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An Analysis of the Flow over Two 45-deg Swept-Back Wings at High Subsonic Speeds, and Comparison with Theory

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An Analysis of the Flow over Two 45-deg Swept-Back Wings at High Subsonic Speeds, and Comparison with Theory

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Summary. This paper presents a comparison of pressure distributions obtained from theory and from wind-tunnel tests on two swept-back wings at various incidences and Mach numbers. The first wing, having NACA 65A004, sections had a taper ratio 0·2, an aspect ratio of 3, and had its quarter-chord line swept back through 45 deg. The second wing had RAE 102 sections, a taper ratio of 1/3, an aspect ratio of $2\sqrt{2}$, and had its semi-chord line swept back through 45 deg. The experimental flow conditions and patterns are presented in greater detail in Refs. 1 and 10, and are briefly discussed here insofar as they are relevant to the comparisons.

1. *Introduction.* At the time of writing this paper there appear to be few reliable experimental pressure distributions for the type of wings considered here, and even fewer comparisons with experiment. The present paper should go some way towards filling this gap, although any conclusions reached are necessarily rather tentative, further work on this subject being required.

According to the theoretical work carried out at the Royal Aircraft Establishment (partly published in Refs. 4, 5 and 11, and partly still unpublished), the local velocity V on a lifting uncambered wing of finite thickness is given by:

$$
\begin{aligned}
\left(\frac{V}{V_0}\right)^2 = {} & \frac{1}{1+\left[\dfrac{S^{(2)}(x)}{\cos\left[(1-|\mu|)\phi_t\right]A}\right]^2}\left[\left[\left\{\cos\alpha_e\left[1+\frac{\tau_a(1+K_1)K(y_a)\cos\phi_t S^{(1)}(x)}{A}-\right.\right.\right.\right. \\
& \left.-\frac{\tau_a\mu(y_a)\cos\phi_t f(\phi_{t_a})}{A}\,\frac{S^{(2)}(x)}{\left(1+\left[\dfrac{S^{(2)}(x)}{A}\right]^2\right)^{1/2}}\right]\pm \\
& \left.\pm\sin\alpha\,\frac{\cos\phi_{c/2}}{(1-M_0^2\cos^2\phi_{c/2})^{1/2}}\left(\frac{1-x}{x}\right)^{n_a}\left[1+\frac{S^{(3)}(x)}{\cos\left[(1-|\mu|)\phi_t\right]A}\right]\right\}^2+ \\
& +\left\{\cos\alpha_e\left[\frac{(1-|\mu(y_a)|)\sin\phi_t S^{(1)}(x)}{A}\right]\pm\right. \\
& \pm\sin\alpha_e\left(\frac{(1-M_0^2\cos^2\phi_{c/2})\cos^2(\lambda_a\phi_{ac/2})-(1-M_0^2)\cos^2\phi_{c/2}}{1-M_0^2\cos^2\phi_{c/2}}\right)^{1/2}\left(\frac{1-x}{x}\right)^{n_a} \\
& \left.\left.\left[1+\frac{S^{(3)}(x)}{\cos\left[(1-|\mu|)\phi_t\right]A}\right]\right\}^2-(1-\mu_a^2)\cos^2\alpha_e\ \sin^2\phi_t\right]+ \\
& +(1-\mu_a^2)\cos^2\alpha_e\ \sin^2\phi_t\,,
\end{aligned}
$$

where

$$A = \{1 - M_0^2 [(1 - C_{p\,i}) \cos^2 \phi_t - C_{p\,i} (1 - |\mu|) \sin^2 \phi_t]\}^{1/2} ,$$

$$C_{p\,i} = C_p (M_0 = \alpha_e = 0) \text{ or } 0 \text{ if } C_p (M_0 = \alpha_e = 0) > 0 ,$$

$$n_a = \frac{1}{2}\left[1 - \lambda(y_a) \frac{\phi_{ac/2}}{\pi/2}\right] .$$

The pressure coefficient C_p is then given by the relation:

$$C_p = -\frac{2}{K}\frac{1}{M_0^2}\left[1 - \left\{1 + \frac{K-1}{2} M_0^2 \left[1 - \left(\frac{V}{V_0}\right)^2\right]\right\}^{K/(K-1)}\right] ,$$

where $K = 1{\cdot}4$.

The suffix $_a$ refers any particular parameter to the analogous wing.

The large number of C_p values needed to define satisfactorily the pressure distribution over a given wing makes the use of the above equation very tedious when evaluated on a desk machine. In the analysis described in this report a DEUCE computer was used to carry out most of the calculations.

Two wings were considered: the first (shown in Fig. 1) which was tested in the Langley 16 ft Transonic Tunnel had an aspect ratio of 3, a taper ratio of 0·2 and NACA 65A004 aerofoil sections. The second wing (Wing 12 of a systematic series of plan-forms devised by Warren) had an aspect ratio of $2\sqrt{2}$, a taper ratio of 1/3, and RAE 102 6 per cent thick sections. It was investigated by Hall and Rogers at the National Physical Laboratory using a half-wing model mounted on the sidewall of the 18 in. × 14 in. High Speed Tunnel. The N.A.C.A. wing was mounted on a body which, at the line of intersection with the wing, had a radius approximately equal to 16 per cent of the wing semi-span.

The above formula holds only for isolated wings and consequently any perturbational velocities induced over the wing by the body have not been taken into account.

2. *Description of Method.* The above equation was evaluated at the same spanwise stations that had been considered in the experiments for an arbitrary range of incidence, and the corresponding chordwise pressure distributions were integrated to give the local lift coefficients. The pressure coefficients for the upper and lower surfaces at particular chordwise stations were plotted against the local lift coefficient for the spanwise station in question, thus enabling the chordwise distribution to be determined once the local lift coefficient was known. It was found that these $C_p \sim C_L$ curves were more or less linear except where movement of the stagnation point introduced a non-linearity for the chordwise stations very near the leading edge.

The spanwise load distributions were calculated using Küchemann's method given in Ref. 4; in the case of the N.A.C.A. wing an extension to the theory (Ref. 2) to take into account the presence of the body had to be used. In the calculations of the span-loading curves a two-dimensional lift curve slope of $a_0 = 2\pi$ was assumed. This implies the assumption that the increase of a_0 due to wing thickness is cancelled by the decrease due to viscosity effects; also that the spanwise variations of boundary-layer effects are negligible. This approximation will become more suspect with increase of incidence. The spanwise load distributions obtained are presented in Figs. 3 to 7, together with those obtained from experiment.

The experimental chordwise distributions were compared with the theoretical distributions obtained by assuming, in the first case, the experimental local lift coefficient and, secondly, the C_L obtained from theory. In those cases where the discrepancy between experimental and theoretical local load coefficients was small the second comparison was omitted.

In the initial application of the above theory the spanwise variant $\tau_a(1 + K_1)$, which takes account of such three-dimensional effects as plan-form and thickness taper, was taken to be unity. This was largely because of the lack of suitable information for its estimation. However, at a later stage several detailed calculations by Newby, Neumark and Collingbourne were discovered (Refs. 6 and 7) from which values of $\tau_a(1 + K_1)$ could be estimated. In the comparisons presented below $\tau_a(1 + K_1)$ was taken to be unity throughout; the effect of substituting a calculated value of $\tau_a(1 + K_1)$ into the theory is demonstrated for the particular case of the Warren Wing at $\alpha = 6$ deg and $M = 0 \cdot 6$ in Fig. 14.

Also shown on the graphs is the curve of the critical pressure coefficient C_p^*, i.e., the value of the coefficient at which the local Mach-number component normal to the local sweep becomes equal to unity.

3. *Discussion of Results.* Since the equation given in Section 1 contains no viscous terms there will always be disagreement between theory and experiment for those cases where viscosity tends to distort the shape of the experimental curves (such an effect is particularly noticeable at the more outboard stations of the N.A.C.A. wing for $M = 0 \cdot 94$ (*see* Fig. 10); where the separated vortex layer affects the loading diagrams).

In Ref. 1 Runckel and Lee attribute the compressions which appear on the N.A.C.A. wing to the presence of shocks, but, for the incidences dealt with here, they are more likely caused by reattachment of the separated vortex layer. This separation starts from the wing tips at comparatively small incidences for $M = 0 \cdot 8$, and with increasing incidence and Mach number spreads both chordwise and inboard. Its effect is evident on the pressure distribution at the 95 per cent semi-span station for the $M = 0 \cdot 8$, 2·4-deg incidence case (*see* Fig. 8) and as the Mach number increases to 0·94 and the incidence to 6·4 deg the separated region extends beyond the wing trailing edge at the tips and inboard very nearly to the bodyside (*see* Fig. 10). The development of the vortex with increase of free-stream Mach number is demonstrated for three spanwise stations in Fig. 13. The chordwise pressure distributions obtained from tests on the Warren Wing 12 show much less marked effects of viscosity than those of the N.A.C.A. wing, particularly for the lower Mach number of 0·6. At $M = 0 \cdot 85$ the region of separated flow begins to develop at the more outboard stations for an incidence of about 4 deg, but even at the highest incidence considered does not greatly affect the shape of the pressure distributions at the inboard stations.

When comparing experiment and theory from the chordwise distributions it should be remembered that two comparisons have been performed: the ordinary broken line is the distribution obtained from theory using the sectional lift coefficient obtained from experiment, while the curve defined by the crosses contains a lift coefficient obtained from theory. The local C_L values for the broken and full lines is the same and is listed alongside the graphs, but the lift which defines the crosses is smaller because of the effect of the vortex in increasing the experimental spanwise sectional lift coefficients. This increase in sectional C_L is due to the low pressures which exist beneath the vortex, and consequently is mostly obtained over the front part of the chord.

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One apparent weakness of the theory, as far as the N.A.C.A. wing comparisons are concerned, is its underestimate of the upper-surface leading-edge suction for all cases considered where viscous effects cannot be blamed for the discrepancies. The discrepancy, which is present at small values of C_L, increases with increasing incidence. This means that a further correction for three-dimensional flow is required near the leading edge on wings of small aspect ratio. The best way of accomplishing this would be to increase n_a and decrease μ_a in this region, and also to make these two quantities vary over the whole plan-form instead of just varying with spanwise position. Putting in the correct values of $\tau_a(1 + K_1)$ from Table 6 would not help to appreciably improve the agreement at the leading-edge stations as it is much too small to be able to produce the increases in negative C_p required. This can be appreciated from Fig. 14 which shows the effect of $\tau_a(1 + K_1)$ for the particular case of the Warren Wing at $M = 0{\cdot}6$ and an incidence of 6 deg. Two-dimensional pressure distributions given in Ref. 8 for N.A.C.A. aerofoil sections similar to the one considered in this report show no 'peakiness' in the C_p distributions near the leading edge suggesting that such leading-edge discrepancies as are evident in the distributions for the 25, 40, 60 and 75 per cent semi-span stations for an incidence of $0{\cdot}4$, for instance, are three-dimensional effects requiring a further three-dimensional correction to the theory.

The pressure distributions for the Warren Wing show better agreement generally than those for the N.A.C.A. wing, quite apart from the less drastic effects of viscosity, but again the theory tends to underestimate the negative pressure coefficients at the leading edge.

Table 6 shows that for all cases considered for the two wings, the value of $\tau_a(1 + K_1)$ is of the same order of magnitude, and as any particular $\tau_a(1 + K_1)$ will multiply quantities of similar order as for $M = 0{\cdot}6$, it may be assumed that the resultant differences in the pressure curves will be similar to those shown in Fig. 6. Consequently $\tau_a(1 + K_1)$ can hardly be regarded as an adequate tool in making an acceptable agreement between the leading-edge pressures for the cases where $\tau_a(1 + K_1)$ had originally been assumed to be unity. In Ref. 6 Newby points out that τ_a is more or less constant across the whole of the chord and therefore it may be evaluated at any chordwise position; in particular, since τ_a is given by:

$$\frac{v_x}{V_0}(x, \eta) = \tau\{(1 + K_1(x, \eta))S^{(1)}(x, \eta) - \mu S^{(2)}(x, 0)f(\phi)\} \cos \phi \,, \tag{2}$$

where $v_x(x, \eta)$ is the chordwise supervelocity at the point (x, η), then performing the calculation of $\tau_a(1 + K_1)$ at the maximum-thickness position where $S^{(2)}(x, o) = 0$ means that $\tau_a(1 + K_1)$ is given simply as:

$$\tau_a(1 + K_1) = \frac{v_x}{V_0}(x_1, \eta) \Big/ S^{(1)}(x_1, \eta) \cos \phi_1 \,, \tag{3}$$

where x_1 is the x co-ordinate of the maximum-thickness position.

For a wing-body combination such as that tested by Runckel and Lee in Ref. 1 the bodyside will reflect the wing for a small distance spanwise and, hence, for stations very near to the bodyside η should be measured from the bodyside when calculating $v_x(x_1\eta)$ in equation (3). Consequently, the 16 per cent semi-span station of the N.A.C.A. wing, which is the bodyside, corresponds to a $\eta = 0$; similarly for stations far removed from the bodyside η should be measured from the body centre-line. Table 6 gives values of $\tau_a(1 + K_1)$ obtained by both these alternatives. For $M = 0{\cdot}8$ the value $\tau_a(1 + K_1) = 0{\cdot}953$ is the correct one for the 95 per cent station, the values at the other stations being obtained by an interpolation of the two curves. Similarly for the other Mach numbers.

In Table 7 are listed the values of the spanwise variants μ_a, n_a and K for the different cases.

4. *Conclusions.* A general apparent weakness of the theory, which is particularly noticeable from the N.A.C.A. wing comparisons is its underestimate of the leading-edge suction over the upper surface. In order to remedy this, a variation chordwise of the parameters μ_a and n_a would be necessary.

As Dr. Weber has previously pointed out, the above theory is applicable only to isolated wings so that in the case of the N.A.C.A. wing-body configuration (shown in Fig. 1), any perturbational pressures due to the presence of the body are not included in the theory and are hence omitted from the comparisons. A theoretical C_p for the body-alone case would quickly diminish with increasing spanwise distance and for a slender body would have no effect on the pressure distributions at the more outboard stations. Treating the body used on the N.A.C.A. model as a nearly cylindrical body with both ends pointed, results published in Ref. 9 for such a body show that at the bodyside the maximum perturbational C_p is of the order of 0·015. Because of the finite thickness of the wing, the body will be effectively waisted at the junction with the wing but, as the maximum waisting is roughly equal to 2 per cent of the body diameter, any influence on the airflow will be negligibly small.

For those cases where viscosity is not important, the agreement obtained between the experimental and theoretical pressure distributions is very encouraging, although the leading-edge pressures clearly require more attention.

