C P No 17
12058
A R C Technical Report

MINISTRY OF SUPPLY

AERONAUTICAL RESEARCH COUNCIL
CURRENT PAPERS

Wind Tunnel Tests on a One-Twelfth Scale Model of a Twin-Engined Military Transport

(Airspeed C 13/45 Ayrshire)

By

R WARDEN, Ph D, M Eng
of the Aerodynamics Division, N P L

LONDON HIS MAJESTY'S STATIONERY OFFICE

1950

Price 3s 6d. net.

Wind Tunnel Tests on a One-twelfth Scale Model of a Twin-engined Military Transport. (Airspeed C 13/45 Ayrshire).

- By -
R. Warden, Ph.D., M.Eng.,
of the Aerodynamics Division, N.P.L.

Introduction and Summary.

13th January, 1949

This report gives the results of wind tunnel tests on a one-twelfth scale model of the A.S.60 - a high wing transport machine having twin engines located in large underslung nacelles. The wing body interference and longitudinal stability were measured. The stability tests included measurements with propellers running.

The results indicate a noticeable fuselage interference effect on the tail and that slipstream has an appreciable destabilising effect under climb conditions.

A comparison of tests made on pitching moment at the R.A.E., and N.P.L., is included.

Details of Tests.

The tests were made in the Duplex wind tunnel at a wind speed of 60 ft. per sec., the equivalent Reynolds number being 0.334×10^6. The main aerodynamic details of the machine are tabulated in Table 1 and a general arrangement is shown in Figure 1.

The first series of tests comprised those without propellers and included tests on wing and fuselage separately, and on various combinations of wing, fuselage, nacelles and empennage. For the tests with propellers running a new wing with the nacelles and part of the fuselage integral with it had to be built to allow the installation of the model propeller drive. This consisted of a single motor, fitted in the fuselage, drawing the propellers through shafts and bevel gears buried in the wing and nacelles.

Tests on Wing Alone.

As this was the first low drag wing to be tested in the Duplex wind tunnel, it was decided to explore the boundary-layer flow by means of the "China clay" and "lead acetate - H_2S" techniques.

The/

Published by permission of the Director, N.P.L.

The main results of these tests are shown in Figure 2. The shaded areas indicate the rear high surface-friction regions as indicated by the "china clay" technique. The boundary line thus determined was so far back from the leading edge that laminar separation was suspected, particularly at low lifts. An exploration using the "lead acetate - H_2S" technique showed that at a C_L of 0.19 laminar separation occurred on the upper surface of the wing at about 0.6 of the local chord from the leading edge. At a C_L of 0.59 the laminar separation could be detected only near the middle of the wing. Careful exploration of the lower surface failed to reveal any signs of laminar separation over the range of incidence tested. It will be observed that at a C_L of 0.92 ($\alpha = 10°.2$) the lower surface is clear of any rear high friction region. At this C_L the outer parts of the upper surface of the wing are completely turbulent, but there is a low friction region at the trailing edge, between 0.25 and 0.5 of the semi-span from the centre line. This is due probably to a turbulent breakaway of the flow.

In all cases there was a tendency for inward flow on the upper surface which became more pronounced as the lift was increased. After consideration of the above results it was decided to fit a 0".020 diameter wire, at 50% of the local chord, on the upper surface only of the wing. All subsequent chemical explorations indicated that transition occurred at the wire. At 12° incidence (C_L approximately 1.0) the flow behind the wire was very disturbed and definite indications of reversed surface flow were obtained over the outer 25% of the span. The approach to this reversed flow had been noted at 10° incidence ($C_L = 0.9$) in the form of a very strong inward flow, almost parallel to the trailing edge, over the outer 20% of the span.

The effect of the wire on the forces measured on the wing alone is very small. On the straight part of the lift curve its effect is equivalent to a change of about 0°.2 in incidence and it has no effect on the value of $dC_L/d\alpha$. Stalling is sharper with the wire than without but occurs at about the same angle. The effect on drag is negligible. The pitching moment is increased by fitting the wire, the increase being roughly equivalent to a forward shift of 0.05c̄ in the centre of pressure between no lift and the initial stall. These results are shown plotted in Figure 3, which also includes the full scale lift against incidence curve as estimated by the firm. The slopes of the model and full scale lift curves are in close agreement, being 0.0925 C_L/degree model scale against an estimated value of 0.0955 C_L/degree full scale. There is a difference of about 0°.5 between the model and estimated full scale "no lift" angles of incidence.

Flow over Nacelles.

Streamer explorations of the flow over the nacelles and adjacent parts of the wing were carried out both with and without the fuselage in position.

With the original design a breakaway began at the nacelle-wing lower surface junction some eight inches ahead of the wing trailing edge on the inboard side of the nacelle. A similar but smaller breakaway on the outboard side of the nacelle began some five inches ahead of the wing trailing edge. At the trailing edge of the wing the disturbed area covered the nacelle and extended some three or four inches along the wing.

To/

To improve this, fillets were fitted and the tail of the nacelle modified as shown in Figure 4. With these alterations, the flow over the wing was good, but there was a small area of disturbed flow on the inboard side of the nacelle near its tail. From these tests it appeared to be advantageous to build up the nacelles somewhat more on the inboard side than the outboard. The force measurements showed a very slight increase in the lift slope with the modified nacelles, compared with that for the original nacelles, but there were no measurable differences in drag. All the complete model tests were made with the modified nacelle shape shown in Figure 4.

Interference Effects.

The wing and fuselage were tested separately and the wing was tested with and without nacelles. Finally the wing and fuselage combination was tested with and without nacelles. From these tests the mutual interference of the various parts could be deduced. A correction was applied to the sums of the separate drags to compensate for the loss of profile drag of that part of the wing covered by the body.

The interference effects on drag are given in Figure 5. It will be noted that there is no significant difference between the wing-fuselage combination and wing alone plus fuselage over the range $0 < C_L < 0.6$. Above a C_L of 0.6 the interference drag increases steadily with C_L. When nacelles are fitted to the wing there is an appreciable interference drag at all positive lifts. At a C_L of 0.3, this interference drag amounts to roughly eight per cent of the drag of the combination.

The effects on lift and pitching moment of adding the several model components to the wing are shown in Figure 6. Although the "no lift" angle changes from $-3.^\circ4$ for the wing alone to $-1.^\circ7$ for the complete model the slope of lift curves remains practically the same.

The curves of pitching moment against lift reveal the destabilising effects of both fuselage and nacelles and show that the negative value of C_{mo} for the wing-fuselage combination is more than twice that of the wing alone.

The values of C_{mo} for several conditions of the model tested are given below:-

Model Condition	C_{mo}
Wing alone (Transition not fixed)	-0.036
Wing alone (Transition fixed at 0.5 local chord)	-0.031
Wing with two nacelles	-0.035
Wing with fuselage	-0.075
Wing with fuselage and nacelles	-0.069

Tests/

Tests with Various Angles of Tail Setting.

The changes in pitching moment due to changes in the tail-setting angle are given in Table 3, for the complete model, and Table 4 for model without nacelles and shown in Figure 7. The most noticeable features due to fitting the nacelles are the loss of stability and the marked decrease in pitching moments, equal to a decrease in C_{mo} of 0.04. The value of $\frac{dC_m}{d\eta_T}$ (C_L const.) for values of C_L between 0 and 0.5 is not affected appreciably by the presence or otherwise of the nacelles and is approximately 0.035 per degree change in η_T.

Effect of Nacelles on Downwash at Tail.

The presence of the nacelles changes the angle of downwash at the tail by $-0\overset{\circ}{.}7$ over the range of C_L from 0 to 0.7. As shown in Figure 8 this change in downwash angle agrees with the variation in lift angle produced by fitting the nacelles to the wing-fuselage combination.

Tests with Various Elevator Angles.

The range of elevator angles covered by these tests was from $-25°$ to $+20°$ and the results are given in Table 5, for flaps set at 0°, and Table 6, for flaps set at 60°. The results are shown plotted in Figure 9. The striking feature of these curves is their irregularity, as opposed to the roughly parallel, straight line curves obtained in tests on other multi-engined models at the same Reynolds numbers in the Duplex wind tunnel.

If the pitching moments be plotted against elevator angle at constant lift for values of C_L between 0 and 0.5 straight line curves can be drawn over the range of elevator angles from $-10°$ to $+5°$ but outside this range the curves are much kinked. For the above range of C_L the value of $dC_m/d\eta$ (C_L const.) is 0.024 per degree elevator movement.

Setting the flaps to 60° increases the pitching moment appreciably over that without flaps at the same lifts. At a lift coefficient of 0.5 and with the elevators set at 0° this increase in pitching moment is equal to a C_m change of 0.115. This change is roughly double that which occurs under similar conditions with the elevators set at $-10°$.

These curves suggested that the flow in the region of the tail might be poor and an examination by means of streamers was made. This exploration revealed a region of dead air which began some six or eight inches in front of the tailplane leading edge and extended rearwards over the fuselage. It is over this region that the sides of the fuselage converge rapidly, possibly too rapidly, towards the sternpost.

Streamers placed in the position normally occupied by the tailplane leading edge showed that the nacelles had a marked effect on the flow in that region. At low lifts there was less downwash behind the nacelles than at a point, on the same lateral line, behind the mid span of the wing. At about 8° incidence the downwash behind these two points was equal and at higher angles of incidence the downwash behind the nacelles was the greater. Behind the nacelles the change in downwash with vertical height of the streamer was marked.

It was suggested that a wing upper surface breakaway in the region of the nacelles and fuselage might be a contributory cause to the trouble, so a streamer exploration of the flow in this region was carried out. The examination revealed a very disturbed flow over the rear part of the wing upper surface at C_L's greater than 0.6. This disturbed region extended outwards beyond the nacelles for some three or four inches.

In an attempt to improve the flow, the wing section in this region was faired so that the outline of the rear part of the upper surface was a straight line from the trailing edge to the tangent of the original profile. This modification much improved the flow as shown by the streamers and indicated an appreciable inflow towards the fuselage. Balance measurements, given below, show that the fairing had very little effect on either forces or pitching moments.

α° Fuselage Datum	C_L	C_D	C_m	C_L	C_D	C_m
	Normal wing profile			Modified wing profile		
-3.5	-0.18	0.0380	0.0343	-0.165	0.0377	0.0324
-1.5	+0.015	0.0326	0.0338	+0.03	0.0326	0.0323
+0.6	0.23	0.0330	0.0267	0.24	0.0331	0.0256
2.7	0.455	0.0376	0.0191	0.465	0.0379	0.0182
6.8	0.805	0.0580	-0.0114	0.805	0.0580	-0.0096
10.9	0.94	0.0920	-0.0031	0.94	0.0920	-0.0033
13.9	1.06	0.168	-0.0337	1.065	0.168	-0.0362

Finally, some total head explorations in the vertical plane through the position of the tailplane quarter chord line were made to determine the energy lost by the air before reaching the tailplane. The total head combs were fixed to the model so that they lay along the fuselage datum line, nominally +0°.2 to the tailplane chord line. The results of these tests are given in Tables 7 and 8 and are shown in Figures 10 and 11.

The effects of incidence changes, with the total head combs in the design position of the tailplane are shown in Figure 10. Up to 4° incidence the loss is negligible but above that angle it increases, steadily with flaps at 0° and rapidly with flaps set at 60°. It will be observed that at high angles of incidence with the flaps set at 0° the loss tends to be greatest near the body, whereas with flaps set at 60° the loss is very much greater at the tip of the tailplane then at the body.

Figure 11 shows the results of explorations made at various distances from the thrust line, with the model set at 12° incidence. The curves show that the flow improves progressively with distance above the design position of the tail and deteriorates with distance below. The body interference effect shows as an appreciable loss of head near the inboard end of the tailplane.

Tests/

Tests with Propellers Running.

For these tests a new wing with part of the fuselage and nacelles made integral with it was fitted. This construction was necessary to allow the incorporation inside the model of the motor and gearing required to drive the propellers.

Owing to the shortness of time available for these tests it was decided to limit them to a C_L range from 0.5 upwards and to cover a T_c range from 0.13 to 0.29. This range covers the normal climb and also take-off with flaps set at 0°. Comparative tests without propellers were made in all cases. The results obtained are given in Tables 9 to 13 and Figures 12 to 16.

The effects of slipstream on lift are shown in Figure 12. With flaps set at 0° a thrust equivalent to a T_c of 0.29 produces a ten per cent increase in lift at five degrees incidence, on the straight part of the lift-incidence curve. Beyond six degrees incidence, where the lift-incidence curve without propellers flattens out, the percentage increase in lift due to a T_c of 0.29 rises steadily and reaches 27% at an incidence of eleven degrees. With flaps set at 30° the maximum lift occurs at an incidence of 10°.5 and the lift increment with a T_c of 0.29 represents a percentage increase of some 26%. At six degrees incidence, on the straight part of the lift curve, the percentage increase in lift due to the above T_c has fallen to about 16%.

The increase in C_L, due to setting the flaps to 30°, ranges from approximately 0.55 without propellers to 0.68 with a T_c of 0.29 at the point of maximum lift with the flaps set at thirty degrees.

On the same diagram is shown part of the lift curve obtained on the original model with flaps set at 60°. The maximum C_L of 1.55 is attained at an incidence of eight degrees. The increment in C_L due to the flaps at this incidence is approximately 0.75.

Finally it will be noted that the lift-incidence curves of the two models without propellers agree extremely well.

The effect of slipstream on pitching moments without tail and with several tail-settings is shown in Figure 13 for model without flaps and Figure 14 for model with flaps set at 30°. The families of curves are reasonably normal and call for no special comment. The effect of slipstream is destabilising and it also tends to reduce the kink which is most marked in the without propeller case.

The angle of downwash at the tail is plotted against C_L in Figure 15, for the several cases in which it was possible to determine it. Without propellers the agreement between the first and second models is good. At a C_L of 0.5 the downwash angle for the second model is 0°.2 greater at 2°.1 than the value derived for the first model. Without flaps and with slipstream the variation of angle of downwash with lift is much greater than without slipstream and there appears to be a variation with T_c. With flaps set at 30° the differences between the without propeller and various T_c cases is much smaller. Due to the paucity of points the curves must be treated as approximate only, but there is no reason to suppose that additional points would change them to any great extent.

The/

The effect of slipstream on the pitching moments due to various elevator settings both without and with flaps set at 30° is shown in Figure 16. Excepting the kink which occurs in the without flap case the family of curves are of reasonably normal form. There are two points, however, which may be of interest. The first is that at the most positive elevator angle tested, namely +5° without flaps and +10° with flaps set at 30°, the T_c effect on pitching moment is very small, as the elevator setting is reduced the effect of T_c on pitching moment increases progressively. The second is that setting the flaps to 30°, besides increasing the pitching moment on the model, reduces the effectiveness of the elevators. Thus at a C_L of 1.25, the increment in C_m for a 10° movement in elevator is 0.29 without flaps against 0.23 with flaps set at 30°.

Longitudinal Stability.

The tests with propellers covered a C_L range from 0.5 upwards. It is therefore not possible to determine the effects of slipstream on longitudinal stability at low lifts, but the information obtained indicates that the slipstream will have a destabilising effect.

The values of $K_n \left(= -\frac{dC_m}{dC_L}, \text{ stick fixed without propellers} \right)$ deduced from the test results are given below:-

Model Condition	K_n at		
	C_L = 0.2	C_L = 0.4	C_L = 0.6
No tail no nacelles	-0.17	-0.15	-0.13
No tail with nacelles	-0.21	-0.20	-0.19
With tail no nacelles	0.06	0.075	0.095
Complete model	0.035	0.04	0.07

From the above results it will be seen that the nacelles have a marked destabilising effect, which is larger when the tail is absent than when it is fitted.

Comparison of R.A.E., and N.P.L., Pitching Moment Test Results.

After the tests at the N.P.L. had been completed the model was transferred to the R.A.E. where rolling and yawing moments were measured and pitching moment tests were repeated at Reynolds numbers (R) up to 1×10^6.

Comparable sets of tests from the two series have been plotted on the same diagrams and the results are shown in Figures 17, 18 and 19.

In Figure 17, C_L and C_D are plotted against the angle of the fuselage datum line. At the same R the lift curves show very good agreement of their straight parts, but in the N.P.L. tests the initial stall begins between one and two degrees earlier than in the R.A.E. tests. The agreement at minimum drag is also very good, but as the incidence is increased the N.P.L. drag becomes the higher, being about 9% greater at an incidence of 11°.

The/

The relation between lift and pitching moment, with and without tail, is shown in Figure 18. The general agreement again is very good. With tail, the N.P.L. interpolated curve for $\eta_T = -0\overset{\circ}{.}9$ agrees more closely with the R.A.E. curves at the higher values of R than with that for the same R.

The effect of setting the elevators (η) is shown in Figure 19. The N.P.L. results have been adjusted to a tail angle of $-0\overset{\circ}{.}9$ from one of $-0\overset{\circ}{.}17$ by adding the pitching moment increment due to the above change in tail angle. This increment was obtained from Figure 18.

With $\eta = 0$ all three curves are in good agreement. But as η is decreased the R effect on the R.A.E. curves increases and at $\eta = -15^\circ$ and -20° these curves are of quite different shapes. The N.P.L. curves agree more closely with the R.A.F. curves obtained at the higher R.

Longitudinal Stability.

