Development of a High Reynolds Number Quiet Tunnel for Transition Research

I. E. BECKWITH*
NASA Langley Research Center, Hampton, Va.

High noise levels in conventional supersonic and hypersonic wind tunnels modify or dominate transition on test models. Transition research and predictions for flight conditions then require at least a partial simulation in wind tunnels of the much lower disturbance levels usually present in flight. The minimum operational limit for Reynolds numbers in a "quiet" wind tunnel will depend on unit Reynolds number and are established by transition Reynolds number correlations of free-flight data. High facility noise levels also dominate fluctuating pressure loads under fully turbulent boundary layers. Reliable measurements of these loads are required for panel vibration and flutter research. By analogy with recent data on the effects of stream turbulence and noise on low-speed turbulent shear layers, the basic structure of supersonic turbulent shear layers may be modified by high facility noise levels. Required research in these areas determines the maximum design Reynolds numbers for a quiet tunnel. Experimental data for current techniques to control and reduce noise levels in supersonic and hypersonic wind tunnels by laminarization of nozzle wall boundary layers and by noise radiation shields are presented. These results and possible effects of Taylor-Göertler vortices observed in a nozzle wall boundary layer are used to predict the limits of quiet performance for a proposed 20 in. quiet tunnel.

Nomenclature

f = frequency
G = Göertler stability parameter, $R_\theta(\theta/r)^{1/2}$
L = nozzle length from throat to exit
M = Mach number
psi = lb per square inch
p = pressure
P_t = pitot pressure
R = radius
Re = local Reynolds number
R_x = length Reynolds number based on x distance
R_θ = local momentum thickness Reynolds number
R/ft = unit Reynolds number per foot
rms = root mean square of fluctuating value
r = longitudinal radius of curvature of nozzle wall
T = absolute temperature
u = streamwise velocity in x direction
x = distance from model leading edge or from nozzle throat
y = distance from centerline of nozzle
δ = boundary-layer thickness
θ = momentum thickness
μ = viscosity coefficient
ρ = density

Subscripts

aw = adiabatic wall
c = convection velocity
cr = critical value
e = local inviscid flow
M = model
max = maximum value
r = radiated sound
TBL = turbulent boundary layer
t = transition
V = vacuum
WT = wind tunnel
w = wall or surface
o = settling chamber conditions
∞ = freestream flow at nozzle exit or test section

$(\bar{\ })$ = time mean value
$(\ )'$ = fluctuating value
$(\tilde{\ })$ = rms value

Presented as Paper 74-135 at the 12th Aerospace Sciences Meeting, Washington, D.C.; submitted February 26, 1974; revision received August 14, 1974.

Index categories: Boundary-Layer Stability and Transition; Nozzle and Channel Flow.

* Head, Gas Dynamics Section, High-Speed Aerodynamics Division. Associate Fellow AIAA.

Introduction

INTENSE sound radiated by nozzle wall turbulent boundary layers in conventional wind tunnels dominates transition on simple bodies at small angles of attack for Mach numbers of about 3 or greater[1-10] (see Fig. 1). Since transition phenomena are highly nonlinear, reliable predictions for flight conditions will depend on the reduction of facility stream disturbances to low levels approaching those of flight.[11] Only then will it be possible to study and hopefully control the mechanisms that cause transition on flight vehicles.

The interaction of turbulent flow with solid boundaries gives rise to radiated noise and fluctuating loads, both of which are of great practical importance.[12] The levels and spectra of fluctuating pressures at a surface under attached turbulent boundary layers are dominated at low frequencies by facility disturbances[11,13] over a wide range of Mach numbers. Kistler and Chen[14] had concluded earlier that facility radiated pressure was "an insignificant contribution to the total signal." However, this conclusion was based on pressure spectra measured at the surface of a flat plate underneath laminar and turbulent boundary layers at $M_\infty = 2.2$ with all frequencies below 1 kHz filtered out. Laufer[15] showed later that the radiated pressure spectra contained significant energy at low frequencies corresponding to the large eddy scales in the tunnel wall boundary layer and that the ratio of radiated to wall pressure decreases rapidly for $M \lesssim 3$. Furthermore, if Laufer's data[15] for radiated pressures are readjusted to represent the total radiation from three tunnel walls to the fourth wall, where Kistler and Chen measured the wall pressures,[14] then the resulting comparison (see Fig. 3 of Ref. 15) would show that the rms radiated pressure is nearly one-third of the rms wall pressure at $M_\infty = 5$. Therefore, the direct application of fluctuating surface pressures measured in conventional wind tunnels to panel vibration and flutter research is questionable since typical resonant frequencies of panels will be at low frequencies. Indeed, by analogy with the measured effects of stream turbulence on low-speed turbulent boundary layers,[16,17] and the direct effects of noise on low-speed turbulent jets[18] and boundary layers,[19] the effects of high noise levels on the basic structure of supersonic turbulent boundary layers and free shear

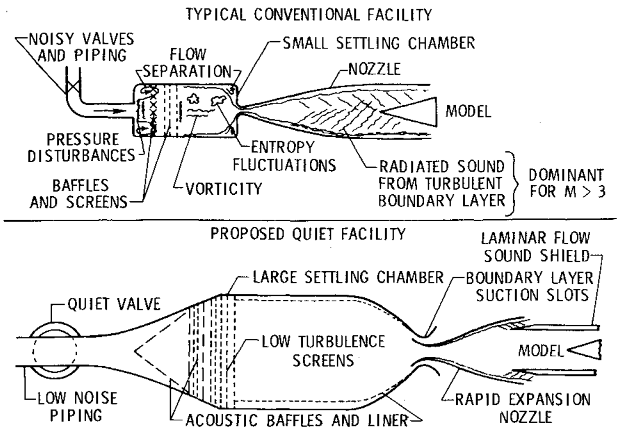


Fig. 1 Schematic sketches of a typical supersonic tunnel and proposed quiet tunnel.

layers can be expected to be of significance. Research in these areas cannot be advanced until methods become available to control and reduce facility disturbances by an order of magnitude for operational Reynolds numbers of 30 to 40 million. Current concepts now being considered or under development for use in a quiet tunnel are schematically illustrated in Fig. 1.

