Rocket Heuristic for Preliminary Design

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Abstract

Rockets design is a iterative process in which the main objective is to reduce the gross lift-off weight (GLOW) and increase the payload ratio making it more efficient. In iterative processes a good first approximation can reduce the efforts in design phase by a significant margin. In this work, a rocket database was constructed with the objective to identify heuristics that can be used in preliminary design of rockets. The main rocket characteristics were compiled, analysed and compared with known heuristics, in order to confirm them, and acquire new ones.
Keywords: Heuristic, Space Launch Vehicles,Rocket Database, Survey

1. Introduction

Space Launch Vehicle (SLV) design is a complex endeavour involving multiple subsystems that all influence each other. A survey of both past and present space launchers was made to confirm known heuristics or find new to be used in preliminary design for launchers. The launch of a spacecraft is fundamental to all space activity, and it is through the development of efficient launch vehicles that the impact of space on many aspects of science, commerce and daily life is possible [11]. Payloads and missions for spacecraft are many and varied. Some have reached the stage of being economically viable, such as satellites for communications, weather and navigation purposes. Other satellites serve the scientific community or are used for military scenarios [6].

The growing interest in launching small satellites into LEO, including constellations of such satellites, whether for scientific or communications purposes, has prompted a renewed interest in small launchers offering flexible operation and making limited infrastructure demands. A number of such vehicles are already available commercially or in development [6].

2. Rocket Dynamics

In this chapter we briefly review the most important aspects of rockets to determine what parameters should be included in the database.

2.1. Tsiolkovsky's equation

The Tsiolkovsky's equation, also known as the rocket equation, allows to calculate the velocity increase ΔV of a rocket through to propellant consumption and the velocity of the exhaust gases (ref)

$$\Delta V = V_e \ln |\frac{m_0}{m}|, \tag{1}$$

where V_e is the exhaust velocity, m_0 the initial mass and m the mass at the point considered. It is only valid for a constant exhaust velocity and with no external forces considered. It can be used as a first approximation in many cases since the main exterior forces, drag and gravity, are relatively small.

2.2. delta-v calculation

The ΔV required to fulfil a mission is

$$\Delta V_{\text{Design}} = \Delta V_{\text{orbit}} + \Delta V_{\text{gravity}} + \Delta V_{\text{drag}}, \tag{2}$$

where ΔV_{orbit} is the velocity that will be required for the desired orbit, while $\Delta V_{gravity}$ and ΔV_{drag} are losses that will occur and that the rocket will have to overdue. Other losses exist, however the two mentioned are the most significant and the only worth considering in a first approximation. The ΔV provided to the launcher is usually higher than the ΔV_{Design} calculated, in order to provide some safety margin.

For a satellite to maintain a circular orbit, it must have the velocity

$$V_{orbit} = \sqrt{\frac{\mu}{R}}, \tag{3}$$

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

Gravity loss is determined by

$$\Delta V_{gravity} = \int g \sin \gamma dt, \tag{4}$$

with g being the gravitational acceleration and γ the flight path angle. Analysing the equation 4 it is possible to see that one way to reduce the gravity loss is to keep the flight path angle zero. This can't be done for the entire flight but it is possible use a trajectory with small γ to gain velocity early in the launch.

The drag losses are [6]

$$\Delta V_{drag} = \int \frac{D}{m} dt, \tag{5}$$

where D is the drag force and m mass of rocket at each time. The drag force present depends on the configuration(size and shape) of the launcher as well as it's speed and angle of attack. The drag force is dependent on the area of the launcher in the direction of the movement, therefore it will increase with the diameter of the launcher. To minimize this losses, the rocket as to rise as slow and vertical as possible. When the rocket reaches a certain height the drag losses became negligible, since the atmospheric density decays exponentially. [6].

2.3. Configuration

The simplest configuration is a single stage rocket (SSTO). However rockets with a single stage can hardly reach the required orbital speed unless the payload is very small [13]. During the powered flight of a single-stage rocket, part of its mass has become 'superfluous', because a significant part of the mass of a rocket is propellant, that requires a lot of structural mass to be carried that becomes empty. So an extra, in fact useless mass has to be accelerated by the rocket engine when it is no longer needed. It is advantageous to discard the useless mass during the flight, using multiple stages. An additional advantage of multiple stages is the possibility of thrust programming, as well as adapting the engines of subsequent stages to the altitude where they are fired, thus reducing losses due to non-ideal expansion [1].

2.3.1 Multistage rockets

Since the performance of a rocket depends significantly on the structural mass of the vehicle, performance can be improved if some way can be found to dispose of useless structural mass whenever possible. The most common method for doing this is to stage the vehicle. Empty tanks and the large engines necessary to lift off from the Earth's surface are shed, and the smaller vehicle proceeds from that point with considerably less parasitic mass [12].

