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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 3166

AN EXTENSION OF THE INVESTIGATION OF THE EFFECTS OF HEAT TRANSFER ON BOUNDARY-LAYER TRANSITION ON A PARABOLIC BODY OF REVOLUTION (NACA RM-10) AT A MACH NUMBER OF 1.61

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Langley Field, Va.

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By K. R. Czarnecki and Archibald R. Sinclair

SUMMARY

The investigation (NACA TN 3165) of the effects of heat transfer on boundary-layer transition on a parabolic body of revolution (NACA RM-10 without fins) has been extended to higher Reynolds numbers, to greater amounts of heating, and to a more extensive study of the effects of surface irregularities and disturbances generated in the airstream. The tests were made at a Mach number of 1.61 and over a Reynolds number range from 2.5×10^6 to 35×10^6. The maximum cooling of the model used in these tests corresponded to a wall-to-free-stream temperature ratio of 1.12, a value somewhat higher than the theoretical value required for infinite boundary-layer stability at this Mach number.

The results indicate that the trend found previously of an increase in boundary-layer transition Reynolds number with increase in model cooling continued to higher Reynolds numbers. The highest transition Reynolds number obtained with cooling was 28.5×10^6. At this Reynolds number, the classical Tollmien-Schlichting wave type of boundary-layer instability was apparently overshadowed by surface roughness effects. The results indicated that, when transition was fixed by surface irregularities or airstream flow disturbances, cooling was not effective in obtaining laminar flow behind the irregularity or disturbance.

INTRODUCTION

In reference 1 are presented the results of a preliminary investigation of the effects of heat transfer on boundary-layer transition at

[1]Supersedes the recently declassified NACA RM L53B25, "An Extension of the Investigation of the Effects of Heat Transfer on Boundary-Layer Transition on a Parabolic Body of Revolution (NACA RM-10) at a Mach Number of 1.61" by K. R. Czarnecki and Archibald R. Sinclair, 1953.

a Mach number of 1.61. The tests were made on a slender parabolic body of revolution (NACA RM-10 without fins) which had a transition Reynolds number of about 11×10^6 for the case without heat transfer. The results indicated that if the boundary-layer transition Reynolds number for zero heat transfer is high, as it was in that investigation, then the sensitivity of transition to heating or cooling is high; if the zero-heat-transfer transition Reynolds number is low, as in the case of other investigations studied, then transition is relatively insensitive to heat-transfer effects. The preliminary investigation also showed that it was possible, by cooling the model an average of about 50° F, to increase the Reynolds number for which laminar flow could be maintained over the entire length of the body from 11×10^6 to 20×10^6, the limit of the tests.

The investigation has since been extended to determine the effectiveness of cooling at higher Reynolds numbers (up to about 35×10^6). In addition, tests were made with greater amounts of heating, and a more extensive study was made of the effects of surface irregularities and airstream disturbances on the ability of heat transfer to influence boundary-layer transition. In addition, the experimental techniques were expanded to include force tests. The results of this extended investigation are presented in this paper.

SYMBOLS

C_{D_F} skin-friction-drag coefficient, $\frac{\text{Skin-friction drag}}{qA}$

A maximum cross-sectional area of body

M free-stream Mach number

q free-stream dynamic pressure

L length of model

x distance along model

R Reynolds number based on body length and free-stream conditions

R_{tr} transition Reynolds number

T_e model equilibrium temperature without heating or cooling, °F

T_w	model surface temperature with heating or cooling, $^{\circ}F$
T_0	stagnation temperature, $^{\circ}F$
ΔT	average temperature difference for entire model, $T_w - T_e$, $^{\circ}F$
$\frac{\Delta T}{T_0'}$	average-temperature-difference ratio for entire model
T_∞	free-stream temperature, $^{\circ}F$
T_w'/T_∞'	average wall-to-free-stream temperature ratio for entire model

A prime mark over a temperature symbol (for example, T_0') indicates absolute temperature.

APPARATUS AND TESTS

Wind Tunnel

The investigation was conducted in the Langley 4- by 4-foot supersonic pressure tunnel which is a rectangular, closed-throat, single-return wind tunnel with provision for the control of the pressure, temperature, and humidity of the enclosed air. Changes in test-section Mach number are obtained by deflecting the top and bottom walls of the supersonic nozzle against fixed interchangeable templates which have been designed to produce uniform flow in the test section. The tunnel operation range is from about $\frac{1}{8}$ to $2\frac{1}{4}$ atmospheres stagnation pressure over a nominal Mach number range from 1.2 to 2.2. For qualitative visual-flow observation, a schlieren optical system is provided.

For the tests reported herein, the nozzle walls were set for a Mach number of 1.61. At this Mach number, the test section has a width of 4.5 feet and a height of 4.4 feet. Calibrations of the flow in the test section indicate that the Mach number variation about the mean value of 1.61 is about ±0.01 in the region occupied by the model and that there are no significant irregularities in stream flow direction.

Model and Techniques

A sketch of the NACA RM-10 model without fins, giving pertinent dimensions and construction details, is shown in figure 1 and a photograph of the model is presented as figure 2. The body has a parabolic-arc profile with a basic fineness ratio of 15. The pointed stern has been cut off at 81.25 percent of the original length, however, so that the actual body has a length of 50 inches and a maximum diameter of 4.096 inches.

A detailed description of the model and testing techniques is given in reference 1. Body contours were estimated to have an average deviation of less than 0.006 inch and a maximum possible deviation of about 0.020 inch. Surface roughness (measured by means of a Physicists Research Co. Profilometer, Model No. 11) varied between 4.5 and 6 microinches root mean square over most of the model and increased to about 12 microinches near the base. In the present tests, the only changes in testing technique from that given in reference 1 involved the substitution of an electrical heating element for the spray tubes and steam when the model was to be heated, the use of an electrical strain-gage balance to determine the body total drag, and the use of a set of pressure tubes to determine the base pressure. In addition, the end of the boundary-layer transition region (where boundary-layer velocity profiles had completed their transition to the turbulent type) was not determined because it was impossible to do so from the force tests and because it was often difficult to determine accurately from the boundary-layer profiles observed at the base of the model.

The heating element consisted of a steel rod wound with heavy resistance wire and was capable of operation to 1,600 watts. Current input into the heating element was controlled by means of a Variac.

For the force tests with the electrical strain-gage balance, base pressures were determined by means of four total-pressure tubes of 0.060-inch outside diameter (0.040-inch inside diameter) mounted on the surface of the sting in the plane of the model base at 90° intervals. The model skin-friction drag was then obtained by subtracting the base drag and a value of forebody pressure drag from the total drag determined by the balance. Values of forebody drag coefficient assumed for the model were 0.041 when the boundary layer was essentially laminar and 0.044 when the boundary layer was turbulent. These values were estimated from pressure measurements made on another model of identical shape.

