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By K. J. TURNER and G. K. HUNT

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Free-Flight Measurements of the Transonic Roll-Damping Characteristics of Three Related Wings of Aspect Ratio 2·83

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COMMUNICATED BY THE DEPUTY CONTROLLER AIRCRAFT (RESEARCH AND DEVELOPMENT), MINISTRY OF AVIATION

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Summary. Measurements of the roll-damping derivative (l_p) of three wing planforms were made by the free-flight roll-balance technique over the Mach number range 0·7 to 1·4. The wings were all of RAE 102 section, 7·5 per cent t/c and aspect ratio 2·83 but varied in sweep and taper ratio.

The two wings of taper ratio 0·33 showed little loss of damping in the transonic region but the 50 deg delta wing suffered a 50 per cent loss of damping at $M = 0 \cdot 96$.

The results have been compared with simple theoretical estimates and the effects of aero-elasticity have been computed.

1. *Introduction.* The problem of control effectiveness at transonic speeds has been investigated by many methods in recent years but, because much of the work has been of an *ad hoc* nature, the separate effects of aspect ratio, taper, sweep, wing section and control geometry have often been obscured. In an unpublished paper Taylor and Thomas proposed a family of three planforms having various control surfaces chosen so that the main features were varied in a geometrically systematic way. It was hoped that such a relatively simple programme would yield useful data on the separate effects of some of the main parameters involved. A free-flight investigation of the aileron effectiveness and roll-damping of this family† of wings was therefore made, the experiments being of two kinds:

(*a*) The measurement of rolling effectiveness, l_ξ/l_p, of wing-tip and trailing-edge flap controls by the steady-rolling technique[1].

(*b*) The measurement of roll-damping, l_p, by the roll-balance technique[2].

The aileron effectiveness, l_ξ, was therefore obtained by combining the results from the two series of experiments.

The present Report records the results of the roll-damping experiments on the three basic planforms and a separate paper[3] deals with the investigation of control effectiveness. The tests were all made under conditions of zero lift and sideslip and at a mean Reynolds number of 3×10^6.

In the light of more recent knowledge of the transonic and supersonic flow regimes over plane wings we now know that such a geometrically systematic series of planforms may have far from systematic aerodynamic characteristics, since the loadings may be largely dependent on local shocks

* Previously issued as R.A.E. Tech. Note No. Aero. 2683—A.R.C. 22,117.

† The aspect ratio was, however, reduced from 3·1 to 2·83 to fit into a wider series of planforms suggested by C. H. E. Warren.

and separations. Therefore, the results may not be representative for wings which are designed to have given aerodynamic properties. Bearing these limitations in mind, the main interest in the present set of results is now seen as the accurate measurement of roll-damping through the transonic region on three particular uncambered and untwisted aerofoils.

A method is given in the Appendix for the estimation of the effect of aero-elastic distortion. Such calculations showed that the wings did not suffer more than 10 per cent loss of roll-damping in the worst case.

2. *Description of the Models.* The three models, mounted on their test vehicles are illustrated in Fig. 2 and their leading dimensions are given in Table 1 and Fig. 3. Each model had a body turned from compressed wood into which the three wings were tongued and glued. The ratio between the body diameter and the wing span was chosen to be the same as that of the rolling-effectiveness models of Ref. 3 but the body length was reduced as much as possible consistent with keeping the wing panels out of the supersonic interference regions emanating from the ogival nose and blunt base. The wings were machined and hand finished from solid aluminium alloy to a profile accuracy of $\pm 0 \cdot 003$ in. but some of them were found to have a certain amount of twist arising from the manufacturing processes. The measured twist distributions across the span of each panel are plotted in Fig. 7 where it can be seen that on models 1 and 2 the twist was less than 3 minutes and on model 3 about 6 minutes from root to tip. Each panel was glued into its body at an incidence calculated to minimise the rolling moment arising from this twist.

