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DESIGN OF AN ACADEMIC SCRAMJET FOR ATMOSPHERIC CAPTIVE FLIGHT AT MACH NUMBER 4.18 COUPLED TO A ROCKET FTI

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Abstract. An (academic) planar symmetric scramjet is being developed (by a Undergraduate Students) at the Universidade Federal do Rio Grande do Norte/UFRN, to perform atmospheric captive flight in high supersonic speed corresponding to Mach number 4.18, at 6.2 km of altitude. The academic scramjet will be launched by the Foguete de Treinamento Intermediário/FTI released from the Brazilian Launch Centers of Barreira do Inferno (CLBI) or of Alcântara (CLA). The Brazilian Launch Centers CLBI and CLA are the only national institutions that have the proper infrastructure (control center, payload-rocket integration lab, launch platform and tracking radars) and expertise in performing the launch of scramjet demonstrators. The FTI is used by CLBI and CLA for training of local operational teams. The FTI reaches speed close to Mach number 4 at 6.2 km of altitude, just the operating speed of the scramjet. It will have no combustion during the firsts atmospheric captive flight of the academic scramjet. Analytical methodology is used to design the academic planar symmetric scramjet. One-dimensional compressible flow (shock wave) theory, one-dimensional flow with heat addition and the one-dimensional compressible flow (expansion wave) theory coupled to the area ratio are applied to design the compression, combustor and expansion sections of the academic scramjet, respectively, considering no effects of boundary layer and calorically perfect gas airflow. For a given design conditions of any scramjet flying at high Mach number in the Earth's dense atmospheric the presented methodology in this work is capable of evaluating the thermodynamic properties and Mach numbers (corresponding to flow velocities) from the leading-edge to trailing-edge of an airframe-integrated scramjet.

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

1. INTRODUCTION

The scramjet is the only airbreathing propulsion system able to provide the thrust needed efficiently in hypersonic flight (Curran, 2001). In addition, it has the advantage over rocket engines not lead to oxidizing substance, reducing vehicle weight. To get an idea of structural weight savings this fact, one should bear in mind that the first stage of the Saturn-1, a rocket widely used by NASA, must carry 285ton of liquid oxygen to burn 125ton of RP-1 (a type of highly refined kerosene for rocket).

The recent success to demonstrate scramjet concept, through the (2004 about 10s burnt hydrogen scramjet-powered at Mach 7 and 10) X-43 flights (Moses et al., 2004; Marshall et al., 2005) and the (2010 about 140s burnt hydrocarbon scramjet-powered at Mach number 5+) X-51 flight (Hanks et al., 2008) provided by the new U.S. hypersonics strategy formulated (after NASP program) by NASA, U.S. Government agencies (Air Force, Army and Navy) and DARPA for the next generation of space transportation systems under NASA Marshall Space Flight Center's Advanced Space Transportation Program (ASTP) gave a fresh renaissance in hypersonic flight.

In 2007, Brazilian researchers, from Laboratório de Aerotermodinâmica e Hipersônica Prof. Henry T. Nagamatsu, at Instituto de Estudos Avançados (IEAv), proposed to design, to develop, to manufacture and to demonstrate, in free flight, a technological scramjet to provide hypersonic airbreathing propulsion system based on supersonic combustion.

The fully airframe-integrated scramjet 14-X waverider (Fig. 1), which include a waverider technology to provide lift to the aerospace vehicle (Rolim et al., 2009), and the scramjet 14-X S (Fig. 2) (Cardoso et al., 2013) are being designed to demonstrate supersonic combustion during atmospheric flight in about 30 km of altitude, in hypersonic speeds corresponding to the numbers of Mach 10 (approximately 3000 m/s) and 7 (approximately 2100 m/s), respectively.

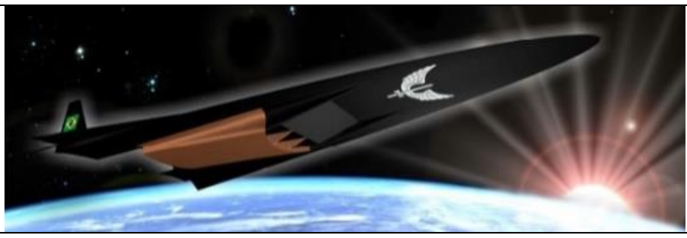

Figure 1. Demonstrator scramjet 14-X waverider.

