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ORIGINAL PAPER

Multi-Criteria Conceptual Design Analysis of Single Stage Suborbital Launch Vehicle

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ABSTRACT

This paper presents the design analysis of a suborbital launch vehicle by performing a multidisciplinary optimization analysis. The design problem consists of establishing the optimized launch scenario using the interior point optimization method. The minimization of the objective function (the total mass of the launch vehicle) is validated by a multi-criteria approach. It is shown that establishing the performance of the launch vehicle should meet the criteria of security and control during the launch mission. The security aspect is represented by adherence to the dynamic pressure for structural matters of the launch vehicle and the acceleration, represented by the G-force, which should remain tolerable. Control is denoted by keeping the vehicle's velocity within the range required for a suborbital flight mission. The approach followed is appealingly constructive for conserving the multidisciplinary design optimization (MDO) formulation for a post-performance analysis.

Keywords: Multidisciplinary design analysis; Suborbital flight; Launch vehicle; Performance optimization; Feasibility model.

INTRODUCTION

Suborbital flights are an interesting segment of space exploration, used for diverse scientific and technological experiment, as well as space tourism (Benjamin 2018; Guerster and Crawley 2018; Niederstrasser and Frick 2015). Consequently, the sub-orbital flight market is experiencing significant growth, driven by the interest and the willingness of individuals for the edge of space exploration and discovery; it is also an interesting research topic for various studies such as microgravity, atmosphere, technology testing, and demonstration. A strong interest and development efforts is shown by Virgin Galagtic, Blue Origin, NASA, and Astrium to accompany the necessary technology advancement regarding reliability, safety, Guidance, Navigation & Control (GNC), as well as cost reduction.

The launched payloads (instruments or personnel) enable the conduction of experiments with the possibility of collecting and transmitting it in flight. This type of launchers is dedicated to near-space missions, as various objectives require reaching higher altitudes (Buddhavarapu *et al.* 2019). However, suborbital launch vehicles are designed to operate in the most performing and optimized scenario. Liquid rockets have the advantage of being modulated and controlled in terms of speed, acceleration, or deceleration during the flight profile (Dresia *et al.* 2021; Huh *et al.* 2017), therefore, based on the given criteria's, liquid propulsion motors were selected for this study as it is providing a solution for the transported payload to record data even during flight.

Through their capabilities, a wide range of mission along with various applications is achieved. Furthermore, sub-orbital launch vehicles represent an important segment of the aerospace industry, it is important to guarantee the mission's success and health safety for human accompanied missions. These launch vehicles are composed of a set of components assembled in a hierarchy of

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subsystems. The overall performance of the launch vehicle is linked to every single subsystem, which depends on the design sets and parameters. Alterations in these design parameters propagate through the subsystems and components, potentially affecting the flight path, payload capability, mission security (Afilipoae *et al.* 2018; Brevault *et al.* 2020; Dépincé *et al.* 2007; Dupont *et al.* 2019). To manage such complexity of these aerospace vehicles, the preliminary conceptual studies relay on dividing this system into several disciplines, we mainly list the trajectory, propulsion, mass sizing, and aerodynamics. Since even the slightest malfunction can lead to disastrous economic, material, and human consequences, ensuring the reliability of each discipline is imperative. Multidisciplinary design optimization (MDO) methods have been applied to launch vehicles in different applications to develop reliable models regarding performance, cost-effectiveness, and security requirements (Brevault 2015; Ullah *et al.* 2013).

In pursuit of performance, the propulsion system of the launch vehicle remains a primary subject of study. Dupont *et al.* (2017) demonstrated the dependency of mass and propulsion efficiency in a single-stage launch vehicle. Establishing this relationship is crucial for enhancing the design within the other disciplines (Afilipoae *et al.* 2018). The structural disposition of the launch vehicle is another key aspect of design analysis, which is also influenced by means of the propulsion and mass efficiency. The launch environment affects the aerodynamic response of the vehicle, making it essential to consider the shape of the launch vehicle as a function of structural mass sizing (Jia *et al.* 2020). Aerodynamic lift and drag coefficients play a crucial role in determining the thrust of the launch vehicle. Huang and Chudoba (2005) illustrated the connection between conceptual design dilemmas and trajectory analysis for launch vehicles. Geometry, structure, fuel consumption, and other interdisciplinary variables were considered for an optimal trajectory scenario of a launch vehicle as it is shown in Park (1998).

