MR May 1941

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WARTIME REPORT

ORIGINALLY ISSUED

May 1941 as
Memorandum Report

A FLIGHT INVESTIGATION OF THE BOUNDARY-LAYER CHARACTERISTICS AND PROFILE DRAG OF THE NACA 35-215 LAMINAR-FLOW AIRFOIL AT HIGH REYNOLDS NUMBERS

By J. W. Wetmore, J. A. Zalovcik, and Robert C. Platt

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WASHINGTON

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

MEMORANDUM REPORT

for the

Army Air Corps

A FLIGHT INVESTIGATION OF THE BOUNDARY-LAYER CHARACTERISTICS AND PROFILE DRAG OF THE NACA 35-215 LAMINAR-FLOW AIRFOIL AT HIGH REYNOLDS NUMBERS

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SUMMARY

Tests have been conducted in flight to determine the boundary-layer characteristics and the profile drag of the NACA 35-215 airfoil section at high Reynolds numbers. These tests were made on a test panel of 17-foot chord mounted on the left wing of a Douglas B-18 airplane just outside of the propeller slipstream. Tests were made to determine the transition points and the boundary-layer velocity profiles for various surface and power conditions over a range of airplane lift coefficients from 0.20 to 0.46 for which the range of corresponding Reynolds numbers was 30,000,000 to 20,000,000. The profile-drag coefficient of the panel was determined for the best surface condition both with power on and with the engines and propellers stopped over a range of airplane lift coefficients from 0.21 to 0.32 with a Reynolds number range of 32,000,000 to 16,000,000. In addition, the profile drag of the upper surface alone was determined for the same power and surface condition and over approximately the same range of airplane lift coefficients and Reynolds numbers.

With the best surface condition and the left engine stopped, the laminar boundary layer was maintained to 42.4 percent of the chord on the upper surface at a lift coefficient of 0.220 and a Reynolds number of 26,700,000. The results of the transition tests indicated a reduction of about 3 percent of the chord in the laminar-flow run over the upper surface due to operation of the engines and propellers. As a result of reducing the indicated amplitude of the transverse waves on the upper surface from 0.005 to 0.001 inch, the transition point moved back from about 32.5 to about 42.5 percent of the chord.

The velocity surveys in the laminar boundary layer indicated that values of boundary-layer Reynolds number R_δ (based on the distance above the surface at which the dynamic pressure in the boundary layer is one-half that just outside the boundary layer) exceeding 8000 are attainable in flight on suitably designed and carefully finished airfoils.

The profile-drag coefficient of the test panel with engines stopped was found to remain substantially constant at a value of about 0.0048 for flight conditions ranging from an airplane lift coefficient of 0.21 and a corresponding Reynolds number of about 30,000,000 to a lift coefficient of 0.32 and a Reynolds number of 24,000,000. Over the same range of conditions the profile-drag coefficient of the upper surface alone varied from about 0.0022 at the lowest lift coefficient tested to 0.0028 at the highest lift coefficient. With both engines operating at full throttle the drag coefficient due to both surfaces and that due to the upper surface alone were both increased on the order of 8 to 10 percent.

The results of the tests indicate the desirability for continued flight research on airfoils at large scale to supplement the development work of the tunnels.

INTRODUCTION

During the earlier stages of the Committee's work on the development of laminar-flow airfoils (reference 1), it was found that by suitably designing the profile of an airfoil a favorable or accelerating pressure gradient could be maintained over as much as 80 percent of the chord back of the leading edge. Tests of some of these airfoils in the wind tunnels and in flight showed that within the lower flight range of Reynolds numbers the laminar boundary layer extended as far back as 80 percent of the chord from the leading edge, with the result that the profile drag was extremely low.

In the higher Reynolds number ranges, say, above 20,000,000, it was expected that other methods might be required to obtain the desired extensive laminar boundary layers and resulting extremely low drags. The present investigation was undertaken with the object of investigating methods of prolonging the laminar flow at high Reynolds numbers and to give data for comparison with wind-tunnel data. Consequently, a suitable wing was chosen with these objects in view rather than with this object of choosing an optimum section for any particular practical application.

This report represents results of the tests of the plain airfoil. These tests covered a range of Reynolds numbers between 20,000,000 and 30,000,000 and included variations in power condition and surface condition. An investigation of the effect of section slots for boundary-layer control will be covered in a subsequent report.

The tests were made with a B-18 airplane which was made available for this project by the Army Air Corps.

APPARATUS

The Douglas B-18 airplane is a bimotored, fully cantilever, midwing monoplane with a wing area of 958.6 square feet and a design gross weight of 23,200 pounds. It is powered with Wright Cyclone R-1820-45 engines (810 horsepower at 2100 rpm and 8700 feet) fitted with 3-blade propellers having a diameter of 11 feet 6 inches. Hamilton Standard, hydraulically controlled, constant-speed propellers are normally used on this airplane, but for most of the present tests, they were replaced by Curtiss electrically controlled full-feathering propellers in order that the engines could be stopped during flight. The weight of the airplane as flown was approximately 22,000 pounds.

A test panel having the NACA 35-215 airfoil section (table I) was mounted on the left wing of the airplane. The chord of the panel was 17 feet and the span was 10 feet at the leading edge, tapering to 5 feet at the trailing edge. It was constructed of laminated white pine in the form of a hollow shell with walls about 2 inches thick; the outside profile was accurately shaped to templet size. The surfaces were sprayed with several coats of lacquer base filler and rubbed down with various grades of water cloth, the final finish being obtained with a No. 400 water cloth. The panel was supported on the wing by rubber pads running along the top and bottom of the wing spars and was secured in place by means of steel straps. The position of the panel was such that the inboard end of the leading edge was about 1 foot outboard of the propeller disk, the leading and trailing edges were normal to the plane of symmetry of the airplane, and the plane of chord lines coincided approximately with the plane of chord lines of the wing. The panel was faired into the wing by means of fabric stretched taut over a wooden framework. The weight of the panel and fairing was 1394 pounds; satisfactory lateral balance for all conditions of flight was obtained by removing all fuel from the left-wing tanks and adding 350 pounds of ballast in the right wing tip. Figure 1 is a photograph of the test panel mounted on the wing; its dimensions and location are shown in figure 2.

The upper surface of the panel was refinished several times during the course of the tests so that various surface conditions are represented in the results, An index of the surface waviness, i. e., the magnitude of the transverse waves, was obtained by measuring the curvature variation along the surface by means of the device shown in figure 3. Finishing the lower surface was found to be very difficult so that no attempt was made to refinish it and no waviness measurements were made on it. The condition of the lower surface throughout the investigation is believed to have been about the same as the initial condition of the upper surface.

Free-stream static and total pressures were measured by means of static- and total-pressure tubes which were calibrated with a static head suspended below the airplane.

The characteristics of the boundary layer were determined by means either of 5-tube or 2-tube racks. The 5-tube racks were each composed of a static-pressure tube and four total-pressure tubes arranged to measure the static pressure just outside the boundary layer and the total pressure close to the surface and at various distances above the surface within the boundary layer; they were used to determine the velocity profile of the boundary layer. In cases where it was desired to determine only the point at which transition occurred the 2-tube racks, each consisting of a static tube located just outside the boundary layer and a total-pressure tube located close to the surface, were used.

