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WIND-TUNNEL TESTS OF THE N.A.C.A. 45-125 AIRFOIL

A THICK AIRFOIL FOR HIGH-SPEED AIRPLANES

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SUMMARY

Investigations of the pressure distribution, the profile drag, and the location of transition for a 30-inch-chord 25-percent-thick N.A.C.A. 45-125 airfoil were made in the N.A.C.A. 8-foot high-speed wind tunnel for the purpose of aiding in the development of a thick wing for high-speed airplanes. The tests were made at a lift coefficient of 0.1 for Reynolds Numbers from 1,750,000 to 8,600,000, corresponding to speeds from 80 to 440 miles per hour at 59° F. The effect on the profile drag of fixing the transition point was also investigated.

The effect of compressibility on the rate of increase of pressure coefficients was found to be greater than that predicted by a simplified theoretical expression for thin wings. The results indicated that, for a lift coefficient of 0.1, the critical speed of the N.A.C.A. 45-125 airfoil was about 460 miles per hour at 59° F.

The value of the profile-drag coefficient at a Reynolds Number of 4,500,000 was 0.0058, or about half as large as the value for the N.A.C.A. 0025 airfoil. The increase in the profile-drag coefficient for a given movement of the transition point was about three times as large as the corresponding increase for the N.A.C.A. 0012 airfoil. Transition determinations indicated that, for Reynolds Numbers up to 7,000,000, laminar boundary layers were maintained over approximately 40 percent of the upper and the lower surfaces of the airfoil.

INTRODUCTION

The minimum static pressure on an airfoil has been used as an index of the critical speed of that airfoil, a high minimum pressure indicating a low critical speed. (See reference 1.) Most conventional airfoils have high peaks of minimum pressure that extend over only a small part of the chord. It should be possible to produce airfoils having high critical speeds by changing the high pressure peaks to a flat pressure distribution with the minimum pressures extending over a larger portion of the airfoil.

Thick wings are desirable for many applications, such as engine-in-wing installations, fuel storage, and recesses for landing gear or other equipment. Thick wings of conventional design, however, have low critical speeds and are inefficient for use at high speeds. A 25-percent-thick airfoil was designed having a pressure distribution intended to give a high critical speed for a wing of this thickness.

This airfoil was tested to determine its critical speed and to obtain data to aid in the design of other thick wings with high critical speeds. The tests were made in the 8-foot high-speed wind tunnel at a lift coefficient of 0.1 and at speeds ranging from 80 to 440 miles per hour. The range of the test Reynolds Number was 1,750,000 to 8,600,000, based on the 30-inch chord. Complete pressure-distribution and drag determinations were made to obtain section coefficients. The location of the transition point was determined along both surfaces of the airfoil throughout the speed range. The method of the surface pitot was used at several positions along the chord to determine the location of the transition point, the point at which the velocity near the surface of the wing started to increase very rapidly because of the onset of turbulent flow. For some tests, transition was artificially fixed at two chord locations.

APPARATUS AND METHOD

The investigation was made in the N.A.C.A. 8-foot high-speed wind tunnel, a single-return closed-throat tunnel of circular cross section. Sphere tests in this tunnel have shown virtually the same critical Reynolds Number as in free air (reference 2).

The N.A.C.A. 45-125 airfoil was used in the tests. (See table I.) The first digit, 4, of the airfoil designation indicates the class of airfoil that has a flat-top pressure distribution, a uniform distribution of lift along the chord at the design lift coefficient, and a cusped trailing edge. The second digit, 5, designates the approximate position at which the adverse pressure gradient begins, 50 percent of the chord. The first digit of the last three numbers represents the design lift coefficient, 0.1, and the last two digits give the maximum thickness, 25 percent of the chord.

The airfoil had a 30-inch chord. The surface was painted and then sanded with No. 500 water sandpaper until it was aerodynamically smooth.

The airfoil completely spanned the tunnel and was fastened directly to the tunnel wall (fig. 1). Two-dimensional flow is approximated with this set-up. The pressures were measured at 31 pressure orifices connected to a photographically recording multiple-tube manometer. The orifices were staggered so that no orifice was in the wake of other orifices; they were so located from the center of the span that the survey rake was not in the wake of any orifices.

The drag of the airfoil was determined from measurements in the wake using Jones' method (reference 3) modified to include the effects of compressibility. Measurements were made with a survey rake supported by a vertical strut and located one-half chord length behind the trailing edge of the wing. The rake consisted of 25 equally spaced total-pressure tubes and 6 static-pressure tubes. The pressure readings were photographically recorded on a multiple-tube manometer.

For some tests the transition point was fixed at 15 and 30 percent of the chord from the leading edge by a 1/4-inch spanwise strip of 0.023-inch carborundum grains secured to both surfaces with thin shellac. The 0.023-inch grains were used because, in a preliminary test, 0.0037-inch carborundum grains failed to produce transition. The failure of the narrow band of 0.0037-inch grains to produce transition is no indication of the permissible roughness for this wing. Pressure distribution, drag, and transition-point location were separately determined. Surface pitot tubes were also used to make certain that transition had occurred at the carborundum strips.

