Multidisciplinary Design Optimization of a De-Boost Hybrid Motor for the Brazilian Recoverable Satellite

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Abstract
This work presents a multidisciplinary optimization technique applied to the design of a hybrid rocket motor for the de-boost system of a small recoverable satellite. For some specific missions, hybrid rocket technology may become a competitive choice due to its inherent advantages over more traditional chemical propulsion systems. The Brazilian Recoverable Satellite Platform (SARA) relies on a chemical propulsion system for orbital de-boost. We considered a hybrid motor as the propulsive technology. An optimization algorithm was developed to help propose a motor configuration. We choose a hybrid motor based on liquefying fuel, solid paraffin, and nitrous oxide propellants, by virtue of the long term tradition in operating such propulsion systems. The multidisciplinary configuration optimization technique was entirely based on geometrical operating parameters of the motor, rather than performance to facilitate further design and fabrication. Results from the code presented a hybrid motor which was considered a competitive alternative for the de-boost engine if compared to the traditional chemical systems, solid and liquid bipropellant and monopropellant. A hybrid propulsion system for the de-boost maneuver weighting less than 26 kg was considered following the design optimization process.

1. Introduction
The SARA satellite was conceived as a microgravity recoverable and reusable research platform by the Space and Aeronautic Institute- Brazil (IAE) and the Brazilian Space Agency (AEB). The satellite will carry a payload mass of 55 kg and was specified to have a total launch mass up to 350 kg. Its orbit will be circular with an altitude of 300 km with two degrees inclination. After conclusion of the orbital experiments, no longer than ten days, the reentry procedure must start, providing the right positioning of the satellite followed by the de-boost impulse. Figures 1-A shows a schematic design of the platform and Fig. 1-B the same system undergoing structural analysis. The spacecraft will soon pass through a series of qualification tests in ballistic flights with 350 km apogee falling at 300 km from the launch site. The system will be accelerated by a Brazilian VS-40 sounding rocket. Figure 2 presents the main characteristics of the rocket and a picture of it taking just after take-off.
After completion of the mission, the spacecraft will re-entry into the atmosphere and final deceleration will take place by a high performance parachute system [1].

To date, solid and liquid rocket propulsion systems were the only options considered as the main alternatives for the de-boost system [2]. Main results of their study are presented in Section 2. Hybrid propulsion system for de-boosting phase was not considered in the work of Villas Bôas et al., [2]. This type of chemical propulsion system has gained a lot of attention from research groups and enterprises around the world.

The University of Brasilia (UnB) Hybrid Propulsion Team has a considerable history in developing and testing hybrid rocket engines and small sounding rockets in which the thrust and burning times (impulse) were quite similar to the SARA de-boost system. Hybrid rockets present an attractive option for the Brazilian space program due to their relative lower development cost, simple construction and safe operation.

In a review paper, Oiknine [3] analyzed what have prevented hybrid propulsion from substituting solid and liquid technologies in commercial space applications, despite several inherent advantages over the classical counterparts, namely:

- Safety (in manufacturing, transporting and storage as a consequence of separate fuel and oxidizer);
- Reliability (due to the larger margin of tolerance in grain imperfections as well as ambient conditions);
- Flexibility (by virtue of stop-restart capabilities and thrust modulation);
- Costs (because of low investments costs for development and operation as well as those costs associate with the materials to fabricate de motor);
- Environment (since combustion products are, often, non-toxic gases and propellants are not hazardous to storage and transport).

In another work, Karabeyoglu [4] claims that hybrid propulsion should be complementary to other chemical rockets, finding specific niches where safety and cost are more important than performance. Space tourism and small launch vehicle propulsion system are instances of such observation.

Our research group was pioneer in studying such propulsive technology in Brazil [5]. Most of the important
features of hybrid propulsion systems, based on liquefying fuels, have been investigated by our research group since the year 2004 [6,7,8,9]. Figure 3 shows a thrust curve against time for a static test of one of our sounding rocket motors along with pictures taken during the firing. The engine design thrust was 1,500 N. In Fig. 3, one picture shows the ignition of the motor and the other the engine exhaust gases plume, taken just after 1.5 s firing time.

