Launch Vehicle - MDO in the development of a Microlauncher

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ABSTRACT

In the Frame of Romanian Incentive Scheme Programme under European Space Agency, a feasibility study of a Small Orbital Launcher was performed by "Politehnica" University of Bucharest. Building on the results obtained in the project, European Space Agency's Space Transportation Directorate through the Future Launchers Preparatory Programme awarded INCAS a Phase 0/A contract for a microlauncher concept. The targeted market is the micro-satellites sector developed by universities, but also by small companies for research purpose and validation of applications/products before commercialization, with a mass lower than 150 kg. This paper presents a conceptual design for a microlauncher having a target mission to deliver a small satellite of 150 kg into a 600 km circular polar orbit with the launching site located in Europe. The microlauncher concept is obtained using an in-house multi-disciplinary optimization tool, which contains four modules: weights and sizing, aerodynamics performance assessment, propulsion, trajectory computation and optimization.

KEYWORDS: small launcher, multi-disciplinary optimization, small satellites, polar orbit

NOMENCLATURE

MDO – Multi-Disciplinary Optimization
LB - lower bounds
UB - upper bounds
GLOW - gross lift off weight
R - radius
V - velocity
SRE - solid rocket engine
LRE - liquid rocket engine
P/L – payload
p - Constraint violation
γ - flight path angle
R_{target} - target orbit radius
V_{target} - target orbit velocity
H_{orbit} - orbit altitude
I_{to} - performance index for the target orbit
I_{pc} - performance index for the path constraint
W_R - orbit radius weight
W_V - orbit velocity weight
W_γ - orbit flight path angle weight
n_a - normal load
n_{amax} - imposed maximum normal load
$m_{payload}$ - payload mass
var_{stage} - stage optimization variables
t_v - vertical flight time
t_{c12} - first coasting time
t_{c23} - second coasting time
Θ_{ij} - control parameters
γ_{target} - target orbit flight path angle

1 INTRODUCTION

At this moment, worldwide, the need to have a dedicated launch option for small satellites arose as the current alternative is a shared ride with the bigger satellites. For this piggy-back ride, the disadvantage is that the primary payload decides the final orbit characteristics. An estimation of the market demand for small satellites as in [1] is shown in Fig.1 (left), indicating a clear interest up to 2020 and beyond.
Unless some other smart ideas emerge, the independent access to space for the small and micro-satellites will remain a distant dream. In order to take advantage of the potential currently present in

the market and the offer-demand situation, a vehicle that would provide the transportation for such technologies into space and which would transform the accessibility to space as being more affordable is considered as a relevant business case, with an economically viable background. Such a vehicle that could meet the market requirements is envisioned as a small launcher. The approach to design-develop such a launcher must take in account the cheapest-available technologies in order to achieve operability "soonest".
Worldwide, there are about 60 on-going initiatives, some of them presented in [1] (right) that try to design and fly a small launcher and the recent events show that no space program is immune to the challenges that are to be overcome in order to achieve a successful flight. Currently, Rocket Lab's Electron performed the most promising orbital flight test, while Vector Space's Vector-R performed two suborbital test flights. It is worth to be mentioned that US (Vanguard), France (Diamond) and India (SLV-3) successfully placed satellites with what is considered today a micro-launcher.

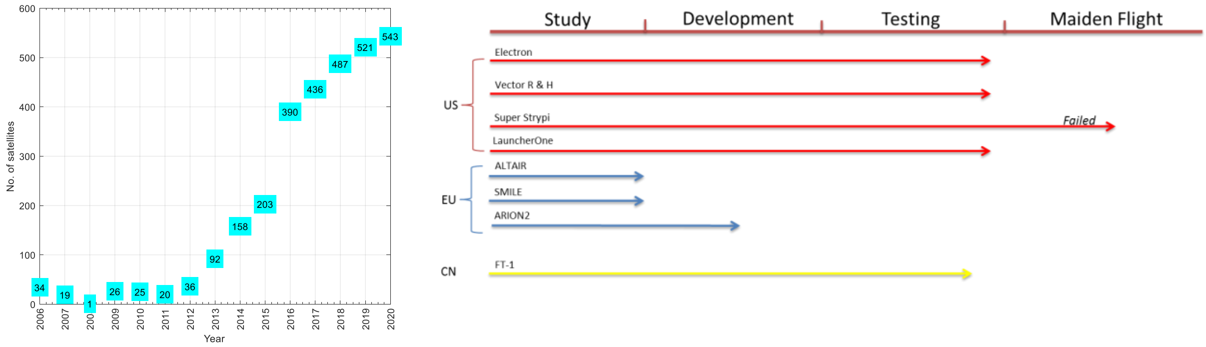

Figure 1 Small satellite market demand [1] (left), current competitors timeline (right)