LIST OF SYMBOLS

x	Distance in terms of chord length from the leading edge measured along the chord
y	Distance spanwise, measured from the vertical plane of symmetry for the wing-alone case
z_t	Section ordinates
ϕ_t	Thickness sweep
M_0	Main-stream Mach number
α_e	Effective angle of incidence
$S^{(1)}(x)$	$= \dfrac{1}{\pi}\displaystyle\int_0^1 \frac{dz_t}{dx'}\frac{dx'}{x-x'}$
$S^{(2)}(x)$	$= dz_t/dx$
$S^{(3)}(x)$	$= \dfrac{1}{\pi}\displaystyle\int_0^1 \left[\frac{dz_t}{dx'} - \frac{dz_t(x')}{1-(1-2x')^2}\right]\frac{dx'}{(x-x')}$
$\lambda(y)$	A function of spanwise position
$\mu(y)$, $K(y)$, $f(\phi_t)$	Functions of wing geometry
$\tau(1+K_1)$	A function of such three-dimensional effects as plan-form and thickness taper (*see* Ref. 6)

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No.	Author	Title, etc.
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TABLE 1

Theoretical Values of Pressure Coefficient—N.A.C.A. Wing

$M = 0 \cdot 8$

x/c	$y/s = 0 \cdot 16$							$y/s = 0 \cdot 25$						
	$\alpha = 0$	$\alpha = 2 \cdot 4$ deg $[C_L = 0 \cdot 100]$		$\alpha = 4 \cdot 4$ deg $[C_L = 0 \cdot 206]$		$\alpha = 6 \cdot 4$ deg $[C_L = 0 \cdot 305]$		$\alpha = 0$	$\alpha = 2 \cdot 4$ deg $[C_L = 0 \cdot 110]$		$\alpha = 4 \cdot 5$ deg $[C_L = 0 \cdot 226]$		$\alpha = 6 \cdot 4$ deg $[C_L = 0 \cdot 348]$	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	+0·268	+0·152	+0·375	+0·031	+0·489	−0·088	0·590	+0·076	−0·110	+0·243	−0·306	+0·408	−0·511	0·568
0·038	0·066	−0·025	0·159	−0·122	0·253	−0·212	0·342	−0·007	−0·136	0·117	−0·271	0·241	−0·417	0·367
0·084	0·038	−0·040	0·112	−0·121	0·192	−0·196	0·269	−0·016	−0·112	0·080	−0·217	0·180	−0·326	0·280
0·146	+0·000	−0·068	0·072	−0·140	0·145	−0·198	0·203	−0·036	−0·124	0·048	−0·250	0·123	−0·287	0·205
0·222	−0·037	−0·098	+0·029	−0·163	0·094	−0·223	0·158	−0·060	−0·132	+0·003	−0·208	0·086	−0·288	0·166
0·309	−0·074	−0·125	−0·008	−0·179	0·060	−0·142	0·102	−0·077	−0·150	−0·021	−0·215	0·050	−0·283	0·100
0·402	−0·101	−0·150	−0·040	−0·198	+0·023	−0·153	0·062	−0·098	−0·152	−0·039	−0·213	+0·024	−0·277	0·072
0·500	−0·124	−0·170	−0·071	−0·219	−0·018	−0·267	0·032	−0·114	−0·163	−0·069	−0·214	−0·009	−0·268	0·049
0·598	−0·124	−0·152	−0·073	−0·200	−0·022	−0·100	0·022	−0·099	−0·143	−0·062	−0·187	−0·004	−0·233	0·027
0·691	−0·114	−0·148	−0·074	−0·178	−0·026	−0·224	0·005	−0·083	−0·097	−0·050	−0·152	+0·000	−0·194	0·025
0·778	−0·097	−0·128	−0·059	−0·163	−0·020	−0·195	0·017	−0·060	−0·090	−0·026	−0·121	0·009	−0·153	0·048
0·854	−0·064	−0·090	−0·034	−0·126	+0·002	−0·149	0·025	−0·027	−0·052	−0·010	−0·074	0·049	−0·099	0·049
0·916	−0·025	−0·050	−0·002	−0·079	0·027	−0·101	0·049	+0·008	−0·022	+0·024	−0·022	0·072	−0·051	0·074
0·962	+0·025	+0·007	+0·048	−0·014	+0·072	−0·033	0·097	+0·053	+0·040	+0·071	+0·024	+0·091	+0·010	0·110

TABLE 1—*continued*

Theoretical Values of Pressure Coefficient—N.A.C.A. Wing

$M = 0 \cdot 8$

x/c	$y/s = 0 \cdot 40$							$y/s = 0 \cdot 60$						
	$\alpha = 0$	$\alpha = 2 \cdot 4$ deg [$C_L = 0 \cdot 126$]		$\alpha = 4 \cdot 4$ deg [$C_L = 0 \cdot 252$]		$\alpha = 6 \cdot 4$ deg [$C_L = 0 \cdot 395$]		$\alpha = 0$	$\alpha = 2 \cdot 4$ deg [$C_L = 0 \cdot 147$]		$\alpha = 4 \cdot 5$ deg [$C_L = 0 \cdot 290$]		$\alpha = 6 \cdot 4$ deg [$C_L = 0 \cdot 481$]	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	−0·004	−0·308	+0·250	−0·623	+0·480	−0·985	0·665	−0·046	−0·519	+0·345	−0·995	+0·580	−1·643	0·690
0·038	−0·044	−0·220	0·115	−0·414	0·279	−0·639	0·443	−0·064	−0·308	0·158	−0·573	0·360	−0·953	0·573
0·084	−0·040	−0·163	0·075	−0·296	0·194	−0·454	0·321	−0·055	−0·217	0·097	−0·388	0·238	−0·634	0·415
0·146	−0·048	−0·150	0·038	−0·246	0·135	−0·374	0·195	−0·070	−0·175	0·051	−0·298	0·162	−0·486	0·174
0·222	−0·073	−0·154	+0·007	−0·240	0·093	−0·340	0·188	−0·079	−0·176	+0·013	−0·273	0·103	−0·420	0·234
0·309	−0·082	−0·162	−0·024	−0·227	0·051	−0·322	0·125	−0·101	−0·175	−0·014	−0·252	0·069	−0·381	0·175
0·402	−0·100	−0·168	−0·047	−0·220	+0·023	−0·295	0·081	−0·111	−0·173	−0·040	−0·238	0·026	−0·338	0·125
0·500	−0·112	−0·159	−0·060	−0·210	−0·007	−0·270	0·058	−0·107	−0·159	−0·056	−0·213	+0·000	−0·292	0·080
0·598	−0·099	−0·131	−0·051	−0·168	−0·004	−0·214	0·052	−0·099	−0·128	−0·047	−0·175	−0·001	−0·225	0·072
0·691	−0·074	−0·102	−0·035	−0·127	+0·002	−0·171	0·050	−0·069	−0·100	−0·026	−0·135	+0·001	−0·170	0·056
0·778	−0·044	−0·071	−0·018	−0·099	0·017	−0·129	0·053	−0·036	−0·060	−0·009	−0·088	0·020	−0·122	0·063
0·854	−0·009	−0·027	0·003	−0·047	0·040	−0·070	0·070	−0·001	−0·017	+0·018	−0·049	0·028	−0·060	0·073
0·916	+0·025	−0·002	0·034	−0·003	0·062	−0·027	0·078	+0·037	+0·021	0·047	−0·006	0·053	−0·008	0·092
0·962	+0·066	+0·057	0·080	+0·048	+0·093	+0·036	0·109	+0·075	+0·068	+0·086	+0·060	+0·098	+0·050	0·114

TABLE 1—*continued*

Theoretical Values of Pressure Coefficient—N.A.C.A. Wing

$M = 0{\cdot}8$

x/c	$y/s = 0{\cdot}75$							$y/s = 0{\cdot}95$						
	$\alpha = 0$	$\alpha = 2{\cdot}4$ deg $[C_L = 0{\cdot}151]$		$\alpha = 4{\cdot}5$ deg $[C_L = 0{\cdot}293]$		$\alpha = 6{\cdot}4$ deg $[C_L = 0{\cdot}484]$		$\alpha = 0$	$\alpha = 2{\cdot}4$ deg $[C_L = 0{\cdot}130]$		$\alpha = 4{\cdot}5$ deg $[C_L = 0{\cdot}257]$		$\alpha = 6{\cdot}4$ deg $[C_L = 0{\cdot}323]$	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	−0·046	−0·592	+0·400	−1·108	+0·603	−1·800	0·673	−0·046	−0·550	+0·399	−1·038	+0·595	−1·299	0·633
0·038	−0·065	−0·331	0·180	−0·610	0·382	−1·017	0·593	−0·065	−0·311	0·152	−0·560	0·348	−0·700	0·347
0·084	−0·057	−0·227	0·101	−0·398	0·244	−0·654	0·427	−0·055	−0·201	0·089	−0·355	0·218	−0·441	0·218
0·146	−0·073	−0·201	0·048	−0·308	0·164	−0·490	0·325	−0·064	−0·155	0·047	−0·254	0·132	−0·340	0·199
0·222	−0·079	−0·179	+0·013	−0·274	0·104	−0·423	0·234	−0·075	−0·161	+0·001	−0·249	0·080	−0·298	0·123
0·309	−0·099	−0·180	−0·025	−0·265	0·050	−0·375	0·175	−0·099	−0·165	−0·025	−0·240	+0·029	−0·275	0·075
0·402	−0·100	−0·175	−0·050	−0·243	+0·003	−0·325	0·123	−0·122	−0·154	−0·049	−0·222	−0·002	−0·250	0·033
0·500	−0·109	−0·160	−0·057	−0·212	−0·005	−0·288	0·071	−0·109	−0·150	−0·066	−0·191	−0·021	−0·220	0·002
0·598	−0·096	−0·128	−0·051	−0·175	−0·005	−0·227	0·003	−0·096	−0·124	−0·051	−0·151	−0·012	−0·178	0·000
0·691	−0·074	−0·098	−0·036	−0·127	−0·001	−0·175	0·049	−0·070	−0·094	−0·040	−0·114	+0·001	−0·142	0·001
0·778	−0·033	−0·060	−0·009	−0·083	+0·017	−0·117	0·057	−0·032	−0·052	−0·017	−0·073	0·006	−0·084	0·020
0·854	−0·010	−0·023	+0·003	−0·041	0·035	−0·073	0·050	−0·003	−0·024	+0·010	−0·026	0·029	−0·049	0·029
0·916	+0·028	+0·014	0·048	−0·001	0·058	−0·027	0·052	+0·027	+0·015	0·042	+0·004	0·050	−0·001	0·050
0·962	+0·075	+0·068	+0·088	+0·062	+0·099	+0·053	0·110	+0·075	+0·070	+0·083	+0·067	+0·091	+0·064	0·097

TABLE 2

Theoretical Values of Pressure Coefficient—N.A.C.A. Wing

$M = 0 \cdot 9$

x/c	$y/s = 0 \cdot 16$							$y/s = 0 \cdot 25$						
	$\alpha = 0$	$\alpha = 2 \cdot 4$ deg [$C_L = 0 \cdot 122$]		$\alpha = 4 \cdot 5$ deg [$C_L = 0 \cdot 244$]		$\alpha = 6 \cdot 4$ deg [$C_L = 0 \cdot 364$]		$\alpha = 0$	$\alpha = 2 \cdot 4$ deg [$C_L = 0 \cdot 130$]		$\alpha = 4 \cdot 5$ deg [$C_L = 0 \cdot 262$]		$\alpha = 6 \cdot 4$ deg [$C_L = 0 \cdot 383$]	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	+0·342	+0·227	+0·452	+0·109	+0·560	−0·005	0·661	+0·132	−0·039	+0·298	−0·211	+0·451	−0·385	0·588
0·038	0·097	−0·001	0·194	−0·100	0·293	−0·196	0·390	0·015	−0·118	0·140	−0·250	0·269	−0·379	0·383
0·084	0·059	−0·024	0·145	−0·110	0·231	−0·193	0·318	+0·000	−0·106	0·104	−0·211	0·210	−0·315	0·305
0·146	+0·020	−0·055	0·100	−0·146	0·176	−0·210	0·152	−0·028	−0·114	0·061	−0·202	0·156	−0·295	0·250
0·222	−0·028	−0·100	0·049	−0·173	0·121	−0·146	0·195	−0·056	−0·137	+0·029	−0·220	0·113	−0·300	0·191
0·309	−0·073	−0·140	+0·002	−0·202	0·074	−0·172	0·145	−0·099	−0·160	−0·007	−0·236	0·064	−0·301	0·147
0·402	−0·109	−0·177	−0·037	−0·232	+0·027	−0·298	0·093	−0·127	−0·178	−0·045	−0·250	+0·025	−0·301	0·099
0·500	−0·144	−0·201	−0·080	−0·259	−0·015	−0·313	0·049	−0·132	−0·194	−0·069	−0·255	−0·002	−0·313	0·059
0·598	−0·151	−0·219	−0·096	−0·151	−0·023	−0·302	0·027	−0·130	−0·197	−0·076	−0·244	−0·002	−0·292	0·050
0·691	−0·150	−0·208	−0·099	−0·147	−0·031	−0·285	0·020	−0·123	−0·175	−0·074	−0·205	+0·000	−0·251	0·042
0·778	−0·131	−0·175	−0·084	−0·219	−0·034	−0·260	0·015	−0·087	−0·128	−0·040	−0·170	0·005	−0·208	0·048
0·854	−0·104	−0·148	−0·052	−0·176	−0·010	−0·215	0·026	−0·057	−0·089	−0·026	−0·116	0·047	−0·149	0·067
0·916	−0·060	−0·092	−0·010	−0·126	+0·027	−0·149	0·052	−0·024	−0·046	+0·003	−0·056	0·075	−0·081	0·090
0·962	+0·015	−0·011	+0·047	−0·039	+0·079	−0·064	0·109	+0·054	+0·032	+0·080	+0·010	+0·106	−0·009	0·129

TABLE 2—*continued*

Theoretical Values of Pressure Coefficient—N.A.C.A. Wing

$M = 0 \cdot 9$

x/c	$y/s = 0 \cdot 40$							$y/s = 0 \cdot 60$						
	$\alpha = 0$	$\alpha = 2 \cdot 4$ deg $[C_L = 0 \cdot 144]$		$\alpha = 4 \cdot 5$ deg $[C_L = 0 \cdot 294]$		$\alpha = 6 \cdot 4$ deg $[C_L = 0 \cdot 432]$		$\alpha = 0$	$\alpha = 2 \cdot 4$ deg $[C_L = 0 \cdot 154]$		$\alpha = 4 \cdot 5$ deg $[C_L = 0 \cdot 330]$		$\alpha = 6 \cdot 4$ deg $[C_L = 0 \cdot 536]$	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	+0·042	−0·238	+0·291	−0·528	0·523	−0·800	0·683	−0·032	−0·562	+0·401	−0·942	0·602	−1·449	0·699
0·038	−0·026	−0·222	0·152	−0·404	0·324	−0·590	0·474	−0·060	−0·370	0·212	−0·592	0·392	−0·935	0·614
0·084	−0·027	−0·165	0·103	−0·310	0·233	−0·447	0·354	−0·054	−0·261	0·139	−0·411	0·273	−0·654	0·461
0·146	−0·047	−0·164	0·072	−0·275	0·175	−0·365	0·270	−0·070	−0·224	0·076	−0·326	0·195	−0·518	0·350
0·222	−0·072	−0·170	+0·024	−0·270	0·123	−0·363	0·218	−0·083	−0·214	0·043	−0·313	0·134	−0·463	0·270
0·309	−0·099	−0·175	−0·002	−0·273	0·076	−0·350	0·150	−0·100	−0·215	+0·020	−0·301	0·027	−0·424	0·197
0·402	−0·123	−0·177	−0·030	−0·268	0·044	−0·335	0·100	−0·118	−0·201	−0·002	−0·288	0·049	−0·376	0·128
0·500	−0·128	−0·189	−0·067	−0·253	0·004	−0·314	0·071	−0·124	−0·200	−0·048	−0·255	0·010	−0·339	0·099
0·598	−0·119	−0·175	−0·054	−0·240	0·000	−0·275	0·056	−0·100	−0·170	−0·025	−0·223	0·002	−0·277	0·075
0·691	−0·093	−0·149	−0·049	−0·200	0·001	−0·226	0·050	−0·074	−0·125	−0·023	−0·172	0·006	−0·212	0·074
0·778	−0·062	−0·100	−0·028	−0·138	0·019	−0·171	0·059	−0·041	−0·080	−0·001	−0·108	0·030	−0·150	0·078
0·854	−0·025	−0·042	+0·003	−0·075	0·040	−0·112	0·074	−0·003	−0·028	+0·026	−0·057	0·048	−0·098	0·082
0·916	+0·000	+0·005	0·050	−0·015	0·068	−0·042	0·097	+0·028	+0·024	0·068	−0·006	0·073	−0·029	0·112
0·962	+0·074	+0·060	+0·092	+0·044	0·113	+0·030	0·137	+0·091	+0·081	+0·110	+0·071	0·123	+0·060	0·145

TABLE 2—*continued*

Theoretical Values of Pressure Coefficient—N.A.C.A. Wing

$M = 0{\cdot}9$

x/c	$y/s = 0{\cdot}75$							$y/s = 0{\cdot}95$						
	$\alpha = 0$	$\alpha = 2{\cdot}4$ deg [$C_L = 0{\cdot}177$]		$\alpha = 4{\cdot}5$ deg [$C_L = 0{\cdot}383$]		$\alpha = 6{\cdot}4$ deg [$C_L = 0{\cdot}586$]		$\alpha = 0$	$\alpha = 2{\cdot}4$ deg [$C_L = 0{\cdot}148$]		$\alpha = 4{\cdot}5$ deg [$C_L = 0{\cdot}397$]		$\alpha = 6{\cdot}4$ deg [$C_L = 0{\cdot}364$]	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	−0·047	−0·621	+0·419	−1·268	0·670	−1·745	0·529	−0·067	−0·670	+0·471	−1·535	0·593	−1·446	0·627
0·038	−0·066	−0·396	0·211	−0·774	0·500	−1·149	0·677	−0·076	−0·384	0·186	−0·908	0·570	−0·840	0·525
0·084	−0·058	−0·247	0·126	−0·514	0·335	−0·781	0·526	−0·067	−0·240	0·100	−0·569	0·373	−0·525	0·340
0·146	−0·067	−0·202	0·075	−0·399	0·247	−0·600	0·417	−0·068	−0·177	0·065	−0·428	0·260	−0·403	0·249
0·222	−0·085	−0·200	+0·029	−0·353	0·170	−0·517	0·308	−0·088	−0·185	+0·001	−0·360	0·172	−0·337	0·150
0·309	−0·101	−0·200	−0·001	−0·325	0·101	−0·460	0·247	−0·099	−0·185	−0·025	−0·326	0·105	−0·303	0·085
0·402	−0·122	−0·195	−0·036	−0·295	0·050	−0·405	0·173	−0·101	−0·176	−0·056	−0·298	0·050	−0·275	0·037
0·500	−0·122	−0·187	−0·060	−0·267	0·020	−0·350	0·109	−0·121	−0·169	−0·081	−0·257	0·015	−0·245	0·002
0·598	−0·100	−0·149	−0·049	−0·211	0·023	−0·278	0·100	−0·087	−0·127	−0·066	−0·208	0·003	−0·200	0·005
0·691	−0·074	−0·102	−0·028	−0·158	0·025	−0·207	0·099	−0·057	−0·091	−0·049	−0·152	0·010	−0·148	0·022
0·778	−0·039	−0·069	−0·008	−0·109	0·034	−0·146	0·081	−0·032	−0·055	−0·012	−0·095	0·030	−0·090	0·024
0·854	+0·000	−0·019	+0·024	−0·049	0·054	−0·066	0·101	+0·002	−0·006	+0·020	−0·045	0·049	−0·027	0·050
0·916	0·037	+0·026	0·053	+0·024	0·092	+0·000	0·125	0·049	+0·045	0·069	+0·006	0·078	+0·026	0·085
0·962	+0·094	+0·088	+0·109	+0·079	0·124	+0·070	0·141	+0·100	+0·097	+0·109	+0·089	0·123	+0·090	0·121

