

## ENCIT-2018-0074

### A NEW APPROACH TO OPTIMIZE THE SCRAMJET INLET DESIGN APPLYING THE TOTAL PRESSURE RECOVERY

**Paulo Gilberto de Paula Toro**

**Alef Giovane Lima do Nascimento Fernandes**

**George Santos Marinho**

Universidade Federal do Rio Grande do Norte/UFRN - Centro de Tecnologia. Av. Senador Salgado Filho, 3000 - Campus Universitário, Lagoa Nova CEP 59.078-970 – Natal/RN - Brasil  
toro@ct.ufrn.br; alefgiovanee@gmail.com; gmarinho@ct.ufrn.br

**Gilvan Luiz Borba**

Universidade Federal do Rio Grande do Norte/UFRN - Centro de Ciências Exatas e da Terra. Av. Senador Salgado Filho, 3000 - Campus Universitário, Lagoa Nova, CEP 59.078-970 - Natal, RN - Brasil  
gilvan@geofisica.ufrn.br

**João Felipe de Araújo Martos**

European Space Research and Technology Centre (ESTEC)-ESA, Keplerlaan 1, 2001 AZ Noordwijk, The Netherlands  
Joao.Martos@esa.int

**Israel da Silveira Rêgo**

Instituto de Estudos Avançados/IEAv - Trevo Coronel Aviador José Alberto Albano do Amarante, nº1 Putim CEP. 12.228-001 São José dos Campos, SP - Brasil  
israel.rego@ieav.ct.br

**Abstract.** *A generic hypersonic airbreathing propulsion based on supersonic combustion ramjet (scramjet) technology has been designed, at the Universidade Federal do Rio Grande do Norte (UFRN), using analytical theoretical analysis (engineering approach). A two-dimensional hydrogen powered generic scramjet inlet has been designed to demonstrate, in atmospheric flight, a supersonic combustion, of atmospheric air (in supersonic speed) with hydrogen, on an acceleration mission to 2050 m/s (Mach number 6.8) at 30 km geometric altitude. In general in a preliminary design, one-dimensional compressible flow (shock wave) theory, which may readily describe many features of compression region of the airbreathing engine, may used to estimate the shock wave angles, thermodynamic properties and the velocities (Mach numbers) of the hypersonic atmospheric air flow, at the generic scramjet inlet. However, the deflection angles of the compression ramps are known. Design and optimization of a 2-D mixed compression generic scramjet models are investigated by the extended Oswatitsch classical optimization of a 2-D supersonic inlet procedure to scramjet inlets. The deflection angles of compression section ramps and the corresponding thermodynamic property ratio (pressure, temperature and specific mass) of the airflow as well as the airflow velocities (and corresponding Mach numbers) are obtained by the present optimization criterion. The present optimization criterion is defined based on the system of (n-1) incident oblique shockwaves, with the equal strength, and one reflected oblique shockwave establish in the external and internal compression section, respectively, which reduce the known hypersonic flow (pressure, temperature, specific mass and flow velocity) to a supersonic flow, with required temperature and velocity, at the entrance of a combustion chamber. Therefore, the maximum shock pressure recovery is obtained as function of the incident oblique shockwaves of equal strength. Thermodynamic property ratio (pressure, temperature and specific mass) as well as the flow velocities (and corresponding Mach numbers) are presented for seven incident oblique shockwaves.*

**Keywords:** *scramjet, supersonic combustion ramjet, hypersonic airbreathing propulsion*

## 1. INTRODUCTION

A new revival in aerospace vehicle design to fly in hypersonic velocities started just after the successfully atmospheric flights of three scramjet demonstrators (HyShot by University of Queensland scramjet at Mach number about 7.5, Australia, 2002; X-43 by NASA scramjet at Mach number 7 and 10, USA, 2004; X-51 by U.S. AirForce scramjet at Mach number close to 5, USA 2010) those burned fuel with atmospheric air in supersonic velocities in the combustor chambers of the scramjet.

