Asst. Dir. Engr.
Vought-Sikorsky Aircraft Div
United Aircraft Corp.

[redacted]

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

SPECIAL REPORT No. 125

PRELIMINARY INVESTIGATION OF CERTAIN LAMINAR-FLOW AIRFOILS FOR APPLICATION AT HIGH SPEEDS AND REYNOLDS NUMBERS

By E. N. Jacobs, Ira H. Abbott, and A. E. von Doenhoff

August 1939

PRELIMINARY INVESTIGATION OF CERTAIN LAMINAR-FLOW AIRFOILS FOR APPLICATION AT HIGH SPEEDS AND REYNOLDS NUMBERS

By E. N. Jacobs, Ira H. Abbott, and A. E. von Doenhoff

SUMMARY

In order to extend the useful range of Reynolds Numbers of airfoils designed to take advantage of the extensive laminar boundary layers possible in an air stream of low turbulence, tests were made of the N.A.C.A. 2412-34 and 1412-34 sections in the N.A.C.A. low-turbulence tunnel. Although the possible extent of the laminar boundary layer on these airfoils is not so great as for specially designed laminar-flow airfoils, it is greater than that for conventional airfoils, and is sufficiently extensive so that at Reynolds Numbers above 11,000,000 the laminar region is expected to be limited by the permissible "Reynolds Number run" and not by laminar separation as is the case with conventional airfoils.

Drag measurements by the wake-survey method and pressure-distribution measurements were made at several lift coefficients throughout a range of Reynolds Numbers up to 11,400,000. The drag scale-effect curve for the N.A.C.A. 1412-34 is extrapolated to a Reynolds Number of 30,000,000 on the basis of theoretical calculations of the skin friction. Comparable skin-friction calculations were made for the N.A.C.A. 23012.

The results indicate that, for certain applications at moderate values of the Reynolds Number, the N.A.C.A. 1412-34 and 2412-34 airfoils offer some advantages over such conventional airfoils as the N.A.C.A. 23012. The possibility of maintaining a more extensive laminar boundary layer on these airfoils should result in a small drag reduction, and the absence of pressure peaks allows higher speeds to be reached before the compressibility burble is encountered. At lower Reynolds Numbers, below about 10,000,000, these airfoils have higher drags than airfoils designed to operate with very extensive laminar boundary layers.

INTRODUCTION

The realization of very low drag coefficients for airfoils designed to take advantage of the unusually extensive laminar boundary layers that may be maintained in the N.A.C.A. low-turbulence tunnel (reference 1) has opened up a new field of airfoil research. These laminar-flow airfoils have been designed to have falling pressures in the downstream direction over a considerable portion of both upper and lower surfaces, thus providing favorable conditions for the maintenance of the laminar boundary layers. These airfoils have very low drag coefficients in the low Reynolds Number range from about 3,000,000 to 6,000,000. At higher Reynolds Numbers, the drag coefficients increase sharply, and the airfoils rapidly lose their advantage over conventional airfoils.

The attempt to obtain drag reductions at higher values of the Reynolds Number is an obvious extension of this work. As the Reynolds Number is increased, the already obtained values of the "Reynolds Number run" for the laminar boundary layer will provide a laminar boundary layer over only a decreasing portion of the airfoil surface. It thus appears, in the light of present knowledge, that the boundary layer will be largely turbulent at high Reynolds Numbers, and the drag reductions obtainable through use of already realized values of the Reynolds Number run for the laminar boundary layer will be correspondingly small. Nevertheless, such reductions should be realized pending a more satisfactory solution of this problem.

These considerations indicated the need for tests of airfoils designed to work with extensive turbulent boundary layers and still permit gains to be obtained from more than usually extensive laminar boundary layers. Conventional airfoils meet the requirement as to the turbulent boundary layers since such airfoils have been designed to provide good pressure recovery with extensive turbulent boundary layers. Most such airfoils, however, have only a very short length of favorable pressure gradient near the leading edge, especially when lifting, and are obviously unsuited for this work because the laminar boundary layer is limited to a very small region by laminar separation.

The foregoing requirements are met by the N.A.C.A. 2412-34 airfoil which has a favorable pressure gradient of moderate length when operating at its ideal lift coefficient. Previous tests of this airfoil (reference 2) were limited to low values of the Reynolds Number. The results were complicated by the presence of then unknown tunnel-wall and end effects which made the published drag coefficients too high. This airfoil was, therefore, selected for test together with a modification, the N.A.C.A. 1412-34 airfoil, having a lower ideal lift coefficient.

These airfoils were tested in the low-turbulence tunnel and, for comparison, in the N.A.C.A. variable-density tunnel. Although Reynolds Numbers higher than about 11,000,000 were not obtainable, it was hoped that the test results would indicate the value of the airfoils at higher Reynolds Numbers. The test results and comparable skin-friction computations made for the N.A.C.A. 1412-34 and 23012 airfoils indicate that in a moderate range of Reynolds Numbers, say about 20,000,000 to 30,000,000, the N.A.C.A. 1412-34 and 2412-34 airfoils should have slightly lower drag coefficients than airfoil sections now in use.

APPARATUS AND TESTS

The N.A.C.A. low-turbulence tunnel has a high, narrow test section (reference 1) and the models extend from wall to wall providing two-dimensional flow (fig. 1). The models used were of 3-foot span and 7.5-foot chord. They were made of wood and were carefully faired and finished with lacquer which was finally rubbed with No. 400 water-cloth in the direction of the air flow until the surface was smooth. The models were not constructed to the ordinates of the airfoils they were to represent, but were made with reduced thickness and camber to compensate approximately for some of the tunnel-wall effects.

