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PRELIMINARY STUDY OF EFFECTS OF REYNOLDS NUMBER AND BOUNDARY-LAYER TRANSITION LOCATION ON SHOCK-INDUCED SEPARATION

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PRELIMINARY STUDY OF EFFECTS OF REYNOLDS NUMBER AND BOUNDARY-LAYER TRANSITION LOCATION ON SHOCK-INDUCED SEPARATION*

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SUMMARY

A two-dimensional experimental and theoretical investigation has been conducted on an NACA 65_1-213 airfoil to determine the effect of Reynolds number and transition location on shock-induced separated flow. The experimental investigation was conducted in the Langley 8-foot transonic pressure tunnel at Mach numbers from 0.60 to 0.80, angles of attack from 0° to 4°, and Reynolds numbers from 3.0×10^{6} to 16.8×10^{6}. Transition locations from 0.05 to 0.50 chord were utilized.

The results indicate that variation of the Reynolds number from full-scale values to the usual wind-tunnel values results in substantial changes in the shock location, trailing-edge pressure recovery, and boundary-layer losses at the trailing edge. By properly fixing the boundary-layer transition point on the wind-tunnel model, full-scale results can be simulated at the usual wind-tunnel Reynolds numbers. The required location of the transition point can be predicted theoretically with acceptable accuracy by simulating the boundary-layer characteristics at the airfoil trailing edge.

INTRODUCTION

Correlations of existing data obtained at the usual wind-tunnel values of Reynolds number on wind-tunnel scale models and of data obtained during flight tests at much higher Reynolds numbers indicate that at speeds where substantial shock-induced separation is present, scale effects may occur. In particular, a large Reynolds number difference may affect the shock-wave location on the model and the presence and extent of shock-induced separation. An example of the problem on a large transport airplane is shown in figure 1 (reproduced from ref. 1). The differences between the wind-tunnel and the full-scale pressure coefficient results on the wing clearly demonstrate the large-scale effects that may occur at supercritical speeds.

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The basic phenomena of scale effects at speeds where shock-induced separation is present are illustrated schematically in figure 2. In flight, at large Reynolds numbers, the boundary layer becomes turbulent near the airfoil leading edge. At transonic Mach numbers, a shock wave forms and moves rearward with increasing Mach number. When the shock is sufficiently strong, the shock induces the boundary layer to separate. This phenomenon is referred to as "shock-induced separation." Presently, for most wind-tunnel investigations, the boundary layer is made turbulent near the leading edge of the airfoil with a fixed boundary-layer transition strip. Since the difference in wind-tunnel and flight Reynolds numbers may be large and the relative thickness of the turbulent boundary layer varies inversely as approximately Reynolds number to the 1/5 power, the relative thickness at any given pércent chord station is greater on a small-scale wind-tunnel model with transition fixed near the leading edge than on a similar full-scale wing with natural transition near the leading edge in flight. Because of the greater thickness of the wind-tunnel boundary layer, the flow separates as a result of the shock wave at a lower Mach number and is more severe than in flight. The separated flow causes a greater displacement of the flow and a loss in circulation. For a given Mach number and angle of attack, this displacement may result in the shock being farther forward on the wind-tunnel model than in flight.

No general procedure has been developed for correcting wind-tunnel data to flight conditions when shock-induced separation is present. Therefore, an experimental wind-tunnel technique is desirable to minimize scale effects for these conditions. The results presented in reference 1 indicate that by moving the fixed boundary-layer transition strip rearward on the wind-tunnel model when the flow has separated, the shock position on the model approached the same location as on the flight airplane. Based on these results, it was reasoned that a technique that could be used to minimize scale effects in shock-induced separated flow might lie in fixing the boundary-layer transition point at the proper location on the wind-tunnel model so that the boundary layer encountered in flight is simulated on the wind-tunnel model in the region of separation.

In order to substantiate this technique and to obtain a better understanding of the factors involved in minimizing the effects of Reynolds number when shock-induced separated flow is present, a comprehensive two-dimensional wind-tunnel experimental investigation and an applied theoretical investigation have been conducted over a wide range of Reynolds numbers with various transition locations.

SYMBOLS

a mean-line designation

b span of wing, feet (meters)

c_l	section lift coefficient
C_p	pressure coefficient, $\frac{p - p_\infty}{q_\infty}$
$C_{p,sonic}$	pressure coefficient corresponding to local Mach number of 1.0
c	chord of airfoil, inches (centimeters)
H	boundary-layer shape factor, δ^*/θ
M	free-stream Mach number
p	local static pressure, pounds/square foot (newtons/meter^2)
p_∞	static pressure in undisturbed stream, pounds/square foot (newtons/meter^2)
Δp_t	total-pressure loss, pounds/square foot (newtons/meters^2)
q_∞	dynamic pressure in undisturbed stream, pounds/square foot (newtons/meter^2)
R	Reynolds number based on local chord
x	ordinate along airfoil reference line measured from airfoil leading edge, inches (centimeters)
z	ordinate vertical to airfoil reference line, inches (centimeters)
α	geometric angle of airfoil reference line, degrees
δ^*	boundary-layer displacement thickness, inches (centimeters)
θ	boundary-layer momentum thickness, inches (centimeters)

Subscripts:

T	denotes boundary-layer transition-strip location
wt	wind tunnel

fs full scale

f fuselage

EXPERIMENTS

Wind Tunnel

The experiments were conducted in the Langley 8-foot transonic pressure tunnel which is well suited to the investigation of the effects of Reynolds number on two-dimensional models at transonic speeds. The 8-foot tunnel is a variable-pressure tunnel and allows investigations to be conducted over a wide range of Reynolds numbers with the same model. Also substantially larger chord models may be tested in this facility than could otherwise be tested in tunnels designed for two-dimensional testing since the wind tunnel is approximately 7 feet (2.80 m) in height. Good results may be obtained in this facility for large chord models since the side walls are solid and the upper and lower walls are slotted. This arrangement allows a development of the flow field in the vertical direction approaching that for free flight. The slot opening at the position of the model was approximately 6.4 percent of the upper and lower surface walls.

Model

The two-dimensional model investigated is shown in figure 3. The model was 3 feet (0.91 m) in length and was tested in an inverted position. The airfoil is the NACA 65_1-213 with $a = 0.5$ mean line. (See table I and fig. 4.) This airfoil shape was selected for several reasons. It is the same shape as the mid semispan section of the T-33 airplane for which flight data are available (ref. 2). A sketch of the T-33 airplane is shown in figure 4. Since the T-33 wing has essentially no sweep (sweep of quarter chord is approximately 4.5°), has a constant airfoil section along the span, and has a high aspect ratio, correlation with two-dimensional data at the mid semispan was expected to be good. The 6-series airfoil was also selected because it represents a class of airfoils that have been recently used in high-subsonic-speed airplane design.

Transition Strips

Boundary-layer transition strips were located at the same position on both the upper and lower surfaces of the model. Results were obtained for transition strips located at 5-, 20-, 30-, 40-, and 50-percent chord. The strips were 0.1 inch (0.25 cm) wide and consisted of carborundum grains set in a plastic adhesive. The size of the carborundum grains for each location was calculated by using reference 3.

Measurements

The lift force acting on the airfoil was obtained from model surface pressure measurements along the center line of the tunnel.

The boundary-layer data presented herein were derived from measurements taken with a total-pressure rake located at the trailing edge of the model. The total-pressure tubes were flattened horizontally and closely spaced.

The total pressure and static pressures were measured with the use of electronically actuated pressure scanning valves. The full-scale range of the gages in the valves was varied, depending on type of measurement and on the wind-tunnel conditions.

Corrections

The primary effect of the wind-tunnel wall on the results presented herein is a substantial upflow at the position of the inverted model so that the real aerodynamic angle of attack is significantly less than the geometric angle. The mean value of this upflow at the midchord of the model, in degrees as determined by the theory of reference 4, is approximately 4.4 times the section lift coefficient. For the design section lift coefficient of 0.20, this angle deviation is approximately -0.88°. For the present investigation where the lift has been obtained by surface pressure measurements, this deviation has little effect on the validity of these results. It merely causes a change of the geometric angle of attack at which a given set of results is obtained. The angles of attack used in the results presented herein have not been corrected for this upflow.

