COBEM 2021 26th International Congress of Mechanical Engineering
ABCM Associação Brasileira de Engenharia e Ciências Mecânicas
November 22-26, 2021, Virtual Congress, Brazil

COB-2021-1183
REVIEW OF FUEL INJECTION FOR HYPERSONIC AIRBREATHING PROPULSION

David Romanelli Pinto
Instituto Tecnológico de Aeronáutica (ITA), Praça Marechal Eduardo Gomes, 50 - Vila das Acacias, CEP: 12228-900, São José dos Campos/SP - Brazil
Prosimulador Tecnologia de Trânsito, Av. das Nações Unidas 14261, 17th floor, ward B - Vila Gertrudes, CEP: 04794-000, São Paulo/SP - Brazil
romanellibr@gmail.com

Luiz Henrique Silva Marques Soares
Instituto Tecnológico de Aeronáutica (ITA), Praça Marechal Eduardo Gomes, 50 - Vila das Acacias, CEP: 12228-900, São José dos Campos/SP - Brazil
luizlhsms@ita.br

Israel da Silveira Rêgo
Instituto de Estudos Avançados (IEAv), Divisão de Aerotermodinâmica e Hipersônica, Trevo Coronel Aviador José Alberto Albano do Amarante, nº1 - Putim, CEP: 12.228-001, São José dos Campos/SP - Brazil
regoisrael@hotmail.com

Pedro Paulo Batista de Araújo
Paulo Gilberto de Paula Toro
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 - Brazil
araujo.projects@gmail.com, toro11pt@gmail.com

Abstract*. The successfully atmospheric flights of the supersonic combustion (scramjet) demonstrations were carried out by Russia (Hypersonic Flying Laboratory, HFL scramjet, 1993, 1994 and 1998), by Australia (HyShot, 2002), by USA (X-43 2002, and X-51, from 2010 to 2014), gave a new revival for a new design of the advanced aerospace vehicles using airbreathing propulsion systems. One of the most relevant issues to be dealt with is how to improve the fuel (hydrogen) injection and hydrogen-air mixing and burning efficiency. As well as make the flame stable in supersonic flows and generate thrust for the scramjet vehicle. The ongoing research efforts on fuel injection techniques in the scramjet combustion chamber have been reviewed. The strut injection, single and double cavity-based supersonic combustors, the backward-facing step are being discussed. Finally, a promising fuel injection technique is being discussed. Typical, Mach number at the combustion chamber entrance ranging Mach number from about 2 to 3, and the hydrogen injected in Mach number 1. In general, the residence time of the supersonic hydrogen-air, in the combustion chamber is in the order of milliseconds. Therefore, the fuel-air mixing processes, chemical reaction processes of hydrogen-air combustion (ignition of the combustible mixtures), stabilization of supersonic combustion flames, the multidimensional interaction between combustion and fluid dynamics are the main issues that must be addressed for hypersonic airbreathing propulsion to be integrated into an aerospace vehicle. Consequently, the hydrogen injection and hydrogen-air mixing section design and resulting mixing length are critical aspects of a scramjet combustion chamber design.*

Keywords: *hydrogen-air supersonic combustion, scramjet, supersonic combustion, hypersonic airbreathing propulsion*

1. INTRODUCTION

Besides the supersonic combustion has been studied from the middle of the 1950s (Weber and Mackay, 1958), only, after 60 years the supersonic combustion was demonstrated in atmospheric flight, in 2002, by the Australian scramjet HyShot (Hass et al., 2005), and in 2004, by NASA scramjet X-43 (McClinton, 2006). Therefore, there is no aerospace vehicle flying at hypersonic velocity, higher than 5 times of flight speed of sound, operating supersonic combustion (scramjet) technology.

Currently, the only way to access space is through multi-stage space vehicles, usually non-reusable, which carry the chemical combustion propulsion system onboard (solid and/or liquid propulsion) with high efficiency (about 97-98%), extracting and converting chemical energy into kinetic energy. Approximately 90% of the vehicle's weight at the time of launch is due to the propulsion system (fuel/oxidizer) and structure (Ketsdever et al., 2010), which became too expensive for space access.

However, the success of the demonstration of the supersonic combustion (scramjet) technology in atmospheric flights gave a new revival for a new design of the advanced aerospace vehicles using airbreathing propulsion systems. Russia demonstrated the supersonic combustion, by the Hypersonic Flying Laboratory (HFL) scramjet, in 1993 (Roudakov et al., 1996), in 1994, and in 1998 in collaboration with France government/industry consortium and NASA, respectively (Voland et al., 1999). University of Queensland (Australia) developed the HyShot scramjet with the undergraduate and graduate students, demonstrating the supersonic combustion in 2002 (Hass et al., 2005). In 2004, the supersonic combustion was successfully demonstrated by the NASA X-43s project, at Mach numbers 7 and 10, respectively (McClinton, 2006). Also, US Air Force, from 2010 to 2014, demonstrated supersonic combustion with the missile X-51 scramjet (Rondeau and Jorris, 2013).

Several research centers around the world are developing a new generation of flight demonstrators aiming not only the supersonic combustion technology but also other hypersonic critical technologies (high-temperature materials, flight stability, and control systems for maneuverability during hypersonic flight). Those technologies are needed to design an aerospace vehicle for flight at hypersonic speed in transatmospheric flight (below 86 km of geometric altitude), using an airbreathing propulsion system fully integrated, where the combustion process occurs at supersonic speed, at the combustion chamber (Bowcutt et al., 2012, Steelant et al., 2018).

Currently, the lessons learned from the demonstration of the supersonic combustion in atmospheric flight, one of the most relevant issues to be dealt with is how to improve the fuel (hydrogen) injection and fuel-air mixing and burning efficiency and make the flame stable in supersonic flows, and generating thrust for the scramjet vehicle.

2. SCRAMJET CHARACTERISTICS

2.1 Supersonic combustion terminology

A mixed external and internal compression system is a typical scramjet inlet configuration, where the air capture area is maximum while the scramjet compression section length is minimum (Fig. 1). Scramjet operates according to (opened) Brayton thermodynamic cycle and can be divided into three main sections, (external and internal) compression, combustion chamber (combustor), and (internal and external) expansion. Stations 0, 1, and 3 are the scramjet and cowl leading edges and combustor chamber entrance, respectively. Stations 4, 9, and 10 are the exit of the combustion chamber, and the cowl and the scramjet trailing edge, respectively (Heiser and Pratt, 1994).

