Design of a Tri-sonic Wind Tunnel

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Abstract: The days of high speed aerodynamics and supersonic propulsion have entered into a new era. Availability of computational techniques has reduced the workload dependencies on other means of testing. Still without experimental data and some physical validation, these may not hold much of a meaning. The wind tunnel for example is one of the best equipment for these applications. Although starting from the same computational bases in the designing phase for a wind tunnel due to cost considerations, a wind tunnel can be designed with a higher accuracy with numerical and theoretical approach as compared to only a theoretical approach. High speed aircrafts particularly supersonic aircrafts undergo radical changes as they move from low subsonic regime to high subsonic compressible regimes and further into transonic and supersonic flow regimes. These transitions pose majority of the design challenges to the aircraft manufacturers. Adhering to these challenges a wind tunnel capable of accelerating from subsonic, transonic and supersonic regimes of flow is probably inevitable equipment in terms of testing parameters and cost. It can provide closest possible results for aircraft designers for the final design of a supersonic aircraft and to obtain the values and parameters for effect of varied flow as the aircraft accelerates from subsonic to supersonic regimes. It could also eliminate the difficulties encountered for such a purpose using different tunnels. The wind tunnel design was carried out and concluded with results from the software providing information of the flow quality through the tunnel and the test section. The designs showed successful simulations of flow of different regimes (subsonic, transonic, supersonic flows). Further experiment with varying diffuser half angle provided more data showing dependency of the flow on diffuser geometry, particularly in supersonic cases. Further nozzle analysis for the tunnel provided more data and conclusion regarding flow dependency on nozzle geometry.

1. INTRODUCTION

A wind tunnel capable of generating the three kinds of flow regimes, namely subsonic, transonic and supersonic flow regimes can perform well if the flow is continuous and accelerates to higher velocities. This particular capability will provide a lot of data on how flow interacts with the aircraft as it moves to higher velocities. This tunnel could prove to be useful as a single model would be required and the setups need not be changed for different tests. It can help design an aircraft capable of producing least drag as it accelerates to supersonic speeds with minimal cost. These qualities were the major reason for designing a single tunnel capable of producing all the flow regimes mentioned above. The design process was started based on simple isentropic relations and theories. The tunnel design was later simulated for real gas and other frictional effects and accordingly modified.

2. BACKGROUND THEORY

2.1 Wind tunnels and their uses [5]

Wind tunnels are an extremely useful tool to conduct aerodynamic tests on aircraft models and other related components of aircrafts. Wind tunnels are not just limited to aircraft design alone, they are also used in design of automobiles, wind turbines, buildings, bridges etc. They are a very integral part of testing and analysis when computational analysis is not feasible. Also, wind tunnels enable easy testing of scaled models of aircrafts so as to understand their behaviour during various phases of flight, before performing a flight test, so that any noticeable defects can be corrected, and the design can be improved. Wind tunnels give us the flexibility to vary the flow conditions like pressure, temperature, speed etc. for a wider range of tests that can be performed.

2.2 Compressible Flow Theory [4][7]

Incompressible flow principles are no longer valid in this type of flow, as most of the flow taking place is compressible in nature. The high velocities used in this project involve large pressure gradients, which lead to changes in local density at various points in the flow. Discontinuous large variations in pressure can lead to formation of shock waves.

For the purpose of theoretical calculations, the flow is assumed to be isentropic, thus isentropic flow relations and shock relations were used during the theoretical design phase.

2.3 Nozzle Flow properties [4][7]

The analysis of isentropic flows across nozzles can be studied and analysed based on the area-velocity relation (derived from the compressible continuity equation), given as follows –

$$\frac{dA}{A} = (M^2 - 1)\frac{du}{u} \qquad [2.1]$$

For the design of supersonic (convergent-divergent) nozzles, the following formula is obtained from the above relation –

$$(\frac{A}{A^*})^2 = \frac{1}{M^2}[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M^2\right)]^{\frac{\gamma+1}{\gamma-1}} \qquad [2.2]$$

This relation tells us that the Mach number at any location in the duct is a function of the ratio of local duct area to the sonic throat area (A^*).

Isentropic relations were used for the determination of properties of the flow. These isentropic relations are as follows –

$$\frac{p_0}{p} = (1+\frac{\gamma-1}{2}M^2)^{\frac{\gamma}{\gamma-1}} \qquad [2.3]$$

$$\frac{\rho_0}{\rho} = (1+\frac{\gamma-1}{2}M^2)^{\frac{1}{\gamma-1}} \qquad [2.4]$$

$$\frac{T_0}{T} = (1+\frac{\gamma-1}{2}M^2) \qquad [2.5]$$

Also, the following normal shock relations were used –

$$\frac{p_2}{p_1} = 1+\frac{2\gamma}{\gamma+1}(M^2-1) \qquad [2.6]$$

$$\frac{\rho_2}{\rho_1} = \frac{(\gamma+1)M_1^2}{2+(\gamma-1)M_1^2} \qquad [2.7]$$

$$\frac{T_2}{T_1} = (\frac{p_2}{p_1})(\frac{\rho_1}{\rho_2}) \qquad [2.8]$$

$$M_2^2 = \frac{1+[\frac{\gamma-1}{2}]M_1^2}{\gamma M_1^2 - \frac{\gamma-1}{2}} \qquad [2.9]$$

Where, subscripts 2 and 1 represent quantity after shock and quantity before shock respectively.

2.4 Method of Characteristics [3][4][7]

Method of characteristics is a numerical method which is generally used to plot the contour of a convergent-divergent supersonic nozzle. It uses non-linear differential equations of velocity potential equations. Characteristic lines are unique lines in the flow where the derivatives of the flow properties become indeterminate and discontinuous, though the flow properties themselves exist as finite values.

General procedure of method of characteristics involves the following steps –

Step 1

Characteristic lines are determined in the flow, i.e., particular directions in the xy space where flow variables are continuous but their derivatives are indeterminate, and sometimes even discontinuous.
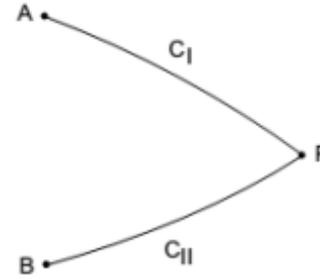
**Figure 1: Characteristic lines – left running (C_{II}) and right running characteristics (C_I)** [3]

Step 2

Partial differential conservation equations are combined in such a fashion that ODEs are obtained which hold only along the characteristic lines. These ODEs are called compatibility equations.

Step 3

These compatibility equations are then solved step-by-step along the characteristic lines, starting from the given initial condition at some point or region in the flow. In this manner, the complete flow field can be mapped out along the characteristics.

