A Tool for Preliminary Design of Rockets

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Abstract

The only way mankind can explore space is with the use of space launch vehicles, commonly known as rockets, which can carry payloads from Earth into Space. The main goal in the rocket design is to reduce the gross lift-off weight (GLOW) and increase the payload ratio, giving them the capacity of having the maximum amount of payload, within its range. The trajectories of spacecrafts and launch vehicles are permanently being optimized, due to the demand of increased payload and reduced propellant requirements. The main objective of this study is to find the best design of a rocket for a specific orbit and payload mass, considering the best trajectory that can be used. A tool was developed allowing the study of different configurations, varying the number of stages and boosters in order to achieve the best rocket design. Beside the different configurations the tool allows the variation of design parameters, calculating for each configuration and parameters the different masses and dimensions that characterize a rocket, and then iteratively with the trajectory will achieve the minimum GLOW and obtain the maximum payload ratio. In order to validate the code comparisons with Vega and Proton K launchers were made. At last an optimization of the Ariane 5 launcher is made by varying its configuration and some design parameters.

Keywords: Multistage Rockets, Gravity Turn, Trajectory optimization, Rocket optimization, Maximize Payload Ratio

1. Introduction

The main objective of rocket design is to reduce GLOW and increase the payload ratio for a specific mission. In this work, a tool is developed for the preliminary design of rockets, where the user can make decisions about the configuration and define the interval of variation of the design parameters. Then, it will obtain a new launcher optimized with respect to the goals desired.

The preliminary design and optimization of rockets has been the focus of several research studies. The minimum GLOW and maximum payload ratio can be obtained by staging and trajectory optimization [6, 7]. The collaborative optimization is an alternative design architecture, whose characteristics are well suited for launch vehicle design [5]. The Multidisciplinary Optimization (MDO) strategies are based in the exploration on the interaction of sub-systems representative models and on the exploitation of their coupling, in order to find the optimal conception parameters. Several works using MDO have been developed in the preliminary design and optimization of rockets [3, 9]. Recently, one work in multi-attribute evaluation provided an interesting approach in the conceptual design of rockets with a cost model [12].

The calculation of the most adequate trajectory will increase the launchers efficiency. By minimizing drag and gravity losses, a trade-off analysis needs to be performed [1, 7]. For the atmospheric flight, several works consider the use of gravity turn (GT) [1, 8, 10]. For the exo-atmospheric flight, there are several methods for trajectory optimization used in works of the rocket design [2, 11].

There are two methods of numerical approaches in trajectory optimization, direct and indirect methods. In a first overview the direct methods consist in discretizing the state and the control and thus reduce the problem to a nonlinear optimization with constraints. The indirect methods consist on solving numerically boundary value problem derived from the application of the Pontryagin Maximum Principle and lead to the shooting methods [4]. The main advantages of using indirect methods are their high solution accuracy and the guarantee of satisfying the optimal conditions of the solution [11].

2. Rocket Design

In order to launch the satellites in orbit it's necessary to escape from the Earth atmosphere and gravity. To accomplish that, a large amount of ΔV is necessary and with the current technology

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only rockets are able to achieve that. Currently, a SSTO is still impossible and only multistage configurations can achieve space, that allow optimize the structure and propulsion of each stage for different conditions. Often, more than 90% of the rocket is propellant, meaning that rockets need a strong structure for accommodate all the propellants and its powerful engines to burn them. Their structure is long and thin allowing the reduction of Drag during the trajectory. However this will bring structural problems when normal loads are applied in the structure.

2.1. Delta-V calculations

The ΔV budget of a mission is represented as

$$\Delta V_{Design} = \Delta V_{orbit} + \Delta V_{gravity} + \Delta V_{drag}, \quad (1)$$

where ΔV_{orbit} is the injection velocity required for the desired orbit, $\Delta V_{gravity}$ and ΔV_{drag} are respectively the total gravity and drag losses. This equation allows us to make an estimate on the required ΔV consumption and thus the required fuel needed to reach the trajectory and LEO. Other losses due to the manoeuvring and static pressure difference at the nozzle exit during flights are considerably smaller compared to gravity and drag losses. This ΔV budget must be larger or in exceptional cases equal to ΔV of the mission, otherwise the payload/spacecraft wouldn't be successful reaching the desired orbit.

The speed of a satellite in a circular orbit is defined by

$$V_{orbit} = \sqrt{\frac{\mu}{R}}. \quad (2)$$

where R is the radius of the orbit and μ the gravitational parameter of the planet.

