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FREE-FLIGHT BOUNDARY LAYER TRANSITION INVESTIGATIONS AT HYPERSONIC SPEEDS

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ABSTRACT

A number of programs have been conducted recently in the NOL ballistics range to determine the boundary layer transition Reynolds number on slender bodies at Mach numbers from 3 to 15. One series of tests has been conducted at Mach numbers of 3 and 5 to determine the effect of high rates of heat transfer on the stability of the laminar boundary layer. For these tests, a sharp 10-degree total-angle cone was launched through a temperature-controlled range. Shadowgraph photographs were obtained of the model in free flight at different heat transfer rates, and transition Reynolds numbers were calculated from the observed length of laminar flow. These tests indicated that the stability of the boundary layer is very sensitive to the heat transfer rate at the conditions tested. Another series of tests has been conducted on slightly blunted 12.6-degree cones at Mach numbers of 9 and 13.3 and on 18-degree cones at Mach numbers of 10 and 15. This combination of high Mach number and slender body results in a bow shock wave that lies so close to the model surface that the boundary layer transition location cannot be observed in the range photographs. Therefore, a method was devised for determining the transition Reynolds number from a series of total drag measurements. This method is explained, and the results of these tests are discussed. The results are also substantiated by observation of the wake flow behind the models.

INTRODUCTION

Boundary layer transition has been of great interest to both the theoretician and the experimentalist. This interest is kindled by the importance of the nature of the boundary layer in a wide variety of aerodynamic problems. The magnitude of drag forces depends upon whether the boundary layer is laminar or turbulent. The aerodynamic heating rate is strongly influenced by the nature of the boundary layer. Other important phenomena that are affected by the location of boundary layer transition include the mass injection rate in boundary layer cooling systems, the location of the boundary layer separation point, and the structure of the wake.

A great deal of effort has been made to theoretically predict the boundary layer transition Reynolds number for a number of situations. However, a theoretical analysis that would include all of the parameters that affect the stability of the boundary layer would be extremely difficult, if not impossible. These parameters include Mach number, surface roughness, nose or leading edge bluntness, heat transfer rate, and pressure gradient. One of the most influential parameters on the location of transition, and one of the most controversial, is the effect of heat transfer rate. A theoretical study of a compressible boundary layer by van Driest (reference (1)) indicated that a stabilizing effect on the boundary layer is produced by increasing the rate of heat transfer.
from the fluid to the body. This result has been generally accepted in certain flow regimes. In addition, it was suggested in references (1) and (2) that at certain Mach numbers the laminar boundary layer can be completely stabilized by the presence of an appropriate amount of cooling of the boundary layer regardless of the Reynolds number. A large amount of experimental evidence substantiated this predicted stabilizing effect for moderate rates of cooling (references (3), (4), (5), and (6)). However, some more recent experiments (references (7), (8), and (9)) have indicated that this stabilizing effect does not exist at extremely high rates of heat transfer.

The boundary layer destabilizing effect, "transition reversal", was first noted on highly cooled models in wind tunnels. Because of the testing techniques required in a wind tunnel facility to obtain the high rates of heat transfer necessary to produce transition reversal, the effect was not completely accepted initially. One area of suspicion was the surface condition of the models. In order to obtain the high heat transfer rates, the model surfaces were cooled to temperatures as low as -340°F. At these temperatures, air components such as oxygen, carbon dioxide, and water vapor would condense on the model surface and produce a form of surface roughness that could act as a boundary layer trip. As the tests would progress, the surface temperature would rise due to aerodynamic heating and the frost would disappear. It was not certain whether the accompanying increase in laminar run was due to the decrease in apparent surface roughness or due to the decrease in heat transfer rate.

A portion of the tests presented in this paper were conducted to investigate the effect of high rates of heat transfer on the stability of the laminar boundary layer. The data were obtained on 10-degree total-angle cones. The cones were launched in a free-flight ballistics range at Mach numbers of 3 and 5. It was possible to obtain heating rates in the ballistics range greater than those of references (7), (8), and (9) without cooling the model due to the naturally high recovery temperature of the air associated with free-flight testing. This precluded any possibility of frost forming on the models and causing premature transition to turbulent flow. The heat transfer rate was varied by changing the velocity while the range temperature was varied appropriately to maintain a constant Mach number for each series of tests.

Another interesting consideration is the effect of Mach number on boundary layer transition. Experimental wind tunnel programs such as those reported in references (10) and (11) indicate that the Mach number effect is relatively small at Mach numbers of less than 2. However, the effect becomes very important at Mach numbers of 4 and greater. The remaining tests presented in this paper were planned to determine the effect of Mach number on the transition Reynolds number for two cone configurations. The tests were performed in a free-flight ballistics range at hypersonic Mach numbers of approximately 9 to 15. The effect of Mach number is pronounced in this regime with an increase in Mach number showing a marked increase in the free-stream transition Reynolds number.

DESCRIPTION OF TESTS

Heat Transfer Tests

Tests have been conducted in the Naval Ordnance Laboratory's Pressurized Ballistics Range No. 3 to study the effect of high rates of heat transfer on boundary layer transition. The tests were performed at Mach numbers of 3 and 5. The test configuration for both numbers was a 10-degree total-angle cone. It was desirable to use as large a model as possible to provide a long surface for boundary layer observations; however, the final size was limited by the capabilities of the launcher. At Mach number 3 it was possible to use models with
base diameters of 1.2 inches while the maximum diameter model that could be successfully launched at Mach number 8 had only a 0.5-inch diameter base. Some of the models had a cylindrical afterbody with fins to increase the stability and decrease the angular motion. It was essential to limit the angle of attack due to the extreme difficulty in analyzing the effect of angle of attack on transition. A sketch of the model configurations used in these tests is shown in figure 1. Figure 2 is a photograph of a typical model and sabot combination.

