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INWARD 2-D SCRAMJET ANALYTIC AND NUMERICAL ANALYSIS AT MACH NUMBER 7AND DIMENSIONAL DESIGN FOR EXPERIMENTAL INVESTIGATIONS

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Abstract. *An inward two-dimensional scramjet inlet has been designed with an axial injection combined with step combustor to be a hydrogen powered scramjet engine on an acceleration mission from 2 to 3 km/s at 30 km altitude. In this preliminary design, one-dimensional compressible flow can readily describe many features of compression region of the airbreathing engine. The engine was divided into several components based on key design parameters to assess the engine performance as a function of these parameters. One of the most important design aspects is the temperature at the inlet of the combustor chamber because the compression must provide high enough temperatures for supersonic combustion at the combustor chamber. On the other hand, very high temperatures could augment the heat loads to impracticable levels. In the present work, it was assumed that will meet both constraints. An inward two-dimensional scramjet inlet model to experimental investigated at the T3 Hypersonic Shock Tunnel will be presented as well as the analytical theoretical analysis and computational simulation based on the compressible flow.*

Keywords: hypersonic airbreathing propulsion, scramjet, hypersonic shock tunnel.

1. INTRODUCTION

The feasibility of a reliable hypersonic airbreathing engine (scramjet) is being pursed by several research centers around the Earth's planet. Many issues related to supersonic combustion are still matters of research and most of them may be performed in ground-based hypersonic facilities, as reflected hypersonic shock tunnels.

Reflected Hypersonic Shock Tunnel is a shock tube fitted with a convergent-divergent nozzle to produce high Mach number and high enthalpy flows in the test section close to those encountered during the flight of a space vehicle into the Earth's atmosphere at hypersonic flight speeds.

Two types of flight test techniques may be adopted: captive or autonomous. From the system design standpoint, captive flight has numerous advantages due to its simplicity whereby vehicle integration can be performed with few modifications in the booster aerodynamics and control systems. Moreover, deploying a hypersonic vehicle definitely increases the system's complexity, introducing the need of an entirely new dynamic analysis for the test vehicle, increasing the risk of failure. In this work, captive technique to a booster while accelerating horizontally from 2 to 3 km/s at 30 km altitude is considered for scramjet flight test.

Inward-turning inlet geometry was adopted for scramjet flight test (Fig. 1) since it minimizes booster dynamics effects on the inlet performance caused by slight variations of the angle of attack, such as mass capture and pressure at the combustor entrance (Rolim and Lu, 2012 a; 2012b).

Moreover, a symmetric ramp system can yield a structure with low products of inertia, which makes eases development of the control system. This type of inlet is also preferable for spin-stabilized flight. Another advantage of this type of configuration is that it uses a shorter length for compression than two-dimensional asymmetric geometries. This reduces not only the structural weight but reduces also the flow path, thereby the boundary-layer thickness and the related displaced mass capture. Finally, another aspect that is advantageous for this kind of inlet is the absence of an Edney type IV shock-shock interaction (Edney, 1968) in the cowl region which may be present in asymmetric designs and constitutes a major heat protection problem.

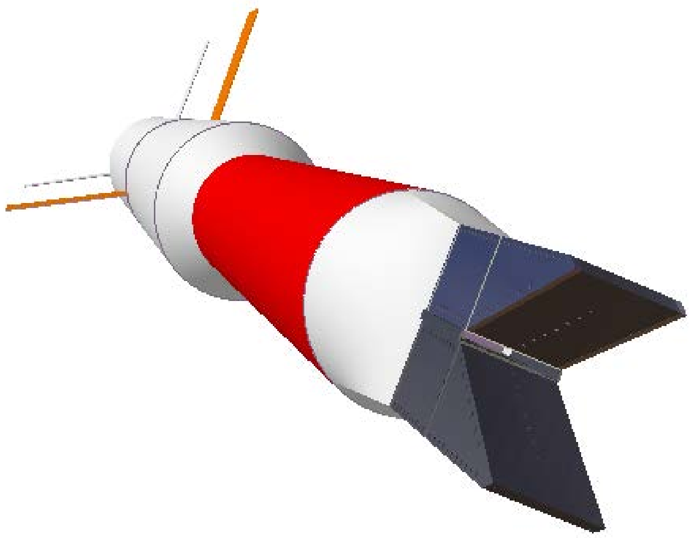

Figure 1: Captive inward-turning inlet geometry scramjet to the two-stage rocket solid engine artistic conception.

Variable geometry, which allows maintaining engine operation throughout the trajectory, is considered. This need is brought by the unique problems with accelerators where inlet off-design performance can lead to boundary-layer separation due to excessive contraction ratios or low, inadequate pressure at the combustor entrance. The combustor control must be robust enough to maintain efficient mixing and heat release during flight without choking the engine. Also, the nozzle geometry must respond to the varying conditions at the combustor and at exit, simultaneously.

The inward-turning inlet scramjet concept to flight demonstration at about 2km/s at 30km altitude is composed by symmetrical oblique convergent inward-turning inlet ramps (Fig. 2), following by the isolator, the combustor and the expansion section.

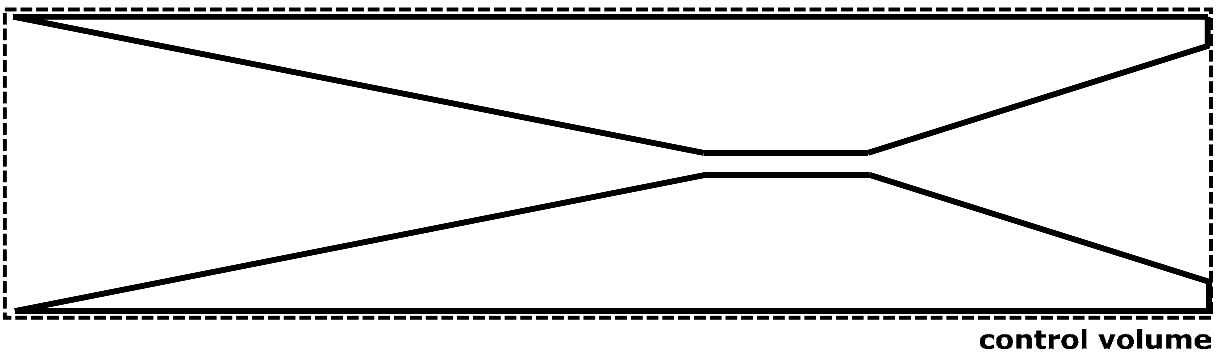

Figure 2: Schematic of the inward-turning inlet scramjet concept.

2. DESIGN OF THE INWARD-TURNING INLET MODEL

Analytic theoretical analysis, computational fluid dynamics simulation and experimental investigation are the methodologies used to design a technological demonstrator, before flight throughout Earth's atmosphere.