In order to eliminate any residual effects of heating and cooling when determining boundary-layer characteristics under equilibrium or adiabatic conditions, all such tests were made as independent runs without heating or cooling, and ample time was allowed for the model surface temperatures to reach an equilibrium state.

Boundary-layer transition was determined from the force tests by plotting skin-friction coefficient against temperature as illustrated in figure 3. Transition was assumed to occur at the intersection of the two basically different segments of the curve. The nearly horizontal portion of the curve corresponds to a completely laminar boundary layer on the body, whereas the sharply sloped portion of the curve at the higher temperatures corresponds to the case where transition has occurred at the base of the body and is moving forward. The transition results thus obtained checked very well with schlieren observations. During the cooling tests, data were analyzed only on the warm-up cycle; during the heating tests, data were analyzed on both the heating and cool-down cycles.

Tests

Tests were made with the model in a smooth surface condition and with circumferential strips of cellophane tape, 0.003 inch thick, at the 3-percent, 25-percent, and 50-percent body-length stations. Care was used to assure that the tape adhered smoothly to the model surface. A series of tests was made with a wedge of 18-inch span mounted on the tunnel floor (see fig. 4) so that the shock off the wedge impinged upon the model usually somewhere on the forward half (x/L from 0.25 to 0.50). This wedge was cut down progressively in angle from about 10° to about 2/3° and in some cases in chord from 8 inches to 2 inches. A few tests were also made with a set of small wing or canard surfaces attached to the model at the 20-percent station (fig. 4). All tests made of configurations other than the basic smooth model were limited to tests with cooling only. The tests were made with the model at zero angle of attack. The tunnel stagnation pressure was varied from about 2 to 30 pounds per square inch absolute, which gave a Reynolds number range, based on the model length of 50 inches, of about 2.5×10^6 to 35×10^6. Tunnel stagnation dew point was usually kept below about -30° F except at the highest test Reynolds numbers when the tunnel air was dried as much as possible (dew point about -45° F).

RESULTS AND DISCUSSION

Tests With Smooth Model

Comparison with previous investigations.- The results of the present investigation of the effects of heating and cooling on boundary-layer transition on the smooth model are presented in figure 5 as a plot of Reynolds number for boundary-layer transition as a function of temperature-difference ratio $\Delta T/T_0'$. Force data and boundary-layer-pressure survey results are differentiated by the use of separate symbols. Included in figure 5 are the results for the beginning of boundary-layer transition obtained in previous tests of the NACA RM-10 model (ref. 1) and some typical results obtained for bodies, wings, and flat plates in other investigations (see refs. 2 to 6) and discussed in reference 1.

A comparison of the force and boundary-layer-pressure results indicates excellent agreement between the two methods of determining boundary-layer transition. The agreement between the results of the present investigation and those of the previous tests on the same model reported in reference 1 is also very good. The results indicate that as the model is heated to high temperatures the rate of change of R_{tr} with $\Delta T/T_0'$ decreases until at the highest temperatures investigated the transition Reynolds number and the rate of change of R_{tr} with $\Delta T/T_0'$ are of the same order of magnitude as those found in previous investigations (other than ref. 1). This result is to be expected since the boundary layer becomes more stable as the Reynolds number is decreased and consequently requires a greater amount of heating for destabilization, and since the curve is asymptotic to the zero Reynolds number axis.

As the model is cooled to lower temperatures, the slope of the curve of R_{tr} plotted against $\Delta T/T_0'$ increases, although the increase is at a slower rate than the decrease in slope encountered with increased model heating. The maximum transition Reynolds number obtained was 28.5×10^6 with a temperature-difference ratio of -0.161, or 92° F of model cooling.

Factors affecting maximum R_{tr} obtainable.- The maximum R_{tr} that could be obtained was apparently limited by two factors. The first, and probably the more important factor insofar as this investigation is concerned, was the great sensitivity of transition to surface roughness that results at high Reynolds numbers since the boundary layer becomes very thin. For greater values of R than 20×10^6, success in obtaining laminar flow by cooling was a random affair dependent upon how smooth the nose of the model was polished; changes in surface roughness between different runs, so minute as to defy detection, apparently determined whether or not laminar flow would be obtained. In many other instances during testing (but not in the runs described above) laminar flow would be obtained for several seconds or more but would disappear before any reliable temperature, force, or pressure data could be obtained. Examination of the model immediately after the run always showed a few minute nicks in the surface due to sandblasting. This sandblasting could not be eliminated at the higher tunnel stagnation pressures even with careful cleaning of the tunnel. Also, during tests at high Reynolds numbers, cooling of the model was so slow that a coat of ice with a rough snowlike surface would often form despite efforts to keep the tunnel unusually dry (dew point of about -45°). This ice probably aided in preventing the attainment of laminar flow. On the basis of these results, therefore, it appears possible that the Tollmien-Schlichting wave type of boundary-layer instability which is probably predominant at the lower Reynolds numbers is overshadowed by effects of surface roughness at higher Reynolds numbers. The sensitivity of laminar boundary-layer stability to surface roughness at high Reynolds numbers with cooling is similar to that experienced at low speeds with boundary-layer suction. This result may be expected because in both cases the boundary layer becomes very thin.

The second factor which influenced the maximum transition Reynolds numbers that could be obtained in this investigation was the lowest temperature that could be obtained near the nose of the model with cooling. This problem is shown in the temperature-distribution plot of figure 6. In some cases the lowest obtainable nose temperature was not as low as the average model temperature. Since at high values of Reynolds number boundary-layer transition occurs near the nose of the model, a deficiency in cooling in this region can easily account for the lack of success in obtaining laminar flow.

Because the average temperature of the model ahead of the point of boundary-layer transition is of considerably greater importance in the study of boundary-layer stability than the average temperature for the whole model as is used in figure 5, it is apparent that the experimental curve is somewhat in error and therefore only qualitative, but it is consistent with the proper trends. On the basis of the average model temperature ahead of the transition point, the slope of the experimental curve will be considerably increased. The proper average temperature that should be used could not be estimated from these tests.

Comparison with theory.- A comparison of the experimental results obtained in this investigation with the theoretical computations for a flat plate as calculated by Van Driest (ref. 7) is presented in figure 7. The comparison shows that the experimental curve of boundary-layer transition follows the trends of the theoretical curve for initial appearance of boundary-layer instability fairly well except for a displacement toward higher Reynolds numbers. If the experimental results are corrected to equivalent flat-plate Reynolds numbers by division of the Reynolds number by a factor somewhat less than 3 (according to ref. 7 the factor 3 applies to cones), the agreement is better. The results thus may be taken to evidence the existence of the classical Tollmien-Schlichting wave type of boundary-layer instability in these tests for Reynolds numbers up to the point where surface-roughness effects become predominant. It may be concluded, also, that Lees' theory of boundary-layer stability in compressible flows (ref. 8) as applied by Van Driest (ref. 7) can predict fairly well the general trends, at least, of the effect of heat transfer on transition.

