DESIGN OF A SUPERSONIC FIRST-THROAT FOR A TRANSONIC WIND TUNNEL AND NUMERICAL EVALUATION

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Abstract. *The Pilot Transonic Wind Tunnel (TTP) of the Aerodynamic Division (ALA) from the Institute of Aeronautics and Space (IAE) is a continuous conventional closed-loop wind tunnel, with tests in Mach numbers up to 1.3. The automatic controls of stagnation pressure (from 0.5 bar to 1.2 bar), stagnation temperature and humidity, guarantee high-quality flow in its test section in order to attain required Mach and Reynolds numbers. Its test section is 30 cm wide and 25 cm high, with slotted walls to allow mass removal which is accomplished by both re-entry flaps positioning and plenum evacuation system. Subsonic and near sonic Mach number conditions are achived by the use of the main compressor with sonic first throat. However, supersonic condition is reached only with the use of a supersonic convergent-divergent first throat nozzle. In this case, a convergent-divergent second throat is adjusted to diminish the overall losses in the aerodynamic circuit. In many industrial installations, the first throat geometry is automatically adjusted to a determined Mach number condition at the test section. However in TTP this is manually done substituting the lateral walls of the first throat by another with different geometry. In a recent project sponsored by CNPq the TTP has designed a supersonic first throat by the Method of Characteristics and Foelsch Method, which allows compression and expansion waves mutual cancellation. This paper describes the steps followed in the method to design the convergent-divergent first-throat geometry for Mach number 1.3. The project is assessed by numerical simulation using a two-dimensional code developed in the Aerodynamics Division to calculate the non-viscous Euler equations. A grid mesh was created to represent the region from the first-throat entrance to the end of the test section. The final results are presented in terms of Mach number in the test section, which is the most significant parameters related to tunnel performance.*

Keywords*: Supersonic Nozzle, Method of Characteristics, Foelsch Method, CFD, Pilot Transonic Wind Tunnel.*

1. INTRODUCTION

The Pilot Transonic Wind Tunnel (TTP) of the Institute of Aeronautics and Space (IAE) has been widely used in aeronautical research since its construction. Many experiments were conducted in order to verify the reliability of the results in subsonic and transonic regimes, as required by the original project, which was developed by a working group created at Aerospace Technology General-Command (DCTA), as well as in partnership with a North American company, Sverdrup Technology Inc., (Sverdrup, 1989).

Many attempts to determine the level of uniformity in the test section were performed, initially with open circuit and with the tunnel being intermittently driven by injection system (Escosteguy, 1998, Zanin *et al.*, 2008). Later on, tests were performed with the tunnel in closed circuit and with injection system (Silva *et al.*, 2009). And in a final configuration many tests were performed with the tunnel continuously driven by a two-stage axial main compressor of 830 kW. All these experiments were conducted with sonic first throat, as well as many other campaigns performed with basic aeronautical models (NACA 0012, AGARD standard) and aerospace vehicles from IAE (Sonda III, VS-40), giving support to research and universities activities.

To continue the improvement and extend the tunnels' applications, TTP is expanding to supersonic researches. The wind tunnel will be able to supply flow at Mach number 1.3, by substituting the sonic first throat by a supersonic converging-divergent throat.

A typical supersonic first-throat is composed of three types of regions, each one with specific functions, as shown in Fig. 1. On the convergent region, the subsonic flow accelerates to unitary Mach number at the smallest area of the nozzle, and then it is accelerated again due to area expansion until the project Mach number. The major trouble occurs due to expansion and compression waves originated at the wall of the initial and straightening section of the divergent curve. The waves reflected on the walls propagate through the whole extension of the nozzle, making the jet non-uniform and non-parallel. Actually, the Method of Characteristics consists of adequately choosing the straightening region geometry to create compression waves in such way to precisely cancel the expansion waves arising in the initial region. This mathematical procedure results in a uniform velocity profile at the test section entrance.

Souza *et al.* (2009) has already made first calculations of the supersonic first-throat for TTP using Foelsh's method (Foelsh, 1949). In this present article better explanations and evaluations varying the main design parameters were made and checked by numerical simulation to check the Mach number deviation in test section region before manufacturing the first-throat panels.

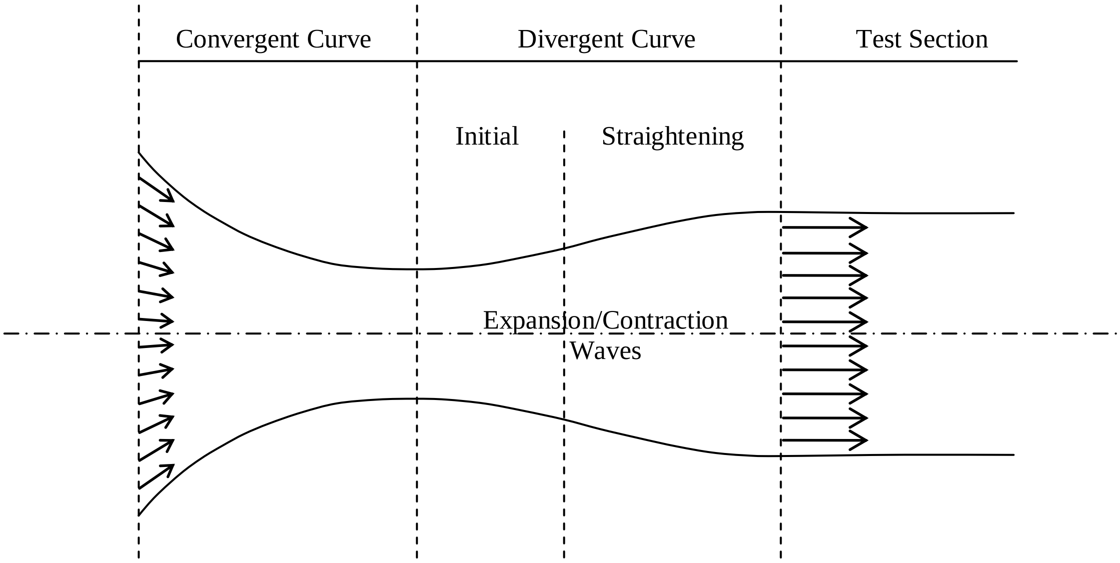

Figure 1. Supersonic convergent-divergent first throat scheme.