Two-dimensional steady state, non-viscous, no heat conduction compressible flow applied to the shock and expansion waves may be used as analytic theoretical analysis. Available commercial CFD codes are able to investigate numerically the turbulent real gas hypersonic airflow under combustion process. Reflected hypersonic shock tunnels are ground-based experimental facilities capable to duplicate the flight conditions.

A nomenclature needed not only in the analytic theoretical analysis but also in the numerical simulation and experimental investigations are presented by Heiser and Pratt (1994) and it is adapted for the inward-turning inlet scramjet, which it is divided in three main components (Fig. 3): external and internal compression section (inlet), combustion chamber (combustor) and internal and external expansion section (outlet).

The goal of the present works is to design an inward two-dimensional scramjet inlet model (Fig. 4), which is consisted by the inlet and the combustor, to experimentally investigate at the T3 Hypersonic Shock Tunnel based on the analytical theoretical analysis and numerical simulation.

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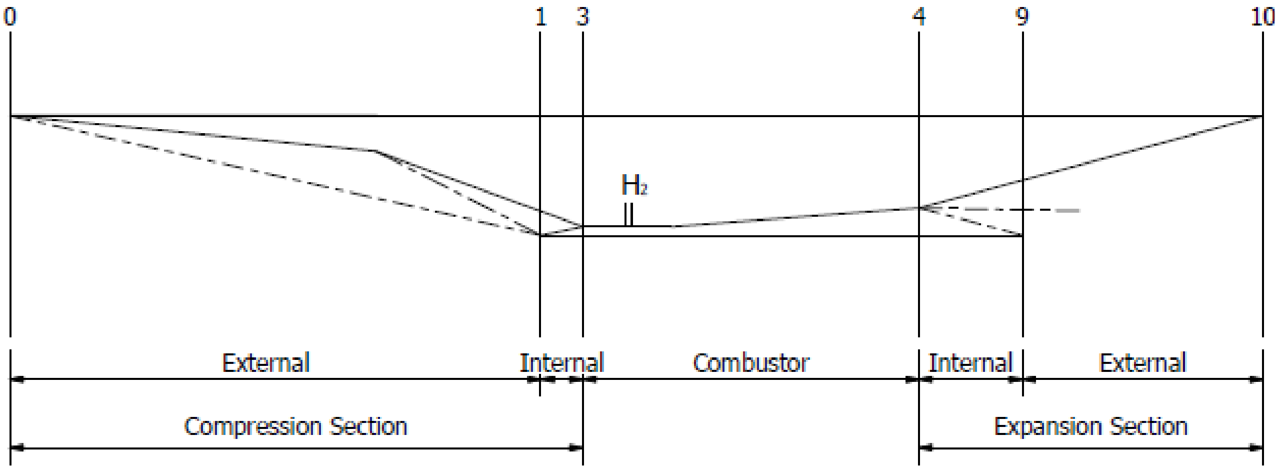

Figure 3: hypersonic vehicle with airframe-integrated scramjet engine stations and reference terminology.

The inward two-dimensional scramjet inlet model (Fig. 4) consists a two-dimensional configuration (Rolim and Lu, 2012 a; 2012b), with a constant cross-section, where the upper and lower flat surfaces, with zero angle of attack, is aligned with the freestream Mach number 6.64 hypersonic airflow. The inward two-dimensional scramjet inlet (internal surface) consists of a leading edge angle of 12^o. The cross-section height is $200mm.$. The combustor chamber is $300mm.$ long with constant area. The constant area combustion chamber is $11.4mm.$ high (to accommodate the airflow captured by the inward two-dimensional scramjet inlet frontal area).

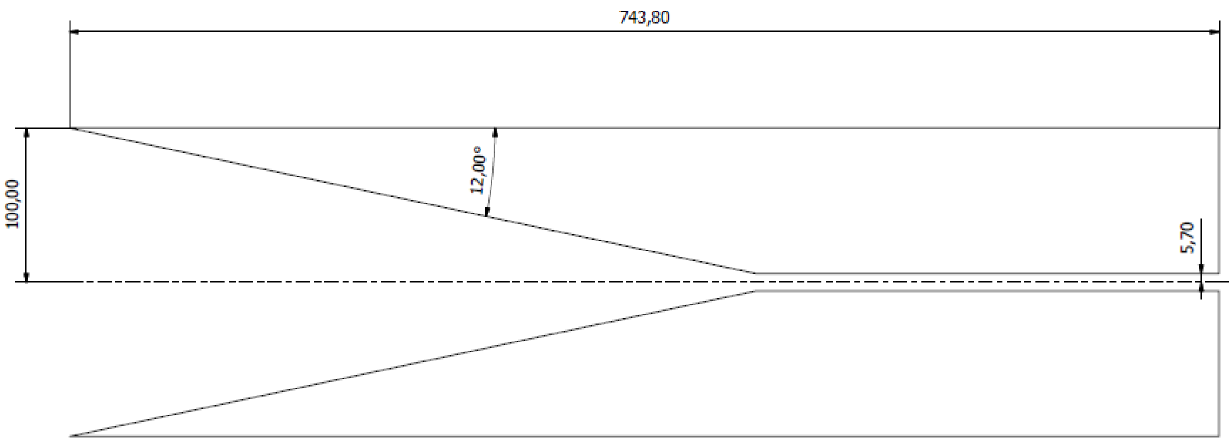

Figure 4: Schematic of the inward-turning inlet scramjet model to experimental investigation.

3. ANALICTIC THEORETICAL ANALYSIS

The baseline geometry (Fig. 5) is taken as a 12^o leading-edge ramp with 100-mm. high at the entrance and 50.8mm. span. The combustor is 5.7mm. high. The design velocity is 2 km/s (Mach number 6.64) at an geometric altitude of 30 km, where the static pressure, static temperature, static density and sound velocity are given by $p = 1197(Pa)$, $T = 226.5(K)$, $\rho = 0.01841(kg/m^3)$, $a = 301{,}7(m/s)$, respectively (U.S. standard Atmosphere, 1976).

The shock wave position and thermodynamics flow properties are evaluated considering a perfect gas air flow assumption $\gamma = 1.4$ $c_p = 1004.5\,[\mathrm{J}/kg - K]$.

In the analytic theoretical analysis, the subscripts in and out are used to identify the upstream (inlet) and the downstream (outlet) conditions, respectively, of the each station (Fig. 5) of the inward-turning inlet scramjet engine baseline.