By following the above iterative procedure, we can obtain the contour of the supersonic nozzle such that no expansion waves or shock waves are formed.
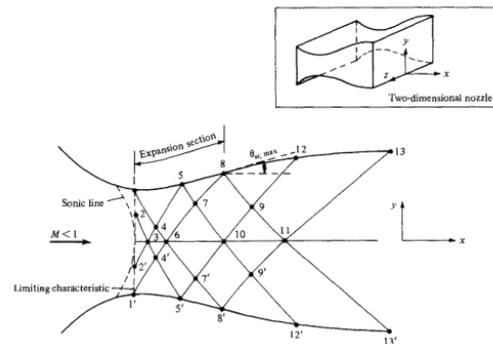
Figure 2- Method to compute contour of divergent section of convergent-divergent nozzle [7]

2.5 Diffusers [4][7]

Diffuser is the section of the wind tunnel responsible for reducing the speed of the flow and to recover pressure. Following classification can be done based on flow velocities for subsonic or supersonic diffusers:

For subsonic diffusers the area Mach number relation dictates that the velocity decreases as area increases for Mach number less than 1.

- For supersonic diffusers, the area Mach number relation dictates that the velocity decreases as area decreases for Mach number greater than 1.

- This suggests that the supersonic diffuser should have convergent divergent section to decelerate the flow from supersonic to subsonic flow again with the diffuser throat Mach number nearing 1.

2.6 Starting problem of supersonic wind tunnel [6][13]

During tunnel start-up, there is zero flow velocity, and constant pressure exists throughout the tunnel. The starting of the tunnel is enabled by changing the back pressure, and consequently the pressure ratio p_o/p_b, by using the compressor. Initially, there will be subsonic flow everywhere.

As we keep increasing ratio of inlet total pressure to back pressure p_o/p_b, the flow velocity keeps increasing, and at a certain point, normal shock formation takes place in the throat of the convergent-divergent nozzle. With subsequent increase in pressure ratio, normal shock gradually moves backward in the test section.

To have the shock to completely disappear from test section, it must completely pass through the second throat. Thus, initially, the second throat area needs to be high (for starting), and once the shock wave is pushed behind, the area of second throat can be reduced for optimal operating conditions. A variable area throat of the diffuser would be most useful in this case.
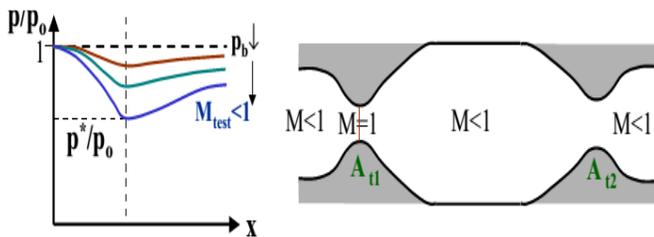

Figure 3: M=1 at first throat [13]
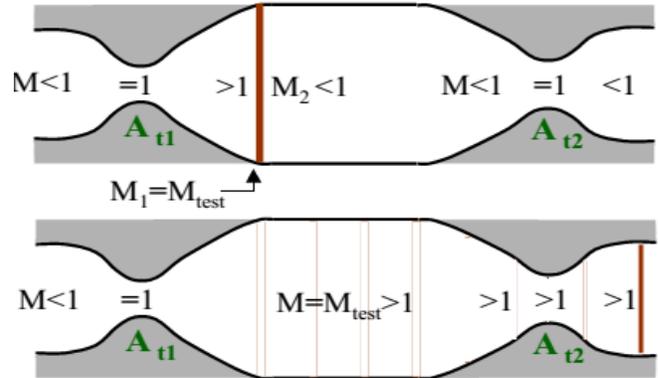

Figure 4: Starting shock [13]

Figure 5: Swallowed shock; tunnel start condition [13]

3. METHODOLOGY

3.1 Procedure carried out

The methodology adopted was primarily based on quasi-one dimensional flow theories. The air flow for the calculation was assumed to be inviscid and the changes assumed to be isentropic. As the major aim to obtain steady state subsonic, transonic and supersonic flow in the wind tunnel, three different tunnels of same length were designed and analyzed. A mechanism which is required to change the geometry of the wind tunnel as the flow accelerates from subsonic to supersonic velocities was also proposed.

The wind tunnels were designed and analyzed using software while the mechanism for varying the geometry was made practically as a demonstration.

3.2 Supersonic flow properties

Principles of compressible flow, nozzle flow and isentropic flow equations were used to design the shape of the wind tunnel and calculate the pressure ratios, temperature ratios, and the area ratios, for each case – supersonic and subsonic.

1. Nozzle and Test section calculations

An inlet Mach number of 0.4 was assumed in the initial calculations. Based on that, the inlet-throat area ratio was calculated from equation 2.2.

The area ratio was found to be:

$$\frac{A_i}{A_{throat}} = 1.5904$$

For a test section Mach number of 2.0, the respective pressure and temperature ratios were calculated and the following values obtained (from equation 2.3 to 2.5) –

$$\frac{p_i}{p_o} = 0.8956143$$

$$\frac{p_{test}}{p_o} = 0.1278045$$

$$\frac{T_i}{T_o} = 0.96899225$$

$$\frac{T_{test}}{T_o} = 0.55556$$

Now, to design the test section dimensions, sea-level conditions were assumed in the test section for a Mach number of 2.0. Accordingly, the following values of pressures and temperatures were obtained –

$$p_{test} = 101325 \text{ Pa}$$

$$T_{test} = 300 \text{ K}$$

$$p_o = 792812.3672 \text{ Pa}$$

$$p_i = 710054.089 \text{ Pa}$$

$$T_o = 540 \text{ K}$$

$$T_i = 523.256 \text{ K}$$

$$\frac{A_{test}}{A_{throat}} = 1.6875$$

Assuming the throat area of convergent-divergent nozzle to be 1 m^2, the rest of the dimensions can be calculated with this reference.

2. Diffuser calculations [7]

A diffuser is a very important part of a supersonic wind-tunnel, as most of the pressure recovery takes place in this section. The most optimal diffuser possible in a supersonic wind tunnel is a convergent-divergent diffuser with a throat, which is designed to minimize the reflected shock waves and combine them into one single normal shock at the end of the diffuser throat. This is done so as to minimize the total pressure losses across the shock waves as much as possible. Subsequently the subsonic flow slowed down to speeds that can be comfortably handled by the fan/compressor.

For isentropic flow in a supersonic wind tunnel, the following relation is valid:

$$\frac{A_{t,2}}{A_{t,1}} = \frac{p_{01}}{p_{02}} \qquad\qquad [3.1]$$

This relation states that ratio of diffuser throat to nozzle throat is equal to the ratio of the total pressure of the flow before and after the normal shock in the diffuser. Since the total pressure after the normal shock is higher than the total pressure preceding it, the second (diffuser) throat area is always greater than the first (nozzle) throat area.