The main challenging task with the design of an efficient small launcher is the cost reduction. The complexity of a launch vehicle is not scalable; therefore, the reduction of the GLOW does not transpose linearly as a reduction in recurring and nonrecurring costs. However, GLOW reduction is the most affordable approach in a preliminary design activity, as in [5], [6].
The Phase O/A Feasibility Study of a Microlauncher has a set of requirements as to identify launcher concepts able to deliver the requested performances by the market. The overall philosophy is to develop a fully functional launcher but broken-down in sub-system analysis for potential use in other markets or developments of other demonstrator technologies that could show a significant cost reduction at system or technological level.
The main technical objective of the study is to design a microlauncher Demonstrator, while making the assessment of the available concepts and technologies, ensuring low cost operations. Also, low TRL concepts and technologies, but with serious growing potential within a frame of 5 years are also a target to be considered in the study.

2 SYSTEM DESIGN PHILOSOPHY

2.1 Requirements

The main requirements that are taken into account for the design of the Microlauncher are listed in Table 1.

Table 1: Main Requirements for the microlauncher

Type	Category	Requirement
Mission Requirements	LEO Mission	The primary mission is to place a satellite with a mass of 100 kg on a Low Earth Orbit starting with 300 km in altitude up to 1000 km.
	Observation Mission	The Microlauncher will be able to deliver to LEO observation satellites.
	Launch rate	The Microlauncher will be able to perform a launch per month.

Performance Requirements	Orbit injection accuracy	The microlauncher is designed to inject the payload with the following accuracy: semi-major axis – less than 2.5 km, eccentricity - less than 10-4 deviation, inclination – less than 0.04 deg, ascending node – less than 0.03 deg.
	Deorbitation	The launch vehicle is designed to be in line with European standards for Debris Control.
Design Requirements	Launch base	The vehicle should be able to Lift-Off from a ground-based launch pad.
Interface Requirements	Fairing	The fairing is designed to accommodate a satellite with a volume of D = 1.5 m and H = 2 m.

2.2 MDO Approach

In the context of this project, INCAS developed a MDO software whose purpose is to find feasible designs for the Microlauncher based on different design parameters. The software employs an intrinsic MATLAB genetic algorithm in order to search for the global optimum for the launch vehicle design problem. Based on several user-defined options related to vehicle configuration, propulsion weights, etc. the genetic algorithm initiates an inner iteration loop where evaluates at every iteration the objective function (or the fitness function/performance index) attempting to minimize it. The objective function evaluation is a multistep process where information is passed between several modules associated with different disciplines like *Weights & Sizing, Propulsion, Aerodynamics and Trajectory*. The propulsion module is considering a 2D grain contour for the solid option, while analytic correlations are available for the liquid version [5]. The degree of confidence is smaller for the solid propulsion module versus the liquid one. The aerodynamic module is based on semi-empirical models which is validated against similar commercial tools. Although the certainty degree of the aerodynamic module is modest when compared to high fidelity results as from CFD, it is the only time efficient and robust way to go in optimization processes. An iteration of the genetic algorithm consists of selecting a new set of decision variables within an a priori defined set of bounds (optimization variables) based on the user inputs and the history of these variables throughout the previous iterations. Using this set of variables as input, each of the previously mentioned modules are evaluated sequentially and, based on their output, the objective function for the current iteration is computed. Due to the high nonlinearity of the problem, inequalities and equalities constraints cannot be fed directly into the genetic algorithm and they have to be included in the objective function as well together with the Gross Lift-Off Weight (GLOW) of the launcher computed in the *Weights & Sizing* module [5], [6]. At this point, the genetic algorithm proceeds to the next iteration attempting to minimize the performance index. To be conservative, some safety margins have been considered, the most important one being increasing the payload capacity with 20%, thus the useful payload having a mass of 180kg. Trajectory module includes a three degrees of freedom model, considered to be adequate for optimization purposes. This is validated against the commercial code ASTOS.
The block scheme used in the development of the MDO tool is shown in Fig.2.