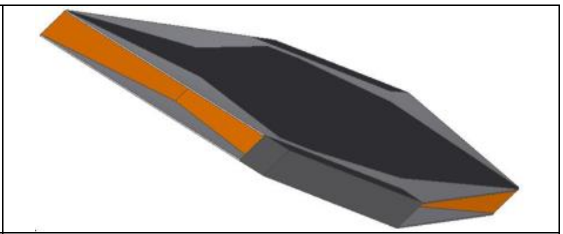

Figure 2. Demonstrator scramjet 14-X S.

Basically, scramjet is a fully integrated airbreathing aeronautical engine, with no moving parts, that uses the oblique/conical shock waves generated during the high-speed flight, to provide compression and deceleration of freestream atmospheric air at the inlet of the scramjet, which are pushed to combustion chamber (Fig. 4), at supersonic speed. 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.

In consequence of the nature of the scramjet engines, scramjets are unable to produce thrust while stationary. Solid-rocket engines may be used to accelerate the scramjet to a speed such that the shock waves produced by the air inlet are able to compress the atmospheric air achieving the operational conditions (Hass et al., 2005). Such approach may provide an affordable path for maturing Brazilian hypersonic airbreathing components and subsystems in flight.

There is no available Brazilian accelerator vehicle to perform Mach number 10 at 30 km of altitude to accelerate the 14-X waverider (Fig. 1). The Brazilian hypersonic accelerator vehicle, which is composed by two-stage (S31 and S30) solid rocket engines, unguided, rail launched, is able to accelerate the 14-X S (Fig. 2) to the predetermined flight test conditions of the scramjet operation (30 km altitude at Mach number 6.8, approximately 2050 m/s) from one of the Brazilian Launch Centers (Centro de Lançamento de Alcântara, CLA, or Centro de Lançamento da Barreira do Inferno, CLBI).

Finally, before launch the IEAv's 14-X S scramjet (Fig. 2) and in order to prepare Brazil in this strategic and emerging area, RD&I in all phases, the IEAv proposes the design of an (academic) scramjet (Pivetta et al., 2016; Santos et al., 2016), considering from the aerothermodynamic configuration conception, aiming the atmospheric flight at altitude of about 6 km using the Foguete de Treinamento Intermediário (FTI).

In addition, in order to accelerate the human resource qualification in this strategic and emerging space area the IEAv created the Instituto Nacional de Ciência e Tecnologia em Propulsão Hipersônica Aspirada (INCT Pro-Hyper) with the objectives to carry out scientific research and technological development for the solution of problems related to the design of an aerospace vehicle that uses hypersonic airbreathing propulsion system based on supersonic combustion (scramjet). The INCT Pro-Hyper proposes transfer the knowledge to the society (Brazilian Universities) and to Brazilian private industries through the design of a new technological scramjet, which involve the research, development and innovation (RD&I) focused on a scramjet technology, with the following tasks: studies of existing scramjet concepts, definition of the appropriate configuration, development and implementation of the necessary methodologies and design of a scramjet to achieve atmospheric flight at the operating conditions, i.e., at the speed corresponding to Mach number 7 in the altitude of 30 km. Additionally, The INCT Pro-Hyper proposes the design of an (academic) scramjet, which involves aerothermodynamic studies, structural and thermal designs, integration with accelerator vehicle, material definitions, stability and trajectory studies, definition of embedded system, definition of monitoring parameters of atmospheric flight, navigation and control system studies, telemetry, post-flight analysis of telemetry data, covering the complete envelope of atmospheric flight from the launching up to hypersonic speeds.

Several researchers from Universidade Federal do Rio Grande do Norte/UFRN, Instituto de Estudos Avançados/IEAv and Centro de Lançamento da Barreira do Inferno/CLBI have been joined, with the objective to conduct scientific research and technological development related to not only to aero-thermo-structural design of an academic scramjet to be integrated to the rocket FTI, but also to launch and to monitor during the atmospheric flight and to analyze the data obtained by real time telemetry. Therefore, the design of the academic scramjet aims to design, using the same methodologies to be applied to the technological scramjet, to manufacture and to perform atmospheric flight, but using the Rocket FTI released from the Brazilian Launch Centers CLBI) and CLA.

The FTI is used by CLBI and CLA for training of local operational teams. The FTI is the speed of Mach number 4, just the operating speed of the supersonic combustion, but it is planned to have no combustion during the first flights. However, these scramjets coupled to FTI launch will allow to capacitation of Undergraduate and Graduate Students, supervised by Researchers or Professors, in the following areas: scramjet design, scramjet integration as payload, flight control, telemetry, launch procedures and post-flight analysis of telemetry data.

