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A SURVEY OF TRANSITION RESEARCH AT AEDC

ARNOLD ENGINEERING DEVELOPMENT CENTER
AIR FORCE SYSTEMS COMMAND
ARNOLD AIR FORCE STATION, TENNESSEE 37389

July 1977

Final Report for Period July 1956 – September 1976

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This report presents a survey of experimental research on transition Reynolds numbers conducted in a large number of ground test facilities. Facilities surveyed included primary wind tunnels used for aerodynamic testing at subsonic, transonic, supersonic, and hypersonic conditions. Measurements have been made on cones and planar bodies, flat plates and hollow cylinders. This report traces the work using cones, which has been more extensive. The primary motivation for this research spanning nearly 20 → next page

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20. ABSTRACT (Continued)

years has been to verify the adequacy of the facilities to simulate flight conditions. This necessarily entailed the study of free-stream disturbances in wind tunnels and the role these disturbances play in altering transition Reynolds number which must be considered when scaling Reynolds-number-sensitive data. Results presented include current experimental efforts as recent as September 1976. In addition to the cited references, a bibliography of relevant publications from AEDC has been included.

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PREFACE

The research reported herein was conducted at the Arnold Engineering Development Center (AEDC), Air Force Systems Command (AFSC), under Program Element 65807F. The results were obtained by ARO, Inc., AEDC Division (a Sverdrup Corporation Company), operating contractor for AEDC, AFSC, Arnold Air Force Station, Tennessee, under ARO Project No. P32A-G2A. The authors of this report were Dr. Jack D. Whitfield, Executive Vice President, ARO, Inc., and N. Sam Dougherty, Jr., Research Engineer, Propulsion Wind Tunnel Facility, ARO, Inc. The manuscript (ARO Control No. ARO-PWT-TR-77-24) was submitted for publication on April 6, 1977.

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1.0 INTRODUCTION

The significance of Reynolds number to the scaling of aerodynamic test results between wind tunnels and free flight is well recognized by all those who base engineering design upon experimental data. However, such scaling, in general, cannot always be based simply upon direct ratios of the above quantities without some error being introduced in the assumption of similitude. Viscous effects are cumulative in the boundary-layer growth on a body and may contain interactions with shock waves and separation as significant features of certain transonic and supersonic flows encountered in modern-day aerodynamic testing. Also the location of laminar-to-turbulent transition on the body will have possibly significant influence on all of the latter events in the flow. The transition Reynolds number, $(U_\infty x_T)/\nu_\infty$, is therefore a key parameter in the overall similitude of flows.

An often-employed practice in wind tunnel testing is to fix transition by means of artificial tripping devices. This is done in cases where it is not possible to match flight Reynolds number on small-scale models in wind tunnels, and it is necessary to resort to a compromise condition approximating gross features of the flow (shock waves, separated regions) even though properties of the turbulent boundary layer are not the same on the model as on the full-scale body. A consequence of such practice can be that the turbulent skin friction, C_f, is altered in the turbulent regions, thereby requiring some analytical correction to skin-friction drag data. The experimenter might resort to some well-known flat-plate skin-friction value for his test Reynolds number, compute skin-friction drag for the particular body shape involved, to be subtracted from the total drag and then added back in at the flight Reynolds number for which there is a lower value of flat plate C_f. The accuracy of such a technique of extrapolation depends upon the degree of similitude afforded by the artificial trip and the methods of computation. In supersonic and hypersonic flows it is imperative that conditions for transition location on a body be nearly matched to those that would occur in free flight because of the high rates of heat transfer involved and the controlling influence of transition on forces experienced by the body. Accordingly, transition Reynolds number considerations are always important to the experimentalist in aerodynamics and must be considered in planning experiments.

In cases of free transition, one cannot expect a constancy to exist in transition Reynolds number $[(U_\infty x_T)/\nu_\infty]$ relative to the characteristic length Reynolds number $[(U_\infty \ell_{ref})/\nu_\infty]$. This fact was recognized early at the Arnold Engineering Development Center (AEDC) by Potter and Whitfield (Ref. 1) in the uncovering of a unit Reynolds number effect on transition sensitive data. In subsequent studies, Whitfield and Potter (Ref. 2) proposed that the unit Reynolds number, U_∞/ν_∞ with dimension 1/unit length, was an important parameter in boundary-layer stability. Reshotko (Ref. 3) and Morkovin (Ref. 4) have discussed how there could be possibly many factors combining in complex ways to produce effects of unit Reynolds numbers which have been observed. Precisely how transition Reynolds numbers vary in wind tunnels with Mach number and unit Reynolds number is important to the conduct of aerodynamic testing with free transition. The possibility of vast differences in transition Reynolds number between wind tunnel and free flight is somewhat disturbing. The lack of precise knowledge about the behavior of artificial trip devices in all types of flow environments is also disturbing.

Wind tunnel flows suffer from the fact that there are inherent free-stream disturbances - noise and turbulence - in the test section which are not representative of the environment found in free flight. These disturbances, furthermore, can be peculiar to the particular facility or class of facilities as to amplitude and frequency composition. Among the significant questions remaining to be answered in regard to similitude are what influence do these disturbances have on Reynolds number scaling and does the degree of influence vary from one facility to another. The primary interest in boundary-layer transition at AEDC grew from the need to verify the adequacy of its high-speed facilities to simulate free-flight conditions. The level of disturbances in the wind tunnel flow are related to the degree of degradation in flow quality if the absence of disturbances can be construed as indicative of pure flow. Since disturbances in free-stream flow influence transition, measurements of transition Reynolds number, it was decided, could serve to provide a quantification of flow quality. Should these measurements be further correlated to direct measurements of the disturbances present in wind tunnels, then there might be some basis for use in planning experiments and interpreting test data. Accordingly, transition Reynolds number measurements have been made in the transonic, supersonic, and hypersonic wind tunnels at AEDC. The philosophy employed was much the same as that employed with the "turbulence sphere" in low-speed facilities where changes in sphere drag coefficient as a function of Re_D reflected turbulence level in the wind tunnel.

Morkovin (Ref. 4) has pointed out that correlation experiments on boundary-layer transition should be performed on the simplest of bodies, i.e., flat plates or cones, and that in all cases an adequate documentation of the facility disturbance environment is essential to the understanding of the results. It should also be recognized that the flight environment is not absolutely disturbance-free, atmospheric conditions vary with time and with geographic location, and values of transition Reynolds number for flight through the atmosphere are not known to good accuracy. Consider the flight data cited by Beckwith et al. in Ref. 5 at $M_\infty = 2.0$, for instance, and one finds scatter in the measurements from 3.0×10^6 to 40×10^6 for the length transition Reynolds number given from several experiments. Such scatter in flight data on transition might be attributable in large part to the degree of difficulty in holding test conditions constant and repeatable in flight. It is particularly difficult to hold the body attitude constant and to measure incidence angles with good accuracy with even moderate atmospheric turbulence. Because so many factors can influence transition, if good correlation from one wind tunnel to another and from wind tunnel to flight is to be realized, the best results should be obtained on a single body of simple geometry with standardization and thorough documentation of experimental methods and instrumentation.

Recognizing that tunnel-to-tunnel correlation must be broad in scope lest possible effects peculiar to any one tunnel be allowed to obscure the results, investigators conducted experiments on transition outside AEDC in a number of other facilities. Measurements were correlated by Pate (Ref. 6) and Pate and Schueler (Ref. 7) in some ten different facilities at M_δ from 3.0 to 8.0 on cones and planar bodies at zero incidence. Pate and Schueler were able to show in wind tunnels ranging from 1-ft x 1-ft to 16-ft x 16-ft size at these Mach numbers that the tunnel wall boundary layer had controlling influence on transition Reynolds number. It was thus inferred that the sound field radiated by turbulent boundary layers in wind tunnels dominated the free-disturbance spectra which influenced transition. Tunnel-to-tunnel correlation was later extended by Credle and Carleton (Ref. 8) and by Dougherty and Steinle (Ref. 9) to subsonic and transonic facilities using a sharp, smooth 10-deg included angle (5-deg half-angle) cone which came to be known as the AEDC Transition Cone. This correlation grew to include twenty-three different wind tunnels in all as this cone became a standard calibration body used by AEDC, the National Aeronautics and Space Administration (NASA), the U. S. Navy, and industry. Seven tunnels were included in Western Europe through cooperation with the governments of the United Kingdom, France, and the Netherlands. The Mach number range was

from 0.2 to 5.5 and thus overlapped with the previous investigation by Pate and Schueler. There were microphones flush-mounted on the cone surface for measurement of wind tunnel flow disturbances and accelerometers mounted internally to measure vibrations.

Such a broad experimental study, involving thirty-one different wind tunnels in all between Pate and Schueler and Dougherty, Credle, and Steinle provided the necessary perspective in tunnel-to-tunnel correlation to reveal a flow disturbance environment influence. Reference flight values for tunnel-to-flight correlation are still as yet unavailable but are planned to be acquired on the AEDC Transition Cone at Edwards Air Force Base, California (Ref. 10). The flight data are planned to be acquired at a variety of altitudes over the Mach number range from approximately 0.4 to approximately 2.0.

Another area of research in transition was performed by Potter (Ref. 11) on cones free-launched in a ballistics range at AEDC. These were cones of 10-deg half-angle, both smooth and roughened. Launches were performed at two Mach numbers, $M_\infty = 2.3$ and 5.0. Range density level could be adjusted to give a very large variation in U_∞/ν_∞. The significance of these transition measurements made in the ballistics range is that they were made during flight through still, undisturbed air since the cones flew at speeds greater than even that for sound transmission through the surrounding structure. The striking result of the range experiments was to reveal the existence of a very strong influence of U_∞/ν_∞ on transition Reynolds number, this in the absence of any significant free-stream disturbances.

Much of the research spanning more than 20 years at AEDC has been focused on transition detection methods by means of schlieren, shadowgraph, surface temperature and heat transfer, microphones, hot wire probes, and pitot probes as well as various sublimation and china clay techniques. Much of this work was in support of specific user test activities; but the main thrust of investigation has been toward improved understanding of the facilities. In order to take truly comparative data between different facilities, some experimental effort must be devoted to parametric studies of tip or leading-edge bluntness, surface roughness - distributed and controlled - and to the effects of incidence, heat transfer, and humidity, all of which have influence on transition. The emphasis in this paper, however, will be on the facility comparison results with other factors discussed only from the standpoint of error analysis (degree of influence).

There was probably no case of sufficient documentation of free-stream disturbance environment in the experiments described herein to lend a clear understanding of what mechanisms were active in the transition process. There were, however, a sufficient number of repeated test points in several of the facilities to gain the confidence that the setting of flow conditions in wind tunnels could be sufficiently repeatable that any aspect of the results in virtually any of the facilities could be explored more fully at a later date.

2.0 SLENDER CONE RESULTS IN WIND TUNNELS

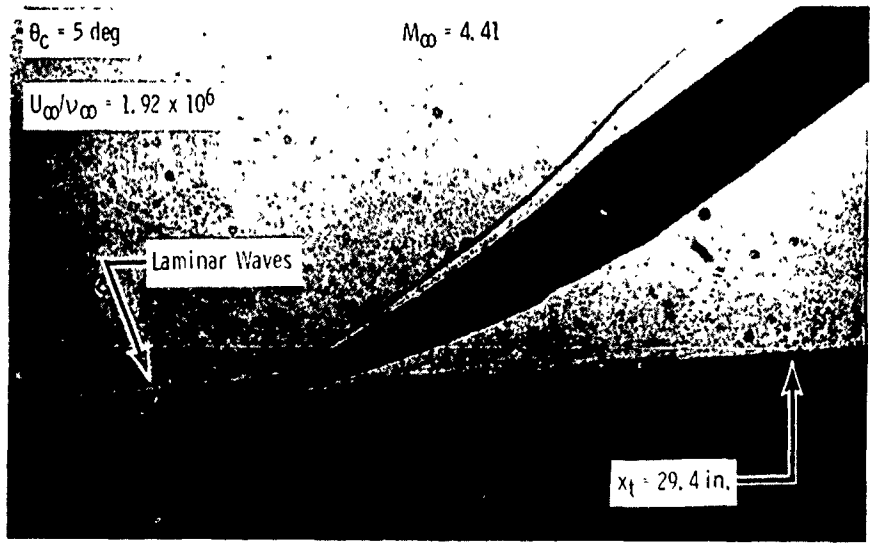
2.1 EXPERIMENTAL METHODS

Shadowgraph views of the boundary-layer flow on 5-deg half-angle cones are shown in Fig. 1 which were taken in the von Kármán Facility Tunnels A and B at AEDC (VKF A and VKF B) at $M_\infty \cong 4.4$ and 8.0. The locations identified x_t and x_T, respectively, will refer to onset and end of transition determined by a traversing pitot probe. The probe is moved axially in contact with the surface in the arrangement as shown in Fig. 2 in AEDC VKF A. Below the cone is a small fixed probe to measure flow incidence angle. A typical set of pitot pressure data from the traversing probe for $M_\infty \cong 4.5$ in VKF A is shown in Fig. 3. End and onset points are defined from the pressure data as indicated in the figure. Each trace represents a particular level of U_∞/ν_∞ at which the traverse was made while holding M_∞ and U_∞/ν_∞ constant. Notice the regular waves that appear in the laminar region in both shadowgraph views well upstream of x_t. More discussion concerning these waves will follow.

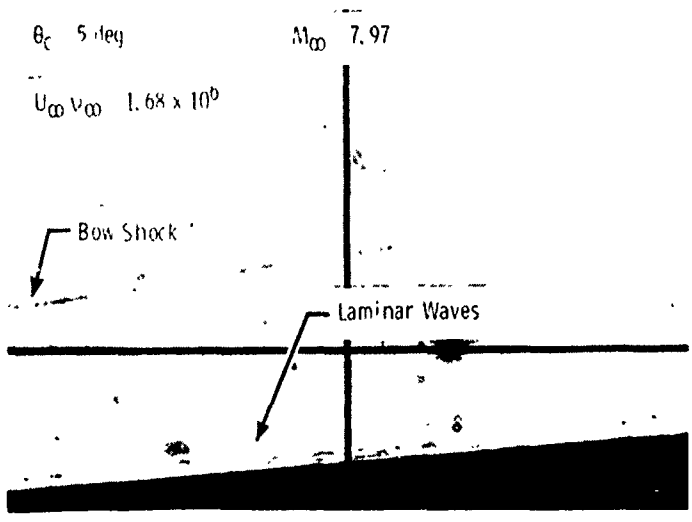
This was the basic experimental procedure employed by Pate and by Dougherty and Credle using the traversing pitot probe: To vary U_∞/ν_∞ while holding M_∞ constant by changing density. Density was adjusted over as much of the tunnel operating envelope as possible through variation in total pressure, p_t, while holding total temperature, T_t, constant. Time was allowed between data points as necessary to allow the cone to reach thermal equilibrium with the airstream and thus adiabatic wall conditions. A pitot traverse was made at each level of U_∞/ν_∞.