TABLE 3

Theoretical Values of Pressure Coefficient—N.A.C.A. Wing

$M = 0{\cdot}94$

x/c	$y/s = 0{\cdot}16$							$y/s = 0{\cdot}25$						
	$\alpha = 0$	$\alpha = 2{\cdot}4$ deg [$C_L = 0{\cdot}129$]		$\alpha = 4{\cdot}4$ deg [$C_L = 0{\cdot}258$]		$\alpha = 6{\cdot}4$ deg [$C_L = 0{\cdot}386$]		$\alpha = 0$	$\alpha = 2{\cdot}4$ deg [$C_L = 0{\cdot}139$]		$\alpha = 4{\cdot}4$ deg [$C_L = 0{\cdot}278$]		$\alpha = 6{\cdot}4$ deg [$C_L = 0{\cdot}417$]	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	+0·367	+0·260	+0·470	+0·149	+0·575	+0·039	+0·675	+0·181	+0·022	+0·334	−0·136	+0·478	−0·291	0·611
0·038	0·112	+0·019	0·209	−0·077	0·306	−0·172	0·402	0·034	−0·091	0·158	−0·215	0·280	−0·339	0·404
0·084	0·074	−0·011	0·160	−0·097	0·247	−0·181	0·334	+0·014	−0·090	0·120	−0·193	0·224	−0·299	0·329
0·146	+0·027	−0·050	0·102	−0·124	0·190	−0·201	0·275	−0·023	−0·101	0·076	−0·198	0·176	−0·299	0·274
0·222	−0·023	−0·098	0·053	−0·171	0·131	−0·245	0·210	−0·051	−0·137	0·034	−0·220	0·121	−0·308	0·211
0·309	−0·074	−0·127	+0·000	−0·201	0·076	−0·276	0·153	−0·092	−0·163	+0·000	−0·229	0·076	−0·324	0·158
0·402	−0·125	−0·175	−0·050	−0·248	+0·026	−0·304	0·103	−0·122	−0·181	−0·044	−0·252	+0·034	−0·326	0·108
0·500	−0·154	−0·217	−0·086	−0·278	−0·017	−0·339	0·050	−0·146	−0·210	−0·073	−0·277	−0·002	−0·341	0·069
0·598	−0·175	−0·228	−0·104	−0·280	−0·050	−0·326	+0·025	−0·150	−0·200	−0·075	−0·253	−0·006	−0·303	0·050
0·691	−0·175	−0·233	−0·121	−0·285	−0·073	−0·323	−0·001	−0·144	−0·182	−0·075	−0·227	−0·009	−0·276	0·035
0·778	−0·168	−0·214	−0·110	−0·261	−0·057	−0·310	−0·002	−0·109	−0·152	−0·059	−0·199	−0·009	−0·245	0·041
0·854	−0·124	−0·162	−0·074	−0·214	−0·025	−0·259	+0·026	−0·090	−0·101	−0·022	−0·137	+0·026	−0·197	0·056
0·916	−0·073	−0·102	−0·023	−0·151	+0·023	−0·201	0·071	−0·012	−0·042	+0·027	−0·074	0·072	−0·128	0·084
0·962	−0·004	−0·037	+0·034	−0·070	+0·071	−0·101	+0·108	+0·049	+0·021	+0·079	−0·004	+0·109	−0·031	0·140

TABLE 3—*continued*

Theoretical Values of Pressure Coefficient—N.A.C.A. Wing

$M = 0 \cdot 94$

x/c	$y/s = 0 \cdot 40$							$y/s = 0 \cdot 60$						
	$\alpha = 0$	$\alpha = 2 \cdot 4$ deg [$C_L = 0 \cdot 157$]		$\alpha = 4 \cdot 4$ deg [$C_L = 0 \cdot 316$]		$\alpha = 6 \cdot 4$ deg [$C_L = 0 \cdot 469$]		$\alpha = 0$	$\alpha = 2 \cdot 4$ deg [$C_L = 0 \cdot 176$]		$\alpha = 4 \cdot 4$ deg [$C_L = 0 \cdot 361$]		$\alpha = 6 \cdot 4$ deg [$C_L = 0 \cdot 558$]	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	+0·082	−0·173	+0·311	−0·433	0·525	−0·682	0·690	−0·015	−0·399	+0·369	−0·830	0·644	−1·243	0·730
0·038	−0·014	−0·198	0·159	−0·384	0·333	−0·561	0·489	−0·044	−0·305	0·195	−0·581	0·429	−0·878	0·620
0·084	−0·020	−0·160	0·116	−0·304	0·252	−0·442	0·380	−0·045	−0·224	0·119	−0·420	0·292	−0·648	0·475
0·146	−0·028	−0·151	0·075	−0·298	0·105	−0·403	0·311	−0·053	−0·198	0·074	−0·350	0·224	−0·522	0·370
0·222	−0·070	−0·172	0·032	−0·277	0·138	−0·379	0·238	−0·083	−0·202	0·035	−0·330	0·160	−0·471	0·291
0·309	−0·100	−0·178	+0·000	−0·287	0·100	−0·375	0·173	−0·103	−0·211	+0·000	−0·309	0·111	−0·447	0·222
0·402	−0·126	−0·200	−0·027	−0·288	0·052	−0·368	0·123	−0·126	−0·218	−0·030	−0·299	0·060	−0·404	0·034
0·500	−0·137	−0·205	−0·061	−0·277	0·013	−0·347	0·087	−0·132	−0·205	−0·062	−0·283	0·018	−0·368	0·108
0·598	−0·126	−0·199	−0·073	−0·250	0·002	−0·302	0·072	−0·129	−0·182	−0·072	−0·245	0·005	−0·319	0·085
0·691	−0·118	−0·160	−0·068	−0·213	0·000	−0·151	0·052	−0·104	−0·140	−0·050	−0·187	0·011	−0·250	0·075
0·778	−0·078	−0·119	−0·026	−0·161	0·022	−0·203	0·069	−0·057	−0·093	−0·017	−0·132	0·029	−0·176	0·080
0·854	−0·032	−0·056	+0·023	−0·103	0·034	−0·126	0·076	+0·000	−0·032	+0·034	−0·053	0·074	−0·099	0·100
0·916	+0·027	−0·006	0·071	−0·037	0·082	−0·054	0·104	0·054	+0·024	0·076	+0·004	0·103	−0·019	0·125
0·962	+0·079	+0·060	+0·101	+0·040	0·128	+0·022	0·150	+0·100	+0·088	+0·119	+0·073	0·139	+0·060	0·160

TABLE 3—*continued*

Theoretical Values of Pressure Coefficient—N.A.C.A. Wing

$M = 0{\cdot}94$

x/c	$y/s = 0{\cdot}75$							$y/s = 0{\cdot}95$						
	$\alpha = 0$	$\alpha = 2{\cdot}4$ deg $[C_L = 0{\cdot}193]$		$\alpha = 4{\cdot}4$ deg $[C_L = 0{\cdot}405]$		$\alpha = 6{\cdot}4$ deg $[C_L = 0{\cdot}632]$		$\alpha = 0$	$\alpha = 2{\cdot}4$ deg $[C_L = 0{\cdot}163]$		$\alpha = 4{\cdot}4$ deg $[C_L = 0{\cdot}371]$		$\alpha = 6{\cdot}4$ deg $[C_L = 0{\cdot}484]$	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	−0·019	−0·581	+0·444	−1·162	0·685	−1·627	0·455	−0·053	−0·723	+0·451	−1·550	+0·629	—	0·262
0·038	−0·063	−0·374	0·230	−0·740	0·511	−1·147	0·701	−0·079	−0·422	0·218	−0·860	0·593	−1·100	0·711
0·084	−0·056	−0·269	0·144	−0·522	0·356	−0·801	0·562	−0·069	−0·288	0·121	−0·563	0·362	−0·715	0·499
0·146	−0·059	−0·226	0·086	−0·401	0·271	−0·640	0·451	−0·075	−0·224	0·061	−0·424	0·247	−0·520	0·342
0·222	−0·088	−0·218	0·042	−0·368	0·184	−0·545	0·348	−0·093	−0·210	+0·015	−0·363	0·152	−0·447	0·229
0·309	−0·110	−0·218	+0·001	−0·335	0·127	−0·482	0·249	−0·105	−0·200	−0·022	−0·331	0·091	−0·392	0·139
0·402	−0·126	−0·217	−0·026	−0·318	0·076	−0·438	0·176	−0·125	−0·196	−0·049	−0·300	0·037	−0·339	0·075
0·500	−0·130	−0·217	−0·056	−0·290	0·029	−0·380	0·134	−0·125	−0·180	−0·069	−0·250	+0·003	−0·289	0·042
0·598	−0·118	−0·169	−0·027	−0·250	0·001	−0·315	0·102	−0·110	−0·150	−0·050	−0·208	−0·024	−0·226	0·027
0·691	−0·089	−0·098	−0·003	−0·190	0·007	−0·234	0·095	−0·078	−0·101	−0·028	−0·152	−0·010	−0·158	0·027
0·778	−0·042	−0·079	−0·007	−0·118	0·043	−0·161	0·098	−0·030	−0·051	−0·005	−0·080	+0·029	−0·094	0·047
0·854	+0·022	+0·024	+0·052	−0·032	0·076	−0·074	0·100	+0·023	+0·016	+0·048	−0·022	0·068	−0·024	0·071
0·916	0·072	0·069	0·091	+0·032	0·110	−0·001	0·125	0·073	0·075	0·097	+0·049	0·100	+0·048	0·101
0·962	+0·111	+0·101	+0·122	+0·090	0·140	+0·077	0·163	+0·120	+0·118	+0·130	+0·111	+0·141	+0·110	0·149

TABLE 4

Theoretical Values of Pressure Coefficient—Warren Wing 12

$M = 0{\cdot}6$

x/c	$y/s = 0{\cdot}1$							$y/s = 0{\cdot}4$						
	$\alpha = 0$	$\alpha = 2$ deg $[C_L = 0{\cdot}095]$		$\alpha = 4$ deg $[C_L = 0{\cdot}177]$		$\alpha = 6$ deg $[C_L = 0{\cdot}268]$		$\alpha = 0$	$\alpha = 2$ deg $[C_L = 0{\cdot}114]$		$\alpha = 4$ deg $[C_L = 0{\cdot}229]$		$\alpha = 6$ deg $[C_L = 0{\cdot}340]$	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	+0·144	+0·030	+0·245	−0·073	+0·329	−0·190	0·419	+0·066	−0·125	+0·235	−0·358	+0·360	−0·619	0·449
0·038	−0·042	−0·168	0·061	−0·276	0·149	−0·397	0·249	−0·098	−0·307	0·089	−0·539	0·246	−0·790	0·370
0·084	−0·069	−0·158	+0·017	−0·237	0·086	−0·323	0·162	−0·108	−0·244	+0·019	−0·388	0·138	−0·539	0·236
0·146	−0·096	−0·153	−0·023	−0·226	0·029	−0·300	0·104	−0·126	−0·224	−0·029	−0·349	0·078	−0·448	0·148
0·222	−0·113	−0·181	−0·052	−0·237	+0·003	−0·299	0·065	−0·135	−0·218	−0·053	−0·303	0·021	−0·390	0·099
0·309	−0·125	−0·193	−0·075	−0·227	−0·020	−0·284	0·036	−0·147	−0·200	−0·074	−0·274	+0·000	−0·342	0·050
0·402	−0·131	−0·178	−0·077	−0·210	−0·028	−0·263	0·020	−0·127	−0·177	−0·075	−0·229	−0·022	−0·287	0·025
0·500	−0·122	−0·167	−0·080	−0·201	−0·041	−0·241	0·000	−0·111	−0·157	−0·070	−0·200	−0·024	−0·242	0·019
0·598	−0·100	−0·134	−0·058	−0·170	−0·027	−0·201	0·002	−0·076	−0·124	−0·045	−0·151	−0·001	−0·186	0·025
0·691	−0·070	−0·100	−0·028	−0·125	−0·010	−0·158	0·022	−0·050	−0·072	−0·014	−0·102	+0·024	−0·128	0·036
0·778	−0·037	−0·061	−0·012	−0·082	+0·011	−0·105	0·040	−0·020	−0·039	+0·005	−0·059	−0·029	−0·079	0·051
0·854	−0·001	−0·022	+0·025	−0·029	0·047	−0·050	0·058	+0·019	+0·000	0·050	−0·008	0·057	−0·019	0·076
0·916	+0·028	+0·010	0·050	+0·002	0·062	−0·003	0·075	0·050	0·036	0·074	+0·027	0·076	+0·027	0·100
0·962	+0·058	+0·044	+0·070	+0·037	+0·083	+0·027	0·098	+0·076	+0·070	+0·088	+0·067	+0·098	+0·060	0·107

(80220)

TABLE 4—*continued*

Theoretical Values of Pressure Coefficient—Warren Wing 12

$M = 0{\cdot}6$

x/c	$y/s = 0{\cdot}7$							$y/s = 0{\cdot}9$						
	$\alpha = 0$	$\alpha = 2$ deg $[C_L = 0{\cdot}138]$		$\alpha = 4$ deg $[C_L = 0{\cdot}271]$		$\alpha = 6$ deg $[C_L = 0{\cdot}395]$		$\alpha = 0$	$\alpha = 2$ deg $[C_L = 0{\cdot}130]$		$\alpha = 4$ deg $[C_L = 0{\cdot}246]$		$\alpha = 6$ deg $[C_L = 0{\cdot}366]$	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	+0·063	−0·228	+0·287	−0·595	+0·427	−0·962	0·481	+0·060	−0·249	+0·288	−0·569	−0·419	−0·950	0·483
0·038	−0·094	−0·380	0·135	−0·693	0·330	−1·008	0·448	−0·095	−0·376	0·142	−0·658	0·310	−0·968	0·436
0·084	−0·104	−0·277	+0·049	−0·463	0·189	−0·636	0·292	−0·102	−0·268	+0·047	−0·427	0·169	−0·610	0·281
0·146	−0·128	−0·249	−0·003	−0·379	0·114	−0·525	0·214	−0·111	−0·238	−0·025	−0·353	0·085	−0·475	0·178
0·222	−0·127	−0·229	−0·028	−0·336	0·060	−0·431	0·140	−0·123	−0·224	−0·032	−0·313	0·047	−0·407	0·125
0·309	−0·126	−0·206	−0·053	−0·301	+0·015	−0·377	0·099	−0·125	−0·200	−0·031	−0·275	+0·000	−0·351	0·093
0·402	−0·123	−0·178	−0·064	−0·252	−0·009	−0·319	0·072	−0·122	−0·175	−0·037	−0·235	−0·014	−0·289	0·052
0·500	−0·101	−0·153	−0·053	−0·205	−0·004	−0·250	0·039	−0·102	−0·148	−0·057	−0·189	−0·015	−0·230	0·028
0·598	−0·075	−0·124	−0·038	−0·153	−0·001	−0·200	0·035	−0·075	−0·116	−0·025	−0·127	+0·001	−0·175	0·025
0·691	−0·047	−0·076	−0·018	−0·101	+0·021	−0·137	0·023	−0·049	−0·075	−0·010	−0·078	0·025	−0·123	0·028
0·778	−0·018	−0·037	+0·010	−0·058	0·035	−0·076	0·058	−0·013	−0·032	+0·009	−0·049	0·029	−0·066	0·049
0·854	+0·024	+0·000	0·042	−0·006	0·057	−0·024	0·075	+0·022	+0·000	0·046	+0·022	0·070	−0·016	0·072
0·916	0·049	0·044	0·062	+0·029	0·076	+0·025	0·099	0·048	0·035	0·048	0·052	0·077	+0·025	0·077
0·962	+0·073	+0·070	+0·084	+0·064	+0·094	+0·060	0·102	+0·072	+0·070	+0·081	+0·065	+0·090	+0·060	0·097