The values of K_m deduced from the R.A.E. and N.P.L. tests are given below:-

Where Made	Vft/sec.	Model Conditions	K_n at $C_L = 0.2$	$C_L = 0.4$	$C_L = 0.6$
R.A.E.	180	less tail	-0.23	-0.23	-0.23
N.P.L.	60	"	-0.23	-0.19	-0.19
R.A.E.	180	Complete Model	0.04	0.055	0.07
N.P.L.	60	"	0.035	0.04	0.065

Table 1/

Table 1

Full scale Dimensions

Model scale = 1/12 Full scale

			Full scale
Wing:-	Gross area	= s	1200 sq.ft.
	Span	= b	115 ft.
	Standard mean chord	= $\bar{c}$	10.43 ft.
	Aspect ratio		11.0
	Taper (tip chord/root chord)		0.289
	Dihedral angle		1°.0
	Wing twist (chord lines)		0°
	Sweepback		Zero on 0.1936$\bar{c}$
	Standard mean chord incidence to fuselage datum (airspeed		4°.1
	Standard mean chord position relative to 25% root chord		$\bar{x}$ = -0.011$\bar{c}$ $\bar{z}$ +0.042$\bar{c}$
	Wing section Root		N.A.C.A. 65_2416
	Wing section Tip		N.A.C.A. 65_2414
Flaps:-	Type		Split
	Outer flap:- area		58.6 sq.ft.
	span		20.8 ft.
	Inner flap:- area		26.8 sq.ft.
	span		7.4 ft.
	Maximum flap deflection		60°
Body:-	Maximum length		80.5 ft.
	" breadth		11.20 ft.
	" height		10.28 ft.
Tailplane:-	Gross area	= S_T	179.4 sq.ft.
	Span		28.0 ft.
	Mean chord		6.41 ft.
	Elevator area		62.5 sq.ft.
	Tail moment arm (from aft C.G.) = ℓ_T		41.5 ft.
	Tail volume = $(S_T\ell_T/S\bar{c})$	=	0.596
	Root thickness/chord ratio	=	15%
Propellers:-	Type:- Constant speed, fully feathering, reversing pitch.		
	Number of propellers		2
	Number of blades per propeller		4
	Diameter		16 ft.
	Solidity at 0.7R		0.114
	Thrust line inclination to fuselage datum		1°5'
	Drawing:- de Havilland		X.P.B.53150
C.G. position:-	Behind L.E. mean chord	=	0.356$\bar{c}$
	Below mean chord line	=	0.265$\bar{c}$
	Behind L.E. root chord	=	4.59 ft.
	Below root chord	=	2.33 ft.

Table 2./

Tests on a 1/12th scale model of the A.S.60.

Wind Speed of Tests = 60ft./second ($R = 0.336 \times 10^6$).
Gills closed in all tests on first model without propellers.

Table No.2. Tests on Components of the Model.

α° Fuselage Datum	C_L	C_D	C_m	C_L	C_D	C_m
Wing alone				Wing alone with 0.020"diam. wire fixed on upper surf.at 50% of chord.		
-5.6	-0.158	0.0158	-0.0551	-0.172	0.0167	-0.0554
-3.5	+0.008	0.0124	-0.0356	-0.009	0.0130	-0.0324
-1.4	0.195	0.0134	-0.0161	+0.177	0.0133	-0.0120
+0.7	0.395	0.0173	-0.0007	0.378	0.0169	+0.0019
2.8	0.593	0.0231	+0.0075	0.580	0.0227	0.0109
4.9	0.788	0.0315	0.0141	0.777	0.0314	0.0159
6.9	0.918	0.0424	0.0206	0.884	0.0430	0.0235
8.9	0.941	0.0538	0.0334	0.888	0.0555	0.0368
11.0	0.988	0.0707	0.0392	0.946	0.0714	0.0404
13.0	1.034	0.0911	0.0399	1.010	0.0908	0.0404
14.0	1.051	0.1040	0.0404	1.044	0.1040	0.0402
Fuselage alone. 0.020"diam.wire fixed on fuselage at 4.5" from nose.				Wing and Fuselage. Wires as before on each component.		
-5.6	-0.016	0.0136	-0.0910	-0.252	0.0284	-0.1330
-3.5	-0.012	0.0126	-0.0669	-0.080	0.0241	-0.0926
-1.4	-0.008	0.0111	-0.0466	+0.106	0.0225	-0.0536
+0.7	-0.004	0.0105	-0.0235	0.311	0.0243	-0.0181
2.8	-0.002	0.0100	+0.0030	0.521	0.0294	+0.0133
4.9	0	0.0099	0.0290	0.725	0.0376	0.0404
6.9	+0.003	0.0100	0.0555	0.844	0.0485	0.0700
8.9	0.007	0.0104	0.0775	0.874	0.0617	0.1015
11.0	0.011	0.0111	0.0990	0.933	0.0781	0.1270
13.0	0.017	0.0121	0.1195	1.015	0.0988	0.1440
14.0	0.019	0.0133	0.1295	1.043	0.1240	0.1445
Wing and two nacelles(final shape). 0.020"diam.wire fixed on upper surface of wings at 50% chord and on nacelles at 2.25" from nose.				Wing, fuselage and nacelles. Wires on each component		
-5.6	-0.217	0.0238	-0.0706	-0.341	0.0374	-0.1590
-3.5	-0.053	0.0182	-0.0439	-0.140	0.0305	-0.1115
-1.4	+0.131	0.0174	-0.0166	+0.040	0.0272	-0.0590
+0.7	0.326	0.0206	+0.0056	0.240	0.0286	-0.0140
2.8	0.532	0.0264	0.0242	0.445	0.0334	+0.0246
4.9	0.722	0.0354	0.0392	0.655	0.0408	0.0672
6.9	0.821	0.0475	0.0519	0.772	0.0515	0.1085
8.9	0.852	0.0605	0.0645	0.812	0.0650	0.1445
11.0	0.917	0.0785	0.0726	0.875	0.0837	0.1722
13.0	0.991	0.1010	0.0785	0.975	0.1260	0.1810
14.0	1.011	0.1230	0.0802	1.012	0.1605	0.1570

Table 3./

Tests on a 1/12th scale model of the A.S.60.

Table No.3. Effect of Varying Tail Angle on Complete Model.
Gills closed

α° Fuselage Datum	C_L	C_m	C_L	C_D	C_m	C_L	C_m
	$\eta_T = -1.80^\circ$			$\eta_T = -0.17^\circ$		$\eta_T = 1.43^\circ$	
-5.6	-0.377	0.0678	-0.363	0.0469	0.0188	-0.348	-0.0274
-3.6	-0.193	0.0702	-0.177	0.0382	0.0173	-0.161	-0.0338
-1.5	-0.025	0.0741	+0.020	0.0331	0.0181	+0.037	-0.0364
+0.6	+0.215	0.0697	+0.232	0.0331	0.0116	0.249	-0.0478
2.6	0.439	0.0666	0.458	0.0378	0.0024	0.475	-0.0621
4.7	0.651	0.0568	0.670	0.0459	-0.0096	0.683	-0.0733
6.8	0.786	0.0395	0.805	0.0585	-0.0247	0.815	-0.0760
8.8	0.845	0.0420	0.862	0.0728	-0.0194	0.871	-0.0613
10.8	0.925	0.0424	0.938	0.0919	-0.0114	0.954	-0.0549
12.9	1.010	0.0347	1.025	0.1385	-0.0157	1.040	-0.0609
13.9	1.045	0.0113	1.060	0.1670	-0.0374	1.070	-0.0831
	$\eta_T = 3.25^\circ$			$\eta_T = 5.30^\circ$		$\eta_T = 7.14^\circ$	
-5.6	-0.332	-0.1005	-0.311		-0.1685	-0.292	-0.2390
-3.6	-0.150	-0.1020	-0.125		-0.1730	-0.102	-0.2455
-1.5	+0.050	-0.1085	+0.075		-0.1810	+0.096	-0.2500
+0.6	0.262	-0.1205	0.288		-0.1880	0.308	-0.2565
2.6	0.488	-0.1320	0.512		-0.1930	0.529	-0.2530
4.7	0.701	-0.1365	0.721		-0.1915	0.709	-0.2450
6.8	0.832	-0.1350	0.849		-0.1825	0.861	-0.2325
8.8	0.888	-0.1220	0.903		-0.1640	0.915	-0.2135
10.8	0.964	-0.1175	0.980		-0.1595	0.994	-0.2180
12.9	1.055	-0.1200	1.065		-0.1600	1.080	-0.2195
13.9	1.085	-0.1445	1.100		-0.1860	1.110	-0.2305

Table 4./

Tests on a 1/12th scale model of the A.S.60.

Table 4. Effect of Varying Tail Angle on Complete Model without Nacelles.

α° Fuselage Datum	C_L	C_D	C_m	C_L	C_m	C_L	C_m
	$\eta_T = -0.17°$			$\eta_T = 1.43°$		$\eta_T = 7.14°$	
-5.6	-0.312	0.0386	0.0627	-0.296	0.0111	-0.239	-0.1850
-3.5	-0.122	0.0316	0.0594	-0.106	0.0029	-0.049	-0.2010
-1.5	+0.076	0.0281	0.0513	+0.092	-0.0046	+0.152	-0.2125
+0.6	0.295	0.0291	0.0412	0.310	-0.0168	0.370	-0.2255
2.7	0.523	0.0343	0.0255	0.537	-0.0342	0.596	-0.2370
4.7	0.738	0.0426	0.0060	0.754	-0.0565	0.810	-0.2475
6.8	0.874	0.0552	-0.0160	0.889	-0.0801	0.936	-0.2510
8.8	0.912	0.0694	-0.0429	0.924	-0.0956	0.967	-0.2440
10.8	0.984	0.0876	-0.0560	0.999	-0.1004	1.035	-0.2430
12.9	1.075	0.1095	-0.0731	1.090	-0.1210	1.125	-0.2620
13.9	1.110	0.1375	-0.0985	1.115	-0.1452	1.150	-0.2715

Table 5./

Tests on a 1/12th scale model of the A.S.60.

Table 5. Effect of Varying Elevator Angle on Complete Model with Tail at -0.17° to Fuselage Datum.

α° Fuselage Datum	C_L	C_m	C_L	C_D	C_m	C_L	C_m
	$\eta = 20^\circ$		$\eta = 15^\circ$			$\eta = 10^\circ$	
-5.6	-0.301	-0.1830	-0.316		-0.1465	-0.337	-0.1700
-3.5	-0.114	-0.1970	-0.132		-0.1460	-0.158	-0.1545
-1.5	+0.085	-0.2020	+0.067		-0.1460	+0.069	-0.1560
+0.6	0.296	-0.2060	0.279		-0.1545	0.276	-0.1430
2.7	0.521	-0.2155	0.506		-0.1645	0.499	-0.1340
4.8	0.738	-0.2540	0.724		-0.2030	0.712	-0.1595
6.8	0.880	-0.3010	0.872		-0.2700	0.857	-0.2215
8.8	0.926	-0.2635	0.919		-0.2350	0.892	-0.2080
10.9	1.000	-0.2525	0.995		-0.2130	0.984	-0.1820
12.9	1.092	-0.2605	1.083		-0.2155	1.070	-0.1735
13.9	1.130	-0.2995	1.118		-0.2475	1.105	-0.1990
	$\eta = 5^\circ$		$\eta = 0^\circ$			$\eta = -5^\circ$	
-5.6	-0.338	-0.0795	-0.365	0.0467	0.0235	-0.392	0.1330
-3.5	-0.147	-0.0860	-0.178	0.0380	0.0227	-0.213	0.1475
-1.5	+0.052	-0.0935	+0.019	0.0328	0.0221	-0.014	0.1485
+0.6	0.263	-0.1020	0.231	0.0331	0.0157	+0.198	0.1385
2.7	0.490	-0.1120	0.457	0.0377	0.0068	0.423	0.1305
4.8	0.704	-0.1355	0.670	0.0459	-0.0063	0.636	0.1200
6.8	0.836	-0.1425	0.805	0.0583	-0.0208	0.774	0.1015
8.8	0.886	-0.1230	0.860	0.0728	-0.0146	0.832	0.0935
10.9	0.967	-0.1125	0.939	0.0921	-0.0101	0.912	0.0910
12.9	1.053	-0.1060	1.024	0.1374	-0.0136	1.002	0.0740
13.9	1.089	-0.1350	1.063	0.1678	-0.0401	1.044	0.0415
	$\eta = -10^\circ$		$\eta = -15^\circ$			$\eta = -25^\circ$	
-5.6	-0.414	0.2035	-0.428		0.2630	-0.465	0.4045
-3.5	-0.236	0.2405	-0.248		0.2830	-0.282	0.4125
-1.5	-0.043	0.2550	-0.056		0.3045	-0.089	0.4310
+0.6	+0.173	0.2335	+0.157		0.2910	+0.117	0.4390
2.7	0.396	0.2270	0.388		0.2685	0.345	0.4225
4.8	0.604	0.2325	0.601		0.2560	0.562	0.3955
6.8	0.740	0.2280	0.730		0.2795	0.698	0.3925
8.8	0.800	0.2150	0.781		0.3015	0.751	0.4170
10.9	0.883	0.2035	0.862		0.2915	0.838	0.3970
12.9	0.974	0.1705	0.955		0.2575	0.935	0.3480
13.9	1.012	0.1355	0.997		0.2195	0.976	0.3015

Table 6./

Tests on a 1/12th scale model of the A.S.60.

Table 6. Tests on Model, with and without Tail, and with Flaps at 60°. Gills closed

$\eta_T = -0.17°$ to Fuselage Datum.

α° Fuselage Datum	C_L	C_D	C_m	C_L	C_m	C_L	C_m
	η = 0°			η = -10°		Model without Tail	
-5.3	0.509	0.1589	0.1185	0.469	0.2720	0.587	-0.1611
-3.2	0.713	0.1576	0.1211	0.671	0.2815	0.782	-0.1198
-1.2	0.927	0.1616	0.1197	0.882	0.2870	0.980	-0.0896
+1.9	1.131	0.1712	0.1238	1.085	0.3050	1.176	-0.0533
3.0	1.350	0.1863	0.1144	1.275	0.3275	1.359	-0.0224
5.0	1.488	0.2058	0.1034	1.426	0.3365	1.520	+0.0046
7.0	1.538	0.2291	0.1023	1.478	0.3290	1.543	0.0402
9.0	1.549	0.2778	0.0839	1.484	0.3030	1.543	0.0634
11.0	1.465	0.4119	0.0274	1.411	0.2285	1.455	0.0630
13.0	1.340	0.4848	0.0058	1.300	0.1605	1.312	0.0710
14.0	1.315	0.5191	-0.0087	1.282	0.1330	1.284	0.0736

Table 7./

Tests on a 1/12th scale model of the A.S.60.

Table 7. Total Head Distribution in Region of Tailplane Position.

The following data applies to all cases.

Mouths of tubes $2\frac{1}{2}$" behind the position of the tailplane L.E. at side of body.
Combs set parallel to the fuselage datum line and normal to plane of symmetry of model.
Distance from centre line of sting to the 26th (innermost) tube = 2.15".
Distance from centre line of sting to the 1st (outermost) tube = 20.9".
Distance between tubes = 0.75".

(a) Flaps set at 0°

	Ratio of Total Head to Total Head of Free Stream = p/q																		
	Tubes 3.75" below L.E. of Tail.				Tubes 1.75" below L.E. of Tail.				Tubes level with L.E. -T.E. chord of Tail.					Tubes 2.0" above L.E. of Tail.				Tubes 4.0" above L.E. of Tail	
Tube No.	α=0°	α=4°	α=8°	α=12°	α=0°	α=4°	α=8°	α=12°	α=0°	α=4°	α=8°	α=10°	α=12°	α=0°	α=4°	α=8°	α=12°	α=8°	α=12°
1	1.01	1.01	0.89	1.00	0.99	0.98	0.98	0.97	1.00	0.99	0.98	0.98	0.82	1.00	Values the same as for α = 0°	1.00	0.96	1.00	1.00
2	1.01	1.01	0.90	0.99	0.99	0.97	0.98	0.96	1.00	0.98	0.98	0.99	0.81	1.00		1.00	0.97		1.00
3	1.01	1.01	0.91	0.99	0.99	0.98	0.97	0.97	1.00	0.98	0.97	0.98	0.82	1.00		1.00	0.97		1.00
4	1.01	1.01	0.83	0.99	0.99	0.97	0.91	0.97	1.00	0.98	0.91	0.95	0.82	1.00		0.99	0.91		0.99
5	1.00	0.97	0.86	0.98	0.98	0.95	0.83	0.96	0.99	0.98	0.82	0.90	0.84	0.98		0.99	0.83		0.94
6	1.01	0.91	0.96	0.99	0.99	0.93	0.81	0.98	1.00	0.98	0.79	0.83	0.87	1.00		0.96	0.82		0.92
7	1.00	0.85	0.96	0.97	0.98	0.91	0.81	0.97	1.00	0.98	0.83	0.82	0.90	0.99		0.95	0.81		0.89
8	0.96	0.84	0.98	0.96	0.99	0.91	0.86	0.89	0.99	0.98	0.88	0.84	0.90	1.00		0.97	0.82		0.88
9	0.80	0.80	0.99	0.89	0.99	0.89	0.87	0.82	1.00	0.98	0.94	0.87	0.86	1.00		0.99	0.83		0.89
10	0.72	0.97	0.96	0.86	0.99	0.86	0.86	0.81	0.99	0.98	0.97	0.88	0.81	0.99		0.99	0.84		0.91
11	0.73	0.91	0.85	0.90	0.98	0.82	0.83	0.84	1.00	0.98	0.97	0.91	0.81	0.99		0.99	0.88		0.94
12	0.77	0.82	0.81	0.96	0.99	0.79	0.82	0.87	0.97	0.98	0.97	0.93	0.84	0.99		Values the same as for α = 0°	0.94		0.97
13	0.89	0.87	0.83	0.98	0.99	0.81	0.84	0.90	1.00	0.99	0.97	0.96	0.89	0.99			0.99		0.99
14	1.02	0.90	0.92	0.99	0.99	0.86	0.87	0.92	1.00	0.99	0.97	0.97	0.92	0.99			1.01		0.99
15	1.03	0.85	0.99	0.99	0.99	0.91	0.88	0.93	1.00	0.99	0.95	0.96	0.91	0.99			1.00		0.99
16	1.03	0.92	0.98	1.00	1.00	0.97	0.85	0.94	1.00	0.99	0.92	0.92	0.88	0.99			0.96		0.99
17	–	–	–	–	1.00	0.98	0.83	0.93	1.00	0.99	0.92	0.88	0.84	0.99			0.94		0.98
18	1.03	0.83	0.91	0.99	1.00	0.99	0.81	0.93	1.00	0.99	0.94	0.87	0.81	1.00			0.90		0.99
19	1.03	0.83	0.88	0.99	1.00	0.99	0.86	0.89	1.00	0.99	0.97	0.90	0.79	1.00			0.90		0.99
20	1.03	0.83	0.86	0.97	1.00	0.99	0.91	0.83	1.00	0.99	0.98	0.91	0.76	1.02			0.90		0.99

Table 7. (Continued)/

Table 7. (Continued).

