Steady flow quality assessment of a modified transonic wind tunnel

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Abstract

An existing operational trisonic wind tunnel is upgraded to improve its performance criterion in the transonic regime. In this research, the test section is modified according to the operational requirements of the various existing transonic wind tunnels. Several perforated walls are designed, manufactured, and installed on the top and bottom sides of the test section. The flow in the test section of the wind tunnel is surveyed for the empty condition prior to testing models. Once satisfactory results regarding the flow quality requirements in the test section under various conditions were achieved, a 2D model, NACA 0012, and a 3D standard model for the transonic wind tunnels, AGARD-B, are manufactured and tested under various conditions for the purpose of integral calibration and validation of the tunnel data. Surface pressure distribution as well as the force and moment data compare well with the existing data from other tunnels for similar models tested under the same conditions.

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1. Introduction

All vehicles that fly supersonically must pass through transonic regime, which is considered one of the most critical phases of their flight. This is due to the formation of normal waves on parts of their surfaces; wing, body, etc., that will cause loss of lift followed by a sharp increase in drag force. These losses must be compensated for increasing the engine thrust. Up to now, various available computational methods are not yet able to predict vehicle performance when operating in this critical flight regime. Therefore, wind tunnel tests are still the most reliable means for predicting aerodynamic forces and moments at transonic speed. However, the aforementioned method has its own difficulties and drawbacks too. When a model is inserted in a transonic wind tunnel with solid walls, it is probable that the blockage will choke the flow near the model in the test section. The accelerated flow over the model surface usually terminates with a normal shock over the model surface. This normal shock will interact with the wind tunnel wall and its reflection might coincide with the model surface again, a phenomenon that will affects aerodynamic force and moment, considerably. Moreover, it may cause flow separation and early stall.

In 1948, NACA researchers demonstrated that a wind tunnel with partially open test section walls could be used not only for extending the choking Mach number of a subsonic tunnel but also for testing at and above a Mach number of unity. In a later study at Cornell University, a wall with discrete openings instead of longitudinal slots was used for the same purposes. The satisfactory cancellation of two-dimensional shock waves was demonstrated at UAC in 1951 using a wall with discrete openings drilled normal to the test section walls. In these studies, optimum results were obtained with walls in which the ratio of open to total area was 22%. Later studies at both the Ohio State University and the Arnold Engineering Development Center (AEDC) demonstrated the effectiveness of poros in cancelling the bow shock reflection from a three-dimensional model as well. These studies were used as a basis for the design, and for the initial sets of walls for the 16-Foot Transonic Circuit of the Propulsion Wind Tunnel (PWT) at the AEDC. Similar studies were conducted to improve the expansion wave cancellations in the 1-Foot Transonic Tunnel at the AEDC in 1954. The tunnel was equipped with perforated walls of 22% porosity. These studies led to the development of a
wall of 6% porosity in which the perforations were slanted 60° from the norm. Significant improvements were achieved in the reduction of expansion wave reflections from the test section walls while good compression wave cancellation properties were retained, particularly at Mach numbers near 1.2. Similar work was continued in the Mach number range from 0.95 to 1.15 and above 1.20 to determine the porosity requirements for walls with slanted holes for Mach numbers different from 1.20 [1–10].

To improve the performance of classical methods in transonic wind tunnel testing with perforated walls, combinations of perforated walls with slotted ones or adaptive walls were used in a few tunnels. In a 2ft by 2ft wind tunnel at the NASA Ames Research Center, side suction on the walls is applied through plenums which are divided into 32 small chambers to control the side suction of every section independently [11–14].

In order to take advantage of these favorable characteristics of perforated walls for converting a 60 cm*60 cm test section wind tunnel that was operating at Mach numbers ranging from 0.4 to 0.75 and 1.4 to 2.5, the test section walls were changed. The flow field in the newly designed and manufactured test section was surveyed and was calibrated to validate the predicted improvements. Furthermore, the pressure distribution at the centerline of the wind tunnel was measured by means of a long tube probe.

