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SUPERSONIC INJECTOR DESIGN FOR USING IN A MIXING CHAMBER

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Abstract. *The Pilot Transonic Wind Tunnel (PTT), located in the Aerodynamic Division (ALA) of the Institute of Aeronautics and Space (IAE), is a modern installation with a conventional closed aerodynamic circuit, continuously driven by a main compressor and with an intermittent injection system which operates in a combined mode to extend the tunnel´s operational envelope. The tunnel has automatic pressure controls (from 0.5 bar to 1.25 bar), Mach number (from 0.2 to 1.3), temperature and humidity, related to its test section (0.30 m wide and 0.25 m height). TTP is an 1/8th scale down from the industrial transonic project, and it was initially designed to study the innovative features of the industrial facility, specifically concerned with the injection system operation in combination with the conventional main compressor operation, but it was also designed for training the technical team in high speed phenomena, to perform basic research and academic research assessment, to produce tests in the development of new vehicles, and others. To explore the capabilities of its injection system, a parametric experimental study was idealized for its supersonic mixing chamber. However, since TTP is already in operation, modifications in the aerodynamic circuit components would interfere with the test campaigns. So, it was recently built a supersonic injection mixing chamber separated from the tunnel´s circuit, but still utilizing its compressed air system to drive one supersonic injector in it. Some of the assessed parameters related with the mixing chamber design are chamber lenght, area ratio, diffuser length and angle, injector Mach number etc. The mixing chamber section is 0.17 m wide and 0.23 m height and it has one supersonic injector with the same geometry of the TTP injection system, with the possibility of changing its angle, and one flat diffuser with approximately the same equivalent performance of the TTP´s conical diffuser. The supersonic nozzle design literature is very rich and one can find many design aspects being addressed for many purposes, since Prandtl e Busemann who laid the foundation for determining the inviscid nozzle contours by the method of characteristics. This work describes the calculation details to design a geometrically suitable supersonic beak to be used in the mixing chamber for four different outlet Mach numbers (1.6, 1.9, 2.2 and 2.5) based on the method of characteristics. The geometry guarantees a perfect fitting over the wall and an easy connection with pressurized air tubing from both sides of the injector stagnation chamber, and it is very useful for many high speed applications. The 1.9-Mach-number injector design is analyzed by a CFD code based on the Euler Equations to assess its flow quality (absence of compresion waves) and its results compared with the experimental data from the injector already constructed.*

Keywords: *Injection Nozzles Design, Mixing Chamber, Supersonic Flow, CFD, Characteristics Method*

1. INTRODUCTION

The Pilot Transonic Tunnel (PTT), located at the Institute of Aeronautics and Space (IAE), one of the institutes that embody the Aerospace Technology General-Command in São José dos Campos – Brazil, is a reduced-scale version of an industrial transonic wind tunnel project. It is a closed loop tunnel with a 0.25 m × 0.30 m × 0.81 m slotted test section, powered by an axial compressor and an injection system that enable the tunnel to reach Mach number from 0.2 to 1.3 in a continuous operation. To verify the capabilities of its injection system, a parametric experimental study was idealized for its supersonic mixing chamber (Silva *et al.*, 2010). However, modifications in the aerodynamic circuit components are of difficult realization since the tunnel is already operational. So, it was recently built a supersonic/subsonic injection mixing chamber separated from the tunnel´s circuit, but still utilizing its compressed air system to drive one supersonic injector in it. This injector has the same geometry used in the PTT, which was developed by Sverdrup (1989).

Supersonic injectors and high-speed nozzles have been used in the modern industry since the beginning of the twentieth century for several engineering applications. A first example of these pioneer applications is the compressed air jet discharge in an annular nozzle as a power supply for wind tunnels operation (Stack, 1933). The use of injectors in the PTT is due to Nogueira (Nogueira *et al.*, 1988), during the conception step of the Brazilian Industrial Transonic Wind Tunnel design, when he was in charge of it.

The rocket nozzle design are, in general, related to axissymmetric geometries that ends in divergence angle to assure the maximal trust, as states innumerous published work in the literature. The same nozzle type is also used to energize a lower-speed flow in a confined geometry, very frequently found in wind tunnels. There are also cases, as the first throats in supersonic wind tunnels, where two-dimensional geometry with the possibility of small changes in it are required to allow variation of nominal Mach number. Many important works on this kind of application has been published and two of them also contain key FORTRAN codes. An example in the design field is the work of Sivells (1978), which addresses the axissymetric and planar nozzle design for supersonic/hypersonic wind tunnels. The computational routine represents the final result of a intense effort dispended along several years at the AEDC ("Arnold Engineering Development Center") and at AFSC ("Air Force Systems Command"), in order to develop a nozzle design methodology with inviscid contour geometry and boundary layer correction that ensures a uniform and parallel flow at the exit. Another example, now in the analysis field, is the work of Anderson (1974), which presents an analytic calculation procedure of the performance and characteristics of the flow field of the supersonic injectors, including the sonic line effect calculation and an analysis of mixing process between the primary and secondary flows of the injector nozzle.

Particularly, the PTT supersonic nozzle design follows the idea of the nozzle exit flowing parallel with respect to the main flow and located close to the tunnel walls, helping to energize the boundary layers and avoiding premature recirculation phenomena in the diffuser walls.

The major problem to be faced in a supersonic nozzle design is to determine the contour of the straightening section, where the divergence angle derivative becomes negative, that guarantees the absence of contraction waves along the nozzle. Although the Fluid Mechanics Theory provides the means to analyze the flow condition for a given contour, the necessary tools to determine the optimum contour are not obvious. Consequently, several papers on this subject have been published in recent decades that widely vary in complexity and applicability.