(a) Flaps set at 0°

	Ratio of Total Head to Total Head of Free Stream = p/q																		
	Tubes 3.75" below L.E. of Tail.				Tubes 1.75" below L.E. of Tail.				Tubes level with L.E. -T.E. chord of Tail.					Tubes 2.0" above L.E. of Tail.				Tubes 4.0" above L.E. of Tail.	
Tube No.	$\alpha=0°$	$\alpha=4°$	$\alpha=8°$	$\alpha=12°$	$\alpha=0°$	$\alpha=4°$	$\alpha=8°$	$\alpha=12°$	$\alpha=0°$	$\alpha=4°$	$\alpha=8°$	$\alpha=10°$	$\alpha=12°$	$\alpha=0°$	$\alpha=4°$	$\alpha=8°$	$\alpha=12°$	$\alpha=8°$	$\alpha=12°$
21	1.02	0.82	0.83	0.95	0.99	0.99	0.93	0.77	0.99	0.98	0.98	0.91	0.72	1.00	Values the same as for $\alpha=0°$	Values the same as for $\alpha=0°$	0.91		0.98
22	1.02	0.82	0.83	0.93	0.99	0.99	0.94	0.72	0.99	0.98	0.98	0.92	0.71	1.00			0.92		0.98
23	1.02	0.80	0.80	0.90	0.97	0.98	0.93	0.70	0.98	0.98	0.98	0.92	0.73	0.99			0.94		0.98
24	1.02	0.88	0.79	0.84	-	-	-	-	0.91	0.98	0.97	0.93	0.78	0.99			0.98		0.99
25	1.01	0.71	0.75	0.75	0.97	0.99	0.72	0.67	0.99	0.99	0.98	0.89	0.83	0.99			1.00		0.99
26	0.64	0.48	0.65	0.64	0.65	0.72	0.61	0.61	0.96	0.98	0.98	0.87	0.85	0.99			1.01	1.00	0.99
Arrange Values of p/q	0.95	0.87	0.88	0.94	0.98	0.93	0.86	0.87	0.99	0.98	0.94	0.91	0.83	1.00	1.00	0.99	0.92	1.00	0.97

Table 8./

Tests on a 1/12th scale model of the A.S.60.

<u>Table 8. Total Head Distribution in Region of Tailplane Position (contd.).</u>

(b) Flaps set at 60°

	p/q															
	Tubes 3.75" below L.E. of Tail.				Tubes 1.75" below L.E. of Tail.			Tubes level with L.E.-T.E. chord of Tail.				Tubes 2.0" above L.E. of Tail.			Tubes 4.0" above L.E. of Tail.	
Tube No.	α=0°	α=4°	α=8°	α=12°	α=4°	α=8°	α=12°	α=4°	α=8°	α=10°	α=12°	α=4°	α=8°	α=12°	α=8°	α=12°
1	0.95	0.95	0.73	0.39	1.00	0.83	0.40	1.00	0.91	0.71	0.36	1.00	1.00	0.42	1.00	0.49
2	0.95	0.95	0.70	0.39		0.82	0.38		0.90	0.66	0.34	1.00	0.99	0.38		0.47
3	0.96	0.95	0.71	0.38		0.82	0.37		0.89	0.63	0.33	1.00	0.99	0.39		0.46
4	0.96	0.95	0.72	0.37		0.85	0.36		0.90	0.61	0.33	1.00	1.00	0.37		0.46
5	0.94	0.93	0.72	0.35		0.85	0.34		0.91	0.59	0.32	0.99	0.98	0.37		0.47
6	0.96	0.95	0.75	0.35		0.88	0.35		0.93	0.58	0.33	1.00	1.00	0.38		0.50
7	0.95	0.95	0.76	0.33		0.89	0.35		0.94	0.59	0.33		0.99	0.39		0.51
8	0.96	0.95	0.80	0.33		0.92	0.36		0.96	0.61	0.35		1.00	0.42		0.53
9	0.96	0.96	0.84	0.32		0.94	0.37		0.98	0.62	0.37		1.00	0.45		0.57
10	0.96	0.96	0.87	0.31		0.97	0.38		0.99	0.66	0.38			0.49		0.59
11	0.96	0.97	0.90	0.30		0.99	0.40		0.99	0.69	0.41			0.53		0.62
12	0.97	0.97	0.93	0.30		0.99	0.42		0.98	0.73	0.43			0.57		0.66
13	0.97	0.97	0.96	0.30		1.01	0.45		0.99	0.79	0.47			0.62		0.70
14	0.97	0.97	0.98	0.31		1.01	0.46		1.00	0.84	0.51			0.67		0.74
15	0.97	0.97	1.00	0.31		1.01	0.50		1.00	0.89	0.55			0.71		0.79
16	0.97	0.97	1.00	0.33		1.02	0.54		1.00	0.95	0.60			0.75		0.81
17	0.97	0.97	1.00	0.34		1.02	0.58		1.00	0.96	0.63			0.80		0.85
18	0.57	0.98	1.00	0.38		1.02	0.61		1.00	0.98	0.67			0.83		0.88
19	0.97	0.98	1.00	0.41		1.02	0.65		1.00	1.00	0.72			0.86		0.90
20	0.97	0.98	0.96	0.44		1.02	0.69		1.00	1.00	0.75			0.88		0.92
21	0.97	0.96	0.98	0.45		1.01	0.74		1.00	1.00	0.78			0.89		0.93
22	0.96	0.93	0.94	0.48		1.01	0.76		1.00	0.99	0.81			0.91		0.95
23	0.93	0.89	0.87	0.45		1.01	0.78		0.99	0.99	0.83			0.94		0.96
24	0.91	0.81	0.78	0.44		1.01	0.79		1.00	1.00	0.86			0.95		0.97
25	0.85	0.68	0.63	0.37		1.00	0.76		1.00	0.99	0.88			0.97		0.98
26	0.45	0.36	0.37	0.25	1.00	0.98	0.70	1.00	0.99	0.99	0.89	1.00	1.00	0.97	1.00	0.98
<u>Averages</u>	0.94	0.92	0.84	0.36	1.00	0.96	0.52	1.00	0.97	0.81	0.55	1.00	1.00	0.65	1.00	0.72

Table 9./

Tests on a 1/12th scale model of the A.S.60.
with airscrews.

Table 9. Complete Model with Various Tail Settings. Gills Open.
Elevators 0°. Blade Angle 25°.
Wind Speed = 60 ft./sec.

J	T_o	α° Fuselage Datum	C_L	C_D	C_m
		Without Tail			
Without Airscrews		3.7	0.533	0.0384	0.0516
		6.8	0.758	0.0540	0.1095
		9.8	0.847	0.0759	0.1545
		12.8	0.983	0.1187	0.1658
0.665	0.29	3.7	0.587	-0.2054	0.0533
		6.8	0.868	-0.1836	0.1251
		9.8	1.050	-0.1508	0.1713
		12.8	1.266	-0.1026	0.2234
0.705	0.24	3.7	0.584	-0.1641	0.0550
		6.8	0.858	-0.1438	0.1211
		9.8	1.026	-0.1129	0.1731
		12.8	1.230	-0.0659	0.2190
0.755	0.19	3.7	0.575	-0.1238	0.0591
		6.8	0.843	-0.1037	0.1257
		9.8	0.999	-0.0733	0.1772
		12.8	1.187	-0.0278	0.2282
0.825	0.13	3.7	0.559	-0.0748	0.0619
		6.8	0.822	-0.0540	0.1286
		9.8	0.963	-0.0275	0.1822
		12.8	1.144	+0.0143	0.2226
		η_T = -1.80°			
Without Airscrews		3.7	0.525	0.0430	0.0833
		6.8	0.768	0.0597	0.0585
		9.8	0.874	0.0822	0.0599
		12.9	1.018	0.1266	0.0429
0.665	0.29	3.7	0.558	-0.1999	0.1535
		6.8	0.862	-0.1779	0.1603
		9.8	1.053	-0.1454	0.1646
		12.9	1.226	-0.0938	0.1551
0.705	0.24	3.7	0.555	-0.1584	0.1444
		6.8	0.850	-0.1383	0.1516
		9.8	1.031	-0.1066	0.1657
		12.9	1.254	-0.0562	0.1442
0.755	0.19	3.7	0.549	-0.1169	0.1332
		6.8	0.836	-0.0974	0.1398
		9.8	0.981	-0.0669	0.1533
		12.9	1.217	-0.0187	0.1335
0.825	0.13	3.7	0.545	-0.0661	0.1222
		6.8	0.824	-0.0480	0.1249
		9.8	0.973	-0.0199	0.1340
		12.9	1.169	+0.0244	0.1224

Table 9. (Continued)/

Tests on a 1/12th scale model of the A.S.60.

Table 9. (Continued.)

J	T_c	$\alpha°$ Fuselage Datum	C_L	C_D	C_m
		η_T =	−0.17°		
Without	Airscrews	3.7	0.554	0.0436	0.0127
		6.8	0.792	0.0615	−0.0097
		9.8	0.887	0.0846	0.0024
		12.9	1.030	0.1283	−0.0199
0.665	0.29	3.7	0.592	−0.2005	0.0666
		6.8	0.892	−0.1781	0.0756
		9.8	1.084	−0.1455	0.0812
		12.9	1.316	−0.0911	0.0699
0.705	0.24	3.7	0.584	−0.1602	0.0608
		6.8	0.878	−0.1381	0.0700
		9.8	1.056	−0.1072	0.0795
		12.9	1.280	−0.0545	0.0631
0.755	0.19	3.7	0.578	−0.1193	0.0556
		6.8	0.863	−0.0968	0.0602
		9.8	1.030	−0.0650	0.0739
		12.9	1.240	−0.0151	0.0553
0.825	0.13	3.7	0.570	−0.0688	0.0473
		6.8	0.849	−0.0482	0.0516
		9.8	0.998	−0.0199	0.0640
		12.9	1.193	+0.0265	0.0503
		η_T =	+1.43°		
Without	Airscrews	3.7	0.559	0.0441	−0.0456
		6.8	0.806	0.0607	−0.0553
		9.8	0.905	0.0853	−0.0423
		12.9	1.040	0.1326	−0.0507
0.665	0.29	3.7	0.601	−0.1994	−0.0043
		6.8	0.906	−0.1743	−0.0028
		9.8	1.098	−0.1412	+0.0020
		12.9	1.323	−0.0874	−0.0023
0.705	0.24	3.7	0.597	−0.1577	−0.0104
		6.8	0.897	−0.1355	−0.0013
		9.8	1.074	−0.1024	−0.0024
		12.9	1.292	−0.0496	−0.0091
0.755	0.19	3.7	0.591	−0.1169	−0.0166
		6.8	0.877	−0.0935	−0.0129
		9.8	1.051	−0.0607	+0.0037
		12.9	1.247	−0.0116	−0.0124
0.825	0.13	3.7	0.582	−0.0665	−0.0199
		6.8	0.860	−0.0450	−0.0204
		9.8	1.012	−0.0162	−0.0095
		12.9	1.209	+0.0313	−0.0085

Table 10./

Tests on a 1/12th scale model of the A.S.60.

Table 10. Complete Model with Various Elevator Settings. Gills Open.
Blade Angle 25° η_T = -0.17°.
Wind Speed = 60 ft./sec.

J	T_c	α° Fuselage Datum	C_L	C_D	C_m
		Elevator Setting -5°			
Without Airscrews		3.7	0.520	0.0432	0.1352
		6.8	0.763	0.0587	0.1067
		9.8	0.860	0.0815	0.1012
		12.9	1.006	0.1300	0.0904
0.665	0.29	3.7	0.535	-0.2003	0.2203
		6.8	0.842	-0.1789	0.2203
		9.8	1.041	-0.1484	0.2381
		12.9	1.269	-0.0968	0.2260
0.705	0.24	3.7	0.535	-0.1598	0.2122
		6.8	0.835	-0.1400	0.2151
		9.8	1.022	-0.1083	0.2276
		12.9	1.237	-0.0594	0.2156
0.755	0.19	3.7	0.529	-0.1185	0.2010
		6.8	0.823	-0.0984	0.2086
		9.8	0.994	-0.0667	0.2154
		12.9	1.200	-0.0203	0.1988
0.825	0.13	3.7	0.519	-0.0682	0.1880
		6.8	0.806	-0.0494	0.1830
		9.8	0.962	-0.0221	0.1985
		12.9	1.153	+0.0220	0.1858
		Elevator Setting +5°			
Without Airscrews		3.7	0.581	0.0453	-0.1026
		6.8	0.824	0.0638	-0.1223
		9.8	0.916	0.0877	-0.0963
		12.9	1.057	0.1381	-0.0991
0.665	0.29	3.7	0.627	-0.1971	-0.0755
		6.8	0.923	-0.1737	-0.0678
		9.8	1.125	-0.1367	-0.0742
		12.9	1.348	-0.0818	-0.0921
0.705	0.24	3.7	0.636	-0.1576	-0.0841
		6.8	0.932	-0.1338	-0.0734
		9.8	1.116	-0.0998	-0.0759
		12.9	1.322	-0.0449	-0.0963
0.755	0.19	3.7	0.637	-0.1155	-0.0880
		6.8	0.910	-0.0936	-0.0737
		9.8	1.086	-0.0586	-0.0709
		12.9	1.289	-0.0063	-0.0912
0.825	0.13	3.7	0.618	-0.0669	-0.0867
		6.8	0.894	-0.0449	-0.0803
		9.8	1.047	-0.0142	-0.0713
		12.9	1.214	+0.0328	-0.0854

Table 11./

Tests on a 1/12th scale model of the A.S.60.

Table 11. Complete Model with Various Elevator Settings. Gills Shut.
Blade Angle 25° η_T = -0.17°.
Wind Speed = 60 ft./sec.

J	T_c	$\alpha°$ Fuselage Datum	C_L	C_D	C_m
		Elevator Setting -5°			
Without	Airscrews	3.7	0.521	0.0429	0.1335
		6.8	0.759	0.0581	0.1122
		9.8	0.864	0.0811	0.1064
		12.9	1.003	0.1276	0.0797
0.665	0.29	3.7	0.555	-0.1984	0.2125
		6.8	0.850	-0.1765	0.2203
		9.8	1.040	-0.1465	0.2359
		12.9	1.267	-0.0951	0.2288
0.705	0.24	3.7	0.550	-0.1582	0.2034
		6.8	0.837	-0.1381	0.2099
		9.8	1.019	-0.1001	0.2251
		12.9	1.237	-0.0601	0.2167
0.755	0.19	3.7	0.543	-0.1186	0.1927
		6.8	0.822	-0.0985	0.1970
		9.8	0.993	-0.0697	0.2128
		12.9	1.205	-0.0208	0.2069
0.825	0.13	3.7	0.535	-0.0695	0.1802
		6.8	0.810	-0.0494	0.1807
		9.8	0.962	-0.0234	0.1995
		12.9	1.157	+0.0205	0.1966
		Elevator Setting 0°			
Without	Airscrews	3.7	0.545	0.0428	0.0185
		6.8	0.794	0.0595	-0.0068
		9.8	0.887	0.0837	+0.0045
		12.9	1.031	0.1306	-0.0038
0.665	0.29	3.7	0.592	-0.2015	0.0724
		6.8	0.891	-0.1776	0.0655
		9.8	1.083	-0.1439	0.0716
		12.9	1.308	-0.0912	0.0726
0.705	0.24	3.7	0.590	-0.1598	0.0665
		6.8	0.878	-0.1391	0.0620
		9.8	1.052	-0.1069	0.0683
		12.9	1.278	-0.0476	0.0572
0.755	0.19	3.7	0.579	-0.1160	0.0564
		6.8	0.860	-0.0976	0.0523
		9.8	1.029	-0.0663	0.0656
		12.9	1.241	-0.0171	0.0567
0.825	0.13	3.7	0.569	-0.0697	0.0520
		6.8	0.845	-0.0486	0.0400
		9.8	0.997	-0.0208	0.0569
		12.9	1.185	+0.0245	0.0575

Table 11. (Continued)/

Tests on a 1/12th scale model of the A.S.60.

Table 11. (Continued).

J	T_c	α° Fuselage Datum	C_L	C_D	C_m
		Elevator Setting +5°			
Without Airscrews		3.7	0.587	0.0447	-0.1067
		6.8	0.823	0.0625	-0.1220
		9.8	0.918	0.0872	-0.0968
		12.9	1.057	0.1354	-0.1022
0.665	0.29	3.7	0.637	-0.1994	-0.0790
		6.8	0.929	-0.1742	-0.0766
		9.8	1.129	-0.1391	-0.0801
		12.9	1.358	-0.0829	-0.0885
0.705	0.24	3.7	0.631	-0.1579	-0.0856
		6.8	0.914	-0.1347	-0.0776
		9.8	1.102	-0.1007	-0.0775
		12.9	1.321	-0.0462	-0.0912
0.755	0.19	3.7	0.620	-0.1175	-0.0895
		6.8	0.900	-0.0941	-0.0792
		9.8	1.068	-0.0602	-0.0740
		12.9	1.277	-0.0092	-0.0885
0.825	0.13	3.7	0.610	-0.0658	-0.0885
		6.8	0.881	-0.0442	-0.0837
		9.8	1.033	-0.0159	-0.0704
		12.9	1.230	+0.0322	-0.0809

Table 12./

Tests on a 1/12th scale model of the A.S.60.