Drag measurements were made by the wake-survey method using a survey rake of total-head tubes connected to an integrating manometer as in reference 1. The drag results presented differ from those of reference 1 because they have been tentatively corrected by a method that gives results nearly equivalent to the Jones method (reference 3). A small correction has also been applied to correct the results for the displaced effective center of the total-head tubes in the wake. Although these corrections are probably only approximately applicable to the test conditions, they are not very large and their application probably results in improved data. It is thought that the data may be directly applied with normal engineering accuracy. Boundary-layer measurements and pressure-distribution points for use in computing the lift coefficients were obtained by means of a "mouse" (reference 1) similar to that used by Jones (reference 4). Tests were made over a range of lift coefficients from -0.06 to +0.89 for the N.A.C.A. 1412-34 airfoil and from 0.03 to 0.56 for the N.A.C.A. 2412-34 airfoil. The Reynolds Number range was from about 4,000,000 to 11,000,000.

The two airfoils were also tested in the usual manner in the N.A.C.A. variable-density tunnel. These results have been fully corrected as described in reference 5.

RESULTS AND DISCUSSION

Lift

The lift coefficient for each test condition was computed from measured pressures at the 15-percent chord point and known values of the basic and additional normal force distributions (reference 6). For each lift, complete pressure distributions were computed using the methods of references 6 and 7. These theoretical pressure distributions are plotted in figures 2 and 3, together with the experimental points. Although there is some slight systematic variation between the theory and experimental data, the agreement is considered satisfactory.

This agreement justifies the method used to correct approximately for some of the tunnel-wall effects by constructing the models thinner and with less camber than the airfoils they were to represent. The object of this procedure was to obtain the same pressure distributions, and accordingly the same flow conditions, in the tunnel on the surfaces of the modified model as would be obtained in free air on the airfoil section. The extent to which this object was realized may be judged from figures 2 and 3. It is believed that the discrepancies are too small to be significant and that the test data may be applied directly at the test lift coefficient.

Drag

The drag results for the two airfoils are presented in figures 4 and 5 where the profile-drag coefficients are plotted against Reynolds Number for several lift coefficients. Minimum drag coefficients are obtained at lift coefficients near or slightly higher than the ideal lift coefficients which are 0.13 and 0.26, respectively, for the N.A.C.A. 1412-34 amd 2412-34 airfoils. The variation of the profile-drag coefficients with lift coefficient is shown for two Reynolds Numbers in figures 6 and 7 which also present the test results from the variable-density tunnel for comparison. The drag results from the low-turbulence tunnel are much lower than those from the variable-density tunnel at the lower lift coefficients as would be expected from the much more extensive laminar boundary layers possible at these lift coefficients in a low-turbulence air stream. For each airfoil, however, at the highest lift coefficient at which drag tests were made in the low-turbulence tunnel, the results from the two tunnels are in fairly good agreement. Figures 2 and 3 show that, at these lift coefficients, pressure peaks have appeared on the upper surfaces of both airfoils. These peaks would cause laminar separation to occur very close to the leading edge and thus prevent extensive laminar boundary layers from existing on these surfaces.

Figure 8 provides a comparison between the drags of the N.A.C.A. 1412-34 and 2412-34 airfoils and several laminar-flow airfoils selected from reference 1. Similar corrections have been applied to all the data. The results taken from reference 1, however, were obtained on models which were not reduced in thickness and camber, and these results, accordingly, are more nearly applicable to somewhat thicker and more highly cambered airfoils than are indicated by the airfoil numbers. It will be noticed that the N.A.C.A. 1412-34 and 2412-34 airfoils are much inferior to the laminar-flow airfoils at the Reynolds Numbers at which the laminar-flow airfoils operate to advantage. This result is at least partly explained by the less extensive laminar boundary layers on the N.A.C.A. 1412-34 and 2412-34 airfoils as shown by a comparison of the transition points of figures 9 and 10 with those of corresponding figures of reference 1.

Extrapolation to Higher Reynolds Numbers

With the exception of the N.A.C.A. 27-215 airfoil with a 0.5c tail extension, the slopes of the drag curves for the laminar-flow airfoils, as plotted against Reynolds Number in figure 8, are definitely higher at the upper end of the test range than for the N.A.C.A. 1412-34 and 2412-34 airfoils. It is, therefore, expected that at higher values of the Reynolds Number the N.A.C.A. 1412-34 and 2412-34 airfoils would be superior to the laminar-flow airfoils. The N.A.C.A. 27-215 airfoil with 0.5c tail extension was designed (reference 1) for use at Reynolds Numbers somewhat above the optimum for the laminar-flow airfoils. At these Reynolds Numbers transition occurs in a region of strong pressure recovery. At higher Reynolds Numbers transition is expected to move forward to a region of decreasing pressure. Under these circumstances, excessively high skin frictions for the fresh turbulent boundary layer are expected to occur, and the flow conditions are similar to those for the N.A.C.A. 27-215 airfoil at Reynolds Numbers above its optimum (reference 1) where the drag coefficient increases rapidly with Reynolds Number. A similar, but less drastic increase, is expected for the airfoil with the tail extension. Accordingly the use of the N.A.C.A. 27-215 airfoil with 0.5c tail extension at Reynolds Numbers appreciably above the test range cannot be recommended in the absence of tests.

At the end of the test range the scale effect on the drag coefficients of the 1412-34 and 2412-34 airfoils is unfavorable. Convential airfoils, as usually tested, show favorable scale effects in this Reynolds Number range. It, accordingly, appears that any attempt to extrapolate these results should be guided by considerations of the details of the boundary-layer flow. Accordingly, the skin friction of the N.A.C.A. 1412-34 airfoil was calculated theoretically for a range of Reynolds Numbers from 12,000,000 to 30,000,000. For comparison, the skin friction of the N.A.C.A. 23012 was also calculated from similar assumptions.

