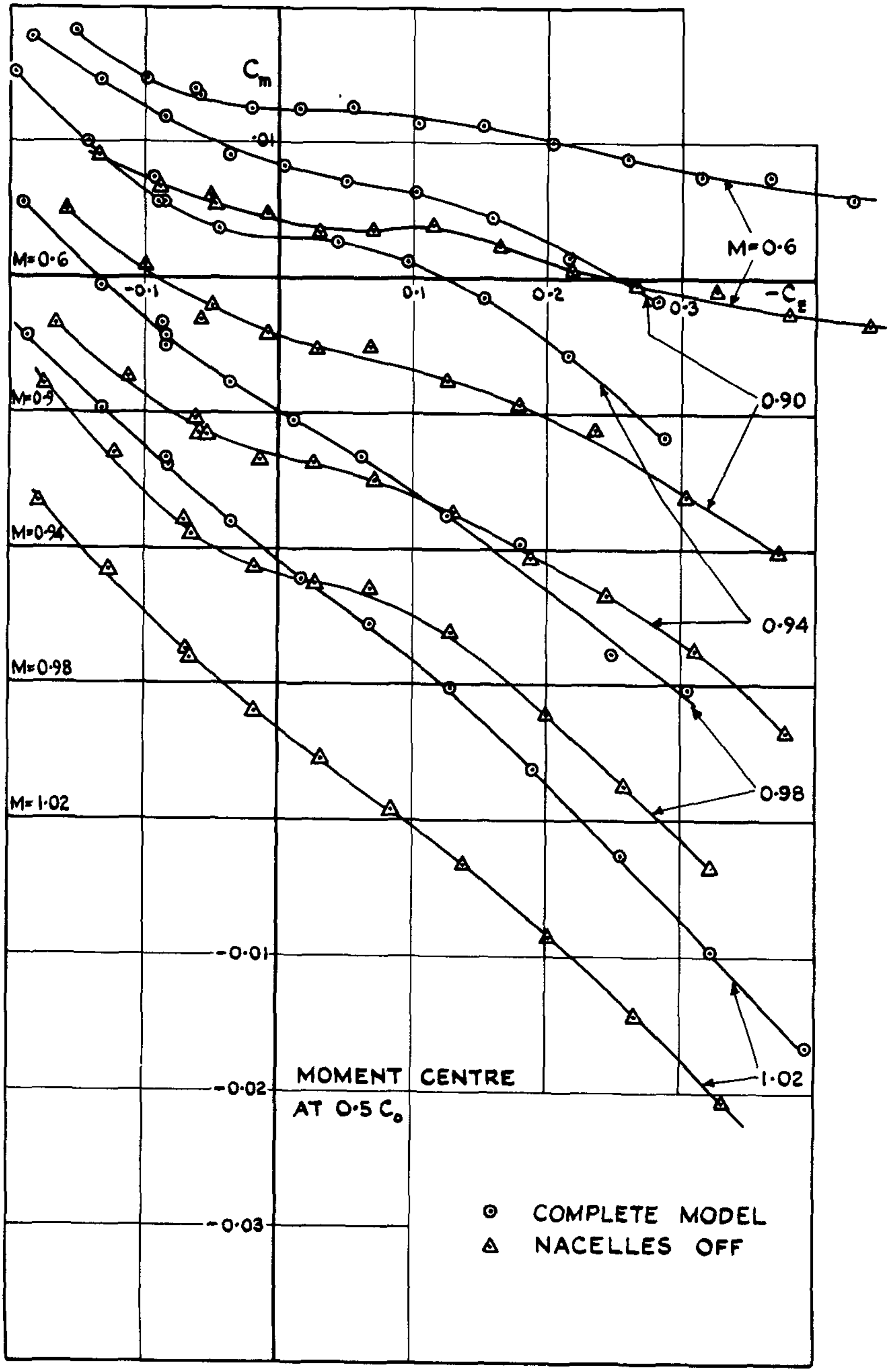

FIG. 8 PITCHING MOMENT vs NORMAL FORCE (EFFECT OF NACELLES)