Table 12. Complete Model with Various Elevator Settings.
Blade Angle 25° η_T = -0.17°. Gills Open. Flaps 30°.
Wind Speed 60 ft./sec.

J	T_c	α° Fuselage Datum	C_L	C_D	C_m
		Elevator Setting 0°			
Without Airscrews		3.9	1.101	0.1145	0.0560
		7.0	1.347	0.1438	0.0228
		10.0	1.404	0.1871	0.0263
		13.0	1.320	0.3332	-0.0172
0.665	0.29	3.9	1.241	-0.1029	0.1429
		7.0	1.541	-0.0601	0.1558
		10.0	1.730	-0.0061	0.1467
		13.0	1.670	+0.1573	0.1308
0.705	0.24	3.9	1.222	-0.0663	0.1361
		7.0	1.517	-0.0266	0.1428
		10.0	1.692	+0.0268	0.1379
		13.0	1.637	0.1843	0.1145
0.755	0.19	3.9	1.198	-0.0289	0.1276
		7.0	1.487	+0.0118	0.1273
		10.0	1.653	0.0632	0.1243
		13.0	1.577	0.2146	0.0989
0.825	0.13	3.9	1.175	0.0168	0.1128
		7.0	1.459	0.0541	0.1118
		10.0	1.592	0.0997	0.1077
		13.0	1.509	0.2441	0.0874
		Elevator Setting +5°			
Without Airscrews		3.9	1.133	0.1176	-0.0610
		7.0	1.378	0.1478	-0.0992
		10.0	1.430	0.1921	-0.0902
		13.0	1.337	0.3336	-0.0820
0.665	0.29	3.9	1.278	-0.0981	+0.0105
		7.0	1.583	-0.0523	0.0129
		10.0	1.769	-0.0197	0.0133
		13.0	1.682	+0.1630	0.0034
0.705	0.24	3.9	1.261	-0.0622	0.0058
		7.0	1.555	-0.0207	0.0020
		10.0	1.733	+0.0128	0.0101
		13.0	1.671	0.1922	-0.0061
0.755	0.19	3.9	1.236	-0.0242	-0.0035
		7.0	1.529	+0.0178	-0.0082
		10.0	1.683	0.0477	+0.0025
		13.0	1.618	0.2244	-0.0190
0.825	0.13	3.9	1.210	0.0214	-0.0123
		7.0	1.491	0.0600	-0.0218
		10.0	1.624	0.0852	-0.0026
		13.0	1.548	0.2564	-0.0170

Table 12. (Continued)./

Tests on a 1/12th scale model of the A.S.60.

Table 12. (Continued).

J	T_c	α° Fuselage Datum	C_L	C_D	C_m
		Elevator Setting +10°			
Without Airscrews		3.9	1.146	0.1212	-0.1167
		7.0	1.388	0.1517	-0.1237
		10.0	1.442	0.1970	-0.1529
		13.0	1.328	0.3390	-0.1496
0.665	0.29	3.9	1.301	-0.0966	-0.0920
		7.0	1.614	-0.0499	-0.1086
		10.0	1.808	-0.0127	-0.1242
		13.0	1.736	+0.1705	-0.1190
0.705	0.24	3.9	1.284	-0.0601	-0.1036
		7.0	1.582	-0.0156	-0.1075
		10.0	1.756	+0.0175	-0.1067
		13.0	1.651	0.1983	-0.1204
0.755	0.19	3.9	1.267	-0.0216	-0.1022
		7.0	1.554	+0.0216	-0.1163
		10.0	1.705	0.0516	-0.0913
		13.0	1.621	0.2220	-0.1187
0.825	0.13	3.9	1.235	0.0238	-0.1261
		7.0	1.517	0.0641	-0.1231
		10.0	1.657	0.0895	-0.0888
		13.0	1.561	0.2595	-0.1193

Table 13./

Tests on a 1/12th scale model of the A.S.60.

Table 13. Complete Model with Various Tail Settings.
Blade Angle 25°. Elevators 0° Gills Open. Flaps 30°.
Wind Speed 60 ft./sec.

J	T_c	α° Fuselage Datum	C_L	C_D	C_m
Tail Setting +1.43° to Fuselage Datum					
Without Airscrews		3.9	1.111	0.1146	-0.0137
		7.0	1.359	0.1456	-0.0300
		10.0	1.421	0.1888	-0.0400
		13.0	1.322	0.3351	-0.0715
0.665	0.29	3.9	1.259	-0.1010	0.0785
		7.0	1.566	-0.0565	0.0812
		10.0	1.761	-0.0006	0.0835
		13.0	1.680	+0.1622	0.0605
0.705	0.24	3.9	1.240	-0.0664	0.0718
		7.0	1.536	-0.0235	0.0685
		10.0	1.713	+0.0300	0.0686
		13.0	1.642	0.1885	0.0542
0.755	0.19	3.9	1.225	-0.0270	0.0617
		7.0	1.509	+0.0140	0.0551
		10.0	1.681	0.0678	0.0574
		13.0	1.588	0.2168	0.0489
0.825	0.13	3.9	1.194	0.0178	0.0487
		7.0	1.481	0.0554	0.0377
		10.0	1.614	0.1029	0.0455
		13.0	1.500	0.2522	0.0427
Tail Setting +3.25° to Fuselage Datum					
Without Airscrews		3.9	1.148	0.1170	-0.0894
		7.0	1.379	0.1480	-0.1079
		10.0	1.428	0.1923	-0.1101
		13.0	1.347	0.3353	-0.0959
0.665	0.29	3.9	1.292	-0.0985	-0.0091
		7.0	1.589	-0.0538	-0.0082
		10.0	1.780	+0.0030	-0.0136
		13.0	1.679	0.1629	-0.0178
0.705	0.24	3.9	1.274	-0.0626	-0.0090
		7.0	1.562	-0.0206	-0.0165
		10.0	1.770	+0.0363	-0.0230
		13.0	1.648	0.1917	-0.0152
0.755	0.19	3.9	1.253	-0.0265	-0.0250
		7.0	1.536	+0.0176	-0.0304
		10.0	1.687	0.0708	-0.0243
		13.0	1.590	0.2489	-0.0258
0.825	0.13	3.9	1.201	0.0199	-0.0334
		7.0	1.500	0.0594	-0.0376
		10.0	1.634	0.1071	-0.0277
		13.0	1.561	0.2455	-0.0219

Table 13. (Continued)./

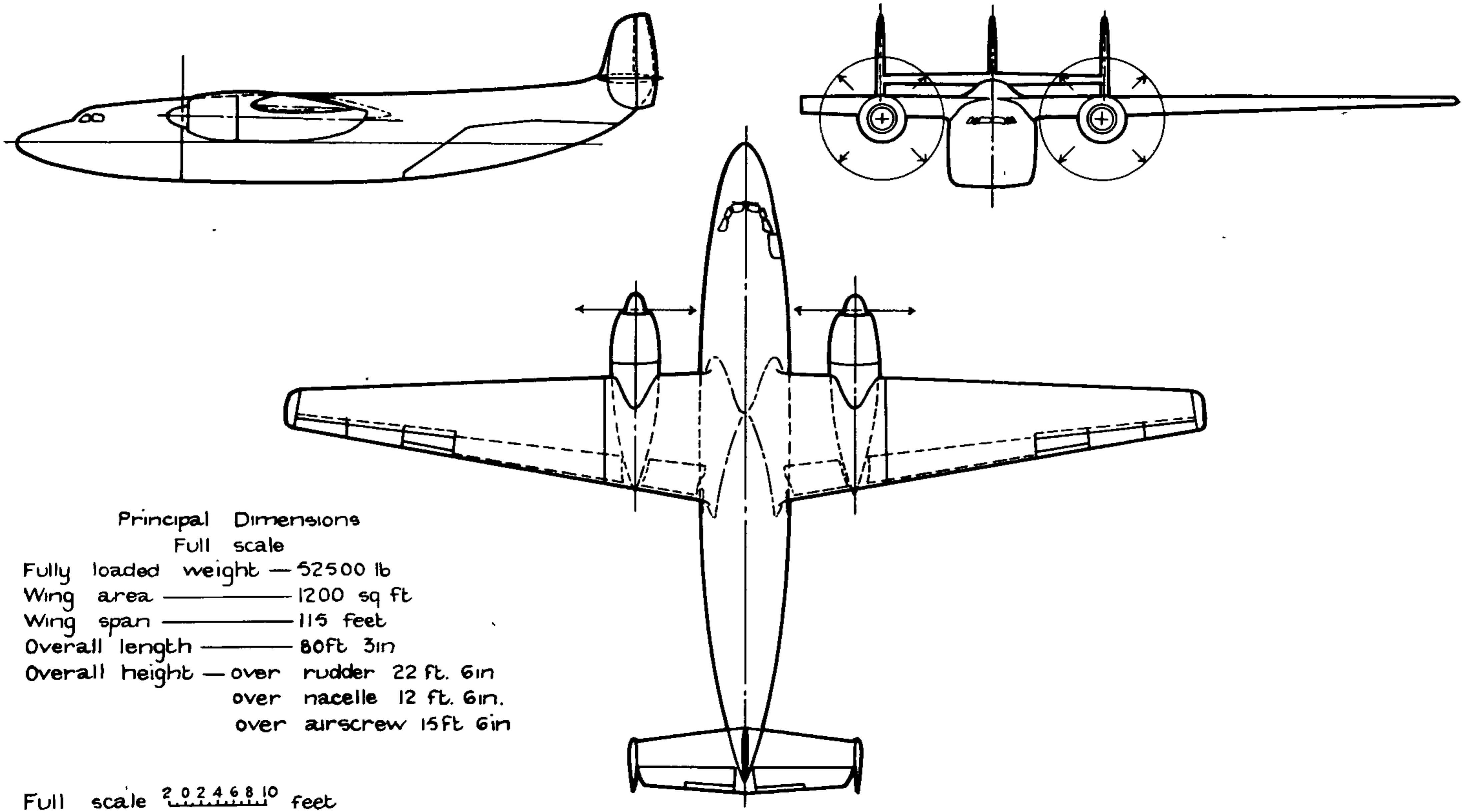
Tests on a 1/12th scale model of the A.S.60.

Table 13. (Continued).

J	T_c	α° Fuselage Datum	C_L	C_D	C_m
		Without Tail			
Without Airscrews		3.9	1.115	0.1119	-0.0197
		7.0	1.348	0.1389	+0.0127
		10.0	1.392	0.1822	0.0631
		13.0	1.322	0.3815	-0.0034
0.665	0.29	3.9	1.294	-0.1038	-0.0435
		7.0	1.573	-0.0611	+0.0106
		10.0	1.743	-0.0066	0.0654
		13.0	1.671	0.1580	0.0775
0.705	0.24	3.9	1.269	-0.0685	-0.0364
		7.0	1.547	-0.0280	+0.0146
		10.0	1.711	+0.0252	0.0672
		13.0	1.621	0.1792	0.0885
0.755	0.19	3.9	1.241	-0.0297	-0.0303
		7.0	1.510	+0.0096	+0.0246
		10.0	1.658	0.0592	0.0724
		13.0	1.571	0.2102	0.0794
0.825	0.13	3.9	1.205	0.0155	-0.0181
		7.0	1.479	0.0511	+0.0293
		10.0	1.599	0.0972	0.0822
		13.0	1.526	0.2414	0.0844

AM.

12,058
FIG 1



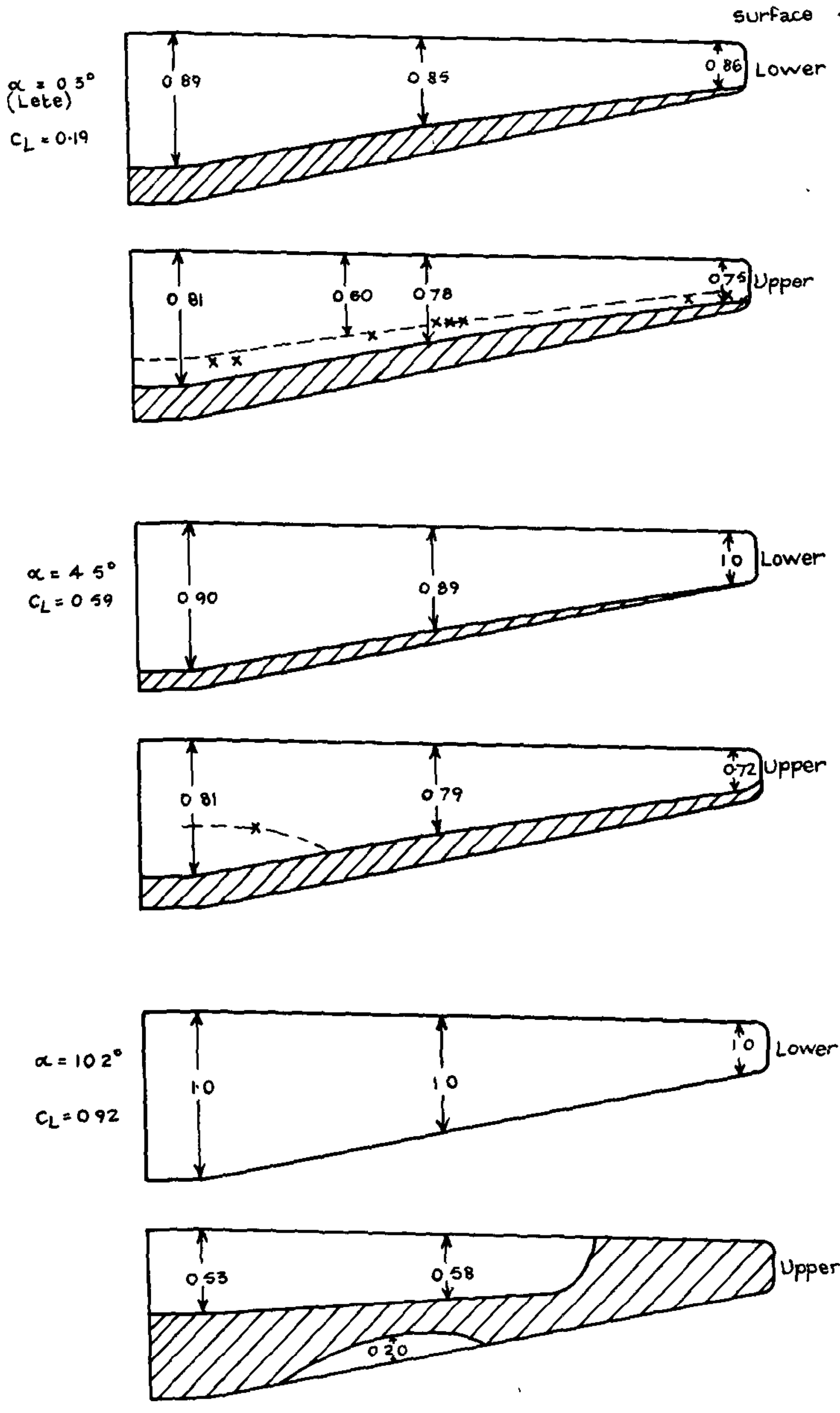
Principal Dimensions
Full scale

Fully loaded weight — 52500 lb
Wing area — 1200 sq ft
Wing span — 115 feet
Overall length — 80ft 3in
Overall height — over rudder 22 ft. 6in
over nacelle 12 ft. 6in.
over airscrew 15ft 6in

