



BRAZILIAN 14-X HYPERSONIC SCRAMJET AEROSPACE VEHICLE EXPERIMENTAL INVESTIGATION AT MACH NUMBER 10

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Abstract. *Several research centers are developing new aerospace vehicles using advanced hypersonic airbreathing propulsion. Supersonic combustion ramjet (scramjet) presents the greatest potential among the technologies being studied. Scramjets carry out only the fuel, usually Hydrogen, using atmospheric air as oxidant acquiring most of the kinetic energy required to reach Earth's orbit during its atmospheric flight. The net thrust produced by the scramjet is the difference between the thrust generated by the expansion of exhaust gases from the rear and the total drag of the engine. Hypersonic shock tunnels are devices capable of generating in the laboratory environment encountered in hypersonic flight, with respect to velocity, chemical composition, temperature and enthalpy of the flow. A 265.1-mm. long, 80-mm. wide and 35-mm. high combustion chamber model instrumented, with piezoelectric pressure transducers on the compression surface and at the combustion chamber, was coupled at the lower surface of the truncated 14-X Hypersonic waverider Aerospace Vehicle. The scramjet model is being experimentally investigated on the equilibrium interface mode operation at the T3 Hypersonic Shock Tunnel at freestream Mach number from 7 to 8. Static pressure measurements on the compression surface and at the combustion chamber as well as the schlieren photographs taken from the 14.5° deflection compression ramp provide experimental data to design the VHA 14-X. Also, schlieren photographs of the injection of on-board Helium gas into the atmospheric supersonic airflow will be presented.*

Keywords: VHA 14-X, hypersonic airbreathing propulsion, scramjet, hypersonic shock tunnel.

1. THE BRAZILIAN 14-X HYPERSONIC AEROSPACE VEHICLE

In 2007 research in hypersonic airbreathing vehicle with airframe-integrated scramjet engine began at the Prof. Henry T. Nagamatsu Laboratory of Aerothermodynamics and Hypersonics, at the Institute for Advanced Studies (IEAv), Brazil (Ricco et al., 2011; Toro et al. 2012). Initially, Brazilian researchers proposed to design, to develop, to manufacture and to demonstrate, in free flight, the technology demonstrator named as 14-X Hypersonic Aerospace Vehicle (Fig. 1), VHA 14-X, using: i) "waverider" technology to provide lift to the aerospace vehicle, and ii) "scramjet" technology to provide hypersonic airbreathing propulsion system based on supersonic combustion.

Today, the 14-X project is a strategic project of the Department of Aerospace Science and Technology (DCTA), where the goal is to demonstrate, in free flight at 30km altitude at Mach number 7 to 10, several technology demonstrators using "scramjet" as an hypersonic airbreathing propulsion system based on supersonic combustion (Toro et al. 2013).

To accelerate the scramjet integrated vehicle to flight to the test conditions (30km altitude at desired Mach number) a Brazilian rocket engines based on solid propulsion, in ballistic trajectory, will be used as a low-cost Brazilian solution.

D. Romanelli Pinto, C. E. O. Silva, S. B. Cruzelles, V. A. B. Galvão, T. V. C. Marcos, G. P. Camillo, and P. G. P. Toro
Brazilian 14-X Hypersonic Scramjet Aerospace Vehicle Experimental Investigation at Mach Number 10

Such approach may provide an affordable path for maturing of the Brazilian hypersonic airbreathing components and systems in flight.

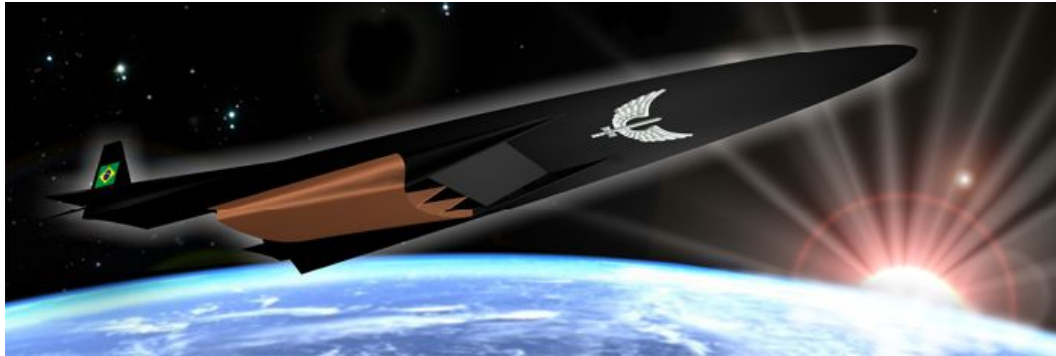


Figure 1: 14-X Hypersonic Aerospace Vehicle.

