

# COB-2023-1931

## PRELIMINARY DESIGN OF A 25 kN PRESSURE-FED LIQUID PROPELLANT ROCKET ENGINE

### Maurício Sá Gontijo

Aeronautics Institute of Technology  
mauricio.sa.gontijo@gmail.com

### Renato de Brito do Nascimento Filho

#### Anieli Juliana Schrammel

DeltaV Engenharia Espacial  
renato.nascimento@deltavengenharia.com , anieli.schrammel@deltavengenharia.com

### Olexiy Shynkarenko

#### Jungpyo Lee

#### Artur Elias de Moraes Bertoldi

University of Brasília  
olexiy@aerospace.unb.br , jpleerocket@gmail.com , bertoldi@unb.br

**Abstract.** *Pressure-fed liquid propellant rocket engines (LPRE) are the simplest configuration of this type of propulsion. Based on this simplicity and on the, consequently, lower costs, it is a natural way to be the first development of a propulsion company. This work consists of a preliminary design of a 25 kN Liquid Oxygen/Ethanol LPRE under pressure-fed condition that is being designed to be used on the new training rocket for launch centers, supported by the Brazilian government through the MCTI (Ministry of Science, Technology and Innovation), AEB (Brazilian Space Agency), FINEP (Financier of Studies and Projects) and FNDCT (National Fund for Scientific and Technological Development), and executed by DeltaV Engenharia Espacial. Training rockets are intended to be a way for the launch centers to keep their personnel trained to deal with any type of rocket. Brazil already has two types of training rockets, both solid propulsion based. With the intention of providing training for liquid rockets, this project was conceptualized. This rocket will be able to supply the launch center teams with experience on handling and operating liquid propellants and high-pressure systems, on tracking and on recovering the vehicle and on ground auxiliary devices operation. In addition, the project must have relatively low cost and the performance is not the one of the major concerns. With that in mind, the pressure-fed system was selected and it also uses ablative cooling, allied with film cooling. With the preliminary design presented in this work, further developments and improvements will be made, followed by tests, integration and prototype flight. For this project, a partnership was consolidated with the experienced team of the Chemical Propulsion Laboratory (CPL) from the University of Brasília.*

**Keywords:** *Liquid Propellant Rocket Engines, Pressure-Fed, Sounding rocket, Training Rocket.*

## 1. INTRODUCTION

Since 2009, Brazilian launch centers (CLA - Alcantara Launch Center and CLBI - Barreira do Inferno Launch Center, acronyms in Portuguese), in order to maintain its personnel under constant practice and to qualify new human resources, operates training rockets. These family of rockets are called FOGTREIN (Training Rockets, in Portuguese), which consists of only two vehicles: FTB and FTI (Basic Training Rocket and Intermediate Training Rocket, respectively in Portuguese) AEB (2009). Both rockets are solid propulsion based and were developed by Avibras Indústria Aeroespacial, derived from its SS30 and SS60 ASTROS rockets, respectively.

In 2022, MCTI, FINEP, AEB and FNDCT launched an funding public notice to support a prototype development of a new training rocket with maximum R\$ 8M and within 3 years (MCTI, 2022). DeltaV Engenharia Espacial was selected as the company responsible for this development. As a first requirement stated by the company, although on the funding public notice there was no such requirement, the rocket must use liquid propulsion. This was defined since the Brazilian launch centers already has two solid training rockets and liquid training rockets provides several other possibilities of operational procedures to be practiced by the personnel.

This type of rocket has the same safety requirements as any other would have to launch on CLA or CLBI. However, its performance requirements aren't as strict as it is in other missions. With this in mind, some inherent complexities of

LPREs were attenuated without compromising the training or the safety of anyone involved. Thereat, a logical simplification was made by using a pressure-fed system. The usage of turbo-pumps or electric pumps would be impractical due to the budget and development time. Other attenuation was made by using ablative thermal protection and film cooling instead of using regenerative cooling, which would increase significantly the unitary cost and complexity of the project.

The company also stated that the engine should work with Liquid Oxygen (LOx) and Ethanol. This selection was made mainly due to the high experience of the Brazilian liquid propulsion program, until now developed only by the Institute of Aeronautics and Space (IAE, in Portuguese), with these propellants (Almeida *et al.*, 1999) (Torres *et al.*, 2009) (Almeida and Pagliuco, 2014) (Pfuetszenreuter *et al.*, 2017) (Almeida *et al.*, 2021). In addition, cryogenic handling training is possible with LOx, it is highly available, non-toxic, green and are extremely used propellants worldwide (or, at least, is similar, as the case of Ethanol being relatively similar to RP-1).

A key partnership was consolidated with CPL, which is composed by a highly competent and experienced team of researchers. Several improvements and developments were implemented due to this interaction.

## 2. METHODOLOGY

An adaptive optimization algorithm was used to design the engine. This algorithm composes a software called LPOT (Liquid Propulsion Optimization Tool) developed initially at Gontijo (2022b). This software optimizes an engine for its highest Thrust to Weight Ratio (TWR). Some adaptations were made for this project, such as the inclusion of impinging injectors, conical nozzles, thermal protections, collapse factor prediction and feed system coupled instabilities.

To calculate thermochemical and theoretical performance parameters, NASA's Chemical Equilibrium with Applications (CEA) software is used (Gordon and McBride, 1988) (McBride and Gordon, 1992) (McBride *et al.*, 1993) (McBride *et al.*, 1994) (Gordon and McBride, 1994) (Svehla, 1995) (McBride and Gordon, 1996) (McBride *et al.*, 2002). In addition, efficiency prediction models were implemented in order to reiterate all calculations and obtain a more realistic design (Vasil'yev and Kudryavtsev, 1969) (Volkov *et al.*, 1970) (Alemasov *et al.*, 1980) (Huzel and Huang, 1992) (Hill and Peterson, 1992) (Kudryavtsev, 1993) (Humble *et al.*, 1995) (Kessaev, 1997) (Dorofeev, 2010) (Ponomarenko, 2013) (Sutton and Biblarz, 1995).