In a multistage rocket the payload of any given stage is considered all the mass above it. This means that the payload of the first stage is all the other stages plus the actual payload meant to be put in orbit, i.e

$$\lambda_N = \frac{m_{0N+1}}{m_{0N}}, \tag{6}$$

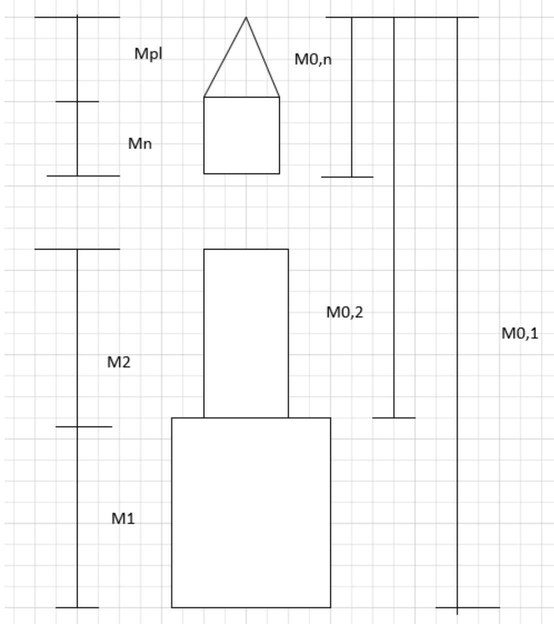

Figure 1: Multistage Rocket Configuration.

where m_{0N} is the total mass and m_{0N+1} is all the mass above that stage. Hence the payload is going to decrease with each stage, typically requiring less thrust for optimum results.

Some parameters about rocket configuration and staging are very important for rocket design

Structural ratio is a measure of the amount of the vehicle that is structure. Usually it is considered for each individually stage

$$\epsilon_N = \frac{m_{sN}}{m_{sN} + m_{pN}}, \tag{7}$$

where m_{sN} and m_{pN} are the masses of the structure and of the propellant for a given stage N

Similarly, the propellant mass ratio gives the relative amount of propellant

$$\varphi_N = \frac{m_{pN}}{m_{0N}} = (1 - \epsilon_N)(1 - \lambda_N), \tag{8}$$

where m_{pN} and m_{0N} are the propellant mass and the total mass of the stage N, respectively

2.3.2 Boosters

Boosters can be added to improve the performance of a stage. This is basically a parallel staging making two stages active at the same time. Usually the boosters have a shorter burn time.

In this cases a zeroth stage, that combines both the boosters and the first stage, is considered. The first stage will be the remaining part of it, once the boosters are released.

The main advantage of parallel staging is the reduction of gravitational losses and an increase in thrust necessary to take of the ground. A disadvantage is that the rocket is likely to be bulky, and for flight through the atmosphere the drag penalty may be large. This, however, is of minor importance for large rockets where the drag losses are very small as compared to gravitational losses. A second disadvantage of parallel staging can the reduction in nozzle efficiency of the engines of the first stage. If these engines are used from the start, their expansion ratio is limited by the atmospheric pressure at low altitudes [1]. In this case, the structural and payload ratios are calculated through:

$$\epsilon_0 = \frac{m_{s0} + m_{s1}}{m_{s0} + m_{s1} + m_{p0} + m_{p1}}, \tag{9}$$

$$\lambda_0 = \frac{m_{01} + m_{ip1}}{m_{00}}, \tag{10}$$

where m_{s0} and m_{p0} are the structural, propellant ratio of the stage respectively and m_{ip1} the remaining propellant of the first stage at the zeroth stage burnout.

After the boosters are released, the first and remaining stages, work as if no parallel staging had happening [12].

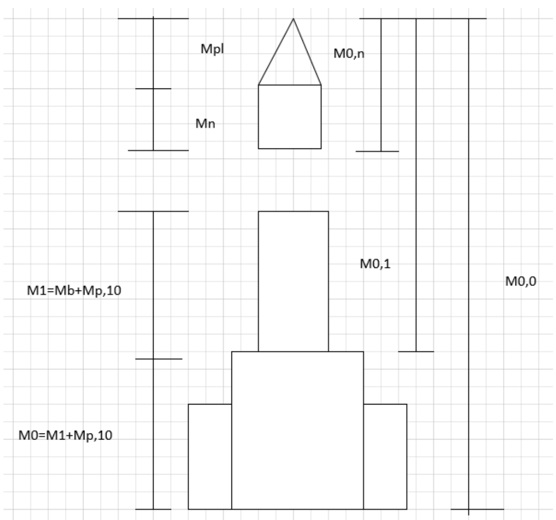

Figure 2: Parellel Staging Configuration.

2.4. Propulsion

Propulsion in a broad sense is the act of changing the motion of a body. Propulsion mechanisms provide a force that moves bodies that are initially at rest, changes a velocity, or overcomes retarding forces. As a result of space being a vacuum and with no standing places, Space Propulsion System (SPS) need to have unique characteristics in order to give an impulse to a vehicle [4, 9].