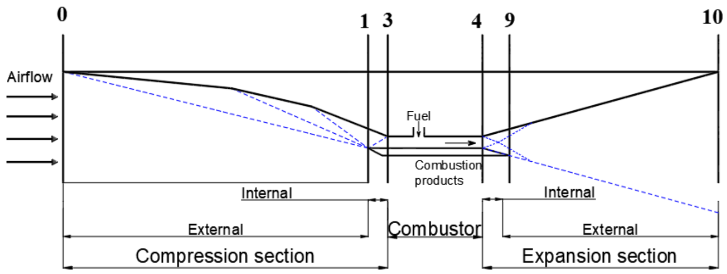

Figure 1. Stations and terminology applied to airframe-integrated scramjet (adapted from Heiser and Pratt, 1994).

In general, the scramjet inlet is designed based on temperature and Mach number at the entrance of the combustion chamber to obtain the maximum total pressure recovery of the external compression section, resulting in the determination of the deflection (turning) angles of the external compression surfaces. Therefore, for mixed compression inlet all incident oblique shock waves, established in the external compression section, converge at the leading edge of the cowl (shock on-lip), and only a single reflected shock wave in the internal compression section converges to the combustion chamber entrance (shock on-corner). Fuel is injected at sonic speed, corresponding to Mach number 1, into supersonic airflow at the combustion chamber, where it is mixed. Fuel-air mixture reaches the equilibrium temperature, burning at fuel spontaneous ignition temperature in supersonic velocity. The combustion products are accelerated at the expansion section. Velocity at the scramjet trailing edge must be higher than the flight velocity providing thrust (Toro et al., 2018a; Araújo et al., 2021).

2.2 Supersonic combustion residence time

Heiser and Pratt (1994) indicated that, for scramjet vehicles flying at a velocity of 3048 m/s (about Mach number 10), the residence time is about 0.002 seconds, considering a scramjet length of 30.5 meters, and the airflow is never far from local equilibrium condition.

A two-dimensional generic hydrogen-powered scramjet was designed at the Universidade Federal do Rio Grande do Norte (UFRN) using an engineering approach to demonstrate supersonic combustion during atmospheric flight. The airflow velocity and temperature at entrance of the combustion chamber, considering perfect gas relation and no viscous effects, were about 1620 m/s and 1010 K (Toro et al., 2018b; 2019c, Carneiro, 2020). Considering that the scramjet combustion chamber was 1 meter long, the residence time was about 0.0006 seconds. Also, considering perfect gas relation and viscous effects, where the displacement thickness was considered, the airflow temperature increased to about 1210 K, and the velocity decreased to about 1450 m/s, corresponding to Mach number 2,15 (Carneiro, 2020). Therefore, for a more realistic case, considering the same combustion chamber length, including the boundary layer effects, the residence time increases to about 0.0007 seconds.

For supersonic combustion, between airflow (Mach number about 2) and hydrogen (Mach number 1), the residence time available in the combustion chamber is in order of milliseconds. Therefore, fuel-air mixing processes, chemical reaction processes of fuel-air combustion, stabilization of supersonic combustion flames, the multidimensional interaction between combustion and fluid dynamics are the main issues that must be addressed for hypersonic airbreathing propulsion to be integrated into an aerospace vehicle. Consequently, the hydrogen injection and hydrogen-air mixing length design and resulting mixing length are critical aspects of a scramjet combustion chamber design.

2.3 Scramjet combustion chamber length

The scramjet combustion chamber length may be divided into the isolator and the hydrogen-air mixture and burning in supersonic velocity (Fig. 2). In general, the isolator length, with a constant area section, is used to reduce the shock train and to uniformize the airflow, coming from the compression section. The constant area right after the hydrogen injection is necessary to mix the fuel up to the equilibrium temperature, and the hydrogen spontaneous ignition. The hydrogen is injected at the sonic speed, with a lower temperature, into the supersonic airflow with a higher temperature. Variable area section is necessary during the burning of hydrogen-air to expand the combustion products and avoid the unstart due to the backpressure.

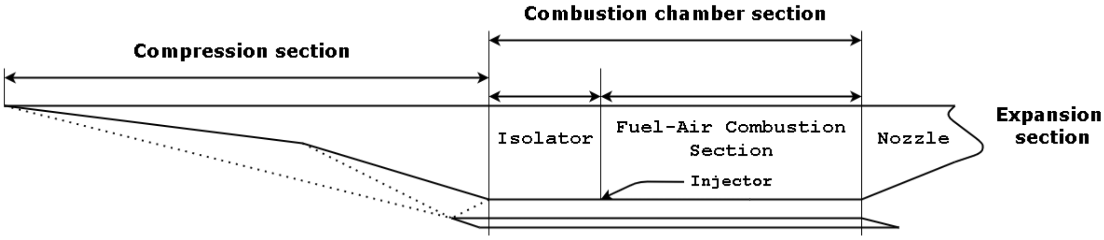

Figure 2. Combuston chamber length dimension.

Heiser and Pratt (1994) commented, for gaseous fuel injection, there are, basically, three main mechanisms: parallel fuel injection within the airflow, normal fuel injection from the walls, and hyper mixer (vortex generator) fuel injectors. However, several modified injection techniques have been tested in different scramjet demonstrators, during the atmospheric flight to demonstrate the supersonic combustion.

2.4 Injection and flame holding techniques

This work discussed and reviewed the fuel injection and flame holding techniques, the strut-based scramjet combustor, the single and double cavity-based supersonic combustors, the backward-facing step, the oblique shock-induced combustion.

Xue et al. (2017) numerically investigated both the flow field and the shock train structure applied to the Rocket-Based Combined-Cycle (RBCC) engine. A hybrid RANS/LES technique, known as Detached Eddy Simulation (DES), was applied to reduce the computation costs, based on the Fluent software package. In the DES approach, the unsteady RANS models, with two-equation realizable $k - \varepsilon$ model and the modified dissipation term, were employed in the boundary layer, while the LES treatment was applied to the separated regions. The LES region was normally associated with the core turbulent region where large unsteady turbulence scales play a dominant role. In this region, the DES models recover LES-like subgrid models. In the near-wall region, the respective RANS models were recovered. To validate the applicability of the turbulent models, a numerical calculation for the DLR scramjet engine with a strut injector (Guerra et al., 1991), similar to the RBCC configuration, was carried out. The operating and boundary conditions for the scramjet engine were consistent with the experiment, and were given by supersonic airflow Mach number 2, temperature 340 K, and pressure 10^5 Pa. For hydrogen fuel jet were Mach number 1, temperature 250 K, and pressure 10^5 Pa.