A cone of 5-deg half-angle, θ_c, has zero pressure gradient at zero incidence at supersonic Mach numbers above the Mach number for shock attachment which is 1.02. Below $M_\infty = 1.02$, there is a slight gradient which becomes more pronounced near the base region at

lower subsonic Mach numbers (see Fig. 4). The results given in Fig. 4 have a theoretical origin from Wu and Lock (Ref. 12). These results indicate that pressure gradient influence cannot be ignored on the aft portion of a θ_c = 5-deg cone at low subsonic Mach numbers if the results are to be compared to a flat-plate case. (Cone transition Reynolds numbers were always greater than those for the planar case for M_∞ below approximately 8.0.)



a. AEDC VKF A



AEDC VKF B

Figure 1. Shadowgraph views of boundary layer on cones.

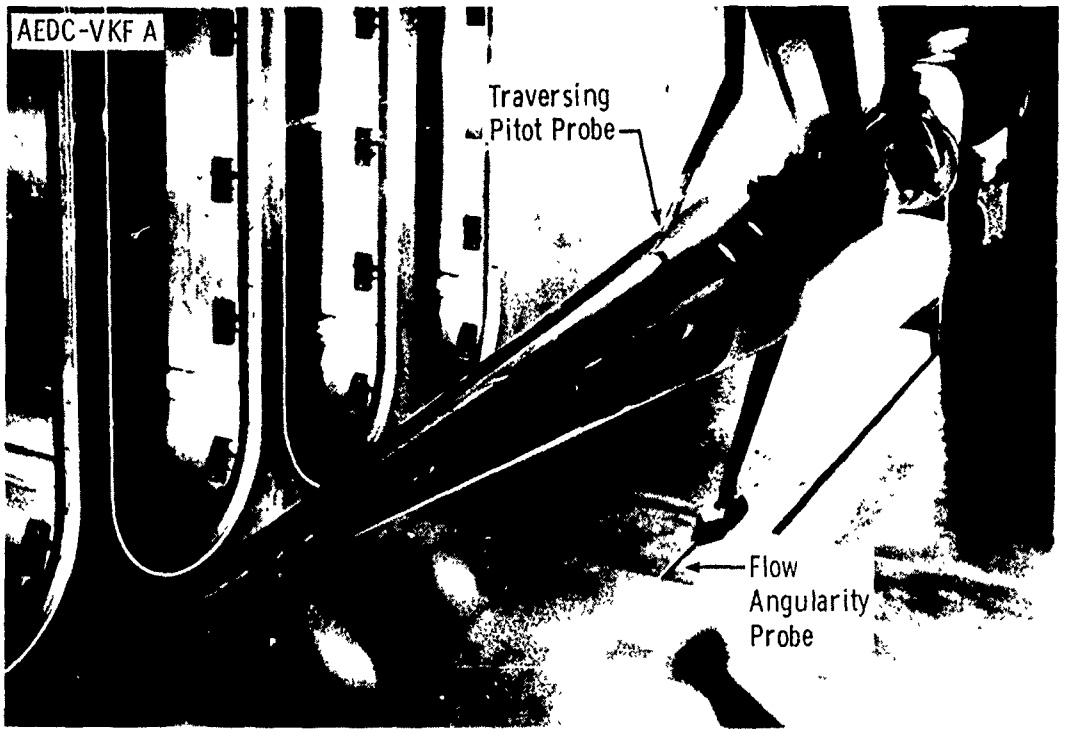


Figure 2. Slender cone used for correlation experiments.

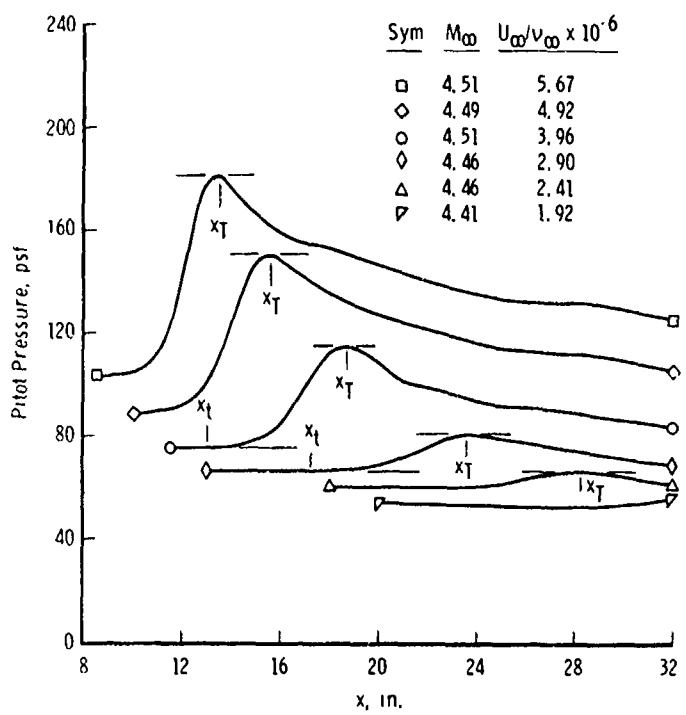


Figure 3. Typical pitot pressure data.

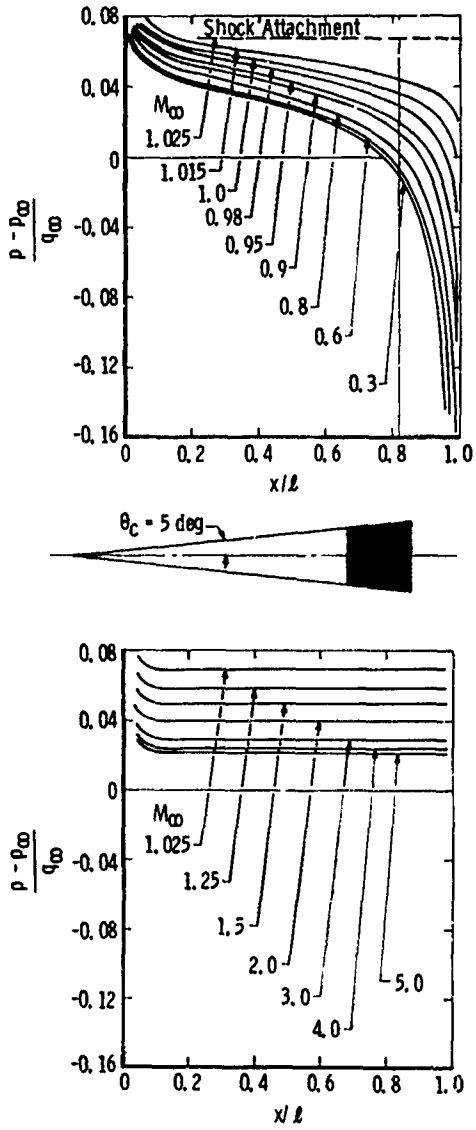


Figure 4. Pressure gradient on the cone at zero incidence.

2.2 TUNNEL-TO-TUNNEL COMPARISONS

Defining a length transition Reynolds number, $Re_T = (U_\infty/\nu_\infty)x_T$, from the horizontal tangency to the peak in the pitot pressure overshoot as was done by Pate and by Dougherty and Credle, Re_T is shown in Fig. 5 as a function of M_∞ for three major wind tunnels at AEDC over a range in M_∞ from 0.3 to 6.0. These data are given for U_∞/ν_∞ held constant at 3.0×10^6, the units on kinematic viscosity being given in ft^2/sec, and the unit length being one foot. The agreement of results by Dougherty in VKF A with those by Pate on a different cone is good. Both cones had nominally 10-μin. root-mean-square (rms) surface finish and tip bluntness less than 0.005 in. equivalent diameter. In addition to the data from VKF A, test results are presented in Fig. 5 from the Propulsion Wind Tunnel Facility Tunnels 16T and 4T (AEDC 16T and 4T). The closed symbols for AEDC 16T and 4T denote results with the test section interior wall surfaces taped over. These two transonic tunnels have perforated test section walls with 60-deg inclined perforations which emit intense discrete edgetones (aerodynamic whistling noise at frequencies in a band from approximately 500 Hz to 5 kHz) as described by Dougherty, Anderson, and Parker (Ref. 13) and by Credle (Ref. 14). The application of tape removed the edgetones from the free-stream disturbance spectra, reducing the overall rms noise amplitude by as much as a factor of three and increasing Re_T at the subsonic Mach numbers by the increments shown.

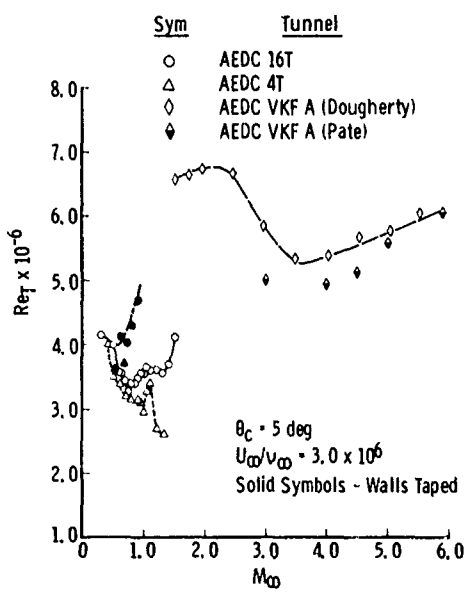


Figure 5. Transition Reynolds numbers on cones in three AEDC wind tunnels.

This was a significant experimental result to show that transition could be influenced by aerodynamic noise sources so low in frequency at these Mach numbers. Relatively poorer flow quality existed in the two PWT transonic facilities, if Re_T is to be the indicator, when compared to the supersonic and hypersonic tunnels of AEDC VKF. The flow quality can be improved by elimination of the edgetones. A perspective was gained through the comparative results obtained on the cone in three slotted-wall transonic tunnels and two tunnels with solid test section walls as shown in Fig. 6. Transonic tunnels were the

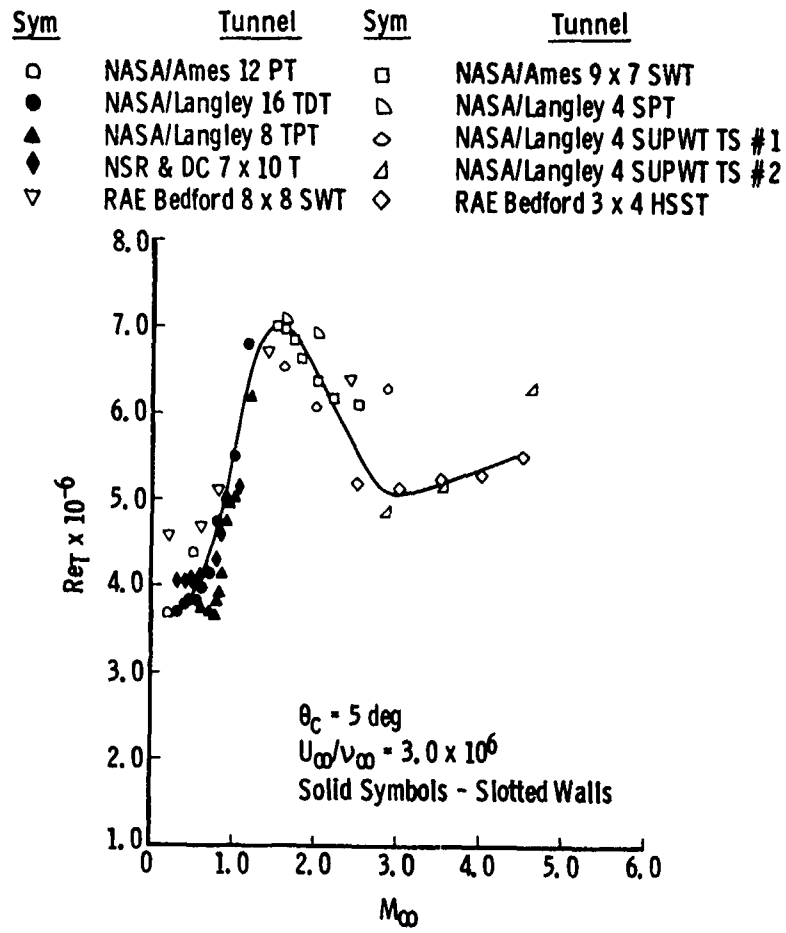


Figure 6. Transition Reynolds number on the AEDC cone in ten other wind tunnels.

NASA/Langley Research Center 16-ft Transonic Dynamics Tunnel and 8-ft Transonic Pressure Tunnel (NASA/Langley 16 TDT and 8 TPT, respectively), and the Naval Ship Research and Development Center

7- x 10-ft Transonic Tunnel (NSR and DC 7 x 10T), all of which have coarsely spaced slots in the test section. The two tunnels with solid test section walls are the NASA/Ames Research Center 12-ft Pressure Tunnel (NASA/Ames 12 PT) and the Royal Aircraft Establishment 8-ft Supersonic Wind Tunnel at Bedford, England (RAE Bedford 8 SWT). Results are shown in Fig. 6 from six other supersonic tunnels which are in close agreement with those from AEDC VKF A (which has a 40-in. x 40-in. test section). These tunnels are the NASA/Ames 9- x 7-ft Supersonic Wind Tunnel (NASA/Ames 9 x 7 SWT), the NASA/Langley 4-ft Supersonic Pressure Tunnel (NASA/Langley 4 SPT), the two test sections of the 4-ft Supersonic Unitary Plan Wind Tunnel at Langley (NASA/Langley 4 SUPWT TS #1 and #2), and the Royal Aircraft Establishment 3- x 4-ft High Speed Supersonic Tunnel (RAE Bedford 3 x 4 HSST).

It should be noted in Figs. 5 and 6 that tunnel size does not appear to have any significance in these results, this being true for M_∞ from 0.2 to approximately 2.5, which is the upper operating limit of the NASA/Ames 9 x 7 SWT. (The four tunnels which can operate above $M_\infty = 2.5$ in this group are all about the same size and thus would not reveal a size variation.) Pate and Schueler (Ref. 7), in fact, showed that there was significant influence of tunnel size in their empirical correlation for $M_\infty \geq 3.0$ in tunnels ranging in size from 1 x 1 ft to 16 x 16 ft. This size parameter apparently was important in the radiation law governing aerodynamic noise intensity radiated to the model from the turbulent boundary layer on the tunnel walls.

Other perforated-wall tunnels gave transition results similar to those in AEDC 4T and 16T as shown in Fig. 7. Data are shown in Fig. 7 acquired in the ONERA 6- x 6-ft S-2 Wind Tunnel at Modane, France, which has 60-deg inclined holes (ONERA 6 x 6 S2MA), and from the Calspan 8-ft Transonic Wind Tunnel (Calspan 8 TWT) and ARA, Ltd., 9- x 8-ft Transonic Tunnel at Bedford (ARA 9 x 8), both of which have normal holes. Results from one other tunnel, the NASA/Ames 11-ft Transonic Wind Tunnel (NASA/Ames 11 TWT), are included here also because this tunnel has finely spaced slots with corrugated baffles that emit an intense organ-pipe whistling disturbance (see Ref. 9). A walls-taped experiment in the NASA/AMES 11 TWT similar to that in AEDC 4T and 16T of covering the slots to eliminate the organ-pipe whistling likewise gave Re_T values at subsonic Mach numbers significantly larger than without the walls taped.