Wake-pressure surveys for the determination of profile drag were accomplished by means of a bank of 25 total-pressure and 6 static-pressure tubes located 12 percent of the chord back of the trailing edge on the panel center line and extending through the entire wake. The total-pressure tubes were spaced 0.60 inch apart. A bank of tubes consisting of 21 total-pressure tubes, spaced 0.25 inch apart, and 3 static-pressure tubes, mounted at the center of the trailing edge and extending only through the upper surface wake was used for the determination of the profile drag of the upper surface alone.

All pressures were measured by means of a multiple-tube alcohol manometer and were recorded photographically.

TESTS

Boundary-layer measurements were made on the upper surface of the test panel over a range of airplane lift coefficients from about 0.20 to 0.46; the range of corresponding Reynolds numbers was from about 30,000,000 to 20,000,000. Several conditions of the panel surface, as indicated in figure 4, and various power conditions were investigated. The power conditions covered were as follows: both engines full throttle; both engines idling; left engine stopped, right engine full throttle; right engine stopped, left engine full throttle; both engines stopped. Only a few tests were made on the lower surface of the panel because of its inferior condition.

The profile drag due to both surfaces and that due to the upper surface alone was determined with the panel surfaces in the final condition and for two power conditions: both engines at full throttle and both engines stopped. The profile-drag measurements covered a range of airplane lift coefficients from 0.21 to 0.32 with a range of corresponding Reynolds numbers from 32,000,000 to 24,000,000.

Inasmuch as it was necessary to dive the airplane in order to attain the low lift coefficients desired, the relative lag of the various pressure tubes and lines was determined by special tests and the results were corrected accordingly.

RESULTS

Results of the investigation are presented in figures 5 to 10 and in tables II to V. In figure 5 the distributions of pressure coefficient, S, $(S=q/q_o)$, over the forward parts of the surfaces are shown. All experimental points in figure 5 are for positions along the center line of the upper and lower surfaces of the test panel and were determined by means of the boundary-layer racks. Transition results are presented in tables II and III for four surface conditions as shown in figure 4, and for various engine and propeller conditions. The ranges of lift coefficient and Reynolds number covered in each test run are included in addition to the particular lift coefficients and Reynolds numbers at which transition occurred. The method of determining the conditions for transition is indicated in figure 6. In figures 7 and 8 the velocity distributions in the laminar-boundary layer are shown for various chordwise and lateral positions on the upper and lower surfaces as plots of u/U against $\frac{y}{c}\sqrt{R}$, where u is the velocity within the boundary layer, U is the velocity just outside the boundary layer, y is the distance from the surface at which u is measured, c is the panel chord, and R is the Reynolds number in terms of the panel chord and the free-stream velocity; this method of plotting eliminates the effect of variations in Reynolds number. Values of R_{δ}, the boundary-layer Reynolds number in terms of U and of the value of y at which $u/U = 0.707$, are listed in table IV for various conditions under which transition to turbulent flow was probably imminent. The profile-drag coefficients for both surfaces and for the upper surface alone are given in figures 9 and 10, respectively, and in table V.

DISCUSSION

The pressure distribution over the forward 53 percent of the chord on the upper surface and over 40 percent of the chord on the lower surface was determined from the static-pressure measurements obtained with the boundary-layer racks. Inasmuch as the section lift coefficients c_l could not be evaluated without pressure-distribution data over the entire panel chord, the results of the investigation are presented in relation to the airplane lift coefficient C_L. A spanwise variation in the surface pressures indicated that the section lift coefficient varied on the order of 4 or 5 percent over the range of spanwise positions covered in the tests, being highest inboard and lowest outboard of the panel center line. The section lift coefficient at the center of the test panel is estimated to be about 0.90 of the airplane lift coefficient.

The experimental pressure distribution shown in figure 5 was obtained at an airplane lift coefficient of 0.238 so that the section lift coefficient was probably about 0.22 as compared to the value of 0.20 at which the airfoil is designed to operate. This small difference in lift coefficient would probably not materially affect the shapes of the curves. The minimum pressure on the upper surface is shown to occur at about 45 percent of the chord.

The transition conditions summarized in tables II and III are defined as the conditions at which, for a given chordwise position, a slight departure from the given lift coefficient-Reynolds number combination would cause transition from laminar to turbulent flow. The transition was generally well defined by an abrupt rise in the velocity close to the surface as illustrated in figure 6.

Comparison of the transition results for the various conditions tested is rather uncertain in some cases owing to the fact that there is no fixed relation between airplane lift coefficient and Reynolds number; i. e., for a quantitative evaluation of the effect, for example, of the power or surface condition on the extent of the laminar-boundary layer, comparison should be made at the same lift coefficient and at the same Reynolds number. There are, however, several conclusions indicated by the results. With the best surface condition tested (condition D, fig. 4) and with the left engine stopped the laminar boundary layer was maintained to 42.4 percent of the chord on the upper surface. As shown in table II, transition was observed at this station at several different combinations of C_L and R owing to the unavoidable variation in the relation of R to C_L between different test runs. At an airplane lift coefficient of 0.220 which most nearly approaches the design lift coefficient of the panel ($c_l = 0.20$), the Reynolds number for transition at 42.4 percent of the chord was 26.7 millions. The transition point on the lower surface was not determined for exactly the foregoing conditions but, as shown in table III, at a lift coefficient of 0.247 and a Reynolds number of 26.8 millions transition occurred at 28.4 percent of the chord so that for $C_L = 0.220$, representing a more unfavorable condition for the lower surface, the extent of the laminary layer would be somewhat less than 28.4 percent of the chord. This result is an indication of the degree of inferiority of the lower surface condition as compared to that of the best upper surface condition.

The influence of surface condition on the position of transition is shown more directly by comparison between the transition results obtained with the different upper surface conditions. With condition A, for which the indicated amplitude of the transverse surface waviness was as much as 0.005 inch, and with the left engine stopped, transition occurred at 32.5 percent of the chord and 24 inches outboard of the panel center line at an airplane lift coefficient of 0.247 and a Reynolds number of 26.4 millions. For surface condition D, with an indicated waviness amplitude of 0.001 inch, and the same power condition the transition occurred at 42.4 percent of the chord at the same Reynolds number and a more unfavorable lift coefficient of 0.256. The result of the improvement in the upper surface condition was therefore an increase in the extent of the laminar boundary layer of at least 10 percent of the chord. The effects of the intermediate surface conditions are not definitely indicated by the results.

Operation of the engines and propellers had an adverse effect on the extent of the laminar layer. Comparison of the results obtained with both engines operating at full throttle with those obtained with both engines stopped indicates a reduction in the laminar-flow run of about 3 percent of the chord.

In figures 7 and 8 boundary-layer velocity distributions, determined for several conditions from the tests, are compared with the theoretical Blasius flap-plate distributions. In general, the experimental points conform to the theoretical profile shape within the probable limits of accuracy of the measurements. The effect of the favorable pressure gradient, which is maintained over the forward 45 percent of the 35-215 airfoil section, is evidenced in figure 7 by the values of equivalent flat-plate length, corresponding to the Blasius profiles, which are generally less than the actual distance along the surface from the stagnation point.

