# Study and Optimization of an Air Launched Nanosatellite Launcher

Guilherme Rodrigues (GuilhermeORodrigues@gmail.com)

**Abstract – Technological developments in miniaturization and electronics have allowed for the reduction of the size of the satellites required for a given mission, as well as in cost, time taken for design and construction and complexity. A particularly interesting class of satellites are the nanosatellites, having a mass between 1 and 10 kg, which are seeing widespread development and use, their development stimulated by initiatives such as the cubesat competitions. Their increasing relevance has not led to an adequate development in launch capabilities, which remain geared towards large and heavy satellites. The present work defines a baseline for the possibility of using a military air plane as a carrier for an air launched rocket optimized for small payloads such as nanossatellites. The most important parameters defining the rocket were studied and near optimal solutions were obtained, in the context of a preliminary design of this type of vehicle.**

## 1 Introduction

With the advances in miniaturization of components, small satellites are capable of tasks that used to require a much larger one [1]. Current technology has allowed for nanosatellites (mass between 1 and 10 kg) to offer capabilities worthy of a space launch, at a lower cost than conventional satellites. Since the present generation of launchers were designed for much larger satellites, the only option for launch is as secondary payload, representing a limiting requirement for the mission. A dedicated launcher for nanosatellites would allow for lower costs and increased number of operations. In the current work, the option of the air launch of small payloads was analysed, as it brings several advantages:

- Flexibility in launch locations
- Removes dependency in launch windows allowing for flexibility in launch
- Does not require the use of dedicated ground stations decreasing costs
- Launching from altitude reduces the required potential energy difference, increases the propulsion system and reduces drag by
- Launching with a given velocity reduces the necessary added kinetic energy

A fully optimized solution is not the objective of this work, what was aimed was to analyze the whole launch system according to the different intervening factors and study their influence from an engineering point of view. As such, everything from the carrier plane to the trajectory was analyzed so as to find a near optimum point. In this kind of work it is common to use multidisciplinary optimization techniques, of which several examples are available [2][3], but usually they limit their scope to a few parameters. We opted to consider the system as a whole, discussing all the main driving system parameters, since that discussion is crucial.

## 2 Parameters

To analyze all the main parameter interactions, they were divided according to type:

- Carrier, which defines the initial conditions
- Trajectory
- Vehicle, the capabilities of the rocket

As each element is almost fully independent, near optimizing each of the three, leads to a good whole system solution.

### 2.1 Carrier Plane

The role of the carrier is to give the rocket the maximum possible initial energy, achieved by going as high and/or fast as possible. This allows for either a lighter and simpler rocket, reducing costs, or it allows the same rocket to carry a heavier payload. A trade-off analysis was performed between the various carrier parameters.

#### 2.1.1 Study Variables

Presenting a quick outline of the main variables:

- Physical constraints – Maximum weight size constraint (length and height).
- Altitude – With direct advantages from potential energy and indirect from lower drag.
- Speed – A higher initial speed is advantageous, but complications arise from a supersonic launch.
- Dock and release

#### 2.1.2 Support Rail and Release

The support rail secures the rocket to the carrier during the flight to the launch location and should allow compatibility with a significant number of aircraft. This last issue should not be a problem given military commonality and will not be further discussed in this work.

Two systems are commonly used for the release of payloads. The first is a drop system, which not only excludes supersonic release but also requires the rocket to have some kind of stabilization (usually aerodynamic). The second method relies on the rail, to which the rocket is fixed, and that allows for a supersonic launch. For this the Rail option was selected.

### 2.1.3 Aeroplane Class Selection

The suitable aeroplanes were split into three separate categories:

- High Performance Fighter Aircraft (F4, F16, Mig 29) – high availability, best performance and flexibility but lower weight capacity
- Large Bomber Aircraft (Tupolev 160, B1, B52) – best carrying capacity, good performance, low availability
- Patrol Craft (Nimrod, P3 – Orion) – middle term in both performance and availability.

Since Large Bombers are available only to a small number of countries we decided not to pursue this alternative. We also decided not to study commercial airliners as alternatives since the retrofitting required would be too expensive to fit with the low cost approach. A number of vehicles of each class were selected and the characteristics of the typical craft calculated[4]:

| Class | Service Ceiling | Top Speed | Range | Payload |
|---|---|---|---|---|
| Fighter | 12 km | 2000 km/h | 1000 km | 2500 Kg |
| Patrol | 10 km | 700 km/h | 5000 km | 4000 kg |

Tab. 1

From tab.1, the only advantages of the Patrol class are a higher payload capacity and range. The first is irrelevant, as the rocket is much lighter than the Fighter's maximum payload, and the extra range is not enough reason for selection. With the wider availability also factored in, the Fighter plane is the logical choice. Considering the supersonic launch is possible, it was selected for its expected advantages.

### 2.1.4 Plane Manoeuvre

The complete definition of the carrier flight path and losses is out of the scope of this work. The final stages of the flight are considered to be a wide loop with a low climb rate, and without loss of forward speed.