One of the most important works published is that written by Prandtl and Busemann (1929), which presents a mathematical approach to the nozzle design based in the concept of the neutralization of the expansion waves. This work is an application of such theory, creating nozzle geometries especially suitable for the perfect assembling with the air compressed tubing and for a good alignment of the jet with the mixing chamber walls. The work of Prandtl and Busemann (1929) in German, is being cited in many relevant works about supersonic nozzle designs (Crown, 1951, Evvard and Marcus, 1952, Sivells, 1978).

2. THE CHARACTERISTICS METHOD

A typical convergent-divergent nozzle is shown at Fig. 1, in which it can be highlighted three main distinct regions: the convergent section in which the flow is accelerated from low subsonic (stagnation condition) (1) to the sonic condition at the throat (2), the divergent section, in which the flow accelerates to the supersonic regime from the throat, passing by the inflexion point (3) until the maximum acceleration condition (design point) (4), and, finally, the outlet flow characterized by a uniform and parallel flow from (4) on.

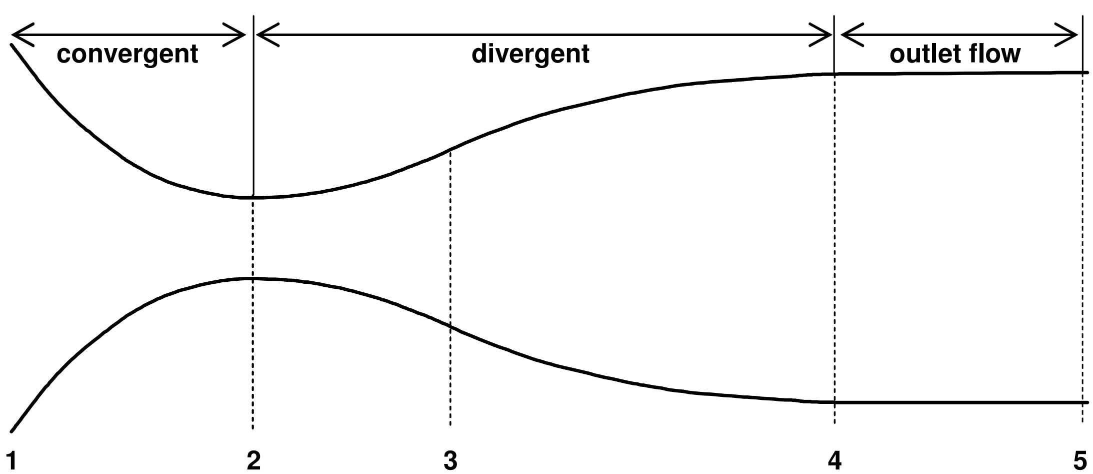

Figure 1. Typical convergent-divergent nozzle.

The main problem to be faced during the design process of a nozzle of this type is to assure that the exit flow to have a good quality: a uniform and parallel flow with absence of significant pressure gradients – without compression waves.

From the throat position on, the flow suffers an expansion until the inflection point (3). Expansion waves can be observed in this region and the flow, now supersonic, follows the nozzle walls. From the inflection point on, the section area continues to increase but the flow near the walls begins to turn into itself generating compression waves that propagates along the flow together with the expansion waves already mentioned. So, the main idea to be followed here is to look for a geometry from the point (3) until the point (4) that propitiates the perfect cancelling of such waves generating a high quality flow. This kind of goal can be achieve by the application of a theoretical tool known as the characteristic method.

Although the method of characteristics is thoroughly discussed in the literature, it will be resubmitted briefly for the specific application of this work.

The Method of Characteristics is basically a mathematical approach to solve partial deferential equations by transforming them into ordinary differential equations. A widely spread application of such theory is in the Fluid Mechanics in the solution of the Euler Equations and in the definition of the boundary conditions of numerical problems. In this work, this tool is used to allow the design of the straightening sector of a supersonic injector nozzle (from (3) to (4) in Fig. 1), so that the appearance of compression waves is avoided.

The basis of this theory is grounded on the establishment of the field directions on which the flow variables derivatives are indeterminate and may even be discontinuous. To verify the existence of such directions, recall to the Two-Dimensional Velocity Potential Equation (that holds for a steady, irrotational, inviscid and isentropic flow) written in terms of velocities (Anderson, 1990) shown in Eq. (1), where u and v, represent, respectively, the flow velocities along the x and y axis, and a represents the local sound speed,

$$\left(1-\frac{u^2}{a^2}\right)\frac{\partial u}{\partial x}+\left(1-\frac{v^2}{a^2}\right)\frac{\partial v}{\partial y}-\frac{2uv}{a^2}\frac{\partial u}{\partial y}=0. \tag{1}$$

If Eq. (1) is solved for $\partial u/\partial x$, it can be obtained

$$\frac{\partial u}{\partial x}=\frac{\frac{2uv}{a^2}\frac{\partial u}{\partial y}-\left(1-\frac{v^2}{a^2}\right)\frac{\partial v}{\partial y}}{\left(1-\frac{u^2}{a^2}\right)}. \tag{2}$$

Note that the velocity derivative is well established with one notable exception: when the local velocity along the x axis is equal to the local sonic speed. Such case is illustrated in Fig. 2.