Thus, using normal shock relations and the earlier total pressure values obtained, the following values were obtained –

$$\frac{A_{t,2}}{A_{t,1}} = \frac{p_{01}}{p_{02}} = \frac{1}{0.7209} = 1.387$$

For all practical cases, the second throat must be larger than the first throat related by the above ratio. Assuming first throat area as 1 m^2, area of second throat is 1.387 m^2, thus, to obtain proper flow in the test section, any diffuser throat with area greater than this will suffice. Thus, an area of $A_{t,2}$ = 1.4 m^2 was chosen.

From the above relation, the total pressure recovered and total temperature behind the normal shock were calculated to be the following –

$$p_{02} = 571602.4782 \text{ Pa}$$

$$T_{02} = 491.8135 \text{ K}$$

3.3 Subsonic flow properties

To estimate the subsonic flow properties, a design test section Mach number of 0.7 was selected and the area ratios were designed appropriately. In this case, test section area is equal to the throat area of the duct. Also, since the test section area has to be same for all three wind tunnel configurations (subsonic, transonic and supersonic), the test section area was fixed to 1.6875 m^2 as obtained from supersonic calculations.

Considering an inlet Mach number of 0.3, using compressible flow properties, the following value was obtained for inlet area to throat area ratio–

$$\frac{A_i}{A_{test}} = 1.9317$$

And the throat to exit area ratio as –

$$\frac{A_{test}}{A_e} = 0.67752$$

Using the isentropic flow relations, the rest of the flow properties were calculated and obtained as follows –

For inlet Mach number 0.3 and throat Mach number 0.7 (from equation 2.6 to 2.8) –

$$\frac{p_i}{p_{test}} = 1.30308$$

$$\frac{T_i}{T_{test}} = 1.0785$$

$$\frac{\rho_i}{\rho_{test}} = 1.2079$$

The above calculations were made for the assumption that flow is adiabatic and isentropic. Test section conditions are sea-level, atmospheric conditions.

3.3 Tools Used

Three major tools were used in the initial calculations and design phase.

- ANSYS ICEM CFD

- ANSYS FLUENT

- MATLAB

ANSYS ICEM CFD was used to make a 2-D CAD model of the tunnels and mesh them for analysis. The analyses were carried out in ANSYS Fluent.

MATLAB was used to design a contour for the divergent section of the convergent-divergent nozzle in the supersonic wind tunnel using a code [11] implementing Method of Characteristics and also for power requirement calculations.

K-epsilon viscous model was used on FLUENT for viscous analysis.

Following were the differential equations used in tools for solution of the required cases:

1. Continuity Equation [14]
2. Navier-Stokes Equations [14]
3. Energy Equation [14]

3.5 Model Generation

Depending on the area and pressure ratios calculated, a simple CAD geometry was made in order to mimic the tunnel with the following parameters –

Supersonic

- Throat area = 1 m^2

- Test section area = 1.6875 m^2

- Inlet area (for Mach 0.4 entry flow) = 1.5904 m^2

- Diffuser exit area (for assumed exit Mach flow of 0.7) = 1.67 m^2

The exit geometry for the nozzle of the supersonic wind tunnel was made using Method of Characteristics. A MATLAB script [11] was utilized in order to generate the required geometry. Later on, the half-angles for the nozzle inlet and the exit diffuser were changed to 2° for supersonic wind tunnel. The final length obtained from above conditions was calculated to be 15.92 m.

Geometry of the wind tunnel was selected to be of rectangular cross-section for the following advantages:
- Better test section visibility
- Easier to vary geometry
- Easier manufacture
- Easier nozzle design
- Can accommodate wider models

Following were the parameters calculated for subsonic case of the wind tunnel:

Subsonic

- Test section area = 1.6875 m^2

- Inlet area = 3.2603 m^2

- Diffuser exit area = 2.491 m^2

- The geometry for generating sonic and near-sonic flows was same as that for subsonic.

The final length of the tunnel was selected as that of the supersonic wind tunnel for all configurations. Another geometry having parabolic intake for flow quality comparison was made for subsonic and transonic flow conditions, using the following equations –

$$y^2 = 4ax$$

With "a" being the constant value of 0.0265, obtained from the above given data of test section and inlet.

3.6 Mesh and Details

The mesh was formed using ANSYS ICEM CFD commercial meshing software. The mesh type was selected to be quad-dominant in order to provide the best geometry capture and low flow distortion.
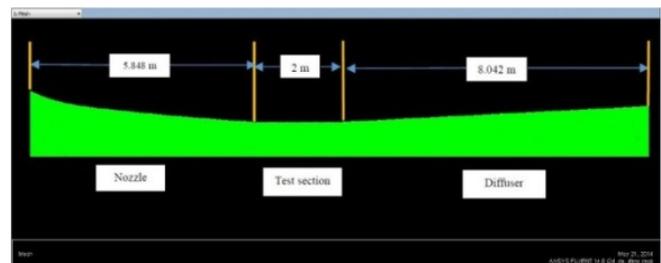

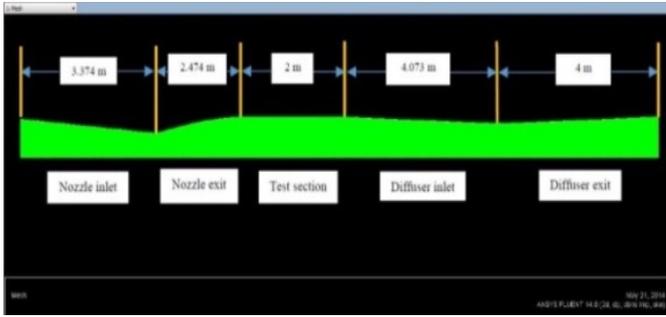**Figure 6: Subsonic wind tunnel with parabolic inlet (with lengths of each section)**

Figure 7: Supersonic wind tunnel (with lengths of each section)

The lengths were obtained based on previous calculations and assumptions.

3.7 Boundary Conditions

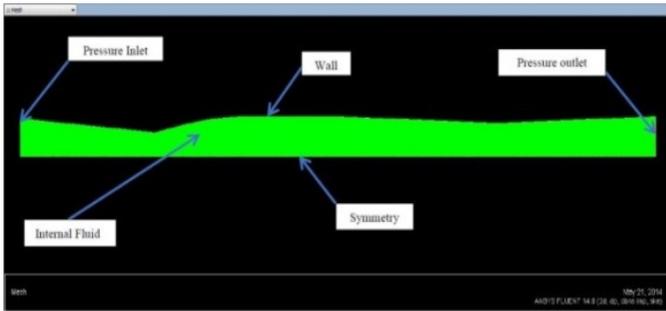Pressure inlet and outlet boundary conditions were used for analysis of the wind tunnel:

Figure 8: General boundary conditions for tunnel

Following were test section parameters assumed:
- Pressure = 101325 Pa
- Temperature = 300 K
- Density = 1.225 Kg/m^2

From the above values and obtained area ratios, inlet boundary conditions were calculated from isentropic relations for supersonic flows.