The values of R_δ derived from the measured velocity distributions in the laminar boundary layer and listed in table IV range from about 7500 to 9000. Although individual values may not be entirely reliable, the results, in general, are sufficiently consistent to permit the conclusion that values of R_δ of at least 8000 are attainable before transition occurs in flight on suitably designed and carefully finished airfoils. The value 8000 represents a considerable increase over the highest values obtained in the original NACA low-turbulence tunnel on laminar-flow airfoils similar to the 35-215 section; this comparison indicates that even with extremely low turbulence in the tunnel air stream, boundary-layer and profile-drag measurements may be subject to considerable revision when applied to flight conditions. It is pointed out that while the value $R_\delta = 8000$ may not be the ultimate attainable, this value has been attained and therefore may be used as a guide in estimating what may be expected in the extent of the laminar boundary layer and hence in profile drag for airfoils having pressure-distribution characteristics generally similar to those of the 35-215 airfoil.

The profile-drag coefficient of the panel was determined from the full-wake surveys in accordance with the momentum method as developed by Jones. (See reference 3.) For the power-off condition the coefficient is substantially constant over the range of lift coefficient and Reynolds number investigated and has a value of about 0.0048. With power on the value is increased to about 0.0052 or 8 percent.

In view of the inferior condition of the lower surface of the panel the profile-drag measurements on the upper surface alone are considered as more nearly representative of the capabilities of the airfoil. The drag coefficients were evaluated from the half-wake surveys by the method of Squire and Young. (See reference 4.) As shown in figure 10, for the power-off condition the coefficient increased from about 0.0022 at an airplane lift coefficient of 0.23 and a Reynolds number of 29,000,000 to 0.0028 at a lift coefficient of 0.32 and a Reynolds number of 24,000,000. It is reasonable to assume that for equally good surface conditions the drag due to the lower surface would be less than that of the upper surface so that the minimum drag coefficient of the airfoil would be somewhat less than 0.0044. The adverse effect on the drag coefficient due to engine and propeller operation is substantiated by the power-on results which show an increase in drag coefficient of about 10 percent over the power-off values.

In reference 4, in addition to the method of determining profile drag from wake surveys, there is developed a method of predicting the drag from a knowledge of the location of the transition point, the laminar boundary-layer velocity distribution immediately forward of the transition point, and the pressure distribution between the transition point and the trailing edge. To make use of this method the experimental pressure-distribution curve for the upper surface given in figure 5 was extended from 53 percent of the chord to the trailing edge where the pressure was known from the half-wake surveys. The profile-drag coefficient of the upper surface was then calculated for the cases of transition at 42.5 percent and 32.5 percent of the chord, both at a Reynolds number of 28,000,000. For the 42.5 percent location the drag coefficient was 0.0023 which is in close agreement with the value obtained by the wake-survey method. With transition at 32.5 percent of the chord the drag coefficient was calculated to be 0.0028. These results indicate a reduction of about 18 percent in the profile drag due to the improvement in surface condition between condition A and condition D.

The significance of the values of profile drag obtained from the tests of the 35-215 airfoil section may become more apparent from suitable comparisons. For example, the theoretical turbulent skin-friction drag coefficient for two sides of a flat plate at the Reynolds number at which the value of 0.0048 was obtained for the test panel is 0.0052 or about 8 percent greater. The minimum profile-drag coefficient for the conventional NACA 0015 airfoil section is estimated to be 0.0057 at the same Reynolds number or about 20 percent greater than that of the 35-215 section. Comparison on the basis of the upper surface drag indicates that the single surface turbulent skin friction of a flat plate is about 12 percent greater and the single surface drag of the 0015 section about 30 percent greater than the upper surface drag of the 35-215 airfoil section.

CONCLUDING REMARKS

A laminar boundary layer was maintained over the upper surface of the NACA 35-215 test panel to $x/c = 0.424$ where transition to turbulent flow occurred at a lift coefficient of 0.220 and a Reynolds number of 26,700,000. Improving the condition of the upper surface so that the indicated amplitude of the transverse waves, as measured with the surface curvature gage, was reduced from 0.005 inch to 0.001 inch resulted in increasing the extent of the laminar boundary layer from 32.5 percent to 42.5 percent of the chord, thereby probably reducing the profile-drag coefficient of the upper surface about 18 percent. The results of the transition tests indicated a forward movement of the transition point of about 3 percent of the chord due to operation of the engines and propellers.

The velocity surveys in the laminar boundary layer indicated that values of boundary-layer Reynolds number R_δ (based on the distance from the surface at which the dynamic pressure in the boundary layer is one-half that just outside the boundary layer) exceeding 8000 are attainable in flight on suitably designed and carefully finished airfoils.

The profile-drag coefficient with power off was very nearly constant with a value of 0.0048 for flight conditions ranging from an airplane lift coefficient of 0.21 and a corresponding Reynolds number of about 30,000,000 to a lift coefficient of 0.32 and a Reynolds number of 24,000,000. For the same range of conditions the profile-drag coefficient of the upper surface alone varied from 0.0022 to 0.0028. The effect of full-throttle operation of the engines and propellers increased the profile-drag coefficients as measured for both surfaces and for the upper surface alone on the order of 8 to 10 percent.

Comparison of the results of the present flight tests on the 35-215 airfoil section with data obtained on generally similar airfoils in the original NACA low-turbulence wind tunnel showed that in flight the laminar boundary layer was maintained to values of R_δ considerably greater than the highest values that were attained in the tunnel. This result indicated that even in tunnel air streams of extremely low turbulence the effect of the residual turbulence might be appreciable, and thereby demonstrated the necessity of continued flight research on airfoils of large scale to supplement the development work of the tunnels.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., May 5, 1941.

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TABLE I
ORDINATES OF NACA 35-215 AIRFOIL

UPPER SURFACE		LOWER SURFACE	
$\frac{x}{c} \times 100$	$\frac{y}{c} \times 100$	$\frac{x}{c} \times 100$	$\frac{y}{c} \times 100$
0	0	0	0
1.085	1.857	1.415	-1.563
2.307	2.619	2.693	-2.101
4.786	3.674	5.214	-2.792
7.278	4.510	7.722	-3.322
9.777	5.211	10.223	-3.759
14.788	6.344	15.212	-4.448
19.809	7.221	20.191	-4.973
24.838	7.899	25.162	-5.375
29.873	8.416	30.127	-5.680
34.913	8.774	35.087	-5.888
39.958	8.961	40.042	-5.989
50.077	8.702	49.923	-5.762
60.150	7.265	59.850	-4.703
70.137	5.277	69.863	-3.295
80.086	3.123	79.914	-1.817
85.056	2.098	84.944	-1.140
90.029	1.175	89.971	-.561
95.009	.436	94.991	-.148
100.000	0	100.000	0

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TABLE II

Summary of Results of Transition Tests on Upper Surface of N.A.C.A. 35-215 Test Panel