No corrections have been made for tunnel-wall blockage since the theory of reference 4 indicates this effect is small.

Range of Tests

The investigation was conducted over a Mach number range from 0.60 to 0.80. The angle of attack varied generally from about 0° to 4° in 1° increments. The Reynolds number of the investigation was varied from 3.0×10^6 which approximates the lowest values usually used for wind-tunnel investigations to 16.8×10^6 which is near the Reynolds number for which full-scale flight results were obtained on the T-33 airplane. (See ref. 2.)

DISCUSSION

Wind-Tunnel Results

Flight—wind-tunnel correlation.- A comparison of the wind-tunnel results and flight results from the T-33 airplane (ref. 2) is presented in figure 5 for a Mach number of 0.8. Other comparisons were made; however, this example may be considered typical of the results obtained. The flight data were obtained at a Reynolds number of 19×10^6 based on the local chord of 6.40 feet (1.95 m) and the two-dimensional wind-tunnel results are presented for a Reynolds number of 16.8×10^6. The transition strip was fixed near the leading edge of the two-dimensional airfoil at 5-percent chord since this position was thought to be representative of the natural transition location of the upper surface flight results. The comparison shown in figure 5 indicates generally good agreement between the flight results and the wind-tunnel results obtained at full-scale Reynolds numbers. Therefore, for the subsequent analysis, the wind-tunnel data taken at 16.8×10^6 are considered to be representative of full-scale results. It should be noted that for the condition presented, a small amount of shock-induced separation is present.

Effect of Reynolds number and transition location.- In figure 6, the effects of Reynolds number and boundary-layer transition location on the section aerodynamics are presented. The results shown indicate (1) the effect of increasing the Mach number at a constant angle of attack ($\alpha = 0^\circ$) from subcritical speeds to a condition with shock-induced separation and (2) the effect of increasing the angle of attack so that shock-induced separation occurs. It is felt that these examples are representative of the data obtained during the investigation. The data presented include pressure distributions on the airfoil and profiles of the boundary-layer total pressure loss $\left(\frac{\Delta p_t}{q_\infty}\right)$ at the trailing edge. Results are presented only for the airfoil upper surface in order to simplify the analysis. The upper surface is generally the most critical for shock-induced separation; however, all conclusions reached regarding the effect of Reynolds number and transition-strip location on shock-induced separation on the airfoil upper surface also apply to the lower surface.

For each comparison, three sets of data are shown. The first set of data was obtained at 16.8×10^6 with the transition fixed at 5-percent chord and represents what are referred to as full-scale results. The second set of data represents data taken at the usual wind-tunnel Reynolds numbers of 3×10^6 with the transition fixed near the leading edge (0.05c). As previously indicated, this procedure is the wind-tunnel technique presently in general use. The third set of data represents data obtained at wind-tunnel Reynolds numbers with the point of transition moved rearward of the leading edge and fixed at a location that best approximates full-scale results.

The results at subcritical speeds (M = 0.70) for an angle of attack of 0° are presented in figure 6(a). These results indicate that at the same transition-strip location (0.05c) for full-scale and wind-tunnel Reynolds numbers, there are only small variations in the pressure distribution over the airfoil. However, it should be noted that for the same transition-strip location (0.05c), the trailing-edge pressure recovery is less than that for the wind-tunnel Reynolds number. Also, as would be expected, the boundary-layer profiles indicate a thicker boundary layer at the airfoil trailing edge for the wind-tunnel Reynolds number. When the transition is moved rearward to the position for the best correlation of the trailing-edge pressure recovery for the high and low Reynolds number (40-percent chord), the boundary-layer profiles also are the same.

The effects of Reynolds number and transition location when the Mach number is increased from 0.70 to 0.80 at $\alpha = 0^{\circ}$ are shown in figure 6(b). For the full-scale case, boundary-layer separation has been induced by the strong shock wave. However, for the data obtained at a Reynolds number of 3×10^{6} with the transition fixed near the leading edge, the separation is substantially greater, the shock wave is farther forward on the airfoil, and the trailing-edge pressure is decreased. When the transition point is moved rearward at wind-tunnel Reynolds numbers, the trailing-edge pressure coefficients, the shock-wave location, and the trailing-edge boundary-layer profiles are all in good agreement with full-scale results for the transition strip located at approximately 45-percent chord. It should be noted that the data presented at $\frac{x_T}{c} = 0.45$ are interpolated from data obtained at $\frac{x_T}{c} = 0.40$ and 0.50 which are not presented.

As another illustration of the effect of Reynolds number and transition location, data are shown for an increase in the angle of attack to 3° at a Mach number of 0.75 in figure 6(c). As in the previous example, good agreement is obtained between the data for the full-scale case and for wind-tunnel Reynolds numbers when the transition strip is moved rearward. For this case, the transition-strip location for best agreement was at the 40-percent chord.

Based on these results and other data not presented, it is concluded that for conditions at which shock-induced separation is present, good agreement can be obtained between full-scale data and results obtained at wind-tunnel Reynolds number with the proper location of the transition strip. However, the results also show that the transition-strip location for the best agreement varies somewhat with changes in the test conditions.

Theoretical Analysis

To allow general utilization of the experimental approach to simulating full-scale shock-induced boundary-layer separation characteristics by moving the boundary-layer transition location, a method must be developed to predetermine the transition-strip location for any airplane or flight condition without resort to experiments. To do so requires a more complete understanding of the fundamental factors governing the boundary-layer development. In order to provide some insight to this problem, a limited analysis using existing theory was undertaken.

The experimental results (fig. 6) indicated that even with shock-induced separation present, the transition-strip locations for best agreement between data at wind-tunnel Reynolds numbers and full-scale Reynolds numbers do not vary appreciably from the transition-strip location for subcritical speeds (40-percent chord). It therefore appeared that an analysis based on subcritical pressure distributions and boundary-layer theory might be applicable. Various theories (refs. 5 to 8) were considered for the boundary-layer analysis. It was found that the results were not significantly different when the theories investigated were used. For the following study, the boundary-layer calculations are based on reference 5 for the laminar portion and reference 8 for the turbulent portion.

Theoretical boundary-layer characteristics for subcritical condition.- By using the subcritical pressure distribution in figure 6(a), theoretical boundary-layer characteristics (δ^* and H) are calculated with these theories and are presented in figure 7. Data are presented for the conditions representing the full-scale results and the wind-tunnel results with the transition strip located at 5-percent and 40-percent chord, the 40-percent-chord transition location being the experimental condition for best wind-tunnel—full-scale correlation of the trailing-edge pressure recovery and boundary-layer profiles. At the trailing edge, the theory also indicates good agreement of the boundary-layer characteristics between the calculations at full scale and the calculations with the transition point fixed at 40-percent chord for wind-tunnel Reynolds numbers.

From the distribution of the theoretical displacement thickness over the chord, it is obvious that the δ^* distribution at the wind-tunnel Reynolds numbers with transition at 40-percent chord is a good approximation to the displacement thickness at full-scale Reynolds numbers over the critical rear portion of the airfoil. The large effect of Reynolds number and transition location on the boundary-layer characteristics is indicated by comparing the full-scale results and wind-tunnel Reynolds number results at $\frac{x_T}{c} = 0.40$ with the wind-tunnel Reynolds number characteristics for the transition located at 5-percent chord.

On the right-hand side of figure 7, the theoretical boundary-layer shape factor H which indicates the boundary-layer separation characteristics (ref. 5) is presented over the rear part of the airfoil. With the transition fixed at 40-percent chord for wind-tunnel Reynolds numbers, the data indicate that the full-scale separation characteristics are adequately simulated over the rear part of the airfoil, in particular they are matched at the airfoil trailing edge. This result is significant since observation of oil-flow patterns during the investigation indicates that the separation resulting from the shock wave initially occurs at the airfoil trailing edge and not at the shock wave. The shape factor indicates the flow for the transition fixed at 5-percent chord for wind-tunnel Reynolds numbers to be substantially nearer separation than that for the full-scale results.