The Brazilian Launch Centers CLBI and CLA are a satellite launching base of the Brazilian Space Agency (AEB), located on Brazil's northeast and north Atlantic coast, respectively, outside Natal and São Luis cities (capital of Rio Grande do Norte and Maranhão States). Both CLBI and CLA are operated by the Departamento de Ciência e Tecnologia Aeroespacial/DCTA (Department of Aerospace Science and Technology). CLBI and CLA are the world's closest launching base to the equator line, which gives the launch sites a significant advantage in launching geosynchronous satellites, an attribute shared only by the Guiana Space Center (utilized by France), and their position nearer the equator offer an advantage over Cape Canaveral (USA).

2. METHODOLOGY

2.1 Scramjet characteristics

First, it is necessary to establish a nomenclature to be used in the scramjet design. Following Heiser and Pratt (1994) the scramjet may be divided in three main components and by several stations (Fig. 3): external (stations 0 to 1, governing by incident shock wave) and internal (stations 1 to 3, governing by reflected shock wave) compression section (inlet), combustion chamber (combustor, stations 3 to 4, governing by one-dimensional flow with heat addition, Rayleigh flow) and internal (stations 4 to 9) and external (stations 4 to 9) expansion section (outlet, governing by expansion wave, Prandtl-Meyer Theory and area ratio).

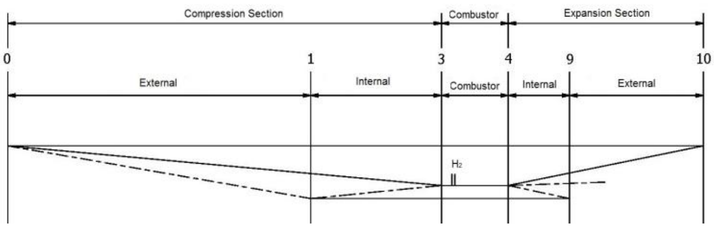

Figure 3: Hypersonic vehicle with airframe-integrated scramjet engine stations and reference terminology.

An important feature of the scramjet is a highly integrated system, where engine and vehicle are indistinguishable. This tight integration 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.

Finally, note that the incident shock wave (Fig. 3), from the external compression section, should incident on the leading-edge of the cowl (shock on-lip) and the reflected shock wave, of the internal compression section, should incident on the entrance of the combustion chamber (shock on-corner). Therefore, the air capture area should be maximum and the length of the scramjet compression section should be minimum.

2.2 Governing equations

Analytical theoretical analysis may be used as first step to calculate the thermodynamic properties (pressure, temperature, density, sound velocity) and Mach number at the flow path from the leading-edge to the trailing-edge of the hypersonic vehicle with airframe-integrated scramjet.

Analytical theoretical analysis provides closed form relationships to determine the thermodynamic air property ratios and Mach number applied to one-dimensional compressible flow (shock wave) theory, one-dimensional flow with heat addition and the one-dimensional compressible flow (expansion wave) theory coupled to area ratio, used to design the compression, combustor and expansion sections of any scramjet, respectively, considering no effects of boundary layer and calorically perfect gas airflow.

In the analytical theoretical analysis, the subscripts in and out are used to identify the upstream (inlet) and the downstream (outlet) conditions, respectively, of each station (Fig. 3) of any scramjet design.

Oblique shock wave relantionships

Considering no boundary-layer effects (non-viscous flow) and calorically perfect gas ($p=\rho RT$, $\gamma=\text{constant}$) the incident oblique shock wave (Fig. 4) relationships, at the leading-edge of the aerospace vehicle, as well as at the intersection of two consecutively plane surfaces and at reflected oblique shock wave, can be easily obtained as closed form of the shock wave angle β , the thermodynamic (static pressure, static density, static temperature, sound velocity) property ratios and Mach number across the oblique shock as function of the incoming local supersonic/hypersonic flow Mach number M_{in}, the gas from the atmosphere γ (air in the Earth's planet, $\gamma=1.4$) and the deflection angle θ (Anderson, 1990).

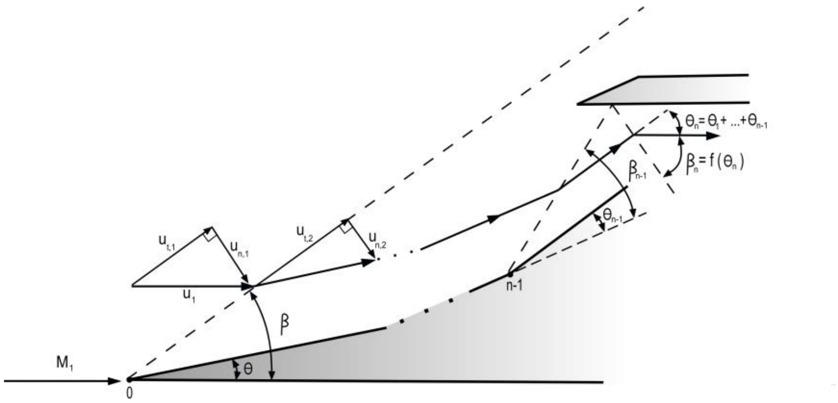

Figure 4. Incident and reflected oblique shockwave geometries.