- *Supersonic tunnel, pressure inlet*
- o Gauge pressure = 792812.3672 Pa
- o initial gauge pressure = 710054.1 Pa
  - o Total temperature = 540 K

These values were obtained from the relations as mentioned previously in section 3.2.

Similarly for subsonic wind tunnel, following were the parameters obtained:
- *Subsonic tunnel, pressure inlet*
  - o Gauge pressure = 140550.8 Pa

- o Initial gauge pressure = 132034.6 Pa
- o Total temperature = 329.374 K

For the outlet conditions, maximum pressure recovery was calculated using isentropic relations and further iterations were performed to get the required pressure recovery for the given conditions.

The wall conditions were given as no-slip, stationary wall for all cases.

3.8 Mechanism of variable geometry

The tri-sonic wind tunnel design has to have a feature to modify the area ratios in order to generate the specific flow conditions for subsonic, transonic or supersonic. A nozzle design algorithm from method of characteristics (MOC) was used in order to obtain the required exit geometry of the nozzle for supersonic flows over various Mach numbers. For subsonic flows, parabolic inlet geometry was formed using the equations stated in section 3.6.

These particular coordinates were fed into servo-motors located at specific positions in order to modify the geometry as and when the flow conditions changed from subsonic through transonic to higher supersonic flows (up to Mach 2.0). The following procedure was followed to modify the geometry as deemed necessary for the flow conditions:
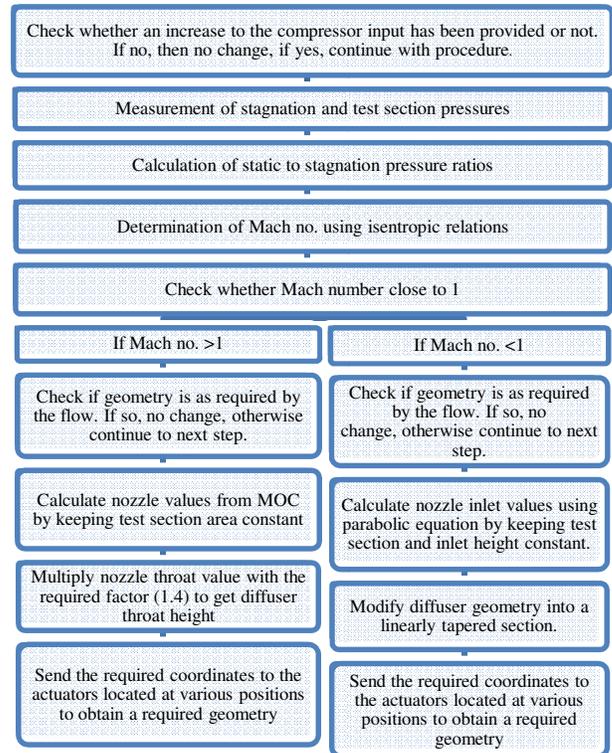

Figure 9: procedure for changing the geometry of the tunnel

4. RESULTS

4.1 Subsonic Contours and Plots

For the subsonic case, two geometries were designed, analyzed and compared in order to obtain the best possible flow quality in the test section. The graphs near the walls for the sections should be as smooth as possible with minimal adverse pressure gradients. Following are the analyses done for the subsonic wind tunnels:

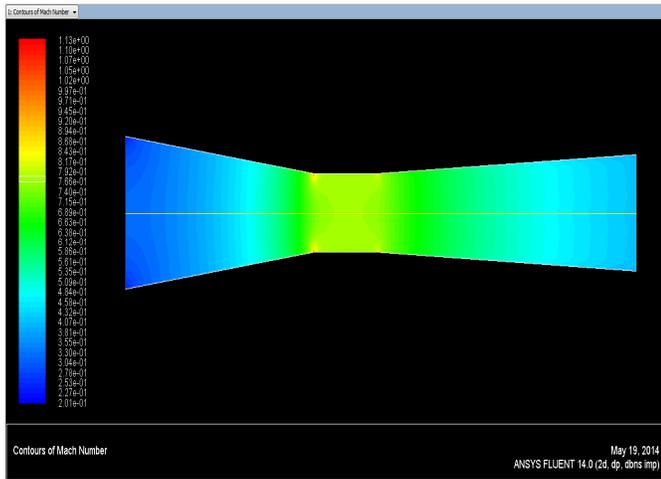1. Inviscid case - stagnation to back pressure ratio of 1.12:
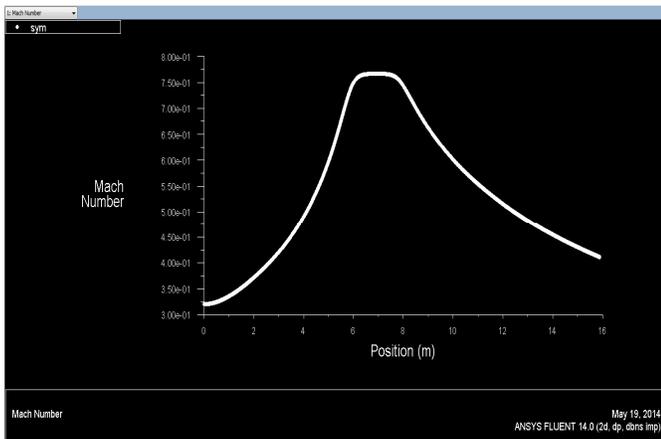
a. Straight tapered inlet:

Figure 10: Contours of Mach number

Figure 11: Mach number plot w.r.t. length along center-line

The graph above (figure 11) represents the Mach number distribution along the center line. The distribution of the flow is nearly ideal as expected. But as the flow is observed closer towards the wall, there is a lot of disturbance and localized expansion corner causes sonic flow conditions (Figure 10).

