

Development of Hypersonic Quiet Tunnels

Steven P. Schneider
Purdue University,
West Lafayette, Indiana 47907-1282

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Nomenclature

a	=	speed of sound
e^N	=	integrated amplification ratio for a linear instability, usually from location of first instability to transition
k	=	roughness height
M_w	=	Mach number at the nozzle wall, at the boundary-layer edge
M_∞	=	Mach number in the freestream
N factor	=	$l_n A/A_0$, the linear-theory amplification ratio from onset of instability A_0 usually to measured transition A
P_{mean}	=	mean pitot pressure
P_t	=	stagnation or total pressure
P'	=	rms pitot-pressure fluctuations
\bar{p}	=	mean static pressure
\bar{p}'	=	rms static-pressure fluctuations
Re_e	=	Reynolds number based on conditions at the boundary-layer edge
Re_k	=	Reynolds number based on roughness height k and local conditions in the undisturbed laminar boundary layer at the height k
$Re_{\Delta x}$	=	quiet length Reynolds number based on freestream conditions and Δx
Re_θ	=	Reynolds number at transition (usually onset) based on momentum thickness and conditions at the boundary-layer edge
Re_∞	=	unit Reynolds number in the freestream; units given where used
U	=	freestream velocity
x	=	axial coordinate for 2-D nozzles, usually from the nozzle throat
y	=	vertical coordinate, in the curvature plane for 2-D nozzles
z	=	for 2-D nozzles, a horizontal coordinate perpendicular to the flat side walls for axisymmetric nozzles, an axial coordinate from the nozzle throat
Δs	=	length along cone from the nosetip to the location in which nozzle wall radiated noise reaches the cone
Δx	=	length along nozzle centerline from the onset of uniform flow to the location in which nozzle-wall radiated noise reaches the centerline

Introduction

CONVENTIONAL hypersonic wind tunnels and shock tunnels suffer from high levels of freestream fluctuations, which are typically 1 to 2 orders of magnitude above flight levels. These freestream fluctuations are generally dominated by acoustic noise radiated from the turbulent boundary layers on the nozzle walls. Although the effects of the noise are often small, and so can be neglected, this noise often has a dramatic effect on laminar-turbulent transition on models, and it can have a significant effect on other phenomena as well.

Quiet-flow wind tunnels provide uniform flow at supersonic and hypersonic speeds with low noise levels comparable to flight. They have been sought for more than 50 years [1]. Low-noise subsonic tunnels were essential to the discovery of the Tollmien–Schlichting waves that lead to low-speed transition on flat plates and many airfoils [2,3]. Low turbulence in subsonic tunnels is often essential to achieving flow conditions representative of flight. Transition is one factor that affects scaling from ground to flight, and tunnel noise was long known to be important for supersonic tunnels also [4]. Thus, the development of low-noise tunnels at supersonic and hypersonic speeds was a natural extension of earlier work.

However, it has been much more difficult to develop comparable low-noise facilities at high speeds, for four main reasons. First, supersonic and hypersonic tunnels are much more expensive to build, modify, and operate, and so the inevitable resource limitations make progress much more difficult. Second, instrumentation for measurement of freestream fluctuations is also much more difficult and expensive at high speeds, due to the high pressures, high temperatures, and high disturbance frequencies associated with high-speed flow. Third, the dominant source of noise in most high-speed tunnels turns out to be acoustic waves radiated from the turbulent boundary layers on the nozzle walls, a phenomena that has proven to be very difficult to control. Fourth, the need for hypersonic quiet flow is obvious only for certain vehicle designs that depend heavily on the location of transition, and the market for quiet tunnels is therefore often assessed as too small to justify the investment.

A visual example of the noise radiated from a turbulent boundary layer is shown in Fig. 1, which shows a magnified portion of a shadowgraph obtained in the Naval Ordnance Lab ballistics range [5]. The sharp cone model is flying at Mach 4.3 near zero angle of attack, at a freestream Reynolds number of $3.2 \times 10^7 \text{ ft}^{-1}$



Steve Schneider received his B.S. from the California Institute of Technology (Caltech) in 1981. From 1981 to 1983, he worked at the Naval Ocean Systems Center in San Diego. Steve returned to Caltech in 1983, receiving a Ph.D. in aeronautics in 1989 under H. W. Liepmann and D. Coles. His thesis involved experimental measurements of low-speed laminar instability and transition. He joined Purdue University as an assistant professor in July 1989. Since then, he has focused on high-speed boundary-layer transition, developing facilities and instrumentation for detailed measurements under quiet-flow conditions. A \$1 million 9.5-in. Mach-6 quiet-flow Ludwig tube was completed in 2001 and achieved high Reynolds number quiet flow in 2006. He was promoted to professor, School of Aeronautics and Astronautics, in 2004, and has written seven review articles on hypersonic transition. He is an Associate Fellow of AIAA.

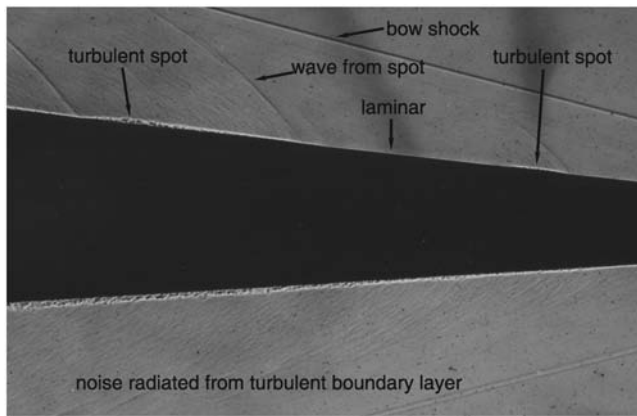


Fig. 1 Shadowgraph of transition on a sharp cone at Mach 4.31.

($1.05 \times 10^8 \text{ m}^{-1}$) [6,7]. The cone is traveling from left to right through still air, and the axial extent of Fig. 1 is about 2.9 in. (74 mm). The lower surface boundary layer is turbulent, and acoustic waves radiated from the turbulent eddies can be seen passing downstream at the Mach angle. This Mach angle is set by the flow speed minus the velocity of the boundary-layer disturbances that generate the acoustic waves. On the upper surface, the boundary layer is intermittently turbulent, with two turbulent spots being visible in the image, interspersed among laminar regions. Larger waves can be seen in front of the turbulent spots, with smaller levels of acoustic noise being radiated from the turbulence within the spots. The acoustic noise is not present above the laminar regions. Thus, it has been necessary to *control* laminar-turbulent transition on the nozzle walls to develop facilities for *measuring* laminar-turbulent transition on models under flight-comparable conditions.

The successful development of quiet tunnels, therefore, took many years, led primarily by Ivan Beckwith of the NASA Langley Research Center, who began a major effort in the late 1960s, which continued into the early 1990s [8]. Because the present author began quiet-tunnel research in late 1989, he was fortunate to be able to work with Ivan for several years to gain much of the benefit of his experience. The present author was then able to develop a low-Reynolds-number Mach-4 quiet Ludwig tube followed by a recently successful high-Reynolds-number Mach-6 quiet Ludwig tube.

The present paper reviews the development of these facilities beginning with early recognition of the need. The scope of the paper is limited to the facilities themselves and the need for them. Although the measurements in these facilities are obviously the reason for developing them, these measurements are described only on occasion, in passing, because the paper is already a long one. It is assumed that the reader is already familiar with the general area of hypersonic instability and transition (for example, see [9–13]). In particular, the reader is assumed to be familiar with the semiempirical e^N method for estimating transition onset by using the linear amplification of a given instability wave by a factor of e^N from the beginning of instability to the observed or predicted transition location [14].

The paper is focused on quiet tunnels for hypersonic and high-supersonic speeds, although tunnel noise issues at low-supersonic speeds are also discussed. Even with this limited scope, the literature in this area is substantial. Although the author has accumulated and studied this literature during 18 years of focused effort, this review is certainly incomplete, and the author would appreciate hearing of errors and omissions.

Background

Sources of Noise in Supersonic and Hypersonic Wind Tunnels

In 1953, Kovasznay showed that small fluctuations in a viscous compressible flow can be analyzed in terms of sound waves

combined with oscillations in vorticity and entropy [15]. Kovasznay also records progress in using hot wires to measure fluctuations in supersonic wind tunnels. However, the paper is concerned with turbulent boundary layers and not with freestream fluctuations in wind tunnels, and so there is no discussion of tunnel noise effects or transition.

During the 1950s and early 1960s, Laufer et al. led a series of studies of supersonic tunnel noise using the wind tunnels at the Jet Propulsion Laboratory (JPL) in Pasadena. In 1954, Laufer varied the fluctuation level in the settling chamber of the JPL 20 in. (0.51 m) tunnel using a grid. He showed that transition on a sharp cone in the test section was independent of the settling-chamber fluctuations for freestream Mach numbers above 2.5 [1]. This tunnel could run at various Mach numbers by adjusting the shape of the flexible-plate walls, which proved to be a very useful property for transition research. This was the first evidence of a new source of test-section fluctuations in supersonic tunnels.

This work is described in more detail in [16], in which Laufer comments, “Unfortunately at the present time no adequate experimental method is available which permits a direct determination of turbulence level in the supersonic test section” ([16], p. 5). This problem with high-frequency instrumentation slowed progress for several years until the supersonic hot-wire technique was developed to provide an adequate method. Laufer attributed the high-Mach insensitivity of transition to settling-chamber noise to one of two factors: 1) the high contraction ratio at high Mach numbers, which tends to reduce propagation of settling-chamber noise and dissipate vorticity through stretching and viscosity, or 2) the generation of acoustic noise from the turbulent boundary layer on the nozzle wall.

Morkovin also treated the problem of supersonic tunnel disturbances in two papers from the middle 1950s, both developed from experience with measurements near Mach 1.76. Morkovin [17] discusses various sources of freestream fluctuations as shown in Fig. 2, redrawn and updated from his Fig. 3. However, he does not appear to realize the importance of sound radiated from the boundary layer on the nozzle wall, perhaps because it does not dominate at this low Mach number. Morkovin [18] again reviews sources of freestream fluctuations and now recognizes the key role of acoustic fluctuations. These are generated in two ways: 1) from quadrupole and dipole radiation from the turbulent boundary layer, and 2) from shivering Mach waves, which are Mach waves generated at flaws in the nozzle contour, which fluctuate in position as turbulence passes over the flaws. Most of the discussion is qualitative, probably because Morkovin could only operate at a single Mach number in the low supersonic region and was not able to run at very low pressures with laminar boundary layers on the nozzle walls.

By 1960, Laufer and Vrebalovich had completed a study of first-mode wave amplification on flat plates in the JPL tunnel, using hot wires and a pulsed air jet through a spanwise slot [19]. Their Fig. 2 showed freestream mass-flow fluctuations increasing rapidly with Mach number, due to noise radiated from the turbulent boundary layer on the nozzle wall. Instability measurements were only carried out below Mach 2.5, where instability-wave oscillations induced by the “natural” disturbances could be detected reliably.

The measurements of noise radiated from the nozzle-wall boundary layer are described in more detail in [20,21]. Laufer measured the sound intensity radiated from the nozzle wall using hot wires and found that the rms fluctuation in mass flow normalized by the mean was approximately proportional to the fourth power of the freestream Mach number [20]. He also reported the first supersonic measurements under quiet flow conditions with laminar nozzle-wall boundary layers, at Mach 4.5 and unit Reynolds numbers below $3.1 \times 10^5 \text{ ft}^{-1}$, where the freestream mass-flow fluctuations decreased by an order of magnitude to about 0.11% ([20], p. 692).

Laufer et al. reviewed these measurements in [22], along with the relevant theory. They give a simple explanation due to Hans W. Liepmann that attributes the noise to the supersonic streamwise motion of Mach waves generated by the irregular streamwise variation of the displacement thickness in the nozzle-wall boundary layer.

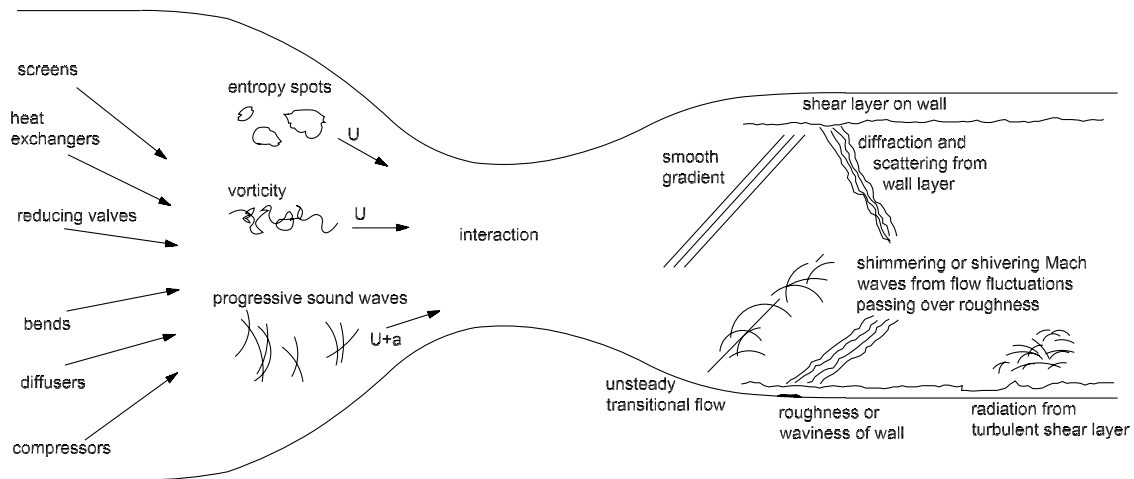


Fig. 2 Freestream disturbances in supersonic wind tunnels.

Need for Hypersonic Quiet Tunnels

Beckwith reviewed the need for high-speed quiet tunnels in several reports, the first of which is [23]. Tunnel noise is shown to have an effect both on laminar-turbulent transition and on the pressure fluctuations under a turbulent boundary layer.

The effects of tunnel noise on supersonic and hypersonic transition were previously reviewed in [24], which discusses fluctuation measurements in flight and in wind tunnels and the effect on transition for various configurations. The present section thus reports only additional information not present in [24].

Morkovin reviewed transition at high speeds in [25] (pp. 55–57). One of the four “open questions” he raises is regarding the effect of sound in the wind tunnels. He reviews various measurements that are discussed in [24] and leaves the question open, as it remains.

Laderman reviewed measurements of pressure fluctuations in various wind tunnels [26]. He discusses the effect on transition but does not show any transition data.

Beckwith discusses the effect of tunnel noise on the pressure fluctuations under a turbulent boundary layer [27] (see also [28]). The tunnel noise can dominate the pressure fluctuations measured at the wall for frequencies below perhaps 20 kHz that are relevant for panel flutter. This issue has received very little attention in more than 30 years, remaining an open question.

Harvey and Bobbitt reviewed the effect of tunnel noise on transition at transonic and supersonic speeds [29]. The length of the transitional zone that is measured in flight is smaller than the length measured in conventional wind tunnels.

Schopper made a detailed study of the eddy-Mach-wave radiation from the turbulent boundary layer on the nozzle wall and of its effect on the laminar boundary layers on models [30,31]. The laminar boundary layer is “strafed” by “miniature sonic booms” generated from coherent turbulent eddies. The waves are focused within the outer part of the laminar boundary layer.

Early Work Toward Development of Quiet-Flow Tunnels

James M. Kendall Jr. worked on supersonic and hypersonic instability and transition for many years during a long career at JPL. Unfortunately, much of this work was recorded only in internal JPL reports that are difficult to obtain.

In 1962, Kendall made the first (and only) measurements of supersonic wake instability under quiet conditions, operating in the JPL supersonic tunnel at Mach 3.7 [32]. At freestream unit Reynolds numbers of $2.3\text{--}3.4 \times 10^5 \text{ ft}^{-1}$ ($7.5\text{--}11.2 \times 10^5 \text{ m}^{-1}$), the boundary layer on the nozzle wall was laminar, and the tunnel was still operable. Hot wires were used to measure the instabilities behind cylinders and spheres.

In 1967, Kendall reported measurements of first- and second-order instability waves on a flat plate in the same tunnel at Mach 4.5 with additional measurements at Mach 3.7 and 2.4 [33]. The waves

were introduced with a glow-discharge perturber, and the tunnel was operated at low pressures where the tunnel-wall boundary layers were laminar. This very short report does not provide any details on the tunnel conditions. Some further detail regarding this work is reported in [34]; however, there is no additional detail regarding the tunnel performance. Additional measurements reported in [35] appear to have been obtained under conventional-noise conditions.

Kendall summarized this work in [36,37], providing substantial additional information. When the JPL tunnel was operated at low quiet pressures at Mach 4.5, transition did not occur on a flat plate whose length Reynolds number was 3.3×10^6 . This is much later than the 1.0×10^6 -length Reynolds number for which transition occurred under conventional-noise conditions. The only data obtained under quiet conditions showed a damped hot-wire response to impact vibrations that disappeared into the noise under conventional-tunnel conditions ([36], p. 8). Kendall apparently coined the term “quiet-flow tunnel” at a Transition Study Group meeting at Case Western Reserve University in the early 1970s.*

Phil Klebanoff of the National Bureau of Standards was one of the first to attempt to develop specially designed supersonic tunnels with low noise levels. His group was apparently funded by NASA for many years to carry out various studies related to transition at subsonic and supersonic speeds. As early as August 1961, Klebanoff et al. reported, “The study of the effect of supersonic wind-tunnel environmental conditions on boundary-layer transition has continued.” Klebanoff et al. pursued a nozzle-wall suction approach, reporting, “The aim is to maintain laminar flow along the walls and thus eliminate the source of the disturbances entirely” ([38], p. 2). This is the earliest report of this goal that is known to the present author.

