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Flight Measurements at Subsonic Speeds of the Aileron Rolling Power and Lateral Stability Derivatives ℓ_v and y_v on a 60° Degree Delta Wing Aircraft (Fairey Delta 2)

by

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June, 1963

FLIGHT MEASUREMENTS AT SUBSONIC SPEEDS OF THE AILERON ROLLING POWER AND LATERAL STABILITY DERIVATIVES ℓ_v AND y_v ON A 60 DEGREE DELTA WING AIRCRAFT (FAIREY DELTA 2)

by

F. W. Dee, A.F.R.Ae.S.

SUMMARY

The aileron rolling power of the Fairey Delta 2 has been measured at subsonic speeds, by a method using asymmetric wingtip weights. The lateral stability derivatives ℓ_v and y_v have also been determined from measurements in steady straight sideslips.

The results have been compared with wind tunnel measurements and some differences found, the tunnel value of $-\ell_\xi$, for example, being about 20% higher than that measured in flight.

The differences could be partially due to aerodynamic interference in the flight tests from the externally mounted wingtip weights. It is hoped to make further tunnel tests with these represented.

Replaces R.A.E. Tech Note No. Aero 2897 - A.R.C.25,204

LIST OF CONTENTS

LIST OF ILLUSTRATIONS

1 INTRODUCTION

The Fairey Delta 2 is a research aircraft, built to investigate the characteristics of a 60 degree delta wing planform over a wide range of lift coefficient and Mach number. The information obtained from the flight research programme is to be used as a basis for comparing these characteristics with wind tunnel measurements on representative models, and with theoretical estimates. This report forms part of the overall lateral stability and control investigation.

The principal lateral stability derivatives have previously been determined dynamically, by analysing the Dutch roll characteristics of the aircraft[1]. The present tests were made to measure the lateral control power by a static method. In addition, the static values of the rolling moment due to sideslip derivative ℓ_v, have been obtained, for comparison with the previous dynamic measurements.

In the present tests, externally fitted wingtip weights were used to provide a known rolling moment, and the aileron rolling power was determined by measuring the aileron angle required to counteract a given moment. Due to the bluffness of the wingtip weight fairings, the tests were limited to an equivalent airspeed of 235 knots.

Further flight tests are planned, using a wingtip parachute, the development of which is described in Ref.2, to supply a known yawing moment; these tests will allow the measurement of the rudder yawing power n_ζ, and also the directional stability derivative, n_v.

2 DESCRIPTION OF AIRCRAFT

The Fairey Delta 2 is a tailless research aircraft with a 60 degree delta wing of thickness-chord ratio 0.04, and is powered by a Rolls Royce RA.28 turbo-jet engine. Figs.1 and 2 show photographs of the aircraft with the wingtip weight canisters fitted. The principal dimensions of the aircraft are given in Table 1; Fig.3 shows a general arrangement of the aircraft.

The aircraft has separate elevators and ailerons. The elevators extend from the fuselage side to 57% semispan, and the ailerons, which are rigged at a nominal angle of 3 degrees to the wing chord, occupy the remainder of the trailing edge. A small amount of aileron aerodynamic balance is provided by a wingtip horn, the ratio of area ahead of the hinge line to total aileron area being 0.034. All the controls are irreversible and power operated; artificial feel is provided by simple springs.

The wingtip weight canister used for the present tests may be fitted to either wing at 11.7 feet from the aircraft centre line. The empty weight of the canister is 65 pounds; lead ballast can be added to the canister in increments of 50 pounds to bring the total weight up to 815 pounds. To maintain aerodynamic symmetry, a similar canister may be fitted to the other wing.

The mean fore-and-aft position of the centre of gravity of the aircraft was maintained at 54% of the wing centre-line chord (31.5% M.A.C.) by the addition of ballast in the front fuselage section.

3 INSTRUMENTATION

The following quantities, relevant to the tests, were recorded on Hussenot A.22 trace recorders running at a nominal paper speed of one inch per second:-

Sideslip angle	range ±5 degrees
Starboard aileron angle*	range 12 degrees up to 10 degrees down, relative to wing chord
Rudder angle	range ±8 degrees
Lateral acceleration	range ±0.25g
Rate of roll	range ±20 degrees per second

In addition, indicated airspeed, altitude, and fuel consumed were obtained from automatic observer instruments, photographed by an Eclair cine camera.

4 FLIGHT TESTS

The tests were made in level flight at two altitudes; 40,000 feet, at equivalent airspeeds of approximately 172, 195, 215 and 235 knots, corresponding to lift coefficients between 0.38 and 0.19, and 20,000 feet, at equivalent airspeeds of approximately 150 and 175 knots, corresponding to lift coefficients of 0.47 and 0.35. The Reynolds number of the tests, based on the wing aerodynamic mean chord, varied between 19.5×10^6 and 26.7×10^6 for the tests at 40,000 feet, and between 22.4×10^6 and 26.1×10^6 for those at 20,000 feet.

Records were obtained in steady level flight at approximately 0, $\pm 2\frac{1}{2}$ and ±5 degrees of sideslip with the empty weight canisters fitted, then with 250, 350 and 450 pounds weight in the port canister, and finally with 450 pounds weight in the starboard canister only. Although the canister could have been ballasted to a total weight of 815 pounds, no tests were made with a weight greater than 515 pounds, because of the extreme difficulty in taxying the aircraft with the asymmetric load. In flight, aileron application was necessary to hold the wings level; the amount required increased with reduction of speed. Therefore, to allow sufficient lateral control during a crosswind approach in turbulent air, flights were restricted to conditions in which the maximum crosswind component was $7\frac{1}{2}$ knots for a canister total weight of 515 pounds. The restriction was progressively relaxed for smaller canister weights; for the aircraft symmetrically loaded, the maximum acceptable crosswind component was 20 knots.

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*It was intended also to measure port aileron angle; however the transducer became unserviceable during the tests and could not be replaced. Thus, no port aileron measurements have been presented.

5 CORRECTIONS

The indicated airspeed and altimeter readings have been corrected for instrument and position errors, the latter corrections being obtained from earlier, unpublished measurements on the aircraft. The sideslip vane readings have been corrected for the effects of flow distortion due to the nose boom and the aircraft fuselage and for the effects of inclination of the vane to the flow direction; the sidewash due to the nose boom has been measured previously in a wind tunnel[3], whereas values of the fuselage sidewash are estimated[4]. The corrected sideslip is 6% less than that indicated by the vane, and at the highest incidence, a further correction of about $1\frac{1}{2}$% is necessary to allow for the inclination of the vane to the flow direction.

6 METHOD OF ANALYSIS

It is assumed that the conditions are those appropriate to steady straight sideslips, i.e., that the aircraft rates of roll and yaw are zero, so that $\ell_p \frac{p\,s}{V}$ and $\ell_r \frac{r\,s}{V}$ are both zero. The validity of this assumption is discussed more fully in section 7.1.

The rolling moment equation in a steady straight sideslip for the aircraft symmetrically loaded is:-

$$\ell_v \beta + \ell_\xi \xi + \ell_\zeta \zeta = 0 \tag{1}$$

and for the aircraft with asymmetric loading is:-

$$\ell_v \beta + \ell_\xi \xi' + \ell_\zeta \zeta' + \frac{L_w}{\rho V^2 S s} = 0 \tag{2}$$

where the moment applied by the wingtip weight is:-

$$L_w = (W_s - W_p)\, y_w$$

where W_s is the total weight carried on the starboard wingtip (pounds)

W_p is the total weight carried on the port wingtip (pounds)

y_w is the moment arm of the wingtip canisters about the aircraft plane of symmetry (feet).

Subtracting equation (1) from equation (2) gives:-

$$\ell_\xi \Delta\xi + \ell_\zeta \Delta\zeta + \frac{L_w}{\rho V^2 S s} = 0 \qquad (3)$$

where $\Delta\xi = \xi' - \xi$, the change in aileron angle to trim at constant sideslip (radians)

$\Delta\zeta = \zeta' - \zeta$, the change in rudder angle to trim at constant sideslip (radians).

Equation (3) may be used to determine the aileron rolling power, ℓ_ξ, provided, in general, ℓ_ζ is known. In this equation, the change in rudder angle to trim, $\Delta\zeta$, arises from the yawing moment due to aileron deflection, $N_\xi \Delta\xi$. If N_ξ is zero, $\Delta\zeta$ is zero, and equation (3) simplifies to:-

$$\ell_\xi \Delta\xi + \frac{L_w}{\rho V^2 S s} = 0 \; . \qquad (4)$$

Once ℓ_ξ is determined, equation (1) may be used to find the rolling moment due to sideslip derivative, ℓ_v, thus:-

$$-\ell_v = \ell_\xi \frac{\xi}{\beta} + \ell_\zeta \frac{\zeta}{\beta} \; . \qquad (5)$$

The value of the derivative, ℓ_ζ, was determined from wind tunnel tests[5], and is presented in Fig.4.

The sideforce equation for a straight, steady sideslip is:-

$$y_v \beta + \tfrac{1}{2} C_L \phi + y_\zeta \zeta + y_\xi \xi = 0 \qquad (6)$$

hence;

$$-y_v = \tfrac{1}{2} C_L \frac{\phi}{\beta} + y_\zeta \frac{\zeta}{\beta} + y_\xi \frac{\xi}{\beta} \; . \qquad (7)$$

Values of y_ζ and y_ξ, determined from wind tunnel tests[5], are presented in Fig.4. The angle of bank, ϕ, is determined from the lateral accelerometer reading, since, in a steady straight sideslip at small angles of bank, $a_y = \phi$, where a_y is the lateral accelerometer reading in 'g' units.

7 RESULTS AND DISCUSSION

7.1 General

The aileron deflections required to trim for various sideslip angles, with and without the wingtip weights, are shown in Fig.5 for each airspeed. It is thought that the accuracy of measurement of aileron angle is within ±0.1 degrees, and of sideslip ±0.05 degrees. Only values of starboard aileron deflection, relative to the wing chord, are presented. It should be remarked that, although the ailerons are rigged at a nominal angle of 3 degrees up, relative to the wing chord, the curves of aileron angle to trim at zero sideslip, with no asymmetric weight, indicate that in flight the starboard aileron trim position corresponded to 4.5 degrees deflection from the wing chord. The difference is thought to be due to airframe asymmetry which necessitated some aileron deflection to achieve trimmed conditions at zero sideslip. The corresponding rudder angles to trim are shown in Fig.6.

The control angles to trim have been corrected to the mean equivalent airspeed quoted for each condition, assuming that the various non-dimensional derivatives are unaffected by small changes in speed. This implies that the changes in aileron and rudder angles to trim the wingtip weight are inversely proportional to dynamic pressure. The largest correction applied to the measured aileron angle at 150 knots E.A.S. was 0.20 degrees, corresponding to a speed correction of 4 knots. Figs.5 and 6 show that, in general, the curves of aileron and rudder angle to trim against sideslip for the various applied rolling moments are parallel straight lines. However, at a speed of 172 knots at 40,000 feet, with wingtip weights of 250 and 350 pounds on the port wing neither the aileron nor rudder angle to trim is linear with sideslip (Fig.5(d) and Fig.6(d)). In both cases however, the control angles to trim at zero sideslip appear to be consistent with those for other wingtip weights.

The reason for the non-linearities in the aileron and rudder trim curves is not clear, as both produce more positive rolling moments than the linear relationship between sideslip angle and control angle to trim indicate. However, flight pressure plotting measurements on the aircraft[6] indicate that two different flow patterns, giving different pressure distributions, are possible under nominally similar conditions. The non-linearities may have been caused by a change of flow pattern during the present tests.

Fig.7 shows the variation of angle of bank with sideslip for the tests at altitudes of 40,000 and 20,000 feet. The accuracy of measurement is thought to be ±0.05 degrees. The measured values have been corrected to the mean equivalent airspeed by assuming that the angle of bank for a given sideslip is inversely proportional to lift coefficient, and that variations in the non-dimensional control derivatives may be neglected.

The angle of bank is linear with sideslip, and at each speed, the points for all wingtip weights lie, within the limits of experimental accuracy, on a single line.

Examination of the flight records showed that during most of the tests, the aircraft was oscillating in roll and yaw in response to small corrective control movements, of the order of ±0.2 degrees of aileron, and ±0.1 degrees for the rudder. The maximum rate of roll induced was of the order of ±3 degrees per second, and the period varied between 1 and 3 seconds; the motion in yaw was much smaller. In view of the small magnitude of the oscillation, no corrections have been applied in the analysis for the aerodynamic moment due to the small terms $\ell_p \frac{ps}{V}$ and $\ell_r \frac{rs}{V}$.

7.2 Aileron rolling power, ℓ_ξ

The aileron and rudder angles to trim, at zero sideslip, for various wingtip weight distributions are presented in Figs.8 and 9 respectively. The control angles to trim are interpolated from the curves of Figs.5 and 6. The rudder angle to trim at zero sideslip, Fig.9, is unaffected by changes in wingtip weight, indicating that the yawing moment due to aileron deflection is negligible for aileron deflections up to $2\frac{1}{2}$ degrees from neutral, and also that the use of the simplified equation (4) of section 6 is justified in determining the aileron rolling power, ℓ_ξ.

Fig.10 shows the variation of $\Delta\xi$, the change of aileron angle required to trim the wingtip weight, with the rolling moment coefficient due to wingtip weight, for each lift coefficient tested. The gradient, $\frac{C_{\ell_w}}{\Delta\xi}$, of the best straight line through the points for each lift coefficient, used in conjunction with equation (4), gives the measured value of ℓ_ξ at that lift coefficient. The values of ℓ_ξ so derived are presented in Fig.11, as a function of C_L, for the aircraft with wingtip weight canisters fitted. The tests at 40,000 feet altitude yield a value of $-\ell_\xi = 0.136$ at $C_L = 0.193$, which rises to 0.148 at $C_L = 0.23$, then falls to 0.143 at $C_L = 0.360$. At $C_L = 0.348$ at 20,000 feet, however, $-\ell_\xi = 0.131$, and at $C_L = 0.473$, $-\ell_\xi$ is reduced to 0.128. When considered as a function of Mach number, $-\ell_\xi$ increases from 0.128 at M = 0.3 to 0.148 at M = 0.755, and then falls sharply to 0.136 at M = 0.825. The reduction in $-\ell_\xi$ above M = 0.755 may be due to compressibility effects as the aerofoil critical Mach number is approached. Flight pressure plotting measurements[6], indicate that the critical Mach number for the wing is about 0.75 in level flight at 40,000 feet, corresponding to a lift coefficient of 0.24.