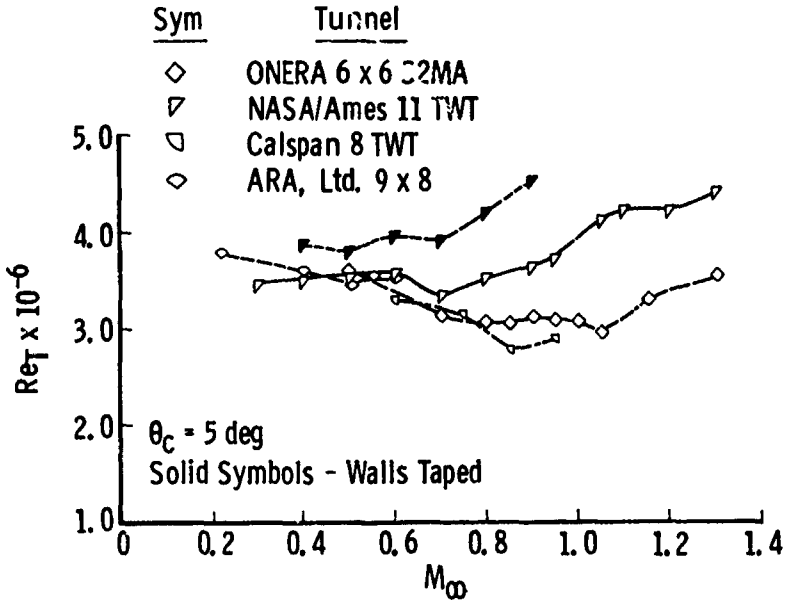
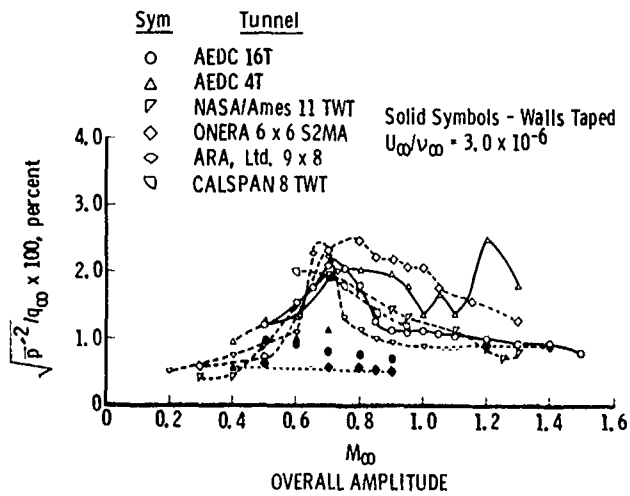


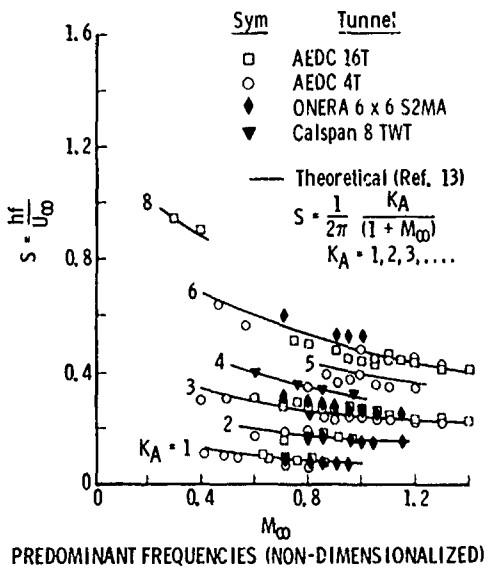
Figure 7. Transition Reynolds numbers on the AEDC cone in other perforated wall tunnels.

Clearly these perforated-wall Re_T values as a family stand apart from those in Fig. 6 as being lower. The reason is apparently that the perforated-wall tunnels have higher test section levels of acoustic disturbance to which the cone boundary layer responds with earlier transition to turbulence. The noise levels were indeed higher in the perforated-wall tunnels than in the slotted-wall tunnels as measured by the cone microphones. Presented in Fig. 8a are overall rms levels of pressure fluctuations detected by the microphones in a bandwidth from approximately 10 Hz to 30 kHz for selected representative tunnels. The pressure fluctuations, $\sqrt{\bar{p}'2}$, have been normalized by q_∞ and given in percent. Strouhal numbers based upon hole size are also shown in Fig. 8b, clearly revealing the holes to be the source of the noise. The advantage of using perforated walls in a transonic tunnel is a reduction in wall interference distortion as explained by Goethert (Ref. 15). The penalty has been that there are excessively high aerodynamic noise disturbances introduced to the free stream; however, as shown by Dougherty, Anderson, and Parker (Ref. 13), the noise can be suppressed by a suitable modification to the holes without compromising the favorable wall interference characteristics. The merit of such a modification to suppress the edgetones would be to bring transition Reynolds number trends in all transonic tunnels more closely in line as is evident in the walls-taped data in Figs. 5 and 7

compared with Fig. 6. Then the size of corrections for effective Reynolds number shift between transonic tunnels might be significantly reduced. As shown in Fig. 8a for AEDC 16T as an example, edgetone suppression with walls taped effectively eliminated an acoustic resonance problem in that tunnel near $M_\infty = 0.71$ and the reduction in noise level was even more dramatic in the ONERA 6 x 6 S2MA.



a. Overall amplitudes



b. Predominant frequencies (nondimensionalized)

Figure 8. Pressure fluctuation measurements in six perforated-wall tunnels.

The slotted-wall transonic tunnels are not without noise problems; but the levels measured by the cone microphones were lower than those in perforated-wall transonic tunnels as shown in Fig. 9. Data presented in Fig. 9 have been selected as typical from the slotted-wall facilities. Results are given from the NASA/Langley 16 TDT, 8 TPT, the NSR & DC 7 x 10 T, and NLR 6.55- x 5.28-ft High Speed Tunnel at Amsterdam (NLR 6.55 x 5.28 HST). Each of these tunnels has an acoustic resonance near $M_\infty = 0.8$; however, inspection of noise spectra in most slotted tunnels revealed the predominant frequencies to be extremely low. In the case of the NASA/Langley 16 TDT, for example, the predominant frequency was 10 Hz. (Here the test medium was Freon® instead of air.) It was concluded the the frequency components coming into resonance in these slotted-wall tunnels were so low (< approximately 200 Hz) that the cone boundary layer was insensitive to them and their influence on transition was nil. This conclusion was based upon the apparent lack of response in Re_T to levels of overall noise amplitude increase by as much as a factor of three in these tunnels. Also presented in Fig. 9 are levels of $\sqrt{\bar{p}^2}/q_\infty$ (in percent) measured in three of the supersonic tunnels - the NASA/Langley 4 SPT, NASA/Ames 9 x 7 SWT, and RAE Bedford 3 x 4 HSST - to show the much lower levels of aerodynamic noise found in the supersonic tunnels, particularly near $M_\infty = 1.5$. These levels represent extrapolated values of noise that would exist under a laminar boundary layer based on readings made at lower levels of U_∞/ν_∞ at each Mach number. Two steps were taken in attempting to obtain noise levels representative of the free-stream background in these tunnels. The first was low-pass filtering of the data to remove components near 48 kHz where microphone diaphragm resonance at low free-stream ambient pressure was excited probably by unstable frequency components (laminar waves) leading to transition. The second was strict avoidance of reporting levels measured under transitional or turbulent boundary-layer conditions, which raised the microphone output because of local turbulent pressure fluctuations. More discussion will follow concerning this first step in regard to the amplification of disturbances by laminar boundary layers and the difficulty of making these measurements using surface-mounted microphones.

An example of the extrapolating technique used on the cone microphones is given in Fig. 10. Data from the two microphones are given at $M_\infty = 2.0$ in the NASA/Langley 4 SPT as a function of U_∞/ν_∞ (Fig. 10a). Transition locations on the cone at these conditions measured using the pitot probe are shown in Fig. 10b. Movement of the transition zone first over the aft microphone at x = 26 in. and then the

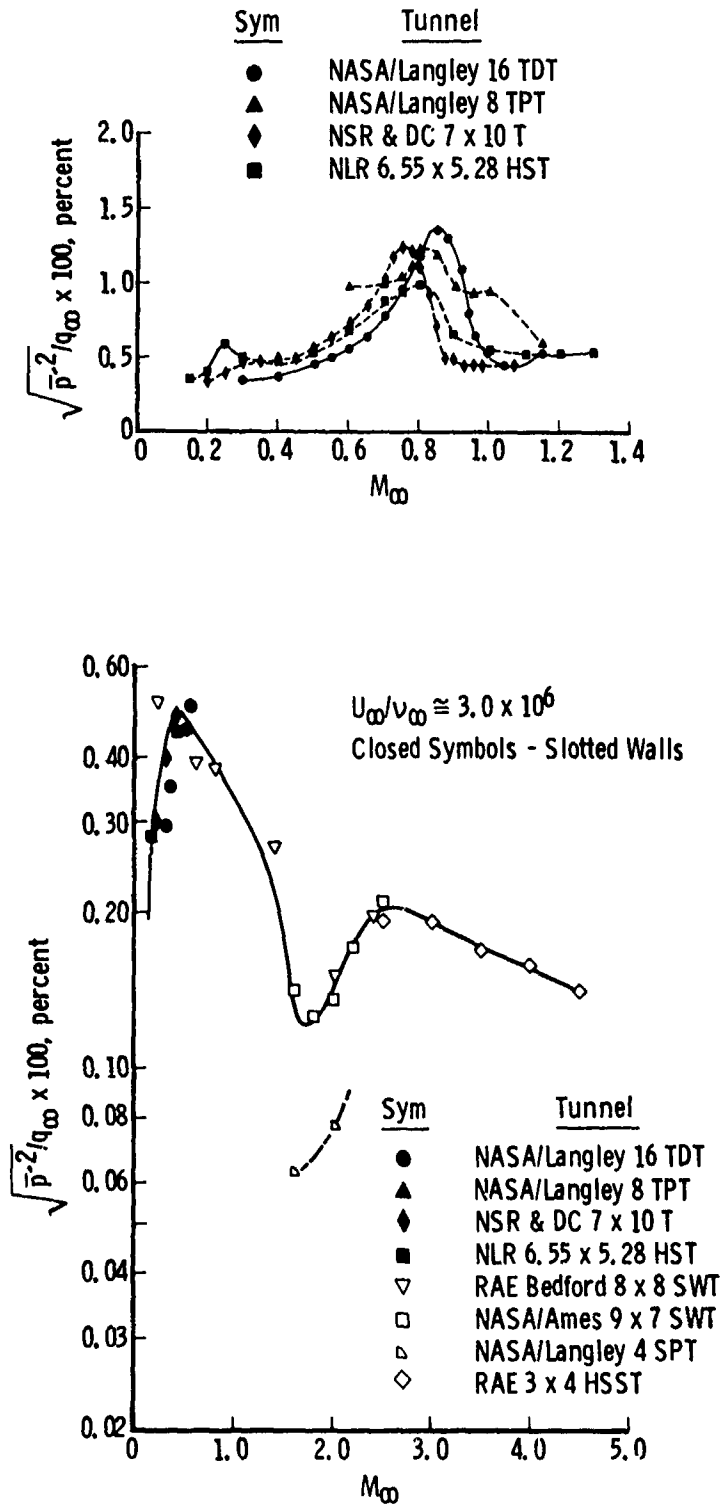
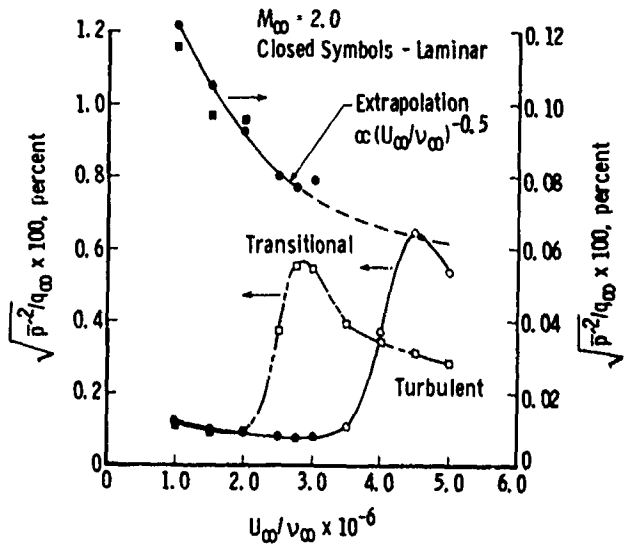
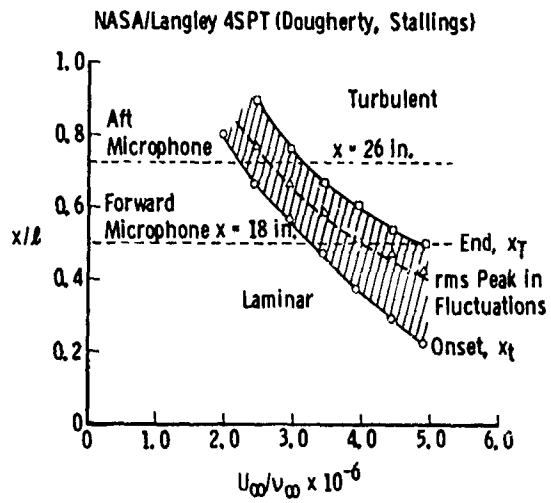


Figure 9. Pressure fluctuation measurement in eight transonic and supersonic tunnels.

forward microphone at x = 18 in. was sensed by the increases in microphone output. The trend of the laminar microphone data is that $\sqrt{\overline{p'^2}}/q_\infty$ decreases (while Re_T increases) with $(U_\infty/\nu_\infty)^{-0.5}$. There is a definite peak in each microphone output near the middle of the transition zone. This was followed by turbulent levels much higher than the laminar trend (open symbols). The extrapolated trend was based solely upon the laminar data (closed symbols) assuming this data trend would continue at higher U_∞/ν_∞ if transition had not occurred.



a. Measured amplitudes



b. Transition location relative to microphones

Figure 10. Pressure fluctuation measurements at M_∞ = 2.0 for varied Reynolds number.