Basic criteria.- Since shock-induced separation has been observed experimentally to occur initially at the trailing edge, the assumed criterion for best wind-tunnel—full-scale correlation in the theoretical analysis to determine the transition location is to match the boundary-layer characteristics at the trailing edge of the wind-tunnel model and at the trailing edge of the full-scale wing. Further, since the primary interest is separation, the transition locations are such that the values of H for wind-tunnel and flight conditions are made equal at the trailing edge. Throughout the following analysis this approach is referred to as the "trailing-edge criterion."

Variation of theoretical transition-strip location with Mach number and pressure distribution.- The experimental transition-strip locations for best wind-tunnel—flight correlation (figs. 6(a) and 6(b)) varied as the free-stream Mach number was increased to supercritical speeds for a constant angle of attack. Therefore, it appears that the transition-strip location for best wind-tunnel—full-scale correlation might be sensitive to the free-stream Mach number or to the change of the shape of the pressure distribution as Mach number is increased. In order to determine the sensitivity of the transition-strip location, theoretical calculations were made by using the trailing-edge criterion to determine the theoretical transition location for a range of pressure distributions for various Mach numbers. The pressure distributions considered are shown in figure 8. The primary variables are (1) shape of leading-edge pressure distribution, (2) trailing-edge pressure, and (3) shape of the aft end pressure recovery. The shape for the pressure distribution with the favorable gradient over the forward portion of the airfoil (solid line) is typical for near zero lift conditions such as obtained in a dive. The rooftop pressure distribution (long- and short-dashed line) represents the conditions generally expected in flight at the higher lift conditions. The variations of the pressure distributions over the aft section of the airfoil are similar to those noted experimentally on various types of airfoils.

By use of the subcritical pressure distributions shown in figure 8, the transition-strip locations were calculated theoretically by using the trailing-edge criterion for various Mach numbers with a full-scale Reynolds number of 16.8×10^6 and a wind-tunnel Reynolds number of 3.0×10^6. The results indicated very little effect of Mach number on transition locations (less than 1 percent). Also the small effect of Mach number on the theoretical transition location appears to be independent of the pressure distribution, since the same conclusions as obtained for the present analysis may be reached by using the flat-plate theory of reference 9.

The effects of the changes in pressure distribution shape on the theoretical transition location have been calculated for a Mach number of 0.70 and are presented as vertical lines on the horizontal scale of figure 8. On the basis of this limited analysis, it can be seen that the changes in the shape of the pressure distributions over the forward part of the airfoil produce significant variations in the theoretical transition locations. However, for the examples shown, the changes in the aft end distributions have only slight effects on these locations.

An analysis of the basic factors involved suggests that the primary influence on the theoretical transition location is the variation of the pressure distribution in the region of transition shift, that is, where the boundary layer is turbulent for the full-scale Reynolds number case and laminar for wind-tunnel Reynolds number condition.

Comparison of Experimental and Calculated Results

The preceding theoretical analysis suggests that for conditions where severe shock and separation are not present, at least, correlation of experimental and calculated transition locations at wind-tunnel Reynolds numbers might be achieved when the pressure distributions over the region of transition shift are similar. For the $M = 0.70$, $\alpha = 0^{\circ}$ experimental case (fig. 6(a)), the pressure distribution over the region of transition shift is similar to that for the schematic distribution of the theoretical analysis as shown by the dashed line in the left-hand side of figure 8. The best experimentally determined transition location (0.40 chord) is the same as the calculated position.

Cases where substantial shocks and separation are present are now considered. For the $M = 0.75$, $\alpha = 3^{\circ}$ experimental case (fig. 6(c)), the pressure distribution over the region of transition shift is also similar to that of the dashed line on the left-hand side of figure 8. Again, the best experimental transition location is the same as that calculated (0.40 chord). For the $M = 0.80$, $\alpha = 0^{\circ}$ experimental condition (fig. 6(b)) the pressure distribution from 0.05 to 0.45 chord is similar to that for the theoretical analysis (solid line). Again, the experimental transition location (0.45 chord) is the same as the calculated value. In these two cases, correlation is achieved even though the nature of the actual flow is substantially different from the assumptions of the theory used for the calculations. The reasons for this agreement are not yet fully understood.

Application

The agreement of the calculated and measured transition locations for simulating full-scale characteristics suggests that the theoretical trailing-edge criterion is a reasonable approach to the prediction of the theoretical transition location. This agreement also indicates that in the calculations only the pressure distribution over the region of transition shift significantly affects the required transition location.

Effect of ratio of wind-tunnel Reynolds number to full-scale Reynolds number.- In order to indicate the variation of the theoretical transition location on the wind-tunnel model for various full-scale Reynolds numbers, figure 9 is presented based on a wind-tunnel Reynolds number of 3×10^{6}. Theoretical calculations indicate that only small variations occur (order of 1 percent) in the results shown for realistic changes in the reference wind-tunnel Reynolds number for a given Reynolds number ratio. Variations representing the two extremes of pressure gradients over the forward portion of the airfoil in figure 8 are shown. Also shown in figure 9 is the transition location curve calculated by using flat-plate theory for momentum thickness (no pressure gradient).

Effect of variations of transition location from the optimum.- In conducting a wind-tunnel test, it is not always convenient or practical to change the transition location with a change of test conditions. Therefore, an attempt has been made to assess the effects of varying from the optimum transition location. An indication of the effects is provided by crossplots of the experimental data (not presented) obtained for various locations. For the cases where shock-induced separated flow is present, a change in the transition of 5-percent chord produces a movement in the shock wave of approximately 1-percent chord.

Limitations of applicability.- Certain comments as to the limitations of the applicability of the method are warranted. Since laminar flow must be maintained ahead of the transition strip, the method is limited to classes of pressure distributions that do not have severe leading-edge peaks at supercritical speeds that would result in natural boundary-layer transition ahead of the transition strip. It should also be noted that the model must be maintained absolutely smooth in front of the transition strip to prevent transition of the boundary layer.

Results of a number of investigations conducted in the 8-foot transonic pressure tunnel have indicated that, when the transition strip is rearward as specified by the trailing-edge criterion and a strong adverse gradient is present ahead of the transition, more severe boundary-layer separation may be present than when the transition is in the normal location near the leading edge. For such conditions, more applicable results are obtained with the transition forward.

Results not presented indicate that when the transition strip is just ahead of the base of the shock, laminar separation occurs ahead of the transition strip. Thus, the maximum rearward movement for which applicable results can be obtained is limited. For the airfoil of the present investigation, the limit is approximately 50-percent chord.

CONCLUSIONS

A two-dimensional experimental and theoretical investigation has been conducted for an NACA 65_1-213 airfoil to determine the effect of Reynolds number and transition location on shock-induced separated flow. The results have led to the following conclusions:

1. Variation of the Reynolds number from full-scale values to the usual wind-tunnel values results in substantial changes of the shock location, trailing-edge pressure recovery, and boundary-layer losses at the trailing edge.

2. By properly fixing the boundary-layer transition point on the wind-tunnel model, full-scale results can be simulated at the usual wind-tunnel Reynolds numbers.

3. The required location of the boundary-layer transition strip can be predicted with acceptable accuracy by theoretically simulating the boundary-layer characteristics at the airfoil trailing edge. The primary influence on the theoretical transition location is the surface pressure distribution in the region where the boundary layer is turbulent for full-scale Reynolds numbers and laminar for the wind-tunnel Reynolds numbers.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., October 28, 1968,
126-13-01-67-23.