## 3. DESIGN

For the optimization, the following assumptions were considered:

1. Ideal gas law;
2. Adiabatic system, except for the thermal protection (combustion chamber) and thermal isolation (LOx lines);
3. Isentropic and one-dimensional flow in the nozzle;
4. Mass and momentum conservation;
5. Finite area chamber;
6. Chemical equilibrium for the combustion;
7. No body forces;
8. Steady state;
9. Pressurant as ideal gas;
10. Propellants as incompressible liquids.

### 3.1 Thrust chamber

In this project, thrust chamber is defined as the set of the following components:

- Injection system;
- Combustion chamber; and
- Nozzle.

The designed one-half axisymmetric combustion chamber and nozzle contour is shown below in Fig. 1, directly obtained from the optimization algorithm:

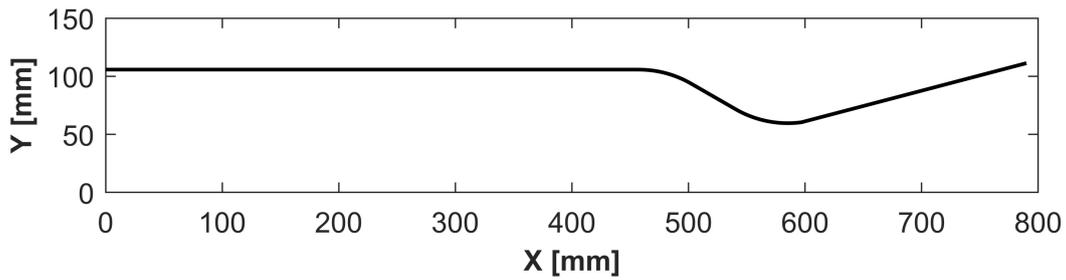


Figure 1: Optimized engine contour.

The chamber length is obtained through the droplet vaporization model developed by Spalding (Spalding, 1958) (Spalding, 1959) (Adler, 1959) (Gontijo *et al.*, 2020) (Gontijo *et al.*, 2021b) (Gontijo *et al.*, 2021a) (Gontijo, 2022a) and the injected droplet size was calculated with theoretical Sauter Mean Diameter (SMD) models for plain orifices (Lefebvre and McDonnell, 2007).

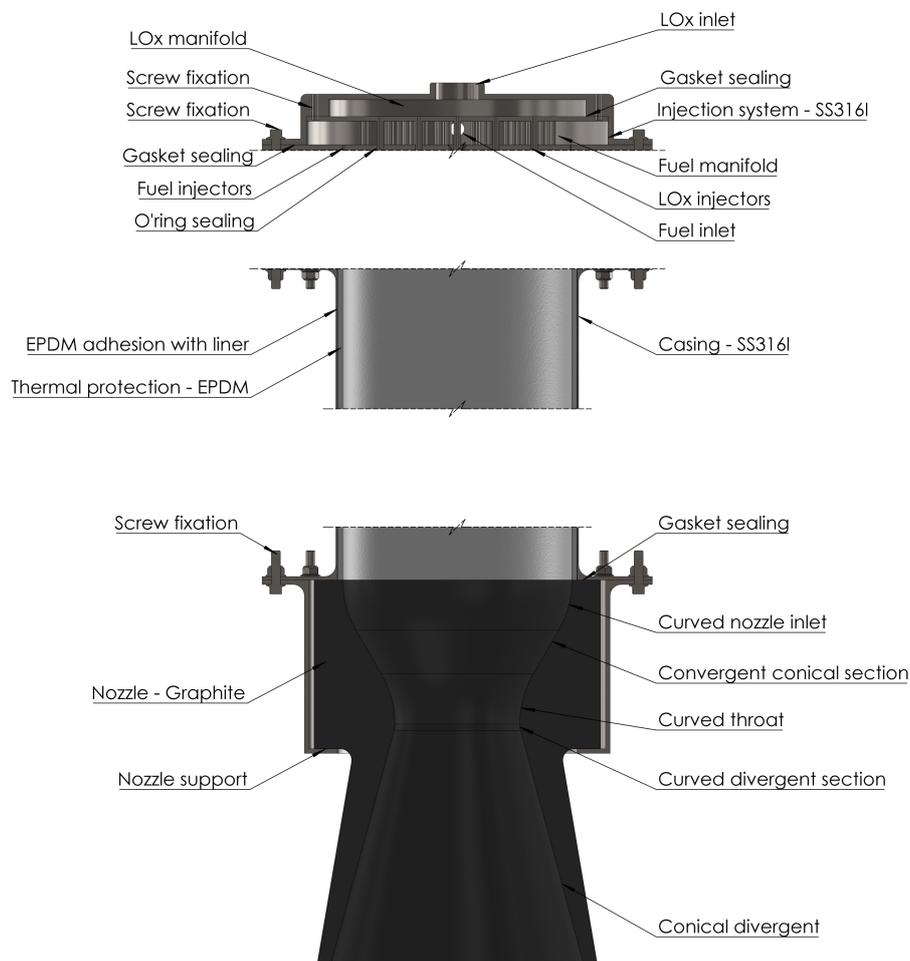


Figure 2: Thrust chamber schematic.