b. Parabolic inlet
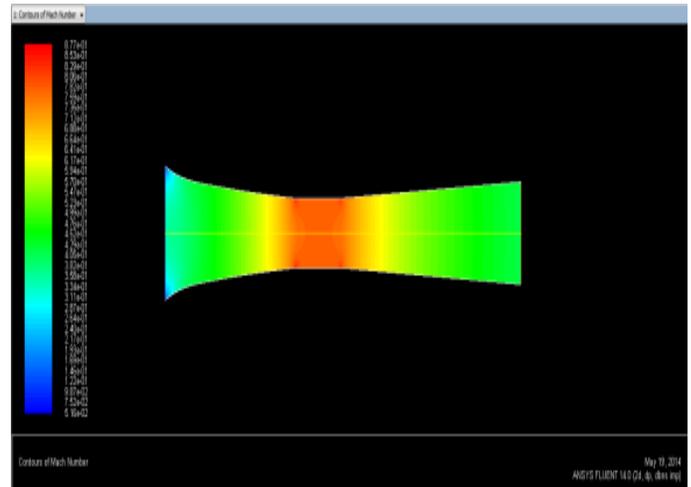
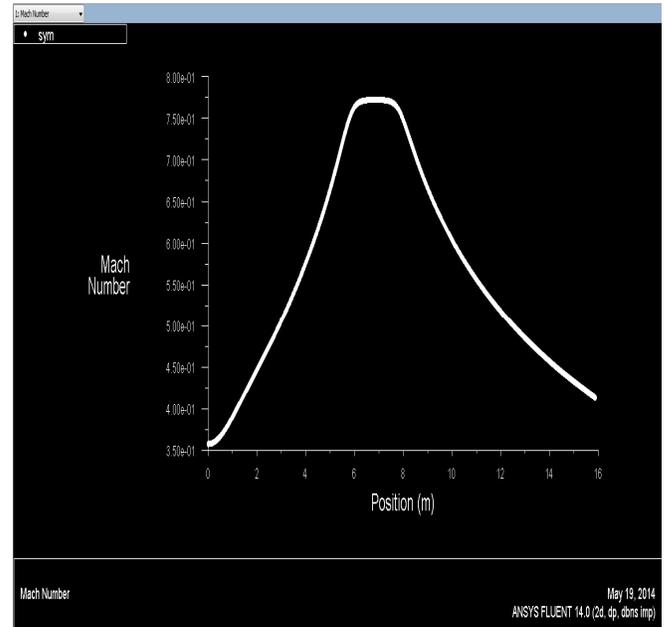**Figure 12: Contours of Mach number**

Figure 13: Mach number plot w.r.t. length along center-line

Here (in Figure 12, 13) there is again a localized increase in Mach number due to a very gradual expansion corner. This results in a rise of Mach number just at the extremities of test section, but the geometry prevents the Mach number from rising up to sonic flows under the same conditions, contrary to the linearly tapered nozzle.

This particular trend highlights the fact that a gradual area variation provides a much smoother flow and less disruption. This also affects the analysis as such effects might disrupt the flow around the model and near the walls.

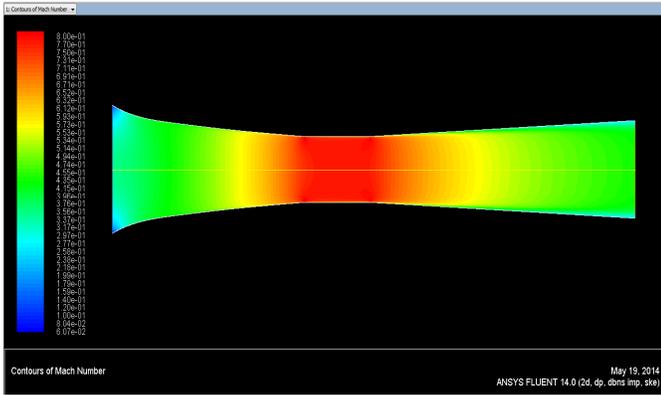*2. Viscous case - stagnation to back pressure ratio of 1.14:*
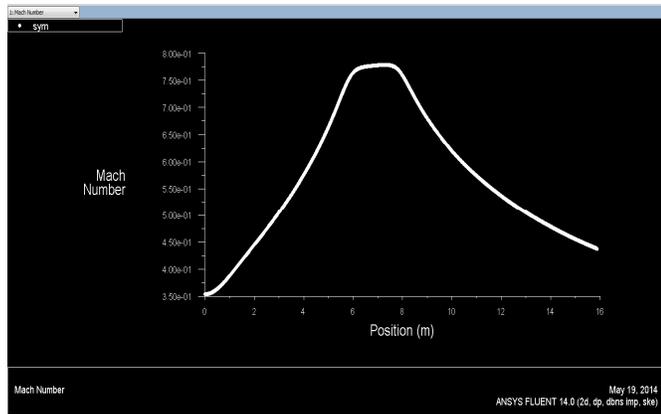

Figure 14: Contours of Mach number

Figure 15: Mach number plot w.r.t. length along center-line

This was the case (refer figure 14, 15), run for subsonic viscous flows. The pressure ratio has resulted in a 0.77 Mach flow. In this case, the viscous losses were calculated to be 1.42 % of the back pressure recovery obtained for inviscid case for a similar Mach number flow in a test section.

Viscous case - stagnation to back pressure ratio of 1.15:
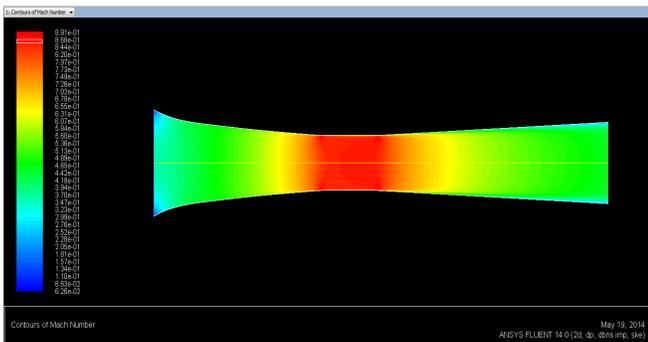
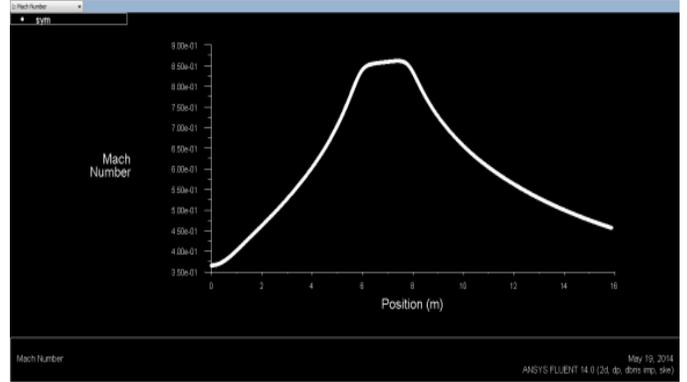**Figure 16: Contours of Mach number**

Figure 17: Mach number plot w.r.t. length along center-line

In above contour and plot (figure 16, 17), the pressure losses were assumed to be 1.4 % of the total back pressure recovery for an expected flow of Mach 0.88. Mach number of 0.86 was achieved in the test section against the predicted value. This indicates that the assumed viscous losses of 1.4 % were not valid and the actual losses were higher.