Extreme care was taken to obtain and preserve both a smooth surface and a sharp tip on the models prior to launching. The models were constructed of tool steel or titanium and the surface was finished by grinding. It was found that center line average surface finishes of 1 to 6 microinches root-mean-square could be obtained consistently by this method and that further attempts to improve the surface by polishing only resulted in introducing waviness in the surface. The surface finish was measured on the aft portion of all models with a Talysurf recorder. Surface finish readings were made over the entire length of a random sampling of the models, and the results showed the entire surfaces to be uniformly smooth. A typical Talysurf reading is shown in figure 3. No attempt was made to produce a radius on the tip of the models. The tips were ground as sharp as possible and inspected immediately prior to launching. Models with bent or broken tips were discarded. A magnified photograph of a model tip is shown in figure 4.

The heat transfer rate was varied by adjusting the ambient air temperature in a 20-foot section of the range. The launch velocity was varied with the range temperature to obtain the desired test Mach number. This was necessary because of the variation in speed of sound with temperature. After the desired range temperature and Mach number had been chosen, the range pressure was varied to produce the required Reynolds number. The transition Reynolds number was obtained by measuring the flow properties such as velocity, temperature, pressure, and length of laminar flow in the boundary layer. The length of laminar flow was obtained from spark shadowgraph photographs taken of the model in free flight. The photographs have sufficient clarity and detail that accurate estimates can be made of the transition location. Figure 5 is a spark shadowgraph photograph of a model in free flight showing transition on the body.

**Hypersonic Transition Tests**

Tests have been conducted in the Naval Ordnance Laboratory's Hyperballistic Range No. 4 to determine boundary layer transition Reynolds numbers for two cone configurations at hypersonic Mach numbers. The configurations tested were a 12.6-degree total-angle cone with a nose to base radius ratio of 0.035 and an 18-degree total-angle cone with a nose to base radius ratio of 0.050. The 12.6-degree cone was tested at free-stream Mach numbers of 9 and 13.3. The 18-degree cone was tested at free-stream Mach numbers of 10 and 15. Two size models of each cone were tested. The 12.6-degree cone had base diameters of 1.00 inches or 1.25 inches while the 18-degree cone had base diameters of 0.525 inches or 1.25 inches. Figure 6 is a sketch of the model configurations used in the hypersonic transition tests and figure 7 is a typical model and sabot combination.

At Mach numbers of approximately 8 or less, the boundary layer can be seen with sufficient detail on the range shadowgraph photographs that the boundary layer transition point can be measured optically. However, at higher Mach numbers a number of factors make an accurate measurement of this type more difficult. First, on slender bodies flying at high Mach numbers the bow shock lies so close to the body surface that the boundary layer is optically distorted on the range photograph. Also, in order to achieve the high launch Mach number, the model size is somewhat reduced. This also makes it more difficult to see the boundary layer.
Finally, at the relatively high pressures required to produce transition at hypersonic Mach numbers, the nose regions of the models are sometimes luminous as a result of the extremely high aerodynamic heating rate. As a result of the luminosity, the range pictures are further reduced in quality. For these reasons, a new method was used to determine the transition Reynolds numbers at hypersonic Mach numbers.

A series of each model configuration was launched over a wide spread in Reynolds number based on body length, \( Re_L \), for each test Mach number. The tests were started at a sufficiently low Reynolds number so that the entire boundary layer was definitely laminar. This could be verified by observation of the wake flow. If transition could be seen in the wake, the flow over the body had to be laminar. As the test Reynolds number was increased from shot to shot, the total drag would decrease as long as the entire boundary layer remained laminar. This decrease in total drag results from a decrease in the laminar skin friction with increasing Reynolds number since the pressure drag is not affected by the Reynolds number. This change in drag can be clearly measured in the range since the total drag is measured extremely accurately. This method is especially applicable since the skin friction drag represents a major portion of the total drag for the present model configuration and test conditions. As shown in reference (12), the skin friction drag is as much as 20 to 30 percent of the total drag in these tests. When the test Reynolds number is increased sufficiently to cause transition to move on to the aft portion of the body, a slight increase in total drag is noted. This is due to the higher magnitude turbulent drag acting on the aft portion of the body. As the Reynolds number is increased farther, the total drag increases as more of the body becomes affected by the higher magnitude turbulent skin friction drag. If the total measured drag is plotted versus body Reynolds number, the Reynolds number at which the drag first starts to increase can be noted. This body Reynolds number is the transition Reynolds number since the body length is also the length of laminar flow at this condition.

**DESCRIPTION OF FACILITIES**

**Pressurized Ballistics Range No. 3**

This facility is a 3-foot diameter, 300-foot long steel tube that can be evacuated to a pressure of about 0.004 atmospheres or pressurized to approximately 6 atmospheres. There are twenty-seven divergent light dual-plane spark shadowgraph stations along the range tube spaced alternately five and eight feet apart. At these stations, high quality shadowgraph photographs are obtained in both the vertical and horizontal planes. A complete projectile trajectory is determined from this photographic information. These photographs have sufficient clarity and detail that accurate estimates can be made of the location of boundary layer transition.

There is a 21-foot long section of the range that contains more flexible instrumentation. A heater can be installed into this section of the range. The present range heater is capable of heating the air in this section to approximately 850°F. Four shadowgraphs can be obtained in the vertical plane and four in the horizontal plane within the heater. The heater is equipped with thermal shields that are placed over the photographic plates until approximately 20 seconds prior to launching the model to protect the film emulsion. There are also doors on both ends of the heater that contain the heat within the special section of the range. These doors remain closed until approximately 1 second before launching. Figure 8 is a photograph of the heater outside of the range.