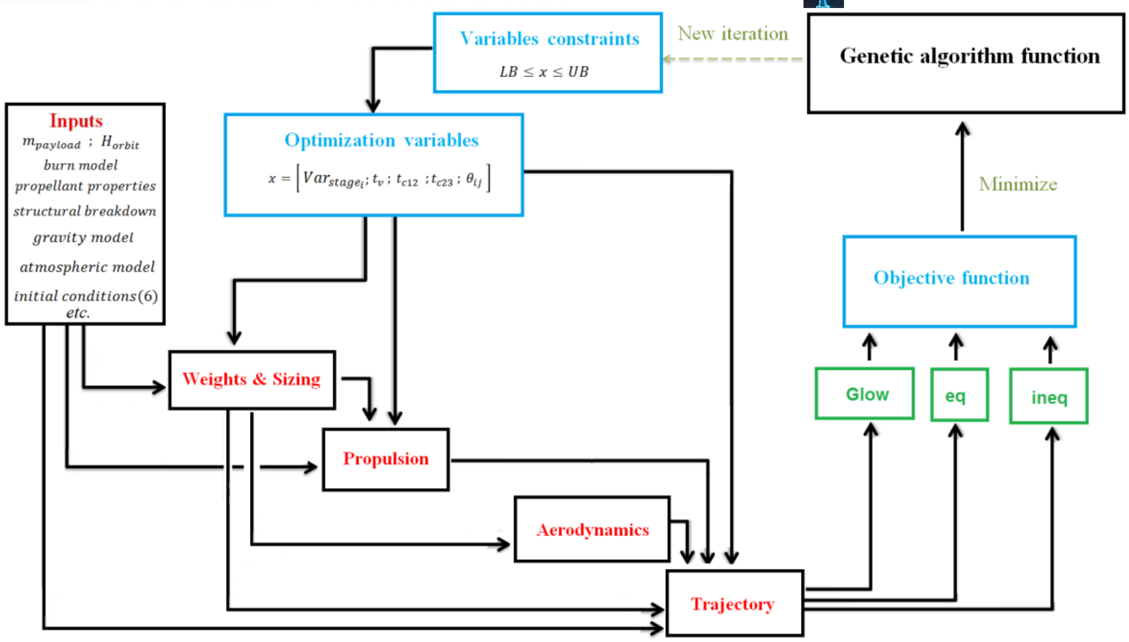

Figure 2 Block scheme of the MDO tool

2.3 Objective function definition

One of the most important functions that must be defined in order for the MDO to solve the complex iterative process is the objective function. Some aspects regarding this issue are presented in Eq.1 to Eq.7. The objective function for the optimization problem has the following form:

$$f = (GLOW + I_{to}) \cdot I_{pc} \tag{1}$$

where I_{pc} and I_{to} represent performance indices defined for the path constraints and the target orbit, respectively. It is intuitive to see that the objective function is constructed in such way that, when the performance index for the target orbit goes to zero, the performance index of the path constraints goes to one and the GLOW is minimized, an optimal solution is found.
The performance index for the target orbit is given by:

$$I_{to} = \sqrt{w_R (R - R_{target})^2 + w_V (V - V_{target})^2 + w_\gamma (\gamma - \gamma_{target})^2} \tag{2}$$

where w_R, w_V and w_γ represent weights chosen for the three parameters defining the target orbit. For the results presented in this report, these are set to:

$$\begin{aligned} w_R &= 1 \\ w_V &= 1 \\ w_\gamma &= 10 \end{aligned} \tag{3}$$

The performance index for the path constraints is defined according to:

$$I_{pc} = \prod_{i=1}^{N_{pc}} I_{pc_i} \tag{4}$$

where I_{pc_i} represent a performance index attributed to the i^{th} path constraint, i.e. for a constraint of the form:

$$n_a \leq n_{a\max} \tag{5}$$

the performance index will be given by:

$$I_{pc_i} = \begin{cases} 1, p \leq 1 \\ p, p > 1 \end{cases} \tag{6}$$

Here, p represents the percentage of the path constraint violation:

$$p = \frac{|n_a - n_{a\max}|}{n_{a\max}} \cdot 100 \tag{7}$$

3 MDO RESULTS

For this paper, the results provided by the in-house MDO for a three stage microlauncher are shown. Several propellant combinations have been studied, the following being analysed:

- HTPB 1912
- Lox + Kerosene
- Lox + Methane
- Lox + Ethanol
- HC + H2O2

The different combinations used in the MDO are set as an input. All studies configurations successfully complete the input mission, which is the insertion of a 180kg satellite in a polar orbit. The different architecture of the launchers is an output of the MDO tool developed. All feasible micro launchers obtained with the MDO tool developed are presented in Fig.3. Methane and Ethanol based launchers, are the longer ones, while the shortest ones are solid based launchers. Kerosene propelled are in between the two. Ethanol offers the lowest Isp enabling low performance index vehicles. Methane enables the highest performance index, followed by kerosene and solid.

