Flow field calibration of a newly developed Laval nozzle in 2.4m transonic wind tunnel

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Abstract. The range of Mach number of 1.3 to 1.5 is defined as low-supersonic flow in wind tunnel testing, which is necessary range for major fighters, aerospace vehicles, and missiles to reach the cruise state. It is also the most difficult test range for conventional ground simulation equipment due to sound barrier and uniformity of artificial air flows. In order to expand the test range of 2.4m transonic wind tunnel, researches on low supersonic flow filed calibration was carried out, and the geometrical parameters of perforated walls in test section were determined. Measured results of flow uniformity are compared with advanced wind tunnels using cross-shaped rake and cone-cylinder model. The results show that uniformed low supersonic flow in model testing area was obtained using newly developed nozzle together with perforated walls in 2.4m transonic wind tunnel, which effectively improve the simulation capability of the facility. Together with the capacity of 2m supersonic wind tunnel, seamless link of test data has been achieved in high speed regime, which has formed an integrated test platform for sub-, trans- and supersonic speeds on the order of 2 meters.

1. Introduction
When the aircraft is flying in the transonic range ($M=0.80\sim1.40$), its aerodynamic characteristics are very complicated, and it is very sensitive to the test simulation parameters (Mach number, Reynolds number, etc.) and test conditions. Therefore, the accuracy of transonic range data has long been a bottleneck affecting the development of flying weapons [1]. Advanced aerospace power countries attach great importance to the design of the aircraft and the evaluation of its aerodynamic characteristics in the range of $M=0.8\sim1.4$. Under this urgent need, aerospace powers such as Europe and the United States have built a number of 2m order transonic wind tunnels with low supersonic testing capabilities [2, 3].

As the largest transonic wind tunnel in Asia, the highest test Mach number for 2.4 m transonic wind tunnel is only up to 1.2. Compared with the 2.0m supersonic wind tunnel ($M=1.5\sim4.25$), there is a simulation blank in the transonic region, and the data cannot be closely connected, which limits the full advantage of the large size of the equipment [4-6].

This paper describes the low supersonic flow field calibrating test of the 2.4m transonic wind tunnel, and compares and analyses with the advanced wind tunnel test results, in order to provide data support for the application of low supersonic flow field in client tests.

2. Test equipment and procedure

2.1. Wind tunnel
The 2.4m transonic wind tunnel is an ejector-driven semi-circuit, intermittent facility. It has a 2.4m square test section, which is capable of operating over a Mach number range of 0.3 to 1.2 using a
conventional convergent nozzle. The stagnation pressure is controlled by a circuit exhaust system, equipped with four main exhaust valves, and the control error is less than 0.3%.

Figure 1. Flow sketch in transonic wind tunnel.

The test section is nominally 7.0m long and 2.4m square. Its top and bottom walls are adjustable in the range of 0 to 0.5 degrees to compensate the boundary layer developing along the longitudinal direction on the walls. All four walls are equipped with slanted holes providing overall porosity of 4.30% in order to decrease the model blockage interference in subsonic region and to minimize wave reflection from solid and open boundaries in transonic region. The Mach number is controlled by the movable fingers mechanism located downstream of the model support system, which have a symmetric, truncated airfoil shape and capacity of extending to restrict the flow and control model area Mach number in transonic region. The Mach number control precision is 0.002~0.003. Figure 1 demonstrates the names of functional sections and the typical flow in a transonic wind tunnel.

2.2. M1.40 convergent-divergent nozzle

2.4 m wind tunnel M1.40 convergent-divergent nozzle has two-dimensional solid block configuration. The profile curve is divided into front part and back part. Sonic at throat is proper developed to form the spring at the turning point, and the supersonic flow is formed into a uniform parallel in the latter stage. The curve of front part of the nozzle is:

\[ y = y^* + \left(\frac{\tan \theta_A}{x_A}\right)x^2 \left(1 - \frac{x}{3x_A}\right) \]

\[ x_A = \frac{3}{2}(y_A - y^*) \cot \theta_A \]  

The back curve is:

\[ x = x_A + r \cos \theta - r_A \cos \theta_A + l_1 \cos(\theta + \mu) \]

\[ y = r \sin \theta + l_1 \sin(\theta + \mu) \]

\[ l_1 = (\theta_A - \theta)Mr \]  

where \( \theta_A \) is the initial expansion angle of point \( A \) located at the turning point. The nozzle structure is shown in Figure 2.

