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RESEARCH MEMORANDUM

BOUNDARY-LAYER TRANSITION AT SUPERSONIC SPEEDS

By George M. Low

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

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~~CONFIDENTIAL~~

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

Recent results of the effects of Mach number, stream turbulence, leading-edge geometry, leading-edge sweep, surface temperature, surface finish, pressure gradient, and angle of attack on boundary-layer transition are summarized.

Factors that delay transition are nose blunting, surface cooling, and favorable pressure gradient. Leading-edge sweep and excessive surface roughness tend to promote early transition.

The effects of leading-edge blunting on two-dimensional surfaces and surface cooling can be predicted adequately by existing theories, at least in the moderate Mach number range.

INTRODUCTION

The importance of the boundary-layer transition problem can hardly be overemphasized. The benefits to be derived from maintaining a laminar as opposed to a turbulent boundary layer are well known. Values of both laminar heat transfer and laminar skin friction are very much lower than the corresponding turbulent values.

A complete understanding of the transition process would enable the designer of high-speed missiles and aircraft to gain two distinct advantages: first, if he were able to predict exactly the location of transition, he would not have to overdesign to allow for turbulent aerodynamic heating rates that may not exist; second, he could incorporate features in the design that would delay transition as far as possible. Unfortunately, such a complete understanding of transition is not yet in sight. However, a large number of experimental observations of transition at supersonic speeds have been made. At first, these observations

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did not present a consistent picture, primarily because a large number of factors influence the transition process. More recently, though, experiments have been conducted which isolate some of the factors affecting transition. These experiments allow us to draw preliminary conclusions concerning the transition process.

This report represents a survey of some of the experimental results obtained during recent years and up to December, 1955. Other surveys have been published by Gazely (ref. 1), Czarnecki and Sinclair (ref. 2), Romig (ref. 3), Eckert (ref. 4), Seiff (ref. 5), and Probstein and Lin (ref. 6).

FACTORS AFFECTING TRANSITION

A complete survey of the boundary-layer transition field is not made in this paper, nor are the possible mechanisms of transition discussed from a theoretical point of view. Instead, some of the more important factors affecting transition are presented, and wherever possible, these are explained in terms of logical correlations.

Perhaps mention should be made of the fact that the theoretical approach to transition is usually through stability theory, which determines whether or not an infinitesimal disturbance will be amplified in a laminar boundary layer. Presumably, if a disturbance is amplified, transition to turbulence will eventually take place. Hence, stability theory is often used to predict qualitatively how transition is affected by a given variable. The point of first instability is generally far upstream of the location of transition. In between lies a region of amplification, which must also influence transition; however, this region is not yet amenable to theoretical analysis. Also, instability of laminar flow is not the only possible mechanism for transition. Other disturbing factors such as flow unsteadiness, shock waves, and effects of surface interferences, to mention only a few, undoubtedly also influence transition. The following parameters that influence transition are discussed herein:

(1) Mach number

(2) Stream turbulence

(3) Leading-edge or nose geometry

(4) Leading-edge sweep

(5) Surface temperature

(6) Surface finish

(7) Pressure gradient

(8) Angle of attack

The effect of isolated roughness elements (ref. 7) is not discussed herein, because this information is not immediately pertinent to the design of high-speed configurations. The effect of an expansion around a corner, which greatly delays transition, is also not presented, because results are not as yet complete (e.g., see refs. 8 to 10).

Mach Number

A summary of wind tunnel data (refs. 10 to 20) showing the effect of Mach number on transition under conditions of no heat transfer is given in figure 1. Only data for sharp-nosed cones (fig. 1(a)) and plates and hollow cylinders with sharp leading edges (fig. 1(b)) are included; in other words, only data for bodies where no pressure gradients exist are shown. Also, the presentation is limited to wind tunnel data, because it is not feasible to obtain high Mach number flight data under conditions of zero heat transfer.

The measured location of transition depends somewhat on the method used in observing transition. In general, the method of locating transition used in this report is that of the particular test being discussed. However, some freedom of choice is available when the temperature rise from a low laminar recovery temperature to a higher turbulent value is used to determine transition on an insulated surface. The results plotted in figure 1 are based on the peak of the longitudinal temperature profile; this peak corresponds approximately to the most frequent location of transition as observed by optical means.

At a given Mach number the spread in transition Reynolds number is appreciable (fig. 1). Part of this spread is undoubtedly due to wind tunnel disturbances. An effect of Reynolds number per unit length u/ν, as shown by the vertical line joining two symbols, is also evident. (All symbols are defined in the appendix.) Therefore, some other length, which may also depend on the tunnel disturbance or perhaps on the leading-edge thickness, is needed to completely correlate the results. However, there appears to be an upper envelope curve for the results, as shown by the dashed curves. (A few isolated points have been omitted purposely in fairing the envelope curves.)

If the spread of the data is caused by disturbing influences that exist only in a wind tunnel and not in flight, then the upper envelope represents the transition Reynolds number that may be expected in free flight. Although the transition Reynolds number at first decreases with increasing Mach number, a definite increase occurs at Mach numbers above 4 (fig. 1).

According to stability theory, the minimum critical Reynolds number (Reynolds number where infinitesimal disturbances are first amplified) for a cone is three times that for a flat plate. Yet, the transition Reynolds number for the cone is only slightly higher than for the plates and hollow cylinders. (Compare envelope curves in fig. 1.) This implies that the rate of amplification of disturbances, which cannot be predicted easily, may be higher for the cone than for the plate; or, transition may not be governed entirely by stability theory. 5003

In the section Leading-Edge or Nose Blunting, it is shown that even slight amounts of leading-edge blunting can substantially increase the transition Reynolds number, especially at high values of u/ν. Although only data obtained on models with leading-edge thicknesses of 0.001 inch or less are included in figure 1(b), it may still be possible that the spread in the data in this figure is due to a leading-edge effect. If this possibility were accepted, then perhaps the lower limit points (corresponding to low values of u/ν) of the flat-plate data (fig. 1(b)) should be compared with the upper envelope curve of the cone data (fig. 1(a)). (The effect of small bluntness on a cone is shown herein to be less significant than on a flat plate.) But, even if such a comparison is made, the transition Reynolds number for the cone is considerably less than three times the transition Reynolds number for plates and hollow cylinders.

Up to this point, the discussion is limited to transition data obtained on insulated surfaces. Data obtained with an ogive cylinder having a cold wall ($T_W/T_\delta \sim 1$) show a contradictory trend in that the transition Reynolds number increases with increasing Mach number for all Mach numbers (ref. 5). However, these data were obtained on an artificially roughened model. Other experiments (e.g., ref. 15) have shown that a given amount of surface roughness has a far greater effect on transition at low Mach numbers than at high Mach numbers. It is therefore suggested that the so-called effect of Mach number on transition of reference 5 is at least partly an effect of surface roughness. Another factor influencing these results is the varying rate of heat transfer that results from operating at a constant wall temperature and a varying Mach number. The effect of surface temperature on transition is discussed in the section Surface Temperature.

Stream Turbulence

A systematic study of the effect of supply-stream turbulence on transition was recently made by Van Driest (ref. 15). The turbulence

level of the supply stream was changed from 0.4 to 9 percent. The results of reference 15 (upper part, fig. 2) show that appreciable amounts of supply-stream turbulence can be tolerated if the test-section Mach number is high. A comparison of these results on the basis of free-stream turbulence is presented in the lower part of figure 2. (A relation between supply- and free-stream turbulences was obtained from ref. 21. This relation neglects dissipation and the generation of turbulence in the wind tunnel nozzle; only the longitudinal fluctuating component of velocity is transformed. A supply-stream Mach number of
5003 0.05 was assumed. Because of these assumptions, the transformation must be considered to be only approximate.) When plotted against free-stream turbulence, the data for Mach numbers of 1.9 and 2.7 fall essentially along a single line, and only one point at Mach 3.65 falls appreciably above that line. Thus, the effect of free-stream turbulence on transition is not reduced appreciably as the Mach number is increased. But, because a given supply-stream turbulence yields a smaller free-stream turbulence at higher Mach numbers, the effect of supply-stream turbulence on transition is reduced with increasing Mach number. We may conclude that wind tunnel transition data at higher Mach numbers are unaffected by turbulence level, if the supply-stream turbulence is reasonably low.

This conclusion does not contradict the discussion in the section entitled Mach Number, where it is surmised that the spread in transition Reynolds number at a given Mach number may be attributed to tunnel disturbances. Tunnel disturbances may be caused not only by supply-stream turbulence, but also by poor surfaces, improper nozzle contours, or leakage in the supersonic portion of the tunnel. Such disturbances may differ widely in different tunnels and may have large effects on transition.

Leading-Edge or Nose Blunting

In an investigation of transition on a hollow-cylinder model at Mach 3.1, Brinich (ref. 10) noted that sizeable delays in transition can be obtained by slightly blunting the leading edge. The explanation for this delay, as proposed by Moeckel (ref. 22), can best be illustrated by figure 3, which shows a typical blunted configuration with its detached shock wave. The total-pressure loss, which exists downstream of the curved portion of the shock, persists (in the absence of viscosity) for the entire length of the body. The static pressure, on the other hand, approaches its free-stream value downstream of the nose. The combination of a low total pressure and a free-stream static pressure leads to a lower Mach number near the surface than at a distance from the surface. Between these limiting values of Mach number exists a profile (forgetting for the moment about the boundary layer) such as is illustrated in the figure. A Reynolds number profile u/ν is similar in appearance to the Mach number profile. Moeckel defines the outer edge of a low Mach number, low Reynolds number layer by the streamline (dashed line, fig. 3) passing through the shock sonic point. This definition is arbitrary, but it guarantees that the Mach and Reynolds numbers will be close to their inviscid surface values throughout the layer so defined (see profile, fig. 3). Even for small nose thicknesses, the inviscid low Reynolds number layer is sufficiently thick to engulf a laminar boundary layer for a considerable length of run. Consequently, the development of this laminar boundary layer is governed not by conditions existing in the free stream, but by conditions existing within the low Reynolds number layer. If it is assumed that the transition Reynolds number is unaffected by blunting, the distance to transition is expected to increase by a factor inversely proportional to the Reynolds number reduction near the surface. The magnitude of the Reynolds number reduction for slender cones and flat plates (ref. 22) is shown as a function of Mach number in figure 4. At a Mach number of 3, for example, blunting causes a Reynolds number ratio of 1/2, which implies that transition can be delayed by a factor of 2. The predicted transition delay increases with increasing Mach number; at Mach 18 a fiftyfold increase in the distance to transtion is indicated. It must be realized, however, that many of the assumptions made in the analysis of reference 22 become invalid at very high Mach numbers. 5003

The preceding discussion is based on the hypothesis that the transition Reynolds number is unaffected by blunting, which may not be strictly valid. In particular, blunting alters the Mach number at the outer edge of the boundary layer. A change in Mach number was shown previously to affect transition. For example, for a free-stream Mach number of 3, the "outer-edge" Mach number obtained by blunting a flat plate is 2.3 (ref. 22). If we accept the effect of Mach number as described by the dashed line in figure 1(b), the transition Reynolds number is expected to increase by a factor of $\frac{3.5}{2.7} = 1.3$. Concurrently, u/ν is halved (fig. 4), and a transition delay of $1.3 \times 2 = 2.6$ is therefore predicted.

In the range of Mach numbers from 3 to 4, nearly all of the predicted effect of blunting has been observed on two-dimensional bodies (fig. 5(a)). The results of Brinich (ref. 10) show a maximum transition delay $(x_T)_b/(x_T)_s$ of nearly 2.2. This is exactly the value predicted in reference 22 for a Mach number of 3.1; however, it is somewhat less than the value 2.85 predicted if the combined effects of Mach number and blunting, as discussed in the preceding paragraph, are considered. The results of reference 10, which represent independent variations of u/ν and leading-edge thickness, are correlated in terms of a Reynolds number based on the leading-edge thickness. (The thickness of the sharp leading edge, used as a reference, was subtracted from all other leading edges.) This correlation implies, for example, that when the boundary layer is thin (high u/ν), less blunting is required to delay transition a given amount than when the boundary layer is thick (low u/ν). The data of reference 18 at first coincide with those of reference 10, but finally point towards larger transition delays. This larger delay is predicted by theory, because the tests of reference 18 were run at a higher Mach number than those of reference 10. Additional correlations of the effect of leading-edge bluntness can be found in reference 23.