Measurements of aileron rolling power on a 1/24th scale model in the 8 foot × 6 foot tunnel at R.A.E. Farnborough[5], give a value of $-\ell_\xi = 0.160$ at $C_L = 0.18$ and a Mach number of 0.85 (Reynolds number = 1.5×10^6 based on mean aerodynamic chord), and tests on a 1/9th scale model in the 8 foot × 8 foot tunnel at R.A.E. Bedford[7], yield a value of 0.169 at a Mach number of 0.82 (Reynolds number = 8×10^6). The two wind tunnel tests are in fair agreement, but they are some 18% - 24% higher than the value measured in the flight tests at the same Mach number, (see Fig.11).

Ground tests have shown that aeroelastic distortion of the ailerons is negligible for loads typical of those imposed during the present tests, and it is thought that the difference may be due partly to aerodynamic interference from the wingtip weight canisters which were fitted for the flight tests, but which were not represented on the wind tunnel models. This difference emphasises the desirability of stowing the wingtip weights internally when possible.

Theoretical estimates of the aileron rolling moment derivative[8], also shown in Fig.11, are in reasonable agreement with the flight measurements at 20,000 feet altitude. The flight values at 40,000 feet, however, start to diverge from the estimated values at low lift coefficients; at $C_L = 0.27$, the predicted value exceeds the flight value by about 9%. No allowance has been made in the estimates for the effects of the wingtip weight canisters, which are difficult to assess, and it is hoped to make further tunnel tests with the canisters represented.

7.3 Rolling moment due to sideslip, ℓ_v

The slopes of the curves of aileron and rudder angle with sideslip, measured from Figs.5 and 6, have been used to derive ℓ_v, in conjunction with equation (5) of section 6. Fig.12 shows ℓ_v as a function of C_L for the two test altitudes, 40,000 and 20,000 feet.

A single line is drawn through the points for both altitudes as the effect of Mach number on ℓ_v is negligible in this range.

At $C_L = 0.19$, $-\ell_v = 0.060$, rising to 0.110 at $C_L = 0.47$. Also shown in Fig.12 are the results of previous Dutch roll flight tests[1] and wind tunnel tests on representative models[5,7]. The Dutch roll tests did not cover as large a range of lift coefficient as the present tests; they show a similar trend with C_L, but are some 25-30% higher. The difference between the static and dynamic values of ℓ_v could be due to a genuine effect of oscillation frequency of the Dutch roll; tunnel tests[9] have shown that differences of this order are possible, but the present difference may be due partly to using estimated values of moments of inertia in the analysis of the dynamic flight tests.

Comparison of wind tunnel measurements of ℓ_v with those of the present tests shows that the former are about 18% higher than the flight results. Part of the difference may be due to aerodynamic interference from the wingtip weight canisters which were not represented in the wind tunnel tests.

7.4 Sideforce due to sideslip, y_v

The variation of y_v with C_L is presented in Fig.13. This was derived from the results of Figs.5, 6 and 7, together with wind tunnel values of the control derivatives (Fig.4). The results are compared with those obtained by analysis of Dutch roll tests[1], and wind tunnel results[5,7]. The present tests yield a value of $-y_v = 0.22$ throughout the range $0.19 < C_L < 0.47$. The tunnel tests are in good agreement with the present flight values; however, the Dutch roll method gives a value of $-y_v = 0.165$ between C_L 0.20 and 0.29, which is about 25% lower than the present tests. In the analysis of the present tests, it has been necessary to use values of the control derivatives y_ξ and y_ζ based on wind tunnel tests[5]. Whilst the contribution of sideforce due to aileron deflection is small, that due to rudder deflection is, at low lift coefficients, as much as 60% of that due to angle of bank; thus any error in the assumed derivative y_ζ, will have a large effect on y_v. At the same time, the effect of oscillation frequency on the derivative y_v deduced from the Dutch roll tests may be significant. Wind tunnel measurements[10] of y_v by static and oscillatory techniques on a 42 degree swept wing of low aspect ratio, and moderate taper ratio exhibit considerable differences. At low incidence, the oscillatory value of $-y_v$ is 22% lower than the static value at sideslip amplitudes typical of those in the Dutch roll oscillation. Further flight tests are planned on the Fairey Delta 2 with a wingtip parachute, which will provide information from which y_ζ can be derived, thus allowing a more accurate determination of y_v.

8 CONCLUSIONS

Flight tests covering a range of lift coefficient from 0.19 to 0.47, and up to a Mach number of 0.82 have been made on the Fairey Delta 2 aircraft to determine the aileron rolling moment derivative, by measuring the aileron angle required to trim a known rolling moment. The rolling moment and sideforce due to sideslip derivatives were also measured. The tests showed that the method is sound, and provided that internal stowage of the rolling weights is possible, is capable of giving accurate results.

The value of $-\ell_\xi$ at $M = 0.82$, with wingtip weight canisters fitted is 0.14, compared with a value of 0.17 from wind tunnel tests on models with no canisters represented. The location of the wingtip weight canisters, immediately ahead of the ailerons, may have been partly responsible for the lower rolling moment due to aileron deflection measured in the flight tests.

The rolling moment due to sideslip derivative, $-\ell_v$, increases from 0.060 at $C_L = 0.19$ to 0.110 at $C_L = 0.47$, about 18% lower than corresponding wind tunnel measurements. The lower value of ℓ_v may be due to aerodynamic interference from the wingtip weight canisters. The analysis of the flight Dutch roll tests gave a value of $-\ell_v = 0.082$ at $C_L = 0.2$, some 30% higher than the present tests. Part of the difference may be due to the effects of oscillation frequency of the Dutch roll.

The sideforce due to sideslip derivative, $-y_v$, is constant at 0.22 between $C_L = 0.19$ and 0.47. The tunnel results, where comparable, are in good agreement with the wingtip weight tests, while the previous Dutch roll test results are about 25% lower, possibly due to the effects of oscillation frequency on the value measured in the Dutch roll tests.

LIST OF SYMBOLS

a_y reading of lateral accelerometer, g units

C_L lift coefficient $= \dfrac{\text{Lift}}{\frac{1}{2}\rho V^2 S}$

C_ℓ rolling moment coefficient $= \dfrac{\text{rolling moment}}{\rho V^2 S s}$

C_{ℓ_w} rolling moment coefficient due to wingtip weight $= \dfrac{L_w}{\rho V^2 S s}$

C_n yawing moment coefficient $= \dfrac{\text{yawing moment}}{\rho V^2 S s}$

C_Y sideforce coefficient $= \dfrac{\text{sideforce}}{\frac{1}{2}\rho V^2 S}$

L_w rolling moment applied by wingtip weight, pounds feet

N_ζ yawing moment due to rudder deflection, pounds feet per radian

p rate of roll, radians per second

r rate of yaw, radians per second

S wing area, square feet

s wing semi-span, feet

V_i equivalent air speed, knots

V true speed, feet per second

β sideslip angle, degrees

ζ rudder deflection from neutral, degrees

LIST OF SYMBOLS (Contd.)

ξ starboard aileron deflection relative to wing chord, degrees

$\Delta\zeta$ change in rudder angle to trim with wingtip weight fitted, degrees

$\Delta\xi$ change in aileron angle to trim with wingtip weight fitted, degrees

ϕ angle of bank, degrees

ρ air density, slugs per cubic foot

ℓ_ζ rolling moment due to rudder deflection derivative $\frac{\partial C_\ell}{\partial \zeta}$, per radian

ℓ_ξ rolling moment due to aileron deflection derivative $\frac{\partial C_\ell}{\partial \xi}$, per radian

ℓ_p rolling moment due to rate of roll derivative $\frac{\partial C_\ell}{\partial (ps/V)}$, per radian

ℓ_r rolling moment due to rate of yaw derivative $\frac{\partial C_\ell}{\partial (rs/V)}$, per radian

ℓ_v rolling moment due to sideslip derivative $\frac{\partial C_\ell}{\partial \beta}$, per radian

n_ζ yawing moment due to rudder deflection derivative $\frac{\partial C_n}{\partial \zeta}$, per radian

y_ζ sideforce due to rudder deflection derivative $\frac{1}{2}\frac{\partial C_Y}{\partial \zeta}$, per radian

y_ξ sideforce due to aileron deflection derivative $\frac{1}{2}\frac{\partial C_Y}{\partial \xi}$, per radian

y_v sideforce due to sideslip derivative $\frac{1}{2}\frac{\partial C_Y}{\partial \beta}$, per radian

N.B. ALL FORCES AND MOMENTS ARE REFERRED TO STABILITY AXES

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TABLE 1

Fairey Delta 2 - principal dimensions

Wing	
Gross area	360 square feet
Span	26.83 feet
Centre line chord	25 feet
Tip chord	1.83 feet
Mean aerodynamic chord	16.75 feet
Leading edge sweep	60 degrees
Dihedral	0 degrees
Twist	0 degrees
Wing body angle	+1.5 degrees
Aileron	
Total area (each)	16.61 square feet
Area forward of hinge line (each)	0.57 square feet
Nominal rigged up angle	3 degrees
Range of movement	±17 degrees relative to rigged up angle
All up weight at take-off, with empty weight canisters, and 2500 lb fuel	14,430 lb
Centre of gravity position (1000 lb of fuel gone)	54% centre line chord or 31.5% mean aerodynamic chord
Moment arm of wingtip weight canister	11.7 feet
Weight of empty canister	65 lb
Length of canister	4.0 feet
Diameter of canister	0.96 feet

Printed in England for Her Majesty's Stationery Office by the Royal Aircraft Establishment, Farnborough. W.T.59.K4.