B

TABLE 5

Theoretical Values of Pressure Coefficient—Warren Wing 12

$M = 0{\cdot}85$

x/c	$y/s = 0{\cdot}10$							$y/s = 0{\cdot}4$						
	$\alpha = 0$	$\alpha = 2$ deg [$C_L = 0{\cdot}085$]		$\alpha = 4$ deg [$C_L = 0{\cdot}177$]		$\alpha = 6$ deg [$C_L = 0{\cdot}267$]		$\alpha = 0$	$\alpha = 2$ deg [$C_L = 0{\cdot}110$]		$\alpha = 4$ deg [$C_L = 0{\cdot}235$]		$\alpha = 6$ deg [$C_L = 0{\cdot}370$]	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	+0·259	+0·183	+0·329	+0·100	+0·403	+0·017	+0·474	+0·120	−0·024	+0·250	−0·212	+0·378	−0·430	0·474
0·038	+0·018	−0·072	0·105	−0·169	0·199	−0·269	0·288	−0·065	−0·241	0·098	−0·455	0·267	−0·685	0·412
0·084	−0·034	−0·104	+0·036	−0·180	0·110	−0·256	0·184	−0·089	−0·211	+0·027	−0·354	0·148	−0·509	0·272
0·146	−0·075	−0·145	−0·009	−0·200	0·051	−0·273	0·127	−0·124	−0·215	−0·022	−0·300	0·090	−0·451	0·193
0·222	−0·105	−0·160	−0·046	−0·219	+0·021	−0·279	0·076	−0·127	−0·208	−0·048	−0·301	0·040	−0·403	0·131
0·309	−0·126	−0·176	−0·073	−0·230	−0·024	−0·283	0·049	−0·129	−0·200	−0·073	−0·276	+0·000	−0·374	0·090
0·402	−0·130	−0·180	−0·099	−0·247	−0·050	−0·287	+0·008	−0·126	−0·185	−0·077	−0·250	−0·014	−0·325	0·011
0·500	−0·146	−0·184	−0·101	−0·228	−0·055	−0·269	−0·009	−0·125	−0·169	−0·078	−0·220	−0·023	−0·277	0·036
0·598	−0·125	−0·169	−0·098	−0·199	−0·049	−0·236	−0·001	−0·100	−0·126	−0·062	−0·174	−0·002	−0·225	0·001
0·691	−0·098	−0·126	−0·074	−0·150	−0·026	−0·190	+0·007	−0·074	−0·090	−0·029	−0·123	+0·001	−0·160	0·020
0·778	−0·063	−0·089	−0·034	−0·114	−0·003	−0·141	0·029	−0·028	−0·049	+0·000	−0·074	0·029	−0·101	0·060
0·854	−0·025	−0·034	−0·002	−0·056	+0·025	−0·076	0·053	+0·002	−0·001	0·038	−0·019	0·052	−0·037	0·077
0·916	+0·005	−0·002	+0·026	−0·024	0·050	−0·035	0·075	0·049	+0·036	0·074	+0·025	0·076	+0·010	0·100
0·962	+0·043	+0·030	+0·060	+0·017	+0·079	+0·001	+0·098	+0·076	+0·069	+0·089	+0·060	+0·100	+0·051	0·115

TABLE 5—*continued*

Theoretical Values of Pressure Coefficient—Warren Wing 12

$M = 0 \cdot 85$

x/c	$y/s = 0 \cdot 70$							$y/s = 0 \cdot 9$						
	$\alpha = 0$	$\alpha = 2$ deg [$C_L = 0 \cdot 140$]		$\alpha = 4$ deg [$C_L = 0 \cdot 266$]		$\alpha = 6$ deg [$C_L = 0 \cdot 408$]		$\alpha = 0$	$\alpha = 2$ deg [$C_L = 0 \cdot 120$]		$\alpha = 4$ deg [$C_L = 0 \cdot 249$]		$\alpha = 6$ deg [$C_L = 0 \cdot 424$]	
		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.		U.S.	L.S.	U.S.	L.S.	U.S.	L.S.
0·010	+0·079	−0·196	+0·300	−0·474	+0·427	−0·812	0·496	+0·054	−0·249	+0·280	−0·620	+0·457	−1·155	0·473
0·038	−0·092	−0·372	0·147	−0·637	0·331	−0·942	0·479	−0·108	−0·384	0·128	−0·697	0·347	−1·123	0·528
0·084	−0·105	−0·280	0·052	−0·443	0·185	−0·639	0·323	−0·115	−0·270	+0·031	−0·448	0·178	−0·717	0·363
0·146	−0·126	−0·164	+0·000	−0·373	0·100	−0·550	0·137	−0·125	−0·224	−0·024	−0·350	0·095	−0·537	0·141
0·222	−0·135	−0·243	−0·038	−0·341	0·052	−0·452	0·151	−0·136	−0·220	−0·051	−0·320	0·038	−0·465	0·161
0·309	−0·147	−0·225	−0·074	−0·308	+0·024	−0·400	0·066	−0·141	−0·200	−0·076	−0·277	+0·000	−0·377	0·004
0·402	−0·136	−0·200	−0·075	−0·168	−0·002	−0·334	0·050	−0·128	−0·176	−0·079	−0·241	−0·024	−0·324	0·050
0·500	−0·118	−0·169	−0·062	−0·217	−0·015	−0·271	0·039	−0·116	−0·150	−0·075	−0·195	−0·027	−0·258	0·039
0·598	−0·088	−0·126	−0·050	−0·174	−0·001	−0·204	0·033	−0·076	−0·103	−0·049	−0·136	−0·010	−0·093	0·026
0·691	−0·050	−0·091	−0·024	−0·121	+0·009	−0·149	0·045	−0·049	−0·071	−0·023	−0·083	+0·001	−0·125	0·040
0·778	−0·014	−0·038	+0·009	−0·059	0·032	−0·081	0·060	−0·010	−0·025	+0·003	−0·041	0·027	−0·067	0·059
0·854	+0·025	+0·000	0·040	−0·007	0·061	−0·026	0·075	+0·025	+0·023	0·051	+0·002	0·065	−0·003	0·075
0·916	0·058	0·045	0·070	+0·028	0·079	+0·024	0·097	0·053	0·057	0·076	0·052	0·089	+0·047	0·096
0·962	+0·085	+0·080	+0·096	+0·075	+0·105	+0·070	0·117	+0·092	+0·090	+0·100	+0·089	+0·107	+0·083	0·117

TABLE 6

Values of $\tau_a(1 + K_1)$

Spanwise station (per cent semi-span)	N.A.C.A. wing		
Values obtained by measuring η from the vertical plane of symmetry			
	$M = 0 \cdot 8$	$M = 0 \cdot 9$	$M = 0 \cdot 94$
16	1·151	1·254	1·346
25	1·162	1·291	1·421
40	1·113	1·230	1·319
60	1·064	1·101	1·145
75	1·056	1·076	1·108
Values obtained by measuring η from the bodyside			
	$M = 0 \cdot 8$	$M = 0 \cdot 9$	$M = 0 \cdot 94$
16	0·953	0·974	1·003
25	1·095	1·163	1·256
40	1·162	1·291	1·416
60	1·104	1·197	1·279
75	1·065	1·105	1·150

Spanwise station (per cent semi-span)	Warren wing		
	$M = 0 \cdot 6$		$M = 0 \cdot 85$
10	1·096		1·180
40	1·130		1·292
70	1·064		1·146
90	1·024		0·868

TABLE 7

Warren Wing

Values of μ_a

M	0·6	0·85
y/s		
0·10	0·382	0·486
0·40	0·028	0·110
0·70	0	0
0·90	−0·020	−0·072

Values of n_a

M	0·6	0·85
y/s		
0·10	0·303	0·235
0·40	0·450	0·394
0·70	0·500	0·489
0·90	0·523	0·569

Values of K_a

M	0·6	0·85
y/s		
0·10	1·0	1·0
0·40	1·0	1·0
0·70	1·0	1·0
0·90	1·0	1·0

N.A.C.A. Wing

Values of μ_a

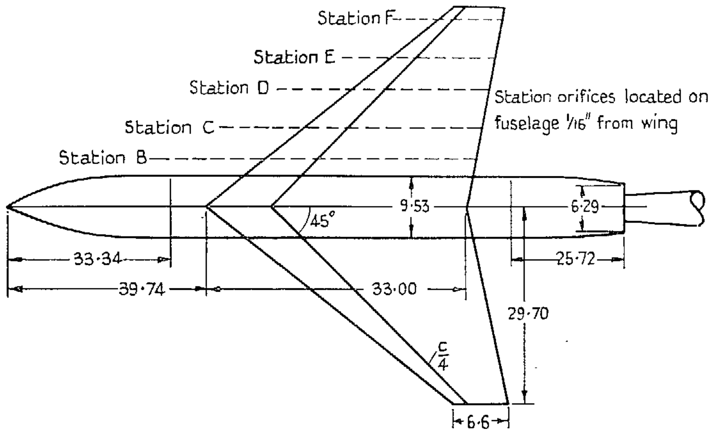
M	0·8	0·9	0·94
y/s			
0·16	1·0	1·0	1·0
0·25	0·421	0·495	0·550
0·40	0·153	0·246	0·275
0·60	0	0·040	0·092
0·75	0	0	0

Values of n_a

M	0·8	0·9	0·94
y/s			
0·16	0·209	0·163	0·132
0·25	0·294	0·239	0·200
0·40	0·389	0·335	0·296
0·60	0·471	0·434	0·397
0·75	0·500	0·483	0·461

Values of K_a

M	0·8	0·9	0·94
y/s			
0·16	1·0	1·0	1·0
0·25	1·0	1·0	1·0
0·40	1·0	1·0	1·0
0·60	1·0	1·0	1·0
0·75	1·0	1·0	1·0
0·95	1·0	1·0	1·0



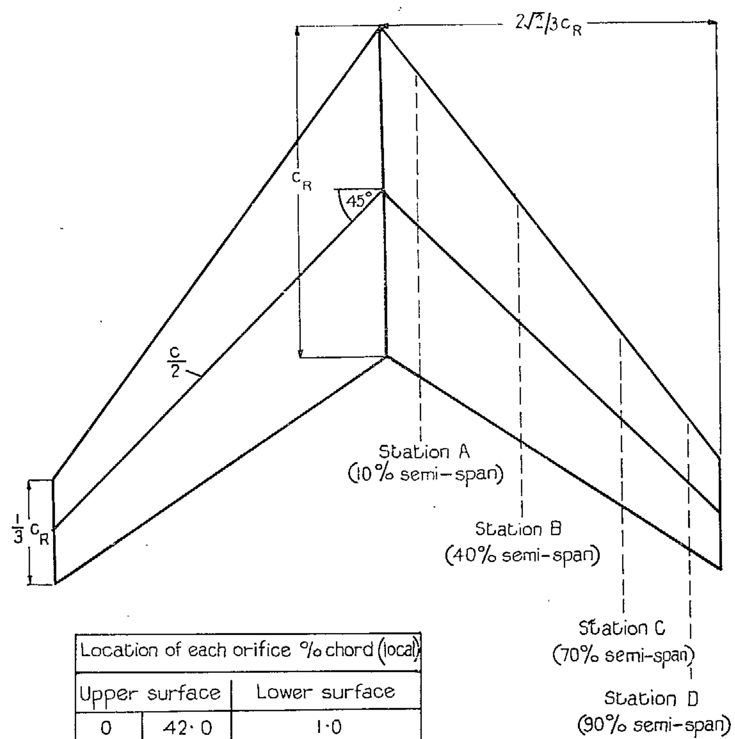
Body ordinates			
x	r	x	r
0·000	0·000	24·000	4·396
0·500	0·144	26·000	4·536
1·000	0·286	28·000	4·643
1·500	0·426	30·000	4·716
2·000	0·564	32·000	4·755
3·000	0·832	33·333	4·763
4·000	1·091	78·582	4·763
5·000	1·341	79·000	4·757
6·000	1·582	79·250	4·752
7·000	1·812	79·500	4·746
8·000	2·035	80·000	4·728
9·000	2·249	80·500	4·708
10·000	2·454	81·000	4·685
10·500	2·551	81·916	4·639
11·000	2·649	83·500	4·557
11·625	2·766	85·250	4·458
12·000	2·834	87·000	4·345
14·000	3·182	88·000	4·278
16·000	3·493	89·000	4·209
18·000	3·770	90·965	4·067
19·000	3·896	97·362	3·624
20·000	4·014	104·300	3·143
22·000	4·223		

Wing data	
Aspect ratio	3·0
Taper ratio	0·2
Wing area	8·165 sq. ft
Aerofoil section	65A 004

Spanwise station location % semi-span	
A	16·00
B	25·00
C	40·00
D	60·00
E	75·00
F	95·00

Location of each orifice % chord		
0	25·00	65·00
1·25	30·00	70·00
2·50	35·00	75·00
5·00	40·00	80·00
7·50	45·00	85·00
10·00	50·00	90·00
15·00	55·00	95·00
20·00	60·00	

FIG. 1. The N.A.C.A. model as tested
(All dimensions in inches unless stated otherwise).



Location of each orifice % chord (local)		
Upper surface		Lower surface
0	42·0	1·0
0·2	48·0	10·0
0·8	60·0	22·0
1·0	65·0	36·0
2·5	70·0	48·0
3·0	80·0	60·0
5·0	85·0	72·0
10·0	90·0	80·0
16·0	95·0	88·0
22·0	100·0	93·0
34·0		95·0

Aspect ratio $= 2\sqrt{2}$
Taper ratio $= \frac{1}{3}$
Aerofoil section = R.A.E. 102

FIG. 2. Plan-form of Warren Wing 12.—Diagrammatic.

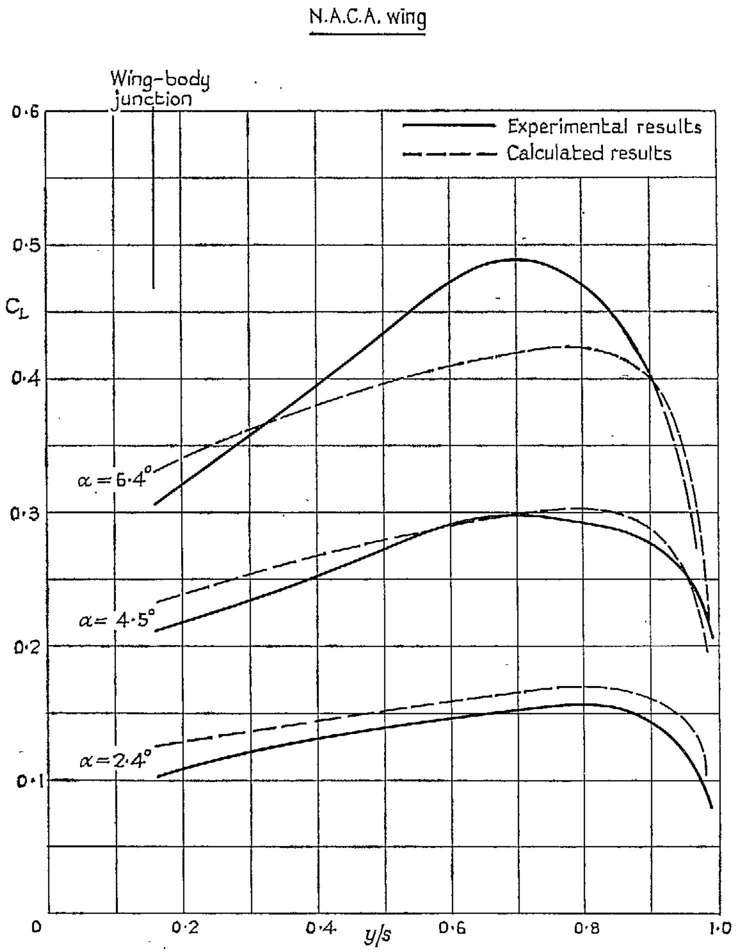


FIG. 3. $M = 0 \cdot 8$, $C_L \sim y/s$, $a_0 = 2\pi$.