Multi-criteria design analysis for suborbital rockets aims to expose overall performance on one hand and ensure the projection of controllability and security of the launch mission on the other. Both aspects must be analyzed simultaneously. Multi-objective studies interpolate the different design variables representing various disciplines while respecting specific constraints. The aim of this study is to optimize the trajectory optimization of a suborbital launch vehicle by addressing multiple criteria, with performance, security, and control as the primary targets for conceptual design analysis.

Launch vehicle conceptual design

Duo to the interconnection of physical phenomena induced by different operations of the launch vehicle, design consideration must account for these links to develop a reliable and high-performance model. MDO is a research field involving engineering applications, numerical advancements, and strategies that integrate multiple disciplines. For each considered discipline, the defined design variables are studied based on the governing phenomena at both the local subsystem level and the global system level. To obtain a comprehensive view of the design set, it is important to define the main operative disciplines, including functional systems that ensure optimal performance mission security. After analyzing various models, launch missions, and environmental conditions, the following disciplines were selected as primary targets.

Mass sizing

It is necessary to define the total mass of the launch vehicle M_{total}, which primarily depends on the amount of required propellant m_p and the structural characterizing mass, as it is stated in Eqs. 1 and 2. Regarding the structural mass, the main components include the tanks, engine, and payload fairing system. An additional mass term is introduced in Eq. 3 to account for other accessory systems of the launch vehicle. To determine the required thrust of the vehicle, mass sizing must consider various structural limitations for ensure stability, as represented in Eq. 4. The given ratio by Eq.5 defines the capability to lift the mass of the vehicle

$$M_{total} = m_p + Ms + m_{payload}. \tag{1}$$

$$m_p = \dot{m}t. \tag{2}$$

$$M_S = m_{engine} + m_{tanks} + m_{payload/fairing} + M_{Add} \tag{3}$$

$$m_p \ = \ m_l \left(exp\left(\frac{\Delta V}{C}\right) \ - 1\right). \tag{4}$$

$$\frac{Thrust}{weight} = \frac{mdotC}{M_{total}g_0} \geq 1.5 \tag{5}$$

Propulsion

The propulsion system is a critical factor influencing other disciplines. Ensuring a high-performance propulsion system is essential to generate sufficient thrust to support the total mass of the vehicle M_{total}. In this study, the RP1-LOX mixture has been selected as the propellant. Table 1 provides the details of the propellant mixture.

Table 1. Propellant energy characteristics.

Propellant	Density (g·mL)	Mixture ratio	Isp sea level (s)	Temperature of combustion (°C)
RP1-LOX	0.820	2.5-2.8	220-265	3,400

Source: Elaborated by the authors.

The mathematical formulation of the generated thrust F_T is given in Eq. 6, where $\dot{m}$represents the propellant mass flow rate, v_e is the exhaust velocity, A_e is the exhaust area, P_e is the exit pressure, and P_a is the atmospheric pressure. Specific impulse (I_{sp}) in relation to exhaust velocity is also defined in Eq. 7.

$$F_T = \dot{m}v_e + A_e(P_e - P_a). \tag{6}$$

$$I_{sp} = \frac{v_e}{g_0}. \tag{7}$$

Aerodynamics

The conceptual design of the sounding rocket should consider the aerodynamic loads, which is represented by the drag D. Variables such as air density, drag coefficient C, as formulated in Eq.9, and characterizing velocity V, calculated by Eq. 8, directly affect the overall behavior of the rocket, as represented by Eq. 7. These variables are interconnected with the other listed disciplines, making the conduction of multidisciplinary analysis is crucial.

