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TECHNICAL NOTES

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

No. 391

TESTS OF N.A.C.A. AIRFOILS IN THE VARIABLE-DENSITY WIND TUNNEL. SERIES 43 AND 63

By Eastman N. Jacobs and Robert M. Pinkerton
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Summary

This note is one of a series covering an investigation of a family of related airfoils. It gives in preliminary form the results obtained from tests in the N.A.C.A. Variable-Density Wind Tunnel of two groups of six airfoils each. One group, the 43 series, has a maximum mean camber of 4 per cent of the chord at a position 0.3 of the chord from the leading edge; the other group, the 63 series, has a maximum mean camber of 6 per cent of the chord at the same position. The members within each group differ only in maximum thickness, the maximum thickness/chord ratios being: 0.06, 0.09, 0.12, 0.15, 0.18, and 0.21. The results are analyzed with a view to indicating the variation of the aerodynamic characteristics with profile thickness for airfoils having a certain mean camber line.

Introduction

An extensive study of the relation between the geometric and the aerodynamic properties of airfoils at a high value of the Reynolds Number is in progress in the Variable-Density Wind Tunnel of the National Advisory Committee for Aeronautics. Tests of a large number of related airfoils are being made at a Reynolds Number of approximately 3,000,000 with a view to establishing definitely the effect of systematic variations in profile shape upon the lift, drag, and pitching moment characteristics of airfoils. For the purpose of this investigation, as discussed in reference 1, airfoil profiles are considered as made up of certain profile thickness forms disposed about certain mean camber line forms. The various N.A.C.A. airfoils for this investigation were developed by changing systematically these two shape variables. Six maximum thickness/chord ratios were chosen: 0.06, 0.09, 0.12, 0.15, 0.18, and 0.21. The mean camber line form depends on two variables, the maximum mean camber and the distance from the leading edge to the position of the maximum mean camber. Three ratios of the maximum mean camber to the chord were chosen: 0.02, 0.04, and 0.06. These were combined with six positions of maximum mean camber: 0.2, 0.3, 0.4, 0.5, 0.6, and 0.7 of the chord from the leading edge. The airfoils so produced are designated by a number of four digits; the first indicates the maximum mean camber; the second, the position of the maximum mean camber; and the last two, the maximum thickness. Thus the N.A.C.A. 6321 airfoil has a maximum mean camber of 6 per cent of the chord at a position 0.3 of the chord from the leading edge, and a maximum thickness of 21 per cent of the chord; the N.A.C.A. 0012 is a symmetrical airfoil having a maximum thickness of 12 per cent of the chord.

The results of tests of the six symmetrical N.A.C.A. airfoils have been published in preliminary form in reference 1. Similar publications presenting data on the other airfoils will follow as the tests are made.

This note presents the results of tests of two series of six airfoils each, the airfoils of each series having the same thickness forms as those of the summetrical series (reference 1), but having curved instead of straight mean camber lines. All twelve airfoils have mean camber lines of such form that the position of the maximum mean camber is 0.3 of the chord behind the leading edge. Six of the airfoils, the 43 series, have a maximum mean camber of 4 per cent of the chord, and the other six, the 63 series, have a maximum mean camber of 6 per cent of the chord.

Description of Airfoils

The method of arriving at the thickness forms used to develop the N.A.C.A. airfoils is described in reference 1. The thickness ordinates are defined by the equation

$$y_t = \frac{t}{0.20}\,(0.296900\sqrt{x} - 0.126000x - 0.351600x^2 + 0.284300x^3 - 0.101500x^4)$$

where t is the ratio of the maximum thickness to the chord. Each mean camber line is defined by two parabolic equations of the form

$$y_c = a + bx + cx^2$$

where the leading end of the mean camber line is at the origin and the trailing end is on the x-axis at $x = 1$. The constants in the above equation are determined by the following conditions,

1. $x = 0$ or $x = 1$, $y_c = 0$
2. x = position of maximum camber, $dy_c/dx = 0$
3. x = position of maximum camber, y_c = maximum camber.

The method of combining the thickness forms with the mean camber line forms is best described by means of the diagram in Figure 1. The line joining the extremities of the mean camber line is chosen as the chord. Referring to the diagram, the ordinate y_t of the thickness form is measured along the perpendicular to the mean camber line from a point on the mean camber line at the corresponding station along the chord. The resulting upper and lower surface points are then designated:

ordinates y_u and y_l

stations x_u and x_l

where the subscripts u and l refer to upper and lower surfaces, respectively. In addition to these symbols, the symbol θ is employed to designate the angle between the tangent to the mean camber line and the x-axis. This angle is given by

$$\theta = \tan^{-1} \frac{dy_c}{dx}$$

The following formulas for calculating the ordinates may now be derived from the diagram

$$y_u = y_c + y_t \cos\theta$$

$$x_u = x - y_t \sin\theta$$

$$y_l = y_c - y_t \cos\theta$$

$$x_l = x + y_t \sin\ \theta$$

Sample calculations are given in Figure 1.

The ordinates of the N.A.C.A. airfoils with which this report deals were obtained in the manner described. The mean camber lines for these sections are

	From $x = 0$ to $x = 0.3$	From $x = 0.3$ to $x = 1$
43 series	$y_c = \frac{1}{9}(2.4x - 4x^2)$	$y_c = \frac{1}{49}(1.6 + 2.4x - 4x^2)$
63 series	$y_c = \frac{1}{9}(3.6x - 6x^2)$	$y_c = \frac{1}{49}(2.4 + 3.6x - 6x^2)$

The ordinates for the airfoils are given in Tables I to XII and profile shapes are shown in Figure 2.

The models, which were constructed of duralumin, have a chord of 5 inches and a span of 30 inches. The method of construction is described in reference 1. The N.A.C.A. 4312 airfoil, however, was constructed before the construction procedure was standardized. The fact that this airfoil was not so carefully made as the others may account for the fact that the plotted results from the tests of this airfoil do not fair in with the other results.

Tests and Results

Routine measurements of lift, drag, and pitching moment about a point one-quarter of the chord behind the leading edge were made at a Reynolds Number of approximately 3,000,000. A description of the tunnel and method of testing is given in reference 1.

The results are presented in the form of coefficients corrected, after the method of reference 2, to give infinite aspect ratio characteristics. Tables XIII to XXIV present the corrected results: lift coefficient C_L, angle of attack for infinite aspect ratio α_o, profile drag coefficient C_{D_o}, and pitching moment coefficient about a point one-quarter of the chord behind the leading edge $C_{m_{c/4}}$. These data are also presented in several figures to facilitate the discussion.

Discussion

Variation of the Aerodynamic characteristics with thickness.- The variation of minimum profile drag coefficient with maximum thickness is shown in Figure 5. This relation may be expressed by the equation

$$C_{D_o\ min.} = 0.0065 + 0.0083t + 0.0972t^2 + k$$

where t is the ratio of the maximum thickness to the chord and k is a constant for the airfoils having the same mean camber line. The first three terms of the above expression give the minimum profile drag coefficient for the six symmetrical N.A.C.A. airfoils. The values of k are given below:

43 series $k = 0.0009$

63 series $k = 0.0020$.

The calculated curves and the test points taken from the faired profile drag curves (figs. 3 and 4) are shown in Figure 5.

Maximum lift coefficients as taken from Figures 6 and 7 are given in the following table:

Airfoil	C_L max.	Airfoil	C_L max.
4306	1.20	6306	1.54
4309	1.60	6309	1.66
4312	1.63	6312	1.64
4315	1.56	6315	1.55
4318	1.46	6318	1.43
4321	1.29	6321	1.37

These results are in agreement with those for the symmetrical airfoils in that the moderately thick airfoils give the highest maximum lift coefficients.

The variation of the slope of the lift curve with thickness is shown in Figure 8. The points on the figure represent the deduced slopes as measured in the angular range of low profile drag for an infinite span wing. It will be noted that all of the values lie below the approximate theoretical value for thin wings, 2π per radian. These results show substantially the same variation as do those from the symmetrical airfoils; that is, the slope decreases with increased thickness.

The pitching moment coefficients at the angles of attacks corresponding to zero lift are given in the following table:

Airfoil	C_{m_o}	Airfoil	C_{m_o}
4306	-0.075	6306	--
4309	- .075	6309	-0.111
4312	- .072	6312	- .109
4315	- .068	6315	- .104
4318	- .065	6318	- .097
4321	- .057	6321	- .091

The calculation of the moment coefficient has commonly been based on the assumption that an airfoil may be replaced by its mean camber line. This assumption, however, would lead to the same moment coefficient for all sections in either one of the above groups, since they have the same mean camber line. It is apparent from the above table that such an assumption leads to erroneous results; actually the magnitude of the diving moment coefficient decreases with increasing thickness.

The ratio of the maximum lift to the minimum profile drag has previously been taken as a measure of the general efficiency of an airfoil section. The variation of this ratio with thickness is shown in Figure 10. The N.A.C.A. 4309 shows the highest value of this ratio.