From our research activities, Bertoldi [9] reported a series of firing using a small hybrid rocket (paraffin based) with just one injector (pressure swirl atomizer) for the nitrous oxide. A correlation equation for the solid fuel regression rate was proposed. Regression rates from our study are higher than those proposed by [10] for the same oxidizer mass flux. Nitrous oxide and liquefying fuels have been the preferred propellants of our propulsion research activities in hybrid rocket systems. Nitrous oxide is a non toxic, long term storable, has good density, a high vapor pressure and has good overall performance, as pointed by Thicksten and Macklin [11] in a paper dedicated on how to handle such oxidizer in hybrid rocket motor testing. Nitrous oxide has also self-pressurizing characteristics thus avoiding pressurization sub-systems. Paraffin is an affordable fuel which handling (solid fuel casting) presents no difficulties. Product emissions from these propellants are also of little harm.

Figure 3: Thrust curve and pictures for a 1500 N hybrid motor under firing conditions [9] using using pressure swirl atomizers for the injection of nitrous oxide.

Following this experience, we proposed a de-booster motor for the SARA spacecraft entirely based on hybrid propulsion technology. This work presents a multidisciplinary optimization of the proposed de-boosting motor in which the propellants were based on nitrous-oxide and solid paraffin.

2. Engine Characteristics
The main re-entry mission aspects were presented and discussed in [2], namely:

- De-boost impulse should produce a velocity reduction of the order of 235 to 250 m/s opposite to the orbital motion;
- Total burning time of the motor should be between 50 and 200 s.

These performance characteristics, though, are not standalone. The system should also follow a geometric configuration as well as total mass limitation, on account of the launch vehicle operational envelop and mission efficiency. Following that, Villas Bôas et al. [2] proposed three different configurations for the engine (de-booster); liquid bi-propellant (LBP), liquid mono-propellant (LMP) and solid propellant (SP). The LBP alternative was composed of a liquid rocket engine system based on unsymmetrical dimethylydrazine (UDMH) and nitrogen tetroxide (NTO) with engine chamber feeding provided by means of an inert gas (nitrogen) pressurization sub-system. The second option (LMP) was a hydrazine mono-propellant system. As for the LBP,
Engine chamber feeding should also be provided by an inert gas (nitrogen) pressurization sub-system. The last configuration (SP) should be based on the technology developed at IAE for the Roll Control System (PCR/S-IV) of the sounding rocket Sonda-IV [2]. Engine thrust comes from the solid propellant, end burn grain type. The propellant grain was conceived as a variable burning area, with the final thrust being about five to six times lower than the initial thrust.

Figure 4-A shows a conceptual design of the spacecraft with its main components. As it can be seen, the engine for de-boosting must meet some dimensional requirements. Figure 4-B shows a drawing of the SP engine, as suggested in [2]. The study conducted by Villas Bôas et al. (1999) for the proposed systems are summarized in Table 1. Table 1 shows propulsion system mass varying from 35.1 to 47.3 kg. Size and volume of the systems are not present but it was assumed that the engine system and sub-system must fit the engine bay in SARA’s platform.

![Figure 4: Conceptual design of the SARA platform (source: IAE) with the main sub-systems and the solid propellant engine for SARA de-boosting as proposed by [2].](image)

Table 1: Summary of the propulsive technologies proposed and investigated by Villas Bôas et al. [2].

<table>
<thead>
<tr>
<th>Parameter</th>
<th>LBP de-boost engine</th>
<th>LMP de-boost engine</th>
<th>SP de-boost engine</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Mass</strong></td>
<td>Higher: 47.3 kg</td>
<td>Medium: 40.3 kg</td>
<td>Lower: 35.1 kg</td>
</tr>
<tr>
<td><strong>Development Term</strong></td>
<td>Higher: high number of parts, more complex, development of new technologies.</td>
<td>Higher: high number of parts, more complex, development of new technologies.</td>
<td>Lower: low number of parts, less complexity, available technologies.</td>
</tr>
<tr>
<td><strong>Development Cost</strong></td>
<td>Lower: utilization of parts of SCR/VLS-1, most of the tests will be performed in the SCR/VLS-1</td>
<td>Medium: utilization of some parts of SCR/VLS-1, some tests will be performed in the SCR/VLS-1</td>
<td>High: new development, higher number of tests.</td>
</tr>
<tr>
<td><strong>Production Cost</strong></td>
<td>Higher: high number of parts</td>
<td>Medium: medium number of parts</td>
<td>Lower: low number of parts</td>
</tr>
<tr>
<td><strong>Operating Precision</strong></td>
<td>Higher</td>
<td>Higher</td>
<td>Lower (may be improved by use of a thrust cutoff system)</td>
</tr>
<tr>
<td><strong>Handling, Safety and Toxicity</strong></td>
<td>Required high level of care, careful operations, high toxic propellants, possibility of spills.</td>
<td>Required high level of care, careful operations, toxic propellant, possibility of spills.</td>
<td>Safe, non-toxic propellant, no possibility of spills.</td>
</tr>
</tbody>
</table>
3. Multidisciplinary Design Optimization

