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High Reynolds Number Tests on an Unswept 11% Thick RAE 101 Section Aerofoil

by

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SUMMARY

Tests were made in the Compressed Air Tunnel at N.P.L. on an 11% RAE 101 wing of aspect ratio 6. Measurements included lift, drag and pitching moments over a Reynolds number range from 0.7×10^6 to 7.7×10^6. Marked scale effects on maximum lift persisted up to a Reynolds number of about 4.5×10^6 and the stalling characteristics resembled those usually found on thinner wings.

Introduction

A lengthy programme of tests was planned on a wing with various arrangements of slotted flaps. Since the incremental components due to the flaps would form a convenient basis for comparing the merits of each flap configuration, the plain wing was tested before being cut to take flaps. Previous experience had shown that a re-assembled cut wing did not always reproduce exactly the aerodynamic characteristics of the original cut wing.

The Model

The wing was of steel construction and was made by Messrs. Vickers Armstrongs Ltd. It was rectangular in planform with a span of 48 in. and an aspect ratio of 6. The section was symmetrical (RAE 101) and had a maximum t/c ratio of 11% at 0.31c from the leading edge. The section co-ordinates are given in Table 1.

No measurements of surface roughness were made, but from visual inspection we should judge that the finish was of similar standard to a previous good quality steel aerofoil which had a centre-line-average value of roughness (B.S. 1134 : 1950) of 4 micro-inches. (Seemingly equally good specimens of wings, one made of aluminium and the other surfaces by an epoxy casting resin gave measurements of 13 and 17 micro-inches respectively.)

It was intended that this should be the basic wing to which slats and flaps would be fitted later. Consequently, substantial support fittings were essential to withstand the high loading expected on the flapped wing. These fittings were incorporated in the model for attachment to the pair of normal overhead supports used in the Compressed Air Tunnel, and to a third rod centrally placed further back. Fig.1 shows the method of holding the wing. The forward fittings were shaped to conform to the aerofoil section, apart from the slots needed to permit the incidence to be changed. Strength and thickness considerations did not allow the rear support pin to be sunk into the wing, and so a protruding fitting of the type shown was unavoidable. Had the prime object of the tests been the measurement of the absolute characteristics of the basic wing, a less obtrusive rear fitting would have been used with advantage.

Presentation/

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Presentation of Results

Table 2 gives the results of the tests. Normal corrections have been applied for wind tunnel interference and special corrections applied for the effects of balance deflection under load. No attempt has been made to convert the results to correspond to infinite aspect ratio conditions, but should this be necessary the numerical corrections due to Glauert would apply, viz.,

$$\alpha = \alpha_o + 3.55^\circ C_L$$

$$C_D = C_{Do} + 0.0555 C_L^2$$

Pitching moment coefficients are quoted about the quarter-chord position. The lift, drag and pitching moment curves are shown in Figs.2 to 5 and the effects of Reynolds number on maximum C_L and minimum C_D in Figs.6 and 7.

Discussion of Results

The lift curves in Fig.2 are noteworthy for the sharpness of the stall, a characteristic more usually found on thinner wings. Loftin and Bursnall[1] have described how, prior to the stall, the laminar boundary layer separates near the nose but manages to reattach after transition. At the stall, the high suction peak suddenly prevents reattachment, leaving complete laminar separation from the nose with a catastrophic loss of lift. This is consistent with the sudden loss of positive pitching moment shown in Figs.4 and 5 when the suction peak near the nose of the aerofoil collapses at the stall.

Although there is virtually no change of $C_{L_{max}}$ from $R \simeq 5 \times 10^6$ to 8×10^6 (see Fig.6), it would not be unreasonable, in view of the work of Loftin and Bursnall, to expect a subsequent slight rise in $C_{L_{max}}$, accompanied by a more gradual stall at Reynolds numbers of about twice these values. For these extremely high Reynolds numbers, transition would occur ahead of the position from which the boundary layer separated at the highest Reynolds numbers of our tests. The resulting turbulent boundary layer would not separate from the nose initially, but from the trailing edge, and the stall would be the result of this separation line moving forward gradually with increasing incidence. It must be remembered that the turbulence level in the Compressed Air Tunnel, at the lower Reynolds numbers, is much higher than in the Langley Two-dimensional Pressure Tunnel in which Loftin and Bursnall made their measurements. Recent measurements of turbulence in the Compressed Air Tunnel have indicated that the values of the three turbulent components lie within the range 0.2% to 0.45% up to 7.5 atmospheres working pressure ($R \simeq 4 \times 10^6$ per foot). At higher pressures, the high wind loading of the hot wires used often caused them to vibrate and break, and the measurements have been temporarily abandoned. The indications were, however, that the turbulence level showed no tendency to increase at higher pressures. In the Langley tunnel, the turbulence level, at 4 atmospheres working pressure, increased from about 0.03% to 0.15% as the tunnel power increased to the maximum. At the 10 atmospheres and full tunnel power necessary to attain the highest Reynolds numbers of Ref.2 it is likely that the turbulence level was approaching the same order as that in the Compressed Air Tunnel. The higher turbulence level in the Compressed Air Tunnel would tend to favour any tendency for transition to occur rather than laminar separation and so the present results in the tunnel might be expected to compare with those for slightly higher Reynolds numbers in free flight or less turbulent tunnels.