By 1965, Klebanoff et al. had extended laminar flow on the two curved nozzle walls of his supersonic tunnel from a freestream Reynolds number of $260,000 \text{ in.}^{-1}$ ($1.02 \times 10^7 \text{ m}^{-1}$) at Mach 1.5 (without control) to $570,000 \text{ in.}^{-1}$ ($2.24 \times 10^7 \text{ m}^{-1}$) at Mach 1.8 (with suction control) [39]. Unfortunately, this effort never became very successful, and Klebanoff et al. never provided a substantial summary of it. The last known descriptions of this effort appear in 1975 ([40], pp. 263–264; [27], p. 303). It appears that Klebanoff et al. originated the concept of a lateral suction slot in the subsonic region upstream of the throat ([27], p. 303). Eli Reshotko recalled this work from early meetings of the Transition Study Group and believes Klebanoff was inspired by Kendall’s earlier measurements in the JPL 20 in. (0.51 m) tunnel under laminar nozzle-wall boundary layers at low Reynolds numbers.†

Reshotko summarized the early activities of the NASA Transition Study Group (TSG) in the lead paper from a special section in the March 1975 issue of the *AIAA Journal* [40]. Beginning in late 1970,

*Kendall, J. M., Jr., private communication, 30 March 2007.

†Reshotko, E., private communication, 30 March 2007.

the TSG met to “develop and implement a program that would do something constructive toward resolving the many observed anomalies in boundary layer transition data” ([40], p. 261). This paper still makes excellent reading after more than 30 years. It includes recommendations for 1) measurements in the JPL supersonic tunnel under quiet conditions at Mach 4.5, and 2) further development of quiet wind tunnels. Reshotko reported that the JPL 20 in. (0.51 m) tunnel ran quiet for unit Reynolds numbers of $4.8\text{--}7.2 \times 10^5 \text{ ft}^{-1}$ ($1.9\text{--}2.8 \times 10^7 \text{ m}^{-1}$) at Mach 4.5 ([40], p. 264). When the boundary layers became laminar, the disturbance levels in the JPL tunnel fell from about 1% to about 0.03%.

Brief Summary of Quiet-Tunnel Development at NASA Langley Research Center

A large number of people worked on quiet-tunnel development at NASA Langley Research Center from the late 1960s through the middle 1990s. Ivan Beckwith provided technical leadership within programs led by Dennis Bushnell, while Mujeeb Malik led the development of stability-based computational methods for estimating transition on the nozzle walls. Beckwith was well prepared for this long effort because he combined a theoretical and computational understanding of boundary layers [41] and supersonic nozzle design [42] with an increasing understanding of experimental issues. This very substantial effort resulted in many publications by many authors over three decades. No accounting of the cost was ever made, but it appears that many millions of dollars were expended.‡ Unfortunately, a comprehensive review of this effort has not been provided by NASA Langley personnel, and most of the relevant personnel are now retired and unavailable. The present section is thus only a first attempt at a brief summary of this large body of work.

Early Developments

The earliest record of the quiet-tunnel development program is presented in [43]. However, this reference also refers to classified reentry flights, and even after 30 years, distribution is still restricted to U.S. nationals so it cannot be described further here. Beckwith also provided an excellent early summary of the NASA Langley program in an unpublished internal working paper [44]. However, because NASA has not yet approved this paper for public release, it will not be described in detail.

The first report of the NASA Langley program in the open literature is thus [23]. Cone transition results using a 4-in. (10-cm) Mach-4 nozzle with porous surfaces and area suction were said to be reported in [43], but transition was no later, and even slightly earlier, than in the equivalent solid nozzle (later used at Purdue University [45]). The failure was thought to be due to the large scale of the nozzle porosity. Suction was then pursued using longitudinal slots between rods, following Klebanoff and also the experiments of Groth with suction models in supersonic wind tunnels [46].

In 1973, Beckwith et al. provided an extensive review of their effort, along with previous related work [47]. Via Morkovin, Beckwith knew of four wind-tunnel nozzles that had run with laminar boundary layers, at least to some extent. Besides the JPL tunnel [20], some laminar nozzle-wall boundary layers had also been achieved in the helium tunnel at NASA Langley [48], in the Mach-8 tunnel at the University of Michigan [49], and in [50]. Winkler and Peresh [50] report a 20-in. (0.5 m) run of laminar flow in tunnel 4 of the Naval Ordnance Lab, along 2-D-wedge nozzle blocks that provide Mach 5, at 15 psia (103 kPa) stagnation pressure. Transition onset along the nozzle wall occurred at $Re_\theta \simeq 400$. However, only the JPL tunnel had a significant region of uniform quiet flow that allowed performing experiments on models. Laminar flow in the Michigan tunnel increased by a factor of more than 4 as the nozzle wall heated up, an effect attributed to the reduced influence of roughness in the thicker boundary layers on a heated wall ([49], Fig. 11).

This 1973 paper also investigated possible laminarization due to the strong acceleration in the diverging nozzle, a concept which apparently led to the “rapid-expansion” nozzles used in early NASA Langley Research Center designs [47] but which turned out to be suboptimal. In addition, the paper discusses the use of lateral suction slots to remove the contraction-wall boundary layer upstream of the throat, a feature of nearly every quiet nozzle since that time ([47], pp. 16–18). Methods for the reduction of settling-chamber noise are also reviewed. The possible dominance of the Görtler instability on the concave nozzle wall is described, and Görtler numbers are computed for four nozzles that showed some laminar flow.

The possibility of using porous-wall suction was also reviewed in ([47], pp. 21–22), leading to a detailed study by Pfenninger and Syberg [51]. Suction continues to promise dramatic improvements in quiet-flow performance but suffers from disturbances generated by roughness and nonuniform suction. It appears that NASA Langley abandoned the suction-based effort in the middle 1970s because it was not working very well due to suction nonuniformity. New microperforated materials may enable successful development of a quiet suction nozzle if the many associated technical challenges can be overcome [52].

Beckwith [27] observed Görtler vortices in transition in a Mach-5 axisymmetric nozzle and again introduced a suction slot upstream of the throat to remove the boundary layer from the contraction wall. This paper again reviewed extensive research into a rod-wall sound shield, which was to be installed within a hypersonic nozzle, arguing that it would be useful for noise reduction at higher unit Reynolds numbers where the nozzle-wall boundary layer cannot be maintained laminar.

An axisymmetric Mach-5 nozzle was used in several early quiet-tunnel development studies [53]. Various modifications to the settling chamber and control valve included the installation of screens, steel wool, and Rigimesh® porous plates. Rigimesh is sintered from stainless-steel mesh to form a dense plate with fine pores. However, even though these changes reduced the disturbances in the settling chamber, they had little effect on transition of the boundary layer on the nozzle wall. A highly polished throat was necessary to retain a laminar nozzle-wall boundary layer to high Reynolds numbers; this made it necessary to install an air filter upstream to remove particulate that otherwise damaged the polished throat. A nozzle with a suction slot to remove the contraction-wall boundary layer performed better and more reliably than a conventional nozzle without the slot. Higher throat temperatures increased the run of laminar flow, again presumably due to the reduced sensitivity to roughness associated with the thicker boundary layer. Roughness associated with joints near the throat region of a lathe-turned nozzle was reduced by use of a nickel nozzle electroformed in one piece on a mandrel. The high accuracy, low waviness, and low roughness of the electroformed nozzle provided much more laminar flow and much lower noise. Thus, most of the key elements necessary to the development of quiet tunnels had been discovered by the time of this excellent 1977 paper.

Görtler vortices on the concave walls of two axisymmetric Mach-5 nozzles were examined in detail in [54]. A high-quality surface finish and a high-quality settling chamber were used for the apparatus in which the vortices were visible. One of the nozzles used a rapid-expansion design with a bleed slot upstream of the throat; the other was of conventional design.

Görtler vortices were visualized using oil flow, which showed that the vortices were always present when the flow was laminar. The vortices disappeared from the oil flow when the flow became turbulent. Figure 3, taken from Fig. 7 in [54], was obtained using the conventional nozzle. The upper pair of photos shows the flow in the entrance of the contraction. At the lower Reynolds number, streamwise vortices are apparently generated by the Görtler instability in the concave region of the contraction, persisting for some distance into the convex region downstream. Nozzles with bleed slots just upstream of the throat are used to remove such disturbances and the risk they pose to the maintenance of laminar flow downstream. The lower pair of photos shows the flow near the nozzle exit. The streaks present in the laminar nozzle-wall boundary

‡Wilkinson, S., NASA Langley Research Center, private communication, May 2007.

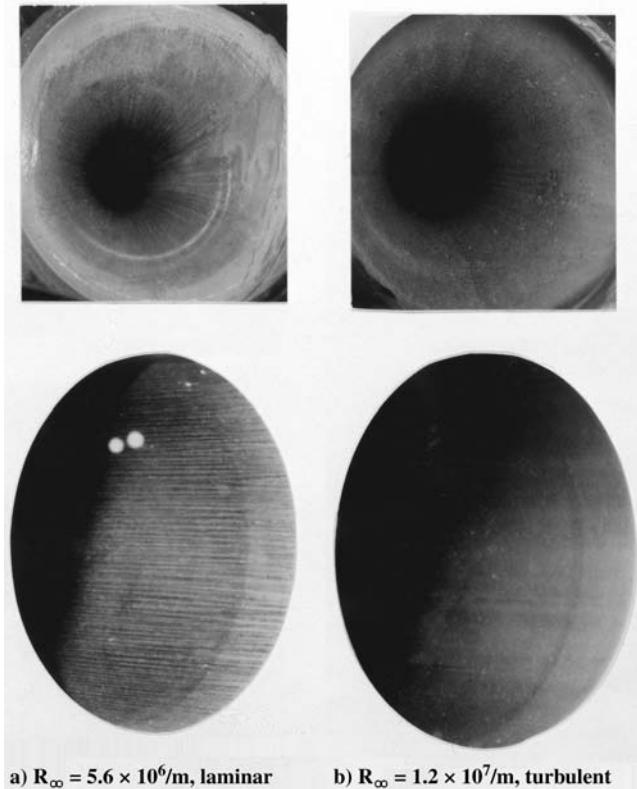


Fig. 3 Oil-flow streaks in contraction entrance and nozzle exit of axisymmetric Mach-5 quiet tunnel.

layer at the lower Reynolds number were clearly associated with the Görtler instability in the concave portion of the supersonic nozzle.

Figure 4, taken from Fig. 9 in [54], shows that the circumferential pattern of the vortices sometimes changed, apparently depending on small variations in the pattern of dust on the nozzle wall. Such very weak sources of streamwise vorticity have been shown to be critical for streamwise-vortex instabilities such as 1) stationary crossflow [55], 2) Görtler [56], and 3) the breakdown of Tollmien–Schlichting waves [57]. Transition onset occurred at Görtler numbers of 5–6. Görtler N factors at transition varied over a large range, from 4–15, perhaps because the boundary-layer profiles were approximated using flat-plate theory.

Anders et al. reported additional measurements in the same pair of axisymmetric Mach-5 nozzles in 1980 [58]. Disturbances in the mean flow were traced to 0.0005-in. (13- μ m) axisymmetric flaws in the contour. These flaws generated Mach waves that focused on the centerline, disturbing the mean flow and generating shimmering Mach waves. These problems led toward work with two-dimensional nozzles. Anders et al. also suggest the use of nozzles with turbulent boundary layers that have low edge Mach numbers and consequent low-noise radiation at the acoustic origin for the model. Although this approach drove the use of rapid-expansion nozzles in later designs, it was dropped eventually because it proved feasible and more effective to eliminate the turbulent boundary layer completely.

Mach-3.5 Quiet Tunnel

The first successful quiet tunnel was then built using a pair of Mach-3.5 two-dimensional nozzle blocks with a 6 by 10 in. (0.15 by 0.25 m) exit [59]. Figure 5, taken from Fig. 3 in [60], shows a schematic of the nozzle, with Mach lines showing how nozzle-wall radiation affects the flow within the test core. A planar section perpendicular to the curved blocks is shown above the centerline, whereas a section perpendicular to the flat sidewalls is shown below the centerline. The “acoustic origin” for a point in the test core (call it “A”) is obtained by tracing a Mach line upstream to the farthest downstream location on the nozzle wall that can radiate sound onto

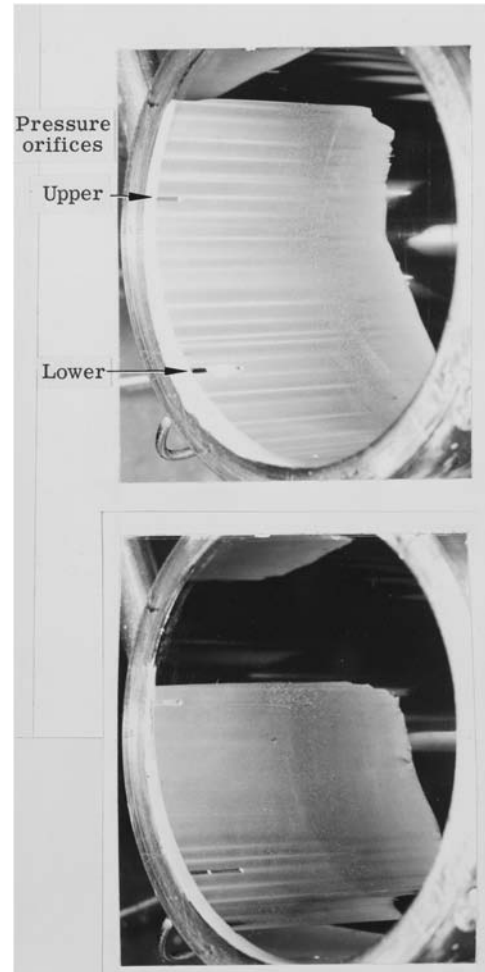


Fig. 4 Oil-flow streaks near the nozzle exit of the slotted Mach-5 quiet tunnel for two different runs.

point A. The curved blocks have a bleed slot ahead of the throat. The flat sidewalls are placed far enough away so that high noise levels radiated from boundary layers with an edge Mach number above perhaps 2.4 do not affect the quiet test core ([60], p. 10).

The critical performance parameter for a quiet tunnel is the Reynolds number based on freestream conditions and the length along the centerline of the region of uniform quiet flow Δx . In Fig. 5, the uniform quiet region begins on the centerline near $x = 5.5$ in. (0.14 m) and ends near $x = 12$ in. (0.30 m), if the onset of transition and radiated noise is taken along the Mach line beginning at $M_w = 2.75$. For an axisymmetric nozzle, this quiet uniform region forms a pair of back-to-back cones, bounded on the upstream end by the beginning of uniform flow and bounded on the downstream end by noise radiated along Mach lines from a nominally symmetric nozzle-wall transition point. The downstream portion of this quiet uniform region is normally of lesser value except for very slender models.

A new settling chamber had a diameter of 2 ft (0.61 m) and a length of 21 ft (6.4 m) and contained acoustic baffles such as steel wool, several dense porous plates that also served as acoustic baffles, and seven screens [60]. The air entering the settling chamber was filtered to remove 99% of all particles larger than 1 μ m. Although the original set of nozzle blocks for this facility suffered from excessive roughness and waviness, they nevertheless provided the first substantial quiet-flow Reynolds numbers. Beckwith and Moore [59] provide the first reported results from this tunnel, measured using high-frequency pressure transducers operated as pitot sensors.

A more complete report on the performance of this tunnel was then provided in [60]. Figure 6, taken from Fig. 2b in [60], shows a photograph of the boundary-layer bleed slots for the side walls and

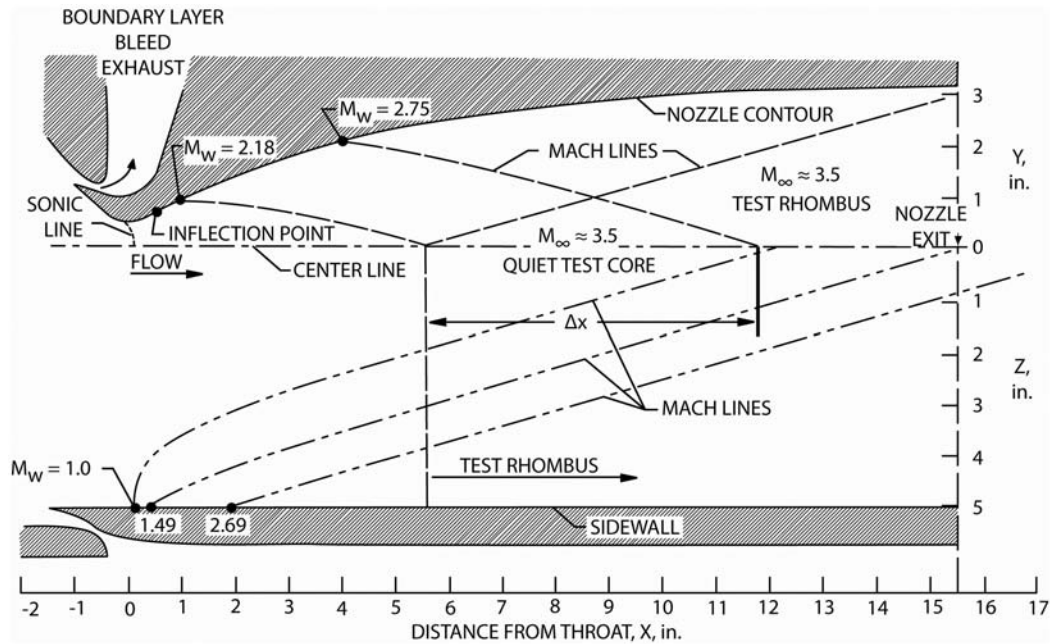


Fig. 5 Nozzle wall cross sections and Mach lines in vertical and horizontal centerplanes of Mach-3.5 quiet nozzle.



