F.M. 9.L

Copy 1

TECHNICAL NOTES

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

No. 392

TESTS OF N.A.C.A. AIRFOILS IN THE VARIABLE-DENSITY WIND TUNNEL. SERIES 45 AND 65

By Eastman N. Jacobs and Robert M. Pinkerton
Langley Memorial Aeronautical Laboratory

Washington
September, 1931

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE NO. 392

TESTS OF N.A.C.A. AIRFOILS IN THE VARIABLE-DENSITY WIND TUNNEL. SERIES 45 AND 65

By Eastman N. Jacobs and Robert M. Pinkerton

Summary

This note is one of a series covering an investigation of a number of related airfoils. It presents 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 45 series, has a maximum mean camber of 4 per cent of the chord at a position 0.5 of the chord behind the leading edge, and the other group, the 65 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 form.

Introduction

A large number of related airfoils are being tested in the Variable-Density Wind Tunnel of the National Advisory Committee for Aeronautics with a view to establishing the relation between the geometric and the aerodynamic characteristics of airfoils at a high value of the Reynolds Number. The method employed to develop the airfoils having varying geometric properties is described in detail in references 1 and 2. Briefly, the profiles are obtained by combining certain thickness forms (reference 1) with several related mean camber line forms (reference 2). The airfoils are designated by a number of four digits: the first indicates the maximum mean camber; the second, the position of maximum mean camber; and the last two, the maximum thickness.

Preliminary results already published include the tests on six symmetrical N.A.C.A. airfoils, 00 series (reference 1), and the tests on the 43 and 63 series (reference 2). Similar publications 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 symmetrical series (reference 1) but having curved instead of straight mean camber lines. All twelve airfoils have mean camber lines of such a form that the position of the maximum mean camber is 0.5 of the chord behind the leading edge. Six of the airfoils, the 45 series, have a maximum mean camber of 4 per cent of the chord, and the other six, the 65 series, have a maximum mean camber of 6 per cent.

Description of Airfoils

The ordinates of the N.A.C.A. airfoils with which this note deals were obtained by the method given in reference 2. The mean camber lines of these airfoils are:

45 series $y_c = 0.16x(1 - x)$

65 series $y_c = 0.24x(1 - x)$

The ordinates, obtained by combining these mean camber lines with the basic thickness forms (reference 1), are given in Tables I to XII, and the profile shapes are shown in Figure 1.

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.

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 are given in reference 1.

The results are presented in the form of coefficients corrected, after the method of reference 3, 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 4. This relation may be expressed by the equation

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

where t is the maximum thickness/chord ratio. The first three terms of the above expression give the minimum profile drag coefficient for the six symmetrical N.A.C.A. airfoils. The value of k is constant for the 45 series at $k = 0.0014$, but for the 65 series, varies from 0.0020 for the N.A.C.A. 6506 to 0.0029 for the N.A.C.A. 6521. The calculated curves, using $k = 0.0025$ (average value) for the 65 series, and the test points taken from the faired profile drag curves (figs. 2 and 3) are shown in Figure 4

Maximum lift coefficients taken from Figures 5 and 6 are given in the table below:

Airfoil	$C_{L\ max}$	Airfoil	$C_{L\ max}$
4506	1.16	6506	1.29
4509	1.56	6509	1.71
4512	1.68	6512	1.75
4515	1.62	6515	1.67
4518	1.54	6518	1.61
4521	1.46	6521	1.49

In agreement with the results previously published (references 1 and 2), these results show that the sections of moderate thickness give the highest maximum lift coefficients.

The variation of the slope of the lift curve with thickness is shown in Figure 7. The points on the figure represent the deduced slopes as measured in the angle-of-attack range in the neighborhood of minimum profile drag for an infinite span wing. It will be noted that all of the points lie below the approximate theoretical value, 2π per radian. These results are in agreement with previous results in that the lift curve slope tends to decrease with thickness.

The pitching moment coefficients at zero lift are given in the following table:

Airfoil	C_{m_o}	Airfoil	C_{m_o}
4506	-0.107	6506	-
4509	- .107	6509	-0.158
4512	- .101	6512	- .152
4515	- .097	6515	- .148
4518	- .094	6518	- .138
4521	- .082	6521	- .128

The calculation of the moment coefficient has commonly been based on the assumption that an airfoil section 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, as 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 $C_{L\,max}/C_{D_o\,min}$ ratio has been used as a measure of the general efficiency of an airfoil section. The variation of this ratio with thickness is shown in Figure 9. It will be noted that the N.A.C.A. 4509 and N.A.C.A. 6509 give the highest value of this ratio.

<u>Variation of the Aerodynamic Characteristics with Lift or Angle</u>.- The variation of profile-drag coefficient with lift coefficient is shown by Figures 2 and 3. Following the procedure given in reference 2, 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 $(C_L - C_{L\,opt})^2$ where $C_{L\,opt}$ is called the optimum lift coefficient; that is, the lift coefficient corresponding to

minimum profile drag coefficient. These plots are given in Figures 10 and 11. It is significant that the same line determined in reference 1 and used in reference 2 may be used here to represent to a reasonable degree of accuracy the additional drag coefficient for the moderately thick airfoils at values of the lift coefficient less than 1. This may not be so apparent from Figure 11; however, a simple calculation shows that for a lift coefficient of 1 for the 6515 airfoil, the value of $(C_L - C_{L\ opt})^2$ is .48 and within that limit the points lie reasonably close to the line. The profile drag coefficient for the moderately thick airfoils may, therefore, be approximated by

$$C_{D_o} = C_{D_o\ min} + 0.0062(C_L - C_{L\ opt})^2.$$

$C_{D_o\ min}$ has been expressed earlier as a function of thickness.

The optimum lift coefficient varies with thickness as well as with camber, the value increasing with camber but decreasing with thickness. The values of $C_{L\ opt}$ are given in the following table:

Airfoil	$C_{L\ opt}$	Airfoil	$C_{L\ opt}$
4506	0.40	6506	0.64
4509	.33	6509	.53
4512	.26	6512	.42
4515	.19	6515	.31
4518	.12	6518	.20
4521	.05	6521	.09

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 change is at the quarter-chord point. However, the curves of $C_{m_{c/4}}$ against angle of attack (fig. 8) show a slope in the normal working range as did the corresponding curves in references 1 and 2. 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)
4506	0.4	6506	0.1
4509	.4	6509	.2
4512	.8	6512	.5
4515	.8	6515	1.2
4518	1.3	6518	1.5
4521	2.3	6521	1.7

In reference 1 the center of pressure for symmetrical airfoils is shown to be farther forward for the thick airfoils than for the thin airfoils. It should be noted here that the center of pressure and the point of constant moment for a symmetrical section are coincident, since the only forces considered as acting on such a section are those due to angular change. The present results may be considered as indicating that with increasing profile thickness there is a similar progressive forward displacement of the center of pressure for that part of the air forces due to angular change.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., September 4, 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, 1931.

2. Jacobs, Eastman N., and Pinkerton, Robert M.: Tests of N.A.C.A. Airfoils in the Variable-Density Wind Tunnel. Series 43 and 63. N.A.C.A. Technical Note No. 391, 1931.

3. 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. 4506
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
1.104	1.134	1.396	-0.738
2.304	1.682	2.696	- .902
4.747	2.520	5.253	-1.000
7.217	3.191	7.783	- .971
9.703	3.763	10.297	- .883
14.702	4.696	15.298	- .616
19.726	5.416	20.274	- .296
29.808	6.355	30.192	+ .365
39.907	6.740	40.093	+ .940
50.000	6.648	50.000	+1.352
60.073	6.121	59.927	+1.559
70.117	5.188	69.883	+1.532
80.125	3.865	79.875	+1.255
90.092	2.158	89.908	+ .722
95.058	1.162	94.942	+ .358
100.010	.062	99.990	- .062
L.E. radius	.394		
Slope of radius passing through end of chord	4/25		

TABLE II

Ordinates for Airfoil N.A.C.A. 4509
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
1.031	1.602	1.469	-1.206
2.205	2.328	2.795	-1.548
4.620	3.401	5.380	-1.881
7.076	4.231	7.924	-2.011
9.554	4.926	10.446	-2.046
14.554	6.026	15.446	-1.946
19.589	6.845	20.411	-1.725
29.712	7.853	30.288	-1.133
39.861	8.189	40.139	- .509
50.000	7.971	50.000	+ .029
60.109	7.261	59.891	+ .419
70.176	6.102	69.824	+ .618
80.188	4.517	79.812	+ .603
90.138	2.518	89.862	+ .362
95.086	1.359	94.914	+ .161
100.015	.094	99.985	- .094
L.E. radius	.887		
Slope of radius passing through end of chord	4/25		