2.3. Test procedure

2.3.1. Flow uniformity. Low supersonic flow uniformity is expressed by three indicators calculated from Mach number distribution in model testing area obtained using static pressure probe or cross shaped rake: the averaged Mach number \( M_{cp} \), standard deviations \( \sigma_M \), and longitudinal gradient \( dM/dx \) are calculated through equations (3) to (5).

\[ M_{cp} = \frac{1}{n} \sum_{i=1}^{n} M_i \]  

(3)
\[
\sigma_M = \sqrt{\frac{1}{n-1} \sum_{i=1}^{n} (M_i - M_{cp})^2}
\]  
(4)

\[
\frac{dM}{dx} = \frac{n \sum_{i=1}^{n} x_i M_i - \sum_{i=1}^{n} x_i \sum_{i=1}^{n} M_i}{n \sum_{i=1}^{n} x_i^2 - (\sum_{i=1}^{n} x_i)^2}
\]  
(5)

where \(M_i\) and \(x_i\) are Mach number and longitudinal coordinate for \(i\)-th measurement point, \(n\) is the total number of measurement points located in model test area.

2.3.2. Flow deviations in Centerline. Mach number distributions along centreline of test section are measured using static pressure probe, which has 256 static pressure taps located on the cylinder surface \([7, 8]\). Combined with the total pressure \(p_{\infty}\) measured in the settling chamber, the local Mach number \(M_i\) is calculated using isentropic relation:

\[
M_i = \sqrt{\frac{2}{\gamma - 1} \left(\frac{p_{\infty}}{p_i}\right)^{\frac{\gamma-1}{\gamma}} - 1}
\]

(6)

where \(p_i\) is \(i\)-th static pressure on probe, \(\gamma\) is specific heats ratio with value of 1.40 for normal air.

The installation of the probe in the test section is shown in Figure 3.

2.3.3. Flow deviations in cross-section. Unlike subsonic flow fields, the perturbation in the supersonic flow field propagates inside the Mach cone. The Mach number distribution along the test section centerline is not enough to represent the flow uniformity of the test section. Therefore, in addition to measuring the Mach number distribution in centerline, the spatial Mach number distribution must also be measured.

In the calibration process, the sectional Mach number distributions in 11 axial positions (cross-section spacing 240 mm) in the range of 2850 mm to 5250 mm from the inlet of the test section were measured using a cross-shaped rake. The front end of the rake was installed with 43 Pitot tubes to measure the total pressure behind the shock wave, which was used to calculate the local Mach numbers through normal shock relation \([9]\):

\[
\frac{p_{02,j}}{p_{01}} = \left[\frac{(\gamma + 1)^2 M_{1,j}^2}{4\gamma M_{1,j}^2 - 2(\gamma - 1)}\right]^{\frac{\gamma}{\gamma-1}} \cdot \left(1 + \frac{\gamma - 1}{2} M_{1,j}^2\right)^{\frac{\gamma-1}{\gamma}} \cdot \frac{1 - \gamma + 2\gamma M_{1,j}^2}{\gamma + 1}
\]

(7)

where \(p_{02,j}\) is \(j\)-th Pitot pressure measured behind the shock wave. Equation (7) is called the Rayleigh Pitot tube formula, which relates the measured Pitot pressure \(p_{02,j}\) and the free stream total pressure \(p_{01}\) to Mach number \(M_{1,j}\). It should be noted that in the calibration process, the free stream total pressure \(p_{01}\) is measured in settling chamber. The installation of the rake is shown in Figure 4.

The local Mach number \(M_i\) in equation (3) is calculated using measured results from cross shaped rake:
\[ M_i = \frac{1}{m} \sum_{j=1}^{m} M_{1,j} \]  \hspace{1cm} (8)

where \( m = 43 \) is the total number of Pitot tubes of the rake.

2.3.4. Wave reflections on the walls. In order to analyse the wall reflections of compression and expansion waves induced by the model, the surface pressure distributions of the 20° cone cylinder model with a blockage of 0.5% was measured. Pressure distributions with identical model obtained in the 16 feet (4.877m) transonic wind tunnel of AEDC were used as interference free data to compare and analyse the wave suppression effect of 2.4m transonic wind tunnel wall on the low supersonic flow field. The installation of the cone cylinder model in the test section is shown in Figure 5.

![Figure 5. Cone cylinder model installation.](image)

3. Discussion

According to the previous researches in 2.4m transonic wind tunnel, the influence of the parameters of the perforated walls in the low supersonic range on the centerline uniformity was analysed [10]. Based on the results, the set of parameters of the test section for the flow field of \( M_{1.40} \) are shown in Table 1.

| \( M \)   | Wall angle | Porosity (2.8m–4.7m) | Fingers |
|-----------|------------|-----------------------|---------|
| 1.35–1.39 | 0°         | 5.2%                  | 0mm     |
| 1.40–1.42 | 0.3°       | 4.3%                  |         |
| 1.43–1.45 |            |                       |         |

3.1. Uniformity of centerline flow

Centerline Mach number distributions ranged from 1.35–1.43 measured using static pressure probe are shown in Figure 6, in which model test region are marked with black dotted lines.