The skin friction was computed as the sum of the laminar and turbulent skin friction along both upper and lower surfaces of the airfoils. The pressure distribution for the N.A.C.A. 1412-34 was taken as that for $c_l = 0.17$ (fig. 7). The calculations for the N.A.C.A. 23012 were carried out with the theoretical pressure distribution for the ideal angle of attack, corresponding to $c_l = 0.383$. The thickness of the laminar boundary layer and the corresponding skin friction were found from equation 1, reference 8. Transition was assumed to occur when the value of the laminar boundary-layer Reynolds Number, R_δ, reached 5,000,

where R_δ is $U\delta/\nu$.

U, the velocity just outside the boundary layer.

δ, the distance from the surface to the point where the boundary-layer velocity is equal to 0.707 the outside velocity.

ν, the kinematic viscosity.

The critical value of R_δ was found, experimentally, to be 5360 at 40 percent of the chord aft of the leading edge on the upper surface of the N.A.C.A. 1412-34.

The turbulent boundary layer was assumed to start at the transition point with the same momentum defect as that of the laminar at the same point. The shape of the turbulent boundary layer was assumed to follow the one-seventh power law. The turbulent skin friction was then found from integration of the von Kármán momentum relation (reference 9).

The results of these computations are given in figure 11. At a Reynolds Number of 12,000,000, the difference between the computed skin friction and the measured drag of the N.A.C.A. 1412-34 is approximately 0.00135. The indicated extrapolation of the drag of this airfoil is based on the assumption that this difference, which is probably the pressure drag, remains constant at somewhat higher Reynolds Numbers.

Comparison of the calculated drags for the N.A.C.A. 23012 and 1412-34 airfoils indicates that the drag of the 1412-34 should be about 5 percent less than that of the 23012 in the Reynolds Number range from 20,000,000 to 30,000,000. Although direct extrapolation of variable-density-tunnel drag results indicates that the drag of the N.A.C.A. 23012 may be slightly lower than that of the N.A.C.A. 1412-34 in this range of Reynolds Numbers, it is felt that the skin-friction calculations give a more reliable estimate of the relative drag of the two airfoils.

At any rate, it appears that the drag difference between the N.A.C.A. 1412-34 and 23012 airfoils will be small at Reynolds Numbers above about 20,000,000. If the drag of the airfoil section is the primary consideration, the N.A.C.A. 1412-34 should probably be selected since this airfoil does allow a possible drag reduction from the existence of a more extensive laminar boundary layer. Moreover, there is always the possibility of more extensive laminar boundary layers being obtained in flight than in the present tests. For high-speed applications, the N.A.C.A. 1412-34 and 2412-34 airfoils have the additional advantage of higher compressibility-burble speeds than conventional airfoils because of the absence of pressure peaks. For instance, the theoretical values of M_c (the ratio of the critical speed to the speed of sound, reference 10) for the N.A.C.A. 1412-34 and 2412-34 airfoils are 0.74 and 0.70, respectively, at the ideal lift coefficients as compared with 0.61 for the N.A.C.A. 23012 airfoil.

The maximum lift coefficients for the N.A.C.A. 1412-34 and 2412-34 airfoils ($c_{l_{max}}$ = 1.12 and 1.22, respectively) are much lower than for airfoils such as the N.A.C.A. 23012 ($c_{l_{max}}$ = 1.74). In cases where the maximum lift coefficient is important, the reduced maximum lift coefficients for these sections will severely limit their application. On the other hand, the advantage of the N.A.C.A. 23012 airfoil in this respect is not as great as would appear because the lift curve for this airfoil breaks sharply from its maximum to a value of about 1.32. The extent to which values of the lift coefficient for this airfoil higher than 1.32 can be used with safety is doubtful.

REASON WHY JAKE DISLIKES 230 SERIES?

CONCLUDING REMARKS

For certain applications at moderate values of the Reynolds Number, the N.A.C.A. 1412-34 and 2412-34 airfoils offer some advantages over such conventional airfoils as the N.A.C.A. 23012. The possibility of maintaining a more extensive laminar boundary layer on these airfoils should result in a small drag reduction and the absence of pressure peaks allows higher speeds to be reached before the compressibility burble is encountered. At lower Reynolds Numbers below about 10,000,000 these airfoils have higher drags than airfoils designed to operate with very extensive laminar boundary layers.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., July 6, 1939.

REFERENCES

1. Jacobs, Eastman N.: Preliminary Report on Laminar-Flow Airfoils and New Methods Adopted for Airfoil and Boundary Layer Investigation. N.A.C.A. confidential report, June 1939.

2. Stack, John, and von Doenhoff, Albert E.: Tests of 16 Related Airfoils at High Speeds. T.R. No. 492, N.A.C.A., 1934.

3. The Cambridge University Aeronautics Laboratory. The Measurement of Profile Drag by the Pitot-Traverse Method. R. & M. No. 1688, British A.R.C., 1936.

4. Jones, B. Melvill: Flight Experiments on the Boundary Layer. Jour. Aero. Sci., vol. 5, no. 3, Jan. 1938, pp. 81-94.

5. Jacobs, Eastman N., and Abbott, Ira H.: Airfoil Section Data Obtained in the N.A.C.A. Variable-Density Tunnel as Affected by Support Interference and Other Corrections. T.R. No. 669, N.A.C.A., 1939.

6. Jacobs, Eastman N., and Rhode, R. V.: Airfoil Section Characteristics as Applied to the Prediction of Air Forces and Their Distribution on Wings. T.R. No. 631, N.A.C.A., 1938.

7. Allen, H. Julian: A Simplified Method for the Calculation of Airfoil Pressure Distribution. T.N. No. 708, N.A.C.A., 1939.

8. von Doenhoff, Albert E.: A Method of Rapidly Estimating the Position of the Laminar Separation Point. T.N. No. 671, N.A.C.A., 1938.

9. Dryden, H. L., and Kuethe, A. M.: Effect of Turbulence in Wind Tunnel Measurements. T.R. No. 342, N.A.C.A., 1930.

10. Jacobs, Eastman N.: Methods Employed in America for the Experimental Investigation of Aerodynamic Phenomena at High Speeds. Misc. Paper No. 42, N.A.C.A., 1936.