TABLE III

Ordinates for Airfoil N.A.C.A. 4512
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.958	2.070	1.542	-1.674
2.107	2.974	2.893	-2.194
4.493	4.282	5.507	-2.762
6.934	5.272	8.066	-3.052
9.405	6.085	10.595	-3.205
14.405	7.352	15.595	-3.272
19.452	8.272	20.548	-3.152
29.617	9.350	30.383	-2.630
39.814	9.640	40.186	-1.960
50.000	9.294	50.000	-1.294
60.146	8.401	59.854	- .721
70.234	7.018	69.766	- .298
80.250	5.169	79.750	- .049
90.184	2.877	89.816	+ .003
95.115	1.558	94.885	- .038
100.020	.124	99.980	- .124
L.E. radius	1.576		
Slope of radius passing through end of chord	4/25		

TABLE IV

Ordinates for Airfoil N.A.C.A. 4515
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.885	2.538	1.615	-2.142
2.009	3.621	2.991	-2.841
4.366	5.161	5.634	-3.641
6.793	6.312	8.207	-4.092
9.257	7.249	10.743	-4.369
14.256	8.681	15.744	-4.601
19.315	9.699	20.685	-4.579
29.521	10.848	30.479	-4.128
39.768	11.090	40.232	-3.410
50.000	10.619	50.000	-2.619
60.182	9.541	59.818	-1.861
70.293	7.931	69.707	-1.211
80.313	5.822	79.687	- .702
90.230	3.236	89.770	- .356
95.144	1.761	94.856	- .241
100.025	.156	99.975	- .156
L.E. radius	2.464		
Slope of radius passing through end of chord	4/25		

TABLE V

Ordinates for Airfoil N.A.C.A. 4518
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.812	3.006	1.688	-2.610
1.911	4.266	3.089	-3.486
4.240	6.041	5.760	-4.521
6.651	7.353	8.349	-5.133
9.108	8.407	10.892	-5.527
14.107	10.011	15.893	-5.931
19.178	11.127	20.822	-6.007
29.425	12.345	30.575	-5.625
39.722	12.538	40.278	-4.858
50.000	11.942	50.000	-3.942
60.219	10.683	59.781	-3.003
70.351	8.844	69.649	-2.124
80.376	6.477	79.624	-1.357
90.276	3.595	89.724	- .715
95.172	1.957	94.828	- .437
100.030	.187	99.970	- .187
L.E. radius	3.549		
Slope of radius passing through end of chord	4/25		

TABLE VI

Ordinates for Airfoil N.A.C.A. 4521
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.739	3.474	1.761	-3.078
1.813	4.912	3.187	-4.132
4.113	6.916	5.887	-5.396
6.510	8.393	8.490	-6.173
8.960	9.568	11.040	-6.688
13.959	11.336	16.041	-7.256
19.041	12.553	20.959	-7.433
29.329	13.842	30.671	-7.122
39.675	13.990	40.325	-6.310
50.000	13.265	50.000	-5.265
60.255	11.821	59.745	-4.141
70.409	9.757	69.591	-3.037
80.439	7.130	79.561	-2.010
90.322	3.955	89.678	-1.075
95.201	2.156	94.799	- .636
100.035	.218	99.965	- .218
L.E. radius	4.830		
Slope of radius passing through end of chord	4/25		

TABLE VII

Ordinates for Airfoil N.A.C.A. 6506
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
1.034	1.218	1.466	-0.626
2.209	1.859	2.791	- .689
4.625	2.878	5.375	- .598
7.080	3.723	7.920	- .393
9.558	4.460	10.442	- .140
14.557	5.696	15.443	+ .424
19.591	6.680	20.409	+1.000
29.713	8.027	30.287	+2.053
39.861	8.658	40.139	+2.862
50.000	8.648	50.000	+3.352
60.109	8.039	59.891	+3.481
70.175	6.864	69.825	+3.216
80.187	5.138	79.813	+2.542
90.137	2.871	89.863	+1.449
95.086	1.537	94.914	+ .743
100.015	.061	99.985	- .061
L.E. radius	.394		
Slope of radius passing through end of chord	6/25		

TABLE VIII

Ordinates for Airfoil N.A.C.A. 6509
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.926	1.680	1.574	-1.088
2.064	2.496	2.936	-1.326
4.437	3.748	5.563	-1.468
6.870	4.751	8.130	-1.421
9.337	5.611	10.663	-1.291
14.335	7.016	15.665	- .896
19.386	8.101	20.614	- .421
29.570	9.521	30.430	+ .559
39.791	10.106	40.209	+1.414
50.000	9.971	50.000	+2.029
60.164	9.179	59.836	+2.341
70.263	7.775	69.737	+2.305
80.280	5.786	79.720	+1.894
90.205	3.227	89.795	+1.093
95.128	1.731	94.872	+ .549
100.022	.092	99.978	- .092
L.E. radius	.887		
Slope of radius passing through end of chord	6/25		

TABLE IX

Ordinates for Airfoil N.A.C.A. 6512
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.818	2.141	1.682	-1.549
1.919	3.134	3.081	-1.964
4.249	4.618	5.751	-2.338
6.661	5.780	8.339	-2.450
9.117	6.759	10.883	-2.439
14.114	8.331	15.886	-2.211
19.182	9.519	20.818	-1.839
29.426	11.015	30.574	- .935
39.722	11.556	40.278	- .036
50.000	11.294	50.000	+ .706
60.219	10.318	59.781	+1.202
70.350	8.688	69.650	+1.392
80.374	6.434	79.626	+1.246
90.273	3.583	89.727	+ .737
95.170	1.928	94.830	+ .352
100.029	.123	99.971	- .123
L.E. radius	1.576		
Slope of radius passing through end of chord	6/25		

TABLE X

Ordinates for Airfoil N.A.C.A. 6515
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.710	2.602	1.790	-2.010
1.714	3.771	3.226	-2.601
4.061	5.486	5.939	-3.206
6.451	6.809	8.549	-3.479
8.896	7.911	11.104	-3.591
13.893	9.651	16.107	-3.531
18.978	10.939	21.022	-3.259
29.283	12.509	30.717	-2.429
39.652	13.006	40.348	-1.486
50.000	12.619	50.000	- .619
60.274	11.457	59.726	+ .063
70.438	9.599	69.562	+ .481
80.467	7.084	79.533	+ .596
90.341	3.939	89.659	+ .382
95.213	2.128	94.787	+ .152
100.037	.154	99.963	- .154
L.E. radius	2.464		
Slope of radius passing through end of chord	6/25		

TABLE XI

Ordinates for Airfoil N.A.C.A. 6518
(Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.602	3.063	1.898	-2.471
1.629	4.407	3.371	-3.237
3.874	6.355	6.126	-4.075
6.241	7.838	8.759	-4.508
8.676	9.058	11.324	-4.738
13.671	10.970	16.329	-4.850
18.773	12.358	21.227	-4.678
29.140	14.002	30.860	-3.922
39.583	14.452	40.417	-2.932
50.000	13.942	50.000	-1.942
60.328	12.598	59.672	-1.078
70.525	10.510	69.475	- .430
80.561	7.735	79.439	- .055
90.410	4.294	89.590	+ .026
95.255	2.322	94.745	- .042
100.044	.184	99.956	- .184
L.E. radius	3.549		
Slope of radius passing through end of chord	6/25		

TABLE XII

Ordinates for Airfoil N.A.C.A. 6521
Dimensions in per cent of chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
-	-	0	0
0.494	3.525	2.006	-2.933
1.483	5.045	3.517	-3.875
3.687	7.220	6.313	-4.940
6.031	8.867	8.969	-5.537
8.455	10.207	11.545	-5.887
13.450	12.285	16.550	-6.165
18.569	13.777	21.431	-6.097
28.996	15.495	31.004	-5.415
39.513	15.903	40.487	-4.383
50.000	15.265	50.000	-3.265
60.383	13.736	59.617	-2.216
70.613	11.421	69.387	-1.341
80.654	8.384	79.346	- .704
90.478	4.651	89.522	- .331
95.298	2.518	94.702	- .238
100.052	.215	99.948	- .215
L.E. radius	4.830		
Slope of radius passing through end of chord	6/25		

TABLE XIII

Airfoil: N.A.C.A. 4506

Average Reynolds Number: 3,050,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.6.