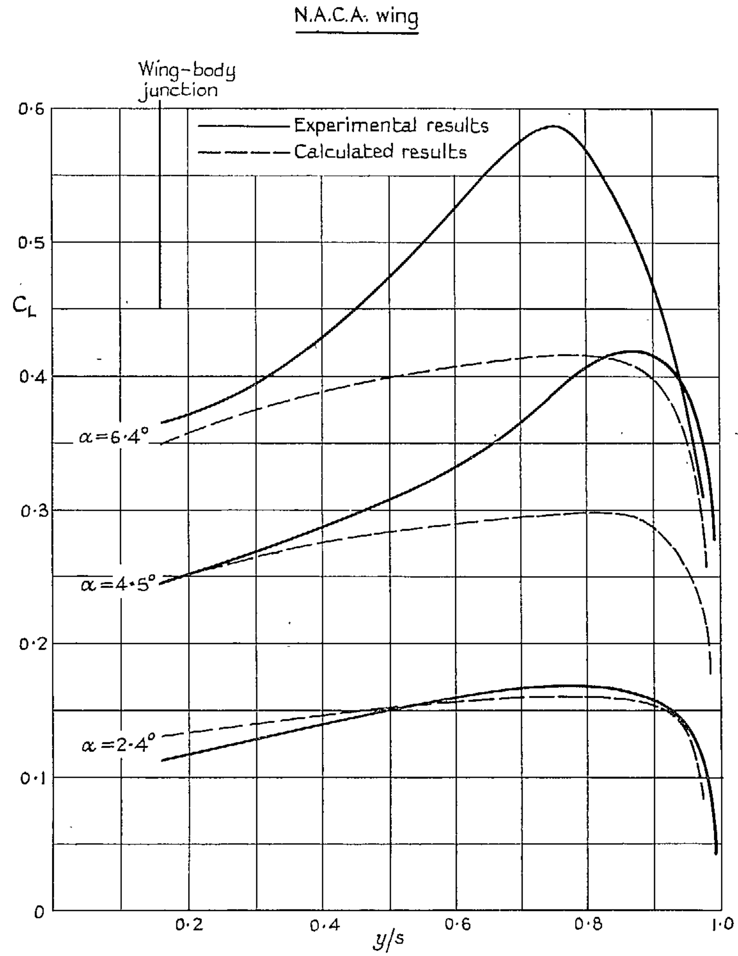


FIG. 4. $M = 0 \cdot 9$, $C_L \sim y/s$, $a_0 = 2\pi$.

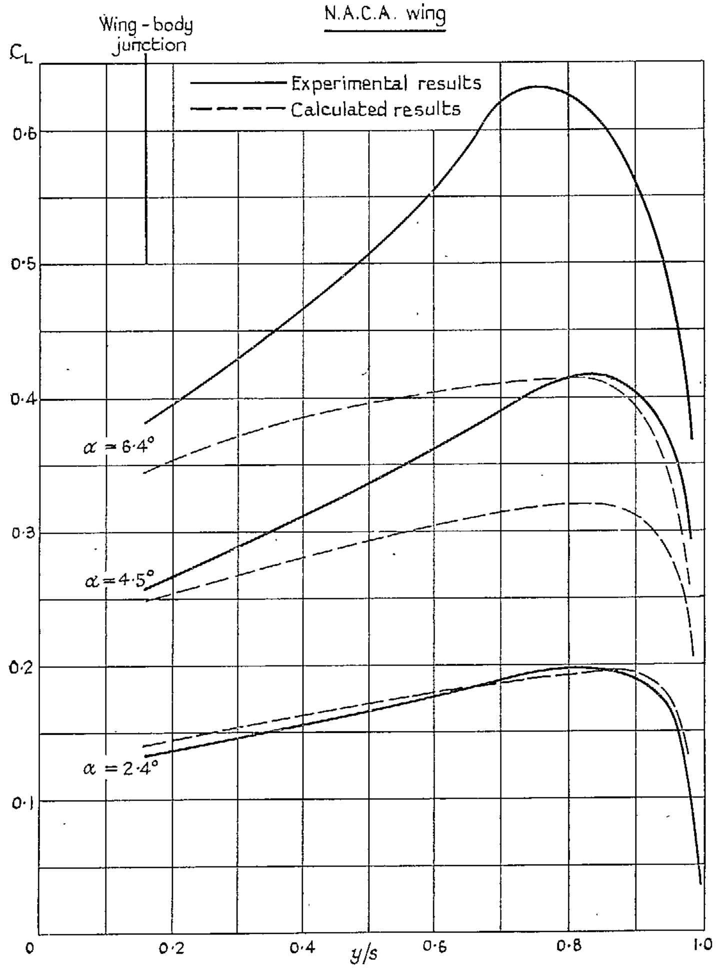


FIG. 5. $M = 0{\cdot}94$, $C_L \sim y/s$, $a_0 = 2\pi$.

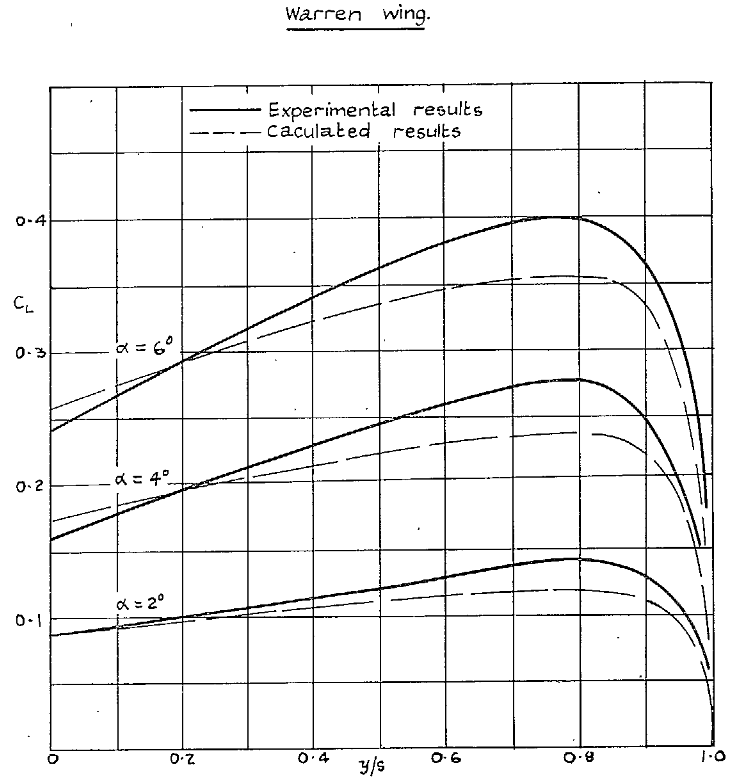


FIG. 6. $M = 0{\cdot}6$, $C_L \sim y/s$, $a_0 = 2\pi$.

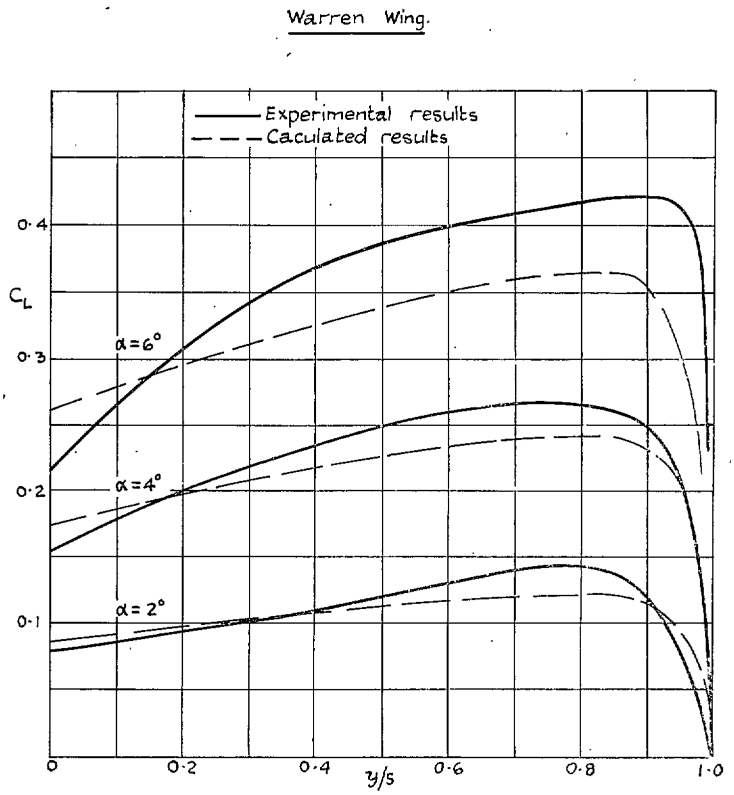
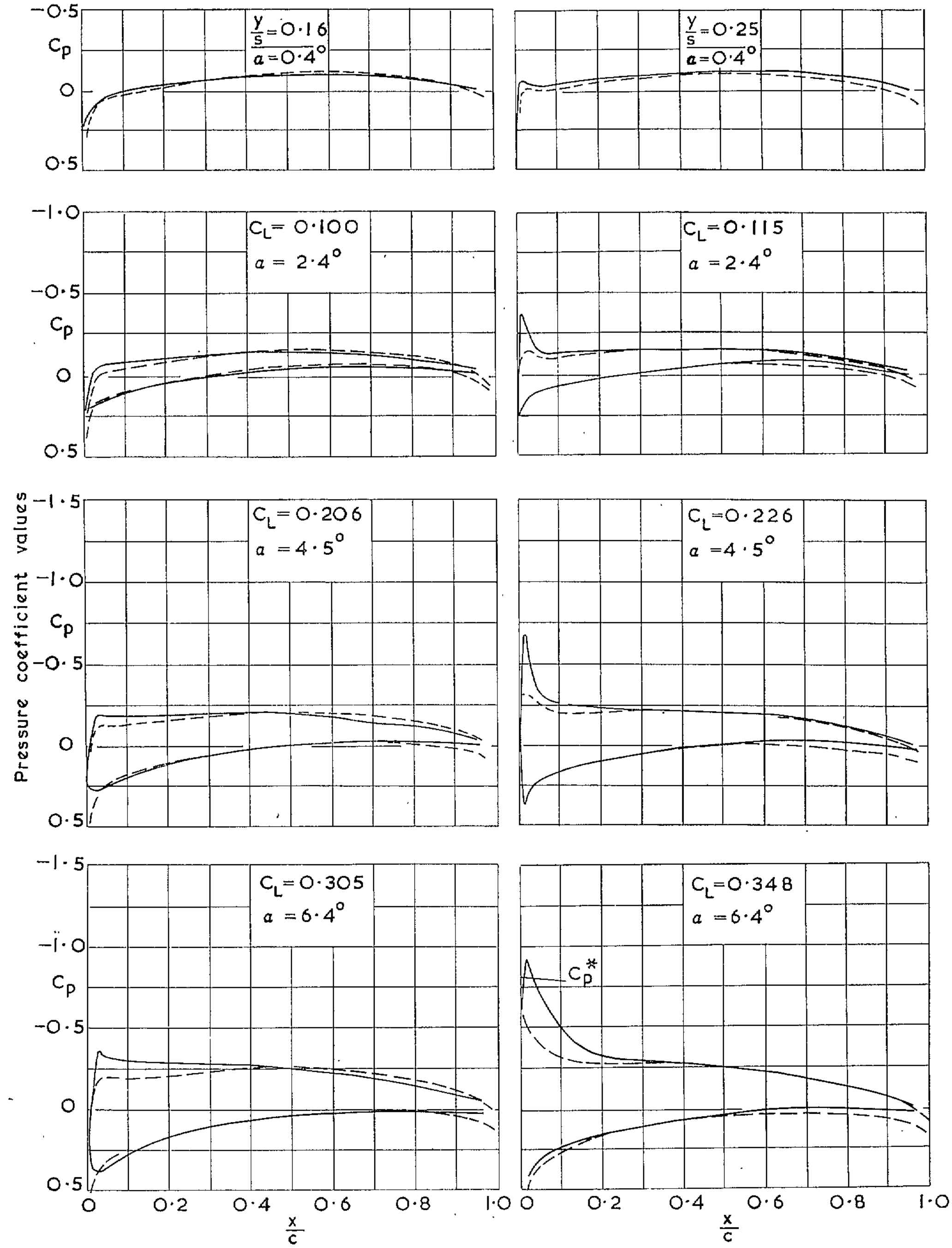


FIG. 7. $M = 0 \cdot 85$, $C_L \sim y/s$, $a_0 = 2\pi$.



——— Experimental results

– – – – Calculated results assuming experimental lift coefficient

x x x Calculated results assuming theoretical lift coefficient

FIG. 8a. Chordwise pressure distributions. $M = 0{\cdot}8$.

N.A.C.A. WING

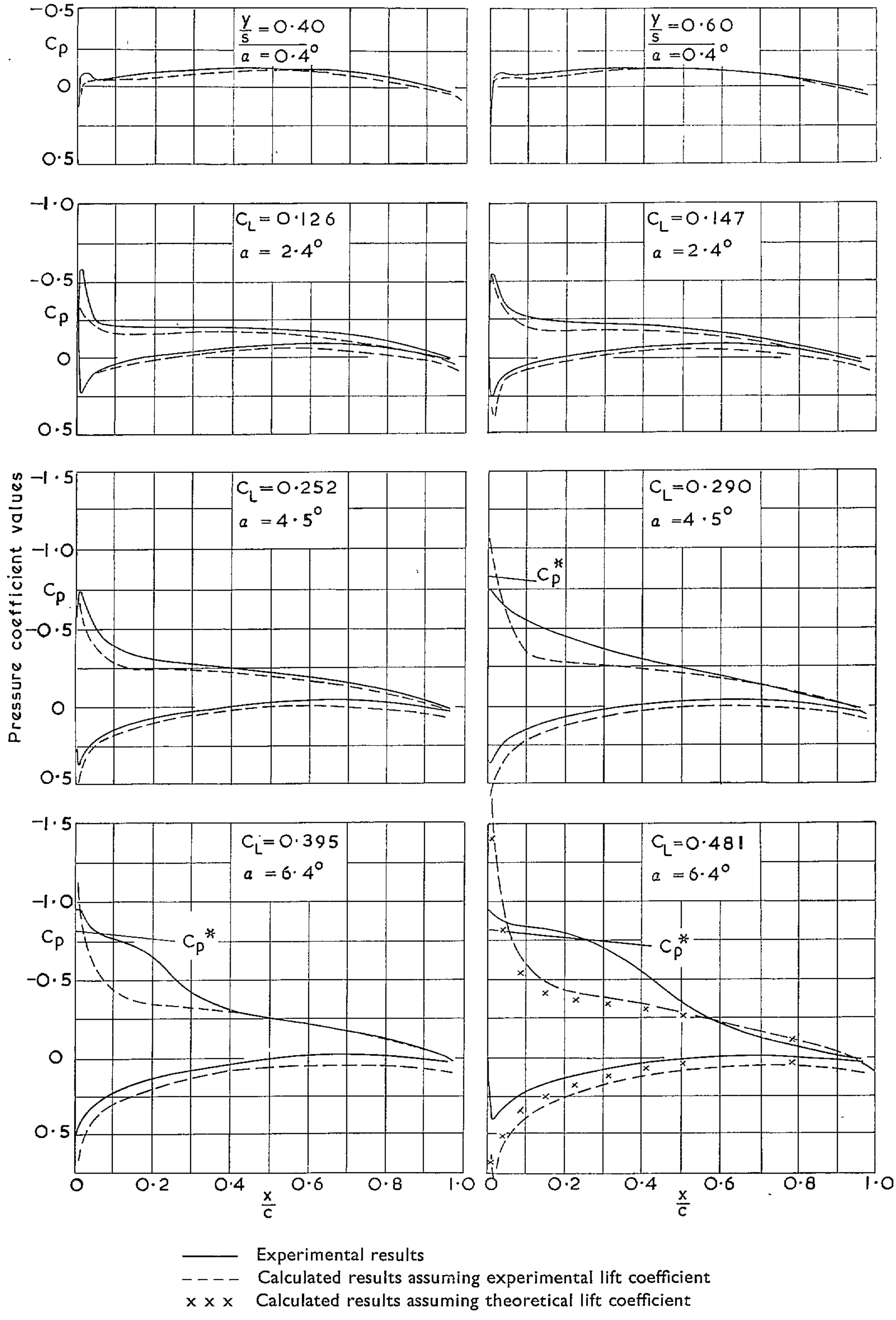


FIG. 8b. Chordwise pressure distributions. $M = 0\cdot8$.

