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Görtler Instability and Hypersonic Quiet Nozzle Design

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A new concept for nozzle design incorporating slow expansion rates and a radial flow region has been implemented for a hypersonic (Mach 6) nozzle. Data indicate that these modifications have the potential for providing a large increase in the length of the quiet test core by postponing the initiation and decreasing the growth of Görtler vortices on the concave region of the nozzle walls. The concept was verified previously by experimental data from the advanced Mach 3.5 axisymmetric quiet nozzle. To verify the concept for hypersonic flow, the new advanced Mach 6 axisymmetric quiet nozzle was built and tested at NASA Langley. Preliminary experimental results on the extent of laminar wall boundary-layer flow are compared with data from the Mach 5 axisymmetric, rapid expansion, pilot quiet nozzle and with theoretical predictions based on linear stability theory. The Reynolds numbers based on the measured length of the quiet test core for the new nozzle are in agreement with the trend of theoretical predictions in the low unit Reynolds number range. With further improvement of the nozzle finish, especially in the critical throat region, the nozzle performance can be expected to approach the design conditions.

Nomenclature

k = peak-to-valley height of a surface roughness defect or a scratch, μ
M = Mach number
N = exponential factor in amplification ratio e^N from linear stability theory
n = wave number, $2\pi r/\lambda$
p = pressure, N/cm^2
R = unit Reynolds number, $\rho u/\mu$
R_k = local Reynold number evaluated at the roughness height, $\rho uk/\mu$
$R_{\Delta X}$ = freestream Reynolds number based on the length ΔX, $(\rho u \Delta X/\mu)_\infty$
r = radius, cm
T = temperature, °C
u = streamwise velocity, m/s
X = axial distance from nozzle throat, cm
X_o = upstream tip of uniform flow test rhombus, cm
ΔX = length of quiet test core, cm
δ = boundary-layer thickness, cm
λ = wavelength, cm
μ = dynamic viscosity, P
ρ = mass density, kg/m^3

Subscripts

c = centerline
G = Görtler vortices
o = stagnation conditions
T = transition onset
TS = Tollmien-Schlichting waves
t = pitot
aw = adiabatic wall
w = wall
∞ = freestream

Superscripts

$\sim$ = rms
$-$ = mean

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I. Introduction

WIND tunnels with very low freestream disturbance levels comparable to free-flight conditions are required for boundary-layer stability and transition research and to ultimately provide reliable predictions of transition for high-speed flight vehicles. High freestream disturbance levels in conventional wind tunnels cause premature boundary-layer transition on test models.[1-5] There can be many sources of these disturbances, but at higher Mach numbers ($M > 2.5$) the primary source is the eddy-Mach-wave radiation from the turbulent boundary layers on the nozzle wall.[6] The mechanism of high-speed turbulent boundary-layer noise radiation was identified in the theoretical work of Phillips[7] and further examined by Ffowcs-Williams and Maidanik.[8] This phenomenon was then confirmed experimentally by Laufer et al.[9]