Fig. 6 Bleed slots in subsonic approach to throat of Mach-3.5 quiet nozzle.

contoured walls. Flow travels from left to right, and only one side of the 2-D throat is shown. The short vertical bleed slot is for the flat sidewalls, whereas the horizontal slots are for the curved walls. Static-pressure fluctuations were inferred from hot-wire measurements with the bleed slots open and closed, at various unit Reynolds numbers, when the nozzle-wall boundary layers were laminar, transitional, and turbulent. The hot-wire data showed good agreement with the earlier pitot-probe measurements.

Figure 7, taken from Fig. 5a in [60], shows typical hot-wire measurements of the freestream fluctuations along the nozzle centerline. Here, the vertical axis on the plot is the rms freestream static-pressure fluctuations divided by the mean, and x is the axial distance from the nozzle throat. The figure also shows measurements of the location of transition on a 5-deg half-angle sharp cone placed on the centerline at two axial positions at unit Reynolds numbers given by the symbol shapes. The upstream end of the uniform flow region begins near $x = 5$ in. (0.13 m). For lower unit Reynolds numbers and farther upstream locations, the noise levels measured in Fig. 7 are below 0.1%, near the electronic-noise limit, because the nozzle-wall boundary layer is laminar at the acoustic origin. Farther downstream or at higher unit Reynolds numbers, the boundary layer becomes turbulent on the nozzle wall, and radiates sound onto the centerline. The consequent increase in noise level begins near $x = 15$ in. (0.38 m) for $R_\infty = 2.5 \times 10^5 \text{ in.}^{-1}$ ($9.8 \times 10^6 \text{ m}^{-1}$) for a quiet length Reynolds number of 2.5×10^6 and near $x = 11$ in. (0.28 m) for $R_\infty = 5.3 \times 10^5 \text{ in.}^{-1}$ ($2.1 \times 10^7 \text{ m}^{-1}$) for a quiet length Reynolds number of 3.2×10^6 .

The most critical parameter for a quiet tunnel is the maximum Reynolds number based on freestream conditions and the length of the quiet uniform region, because this controls the highest Reynolds number under which measurements can be made on a slender model under quiet conditions. The highest value reported was 4.0×10^6 at a unit Reynolds number of $7.9 \times 10^5 \text{ in.}^{-1}$ ($3.1 \times 10^7 \text{ m}^{-1}$) with a very short uniform quiet length of 5 in. (0.13 m) ([60], Fig. 5b), although this length is too short to be of much use for most purposes.

Many measurements of the effect of tunnel noise on transition were made in this tunnel over many years. It is important to note that for many of these measurements, only the nose of the model was in the quiet-flow region, and so transition occurred under conditions that were only partially quiet. The upper part of Fig. 7 also illustrates this effect. When the tip of the cone was placed about 5 in. (0.13 m) downstream of the throat, transition occurred near the base of the cone, at about 12 in. (0.30 m) from the tip, at $x \approx 17$ in. (0.43 m) at $R_\infty = 5.3 \times 10^5 \text{ in.}^{-1}$ ($2.1 \times 10^7 \text{ m}^{-1}$). Note that this high transition Reynolds number of roughly 6.4×10^6 (based on freestream rather than edge conditions) is measured whereas the aft two-thirds of the model is bathed by noise radiated from the turbulent nozzle-wall boundary layer. When the cone was moved 3 in. (0.076 m) aft, transition occurred forward of the cone base, about 10 in. (0.25 m) from the tip, for a transition Reynolds number of about 5.3×10^6 . Although the tunnel performance was later improved somewhat, *the limited ability of the tunnel to provide quiet flow at high Reynolds numbers means that the transition location cannot be measured on a sharp cone at zero angle of attack in this tunnel under fully quiet conditions.* These measurements can say that only under fully quiet conditions, the transition Reynolds number on a sharp 5-deg cone would exceed 6.5×10^6 .

The effect of these streamwise variations in noise level was examined in detail in [61,62]. Transition on sharp cones at angle of attack was dominated by the noise incident on the cone boundary layer upstream of the neutral stability point. This result justified the measurement of transition on many models in which only the forward portion of the model was in the quiet-flow region. It can be argued that tunnel noise is less important for regions in which the amplitude of the instability waves is higher than the amplitude of the radiated tunnel-wall noise.[§] However, the high levels of noise that impinge on the model farther aft probably still have a substantial effect, which has never been quantified, because natural transition

[§]Balakumar, P., and Wilkinson, S., NASA Langley Research Center, private communication, April 2007.

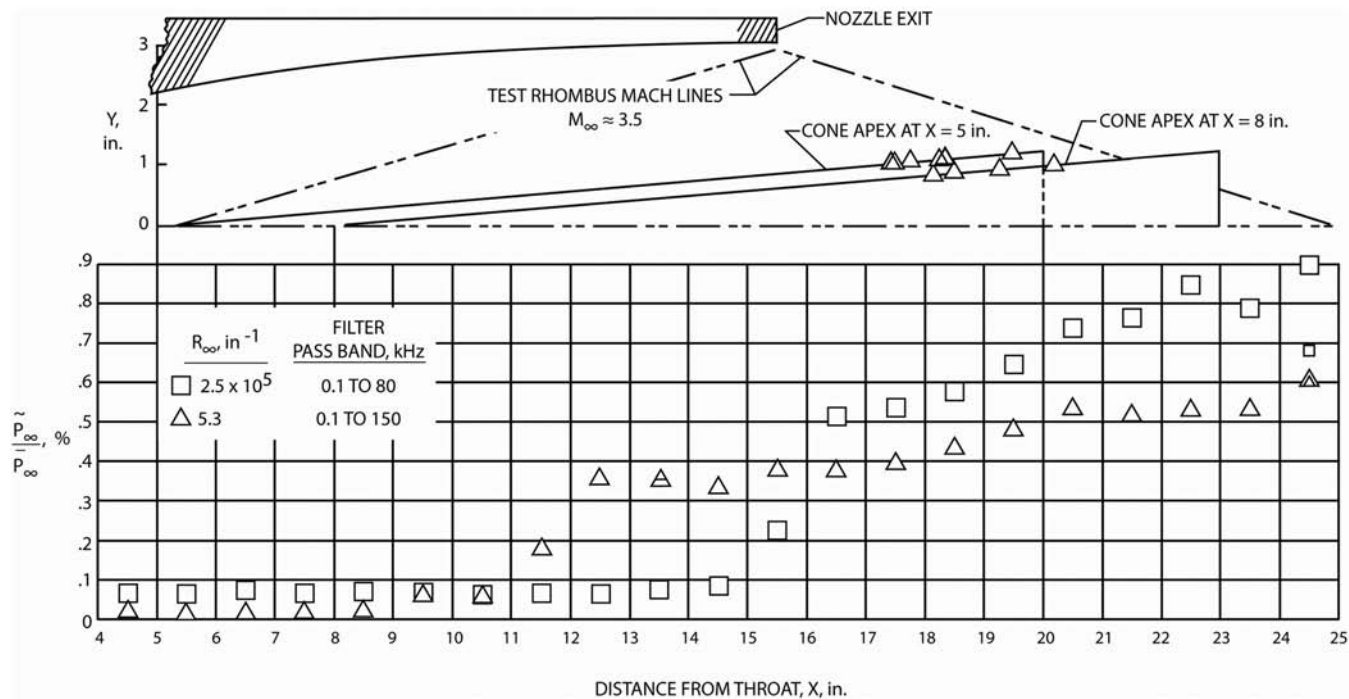


Fig. 7 Variation along centerline of normalized rms static pressure in Mach-3.5 quiet tunnel. Upper part of figure shows locations of transition on the cone in its two test positions.

has never been obtained on a sharp cone at zero angle of attack under fully quiet conditions in which the nozzle-wall boundary layer is maintained fully laminar.

This issue is highlighted in Fig. 8, taken from Fig. 5 in [63], which shows the transition Reynolds number on a sharp 5-deg half-angle cone at zero angle of attack, based on edge conditions, plotted against the unit Reynolds number based on edge conditions. Data from the Mach-3.5 quiet tunnel is shown both before and after the nozzle blocks were repolished. Transition is in the low range of the flight data for moderate unit Reynolds numbers but moves to lower Reynolds number at higher unit Reynolds numbers as transition moves forward on the nozzle walls and the tunnel noise level rises. The plot also includes curves for the Reynolds number based on edge conditions and the length of the cone that is within the fully quiet region Δs . The length of the fully quiet region is defined in two ways: based on a marginally quiet pressure-fluctuation level of 0.1% or on a fully quiet level of 0.03%. The plot makes clear that the downstream half of the laminar boundary layer on a sharp cone sees high levels of noise when the transition Reynolds number on the cone is in the low range of the flight data. To the present author, this suggests that transition would probably occur substantially later if the entire process could take place without contamination from high levels of tunnel noise. Unfortunately, no quiet tunnel has yet achieved this level of performance.

Beginning with [64], Beckwith et al. argued, “very low stream noise levels can be achieved only when the nozzle wall boundary layers are laminar,” abandoning the idea of measuring under any type of turbulent nozzle-wall boundary layers. The Mach-3.5 nozzle blocks were polished to a 3 $\mu\text{in.}$ (0.08 μm) rms finish with 20 $\mu\text{in.}$ (0.51 μm) peak-valley flaws, providing quiet flow at higher unit Reynolds numbers and improving the quiet flow length Reynolds number from about 3.8×10^6 to about 6.7×10^6 ([64], Fig. 3). To maintain this performance, it became critical to maintain the air flow and nozzle surfaces free of particulate and dust, with page 3 of the reference providing one of the first discussions of a long and ongoing effort to maintain very high surface finishes in quiet nozzles by minimizing the largest local microflow.

Oil-flow images again showed that Görtler vortices were present on the 2-D Mach-3.5 nozzle upstream of transition. Figure 9 shows these vortices at two Reynolds numbers under quiet conditions with the bleed slot open and shows they disappear when the nozzle-wall

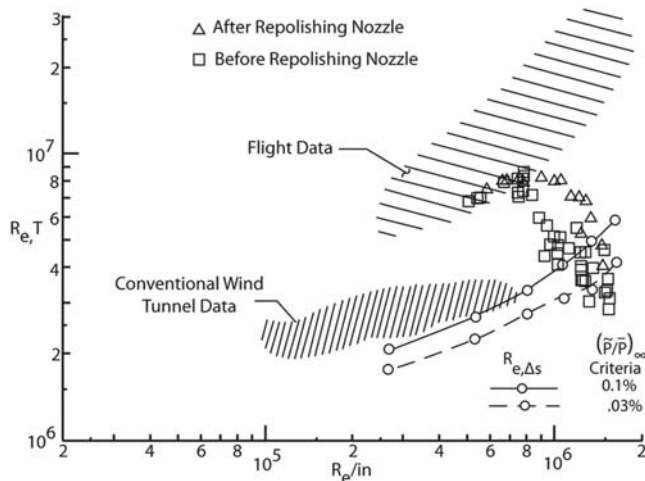


Fig. 8 Variation of transition Reynolds number on sharp-tip cones in the Mach-3.5 tunnel under partially quiet flow.

boundary layer becomes turbulent with the bleed slot closed. The figure is from NASA Langley Research Center photograph L-6105-7, a color version of Fig. 5 from [64]. These Görtler vortices were thought to dominate transition.

The Görtler number, computed as a first estimate of the performance of several nozzles of varying length, suggested that shorter nozzles would provide more quiet-flow performance. Simplified instability computations were then used to compute the development of the Görtler and first-mode instabilities to optimize the nozzle designs to maintain laminar flow. The large favorable pressure gradient associated with the expanding flow minimized the growth of the first-mode waves, which were thus thought to be small. N factors based on the integrated growth of the Görtler instability provided nozzle-performance trends opposite to those predicted by the simpler Görtler number: longer nozzles were predicted to perform better. Although this apparent contradiction still remains an open question after many years, the computations gave the first impetus to the consideration of very long nozzles as opposed to the rapid-expansion nozzles considered previously. The paper also introduced the idea of using nozzles with radial-flow regions

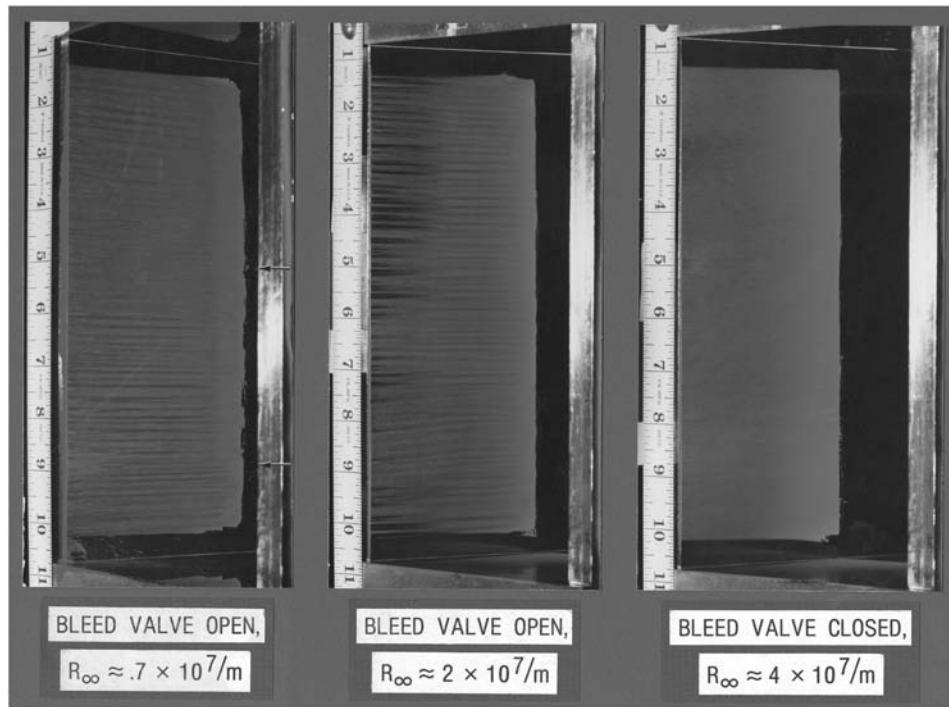


Fig. 9 Oil-flow photographs over a downstream portion of the curved blocks of the Mach-3.5 rapid-expansion nozzle.

between the initial convex expansion and the growth of the Görtler vortices in the final concave region that turns the flow back to parallel.

Analyses of laminar instability in various nozzles were also reported in [64]. A new computer code was developed to analyze the Görtler instability using computed boundary-layer profiles rather than equivalent Blasius profiles. Computer codes were being developed by Malik to design quiet nozzles based on the analysis of the instabilities on the nozzle walls [65]; these codes played an increasingly important role in the design of the nozzles. A Mach-3.5 rod-wall nozzle was still being developed, and so this approach had not yet been abandoned in 1984.

Creel et al. reviewed the effect of roughness in the pilot nozzle throat [66]. Roughness was introduced via flaws in machining or polishing, flaws in the material itself (such as porosity), or particulate that impinged or deposited from upstream. Creel et al. [66] provide a detailed discussion of the development of a system for filtering the air to remove particles larger than $1 \mu\text{m}$. Contaminant samples were collected on 1-in-diam flat-faced cylinders coated with wax or an oily film so that particles remained attached to the surface. It was necessary to clean the nozzle blocks before each run to keep them clear of particulate entering from the room. Later on, clean bleed air was used to purge the nozzle when the test section was opened, reducing the need for frequent cleanings. Frequent cleaning reduces productivity and increases the risk of scratching the finish. Creel et al. also show that repolishing the nozzle blocks from an rms of $2\text{--}10 \mu\text{in}$. ($0.051\text{--}0.25 \mu\text{m}$) to an rms of $1\text{--}3 \mu\text{in}$. ($0.025\text{--}0.08 \mu\text{m}$) improved the quiet Reynolds number by a factor of about 2. The maximum flaw in the nozzle blocks was reduced from about $100 \mu\text{in}$. ($2.5 \mu\text{m}$) to about $40 \mu\text{in}$. ($1 \mu\text{m}$).

Chen et al. computed N factors for the integrated amplification of first-mode and Görtler instabilities on the walls of four supersonic nozzles for comparison to experimental transition data [67]. The boundary-layer profiles were obtained from finite-difference solutions. The Görtler instability was found to dominate, with N factors at transition ranging from 3.5 to 10.9. The lower values were probably caused by excessive roughness and waviness for the nozzles having upstream bleed slots or by residual streamwise vorticity for the nozzle that lacked a bleed slot. The first-mode N factors were about 1 or less for all the nozzles. A Görtler N factor of 9.2 was then used to design several long nozzles with radial-flow sections, showing for the first time the theoretical advantage of

small nozzle-wall inflection angles. This paper also contains measurements from a Mach-3 axisymmetric nozzle, which show that contour waviness with a slope of $0.008 \text{ in./in. (mm/mm)}$ causes Mach waves that focus on the centerline and create disturbances in the mean flow and increases in fluctuations. These results led towards tight specifications for contour waviness in future nozzles.

