Universidad Rey Juan Carlos

Aerospace Engineering in Aerospace Vehicles

Conceptual Vehicle Design of a Microsatellite Launcher

Design Project

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1 Introduction

1.1 Context and Rationale

In this proyect we are going to study a ground launch into orbit of a small launch vehicle or microsatellite in a sun-synchronous low orbit. The study of this project is very important because currently a large majority of the satellites are placed in low orbits (LEO type). This is because it is easier to inject a satellite into a low Earth orbit because less power is required and data transfers are achieved without the need to use very powerful transmitters.

The goals of this project will be to conceptually design our vehicle and develop it technologically so we use the best technologies with low costs to develop an economical and feasible option in the market.

The objectives of carrying out this work will be to get the basic knowledge on the design of small launchers and develop the ability to choose among so many options which is the most appropriate to develop our mission.

The development of small launch vehicles can provide a unique opportunity to innovate and explore new opportunities in the field of space exploration and satellite launch. The benefits include lower cost, greater flexibility, new business opportunities, and technological advances.

1.2 Background and State of Art

For the realization of our project we have taken as a reference small launching vehicles with payloads similar to ours as a starting point. Mainly we have based the main reference on the Electron which has a payload of 200kg in a LEO-SSO orbit and is a small launcher with 3 liquid stages. The first 2 stages are for LOx and kerosene fuel and the last one is a Kick Stage (inject it into orbit) of Liquid bi-propellant. In addition to the Electron we have also taken references from launchers such as the Vega, the Delta IV small, the Falcon 9, the LauncherOne...

Comparing all these launchers we define our number of stages and the type of fuel for each one. Almost all small launchers consist of 2 stages plus boosters or 3 stages, so we decided to design a 3-stage launcher where the last stage will be to put the payload into orbit.

For the choice of the type of fuel used in each stage we compare the advantages and disadvantages of solid and liquid fuel:

- Disadvantages of solid fuel:

It gives fewer specific impulses, less thrust, and needs more mass to achieve the required thrust.

It is not controllable since once it is turned on we have to consume all the fuel, so it is not recommended for stages higher than the first for low orbit missions.

They are not reusable

It generates many pollutants that are harmful to the environment, which implies a great disadvantage in terms of the environmental issue.

Higher internal pressure is generated in the tanks compared to liquid fuel

Combustion is less prolonged because there is no refrigeration

- Disadvantages of liquid fuel:

The storage is very complex because it must be in a cryogenic state

They are much more expensive

Liquid hydrogen LOx requires very large tanks because of the low density of hydrogen

- Advantages of solid fuel:

Simplicity

Low cost

Greater reliability

Easy storage

- Advantages of liquid fuel:

It is less polluting

Provides a higher specific impulse

We have control of the thrust once the engine is started

Can be refrigerated for longer burning

Later in the propulsion section, it will be specified what types of fuels we will use in each stage and the reason for our choice.

1.3 Approach of the Project

1.3.1 Scope

Our launcher must be able to take off from a specific launch site, in our case from the spaceport of Kourou, France, and must eventually inject itself into a sun-synchronous LEO orbit with a payload of 100kg. To do this, we must correctly carry out the launch maneuver taking into account the precise coordinates and parking maneuvers. We are going to base our launcher on current European competitions.

Another thing to take into account would be to define the type of propulsion that we will use in our engines and define the structural and aerodynamic characteristics of our launcher.

Finally, we must take into account the current market situation in everything related to our project.

The scope of the practice will be to calculate: the sizing, the geometry of the vehicle, determine the aerodynamic coefficients, the trajectories, the mass distributions, the assess vehicle loads, the structural design, the assess vehicle control, the operational modes, the costs and the trade studies.

1.3.2 Limitations

The design of the rocket vehicle is conceptual.

Because it is a conceptual project, no experimental prototype will be made.

Part thickness and tolerance calculations are not going to be made because we focus on their global dimensioning of masses, stages, and thrust, and with this the trajectory in which it will be used is associated.

Specific calculations are not made for turbines and engine nozzles.

Financial limitations where we must select features of the launcher so that we do not exceed the average budget for small launchers.

Technical limitations such as the possible unavailability of certain materials or technologies.

Legal limitations where we must comply with the regulations established in our field.

Temporary limitations where we must deliver our project around the established date.

1.4 Report's Structure

The objective of this document is to describe and develop the steps needed to overtake a space launch vehicle design project. A detailed description of the structure can be seen in the contents.

The following section describes a global perspective of the project including the main goals and objectives desired. In addition, the baseline point design is described from which a more detailed design is carried out for each of the subsystems of the vehicle.

In this document, in addition to carrying out a preliminary design of the launcher vehicle itself, an analysis has been conducted on aspects such as the costs of a mission of this level, the manufacturing process and launch site, the associated risks, and the viability of this project.

The last section of the report presents the topics related to project management, including the project planing and monitoring.

1.5 Requirements Format

In this report, the requirements of each part of the development of our vehicle are developed, but in a real design of a launch vehicle, applications such as Reqview are used to have an organization of all the requirements that our project must meet. An example of organizing these requirements would be:

Requirements format				
ID	**Type**	**Description**	**Priority**	**Verification Method**
LV-ORB-FUN-01	Type of orbit	Low type orbit (LEO) and sun-synchronous	High	Analisys
LV-ORB-FUN-02	Altitude	Height from the Earth of 560km	High	Analisys
LV-ORB-FUN-03	Orbital period	Less than 120min	High	Analisys
LV-ORB-FUN-04	Orbital speed	The speed required is 121km/s	High	Analisys
LV-ORB-FUN-05	Launch window	Period of time when the rocket can be launched	High	Analisys
LV-ORB-FUN-06	Transfer orbit	It will be injected into an elliptical transfer orbit	High	Analisys

Table 1: Requirements Format

ID: is the requirement identifier LV-UUU-TTT-DD: Launch Vehicle,Stands for the design part to be specified,Stands for the type of requirement and Stands for the number of the requirement.

- ORB: Orbit
- FUN: Functional

2 Executive Summary

2.1 Background and Mission Objectives

The project 'Conceptual Vehicle Design of a Microsatellite Launcher' is part of the subject Aerospace Design and Manufacturing Project at the Rey Juan Carlos University.

The main aim of this report is to study and analyse the technological feasibility of launching Small Sats into LEO orbits. Therefore, technological developments, market growth, current geopolitical situation among others must be taken into consideration in order to offer the most approachable design as possible.

The project's scope is to develop a conceptual ground launcher whose mission is to deliver 100 kilograms of payload to a LEO orbit at 500 kilometres.

Our 2-stage launcher will be powered by Hydrolox and Kerolox (RP-1). It also counts with a third stage known as Kick Stage whose function is to circularize and raise orbits to deploy payloads to unique and precise orbital destinations.

Other actual vehicles operating nowadays will be taken as a reference for a starting point. This study must prioritize the following requirements:

- Implement European technologies and materials whenever possible.

- Deliver 100kg payload to a LEO orbit (500km) from French Guiana.

- Design a competitive LV reducing costs.

- Prior to any of the aforementioned points, the focal responsibility of this team of engineers is guaranteeing the maximum degree of safety possible throughout our systems.

2.2 Baseline Point Design

As mentioned above, other actual vehicles will be taken as reference. Therefore, the following table collects the initial estimated data used for the mass optimization:

Baseline data					
Stage	Propulsion	Structural Mass (kg)	Propellant mass (kg)	Isp (s)	T/W
Stage I	Liquid	1950.61	10106.99	311	1.7643
Stage II	Liquid	635.41	2381.87	343	1.6702
Stage III	Liquid	206.98	1436.38	310	0.599

Table 2: Baseline data

The reason behind choosing these values as starting point will be explained in more detail later in this report. This team has assumed that our initial stage is propelled by 9 Rutherford Sea Level Engines that provide 25 kN thrust/engine. The next stage will be propelled by 3 Rutherford Vacuum Engines that provide 26 kN thrust/engine and the kick stage is propelled by Curie, a liquid-propellant rocket engine designed by Rocket Lab. It will provide with 10 kN.

2.3 Recommendations for Further Study

As already mentioned, this study consists in a conceptual design of a launch vehicle, therefore are different areas that could be deepen if interested such as:

- Increasing the capacity of our LV in other to carry larger payloads and thus be able to compete in a more competitive market.

- Explore different alternatives of materials that could reduce weight and increase strength, as well as more economic fabrication processes.

- Carry out a computational fluid dynamics study to approach greater accuracy in aerodynamic or thermal coefficients.

- Development of an algorithm for GNC.

- 3D model for a graphical visualization of the main components of the vehicle.

3 Market Assessment

3.1 Background

An artificial satellite is a space system developed through a design process followed by a construction and finally an insertion into its operational orbit. For the launch of the satellite, a launch vehicle is required, which will be the focus of study of this project. For the specific trajectory and precise dynamics of the insertion of the final orbit orbital mechanics will be used.

Satellites can be classified by their size, target or even range of use, but perhaps the most commonly used measurements for their classification are weight and orbit. In the context of our project, the project will focus on small satellites in LEO orbits.

LEO satellites occupy the lowest orbit of all satellite types, operating between 200 km and 2000 km above sea level. This proximity to Earth makes them ideal for very high-speed, low-latency communications, which often feature a delay of just 0.05seconds, thus causing information to be transmitted more quickly than it would through fiber-optic cables or satellites in higher orbits. In addition, satellites in LEO orbits are relatively inexpensive compared to satellites in higher orbits. Satellites in LEO orbits can provide global coverage for Earth communication, observation and tracking. Also, we must highlight its ease of maintenance and repair with respect to higher orbits satellites.

Figure 1: Earth satellite network

However, individual LEO satellites are less useful for tasks such as telecommunications, because they move very fast across the sky and therefore require a lot of effort to track from ground stations.

Instead, communications satellites in LEO often function as part of a large combination, or constellation, of multiple satellites to give constant coverage. In order to increase coverage, sometimes constellations like this, consisting of several equal or similar satellites, are launched together to create a "network" around Earth. This allows them to cover large areas of the Earth simultaneously working together.

It is for all this that, in order to meet the high demand for these services in recent years, these orbits are becoming increasingly saturated with a large number of these satellites whose objective is to provide broadband internet service, low latency and global coverage at a relatively low cost. This, in turn, carries an even greater risk of collision compared to higher orbits. This can lead to the creation of space junk and other safety issues.

Another major problem with LEO is aerodynamic drag, it limits the amount of time satellites can orbit because friction causes satellites to fall into lower orbits, with its effect increasing as altitude decreases, if a satellite is allowed to drift into anorbit. LEO will end up falling into the atmosphere. Another aspect to consider is the orbital velocity, since the gravitational attraction becomes stronger the closer it gets to the earth, to stay in orbit the tangential speed has to be faster to compensate for the drag paradox. Drag actually accelerates objects, as they fall into lower orbits they gain the speed corresponding to that orbit. For this reason, satellites orbiting in LEO orbit tend to have short orbital periods and very high orbital velocities.

Due to the very high orbital speeds, satellites tend to stay in view from the ground station for about 5 minutes. This small window of time makes it necessary to track the position and trajectory of the satellite at all times, especially the time of passage over the ground station.

3.2 Industry Evaluation

In recent years, space launch vehicle missions have become increasingly important due to their role in meeting market needs for space-based services, scientific research and national security. As a result, a number of private companies and government agencies have invested heavily in developing new launch vehicles and technologies to meet these demands.

Another major driver of the space launch vehicle market is the need for scientific research and exploration. With the continued exploration of our solar system, there is a growing demand for launch vehicles capable of carrying increasingly sophisticated instruments and equipment to destinations such as Mars and the Moon. Among other missions, last year we saw the Perseverance rover landing in Mars and at the moment, the project Artemis is almost ready to carry astronauts to the Moon for the seventh time in history. This has led to the development of new, more powerful launch vehicles capable of carrying larger payloads over longer distances.

In addition to these market needs, there are plenty of other factors driving the development of new launch vehicles. These include the desire for greater efficiency and cost savings, considering that global warming is an essential issue witch involves reducing the CO2 footprint. To meet these demands, many companies and government agencies are investing in new technologies such as reusable rockets, which can significantly reduce the cost of launching payloads into space.

In the last years, companies such SpaceX have made a huge evolution with reusable rockets and cutting costs. The Falcon 9 rocket has made history as the first stage, witch should be the most expensive one, has been designed to be reusable. The recovery of these boosters has achieved a success rate of over 90%, with the remarkable figure of 99 recoveries at the date. Added to this, it has built the biggest, most heavy and cheapest rocket in history.

Figure 2: Starlink satellites ready for launch on the Falcon 9 rocket

Despite the many opportunities in the space launch vehicle market, there are also a number of challenges that must be addressed. One of the biggest challenges facing the industry is the high cost of launch vehicles and associated infrastructure. In order to address this issue, many companies are exploring new business models, such as launching multiple payloads on a single rocket, or partnering with other companies to share costs.

Another challenge facing the space launch vehicle industry is the need to ensure safety and reliability. With the high stakes involved in launching payloads into space, it is essential that launch vehicles are designed and tested to the highest standards of safety and reliability. This requires significant investment in research and development, as well as rigorous testing and quality control processes.

In conclusion, the space launch vehicle market is a dynamic and rapidly evolving industry, driven by a range of market needs, technological advancements, and national security concerns. While there are many opportunities for growth and innovation, there are also a number of challenges that must be addressed in order to ensure the continued success of the industry. As such, continued investment in research and development, as well as collaboration and partnership between companies and government agencies, will be essential for the future of space launch vehicle missions.

3.3 Mission Justification

The current market for constellations of small satellites in LEO orbits is increasingly competitive, as several companies are investing in this technology with the aim of offering high-speed global connectivity services, Earth observation, communications and other applications.

One key driver of the current space launch vehicle market is the need for communication and imaging services. With the escalation of smartphones and other devices that rely on satellite connectivity, there is a growing need for reliable and affordable satellite-based communication services. This has led to the development of a number of new launch vehicles designed specifically for small satellite payloads, as well as the creation of constellations of satellites that can provide global coverage. For instance, the Starlink project developed by SpaceX, a space-based internet service developed to provide high-speed, low-latency internet connectivity to users around the world, particularly those in remote or underserved areas. This project already has more than 1,500 satellites in orbit and plans to have more than 12,000.

Other major companies in the small satellite constellation market include OneWeb, which has already launched more than 200 satellites and has plans to operate more than 650 satellites, and Amazon with its Kuiper project, which seeks to launch more than 3,000 satellites to offer global connectivity services. In addition, companies such as Planet Labs and Spire Global are operating constellations of satellites focused on Earth observation and obtaining meteorological and global positioning data.

The market for small satellite constellations in LEO orbits is constantly evolving and is expected to continue growing in the coming years due to the increasing demand for connectivity, Earth observation and communications services across the globe. In addition, competition and innovation in satellite technologies are expected to advance, which could result in reduced satellite launch and operation costs.

4 Conceptual design and Project Goals

4.1 Mission Objectives

Throughout the project, we are searching for develop a small launcher that has sufficient capacity to transport a payload of 100kg and be able to inject it into a low orbit (560 km). However, it is necessary that it achives certain requirements and that offers good statistics in the market since, even if we managed to develop it, if it were not an optimal launcher in terms of economics, it would not be efficient.For this reason, it is necessary to reduce the launching and construction costs of our launcher in order to get a launcher with general costs within the margins of the European competitions and offer improvements to them in order to obtein a competitive product.

The main goal of developing a launch vehicle for low Earth orbit is to provide an efficient and cost-effective way to launch satellites into space. Low-orbit launch vehicles can be used for a variety of purposes such as earth observation, scientific research and satellite communication. In addition, these vehicles can be used to carry out manned missions to space such as the International Space Station (ISS). The development of efficient and reliable launch vehicles for Leo orbits may also have economic and military implications including the possibility of commercializing access to space and the use of these technologies for surveillance defense. In short, the main objective of developing a launch vehicle for a low orbit is to improve our ability to operate in space and take advantage of the opportunities offered by this environment.

The main features of a Low Earth Orbit (LEO) launcher include:

1. Load capacity: This is the amount of weight the launcher can carry in a single flight.

2. Orbit height: A low orbit launcher is used to put objects into orbit at heights of up to 2000 kilometers.

3. Reusability: Some launchers can be reused on multiple flights instead of being discarded after a single use.

4. Efficiency: Modern launchers are designed to be more efficient in terms of fuel consumption and cost reduction.

5. Reliability: An important characteristic of a launcher is its reliability, which is achieved through rigorous testing and validation.

6. Propulsion technology: Launchers can use different types of propulsion technologies, such as solid or liquid fuel, or a combination of both.

Competition in the small satellite launcher market is driving innovation and the development of new technologies to make space more efficient and accessible because this sector is constantly developing.

In conclusion, the key features of a low-orbit launcher include its payload capacity, the orbit height at which it can put objects, its energy efficiency, its reliability, and the propulsion technology it uses.

4.2 Mission Requirements

The requirements to launch a rocket into Low Earth Orbit (LEO) can change depending on the specific mission. For our mission we have the following requirements:

1. Payload: The rocket must be able to carry the payload of 100 kg for the mission.

2. Payload mass: The wet mass of the payload shall not exceed 150 kg.

3. Orbit height and speed: the rocket must be able to reach the necessary height and speed to enter the desired orbit.

4. Orientation and control: The rocket must be able to properly orient itself during launch and orbit insertion, as well as control its speed and trajectory.

5. Stability: The vehicle must maintain stability throughout its operation.

6. Safety: The rocket must comply with current safety rules and regulations, and measures to avoid collisions with other objects in orbit must be considered.

7. Nominal position: The upper stage must inject the payload in its final position.

8. Launch date: the launch must be done on the stipulated date. This date will be chosen taking into account a series of factors that could harm us in our launch, such as the presence of an eclipse.

9. Reliability: The vehicle must be designed to be reliable during all phases of operation.

10. Launch cost: The launch cost will not exceed 210 million euros.

4.3 Baseline Design

In this section we are going to deal with the specific dimensions of our vehicle such as tanks, interstages, domes and nozzles. This process includes the selection of the vehicle architecture, the determination of the system parameters, the identification of the main components, the forecast of performance requirements to each component and the evaluation of the integration and operation capacity of the vehicle. The baseline must be designed under a systematic and rigorous approach that guarantees the effectiveness, efficiency and economic viability of the project. Additionally, the baseline design must be validated through laboratory testing and flight simulations before the actual launch takes place. We first start by determining the size of the deposits. For this, it has been necessary to calculate the inert masses in each of our stages, obtaining the following results:

Inert mass			
Stages	Stage I	Stage II	Stage III
m_{inert} [kg]	805.267	459.7484	442.1647

Table 3: Inert mass

The results obtained are physical since in the first stage more fuel is used and so on, and the values obtained decrease in meaning in the same way.

Another thing we have to take into account is that the diameter of the tanks has a negative design allowance as they allow us to create an efficient and compact layout of the components, which allows us to maximize the space available inside the vehicle. When designing our launcher's fuel tanks we use negative design allowances to make the tanks conform more frequently to the shape of the vehicle, allowing them to be larger or better fit the size of the launcher. Once we have obtained the design margins that meet our characteristics, we take our diameter and calculate the dimensions of the dome at each stage, which will be calculated in the more in-depth geometric design section.

Also, keep in mind that the domes will be placed one on top of the other leaving the space between them the diameter that is necessary as seen in the in-betweens. In addition, in the lower part of each tank the nozzle will go and in the upper part the ignition system, made up of everything for each stage.

Finally, when choosing the type of dome that our launcher will carry, we can opt for hemispherical, elliptical or semi-elliptical domes that can be used for the head of the rocket or the payload. These shapes are used in fuel tank construction due to their ability to withstand extreme loads and distribute weight evenly. Elliptical and semi-elliptical domes can be used to reduce the overall size of the launch vehicle and improve aerodynamic efficiency during launch. However, the fuel tanks cannot be spherical because this would result in a very short and very wide launcher and, furthermore, the design margin when calculating the measurements by the MERS criteria would not be positive. This leads us to opt for the option of using hemispherical domes.The advantages of using hemispherical domes are:

1. Can reduce the amount of material needed to build the rocket head, which can result in cost savings and a lighter overall weight for the launch vehicle.

2. They provide a greater area of contact with the support surfaces, which helps to better distribute the forces of compression and tension during the launch.

3. They help protect the payload during the initial stages of launch by damping vibrations, impacts, and temperature changes.

4. They may be easier to fabricate and fit than other types of payload protection.

5 Mission Architecture

5.1 Nominal orbit Design

After having discussed the justification of the mission and having researched the market to be exploited, the mission is defined as follows: To develop a launch vehicle that allows injecting a 100 kg observation payload into a SunSynchronous orbit, in order to provide an optimized way to inject SmallSats and take advantage of its market. Once the mission to be carried out has been defined, the trajectory must be analyzed to reach the nominal orbit.

First, the nominal orbit into which it has been decided to inject the payload is synchronous with the Sun. An SSO orbit is a near-polar orbit, in which combining altitude and inclination makes an object in that orbit pass over a certain terrestrial latitude at the same local solar time. This allows the payload to perform efficient work when conducting an observation mission. More technically, it is an orbit arranged so that it precesses a full revolution each year, so it always maintains the same relationship with the Sun.

A synchronous orbit with the Sun is achieved by causing the osculating orbital plane to rush about one degree eastward each day with respect to the celestial sphere to keep pace with the Earth's motion around the Sun. This precession does not need any impulsive maneuver, it is achieved by adjusting the inclination to the altitude of the orbit (we will see below) in such a way that the equatorial bulge of the Earth, which disturbs the inclined orbits, causes the orbital plane of the spacecraft to precess with the desired speed. The plane of the orbit is not fixed in space relative to distant stars, but rotates slowly on the Earth's axis.

$$\Lambda\Omega = -3\pi \frac{J_2 R_E^2}{p^2} \cos(i)$$

Where: J_2 is the coefficient for the second zonal term (1.0826310^3) related to the oblatidad of the Earth RE is the mean radius of the Earth, approximately 6378 km p is the straight semi-latto of the orbit i is the inclination of the orbit to the equator An orbit will be synchronous with the Sun when the precession rate (360° per year) is equal to the average motion of the Earth around the Sun, which is 360° per sidereal year ($1.9909687110^7 rad/s$):

$$\rho = \frac{d\Omega}{dt}$$

$$\rho = \frac{\Lambda\Omega}{T}$$

Where T is the orbital period. For a circular or nearly circular orbit, it follows that:

$$\cos(i) = -\frac{T}{3.795h}^{\frac{7}{3}}$$

Typical synchronous orbits with the Sun around the Earth are about 600-800 km altitude, with periods in the range of 96-100 minutes and inclinations of around 98°

As for the needs of the market and the altitude of similar payloads, the altitude of the chosen orbit is 560 km. Using the equations defined with the altitude already defined, the inclination of the nominal SSO orbit will be 98 degrees. The eccentricity of this orbit is almost zero, so it can be considered circular.

5.1.1 Tilt Conditions and Launch Windows

Theorbital inclination that we can have depends strongly on what the latitude of the launch base is. The latitude of the launch base must be less than or equal to the inclination of our orbit (as will be demonstrated below), otherwise, we will be forced to perform a tilt change maneuver, which is not interesting due to its high cost. However, our nominal orbit has an inclination around 98 degrees, so we do notsave you from performing an inclination change maneuver

Making use of spherical trigonometry:

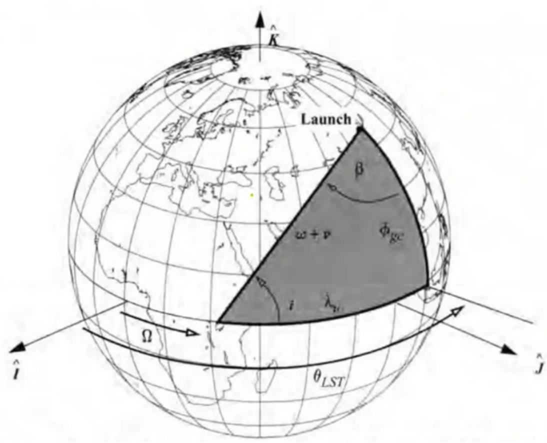

Figure 3: Tilt conditions

The launch point can be identified by 2 coordinates, the longitude and latitude of this launch point. The longitude corresponds to the horizontal arc segment measured from the Greenwich meridian, and the latitude corresponds to the vertical arc segment resulting from cutting the sphere by a plane that passes through the center of the sphere and is perpendicular to the equatorial plane. The third side of the spherical triangle corresponds to the intersection of the vehicle's trace with the sphere.

