DOI: 10.13009/EUCASS2022-4879

Overview of the Past and Current Research on Hybrid Rocket Propulsion at the University of Brasília

Artur Elias DE MORAIS BERTOLDI[1], Carlos Alberto GURGEL VERAS[2], Olexiy SHYNKARENKO[1], Artem ANDRIANOV[1], Jungpyo LEE[1] and Domenico SIMONE[1]

[1]University of Brasília, Faculty of Gama, AE Projeção A, 72444-240, Brasília, Brazil

[2]University of Brasília, Mechanical Engineering Department, 70910-900, Brasília, Brazil

Abstract

Chemical propulsion research at the University of Brasilia started in 1999, mainly on hybrid propellant rocket engines. Over the years the group has conducted scientific and technological studies covering motor design, solid fuel regression rate, injectors, ignition systems, internal ballistics, combustion instability, thrust control, thermal insulators, small satelites launch vehicle preliminary design, and additive manufacturing. The team has also elaborated new safety measures for hybrid and missile propulsion systems testing, including hardware and software solutions in test preparation, execution, and termination. Collectively, the test benchs and simulation tools form the most advanced hybrid propellant research university laboratory in Brazil. This paper addresses the main results and their contribution to developing hybrid rocket motors and solid ramjet engines technologies.

1. Introduction

Hybrid propellant rocket technology is a class of chemical propulsion in which fuel and oxidizer are in different states of matter. In the classical concept, the oxidizer is liquid, stored in external tanks, and the fuel is solid, placed in the combustion chamber, before the nozzle. Hybrid rocket engines (HREs) have been exhaustively investigated in the past decades motivated by some specific technical advantages vis-à-vis over liquid rocket engines (LREs) or solid rocket motors (SRMs). Safety, higher performance in terms of specific impulse, low environmental impact, throttling and shut-off are key advantages over solid propellant motors. Reduced fabrication costs, complexity and development effort are the main advantages of hybrid engines over the liquid counterpart. The main drawbacks of classical hybrid propulsion systems are the low regression rate of the solid fuel, poor combustion efficiency, and the *O/F* shift during operation [1-2].

The University of Brasilia (UnB) initiated studies on hybrid rocket propulsion systems in the year 1999. Since then, many combinations of propellants and thrust levels have been under investigation, with special attention to high regression rate paraffin. During this time, a great number of students, undergraduate and graduate, has been enrolled in rocket propulsion research and development.

Historically, Brazil has been conductiong, with great success, research and delopment of solid propellant motors applied to sounding rockets and launch vehicles. Modest investiment, though, have been granted in supporting liquid propellant rocket engines. The higher cost of such propulsion systems and the lack of critical mass in terms of qualified human resources have delayed technological progresses for this technology. Brazil still lacks a modern and competitive rocket propulsion industry. Therefore, cheaper and less compex propulsive systems would demand less overall development effort, thus increasing the opportunities for public financing. With that in mind, HRE would be a competitive alternative for rocket engines development in Brazil.

This paper addresses the Chemical Propulsion Laboratory activities (CPL) and the Energy and Environment Laboratory (LEA) decades of contribution to the development of hybrid rocket propulsion systems. Over the years hundreds of tests with different hybrid rocket motors were carried out by the group in many fields, such as solid fuel regression rate using diverse fuels mixtures, injectors design, internal ballistics, combustion instability, thrust control, ignition, and thermal insulators. The university and its partners have made numerous efforts to design, manufacture, and test rocket motors, from small and low-altitude sounding hybrid rockets to a propulsive decelerator test motor for the Brazilian space program.

DOI: 10.13009/EUCASS2022-4879

2. Early Research and Projects

The UnB/LEA laboratory started hybrid propulsion activities at the University of Brasília, studying rocket fuel regression rate of GOX (gaseous oxygen) and polyethylene. For that, the team developed and indigenous test apparatus of very low cost. Late in 2004 the Energy and Environment Laboratory introduces paraffin-based fuel to the university propulsion activities. Based on these scientific research results, a set of small experimental rockets were designed, manufactured, and launched. In 2012 most of the activities of the Hybrid Propulsion Team (HPT) moved to the aerospace engineering course at the Faculty of Gama, a new campus of the University of Brasília established in 2011 and located in the Gama city, approximately 40 km from the main campus. The aerospace engineering course was proposed to the University's Provost as a natural evolution of the integrated research conducted by professors from the physics, mechanical e electrical depatments. Couple of years later, the Chemical Propulsion Laboratory (CPL) was inaugurated to enclosing the general activities of all research groups dealing with aerospace propulsion.

More specifically, the first activities of the University of Brasília related to hybrid rocket motor was an engine demonstrator with 800 N thrust using polyethylene-based solid fuel, with several combustion ports, and GOx (gaseous oxygen). Nitrous oxide was late introduces as the oxidizer primary choice for applications in small sounding rockets. To support the activities a test stand was built and several firings were performed to evaluate the ignition, combustion characteristics of the propellants, and system's overall performance [3]. In 2003 paraffin-based fuels started to be studied by the group using GOx as the oxidizer. A detailed fuel processing procedure was implemented in order to have the paraffin-based solid grain. By such means, the performance of the solid fuel for static tests, upon combustion, showed to be reliable. A series of tests using a 200 N engine was conducted between 2003 and 2004 to increase the expertise of the team in topics such as fuel grain manufacturing, operation with rocket engines, and data acquisition system [4].

In 2005 a new test bench was developed to test hybrid rocket engines in vertical position. A pneumatic valve remotely controls the oxidizer injection system and a nitrous oxide tank is turned upside down in order to fill the injection system with liquid oxidizer. The main parameters obtained were the thrust, feed system and main chamber pressures respectively by a load cell and pressure transducers. All the data is processed under the LabView platform and post-processed in a worksheet chart. Three injectors were fabricated with 10 elements of 0.5, 0.78, and 1.4 mm diameter, with calculated flow rates of 60, 150, and 460 g/s, respectively. The mass flow rate was verified experimentally with a pressurized water tank for three different levels of pressure drop, 10, 15, and 20 bar. The measured discharge coefficient (c_d) was calculated as 0.55. The average thrust was between 150 and 200 N and the chamber pressure varies from 12 to 16 bar [5]. Table 1 summarises the results obtained, where $\bar{G}$ is the average oxidizer mass flux and $\bar{r}$ the average regression rate.

Table 1: Experimental results N_2O – paraffin [5], *with modifications*

Test number	Injector Plate (mm)	$\bar{G}\ \left({}^{g}/_{cm^2 s}\right)$	$\bar{r}\ \left({}^{mm}/_{s}\right)$	Notes
1	0.78	18.6	2.0	N2O Tank 1
2	0.78	20.2	2.1	N2O Tank 1
3	0.78	9.6	1.6	N2O Tank 1
4	0.78	12.0	1.5	N2O Tank 1
5	0.78	20.2	1.4	N2O Tank 1
6	0.78	N.V.	N.V.	Tank 1 empty
7	0.78	N.V.	N.V.	Tank 1 empty
8	0.78	13.5	2.7	N2O Tank 2
9	0.78	15.0	2.8	N2O Tank 2
10	0.78	17.0	3.7	N2O Tank 2
11	1.40	50.5	4.4	N2O Tank 2
12	1.40	50.2	4.3	N2O Tank 2
13	1.40	58.5	4.6	N2O Tank 2

Note: the injector plate with 0.5 mm holes did not match the available nozzles.