2. SUPERSONIC NOZZLE DESIGN

Supersonic wind tunnel uses expansion nozzle, exactly like those found on rocket engines, to expand a relatively slow stream of air to supersonic speeds in its test section. There are two general categories of nozzles used in supersonic wind tunnels: two-dimensional and axisymmetric, as it can be seen in Fig. 2.

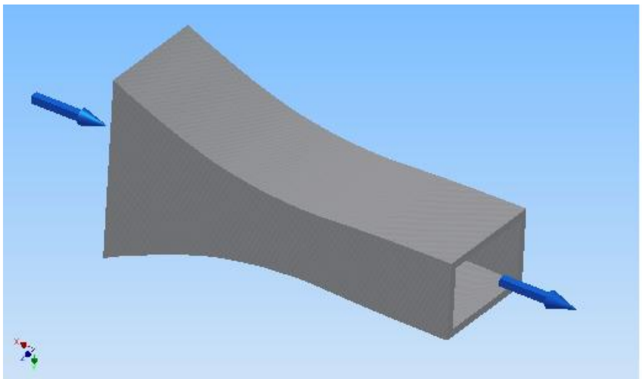
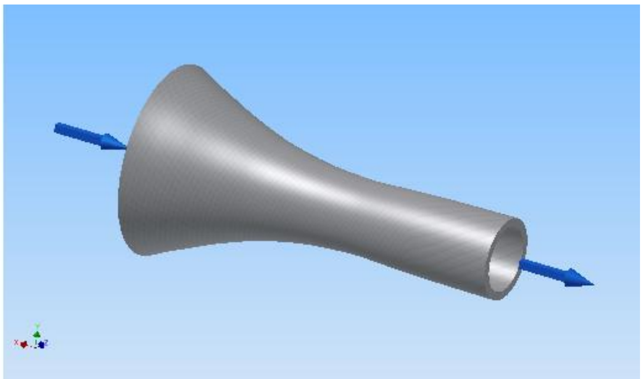

Figure 2. Two-dimensional (left) and axisymmetric (right) wind tunnel first throat.

The two-dimensional nozzle, also known as planar nozzle, is formed by two straight walls and two contoured walls. This configuration is found in most of the early supersonic nozzles, from the state of the art of that time's manufacturing methods were unable to create complex geometries such as axisymmetric nozzles, as well as the fact that the expansion of the air occurs in only one plane, simplifying the calculations. Axisymmetric nozzles are used in higher Mach numbers, when the stagnation temperature is raised, and also in low-density wind tunnels, since their boundary-layer growth is more uniform than that of planar nozzles. The major difference between two dimensional and axisymmetric nozzles is that while two-dimensional wind tunnels can easily modify their geometry to vary the test section Mach number, the axisymmetric version works only on the designed Mach number, and in order to modify it the entire throat has to be redesigned. This paper will analyze only the two-dimensional case, since the TTP has a planar first-throat.

One of the major problems of a supersonic wind tunnel design is to determine the best contour that creates uniform, parallel and shock-free flow in the test section. Many attempts to define this contour can be found since 1930, and even today this issue is still being studied as specific projects are needed (Shope, 2006, Shope and Aboulmouna, 2008, Dutta *et al.*, 2008).

The first who started designing the curvature of wind tunnel throat using the Method of Characteristics was Prandtl and Busemann (1929). The graphical method determined the walls contour that would transform a uniform flow at sonic Mach number to a uniform shock-free flow at a desired supersonic speed. The method consists of firstly finding the angle of Prandtl-Meyer (v) corresponding to the Mach number at test section and then, starting at the throat and proceeding downstream, determine the flow field in terms of the local Prandtl-Meyer angle and wall angle (v and θ), herein called by MoC (Method of Characteristics). Later in 1949 Foelsch demonstrated analytically how to construct the wall curvature solving the characteristics starting at the inflection point and proceeding to both sides based on Prandtl and Busemann method, simplifying extensive and tiring graphical works to approximate analytical results.

2.1. MoC – The Method of Characteristics

To understand the construction process of the divergent curve, an example is shown using five Characteristics lines in a two-dimensional supersonic axisymmetric nozzle with a required Mach number of 1.3 at test section. The strategy is to solve the characteristics in each point varying the angle of the wall from the throat to test section, well known as lattice-point method (Saad, 1993, Shapiro, 1953). Figure 3 shows this example in a two-dimensional supersonic nozzle. The origin is located at the throat (O), with the height of y_o, and with the length L until the test section entrance. The first step is to choose a mathematical function to represent the initial region contour in such way as to guarantee that the maximum wall deflection angle at its end is given by $\theta_i = \nu_{TS}/2$, where ν_{TS} is the Prandtl-Meyer angle related to the Mach number at the test section. Good candidates for this function could be an arc of circumference with adequate radius, a polynomial or other function that could be well adapted to the geometric conditions. The second step is to choose points in the initial region contour. To simplify the idea, Fig. 3 shows the region divided into five equally spaced sections relating to the symmetric direction, with five points defined, “*a*”, “*b*”, “*c*”, “*d*”, and “*e*”. The last point corresponds to the inflection point. According to the local Mach number and contour deflection, each contour point originates an expansion wave which travels in a determined direction. As the wave crosses other waves originated from other points, its direction is diverted many times (“1”, “2”, “3”, “4”, “5”, “7”, “8”, “9”, “10”, “12”, “13”, “14”, “16”, “17” and “19”) until reaching the straightening region contour (“6”, “11”, “15”, “18” and “20”), where a mathematical procedure is applied in order to create a compression wave to cancel the coming expansion wave.