3. *The Experimental Technique.* A full description of the experimental technique is given in Ref. 2. However, the present test vehicles, Figs. 1 and 2, differed from the earlier ones, Fig. 4 of Ref. 2, in that the centre of gravity was further aft, the fin-assembly fairing tube was extended to cover the whole length of the boost motor, and the stabilising fins were flat plates set at $1 \cdot 8$ degrees incidence relative to the body axis instead of having tip controls set at 6 degrees. These modifications were incorporated to overcome the coupled pitch-yaw-roll instability encountered on the earlier test vehicles, and are explained more fully in Ref. 2. No sign of such instability was apparent during the present test programme and it is now believed that the effectiveness of the modifications was mainly due to the separation of the rolling frequency from the pitch-yaw frequency which had been nearly synchronous near $M = 1 \cdot 3$. The fins were modified to increase the rate of roll and the static margin was reduced to lower the pitch-yaw frequency. A typical test-vehicle performance is illustrated in Fig. 4; only curves for model 1 are given as the others were nearly identical.

The model wings were offset 40 degrees in roll, relative to the stabilising fins in order to minimise the interference between the two sets of aerofoil surfaces. The models were sting-mounted on a torsion-bar balance which measured the rolling-moment reaction, L, between the model and the body of the test vehicle. The rate of roll p, and velocity V, were measured from the ground. Then,

$$l_p = \frac{4L}{\rho V b^2 S p},$$

where ρ is the local air density,

b is the diameter of the circle circumscribing the wing-tips of the models,

S is the gross area of the three wing panels of the model.

Measurements were evaluated for the coasting part of the flight only.

4. *Discussion of Results.* The roll-damping derivative (l_p) is plotted against Mach number in Fig. 5 and curves of the corresponding tip-helix angles ($pb/2V$) are given in Fig. 6. All the results were obtained under conditions of zero net lift and at low tip incidence of between $1\cdot3$ and $1\cdot8$ degrees. It is interesting to note that the transition of l_p from its subsonic level to its supersonic level is quite different for each planform. The nearly unswept planform of model 1 shows a very small transonic loss of damping between $M = 0\cdot9$ and $0\cdot94$ and the curve then rises steeply to the supersonic value; the l_p curve for the swept planform of model 2 shows a small but distinct 'dip' at $M = 0\cdot97$ on its more shallow rise to the supersonic level; the cropped delta planform of model 3 exhibits a marked loss of damping, some 50 per cent in the transonic region.

The difference between models 1 and 2 is in sweepback only and the well known effectiveness of sweepback in delaying the establishment of supersonic-flow characteristics is clearly apparent.

The aerodynamic characteristics of the model 2 planform have been investigated in considerable detail by Hall and Rogers[4] for the case of symmetrical loading produced by uniform incidence across the span. Although their results are not directly applicable to the present anti-symmetric case they do indicate that leading-edge separation is unlikely at the incidences of the present test and that a small shock may appear near the tip at $M = 0\cdot95$. A local flow separation arising from such a shock could account for the small dip in the l_p curve at $M = 0\cdot97$. The fact that the dip is rather more marked on model 2 than model 1 follows from the sweep effect of shifting the aerodynamic loading outwards so producing higher suctions and hence stronger adverse pressure gradients on the tip sections of the swept wing. Theoretical estimates[5] give a 10 per cent increase in the local lift coefficient in the tip regions of model 2 compared with model 1.

In comparing the results from models 2 and 3 we have to take into account the effects of both taper ratio and sweep. The sudden loss of damping of model 3 at $M = 0\cdot97$ is clearly a transonic effect and in this region a useful parameter for comparison is the sweep of the quarter-chord line. In this respect the models do not differ very much, model 2 having 49 deg 39 min quarter-chord sweep and model 3, 41 deg 45 min sweep (Table 1). The reduction of taper ratio, however, from $0\cdot33$ to $0\cdot086$ produces a considerable increase of the tip loading; theoretical estimates[5] show a 50 per cent increase in local lift coefficient at 90 per cent semi-span. The associated pressure gradients have apparently been strong enough to promote a considerable shock-induced separation which has more than halved the roll-damping effectiveness in this region. At higher tip incidences the wings of models 1 and 2 would probably also show a similar loss of effectiveness at transoinc speeds but it is not possible from the present results to say which would be the worse in this respect.