The slight loss of maximum lift at the highest Reynolds number, (Fig.6), usually indicates that the aerofoil was just beginning to bend under load, and in more serious cases the results at the next lower Reynolds number are usually accepted with more confidence.

The/

The value of the lift slope, above the critical Reynolds number, was 4.29 per radian, which compares quite well with a value of 4.24 estimated, for aspect ratio 6 in the absence of sweep back, from an expression due to Garner[2].

The drag curves (Fig.3) from about 4° to 17°, at the highest Reynolds numbers, can be represented by the expression

$C_D = 0.0060 + \frac{1.170}{\pi A} C_L^2$ (where A is the aspect ratio). The curves are not quite symmetrical about zero incidence, and at low incidences, the curves for different Reynolds numbers cross over. This could be through lack of symmetry in the model but is more likely due to the protruding rear fitting on the lower surface (uppermost in the tunnel). At small negative incidences, and for the lowest Reynolds numbers of the test, the boundary layer in the region of this fitting would have a thickness of the same order as the height of the fitting. Thus the drag of the fitting would be small, the fitting being mostly in a region of reduced velocity. Increase of Reynolds number would reduce the boundary layer thickness and cause the drag of the fitting to rise asymptotically to the free stream value. At slightly more positive incidences, the fitting would be in a very thin boundary layer over the whole Reynolds number range and the drag of the fitting would be nearly constant. This explanation would account for the juxtaposition of the drag curves at low incidences and should help when deciding which portions of the drag curve are least affected by the rear support fitting.

From the curves of Fig.5, values of $\frac{dC_m}{d\alpha}$ at zero lift for the higher Reynolds numbers locate the aerodynamic centre at 0.235c.

Conclusion

The main features of the results of these tests are the "thin aerofoil" stalling characteristics and the scale effect on $C_{L_{max}}$ which extends up to Reynolds numbers of about 4.5×10^6.

References

No.	Author(s)	Title, etc.
1	Laurence K. Loftin, Jnr. and William J. Bursnall	The effects of variations in Reynolds number between 3×10^6 and 25×10^6 upon the aerodynamic characteristics of a number of NACA 6-series airfoil sections. N.A.C.A. Tech. Note 1773, December, 1948.
2	H. C. Garner	Swept-wing loading. A critical comparison of four subsonic vortex sheet theories. A.R.C. C.P.102, October, 1951.

Table 1/

Table 1

Theoretical Co-ordinates for 11% RAE 101 Wing

Span 48 inches Chord 8 inches

x/c	y_{upper}/c	Distance from L.E. (inches)	Upper surface ordinate (inches)
0	0	0	0
0.001	0.0042955	0.008	0.0344
.005	.0095755	.040	.0766
.0125	.0150557	.100	.1204
.025	.0210914	.200	.1687
.05	.0292523	.400	.2340
.1	.0396803	.800	.3174
.15	.0486222	1.200	.3890
.2	.0509344	1.600	.4075
.25	.0537361	2.000	.4299
.3	.0549659	2.400	.4397
.35	.0544896	2.800	.4359
.4	.0528055	3.200	.4224
.45	.0502227	3.600	.4018
.5	.0469370	4.000	.3755
.55	.0431013	4.400	.3448
.6	.0388410	4.800	.3107
.65	.0342694	5.200	.2742
.7	.0294877	5.600	.2359
.75	.0245927	6.000	.1967
.8	.0196746	6.400	.1574
.85	.0147554	6.800	.1180
.9	.0098373	7.200	.0787
.95	.0049181	7.600	.0393
.975	.0024591	7.800	.0197
.9875	.0012295	7.900	.0098
.995	.0004918	7.960	.0039
.999	.0000984	7.992	.0008
1.0	0	8.000	0