Figure 1.- Airfoil set-up in the low-turbulence tunnel.

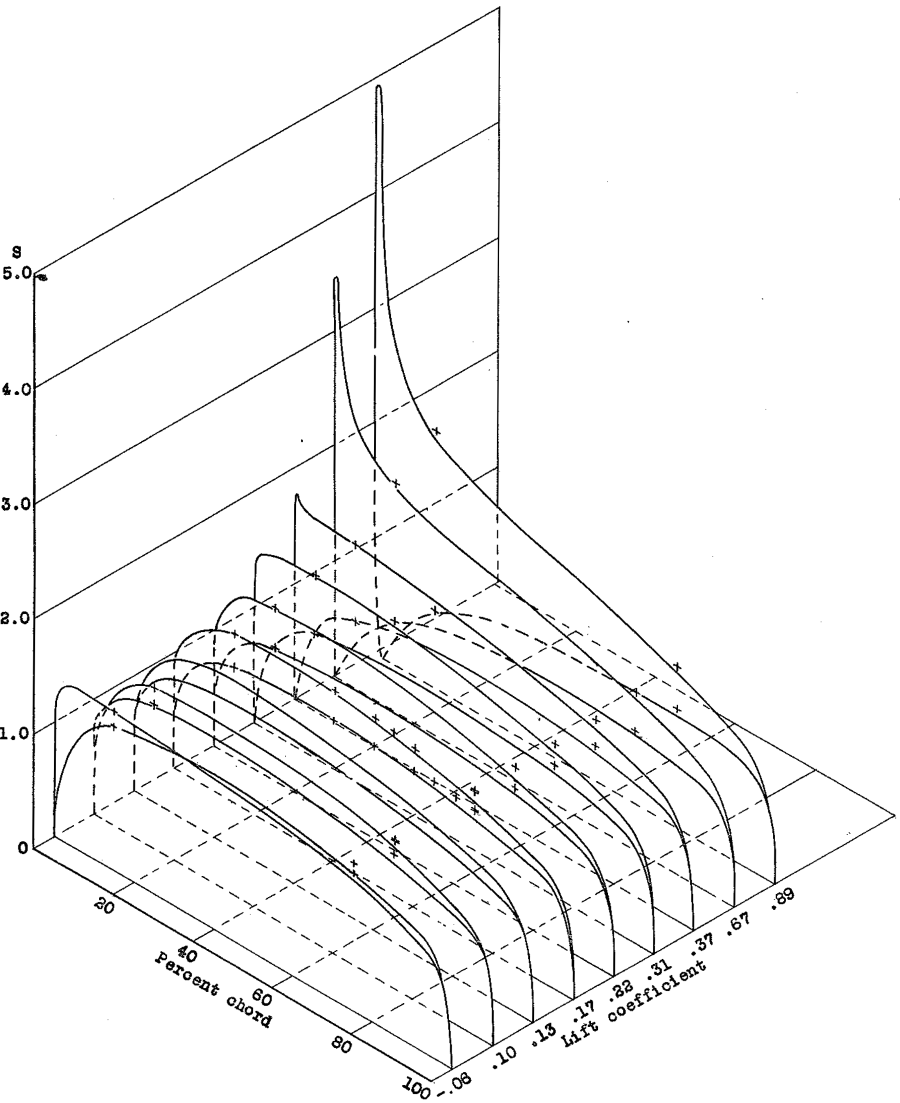


Figure 2.- Theoretical pressure distributions and experimental points for N.A.C.A. 1412-34 airfoil.

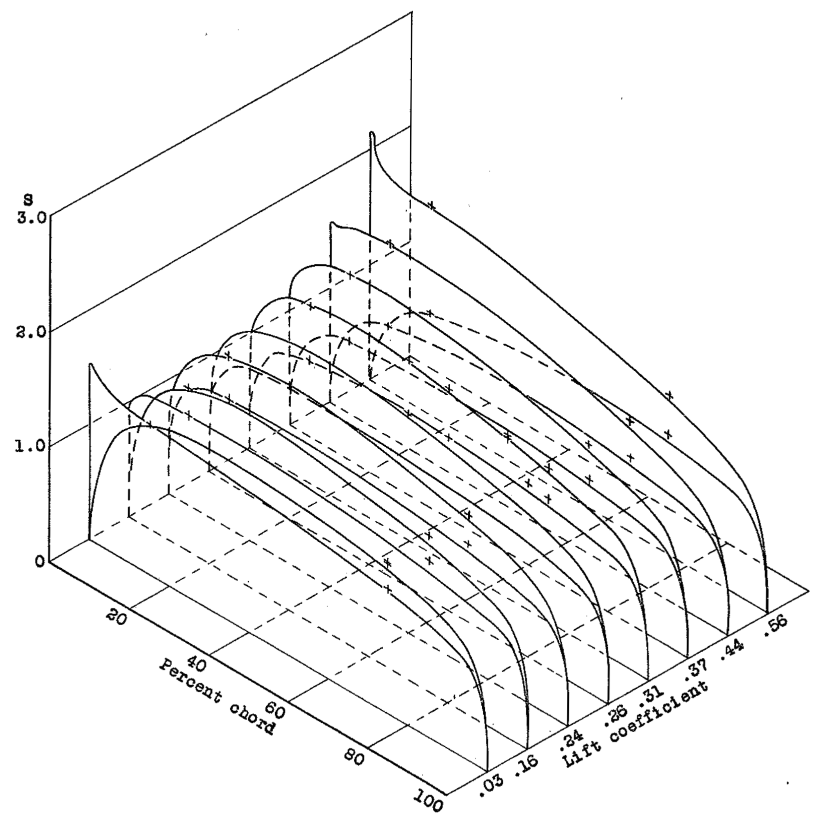


Figure 3.- Theoretical pressure distributions and experimental points for N.A.C.A. 2412-34 airfoil.