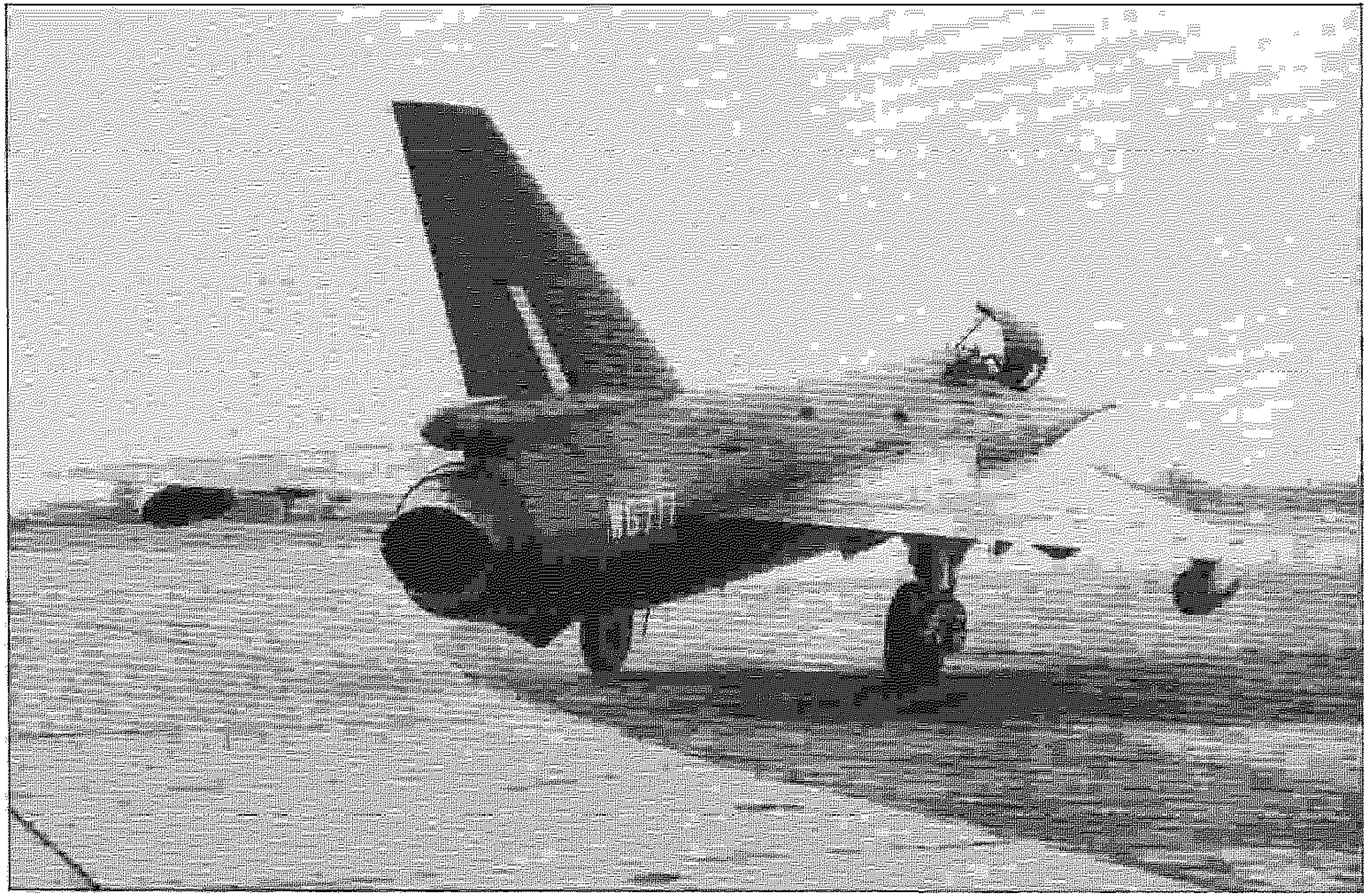


FIG. I. FAIREY DELTA 2 WITH WINGTIP WEIGHT CANISTERS FITTED

FIG.2. FAIREY DELTA 2 SHOWING STARBOARD WINGTIP WEIGHT CANISTER

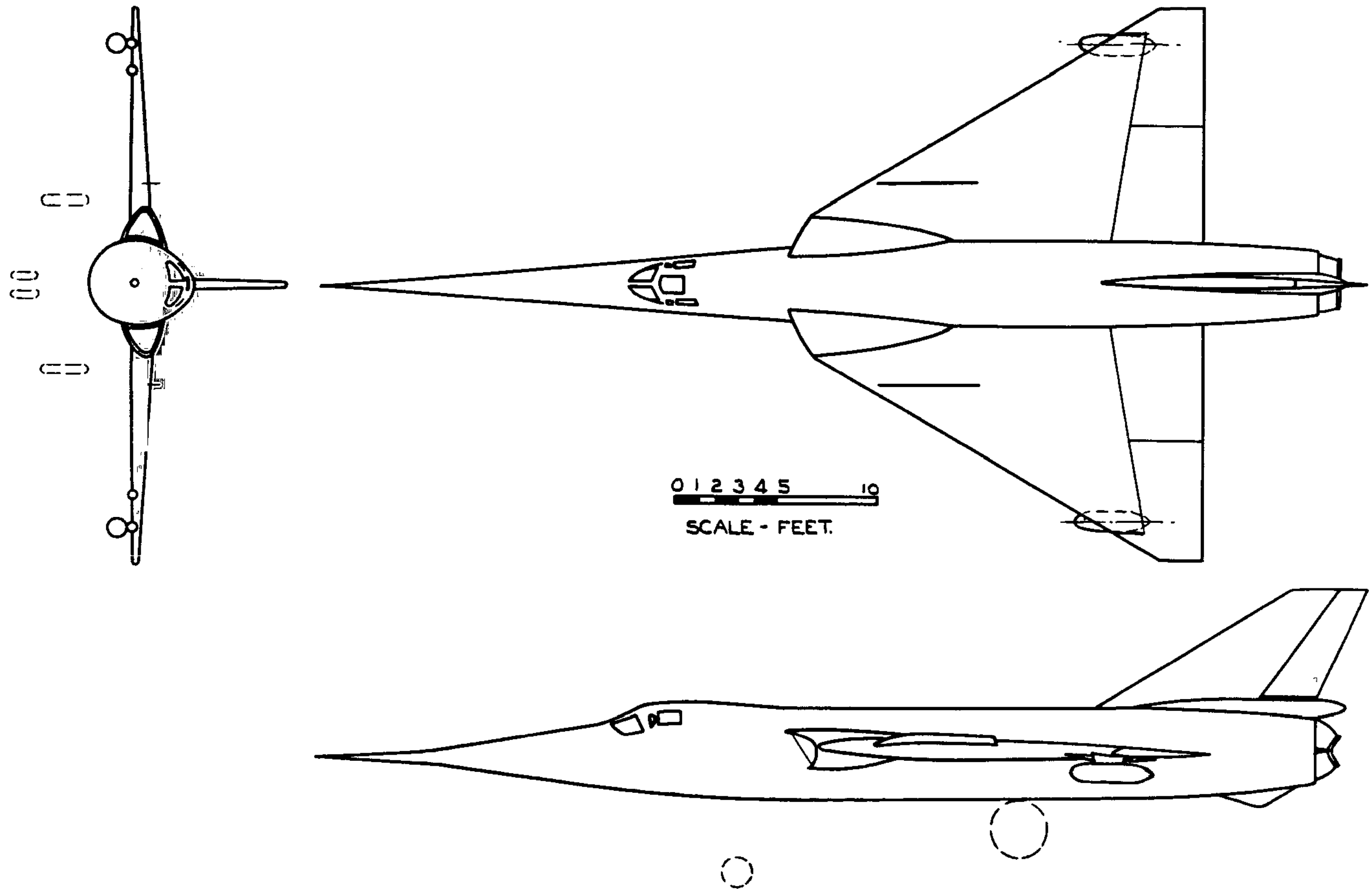


FIG. 3. GENERAL ARRANGEMENT OF FAIREY DELTA 2 IN TEST CONFIGURATION.

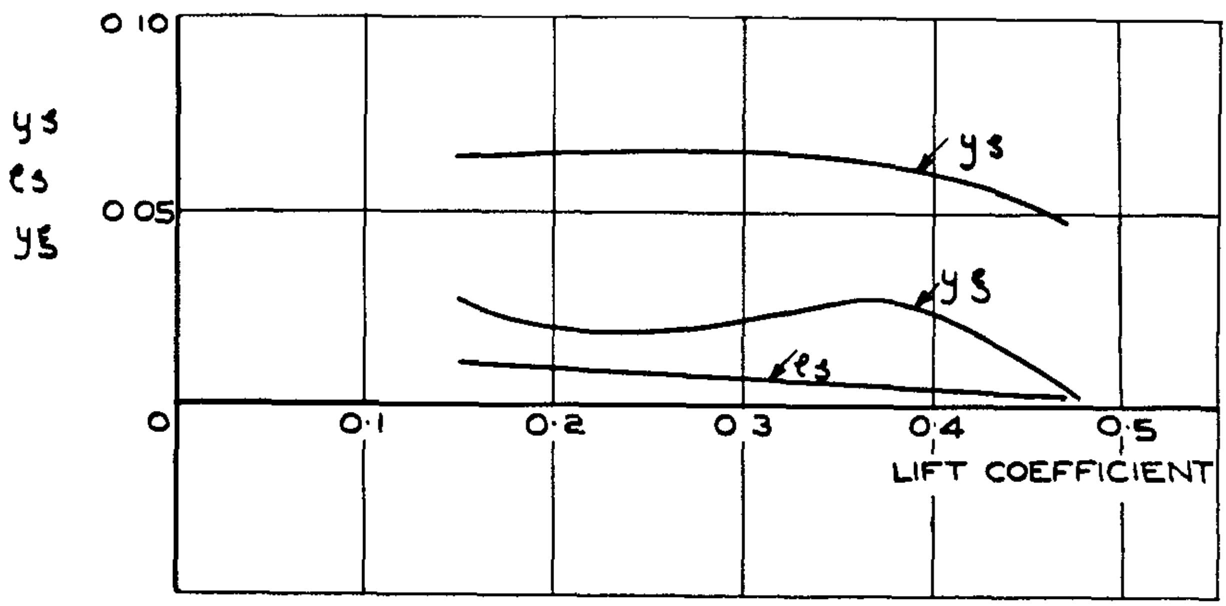
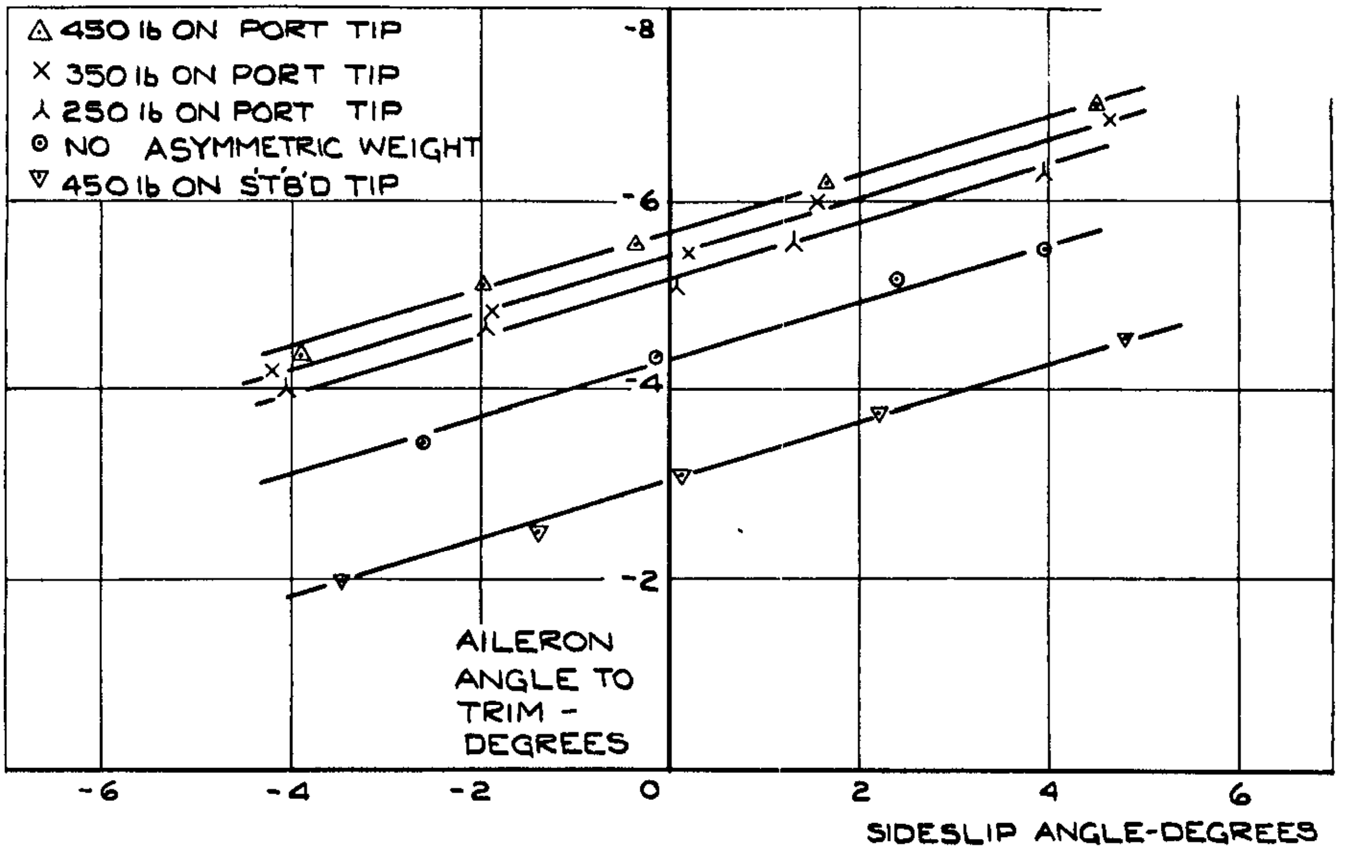
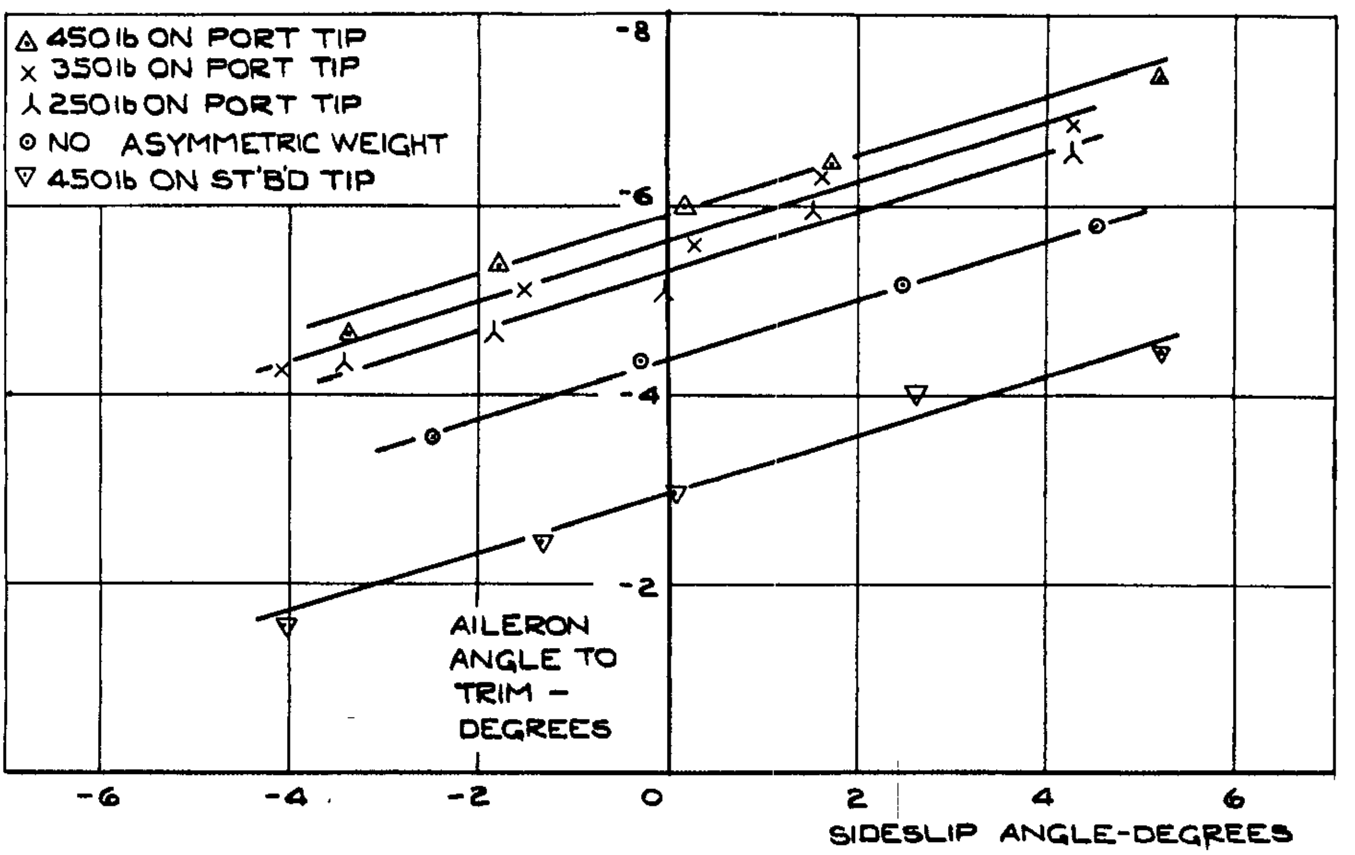


FIG.4. ASSUMED VALUES OF CONTROL DERIVATIVES y_ζ, l_ζ AND y_ξ DERIVED FROM WIND TUNNEL TESTS (REF. 5) AT M=0·85.