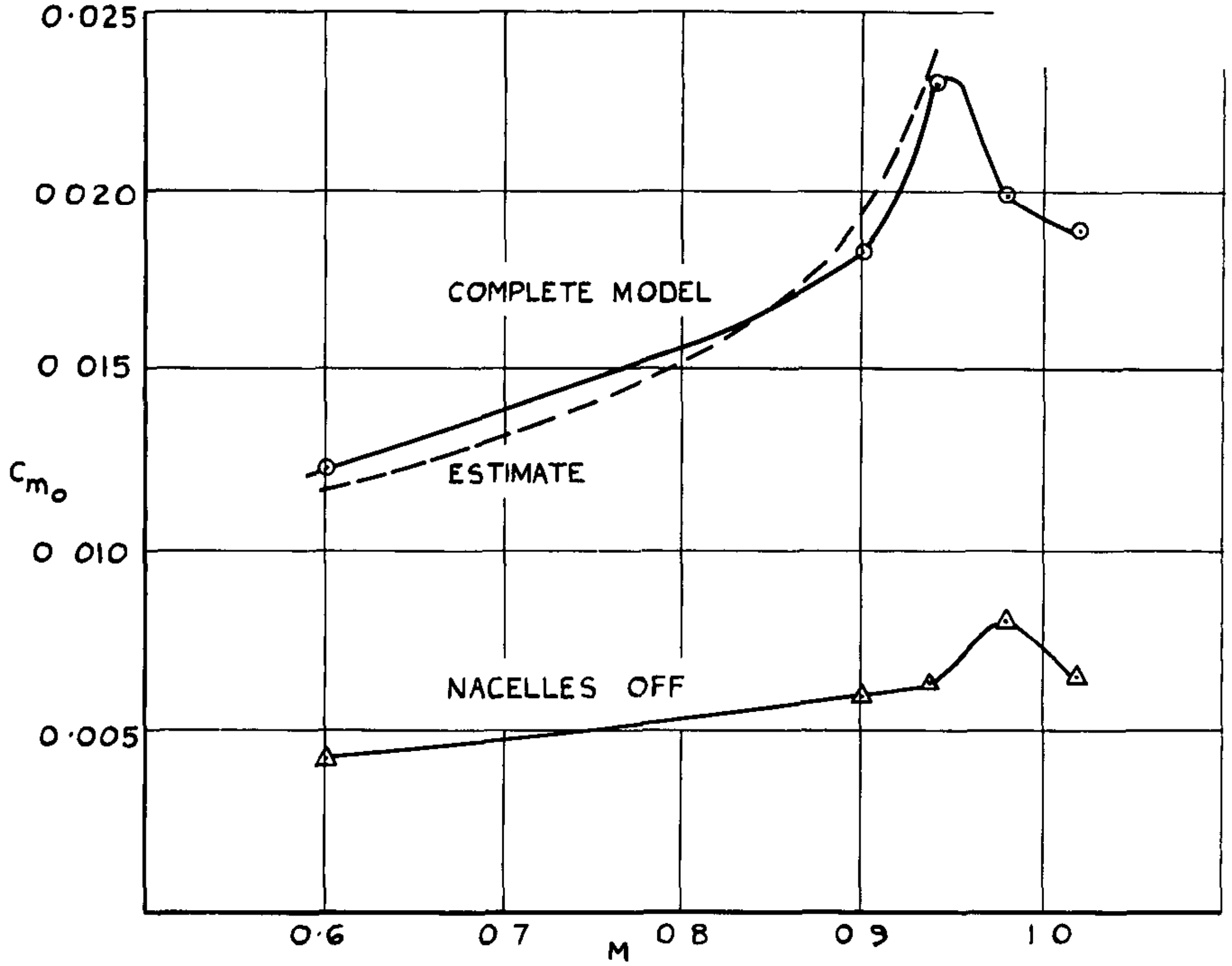

FIG. 9 PITCHING MOMENT AT ZERO LIFT vs MACH NUMBER

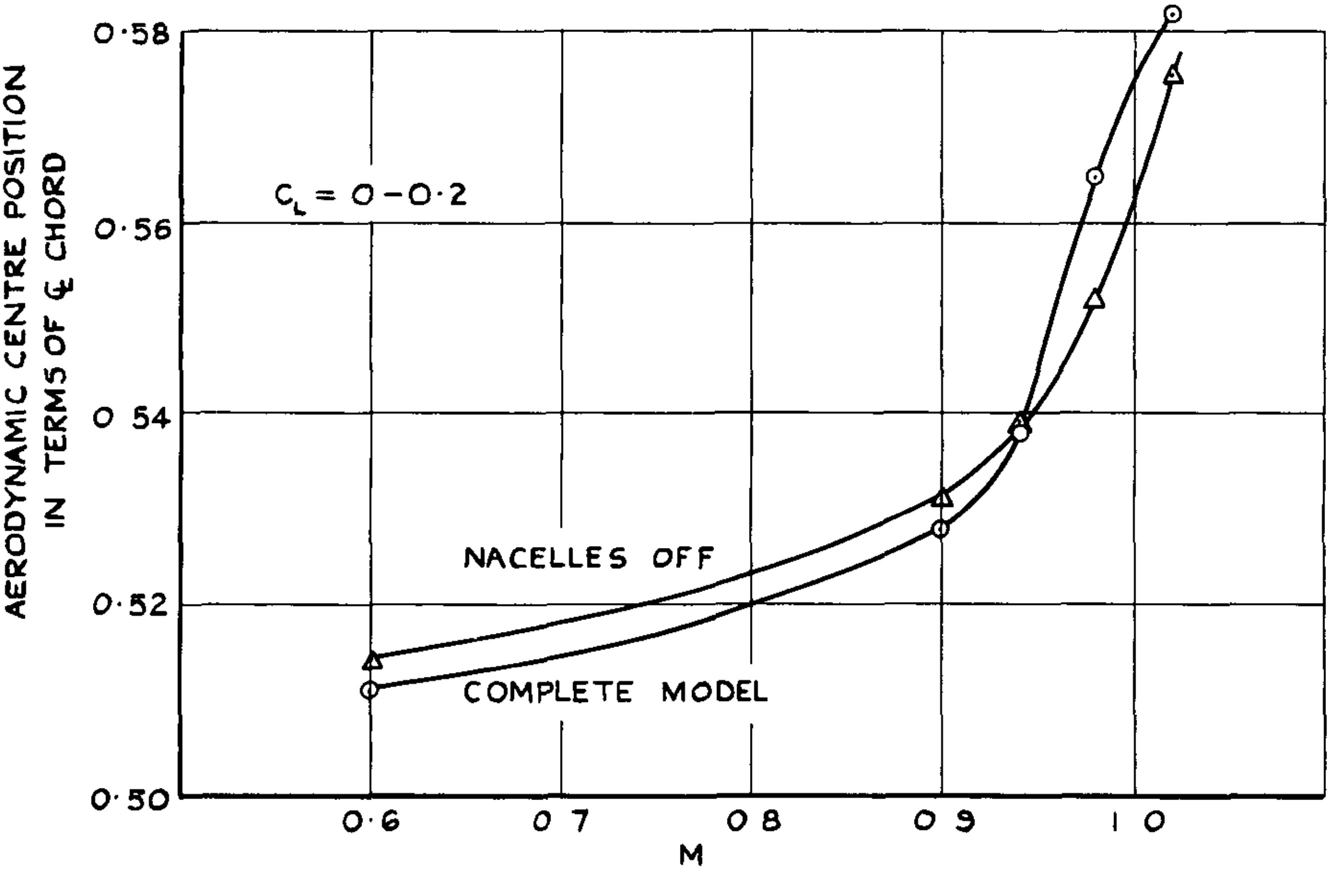

FIG.10 AERODYNAMIC CENTRE POSITION vs MACH NUMBER