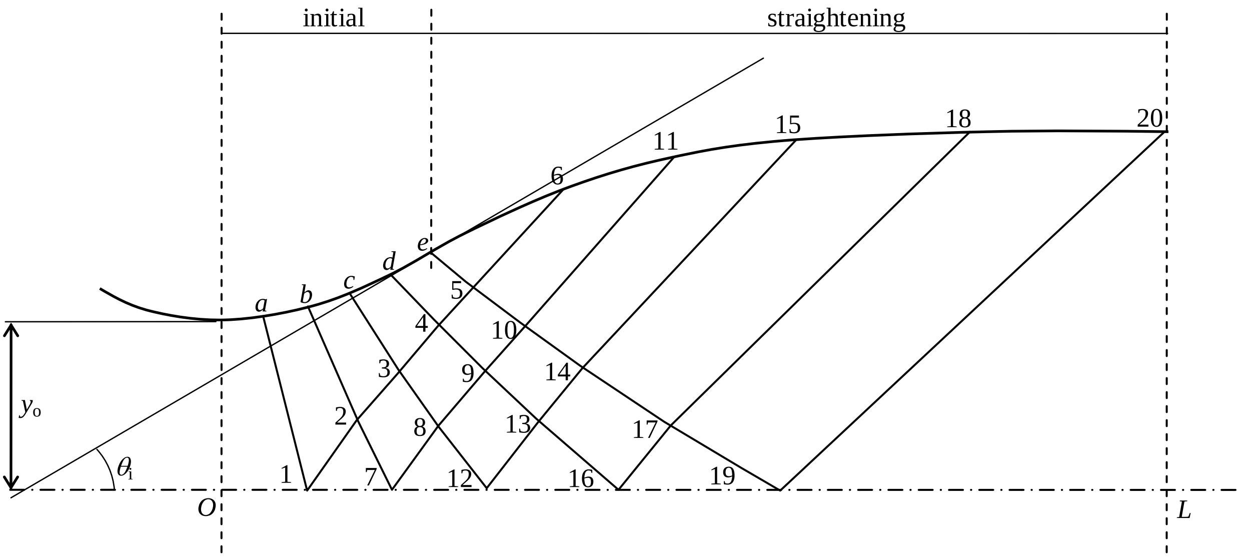

Figure 3. Scheme of supersonic nozzle design by the Method of Characteristics.

The calculation of the interference between crossing waves is facilitated by the use of characteristics lines, where the interception of the characteristics lines represents the values of *CI* e *CII* in the regions of the nozzle. Then, for any supersonic Mach number at point 1, the hodograph plane (mathematically represents the characteristics coordinates) is showed in Fig. 4. The conditions of the positive and negative characteristics *CI* and *CII* are established easily for the whole field: for $M = 1$: *CI* and *CII* = 0.

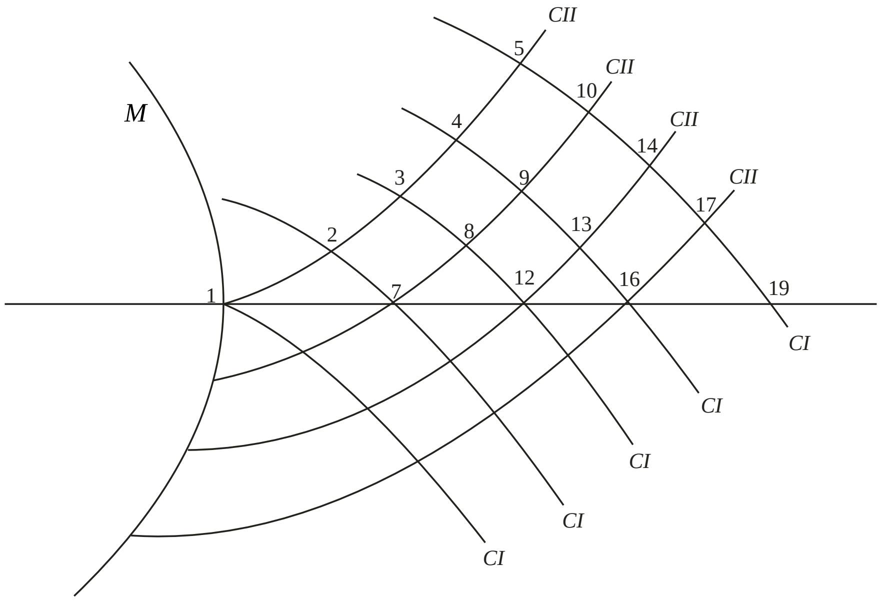

Figure 4. Hodograph plane showing the propagation of the characteristics.

By hodograph positive characteristic curves *I*: $CI_2 = CI_7$; $CI_3 = CI_8 = CI_{12}$; $CI_4 = CI_9 = CI_{13} = CI_{16}$; $CI_5 = CI_{10} = CI_{14} = CI_{17} = CI_{19}$. And by hodograph negative characteristic curves *II*: $CII_1 = CII_2 = CII_3 = CII_4 = CII_5$; $CII_7 = CII_8 = CII_9 = CII_{10}$; $CII_{12} = CII_{13} = CII_{14}$, $CII_{16} = CII_{17}$

And to conclude the constraints for wall curvature points: $CI_5 = CI_6$ and $CII_5 = CII_6$; $CI_{10} = CI_{11}$ and $CII_{10} = CII_{11}$; $CI_{14} = CI_{15}$ and $CII_{14} = CII_{15}$; $CI_{17} = CI_{18}$ and $CII_{17} = CII_{18}$; $CI_{19} = CI_{20}$ and $CII_{19} = CII_{20}$. Notice that the points on the wall own the same characteristics values as the previous regions, and this happens in order to guarantee that the compression wave originated at each point will cancel the expansion wave reaching it.