The NASA Langley Research Center has conducted high-speed quiet-tunnel research and development for over 18 years.[10-25] These studies indicated that mere reduction of the rms noise levels (e.g., through control of the turbulence field in the nozzle wall boundary layer) was not adequate, since the high-frequency content of the incident noise field still affected transition on test models. The only feasible technique to significantly reduce this high-frequency noise is to maintain the nozzle wall boundary layer in a laminar state. Therefore, quiet-tunnel development is essentially a laminar flow control (LFC) problem. Unfortunately, the general LFC technique of local suction cannot be used to accomplish laminarization in supersonic nozzles due to consequent production of finite strength waves at the suction site. This wave system usually causes stream disturbance levels even higher than the turbulent wall radiated noise.[26] By 1983, a viable Mach 3.5 two-dimensional pilot quiet tunnel[19] was produced by using a combination of techniques including 1) a large settling chamber with dense porous plates as acoustic baffles to attenuate upstream pipe/valve noise and turbulence screens to reduce vorticity disturbances upstream of the nozzle entrance, 2) boundary-layer suction slots upstream of the throat to remove the turbulent boundary layers in the subsonic approach to the nozzle, and 3) rapid expansion contours to limit the growth of both Tollmien-Schlichting and Görtler modes. In the Mach 3.5 two-dimensional pilot quiet nozzle, the laminar boundary layer downstream of the suction slots could be extended to a limited distance downstream of the throat due to strong favorable pressure gradients along the rapid expansion contours, provided that the nozzle walls wer maintained clean and highly polished. A small quiet test core with very low noise levels in the upstream regions of the test rhombus was obtained because the corresponding upstream "acoustic origin" regions of the nozzle wall boundary layers were within the aforementioned laminar range. The hot-wire measurements of the noise field in this nozzle showed that the maximum length of the quiet test core that could be maintained up to the highest test freestream unit Reynolds number, $R_\infty = 8 \times 10^7$/m, was approximately 12 cm. Even with such a short length of quiet test core, transition data measured in this pilot quiet tunnel reached the low range of flight data.[27]

The short length of the quiet test core in the rapid expansion, pilot quiet nozzle limits the length of test models that can be exposed to the low noise flow. To develop an efficient technique to enlarge the quiet test core, it is essential to understand the instability mechanisms involved. Previous investigations[20,21] indicated that transition in the wall boundary layers of nozzles for Mach numbers from 3 to 5 was caused by the Görtler instability mechanism in the concave curvature regions rather than Tollmien-Schlichting waves. Based on this finding, a new concept for quiet nozzle design was proposed.[25] The new concept was to insert a straight wall, radial flow section upstream of the inflection point of the contoured nozzle wall. The initiation of the Görtler instability was then delayed until the beginning of the concave nozzle wall and the resulting slower expansion and larger radii of curvature reduced the growth rate of Görtler vortices. This concept was used to design an advanced Mach 3.5 axisymmetric quiet nozzle. Wind-tunnel tests of the new nozzle confirmed the improved nozzle performance compared with other pilot quiet nozzles.[25] The success of the advanced quiet nozzle in the supersonic range was the basis for application of the new design concept

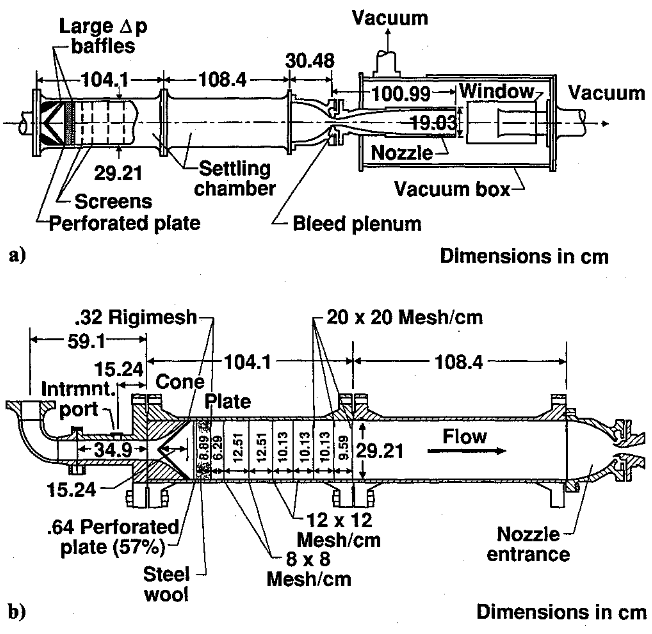

Fig. 1 Nozzle test chamber: a) general arrangement of test facility, b) settling chamber.