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TABLE I.- ORDINATES OF NACA 65_1-213 (a = 0.5) AIRFOIL

[All stations and ordinates in percent chord]

Station	Ordinate	Station	Ordinate
Upper surface		Lower surface	
0	0	0	0
.38	1.06	.62	-.92
.62	1.29	.88	-1.10
1.10	1.64	1.40	-1.35
2.34	2.28	2.66	-1.76
4.81	3.26	5.19	-2.38
7.31	4.02	7.69	-2.84
9.80	4.67	10.20	-3.22
14.81	5.71	15.19	-3.82
19.83	6.51	20.17	-4.26
24.86	7.12	25.14	-4.59
29.89	7.56	30.11	-4.82
34.92	7.85	35.08	-4.96
39.96	7.98	40.04	-5.01
45.01	7.94	44.99	-4.95
50.07	7.71	49.93	-4.77
55.11	7.26	54.89	-4.47
60.13	6.63	59.87	-4.07
65.14	5.89	64.86	-3.60
70.13	5.04	69.87	-3.06
75.11	4.14	74.89	-2.49
80.09	3.19	79.91	-1.88
85.06	2.24	84.94	-1.29
90.04	1.33	89.97	-.72
95.01	.53	94.99	-.24
100.00	0	100.00	0
Leading-edge radius, 1.174; slope of radius through leading edge, 0.084			

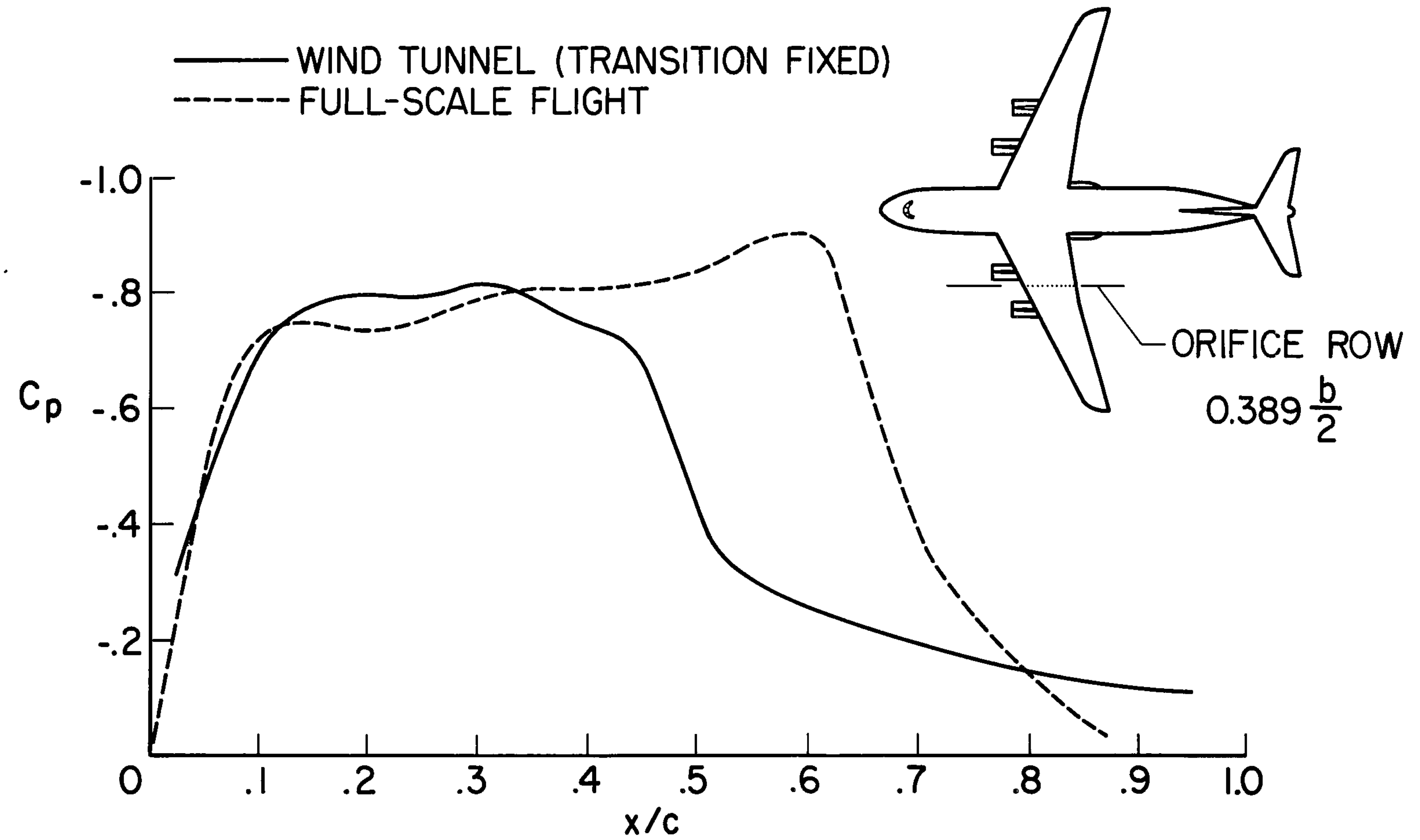


Figure 1.- Supercritical pressure distribution. $M = 0.85$; $\alpha_f \approx 0^0$.

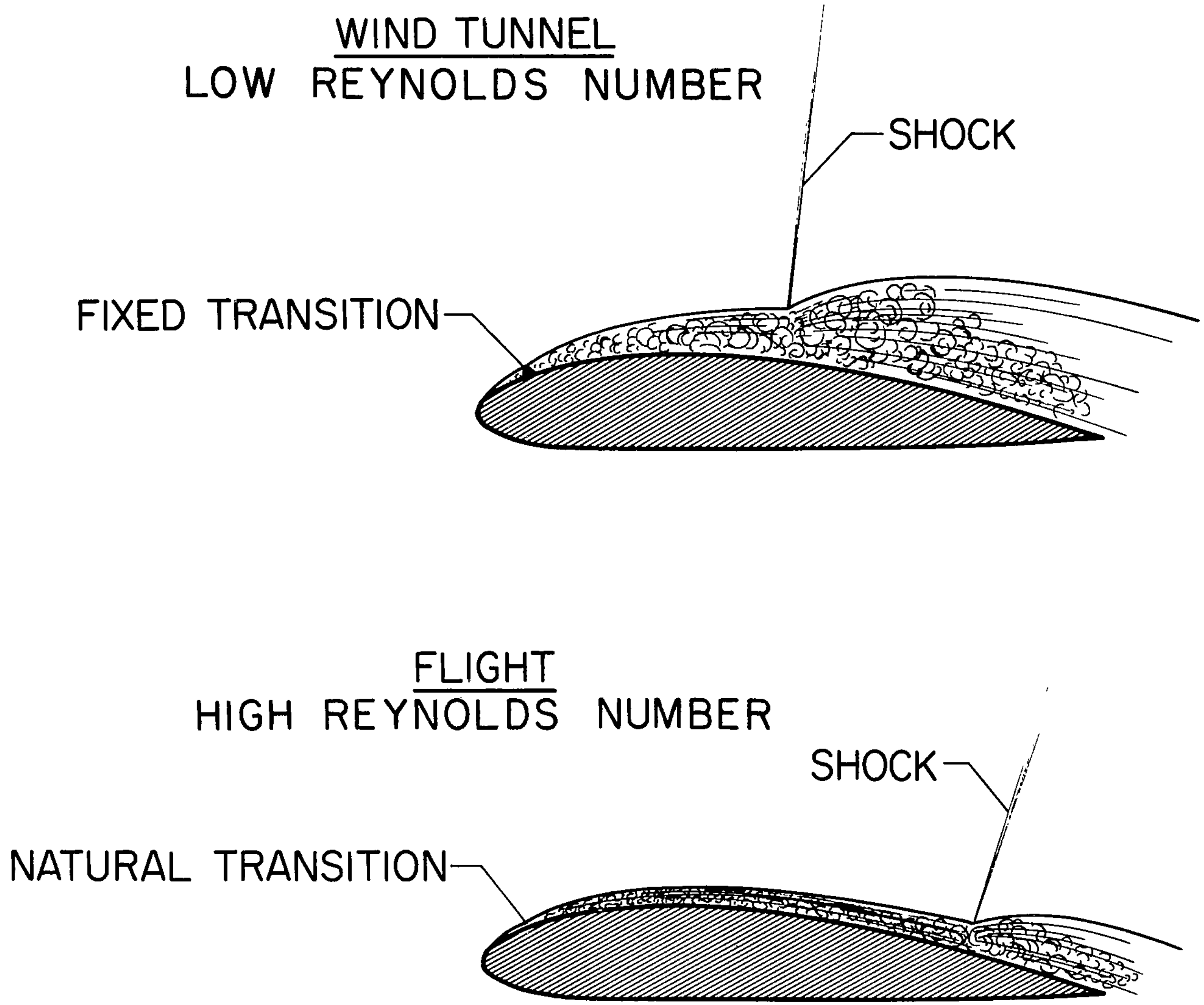


Figure 2.- Effect of boundary layer on shock-induced separation.