The gravity loss is by definition

$$\Delta V_{gravity} = \int g \sin\gamma dt, \quad (3)$$

where g is the gravitational acceleration γ and the flight path angle. The drag loss is by definition

$$\Delta V_{drag} = \int \frac{D}{M} dt, \quad (4)$$

where D is the drag force and M the mass of rocket at each time.

A preliminary estimation of gravity and drag losses is necessary for the development of the tool. These heuristics are defined by:

$$\Delta V_{gravity} = 0.08 < V_{orbit} < 0.12 \quad (5)$$

$$\Delta V_{drag} = 0.008 < V_{orbit} < 0.012 \quad (6)$$

2.2. Rocket equations

The mass of the rocket is the sum of the payload mass with the structural and propellant masses of each stage. It's possible to define the following ratios using mentioned masses.

The payload of a specific stage is the mass of everything above that stage.

$$\lambda_N = \frac{M_{0N+1}}{M_{0N}}, \quad (7)$$

where M_{0N} is the total mass and M_{0N+1} is the mass of everything above that stage.

Total Payload Ratio is equal to the product of all payload ratios of all rocket sections and is equal to

$$\lambda_{tot} = \prod_{i=1}^{N} \lambda_i. \quad (8)$$

The Structural Ratio measures how much of the launch vehicle is structure, depends exclusively of the stage.

$$\varepsilon_N = \frac{M_{sN}}{M_{sN} + M_{pN}}, \quad (9)$$

where M_{sN} and M_{sN} are respectively the structural and propellant mass of the N stage.

The Propellant Mass Ratio measures how much of the vehicle is propellant.

$$\varphi_N = \frac{M_{pN}}{M_{0N}} = (1 - \varepsilon_N)(1 - \lambda_N), \quad (10)$$

where M_{pN} and M_{0N} are respectively the propellant mass and the total mass of the N stage.

The Mass ratio is a measure of the efficiency of a rocket for any given efficiency a higher mass ratio typically permits the vehicle to achieve higher ΔV.

$$\Lambda_N = \frac{M_{0N}}{M_{0N} - M_{pN}} = \frac{1}{1 - \varphi_N} \quad (11)$$

where the M_{pN} and M_{0N} are respectively the propellant mass and the total mass of the N stage and φ is the propellant mass ratio.

3. Trajectory

The rocket flight can be divided in two main phases: the atmospheric flight and the exo-atmospheric flight. The frontier between them is not well defined, but can be linked to an altitude around 120 kilometres, where atmosphere forces can be neglected.

3.1. Ascent Phase

A typical launcher starts its mission locked to the Launchpad. The engines are not yet ignited or the thrust provided is still lower than the vehicle weight. The fist phase is a vertical lift off, required to gain velocity before the gravity turn phase and avoid any contact with the launchpad tower. The duration of

the ascent phase varies from vehicle to vehicle and the current tendencies is to decrease, because of the developments made in space technology.

In the figure 1 is represented the launch sequence between lift-off and the final orbit, and the division of altitude in two parts, one where the atmosphere can't be neglected and other where the atmosphere is neglected.

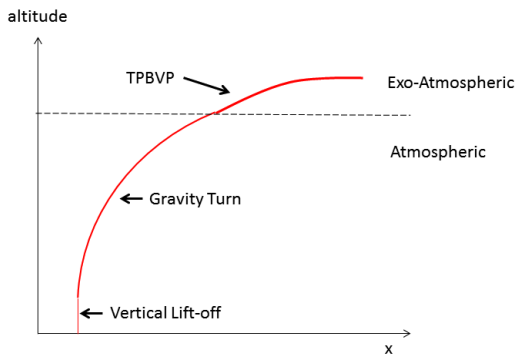

Figure 1: Launch Sequence.

3.2. Gravity Turn

The gravity turn phase, also named "zero lift trajectory" is defined as a non-guided trajectory. Primarily, the launch vehicle crosses the atmosphere, only influenced by the gravity force. The angle of attack must be zero, because even a small angle of attack can lead to structural failure of the vehicle. Most launch vehicles have requirements of strength, regarding axial direction, that are fulfilled by compromising the structural strength in the transverse direction. As a result, they cannot fly at any significant angle of attack, without risks of catastrophic failure of the whole structure. The trajectory data includes altitude, flight path angle variation, thrust, drag and total system mass. The equations of motion, for an object in flight with gravity turn trajectory, are shown below:

$$\dot{X} = V cos\gamma, \tag{12}$$

$$\dot{H} = V sin\gamma, \tag{13}$$

$$m\dot{V} = T - D - (mg - \frac{mV^2}{Re + h})sin(\gamma), \tag{14}$$

$$\dot{\gamma} = -\frac{1}{v}(g - \frac{V^2}{Re + h})cos(\gamma), \tag{15}$$

where X, H, V, γ, T, D, m, g and Re are respectively downrange distance, altitude, velocity, flight path angle, thrust, drag, mass, gravity and earth radius.