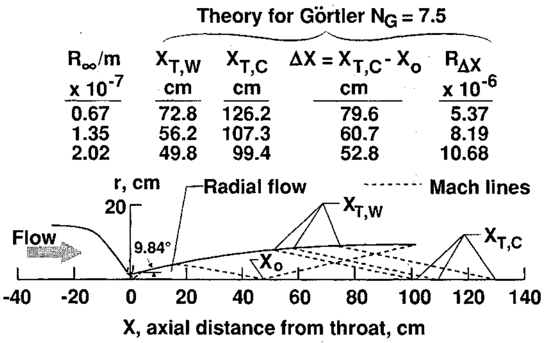

Fig. 2 Advanced Mach 6 axisymmetric quiet nozzle.

to the hypersonic speed range. The advanced Mach 6 axisymmetric quiet nozzle is the first hypersonic quiet nozzle designed by the new concept and preliminary test results are presented in this paper.

II. Test Facility

The advanced Mach 6 axisymmetric quiet nozzle was designed to be used in the existing nozzle test chamber located in the Gas Dynamics Laboratory at the NASA Langley Research Center. The general arrangement of the test facility is shown in Fig. 1a. The supply air for this blowdown facility comes from the high-pressure tank field through the test facility and is exhausted to the downstream vacuum spheres. Detailed descriptions of the supply air system were given in Ref. 18. The settling chamber and the subsonic approach to the suction slot are similar to those used previously for the Mach 5 axisymmetric, rapid expansion, pilot quiet nozzle[13-15,18] except for the length of the downstream section of the settling chamber that was reduced from 154.3 to 108.4 cm. A 1-μ filter has also been installed upstream of the settling chamber.

A detailed drawing of the settling chamber is shown in Fig. 1b. The upstream section of the settling chamber contains a porous entrance cone (Rigimesh), a perforated plate, a section filled with steel wool, a porous plate (Rigimesh), and seven screens. The steel wool and porous components were used as acoustic baffles to attenuate upstream pipe/valve noise. The rms fluctuating pressure levels measured by pressure transducers mounted flush with the wall just upstream of the nozzle entrance were less than 0.004% of stagnation pressure.[13,17] The turbulent screens were used to reduce vorticity disturbances. The streamwise velocity fluctuations measured with a hot wire upstream of the nozzle entrance were between 0.2–0.4%.[13,17] Therefore, good flow quality is assured before the nozzle entrance.

III. Nozzle Configurations

The advanced Mach 6 axisymmetric quiet nozzle was designed by the same procedures used previously for the advanced Mach 3.5 axisymmetric quiet nozzle. Detailed descriptions of the design procedures and the boundary-layer stability analysis were given in Ref. 25. This new hypersonic quiet nozzle was fabricated by electroforming nickel onto a stainless steel mandrel. Figure 2 shows the wall contours and Mach lines for different flow regions in the nozzle. The throat diameter is 2.54 cm, and the nozzle exit diameter is 19.03 cm. The nozzle length from throat to exit is 100.99 cm. The radial flow region with its straight line wall section inclined at 9.84 deg is identified. The beginning of the concave curvature along the nozzle wall is located at the upstream point of origin of the Mach line that terminates on the centerline at X_o. This point X_o is the upstream tip of the uniform flow test rhombus that is located at $X = 46.60$ cm from the nozzle throat. Since the Görtler vortices can only form along the concave wall, their onset is delayed until the end of the straight wall section is reached. The amplification rates of the vortices are then

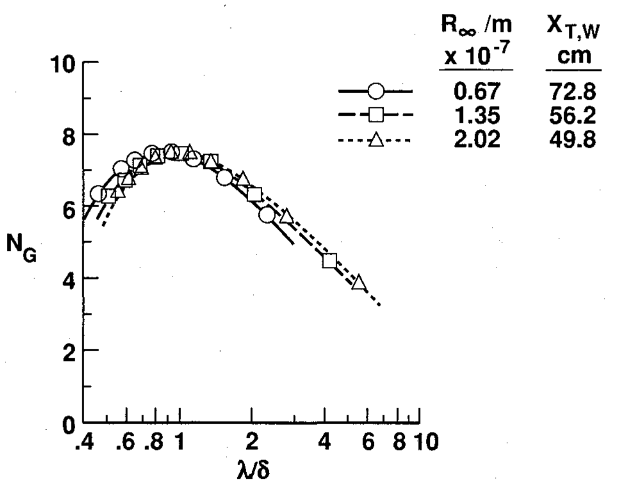

Fig. 3 Calculated variation of N_G factor with wavelength of Görtler vortices at predicated locations of transition.