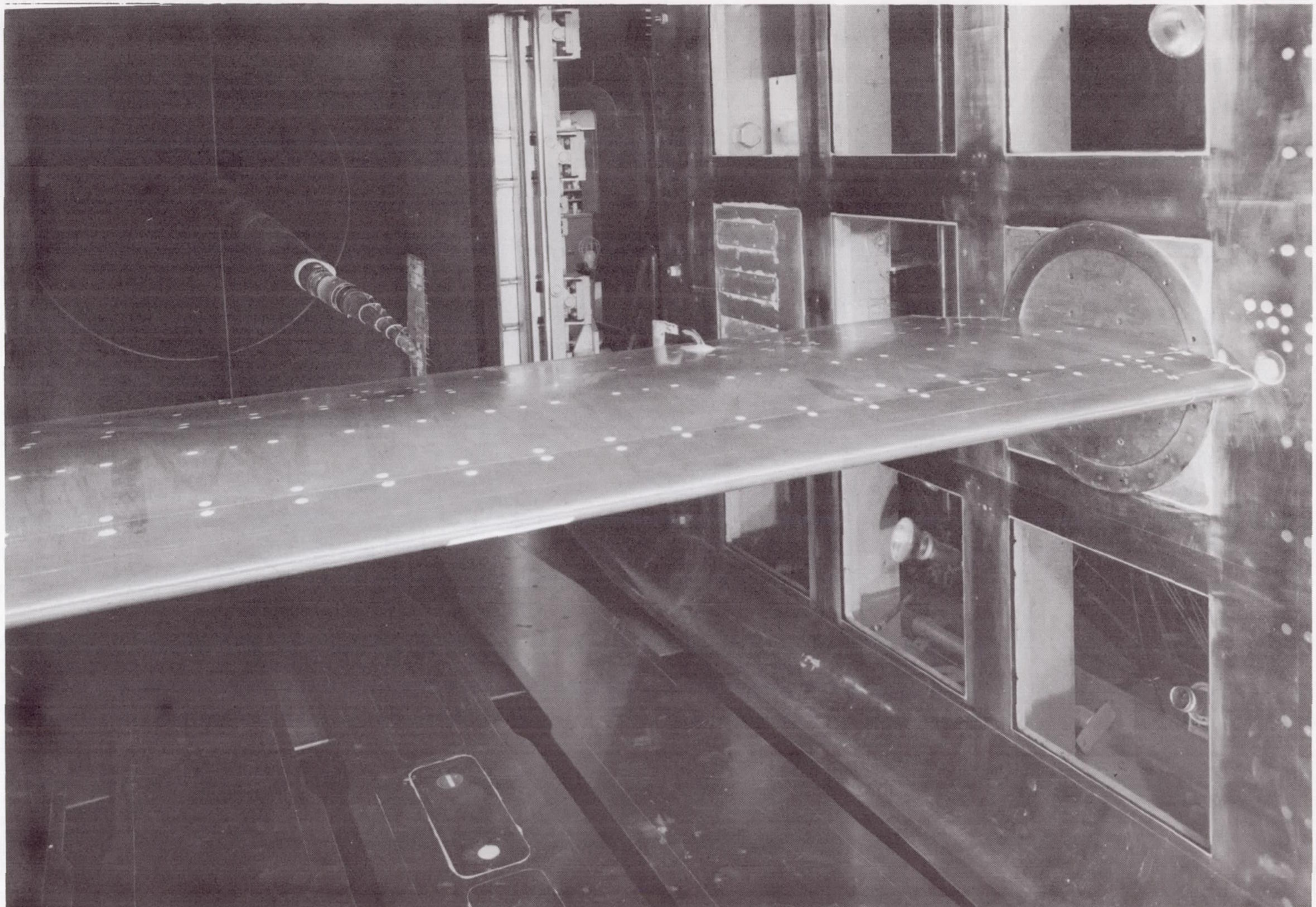
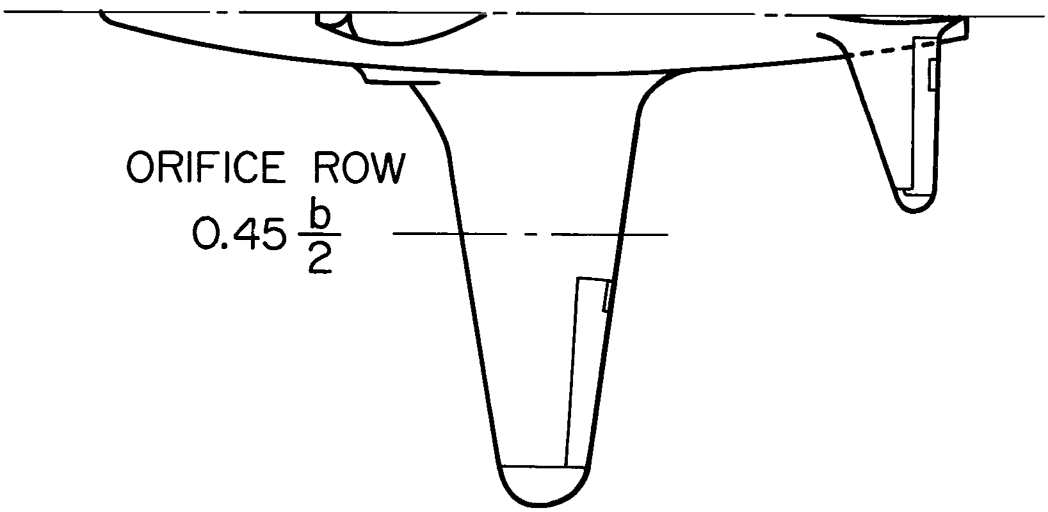


Figure 3.- Wind-tunnel installation of two-dimensional model. L-3085-3



NACA 65_1-213 AIRFOIL (a = 0.5)

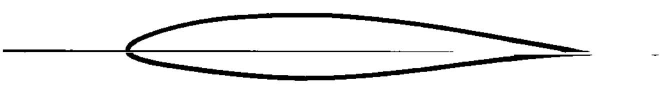


Figure 4.- Flight configuration.