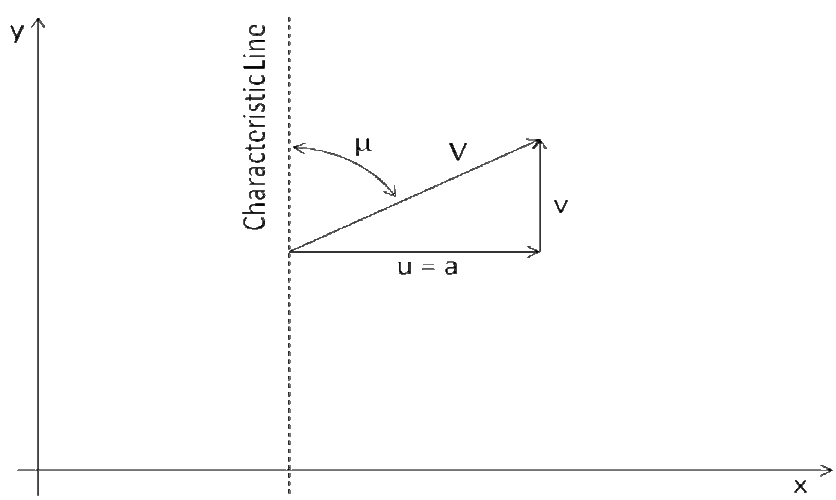

Figure 2. Illustration of a characteristic direction.

From the geometry in Fig. 2 one can show that the angle μ corresponds to the Mach angle, given by

$$\sin\mu = \frac{u}{V} = \frac{a}{V} = \frac{1}{M} . \tag{3}$$

Note that the coordinate system used was arbitrary and so, it can be concluded that along the direction that makes a Mach angle with respect to the streamline direction, the derivative of the velocity u is indeterminate. Such directions are known as characteristic directions. Furthermore, one can show that along these directions some variables, known as characteristic variables (K), are conserved, *i.e.*, equals a constant. According to Anderson (1990), these variables can be expressed by Eq.(4), where θ is the angle of the streamline with respect to the x axis and $\nu(M)$ is the Prandtl-Meyer function with respect to the local Mach number M,

$$K = \theta + \nu(M) = cte . \tag{4}$$

Consider now a two-dimensional flow about a corner of infinitesimal angle as shown in Fig. 3. If the flow is turned into itself, it appears an infinitely weak compression wave that corresponds to a Mach wave. Analogously, if the flow is turned out of itself, it occurs an infinitely weak expansion wave which direction can be also approximated by a Mach wave (Crown, 1948). It has to be remembered too that such waves can be reflected by boundaries such as walls and in these cases the well-known mirror-image concept can be applied as illustrated in Fig. 4.

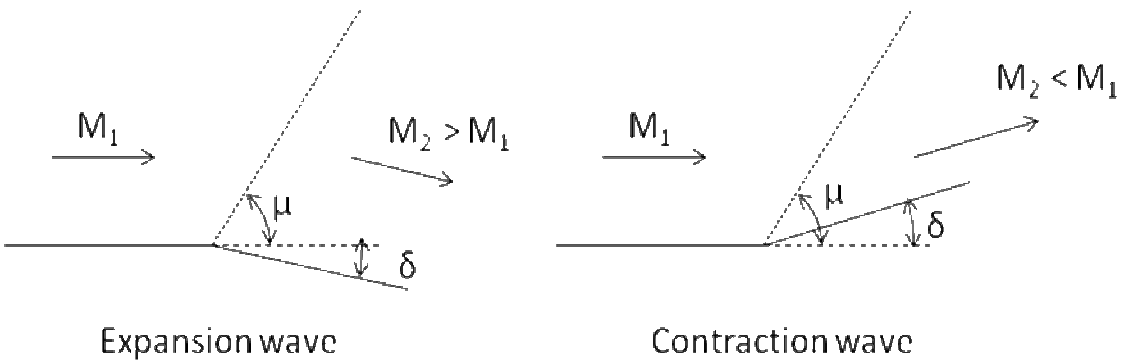

Figure 3. Flow about an infinitely small corner.

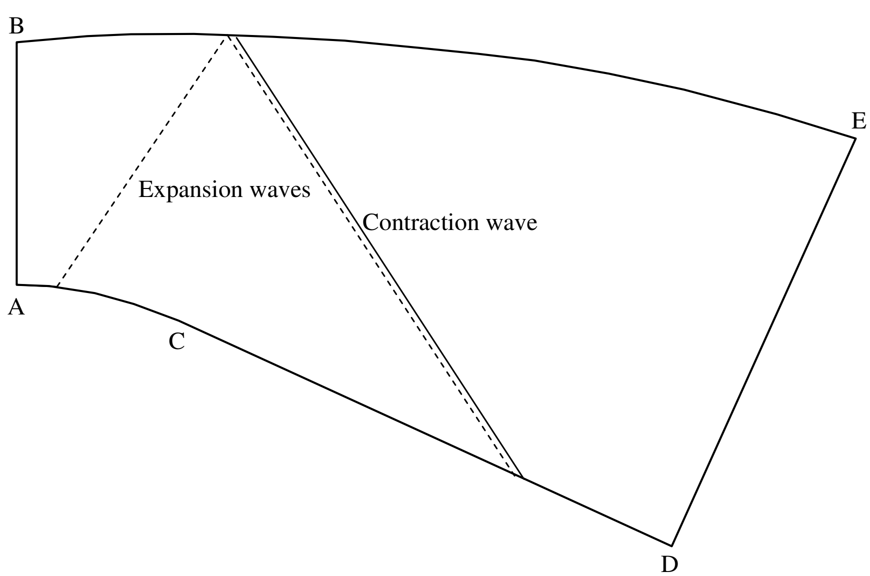

Figure 4. Neutralization of a contraction-expansion wave pair inside the injector nozzle.