SYMBOLS

The symbols used in this report are defined as follows:

V, air speed.

M, Mach number, the ratio of the air speed to the speed of sound in air at the temperature of the tests.

- c , wing chord.
 x , distance measured from leading edge along chord of wing.
 R , Reynolds Number based on chord.
 H , total pressure.
 p , local static pressure on the airfoil.
 q , dynamic pressure of the air stream $(1/2 \rho V^2)$.
 S , pressure coefficient $\left(\frac{H - p}{q} \right)$.
 S_0 , value of S at $M = 0$.
 S_{cr} , pressure coefficient at which the speed of sound is reached at some point on the airfoil.
 M_{cr} , Mach number corresponding to S_{cr} .
 α , angle of attack of airfoil.
 c_l , section lift coefficient.
 c_{d0} , section profile-drag coefficient.

RESULTS

The pressure distributions on both surfaces of the airfoil for $c_l = 0.1$ ($\alpha = 0^\circ$) are shown in figure 2 for various values of M . Figure 3 shows comparisons between the experimental variation with speed of the pressure coefficient for the airfoil tested and Ackeret's theoretical variation for thin airfoils (reference 4) as

given by the factor $1 - \frac{1 - S_0}{\sqrt{1 - M^2}}$. The computed pressures

on the upper surface of the N.A.C.A. 0025 airfoil are compared in figure 4 with the measured pressures on the N.A.C.A. 45-125 for a lift coefficient of 0.1 at low speed.

The variation of the Reynolds Number with Mach number is shown in figure 5. In figure 6, the location of the

transition point is shown for various values of the Reynolds Number. Curves of c_{d0} for the various surface conditions are shown in figure 7; in addition, curves of c_{d0} for the N.A.C.A. 0025 airfoil (reference 5) and the laminar and turbulent skin-friction curves for flat plates are shown. Figure 8 shows the variation of c_{d0} with the transition-point location for various Reynolds Numbers.

All tests reported herein were made with the airfoil at an angle of attack of 0° , which gave approximately the design lift coefficient of 0.1. In the tests with roughness at 0.15c, however, a loss in lift occurred at high speed, probably because of turbulent separation.

Inasmuch as the wing was thick relative to the tunnel diameter, a correction for constriction was made to the pressure coefficients and to the profile-drag coefficients. This correction varied from 2 percent at low speeds to 5 percent at high speeds. A correction was also made for the departure of the effective centers of the total-pressure tubes from the geometric centers in the transverse pressure gradient (reference 6). The profile-drag coefficients for the airfoil with roughness at 0.15c and 0.30c have not been corrected for the direct drag of the carborundum strips that were used to produce and to fix transition; this correction is small and unimportant.

DISCUSSION

Critical speed.- The critical speed is defined as the speed at which a break-down in flow, caused by compressibility effects and known as the compressibility burble, occurs. This flow change is usually evidenced by a rapid rise in the drag coefficient. It has been pointed out in reference 7 that, for practical purposes, the critical speed can be taken as the value of the translational speed at which the sum of the translational and the induced velocities equals the local speed of sound.

Extrapolation of the curves given in figure 3 showing the variation with M of the maximum pressure coefficient to the critical pressure coefficient (the pressure coefficient at which the speed of sound is locally reached) indicates that the critical speed of the N.A.C.A. 45-125

airfoil is about 460 miles per hour at 59° F. ($M_{cr} = 0.61$). The critical speed obtained by using Ackeret's theoretical variation of the pressure coefficient (reference 4),

$$S = 1 - \frac{1 - S_0}{\sqrt{1 - M^2}}, \text{ is about 7 percent higher. The most}$$

probable cause of this discrepancy is the assumption made in the development of the theory that the induced velocities are negligibly small. It is believed that the critical speed of the N.A.C.A. 45-125 airfoil can be increased by so modifying the design that the peak pressure coefficients on both surfaces are reduced to give a flat-peak pressure distribution.

Pressure distribution.— The shapes of the pressure-distribution curves remained practically the same throughout the speed range (fig. 2). As the speed was increased, all the pressure coefficients at points along the tops of the curves increased at almost the same rate except for speeds above 400 miles per hour ($M = 0.53$), where the increase was greater on the lower surface, as is indicated in figure 3. The adverse pressure gradients over the rear 50 percent of the airfoil increased when the speed was increased but apparently caused no separation. At high speeds, some unreported pressure-distribution results, with roughness at 0.15c on both surfaces of the airfoil, showed a loss in lift that was probably caused by separation; no such loss in lift was noted with roughness at 0.30c.

Drag.— The minimum profile-drag coefficient of the smooth N.A.C.A. 45-125 airfoil, for a lift coefficient of 0.1, was 0.0058 at a Reynolds Number of 4,500,000 (fig. 7), or about one-half as large as the profile-drag coefficient of the N.A.C.A. 0025 airfoil (reference 5). The value of c_{d_0} was 0.0068 at the lowest Reynolds Number and it remained below 0.0063 even at the highest Reynolds Number, which corresponded to a speed of 440 miles per hour at 59° F., indicating that compressibility effects were small. These low drag values are due to extensive laminar flows extending over 40 percent of the airfoil up to a Reynolds Number of at least 7,000,000 (fig. 6).