The use of the local characteristic, Eqs. (1) and (2), are the simplest and easiest way to solve the problem and to find the geometry of the nozzle contour by the characteristics lines, given in terms of deflection angle, θ, and Prantl-Meyer function, ν.

$$CI = \theta + \nu, \tag{1}$$

$$CII = \nu - \theta. \tag{2}$$

The Prandtl-Meyer angle is very important in compressible flow problems at high speeds because it is the key to the calculation of changes across an expansion wave (Anderson, 2001), and it is given by

$$\nu = \sqrt{\frac{\gamma+1}{\gamma-1}}\tan^{-1}\sqrt{\frac{\gamma-1}{\gamma+1}\left(M^2-1\right)} - \tan^{-1}\sqrt{M^2-1}, \tag{3}$$

where γ is the ratio of the specific heats of the gas and M is the local Mach number.

This methodology could cause bad formations at the beginning of the curve formed by wave reflections, and this can occur for two reasons: excessive discretization points and an initial curve arc with extremely long radius. In both cases it would result in the new points being determined still on the initial curve. Otherwise, if the necessary quantity of points is not attempted or the arc is too small, enormous blank spaces are created mainly between the inflection point and the first point in the straightening region resulting in undesired corners.

2.2. Foelsch Method

The Foelsch Method uses the same ideas already presented of canceling the expansion waves created in the initial region by compression waves mathematically calculated in the straightening region, but has analytical advantages. The equations for the nozzle's contours are derived by integrating the characteristic equations of the axially symmetric flow generated at a conical source, comparing the conditions of the flow in this cone with those in the nozzle, as a linearization of the characteristic equations. The cylindrical flow section is converted from a two-dimensional section to a straight circular section.

Figure 5 shows the geometric definition parameters for Foelsh method, which is based on an analogy of a conical propagation of the flow from a virtual origin *O'*, which is located upstream the nozzle throat located at point *O*. At the inflection point the contour angle is maximum (θ_{inf}) and it corresponds to a determined local Mach number. From this method, in the region between the arc formed from the inflection point to the channel center line and the Mach line with length *L*, the Mach number is only a function of the radius. In the figure, the radius evolution line shows the locus of the radius corresponding to Mach number variation from the inflection point to test section entrance, given by coordinates (x_2, y_2). And one can still consider a virtual value corresponding to sonic Mach number, R_o. At a generic point the Mach number *M* corresponds to the expansion from sonic Mach number through an area ratio given by R/R_o, reaching the test section Mach number value at the channel center line.

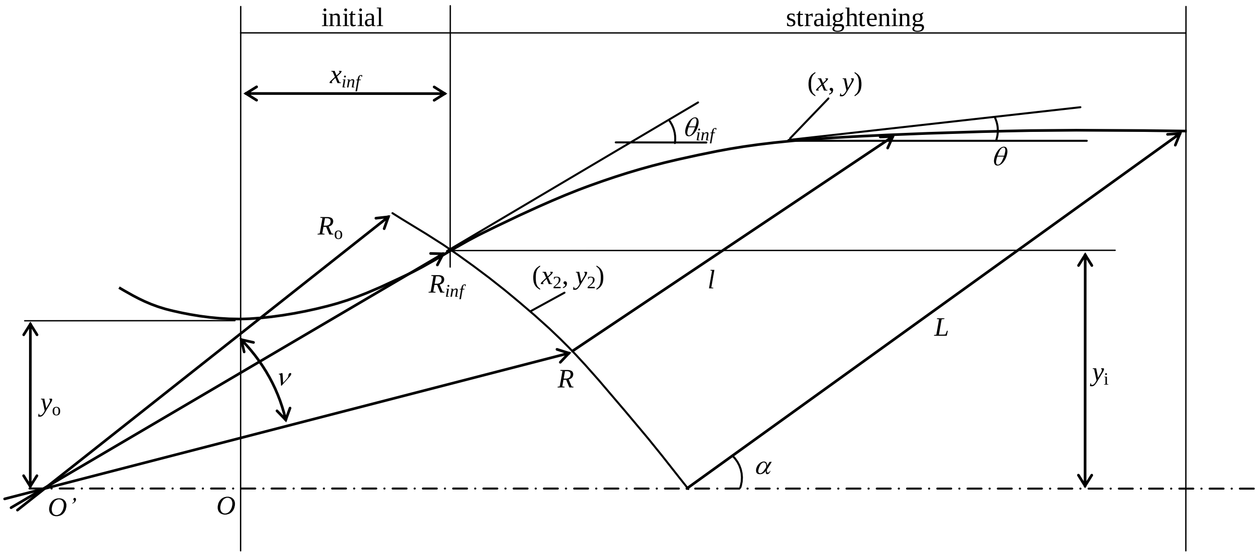

Figure 5. Variables used in the Foelsh's scheme.

From Figure 5, the main parameters like the hypothetical distance for unitary Mach number (R_0), and the distance between the origin arbitrary point and any point located on inflection Mach line (*R*) are given by

$$R_0 = \frac{y_0}{\theta_{\text{inf}}}, \tag{4}$$

$$R = R_0 (A/A_0), \tag{5}$$

the index (0) represents the throat conditions, and the area ratio as a function of the Mach number well known in Fluid Mechanics, is given by

$$\frac{A}{A_0} = \sqrt{\frac{1}{M^2}\left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M^2\right)\right]^{\gamma+1/\gamma-1}}. \tag{6}$$