The energy source most useful to rocket propulsion is chemical combustion. [9]. A rocket is generally (traditionally) defined as a propulsion system that carries both fuel and oxidizer as storage within the vehicle, burning the propellant as required to produce a high-speed exhaust jet that delivers the needed thrust [2]. Chemical combustion systems are the most common systems for space applications and can be divided into three basic categories: solid, liquid and hybrid. The terminology refers to the physical state of the stored propellants [3].

The standard one-dimensional thrust equation, for thrust generated by a rocket's propulsive exhaust nozzle, is given by

$$T = \dot{m}_e u_e + (p_e - p_a) A_e, \tag{11}$$

where the first term is the momentum thrust represented by the product of the propellant mass flow rate and its exhaust velocity relative to the vehicle. The second term represents the pressure thrust consisting of the product of the cross-sectional area at the nozzle exit A_e (where the exhaust jet leaves the vehicle) and the difference between the exhaust gas pressure at the exit and the ambient fluid pressure. If the exhaust pressure is less than the surrounding fluid pressure, the pressure thrust is negative. Because from this condition results a low thrust and is undesirable, the rocket nozzle is usually designed in a way that the exhaust pressure is equal or slightly higher than the ambient fluid pressure [9].

Other key parameter for propulsion is the specific impulse. It's defined the change in linear momentum per unit weight of the propellant consumed

$$I_{sp} = \frac{V_e}{g_0}, \tag{12}$$

where g_0 is the acceleration due to gravity at standard sea level of the Earth, and V_e is the velocity of the exhaust gases [10].

Having clear the atmosphere, the gravity turn manoeuvrer id no longer required. It is usually at this point that the fairing can be jettison since there are no longer atmospheric forces being applied.

3. Space Launcher Database

In order to compare different SLV and to understand the trends they follow a database was constructed gathering some key characteristics of rockets. We also calculate or estimate some parameters when they were not available at the source using the ones available, if possible. The database was constructed with information gathered from rocket launcher user guides, books on the subject and, as a last resort, information on the internet. Rockets launched horizontally weren't considered since they present some considerable differences in their trajectory and design philosophies. Reusable rockets were also not included, due to small data sample.

We also gathered from literature some heuristics about rockets that can be used as a guide for preliminary design. The database built can be used to confirm or disprove the existing heuristics, as well as to identify new ones.

3.1. Launch Vehicles
To build the database, a survey of both existing and retired rockets was conducted. Some launch vehicles expected to perform their first launch in the near future, for which information was already available, were also included.

Information about a total of 43 launch vehicle was collected. They were separated into 3 different categories according to their payload capability to a certain orbit.

Small rockets are able to deliver up to 2 tons to Low Earth Orbit (LEO) [13]. A total of 11 small rockets are present in the database.

Medium launchers serve to place satellites into all Earth orbits: LEO, including polar orbits, mMdium Earth Orbits (MEOs), Geostationary Transfer Orbit (GTO), Geostationary Orbit (GEO) and Earth escape missions. They can deliver between 2 and 15 tons at LEO and 3 to 6 tons at GEO [13]. There are 20 medium rockets present in the database.

Heavy-lift launch vehicles (HLLVs) mainly launch communications satellites into GTOs and are used specifically for launching very heavy payloads. They can deliver more than 15 tons at LEO and 6 tons at GEO [13]. A total of 11 heavy rockets are present in the database.

3.2. Parameters Gathered
The key characteristics gathered for each rocket can be found in Table 1. The information was collected for each individual stage. Some characteristics were available for some rockets, such as interstage sections and payload adapters, that were also gathered but for which the sample size isn't enough in order to look for regularities.

With the information gathered some derived relevant characteristics where calculated (see Table 2).

3.2.1 Trajectory

Additionally some aspects of the trajectory were also included in the database when available, such as the altitude, velocity and time of flight(ToF) of some relevant events of the ascent.

3.3. Heuristics Gathered from Literature
The collected information is a historical recoil of the most rockets and can be used to confirm or disprove a a posteriori known heuristics about rockets. For that purpose, some heuristics were compiled from literature [3, 13, 9, 6]. For easy reference a code was attributed to each one.

- SR1 Structural Mass Ratio 14% total mass[3]
- SR2 Structural Mass Ratio 5%-15% total mass[13]
- PR1 Propallant Mass Ratio 85% total mass[3]
- PR2 Propallant Mass Ratio 91% total mass[9]
- PRL1 Payload Mass Ratio 1% of total mass[3]
- PR2 Payload Mass Ratio 1%-5% total mass[13]
- PR3 Payload Mass Ratio larger vehicles are able to archive higher payload fractions[6]
- IV I_{sp} Usually higher for higher stages[3]
- ST Structure Lower stages are longer and often have a larger diameter than upper stages [13]

Characteristic	Units
Mass Propellant	kg
Mass Structure	kg
Total Mass	kg
Total mass at lift-off	kg
Length	m
Diameter	m
Engine	-
Engine number	-
Thrust	kN
Isp	s
Propellants	-
Burn time	s
Payload Capability	kg
First Flight date	-