3.3. Free-flight phase

As the vehicle emerges from the atmosphere a better trajectory than the gravity turn can be used. It has a (present) position and velocity, and it is desired to achieve a different position and velocity at thrust termination. The only two forces that can cause the vehicle to accomplish the desired position and velocity changes are thrust and gravity. Continuous problems of trajectory optimization are traditionally solved by direct or indirect methods. We want to minimize the final time i.e. the duration of the free-flight phase with a TPBVP, less time means less propellant because we assumed constant thrust, then this will result in minimize the GLOW.

For minimum time problems, the cost function can be written as

$$J = t_f \tag{16}$$

the initial conditions of TPBVP are the final conditions of gravity turn.

$$t_0 = t \tag{17}$$

$$x_0 = x \tag{18}$$

$$y_0 = y \tag{19}$$

$$V_{x0} = V_x \tag{20}$$

$$V_{y0} = V_y \tag{21}$$

the final conditions, for the particular case of circular orbit are

$$y_f = h \tag{22}$$

$$V_{xf} = Vc \tag{23}$$

$$V_{yf} = 0 \tag{24}$$

Applying transversality condition and the Minimum Principle, which states that the Hamiltonian must be minimized. We have a $\lambda_1 = c_1$ and $\lambda_{1f} = c_0$, so $c_1 = 0$.

We have a well defined TPBVP, with four state equations, and four costate equations.

$$\dot{x} = V_x \tag{25}$$

$$\dot{y} = V_y \tag{26}$$

$$\dot{V_x} = \frac{F}{m_0 - \dot{m}t}(\frac{-\lambda_3}{\sqrt{\lambda_3^2 + \lambda_4^2}}) \tag{27}$$

$$\dot{V_y} = \frac{F}{m_0 - \dot{m}t}(\frac{-\lambda_4}{\sqrt{\lambda_3^2 + \lambda_4^2}}) - g \tag{28}$$

$$\dot{\lambda}_1 = 0 \tag{29}$$

$$\dot{\lambda}_2 = 0 \tag{30}$$

$$\dot{\lambda}_3 = -\lambda_1 \tag{31}$$

$$\dot{\lambda}_4 = -\lambda_2 \tag{32}$$

Thus there are eight differential equations.

$$\dot{x} = f(t, x, \lambda) \tag{33}$$

$$\dot{\lambda} = g(t, x, \lambda) \tag{34}$$

Now there are five initial conditions and five final conditions.

The steps that shooting method used are:

1. Guess λ_{10} , λ_{20} , λ_{30} , λ_{40} *and* t_f
2. Integrate the $\dot{x}$ and $\dot{\lambda}$ forward to $t = t_f$
3. Compute the final conditions:

$$\Psi 1 = y_f - h_{circ} + R_{Earth} \tag{35}$$
$$\Psi 2 = V_{xf} - V_{circ} \tag{36}$$
$$\Psi 3 = V_{yf} \tag{37}$$
$$\Psi 4 = \lambda_{1f} \tag{38}$$
$$\Psi 5 = H_f + 1 \tag{39}$$

   where λ and tf are changed iteratively

4. Until the convergence on $\Psi i(\lambda_{0,tf}) = 0$ for $i = 1, .., 5$

In the figure 2 is possible to observe, the initial conditions, and a few iterations of Shooting Method until the convergence has achieved.

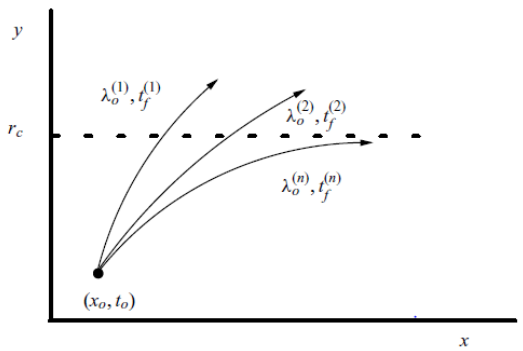

Figure 2: Shooting Method for TPBVP

3.4. Drag Model

The drag coefficient is a function of the atmospheric conditions and the geometry of the rocket. An assumption was made, the drag force is independent of the length of the vehicle (assuming skin friction drag to be negligible).

$$D = \frac{1}{2}\rho C_D S_{ref} V^2, \tag{40}$$

where ρ is the density of the atmosphere, the C_d is the drag coefficient, S_{ref} is the cross-sectional area of the body and V is the velocity of the vehicle. A simple model for the C_d was adopted and its variation only depends on the Mach number.