The profile-drag coefficients of the airfoil with roughness at 0.15c and 0.30c were, respectively, 125 percent and 75 percent larger than the drag coefficients of the smooth airfoil for Reynolds Numbers up to 7,000,000. The rapid increase in drag for Reynolds Numbers greater

than 7,000,000 is believed to be due to turbulent separation and not to compressibility effects. This belief is substantiated by the results of the pressure-distribution measurements on the airfoil with roughness at 0.15c, as has previously been discussed, and also by the fact that the critical speed was not reached. It is not known whether there would have been separation if transition had occurred at the same chord position because of large scale rather than roughness.

The variation of c_{d0} with the transition-point location was almost linear for Reynolds Numbers up to 5,000,000 (fig. 8). The drag increase for a given movement of the transition point was about three times as large as the increase for the same movement of the transition point on the N.A.C.A. 0012 airfoil (reference 8). This drag increase is greater than can be accounted for by increased skin friction alone and is probably due mainly to pressure drag.

CONCLUSIONS

1. The pressure-distribution results indicate that the critical speed of the 25-percent-thick N.A.C.A. 45-125 airfoil was about 460 miles per hour at 59° F. for a lift coefficient of approximately 0.1. The test results indicate that it will be possible further to increase the critical speed by so modifying the airfoil that the peak pressures will be reduced to a more uniform pressure distribution.

2. The value of the profile-drag coefficient for the N.A.C.A. 45-125 airfoil at a Reynolds Number of 4,500,000 was 0.0058, or about half as large as the value for the N.A.C.A. 0025 airfoil.

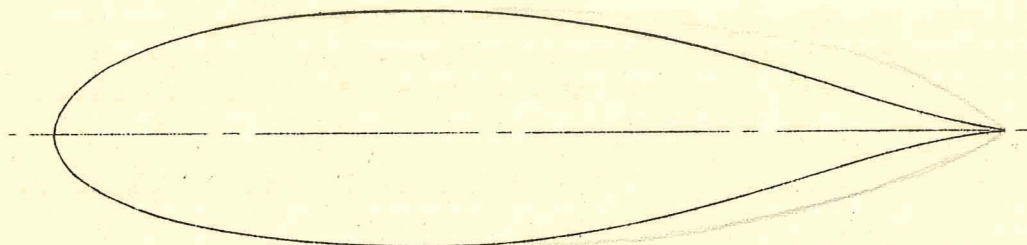
3. Transition determinations indicated that, for Reynolds Numbers up to 7,000,000, laminar boundary layers were maintained over approximately 40 percent of the upper and the lower surfaces of the airfoil.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., November 3, 1939.

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6. Young, A. D., and Maas, J. N.: The Behaviour of a Pitot Tube in a Transverse Total-Pressure Gradient. R. & M. No. 1770, British A.R.C., 1937.
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TABLE I.- ORDINATES OF THE N.A.C.A. 45-125 AIRFOIL



[Stations and ordinates given in percent chord]

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
1.132	3.448	1.368	-3.340
2.364	4.762	2.636	-4.576
4.851	6.528	5.149	-6.212
7.348	7.831	7.652	-7.407
9.850	8.849	10.150	-8.333
14.862	10.327	15.138	-9.655
19.880	11.314	20.120	-10.518
29.919	12.531	30.081	-11.559
39.960	12.906	40.040	-11.834
50.000	12.440	50.000	-11.336
60.033	10.760	59.967	-9.688
70.052	8.223	69.948	-7.251
80.053	5.165	79.947	-4.369
90.032	2.076	89.968	-1.560
95.015	.819	94.985	-.503
100.000	0	100.000	0

L.E. radius : 4.69

Slope of radius through end of chord: $1/20$

N.A.C.A.