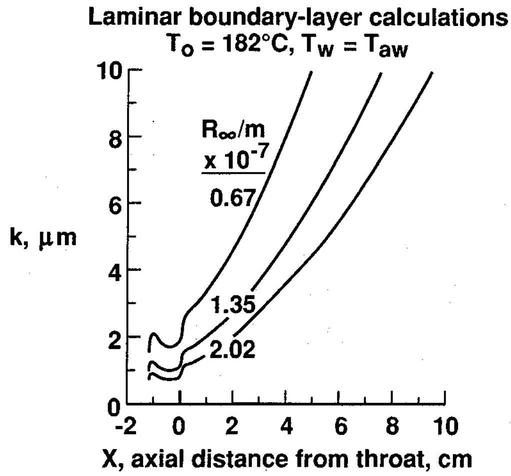

Fig. 4 Allowable surface roughness variations for $R_k = 10$.

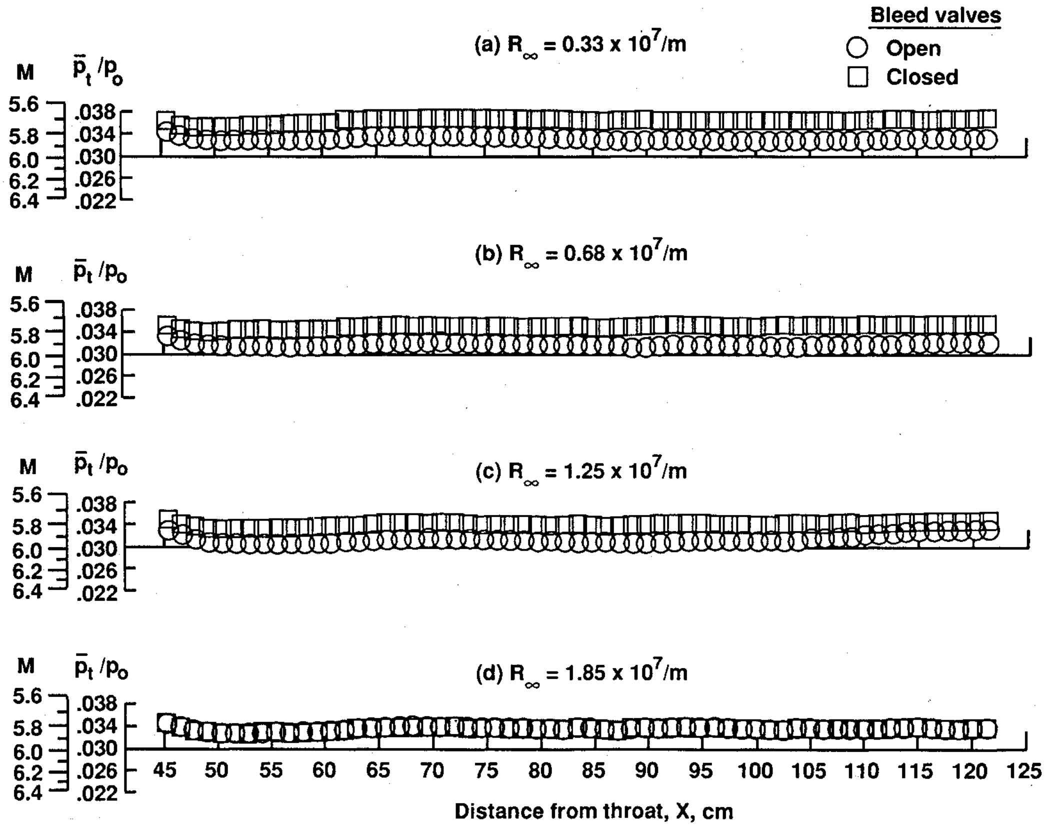

Fig. 5 Mean pitot pressure distributions along the centerline ($r = 0$).