3.5. Atmospheric Model

In this work, the U.S. Standard Atmosphere Convention of 1976 was used as a reference for altitudes lower than 86 kilometres, and the 1962 U.S. Standard Atmosphere was followed for altitudes above 86 kilometres, it models atmosphere up to 2000 kilometres.

The Knudsen number will define the boundary between the atmospheric and exo-Atmospheric flight, i.e. it will define the end of gravity turn and the beginning of the free-flight phase of the flight. The Knudsen number is dimensionless defined as

$$Kn = \frac{\lambda_{kn}}{L}, \tag{41}$$

where the λ_{kn} is the mean free path of free-stream molecules and L is a length characteristic, the value assumed was the cone radius that it's the same of last stage radius.

3.6. Gravitational Model

The external force caused by gravity is the main force that acts on the launch vehicle. Gravity is the natural phenomenon by which physical bodies are attracted to each other with a force proportional to their masses.

The local gravity is calculated with the following equation

$$g = \frac{g_0}{1 + \frac{h}{R_e}}, \tag{42}$$

where h is the current altitude, R_e the radius of Earth and g_0 the surface gravity.

4. Rocket Knowledge Database

For a better understanding of the differences between launchers, and to compare converging parameters between them, a database of well-known launchers was developed. It gathers information about their different masses, dimensions and propulsion, to understand if the concept of new designed rockets is "real", and if their values can be compared with similar launchers.

For each stage of each launcher a lot of information is available in its user's manual but only are selected the principal characteristics that define a rocket. In the table 1 is a list of the information gathered.

Some information wasn't available for all launchers, for example the nozzle area ratio is one characteristic that is very difficult to obtain, such as aerodynamic performance. The most important information gathered is about the structural factor of each stage, that allows us to know how much mass of each stage is structure and to give an appropriate range of values of Structural factor of each stage for mass model developed in this work. The average value obtained was used in mass model as first value for the iterations of Structural factor.

5. Tool for Rocket Design

The computer simulation program was designed to perform a rocket optimization. This means that it optimizes all the masses and the dimensions of the vehicle as well as the last part of the trajectory. To develop this work a Matlab code was developed, in

Characteristics	Units
Class	Small/Medium/Heavy
Year	-
Engine	-
Length	m
Diameter	m
Take-off mass	kg
Propellant mass	kg
Structural mass	kg
Structural factor	kg
Propellants	-
Burning duration	s
Thrust (vac)	kN
Isp (vac)	s

Table 1: Characteristics gathered for each Stage of each Launcher.

order to design the rocket and simulate trajectories into orbit. In this section is described how the tool achieves the best configuration for one combination of parameters and configuration. This algorithm is repeated for all the combination of parameters and configurations.

First, the user inputs his objectives: the orbit altitude and the payload mass. After this he has to make basic decisions about the configuration: how many stages there will be, and if boosters will be used in the first stage and how many there will be. It is assumed that the launch system starts from ground, with velocity equal to zero and launch from vertical.

The first ΔV evaluation assumes an estimate for gravity and drag losses. A good preliminary estimate of ΔV losses increase the velocity of program, i.e. the tool will need less iterations to find the best solution.

Before the user enters the mass model, he had already selected the configuration, some parameters, shown in table 2, and its variation for each stage,

Characteristics	Units
Number of Stages	-
Number of Boosters	-
Diameter Booster	m
Diameter	m
Nozzle Area Ratio	m
Propellant Selection	-
Thrust	kN
Thrust Booster	kN
ΔV division	-

Table 2: Configuration and Parameters introduced in the tool.

The propulsion parameter selected was the thrust instead of burn time. Due to the creation of database, the user can have an insightful idea of the values of thrust for each stage and different payload masses. The burn time is obtained after defining the thrust. The dimension parameter selected was the diameter instead of length, because defining the diameter will give an idea to the user of how much drag will be generated.

Afterwards, in Mass Model the program will compare the masses obtained with heuristic equations i.e. Mass Estimation Relationships from the Akin model, and will iterate the structural ratio, until it converges. During the mass model the program calculate in parallel the dimensions of the vehicle. The user already defined the diameter of each stage. Then the volume is calculated and will iterate inside Mass Model. After this all the masses and dimensions of the launcher are specified, the launcher is ready to trajectory simulations.