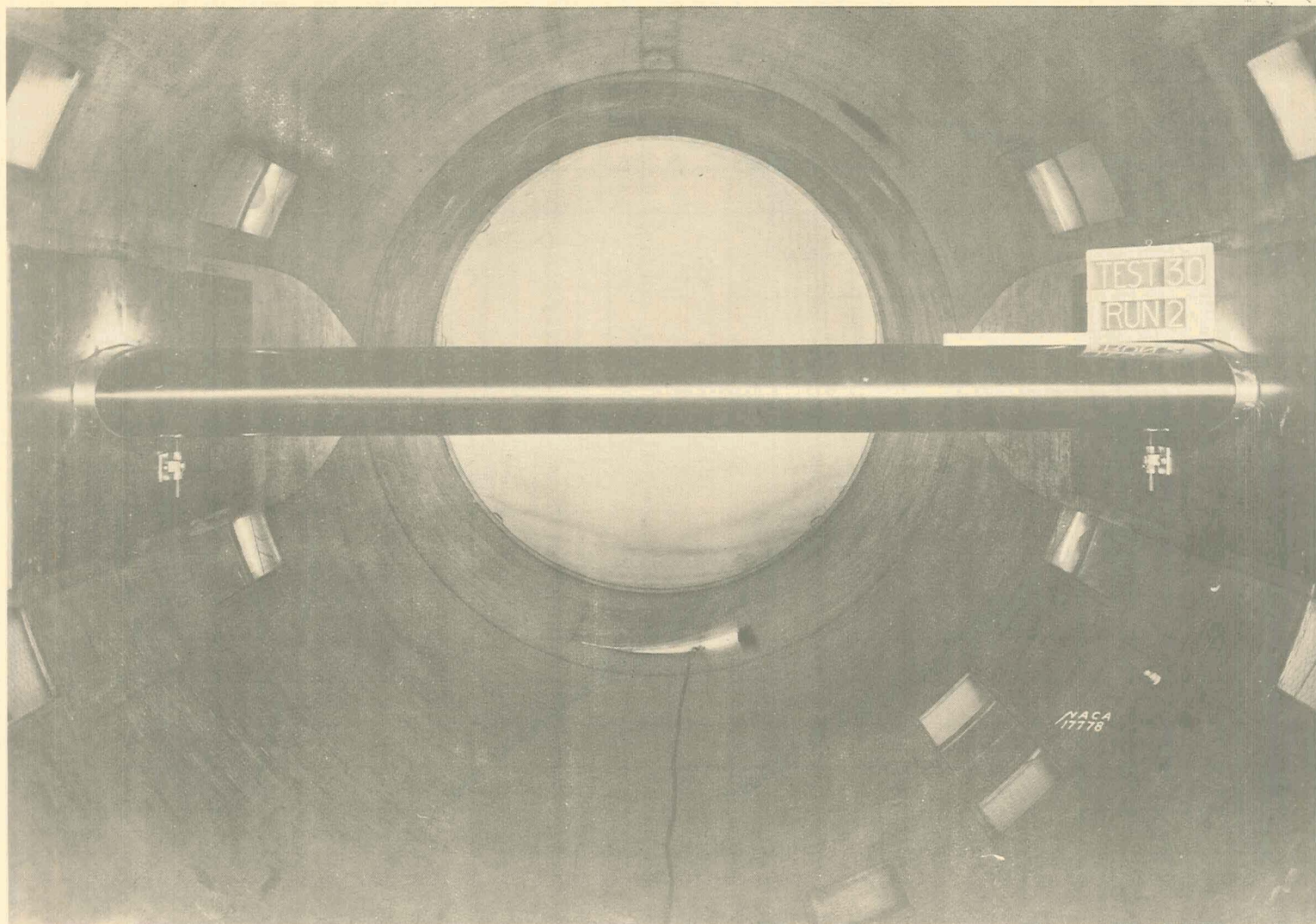


FIG. 1

Figure 1.- The N.A.C.A. 45-125 airfoil in the 8-foot high-speed wind tunnel.