Table 2/

Table 2

Results of Tests on 11% RAE 101 Wing of Aspect Ratio 6

Pitching Moment Coefficients given about $\frac{1}{4}$ Chord Position

P = 2.89 Atmos. $\frac{1}{2}\rho V^2$ = 12.24 lb/sq ft				P = 4.39 Atmos. $\frac{1}{2}\rho V^2$ = 24.44 lb/sq ft			
V = 60.55 F.P.S. R = 0.704 × 10^6				V = 69.4 F.P.S. R = 1.228 × 10^6			
$\alpha°$	C_L	C_D	C_M	$\alpha°$	C_L	C_D	C_M
- 2.95	-0.251	0.0087	-0.0122	- 2.95	-0.214	0.0097	-0.0106
- 1.7	- .162	.0069	- .0101	- 1.75	- .125	.0071	- .0093
- 0.45	- .072	.0056	- .0080	- 0.45	- .036	.0058	- .0075
+ 0.8	+ .022	.0060	- .0060	+ 0.8	+ .056	.0058	- .0065
2.05	.110	.0071	- .0043	2.0	.148	.0068	- .0045
4.5	.285	.0130	- .004	3.25	.237	.0086	- .0024
6.95	.466	.0226	+ .0031	4.5	.323	.0119	- .0009
9.4	.645	.0381	.0036	5.7	.408	.0161	+ .0012
10.65	.724	.0468	.0047	6.95	.500	.0216	.0026
11.25	.763	.0513	.0064	8.15	.590	.0279	.0025
11.9	.795	.0560	.0075	9.4	.679	.0358	.0015
12.5	.825	.0610	.0087	10.6	.770	.0446	.0026
12.7	.760	.109	- .0121	11.85	.853	.0536	.0044
13.2	.752	.129	- .0250	13.1	.929	.0636	.0057
14.5	.703	.170	- .0536	14.35	1.000	.0759	.0061
17.15	.601	.214	- .0765	14.65	1.011	.0787	.0055
19.75	.593	.253	- .0855	14.95	1.016	.0821	.0046
22.3	.611	.296	- .0542	15.3	1.036	.0850	.0049
24.9	.636	.339	- .0494	15.5	.649	.196	- .0689
				15.9	.644	.199	- .0734
				17.15	.616	.220	- .0817
				18.45	.608	.239	- .0856
				19.75	.611	.258	- .0884
				21.0	.620	.281	- .0942
				22.3	.626	.301	- .0953
				23.5	.638	.323	- .0975
				24.9	.655	.350	- .1015

continued/

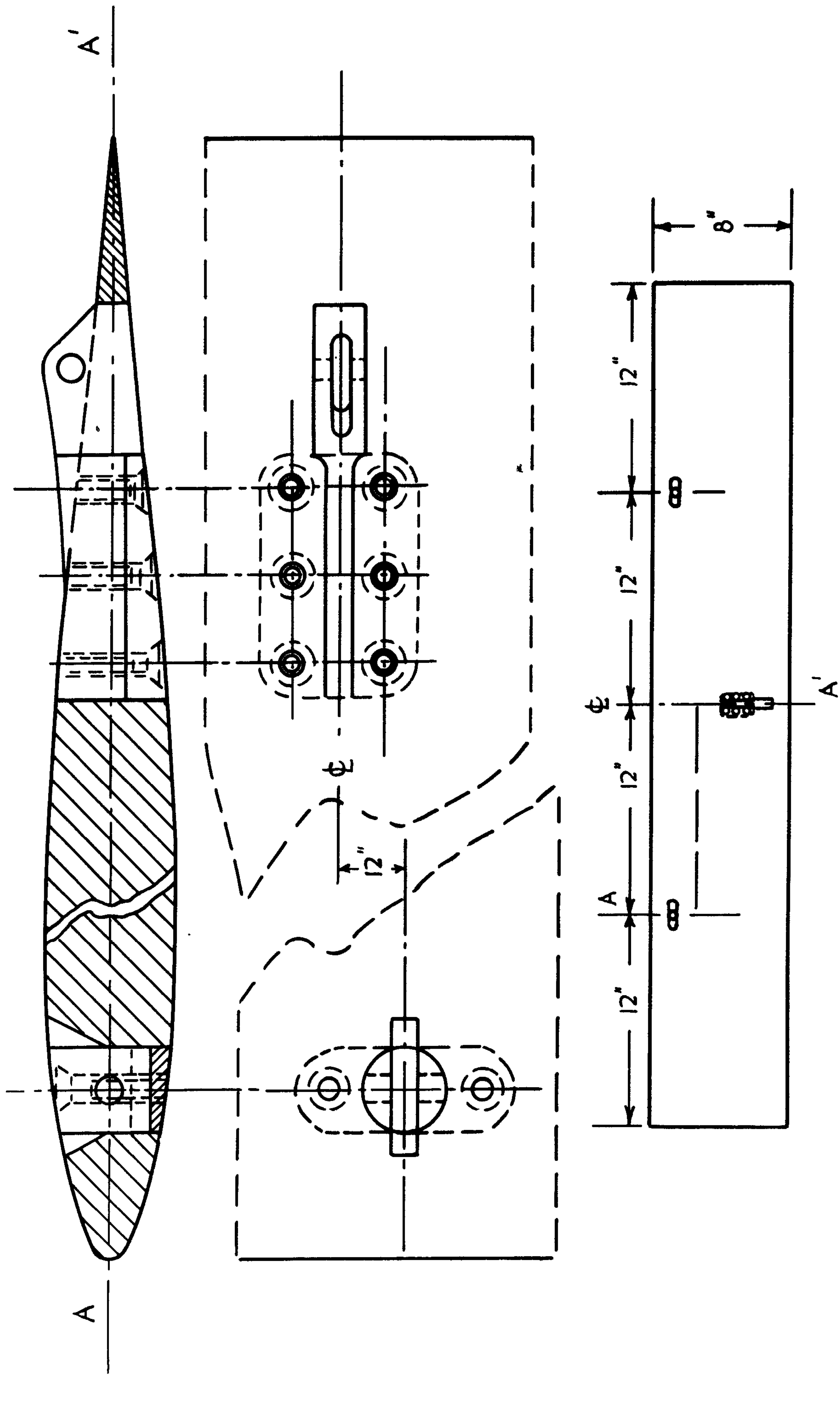
Table 2 (contd.)