Test No.: 568 Variable-Density Tunnel. April 15, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.132	-5.6	0.0200	-0.105
+ .023	-4.1	.0099	- .109
.181	-2.6	.0092	- .109
.335	-1.1	.0087	- .108
.491	+ .4	.0088	- .109
.639	2.0	.0091	- .106
.940	5.0	.0118	- .106
1.053	6.6	.0252	- .101
1.105	8.5	.0602	- .090
1.154	12.3	.1779	- .127
1.123	16.4	.2994	- .176
1.025	20.7	.3862	- .197
.984	26.9	.5064	- .209

TABLE XIV

Airfoil: N.A.C.A. 4509

Average Reynolds Number: 3,120,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.9.

Test No.: 569 Variable-Density Tunnel. April 15, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.141	-5.6	0.0106	-0.107
+ .012	-4.0	.0100	- .106
.166	-2.5	.0095	- .106
.320	-1.0	.0092	- .105
.474	+ .5	.0095	- .105
.622	2.0	.0098	- .104
.919	5.1	.0123	- .104
1.208	8.2	.0165	- .102
1.461	11.4	.0260	- .103
1.559	13.0	.0353	- .099
1.516	15.2	.1159	- .110
1.263	20.0	.2261	- .175
1.000	26.8	.4938	- .192

TABLE XV

Airfoil: N.A.C.A. 4512

Average Reynolds Number: 3,100,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.6.

Test No.: 570 Variable-Density Tunnel. April 16, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.148	-5.5	0.0111	-0.102
+ .003	-4.0	.0106	- .101
.156	-2.5	.0104	- .100
.310	-1.0	.0103	- .100
.462	+ .5	.0108	- .099
.611	2.1	.0112	- .097
.904	5.1	.0133	- .095
1.180	8.2	.0177	- .092
1.440	11.4	.0256	- .090
1.555	13.1	.0324	- .089
1.648	14.8	.0435	- .088
1.685	16.6	.0674	- .094
1.568	19.0	.1729	- .129
1.159	26.3	.4098	- .182

TABLE XVI

Airfoil: N.A.C.A. 4515

Average Reynolds Number: 3,080,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.7.

Test No. 571 Variable-Density Tunnel. April 16, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.297	-7.1	0.0154	-0.100
+ .007	-4.0	.0116	- .097
.159	-2.5	.0115	- .095
.310	-1.0	.0113	- .094
.462	+ .5	.0117	- .093
.606	2.1	.0126	- .091
.896	5.2	.0150	- .089
1.170	8.3	.0194	- .086
1.419	11.5	.0287	- .085
1.604	14.9	.0537	- .089
1.619	16.9	.0915	- .099
1.581	19.0	.1606	- .119
1.227	26.1	.3837	- .172

TABLE XVII

Airfoil: N.A.C.A. 4518

Average Reynolds Number: 3,130,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 21.0.

Test No.: 572 Variable-Density Tunnel. April 16, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.158	-5.5	0.0130	-0.096
- .013	-4.0	.0126	- .094
+ .138	-2.4	.0125	- .092
.287	- .9	.0126	- .090
.580	+2.2	.0140	- .085
.864	5.3	.0168	- .082
1.129	8.4	.0224	- .077
1.361	11.7	.0349	- .077
1.457	13.4	.0464	- .078
1.520	15.2	.0690	- .085
1.539	17.1	.1092	- .095
1.515	19.2	.1633	- .111
1.334	25.8	.3417	- .153

TABLE XVIII

Airfoil: N.A.C.A. 4521

Average Reynolds Number: 3,150,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.8.

Test No.: 573 Variable-Density Tunnel. April 17, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.186	-5.4	0.0142	-0.086
- .046	-3.9	.0139	- .082
+ .103	-2.3	.0138	- .080
.248	- .8	.0140	- .077
.535	+2.3	.0159	- .072
.812	5.4	.0190	- .069
1.069	8.6	.0260	- .063
1.291	11.9	.0419	- .067
1.377	13.6	.0571	- .071
1.429	15.5	.0819	- .077
1.453	17.4	.1196	- .087
1.455	19.4	.1659	- .099
1.437	21.4	.2135	- .113
1.325	25.8	.3198	- .136

TABLE XIX

Airfoil: N.A.C.A. 6506

Average Reynolds Number: 3,170,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 21.1.

Test No. 586 Variable-Density Tunnel. April 23, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.162	-7.5	0.0704	-0.113
+ .011	-6.0	.0216	- .151
.177	-4.6	.0105	- .159
.332	-3.1	.0099	- .159
.486	-1.5	.0096	- .159
.640	0.0	.0093	- .162
.784	1.5	.0099	- .159
1.077	4.6	.0126	- .159
1.208	6.2	.0171	- .157
1.282	7.9	.0369	- .145
1.290	9.9	.0793	- .132
1.279	11.9	.1376	- .142
1.285	13.9	.2082	- .165
1.260	16.0	.2810	- .192
1.165	20.3	.3959	- .222
1.048	26.7	.5327	- .238

TABLE XX

Airfoil: N.A.C.A. 6509

Average Reynolds Number: 3,110,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.8.

Test No.: 585 Variable-Density Tunnel. April 22, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.132	-7.6	0.0131	-0.157
+ .022	-6.1	.0119	- .158
.182	-4.6	.0110	- .158
.336	-3.1	.0104	- .159
.489	-1.6	.0100	- .159
.645	- .1	.0101	- .159
.789	+1.5	.0108	- .158
.934	3.0	.0122	- .158
1.072	4.6	.0139	- .157
1.339	7.7	.0194	- .154
1.572	11.0	.0312	- .149
1.698	13.8	.0521	- .146
1.711	14.6	.0617	- .146
1.650	16.8	.1392	- .171
1.481	19.3	.2788	- .201
1.119	26.4	.5062	- .223

TABLE XXI

Airfoil: N.A.C.A. 6512

Average Reynolds Number: 3,070,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.6.

Test No. 584 Variable-Density Tunnel. April 22, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.135	-7.6	0.0137	-0.153
+ .174	-4.6	.0116	- .153
.325	-3.0	.0112	- .153
.480	-1.5	.0111	- .152
.631	0.0	.0116	- .151
.769	1.6	.0124	- .150
1.053	4.6	.0157	- .148
1.320	7.8	.0203	- .142
1.548	11.1	.0331	- .140
1.645	12.8	.0441	- .138
1.716	14.5	.0599	- .140
1.747	16.5	.0895	- .145
1.735	18.5	.1448	- .160
1.265	26.0	.4137	- .215

TABLE XXII

Airfoil: N.A.C.A. 6515

Average Reynolds Number: 3,100,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.8.

Test No.: 583 Variable-Density Tunnel. April 22, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.157	-7.5	0.0141	-0.149
+ .148	-4.5	.0129	- .147
.298	-2.9	.0128	- .145
.451	-1.4	.0127	- .144
.599	+ .1	.0134	- .141
.734	1.7	.0143	- .140
1.016	4.8	.0175	- .133
1.268	8.0	.0245	- .128
1.489	11.3	.0394	- .126
1.577	13.0	.0524	- .127
1.636	14.8	.0739	- .128
1.667	16.7	.1049	- .134
1.668	18.7	.1515	- .145
1.632	20.8	.2130	- .162
1.475	25.3	.3549	- .190

TABLE XXIII

Airfoil: N.A.C.A. 6518

Average Reynolds Number: 3,100,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.8.

Test No.: 582 Variable-Density Tunnel. April 21, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.169	-7.5	0.0148	-0.141
- .020	-5.9	.0142	- .139
+ .130	-4.4	.0141	- .137
.274	-2.9	.0141	- .135
.420	-1.3	.0142	- .132
.567	+ .2	.0150	- .129
.703	1.8	.0161	- .126
.975	4.9	.0206	- .121
1.226	8.1	.0293	- .116
1.435	11.4	.0471	- .118
1.514	13.2	.0635	- .119
1.576	15.0	.0861	- .120
1.603	16.9	.1170	- .127
1.609	18.9	.1592	- .137
1.593	20.9	.2098	- .150
1.465	25.3	.3355	- .176

TABLE XXIV

Airfoil: N.A.C.A. 6521

Average Reynolds Number: 3,080,000.

Size of model: 5 x 30 inches.

Pressure, Standard Atmospheres: 20.6.

Test No.: 581 Variable-Density Tunnel. April 21, 1931.