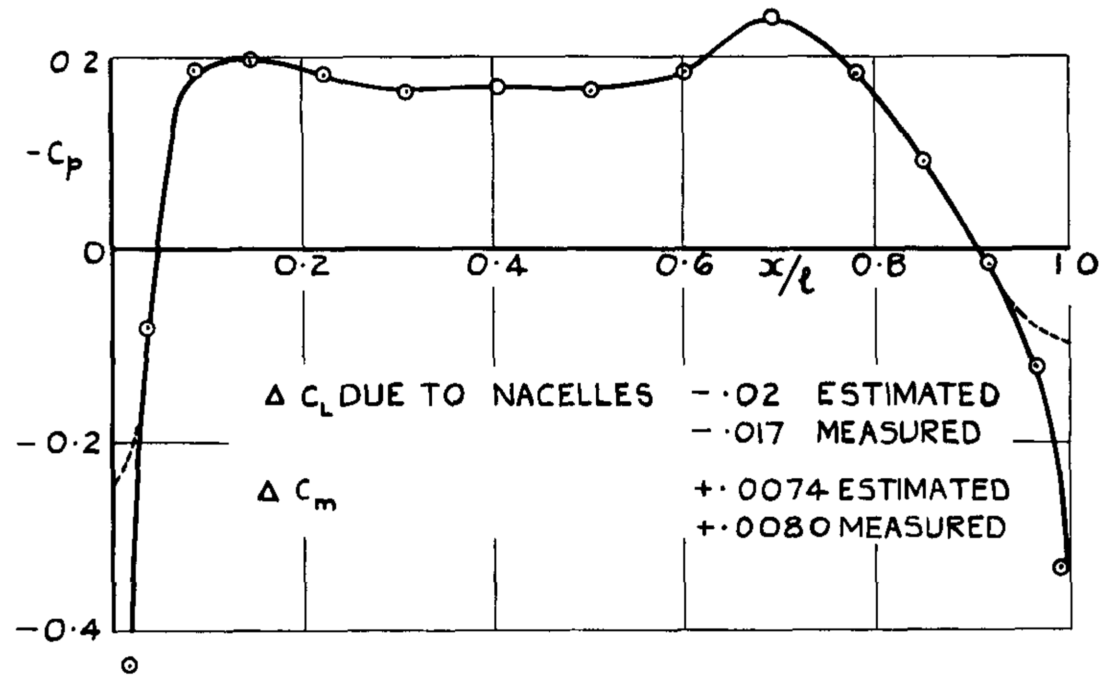

FIG. 11 (a) RRESSURE DISTRIBUTION OVER NACELLES (M = 0·6)

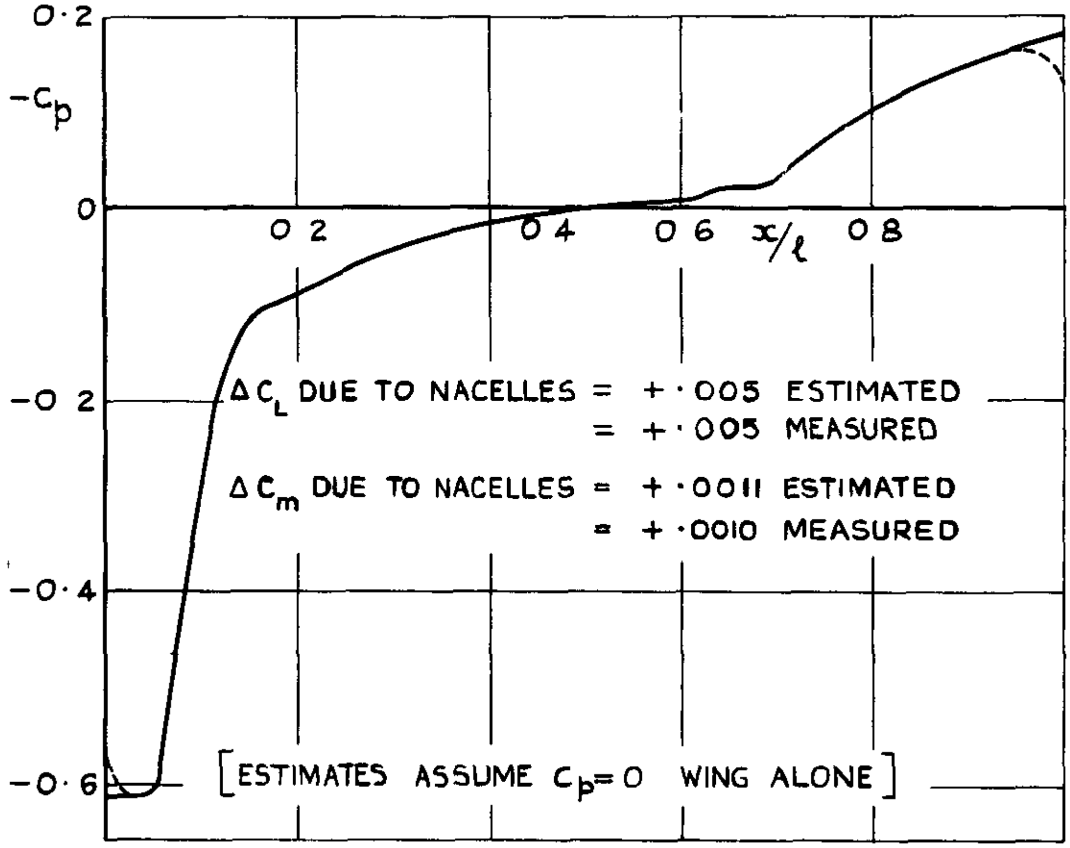

FIG. 11 (b) PRESSURE DISTRIBUTION OVER NACELLE (M = 1·6)

