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FLOW QUALITY IMPROVEMENT AT MACH 8
IN THE VKF 50-INCH HYPersonic Wind Tunnel B

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FOREWORD

The work reported herein was sponsored by the Arnold Engineering Development Center (AEDC), Air Force Systems Command (AFSC) under Program Element 65402234.

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Flow quality within the 50-in. hypersonic tunnel at a nominal Mach number of 8 is discussed. The nozzle and throat were recently remachined to closer tolerances which resulted in improved test section flow quality. Measurements of the contoured nozzle ordinates and Mach number distributions are presented along with results from method of characteristics solutions of design and actual measured wall contours.
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TABLE

I. Coordinates of Tunnel B (M = 8) ........ 14
NOMENCLATURE

$M$  
Mach number

$M_Q$  
Mach number on axial centerline of tunnel

$p_0$  
Stilling chamber pressure, psia

$p_0'$  
Total pressure behind normal shock wave, psia

$T_0$  
Stilling chamber temperature, °R

$y$  
Ordinate perpendicular to axial centerline of tunnel, in.

$\Delta y$  
Ordinate measured minus design ordinate, in.

$Z$  
Measure of curvature defined in Fig. 4, in.

$\gamma$  
Specific heat ratio

$\delta$  
Boundary-layer thickness, in.

$\delta^*$  
Boundary-layer displacement thickness, in.
SECTION I
INTRODUCTION

The initial calibration of the 50-in. hypersonic Mach 8 tunnel (Gas Dynamic Wind Tunnel, Hypersonic (B)) in 1959 showed that the flow was nearly uniform everywhere excluding a small core in proximity to the axial centerline. At that location a sine-wave-shaped distribution in Mach number with an amplitude several times the off-center deviation was evidenced. This flow situation was quite common to axisymmetric supersonic wind tunnels and was more or less accepted until 1963 when the initial calibration of the Mach 10 tunnel (Gas Dynamic Wind Tunnel, Hypersonic (C)) showed negligible centerline focusing. Of course, Tunnel C had a much thicker boundary layer because of its higher Mach number, lower expansion angle, and increased length. The thicker boundary layer was thought to perhaps absorb any machining errors more readily. With the use of the AEDC high-speed digital computer (IBM 7074), after the formation of Scientific Computing Services, and with the advent of increased precision requirements of wind tunnel testing, it was decided to investigate the disturbance focusing problem in Tunnel B. The problem has been solved to the extent that the flow quality in Tunnel B is as good or better than that of Tunnel C and is discussed herein.

SECTION II
WIND TUNNEL

Tunnel B is an axisymmetric, continuous, variable-density, hypersonic wind tunnel with a 50-in. diam test section. Interchangeable throats provide nominal test Mach numbers of 6 and 8. At Mach 6, the stilling chamber pressure can be varied from 10 to 300 psia. Stagnation temperatures up to 1260°R are available. At Mach 8, the stilling chamber pressure can be varied from 50 to 900 psia at a stagnation temperature up to 1360°R.

Details of Tunnel B and associated equipment are shown in Fig. 1. As illustrated, the tunnel is equipped with a mechanism which can inject a model into the test section for a test run and then retract it into a chamber where model cooling or changes can be accomplished while the tunnel is running.

The aerodynamic design and operation of the tunnel is discussed in Ref. 1. The coordinates of the nozzle were obtained by adding a boundary-layer displacement thickness (δ*) to an inviscid expansion
determined by the method of characteristics. The inviscid design is basically that of Cresci (Ref. 2) where an initial expansion section is required to produce radial flow (that of a spherical source). The downstream portion of the nozzle accepts the radial flow and produces uniform parallel flow at the exit as shown in Refs. 1 and 2. The boundary-layer displacement thickness was determined by the Sivells-Payne method of Ref. 3. Both the Cresci and the Sivells-Payne schemes have been programmed on the AEDC high-speed digital computer (IBM 7074). Table I contains an abbreviated listing of coordinates applicable to the new throat and the remachined downstream contour of Tunnel B.