### 3.1.1 Injection system

The injection system is defined as the set of injection chamber (manifold) and the injection plate. The manifold was designed to equally distribute the propellants at the injection plate orifices. Therefore, the propellant must be evenly distributed in all orifices, with a pressure variation between the manifold inlet and the diametrically opposite orifices of 1% maximum. In Section 3.1.3 it is possible to observe a model of the manifold.

An uneven distribution in the fuel section is expected, since the inlet is made laterally. In the case that the simulation results show a higher than acceptable non-uniformity, it is recommended to add a diametrically opposite second inlet.

The manifold has an arrangement of holes distributed in 6 circles. These 6 circles are distributed in 3 regions. The first region is the central one, that is composed by four circles and the central injector, and has a mixture ratio closer to the

design point. The second region is composed by the fifth circle and is oxidizer rich, since the third region has only fuel injection. Finally, the third region is made up of film cooling injectors.

As shown in Fig. 3, the injectors in the first region have two orifices inclined at 35° for fuel for each central oxidizer orifice. The circles on the first region have 6, 12, 18 and 24 sets of bi-propellant injectors, beyond the core set. The second region has 16 sets with a fuel injector inclined at 35° and a oxidizer injector inclined at 12°, each. The third region has 18 fuel showerhead injectors for film cooling.

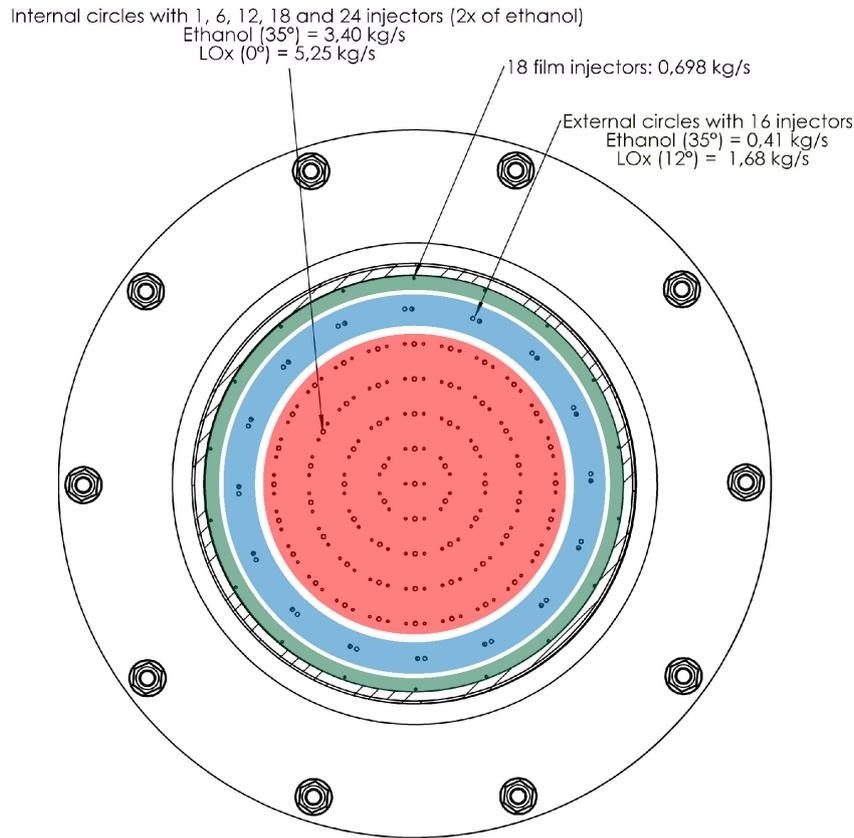


Figure 3: Injector regions.

Table 1 shows some geometrical parameters of the injectors shown in Fig. 3 and its SMD. The table is divided into Fuel and Oxidizer injectors for each column. Also, the hydraulic diameter is presented since, specially for the film cooling injectors, the orifice shape can be changed in future.

Table 1: Injector geometrical parameters and SMD.

Region	Hydraulic $\varnothing$ [mm]		Inclination [°]		# Injectors [-]		SMD [ $\mu\text{m}$ ]	
	Fuel	Ox.	Fuel	Ox.	Fuel	Ox.	Fuel	Ox.
First	1.20	2.10	35	0	122	61	300.3	623.1
Second	1.15	1.90	35	12	16	16	285.3	370.7
Third	1.40	-	0	-	18	-	-	-

As it is possible to observe above, no SMD values are informed on the third region, since there are only liquid film injectors on that region. The configuration shown in Tab. 1 and in Fig. 3 provides, relatively, high overall performance, low characteristic length, provides the desired propellant mass flow rate and also protects the chamber wall with a lower adiabatic flame temperature on second region (in comparison with the first region that has the optimum mixture ratio) and with the film cooling.

Figure 4 displays the dimensionless Reynolds, Weber and Ohnesorge numbers for the injectors of the first two regions shown in Fig. 3 and Tab. 1. These numbers are important in order to design high performance injectors and relates the inertia, viscous and surface forces ( $Re = \rho U D / \mu$ ;  $Oh = \mu / \sqrt{\rho \sigma D}$ ;  $We = \rho U^2 D / \sigma$ ; where  $\rho$  is the density,  $\mu$  dynamic viscosity,  $\sigma$  surface tension and  $U$  injection velocity -  $U = \sqrt{\Delta P / \rho}$ , where  $\Delta P$  is the pressure drop along the injector).