Athithan et al. (2121) numerically studied the influence of the ramp located at different locations upstream of a strut-based scramjet combustor under-reacting flow conditions. The temperature and supersonic velocity airflow at the entrance of the combustion chamber was, respectively, 300 K and 706 m/s (corresponding to Mach number 2). The hydrogen was injected at 250 K with 1240 m/s (Mach 1). The numerical study was performed using ANSYS Fluent commercial software. A 2D compressible Reynolds Averaged Navier Stokes (RANS) equation with the Shear Stress

Transport (SST) k-ω turbulence model was adopted. The numerical results were validated with the DLR experimental values (Guerra et al., 1991; Waidmann et al., 1994), showing good agreement within the range, indicating that the adopted numerical simulation method can be extended for other investigations. Shock structure, wall pressure distribution, temperature distribution across the combustor, combustion efficiency, and total pressure loss were discussed. More shock-to-shock, shock-to-shear layer, and shock-to-boundary layer interactions were revealed by the numerical shadowgraph images for the double ramp and strut injector scramjet combustors compared to the DLR scramjet model. On the other hand, more vortex regions were found in the combustor with ramps.

Huang et al. (2010) numerically simulated the flow field of the hydrogen fueled scramjet combustor with a cavity flame holder (single cavity-based supersonic combustor) under cold flow and fuel ignition conditions. The 2D coupled implicit Navier-Stokes equations with the standard k-ε turbulence model and the finite-rate and eddy-dissipation reaction model were adopted. The incoming airflow condition was Mach number of 4.5, static temperature of 1300 K and static pressure of 101,325 Pa. The hydrogen was injected into the supersonic airflow at sonic velocity, the static temperature of 1000 K, and static pressure of 506,625 Pa. A typical two-dimensional scramjet combustor was built with 0.666 m long, with an isolator length of 0.22 m, and a height of 0.032 m. The combustor, with a height of 0.0384 m, was located 0.22 m from the entrance of the isolator. The divergent section, located at 0.316 m from the entrance of the isolator, with a turning angle of 1.7°, on both sides, on the top and bottom walls, of the combustor. A step, on the top wall of the combustor, was built with a height of 0.0032 m, located downstream from the entrance of the combustor. Hydrogen was injected from the bottom wall at 0.2072 m (from the entrance of the isolator), which is 0.0128 m upstream of the entrance to the combustor, and the width of the injection slot is 0.001 m. There exists a cavity flame holder on the bottom wall of the combustor with the upstream depth of 0.0032 m, the ratio of the downstream depth to the upstream depth of 1, the ratio of the length to the upstream depth of 5, and the swept angle θ of 45°. The distance between the injection slot and the leading-edge of the cavity flame holder was 0.005 m. The numerical results showed the static pressure distribution along the top and bottom walls for the case under the condition of ignition was much higher than that for the cold flow case condition, with three clear pressure rises on the top and bottom walls of the scramjet combustor. The eddy generated in the cavity acted as a flame holder in the combustor, which can prolong the residence time of the hydrogen-air mixture in supersonic velocity.

Ombrello et al. (2015) experimentally investigated the influence of the incoming airflow distortion on ignition in a single cavity-based supersonic combustor. A supersonic airflow with Mach number 3, the total temperature of 616 K, and the total pressure of 1379 kPa approached the wedge with a turning angle of 8° mounted on the opposite side of the single cavity. The wedge could be moved in the streamwise direction to produce a shock that impinged at different locations relative to the cavity. Three positions were chosen to impinge the incident shock far upstream of the cavity, just upstream of the cavity on the fuel injector plume, and on the cavity shear layer. An incident oblique shock wave was established impinging upstream and over a cavity. Fuel (C_2H_4) was injected into the cavity from eleven holes (disposed along the flow path 0.152 m wide) in the cavity closeout ramp at various flow rates. Spark discharge and a pulse detonator were used to provide two different forms of energy deposition, needed to ignite the fuel-air mixtures. The sensitivity of each ignition device was determined by varying the cavity fueling rates. Since the cavity fueling was symmetric with respect to the spanwise distribution the mixture was assumed to be the same at the location of both ignition devices. The spark discharge provided an ignition strongly dependent on the flow field and therefore much more sensitive to the effects of distortion. The pulse detonator generated a strong disruption of the flow field and generally ignited the fuel in the single cavity over a wide range of fueling flow rates. Therefore, the pulse detonator allowed ignition success across a wide range of fueling conditions. This was more remarkable than for the wedge in the downstream position. Also, the shock (from the wedge) intersection with the cavity shear layer generated a large cavity volume. Finally, the investigation emphasized a careful placement of an energy deposition device, especially the spark discharge, to provide a robust ignition solution across a range of fueling and inflow distortion conditions.

Wang et al. (2015) numerically investigated a hydrogen-fueled scramjet combustor with a dual cavity, applying the Reynolds-Averaged Navier–Stokes (RANS) model was used for near-wall treatment. The supersonic airflow was Mach number 2.52, with static temperature and static pressure of 1486 K and 1.6 MPa, respectively. Hydrogen was injected at the speed of sound, with 300 K and 0.8 MPa. The isolator was 180 mm long, and the width and height were 50 mm and 40 mm, respectively. Two cavities with depth D 8 mm, length-to-depth ratio L/D 7, and aft angle of 45°, were mounted on the top wall and bottom wall, respectively. Gaseous hydrogen was injected sonically from a 2 mm diameter injector located 10 mm upstream of each cavity. The observed flow and combustion structures were reasonably well captured and explained by the numerical simulation. The intersection of the bow shock waves and the concentrated heat release generated a high-pressure region between the cavities, which induces great pressure gradients as well as in the transverse direction, pushing the fuel jets towards the combustor walls. The combustion was stabilized around the cavities, suggesting that the cavities acted as flame holders. Compared with the single-cavity results (Wang et al., 2013) in the same combustor, it was found that the flame length was greatly reduced by the dual cavity flame holders.

Choubey et al. (2018) numerically investigated the influence of four different boundary conditions of inlet air and four of hydrogen fuel on the flow field of parallel double-cavity scramjet combustor, by employing seven-step reaction mechanisms of hydrogen-air coupled with 2D compressible RANS equation as well as finite rate/eddy dissipation model and two-equation k-ε turbulence model. The numerical simulations were validated with experimental schlieren photograph (Wang et al., 2015; Yang et al., 2015), experimental flame luminosity image (Wang et al., 2015), and the experimental wall pressure distribution curve (Yang et al., 2015), using the same boundary conditions from them. Eight different cases were studied. The first four cases were studied for the variation of hydrogen fuel inlet boundary conditions. The variation of air inlet boundary conditions had a strong influence on the flow field of parallel double-cavity scramjet combustor. The formation of a high-pressure region around the cavities for air inlet boundary (cases 3 and 4) condition helped to push the air to mix with an adequate amount of hydrogen fuel for stable combustion. The intersection between the bow shock waves and the concentrated heat release was the primary reason for the formation of the high-pressure region around the cavities.