The experimental results have been compared with theoretical estimates from Weissinger's simplified lifting-surface theory[5] for the subsonic values and from linearised supersonic-flow theory[6] for the supersonic values. The estimates include an allowance for interference between the three wing panels at supersonic speeds only; subsonically the interference was assumed to be negligible.

The generally good agreement between the experimental results and the theoretical estimates is perhaps somewhat fortuitous because neither of the theories adequately deal with the real flow conditions over such planforms at these Mach numbers. The theoretical estimates have, however, been given to provide a frame of reference for the experimental results.

The experimental results are influenced by aero-elastic distortion of the wing panels and this effect has been investigated by the method given in the Appendix. Calculations for Mach numbers of $0\cdot8$ and $1\cdot4$ indicate that the loss of l_p from aero-elasticity is negligible for model 1 and, at subsonic speeds for models 2 and 3. At $M = 1 \cdot 4$ model 2 suffers a loss of 10 per cent and model 3 a loss of about 6 per cent of their rigid-wing values. These corrections have been applied to the experimental results at $M = 1 \cdot 4$ on Fig. 5 and would get progressively less with decreasing Mach number.

Apart from these aero-elastic effects the overall accuracy of the experimental results should be within $\pm$ 5 per cent.

5. *Conclusions.* The successful measurement of the roll-damping derivatives on three related wings has shown that a simple and reliable technique has been evolved using a non-separating rocket-powered test vehicle equipped with a roll balance. Earlier instability troubles experienced with this type of test vehicle have been completely overcome.

For the three planforms tested the character of the transition of l_p from subsonic to supersonic flow differed in each case. The wings of moderate taper ratio showed little loss of damping in the transonic region whereas the cropped delta planform suffered a 50 per cent loss of damping between $M = 0 \cdot 95$ and $1 \cdot 0$. The general trends were in agreement with existing knowledge of the flow characteristics over swept wings at transonic speeds.

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APPENDIX

The Estimation of Aero-Elastic Effects

The loss of rolling moment arising from aero-elastic distortion of the wings was calculated by an iterative method, derived from that of Broadbent[7], which considers the wing as being divided into a number of chordwise strips, each rigid in itself but torsionally spring-restrained to its neighbours. Thus the effect of wing twist is taken into account but spanwise bending is neglected as it has very little effect on the aerodynamic load distribution.

The wing flexibilities were determined experimentally by applying concentrated loads at the quarter-chord and half-chord points of the three stations, A, B, C shown in Fig. 8 and measuring the twists induced at all these stations each time. Thus the flexibility matrices were found:

$$[\theta_{1/4}] = \begin{bmatrix} \theta_{AA} & \theta_{BA} & \theta_{CA} \\ \theta_{AB} & \theta_{BB} & \theta_{CB} \\ \theta_{AC} & \theta_{BC} & \theta_{CC} \end{bmatrix}$$

where θ_{RS} is the rotation of station R due to unit load applied at the quarter-chord point of station S. Similarly $\theta_{1/2}$ was determined for the loading at the half-chord points. Curves for $\theta_{1/4}$ and $\theta_{1/2}$ are given in Fig. 8c to f.

For the purposes of computation, the wing panels were divided into six chordwise strips of equal width and the curves of Fig. 8c to f were interpolated to obtain the flexibility coefficients for the centre-lines of each of the six strips; thus new square matrices were formed of order six.

The aerodynamic loadings over the wings were obtained theoretically (Refs. 5 and 6) for the rigid rolling wing, *i.e.*, for a linear distribution of incidence across the span. The loading on each strip was then assumed to act at the quarter-chord point for the subsonic case and at the calculated centre of pressure position for the supersonic case. Where the local centre of pressure positions fell between the quarter- and half-chord points the appropriate new flexibility coefficients were obtained by interpolation from the measured stiffness data.

The local lift, ΔL_S, on strip S causes the incidence of strip R to change by an amount $\Delta L_S \,.\, \phi_{RS}$, therefore the total change of incidence of strip R is

$$\Delta\alpha_R = \sum_{S=1}^{6} \Delta L_S \, \phi_{RS}$$

This calculation is done for each of the strips. Thus a new spanwise incidence distribution is obtained and hence a modified loading distribution. This process is repeated until it converges, generally after only three iterations, to a sufficient degree of approximation. The ratio of the final rolling moment to the original one for the rigid wing gives the aero-elasticity factor.