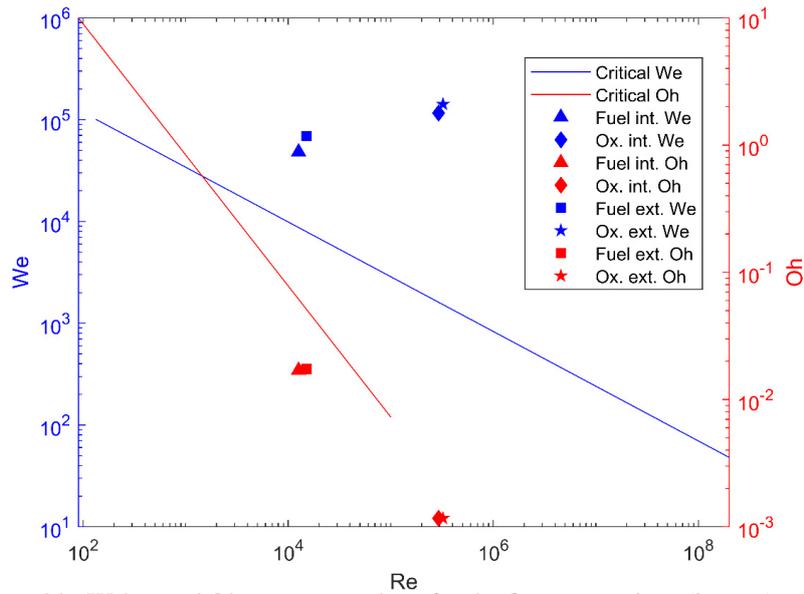


Figure 4: Reynolds, Weber and Ohnesorge numbers for the first two regions (int. = 1st & ext. = 2nd).

The We Vs Re Vs Oh chart above is displaying the critical lines in order to obtain atomization, and practically skip the breakups (Schmidt, 1984) (Saeedipour *et al.*, 2014) (Wang, 2016) (Gontijo and Shynkarenko, 2020) (Gontijo, 2022b). In general, the We, Re and Oh numbers should be above its critical lines, otherwise bad atomization is achieved, a larger chamber is required and combustion instabilities are more susceptible (Hill and Peterson, 1992) (Gontijo *et al.*, 2020) (Gontijo, 2022a). As it is possible to analyze in Fig. 4, the Ohnesorge numbers are close to the critical line for atomization, which is acceptable. In the case of the Weber numbers, the results are much better. Therefore, this set of results are satisfactory. Better results could not be obtained due to manufacturing limitations. For the film cooling injectors, the design point is defined in the Figs. 5a and 5b below, with 18 fuel orifices. Mainly, the recommendations made by Stechman *et al.* (1969) were followed to design the film cooling.

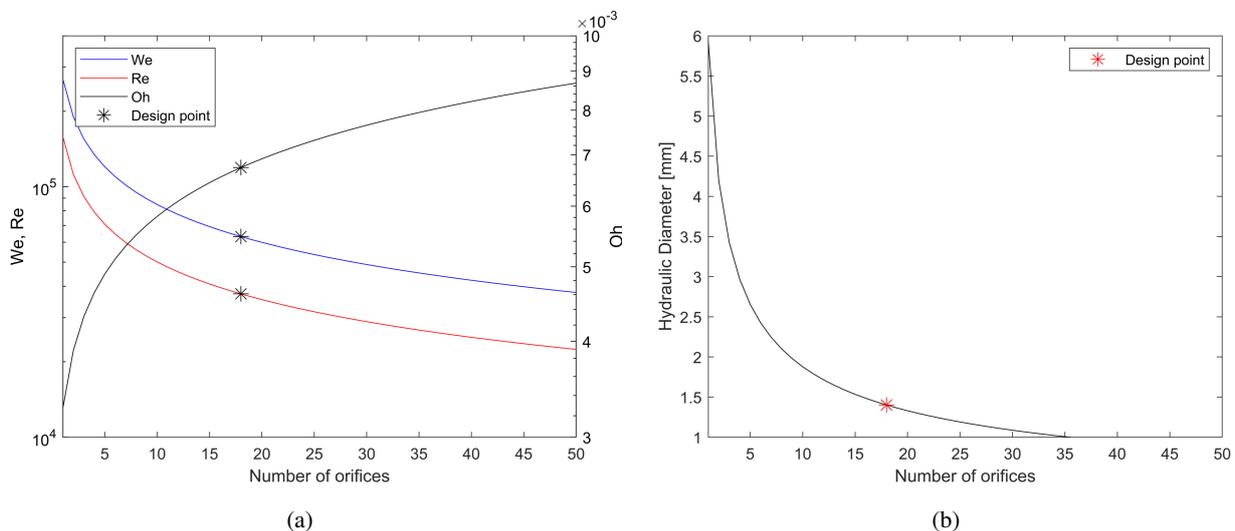


Figure 5: (a) Reynolds, Weber and Ohnesorge numbers for the film injectors, (b) Hydraulic diameter variation with the number of injectors.

With the above analyses, a hydraulic diameter could be calculated in order to obtain a liquid film that protects the chamber wall. In this case, atomization is not desired, since a liquid film is required. This means that lower We, Re and Oh numbers are acceptable.

### 3.1.2 Combustion chamber

The combustion chamber is cylindrical, made of 316L stainless steel and protected, besides the film cooling, by an EPDM (Ethylene-Propylene-Diene) thermal protection. EPDM was selected as insulating material due to its high usage

for such application, its low cost, high availability, no toxicity, and reasonable ablative properties when compared to other materials, such as phenolic-based.