Inspection of figure 7 shows that the curves of boundary-layer transition (experimental curve) and boundary-layer instability (theoretical curve) apparently become asymptotic to some critical value or values of wall-to-free-stream temperature ratio. Theoretically, the boundary layer will then be stable for all Reynolds numbers (to infinity) for temperature ratios less than this critical value. Since the most powerful effect of cooling on boundary-layer stability or transition occurs in the small-temperature-ratio range where the curves approach this asymptotic condition, it is possible that in this range damping would occur for disturbances of appreciable magnitude. Thus, if sufficient cooling were applied to cool the model below the critical temperature for complete stability, then the boundary layer might conceivably lose much of its sensitivity to surface roughness and traverse relatively rough surfaces without undergoing transition. The small amount of additional cooling required in the present case to test this possibility can be seen from figure 8, which shows the average heating and cooling ranges covered in this investigation and the theoretical wall-to-free-stream ratio required to stabilize completely the boundary layer. A margin to allow for inaccuracy in the theory is desirable.

Tests With Surface Roughness and Tunnel Flow Disturbances

Transition strips.- The results of the force tests made with cellophane-tape transition strips at the 3-, 25-, and 50-percent body-length stations are presented in figure 9. The theoretical curves were obtained by means of the extended Frankl and Voishel method (ref. 9) for the turbulent boundary layer and the Chapman and Rubesin method (ref. 10) for the laminar boundary layer. Mangler's transformation (ref. 11) was used in order to apply the flat-plate calculations to three-dimensional bodies. The short-dashed lines indicate cooling at constant Reynolds number and the arrows indicate the direction of change in skin-friction drag with decreasing temperature. Too much emphasis should not be placed upon the quantitative values of skin-friction coefficient with cooling, as it is believed that the quantitative accuracy of the balance deteriorates somewhat at low values of temperature. The direction of the trends, however, is not affected.

An analysis of the results for the adiabatic or equilibrium conditions (zero heat transfer) shows that the cellophane tape at the 3- and 25-percent body-length stations caused earlier-than-normal transition, whereas the strip at the 50-percent station had little or no effect. Attempts to obtain completely laminar flow by cooling for the cases with cellophane tape at the two forward locations were unsuccessful, even at Reynolds numbers only slightly above those at which transition first appeared. For the case of cellophane tape at the 50-percent station an attempt was made to obtain completely laminar flow by cooling at $R = 25.5 \times 10^6$. It was estimated that at this Reynolds number transition was slightly ahead of the 50-percent body station for the uncooled or adiabatic condition. The attempt was partially successful in that laminar flow was apparently established up to the strip of cellophane tape although not beyond. These results with surface roughness are apparently analogous to those obtained for the smooth body at high Reynolds numbers in that boundary-layer cooling is not effective in delaying transition when boundary-layer instability is associated predominantly with surface roughness.

Canard surfaces.- In practical airplane and missile configurations wings or small canard surfaces will be placed well forward on the body. In order to investigate the effects of such surfaces on transition with cooling, tests were made with small canard surfaces placed with the leading edge at the 16-percent body-length station (fig. 4) at zero angle of incidence. The results indicated that the surfaces strongly fixed transition at this location for Reynolds numbers as low as 2.5×10^6 and that cooling will be of little avail in obtaining laminar flow behind the surfaces.

Tunnel disturbances.- Past experience has indicated that laminar boundary layers become increasingly susceptible to separation, usually followed by transition, as the Reynolds number is increased. (For example, see ref. 12.) In fact, the indications are that at Reynolds numbers of the order of 20×10^6 to 30×10^6 laminar separation will occur as a result of a static-pressure rise relative to stream dynamic pressure of about 0.5 percent. This pressure rise can be generated by a shock having a turning angle of less than $1/5^{\circ}$. Thus, at these high test Reynolds numbers the laminar boundary layer will separate for pressure rises closely approaching the magnitude of the pressure disturbances that may exist in supersonic wind tunnels. In order to check the validity of this prediction, a series of tests was made with a wooden wedge of 18-inch span mounted on the tunnel floor so that the shock from the leading edge of the wedge would impinge somewhere on the forward half of the model (fig. 4).

The detailed results are not presented but they indicate that even the smallest wedge that could be tested (about $2/3^{\circ}$ with a chord of 2 inches) precipitated earlier-than-normal transition under adiabatic or zero-heat-transfer conditions. Also, cooling the model was ineffectual in obtaining laminar flow behind the point where the shock off the wedge impinged upon the model. Tests with a double thickness of cellophane tape replacing the wedge on the tunnel floor showed that the disturbance produced was so small as to have no effect under both the no-heat-transfer and the cooling conditions as compared with the smooth model without the sp cially induced disturbances. Apparently, the effects of finite disturbances that could originate in a test section of a supersonic tunnel are very similar to the effects of surface roughness on the ability of heat transfer to influence boundary-layer transition. An analysis, on the basis of reference 12, of the air flow in the region of the test section occupied by the model revealed that considerably higher values of R_{tr} than those obtained in the present investigation should be attainable before the flow disturbances present in the 4- by 4-foot supersonic pressure tunnel would have an effect.

SUMMARY OF RESULTS

An investigation of the effects of heating, cooling, surface irregularities, and airstream disturbances on boundary-layer transition on a parabolic body of revolution has been carried out at Reynolds numbers ranging from 2.5×10^6 to 35×10^6 in the Langley 4- by 4-foot supersonic pressure tunnel.

The following results were obtained:

1. The trend found previously (NACA TN 3165) of an increase in boundary-layer transition Reynolds number with increase in model cooling continued to higher Reynolds numbers. The trend of the results is in agreement with theoretical predictions.

2. The highest transition Reynolds number obtained in this investigation with cooling was 28.5×10^6. At this Reynolds number the classical Tollmien-Schlichting wave type of boundary-layer instability was apparently overshadowed by surface roughness effects.

3. In the presence of airstream disturbances (generated by thin wedges mounted on the test-section floor) and surface irregularities such as circumferential strips of cellophane tape and small canard surfaces, it was not possible to obtain laminar flow downstream of the irregularity or disturbance by application of the maximum cooling available in the present tests. It should be noted, however, that the lowest wall temperature in these tests was somewhat higher than the theoretical value for infinite stability at a free-stream Mach number of 1.61. It is possible, therefore, that some further reduction in wall temperature might alter this result.