In the previous works (Rolim 2009, Rolim et al. 2009, 2011) a 718.28-mm. long stainless steel “waverider” model instrumented with seven piezoelectric pressure transducers on the compression surface, had been experimental investigated on the equilibrium interface mode operation at the IEAv 0.60-m. nozzle exit diameter Hypersonic Shock Tunnel T3 at freestream $Re_\infty = 2.25 \text{ to } 8.76 \cdot 10^6 \left(m^{-1} \right)$ and $Kn_\infty = 0.06 \text{ to } 0.19$ at Mach number from 8.9 to 10 with stagnation pressures between 2176-2938 psi and temperatures at the range of 1558-2150 K. The 718.28-mm long waverider model has 5.5° leading edge angle, 14.5° deflection compression ramp and 15° expansion ramp. The schlieren photographs taken at the 5.5° pure waverider leading-edge and at the 14.5° deflection compression ramp allowed to locate the position of the combustor chamber (Fig. 2) of the VHA 14-X (Costa, 2011; Costa et al., 2012; Costa et al., 2013).

2. EXPERIMENTAL INVESTIGATION AT THE T3 HYPERSONIC SHOCK TUNNEL

2.1 T3 0.60-m. nozzle exit diameter Hypersonic Shock Tunnel

The T3 0.60-m. nozzle exit diameter Reflected Hypersonic Shock Tunnel, is new Hypersonic High Enthalpy Real Gas Pulsed Reflected Shock Tunnel (Fig. 2), funded by São Paulo Research Foundation (FAPESP, process n° 2004/00525-7), 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 (R& D) facility for basic investigations in supersonic combustion applied to high-speed advanced airbreathing propulsion.



Figure 2: T3 0.60-m. nozzle exit diameter Hypersonic Shock Tunnel.

The T3 Hypersonic Shock Tunnel (Fig. 2) was designed (Toro et al, 2005; 2007) to be operated in the 6-25 flow Mach number range, generating reservoir enthalpies in excess of 10 MJ/Kg and reservoir pressures up to 25 MPa when

operated in the equilibrium interface mode, with estimated useful test time of 2 ms to 10 ms. These conditions are estimated based on the driver-to-driven area ratio, $\left(\frac{\phi_{driver}}{\phi_{driven}}\right)^2 = 2.25$, on the driven length equal 100 times the driven internal diameter, $\phi_{driven} = 127\text{mm}$, and on the operating driver pressure 5000 psi.

First, the T3 Hypersonic Shock Tunnel (Fig. 2) was used to conduct the Experimental Investigation of Hypersonic Flow Characterization (Romanelli Pinto et al. 2011a, 2011b), which test condition may be used in the present experimental investigation on the truncated 14-X scramjet.

In the hypersonic flow characterization investigation, of high enthalpy runs, helium is used as the driver gas and the tunnel operates in the equilibrium interface condition to produce a useful test time of roughly 1 ms. The test section airflow Mach number is 8 to 10 in the high enthalpy tests.

To verify the uniformity of the hypersonic flow in the test section of the T3 Hypersonic Shock Tunnel, a special pitot pressure rake (Fig. 3) was fabricated to house twenty-nine (29) piezoelectric pressure transducers (Romanelli Pinto et al. 2011a, 2011b).

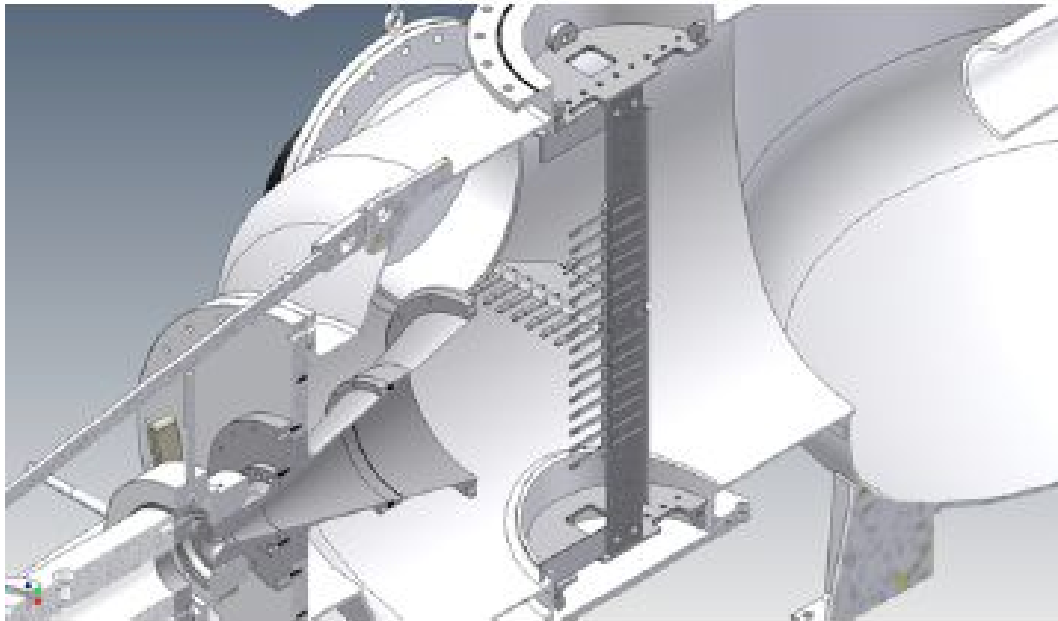


Figure 3: Schematic view of the pitot pressure rake mounted in the test section of the T3 Hypersonic Shock Tunnel.

