Design of Supersonic and Hybrid engine based Advanced Rocket (SHAR)

Naresh Relangi1*, Lakshmi Narayana Phaneendra Peri2, Caio Henrique Franco Levi Domingos3, Amalia Fossella4, Júlia Meria Leite Henriques5, Antonella Ingenito6

1,2,3,4,5,6 School of Aerospace Engineering, University of Rome “La Sapienza”, 00138 Rome, Italy

Abstract. The paper deals with the design of a two-stage to orbit rocket launcher loaded with a solid rocket booster, scramjet, and hybrid rocket for delivering a 100kg payload in 200 km circular orbit. The possibility of implementing a cavity-based axisymmetric circular combustor in a scramjet is proposed. Computational analysis on various injector locations in a circular combustor and their validation with the test bench results were performed. The utilisation of a hybrid rocket in the final stage of the launcher to deliver the payload is discussed and the performance characteristics of the circular scramjet combustor and the hybrid rocket are shown. The overall mission proposed based on the sustainable and reusable characteristics.

1. Introduction:
RBCC engines are an integration of usual rockets and air-breathing systems, obtaining the best performance characteristics (higher thrust to weight ratio and the high specific impulse) [1] while also allowing easier access to the low earth orbit (LEO). The increase of the net specific impulse of the trans-atmospheric propulsion system can reduce the required propellant mass and thus increase the mass available for the structure & payload [2, 3].

With the recent technological advancements employed in the scramjets and the proven atmospheric test flights [4, 5], hypersonic air-breathing engines are more likely to be utilized to transport the payloads into the upper atmospheres. The advantage of burning the available atmospheric oxygen results in the reduction of the overall mass of the spacecraft, which in turn increases the performance characteristics of the vehicle [6]. In addition to these factors, the reusability of the vehicle for delivering a satellite to orbit drives the cost of launching down. Despite the current challenges on aerodynamic heating, control issues, the probability of anticipating them in future space launch systems are likely to be encouraging. A couple of vehicles of such successful combined-cycle systems are the NASP [7] and the SÄNGER vehicles [8] which do not relay completely on the ramjet and scramjet but also the turbojet. But due to the involved complexities, this kind of concept usage is limited. However, the scope for the further research on utilization of turbojets during the launch phase is encouraging.
The role of the combustor in the scramjet is significant. The major concerns are related to the complex fuel-air mixing phenomena inside the combustor as the residence time is in the order of milliseconds. Earlier researches were mainly focused on the injection techniques to improve the mixing characteristics [9,10]. High injection angles tend to a strong bow shock, enhancing the vorticity and mixing inside the flow, causing the total pressure loss on the other hand. Lowering the injection angles (30° and 60° rather than 90°) in a supersonic flow could reduce the effect of the bow shock which results in the reduction of stagnation pressure loss and maximizes the net thrust. Studies [11, 12] show that significant improvement in the combustion efficiency, flame holding, and flame stabilization can be observed by bringing in the concept of cavity after the injector since the injected fuel jet separates the boundary layer and increases the recirculation zones. Besides, with this kind of technique, jet penetration depth would be higher and fuel-air residence time will be increased [13,14]. Moreover, a numerous combination of fuels (solid [15] and liquid [16] state of hydrocarbon-based to several phases of hydrogen fuels [17]) have been studied from the literature in scramjet propulsion systems. Hydrocarbon fuels have an advantage of higher density compared to hydrogen [18], with 11 times higher storage density, making tanks and as well as the entire vehicle compact.

To achieve precise satellite insertions into the orbit requires propulsion system with less complexity and greater performance. Hybrid rocket engines (HRE) which run by solid fuel and liquid/gaseous oxidizers, are one of the most promising systems that possess almost all the advantages of the LRE’s and SRM’s [19]. The most desired qualities of the hybrid propulsion are the simplicity (the use of one liquid propellant require only a single liquid line and relatively simpler injection system), safety (the fuel and oxidizer are located at different places), multiple times ignition ability, and the possibility of throttling (the ability to control oxidizer flow rate allows multiple starts and precise control of thrust level) [20]. Fuel regression rate is the primary parameter for the characterization of hybrid rocket performance [21]. The Paraffin as a fuel for HRE has a decent specific impulse with lower impact on the environment due to the absence of Hydrogen chloride (HCl), Sulfur oxides (Sox). Additionally, significant research has been existed on paraffin usage in HRE [22]. Nitrous Oxide (N2O) as the oxidizer avoids the usage of an additional pressurizing system [23], lowers the vehicle mass.