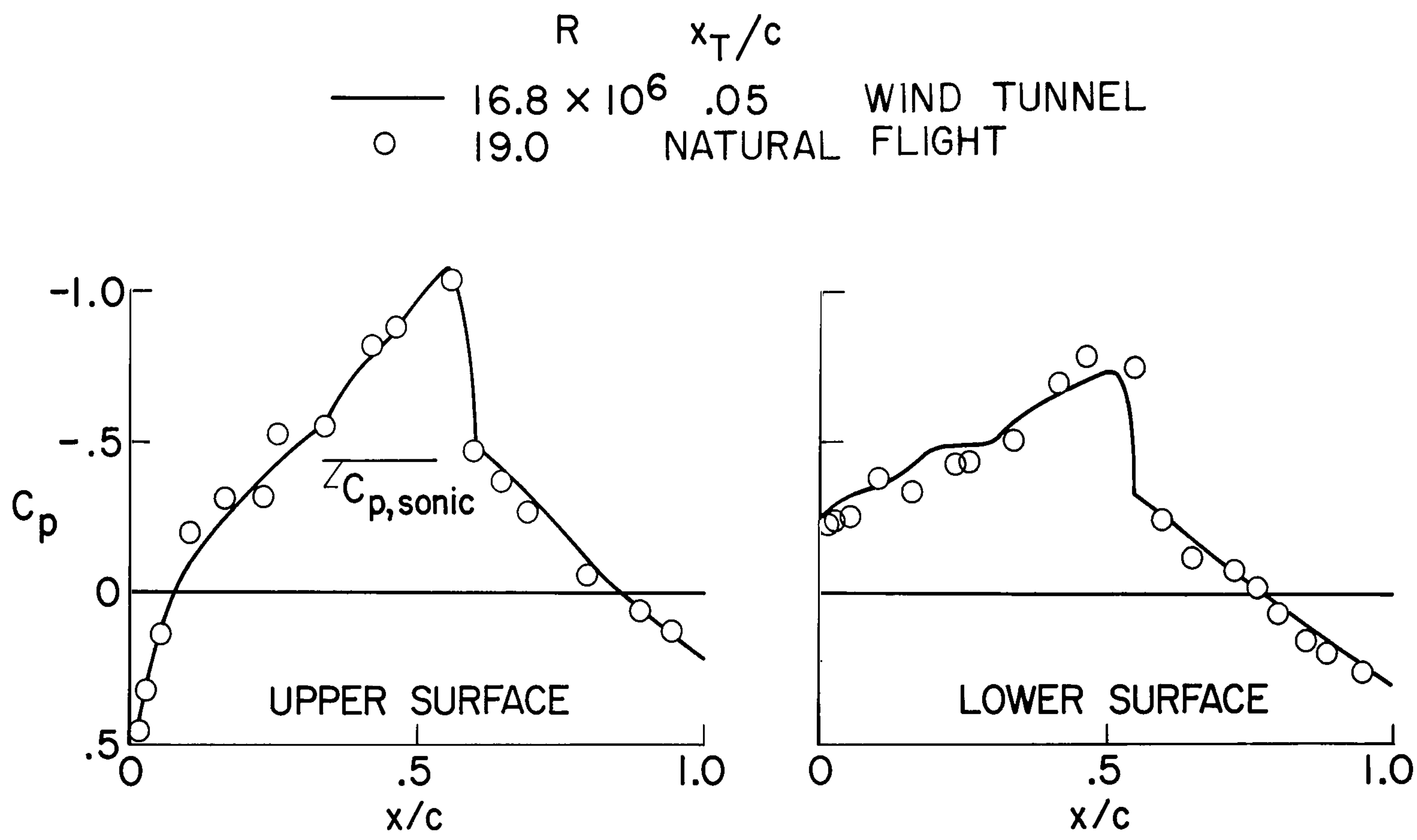
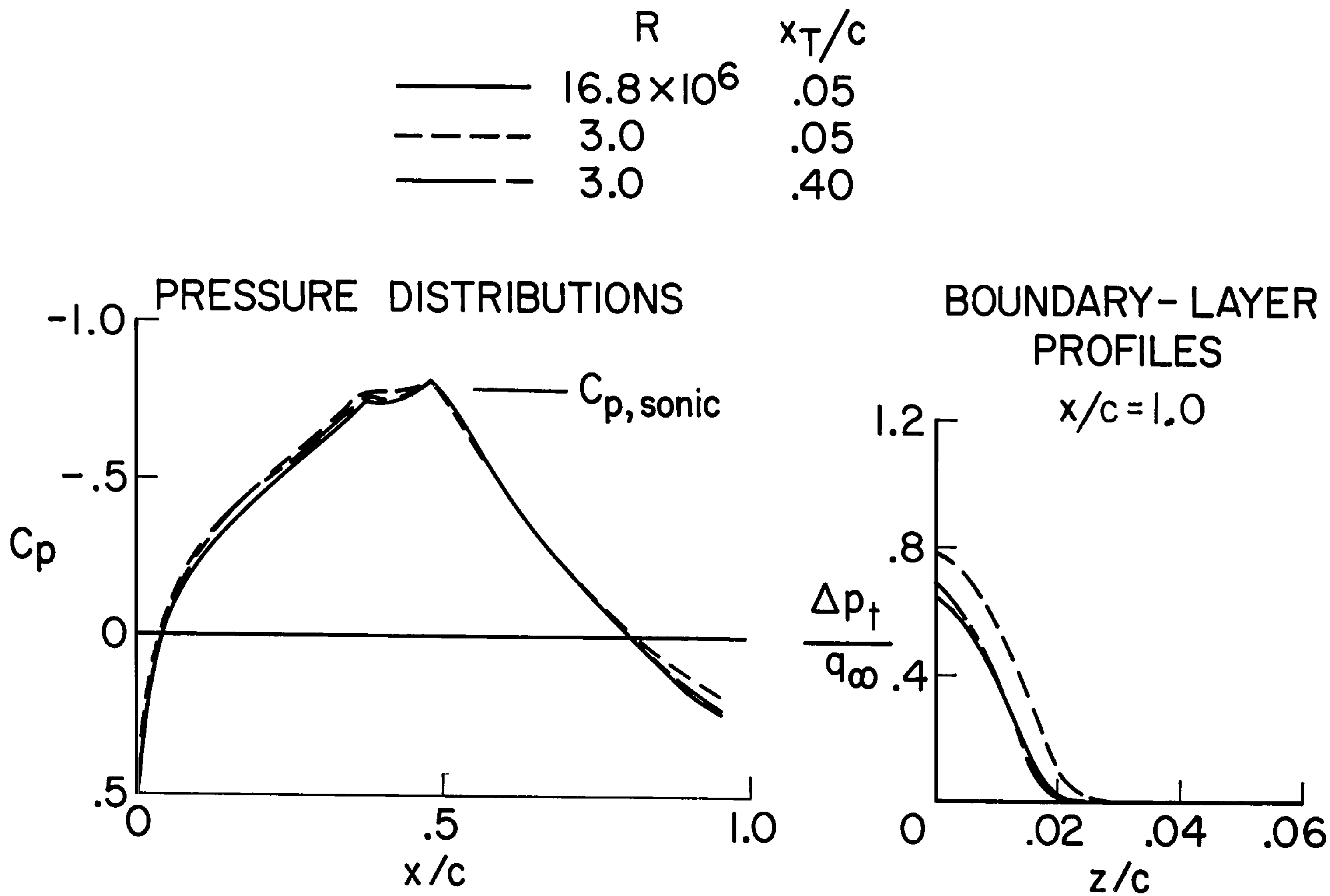
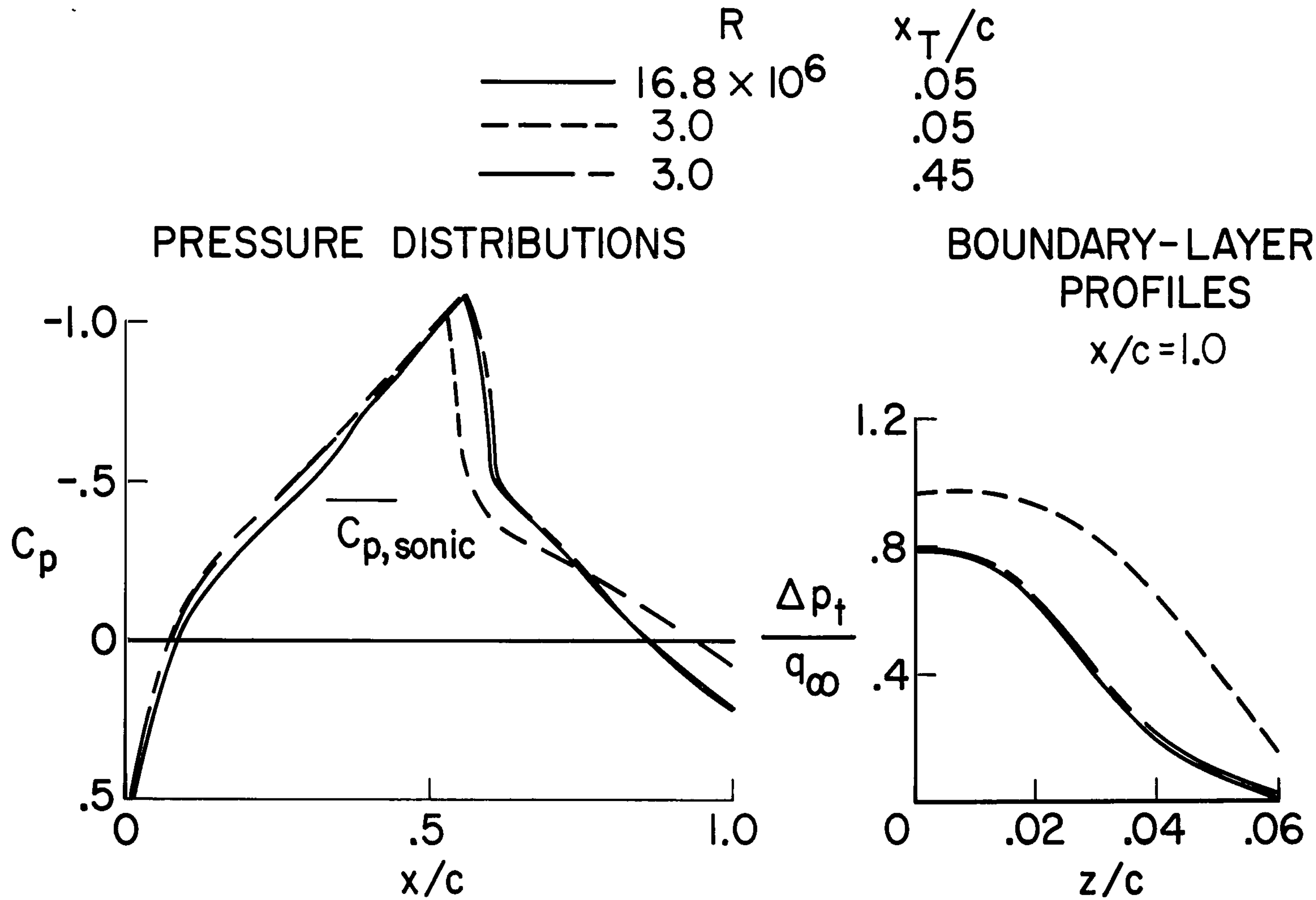


Figure 5.- Correlation of wind-tunnel and flight results. $M \approx 0.80$; $c_l = 0.056$.