The shock wave angle β (Fig. 4), the Mach number after the shock wave and the thermodynamic property ratios are given by:

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

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

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

$$\frac{\rho_{out}}{\rho_{in}} = \frac{\left[(\gamma+1)(M_{in}\ sen\beta)^2\right]}{\left[(\gamma-1)(M_{in}\ sen\beta)^2+2\right]} \tag{4}$$

$$\frac{T_{out}}{T_{in}} = \frac{\dfrac{p_{out}}{p_{in}}}{\dfrac{\rho_{out}}{\rho_{in}}} = \left\{1+\frac{2\gamma}{(\gamma+1)}\left[(M_{in}\ sen\beta)^2-1\right]\right\}\left\{\frac{\left[(\gamma-1)(M_{in}\ sen\beta)^2+2\right]}{\left[(\gamma+1)(M_{in}\ sen\beta)^2\right]}\right\} \tag{5}$$

It is important point out the plane oblique shock wave theory may be used to determine the flow conditions after the incident shock wave established (attached) at the leading-edge with the turning angle θ_1. The same theory may be used not only to the incident shock wave established (attached) at the intersection of two compression surfaces with the turning angle θ_2, but also to the reflected shock wave with the turning angle θ_3. One may observe the flow, after the reflected shock wave, should be aligned to the confined structure (Fig. 4).

The thermodynamic property ratios may be obtained in any gas dynamic textbooks, ex: Anderson (1990). Also, one may 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 compression section (Fig. 4) of the hypersonic vehicle with airframe-integrated scramjet engine lower surface (Fig. 4).

One-dimensional flow with heat addition

One-dimensional with constant-area heat addition, Rayleigh Flow, (Fig. 5) may be applied to combustion processes between the entrance and the exit of the scramjet combustor (Fig. 3), where the combustion processes correspond to heat addition at constant pressure, constant density, constant temperature and constant Mach number, respectively, at the inlet of the scramjet combustor.

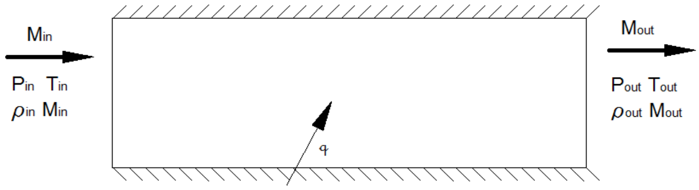

Figure 5. Rayleigh Flow, one-dimensional with constant-area heat addition.

The conditions at the entrance of the combustion chamber (subscripts in) are known, given by the thermodynamic properties and speed (Mach number) estimated just right after the reflected shock wave at the internal compression section (Fig. 3). For the calorically perfect gas ($p = \rho RT$, $\gamma = \text{constant}$) and no boundary-layer effects analytical closed-form relationships, of the thermodynamic property ratios, for one-dimensional flow with heat addition, which may be obtained in any gas dynamic textbooks, are easily obtained as function of the incoming local supersonic flow Mach number M_{in}, the gas from the atmosphere γ (air in the Earth's planet, $\gamma = 1.4$) (Anderson, 1990).

The heat added to the flow due to change the total energy (temperature), is given by:

$$q = \dot{m}_0 \, c_p \left(T_{o,out} - T_{o,in} \right) \tag{6}$$

where: the captured air mass flow rate and total temperature is given by:

$$\frac{T_o}{T} = 1 + \frac{\gamma - 1}{2} M^2 \tag{7}$$

$$\dot{m}_0 = \rho_0 u_0 A_0 \tag{8}$$

Closed-form of the thermodynamic property (static pressure, static density, static temperature and total temperature) ratios across constant-area heat addition are given by:

$$\frac{p_{out}}{p_{in}} = \left(\frac{1 + \gamma M_{in}^2}{1 + \gamma M_{out}^2} \right) \tag{9}$$

$$\frac{\rho_{out}}{\rho_{in}} = \left(\frac{1 + \gamma M_{out}^2}{1 + \gamma M_{in}^2} \right) \left(\frac{M_{in}}{M_{out}} \right)^2 \tag{10}$$