A simulation was conducted by an algorithm developed by the team, in which the control volume is discretized into a mesh of cells. The main governing equations were the 1-D cylindrical heat conduction, 1-D energy conservation with convection, Newton's law of cooling and the convective heat coefficient was calculated using Bartz equation (Bartz, 1968). To solve the equations, an initial thermal protection must be informed and the converging criterion is the maximum desired inner side casing wall temperature. The main result of this algorithm is shown below:

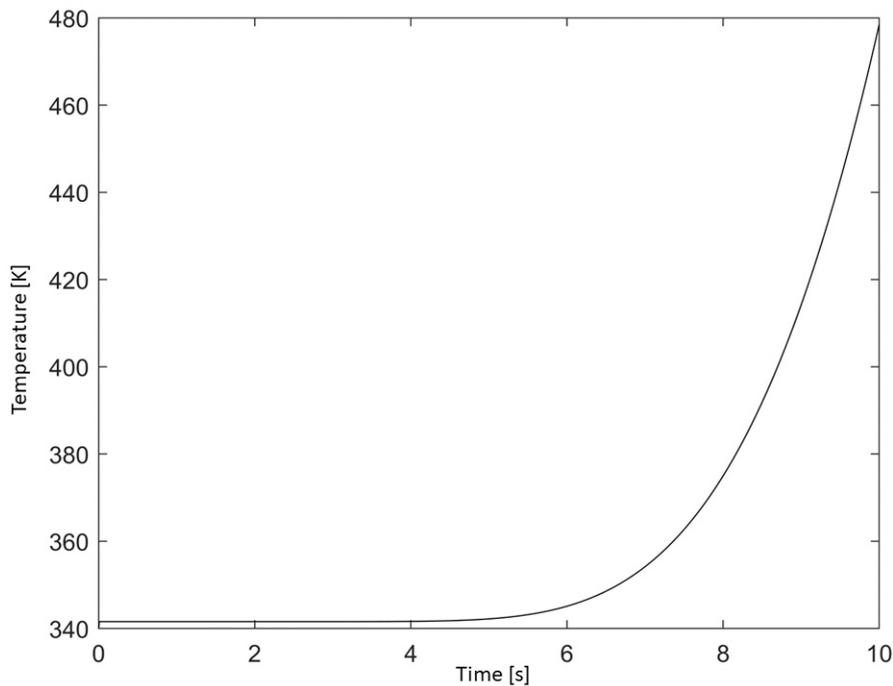


Figure 6: Inner side casing wall temperature in function of burning time with 4.5 mm of EPDM.

Figure 6 shows how the temperature of the inner side casing wall grows with burning time for 4.5mm of EPDM. This thickness protects the casing wall and prevents the mechanical properties of the metal from being compromised.

This analysis considered a constant thickness for the thermal protection material. In the future, for more accurate results, the regression of the EPDM with time of operation will be considered and its effects on the heat transfer to the inner side casing wall along the burning time.

### 3.1.3 Nozzle

The nozzle shape initially was conceived to be bell, however on later revisions it was changed to be conical. This modification was made in order to decrease costs and manufacturing complexity, since the difference in performance that this change brings does not affect the mission objective. It is made of a single part of low porosity graphite, nevertheless an update will be made in order to make a graphite convergent and throat, and a steel divergent. By using a two-part nozzle the complexity slightly increases, however the decrease in mass is more expressive.

### 3.2 Feed system

As mentioned previously, the LPRE operates under pressure-fed condition. This means that, at least, three tanks are reacquired on a bi-propellant system. Since, for this mission, a low cost product is more relevant than performance, gaseous nitrogen was selected to work as pressurant. In order to maintain the propellant tanks with the desired pressure during the engine's operation, pressure regulators are being used. In addition, pressure, temperature and level sensors, and pressure relief, bleeding and venting valves are assembled to guarantee safety and proper filling. Furthermore, filters and check valves were implemented to guarantee operation safety. The schematic below presents the feed system:

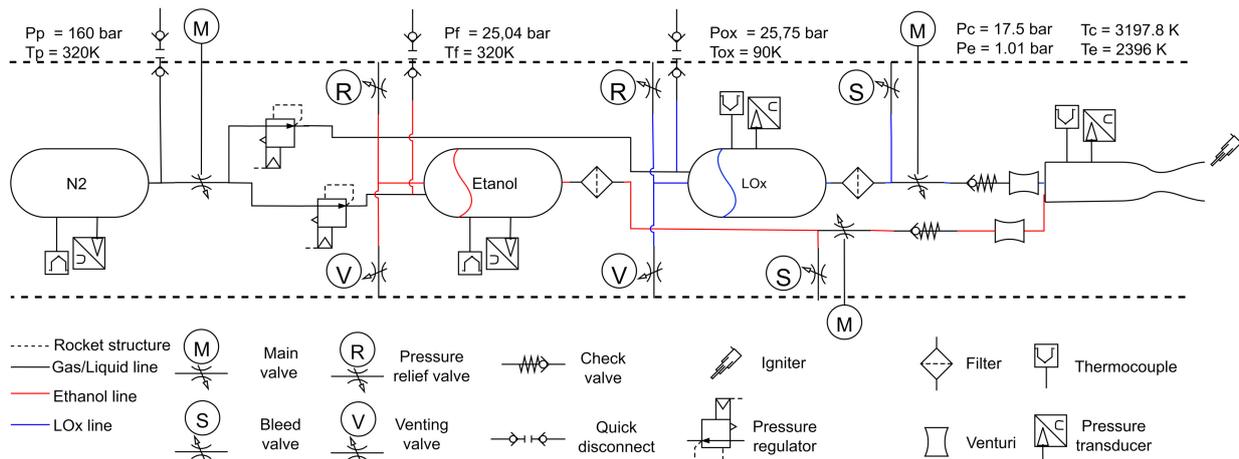


Figure 7: Feed system schematic.