reduced as compared with those in the Mach 5 axisymmetric, rapid expansion, pilot quiet nozzle[18] due to the larger radii of curvature along the concave wall region of the new advanced quiet nozzle. The boundary-layer transition points due to the growth of Görtler vortices as predicted by the e^N method for three different unit Reynolds numbers are listed in the figure. In these calculations, $N_G = 7.5$ has been used rather than the previous value[25] of $N_G = 9$ to give better agreement with current data to be presented in the following sections. The predicted locations of transition on the nozzle wall are denoted by $X_{T,W}$. The Mach lines that originate at $X_{T,W}$ intersect the centerline at $X_{T,C}$. The resulting predicted axial lengths ΔX of the quiet regions are then the difference between $X_{T,C}$ and X_o. Also listed in the figure are the Reynolds numbers, $R_{\Delta X}$, based on freestream conditions and the lengths, ΔX. The values of $R_{\Delta X}$ are useful in evaluating the performance of quiet nozzles. To assess the possible contributions of Tollmien-Schlichting waves, their growth has also been computed. The corresponding largest amplification of Tollmien-Schlichting waves occurs at the highest unit Reynold number where N_{TS} is less than 3.5 at the predicted transition point. This result confirms the dominance of Görtler instability in the nozzle wall boundary-layer transition.

Figure 3 shows the calculated variation of the N_G factor with the normalized wavelength of Görtler vortices at the predicted locations of transition on the nozzle wall. The computational criterion used for the streamwise development of the vortices was constant number of waves around the nozzle periphery ($n = 2\pi r/\lambda$). Therefore, for a given value of n, the Görtler wavelength increases with downstream distance along the nozzle wall. The calculated N_G factors at the transition locations are correlated with the ratio of Görtler wavelength to boundary-layer thickness, λ/δ, over a range of computed flow conditions. As shown in the figure, the specified peak value of N_G occurs within a narrow wavelength range. Similar results were obtained in both the Mach 3 and 3.5, rapid expansion, pilot quiet nozzles[20,21] and the Mach 3.5, slow expansion, advanced quiet nozzle.[25] These results indicate that the boundary layer along the nozzle wall selects a narrow range of vortex sizes that presumably match the mechanisms and boundary-layer thickness required for maximum integrated amplification of the vortices at the transition location.

As a guide for the surface finish specifications on the nozzle wall, values of k (roughness height) were computed from the same laminar boundary-layer solutions used for stability computations. The value of roughness Reynolds number, $R_k = 10$, was chosen as the criterion of allowable surface roughness based on experience with the rapid expansion, pilot quiet nozzles.[23] The allowable surface roughness k at each axial station X was then obtained by interpolating the normal distance from the wall in the local Reynolds number profile for $R_k = 10$. Results are shown in Fig. 4. With this criterion, the maximum allowable peak-to-valley roughness for the new nozzle at the highest unit Reynolds number is about 0.8 μ in the throat region. The effect of surface finish on the nozzle performance is discussed in Sec. V.

IV. Experimental Results

The boundary-layer bleed flow into the suction slot upstream of the nozzle throat (see Figs. 1 and 2) is controlled by four on-off valves in four bleed pipes located downstream of the bleed plenum. A quiet test core in the upstream region of the test rhombus can only be obtained when the bleed valves are open. When the bleed valves are closed, the turbulent boundary layer in the subsonic approach spills around the slot leading edge. The entire boundary layer on the nozzle wall is then turbulent, and no quiet test core exists within the test rhombus.