After designing and manufacturing the perforated walls, a series of tests were conducted to survey the efficiency of the walls and the side suction system, maximum attainable transonic Mach number, and Mach number distribution in the test section of the wind tunnel with new facilities. All tests were performed under empty test section conditions. The pressure and Mach number distribution in the test section and along the nozzle were measured using a specially designed long tube rake. In addition, two standard models that were tested in several transonic wind tunnels (information about their surface pressure distribution, force and moments are available) were tested and the acquired data are compared with existing ones. The calibration results are presented in Ref. [12]. In this paper, however, force and moment data of two different models are compared with the existing data from other tunnels. This information is used as a baseline for future tests.

2. Experimental facilities and instrumentation

Various equipment was used in this investigation. All of them except the wind tunnel were designed and manufactured specifically for these tests. In the following sections, a brief description of some of this equipment is presented.

2.1. Wind tunnel

All tests were performed in a wind tunnel with a test section of (60 cm(W) ∗ 60 cm(H) ∗ 150 cm(L)). The tunnel is of open circuit suction type. Two turbofan engines that eject their exhaust gases downstream of the test section through ejector systems supply the main circuit power of the tunnel. Side suction from the perforated walls is supplied by a smaller turbofan engine. Figure 1 shows the schematic of the wind tunnel. Both sidewalls are solid and there are three types of interchangeable upper and lower wall; closed, normal perforated and inclined perforated walls, [1,2]. Two plenum chambers are installed above the upper and at the bottom of the lower walls of the test section.

2.2. Standard models

Both static and total pressure distribution along the nozzle and tunnel tests section at various Mach numbers, 0.8 ≤ M∞ ≤ 1.2, were measured. From these data, velocity and Mach number distributions along the test section were calculated [1,2]. In this paper, the results for two standard models that were tested in various tunnels and their data are used as a baseline and are compared with available data. As mentioned, the tunnel is utilized for 2D and 3D tests. Thus, for this part of the tests 2D and 3D models are selected for integral calibration of the tunnel. The following criteria were considered in the selection of these models:

- Compatibility of models with the transonic regime
- Frequency of reported data
- Diversity of test conditions
- Similarity of referenced wind tunnels with present tunnel
- Ease of manufacturing.

From the above criteria, a NACA 0012 airfoil was selected as a 2D calibration model and AGARD-B as a 3D model.

2.3. 3D Calibration model (AGARD-B)

There are several 3D models which have been tested at transonic speeds, ranging from conventional simple rockets to high-tech complicated airplanes such as F-18 and F-22. One of the most popular transonic models which is specifically designed by the AGARD group for calibration of transonic wind tunnels is called AGARD-B. This model has a delta wing with a span four times its body diameter. It has a cylindrical body of revolution with an ogive nose Figure 4. The designed and manufactured model for the present tunnel has a base diameter of 33.2 mm corresponding to a blockage ratio of 0.29% when set at zero degrees angle of attack. However, it should be noted that the blockage ratio is usually considered for the highest angle of attack; which for the present model is lower than 0.5%, which is compatible with the limitations proposed in Ref. [7].

2.4. 2D Calibration model (NACA 0012)

Among 2D models, the most distinguished one is the NACA-0012 that is used for calibration purposes in various flow regimes; (subsonic, transonic and supersonic models along with various CFD results for all cases are available in the literature). Consequently, a variety of data from pressure distribution to force and moment results for this airfoil are available which can be used for calibration purposes. The designed model for the present tests had a span of 600 mm,
Figure 1: The present wind tunnel main circuit.

which is equal to the width of the present test section. The model chord is equal to 150 mm that results in $b/c = 4$. There are 23 pressure taps on each side of the model that can scan the pressure distribution over both surfaces. The pressure ports are located on a $30^\circ$ slanted line with respect to the model chord to provide more space for drilling and further to reduce interaction between the ports. In addition, eight pressure ports were located along the upper surface span of the model at four different locations. Figure 5 shows a schematic of the corresponding model as well as the pressure ports that are located on the upper surface in both chordwise and spanwise directions.