Choi et al. (2005) numerically investigated a two-dimensional scramjet with a backward-facing step, combustor configuration with transverse fuel injection and a cavity flame holder. Considering the flight Mach number of 5 to 6 at an altitude of 20 km, the incoming airflow at the combustion chamber was about Mach number 3 at 600 K and 0.1 MPa. Gaseous hydrogen was injected transversely through a chocked slot of 1 mm in width to the combustor. The hydrogen temperature was 151 K, and the injector exit pressures were 0.5, 1.0, and 1.5 MPa, and the overall equivalence ratios of the overall fuel/air mixture ranges were 0.167, 0.33, and 0.5. The Richtmyer–Meshkov shear layer instability or cavity-driven instability, those were unavoidable in the supersonic combustor, triggers the injector flow instability and the disturbing injector flow greatly enhanced the hydrogen-air mixing and combustion.

Sainte-Rose et al. (2009) numerically investigated backward-facing step reactive flow by using the in-house CEDRE code, based on hybrid Reynolds Averaged Navier Stokes–Large Eddy, which was a trend of common use in aerodynamics but has seldom been employed to simulate reactive flows. Such methods, like the Delayed Detached Eddy Simulation (DDES) have been created to treat near-wall flows with a RANS approach while switching to LES in the separated flow region. It was indeed an affordable solution to simulate complex and unsteady compressible flows and to have access to accurate skin friction and wall thermal fluxes. In order to validate this technique in combustion, a simple and well-documented backward-facing step combustor was chosen. To account for the turbulent combustion a dynamic thickened flame was used. The results obtained showed a good agreement with the experimental database and were of the same quality as LES in the separated region for both inert and reactive flow.

3. ANALYSIS OF THE HYDROGEN INJECTION, AND HYDROGEN-AIR COMBUSTION

3.1 Analysis of the fuel injection techniques

A strut-based scramjet combustor is possible to perform a numerical investigation. However, the support needed to flow the fuel from the fuel tank to be injected into the supersonic airflow becomes not adequate to install in the real scramjet combustor, due to the high perturbation into the airflow, including may block the hydrogen-air flow (Fig. 3a).

The single or double cavity-based supersonic combustors are a very adequate fuel injection technique. The single cavity-based fuel injection (Fig. 3b) was tested in a joint Russian/French dual-mode scramjet flight-test, at approximately Mach number 5.6 (Roudakov et al., 1996; Huang et al., 2010).

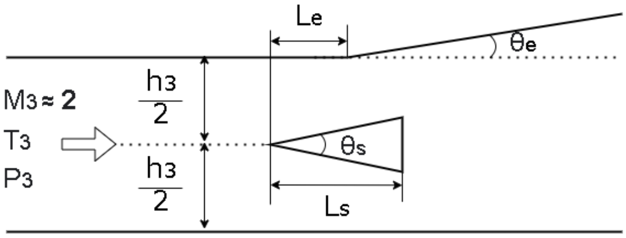

Figure 3a. Strut-based fuel injection.

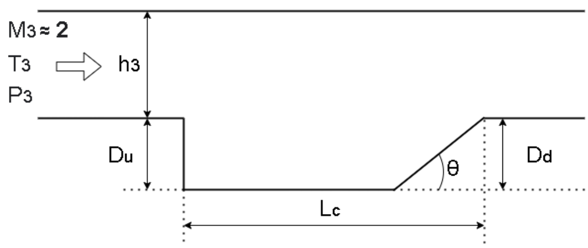

Figure 3b. Single cavity-based fuel injection.

The two opposed cavity-based flame holders (Fig. 4a) were used in the HIFiRE-2 (Hypersonic International Flight Research Experimentation) configuration, which enabled combustion stabilization and dual-mode transition. Two sets of fuel injectors, corresponding to the primary and secondary stages upstream and downstream of the cavity, respectively, supplied a gaseous mixture made up of 64% ethylene (C_2H_4) and 36% methane (CH_4) content by volume. In 2012, the successfully HIFiRE-2 atmospheric flight burning the hydrocarbon, from Mach numbers 6 to 8, where the scramjet was mounted on a sounding rocket (Jackson et al., 2014). The backward-facing step (Fig. 4b) was, somehow, a variation of the single or double cavity-based supersonic combustion.

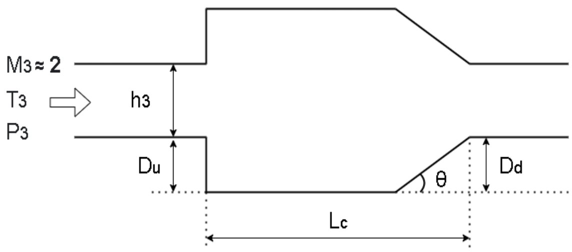

Figure 4a. Double cavity-based fuel injection.

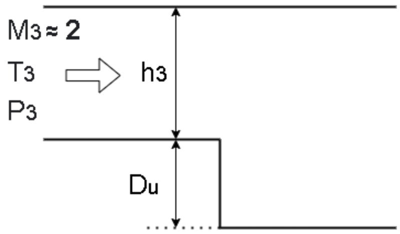

Figure 4b. Backward-facing stepfuel injection.

3.2 Previous works at UFRN

Hypersonic airbreathing propulsion (scramjet) is not capable of flight beyond the about altitude of 60-70 km, because there is no air enough to be captured by the scramjet inlet (Heiser and Pratt, 1994). Due to the propulsion system is carried (solid or liquid fuel) onboard, the rocket engines are the only propulsion system capable to reach Earth orbit. Although the high efficiency of 97-98%, about 90% of the weight of the spacecraft at the time of launch, is due to the propulsion system (and the respective structure) be part of the vehicle, only 5% to 10% is due to the payload, usually satellites, equipment to ISS and astronauts (Ketsdever, 2010). Rocket engines have a constant low specific impulse, from launch to 86 km of altitude, compared to airbreathing propulsion (Fry, 2004). The scramjet concept results from the observation that a good part of the flight trajectory of an aerospace vehicle towards Earth orbits occurs in the atmosphere rich in oxygen (below 86 km of geometric altitude), which is an oxidizer by nature. Therefore, a propulsion stage that could make use of this oxygen presents in the Earth's atmosphere, to pass through the mesosphere and stratosphere up to about 60-70 km, instead of carrying the oxygen onboard would make it more efficient than the conventional propulsion system based on solid propulsion and/or liquid propulsion (Heiser and Pratt, 1994). Scramjet requires a minimum of at least Mach 4 to operate. The upper limit on the operational Mach number is not accurately known, but is estimated to be between Mach number close to 8 for hydrocarbon fuels and could be up to Mach 25 for hydrogen fuel (Fry, 2004).