C_L	α_o (degrees)	C_{D_o}	$C_{m_{c/4}}$
-0.201	-7.4	0.0159	-0.133
+ .090	-4.3	.0154	- .128
.234	-2.7	.0156	- .124
.383	-1.2	.0162	- .121
.521	+ .3	.0176	- .118
.657	1.9	.0193	- .116
.922	5.1	.0252	- .113
1.168	8.3	.0355	- .109
1.354	11.7	.0611	- .112
1.416	13.5	.0827	- .115
1.462	15.4	.1098	- .122
1.486	17.3	.1483	- .127
1.489	19.3	.1870	- .137
1.480	21.3	.2332	- .149
1.288	25.9	.3500	- .168

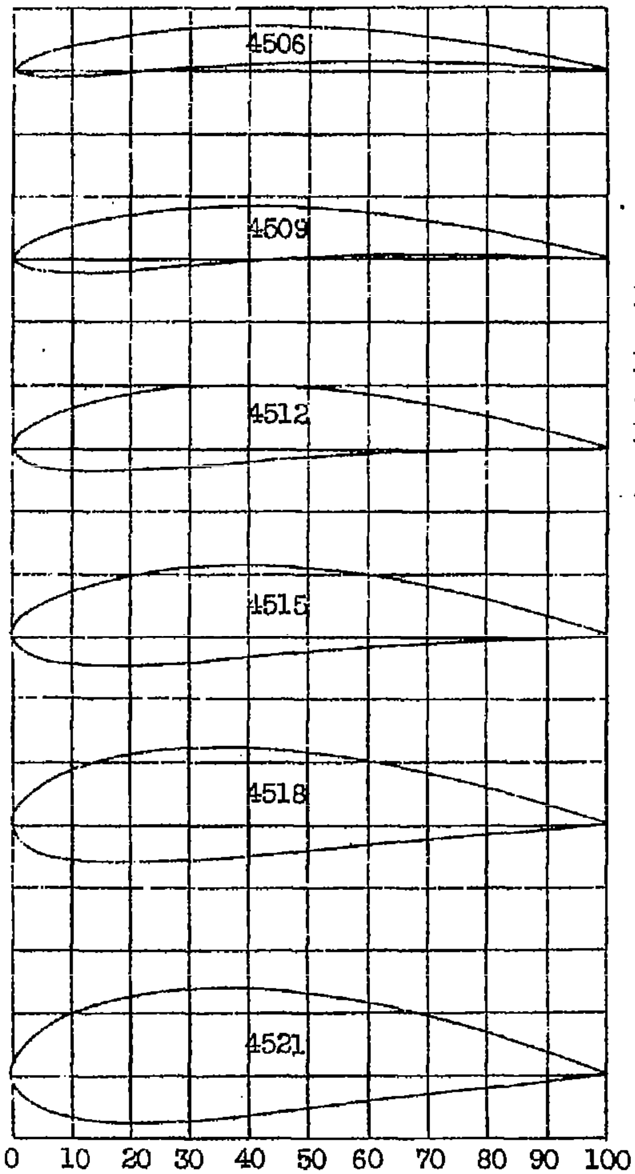
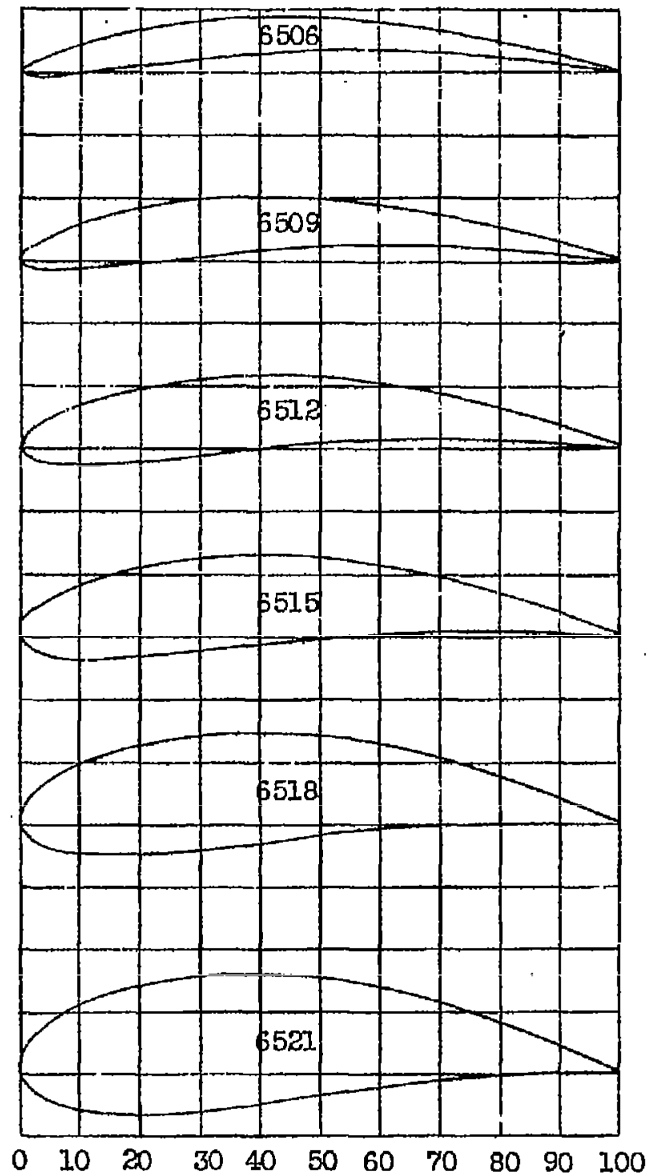


Fig.1

N.A.C.A. airfoil profiles. Series 45 and 65.



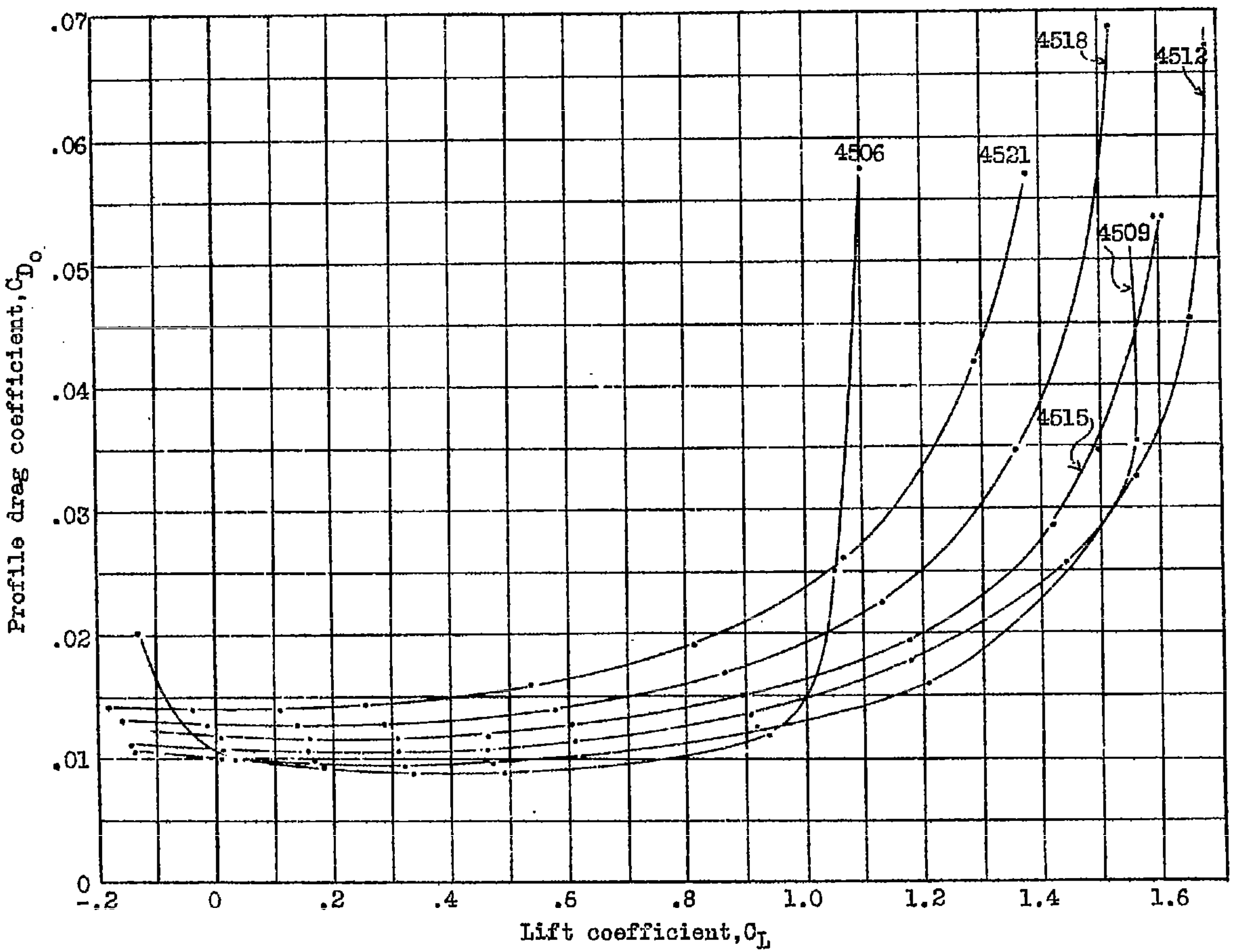


Fig.2 Profile drag curves for N.A.C.A.45 series airfoils.