5003

Most of the theoretically predicted transition delay has therefore been realized on two-dimensional bodies in the moderate Mach number range. However, only a small fraction of the predicted delay was observed on a hemispherically blunted cone recently tested by Brinich at a free-stream Mach number of 3.12 (unpublished). Results of this test are shown in figure 5(b). We see that the maximum transition delay is 1.27, whereas a theoretical delay of between 2 and 3 was anticipated. (With the assumption of a constant transition Reynolds number, a theoretical delay of 2 is predicted. But if the Mach number effect of fig. 1(a) is included, a delay of nearly 3 is expected.) This discrepancy between theory and experiment may be attributed to the adverse pressure gradient existing near the nose, which may partially counteract the favorable effects of blunting. If this explanation is valid, then a larger portion of the predicted transition delays may be achieved at higher Mach numbers. At hypersonic speeds the overexpansion around the nose, and, hence, the resulting adverse pressure gradient, is milder than at lower Mach numbers.

Much larger amounts of blunting are required on the cone than on two-dimensional bodies (fig. 5). This is expected, because a given amount of blunting produces a low Reynolds number layer of a fixed area. On a cone this area is distributed over an increasing perimeter, and, thus, the thickness of the low Reynolds number layer decreases along the length of the cone. A method for predicting the amount of blunting to produce a low Reynolds number layer of sufficient thickness is given in reference 22.

In addition to the hemispherical blunting, Brinich also blunted the cone tip to a flat face perpendicular to the cone axis. With this type of blunting, transition was often moved forward of its position on the sharp-nosed configuration.

Leading-Edge Sweep

Data of Dunning and Ulmann (ref. 18), showing the effect of leading-edge sweep on transition, are reproduced in figure 6. A transition ratio, representing the distance to transition measured normal to the leading edge referred to the distance to transition for an unswept wing, is shown as a function of sweep angle. A very rapid forward movement of transition with increasing sweep angle, well represented by the cube of the cosine of that angle, is evident. Had the data been represented in terms of the Reynolds number normal to the leading edge rather than distance normal to the leading edge, a decrease as $\cos^4\Lambda$ (sweep angle, fig. 6) would be noted.

There are two possible explanations for the rapid forward movement of transition with increasing sweep angle. In the first place, the aspect ratio of the wings tested in reference 18 ranged from 2.3 to 4. These ratios are rather low, so that end effects may have influenced transition. Secondly, a three-dimensional boundary layer, such as exists on a swept wing, is generally less stable than a two-dimensional 5003
profile. Moore, in reference 24, suggests that it always may be possible to select a coordinate system representing a boundary layer with secondary flow such that an inflection point exists in the profile. He states further that the stability problem may be treated as a two-dimensional problem, governed by the boundary-layer profile measured in the direction of an assumed disturbance. Hence, a velocity profile with an inflection point may always enter into the stability calculations for a swept wing, and such a profile is very unstable.

Surface Temperature

Experimental results of the effect of surface temperature on transition are presented in figure 7. The wind tunnel data of reference 25, obtained on a cone-cylinder model, show that the transition Reynolds number Re_T can be increased by a factor of 5 by cooling the model from the insulated surface condition ($T_W/T_\infty \sim 2.6$) to a temperature ratio of about 1.4. Further cooling would have moved transition off the model. The shape of the curve suggests that small additional amounts of cooling may yield exceedingly high transition Reynolds numbers. The results represent data obtained on both the cone and the cylinder portions of the model with no significant difference in Re_T. These wind tunnel data are extended by the flight data of reference 26 obtained on a cone. Transition Reynolds numbers as high as 32×10^6 were obtained by cooling to a temperature ratio between 1.2 and 1.3.

The solid symbols in figure 7 represent data obtained from an unpublished investigation by Disher and Rabb in flight on a two-stage rocket-propelled test vehicle with a highly polished cone-cylinder as its second stage. The tip of the 15° included-angle cone was blunted to a diameter of 7/8 inch, whereas the cylinder diameter was 6 inches. A peak Mach number slightly above 8 was attained in this flight. At that time, the wall-to-stream temperature ratio was 1.5 and the boundary layer at all measuring stations was found to be laminar; thus, the transition Reynolds number was at least 38.5×10^6. As the missile decelerated, transition passed over the last measuring station at a Reynolds number of 27.5×10^6, a temperature ratio of 1.9, and a Mach number of 3.6.

Transition was observed at a measuring station located downstream of the cone-cylinder juncture. The reason for transition being observed at a temperature ratio of 1.9, rather than 1.2 as previously noted on a sharp cone, is given in the section Combined Effects of Cooling and Blunting.

The effect of cooling, as obtained in various wind tunnels, is correlated in figure 8. The wall temperature divided by the adiabatic wall temperature is plotted against the transition Reynolds number divided by its value existing on an insulated body. Data are presented in this manner in order to eliminate any Mach number effect (and for the data of ref. 27, pressure gradient effect) under conditions of zero heat transfer. At a given Mach number, the data are quite well represented by a form of a hyperbola. The data were fitted with an analytic curve primarily to allow extrapolation with consistency to much higher Reynolds numbers. Asymptotes of the extrapolated curves are the temperature ratios that may yield infinite Reynolds numbers. These are needed for a comparison of the data with stability theory (fig. 9).

The solid curve in figure 9 delineates the region of complete stability to two-dimensional disturbances as given in reference 28.[1] Above the curve disturbances are amplified if the Reynolds number is sufficiently high; below the curve all two-dimensional disturbances are damped, no matter how high the Reynolds number. For conditions below the curve we may therefore presume that transition as resulting from laminar instability will not occur. With the exception of the low Mach number data ($M = 1.61, 1.9$), both the asymptotic values of the wind tunnel data and the flight data agree reasonably well with stability theory.

Dunn and Lin in reference 28 have shown that, even though all two-dimensional disturbances in a boundary layer may be damped at a sufficiently low temperature ratio, certain three-dimensional disturbances are always amplified at high Reynolds numbers. But by cooling to a temperature ratio slightly below that required for complete two-dimensional stability, the minimum critical Reynolds number (Reynolds number where disturbances are first amplified) for all disturbances becomes exceedingly high (of the order of 10^{12}). For all practical purposes the boundary layer is then stable for all disturbances. A typical point that includes these three-dimensional effects is shown in figure 9 and also represents well the experimental points. The agreement of the experimental points in the range of Mach numbers from 2.5 to 4 with the theoretical stability curve lends hope to the possibility that stability theory can be used to predict the amount of cooling necessary to delay transition to very high Reynolds numbers.

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[1]Dr. C. C. Lin, in a private communication, has informed the author of a recent revision of the stability theory. However, the differences between the revised and original curves, for $M > 2.5$, are less than the uncertainties in the reported data. Therefore, only the original theoretical curve is included in figure 9.

Combined Effects of Cooling and Blunting

Additional benefits to be derived from nose blunting are that less cooling should be required to delay transition, and that cooling can be made to be effective at Mach numbers above the upper limit of the stability curve. These effects are fully described in reference 22 and are discussed only briefly herein. Consider, for example, a flight Mach number of 6 and a surface that is cooled for structural reasons to a temperature equal to 0.4 of its adiabatic temperature. This point is located well outside the region of infinite stability (fig. 10). If the 5003
nose is blunted, the Mach number at the outer edge of the boundary layer is reduced to about 3.25; the adiabatic wall temperature changes only by a small amount. For the same wall temperature the point is now well within the stable region.

An explanation of the discrepancy between two points at a transition Reynolds number of 27.5 shown in figure 7 can be made along similar lines. One data point was obtained on a sharp cone, the other on a blunted cone-cylinder. The temperature ratio T_w/T_∞ for the blunted configuration equalled about 1.9 when transition passed over the measuring station. Because of the blunt nose, however, the Mach number at the outer edge of the boundary layer was considerably below its free-stream value, and, consequently, the outer-edge temperature T_δ was considerably higher than T_∞. The appropriate temperature ratio T_w/T_δ was 1.2. The transition point for the blunted configuration, when corrected for the true outer-edge conditions, then falls in line with the sharp-tipped-cone data. Since the point falls on a flat portion of the curve, we cannot say, however, that the theoretically predicted transition delay due to blunting was realized.

Wind tunnel data obtained at the NACA Lewis laboratory showing the combined effect of cooling and blunting are presented in figure 11. These results were obtained on the cone-cylinder model of reference 25, blunted hemispherically to a tip diameter of 3/16 inch. For reference purposes, a curve faired through the data for the sharp-tipped configuration is included in addition to a curve representing the theoretically predicted transition delay due to blunting. The latter curve was obtained as follows: Under equilibrium conditions $(T_w/T_{aw})_s = 1$, the theory of reference 22 indicates a transition delay by a factor of 2. At the same time, the Mach number is decreased; consequently, there is a further delay by a factor of 1.5 (fig. 1(a)), while the temperature ratio $(T_w/T_{aw})_b$ becomes 1.025 (point A, fig. 11). At the high Reynolds number end of the curve the stability theory must be used. From figure 9 the wall temperature ratio for the sharp body (outer-edge Mach number $M_\delta = 3.1$) is 0.61, for the blunt body ($M_\delta = 2.3$), $T_w/T_{aw} = 0.71$. Hence,

$$\left(\frac{T_w}{T_{aw}}\right)_b = \left(\frac{T_w}{T_{aw}}\right)_s \times \frac{0.71}{0.61} \times 1.025 = 1.2\left(\frac{T_w}{T_{aw}}\right)_s \tag{1}$$

The factor 1.025 represents the change in adiabatic wall temperature due to blunting.

Equation (1) fixes a point (say B) on the asymptotic portion of the curve (fig. 11). Between points A and B, the curve is faired with a transition delay factor of 3 and a temperature ratio factor ranging from 1.025 to 1.2. Again, only part of the predicted downstream movement of transition was observed. The difference between theory and experiment may be attributed to the fact that the model was not sufficiently blunt and to the adverse pressure gradient existing at the nose, as previously discussed.

We observe also that a larger increase in transition Reynolds number was obtained on the cooled body than on the insulated body. On the insulated body, transition took place on the conical portion of the model, and results here are in fair agreement with the cone data of figure 5(b). All points on the cooled body ($T_w/T_{aw} < 1.0$) were obtained on the cylindrical portion of the body. These results are in reasonable agreement with the hollow-cylinder data of reference 10 (fig. 5(a)).

Effect of Surface Finish

The question naturally arises as to how smooth a surface must be before cooling can be used as an effective means of delaying transition. Results of a study recently completed at the NACA Lewis laboratory of the effect of uniformly distributed surface roughness on transition are shown in figure 12 (unpublished investigation). The test model was the blunted cone-cylinder discussed in connection with figure 11, and the test Mach number was 3.1. The model was sanded, then sandblasted, and finally tested with an applied Carborundum grit. A variation in surface finish from less than 16 to 1250 microinches was thereby obtained. (For the polished, sanded, and sandblasted finishes, the roughness height h was determined with a Brush Surf Indicator using a 0.0005-inch stylus. The height of the grit finish is the maximum particle size.)

A correlating parameter for the data is a Reynolds number based on the roughness height hu_0/ν_0. Since neither the model geometry nor the Mach number was varied, the roughness Reynolds number is proportional to $\frac{h}{\delta}\sqrt{Re}$, which is the correlating parameter suggested in reference 5.

For values of hu_0/ν_0 less than 120, cooling was found to be very effective in delaying transition. This delay was obtained with surface finishes as high as 300 microinches. For sufficiently low values of u_0/ν_0, laminar flow was maintained over the entire model, even with the 1250-microinch finish. (These points do not appear on fig. 12). When the roughness Reynolds number was increased above 120, cooling became less and less effective. Transition could not be delayed by cooling when hu_0/ν_0 was equal to or greater than 840. The amount of roughness that can be tolerated under other conditions is expected to be a function of Mach number and body geometry. 5003

Combined Effect of Cooling and Favorable Pressure Gradient

Combined effects of cooling and a favorable pressure gradient at Mach 3.1 can be determined from the results of reference 25, which are reproduced for the forebodies of two models in figure 13. One of these is a cone, the other a parabolic nose of a fineness ratio of 6. The parabolic body, which has a favorable pressure gradient, has a transition Reynolds number about twice that of the cone, regardless of the amount of cooling, the location of transition, or the Reynolds number per unit length.