Emissions from combustion processes have a significant impact on air quality, the environment, and human health. Most of the rocket engines have historically been fueled by fossil fuels. However, with production from renewable energy, sustainable fuels such as Hydrogen, Hydrogen Peroxide, and Methane yield less greenhouse gases. This project aims to design a launch vehicle for payload transportation to the LEO by accessing sustainable and re-usable characteristics.

2. Launch vehicle description:
A three-stage launcher with a total mass (M) estimation of 9811.55 kg has been designed with a solid booster at its first stage, a reusable scramjet in the second and a hybrid rocket as a third/final stage which is encapsulated in the scramjet stage to inject the payload into its orbit.

The required mass of the propellant (M_{prop}) for the first stage, considering the payload mass (M_{pay}) to be the second, third stages mass, is estimated to be 5897.05 kg assuming the inert mass fraction of ‘0.1’ using the Eq. (1)

$$M_{prop} = \frac{M_{pay}(1 - K\epsilon)(e^{\frac{AV}{Kf}} - 1)}{(1 - K\epsilon, e^{\frac{AV}{Kf}} - 1)}$$ (1)

Upon all the available solid fuels, the heterogeneous mixture of HTPB/AP/Al, combination has been chosen to boost the first stage because of its and decent specific impulse. The characteristics of
the fuel when burnt have been analysed with various weight proportions using NASA CEA and the optimum specific impulse (Isp) observed is 230 seconds.

2.1. Trans-atmospheric flight phase:
The second stage of the launcher is equipped with a scramjet designed in the form of a waverider in order to take the advantage of the compression lift (using the shock waves generated by its own supersonic flight to generate the lift) to improve the lift-to-drag ratio [24].

The propellant mass (M_{2,prop}) for the second stage is estimated to be 108.47 kg with an assumption of the inert mass fraction of 0.4 and the payload mass as, the masses of upper stage. The second stage total mass (Mt) including its payload is estimated to 3259.27 kg. Since hydrocarbon fuels deliver greater density and less complexity, the possibility of reducing the vehicle size is anticipated. The scramjet will be fuelled by Methane (CH4) which yields clean combustion products. The estimated specific impulse (Isp) for the flight is 859 seconds. The scramjet engine is incorporated with axisymmetric combustor based on aft wall cavities. To overcome the corner boundary layer separation and flow distortion issues encountered in two dimensional combustors, circular combustors have been introduced.

As the residence time is shorter (in the millisecond range), the fuel injection, fuel-air mixing, and combustion must all take place within this time frame. This complete phenomenon occurs in the combustor, so combustor design optimization is strongly advised. A circular scramjet combustor based on axisymmetric cavities has been introduced to address fuel injection strategies, fuel-air mixing, and greater combustion efficiency with nominal total pressure losses. The combustor is designed as follows. To comprehend the effect of the circular combustor on the flow field, numerical simulations are used.

![Figure 1. The circular combustor configuration](image)

After the scramjet reaches its final operating point, the deployment of the final stage is carried out by elevating the payload bay at an inclination suitable for the launch with a dedicated mechanism.

2.2. Upper stage hybrid rocket:
The final stage of the launcher is propelled by a hybrid rocket engine. This stage is encapsulated in the scramjet payload bay and will be ignited after reaching its launching condition. Nitrous oxide is considered as oxidizer, due to its two-phase behavioral characteristics (because of its operating temperature nearer to the critical point) [25]. It means that, when stored in the tank, part of the total volume will be occupied by the liquid state and the other part by the vapor state which avoids the necessity of an external pressurizing tank. Paraffin-based fuel was chosen because of its high regression rate that burns 3 to 5 times faster than the other available hybrid fuels [26].