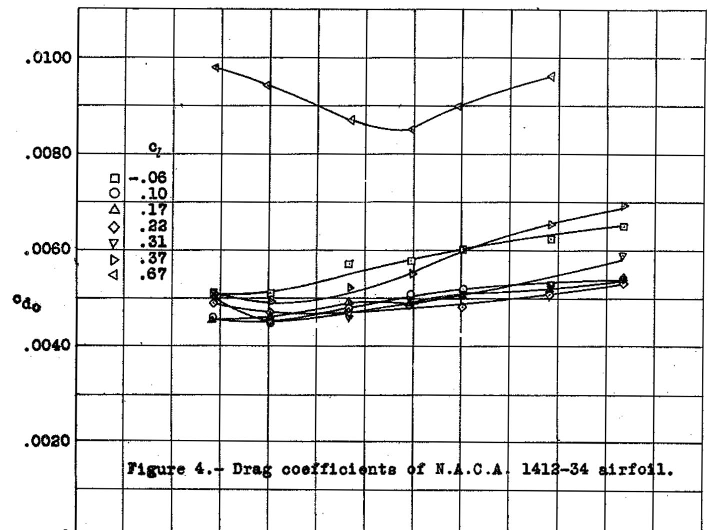


Figure 4.- Drag coefficients of N.A.C.A. 1412-34 airfoil.

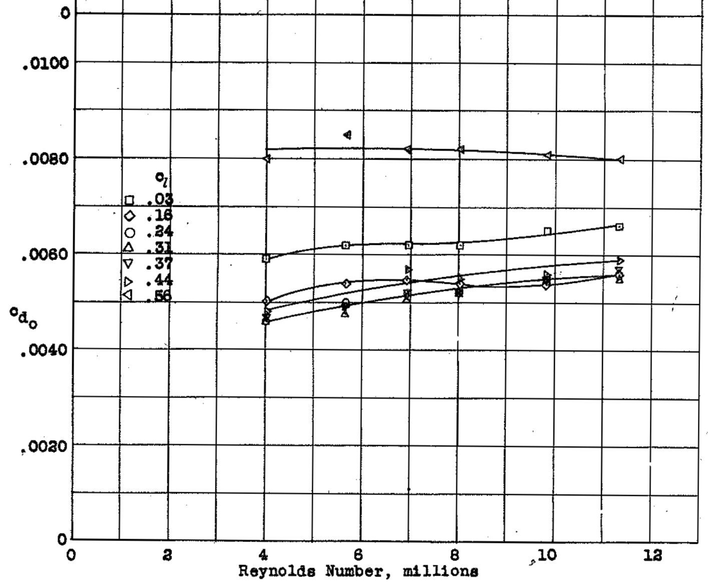


Figure 5.- Drag coefficients of N.A.C.A. 2412-34 airfoil.

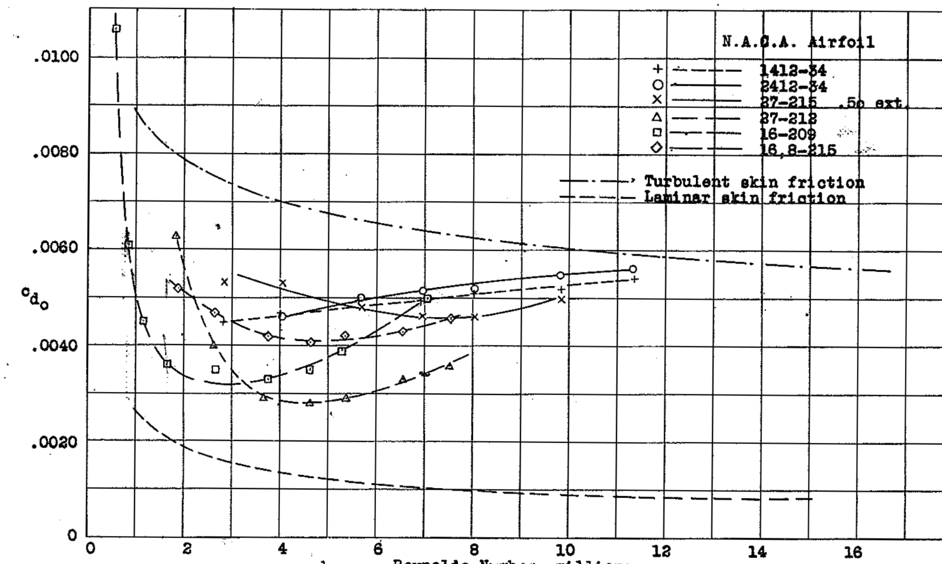


Figure 8.- Drag coefficients near ideal lift coefficients for several airfoils.