The required static temperature (T_3) and Mach number (M_3) of the airflow, at the entrance of the combustion chamber (Fig. 1) conditions needed to burn the fuel (hydrogen) spontaneously, at the hydrogen ignition temperature of 845.45 K, in supersonic velocity, may be estimated applying the zeroth and first Laws of Thermodynamics, and the energy conservation law, considering calorically perfect gas (no real gas effects), inviscid flow (no boundary layer effects) (Araújo et al., 2021).

Carneiro (2020) presented the conceptual design of the supersonic combustion demonstrator, using two approaches. The first was carried out the study considering calorically perfect gas and no boundary layer formation. In the second approach, viscous effects were considered, and the thickness of the boundary layer was incorporated, using the Chapman-Rubesin model. In both approaches, the aerospace vehicle was in atmospheric flight at a speed of 2051 m/s (corresponding to Mach 6.8) at a geometric altitude of 30 km, wherein the combustion chamber was burning hydrogen fuel with airflow at supersonic speed. Theories of oblique plane shockwave, of addition of heat in one-dimensional (Rayleigh) flow, of (Prandtl-Meyer) expansion wave and of the area ratio, were applied to the design of the supersonic combustion demonstrator in a given altitude and hypersonic speed (Fig. 5).

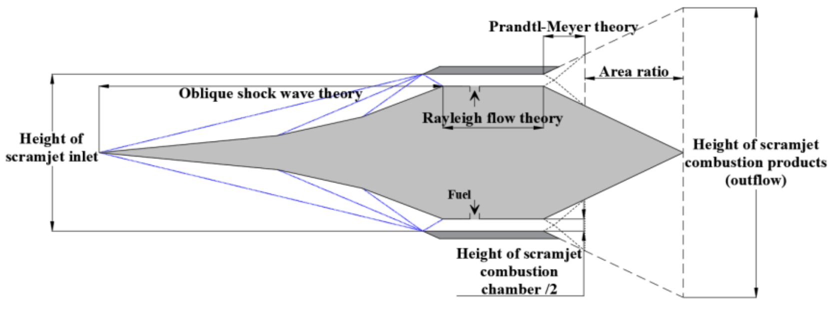

Figure 5. Theories used from leading-to-trailing edges of the scramjet (Carneiro, 2020).

Applying the design conditions developed by Carneiro (2020), Bezerra (2020) numerically investigated the variation of the flight speed of the scramjet vehicle, burning hydrogen-air mixture in supersonic speed. Three flight conditions of the supersonic combustion demonstrator were evaluated, at an altitude of 30 km: flight speeds corresponding to Mach numbers 6.4 (below design speed), 6.8 (design speed), and 7.2 (above design speed), considering single and double transverse hydrogen injection (Figs. 6a; 6b). Both analytical and numerical investigations were in good compliance with the hydrogen-air combustion temperature.

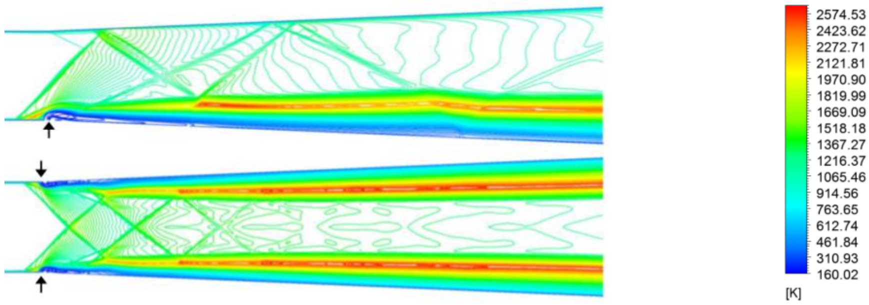

Figure 6. (a) Temperature contours (Bezerra, 2020).

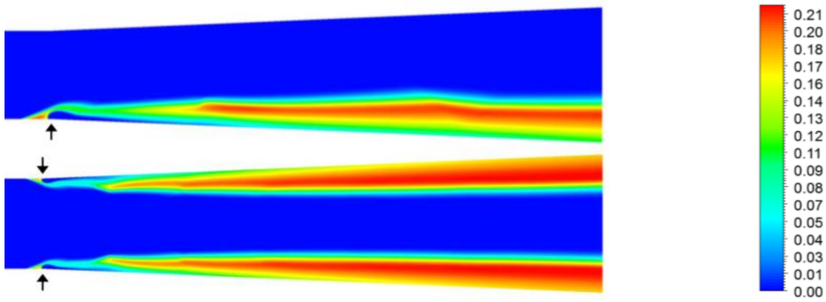

Figure 6. (b) H_2O mass fraction contours (Bezerra, 2020).

Considering the on-design flight Mach number 6.8, hydrogen was transversely injected at 300 K and sonic speed of 1318 m/s, corresponding to Mach number 1, into airflow with temperature and velocity of 1210 K and of 1494 m/s, corresponding to Mach number 2.14. The temperature (Fig. 6a), velocity (Mach number) of the combustion products were ranging from 1190 K and 1400 m/s (M 1.97) to 1413 K, and 1370 m/s (Mach 1.73), for single and double transverse hydrogen injection. The temperature (Fig. 56) and H_2O (Fig. 6b) mass fraction contours showed the hydrogen-air was burning close to the wall surface. For double hydrogen injection, only 22% of hydrogen was burning. The core of the hydrogen-air had no high temperature enough to combustion (Bezerra, 2020).

4. PROPOSAL FOR HYDROGEN FUEL INJECTION

The transverse under expanded hydrogen injection in a supersonic airstream established, just upstream of the hydrogen injected, a detached normal shock wave, and downstream of the hydrogen injected was established a bluff-body wake region, which a recirculation flow in the wake acted as a flame holding. Also, a bow (oblique) shock wave was established, generating a core where the high temperature of the hydrogen-air mixing was enough to burn the hydrogen. In the bluff-body wake region, hydrogen-air combustion was established. However, the flame was no holding (Fig. 6) and was extinguished, but due to the shock train (Fig. 6a) the mixed hydrogen-air temperature increased and the flame became stable (Fig. 6b). The temperature and H_2O mass fraction contours indicated the mixed hydrogen-air burned near the wall surface (Fig. 6), and at the mainstream core, there was no hydrogen-air, consequently no combustion.

One dimensional (Rayleigh) flow with heat addition, with no fuel mass addition, was preliminary used to analytically simulation the fuel-air combustion process. Considering calorically perfect gas, with no boundary layer effects, and constant area combustor, closed-form analytical equations were presented (Anderson, 2003), and can be found in any gas dynamic textbook. Knowing the conditions at the combustion chamber entrance, and the heat added per unit mass, the effect of heat addition was directly change the total temperature (energy) of the airflow (with no fuel mass addition).