Model launchers available for use with this facility include a two-stage light-gas gun with a smooth bore launch tube of 20 mm diameter and a two-stage light-gas gun with a smooth bore launch tube of 1.25 inches diameter. With these guns models have been launched at velocities in
excess of 20,000 feet per second. Figure 9 is a photograph of this facility.

**Hyperballistics Range No. 4**

This facility is a 10-foot diameter, 1,000-foot long steel tube that can be evacuated to a pressure of about 0.2 mm Hg. The instrumentation in this range includes twenty-seven dual-plane spark shadowgraph stations which give a coverage of 430 feet of testing length. The spacing between the first twenty-six stations varies and is either 7, 13, or 20 feet for a total testing length of 340 feet. There is then one additional dual-plane station at a distance of 430 feet from the first station. For the remaining length of the range there are four single-plane spark shadowgraph stations. These extend the measuring length for drag measurements to 790 feet.

There are two smooth bore two-stage light-gas guns available for launching the models in this facility. The launch tubes for these guns are 1.6 and 2.5 inches in diameter. Scaled models for aerodynamic testing have been launched at velocities above 20,000 feet per second. Figure 10 is an illustration of this facility.

**DATA REDUCTION PROCEDURES**

**Heat Transfer Tests**

The location of boundary layer transition on the model was determined from spark shadowgraph photographs taken of the model during its flight down the range. The photographic plates used in these tests consisted of a film emulsion on a 14 by 17-inch glass plate. Photographs were obtained of the model at each data station. Depending upon the location of the model between the light source and the photographic plate, the magnification of the model by the optical system can vary from 1 to 1.3 times its actual size. The magnification of the model on each photographic plate was measured by comparing the size of the image to the true size of the model. The location of transition was measured from the nose of the image directly from the glass photographic plate. This length was multiplied by the magnification factor for the plate to obtain the actual location of transition on the model.

The velocity, Mach number, and angle of attack were obtained through a visual observation of the shadowgraph plates and a standard data reduction program described in reference (13). The recovery temperature of the air in the heated section of the range was obtained from the equation:

\[ T_x = T_H \left(1 + r \frac{Y-1}{2} M_a^2 \right) \]  

The recovery factor, \( r \), was assumed to be 0.85 in the region of laminar flow and 0.88 in the turbulent region.

The surface temperature of the model was calculated with the aid of an IBM-7090 computer. The method of calculation is presented in reference (14). The calculations were made by dividing the model into sections along the center line of the model. The temperature and physical properties of the model were considered constant within each section. The temperature history of each section was then calculated by considering the convective heat flow into each section due to aerodynamic heating and conductive heat flow from section to section due to the possible unequal temperature of neighboring sections. The results of a typical calculation are shown in figure 11. From figure 11, it can be seen that aside from the immediate area of the tip the model temperature increases only slightly during the test. The small temperature rise of the model is limited by the short time required for the model to travel through the range at the test Mach numbers.

The Reynolds number based upon the length of laminar flow on the model and the local flow properties was chosen as the parameter to indicate
the degree of stability of the boundary layer. The local Reynolds number based on the length of laminar flow (transition Reynolds number) is:

\[ R_{tr} = \frac{p_0 U_e X_{tr}}{\mu e} \quad (2) \]

Upon introducing Sutherland's viscosity law and the equation of state for a perfect gas, equation (2) may be rewritten as

\[ R_{tr} = A \frac{p_0 U_e X_{tr}}{T_e^{1/2}} (T_e + C) \quad (3) \]

In order to obtain the local pressure, velocity, and temperature, it was first necessary to know the corresponding free-stream conditions. The range pressure was measured directly by pressure gages attached to the range. The free-stream velocity was determined at each data station by measuring the time required for the model to travel from one data station to another. Two thermocouples were placed at each data station within the heated section of the range to measure the ambient temperature at the stations. Using these free-stream conditions, the local properties that appear in equation (3) were calculated with the aid of tables of flow properties for yawed cones (reference (15)) and tables of isentropic flow (reference (16)). Two rays of the boundary layer could be seen on each shadowgraph photograph, one on the windward surface of the model and one on the leeward surface. In general, the model would be at some angle of attack to the free stream. In order to obtain the local flow properties along each ray, the angular orientation of the ray with respect to the pitching plane was determined from the horizontal and vertical components of the angle of attack. These angle of attack components are calculated at each data station. The method of calculation is presented in reference (13). After the orientation of the ray has been determined, the local flow conditions can then be determined from references (15) and (16).

The Reynolds number based on the momentum thickness and local flow properties at the location of transition was also calculated. By applying Mangler's transformation to flow over a flat plate, it can be shown that the boundary layer momentum thickness for a cone is given in terms of the momentum thickness on a flat plate obtained at the same local Mach number, local Reynolds number, and wall to local temperature ratio by the following simple relation

\[ \theta_{cone} = \sqrt{3} \theta_{flat \ plate} \quad (4) \]

The flat plate momentum thickness is given by

\[ \theta_{flat \ plate} = \frac{C_F X}{2} \quad (5) \]

Equation (5) is obtained by integrating the momentum equation. The laminar value of the mean skin friction coefficient used in equation (5) was obtained from reference (17). By using the momentum thickness in equation (3) in place of the length of laminar flow, \( X_{tr} \), a local Reynolds number based on momentum thickness can be expressed as

\[ R_{tr} = A \frac{p_0 U_e \theta}{T_e^{1/2}} (T_e + C) \quad (6) \]