The nozzle flow and noise in the test rhombus were measured with the bleed valves open and closed. The mean flow data were obtained with the pitot pressure probe. Figure 5 shows the axial variations of mean pitot pressure normalized by stagnation pressure $\bar{p}_t/p_0$ and the corresponding isentropic Mach number M on the centerline ($r = 0$) for four different freestream unit Reynolds numbers. The levels of $\bar{p}_t/p_0$ are higher with the bleed valves closed than those with the bleed valves open when the freestream unit Reynolds number R_∞ is low (see Figs. 5a and 5b). This result shows that the boundary layer on the nozzle wall is thicker (in other words, turbulent) when the bleed valves are closed. When the nozzle wall boundary layer undergoes transition from laminar to turbulent with the bleed valves open at higher freestream unit Reynolds number R_∞, the boundary layer becomes thicker so that the levels of $\bar{p}_t/p_o$ increase and approach those values with the bleed valves closed for $X > 100$ cm (see Fig. 5c). When the nozzle wall boundary layer is fully turbulent with bleed valves open at even higher freestream unit Reynolds number R_∞, the levels of $\bar{p}_t/p_0$ coincide with those values with the bleed valves closed (see Fig. 5d). Although these mean pitot pressure data do provide an indication of boundary-layer transition, the hot-wire data that will be presented later are more accurate for this purpose. The increase in the mean pitot pressure data at the upstream end of the probe survey range occurs for survey stations that are outside of the uniform flow test rhombus (see Fig. 2). Generally, this nozzle exhibits excellent flow uniformity and essentially no centerline focusing of mean flow disturbances due to the wall waviness criteria of 0.002 mm/cm that was required during the machining of the mandrel.

The dominant unsteady disturbances in the freestream of hypersonic nozzles are pressure flucturations caused by weak shock waves radiating from moving eddies in the nozzle wall boundary layer. The magnitude of the pressure fluctuations in the uniform flow test rhombus can be determined by mode diagram analysis of measured hot-wire signals.[6] Figures 6a–6d show typical variations along the nozzle centerline of rms static pressure fluctuations normalized by the mean static pressure $(\tilde{p}/\bar{p})_\infty$ obtained from constant current hot-wire anemometer data. In the quiet region, the measured hot-wire