$\frac{x}{c}\times 100$	Flt No	Surface Cond.	Range of Flt. Cond. From C_L	From $R\times10^{-6}$	To C_L	To $R\times10^{-6}$	Outboard 24 in. C_L	Outboard 24 in. $R/10^6$	Outboard 18 in. C_L	Outboard 18 in. $R/10^6$	Outboard 12 in. C_L	Outboard 12 in. $R/10^6$	Center line C_L	Center line $R/10^6$	Inboard 12 in. C_L	Inboard 12 in. $R/10^6$	Inboard 18 in. C_L	Inboard 18 in. $R/10^6$	Inboard 24 in. C_L	Inboard 24 in. $R/10^6$	Inboard 34 in. C_L	Inboard 34 in. $R/10^6$
Both Engines Full Throttle																						
16.9	3	A	.265	27.0	.459	20.3									.425	20.9						
22.7	4	A	.245	27.3	.466	19.9									.364	22.3						
24.7	5	A	.243	27.8	.373	22.3					.269	26.4					.317	24.1				
27.7	3	A	.265	27.0	.459	20.3							T									
	4	A	.245	27.3	.466	19.9							T									
	5	A	.243	27.8	.373	22.3	.243	27.8					T								T	
			.245	25.3	.267	24.0	.256	24.7					T								T	
			.251	24.3	.382	20.0	.335	21.2					T								T	
	6	A	.247	27.3	.257	24.6											.258	24.6				
	7	A	.211	27.7	.415	20.6													.269 .296	26.2 24.3		
32.5	6	A	.247	27.3	.257	24.6	.251	24.6														
	7	A	.211	27.7	.415	20.6	T						T									
	8	A	.238	26.8	.256	25.8							T						T			
	9	B	.256	27.3	.288	26.2							T						T			
	10	B	.217	29.5	.261	26.8							T						T			
	11	C	.229	28.3	.269	26.0					.238	27.8	.238	27.8	T							
	14	D	.234	31.4	.326	25.4							.271	26.6								
37.4	7	A	.211	27.7	.415	20.6					T											
	8	A	.238	26.8	.256	25.8			T													
	9	B	.256	27.3	.288	26.2			T													
	10	B	.217	29.5	.261	26.8			T													
	14	D	.234	31.4	.326	25.4					T											
	15	D	.207	29.7	.317	24.3					.256	27.0										
38.8	3	A	.265	27.0	.459	20.3					T											
	4	A	.245	27.3	.466	19.9					T											
40.0	15	D	.207	29.7	.317	24.3							.207	29.7								
42.4	15	D	.207	29.7	.317	24.3									T							
44.1	3	A	.265	27.0	.459	20.3			T								T					
	4	A	.245	27.3	.466	19.9			T								T					
Right Engine Stopped; Left Engine Full Throttle																						
32.5	11	C	.221	29.4	.292	25.6					.256	27.1	.256	27.1	T							
Left Engine Stopped; Right Engine Full Throttle																						
27.7	7	A	.234	27.8	.375	22.5													.247	26.8		
32.5	7	A	.234	27.8	.375	22.5	.247	26.4					T									
	8	A	.210	28.0	.261	26.4							.210	28.0					T			
	9	B	.264	26.6	.310	24.8							T						T			
	10	B	.226	29.1	.277	26.5							T						T			
	11	C	.221	29.5	.304	26.0					.269	26.8	.269	26.8	T							
37.4	7	A	.234	27.8	.375	22.5					T											
	8	A	.210	28.0	.261	26.4			T													
	9	B	.264	26.6	.310	24.8			T													
	10	B	.226	29.1	.277	26.5			T													
40.0	19	D	.220	26.7	.323	23.8							.274	24.6								
			.229	28.1	.248	26.7							.243	27.0								
			.256	26.5	.287	25.4							.258	26.5								
42.4	19	D	.220	26.7	.323	23.8					.269	24.8			.220	26.7						
			.229	28.1	.248	26.7					.238	27.4			T							
			.256	26.5	.287	25.4					.256	26.5			T							
			.225	28.5	.262	27.6					T				T							
Both Engines Stopped																						
27.7	7	A	.238	28.2	.340	22.2													.260	26.6		
32.5	7	A	.238	28.2	.340	22.2	T						T									
	8	A	.246	26.5	.274	26.3							T						T			
	11	C	.225	31.7	.256	28.5					.250	28.5	.250	28.5	T							
	14	D	.211	29.6	.280	26.7							L									
37.4	7	A	.238	28.2	.340	22.2					T											
	8	A	.246	26.5	.274	26.3			T													
	14	D	.211	29.6	.280	26.7					.221	29.0										
	15	D	.212	29.9	.321	25.5					.300	25.8										
40.0	15	D	.212	29.9	.321	25.5							.251	27.4								
42.4	14	D	.211	29.6	.280	26.7									T							
	15	D	.212	29.9	.321	25.5									T							
Both Engines Idling																						
29.7	7	A	.256	25.0	.433	20.2													.256 .379	25.0 21.5		
32.5	7	A	.256	25.0	.433	20.2	T						T									
37.4	7	A	.256	25.0	.433	20.2					T											

Note: T indicates turbulent layer and L, laminer layer at a given position for the given range of flight conditions.

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TABLE III

Summary of Results of Transition Tests on Lower Surface of N.A.C.A. 35-215 Test Panel.

Lateral Position of Racks						Outboard ~ inches 24		Outboard ~ inches 12		Center line		Inboard 24 inches	
$\frac{x}{c}$x100	Flt No	Range of Flight Cond. from C_L	from $R/10^6$	to C_L	to $R/10^6$	C_L	$R/10^6$	C_L	$R/10^6$	C_L	$R/10^6$	C_L	$R/10^6$
Both Engines Full Throttle													
24.7	3	.265	27.0	.459	20.3					.295	25.2		
27.5	15	.207	29.7	.317	24.3			.256	27.0				
28.4	7	.211	27.7	.415	20.6							.291	24.5
30.3	4	.245	27.3	.466	19.9					.295	24.9		
33.2	7	.211	27.7	.415	20.6	.353	22.3			.309	23.8		
	8	.238	26.8	.256	25.8	T				T			
	9	.256	27.3	.288	26.2	T				T			
	10	.217	29.5	.261	26.8	T				T			
	14	.234	31.4	.326	25.4					.295	25.6		
	15	.207	29.7	.317	24.3					.322	24.2		
38.1	8	.238	26.8	.256	25.8							T	
	9	.256	27.3	.288	26.2							T	
	10	.217	29.5	.261	26.8							T	
Left Engine Stopped; Right Engine Full Throttle													
28.4	7	.234	27.8	.375	22.5							.247	26.8
33.2	7	.234	27.8	.375	22.5	.265	25.8			.265	25.8		
	8	.210	28.0	.261	26.4	T				T			
	9	.264	26.6	.310	24.8	T				T			
	10	.226	29.1	.277	26.5	T				T			
	19	.220	26.7	.323	23.8					.282	24.2		
		.256	26.5	.287	25.4					.273	25.9		
38.1	8	.210	28.0	.261	26.4							T	
	9	.264	26.6	.310	24.8							T	
	10	.226	29.1	.277	26.5							T	
39.2	19	.220	26.7	.323	23.8			.295	23.8				
		.256	26.5	.287	25.4			T					
Both Engines Stopped													
27.5	15	.212	29.9	.321	25.5			.260	26.6				
28.4	7	.238	28.2	.340	22.2							.260	25.6
33.2	7	.238	28.2	.340	22.2	.265	25.8			.265	25.8		
	8	.246	26.5	.274	26.3	T							
	14	.211	29.6	.280	26.7					T			
	15	.212	29.9	.321	25.5					.300	25.8		
38.1	8	.246	26.5	.274	26.3							T	
Both Engines Idling													
28.4	7	.256	25.0	.433	20.2							.273	24.5
33.2	7	.256	25.0	.433	20.2	.344	22.3			.295	23.5		

Note T indicates turbulent layer at given position for the given range of flight conditions.