The purpose of this paper is to report on a continuing research program† at the Langley Research Center to develop and test techniques to control and reduce disturbance levels in supersonic and hypersonic wind tunnels. Available data required to determine the range of operating conditions and disturbance levels for a quiet tunnel are reviewed. New data on rms pressure levels and spectra under a turbulent boundary layer in a conventional noisy Mach 6 wind tunnel are presented. Current techniques and new data on methods for reducing facility noise levels are presented.

Operating Limits for a Quiet Tunnel

Fluctuating Pressures Under Turbulent Boundary Layers

Previous investigations[11,13] have shown that over-all rms pressure intensities under laminar and turbulent boundary layers on models in the same facility may be nearly the same. The power spectra under the turbulent boundary layer in these investigations were dominated by energy at low frequencies. This low-frequency energy was clearly the result of sound radiated from the turbulent boundary layers on the wind-tunnel walls. The data of Dods and Hanly[11] at Mach 2 indicated the turbulent boundary layer may have interacted with the tunnel noise, since at frequencies below about 200 Hz, the power spectra measured under the turbulent boundary layer were below that of the laminar boundary layer. Of course, a laminar boundary layer on a model may also modify the facility noise as shown by Mack,[7] depending on the frequencies and inclination angles of the input pressure waves and on the boundary-layer thickness.

Preliminary results of an investigation by Stainback in the Mach 6 High Reynolds Number Tunnel at Langley are presented in Fig. 2. The basic test and calibration techniques and data reduction procedures are the same as developed previously by Stainback[8] except that the gage size used here is $\frac{1}{8}$ in.-diam with a frequency response to about 160 kHz. The variation with unit Reynolds number of the rms pressure (uncorrected for finite transducer size) at the cone surface normalized with local cone static pressure is shown in Fig. 2a. Previous data[8] obtained with a 0.44-in.-diam transducer also mounted flush with the surface on the same cone and in the same wind tunnel are included for comparison. The rms levels of the new data from the smaller transducer are higher than the previous values, presumably because of the improved frequency response of the new gage. The present results show the boundary layer on the rear gage is fully turbulent for $Re/\text{ft} > 10^7$. Comparison with the corresponding levels measured on the forward gage indicates that as the Reynolds number is increased, the levels under the turbulent boundary layer are only 10% to 20% higher than those at the forward gage before transition occurs there at $Re/\text{ft} \approx 1.8 \times 10^7$.

Selected spectra from the two surface gages are given in Fig. 2b. The large energies in these spectra below about 20 kHz are due to sound radiated from the tunnel wall boundary layer as indicated by comparison with the spectra from a $\frac{1}{4}$-in.-diam transducer mounted in a pitot tube in the freestream. The increase in energy at the 4-in. cone station relative to the free-stream spectra for $f > 20$ kHz is presumably due to amplification of tunnel noise by the laminar boundary layer on the cone. The mechanism involved here may be analogous to the "forced response" of a laminar boundary layer to an external sound field as calculated by Mack[7] from linear stability theory and measured by Kendall[6] with a hot wire in a laminar boundary layer on a flat plate.

At $Re/\text{ft} \simeq 14 \times 10^6$ $(R_\infty/\text{ft} \simeq 10 \times 10^6)$, the tunnel wall boundary layer is about 1 in. thick.[20] With the assumption that the convection velocity of disturbances or eddies is 0.6 of local stream velocity, and that their scales are of order δ, the dominant frequencies of radiated sound would be approximately

$$f_r \approx (u_c/\delta)_{WT} \approx 20 \text{ kHz}$$

The boundary-layer thickness on the model at the rear gage for these same conditions is estimated to be $\delta_M \approx 0.1$ in. which gives