Transition data acquired in supersonic and hypersonic tunnels indicated the existence of a trend that $Re_T \propto (U_\infty/\nu_\infty)^m$, where m has some variation with M_∞ in a given tunnel and some variation from tunnel to tunnel. In order to illustrate this trend, results from the AEDC VKF A and the NASA/Ames 9 x 7 SWT are presented in Fig. 11 as a function of M_∞ at constant levels of U_∞/ν_∞. A broad envelope of Re_T versus M_∞ and U_∞/ν_∞ existed in the NASA/Ames 9 x 7 SWT as shown in Fig. 12. In contrast to these results given in Fig. 11 was the fact that no such trends with U_∞/ν_∞ were found in the transonic slotted-wall tunnels as if $m \cong 0$ for $M_\infty \leq 1.3$. There were trends at high subsonic Mach numbers with U_∞/ν_∞ in AEDC 4T and AEDC 16T only; however, these could be explained by variations in edgetone noise amplitude with U_∞/ν_∞ and that Re_T appeared to be a function of $\sqrt{\bar{p}'^2}/q_\infty$. There was no trend with U_∞/ν_∞ in the NASA/Ames 11 TWT for U_∞/ν_∞ varied over a range from 1.5 to 10^6/ft to 6.0 x 10^6/ft; likewise, there was no trend to be found in $\sqrt{\bar{p}'^2}/q_\infty$ with U_∞/ν_∞, again suggesting that Re_T was controlled by $\sqrt{\bar{p}'^2}/q_\infty$ with $\sqrt{\bar{p}'^2}/q_\infty$ having variation only with M_∞ in this tunnel. The cone was tested in three transonic tunnels that operated at atmospheric total pressure only; thus there was no capability to control density. These tunnels were the ARA, Ltd. 9 x 8, the NASA/Ames 14-ft Transonic Wind Tunnel (NASA/Ames 14 TWT), and the NASA/Langley 16-ft Transonic Tunnel (NASA/Langley 16TT). In these tunnels, U_∞/ν_∞ varied as a function of M_∞ and Re_T appeared to vary as a function of $\sqrt{\bar{p}'^2}/q_\infty$.

The exponent m, where $Re_T \propto (U_\infty/\nu_\infty)^m$, found in supersonic and hypersonic tunnels is shown in Fig. 13. Shown in Fig. 13 are selected data acquired in the NASA/Ames 9 x 7 SWT, AEDC VKF A, and the three NASA/Langley supersonic tunnels. The range in U_∞/ν_∞ available in these tunnels to determine trends was from 1.5 x 10^6/ft to 7.0 x 10^6/ft maximum. The trends shown were based upon the measurements of end-of-transition Reynolds number, Re_T, only as measurements of the point-of-transition onset, x_t, could not be made with the same fidelity as the end-of-transition point, x_T, using the traversing pitot probe. The data included are all from the same 5-deg half-angle cone, the AEDC Transition Cone, with additional data from Pate's cone given for AEDC VKF A. The value of m, as shown in the Appendix (Fig. A-3), can change significantly with only very small changes in bluntness on both cone and planar bodies, the effect of increasing bluntness being generally to increase m. Some additional measurements to derive m which were taken in the AEDC VKF Tunnels D and

E (which both have 12- x 12-in. test sections) are also shown in Fig. 13. These data are given as examples from eight supersonic tunnels. This is still too small a sampling for clarification of a trend. The local Mach number, M_δ, is used in this correlation of results in Fig. 13, the subscript δ being used to denote local Mach number obtained from theoretical cone tables given in Ref. 16.

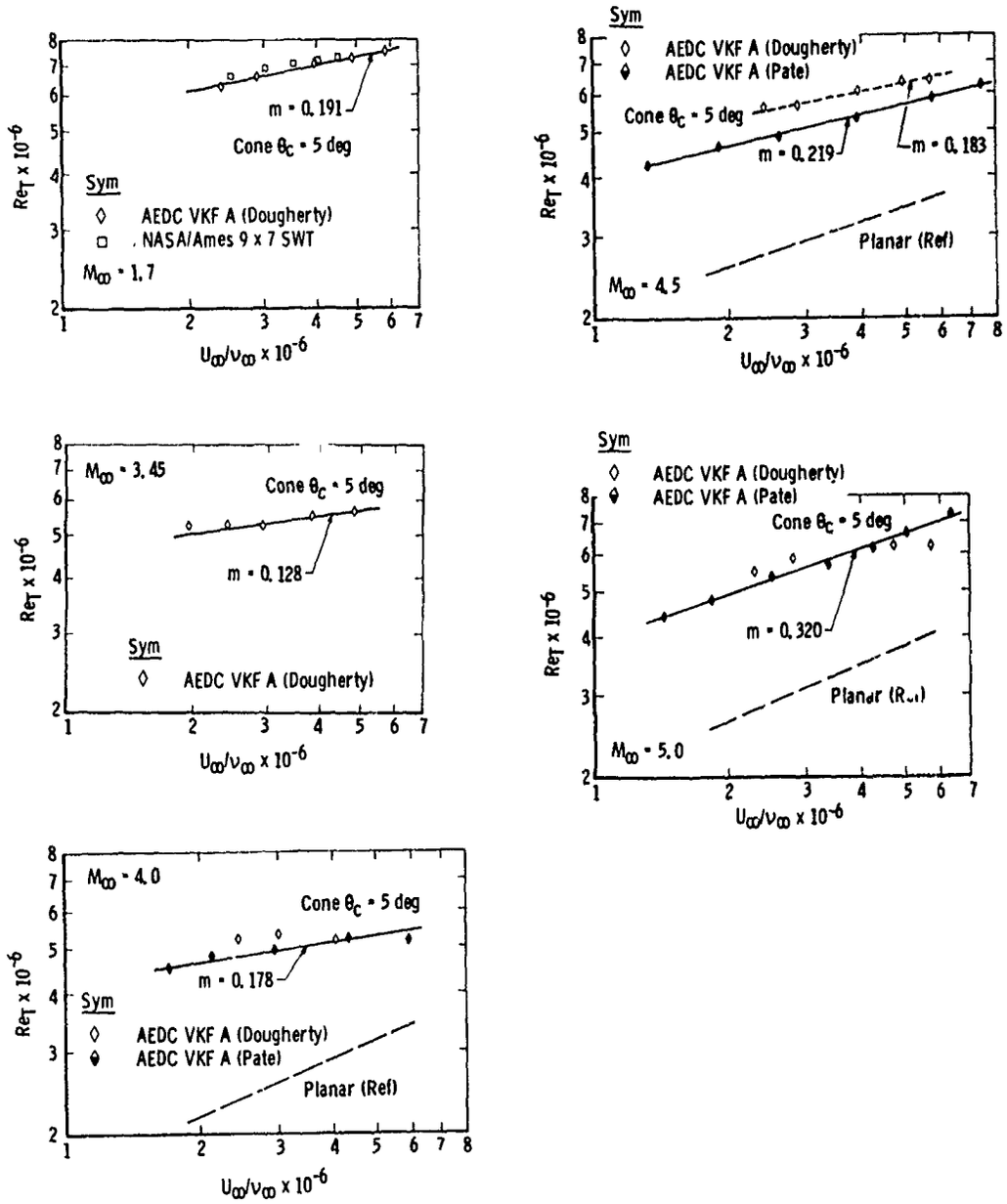


Figure 11. Transition Reynolds number as a function of unit Reynolds number (supersonic tunnel example).

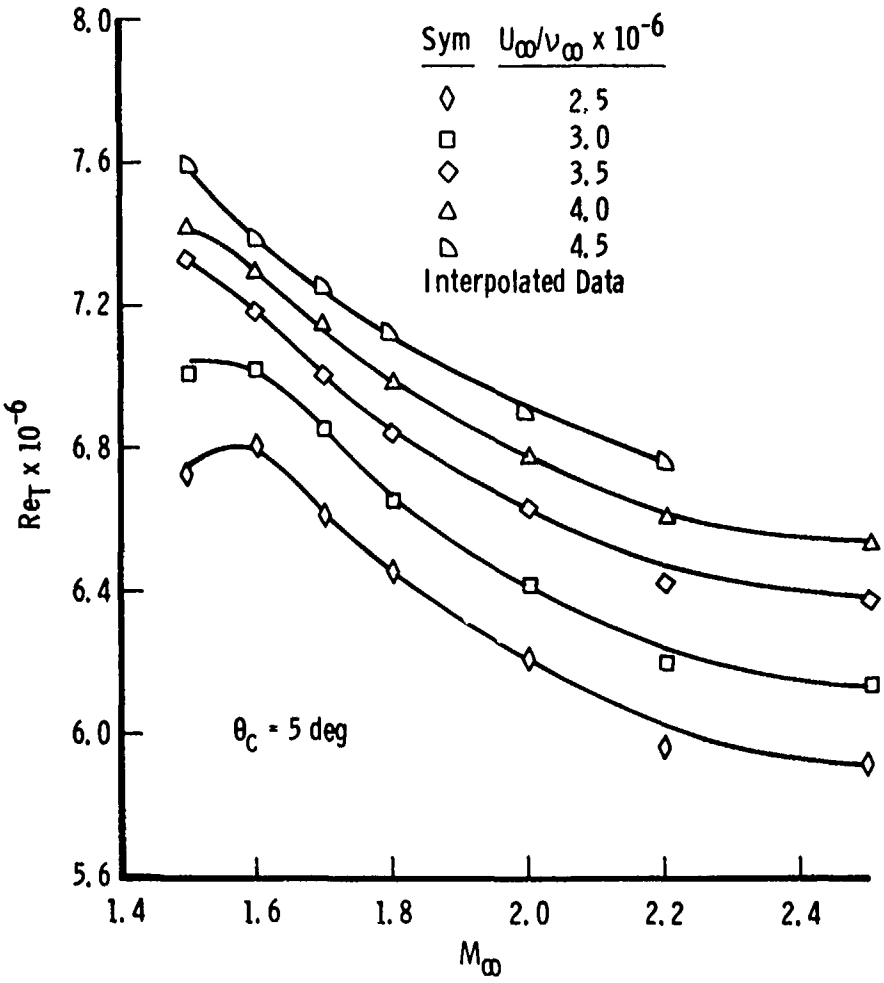


Figure 12. Envelope of cone transition Reynolds numbers in the NASA/Ames 9 x 7 SWT.

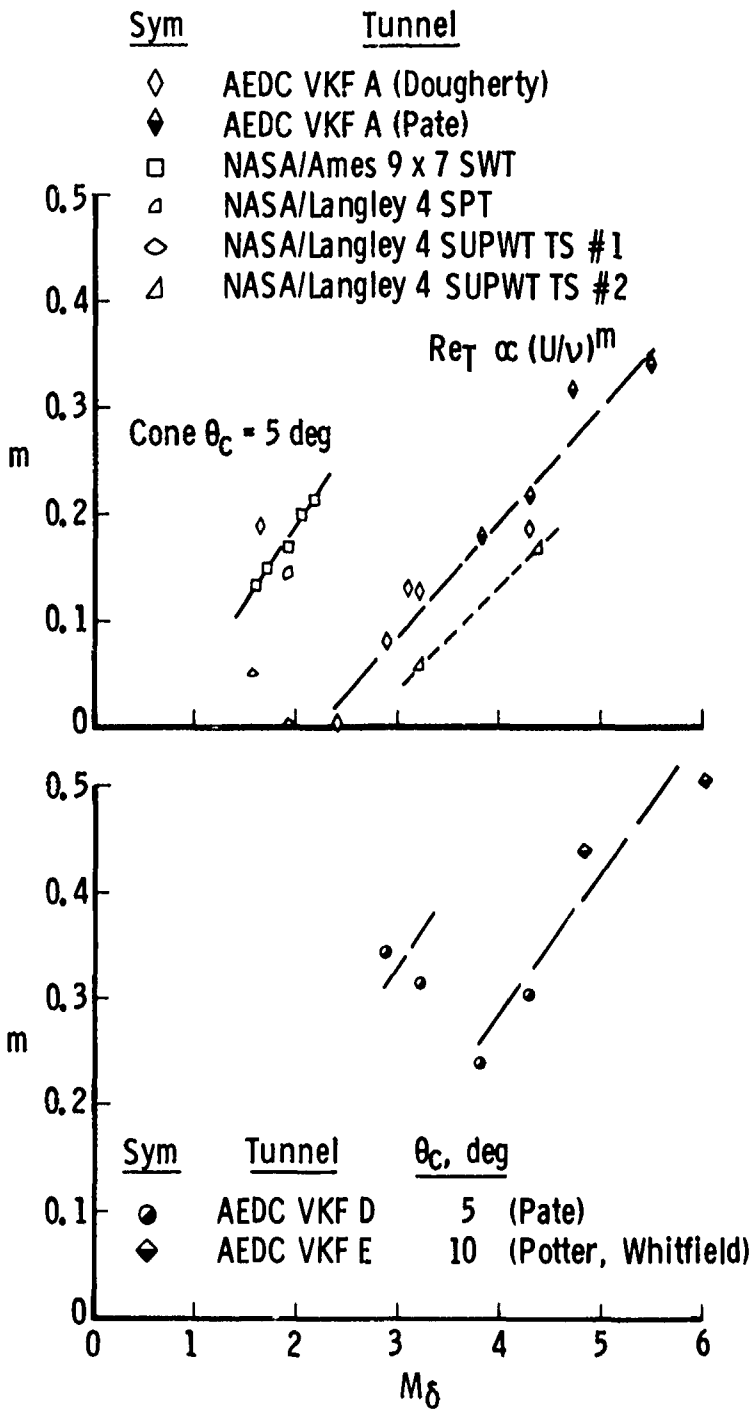


Figure 13. Variation of exponent m on cones in supersonic tunnels.

Less important than absolute values for m is the observation that m tends to increase with M_δ on cones and exhibits some variation from tunnel to tunnel. Much more data on one clearly defined body configuration (i.e., bluntness, surface finish, half angle) is needed to clear the issue of the unit Reynolds number effect in wind tunnels. Differences in apparent trend of m with M_∞ above and below approximately $M_\infty = 2.0$ on a 5-deg half-angle cone may possibly be related to the active mode of instability leading to transition. Modes are discussed in Section 3.0.

In closing this section on cone results in wind tunnels, a qualitative illustration taken from Potter and Whitfield (Ref. 17) shows how transition Reynolds number continues to increase at hypersonic Mach numbers (Fig. 14). These data are given for local flow conditions (δ) for various cone angles and tunnels at constant unit Reynolds number. These were all smooth, sharp cones.