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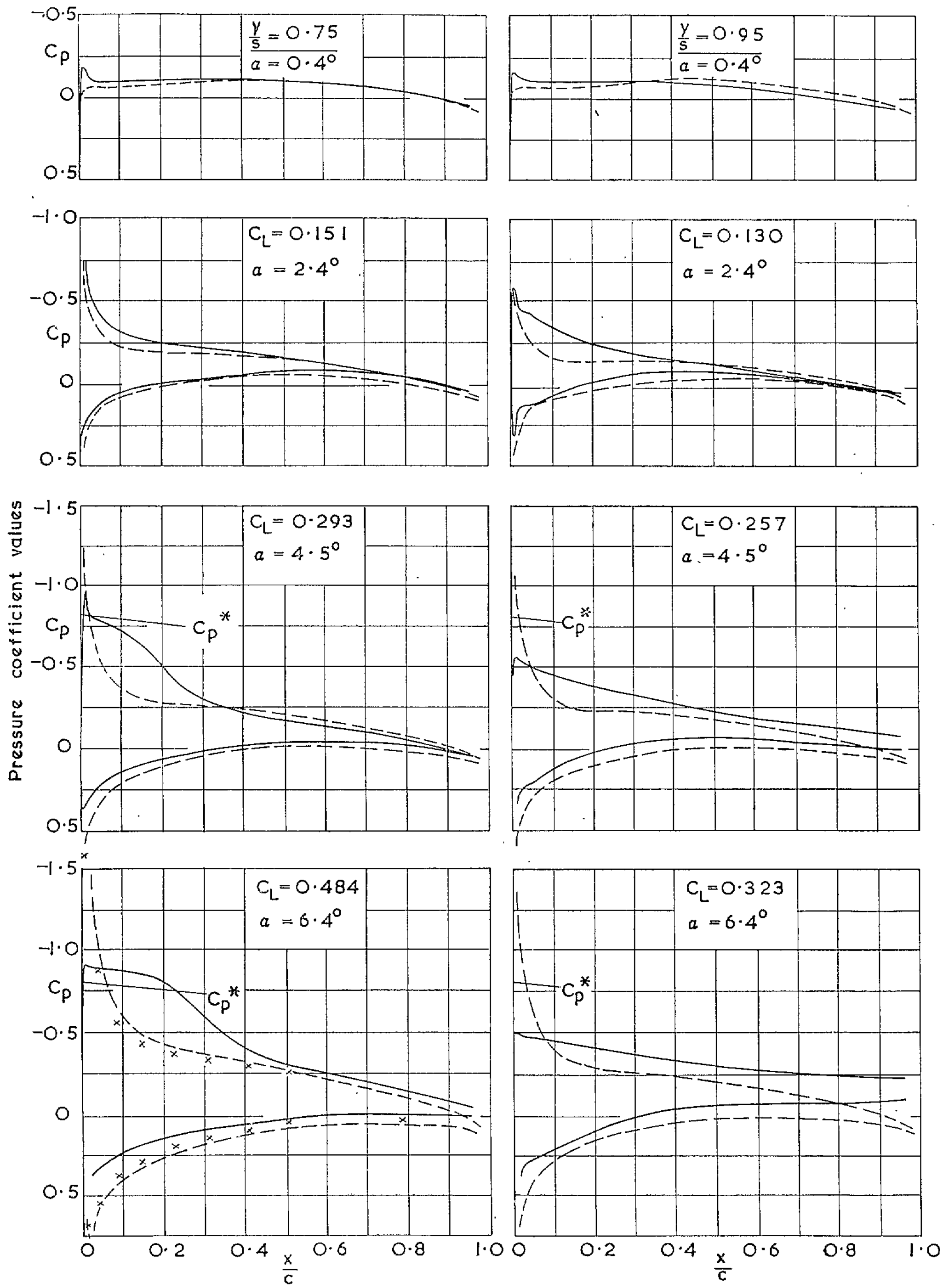

——— Experimental results
– – – – Calculated results assuming experimental lift coefficient
x x x Calculated results assuming theoretical lift coefficient

FIG. 8c. Chordwise pressure distributions. $M = 0 \cdot 8$.

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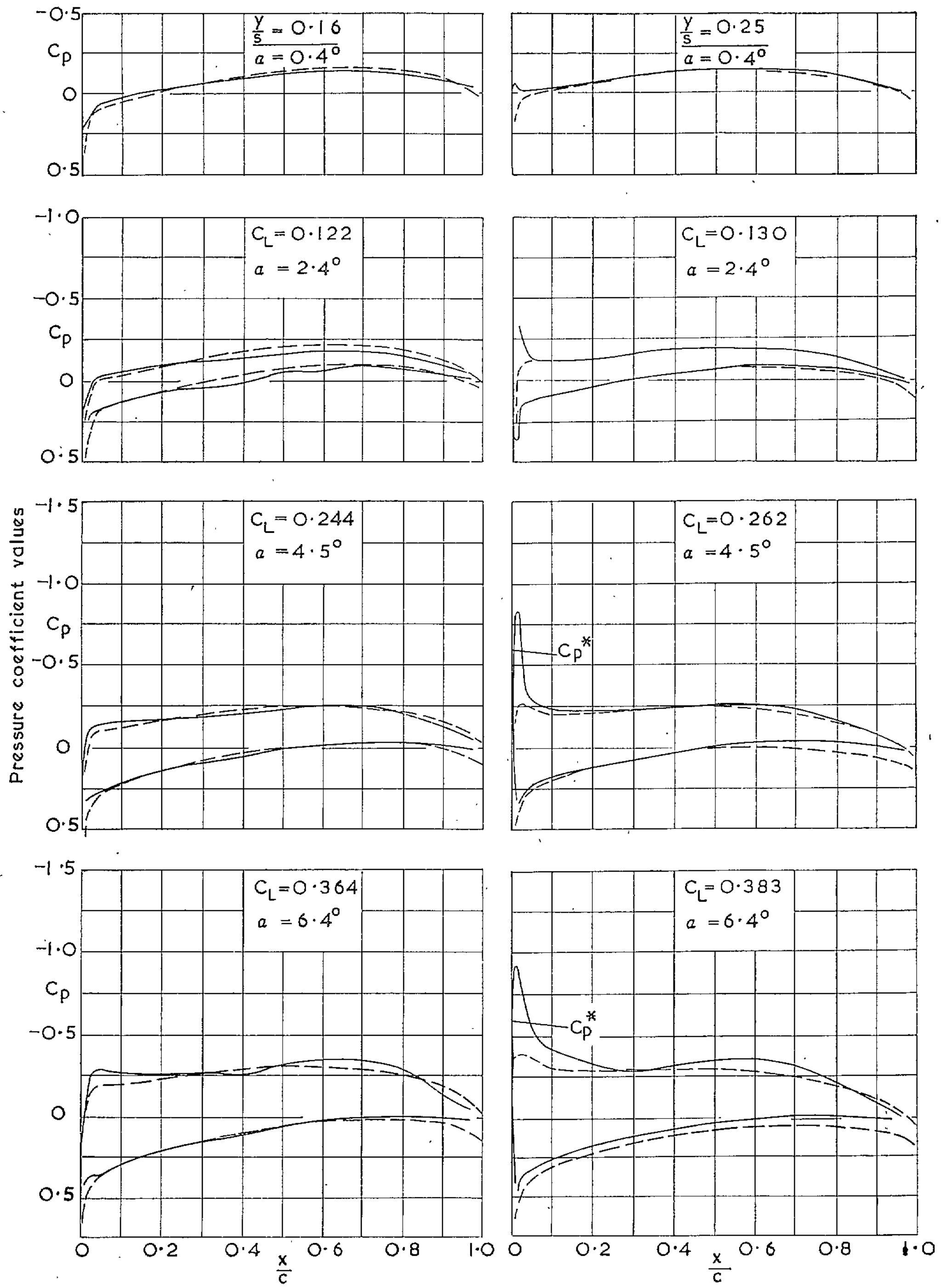
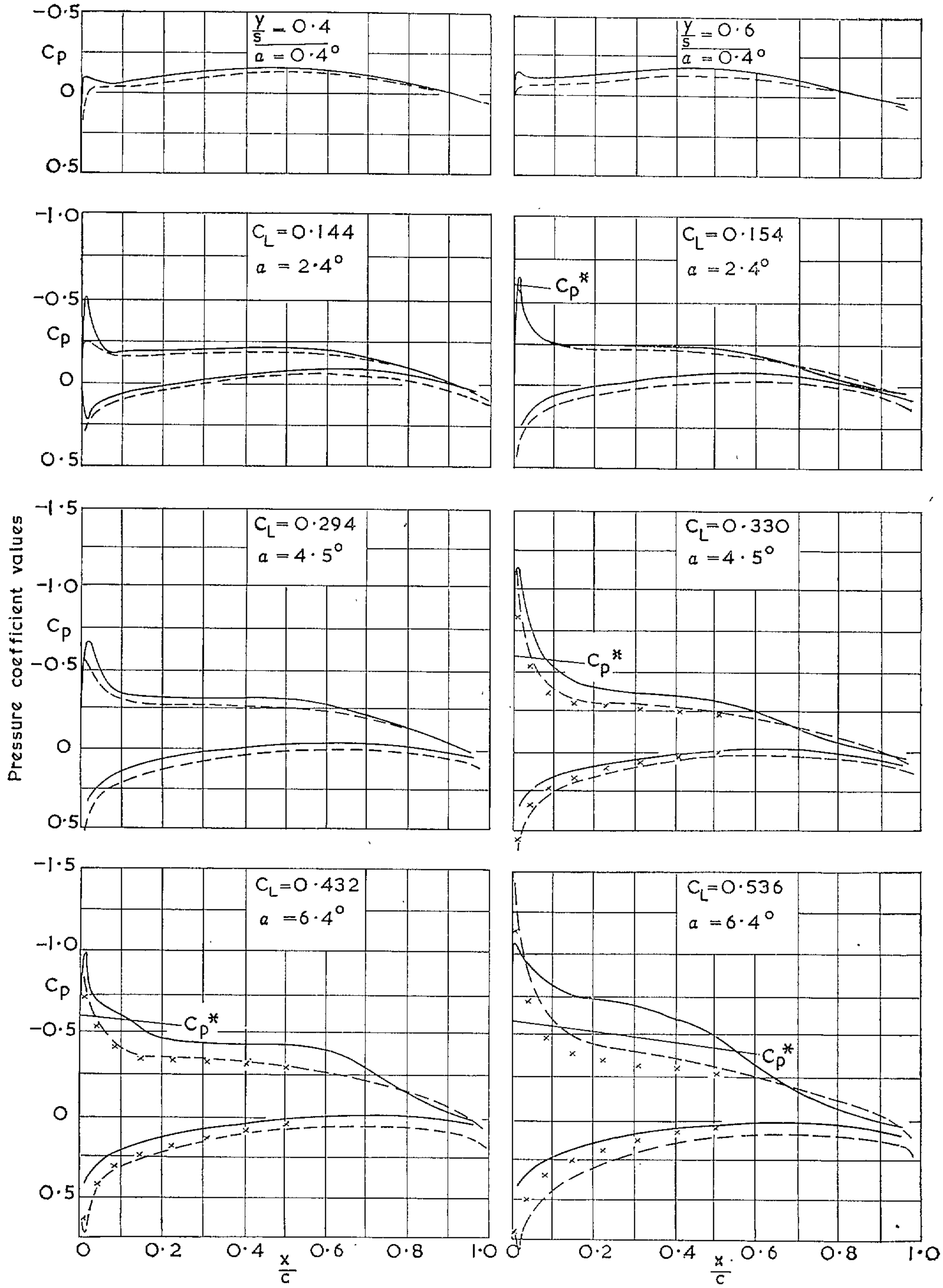


—— Experimental results
– – – – Calculated results assuming experimental lift coefficient
x x x Calculated results assuming theoretical lift coefficient

FIG. 9a. Chordwise pressure distributions. $M = 0{\cdot}9$.

N.A.C.A. WING



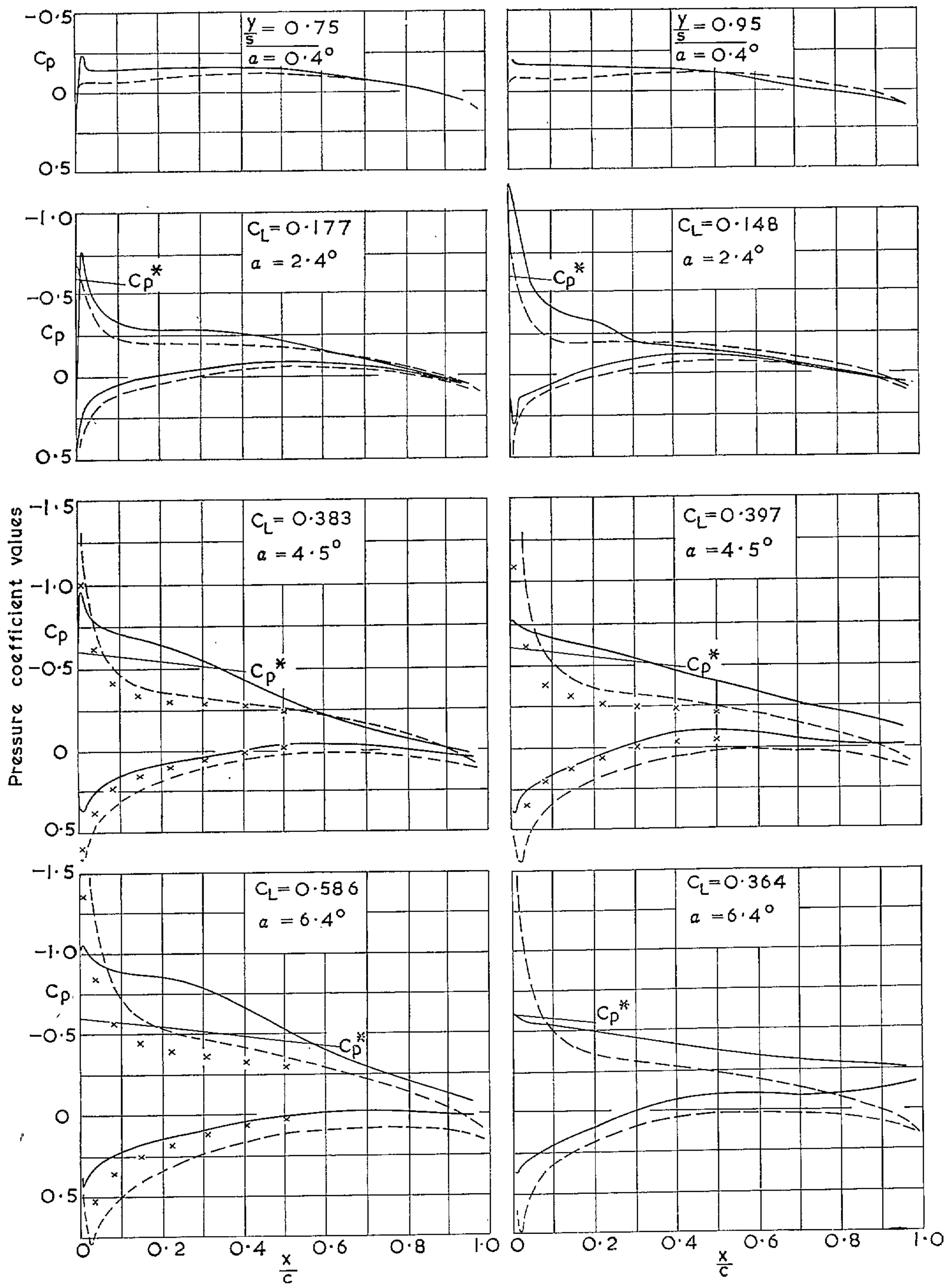
——— Experimental results

– – – – Calculated results assuming experimental lift coefficient

x x x Calculated results assuming theoretical lift coefficient

FIG. 9b. Chordwise pressure distributions. $M = 0 \cdot 9$.

N.A.C.A. WING



——— Experimental results

– – – – Calculated results assuming experimental lift coefficient

x x x Calculated results assuming theoretical lift coefficient

FIG. 9c. Chordwise pressure distributions. $M = 0{\cdot}9$.