Examining the spherical triangle we arrive at the following relationship between geocentric latitude and orbital inclination:

$$\cos(i) = \cos(\varphi_{gc})\sin(\beta)$$

Where:

- φ_{gc} is the geocentric latitude.
- i corresponds to the orbital inclination.

By the geometry described, the value of β does not interest us, we are only interested in realizing that the modulus of $sin(\beta) <= 1$ and, because of this, it must be satisfied that $cos(\varphi_{gc}) \; >= cos(i)$, which implies that i $>= \varphi_{gc}$.

That is, if we have a launch baseor at a certain latitude, the inclination we can have will be at least equal to the launch latitude, therefore, only the highest inclinations are directly achievable. Particularly, in our case we want to have an inclination of 98 degrees, so we are not going to have this problem.

As for the launch window, it can be said that since the orbital plane is a fixed reference in space, the rotation of the Earth means that the launch is only possible when the launch site is passing through said plane. Based on this, it can be established that:

- There will be no launch window if $\varphi_{gc} > 1$ for the direct orbits and $\varphi_{gc} < 180 - i$ for retrograde orbits.
- A startup window exists if $\varphi_{gc} = i$ or $\varphi_{gc} < 180 - i$
- There are two launch windows if $\varphi_{gc} < i$ or $\varphi_{gc} = 180 - i$, since the satellite is launched directly towards the ascending node or the descending node. It will be the restrictions associated with the launch base that impose which of the two launch windows we should use.

To determine both the launch window and the inclination conditions obtained at the end of the first phase of flight, a launch base must be established.

5.1.2 Base and Launch Azimut

Due to the rotational motion of the Earth, all points on the surface of the planet naturally have a certain tangential velocity. Some space missions can take advantage of this rotation to reduce the total energy required for their launch.

There are numerous launch bases associated with certain launch vehicles. However, most of these launch bases are located as close as possible to the equator, because the magnitude of the speed that the earth will communicate to the launcher and therefore to the satellite will be greater since the latitude is lower.

In theory, any angle of orbital inclination can be reached from a base right at the equator. However, other criteria become important when selecting the launch base, such as accessibility, security or political reasons.

On the other hand, launch bases located at higher latitudes do not allow direct access to orbits with inclinations much lower than their latitude, so that launches intended to reach orbits with higher inclinations do so to the detriment of their speed and the payload capacity they can launch.

Although a low-latitude launch base is not needed because the nominal orbit has a high inclination (98 degrees), the Kourou Spaceport in French Guiana is a good choice due to its equatorial location and open sea to the east. In addition, multiple launch vehicles are launched there (Ariane 5, Soyuz, Vega). Therefore, the selected launch base will be Kourou Spaceport.

Kourou has a low latitude (5º3’ N), so it allows the launch to an orbit with 98º inclination, as mentioned before.

Azimuth is a fundamental parameter in the design of a launch base for a launch vehicle. The proper choice of azimuth can significantly affect the flight path and efficiency of the vehicle in reaching a specific orbit. In addition, azimuth also has an impact on launch safety and the protection of populated areas and ecosystems.

If an inclined orbit is desired, the azimuth must be chosen to meet the specific requirements of that orbit.

Figure 4: Kourou Spaceport

Choosing the right azimuth is also important to ensure launch safety. The launch vehicle must be directed to a safe area away from populated areas, nature reserves, military zones or any other area that may pose a danger to the launch. Therefore, azimuth must be carefully selected to avoid any possible danger.

In addition, the right choice of azimuth can help minimize fuel consumption and thus reduce launch costs. The azimuth can be chosen to take advantage of the Earth’s rotation and thus reduce the amount of fuel needed to reach the desired orbit.

Using spherical trigonometry as defined in section 5.1.1, the inclination of our nominal orbita (98º) and the latitude of the Kourou spatial base (5º 3’), the azimuth takes the following value:

$$\cos(i) = \cos(\varphi_{gc})\sin(\beta)$$
$$\beta = -8.031$$

5.2 Study Assumptions.

For the study of the trajectory followed by the launch vehicle and the launch into orbit of the payload, a series of assumptions have been made:

- First, it has been considered that the nominal orbit to be reached is circular. This has been considered because its eccentricity is small enough to be disregarded from the calculations without these hardly changing. In this way, the transfer orbit will always share an apse line with the nominal one, facilitating in part, the impulsive maneuver.

- It is considered that at the end of the pitch maneuver the launch vehicle is injected into a transfer orbit, so that it is only necessary to perform an impulsive maneuver at the apogee of this to move to the nominal orbit.

- One of the main hypotheses is that it has been considered that the propulsion system of the vehicle is capable of supplying the necessary thrust to complete the defined trajectory, otherwise, a resizing process would be followed.

- An eplerian K motion has been assumed, neglecting all kinds of orbital perturbations, that is, the term perturbation acceleration has been neglected in the equation of motion. That is why, so it has been considered that the trajectory followed represents a conic section, when in fact it is not, in fact, in most cases it does not become a closed trajectory.

The term of acceleration of disturbance can be given by different causes, some to highlight are:

- The non-uniformity of the gravitational field, as a consequence of the fact that the earth is neither a point mass nor a sphere of homogeneous density.

- Disturbances caused by the presence of a third body.

- Solar radiation pressure.

- Braking by atmospheric friction.

5.3 Trajectory Phases and Sequence of the Mission.

There are several options available to reach the nominal orbit, some of the most efficient in terms of fuel savings and time are shown in the following figure:

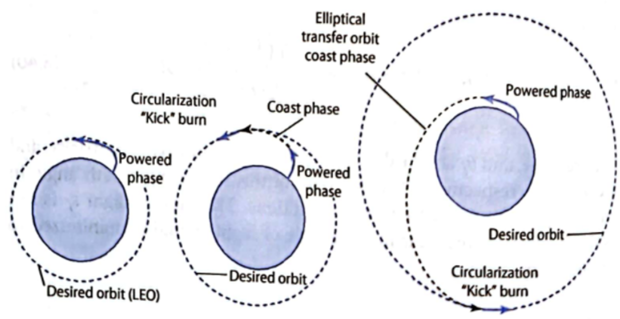

Figure 5: Options available to reach the nominal orbit

The option chosen for energy savings and compliance with requirements is the third shown in the image.

The launcher will perform a classic direct ascent consisting of two phases, a first phase based on an initial vertical flight and then a pitch maneuver, until reaching an altitude and speed necessary to place the payload in a transfer orbit prior to the final. Once the transfer orbit is reached, a second unpropelled phase will begin until it reaches the altitude of the design orbit. The trajectory of this second phase is part of an elliptical orbit whose perigee coincides with the altitude at which the pitch maneuver ends and its apogee with the altitude of the final sunsynchronous orbit (560 km). Finally, once reached apogee, the last stage of the launcher must provide the necessary boost to inject the payload into the desired orbit.

5.3.1 Direct Ascent Phase

As mentioned, it has been chosen to perform a classic direct ascent until reaching an adequate altitude and speed to place the payload in an elliptical transfer orbit to later be able to reach the nominal orbit. In this phase, the launch vehicle must be launched upright and turned with a pitch manoeuvre in such a way that it reaches the required speed, altitude and flight path angle. This maneuver consists of the following phases:

- Vertical flight. In the first seconds of flight, the high atmospheric density makes it necessary for structural reasons to fly with very low or no angles of attack. This ensures that the transverse aerodynamic forces are as small as possible. For this reason, it initially follows a vertical path until it reaches less dense layers of the atmosphere. During this part of the launch, gravity acts directly against the rocket's thrust, decreasing its vertical acceleration. The velocity losses associated with this deceleration are known as gravity losses, and can be minimized by running the next phase of the launch, the pitch maneuver, as soon as possible.

- Gravity turn. This manoeuvre must be performed while the vertical speed is small to avoid large downforce loads on the vehicle during the manoeuvre. After this vertical flight, the launcher receives a slight inclination in such a way that gravity is used to contribute to the rotation of the vehicle thanks to the misalignment that occurs between the thrust and weight vectors. This force creates a net torque in the craft, rotating it so that it no longer points vertically. The turn is performed at zero angle of attack to reduce lateral downforce and produce negligible lifting force during climb.

This is a trajectory optimization that uses gravity to steer the vehicle toward the desired trajectory. It offers two main advantages over a trajectory controlled solely through the vehicle's own thrust. First, thrust is not used to change the direction of the spacecraft, so it is used more to accelerate the vehicle to orbit. Secondly, and more importantly, during the initial ascent phase, the vehicle can maintain a low or even zero angle of attack. This minimizes transverse aerodynamic stress on the launch vehicle, allowing for a lighter launch vehicle.

During this first phase of the trajectory the pitch angle executed by the vehicle and the vertical flight time prior to this turn are critical design parameters to achieve the speed, altitude and flight path angle required in the transfer orbit and are conditioned by the transfer orbite.

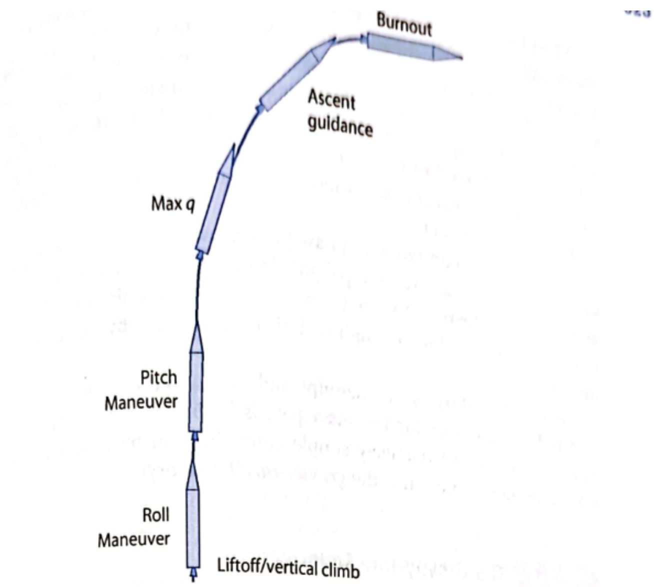

Figure 6: Direct ascent phase

5.3.2 Transfer Orbit

The second phase of the trajectory to follow is based on traveling part of an elliptical transfer orbit by means of a non-propelled coasting maneuver until reaching the apogee of this, where it will now be applied, with the third stage of the vehicle , an impulsive Hohmann maneuver to inject the payload into the nominal orbit of 560 km altitude and 98º of inclination.

As defined above, the nominal orbit of the mission is considered to be circular, as it has a very small eccentricity. Therefore, when defining this transfer orbit, it will always share the apsis line with the nominal one, so it will always be possible to perform the impulsive maneuver from the apogee of the transfer orbit.

The Hohmann transfer orbit has the following characteristics:

- Its perigee corresponds to the point at which the gravity turning maneuver has ended.

- Its apogee radius must be equal to the nominal altitude of the orbit (560 km), to apply an impulse upon reaching apogee andinject the payload into the nominal orbit.

- During the trajectory followed inthis orbit the engines will remain switched off to continue on the correct trajectory.

The altitude at which the turn of gravity will end is not yet known, because for this it is necessary to know the propulsion of the launch vehicle (it will be seen in the following sections). However, it can be said that, from that altitude, the vehicle will be injected into the Hohmann transfer orbit, without the need to perform any impulsive maneuver, only controlling the speed and altitude that the vehicle has at the end of the gravity rotation maneuver.

The speed and altitude with which the vehicle reaches the transfer orbit will be key when it comes to having to apply the impulsive maneuver at its peak to move to the nominal orbit. The higher your altitude and speed, the lower the boost you will need to give the third stage.

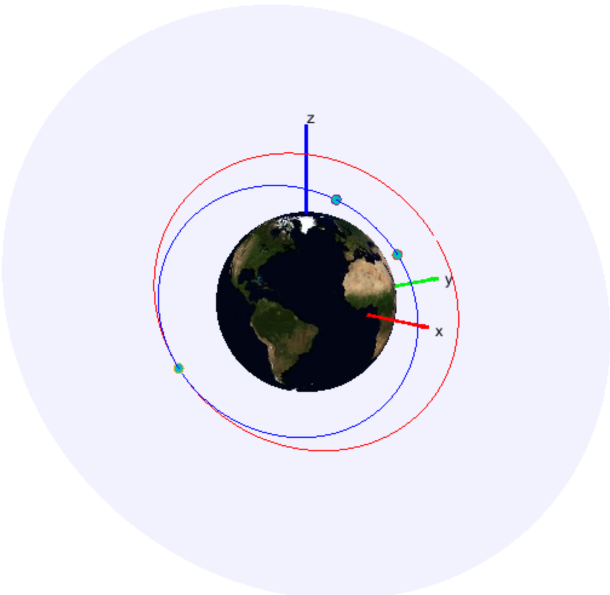

Figure 7: Hohmann trasfer by Matlab

6 Launch Vehicle Systems and Design

6.1 Vehicle Systems and Configuration

This report focuses on a ground microsatellite LV. Therefore, it is important to define the launch platform that will support the design.

A vertical launch platform offers greater flexibility, accessibility and efficiency compared to other launch methods, making it an attractive choice for our space mission.

Once the platform is designed, a further study must be made to design the configuration of the vehicle.

Several mathematical methods were used to obtain the mass distribution of the vehicle as well as the relations of fuel to oxidizer.

This section of the report focuses on the propulsion of the LV as well as its geometry, mass distribution and staging configuration.

Figure 8: Example of OmegA rocket rolling out on the modified Apollo and space shuttle Mobile Launch Platform-3 at NASA's Kennedy Space Center in Florida (Northrop Grumman)

6.2 Propulsion of the Vehicle: Assumptions and Trade-Offs

The launcher will take advantage of the advanced facilities built in the French Guiana. The operational control centre will be situated at the Kourou spaceport, where the operation and monitoring systems will be housed. These mentioned facilities contain the unpacking of satellites and control equipment, electrical and mechanical inspections, and the verification of the launching platform.

Regarding the mass optimization of the LV, the Lagrange Multiplier Method was utilized. However, a more detailed explanation of this mathematical procedure will be provided in section 6.4 which covers the geometry and mass distribution.

Our starting point is: $m_0 = 13000kg$

Mass distribution			
Data	Stage I	Stage II	Kick Stage
Isp [s]	311	343	310
Propellant	LOX	Kerolox	Liquid bi-propellant
Mpropellant/stage [kg]	6288.73	2381.872	1436.38
Mstructure [kg]	1950.61	635.41	206.98
Mpropellant/total [kg]	10106.99	3818.26	1436.98

Table 4: Mass distribution

Stage I:

$$\triangle\ V_1 = g \cdot Isp_1 \cdot ln(\frac{m_{01}}{m_{f1}})$$

$$m_{01} = M_1 + M_2 + M_3 + M_{payload} + M_{p1} + M_{p2} + M_{p3} = 13000kg$$

$$m_{f1} = M_1 + M_2 + M_3 + M_{payload} + M_{p2} + M_{p3} = 6711.69kg$$

Stage II:

$$\triangle\ V_2 = g \cdot Isp_2 \cdot ln(\frac{m_{02}}{m_{f2}})$$

$$m_{02} = m_{f1} - M_1 = 4760.655kg$$

$$m_{f2} = m_{02} - M_{p2} = 2378.783kg$$

Stage III:

$$\triangle\ V_3 = g \cdot Isp_3 \cdot ln(\frac{m_0 3}{m_f 3})$$

$$m_{03} = m_{f2} - M_2 - M_{fairing} = 1699.37kg$$

$$m_{f3} = m_{03} - M_{p3} = 262.98kg$$

$M_{fairing}$ is assumed to be 44 kg.

6.2.1 Staging Configuration

When it comes to staging configuration, once again we find a large number of possible arrangements, such as tandem configuration, parallel staging, stages consisting solely of engines or fuel tanks.

However, this report focuses on series staging configuration as it offers several advantages such as:

- Enhanced efficiency: the series configuration enables the launcher vehicle to shed weight as it ascends, thus improving its efficiency and payload capacity.
- Greater design flexibility: this configuration allows for a higher degree of design flexibility, as different types of engines and fuels can be employed in each stage.
- Improved safety: separating the stages in series decreases the likelihood of a catastrophic failure during launch, as each stage can be equipped with its own safety systems.
- Increased range: the series configuration allows the launcher vehicle to achieve a greater range and reach higher altitudes, thereby enhancing its capability to undertake more demanding space missions.

Initially, a 3 staged rocket was considered. The first stage was going to be propelled by a solid propellant in order to obtain one only impulse allowing us to reach the necessary ΛV, minimizing costs. The next two stages were going to be propelled by liquid propellants. However, the 3rd stage has to be smaller and therefore carry less amount of propellant. This means that this stage would provide with a higher thrust.

After assessing the different pros and cons of both 2 staged and 3 staged LV, the ultimate decision is to design a 2 staged launch vehicle plus an additional kick stage.

Using 2 stages rather than 3 has several conveniences such as:

- Reduction in system complexity, resulting in lower costs and increased reliability.
- Increased efficiency due to the reduction of weight and payload required to accomplish a specific mission.
- Enhanced safety by reducing the number of stages that need to operate correctly during spacecraft launch and separation.
- Increased productivity, enabling earlier entry into service.

6.2.2 Propellant Selection

In order to complete the propellant selection and design optimization studies, it is essential to further define the relationship between combustion efficiency, injector design and the chamber heat transfer characteristics.

As for the staging configuration, a study has been done in order to decide what type of propellant will be used in our LV. After evaluating the different pros and cons of both solid and liquid propellants, our LV will be totally propelled by liquid propellant because we obtain:

- Greater propulsion control which improves the accuracy and effectiveness of the LV.
- Fuel amount adjustment during flight, optimizing the performance of the vehicle and payload.
- Flexibility in fuel selection, which helps in the costs reduction and increases the availability of fuels.
- Improved safety as they are easier to stop and control in case of emergency. However, their incendiary risk must be taken into consideration.

Liquid propellant systems are more complex than solid ones, but they provide greater versatility and capacity. Because liquid propellants are stored in separate tanks, a complex system of feed lines, pipes, and injectors is required to transport them to the combustion chamber. To provide the necessary pressure in these systems, a pump is used that receives energy from a smaller combustion chamber present in the rocket and uses part of the propellant. Despite their complexity, this system provides greater efficiency than solid rocket motors, especially due to their higher specific impulses.

The selected liquid propellant for the first stage is Hydrolox, a binary rocket fuel composed of oxygen as oxidizer and liquid hydrogen combustible. This propellant keeps the temperature in the combustion chamber lower. The selected liquid propellant for the second stage is Kerolox, a highly refined form of kerosene. It provides a lower specific impulse than Hydrolox but is cheaper. As we can observe in the following table, it is much denser than Hydrolox.

Propellant propoerties				
Propellant	Combution temperature [K]	Density [g/ml]	Specific Impulse [s]	O/F
LOX/Methane	3000	0.424	250	2.5-3.6
LOX/LH2	2985	0.28	451	6
LOX/RP1	3670	0.81-1.02	353	2.56

Table 5: Propellant Properties

None has a similar specific impulse as Hydrolox and Kerolox, that is the reason why we have chosen both to propel our LV.

6.3 Geometry and Mass Distribution

To find the masses of each stage of our launcher we use a method called the Lagrange multipliers method. This method is a procedure for finding the maxima and minima of functions of multiple variables subject to constraints. This method reduces the restricted problem with n variables to one without restrictions of n + k variables, where k is equal to the number of restrictions, and whose effects can be easily resolved.

Initially, we know the specific impulses of each stage, the type of fuel that will be used in each stage and the initial mass of the first stage, which is obtained by taking another vehicle as a reference whose characteristics are very similar to those of our mission and, through a rule of 3 and taking into account that its initial mass was 11926 kg and its initial propellant mass was 9950kg, we obtain that our initial propellant mass is going to be 10846.051kg.

Inicial data			
	Stage I	Stage II	Stage III
Isp [s]	311	343	310
Propellent	Lox	Lox/keroseno	viscous liquid bipropellent
$m_p[kg]$	10846.051	?	?

Table 6: Inicial data

We would have the following structure when solving by Lagrange:

Problem structure			
	Stage I	Stage II	Stage III
$m_0[kg]$	M_1	M_2	M_3
$m_{payload}$ [kg]	M_2+M_3+U	M_3+U	U
m_f [kg]	$SM_1+M_2+M_3+U$	SM_2+M_3+U	SM_3+U

Table 7: Problem structure

Were we have that $M = M_1 + M_2 + M_3$, U= 100kg and $S = 0.06$. With this and knowing the equation derived from Newton's second law of motion governs the relationship between the propellant consumption and the velocities that can be attained by the rocket vehicle and is given by:

$$\Lambda V_{vehicle} = C * ln(\Lambda))$$

And knowing that C = g*Isp, we substitute these expressions at each stage and make their summation obtaining:

$$\Lambda V_{total} = g_0*[Isp_1*ln(\frac{M_1+M_2+M_3+U}{SM_1+M_2+M_3+U})+Isp_2*ln(\frac{M_2+M_3+U}{SM_2+M_3+U})+Isp_2*ln(\frac{M_3+U}{SM_3+U})]$$

It must be fulfilled that:

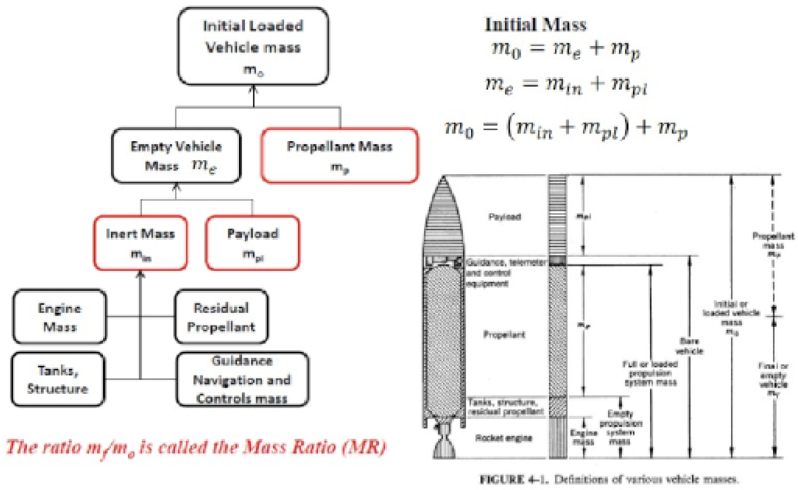

Figure 9: Mass of a launcher vehicle

With this we begin our method of Lagrange multipliers by grouping terms as follows:

$$z_1 = \frac{M_1 + M_2 + M_3 + U}{SM_1 + M_2 + M_3 + U}$$

$$z_2 = \frac{M_2 + M_3 + U}{SM_2 + M_3 + U}$$

$$z_3 = \frac{M_3 + U}{SM_3 + U}$$

The mean specific impulse will be 3152.28s so we simplify M1,M2 and M3 to:

$$M_1 = \frac{U * (1-S)^2 * (e^{\frac{\Delta V}{3*Isp_{medio}*g}} - 1) * e^{\frac{2\Delta V}{3*Isp_{medio}*g}}}{(1 - S * e^{\frac{\Delta V}{3*Isp_{medio}*g}})^3} = 1950.613kg$$

$$M_2 = \frac{U * (1-S) * (e^{\frac{\Delta V}{3*Isp_{medio}*g}} - 1) * e^{\frac{\Delta V}{3*Isp_{medio}*g}}}{(1 - S * e^{\frac{\Delta V}{3*Isp_{medio}*g}})^2} = 635.4107kg$$

$$M_3 = \frac{U * (e^{\frac{\Delta V}{3*Isp_{medio}*g}} - 1)}{1 - S * e^{\frac{\Delta V}{3*Isp_{medio}*g}}} = 206.9846kg$$

It is true that:

$$M_2^2 - M_1 * M_3 = 0$$

With these data we substitute in the expressions and obtain that:

$$m_0 = m_p + m_s + m_{payload}; 13000 = m_p + M_1 + M_2 + M_3 + U; m_{p1} = 10106.99166kg$$

$$z_1 = \frac{m_0}{m_f} = \frac{m_{p1} + m_s + m_{payload}}{m_{p2} + m_s + m_{payload}}; m_{p2} = 3818.2606kg$$

$$m_{p3} = 1436.3881kg$$

Mass calculation			
Variable	Stage I	Stage II	Stage III
Isp [s]	311	343	310
$m_p[kg]$	6288.73	2381.872	1436.38
$m_{structure}$ [kg]	1950.61	635.41	206.98
m_{stage} [kg]	8239.34	3017.28	1643.37
$m_{p,total}$ [kg]	10106.99	3818.26	1436.38
m_{total} [kg]	13000	4660.66	1643.37

Table 8: Mass calculation

We assume that the weight of the mob-cap is 44 kg. Once we have our masses calculated we can calculate ΛV:

$$\Lambda V_1 = g * Isp1 * ln(\frac{m_{01}}{m_{f1}})$$

$m_{01} = M_1 + M_2 + M_3 + m_{payload} + m_{p1} + m_{p2} + m_{p3} =$ 13000 kg

$m_{f1} = M_1 + M_2 + M_3 + m_{payload} + m_{p2} + m_{p3} =$ 6711.26894 kg

$\Lambda V_1 =$ 2017.14366 m/s

$$\Lambda V_2 = g * Isp2 * ln(\frac{m_{02}}{m_{f2}})$$

$m_{02} = m_{f1} - M_1 =$ 4760.655923 kg

$m_{f2} = m_{02} - m_{p2} =$ 2378.783443 kg $\Lambda V_2 =$ 2334.506467 m/s

$$\Lambda V_3 = g * Isp3 * ln(\frac{m_{03}}{m_{f3}})$$

$m_{03} = m_{f2} - M_2$- mobcap $=$ 1699.372696 kg

$m_{f3} = m_{03}$ - $m_{p3} =$ 262.98458 kg

$\Lambda V_3 =$ 5674.446454 m/s

$$\Lambda V_{total} = 10.026km/s$$

Finally, we are going to calculate the thrust-weight relationships of each stage.

$$\frac{T}{W}_1 = \frac{T * n_{motors}}{m_{01} * g} = \frac{25000 * 9}{13000 * 9.81} = 1.7643$$

$$\frac{T}{W}_2 = \frac{T * n_{motors}}{m_{02} * g} = \frac{26000 * 3}{4760.655923 * 9.81} = 1.6702$$
$$\frac{T}{W}_3 = \frac{T * n_{motors}}{m_{03} * g} = \frac{10000}{1699.372696 * 9.81} = 0.599$$

6.4 Propulsion Requirements

The requirements that must be taken into account in the field of propulsion are:

1. Launch inclination: The vehicle will be launched vertically, that is, the inclination of the vehicle will be 90º with respect to the local horizontal at launch.