**Hypersonic Transition Tests**

The hypersonic transition tests were conducted in the 1,000-foot Hyperballistics Range No. 4. The main difference in the method of obtaining data between this facility and the Pressurized Ballistics Range used in the heat transfer tests is the size of the spark shadowgraph photograph. In this facility, a shadowgraph image of the model and flow field is cast on a section of a spherical mirror and a photograph is taken of this image. The photographic plates used in this system are only 5 by 7 inches. The
image, as it finally appears on the range plates, is smaller than the actual model. Therefore, it is somewhat more difficult to obtain some of the flow details in this facility than it is with the enlarged photographs obtained directly in the Pressurized Ballistics Range. However, observations were made of both boundary layer and wake details with good results. The location of transition, in either the boundary layer or the wake, was determined essentially the same as in the heat transfer tests with the exception that the image magnification factor was always less than unity. The data reduction method of reference (13) is also used in this facility to obtain the velocity, Mach number, and angle of attack at each data station.

The drag coefficient can be measured to a high degree of accuracy in a ballistics range. This accuracy can be attributed to the fact that the downrange trajectory as a function of time can be determined very closely. The location of the model can be determined from the photographic plate to ± 0.002 feet and the time at which the photograph was taken is known within ± 2 x 10^{-7} seconds. As discussed in reference (12), the equation which describes the longitudinal motion of the model, assuming a constant drag coefficient and the square law of drag, can be approximated by

\[ z' = z_{0} + \frac{V_{m}}{2} \left( t - t_{m} \right)^2 + \frac{1}{3} \frac{2}{V_{m}}^{3} \left( t - t_{m} \right)^3 \tag{7} \]

where \( g = \frac{C_{D} A_{m}}{m} \frac{p}{RT} \tag{6} \)

and \( z_{0} \) and \( t_{m} \) are the values of distance and time for the station nearest midrange. Equation (7) is fitted by the method of least squares to the measured time-distance data. The drag coefficient and velocity that are computed by equation (7) are the values at the station nearest midrange.

Since it is not possible to maintain zero angle of attack in ballistics range tests, the drag coefficient obtained in equation (7) contains the effect of angle of attack. The mean squared angle of attack for each launching can be defined by

\[ \bar{\alpha}^2 = \frac{1}{Z_{1} - Z_{F}} \int_{Z_{F}}^{Z_{1}} \alpha^2 \, dZ \tag{9} \]

where \( Z_{F} \) and \( Z_{1} \) are the location of the first and last range stations, respectively. In order to eliminate the effect of angle of attack from the measured drag coefficients, the coefficients were plotted as a function of the mean squared angle of attack. Straight lines were then faired through each group of data obtained at the same flow conditions of Mach number and Reynolds number. The slope of these lines, \( m \), was then used to reduce the drag coefficient measured for each shot to an equivalent zero angle of attack value with the equation

\[ C_{D_{0}} = C_{D} - m \bar{\alpha}^2 \tag{10} \]

Calculations were made of the components of the total drag (pressure drag, friction drag, and base drag) and compared with the measured total drag obtained from the 12.6-degree and 18-degree cones. The method of calculating the drag in the laminar flow region is described in reference (16). This method takes into account the curved bow shockwave that exists for slightly blunted slender bodies and allows a variation in total pressure along the outer edge of the boundary layer on the conical portion of the body. It assumes, however, that the static pressure along this surface is constant and equal to the inviscid sharp nosed cone value. An addition to this method was used in the turbulent flow region. In this addition, the calculation is started at the transition point of a turbulent boundary layer with a starting momentum thickness that is equal to the momentum thickness of the laminar boundary layer at that point. The calculations were performed using an IBM 7090.
DISCUSSION OF RESULTS

Factors Affecting Transition

Boundary layer transition is affected by a number of factors including surface roughness, tip blunting, pressure gradient, Mach number, and heat transfer rate. An experimental investigation into the effect of any one or group of these parameters is made more difficult if the others cannot be properly controlled or accounted for. One of the primary concerns in the present tests was to vary as few parameters as possible at a time so that the results could be interpreted more easily.

One of the parameters that has been studied by a great number of investigations is surface roughness. A number of tests, references (10), (19), and (20), in which the surface roughness was varied in a controlled fashion have shown that transition can be created prematurely by increasing the surface roughness. However, other tests such as those of references (21) and (22) show that roughness of a smaller order of magnitude can actually produce greater laminar runs than are obtained on a smooth surface. Since it did not seem possible to quantitatively predict the effects of surface roughness for the present tests, it was decided to minimize the effects by making the surface as smooth as practical. This approach seemed appropriate for another reason. At the high rates of heat transfer to the surface of the model that existed in these tests, the boundary layer was relatively thin. This would indicate that the effect of a particular size roughness would be more pronounced than in a case of small or no heat transfer. The maximum size of a discrete roughness protuberance found on the 10-degree cones was approximately 30 microinches. The 12.6-degree and 18-degree cones had some protuberances as large as 120 microinches. An analysis was made to see if these protuberances were large enough to affect transition location.