$$D = \frac{1}{2}\,\rho\,C\,V^2 \tag{8}$$

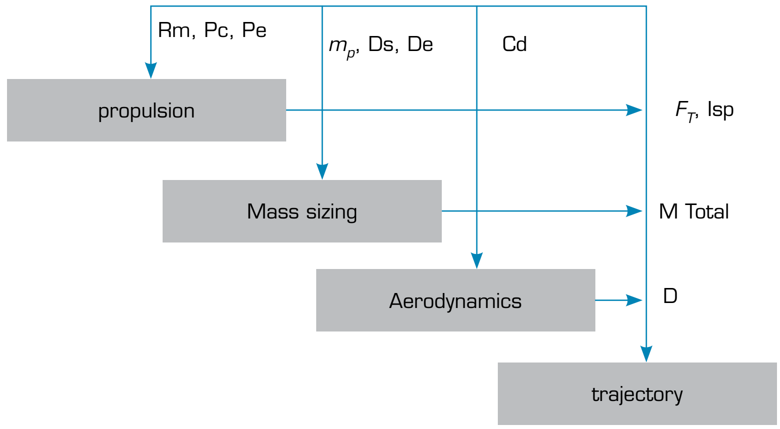
$$C = A_t P/mdot \tag{9}$$

Trajectory

Altitude, velocity, and the variable mass of the rocket during the launch profile will be analyzed in this study. The previously listed disciplines will be explored for design optimization, and the resulting outcomes will be reintegrated into the trajectory analysis to define the best launch scenario. Equation 10 indicates the dynamic behavior of the launch vehicle:

$$F_T - mg_0 - D = ma \tag{10}$$

For conceptual design purposes, the listed disciplines function within a hierarchical architecture system. From a performance perspective, the different disciplines are analyzed following that hierarchical order, as presented in Fig. 1; the design variables do affect the performance of each subsystem locally and the overall design entity. For stability analysis, the main goal is to ensure a safe launch mission. Regarding control, it is necessary to manage the functioning of the design set discipline by discipline. For the propulsion discipline, the controllability is explained by the regulation system for propellant flow, mass management, combustion rate, and more. Controllability is also a key factor in trajectory analysis. For a defined path, the launch vehicle must maintain its modeled position and time sequence for a specified velocity. Following the proposed approach, the propulsion discipline is the first operative system to be managed, where performance remains a primary objective.



Source: Elaborated by the authors.

Figure 1. Launch vehicle multidisciplinary design setups.

To handle the listed disciplines, the proposed approach targets the propulsion module. The performance is examined in an early phase of design, and for the other disciplines, the outputs of propulsion are injected to verify their consistency with structure, aerodynamics, and trajectory. The launch vehicle is fundamentally designed to accomplish the launch mission, and it is necessary to establish the requirements for a safe, high-performing, and optimized launch scenario. Reviewing some proven suborbital launch vehicles in the market is essential to extract the performance requirements for such a mission. As seen in Table 2, regarding payload capacity and the reached altitude, some launch vehicles are equipped with a cabinet or capsule for human occupation.

Table 2. Suborbital launch vehicle performance review.

	Mercury-Redstone	SpaceShipOne	Blue Origin NS	ARCA Haas 2CA (under developments)
Total mass (kg)	30,000	3,600	75,000	2,420
Payload to suborbital (kg)	1,800	Crewed cabinet	Crewed capsule up to 6 people	120
Maximum thrust (kN)	350.0	73.5	1,020.0	40.0
Specific impulse (s)	215.0	250.0	260.0	230.0
Burn time (s)	143.5	87.0	141.0	85.0

Source: Elaborated by the authors.