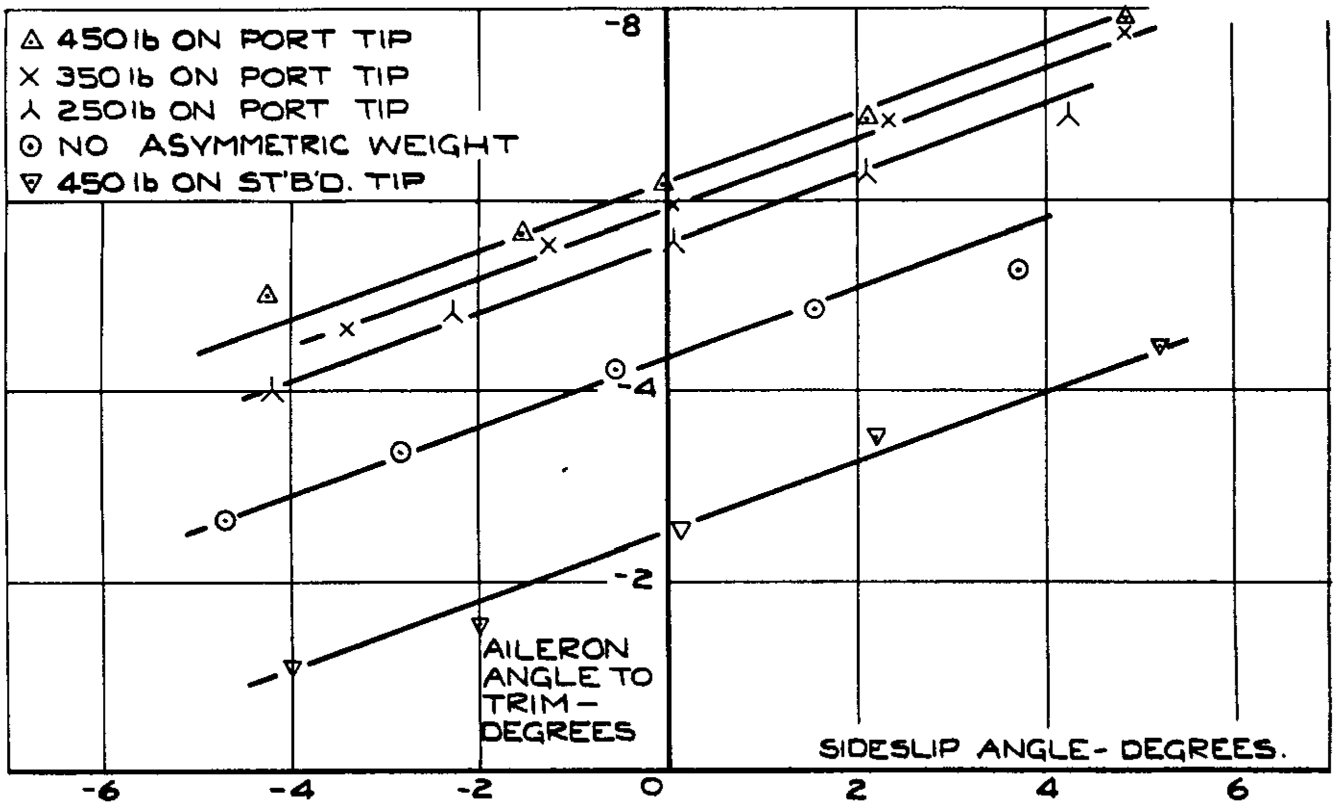


(a) E.A.S.=235 KNOTS.

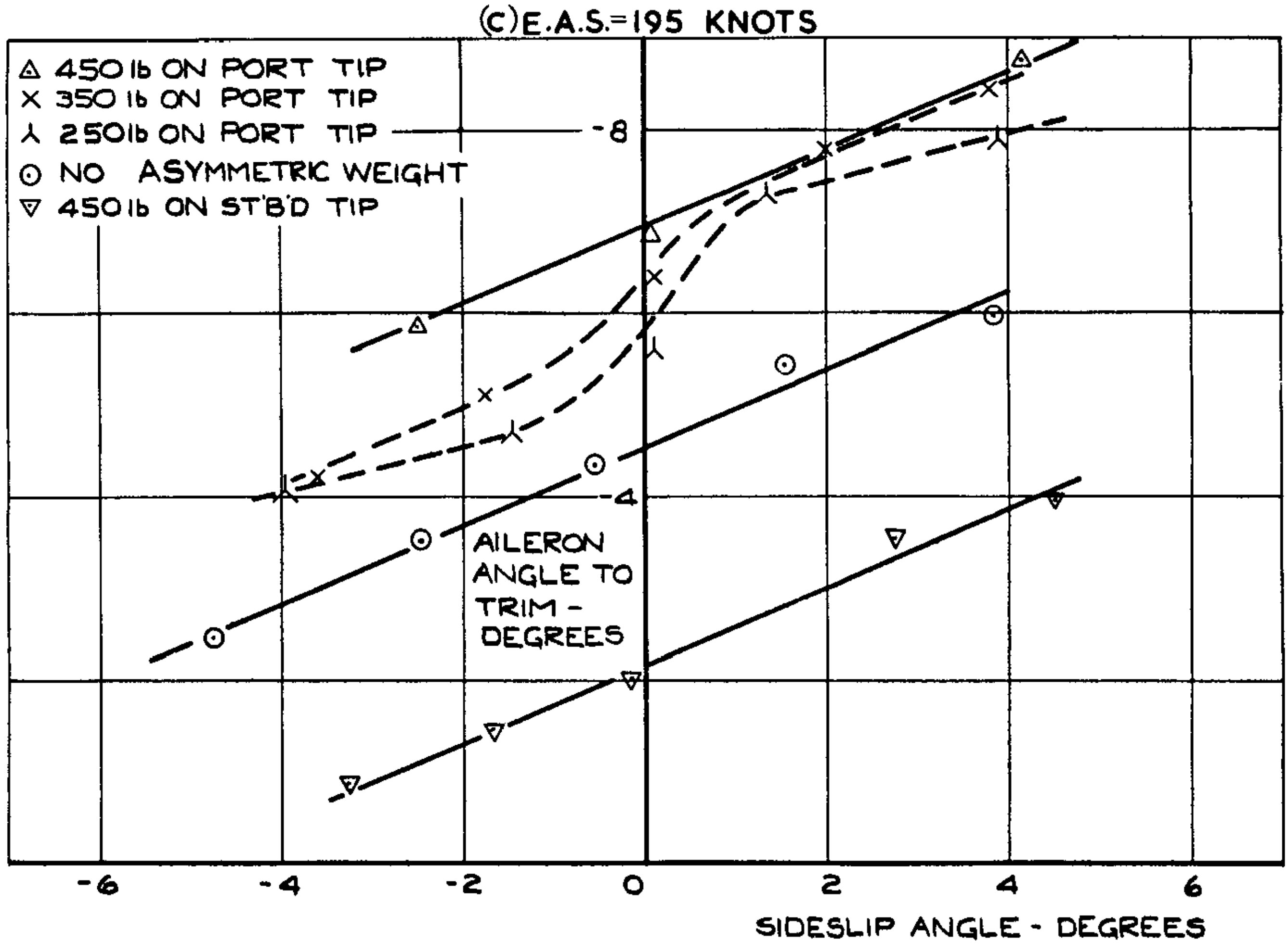


(b) E.A.S.=215 KNOTS

FIG.5. AILERON ANGLE TO TRIM STRAIGHT SIDESLIPS WITH AND WITHOUT WINGTIP WEIGHT. 40,000 FEET.

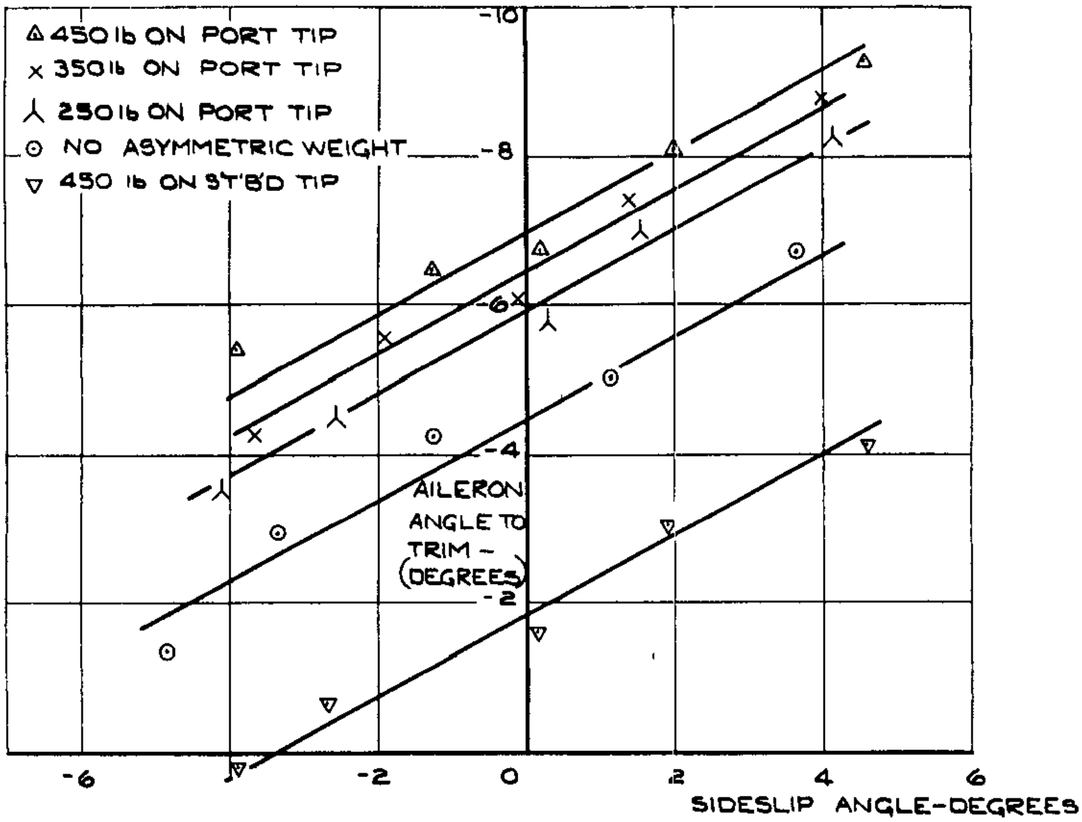


(c) E.A.S.=195 KNOTS

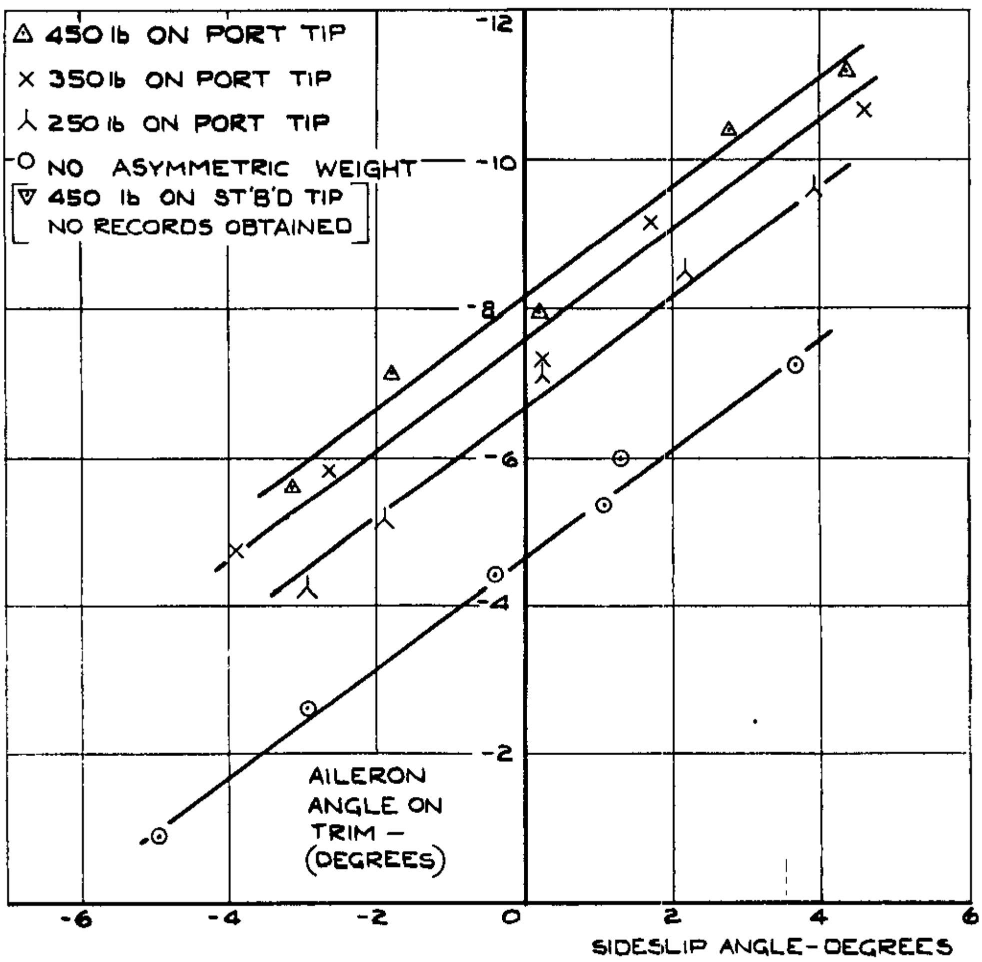


(d) E.A.S.=172 KNOTS

FIG. 5. (CONT'D) AILERON ANGLE TO TRIM STRAIGHT SIDESLIPS WITH AND WITHOUT WINGTIP WEIGHT. 40,000 FEET.