Basically, scramjet is a fully highly integrated airbreathing aeronautical system, where engine and vehicle are indistinguishable (Fig. 1), with no moving parts, that uses the oblique/conical shock waves generated during the hypersonic flight, to provide compression and deceleration of freestream atmospheric air at the inlet of the scramjet, which are pushed to combustion chamber. Fuel, at least sonic speed, may be injected into the supersonic airflow just downstream of the inlet or at the beginning of the combustion chamber (combustor). Right after, both oxygen (from the atmospheric air) and on-board hydrogen fuel are mixed. The combination of the high energies of the fuel and of the oncoming supersonic airflow starts the combustion at supersonic speed. Finally, the divergent exhaust nozzle at the afterbody vehicle accelerates the exhaust gases, providing thrust.

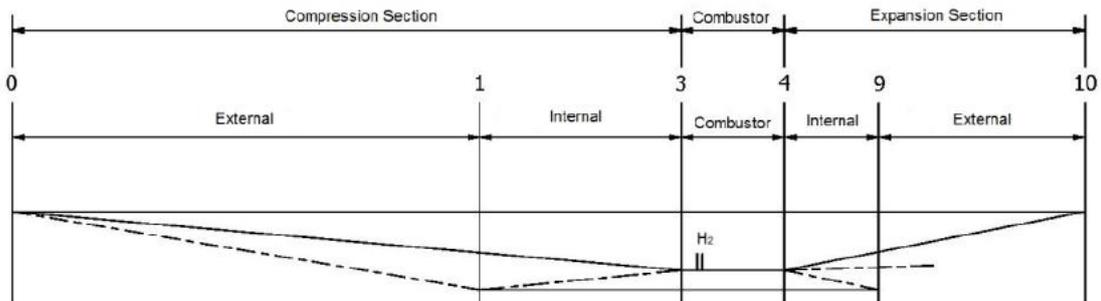


Figure 1. Generic fully airframe-integrated scramjet, adapted from Heiser and Pratt (1994).

Theoretically, the scramjet engine can fly from Mach number 4 to 20, which can push the existing maximum flight speed limit for airbreathing propulsion based on gas turbine engines, those may reach Mach number 3.5.

Scramjet may be divided in three main components (Fig. 1): external and internal compression section (inlet), combustion chamber (combustor), and internal and external expansion section (outlet).

Stations 0 and 1 are the leading edges of the scramjet and of the cowl, respectively. Stations 3 and 4 are the entrance and exit of the combustion chamber. Stations 9 and 10 are the trailing edges of the cowl and the scramjet, respectively.

Since scramjet rely on the combustion of the hydrogen fuel and the oxygen from the supersonic atmospheric airflow compressed by the external compression section (Fig. 1) the optimization of the scramjet inlet is one of the most important feature for tip-to-tail fully highly integrated scramjet engine design.

The basic function of the compression section is to provide the desired temperature of the atmospheric air in supersonic velocity at the entrance of the combustor chamber higher than the ignition temperature of the fuel (hydrogen). Therefore, the scramjet inlet must be able to capture a designed amount of surrounding atmospheric air with maximum compression efficiency and minimum loss of the total pressure.

The (Instituto de Estudos Avançados) IEAv's Brazilian researchers are being designed the fully airframe-integrated scramjet 14-X S, to demonstrate, in free flight, the supersonic combustion (scramjet technology) of the on-board hydrogen with the atmospheric air in supersonic velocity, at 30 km altitude with Mach number 7 (approximately 2100 m/s).

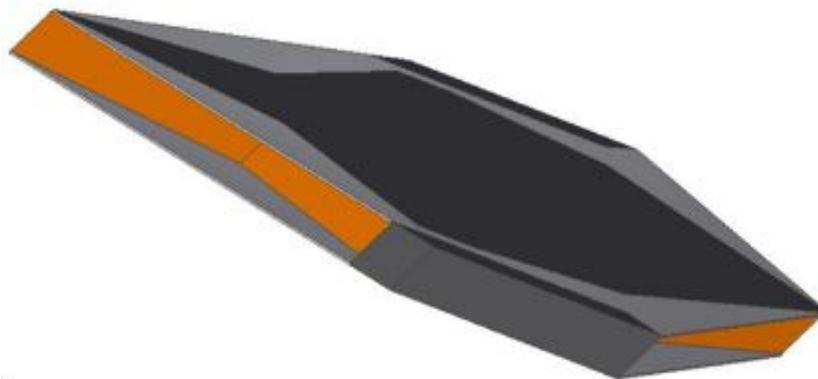


Figure 2. Demonstrator scramjet 14-X S, with two compression ramps.