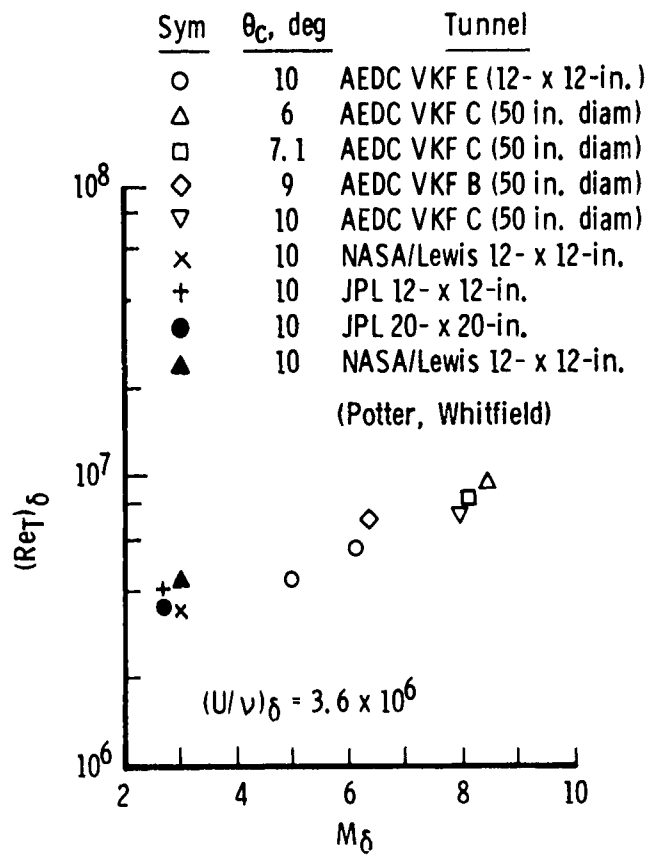


Figure 14. Transition Reynolds numbers on cones in six hypersonic wind tunnels.

3.0 AEROBALLISTICS RANGE RESULTS VERSUS THE WIND TUNNEL

A valuable set of data were those of Potter (Ref. 11) on cones in free flight in the AEDC Aeroballistics Range K of the von Kármán Facility because they showed transition Reynolds number to be a function of $(U_\infty/\nu_\infty)^n$, n being a value for transition in still, undisturbed air. Unfortunately, there was not a direct correspondence in the precise point in the transition process to be documented between wind tunnel and range measurements for it was necessary to rely upon shadowgraph and schlieren photographs for these cones in free flight. It was also necessary to correct the measurements on free-flight cones in the range for small angles of attack because of oscillations in incidence angle during flight. This could be done accurately in the plane of the photographic film only, such that there remained some error with respect to yaw angle. It is known that different instrumental techniques for transition detection give somewhat different locations for the most definable points in the transition process, where photographic techniques give a point early in the transition process as it is defined by pitot probes. Accordingly, no attempt will be made to reconcile precisely what point with respect to pitot pressure profiles was documented by Potter except to remark that his transition Reynolds number, Re_t, was relatively low compared to the above values given for wind tunnels. Rather than absolute value for transition Reynolds number, the important aspect of comparing tunnel and range measurements was the trend to be seen with U_∞/ν_∞. Virtually all experimental investigations of transition have indicated near constancy for the extent of the transition process; this is a ratio of $U_\infty x/\nu_\infty$ of approximately two for end of transition Reynolds number divided by the beginning of transition Reynolds number. Thus, there can be an expanse of a factor of two in $U_\infty x/\nu_\infty$ data given by various techniques.

Potter's data (Ref. 11) are given in Fig. 15. The cone half-angle, θ_c, is 10 deg. The curve fit through these results gave $Re_t \propto (U_\infty/\nu_\infty)^n$, n being approximately 0.63. It can then be surmised that free transition in the absence of imposed disturbances is a function of U_∞/ν_∞ although the reason that such a function should exist is not known. That transition Reynolds numbers should have different trends in wind tunnels in the presence of free-stream disturbances suggests simply that free-stream disturbances have controlling effects on transition.

Some of the possibly most elucidating measurements of the effects of aerodynamic noise on transition which have been made are the

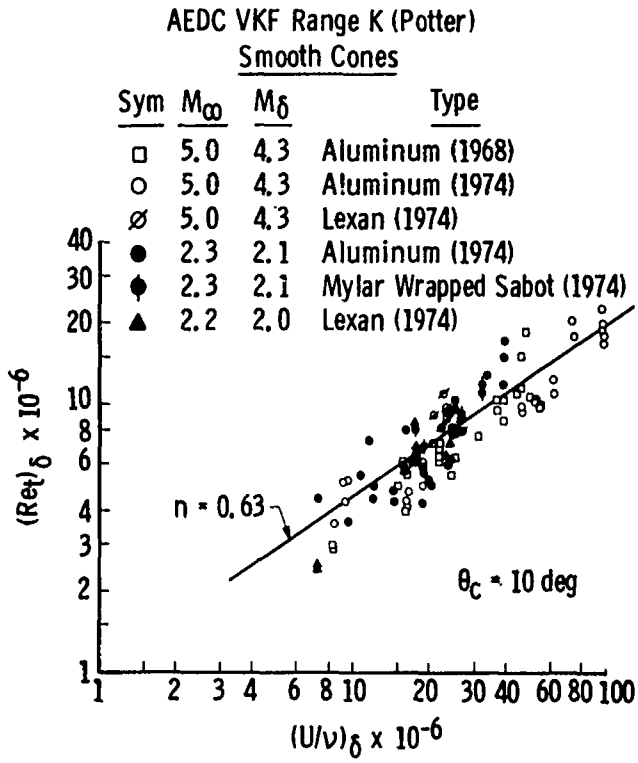


Figure 15. Transition Reynolds numbers on free-flight cones in AEDC Range K.

microscopic measurements of Kendall (Ref. 18). Kendall's data showed excellent agreement with the linear stability theory developed by Mack (Ref. 19) at supersonic and hypersonic speeds. Kendall correlated hot wire measurements made in the transitioning boundary layer with measurements in the free stream in the 20-in. and 21-in. tunnels at the Jet Propulsion Laboratory (JPL) at Mach numbers between 1.65 and 8.5. These measurements concerned transition on a flat plate. Kendall was able to show that the free-stream disturbance field radiated from the tunnel wall turbulent boundary layer was driving the boundary layer at Mach numbers in the range from 3.0 to 5.6. A lower level of correlation was reported by Kendall at Mach numbers in the range from 1.6 to 2.2. Mack, in turn, was able to predict analytically the transition Reynolds numbers measured by Kendall using linear stability theory and measured free-stream disturbance spectra in the wind tunnel. The theory predicts that the first mode for oblique three-dimensional wave propagation (ψ = 60 deg) should be most unstable around M_∞ = 3.0. Then a second mode at ψ = 0 deg with higher unstable frequencies should be active around M_∞ = 5.0 to about M_∞ = 8.0. Amplification level at a particular

$U_\infty x/\nu_\infty$ and predicted transition Reynolds number vary with M_∞ as shown in Fig. 16. The flat-plate trends (theoretical in Fig. 16b) bear strong resemblance to those on the θ_c = 5-deg cone given in Fig. 6 above, although the relative levels are lower for a flat plate than were measured on the cone. Furthermore, Mack (Ref. 19) found the exponent m to be a function of M_∞ for the flat-plate case with the JPL tunnel disturbance spectra used as an input.

The laminar waves observed in AEDC VKF A with a high-resolution shadowgraph system and shown in Fig. 1a are possibly the most amplified waves of the first ψ = 60-deg mode predicted by Mack for flat-plate transition but appearing on a cone also. The high-resolution photographs of laminar waves in AEDC VKF B (Fig. 16) have been reported by Demetriades (Ref. 20) at M_∞ = 8.0. In the VKF B photographs the wavelength was shorter and the wave inclination angle steeper, and as observed by Demetriades possibly corresponded to the second ψ = 0-deg mode predicted by Mack. Regular waveforms have also been observed in the photographs of cone boundary-layer flow in the aeroballistics range (see Ref. 11). That such regular waveforms should be observed experimentally with such regular wavelength intervals in both wind tunnel and range tests bears strong testimony to the theoretical concept of selective wave amplification modes in laminar boundary layers leading to transition. It is remarkable that regular wave patterns should be seen in the wind tunnels as well as in the range, the flow conditions in the range being apparently disturbance-free, those in the wind tunnels being apparently dominated by broadband continuous random-type noise spectra.

Some additional comments about the range Re_t data are that the cones were only 5 in. in length and the extremely high unit Reynolds numbers, $7 \times 10^6 \leq U_\infty/\nu_\infty \leq 90 \times 10^6$, produced very thin boundary layers. At $U_\infty/\nu_\infty = 3.0 \times 10^6$ in the wind tunnels, for instance, the boundary layer at transition was much thicker than on the cones in the range relative to two important parameters, tip bluntness and surface finish. The factor of two scatter in the range data points at any given U_∞/ν_∞ is not excessive, considering the extremely small lengths to be measured on these small cones, that photographic measurements are essentially instantaneous and not time averaged, and that transition movements with respect to incidence angle variations in free flight can be large. Still, it is clear in these range data on cones that Re_t had a very strong variation with U_∞/ν_∞, much stronger than in any of the wind tunnels.

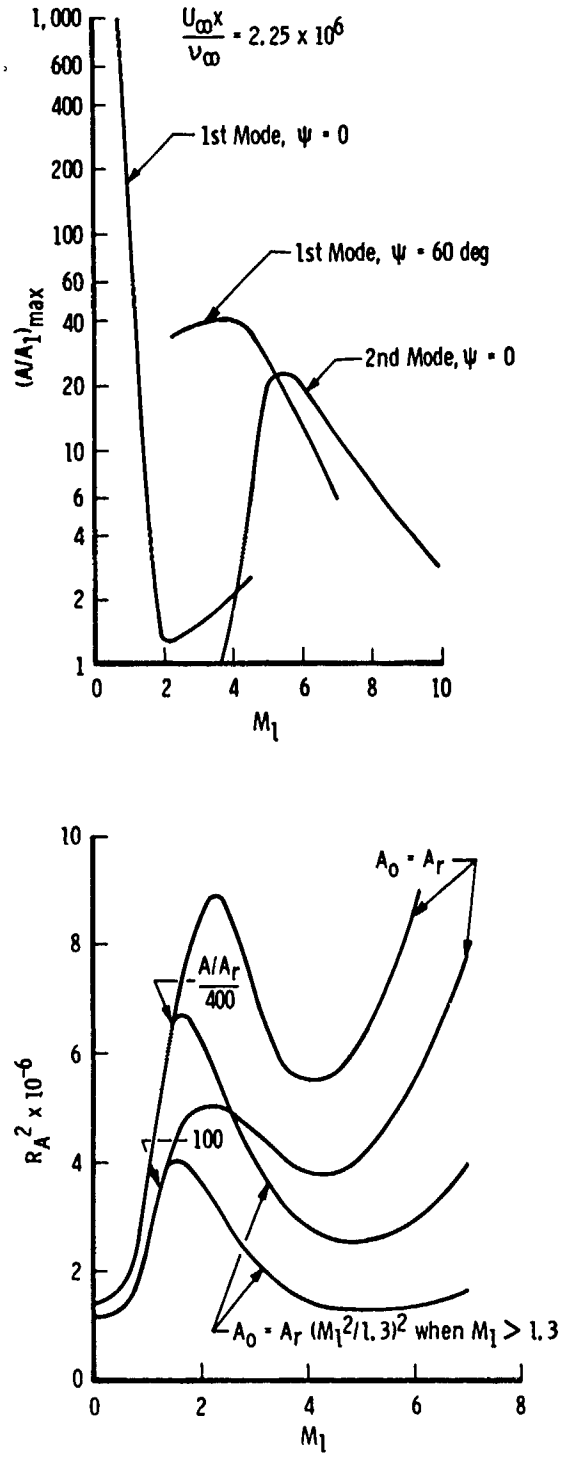


Figure 16. Theoretical effects of Mach number on instability and amplification.

More subtle considerations in the comparison of range and wind tunnel results on these cones include greater tip bluntness relative to boundary-layer thickness, δ, at transition for the free-launched cones than for the cones in the wind tunnels. To what degree that tip bluntness effects may have influenced the range data is an open question. The nose radius of 0.005 in. was selected for the free-launched cones because it was large enough to be reproducible. Tip bluntness, it is suspected only by conjecture, may have been more likely to have had an influence in the range data than surface finish with respect to distributed roughness, the possible exception being minute surface blemishes near the tip. Another consideration is moisture in the free-stream flow about the cones, which introduces losses across the bow shock and some departure in local flow conditions on the cones from the theoretical local values. Yet another consideration is the thermal effect in the boundary layer which can be different between cones at thermal equilibrium in wind tunnels and cones which have been rapidly accelerated to terminal Mach number during launch in the range. This effect would appear in T_w/T_{aw} influence as the influence of heat transfer alters the stability characteristics of a laminar boundary. Still another more subtle consideration is local tip heating at the very high dynamic pressures in the range.

The documentation concerning control over the range experiments was extensive and may be found in Ref. 11. The ability to exercise control in ground test free-flight experiments relative to other methods of obtaining free-flight data makes the aeroballistics range a significant investigative tool.

4.0 CORRELATION OF WIND TUNNEL DATA

The empirical correlation of Pate and Schueler (Refs. 6 and 7) has as its basis that the aerodynamic noise radiated by turbulent boundary layers on wind tunnel walls has controlling influence on transition for $3.0 \leq M_\infty \leq 8.0$. The correlation is given as follows:

$$\left[Re_T\right]_{planar} = \frac{0.0141\,(C_f)^{-2.55}\left[0.56 + 0.44\left(\frac{c_1}{c}\right)\right]}{\sqrt{\frac{\delta^*}{c}}}$$

and

$$\left[(Re_T)\right]_{cone} = \frac{10.5\,(C_f)^{-1.66}\left[0.56 + 0.44\left(\frac{c_1}{c}\right)\right]}{\sqrt{\frac{\delta^*}{c}}}$$

These two trend curves tend to come together for the tunnel wall boundary-layer conditions that occur at $M_\infty \rightarrow 8.0$ (low C_f) and diverge as $M_\infty \rightarrow 3.0$ (higher C_f), implying that $\left[(Re_T)_\delta\right]_{cone} \Big/ \left[Re_T\right]_{planar}$ is a function of M_∞ and then has dependency on U_∞/ν_∞ and tunnel size as well. Pate (Ref. 21) has shown that slight adjustment of constants in the above equation for cones gives an adequate fit to transition data at even higher hypersonic Mach numbers. Dougherty (Ref. 22) has shown that the planar equation gives an adequate fit to data at lower M_∞ down to 2.0 for a tunnel of low free-stream turbulence level (AEDC 16S). This implication of low turbulence level in AEDC 16S stems from the fact that Laufer (Ref. 23) had demonstrated that high stilling chamber turbulence level in a wind tunnel has pronounced effect on transition at $M_\infty \leq 2.5$ but no measurable effect for $M_\infty \geq 2.5$.

Pate's cone/planar correlation for $3.0 \leq M_\infty \leq 8.0$ is given in the Appendix (Fig. A-2). The utility of this correlation with tunnel wall boundary layer C_f and $\delta^{*1/2}$ lies in the fact that these boundary-layer properties can be measured accurately. There has been great difficulty associated with making representative measurements of free-stream background noise in supersonic wind tunnels. This is because of the inherent dependency of the measurements of noise on characteristics of the particular transducer and precisely how and where the measurements are made. The effect of noise on transition is implied herein from the observation that the noise imposed on the test model in these tunnels has primary dependence upon the tunnel wall skin friction, C_f, and then dependence upon boundary-layer thickness and tunnel size. The values used for C_f represent a mean turbulent skin friction near the tunnel test section midlength, recognizing that in some tunnels there can be significant differences between boundary-layer development on the vertical side walls and the horizontal walls, depending upon details of the nozzle design.