The height of inflection point is

$$y_{\text{inf}} = \frac{y_0 \sin\theta_{\text{inf}}}{\theta_1}\left(\frac{A_{\text{inf}}}{A_0}\right), \tag{7}$$

and the length of Mach line between the Mach line formed at inflection point and the final curve is

$$l = M\ R\left(\nu - \nu_{\text{inf}}\right). \tag{8}$$

The coordinates of the Mach line formed at inflection point are

$$x_2 = x_{\inf} - R_{\inf} \cos\theta_{\inf} + R\cos(\nu_{TS} - \nu), \tag{9}$$

$$y_2 = R\sin(\nu_{TS} - \nu). \tag{10}$$

To find the final coordinates for wall curvature it is needed to know the Mach angle, given by

$$\alpha = \sin^{-1}\frac{1}{M}. \tag{11}$$

Finally, the wall contour can be found by

$$x = x_2 + l\cos(\nu_{TS} - \nu + \alpha), \tag{12}$$

$$y = y_2 + l\sin(\nu_{TS} - \nu + \alpha). \tag{13}$$

2.2.1. Inflection Point

While generating nozzle walls, it is normal that imperfections and discontinuities on inflection point appears, where there is a junction of two different curves and change from radial to planar flow, as can be seen in Fig. 6. The fact that this method uses curves with distinct behavior creates a difficulty to fit their intersection, but according to Evvard and Marcus (1952) some auxiliary restrictions acting like boundary conditions in the routine should be necessary and sufficient to avoid these complications. *A priori* the imposed conditions to the inflection point was defined by Eqs. (14) and (15). For the present problem the gap observed in the inflection point was 1.6 mm, and in some cases this value does not cause significant error in Mach number distribution.

$$\nu_{\inf} = \nu_{TS} - \theta_{\inf}, \tag{14}$$

$$\theta_{\inf} = \nu_{TS}/2. \tag{15}$$

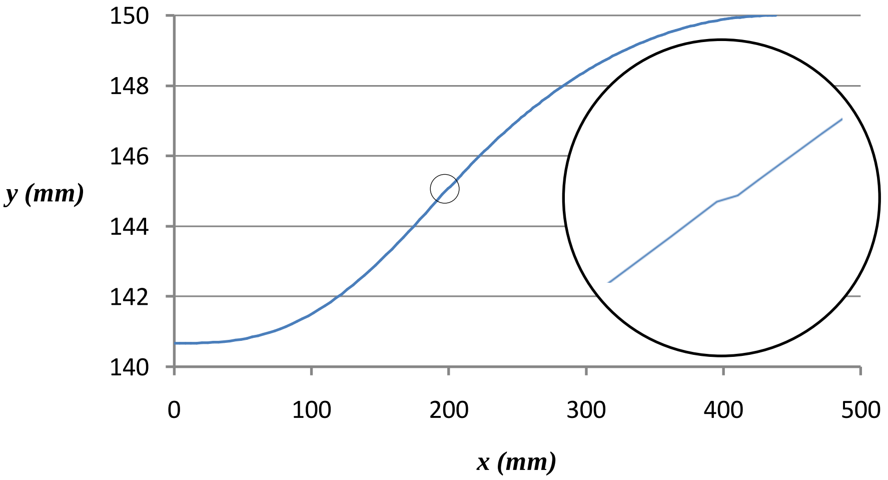

Figure 6. Wall curvature output with detail on discontinuity of the inflection point.

2.2.2. Initial Curve

According to Crown (1948), a good approximation of the initial curve can be given by the third degree equation, where experiences have shown reliable results in designs based on Puckett (1946) and Foelsh methods. In this case, the geometry is function of x in each point, deflection angle and position of the inflection point on the wall and throat's half-height. In this step of calculation it is very important to achieve the main objective of the project and, thus the increment *dx* has to guarantee a smooth and monotonic curvature to avoid the strong expansion waves originated from the corners. One legitimate candidate is

$$y = y_0 + \left[\frac{\tan\theta_{\inf}}{x_{\inf}^{\,2}} x^3 \left(1 - \frac{x}{2x_{\inf}} \right) \right]. \tag{16}$$

The inflection point is found by

$$x_{\inf} = \frac{2}{\tan\theta_{\inf}} \left(y_{\inf} - y_0 \right), \tag{17}$$

provided that $y_{\inf}$ is obtained from the area ratio $R_{\inf}/R_0$.

Another initial curve also used and found in literature is the arc of circumference. The arc is generated by a radius *R*, which satisfies the tangent condition equal to zero at the throat and fits perfectly in the inflection point. These geometric restrictions are described by Eqs. (18) and (19).

$$y = y_0 - R + \sqrt{R^2 + x^2}\ , \tag{18}$$

$$R = \sqrt{\frac{x_{\inf}^{\,2}}{\tan^2\theta_{\inf}} + x_{\inf}^{\,2}}\ . \tag{19}$$

One more initial curve was tested: a fourth degree polynomial, with the slope increasing monotonically from the throat to the inflection point, where θ is maximum, and decreasing monotonically from the inflection point to the test section entrance. It is given by

$$y = Ax^4 + Bx^3 + Cx^2 + y_0\,. \tag{20}$$

Figure 7 shows the first derivative of the contour when Eq. (20) is applied to the problem, demonstrating the good behavior of the selected function, as required for good throat designs (Shope, 2006).

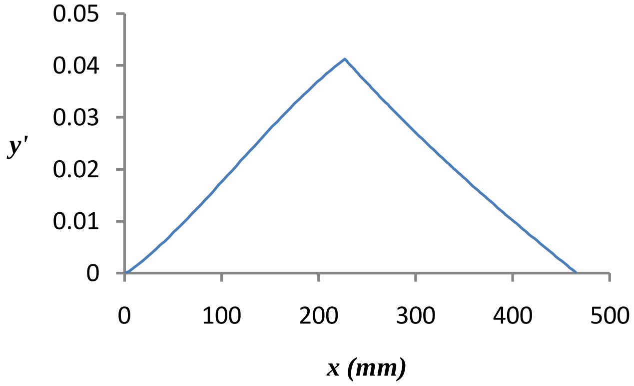

Figure 7. Slope of the supersonic first-throat surface for fourth degree polynomial in the initial curve.