Recalling what was said in the beginning of this section, the main goal the method here presented is to cancel the reflection of the expansion wave generated by the acceleration of the flow with the contraction wave generated by the straightening section (decreasing divergence angle). So, in this case, the only waves present in the nozzle are the expansion waves due to the acceleration of the flow. For such a situation the characteristic lines are straight lines (if two waves intersect at a point, the directions of both waves change and, consequently, the characteristic directions, remembering that both coincide for infinitely weak waves). It is exactly this theoretical tool that is especially useful to this nozzle design methodology.

3. THE INJECTOR NOZZLE DESIGN

The design method here presented has the accomplishment of being simple and also guarantees a well established flow in the nozzle exit. It is mainly indicated to injection systems where it is desirable that the supersonic jet to be parallel to the wall surface and present a good solution to the problem of the injector/compressed air system linkage.

The methodology needs the following initial parameters: air compressed tubing radius (r), the number of Mach in the exit of the injector (M_E), the specific heat ratio for the gas (γ) and the height of the exit section of the injector (H). Figure (5) represents the general shape of the injector, as the notable points in the design.

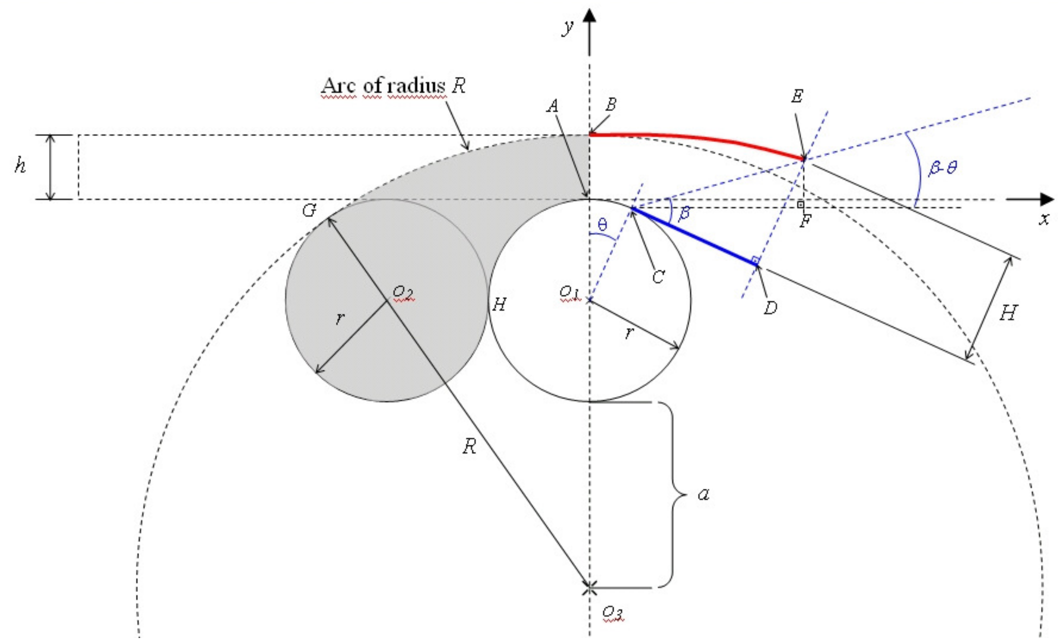

Figure 5. Geometric representation of the supersonic nozzle design.

Note that the geometry is basically laid upon three arcs of circunference (two of radius r and one of radius R), a line segment between the points C and D ($\overline{CD}$), and a point to point calculated curve between the points B and E. The points H and G are points of tangency between the circunferences and C is a point of tangency between the line segment $\overline{CD}$ and the circunference of center at O_1.

Initialy, the coordinates of the point A are set to be (0,0) and the other points calculated with referece to it. So, the nine following steps should be observed:

1. The center of the first circunference of radius r is set at the point O_1 at the coordinates given by

$$x_{O_1} = 0 \quad , \quad y_{O_1} = -r \,. \tag{5}$$

2. The center of the second cincunference of radius r is set in the point O_2 at at the coordinates given by

$$x_{O_2} = -2r \quad , \quad y_{O_2} = -r \,. \tag{6}$$

3. The throat height h is calculated from the exit section height H and the exit Mach number M_E using the sonic area ratio equation (Shapiro, 1953), remembering that the depth of the injector is constant,

$$\left(\frac{H}{h}\right)^2 = \frac{1}{{M_E}^2}\left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}{M_E}^2\right)\right]^{\frac{\gamma+1}{\gamma-1}}. \tag{7}$$

4. The point B is, then, set at the coordinates given by

$$x_B = 0 \quad , \quad y_B = h. \tag{8}$$

5. Taking the auxiliary parameter a as defined by the Fig. 5, the bigger circumference radius R is given by

$$R = a + 2r + h. \tag{9}$$

If the triangle defined by the points O_1, O_2 and O_3 is taken, it can be written

$$(R-r)^2 = (2r)^2 + (a+r)^2. \tag{10}$$

Finally, using the previous two equations the value of R can be obtained as expressed by

$$R = \frac{h}{2} + r + \frac{2r^2}{h}. \tag{11}$$

The coordinates of the point O_3 can be set as expressed by

$$x_{O_3} = 0 \quad , \quad y_{O_3} = -R. \tag{12}$$

6. With the data assembled so far, the contour of the hatched region of Fig. 5 can be obtained simply by construction of three arcs of circumference with radius already determined.