The thermodynamic and other parameters were calculated using NASA CEA and the specific impulse is estimated to be 331.75 seconds. To carry the final payload to its destination orbit, the mass of the propellant \( M_{\text{prop}} \) necessary is estimated to be 416.97 kg assuming the inert mass fraction of 0.1 and a payload mass of 100 kg. The total mass of the third stage \( M_{t3} \) is 563.30 kg.

3. Mission profile:
The launcher is designed with the capability of delivering a payload of 100 kg into the 200 km circular orbit from the ground. The required inertial velocity is estimated to be 7784 m/s at an altitude of 200 km. The first stage solid booster burns for a time period of 60.59 seconds to achieve the supersonic speeds of Mach 6.1 with a ∆V of 2075.1 m/s. The required thrust \( F \), propellant mass flow rate \( M_{\text{prop}} \), and chamber pressure are 2,40,383.16 N, 106.44 kg/s, 8.10^6 Pa. The calculation of ‘∆V’, ‘F’ and ‘\( M_{\text{prop}} \)’ are done using Eq. (2), Eq. (3) and Eq. (4)

\[
\Delta V = \sqrt{\left( Ve^2 \log_\frac{M_0}{M} - 2Ve \cdot t \cdot \sin \theta \log_\frac{M_0}{M} + g^2 \cdot t^2 \right)}
\]  
(2)

\[
F = \frac{F}{W} \cdot M \cdot g
\]  
(3)

Where F/W i.e., thrust to weight ratio considered to be 2.5

\[
M_{\text{prop}} = \frac{F}{I_{\text{sp} \cdot g}}
\]  
(4)

The scramjet is programmed to ignite at an altitude of 25 km and accelerate to an altitude of 50 km with the estimated \( \Delta V \) of 1326.4 m/s, the flight path angle of 1.5°, and thrust \( (F) \) of 79852.28 N. The mass flow rate of the propellant \( (M_{\text{prop}}) \) is estimated to be 9.48 kg/s.

After the ejection of the final stage hybrid rocket from the scramjet at an altitude of 50 km, it accelerates to the target orbit at 200 km altitude with an estimated \( \Delta V \) of 4382.4 m/s producing a thrust \( (F) \) of 13800.98 N with a chamber pressure of 3,10^6 Pa and at a burning time of 98.228 seconds. A MATLAB code was developed to derive the above characteristics of the hybrid rocket. After reaching the desired altitude, the final payload will be injected into the designed orbit.

4. Results & discussions:
It's possible to use a 3D combustor with axisymmetric cavities in the scramjets. The inclusion of the hybrid rocket motors for the payload delivery is efficient and economical (mass budget). To understand the effect of the circular combustors on the flow field, numerical simulations have been adopted. Investigations were carried out, in particular, to analyze the fueling techniques and their impact. HRE has been specifically designed for the mission's third stage.

4.1. Computational analysis on circular combustor: Computational studies have been carried out under steady-state conditions to measure the implications of injection location, angle, and their effect. The boundary conditions are determined by the test bench configuration [27]. To solve the RANS equations, Ansys Fluent commercial code with a density-based solver and the SST K-Omega
turbulence model is used. Non-reacting flow numerical investigations have been validated against experimental bench results [28].

Four injection strategies were chosen: 45° angled injection into the cavity (case I), parallel injection into the cavity (case II), normal injection into the cavity (case III), and upstream injection into the cavity (case IV) (case IV). Each injector has a diameter of 1mm, and four uniformly spaced injectors are placed around the combustor in every case. The investigation yielded the following results.