Table 1: Characteristics gathered for each stage

Characteristic	Units
Class	-
Mass Propellant	kg
Mass Structure	kg
Total Mass	kg
Mass stage and above	kg
Weight	N
Total thrust	kN
T/W	-
Structural Ratio	-
Mass Stage Ratio	-
Propellant Ratio	-
Payload Ratio	-
Total Length	m
Length/Diameter	-
Length Stage Ratio	-
Exit gasses Velocity	m/s
Ideal Velocity Increment	km/s

Table 2: Characteristics derived for each stage

Trajectory Events
Maximum Dynamic Pressure
Fairing Jettison
Stage Engine Cut-off
Stage Engine Ignition
Stage Separation
Spacecraft Separation

Table 3: Trajectory relevant units for which ToF, altitude and velocity were gathered

- TW1 Thrust over Weight First stage might typically have an T/W ratio less then 1,5 [6]
- TW2 Thrust over Weight T/W has to be higher than 1. $10\% - 20\%$ higher take off thrust helps the launcher [13]
- BT1-Burn Time - Booster engines operate in a duration of 1-3 minutes[13]
- BT2-Burn Time - Main engine operate in a duration of 400-500s[13]

4. Data Analysis

In this chapter the database is used to analyse the different characteristics of the launchers and try to find regularities. The heuristics gathered from literature are compared with the data from the launchers and new ones are obtained.

4.1. Analysis Criteria

In order to analyse the data acquired, a set of criteria must be placed in order to keep the process consistent.

When describing numerical data, it is common to report a value that is representative of the observations. Such a number describes roughly where the data are located or "centered" along the number line, and it is called a measure of center. The two most popular measures of center are the mean and the median [5]. It was chosen the median value of the population to estimate a launcher characteristic. The median of a population is the middle value of data when rearranged in crescent order, and divides it into two equal parts. This way it is the best single number approximation because it isn't as sensitive to outliers as the mean [5].

However a point estimation of a population characteristic will depend on the sample. Instead a confidence interval for the median is calculated. It is constructed so that, with a chosen degree of confidence, the value of the characteristic will be captured between the lower and upper endpoints of the interval. A confidence level has to be established before examining the data. A 95% confidence level was chosen since it is the most commonly used (ref).

The sample of launchers obtained is not very large and that is further evidenced when divided into classes. Therefore it is assumed that a t distribution is in place, and the confidence interval can be calculated using

$$x \pm t\frac{s}{\sqrt{n}}, \tag{13}$$

where x is the median, t is the confidence coefficient that is dependent on the confidence level and the number of the sample, s is the standard deviation and n is the number of the sample [8].

The mode of the distribution is not presented in this study. It was thought that studying the data mode would be interesting, however the only significant observation resulting was when launchers from the same family where present. SLV from the same family often use the same design for a single stage. For example, the first and second stages from Titan 401B and Titan 403B are the same, with Titan 401B having an extra stage, or Proton-k and Proton M in which the first stage of Proton M is an improvement over Proton-k, while the rest of the stages remain the same. Meaning that the results obtained from mode are not for characteristics that happen to be the same but from the same design.

Outliers are expected in every sample. In order to identify them it was used Tukey's method, because it can be applicable to skewed or non mound-shaped data since it makes no distributional assumptions and it does not depend on a mean or standard deviation because it uses quartiles which are resistant to extreme values [7]. Also this method fits in the display of results used since it allows the verification with boxplots which is the chosen method to visualize the results.

To find outliers it is necessary to calculate the Inter Quartile Range (IQR), which is the distance between the lower quartile (Q1) and upper quartile (Q3). If a point is at a distance of 1,5 times the IQR below Q1 and above Q3 it is considered a possible outlier and for a distance of 3 times the IQR it is considered a probable outlier [7].

4.2. Data Overview

4.2.1 Structural Ratio

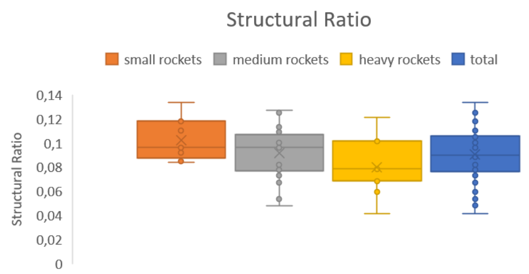

Figure 3: Structural Ratio.

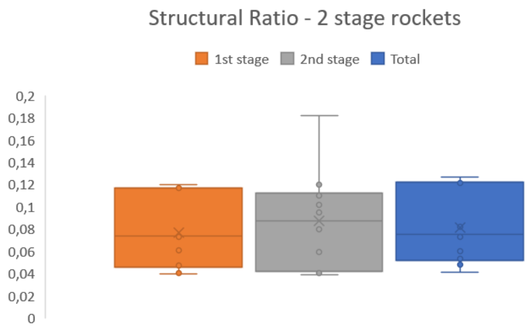

Figure 4: Structural Ratio of 2 stage rockets.