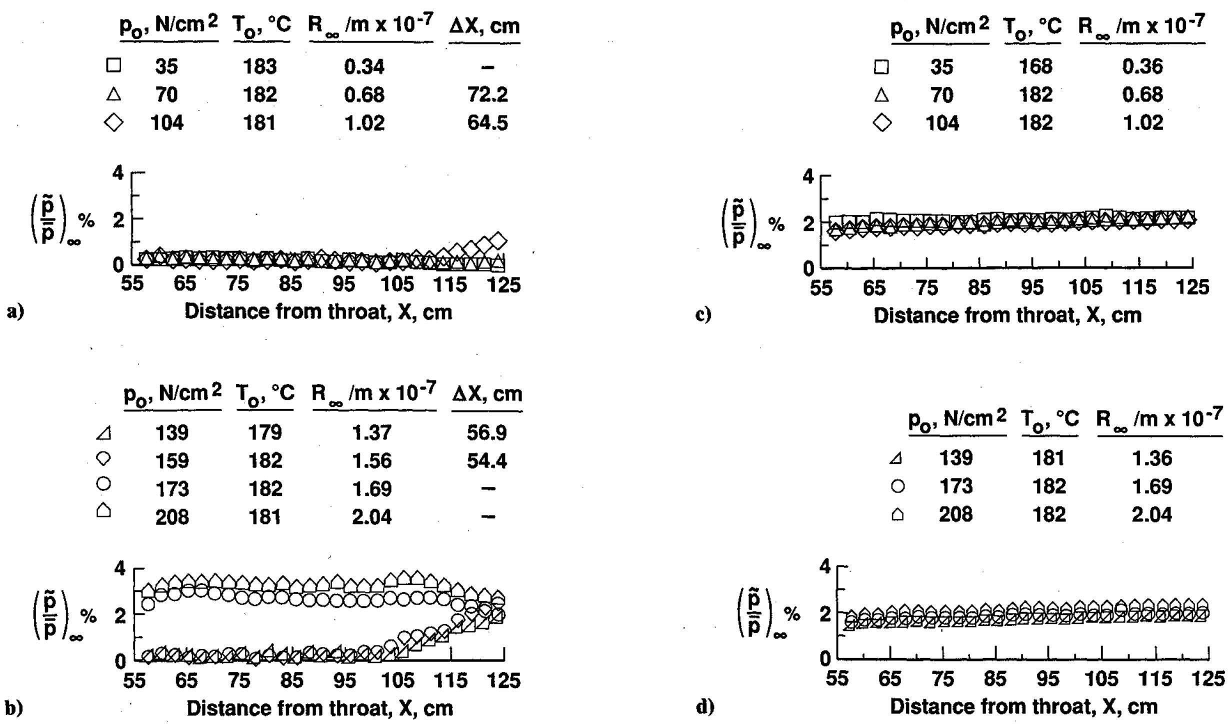

Fig. 6 Variation along the centerline ($r = 0$) of the normalized rms static pressure: a) quiet flow with bleed valves open, b) quiet/noisy flow with bleed valves open, c) noisy flow with bleed valves closed, and d) noisy flow with bleed valves closed.

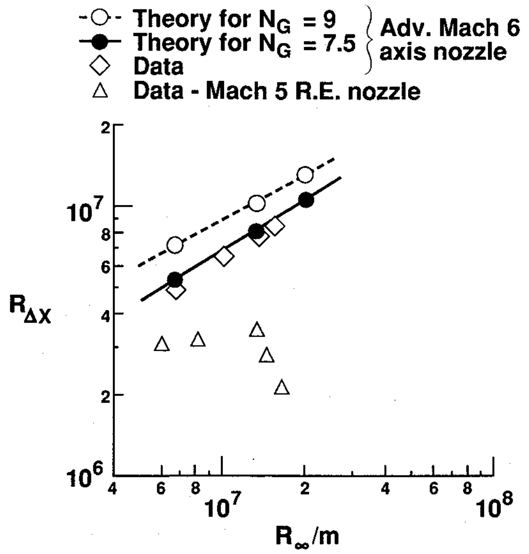

Fig. 7 Quiet test core length Reynolds numbers.

signals were very close to the instrument noise levels. The location where disturbance levels begin to increase is defined as the onset of transition, denoted as $X_{T,C}$ in Fig. 2. The measured lengths ΔX of the quiet test core are again defined as $X_{T,C}-X_o$. In Figs. 6a and 6b, values of ΔX are listed for the low test range of R_∞ with the bleed valves open. For the lowest test Reynolds number, the value of ΔX was not obtained because the point $X_{T,C}$ was downstream of the hot-wire probe survey mechanism range. The measured values of ΔX decrease when the unit Reynolds number increases. Hence, the transition point on the nozzle wall moves upstream with increasing unit Reynolds number and the length of the quiet test core is reduced. For $R_\infty > 1.56 \times 10^7$/m, the quiet region in the test rhombus is lost completely as shown in Fig. 6b. The loss of quiet flow occurred suddently with just a slight increase of p_o over 159 N/cm^2. This rapid upstream movement of transition and stronger disturbance levels than the fully turbulent cases with the bleed valves closed, as shown in Figs. 6c and 6d, suggest a roughness-triggered transition bypass. Similar behavior was observed in the advanced Mach 3.5 axisymmetric quiet nozzle.[25]

V. Nozzle Performance

The performance of a quiet nozzle is characterized by the length of the quiet test core at useful test Reynolds numbers. Therefore, the freestream Reynolds number $R_{\Delta X}$, based on the length of the quiet test core ΔX, provides the best criterion for comparison of different quiet nozzles. Figure 7 shows $R_{\Delta X}$ from test data for the Mach 5 axisymmetric, rapid expansion, pilot quiet nozzle[18] and for the present advanced Mach 6 axisymmetric quiet nozzle over the test range of unit Reynolds numbers R_∞. Also, the predicted values of $R_{\Delta X}$ for $N_G = 7.5$ (from Fig. 2) and 9.0 for the present nozzle are included for comparison with the experimental reuslts.