For this case, it was used a lower value for the inclination at the inflection point ($\theta_{inf} = \nu_{ts} / 2.6$), which has proved to be the best solution for the throat design. The value for the constant *A* which represents a best fit between the two curves at the inflection point was -0.4472.

From Eqs. (21) and (22) the other constants were

$$B = \frac{y_{\inf} - y_0}{1.5\, x_{\inf}^{\;3}} + \frac{2\, A\, x_{\inf}^{\;2}}{3} - \frac{3 \tan \theta_{\inf}}{x_{\inf}^{\;2}}\,, \tag{21}$$

$$C = \frac{\tan \theta_{\inf}}{2\, x_{\inf}} - 2\, A\, x_{\inf}^{\;2} - 1.5\, B\, x_{\inf}\;. \tag{22}$$

3. RESULTS

Initially, an evaluation of the Mach number distribution in the test section region was performed to verify which method is more effective in transonic flow. The two methods were applied to the design of a first throat for Mach number 1.3 at test section. Figure 8 shows a comparison of the two methods in terms of Mach number at center line for the best solutions of MoC and Foelsch methods without any adjustments. For MoC curve, a radius of eight times the half of throat height was used. It is important to emphasize that an increase of the nozzle length results in better flow behavior, where weaker reflective waves can be observed. Many attempts were done to vary the parameters of the MoC method. Although all modifications made in it, the Mach number distribution found was very close to the Foelsch method result, without any adjustments. For the nominal test section region, the Mach number observed in MoC was 1.3017± 0.0033, with a maximum Mach number variation of 0.0104; and in Foelsch method was 1.3008± 0.0042, with a maximum variation of 0.0138.

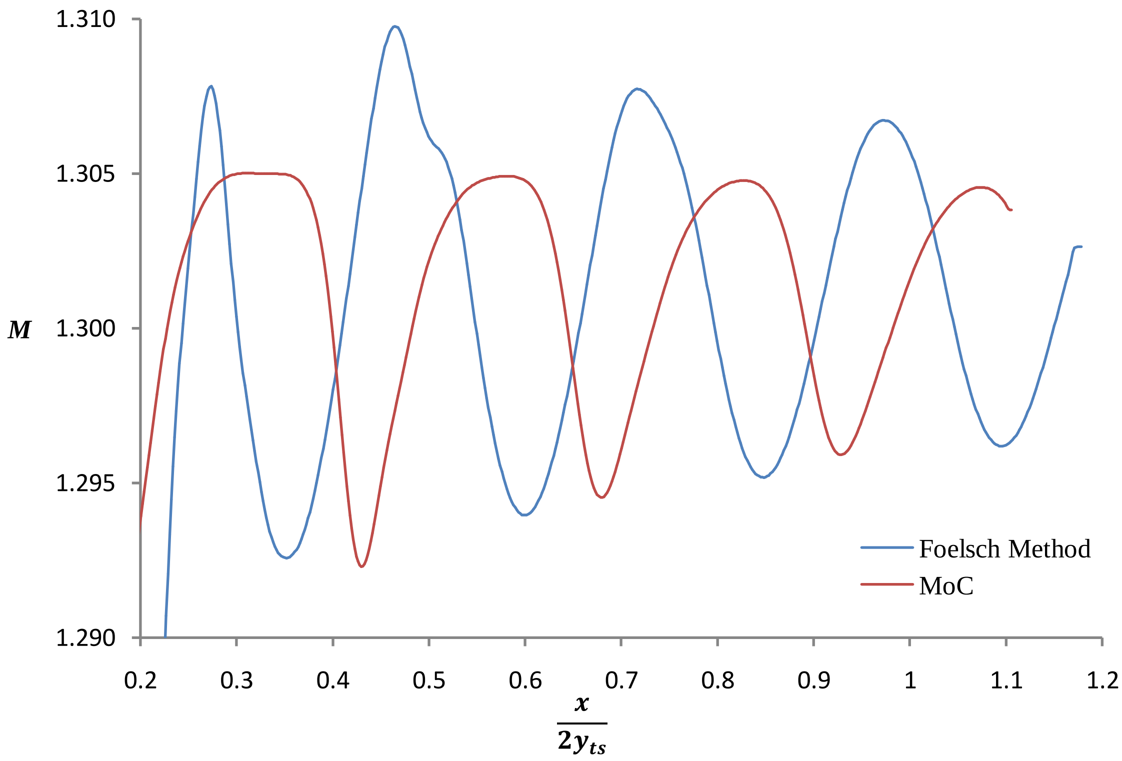

Figure 8. Comparison of Mach number distribution between MoC and Foelsch method.

The following research explored the parameters of Foelsch method, varying θ_{inf}, which resulted in length increasing, and the initial curve. Remembering that the TTP is already installed, and it has already defined dimensions and also because increasing the throat's length too much would result in an excessive pressure loss, the variation of θ_{inf} was limited to $\nu_{ts}/2.6$. The results of four different inflection point angles are shown in Fig. 9, and prove that the flow becomes more uniform when the inflection point is located forward, making the change from expansion to compression flow slightly smoother.