The losses were plotted for different pressure ratios against the Mach no. achieved in the test section.
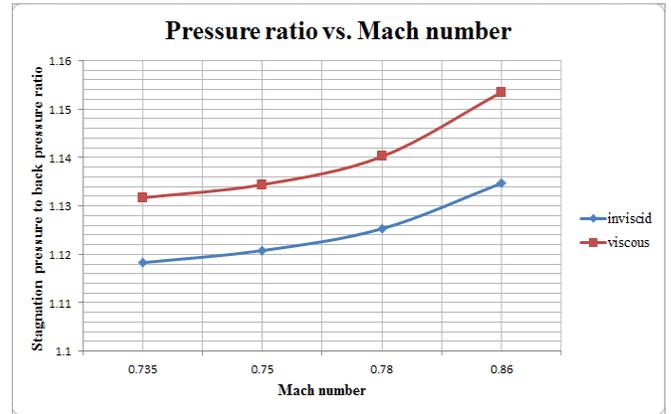

Figure 18: Stagnation to back pressure ratio vs. Mach number
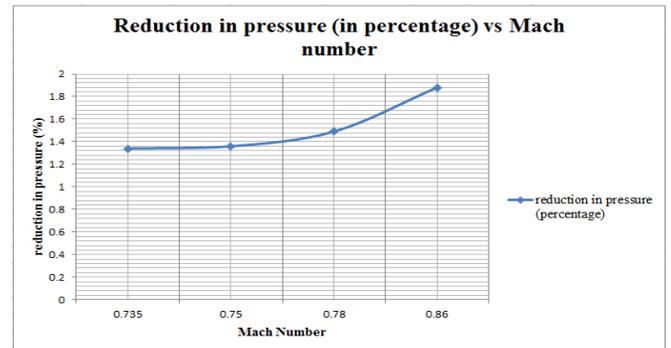

Figure 19: Reduction in pressure ratio (in %) vs. Mach number

It is depicted in the graphs above (figures 18, 19), as the flow velocity increases, the amount of viscous losses also tend to increase.
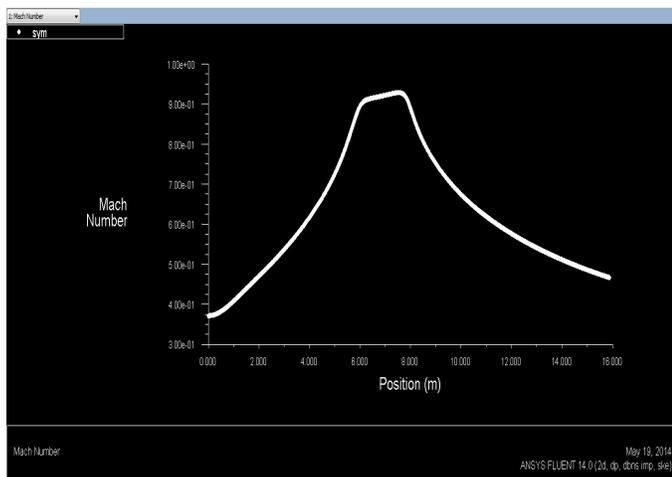
4.2 Transonic contours and plots

For transonic analyses, the subsonic geometry was used. Following analyses were carried out.

### 1. Viscous case - stagnation to back pressure ratio of 1.161:
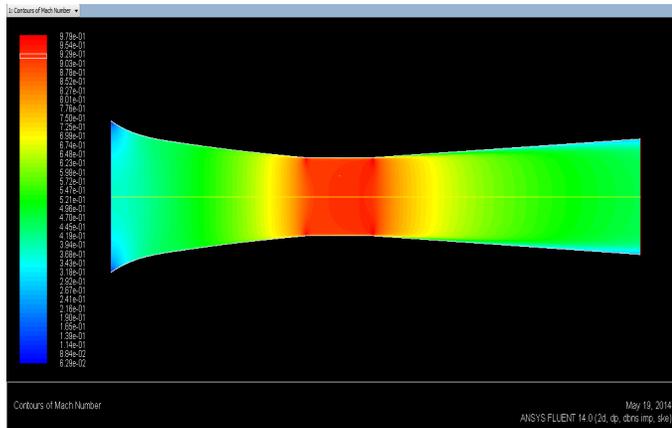
**Figure 20: Contours of Mach number**

Figure 21: Mach number plot w.r.t. length along center-line

Here in figure 20 and figure 21, a Mach number of 0.93 was achieved in the test section. As it can be seen in the graph, for Mach number vs position, there is a gradual increase in the test section Mach number over the length of the test section. This particular finding leads to the conclusion that the boundary tends to, very gradually, increase in thickness, thus producing a secondary nozzle in the test section, accelerating the flow. The thickness, however, does not change very drastically, and is low enough to not have a considerable change in the Mach number.

2. Viscous case - stagnation to back pressure ratio of 1.164:
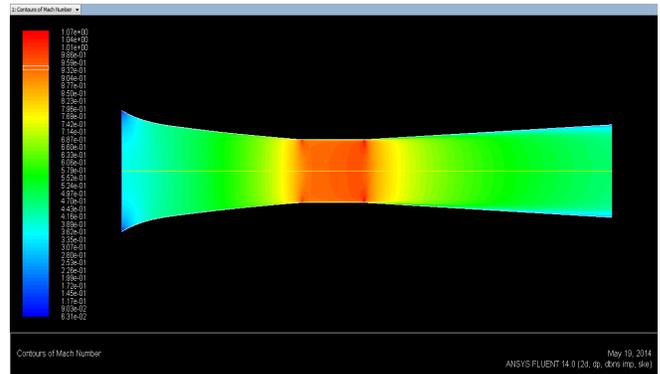
Figure 22: Contours of Mach numbers
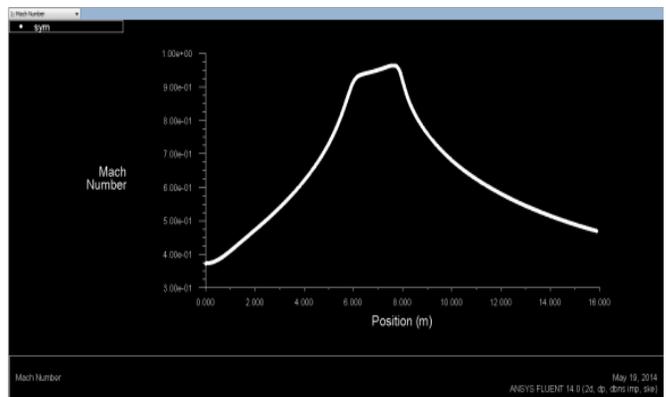
Figure 23: Mach number plot w.r.t. length along center-line

A Mach number of about 0.95 was achieved in the starting of the test section. As explained before, there is a Mach number rise in the test section itself, after the nozzle has ended (refer figure 22 and figure 23), due to slight increase in boundary layer thickness across the test section. In this particular case, due to a higher flow velocity, the boundary layer tends to have a higher thickness (though negligibly) than the previous case. Even this negligible increase in thickness is enough to cause the test section to act as a nozzle, and have a higher Mach number towards its end.