$$\frac{T_{out}}{T_{in}} = \frac{\dfrac{p_{out}}{p_{in}}}{\dfrac{\rho_{out}}{\rho_{in}}} = \left(\frac{1 + \gamma M_{in}^2}{1 + \gamma M_{out}^2} \right)^2 \left(\frac{M_{out}}{M_{in}} \right)^2 \tag{11}$$

$$\frac{T_{o,out}}{T_{o,in}} = \left(\frac{1+\gamma M_{in}^2}{1+\gamma M_{out}^2}\right)^2 \left(\frac{M_{out}^2}{M_{in}^2}\right)^2 \left(\frac{1+\frac{\gamma-1}{2}M_{out}^2}{1+\frac{\gamma-1}{2}M_{in}^2}\right) \tag{12}$$

Rayleigh flow (one-dimensional flow with heat addition) may be applied to the combustion process to burn H_2 and O_2 in supersonic speed, resulting at the exit of the combustor chamber (outlet) an increase in static pressure, static density, static temperature and a decrease the Mach number (speed) of the combustion product flow.

Burn hydrogen with supersonic airflow

The chemical energy rate available to the scramjet engine is given by (Heiser e Pratt, 1994)

$$\text{chemical energy rate} = \dot{m}_{\text{fuel}}\ h_{pr} \tag{13}$$

where: h_{pr} is the heats of reaction, and for hydrogen is $119{,}954{,}000\ [\text{J/kg fuel}]$(Heiser and Pratt, 1994). The chemical energy rate should be the heat addition estimated by the supersonic airflow at the entrance of the combustor (Eq. 3).

Expansion wave and area ratio

Considering no boundary-layer effects (non-viscous flow) and calorically perfect gas ($p = \rho RT$, $\gamma = \text{constant}$) closed-form of the thermodynamic property (static pressure, static density, static temperature) ratios across the expansion wave (Fig. 6), which is limited by the head and tail of the expansion wave defined by the Mach angle μ_{head} , μ_{tail} , respectively, given by (Anderson, 1990):

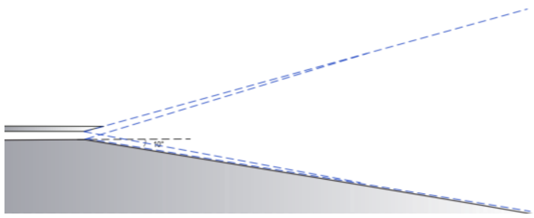

Figure 6: Expansion wave geometry and confined expansion wave geometry.

$$\mu_{head} = \text{arcsen}\left(\frac{1}{M_{in}}\right) \tag{14}$$

$$\mu_{tail} = \text{arcsen}\left(\frac{1}{M_{out}}\right) \tag{15}$$

The expansion deflection angle θ_{e} is given by the Prandtl-Meyer function $\upsilon(\text{M})$:

$$\theta_e = \upsilon(M_{out}) - \upsilon(M_{in}) \tag{16}$$

where the Prandtl-Meyer function $\upsilon(\text{M})$, which is function of the Mach number, is given by:

$$\upsilon(M) = \sqrt{\frac{\gamma+1}{\gamma-1}}\ tg^{-1}\sqrt{\frac{\gamma-1}{\gamma+1}\left[M^2-1\right]} - tg^{-1}\sqrt{M^2-1} \tag{17}$$

Once Mach number after expansion wave M_{out} is determined the closed form of the thermodynamic property (static pressure, static density and static temperature) ration across, which may be obtained in any gas dynamic textbooks, ex: Anderson (1990), the expansion wave may be obtained by the isentropic relationships given by:

$$\frac{T_{out}}{T_{in}} = \left(\frac{1 + \frac{\gamma - 1}{2} M_{in}^2}{1 + \frac{\gamma - 1}{2} M_{out}^2} \right) \tag{18}$$

$$\frac{p_{out}}{p_{in}} = \left(\frac{T_{out}}{T_{in}} \right)^{\frac{\gamma}{\gamma - 1}} \tag{19}$$

$$\frac{\rho_{out}}{\rho_{in}} = \frac{\frac{p_{out}}{p_{in}}}{\frac{T_{out}}{T_{in}}} = \left(\frac{T_{out}}{T_{in}} \right)^{\frac{1}{\gamma - 1}} \tag{20}$$

Note the flow across the expansion wave promote a decrease of static pressure, static density, static temperature, and an increase of Mach number. The flow remains supersonic and parallel to the flat surface of the internal and external expansion section (Fig. 6) of the hypersonic vehicle with airframe-integrated scramjet engine lower surface.