### 2.1.5 Initial Angle ($\gamma 0$)

It was determined that for a certain initial angle a minimum value of thrust is needed leading to a minimum mass flow ratio $\dot{m}$, however, an increase in $\dot{m}$ increases the structural weight and therefore leads to worse results. A trade-off analysis was performed to optimize the solution and will be presented in the propulsion section, section 2.3.4.

### 2.1.6 Initial Point

We decided to use a Fighter Jet as the carrier plane for the much higher performance offered. Considering a significant number of available options, the initial point was defined as seen in Table 2.

| | |
|---|---|
| Initial Speed | 550 m/s |
| Initial Altitude | 12 km |
| Initial Angle | 62.5º |

Tab. 2

## 2.2 Trajectory

Much depends on the trajectory, which has to minimize fuel mass by attaining the right balance between lower altitude burns (more energetically efficient) or higher burns (less drag). As the trajectory also determines the loads that the Rocket is subjected to, care must be taken to ensure that the force is kept as axially aligned as possible. This represents a complex optimization that goes beyond the scope of this work and only a simplified but commonly used approach is adopted [5].

### 2.2.1 Gravity turn

A possible solution to this problem is to have the vehicle perform the so-called gravity turn trajectory during the atmospheric flight. This consists in maintaining the thrust and velocity vectors aligned at all times thus ensuring zero angle of attack. An initial angle is given to the rocket and will increase during the flight by the action of the gravity force [6].

### 2.2.2 Trajectory calculation

The selected trajectory is split into three parts, determined using the cartesian approximation:

- The first burn will give the rocket enough potential energy to reach the target altitude - gravity turn type of trajectory for the whole burn.
- Between burns, a small coasting phase (20 seconds) for stage separation and the rocket manoeuvres itself for the second burn.
- A second burn, horizontal - accelerates the payload up to orbital velocity.

This provides a realistic if not completely real path for the rocket, approximating what usually takes place in a launch. It has the advantage of being simple since the horizontal impulse (second burn) does not interfere with the vertical speed provided to increase potential energy, due to the cartesian approximation used. While not a completely accurate and optimized solution for the trajectory problem, it is sufficiently good for the goal intended.

## 2.3 Rocket

The basic functioning of a rocket can be easily expressed by Tsialkovsky's Rocket equation, where $v_e$ represents the exhaust velocity and the various m variables represent masses.

$$\Delta v = v_e * \ln\left(\frac{m_{pay} + m_s + m_{propellant}}{m_{pay} + m_s}\right) \quad [2.1]$$

This equation [6] provides an exact solution in vacuum, with correction for other conditions represented by extra Δv terms added to the left side of the equation, to account for effects such as drag or gravity losses.

### 2.3.1 Configuration

Since the performance of the rocket is extremely dependent of the structural mass (seen from eq [2.1]) methods are used to reduce it as much as possible. By separating the rocket into several stages it is possible to achieve a much higher performance [6]. In the present case, the carrier plane may be considered as a first stage and the final number of stages was studied to verify the best configuration. The options are:

- 1 - single stage rocket
- 2 - linearly staged rocket
- 3 - single stage rocket with boosters
- 4 - linearly staged rocket with boosters

As boosters should be complicated to implement for the air launch, only option 1 and 2 were tested. As can be seen in Fig 1, the staged option is much more efficient. A higher number of stages would provide even better performance, but the gain vs complexity ratio would be poor, justifying the choice for 2. The stages' division follows the trajectory plan; the first stage fulfils the first burn and the second stage the second.

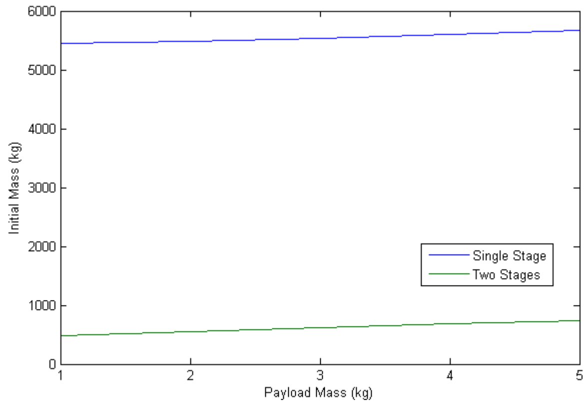


**Fig. 1**

### 2.3.2 Propellant

The selection of propellant is one of the most important and complex choices of the whole vehicle, both for the importance it has in defining the performance as for the number of choices available. The first factor is the propellant type, of which there are 3 [7]:

Solid Propellants – pros: the simplicity of having fuel and oxidizer in a single solid mixture, the low maintenance required and high density. Cons: the lowest efficiency, require a heavier casing, prone to accidents from impact or from grain defects and there is no thrust control.

Liquid Propellants – pros: a wide variety of choices, the availability of thrust control including shutdown and re-ignition and the highest efficiency of chemical rockets. Cons: a smaller density, the added weight of both the pressurization system, valves and plumbing, some have added constraints regarding use or storage and many have high toxicity.