In reference (23), a method is described through which it is possible to calculate whether or not a particular roughness size can influence the location of transition at a given set of flow conditions. In order to apply the method, a value is chosen for the critical roughness Reynolds number, \( R_\text{k} \), based on roughness height and flow conditions within the boundary layer at a distance \( k \) from the wall. Using this value and local flow conditions that exist at the outer edge of the boundary layer, a minimum value of \( R_\text{k}^{\text{e}} \) that can cause premature boundary layer transition can be calculated. This value of \( R_\text{k}^{\text{e}} \) can then be compared with the actual value of \( R_\text{k} \) for the given flow conditions and roughness height. If the actual value is less than the calculated minimum value, the given roughness size is not sufficiently large to trip the boundary layer. There is a great deal of uncertainty in the proper value of \( R_\text{k}^{\text{e}} \). Values of \( R_\text{k}^{\text{e}} \) obtained from references (10), (20), and (24) vary from 400 to 10,000 at a local Mach number of 3 and from 700 to 40,000 at a local Mach number of 6. In order to be conservative a value of \( R_\text{k}^{\text{e}} \) of 400 was used in calculating the effect of roughness on the 10-degree cones and a value of 700 for the 12.6-degree and 18-degree cones. The results are as follows:

<table>
<thead>
<tr>
<th>( \theta )</th>
<th>( M_\infty )</th>
<th>( R_\text{k} )</th>
<th>Calculated</th>
<th>Actual</th>
</tr>
</thead>
<tbody>
<tr>
<td>10°</td>
<td>3</td>
<td>400</td>
<td>1,200</td>
<td>43</td>
</tr>
<tr>
<td>10°</td>
<td>5</td>
<td>400</td>
<td>4,000</td>
<td>150</td>
</tr>
<tr>
<td>12.6°</td>
<td>9</td>
<td>700</td>
<td>7,000</td>
<td>140</td>
</tr>
<tr>
<td>12.6°</td>
<td>13.3</td>
<td>700</td>
<td>10,500</td>
<td>140</td>
</tr>
<tr>
<td>18°</td>
<td>10</td>
<td>700</td>
<td>3,500</td>
<td>160</td>
</tr>
<tr>
<td>18°</td>
<td>15</td>
<td>700</td>
<td>4,200</td>
<td>180</td>
</tr>
</tbody>
</table>

The conservative calculations indicate that for all test conditions, the surface roughness was at least an order of magnitude less than that required to trip the boundary layer.

Another area of great importance is the effect of tip blunting on boundary layer transition. In general, it has been found that slight blunting moves the location of transition downstream from its location.
with a sharp tip (references (10), (25), (26), and (27)). This downstream movement has been partially explained by the unit Reynolds number effect. That is, an increase in tip bluntness results in both a decrease in local unit Reynolds number downstream of the tip and an increase in the length of laminar flow. These two quantities vary in such a manner that their product, the local transition Reynolds number, remains approximately the same. In the case of the 10-degree cones, the initial condition of the tip was extremely sharp with maximum tip diameters of less than 0.002 inch. During the relatively short flight times required for the model to travel to the end of the heated section of the range, only slight blunting due to aerodynamic heating could be noticed. Under the most severe heating conditions the nose radius increased to only 0.008 inches before the model completed its flight through the heater. This maximum amount of blunting encountered in the heat transfer tests reduces the local Reynolds number at the point of transition, as calculated by the method of reference (18), by approximately 4 percent from the value based on sharp cone properties. Therefore, the small bluntnesses associated with these low Mach number tests should not affect transition appreciably.

The 12.6-degree and 18-degree cones had considerably more initial blunting than the 10-degree cones. For the 12.6-degree cone at Mach number 9 and the 18-degree cone at both Mach number 10 and 15, no appreciable increase in blunting due to aerodynamic heating occurred until after the Reynolds number was increased sufficiently to move transition onto the body. Therefore, the transition Reynolds number determined in these tests should not be affected by a change in bluntness. For the 12.6-degree cone at a Mach number of 13.3, some slight blunting occurred at Reynolds numbers at which the flow was still laminar. To determine if the bluntness encountered in these tests affected the transition location, the shadowgraphs obtained of the models in flight were examined to see if the transition location changed as the bluntness changed. Since the additional blunting was caused by aerodynamic heating, the models became blunter as they progressed down the range. Although the location of transition could only be measured with an accuracy of ± 5 percent of the body length, no change in transition location with increase in nose blunting could be determined for these tests.

Two other factors, which have an undetermined effect on the location of transition, have been eliminated in these tests. They are free-stream turbulence associated with wind tunnel tests and an undesired surface roughness created by the condensation of air on very cool model walls. The first factor is important in any transition tests while the second is usually only critical in tests at high heat transfer rates such as those described here. The problem of free-stream turbulence was eliminated by conducting the present tests in a ballistics range where the turbulence level is essentially zero. The second condition was eliminated by creating a highly cooled boundary layer by increasing the recovery temperature of the air rather than by cooling the walls of the model and thereby eliminating any possibility of the formation of frost or condensate on the model surface.

Perhaps the most undesirable element in a ballistics range transition program is control of the angle of attack of the model. For the present tests it was desirable to keep the angle of attack as low as possible. The sabots for the models were designed to give as small an initial angle of attack as possible. The aft portion of all of the models was made hollow to shift the center of gravity as far forward as possible and fins were added to the heat transfer models to increase their static stability and damping rate. A further attempt to remove the effect of angle of attack was made for the hypersonic transition tests by computing the zero angle of attack drag coefficient for each launching by equation (10). For the heat transfer tests, only data that were collected at angles of attack of less than 1 degree were considered.
Heat Transfer Test Results

Table I presents the test conditions and local transition Reynolds numbers obtained on the 10-degree cones. Since the location of transition would vary as the model traveled downrange, partially due to the change in angle of attack, it was necessary to obtain an average length of laminar flow for each round from which an average transition Reynolds number could be calculated. In each shadowgraph obtained, the boundary layer could be seen on both the most windward ray and the most leeward ray of the model in the plane of the photograph. For test conditions that required an elevated range temperature, only the transition readings obtained from the eight data stations in the heated section of the range were considered.