Following the experience obtained in the static test campaigns, the group suggested a twelve months project to design, build and test a model rocket based entirely on hybrid technology employing paraffin as the solid fuel and nitrous oxide as the oxidant. To our knowledge, this attempt was the first-ever conducted in the southern hemisphere. The challenge was proposed in the context of an undergraduate project (course completion requirement) and was carried out by two

DOI: 10.13009/EUCASS2022-4879

students from the Mechanical Engineering Department, University of Brasília, supervised by a professor from the same department [6]. The study was carried out in small steps: (a) extent literature review with emphasis on similar projects; (b) adapt the test stand to work with a liquid oxidizer (N_2O); (c) improve the fuel processing technology; (d) improve the ignition system; (e) implement mathematical models for rocket motor performance analysis; (f) identify and acquire model rocket design and flight simulation system; (g) static test with up to the 500 N thrust engine; (h) design the model rocket; (i) design the oxidizer filling and ignition system; (j) static test the model hybrid rocket; (k) integrate all the subsystem and flight test the rocket; (l) overall performance analysis.
The rocket motor conception was entirely based on the engine tested statically, but with another oxidizer injector. Based on the simulation and the experimental results shown in Figure 1, it was decided to build a 300 N hybrid motor. A model rocket was then conceived for such engine. The motor parameters are listed in Table 2, more detail on the motor design can be seen in Cozac and Santos [6].

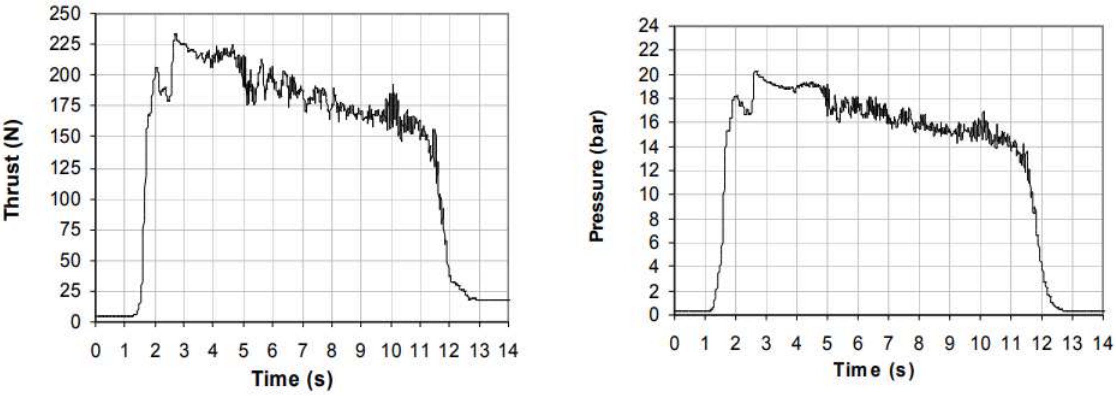


Figure 1. Motor thrust and chamber pressure.

Table 2: Rocket engine main parameters

Propellant	**Combustion**	**Nozzle**
N2O mass = 0.8 kg	Characteristic velocity = 1485 m/s	Throat diameter = 11.1 mm
Fuel mass = 0.077 kg	Initial chamber pressure = 20 bar	Entrance diameter = 30.9 mm
O/F = 9.8	Specific Impulse = 224.6 s	Exit diameter = 22.36 mm
Fuel Density = 0,745kg/m^3	Total Impulse = 1500 Ns	Convergence angle = 24°
Total mass = 0.827 kg	**Oxidizer Tank**	Divergence angle = 95°
Grain length = 150 mm	Initial tank pressure = 50 bar	Length = 45.56 mm
Port Diameter = 25 mm	Oxidizer flow = 0.180 dm^3/s	External diameter = 40.9mm
Fuel = solid paraffin	Material = Al alloy 6101 T6	Material = graphite

The oxidizer tank was made of 561.5 mm aluminium pipe, with a 48.26 external diameter, and composed of a welded end cap and a piston-like injector, which also defined the combustion chamber volume. The length of the oxidizer tank was calculated using the density of the nitrous oxide, at ambient temperature and the total mass of the oxidizer. This mass was estimated as 0.8 kg. The minimum length was set to 500 mm, including 5% of ullage volume, for gaseous nitrous oxide, the remaining is filled with liquid N_2O. The oxidizer injector was an aluminium piston-like device with 30 mm in length and 40.9 mm of external diameter. The combustion chamber, placed at the opposite end of the monotube was composed of the pre-combustion chamber (30 mm), the fuel grain (150 mm), and a post-combustion

DOI: 10.13009/EUCASS2022-4879

chamber (50 mm). PVC (polyvinyl chloride) was used for the pre-and post-combustion chambers and the solid grain case. The exhaust gases nozzle was constructed of graphite. The nozzle was machined with a 90° convergence angle and 24° divergence angle. An O-ring was placed in the nozzle for sealing the combustion chamber. For closing the combustion chamber, an end cap was threaded. Figure 2 shows the rocket's main components. The two prototypes received the tag name LILE. To our understanding, this was the first model rocket, powered by a hybrid rocket engine, succefully launched in the Southern Hemisphere.

Fig. 2. Model hybrid rocket engine.

With the rocket, and subsystems assembled the group travel to small farm in the Santo Antônio do Descoberto, Brazil, for the launch operation. The weather was cloudy and mildly windy. The launch angle was estimated at less than about 85 degrees bending towards the wind direction. The group was protected by a small construction that worked as a bunker for safety precautions. The rocket, named LILE-1, was succesfuly launched, crashing about 700 m, in the direction of the wind and the launch pad angle. The motor was damaged in the upper part, due to the impact. It was concluded that the rocket hit the ground in a ballistic trajectory, flying nose down. A week later another rocket (LILE 2) was also launched.

2. UNIESPAÇO 1 and the Santos Dumont (SD) Project

The UNIESPAÇO program was established in 1997 by the Brazilian Space Agency. The program selects projects covering different areas of aerospace science and engineering from main Brazilian state universities and research institute centres. In the aims of the UNIESPAÇO 1 project, in 2005, the *Santos Dumont* (SD) sounding rocket program begins at the University of Brasília, intending to develop sounding rockets capable of reaching altitudes as high as 8 km. The program was named in tribute to Mr. Alberto Santos Dumont, a Brazilian inventor and aeronaut that, between 1873 and 1932, had won various prices and honours such as the Aero Club of France, the and the Deutsch-Archdeacon price, and was the first aeronaut who performed a certified flight in 1906 with an airplane, the 14-bis, also known as *Oiseau de Proie,* winning the Archdeacon price in Paris, France.

The *Santos Dumont* project was divided into two phases, design, construct and test two model rocktes named SD-1 and SD-2. The first one (SD-1) was a based on 500 N rocket, for low altitude flights, and the SD-2 relyied on 1500 N thrust hybrid rocket (Figure 3). In addition to the rockets a test bench apparatus was also constructed to characterize the hybrid engines before launch. The motor of the SD-1 rocket (500 N) was manufactured and further tested several times. In 2006, the tests with the SD-2 class motor had begun and the motor was fired more than 30 times in horizontal and vertical positions. In 2008 a group of students from the LEA, the *Hybrid Propulsion Team* (HPT), of the University of Brasília, Mechanical Engineering Department, went to São José dos Campos, São Paulo, with the mission to validate the aerodynamic profile of the SD-2 rocket in the Brazilian Aeronautics and Space Institute [7]. Several improvements were made to the rocket with the knowledge gained in this short period.Table 2 shows the main parameters of the SD-2 rocket. Unfortunately, the program was cancelled in 2009 due to a lack of funds and SD-2 was never launched.

Parallel to the Santos Dumont rocket project, an advanced test bench was developed. The concept employed was based on a refurbished INPE (National Institute of Space Studies) bench that a group of UnB/LEA students operated in 2007, and it was one of the greatest rocket test campaigns ever made in Brazil with more than 50 test-fires made in a month. With the knowledge gained in the INPE campaign, a new versatile test bench was designed by the UnB/LEA/HPT. The new bench was designed for a modular operation in a way that new experiments could easily be adapted and

DOI: 10.13009/EUCASS2022-4879

realized. The modular test bench has been used in experiments as diverse as; thrust modulation, bio-paraffin research, composite nozzles, nozzle refrigeration, and combustion instability characterization.

Figure 3. SD-2 sounding rockets, versions A (right) and B (left).