Effects of Angle of Attack and Adverse Pressure Gradient

One of the most important factors affecting transition, and one that is least understood, is the effect of angle of attack. This effect has been studied extensively by Seiff and his co-workers at the NACA Ames laboratory. A typical model at angle of attack, as described in reference 29, is shown in figure 14. Along the windward side of the model, the boundary layer, as observed by optical means, appears typically laminar; along the sheltered side, the boundary layer is laminar to point B, where it abruptly thickens. This transition point can be correlated with the pressure rise along a streamline from the pressure minimum to point B. (See insert, fig. 14.)

Carros (ref. 30) was further able to correlate transition with a pressure-rise coefficient $\Delta p/p_0$, even if the pressure rise was not the result of angle of attack. (In the latter case, the adverse pressure gradient follows an overexpansion around the corner of a cone-cylinder or on an ogive cylinder.) The correlation of reference 29, as presented in reference 5, is shown in figure 15 with data from references 30 to 32. The critical pressure-rise coefficient that causes transition is independent of Mach number and has a value between 0.1 and 0.2. It is interesting to note that these same values of pressure-rise coefficient describe the limits of $\Delta p/p_0$ required to separate a laminar boundary layer ahead of forward facing steps (ref. 33). This may imply that transition, when caused by pressure rise, is always triggered by incipient separation.

An interesting corollary to the problem of transition at angle of attack has been obtained in heat-transfer studies conducted at the NACA Lewis laboratory. The cone-cylinder model of reference 25 was tested at angles of attack up to 18°, while heat-transfer coefficients were measured along five rays of the model. Contours of Stanton number on
5003 the model at an angle of attack of 12° are shown in figure 16(b). For purposes of comparison, Stanton number distributions at zero angle of attack for wholly laminar and for wholly turbulent flow are also shown. We note that the Stanton numbers on the windward side of the body at angle of attack are slightly higher than laminar values at zero angle of attack; on the sheltered side towards the rear of the model, they are somewhat higher than on the windward side. Yet nowhere on the body do the heat-transfer coefficients approach the turbulent values obtained at zero angle of attack; nor is there any evidence of the abrupt rise in heat transfer usually associated with transition. The calculated pressure-rise coefficient for the conical nose at this angle of attack is 0.6, a value considerably greater than required for pressure-rise transition. It may be concluded, therefore, that even under conditions that indicate transition at angle of attack, the measured heat-transfer coefficients can fall considerably below the turbulent values.

CONCLUDING REMARKS

A fitting closure would be to present a method of predicting the transition Reynolds number for any given set of conditions. But it should be clear from this summary that such a method cannot be forthcoming at this time. However, this report points out the desirable factors for delaying transition and some of the undesirable factors that advance transition. Long laminar runs can be achieved by nose blunting, by cooling, and by making use of a favorable pressure gradient. Leading-edge sweep and excessive surface roughness, on the other hand, tend to promote early transition.

The quantitative results of this report can be used to obtain reasonable approximations of the transition Reynolds number, insofar as experimental evidence exists. The effect of combining all of the transition delaying factors is as yet unknown. If all favorable factors are multiplicative, and if the theoretically predicted delay due to blunting can be realized at higher Mach numbers, then extremely long laminar runs are feasible. Further experimentation is required before these long laminar runs can be predicted with confidence.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, May 15, 1956

APPENDIX - SYMBOLS

b	leading edge or nose thickness
h	roughness height
M	Mach number
p	static pressure
Re	Reynolds number
Re_T	transition Reynolds number
St	Stanton number
T	temperature
u	velocity
u'	fluctuating velocity in streamwise direction
x	distance from leading edge
ν	kinematic viscosity

Subscripts:

aw	adiabatic wall condition
b	blunted configuration
cr	critical
s	sharp configuration
T	transition
w	wall
δ	condition at outer edge of boundary layer
∞	conditions at a distance from surface
0	upstream or ambient conditions

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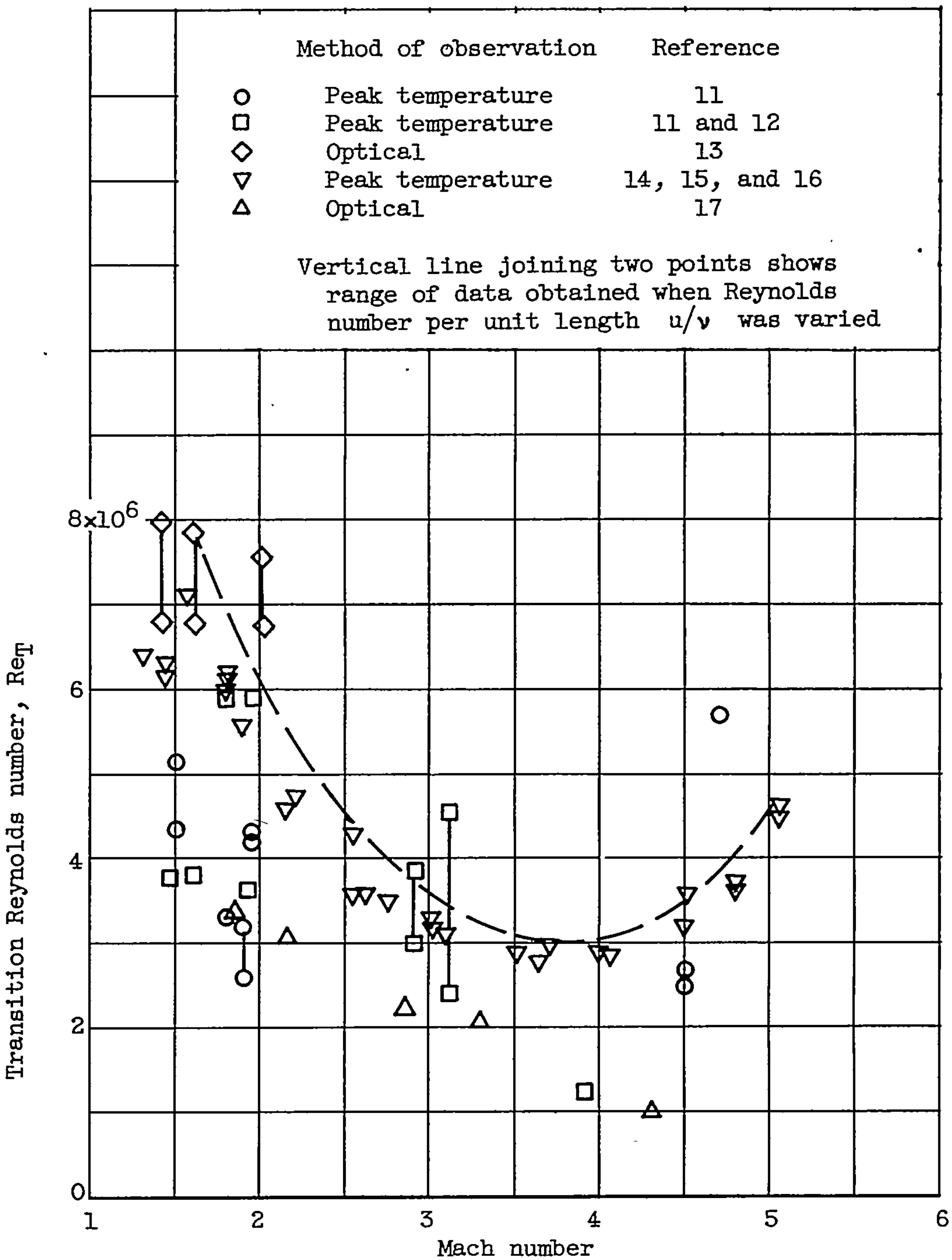

(a) Insulated cones.

Figure 1. - Effect of Mach number.

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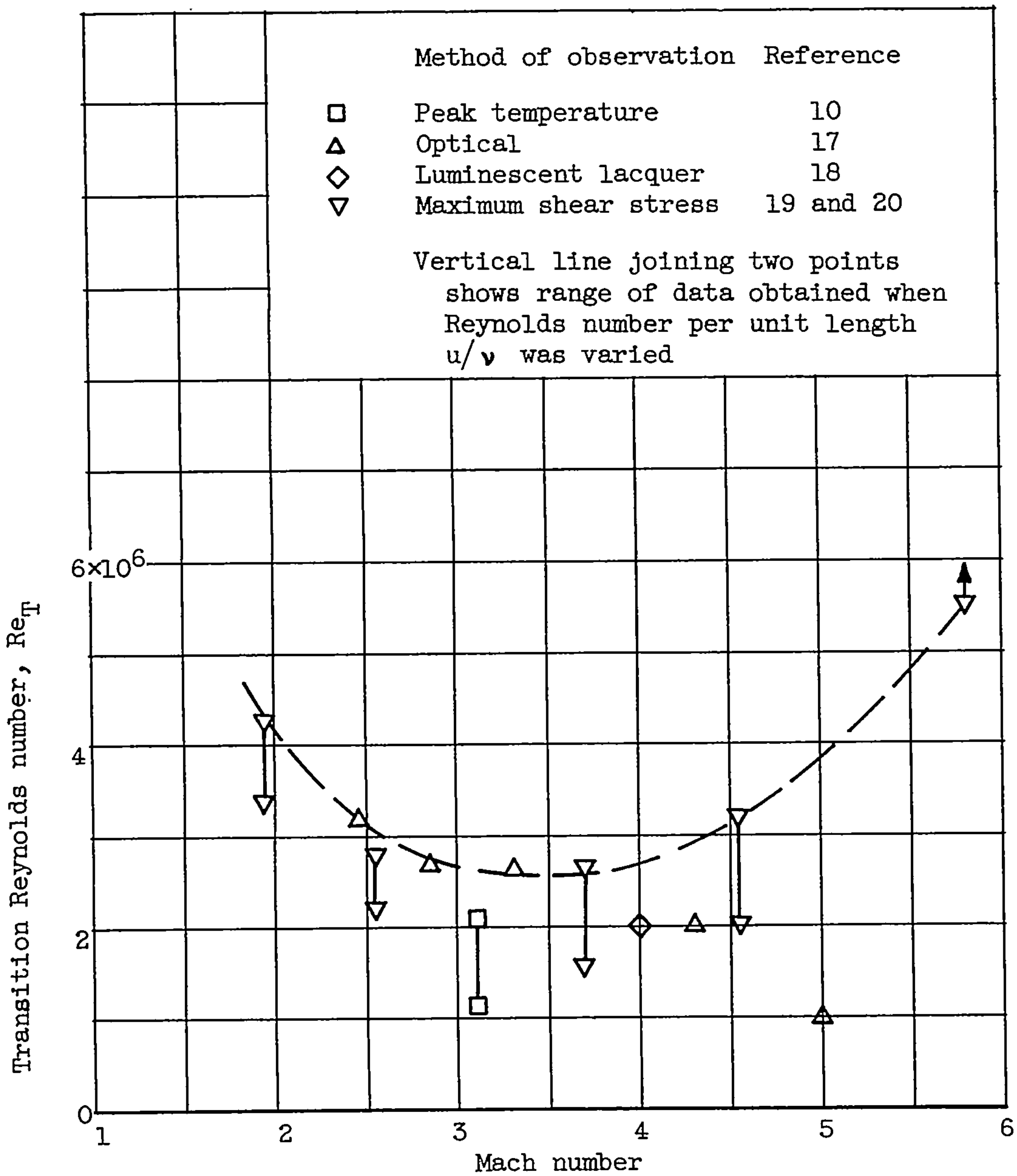

(b) Insulated plates and hollow cylinders.

Figure 1. - Concluded. Effect of Mach number.