2. Thrust Capability: The propulsion system must be capable of providing the appropriate amount of thrust to achieve the target payload speed and altitude. For our first stage we will have a thrust of each engine of 25KN, for the second of 26 KN each engine, and for the third of 10 KN.

3. Velocity increment: The first/second/third stage will deliver a gross velocity increment of $\Lambda V = 10.026$ km/s.

4. Engine Efficiency: The engine must be efficient enough to maximize the amount of power converted to thrust and minimize the amount of fuel needed.

5. Reliability: The propulsion system must be reliable and safe to ensure mission success.

6. Flexibility: The propulsion system must be flexible in terms of adjusting the amount of thrust and burn time to meet mission requirements.

7. Cost: The propulsion system must be affordable and profitable for our mission.

7 Trajectory Design

7.1 Assumptions

- The trajectory is contained in a plane that passes through the center of the Earth. There are no forces outside the plane.
- The Earth does not rotate at any time and has a constant density.
- Alterations of the third body are considered insignificant.
- The vehicle is considered totally rigid for control cases.
- The vehicle is considered a point during the trajectory.
- An average specific impulse is assumed during the low vacuum impulse in the atmosphere.
- Steering losses are negligible.
- The density and pressure behavior of the atmosphere are modeled with exponential functions.
- Drag is only assumed up to 70 km, just like atmosphere.

7.2 Initial Trajectory From Mass Optimization

Starting from all the data known through mass optimization, and knowing the characterization of flight through the differential equation model, you can try to design a trajectory.

As for the vertical flight phase, it is very ineffective in terms of obtaining a speed increase, since the energy obtained from the combustion of the propellant is being used to obtain a height increase in order to maximize the potential energy. On the other hand, this phase must be sufficient to move away from the launch pad and ground station, as well as to achieve sufficient height at the end of the maneuver to initiate the Hohmann transfer.

Since the pitch over angles should not be very high, a method to obtain 0º of flight path angle at the end of the gravitational rotation is to carry out a coasting stage where the ejection of a stage is used to maintain an unpropelled movement for a few seconds. For this reason, the spin of gravity will continue to decrease the angle of the flight path, so that as soon as the engines are turned on, the flight will have a smaller vertical component and a larger horizontal component.

The maneuver will always end with a flight path angle of 0º, so that depending on how the coasting stage is distributed, a greater transmission of potential energy to kinetic energy can be made, then this inertia time will be a design parameter when defining the trajectory.

On the transfer orbit side, the most interesting thing from a mission point of view will be that the maneuver that places the vehicle at the perigee of the transfer orbit is at the highest possible height and with the highest transverse velocity, since these hypotheses will imply a lower propellant cost for the last stage. Speed increases can be known as a direct relationship to the fuel consumed by means of the Tsiolkovsky equation, which can be written as:

$$\Lambda V = v_e \cdot \ln(\frac{m_0}{m_f}) = I_{sp} \cdot g_0 \cdot \ln(\frac{m_0}{m_f})$$

One of the main requirements to take into account is that the remaining propellant mass after finishing the pitch maneuver, that is, that of the third stage must be able to give us a sufficient AV to first inject us into the Hohmann transfer orbit and then perform the circularization maneuver to move to the nominal orbit.

Next, a series of graphs corresponding to the nominal path followed by the launch vehicle will be presented. These have been carried out with the numerical calculation tool MATLAB, and with the tool "ode45" the system of differential equations that govern the first phase of the trajectory has been solved, that is, the vertical flight stage and the one corresponding to gravity rotation. In this code, we have tried to arrive at the most optimal solution by modifying the following parameters: vertical flight time, inclination angle and inertia time(s).

In the graphs you can also see the trajectory of fall of each of the stages already expelled, which is of great importance in order to prevent you from ending up in populated areas, nature reserves or any other area in which it may be a serious danger.

7.2.1 Simulation 1:

First, a trajectory has been developed in which the vertical flight time is 10 seconds. The pitch angle will be defined as 8^{o}, and a coasting that will last 4 seconds. From these data, the resulting trajectory obtained is as follows:

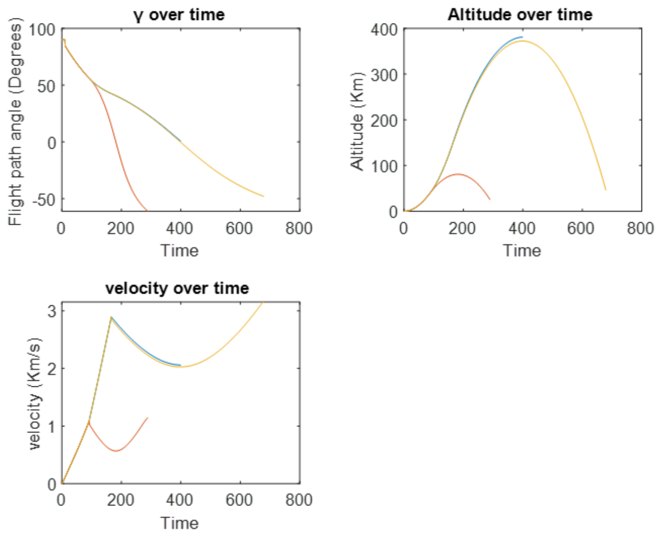

Figure 10: Flight path angle, velocity and altitude of Simulation 1

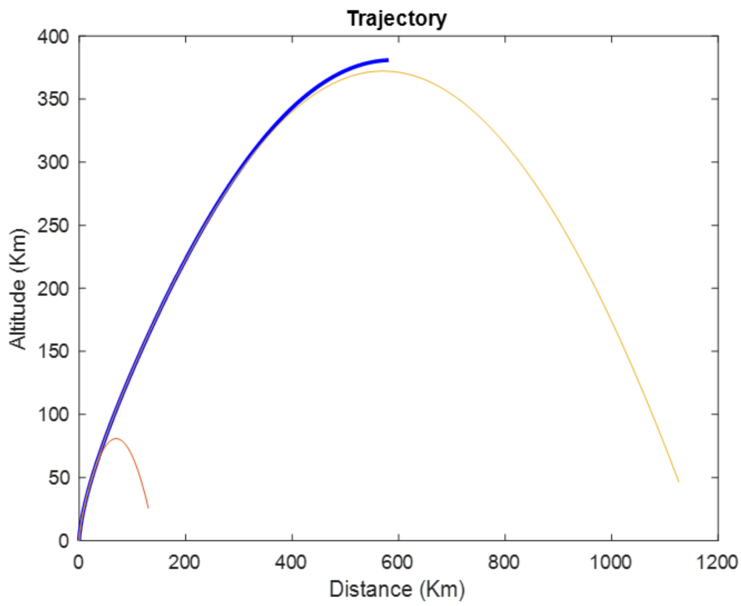

Figure 11: Trajectory Simulation 1

It obtains a height 380.354 km and a speed of 2.06024 km / s. However, performing the corresponding calculations, we can see how the Hohmann maneuver will require a higher speed than we can obtain after consuming the entire mass of propellant of the third stage, therefore, this trajectory is not valid.

7.2.2 Simulation 2:

In this case, it has been decided to reduce the vertical flight time, to reduce the losses associated with it and try to get the vehicle to finish the direct ascent phase with a higher tangential speed. Thus, the vertical flight time has been reduced to 7 sec and, therefore, the pitch maneuver has been started earlier, likewise , the pitch angle has also been increased to 10 º. In this way, it has been possible to reach a height of 275.105 km and a tangential speed of 2.5983 km/s.

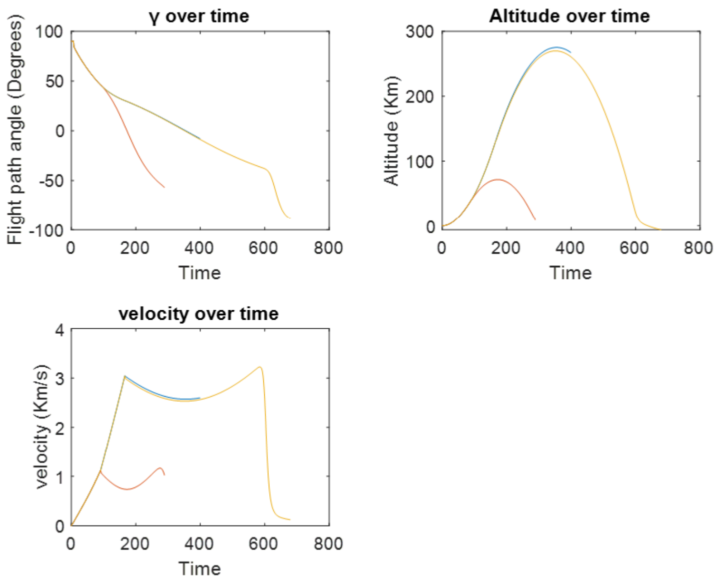

Figure 12: Flight path angle, velocity and altitude of Simulation 2

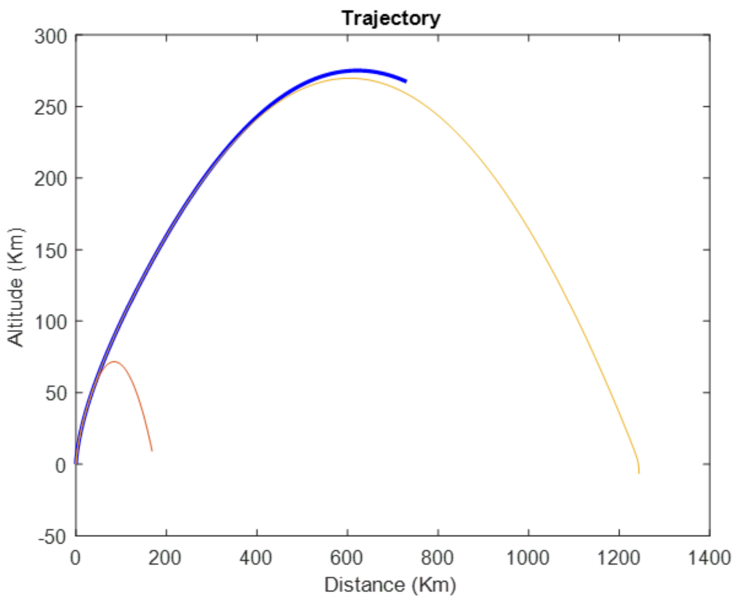

Figure 13: Trajectory Simulation 2

By reducing the vertical flight time, load losses due to friction are reduced, thus we manage to finish the pitch maneuver at a higher speed, and at a lower height.

However, no great progress has been made, because although the final tangential velocity has been maximized, the maximum height at which the vehicle reaches (distance from the apogee of the transfer orbit) in this case is much lower, therefore, the Hohmann transfer will require a higher speed at its perigee.

The tangential speed has been maximized, but not so much that by consuming the entire mass of propellant of the third stage we can inject ourselves into this transfer orbit and then perform the circularization maneuver to move to the nominal orbit.

7.2.3 Simulation 3 and Mass Reconfiguration:

After performing a series of simulations of the trajectories changing the parameters already mentioned, only results similar to those already shown have been achieved. Therefore, and as demonstrated in simulation 1 and 2, it has been concluded that the problem lies in the configuration of masses obtained during the process of optimizing these.

Thus, an iterative process of mass distribution has been carried out in stages until finding a distribution with which it is possible to perform the Hohmann transfer maneuver and subsequently inject the payload into the nominal orbit.

Special emphasis has been placed on the propellant masses, but taking into account how the inert masses of the stages have to vary and respecting the structural relationships obtained in the optimization.

In this way, the following distribution of propellant masses by stages has been reached:

	Etapa 1	Etapa 2	Etapa 3
Masa de combustible(kg)	5765.512	2483.628	550.000

Figure 14: mass reconfiguration

Thus, the nominal trajectory obtained is:

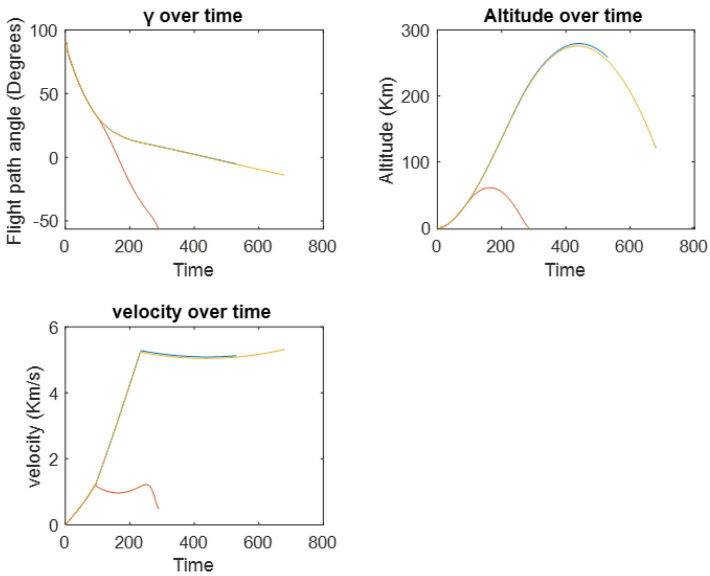

Figure 15: Flight path angle, velocity and altitude of Simulation 3

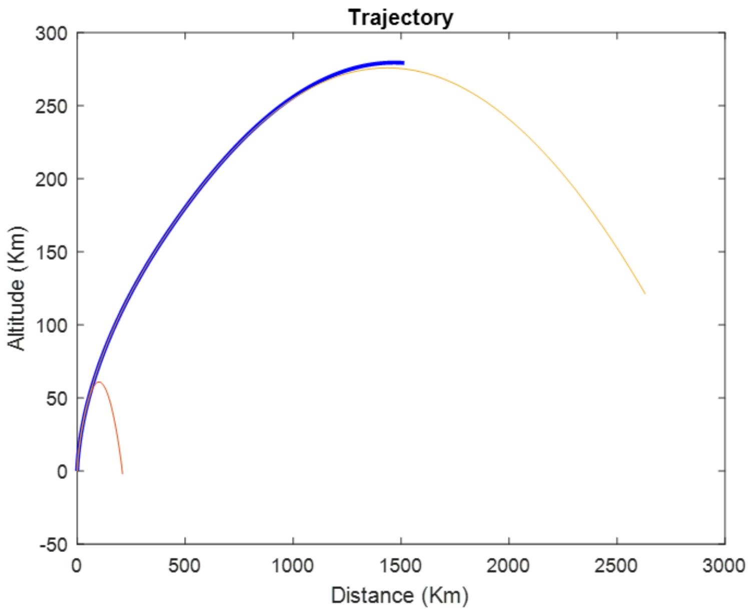

Figure 16: Trajectory Simulation 3

Thus, the vehicle completes the ascent maneuver at an altitude of 279.444 km with a speed of 5. 121 km/s , it has therefore been possible to maximize both altitude and speed.

Thus, the transfer orbit will have its perigee at an altitude of 6657.444 km, its apogee at 6938 km with a semi-major axis of 6797.722 km. Which corresponds to a perigee velocity Vp = 7.8171839 km/s, so you would have to perform an impulsive maneuver ΛV

= 2.6971839 km/s.

On the other hand, the speed at apogee is Va = 7.5010791 km / s and the speed to stay in the nominal circular orbit is 7.579691 km / s therefore, the second impulsive maneuver involves ΛV = 0.078615088 km/s.

The propellant mass of the third stage had an ΛV = 3.0549 therefore the two maneuvers can be performed perfectly, leaving us an ΛV = 0.279101012 km/s.

7.3 Nominal trajectory Requirements

1. Maximum pitch angle: The pitch angle cannot exceed 10 degrees.

2. Mass available after ascension flight: final altitude and speed should lead to a Hohmann transfer that can be performed with the remaining propellant mass in the third stage.

3. Factors such as desired orbit altitude, orbital chelation, orbital eccentricity, and time to reach desired orbit are included.

4. The trajectory must be kept within safety limits.

5. Perturbations must be taken into account, such as the difference in atmospheric pressure and atmospheric friction that can affect the trajectory of our vehicle.

7.4 Orbital Maneuvers

In this section more emphasis will be placed on the orbital maneuvers performed, thespeed increases made in each of them, as well as the masses of fuel consumed and the surplus for this third stage will be calculated, thus verifying that impulsive maneuvers can be performed with the mass of fuel and the AV that provides us with this third stage.

As has already been mentioned on numerous occasions, two impulsive maneuvers will need to be performed.

First, one will be made from the point where the pitch maneuver ends, coinciding in turn with the perigee of the Hohmann transfer orbit, to incorporate us into this because it requires more energy than the vehicle carries at that height.

Once the perigee of the transfer orbit is reached, a coasting maneuver will be carried out, not propelled until reaching the apogee of this orbit, which coincides with the radius of our nominal circular orbit.

And once the apogee is reached, now, we will perform a second impulsive maneuver that allows us to inject ourselves into the nominal orbit.

The speed increases corresponding to the two impulsive maneuvers are those already named in simulation 3: ΛV_1 = 2.6971839 km/s and ΛV_2 = 0.279101012 km/s. Knowing these and making use of the Tsiolkovski equation can obtain the fuel consumed in each of the impulses.

Maniobra	ISP(s)	Altitud(km)	g0(m/s^2)	m0(kg)	AV(km/s)	mf(kg)
Transferencia Hohmann	310	279.444	8.99	770.372696	2,6971839	292.6760880
Órbita nominal/Circularización	310	560	8.29	477.696608	0.279101012	262.5554731

Figure 17: fuel consumed per maneuver

Therefore, the amount of fuel mass consumed and left over is:

Maniobra	m0(kg)	mf(kg)	Masa de combustible consumida(kg)
Transferencia Hohmann	770.372696	292.6758895	477.6968065
Circularización	292.6758895	262.5554731	30.1204164

Figure 18: mass consumed and left over

We had 550kg of fuel mass and the total mass consumed in both maneuvers is 507.8172229 kg, therefore we have a remaining fuel mass of 42.1827771 kg.

Moreover, the acceleration of the nominal orbit is given by:

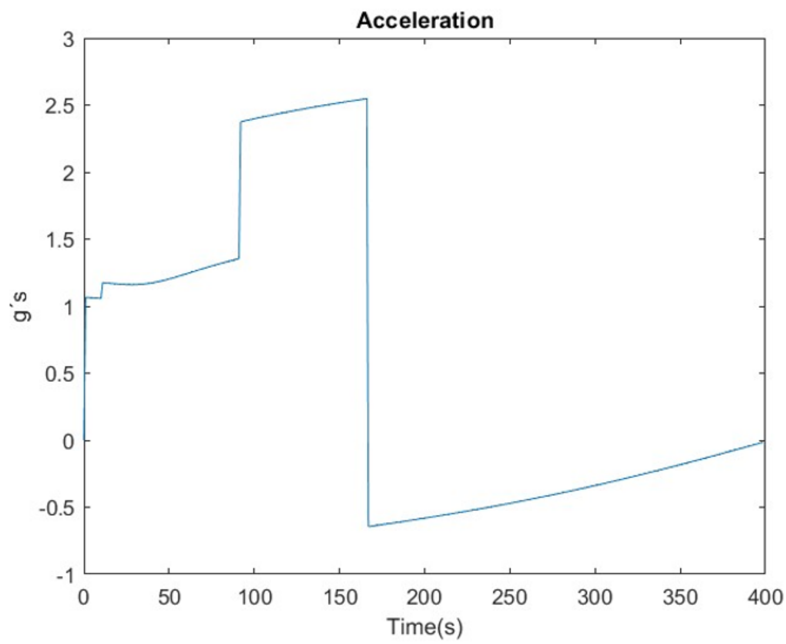

Figure 19: Aceleration of the nominal orbit

7.5 Orbit Requirements

We must take into account that the launches of a launcher directly depend on the type of orbit in which it is going to be injected and we have to meet a series of requirements to be able to carry out our mission correctly.

1. Type of orbit: our orbit is going to be a low type orbit (LEO) and sun-synchronous.

2. Altitude of the orbit: the orbit in which we want to inject our launcher has a height from the earth of 560 km.

3. Orbital period: Our orbit will have a period of less than 120 min.

4. Orbital speed: it is the speed required to maintain a circular orbit around the Earth and in our case it will be 121 km/s.

5. Launch Window: This is the period of time during which the rocket can be launched to achieve the desired orbit. It depends on several factors, such as the relative position of the Earth and the satellite in its orbit, and the availability of the launch support team.

6. Transfer orbit: The LV will be injected into an elliptical transfer orbit, to then be injected into the nominal orbit applying a V at the apogee of the transfer orbit.

8 Aerodynamics and Loads Distribution

8.1 Initial Design Requirements

When designing a launch vehicle, it is necessary to establish which are the most important requirements, since in this way it is possible to guarantee the effective reach of the mission objectives.Among these objectives are those already mentioned with respect to the trajectory, and the limitations and standards that the launch platform must comply with. However, at the aerodynamic level, there are also certain restrictions or requirements that the launch vehicle must satisfy.

First of all, it is necessary to assess at what atmospheric range the mission will be carried out, since this will determine the properties of the atmosphere and its effects on the launch vehicle.

In this case, most of the flight time will be within the atmosphere, so this still imposes new requirements.Below is a table with the main aerodynamic requirements:

Req.Id	Req.Title	Description
LV-AER-FUN-01	Stability	The vehicle must be stable at the time of takeoff and the first stage with engines at maximum thrust, and avoid collisions with the launch platform.
LV-AER-FUN-02	Pitch maneuver	Possibility of changing planes to be able to perform the pitch maneuver.
LV-AER-FUN-03	Counter push vector	Ability to counter negative effects that can appear as a result of vector thrust.
LV-AER-FUN-04	Vertical launch	Possibility of carrying out the vertical launch without the need to modify the already existing launch bases.