For the other test conditions at normal range operating temperature, approximately 80°F, transition was obtained from all twenty-seven data stations. Figure 12 shows the variation of both the transition Reynolds number and the horizontal and vertical components of the angle of attack with distance downrange for round number 5043. Since the required range temperature was approximately 80°F, it was possible to use data from the entire length of the range for this launching. From this figure it can be seen that the transition Reynolds number tends to vary in a sinusoidal fashion much like the yawing motion of the model with both variations having approximately the same frequency. The average transition Reynolds number for this round number was obtained at complex angles of attack of less than 1 degree.

Figure 13 shows the effect of the ratio of wall temperature to recovery temperature of the air on the local transition Reynolds number. Included in this figure is wind tunnel data obtained on the same configuration at a Mach number of approximately 3. Even though there are effects of angle of attack reflected in the scatter of the Mach 3 data obtained during the free-flight ballistics range tests, an agreement of these data with the data obtained in a wind tunnel (reference (7)) is obvious. Although the scatter of the present Mach number 3 free-flight data makes it difficult to determine a meaningful curve, there appears to be a destabilizing effect on the boundary layer caused by decreasing the temperature ratio. A comparison of the present Mach number 3 data with the low temperature ratio wind tunnel data makes it even more justifiable to draw the destabilizing curve presented in figure 13. It is apparent that the Mach number 3 free-flight data cannot be associated with the upper wind tunnel curve which shows a stabilizing effect of decreasing the temperature rate, and, therefore, some destabilizing effects appear to be present at the conditions of extreme cooling that are not significant for the condition of moderate cooling.

The Mach number 5 data are also presented in figure 13. Due to the much higher recovery temperature obtained at Mach number 5 than at Mach number 3, it was not possible to obtain the same temperature ratios for both series of tests without heating the model during the Mach number 5 tests. The temperature ratio for the Mach number 5 tests varied from approximately 0.20 to 0.09. For temperature ratios of approximately 0.20 to 0.13, a destabilizing effect of decreasing temperature ratio on transition Reynolds number, very similar to that observed in the Mach number 3 test, was obtained. However, the slope of the data as presented in figure 13 is steeper and there is less scatter in the data. At temperature ratios of less than 0.13, the destabilizing effect was not noticed. Instead the boundary layer became extremely stable and the boundary layer remained laminar over the entire surface. A minimum transition Reynolds number was formed for these data based on local flow properties and the length of the cone and represents the minimum value that the transition Reynolds number can assume at the given flow conditions. The minimum transition Reynolds numbers are indicated in figure 13 by an arrow attached to the symbol.
A Reynolds number based on local flow properties and the momentum
thickness at the location of transition was also calculated for both the
Mach number 3 and Mach number 5 tests. These data are presented in
figure 14. These calculations were only made for launchings in which
transition occurred on the model. Again the data show a decrease in $R_{\theta}^{*}$
with a decrease in temperature ratio at Mach number 5. The effect of
temperature ratio on $R_{\theta}^{*}$ cannot definitely be determined from the Mach
number 3 data due to the scatter. However, by eliminating the data point
at a value of $R_{\theta}^{*}$ of approximately 900, the same trend is observed for
the Mach number 3 data as for the Mach number 5 data. The better agreement
in the Mach number 5 data is probably a factor of the angle of attack.
The models used in the Mach number 3 tests, which were conducted before
the Mach number 5 tests, did not have fins. After noticing the large
number of data points that could not be used in the Mach number 3 tests due
to excessive yaw, finned models were used for the Mach number 5 tests.
This reduced the angle of attack considerably and resulted in more
useful data from each launching.

**Hypersonic Transition Test Results**

Table II presents the test conditions and drag data for the
12.6-degree and 18-degree cones. Since the measured drag coefficients
contained the effects of the yawing motion of each launching, a correction
had to be applied to the coefficient to remove this effect. This was
done through equation (10). The variation of the total drag coefficient
with the mean squared angle of attack is presented in figure 15 for the
12.6-degree cones and in figure 16 for the 18-degree cones. In these
figures the data are grouped in terms of Mach number, Reynolds number
based on free-stream properties and body length, and cone angle. For each
cone angle, straight lines were drawn through each group of data at
similar flow conditions. The value of the slope of these lines was then
used for $m$ in equation (10) to calculate the corrected zero angle of
attack drag coefficient, $C_{D0}$. The corrected drag coefficient was then
plotted as a function of length Reynolds number for each Mach number and
is presented in figure 17. Figure 17 also contains some drag data on
12.6-degree cones obtained at NCL in the Pressurized Ballistics Range
during a previous test. The data were obtained at a Mach number of 9 and
at body Reynolds numbers of $2 \times 10^6$ to $7 \times 10^6$ and is reported in
reference (28). From figures 15 and 16 it can be seen that it was not
possible to choose a single slope that would satisfy all groups of data
exactly for either cone angle. Therefore, an average slope was chosen
for each cone angle. Since the majority of the data were obtained at
small angles of attack, $\theta^2 \leq 4$, a small variation in $m$ gave only a very
small change in $C_{D0}$. These small changes in $C_{D0}$ tended only to displace
the curves of $C_{D0}$ versus length Reynolds numbers presented in figure 17
in the vertical direction and not affect the points of intersection of
the laminar and laminar-turbulent curves. The points of intersection
are the critical values for determining the transition Reynolds number
and not the vertical location of the curves.

While the points of intersection of the laminar and laminar-turbulent
curves are well defined for both Mach numbers at each cone angle, the
method of determining the transition Reynolds number from the variation
in drag appears to be better suited to the 12.6-degree cone. This is
due to the fact that the skin friction drag component represents a large
percent of the total drag for the 12.6-degree cone than for the blunter,
less slender 18-degree cone.