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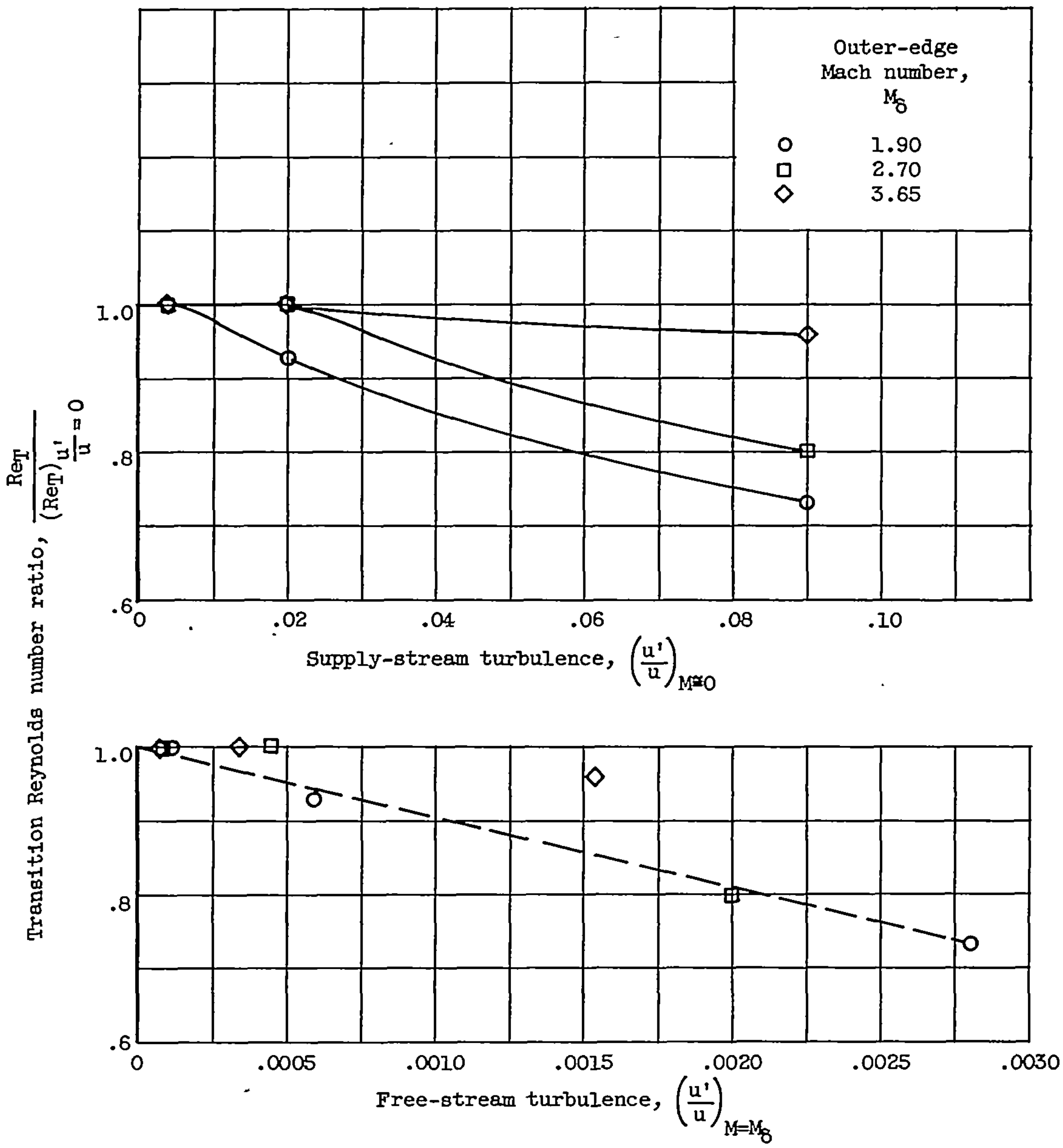

Figure 2. - Effect of turbulence level (data from ref. 15).

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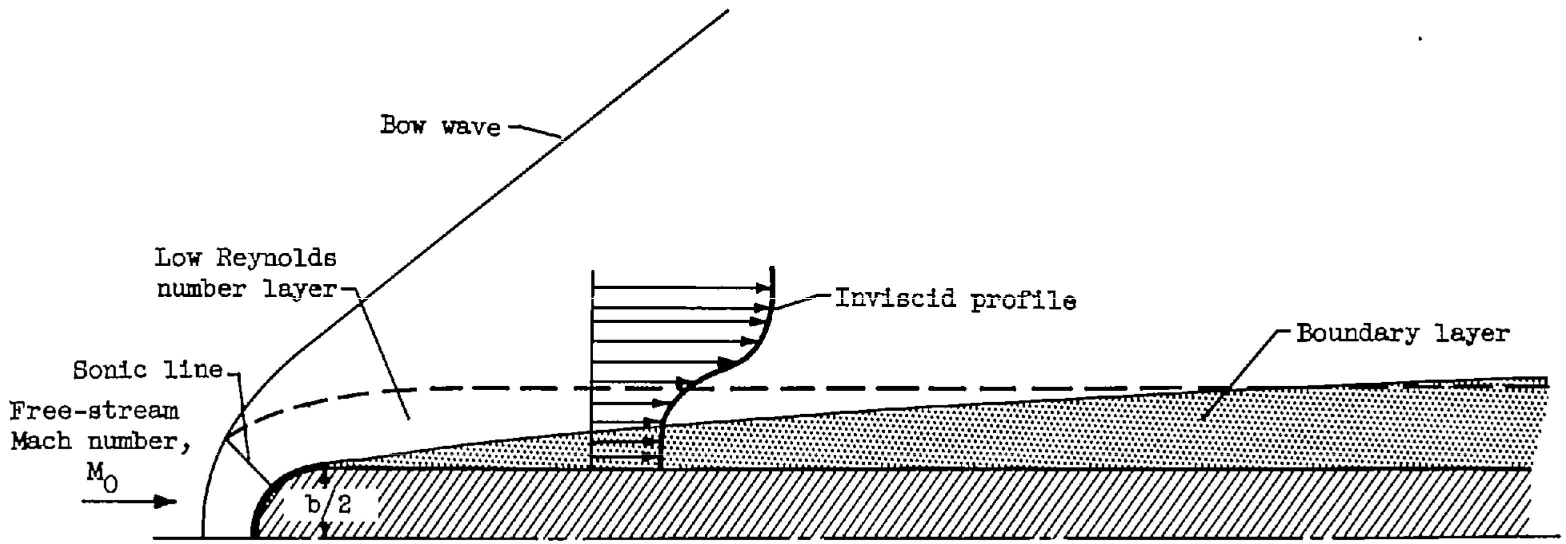

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Figure 3. - Illustration of effect of blunting.

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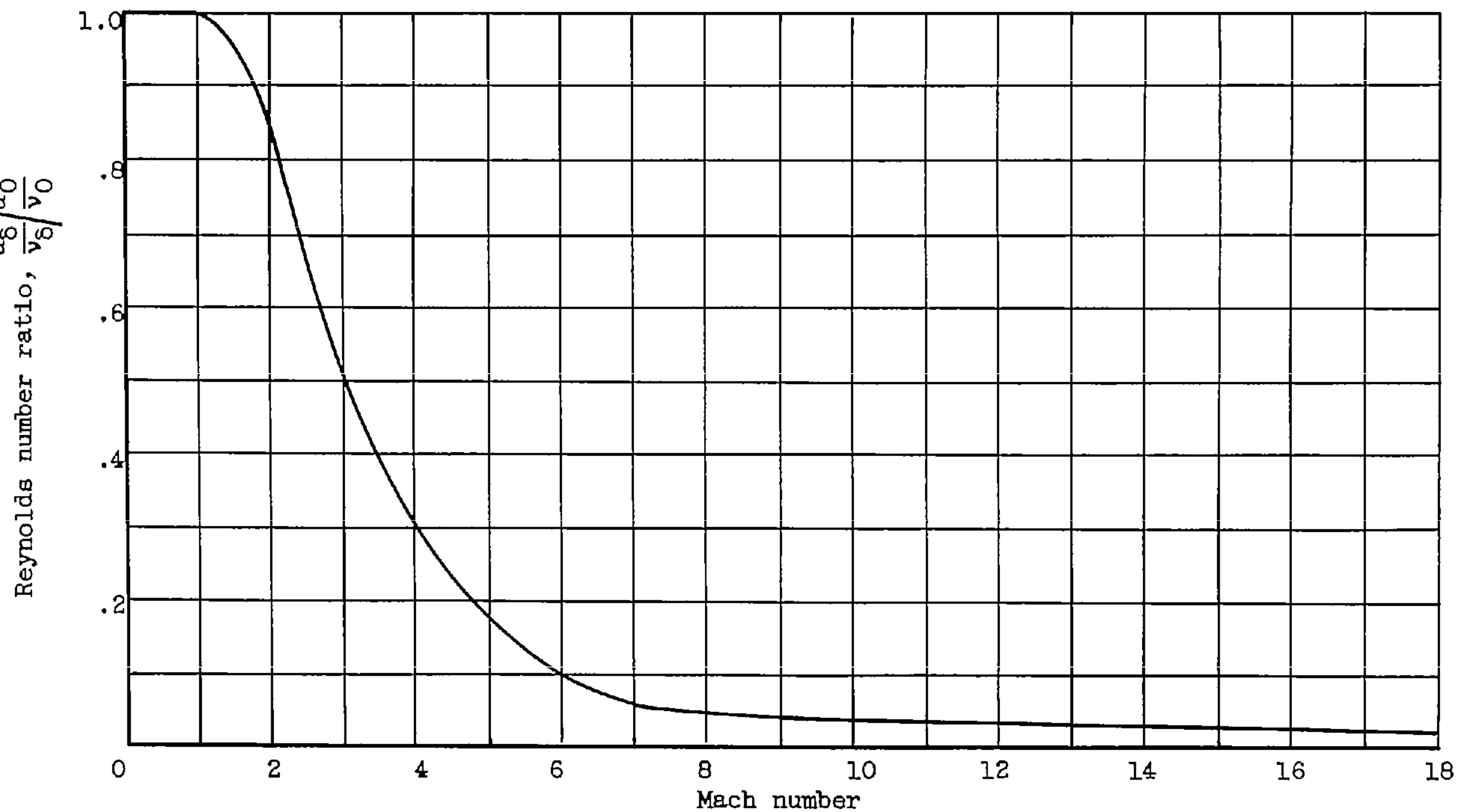

Figure 4. - Reynolds number reduction due to blunting (ref. 22).

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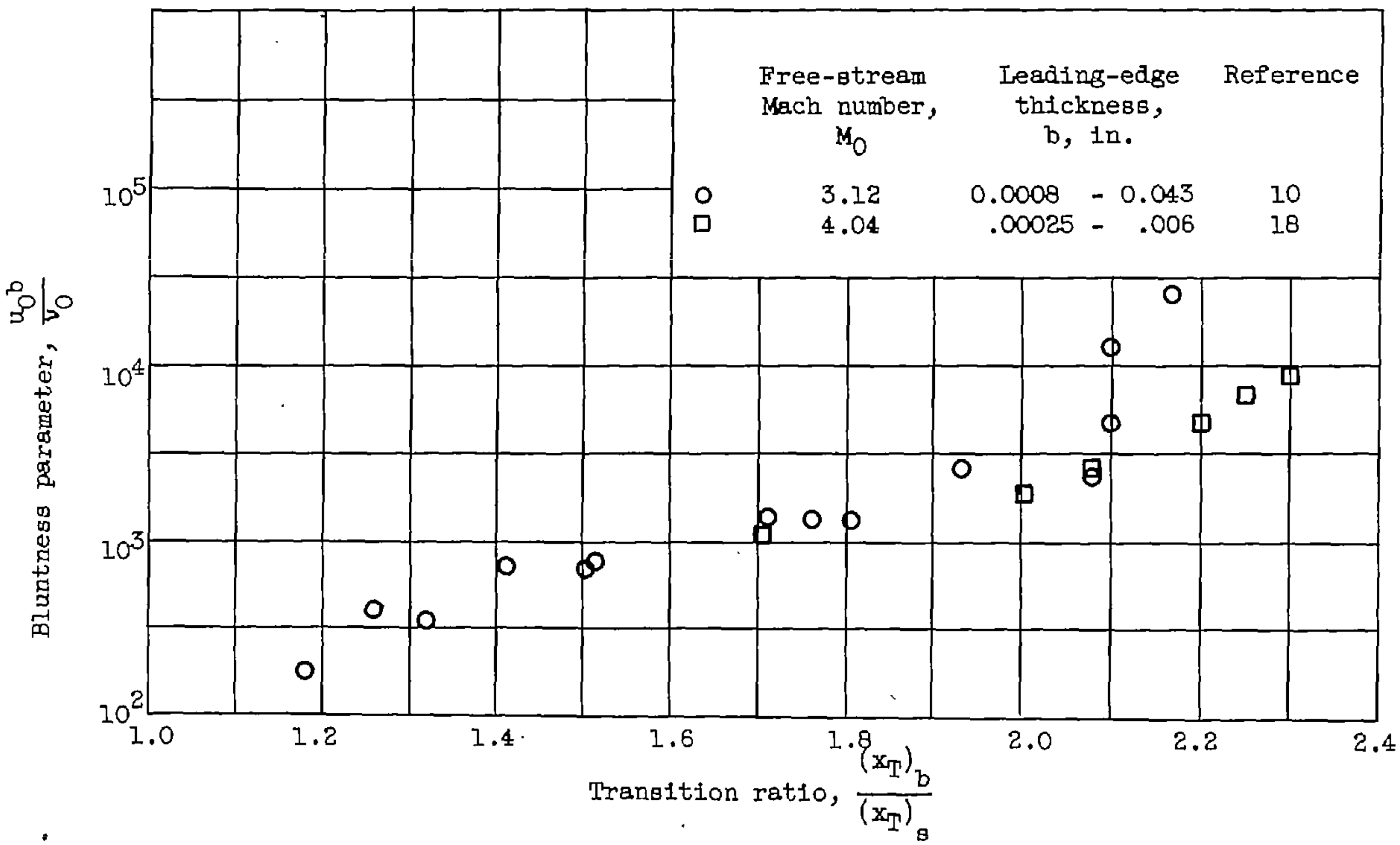

(a) Two-dimensional bodies.