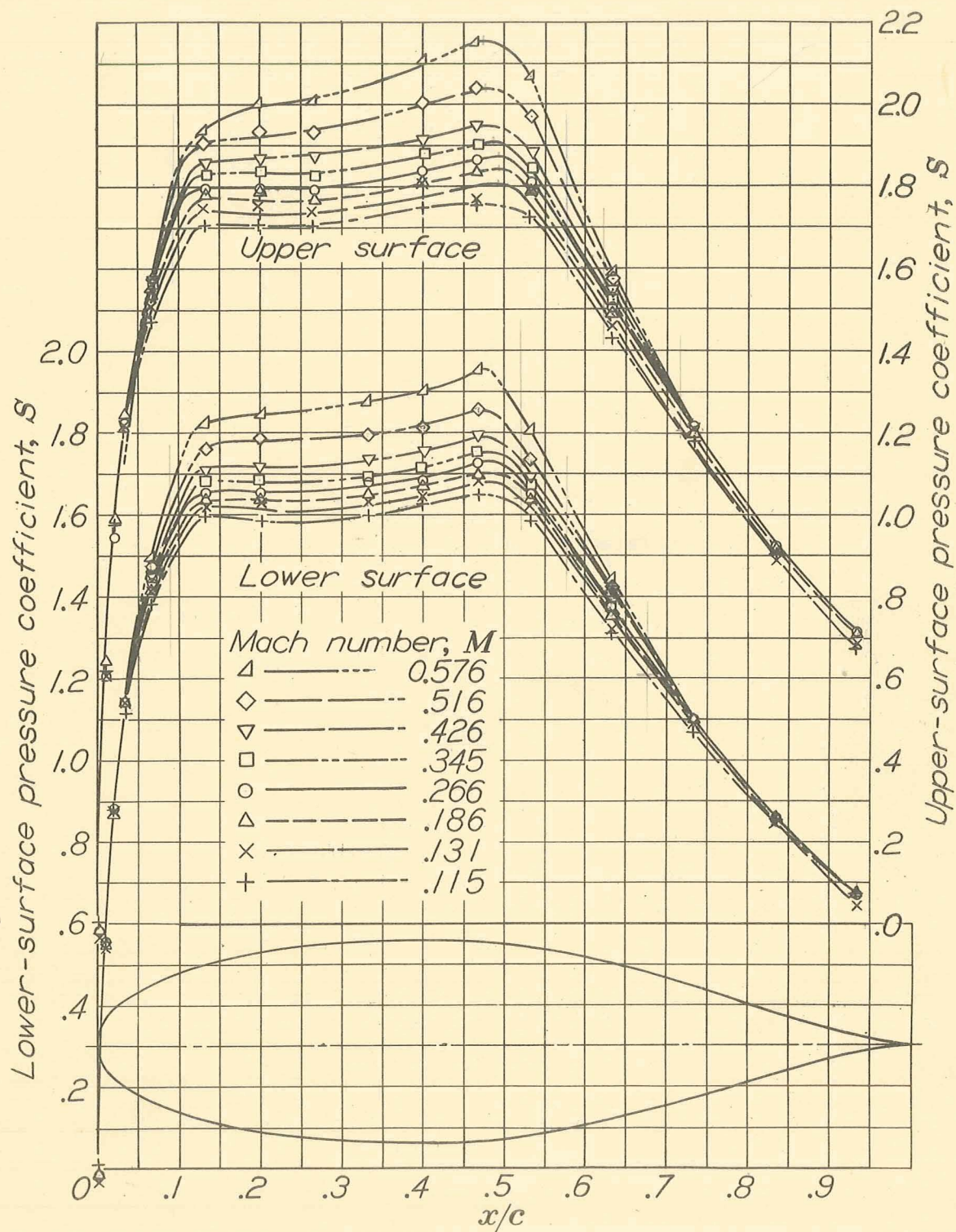


Figure 2.- Pressure distribution on the N.A.C.A. 45-125 airfoil.
 c_l , 0.1.