Area ratio (expansion wave)

If the supersonic flow, which establishes the expansion wave, is confined (Fig. 6), the head Mach wave strikes the lower surface of this confined environment and is reflected toward the upper surface. In this case the Prandtl-Meyer theory is not valid any more, and the area ratio should be applied (Heiser and Pratt, 1994). The reflected head Mach wave is assumed to be the same as the incident head Mach wave, which is given by

$$\frac{A_{out}}{A_{in}} = \frac{M_{in}}{M_{out}} \left(\frac{1 + \frac{\gamma - 1}{2} M_{out}^2}{1 + \frac{\gamma - 1}{2} M_{in}^2} \right)^{\frac{\gamma + 1}{2(\gamma - 1)}} \tag{21}$$

3. RESULTS AND COMMENTARIES

First, it is necessary to define the thermodynamic atmospheric air properties, which the generic scramjet will perform atmospheric flight, at 6.2 km geometric altitude (Tab. 1) and speed corresponding to Mach number 4.18.

Table 1. Thermodynamic atmospheric properties at 30 km altitude (U.S. Standard Atmosphere, 1976).

Altitude	Temperature	Pressure	Density	Sound speed
km	K	Pa	kg/m^3	m/s
6.2	247.85	45,901.4	0.64517	315.6

The $\theta - \beta - M$ relation (Eq. 1) is applied to obtain the incident oblique shockwave angles as well as the reflected oblique shockwave angle (Tab. 2). Following, Mach number (and corresponding flow velocity) and the thermodynamic air properties after each event (Tab. 2), at external (incident shock waves) and internal (reflected shock wave) supersonic compression section are evaluated by the thermodynamic property ratios (Eqs. 2-5) and the freestream thermodynamic properties (Tab. 1). Then, knowing the constraints of the accelerator vehicle used to accelerate the academic scramjet to 6.2 km, in velocity corresponding to Mach number 4.18, and imposed the Mach number at the end of the combustion chamber is supersonic (Mach number 1.2), the captured air mass flow rate (Eq. 8) at the inlet and the heat addition (Eq. 7), needed to burn the hydrogen and supersonic airflow at the combustor chamber, are evaluated using the one-dimensional flow with heat addition. After, the fuel mass flow rate is evaluated (Eq. 13). Finally, the Mach number (and corresponding flow velocity) and the thermodynamic air properties after the expansion section (Tab. 2), are estimated by expansion wave (Prandtl-Meyer) theory. But, if the head of the expansion wave is reflected and hits the expansion surface (Fig. 6), it is necessary to couple the area ratio equation to evaluate the properties and the velocity (and corresponding Mach number) at the trailing-edge of the academic scramjet (Eq. 21).

Table 2. Thermodynamic properties, at the compression section, of the academic scramjet engine, non-viscous flow, calorically perfect gas ($p = \rho RT$, $\gamma = 1.4$).

		Freestream	External compression surface (incident shock waves)				Internal compression surface (reflected shock wave)
			Ramp 5°	Ramp 5.5°	Ramp 6°	Ramp 6.5°	
M_{in}		4.18	4.18	3.79	3.43	3.07	2.74
θ_{in}	o		5	5.5	6.0	6.5	23
β_{out}	o		17.38	19.23	21.35	23.83	43.92
M_{out}	-		3.79	3.43	3.07	2.74	1.67
p_{out}	Pa	45,901	75,835	125,223	206,940	339,369	1,373,667
ρ_{out}	kg/m^3	0.64517	0.92001	1.31165	1.87092	2.65449	6.68
T_{out}	K	247.85	287.12	332.80	385.28	444.86	716.08
a_{out}	m/s	315.60	339.65	365.67	393.45	422.78	536.40
u_{out}	m/s	1319.20	1287.27	1254,25	1211.83	1158.42	865.79

The academic scramjet, with four ramps at the compression section (Fig. 7), with leading-edge angle of 5º, followed by three deflection angles of 5.5º, 6º and 6.5º, flying at 6.2 km of geometric altitude with speed corresponding at Mach number 4.18 is capable to generate a supersonic velocity of 865.79 m/s, corresponding to Mach number of 1.67, and a static temperature of 716.08 K (Tab. 2) at the entrance of combustion chamber, lower than 845.15 K, the ignition temperature of hydrogen (Coordinating Research Council, 1983). Consequently, it is not possible to demonstrate a supersonic combustion when hydrogen may burn (spontaneously) the supersonic atmospheric air, at the combustion chamber, during the atmospheric hypersonic flight of this academic scramjet. In this case, it is necessary to use an igniter to start the ignition.