TABLE IV

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

VALUES OF R_δ DETERMINED FROM BOUNDARY LAYER MEASUREMENTS ON NACA 35-215 AIRFOIL.

POWER CONDITION	SURFACE CONDITION	$\frac{X}{C} \times 100$	SPANWISE POSITION	C_L	R_δ
UPPER SURFACE					
BOTH ENGINES FULL-THROTTLE	B	32.5	CENTER	.229	8100
	C	32.5	CENTER	.227	8000
				.239	8100
				.261	8300
		37.4	12 INCHES OUTBOARD	.211	8600
				.216	8100
LEFT ENGINE STOPPED; RIGHT ENGINE FULL THROTTLE	B	32.5	CENTER	.220	8000
				.250	7700
	D	40.0	CENTER	.226	7800
				.229	8000
				.259	7700
				.280	7400
		42.4	12 INCHES OUTBOARD	.226	8100
				.229	8200
				.259	8000
BOTH ENGINES STOPPED	C	32.5	CENTER	.215	8600
				.224	8400
				.270	8100
				.272	8100
		37.4	12 INCHES OUTBOARD	.213	8400
				.215	9200
				.228	8300
				.255	8300
				.279	8000
	D	40.0	CENTER	.213	8500
LOWER SURFACE					
BOTH ENGINES FULL THROTTLE	—	33.2	CENTER	.309	6700
				.317	6500
				.342	6400
LEFT ENG. STOPPED; RT. ENG. FULL THROT.	—	33.2	CENTER	.287	6900
				.293	6900
				.323	6700

TABLE V

SUMMARY OF RESULTS OF PROFILE DRAG TESTS ON NACA 35-215 AIRFOIL

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

POWER CONDITION	C_L	$R/10^6$	C_{d_0}
BOTH SURFACES			
BOTH ENGINES STOPPED	.208	31.5	.0050
	.232	29.7	.0047
	.256	27.3	.0049
	.256	28.3	.0050
	.258	28.0	.0048
	.260	27.3	.0048
	.260	27.6	.0047
	.288	26.0	.0048
	.300	25.3	.0049
	.322	24.3	.0048
BOTH ENGINES FULL-THROTTLE	.214	29.7	.0051
	.220	31.7	.0053
	.249	27.5	.0049
	.267	30.1	.0053
	.282	26.1	.0052
	.311	24.3	.0053
UPPER SURFACE ALONE			
BOTH ENGINES STOPPED	.226	27.5	.0023
	.227	29.2	.0020
	.236	29.0	.0022
	.258	27.3	.0022
	.262	26.1	.0023
	.270	26.0	.0025
	.293	24.5	.0026
	.322	23.6	.0028
BOTH ENGINES FULL THROTTLE	.213	31.4	.0025
	.226	27.6	.0023
	.258	25.9	.0025

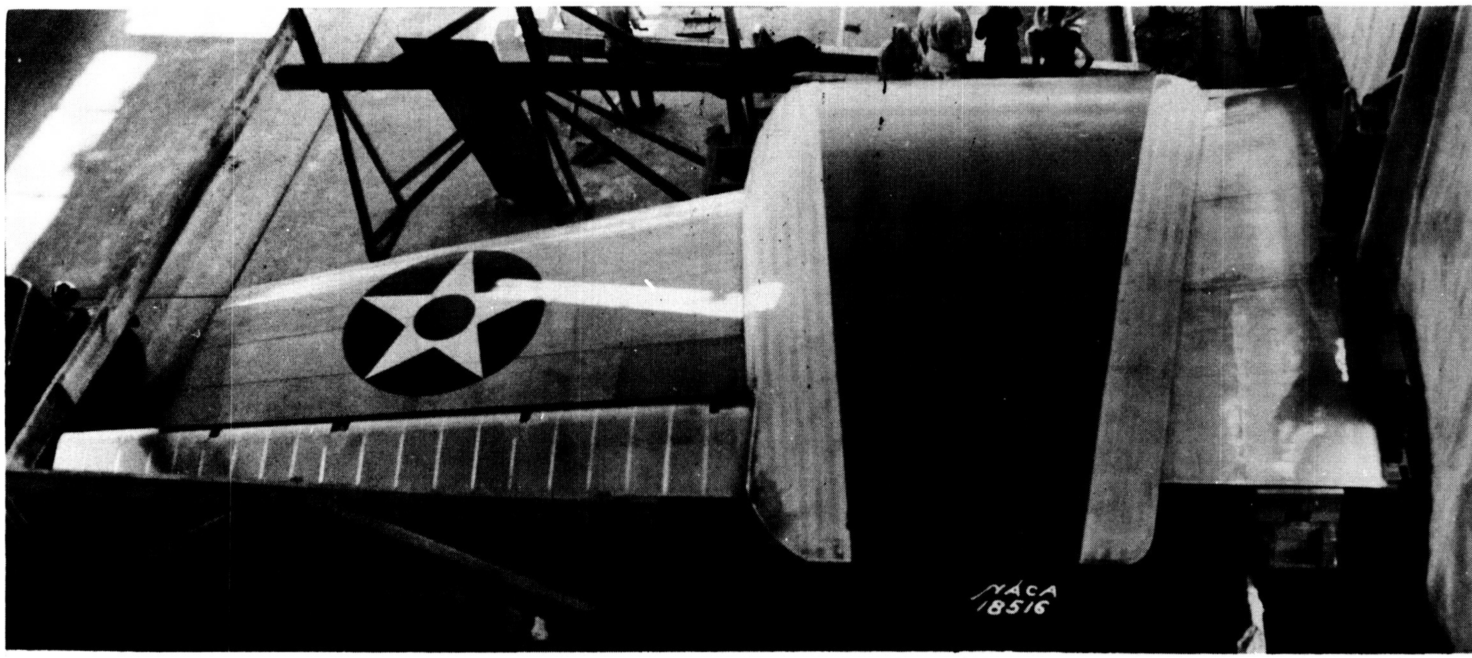

Figure 1.- NACA 35-215 test panel mounted on wing of a Douglas B-18 airplane.