$$f_{TBL} \approx (u_c/\delta)_M \approx 200 \text{ kHz}$$

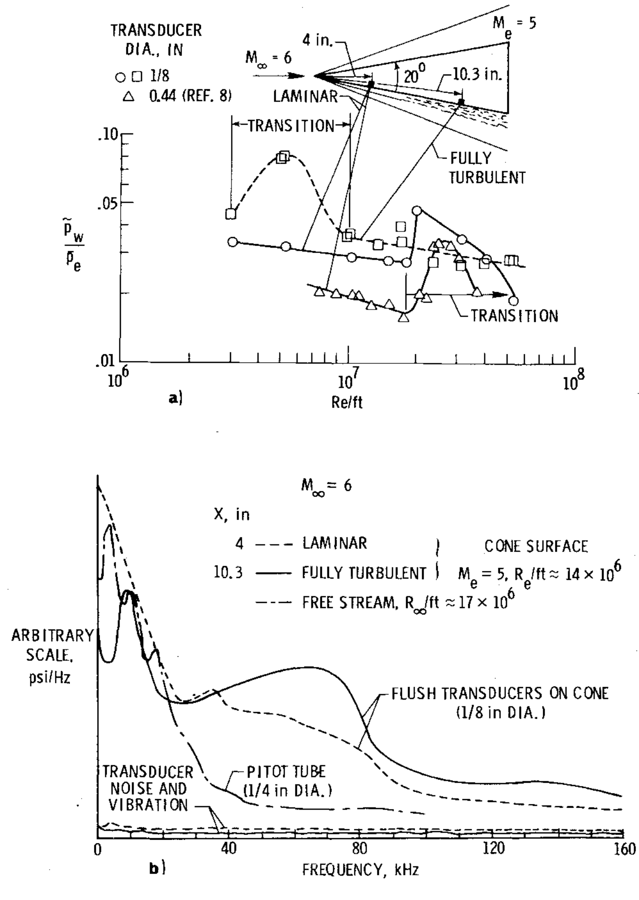


Fig. 2 Measurements of fluctuating pressures under laminar and turbulent boundary layers on a sharp cone in Mach 6 high Reynolds number tunnel at NASA Langley. a) Variation of over-all rms pressure with unit Reynolds number. b) Spectra of pressure fluctuations with $\Delta f = 200$ Hz.

† Results in this program have been achieved by a team effort and individual contributions from all members of the Gas Dynamics Section at Langley, particularly P. C. Stainback, W. D. Harvey, and J. B. Anders.

While the response of the flush gages is only 160 kHz, the location of peaks in these spectral curves ($f \approx 5$ to 8 kHz at $x = 4$ in. and $f \approx 70$ kHz at $x = 10.3$ in., respectively; Fig. 2b) indicates the dominant eddy scales in the turbulent boundary layers on both the model and the tunnel wall are roughly three times larger than the boundary-layer thickness. In agreement with the results of Dods and Hanly[11] at Mach 2, the present results indicate the turbulent boundary layer on the model may attenuate the low-frequency sound generated by the facility. Obviously, much more research is required before the effects of acoustic inputs on supersonic turbulent boundary layers can be understood. A quiet tunnel with controlled and reduced disturbances but with sufficiently high test Reynolds numbers to maintain fully turbulent boundary layers on models will be essential in this important research work.

Transition Reynolds Number Correlations

Figure 3 shows the variation of transition Reynolds number with tunnel disturbance intensities for flows with zero streamwise pressure gradients. The data of Stainback,[8,10] were taken in four different wind tunnels. Disturbance levels were measured with surface pressure transducers under laminar boundary layers on sharp cones. The stream Mach number varied from 6 to 8 while the local Mach number was maintained at 5 by adjusting the cone angle. The data of Fischer and Wagner[8,21] were obtained in two different helium tunnels and the disturbance levels were measured with hot wires in the freestream and cone flowfield. The dominant disturbance mode was sound radiated by the tunnel wall turbulent boundary layers. The results of these investigations suggest that when other parameters are held fixed such as local Mach number, wall to stagnation temperature ratio, and pressure gradient, the transition Reynolds numbers were dominated by the rms sound level for the conditions of these tests.[8,10,21]

The low-speed ($M \simeq 0$) results in Fig. 3 are based on data[22-24] for zero pressure gradient and suggest that transition Reynolds number depends mainly on stream turbulence levels for $\tilde{u}/u_e > 3 \times 10^{-3}$. For lower turbulence levels the transition Reynolds number was strongly dependent on the spectral content of acoustic disturbances.[23,24] Whether similar results can be expected at supersonic and hypersonic speeds must remain speculative until the proper research equipment can be developed. While the disturbance levels for the flight data shown in Fig. 3 are not known, a few measurements such as those of Dods and Hanly[11] indicate levels well below 0.5%. In any case, when disturbances are reduced below this level, there is little doubt that the spectral content of remaining disturbances will affect transition as pointed out by Morkovin.[1,2] The only way to determine which frequencies and types of disturbances are critical in the nonlinear transition process is to simulate disturbances in a quiet tunnel with rms pressure intensities reduced to at least 0.5%. Levels below 0.5% will probably be required to simulate flight conditions.

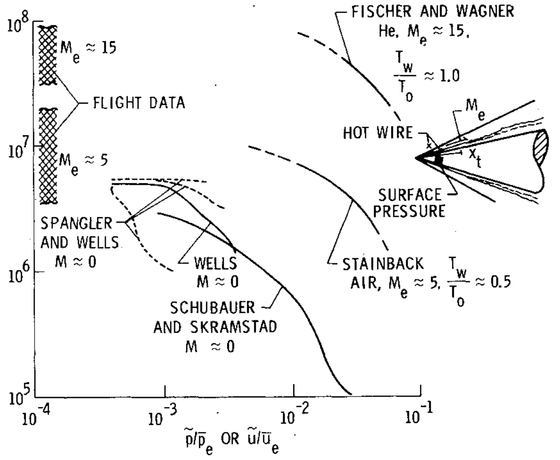


Fig. 3 Variation of transition Reynolds number with rms disturbance levels for $d\bar{p}/dx = 0$.

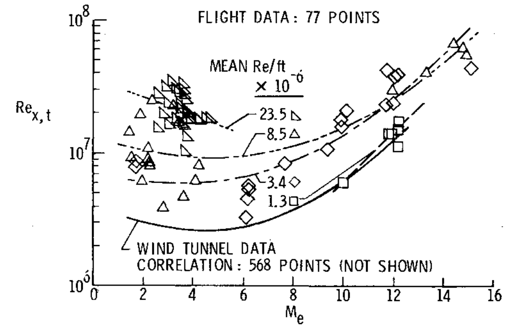


Fig. 4 Correlation of transition Reynolds number data on sharp cones in wind tunnels and in flight.[9]

To determine the minimum operational Reynolds numbers for a quiet tunnel, it is necessary to consider free-flight data for transition Reynolds numbers. Data from sharp cones at small angle of attack are particularly well suited for this purpose since the effects of geometry, leading-edge bluntness, side boundaries, and crossflow or three-dimensional flows are minimized. The variation of transition onset with local Mach number and local unit Reynolds number is shown in Fig. 4. The dashed faired lines are least-squares curve fits of the free-flight data shown. The correlation procedures and data are the same as used in Ref. 9. The mean *Re*/ft values shown for the flight data are intermediate values for data which are included in four ranges of $Re/\text{ft} \times 10^{-6}$ as follows: 0.7 to 1.8, 1.8 to 5.0, 5.0 to 12, and 12 to 35.