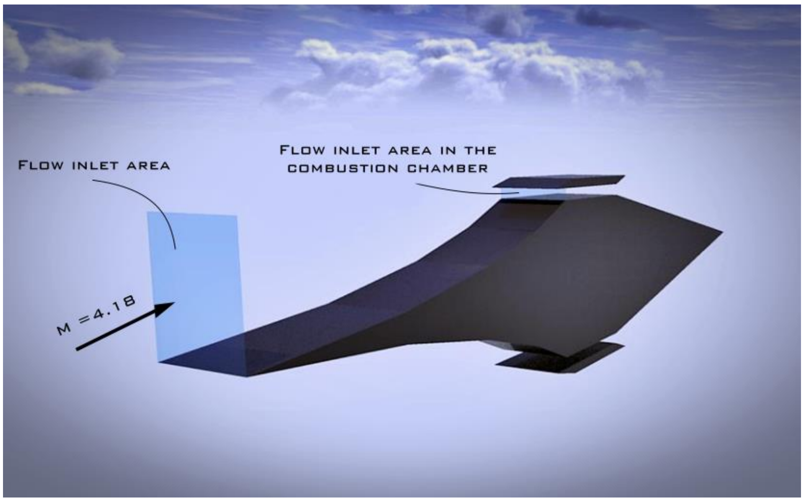

Figure 7. Academic scramjet flying at Mach number 4.18 to demonstrate supersonic combustion at 6.2 km altitude.

As mentioned before, the supersonic airflow across the plane attached shockwave promotes an increase of thermodynamic properties (pressure, temperature, density) and a decrease of flow velocity (Mach number), and the plane streamline of the airflow remains supersonic and parallel to the flat surface (Tab. 2) of the academic scramjet starting at 5º leading-edge deflected angle.

Observe the total temperature at each event of oblique incident and reflected shockwaves are constants (Tab. 2), as one may expect since the energy is conserved. Also, note the total pressure recovery (Eq. 13) is approximately constants across the internal compression surface (with turning angles of 5º, 5.5º, 6º and 6.5º) related to incident shock waves.

In the present work, for a given leading-edge turning angle of 5º the optimization criterion to maximize total pressure recovery was applied for hypersonic scramjet inlet to obtain the others turning angles of internal compression section (Fig. 2).

After the conditions at the entrance of the combustion chamber were obtained, the heat addition (Eq. 3), at the combustor section due to burn, using probably an igniter, the hydrogen and atmospheric air in supersonic speed, and fuel mass flow rate (Eq. 7) were evaluated. The (captured area of the scramjet) air inlet height was defined to be 0.255 [m], therefore, for constant width, the combustor height is 0.0362 [m]. The inlet and the combustor air mass flow rates are equal to 21.7 [kg/s]. To avoid the chocked flow after combustion, it is assumed the combustion product is supersonic as Mach number 1.2. For this case, the heat addition (Eq. 3) necessary to stoichiometrically burn the hydrogen is 3,003,858 [J/kg air]. Then, the fuel mass flow rate is 0.02504 [kg/s] (Eq. 7), where the chemical energy rate is assumed as the heat addition by the supersonic airflow at the entrance of the combustor (Eq. 3).

After, it was applied the Prandtl-Meyer theory and was determined the length of this theory is valid, considering the expansion waves those occurs at the expansion section of the scramjet and of the cowl (Fig. 8). The reflected expansion wave related to head of the expansion Mach angle is the same as the incident head of the expansion Mach angle (Eq. 8).

For the length from the head of the incident to reflected Mach angle was applied the Prandtl-Meyer theory (Eqs. 8-11), after the area ratio was applied (Eq. 12) and the conditions at the trailing-edge of the scramjet (point 10, Fig. 2) were determined (Tab. 3), considering the expansion deflection angle of 10º.

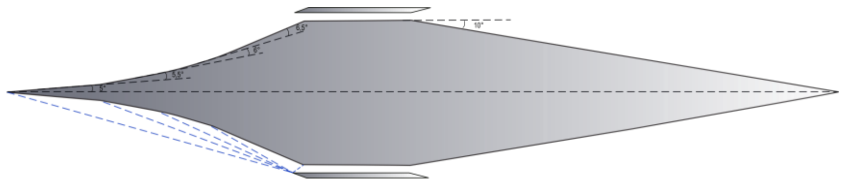

Figure 8. Cross section of the generic scramjet inlet.