Figure 20: Main aerodynamics requirements

In addition to the requirements already stated, the following considerations regarding the geometry of the vehicle have been carried out, with the aim of simplifying the design process.

- Trapezoidal type interstages are neglected, which allows the radius of the vehicle to be constant at all times, and in this way certain aerodynamic effects are eliminated that may lead to the appearance of shock waves that are harmful to the flight of the vehicle.

This assumption is becoming more common, as can be seen in recent releases such as the Electron, or the Miura I..

Figure 21: Miura 1

- Segments having a constant diameter are assumed to have negligible lift and generate only drag. In most of the launch vehicles that are in phase 0/A this hypothesis is considered.

- At the level of control and stability, it has been decided not to use aerodynamic surfaces such as spoilers, since this makes the aerodynamic study of the launch vehicle very difficult. This lack of aerodynamic surfaces can be compensated with corrections or maneuvers that will need other elements such as vector thrust nozzles and/or compressed air channels that generate a couple of forces in the upper part of the vehicle, generating a moment that modifies its trajectory.

This last element would be located in the upper part of the vehicle, since, in this way, for the same amount of force, it generates a greater moment.

Figure 22: Scout

After surveying different launch vehicles that put a payload of similar mass into orbit, it has been possible to determine that the coefficient of drag is approximately 0.2. The coefficient of friction that depends on the Mach number and the surface qualities will not be taken into account. About the launch vehicles examined, it has been observed that the Scout/Electron model is the closest to the LV.

Figure 23: Electron

- Segments having a constant diameter are assumed to have negligible lift and generate only drag. In most of the launch vehicles that are in phase 0/A this hypothesis is considered..

8.2 Aerodynamics Coefficients and Loads

Although the launch vehicle is intended for an off-Earth mission in a space environment, before reaching this point the vehicle must be built, and its components must be designed to be capable of not breaking during manufacture.

Below is a detailed table of the loads that the launcher vehicle must support during the manufacturing, assembly and transportation processes:

Operation		Loads (x-,y-,z-,g)		
Clean Room Handling	Dolly	1.0	0.75	-1+0.5
	Movement	0.2	0.2	-1+0.5
	Vertical turn	0.2	0.2	-1+0.5
	LV mate	0.5	0.5	-2
Ship	Slamming	0.0	0.0	-1.8
	Waves	0.3	0.5	-1.6
	Vibration	0.1	0.1	0.1
	Shock load	2	2	3

Figure 24: Main aerodynamics loads

Once the loads produced by the transport are known, the loads in flight and the coefficients are obtained. It is necessary to determine the nature of each of the loads to characterize them. The loads will be calculated for the most critical case, so the rest of the loads that appear will not be a problem for the vehicle. In this case, the most critical loads are the following: aerodynamic loads, the loads produced by the effect of the wind, the induced angle of attack and the vibrations of the vehicle itself.

In order to obtain the angle of attack and the normal forces during the ascent flight, it is necessary to know how the atmosphere behaves in this region, that is, the characteristics of the wind.

The wind model to use is the AMR. The model is an envelope that generates a wind. The following profile with an approximate probability of 95% and is applicable up to 40 km altitude.

The evolution of air speed in the atmosphere as a function of altitude is shown graphically below:

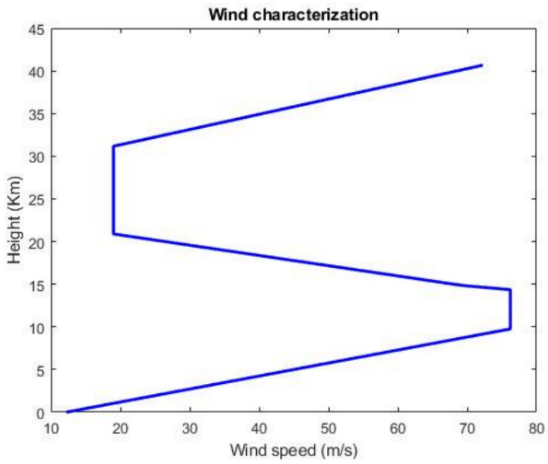

Figure 25: Wind speed vs altitude

However, it is necessary to study the effect of two parameters due to their importance in the behavior of the launch vehicle. These two parameters are the angle of attack and the dynamic pressure.

$$\alpha = arctang\left(\frac{v_w}{v}\right)$$

$$q = \frac{1}{2}\rho v^2$$

Taking these relationships into account, it is possible to obtain the value of the dynamic pressure over time:

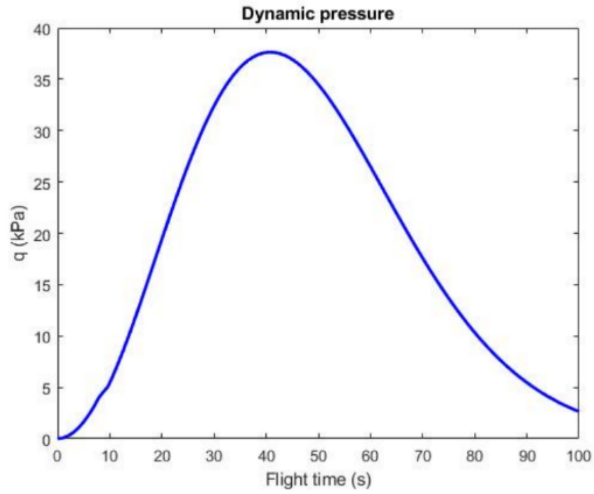

Figure 26: Evolution of dynamic pressure vs time

Although it is also possible to study the effect of angle of attack and dynamic pressure together, which results in the parameter , whose temporal evolution is represented below:

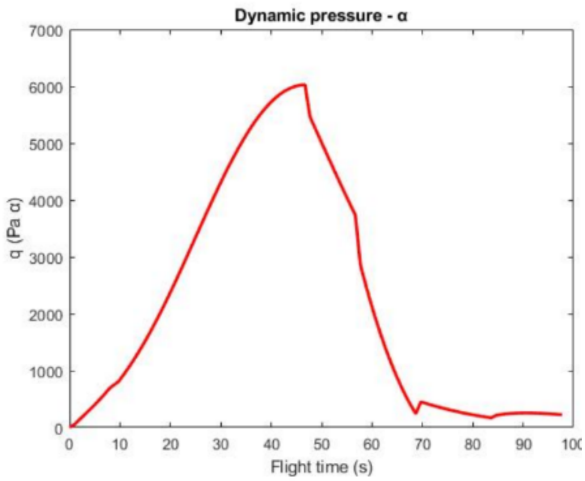

Figure 27: Evolution of parameter dynamic presure*alpha

However, before studying the aerodynamic loads supported by the structure, it is necessary to know the values of the aerodynamic coefficients and forces.

There are different methods to obtain the aerodynamic forces. In this case, tabulated tables will be used to obtain the coefficients that are needed.

To calculate the lift and resistance in each of the segments of the vehicle, it is necessary to use the following expressions:

$$C_{L_k} = C_{N_{\alpha k}} \alpha$$

$$L_k = C_{L_k} q A_k$$

$$\begin{cases} Nose\ cones \rightarrow C_{N\alpha k} = 2 \\ Skirts \rightarrow\ C_{N\alpha k} = \frac{8(S_2 - S_1)}{S_{ref}} \\ Cilinder \rightarrow C_{N\alpha k} = 0 \end{cases}$$

$$D_k = C_D q A_k$$

The drag coefficient for small launch vehicles is usually close to 0.2. However, for large pitchers this value becomes 0.7.

Once this is known, the aerodynamic loads to which the LV is subjected can be calculated. To do this, the following procedure will be followed:

1. The aerodynamic loads in wind axes must be determined. For this, the formulas previously proposed for lift and resistance will be used. Therefore, due to the absence of skirts, only the fairing will generate lift, since constant diameter sections are assumed to have zero lift.

2. The second step is to convert these loads from wind axes to body axes (axial and shear loads), where the axial corresponds to the parallel to the longitudinal axis of the body and the other axis is perpendicular to the longitudinal ones.

Taking into account that the rotation is carried out with an angle -alpha.

To carry out this change, it is necessary to use the following formulas, in which the maximum angle of attack conditions are used, so that the maximum axial and shear loads are obtained:

$$Shear_{loads} = Dsen\left(\alpha_{\max}\right) + Lcos(\alpha_{\max})$$

$$Axial_{loads} = D\cos\left(\alpha_{\max}\right) - Lsen(\alpha_{\max})$$

If the results are represented graphically as a function of their position on the vehicle, we have:

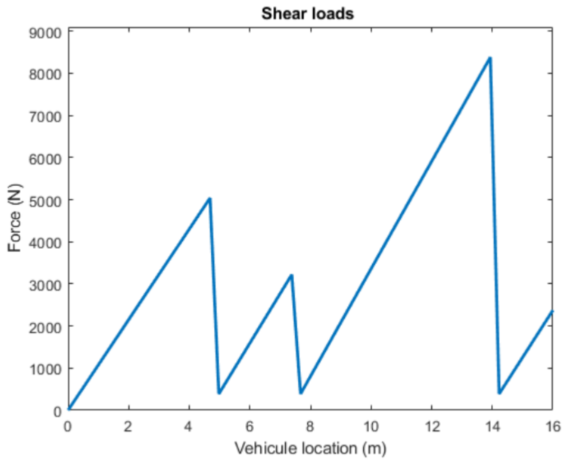

Figure 28: Force through position over the vehicle

Additionally, if you want to calculate the bending moment along the structure, it is necessary to know the value of the center of mass of each section of the structure, whose calculation depends on the following formula:

$$x_{CM(t)} = \sum \frac{m_i(t)l_i(t)}{m_i(t)}$$

If this relationship is represented graphically, something similar to the figure shown below will be obtained:

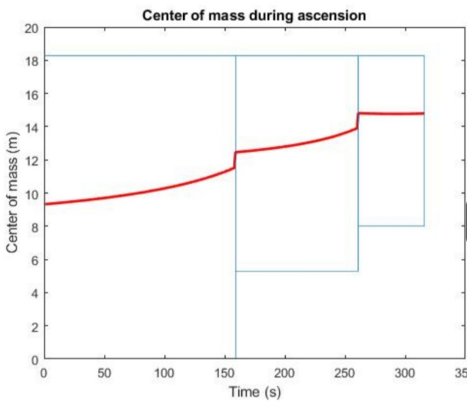

Figure 29: Center of mass position over time

In this way, once the value of the center of mass is known, it is possible to determine the value of the bending moment with respect to this point:

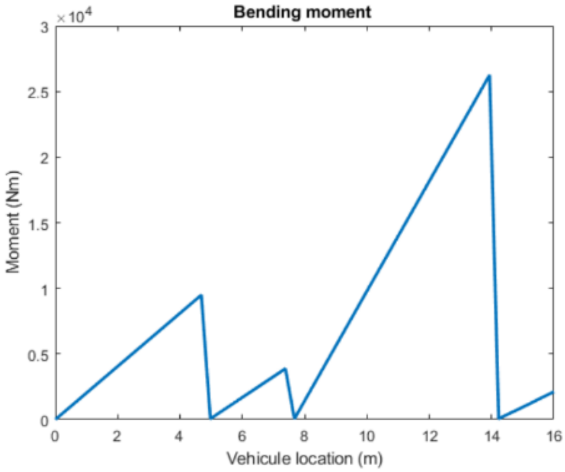

Figure 30: Moment vs vehicle location

As previously mentioned, the analysis of these moments is of vital importance when designing the structure, since they will condition the point where the maximum loads occur.It is very important to take into account all these types of charges, since, added to the effect of the angle of attack, they can put the entire mission at risk.

9 Structural Analysis

9.1 Geometric Design

It is necessary to carry out a series of working hypotheses for the study and structural analysis of the launch vehicle in terms of: the calculation of the inert mass, the calculation of the size of the tank, the calculation of the size of the nozzle and the calculation of the thickness of the tank.

Furthermore, we will assume that the openings in the propellant grain for thrust matching consume about 10% of the internal volume of the engine casing.

The different relationships that allow calculating the rest of the geometric parameters necessary to define the dimensions of a launch vehicle are shown below:

First, we start with the assumptions used to calculate the inert mass of each stage. When calculating the volume of the fuel tank and the oxidizer tank of the third liquid section, it is recommended to increase the volume by 3%.

Recommended for vacuum in tanks. Also, the effect of tank pressurization during the flight leads to an increase in its geometry. Titanium is used to make tanks.

Therefore, the variation of the material constant with temperature will not be considered. If another material is used, they will be different and this must be taken into account.

In addition, it is necessary to use MERs (Mass Estimating Relationships), which are mathematical formulas that relate dimensions and known materials to calculate the mass of each component. MERs are based on curve fitting to previously built components or vehicle components, so they may be somewhat conservative in that they do not indicate technological advances that may have occurred since the component was manufactured. Correlations are shown to calculate the mass of sediments in the liquid phase according to the MER method.

Furthermore, we will assume that the openings in the propellant grain for thrust matching consume about 10% of the internal volume of the engine casing. It is also assumed that the insulation required to protect the ignition system adds up to 2% volume.

The mathematical relationships that will be used to calculate the inert masses of the liquid stages are shown below:

$$m_{avionics} = 10m_0^{0.361}$$

$$m_{fairing} = 13.3A_{fairing}$$

$$m_{thrustestructure} = 2.5510^{-4}T$$

$$m_{wiring} = 1 - 058 m_0^{\frac{1}{2}} l^{\frac{1}{4}}$$

$$m_{engines} = 237.8 \left[\frac{T}{P_c} \right]^{0.9375}$$

The different relationships that allow calculating the rest of the geometric parameters necessary to define the dimensions of a launch vehicle are shown below:

$$A = 2\pi R t \longrightarrow t = R_{out} - R_{in}$$

$$I = \pi R^3 t$$

$$h_{dome} = \frac{D}{2} = R$$

$$vol_{sphericaldome} = 2\pi \frac{R^3}{3}$$

$$vol_{cyl} = \pi R^2 h_{cyl}$$

$$vol_{tank} = vol_{cyl} + 2vol_{sphericaldome} = \pi R^2 h_{cyl} + 2 * 2\pi \frac{R^3}{3}$$

$$h_{tank} = h_{cyl} + 2h_{dome} = h_{cyl} + 2R = h_{cyl} + D$$

However, in this case, as previously mentioned, the "Electron" vehicle has been taken as a reference, whose final dimensions are shown in the following figures:

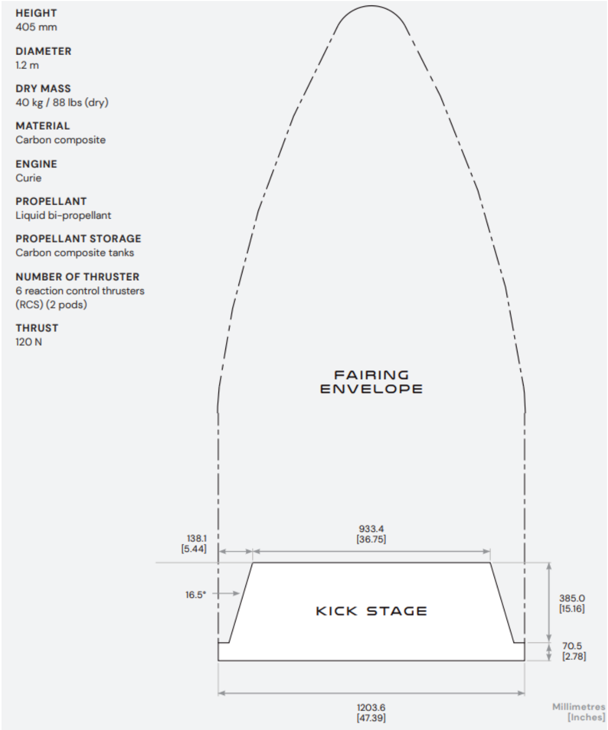

Figure 31: Kick stage placement inside the dolly

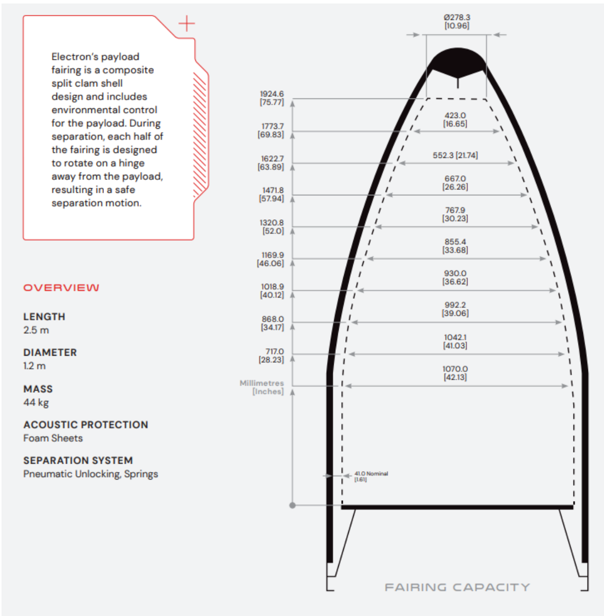

Figure 32: Dolly dimensions

9.2 Technology Requirements

The requirements that we have to take into account to make a good development in the technological aspect in our launcher are:

1. Safety: the launcher must be designed to minimize the risk of failures and accidents,

which implies the implementation of redundancy systems, quality controls, tests and exhaustive verifications.

2. Fuel Efficiency: The launcher must be efficient in terms of fuel consumption, as this affects both the cost of the launch and the feasibility of the project itself.

3. Reliability: The reliability of the launcher is critical as it ensures that the payload reaches its intended destination without issue. This involves careful design of critical systems, implementation of security measures, and establishment of maintenance and repair protocols.

4. New technologies: The launcher design must take into account the possibility of incorporating new technologies, such as more efficient engines, new materials and advanced control systems.

5. Satisfy technological regulations: the launcher must comply with the norms and regulations established for the launch of vehicles and payload, both nationally and internationally.

6. Sustainability: Launcher design must also take into account environmental sustainability, minimizing the amount of waste and emissions generated during launch and production of the vehicle itself.

9.3 Loads Response Analysis

Once the geometry of the structure has been analyzed, it is necessary to study how the loads to which it is subjected affect it. For this, the following relationships must be taken into account:

$$Total\ stress \rightarrow\ \sigma_{tot} = |\sigma_a\ | + \sigma_b$$

$$Alloawble\ stress \rightarrow\ \sigma_{allow} = \frac{\sigma_{tot}}{(\sigma_{allow} - T_i\ )}$$

$$Shear\ stress \rightarrow\ \sigma_s = (\frac{F}{A})FS$$

$$Minimun\ thickness\ for\ strength \rightarrow t_{Ti} = \frac{(\frac{P/2}{\pi R} + \frac{M}{(\pi R^2\ )})}{\sigma_{allow}}$$

A structure, even if it is adequate in strength and rigidity, can present instability under load. Elastic stability or buckling serve to reduce allowable stress levels.

It is essential that elastic stability is guaranteed, that is, the balance of forces and moments, to avoid instabilities in any circumstance.

Critical stress (σ_c) includes stability considerations and can reduce allowable loads below those allowed for strength and stiffness.

Finally, the following table shows the dimensions of each stage and the loads suffered by each of them:

Nozzle dimensions(m)	Stage 1	Stage 2	Stage 3
$D_{chamber}$	0.668	0.4661	0.284
$L_{combchamber}$	0.6029	0.4195	0.256
$L_{convsection}$	0.0867	0.117	0.11
D_{throat}	0.56989	0.33	0.16
L_{nozzle}	0.9242	0.74	0.62

Figure 33: Geometrical dimensions

Loads	Stage 1	Stage 2	Stage 3
$P(N)$	115430	50023	35987
$F_b(N)$	9234.1	4059.23	2904.5
$M(Nm)$	7387.3	3259.3	2323.4
$\sigma_a(MPa)$	5.5362	2.447	1.742
$\sigma_b(MPa)$	0.6226	0.2747	0.2001
$\sigma_{tot}(MPa)$	6.1587	2.3456	1.9369
$\sigma_{allow}(MPa)$	0.58734	0.26249	0.1902
$\sigma_s(MPa)$	0.43223	0.18345	0.13931
$\sigma_c(MPa)$	0.39523	0.11411	0.069798

Figure 34: Loads through stages

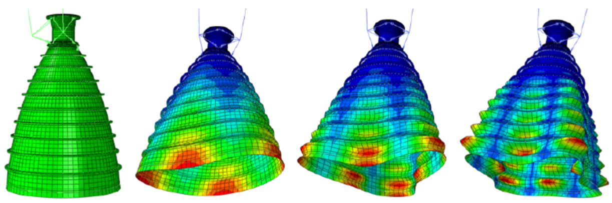

Figure 35: Deformation of the nozzles due to the action of the loads

9.4 Other Effects to Study

In addition to the study and analysis of the structural loads experienced by the launch vehicle throughout its mission, it is also possible to study other effects, which, although less important, also influence the correct development of the mission.

Among all of them, the effects of: shock waves, vibrations and lateral "g" force/load stand out.

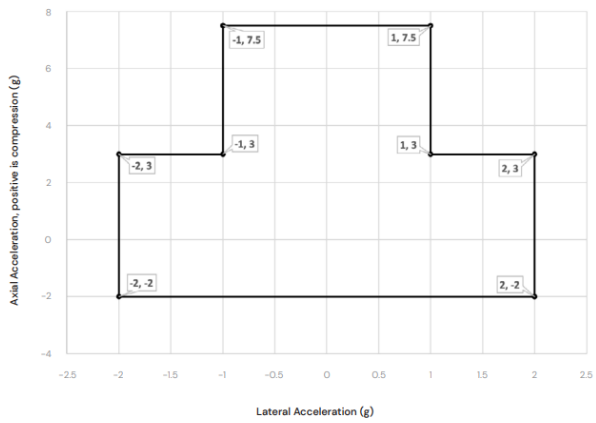

Figure 36: Lateral acceleration

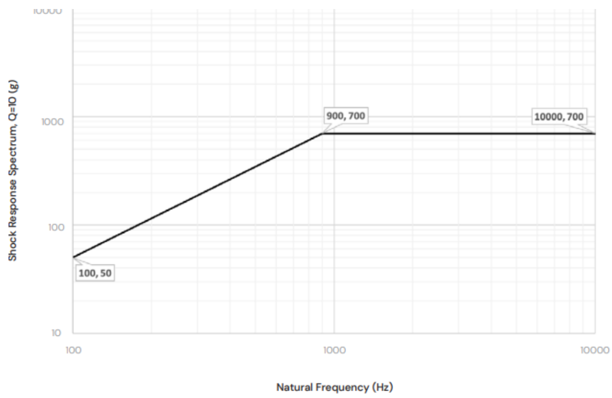

Figure 37: Natural frequency of the vehicle

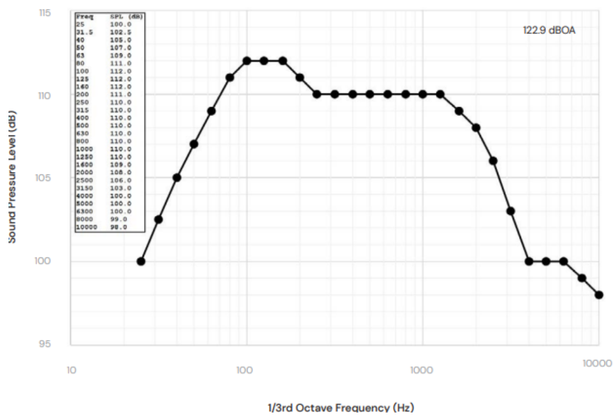

Figure 38: Octaves of vibration vehicle

9.5 Heat Transfer and Thermal Protection System

The main thermal component to be studied in the launch vehicle is convective heat transfer.

Part of the kinetic energy is dissipated as thermal energy, the tip of the casing the element that suffers the most from this effect

This thermal phenomenon depends mainly on:

i. The coefficient of friction of the vehicle surface.

ii. Type of flow and transition points.

iii. Other influencing factors: flight time, weather conditions and/or internal heat sources.