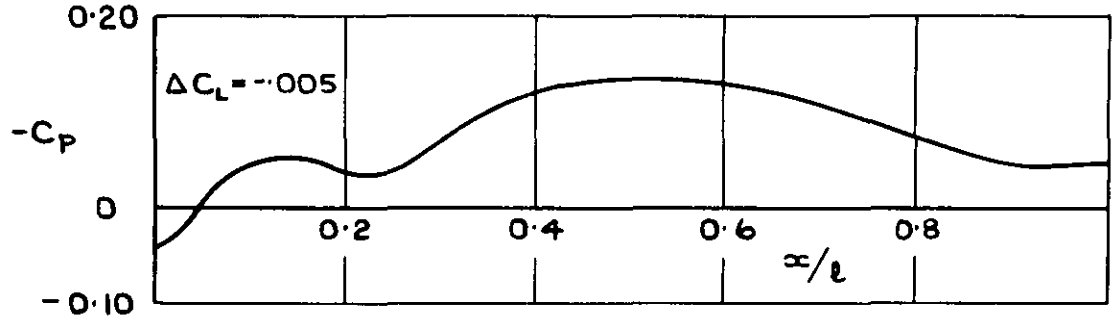

FIG. 12 (a) PRESSURE DISTRIBUTION OVER Mk II NACELLE (M=0·6)

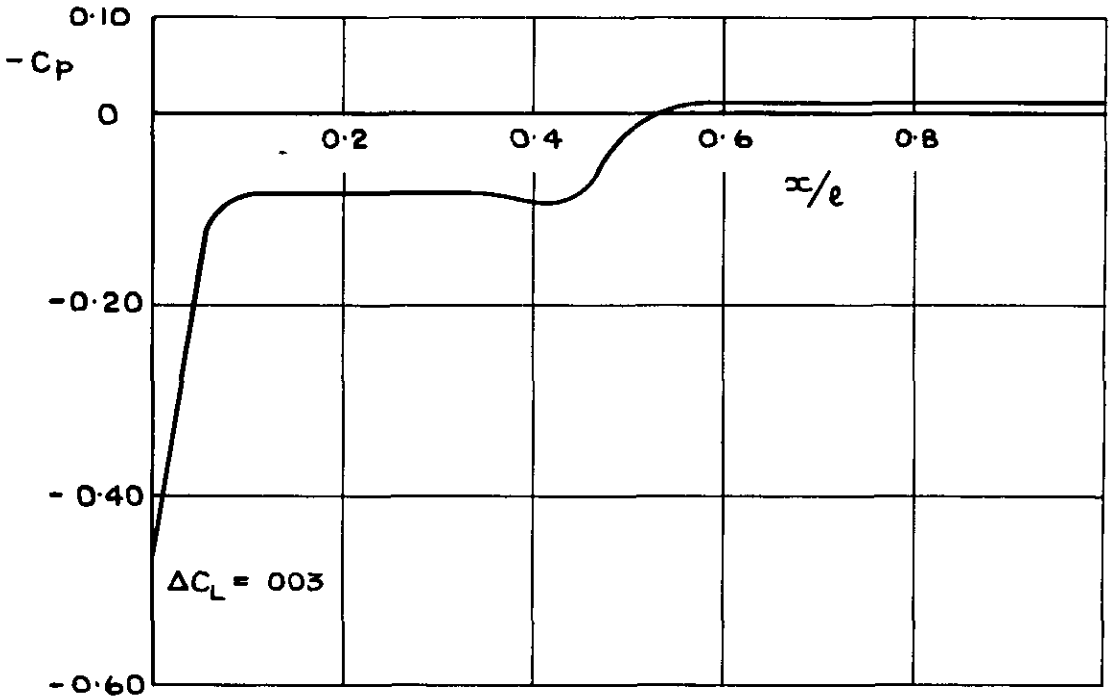

FIG. 12 (b) PRESSURE DISTRIBUTION OVER Mk II NACELLE (M=1·6)