Table 2: SD-2 rocket main parameters

Propellant	**Combustion**	**Oxidizer Tank**
N_2O mass = 10.5 kg	Thrust (average) = 1100 N	Initial tank pressure = 52 bar
Paraffin mass = 1.6 kg	Chamber pressure (average) = 15 bar	Oxidizer mass rate = 800 g/s
O/F (initial) = 4.2	Specific Impulse (theoretical) = 215 s	Material = Al alloy 6101 T6
Fuel mass rate = 195 g/s	Total Impulse (theoretical) = 1300 Ns	**Apogee**
Port Diameter = 25 mm	Burn time = 12 s	8200 meters

3. UNIESPAÇO 2 and the SARA Motor

The Brazilian Institute of Aeronautics and Space (IAS) designed a recoverable satellite (SARA) for long duration microgavity experiments. The satellite was planned to carry up to 55 kg of payload mass, with total launch mass not exceeding 350 kg. Missions would follow a circular orbits less than 300 km altitude with two-degree inclination. After mission completion, of about ten days, the reentry procedure is started providing the right positioning of the satellite followed by a deboost impulse. The IAS developing team investigated deboost engines based on solid and liquid rocket engines while CPL from University of Brasilia proposed a hybrid engine for the reentry system. The motor was named “SARA” and its development followed several constraints, such as safety, research purposes of the project, available

DOI: 10.13009/EUCASS2022-4879

resources, access to structural materials and manufacturing technologies, cost, compatibility with the test bench, and data acquisition system. The project team built the paraffin - nitrous oxide hybrid engine [8], [9] with an average thrust of 1 kN and operation time of 12 *s* to satisfy the mentioned requirements. The engine is self-pressurized by an oxidizer kept at a temperature from 20 to 25 degrees Celsius, creating the feeding pressure above 52 bar in the line and 30 bar in the combustion chamber. The low-cost showerhead injection plate was designed for the engine. Its simple design allowed to reduce the total cost and time of manufacturing. The optimal injectors' distribution showed high overall engine performance within the tests. The motor had a modular structure with fast connections, allowing to substitute its components for testing of various motor configurations. The disadvantage of such design is an excess of weight of the test motor and the necessity of large sealing rings. The major benefit of such design, built with thick walls, was the higher safety factor of the engine.

An axial capsule pyrotechnic igniter was designed to start the engine in static firings. However, the lack of solid-fuel grains suppliers in Brazil made its continues utilization unfeasible. During the first years of the project development, the simple rocket-candy igniter installed in the pre-combustion chamber was used in the tests. Later, the development of the gas torch ignition system [10], [11], as depiced in Figure 4, improved motor ignition.

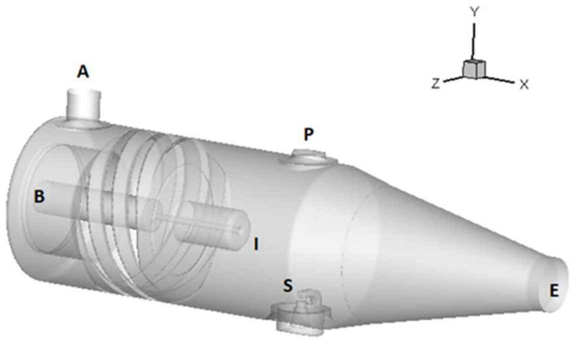


Figure 4. Igniter for the hybrid engine: A, B, E, I, P, S respectively O2 inlet, CH4 inlet, exit section, CH4 injector exit section, probe hole and spark plug head.

The single-port paraffin grain was used for the proposed SARA engine. The technique of uniform solid grain manufacturing was created to provide high-quality fuel. Later, the authors studied grain manufacturing quality, crack propagation, and its influence on engine performance and safety [12]. Intensive testing of the SARA motor showed a necessity to use fuels with higher structural characteristics, such as polyethylene (PE, HDPE). The chamber extensions allowing PE and HDPE utilization were constructed and tested. The engine had a smooth aerodynamic shape constructed according to the recommendations for efficient motor design. The convergent part of the pre-combustion chamber and optimal injectors distribution provide a uniform flow of oxidizer near the fuel grain. The post-combustion chamber had an elongated profile to increase the fuel droplets' residence time and ensure complete combustion. The pre-and post-combustion chamber profiles were designed using CFD modelling [10] to ensure the optimal geometrical shape of the engine. Further experimental tests proved tha such approach was favourable in improving the overall performance of the system. The nozzle of the SARA engine consisted of the graphite insert and the supersonic high-expansion stainless steel or carbon composite nozzle. The heat insulation of the SARA motor was accomplished from organic fibre composite material, "celeron" and further substituted by a silicon rubber. The insulation of 5 mm thick allowed to extend the motor operation time up to 40 *s* with the use of HDPE as a fuel. The use of the silicon rubber allowed in-house manufacturing of the insulation, thus reducing costs and time for project development.

3.1 Feed System and Valve

A complex feed system, compatible with earlier developed engines, was developed to safely and efficiently issue a liquid or gaseous oxidizer to the rocket engines. The system was designed to control the flow of oxidizer and hence, motor thrust modulation using PID and real-time oxidizer injection. System development was based on "safety principles" – broad regime operation, through system integrity check-up, testing preparation, test execution, and termination by the the programed engine shut-down. Depowering should happen in automatic mode without a human presence on the test site. A system observation and control were realized from a dedicated safe control room.

The flow control valve was the key component of the thrust control system. It was composed of a valve body operated with external gears augmented by a high-torque servo motor and embedded electronics with feedback. An aluminium

DOI: 10.13009/EUCASS2022-4879

sealed case was fabricated to mount the valve either on a rocket motor system or test bench. The case protects the control valve components from an external environment, such as vapor or liquid, dust, mechanical and electrical impact, etc. Two valve models were built for different propellant flow rate levels. The valves were designed to fulfil mass/performance, volume/performance, cost/kg, reliability, and compatibility requirements. More than 100 hot tests were performed to validate the flow control system. Its operation was proven in various regimes of ramjet operation: cold flow, combustion initiation, solid fuel ignition, nominal motor operation, and combustion termination.Some test results were previously published in [13], [14].

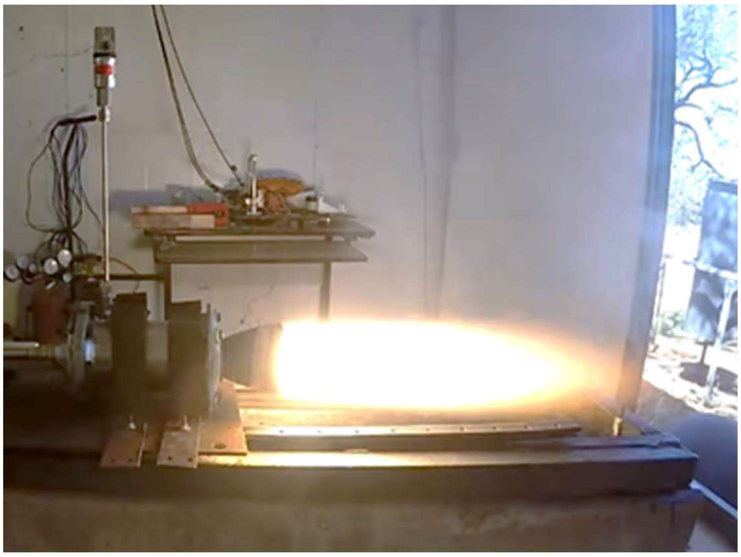


Figure 5. Hybrid rocket motor controlled by the valve designed at CPL

Moreover, the team also verified valve component's resistance to long periods of exposure to aggressive environment [15]. Variation of the external pressure and temperature, humidity, dust resistance, external loads, waterproof capabilities, etc., had been verified. The system operation and performance limits were defined experimentally, showing its stable and reliable performance in laboratory conditions and simulated environments. Thrust control has been implemented using two techniques: "reverse flow control function" and PID algorithm. Flow control in the hybrid rocket motor is realized altering the oxidizer mass flow rate, and consequently the pressure downwards of the control valve, Figure 6.