To alleviate the effects of this phenomenon, thermal protection systems are used, which are based on the following points:

- RCC: Highly reinforced carbon composite that withstands high temperatures and can be exposed to the aerodynamic forces generated at launch.

- FRSI: is a type of reusable felt protective blanket that protects different surfaces up to 350 degrees. It is often used on surfaces with complicated shapes thanks to its adaptability.

- SOFI: is a chemical product made up of polyol resin and isocyanate that, when mixed, increase their volume about 50 times, which allows thermal insulation. Because it does not support very high temperatures, it is used in the coldest areas, it also has a great versatility of adaptation to all shapes.

To join the different pieces of thermal insulation, an adhesive from the silicone family is usually used, since it supports both high and low temperatures quite well (launch and orbit).

The maximum temperature to which it will be subjected can be approximated in a certain way.

The fairing is the area that suffers the most extreme temperature rise, along with the areas near the nozzles through which the high-temperature airflow exits. Therefore, these two areas will be the ones that require the greatest thermal protection. The fairing requires special attention since the payload and a large part of the systems and sensors that send information to the ground for the correct control of the launch vehicle.

Temperature gradients, in addition to generating temperature increases, can also generate displacements. These displacements are due to the coefficient of thermal expansion presented by each of the different materials used for the construction of the launch vehicle. However, it is necessary to emphasize that not all materials are the same, since materials such as aluminum generate greater displacements than carbon fiber due to its higher coefficient of thermal expansion.

Displacements must also be taken into account to avoid the possible collapse of any of the structures/skins.

Below is a thermal study for a very common type of hood on shuttles. This study allows us to know the order of magnitude of the displacements to which the casing of the launch vehicle that is being designed will be subjected.

It must be taken into account that the heat produced by aerodynamic drag must be taken into account until approximately 80 km, at which time the atmosphere is left.

Beyond that distance, other agents will cause the temperature gradients. One of the most notable agents is solar radiation.

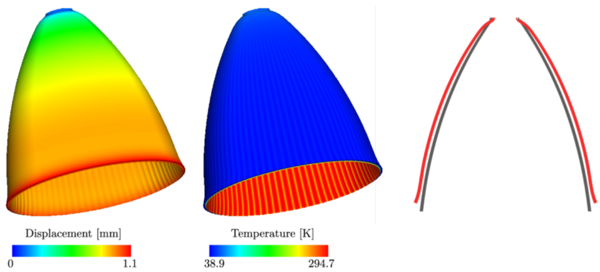

Figure 39: Displacements and temperature gradients on the dolly

9.6 Structural Requirements

The requirements that we have to take into account in the structural aspect are:

1. Optimum Payload Structure: The payload structure must survive the launch process, vibration and shock environment.

2. Design margin: The design margin of the three stages must be positive in order to size our vehicle.

3. Geometry of the domes: Fuel and oxidizer tanks with spherical geometry cannot be assumed.

4. Payload Fairing: The payload fairing must be precisely dimensioned to fit the payload perfectly and must be large enough to store the payload.

5. The resistance to rigidity of the materials used must be taken into account, as well as that of the structures of our launcher. Our materials must exist are the vibrations and the factors that will affect it when carrying out a maneuver of this type.

6. We must take into account the reduction of bstructural mass and the optimization of mass and volume of the payload while maintaining the safety and reliability of the system.

7. Structural analysis and computer simulation are normally used to ensure compliance with structural requirements and risk reduction.

10 Guidance, Navegation and Control

During any mission involving a launch vehicle a GNC subsystem must be included. The reason for this lies in the uncertainty of the environment during the ascent.

Predominantly lateral winds, as well as other unexpected ones, could give rise to disturbances, destabilizing the launch vehicle and putting the entire mission at risk.

The objective of the GNC subsystem is to minimize the effect of said disturbances, making the real trajectory resemble as much as possible the nominal theoretical trajectory calculated by the equipment.

f external disturbances cannot be fully mitigated, the guidance system will need to be able to track errors, in position, attitude, and speed, and then update the estimated trajectory the launch vehicle will need to follow to reach its destination.

However, the design of a competent GNC system is very complex and requires a complete study. To study stability and control, we first need to size our launch vehicle and locate its center of mass.

This calculation has already been carried out in previous sections, and it can be seen that since the mass decreases with the fuel burned and the stages are discarded, the center of mass will change.

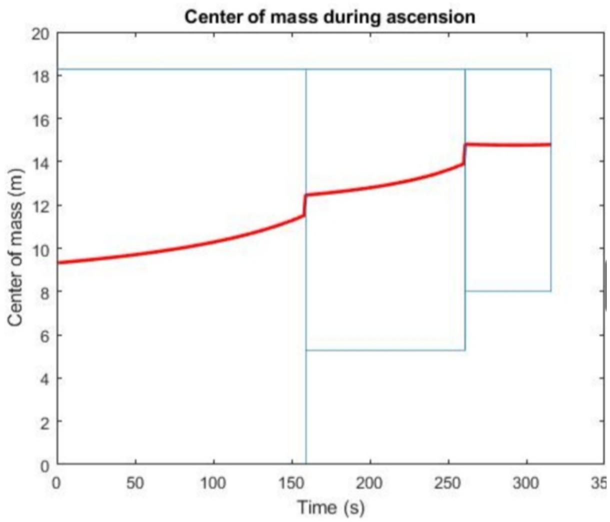

Figure 40: Evolution of the center of mass over time

The red line being the value of the center of mass for each second, the blue vertical lines indicate what time causes the first and second stages to separate from the LV.

Finally, the horizontal blue lines were used to understand the overall size of the rocket remaining. As the scenes are lowered, the horizontal line that marks the bottom of the LV rises, that way it is easier to visualize the LV and how the center of focus is changing.

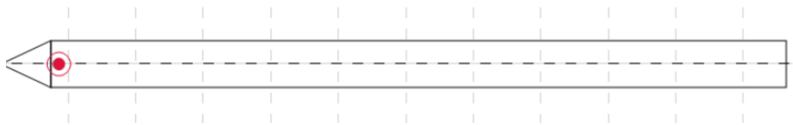

Figure 41: Position of the center of mass

Calculating the center of pressure for a 3D model is a complex process that requires a supersonic fluid dynamics simulation. However this can be avoided, RASAero is software that allows us to locate the estimated center of pressure for a rocket of our own design over a range of Mach numbers.

However, it is reasonable to believe that the launcher vehicle will be controllable if further study is done, mainly because its configuration is very similar to other small LVs. While the exact numbers regarding performance will vary, its general behavior will resemble the LVs it was based on, eg Scout LV and Epsilon LV.

10.1 Guidance, Navigation and Control Requirements

1. Precise adjustments must be made during the trajectory of the vehicle's orientation in real time.

2. We must have the ability to compensate for atmospheric and gravitational disturbances.

3. We must have the ability to maintain communication with the payload throughout the launch.

4. It must be possible to send telemetry to the ground, to have knowledge of maintenance data and other flight data.

5. The launch vehicle will not have redundancies in avionics or undocking systems due to the short flight duration and increased inert energy.

6. The parameters of the launch vehicle must be monitored and we must follow the flight.

11 Manufacturing and Assembly

11.1 Launch Vehicle Fabrication

The fabrication of a launch vehicle involves a series of complex steps that require careful planning, engineering and manufacturing processes. Some of the steps are:

1. Design and Planning:

   - The fabrication process begins with the design phase. Engineers and designers work together to develop a detailed plan for the launch vehicle, considering factors such as payload capacity, mission requirements, safety and efficiency.

   - Computer-aided design (CAD) software is commonly used to create precise 3D models of the launch vehicle, which serve as the blueprint for fabrication.

2. Material Selection:

   - Once the design is finalized, appropriate materials are chosen based on their properties, such as strength, weight and heat resistance.

   - Common materials used in launch vehicle fabrication include aluminum alloys, titanium, carbon fiber composites and high-temperature resistant materials like Inconel.

3. Structural Manufacturing:

   - The fabrication process typically starts with the manufacturing of the primary structures, such as the fuselage, stages and engine sections.

   - Techniques like CNC machining, milling, and turning are used to shape the raw materials into the desired components. Computer-controlled machines ensure precision and consistency in manufacturing.

4. Welding and Joining:

   - Some components of the launch vehicle, particularly metal structures, require welding to join them together. Specialized welding techniques like TIG (Tungsten Inert Gas) welding or electron beam welding may be used.

   - Welding is performed by skilled technicians who carefully follow the design specifications to create strong and reliable joints.

5. Propulsion System Integration:

   - The propulsion system, including engines, fuel tanks, and associated plumbing, is integrated into the launch vehicle structure.

   - Engines are often manufactured separately by engine manufacturers and then integrated into the launch vehicle during assembly.

6. Avionics and Electronics Integration:

   - Avionics and electronic subsystems, including guidance, navigation and control systems, are integrated into the launch vehicle.

   - Sensors, wiring, circuit boards, and other electronic components are installed in designated areas of the vehicle, following precise wiring diagrams and connection instructions.

7. Payload Integration:

   - If the launch vehicle is designed to carry a payload, a dedicated section is provided for payload integration.

   - This area is equipped with mechanisms to secure and protect the payload during launch and deployment. Payload adapters, fairings, and separation systems are installed.

8. Final Assembly:

   - The various components, subsystems, and structures are brought together for final assembly.

   - This includes attaching wings, fins, fairings and other external features as per the design specifications.

9. Quality Assurance and Testing:

   - Rigorous quality control processes and inspections are conducted throughout the fabrication process to ensure the components meet the required specifications.

   - Non-destructive testing methods, such as X-ray, ultrasonic testing, and visual inspections are employed to detect any defects or anomalies.

   - The completed launch vehicle undergoes a series of tests, including static firing of the engines, vibration testing, and simulated flight tests, to verify its performance and functionality.

10. Launch Preparation:

    - Once the launch vehicle successfully passes all tests and inspections, it is prepared for its intended mission.

    - This includes fueling, final inspections, integration with the launch pad or transport vehicle and overall readiness for launch.

The fabrication of a launch vehicle is a highly intricate and specialized process that involves the collaboration of various engineering disciplines and manufacturing techniques. Each step requires meticulous attention to detail and adherence to strict quality standards to ensure the final product is reliable and capable of carrying out its intended mission.

Figure 42: Rocket Lab Launch Vehicle Fabrication

11.2 Vehicle Stacking and Assembly

There are several important things to keep in mind when manufacturing composite materials to be used in a launch vehicle, such as:

1. Appropriate selection of high quality materials and resins in accordance with the design specifications. This includes choosing lightweight materials capable of withstanding the extreme temperatures and stresses experienced during launch into orbit.

2. Careful design of the composite structure, including fiber thickness and orientation, to ensure adequate strength and durability.

3. A well controlled and precise manufacturing process, with attention to the uniform distribution of the resin and the removal of any air bubbles from the material.

4. Rigorous and continuous quality control during the production and testing process to ensure the safety and performance of the composite material.

5. Subsequent tests and tests to verify the mechanical properties of the material and ensure its quality before its use in the construction of the launch vehicle.

The formation of air pockets within a material used in the construction of a launch vehicle can impair the safety and durability of the system. Therefore, it is important to take preventive measures to avoid the formation of air pockets in the materials applied in the development of our launcher. Some measures that can be implemented include:

1. Use a suitable mold and make sure it is clean and dry before placing the material.

2. Make sure there is an even distribution of resin and fibers in the material, removing any trapped air bubbles.

3. Use a proper compaction technique, such as the use of rollers, to aid in the removal of air bubbles.

4. Ensure that the temperature and humidity in the manufacturing area are properly controlled and regulated.

5. Curing times must be followed carefully to avoid the formation of air bubbles.

In addition, it is important to carry out regular inspections and tests during the production process and subsequent tests to verify the integrity of the material used.

The selection of materials used in the development of a launch vehicle is crucial to ensure the safety, reliability, durability and efficiency of the system. The materials used must be lightweight, resistant to corrosion and fatigue and capable of withstanding the loads and extreme temperatures to which they are subjected during launch and in-orbit operation. The most used materials are:

- Carbon fibers and Kevlar for the manufacture of panels and reinforcement structures.

- Aluminum, titanium and magnesium alloys for structural parts and support systems

- Ceramic and ceramic composite materials for thermal protection and heat shield.

- Insulating materials for temperature control and elimination of generated heat.

- Resins, adhesives and bonding agents for joining various parts and structures.

Most of the parts on our launcher today are carbon fiber. Carbon fiber parts are manufactured using the Wet Hand Lay-Up process, also known as Wet Lay-Up or Hand Lay-Up, it is a manual and economical process of manufacturing composite materials that involves the manual laying of layers of fibers in an open mold. In this process, liquid epoxy resin is used to position the fiber layers, resulting in a final fiber-reinforced part. The Wet Hand Lay-Up is commonly used in the composites industry to manufacture large, complex parts at low cost.

Finally on the subject of materials, liquid rocket propellants can be composed of different materials depending on the rocket design and type. However, some of the most commonly used materials in the manufacture of liquid rocket propellants include:

1. Liquid oxidizers: These are usually liquid oxygen (LOX) compounds and used in conjunction with a liquid fuel in the rocket engine.

2. Liquid fuels: various chemical compounds are used, such as liquid hydrogen, kerosene, alcohol or hydrazine.

3. Metals: Some liquid propellants, such as gasoline or kerosene, often contain small amounts of metal, primarily aluminum, which increases the energy density of the propellant.

4. Ceramics: Oxides of beryllium or zirconium are sometimes used to recuperate combustion chambers to protect them from the extreme heat generated during combustion.

As for the Kourou spaceport infrastructures, we have a wide range of facilities and services, including launch pads, assembly and preparation buildings, fuel facilities, tracking radars and communications antennas.Within the launch zones we have: the Vega launch zone (ZLV), the launch zone 2 (ZL2), the Ariane launch zone (ZLA), the Soyuz launch zone (ZLS) and the ELA-launch zone 4. In addition, the CSG has adequate infrastructure for transportation and logistics, including nearby ports and airports.The Guyana Space Center has several facilities for monitoring rocket and satellite launches. Both indoors and outdoors the CSG area, there are weather stations that measure and forecast the weather. Near the ELA-4 launch site is the Kourou station of the ESTRACK network, known as Diane Station, and an auxiliary control room. The space station also has a space station linked to the Galileo satellite system. Outside the main CSG area there are two monitoring stations for monitoring the launchers after takeoff, Grand Leblond Station and Galliot Station.It also has control centers, facilities dedicated to the preparation of satellites and their storage, and buildings that seek to organize activities in the CSG.

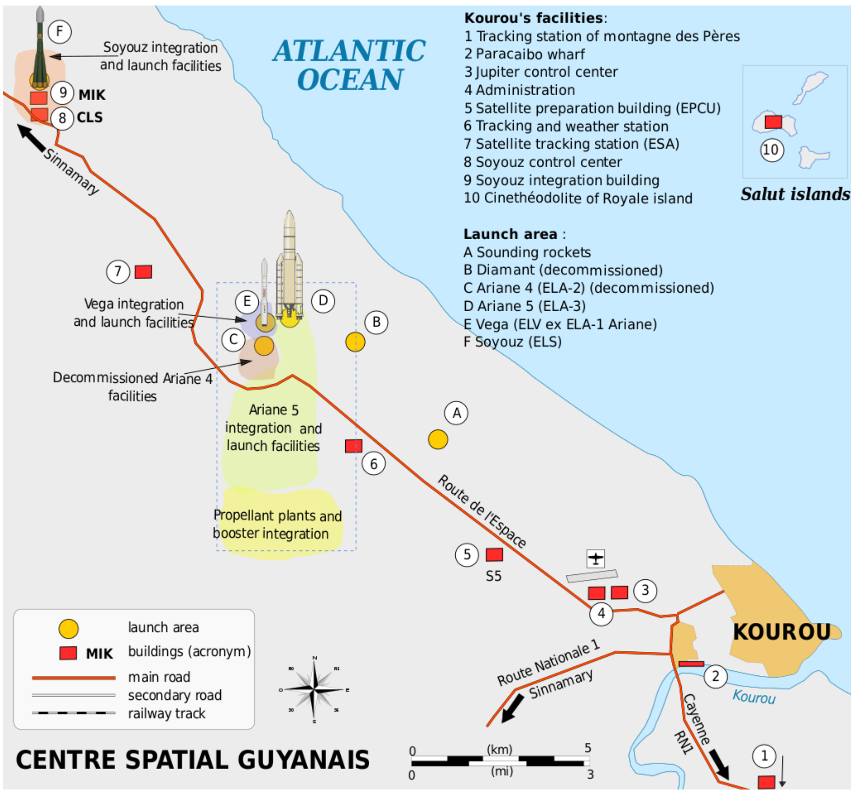

Figure 43: Kourou spaceport infrastructures

11.3 Launch Facilities

The launch facilities of a launch vehicle are all the infrastructures necessary to carry out the launch of the vehicle successfully. This may include launch pads, service towers, fuel supply and cooling systems, real-time monitoring and security control systems, and other specific equipment required for the vehicle in question. These facilities can be expensive and complex to develop and maintain, and they are generally built and operated by government agencies or companies specializing in rockets and launch systems. In some cases, these facilities may be shared among multiple launch vehicles or used by commercial vehicles and governments alike.

The facilities and equipment of a ground segment in the launch operations of a launch vehicle are of great importance. The ground segment includes all the necessary facilities and equipment to carry out the preparation, launch and monitoring of the vehicle. This may include the launch pad, service tower, fuel supply system, cooling facilities, and security and real-time control equipment.

Launch facilities are located in specific areas, called launch bases. The location of the launch site is important, as it must have enough space for the installation and launch of the vehicle, and be far enough away from populated areas.

The ground segment may also include additional facilities and systems to monitor the vehicle's flight and collect relevant data during its mission in space.

The primary elements of a ground segment are: ground stations, mission control centers, ground networks,remote terminals,spacecraft integration and test facilities and launch facilities.

Figure 44: Launch Facilities

11.4 Concept of Operations

The CONOPS for a launch vehicle defines the operational concepts and procedures associated with the entire launch process, encompassing pre-launch activities through post-launch operations. While specific details may vary based on the launch vehicle type and mission requirements, here is a general overview:

1. Mission Planning and Design:

   - Establish mission objectives, payload specifications, and target orbit parameters: deliver 100kg of payload to a LEO orbit.

   - Select an appropriate launch vehicle based on payload characteristics and performance capabilities: ground microsatellite launch vehicle.

   - Evaluate trajectory options and perform mission simulations.

2. Pre-Launch Preparations:

   - Conduct system checks and integrate the payload with the launch vehicle, as mentioned above, in our space base in Kourou.

   - Perform environmental tests, including vibration and thermal vacuum testing.

   - Load propellants, execute fueling operations, and verify system readiness.

   - Conduct final inspections and safety reviews.

   The forward points are common to any launch vehicle project, therefore are generalised. Every following operation will take place in Kourou.

3. Launch Readiness Level:

   - Coordinate with range personnel and obtain necessary launch approvals.

   - Integrate the launch vehicle with the launch pad or mobile platform.

   - Conduct comprehensive system checks, including electrical, mechanical and avionics verification.

   - Ensure operational communication systems for telemetry, tracking, and command.

4. Launch Sequence:

   - Activate launch vehicle systems and validate their functionality.

   - Initiate the countdown sequence, including propellant pressurization and engine ignition.

   - Monitor critical parameters, such as engine performance, trajectory and telemetry data.

   - Progress through various stages of flight, discarding spent rocket stages when required.

5. Spacecraft Deployment or Payload Delivery:

   - Execute the payload deployment sequence, such as separating satellites or releasing spacecraft.

   - Verify successful separation and assess payload health and functionality.

   - Perform any necessary post-deployment maneuvers or operations.

6. Post-Launch Operations:

   - Continuously track and monitor the launch vehicle and its payload.

   - Analyze telemetry and data to ensure mission success and verify orbit insertion accuracy.

   - Conduct orbital adjustments or repositioning maneuvers as necessary.

7. Mission Completion and Deorbit:

   - Fulfill mission objectives, such as data collection, satellite deployment or space exploration.

   - Determine the end of the spacecraft's operational mission life or payload's objectives.

   - Plan for spacecraft deorbit, disposal or execute re-entry procedures when applicable.

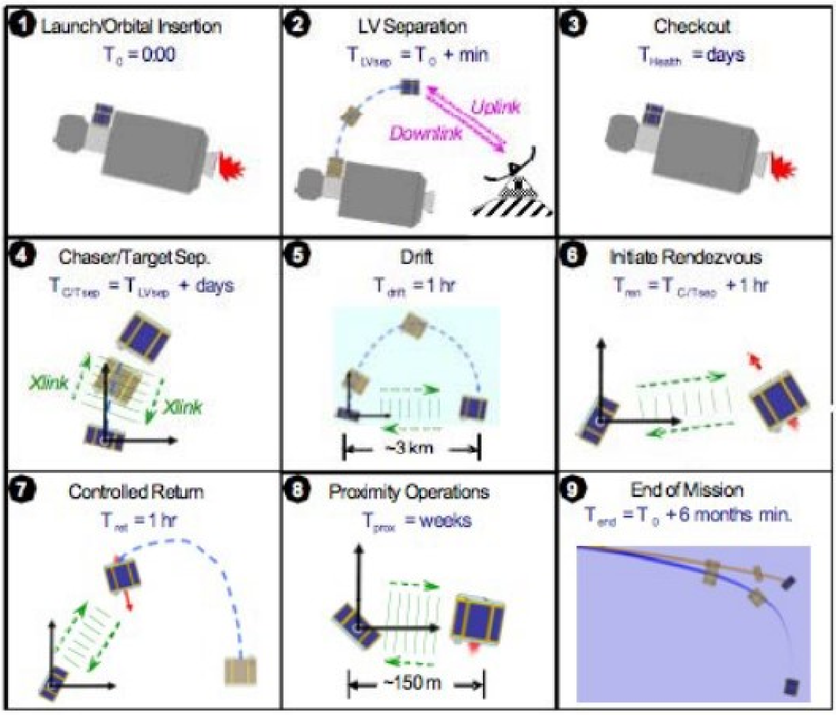

Figure 45: Source: Texas 2-Step Mission, Project Plan, 200

The CONOPS may also encompass provisions for contingency scenarios, anomaly response procedures, and abort contingencies to address unexpected events during launch or mission operations. It serves as a comprehensive guide for ensuring a safe and successful launch, while promoting effective collaboration among mission planners, operators, and stakeholders throughout the launch campaign.

12 Risks and Viability

12.1 Risk Management System

The risk management system is a systematic approach used to identify, assess and mitigate potential risks in the development of a launch vehicle design program. It involves the identification of various risks that could impact the success of the program and the implementation of strategies to minimize or eliminate those risks.

A strong risk management system holds significant importance as it enables the identification and resolution of potential issues before they escalate into major failures. By methodically evaluating risks, the design program gains valuable insights to make well-informed decisions and allocate the necessary resources to effectively mitigate or manage those risks. This systematic approach ensures that potential challenges are identified early on and addressed on time, ensuring the success and progress of the project.

The risk management systems follows a closed loop process which need to be executed all along the project design. Early risks identification brings the ability to apply the lessons learnt to potential further risks. The phases of this system are represented in figure 46.

Figure 46: Risk Management Cirlce

Risks can be classified based on their probability and impact, which helps visualizing their magnitude and impact to prioritize the most important ones. The classification typically involves assessing the likelihood of a risk event occurring and the potential impact it could have on the project or organization.

The classification of risks based on what is the probability on it happening is the following:

- Rare risks: likelihood under 20%.
- Unlikely risks: likelihood between 20% and 40%.
- Moderate risks: likelihood between 40% and 60%.
- Likely risks: likelihood between 60% and 80%.
- Almost certain risks: likelihood between 80% and 100%.

The classification of risks based on how severe would be the outcome if the risk occurred is the following:

- Insignificant: risks with minimal potential to cause significant disruptions.
- Minor: risk which might cause noticeable impact if they occur.
- Significant: risks which have a notable impact on project objectives. While the consequences are not severe, they can affect timelines, resources or deliverables.
- Mayor: risks with relatively high impact on project development. Mitigation efforts are necessary to manage these risks effectively
- Critical: risks with high potential to cause significant consequences in the project. They are the most important to prioritize to avoid catastrophic results.