Langley Aeronautical Laboratory,
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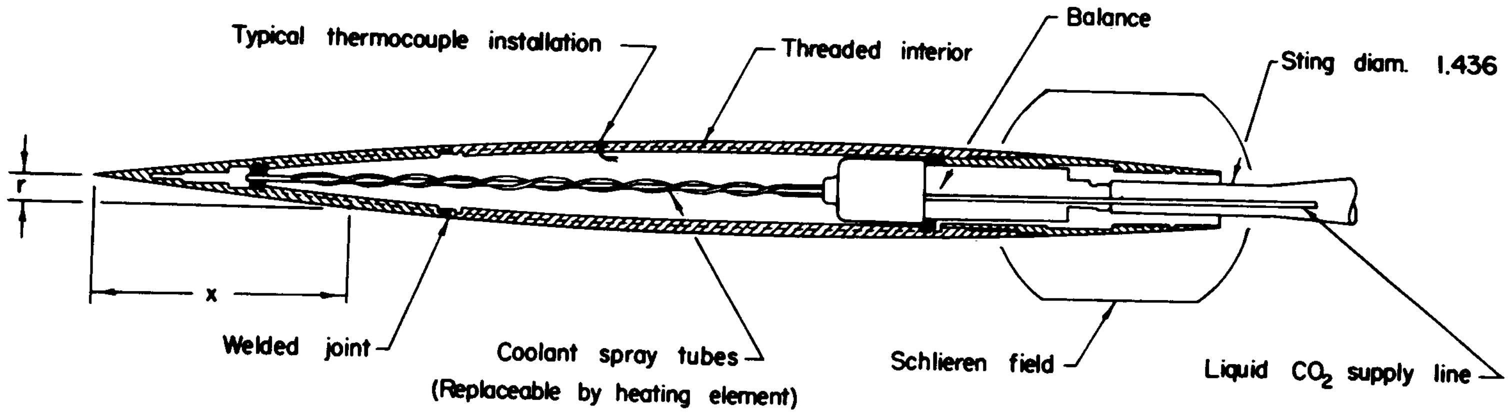

Body profile equation: $r = 0.1333x - 0.00217x^2$
Model length 50.0
Max. diam. 4.096

Thermocouple	Locations	
Station	No.	Spacing
3.0	2	180°
12.6	2	180°
22.4	4	90°
32.0	2	180°
37.1	2	180°
46.0	2	180°

Figure 1.- Sketch of NACA RM-10 model and apparatus for heating and cooling. All dimensions are in inches.

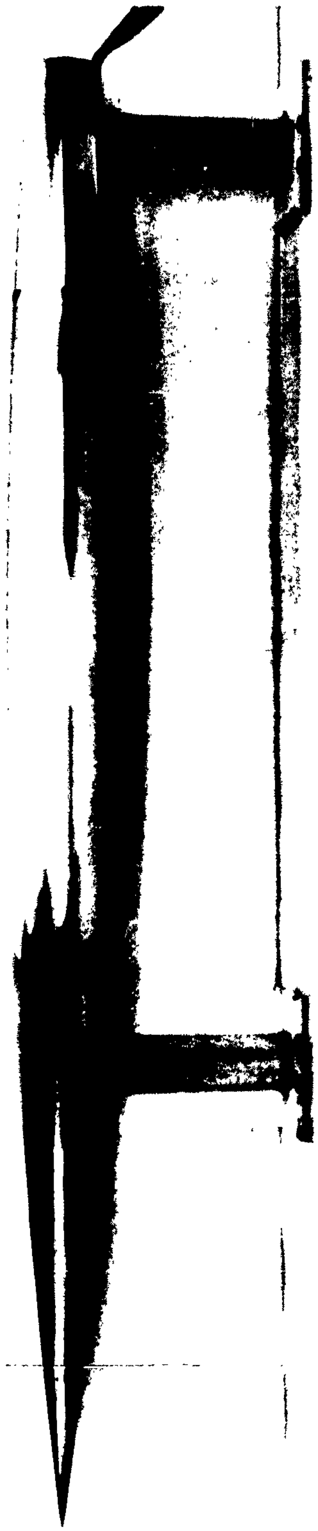

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Figure 2.- NACA RM-10 model.