P = 8.01 Atmos. $\frac{1}{2}\rho V^2$ = 43.14 lb/sq ft			
V = 68.77 F.P.S. R = 2.18 × 10^6			
$\alpha°$	C_L	C_D	C_M
- 2.95	-0.223	0.0091	-0.0107
- 1.7	- .131	.0072	- .0094
- 0.45	- .039	.0059	- .0079
+ 0.8	+ .055	.0054	- .0061
2.0	.148	.0067	- .0051
4.5	.325	.0122	- .0017
6.95	.507	.0219	- .0005
9.4	.689	.0356	+ .0011
11.85	.866	.0529	.0022
14.35	1.038	.0760	.0020
15.55	1.118	.0871	.0022
16.8	1.188	.102	.0025
18.05	1.242	.117	.0010
18.8	.650	.252	- .0913
19.75	.621	.262	- .0929
22.35	.638	.306	- .0989
24.95	.658	.351	- .1035

P = 25.0 Atmos. $\frac{1}{2}\rho V^2$ = 67.2 lb/sq ft			
V = 48.66 F.P.S. R = 4.76 × 10^6			
$\alpha°$	C_L	C_D	C_M
- 2.95	-0.227	0.0086	-0.0086
- 1.7	- .134	.0067	- .0084
- 0.45	- .039	.0054	- .0075
+ 0.8	+ .054	.0052	- .0060
2.0	.146	.0067	- .0044
4.5	.326	.0125	- .0020
6.95	.516	.0222	+ .0007
9.4	.694	.0356	.0009
11.85	.885	.0538	.0023
13.1	.975	.0651	.0024
14.35	1.064	.0763	.0027
15.55	1.156	.0890	.0029
16.8	1.237	.102	.0039
18.05	1.309	.117	.0034
19.3	1.388	.132	.0056
20.55	1.451	.146	.0068
21.4	1.018	.253	- .0556
22.3	.783	.289	- .0834
23.55	.818	.329	- .0875
24.9	.792	.342	- .0842

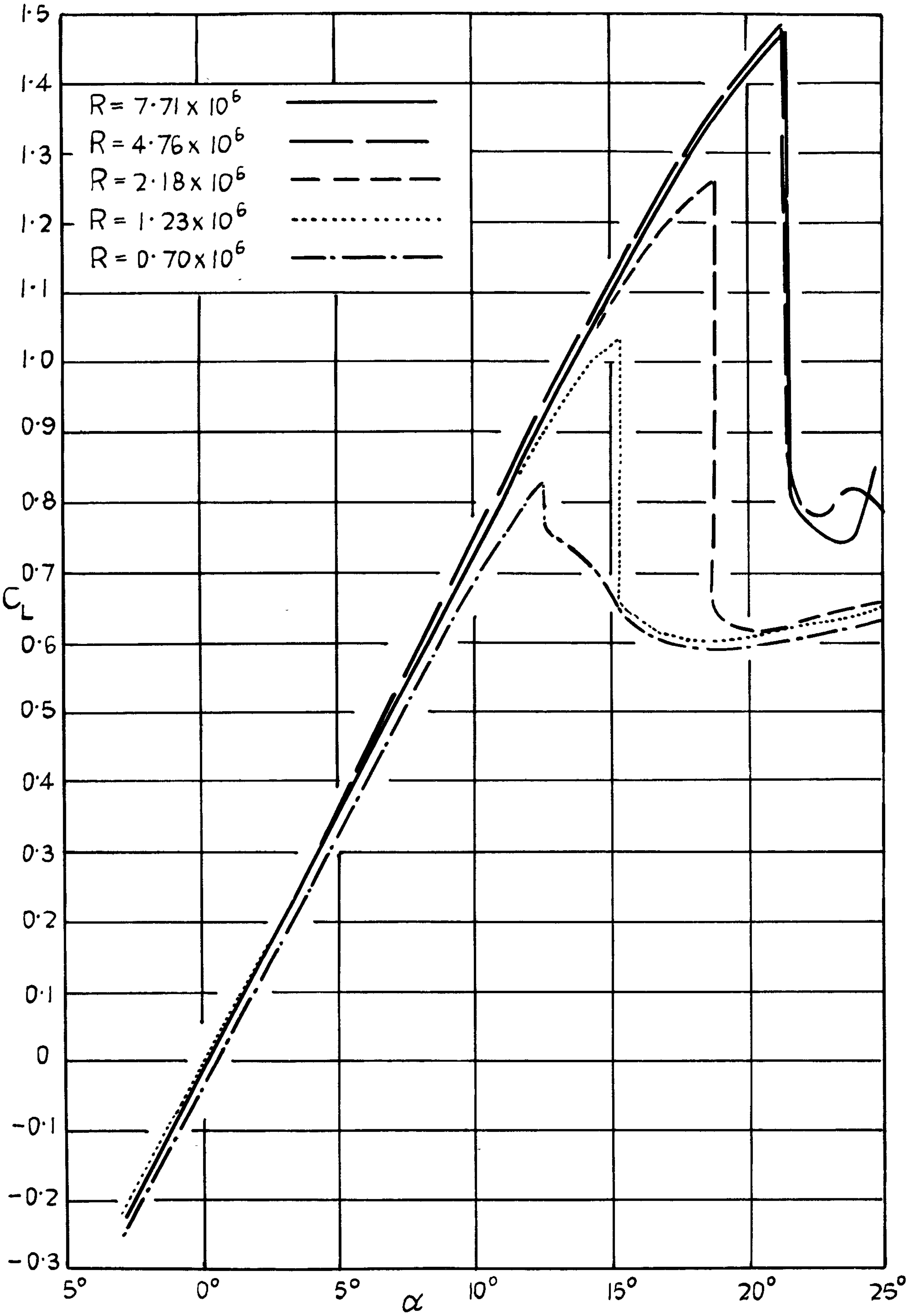
P = 25.42 Atmos. $\frac{1}{2}\rho V^2$ = 179.1 lb/sq ft			
V = 79.1 F.P.S. R = 7.71 × 10^6			
$\alpha°$	C_L	C_D	C_M
- 3.0	-0.229	0.0084	-0.0095
- 1.75	- .139	.0064	- .0078
- 0.45	- .044	.0048	- .0069
+ 0.8	+ .049	.0047	- .0061
2.05	.142	.0060	- .0038
4.5	.323	.0120	- .0021
7.0	.507	.0212	- .0008
9.5	.692	.0352	+ .0004
11.95	.876	.0525	.0022
14.45	1.054	.0747	.0023
16.95	1.224	.101	.0044
18.15	1.306	.115	.0054
19.45	1.387	.131	.0065
20.7	1.444	.147	.0074
21.5	.805	.254	- .0783
22.15	.777	.266	- .0817
23.45	.742	.288	- .0871
24.7	.858	.340	- .1030