The previous classification leads to a risk assessment matrix, which relates the potential risks according to impact and probability and is a useful tool to prioritize risks. In any project, the interest is to have the risks have to be in the region where probability and impact are low (usually represented in green). To reach this, it's necessary to conduct mitigation strategies and measures to avoid, transfer, mitigate or accept these.

	Insignificant 1	Minor 2	Significant 3	Major 4	Severe 5
5 Almost Certain	Medium 5	High 10	Very high 15	Extreme 20	Extreme 25
4 Likely	Medium 4	Medium 8	High 12	Very high 16	Extreme 20
3 Moderate	Low 3	Medium 6	Medium 9	High 12	Very high 15
2 Unlikely	Very low 2	Low 4	Medium 6	Medium 8	High 10
1 Rare	Very low 1	Very low 2	Low 3	Medium 4	Medium 5

Figure 47: Risk 5x5 Assessment Matrix

12.2 Risk Analysis

12.2.1 Global Mission Risks

Some of the most important risks identified in relation with the global project are:

1.1 Schedule delays: Delays in critical activities, such as component manufacturing, testing, or system integration, can impact the project's timeline and potentially result in missed launch windows or contractual obligations.

1.2 Cost overruns: Unexpected cost increases in various project aspects, including material procurement, manufacturing, testing, or operational expenses, can strain the project's budget and financial viability.

1.3 Supply chain disruptions: Disruptions in the supply chain, such as delays in the delivery of critical components or dependencies on limited suppliers, can impact the project's progress and result in schedule delays and cost implications.

1.4 Safety and regulatory compliance: Non-compliance with safety regulations, failure to meet quality standards, or inadequate risk management practices can lead to safety incidents, legal repercussions and potential delays or cancellations of the project.

1.5 Environmental impact: Inadequate environmental assessments or failure to comply with environmental regulations can result in negative environmental impacts, legal penalties and reputational damage.

12.2.2 Structural Risks

Some of the most important structural risks identified are:

2.1 Structural failure: structural components or systems failing to withstand the operational loads and stresses during launch or in space.

2.2 Material degradation: materials used in the vehicle's structure can experience degradation over time due to exposure to extreme temperatures, vibrations, radiation or other environmental factors.

2.3 Manufacturing defects: errors or defects in the fabrication of structural components, such as improper assembly, incorrect dimensions or material inconsistencies.

2.4 Propellant leakage: leakage or seepage of propellant during storage, handling or operation, leading to potential safety hazards, equipment damage or mission failure.

2.5 Propellant storage and handling: the proper storage, handling and transportation of propellants, including potential accidents, spills or mishandling that can result in safety hazards, environmental damage or property loss.

2.6 Incompatibility between the launch vehicle and the launch structure.

12.2.3 Propulsion Systems Risks

Some of the most important propulsion risk among others are:

3.1 Engine Failure: propulsion system failure, including issues with ignition, combustion instability or component malfunction, leading to loss of thrust and potential mission failure.

3.2 Thrust vector control failure: failure or malfunction of the thrust vector control system, resulting in the inability to control or steer the vehicle properly, affecting trajectory, stability and mission success.

3.3 Fuel or Oxidizer Contamination due to impurities, foreign particles or inadequate storage conditions, potentially leading to engine damage, combustion issues or reduced performance.

3.4 Inappropriate mixture ratio: deviations in the ratio of fuel and oxidizer during the propellant mixing process or combustion can lead to suboptimal or unstable engine performance. Variations in the mixture ratio can affect thrust, specific impulse, combustion efficiency and overall engine stability.

3.5 Insufficient fuel or oxidizer reserves: inadequate fuel or oxidizer reserves for the duration of the mission, potentially leading to premature engine shutdown or an inability to accomplish mission objectives.

12.2.4 Navigation and Control Risks

Some of the most important risks related to the GNC systems are:

4.1 GNC system failure: malfunction of the guidance system, including sensors, navigation algorithms or control mechanisms, leading to inaccuracies in trajectory, navigation errors or loss of control.

4.2 Communication loss between the launch vehicle and ground control, impacting real-time telemetry data transmission, remote control commands or system monitoring during critical mission phases.

4.3 Navigation inaccuracy: The risk of insufficient precision in navigation, resulting in deviations from the intended trajectory, misalignment with orbital targets, or the inability to achieve accurate positioning for payload deployment or mission objectives.

12.2.5 Aerodynamics and Loads Risks

Among aerodynamic risks, some of the most importan are:

5.1 Aerodynamic instability: unstable flight conditions due to inadequate aerodynamic design result in unpredictable vehicle behavior and potential loss of control.

5.2 Aerodynamic loads exceeding design limits during launch, max Q or reentry, leading to structural damage, performance degradation or mission failure.

5.3 Structural failure due to dynamic loads: structural failure caused by vibrations, oscillations, or resonance effects, which can lead to component or system malfunctions.

5.4 Thermal Effects: thermal stresses and deformations caused by aerodynamic heating during ascent or reentry, potentially leading to structural integrity issues, material degradation or thermal management challenges.

12.2.6 Project Risk Index Chart

The previous mentioned risks can be represented in a project risk index chart to visualize the probability of each risk, the impact on the project and the PRI value on a scale from 1 to 10.

Risk ID	Subsystem	Probability (%)	Impact	PRI
1.1	Global Mission	30	Critical	15
1.2	Global Mission	35	Critical	17
1.3	Global Mission	20	High	11
1.4	Global Mission	10	High	10
1.5	Global Mission	9	High	9
2.1	Structural	10	Critical	10
2.2	Structural	10	High	10
2.3	Structural	5	High	11
2.4	Structural	20	Critical	13
2.5	Structural	15	High	11
2.6	Structural	10	High	12
3.1	Propulsion	15	Critical	15
3.2	Propulsion	15	Critical	14
3.3	Propulsion	30	High	10
3.4	Propulsion	10	High	8
3.5	Propulsion	5	Critical	9
4.1	GNC System	8	High	9
4.2	GNC System	30	Critical	14
4.3	GNC System	25	Medium	12
5.1	Aerodynamics and loads	8	High	8
5.2	Aerodynamics and loads	20	High	10
5.3	Aerodynamics and loads	15	Critical	15
5.4	Aerodynamics and loads	10	Medium	6

Table 9: Project Risk Index Chart

12.3 Risk Mitigation Strategy

To effectively mitigate project risks, it is crucial to prioritize those with the highest risk impacts, considering their significance level. The appropriate mitigation strategy or action plan depends on both the risk impact and likelihood. Each risk may have unique characteristics, necessitating a tailored mitigation plan.

There are four general strategies for risk mitigation:

- **Avoid**: This strategy involves eliminating the risk source when the potential rewards do not justify the expected costs. It is a viable approach when taking on a particular risk is not economically feasible.

- **Transfer**: Risk transfer involves sharing or reducing a risk by partially shifting its responsibility to another business partner. This strategy can help distribute the potential impact and decrease the organization's overall risk exposure.

- **Accept**: Informed decision-making is crucial when choosing to accept both the impact and likelihood of risk events. This strategy acknowledges the risks but consciously decides not to take specific mitigation measures.

- **Mitigate**: Mitigation aims to limit the impact of risks or reduce the probability of their occurrence. This strategy involves implementing measures and controls to minimize the adverse effects of identified risks.

In many cases, a combination of these strategies may be necessary to develop the most effective mitigation plan.

12.3.1 Analyzed risks mitigation measures

1.1 Schedule Delays:

- Implement effective project management practices, such as creating a detailed schedule with milestones and deadlines.
- Regularly monitor and track project progress to identify potential delays early on.
- Allocate sufficient resources and manpower to critical tasks.
- Develop contingency plans and alternative approaches to mitigate schedule delays.

1.2 Cost Overruns:

- Conduct comprehensive cost estimation and budgeting during the planning phase.
- Implement rigorous cost tracking and monitoring mechanisms.
- Regularly review project finances and make necessary adjustments to stay within budget.

1.3 Supply Chain Disruptions:

- Diversify suppliers and establish relationships with multiple vendors.

- Conduct regular assessments of supplier capabilities, reliability and financial stability.
- Develop backup plans for sourcing critical components or materials.
- Maintain open communication channels with suppliers to address any potential disruptions proactively.

1.4 **Safety and Regulatory Compliance**:

- Establish a robust safety management system.
- Stay updated on regulatory requirements and ensure compliance throughout the project.
- Provide comprehensive training to personnel on safety protocols and procedures.

1.5 **Environmental Impact**:

- Conduct thorough environmental impact assessments and follow applicable environmental regulations.
- Implement sustainable practices and technologies to minimize environmental footprint.

2.1 **Structural Failure**:

- Conduct comprehensive structural analysis and testing during the design phase.
- Implement robust quality control measures during manufacturing and assembly.
- Perform regular inspections and maintenance of structural components.
- Implement redundant and backup systems to mitigate the consequences of structural failures.

2.2 **Material Degradation**:

- Select materials suitable for the operational environment and expected lifespan.
- Implement appropriate material protection and preservation techniques.
- Conduct regular inspections and testing to detect material degradation.
- Develop contingency plans and procedures to replace or repair degraded materials.

2.3 **Manufacturing Defects**:

- Implement rigorous quality control processes during manufacturing and assembly.
- Conduct comprehensive testing and inspections at various stages of production.
- Establish clear standards and specifications for manufacturing processes.
- Continuously monitor and analyze manufacturing data to identify and address potential defects.

2.4 **Propellant Leakage**:

- Implement robust sealing and containment systems for propellant tanks and lines.
- Conduct thorough inspections and testing of propellant systems.
- Use reliable and tested propellant materials and components.
- Develop emergency response plans and procedures to handle propellant leakage

incidents.

2.5 Propellant Storage and Handling:

- Establish proper storage and handling procedures for propellants.
- Train personnel on safe practices for propellant storage, transportation and handling.
- Implement monitoring systems to detect any abnormalities or leaks.
- Regularly inspect and maintain propellant storage facilities and equipment.

2.6 Incompatibility Between Launch Vehicle and Launch Site:

- Conduct thorough compatibility assessments between the launch vehicle and intended launch sites.
- Coordinate with launch site operators to ensure infrastructure and support capabilities align with the vehicle requirements.
- Implement necessary modifications or adaptations to the launch vehicle or launch site as needed.
- Conduct comprehensive tests and simulations to validate compatibility before launch.

3.1 Engine Failure:

- Conduct rigorous testing and validation of engines during the development phase.
- Implement redundant engine systems to provide backup capabilities.
- Regularly monitor engine performance and conduct inspections and maintenance as per manufacturer guidelines.
- Develop emergency shutdown and abort procedures in case of engine failures.

3.2 Thrust Vector Control Failure:

- Implement redundant thrust vector control systems to ensure control authority.
- Conduct thorough testing and calibration of thrust vector control mechanisms.
- Regularly inspect and maintain thrust vector control components.
- Develop contingency plans and procedures to mitigate the consequences of thrust vector control failures.

3.3 Fuel or Oxidizer Contamination:

- Establish strict protocols for fuel and oxidizer handling, storage and transportation.
- Implement reliable filtration and purification systems for fuel and oxidizer.
- Conduct regular inspections and testing to detect and address any contamination issues.
- Develop contingency plans and procedures to respond to fuel or oxidizer contamination incidents.

3.4 Inappropriate Mixture Ratio:

- Conduct thorough propellant mixing and testing to ensure the desired mixture ratio.
- Implement automated mixing and monitoring systems to maintain accurate mixture ratios.
- Perform regular inspections and testing to verify mixture ratio integrity.
- Develop contingency plans and procedures to address mixture ratio deviations.

3.5 Insufficient Fuel or Oxidizer Reserves:

- Conduct fuel and oxidizer consumption analysis and estimation.
- Implement redundant fuel and oxidizer storage systems to provide backup reserves.
- Continuously monitor fuel and oxidizer levels during operations.
- Develop accurate consumption models and mass margins to ensure sufficient reserves.

4.1 GNC System Failure:

- Implement redundant guidance, navigation and control (GNC) systems.
- Conduct rigorous testing and validation of GNC algorithms and hardware.
- Develop contingency plans and procedures for GNC system failures.
- Regularly calibrate and maintain GNC sensors and components.

4.2 Communication Loss:

- Establish redundant communication systems and networks.
- Conduct communication tests and range assessments.
- Implement robust error detection and correction mechanisms in communication protocols.
- Develop contingency procedures to reestablish communication in case of loss.

4.3 Navigation Inaccuracy:

- Implement redundant navigation systems, multiple sensors and algorithms.
- Conduct thorough testing and calibration of navigation systems.
- Continuously monitor and analyze navigation data for accuracy and consistency.
- Develop contingency procedures to address navigation inaccuracies during operations.

4.3 Aerodynamic Instability:

- Conduct extensive aerodynamic analysis and testing during the design phase.
- Implement appropriate stability augmentation systems and control mechanisms.
- Regularly monitor and analyze flight data to detect any signs of instability.
- Develop contingency procedures to address aerodynamic instability issues.

5.1 Aerodynamic Loads Exceeding Design Limits:

- Conduct comprehensive aerodynamic load analysis during the design phase.

- Implement robust structural design and reinforcement to resist anticipated loads.
- Regularly monitor and analyze flight data to ensure loads are within design limits.
- Develop contingency plans and procedures to address excessive aerodynamic loads.

5.2 Structural Failure Due to Dynamic Loads:

- Conduct thorough dynamic load analysis during the design phase.
- Implement robust structural design and reinforcement to resist anticipated dynamic loads.
- Regularly monitor and analyze structural response during operations.
- Develop contingency plans and procedures to address structural failures caused by dynamic loads.

5.3 Thermal Effects:

- Conduct thermal analysis during the design phase.
- Implement proper insulation and thermal protection systems.
- Regularly monitor and analyze thermal conditions during operations.
- Develop contingency plans and procedures to mitigate adverse thermal effects.

12.4 Opportunities

In a space launch vehicle project, there are several potential opportunities that can arise throughout its development. These opportunities can have a significant impact on the success and advancement of the project. Below is a detailed analysis of the possible opportunities in such a project:

- **Technological Advancements:** The rapidly evolving field of aerospace technology offers numerous opportunities for innovation and improvement. Advancements in propulsion systems, materials, and manufacturing techniques can lead to significant enhancements in performance, efficiency, and cost-effectiveness of the launch vehicle.

- **Collaboration and Partnerships:** Collaborating with other organizations, both within the aerospace industry and outside, can provide opportunities for knowledge sharing, resource pooling, and leveraging expertise. Partnering with research institutions, universities, and industry leaders can lead to breakthroughs in design, engineering, and operational capabilities.

- **Regulatory Support:** Favorable regulatory environments and government support can create opportunities for streamlined processes, funding assistance, and expedited approvals. Engaging with regulatory bodies and ensuring compliance with applicable regulations can open doors to smoother operations and reduced barriers.

- **Market Demand:** Identifying and capitalizing on market demand for satellite launches and space exploration can present significant opportunities. Conducting market assessments and understanding customer needs can help tailor the launch vehicle design and services to cater to specific market segments and gain a competitive advantage.

- **Cost Reduction Strategies:** Developing innovative cost reduction strategies, such as reusable launch systems or optimized manufacturing processes, can lead to increased affordability and market competitiveness. Exploring cost-saving measures in the design, production, and operational phases of the project can unlock opportunities for cost-effective solutions.

- **New Business Models:** Introducing disruptive business models, such as rideshare services or satellite constellation deployment, can create new revenue streams and expand market reach. Exploring alternative ways of monetizing the launch vehicle's capabilities can open doors to diverse business opportunities.

- **Talent Acquisition and Development:** Attracting and retaining skilled professionals in the aerospace industry is crucial for project success. Establishing comprehensive talent acquisition and development strategies, including internships, training programs and partnerships with educational institutions, can ensure access to a talented workforce and foster innovation.

- **Intellectual Property:** Developing proprietary technologies and securing intellectual property rights can provide opportunities for licensing and royalties. Protecting innovative ideas and solutions can give the project a competitive edge and generate additional revenue streams.

- **Global Market Expansion:** Exploring international markets and establishing partnerships with international clients and space agencies can expand the project's reach and create opportunities for global operations and collaborations.

- **Environmental Sustainability:** Incorporating environmentally sustainable practices into the design, manufacturing and operational phases of the project can attract environmentally conscious customers and stakeholders. Embracing clean energy solutions, waste reduction strategies and carbon footprint mitigation can lead to favorable market positioning and partnerships.

12.5 Reliability Requirements

1. Reliability of the propulsion system and turbines to ensure that the vehicle can achieve the speed and altitude necessary for the desired orbit.

2. Reliability of navigation and control systems to ensure that the vehicle can be accurately directed to its destination.

3. Reliability of communication systems to ensure that the vehicle can maintain communication with operators on the ground and receive data and commands reliably.

4. Resistance to vibrations, accelerations and high temperatures during launch and re-entry to ensure that the vehicle and its payload are not damaged.

5. Reliability of the payload restraint and release systems to ensure that the vehicle is detached safely and at the right time.

6. Overall Reliability: The overall reliability of the launch system will be greater than 95% at the end of its useful life.

7. Failure propagation: The failure of one component must not cause failure or damage to another component or subsystem.

13 Costs Analysis

The purpose of building launch vehicles is to achieve high performance with the lowest possible weight and cost.

The high cost of space transportation is a major obstacle to space exploration and commercialization. To reduce these costs, numerous studies have been carried out, such as those conducted by NASA and ESA. In addition, various tools have been developed to estimate production costs, including Koelle's TRANSCOST and other models such as Unmanned Space Vehicle Cost Model (USCM), SEER-H, Small Satellite Cost Model (SSCM), Aces de 4cost, TruePlanner from PRICE Systems Solutions, NAFCOM from NASA Air Force and LVCM.

Of these tools, TRANSCOST is the only one that is public, while the others are government or commercial payment tools.

Since 1980 technological miniaturization has allowed the development and popularization of small satellites, which has led to the emergence of manufacturers that cover applications in the civil field of Earth observation and monitoring, as well as in the search for energy resources.

In recent years, there has been an increase in the number of companies interested in launching small satellites into low orbits, which has led to competition in the cost of these missions. One of the main objectives of these companies is to achieve the lowest possible cost per flight (CPF).

In our case, we have mainly used the tools of TRANSCOST and SOLSTICE for cost development.

In the space sector, we need an active variable to determine the costs of our technical systems and the program, so that we can update them as we undergo changes in the characteristics of the system, schedule or in the management of our project.

It is very important to have a very precise system because the viability of the project is usually measured mainly based on its cost, so that, if the cost is too high, it may not have financing to develop it. In addition, we must be careful with the cost estimate, in case of underestimating the project, we could run out of budget and not be able to finish it.

Companies are currently competing to optimize this design process and project life-cycle. There are three methodologies for cost estimation in the aerospace industry:

1. Accumulation of energy.
2. Analogy cost estimation.
3. Parametric cost estimation.

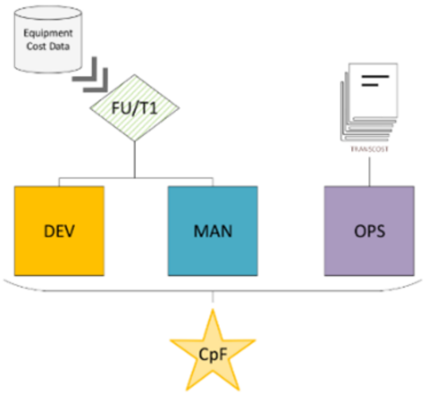

Figure 48: Cost per Flight estimating process.

The estimation of parametric costs is the one we are going to use, where a parameter relates a variable (mass is our choice), with the final cost of the component proportionally through linear regressions. This system is more complex due to the reduction of mass of some components where we should add new technologies and take this into account.

This process will help us to determine the development and manufacturing costs based on the concept of Flight Unit, which is also usually called First Theorical Unit (FU/T1)

13.1 FU/T1 Cost Estimate

These are the parts of a liquid launch vehicle that are identified:

Unit	Equipment	Part	Part ID
Stage	Pressurizant Tank		S-PT-
	Fuel Tank		S-FT-
	Oxidizer Tank		S-OT-
	Stage Structure	Thrust Cone	S-SS-TRC
		Skirt	S-SS-SKI
		Thermal Control	S-SS-THM
	Engine(s)		S-EN-
	Thrust Vector Control		S-TV-
	Propellant	Fuel	S-PR-FUE
		Oxidizer	S-PR-OXI
		Pressurizant	S-PR-PRE
	Pressurization System		S-PS-
	Pipes & Valves	Pipes	S-PV-PIP
		Valves	S-PV-VAL
	Stage Harness		S-SH-
Interstage	Interstage Structure		I-IS-
Payload	Payload Adapter		P-PA-
	Payload Fairing		P-PF-
Avionics	Avionics	Comms	A-AV-COM
		Power	A-AV-PWR
		Data Handling	A-AV-DHL
		GNC	A-AV-GNC
		Avionics Harness	A-AV-HNS
Attitude Control	Attitude Control Module		C-AC-

Figure 49: Liquid Launch Vehicle Breakdown

For the estimation as we have already explained before, a linear regression will be used on the cost of the equipment system over the years, so that all will have the following relationship:

$$C = aM^b \tag{1}$$

The values of a and b have been estimated using historical data from previous releases, standard logarithmic errors and cost modeling accuracy collected at the SOLSTICE:

					Regression		
				a value	b value	Log. SE	RSE/CMA
Stage							
	Solid Casing		S-SC-	90.72782	0.44422	0.128	12.9%
	Pressurizant Tank		S-PT-	19.99465	0.71253	0.271	27.6%
	Fuel Tank		S-FT-	19.99465	0.71253	0.271	27.6%
	Oxidizer Tank		S-OT-	19.99465	0.71253	0.271	27.6%
	Stage Structure						
		Thrust Cone	S-SS-TRC	2.79930	0.91199	0.125	12.6%
		Skirt	S-SS-SKI	2.79930	0.91199	0.125	12.6%
		Thermal Control	S-SS-THM	2.79930	0.91199	0.125	12.6%
	Engine(s)		S-EN-	31.48271	0.78811	0.347	35.8%
	Thrust Vector Control		S-TV-	33.90978	0.60977	0.136	13.7%
	Pressurization System		S-PS-	11.50618	1.06948	0.471	49.8%
	Pipes & Valves						
		Pipes	S-PV-PIP	8.95877	0.68815	0.333	34.3%
		Valves	S-PV-VAL	8.95877	0.68815	0.333	34.3%
	Stage Harness		S-SH-	27.45211	0.44623	0.339	34.9%
Payload							
	Payload Adapter		P-PA-	124.86209	0.31031	0.129	13.0%
	Payload Fairing		P-PF-	4.09558	0.96587	0.091	9.2%
Avionics							
	Avionics						
		Comms	A-AV-COM		*One data point only*		
		Power	A-AV-PWR	56.13918	0.66916	*Two data points only*	
		Data Handling	A-AV-DHL	141.82428	0.79249	0.162	16.3%
		GNC	A-AV-GNC	69.05491	0.82458	0.235	23.8%
		Avionics Harness	A-AV-HNS	27.45211	0.44623	0.339	34.9%
Attitude Control							
	Attitude Control Module		C-AC-	44.04074	1.06207	0.761	88.6%
Interstage							
	Interstage Structure		I-IS-	6.70369	0.68041	0.191	19.3%