Variation of the characteristics with lift or angle.- The variation of profile drag coefficient with lift coefficient is shown by Figures 3 and 4. In accordance with the procedure given in reference 1, the variation of the additional drag coefficient due to lift has been studied by plotting values of $C_{D_o} - C_{D_o\ min.}$ against the square of the lift coefficient as measured from the lift coefficient corresponding to the minimum profile drag coefficient. These plots are given in Figures 11 and 12. It may be significant to note that the same line determined for the symmetrical airfoils fits the present cases to a fair degree of accuracy. It is now possible to write the profile drag coefficient as

$$C_{D_c} = C_{D_{o\,min.}} + 0.0062(C_L - C_{L\,opt.})^2$$

where $C_{L\,opt.}$ may be called the optimum lift coefficient; that is, the lift coefficient corresponding to the minimum profile drag coefficient. The optimum lift coefficient varies with thickness as well as with camber, the value increasing with camber but decreasing with thickness. $C_{L\,opt.}$ varies from 0.40 for the 4306 to 0 for the 4321 and from 0.70 for the 6306 to 0.10 for the 6321. These variations may be expressed by the following formulas:

43 series $\quad C_{L\,opt.} = 0.56 - \frac{8}{3}\,t$

63 series $\quad C_{L\,opt.} = 0.94 - 4\,t$

The variation of the pitching-moment coefficient with angle or lift may be best studied with reference to thin airfoil theory, which predicts a constant pitching moment about a point one-quarter of the chord behind the leading edge. The theory indicates that the moment about this point is constant because the center of pressure of that part of the air force which is due to angular changes is at the quarter chord point. However, the curves of $C_{m_{c/4}}$ against angle of attack (fig. 9) show a slight slope in the normal working range, as did the corresponding curves for the symmetrical airfoils. (Reference 1.) The point of constant moment is, therefore, not exactly at the quarter chord point, but displaced forward from it as indicated in the following table:

Airfoil	Displacement (Per cent chord)	Airfoil	Displacement (Per cent chord)
4306	0.3	6306	0.1
4309	.5	6309	.1
4312	.2	6312	.4
4315	.5	6315	.6

4318	.9	6318	1.0
4321	1.6	6321	1.6

In reference 1, the center of pressure for symmetrical airfoils is shown to be farther forward for the thick airfoils. The preceding results may, therefore, be considered as indicating a displacement of the center of pressure for that part of the air forces due to angular changes.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., August 26, 1931.

References

1. Jacobs, Eastman N.: Tests of Six Symmetrical Airfoils in the Variable Density Wind Tunnel. N.A.C.A. Technical Note No. 385, July, 1931.

2. Jacobs, Eastman N., and Anderson, Raymond F.: Large-Scale Aerodynamic Characteristics of Airfoils as Tested in the Variable Density Wind Tunnel. N.A.C.A. Technical Report No. 352, 1930.

TABLE I

Ordinates for Airfoil N.A.C.A. 4306

(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
1.016	1.244	1.484	-0.592
2.190	1.908	2.810	- .632
4.614	2.958	5.386	- .514
7.088	3.809	7.912	- .309
9.590	4.528	10.410	- .084
14.647	5.650	15.353	+ .350
19.746	6.414	20.254	+ .698
30.000	7.001	30.000	+ .999
40.047	6.819	39.953	+1.017
50.086	6.321	49.914	+1.027
60.112	5.545	59.888	+ .987
70.119	4.522	69.881	+ .866
80.107	3.266	79.893	+ .652
90.071	1.782	89.929	+ .340
95.043	.955	94.957	+ .147
100.007	.063	99.993	- .063
L.E. radius	.394		
Slope of radius passing through end of chord	4/15		

TABLE II

Ordinates for Airfoil N.A.C.A. 4309
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.898	1.703	1.602	-1.051
2.035	2.542	2.965	-1.266
4.421	3.826	5.579	-1.382
6.882	4.839	8.118	-1.339
9.385	5.682	10.615	-1.238
14.470	6.976	15.530	- .976
19.619	7.844	20.381	- .732
30.000	8.502	30.000	- .502
40.071	8.268	39.929	- .432
50.130	7.643	49.870	- .295
60.167	6.685	59.833	- .153
70.179	5.436	69.821	- .048
80.160	3.918	79.840	.000
90.106	2.143	89.894	- .021
95.064	1.153	94.936	- .051
100.011	.094	99.989	- .094
L.E. radius	.887		
Slope of radius passing through end of chord	4/15		

TABLE III

Ordinates for Airfoil N.A.C.A. 4312
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.781	2.162	1.719	-1.510
1.879	3.177	3.121	-1.901
4.228	4.692	5.772	-2.251
6.676	5.868	8.324	-2.368
9.180	6.833	10.820	-2.389
14.294	8.298	15.706	-2.298
19.492	9.272	20.508	-2.160
30.000	10.002	30.000	-2.002
40.095	9.720	39.905	-1.884
50.173	8.965	49.827	-1.617
60.223	7.824	59.777	-1.292
70.239	6.351	69.761	- .963
80.213	4.571	79.787	- .653
90.141	2.503	89.859	- .381
95.085	1.353	94.915	- .251
100.014	.125	99.986	- .125
L.E. radius	1.576		
Slope of radius passing through end of chord	4/15		

TABLE IV

Ordinates for Airfoil N.A.C.A. 4315
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.664	2.620	1.836	-1.968
1.724	3.813	3.276	-2.537
4.036	5.562	5.964	-3.118
6.470	6.898	8.530	-3.398
8.975	7.988	11.025	-3.544
14.117	9.624	15.883	-3.624
19.365	10.700	20.635	-3.588
30.000	11.503	30.000	-3.503
40.118	11.171	39.882	-3.335
50.216	10.289	49.784	-2.941
60.279	8.963	59.721	-2.431
70.298	7.264	69.702	-1.876
80.267	5.225	79.733	-1.307
90.177	2.863	89.823	- .741
95.107	1.556	94.893	- .454
100.018	.157	99.982	- .157
L.E. radius	2.464		
Slope of radius passing through end of chord	4/15		

TABLE V

Ordinates for Airfoil N.A.C.A. 4318
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.546	3.080	1.954	-2.428
1.569	4.446	3.431	-3.170
3.843	6.430	6.157	-3.986
6.264	7.928	8.736	-4.428
8.771	9.138	11.229	-4.694
13.940	10.951	16.060	-4.951
19.238	12.128	20.762	-5.016
30.000	13.003	30.000	-5.003
40.142	12.619	39.858	-4.783
50.259	11.612	49.741	-4.264
60.335	10.104	59.665	-3.572
70.358	8.177	69.642	-2.789
80.320	5.881	79.680	-1.963
90.212	3.224	89.788	-1.102
95.128	1.753	94.872	- .651
100.021	.188	99.979	- .188
L.E. radius	3.549		
Slope of radius passing through end of chord	4/15		

TABLE VI

Ordinates for Airfoil N.A.C.A. 4321
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.429	3.539	2.071	-2.887
1.414	5.081	3.586	-3.805
3.651	7.294	6.349	-4.850
6.059	8.957	8.941	-5.457
8.566	10.290	11.434	-5.846
13.764	12.272	16.236	-6.272
19.111	13.556	20.889	-6.444
30.000	14.503	30.000	-6.503
40.166	14.072	39.834	-6.236
50.302	12.934	49.698	-5.586
60.391	11.241	59.609	-4.709
70.418	9.090	69.582	-3.702
80.374	6.535	79.626	-2.617
90.247	3.585	89.753	-1.463
95.149	1.953	94.851	- .851
100.025	.220	99.975	- .220
L.E. radius	4.830		
Slope of radius passing through end of chord	4/15		

TABLE VII

Ordinates for Airfoil N.A.C.A. 6306
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.911	1.373	1.589	-0.395
2.050	2.185	2.950	- .269
4.438	3.521	5.562	+ .147
6.897	4.636	8.103	+ .614
9.397	5.596	10.603	+1.070
14.476	7.121	15.524	+1.879
19.621	8.177	20.379	+2.489
30.000	9.001	30.000	+2.999
40.071	8.778	39.929	+2.978
50.130	8.157	49.870	+2.867
60.167	7.174	59.833	+2.622
70.179	5.863	69.821	+2.217
80.159	4.240	79.841	+1.638
90.105	2.308	89.895	+ .876
95.064	1.227	94.936	+ .425
100.011	.062	99.989	- .062
L.E. radius	.394		
Slope of radius passing through end of chord	6/15		

TABLE VIII

Ordinates for Airfoil N.A.C.A. 6309
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.741	1.816	1.759	-0.838
1.825	2.798	3.175	- .882
4.157	4.365	5.843	- .697
6.595	5.642	8.405	- .392
9.095	6.728	10.905	- .062
14.213	8.433	15.787	+ .567
19.431	9.600	20.569	+1.066
30.000	10.503	30.000	+1.497
40.106	10.228	39.894	+1.528
50.194	9.478	49.806	+1.546
60.251	8.312	59.749	+1.484
70.268	6.775	69.732	+1.305
80.239	4.890	79.761	+ .988
90.158	2.667	89.842	+ .517
95.095	1.423	94.905	+ .229
100.016	.094	99.984	- .094
L.E. radius	.887		
Slope of radius passing through end of chord	6/15		

TABLE IX

Ordinates for Airfoil N.A.C.A. 6312
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.572	2.258	1.928	-1.280
1.600	3.412	3.400	-1.496
3.875	5.209	6.125	-1.541
6.293	6.648	8.707	-1.398
8.794	7.858	11.206	-1.192
13.951	9.741	16.049	- .741
19.242	11.021	20.758	- .355
30.000	12.002	30.000	- .002
40.142	11.682	39.858	+ .074
50.259	10.800	49.741	+ .224
60.334	9.449	59.666	+ .347
70.357	7.688	69.643	+ .392
80.319	5.541	79.681	+ .337
90.211	3.026	89.789	+ .158
95.127	1.622	94.873	+ .030
100.021	.124	99.979	- .124
L.E. radius	1.576		
Slope of radius passing through end of chord	6/15		