7. The angles β and θ are, respectively, the Mach angle and the Prandtl-Meyer function corresponding to the exit section Mach number given by (Zucker, 1977)

$$\beta = arcsen\frac{1}{M_S}, \tag{13}$$

$$\theta = \sqrt{\frac{\gamma+1}{\gamma-1}}\, arctan\left[\sqrt{\frac{\gamma-1}{\gamma+1}}\,\sqrt{M_S^2-1}\right] - arctan\sqrt{M_S^2-1}\,. \tag{14}$$

So the coordinates of the point C are set as given by

$$x_C = r\,sin\,\theta \quad , \quad y_C = r\,(cos\,\theta - 1). \tag{15}$$

8. The point E is determined considering that the Mach line calculated at the point C includes the point E. So if the triangle ΔCDE is taken, the length of the line segment $\overline{CD}$ is given by

$$\overline{CD} = \frac{H}{tan\,\beta}, \tag{16}$$

and the coordinates of points D can now be set as expressed by

$$x_D = x_C + \overline{CD}\,cos\,\theta \quad , \quad y_D = y_C - \overline{CD}\,sin\,\theta\,. \tag{17}$$

Finaly, the coordinates of point E can be obtained from the coordinates of point D by

$$x_E = x_D + H\,sin\theta \quad , \quad y_E = y_D + H\,cos\theta \,. \tag{18}$$

9. Each point in the arc $\widehat{AC}$ correspond to a point in the curve $\widehat{BE}$ determined by the Mach line corresponding to the local Mach number. So the coordinates of each point can be calculated using a method analogous to that used to calculate the coordinates of point *D*. Let *n* be the desired number of points in the curve $\widehat{BE}$ – the bigger the number of points the more exact will be the determination of curve. Then the Mach number increment and the local Mach number are given by

$$\Delta M = \frac{M_{E'} - 1}{n+1}\,, \tag{19}$$

$$M_i = 1 + (i-1)\,\Delta M \,. \tag{20}$$

The point *E*’ corresponding to the local Mach number is obtained using the Eq. (7), and Eq. (13) to Eq (18) again just changing the Mach number at the exit section by the local Mach number calculated until the local Mach number reaches the nozzle nominal exit Mach number.

A computational routine written in Fortran was developed using the methodology presented to generate the nozzle geometry as much as the computational mesh for the CFD evaluation to demonstrate the flow quality along the nozzle. The geometries given in Fig. 6 were calculated for the exit section Mach number of 1.6, 1.9, 2.2 and 2.5, tubing radius of 34.94 mm, exit section height of 22.57 mm, and the specific heat ratio γ equal to 1.4, as it appears in Eqs. (7) and (14). It is important to highlight that the bigger the number of points chosen to perform the calculus, the better the resolution of the Mach number will be in the divergent section and the higher will be its flow quality (remember that the smaller the iteration step in terms of Mach number, the better the hypothesis of infinitely small corners will be satisfied).

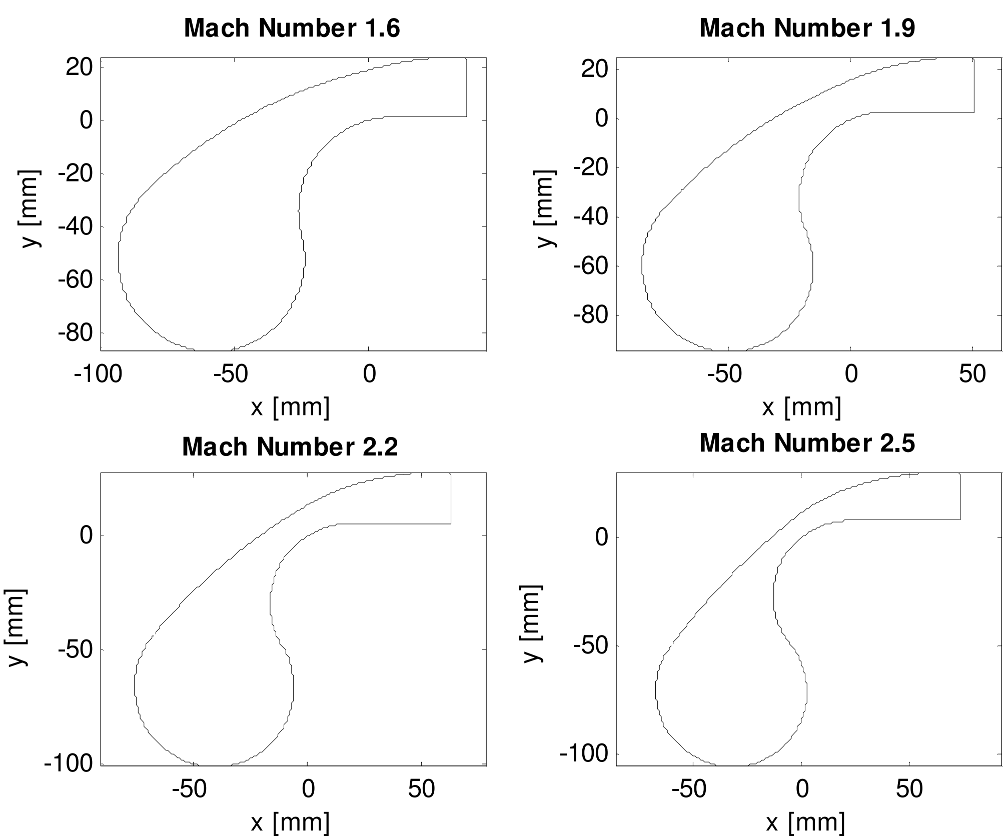

Figure 6. Calculated geometry for different Mach numbers.