![Figure 6. Centreline Mach number distributions.](image)
Results demonstrate that under the current equipment parameters, the Mach number in the model area has reached the target value, forming a uniform low supersonic flow field. Table 2 shows the calculated uniformities in the model area. In the range of $M=1.35$–$1.45$, the standard deviation of Mach number distribution is about 0.0110, which has met the client test requirements. The magnitudes of longitudinal gradient for all test conditions are less than 0.0040, indicating that the effect of buoyancy drag caused by the non-uniformity of the flow field is small and can be ignored for moderate models.

**Table 2.** Centerline uniformities in $M=1.35$–$1.45$.

| $M$   | $M_{cp}$ | $\sigma_M$ | $dM/dx$ |
|-------|----------|------------|---------|
| 1.35  | 1.35231  | 0.01110    | -0.00039 |
| 1.36  | 1.36145  | 0.01078    | 0.00023  |
| 1.38  | 1.37865  | 0.01024    | 0.00342  |
| 1.39  | 1.38848  | 0.01152    | 0.00363  |
| 1.40  | 1.40590  | 0.01212    | 0.00391  |
| 1.42  | 1.42204  | 0.01170    | -0.00345 |
| 1.43  | 1.42999  | 0.01076    | -0.00101 |
| 1.44  | 1.44009  | 0.01000    | -0.00073 |
| 1.45  | 1.45010  | 0.01112    | 0.00119  |

3.2. *Sectional distribution of Mach number*

The sectional distribution of Mach number in the model area is verified by the cross shaped rake. The Mach number distributions of each section in the horizontal direction and the vertical direction are shown in Figure 7 and Figure 8, respectively. The average value of the Mach number, the standard deviation and axial gradient are shown in Table 3.

![Figure 7. Sectional distribution of Mach number in horizontal direction.](image1)

![Figure 8. Sectional distribution of Mach number in vertical direction.](image2)

The results show that the deviation of sectional Mach number in model area are less than 0.003, and there is no obvious wave interference in the flow field. Therefore, the $M1.40$ nozzle is capable of producing uniform low supersonic flow field, and the chosen structural parameters are reasonable. The Mach number distribution is uniform, and the influence of the boundary layer and the wave reflection interference of the perforated wall are effectively controlled.

In order to compare the uniformity of the low supersonic flow field, the Mach number distribution in centerline is compared with the convergent-divergent nozzle of the IAE TTP transonic wind tunnel in Brazil [11]. The result is shown in Figure 9 from the perspective of Mach number uniformity, the
accuracy of average Mach number and standard deviation of the 2.4 m transonic wind tunnel are better than the TTP transonic wind tunnel.

### Table 3. Sectional distribution of $M=1.43$

| $x$/m | $M_{cp}$ | $\sigma_M$ | Uniformities |
|-------|----------|------------|--------------|
| 2.85  | 1.43021  | 0.00970    |              |
| 3.09  | 1.43007  | 0.01072    |              |
| 3.33  | 1.42640  | 0.00954    |              |
| 3.57  | 1.43017  | 0.01032    | $M_{cp}=1.42844$ |
| 3.81  | 1.43063  | 0.01121    |              |
| 4.05  | 1.43269  | 0.00962    | $\sigma_M=0.01076$ |
| 4.29  | 1.42032  | 0.01146    |              |
| 4.53  | 1.42907  | 0.01019    | $dM/dx=-0.00119$ l/m |
| 4.77  | 1.42873  | 0.01440    |              |
| 5.01  | 1.42714  | 0.01075    |              |
| 5.25  | 1.42739  | 0.01043    |              |

### 3.3. Wave reflection suppression

In order to analyse the wave reflections on the wall, the pressure distributions on surface of the cone cylinder model is measured. Figure 10 shows the results. For comparative analysis, the surface pressure distribution of the cone cylinder model with the same shape and blockage the AEDC 12-inch (0.305 m) transonic wind tunnel [9] (60° slant hole wall, opening and closing ratio 6.0%) is also shown in picture.

Results illustrate that, in the region between initial shock wave and the shoulder expansion waves, the experimental and interference free data are very close, and the entire compression wave in this region is well suppressed by the perforated wall. The test results fluctuate slightly in downstream region, but the difference does not exceed 1.50% of the total pressure. Therefore, under the chosen wall structures, the compression and the expansion waves induced by the model in the $M=1.43$ flow field can be well suppressed. The wave-eliminating characteristics are close to those of the AEDC 12-inch transonic wind tunnel in the United States.

![Figure 9. Comparison of Mach number distributions with IAE TTP.](image)

![Figure 10. Pressure distributions on surface of Cone-cylinder model.](image)

### 4. Conclusions

The low supersonic flow field calibrating of the 2.4m transonic wind tunnel was completed. The method and process of parameter determination and optimization in the low supersonic flow field calibrating of the perforated walls, the uniformity of the sectional Mach number distributions, and the waves suppression characteristics area are proposed. According to the results, the operational range of 2.4m...
transonic wind tunnel is expanded, and a sub-, trans-, and supersonic integrated ground test platform is formed together with the 2m supersonic wind tunnel, which provides a wide range of test Mach numbers for model tests.

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