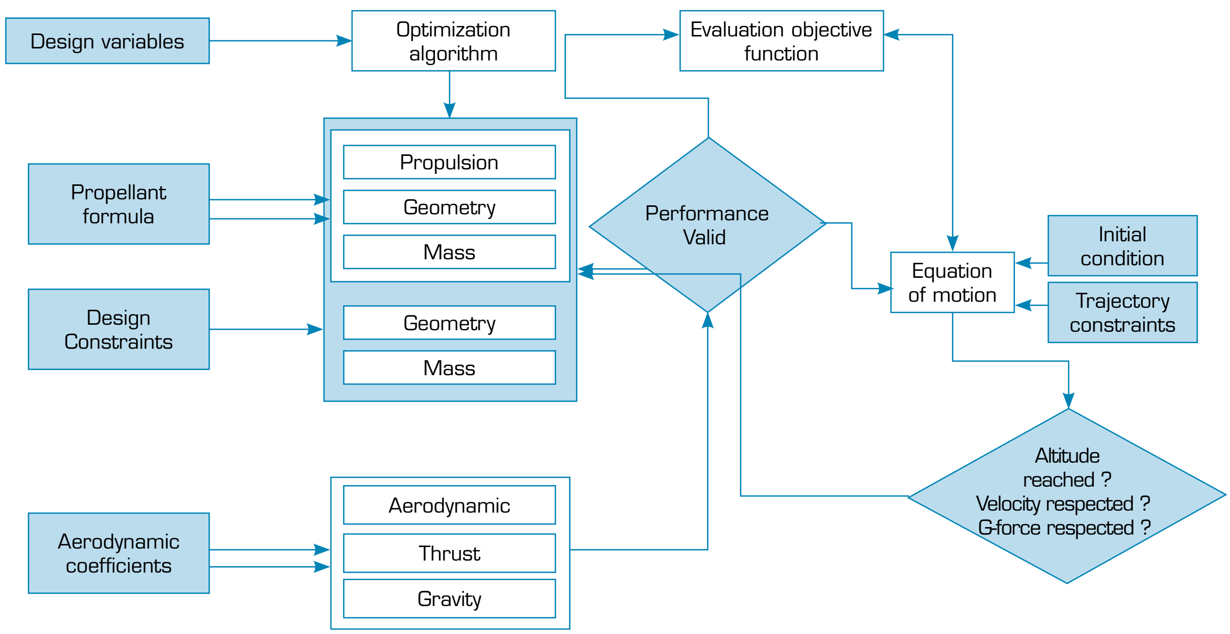
The literature review indicates that thrust is a function of the total mass of the launch vehicle. The specific impulse, which represents the efficiency of a propulsion system for the listed vehicles, varies from 215 to 260 seconds. After analyzing the performance of the listed launch vehicles, it is necessary to establish the design. This is explained by the performance given in Table 3, where the satisfying those requirements is essential for lifting a payload as mentioned. This capacity is estimated to be satisfactory for handling a human experience, which is addressed through the design analysis. The multidisciplinary approach is the most effective method for achieving these listed requirements.

Table 3. Launch mission requirements.

Launch vehicle design	Designed value
Reached velocity	1.5 km·s
Altitude	Up to 100 km
Generated Thrust	1,200 kN
Total mass not to exceed	75 10^3 kg
Burning time	Up to 220 s
Payload capacity	200 kg

Source: Elaborated by the authors.

The architectural disposition of the prefeasibility model is set from local to global. For each discipline, the analysis aims to study the influence of the design variables under the governing principles of each discipline. Since these disciplines are integrated into a hierarchical functional system, verifying the coupling model is important – both from local to local and from local to global interactions. The global system is the assembly of the different local-global interactions, and as defined in the previous section, performance, security, and control should be assessed at every phase of the study. Figure 2 illustrates the process of the launch vehicle design analysis; the first step is characterizing the design variables, which are considered crucial for an optimized model. By defining the optimization algorithm, the listed disciplines cooperate to achieve the required performance; this is the first selection criterion for this study. By evaluating of the objective function representing the performance, the trajectory analysis of the launch vehicle is conducted. Reached velocity, altitude, and adherence to the vehicle's G-force constraints are considered the second design criterion in the post-performance analysis.



Source: Elaborated by the authors.

Figure 2. Launch vehicle design analysis and optimization procedure.

Proposed multidisciplinary optimization approach

Interior point methods are numerical iterative solutions used for solving both linear and nonlinear convex optimization problems that include inequality constraints and multiple design variables. Interior point methods approach a solution for each design variable from the interior or exterior of the feasible region (optimization domain) but are never on the boundary (upper or lower limits). There are two important interior point algorithms: the barrier method and the primal-dual IP method. The primal-dual method is preferred due to its efficiency and accuracy. The choice of the interior point method to address our study owing to its numerous advantages over other methods in particular the genetic algorithm method or simulated annealing method. Overall, the interior point method is designed to solve large-scale optimization problems with a large number of variables and constraints, linear or nonlinear. In contrast, the genetic algorithm method and the simulated annealing method require a larger computational effort when the problem size increases, and they also require a longer convergence time than the interior point method. Moreover, the interior point method is effective in addressing problems containing a large number of linear and nonlinear constraints. On the other hand, the other two methods are used for unconstrained or less-constrained problems, where they require more complex mechanisms to address this type of constrained problems, such as a penalty function, which increases the complexity of the optimization problem. As final point, using the internal point method on the MATLAB platform is more mature and simpler in comparison with other methods that require additional steps and add complexity to implement these methods, the formulation of the optimization problem is given as follows:

$$\begin{aligned}&Objective\ function: min\ M_{total} = M_s + M_p + M_{payload}\\&With\ respect\ to\ (P_c, P_e, M_p, R_m, D_s, D_e)\\&Subjected\ to: 0.4 * Pe - Pa \leq 0\\&0.8 * De - Ds \leq 0\\&Initial\ guess\ = (lb + ub)/2\end{aligned} \tag{11}$$

The objective function M_{total} is linked to the different design variables in a manner that defines the total mass of the launch vehicle. With the dimensioning of the propulsion motor, propellant mass, and other structural variables, the total mass is established.

Regarding the optimization problem, as formulated in Eq.11, the objective function is the minimization of the total mass M_{total} of the launch vehicle, integrating all the required equipment at the moment of the launch. By minimizing this function, the generated thrust is capable of lifting the launch vehicle and ensuring a safe mission. This is deemed satisfactory by verifying the thrust-to-weight ratio mentioned in Eq. 4. Two major constraints can be listed for this optimization problem: the first set is related to the propulsion discipline, specifically the ratio defining the combustion chamber pressure (P_c), exit diameter (P_e), and mixture ratio (R_m). Another constraint is related to the structural stability, specifically the structural diameter of the launch vehicle (D_s). After defining the objective function, Table 4 introduces the design variables of the launch vehicle. These variables are related to different design disciplines, as they affect the optimization procedure. The given upper and lower bounds represent the optimization range for each design variable, while the initial guess is used to initiate the optimization process. These variable bounds will be introduced to the MATLAB IP optimization algorithm.

Table 4. Targeted design variables for MDO optimization procedure.

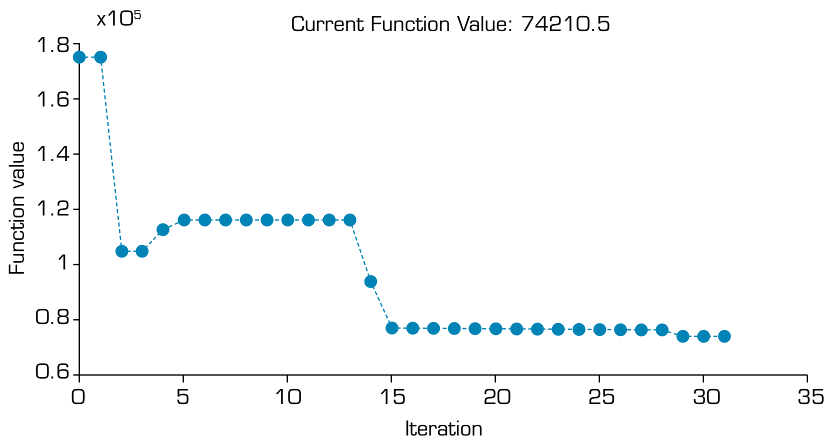
Disciplines variables	Variable upper bound	Variable lower bound	Initial guess
Combustion chamber pressure (Pascal) (P_c)	80 10^5	120 10^5	100 10^5
Nozzle exit pressure (Pascal) (P_e)	0.1 10^5	1 10^5	0.5 10^5
Propellant mixture ratio (R_m)	2.6	2.8	2.7
Thrust-to-weight ratio T/W (criteria)	1.3	1.8	1.3
Exit diameter (m) (D_e)	1.0	2.0	1.5
Mass propellant (kg)	34,000	42,800	38,400
Launcher structural diameter (D_s)	3	5	4
Total mass (kg)	-	-	174,640

Source: Elaborated by the authors.

RESULTS AND DISCUSSION

As defined in the section, the requirements of a multi-criteria design model have been established, ensuring a safe, high-performing, and optimized design regarding multiple aspects: vehicle velocity, achieved altitude, generated thrust, and total vehicle mass. By conducting the MDO analysis, the first criterion of the study, which is the minimization of the total mass function M_{total}, has been achieved as presented in Fig. 3.