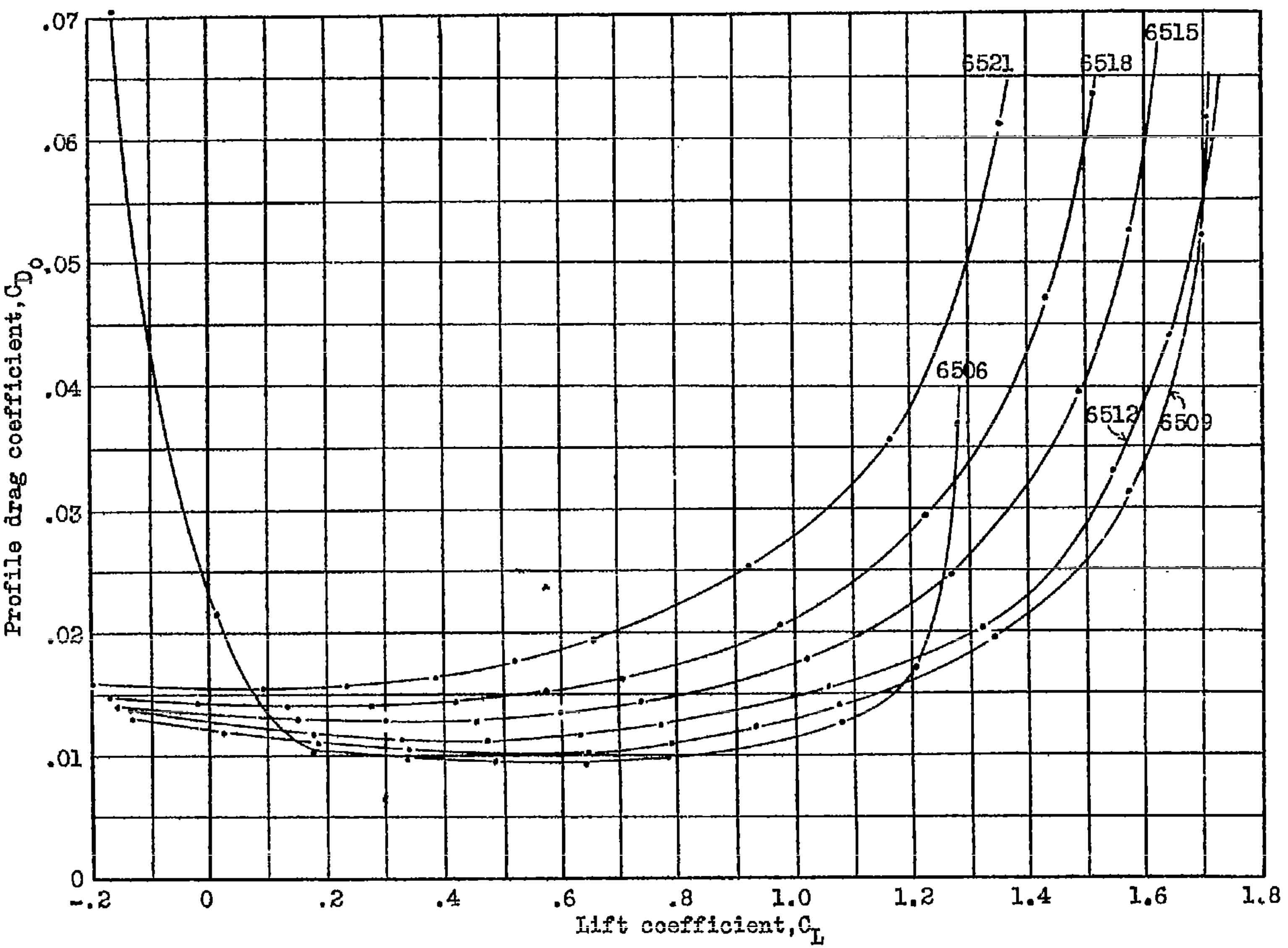


Fig.3 Profile drag curves for N.A.C.A.65 series airfoils.

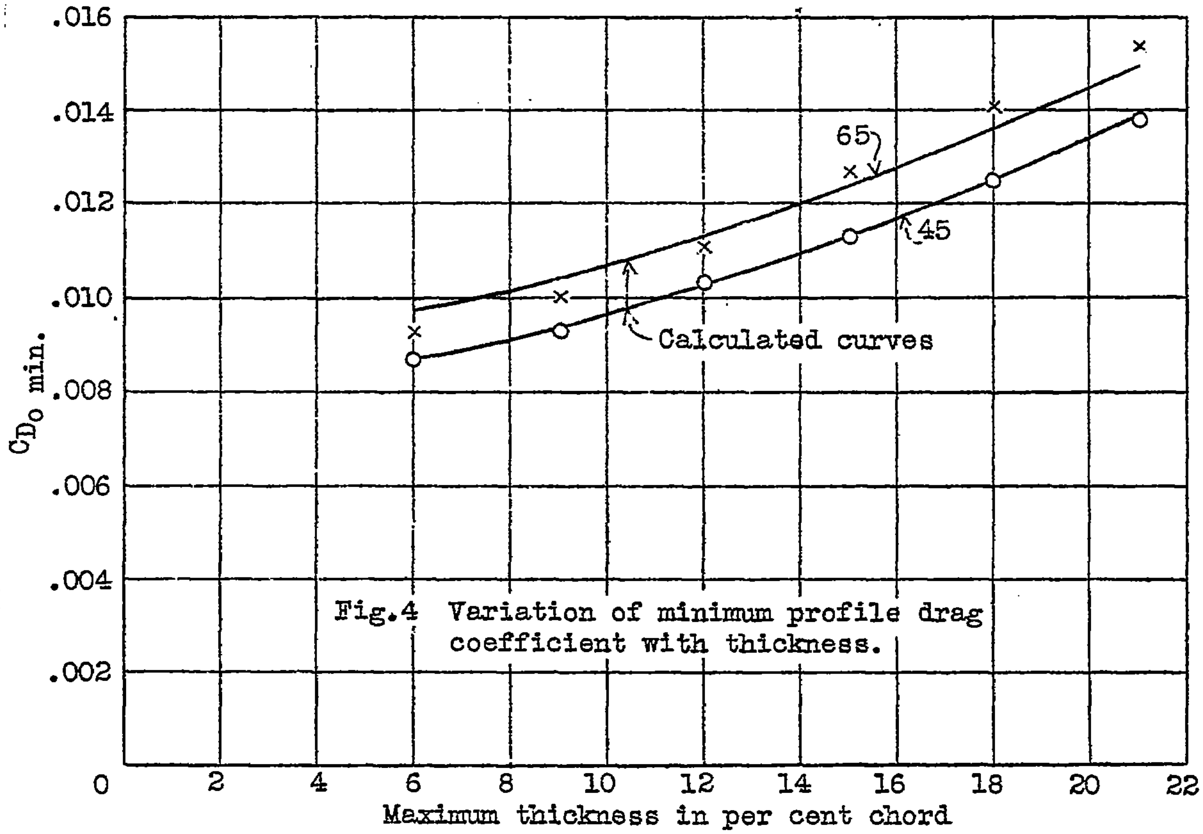


Fig.4 Variation of minimum profile drag coefficient with thickness.