Hybrid Propellants – these stand in the middle, usually consist of a solid fuel with a liquid propellant. Comparatively to Solid Propellants the greatest changes are a higher efficiency and some burn control, while compared to a Liquid solution they present a higher density and are mechanically simpler.

We adopted a Liquid Propellant for its superior performance, safety and versatility. Further options are available, beginning with the number of substances used:

- Monopropellants – single substance. Pros: simplicity in their use. Cons: low efficiency.
- Bipropellants – separate fuel and oxidizer, mixed and ignited. More complex, but much more efficient.

- Tripropellants – usually bipropellants with an extra chemical added. Pros: the most efficient chemical fuels. Cons: extremely complex, never been used in a standard vehicle and usually extremely toxic.

And according to storage temperature:

- Earth Storable – liquid at room temperatures.
- Cryogenic – need to be stored at very low temperatures to remain liquid. The best mixtures available but add complexity.

In keeping with the design goal of a simple system an Earth Storable Bipropellant Solution was selected, as it provides a good balance between performance and complexity while complying with operation procedures. From the available propellants in that class a choice was made, the relevant parameters being a high density, preferably a low Chamber Temperature and a high Isp. With the cryogenic solution rejected, and both Fluorine and Red fuming nitric acid extremely toxic, we decided to use liquid nitrogen tetroxide as oxidizer for which there are two suitable fuels: Rp-1 [fuel 2] and an equal mix of unsymmetrical dimethylhydrazine and hydrazine [fuel 1].

This issue goes beyond merely engineering or performance factors. Fuel 1 offers the best performance [fig 2], but it is also extremely toxic requiring specialized handling. Fuel 2 is a less efficient option but much safer one. The fuel selected was Rp-1. It is more readily available, cheaper, much safer to handle, and the performance penalty is not significant. The relevant data is shown in table 3 [7].

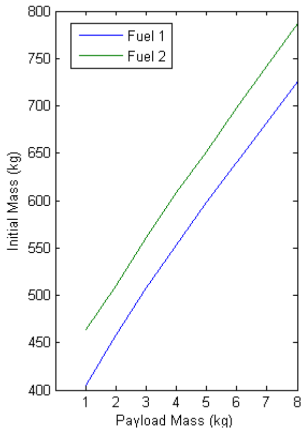


**Fig. 2**

| | |
|---|---|
| Fuel Density | 915 $Kg/m^3$ |
| Oxidizer Density | 1413.5 $Kg/m^3$ |
| Oxidizer to Fuel Mass Ratio | 3.4 |
| Propellant Molar Mass | 24.1 Kg/mol |
| Specific heat ratio (k) | 1.23 |
| Chamber Temperature | 3290 K |
| Chamber Pressure | 6894757.29 Pa |

**Tab. 3**

### 2.3.3 Nozzle

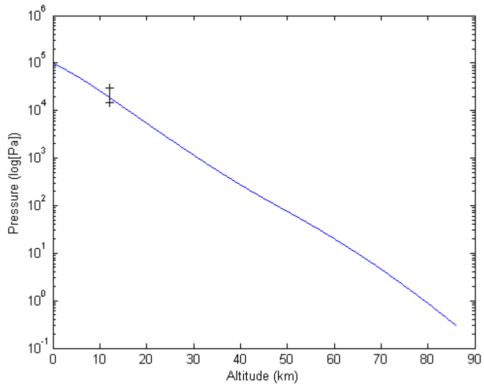


**Fig. 3**

The main driving parameter of the nozzle is the design altitude at which it operates at maximum efficiency, from which the area ratio between the nozzle throat and exit is derived. A balance has to be found between a smaller area ratio for low altitude operation and a large area ratio for vacuum, also minimizing mass and physical dimensions. The air launch proves extremely advantageous, as pressure variations are much lower at high altitudes, as can be observed in Fig.3 [8].

For each area ratio we calculated the thrust efficiency at each altitude (represented by the Isp) as well as the physical dimensions, shown in Fig 4. As the physical characteristics depend on the propellant mass ratio, a dimensionless plot was obtained. We saw the efficiency does not change significantly whether the nozzle is optimized for the launch altitude or space (under 10% variation) while the physical dimensions do. After analysis the best area ratio was determined and set at 80.