The total error of mass flow measurement took into account time delays of system response, positioning of the control valves, sensor error, and signal noise. It was found that total associated error was strictly related to valve opening levels and is practically independent of propellants inlet pressure. The PID control algorithm used the gas pressure after the valve as an input, which was filtered for noise, and the valve opening level as the output.

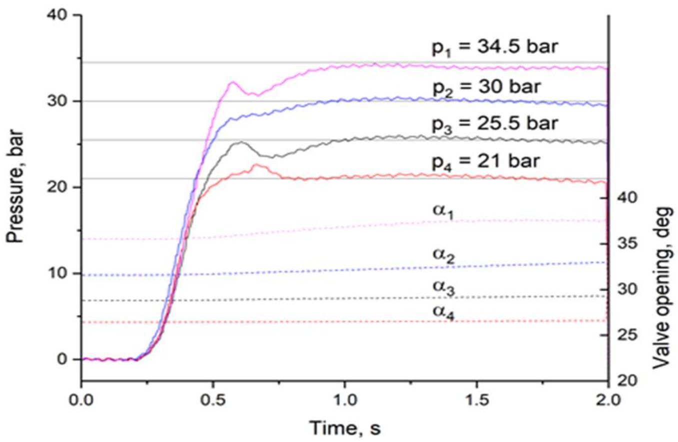


Figure 6. The behaviour of pressure after the control valve for various goal pressure levels: 21 bar, 25.5 bar, 30 bar, 34.5 bar, and corresponding valve opening levels

DOI: 10.13009/EUCASS2022-4879

3.2. Motor Insulation

Most of the experimental hot tests were done with relatively short burning times when paraffin was employed a the solid fuel, due to its higher regression rate. To increase the motor operational time for scientific analysis as well as for application in high-altitude rocket missions a heat insulation capable to protect the structure of the combustion chamber from overheating and damage was erquired.

Substantial experience gained from previous study of thermal protection systems for solid propellant motors in the early years [16] and a relatively recent survey [17] provide comprehensive information about ablation materials. However, it must be emphasized that the thermal protection of hybrid motors is somewhat different from solid fuel due to the presence of a pre-and post-chamber. While the solid propellant makes a part of a thermal protection system, the walls of pre-and post-chambers of a hybrid propellant motor are not covered by the solid grain. The problem of insufficient information on the behaviour of ablative polymeric composites of thermal protection in hybrid-propellant motors was outlined in the early 60s [18] and since this time few data was provided on the issue. Thus, several experimental works were begun in the laboratory in order to solve the problem of heat insulation for long burning times, up to 40 s. These works included the estimation of heat fluxes [19] in the combustion chamber of a low thrust hybrid propellant motor and measurements of ablation rates of various low-cost, non-toxic and thermosetting resins and rubbers available on the Brazilian market, such as epoxies, polyesters, polyurethanes and silicones [20].

Silicones are widely used in various thermal protection systems showing significant advantages over materials based on butyl rubber, polyurethanes, and epoxy resins [21]. Experimental tests show that composite silicon-based insulators can have better performance in some conditions than composites based on phenolic or epoxy resin [22], [23]. The choice of other materials was mainly guided by low cost, availability on the market, and accessible manufacturing technology.

Heat flux estimations in the most stressed part of the combustion chamber was based on the solution of an inverse heat conduction problem, whose approach included mathematical processing of a measured temperature history inside a heat-conducting solid [24]. The composite insulator consisted of phenolic resin reinforced by cotton fabric and the graphite nozzle, both with known thermal conductivity properties, were used as a heat-conducting solid. The insulator was installed in the post chamber of a modular hybrid propellant test-motor, whose design is given in reference [9]. The principal propellant components were liquid nitrous oxide and paraffin. The procedure included temperature measurements on the external surface of the insulator by K- and N-type thermocouples (Figure 7, a) during the burning test with a duration of near 11 s. The experimentally obtained temperature-time data was used to compute the value of the heat flux as a boundary condition at the internal surface of the insulator by a finite-element method. Before using the numerical predicition for the heat flux estimation, the code had been validated by an analytical solution of the heat equation in the cylindrical domain.

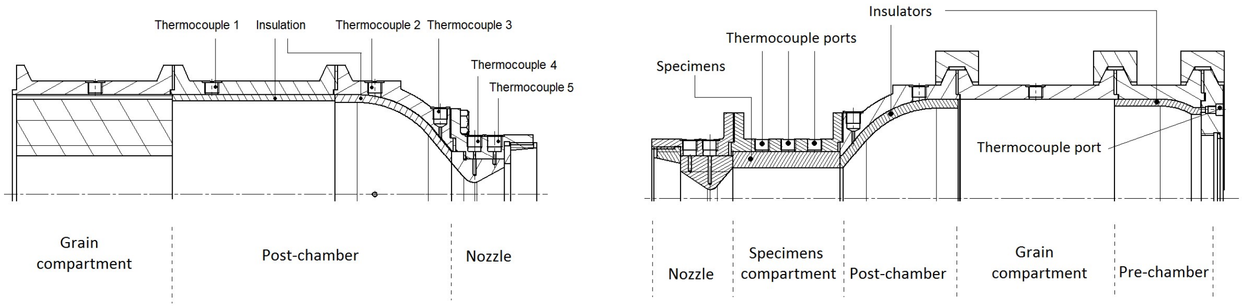
The results of the heat flux evaluation at various parts of the test-motor were in order of tens of kW/m^2 for the oxidizer-to-fuel ratio of 1.5, which was adopted for the initial tests to diminish the influence of ablation on the thickness of the insulator. It is necessary to note that the methodology adopted for the study would provide underestimated values of temperature on the external surface of the insulator in case of longer burning periods due to erosion of the insulator. The approach was used for the preliminary estimation of insulator geometry and subjected to correction after the determination of its ablation rates.

Another configuration of the test-motor was used for the measurement of ablation rates. Specimens with the shape of an annular sector were fixed in a specially designed extension (Figure 8, a) installed between the post-chamber and nozzle insert (Figure 7, b). Numerical simulations were conducted to estimate the average velocity of combustion gas and static temperature for a given configuration of the test-motor resulting on 100-110 m/s and 3300 K, respectively. The ablation rate was measured after comparing the thickness of the insulator before and after a predetermined firing time. The average thickness was considered for the evaluation of ablation rates since the residual thickness was not uniform, increasing downeards (Figure 8, b). The insulators of the pre-and post-chambers were manufactured from the same compositions as the tested specimens (Figure 8, c).

The fire tests showed that the ablation rates of non-reinforced resins and rubbers were of the same order as a regression rate of some solid fuels, i.e., in the range of 0.3 – 0.8 mm/s depending on the test conditions. From the specimens tested in conditions of oxidizer's excess, the polyurethane composition had the greatest ablation rates in the range of 0.82 – 0.83 mm/s in contrast with the silicone composition with the least ablation rates in the range of 0.33 – 0.46 mm/s. The lowest and highest values in the range corresponding to the burning time of 6 and 10 s respectively. Presumably, the

DOI: 10.13009/EUCASS2022-4879

burning time affected the ablation rate because of the inertial nature of the heat transfer to the specimens with different decomposition temperatures and hardness.



a) b)

Figure 7. The configurations of the test-motor adopted for the study: a) arrangement of the thermocouples; b) the position of extension for the ablation specimens (specimen compartment)

a) b) c)

Figure 8. Specimens of the silicone compositions fixed in the extension of the test-motor before (a) and after burning (b) and the bell-shaped silicone insulator in the post-chamber (c)

Moreover, a 6-second fire test ablation of polyurethane showed the effect of gravity on the ablation rate. The bottom sector of the polyurethane specimen had the least residual thickness in comparison to other parts of the insulator. Before expensive engine testing of the fibre-reinforced silicone insulators, its thermal insulation efficiency had been investigated using an oxyacetylene torch [25]. Slicone rubber was reinforced with glass, carbon, ceramic, and silica fibres. The objective of the preliminary study was to compare the back-face temperature of the plane silicone specimens, whose frontal face was subjected to oxyacetylene flame for over 40 s. The methodology of the ASTM standard [26] with some alterations was used as a reference for the study.