TABLE X

Ordinates for Airfoil N.A.C.A. 6315
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.402	2.700	2.098	-1.722
1.375	4.026	3.625	-2.110
3.594	6.052	6.406	-2.384
5.991	7.654	9.009	-2.404
8.491	8.991	11.509	-2.325
13.689	11.053	16.311	-2.053
19.053	12.442	20.947	-1.776
30.000	13.503	30.000	-1.503
40.177	13.130	39.823	-1.374
50.324	12.123	49.676	-1.099
60.418	10.587	59.582	- .791
70.447	8.598	69.553	- .518
80.398	6.192	79.602	- .314
90.263	3.384	89.737	- .200
95.159	1.824	94.841	- .172
100.027	.156	99.973	- .156
L.E. radius	2.464		
Slope of radius passing through end of chord	6/15		

TABLE XI

Ordinates for Airfoil N.A.C.A. 6318
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.233	3.143	2.267	-2.165
1.151	4.638	3.849	-2.722
3.313	6.895	6.687	-3.227
5.690	8.659	9.310	-3.409
8.191	10.120	11.809	-3.454
13.426	12.365	16.574	-3.365
18.863	13.864	21.137	-3.198
30.000	15.003	30.000	-3.003
40.213	14.577	39.787	-2.821
50.389	13.445	49.611	-2.421
60.502	11.726	59.498	-1.930
70.536	9.509	69.464	-1.429
80.478	6.845	79.522	- .967
90.316	3.742	89.684	- .558
95.190	2.020	94.810	- .368
100.032	.186	99.968	- .186
L.E. radius	3.549		
Slope of radius passing through end of chord	6/15		

TABLE XII

Ordinates for Airfoil N.A.C.A. 6321
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.063	3.585	2.437	-2.607
0.925	5.252	4.075	-3.336
3.033	7.735	6.967	-4.067
5.388	9.665	9.612	-4.415
7.889	11.251	12.111	-4.585
13.165	13.672	16.835	-4.672
18.674	15.284	21.326	-4.618
30.000	16.504	30.000	-4.504
40.248	16.030	39.752	-4.274
50.453	14.766	49.547	-3.742
60.585	12.862	59.415	-3.066
70.625	10.419	69.375	-2.339
80.558	7.496	79.442	-1.618
90.369	4.101	89.631	- .917
95.221	2.218	94.778	- .566
100.037	.218	99.963	- .218
L.E. radius	4.830		
Slope of radius passing through end of chord	6/15		

TABLE XIII

Airfoil: N.A.C.A. 4306

Average Reynolds Number: 3,080,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.3.

Test No.:561 Variable Density Tunnel. Date:April 11, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.514	-10.1	0.1399	-0.019
- .298	- 6.8	.0714	- .051
.009	- 3.7	.0100	- .075
.317	- .7	.0080	- .073
.472	.8	.0082	- .071
.627	2.3	.0087	- .071
.933	5.3	.0111	- .073
1.191	8.5	.0263	- .071
1.198	10.5	.0730	- .073
1.172	12.6	.1471	- .096
1.128	16.7	.2908	- .151
1.032	21.0	.3918	- .178
.963	27.2	.5194	- .188

TABLE XIV

Airfoil: N.A.C.A. 4309

Average Reynolds Number: 3,080,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.6.

Test No.: 563 Variable Density Tunnel. Date: April 13, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.342	-6.9	0.0117	-0.078
- .036	-3.9	.0096	- .075
.120	-2.4	.0090	- .076
.277	- .9	.0089	- .067
.429	.6	.0090	- .071
.581	2.2	.0095	- .065
.887	5.2	.0111	- .069
1.176	8.3	.0155	- .074
1.444	11.4	.0238	- .071
1.558	13.0	.0308	- .064
1.603	14.9	.0622	- .085
1.559	15.5	.0904	- .094
1.505	17.2	.1590	- .115
1.393	19.6	.2594	- .151
1.051	26.7	.4767	- .173

TABLE XV

Airfoil: N.A.C.A. 4312

Average Reynolds Number: 3,110,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.7.

Test No.: 564 Variable Density Tunnel. Date:April 13, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.178	-5.4	0.0110	-0.074
- .023	-3.9	.0105	- .070
.127	-2.4	.0101	- .070
.283	- .9	.0101	- .069
.440	.6	.0104	- .065
.591	2.1	.0110	- .071
.894	5.2	.0129	- .071
1.181	8.2	.0174	- .069
1.318	9.8	.0196	- .069
1.446	11.4	.0246	- .070
1.561	13.0	.0324	- .071
1.626	14.8	.0551	- .078
1.541	17.1	.1331	- .105
1.486	19.3	.1927	- .121
1.154	26.3	.4034	- .168

TABLE XVI

Airfoil: N.A.C.A. 4315

Average Reynolds Number: 3,130,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmosphere: 20.8.

Test No.: 565 Variable Density Tunnel. Date: April 14, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.335	-6.9	0.0123	-0.072
- .034	-3.9	.0109	- .068
.124	-2.4	.0107	- .066
.279	- .9	.0109	- .066
.432	.6	.0113	- .065
.585	2.1	.0117	- .064
.876	5.2	.0135	- .062
1.161	8.3	.0174	- .061
1.413	11.5	.0266	- .062
1.527	13.1	.0345	- .063
1.558	14.8	.0650	- .073
1.531	15.1	.0743	- .078
1.492	17.3	.1314	- .093
1.461	19.4	.1822	- .107
1.228	26.1	.2650	- .151

TABLE XVII

Airfoil: N.A.C.A. 4318

Average Reynolds Number: 3,090,000,

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.8.

Test No.: 566 Variable Density Tunnel. Date: April 14, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.190	-5.4	0.0136	-0.068
- .039	-3.9	.0122	- .065
.107	-2.3	.0120	- .064
.260	- .8	.0119	- .061
.412	.7	.0123	- .059
.557	2.2	.0132	- .058
.849	5.3	.0156	- .054
1.119	8.4	.0207	- .054
1.362	11.7	.0320	- .055
1.455	13.4	.0472	- .059
1.425	15.5	.0978	- .075
1.404	19.5	.1796	- .095
1.217	26.1	.3337	- .133

TABLE XVIII

Airfoil: N.A.C.A. 4321

Average Reynolds Number: 3,120,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.7.

Test No.: 567 Variable Density Tunnel. Date: April 15, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.325	-7.0	0.0142	-0.069
- .032	-3.9	.0134	- .058
.113	-2.4	.0134	- .056
.263	- .8	.0137	- .053
.405	.7	.0142	- .050
.550	2.3	.0152	- .047
.830	5.4	.0182	- .046
1.086	8.5	.0253	- .041
1.199	10.2	.0327	- .040
1.280	11.9	.0508	- .052
1.291	13.9	.0903	- .060
1.292	15.9	.1287	- .070
1.269	20.0	.2111	- .097
1.113	26.5	.3469	- .119

TABLE XIX

Airfoil: N.A.C.A. 6306

Average Reynolds Number: 3,080,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.6.

Test No.: 575 Variable Density Tunnel. Date: April 17, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.263	-7.2	0.1017	-0.034
.063	-4.2	.0513	- .096
.242	-2.8	.0197	- .109
.408	-1.3	.0097	- .111
.565	.2	.0092	- .111
.722	1.7	.0092	- .111
1.031	4.7	.0106	- .114
1.317	7.8	.0160	- .114
1.539	11.1	.0360	- .113
1.496	13.3	.0986	- .129
1.407	15.5	.1847	- .152
1.254	20.0	.2554	- .194
1.063	26.6	.5148	- .211

TABLE XX

Airfoil: N.A.C.A. 6309

Average Reynolds Number: 3,110,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.8.

Test No.: 576 Variable Density Tunnel. Date: April 18, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.182	-7.4	0.0541	-0.096
- .042	-5.9	.0130	- .111
.109	-4.3	.0110	- .110
.265	-2.8	.0102	- .109
.421	-1.3	.0101	- .108
.728	1.7	.0104	- .108
1.033	4.7	.0116	- .109
1.321	7.8	.0163	- .109
1.572	11.0	.0263	- .110
1.665	12.7	.0375	- .110
1.598	14.9	.1034	- .131
1.497	19.2	.2227	- .164
1.238	26.1	.4793	- .211

TABLE XXI

Airfoil: N.A.C.A. 6312

Average Reynolds Number: 3,170,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.6.

Test No.: 577 Variable Density Tunnel. Date:April 20, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.200	-7.4	0.0132	-0.110
- .046	-5.9	.0121	- .108
.109	-4.3	.0114	- .108
.260	-2.8	.0108	- .107
.420	-1.3	.0108	- .106
.572	.2	.0110	- .103
.721	1.7	.0113	- .104
1.015	4.8	.0134	- .104
1.296	7.9	.0192	- .103
1.545	11.1	.0289	- .103
1.635	12.8	.0414	- .104
1.604	14.9	.0958	- .123
1.530	19.1	.1976	- .143
1.314	25.8	.3760	- .184

TABLE XXII

Airfoil: N.A.C.A. 6315

Average Reynolds Number: 3,100,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.5.

Test No.: 578 Variable Density Tunnel. Date: April 20, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.047	-5.9	0.0127	-0.105
.105	-4.3	.0122	- .104
.258	-2.8	.0120	- .102
.417	-1.3	.0120	- .101
.568	.2	.0121	- .099
.715	1.7	.0128	- .099
1.005	4.8	.0154	- .096
1.276	7.9	.0206	- .096
1.501	11.2	.0358	- .096
1.551	13.1	.0626	- .103
1.496	15.2	.1197	- .117
1.444	19.4	.2074	- .135
1.292	25.9	.3555	- .169

TABLE XXIII

Airfoil: N.A.C.A. 6318

Average Reynolds Number: 3,080,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 21.0.