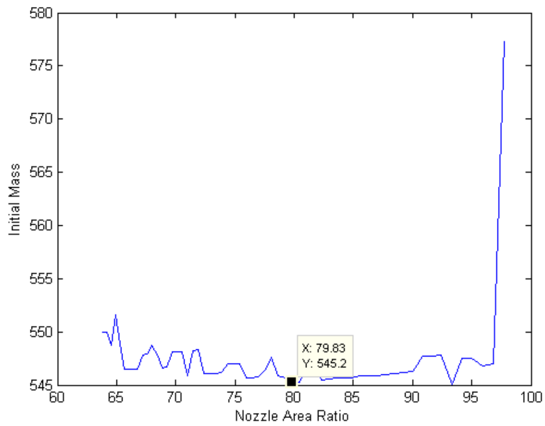


**Fig. 4**

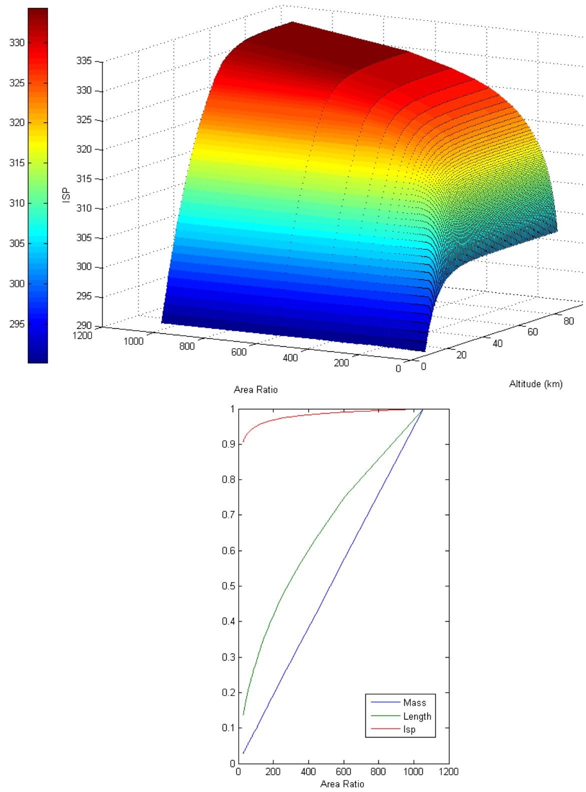


Fig. 5

### 2.3.4 Angle and mass flow rate

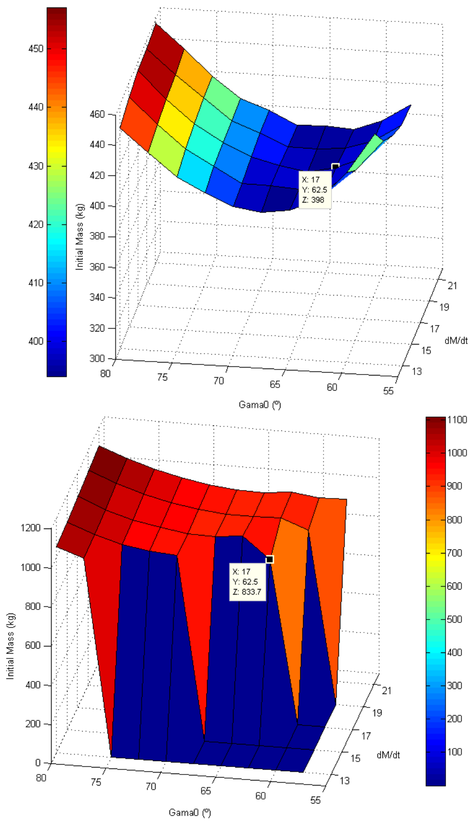


Fig. 6

The minimum allowable propellant mass flow is dependent of the initial angle – lowering the initial angle lowers the initial mass but requires a higher propellant mass flow and a lower propellant mass flow lowers the initial mass but requires a higher initial angle. Both parameters were tested together for the minimum mission (1 kg and a 300 km high orbit - top) and the maximum mission (10 kg at a 700 km orbit - bottom), shown in Fig. 6, with a more precise analysis shown in Fig. 7.

Analysing both graphics we can see that the maximum efficiency point is similar. The selected values are 62.5º for the initial angle and 17 kg/s for the mass flow rate. While this is not the most efficient point for the lower mission, it is fairly close and proves to be highly advantageous for the heavier one.

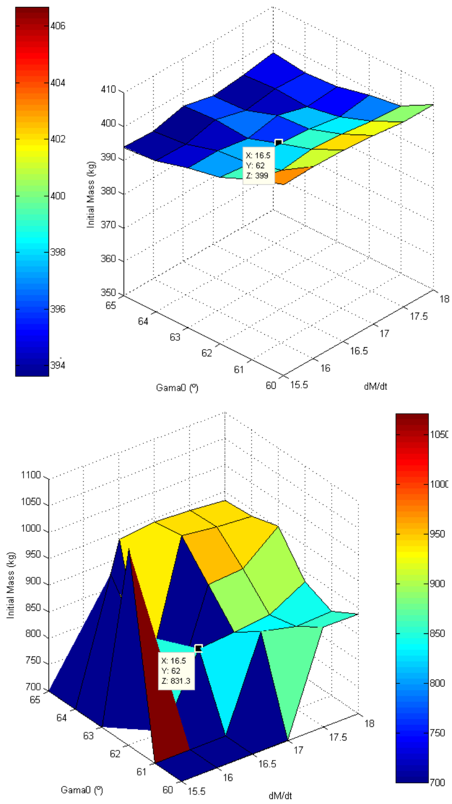


Fig. 7

The mass flow ratio of the second stage was also analysed with results showing that the most efficient point is dependent on the mission, shown by the higher horizontal error for lower values. Since the objectives emphasize versatility, the selected value was 0.5 kg/s as it fulfils the most demanding mission. Fig. 8 shows the initial mass made dimensionless by the maximum value top and the horizontal error bottom, both versus propellant mass flow ratio.