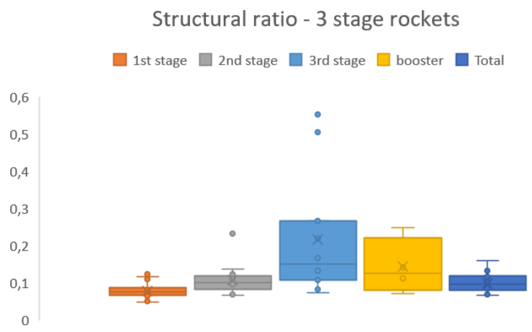

Figure 5: Structural Ratio of 3 stage rockets.

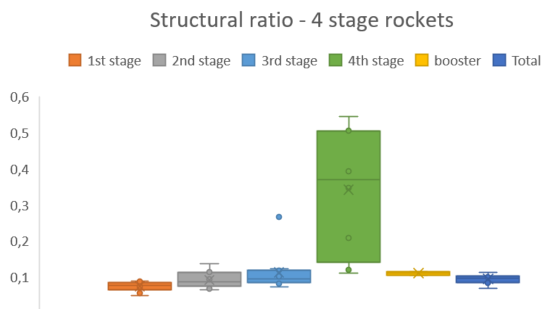

Figure 6: Structural Ratio of 4 stage rockets.

Figures 3, 4, 5 and 6 shows how the structural ratio fluctuates with the class of the launcher and the number of stages it as. Considering the entire population and comparing with **SR2** it is possible to observe that the range it provides is wider than the range between the 5th and 95th percentile, meaning that involves more than 90% of all values gathered. We can use this heuristics as a benchmark of values, however since it includes the majority of values gathered this interval can still be refined in order to have a narrower set of values for initial design. The estimation interval value for structural mass is 0,086 to 0,101 with a median of 0,094 which can be used as a good first value for preliminary design. According to **SR1** this value should be 0,14. However this value is outside the 95th percentile of the data and therefore it is not verified by it.

Small and medium launchers present a similar structural ratio even having the same median value of 0,096. On heavy launchers this value slightly decrease to 0,081 meaning it could decrease inversely to the gross mass of the SLV.

Upper stages present a wider range of values, which can be explained by their smaller size and lesser requirement of propellant due to operate without drag.

Using the criteria defined we can calculate both the first value for iteration and the range of values expected.

4.2.2 Propellant Ratio

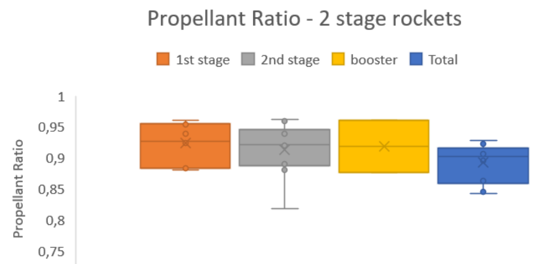

Figure 7: Propellant Ratio of 2 stage rockets.

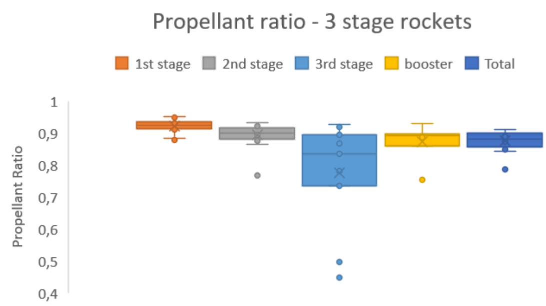

Figure 8: Propellant Ratio of 3 stage rockets.

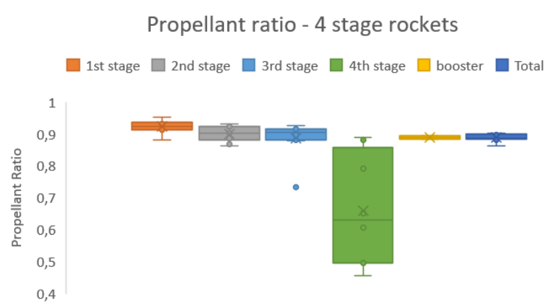

Figure 9: Propellant Ratio of 4 stage rockets.

From Figures 7, 8 and 9 it is possible to observe that propellants provide the greatest contribution to the launcher total mass. Upper stages have a wider range of values. For the totality of the SLV, propellant ratio does not vary significantly with the class of the launcher. The median value of 0,888 is a good approximation for a first iterative value to be used, with the estimate interval being 0,881 to 0,896.

4.2.3 Payload Mass Fraction

The maximum payload of a launch vehicle will depend on the target orbit. In the database some launchers present more than one option for the payload capabilities corresponding to different possible orbits. For this calculations it was considered the data for highest payload value.

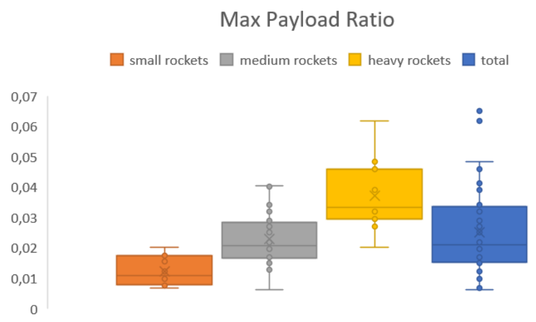

Figure 10: Max Payload Ratio.