Full scale 2 0 2 4 6 8 10 feet

General arrangement of Airspeed AS 60. Model scale = 1/12 Full scale

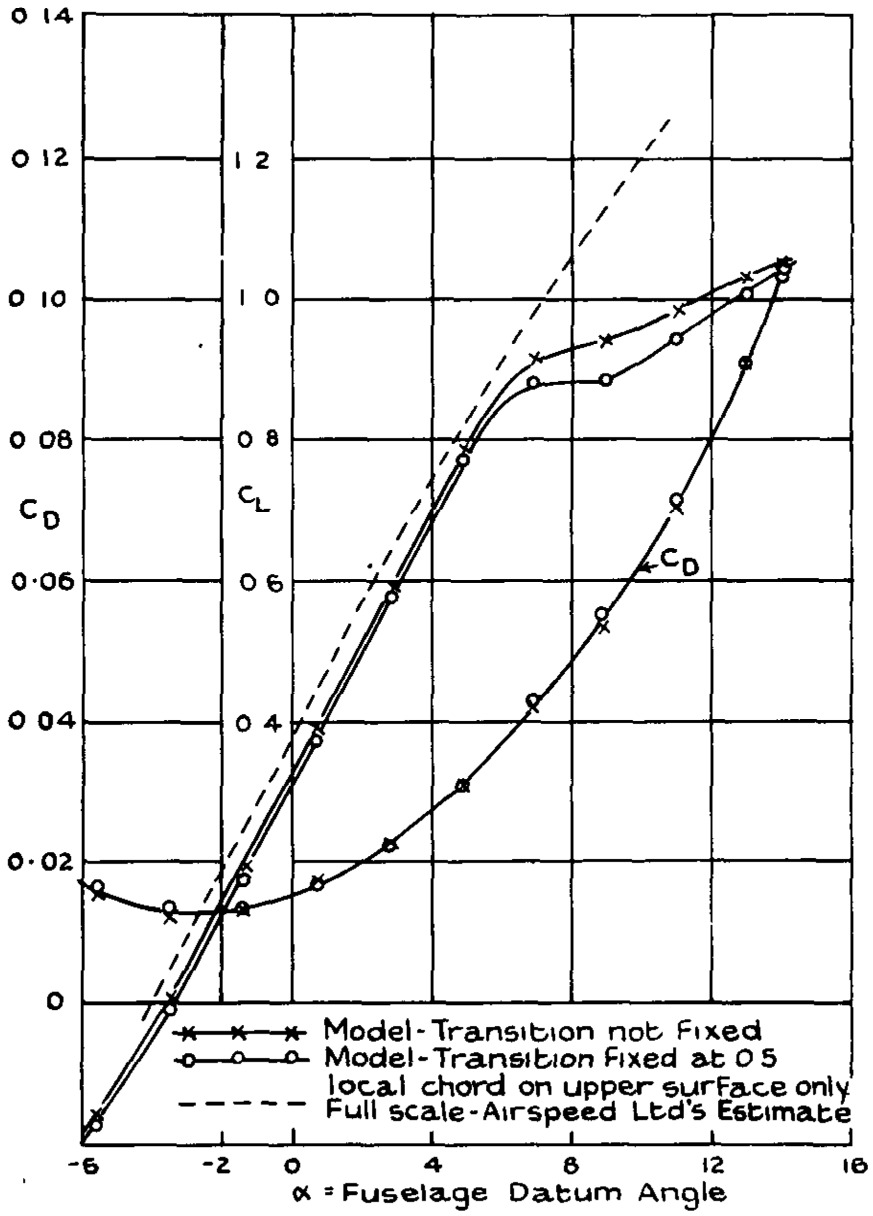
E.A.S



Shaded areas represent the high surface friction areas as indicated by the china clay - nitrobenzene technique
Figures (0 89 etc) give distance of boundary from L E in terms of the local chord
-x-x-x- Approximate position of laminar breakaway as indicated by the lead acetate - H_2S technique

Exploration of flow over plain model wing

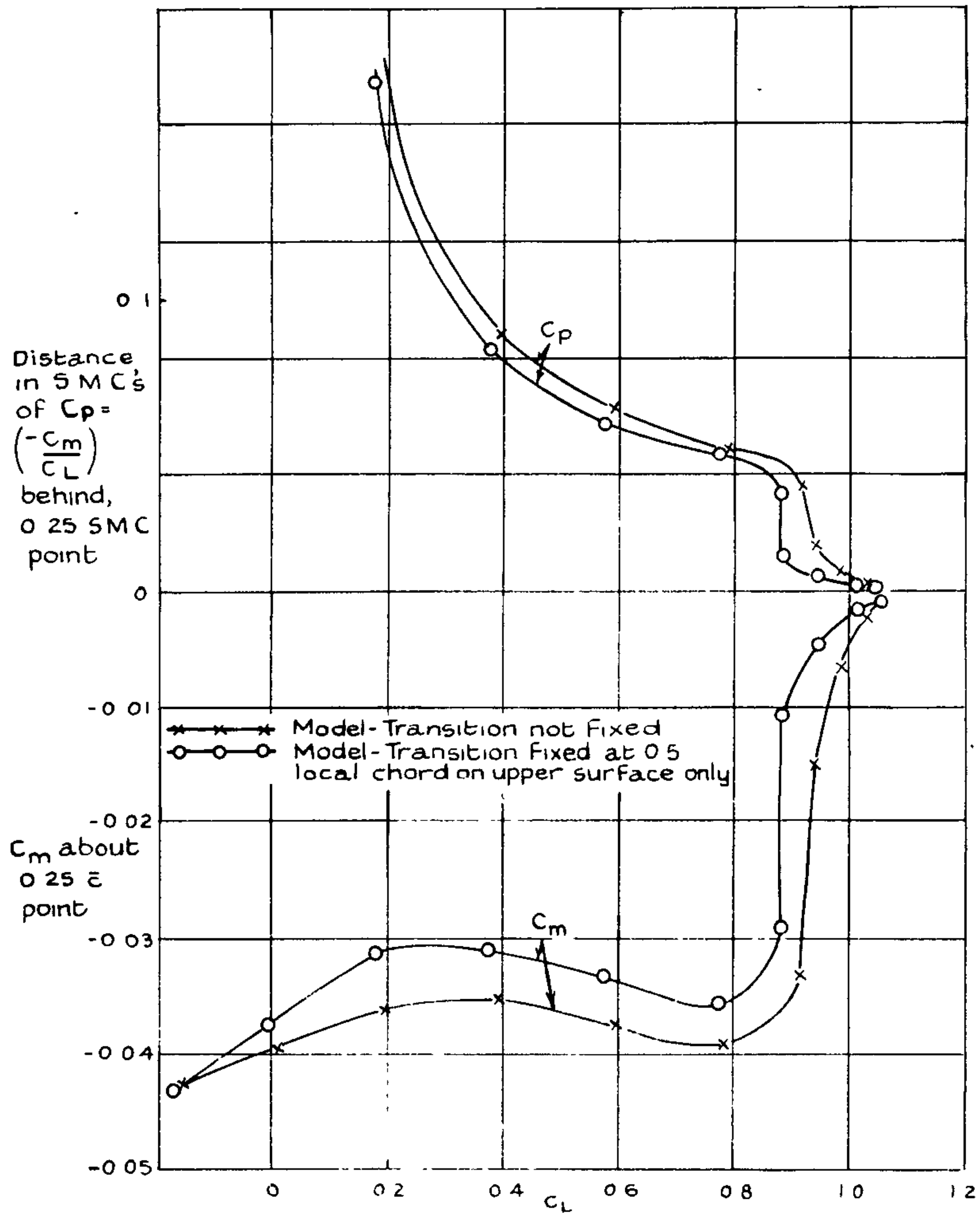
EPS



$\frac{dC_L}{d\alpha}$ { Model ($-0.1 < C_L < 0.75$) = 0·0925 per degree
Full Scale ($0.2 < C_L < 0.6$) = 0 0955 per degree
Firm's Estimate

Lift and Drag against α° (Fuselage Datum)

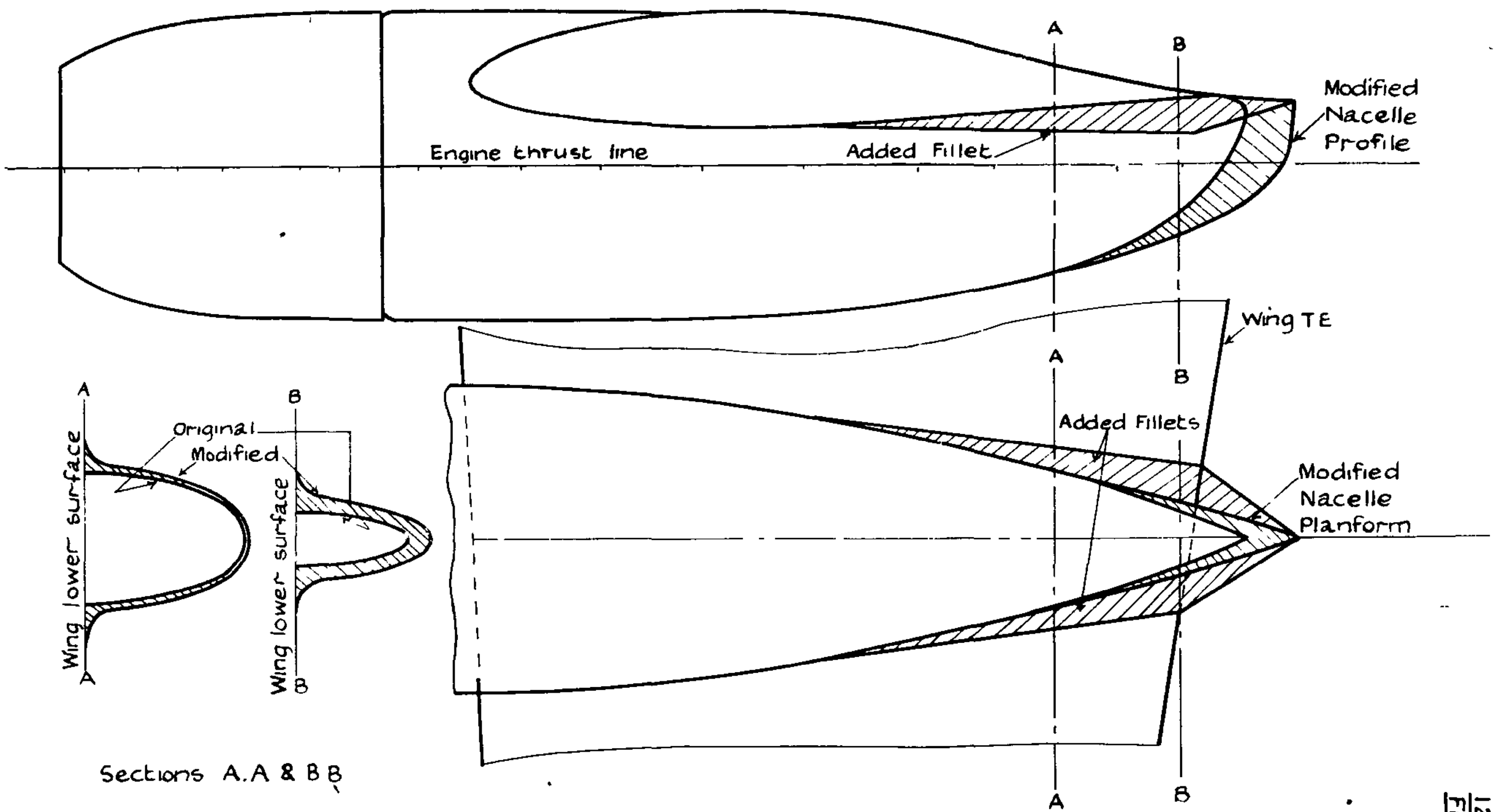
Effect of Fitting Transition wire at 0 5 chord on upper surface of wing - (Wing alone tests)



Pitching moment and Centre of Pressure against Lift

Effect of Fitting Transition wire at 0 5 Chord on Upper Surface of Wing -(Wing alone Tests)