The subsonic-supersonic (hypersonic) isentropic flow through variable-area ducts (convergent-divergent nozzle) may be used to determine the flow at the nozzle exit (Anderson, 1990), which will establish the hypersonic flow in the test section (Fig. 4).

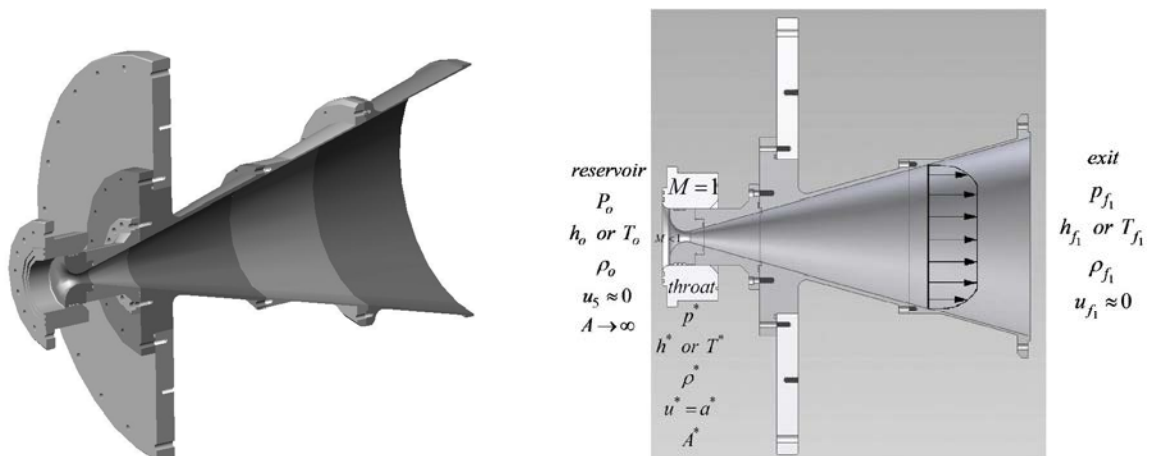


Figure 4: Schematic for subsonic-supersonic (hypersonic) isentropic flow through variable-area ducts (convergent-divergent nozzle).

D. Romanelli Pinto, C. E. O. Silva, S. B. Cruzelles, V. A. B. Galvão, T. V. C. Marcos, G. P. Camillo, and P. G. P. Toro
Brazilian 14-X Hypersonic Scramjet Aerospace Vehicle Experimental Investigation at Mach Number 10

The flow at the inlet of the nozzle comes from the compressed test gas (from the driven section, reservoir, of the T3 Hypersonic Shock Tunnel). The flow expands isentropically to supersonic (hypersonic) speeds at the nozzle exit, and the high stagnation enthalpy of the oncoming flow is converted into a freestream high speeds in the test section.

Figure 5 display typical pressure traces for the present investigation for high enthalpy air conditions. The first two traces (from left) measure the time of the incident shock wave in the driven section. The third trace measures the reservoir pressure and the fourth the impact pressure (pitot pressure).

The Schlieren photograph (Fig. 6) of Mach number 9.16 flow over a “pitot pressure” at the pitot pressure rake, shows that the shock is symmetrical, and the experimental stand-off distance of $\delta = 1.04 \text{ mm}$ (Fig. 6), agrees with the estimate (12%) given by the correlation of Billing (Anderson, 1990) $\delta = 1.18 \text{ mm}$, which is based on experimental data given by:

$$\frac{\delta}{R} = 0.143 \exp\left(\frac{3.24}{M_\infty^2}\right) \quad (1)$$

where R is the radius of the nose of the pitot pressure model and M_∞ is the Mach number of undisturbed flow.

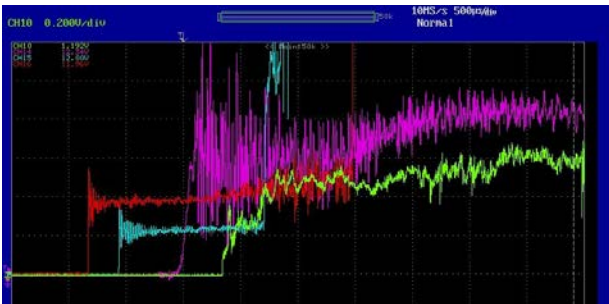


Figure 5: Typical pressure traces (voltage trace) for the present investigation.



Figure 6: Schlieren photograph of the pitot pressure of the pitot pressure rake.

The pressure distribution (Fig. 7) of hypersonic Mach number 9.16 flow at the pitot pressure rake at the exit of the nozzle, shows that the shock is symmetrical. The boundary layer is in between the external circle (pressure transducers 1, 13, 7, 8) and the internal circle (pressure transducers 2, 12, 6, 9). More data between both circles to clear the position of the boundary layer is needed (Romanelli Pinto et al. 2011a).