For calorically and/or thermally perfect gas ($p = \rho RT$, $\gamma = \text{constant}$) the oblique shock relationships (Fig. 6) can be easily obtained as closed form of the thermodynamics properties (static pressure, static density and static temperature) ratios and Mach number across the oblique shock given by (Anderson, 2003):

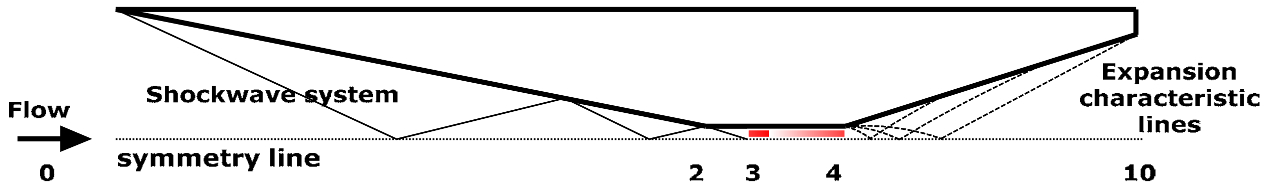

Figure 5: inward-turning inlet scramjet baseline and reference stations.

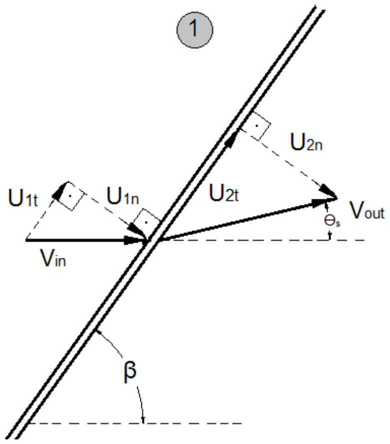

Figure 6: Leading edge incident Oblique Shock wave geometry.

$$\frac{p_{out}}{p_{in}} = 1 + \frac{2\gamma}{(\gamma+1)}\left[(M_{in}\ sen\beta)^2 - 1\right] \tag{1}$$

$$\frac{\rho_{out}}{\rho_{in}} = \frac{(\gamma+1)(M_{in}\ sen\beta)^2}{\left[(\gamma-1)(M_{in}\ sen\beta)^2 + 2\right]} \tag{2}$$

$$\frac{T_{out}}{T_{in}} = \frac{p_{out}}{p_{in}}\ \frac{\rho_{in}}{\rho_{out}} = 1 + \frac{2\gamma}{(\gamma+1)}\left[(M_{in}\ sen\beta)^2 - 1\right]\ \frac{\left[(\gamma-1)(M_{in}\ sen\beta)^2 + 2\right]}{(\gamma+1)(M_{in}\ sen\beta)^2} \tag{3}$$

$$M_{out} = \frac{\sqrt{\dfrac{(M_{in}\ sen\beta)^2 + \dfrac{2}{(\gamma-1)}}{\dfrac{2\gamma}{(\gamma-1)}(M_{in}\ sen\beta)^2 - 1}}}{sen(\beta - \theta_s)} \tag{4}$$

where: θ_s , β are the deflection and shock wave angles, respectively. Additionally, the shock wave angle β with respect to the local flow direction θ_s may be obtained iteratively with the relationship given by:

$$tg\theta_s = 2(\cot\beta)\left[\frac{(M_{in}\ sen\beta)^2 - 1}{M_{in}^2(\gamma + \cos 2\beta) + 2}\right] \tag{5}$$

Note, the flow across the oblique shock wave promote an increase of pressure, density, temperature, and a decrease of Mach number, however the flow remains supersonic/hypersonic and parallel to the flat surface of the compression ramp section (Fig. 5) of the inward-turning inlet scramjet engine baseline.

Also, the leading edge incident Oblique Shock wave methodology may be used for reflected shock wave (Fig. 7).

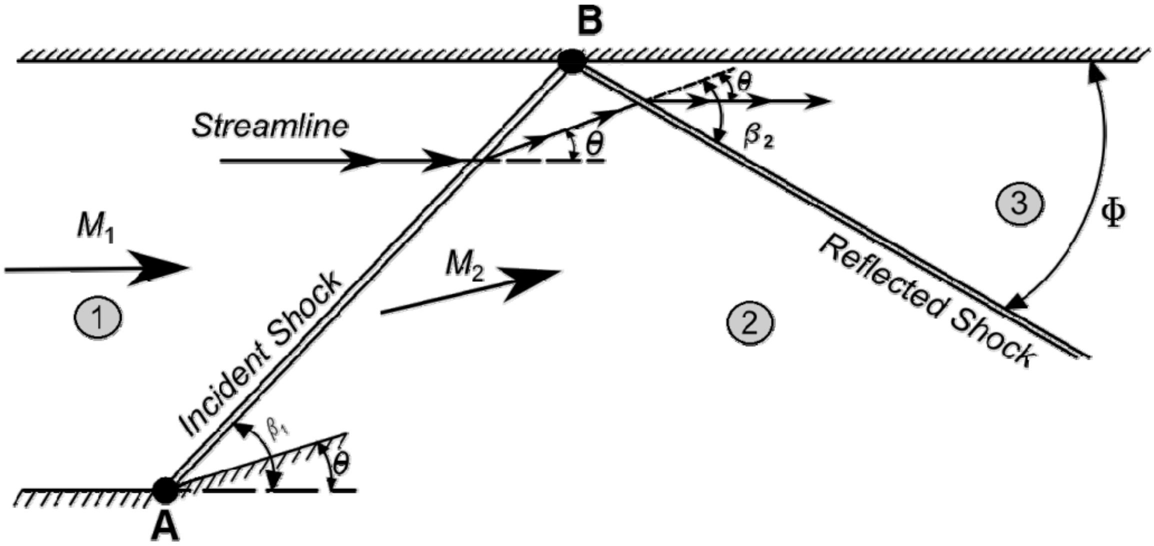

Figure 7: Reflected Oblique Shock wave geometry.

4. ANALICTIC THEORETICAL ANALYSIS RESULTS

The thermodynamics properties at the inward-turning inlet scramjet engine baseline (Fig. 5) may be determined based on the two-dimensional compressible flow (oblique shock wave relationships) considering the simplest case, i. e., no viscous flow, calorically perfect air gas ($\gamma = 1.4$) and scramjet engine with power off (Table 1). It was applied the following restriction: there is incident shock waves generated at the 12° leading-edge deflection angle due the alignment of the leading-edge with the undisturbed freestream air flow; the last reflected shock wave generated at symmetric center line hits the entrance of the combustor chamber, which the flow at the combustor chamber will start with the constant static pressure, constant static temperature, constant static density and constant supersonic airflow.

The analytical theoretical analysis related to the inward-turning inlet scramjet engine flying at Mach number 6.64 at 30km geometric altitude (U.S. standard Atmosphere, 1976) where the static pressure, static temperature, static density and sound velocity are given by $p = 1197(Pa)$, $T = 226.5(K)$, $\rho = 0.01841\left(kg/m^3\right)$, $a = 301{,}7\,(m/s)$, respectively.