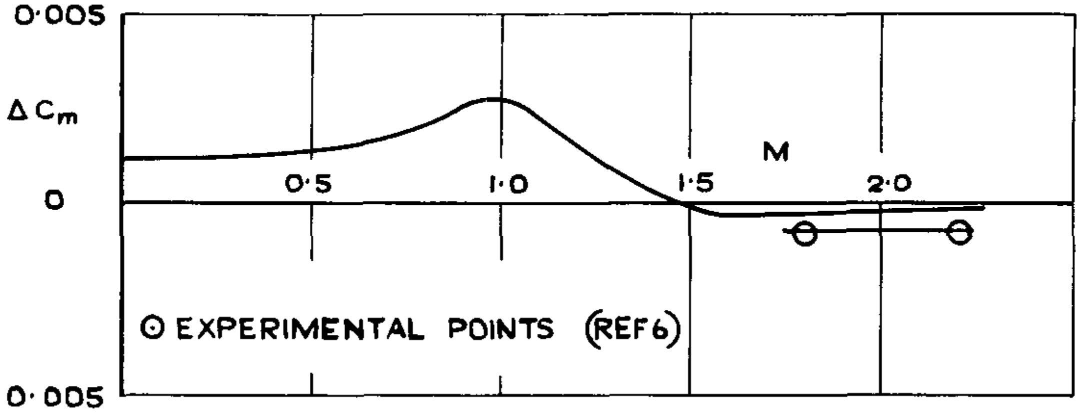

FIG. 12 (c) PITCHING MOMENT DUE TO Mk II NACELLES vs MACH NUMBER

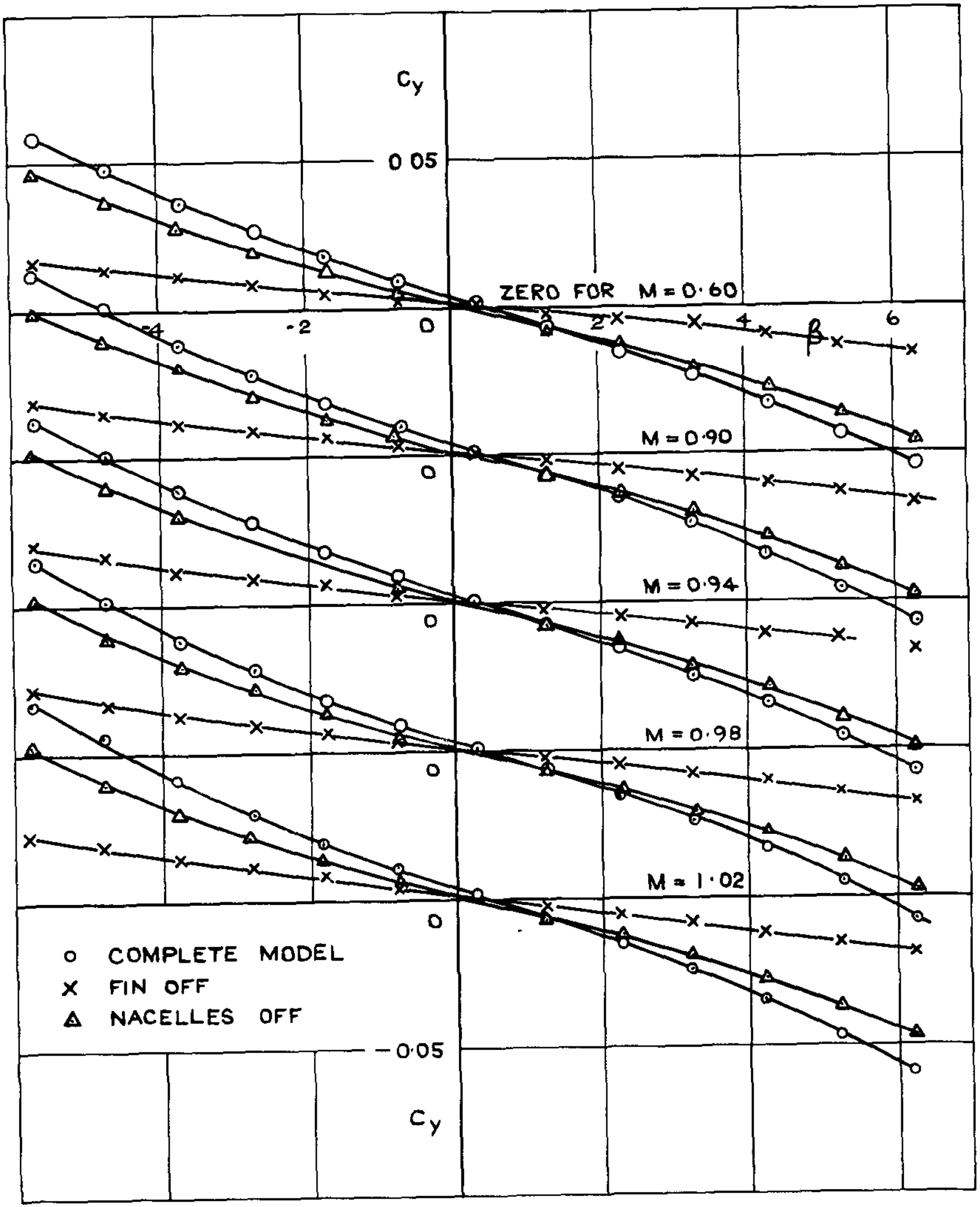

FIG. 13 SIDEFORCE vs SIDESLIP AT $6\frac{1}{2}^{\circ}$ INCIDENCE