3. Preliminary tests

The flow field in the test with the newly implemented walls was investigated for two different cases; empty test section and one with a long tube probe. The measured data included Mach number distribution along the centerline of the test section obtained from the long tube pressure taps, 3D distribution of Mach number in the test section acquired by a rake and the Mach number along the side walls of the tunnel. Figures 2 and 3 show variations of the Mach number along the test section for closed and normal perforated wall cases. Similar data are obtained for the inclined walls too, but are not presented in this paper. As seen from these figures the standard deviation of this parameter is less than one percent for Mach numbers up to 1.05 which is satisfactory.

These results showed that with solid walls, maximum attainable Mach number in the test section of the tunnel is about 0.85, Figure 2. However, when the normal perforated sidewalls are installed in the test section, the free stream Mach number was increased to about 0.95, Figure 3. With the exertion of the side suction, maximum Mach number was about 1.18. The effects of porosity and side suction were optimized when the normal perforated walls were replaced by those with $60^\circ$ slanted holes. In this case, the maximum attainable Mach number in the test section was increased to 1.25.

For further information, please see Refs. [1,2].

4. Results and discussion

Each model has been tested at different angles of attack and at different free stream Mach numbers for calibration purposes. In the following sections, the results for various cases will be presented and are compared with available data from other sources.