4.1.1. Fuel mass fraction: The contours of the fuel mass fraction with streamline formation are depicted in the figure below. Aside from Case IV, the remaining three cases are direct injection techniques in which the majority of the fuel is contained within the cavity volume and is a fuel-rich zone. The fuel-air interaction is improved by the leading-edge, fuel jet-induced shocks and the shock train downstream of the cavity. In cases I, II, and IV, the subsonic regions within the cavity, aid in achieving greater mixing. In case III, the formation of counter-rotating vortices inside the cavity has a significant impact on mixing efficiency and improves combustion, as evidenced by the average fuel mass fraction of 0.00046 in the cavity region. Particularly in the case of upstream injection (case IV), fuel-jet induced bow shock separated the boundary layer, allowing greater amounts of air-fuel mixture to be entrained into the cavity, resulting in higher mixing rates and combustion efficiency. When compared to other strategies, the penetration of the fuel jet is much better in this technique, resulting in a homogeneous distribution of the combustion products across the combustor with an average fuel mass fraction of 0.0007 within the cavity volume. Since the fuel is moving parallel to the supersonic stream in Case II, the jet penetration is shorter and there is less air-fuel interaction. Case I has a significant amount of fuel in the shear layer, but the penetration is not as high as in Case IV. Besides average fuel mass fractions reside inside the cavity for these 2 cases are 0.0017, 0.0008.

4.1.2. Combustion efficiency and stagnation pressure loss: The following figures depict the effectiveness of fuel injection techniques in terms of combustion efficiency and total pressure losses. In all cases, the overall combustion efficiency is greater than 75%. The transverse injection cases III and IV, reported 100 % combustion efficiency within the combustor, indicating desired mixing and fuel jet penetration. The 45° angled injection into the cavity technique resulted in a combustion efficiency around 92 %. Because of majority of the fuel is moving with the supersonic stream, Case II did not reach the intended combustion efficiency and achieved 82 %.

![Image of fuel mass fraction contours](image)
The stagnation pressure losses have been calculated at various locations across the combustor. Due to the jet-induced bow shocks and higher jet penetration, Cases III and IV encountered greater pressure losses inside the cavity volume. Because of the poor fuel jet-supersonic stream interaction, parallel injection into the cavity resulted in lower pressure losses than all other injection strategies. The pressure losses in the cavity region for case I are lower than in cases III and IV, but higher than in case II. Nonetheless, at the combustor's exit, all strategies reported almost same total pressure losses with a minor variation.

![Figure 2. Combustion efficiency](image1)

![Figure 3. Stagnation pressure loss](image2)

4.2. Hybrid rocket design:
The chamber pressure is a design choice, it is assumed $3 \times 10^6$ Pa. Using it as input in the NASA’s code Chemical Equilibrium with Applications (CEA), it is possible to obtain the propulsion efficiency parameters required to size the fuel grain, oxidizer tank, injector plate and nozzle.

Assuming the initial oxidiser mass flux ($G_{ox}$) as $225 \text{ kg/ m}^2\text{.s}$ and knowing the burning time (98.228 seconds), the initial and final radius of the grain can be calculated. Applying geometric relations in combination with the fuel physical properties, it is possible to obtain the length, mass, and mass flow rate of the solid propellant. Table 1 shows the results. The oxidizer is stored at, approximately, $6 \times 10^6$ Pa in the tank. Since nitrous oxide presents a two-phase behaviour, part of the volume is used to store the gas phase, while the rest is for the liquid. Assuming ullage volume of 12%, the result of the oxidizer tank sizing is shown in Table 2.

| Table 1. Fuel design | Table 2. Tank design |
|----------------------|----------------------|
| Fuel initial port radius | 0.072756 m |
| Fuel final radius (chamber internal radius) | 0.224569 m |
| Fuel length | 0.383980 m |
| Fuel mass | 49.005184 kg |
| Fuel mass flow rate | 0.294114 kg/s |
| Oxidizer mass | 367.538882 kg |
| Tank volume | 0.552913 m$^3$ |
| Tank radius | 0.426308 m |
| Tank length | 0.40 m |