The test section Mach number distribution was measured using a top-window-mounted, cantilevered rake containing 17 impact pressure probes, 3/32-in. in diameter, spaced 1 in. apart. The rake was carefully aligned so that the center probe was on the geometrical longitudinal tunnel centerline when traversed fore and aft. This was considered important because of the inherent focusing of flow in axisymmetric wind tunnels.

Mach number was calculated from

\[ \text{Mach number} = \left( \frac{p_0 - p_0'}{p_0} \right)^{\frac{\gamma}{\gamma-1}} \left[ \frac{\gamma + 1}{2\gamma M^2 - (\gamma - 1)} \right]^{\frac{1}{\gamma-1}} \]

The specific heat ratio, \( \gamma \), was taken to be 1.4. The precision of \( p_0' \) and \( p_0 \) measurements was estimated to be within 1 percent of the reading.

**SECTION III
RESULTS AND DISCUSSION**

The aerodynamic design and the flow quality associated with the 50-in. hypersonic tunnel is discussed in Ref. 1. The reference indicates that along the axial centerline of the tunnel in an area of frequent model testing, there was a nonuniformity in Mach number of approximately 0.3. Figure 2 illustrates the centerline Mach number distribution near the midpoint of the 54-in.-long test section (\( p_0 = 600 \) psia). This nonuniformity was confined to the tunnel centerline and was shown (Ref. 1) to be somewhat detrimental to local measurements on some wind tunnel models.

The axisymmetric Tunnel B nozzle is fabricated in several sections (Fig. 2): a throat section, five contoured sections, and a cone-frustum.
test section. The centerline focused nonuniformity of concern occurs at station 270. Characteristics associated with the disturbance (obtained from the design inviscid flow field) are illustrated in Fig. 2. If a contour error were to exist, it would have to be either near station 120 or in the throat section as shown in the figure.

The first step in seeking flow improvement was to build a new throat section as part of a general maintenance and modernization program as indicated in Ref. 1. The old throat had a radius of curvature to throat radius ratio of 5, and was defined by an analytical expression (a cubic for the entrance and a semicubic for the supersonic portion, Ref. 1). The new throat section was designed to have a radius of curvature to throat radius ratio of 20 and was defined by (1) the method of characteristics (Ref. 4) where the flow was supersonic, (2) transonic theory (Ref. 5) at the throat, and (3) a cubic for the entrance region. These changes in aerodynamic design of the throat region were believed to give better assurance that radial flow would be achieved at the inflection point, and the improved flow experienced at Mach 10 (Ref. 1) was partly attributed to the larger radius of curvature.

As shown in Fig. 2, however, the replacement of throat sections did not improve the flow uniformity. A slight over-all level shift was caused by a small change in throat diameter. The conclusion then was that an axisymmetric error existed in the contour near station 120. Figure 3 shows the results of a simple experiment where an axisymmetric disturbance, consisting of three layers of stairstep-fashioned pressure-sensitive tape (Scotch® No. 33), was affixed to the wall at station 117 (a fabrication joint). Centerline Mach number disturbance with an amplitude of five times that obtained with no tape resulted (Fig. 3), indicating that a contour error of a few thousandths of an inch was probably present in this vicinity.

Since absolute ordinate measurements of the nozzle near station 120 would be difficult to obtain without tunnel disassembly, a simple curvature gage was used to inspect the contour. The gage was 24 in. in length, with a dial indicator at the midpoint. Two legs at each end (1 in. apart) insured that the gage was always aligned with the tunnel centerline. The results of these measurements along with the calculated desired curvature are shown in the lower curve in Fig. 4. The measurements were repeatable to 0.001 in.

Although Fig. 4 shows only relative changes along the nozzle, it does imply that the contour, when assembled, was not within the specified ±0.002 tolerance. Based on these measurements along with some diametrical measurements, it was decided to remachine the contour (excepting the throat section) to as close a tolerance as possible with the available equipment. Curvature measurements after machining,
given in the upper curve in Fig. 4, indicated a much better contour than before. Also, additional absolute ordinate measurements from station 60 to 160 showed that the contour was now within the ±0.002-in. tolerance.