4. CFD EVALUATION AND EXPERIMENTAL PARALLEL

In order to verify if the flow quality in terms of absence of compression waves in the divergent section of the injector nozzle design by the methodology presented, the 1.9-Mach-number injector design was analyzed by a CFD code based on the Euler Equations. The numerical algorithm is based in the implicit approach suggested by Beam and Warming (1978) and follows closely the main lines of the finite-difference, diagonal scheme due to Pulliam and Chaussee (1981), complemented by a nonlinear, spectral-radius-based artificial dissipation strategy due to Pulliam (1986), which presents good shock wave capturing ability. The medium is the air, considered as an isotropic and Newtonian fluid and as a thermally and calorically perfect gas.

The boundary conditions was set at the entrance to correspond to the conditions in the air compressed tubing, fixing the stagnation pressure, the stagnation temperature and the flow angularity and obtaining the velocity along the x-axis by zero-order extrapolation from the mesh interior. In the exit section the flow density and velocities components along both the axis directions are also obtained from inside the mesh by a zero-order extrapolation. The pressure at the exit is initially set to be one third of the pressure at the entrance until the flow at the exit becomes supersonic, when the pressure begin to be also extrapolated as the others variables, correctly interpreting the supersonic flow behavior.

The mesh was generated by an extension of the Fortran routine that calculated the geometry of the injector nozzle. The curves were reparametrized by its lengths so that a Gaussian-distributed stretching with variance in the order of the total length of the curve could be imposed to both the direction of the mesh. Consequently, a mesh greater resolution is guaranteed near the walls and at the throat. A prolongation of mesh after the exit section was made to study the flow after it leaves the injector nozzle. An example of a 162×33 points mesh illustration is shown in Fig. 7.

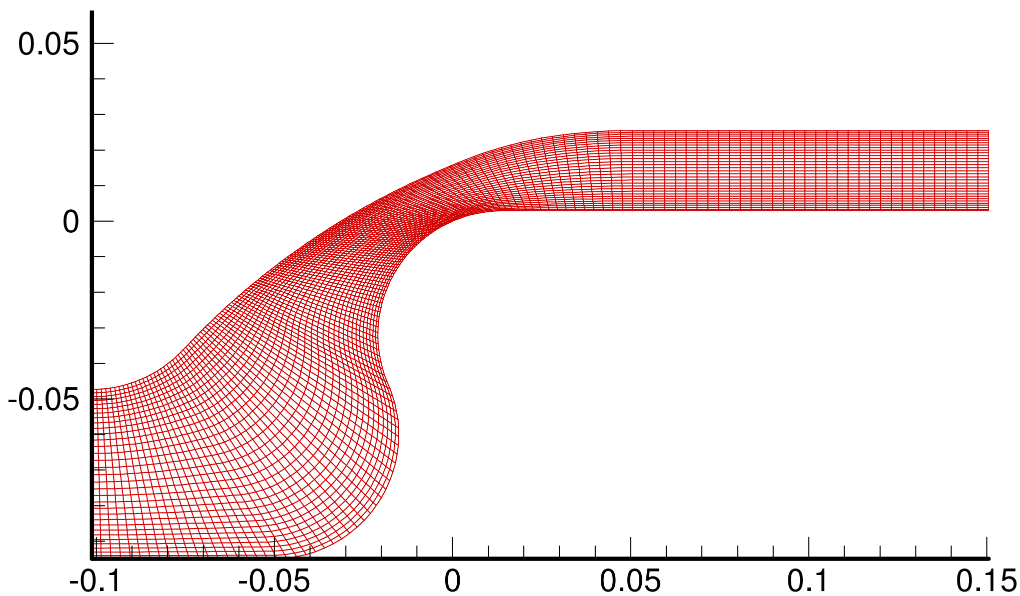

Figure 7. Illustration of a 162×33 points mesh for the 1.9-Mach-number injector nozzle.

A mesh refinement study was made to ensure that the work mesh is fine enough to guarantee the perfect convergence of the code, starting with the 81×17 points mesh as the coarsest one. The number of points in both directions of the mesh excluding the prolongation was successively doubled (the prolongation section keep the same spacing along the x-axis used at the exit section of the injector nozzle) generating a set of five different meshes. The minimum displacement near the wall varied from 1.6×10^{-3} in the coarsest mesh to 1.0×10^{-4} in the finer one.

The Mach number distribution along the upper wall plotted against the curve length to verify the convergence are shown at Fig. 8. Note that the 643×129 mesh has already obtained a good convergence, being practically coincident with the 1285×257 mesh result.