The injector plate is responsible to atomize the oxidizer due to a pressure drop between the feeding line and the combustion chamber. In order to provide a pressure, drop of $3 \times 10^6$ Pa with a discharge coefficient of 0.66, the injector was designed, and the results are represented in Table 3. The preliminary nozzle design considers a conical shape with convergent angle of 45° and divergent angle of 15°. Using the results from CEA and the engine design assumptions, the nozzle sizing results are shown in Table 4.
5. Conclusion:
A comprehensive investigation has been conducted in order to design a launch vehicle capable of delivering a 100 kg payload in 200 km LEO, with a scramjet engine in the second stage and a hybrid rocket in the final stage. The overall mass of the launching vehicle quantified as 981.55 kg. A wave rider shaped scramjet with an axisymmetric cavity-based combustor is employed in the launcher and a computational analysis on various injection angles in a circular combustor is carried out to understand the performance of the injection strategies. The observations drawn shows that the upstream injection reported maximum combustion efficiency with greater fuel jet penetration. The inclusion of the hybrid rocket in the third stage of the mission, benefits the launcher in terms of the performance and mass budget. Moreover, the inclusion of green propellants contributes to sustainable space transportation.

6. Nomenclature:

| Symbol | Definition |
|--------|------------|
| ∆V    | Change in Velocity |
| Isp   | Specific Impulse |
| Ks    | Inert mass fraction |
| g     | Gravitational acceleration constant |
| V_e   | Velocity at exit |
| M_0   | Initial mass |
| HTPB  | Hydroxyl-Terminated Polybutadiene |
| M_prop| Mass of the Propellant |
| M_pay | Mass of the Payload |
| M/Mt  | Total Mass |
| Ṁprop| Mass flow rate of the propellant |
| Gox   | Oxidiser mass flux |
| F     | Thrust |
| t_b   | Burning time |
| Pc    | Chamber Pressure |
| RBCC  | Rocket Based Combined Cycle |
| AP    | Ammonium Perchlorate |
| Al    | Aluminium |
| HRE   | Hybrid Rocket Engine |
| LRE   | Liquid Rocket Engine |
| SRM   | Solid Rocket Motor |
| LEO   | Low Earth Orbit |
| CEA   | Chemical Equilibrium with Applications |

References
[1] Zeyu Dong, Mingbo Sun, Zhenguo Wang, Jian Chen, Zun Cai. Survey on key techniques of rocket-based combiner cycle engine in ejector mode. *Acta Astronautica* 164 (2019) 51-68.
[2] W. H. Heiser and D. T. Pratt, Hypersonic Airbreathing Propulsion. *AIAA Education Series, American Institute of Aeronautics and Astronautics, Inc.*, 1994.
[3] R. Daines and C. Segal, “Combined rocket and airbreathing propulsion systems for space-launch applications,” *Journal of Propulsion and Power*, vol. 14, no. 5, 1998.
[4] Boyce, R., S. Gerard, and A. Paull. The HyShot Scramjet Flight Experiment - Flight Data and CFD Calculations Compared. in *12th AIAA International Space Planes and Hypersonic Systems and Technologies*. 2003. Norfolk, Virginia: AIAA.
[5] Moses, P.L., et al., NASA Hypersonic Flight Demonstrators - Overview, Status, and Future Plans. *Acta Astronautica*, 2004. 55: p. 619-630.
[6] Fortescue, P. and J. Stark, *Spacecraft System Engineering*. 1995: Wiley.
[7] Moses, P.L., et al. An Airbreathing Launch Vehicle Design with Turbine-Based Low-Speed Propulsion and Dual Mode Scramjet High-Speed Propulsion. in *9th International Space Planes and Hypersonic Systems and Technologies Conference*. 1999. Norfolk, VA: AIAA.
[8] Kuczera, H., P. Krammer, and P.W. Sacher. SANGER and the German Hypersonics Technology Program - Status Report 1991. in *International Astronautical Congress. 1991: International Astronautical Federation*.
[9] Seleznev, R.K.; Surzhikov, S.T.; Shang, J.S. A review of the scramjet experimental data base. *Prog. Aerosp. Sci.* 2019, **106**, 43–70.

[10] Ren, Z.; Wang, B.; Xiang, G.; Zhao, D.; Zheng, L. Supersonic spray combustion subject to scramjets: Progress and challenges. *Prog. Aerosp. Sci.* 2019, **105**, 40–59.

[11] Ben Yaker, A.; Hanson, R.K. Experimental Investigation of Flame-Holding Capability of Hydrogen Transverse Jet in Supersonic Cross Flow. *Symp. Int. Comb.* 1998, **27**, 2173–2180.