Beckwith et al. [63] provide a detailed discussion of noise convected from the settling chamber. The settling chamber was redesigned to improve the screen mounting frames, reducing the noise levels. Acoustical theory is used to correlate the propagation of noise from the settling chamber into the test section.

Beckwith et al. [68] provide a detailed discussion of the roughness measured in various NASA Langley nozzles over several years and of the effect of this roughness on transition and quiet flow. A value of the roughness Reynolds number of $Re_k \simeq 10$ is recommended to maintain laminar boundary layers on the contoured nozzle walls ([68], p. 4). Here, Re_k is a Reynolds number based on the roughness height k and conditions in the undisturbed laminar boundary layer at the roughness height [13]. This value of Re_k is much lower than that used elsewhere, presumably because small streamwise vortices from the roughness can amplify via the Görtler instability [13]. Considerable effort is also required to maintain extreme cleanliness on these highly polished surfaces; earlier difficulties with maintaining laminar nozzle-wall boundary layers at high unit Reynolds numbers were now attributed not only to roughness but also to insufficient nozzle-cleaning procedures ([68], p. 3).

Critical Properties of Successful Mach-3.5 Quiet Tunnel

The overall effort in the Mach-3.5 quiet tunnel is well summarized in [69]. This appears to be the last paper describing the development of this tunnel. The first quiet tunnel became successful by combining the following properties: 1) a clean flow entering the nozzle throat with low fluctuations and almost no particles, 2) a bleed slot just upstream of the throat to remove residual disturbances in the contraction-wall boundary layer, 3) a nozzle that is highly polished, particularly near the throat, 4) a 2-D nozzle that is specially designed to obtain laminar flow on the curved nozzle walls by controlling relevant instabilities, and 5) placement of the flat nozzle side walls far enough away that noise radiated from the turbulence at higher local Mach numbers does not affect the quiet-flow core.

Brief Summary of Measurements in Mach-3.5 Quiet Tunnel

The Mach-3.5 quiet tunnel has now been operational for more than 25 years. However, most of the measurements in this tunnel were reported from the 1980s through the middle 1990s, with activity falling off dramatically after the end of the Cold War. The present section is only an incomplete effort to provide a brief summary of the most important measurements in this tunnel. To the author's knowledge, such a summary has not been provided previously.

In nearly all cases, measurements in this tunnel consisted of the transition location measured under noisy and mostly quiet flow, with the bleed slots closed and open. For nearly all the quiet-flow measurements, only the forward portion of the model was in the quiet region, although this is not always clear from the papers. The most well-known example showed that transition on a sharp flat plate occurred later than on a sharp circular cone under quiet conditions [70,71]. This agreed well with semi-empirical e^N theory based on $N = 10$ and was opposite to the trend observed in conventional tunnels.

Creel et al. measured transition on swept cylinders under quiet and noisy conditions with and without roughness and endplates [72]. Without endplates, transition on the smooth cylinders was independent of noise level, but transition generated in part by roughness depended also on the noise level. This geometry simulates the leading edge of winglike shapes like slab delta wings.

Chen measured the extent of the intermittent region between the onset and end of transition, for cones and flat plates, under both noisy and quiet conditions [73]. Under quiet conditions, the transitional region is much shorter than under noisy conditions.

King measured transition on a sharp cone at angle of attack, under both quiet and noisy conditions [74]. The effect of noise was found to depend on angle of attack, and to be smaller at higher angles of attack.

King et al. measured transition in the free shear layer above a cavity, under both quiet and noisy conditions [75]. Tunnel noise was found to have little effect on the transition location. It seems possible that transition was dominated by subsonic cavity-feedback effects or end effects.

Development of Slow-Expansion Axisymmetric Nozzle Concept

Beckwith et al. [68] also describe the design and performance of several long axisymmetric nozzles designed with radial-flow sections and minimal curvature to minimize the Görtler instability. The design and performance of the long Mach-3.5 axisymmetric nozzle is described in detail in [76,77]. The exit diameter was 6.87 in. (0.17 m) and the length was 29.2 in. (0.74 m) from the throat. The nozzle provided a quiet length Reynolds number of 14.0×10^6 at a unit Reynolds number of about $11.0 \times 10^6 \text{ ft}^{-1}$ ([77], Fig. 2). At unit Reynolds numbers above $11.0 \times 10^6 \text{ ft}^{-1}$ ($36.0 \times 10^6 \text{ m}^{-1}$), quiet flow abruptly disappeared, presumably due to the effect of roughness in the nozzle throat ([76], p. 5). This performance was substantially better than that of the 2-D Mach-3.5 quiet nozzle, but the nozzle was apparently never used for measurements on models. The nozzle was electroformed on a mandrel, which had a flawed joint at 22.8 in. (0.58 m) from the throat; the consequent contour flaw focused waves on the nozzle centerline, disturbing the mean flow. This flaw was not easily repaired, and the mean-flow distortion discouraged use of the nozzle for measurements on models; the extended length of the nozzle also caused difficulties with tunnel installation.[†]

Beginning in the middle 1980s, the National Aerospace Plane (NASP) program provided substantial motivation and funding to drive the development of hypersonic quiet tunnels [78–80]. This effort seems to have peaked circa 1990, when Beckwith et al. reviewed the development of hypersonic quiet tunnels for Mach 6, 8, and 18, although the large-scale High Speed Low Disturbance Tunnel described in [81] (pp. 6–8) was never built. A brief summary was also reported in [8].

The Mach-8 nozzle had an 18-in. (0.46-m) exit diameter and a 11.7 ft (3.57 m) length [81]. It was built of Inconel 600 to handle the 1050°F stagnation temperatures expected. It took several years to

successfully fabricate the \$1.5 million nozzle assembly, due to very tight specifications on contour accuracy, waviness, and surface finish [82,83]. Very high surface finishes with 15- $\mu\text{in.}$ (0.38- μm) peak flaws would have been required in the throat to achieve the design performance, due in part to the high Mach number. The present author attended the mechanical design review in the summer of 1990, and failed to foresee any difficulties. Unfortunately, the throat corroded under supersonic flow even though Inconel specimens had not corroded in oven tests at the same temperatures. The nozzle also suffered from temperature-induced distortions in shape that were larger than expected, causing leaks at the nozzle joints.^{**} By the middle and late 1990s, when these problems were becoming clear, the NASP program had ended, funding was scarce, and the nozzle was slowly abandoned as a failure [84]. In hindsight, the Inconel 700 series, which are hardenable, might be an alternate choice, although these materials are not approved by the American Society of Mechanical Engineers (ASME) Boiler and Pressure Vessel Code. Hot-flow tests would be needed to determine if a material can be found to preserve a highly polished finish under the high-temperature throat flow, and improved high-temperature joint designs would also be needed.^{††}

The Mach-18 quiet tunnel was to consist of a new conical slotted nozzle for the NASA Langley helium tunnels ([81], pp. 5–6). The nozzle was nearly 80 in. (2.0 m) long with a 14-in. (0.36-m) exit diameter. The highly polished stainless-steel throat included bleed slots. More detail regarding the design and shakedown is reported in [82,83,85]. Unfortunately, this nozzle never provided quiet flow, and efforts to repair it ended in the early 1990s with the end of the NASP program. The lack of quiet flow was thought to be due to flaws in either the bleed-slot design or a mechanical joint near the throat, although the problem was never definitively resolved.^{**} The helium tunnels have now been completely decommissioned.

Mach-6 Quiet Nozzle

A Mach-6 nozzle with a 7.5-in. (0.19-m) exit diameter was also built for the NASP program [86]. This nozzle has a 1-in.-diam (25-mm-diam) throat and is 39.76 in. (1.00 m) long, from throat to exit. There is a radial-flow region with a 9.84-deg expansion angle between the convex throat region and the concave exit region. The nozzle was electroformed in one piece using a very precise mandrel that was ground to a waviness of 0.0002 in./in.. This provided very uniform flow with no steps or gaps along the wall. Transition onset along the nozzle walls correlated to a Görtler N factor of about 7.5. The quiet-flow length Reynolds number increased with unit Reynolds number to a maximum of about 8×10^6 at a unit Reynolds number of $4.7 \times 10^6 \text{ ft}^{-1}$ ($15.4 \times 10^6 \text{ m}^{-1}$). Here, the stagnation pressure was 230.6 psia (1590 kPa), the stagnation temperature was 182°C, and the length along the centerline of the quiet-flow region was 21.4 in. (0.54 m) ([86], Fig. 7b). For higher unit Reynolds numbers, transition moved abruptly forward, probably due to the increasing effect of throat roughness in the thinner boundary layers. The largest surface defect in the throat region was estimated to have a height of 7 micrometers, or 280 $\mu\text{in.}$ [87].

However, this level of performance required extremely high surface finishes in the throat and extremely clean flow, and it proved erratic and difficult to repeat [82]. A new purge-air system was installed to bleed clean, dry air out of the nozzle whenever it was exposed to the atmosphere. An attempt was also made to plate the nickel nozzle with a hard nickel-phosphorous alloy to improve resistance to corrosion. However, the plating process added waviness and pits, and the nozzle never provided the same high quiet Reynolds numbers as initially measured, even when extensively repolished throughout. Wilkinson et al. [82] also provide a good overview of the design procedures used for the NASA Langley quiet nozzles.

^{**}Wilkinson, S., NASA Langley Research Center, private communication, June 2004.

^{††}Wilkinson, S., private communication, May 2007.

^{**}Wilkinson, S., private communication, April 2007.

[†]Wilkinson, S., and Chen, F., private communication, May 2007.

Wilkinson et al. [83] reviewed the NASA Langley quiet tunnels as of 1994. The Mach-6 nozzle remained fully quiet only to a stagnation pressure of 130 psia (896 kPa) for a quiet length Reynolds number of about 6×10^6 . Figure 1 in [83] shows hot-wire time traces for uncalibrated mass-flow fluctuations on the centerline at the nozzle exit for increasing pressures. As the pressure rises, turbulent spots appear in the boundary layer on the nozzle wall and radiate spiked fluctuations onto the hot wire. The number of spikes increases with pressure through the intermittent region and then decreases as the nozzle boundary layer becomes turbulent.

The performance of the Mach-6 nozzle was characterized for [88]. The nozzle produces very uniform flow at a Mach number of about 5.91. The quiet-flow Reynolds number reached a maximum of about 6.2×10^6 at a unit Reynolds number of about $3.0 \times 10^6 \text{ ft}^{-1}$ ($10.0 \times 10^6 \text{ m}^{-1}$).

Three major projects were carried out in this tunnel before it was shut down. Blanchard and Selby measured wall-cooling effects on a circular cone [89], Lachowicz et al. measured instability and transition on a flared cone [90,91], and Doggett et al. measured instabilities on a cone at angle of attack [92]. The measurements in the Mach-6 quiet tunnel were summarized by Wilkinson [93].

The Mach-6 quiet nozzle had been installed in the Nozzle Test Chamber, which shared a test cell with the 20-in. Mach-6 tunnel. Because the Mach-8 quiet nozzle was expected to supersede the Mach-6 nozzle, and the Nozzle Test Chamber operations conflicted with the 20-in. Mach-6 tunnel, the Nozzle Test Chamber was shut down along with the Mach-6 quiet nozzle. After the termination of the NASP program, NASA's interest in quiet tunnels and hypersonic transition went into a long decline. The Mach-8 quiet nozzle failed to perform satisfactorily, and the Mach-6 nozzle remained in a crate. Eventually, the Mach-6 quiet nozzle and associated hardware were shipped to Texas A&M University, where they are being reinstalled with support from the Air Force Office of Scientific Research (AFOSR).

Other NASA Langley Research Center Efforts

The last portion of the NASA Langley Research Center quiet-tunnel development effort took place in the early 1990s, when supersonic applications became the dominant interest. A large Mach-2.4 quiet nozzle was designed for supersonic transport applications [94]. The quiet-flow Reynolds number of the axisymmetric Mach-2.4 nozzle was expected to exceed 80.0×10^6 . Chen and Wilkinson [94] (p. 3) also report the first use of an elliptical leading edge for the bleed-slot lip, because multiblock Navier–Stokes computations predicted a separation bubble on the nozzle side when the usual circular arc was specified.

The possible use of supersonic quiet nozzles with square cross sections was also investigated using three-dimensional Navier–Stokes analyses [95]. However, crossflow instabilities on the flat sidewalls and other instabilities in the corners eventually made this approach seem unpromising.

Other Quiet-Tunnel Developments

NASA Ames Research Center worked to develop a supersonic quiet tunnel from the late 1980s through the middle 1990s [96–98]. The original concept was to run near Mach 2.4, but the vacuum system could not provide sufficient pressure ratio, and the tunnel eventually operated near Mach 1.6. Low noise flow was achieved at these low Mach numbers at which nozzle-wall acoustic radiation is, in any case, expected to be small. Coleman et al. made measurements on a swept cylinder in this tunnel before it was decommissioned [99,100].

The Japanese National Aerospace Laboratory also built a supersonic tunnel during the middle to late 1990s [101]. The continuous-flow cryogenic tunnel has a 0.2 by 0.2 m test section with a flexible-plate nozzle that can be adjusted from Mach 1.5 to 2.5. A high-quality settling chamber and a highly polished nozzle were

combined with a suction slot upstream of the throat in an attempt to obtain quiet flow. The rms pressure fluctuations in the test section dropped to near 0.1% for Mach numbers below about 1.8 ([102], Fig. 17). Furthermore, later addition of a screen downstream of the compressor successfully reduced the test-section pressure fluctuations below 0.1% for $1.5 < M_\infty < 1.8$ [103].^{§§} However, as the Mach number rises from 1.8 to 2.5, the pitot-pressure fluctuations rise from 0.1 to 0.3% for stagnation pressures from 55 to 100 kPa (8.0 to 14.5 psia). This suggests that the nozzle-wall boundary layers are turbulent, with acoustic radiation becoming significant above Mach 1.8. The measurements are in many ways similar to those of Laufer [20].

The ONERA laboratory at Chalais-Meudon in France has also been engaged since the early 1990s in an effort to develop a supersonic quiet tunnel [104]. The Mach-3 nozzle has an 11.8-in. (300-mm) exit diameter, placed 59.0 in. (1500 mm) downstream of the polished throat, and the nozzle is equipped with an upstream suction slot [105]. Poor performance of the suction slot was suspected as the cause of the initial lack of quiet flow, and detailed simulations of the slot flow were obtained computationally [106] and using a water-tunnel simulation [107]. If the boundary layer on the bleed lip separates on either the main-flow or suction-slot sides, unsteadiness will almost certainly arise, and this is likely to trip the nozzle-wall boundary layer. Both of these simulations helped to show the conditions under which either form of separation is likely to occur.

The settling chamber, bleed lip, and nozzle polish were then improved, and for the first time, the boundary layer was then found to be laminar, at 7.87 in. (200 mm) downstream of the throat, at a stagnation pressure of 60 kPa (8.7 psia) [108]. Further improvements in instrumentation and the suction system led to detection of transition on the nozzle wall about 19.7 in. (500 mm) downstream of the throat, at a stagnation pressure of 60 kPa (8.7 psia) [109]. However, this performance is well short of the goal of laminar flow to 39.4 in. (1000 mm) from the nozzle throat at a stagnation pressure of 150 kPa (22 psia), and so this effort appears to continue at a low level with limited funding. The design and analysis of the tunnel and the bleed-slot flowfield was recently summarized in [105], and a short summary was presented in [110].

In the early 1990s, Demetriades et al. carried out a series of quiet-tunnel experiments in a 3.1 by 3.2 in. (7.9 by 8.1 cm) Mach-3 continuous-flow vacuum-indraft tunnel at Montana State University [111]. The nozzle throat had a radius of curvature of 11.8 in. (30.0 cm), and the nozzle was 15.2 in. (38.5 cm) long. The stagnation pressure could be adjusted from 4.8 to 11.9 psia (33 to 82 kPa). Measurements were made with surface hot films, high-frequency pitot-pressure sensors, and liquid crystals. Navier–Stokes computations were made for comparison [112]. The nozzle-wall boundary layer was laminar to the nozzle exit to about 8.7 psia (60 kPa) ([112], Fig. 10). Later measurements showed that heating the nozzle throat delayed transition, presumably because the throat roughness appears smaller to the thicker boundary layer [113]. Additional measurements of pitot-pressure fluctuations were reported in [114]. The contraction had a number of steps in the contour, which probably generated separation bubbles that created the residual noise measured when the nozzle-wall boundary layers were laminar.^{¶¶} After Demetriades retired, the tunnel was later disassembled.

Purdue University Quiet Ludwig Tubes

Introduction

When the present author arrived at Purdue University in July 1989, it was natural to seek to continue experimental research on laminar-turbulent transition [115]. Unfortunately, Purdue did not have a low-turbulence wind tunnel suitable for instability measurements. Furthermore, the end of the Cold War was leading to reductions in research funding, and other universities had low-turbulence low-

^{§§}Sawada, H., private communication, 7 May 2007.

^{¶¶}Brogan, T., private communication, June 1998.

speed tunnels, and so there was little prospect of funding to build a new low-turbulence tunnel for low-speed transition research.