Comparing Figure 10 with heuristics gathered in section3.3, the payload ratio from all types of rockets fit into **PLR2**(1%-5% of total mass). This range includes 90% of the values gathered making it representative of most possible values.

Looking at **PRL1** (1% of total mass), it doesn't apply to the generality of the launchers, however if we only take the small rockets into account this value is within the estimate interval for that category. The payload ratio actually has different levels for each category, making it possible to estimate heuristics for each one. Small launchers mass ratio interval is 0,8% to 1,4%, which includes **PRL1**. Medium launchers 1,7% to 2,4% and heavy launchers 3,1% to 4,7%.

This different levels for each category allow to verify **PLR3** which stated that heavy launchers have a higher payload ratio. The propellant ratio being constant for all classes means that the payload ratio and the structural ratio have an inverse relation.

4.2.4 Isp

From Figures 11, 12 and 13 it is possible to observe that the values for I_{sp} are lower for the first stage and boosters verifying **IV** (usually higher values for higher stages). Lower stages main concern

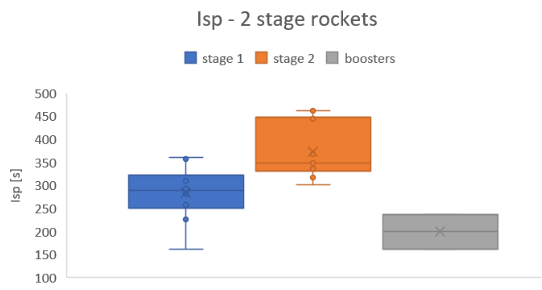

Figure 11: Isp of 2 stage rockets.

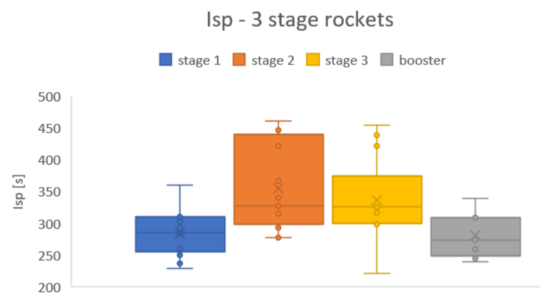

Figure 12: Isp of 3 stage rockets.

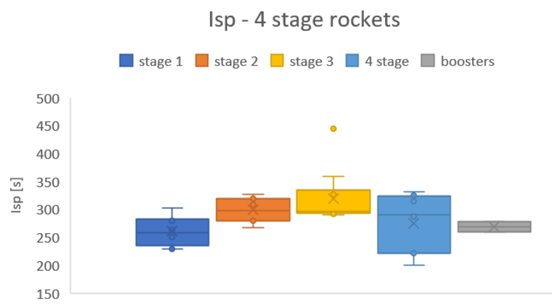

Figure 13: Isp of 4 stage rockets.

is to have enough trust to be able to surpass the weight of the launcher. In higher stages the focus is on fuel efficiency, to which a high I_{sp} value translates. Engine manufactures often present values for its operational value in vacuum and at sea level, with I_{sp} for vacuum always higher. Since upper stage engines operate in rarefied atmosphere its I_{sp} can be higher. For both boosters and first stage the sea level value are taken into account.

The differences in I_{sp} are minimal since the propulsion technology used for every engine in this database is chemical. This means that the main contributing factor is the propellants used.

The 1st stage and boosters have a narrower set of values for I_{sp} since the altitude in which they operate is always the same, at sea level. Depending on trajectory other stages can ignited at different points, which allows for optimization according to their operation altitude. Considering the totality of launchers this can be verified with the increase of standard deviation, which translates to values being more scattered, even doubling in some cases.

Due to the possibility of optimization for the altitude the engine will operate, obtaining heuristics values for I_{sp} is not a productive task. However for sea level, where the 1st stage and boosters start, we can use the criteria defined to estimate a interval of 272 to 293 seconds, with a median value of 283 for 1st stages.

4.2.5 structure

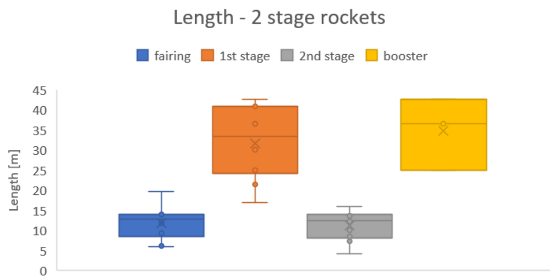

Figure 14: Length of 2 stage rockets.