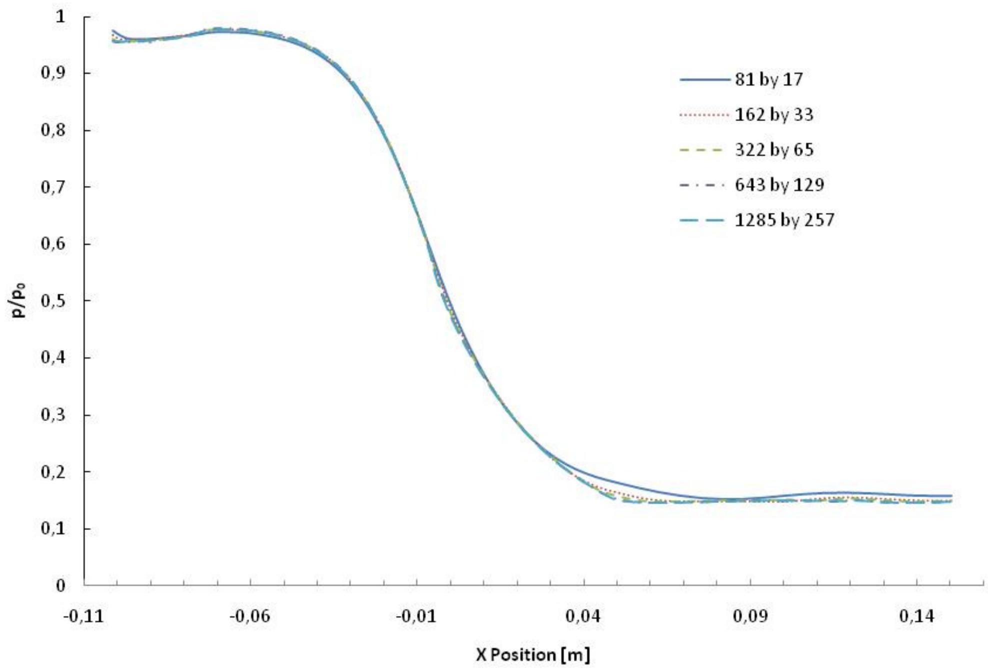

Figure 8. Convergence study for the mesh refinement: Pressure profile along the upper wall.

The result with the finer calculated mesh is shown at Fig. 9. The exit section is highlighted by a straight line. At this line the average Mach number is 1.904 ± 0.003, which represents an error of only 0.19%, testifying the good quality of such a simple project.

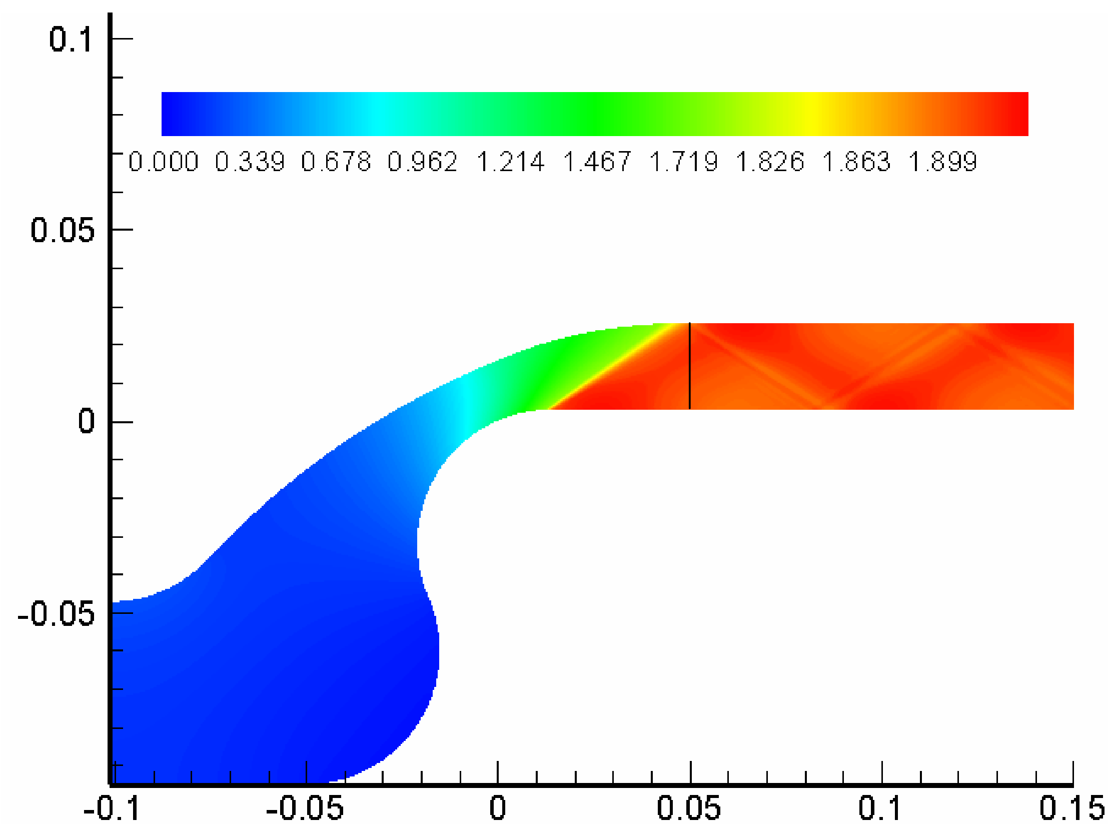

Figure 9. CFD results: Mach number distribution along the injector nozzle.

In Fig. 9 it was plotted 250 contours levels of which 130 are in the interval of Mach number 1.8 to 1.95. This choice was made to amplify the Mach number variation in the supersonic portion of the nozzle and better analyze the existing shock/expansion waves. Note that the waves observed in the prolongation of the injector nozzle do not exist in the real experimental set because there are no walls in the physical facility but the open space. Note also that the appearing of such waves do not influence the upstream flow due to the hyperbolic characteristic of the governing equations of the supersonic flow. In the finest mesh, the compression waves present in the prolongation section are so weak that the Mach number decrement is of about 0.1%.

Table 1 shows up the evolution of the calculated flow quality at the outlet section of the injector nozzle in terms of the average Mach number and its deviation as the mesh is refined.

Table 1. Outlet Flow quality evolution with the mesh refinement.

Mesh Size	Average Outlet Mach Number
81×17	1.83 ± 0.04
162×33	1.88 ± 0.02
322×65	1.898 ± 0.009
693×129	1.903 ± 0.004
1285×257	1.904 ± 0.003

The computational result can also be compared with the Mach number experimentally obtained by static pressure measurements at the wall of the injector nozzle already operational in the experimental mixing chamber of IAE in order to verify how close the results obtained are with the physical reality. This comparison can be found in Fig. 10.