Test No. 579 Variable Density Tunnel. Date:April 20, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.062	-5.8	0.0135	-0.099
.086	-4.3	.0131	- .096
.236	-2.8	.0130	- .093
.390	-1.2	.0131	- .092
.683	1.8	.0141	- .089
.969	4.9	.0174	- .087
1.228	8.1	.0242	- .084
1.424	11.5	.0481	- .090
1.431	13.4	.0900	- .098
1.428	15.5	.1276	- .107
1.372	19.6	.2148	- .126
1.252	26.0	.3423	- .152

TABLE XXIV

Airfoil: N.A.C.A. 6321

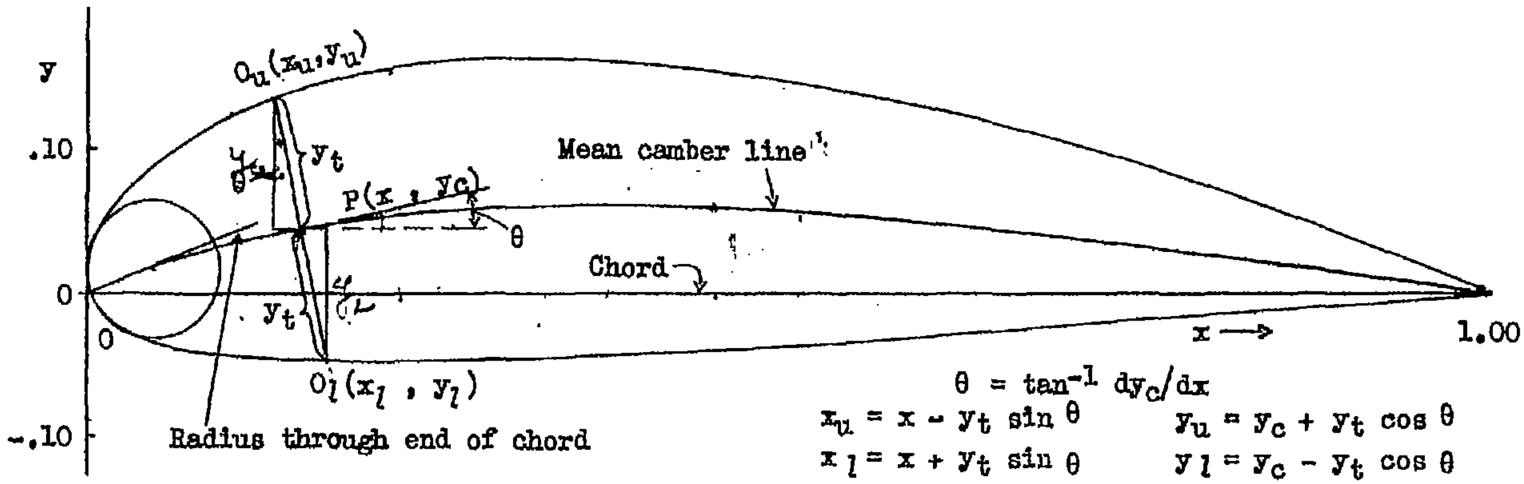
Average Reynolds Number: 3,140,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.8.

Test No.: 580 Variable Density Tunnel. Date: April 21, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.207	-7.3	0.0154	-0.094
.085	-4.3	.0144	- .089
.237	-2.8	.0145	- .087
.378	-1.2	.0148	- .084
.527	.3	.0156	- .080
.667	1.9	.0166	- .079
.943	5.0	.0206	- .076
1.188	8.2	.0300	- .075
1.336	11.8	.0686	- .084
1.354	13.7	.1042	- .091
1.372	15.6	.1261	- .098
1.373	17.6	.1734	- .106
1.332	19.8	.2149	- .113
1.234	26.1	.3283	- .137



Calculation of ordinates N.A.C.A. 6321

x	y_t	y_c	$\tan\theta$	$\sin\theta$	$\cos\theta$	$y_t \sin\theta$	$y_t \cos\theta$	x_u	y_u	x_l	y_l
.0125	.03316	.00489	.38333	.35795	.93374	.01187	.00096	.00063	.03585	.02437	-.02607
.30	.10504	.06000	0	0	1	0	.10504	.30	.16504	.30	-.04504
.60	.07985	.04898	-.07347	-.07326	.99731	-.00585	.00964	.60585	.12862	.59415	-.03066
1.00	.00221	0	-.17143	-.16900	.98562	-.00037	.00218	1.00037	.00218	.99963	-.00218

Fig.1 Diagram and example showing method of calculating the ordinates of N.A.C.A. cambered airfoils.

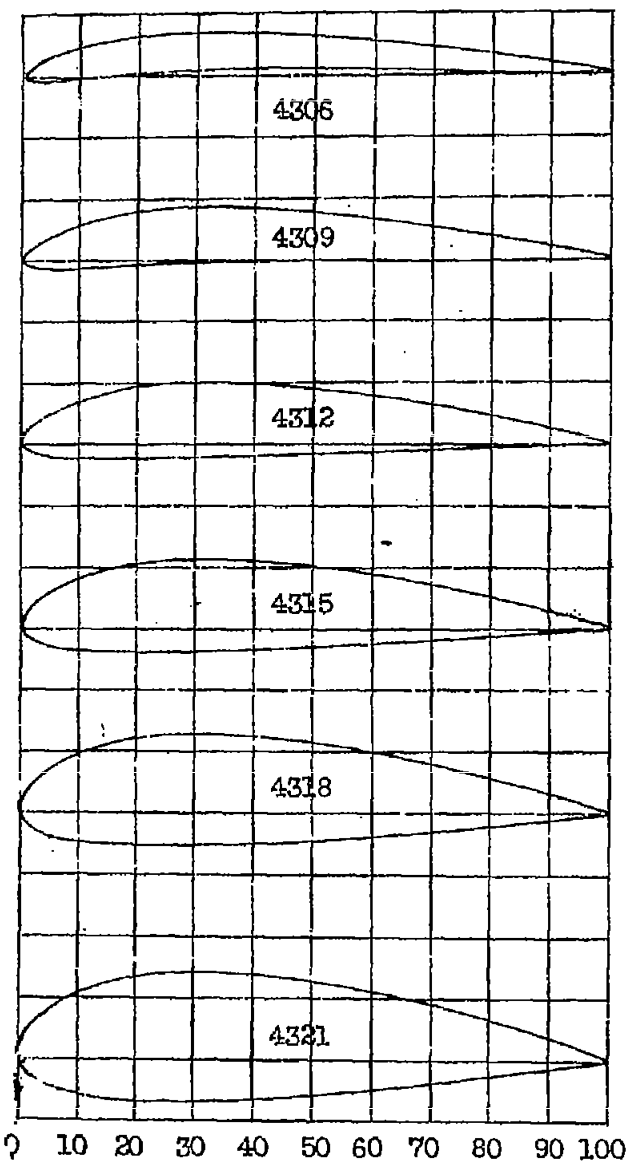
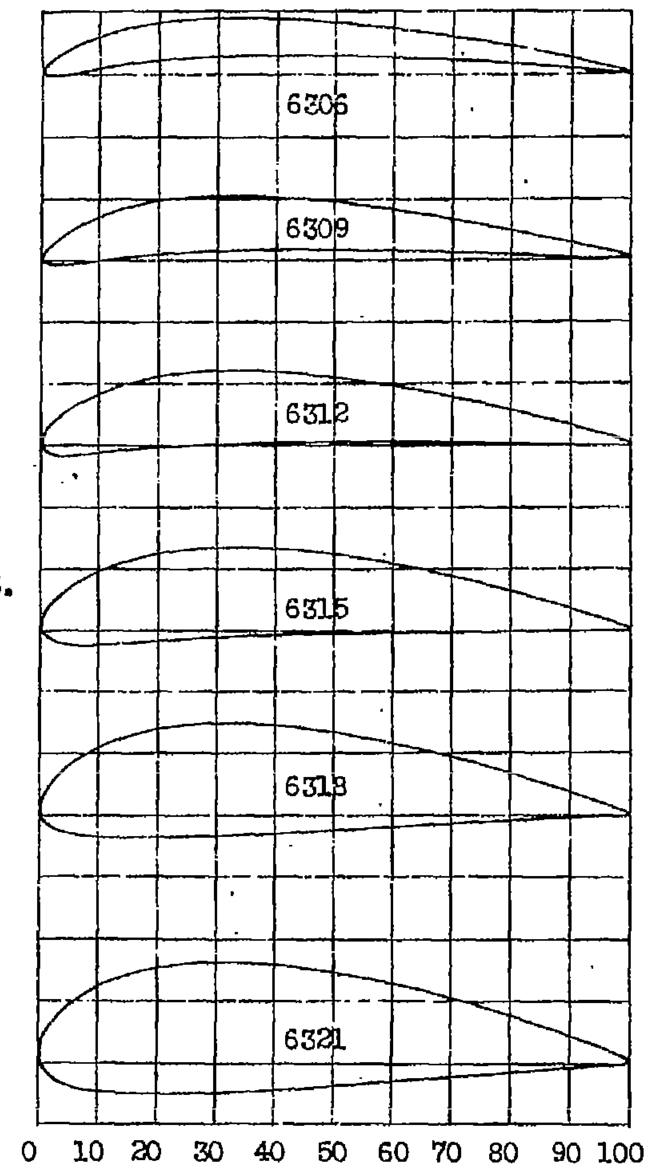


Fig.2 N.A.C.A. airfoil profiles. Series 43 and 63.