The flight data are generally higher than the correlation line for the wind-tunnel data,[9] presumably due to large disturbance levels in wind tunnels. The large increase in transition Reynolds number with *Re*/ft is clearly evident in the flight data. This trend of the flight data implies that if only lower values of R_∞/ft with quiet operation ($\tilde{p}/p_e < 0.005$) in wind tunnels can be realized, useful transition data can still be obtained if the facility can accommodate models of sufficient length. Thus, for example, if $R_\infty/\text{ft} \leqq 2.5 \times 10^6$ for quiet operation, then the onset of transition would be expected (Fig. 4) at about $R_{\infty,x} \simeq 5 \times 10^6$ in the Mach number range from 2 to 7. The entire transition process could then be observed if the Reynolds number capability extended up to $R_{\infty,x} \simeq 10 \times 10^6$ which would require a model of about 4 ft in length and therefore a test section diameter of about 18 to 20 in. for $M_\infty \approx 5$ (allowing for tunnel wall boundary-layer thickness and typical model sizes).

Laminarization of Nozzle Wall Boundary Layers

Measurements in the JPL tunnel[3] and in the 4 in. Mach 5 nozzle at Langley[25,26] have shown that when the sidewall boundary layers are laminar, the stream disturbance levels are reduced by an order of magnitude. Therefore, one of the principal design objectives for a quiet tunnel is to maintain laminar boundary layers on the walls of the nozzle and test section up to sufficiently large Reynolds numbers so that the "natural" transition process can be completed in a model boundary layer or shear layer. Laminar boundary layers have been observed on the walls of several nozzles. Since a detailed review of the various factors involved and conditions required to obtain laminar boundary layers in supersonic and hypersonic nozzles is now available,[25] only brief comments and some new results will be included herein.

Conventional Nozzles

As the unit Reynolds number or operating pressure of a tunnel is increased, the wall boundary layers usually become turbulent before the test section unit Reynolds number reaches 1×10^6 per foot. The Reynolds number for transition in the wall boundary

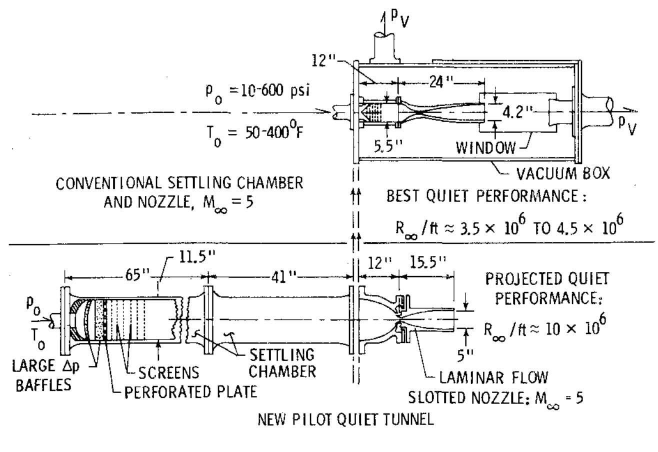
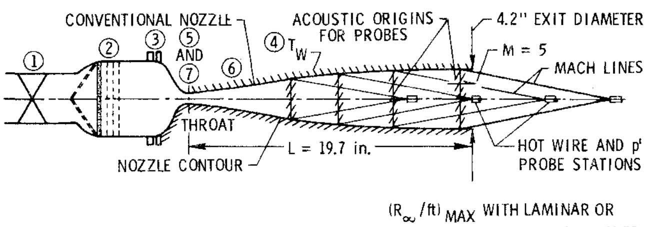


Fig. 5 Facility now being used to test and develop quiet tunnel.

layer depends on the Mach number, nozzle size, and several other factors such as wall roughness, acceleration in the nozzle approach, settling chamber design, wall cooling, and so forth. The largest Reynolds number yet reported for transition in a nozzle boundary layer was obtained recently in a Mach 5 conventional nozzle at Langley.[26] The small-scale facility used in the investigation is shown in Fig. 5. The new laminar flow slotted nozzle shown in the lower part of Fig. 5 will be discussed in a following section of this report.

Several factors thought to be of importance in transition of nozzle wall boundary layers were investigated as listed in Fig. 6. The dominant factor in this case was the wall roughness. Also of interest was the behavior of transition for the tests with the nominal "unpolished" wall. The results[25,26] indicated that transition occurred simultaneously throughout the nozzle and was therefore apparently initiated in or upstream of the throat. Since cleaning and polishing the nozzle throat increased the transition Reynolds number by about 50%, results with the unpolished wall were probably roughness dominated.