Figure 5. - Effect of blunting.

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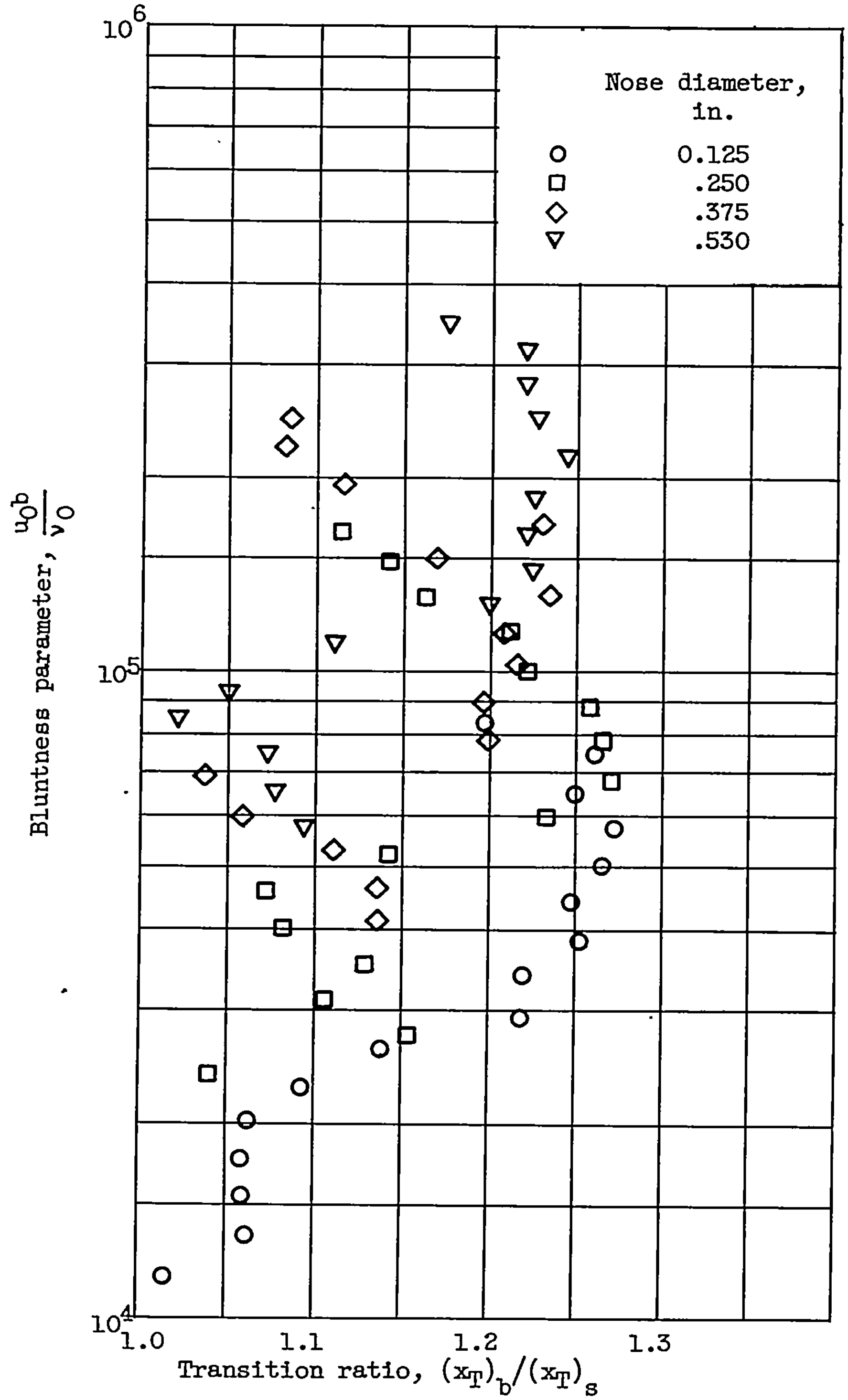

(b) Cone. Free-stream Mach number, 3.1.

Figure 5. - Concluded. Effect of blunting.

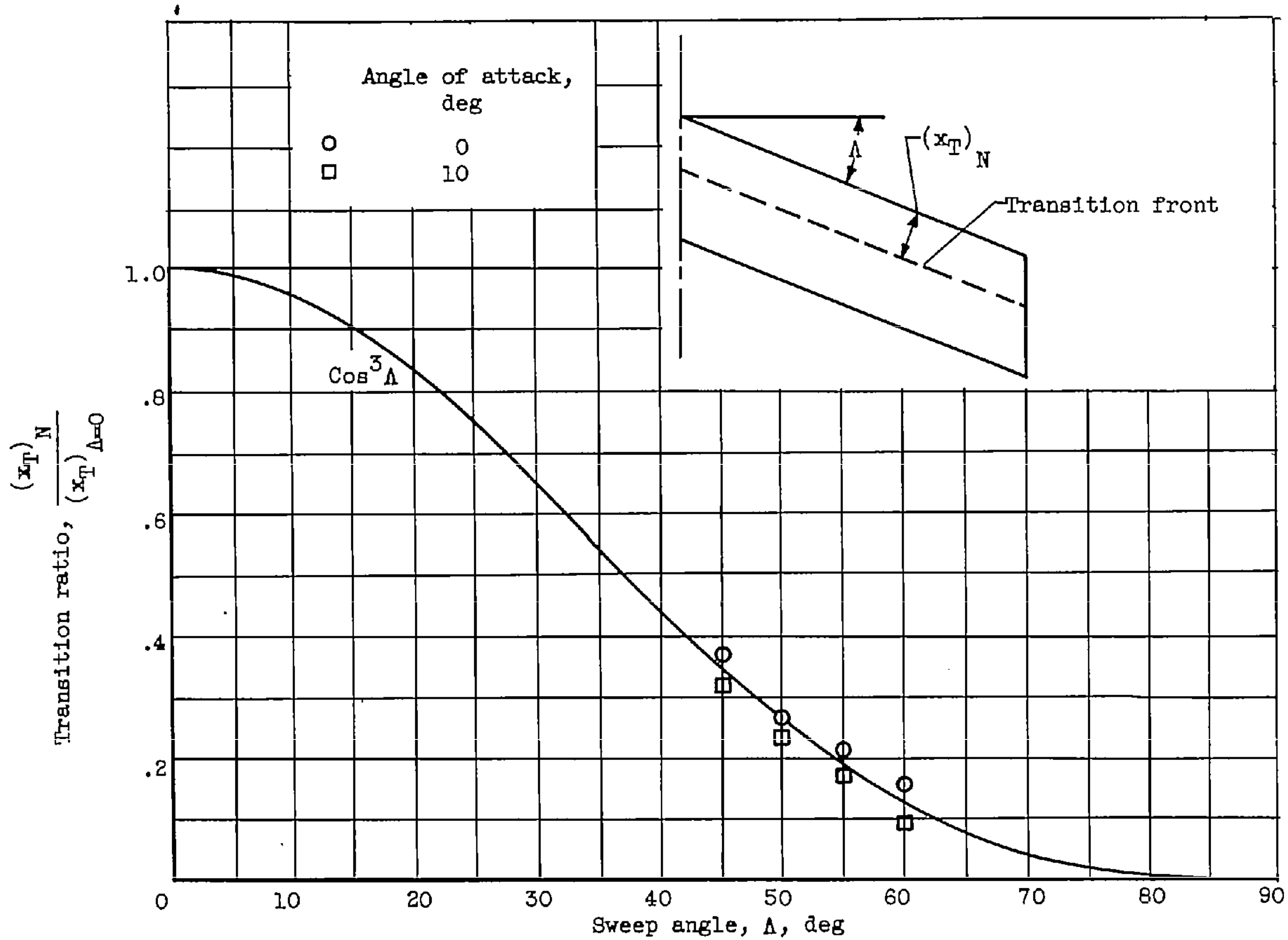

Figure 6. - Effect of sweep (data from ref. 18). Free-stream Mach number, 4.04.

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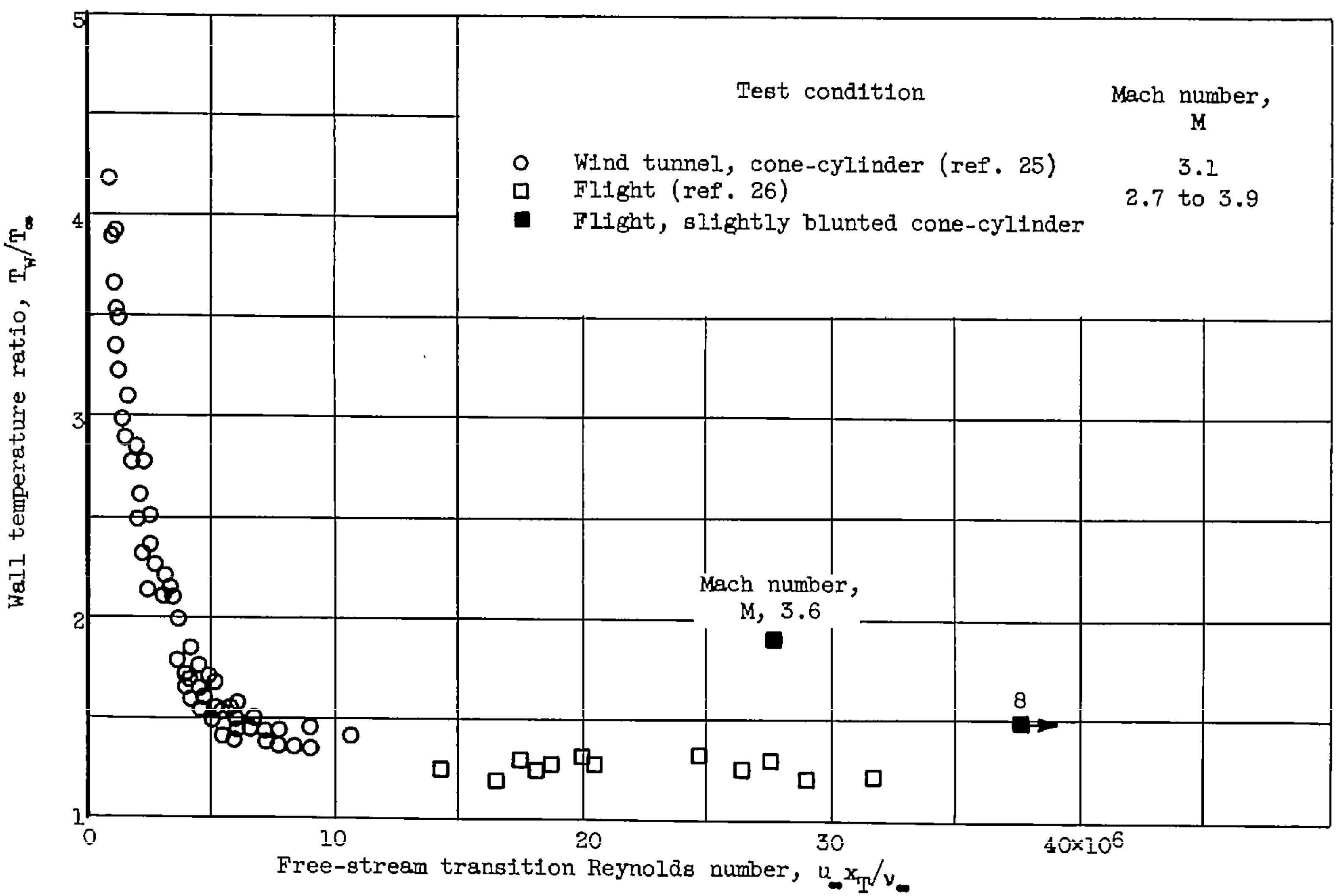

Figure 7. - Effect of surface temperature.