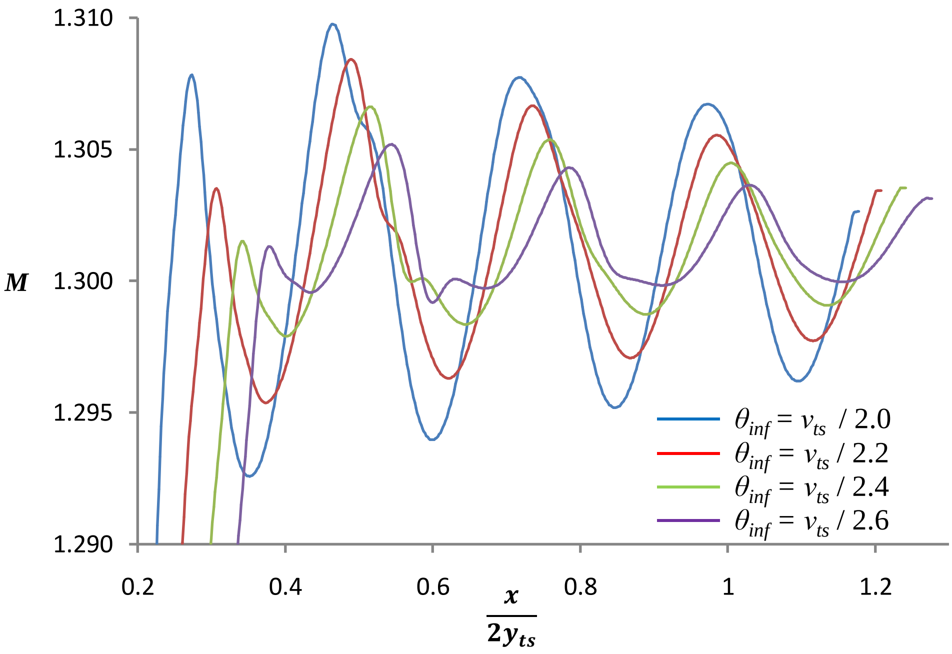

Figure 9. Results of Mach number for different inflection point angle.

Some authors like Foelsch (1949) and Shope (2006) used an arc of circumference to construct the initial curve. Others, like Crown (1948) and Belderrain and Abreu (1978) used a third degree polynomial, so these two well-known initial curves and a new methodology with a fourth degree polynomial were calculated. Figure 10 shows the results of Mach number distribution in the test section for each type of initial curve, and shows clearly the improvement of the fourth degree polynomial in comparison with the others. The Mach number in the test section was 1.3011±0.0014 with maximum ΔM = 0.0046. This is the best configuration found until now, and it represents the best approximation to the optimum criteria adopted in well-designed transonic wind tunnels (Davis *et al.*, 1986): two-sigma deviation in Mach number over the region of the model would be $|2\sigma_M| \leq 0.001$ and the maximum ΔM = 0.001. The overview result of the whole supersonic first-throat is shown in Fig. 11 and a numerical simulation result using non-viscous Euler equations formulation.

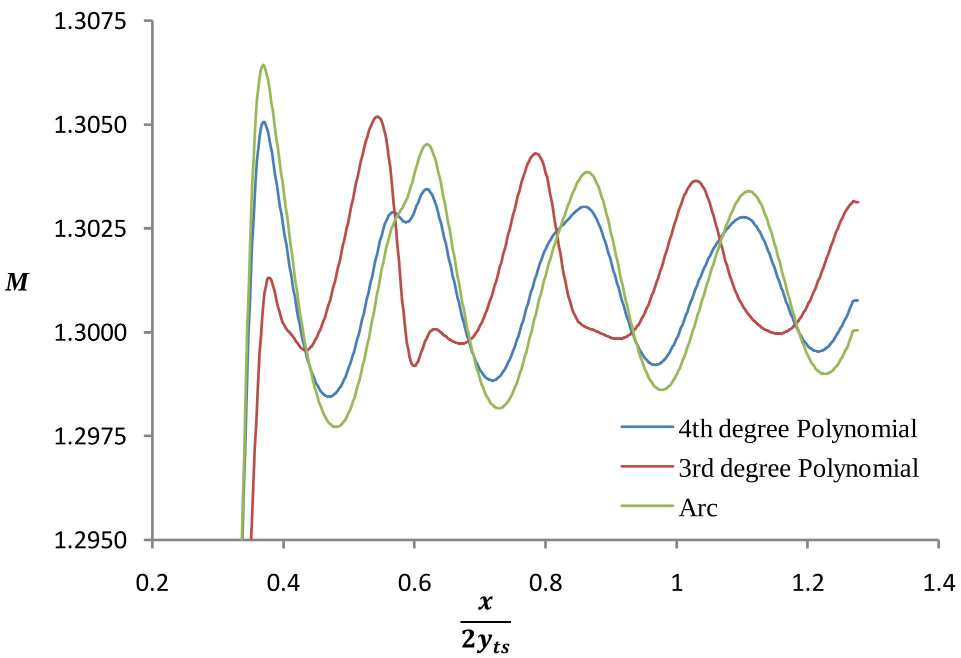

Figure 10. Results of Mach number distribution at test section for three kind of initial curves.