However, NASA Langley Research Center had recently developed the first low-noise wind tunnels for supersonic and hypersonic transition research, and no other lab had facilities of this type. Furthermore, 1) there were several substantial long-term applications for hypersonic transition research, which suggested that funding might remain available, 2) most U.S. experts in this area were nearing retirement, which suggested there might be a long-term demand for the development of new expertise, 3) the existing NASA Langley quiet tunnels had long run times in large test sections that required very expensive compressed-air and vacuum systems, and 4) the Purdue aerospace department had a large building at a remote site with plenty of low-cost space in which to build a new facility [116]. It appeared that there might be a niche market for a university quiet tunnel with a fairly large test section (sufficient for measurements on models) and a short run time (to minimize operating costs).

Beginning in fall 1989, hypersonic transition research was pursued at Purdue to serve this niche market. It was clear from the beginning that the development of the desired facility would be difficult, time consuming, expensive, and risky. The actual development took far longer and cost far more than was originally expected. The risk of failure was always substantial. The eventual success of the Boeing/AFOSR Mach-6 Quiet Tunnel has only been possible due to the contributions of many people from many different organizations. The first of these people was Hans Hornung from the California Institute of Technology, who suggested the use of a Ludwig tube at a meeting in November 1989.

Ludwig Tubes

A Ludwig tube is a long pipe with a converging-diverging nozzle on the end, from which flow exits into the test section and second throat, as shown in Fig. 10. A diaphragm is placed downstream of the test section. When the diaphragm bursts, an expansion wave travels upstream through the test section into the driver tube, and the measurements are carried out in the gas behind the expansion wave.

The concept of a short-duration wind tunnel driven by gas expanding from a long tube originated circa 1955 with H. Ludwig [117]. The short flow duration, on the order of a second, results in lower operating costs [118]. Supersonic and hypersonic Ludwig tubes were built at the DFVLR in Germany in the 1960s [119]. Several other Ludwig tubes were designed and/or built over the years (see, for example, [120–122]). To the author's knowledge, the longest is the transonic-supersonic facility at NASA Marshall Space Flight Center, which has a tube with a 52-in. inside diameter that is 386 ft long [123]. The large "shock wind tunnel" at the University of Stuttgart is of similar size and is very similar to a Ludwig tube, differing only in having a long pipe to replace the usual dump tank [124]. This difference increases the run time for a given dump-tank

volume [125]. Lukasiewicz summarized Ludwig-tube facilities in 1973 [126].

Many of these Ludwig tubes have upstream valves, which seem to reduce the starting times and make for a simpler, unpressurized test section [119]. However, some have a valve downstream of the test section, which reduces the sources of disturbances in the test-section flow, although it seems to lengthen the starting time and makes for a more complex test section that must sustain stagnation pressure before tunnel start [123,127]. A Ludwig tube with a downstream valve produces a naturally low-noise acceleration from nominally stagnant flow into the test section. When this is combined with the moderately short run time, this type of facility is a natural choice for a quiet-flow tunnel.

Four-Inch Mach-4 Quiet Ludwig Tube

In January 1990, the present author visited NASA Langley Research Center and proposed developing a quiet-flow Ludwig tube; the summer of 1990 was then spent at NASA Langley learning about quiet tunnels from Beckwith, Chen, Wilkinson, and others. Schneider [128] reports initial experience with the computational methods to be used for the design of the quiet-flow nozzle. Preliminary designs were also developed for a large Ludwig tube that was to have a 200-ft-long (61-m-long) driver tube, 2–4 ft (0.6–1.2 m) in diameter, and a smaller Ludwig tube with an 80 ft (24 m) driver, 1 ft (0.3 m) in diameter. A 500 ft³ (14.2 m³) tank was procured and installed; in the long term, this was to provide vacuum for the eventual Ludwig tube, whereas in the short term it enabled operation of the 2-in. supersonic blowdown tunnel at low Reynolds number. Schneider [129] then reported the justification for building a quiet-flow Ludwig tube, some initial design concepts, and a description of the methods to be used for design of the laminar-flow quiet nozzle.

Funding then became available to build the smaller Ludwig tube, which was constructed during 1990–1992 [130]. Vacuum was generated using the 500 ft³ (14.2 m³) tank placed just outside the building, combined with a surplus 5 hp (3.7 kW) pump. Compressed air was obtained from the system previously established for the supersonic teaching tunnel. The driver tube was built of carbon steel, 68 ft (20.7 m) long and 12 in. (0.30 m) in diameter, elevated 8 ft (2.4 m) above the floor, above several teaching labs. The driver tube was painted on the inside as well as the outside to reduce rust-induced particulate. The last 8 ft (2.4 m) section of the driver tube was honed and machined to fit the contraction that was inserted into the end of it. The 3.8 × 4.3-in. (97 × 109-mm) rectangular nozzle was provided by NASA Langley; it was the solid counterpart to the porous nozzle mentioned in [23] and is of conventional design. Openings were added near the exit for two pairs of 3 in. (76 mm) windows. Downstream of the nozzle is a rectangular test section, also 3.8 × 4.3 in. (97 × 109 mm) with three 3-in. (76-mm) window openings at the sides and bottom and a probe-traverse slot on top. A fixed second

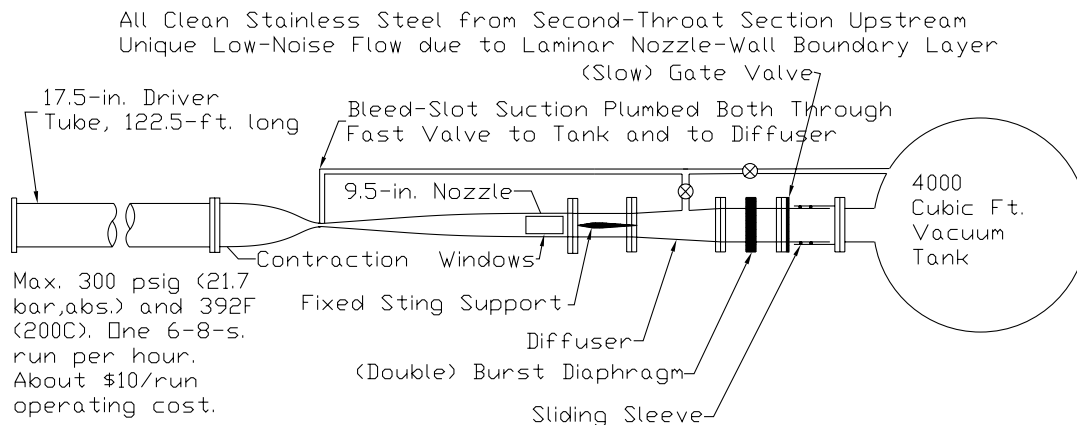


Fig. 10 Schematic of Boeing/AFOSR Mach-6 quiet tunnel.

throat was used with a double-diaphragm apparatus for starting up the flow.

The contraction was designed using inviscid computations to minimize pressure gradients that might otherwise cause boundary-layer separation. The contraction ratio is 83, leading to a very low driver-tube Mach number of 0.0069. This unusually low value was the result of oversizing the driver tube for later use with a larger test section. During the first 0.1 s of the run, the gas in the tube flows only about 9.4 in. (0.24 m). The oversized vacuum tank was also the result of designing for long-term use of larger nozzles; the combination turned out to be very useful for generating longer-duration runs, although this was serendipitous. Analyses of the potential for larger test sections were reported for various Mach numbers.

Computations showed that the roughness Reynolds number Re_k was no more than 6 in the throat, for a stagnation pressure of $P_t = 60$ psia (414 kPa) for an estimated maximum local roughness of $64 \mu\text{in.}$ ($1.6 \mu\text{m}$) ([130], p. 8). For $P_t = 15$ psia (103 kPa), $Re_k \approx 0.8$. The throat was later polished to a mirror finish with a nominal rms of $1\text{--}2 \mu\text{in.}$ ($0.03\text{--}0.06 \mu\text{m}$). This high polish turned out to be the key to achieving quiet flow and roughly followed Beckwith's estimate that the peak roughness heights are typically 15–30 times higher than the rms. Although no published record of this Beckwith estimate could be found, similar information is available in [63,131].

The peak Görtler number in the contraction varied from about 3–10, depending on the fetch and stagnation pressures ([130], Fig. 11). Later during a run, the boundary layer in the contraction is the result of gas that originates farther upstream, with a longer fetch, resulting in a thicker boundary layer and higher Görtler numbers. These high Görtler numbers provide some explanation for the streamwise vortices present in the NASA Langley contraction, visible in the upper-left portion of Fig. 3. The Görtler number on the curved nozzle walls ranged up to about 9 near the exit at $P_t = 15$ psia (103 kPa). The crossflow instability on the flat sidewalls was not computed, partly because there was no feasible way to do the computation and partly because its importance was not fully appreciated.

When the Mach-4 Ludwieg tube was first completed, it did not provide quiet flow at high Reynolds numbers as planned. Shakedown involved development of operating procedures and instrumentation, which was not trivial because a wide range of operating pressures had to be tested out using new burst diaphragms developed for that purpose. In late 1993, quiet flow was achieved for the first time, although only at low stagnation pressures of about 15 psia (100 kPa) [132]. A high-quality polish in the contraction and nozzle were critical to obtaining quiet flow, with a particularly high level of polish and cleanliness being necessary in the throat.

The flow in the nozzle was measured using Kulite pressure transducers combined with custom-built low-noise electronics that provide a gain of 100 to the dc output, and a total gain of 10,000 to a second output that is high-pass filtered near 800 Hz. The output was recorded with a Tektronix TDS420 digital oscilloscope with hi-res mode and a record length of 15,000 words per channel. In hi-res mode, these 8-bit Tektronix scopes sample at full speed, averaging the data on the fly via a digital-signal-processing chip and saving the average to memory at the requested sampling rate. This provides digital filtering of high-frequency noise and increases the voltage resolution to 10–13 bits, both properties that are often critical for capturing the very small signals often present in quiet-flow tunnels. Although only 60 ms of data could be recorded at 250 kHz, the startup transient was about 40 ms, and the first part of the run showed that the mean Mach number was about 3.8, slightly below Beckwith's earlier measurements with the same nozzle at NASA Langley. Compared with Beckwith's measurements during very long run times, the mean Mach-number measurements at Purdue scattered more and were slightly lower, in part due to inadequate equipment for calibrating the Kulite transducers.

Following a suggestion from Beckwith, the tunnel noise was characterized using the rms pitot pressure divided by the mean, just as in [59], because this was much simpler than developing hot-wire calibrations. "An interactive process of finishing and measurement

was carried out after the facility was first assembled in the summer of 1992" ([132], p. 8). The first two rounds of polishing were focused on the contraction and were carried out in house, producing rms pitot pressures ranging from 0.1 to 0.4% for P_t ranging from 10 to 20 psia (69–138 kPa) with scatter of a factor of 3. Visible streamwise scratches remained in the throat of the nozzle that was obtained from NASA Langley. It was clear that professional expertise was needed to obtain the best and most uniform finish, particularly near the nozzle throat. In October 1993, the nozzle and contraction were polished from $16 \mu\text{in.}$ ($0.4 \mu\text{m}$) rms to about $1\text{--}2 \mu\text{in.}$ ($0.03\text{--}0.05 \mu\text{m}$) by Optical Technologies, near Chicago, Illinois. Following this polish, low quiet pitot fluctuations of 0.05 to 0.08% were successfully obtained for stagnation pressures below 14 psia (97 kPa) ([132], Fig. 14). This provided uniform quiet flow for a freestream length Reynolds number of about $1.3 \times 10^6 \text{ ft}^{-1}$ ($4.2 \times 10^6 \text{ m}^{-1}$) over a length of about 3.9 in. (0.10 m) for a quiet-flow length Reynolds number of about 400,000 [133].

Maintaining a clean and highly polished throat was essential to this performance and remains essential to the operation of every quiet tunnel. The performance of quiet tunnels *must* be monitored on a regular basis, preferably during each run, to ensure that contaminants have not tripped the boundary layer and eliminated quiet flow. The air supply, contraction, and nozzle must be treated like a clean room, with extraordinary care to obtain air free of particulates and condensable vapors. The throat of the tunnel is a cold trap for oil vapors generated at the air compressor and incompletely removed by filtering. Grease and oil deposits in various places in the contraction and nozzle would sometimes but not always trip the boundary layer. Dust was often trapped by the grease or oil, after which the dust would often trip the boundary layer. This was particularly a problem for the later measurements with heated air in a heated driver tube [134].

The Mach-4 nozzle was regularly cleaned from downstream, using alcohol and soft, lint-free cloths attached to makeshift tools composed of soft materials like wood or plastic. Perhaps one or twice a year, it was also necessary to remove the contraction to clean oil and particulate that were not accessible from downstream. Although the oil from the air compressor was mostly removed by two filters, a twin-tower dryer and an air filter rated to remove 99.9999% of all particles larger than $23 \mu\text{in.}$ ($0.6 \mu\text{m}$), the original air supply system and painted carbon-steel driver tube provided air of marginal quality.

Traces of the fluctuating pitot pressure divided by the mean were obtained for increasing stagnation pressures. These reflect the transition process on the nozzle wall, confirming the presence of quiet flow as in the NASA Langley Mach-6 tunnel ([83], Fig. 1). Figure 11, redrawn from Fig. 8 of [45], is typical of the truly quiet pressure-fluctuation data. Even the peaks in the fluctuations are rarely above 0.1% of the mean pitot pressure. As the pressure is

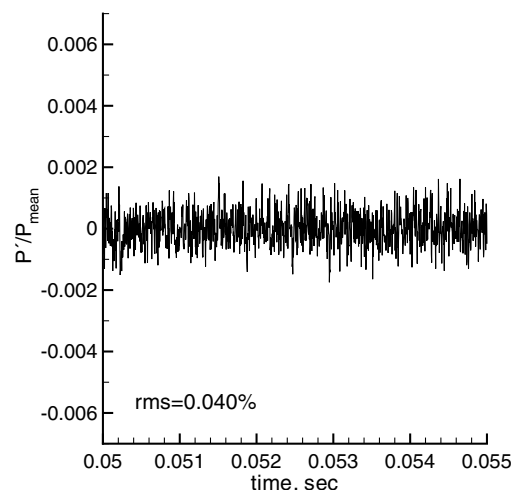


Fig. 11 Pitot-pressure fluctuations in Mach-4 quiet nozzle at $P_t = 74.5$ kPa.

increased, transition in the nozzle-wall boundary layer begins to move upstream, and the signature of turbulent spots in the nozzle-wall boundary layer begins to be apparent in the pressure records on the centerline. A weak pulse of this type is present in Fig. 12 at 0.054 s; this figure is redrawn from Fig. 9 of [45]. The rms noise for this record has increased to 0.045%, but is still small because only a single small pulse is present. A further increase in pressure produces Fig. 13, which contains several pulses whose peaks are an order of magnitude above the background; the figure was redrawn from Fig. 10 of [45]. The rms noise in this record is still only 0.059%, but the flow is no longer perfectly quiet. The pressure fluctuations in the whole trace are above 0.2% of the mean for only 0.6% of the time.

Figure 14, redrawn from Fig. 11 of [45], shows what happens when the total pressure is raised further. The noise level is now 0.38%, almost an order of magnitude higher than in Fig. 11, and 48% of the signal is above 0.2% of the mean. The acoustic origin on the nozzle walls is probably now turbulent. Note that the scales are the same in Figs. 11–13, but the vertical scale was increased by a factor of 2 in Fig. 14 to accommodate the larger signal; in all four figures, only a portion of the record is shown, for clarity.

This qualitative change in the fluctuations as the stagnation pressure rises is the best evidence of a laminar nozzle-wall boundary layer at the lower stagnation pressures. Clearly, the pressure fluctuations felt on the nozzle centerline due to the passage of turbulent spots on the nozzle walls must depend on the geometry of the nozzle as well as on the general flow conditions. For the Mach-4 quiet tunnel, it seems conservative to say that the flow is quiet when P'/P_{mean} is less than 0.05%. Although the occasional pressure pulse

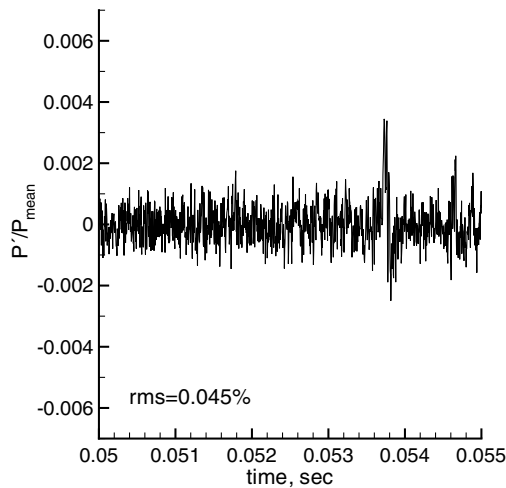


Fig. 12 Pitot-pressure fluctuations in Mach-4 quiet nozzle at $P_t = 81.2$ kPa.

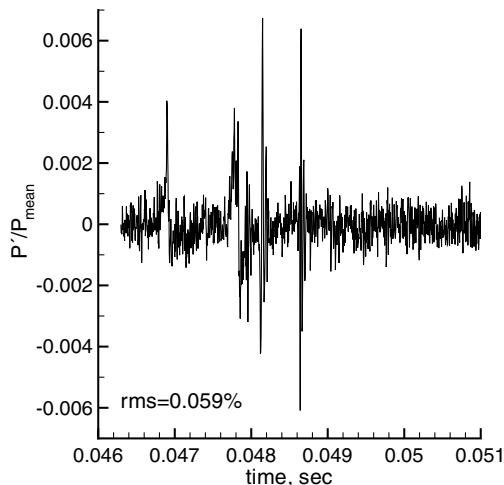


Fig. 13 Pitot-pressure fluctuations in Mach-4 quiet nozzle at $P_t = 94.3$ kPa.