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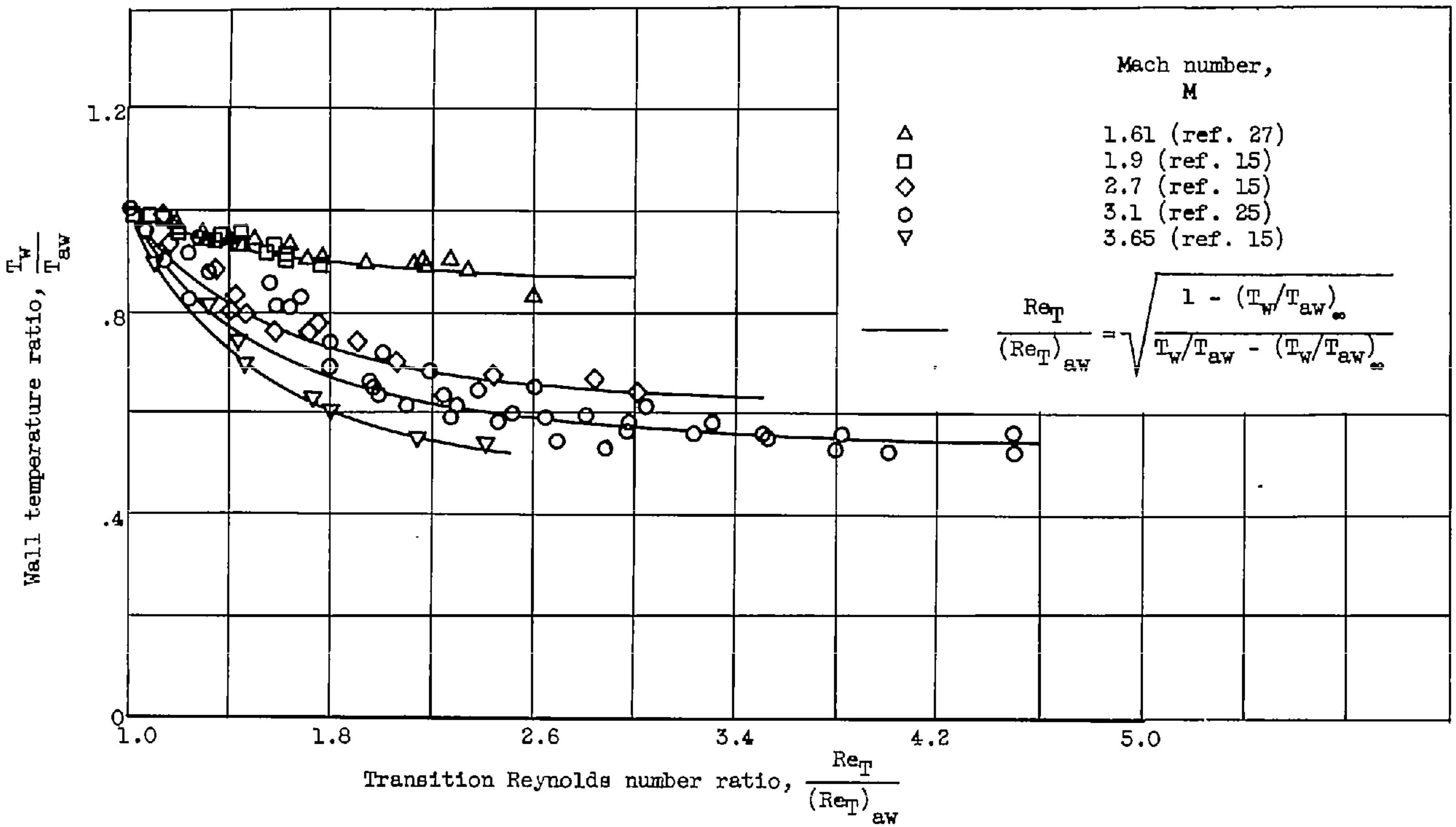

Figure 8. - Correlation of cooling effect.

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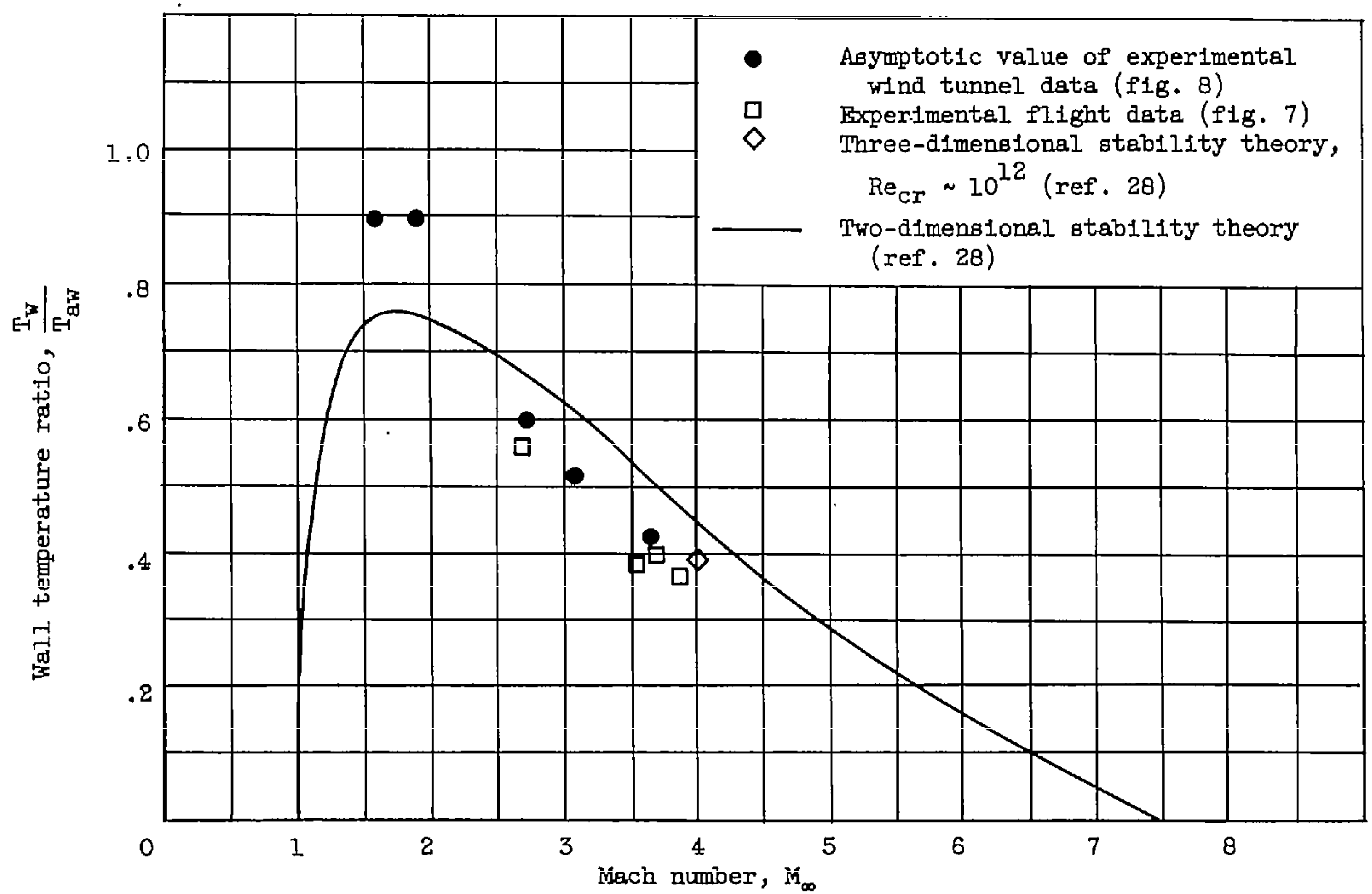

Figure 9. - Cooling effect compared with stability theory.

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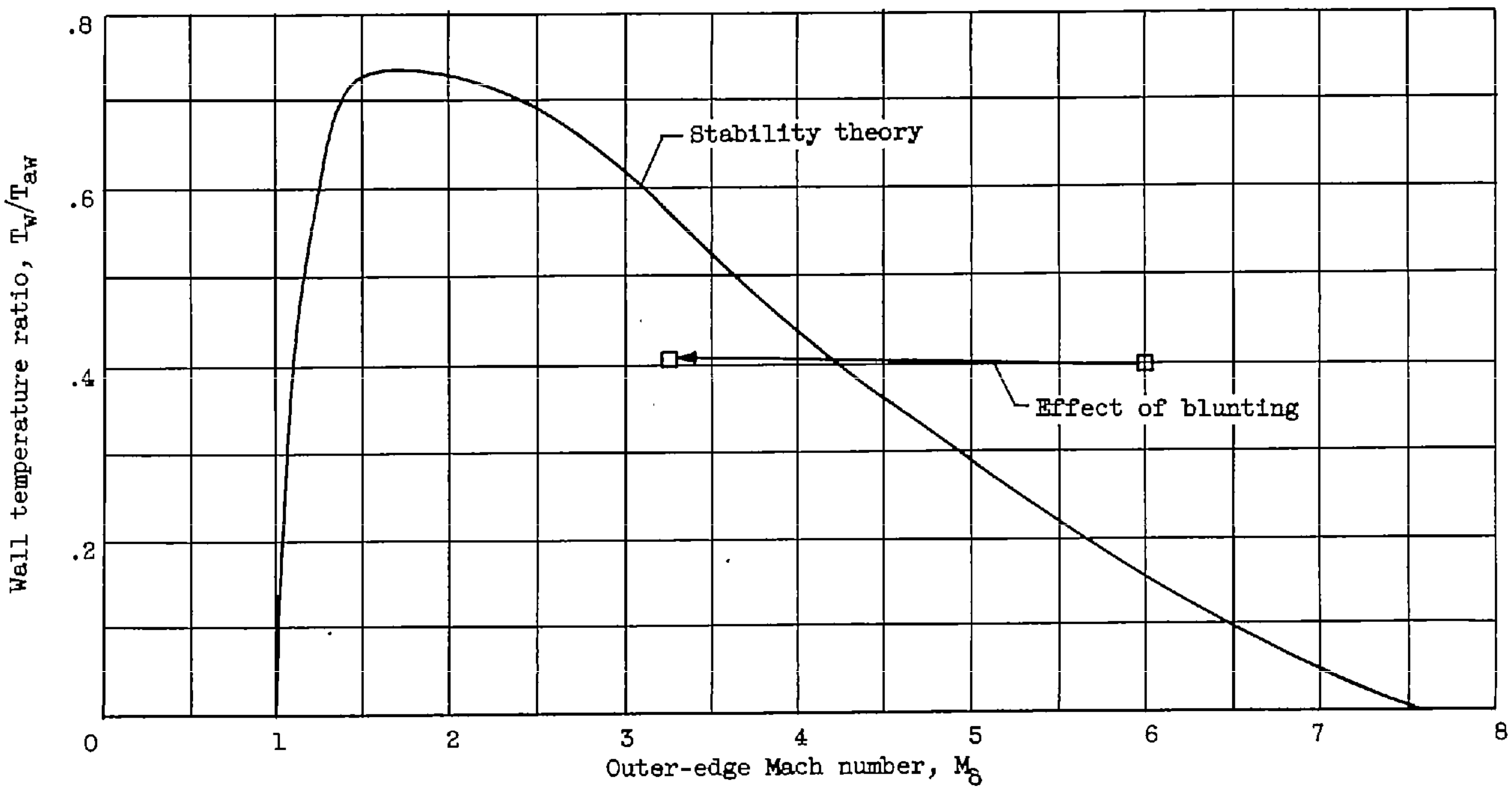

Figure 10. - Effect of nose blunting with respect to stability theory.