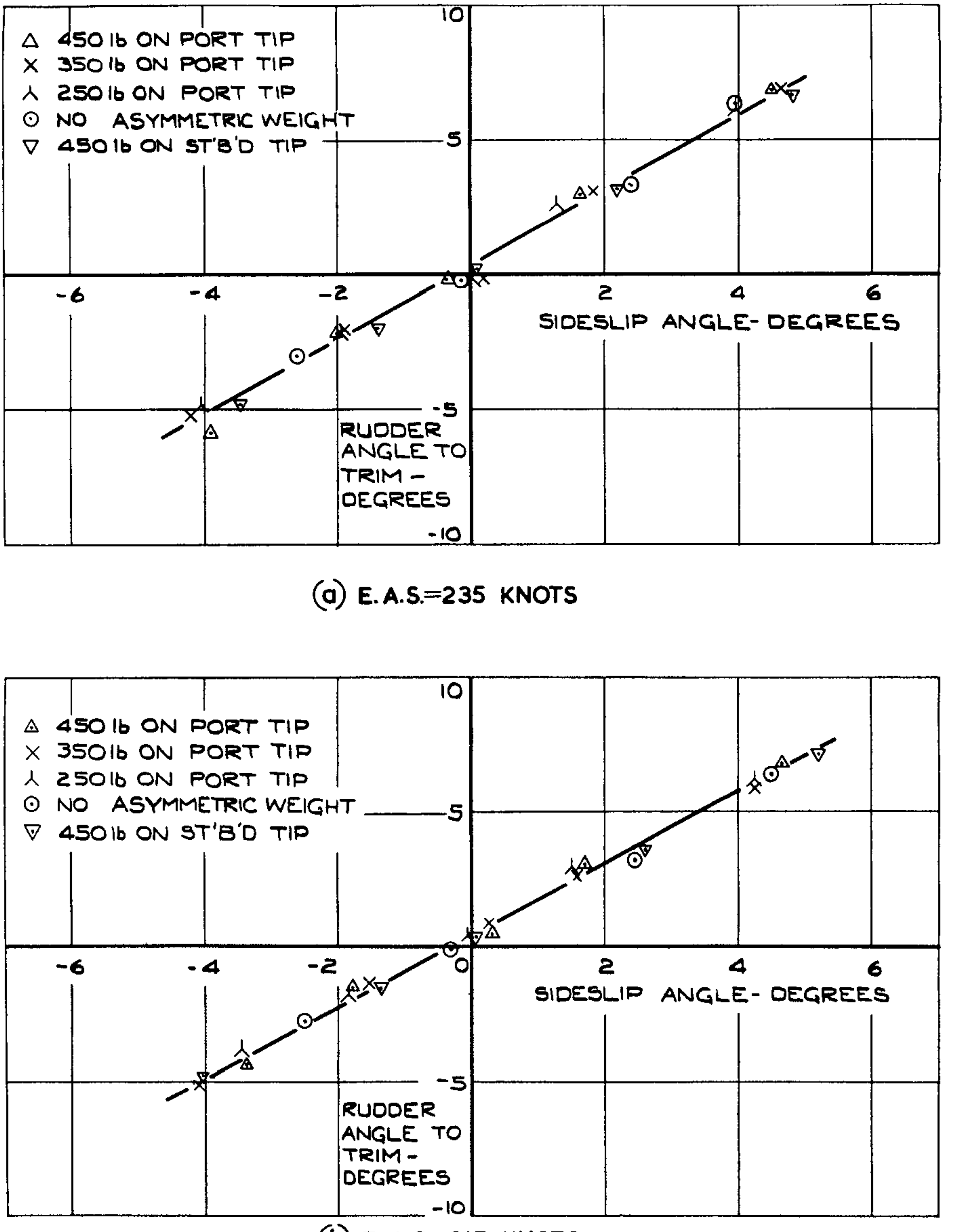


(e) E A.S.=175 KNOTS



(f) E A.S.=150 KNOTS

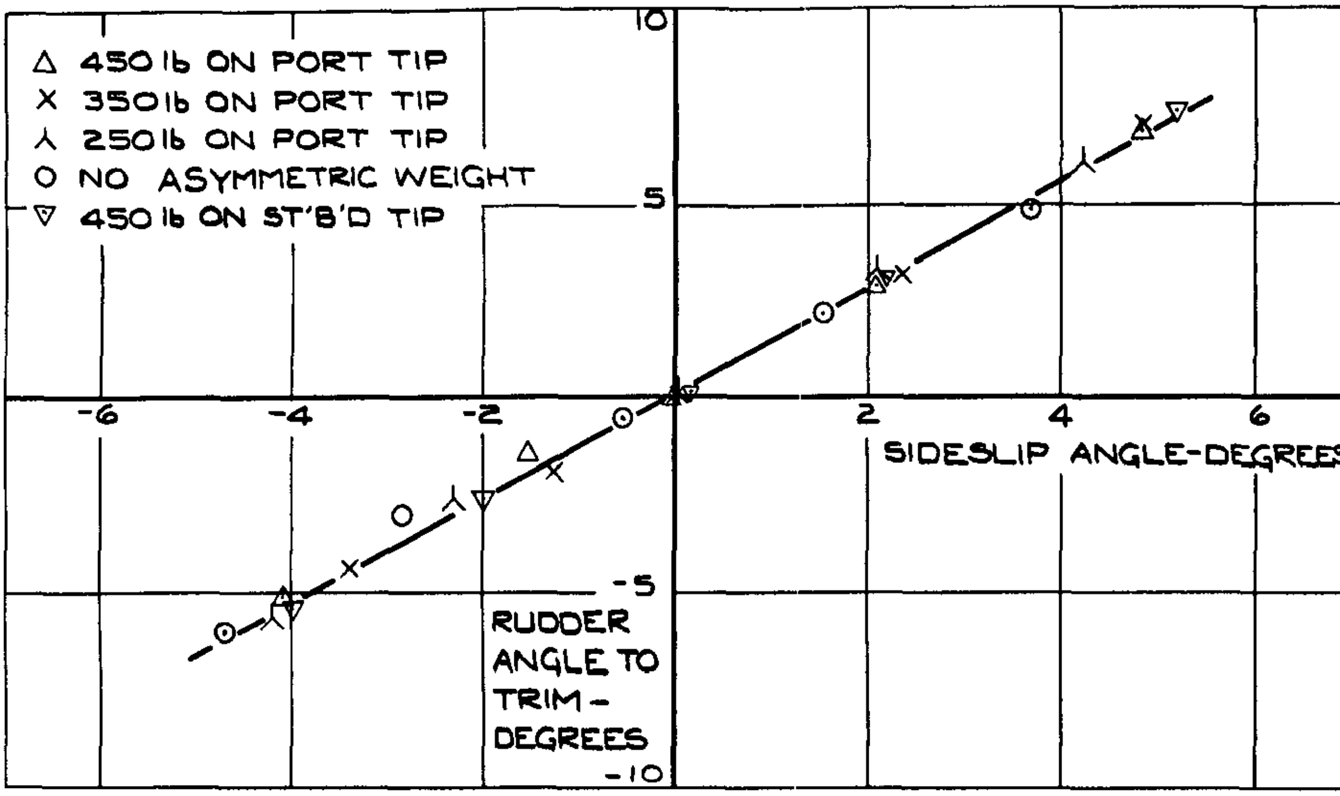
FIG.5 (CONCL'D) AILERON ANGLE TO TRIM STRAIGHT SIDESLIPS WITH AND WITHOUT WINGTIP WEIGHT. 20,000'



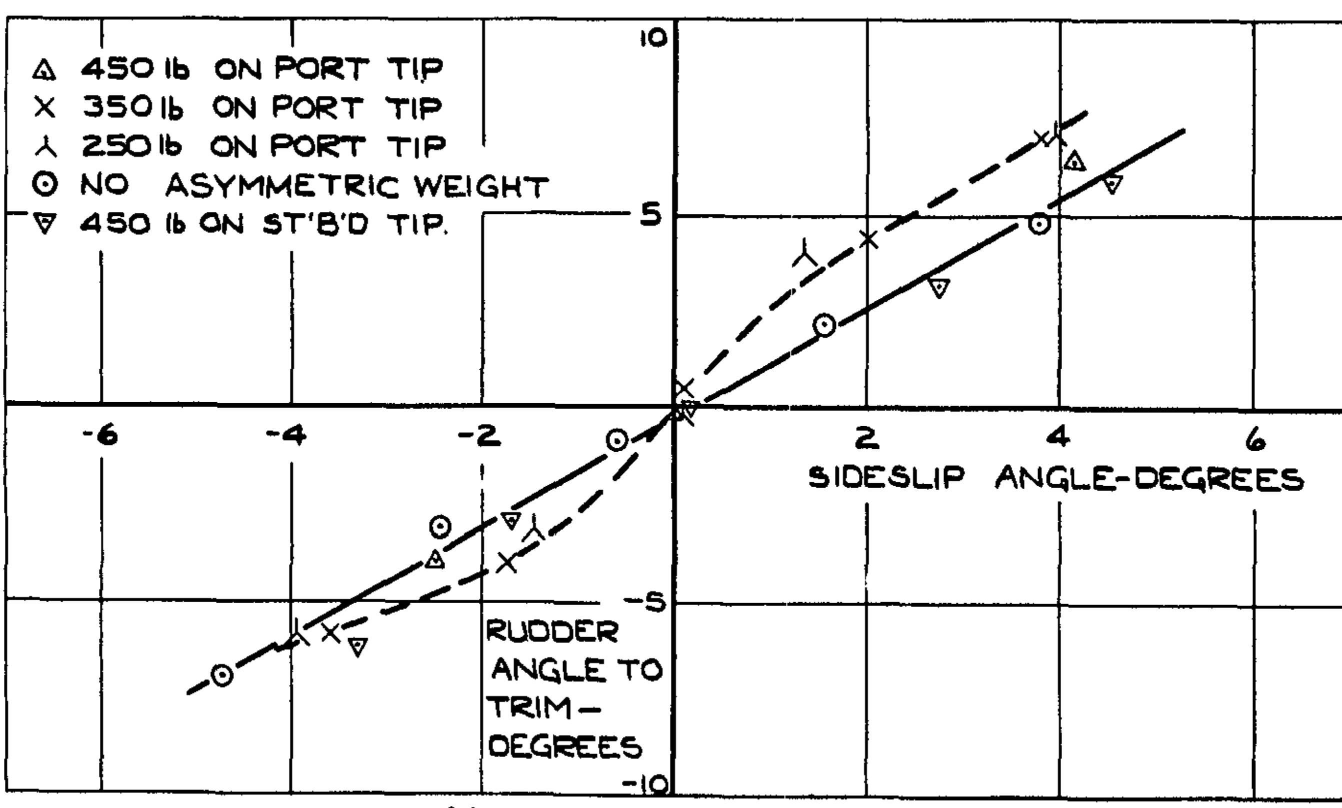
(a) E.A.S.=235 KNOTS

(b) E.A.S.=215 KNOTS

FIG. 6. RUDDER ANGLE TO TRIM STRAIGHT SIDESLIPS WITH AND WITHOUT WINGTIP WEIGHT. 40,000 FEET.

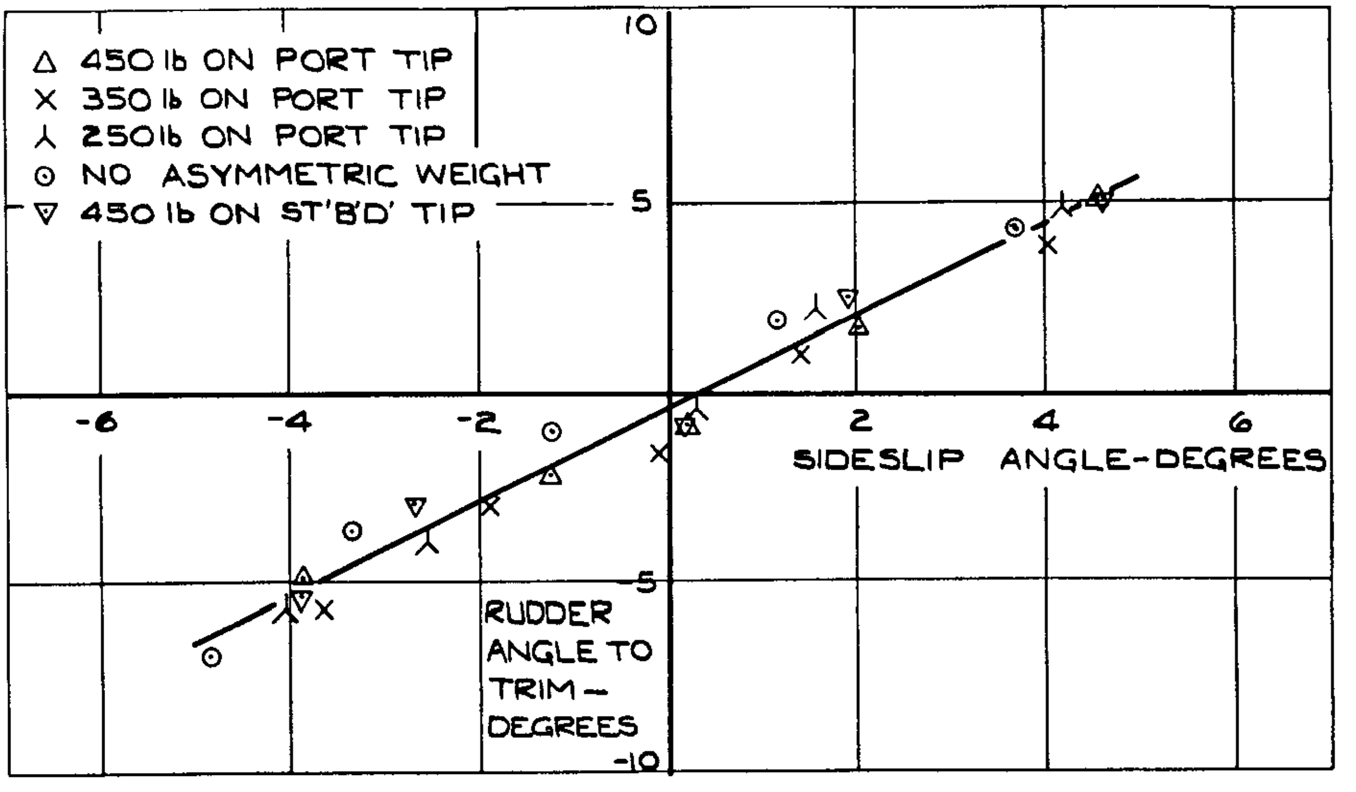


(c) E.A.S.=195 KNOTS.