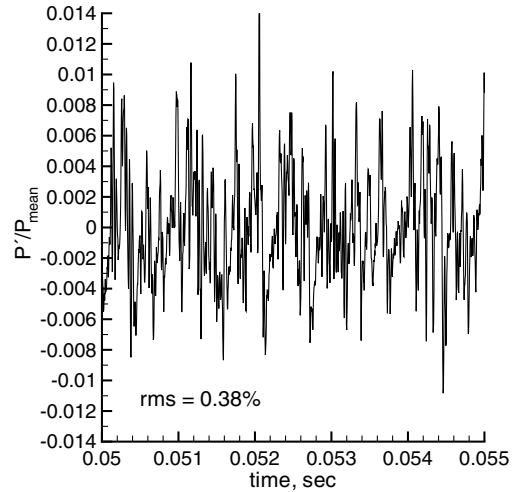


Fig. 14 Pitot-pressure fluctuations in Mach-4 quiet nozzle at $P_t = 122.4$ kPa.

may still pass the measuring station even under these conditions, ensemble averaging or conditional sampling techniques should be able to handle this small proportion of noise in the flow.

Measurements were then taken later on during the run, which turned out to last about 3–4 s, during which the stagnation pressure drops about 34% [135] and the stagnation temperature drops about 12% [136]. A simple model of choked flow from a reservoir gave a good fit to the measured pressure and temperature drops [136]. Surprisingly, the tunnel remained quiet later in the run, despite the repeated reflections of the expansion wave within the driver tube [135] (see also [137], Fig. 2). This unexpected result enabled quiet-flow measurements during the extended runtime possible with the oversized driver tube and vacuum tank. The pitot pressure decreases in stair steps, every 121 ms, when the expansion wave reflects from the contraction; these steps slowly spread out as the expansion wave thickens. The 121-ms steady periods are more than sufficient for quasi-steady measurements of instability and transition, and the slow decrease in pitot pressure conveniently provides a range of Reynolds numbers during a single run.

Schneider et al. [135,136] also report the development of a new diaphragm-breaking apparatus that greatly improved operations at the low quiet pressures. Thin nichrome wires were taped across thin mylar diaphragms clamped in a special ring. A pulse of current was discharged through the wires to melt the mylar and initiate a run on command. A sharp cone with a base diameter of about 1.85 in. successfully started in the tunnel ([135], p. 12), although base diameters of about 1 in. were used for most measurements, because the larger cones would not fit completely within the quiet region [138]. Improvements were also made to the tunnel suspension system, easing alignment of the various sections, and to the hydraulically driven telescoping section that is used when changing burst diaphragms or installing models ([136], pp. 4–5).

The Mach-4 tunnel was used to develop a glow-discharge perturber [138], a laser perturber [139], a laser differential interferometer [140], and methods for measuring with hot wires and hot-film arrays [141]. However, the low quiet Reynolds numbers meant that instability waves amplified by less than a factor of 10 under quiet conditions [138], and the small test section resulted in small models with thin boundary layers. In addition, most applications were for hypersonic flows, and the critical hypersonic second-mode instability could not be studied at Mach 4 ([136], p. 7). A larger tunnel was needed with higher quiet Reynolds numbers and a hypersonic Mach number.

One of the risks involved in developing a hypersonic Ludwig tube was associated with unsteady free-convection flows that result within the driver tube when it is heated. Free convection sets up spatial variations in density and temperature that transform into flow fluctuations when they are accelerated into the nozzle. To assess this problem, the downstream end of the Mach-4 Ludwig tube was

heated, and measurements of the rms pitot pressure were made near the nozzle exit [134,137]. The experiments did not reveal any major difficulties. Heating the air by 30% did make the nozzle throat look cold, reducing the quiet-flow Reynolds number by about 15%, which appeared generally consistent with the earlier measurements of Demetriades [113], Anders et al. [53], and Harvey et al. [142].

Boeing/Air Force Office of Scientific Research Mach-6 Quiet Tunnel

Design and Preparation

In fall 1993, the Boeing Company announced a \$0.5 million gift to Purdue University to support a transonic–supersonic–hypersonic Ludwig tube, with the funds to arrive during 1994–1998. It soon became clear that these funds would have to be matched with substantial federal funds if a state-of-the-art facility was to result. With the impending termination of NASA's High Speed Civil Transport program, the market for supersonic transition research appeared very limited, and plans for a supersonic nozzle were soon eliminated. Although transonic viscous-flow problems could be effectively addressed in a large transonic Ludwig tube with an adaptive-wall test section, prospects for federal funding to support research in such a facility were also limited, and the transonic test section was also dropped. The large, low-pressure driver tube needed for the transonic test section was not compatible with the smaller, high-pressure driver tube needed for the hypersonic test section.

However, AFOSR remained interested in hypersonic transition and expressed a willingness to support the development of both a prototype Mach-6 quiet nozzle and a larger, full-scale Mach-6 quiet nozzle with a 24-in. exit diameter ([136], p. 21). Thus, the effort rapidly focused on development of a hypersonic quiet tunnel.

During 1995–1998, much of the preparatory work was carried out. The building entrance was relocated to create an enclosed space for the Mach-6 test section at the end of the building. An air-cooled pump room was built for the vacuum pump, main compressor, and boost compressor. A 4000 ft³ (113 m³) propane tank was obtained surplus, installed on footings, converted for use at full vacuum, and repainted. Structure was built to support the large Ludwig tube and provide a working platform in the test area. Quiet nozzle design studies were also carried out for both tunnels.

The nozzle design studies are reported in [143,144]. The need for a high-Reynolds-number quiet tunnel was reiterated, and it was shown that a large improvement in quiet Reynolds number is needed to achieve natural transition on near-symmetric geometries under fully quiet flow. Mach 6 was chosen for the nozzle because the hypersonic second-mode instability can already dominate for a cone at zero angle of attack, and because the hypersonic insensitivity to roughness effects is already observed there, yet the temperatures required to reach Mach 6 are moderate. These moderate temperatures reduce risk and cost by permitting the use of stainless-steel nozzles and pressure vessels, and by permitting the use of instrumentation and plastic sealing materials that are not feasible at Mach 8. Axisymmetric nozzles were to be used to eliminate crossflow and corner-flow instabilities.

Schneider [143] (p. 5) also notes that the throat roughness Reynolds numbers for quiet tunnels depend only on upstream pressure and temperature and not on the Mach number at the nozzle exit. Feasible throat finishes require that high quiet Reynolds numbers be obtained using larger nozzles at moderate Mach numbers. Thus, stagnation pressures were limited to 150 psia (1034 kPa), and stagnation temperatures were taken as the minimum required to avoid static liquefaction, about 820R (456K). Large nozzles also enable larger models with thicker boundary layers that ease measurements. The high mass flow of larger nozzles can be accommodated with acceptable operating costs if the runtime is reasonably short.

Extensive computations were carried out using the method of characteristics, a finite-difference boundary-layer code, and the e**Malik code for computation of first, second, and Görtler instabilities [65]. Transition was estimated when the square root of the sum of the squares of the three individual N factors exceeded 7.5. A very long Mach-6 prototype nozzle was designed using with an

angle at the inflection point of 4.0 deg, a length of 102 in. (2.6 m), and an exit diameter of 9.5 in. (0.24 m). This was predicted to perform twice as well as the NASA Langley Mach-6 quiet nozzle, mainly because the reduction in curvature was predicted to damp the Görtler instability without excessive increases in the first- and second-mode instabilities. Use of a heated throat and a room-temperature exit improves quiet-flow performance and eases both operations and the design of conformal windows provided near the nozzle exit. The decrease in temperature along the nozzle wall damps the first-mode instability without excessive adverse impact on the second mode, apparently because the second mode does not begin to amplify until near the nozzle exit.

Figure 15 shows a schematic of the nozzle exit. A slender cone is drawn near the maximum size that successfully starts at zero angle of attack. The bow shock from the cone is also drawn. Uniform flow begins near $z = 75.13$ in. (1.91 m) downstream of the throat, the nozzle ends at $z = 101.975$ in. (2.59 m), and the curved part of the nozzle ends just upstream at $z = 101.734$ in. (2.58 m). The rectangular outlines show the location of the eight window ports, most of which are presently filled with solid metal blanks.

Mach lines are drawn on Fig. 15 to show the onset of radiated noise for various estimates of transition on the nozzle wall. Case “m1,” not shown in Fig. 15, assumes a throat heated to 1000R (556K) for the first foot (0.30 m), followed by a linear taper to 540R (300K), where the wall temperature is held for the last 2 ft (0.61 m). For this case, the boundary layer is predicted to be laminar past the nozzle exit, for a quiet Reynolds number of 13.2×10^6 based on a length of 4.3 ft (1.31 m) from the onset of uniform flow to the location where the noise impinges on the centerline. Case “m2” assumes an unheated throat with a wall temperature that tapers linearly from 820R (456K) at the bleed-lip tip to 540R (300K) at the exit. As shown on Fig. 15, this suggests transition will begin near the upstream end of the large windows, for a quiet Reynolds number of 10.2×10^6 based on a length of 3.3 ft (1.01 m) and noise that impinges near the aft end of the cone. Case “m5,” also shown on the figure, was computed assuming a stagnation pressure of 250 psia (1724 kPa) instead of 150 psia (1034 kPa) but the same stagnation temperature 820R (456K). Here, the wall temperature was assumed to be at 1000R (556K) for the first foot of the nozzle, followed by a linear taper to 540R (300K), where it is held for the last 2 ft (0.61 m). This case predicts the onset of transition at about $z = 93.6$ in. (2.38 m) at the nozzle wall and a quiet Reynolds number of 19.4×10^6 based on a length of 3.77 ft (1.15 m), assuming the throat roughness does not trip the flow at this higher unit Reynolds number. The noise would impinge aft of the end of the cone.

Thus, the quiet length Reynolds number was predicted to increase from about 7.0×10^6 for the NASA Langley nozzle to about 13.0×10^6 for the longer Purdue nozzle using a heated throat. Doubling the length of the nozzle is predicted to yield an approximate doubling of the quiet Reynolds number. Figure 16 shows the quiet length Reynolds number $R_{\Delta x}$ plotted vs the unit Reynolds number in the nozzle freestream $R_{\infty} \text{ m}^{-1}$. The computations for the NASA Langley nozzle were based on a Görtler N factor of 7.5 [86]. The measurements in the NASA Langley nozzle were digitized from [87,88]. Most of the actual measurements in the NASA Langley quiet nozzle were made at the unit Reynolds number shown by the vertical dashed line at a stagnation temperature of 810°R (450K) and a stagnation pressure of 130 psia (896 kPa) ([90], p. 3). For the nominal wall-temperature distributions with a combined $N = 7.5$ the Purdue computations estimate that the quiet Reynolds number is more than 13.0×10^6 for the two lower unit Reynolds numbers because the boundary layer was expected to be laminar at the nozzle exit; thus, the upward arrows above those two symbols. At the higher unit Reynolds numbers, quiet Reynolds numbers near 19.0×10^6 are estimated. Remarkably, this higher value is independent of unit Reynolds number due to the combination of factors involved, although it assumes that the roughness in the nozzle throat does not become a limiting factor. Heating the nozzle throat from 820°R (456K) to 1000°R (556K) enables a given throat roughness k to provide nearly the same values of Re_k at a stagnation pressure of 200 psia (1379 kPa) instead of 150 psia (1034 kPa) ([143], p. 28).

Sketch of Small Bluntness Cone in Nozzle. Dimensions in inches.

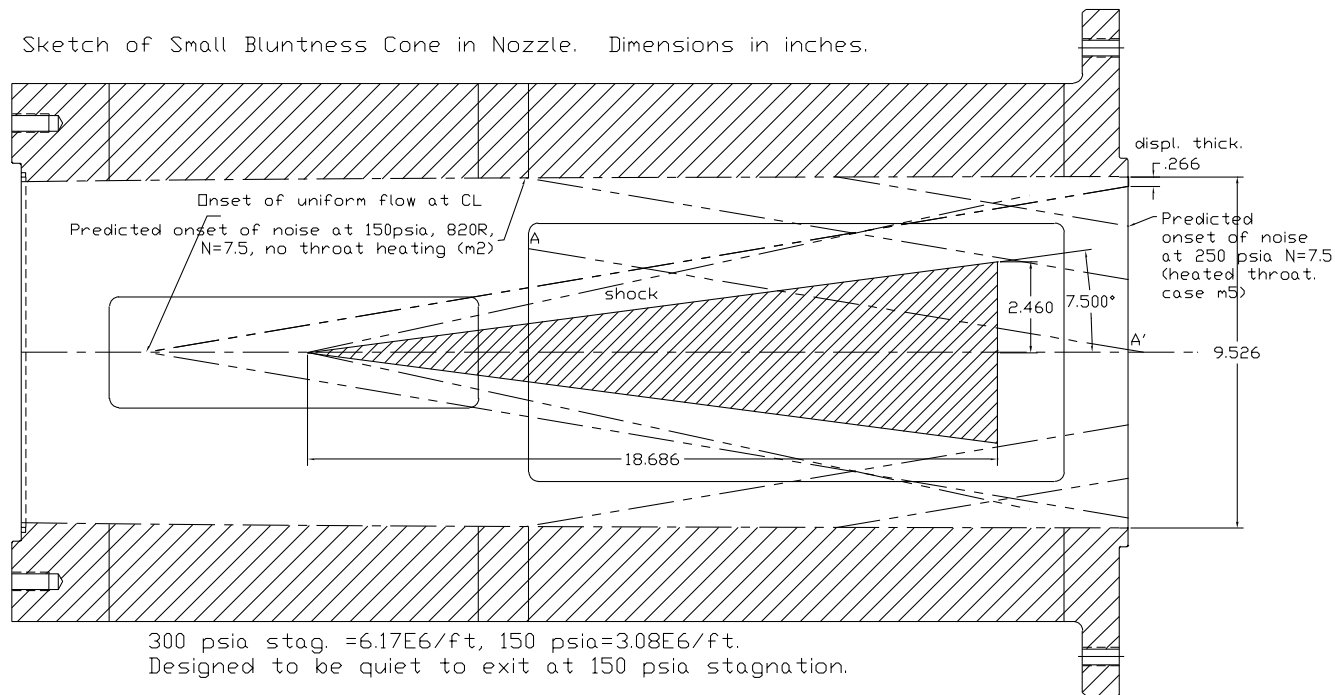


Fig. 15 Schematic of Mach-6 quiet nozzle with slender cone showing estimated transition locations.

The measured performance of the Purdue tunnel still needs to be added to Fig. 16 because the spatial extent of quiet flow has not yet been measured. However, the symbol labeled “expt. temp.” shows a computation for the Purdue tunnel using a good approximation to the actual temperature distribution along the nozzle at a stagnation temperature of 780°R (433K) and a stagnation pressure of 150 psia (1034 kPa) [145]; the estimated quiet Reynolds number is much lower and approximately the same as that in the NASA Langley nozzle. A full-scale Mach-6 nozzle that is 33 ft (10 m) long with a 24-in. (0.61-m) exit diameter was predicted to produce quiet flow to a length Reynolds number in excess of 36.0×10^6 .

Design and Initial Fabrication

When available funding did not permit fabrication of the large Ludwig tube, the prototype was built in the space originally

reserved for the larger tunnel. The driver tube was built from 122.5 ft (37.3 m) of 17.5-in. (0.44-m) inside diameter stainless-steel pipe. This was large enough to permit a person to crawl through it to clean it out, and the use of stainless steel eliminated the problems with flaking paint that were observed in the Mach-4 Ludwig tube. The pipe could be rated for 300 psig (2170 kPa) at 392°F (473K), without additional cost, permitting operation at Reynolds numbers about twice those for which quiet flow was expected, to permit taking advantage of any serendipitous performance. Considerable attention was paid to providing a smooth driver tube, without large steps at the joints, to minimize the risk of transition in the boundary layer on the driver-tube wall and any other possible disturbance sources such as boundary-layer separation at joints. The risk of boundary-layer separation was to be minimized, because it often leads to low-frequency unsteadiness in wind tunnels. An axisymmetric panel method was combined with a laminar boundary-layer analysis to design a long contraction that avoids the boundary-layer separation that is predicted in the short contraction used with the NASA Langley Mach-6 quiet nozzle.

An attempt was also made to design the bleed lip to avoid separation ([146], pp. 13–14). It was noted that the hemicylindrical leading edge used in successful NASA Langley designs can be expected to produce a small boundary-layer separation at the shoulder of the bleed lip. Because no difficulty had been observed in the NASA Langley designs, and because it appeared very difficult to machine an elliptical leading edge for the 0.030-in.-diam (0.76 mm) bleed lip, the hemicylindrical lip design was followed. An attempt was made to design the location of the stagnation point on the bleed lip to eliminate any separation on the main-flow side of the lip. However, available computational methods did not permit a detailed analysis of the viscous transonic flow in the throat.