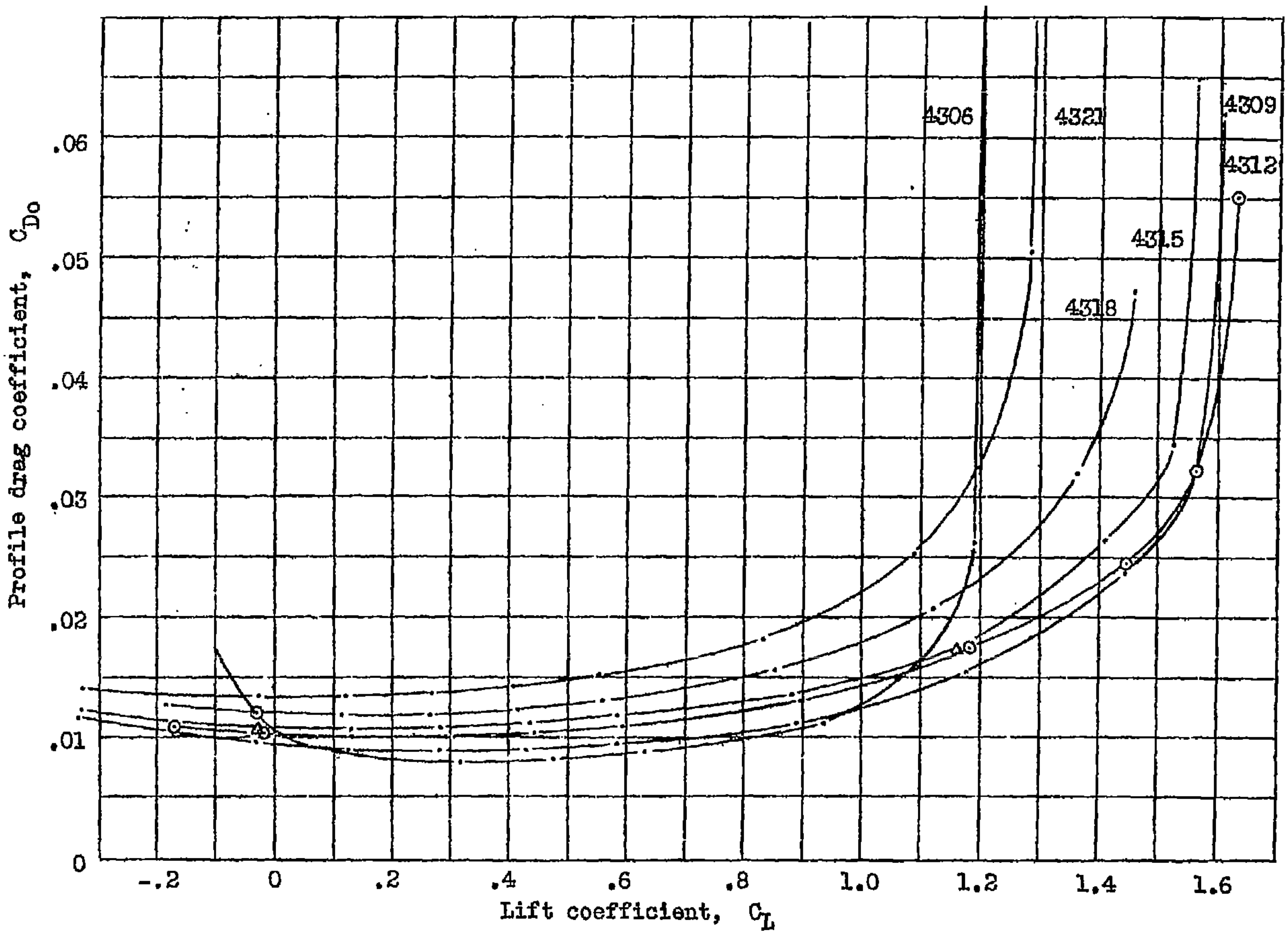


Fig. 3 Profile drag curves for N.A.C.A. 43 series airfoils.

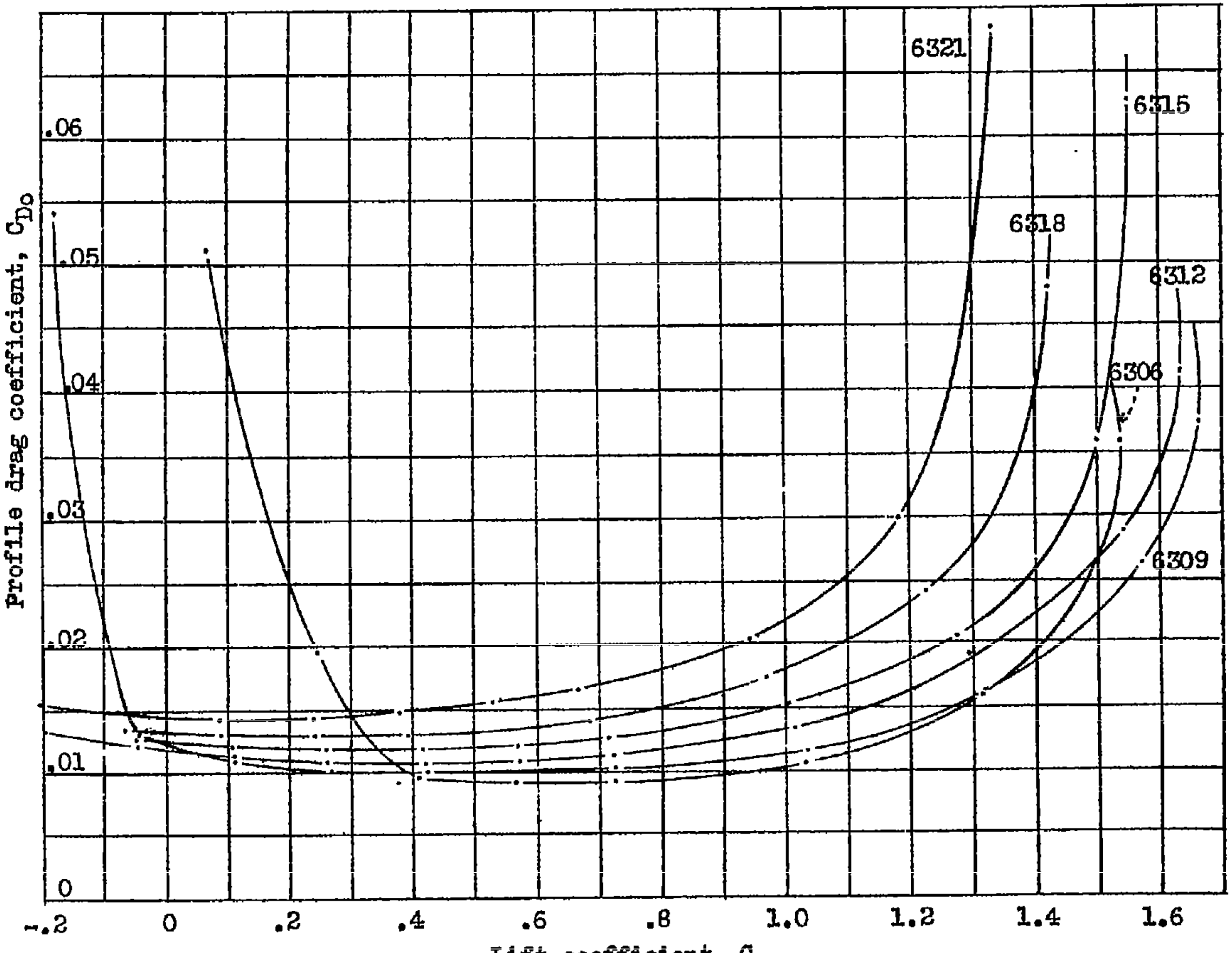


Fig. 4 Profile drag curves for N.A.C.A. 63 series airfoils.