As the mass flow ratio is constant during each stage burn, and as we approximated the Isp constant with the value of the vacuum Isp, the thrust is also constant during the burns.

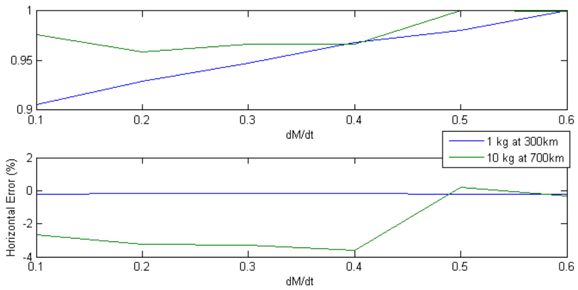


Fig. 8

### 2.3.5 Pressure system

To pressurize the propellants there are two main types of systems, the use of a turbo pump assembly or of a tank of pressurized gas. The first option offers the best propulsive performance but its complexity increases price and possible failure rates. Its mass is almost fixed for a given thrust and irrelevant of fuel mass, so no judgement of the matter can be held beforehand. The second solution consists a tank of an inert gas (usually helium) connected to the fuel and oxidizer tanks. While it is a simpler solution, the tank is rather heavy due to the required pressure, and the propellant tanks have to also be kept at a higher pressure needing reinforcement. We selected the pressurized gas tank option. While the mass cannot be discussed, the difference in complexity was the reason for the choice.

## 2.4 Aerodynamics

Aerodynamic calculations are computationally demanding, usually requiring time consuming CFD analyses to provide accurate solutions. To reduce the numerical workload a simple aerodynamic model from [9] was used. As the rocket has no aerodynamic stabilizers, only the sections pertaining to longitudinal force were considered.

Little optimization was required. The shape is that of a completely conventional rocket with a nosecone on the top and a cylinder behind. The only structural optimizations are the finesse ratio of both the nosecone and the cylinder section which have to be chosen balancing drag and structural weight. Regarding the nosecone, the analysis is presented in Fig 9 which shows the dimensionless mass vs the finesse ratio, with the final value of 4.6 having been chosen adequately. The analyses regarding the cylindrical section will be presented in the structural section.

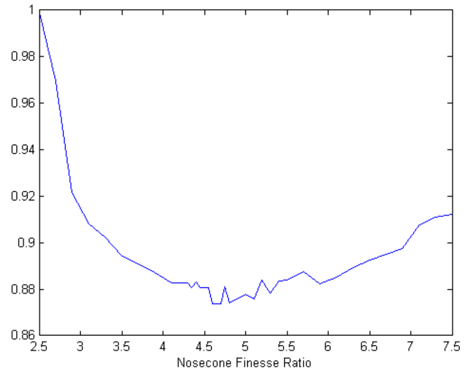


Fig. 9

## 2.5 Structure

### 2.5.1 General Description

The structure component of the rocket has two main tasks: to withstand the structural stresses and to cover all the vehicle's components. It must be strong and stiff enough to accomplish the first while remaining light enough not to hinder the system's performance. The main components are:

General:

- Casing – the outer shell of the vehicle, shaped like a cylinder. It encloses all the subsystems and supports the loads
- Electronics – including the guidance and communication systems.
- Nosecone – the top of the rocket, it houses the payload and is shaped as a parabola

Propulsion:

- Nozzle
- Propellant tanks – of cylindrical shape
- Pressure gas tank – a spherical tank
- Supplementary mass – plumbing, actuators, thrust chambers, cooling...

### 2.5.2 Structural Stresses

The stresses applied on the vehicle were divided according to type/source.

Axial tensions – both lengthwise and perpendicular.

- Lengthwise – Thrust and Drag and a component of the Weight
- Transverse – during the total weight; during flight a component of the vehicle Weight.

Manoeuvring torques – during attitude changes.

As there is only one significant manoeuvre and it occurs during a coasting phase we considered it to be performed softly enough that the stresses generated are lower than from normal operation and can be ignored.

Propellant pressure

Since the propellant tanks are integrated with the casing it is also subjected to the pressure. Since the pressurizing gas has a higher pressure, an independent spherical tank was used, its mass calculated separately. The propellant tanks are both pressurized to 0.5 MPa and the gas tank to 6MPa, values comparable to the conventionally used [6].

### 2.5.3 Finesse ratio optimization

The structural finesse ratio influences both the structural weight and aerodynamic drag, a trade-off must be performed. The evolution of both factors, in a dimensionless form, is depicted in Fig 10.