4.3 Supersonic contours and plots

For supersonic analyses, two separate geometries, both containing nozzles designed by method of characteristics, were made. Both had identical overall dimension with difference in the diffuser section where one of the tunnels had the converging diffuser half-angle of 2°, while the other had converging diffuser half-angle of 4°. The diffuser with 4° converging half angle accommodated a straight diffuser throat section before the expansion cone. This was to allow for shock reflections and smoother deceleration of the flow. Following analyses were carried out:

Viscous case - stagnation to back pressure ratio of 1.98 (Converging diffuser half-angle 2°):

This static pressure graph (figure 112) depicts a smooth drop in pressure along the center-line of the wind tunnel until the test section where it is consistent as with the flow. The sudden rise as observed is due to the shock being formed at that particular position in the diffuser section of the wind tunnel.
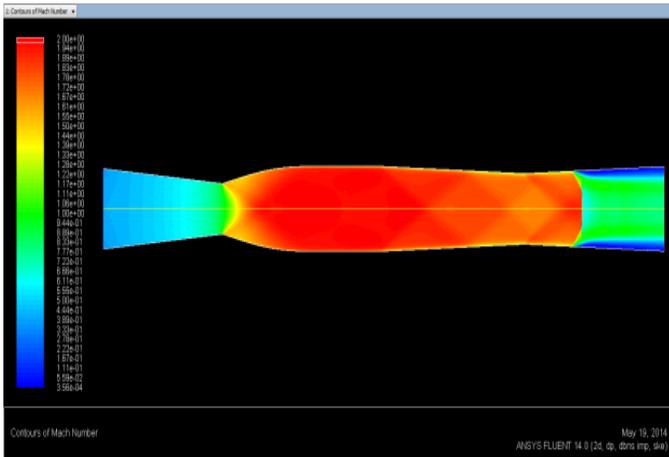

Figure 24: Contours of Mach number

This Mach number contour (figure 24) as depicted above shows a uniform distribution of flow in the test section. There is a gradual decrease in flow velocity as it nears the wall reaching 0 as there is no slip condition on the wall.
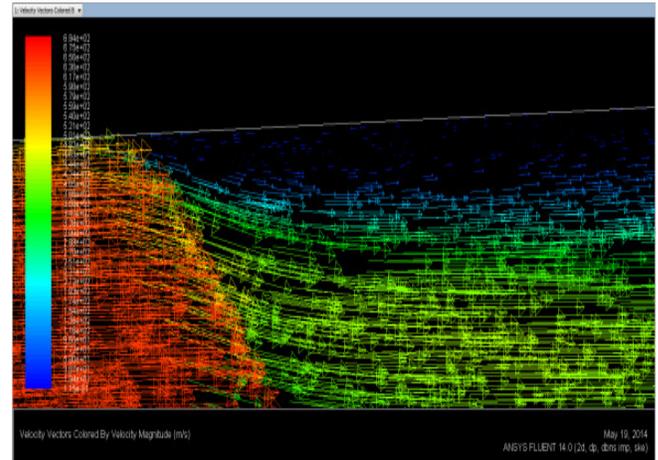

Figure 25: Mach number plot w.r.t. length along center-line (lines indicate test section)

The graph above (figure 25) shows a smooth increase in Mach number smoothly through the nozzle into the test section, where there is very little variation. The plot depicts that there

is a drop in the diffuser as expected and Mach number drops from 1.8 to nearly Mach 0.6.

Figure 26: Velocity vectors indicating flow separation

Figure 26 is a close-up of vector contour of velocity magnitude in diffuser depicting the shock occurring in the diffuser. The slight deviation from normal shock occurs due to boundary layer giving it a lambda shape. The vectors, coloured by velocity magnitude are observed to turn inwards towards the flow due to boundary layer separation. The vectors are observed to have an eddy formation near the wall.
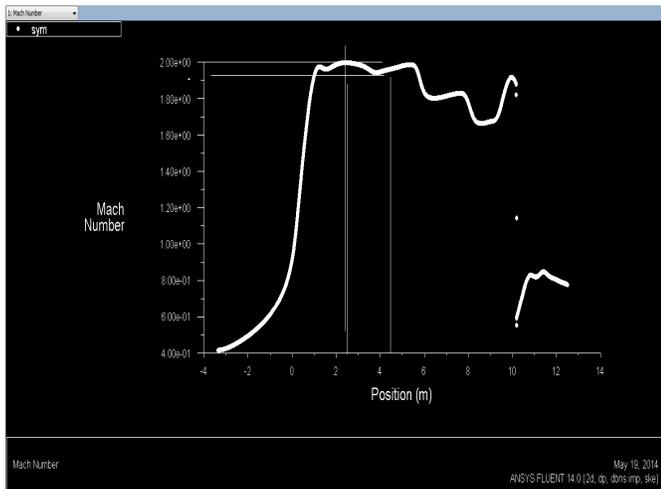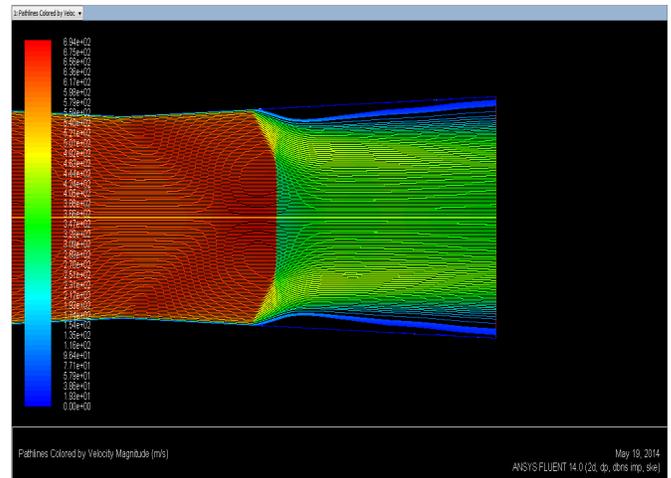

Figure 27: Path lines showing separation of flow at diffuser throat

The path lines in the above contour (figure 27) depict the flow separation and re attachment as the flow progresses further. This trend is as expected for the diffuser and reattachment is necessary for diffuser to work properly.