For this work it was decided the use of a genetic optimization algorithm to optimize the configuration of the SARA de-boost engine, targeting for the lightest configuration. Comparatively to the other propulsion system hybrid rocket engines allow unification of principal assemblies and units to a greater extent than solid and liquid motor [12]. Therefore, hybrid motors are much simpler to design leading to a simpler optimization process.

Hybrid rocket systems are considerably sensitive to grain configuration, which makes it attractive for multidisciplinary optimization. The SARA de-boost engine was selected as our first attempt on carrying multidisciplinary configuration optimization given that the mission performance requirements are very well defined along with design constraints. Optimization techniques were suggested for the design of a hybrid motor for spacecraft de-boosting. In most of the cases, rocket engine design is focused on searching the minimal mass for the system. Also, for the specific case studied here, minimal length is also an important aspect to be taken into account.

3.1. Optimization Formulation

An optimization problem can be expressed as the minimization of an objective function under certain equality and inequality constraints. The formulation of a generic optimization problem is defined as:

\[
\min_{\mathbf{s} \in \mathbb{S}} z(\mathbf{s}, q(\mathbf{s})) \\
h(\mathbf{s}, q(\mathbf{s})) = 0 \quad h \in \mathbb{R}^n \\
g(\mathbf{s}, q(\mathbf{s})) \leq 0 \quad g \in \mathbb{R}^m \\
\mathbf{S} = \left\{ \mathbf{s} \in \mathbb{R}^n \mid \mathbf{s}_l \leq \mathbf{s} \leq \mathbf{s}_u \right\}
\]

where \( z \) is the objective function, \( \mathbf{s} \) is the set of design variables, \( q \) is the set of design criteria, and \( h \) and \( g \) are respectively equality and inequality constraints. The design variables and constraints belong to a set of real numbers whose dimensions are represented by \( n_s \), \( n_h \), and \( n_g \), respectively. The values of the design variables are limited by lower and upper bounds \( [\mathbf{s}_l, \mathbf{s}_u] \), thus defining the so called box constraints. The objective function and the constraints are built as functions of design criteria and design variables. In a multidisciplinary design optimization setup, the design variables are defined by parameters that represent different physical aspects. For instance, in hybrid propulsion systems parameters such as geometry or shape, dimensions and pressure of the combustion chamber and the mass flow rate of oxidant are possible choices for design variables.

The design criteria are functions of design variables. In a multidisciplinary optimization framework, the design criteria are associated with quantities which describe system performance and behavior. For example, in hybrid propulsion systems quantities such as trust, burning time, variation of velocity and mass are considered as design criteria. In general, for a multidisciplinary optimization problem, the design criteria are dependent on the response of the multidisciplinary system that can also be a function of the design variables.

3.2. Optimization Procedure

3.2.1. Optimization Model

To successfully perform the re-entry maneuver it is necessary to provide velocity variation opposite to the orbit trajectory raging from 235 to 250 m/s [2] the re-entry maneuver must be performed in a time span of 50 to 200 s. The multidisciplinary optimization was then proposed to find a hybrid motor capable of responding all the mission requirements.