As was mentioned earlier, the bow shock lies very close to the body
of the model for the combination of a slender body and a high free-stream
Mach number and the boundary layer is optically distorted. This makes it
extremely difficult to determine the location of boundary layer transition
directly from the range shadowgraphs at these conditions. However, the structure of the near wake can be seen quite clearly. Readings were made of the near wake for all launchings to determine whether the base flow was laminar or turbulent. From these readings it could be determined if transition occurred in the boundary layer or in the wake. If the base flow was laminar, the boundary layer must also have been laminar. If the base flow was turbulent, transition must have occurred on the body. For launchings that were made at flow conditions such that transition occurred very near the base, either in the boundary layer or in the wake, a good deal of scatter was noticed in the wake measurements. That is, photographs of both laminar and turbulent wakes would be obtained from the same launching. As the flow conditions were changed to move transition farther from the base, either forward in the boundary layer or aft in the wake, the scatter in the wake readings was reduced. The scatter was partially caused by transition moving on and off the base of the model as the angle of attack varied during the flight. The launchings for which the wake flow was predominantly laminar are indicated in figure 17 by open symbols and the launchings with predominantly turbulent wakes are indicated by solid symbols. The wake data confirms the transition Reynolds numbers determined from the drag data. As can be seen in figure 17, all data at conditions where the total drag coefficient was decreasing as the length Reynolds number increased was obtained from launchings with predominantly laminar wakes and therefore completely laminar boundary layers. When the length Reynolds number was increased sufficiently to cause a noticeable increase in the total drag coefficient, the wake flow became predominantly turbulent indicating that transition had moved onto the body and that part of the boundary layer was laminar. Figure 18 is a shadowgraph of a cone in free flight showing a laminar base flow, and figure 19 is a similar shadowgraph showing turbulent flow.

Calculations were made of total drag coefficients for the flow conditions of the present tests using the method described in reference (18) and a modification discussed earlier to allow for turbulent flow over a portion of the body. In order to carry out the calculations, a transition Reynolds number was determined from the drag and wake data for each Mach number and cone angle and introduced into the calculations. The results of these calculations are presented in figure 20. The broken lines are the sum of the pressure drag coefficient and the skin friction drag coefficient. The solid lines are total drag coefficients obtained by summing with the pressure and skin friction components a base pressure drag coefficient based on a base pressure of zero. The calculated total drag, as indicated by the solid curves in figure 20, agrees very well with the measured total drag coefficients.

The local flow conditions were calculated at the transition location for each Mach number and cone angle tested. The calculations were made for each test condition at a free-stream Reynolds number that produced transition at the base of the model. The free-stream properties and the calculated local properties are listed below.

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<tr>
<th>θ</th>
<th>RN/Rb</th>
<th>M_e</th>
<th>RL_e</th>
<th>RTR_e</th>
<th>Rp_e</th>
<th>M_p</th>
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<td>.035</td>
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<td>6.4 x 10^6</td>
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<td>824</td>
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<tr>
<td>12.6°</td>
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<td>16°</td>
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</table>

It can be seen that with nearly a 50 percent increase in the Mach number the local transition Reynolds number and the local Reynolds number based on the momentum thickness at the location of transition vary by less than a maximum of 6 percent for either cone angle.
CONCLUSIONS

Free-flight transition Reynolds numbers have been determined for three configurations from Mach number 3 to 15. A sharp 10-degree total-angle cone was tested at Mach numbers of 3 and 5 to determine the effect of ratio of wall to recovery temperature on the boundary layer stability. The Mach number 3 data were obtained at ambient range temperatures of 80°F to 450°F. Within this range a destabilizing effect on the boundary layer was obtained as the temperature ratio was decreased. Some of the scatter that was obtained can be accounted for by an undesirable amount of angle of attack in some of the tests. The Mach number 3 data were compared with wind tunnel data obtained at similar conditions and the agreement was found to be very good. The Mach number 5 data were obtained at ambient range temperatures of 80°F to 600°F and temperature ratios of .20 to .09. At temperature ratios of .20 to .13, the same destabilizing effect of increasing heat transfer rate on boundary layer stability was noticed as in the Mach number 3 tests. However, at temperature ratios of less than .13 the boundary layer became extremely stable and transition could not be obtained on the body.

An analysis of both the surface roughness and the tip blunting due to aerodynamic heating indicates that for the present test conditions their effect on the transition Reynolds number is negligible. However, it is desirable to have as low an angle of attack as possible. At angles of attack of less than 1 degree, a variation in transition Reynolds number with angle of attack was noticed. Both angle of attack and transition Reynolds number varied in a sinusoidal manner when plotted versus distance downrange, and both appear to have approximately the same frequency.

Slightly blunted 12.6-degree and 18-degree total-angle cones were tested at Mach numbers from 9 to 15. The transition Reynolds number was determined from a series of drag measurements. By measuring the total drag coefficient over a range of increasing length Reynolds numbers, it was possible to note the conditions at which transition first began to appear on the aft portion of the model by observing an accompanying increase in drag. This increase in drag was due to a portion of the boundary layer becoming turbulent. Calculations were made of the total drag for the conditions tested and compared with the experimental data. The agreement was very good.

Observations were made of the flow in the near wake for each launching to determine whether the base flow was laminar or turbulent. At all conditions where the total drag was decreasing as the length Reynolds number increased, the flow at the base appeared to be laminar indicating that the boundary layer was also laminar. As soon as the total drag began to increase with increasing length Reynolds number, the base flow became turbulent indicating that transition had occurred in the boundary layer.

The local flow conditions were calculated at the transition point for each combination of Mach number and cone angle tested. For the 12.6-degree cone tests it was observed that when the free-stream Mach number was increased by approximately 50 percent, from 9 to 13.3, the maximum variation in either the local transition Reynolds number or the local Reynolds number based on the momentum thickness at the location of transition was less than 4 percent. Similarly, when the free-stream Mach number was changed from 10 to 15 for the 18-degree cone tests the maximum variation in the local transition Reynolds number or the momentum thickness Reynolds number was less than 6 percent.