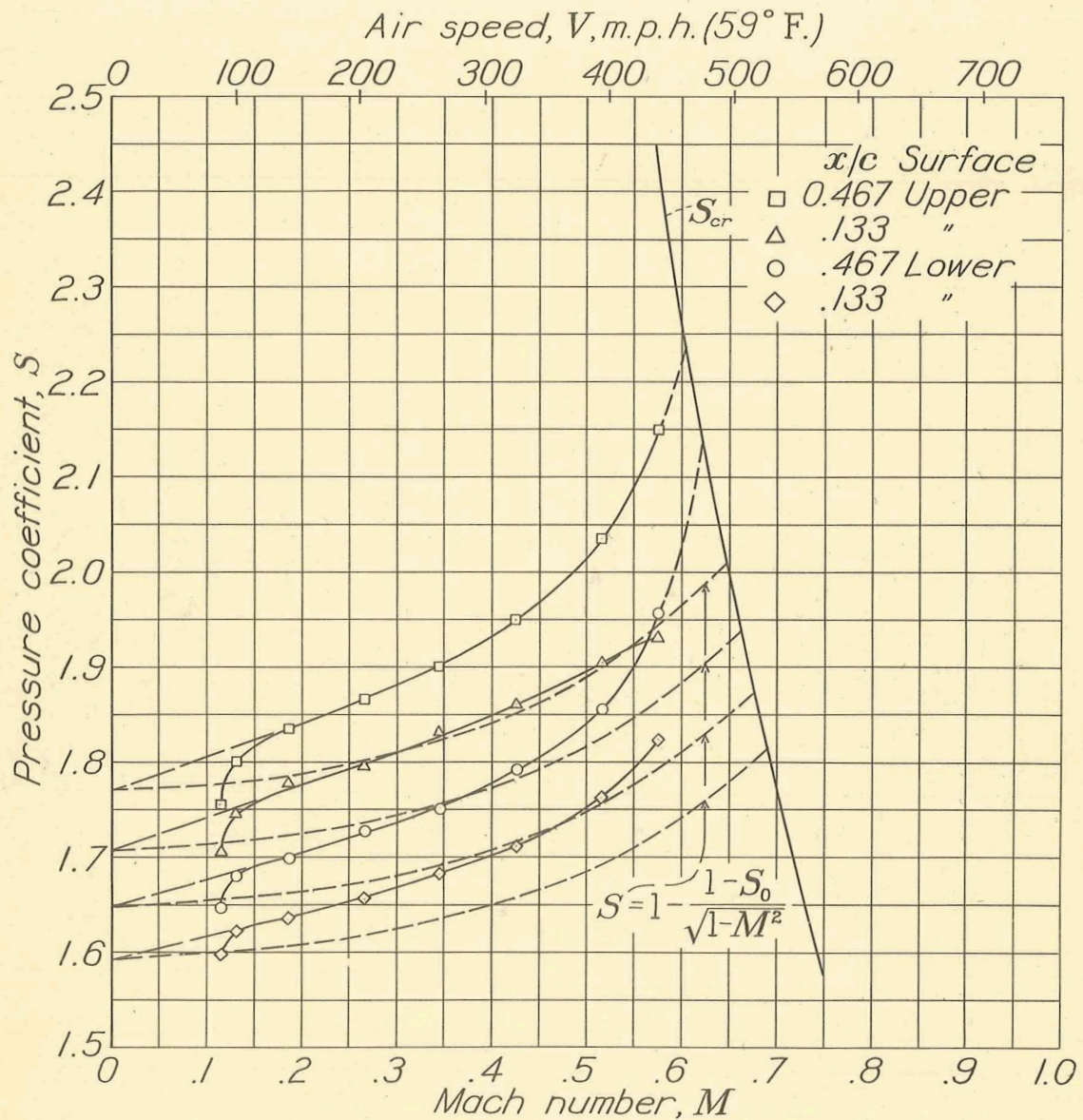


Figure 3.- Effect of compressibility on the pressure coefficient.
 c_l , 0.1.

30" chord

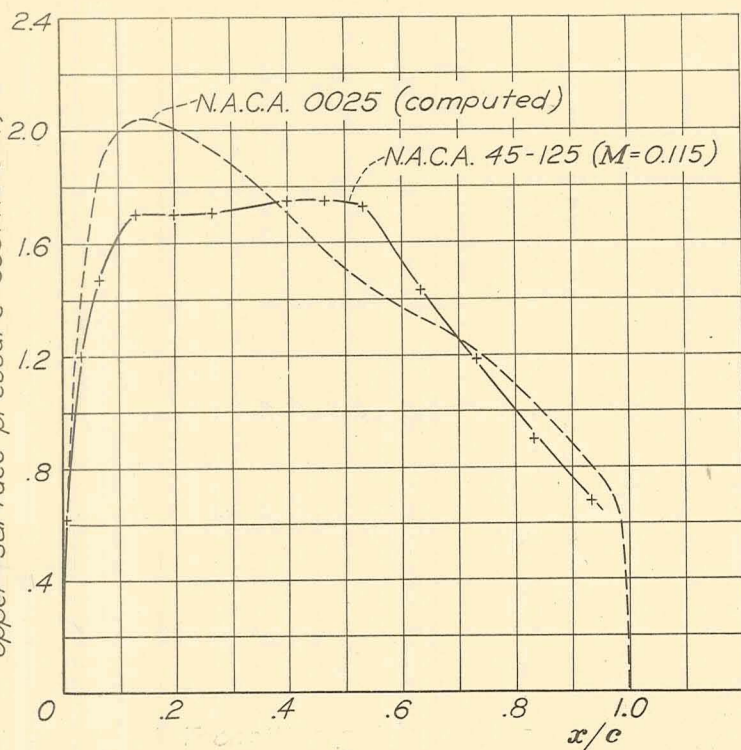


Figure 4.- Pressure distribution on the upper surface of the N.A.C.A. 45-125 airfoil at low speed and theoretical pressure distribution on the N.A.C.A. 0025 airfoil. $C_f, 0.1$.