Figure 50: Systems Elements Cost Estimating Relationships

To verify that the result is correct we must make sure that the standard error of a and b is less than 5%. Alternatively, we can use NAFCOM data, where the values of a are adjusted and the values of b are constant

	b-values		Variance of b-values	
Subsystem/Group	DDT&E	Flight Unit	DDT&E	Flight Unit
Antenna Subsystem	0.85	0.80	0.0338	0.0161
Aerospace Support Equipment	0.55	0.70	0.0698	0.0982
Attitude Control/ Guidance and Navigation Subsystem	0.75	0.85	0.0120	0.0059
Avionics Group	0.90	0.80	0.0055	0.0044
Communications and Command and Data Handling Group	0.85	0.80	0.0053	0.0003
Communications Subsystem	0.85	0.80	0.0208	0.0141
Crew Accommodations Subsystem	0.55	0.70	0.0826	0.0565
Data Management Subsystem	0.85	0.80	0.0048	0.0025
Environmental Control and Life Support Subsystem	0.50	0.80	0.0048	0.2662
Electrical Power and Distribution Group	0.65	0.75	0.0064	0.0043
Electrical Power Subsystem	0.65	0.75	0.0878	0.0161
Instrumentation Display and Control Subsystem	0.85	0.80	0.1009	0.0665
Launch and Landing Safety	0.55	0.70	0.0960	0.0371
Liquid Rocket Engines Subsystem	0.30	0.50	0.1234	0.0483
Mechanisms Subsystem	0.55	0.70	0.0050	0.0167
Miscellaneous	0.50	0.70	0.0686	0.0784
Power Distribution and Control Subsystem	0.65	0.75	0.0106	0.0053
Propulsion Subsystem	0.55	0.60	0.2656	0.1730
Range Safety Subsystem	0.65	0.75	-	-
Reaction Control Subsystem	0.55	0.60	0.0144	0.0092
Separation Subsystem	0.50	0.85	-	-
Solid/ Kick Motor Subsystem	0.50	0.30	0.0302	0.0105
Structures Subsystem	0.55	0.70	0.0064	0.0038
Structures/Mechanical Group	0.55	0.70	0.0029	0.0023
Thermal Control Subsystem	0.50	0.80	0.0055	0.0045
Thrust Vector Control Subsystem	0.50	0.60	0.7981	0.0234

Figure 51: b-values for NAFCOM regression curves

Knowing all this, the cost of the flight unit concept for the launch vehicle can be calculated:

Unit	Equipment	Part	Part ID	Stage 1 [Kg]	Stage 2 [Kg]	Stage 3 [Kg]	Cost [K€]
Stage							
	Pressurizant Tank		S-PT				0
	Fuel Tank		S-FT	261,4416	99,0217	59,715	1479,752094
	Oxidizer Tank		S-OT	48,3879	18,327	11,0521	444,7973733
	Stage Structure	Trhust Cone	S-SS-TRIC	140,136	46,712	15,5706	355,0804501
		Skirt	S-SS-SKI	73,5586	73,5586	73,5586	384,1760632
		Thermal Control	S-SS-THM				0
	Engine(s)		S-EN	740,635	249,6735	68,3171	7619,067752
	Thrust Vector Control		S-TV				0
	Pressurization System		S-PS				0
	Pipes and Valves	Pipes	S-PV-PIP				0
		Valves	S-PV-VAL				0
	Stage Harness		S-SH				0
Interstage							
	Interstage Structure		I-IS	63,5608	63,5608	44,4926	222,1909058
Payload							
	Payload Adapter		P-PA			57,55	439,1271264
	Payload Fairing		P-PF				0
Avionics							
	Avionics	Comms	A-AV-COM	65			574,6621338
		Power	A-AV-PWR	140			1532,389266
		Data Handling	A-AV-DHL	43			2794,203317
		GNC	A-AV-GNC	43			1535,041011
		Avionics Harness	A-AV-HNS	32			128,8898758
Attitude Control							
	Attitude Control Module		C-AC	40			2214,901731
						Cost	**19724,2791**

The price per flight unit of the launch vehicle is **19,724 million.**

13.2 Development, Production and Ground and Flight Operations Cost

For this part, we will use the notation of TRANSCOST, where they work through "Work – Year effort" as the main value. This way of evaluating the cost can be converted to US dollar or Euro using the following table:

YEAR	USA*) US$	Europe**) Euro(ECU /AU)	Japan***) MillionYen
1960	26 000	18 000	
1961	27 000	18 900	
1962	28 000	20 000	
1963	29 000	21 000	
1964	30 000	22 000	
1965	31 000	23 200	
1966	32 300	24 400	
1967	33 200	25 700	
1968	34 300	27 400	
1969	36 000	29 100	
1970	38 000	31 000	
1971	40 000	33 050	
1972	44 000	35 900	
1973	50 000	38 700	
1974	55 000	43 600	
1975	59 500	50 000	
1976	66 000	55 100	
1977	72 000	60 500	
1978	79 700	65 150	
1979	86 300#	71 800	
1980	92 200	79 600	
1981	98 770	86 700	
1982	105 300	92 400	
1983	113 000	98 300	
1984	120 800	104 300	14.6
1985	127 400	108 900	15.2
1986	132 400	114 350	15.8
1987	137 700	120 000	16.4
1988	143 500	126 000	17.1
1989	150 000	133 000	17.6
1990	156 200	139 650	18.1
1991	162 500	145 900	18.6
1992	168 200	151 800	19.0
1993	172 900	156 800	19.5
1994	177 200	160 800	20.0
1995	182 000	167 300	20.5
1996	186 900	172 500	21.0
1997	191 600##	177 650	21.5
1998	197 300	181 900	22.0
1999	203 000	186 300	22.6
2000	208 700	190 750	23.2
2001	214 500	195 900	23.8
2002	220 500	201 200	24.4
2003	226 400	207 000	25.0
2004	232 100	212 200	25,6
2005	240 000	217 500	26,3
2006	246 000	222 300	26,9
2007 estimate	252 000	228 500	27,5

Figure 52: Work Year Transcost 7.2

In this table not all the data because it is not the latest version, but the estimated conversion for the year 2015 is 300800 Euros for each year, this will be the data used.

13.2.1 Development costs

The estimation of these costs is very complicated due to the subjective influence when developing the program. There are some criteria to consider such as: the degree of cost engineering, the technology status, the realism of dry mass estimates or margin strategy, testing and the number of test units.

The main criteria influencing the development cost for a launch vehicle are:

1. Vehicle launch mass and size.

2. Number and type of stages.

3. Technology readiness / scope of existing subsystems or components
4. Type and number of engines.
5. Reliability and safety requirements.
6. Verification and test strategy.
7. Number off light units and flight tests.
8. Company and team experience.
9. Program organization and management procedures.
10. Program Budget planning and schedule/delays.
11. Technical changes required or ordered by customer
12. Contract conditions

Development costs include all detailed design activities and implementation, hardware verification, including all test models built.

In our case, the engines have not been considered since we use an already created model and its development is not necessary.

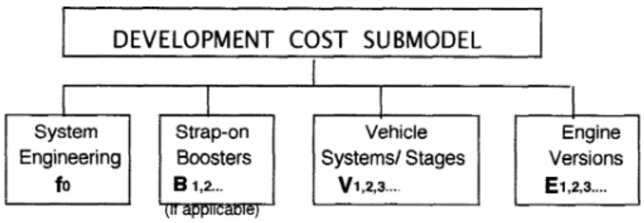

Figure 53: Development Cost Sub model Elements

To calculate the cost, we need to use the following formula:

$$C_D = f_0(\sum H_b + \sum H_v + \sum H_e) f_6 f_7 f_8 \tag{2}$$

Being the HB the development costs of Boosters, H V those of the vehicle and HE of the engines.

In addition, a system engineering factor f0 appears, which has a value of 1.04N where N is the number of stages.

And finally, there are programmatic cost impact factors f 6, f7, f8 that will be explained later.

Next, to calculate the values of H, we will need to know the Development Cost Factors

1. Development Standard Factor f1:

   It is influenced by the relative status of the Project in comparison to the state of the art, or the relation to other similar projects.

   The following numerical values could apply:

f_1	First generation system, new concept approach, involving new techniques and new technologies	f_1 = 1.3 to 1.4
	New design with some new technical and/ or operational features	f_1 = 1.1 to 1.2
	Standard projects, state-of -the-art (similar systems are already in operation)	f_1 = 0.9 to 1.1
	Design modification of existing systems,	f_1 = 0.7 to 0.9
	Minor variation of existing projects	f_1 = 0.4 to 0.6

Figure 54: Development Factor f1

2. Technical Quality Factor f2:

   In contrast to factors f1 and f3, this factor is different and characteristic for each technical system and it may be the most important cost driver. It is based either on the relative net mass fraction, the performance or another important cost factor.

f_2 = Specific for each system (or element type), defined by an inherent technical criterion

Figure 55: Development Factor f2

3. Team Experience Factor f3:

   Clearly an inexperienced new team will need a higher development effort.

   Some experience based values for the f3 factor are as follows:

f_3	New team, no relevant direct company experience	f_3 = 1.3 to 1.4
	Partially new project activities for the team	f_3 = 1.1 to 1.2
	Company / industry team with some related experience	f_3 = 1.0
	Team has performed development of similar projects	f_3 = 0.8 to 0.9
	Team has superiour experience with this type of project	f_3 = 0.7 to 0.8

Figure 56: Development Factor f3

Propulsion Development CERs

As mentioned above, for our launch vehicle we will use existing and developed engines, saving us a cost about EXP(8) orders of magnitude in euros, that will allow us to cut the budget to carry out the project.

Vehicle systems development CERs

1. Liquid propulsion modules

   This group of space systems comprises both propulsion modules and bipropellant systems. Propulsion modules are propulsion systems with their own basic structure, but no external structure, no own power supply and no own intelligence equipment like telemetry or guidance and control, these functions are taken over by a separate module or by another system.

$$H_v p = 14.2 M^{0.577} f_1 f_3 \tag{3}$$

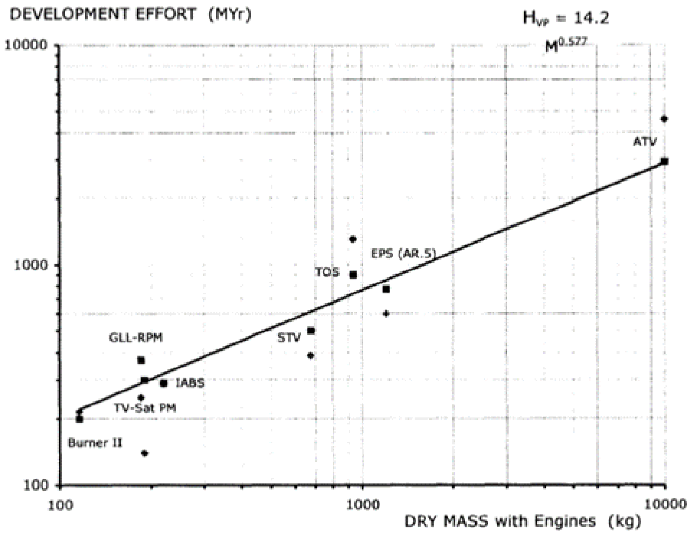

Figure 57: Liquid propulsion modules

2. Expendable ballistic stages and transfer vehicles

   This group comprises launch vehicles used at launch or higher stages. The mass of the engines is not added for this calculation because existing engines are often used. In addition, there are single and multi-engine vehicles, which is a feature that influences the development cost of the complete vehicle.

$$H_V E = 100 M^{0.555} f_1 f_2 f_3 \tag{4}$$

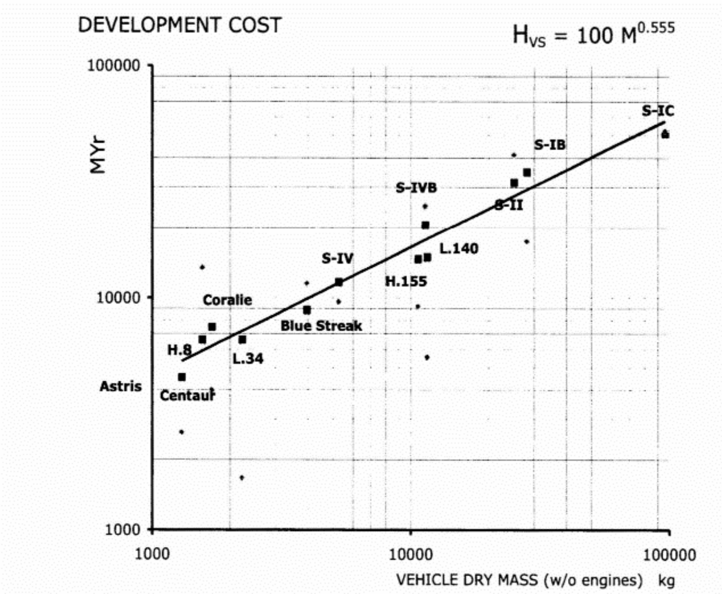

Figure 58: Basic CER for expendable Stage vehicles

In this case, for the calculation of the f2 factor with cryogenic engines is

$$f_2 = \frac{K_{ref}}{K_{eff}} \tag{5}$$

Kref average reference net mass fraction is defined in the next figure

Keef effective net mass fraction of the vehicle concept or the existing vehicle

Cost impact of development program preparation, organization and Schedule

1. Program organization

   To start the organisation of a project of this technical magnitude requires a clear contractor/subcontractor relationship with well-defined responsibilities, that define the following factor:

   $$f_7 = n^2 \tag{6}$$

   Where n is the number of participants from parallel organizations

   Sometimes the contract conditions can have an impact on the project cost. There are the following major contract types:

   (a) Cost plus a Percentage Fee (5 to 7

   (b) Cost plus Fixed Fee

   (c) Cost plus Award Fee

(d) Firm Fixed Price

2. The optimum development Schedule

   Next figure shows development cost or Budget profiles for 6, 7, 8 and 9 years program from the start. The growth in total cost is because inflation and efficient work procedures.

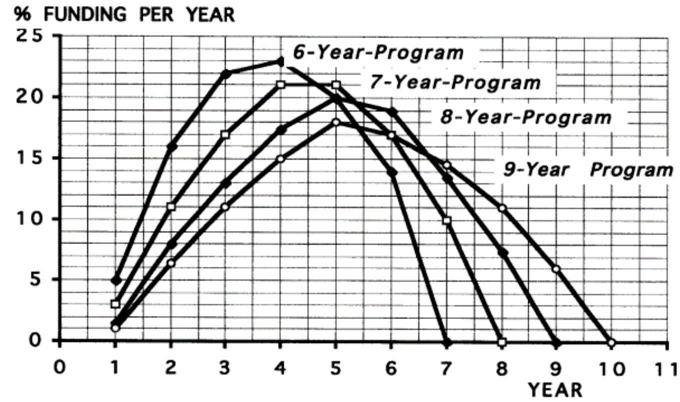

Figure 59: Optimal profiles for development programs

According to historical cases, vehicles requiring 8 to 12 billion euros should be planed for a development period f 5 to 6 years, 15 to 25 billion euros for a period f 6 to 7 years and vehicles requiring 35 to 50 billion euros for period of 7 to 8 years .

If the optimal schedule is delayed or extended by 20%, the cost will increase by 10-15%, and if the delay is 40%, the cost will increase by 30-35%

Accelerated program will cause a higher cost due to overtime of shift activities that have to be paid for, and some additional parallel work

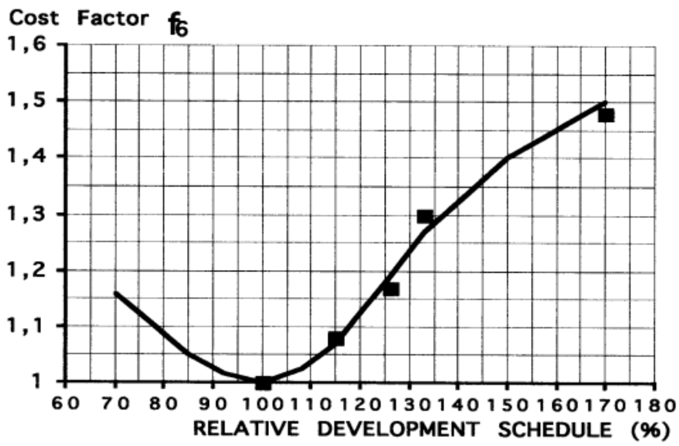

Figure 60: Development factor f6

Impact of productivity in different countries/regions

The productivity levels vary across regions of the world, potentially impacting the expenses associated with completing a particular task.

The factor f8 improves the consistency of international benchmark data.

TABLE 1-IV: **Productivity Model for Different Countries/Regions (f_8) 1980-99**

COUNTRY	(1) Effective Working Hours per year		(2) Relative Education	(3) Relative Dedication	Relative Productivity (Product of items1-3)	MYr Correction Factor f_8
USA (REF.)	1847,	$h^{0.7}$ =1.94	1.00	1.00	194 = 1.00	**1.00 (Ref.)**
Europe (ESA)	1583	174	1.20	1.08	225 = 1.16	**0.86**
France	1611	176	1.30	1.10	252 = 1,30	**0,77**
	1561*	172	1.30	1.10	246 = 1.27	**0.79**
Germany	1568	172	1.30	1.13	252 = 1,30	**0,77**
	1674*	180,5	1.30	1.13	265 = 1.37	**0,73**
Japan	2052	208	1.13	1.18	278 = 1.43	**0.70**
Russia	est. 1600	175	0.75	0.70	92 = 0.47	**2.11**
China	1958	201	0.85	0.85	145 = 0.75	**1.34**

Figure 61: Development factor f8

Calculation Development costs

First, we must decide what the value of our factors will be:

1. f1: Standard Project = 1.1
2. f2: As proposed in TRANSCOST the number of tests is 500 and the reference value have been chosen is unitary
3. f3: Due to our experience, f3 will be the maximum value f3 = 1.4
4. f6: In order not to add extra costs we assume that the project is carried out in the estimated time
5. f7: Base on other launch vehicles, people working on the project is around 500.
6. f8: We have an European launcher so f8 = 0.86

CERs	Name	Mass [Kg]	Cost [WYr]
H_B	Boosters		
H_{EL}	Liquid propellant rocket motors		
H_{VP}	Liquid propellant propulsión system	2793.0083	2129.0837
H_{VE}	Expendable ballistic stages and transfer vehicles	2338.0083	11408.5455

The development cost are 14633.1499 WYr, that passing it to euros through the conversion of the TRANSCOST gives us **13652.4276 million euros**

13.2.2 Production costs

Production cost include material cost, processing and manufacturing cost, assembly and verification and acceptance testing costs as well as engineering support and quality assurance costs, as shown in the following figure:

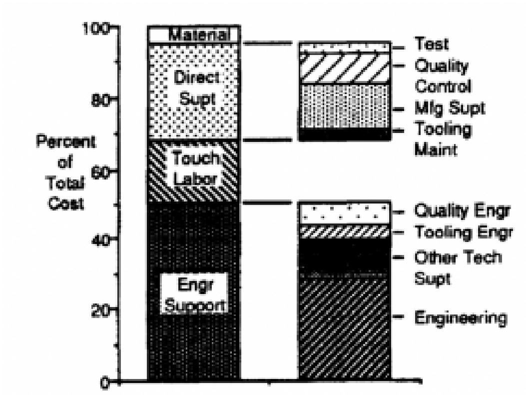

Figure 62: Production cost breakdown

Learning factor/production quantify impact

The Learning Factor takes into account the reduction of the effort required for manufacturing by comparing the first unit with subsequent units.

Thus, for a p-value = 0.8, the cost of the following units is reduced by 80%. Using these estimates, we can define the last factor f4 based on the units built:

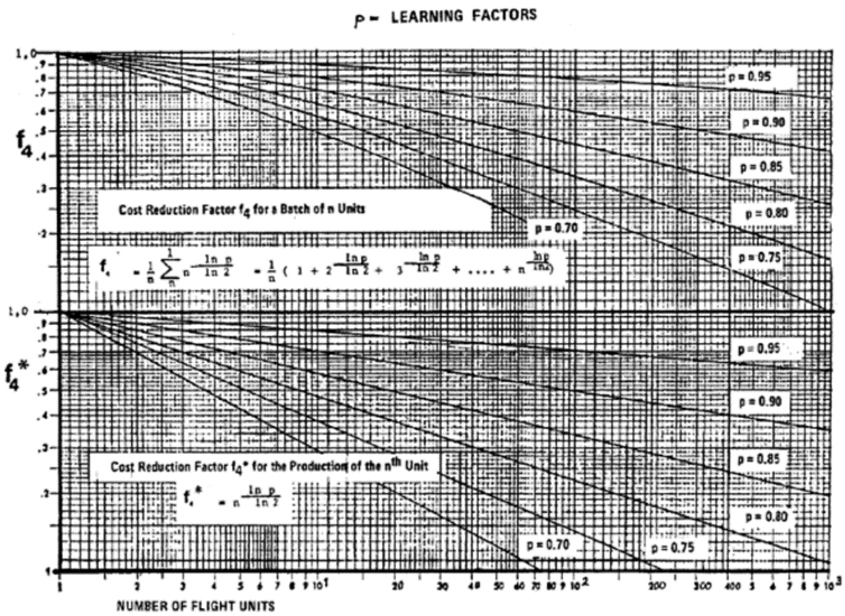

Figure 63: Development factor f4 vs number of units built

The values of the learning factor applicable to space systems are between 0.80 and 1.0. The specific value depends on the size of the unit (mass) as well as the annual production rate.

In our case we will asume that f4 = 1.

Complete launch vehicles production costs

The basic recurring costs ERC are structured as follows

$$F = naM^x f_4 \tag{7}$$

n: number of units to be built.

X: specific cost/mass sensitivity value for each hardware group.

M: reference mass.

f4: cost reduction factor for series production

The total industrial fabrication, assembly and tests costo f a complete launch vehicle are defined:

$$C_F = f_0^N (\sum_1^n F_S + \sum_1^n F_E) f_8 \tag{8}$$

N is the number of stages or system elements n is the number of identical units per element

EXPENDABLE launch Vehicles have the advantage of a continuous production line with cost reductions vs time due to the learning effect.

In case of REUSABLE launch Vehicles the situation is different, there are only two options:

1. All vehicles and spares required for the planned operational period are produced in an optimum time period and put into storage until they are needed

2. A continuous production activity is maintained wich means the scheduled introduction of new vehicles into the program. This however, requires either a continuous growth of operational capability and a limited lifetime for each vehicle

Engine Production CERs

Liquid-Propellant Rocket Engines

The first cost evaluation of rocket engines was carried out to find the best correlation cost, whose final conclusion was engine dry mass.

In the other hand, a study was carried out where it could be seen that low chamber pressure does not reduce engine cost but above 120 does lead to an increase.

As function of pressure and dry mass, depending on the type of engine, the engine production costs CERs have been derived as follow:

Cryo Engines with LH2:

$$F_{EL(c)} = 5.16nM^{0.45}f_4 \tag{9}$$

Engines with storage propellants:

$$F_{EL(s)} = 1.9nM^{0.535}f_4 \tag{10}$$

Mono-propellant Engines:

$$F_{EL(m)} = 1.13nM^{0.535}f_4 \tag{11}$$

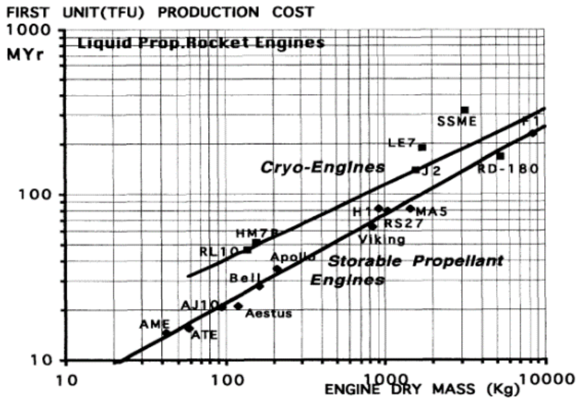

Figure 64: Production for Liquid Propellant Engines

Launch Vehicle Systems production CERs

Propulsion Modules

Propulsion Modules are defined as in the case of development, but through linear regressions of previous models, the following formula has been created that indicates only the cost of production:

$$F_{VP} = 4.65nM^{0.49}f_4 \tag{12}$$

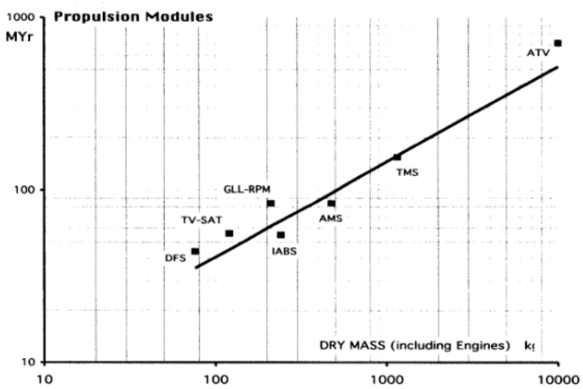

Figure 65: Production for Liquid propulsion modules

Ballistic vehicles/stages

As in the previous case, by referencing 16 existing projects, a linear regression has been created that defines the cost of production for ballistic systems:

$$0.83nM^{0.65}f_4 \tag{13}$$

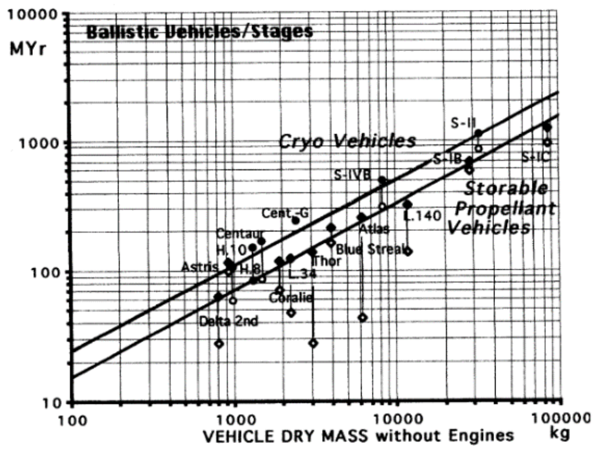

Figure 66: Production for expendable Stage vehicles

The CERs are applicable to new and reusable systems, but it should be noted that reusable vehicles require 40% more dry mass and are therefore more expensive to build. Hydrogen fuel vehicle systems are approximately 60% more expensive than those using storable propellants.