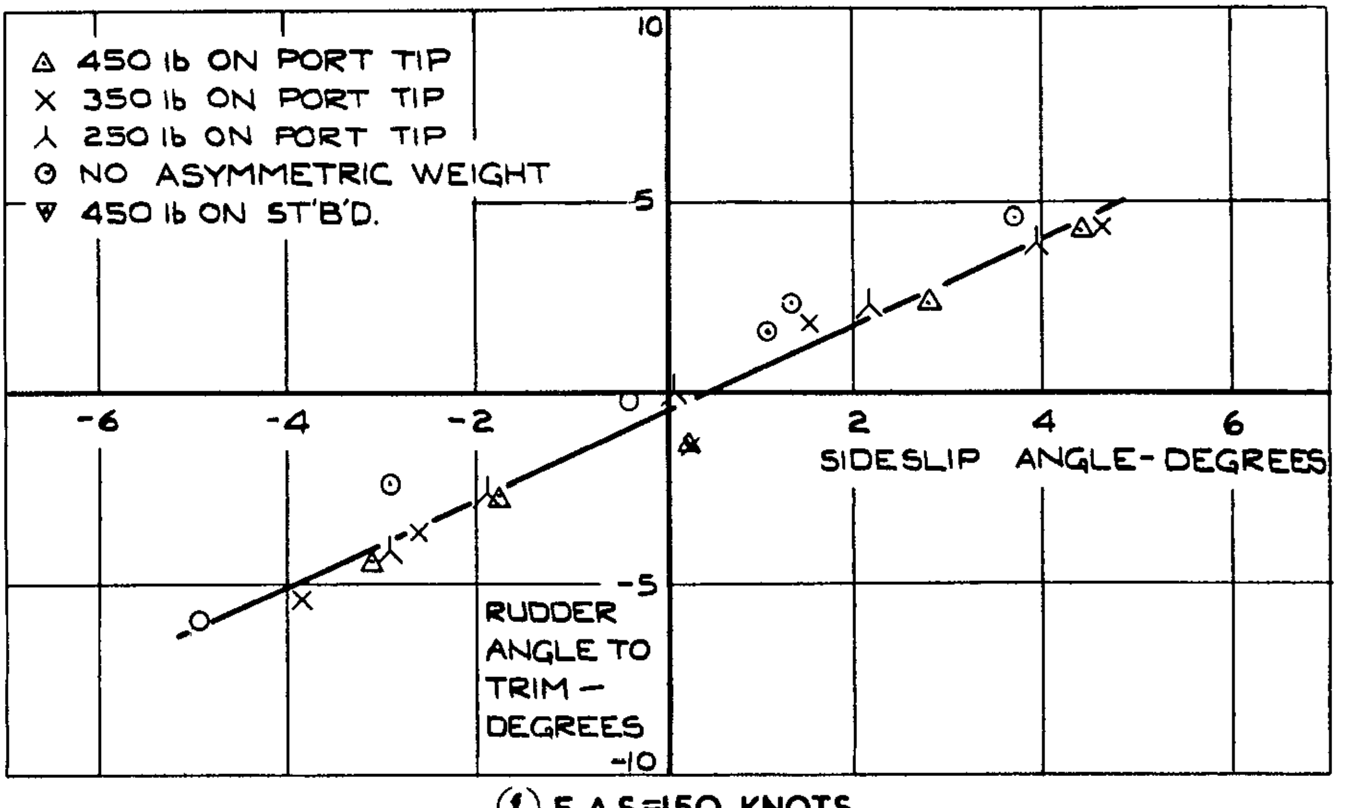


(d) E.A.S.=172 KNOTS.

FIG. 6. (CONT'D) RUDDER ANGLE TO TRIM STRAIGHT SIDESLIPS WITH AND WITHOUT WINGTIP WEIGHT. 40,000'

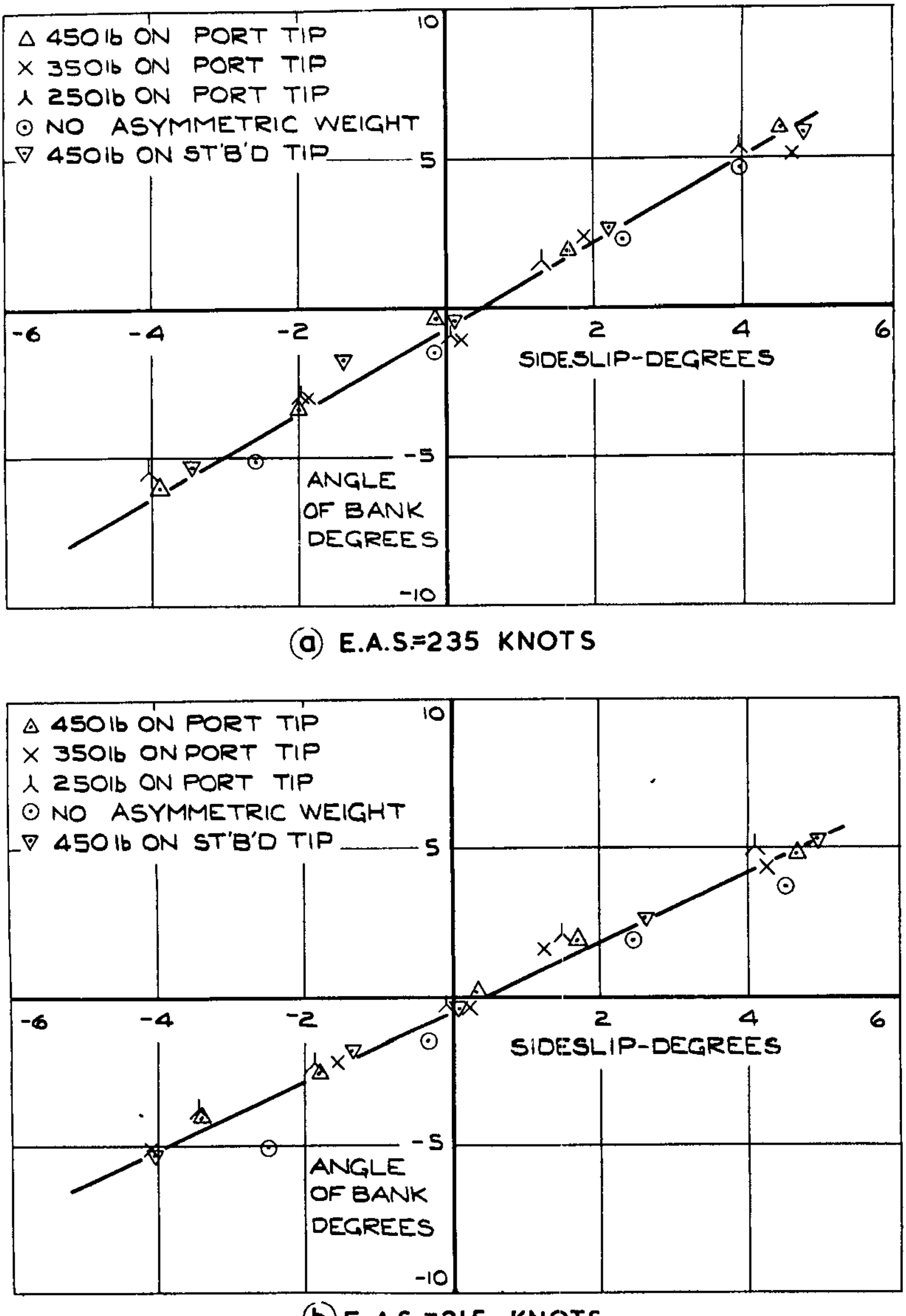


(e) E.A.S.=175 KNOTS.



(f) E.A.S.=150 KNOTS.

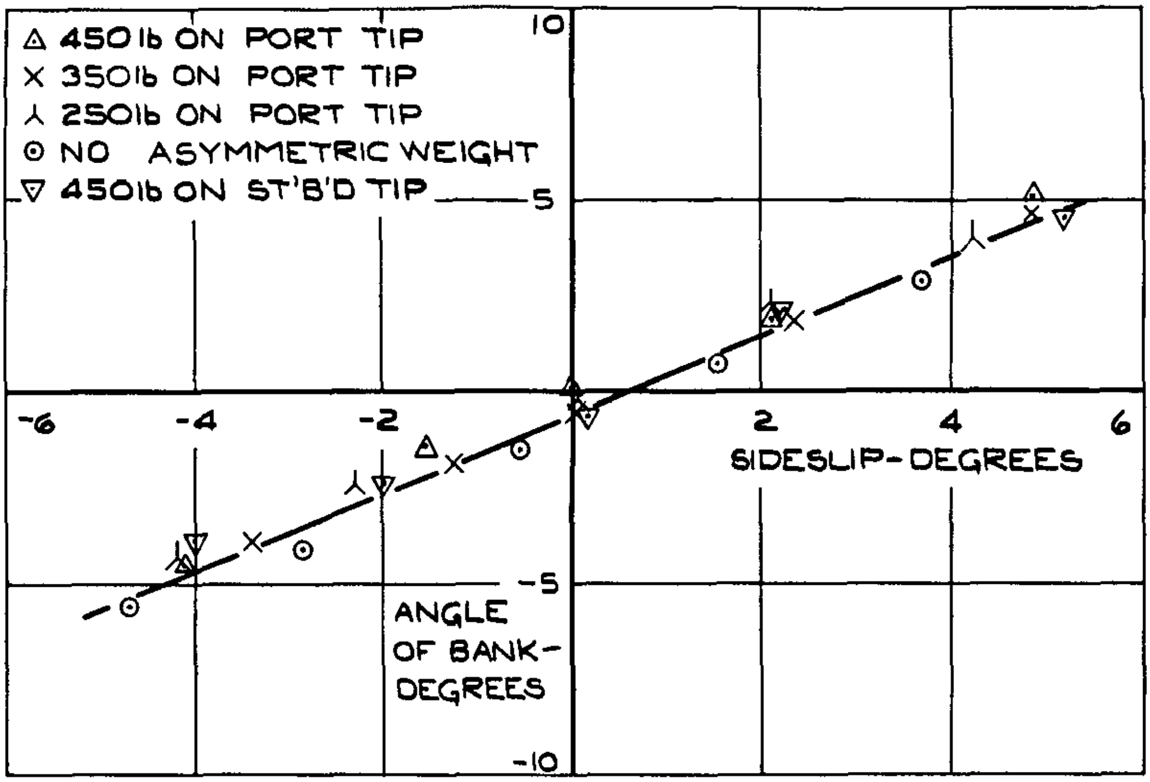
FIG. 6. (CONCLUDED) RUDDER ANGLE TO TRIM STRAIGHT SIDESLIPS WITH AND WITHOUT WINGTIP WEIGHT. 20,000'.



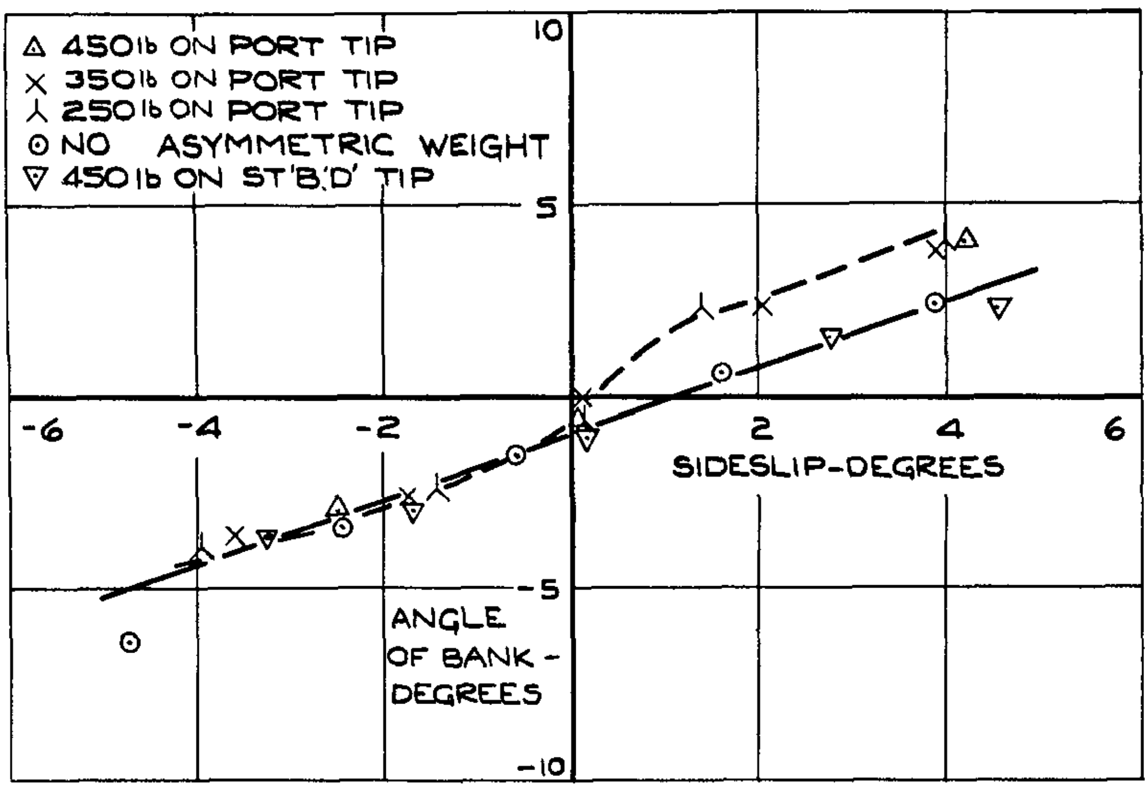
(a) E.A.S.=235 KNOTS

(b) E.A.S.=215 KNOTS

FIG.7. VARIATION OF ANGLE OF BANK WITH SIDESLIP. 40,000'.