Silicone composites reinforced with E-glass, carbon, silica, and ceramic fibres with a perpendicular orientation of the plies in relation to flame direction demonstrated almost the same insulation efficiency subjected to a heat flux of 114.5±24 kW/m2 provides by the oxyacetylene torch during 40 s. It became obvious that the low-cost glass-fibre reinforced silicone composites could be used for thermal protection of hybrid propellant test-motor for the laboratory long firings tests without significant losses in their efficiency.

The ablation efficiency of the glass-fibre reinforced silicone composites was checked during 14 *s* burning test with the use of the test-motor configuration shown in Figure 7, b. The ablation rates of silicon reinforced by fiberglass mat (1.14 g/cm^3), and those reinforced with fabric and plies parallel (1.48 g/cm^3) and perpendicular (1.55 g/cm^3) to the flux were 0.51, 0.39 and 0.13 mm/s, respectively. These data helped to predict the thickness and orientation of the insulators in the test-motor for the 40 *s* burning test. The test was realized with a propellant grain made of high-density polyethylene (HDPE). The pre-chamber was isolated with a silicone insulator reinforced by a fiberglass mat, while the silicone insulator of the post-chamber was reinforced by fiberglass fabric. Both insulators were fabricated by a manual layup with the use of steel molds. The fire test was interrupted at 41 *s* due to overheating of the graphite nozzle (above 750°C). Graphite is a material with a high decomposition temperature and ablation resistance, but it is a good heat conductor too. Since the thermal insulator was not foreseen for the graphite insert the steel structure of the nozzle could be damaged. However, the temperature behind the post-chamber thermal insulator was at an acceptable level (below 50°C). A relatively high temperature of 324°C was measured at the injection plate. Initially, it was expected that the

DOI: 10.13009/EUCASS2022-4879

injection plate would be cooled by the low temperature of the issuing nitrous oxide. This observation required a more detailed thermal analysis of this part of the motor. During the operation of the test motor, the peak pressure of 40 bar was recorded (Figure 9, a). At the end of the burn, the pressure decreased to 13 bar, caused by the ablation (increase in diameter) of the nozzle's critical section. However, the thrust increased because of the increase in the area of the burning surface of the propellant grain (Figure 9, b).

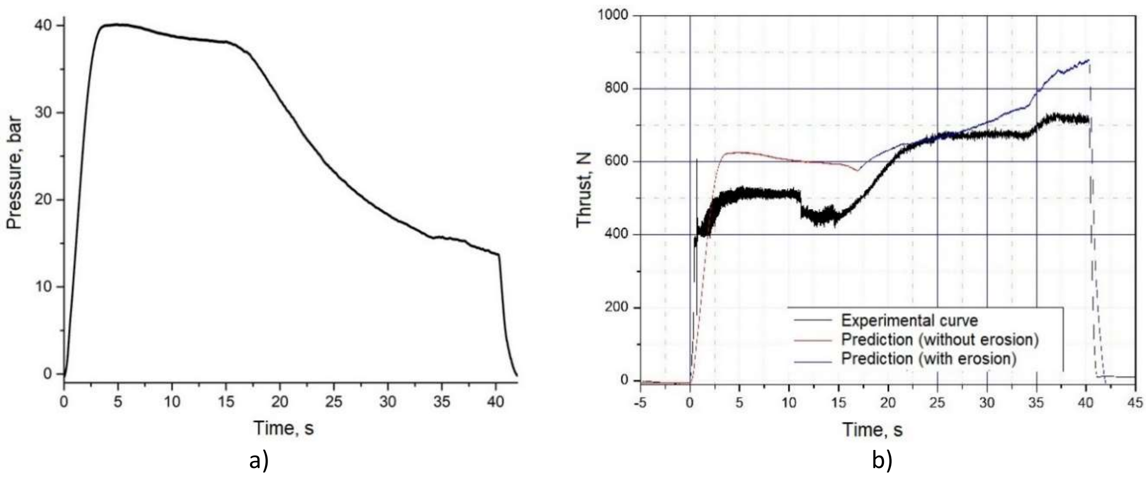


Figure 9. The measurements during the 41 s fire test: a) pressure inside the combustion chamber, b) thrust

The main objective of extending the burning time of a low thrust hybrid propellant motor was achieved with a thermal insulator based on silicone rubber reinforced by glass fibres. The experience gained and the associated equipment built for the studies would accelerate the development of more effective insulators based on carbon or ceramic fibres.

4. Injectors Design and Characterization

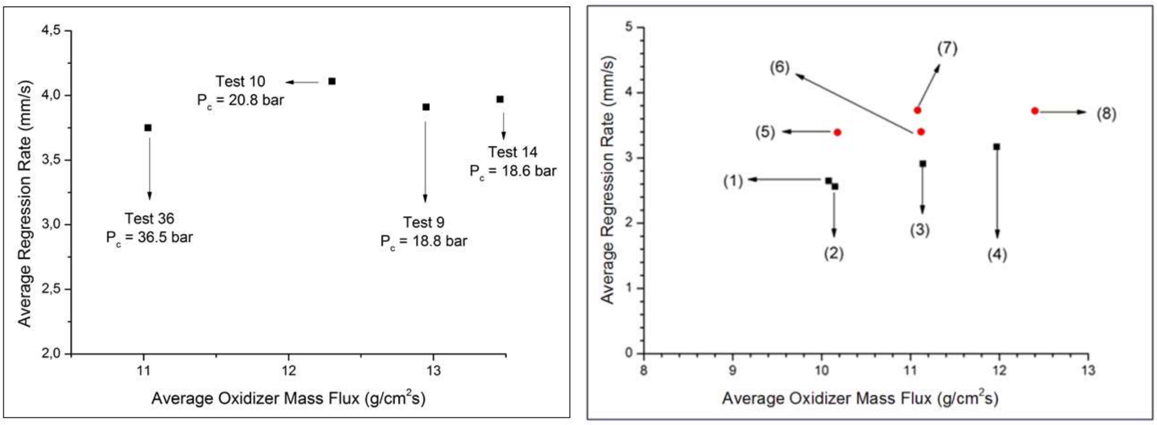
The injector characteristics play a substantial role in hybrid rocket motors because the flow profile can significantly affect the overall behaviour of the motor in terms of thrust, combustion efficiency, and combustion instability. Hybrid rockets can potentially use any injector initially designed for liquid engines, with required modifications due to the absence of one propellant component [27]. Usually, for many laboratory-scale applications, the showerhead injector is the common choice due to the simplicity of design and manufacturing, and also because it is largely spread in hybrid rocket community thus allowing system performance comparisons among research groups around the world.

However, showerhead design is not the optimized solution for the atomization process and combustion efficiency [27]. Experiments were carried out at the University of Brasília to evaluate the impact of the injector on the regression rate of paraffin-based fuels. The pressure swirl atomizer was applied in a lab-scale hybrid rocket (~ 200 N), based on N_2O–paraffin propellants, in order to study the effect of the injector system on the solid fuel regression rate.

Based on a substantial series of experimental data using the pressure-swirl injector (PSW), it was noticed that a 20% increase in paraffin regression rate in comparison with showerhead (SH). The effect of the chamber pressure on the regression rate was observed (Figure 10, a), but grain length (Lg) had little influence on that parameter (Figure 10, b) [28]. In Figure 10b, the solid fuel grain lengths are: (1) Lg = 160 mm; (2) Lg = 161 mm; (3) Lg = 161 mm; (4) Lg = 169 mm; (5) Lg = 133 mm; (6) Lg = 128 mm (7) Lg = 131 mm (8) Lg = 134 mm. Table 3 summarizes the results, where $\bar{r}$ is the solid fuel average regression rate and $\bar{G}$ is the oxidizer average mass flux.