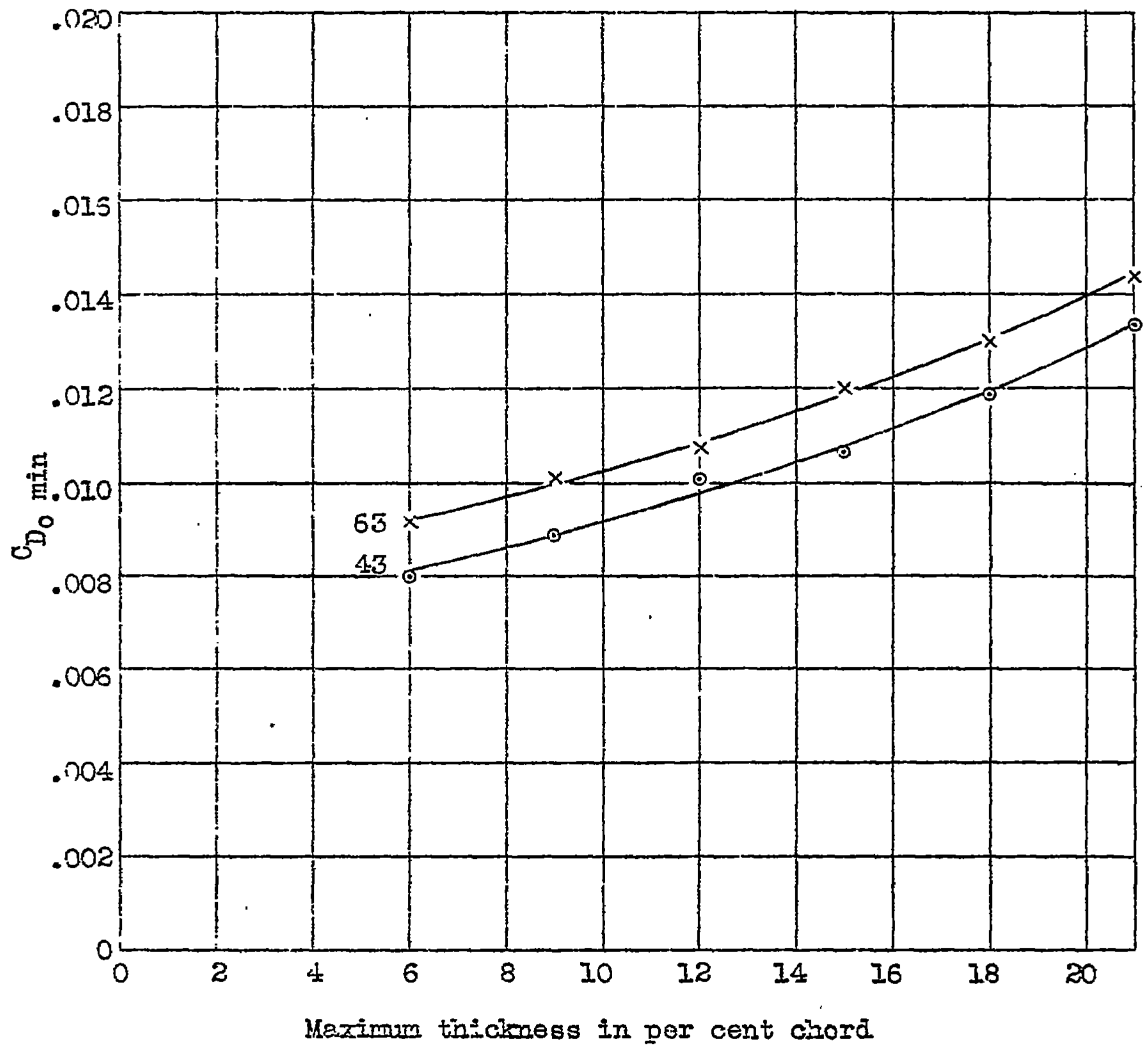


Fig. 5 Variation of minimum profile drag coefficient with thickness.

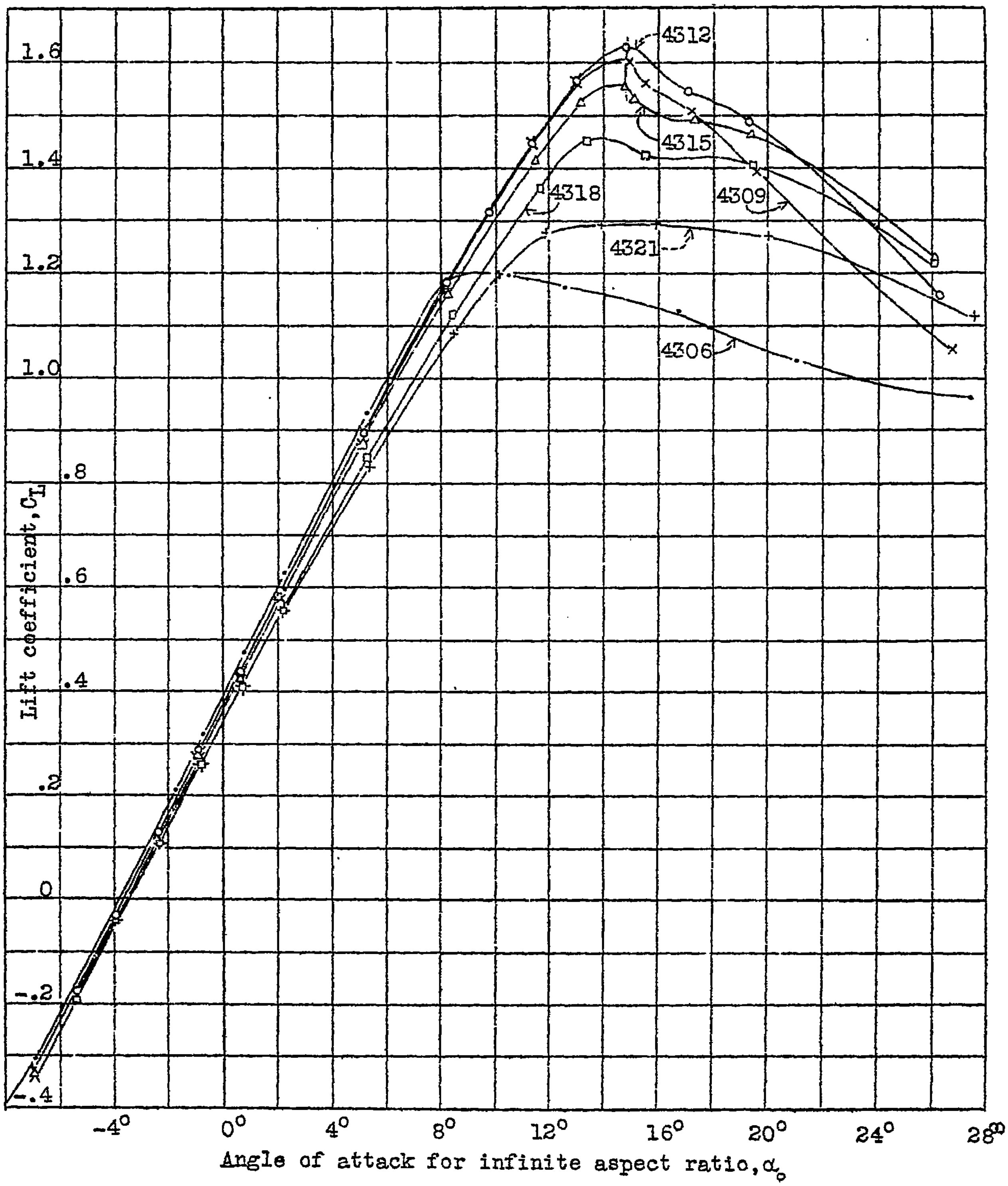


Fig.6 Lift curves for N.A.C.A.48 series airfoils.

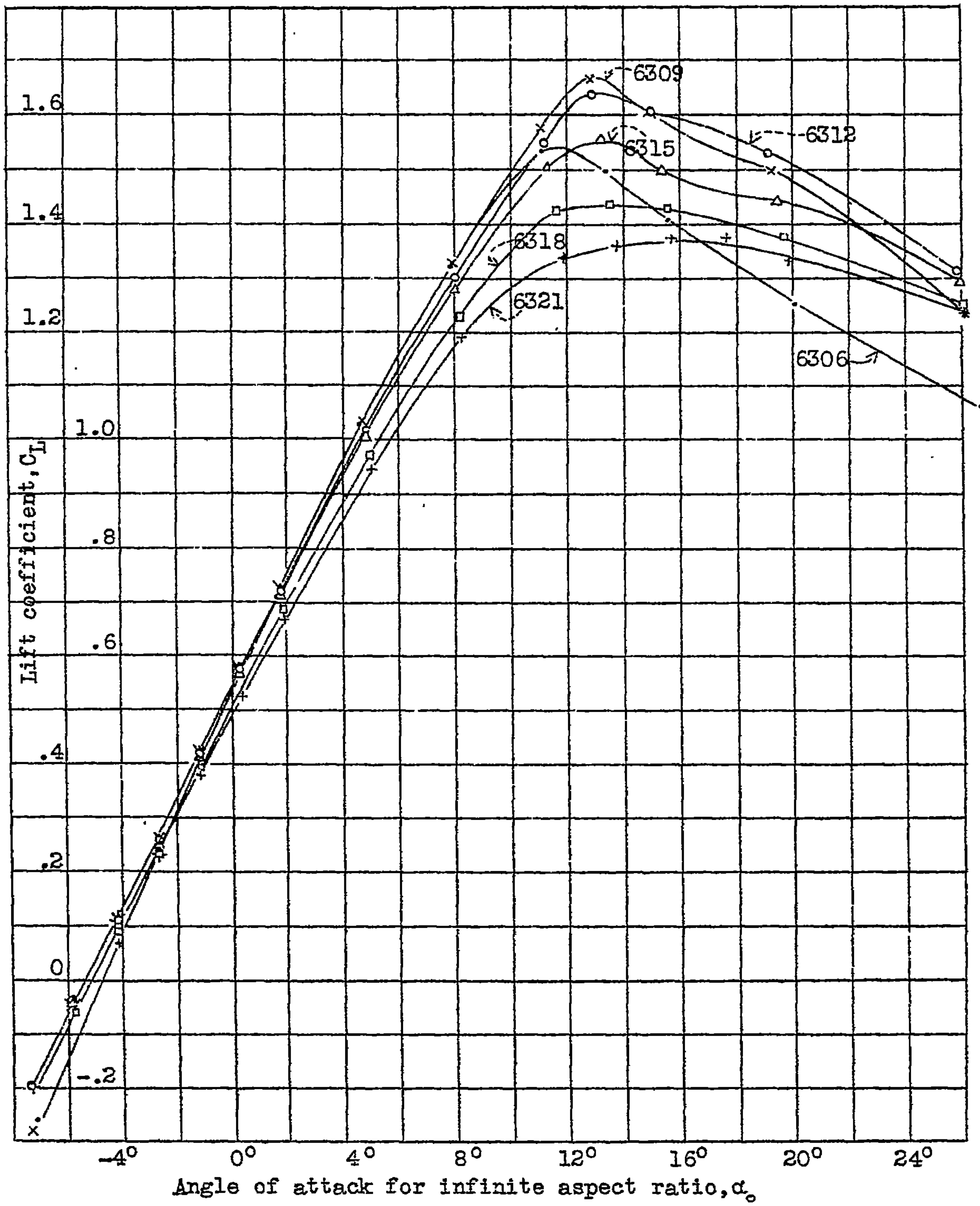


Fig.7 Lift curves for N.A.C.A. 63 series airfoils.