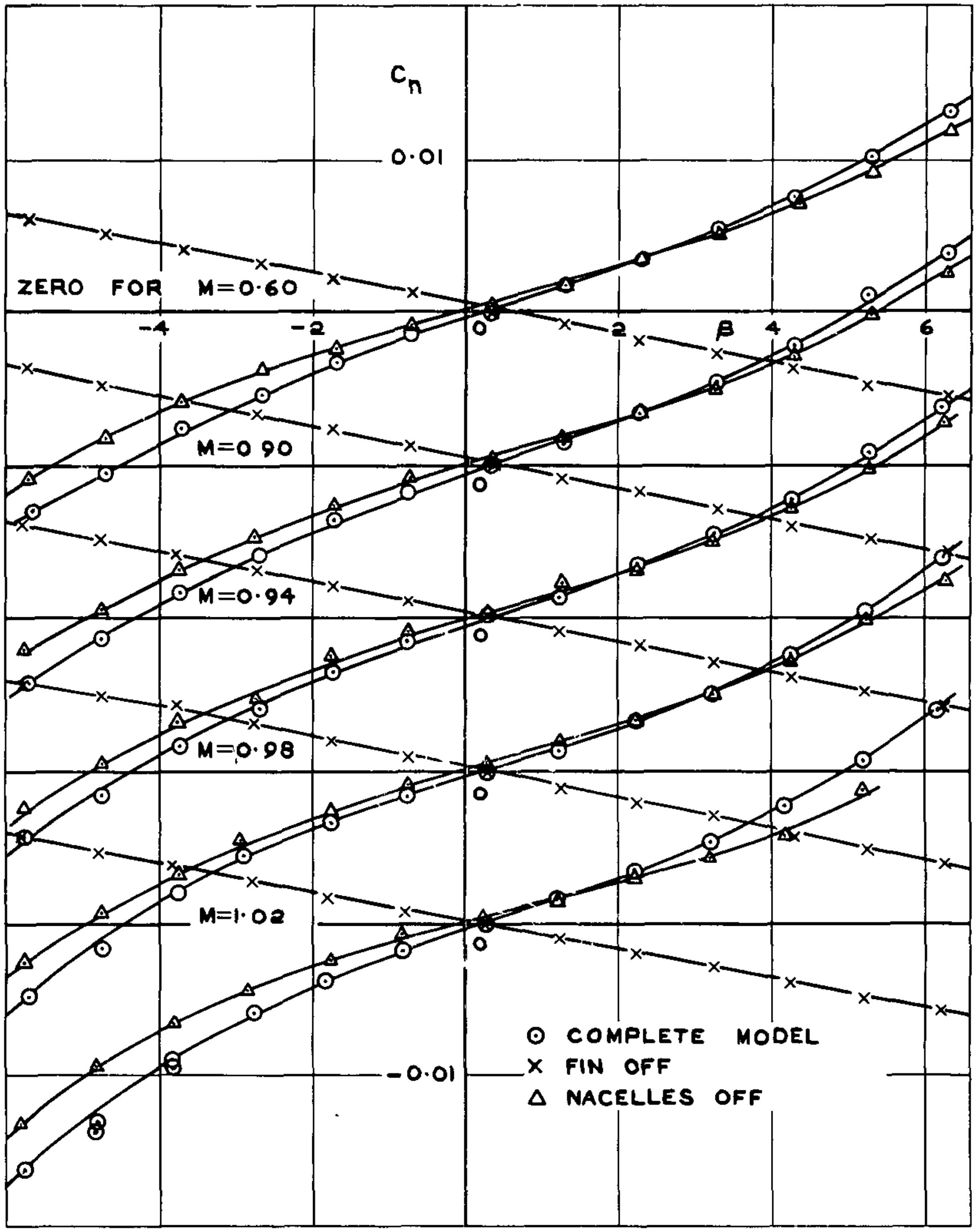

FIG. 14 YAWING MOMENT vs SIDESLIP AT 6½° INCIDENCE

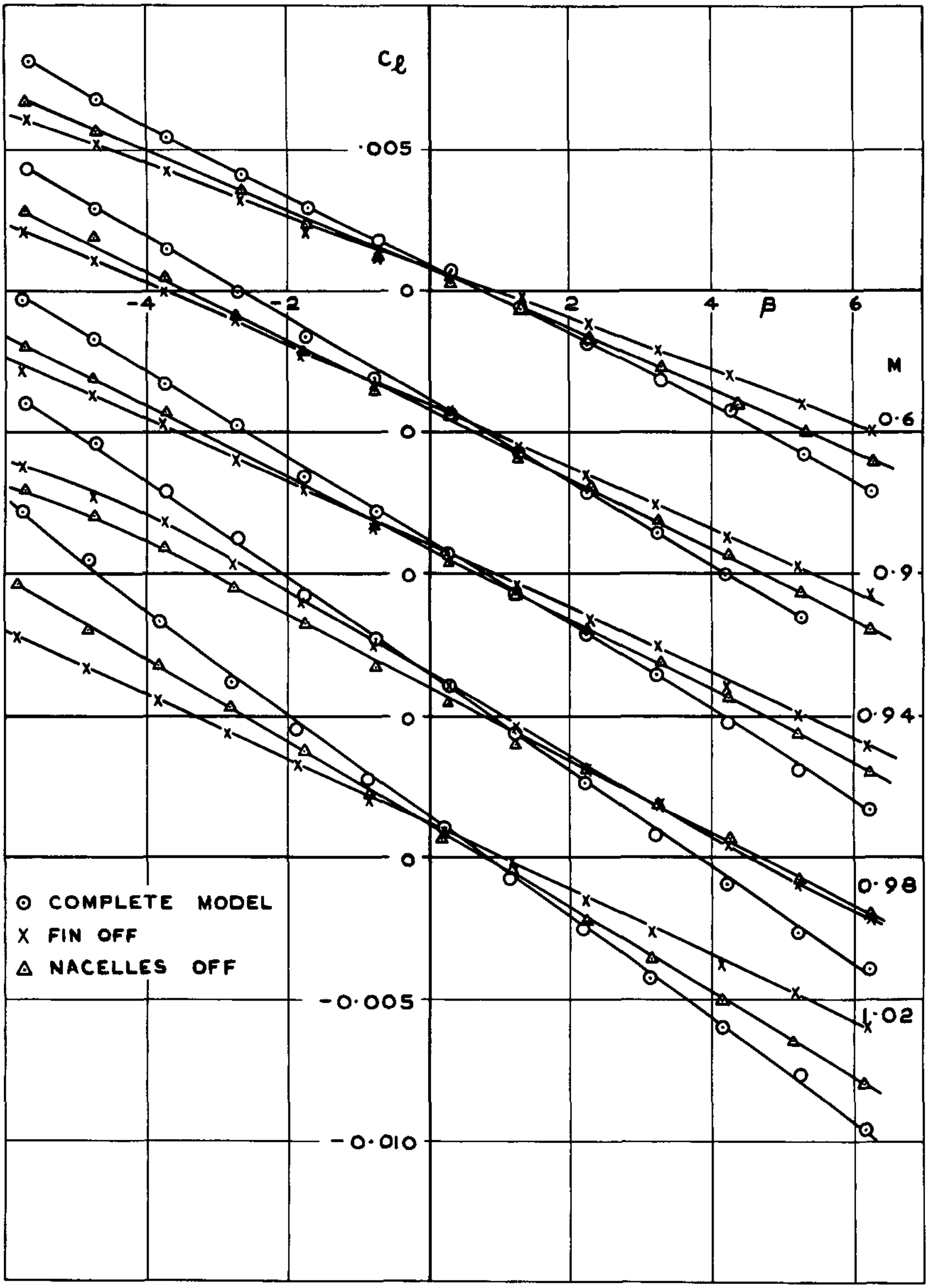