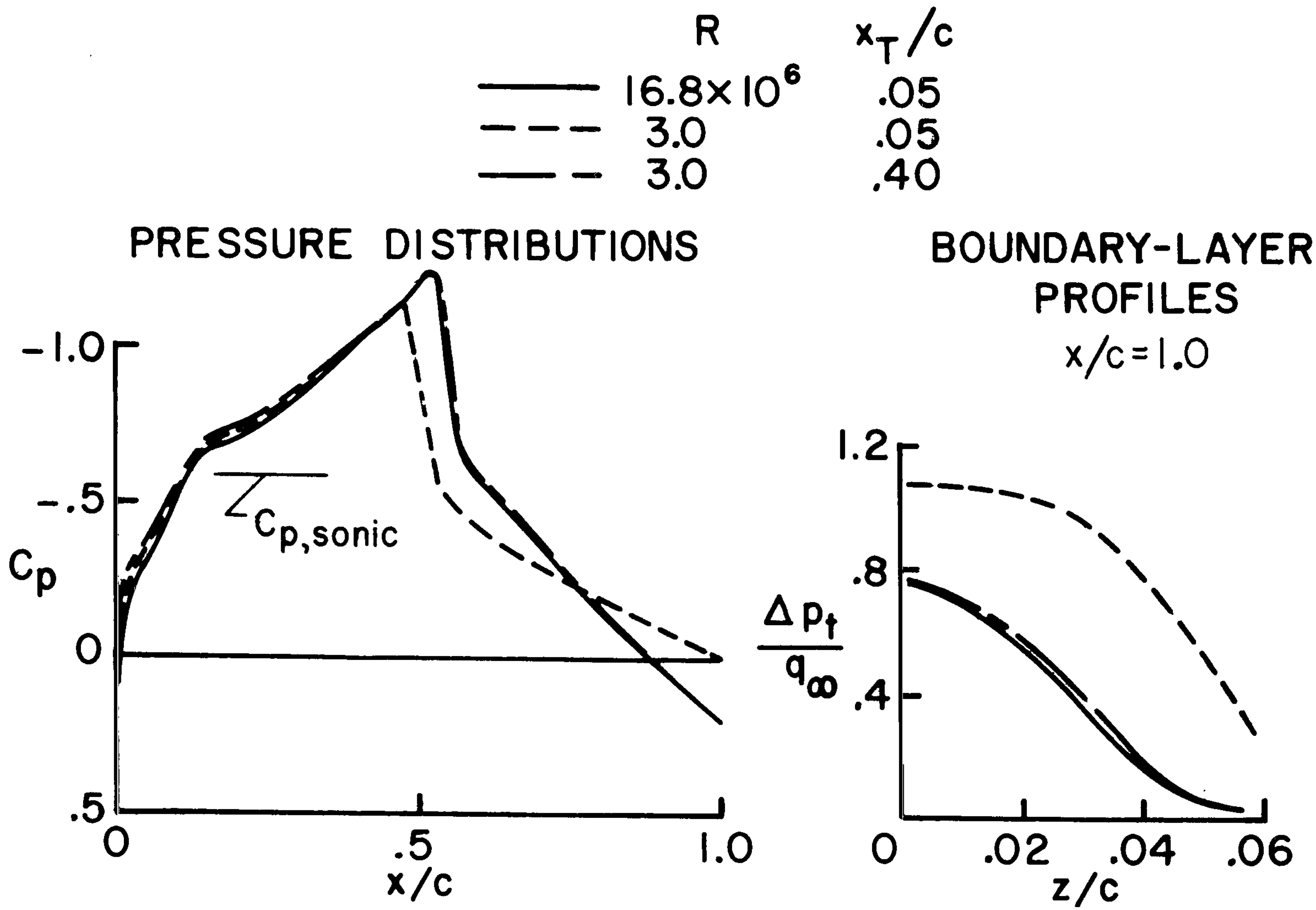
(a) $M = 0.70$; $\alpha = 0^{\circ}$.

Figure 6.- Effect of Reynolds number and transition location.



(b) M = 0.80; α = 0°.

Figure 6.- Continued.



(c) $M = 0.75$; $\alpha = 3^0$.

Figure 6.- Concluded.

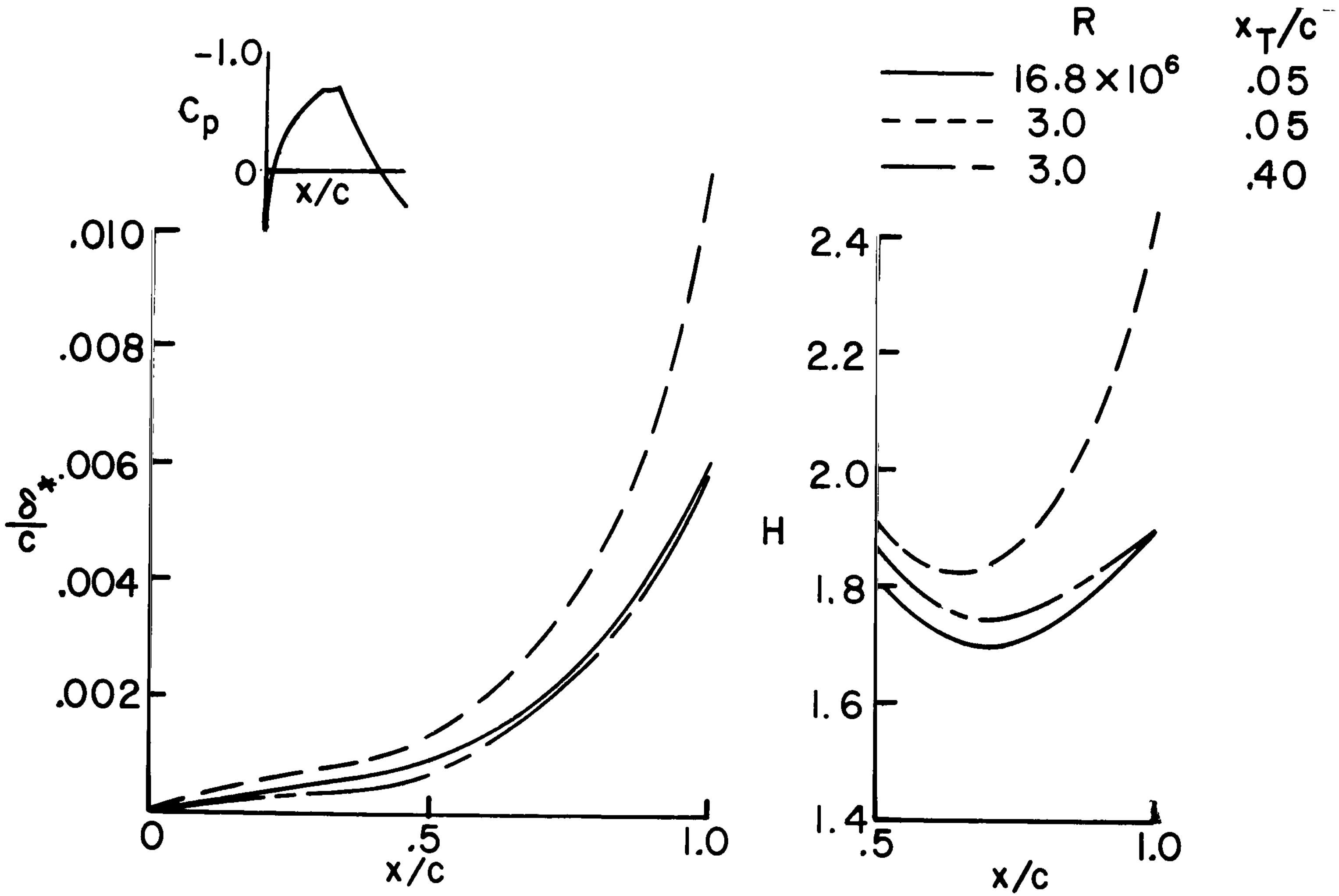


Figure 7.- Theoretical boundary-layer characteristics. $M = 0.70$; $\alpha = 0^0$.