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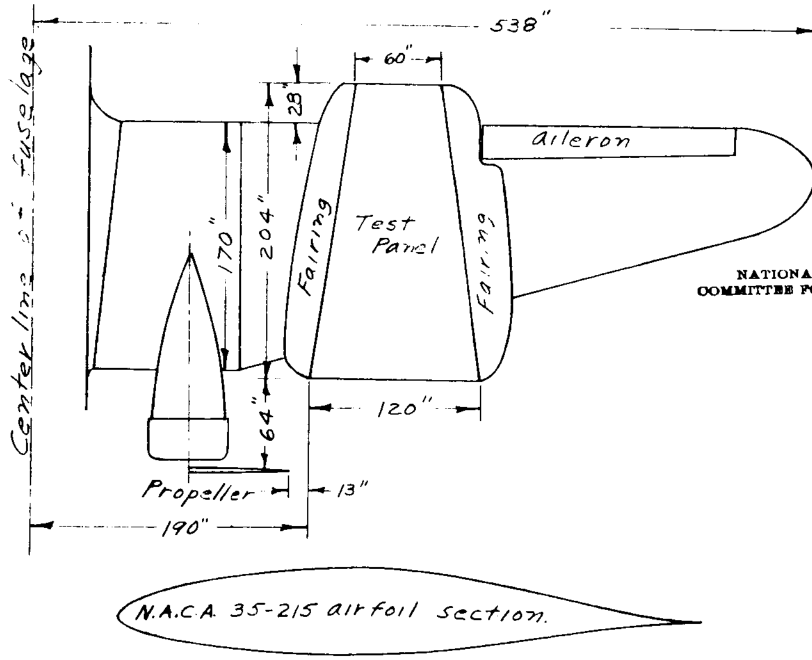

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Figure 2. Sketch showing position of test panel on wing of Douglas B-18 airplane and profile of N.A.C.A 35-215 airfoil section.

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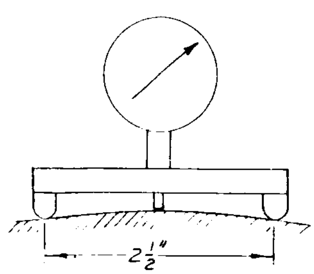

Figure 3. Sketch of curvature gauge used in making surface waviness measurements.

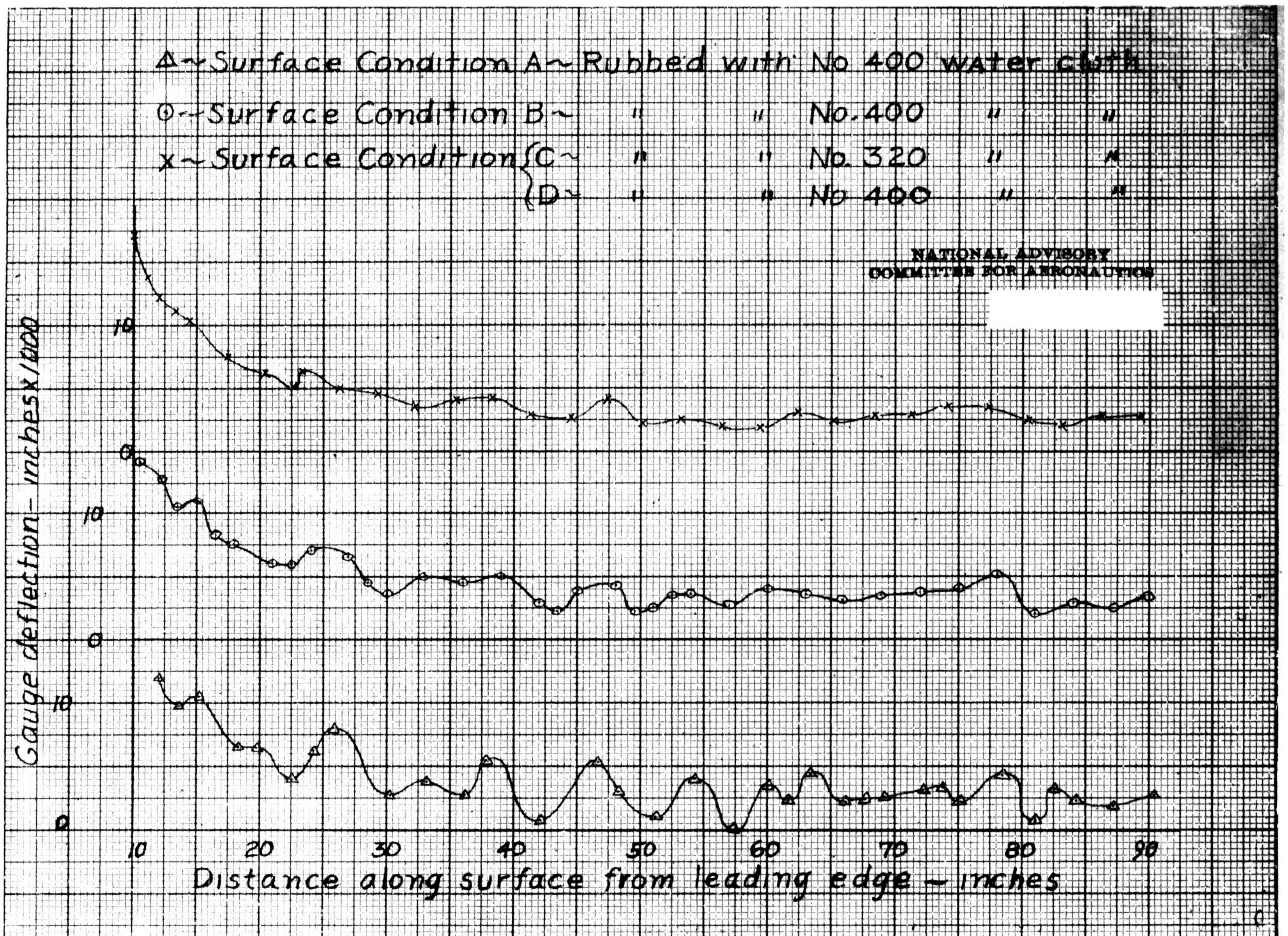

Figure 4. Measurements of surface waviness on upper surface of 35-215 test panel.

L-532

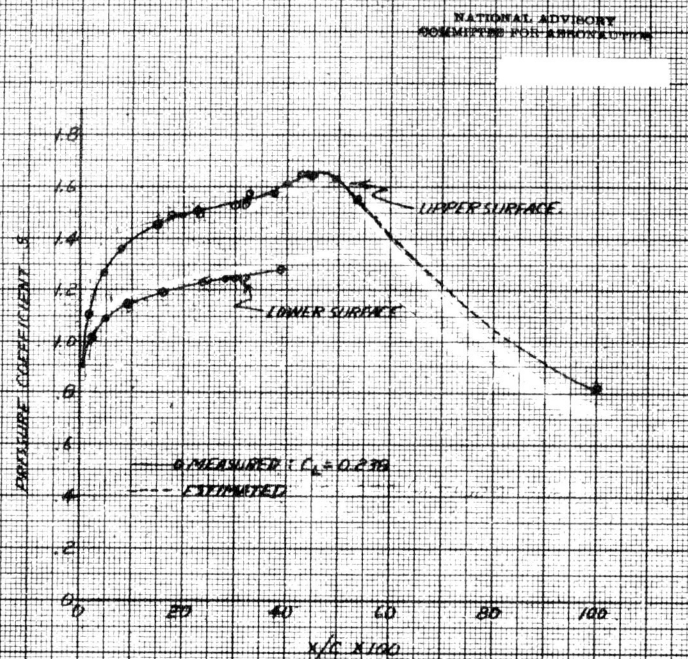

FIG. 5 Pressure distribution over NACA 35-215 test panel.