The original 14-X S consists a two-dimensional configuration (Fig. 2), with a constant cross-section (Fig. 3), which shows only the half of the constant cross-section. The scramjet consists of the compression section with a leading-edge angle of  $5.5^\circ$ , compression ramp angle of  $14.5^\circ$  (related to the angle of the leading edge), the internal expansion chamber combustion angle of  $4.27^\circ$  and external expansion angle of  $10.73^\circ$  (related to the angle of internal expansion).

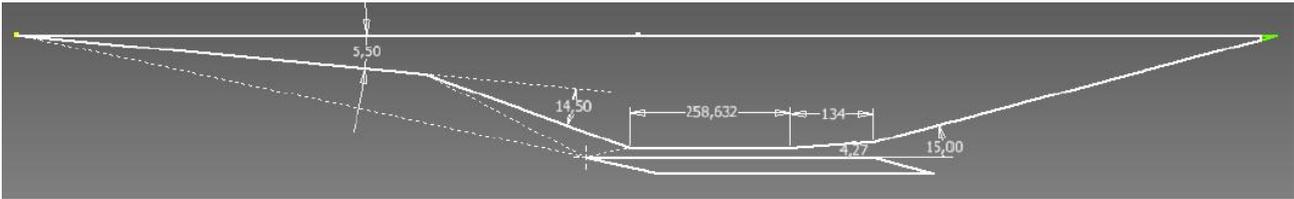


Figure 3. Cross-section of the 14-X S (with two compression ramps) flying at 30km altitude at Mach number 7.

The IEAv's Brazilian researchers decided to design a new version of the scramjet 14-X S, with three compression ramps (Fig. 4), with the turning angles of  $5.5^\circ$ ,  $7^\circ$  and  $8.5^\circ$ , to demonstrate the scramjet technology to fly in the Earth's atmosphere, at 30km altitude and velocity corresponding the Mach number 6.8.

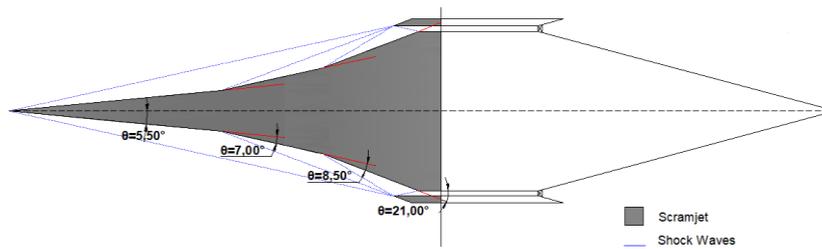


Figure 4. Scramjet 14-X S flying at Mach number 6.8, to demonstrate supersonic combustion at 30 km altitude.

This scientific paper presents the optimization method to obtain the turning angles of the compression ramps of the scramjet inlet to provide the desired temperature of the atmospheric air in supersonic velocity at the entrance of the combustor chamber higher than the ignition temperature of the fuel (hydrogen), capturing a designed amount of atmospheric air with maximum compression efficiency.

## 2. METHODOLOGY

### 2.1 Scramjet characteristics

First, it is necessary to establish a nomenclature to be used in the scramjet design. Heiser and Pratt (1994) present the terminology of the scramjet, which (as mentioned) may be divided in three main components (Fig. 1): external and internal compression section (inlet), combustion chamber (combustor), and internal and external expansion section (outlet).

The external compression section is governed by incident shock wave, while the internal is governed by reflected shock wave. The internal and external expansion section is governed by expansion wave, Prandtl-Meyer Theory, and area ratio. The constant area section of the combustion chamber is called as isolator and is used to uniformize the flow from the compression section. Fuel is injected right after the isolator used to expand the gases from burning the fuel and the oxygen. In general, one-dimensional flow with heat addition, Rayleigh flow, is used to simulate the burning the fuel and the oxygen.

This fully airframe-integrated scramjet, where engine and vehicle are indistinguishable, is caused by the fact that the front section of the vehicle contributes to the compression of atmospheric air, while the rear contributes to the generation of thrust. The net thrust produced by the scramjet is the difference between the thrust (force that propels the vehicle) generated by the expansion of exhaust gases from the rear of the engine and the total drag (force that resists the movement of the vehicle). These forces may produce thrust to the flight of the vehicle or not depending on the balance of these forces in engine design in question.