As mentioned early, Carneiro (2020) presented the design of an aerospace vehicle using an airbreathing propulsion system to demonstrate the supersonic combustion (scramjet) technology, during the hypersonic flight velocity of 2050 m/s (corresponding to Mach number 6.8) at a 30 km geometric altitude. The combustion process, in the scramjet combustion chamber, was based on one-dimensional compressible (Rayleigh) flow with heat addition (Tab. 1), and no mass addition. The scramjet inlet capture airflow was deflected to the combustion chamber entrance by, first, using the oblique shock wave theory, considering calorically perfect gas and no viscous effects. Following, the viscous effects were included based on the displacement thickness, which enabled to consider the new slope of the wall ramp, as described by the Chapman-Rubesin theory (Chapman et al., 1958).

The viscous effects increased the static temperature and decreased the airflow velocity (Mach number), at the entrance of the combustion chamber from 1090 K, 1574 m/s (Mach 2.37) to 1210 K, 1494 m/s (Mach 2.14), respectively. Considering heat addition on one-dimensional (Rayleigh) flow and viscous effects, the velocity, after the combustion, was supersonic (Mach number equal 1.2), the hydrogen-air spontaneous combustion temperature was about 2300 K, the scramjet trailing-edge velocity was about 2350 m/s, higher than the flight velocity of 2050 m/s, generating a positive uninstalled thrust of 319,22 N (Carneiro, 2020).

Table 1. Thermodynamic properties of the airflow and the combustion products (Carneiro, 2020).

			no viscous effects			viscous effects		
		Flight conditions	Combustor entrance	Combustor exit (Power-off)	Combustor exit (Power-on)	Combustor entrance	Combustor exit (Power-off)	Combustor exit (Power-on)
M		6.8	2.372	2.372	1.2	2.143	2.143	1.2
p	Pa	1197	130594.20	130594.20	384404.96	167.608.12	167.608.12	413.035.56
T	K	226.5	1092.12	1092.12	2421.57	1209.11	1209.11	2301.31
ρ	kg/m^3	0.01841	0.41657	0.41657	0.55301	0.48289	0.48289	0.62523
a	m/s	301.7	662.4	662.4	986.4	641.99	697.03	961.60
u	m/s	2051.7	1571.4	1571.4	1183.68	1494.02	1494.02	1154.91
T_{total}	K	2321.17	2321.17	2321.17	3118.98	2321.41	2321.41	2964

Numerical simulation was carried out, at the same combustion chamber conditions as Carneiro (2020), considering the calorically perfect gas, with viscous effects based on the displacement thickness (Chapman et al., 1958), with power-on, means, at the hydrogen injection location, a uniform and constant heat addition across the height of the combustion chamber was assumed as 950,000 J/s. The pressure increased from 107,718 Pa to 148,216 Pa (Fig. 7).

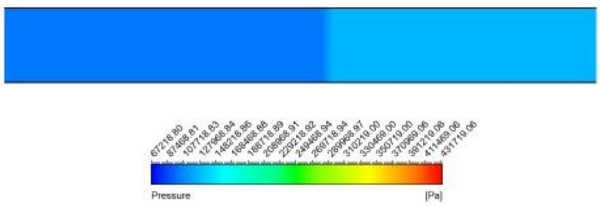

Figure 7. Pressure contour for constant and uniform heat addition across the height of the combustion chamber.

However, when the same heat was added in one station close to the wall surface, which was a better simulation of transverse hydrogen injection, the pressure and temperature increased to about 320,000 Pa (Fig. 8a) and 2400 K (Fig. 8b), respectively, as expected. Also, in this case, a shock train was established at the heat added, similar to the shock train established from real transverse hydrogen injection (Fig. 3a).

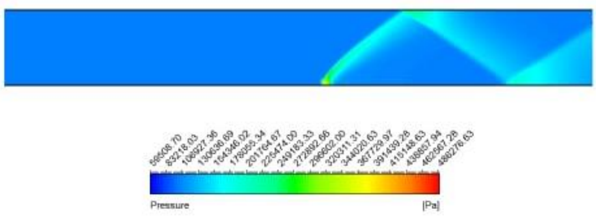

Figure 8. (a) Pressure contours for heat added close to the wall surface.

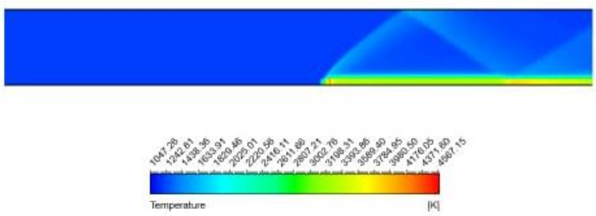

Figure 8. (b) Temperature contours for heat added close to the wall surface.

The objective of this paper was to preliminary explore the influence of the shock impinging on transverse, under expanded, hydrogen injection (Fig. 9) into high enthalpy airflow conditions (u_3 = 1300 m/s, T_3 =1100 K, M_3= 2, simulating the condition of flight velocity 2050 m/s (Mach number 6.8 at 30 km altitude), same airflow conditions of the numerical investigation from Bezerra (2020) and of the analytical approach from Carneiro (2020).

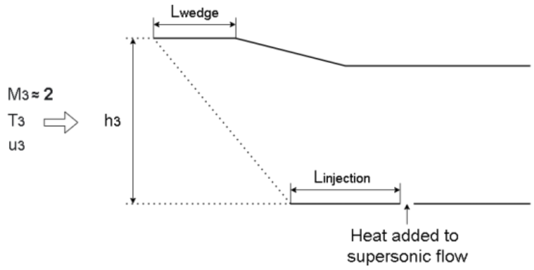

Figure 9. Shock impinging on transverse hydrogen injection.

When a wedge was installed downstream and at the opposite side surface of the heat addition position (Fig. 9), an incident oblique shock wave was established and converged right away (Fig. 10a) on the heat added (simulating the transverse hydrogen injection), increasing the pressure (Fig. 10a) and temperature (Fig. 10b) at the heat addition (simulating the hydrogen-air combustion).

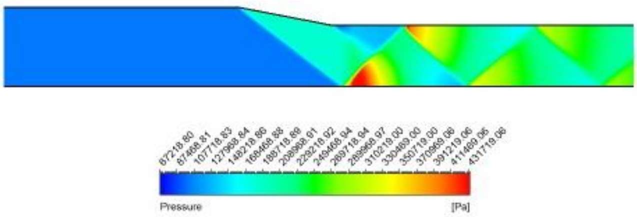

Figure 10. (a) Pressure contours of an oblique shock impinging ahead of the heat added close to the wall surface.