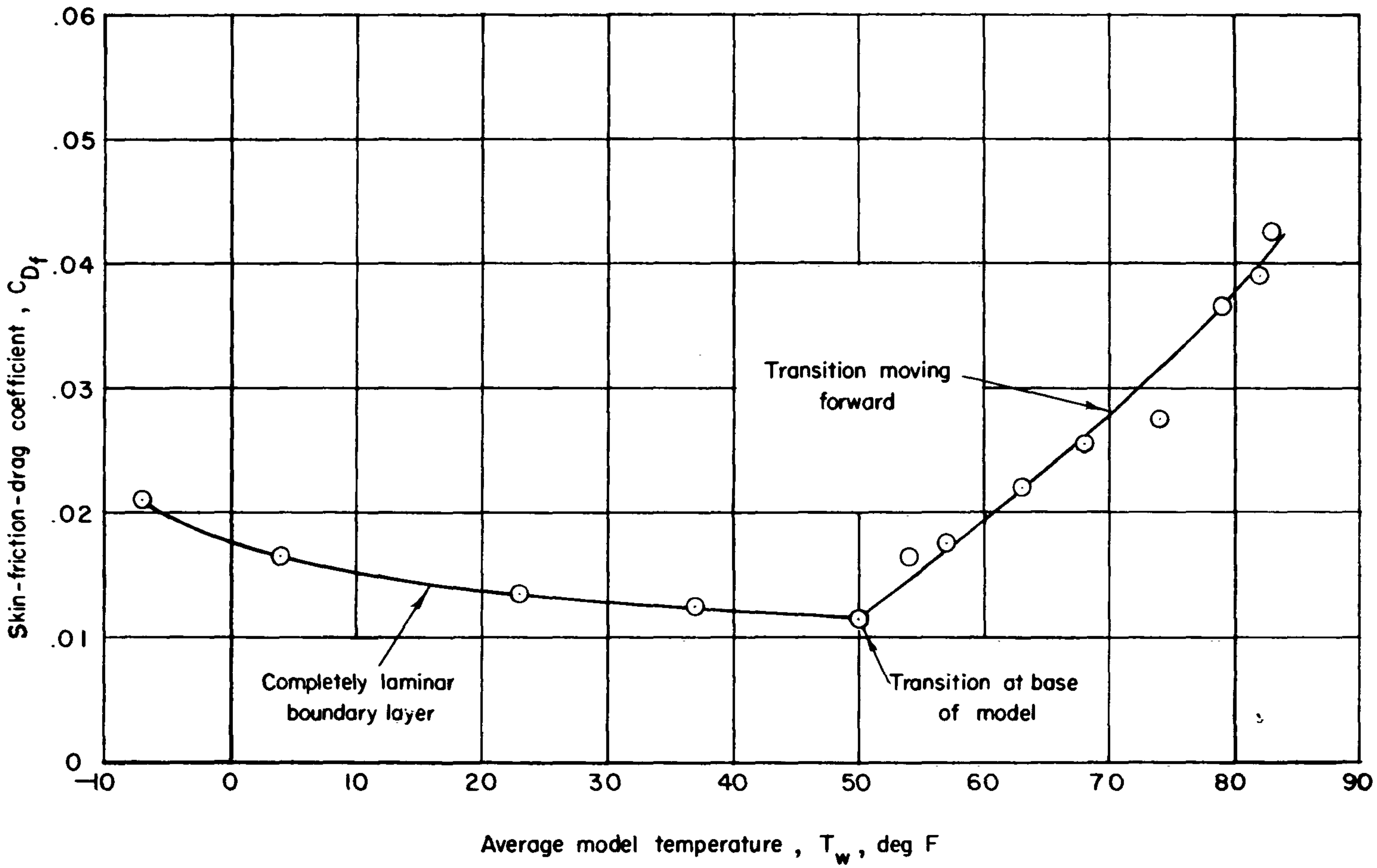

Figure 3.- Typical variation of skin-friction-drag coefficient with model surface temperature at constant Reynolds number. Force tests; $R = 19 \times 10^6$; $M = 1.61$.

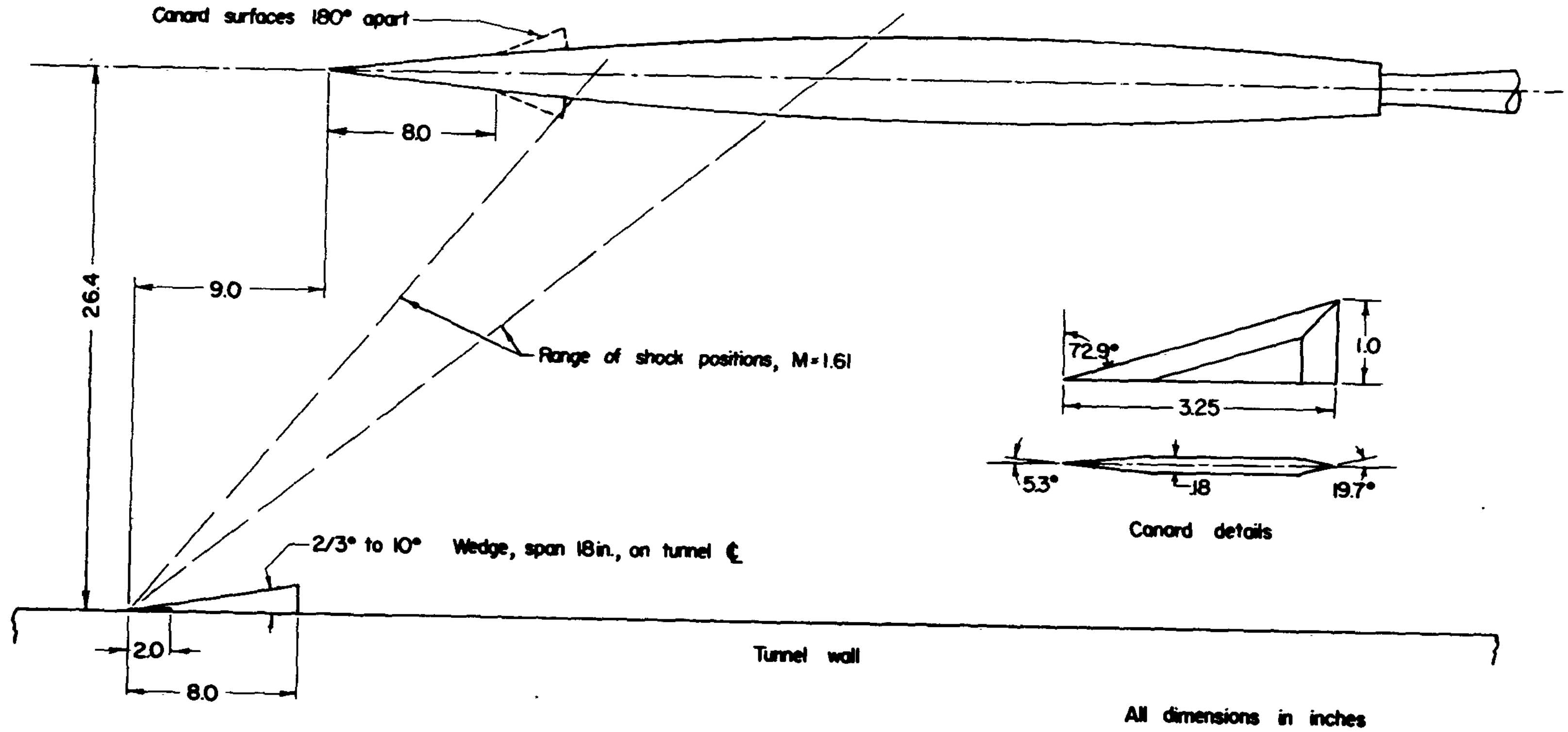

Figure 4.- Diagrammatic sketch showing canard location on model and wedge setup used in investigating effects of tunnel flow disturbances.