Correlation of direct noise measurements to wall boundary-layer properties in wind tunnels has been sketchy and the results sometimes

ambiguous. Notable among the attempts to quantify noise levels in supersonic and hypersonic tunnels has been the recent empirical correlation by Stainback and Rainey (Ref. 24), which has helped to clarify the issue. One of the first to recognize that the aerodynamic noise field radiated from turbulent boundary layers on the walls of high-speed tunnels tended to scale in amplitude with C_f was Laufer (Ref. 25).

Although it can be imagined that the distribution of eddy vorticity in turbulent boundary layers will follow certain definable growth patterns and provide sound field spectra from these vortex sources of a prescribed spectral distribution, the variation in amplitude of the sound field with M_∞ and U_∞/ν_∞ has not been clearly defined. In supersonic flows, the sound waves are propagated across the free stream at the Mach angle as a fluctuating Mach wave pattern from a moving, spatially random, time-stationary pattern of turbulent sources. Statistical correlation tends to reveal consistently a downstream apparent convection velocity of disturbances, U_c, from turbulent boundary layers of $0.6\ U_\infty$ (see Ref. 26, for example) regardless of what portion of the spectrum is examined and regardless of M_∞ being greater or less than 1.0. One might then imagine an apparent height within the boundary layer from which the disturbances have origin, that corresponding to $U/U_\infty = 0.6$. This location is close to the wall for all M_∞, but there is a critical M_∞ below which $0.6\ U_\infty$ has a local Mach number less than unity and above which exceeds unity. Is there a significant difference in sound generation characteristics for vorticity sources in subsonic regions of flow from those in supersonic regions is a question germane to the determination of the sound field far from the source. Furthermore, there is a compressibility effect at higher Mach numbers associated with how the local density at the site of sound-producing vortex eddies varies relative to free-stream density. This density variation can be estimated from a direct ratio of temperatures throughout the boundary layer (density ratio having proportionality with the inverse of temperature ratio) as was done by Lowson (Ref. 27). The Crocco velocity-temperature relationship and observation of whether the wall is insulated or conducting defines the temperature.

The complexity in defining how the boundary-layer sound field varies is soon appreciated in just attempting to define the source strength. Greater complexity still is encountered upon attempting to define propagation characteristics. Interest in this sound field, of course, stems from the inferred correlation by Pate and Schueler that the sound controls transition Reynolds number and is represented

so simply by C_f and $\delta^{*1/2}$. Judgment in attempting to measure this sound field is extremely important because the overall rms amplitude propagated to the far field is not the same as that produced locally at the wall. As a first approximation, a number of investigators, Lilley (Ref. 28), for example, have proposed that $\sqrt{\bar{p}'^2}$ has a prescribed proportionality to the wall shear stress, τ_W, or that $\sqrt{(\bar{p}^2/q)/(\tau_W/q)}$ = constant within a certain range of $U_\infty x/\nu_\infty$. The value of this constant appears to vary with M_∞, and more experimental data are needed to verify what this value is over a broad range of M_∞ and U_∞/ν_∞ and indeed if the approximation is valid in the far-field where free-stream disturbances arrive at the test model. Lilley (Ref. 28) proposed that $\sqrt{\bar{p}'^2}/\tau_W$ at the wall increases from 2.2 for M_∞ close to 0 to 5.6 at M_∞ = 10. Laderman (Ref. 29) has shown $\sqrt{\bar{p}'^2}/\tau_W$ to be about the same as proposed by Lilley at the wall but to be substantially lower in the free stream: e.g., 0.4 at M_∞ = 3.0 increasing about 1.1 for M_∞ = 9.0. Maestrello et al. (Ref. 30) made measurements in AEDC 16S at Mach numbers 1.6 and 2.2 showing $\sqrt{\bar{p}'^2}/\tau_W$ to be about 5.0 on the tunnel wall which is consistent with Lilley's observations.

The correlation of data in supersonic and hypersonic tunnels obtained by Pate and Schueler is shown in Fig. 17. The planar data are shown for reference in the figure. The trends of the planar data may be found in Ref. 7 and other appropriate references and will not be elaborated in this report. The free-stream disturbances in these tunnels have continuous, broad band random-type spectra, free of significant discrete-type disturbances and it can be inferred, since correlation is achieved with properties of the tunnel wall boundary layer alone, that there are no contributions of stilling chamber vorticity or temperature stratification acting on transition to any significant extent in these tunnels.

In other tunnels, such as the perforated wall transonic tunnels, it is necessary to consider direct measurements of disturbance level such as microphone measurements of fluctuating pressure because of the presence of discrete-type disturbances such as edgetones. No simple radiation law can be formulated for subsonic flows containing these disturbances because duct acoustic reverberation characteristics of the test section play a strong role in disturbance amplitudes by selective wave amplication (see Ref. 9).

The disturbances originating from perforations and slots have origin from vortex interactions with the wall geometry to produce the discrete tones and these tones are superimposed on otherwise continuous broadband boundary-layer noise. Thus, q_∞ is the natural choice for a normalizing parameter in characterizing vortex strength in the regular vortex boundary-layer interactions with the wall just as it is for random vortex structures within the layer. At subsonic Mach numbers, when $\sqrt{\overline{p'^2}}/q_\infty$ was much above 0.6 percent, the spectra virtually always revealed predominance of certain discrete or clustered disturbances in narrow bandwidths, and resonance is to be characterized by increases in $\sqrt{\overline{p'^2}}/q_\infty$ above that approximately constant level of 0.6 in subsonic flows.

Sym	M_∞	M_δ	θ_c, deg	Tunnel
○	3.0	2.9	5	AEDC VKF D
△	3.5	3.4		
□	4.0	3.8		
◊	4.55	4.3		
●	3.0	2.9		AEDC VKF A
■	4.0	3.8		
◆	4.5	4.3		
◥	5.0	4.7		
◆	5.9	5.5		

Sym	M_∞	M_δ	θ_c, deg	Tunnel
◆	6.0	5.5	6	AEDC VKF B
◖	8.0	7.0	6	
◖	8.0	6.4	9	
◥	10.0	8.5	6	AEDC VKF C
◥	10.0	7.5	9	AEDC VKF C
△	6.0	5.0	10	AEDC VKF E
◖	8.0	6.2	10	AEDC VKF E
◆	10.2	9.2	3.75	NASA/Langley 31- x 31-in.
×	3.1	3.0	5	NASA/Lewis 12- x 12-in.
+	5.0	4.9	2.5	NASA/Lewis 12- x 12-in.

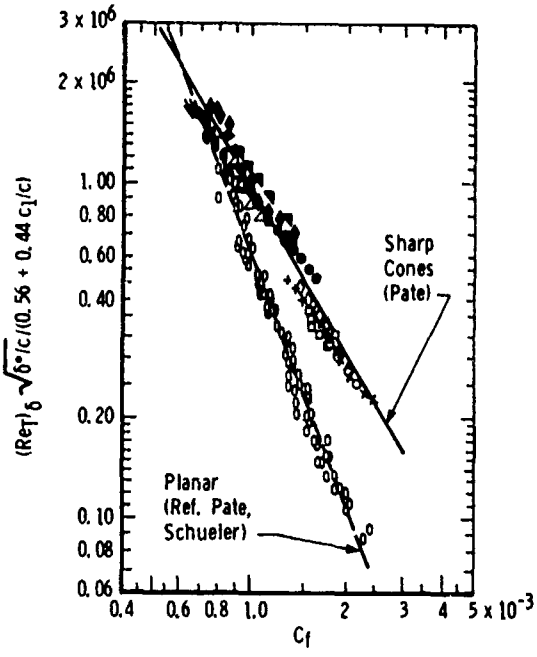


Figure 17. Correlation of transition Reynolds number in supersonic and hypersonic wind tunnels.

The correlation of θ_c = 5-deg cone data obtained by Dougherty is shown in Fig. 18 with

$$Re_T = (3.7 \times 10^6) \left[\sqrt{\bar{p}'^2}/q_\infty \times 100\right]^{-0.25}$$

giving an adequate fit to a large collection of data over the Mach number range from 0.3 to 4.5 and unit Reynolds number range from 1.5×10^6 to 7.0×10^6. The level of $\sqrt{\bar{p}'^2}/q_\infty$ ranges nearly two orders of magnitude, whereas Re_{T_∞} (based upon free stream rather than local, δ, flow properties) for this correlation was due to the fact that the correlation excludes hypersonic conditions where $(Re_T)_\delta/Re_{T_\infty}$ deviates appreciably from unity and that the effort to estimate $(U/\nu)_\delta$ locally at transition at subsonic conditions was not made. This correlation as it is presented is therefore independent of Mach number, unit Reynolds number, and tunnel size. While it is clear that Re_T has a definite dependency upon the disturbance levels in wind tunnels from the data given in Fig. 18, the $\left(\sqrt{\bar{p}'^2}/q_\infty\right)^{-0.25}$ trend shown contains a gross oversimplification of the actual problem. There is a great multiplicity of types of disturbances involved considering spectral composition, phase, and directivity of the disturbances.

The only data which have been excluded from the correlation in Fig. 18 for which both Re_T and $\sqrt{\bar{p}'^2}/q_\infty$ were available are the data in slotted-wall transonic tunnels within the range from $1/\sqrt{2}\, M_{\infty_{RES}}$ to $\sqrt{2}\, M_{\infty_{RES}}$, where $M_{\infty_{RES}}$ is the Mach number at which $\sqrt{\bar{p}'^2}/q_\infty$ peaks. These are the data for which the resonance in $\sqrt{\bar{p}'^2}$ was identified to occur at low frequencies (< 200 Hz) which did not appear to be influencing transition as mentioned earlier. A more proper correlation might have been based upon high-pass filtered measurements of $\sqrt{\bar{p}'^2}$ to remove these disturbances in slotted wall tunnels, but this would have introduced some arbitrariness in the selection of filter cutoff frequency which could not be defended rigorously on a theoretical basis. The usual basis for bandwidth considerations of admissible disturbance frequency has been the neutral stability boundary for the particular boundary layer under study. However,

the large amplitude edgetones in the perforated wall tunnels lie below theoretical neutral stability limits and yet had an influence on Re_T (as verified by the walls-taped experiments). These edgetones may represent a case of forced oscillations in the boundary layer as opposed to natural growth of infinitesimally small disturbances treated by stability theory.

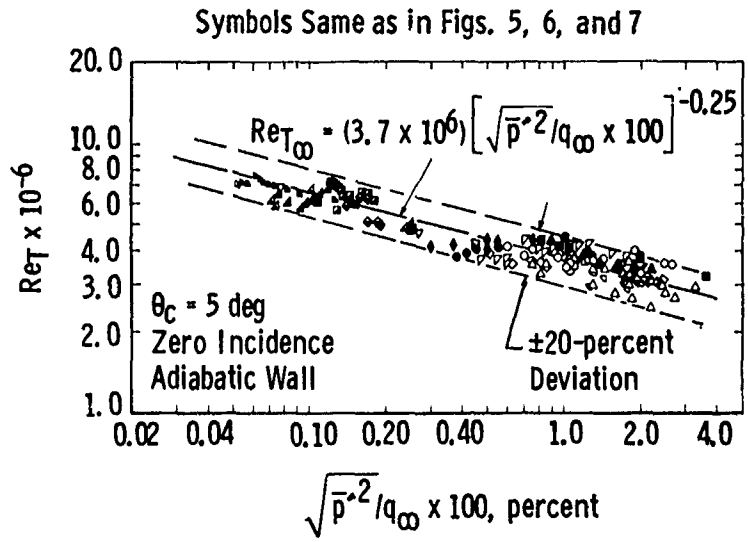


Figure 18. Correlation of cone transition Reynolds numbers with pressure fluctuation level.

The basic objective to show how transition Reynolds numbers vary in wind tunnels and how transition is influenced by the free-stream disturbances has been satisfied for gross engineering purposes of being able to estimate transition locations in wind unnels on a particular body of defined geometry (i.e., sharp slender cone). This has now been done for a broad range of flow conditions and a large number of tunnels, including most of the tunnels at AEDC and many outside AEDC. Still, even for such simple model geometry, little is known about the physics of the phenomena observed over most of the range of variables considered in the experiments. If one wishes to improve upon the accuracy with which transition predictions might be made or to extend what has been learned from the cone experiments to bodies of some other geometry, the great difficulty in doing so can be appreciated by reading once again Morkovin's critical review in Ref. 4.

5.0 CURRENT STATE OF CONFUSION

The experimental data show that the natural disturbances found in conventional wind tunnels have an influence on transition Reynolds numbers, this being true for subsonic, transonic, supersonic, and hypersonic tunnels. There is correlation between the amplitude of disturbances, where measurements are available, to the transition Reynolds number scaling of transition sensitive aerodynamic data. In many tunnels, supersonic and hypersonic in particular, Re_T is not constant with U_∞/ν_∞ and complicates the scaling. Furthermore, there appears to be significant variation in the $U_\infty\nu_\infty$ trend on a given body with M_∞ and from one tunnel to another at the same M_∞. The degree of sensitivity, reflected in $(Re_T)_\delta \propto (U/\nu)_\delta^m$ or $Re_{T_\infty} \propto (U_\infty/\nu_\infty)^m$, is strongly affected by even slight changes in geometry of the body, e.g., tip or leading-edge bluntness.

Available free-flight data on cones through still air in an aeroballistics range indicate the existence of a strong $(U/\nu)_\delta$ effect on $(Re_t)_\delta \propto (U/\nu)_\delta^n$, n being much greater than in wind tunnels at comparable M_∞. Although the variations in Re_T with M_∞ and U_∞/ν_∞ in wind tunnels might be explained by correlation with absolute amplitude of disturbances in wind tunnels and how the disturbances vary with M_∞ and U_∞/ν_∞, a point of confusion arises upon attempting to reconcile the experimental observations with the known stability characteristics (receptivity) of laminar boundary layers. This confusion centers about the question of which has the greater influence on determining transition Reynolds number - the absolute level of free-stream disturbances in the frequency range that can be amplified or the inherent stability characteristics of the boundary layer. The answer undoubtedly lies in there being some combination of both.

Consider again the relative amplification factor for each of the three predominant modes given by linear stability theory $(A/A_1)_{max}$, in Fig. 16 and one sees strong compressibility effect in the level of $(A/A_1)_{max}$ as a function of Mach number, M_1. The first mode is excited to much greater amplification as $M_1 \rightarrow 0$ than it is at $M_1 = 2.0$ by two orders of magnitude or more. This is consistent with the cone data where Re_T is approximately one-half the level near Mach 2.0 as the Mach number goes to 0. The local minimum in Re_T on the cone near $M_\infty = 3.0$ (Figs. 5 and 6) is consistent with the appearance of the first mode at $\psi = 60$ deg with an amplification factor nearly 40 times that at Mach 2.0. The gradual shift from the first $\psi = 60$-deg

mode to the second ψ = 0-deg mode at an amplification factor nearly 20 times greater occurs near Mach 5.0.