It is also possible to visualize in Fig. 7 that it is being used an external ignition system and that the main propellant and pressurant valves are actuated externally. This solutions is being taken in consideration in order to decrease weight and costs per launch, since the same system could be used in several launches with low chances of damaging. In addition, the complexity and failure risks are diminished, since the required on-board battery budget is also decreased and externally there is no dimensions or weight restrictions. This means that the actuators could be robust and possible risks of the actuator not being able to open the valve, due to low torque or valve freezing, are almost 100% mitigated.

Thermal protection is applied on the oxidizer feed lines in order to avoid evaporation. Additionally, the fuel lines and manifold are thermally isolated from the oxidizer lines and manifold, since the freezing point of ethanol is around 160 K and the oxidizer is maintained under 90 K. Moreover, the thermophysical properties are highly sensitive to fluid temperature, so the engine would lose performance. Futhermore, before ignition, a small portion of pressurant is purged through the lines with the objective of removing any existing solid particles and it cools down the oxidizer lines and prevents it from evaporating.

Another component of the feed system are the venturis. The venturis were implemented in order to avoid feed system coupled instabilities (Summerfield, 1951) (Waxman, 2014). Moreover, the engine was designed in such way that this type of instability shouldn't happen. So, the venturis are a redundant component. The graph below shows the design point:

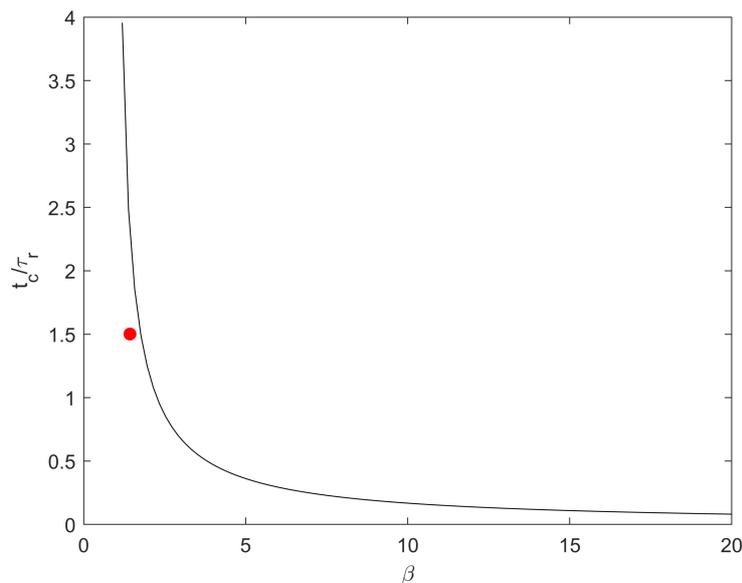


Figure 8: Feed system coupled instability analysis.

where  $\beta = \bar{P}_c / 2\Delta P$ , in which  $\bar{P}_c$  is the average chamber pressure and  $\Delta P$  is the pressure drop along the injector,  $t_c$  is the combustion time lag and  $\tau_r$  is the residence time. Additionally to the stability analysis shown in Fig. 8, the system attends the Summerfield stability criterion. Moreover, further stability analysis and experimental characterization will be conducted in future design and qualification phases.

#### 4. RESULTS

After running the optimization algorithm, some more iterations were performed in order to obtain a manufacturable propulsive system, since there are some limitations of budget and available technologies. As a result, the main final design parameters of the LPRE are compiled in Tabs. 2, 3 and 4. In these tables, NI means Nozzle Inlet.

Table 2: Propulsion system preliminary design global parameters.

Parameter	Value	Parameter	Value
Thrust, kN	25.95	TWR	6.62
Chamber Pressure, bar	17.5	Injection Pressure Drop, bar	6.125
Burning Time, s	10	Specific Impulse, s	231.42
Mixture Ratio	1.54	Fuel Mass Flow Rate, kg/s	4.51
Contraction Ratio	3.15	Expansion Ratio	3.53
Characteristic Length, m	1.44	Thermal Protection Thickness, mm	4.5

Table 3: Combustion chamber thermochemical parameters.

Parameter	Injector	NI	Throat	Exit
Temperature, K	3197.83	3189.70	3042.07	2393.04
Pressure, bar	17.5	16.79	9.93	1.01
Density, kg/m <sup>3</sup>	1.496	1.439	0.903	0.119
Specific Heat Ratio	1.127	1.126	1.126	1.157
Specific Heat at Constant Pressure, kJ/kgK	6.359	6.363	5.892	2.961
Thermal Conductivity, W/mK	1.247	1.246	1.105	0.445

Table 4: Expected efficiency.

Efficiency	Value, %
Nozzle divergence	98.29
Friction	99.93
Chamber Diameter	98.19
Film cooling	96.00
Combustion	97.00
<b>TOTAL</b>	<b>89.81</b>

The designed engine with the above results is capable of powering the training rocket and attend all requirements. Looking at the results shown in Tab. 2, with just 10 seconds of burning time it is possible to reduce the effects of exposing the combustion chamber material to high temperature, avoiding the need for a complex cooling system, whilst still achieving sufficient performance to carry out the mission. In addition, working with lower chamber pressure makes it safer and the value itself is in the range recommended by (Kessaev, 1997).

Table 3 shows how important thermochemical parameters changes along the engine. This results are later used to improve the thermal analysis and to make sure the cooling system is capable of protecting the chamber walls, providing a safe operation of the system.

Comparing the above results with propulsion systems of similar applications and design, one can see that the presented performance parameters can be considered satisfactory, in this phase of development. A TWR of 6.62 is a value that lies inside the range of similar projects such as (Gontijo, 2022b) (CopenhagenSuborbitals, 2023) (Altafim *et al.*, 2019). It is important to mention that this is the preliminary design of this rocket engine and further developments will provide a more complete and refined project.