### 2.2 Scramjet inlet design by total pressure recovery

Ran and Mavris (2005) defined an optimization criterion used to determine the ramp angles of the oblique shock waves for a maximum pressure recovery of the supersonic inlet. The optimization criterion was, first, proposed by Oswatitsch (1944), a Germany aerodynamicist, which was applied for a system of  $n-1$  oblique shocks and one terminal normal shock in two dimensions (Fig. 5), reducing a supersonic freestream airflow to a subsonic atmospheric air at the combustion chamber. The maximum shock pressure recovery is obtained when the shock waves are of equal strength, i.e., the Mach numbers perpendicular to the individual shock waves are equal, which the relationship is given by:

$$M_1 \text{sen}\beta_1 = M_2 \text{sen}\beta_2 = \dots = M_{n-1} \text{sen}\beta_{n-1} \quad (1)$$

The modified optimization criterion to maximize total pressure recovery, developed by Ran and Mavris (2005), was applied for hypersonic scramjet inlet (Martos, 2017), which the freestream hypersonic airflow is decelerated to the supersonic speed, at the combustion chamber, through a suitable shock system (including only oblique shock waves).

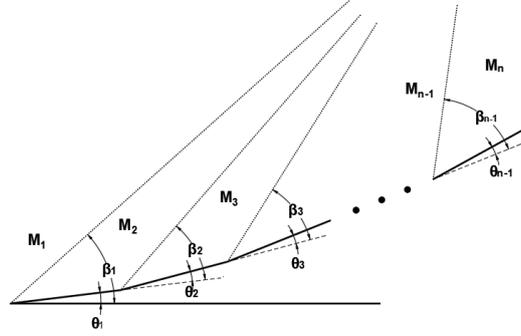


Figure 5. The optimization criterion to maximize total pressure recovery (Ran and Mavris, 2005).

The total pressure recovery is presented by Heiser and Pratt (1994) as a parameter  $\pi$ , given by:

$$\pi = \frac{p_{te}}{p_{ti}} = \frac{p_e}{p_i} \left[ \frac{\left(1 + \frac{\gamma-1}{2} M_e^2\right)^{\frac{\gamma}{\gamma-1}}}{\left(1 + \frac{\gamma-1}{2} M_i^2\right)^{\frac{\gamma}{\gamma-1}}} \right] \quad (2)$$

The modified oblique shock equations used by Ran and Mavris (2005), to solve the  $n-1$  oblique shocks (Fig. 5), may be obtained from NACA TR 1135 (1953), a Ames Aeronautical Laboratory report.

In this paper the conventional oblique shock wave relationships, obtained from any textbook, will be applied coupled to the required temperature of the atmospheric air (higher than the ignition temperature of the desired fuel) with supersonic Mach number to burn at stoichiometrically the fuel/air mass flow ratio, which may be based on the products of burning hydrogen and air.

### 2.3 Oblique shock wave relationships

Considering no boundary-layer effects (inviscid flow) and for calorically perfect gas ( $p = \rho RT$ ,  $\gamma = \text{constant}$ ) the mass, momentum and energy conservation laws (Anderson, 1990) in two-dimensional steady state, non-viscous, no heat conduction compressible flow, may be applied to plane oblique shock wave (Fig. 6).

The normal component of the upstream velocity  $V_{in}$  (corresponding to Mach number  $M_{in}$ ) is given by

$$M_{in_{in}} = M_{in} \text{sen}\beta \quad (3)$$

The shock wave angle  $\beta$  (Fig. 6) is a function of the incoming local supersonic/hypersonic flow Mach number  $M_{in}$ , the gas from the atmosphere  $\gamma$  (air in the Earth's planet,  $\gamma=1.4$ ) and the deflection angle  $\theta_s$ , and it may be obtained iteratively with the relationship given by (Anderson, 1990):

$$\text{tg}\theta_s = 2(\cotg \beta) \left[ \frac{(M_{in} \text{sen}\beta)^2 - 1}{M_{in}^2(\gamma + \cos 2\beta) + 2} \right] \quad (4)$$

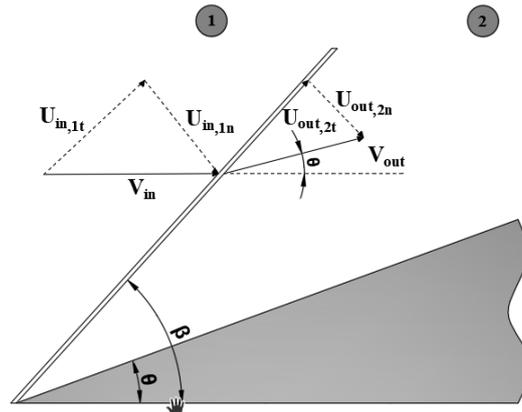


Figure 6. Leading edge incident plane oblique shock wave geometry.