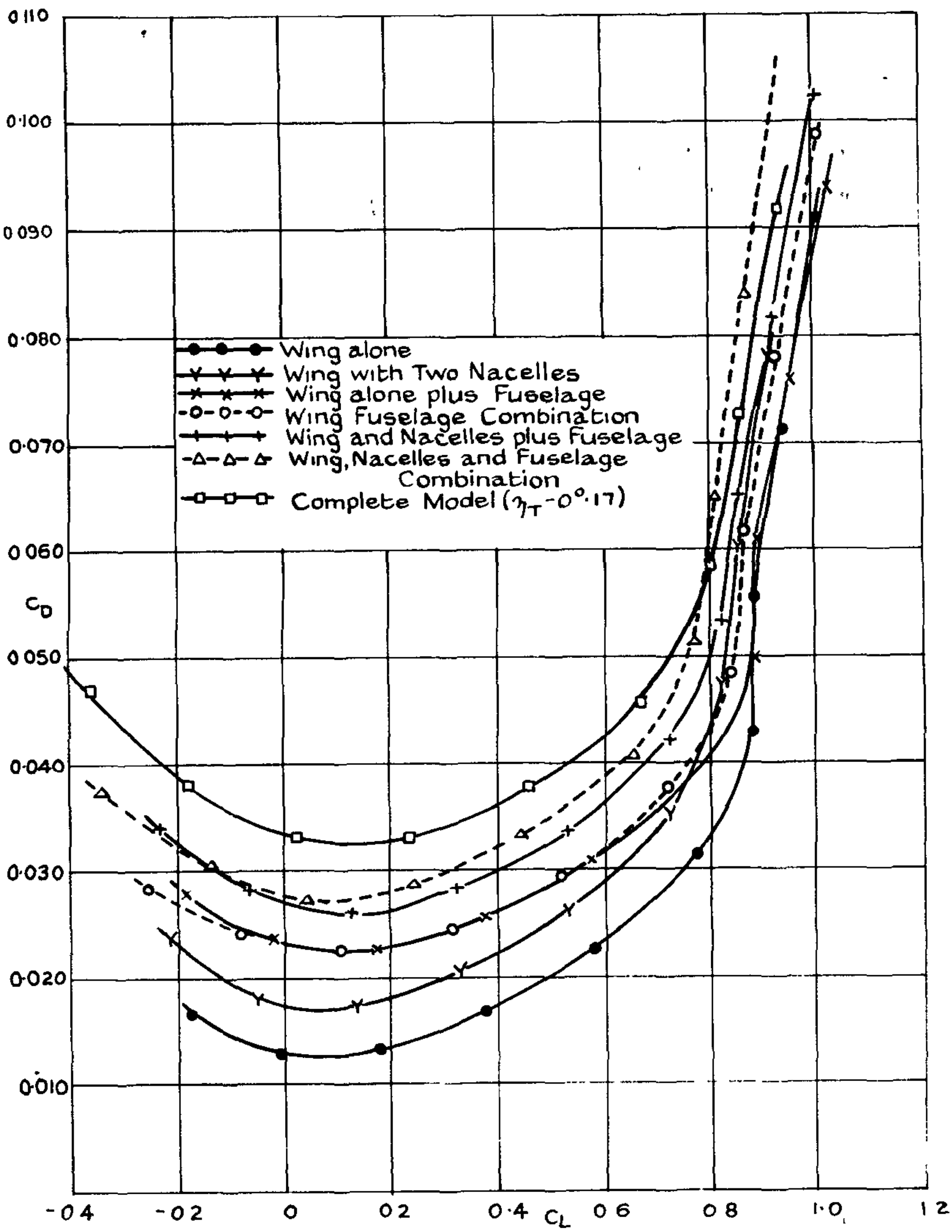
PH



Modifications made to Nacelle Shape

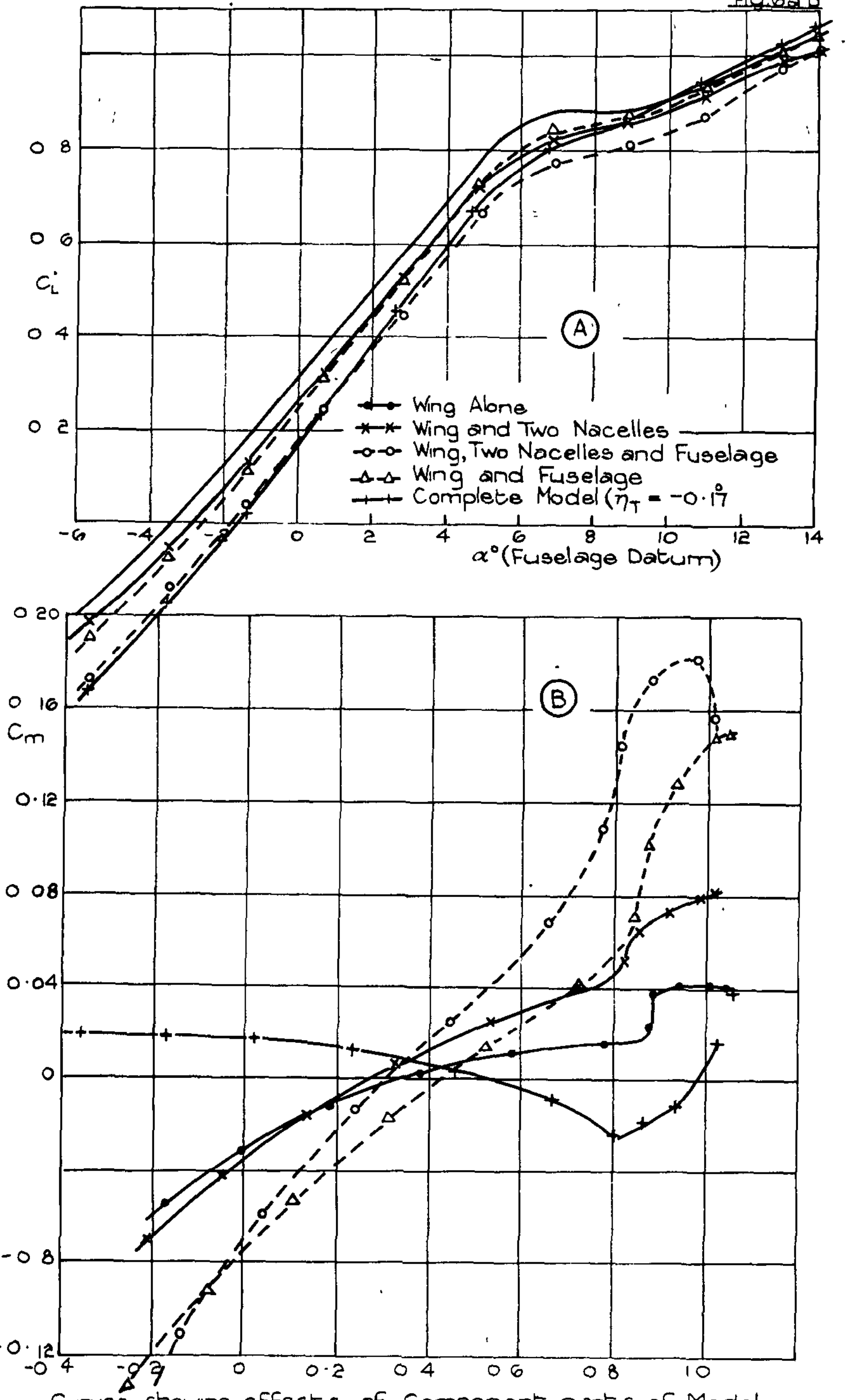
12,058
Fig 4

12,058
FIG 5

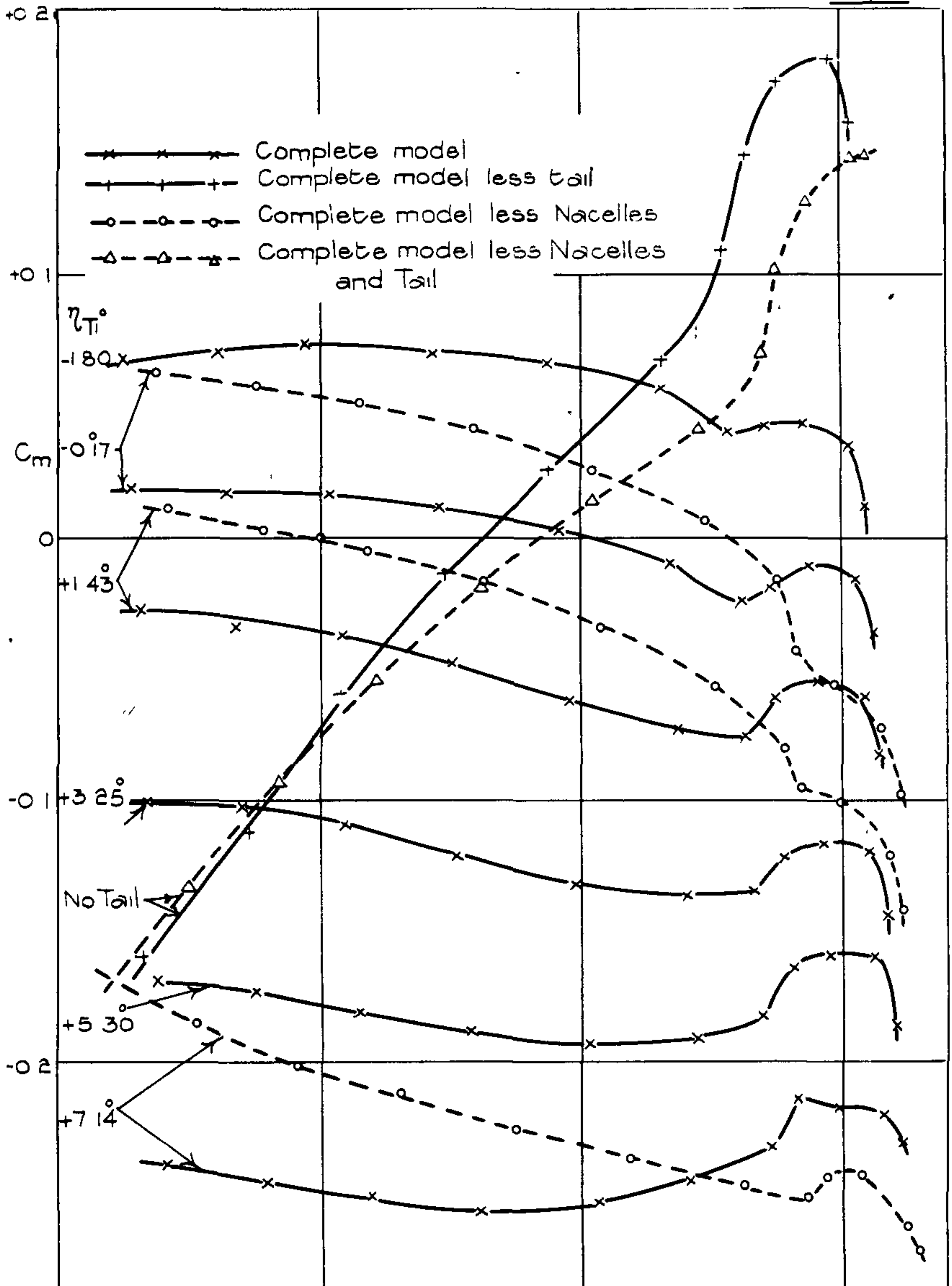


Drag against Lift showing Interference Effects of Nacelles and Fuselage

RM

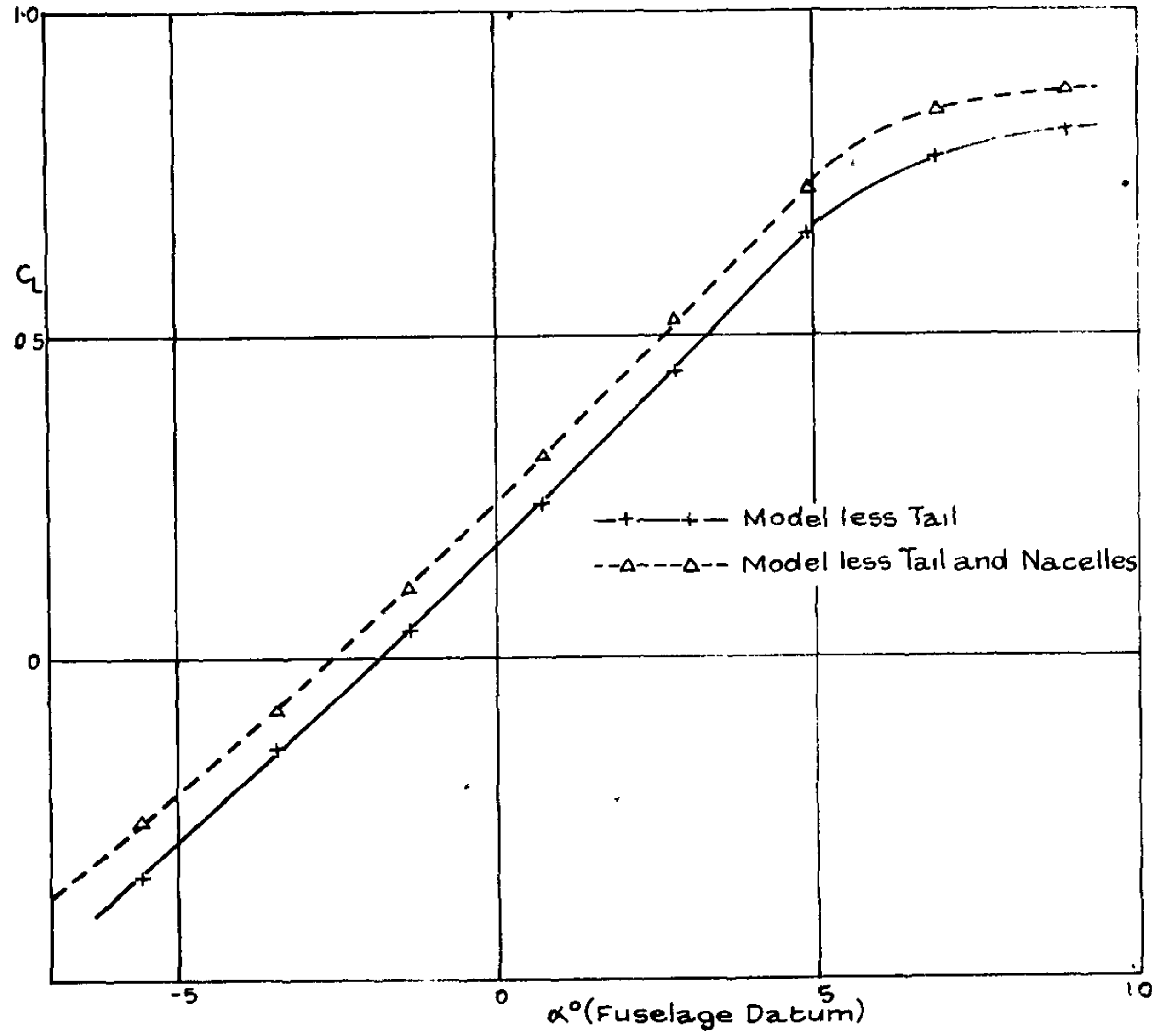


Curves showing effects of Component parts of Model
(A) On Lift (B) On Pitching Moment



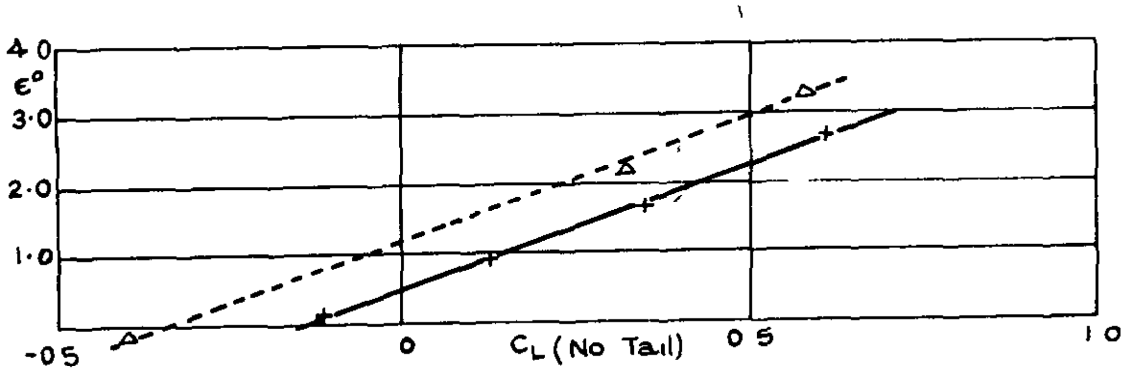
Pitching Moment against Lift for various Tailplane settings with and without nacelles No Propellers

8R



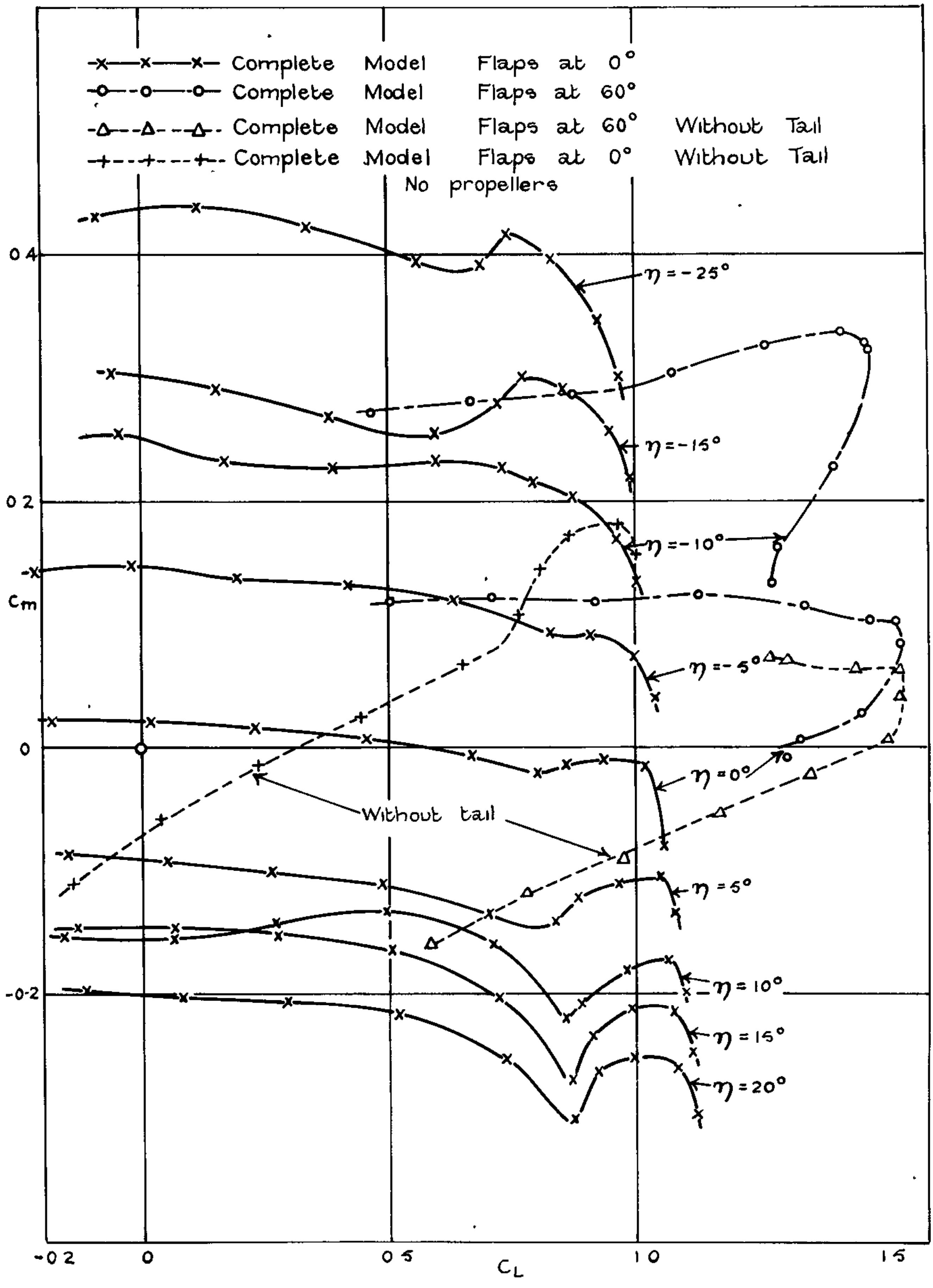
Lift against Fuselage Datum Angle for model without tail

Fuselage Datum	C_L (No Tail)	η_T°	ϵ°	
4 45	0·610	-1·8	2·65	Complete Model
1 80	0 346	-0 17	1 63	
-0 55	0 119	+1·43	0 88	
-3·10	-0·118	3 25	0 15	
3·45	0 580	-0·17	3·28	Model less Nacelles
0·75	0·320	+1 43	2 18	
-7·30	-0·396	+7 14	-0·16	



Downwash with Tail against Lift without Tail

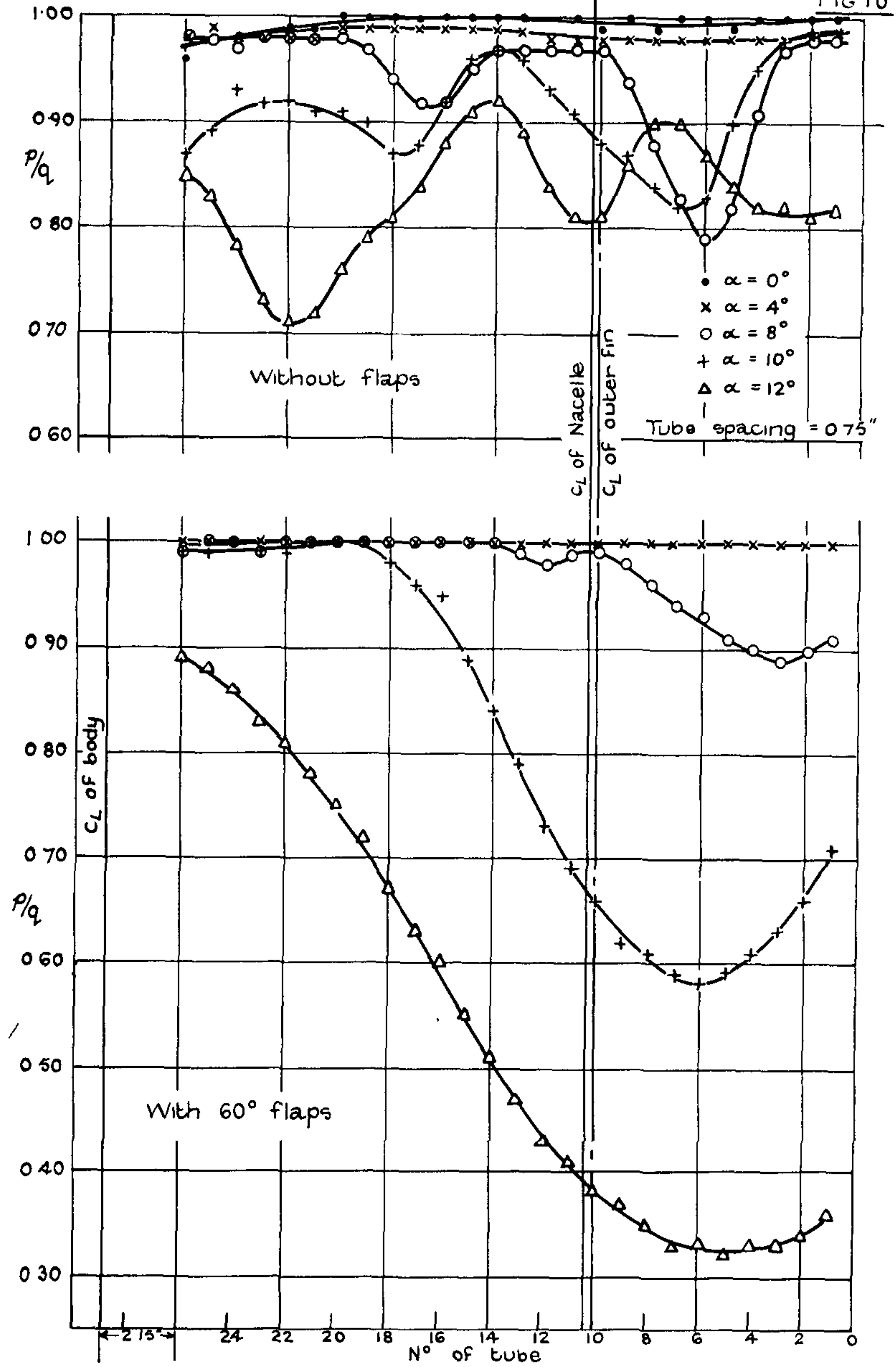
PM



Pitching Moment against Lift for Different Elevator Angles

$\eta_T = -0.17°$ to Fuselage Datum

EPS



$p/q = \dfrac{\text{Total head in position explored}}{\text{Free stream total head}}$