#### 5. CONCLUSIONS

Basing the whole methodology and calculations on training requirements, limitations and desires of the launch centers, a liquid propellant rocket engine using liquid oxygen and ethanol could be designed. Some key design simplifications and choices were fundamental to achieve a relative low cost and low complexity/risk system. Then, it is expected to obtain a reliable, useful and easily accessible training rocket for CLA and CLBI.

Additionally, great partnerships firmned in this project demonstrated to be very promising and relevant improvements and innovations were already implemented. Furthermore, it is expected that more iterations of the project are to come

and an enhancements will be reached. One of the main sources of upgrades are the numerical simulations that will be performed and validate and propose modifications. In addition, revisions will be made alongside the CPL team and feedbacks will be received from a technical team formed by AEB on every end of systems engineering phases.

Finally, this type of funding is proving itself as the right way to promote new initiatives in Brazilian aerospace development and to create and evolve an entire industrial chain for growing on space sector. A great opportunity provided by MCTI, AEB, FINEP and FNDCT is being executed by the team on the best possible way, attending all requirements and overcoming technological limitations.

## 6. ACKNOWLEDGEMENTS

The authors would like to thank the MCTI (Ministry of Science, Technology and Innovation), AEB (Brazilian Space Agency), FINEP (Financier of Studies and Projects) and FNDCT (National Fund for Scientific and Technological Development) for funding (Grant term 03.22.0491.00 - FINEP reference project n° 1406/22 - Economic Subsidy for Innovation - n° 03/2022) the project and DeltaV Engenharia Espacial for the opportunity of developing such great work.

## 7. REFERENCES

- Adler, J., 1959. "A one-dimensional theory of liquid-fuel rocket combustion. part ii. the influence of chemical reaction". *Imperial College of Science and Technology*.
- AEB, 2009. "400 years searching for technology (in portuguese)". Agência Espacial Brasileira e Financiadora de Estudos e Projetos, <https://www.gov.br/aeb/pt-br/centrais-de-conteudo/publicacoes/institucional/revistas-pdf/006revista-peb-jul-ago-set-2009.pdf>. Accessed 02 June 2023.
- Alemasov, V.E., Dregalin, A.F. and Tishin, A.P., 1980. *Theory of rocket engines (in Russian)*. MOCKBA, Mashinostroenie.
- Almeida, D.S. and Pagliuco, C.M.M., 2014. "Development status of I75: A brazilian liquid propellant rocket engine". *Journal of Aerospace Technology and Management*, Vol. 6.
- Almeida, D.S., Santos, E.A. and Langel, G., 2021. "Upper stage liquid propellant rocket engine: A case analysis". *Journal of Aerospace Technology and Management*, Vol. 13.
- Almeida, D.S., Shimote, W.K. and Niwa, M., 1999. "Selection of materials for combustion chamber of liquid propellant rocket engine". In *15th Brazilian Congress of Mechanical Engineering*. ABCM - Brazilian Society of Mechanical Sciences and Engineering, Águas de Lindóia, SP, Brazil.
- Altafim, R.M., Belazora, L.A., Goulart, A., Borges, V., Tomazeti, C.A., Santos, G.C. and Vaz, C.C., 2019. "Design and development of the liquid training rocket (ftb-l) - (in portuguese)". In *II Brazilian Aerospace Congress, 2019*. Santa Maria, Brazil.
- Bartz, D.R., 1968. "Survey of relationships between theory and experiment for convective heat transfer in rocket combustion gases". *Advances in Rocket Propulsion, AGARD*.
- CopenhagenSuborbitals, 2023. "Spica engine." Available at: <https://copenhagensuborbitals.com/missions/spica-engine/>. Accessed 01 October 2023.
- Dorofeev, A.A., 2010. *Basics of the theory of thermal rocket engines (in Russian)*. MOCKBA, Moscow.
- Gontijo, M.S., 2022a. "A review of vaporization models as design criterion for bipropellant thrust chambers". *Aerospace Technic and Technology*, Vol. 4.
- Gontijo, M.S., Fischer, G.A.A. and Costa, F.S., 2020. "Evaluation of smd effects on characteristic lengths of liquid rocket engines using ethanol/lox and rp-1/lox". In *18th Brazilian Congress of Thermal Sciences and Engineering*. ABCM - Brazilian Society of Mechanical Sciences and Engineering.
- Gontijo, M.S., Fischer, G.A.A. and Costa, F.S., 2021a. "Characteristic lengths of liquid propellant rocket engines and the influence of chemical reactions". In *26th International Congress of Mechanical Engineering*. ABCM - Brazilian Society of Mechanical Sciences and Engineering.
- Gontijo, M.S., Fischer, G.A.A. and Costa, F.S., 2021b. "Influence of smd on characteristic lengths of liquid propellant rocket engines". In *8th School of Combustion*. RNC - Rede Nacional de Combustão.
- Gontijo, M.S. and Shynkarenko, O., 2020. "Investigation of a dual-fuel hybrid rocket engine for missile and rocket applications". In *71st International Astronautical Congress(IAC) – The CyberSpace Edition*. IAF - International Astronautical Federation.
- Gontijo, M.S., 2022b. *Preliminary Design, Analysis and Optimization of a Liquid Propellant Rocket Engine for Sounding Rockets (in Portuguese)*. Distrito Federal, Brasil.
- Gordon, S. and McBride, B.J., 1988. "Finite area combustor theoretical rocket performance". *NASA Technical Memorandum*, Vol. 100785.
- Gordon, S. and McBride, B.J., 1994. "Computer program for calculation of complex chemical equilibrium compositions and applications - i analysis". *NASA Reference Publication*, Vol. 1311.
- Hill, P. and Peterson, C., 1992. *Mechanics and thermodynamics of propulsion*. Addison Wesley Longman, 2nd. Ed.