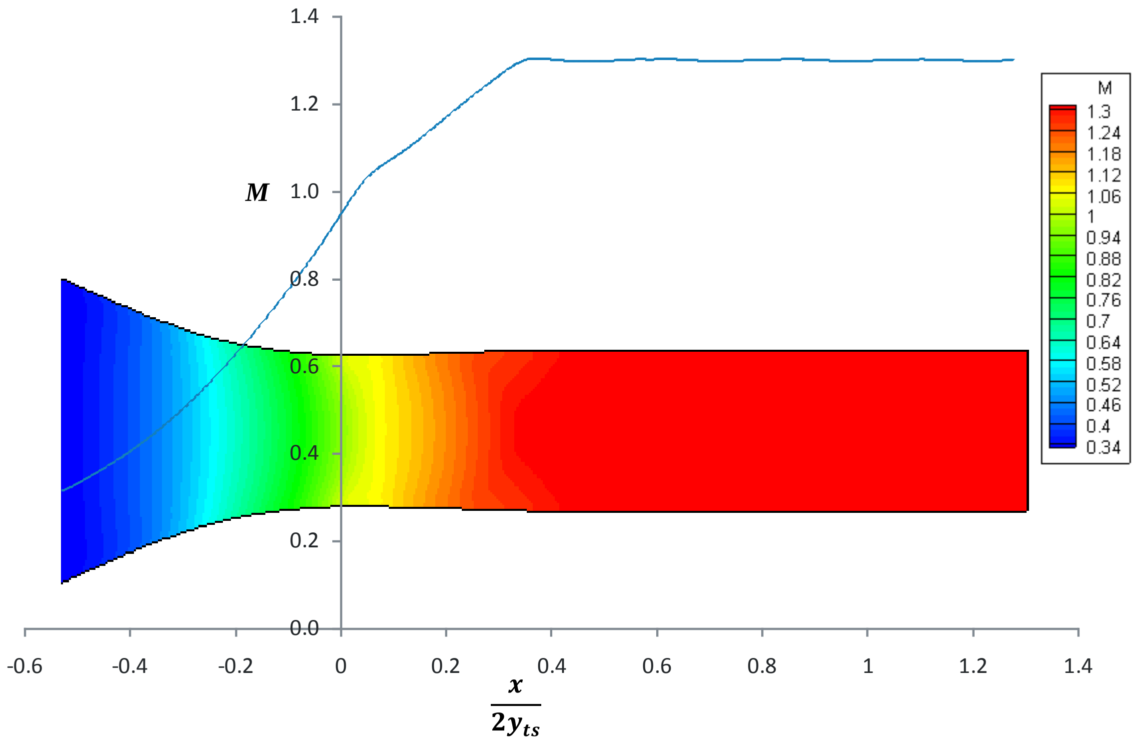

Figure 11. Result of numerical simulation with Mach number output at center line.

4. CONCLUSIONS

The supersonic first throat design procedures using MoC and Foelsch methods were accomplished. Comparisons of Mach number distribution at center line were performed varying main parameters in the initial curve and inflection point angle. The first throat contours were simulated by numerical code based on Euler equations, and the best configuration for TTP was found: Foelsch method with inflection point angle of $\nu_{ts}/2.6$ using a fourth degree polynomial in the initial curve, which demonstrated good performance with low deviation and variation of Mach number distribution.

5. ACKNOWLEDGEMENTS

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6. REFERENCES

Anderson, J.D., 2001, *Fundamentals of Aerodynamics*, McGraw-Hill Series in Aeronautical and Aerospace Engineering, 3rd. ed.

Belderrain, J.L.R., Abreu, F.L., 1978, “Anteprojeto de um Túnel de Vento Supersônico”, Trabalho de Graduação, Instituto Tecnológico de Aeronáutica.

Crown, J.C., 1948, “Supersonic Nozzle Design”, NACA TN-1651.

Davis, M. W., Gunn, J. A., Herron, R. D., Kraft, E. M., 1986, “Optimum transonic wind tunnel,” AIAA 14th Aerodynamic Testing Conference, 14, West Palm Beach, *AIAA*-86-0756-CP.

Dutta, V., Dutta, P.K., Sajjan, S,V., 2008, “RANS Computations for Flow Through a Variable Mach Number Flexible Nozzle at Mach 4”, Proceedings of the International Conference on Aerospace Science and Technology, INCAST 2008-056, India.

Escosteguy, J.P.C., 1997, “Ensaios Iniciais no Túnel Transônico Piloto do CTA”, Proceedings of 7th Brazilian Congress of Engineering and Thermal Sciences, vol.1, Rio de Janeiro-RJ, Brazil.

Evvard J.C., Marcus L.R., 1952, “Achievement of Continuous Wall Curvature in Design of 2-D Symmetrical Supersonic Nozzles”, NACA TN-2616.

Foelsch, K., 1949, "The Analytical Design of an Axially Symmetric Laval Nozzle for a Parallel and Uniform Jet." *Journal of the Aeronautical Sciences*, pp. 161-166.

Prandtl, L., Busemann, A., 1929, "Näherungsverfahren zur Zeichnerischen Ermittlung von Ebenen Strömungen mit Überschallgeschwindigkeit", *Stodola Festschrift* (Zurich), 1929, S. 499-509. – Cited by: Evvard and Marcus (1952).

Puckett, A.E., 1946, "Supersonic Nozzle Design", *Journal of Applied Mechanics*, vol. 13, n. 4, pp. A-265-270.

Saad, M.A., 1993, *Compressible Fluid Flow*, Prentice Hall, 2nd. ed.

Shapiro, A.H., 1953, *The Dynamics and Thermodynamics of Compressible Fluid Flow*, vol. 1, The Ronald Press Company, New York.

Shope, F.L., 2006, "Contour Design Techniques for Super/Hypersonic Wind Tunnel Nozzles", 24th Applied Aerodynamics Conference, AIAA 2006-3665.

Shope, F.L., Aboulmouna, M.E., 2008, "On the Importance of Contraction Design for Supersonic Wind Tunnel Nozzles", 26th AIAA Aerodynamic Measurement Technology and Ground Testing Conference, *AIAA* 2008-3940.

Silva, A.F., Braz, R.O., Avelar, A.C., Falcão Filho, J.B.P., 2009, "Study of the Mach Number Uniformity over a Horizontal Plane inside the Test Section of a Pilot Transonic Wind Tunnel", Proceedings of the 20th. International Congress of Mechanical Engineering, Gramado-RS, Brazil.

Souza, F.M., Falcão Filho, J.B.P., De Oliveira Neto, P.J., 2009, "First Throat Design of a Transonic Wind Tunnel", Proceedings of the 20th. International Congress of Mechanical Engineering, Gramado-RS, Brazil.

Sverdrup Technology Inc., 1989, "Brazilian transonic wind tunnel concept definition study", São José dos Campos: CTA-IAE, Contractor Report for TTS and TTP Projects.

Zanin, R.B., Reis, M.L.C.C., Falcão Filho, J.B.P., 2008, "Análise da Uniformidade Longitudinal do Número de Mach na Seção de Testes do Túnel Transônico Piloto do IAE em Circuito Aberto", Proceedings of 5th National Congress of Mechanical Engineering, Salvador-BA, Brazil..

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