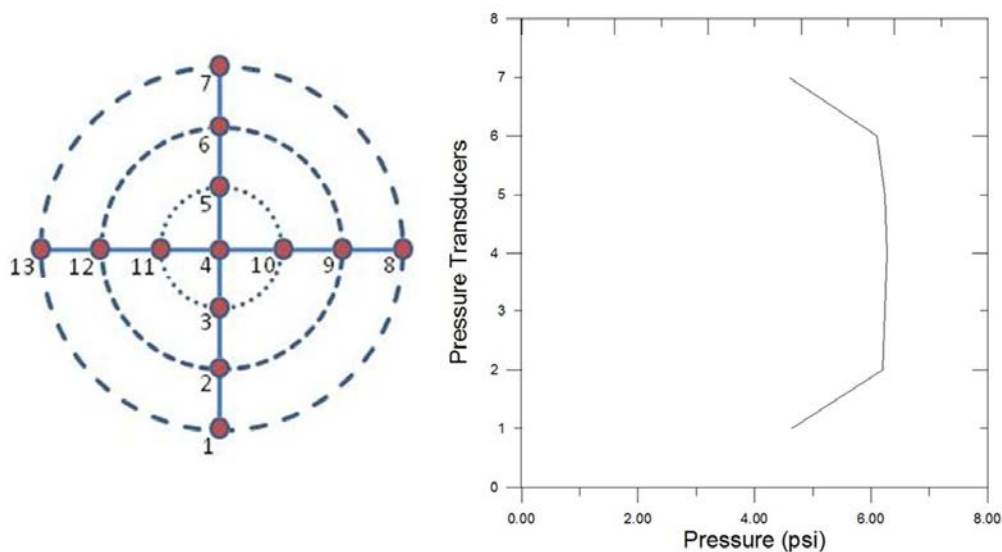


Figure 7: Pressure measurements at the nozzle exit for Mach number 9.16.

2.2 Dimensional design of the truncated 14-X Hypersonic scramjet

The combustor chamber design (Fig. 8), based on the numerical (Hyslop, 1998) and on the experimental (Kasal et al., 2002), consists by a duct with 265.1-mm. long constant area following by a duct with 134.0-mm. long with 4.27° (to accommodate the boundary layer and expansion due H_2 and O_2 combustion). The combustion chamber is coupled at the

lower surface of the 14-X Hypersonic waverider Aerospace Vehicle between the end of the 14.5° deflection compression ramp and the beginning at the 15° expansion ramp (related to the upper surface) defined by the 14-X waverider aerodynamic experimental results (Rolim, 2009, Rolim et al., 2009, Rolim et al., 2011).

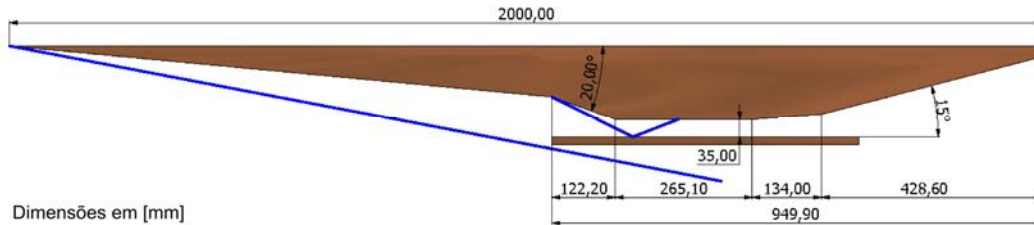


Figure 8: The 14-X waverider scramjet Mach number 10 model main dimensions.

To accommodate the model inside of the T3 Hypersonic Shock Tunnel, which has 0.60-m. diameter by 1.40-m. long test section, a 1.5-m. long truncated at the 5.5° waverider leading edge surface (Fig. 9) of the 14-X waverider scramjet model was designed (Costa, 2011; Costa et al., 2012).



Figure 9: The truncated 14-X waverider scramjet Mach number 10 model concept.

A 1.5-m. long truncated 14-X waverider scramjet stainless steel model was manufactured (Fig. 10). The 5.5° waverider leading edge surface of the truncated model was aligned with the free stream hypersonic flow, so that there is not conical shock wave generated by the 5.5° waverider leading edge. In this configuration only the boundary layer will be established at the 5.5° waverider leading edge surface during the experimental runs.

The truncated 14-X waverider scramjet model (Fig. 10) instrumented with 29 piezoelectric pressure transducers on the compression surface, had been experimental investigated (Fig. 11) on the equilibrium interface mode operation at the IEAv 0.60-m. nozzle exit diameter Hypersonic Shock Tunnel T3 at freestream at Mach numbers from 8 to 10. Quartz windows were designed and added to the combustion chamber to obtain schlieren photographs during the supersonic combustion runs.