No results are given for model 1 since an approximate calculation showed that it would suffer negligible aero-elastic distortion within the Mach number range covered in these tests.

TABLE 1

Model Data

Model	1	2	3
Span (ft)	1·277	1·277	1·096
Exposed semi-span (ft)	0·548	0·548	0·458
Centre-line chord (ft)	0·677	0·677	0·714
Root chord (ft)	0·613	0·613	0·606
Tip chord (ft)	0·226	0·226	0·061
Leading-edge sweep	27° 57′	53° 33′	49° 58′
Quarter-chord sweep	19° 29′	49° 39′	41° 45′
Half-chord sweep	10° 1′	45°	30° 45′
Trailing-edge sweep	−10° 1′	32° 53′	0
Gross aspect ratio	2·83	2·83	2·83
Gross taper ratio	0·33	0·33	0·086
Wing section	RAE 102, t/c 7·5 per cent		
Body length (ft)	1·750		
Body diameter (ft)	0·181		
Body nose profile	4 calibre ogive		

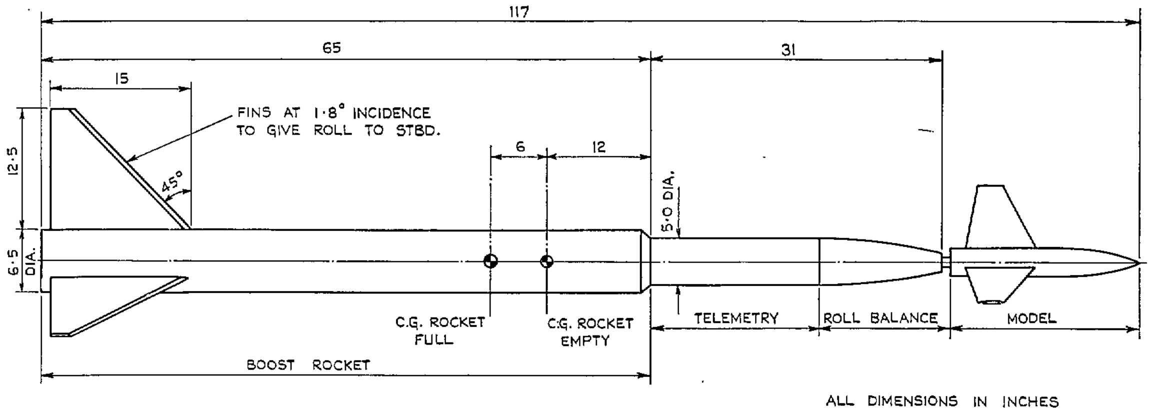

FIG. 1. General arrangement of test vehicle.