Considering a hypersonic vehicle with 12° leading-edge angle flying at a Mach number ($M = 6.64$) throughout Earth's Atmosphere ($\gamma = 1.4$) at 30km geometric altitude, the attached oblique incident shock wave angle as well as the oblique reflected shock wave angles may be evaluated by the $\theta - \beta - M$ relation (Eq. 5). Following, the thermodynamics air properties at each event may be evaluated by the thermodynamics properties ratio (Eqs. 1-3). The Mach number after each incident and reflected shock waves may be evaluated by Eq. 4.

Table 1: Thermodynamic properties at the inward-turning inlet scramjet engine baseline, power off, inviscid, $\gamma = 1.4$.

		station 0	station 1	1st reflection	2nd reflection	station 3
M_{in}		6.64	6.64	4.717	3.6226	2.8754
θ_{in}	o		12	12	12	12
β_{out}	o		18.7682	9.9375	25.6456	18.2029
M_{out}			4.717	3.6226	2.8754	2.3078
T_{out}	K	226.5	408	613.5	838	1076.8
p_{out}	Pa	1197	6174	21340	57646	131111
ρ_{out}	kg/m^3	0.01841	0.05271	0.12117	0.23963	0.42417
a_{out}	m/s	301.7	404.9	496.5	580.3	657.8
u_{out}	m/s	2003.1	1909.9	1798.6	1668.5	1518

Figure 8 shows the incident shock waves generated at the 12° leading-edge deflection angle due the alignment of the leading-edge with the undisturbed freestream air flow; the reflections shock waves and finally the last reflected shock wave generated at symmetric center line hits the entrance of the combustor chamber.

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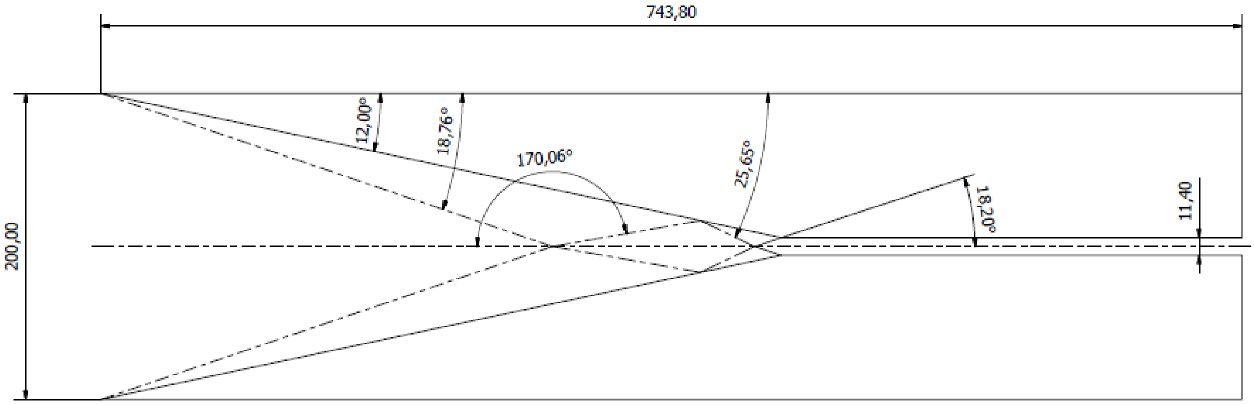


Figure 8: Incident and reflected shock wave angles at the inward-turning inlet scramjet model.

5. NUMERICAL THEORETICAL ANALYSIS

The commercial "Fluent" software, which solves the mass, momentum and energy conservation equations, considering reacting flow and includes routines (solvers) that accurately simulate the behavior of flow, single phase and multiphase, Newtonian and non-Newtonian flow from subsonic to hypersonic speed, is able to perform a numerical theoretical simulation applied to the inward-turning inlet scramjet engine model.

Two-dimensional, steady state, non-viscous, no heat conduction compressible flow using implicit second order upwind spatial discretization are used to nose-to-tail numerical modeling of inward-turning inlet scramjet engine (Fig. 8) geometry, considering a hypersonic vehicle (inward-turning inlet scramjet engine) with 12° leading-edge angle flying at a Mach number ($M = 6.64$) through Earth's Atmosphere ($\gamma = 1.4$) at 30km geometric altitude (U.S. standard Atmosphere, 1976.

Also, for the present numerical test case, with power-off scramjet engine, the flow from the external and internal compression section are deflected to the combustor entrance (Fig. 8) at supersonic speed (at constant pressure, constant density, constant temperature and constant Mach number) and remains constant at the exit of the combustor.

6. NUMERICAL THEORETICAL ANALYSIS RESULTS

Note that both attached incident oblique shock waves, at the 12° leading-edge deflection angle focus at the center line of the inward-turning inlet scramjet engine, promoting a shock train before the last reflection shock wave hits the entrance of the combustor chamber. Also, the Mach number at combustor chamber is higher than Mach number 2.

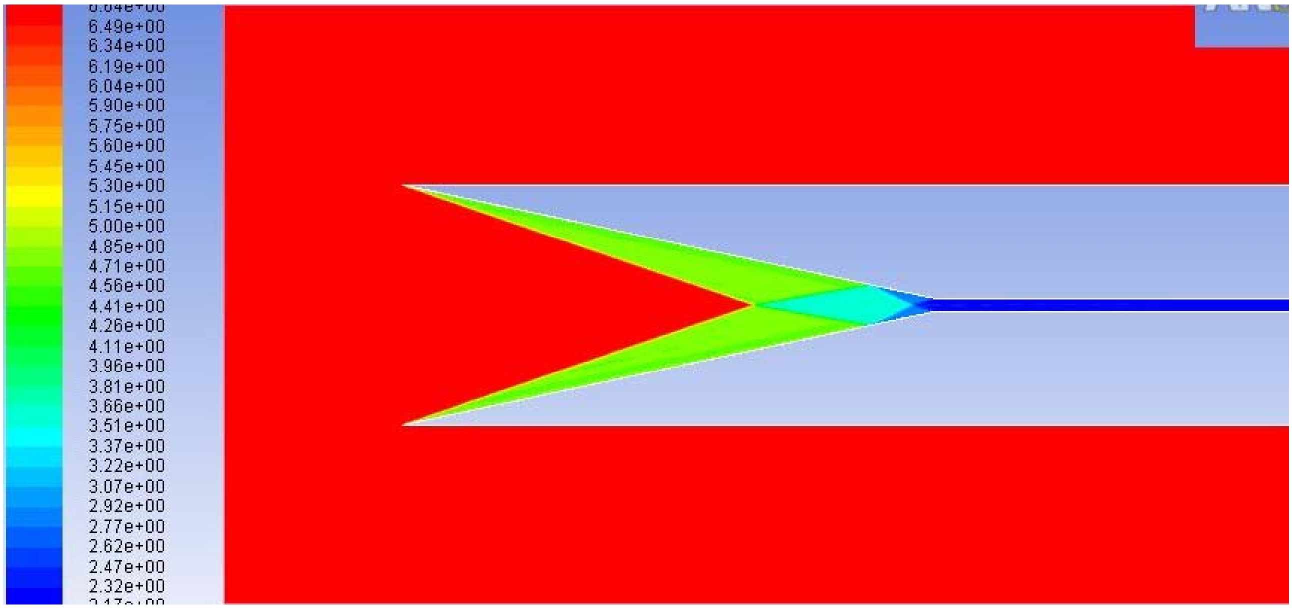


Figure 9: Mach number countor plot.