4.1. AGARD-B tests

The 3D model, illustrated in Figure 4, was used to provide forces and moment data; lift, drag and pitching moment. Note that since the sideslip angle was set to zero, there is no need for presenting lateral force and moments. In addition, only one pressure port was used for the base pressure sensing. The model was tested at different free stream Mach numbers ranging from 0.51 up to one. In each set of tests, the angle of attack was
show large deviations, Figure 7(c). The acceptable lift results
agreement for all cases. However, the pitching moment data
forces and moment as well as the performance of
varied from $-6^\circ$ to $16^\circ$ with steps of $2^\circ$ and the corresponding
forces and moment were recorded. The lift, fore body drag and
pitching moment coefficients were then calculated, corrected,
and are presented in Figures 6 and 7. The results are compared
with the results of the CSIR Laboratory [10], the T-38 wind
tunnel [6], AEDC [11], and the NAL 4’ × 4’ trisonic wind tunnel,
In these tests, the effect of angle of attack on the
aerodynamic forces and moment as well as the performance of
the model; $C_l$ vs. $C_D$, are studied. The results show that for Mach
numbers below 0.73, the flow over the entire model surface is
subsonic. For these cases, wall porosity was zero, similar
to solid walls, hence no suction was applied. All data, except
those for $C_m$ vs. $\alpha$, are compared with those of NAL. Further,
Mach standard deviation of the measured data is lower than one
percent, which is acceptable for this type of wind tunnel [1,2].
The slight difference between the present data and those of
NAL is due to the Reynolds number and free stream turbulence.
The differences in the acquired data are more pronounced for
the $C_D$ and $C_m$ data which are more sensitive to the free stream
turbulence and Reynolds number. From these data, it is clearly
seen that the deviation between the present data and those of
various references for lift cases starts around $\alpha > 12^\circ$ while for
$\alpha < 12^\circ$, the $C_l$ data are comparable with those available in the
references. For $\alpha > 12^\circ$ apparently, the flow over a portion of
the model is separated and as is known separation is a function
of Reynolds number which is different from Reynolds numbers
cited in reference cases. Furthermore, free stream turbulence
level has a significant effect on separation, too.
Aerodynamic forces and moments for other Mach numbers,
where the flow was fully subsonic were also acquired, but
are not presented in this paper. From these data, it could be
concluded that for these ranges of free stream Mach numbers
and for the present model, AGARD - B, with blockage ratio close
to the current case, there is no need for side suction to be
applied.
For higher free stream Mach numbers, $M_\infty > 0.8$, both
side suction and wall porosity must be applied to avoid choking
and shock wave formation as well as shock reflection from the
wall over the model surface. The model is further tested at free
stream Mach numbers ranging from, $0.81 < M_\infty < 1$. For
all cases, both side suction and porosity were applied and the
data are compared with those of other tunnels. For similar cases
as seen from Figure 7, variations of $C_l$ vs. $\alpha$ and $C_l$ vs. $C_D$ for
$M_\infty = 1$ are in good agreements with similar cases obtained
from different tunnels except for those of the NAL tunnel. It is
seen that $C_l$ vs. $\alpha$ data for the NAL tunnel for $M_\infty \approx 1$ are slightly
higher than the present data. The differences seems to be due
to a small angle of attack that has shifted the NAL tunnel data
(i.e. at zero degrees angle of attack for the present model which
is symmetric, there should be no $C_l$ while for NAL data a small
shift is present). It seems that for low angles of attack, the slope
of the $C_D$ vs. $\alpha$ diagram could be used as a comparison for all
cases and as seen from Figure 7(a) the values of $C_m$, are in good
agreement for all cases. However, the pitching moment data
show large deviations, Figure 7(c). The acceptable lift results
indicate that the applied suction and porosity of 2.5% for these
cases are optimized. However, the pitching moment data is
affected by other parameter(s), i.e. Reynolds number, blockage
for this Mach number, shock wave location etc., which need
further investigation. After inspection, the problem was found
to be related to the difference between the Reynolds number
of the present tunnel and that of those referenced. The present
wind tunnel is of a suction type while the reference tunnels are
of blow down type with pressurized air which results in higher
Reynolds number. At the same time, since the test section of the
present tunnel is smaller than that of all other tunnels, smaller
models must be tested in its test section to avoid blockage
which will intensify the problem of low Reynolds number. For
instance, the Reynolds number based on the mean chord of the
model in this tunnel was about $0.7 \times 10^6$ for $M_\infty = 0.8$
while for other tunnels, the Reynolds number is in the order
of $6 \times 10^6$. Consequently, the boundary layer on the model is
affected significantly and the normal shock location for $M_\infty \geq
0.9$ is not the same as that of the other tunnels. As a result, the
center of pressure is closer to the leading edge and the resulting
pitching moment is smaller than the referenced data.
From the above discussion, lift and drag coefficients for 3D
tests are in good agreement with the published data and it
seems that no correction is needed. For high Mach numbers, $M_\infty > 0.9$, however, correction should be applied and the correction factor has been calculated as a function of Mach number and angle of attack.

Because of large amounts of acquired data from the tunnel, it is impossible to present the results for all tested Mach numbers. A summary of the results is presented in Figure 8 through Figure 10. Figure 8 shows the effect of free stream Mach number on the lift curve slope of the AGARD-B model for the linear range of $C_L$ vs. $\alpha$ curve. The data for a few other tunnels are shown for comparison. The figure shows that for all cases, the lift curve slope of the model increases with increasing the free stream Mach number up to about 0.95, where it reaches its maximum value. For higher Mach numbers, $M_\infty > 0.95$, the lift curve slope decreases with increasing the free stream Mach number.

Figure 8. Comparison between the present data and those of the references from NAL and AEDC wind tunnels show that the acquired data, lies within the scatter of the referenced data. The slight differences are due to the Reynolds number, surface roughness, shock location, blockage effect and instrument error.