The primary objective of the second phase of this study is to incorporate the results of this function to enhance the trajectory details of the launch vehicle. Table 5 lists the results of the optimization method regarding the total mass function M_{total}. As explained in the previous section, by minimizing this function, the performance of the propulsion discipline, in terms of the generated thrust, is optimized. The minimization is significant compared to the initial statistical value before initiating the optimization procedure, as presented in Table 4. The various listed design variables listed in the table are aim to ensure that optimized value. The required propellant mass for a safe mission approximately 40,000 kg, with different propulsion and structural design variables managed through the optimization formula.



Source: Elaborated by the authors.

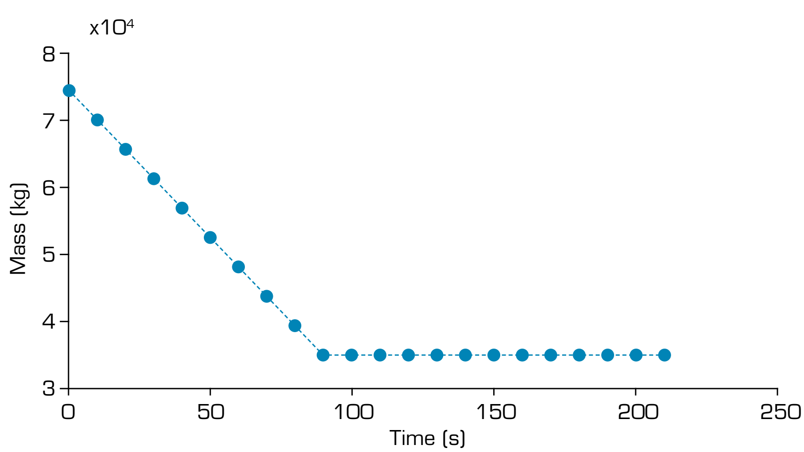
Figure 3. Evaluation of the objective function M_{total} (total mass of the vehicle).

Table 5. Launch vehicle trajectory specifications (optimization results).

Vehicle performances optimization results	
Propellant mass M_p (kg)	39,365
P_c (Pascal)	106.889810^5
P_e (Pascal)	0.110^5
D_s (m)	0.58
Isp (s)	265
Total mass (kg)	74,210

Source: Elaborated by the authors.

The results of this study indicate the minimum required mass for this launch vehicle, which is set as the objective function of this model. The total mass of the vehicle is determined to be 74,210 kg. It is evident that, aside from the structural mass of the vehicle, a substantial amount of it is required for propellant, specifically 39,365 kg in this case study. The total vehicle mass decreases during the launch operation due to propellant consumption, as illustrated in Fig. 4.

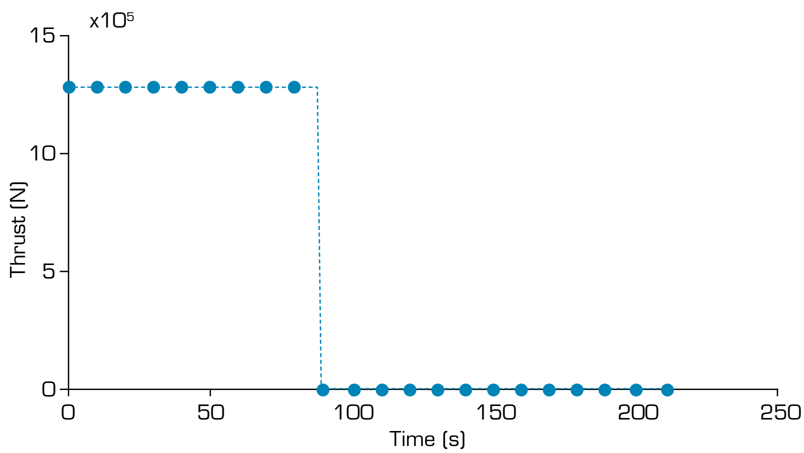


Source: Elaborated by the authors.

Figure 4. Propellant mass consumption during the launch mission.