ISSN 2176-5480

The inward-turning inlet scramjet engine with 12° leading-edge deflection angle (Fig. 8) is capable to genetarate a static temperature about 1000K at the combustor (Fig. 10) higher than the ignition temperature of 845K aproximately for Hydrogen (Kuchta, 1985), with supersonic Mach number higher than 2 at the combustor (Fig. 9).

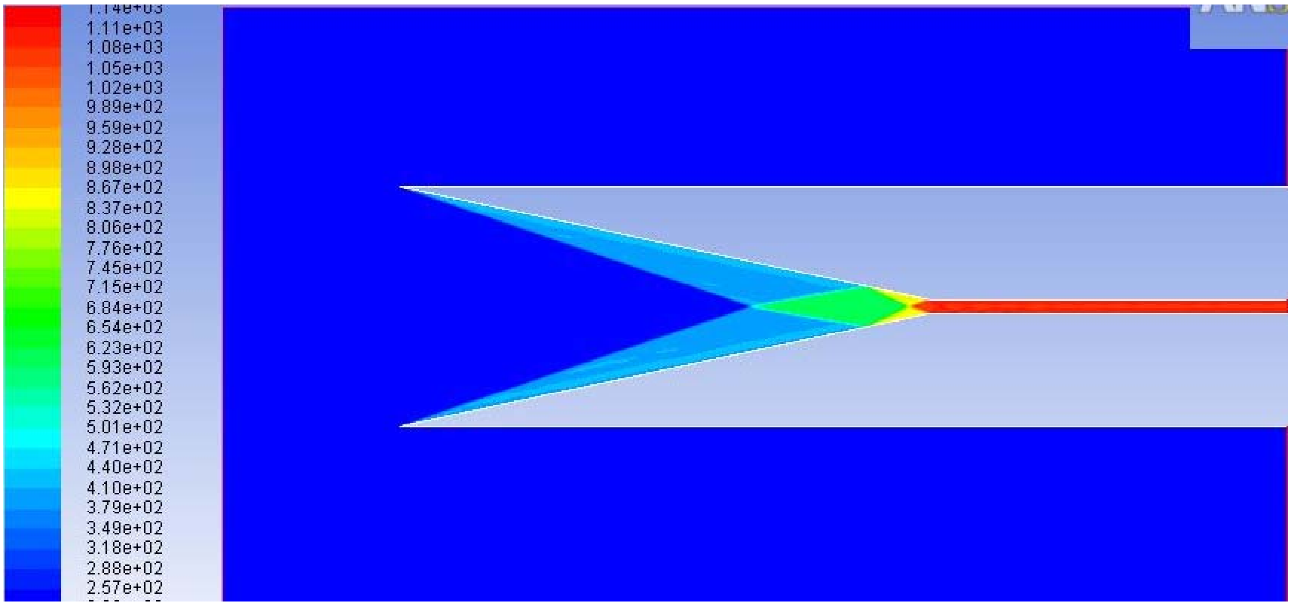
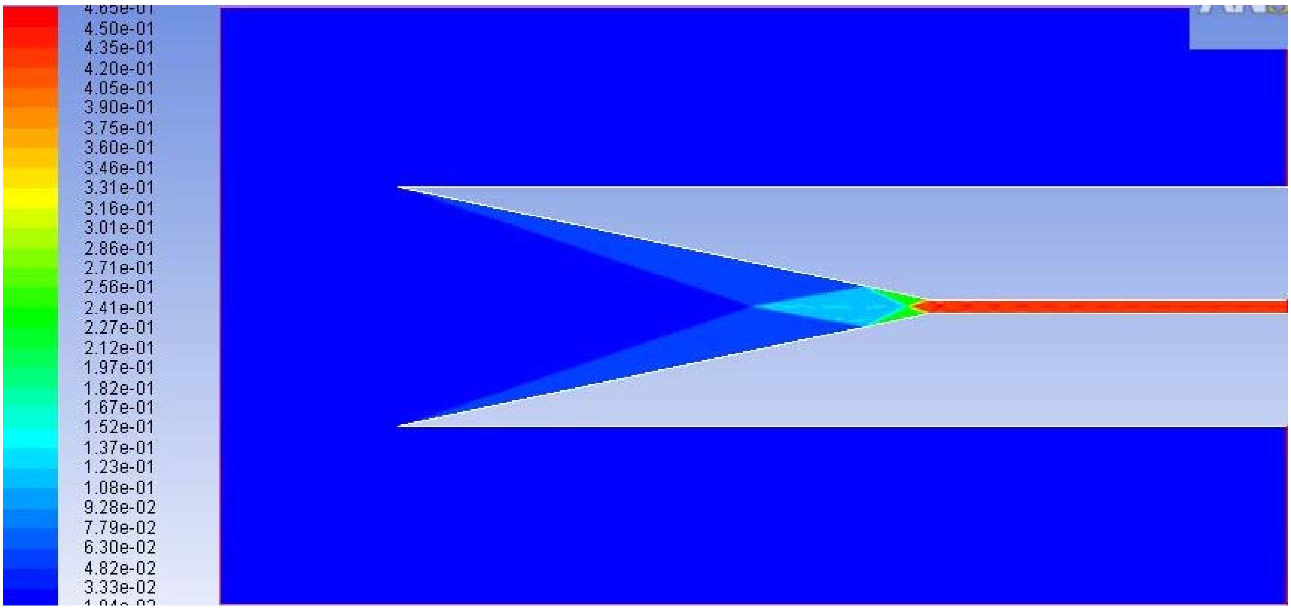
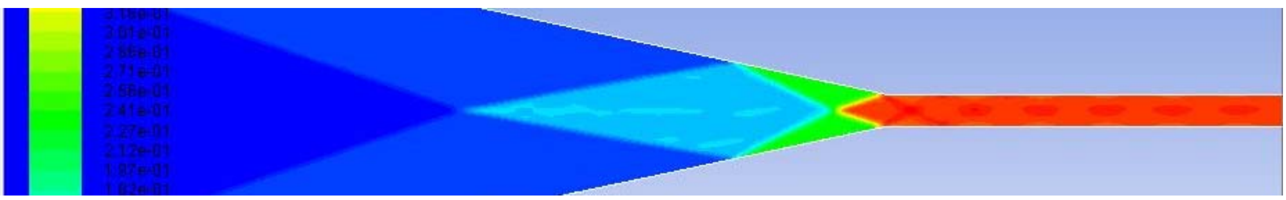


Figure 10: Countors plot of static temperature.