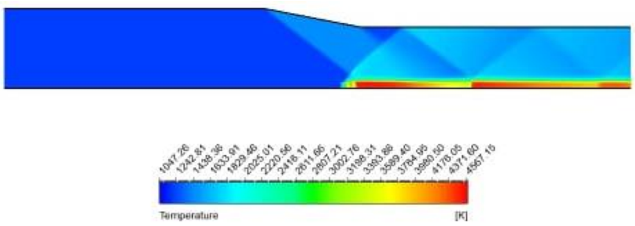

Figure 10. (b) Temperature contours of an oblique shock impinging ahead of the heat added close to the wall surface.

A combination of both, the expansion wave from the wedge, the reflected shock originated from the incident shock, and the shock wave due to transverse hydrogen may provide better fuel-air mixing and combustion. Naturally, the viscous effects based on the displacement thickness did not establish a boundary layer; thereby there was no shock-wave boundary-layer interaction, no boundary-layer separation, no boundary-layer reattachment. However, the oblique shock wave impinging on the transverse fuel injection will deepen the mixing of the hydrogen-air, increasing the temperature higher than the hydrogen ignition temperature, and probably higher combustion efficiency.

5. CONCLUSION

To improve the fuel (hydrogen) injection, hydrogen-air mixing, burning efficiency, and make the flame stable in supersonic flows, which influence the thrust for the scramjet vehicle, a brief review of the principal fuel injection techniques, which some of them were used in the supersonic combustion demonstrators, like the joint Russian/French dual-mode scramjet flight-test and in the HIFiRE 2 (Hypersonic International Flight Research Experimentation), under development by Air Force Research Laboratory (AFRL), in partnership with the Australian Defense Science and Technology Organization (DSTO).

Typical, supersonic speeds, corresponding to supersonic Mach number, at the combustion chamber entrance ranging from about 2 to 3, and the hydrogen injected in sound speed, Mach number 1, at a lower temperature than the corresponding ignition temperature. In general, the residence time of the supersonic hydrogen-air, in the combustion chamber is in the order of milliseconds. Therefore, the fuel-air mixing processes, chemical reaction processes of hydrogen-air combustion (ignition of the combustible mixtures), stabilization of supersonic combustion flames, the multidimensional interaction between combustion and fluid dynamics are the main issues that must be addressed for hypersonic airbreathing propulsion to be integrated into an aerospace vehicle. Consequently, the hydrogen injection and hydrogen-air mixing section design and resulting mixing length is a critical aspect for the scramjet combustion chamber.

A strut-based injection technique, is, in general, numerically investigated, without the support needed to flow the fuel from the fuel tank to be injected into the supersonic airflow, showed not adequate to install in the real scramjet combustor, due to the high perturbation into the airflow, including may block the hydrogen-air flow. While the backward-facing step injection technique is a variation of the single or double cavity-based supersonic combustion.

It was decided in this preliminary work, explore the influence of the shock impinging on transverse, under expanded, hydrogen injection into high enthalpy airflow conditions (u_3 = 1300 m/s, T_3 =1100 K, M_3= 2, simulating the condition of flight velocity 2050 m/s (Mach number 6.8 at 30 km altitude), same airflow conditions analytically and numerically investigated at the Graduate Program in Mechanical Engineering, at the Universidade Federal do Rio Grande do Norte (UFRN).

The incident oblique shock wave impinging on the transverse fuel injection right away on the heat added (simulating the transverse hydrogen injection), increased the pressure and temperature at the heat addition (simulating the hydrogen-air combustion). The viscous effects based on the displacement thickness did not establish a boundary layer; thereby no shock-wave boundary-layer interaction, no boundary-layer separation, no boundary-layer reattachment. However, the oblique shock wave impinging on the transverse fuel injection provided a simultaneous combination of the expansion wave from the wedge, the reflected shock originated from the incident shock, and the bow shock due to transverse hydrogen, which, as shown, intensifying the mixing of the hydrogen-air, increasing the temperature higher than the hydrogen ignition temperature, and may provide an insight to the hydrogen-air combustion flame holding capability due to better hydrogen-air mixing and combustion.

6. ACKNOWLEDGEMENTS

This study was financed in part by the Coordenação de Aperfeiçoamento de Pessoal de Nível Superior - Brasil (CAPES) - Finance Code 001. Also, the authors would like to thank CAPES the support given by the Project PROCAD-DEFESA 88887.626266/2021-00. The authors would like to thank the Instituto Tecnológico de Aeronáutica (ITA) to support granted to carry out research on hypersonic airbreathing propulsion. Also, the third author would like to thanks the Instituto de Estudos Avançados (IEAv). The last two authors extend their gratitude to the Universidade Federal do Rio Grande do Norte (UFRN). Finally, the first author would like to thank Prosimulador Tecnologia de Trânsito and Grupo Tecnowise for the enthusiasm dedicated to Hypersonics.

7. REFERENCES

Athithan, A.A.; Jeyakumar, S.; Sczygiol, N.; Urbanski, M.; Hariharasudan, A. (2021). The Combustion Characteristics of Double Ramps in a Strut-Based Scramjet Combustor. Energies 2021, 14, 831. https://doi.org/10.3390/en14040831

Araújo, P. P. B., Pereira, M. V. S., Marinho, G. S., Martos, J. F. A., Toro, P. G. P. (2021). Optimization of scramjet inlet based on temperature and Mach number of supersonic combustion. Aerospace Science and Technology.Volume 116. September 2021. 106864. https://doi.org/10.1016/j.ast.2021.106864.

Bezerra, I. S. A. (2020). Análise numérica da influência da velocidade na combustão supersônica em um demonstrador scramjet. Master Dissertation (in Portuguese). Graduate Program in Mechanical Engineering (UFRN).

Bowcutt, K.; Pall, A.; Dolvin, D.; Smart, M. (2012). HIFIRE: an International Collaboration to Advance the Science and Technology of Hypersonic Flight. 28th International Congress of the Aeronautical Sciences (ICAS).

Carneiro, R. (2020). Estudo analítico de um demonstrador da tecnologia da combustão supersônica. Master Dissertation (in Portuguese). Graduate Program in Mechanical Engineering (UFRN).

Chapman D.R., Kuehn D.M. and Larson H.K., (1958). Investigation of Separated Flows in Supersonic and Subsonic Streams with Emphasis on the Effect of Transition. NACA report No. 1356, 1958.