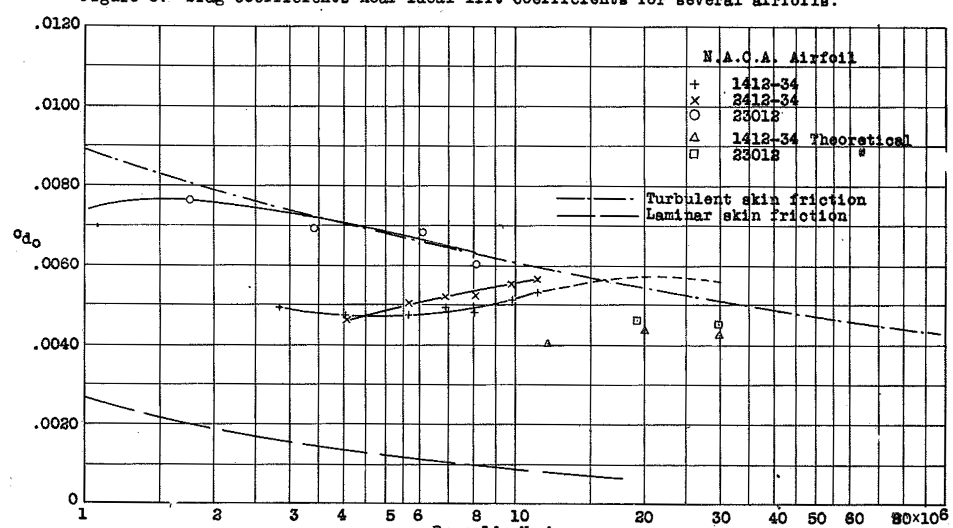
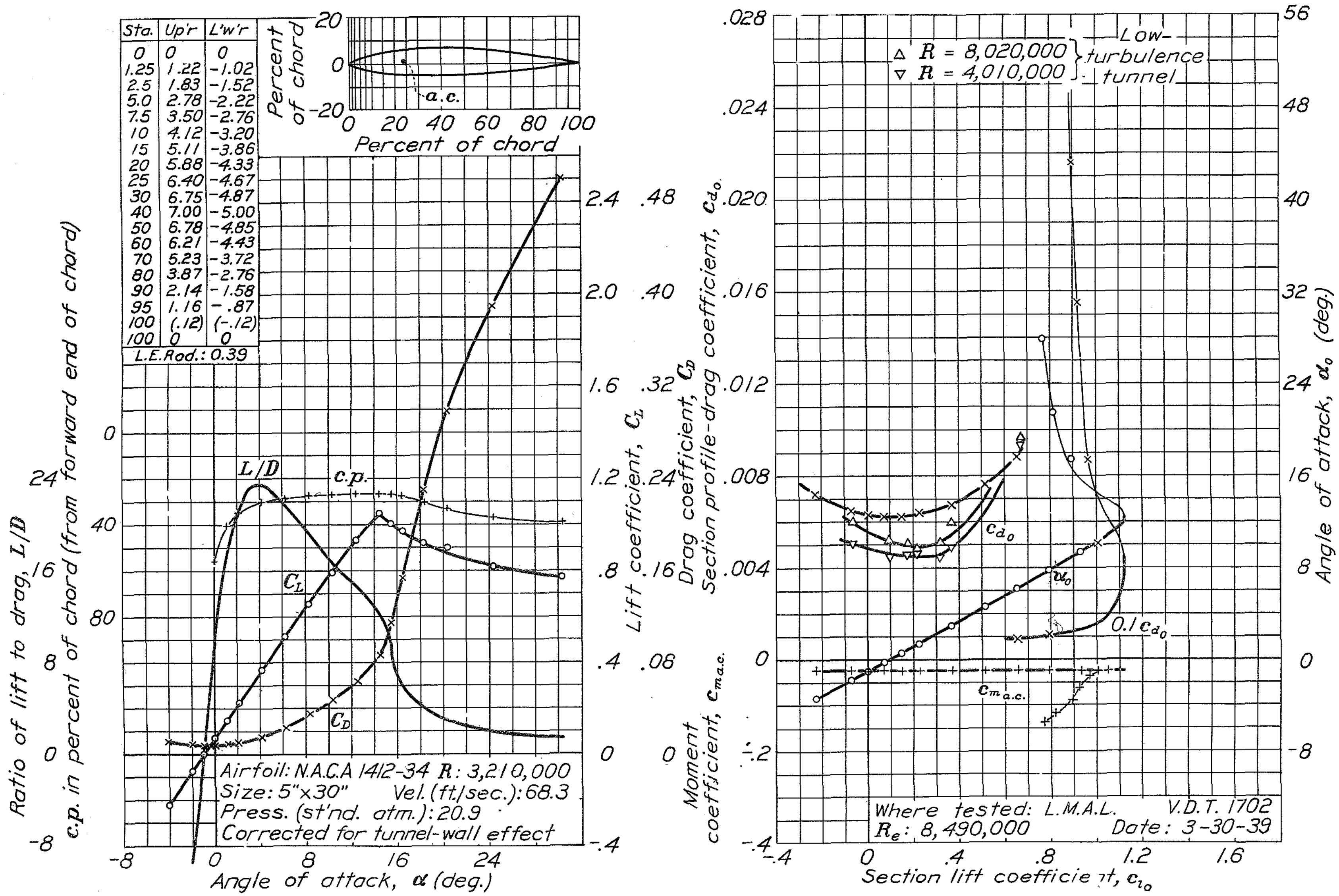