FIG. 15 ROLLING MOMENT vs SIDESLIP AT $6\frac{1}{2}°$ INCIDENCE

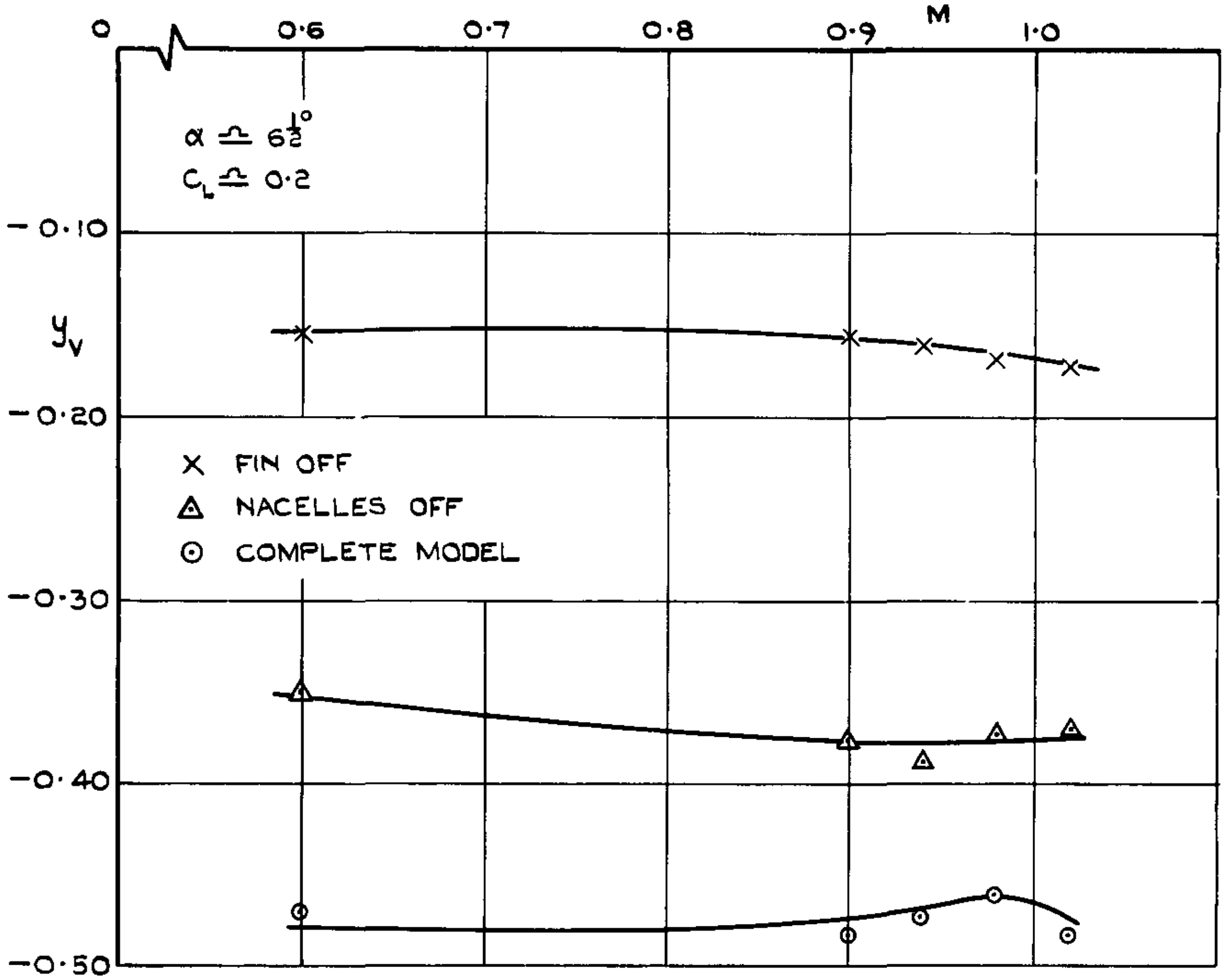

FIG. 16 SIDE FORCE DUE TO SLIDESLIP vs MACH NUMBER

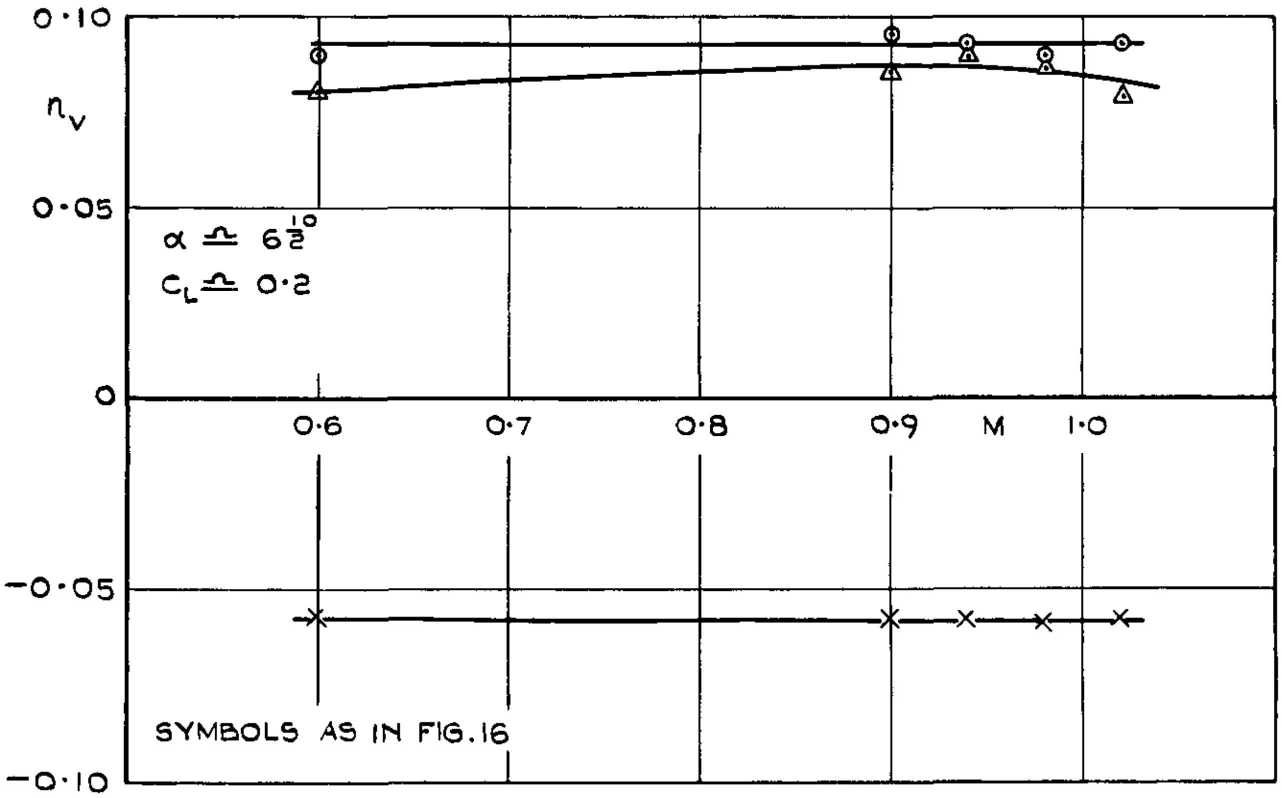