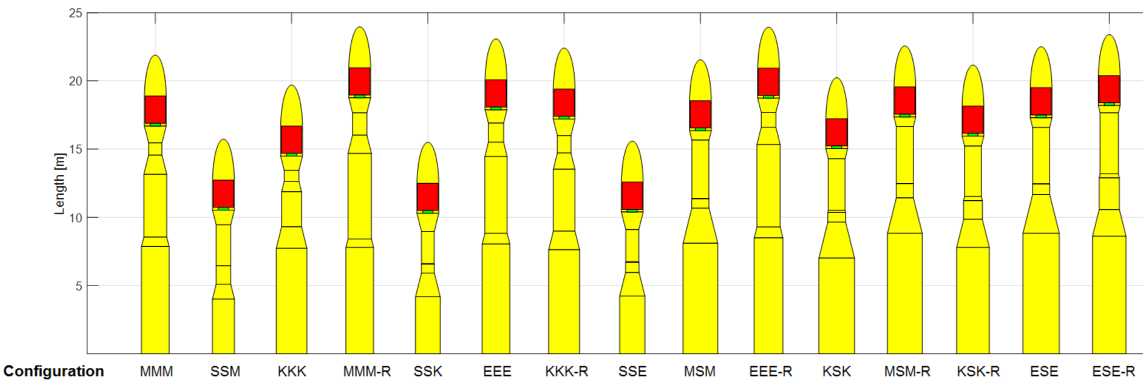

Figure 3 Feasible launchers – MDO output

The best two solutions are selected not only based on pure performance index, but also based on cost, technology maturity and available technologies. The two configurations are further optimized within the MDO, based on a more accurate propulsion model for the solid version. Thus, the final output of the MDO tool is presented in Fig.4. The constant diameter launcher (backup) is propelled by HC+H2O2, having the advantage of affordable, green fuel, more important in the future. The solid version (baseline) is adapted to the local industry capabilities and includes a Methane upper stage, with the potential to be replaced with a green propellant system.

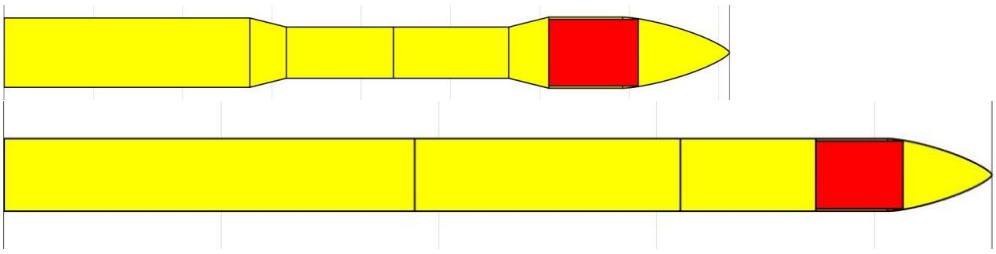

Figure 4 Best configurations from the MDO tool

Some of the most important characteristics of the launchers obtained with the MDO tool are presented in Table 2.

Table 2: MDO output

Characteristics	Values		Units
	Baseline	Backup	
Gross lift-off weight	19.17	22.72	[t]
Launcher length	16.24	22.69	[m]
Initial thrust to weight ratio	3.22	1.60	[-]
Total payload mass	335	335	[kg]
Useful payload mass	180	180	[kg]
Fairing mass	100	100	[kg]
Adapter mass	30	30	[kg]
Safety mass	25	25	[kg]

At convergence, the configurations studied in the MDO must minimize the objective function. The value of the performance index for the target orbit must be close to zero, admitting a small tolerance. Thus, the converged solution for the Baseline launcher is shown in Fig.5.