Then the trajectory has to be run, according to the three distinct phases of flight, firstly Vertical flight, secondly Gravity turn and then the free-flight phase, until the objective is fulfilled. After knowing the best trajectory new values of ΔV drag and gravity and how much Propellant mass of the last stage remains or need to be added, the program will transform those propellant in a ΔV, and then will iterate with those new values, regarding the Mass model. The program will iterate until it finds a solution, i.e. when the ΔV losses used in the development of the launcher, are equal to the losses obtained in the trajectory and the payload reaches its objective with the right amount of propellant. The Launch Vehicle is optimized considering the initial objectives.

In the figure 3 it's represented the algorithm of the tool only for one combination of parameters and configuration. For each combination of these the tool will run the algorithm described earlier and presented next. In the end, it will select the combination of parameters and configuration with minimum GLOW and maximum payload ratio.

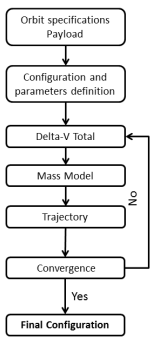

Figure 3: Algorithm for the Design of the rocket

5.1. Mass Model

The mass model of this tool was developed to calculate all the masses, dimensions and parameters of the rocket needed to perform the trajectory for each combination of parameters previously selected. When the mass model starts, the tool has already defined the parameters and the configuration. Those parameters will be used later for the Heuristics MERs. For given stage velocities, the mass of the structure and propellants can be estimated using the ideal velocity equations, known as the Tsiolkovsky's Equation. The Lift-off weight is the sum of payload with all the calculated masses. An initial estimate of the structural factor is used based on heuristics selected from database. The structural and propellant mass of each stage are calculated with the following two equations:

$$m_{s_n} = \frac{\epsilon_n(k_n - 1)}{1 - \epsilon_n k_n} m_{pl}, \tag{43}$$

$$m_{p_n} = \frac{(k_n - 1)(1 - \epsilon_n)}{1 - k_n \epsilon_n} m_{pl}, \tag{44}$$

$$k_n = e^{\frac{\Delta V}{g I sp_n}}, \tag{45}$$

where ϵ_n and k_n are respectively the structural factor and the mass ratio for each stage.

Starting from the last stage, and knowing the Initial structural factor, the mass ratio k_n and payload mass, it is possible to estimate the structural and Propellant Mass of that stage. Then a heuristic comparison is performed using the values generated (propellant and structural mass, volume of that stage) and the parameters previous defined (Thrust and Nozzle Area Ratio). The Heuristic Mass generated will provide a new Structural factor, then the program iterates using this new value. The program will stop when the iteration of the Structural factor converges with a maximum tolerance of 0.1%. This will define the last stage. Now, the program has the conditions to estimate the masses of next stage, where the Mass of the Payload will be the sum of masses of everything above. The same process is applied until all masses are defined.

When the user chooses to use boosters during the first stage, he will select two ΔV, one for the "zeroth" stage considering the boosters and first stage burning simultaneously and one only for the first stage, after the boosters were discarded. A specific heuristic comparison for boosters is performed to validate the configuration.

In order to get appropriate dimensions for the Launcher, all the main dimensions are calculated. For each iteration of Structural factor in Mass model, a new propellant mass is generated. Knowing the density of Propellant, we're in conditions to know the volume and calculate the involving structure for each stage.

The figure 4 shows the iterative process of how the mass model calculates the propellant and structural masses for each stage (rocket section mass), where the ΔV is the ΔV assigned for the specific stage.

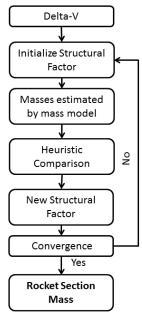

Figure 4: Mass model loop

5.2. Trajectory

After the lift-off the launcher follows a vertical trajectory until a designed altitude where Gravity Turn starts. Although the altitude was fixed for running the simulations presented in this thesis, it can be changed and be a parameter. After the end of the gravity turn i.e. the end of considerable atmosphere, the free-flight phase starts and the position and velocity of the vehicle is known, along with its destination (velocity and position). Considering the stages and respective thrust, the program will calculate how much burn time is necessary to achieve the desired orbit. Three different results can appear: arriving at a circular orbit at burnout, burn time necessary is less than was previous calculated by the mass model; burn time is not enough to achieve the desired orbit. These cases still depend on the effective gravity and drag losses determined during the trajectory. If their values aren't correct the mass model will iterate until the both values converge. The propellant mass missing or in excess in the last stage is converted to ΔV by Tsiolkovsky's equation. This will generate a new ΔV total. Then, the program will iterate until the value of ΔV converges with a tolerance of 0.01%.

When the program finishes, a complete launcher has been developed, and its data can be saved to try different orbits regarding that the maximum payload is the previously used for the initial design.