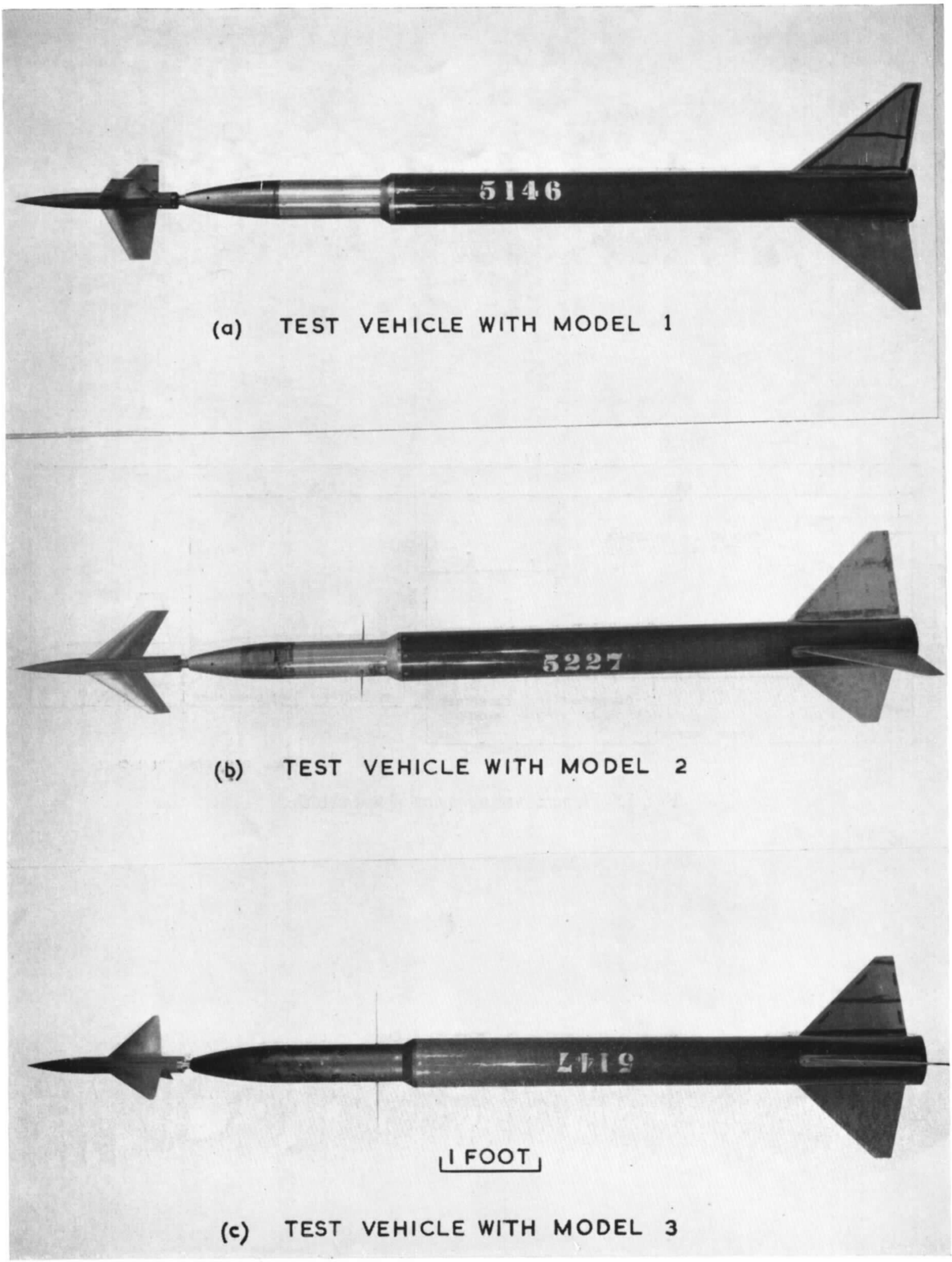

FIG. 2. Test vehicles.

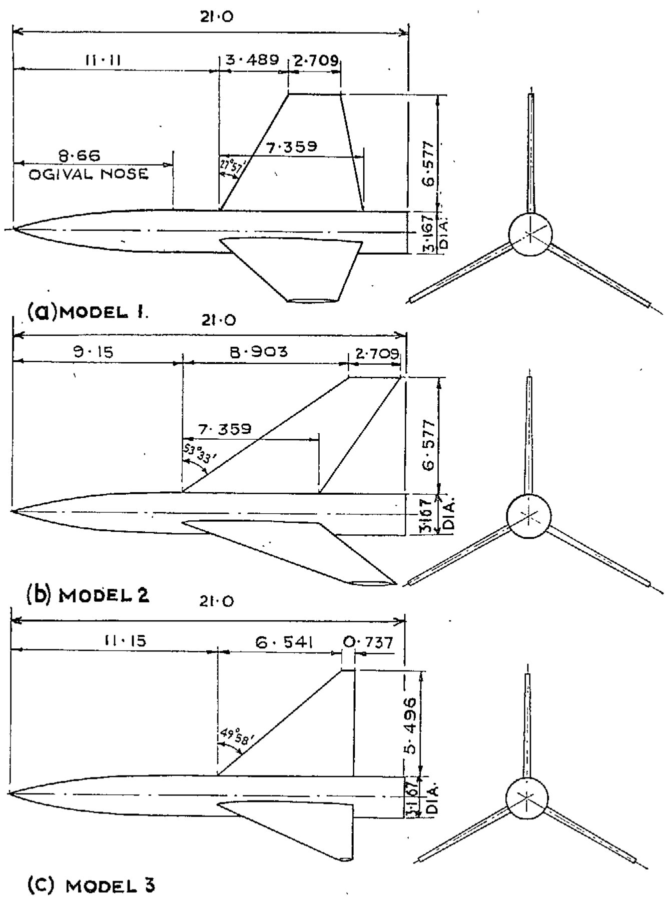

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FIG. 3. General arrangements of models.