Two additional tests were conducted to study the upper limit of initial oxidizer mass flux on regression rate. The motor was operated with an initial mass flux value of 157.9 g/cm^2s, that is 2.8 times higher than the value suggested by [29]. For this initial oxidizer mass flux, we found a regression rate of 9.95 mm/s for a 3 second test. These values for the regression rate need to be analysed carefully. Since the oxidizer mass flux was very high and the burning time was short (around 3 seconds), and as the regression rate tends to be high in the first instants of motor operation, the data presented here just illustrates this behaviour.

DOI: 10.13009/EUCASS2022-4879



a) b)

Figure 10. a) Effect of chamber pressure on the average regression rate, b) Effect of grain length over the average regression rate

Table 3: Comparison between the injection system, PSW: pressure-swirl, and SH: showerhead

Test	Injector	$\overline{G}\ (g/cm^2s)$	$\overline{r}\ (mm/s)$	O/F
10	PWS	12.30	4.11	3.3
32	PWS	13.70	4.18	3.2
33	PWS	15.25	4.66	3.1
52	SH	14.98	3.30	4.7
53	SH	16.20	3.05	5.6
54	SH	15.36	3.10	5.1

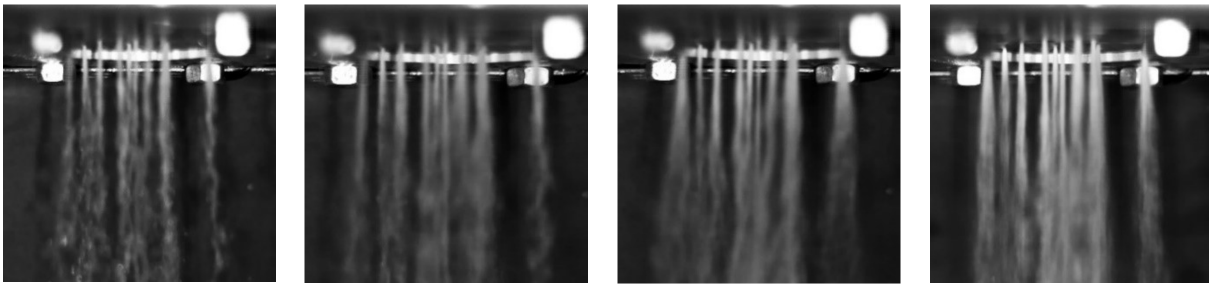
Based on the results of this research the University of Brasília Chemical Propulsion Laboratory (UnB/CPL) and the Université libre de Bruxelles Aero-Thermo-Mechanics Department (ULB/ATM) carried out cold and hot conditions tests with four types of injectors designed for hybrid rocket motors, which were: showerhead (SH), hollow-cone (HC), pressure-swirl (PSW) and vortex atomizer (VOR). The injectors were designed based on the theoretical parameters of the ULB-HRE motor and SARA motor to deliver 550 g/s based on an injector pressure drop of 25 bar.

Each element of the SH injector has a 7 mm length and orifice diameter of 1.4 mm and it is used as a reference to evaluate the performance of the other types of injectors. Figure 11 shows the flow pattern of the water as a function of the pressure drop, in the range of 10 bar to 40 bar. Figure 11 shows the change in the flow structure before and after 20 bar. The Nitrous oxide behaviour at 45 bar is illustrated in Figure 12 and, because of the fast expansion of the N_2O at ambient temperature and pressure, a clear separation between the liquid layer and the gas plume is not visible.

The pressure-swirl injector plate (PSW) was composed of 6 individual PSW elements, to deliver the same oxidizer mass flow rate of 550 g/s. However, to characterize the injector under cold conditions it was used an individual PSW element set down in the centre of the PSW injector plate. Figure 13 shows the flow pattern of the water as a function of the pressure drop while Figure 14 shows the patterns for nitrous oxide.

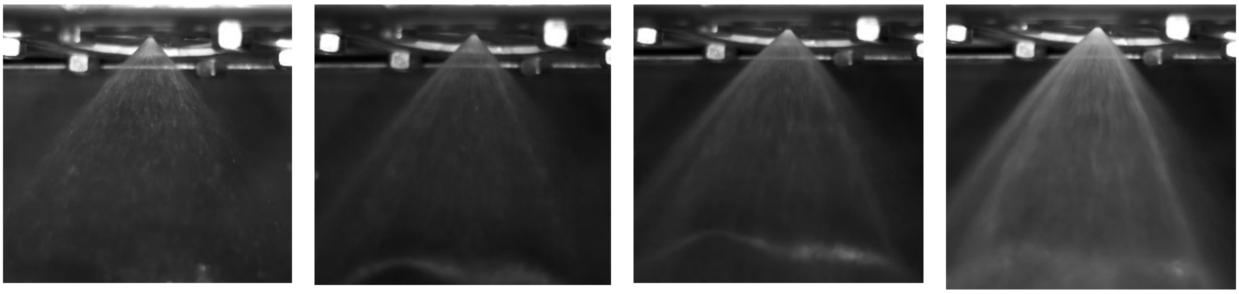
Radial axial injection, VOR, was composed of 6 orifices with 45° degrees of inclination from the injector plane. Figure 15 shows the evolution of water flow with injector pressure difference from 10 to 30 bar. The results with Nitrous oxide are presented in Figure 16.

DOI: 10.13009/EUCASS2022-4879



a) 10 bar b) 20 bar c) 30 bar d) 40 bar

Figure 11. Showerhead injector water flow pattern – pressure from 10 to 40 bar



a) 10 bar b) 20 bar c) 30 bar d) 40 bar

Figure 13. Water discharged through a pressure swirl injector from 10 to 40 bar

a) 10 bar b) 20 bar c) 30 bar

Figure 15. Water discharged through vortex injector from 5 to 35 bar using HSC

Figure 12. Showerhead injector using N_2O

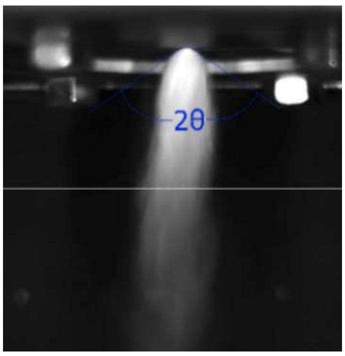


Figure 14. Liquid N_2O discharged through a pressure swirl injector

Figure 16. Liquid N_2O discharged through vortex injector

DOI: 10.13009/EUCASS2022-4879

The flow pattern was firstly investigated by the use of a high-speed camera with a sample rate of 5000 fps. In this case, we noticed a considerable discrepancy between the flow pattern of water and nitrous oxide as duty fluid, mainly with the PSW. Using SH and VOR, however, the water stream profile was more similar to that of N_2O. The higher influence on the injector's discharge coefficient due to pressure was in the SH and the lower was in the PSW and VOR.

A more detailed analysis was carried out with the pressure-swirl atomizer to estimate the spray-semi angle and the Sauter Mean Diameter (SMD). The results with water gave 43° and 40.71 µm at 40 bar, respectively [30]. Using the laser scattering we also obtained the droplet size distribution of the spray. The values of the spray semi-angle, SMD, and the droplet size distribution also showed good agreement with the literature [31], these results for the nitrous oxide were inconclusive due to the optical concentration in the laser detector.