The normal component of the downstream velocity  $V_{out}$  (corresponding to Mach number  $M_{out}$ ) is given by

$$M_{out\ 2n}^2 = \frac{(M_{in\ sen\beta})^2 + \frac{2}{(\gamma-1)}}{\frac{2\gamma}{(\gamma-1)} (M_{in\ sen\beta})^2 - 1} \quad (5)$$

Additionally, the supersonic Mach number and the thermodynamic property (static pressure, static density, static temperature, ...) ratios, across the oblique shock, are given by:

$$M_{out} = \frac{M_{out\ 2n}}{sen(\beta - \theta_s)} = \frac{\sqrt{\frac{(M_{in\ sen\beta})^2 + \frac{2}{(\gamma-1)}}{\frac{2\gamma}{(\gamma-1)} (M_{in\ sen\beta})^2 - 1}}}{sen(\beta - \theta_s)} \quad (6)$$

$$\frac{p_{out}}{p_{in}} = 1 + \frac{2\gamma}{(\gamma+1)} \left[ (M_{in\ sen\beta})^2 - 1 \right] \quad (7)$$

$$\frac{\rho_{out}}{\rho_{in}} = \frac{(\gamma+1)(M_{in\ sen\beta})^2}{\left[ (\gamma-1)(M_{in\ sen\beta})^2 + 2 \right]} \quad (8)$$

$$\frac{T_{out}}{T_{in}} = \frac{p_{out}}{p_{in}} \frac{\rho_{in}}{\rho_{out}} = 1 + \frac{2\gamma}{(\gamma+1)} \left[ (M_{in\ sen\beta})^2 - 1 \right] \frac{\left[ (\gamma-1)(M_{in\ sen\beta})^2 + 2 \right]}{(\gamma+1)(M_{in\ sen\beta})^2} \quad (9)$$

where:  $\rho$ ,  $p$ ,  $u_{1n}$ ,  $u_{1t}$ ,  $h$  are density, pressure, normal and tangential velocities across the plane oblique shock wave and enthalpy of the gas, respectively.

Note, the flow across the plane 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 compression section (Fig. 6) of the hypersonic vehicle with airframe-integrated scramjet engine lower surface.

Also, the leading-edge incident plane oblique shock wave theory may be used for incident oblique planar shock wave (compression ramp angle) and the reflected shock wave.

## 2.4 Required Mach number and temperature of the atmospheric air at the entrance of combustion chamber

The required temperature of the atmospheric air at the entrance of combustion chamber can be obtained by the total (stagnation) temperature  $T_o$  across the oblique shockwave, which is given by

$$T_o = \left(1 + \frac{\gamma-1}{2} M_0^2\right) T_0 = \left(1 + \frac{\gamma-1}{2} M_3^2\right) T_3 = (\text{constant}) \quad (10)$$

where:  $T_o$  the total (stagnation) temperature, or

$$T_3 = \frac{1 + \frac{\gamma-1}{2} M_0^2}{1 + \frac{\gamma-1}{2} M_3^2} T_o \quad (11)$$

where:  $M_0$  and  $M_3$  are the Mach numbers, and  $T_0$  and  $T_3$  are the Mach numbers and static temperatures at the freestream (flight velocity of the scramjet) and combustion chamber conditions, respectively.

The required Mach number and temperature at the entrance of combustion chamber are assumed as 2.09 and 1237.14 [K], respectively.

## 2.5 Significant properties of hydrogen and hydrocarbon

Scramjet is a reaction jet engine with high specific impulse used for airbreathing hypersonic flight that may provide higher the specific impulse than the conventional rockets by burning hydrogen fuel in a supersonic atmospheric compressed airflow (Heiser and Pratt, 1994). Also, safety, storage and handling of hydrocarbon (except for cryogenic methane) are easier than the hydrogen.