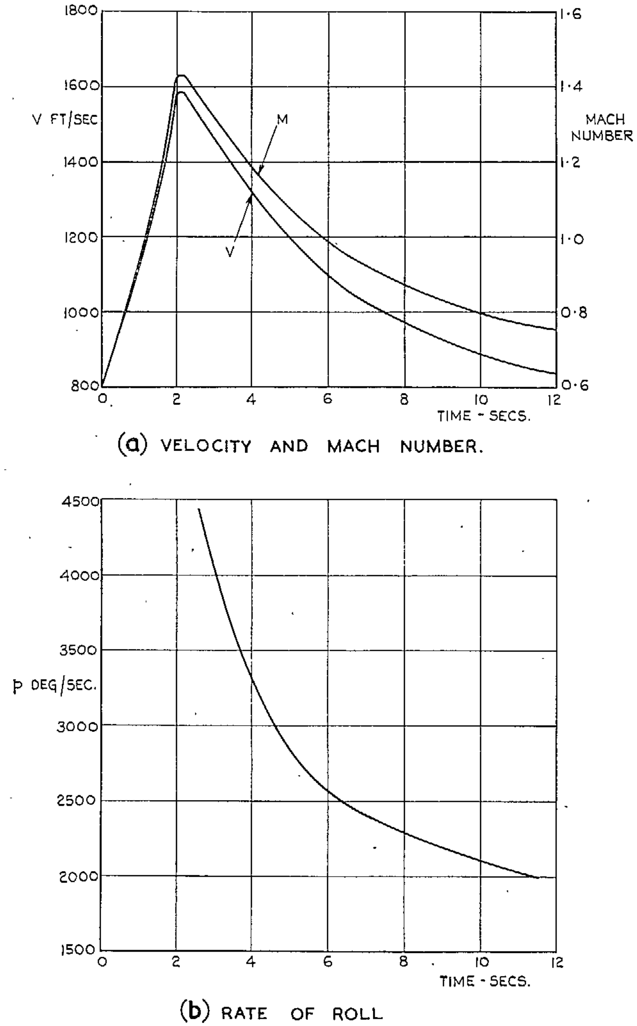
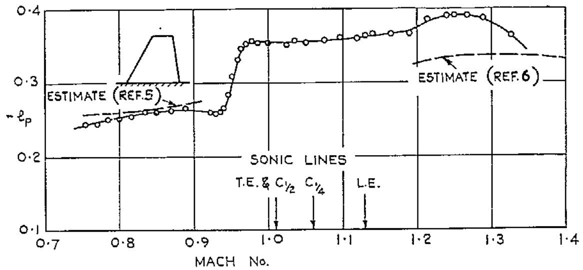
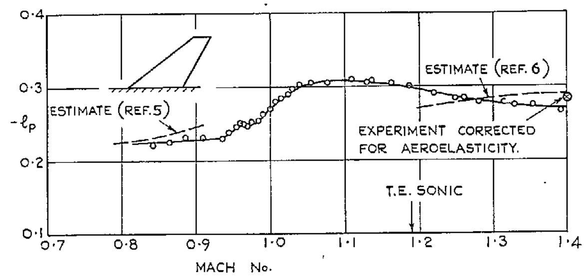


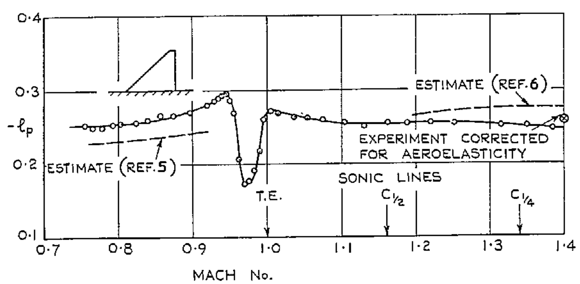
FIG. 4. Test vehicle performance (Model 1).