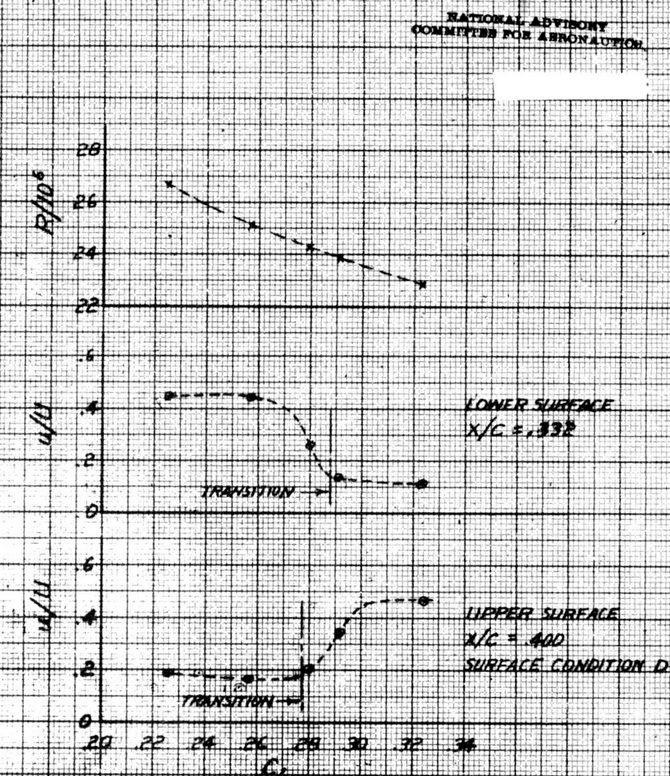

FIG. 6. ILLUSTRATION OF METHOD OF DETERMINING CONDITIONS AT WHICH TRANSITION OCCURS. 35-215 TEST PANEL; LEFT ENGINE STOPPED; y = 0.08 INCHES.

L-532

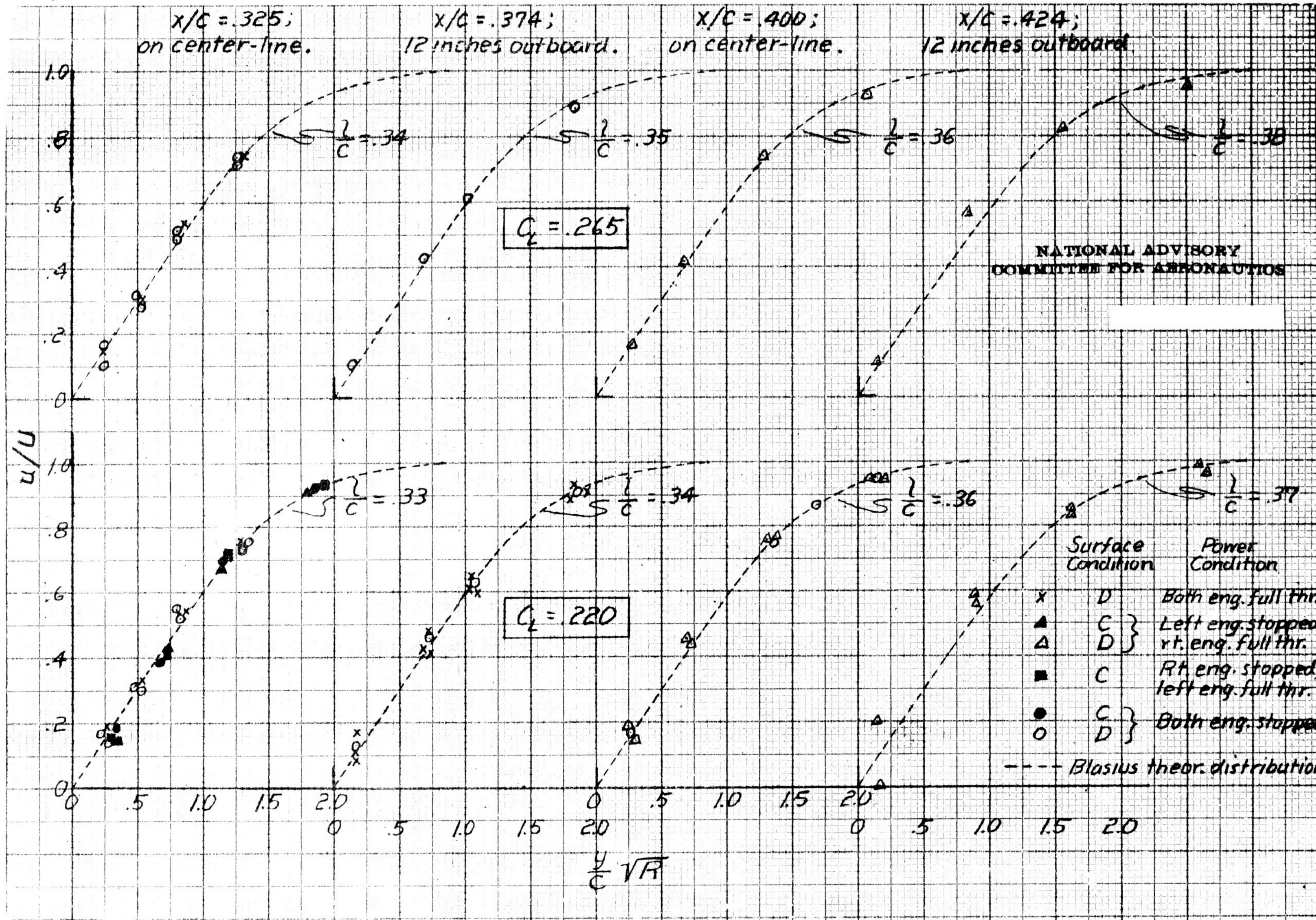

Fig. 7 Velocity distribution in the laminar boundary layer over upper surface of NACA 35-215 test panel. l is flat plate length corresponding to Blasius profile.