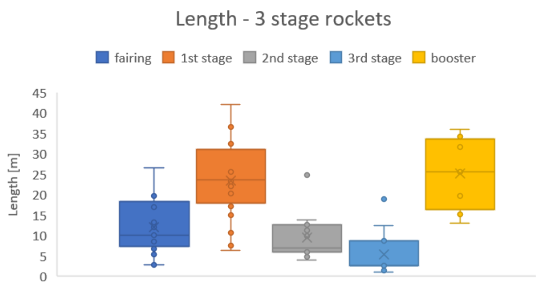

Figure 15: Length of 3 stage rockets.

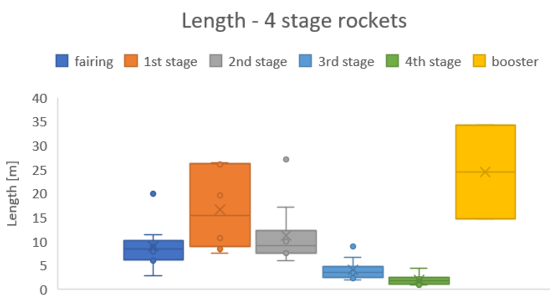

Figure 16: Length of 4 stage rockets.

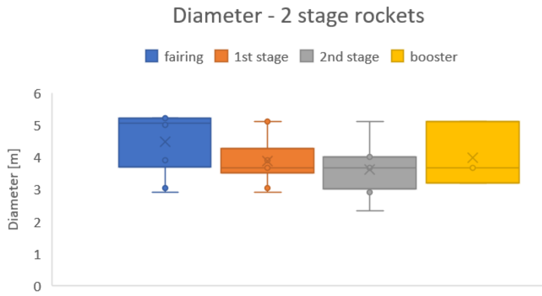

Figure 17: Diameter of 2 stage rockets.

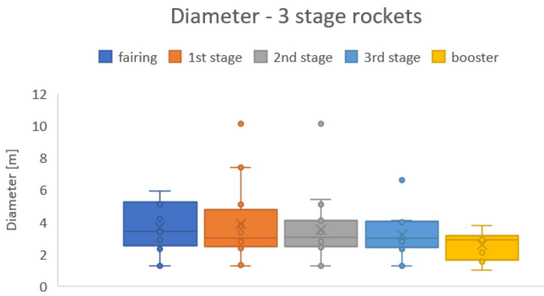

Figure 18: Diameter of 3 stage rockets.

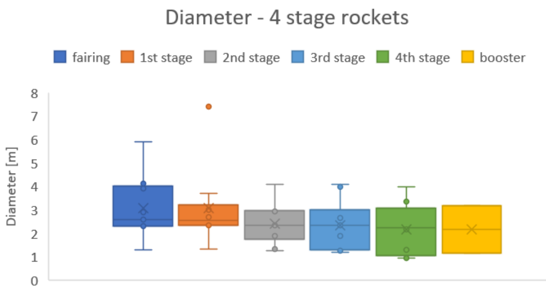

Figure 19: Diameter of 4 stage rockets.

Figures 14, 15, 16, 17, 18 and 19 display the size (length and diameter) of different stages and how the size of a SLV can change by having a different number of stages. With this information is possible to verify the heuristic **ST**, as it stated that lower stages are longer and often have a larger diameter than upper stages, which matches the information gathered. Both first and second stages have the same median value of 3,04m with this value lowering to 2,5m and 2,18m for the third and fourth stages. Regarding the length this is even more noticeable with the median value for the first stage being 23,5m and the second stage median having a value that is less than half at 9,2m. It continues lower with higher stages with values of 2,8m and 2,04m.

It is possible to see a clear outlier on 3 stage launchers corresponding to Saturn V. Other curious outlier the the fairing on 4 stage vehicles. The launcher is Titan 401B which performed the Cassini-Huygens mission delivering a payload of 5712 kg to Saturn's largest moon Titan.

4.2.6 T/W

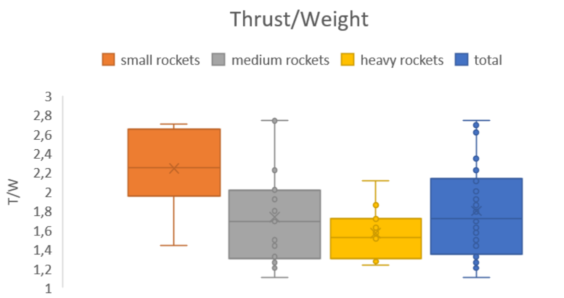

Figure 20: T/W.

We can use the values on figure 20 to study the heuristics gathered. **TW2** says that thrust has to be, at least, equal to the mass and that a 10% to 20% higher take off thrust helps to maintain a good trajectory. The first part is verified since the minimum value for T/W is 1,092 from LVM3. In fact this first part can be considered more of a requirement than an heuristic because if the weight is superior to the thrust the SLV wouldn't be able to take off. Even so, after take off and for upper stages it is possible to see that multiple rockets have a T/W inferior to 1.

The 10%-20% margin would correspond to a T/W value of 1,1 to 1,2. Using the entire sample the estimate value for T/W is from 1,6 to 1,8 which is higher than the expected from **TW2**. The 20% margin matches with the 5th percentile, meaning that 95% of the launchers in the database use values superior the it.