Figure 5 shows the final centerline calibration with a total deviation of 0.09 in Mach number, whereas the deviation before machining was 0.24. The off-center Mach number is within 0.02 compared to 0.04 prior to remachining. Over-all axial and lateral gradients proved to be essentially nonexistent in the test region.

The method of characteristics using the AEDC digital computer was the basic tool for obtaining the inviscid outermost streamline onto which the turbulent boundary-layer displacement thickness was added. Given this inviscid streamline and an initial line of known flow (from transonic theory), the computer can generate a characteristics network, thereby defining the downstream flow. The results of two solutions of this type are presented in Fig. 6a. The first computation was applied to the inviscid design contour to serve as a basis of comparison for the second. As expected, the computed flow is shown to be very close to that desired. The second computation retained the identical inviscid boundary from the throat to station 109. From station 109 to 146, however, deviations of the measured wall from the design wall were applied to the inviscid outermost streamline, just as if the boundary layer conformed to the wavy wall but provided no cushioning. The top curve in Fig. 6a shows the streamline perturbation $\Delta y$ in a region $AB$ which has a direct effect on $M_q$ in region $A'B'$. The points presented on the two curves between $AB$ and $A'B'$ are in correspondence with respect to characteristics. The computed centerline Mach number distribution based on the perturbed streamline is in fair agreement with the measured Mach number, thereby indicating that small contour deviations can produce centerline nonuniformities on the order of those measured.

Thus far only test section centerline flow has been discussed. As mentioned previously, off-centerline flow is highly uniform. Figure 6b shows a lateral distribution at station 270 where, according to Fig. 6a, the centerline Mach number deviates most from that desired. The agreement between calculated and measured Mach numbers is considered excellent within the 32-in.-diam usable core. Of course, outside of the 32-in. test core a real boundary layer of thickness $\delta$ cannot be compared to the fictitious, but useful, concept of divorcing viscous and nonviscous flows. The reason for the drop in Mach number near the wall in the computed curve is the fore-shortened nozzle. Lateral distributions at all stations are typically the same except for less severe centerline focusing.
All measurements and calculations in this report are based on a
design condition of a fixed stilling chamber pressure and temperature
at Mach 8, namely 600 psia and 1310°F, respectively. The tunnel is
being operated successfully over a stilling chamber pressure range
of 50 to 900 psia. The quality of flow within this range is essentially
the same as that presented, except for an over-all level change in
Mach number. The average Mach number is 7.85 at 50 psia, 8.00 at
600 psia, and 8.02 at 900 psia.

SECTION IV
CONCLUSIONS

Based on this investigation of the flow in Tunnel B, the following
results were obtained:

1. The semicubic throat discussed in Ref. 1 produced flow of
equal quality to that of a more exotic aerodynamically
designed throat with a larger radius of curvature.

2. Close tolerance machining is important, at least to point B
(Fig. 6a), to keep centerline focusing to a minimum.

3. Masking or cushioning of the turbulent boundary layer is
apparently small, with regard to machining errors of the
type illustrated, even though the boundary layer is thick
compared to the error.

4. The turbulent boundary layer scheme of Ref. 3 apparently
predicted δ* extremely well at all stations as evidenced
by the level and lack of gradients in Mach number at design
conditions.

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for Test Section Mach Numbers of 8, 12, and 20." WADC


a. Tunnel Assembly

b. Tunnel Test Section

Fig. 1 Tunnel B
Fig. 2 Test Section Centerline Calibration with Old and New Throat Sections
Fig. 3 Test Section Centerline Calibration with an Artificial Disturbance
Fig. 4 Relative Smoothness of Nozzle before and after Remachining
Fig. 5 Test Section Centerline Calibration before and after Remachining
Fig. 6 Computer-Analysis of Centerline Focusing

a. Longitudinal Distribution
O Measured Wall/Computed Flow
△ Measured Flow

b. Lateral Distribution
Fig. 6 Concluded
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