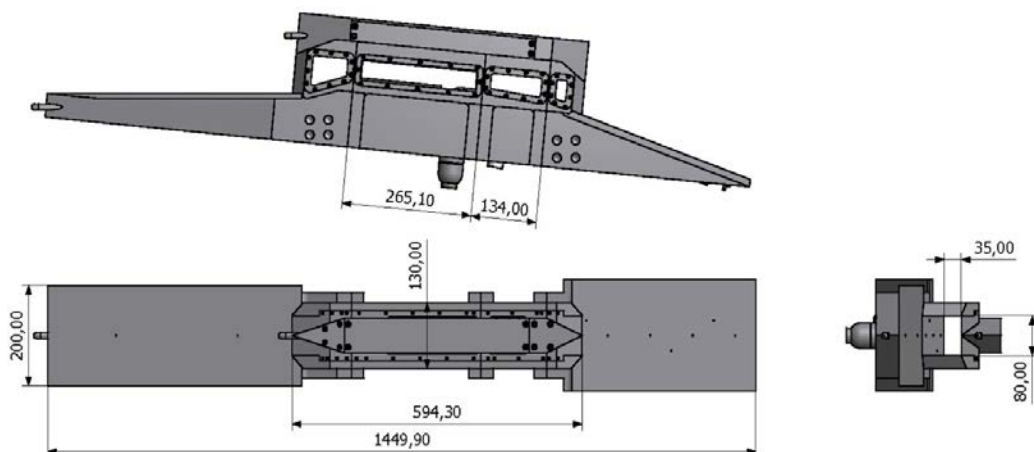


Figure 10: The truncated 14-X waverider scramjet Mach number 8.9 model main dimensions.

An ultra high-speed camera, manufactured by CORDIN, model 550, coupled with mirror-based schlieren 'Z' configuration (Fig. 12, left) with the schlieren light beam path and placement of the Cordin camera (Fig. 12, right) 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.

D. Romanelli Pinto, C. E. O. Silva, S. B. Cruzelles, V. A. B. Galvão, T. V. C. Marcos, G. P. Camillo, and P. G. P. Toro
Brazilian 14-X Hypersonic Scramjet Aerospace Vehicle Experimental Investigation at Mach Number 10

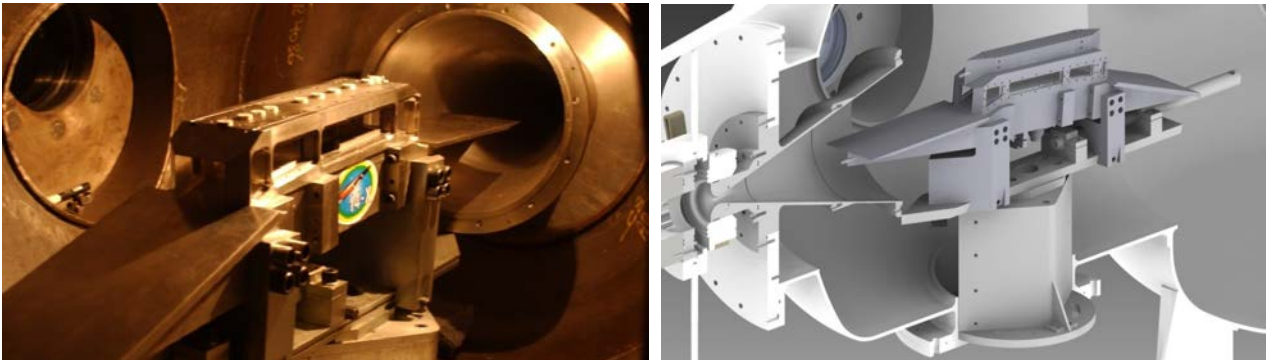


Figure 11: Instrumented truncated 14-X waverider scramjet model installed at the T3 Hypersonic Shock Tunnel.

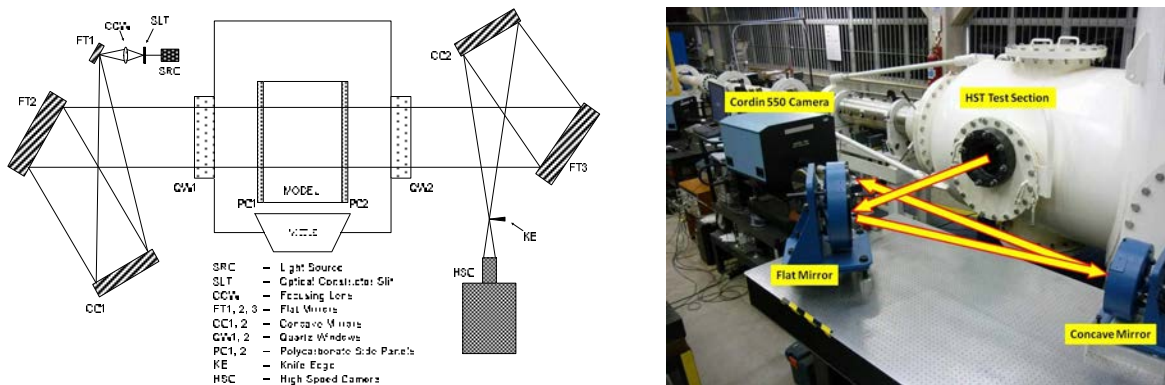


Figure 12: (Left) Schlieren visualization and high speed Cordin 550 camera system setup and (right) schlieren light beam path and placement of the speed Cordin 550 camera.