Looking at the numbers for each individual stage we can find some of the minimum values unexpectedly low, even being lower than 0,1 in some upper stages. This cases, which correspond to Athena 2c, Proton M/Briz-M, and both Angara A3 and A5, can be explained by the fact that their are the last stage with high burn times. These stages will operate at altitudes in which the atmospheric drag is no longer a concern so the required thrust to change the trajectory is greatly reduced, choosing to perform a longer burn with reduced thrust.

4.2.7 Burn Time

Figure 21 shows the burn time for the first stage and boosters. On launchers without boosters we can observe a burn time for the first stage similar to the one presented by boosters, while the 1st stage of launchers with boosters continue to burn even after the boosters are released.

The maximum burn time for boosters is 325,2 corresponding to Angara A3 and Angara A5. According to the criteria defined this values are outliers. If we remove them, the maximum value is 162

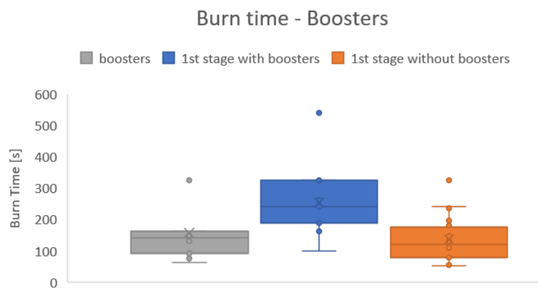

Figure 21: Burn Time.

seconds, which is around half the time of those two. Comparing with **BT1**, which stated that boosters operate in a duration of 1 to 3 minutes, 100% of the values are within the range. As a benchmark of values this heuristic can be useful, however as a design value it provides a range of values that is too wide to be effective. Removing the outliers and using the criteria defined, it is possible to obtain a narrower range that goes from 108 to 151 seconds.

The main engine should operate in a duration of 400 to 500 seconds according to **BT2**. The only launcher that is within the range of values is Arianne 5, which is a outlier from the rest of the sample. If we remove it there are no launchers that verify **BR2**, with the maximum value being 325,2 for the first stage of Angara family launchers.

5. Conclusions

Each launcher vehicle is designed for a different set of objectives, according to payload and orbit capabilities. They are complex machines that require an iterative process to be designed. In this work, we gather rocket historical information and used it to establish heuristics, and confirm or disprove existing ones, to be used as guidelines for preliminary rocket design.

New heuristics were established and existing ones were confirmed or disproved and thus some empirical relations were established.

Heuristic Verification:

- SR1 - Not Verified
- SR2 - Verified
- PR1 - Not Verified
- PR2 - Not Verified
- PLR1 - Partially Verified
- PLR2 - Verified
- PLR3 - Verified
- IV - Verified
- ST - Verified
- TW1 - Partially Verified
- TW2 - Partially Verified
- BT1 - Verified
- BT2 - Not Verified

- NSR1 Range of structural ratio 8,6% - 10,1%
- NSR2 Upper stages have a wide range of results going up to 50% in some cases
- NSR3 Structural Ratio is slightly lower in heavy launchers with a estimate interval of 6,3%-9,9%
- NSR4 Small launcher structural ratio tends to evolve with the gross mass with a logarithmic relation
- NPR1 Range of Propellant Ratio 88% - 89,5%
- NPR2 Class of the launcher has no influence on the Propellant Ratio
- NPR3 The propellant mass has a clear linear trend line
- NPRL1 Payload Mass Ratio small SLV 0,8%-1,4%
- NPLR2 Payload Mass Ratio medium SLV 1,7%-2,4%
- NPLR3 Payload Mass Ratio heavy SLV 3,1%-4,7%
- NIV - I_{sp} for the 1st stage goes from 272s-293s
- FL1 - Fairing length small SLV 4,8m-7,9m
- FL2 - Fairing length medium SLV 7,8m-11,3m
- FL3 - Fairing length heavy SLV 13,5m-20,5m
- FD1 - Fairing diameter small SLV 2m-2,6m
- FD2 -Fairing diameter medium SLV 3,5m-4,3m
- FD3 - Fairing diameter heavy SLV 4,9m-5,4m
- NTW1 - T/W for small SLV 2,004-2,503
- NTW2 - T/W for medium SLV 1,291-1,643
- NTW3 - T/W for heavy SLV 1,235-1,643
- NBT1 - 1st stage burn time for small SLV is 53-82 seconds
- NBT2 - Burn time boosters is 108-151 seconds
- NBT3 - Upper stages burn time is significantly superior to the lower stages
- DPvel - Maximum dynamic pressure at 0,5km/s-0,75km/s
- DPtime - Maximum dynamic pressure at 58-79 seconds of flight
- DPalt - Maximum dynamic pressure at 10km-12,5km altitude
- FJvel - Fairing jettison at velocities of 3,5km/s-4,6km/s
- FJtime - Fairing jettison 213-271 seconds of flight
- FJalt -Fairing jettison at 112km-142km altitude

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