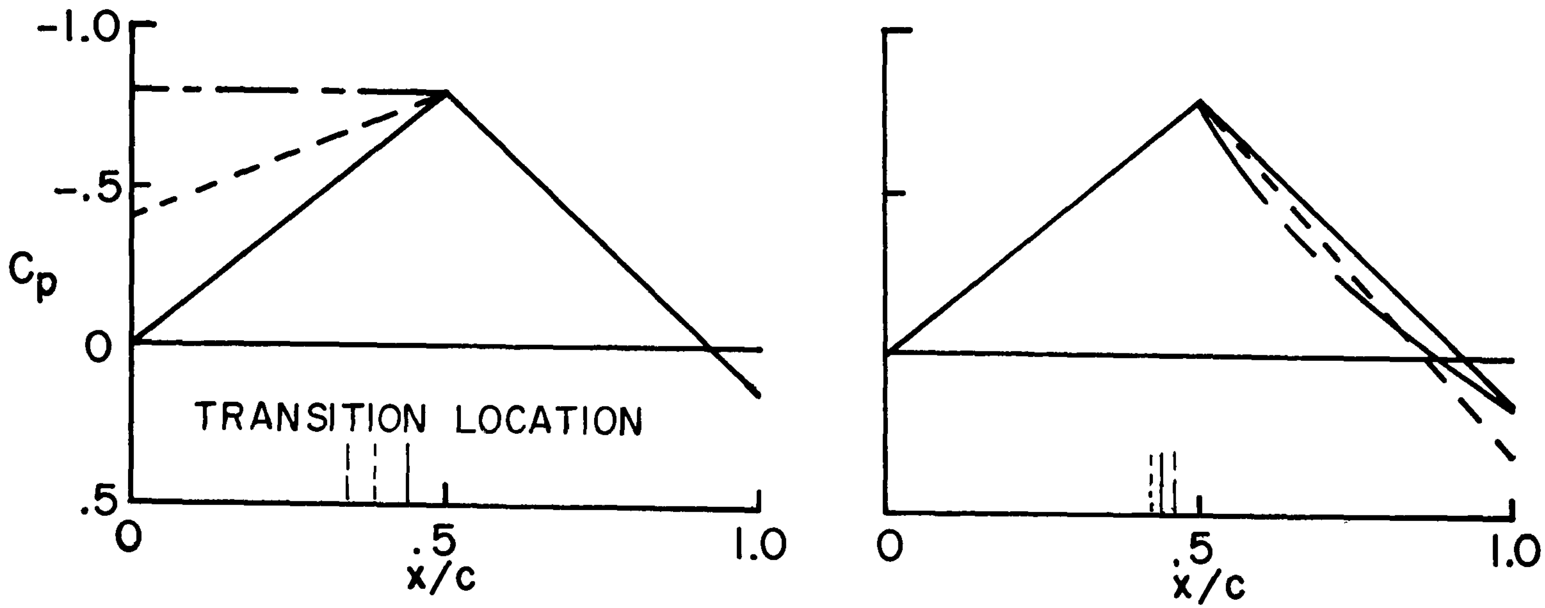


Figure 8.- Effect of pressure distribution on theoretical transition location. $M = 0.70$; $R_{wt} = 3 \times 10^6$; $R_{fs} = 16.8 \times 10^6$.

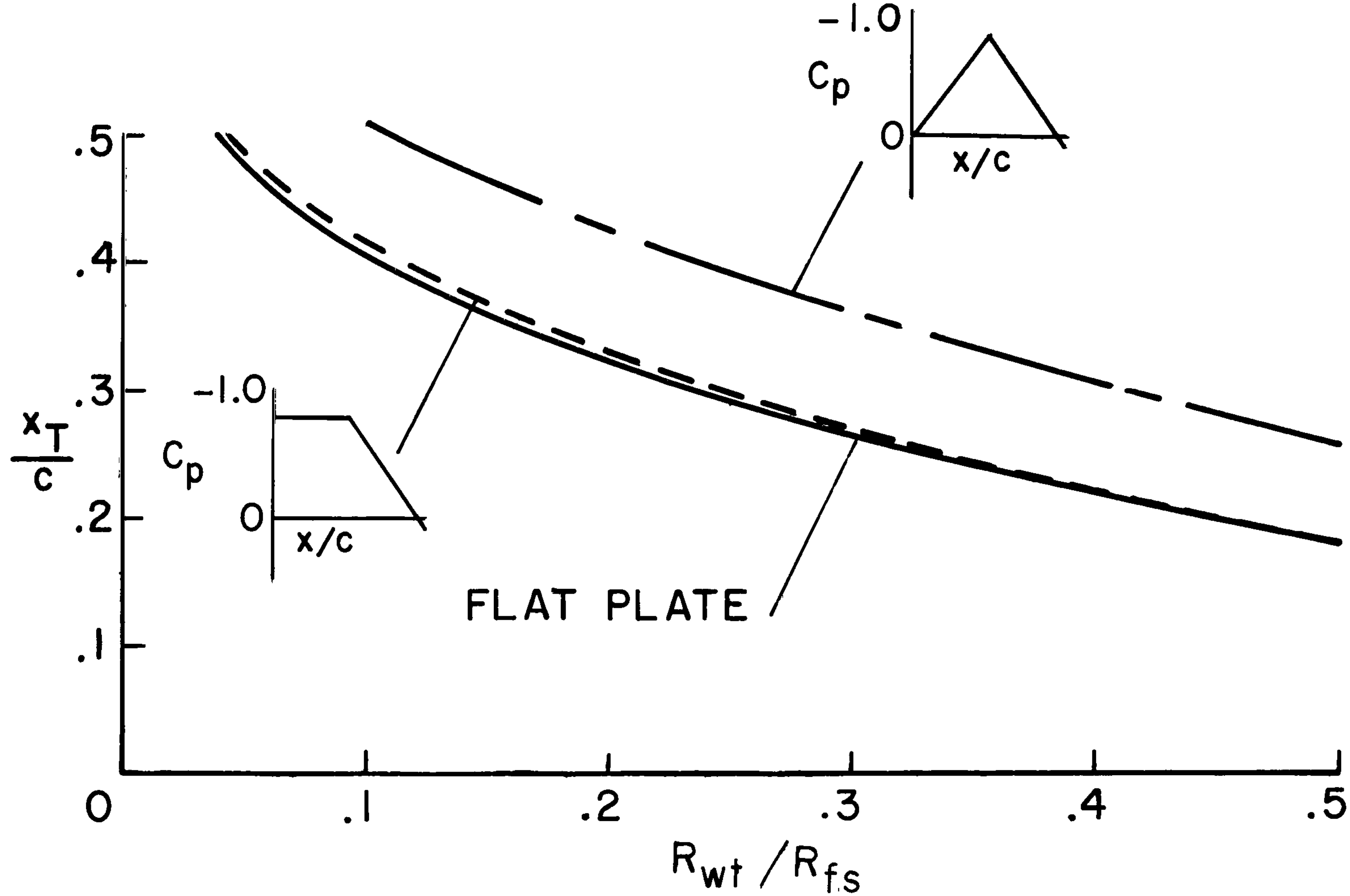


Figure 9.- Effect of Reynolds number on theoretical transition location. $M = 0.70$; $R_{wt} = 3 \times 10^6$.

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