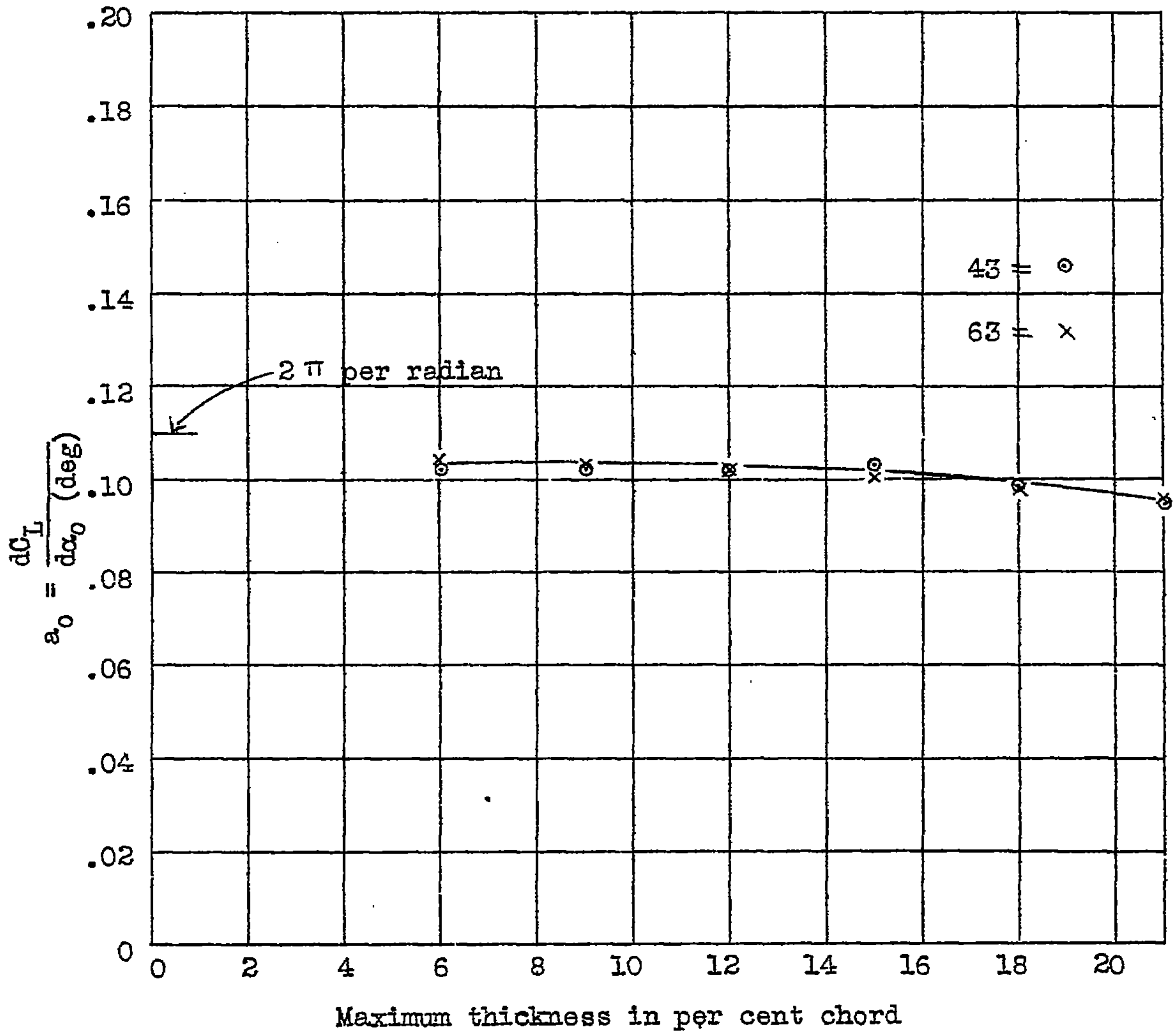


Fig. 8 Variation of lift curve slope with thickness

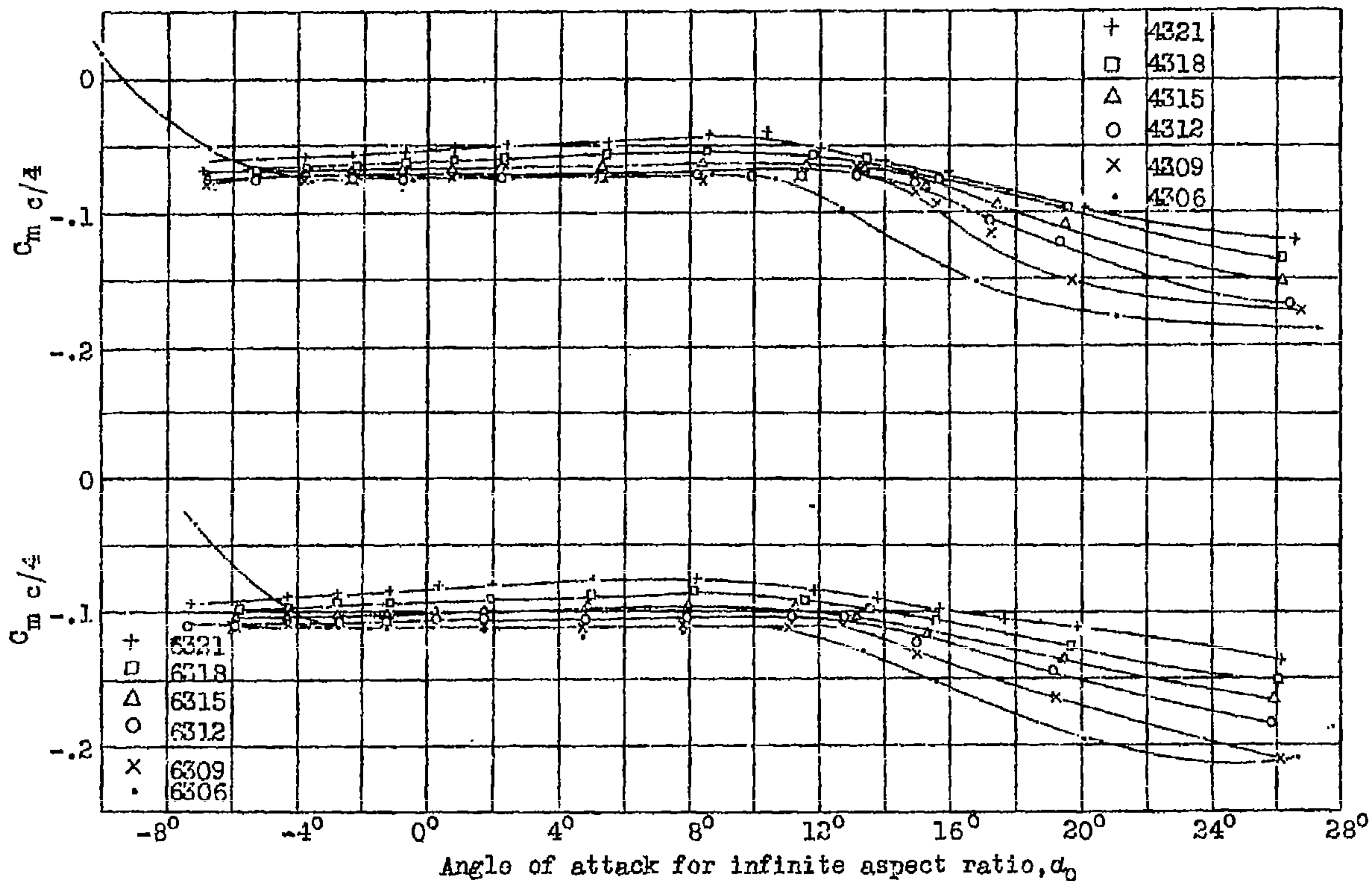


Fig.9 Moment coefficients about a point one-quarter of the chord behind the leading edge.