The contours of the static density gradients across the lower surface of the inward-turning inlet scramjet engine (Figs. 11 and 12) shows there is no shock train at the combustor chamber, which is the characteristics of the optimum combustor chamber design, with the flow mass rate at the entrance of the combustor chamber is the same at the entrance of the inward-turning inlet scramjet engine. Also, the static density gradients across the lower surface of the inward-turning inlet scramjet engine may be compare with the schlieren pictures taken from the experimental investigation.



Figire 11: Countors plot of static density.



Figire 11: Detailed of the countors plot of static density at the combustor chamber.

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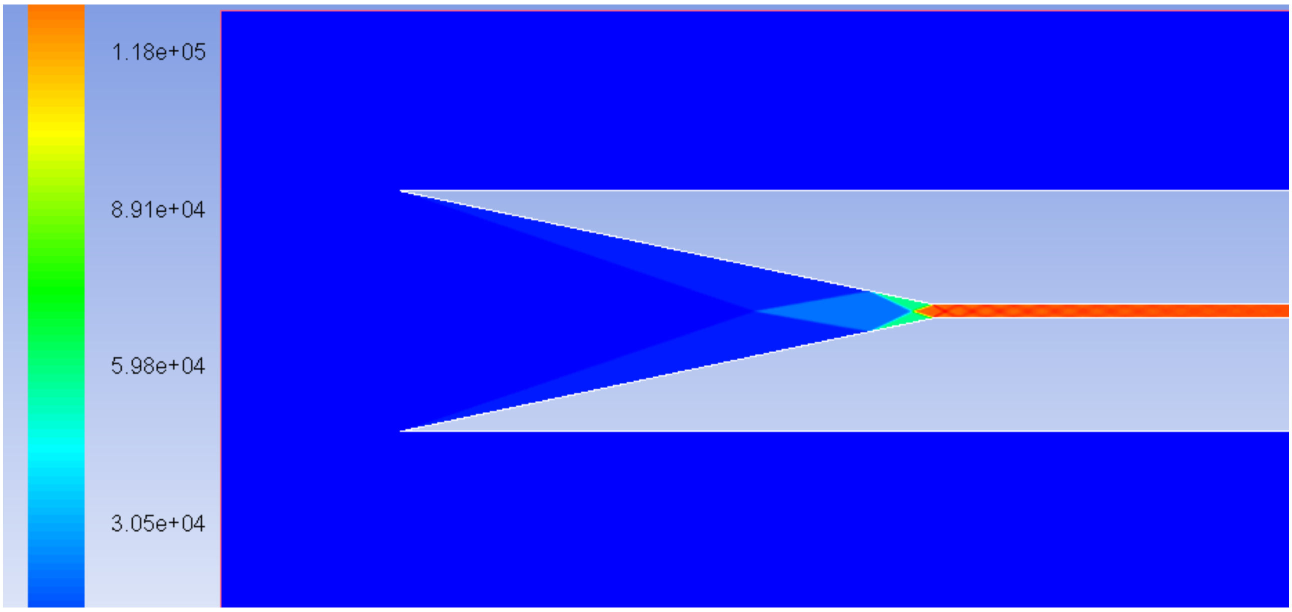
The maximun static pressure occurs (Fig. 12) at the combustor and may be used as a guide to specify the structural material of the inward-turning inlet scramjet engine.



Figire 12: Ccountors plot of static pressure.

Figure 13 shows the pressure numerical simulation results are comparable to the pressure analytic theoretical analysis (Table 1) applied to the inward-turning inlet scramjet engine. Also, the pressure distribution from the numerical simulation shows the shock train at the combustor section.