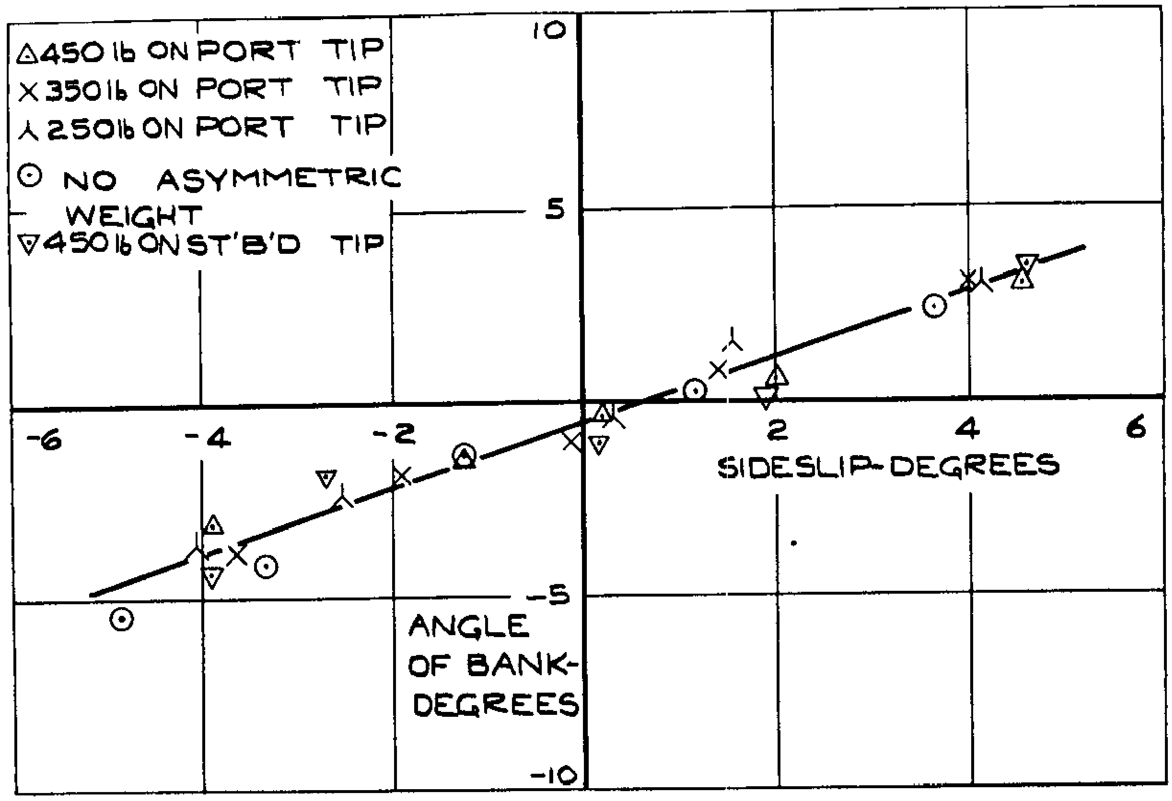


(c) E.A.S.=195 KNOTS

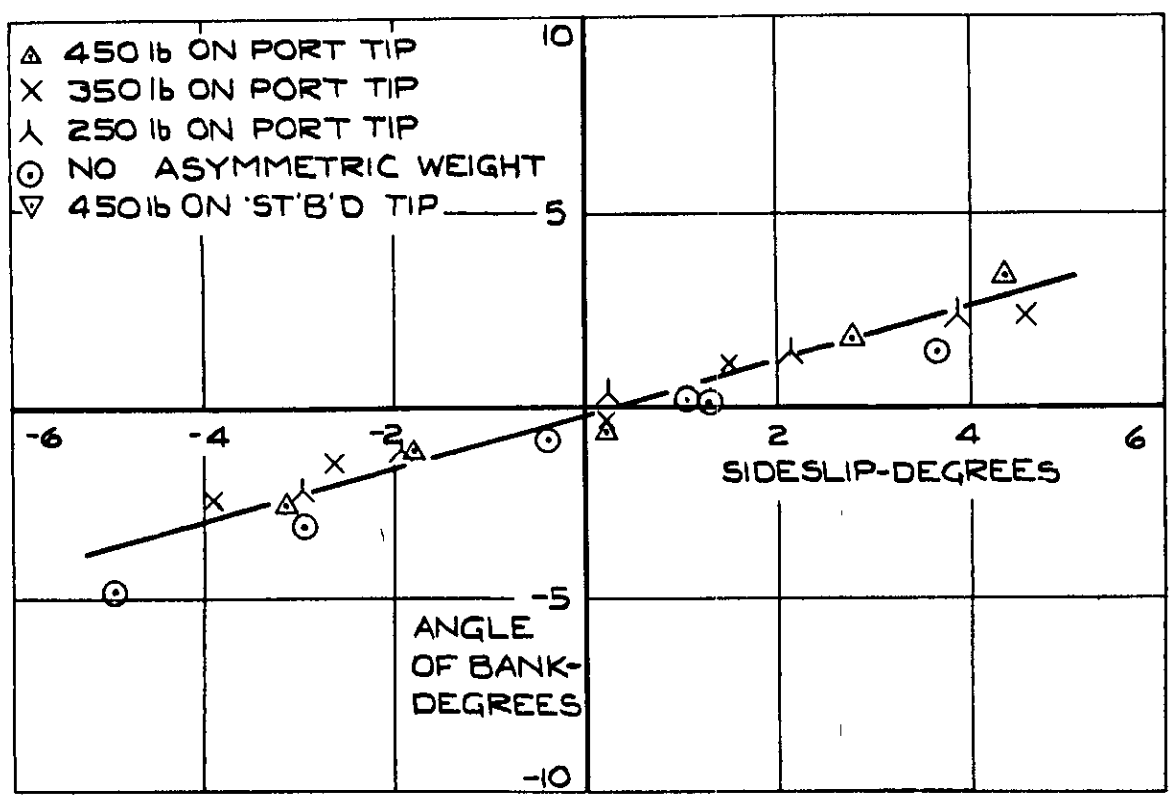


(d) E.A.S.-172 KNOTS

FIG.7.(CONT'D) VARIATION OF ANGLE OF BANK WITH SIDESLIP. 40,000'



(e) E.A.S.=175 KNOTS



(f) E.A.S.=150 KNOTS

FIG.7(CONCL'D) VARIATION OF ANGLE OF BANK WITH SIDESLIP. 20,000'.

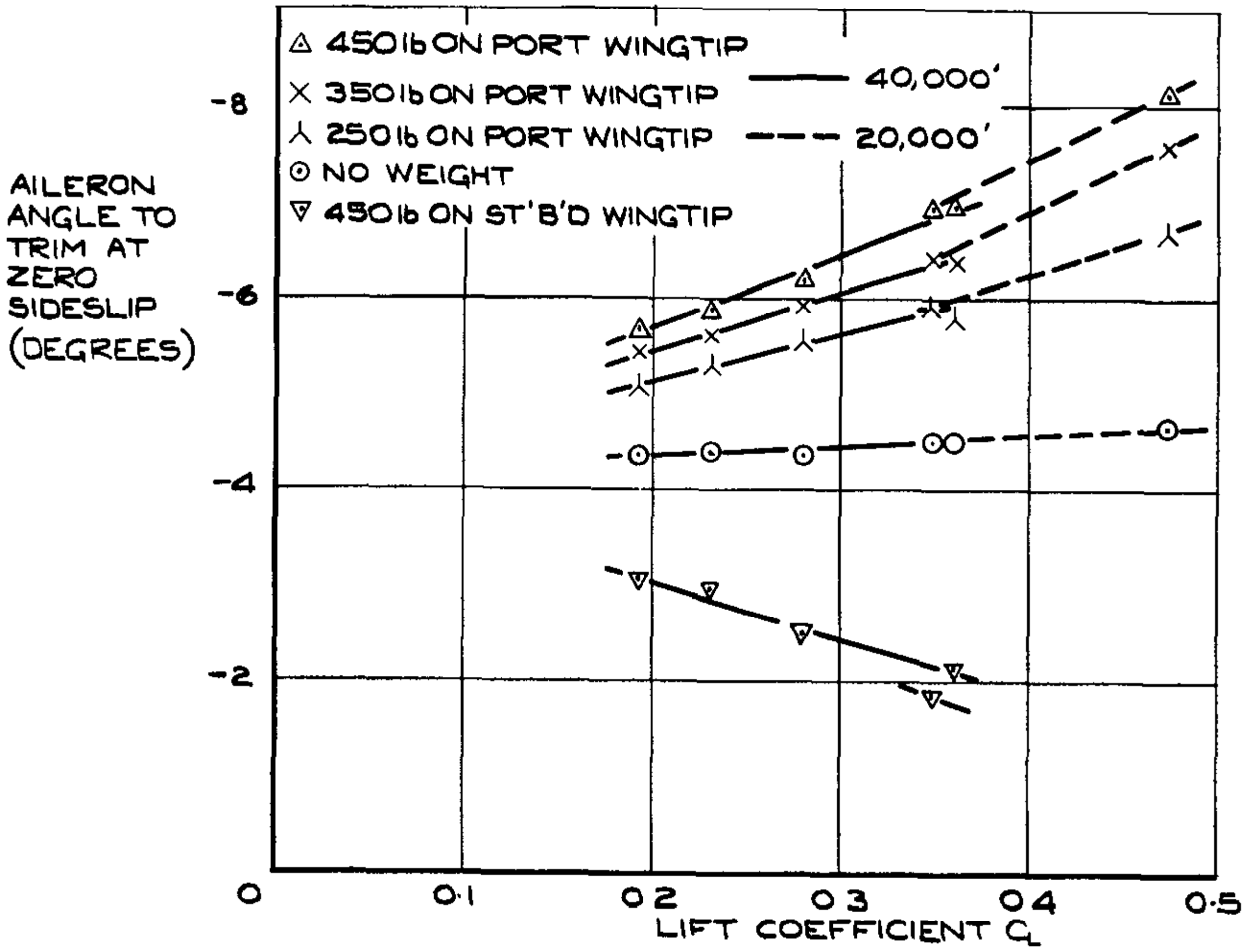


FIG.8. AILERON ANGLE TO TRIM ASYMMETRIC WEIGHT AT ZERO SIDESLIP. 40,000 AND 20,000 FEET.

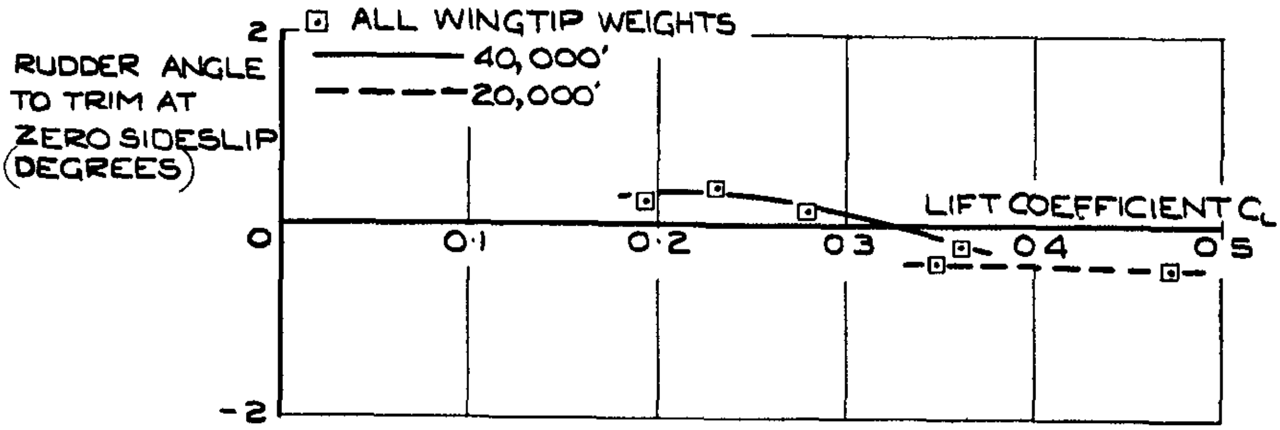


FIG.9. RUDDER ANGLE TO TRIM ASYMMETRIC WEIGHT AT ZERO SIDESLIP. 40,000 AND 20,000 FEET.