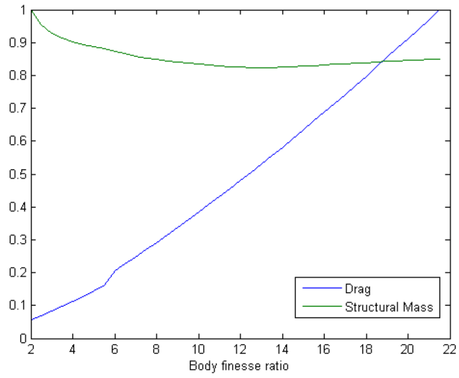


**Fig. 10**

While the vehicle's structural mass has a minimum, the drag force increases constantly with the finesse ratio. Because of this, 3 different missions were tested, varying the finesse ratio and obtaining the initial mass required for each. Seen in Fig 11, all three show similar evolution and the final selected value was 8, as to minimize weight.

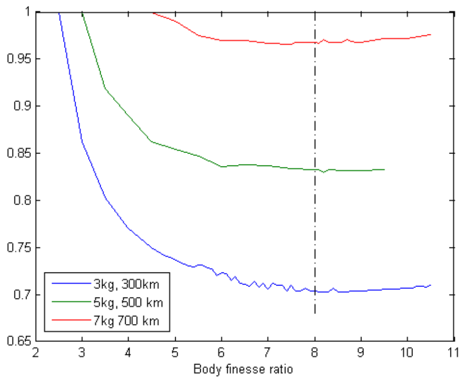


**Fig. 11**

### 2.5.4 Structural layout

The interior layout is simplified. We calculated the oxidizer and fuel masses and their volumes, increased by 10% as extra volume [6]. The structural radius was so obtained, increased in 10% to account for plumbing and other systems not explicitly detailed, the rest being calculated from geometrical constraints. The casing layout was kept simple, with the cylindrical outer shell and the tanks simulated by three circular ends inside the shell having the same thickness. This leads to an over dimensioning of the structure, since the tank deposits aren't required to support the same kind of loads, giving the results a certain extra degree of safety.

### 2.5.5 Materials

The structural algorithm performs the calculations for all the available materials and then the lightest solution chosen. The material list is presented below in Tab. 7. We found the lightest option to be the titanium alloy, results (divided by the lightest) shown in figure 12.

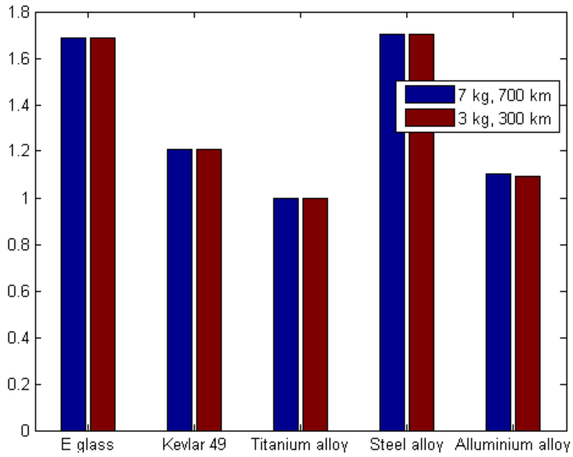


**Fig. 12**

| Material | Ultimate Strength (MPa) | Density (Kg/m$^3$) | Minimum thickness (mm) |
|---|---|---|---|
| E glass reinforced plastic | 1030 | 1940 | 1 |
| Kevlar 49 reinforced plastic | 1310 | 1390 | 1 |
| Titanium alloy | 1240 | 4600 | 0.25 |
| Steel alloy | 2000 | 7840 | 0.25 |
| Aluminium alloy | 455 | 2790 | 0.25 |

**Tab. 4**

For the nozzles and burn chamber a material capable of withstanding high temperature is required, and our choice was a Niobium C130 [10] alloy, similar to the one used in the Space Shuttle OMS system's nozzles.

| | |
|---|---|
| Density | 8850 kg/m$^3$ |
| Melting Point | 2350 +-50 ºC |

Tab. 5

### 2.5.6 Electronics and guidance

It is somewhat complicated to estimate the mass required for the electronics of the system. Searching for cubesat components, we have found communications sets weighting from 80 to 200 g. We thus estimated a total mass of 300 g for all the required electronics.

# 3 Design

With all the main design variables defined we may now study the rocket as an integrated system, the vehicle's size being dependent on the mission desired. As a generic case, the mission chosen was a circular orbit of a given payload mass at a given altitude. The mission variables were selected as:

- Payload mass: 1 to 10 kg, which compromise the limits for the nanosatellite class.
- Orbital Altitude: 250 to 700 km, low Earth orbits. The lower limit is high enough for a high orbital life while the upper limit is still admissible for the class.

The results are shown in Fig 13.