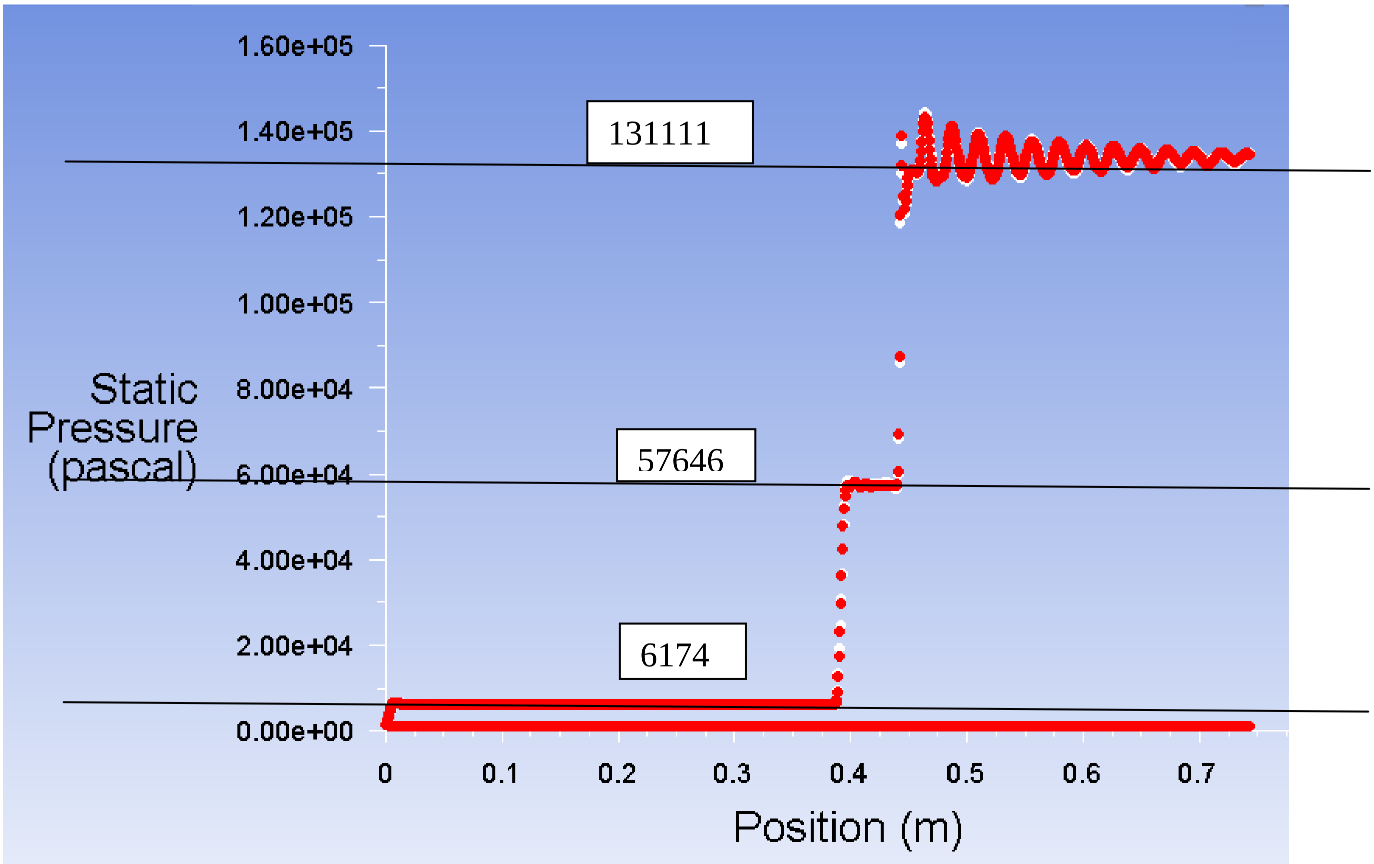


Figure 13: Numerical and analytical pressure distribution of the lower surface of the inward-turning inlet scramjet engine.

ISSN 2176-5480

7. DESIGN OF THE INWARD-TURNING INLET MODEL FOR EXPERIMENTAL INVESTIGATION

Hypersonic Shock Tunnel is a shock tube fitted with a convergent-divergent nozzle to produce high Mach number and high enthalpy flows in the test section close to those encountered during the flight of a space vehicle into the Earth's atmosphere at hypersonic flight speeds.

The T3 0.60-m. nozzle exit diameter Reflected Hypersonic Shock Tunnel, which is a new Hypersonic High Enthalpy Real Gas Pulsed Reflected Shock Tunnel (Fig. 14) funded by São Paulo Research Foundation (FAPESP, process nº 2004/00525-7), was designed by Toro et al (2005;2007) at the Prof. Henry T. Nagamatsu Laboratory of Aerothermodynamics and Hypersonics, primarily as Research and Development (R& D) facility for basic investigations in supersonic combustion applied to high-speed advanced airbreathing propulsion.

Figure 14: T3 0.60-m. nozzle exit diameter Hypersonic Shock Tunnel.

A 743.8-mm. long stainless steel of the inward-turning inlet scramjet engine model (Fig. 15) instrumented with twenty-two piezoelectric pressure transducers on the compression surface and the combustion chamber is being manufactured and will be experimentally investigated on the equilibrium interface mode operation at the T3 Hypersonic Shock Tunnel at freestream Mach number from 7 to 8.

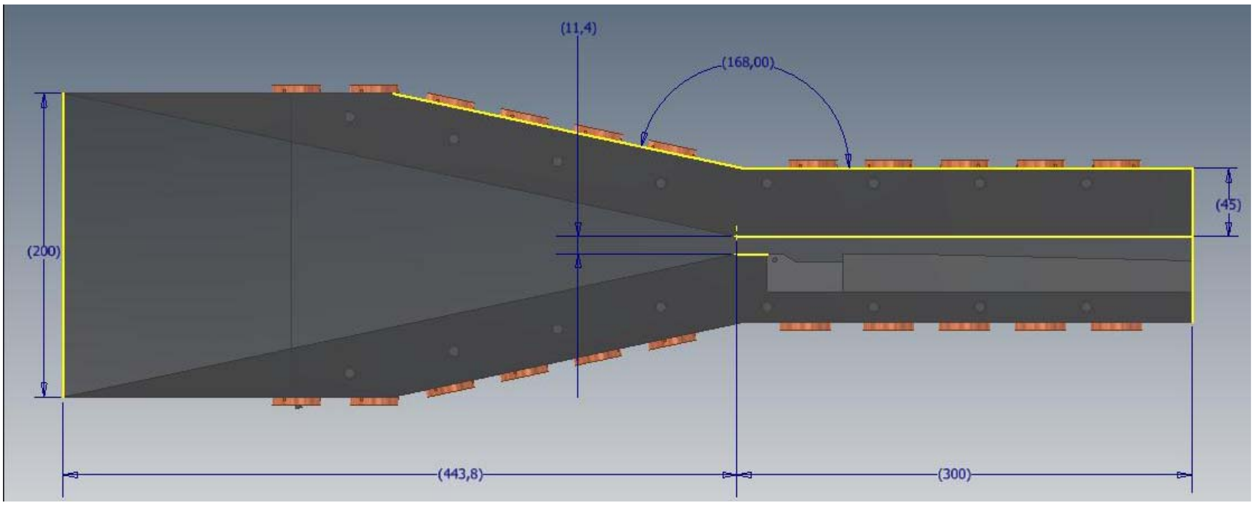


Figure 15: Inward-turning inlet scramjet engine model.

ISSN 2176-5480

The subsonic-supersonic (hypersonic) isentropic no viscous flow through variable-area ducts (convergent-divergent nozzle) may be used to established the Mach numbers 7 and 8 hypersonic airflow, in the test section of the T3 Hypersonic Shock Tunnel.

The inward-turning inlet scramjet engine model will be installed on the heavy steel stand (Fig. 16), specially designed to hold firmly the model, in the test section of the T3 Hypersonic Shock Tunnel (Fig. 17). Two symmetrically windows were designed on the sides of the horizontal derivation of the tank dump, and they will be used as flow visualization windows.