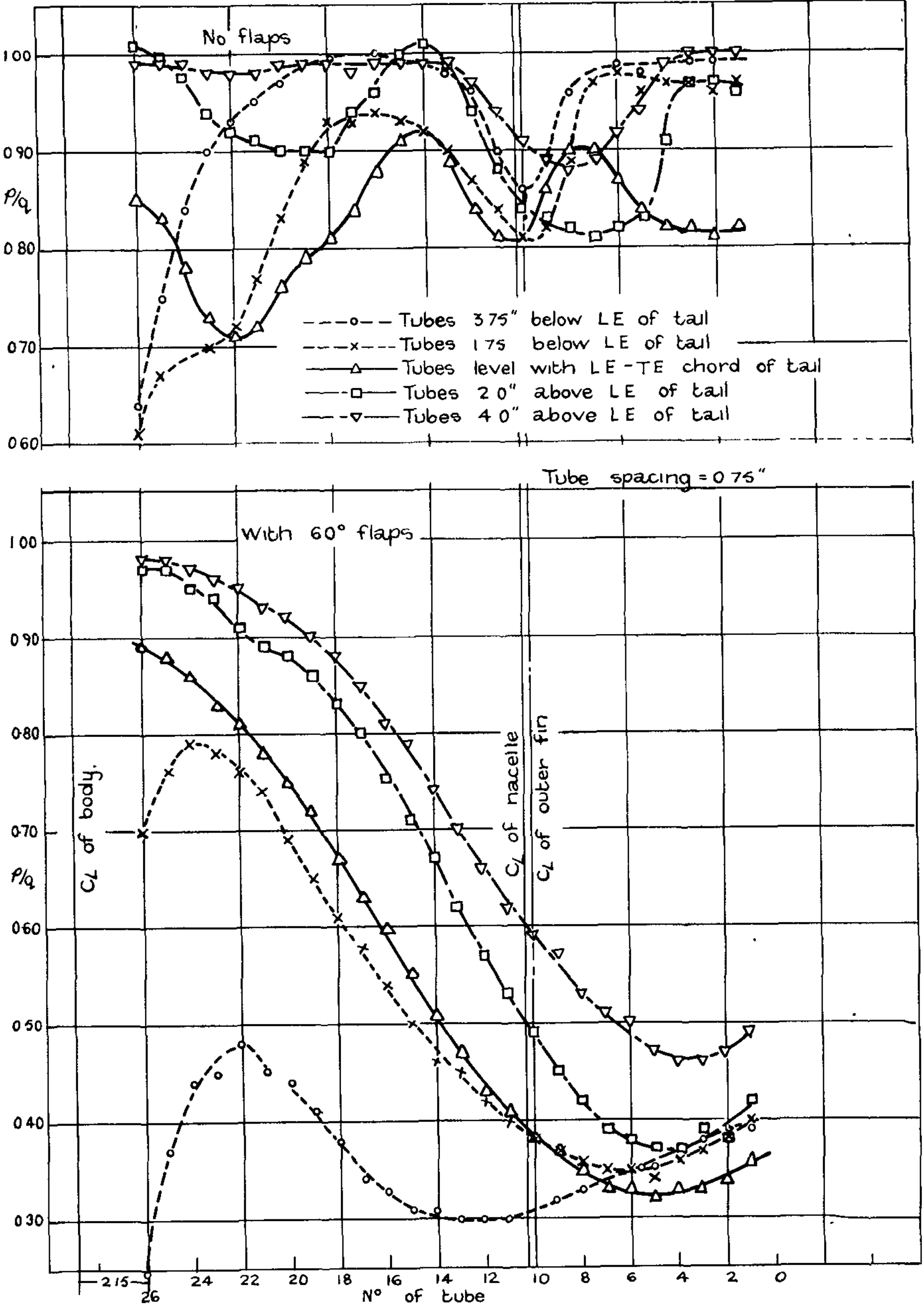
Tubes in the plane of the tailplane Mouth of tubes at 1/4 chord point

Variation of p/q with α

Tests on a 1/12TH Scale Model of the A S 60. Total Head Distribution in Region of Tailplane Position

EPS

12,058.
FIG 11

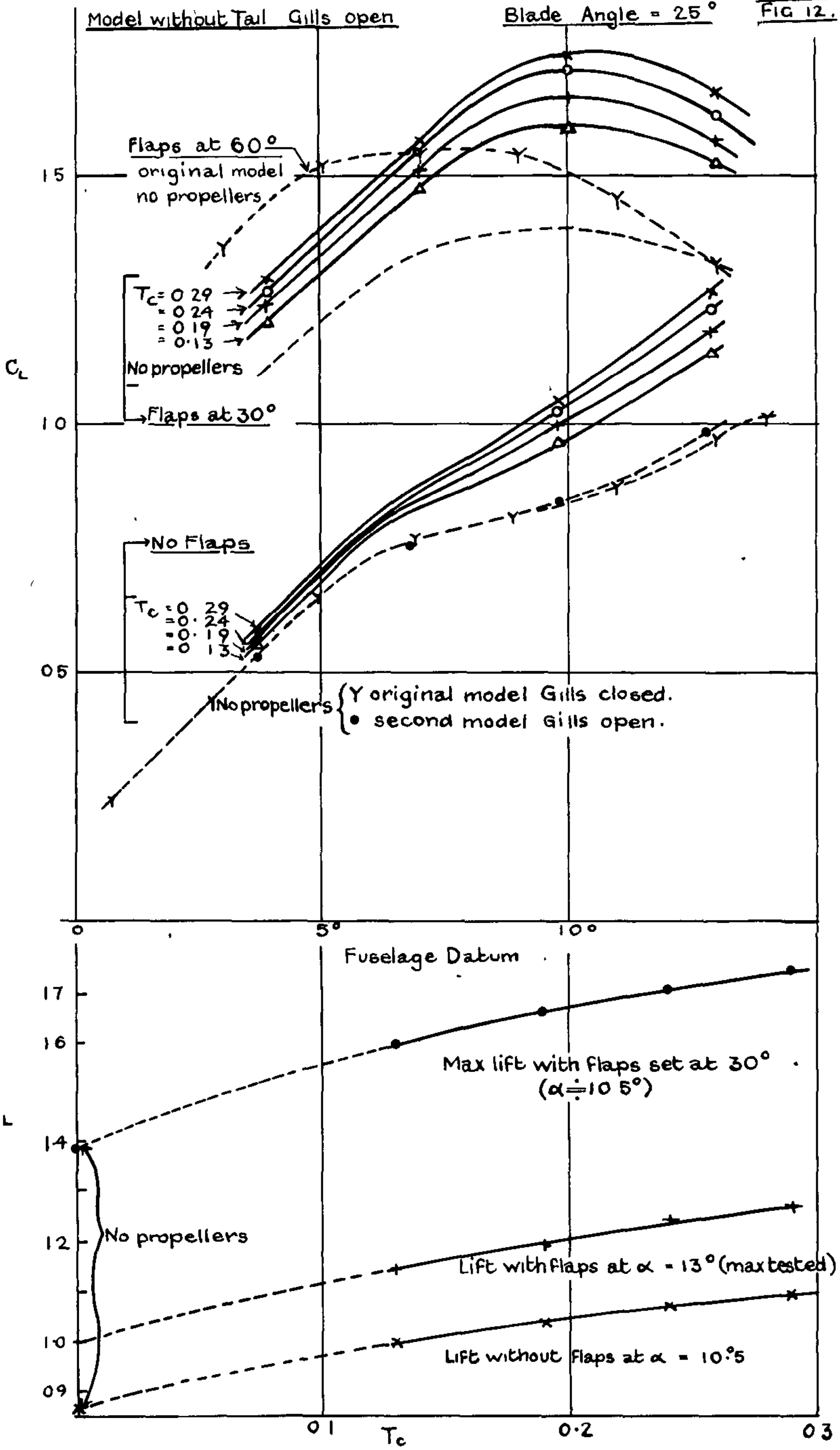


Variation of p/q at $\alpha = 12°$

where $p/q = \dfrac{\text{Total head in position explored}}{\text{Free stream total head}}$

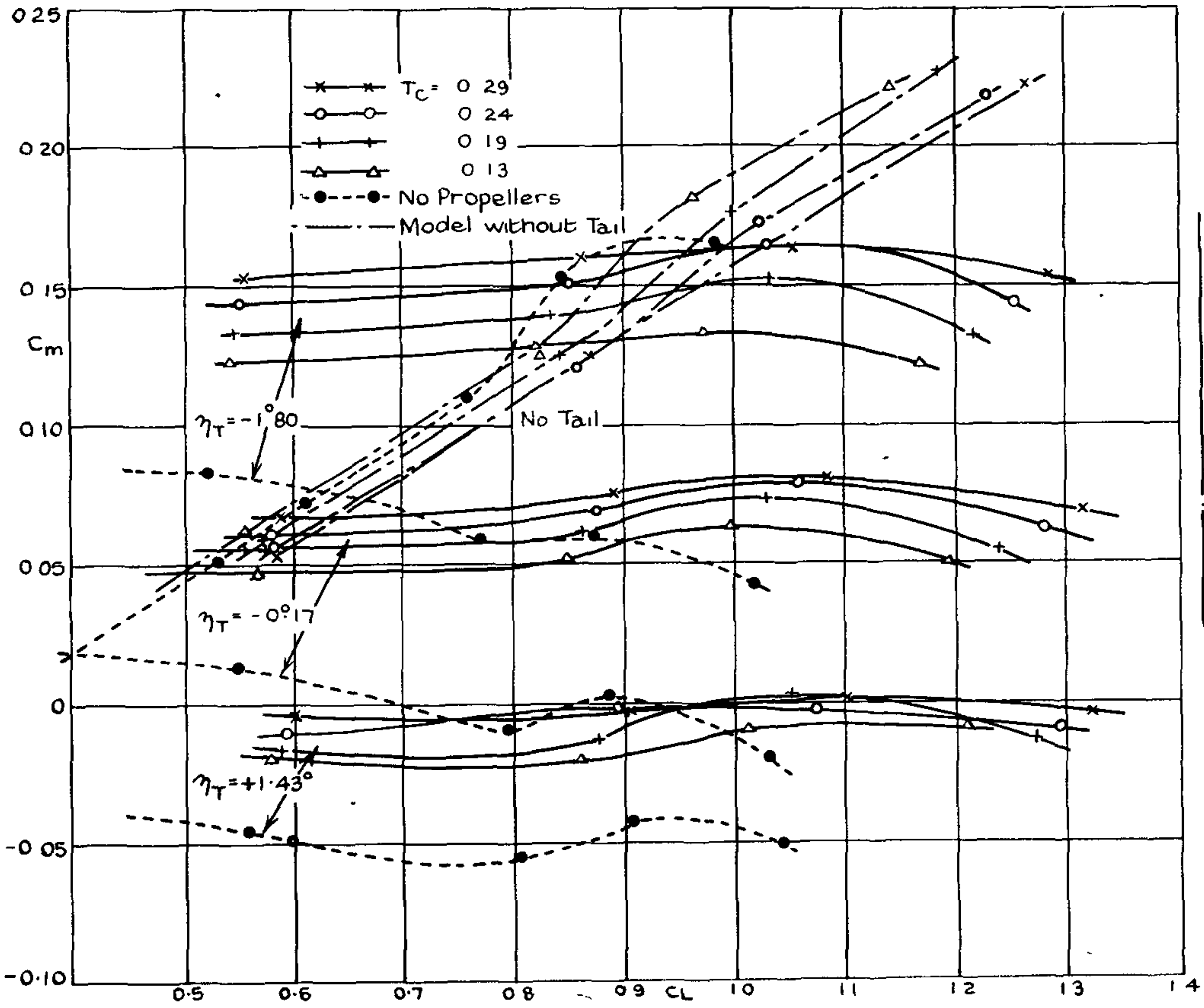
Tests on a 1/12TH Scale Model of the AS 60 Total head Distribution in Region of Tailplane Position

EPS



Effect of Slipstream and Split Flaps on Lift.

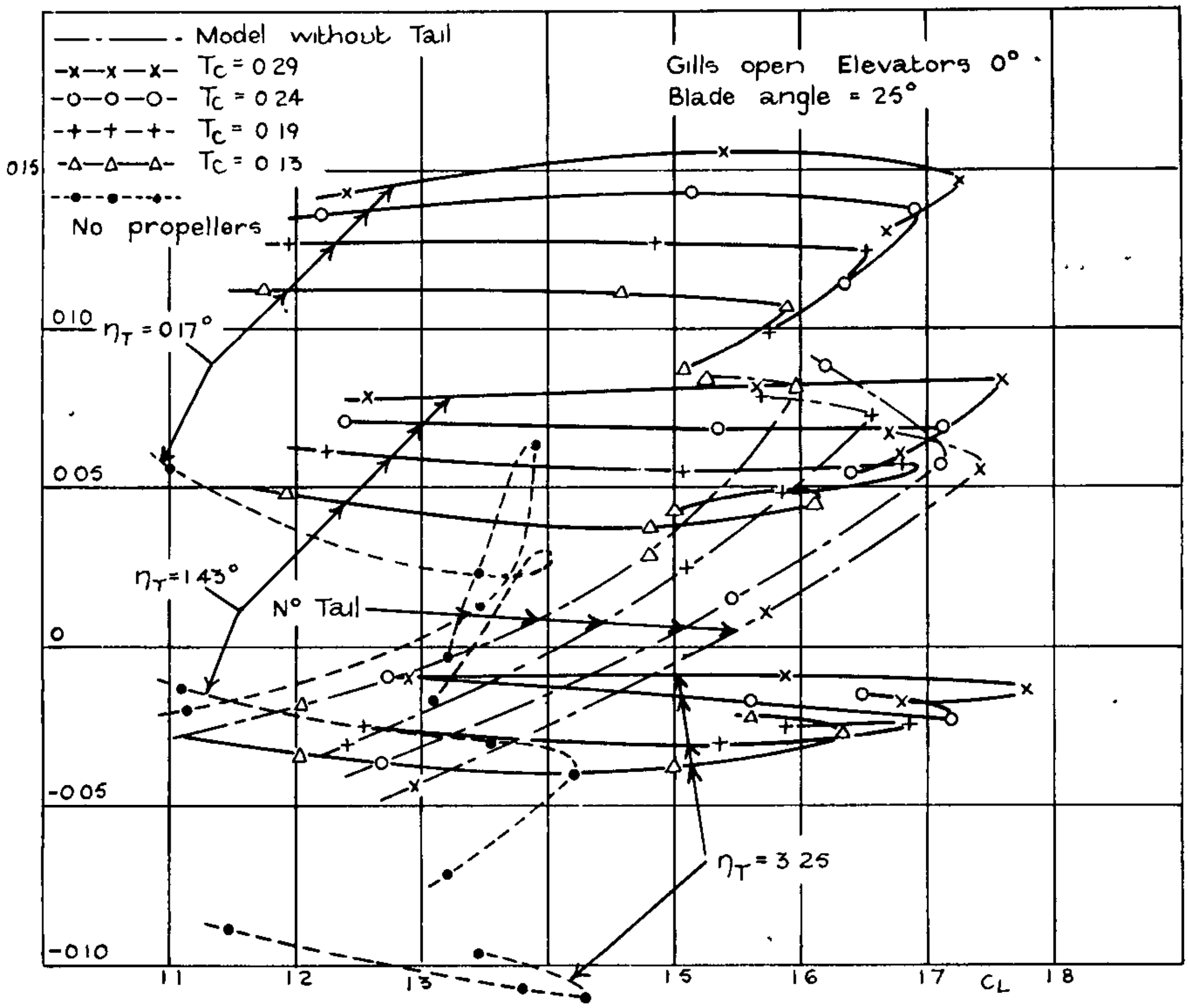
Pitching Moment against Lift for Various Tail plane settings with and without slipstream