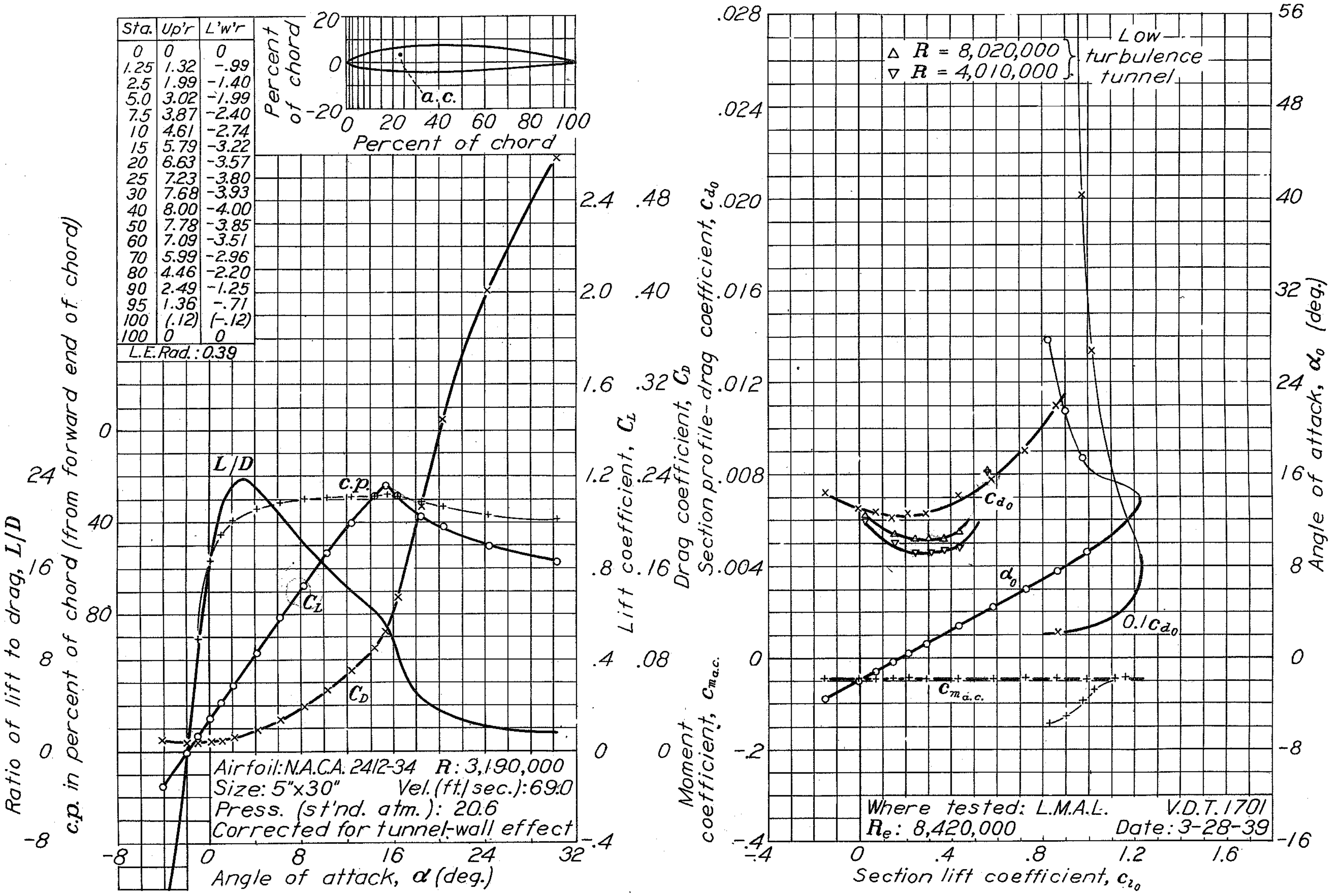
Figure 11.- Scale effect on drag coefficients.



Sta.	Up'r	L'w'r
0	0	0
1.25	1.22	-1.02
2.5	1.83	-1.52
5.0	2.78	-2.22
7.5	3.50	-2.76
10	4.12	-3.20
15	5.11	-3.86
20	5.88	-4.33
25	6.40	-4.67
30	6.75	-4.87
40	7.00	-5.00
50	6.78	-4.85
60	6.21	-4.43
70	5.23	-3.72
80	3.87	-2.76
90	2.14	-1.58
95	1.16	-.87
100	(.12)	(-.12)
100	0	0

L.E. Rad.: 0.39

Figure 6.- N.A.C.A. 1412-34 airfoil.



Sta.	Up'r	L'w'r
0	0	0
1.25	1.32	-.99
2.5	1.99	-1.40
5.0	3.02	-1.99
7.5	3.87	-2.40
10	4.61	-2.74
15	5.79	-3.22
20	6.63	-3.57
25	7.23	-3.80
30	7.68	-3.93
40	8.00	-4.00
50	7.78	-3.85
60	7.09	-3.51
70	5.99	-2.96
80	4.46	-2.20
90	2.49	-1.25
95	1.36	-.71
100	(.12)	(-.12)
100	0	0

L.E. Rad.: 0.39

Figure 7.- N.A.C.A. 2412-34 airfoil.