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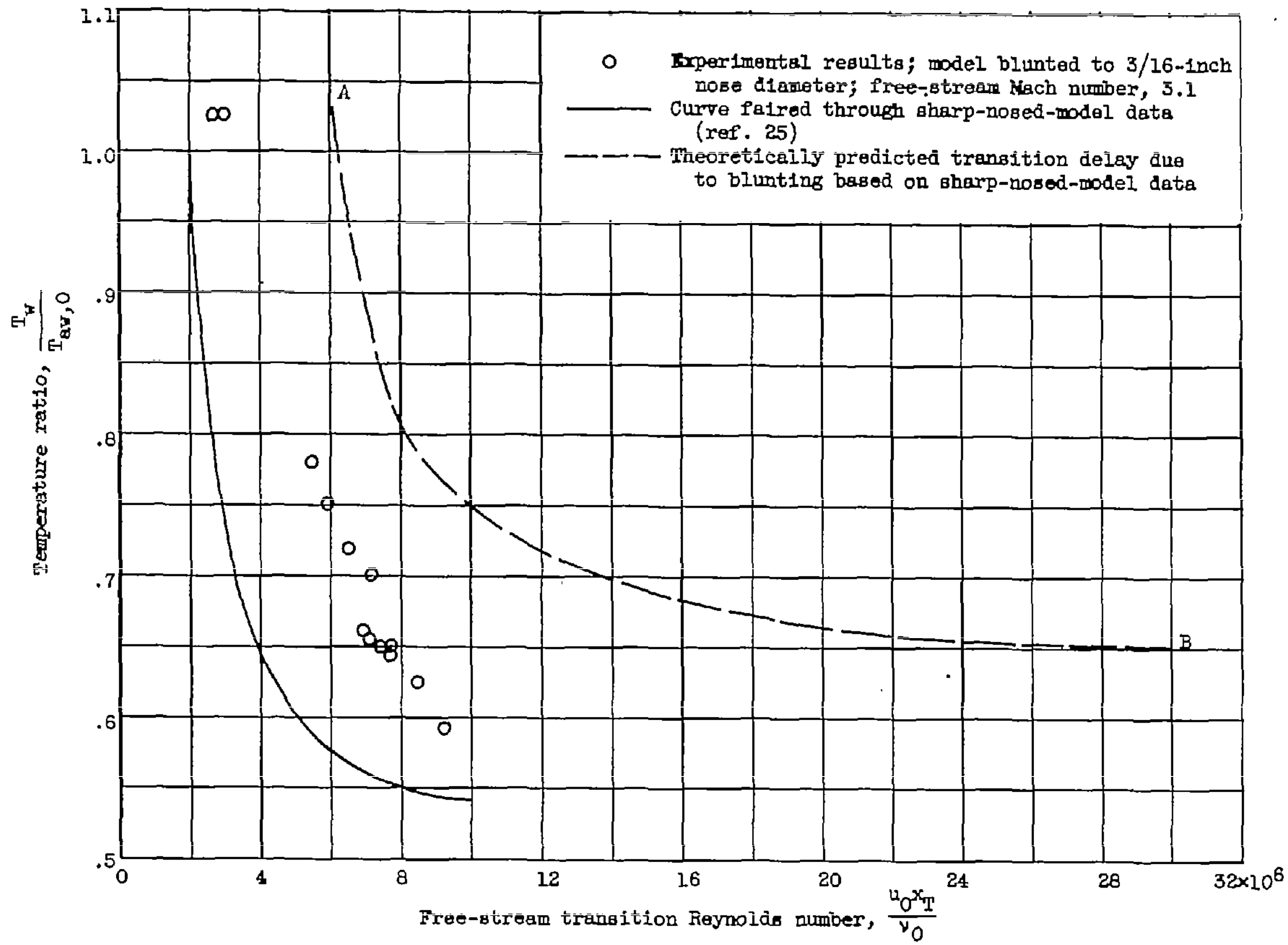

Figure 11. - Combined effect of blunting and cooling. 5003

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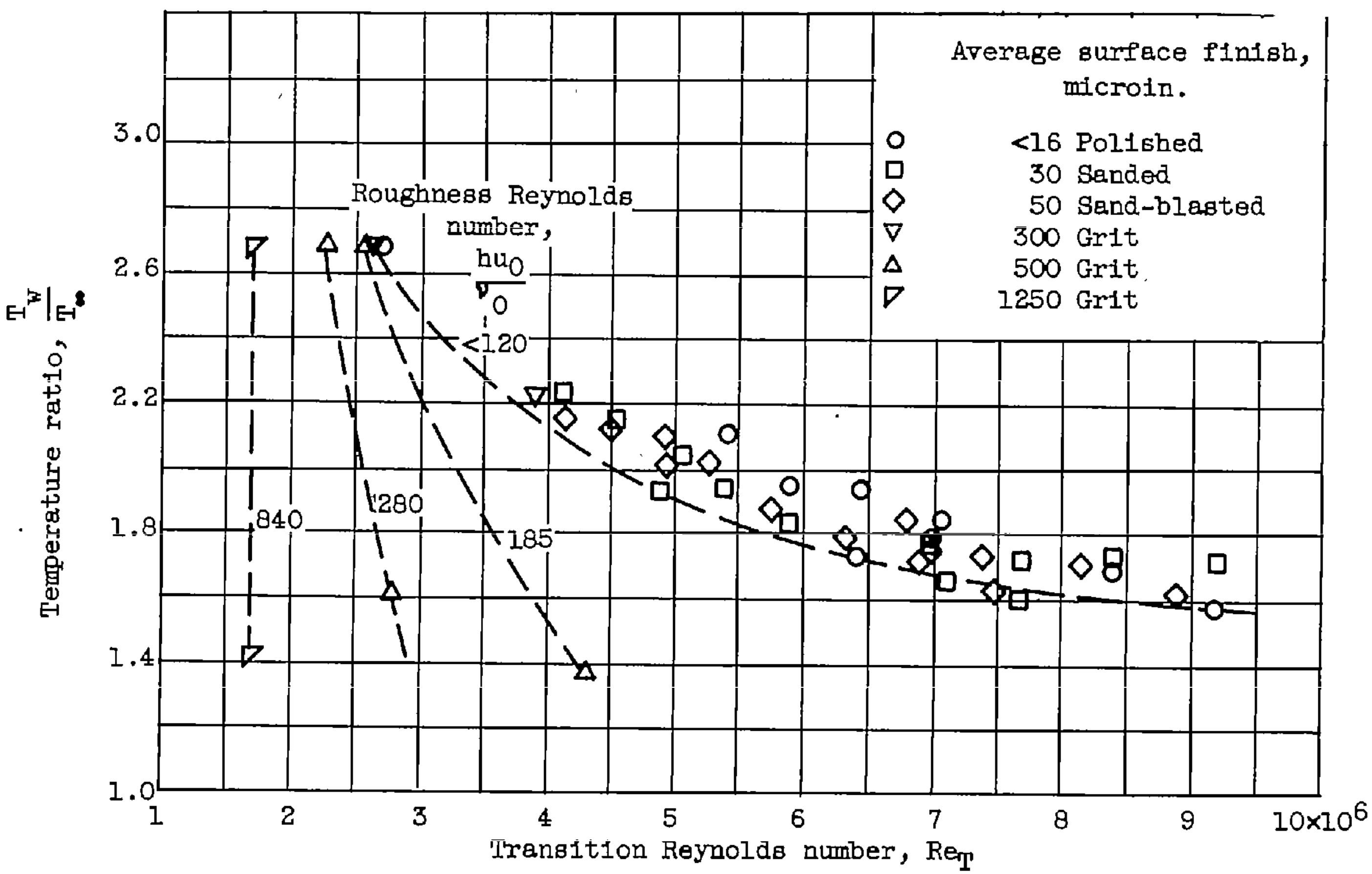

Figure 12. - Effect of surface finish. Free-stream Mach number, 3.1.

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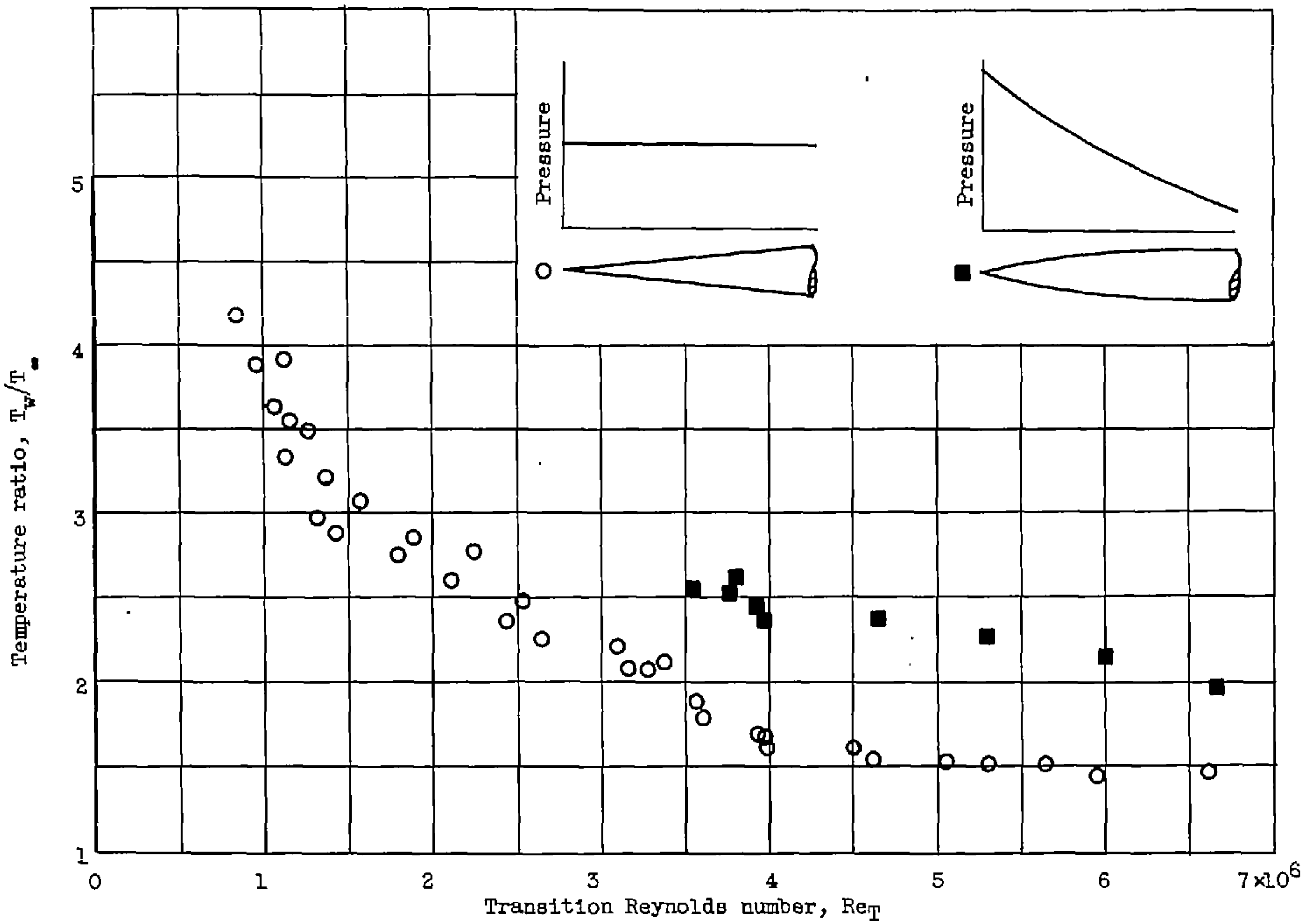

Figure 13. - Effect of favorable pressure gradient (ref. 25). Free-stream Mach number, 3.1.