Calculation Production costs

Now the production cost can be calculated

CERs	Name	Mass [Kg]	Cost [WYr]
$F_{EL(s)}$	Liquid – Propellant rocket motors and boosters	35	152.7606
F_{VP}	Liquid propellant propulsión system	2793.0083	227.0013
F_{VE}	Ballistic vehicles/stages	2338.0083	128.4778

Figure 67: Production for expendable Stage vehicles

The production cost are 622.1098 WYr, that passing it to euros through the conversion of the TRANSCOST gives us **187.1306 million euros.**

13.2.3 Ground and flight operations costs

Direct Operations Costs

There are five major areas including the fees and charges covering launch aborts and vehicles failures.

1. The size and complexity of the vehicle, especially the number of stages and auxiliary boost units.

2. The fact wheter it is a crewed or an automated vehicle.

3. The assembly and mating and transportation.

4. The launch mode and type of facility

5. The number of launches per year

Ground Operations Costs

Base don the experience from the SPACE SHUTTLE and even Small experimental vehicles, the following provisional CER has been established:

$$C_{PLO} = 8M_0^{0.67}L^{-0.9}N^{0.7}f_v f_c f_4 f_8 \tag{14}$$

M0 is the launch mass in Mg.

The factor L determinantes the required launch team size vs number of launcher per year.

N determinantes the number of stages, and the exponent begin 0.7 says that a three stage vehicle requires a 115% higher effort than a Single Stage vehicle.

The factor fv describes the impact of the launch vehicle type:

1. Expendable multistage vehicles
   (a) Liquid propellant vehicles, cryogenic props = 1
   (b) Liquid propellant vehicles, storable props = 0.8
   (c) Solid propellant vehicles = 0.3
2. Reusable launch systems
   (a) Automated cargo vehicles = 0.7
   (b) Crewed/ piloted vehicles = 1.8

The factor fc indicates the impact of the assembly and integration mode:

1. Vertical assembly and checkout on the launch pad = 1
2. Vertical assembly and checkout then transport to launch pad = 0.7
3. Horizontal assembly and checkout transport to pad = 0.5

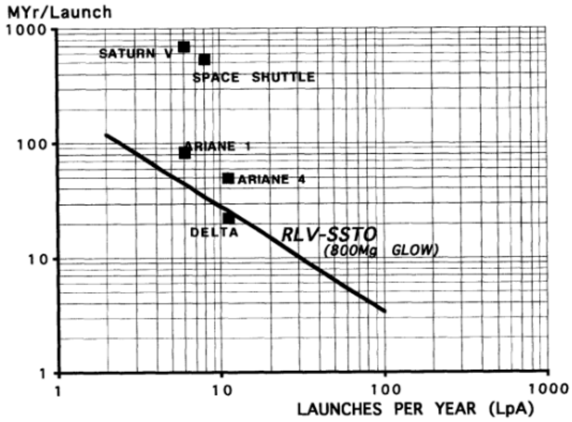

Figure 68: Cost/Launches per year

Cost of Propellants and Gases

Represent only a small fraction of the total cost per flight.

The costs and other propellants as valid in the USA in 1998/99 are:

LH2 Liquid Hydrogen		2.8 to 5 \$ / kg	15 - 25 MYr / Gg
LOX Liquid Oxygen		0.11- 0.15 \$ / kg	0.55- 0.75 MYr / Gg
Kerosene - RP-1		0.77 \$ / kg	3.9 MYr / Gg
Kerosene - JP-5	(ref.72)	0.26 \$ / kg	1.3 MYr / Gg
Kerosene - JP-7	(ref.72)	0.18 \$ / kg	0.9 MYr / Gg
N_2O_4 - Nitrogen Tetroxyde		7,6 \$ / kg*	38 MYr / Gg
N_2H_4 - Hydrazine		40-55 \$ / kg*	200 - 275 MYr / Gg
MMH - Monomethyhydrazine		100-130 \$ / kg*	500 - 650 MYr / Gg
H_2O_2 - Hydrogen-Peroxyde (85 %)		5,5 \$ / kg*	27,5 MYr / Gg
Helium-Gas		18 \$ / kg	90 MYr / Gg
Nitrogen-Gas		0.11 \$ / kg	0.55 MYr / Gg

*) according to Space news, 17.8.98

Figure 69: Propellant cost in 1998/99

Launch, Flight and Mission Operations Costs

The effort in this area is composed of

1. Mission planning and preparation
2. Launch and ascent flight control unit payload separation
3. Orbital and return flight operations in case of reusable, launch systems
4. Flight safety control and tracking

For expendable launch vehicles the mission planning, launch and flight operations effort is relative small. The flight time is less than 15 min for ascent to LEO.

For reusable launch vehicles the task becomes more demanding due the much longer missions period, including orbital operations, data transmissions and the return flight phase.

The major cost for unmanned launch and mission operations has been conceived as follows:

$$C_M = 20(\sum Q_N)L^{-0.65} f_4 f_8 \tag{15}$$

With L = Launch rate and Q = specific value depending on the complexity:

1. Small Solid Motor Stages = 0.15
2. Expendable Liquid Prop Stages or Large Boosters = 0.4
3. Recoverable or Fly back systems = 1
4. Unmanned Reusable Orbital Systems = 2
5. Crewed Orbital Vehicles = 3

Calculation Ground and Flight Operation costs

We consider that we perform only a punctual launch with QN = 0.4, we can estimate the cost:

CERs	Name	Mass [Kg]	Cost [M$]
C_{PLO}	Ground Operations Costs	13000	
C_{LOX}	Oxidizer Cost	7267.9	9.4482EXP (-4)
C_{RP1}	Fuel Cost	2839	2.1860EXP (-3)
C_M	Launch Flight and Mission Operations		2.0695

The Ground and flight operation cost are 2.0726 million dollars.

There are many more costs that would have to be considered but due to lack of data and development they are simply not assumed in order to make an estimate of the overall cost of the project.

14 Project Management

14.1 Project Planing

The following section provides an overview of each of the topics to be considered for project planning.

Project management provides advantages when it comes to carry out any type of project. Some of these benefits include having a clear view of the project objectives and it's definition or facilitation of the communication between team members, steakholders and other parts involved in the project. This ensures that everyone is aligned and working towards the same goal, while also allowing potential issues to be identified and resolved without delay.

Apart from this, risk management involves identifying potential risks and developing strategies to mitigate them. This helps to minimize the impact of risks on the project and ensure its success.

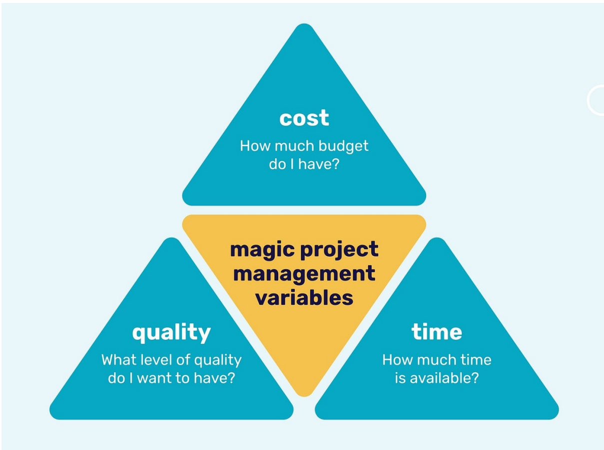

Figure 70: The project magic triangle

Some of the most important parts to take care of in project management are:

- Making a market assessment to define what are the market needs and opportunities to develop the project.
- Defining the goals and scope of the project.
- Creating deliverables and defining the timeline of the project.
- Identifying any variances from the plan and taking corrective action as needed to keep the project on track.
- Risk management to identify potential risks to the project and developing strategies to mitigate those risks.
- Project closure: This involves wrapping up the project, completing any final deliv-

erables, conducting a post-project review and archiving project documentation.

14.2 Stakeholders

In a launch vehicle design project, there are multiple stakeholders involved, both internal and external with different roles and interest in the project. Among the most important ones are the following.

Internal stakeholders:

- Project Team: This includes the engineers, designers, technicians, and other professionals directly involved in the design and development of the launch vehicle.
- Project Manager: Responsible for overseeing the project, managing resources and ensuring it's successful execution.
- Management and Executives: Leaders within the organization who provide guidance, allocate resources and make key decisions related to the project.
- Quality Assurance Team: Ensures that the design and manufacturing processes meet the required standards and specifications.
- Testing and Validation Team: Conducts tests and evaluations to verify the performance and safety of the launch vehicle.

External stakeholders

- Customers: The organization or entity that commissioned the launch vehicle design project.
- Government Agencies: Regulatory bodies and government organizations responsible for granting licenses, permits, and ensuring compliance with applicable regulations.
- Investors and Financial Institutions: Entities providing funding and financial support for the project.
- Suppliers and Vendors: Companies or individuals supplying components, materials, and services necessary for the design and construction of the launch vehicle.
- Partners and Collaborators: Other organizations or entities collaborating with the project team, such as research institutions, universities, or other aerospace companies.

14.3 Organizational Breakdown Structure (OBS)

This section provides an overview of the project's organizational structure, roles, and responsibilities. It outlines the key team members involved in the design of the space launch vehicle, including the project manager, propulsion experts, trajectory designers, cost analysts, systems engineers and aerodynamics specialists.

By establishing a clear organizational structure, the project team can efficiently allocate resources, manage risks and maintain effective communication channels.

14.3.1 Project Management

The project manager is Pablo Negro. This role makes him responsible for overseeing the overall execution of the launch vehicle design project. He ensures that the project remains on schedule, within budget, and meets the specified objectives. Pablo coordinates with various team members and stakeholders and facilitates effective communication and collaboration.

14.3.2 Propulsion

The members of this team are Carmen Cornejo, Sara Sánchez and Pablo Negro apart from his role as project manager. The members of this team collaborate to design and evaluate propulsion system components, including engines, fuel systems and propulsion subsystems. This team also develops propulsion systems analysis and optimization, guaranteeing the efficient and dependable performance of the launch vehicle.

14.3.3 Trajectory Design

The members of this team are Alejandro Sánchez and Adrián Martín. They are responsible for the trajectory design of the launch vehicle. They analyze various factors such as mission requirements, payload characteristics and launch site conditions to determine the optimal trajectory for achieving the desired mission objectives.

14.3.4 Systems Engineer

The member of this team is Francisco. He is responsible for ensuring the proper integration and coordination of various systems and subsystems of the launch vehicle. Francisco also conducts conducts system-level analyses, manages interfaces between different components and oversees the overall system performance.

14.3.5 Aerodynamics and Loads Distribution

The member of this team is Francisco. He assesses the aerodynamic characteristics of the launch vehicle, analyzes the loads experienced during different mission phases and ensures structural integrity and safety.

14.3.6 Cost Analysis

The member of this team is Adrián Martín. He assesses the financial implications of various design choices, materials, manufacturing processes and operational considerations.

Additionally, Adrian contributes to the trajectory design phase, ensuring that cost considerations are integrated into the trajectory optimization process.

14.4 Project Scheduling

This section outlines the timeline and sequence of activities for the launch vehicle design project. It provides a comprehensive overview of the planned tasks, milestones and their respective durations. Effective project scheduling ensures efficient resource distribution, timely completion of deliverables and effective coordination among team members. This section serves as a roadmap for the project, enabling stakeholders to track progress and make informed decisions to ensure project success.

The design process of a space launch vehicle is divided into multiple phases. The first three and most important of these are the following:

1. Conceptual design
    - Requirements definition
    - Definition of initial configuration
    - Study basic design alternatives
    - Mass distribution and performance
2. Preliminary deign
    - Configuration downselect and freeze
    - Nominal orbit and trajectory Design
    - Vehicle Analysis
    - Detailed cost evaluation
3. Detail design
    - Technology Readiness Level (TRL) assessment
    - Design parts to be built and make assembly instructions
    - Component testing
    - Refinement of financial and risk analyses

Additional phases of the project exist beyond the primary focus on the launch vehicle design itself and the development of these systems is carried out in parallel with the launch vehicle design.

- Conduct risk analysis and implement risk management plan

- Manufacturing and testing of the launch vehicle components

- Integration and assembly of the launch vehicle

- Testing and operation of the launch vehicle

- Obtain regulatory and safety approvals

- Launch operation planning and testing

- Operations and sustainment

- Closeout

14.4.1 Work packages and Scheduling

Work packages are an indispensable tool for effective project control and monitoring. Their use facilitates the visual tracking and management of ongoing and upcoming tasks in a highly efficient manner.

To ensure proper project management, the necessary tasks have been divided into these mentioned discrete work packages (Table 10). These packages are designated to various subsystem groups working in parallel to ensure timely and efficient completion. Each package is assigned an estimated duration, start and end dates and strict adherence to these timelines is essential to maintain the project's overall schedule.

In the matter on scheduling, some important considerations include:

- **Task dependencies:** Identification of dependencies between work packages is crucial to ensure that tasks are scheduled in the correct order. It's important considering both predecessor tasks (tasks that must be completed before) and successor tasks (tasks that depend on the completion of other tasks).

- **Resource availability:** It's important to take into account the availability and capacity of resources when scheduling work packages. Also ensuring that resources are not overloaded or double-booked and considering any constraints or limitations that may affect their availability.

- **Buffer time:** Allocating buffer time for unforeseen delays or disruptions to mitigate risks. This can help accommodate unexpected changes, such as resource constraints, technical issues or external factors beyond control.

- **Regular monitoring and updates:** It's essential to be continuously monitoring the progress of work packages and updating the schedule as needed. Regularly communicating with the project team to assess progress, address challenges and make necessary adjustments to keep the project on track.

WP.1 Mission Justification
Market assessment
Competitors analysis
Project planning and scoping
Project objectives and requirements
Similar launch vehicles research
WP.2 Mission Design
Assumptions
Orbit tradeoff
Nominal orbit
WP.3 Launch Vehicle Design
Payload requirements
Staging tradeoffs
Mass distribution and optimization
Geometry
Configuration downselect and freeze
WP.4 Aerodynamics and Loads
Trajectory design
Coefficients and loads
Thermal analysis
Loads analysis
Propellant tanks
Payload fairing
WB.5 GNC
GNC requirements
GNC design process
GNC subsystem integration
WP.6 Manufacturing
Components manufacture
Assembly
Integration of propulsion system
Testing of the launch vehicle
WB.7 Project Management
Project scheduling
Objectives and scope
Costs analysis
Risk analysis and implement risk management plan
Project closure

Table 10: Work packages

The previous work packages have been designed to have a sequential order. Some of them are collapsed as all the project members are divided into smaller groups or teams which have an assigned task or related group of tasks. The following table shows the work packages with it's duration, start and end date.

Task	**Duration**	**Start Date**	**End date**
Market assessment	1 week	02/07/2023	02/14/2023
Competitors analysis	5 days	02/07/2023	02/12/2023
Similar LV research	12 days	02/14/2023	02/26/2023
Project objectives and requirements	1 week	02/14/2023	02/21/2023
Project planning and scoping	86 days	02/21/2023	05/18/2023
Project scheduling	5 days	02/28/2023	03/04/2023
Assumptions	1 week	02/24/2023	03/02/2023
Orbit tradeoffs	10 days	02/24/2023	03/05/2023
Nominal orbit	6 days	03/05/2023	03/10/2023
Payload requirements	4 days	02/24/2023	02/28/2023
Staging tradeoffs	2 weeks	03/01/2023	03/14/2023
Geometry	1 week	03/14/2023	03/21/2023
Mass distribution and optimization	18 days	03/14/2023	03/31/2023
Configuration downselect and freeze	6 days	04/03/2023	04/08/2023
Trajectory Design	3 weeks	04/08/2023	04/28/2023
Propellant tanks	4 days	04/08/2023	04/11/2023
Coefficients and Loads	1 week	04/08/2023	04/14/2023
Thermal analysis	5 days	04/14/2023	04/18/2023
Loads analysis	12 days	04/14/2023	04/25/2023
Payload fairing	5 days	04/17/2023	04/21/2023
GNC requirements	4 days	04/25/2023	04/28/2023
GNC System Design	5 days	04/27/2023	05/01/2023
GNC Subsystem Integration	3 days	04/26/2023	04/28/2023
Components manufacture	5 days	04/28/2023	05/02/2023
Assembly	5 days	05/01/2023	05/05/2023
Integration of propulsion systems	5 days	05/01/2023	05/05/2023
LV Testing Process	1 week	05/05/2023	05/11/2023
Costs analysis	80 days	02/24/2023	05/15/2023
Risk analysis and risk management plan	80 days	02/24/2023	05/15/2023
Project closure	3 days	05/15/2023	05/18/2023

Table 11: Work packages duration and scheduling

14.4.2 Gantt Chart

To meet all the needs of managing the previously mentioned work packages and the scheduled dates for each of them, a valuable tool that can be used is a Gantt chart. This allows for a comprehensive visualization of the project, divided into different phases and subphases that are sequentially completed, while monitoring their progress through work packages.

By utilizing a Gantt chart, the project team gains a clear overview of the entire project timeline, enabling effective planning and coordination of tasks. The chart displays the start and end dates of each work package, highlighting any overlaps or gaps in the schedule. Dependencies between tasks are visualized, ensuring that sequential activities are properly sequenced and any interdependencies are managed effectively.

Additionally, the Gantt chart allows for easy tracking of progress, as each work package's starrt and completion date can be marked on the chart. This provides a visual representation of how the project is advancing and helps identify any potential delays or issues that may arise. By regularly updating the Gantt chart, project managers and stakeholders can stay informed about the project's status and take appropriate actions to ensure timely completion.

A Gantt chart has been developed for this launch vehicle design project (figure 71), providing a detailed timeline of activities, dependencies and milestones represented in table 10.

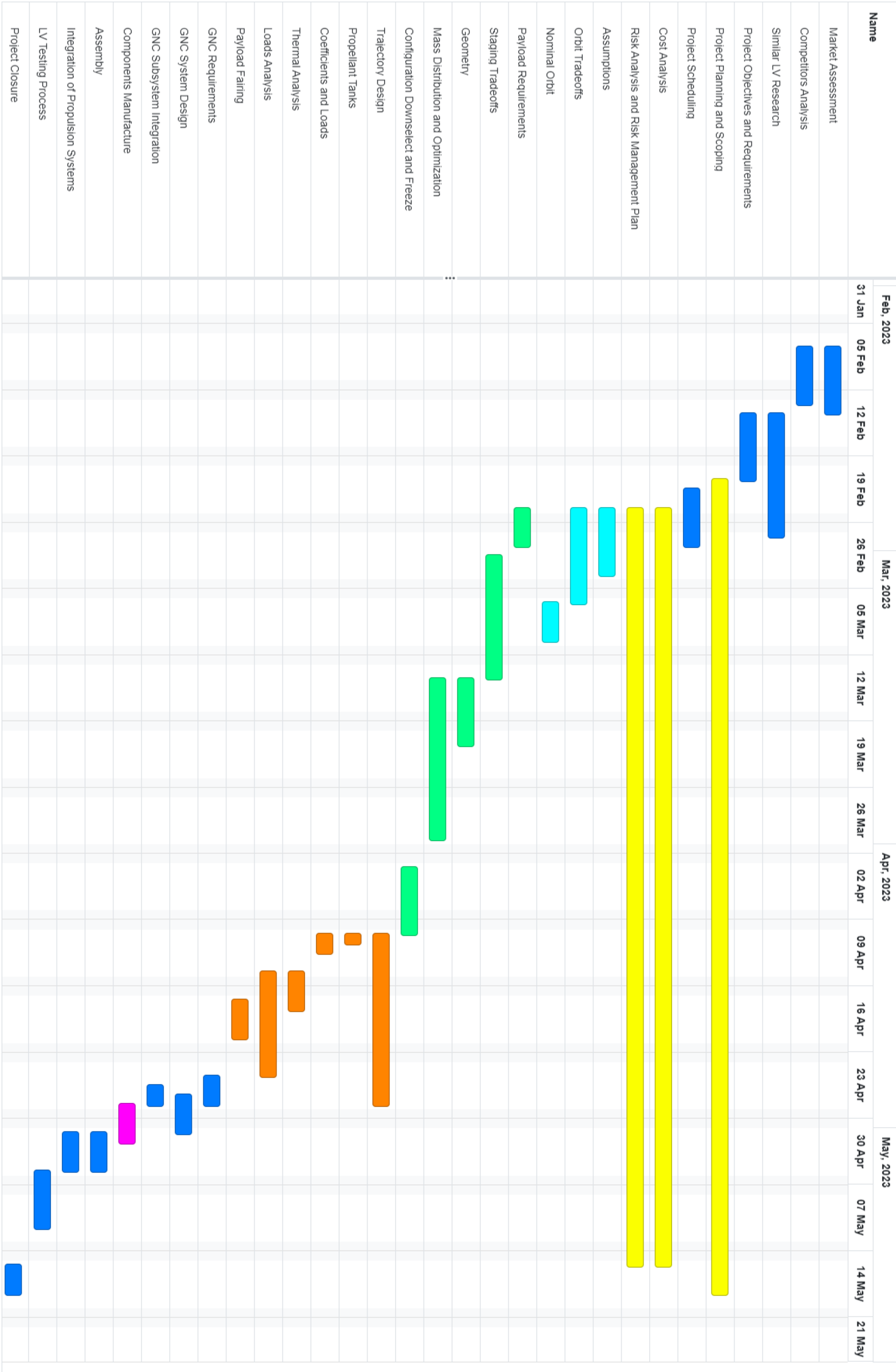

Figure 71: Gantt Chart

14.5 Conclusions

This final chapter contains a detailed examination of the project's goals and the application's progress. In addition, potential areas of enhancement will be discussed. It has been noted that designing a launch vehicle's concept, specifically for a certain payload, is far from a straightforward process. It entails assessing numerous interconnected factors, followed by carefully choosing from among several alternatives while weighing their respective benefits and drawbacks.

Definition of the requirements and constraints placed on the satellite to be launched, as well as the task that it must carry out once it is in orbit, must come first. The launcher must be able to lift the required amount of weight and fit the satellite inside and it must accomplish all of this with the highest level of accuracy, dependability and economy.

Last but not least, the best launch vehicle must be equipped with a propelling system that enables the launcher to generate the necessary impulse to place the satellite into orbit. For a staggered vehicle, the number of stages and the individual stages' propelling properties must be specified. All of these components have been examined and conclusions regarding the perfect launcher configuration and potential launch sites for the vehicle have been reached.

We used mass optimization to reduce the amount of fuel needed to move the vehicle at the desired speed with a 100 kg payload in order to arrive at this solution. In conclusion, the current work has made it possible to assess each factor that affects the conceptual design of a launch vehicle system for a particular mission. Each factor that affects this process has been carefully explained and its significance delineated. Additionally, it has made it possible to get familiar with and employ project management tools and procedures and to recognize their value.

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