The effect of Taylor-Göertler vortices on transition in the concave regions of nozzle wall boundary layers has not been reported in the literature. Therefore, we conducted some oil-flow tests with the small Mach 5 conventional nozzle. The wall roughness was the nominal "unpolished" condition for these tests. The results are shown in Fig. 7. A thin film of oil pigmented with alumina oxide was applied uniformly to the nozzle wall upstream of the throat and near the exit. Constant operating pressure and temperature were then maintained until the oil film patterns were stabilized. Evidence of longitudinal vortices just downstream of the concave portion of the subsonic approach (upper photographs) and in the downstream supersonic region



FACTORS INVESTIGATED		$(R_\infty/ft)_{MAX}$ WITH LAMINAR OR TRANSITIONAL WALL BOUNDARY LAYER
1. UPSTREAM VALVES AND PIPING	NOMINAL UNPOLISHED WALL	3.0×10^6 PER ft
2. SETTLING CHAMBER SCREENS AND BAFFLES		3.0×10^6
3. APPROACH MISALIGNMENT		3.0×10^6
4. HEATING NOZZLE WALL		3.8×10^6
5. WALL ROUGHNESS IN THROAT		2.2×10^6
6. RAPID EXPANSION NOZZLE, L = 15.3 in.		3.9×10^6

TRANSITION BEHAVIOR: OCCURS SIMULTANEOUSLY THROUGHOUT NOZZLE --- HENCE IS INITIATED IN OR UPSTREAM OF THROAT

7. POLISHED THROAT REGION (CONVENTIONAL NOZZLE)	4.5×10^6

Fig. 6 Effect of several factors on transition in the boundary layer on the wall of a conventional nozzle.[26]

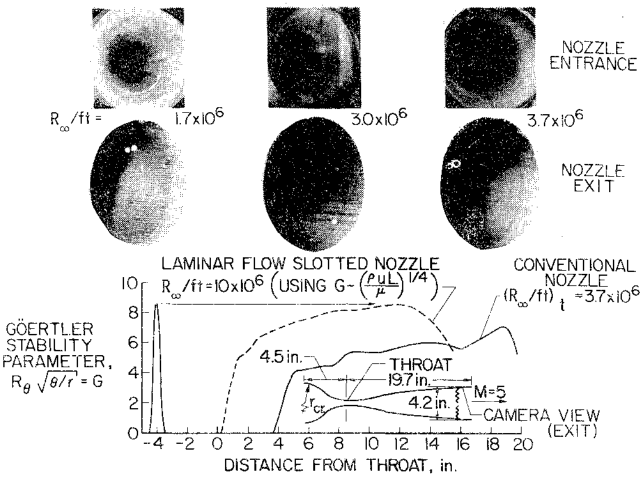


Fig. 7 Evidence of Taylor-Göertler vortices in a nozzle wall boundary layer and projected effects of these vortices on transition in the laminar flow slotted nozzle.

(lower photographs) are apparent at the two lower values of R_∞/ft. As the R_∞/ft was increased, the streaks tended to diffuse and disappear as indicated by the photographs at $R_\infty/ft \simeq 3.7 \times 10^6$. Since transition was observed[26] by three independent methods at $R_\infty/ft \leqq 3.5 \times 10^6$ it is possible that the Taylor-Göertler vortices were involved in transition of the nozzle wall boundary layer for the conditions of these tests. Therefore, the distribution of the Göertler stability parameter G for this nozzle was computed and the results are shown in the lower graph of Fig. 7. For $R_\infty/ft \simeq 3.7 \times 10^6$, the peak value of G upstream of the throat exceeds the values observed[27] for transition in low-speed flow by about 20%. The peak value in the downstream concave region of the nozzle is about 7 which is the transitional value observed by Liepmann.[27] We have therefore concluded tentatively that Taylor-Göertler vortices were an important factor in transition of the nozzle wall boundary layer for these tests. Before the projections of these results to the new laminar flow slotted nozzle are discussed, we will describe briefly the purpose and design of this nozzle.

Laminar Flow Slotted Nozzle

Klebanoff and Spangenburg at the National Bureau of Standards (NBS) have maintained laminar boundary layers on the walls of a small Mach 2 nozzle up to $R_{\infty,x} \simeq 2 \times 10^6$ (where x is the distance from the nozzle throat) by the use of lateral boundary-layer scoops or slots upstream of the nozzle throat. Longer runs of laminar flow up to $R_{\infty,x} \simeq 3.3 \times 10^6$ were obtained with longitudinal suction slots in the tunnel sidewall. (This work at the NBS is described briefly in Ref. 9.) The function of the subsonic lateral slot was to remove the turbulent boundary layer that forms in the settling chamber and nozzle approach just before the flow is expanded to supersonic speeds. Thus, any stream disturbances that would be caused by a lateral slot in supersonic flow are avoided and a new laminar boundary layer is started at the lateral suction slot. The longitudinal suction slots were intended to maintain laminar flow without introducing stream disturbances into the supersonic flow region.

Both concepts have been utilized in the design of the laminar flow slotted nozzle as illustrated in Figs. 5 and 8. Figure 8a shows the subsonic lateral slot and rapid expansion nozzle with a rodded sound shield installed at the nozzle exit. This sound shield and test results of a conceptual model will be described in the next section of this paper.