These estimated amplification factors given above are for the most-amplified frequency component of the spectrum where actually there is a fairly broad band of frequencies above and below the most amplified extending to the neutral stability limits. There are successively lower amplification rates for the frequencies approaching these limits. Selective filtering of the disturbance data in the attempt to remove frequencies in the bands which would be amplified may not be actually feasible in laminar boundary layers. The pressure flucturation measurements of Fig. 9 are influenced both by the disturbances within the laminar boundary layer and by the free-stream disturbances. The latter disturbances were measured, of course, after they had passed through the laminar boundary layer and thus the resulting measurements have some unknown inherent distortions. The correlation of pressure measurements is based on the assumption that the laminar boundary-layer disturbances and the distortions in free-stream disturbances are similar in all cases. The levels found in transonic tunnels, i.e., 0.4 to 3.0 percent $\sqrt{\bar{p}^2}/q_\infty$, seem to be repeatable with good fidelity under either laminar or turbulent boundary-layer conditions, so long as the transition zone itself is avoided. An accurate measurement in a laminar layer may be impossible, not from lack of dynamic range of the sensor but from the fact that it rivals the level to which small disturbances in boundary layers are amplified naturally.

Answers to many of these questions may be obtained in the next planned major experimental research effort involving AEDC. This is to acquire flight test data on the same θ_c = 5-deg cone (the AEDC Transition Cone) over an envelope in M_∞ from 0.4 to 2.0 using the same instrumentation as used in the wind tunnels. With representative free-flight data in hand on a body for which representative wind tunnel data have been obtained, this correlation presented in Fig. 18 may be validated with flight reference values to give these wind tunnel data an absolute rather than just a relative quantification. Until such data are available, uncertainties in how to correlate Re_T from one body to another, in correlating one detection technique with another, in correlating data from facilities of different size are large enough to rival the ±20 percent uncertainty band in Fig. 18 from the same cone and render the usefulness of the needed reference point nil.

Research to date has indicated that transition Reynolds number does respond to free-stream disturbances in wind tunnels to a sufficient extent as to provide a good indicator of certain aspects of

wind-tunnel flow quality. More research is needed in free-stream disturbance measurements over very broad bandwidths including instruments other than surface-mounted microphones to clear the issue of precisely what it is that the instruments measure and how can the disturbances best be characterized. For compressible flows such measurements must include pressure, velocity, density, and perhaps temperature sensing to sort out the relative coupling occurring between any two of the above variables from these disturbances associated with acoustic wave propagation and those associated with the convection of vorticity. The eventual successful application of stability theory to more complex flows will require a much more complete knowledge of the disturbance environment contained in that flow. The long-range benefits to be realized from this research will be in pay-offs from improved understanding in aerodynamic testing and in the prediction of flight vehicle performance.

The focus of investigation on the end of transition Reynolds number, Re_T, as stated earlier, has merit from the standpoint of greatest fidelity in measurements. There is the additional merit that it represents a point in boundary-layer development where all of the factors influencing transition have completed their action in both linear and nonlinear processes. Certainly more microscopic experiments such as those which have been performed by Kendall are needed for verification of regimes of transitioning flow where compressible linear stability theory, compressible nonlinear stability theory, and descriptions of eventual spreading of turbulent spots to fully developed turbulence may be applicable for high-speed flows.

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APPENDIX
TRANSITION EXPERIMENTS – ERROR ANALYSIS – SECONDARY EFFECTS

Body Geometry

First and foremost in correlation experiments is the necessity for control over body geometry. Since it is impossible to fabricate two bodies of perfectly identical geometry, e.g., surface finish, bluntness, surface flatness, symmetry, best correlation should be obtained on a single body which is maintained in a constant configuration throughout the sequence of tests. Barring no occurrences of damage, for smooth bodies maintenance means careful polishing with fine-grade rouge to within finish tolerances and removal of dirt, grease, and even fingerprints from the surface. Still the surface finish will have some unevenness, and the transition front will inevitably have some nonuniformity. Tests of the effects of roughness by Potter (Ref. 11) and by Potter and Whitfield (Ref. 31) have given results such as those shown in Fig. A-1. These results indicate that there is a critical roughness element height, k_r, above which transition will be influenced and below which there is little influence. Thus, the degree of polishing required for repeatable results is somewhat forgiving and the model can withstand even some particulate impact roughness from contamination in wind tunnel flow without deleterious effects. Polishing to a surface finish of 5 to 10 μin. in these cone experiments (determined by a surface profilometer with a diamond-point stylus of 500-μin. radius) was probably sufficient that roughness variations over these limits were not a serious contributor to data scatter.

Albeit true that the disturbance environment in wind tunnels has an influence on transition, the wind tunnel can be used for parametric study of geometric effects (bluntness, conical vs planar, pressure gradient, sweep angle) with controlled surface finish. Changes can even be studied with time as in the case of ablation where bluntness, shape, and surface finish are all changing. Furthermore, useful experiments on artificial trip devices for transition fixing can be made for a constant-disturbance-environment flow, recognizing that there could be interaction between the action of the trip device and the action of the disturbance environment. As pointed out by Pate (Ref. 6), the correlation of transition between flat plates and cones is not even easily defined, but data for $M \geq 3.0$ indicate a variation to exist with M_∞ as shown in Fig. A-2. To be able to predict transition on complex three-dimensional bodies with pressure gradients will be even more difficult. Progress in this area will likely draw heavily from empirical data correlations.

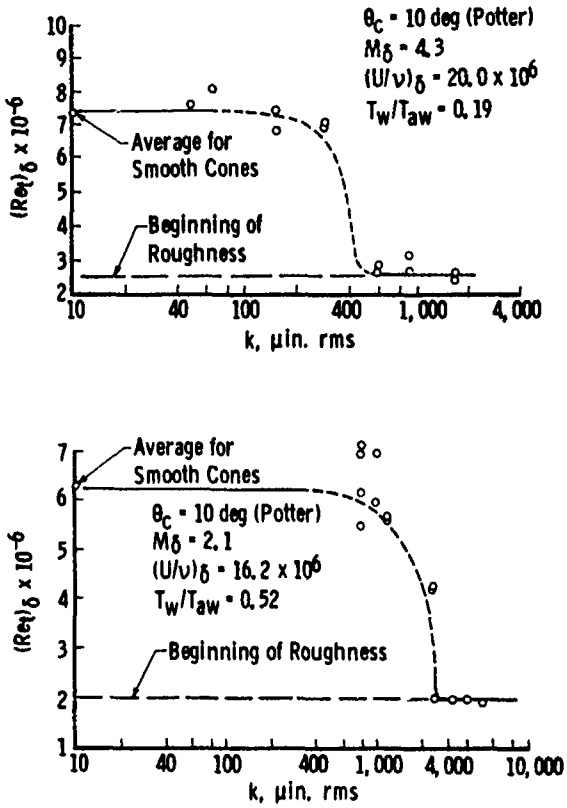


Figure A-1. Effects of controlled roughness on cone transition at high Mach numbers.

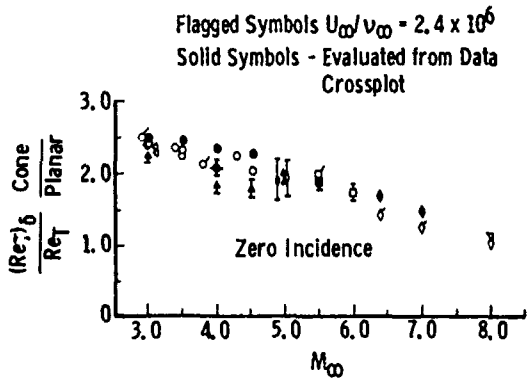


Figure A-2. Correlation of cone/planar transition differences at high Mach number.

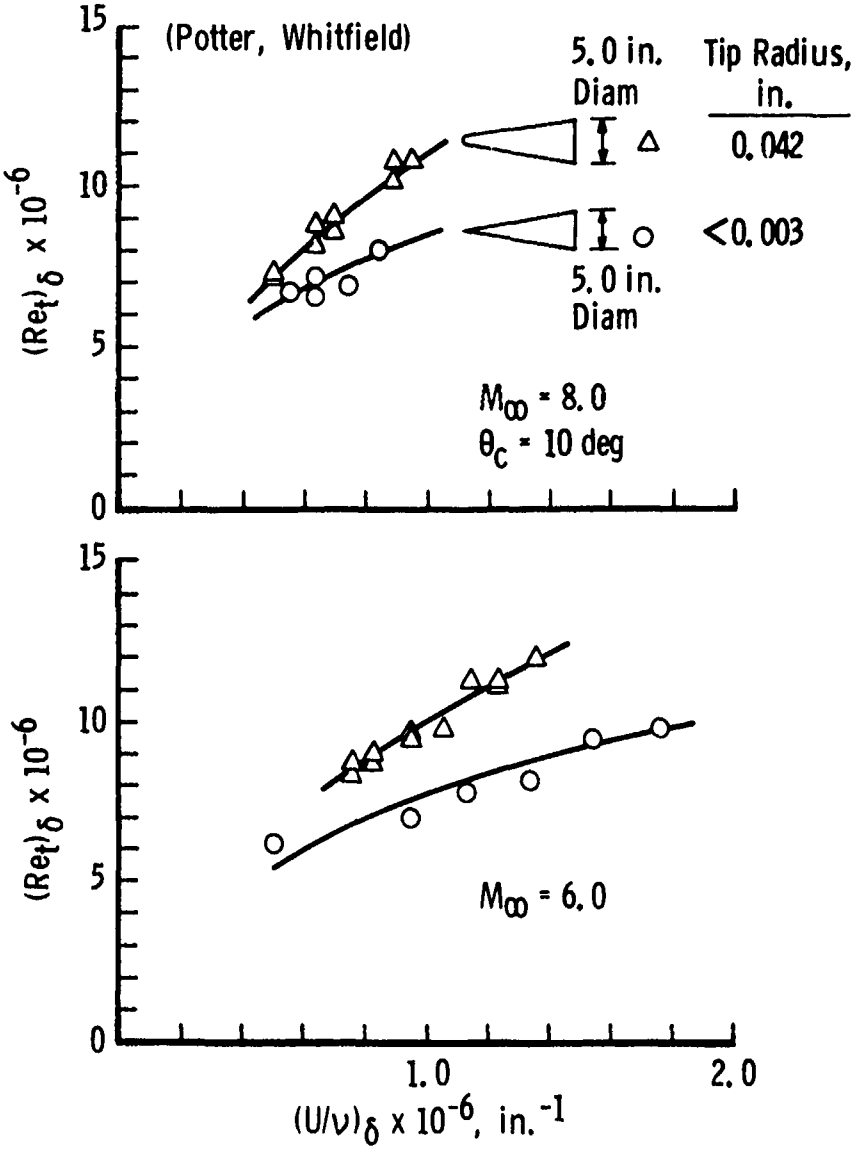
A set of experimental data on leading-edge bluntness by Potter and Whitfield (Ref. 31) is given in Fig. A-3. These data indicate that bluntness has significant influence on boundary-layer growth and transition and that bluntness control is a significant factor in obtaining comparative data between facilities or between facilities and free-flight. In the limit, a sharp leading edge is preferred. While the limiting case of zero bluntness is impossible to realize in practice, acquisition of data at several bluntness values permits extrapolation to the limiting sharp case.

Incidence Angle

Data obtained by Dougherty and Steinle on the sharp, smooth, θ_c = 5-deg cone used in these correlation studies (the AEDC Transition Cone) are shown in Fig. A-4 to illustrate that small incidence angle at supersonic speeds has pronounced influence on transition. The degree of influence apparently increases to a maximum at supersonic speeds near $M_\infty = 2.2$. If repeatability in transition results is to be achieved at these high speeds, control over incidence angle to within approximately ±0.2 deg is critical to hold the transition zone within about 15 percent on a slender cone. Variation in sensitivity to incidence with M_∞ may be related to particular modes of instability as well as the cross-flow Reynolds numbers as parameters.

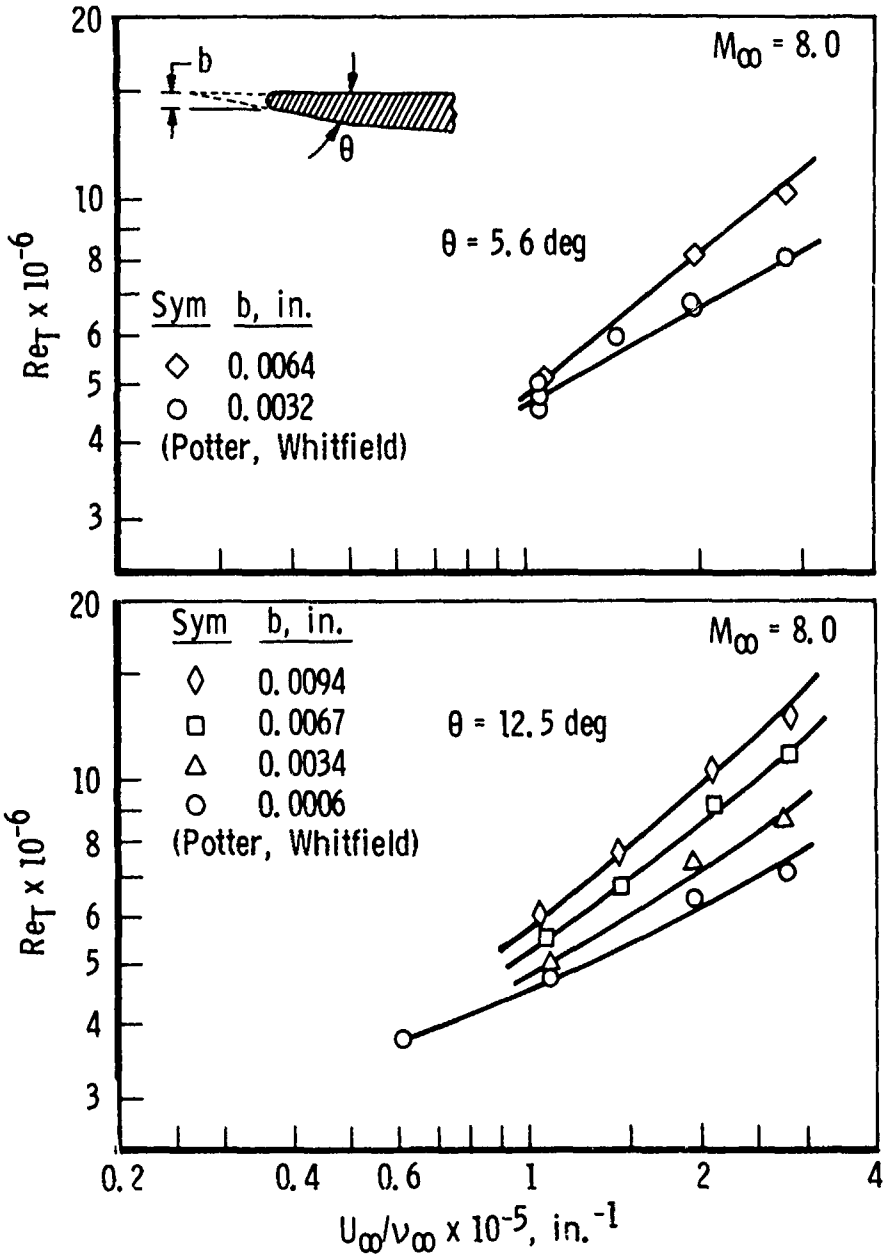
Heat Transfer

The influence of nonequilibrium in thermal conditions can produce significant transient movement in free transition. Heat transfer effects (small changes) are: (a) heating of the body by the free-stream results in earlier transition and (b) cooling of the body will delay transition. Adiabatic wall conditions are desirable for any experiments which would isolate effects of free-stream disturbances on transition. At hypersonic speeds, flow conditions in wind tunnels generally tend toward large departure in the ratio T_w/T_{aw} from unity to values of 0.5 or even less. At those conditions, effects of heat transfer necessarily require some attention because T_w/T_{aw} is much less than unity. For $M_\infty \leq 4$, approximately, $T_w/T_{aw} \rightarrow 1.0$ is realizable provided sufficient time is allowed in the conduct of the experiment to achieve thermal equilibrium.



a. Cones

Figure A-3. Effects of tip and leading-edge bluntness on transition at high Mach numbers.



b. Planar
Figure A-3. Concluded.