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.

3. RESULTS AND COMMENTS

3.1 Non Intrusive Schlieren Flow Visualization Technique

The 14.5° deflection compression ramp and the combustion chamber of the truncated 14-X Hypersonic scramjet model were aligned to the quartz windows of the test section of the T3 Tunnel (in different runs) to obtain the schlieren photographs (Figs. 13 and 14), respectively. The schlieren photographs are based on density gradient taken from the experimental investigations of a desired hypersonic Mach number.

Figure 15 shows a 19.85° oblique shock wave established at the manufactured 13.6° (designed 14.5°) deflection compression ramp.

A montage of the Fig. 13 and Fig. 14 was made to obtain the schlieren photograph (Fig. 16), which provides enough data to developed analytical theoretical analysis.

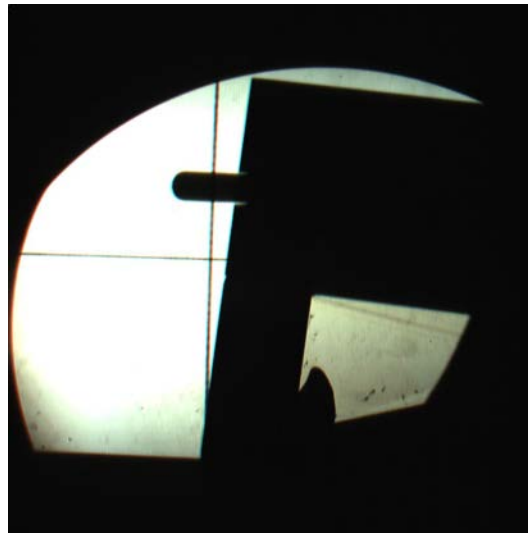


Figure 13: Schlieren photograph of the 14.5° deflection compression ramp.

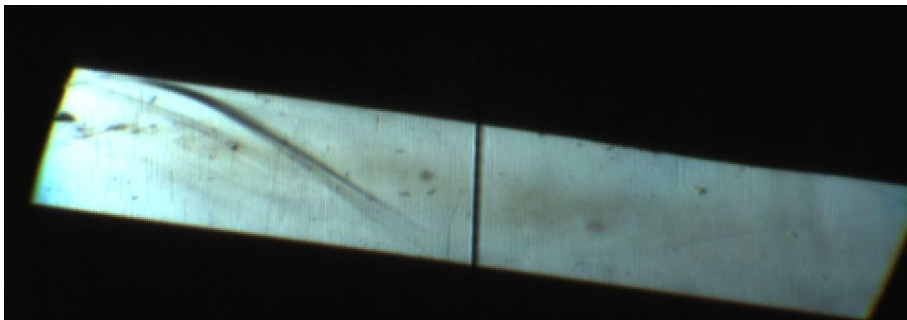


Figure 14: Schlieren photograph of the combustion chamber.

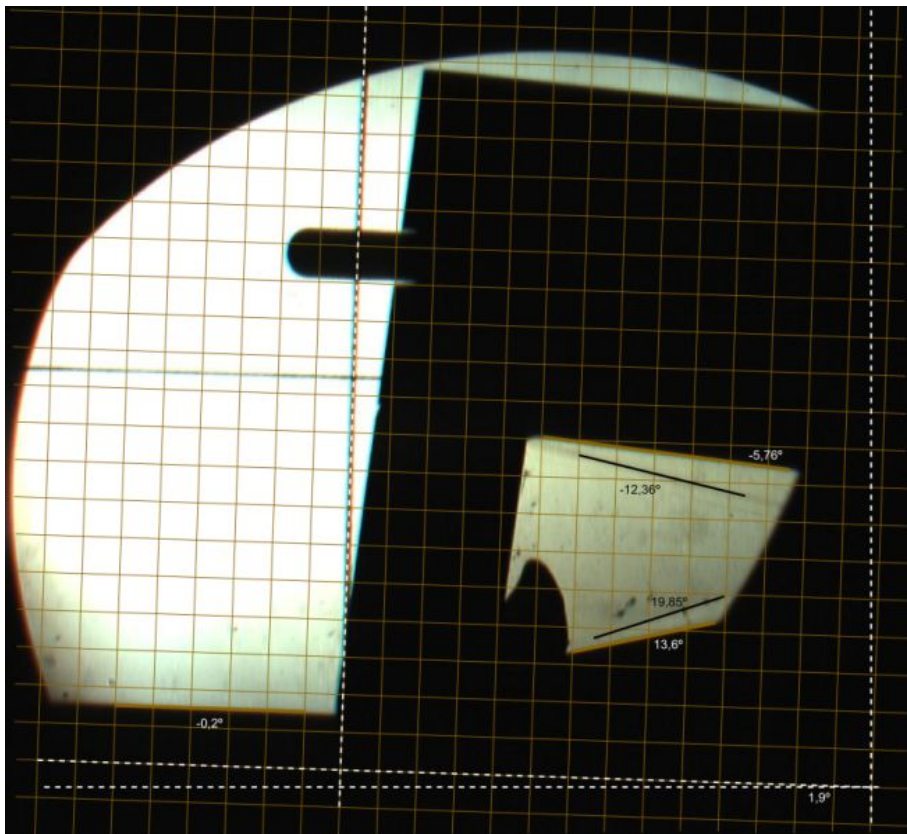


Figure 15: Oblique shock wave angle obtained from the schlieren photograph of the 14.5° deflection compression ramp.