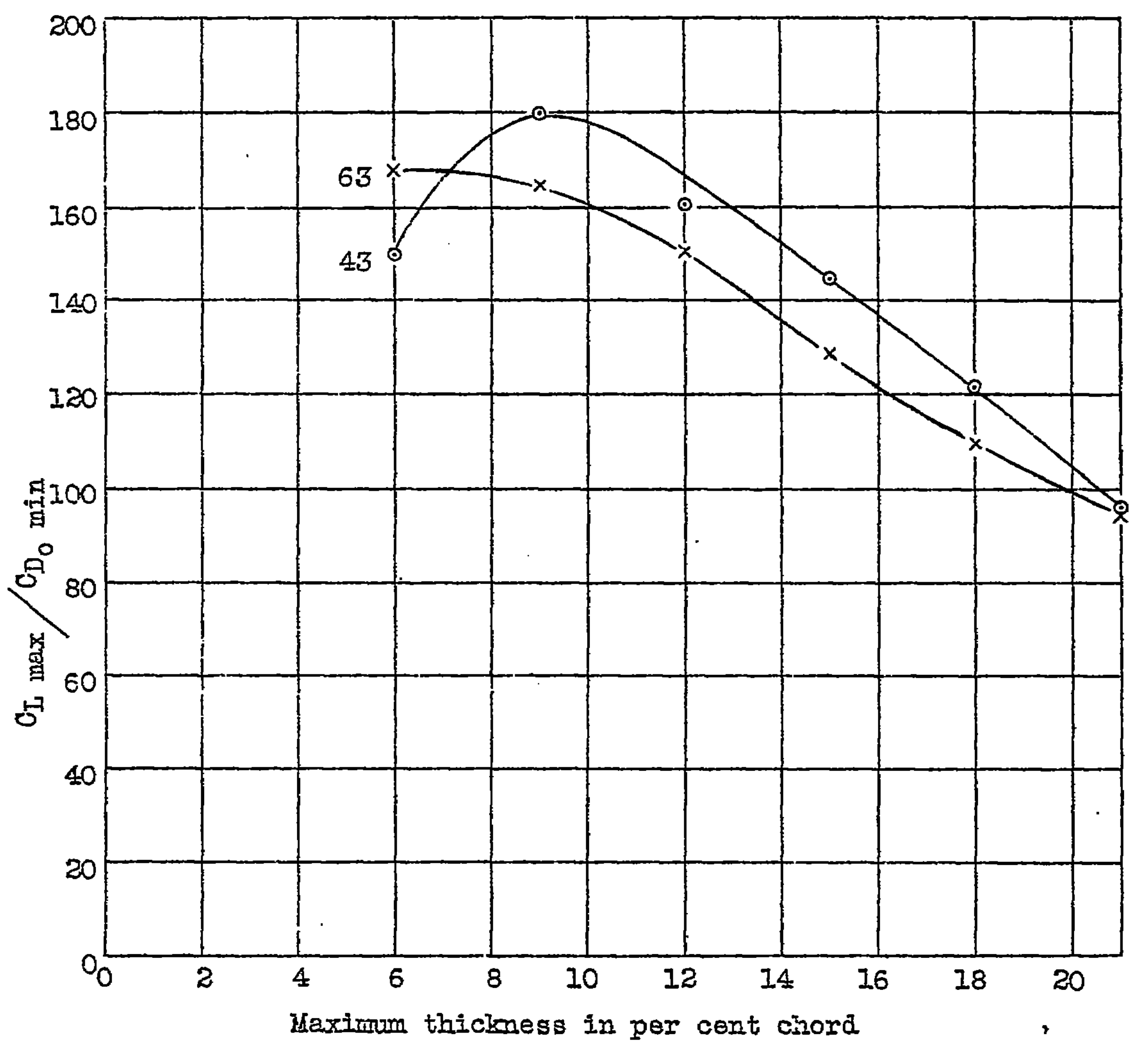


Fig. 10, Ratio of maximum lift to minimum profile drag.

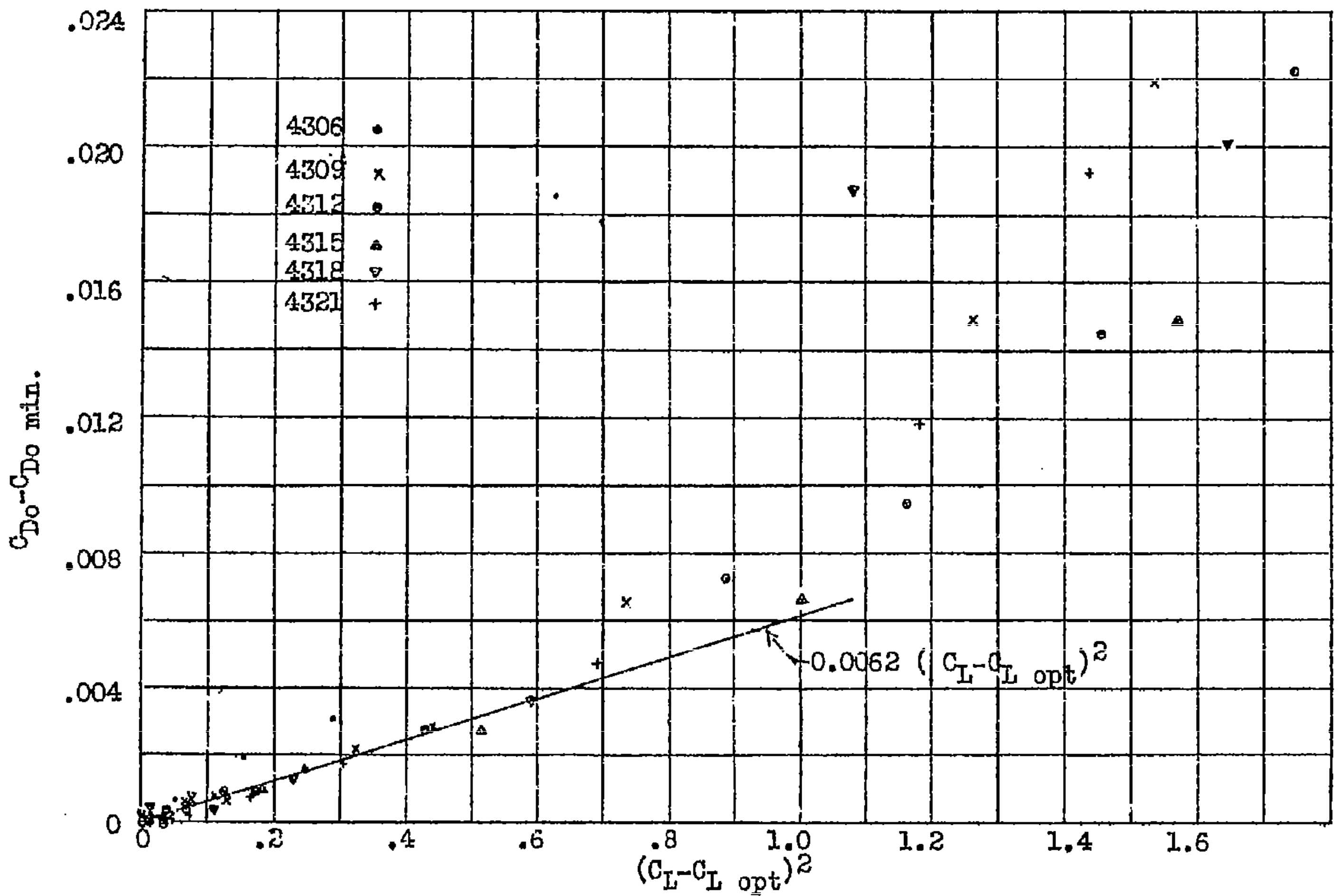


Fig. 11 Increase of profile drag coefficient with lift.

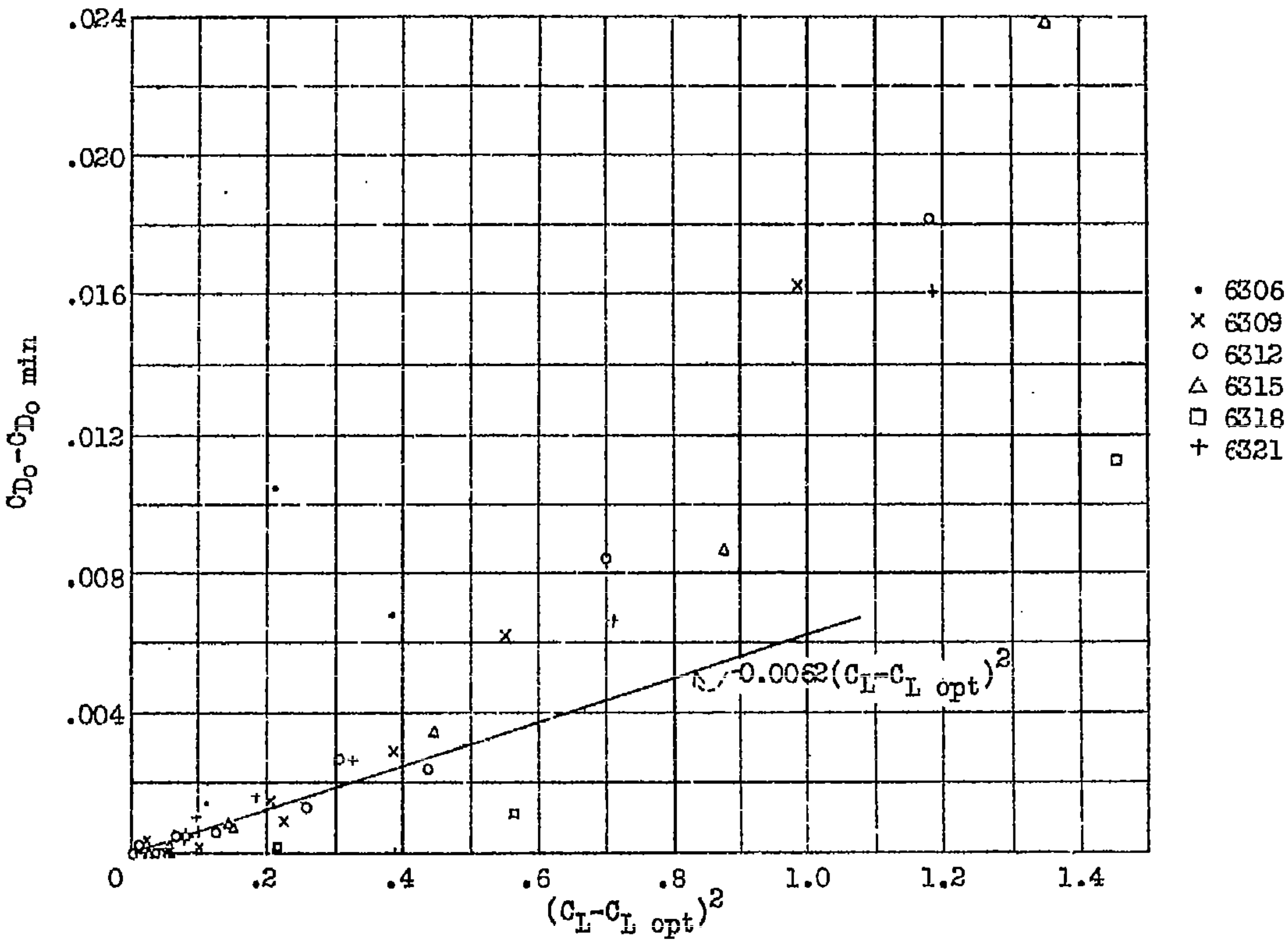


Fig.12 Increase of profile drag coefficient with lift.