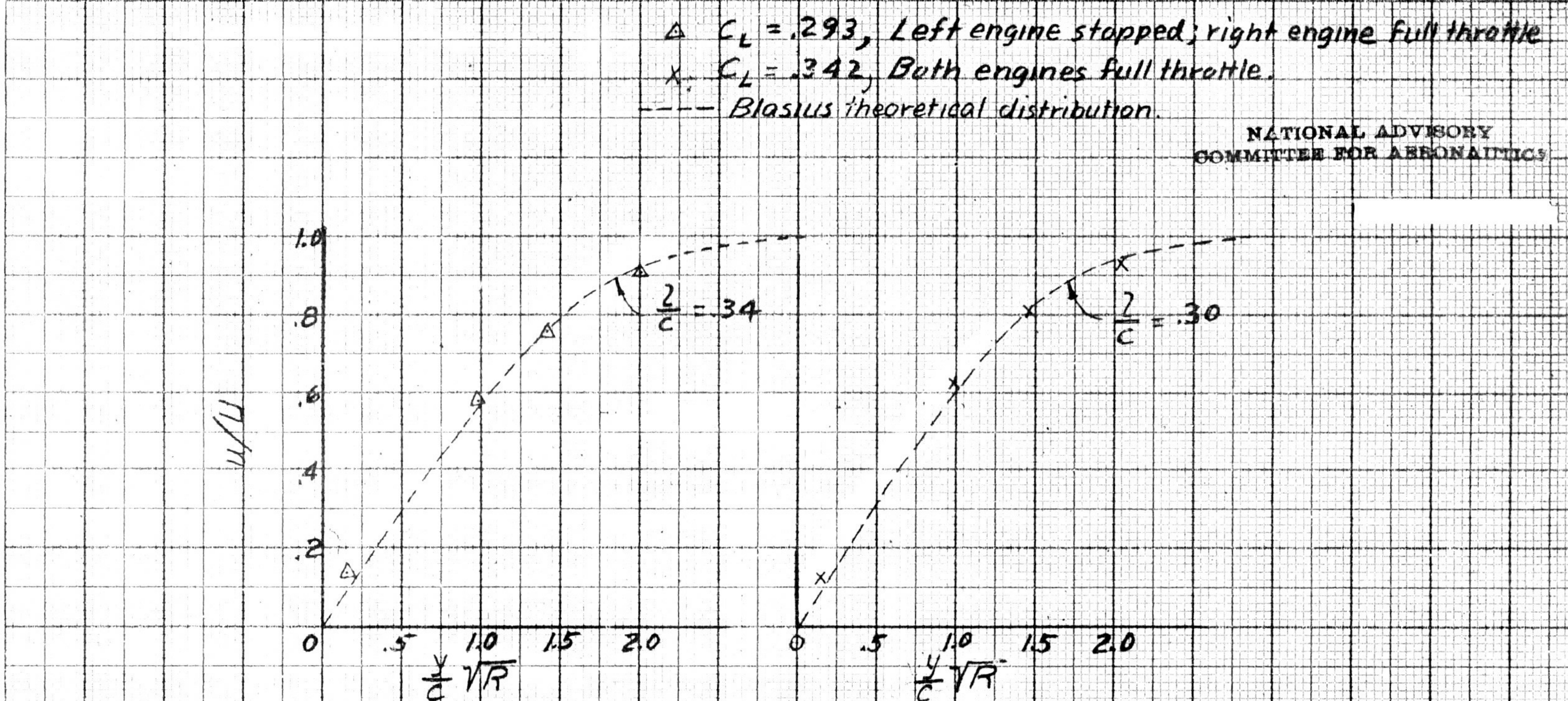

Fig. 8 Velocity distribution in the laminar boundary layer over lower surface of NACA 35-215 test panel. Measured at x/c = .333 on center-line. l is flat plate length corresponding to Blasius distribution.

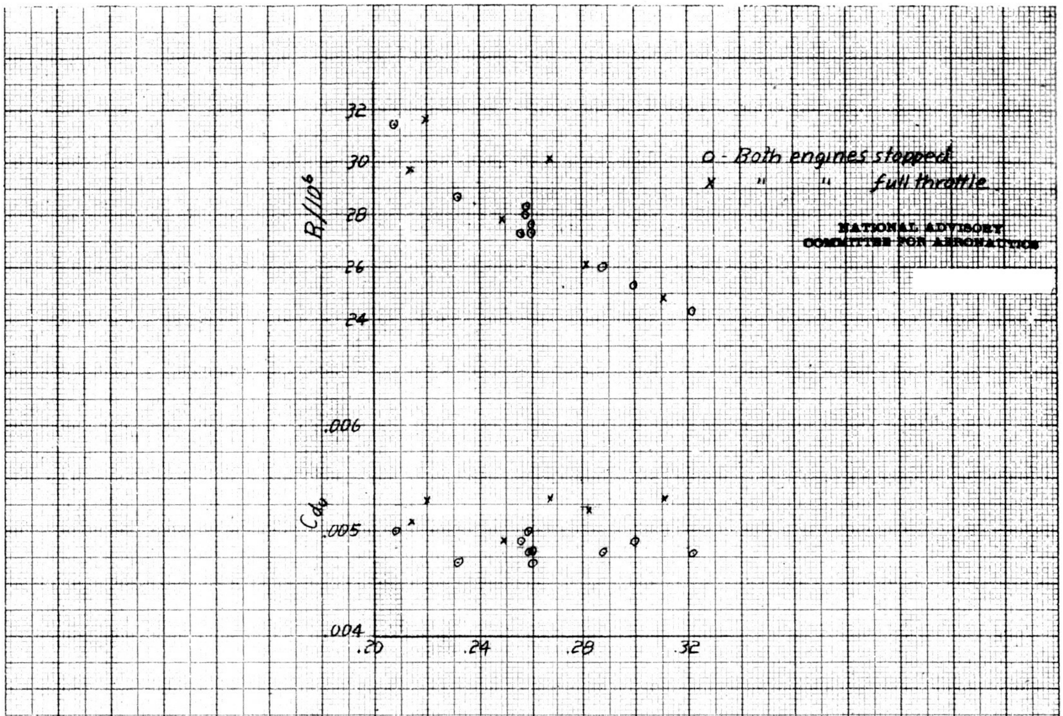

Fig. 9 Profile-drag coefficients of NACA 35-215 test panel.

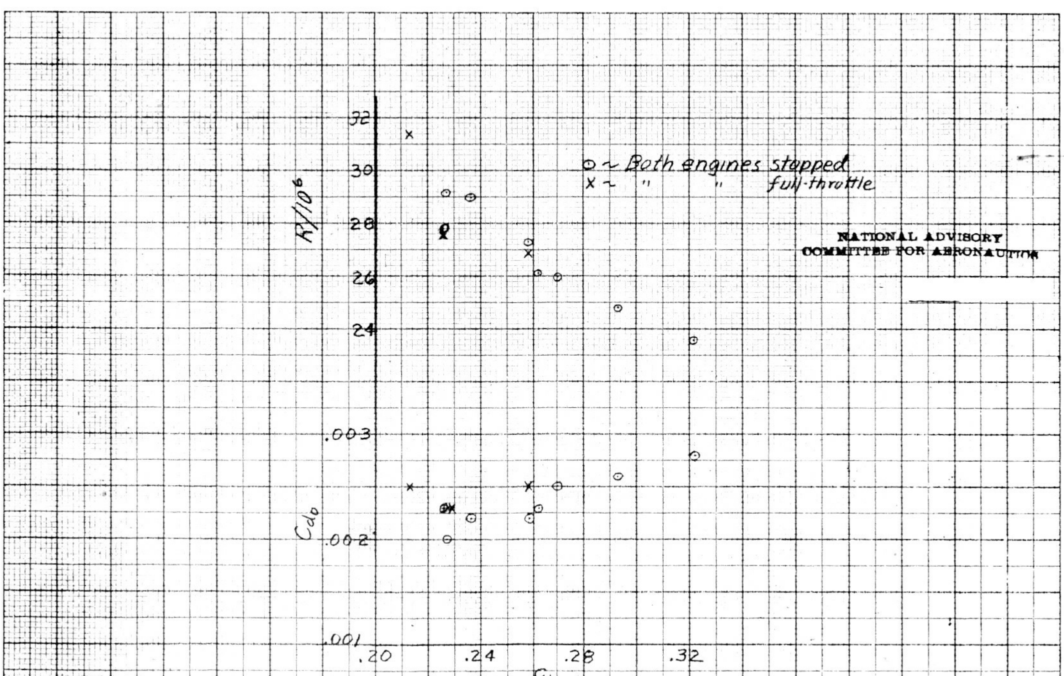

Figure 10 Profile-drag coefficient of upper surface of N.A.C.A 35-215 test panel.