As pointed by Kwon et al. [13], in designing a hybrid motor the grain configuration, the combustion performance, the oxidizer tank pressure, and the nozzle configuration are key elements of the performance of the rocket in addition to its geometrical configuration. In their work, the authors selected the number of ports, the initial oxidizer flux, the combustion chamber pressure, the nozzle expansion ratio, and the average O/F ratio as initial candidate design variables. They performed a preliminary sensitivity analyses to identify the dependence of some candidate design variables to the design constraints and the objectives, namely; rocket length, diameter, total mass, and nozzle exit diameter. As shown in the figure, the number of ports has a big influence on the rocket length and diameter. Their sensitivity analyses for multi port hybrid engines brought to the following:

- initial oxidizer flux could dominantly affect the length and the diameter of rocket simultaneously;
- nozzle exit diameter was mainly affected by the combustion chamber pressure and nozzle expansion ratio;
the average OF ratio had a small influence on the response parameters like, rocket length, diameter and nozzle exit diameter.

Taking fabrication feasibility as a major project concern, design variables were selected in order to facilitate the geometrical configuration of the motor itself and the pressurization sub-systems. The higher regression rates of paraffin based fuels allow the use of only one combustion port, greatly simplifying grain configuration. It was avoided the use of subjective variable such as thrust specific impulse. These parameters, though, were considered along the optimization process. Based on that, the chosen design variables were:

- Solid fuel external diameter - \(D\);
- Solid fuel length - \(L_g\);
- Internal port diameter - \(D_i\);
- Initial combustion chamber pressure - \(p_{ci}\);
- Oxidizer mass flow rate - \(m_{oxi}\).

The solid fuel external diameter was chosen as a main design variable due to its intrinsic relation with the volume of the combustion chamber. In order to evaluate a broad range of configurations, the grain external diameter was allowed to vary from 100 to 1,000 mm, which is lower than the diameter of SB-40 sounding rocket. This wide range of diameters was selected to allow a high degree of freedom since computational cost was a minor concern. Grain length was also selected as a major design variable due to its direct influence on mixture ratio and size constraint of the system. This parameter was then allowed to vary from 10 to 950 mm. The grain initial port diameter or the grain thickness is a measure of the available burning radius, or burning time depending on the average oxidizer mass flux. A constraint is imposed to the initial port diameter as to avoid erosive burn. Initial oxidizer mass flux should be less than 600 kg/(m\(^2\)s). The initial chamber pressure influences the thrust of the motor, the thickness of the combustion chamber wall, the oxidizer tank pressure and its wall thickness. The chamber pressure was set to vary from 3 to 100 bar.

3.2.2. Optimization Routine
A routine using ESTECO’s ModeFRONTIER software (ModeFRONTIERv4) was proposed to help generate, evaluate and select individuals along the optimization process. The ModeFRONTIER workflow showed in Fig. 5 consists of a performance prediction module, a set of input variables, a block with three “IF” structures for the design constraints, a genetic algorithm scheduler and a DOE block. The performance prediction module was implemented in the Engineering Equation Solver (EES Academic Professional V8-3D). This module has the internal ballistic model proposed for this work. The five input variables blocks represent the respective design variables described previously. The three constrains represent respectively the \(D > D_{gr}\), and the time and velocity variation constraints.

![Figure 5: Processes diagram, ModeFRONTIER.](image)

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4. Hybrid Propulsion System Analysis

4.1. Rocket Configuration Model
In order to accurately define the rocket system several considerations were made, resulting in a simple propulsion system configuration as shown in Fig. 6. The rocket will be consisted of a combustion chamber and nozzle unit, a pressurized nitrous oxide tank and the pressurization subsystem (helium tank and control valve). In its final design, the shape and positioning of the tanks would consider available space in the engine bay of the SARA spacecraft. A jet swirl injector plate for the atomization of the oxidizer was chosen. This system would significantly increase the solid fuel regression rate as compared to shower head injector type. The hybrid motor is made of cylindrical container with spherical ends. The nozzle would be conical and made of aluminum with carbon phenolic insert for thermal protection. The combustion chamber should be also made of aluminum with added thermal insulation for the post-combustion chamber.

![Figure 6: Hybrid rocket engine schematics.](image)

4.2. Internal Motor Ballistic
The ballistic model for the hybrid motor optimization process was based on that proposed by Casalino and Pastrone [13]. The main parameters are shown in Fig. 7.

![Figure 7: Pressure and areas for the ballistic model.](image)

Solid fuel regression rate is a strong function of the oxidizer mass flux

\[
\dot{r} = nG^n.
\]

Table 2 presents different values of \( a \) and \( n \) parameters as proposed by our research group [9], for paraffin and nitrous oxide, and reference [10].