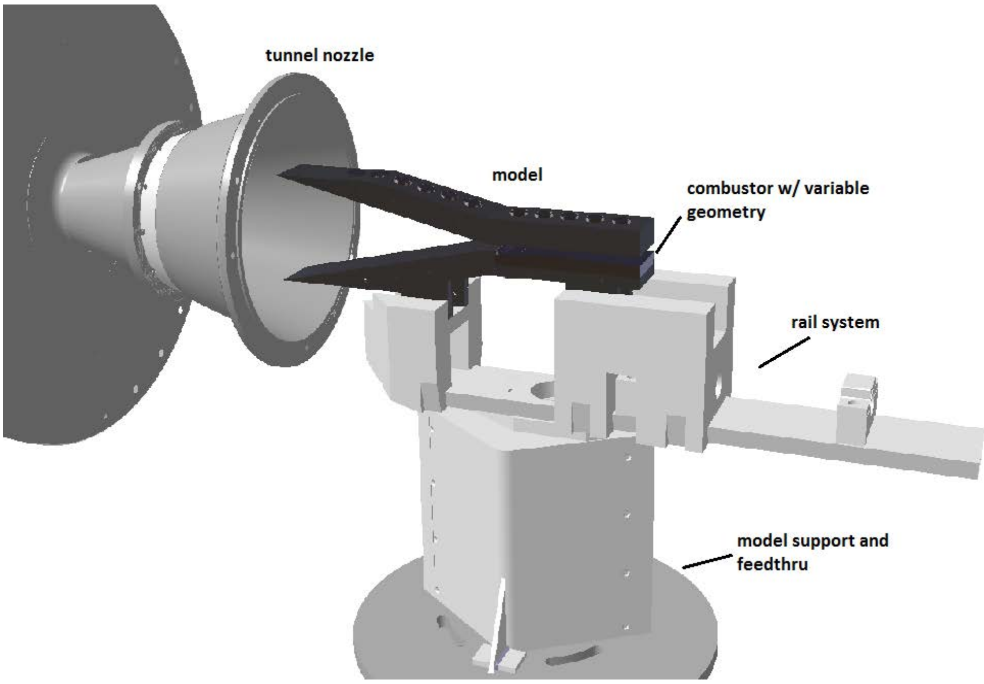


Figure 16: Inward-turning inlet scramjet engine model installed on the heavy steel stand.

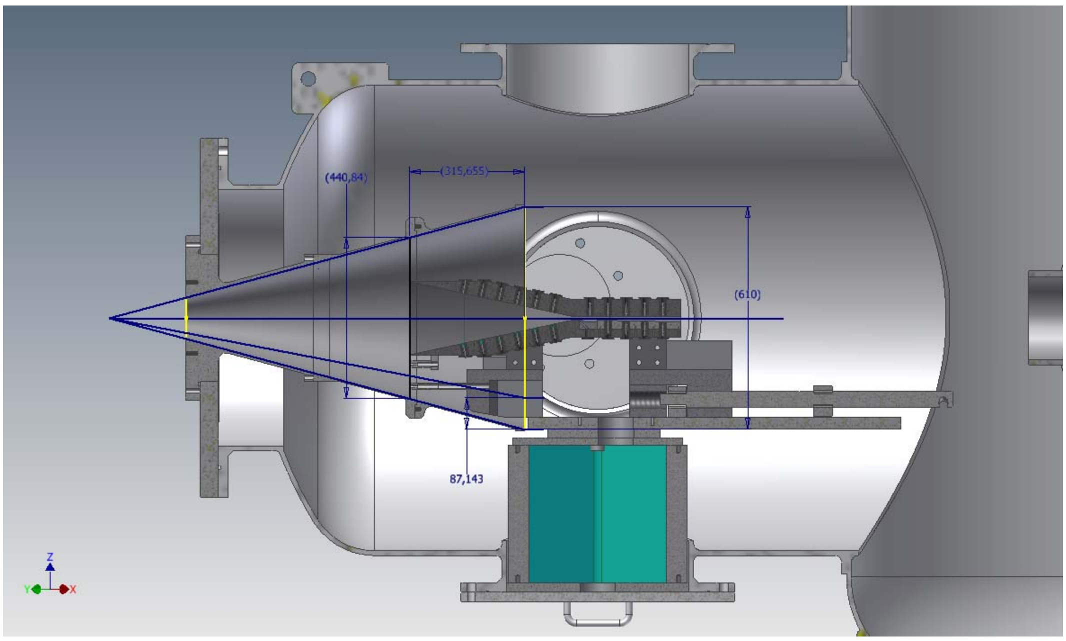


Figure 17: Inward-turning inlet scramjet engine model installed at the T3 Hypersonic Shock Tunnel.

An ultra high-speed camera, manufactured by CORDIN, model 550, coupled with mirror-based schlieren 'Z' configuration with the schlieren light beam path and placement of the Cordin camera has been used for dynamic flow visualization. The schlieren system is composed of a pulsed xenon flash lamp, an optical slit and focusing lens, two parabolic and three flat mirrors, the knife edge which provides the necessary light cut-off to the Cordin 550 ultra-high speed camera.

The Cordin 550 camera acquires 32 frames with a maximum resolution of 1000 x 1000 pixels at up to 2 million frames per second (fps) in full color. Such frame rates are achieved by a multi-faceted mirror spinning at high speeds, surrounded by 32 CCD elements which acquire images as the mirror rotates. Mirror rotation is driven by a turbine wheel supplied with high pressure N_2 for frame rates up to 500,000 fps, and pressurized He for the highest speeds. Even though extremely high speeds can be achieved, the present work demanded more modest 50,000 to 100,000 fps.

A multichannel time-delay generator is used to synchronize all the equipment used in the experiment (data acquisition system, and schlieren system) within the useful shock-tunnel time. The unit was triggered by a Kistler piezoelectric pressure transducer (model 701 A) located immediately upstream of the nozzle entrance. Also, this transducer supplies the reservoir pressure of the nozzle. Two other Kistler 701A transducers, located 0.314-m. apart at the end of the tunnel-driven section, mounted flush with the shock tube (heavy section) inner wall, were used to time the incident shock wave.

8. CONCLUSION

Analytic theoretical analysis and computational fluid dynamics simulation are developed to evaluate the thermodynamic properties of an inward two-dimensional scramjet inlet with a leading edge angle of 12^{o}, constant cross-section, zero angle of attack, with $300mm.$ long constant area and $11.4mm.$ high combustor chamber flying through Earth's Atmosphere at 30km altitude aligned with the freestream Mach number 6.64 hypersonic speed.

Analytic theoretical and numerical computational fluid dynamics results show the inward-turning inlet scramjet engine with 12° leading-edge deflection angle is capable to genetarate a static temperature about 1000K at the combustor higher than the ignition temperature of 845K aproximately for Hydrogen, with supersonic Mach number higher than 2 at the combustor.

An inward two-dimensional scramjet inlet model to experimental investigated at the T3 Hypersonic Shock Tunnel is proposed based on the analytic theoretical analysis and numerical computational fluid dynamics results.

The T3 0.60-m. nozzle exit diameter Reflected Hypersonic Shock Tunnel, is new Hypersonic High Enthalpy Real Gas Pulsed Reflected Shock Tunnel, funded by São Paulo Research Foundation (FAPESP), and it was designed at the Prof. Henry T. Nagamatsu Laboratory of Aerothermodynamics and Hypersonics, at Institute for Advanced Studies (IEAv), primarily, as Research and Development facility for basic investigations in supersonic combustion applied to high-speed advanced airbreathing propulsion.

A 743.8-mm. long stainless steel of the inward-turning inlet scramjet engine model instrumented with twenty-two piezoelectric pressure transducers on the compression surface and the combustion chamber is being manufactured and will be experimentally investigated on the equilibrium interface mode operation at the T3 Hypersonic Shock Tunnel at freestream Mach number from 7 to 8.

9. ACKNOWLEDGEMENTS

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