FIG. 17 YAWING MOMENT DUE TO SIDESLIP vs MACH NUMBER

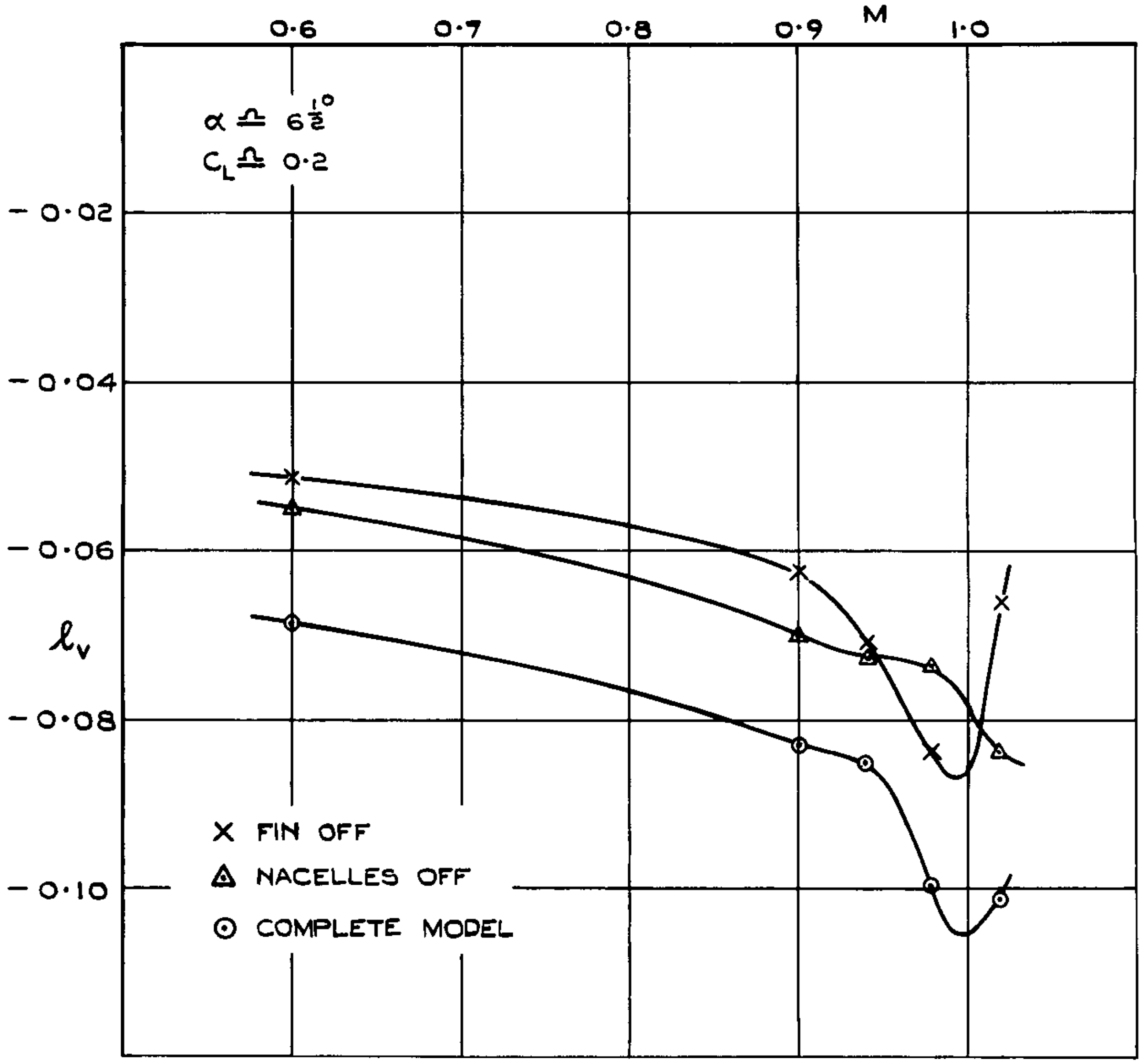

FIG. 18 ROLLING MOMENT DUE TO SIDESLIP vs MACH NUMBER

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