D. Romanelli Pinto, C. E. O. Silva, S. B. Cruzelles, V. A. B. Galvão, T. V. C. Marcos, G. P. Camillo, and P. G. P. Toro
Brazilian 14-X Hypersonic Scramjet Aerospace Vehicle Experimental Investigation at Mach Number 10

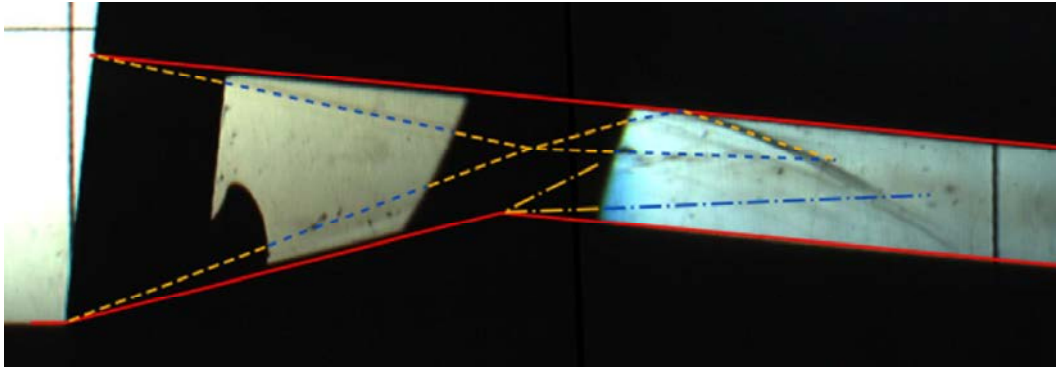


Figure 16: Mach number 10 flow past the model leading edge aligned with the horizontal.

4. ANALYTIC THEORETICAL ANALYSIS

In this analytical 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. 10) of the hypersonic vehicle with airframe-integrated scramjet engine lower surface.

4.1 External and Internal Compression Section (Oblique Shock Wave)

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

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

$$\frac{\rho_{out}}{\rho_{in}} = \frac{(\gamma+1)(M_{in} \sin\beta)^2}{(\gamma-1)(M_{in} \sin\beta)^2 + 2} \quad (2)$$

$$M_{out} = \frac{\sqrt{\frac{(M_{in} \sin\beta)^2 + \frac{2}{(\gamma-1)}}{\frac{2\gamma}{(\gamma-1)}(M_{in} \sin\beta)^2 - 1}}}{\sin(\beta - \theta_s)} \quad (3)$$

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:

$$\tan\theta_s = 2(\cot\beta) \left[\frac{(M_{in} \sin\beta)^2 - 1}{M_{in}^2(\gamma + \cos 2\beta) + 2} \right] \quad (4)$$

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 external and internal compression section of the hypersonic vehicle with airframe-integrated scramjet engine lower surface.

Note the $\frac{T_{out}}{T_{in}}$ may be calculated using the perfect gas relation $p = \rho RT$, therefore $\frac{T_{out}}{T_{in}} = \frac{p_{out}}{p_{in}} \frac{\rho_{in}}{\rho_{out}}$. This approach will be used in this work.

4.2 Internal and External Expansion Section (Expansion Wave)

The Prandtl-Meyer theory may be applied to the (internal and external expansion section, Fig. 10) expansion waves,

An isentropic expansion wave is limited by the head and tail of the expansion wave defined by the Mach angle μ_{head} , μ_{tail} , respectively, and they are given by:

$$\mu_{head} = \arcsen\left(\frac{1}{M_{in}}\right) \quad (5)$$

$$\mu_{tail} = \arcsen\left(\frac{1}{M_{out}}\right) \quad (6)$$

The expansion deflection angle θ_e is given by the Prandtl-Meyer function $\nu(M)$:

$$\theta_e = \nu(M_{out}) - \nu(M_{in}) \quad (7)$$

where: The Prandtl-Meyer function $\nu(M)$, which is function of the Mach number, is given by:

$$\nu(M) = \sqrt{\frac{\gamma+1}{\gamma-1}} \operatorname{tg}^{-1} \sqrt{\frac{\gamma-1}{\gamma+1} [M^2 - 1]} - \operatorname{tg}^{-1} \sqrt{M^2 - 1} \quad (8)$$

Once Mach number after expansion wave M_{out} is determined the closed form of the thermodynamic property (pressure, density and temperature) ratios across the expansion wave may be obtained by the isentropic relationships given by:

$$\frac{T_{out}}{T_{in}} = \left(\frac{1 + \frac{\gamma-1}{2} M_{in}^2}{1 + \frac{\gamma-1}{2} M_{out}^2} \right) \quad (9)$$

$$\frac{p_{out}}{p_{in}} = \left(\frac{T_{out}}{T_{in}} \right)^{\frac{\gamma}{\gamma-1}} \quad (10)$$

Note the flow across the expansion wave promote a decrease of pressure, density, temperature, and an increase of Mach number. The flow remains supersonic/hypersonic and parallel to the flat surface of the internal and external expansion section of the hypersonic vehicle with airframe-integrated scramjet engine lower surface.