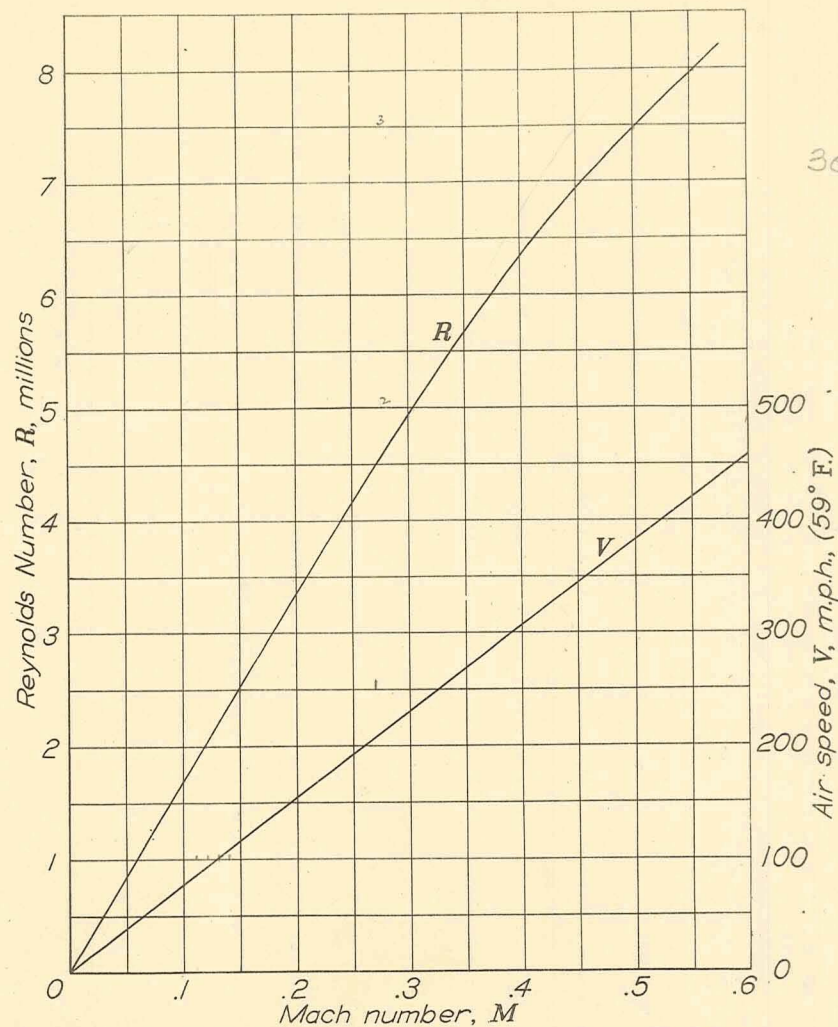


Figure 5.- Variation of Reynolds Number and air speed with Mach number.

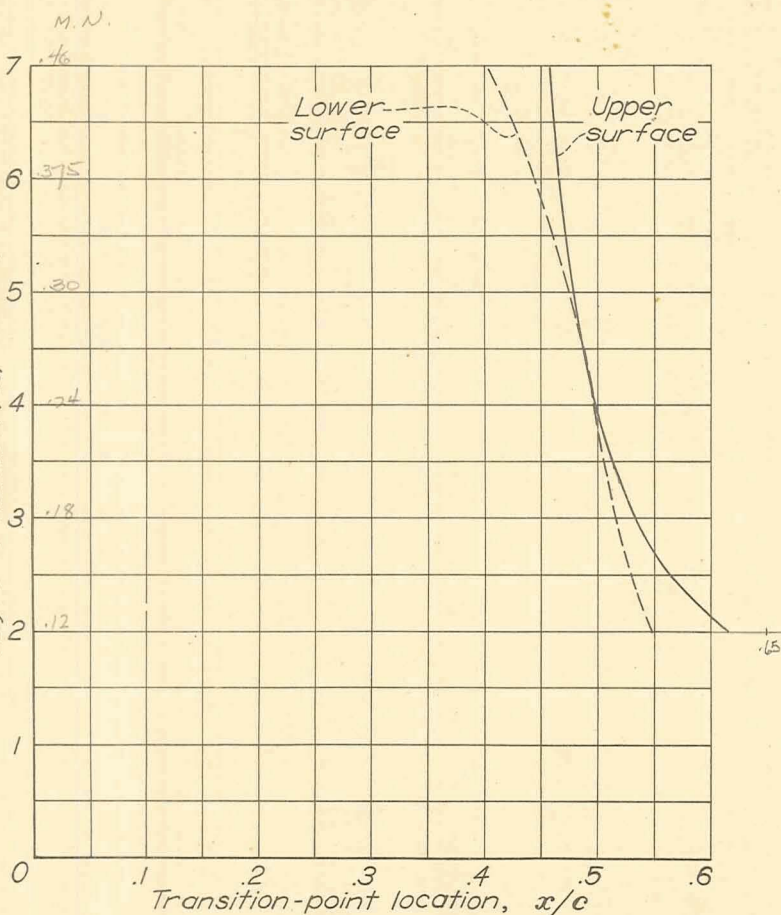


Figure 6.- Effect of Reynolds Number on transition-point location. $c_l, 0.1$.