Table 3. Thermodynamic properties, at the combustor and expansion sections, of the academic scramjet engine, non-viscous flow, calorically perfect gas ($p = \rho RT$, $\gamma = 1.4$).

					$\theta_{Expansion} = 10^{\circ}$	
		Freestream	Combustor entrance (Tab. 2)	Combustor exit	Expansion section (Prandtl-Meyer)	Expansion section (area ratio)
M		4.18	1.67	1.2	1.556	4.14
p	Pa	45,901	1,373,667	1,672,055	1,017,899	22,281.18
ρ	kg/m^3	0.64517	6.68	7.99472	5.60844	0.36584
T	K	247.85	716.08	971.85	843.36	283.00
a	m/s	315.60	536.40	624.95	582.18	337.24
u	m/s	1319.20	865.79	749.94	905.87	1395.19
T_{total}	K		----	1251.74	1251.74	1251.74

Besides of the Mach number at the trailing-edge is lower than the Mach number at the leading-edge, the velocity of the burning products, of hydrogen and supersonic airflow (at the combustor), at the trailing-edge is slightly higher than the velocity of the academic scramjet. The thermodynamic properties of pressure and density, at the trailing-edge, are 22% and 20% higher than the Earth's pressure and density at the 6.2 km geometric altitude. But, the temperature is 30% lower than the Earth's temperature at the 6.2 km altitude. The thermodynamic properties, from tip-to-tail, of the academic scramjet (Tabs. 2 and 3) is summarized by Fig. 9.

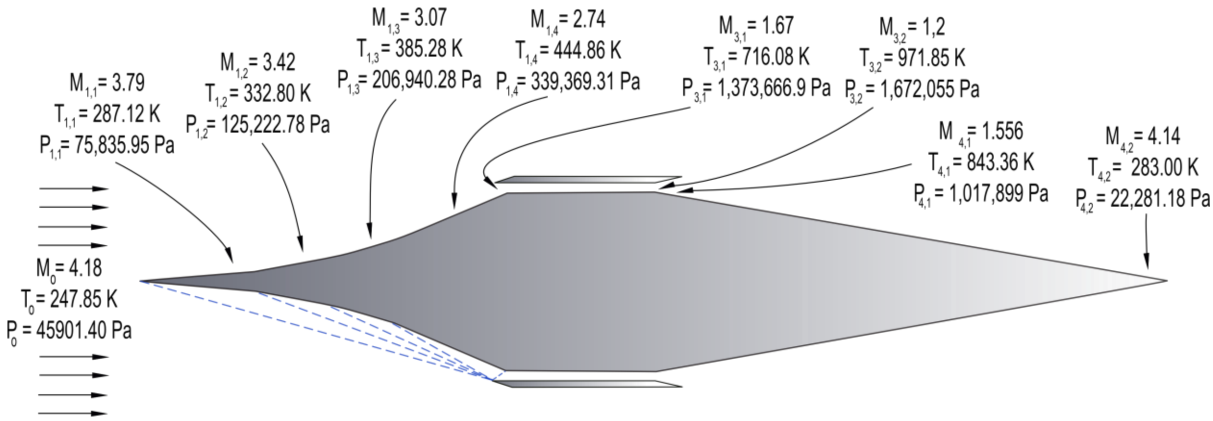

Figure 9. Thermodynamic properties and Mach number of the air flow from tip-to-tail of the academic scramjet.

4. CONCLUSION AND OUTLOOK FOR FUTURE PROJECTS

The thermodynamic properties and airflow Mach number, from leading-edge to trailing-edge, of the academic scramjet flying at 6.2 km altitude with Mach number 4.18 (1319.20 m/s), designed at the Universidade Federal do Rio Grande do Norte (UFRN) using analytical theoretical analysis (engineering approach), are estimated applying the one-dimensional compressible flow (shock wave) theory, one-dimensional flow with heat addition and the one-dimensional compressible flow (expansion wave) theory coupled to the area ratio to the compression, combustor and expansion sections, respectively, considering no effects of boundary layer and the airflow behaves as calorically perfect gas.

The academic scramjet, with four ramps (5º, 5.5º, 6º and 6.5º) at the compression section and one ramp (10º) at the expansion section (Fig. 8), flying at 6.2 km of geometric altitude with speed corresponding at Mach number 4.18 is capable to generate thrust, when hydrogen is burning by a supersonic velocity of 865.79 m/s, corresponding to Mach number of 1.67, and a static temperature of 716.08 K at the combustion chamber.

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