Choi, J.-Y., Ma, F., Yang, V. (2005). Combustion oscillations in a scramjet engine combustor with transverse fuel injection. Proceedings of the Combustion Institute, 30(2), 2851–2858. doi:10.1016/j.proci.2004.08.250

Choubey, G.and Pandey, K. M. (2018). Effect of variation of inlet boundary conditions on the combustion flow-field of a typical double cavity scramjet combustor. International Journal of Hydrogen Energy, 43(16), 8139–8151. doi:10.1016/j.ijhydene.2018.03.062

Guerra, R., Waidmann, W., Laible, C. (1991). An experimental investigation of the combustion of a hydrogen jet injected parallel in a supersonic air stream. 3rd International Aerospace Planes Conference. doi:10.2514/6.1991-5102

Hass, N., Smart, M., &Paull, A. (2005). Flight Data Analysis of the HyShot 2. AIAA/CIRA 13th International Space Planes and Hypersonics Systems and Technologies Conference. doi:10.2514/6.2005-3354

Heiser, W.; Pratt, D.; Daley, D.; Mehta, U. (1994) Hypersonic Airbreathing Propulsion. AIAA Education Series. ISBN 1-56347-035-7. https://doi.org/10.2514/4.470356

Huang, W., Luo, S., Pourkashanian, M., Ma, L., Ingham, D. B., Liu, J., Wang, Z. (2010). Numerical simulations of a typical hydrogen fueled scramjet combustor with a cavity flame holder. Proceedings of the World Congress on Engineering 2010 Vol II. London, U.K.

Jackson, K. R., Gruber, M. R., and Buccellato, S. (2014). Mach 6–8+ Hydrocarbon-Fueled Scramjet Flight Experiment: The HIFiRE Flight 2 Project. Journal of Propulsion and Power, 31(1), 36–53. doi:10.2514/1.b35350

Ketsdever, A. D., Young, M. P., Mossman, J. B., and Pancotti, A. P. (2010). Overview of Advanced Concepts for Space Access. Journal of Spacecraft and Rockets, 47(2), 238–250. doi:10.2514/1.46148

McClinton, C. R. (2006). X-43–Scramjet Power Breaks the Hypersonic Barrier Dryden Lectureship in Research for 2006. 44th AIAA Aerospace Sciences Meeting and Exhibit. AIAA 2006-0001

Ombrello, T., Peltier, S., Carter, C. (2015). Effects of Inlet Distortion on Cavity Ignition in Supersonic Flow. AIAA SciTech. 53rd AIAA Aerospace Sciences Meeting. AIAA 2015-0882.

Rondeau, C. M.; Jorris, T. R. (2013). X-51A Scramjet Demonstrator Program: Waverider Ground and Flight Test. SFTE 44th International / SETP Southwest Flight Test Symposium.

Roudakov, A. S., Semenov, V., Kopchenov, V., Hicks, J. W. (1996). Future Flight Test Plans of an Axisymmetric Hydrogen-Fueled Scramjet Engine on the Hypersonic Flying Laboratory. 7th International Space Planes and Hypersonics Systems & Technology Conference. AIAA 1996-4572.

Sainte-Rose, B., Bertier, N., Deck, S., Dupoirieux, F. (2009). A DES method applied to a Backward Facing Step reactive flow. Comptes Rendus Mécanique, 337(6-7), 340–351. doi:10.1016/j.crme.2009.06.017

Steelant, J.; Villace, V.; Kallenbach, A.; Wagner, A.; Andro, J.-Y.; Benedetto, S. di; Saracoglu, B.; Chernyshev, S. L.; Gubanov, A.A.; Talyzin, V. A.; Voevodenko, N. V.; Kukshinov, N. V.; Prokhorov, A. N.; Grigoriev, N. V.; Neely, A. J.; Verstraete, D.; Buttsworth, D. (2018). Flight Testing Designs in HEXAFLY-INT for High-Speed Transportation. International Conference on High-Speed Vehicle Science & Technology (HiSST). Moscow, Russia.

Toro, P. G. P., Oliveira, G. I. A., Carvalho, L. A. S., Santos, L. M. C., Farias, R. L. A., Carneiro, R., Marinho, G. S., Borba, G. L., Rego, I. S. (2018a). Equilibrium and perfect gas air behavior at the stagnation region. ENCIT-2018-0817.

Toro, P. G. P., Carneiro, R., Silva Araújo, J. W., Marinho, G. S., Borba, G. L., Martos, J. F. A., Rego, I. S. (2018b). Design and analysis of a generic scramjet air inlet. ENCIT-2018-0751.

Toro, P. G. P., Carneiro, R., Silva Araújo, J. W., Marinho, G. S., Borba, G. L., Martos, J. F. A., Rego, I. S. (2018c). Design of the generic scramjet combustion chamber. ENCIT-2018-0056.

Voland, R., Auslender, A., Smart, M., Roudakov, A., Semenov, V., & Kopchenov, V. (1999). CIAM/NASA Mach 6.5 scramjet flight and ground test. 9th International Space Planes and Hypersonic Systems and Technologies Conference. doi:10.2514/6.1999-4848

Xue, R., Wei, X., He, G., Hu, C., & Tang, X. (2017). Effect of parallel-jet addition on the shock train characteristics in a central-strut isolator by detached eddy simulation. International Journal of Heat and Mass Transfer, 114, 1159–1168. doi:10.1016/j.ijheatmasstransfer.2017.06.074

Yang, Y., Wang, Z., Sun, M., Wang, H., and Li, L. (2015). Numerical and experimental study on flame structure characteristics in a supersonic combustor with dual-cavity. Acta Astronautica, 117, 376–389. doi:10.1016/j.actaastro.2015.09.005

Waidmann, W.; Alff, F.; Brummund, U.; Bohm, M.; Clauss, W.; Oschwald, M. (1994). Experimental Investigation of the Combustion Process in a Supersonic Combustion Ramjet (Scramjet); DGLR Jahrbuch: Erlangen, Germany, 1994; pp. 629–638.

Wang, H., Wang, Z., Sun, M., & Qin, N. (2013). Combustion characteristics in a supersonic combustor with hydrogen injection upstream of cavity flame holder. Proceedings of the Combustion Institute, 34(2), 2073–2082. doi:10.1016/j.proci.2012.06.049

Wang, H., Wang, Z., Sun, M., Qin, N. (2015). Large eddy simulation of a hydrogen-fueled scramjet combustor with dual cavity. Acta Astronautica 108 (2015) 119–128

Weber, R. J. e Mackay, J. (1958) An Analysis of Ramjet Engines Using Supersonic Combustion. NACA TN 4386.

8. RESPONSIBILITY NOTICE

The Authors are the only responsible for the printed material included in this scientific article.

The Authors declare that there is no conflict of interest regarding the publication of this scientific article.