The figure 5 represents the trajectory part of the program described earlier.

6. Results

Before using the tool for its purpose, its validation was necessary. It was performed in two separated phases, starting with the validation of the trajectory and ending with the validation of the mass

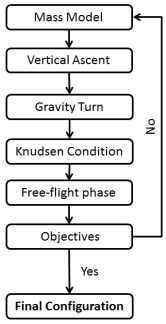

Figure 5: Trajectory

model. The main objective of the simulation with Vega launcher was to place 1500 kilograms of payload into a circular orbit, at an altitude of 700 kilometres, and the main objective of the Proton K was to place 19360 kilograms of payload in a circular orbit at an altitude of 200 kilometres. Posteriorly, a set of simulations was made for the preliminary design and optimization of a new launcher, using as initial parameters the values of Ariane 5. The optimization was based on the variation of some of the original parameters and the result was a new launcher.

6.1. Trajectory and mass model validation

To validate the trajectory, firstly, a simulation was done with each launcher, using the parameters presented in user's manual of the Vega and Proton K, to confirm if it managed to use the proposed trajectory.

The final times obtained at the end of the trajectories were lower than the sum of all burning and coast times, and in the last stage 34% and 38% respectively of the propellant mass wasn't burned, meaning that the trajectory used was considered validated for this two launchers. After the successful validation of the trajectory for each launcher. It's time to perform the validation of the mass model and of the entire tool.

The parameters introduced in the tool for the two simulations are presented in the table 2, all the parameters were equal to the ones presented in the respective user's manual.

The deviation obtained in the different masses are shown in the table 3. The most significant differences were obtained in structural mass, specially in the second stage where the difference was almost 30%. The structural and propellant mass of last stage are the ones that have more differences, specially the structural mass because last stage usually have "more structure", i.e. more avionics, wiring and a complicated thrust attitude controller for the re-ignitions, so the MERs used in Mass model doesn't have the same accuracy for upper stages as they have for lower stages.

-	m_p	m_s
1 Stage	-5.83	15.6
2 Stage	-4.1	29.8
3 Stage	-15.3	8.1
4 Stage	40.7	57.6

Table 3: % Deviation of the structural and propellant masses - Vega.

The program overestimates all the structural masses but on the other hand the propellant masses calculated were inferior with the exception of the last stage once again, that can be explained by the fact that the last stage of the original launcher have multiple burns and long coast times and that analysis isn't provided in the trajectory and for the difference in the others stages are explained for the non existence of the propellant margin. The biggest difference in the propellant masses excluding last stage was in third stage. The GLOW was a difference of 4.8%, that is inferior to 10% and for a tool of preliminary design the differences are relatively small. The mass model was considered validated for the Vega case.

The deviation obtained in the different masses for the Proton K are shown in the table 4. All the propellant masses calculated were smaller than the original Launcher, the biggest difference being in the last stage and the smallest the first stage. The differences are explained for the non existence of the propellant margin. The structural mass calculated were always larger than the real Launcher: the biggest difference was in the last stage and the smallest was in the second stage. The GLOW had a difference of 6.2%. For a tool of preliminary design the differences are relatively small and the mass model was considered validated for this case.

-	m_p	m_s
1 Stage	-5.7	19.5
2 Stage	-8.5	5.2
3 Stage	-26.7	26.2

Table 4: % Deviation of the structural and propellant masses - Proton K.

The following figures show different variables of the trajectory of Vega for the last simulation. In the figure 6 is possible to observe the evolution of altitude with time and when the GT ends and then TPBVP starts. In the figure 7 is possible to observe the Velocity function of time and observe how the velocity varies in GT and TPBVP and the change that happens after the end of Gravity turn. The

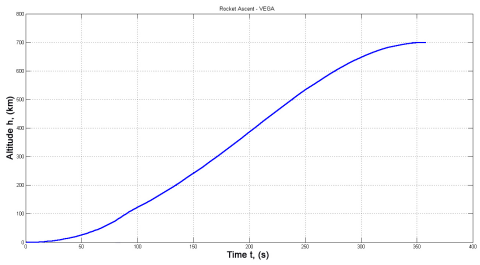

Figure 6: Altitude vs Time - Vega

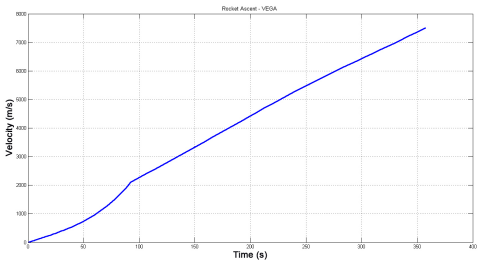

Figure 7: Velocity vs Time - Vega

following figures show different variables of the trajectory of the last simulation of Proton K. In figure 8 are represented the Altitude of the trajectory vs time and the altitude vs downrange. It's possible to observe that the payload reaches the desired orbit and time. The figure 9 represents velocity of