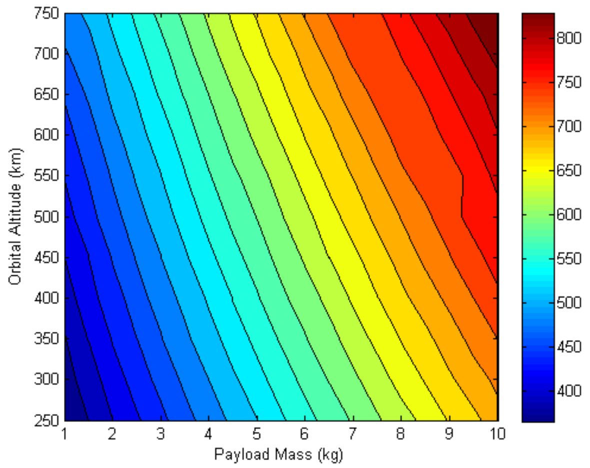


Fig. 13

We can see that there is a regular evolution of mass with the desired final mission, with a clear bias in the curves' slope towards the payload mass axis. In the interest of flexibility and providing a range of solutions we decided to analyze three significantly different rockets, each optimized for a different mission: 3 kg at 300 km altitude (Rocket 1), 5 kg at 500 km altitude (Rocket 2) and 7.5 kg at 700 km altitude (Rocket 3). The most relevant data of each rocket is presented in Tab 6:

| | Rocket 1 | Rocket 2 | Rocket 3 |
|---|---|---|---|
| Initial mass (minus payload) | 443.61 kg | 578.00 kg | 706.78 kg |
| First stage fuel mass | 331.5 kg | 464.53 kg | 595.43 kg |
| First stage structural mass | 34.94 kg | 42.00 kg | 48.53 kg |
| Second stage fuel mass | 69.28 kg | 63.60 kg | 55.21 kg |
| Second stage structural mass | 7.895 kg | 7.868 kg | 7.608 kg |
| Structural mass | 42.83 kg | 49.87 kg | 56.14 kg |
| Length | 3.480 m | 3.823 m | 4.103 m |
| Structural radius | 0.218 m | 0.239 m | 0.256 m |
| Single volume payload capacity | 0.0744 m$^3$ | 0.1013 m$^3$ | 0.1253 m$^3$ |
| Nosecone Length | 2.001 m | 2.198 m | 2.359 m |

Tab. 6

It is easy to see that the relative size of the stages changes with the mission: the first stage's mass increases with a more demanding mission (more energy is required to raise a higher mass higher) while the second stage becomes lighter for the more extreme mission (lower orbital velocity at higher altitudes and higher velocity at the end of the gravity turn manoeuvre). The initial mass always increases with the more demanding the mission (heavier load / higher altitude), leading to a larger rocket.

## 3.1 Inverse Calculation

Having selected those three representative rockets we analyzed the full range of capabilities of each – the spectrum of missions each is capable of fulfilling. The calculations were performed by assigning a target altitude and calculating the allowable payload mass, the results for the Rocket 2 shown in Fig14.

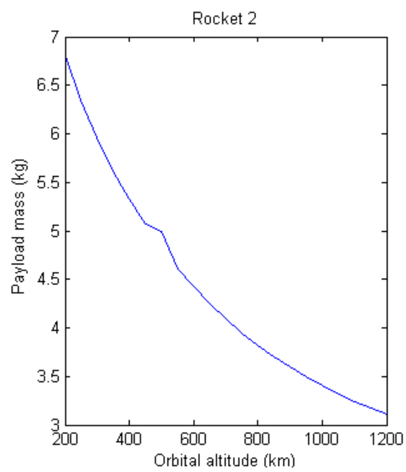


Fig. 14

We found that the rocket presents a rather smooth evolution in terms of payload vs altitude, with a blip representing the optimized point showing a higher carrying capacity than would be extrapolated from the rest of the curve. We also calculated the effect of doubling or halving the altitude (compared to the design point) and as seen in Tab 7, the larger the rocket the lower the performance loss.

| Rocket | Altitude*2 | % | Altitude/2 | % |
|---|---|---|---|---|
| 1 | 1.341 kg | 45 | 3.699 kg | 123 |
| 2 | 3.403 kg | 68 | 6.35 kg | 127 |
| 3 | 5.44 kg | 78 | 8.522 kg | 122 |

Tab. 7

## 3.2 Performance improvement

We also calculated the increase in performance possible by using a more capable carrier, using Rocket 3 for the analysis as it is less sensitive to changes. We obtained Fig 15 where the Payload mass as a % of the design payload is shown.

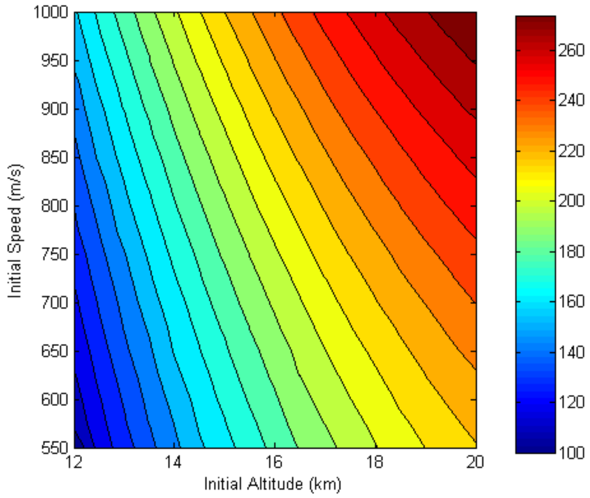


Fig. 15

As supposed, a higher performing carrier allows for a higher payload mass. We also discovered that increasing the altitude brings higher benefits than the initial velocity, increasing the velocity by 25% leads to an 11% increase in payload, while the same increase in the initial altitude increases payload by 49%. This is easily explainable as increasing the altitude increases the initial potential energy and allows the rocket to operate in a lower drag area (both beneficial). Increasing the initial velocity increases kinetic energy but also increases drag (contradicting effects)

## 3.3 Use considerations

Even though the current work is a preliminary study, it is still useful to take some time to consider some aspects of the use of a vehicle like the one designed.