FIG. 1.



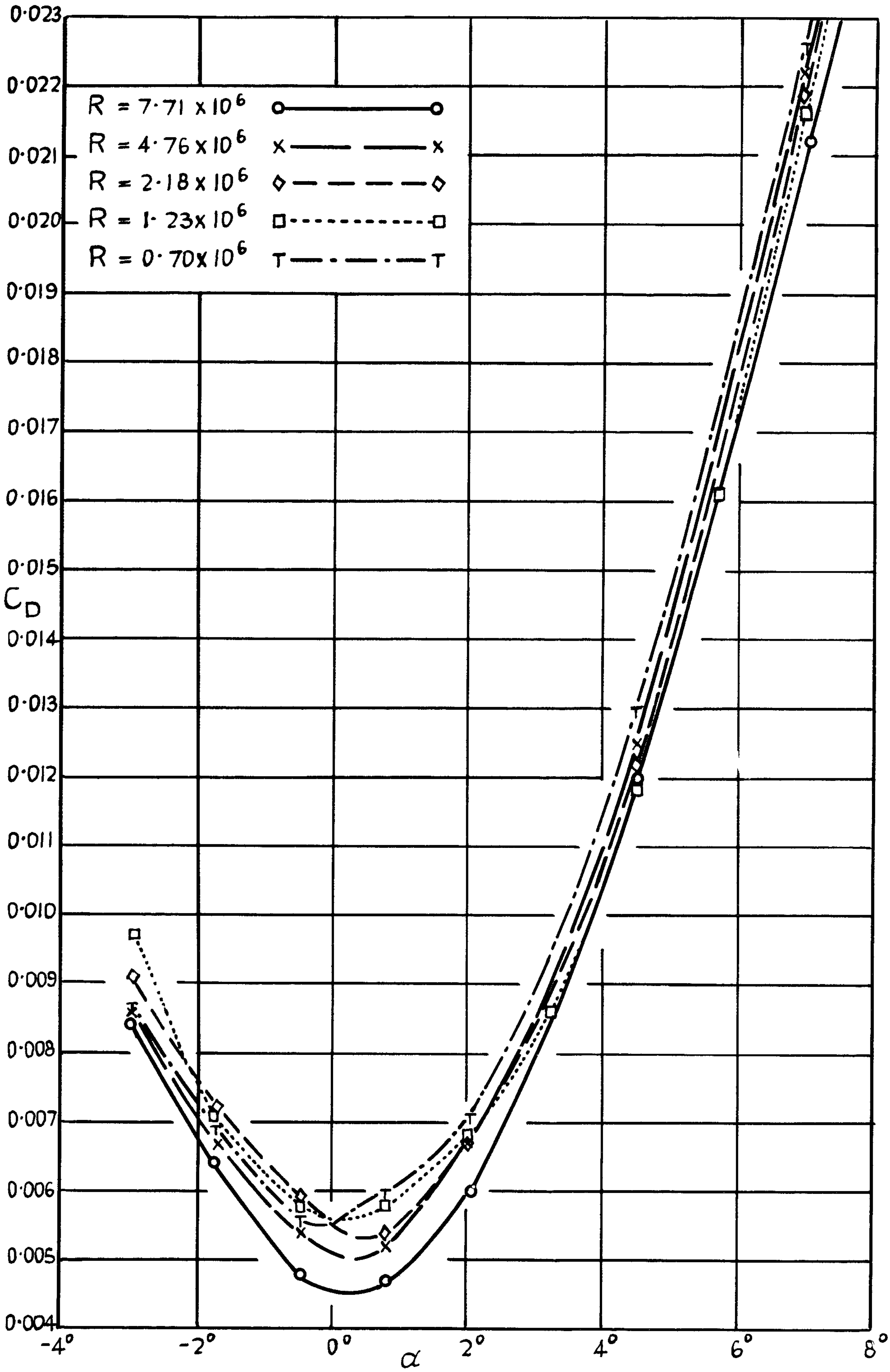
Arrangement of support fittings on 11% RAE 101.