5. Combustion Instability in Hybrid Rocket Motors

Preliminary studies of the combustion instabilities from UnB/CPL have indicated the influence of the combustion chamber design parameters on the combustion instabilities of hybrid rocket systems. Therefore, a detailed investigation of the effects of pre-combustion chamber configuration on the feed-system coupled instabilities in hybrid systems was performed both theoretically and experimentally [32-34].
A methodology to analyse the feed-system coupled instabilities in hybrid rockets was developed, and an extended mathematical model for the instability related to the coupling between the combustion chamber and feed system was introduced, including the parameters of the pre-combustion chamber residence time, the gas residence time, the injector pressure drop, the combustion time lag, and the O/F ratio. The stability limit analysis was performed using the root locus method, and the sensitivity of the system to the variation of some design parameters of the engine was analyzed. A correlation to predict the instability frequencies was proposed, which is expressed as the pre-combustion chamber residence time (τ_{pre}), as shown in Equation 1. Firing tests were conducted to validate the analysis methodology for the feed-system instabilities and the frequency prediction model (Figure 17) at the University of Brasília (Brazil) and the Free University of Brussels (Belgium) campaigns.

$$f = \frac{1.73}{\tau_{pre}^{0.6}} = 1.73\left(u_{pre} / L_{pre}\right)^{0.6} \quad (1)$$

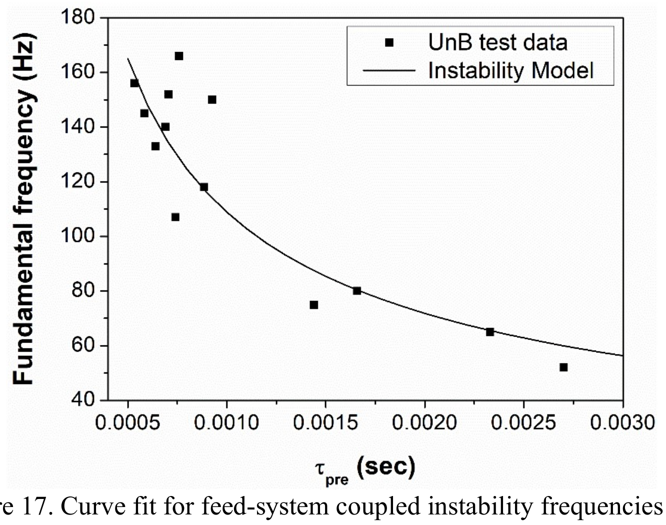


Figure 17. Curve fit for feed-system coupled instability frequencies [34].

The stability analysis based on the proposed extended model could be an effective method to predict the condition of stable and unstable regimes in a typical hybrid motor configuration. The analysis confirmed that parameters such as the ratio of the injector pressure drop to the operating chamber pressure, the gas residence time, the precombustion chamber residence time, and the combustion time lag affect the pressure oscillation amplitude and the frequency of the feed-system coupled instabilities for hybrid rockets. The large injector pressure drop suppressed the feed-system coupled instabilities. The injector design is also a key element for the stability of the system, however, this influence is much less important to the feed-system coupled instability under a large pressure drop. The long pre-combustion

DOI: 10.13009/EUCASS2022-4879

chamber contributed to the stabilization of the system. The pre-combustion chamber residence time which is a function of the pre-combustion chamber length and the local flow velocity in Equation 1 plays an important role in the frequency response of the feed-system coupled instability in hybrid rocket engines, whereas the combustion time lag had a small effect on the oscillation frequency. It is worth noticing that hybrid rocket engines show a shift in the feed system instability frequencies over time, a particular characteristic not observed in liquid rocket engines [34].

6 Other Projects

6.1 Ramjet motor

There are two ramjet motors developed in the Laboratory: solid-fuel ramjet (SFRJ) and liquid-fuel ramjet. The SFRJ [35], [36] (Figure 18) operates on paraffin-polyethylene grain and has a thrust level of 120 N. It is fed from a compressed air tank and utilizes a chemical heater (combustor) to simulate the flow conditions after the supersonic diffuser for a fiven flight regime. In the solid-fuel configuration, the fuel grain is inserted into the cavity of the engine, operating as the combustion chamber itself. This setup is inherently simple and absent fuel lines, pumps, and injectors. Since the tank's pressure falls during test execution, means were provided to stabilize it at the combustor entrance. An effective pressure control algorithm was designed to maintain a prescribed air pressure upstream of the motor. The air heater section has thermocouples for ignition detection, hot air temperature measurements, and a pressure sensor used to control the primary airflow to the ramjet motor. Heating of the air occurs in the first part of the motor, separated from the main combustion chamber by a diaphragm similar for better mixing quality. The diaphragm (flame holder) causes a recirculation zone inside the combustion chamber for flame stabilization. The motor provides a thrust modulation by the grain design, for example, using an additive manufacturing technology. A combination of the experimental and numerical techniques was employed to obtain the thermodynamic properties inside the combustion chamber, including the combustion products. To our knowledge, University of Brasilia was the first institution to operate, in regular bases, ramjet using solid paraffin. The results were presented at the Student Poster Competition during the 11th Wernher von Braun Memorial Symposium held in Huntsville, Alabana, 2018. Two students from the CPL-UnB team were awarded with the second prize in that competition.

Figure 18. Ramjet test stand with engine operation in nominal regime

6.2 Development of Fuels for Hybrid Rockets

Many recent Hybrid Rocket Engines (HREs) use paraffin as the main component of the fuel grain, because of the low cost and high regression rate advantages. However, pure paraffin has poor structural characteristics and sometimes low performance due to the fuel's internal ballistics behaviour. The UnB/CPL team and the ULB/ATM are studying ways to address the problem by the use of metal particles, and additive manufacturing, and its impact on the fuel regression rate, O/F ratio, specific impulse, and the overall performance of the engine.

In HREs, the fact that the fuel is in the solid phase makes it very easy to add solid performance-enhancing materials such as Magnesium-Diboride (MgB_2). This enables the hybrid rockets to gain a specific impulse (*Isp*) and density advantage over a comparable hydrocarbon-fuel liquid system. Theoretically, any additive used in solid proposition motors can be applied in hybrids.

DOI: 10.13009/EUCASS2022-4879

While metal additives such as aluminium can increase the nozzle erosion, certain metal hydrides, such as magnesium hydride, can reduce this effect, by decreasing the oxidising species in the nozzle. Additives that affect the material strength can be added to improve the structural performance of the fuel grain if it is prone to failure, but in turn, they may increase the melt layer viscosity, which can reduce entrainment and thus regression rate [37].

6.3 Final Considerations

The University of Brasília has a long history in aerospace propulsion mostly focused on hybrid propellant rocket engines. During the last decades,' the Energy and Environment Laboratory (LEA) and more recently the Chemical Propulsion Laboratory (CPL) carried out research in many different fields such as rocket engine design, experimental techniques for rocket propulsion, and performance analysis. In several aspects, the University of Brasília has been at the forefront of developing hybrid propulsion in the Americas.

Nowadays, hybrid propulsion is the main activity in rocket propulsion at UnB and, from the 2000s to now, various projects have been developed and fomented by government actors (such as CNPq, FAPDF, AEB, and others). These activities imply a large impact on the scientific production as many journals' papers, patents, and products. The team has cooperation with private companies, universities, and research institutions overseas.

References

[1] Bouziane, M., Bertoldi, A. E. M., Hendrick, P., Lefebvre, M. 2021. Experimental investigation of the axial oxidizer injectors geometry on a 1-kN paraffin-fueled hybrid rocket motor. *Journal of FirePhysChem*. Vol.1, Issue 4: 231-243. https://doi.org/10.1016/j.fpc.2021.11.012

[2] Mazzetti, A. Merotto, L., Pinarello, G. 2016. Paraffin-based hybrid rocket engines applications: A review and a market perspective. Jounal ActaAstronautica. Vol. 126: 286-297. https://doi.org/10.1016/j.actaastro.2016.04.036

[3] Viegas, F.L., Salemi, L.C. 2000. Design and Construction of a Test Bench for Hybrid Rocket Static Firings and its Hybrid Engine. Under-graduate report. Mechanical Enginering Department, University of Brasilia. *In Portuguese.*

[4] Santos, L.M.C., Almeida, L. A. R., Fraga, A. M., Veras, C. A. G. 2004. Experimental Investigation of a Paraffin Based Hybrid Rocket. In: *10th Brazilian Congress of Thermal Sciences and Engineering* – ENCIT.