(a) MODEL 1 $A = 2 \cdot 83$ $\lambda = \cdot 333$ $\Lambda_{LE} = 27° 57'$

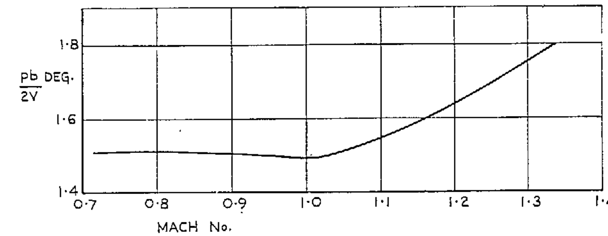


(b) MODEL 2 $A = 2 \cdot 83$ $\lambda = \cdot 333$ $\Lambda_{LE} = 53° 33'$

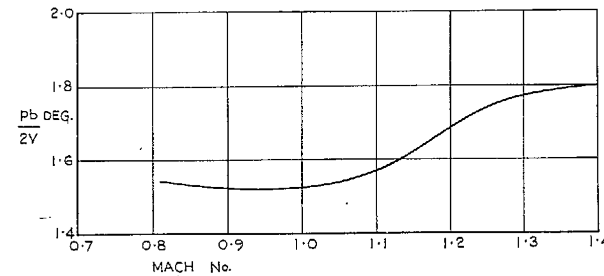


(c) MODEL 3 $A = 2 \cdot 83$ $\lambda = 0 \cdot 086$ $\Lambda_{LE} = 49° 58'$

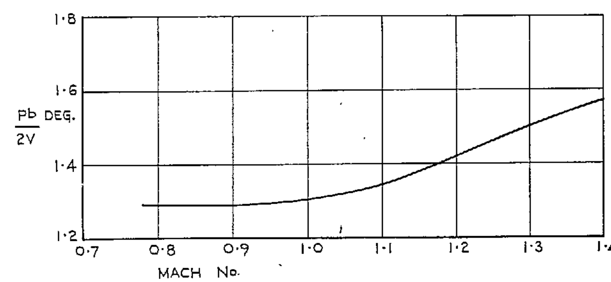
FIG. 5. Roll-damping derivative l_p. $R_e = 3 \times 10^6$ at $M = 1$.



(a) MODEL 1



(b) MODEL 2



(c) MODEL 3

FIG. 6. Tip-helix angle.

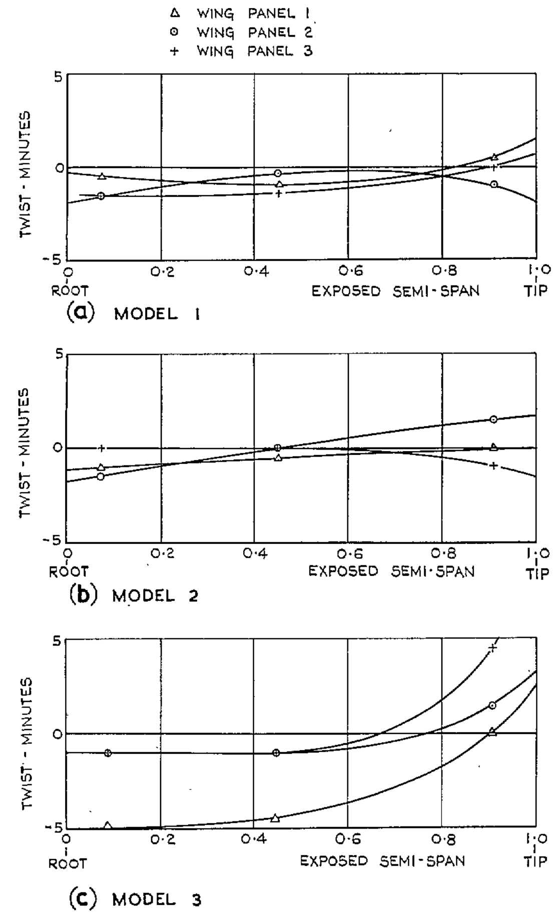


FIG. 7. Spanwise twist distributions (arising from manufacturing errors).