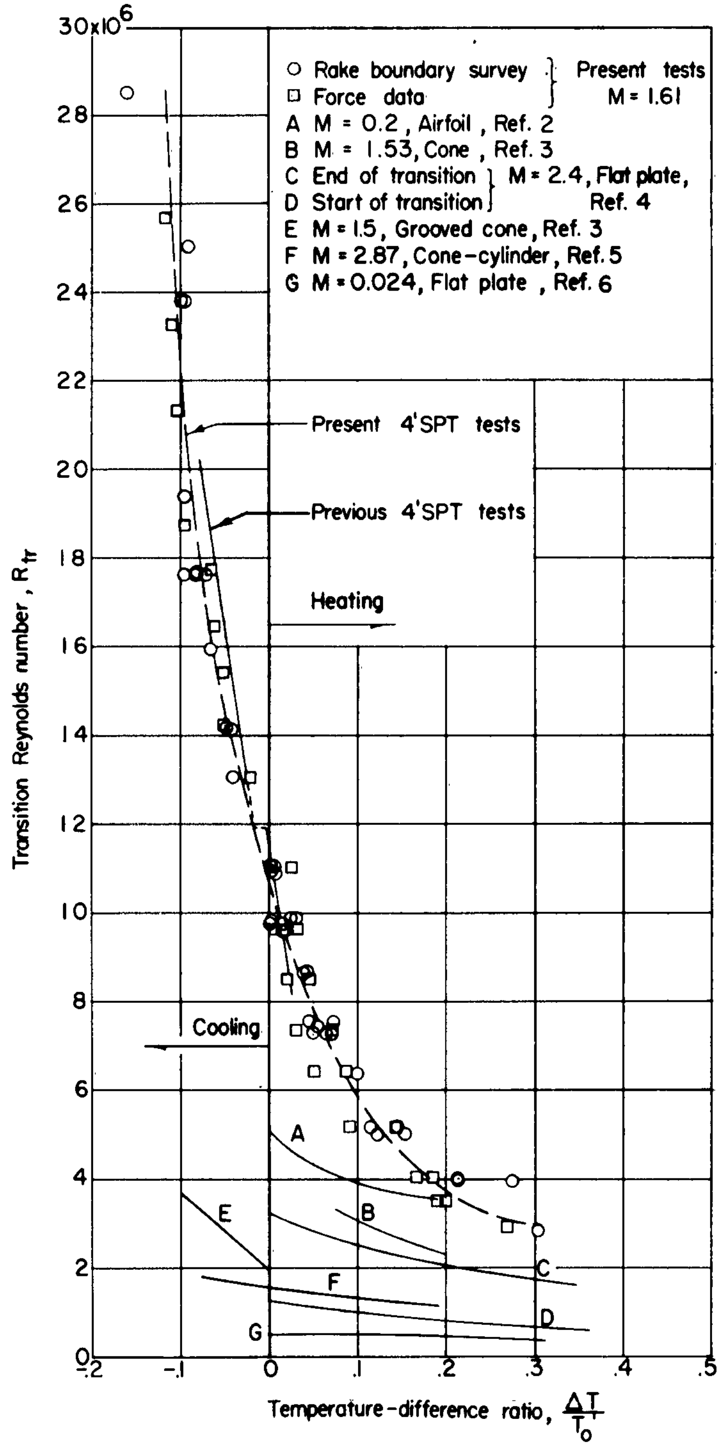

Figure 5.- Effect of heating and cooling on boundary-layer transition for present tests and comparison with results from other sources.

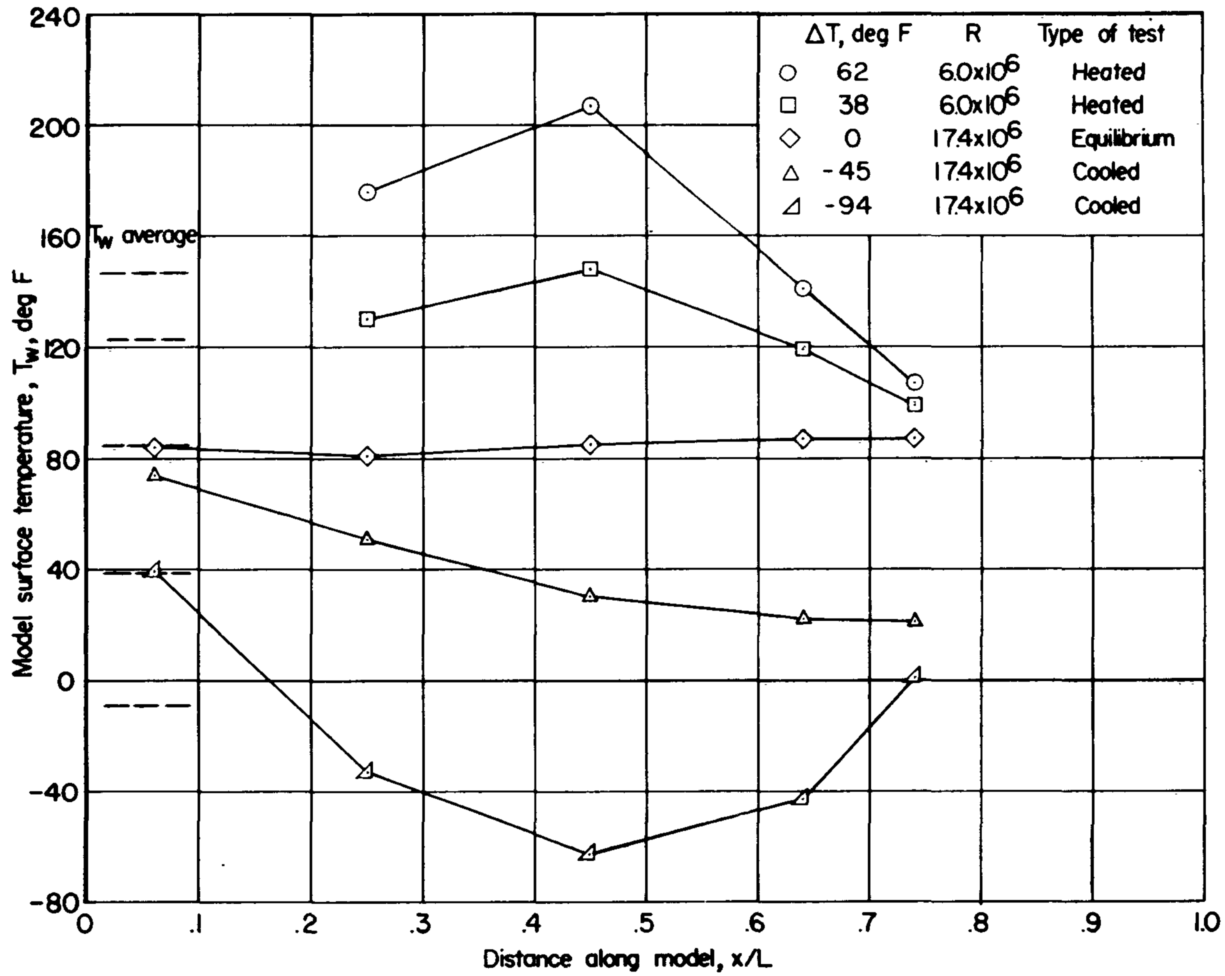

Figure 6.- Typical temperature distributions on model surface. $M = 1.61$; $T_0 = 110^\circ$ F.