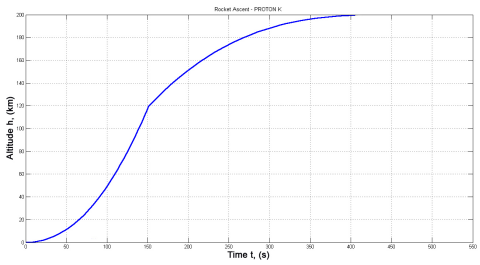

Figure 8: Altitude vs Time - Proton K

Launcher vs time, where it's also possible to observe the end of GT and the start of TPBVP and how the launcher rapidly ascents to the desired orbit.

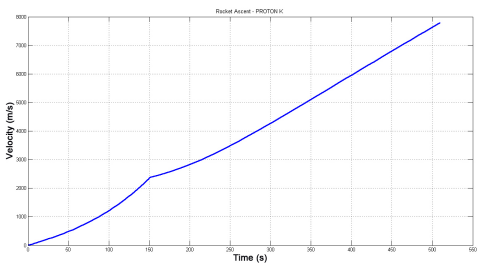

Figure 9: Velocity vs Time - Proton K

6.2. Preliminary design and optimization

In the last case, a set of simulations was done with the objectives of Ariane 5 for LEO, an altitude of 200 km and a payload mass of 19.3 tons, but in this set of simulations some parameters were varied with the goal of obtaining an optimal configuration. This means the minimum GLOW that can satisfy the desired objectives.

Three groups of different simulations were run. In the first group, the diameter and Thrust of each stage were fixed but on the other hand the number of boosters and its characteristics, Thrust, diameter and burn time were varied in a certain range. In parallel with the first group simulation, the second group simulation was run with all the parameters fixed except the diameter of core stages. The third group simulation used the best result of the first group simulation for the boosters (its thrust and dimensions) and the best diameter obtained in the second group simulation with the variation of ΔV division and respective thrust for each stage. In the end, an optimal solution was found and its configuration was the one with minimum GLOW that achieves the objectives early proposed. A total of 170 simulations were run.

The objective was to optimize an existing launcher and not to validate the code, the objective wasn't to see if the results match but how different will be the new optimized launcher compared with the original Ariane 5, how lighter will be, and how the volume will vary. The values achieved for the best configuration are presented in the table 5.

Parameters	Final Value
Diameter [m]	3.9
Thrust [kN]	1700 and 58.5
$\Delta V division$ [%]	[25% 49% 26%]
Number of boosters	2
Thrust booster [kN]	8000
Diameter booster [m]	2.625
Burn time Booster [%]	25%

Table 5: Optimal Configuration Achieved.

The deviation obtained in the different masses for the last simulation are shown in the table 6.

-	m_p	m_s
Booster	-4.5	-3.5
1 Stage	-27.2	-2
2 Stage	-85	-5

Table 6: % Deviation of the structural and propellant masses.

In this optimization all the propellant masses cal-

culated were inferior than the ones from the original Launcher. Comparing this differences, with those obtained for the validations of the model it's possible to state that optimization of the parameters decrease the propellant necessary to achieve the orbit. The structural masses calculated were slightly inferior than the ones from the real Launcher, these difference can be explained by the reduction of the mass of the propellant and its direct influence in decreasing the mass of the propellant tanks. The goal of this optimization was the reduction of GLOW, the final simulation had a difference of 11%, which allows to reduce 84 tons the weight of the Launcher at lift-off and compared with the other simulations, where the objective was to validate the tool, it possible to say that the original Ariane 5 was optimized and consequently its GLOW reduced.

The dimensions obtained were inferior compared with Ariane 5 original because the propellant masses of each stage were inferior due to optimization. The differences are higher in upper stage because that was the stage that had more propellant reduction. The overall length was 3.1 meters higher than original Ariane 5, increasing 5.5% the total length of the Launcher.

The following figures show the evolution of different variables of the final trajectory simulation. In the figure 10 is represented the evolution of the altitude of the trajectory vs time where its possible to observe the end of GT and the start of TPBVP. The figure 11 represents the velocity of launcher vs

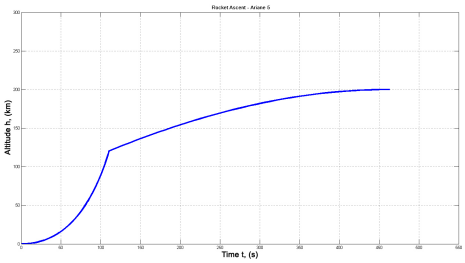

Figure 10: Altitude vs Time - Ariane 5

time, where it's possible to observe the end of GT and the start of free-flight phase when the velocity initially raises slowly and then accelerates until the desired orbit.