ACKNOWLEDGMENTS

The author would like to acknowledge the work performed by Messrs. Leonard Crogan and Stuart Hanlein, who designed the models and sabots used in these tests. Appreciation is expressed to
Miss Amy Chamberlin, who performed many of the numerical calculations used in the reduction and analysis of the data and to Mr. Hensel Brown for coding the computer programs used in the analysis of the data. The author would also like to acknowledge the direct participation of Mr. W. Carson Lyons, Jr., in a portion of the programs discussed in this paper and his helpful suggestions contributed during the others.

REFERENCES


NOMENCLATURE

\begin{itemize}
\item $A, C$: Constants in equations (3) and (6)
\item $C_D$: Drag coefficient
\item $C_{D_0}$: Drag coefficient at zero angle of attack
\item $C_{D_B}$: Base drag coefficient
\item $C_{D_F}$: Friction drag coefficient
\item $C_{D_P}$: Pressure drag coefficient
\item $C_F$: Mean skin friction coefficient
\item $k$: Height of roughness element
\item $m$: Mass of model
\item $M$: Mach number
\item $P$: Static pressure
\item $r$: Recovery factor
\item $R$: Gas constant
\item $R_B$: Model base radius
\item $R_L$: Free-stream Reynolds number based on model length
\item $R_N$: Model nose radius
\item $R_{k_e}$: Local Reynolds number based on $k$
\end{itemize}
Reynolds number based on k and flow conditions within
  boundary layer at height k
Local Reynolds number based on length of laminar boundary layer
  flow
Local Reynolds number based on boundary layer momentum thickness

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<th>Subscripts</th>
<th>Description</th>
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<td>Local flow conditions just outside the boundary layer</td>
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<td>H</td>
<td>Ambient conditions in the heated section of the range</td>
</tr>
<tr>
<td>k</td>
<td>Flow conditions within the boundary layer at height k</td>
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<td>m</td>
<td>Ambient range conditions</td>
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### TABLE I

**TEST CONDITIONS AND LOCAL TRANSITION REYNOLDS NUMBERS FOR 10-DEGREE CONES**

<table>
<thead>
<tr>
<th>Round Number</th>
<th>( V_m ) (ft/sec)</th>
<th>( P_m ) (mm Hg)</th>
<th>( T_e ) (°R)</th>
<th>( R_e )</th>
<th>( T_w/T_x )</th>
<th>( R_{tr} ) ( \times 10^{-6} )</th>
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TABLE II (a)

TEST CONDITIONS AND DRAG DATA FOR 12.6-DEGREE CONES
($R_e/R_g = 0.035$)

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<th>Round Number</th>
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<th>$C_D$</th>
<th>$I^2$ (Inch $^2$)</th>
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TABLE II (b)

TEST CONDITIONS AND DRAG DATA FOR 18-DEGREE CONES
($R_e/R_g = 0.05$)

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<th>Round Number</th>
<th>Base Dia. (in.)</th>
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<th>Range Temperature ($^\circ$F)</th>
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FIG. 1 SKETCH OF MODEL CONFIGURATION USED IN HEAT TRANSFER TESTS.
FIG. 2 TYPICAL MODEL AND SABOT COMBINATION USED HEAT TRANSFER TESTS

FIG. 3 SURFACE ROUGHNESS MEASUREMENTS OVER THE MODEL SURFACE

FIG. 4 MAGNIFIED PHOTOGRAPH OF MODEL TIP
FIG. 5 TYPICAL SPARK SHADOWGRAPH OF HEAT TRANSFER MODEL.

FIG. 6 SKETCH OF MODEL CONFIGURATION USED IN HYPERSONIC TRANSITION TESTS.

FIG. 7 TYPICAL MODEL AND SABOT COMBINATION USED IN HYPERSONIC TRANSITION TESTS.
FIG. 8 RANGE HEATER

FIG. 9 PRESSURIZED BALLISTICS RANGE NO. 3

FIG. 10 HYPERBALLISTICS RANGE NO. 4

FIG. 11 TYPICAL WALL TEMPERATURE DISTRIBUTION ALONG THE LENGTH OF A MODEL
FIG. 12 RELATION OF TRANSITION TO YAWING MOTION OF MODEL

FIG. 13 EFFECT OF TEMPERATURE RATIO ON $R_{\theta_1/e}$ FOR 10° CONES

FIG. 14 EFFECT OF TEMPERATURE RATIO ON $R_{\theta_1/e}$ FOR 10° CONES
FIG. 15 VARIATION OF TOTAL DRAG COEFFICIENT WITH MEAN SQUARED ANGLE OF ATTACK

FIG. 16 VARIATION OF TOTAL DRAG COEFFICIENT WITH MEAN SQUARED ANGLE OF ATTACK FOR 15° CONES

FIG. 17 VARIATION OF DRAG COEFFICIENT WITH LENGTH REYNOLDS NUMBER
FIG. 12 RELATION OF TRANSITION TO YAWING MOTION OF MODEL

○ READINGS FROM VERTICAL PLATES
□ READINGS FROM HORIZONTAL PLATES

FIG. 13 EFFECT OF TEMPERATURE RATIO ON \( R_{tr} \) FOR 10° CONES

FIG. 14 EFFECT OF TEMPERATURE RATIO ON \( R_{th} \) FOR 10° CONES
Fig. 18 Photograph showing laminar base flow

Fig. 19 Photograph showing turbulent base flow

Fig. 20 Comparison of calculated drag with experimental data