N.A.C.A. WING

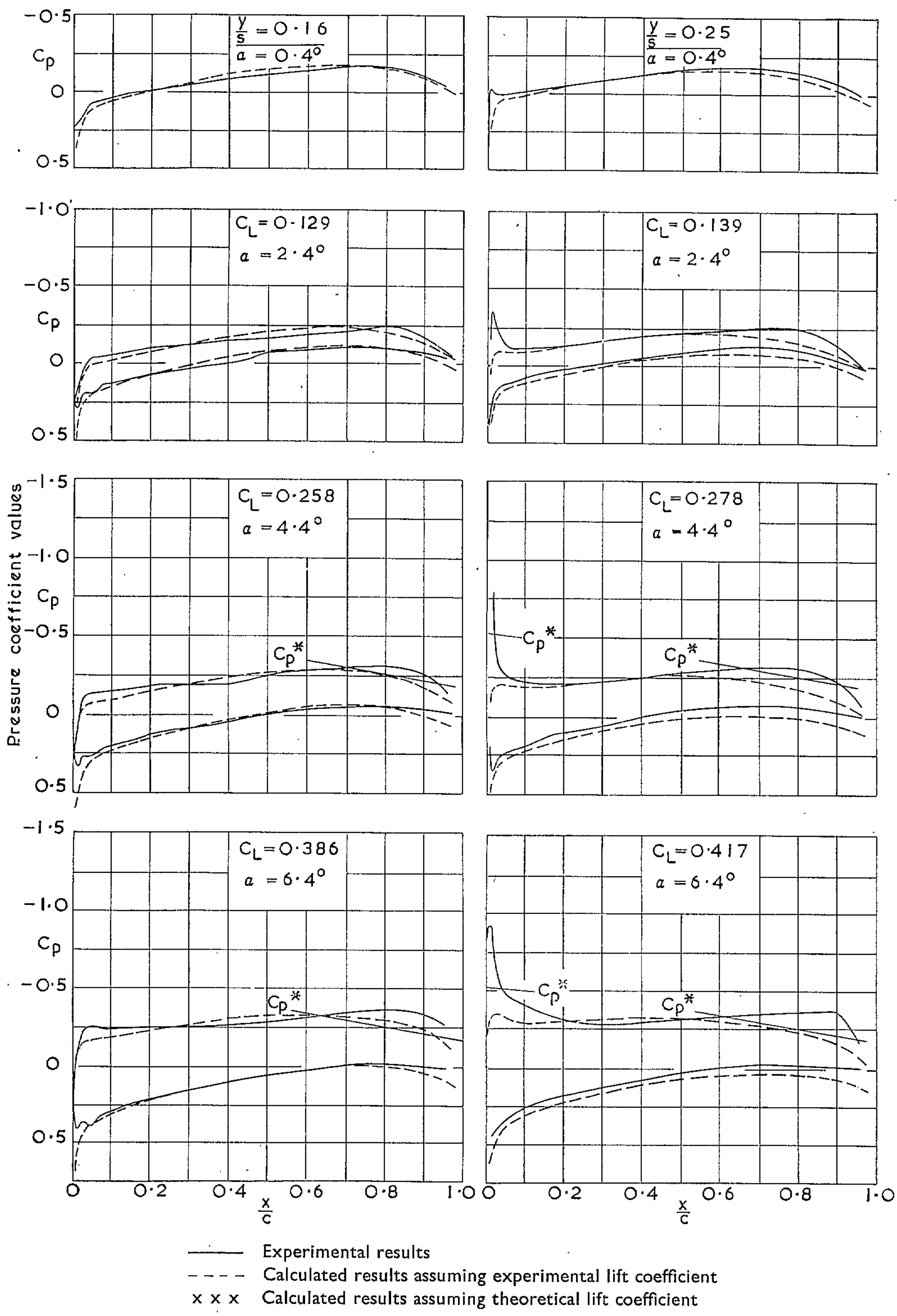
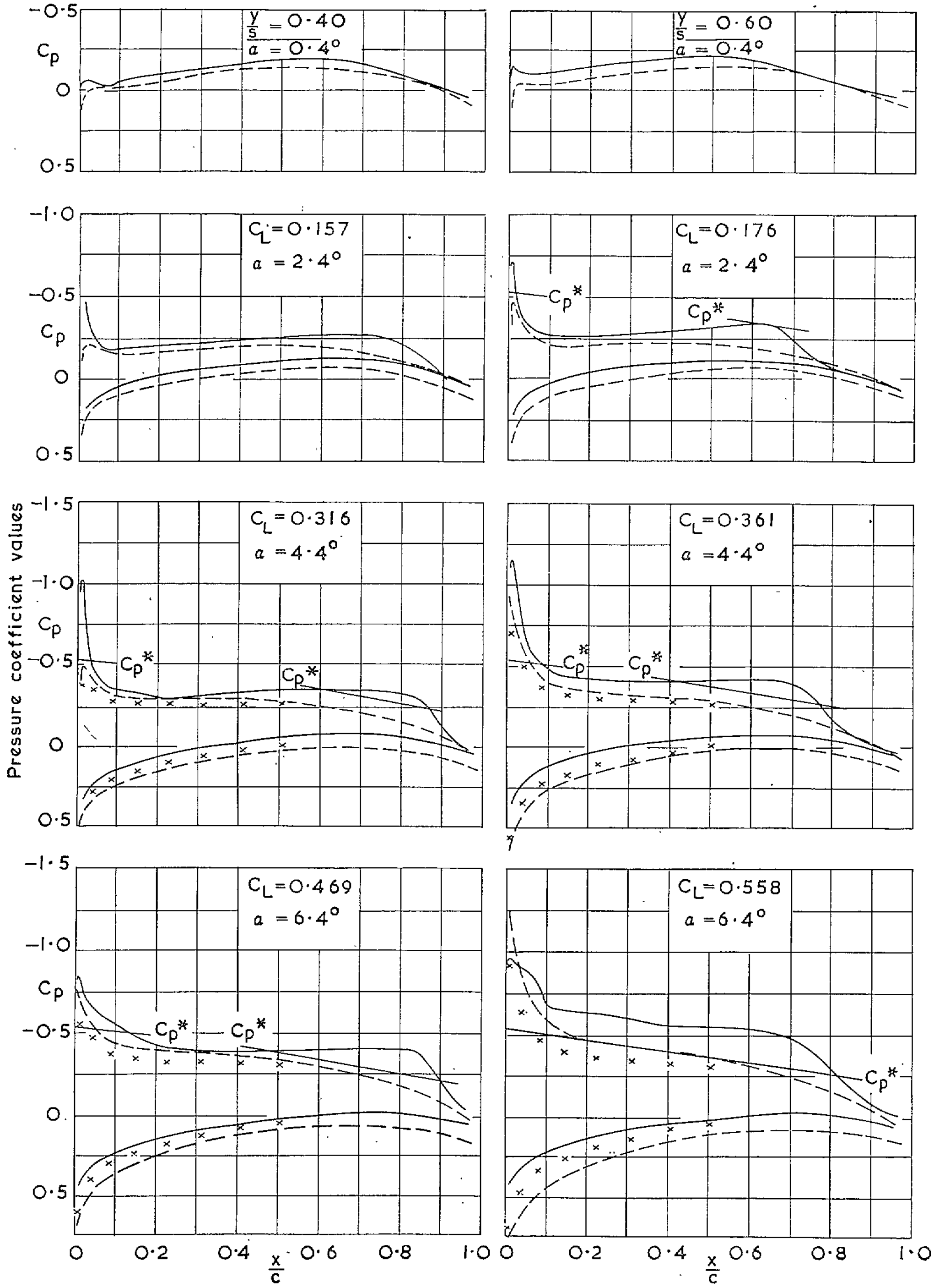


FIG. 10a. Chordwise pressure distributions. $M = 0 \cdot 94$.

N.A.C.A. WING



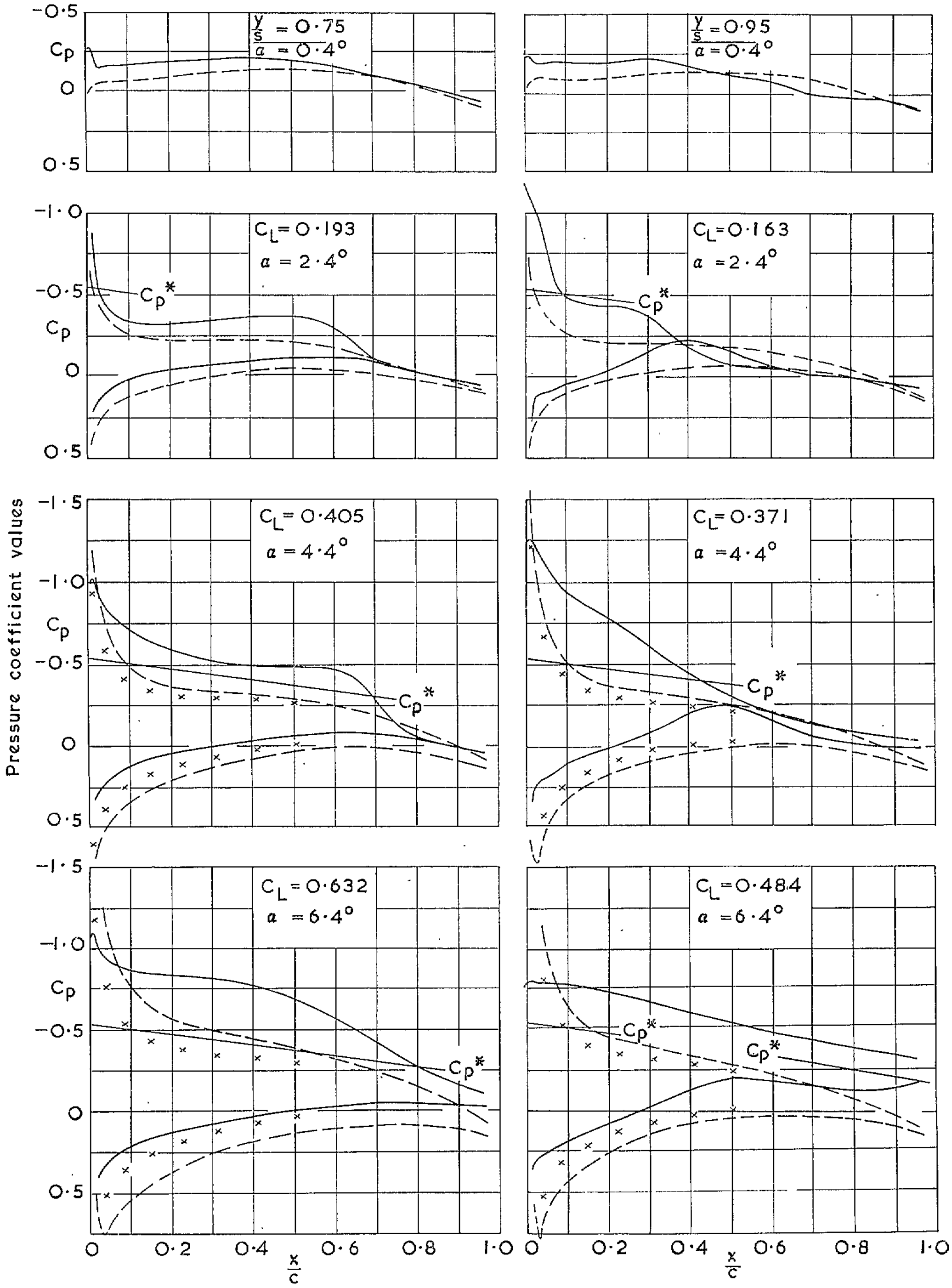
——— Experimental results

– – – – Calculated results assuming experimental lift coefficient

x x x Calculated results assuming theoretical lift coefficient

FIG. 10b. Chordwise pressure distributions. $M = 0 \cdot 94$.

N.A.C.A. WING



——— Experimental results

– – – – Calculated results assuming experimental lift coefficient

x x x Calculated results assuming theoretical lift coefficient

FIG. 10c. Chordwise pressure distributions. $M = 0{\cdot}94$.

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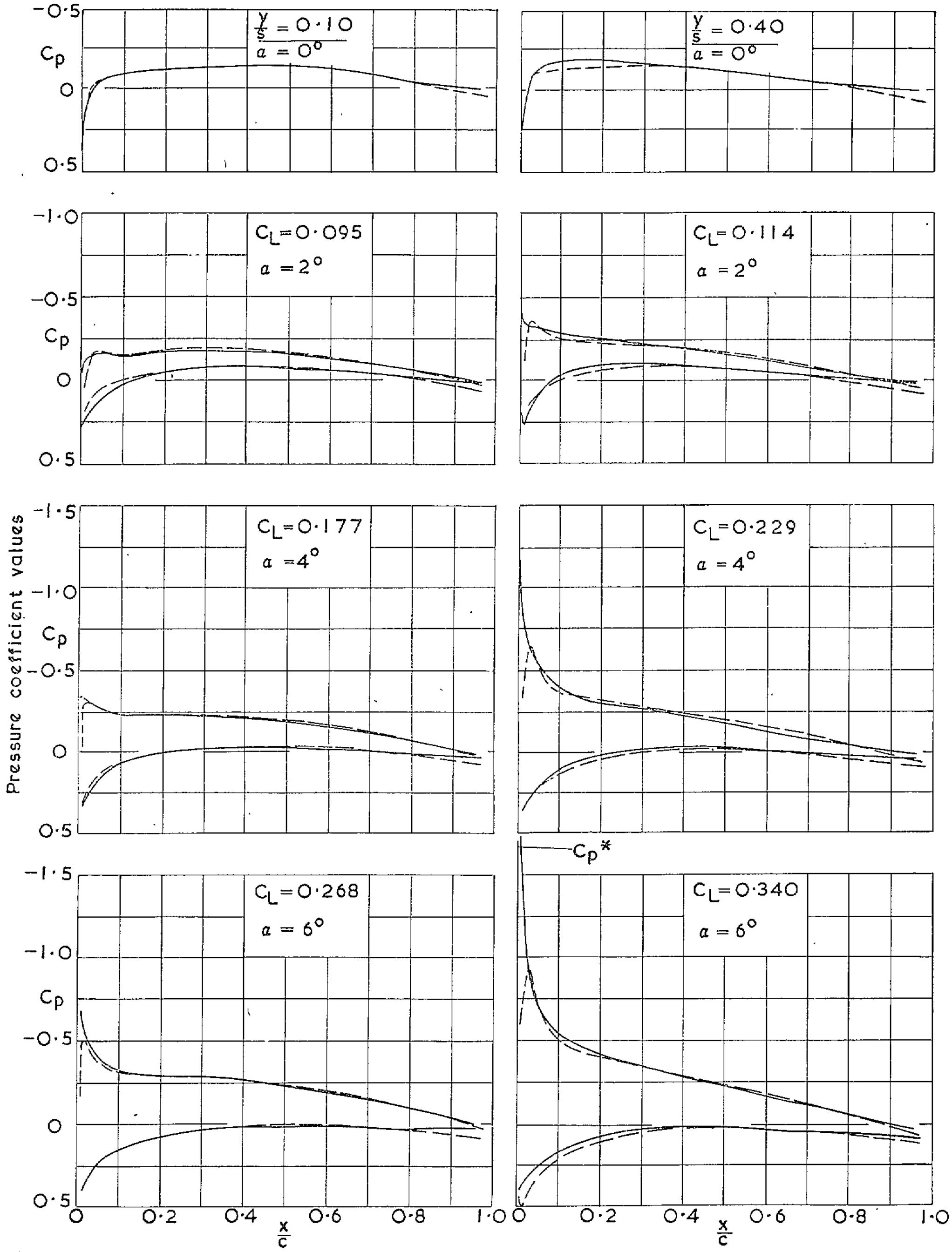
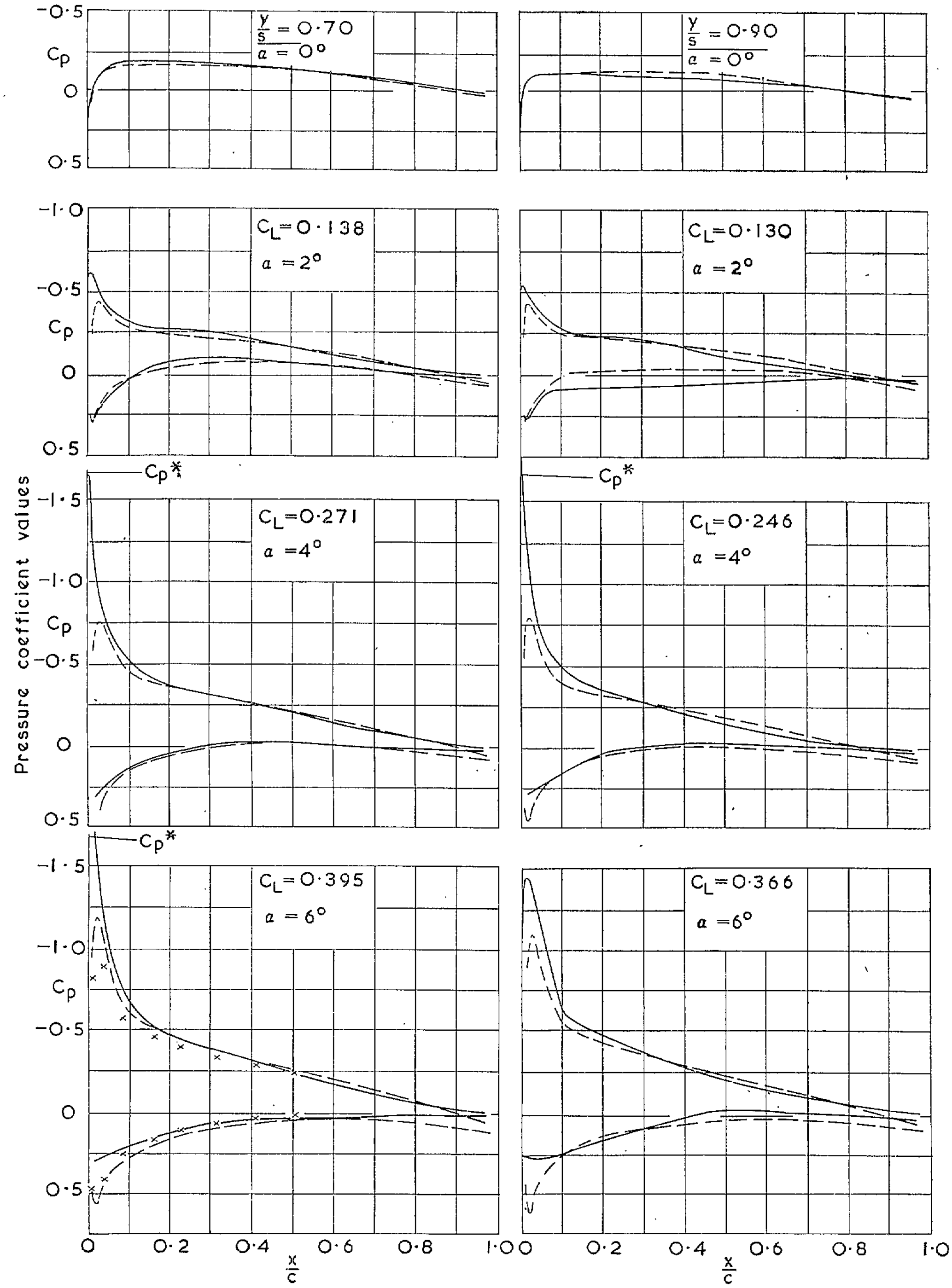


——— Experimental results
– – – – Calculated results assuming experimental lift coefficient
x x x Calculated results assuming theoretical lift coefficient

FIG. 11a. Chordwise pressure distributions. $M = 0 \cdot 6$.