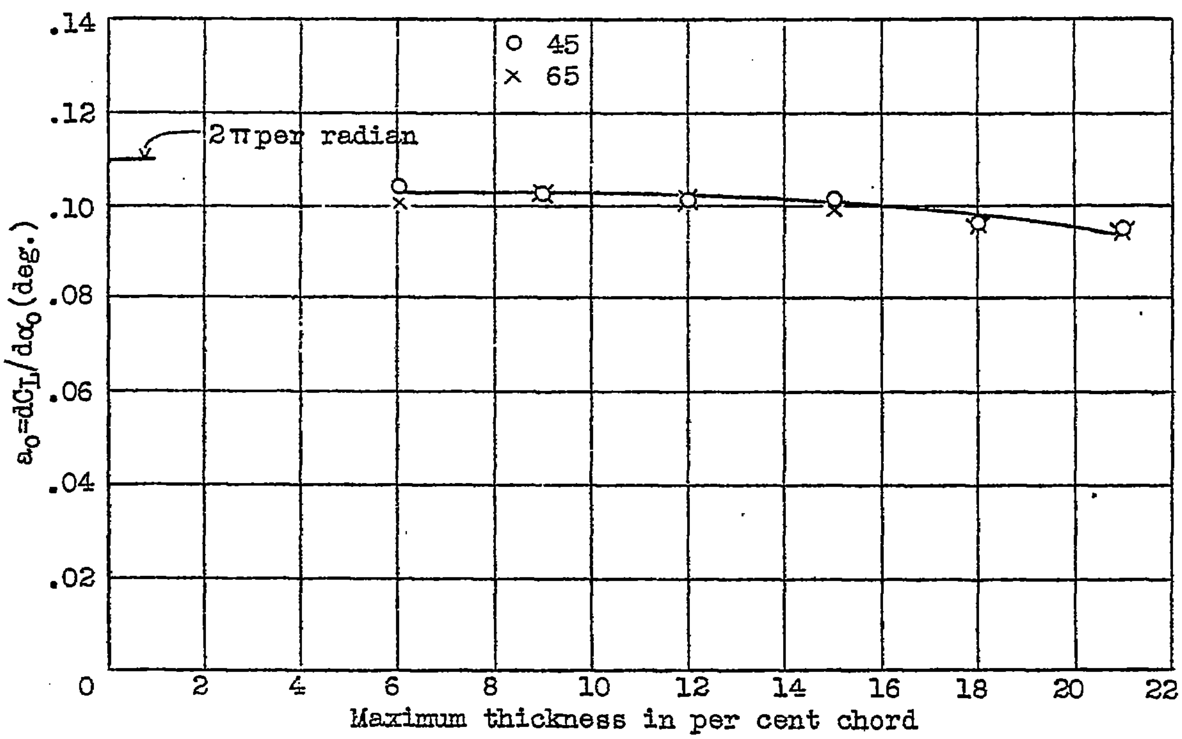


Fig.7 Variation of lift curve slope with thickness.

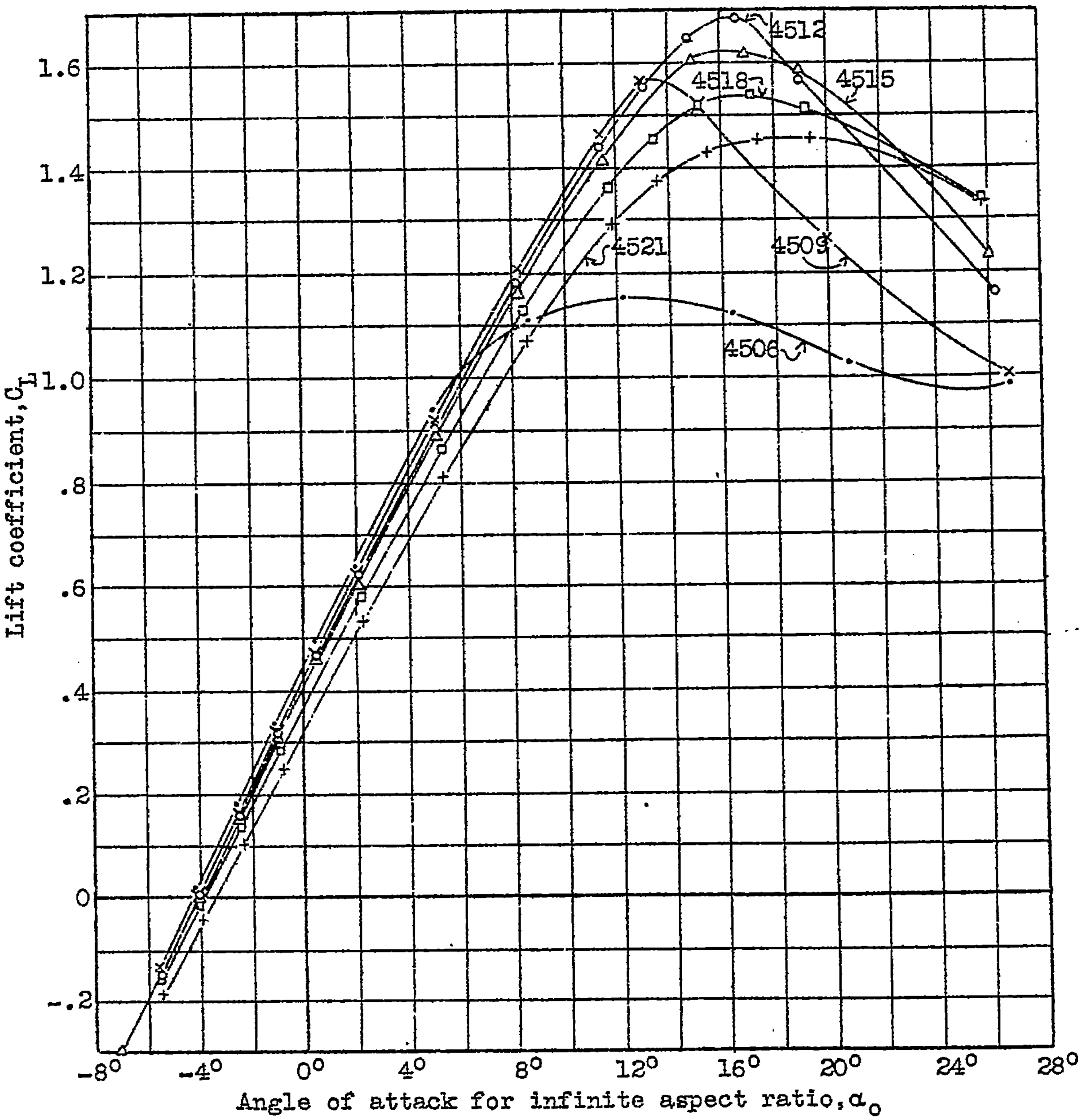


Fig.5 Lift curves for N.A.C.A.45 series airfoils.