7. Conclusions

The main goal of this research was to develop a tool to assist the preliminary design of rockets. The tool was modelled using a MATLAB code, which applies a simplified optimization, and covers two major fields: the mass model and the trajectory.

The database created during this study allowed the understanding of some important variables, such as length, diameter, thrust, burning time, specific masses and others, providing their range of val-

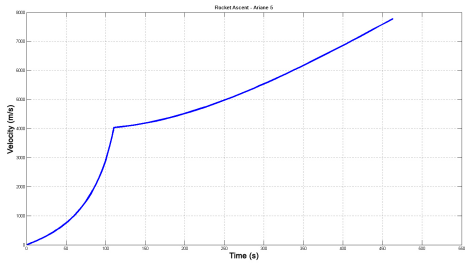

Figure 11: Velocity vs Time - Ariane 5

ues and ensuring the viability of the final result. Some of these values were used in the mass model, as initial structural factors, for the iteration of the mass model.

The mass model created, showed that the use of Tsiolkovsky's equation with the MERs heuristics, provides realistic results for the structural masses of the launcher. This model also calculates the dimensions of the launcher, with the goal of adding more realism to the final design. When the mass model is concluded, all the masses and dimensions needed for the trajectory of the launcher have been defined. The trajectory was divided into three distinct phases. The vertical ascent, the gravity turn and at last the free flight phase. A simplified drag model was used, where the C_d only varies with Mach number. The same happens for gravity model where the gravity only varies with altitude.

The Model validation is important to determine the accuracy of the results obtained. In this study, the model validation was done regarding two examples, Vega and Proton K, targeting diverse objectives (Orbit and Payload Mass) and characterised by different configurations. Both validations were done with the values presented in the user's manual. First, an isolated simulation of the trajectory was done, i.e. using the mass and dimension values from the user's manual. Both trajectory simulations were successful.

After the validation of the trajectory, it was possible to validate the mass model. All the parameters selected were the same of the respective User's manual. The results showed that almost all the propellant masses were inferior, when compared to real launchers, explained by the fact that the mass model only adds propellant margin for the last stage, instead of adding it for all stages. The structural masses were almost all superior, when compared with real launchers, meaning the mass model is over estimating the structural weight for the selected launchers. The GLOW of each launcher was inferior, and that is explained by the reasons presented for the deviation regarding the propellant masses.

After the validation of the developed tool, this

was used for the preliminary design of a rocket. The original parameters used were based on launcher Ariane 5, and some were modified, to optimize the rocket i.e. increase the payload ratio by reducing the GLOW. This allowed the understanding of which parameters influence more the difficult task of designing a rocket. The objectives of the mission were the same as those defined for Ariane 5 and in the end of the optimization all masses and dimensions were compared with the user's manual. All the masses were smaller; the propellant masses differences can be explained by the non-existence of propellant margin and also by the reduction of the diameter, which allowed the free-flight phase to start earlier; the structural masses were also inferior because of the reduction of all propellants masses, the lighter tanks and the reduction of the diameter, which decreased the weight of the structure and allowed the free-flight phase of the trajectory to start earlier, optimizing the fuel consumption. In conclusion, the reduction of GLOW was significant, meaning that the optimization was successful.

The ΔV division and thrust are the parameters that most influence the GLOW. The definition of diameter effects drag and trajectory, because of the definition of the Knudsen number and must be carefully selected, due to the fact that in this study a structural analysis wasn't developed.

7.1. Future work

It would be interesting to include the chamber and exit pressure and the Nozzle areas in the Mass model, currently its effect is neglected on the nozzle mass. Also in mass model development of a heuristic by a student, creating a database for different components of launchers (Tanks, Engines Nozzles, Interstage adapters, Avionics, Wiring and others).

Implement constrains in the Trajectory model, such as Max. Dynamic Pressure, Heat Flux, Bending Load and Axial Acceleration. Improve the Drag model, a more realistic Drag model can be achieved with analytical equations for each Nose Cone Configuration.

Study air-launched rockets (Pegasus), transfer orbits to GEO and interplanetary trajectories that will include long coast phases and improve the trajectory. Develop a Cost Model, in order to obtain a realistic insight of Launch Vehicle Design.

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