It is not possible to machine the inside of a long axisymmetric shape using a long boring bar without introducing chatter and excessive deflection. To maintain high accuracy in the internal contour and to control costs, it was necessary to machine the 102-in.-long (2.59 m) Mach-6 nozzle in eight sections. Great care was taken with the design of these sections to provide the smoothest possible lap joint. Starting from upstream, the internal contour was to be machined up to near the joint, then the next downstream section was to be added, carefully aligned in the lathe, and then machined up to near the next joint, repeating until the nozzle was complete. A test

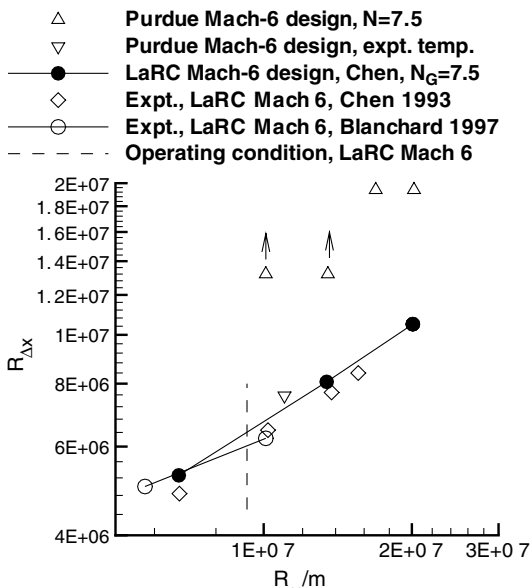


Fig. 16 Quiet Reynolds numbers for Mach-6 nozzles at NASA Langley Research Center and Purdue University.

specimen was designed and machined to prove out the joint concept [146].

The overall design of the facility was summarized in [147]. It was noted that heating the throat increases the predicted quiet Reynolds number from 10.0×10^6 – 13.0×10^6 .

A detailed review of the roughness and waviness requirements was carried out during fabrication of the nozzle ([148], pp. 3–6). Axisymmetric focusing of contour disturbances was considered along with the effect on nozzle-wall transition. The Purdue nozzle was fabricated to keep the peak roughness within $Re_k = 12$, with the peak estimated to be perhaps 30 times the rms. The contour waviness was to be less than 0.001 in./in..

Contour measurements on the test joint were also reported ([148], pp. 6–10). No step could be detected at the mechanical joint, either before or after thermal cycling. However, the internal contour was also machined in two setups, simulating the process that was to be followed for nozzle fabrication, and a 0.0034 in./in. wave appeared at the joint between the two machining steps. The test joint was successfully polished to 0.6 $\mu\text{in.}$ (0.015 μm) with a peak flaw of 3.3 $\mu\text{in.}$ (0.084 μm) according to a single linear pull of a Talysurf profilometer. Only about 120 $\mu\text{in.}$ (3 μm) of material was removed during the polishing, demonstrating that the polishing has little effect on the contour accuracy.

Despite this success, it was decided to build the nozzle with an electroformed throat, eliminating mechanical joints at $z = 3.76$ and $z = 9.01$ in. (95.5 and 228.9 mm) downstream of the throat and placing the first interior joint at $z = 19.3$ in. (490 mm), where the allowable peak roughness was 0.0029 in. (74 μm) ([148], pp. 10–17). Extensive tests were carried out with specimens to characterize and develop the fabrication processes. However, the first two mandrels for the throat electroform were still both failures. When the mandrel is ground to high accuracy, the surface must be excellent if it is to be polished to less than a microinch (0.025 μm) rms without removing more than 0.0001–0.0002 in. (2.5 to 5.0 μm) of material and damaging the contour. Unfortunately, the first mandrel suffered from pits that were revealed during the polishing process; this problem was apparently due to a flaw in the material. The second mandrel suffered from gouges introduced during the grinding process, which again could not be polished out without excessive material removal.

Detailed measurements of the as-machined contraction contour were also reported ([148], pp. 17–20). The accuracy was excellent, except near $z = -23$ in. (–584 mm), where the slope was 0.015 in./in. (mm/mm). The larger flaw at this location was the result of the need to blend the final cut in the middle of the contraction to final-machined surfaces both upstream and downstream; this process overconstrained the alignment of the final cut and made it nearly impossible to achieve sufficient accuracy. It was clear that the final internal nozzle contour *must* be machined only in successive steps starting from the throat.

Schneider [148] (pp. 20–21) also reports details of the apparatus used to heat the driver tube by passing a large dc current through it, along with initial measurements of the temperature distribution in the driver tube. It is necessary to heat the tube for more than 14 h before the thick flanges at the joints reach equilibrium, and so the driver tube is kept at 160°C all the time. The compressed air is passed through an aftercooler, dryer, and five filters before passing into a stainless-steel circulation heater, which heats it to near 160°C before entering the driver tube. Minimum operating temperatures are also discussed.

The third mandrel for the electroformed throat was successfully ground from Optimax by the Schmiede Corporation [149]. Flaws were within 0.001 in. (25 μm), and waviness was within 0.001 in./in.. There was a flaw perhaps 1/16 in. (0.2 mm) long about 5 in. (1/8 m) downstream of the throat, with a depth of perhaps 0.001 in. (25 μm), which was a concern, but it was decided to proceed with electroforming.

Because the shape of the region of uniform quiet flow downstream of the nozzle exit makes it less useful, initial plans for a “test section” downstream of the nozzle were omitted to reduce costs, and the nozzle was connected directly to a sting-support section that also

serves as a diffuser ([149], p. 4). Schneider [149] also reports the design of the support structure, vacuum system, diffuser, and second-throat section. The bleed air from the throat suction slots was designed to reenter the main flow at the diffuser, providing a passive means of timing the beginning of suction.

Schneider et al. [145] describe the system for heating the contraction and computations of the temperature distribution along the first part of the nozzle. The temperature drops below 40°C by about 20 in. (0.5 m) downstream of the throat. This temperature distribution was used to recompute the expected location of transition on the wall (see the data for “expt. temp.” in Fig. 16). In Fig. 15, the onset of noise from the upper wall at 150 psia (1034 kPa) and 780°R (433K) is drawn as the line A–A'. The predicted quiet-flow Reynolds number decreased to 7.6×10^6 from the previous value of 10.2×10^6 , which had assumed that the wall temperature decreased linearly from the throat to the exit. Clearly, wall temperature is expected to have a dramatic effect, and experimental investigation is needed.

The electroformed throat was plated from a hard-nickel bath that improves polishability and reduces the risk of scratching. Unfortunately, the maximum temperature for the hard nickel was found to be only slightly higher than the stagnation temperature (typically 433K) due to an unexpected state change in this nickel near 470 K ([145], pp. 12–13). At temperatures above 470K, the hard-nickel electroform softens, precluding the use of the throat heating that was predicted to have such favorable effects. However, this does not preclude heating the sections of the nozzle that are farther downstream.

Nozzle sections 4 and 5 are immediately downstream of the electroformed throat and were successfully machined with contour errors of about 0.0013 in. (33 μm) or less, and waviness of 0.001 in./in. or less, except near one joint ([145], pp. 6–8). At the joint between the final cut on section 4 and the final cut on section 5, the waviness increased to near 0.002 in./in. over a short distance of 0.2 in. (5 mm) with a limited total change in contour deviation of only 0.0005 in. (13 μm) or less.

Schneider et al. [145] (pp. 8–12) and Rufer [150] describe the burst-diaphragm apparatus in some detail. Two diaphragms are used to have full control over the break pressure. Three materials cover the range from 35 to 150 psia (240 to 1030 kPa). Schneider et al. [145] also describe the electroforming of the nozzle throat and some details of the bleed-slot vacuum lines. The interior of the electroform was black when removed from the mandrel, which was very disturbing, but the color was caused by a leakage of cutting fluid that did not in the end cause a problem.

Shakedown and Modifications for Quiet Flow

The Mach-6 tunnel was finally completed in spring 2001 [151]. The tunnel successfully ran for about 10 s with low operating costs. However, the flow was not quiet. It then took more than 5 years to shake down the tunnel and achieve quiet flow at high Reynolds numbers; this tried the patience of both researchers and funding agencies. Many different factors can cause early transition, and it was not obvious which was causing the low-Reynolds-number transition on the nozzle walls. Different possibilities were therefore systematically investigated, one by one. At the same time, tunnel operating procedures were being developed along with tunnel instrumentation. The following description provides a brief chronological summary of a long process that is described in detail in a series of theses and AIAA Papers.

Fabrication of the nozzle and contraction took 2.25 years [151], much longer than expected. The original fabrication budget doubled, and then doubled again, as the preliminary design proceeded to detailed design, bids, and fabrication. Measurements of the contour of the electroformed throat were obtained only in a 1.5 in. (38 mm) region near the minimum radius, in part for fear of damaging the mirror finish. These showed contour errors of less than 0.0013 in. (33 μm) ([151], pp. 3–5). The electroform distorted about 0.0015 in. (38 μm) out of round after being removed from the mandrel, apparently due to residual stresses. This created a flaw near the

downstream end of the electroform, which was nearly 0.002 in. (51 μm) high, just within the 0.0028 in. (71 μm) allowed by the $Re_k = 12$ criterion. It could not be honed out due to the difference in hardness between the nickel electroform and the 15-5PH H1100 stainless-steel used for the rest of the nozzle. Thus, it was accepted.

Unfortunately, the bleed-lip contour was also machined into the electroform as if the electroform were round, and the contour of the bleed lip was not measured for fear of introducing flaws into the mirror finish. Measuring the lip involves multiple automated contacts between a stylus and the lip; the consequent risk of scratches is actually not as large as was feared. This critical error was made by the present author during one of many short discussions with the project manager at the fabrication shop. Transition can be very sensitive to small flaws in geometry and fabrication, in a way that is often poorly understood. Limits in budget and schedule do not permit attaining the best-possible quality for every single tunnel element. In the case of the contemplated bleed-lip measurements, finish and cost were thought to be more important than contour accuracy. It took almost five years to realize the critical importance of this failure to measure the as-built bleed-lip contour.

Contour measurements were also reported for nozzle sections 6, 7, and 8 ([151], pp. 6–9). The largest contour errors were within 0.0015 in. (38 μm) with waviness also within 0.0015 in./in., except that some portions of section 8 were oversized by slightly less than 0.002 in. (51 μm). The joints between the sections are barely perceptible to the touch, with steps of less than 0.001 in. (25 μm), even after repeated disassembly and reassembly. Thus, the joint design was very successful.

Initial measurements of the flow in the tunnel showed that the bleed-slot vacuum timing worked well ([151], pp. 10–17). The run lasted nearly 10 s with the initial bleed-slot design. Unfortunately, the pitot-pressure fluctuations ranged from about 4% near 15 psia (103 kPa) stagnation pressure to about 1% near 150 (1034 kPa) psia. No quiet flow was observed. Concern immediately focused on fluctuations that might be caused by separation near the bleed-slot lip. However, measurements from the single pressure transducer located near the minimum in the bleed slot were not sufficient to diagnose the problem [152].

A traverse mechanism was then devised to move probes within the vertical centerplane of the aft end of the nozzle [153,154]. A corrected plot showing the relative timing of the startup pressure traces in the nozzle, suction plenum, and bleed-slot throat is reported ([153], pp. 5–6).

Concern about separation near the bleed-slot lip led to experiments with bleed-slot geometries 1 to 5 ([153], pp. 6–14). The mass flow sucked through the bleed slot was increased from the original 10% to 19%. The last section of the contraction was modified to add larger inserts that enabled a wide variety of bleed-slot configurations.

Pitot measurements near the nozzle exit showed fluctuations falling from about 3% at $P_t = 15$ psia (103 kPa) to 1% at $P_t = 150$ psia (1034 kPa). The substantial scatter might well have been due to condensation effects because several measurements were carried out at reduced driver-tube temperatures. These lower temperatures were the result of operating with only two of the three power supplies that heat the driver tube, while awaiting repair of the third. The pitot-pressure fluctuations fell by nearly a factor of 4 when the stagnation temperature decreased from 170 to 24°C, probably due to temperature-induced decreases in Mach-wave radiation from the turbulent boundary layer on the nozzle wall ([153], Table 3). However, all the measurements yielded noisy flow, although the stagnation pressure was never reduced below 1 atm (100 kPa).

Schneider et al. [153] (pp. 14–18) also report the first surveys of the flow properties near the nozzle exit. The stagnation temperature dropped about 10% during a 10 s run. Both the noise and the Mach number were fairly uniform across the inviscid core, outside of the nozzle-wall boundary layers. Kwon and Schneider [155] reported a detailed stress analysis of a 7×14 -in. (0.18 \times 0.36-m) Plexiglas window to provide optical access to the downstream portion of the nozzle. The Purdue University facility is the first quiet tunnel with optical access within the nozzle.

Schneider et al. [156] report progress with the traversing mechanism and hot-wire instrumentation along with blockage tests with both a slab delta at angle of attack and a circular cone at zero angle of attack. The slab delta at angle of attack caused high levels of unsteadiness in the supersonic flow, apparently due to separation induced upstream by shock/boundary-layer interactions on the nozzle wall. The conventional-noise pressure fluctuations in the nozzle decreased from about 1.6% to about 0.5% as the stagnation temperature was reduced from about 430 to about 300 K, showing that the noise radiated in conventional tunnels depends not only on unit Reynolds number but also on temperature or temperature ratio.

Six possible causes were listed for the early transition on the nozzle wall [156]. The distribution of wall temperature in the nozzle was varied by as much as 30 K using various distributions of insulation, but the flow was still always noisy at a total pressure of 15 psia (103 kPa). A sixth bleed-slot geometry was designed in an attempt to eliminate possible separation-induced unsteadiness. The slot mass flow was increased to 30%, which reduced the runtime to about 9 s from the previous 10 s. Quiet flow was observed in the tunnel for the first time, late in a low-pressure run, when the stagnation pressure had dropped to about 7.5 psia (52 kPa).

Schneider et al. [157] report the design and performance of the case 7 bleed-slot geometry, which increased the suction mass flow to 38%. It also reports the polishing of the entire portion of the nozzle downstream of the electroformed throat. However, the tunnel remained quiet only below 8 psia (55 kPa), with no change in the quiet pressure, although there appeared to be some reduction in the noise under quiet conditions. Under some conditions, separation seems to have occurred in the nozzle-wall boundary layers when they became laminar as the pressure dropped during the run ([157], Fig. 11 and pp. 22–24). Under noisy conditions at a total pressure of 75 or 135 psia (52 or 931 kPa), the Mach number in the nozzle core was about 5.87 with good uniformity. Nine possible causes were listed for the early transition on the nozzle walls.

Schneider et al. [158] report measurements with a pitot mounted halfway between the throat and the nozzle exit. Nozzle-wall transition occurred at about the same pressure both near the exit and halfway upstream, indicating a bypass of the usual linear instability processes and suggesting a problem with roughness or separation near the throat. When the nozzle-wall boundary layer dropped laminar, the second-throat centerbody seemed to induce separation on the nozzle walls upstream, and so it was replaced with a sting-support section that was much more streamlined. Eleven possible causes were listed for the early transition on the nozzle walls.

Schneider et al. [159] report additional measurements with a long pitot placed about halfway down the nozzle. Nozzle-wall transition was again observed at a total pressure of about 8 psia (55 kPa), both near the exit and halfway down the nozzle, indicating a bypass mechanism. The bleed-slot flow was for the first time plumbed directly to the vacuum tank, using a moderately fast valve to time initiation; this had no effect on the pressure at which quiet flow began. A new streamlined sting-support system was installed, eliminating upstream separation effects observed earlier when the nozzle-wall boundary layer dropped laminar. Additional measurements in the diffuser with bleeds open, closed, and throttled began to make clear that disturbances propagating upstream within the nozzle-wall boundary layers were not the cause of early transition on the nozzle walls.

Preliminary measurements of condensation by observing scattering from a helium-neon laser showed that supercooling might permit operation to pressures 50% above the static liquefaction line ([159], pp. 7–8). Preliminary measurements of the flow in the contraction showed small but possibly significant nonuniformities due to free convection before flow initiation.

Schneider et al. [160] report additional measurements of the free convection in the contraction before the run, which creates spatial and temporal nonuniformities that convect into the nozzle after the run begins. Most of the fluctuations appeared to occur near the wall. A small jet of air blowing through the normal fill line into the upstream end of the driver tube did not affect the 8 psia (55 kPa)

stagnation pressure at which the nozzle flow dropped quiet ([160], pp. 9–10). Although it is known from prior experience in the JPL tunnel that leaks in the settling chamber can cause jets of air that precluded quiet flow, leak tests in the Purdue tunnel using both pressure-drop measurements and a helium sniffer did not detect any leaks that seemed likely to preclude quiet flow ([160], pp. 10–12). The first operations with the new boost compressor are also reported ([160], pp. 12–14). This enables operation to the maximum stagnation pressure of 315 psia (2172 kPa).

Small transverse jets introduced via holes in the diffuser wall were able to separate the nozzle-wall boundary layer far upstream ([160], pp. 14–16). However, there was no evidence they could trip an unseparated boundary layer, upstream of the jet (cf. [161]). The effects of temperature on the pressures measured with Kulite pressure sensors were investigated with the help of a temperature-sensing readout from the Kulite diaphragm itself ([160], pp. 16–19). At stagnation pressures below 30 psia (270 kPa), the temperature variations could cause errors in inferred mean Mach number of several percent.