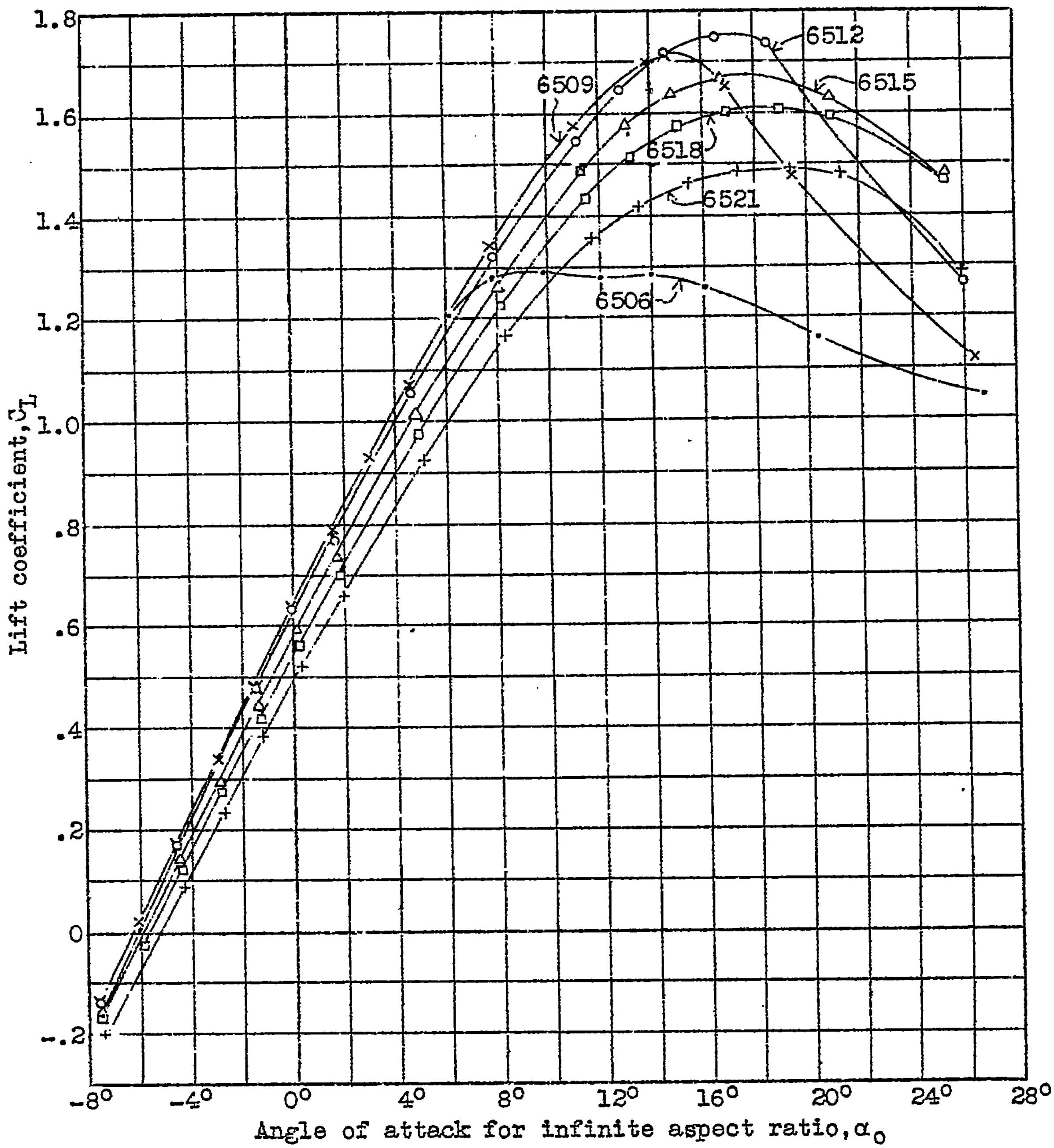


Fig.6 Lift curves for N.A.C.A.65 series airfoils.

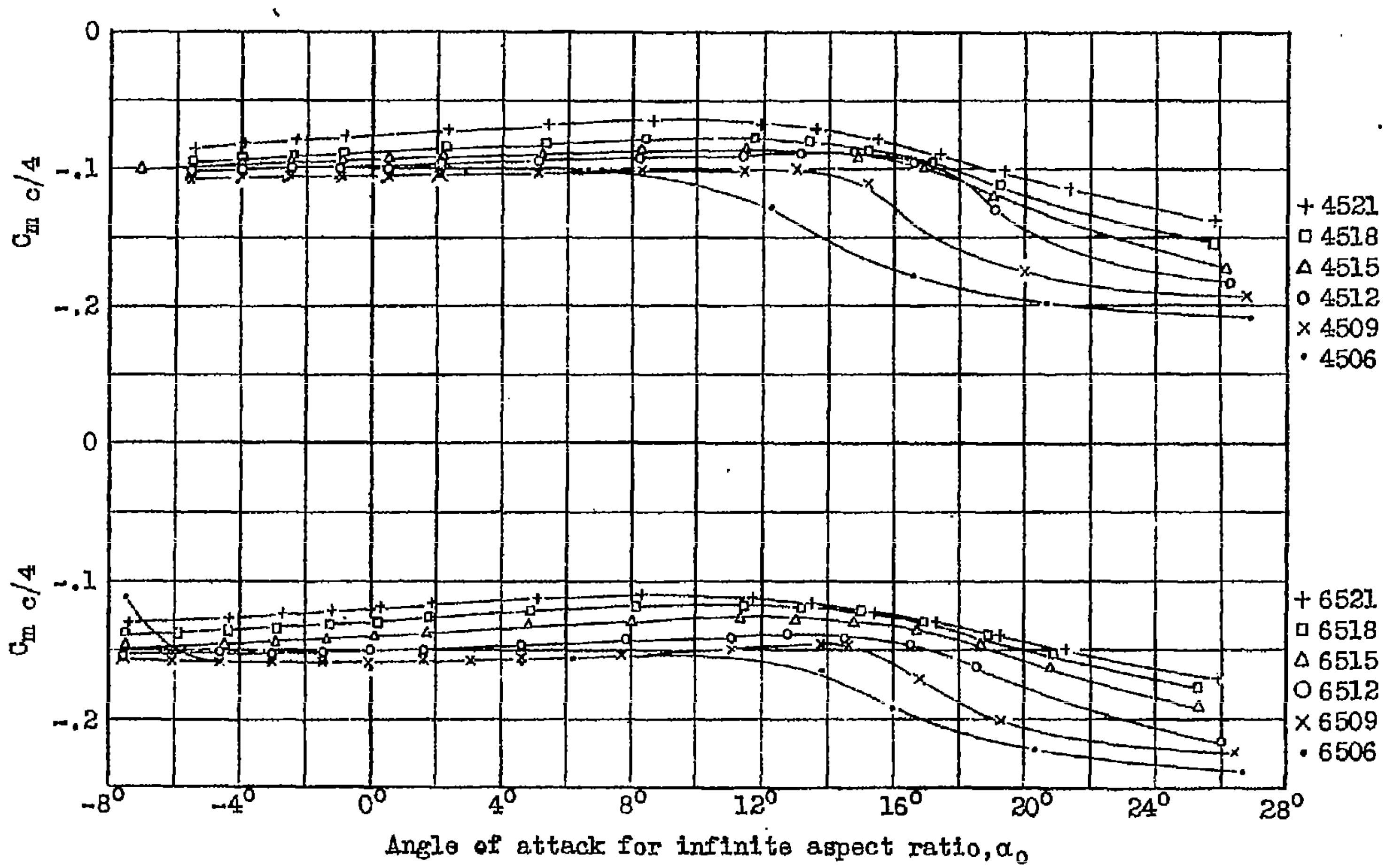


Fig.8 Moment coefficients about a point one-quarter of the chord behind the leading edge.

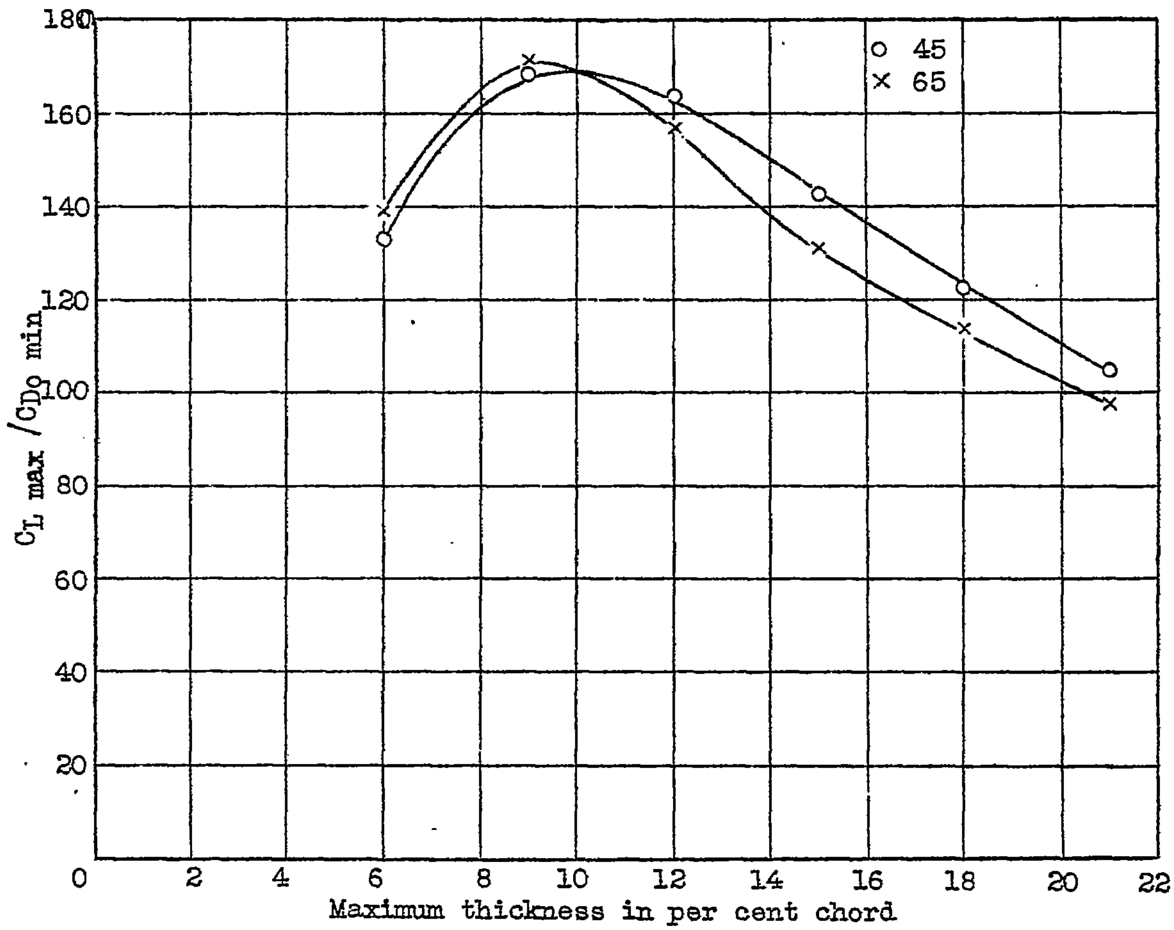


Fig.9 Ratio of maximum lift to minimum profile drag.