No Flaps Gills open Elevators at 0° Blade Angle = 25°

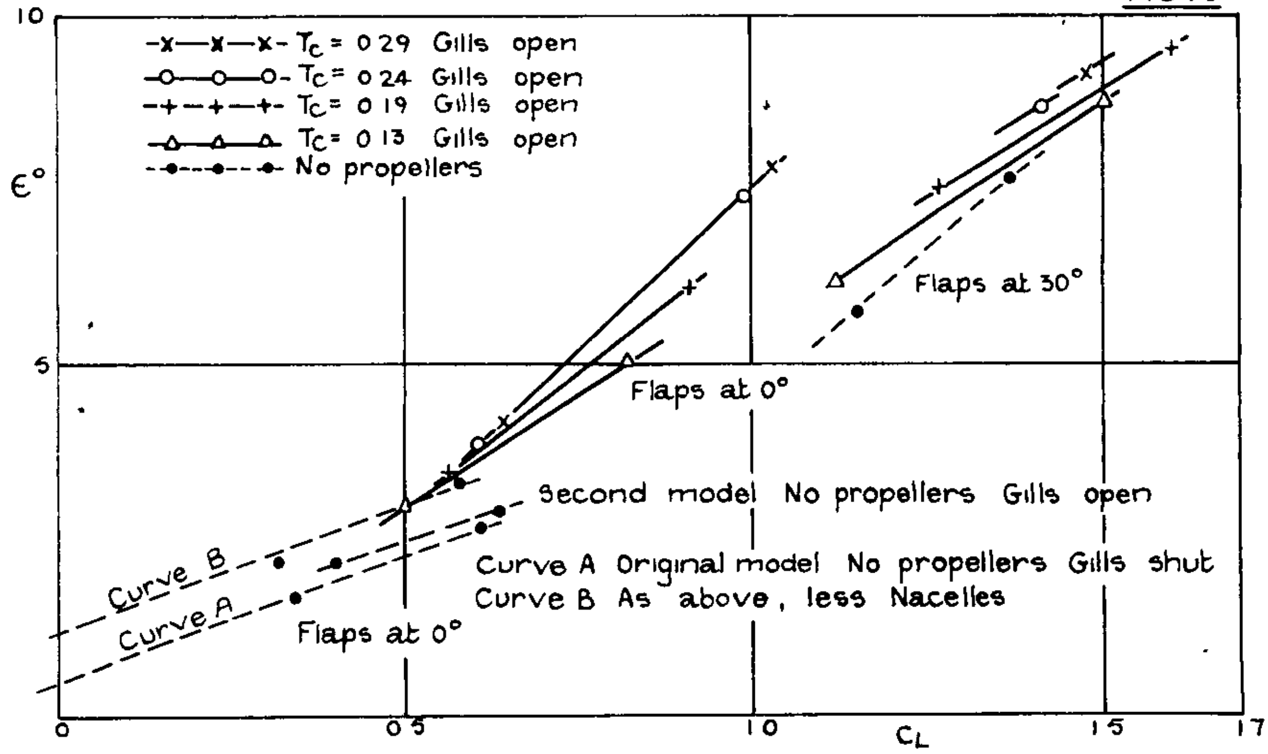
12,058
FIG 13

FIG 14



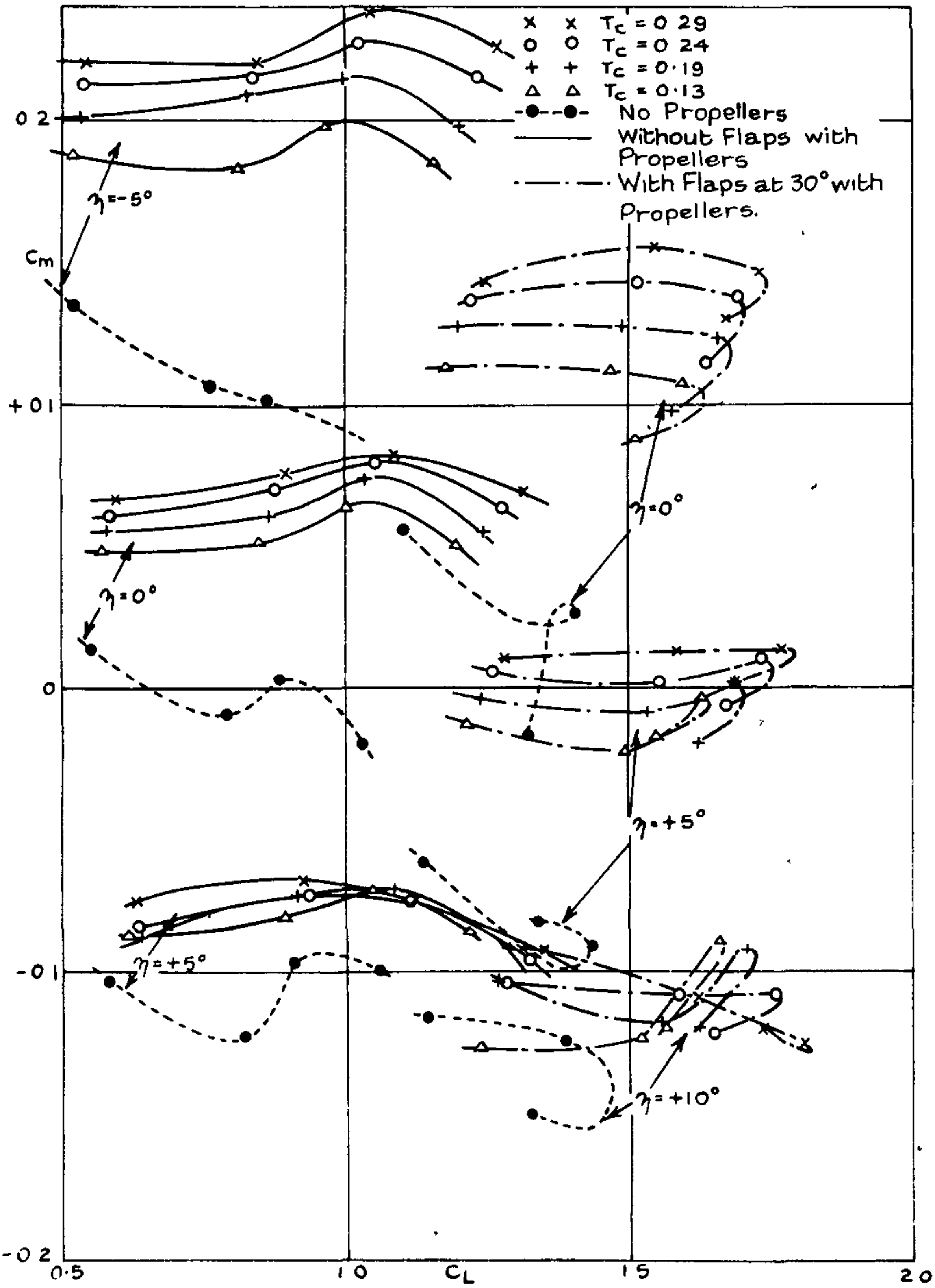
Pitching moment against Lift, with Flaps set at 30°, for various tail plane settings, with and without slipstream

FIG 15



Angle of Downwash at Tail (ε°) against Lift without tail

EPS

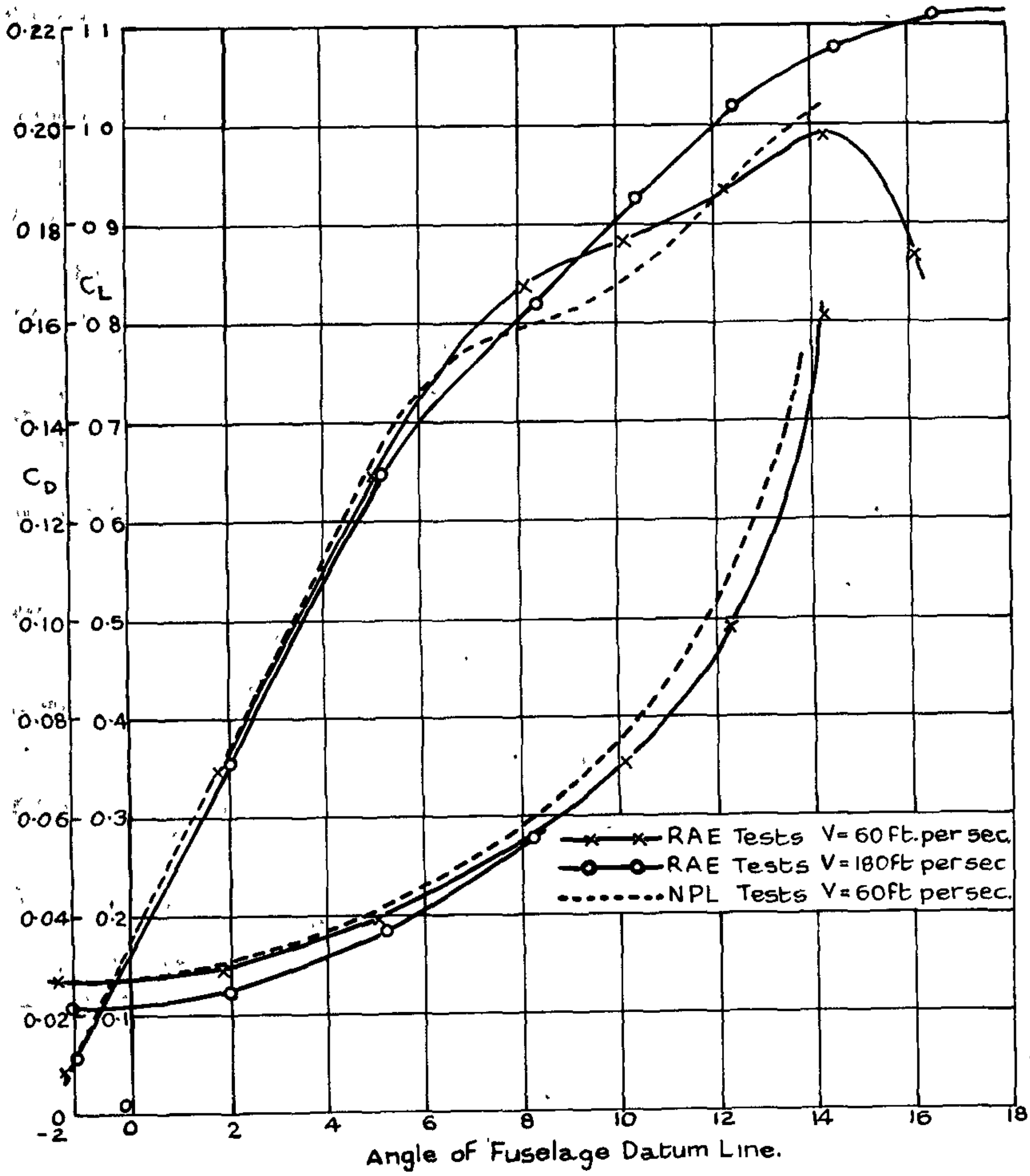


Pitching Moment against Lift for Various Elevator Angles with and without Flaps

η_T = -0°17   Gills open   Blade Angle = 25°

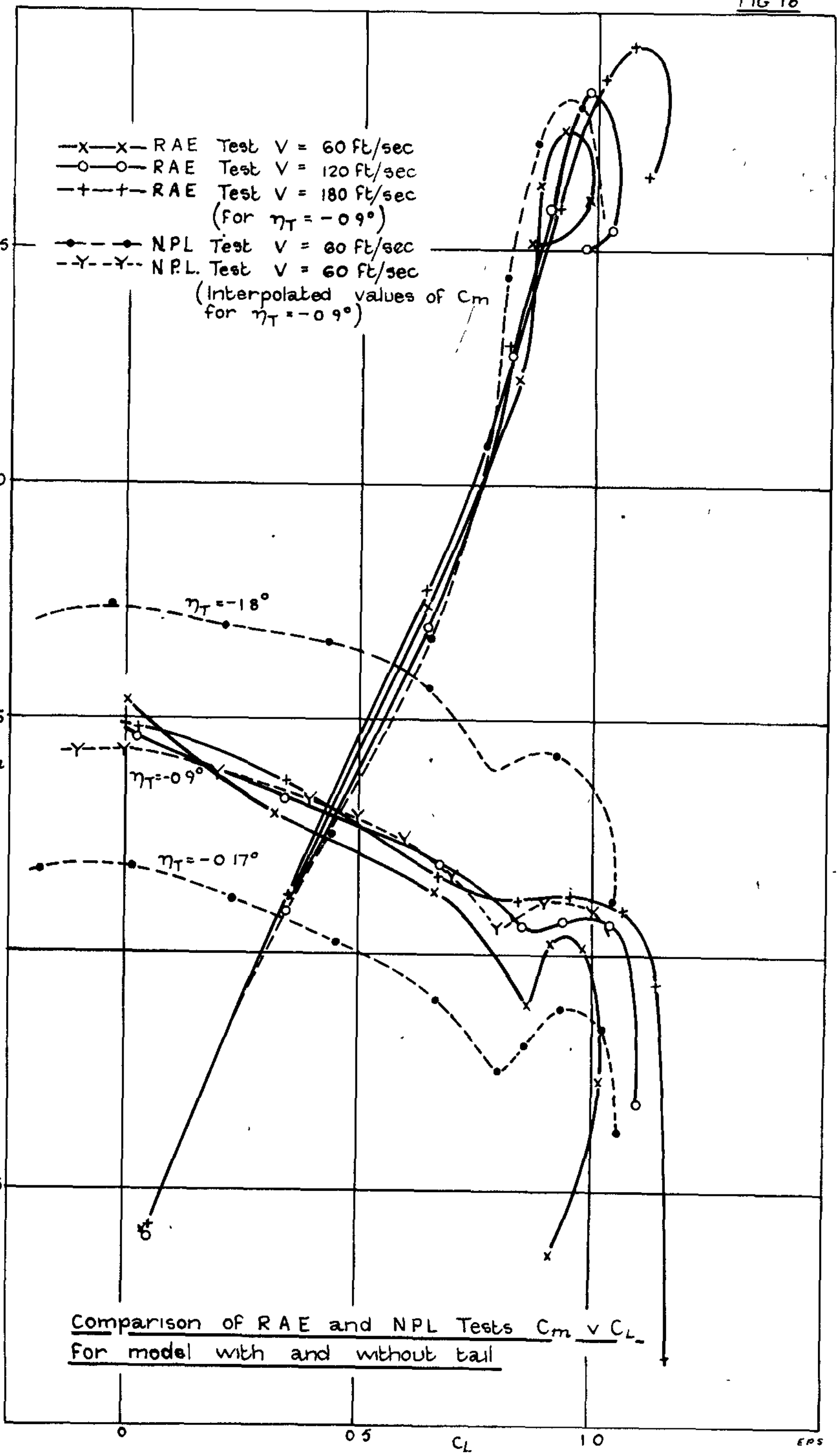
PM

12,058.
FIG.17



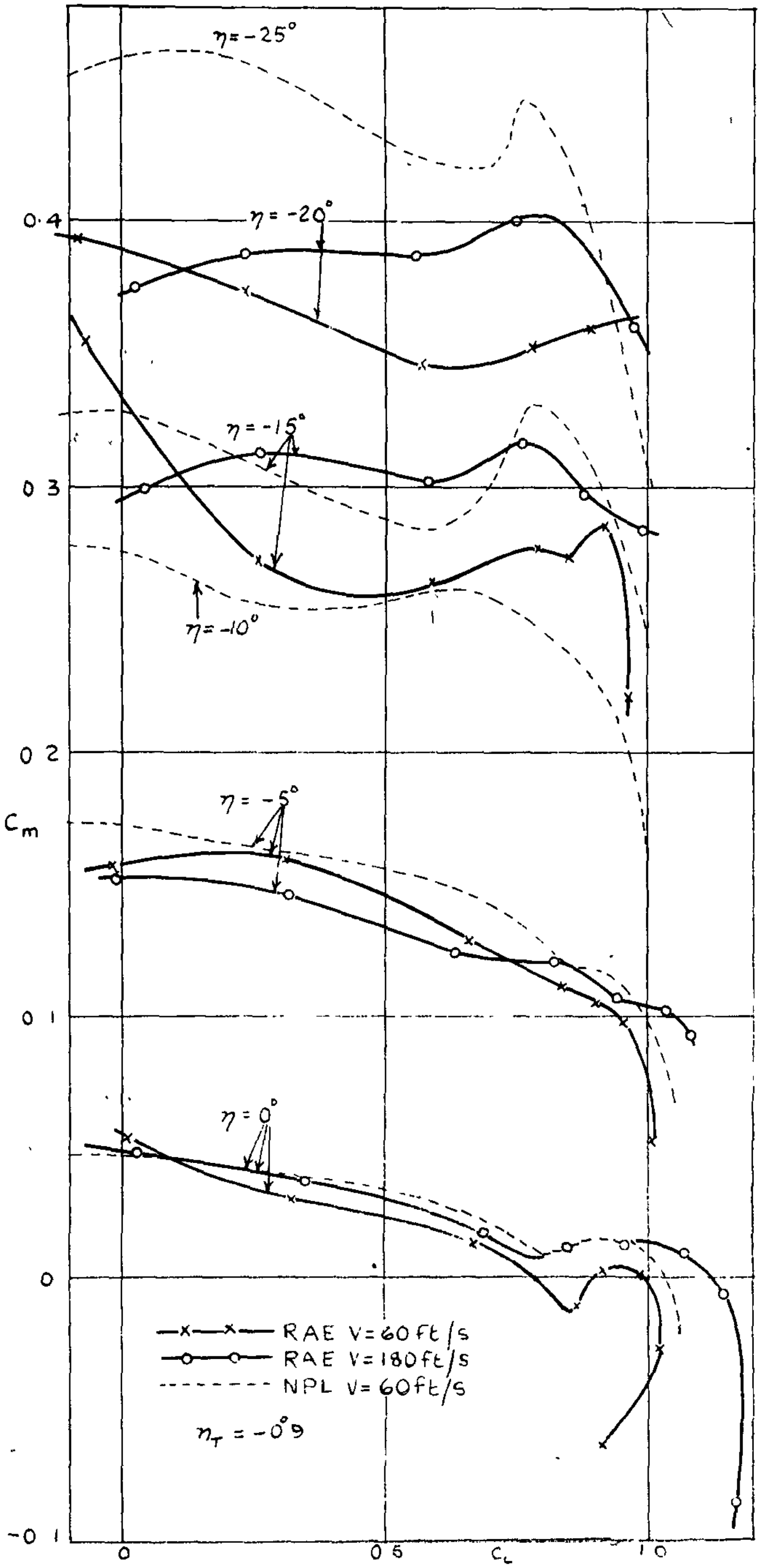
Comparison between RAE and NPL tests.
C_L and C_D v $\alpha°$ (Fuselage Datum Angle)
Model without empennage

PM.



Comparison of R A E and N P L Tests C_m v C_L for model with and without tail

12,058
FIG 19



Comparison of RAE with NPL tests
C_m v C_L for various elevator settings

C.P. No. 17
12058
A.R.C. Technical Report

PRINTED AND PUBLISHED BY HIS MAJESTY'S STATIONERY OFFICE
To be purchased from
York House, Kingsway, LONDON, W.C.2 429 Oxford Street, LONDON, W.1
P.O. Box 569, LONDON, S.E.1
13a Castle Street, EDINBURGH, 2 1 St. Andrew's Crescent, CARDIFF
39 King Street, MANCHESTER, 2 1 Tower Lane, BRISTOL, 1
2 Edmund Street, BIRMINGHAM, 3 80 Chichester Street, BELFAST
or from any Bookseller

1950

Price 3s. 6d. net.

PRINTED IN GREAT BRITAIN

S.O. Code No. 23-9006-17