For launch vehicle's performance optimization, Table 5 lists the various characteristics established for this mission. The vehicle's thrust reaches 120 kN, as shown in Fig. 5. The calculated thrust of this launch vehicle is aimed to stand against the gross lift-off weight of the vehicle, which represents the heaviest weight of the launch vehicle during the launch time; the extraction of the exhaust reactions gazes will lighten the weight gradually. Additionally, the required fuel for the launch is detailed. With this fuel quantity, the vehicle reaches an altitude of 100km, as depicted in Fig.6.This altitude is deemed adequate, as the study is focused on suborbital flights. The velocity of the launch vehicle progressively increases to approximately 1.2 km·s^{-1} (Fig. 7), it is important to highlight that this velocity was studied under its influence for possible crewed mission and kept in range during the optimization process. The duration of the launch operation, meaning the time required to burn the specified amount of propellant, is recorded as 202 seconds.

Regarding mission safety, as previously explained, aerodynamic stress must be considered for specific suborbital flights. Figure 8 presents the dynamic pressure, which reaches its peak at 50 seconds but remains within safe and manageable limits. Another aspect analyzed in response to safety concerns is the acceleration of the launch vehicle, represented in this case by the G-force. Figure 9 illustrates the evaluation of this generated force during the mission, peaking at 2.6 g, ensuring the vehicle remains safe for human space travel mission. To summarize the results of this study, the launch vehicle meets the addressed requirements outlined in considering a vertical flight path.



Source: Elaborated by the authors.

Figure 5. Generated thrust during the launch mission.

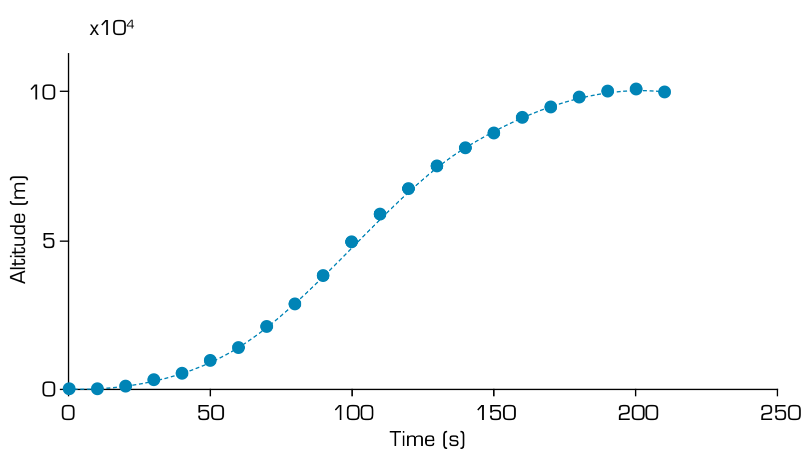


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Figure 6. Reached altitude during the launch mission.

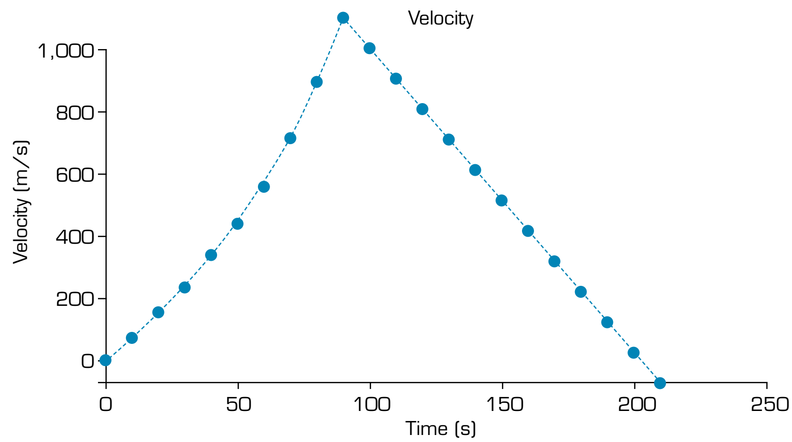

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Figure7. Evaluation of the vehicle resulted velocity during the launch.