Viscous case - stagnation to back pressure ratio of 1.98 (Converging diffuser half-angle 4°):
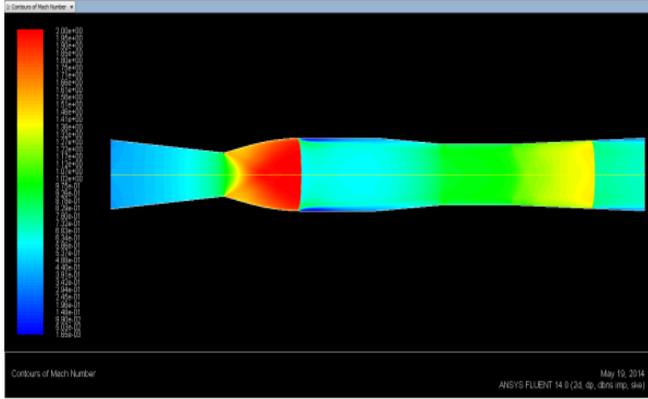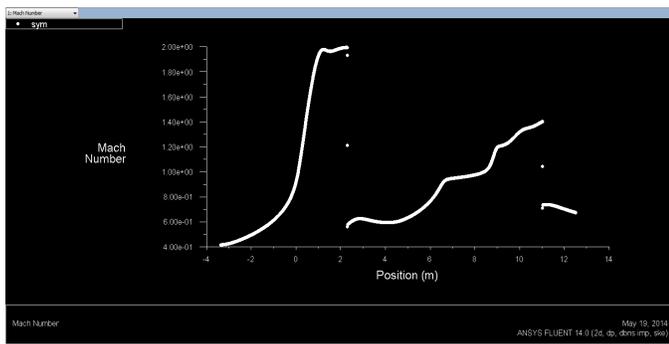
Figure 28: Contours of Mach number

Figure 29: Mach number plot w.r.t. length along center-line

The Mach number contour and graph in figure 28 and figure 29 depict a trend expected during an unstart condition of the wind tunnel. There is a smooth rise in Mach no. across the nozzle (refer figure 29) due to the design of the. The flow through the test section is subsonic.
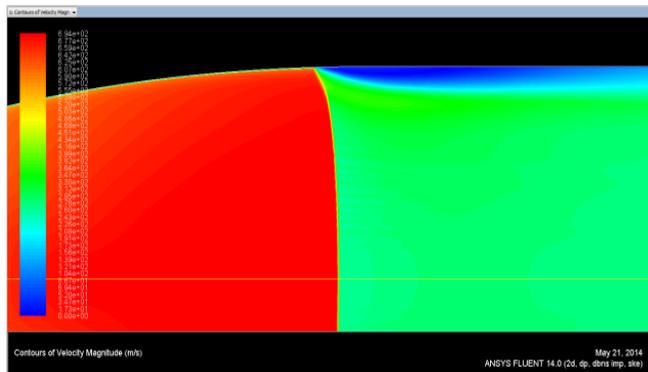

Figure 30: Contours of Velocity magnitude, shock in nozzle magnified

Figure 30 is a close-up view of the shock which occurs in the nozzle. A flow separation can be seen right after the shock. This is the unstart condition of wind tunnel. The possible reason for this is expected to be change in the diffuser inlet angle which was increased to be 4 degrees in this case.

4.4 Sealing problem

The variation in geometry could cause a problem regarding proper sealing of the tunnel (in order to prevent pressure leaks). This problem could be mitigated by either using different sealing techniques which can prevent the pressure leak. A method of sealing is suggested to have a flexible wall at the sides of the wind tunnel. The flexible walls would result in change of internal volume as the geometry moves. This would require particular corrections which can be included in the initial calculations itself to prevent any adverse variation from the expected results.

4.5 Power Requirements

The maximum power requirements for a wind tunnel occur during start-up (for a supersonic case). For continuous operation, these power requirements drastically reduce. A MATLAB script was used to calculate the required power for 1:1 scale wind tunnel and the following results were obtained:

Figure 31: Power requirements of scale 1:1 cross section

It is observed from the graph in figure 31 that power requirements are extremely high. This is due to the fact that at 1:1 scale, there is a huge amount of mass of air to be moved through the tunnel, as the cross section area is 1.6875×8.4375 m^2. For practical purposes, the test section area needs to be reduced. Thus, the lengths are scaled down by a factor of 7, to give a cross section area of 0.2411×1.2054 m^2. The following was the graph obtained for the smaller test section.

Figure 32: Power requirement of scale 1:49 cross section area

It is observed from figure 32 that the more realistic value of 12 MW is required for running the wind tunnel when the lengths are scaled down by a factor of 7.

5. CONCLUSION AND FUTURE SCOPE OF WORK

5.1 Brief Summary of work

The objective of the project was to design a tri-sonic wind tunnel capable of generating subsonic, transonic and supersonic flows in the test section. The designs were made and analyses were carried out for several conditions and geometries. The geometries were designed based on different conditions and various equations were used for designing the contours.

The boundary conditions were theoretically calculated using isentropic relations. The maximum value of back pressure recovery (in case of supersonic Mach 2.0 tunnel) was obtained to be 0.721 times the stagnation pressure at inlet. Therefore, any back pressure value obtained will be lesser than the above mentioned pressure ratio.

6. CONCLUSION

The objective of designing a tri-sonic wind tunnel was met with designs of subsonic, transonic and supersonic tunnels providing the required results. Following were the conclusions derived from the results obtained:

1. Supersonic

The method of characteristics (for supersonic nozzle exit geometry) produced better results than linearly tapered nozzles. A stable Mach number of 2.0 was obtained in the supersonic wind tunnel.

In case of viscous flow through the supersonic tunnel, very little boundary layer thickening was observed. The flow quality wasn't much affected by it. In case of 2° diffuser inlet geometry, for viscous flows, the shock was existent just behind the diffuser throat, rather than the exit of the diffuser. A lambda shock seemed to have formed due to viscous effects near the wall. Also, in the diffuser, there was a region of flow separation right after the shock, which resulted in eddy formations. This flow was reattached to the wall as the flow progressed further downstream. In case of geometry with 4° diffuser inlet half-angle, the tunnel remained in unstart condition with subsonic flow throughout the test section.

2. Subsonic:

Different Mach numbers were achieved for various cases designed for subsonic flow by changing the pressure ratio across the tunnel.

There seemed to be a trend where the viscous losses increased with the Mach number in the wind tunnel. This is due to the fact that, with the increase in flow speeds, the viscous energy dissipation also increases. The boundary layer thickening in case of transonic wind tunnel resulted in increasing the Mach number of the flow through the test section. This particular condition was obtained due to the increase in the boundary layer thickness as the flow progressed through the test section. This thickening of boundary layer being low resulted in a slight convergent section in the test section.

The parabolic geometry produced much better flow quality and lesser adverse pressure gradients.

7. FUTURE SCOPE OF WORK

Complete propulsion system design can be carried out. Model tolerances for the tunnel can also be calculated and different geometries analyzed in order to get the required result. Instrumentation system can be designed in order to get the measurements and readings from the wind tunnels.

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