4.3 Analytical Theoretical Results

The thermodynamic properties at the truncated VHA 14-X model lower surface may be determined based on the two-dimensional compressible flow (oblique shock wave relationships) and expansion wave (Prandtl-Meyer theory) considering the simplest case, i. e., no viscous flow, calorically perfect air ($\gamma = 1.4$) and scramjet engine with power off (Table 1). It was applied the following restriction: there is incident shock waves generated at the 5.5° attached leading-edge deflection angle due the alignment of the leading-edge with the undisturbed freestream air flow. A incident shock wave is established at the 14.5° deflection (following the leading-edge deflection). The reflected shock wave generated at the cowl leading-edge hits the entrance of the combustor station with the constant static pressure, static temperature, static density and constant supersonic airflow (Fig. 17).

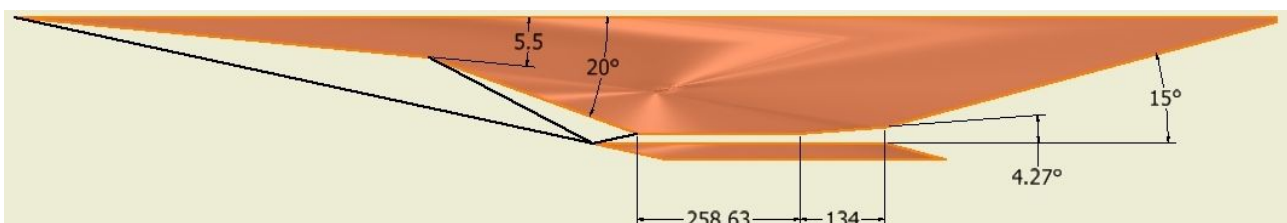


Figure 17: Cross-section of the truncated VHA 14-X.

D. Romanelli Pinto, C. E. O. Silva, S. B. Cruzelles, V. A. B. Galvão, T. V. C. Marcos, G. P. Camillo, and P. G. P. Toro
Brazilian 14-X Hypersonic Scramjet Aerospace Vehicle Experimental Investigation at Mach Number 10

Table 1: Thermodynamic properties at the truncated VHA 14-X model lower surfaces, power off, inviscid, $\gamma = 1.4$.

		station 0	station 1 (deflection 5.5°)	station 2 (deflection 14.5°)	station 3 (deflection 20°)	station 4 (Power off) (deflection 4.27°)	station 4 (deflection 10.73°)
M_{in}		7	7	6.0188	4.0645	2.6012	2.7981
θ_{in}	°		5.5	14.5	20	4.27	10.73
β_{out}	°		12.2429	22.1143	32.2384		
M_{out}			6.0188	4.0645	2.6012	2.7981	3.3715
T_{out}	K	226.5	296.6924	568.3735	1039.555	953.2721	747.1747
p_{out}	Pa	1197	2877.588	16755.91	89104.56	65803.72	28052.13
ρ_{out}	kg/m ³	0.01841	0.033788	0.102702	0.298605	0.240467	0.13079
a_{out}	m/s	301.7	345.9846	478.8732	647.6307	620.1719	549.0535
u_{out}	m/s	2111.9	2082.412	1946.38	1684.617	1735.303	1851.134
μ_{head}	°					22.6107	20.9405
μ_{tail}	°					20.9394	17.2536

5. CONCLUSION

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 The State of 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.

The Brazilian 14-X Hypersonic Aerospace Vehicle, designed at the Prof. Henry T. Nagamatsu Laboratory of Aerothermodynamics and Hypersonics, is part of the continuing effort of the Department of Aerospace Science and Technology (DCTA), to develop a technologic demonstrator using: i) "waverider" technology to provide lift to the aerospace vehicle, and ii) "scramjet" technology to provide hypersonic airbreathing propulsion system based on supersonic combustion.

Schlieren pictures supply by Mach number 10 flow over a 265.1-mm. long combustion chamber coupled at the lower surface of the truncated 14-X Hypersonic waverider Aerospace Vehicle agreed with the analytical theoretical results based on oblique shock wave.

Also, it is presented the methodology to design the lower surface of the truncated 14-X Hypersonic waverider Aerospace Vehicle using the analytical theoretical analysis based on the two-dimensional (plane) oblique shock wave compressible flow, and Prandtl-Meyer theory.

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