### 3.3.1 Handling and risks

When studying a project whose results are desired to have a global use, special care has to be taken regarding the risks in both use and handling. This was already considered in the fuel selection, where it was opted to use a non toxic option, which also has the added advantage of facilitating and reducing the risk of the transport operations of the vehicle; it may be carried empty and filled on location. There is always a certain degree of risk when working with rockets, it is inherent to space exploration and will most certainly never be fully eliminated. However the decisions made, from a non toxic fuel and to non hypergolic propellants suggest a low and admissible risk rate in terms of ground handling and storage.

### 3.3.2 Storage and reliability

The vehicle is designed to be kept in storage ready to launch when required. The use of earth storable propellants was fundamental in this matter, as the rocket may be kept completely fuelled for a significant period of time (estimated to be around 10 years for the oxidizer [6]). As the design choices were made with simplicity in mind, there is no reason to expect reliability issues, if the vehicle is adequately stored and periodically checked.

### 3.3.3 Quick response estimate

Two of the most interesting and useful aspects of an air launch are the reduced dependence on launch windows, and the absence of weather interference. This represents the ability to operate on a short notice, a rising demand in the field of space launches. By having the rocket in storage fuelled and even with the payload already inside the time required for the launch is reduced to the time required to prepare the carrier for launch and the ferry to launch point – hours, as opposed to weeks, months or even years that missions usually take. This allows for a quick response to events, such as terrorist attacks or natural disasters, and a much easier real time monitoring of situations.

# 4 Conclusions

In this work we studied the main parameters influencing the performance of an air launched rocket. With trade-off analyses we observed interactions and obtained a near optimal solution for the rocket as a whole. This analysis constitutes a baseline for further work in the subject.

Focusing in the engineering aspects of the problem as a whole, and by performing a heuristic analysis, there are some limitations in the work. While all the decisions were valid and justified, there is room for improvement. A particular source of error is the type of approximations considered. The drag model is simple, thrust was considered constant and the structural analysis was reduced to a single kind of stress. All these limit the possible precision. The trajectory chosen is also simplified.

These limitations do not invalidate the results obtained. The interactions between the parameters remain the same regardless of the optimization level, as do most if not all of the engineering decisions.

This work has shown that an air launch is a viable solution to the nanosatellite launch problem, being possible to obtain a responsive and relatively simple system by making use of broadly available technology, putting such system within reach of many countries and organisations. From this point of view, this type of systems together with the development of the nanosatellites themselves, can promote a broader access to space and its advantages.

## 5 References

[1] **Stephens, J. Paul.** *Developing National Space Capability with Small Low Cost Satellites*. Proceedings of International Conference on Recent Advances in Space Technologies, IEEE, 2003. pp. 102 - 111.

[2] **Choi, Y. C., et al.** Optimal air-launching rocket design using system trades and a multidisciplinary optimization approach. *Aerospace Science and Technology.* 2009.

[3] **Ridolfi, L., Pontani, M. e Teofilatto, P.** Effect of different flight conditions at the release of a small spacecraft from a high performance aircraft. *Acta Astronautica.* 2009.

[4] **Jackson, Paul, Peacock, Lindsay T. e Munson, Kenneth.** *Jane's All the World's Aircraft.* Surrey, Uk : Jane's Information Group, 2004. ISBN 0710626142.

[5] **Callaway, David W.** *Coplanar Air Launch with Gravity-turn.* s.l. : Airforce Institute of Technology, 2004.

[6] **Wiesel, William E.** *Spaceflight Dynamics.* s.l. : McGraw-Hill, 1997. ISBN 0-07-070110-5.

[7] **Sutton, George P. e Biblarz, Oscar.** *Rocket Propulsion Elements.* s.l. : John Wiley & Sons, 2001. ISBN 978-0471326427.

[8] *U.S. Standard Atmosphere.* Washington : U.S. Government Printing Office, 1976.

[9] **Al-Garni, Ahmed Z., Kassem, Ayman H. e Abdallah, Ayman M.** Aerodynamic-Shape Optimization of Supersonic-Missiles Using Monte-Carlo. *International Review of Aerospace Engineering.* 2008, Vols. Vol.1, N. 1.

[10] **ATI Wah Chang.** *Nb/Nb Alloy Products Technical Data Sheet.* 2003.