WARREN WING 12



——— Experimental results
– – – – Calculated results assuming experimental lift coefficient
x x x Calculated results assuming theoretical lift coefficient

FIG. 11b. Chordwise pressure distributions. $M = 0 \cdot 6$.

WARREN WING 12

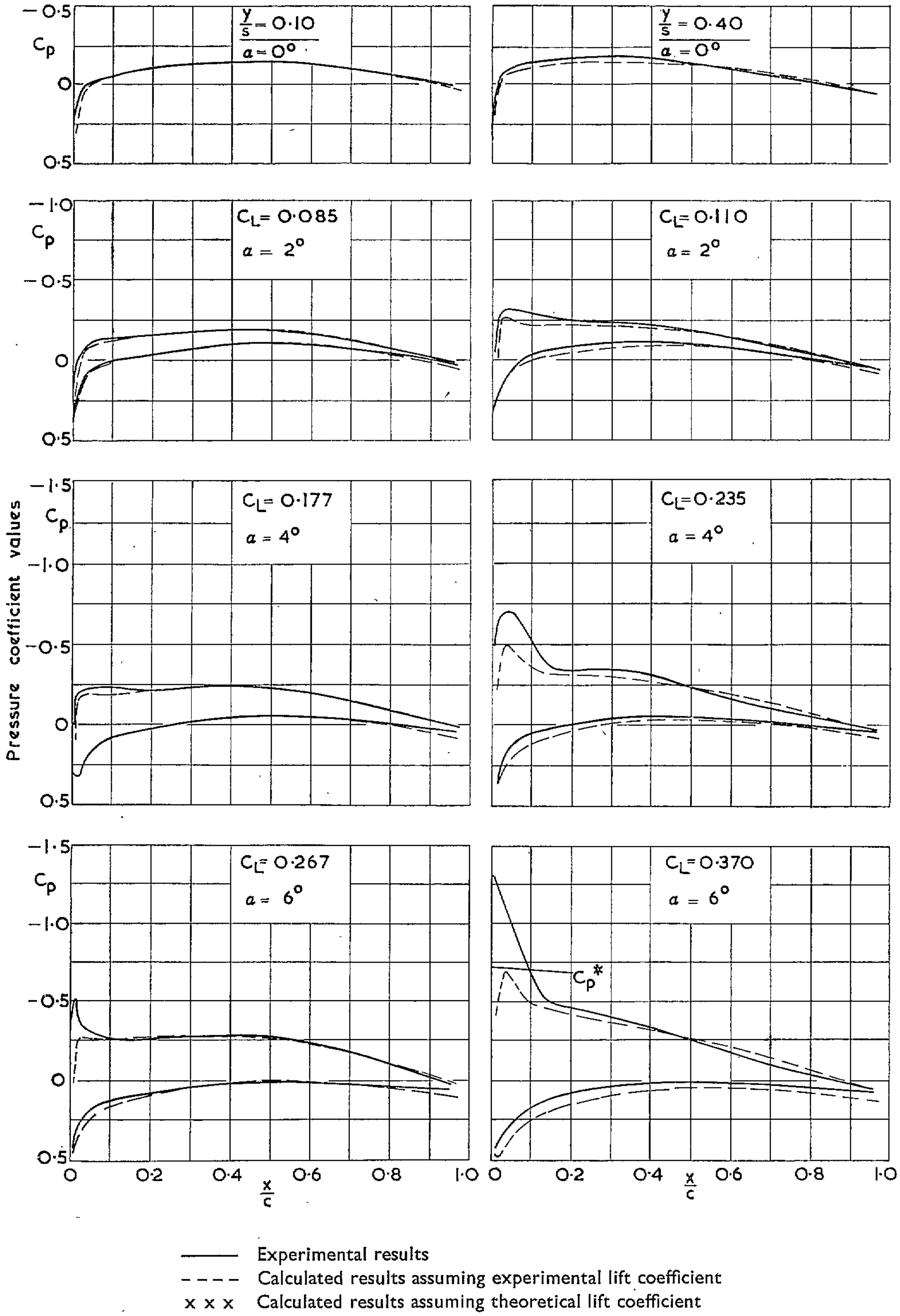
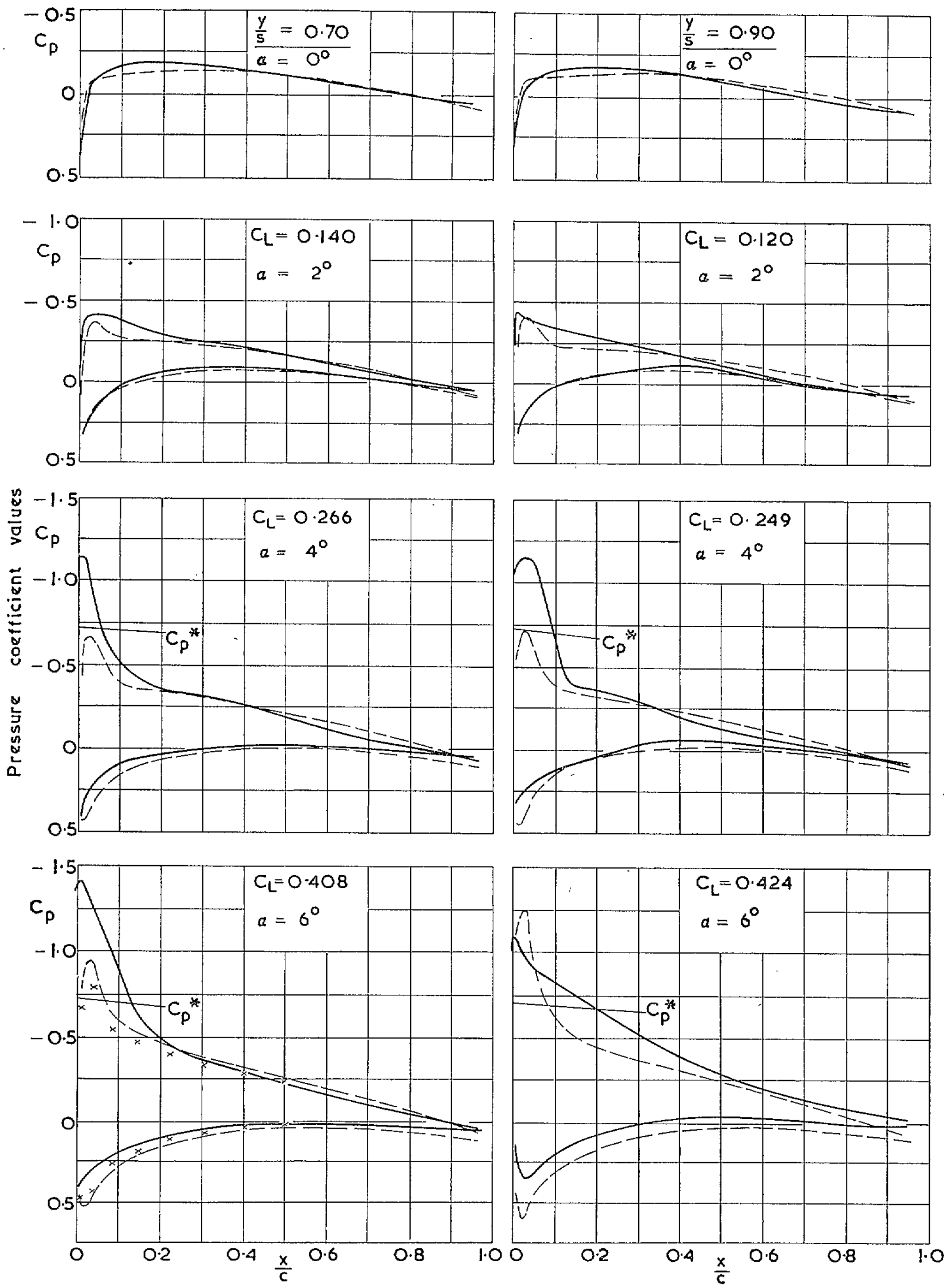


FIG. 12a. Chordwise pressure distributions. $M = 0{\cdot}85$.

WARREN WING 12



——— Experimental results
– – – – Calculated results assuming experimental lift coefficient
x x x Calculated results assuming theoretical lift coefficient

FIG. 12b. Chordwise pressure distributions. $M = 0 \cdot 85$.

WARREN WING 12

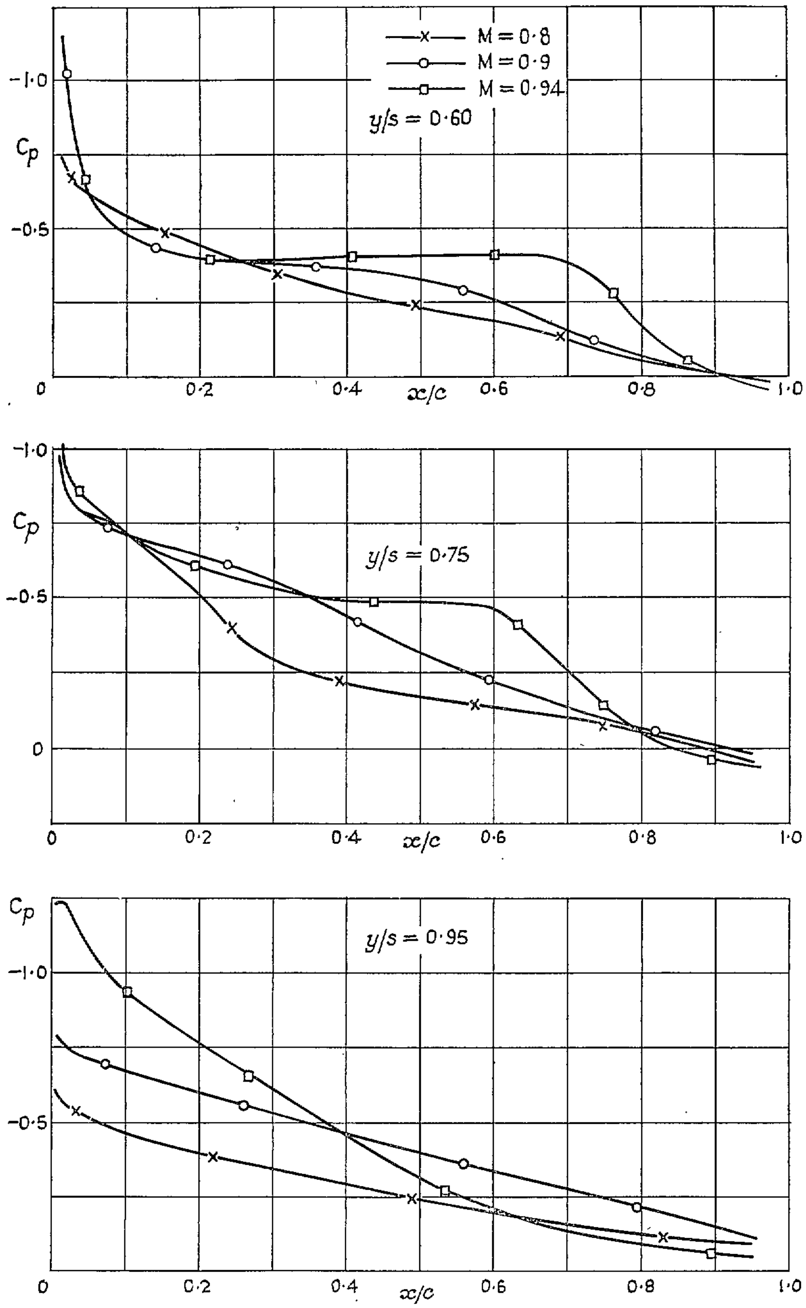


FIG. 13. Upper-surface pressure distributions. $\alpha = 4 \cdot 5$ deg.

N.A.C.A. WING

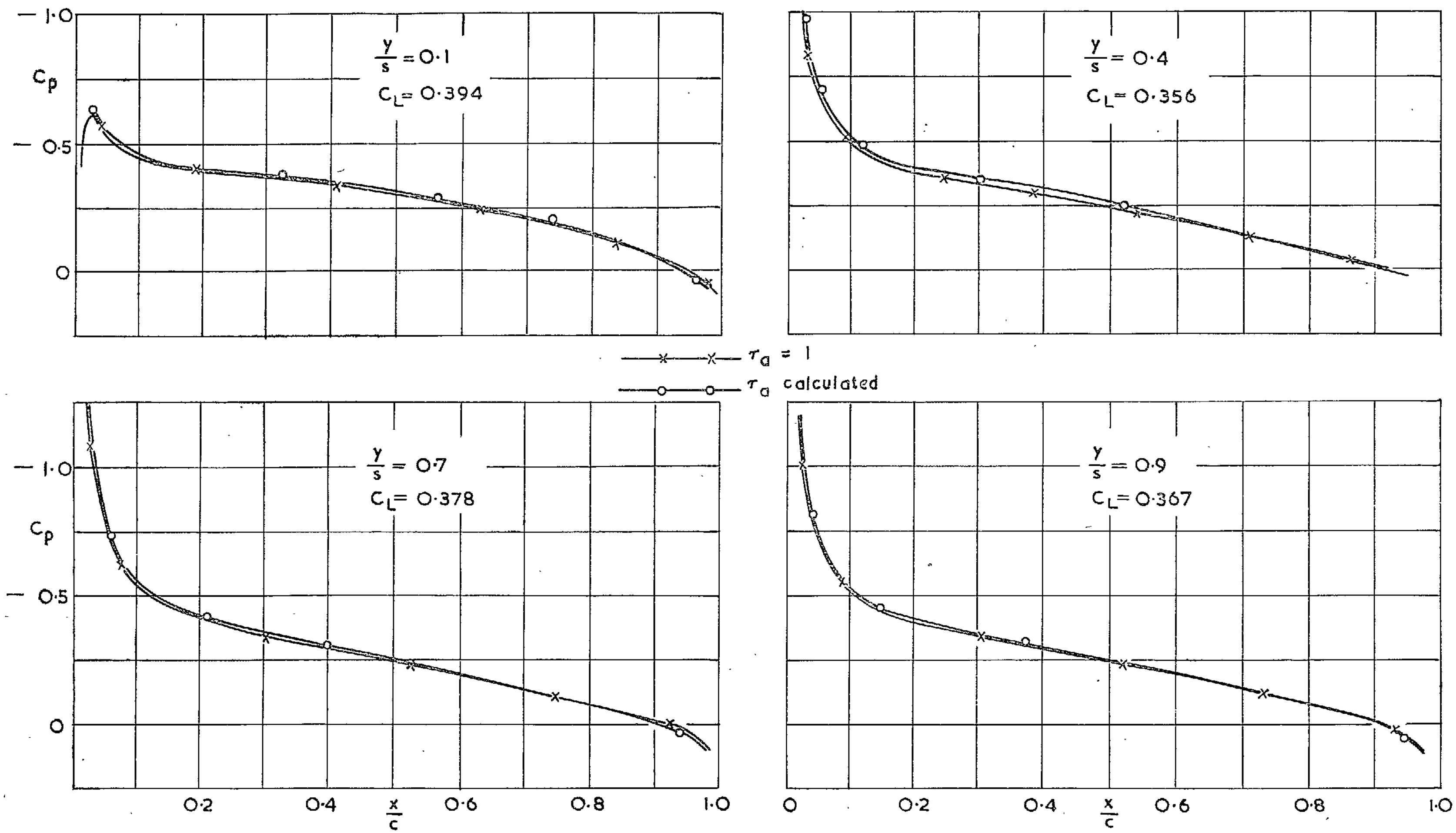


FIG. 14. Theoretical upper-surface $C_p \sim x/c$ (Warren Wing 12; $M = 0{\cdot}6$; $\alpha = 6$ deg).

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