Figure 9 shows variations of the fore body drag coefficient at zero degree angle of attack, $C_L = 0$, versus the free stream Mach number. From this figure, it is clearly seen that again the presented data are in good agreement with those of other tunnels for similar cases. Further, this figure shows that the value of $C_Df$ at zero lift coefficient, $C_L = 0$, is almost constant up to $M_\infty \approx 0.95$. For higher Mach numbers, $C_Df$ increases, reaching its maximum value at $M_\infty \approx 1.1$. This increase in $C_Df$ is
of course related to the formation of normal shock waves on the model surface where its strength increases with increasing the free stream Mach number. For Mach numbers higher than 1.2, it is expected that \( C_D \) decreases since the normal shock will move toward the tip of the nose and becomes an oblique shock. The losses through an oblique shock wave are much less than the corresponding ones through a normal shock, thus it is expected that \( C_D \) will decrease at higher Mach numbers, \( M_\infty > 1.2 \).

Figure 10 shows the effect of free stream Mach number on the pitching moment stability derivative (\( C_{m\alpha} \)). According to the diagram, \(|C_{m\alpha}|\) monotonically decreases as the free stream Mach number is increased up to \( M_\infty \approx 0.9 \). However, the model is stable even at sonic speed, \( M_\infty \approx 1 \), but the stability margin decreases as the free stream Mach number is increased. This figure again shows that the results are in good agreement with those of the NAL tunnel.

Figure 10 shows that in the vicinity of \( M_\infty \approx 0.925 \), \(|C_{p}\)| increases sharply and then remains almost constant with further increasing \( M_\infty \) for Mach numbers higher than 0.925. The results presented in Figure 9 show that at the same Mach number the \( C_D \) rises sharply. Thus, it could be concluded that the free stream Mach number of \( M_\infty \approx 0.925 \), is drag divergence Mach number for this model under this condition. Beyond this Mach number, \( M_\infty \approx 0.925 \), the position of normal shock, apparently does not vary significantly with increasing the free stream Mach number, Figure 10. However, its strength will increase, Figure 9.

4.2. NACA0012 tests

The 2D NACA0012 airfoil model, discussed previously, was used to measure the static pressure distribution and investigate the effects of porosity and side suction on the wind tunnel operation in the transonic regime. The designed and manufactured model had a span of 600 mm which is equal to the wind tunnel test section width and had a chord of about 150 mm that resulted in a test section height to chord ratio of about 4. Pressure port arrangement was discussed in the model description section. The designed and manufactured model was tested at different free stream Mach numbers, ranging from 0.4 to 0.95, and the angle of attack was varied from 0° to 4°. Surface pressure distribution on both upper and lower surfaces of the airfoil as well as the schlieren pictures are presented in this paper.

An important problem in the transonic tests is related to the sensitivity of the shock location and transition point caused by the flow variables such as precision of the free stream Mach number setting, Reynolds number, surface roughness, turbulence level of tunnels and even tunnel acoustic level. For instance, the pressure distribution on the NACA0012 airfoil in two different wind tunnels at NASA Langley is illustrated in Figure 11 for the same Reynolds numbers of about 2.1 Million. As seen from this figure, shock location in different wind tunnels varies up to 15% of the model chord. The location of the shock wave from the \( C_p \) data is where there is a jump, increase in \( C_D \), as illustrated in Figure 11a. Note that in this figure for a free stream Mach number difference of about 0.003, the shock wave location varies 15% of the chord. In addition, when the tests are repeated, the shock location may vary too, 5% of the chord as seen from Figure 11(a). Figure 11(b) shows a similar trend. The redundancy of the tests in the transonic regime is another problem, which is due to the occurrence of the shock wave on the model surface and high sensitivity of the shock location to other variables.

The presented results are compared with the published data of ATA, Langley wind tunnels (4 by 18 inch and 6 by 19 inch) and their results are presented in Ref. [12]. The present experimental results for the zero degree angle of attack are shown in Figures 12 through 14 and are compared with the available experimental data from other wind tunnels.