The data for the present nozzle, as shown in Fig. 7, indicate an increasing trend of the quiet test core length Reynolds number $R_{\Delta X}$ with the unit Reynolds number R_∞. The effect of unit Reynolds number on $R_{\Delta X}$ is caused by the increasing local favorable pressure gradients that damp the growth of Görtler vortices as transition moves upstream along the nozzle wall with increasing R_∞. The values of $R_{\Delta X}$ obtained from the present nozzle are in good agreement with the theoretical predictions for $N_G = 7.5$ instead of 9, which was previously used as a transition criterion for the advanced supersonic quiet nozzle.[25] The discrepancy between the data for the present nozzle and the theoretical predictions for $N_G = 9$ may be attributed to the early transition promoted by the surface roughness as that observed in the Mach 3.5 two-dimensional pilot quiet nozzle.[25] Data from the present nozzle show the complete loss of any quiet test core for $R_\infty > 1.56 \times 10^7$/m. This is apparently caused by the transition bypass triggered by surface roughness. A microphotographic examination of the surface finish indicated the existence of many scratches and pits on the nozzle surface in the visually accessible region upstream of the throat region. The largest surface roughness defect in this region was estimated to be $k \simeq 7\ \mu$. This value of k is much greater than the maximum allowable surface roughness as shown in Fig. 4. With further improvement of the nozzle surface finish, especially in the critical throat region, it is expected that the nozzle performance will show better agreement with the theoretical predictions for $N_G = 9$ and reach higher values of $R_{\Delta X}$ in the high unit Reynolds number range. Except for the problem of surface finish, values of $R_{\Delta X}$ for the present nozzle are much higher than those for the Mach 5 axisymmetric, rapid expansion, pilot quiet nozzle. This improved performance verifies that the new design concept of the advanced quiet nozzle is feasible in the hypersonic speed range.

VI. Conclusions

The development of high-speed quiet tunnels is essentially a laminar flow control problem. Several techniques have been developed and combined for controlling and increasing the extent of laminar boundary-layer flow on the nozzle wall. The boundary-layer suction slot upstream of the nozzle throat is necessary to remove upstream turbulent boundary layers and initialize a new boundary layer on the downstream nozzle wall. The laminar boundary-layer flow can then be extended more effectively farther downstream by a slow expansion contour based on the new design concept than by a rapid expansion contour used in the previous pilot quiet nozzles. This new slow expansion contour is obtained by inserting a radial-flow, straight-wall section upstream of the inflection point. Therefore, the initiation of Görtler vortices is delayed until the beginning of the concave wall and the growth of Görtler vortices is decreased by the larger radii of curvature of the concave wall. This new design concept greatly improves the performance of quiet nozzles not only in the supersonic speed range but also in the hypersonic speed range.

The advanced Mach 6 axisymmetric quiet nozzle has verified the application of the new design concept of advanced quiet nozzles to the hypersonic speed range; consequently, transition research in ground facilities that simulate flight conditions can be extended to higher Mach numbers.

A high-quality nozzle surface finish is essential, especially in the throat region. Surface roughness defects can 1) promote early transition and degrade nozzle performance in the low unit Reynolds number range and 2) trigger transition bypass and completely eliminate the quiet test core in the high unit Reynolds number range. The requirements of surface finish in the critical throat region become more stringent for higher Mach number with increasing unit Reynolds numbers and thinner boundary layers in smaller nozzles.

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