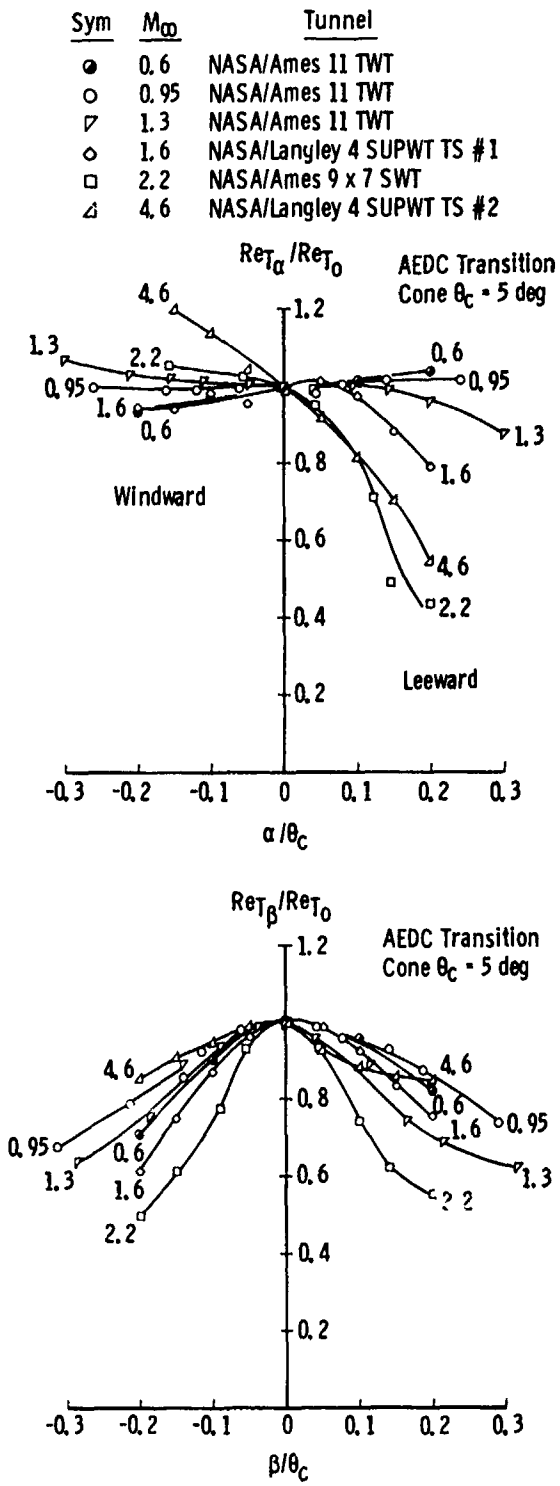


Figure A-4. Effects of incidence angle on cone transition.

Heat-transfer effects can be significant at subsonic Mach numbers as was well illustrated by one simple test in AEDC 4T by Dougherty and Credle at $M_\infty = 0.6$. An excursion in total temperature, T_t, of approximately 30 min produced the shift in transition Reynolds number shown in Fig. A-5 of 50 percent. Heat transfer is a critical parameter in transition correlation with potentially large influence if not controlled.

Humidity

The specific humidity of water vapor in a wind tunnel airstream is difficult to measure, much less to isolate as an influence variable on transition. The first problem with humidity arises from alteration of Mach number and Reynolds number locally about the body because of losses in total pressure recovery across the bow shock when condensation is present. Condensation in normal stagnation bow shocks (blunt body) begins to become a problem at approximately $M_\infty \geq 1.75$. Very little experimental data are available for quantification of isolated effects of humidity and condensation on transition.

General Observations

Even with the very best of control over secondary influence parameters to minimize their effects on data to be correlated, it must be remembered that transition is a dynamic phenomenon which is inherently nonstationary, both in temporal and spatial context. Transition fronts have natural restlessness if viewed instantaneously, and some time as well as spatial averaging can be useful for improving statistical certainty in defining a transition location. How the averaging that is best done varies with what aspect of the transition process is being sensed by the instrument used for transition detection (i.e., pitot pressure, velocity or pressure fluctuations, surface heat flux, or density gradients). A correlation of transition data within ±20 percent probably approaches as good an accuracy as should be expected.

The subject of transition detection is a whole treatise in itself as each method tells something different about the transition process, the sum total of them adding together to characterize the whole of events taking place. Of all of them there is not one that is wholly suitable for all situations. The best idea, therefore, is to pick one method that gives some point that can be defined the same way all of the time and to stick with it throughout. This is an idea to which the authors subscribe. Choice of detection means should be guided by the specific objectives of the experiments.

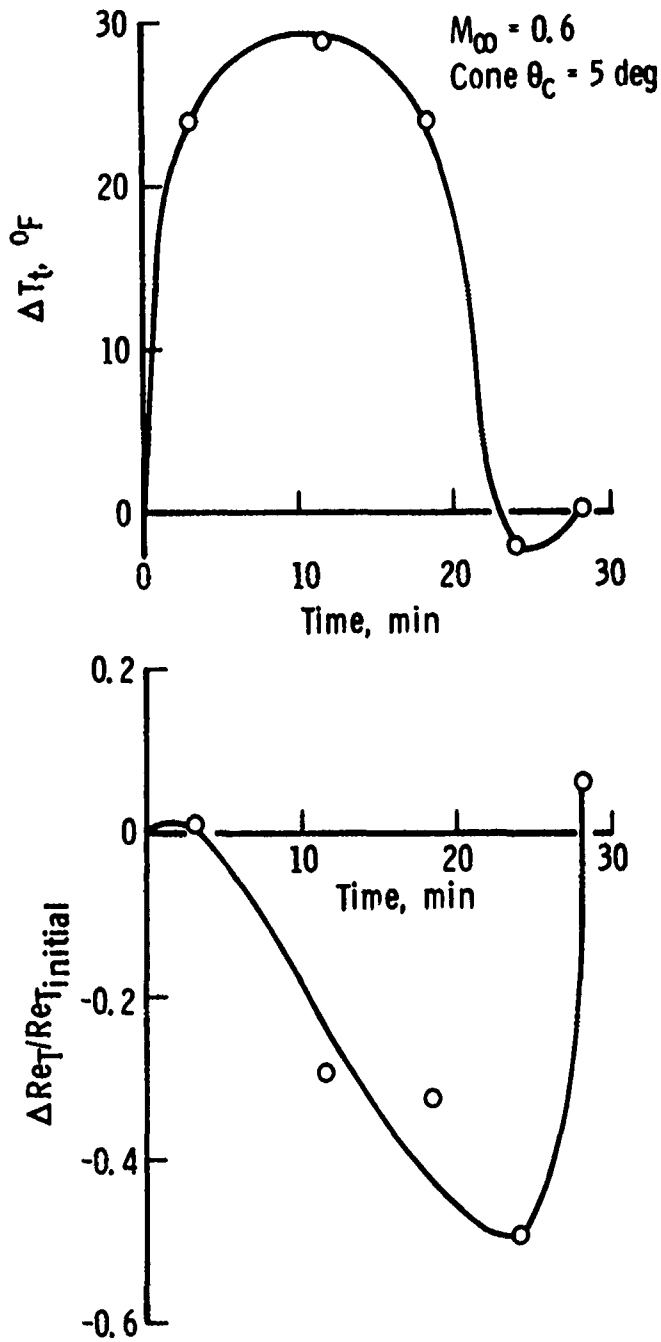


Figure A-5. An example of heat-transfer effect on transition.

There still exists the need for both macroscopic and microscopic experimentation in transition. Detailed microscopic experiments have provided the needed experimental basis for verification of theories which will eventually describe the transition phenomenon. The meticulous detail required to perform such experiments precludes testing at many test conditions because of the time and expense involved in making such measurements. Macroscopic experiments featuring less detail about the transition process but revealing the behavior of transition over a broad range of test conditions are also useful, relying upon microscopic experiments at selected conditions for discovery of the active mechanisms of transition. The two types of experiments are therefore to be viewed as being complementary. The primary contribution of the AEDC experiments has been macroscopic but fairly comprehensive in the scope of test conditions and eligible variables considered.

NOMENCLATURE

A_1	Initial (or reference) amplitude of an arbitrary disturbance
$\left(\frac{A}{A_1}\right)_{max}$	Maximum amplification factor for A_1 at a specified $U_\infty x/\nu_\infty$
A_o	Amplitude of a disturbance at the neutral stability point
A_r	Arbitrary constant reference amplitude
b	Leading-edge radius of a planar body, in.
C_f	Turbulent skin-friction coefficient, τ_w/q_∞
c	Tunnel test section perimeter, ft
c_1	Reference tunnel test section perimeter of 4 ft
f	Frequency, Hz
h	Characteristic length dimension of a perforation which emits edgetones, ft
K_A	Wave number, an integer, 1, 2, 3, ---
k	Roughness element height, root-mean-square, μin.
ℓ	Cone length, ft
ℓ_{ref}	Arbitrary reference length dimension, ft
M_δ	Local Mach number on a cone
M_∞	Free-stream Mach number [Also M_1, see Mack (Ref. 19)]
$M_{\infty_{res}}$	Free-stream Mach number at which resonance occurs for certain components of the spectrum $\sqrt{\bar{p}'^2}$
m	An exponent, determined empirically
n	An exponent, determined empirically
p	Local static pressure, psfa
p_t	Total (or stagnation) pressure, psfa

$\sqrt{\bar{p}'^2}$	Fluctuating static pressure level measured locally by a microphone flush-mounted on a surface, root-mean-square (time-average, frequency-integrated over a bandwidth from approximately 10 Hz to approximately 30 kHz), psf
p_∞	Free-stream static pressure, psfa
q	Dynamic pressure based upon fluid density at an arbitrary location, $1/2\ \rho\ U_\infty^2$, psf
q_∞	Dynamic pressure of the free-stream flow, $1/2\ \rho_\infty\ U_\infty^2 = \gamma/2\ p_\infty\ M_\infty^2$, psf
R_A^2	A relative length of transition Reynolds number computed from linear stability theory (see Mack, Ref. 19)
Re_T	End-of-transition length Reynolds number based upon free-stream conditions (unless otherwise specified), $(U_\infty x_T)/\nu_\infty$
Re_{T_α}/Re_{T_o}	Ratio of end-of-transition Reynolds number at angle of attack to that at zero incidence
Re_{T_β}/Re_{T_o}	Ratio of end-of-transition Reynolds number at yaw angle to that at zero incidence
$(Re_T)_\delta$	End of transition length Reynolds number based upon cone local conditions, $(U/\nu)_\delta\ x_T$
(Re_t)	A length-of-transition Reynolds number obtained by schlieren or shadowgraph near the beginning of transition, $(U/\nu)_\delta\ x_t$
S	Strouhal number, nondimensionalized frequency for edgetones, hf/U_∞
T_{a_w}	Adiabatic wall recovery temperature, °R
T_t	Total (or stagnation) temperature, °R
T_w	Body surface temperature, °R
T_∞	Free-stream static temperature, °R
U_c	Apparent convection velocity for measured disturbances (different from U_∞), ft/sec
U_δ	Local velocity on a cone, ft/sec
U_∞	Free-stream velocity, ft/sec

$(U/\nu)_\delta$	Local unit Reynolds number on a cone, ft^{-1} (or $in.^{-1}$, if specified)
U_∞/ν_∞	Unit Reynolds number based upon free-stream conditions with dimensions (unless otherwise specified), ft^{-1}
$(U_\infty \ell_{ref})/\nu_\infty$	An arbitrary length Reynolds number based upon free-stream conditions
$U_\infty x/\nu_\infty$	A length Reynolds number for specified x based upon free-stream conditions
w	A subscript to denote local conditions at the wall
x	A length dimension measured along a surface from the body tip or leading edge (stagnation point), in the direction of flow, ft or in.
x_T	End-of-transition distance from tip or leading edge, ft or in.
x_t	Onset or beginning-of-transition distance from tip or leading edge, ft or in.
α	Angle of attack, deg
β	Yaw angle, deg
γ	Ratio of specific heats of fluid medium
δ	Local boundary-layer thickness in ft, or subscript used to denote local flow conditions on a cone
δ^*	Local boundary-layer displacement thickness, ft or in.
θ_c	Cone semivertex angle, deg
μ	Dynamic viscosity, $(lbf\text{-}sec)/ft^2$
μ_δ	Dynamic viscosity evaluated locally on a cone, $(lbf\text{-}sec)/ft^2$
μ_∞	Dynamic viscosity evaluated at free-stream conditions, $(lbf\text{-}sec)/ft^2$
ν	Kinematic viscosity, μ/ρ, ft^2/sec
ν_δ	Kinematic viscosity evaluated locally on a cone, μ_δ/ρ_δ, ft^2/sec
ν_∞	Kinematic viscosity evaluated at free-stream conditions, μ_∞/ρ_∞, ft^2/sec

ρ	Density of the fluid medium at an arbitrary location, lbm/ft^3
ρ_∞	Density of the fluid medium in the free stream, lbm/ft^3
τ_w	Turbulent boundary-layer shear stress at the wall, psf
ψ	Propagation angle for disturbances in a laminar boundary layer leading to transition, deg