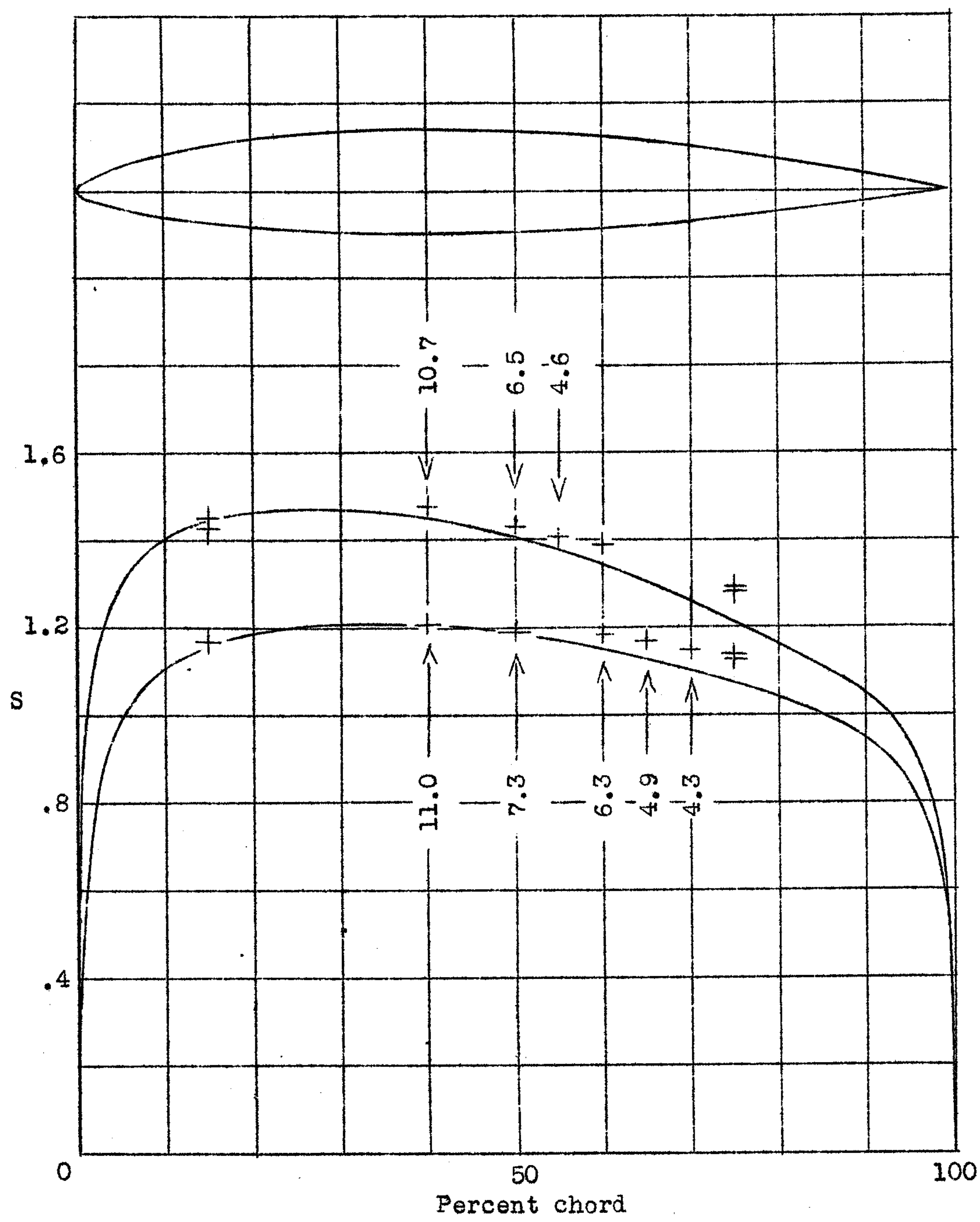


Figure 9.- N.A.C.A. 1412-34 airfoil. c_l = 0.17 Arrows indicate location of transition corresponding to the Reynolds Number in millions. M_c = 0.73.

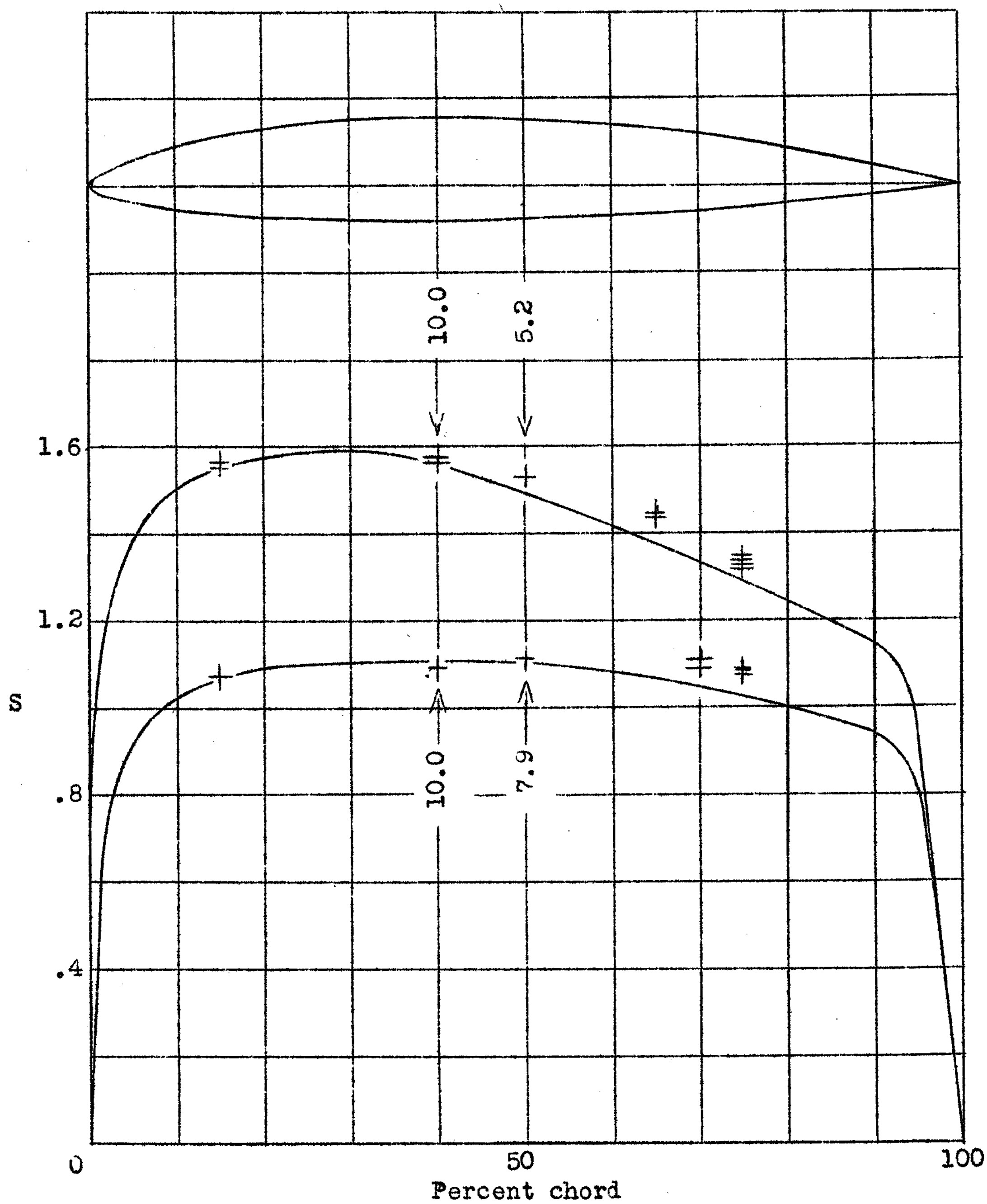


Figure 10.- N.A.C.A. 2412-34 airfoil. c_l = 0.31 Arrows indicate location of transition corresponding to the Reynolds Number in millions. M_c = 0.69.