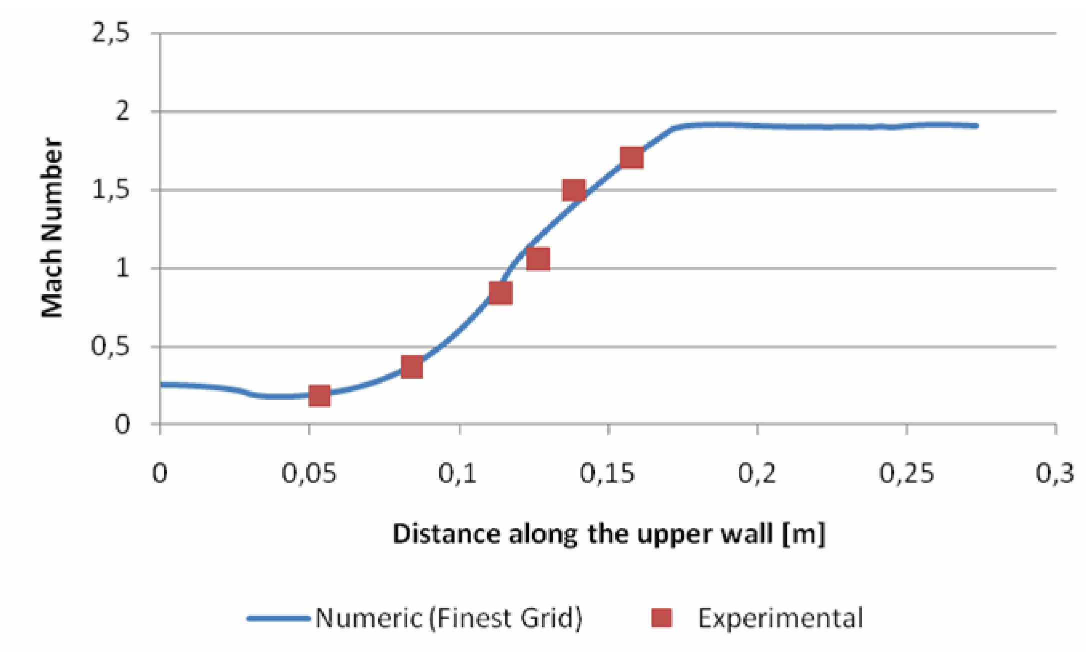

Figure 10. CFD results: Mach number distribution along the injector nozzle.

Note that the results achieved good agreement. The disparities, especially in the throat region, may be explained by the presence of the boundary layer, which can shift the effective position of the throat, an effect that cannot be previewed by the Euler model.

5. CONCLUDING REMARKS

A brief introduction to the Method of characteristics is presented and its basic concepts applied to the problem of a supersonic injector nozzle design. A design methodology is then developed in order to provide an accessible way to design an injector nozzle which provides a good assembling with the air compressed tubing and the mixing chamber geometry. The computational preliminary results obtained were beyond satisfactory even when compared with the experimental results, which proves the representativeness of the Euler model in this case.

In later works, a boundary layer correction should be added to the design methodology to correct the effect of this viscosity in the flow quality inside the injector nozzle. The new geometry could then be tested with a turbulent code to verify the accomplishment of the goal.

6. ACKNOWLEDGEMENTS

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7. REFERENCES

Anderson, B. H., 1974, "Computer Program for Calculating the Flowfield of Supersonic Ejector Nozzles," Lewis Research Center, NASA TN-D 7602.

Anderson, J. D. Jr., 1990, "Modern Compressible Flow with Historical Perspective", 2ªed., McGraw-Hill Book Company, New York, 650 p.

Beam, R. M., Warming, R. F., 1978, "An implicit factored scheme for the compressible navier-Stokes equations", AIAA Journal, v. 16, n. 4, p. 393-402.

Crown, J.C., 1948, "Supersonic Nozzle Design", NACA TN 1651.

Prandtl, L., Busemann, A., 1929, "Näherungsverfahren zur Zeichnerischen Ermittlung von Ebenen Strömungen mit Überschallgeschwindigkeit," Stodola Festschrift (Zurich), 1929, S. 499-509. – Cited by: Evvard, J. C., Marcus, L. R., "Achievement of Continuous Wall Curvature in Design of Two-Dimensional Symmetrical Supersonic Nozzles," NACA TN-2616.

Pulliam, T. H., and Chaussee, D. S., 1981, "A Diagonal Form of an Implicit Approximate-Factorization Algorithm", Comput. Phys., 39, pp. 347-363.

Pulliam, T. H., 1986, "Artificial Dissipation Models for the Euler Equations," AIAA J., 24(12), pp 1931-1940.

Silva, A. F. C., Braz, R. O., Rocha, A., Falcão Filho, J. B. P., 2010, "Descrição de Câmara de Mistura Supersônica/Subsônica e Primeiros Resultados," Proceedings of VI National Congress of Mechanical Engineering (accepted for publication).

Sivels, J. C., 1978, "A Computer Program for the Aerodynamic Design of Axisymmetric and Planar Nozzles for Supersonic and Hypersonic Wind Tunnels," Arnold Engineering Development Center, AEDC-TR-78-63.

Stack, J., 1933, "The N.A.C.A. High-Speed Wind Tunnel and Tests of Six Propeller Sections," NACA Report no. 463, pp. 399-420.

Sverdrup, 1989, "Brazilian Transonic wind Tunnel Concept Definition Study," Sverdrup Technology Inc. – Contractor Report for TTS and TTP Projects, CTA-IAE, Aeronautics and Space Institute, São José dos Campos, SP, Brazil

Zucker, R. D., 1977, "Fundamentals of Gas Dynamics", Matrix Publishers, Inc., 443 p.

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