Due to high heats of reaction (Tab. 1) hydrogen is the only fuel that might deliver net positive thrust up to Mach number 24, near orbital velocity. However, due to high density (Tab. 1) hydrocarbon fuels may result in superior overall performance for Mach number below 8. Hydrogen will be used for the present case study of Mach number 6.8.

Table 1. Ignition temperature and heats of reaction for typical gaseous hydrogen and gaseous hydrocarbon fuels reacting with air at the standard reference state.

Fuel	Chemical Formula	Ignition temperature	(heats of reaction)		Density	$\gamma$	Gas constant
		$T_{ig}$	$h_{pr}$	$f_{st}$			
		CRC (1985)	Heiser e Pratt (1994)				
	$C_xH_y$	(K)	(MJ/kg)	-	kg/m <sup>3</sup>	-	(J/kg.K)
Hydrogenic	H <sub>2</sub>	845.15	119.954	0.0291	82	1.404	4124.16
Methane	CH <sub>4</sub>	810.15	50.010		424	1.32	518.35
Ethane	C <sub>2</sub> H <sub>6</sub>	745.15	47.484			1.183	276.5
Hexane	C <sub>6</sub> H <sub>14</sub>	498.15	45.100				96.48
Octane	C <sub>8</sub> H <sub>18</sub>	479.15	44.786	0.0664	703	1.044	72.79
JP-7	C <sub>12</sub> H <sub>25</sub>	514.15	43.90325		790		
JP-10	C <sub>10</sub> H <sub>16</sub>	518.15	42.100				

## 3. RESULTS AND COMMENTARIES

First is necessary to define the thermodynamic atmospheric air properties, which the generic scramjet will perform atmospheric flight, at 30 km geometric altitude (Tab. 2) and speed corresponding to Mach number 6.8.

Table 2. Thermodynamic atmospheric properties at 30 km altitude (U.S. Standard Atmosphere, 1976).

Altitude	Temperature	Pressure	Density	Mean free path	Sound speed	Dynamic viscosity
km	K	Pa	kg/m <sup>3</sup>	m	m/s	N s/m <sup>2</sup>
30	226.5	1197	0.01841	0.000004413	301.7	0.000014753



$M_{1n}^i$	3.38121388	1.98581626	1.60438996	1.43098475	1.33333081	1.27114989	1.22825291
$M_{2n}^i$	0.45599707	0.57979880	0.66712054	0.72701548	0.76956796	0.80102613	0.82510580
$\frac{p_{out}^i}{p_{in}}$	13.1713752	4.43404396	2.83641167	2.22233691	1.90739956	1.71845905	1.59337277
$\frac{T_{out}^i}{T_{in}}$	3.15530294	1.67600954	1.39100052	1.27478550	1.21199685	1.17267650	1.14571991
$\frac{\rho_{out}^i}{\rho_{in}}$	4.17436153	2.64559588	2.03911618	1.74330262	1.57376609	1.46541612	1.39071753
$\frac{p_{oout}^i}{p_{oin}}$	0.23604097	0.72749880	0.89356004	0.95012334	0.97316494	0.98404601	0.98978934
Internal compression section (reflected shock waves)							
	1 ramps	2 ramps	3 ramps	4 ramps	5 ramps	6 ramps	7 ramps
$M_{in}^r$	3.35251072	3.63886175	3.74676647	3.79507985	3.82008266	3.83449287	3.84349077
$\theta_s^r$	21.999983	24.349723	25.096946	25.410737	25.568402	25.657851	25.713187
$\beta_s^r$	37.7618940	39.1507305	39.6046959	39.7970380	39.8940553	39.9492067	39.9833654
$M_{out}^r$	2.09313097	2.09313091	2.09313097	2.09313092	2.09313088	2.09313093	2.09313094
$\frac{p_{out}^r}{p_{in}}$	4.75068346	5.99127711	6.48917365	6.71737587	6.83674470	6.90593436	6.94928267
$\frac{T_{out}^r}{T_{in}}$	1.73105069	1.94445491	2.02940607	2.06824680	2.08854280	2.10030075	2.10766503
$\frac{\rho_{out}^r}{\rho_{in}}$	2.74439303	3.08121164	3.19757279	3.24785990	3.27345204	3.28806927	3.29714758
$\frac{p_{oout}^r}{p_{oin}}$	0.69610033	0.58442237	0.54500194	0.52794828	0.51927514	0.51432452	0.51125128
$M_{1n}^r$	2.05301523	2.29744151	2.38851543	2.42911669	2.45008653	2.46215951	2.46969333
$M_{2n}^r$	0.56857920	0.53471700	0.52435224	0.52005575	0.51790934	0.51669526	0.51594552
Supersonic combustion conditions							
$T_3$ [K]	1237.140584	1237.140589	1237.140537	1237.140488	1237.140548	1237.140506	1237.14048
$M_3$	2.09313097	2.09313091	2.09313097	2.09313092	2.09313088	2.09313093	2.09313094