- Humble, R.W., Henry, G.N. and Larson, W.J., 1995. *Space propulsion analysis and design*. Learning Solutions, 1st Ed.
- Huzel, D.K. and Huang, D.H., 1992. *Modern engineering for design of liquid-propellant rocket engines*. AIAA - American Institute of Aeronautics and Astronautics, Progress in Astronautics and Aeronautics, Vol. 147.
- Kessaev, J.V., 1997. *Theory and calculation of liquid-propellant engines*. Moscow Aviation Institute, São José dos Campos.
- Kudryavtsev, V.M., 1993. *Fundamentals of the theory and calculation of liquid rocket engines (in Russian)*. MOCKBA, Moscow.
- Lefebvre, A.H. and McDonell, V.G., 2007. *Atomization and Sprays*. CRC Press, 2nd Ed., New York.
- McBride, B.J. and Gordon, S., 1992. "Computer program for calculating and fitting thermodynamic functions". *NASA Reference Publication*, Vol. 1271.
- McBride, B.J. and Gordon, S., 1996. "Computer program for calculation of complex chemical equilibrium compositions and applications - ii users manual and program description". *NASA Reference Publication*, Vol. 1311.
- McBride, B.J., Gordon, S. and Reno, M.A., 1993. "Coefficients for calculating thermodynamic and transport properties of individual species". *NASA Technical Memorandum*, Vol. 4513.
- McBride, B.J., Reno, M.A. and Gordon, S., 1994. "Cet93 and cetpc: An interim updated version of the nasa lewis computer program for calculating complex chemical equilibria with applications". *NASA Technical Memorandum*, Vol. 4557.
- McBride, B.J., Zehe, M.J. and Gordon, S., 2002. "Nasa glenn coefficients for calculating thermodynamic properties of individual species". *NASA Technical Papers*, Vol. 211556.
- MCTI, 2022. "Capacitation rocket prototype (in portuguese)". Ministério da Ciência, Tecnologia e Inovações, Agência Espacial Brasileira e Financiadora de Estudos e Projetos, <http://www.finep.gov.br/chamadas-publicas/chamadapublica/673>. Accessed 02 June 2023.
- Pfuetzenreuter, L., Burkhardt, H., Lippert, C., Wagner, B., Almeida, D.S., Pagliuco, C.M.M., Nascimento, L.B., Souza, B., D, Z.R., Araujo, T.B., Alting, J., Preuss, A. and Langel, G., 2017. "L75 lox ethanol engine: Current status of thrust chamber and turbopump cooperative development". In *53rd, AIAA/SAE/ASEE Joint Propulsion Conference*. AIAA - American Institute of Aeronautics and Astronautics, Atlanta, Ga, United States of America.
- Ponomarenko, A., 2013. "Assessment of delivered performance of thrust chamber". RPA: Tool for Rocket Propulsion Analysis, [https://www.rocket-propulsion.com/downloads/pub/RPA\\_AssessmentOfDeliveredPerformance.pdf](https://www.rocket-propulsion.com/downloads/pub/RPA_AssessmentOfDeliveredPerformance.pdf). Accessed 02 June 2023.
- Saeedipour, M., Schneiderbauer, S., Pirker, S. and Bozorgi, S., 2014. "A numerical and experimental study of flow behavior in high pressure die casting". *Magnesium Technology*.
- Schmidt, P.W., 1984. "Atomization of liquids (in german)". *Physik in Unserer Zeit*, Vol. 15.
- Spalding, D.B., 1958. "Combustion in liquid-fuel rocket motors". *Imperial College of Science and Technology*.
- Spalding, D.B., 1959. "A one-dimensional theory of liquid-fuel rocket combustion". *Imperial College of Science and Technology*.
- Stechman, R.C., Oberstone, J. and Howell, J.C., 1969. "Design criteria for film cooling for small liquid-propellant rocket engines". *NASA Technical Memorandum*, Vol. 6.
- Summerfield, M.A., 1951. "Theory of unstable combustion in liquid propellant rocket systems". *Journal of the American Rocket Society*, Vol. 21.
- Sutton, G.P. and Biblarz, O., 1995. *Rocket propulsion elements*. John Wiley Sons, 9th Ed., Hoboken.
- Svehla, R.A., 1995. "Transport coefficients for the nasa lewis chemical equilibrium program". *NASA Technical Memorandum*, Vol. 4647.
- Torres, M.F.C., Almeida, D.S., Krishna, Y.S.R., Silva, L.A. and Shimote, W.K., 2009. "Liquid propulsion in iae: Vision of activities and future perspectives (in portuguese)". *Journal of Aerospace Technology and Management*, Vol. 1.
- Vasil'yev, A.P. and Kudryavtsev, V.M., 1969. *Fundamentals of theory and calculation of liquid-propellant rocket engines - Vols. I-III*. National Technical Information Service, Springfield.
- Volkov, E.B., Golovkov, L.G. and Syritysyn, T.A., 1970. *Liquid propellant rocket engines (in Russian)*. MOCKBA, Moscow.
- Wang, Z.G., 2016. *Internal Combustion processes of Liquid Rocket Engines: Modeling and Numerical Simulations*. John Wiley Sons, Singapore.
- Waxman, B.S., 2014. *An investigation of injectors for use with high vapor pressure propellants with applications to hybrid rockets*. Ph.D. thesis, Department of Aeronautics & Astronautics, Graduate Studies of Stanford University, Stanford, United States of America.

## 8. RESPONSIBILITY NOTICE

The authors are solely responsible for the printed material included in this paper.