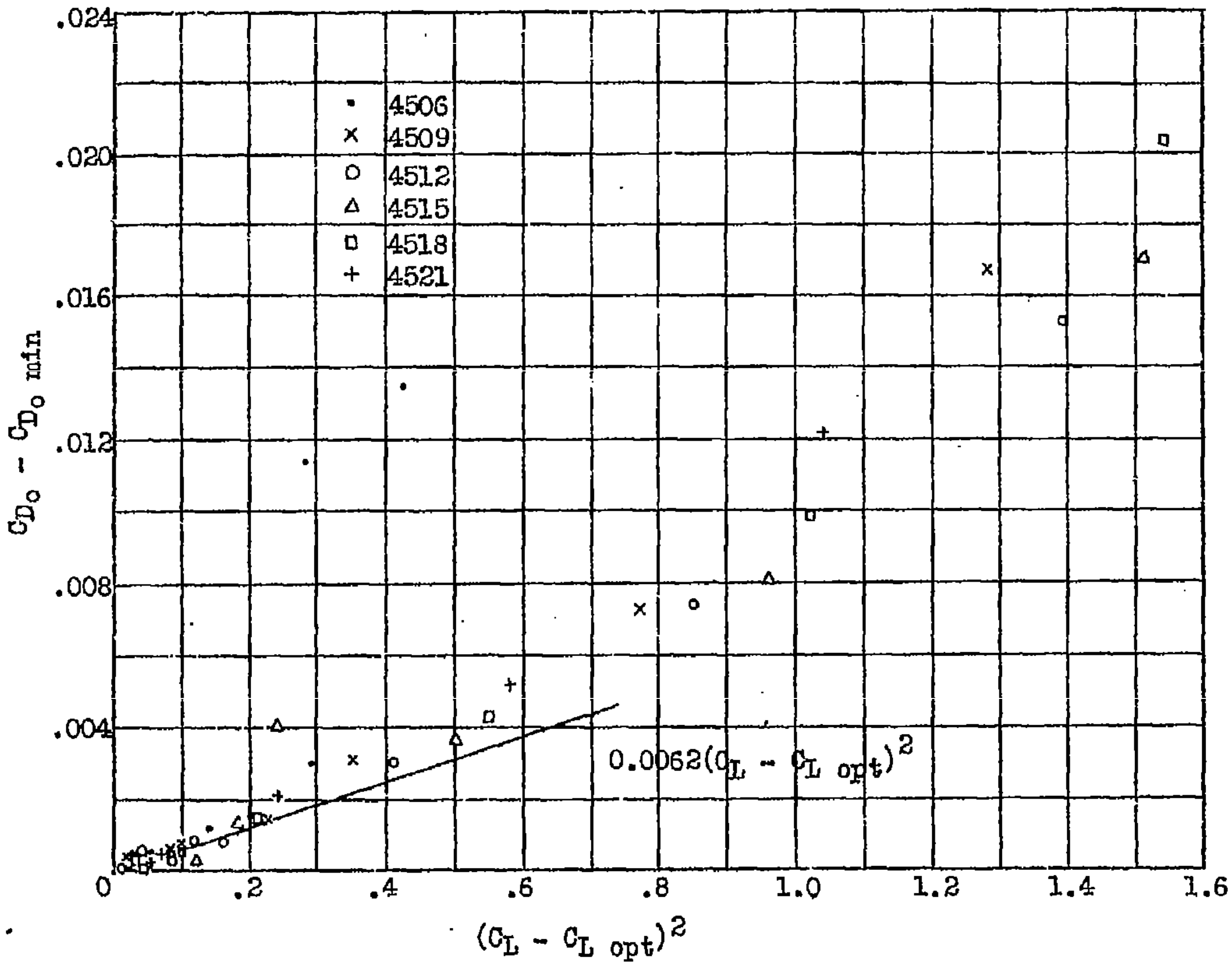


Fig.10 Increase of profile drag coefficient with lift.

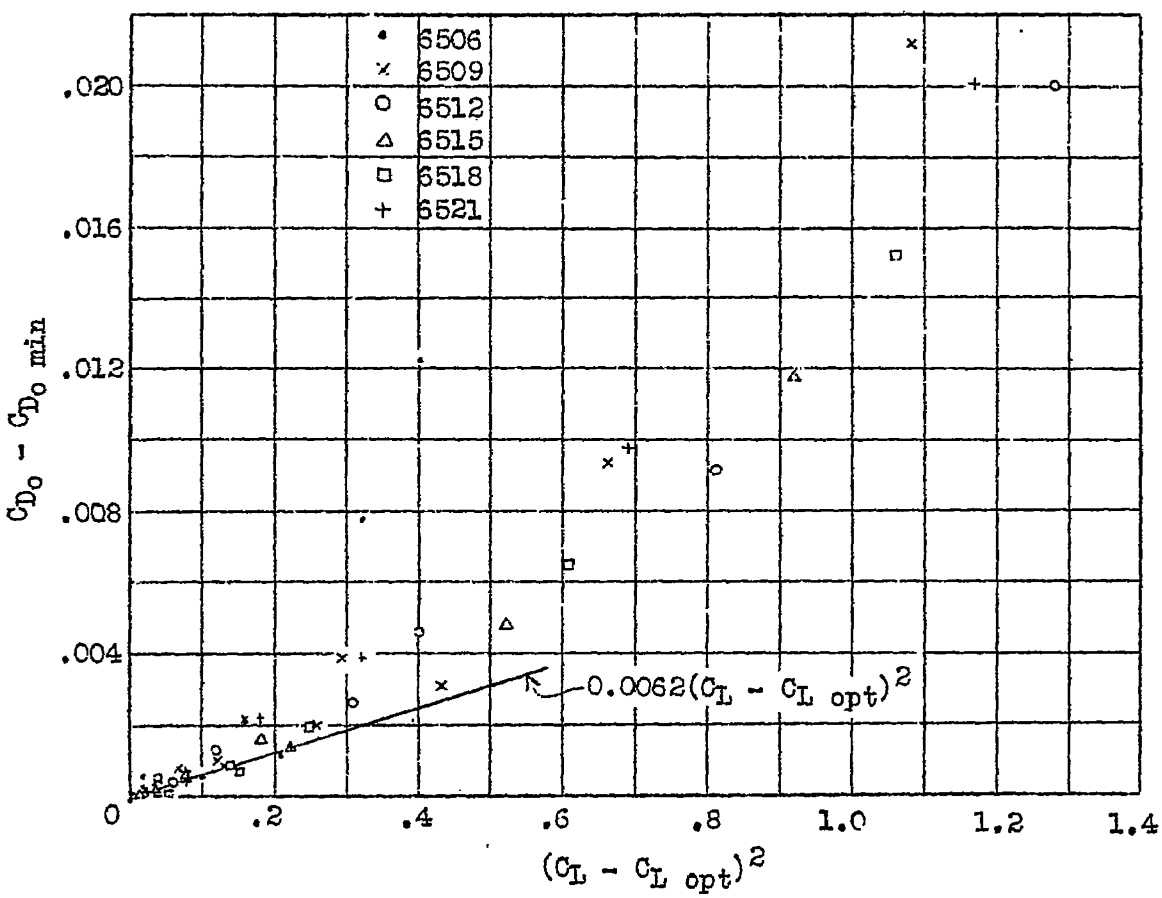


Fig.11 Increase of profile drag coefficient with lift.