FIG. 2.



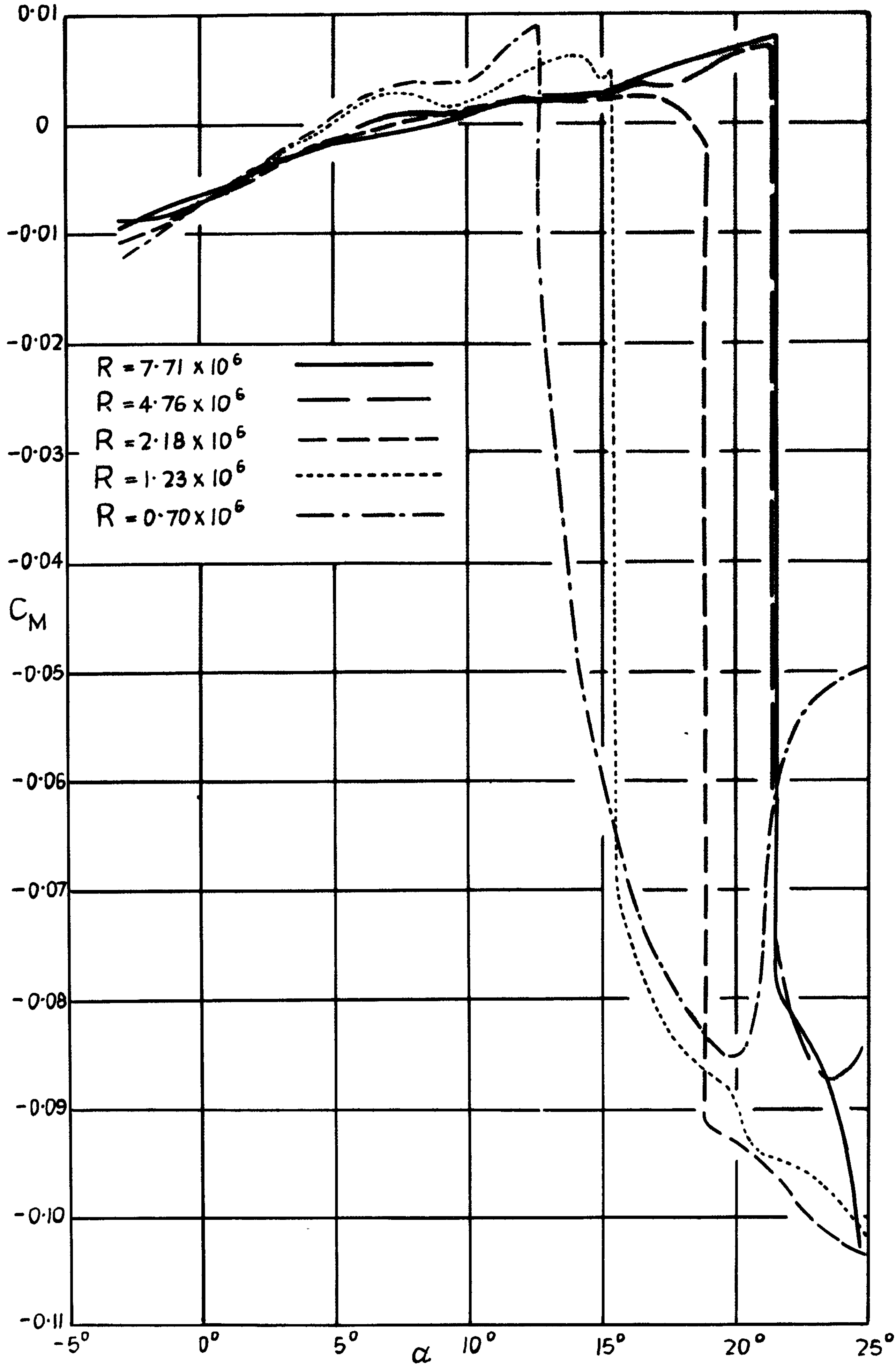
Variation of lift coefficient with incidence for 11% RAE 101 wing of aspect ratio 6.