Details of the subsonic lateral slot are shown in Fig. 8b. This slot is designed to completely remove the boundary layer that forms in the settling chamber and nozzle approach, including the Taylor-Göertler vortices (Fig. 7) that would be present due to the concave curvature in the approach. The slot lip was placed just downstream of the inflection point in the streamlines. The new

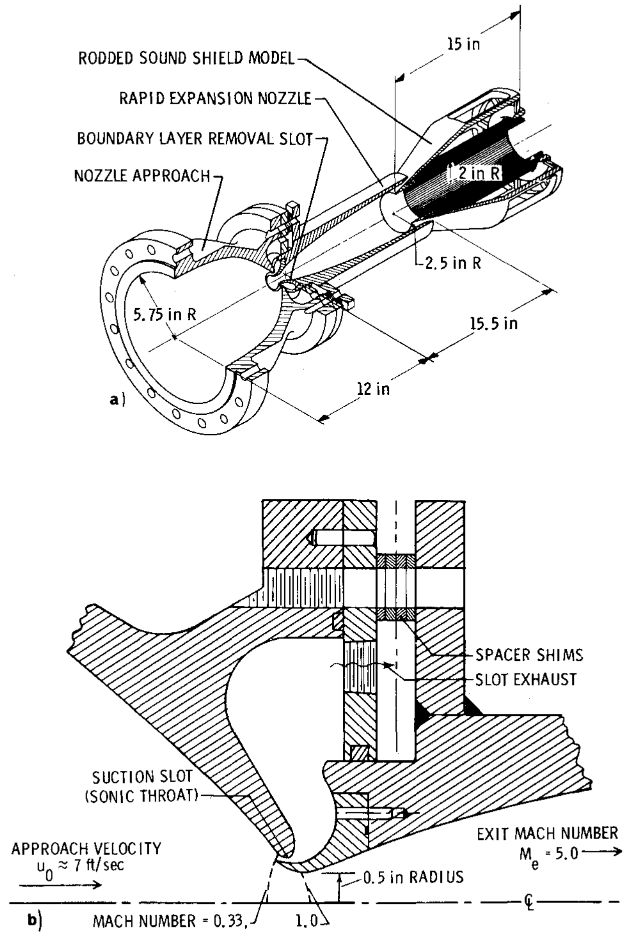


Fig. 8 Design details of Mach 5 pilot quiet tunnel. a) Slotted nozzle with rodded wall sound shield installed in test section. b) Transonic region of nozzle.

boundary layer is therefore not subjected to concave curvature until far downstream where the density and local Reynolds number are smaller than in the transonic region.

After careful consideration and analysis of several other possible methods for construction of laminar flow nozzles, the present design was chosen[25] as the most practical. Critical design problems of this nozzle have been analyzed[25,28] including the shape and height of the slot required to remove the entire boundary layer. Predictions[25,26] for laminar boundary layers throughout the supersonic part of the nozzle range from $R_\infty/\text{ft} \simeq 10 \times 10^6$ based on local R_θ values and laminarization due to rapid expansion down to $R_\infty/\text{ft} \simeq 7 \times 10^6$ based on finite-difference solutions of the boundary-layer equations by the transitional method of Shamroth and McDonald.[29] (This method cannot account for the detailed effects of Taylor-Göertler vortices on transition.) These predictions indicate the slot design height (Fig. 8b) is barely large enough to remove the velocity boundary layer at this station. The spacer shims shown in Fig. 8b provide for axial adjustment of the slot lip with respect to the nozzle approach.

The predictions in Fig. 7 of laminar boundary layers up to $R_\infty/\text{ft} \simeq 10 \times 10^6$ for the pilot slotted nozzle are based on the apparent critical value of $G \simeq 8.5$ in the subsonic approach of the conventional nozzle. That is, this critical value of G is assumed to apply in the supersonic concave region of the slotted nozzle as indicated in the figure. The proportional relation between G and Reynolds number (shown in the figure) has been utilized in this extrapolation procedure.

Preliminary tests of the pilot slotted nozzle indicated that transition occurred at $R_\infty/\text{ft} \approx 2.5 \times 10^6$. This poor performance is tentatively attributed[26] to improper spacing or design of the slot. Later tests with the slot throat opened up by 0.020 in. increased transition to $R_\infty/\text{ft} \approx 3.5 \times 10^6$. At $R_\infty/\text{ft} = 13 \times 10^6$ where the boundary layer was fully turbulent, the measured[26] rms pitot pressure was 0.5% of mean pitot pressure; considerably smaller than values in conventional nozzles. However, to reduce the disturbance levels below 0.1% without sound shields requires laminar boundary layers on the nozzle wall.

Application of these preliminary results to the proposed 20 in. tunnel indicates that the desired minimum unit Reynolds number for quiet operation of 2.5×10^6 per ft (see the previous section; *Transition Reynolds Number Correlations*) will be very difficult to obtain since the corresponding value in the 5 in. pilot tunnel should be approximately four times larger. That is, to delay transition in the pilot nozzle to $R_\infty/\text{ft} \simeq 10 \times 10^6$ (and $G \simeq 8.5$) will require extremely small stream disturbance levels, a "mirror" finish on the walls, and minimum wall waviness. We therefore conclude tentatively that a practical alternative technique to obtain low disturbance levels in the 20 in. quiet tunnel is by the use of a sound shield which, in any case, is required at the larger Reynolds numbers necessary to study fully turbulent boundary layers.

Development of Noise Shields

Streamwise Suction Slots

Tests by Pate and Schueler[4] at $M_\infty = 3.0$ have shown that rms pressure levels within the shielded region of an axisymmetric shroud (i.e., downstream of Mach line disturbances from the shroud leading edge) are reduced by about 50% below stream levels when the boundary layer on the inside of the shroud is laminar. These results indicate that about one-half the local sound intensity for these tests[4] entered the shielded region presumably by flow convection or by reflection at or transmission through the walls of the shroud. Since the boundary layer on even the optimum design laminar flow nozzle will become turbulent at sufficiently high operating pressures, the shroud or sound shield technique offers a way to reduce the stream sound levels at high operating pressures if the boundary layer on the inside of the shield can be maintained laminar.