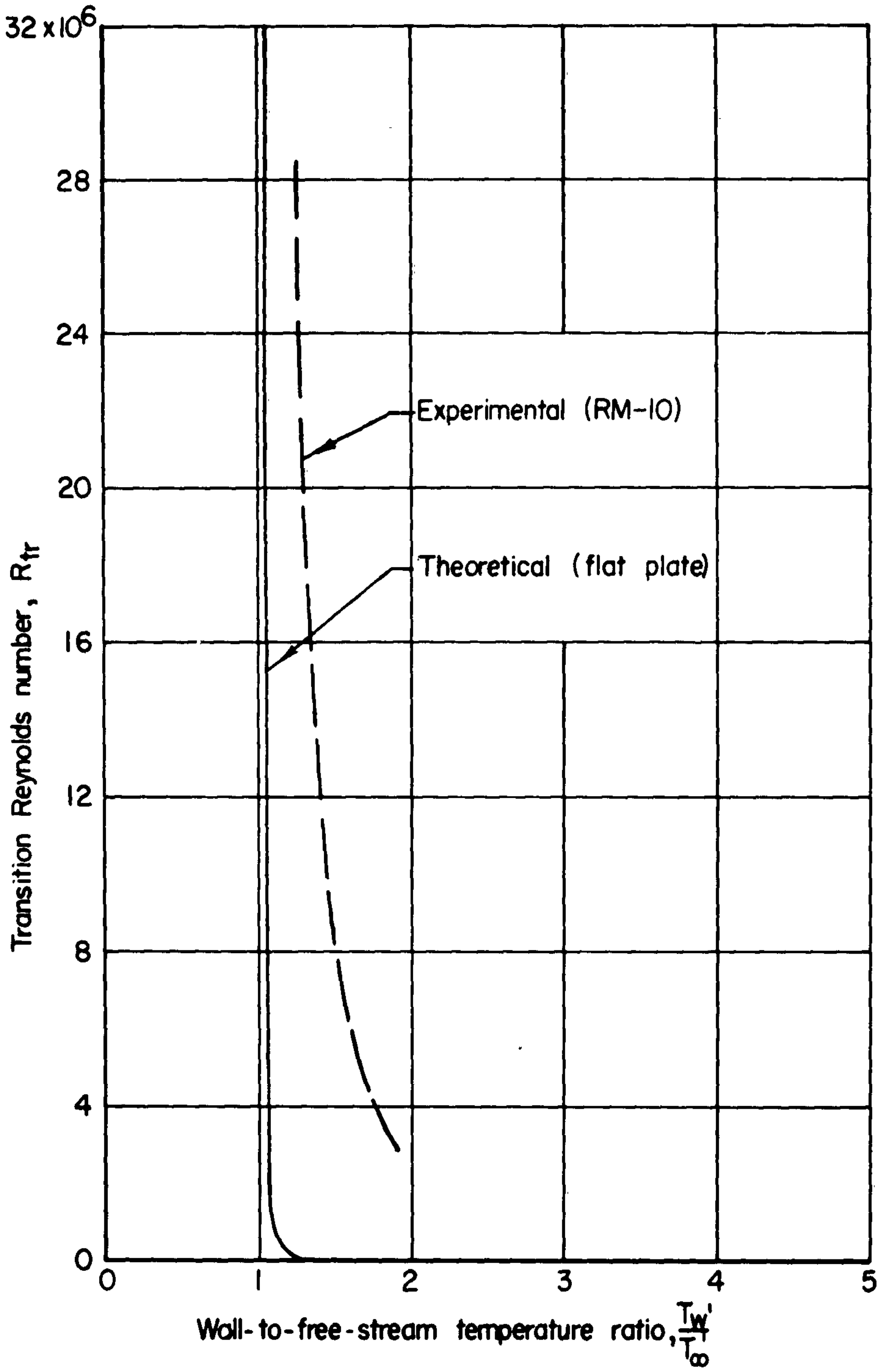

Figure 7.- Comparison of experimental transition Reynolds number for NACA RM-10 model and theoretical calculations of boundary-layer stability for flat plate (ref. 7).

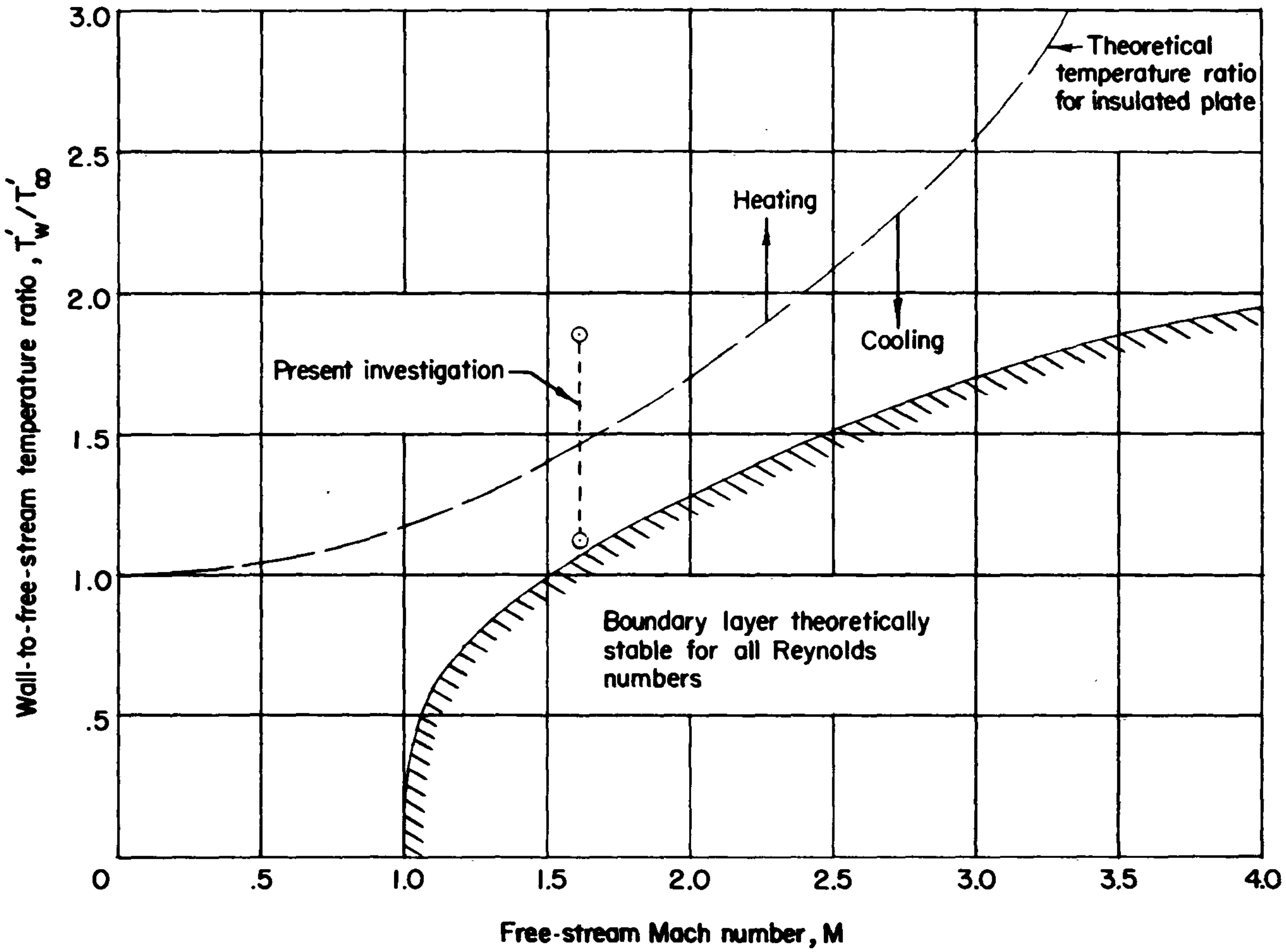

Figure 8.- Heating and cooling range covered in boundary-layer-transition investigation on NACA RM-10 model. Theoretical curve from reference 7.