[5] Santos, L.M.C., Contaifer, R., Bertoldi, A. E. M., Medeiros, E. G., Sousa, L. C. P. C., Borges, A., Veras, C. A. G. 2005. Regression Rate Studies of a Paraffin-Based Hybrid Rocket. In: *18th International Congress of Mechanical Engineering*.

[6] Santos, L.M.C., Almeida, L. A. R., Veras, C. A. G. 2005. Design and Flight Test of a Paraffin Based Hybrid Rocket. In: *18th International Congress of Mechanical Engineering*.

[7] WordPress. 2022. Hybrid Propulsion Team History – University of Brasília. Access: 06/06/2022 (*https://hybridteam.wordpress.com/history/*).

[8] Shynkarenko, O.; Andrianov, A.; Bertoldi, A.E.M. 2015. Low-thrust hybrid motor efficiency research for design optimization purposes. *In: Proceedings of the 51st AIAA/SAE/ASEE Joint Propulsion Conference*, Orlando, FL, USA.

[9] A. Andrianov, O. Shynkarenko, A. E. M. Bertoldi, Jr. M. N. D. Barcelos. 2015. Concept and Design of the Hybrid Test-motor for the Development of a Propulsive Decelerator of SARA Reentry Capsule. *In: 51st AIAA/SAE/ASEE Joint Propulsion Conference*, Orlando, USA.

[10] O. Shynkarenko, D. Simone. 2020. Oxygen-Methane Torch Ignition System for Aerospace Applications, MDPI Aerospace, August.

[11] O. Shynkarenko, D. Simone, J. Lee, A. E. M. Bertoldi. 2022. Experimental and Numerical Study of the Flammability Limits in a CH4/O2 Torch Ignition System, Energies.

[12] Andrianov, A., Lee, J., Shynkarenko, O., Simone, D. and de Morais Bertoldi, A.E., 2019. "Experimental study of severity level of structural discontinuities in paraffin grains of hybrid propellant rocket". Acta Astronautica.

[13] Azevedo, V. A., Shynkarenko, O., Veras, C.A.G., Alves, I.P.M. 2019. Experimental Investigation of High Regression Rate Paraffin for Solid Fuel Ramjet Propulsion, *In: AIAA Energy and Propulsion Forum*, Indianapolis, USA.

[14] O. Shynkarenko, V.A. Azevedo, C.A.G. Veras, I.P.M. Alves. 2019. Experimental Investigation of Hydrocarbon Based Fuels in Solid Fuel Ramjet Propulsion. *In: 70th International Astronautical Congress* (IAC), Washington, USA.

[15] K. N. Silva. 2019. Desenvolvimento da Válvula de Controle de Vazão para Motor de Foguete Híbrido, B.Sc. Thesis, University of Brasilia.

DOI: 10.13009/EUCASS2022-4879

[16] The National Aeronautics and Space Administration, NASA SP-8093. 1977. Solid rocket motor internal insulation. NASA-Langley, 116 p.

[17] Nataly, M.; Kenny, J.M.; Torre, L. 2016. Science and technology of polymeric ablative materials for thermal protection systems and propulsion devices: A review. Progress in Material Science, v. 84, p. 192-275.

[18] Schmidt, D.L. 1969. Ablative polymers in aerospace technology. Journal of Macromolecular Science: Part A – Chemistry, v. 3, n. 3, p. 327-365.

[19] A Nunes, GP Milhomem, VC Rispoli, A Andrianov. 2017. Application of an Inverse Method for the Estimation of Heat Flux in Low-Thrust Hybrid Propellant Rocket Motor and Its Analytical Validation - XXXVIII CILAMCE, Florianópolis.

[20] Gabriel Pachêco Milhomem, Ana Paula Castro de Paula Nunes, Artem Andrianov, Olexiy Shynkarenko, Jungpyo Lee. 2017. Experimental Studies of Heat Insulation Materials for Hybrid Propellant Rocket Motors, COBEM-1706.

[21] Donskoy, A., A. 1996. "Elastomeric heat-shielding materials for internal surfaces of missile engines". International journal of polymeric materials and polymeric biomaterials, Vol. 31, Is. 1-4, p. 215 – 236.

[22] Koo, J.H., Miller, M.J., Weispfenning, J., Blackmon, C. 2011. "Silicone polymer composites for thermal protection system: fiber reinforcements and microstructures". Journal of composite materials, Vol. 45, Is. 13, p. 1363 – 1380.

[23] CHAPMAN, A.J. 1966. Evaluation of several silicone, phenolic, and epoxy base heat-shield materials at various heat transfer rates and dynamic pressures. Washington, D.C.: NASA.

[24] Beck, J. 1970. Nonlinear estimation applied to the nonlinear inverse heat conduction problem. International Journal of Heat and Mass Transfer, vol. 13, n. 4, pp. 703-716.

[25] Andrianov, A.; Lee, J.; Possa, G.; Silva, H.O., 2020. Experimental Study of The Insulating Effectiveness of Silicone Rubber Composites by Oxyacetylene Ablation Testing. JATM, vol. 12.

[26] American Society for Testing and Materials. 2008. ASTM E285 – 08: Standard test method for oxyacetylene ablation testing of thermal insulation materials. West Conshohocken.

[27] Bouziane, M., Bertoldi, A.E.M., Milova, P., Hendrick, P., Lefebvre, M. 2019. Performance comparison of Oxidizer Injectors in a 1-kN Paraffin-Fueled Hybrid Rocket Motor. *Aerospace Science and Technology*. 89: 392-406.

[28] Bertoldi, A. E. M., Veras, C. A. G., Hendrick, P. 2017. Experimental Evaluation of Pressure-Swirl Injection System Over Solid Fuel Regression Rate in Hybrid Rockets. In: *7th EUCASS European Conference for Aeronautics and Space Sciences.*

[29] Colburn, W. H. 2005. A Manual for Hybrid Propulsion System Design. Edited by Steven Zwaska.

[30] Bouziane, M., Bertoldi, A.E.M., Lee, D., Milova, P., Hendrick, P., Lefebvre, M. 2017. In: *7th EUCASS European Conference for Aeronautics and Space Sciences.*

[31] Lacava, P. T., Bastos-Neto, D., Pimenta, A. P., 2004. Design Procedure and Experimental Evaluation of Pressure-Swirl Atomizers. In: 24th International Congress of the Aeronautical Sciences, ICAS.

[32] Bertoldi, A. E. M. Bouziane, M., Veras, C. A. G., LEE, J., COSTA, P. D. M. V. C. 2018. Experimental Investigation of the Feed System Instabilities in Hybrid Rocket Motors*. In: 69th International Astronautical Congress,* 2018, Bremen.

[33] De Morais Bertoldi, A. E., BouzianeE, M., Lee, J., Veras, C. A. G., Hendrick, P., Simone, D. 2019 . Theoretical and Experimental Study of Combustion Instability in Hybrid Rocket Motors. *In: 8th European Conference for Aeronautics and Space Sciences*, Madri.

[34] Lee, J., De Morais Bertoldi, A. E., Andrianov, A., Borges, R. A., Veras, C. A. G., Battistini, S., Morita, T., Hendrick, P. 2020. Role of Precombustion Chamber Design in Feed-System Coupled Instabilities of Hybrid Rockets. JOURNAL OF PROPULSION AND POWER, v. 36, p. 796-805.

[35] Azevedo, V.A., Shynkarenko, O., Veras, C.A.G., Alves, I.P.M. 2019. Experimental Investigation of High Regression Rate Paraffin for Solid Fuel Ramjet Propulsion, *2019 AIAA Energy and Propulsion Forum, Indianapolis*, USA.

[36] Shynkarenko, O., Azevedo, V.A., Veras, C.A.G., Alves I.P.M. 2019. Experimental Investigation of Hydrocarbon Based Fuels in Solid Fuel Ramjet Propulsion, *70th International Astronautical Congress* (IAC 2019), Washington, USA.

[37] Veale, K., Adali, S., Pitot, J., Brooks, M. 2017. A review of the performance and structural considerations of paraffin wax hybrid fuel with additives. Acta Astronautica.