The rodded sound shield illustrated in Fig. 8a is designed to maintain a laminar boundary layer on its inside walls. This shield encloses the test section and consists of an axisymmetric array of small rods with adjustable gaps between the rods. The nozzle and shield are mounted in a vacuum chamber (see Fig. 5) so that the nozzle wall boundary layer can be removed before the flow enters the shielded region. The new boundary layer that forms on the longitudinal rods is then partially removed through gaps between the rods by the pressure drop from the stream side to the vacuum side of the rod array. This sound shield is not yet available for tests which will be primarily at the higher operating pressures for which the nozzle wall boundary layer would be turbulent.

Detailed analysis and tests results at Mach 6 and 8 of a conceptual rodded shield model are given in Ref. 30. These results were obtained with the model shown in Fig. 9. Both round and wedge-shaped rods were tested. The model is tested at angle of attack such that the pressure drop across the rod array is large enough to generate sonic crossflow through the gaps between the rods. Therefore, noise radiated directly from the tunnel wall boundary layer or shear layer and noise generated on the lee side of the rods would not be transmitted through the sonic portion of the gap flow. Presumably, some noise could pass through the thin subsonic viscous layer in the gaps and some low-frequency sound may be transmitted through the rods depending on their size and how they are supported. Noise measurements in the flowfield of a smaller model similar to the one in Fig. 9 showed that rms pressure levels were reduced by about 30%.[9] However, localized disturbances from the rod supports for this small model compromised the results. An improved support system has been developed for the new model and measurements in the shielded region show the noise levels are reduced by nearly 50% with the round rods at Mach 6 when the boundary layers on the rods are laminar.[30] The local unit

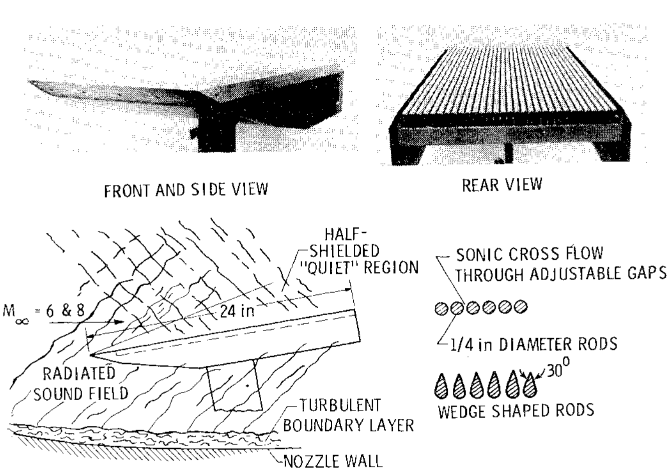


Fig. 9 Conceptual noise shield model with streamwise slots.[30]

Reynolds number for transition on the rods was 8×10^6 per ft. Analysis[30] of these results at Mach 6 show that this reduction in stream noise approached the maximum possible for an ideal flat shield with nearly zero reflection of incident sound and nearly zero generation and transmission of radiated sound into the shielded region. Furthermore, by analogy with swept cylinder flow,[30] transition on the rods should be essentially independent of the rod length. Hence, the application of this sound shield concept to a wind tunnel as illustrated in Fig. 8a would result in substantial reductions in rms pressures of at least 90% (or by approximately 20 dB in sound pressure level) up to length Reynolds numbers that would be determined approximately by the length of the shield times the unit Reynolds number for transition.

Application of these results to the proposed 20 in. tunnel then indicates that rms pitot pressure levels of nearly 0.05% could be achieved (recall that levels of 0.5% were measured in the pilot slotted nozzle with fully turbulent wall boundary layers) at Reynolds numbers of 30 to 40 million with a 4 to 5-foot-long shield.

Aerodynamic Noise Shields

It is well known that total reflection of incident sound waves at a moving interface will occur for certain combinations of incident wave angles and relative Mach numbers.[31,32] For example, at incident angles above 30°, total reflection occurs for $1 < M_r < 3$, where M_r is the relative Mach number between the moving layers. Thus, if a wake can be established within a supersonic flow such that the difference in Mach number within the wake and the surrounding flow is supersonic, the wake becomes a barrier to transmitted sound and also blocks transmission of aerodynamic sound that may be generated within the wake itself.

Hot-wire measurements in the wake of a 15° half-angle wedge by Wagner and Weinstein[33] in a Mach 15 helium tunnel showed that the hot inviscid wake did not transmit noise emanating from the nozzle sidewall boundary layer. These measurements were made at 68 base thicknesses downstream of the wedge, and no signal above instrument noise was detected outside the turbulent wake. This result is attributed by Wagner to the very small velocity defect within the wake, which prevents eddy Mach wave radiation of sound from the turbulence in the wake. The wake then becomes an aerodynamic shield to both sound transmission and to radiation of internally generated sound.

This concept of a hot-wake aerodynamic sound shield has been tested in the nozzle test chamber (Fig. 5). A thin wall split ring (max wall thickness, $\frac{1}{16}$ in.) shown in Fig. 10a was fabricated of nickel alloy and mounted at the exit of the Mach 5 conventional nozzle. The ring could be heated to 2300 R by a high amperage, low voltage, 60 cycle current. Fluctuating pitot pressures were measured 8.5 in. downstream of the ring trailing edge.

A Schlieren photograph of the hot wake and flowfield downstream of the ring is shown in Fig. 10b. The flowfield is obviously not suitable for model tests due to large mean flow disturbances. Since the only purpose of this investigation was to test the concept of noise reduction by a hot wake, the probe was located in a relatively undisturbed region downstream of the forward Mach wave boundary of the wake. For the conditions of Fig. 10b, the probe "sees" a turbulent wake since, from a careful examination of the original photograph, transition apparently occurred at 1 to 2 in. downstream of the ring trailing edge.