Measurements of the mass-flow fluctuations in the contraction inlet yielded values of 0.13 to 0.19% ([162], pp. 3–7). Although these are significant and larger than desired, they are comparable to the values measured by Beckwith et al. in the successful Mach-3.5 quiet tunnel, and so they were not thought to be the cause of early transition on the nozzle wall [163]. An eighth throat insert was designed to further vary the bleed geometry and remove about 50% of the total mass flow. However, the high slot mass flow exceeded the capabilities of the suction system, and sonic flow could not be achieved in the minimum of the suction slot, a necessary condition for precluding upstream propagation of disturbances in the bleed system piping ([162], pp. 7–9).

A porthole window was designed and built to provide optical access above 158 psia (1087 kPa) stagnation pressure ([162], pp. 10–11). Further measurements with sonic transverse jets and blunt blockage models were again able to generate separation in the nozzle-wall boundary layer upstream, but in all cases, separation occurred before transition did ([162], pp. 11–15).

A detailed and fairly thorough analysis of disturbances generated downstream in the tunnel is presented in [164,165]. At low quiet pressures, to 20 psia (138 kPa) stagnation, the struts in the sting-support section caused separation upstream in the nozzle when the boundary layer dropped laminar. When the sting-support struts were removed, the nozzle-wall boundary layer was laminar and the profiles were in good agreement with computations. With the struts removed, transverse jets from the usual configuration of the throat bleed system would enter the diffuser boundary layer (see Fig. 8 in [149]). Remarkably, these transverse jets caused separation at least 120 in. (3.05 m) upstream in the nozzle. A compression ring was not useful for blocking the upstream propagation of these disturbances (see [165]).

Achieving Quiet Flow at High Reynolds Number

Beginning in 2005, the critical tasks necessary for achieving high-Reynolds-number quiet flow were finally identified. Doyle Knight of Rutgers University and Garry Brown of Princeton University played key roles.

Doyle Knight's group at Rutgers University was kind enough to undertake the task of computing the flow in the bleed slots to identify any separation bubbles that might occur. References [166,167] report computations for the case 7 bleed geometry at 8 and 14 psia (56 and 97 kPa) stagnation pressures and for the geometry of the NASA Langley Mach-6 quiet tunnel at 14 and 150 psia (96.6 and 1034 kPa) stagnation pressures. For five of the six cases that were computed, small stagnation bubbles were observed just behind the bleed lip on both the outer bleed-slot side and the inner main-flow side. For the NASA Langley nozzle at the lower pressure, no separation was observed on either side of the lip [167]. These small separation bubbles became the most likely cause of the early transition in the Purdue nozzle. The NASA Langley nozzle has smaller bubbles and a larger contraction ratio from the bleed lip to the throat; it seemed

possible that these two factors might explain why the NASA Langley nozzle was successful when the Purdue nozzle was not.

Although it had been clear for some time that there was a good probability that the lack of quiet flow was associated with flaws in the bleed-slot flowfield, numerous attempts to modify this flowfield by changing the geometry of the end of the contraction had not yielded much progress. The possibility of modifying or instrumenting the upstream portion of the nozzle itself had been considered, but had been rejected due to the high cost and the long time that had been necessary to develop the precise and highly polished nozzle with its electroformed throat. It was thought to be just too risky to make changes to the nozzle near the throat.

Garry Brown then suggested the key experimental concept: fabrication of a surrogate or alternate throat section, which could be used to replace the original electroformed throat. Aluminum for a simple surrogate throat was procured in early 2004 with the idea of fabricating a low-cost throat in sections. If the larger roughness that was inevitable with this process did not cause excessive difficulty at the low quiet pressures that were of interest, it was thought that this low-cost throat might be modified to insert instrumentation that would help to diagnose the cause of early transition. Because of limited funds, it took until January 2005 to complete the surrogate throat, which was machined on the numerically controlled lathe in the departmental shop. A minimal polish was then produced by the departmental machinist using 2000-grit paper and Scotchbrite.

Surprisingly, quiet flow was obtained at 20 psia (138 kPa) stagnation pressure using this simple aluminum throat ([164], pp. 5–8). This clearly showed there was some defect in the sophisticated electroformed throat. In addition, the quiet-flow pressure fluctuations were about 20% higher using the aluminum throat, and the intermittent region between noisy and quiet flow was qualitatively different for the two throats ([164], Figs. 8, 9). Finally, the quiet pressure for the aluminum throat showed a weak dependence on the time during the run at which that pressure was achieved, a property that had not been observed with the electroformed throat. However, the stagnation pressure at which the tunnel became quiet was still independent of measurement position, both near the nozzle exit and halfway up the nozzle, again suggesting some sort of bypass mechanism.

An aft-facing step with a height of about 0.002 in. (51 μm) was observed at the downstream end of the aluminum throat, at $z = 30.265$ in. (769 mm) [168]. Although this was within the $Re_k = 12$ criteria, it was removed by adding a very small taper beginning at $z = 23.675$ in. (601 mm). Surprisingly again, the quiet stagnation pressure increased to about 37 psia (255 kPa). It appears that the small aft-facing step was generating small streamwise vortices that were amplified by the Görtler instability.

The aluminum throat was then polished professionally, which dramatically improved the surface finish of the interior contour. However, at some point, a small nick in the bleed lip was introduced ([168], Fig. 5). The nozzle was then quiet only at 12 psia (83 kPa) ([168], p. 5). The throat was then repolished again in the department shop, and the lip was hand worked to remove the nick. This improved the quiet pressure to about 34 psia (234 kPa), which was still below the value attained earlier.

A second professional polishing was then carried out on the aluminum throat [168]. This yielded quiet flow at about 95 psia (655 kPa) stagnation pressure, nearly two-thirds of the value expected from the design ([168], pp. 5–7). It was then very clear that the lip geometry as well as the throat finish were both very critical to achieving high-Reynolds-number quiet flow. For all these measurements in the Mach-6 tunnel, the stagnation temperature was held near 160°C.

The bleed lips of the nozzle throats were therefore measured to check the coordinates. The geometry of the aluminum throat was nearly as designed. The nickel throat was then measured as shown in Fig. 17 (here, the axial coordinate is zero at the throat, but the sign is reversed from z , and it increases going upstream). Whereas the coordinates are generally as designed, there is a 0.001 in. (25 μm) kink near the inner shoulder at the 90 deg azimuth (and also, to a lesser extent, near the 270 deg azimuth). This kink was the only

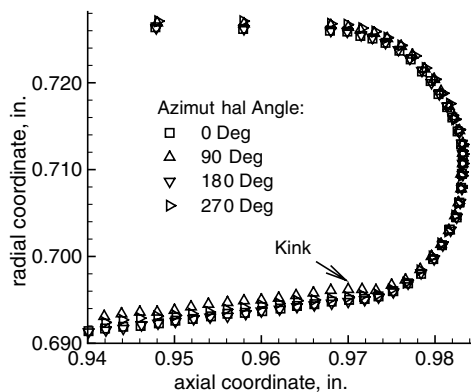


Fig. 17 Measured coordinates of electroformed lip.

known feature in which the nickel nozzle was inferior to the surrogate, and so it appeared to account for the difference in performance.

It appears that this small kink aggravated the problem with separation near the lip and apparently caused the early nozzle-wall transition at 8 psia (55 kPa). The lip of the nickel nozzle had not been measured previously due to concerns about damaging the mirror finish (1 $\mu\text{in.}$ or 0.03 $\mu\text{m rms}$). Schneider et al. [151] show that the electroform distorted about 0.002 in. (51 μm) out of round when it was removed from the mandrel (about 0.1%). Although detailed records no longer exist for the subsequent machining process,** it appears that the electroform was aligned in the lathe as though it were axisymmetric before the tip and upper surface of the bleed lip were machined. The flaw that was then machined into the lip was not detected due to concerns about damaging the surface finish. In hindsight, these concerns were excessive and unwarranted because a high-quality measurement need not damage the finish anyway.

Borg et al. [168] (pp. 7–21) also report additional measurements of the flow in the contraction inlet; although this is less uniform than desirable, it appears to be acceptable. The temperature drop during the run agrees with the isentropic theory ([168], pp. 13–15).

Rutgers University then designed a new bleed-lip shape for the electroformed nickel throat to eliminate the boundary-layer separation [169,170]. Transonic Navier–Stokes computations were carried out on near-semi-elliptical geometries to find one that would 1) eliminate the separation bubbles at all likely operating conditions and 2) involve machining away the kink previously measured in the lip. Time-dependent computations found that the flow was steady when unseparated. The new geometry eliminated separation bubbles up to the maximum stagnation pressure, near 300 psia (2068 kPa).

The aluminum nozzle throat was then rerated for operation to 185 psia (1276 kPa) stagnation pressure because in February 2006, this throat ran quiet to the maximum allowable pressure of 107 psia (738 kPa) [171]. Here, “rerating” refers to a process of analyzing and testing the strength of the nozzle and approving operation to a higher stagnation pressure. After rerating, this throat ran quiet to a maximum stagnation pressure of 130 psia (896 kPa), although this performance was not reliable ([171], pp. 5–9). At 130 psia (896 kPa), the freestream unit Reynolds number is about $2.7 \times 10^6 \text{ ft}^{-1}$ ($8.9 \times 10^6 \text{ m}^{-1}$), and $Re_\theta = 2700$ in the laminar boundary layer at the nozzle exit where $M_e = 6.0$. Again, the flow dropped quiet at 130 psia (896 kPa) both halfway up the nozzle and near the exit. A change in the time allowed to settle the driver-tube air did not affect the noise measured under quiet flow. Schneider et al. [171] also report additional measurements of the temperature distribution in the contraction inlet.

The bleed lip of the electroformed throat was then recut to the new semielliptical shape very carefully ([171], pp. 12–15). The small kink was removed from the lip [172]. The separation bubble that was computed to be present even on a perfect half-circle lip was eliminated for the elliptical profile. Less than 0.010 in. (0.25 mm) of

material was removed over a small region within 0.25 in. (6 mm) of the tip. A series of measurements were then carried out using this electroformed throat before and after a series of polishings, removals, and reinstallations [172–174]. Juliano et al. [172] show that the tunnel then became quiet at stagnation pressures of about 145 psia (1000 kPa).

Juliano et al. [173] (pp. 10–12) also report nozzle-wall hot-film measurements of separation and reattachment under various conditions. During late spring 2007, the maximum quiet-flow stagnation pressure fell to about 80 psia (550 kPa) apparently due to flaws in the throat finish [175]. However, a late-summer repolishing of the nozzle throat restored quiet flow to a stagnation pressure of about 135 psia (930 kPa) [176].

Recommendations for Future Work

Research Needs for Boeing/Air Force Office of Scientific Research Mach-6 Quiet Tunnel

1) Although it is clear that quiet flow has been obtained in this facility, the spatial extent of quiet uniform flow needs to be mapped out for various stagnation pressures.

2) It is clear that wall temperature is expected to have a large effect on nozzle-wall transition. The effect of varying the nozzle wall temperature should be measured experimentally in combination with computations of the temperature distribution and its effect on transition.

3) A new nozzle throat that is machined from stainless-steel sections might result in joint roughness that is still acceptable. Such a throat would permit higher throat temperatures that might also permit marked increases in quiet Reynolds number.

4) The use of a nozzle section that incorporates boundary-layer suction using microperforated walls might enable a new method of obtaining high-Reynolds-number quiet flow. Advanced materials, fabrication processes, and design methods might enable the removal of boundary-layer fluid without introducing excessive disturbances. It seems feasible to test this in the present facility by replacing one of the sections that has conical nozzle walls.

General Needs

Although quiet tunnels now exist to measure cold hypersonic flow for moderate Reynolds numbers at Mach 3.5 and 6.0, much remains to be done. A larger tunnel is clearly needed to obtain high quiet Reynolds numbers on large models with thick boundary layers, and a large Mach-6 quiet tunnel seems very feasible [144]. The use of a large nozzle at moderate stagnation pressures should alleviate the difficulties with roughness in the throat. Because this is a high-risk undertaking that will involve the expenditure of millions of dollars over several years, it is likely to be very difficult to find a source of funds. Mach number effects may be examined via validated computations.

It would be very desirable to develop a quiet tunnel for high-enthalpy flow comparable to reentry flight conditions, and the expansion tunnel has been proposed as a possible approach [177]. However, it appears extremely difficult to maintain a sufficiently smooth nozzle throat, and thus laminar nozzle-wall boundary layers, at the high temperatures necessary to operate at high enthalpy. It will also be necessary to eliminate freestream particulate to maintain this high surface finish, and this also seems very difficult in high-enthalpy flows. If some method of overcoming these difficulties can be developed, it seems conceivable that a quiet nozzle might be developed for the large expansion tunnel recently developed at Calspan—University at Buffalo Research Center (CUBRC) [178].

Lacking a high-enthalpy quiet tunnel, one could pursue quiet tunnels at higher Mach numbers nearer to flight conditions. However, even for cold flow, it will again be difficult to obtain and maintain sufficiently smooth nozzle throats at the high temperatures needed to operate at higher Mach numbers. It would not be easy to improve on the throat design used for the NASA Langley Mach-8 nozzle, which failed due to difficulties with materials and thermal expansion while using the same aerodynamic design methods proven in previous nozzles.

***DeMeno, L., Allied Aerospace, private communication, July 2005.

Finally, new approaches might well be feasible for this hypersonic nozzle-wall laminar-flow-control problem. Because the problem is difficult and expensive, very few workers have attempted solutions. Modern computational tools are far more capable than those that were available when the existing tunnels were designed. Much remains to be learned about the transition process on the nozzle wall, which is poorly understood. A creative new approach, probably using some form of suction, might well lead toward new forms of quiet tunnels that might be more affordable. The successful use of suction in quiet nozzles will require a multidisciplinary development process using new materials with uniform microscopic suction.

Conclusions

The development of quiet-flow hypersonic wind tunnels with low noise levels comparable to flight has been a long and challenging process because it has required control of laminar-turbulent transition on the nozzle walls, the very process the tunnels are built to study. The tunnels are large, complex, and expensive. Controlling transition on the nozzle walls is risky fundamental research that must nevertheless expend substantial funds over long periods to perform the necessary experiments. The NASA Langley Research Center team led by Ivan Beckwith expended many millions of dollars over three decades developing the first supersonic and hypersonic quiet tunnels. Building on these achievements, the Purdue University effort still required more than \$3 million in external support over nearly two decades to develop the first hypersonic quiet tunnel with low operating costs.

Hypersonic quiet tunnels require low disturbance levels in the core flow entering through the nozzle throat. This core flow must also be free of particulate. The nozzle contour must be designed to maintain laminar boundary layers to the maximum feasible Reynolds number. High quiet Reynolds numbers seem to be possible by using axisymmetric nozzles to eliminate crossflow and corner flows, long nozzles with smaller curvature to damp the Görtler instability, axially falling temperatures to damp the first-mode instability, highly polished throats to reduce roughness effects and small nozzle-wall waviness and roughness. All successful hypersonic quiet tunnels have used bleed slots upstream of the throat to remove streamwise vorticity that may be generated in the contraction-wall boundary layer. It is further necessary to eliminate or minimize boundary layer separation in the contraction and near the nozzle lip to eliminate the associated unsteadiness.

Transition has never been measured on a smooth symmetric circular cone or flat plate under fully quiet conditions at supersonic or hypersonic speeds. As stagnation pressure rises, the nozzle-wall boundary layer transitions before the flow on the cone does. Larger and better-performing quiet tunnels will be required to reach this long-sought goal, which was reached at low speeds many decades ago.

The development of hypersonic quiet tunnels promises to remain a very challenging field requiring much patience and substantial funding. However, it should always be remembered that even the most expensive quiet tunnel costs less than a single one-shot hypersonic flight test. With sufficient care, patience, and funding, it is possible to make progress addressing the difficult challenge of controlling laminar-turbulent transition on the nozzle walls of hypersonic quiet tunnels.

Acknowledgments

The development of these complex and difficult facilities over the past 50 years would not have been possible without the support of many people, too numerous to mention here. John Laufer and James Kendall of the Jet Propulsion Laboratory developed the first good understanding of noise effects and made the first quiet-tunnel measurements. Ivan Beckwith and Dennis Bushnell of the NASA Langley Research Center led the development of high-Reynolds-number quiet tunnels for almost three decades. Steve Wilkinson and Ivan Beckwith from the NASA Langley Research Center have helped the present author in many ways over the last 18 years,

passing along some of the information learned during the NASA Langley development program. Steve provided a high-quality scan of Fig. 9. Outstanding support was provided by several critical vendors, including Larry DeMeno et al. from Dynamic Engineering, Inc. (nozzle design and fabrication), Paul Thomas et al. from Optek (polishing), Terry Kubly et al. from Monticello Exchanger (aerodynamic pressure vessels), and Noah Risner et al. from ATK/Microcraft (remachining of bleed lip). Alicia Bagby provided several photographs from the NASA Langley archive.

Interactions with the ONERA quiet-tunnel group played a significant role in identifying the problem with bleed-lip separation in the Purdue University quiet tunnel. Doyle Knight's group from Rutgers University performed critical computations of the nozzle throat flow, Garry Brown from Princeton University suggested the use of a surrogate throat, and Hans Hornung from the California Institute of Technology suggested the use of a Ludwig tube in November 1989. Eli Reshotko from Case Western has supported and encouraged the development of quiet tunnels for more than three decades. Len Sakell, Steve Walker, and John Schmisser provided Air Force Office of Scientific Research funding for the Purdue effort over more than 13 years. Without the patient and visionary support of the Air Force Office of Scientific Research and the other sponsors over the last 50 years, quiet tunnels would not have become a reality.

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R. Kimmel
Associate Editor