[12] Owens, M.G.; Tehranian, S.; Segal, C.; Vinogradov, V.A. Flame-Holding Configurations for Kerosene Combustion in a Mach 1.8 Airflow. *J. Propuls. Power* 1998, **14**, 456–461.

[13] Zhang, J.; Chang, J.; Tian, H.; Li, J.; Bao, W. Flame Interaction Characteristics in Scramjet Combustor Equipped with Strut/Wall Combined Fuel Injectors. *Combust. Sci. Technol.* 2020, **192**, 1863–1886.

[14] Kim, C.H.; Jeung, I.S. Forced combustion characteristics related to different injection locations in unheated supersonic flow. *Energies* 2019, **12**, 1746

[15] Cohen-Zur, A. and B. Natan, Experimental Investigation of a Supersonic Combustion Solid Fuel Ramjet. *Journal of Propulsion and Power*, 1998. **14**(6): p. 880-889.

[16] Savino, R. and G. Pezzella, Numerical Analysis of Supersonic Combustion Ramjet with Upstream Fuel Injection. *International Journal for Numerical Methods in Fluids*, 2003. **43**: p. 165-181.

[17] Engelund, W.C., et al. Propulsion System Airframe Integration Issues and Aerodynamic Database Development for the Hyper-X Flight Research Vehicle. in XIV ISOABE Conference. 1999. Florence.

[18] Lewis, M.J., Significance of Fuel Selection for Hypersonic Vehicle Range. *Journal of Propulsion and Power*, 2001. **17**(6): p. 1214-1221.

[19] N.A. Davydenko, R.G. Gollender, A.M. Gubertov, V.V. Mironov, N.N. Volkov. HybridRocket Engine: The benefits and prospects. *Aerospace Science and Technology* **11** (2007) 55-60.

[20] Fernando de Souza Costa and Ricardo Vieira. Preliminary Analysis of Hybrid Rockets for Launching Nanosats in LEO. *J. of the Braz. Soc. of Mech. Sci. & Eng. October-December 2010, Vol. XXXII*, No. 4/509.

[21] Sutton, G.P.; Biblarz, O. Hybrid Propellant Rockets. In *Rocket Propulsion Elements*, 7th ed.; John Wiley & Sons: New York, NY, USA, 2001; pp. 585–593

[22] Mario Kobald, Christian Schmierer, Helmut Ciezki, Stefan Schlechtriem, Elena Toson, and Luigi De Luca. “Viscosity and Regression Rate of Liquefying Hybrid Rocket Fuels”. In: *Journal of Propulsion and Power* **33.5**(2017), pp. 1245–1251

[23] Heejang Moon, Seongjoo Han, Youngjun You, Minchan Kwon. Hybrid Rocket Underwater Propulsion: A Preliminary Assessment. *Aerospace* 2019, **6**, 28, doi: 10.3390/aerospace6030028.

[24] Cockrell, C.E., L.D. Huebner, and D.B. Finley, Aerodynamic Characteristics of Two Waverider- Derived Hypersonic Cruise Configurations. 1996, NASA: Hampton, Virginia.

[25] Stephen A. Whitmore and Spencer N. Chandler. Engineering model for selfpressurizing saturated-n2o-propellant feed systems. *Journal of Propulsion and Power*, **26**(4):706–714, 2010.

[26] M. Karabeyoglu, B. Cantwell, and D. Altman. *Development and testing of paraffin-based hybrid rocket fuels.*

[27] Assis, S.M.; Suppandipillai, J.; Kandasamy, J. Transverse Injection Experiments within an Axisymmetric Scramjet Combustor. Int. *J. Turbo Jet-Engines* 2019.

[28] Relangi, N.; Garimella, D.; Jayaraman, K.; Venkatesan, J.; Jeyakumar, S.; Ingenito, A. Numerical simulations of axisymmetric aft wall cavity in supersonic combustion ramjets. In *Proceedings of the AIAA Propuls. Energy* 2020 Forum, Virtual Event, 24–28 August 2020; pp. 1–15.