FIG. 3

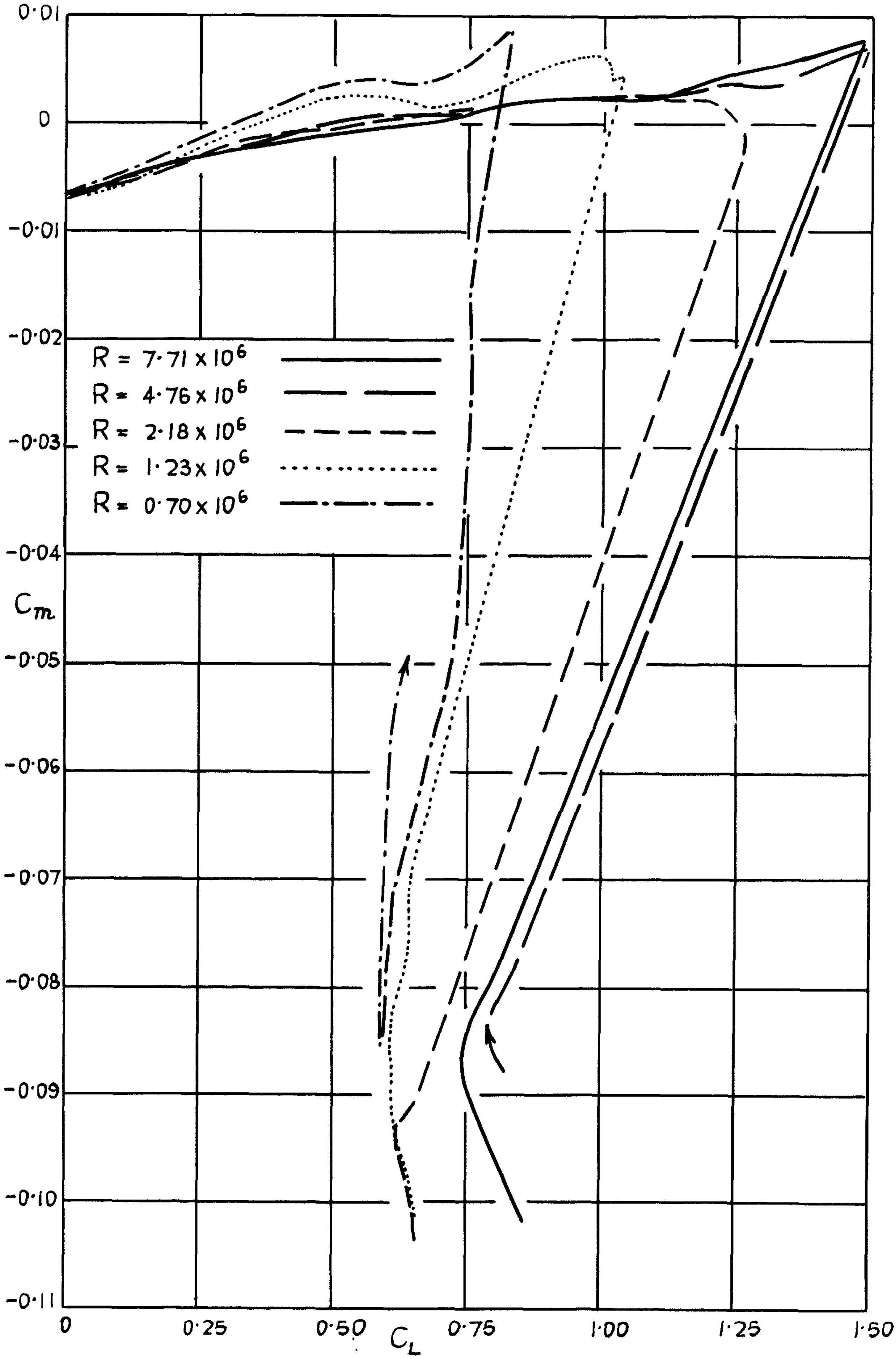


Variation of drag coefficient with incidence for 11% RAE 101 wing of aspect ratio 6.

Fig. 4.



Variation of pitching moment coefficient with incidence for 11% RAE 101 wing of aspect ratio 6.



Variation of pitching moment coefficient with lift coefficient for 11% RAE 101 wing of aspect ratio 6.

FIGS. 6 & 7.

FIG. 6.