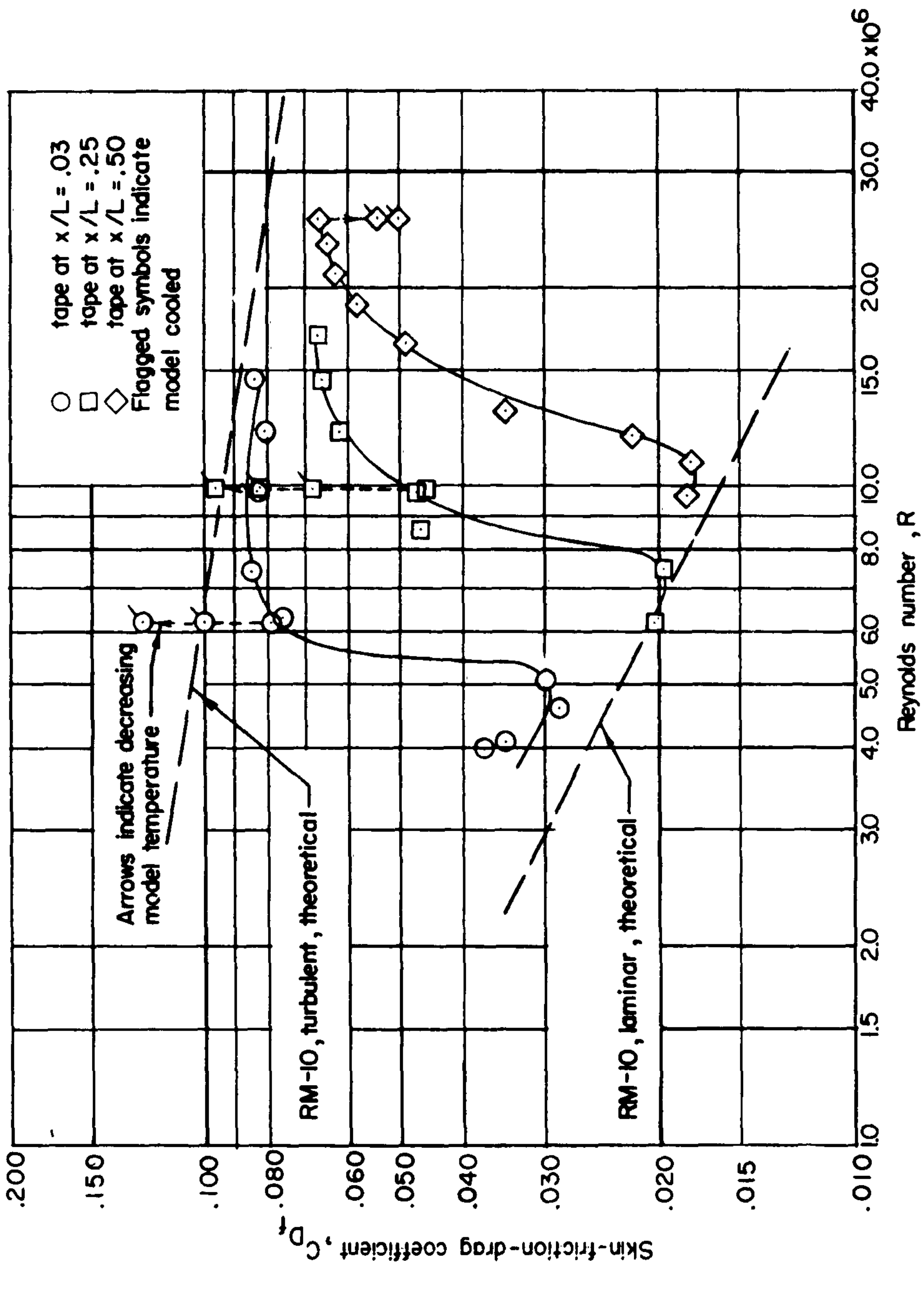

Figure 9.- Variation of skin-friction-drag coefficient with Reynolds number for various locations of transition strip, including the effects of cooling. M = 1.61.

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This paper covers the extension of a previous investigation of the effects of heat transfer on boundary-layer transition to higher Reynolds numbers, to greater amounts of heating, and to a more extensive study of the effects of surface roughness and wind-tunnel flow disturbances. The tests were made at a Mach number of 1.6 and over a Reynolds number range from 2.5×10^6 to 35×10^6: A comparison is made between the experimental results and theory.

Copies obtainable from NACA, Washington

1. Flow, Laminar (1.1.3.1)
2. Flow, Turbulent (1.1.3.2)
3. Aerodynamics with Heat (1.1.4)
4. Bodies - Surface Conditions (1.3.2.4)

I. Czarnecki, Kazimierz Roman
II. Sinclair, Archibald R.
III. NACA TN 3166
IV. NACA RM L53B25

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AN EXTENSION OF THE INVESTIGATION OF THE EFFECTS OF HEAT TRANSFER ON BOUNDARY-LAYER TRANSITION ON A PARABOLIC BODY OF REVOLUTION (NACA RM-10) AT A MACH NUMBER OF 1.61. K. R. Czarnecki and Archibald R. Sinclair. April 1954. 21p. diagrs., photo. (NACA TN 3166. Formerly NACA RM L53B25)

This paper covers the extension of a previous investigation of the effects of heat transfer on boundary-layer transition to higher Reynolds numbers, to greater amounts of heating, and to a more extensive study of the effects of surface roughness and wind-tunnel flow disturbances. The tests were made at a Mach number of 1.6 and over a Reynolds number range from 2.5×10^6 to 35×10^6: A comparison is made between the experimental results and theory.

Copies obtainable from NACA, Washington

1. Flow, Laminar (1.1.3.1)
2. Flow, Turbulent (1.1.3.2)
3. Aerodynamics with Heat (1.1.4)
4. Bodies - Surface Conditions (1.3.2.4)

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