Figure 12 shows variations of the pressure coefficient with \( x/c \) for the lowest possible free stream Mach number in this
The effect of increasing the free stream Mach number can be clearly seen from the jump in the $|C_p|$ value, $|C_p|$ decreases suddenly at the shock location over the surface of the model, as can be seen in Figures 12–14. By careful examination of these figures, one can clearly realize the sensitivity of the surface pressure with variations of the free stream Mach number. In addition, from these figures it is clearly seen that the present $C_p$ data with the aforementioned test conditions compares very well under the available experimental data for $\frac{C}{\delta}$'s from the leading edge until the point where the shock wave is located which varies with the free stream Mach numbers. At $\frac{C}{\delta}$ where the shock is located, $C_p$ data differs slightly. However, as seen from Figure 11, the pressure distribution for the referenced tunnel varies too and by careful inspection of Figure 12 it is clearly seen that the differences between the present $C_p$ data and those of the other tunnels are much less than those presented in Figure 11.

Figure 14 shows the pressure distribution over the model surface for $M_\infty = 0.91$ and as seen from this figure, the present data are comparable with those from the Langley and ATA findings. Moreover, the Schlieren photo shows no shock reflection from the upper and lower walls which indicate the effectiveness of porosity and side suction from the test section walls. From the $C_p$ data, it is apparent that the shock location on the model surface for this Mach number is about 0.74 of the chord which is in good agreement with the reference data of Langley.

Figure 15 which is presented in Ref. [13] shows the effects of the free stream Reynolds number on the shock position for the NACA0012 airfoil when tested at $M_\infty = 0.81$. The data are for various wind tunnels all over the world. The present data is added to the figure too. As seen from this figure, the scatter of the data is significant which is due to the various sources mentioned previously. For the present test, as mentioned in the article, the acceptable range of shock wave location for this airfoil when tested at $M_\infty = 0.8$ and $\alpha = 0^\circ$ is $\frac{C}{\delta} = 0.46 \pm 0.02$. As seen from Figure 15 the present data lies within the acceptable range which indicates that operation of the tunnel with the applied side suction is acceptable.

Figure 16 shows the effect of the angle of attack and free stream Mach number on the normal shock location over the airfoil. In addition, the data are compared with the available data from other tunnels. As seen, the acquired data of the present tunnel for angle of attack of two degrees are also in good agreement with those of other tunnels. The results for other Mach numbers and for angle of attack of $4^\circ$ showed similar trends, but are not presented in this paper.

5. Conclusion

Intensive experiments over both 2-D and 3-D calibration models were performed to investigate the flow improvements and performance of an upgraded transonic wind tunnel. Surface pressure data over a 2-D model at various free stream Mach numbers and angles of attack from subsonic through transonic regimes are obtained and are compared with the existing data of other tunnels for similar cases. From these comparisons, it is concluded that the data of the upgraded tunnel in the transonic regime is reliable when surface porosity and suction are applied. Similar results were obtained for the 3-D model. However, the pitching moment data of the 3-D model was not comparable with the available data of similar tests in the transonic regime. Therefore, further investigations are needed to find and fix this problem. However, other data such as lift and drag ($C_{Df}$) compare excellently. The discrepancy of the pitching moment is, however, in an acceptable range, while the
Figure 15: Shock location on NACA-0012 airfoil in different wind tunnels in $M = 0.81$.

Figure 16: Pressure distribution over NACA 0012 at $M = 0.74$ and at $\alpha = 2$.

authors expected to obtain better accuracy. In conclusion, with the aforementioned changes in the present wind tunnel, the acquired data in the transonic tunnel are accurate and can be used if the right conditions are met.

References


Kaveh Amiri is a graduate student from Sharif University of Technology, Department of Aerospace Engineering. Mr. Amiri’s specialty is in the area of experimental aerodynamic, specifically transonic wind tunnel testing. He has designed and tested a new test section for the existing trisonic wind tunnel.

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