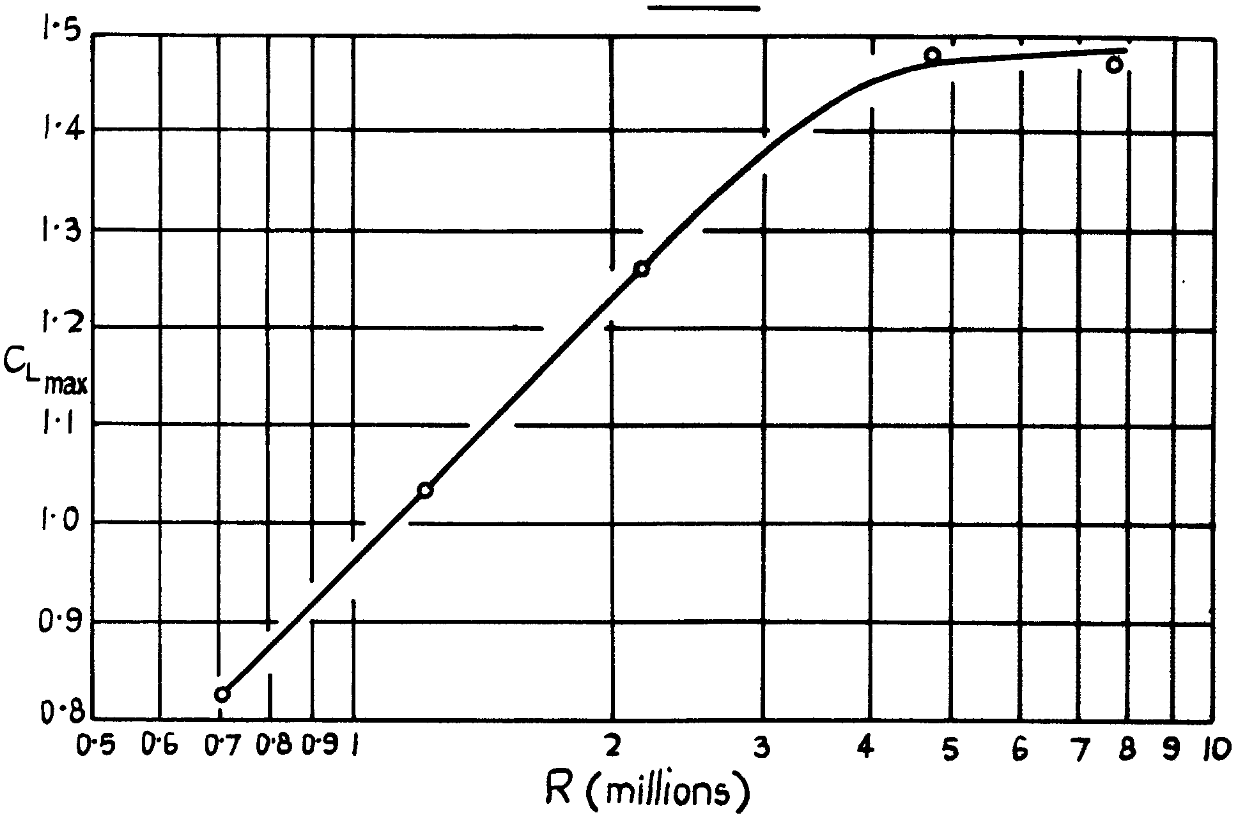
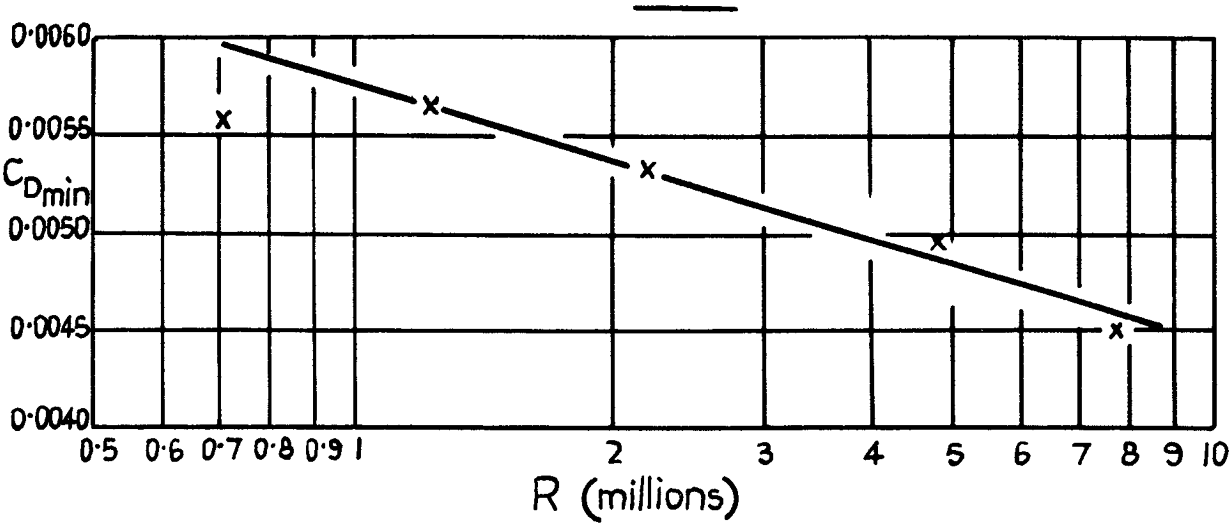


FIG. 7.



Effect of Reynolds number on $C_{L_{max}}$ and $C_{D_{min}}$.

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