First, the optimization criteria defined of a system of (n-1) incident oblique shockwaves, with the same equal strength, given by  $M_{in} \sin\beta$  (normal component  $M_{1n}^i = M_{in} \sin\beta$  of the upstream velocity  $V_1^{\text{freestream}}$ , corresponding to  $M_1$ ), and one reflected oblique shockwave establish in the external and internal compression section (Fig. 5), respectively, reduce (Tab. 3) the known hypersonic flow  $p_1^Z, T_1^Z, \rho_1^Z$  and  $M_1$  to a supersonic flow, with required temperature  $T_3$  higher than the ignition temperature of the hydrogen  $T_{H_2}^{\text{ignition}} = 845.17[\text{K}]$ , (Tab. 1), and velocity corresponding to  $M_3$ , at the entrance of a combustion chamber.

Also, the deflection angles (turning flow), the incident oblique shockwaves and the Mach number after the incident shockwaves are obtained by the present optimization criteria.

Finally, for a given number of ramps at the external compression section, not only the normal component  $M_{in} \sin\beta$  are equal across ( $M_{1n}^i, M_{2n}^i$ ) the incident shockwaves, but also all thermodynamic property ratios  $\frac{p_{out}^i}{p_{in}^i}, \frac{T_{out}^i}{T_{in}^i}, \frac{\rho_{out}^i}{\rho_{in}^i}$

(Tab. 3). Automatically, the total pressure ratio  $\frac{p_{oout}^i}{p_{oin}^i}$  across the incident shockwaves are constant.

As the number of the ramps increase, the normal components before and after the incident shockwaves decrease and increase, respectively. Also, the normal components before and after the incident shockwaves are always Higher and lower than the Mach number 1, as expected. Finally, as the number of the ramps increase the deflection angles of the external compression section, the incident shockwave angles and the thermodynamic property ratios decrease and the Mach number after the shockwaves and the total pressure ratios increase.

#### 4. CONCLUSION AND OUTLOOK FOR FUTURE PROJECTS

A two-dimensional hydrogen powered generic scramjet, designed at the Universidade Federal do Rio Grande do Norte (UFRN), based on the technological demonstrator scramjet 14-X S in development at the Instituto de Estudos Avançados (IEAv), has been considering to demonstrate, in atmospheric flight, a supersonic combustion, of atmospheric air (in supersonic speed) with hydrogen, on an acceleration mission to 2050 m/s (Mach number 6.8) at 30 km geometric altitude.

The supersonic flow conditions at the entrance of the combustion chamber are assumed as supersonic Mach number 2.09 and temperature of 1237.14 [K] higher than the ignition temperature of the hydrogen of 845.15 [K].

The deflection angles (turning flow), the incident oblique shockwaves and the Mach number after the incident shockwaves of the 2-D mixed compression generic scramjet models are determined applying the present optimization criteria, which use as initial conditions the freestream conditions to reaches the supersonic combustion conditions.

From the analysis obtained for seven scramjet configurations studied in the present paper, one may conclude as the number of the ramps goes to infinite, the total pressure ratios approaches to 1.

#### 5. ACKNOWLEDGEMENTS

The first four authors would like to thanks to Universidade Federal do Rio Grande do Norte (UFRN) the support given to complete this scientific work. The first author (as Visiting Professor) would like to express appreciation to UFRN the possibility to apply the knowledge in aerothermodynamics and hypersonics in the research area in hypersonic airbreathing propulsion based on supersonic combustion ramjet (scramjet) technology. This work has been partially sponsored by European Space Research and Technology Centre (ESTEC) and the fifth author would like to extend his appreciation to support this project. Finally, a portion of this effort was supported by Instituto de Estudos Avançados (IEAv) and by the Propulsão Hipersônica 14-X (PropHiper) Project and the last author would like express his appreciation.

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