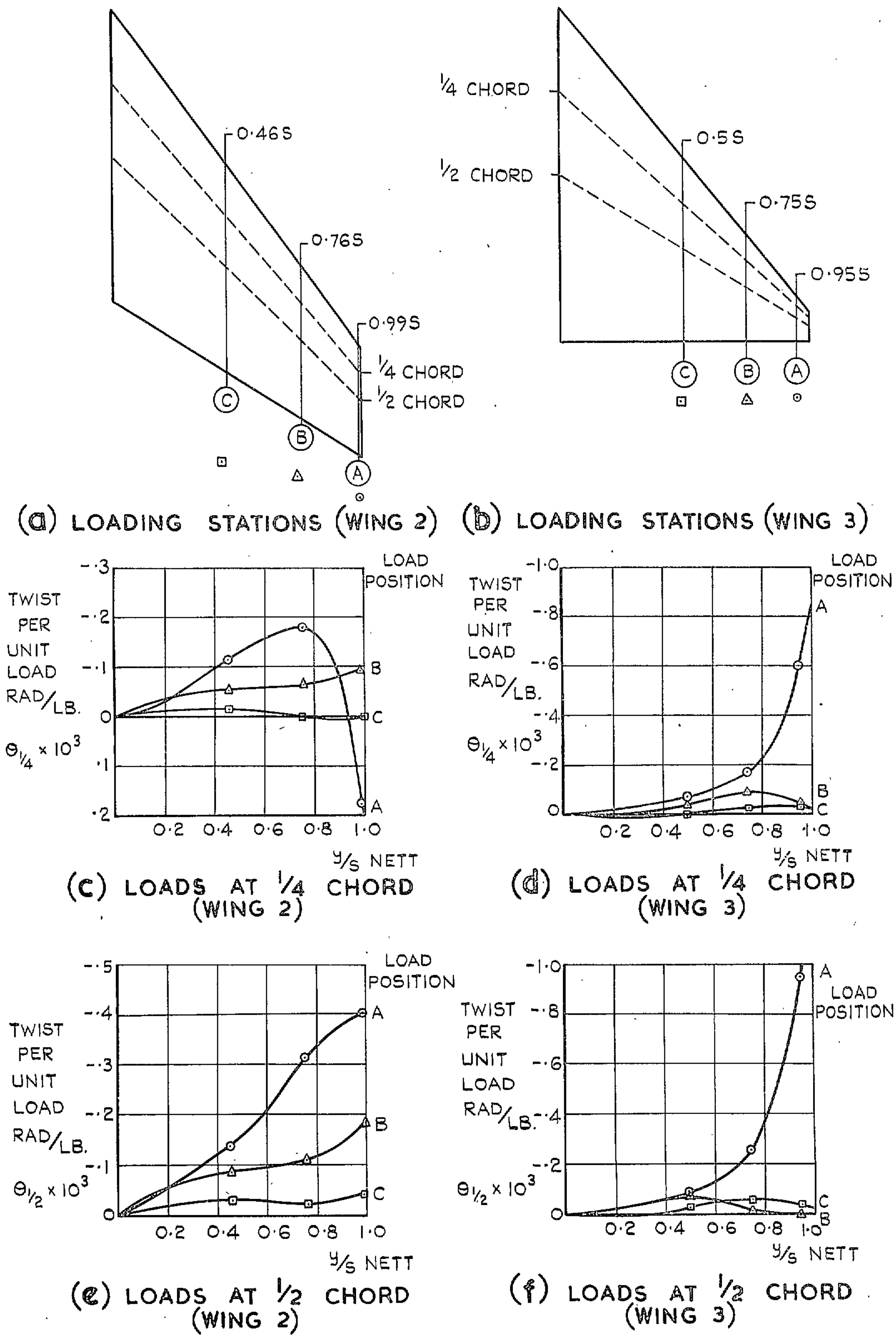


FIG. 8. Measured stiffness of model wings. (θ positive when positive lift force increases local incidence.)

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