<table>
<thead>
<tr>
<th>Propellants</th>
<th>( a )</th>
<th>( n )</th>
<th>Reference</th>
</tr>
</thead>
<tbody>
<tr>
<td>Liquid N(_2)O/paraffin</td>
<td>0.722</td>
<td>0.67</td>
<td>[9]</td>
</tr>
<tr>
<td>N(_2)O/paraffin</td>
<td>0.488</td>
<td>0.62</td>
<td>[10]</td>
</tr>
</tbody>
</table>
Due to the high regression rate of paraffin, compared to other traditional solid fuel for hybrid propulsion, only one combustion port is needed. Therefore, the variation of the internal grain diameter ($R$) is given by

$$dR/dt = \dot{r}. \hspace{1cm} (2)$$

Pressure decay inside the combustion chamber is given by

$$p_1 = \left[ 1 + 0.2 \left( \frac{\Delta}{A_p} \right)^2 \right] p_c. \hspace{1cm} (3)$$

In Eq. (3), $p_1$ is the head end pressure and $p_c$ the combustion chamber pressure just before the expansion process. Also, $A_t$ and $A_p$ refer to the area of the throat and grain port, respectively. Oxidizer mass flow is a function of the hydraulic resistance of the injection plate ($Z_{inj}$), estimated by the following equation

$$\dot{m}_{oxi} = \sqrt{(p_t - p_1)/Z_{inj}} \hspace{1cm} (4)$$

where $p_t$ is the oxidizer tank pressure. The fuel mass flow can be calculated by

$$\dot{m}_{fuel} = \rho_{fuel} \dot{r} A_b. \hspace{1cm} (5)$$

The fuel mass flow in Eq. (5) is a function of the density of the paraffin, the regression rate and the internal burning area of the combustion port, respectively. The ratio between $\dot{m}_{oxi}$ and $\dot{m}_{fuel}$ gives the mixture ratio of oxidizer and fuel accordingly

$$OF = \frac{\dot{m}_{oxi}}{\dot{m}_{fuel}}. \hspace{1cm} (6)$$

3. Results and Discussion

For this work, the genetic optimization algorithm was chosen in order to provide an expertise background for an intended future work in which a larger number of variables will be considered. During the design optimization process, on account of the excessive number of values for the design variables and the box constraints, a considerable number of individuals failed the design constraints. The resulting candidates presented here are the best from each generation. Four individuals were eliminated because they were located in a zone where the physical model was not pertinent. Several tendencies were observed among the lightest individuals. As shown in Fig. 8, shorter grain lengths would give lighter propulsive system with, the same velocity variation. The square represents, after the optimization process, the best candidate for the missions. It can be seen in Fig. 8 a correlation between grain length and the mass of the system would emerge.

![Figure 8: Non-dimensional mass of the candidates versus solid fuel length.](image)

Figure 9 presents the non-dimensional mass as a function of burning times. As regard to rocket engine burning times the candidates are located in the time span of 60 to 190 s with most of them concentrated around 80 s. The best candidate would operate with a burning time of around 75 s.
Figure 9: Non-dimensional mass of the candidates versus engine burning times

Figure 10 shows the mass of the system versus oxidizer mass flow. In this case, the optimization process indicates the best individual to operate with an oxidizer mass flow near 0.18 kg/s. As regard to the combustion chamber pressure, the optimizations process located the candidates from around 18 to 50 bar. Most of the candidate were located in the vicinity of 30 bar, including the best individual.

Figure 10: Non-dimensional mass of the candidates versus engine oxidizer mass flow.

The grain internal diameter of the best candidate, as shown on Fig. 11, was about 130 mm, though, the population could vary from about 50 to 160 mm.

Figure 11: Non-dimensional mass of the candidates versus grain initial port diameter.
4. Conclusion

After applying the proposed a multidisciplinary design optimization algorithm the optimal design for a de-booster engine based on hybrid propulsion technology was identified. The best candidate able to fit mission requirements was less than 26 kg total mass. The operating parameters of the engine, for this mission were all feasible and near to what we have been working in terms of engine configuration, oxidizer injection system, thrust and combustion chamber and inert gas pressure. The best candidate weights less than those proposed by reference [2] using more traditional chemical propulsive systems. Liquid bi-propellant option was as heavy as 47 kg, for instance.

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References