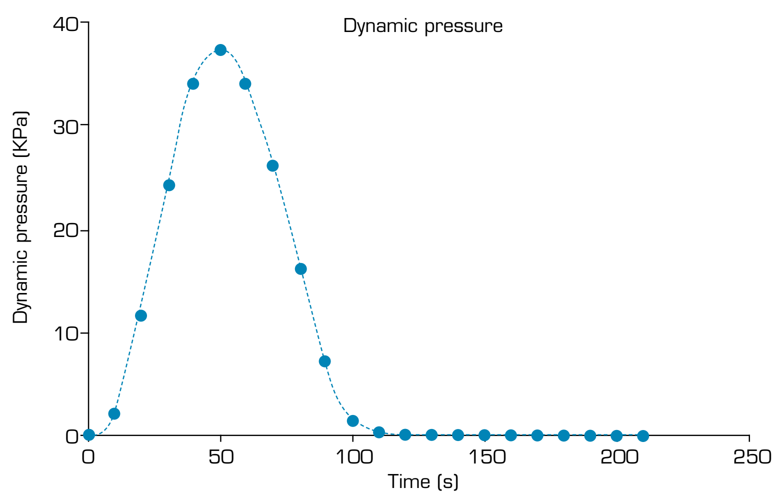

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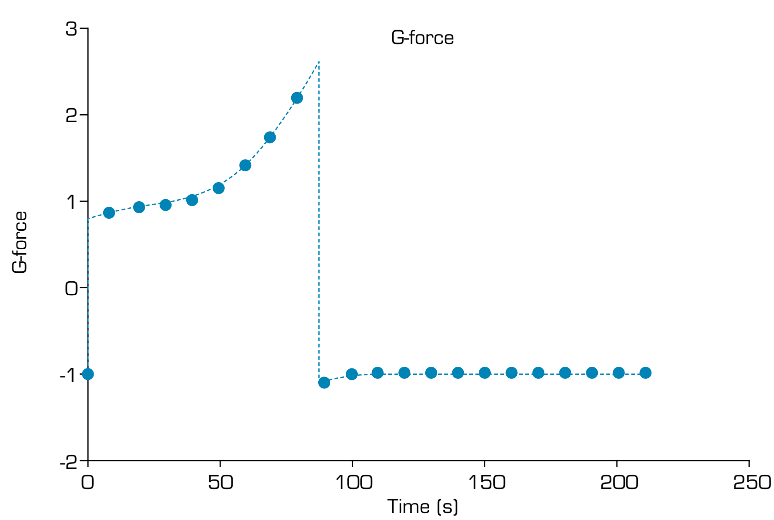
Figure 8. Evaluation of the dynamic pressure during the launch mission.



Source: Elaborated by the authors.

Figure 9. Evaluation of G-force during the launch mission.

CONCLUSIONS

In this work, an optimization procedure was presented using MDO analysis. The performance of the suborbital launch vehicle was enhanced through the minimization of the total mass of the vehicle M_{total}. The incorporation of various design variables to establish the optimized solution was executed. The generation of sufficient thrust for an appropriate fuel consumption was the first step toward achieving the study's objectives, resulting in significant weight savings. Achieving high launch vehicle performance in multidisciplinary terms alone was insufficient to complete this study. As part of the post-performance analysis, the implications of performance for a launch mission were assessed to ensure compliance with safety and control criteria. After analyzing the different design variables of the launch vehicle for a vertical flight, the study of the trajectory of the lunch vehicle with different angles remain as a perspective. Additionally, considering trajectory optimization for various launch scenarios is essential, with a focus on the potential reusability of major vehicle components, such as propulsion motors, to overcome the limitations of this work.

CONFLICT OF INTEREST

Nothing to declare.

AUTHORS' CONTRIBUTION

Conceptualization: Hemza L; **Methodology:** Hemza L; **Software:** Mohamed Amine B; **Validation:** Hemza L, Mohamed Amine B; Khaled T, and Hatem H; **Formal analysis:** Hemza L, Mohamed Amine B, Khaled T, and Hatem H; **Investigation:** Hemza L, Mohamed Amine B, Khaled T, and Hatem H; **Data Curation:** Hemza L, Mohamed Amine B, Khaled T, and Hatem H; **Writing - Original Draft:** Hemza L, Mohamed Amine B; **Writing - Review & Editing:** Hemza L, Mohamed Amine B, Khaled T, and Hatem H; **Supervision:** Hemza L; **Final aproval:** Hemza L.

DATA AVAILABILITY STATEMENT

All data sets were generated or analyzed in the current study.

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