Results of the rms fluctuating pitot measurements normalized by the rms value at the same point in the flow with the ring unheated ($T_{ring} = T_{aw,ring}$) are given in Fig. 10c. While there is considerable scatter in the data depending on probe location, the effect of heating the ring was to reduce the rms disturbance levels by 20% to 45%. The R_∞/ft values for these tests are above transitional values[25,26] for the nominal unpolished nozzle wall and hence the nozzle wall boundary layer and free jet shear layer were turbulent. Thus, if a hot wake can be generated without introducing mean flow disturbances, the desired sound shielding effect for a quiet tunnel can be obtained.

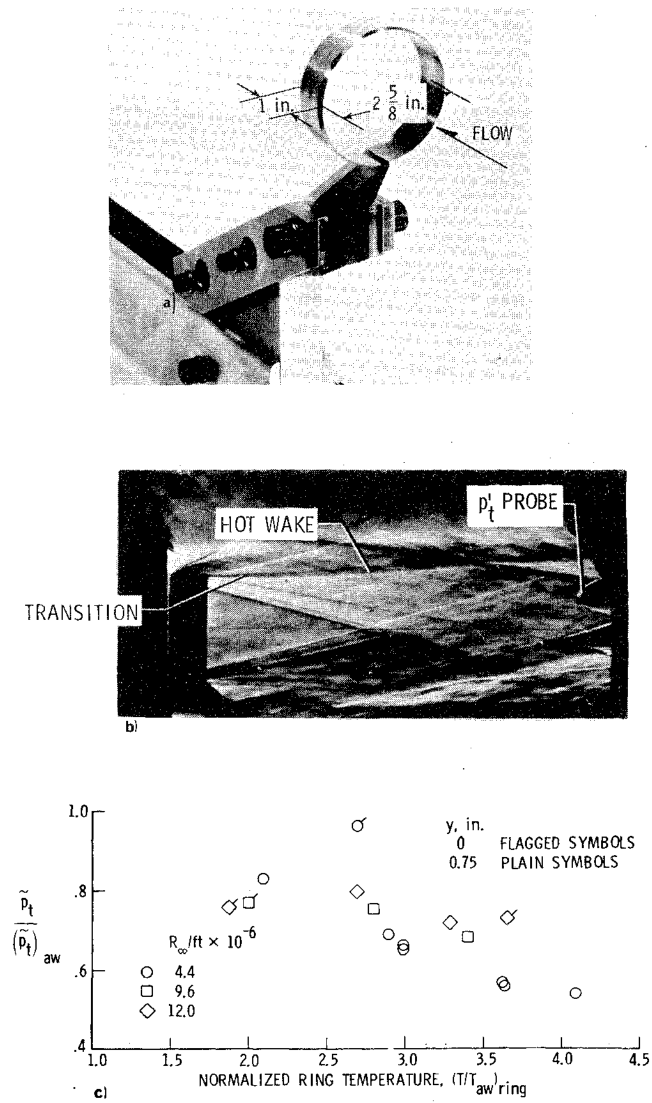


Fig. 10 Development of aerodynamic noise shields. a) Heated ring model for producing hot-wake noise shield; b) schlieren photograph of heated ring model, $M_\infty \approx 5$, $R_\infty/\mathrm{ft} \approx 12 \times 10^6$, $(T/T_{aw})_{ring} \approx 3.3$; c) effect of hot-wake shield on rms pitot pressures, $M_\infty = 5.3$, $x = 8.5$ in. downstream of ring, $T_0 \approx 680°$R.

Conclusions

Review of flight transition data and the effects of rms disturbances on transition in wind tunnels has indicated that flight transition conditions at high altitudes can be simulated in a wind tunnel at unit Reynolds numbers as low as 2.5×10^6 per foot with length Reynolds numbers up to 10×10^6 with disturbance levels of 0.1% or less.

New data on fluctuating pressures under turbulent boundary layers have emphasized the possibility that high levels of wind-tunnel noise may modify the structure of fully turbulent shear layers at supersonic speeds. Already it is known that fluctuating surface loads measured under turbulent boundary layers in wind tunnels are dominated by tunnel disturbances. Further research in these areas will require a quiet tunnel with length Reynolds numbers up to 40×10^6 and disturbance levels from 0.1% to 0.5%.

Experimental data on current techniques for achieving these requirements are reviewed. Preliminary tests on a Mach 5 pilot slotted nozzle showed that transition in the nozzle wall boundary layer occurred at $R_\infty/\text{ft} \approx 2.5 \times 10^6$. Modifications to the slot shape and other improvements in the nozzle will be required before the predicted value of $R_\infty/\text{ft} \approx 10 \times 10^6$ at transition can be achieved. On the other hand, the measured stream disturbance levels at $R_\infty/\text{ft} = 13 \times 10^6$ were considerably lower than previously observed in conventional nozzles.

To obtain higher test Reynolds numbers, some type of radiation noise shield will be required. Results on two types of shields are discussed. Analysis of the data on a conceptual model of a longitudinal slotted shield has shown that laminar boundary layers can be maintained on the walls of the shield up to $R_\infty/\text{ft} \simeq 8 \times 10^6$. Measured disturbance levels have been reduced by nearly 50% within the "half" shielded flow region of this rodded model. Application of this concept to a complete wind-tunnel shield would therefore be expected to reduce sound pressure levels in the test section by about 20 dB up to Reynolds numbers determined approximately by the length of the shield times the unit Reynolds number for transition. Therefore, the development of a rodded wall sound shield for the proposed 20 in. Mach 5 tunnel is a practical approach to the requirement of disturbance levels below 0.1% at test Reynolds numbers approaching 40 million.

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