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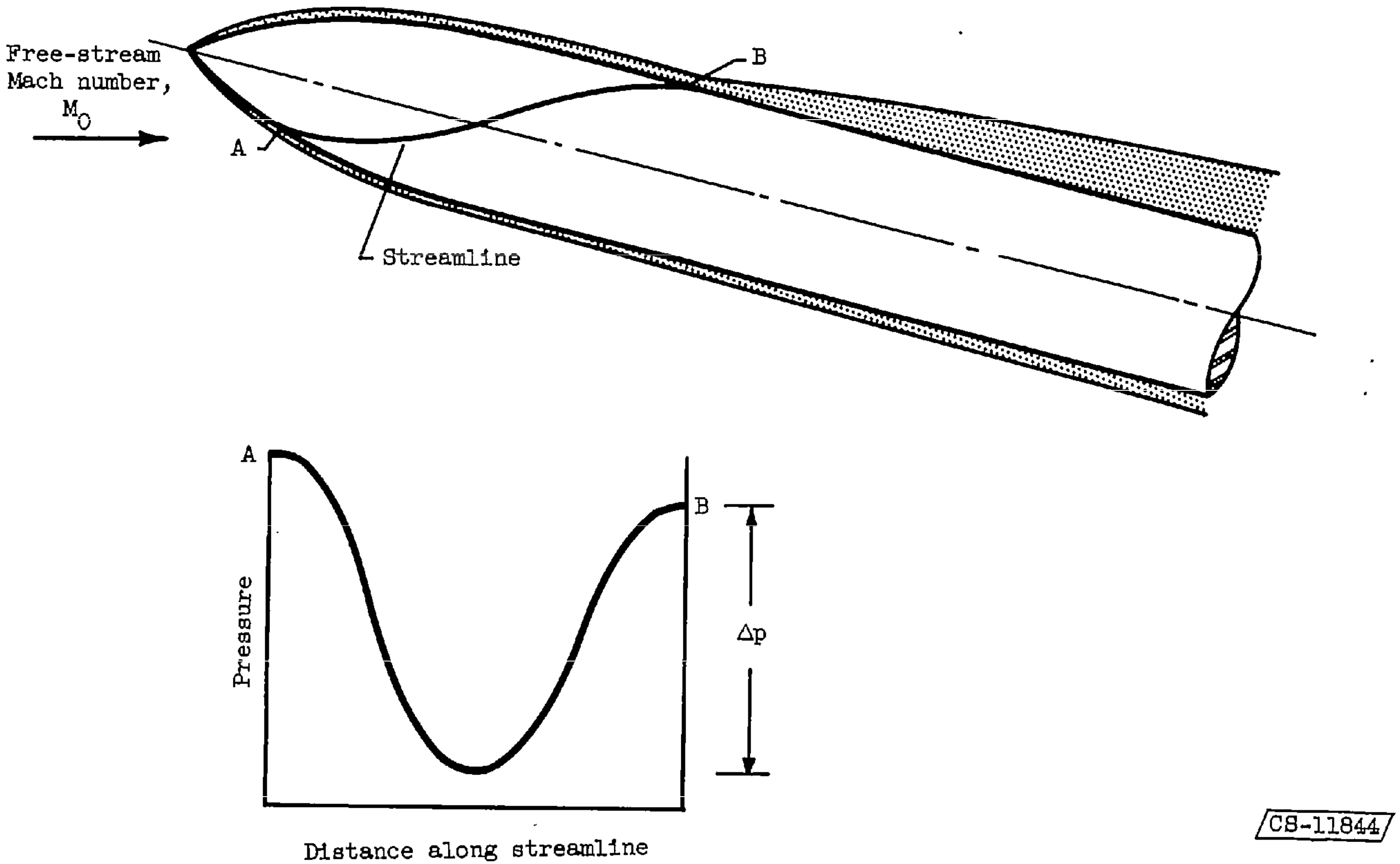

Figure 14. - Effect of angle of attack (ref. 29).

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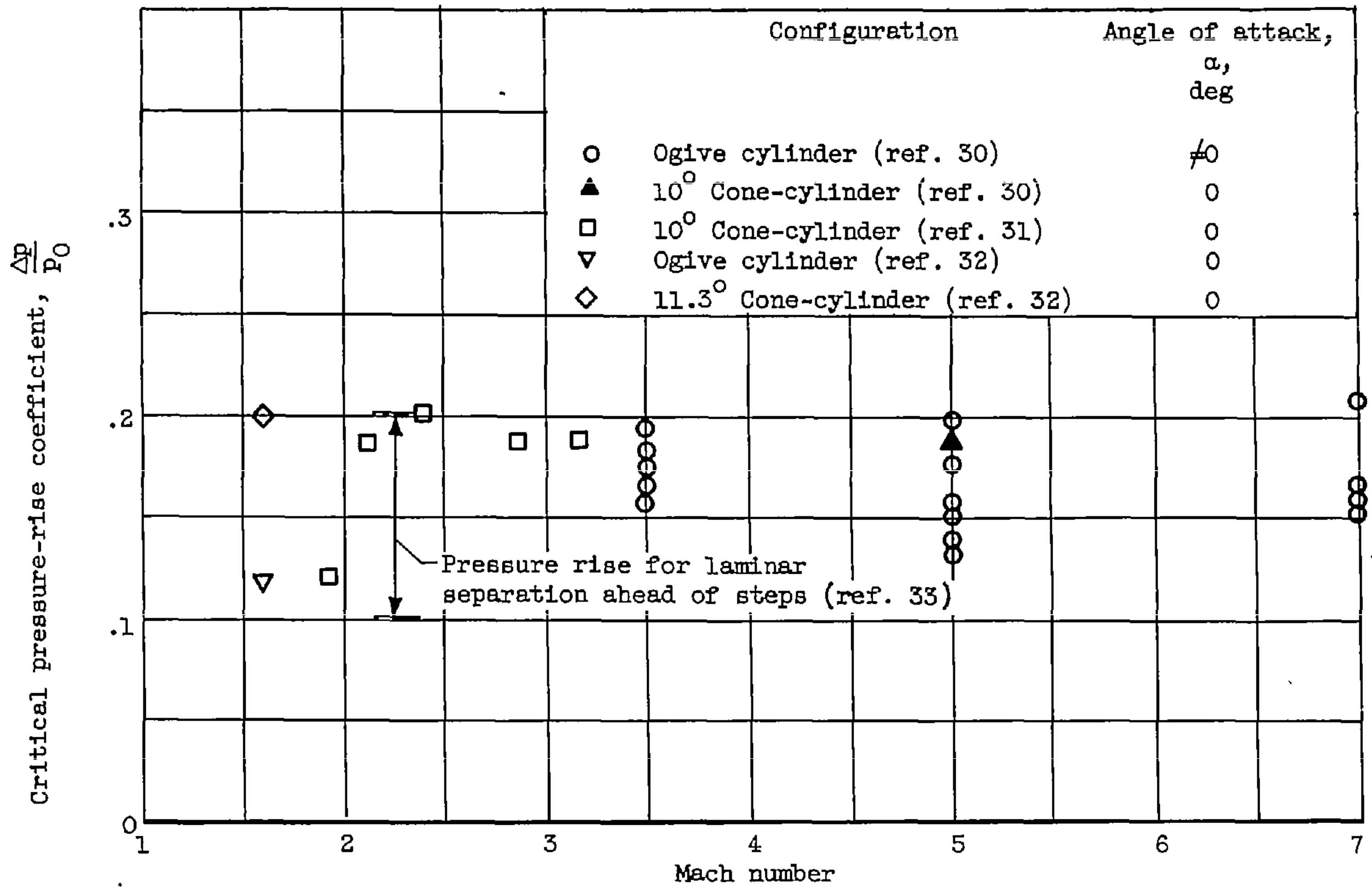

Figure 15. - Effect of pressure rise.

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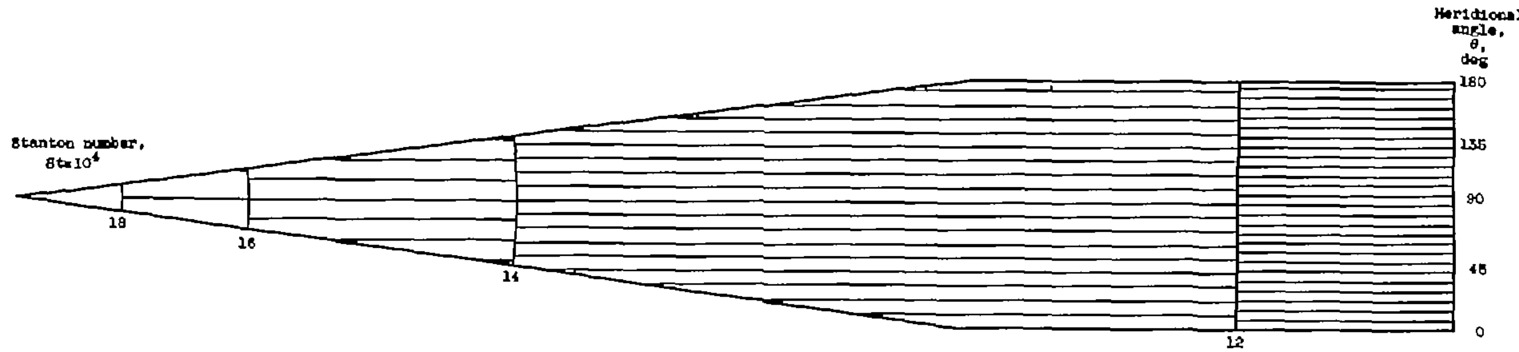
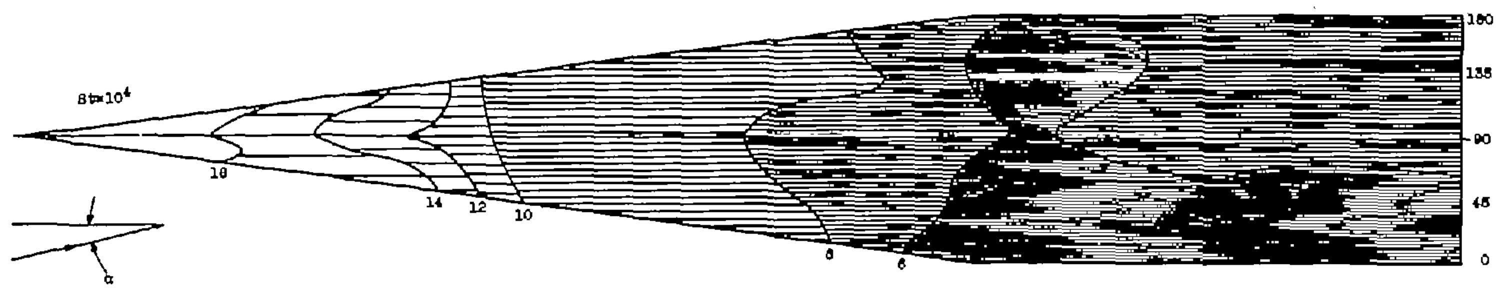
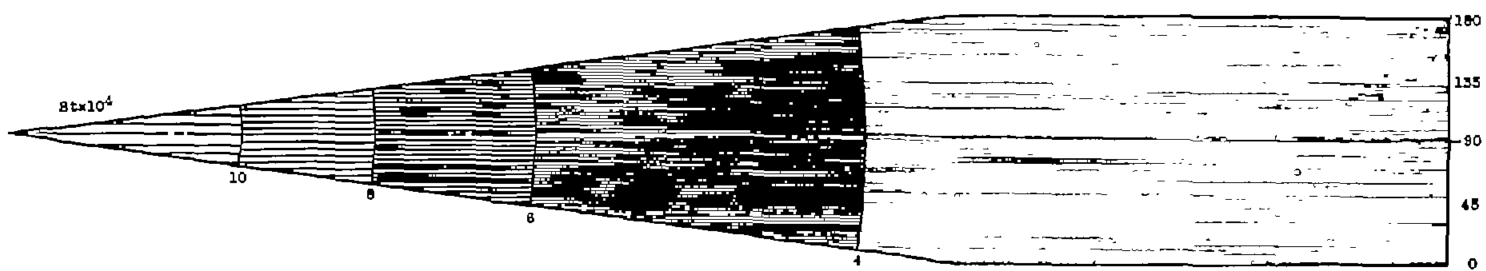


(a) Turbulent flow. Angle of attack, 0°.



(b) Actual flow. Angle of attack, 12°.



(c) Laminar flow. Angle of attack, 0°.

Figure 16. - Heat transfer at angle of attack. Free-stream Mach number, 3.1; Reynolds number $\frac{u_0}{\nu_0}$, 4.5×10^6 per foot.

NACA - Langley Field, Va.