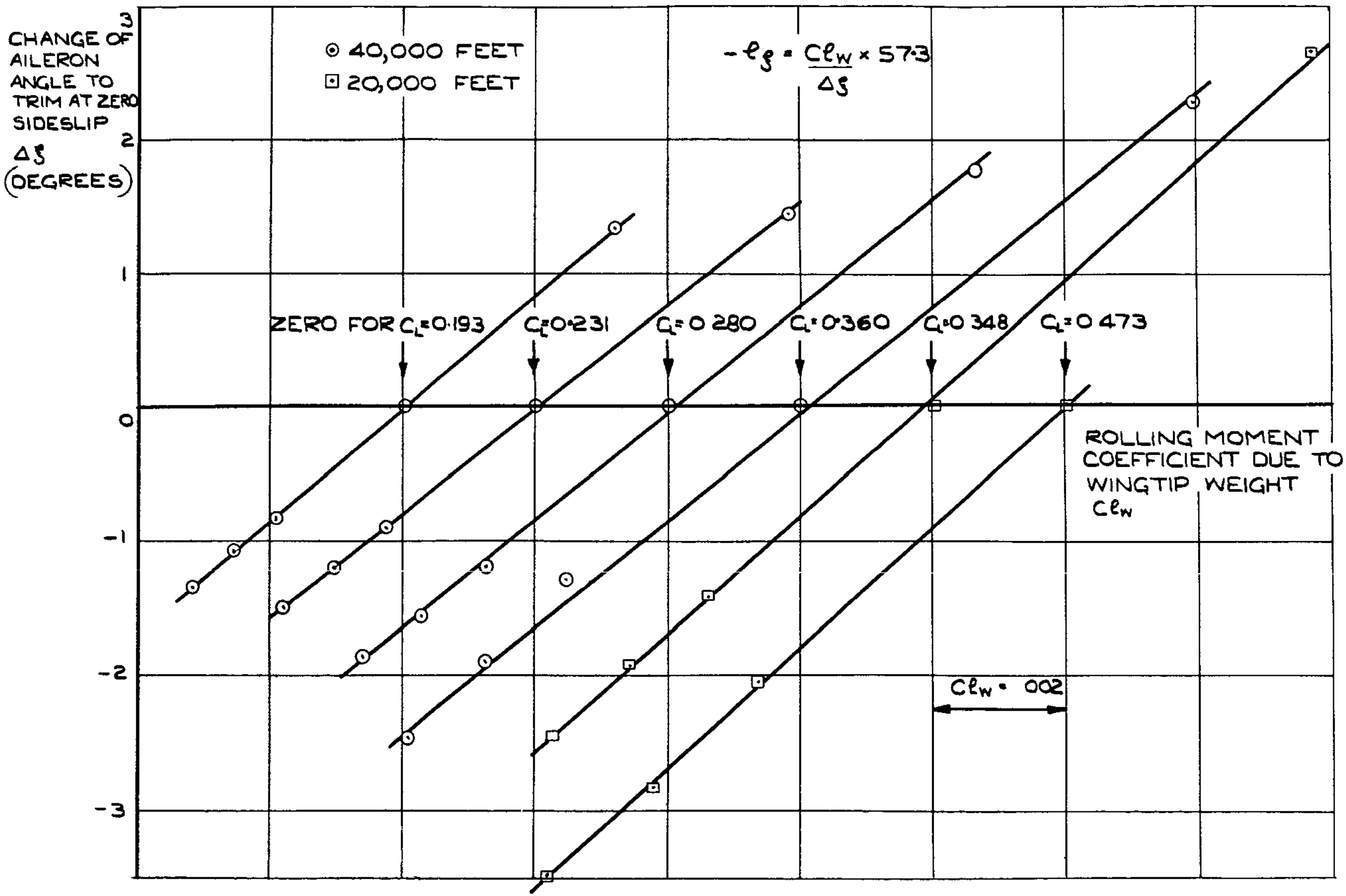


FIG. 10. CHANGE OF AILERON ANGLE REQUIRED TO TRIM ASYMMETRIC WEIGHT AT ZERO SIDESLIP.

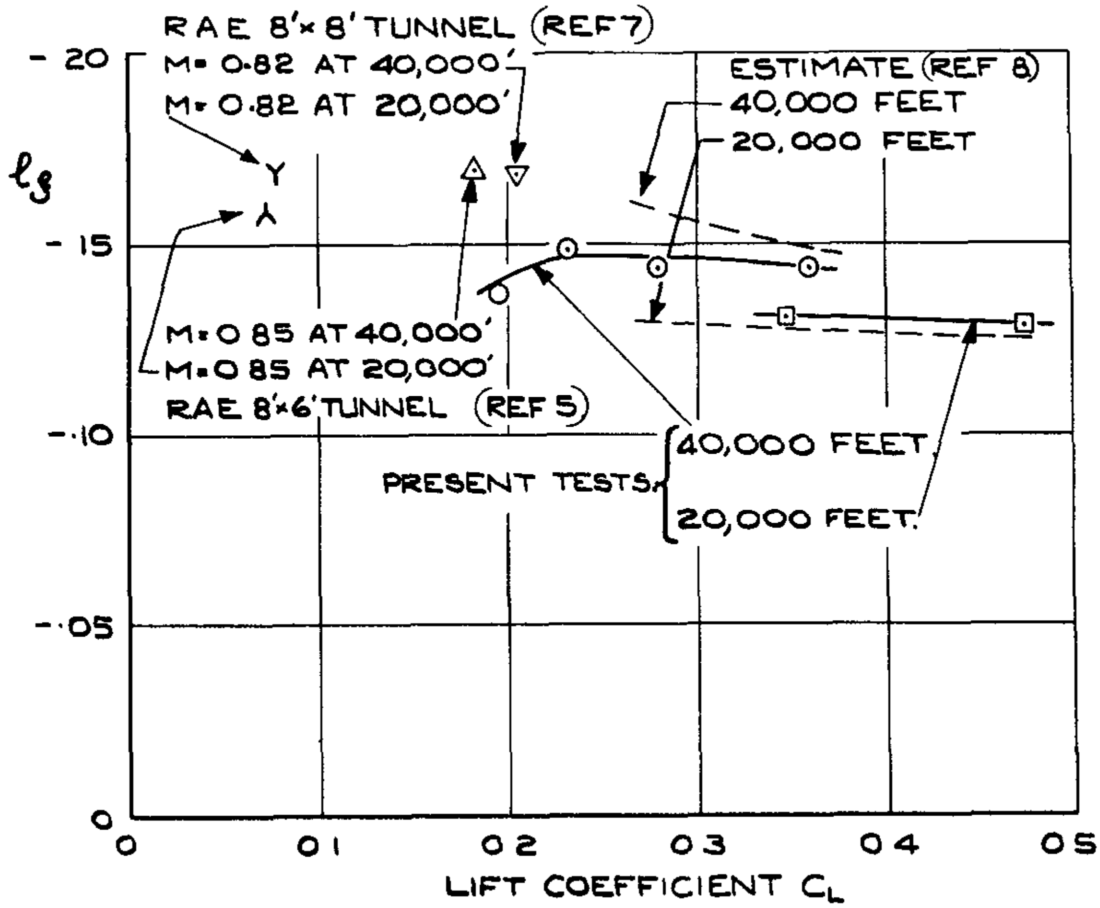


FIG. 11. AILERON ROLLING POWER, l_ξ.

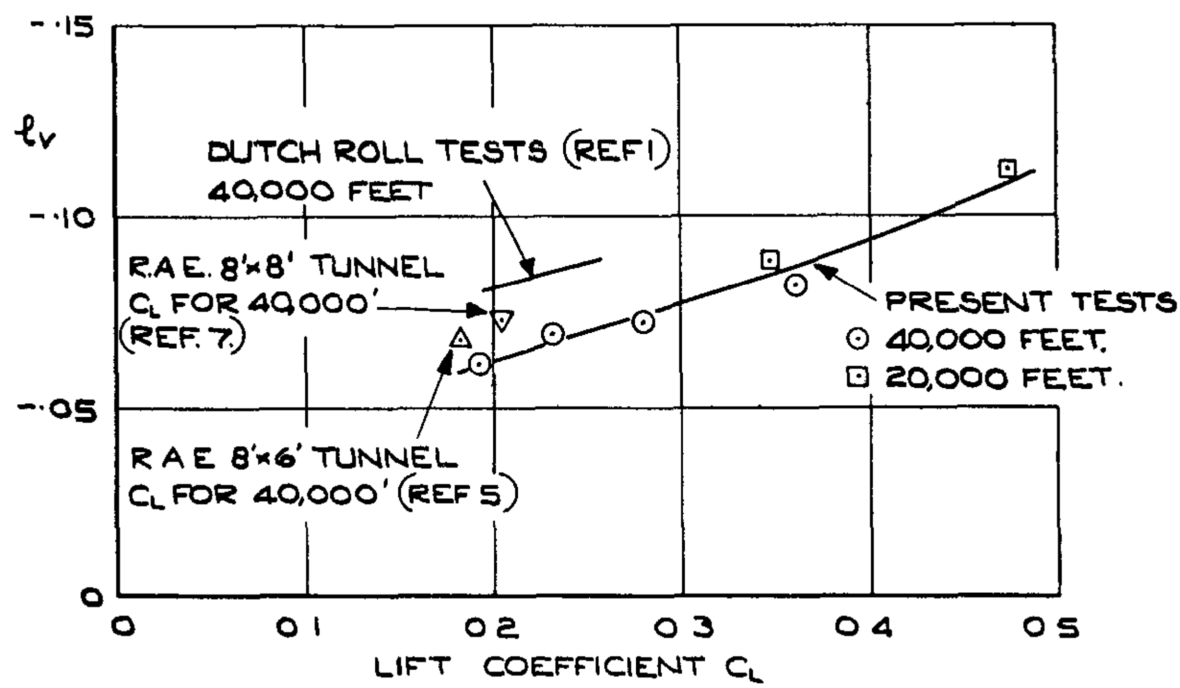


FIG. 12. ROLLING MOMENT DUE TO SIDESLIP DERIVATIVE, l_v.

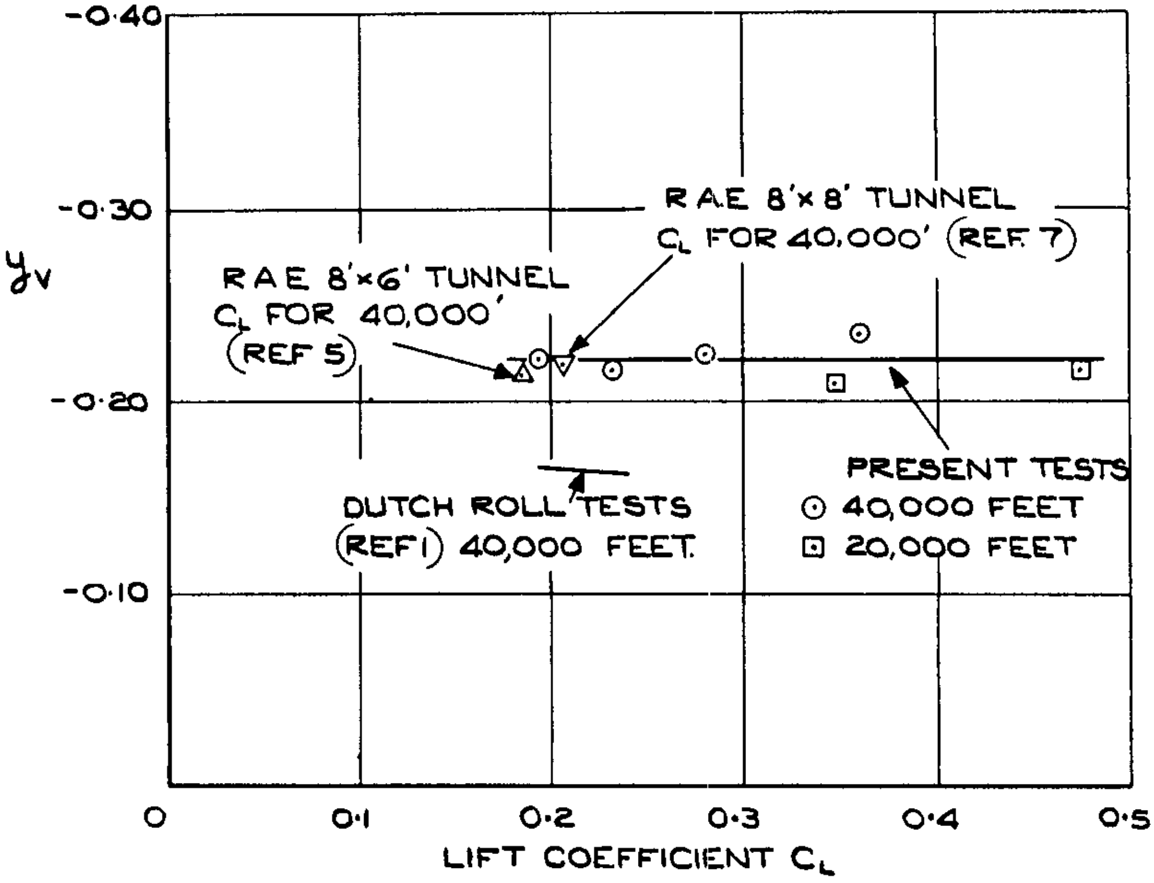


FIG. 13. SIDEFORCE DUE TO SIDESLIP DERIVATIVE y_v.

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AI(42) Fairey Delta 2
533.652.1 :
533.6.013.413 :
533.6.013.417 :
533.6.011.34

FLIGHT MEASUREMENTS AT SUBSONIC SPEEDS OF THE AILERON ROLLING POWER AND LATERAL STABILITY DERIVATIVES ℓ_v AND y_v ON A 60 DEGREE DELTA WING AIRCRAFT (FAIREY DELTA 2). Dee, F.W. June, 1963.

The aileron rolling power of the Fairey Delta 2 has been measured at subsonic speeds, by a method using asymmetric wingtip weights. The lateral stability derivatives ℓ_v and y_v have also been determined from measurements in steady straight sideslips.

The results have been compared with wind tunnel measurements and some differences found, the tunnel value of $-\ell_\xi$, for example, being about 20% higher than that measured in flight.

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The differences could be
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The differences could be partially due to aerodynamic interference in the flight tests from the externally mounted wingtip weights. It is hoped to make further tunnel tests with these represented.

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