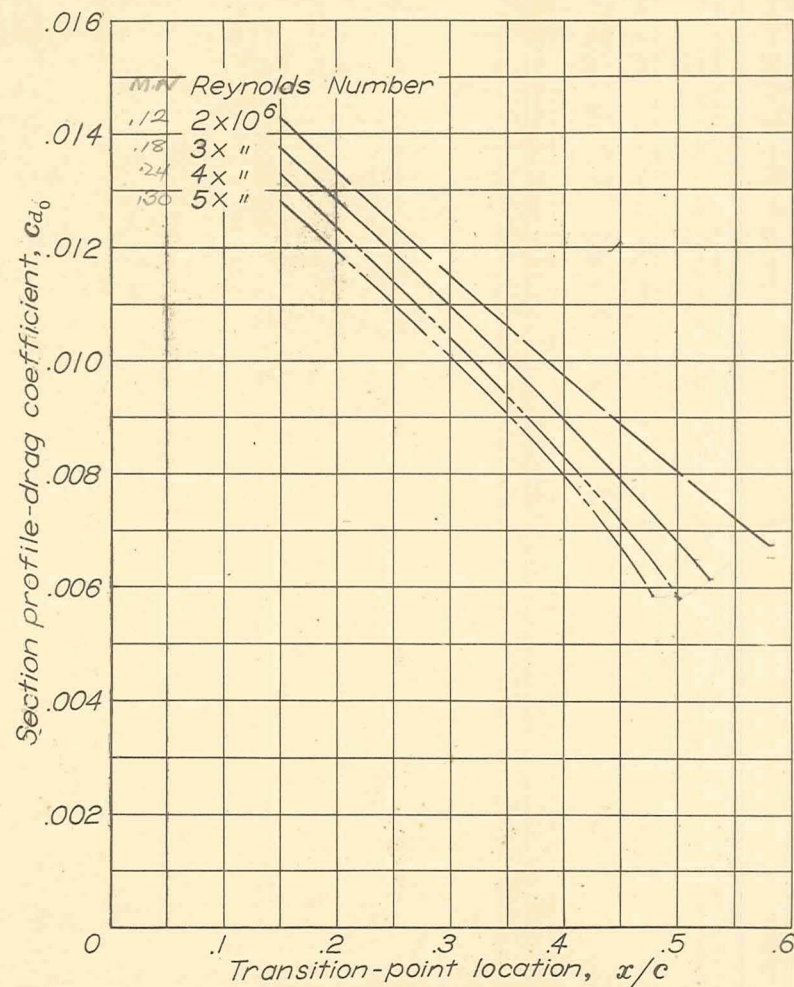


Figure 8.- Variation of the section profile-drag coefficient with transition-point location, $c_l, 0.1$.

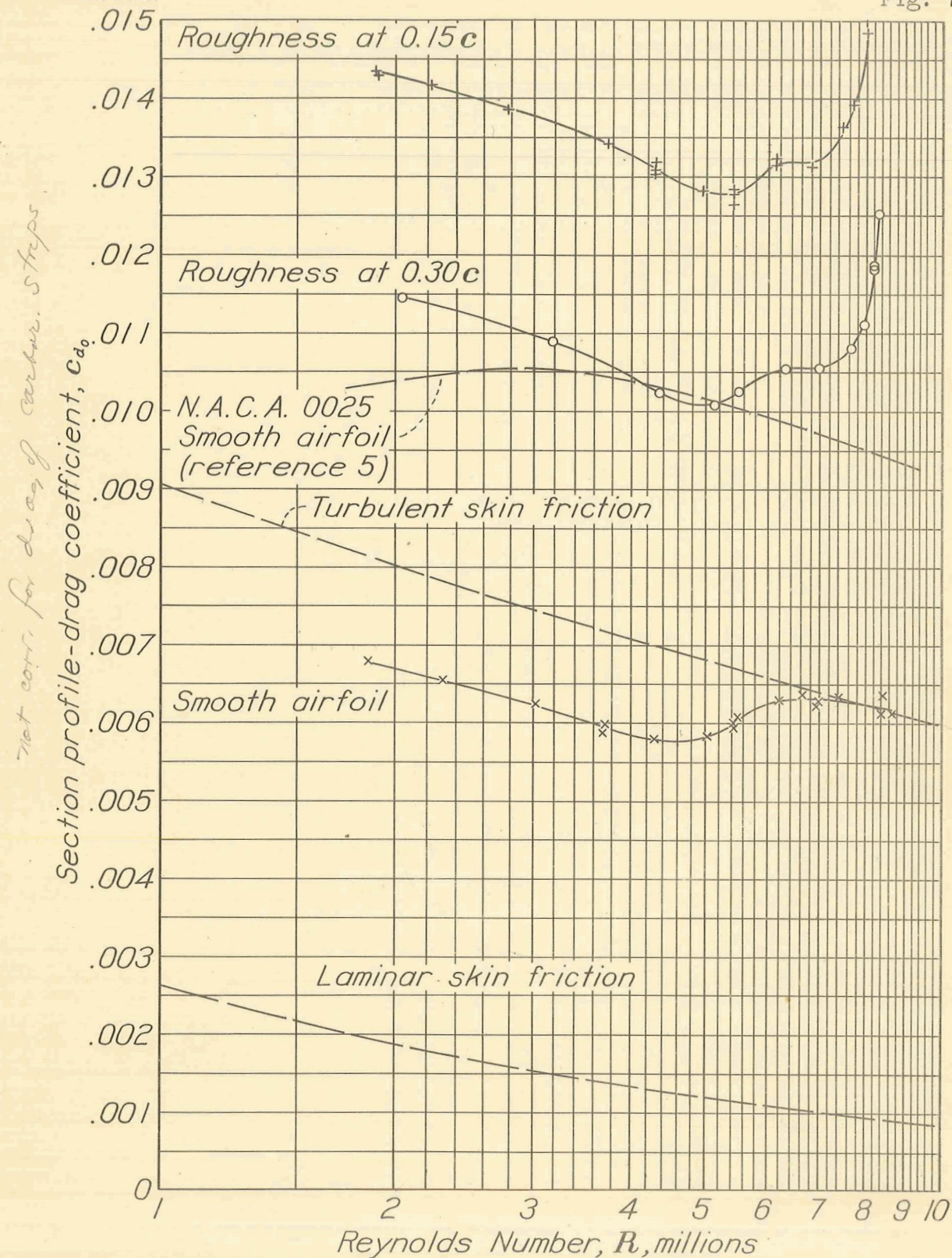


Figure 7.- Section profile-drag coefficient of the N.A.C.A. 45-125 airfoil for various surface conditions.