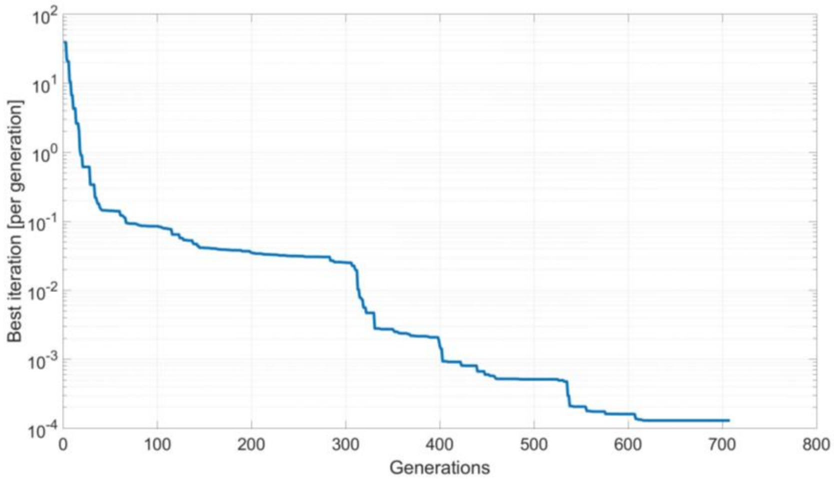

Figure 5 Baseline configuration - performance index I_{to}

For the Baseline configuration, the inertial velocity variation in time is shown in Fig.6, the FPA variation in time is shown in Fig.7 and the local altitude variation in time is shown in Fig.8. It can be observed that the mission requirements (orbital parameters) have been fulfilled. The optimal launcher mass time history is shown in Fig.9, while Fig.10 presents the orbit propagation in time indicating the circularity of the orbit. The difference between the apogee and the perigee is around 25km which corresponds to a 0.001 orbit eccentricity, in line with the mission requirements.

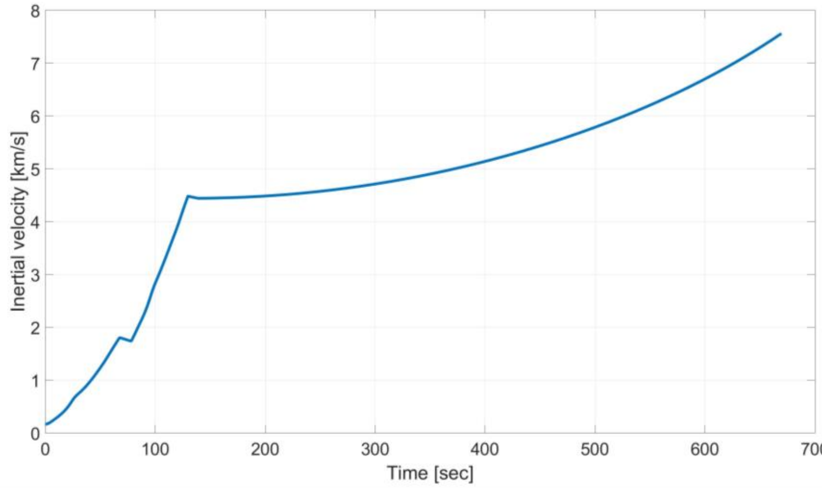

Figure 6 Baseline configuration - velocity vs. time

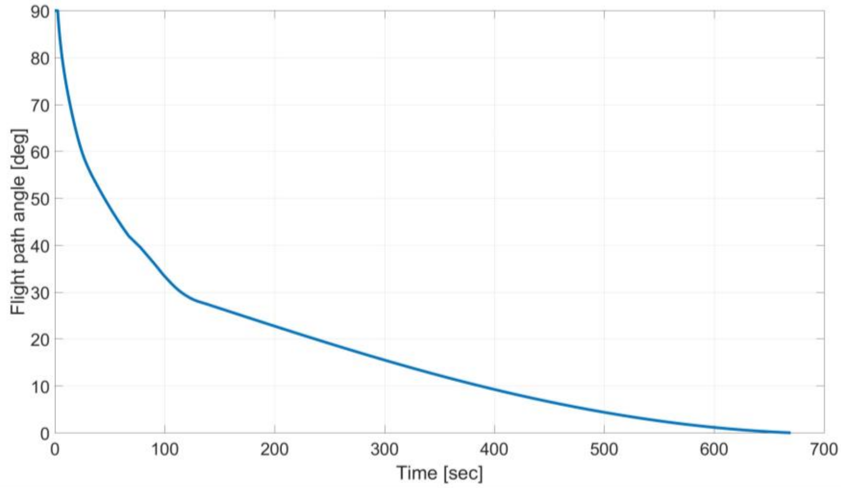

Figure 7 Baseline configuration - Fpa vs. time

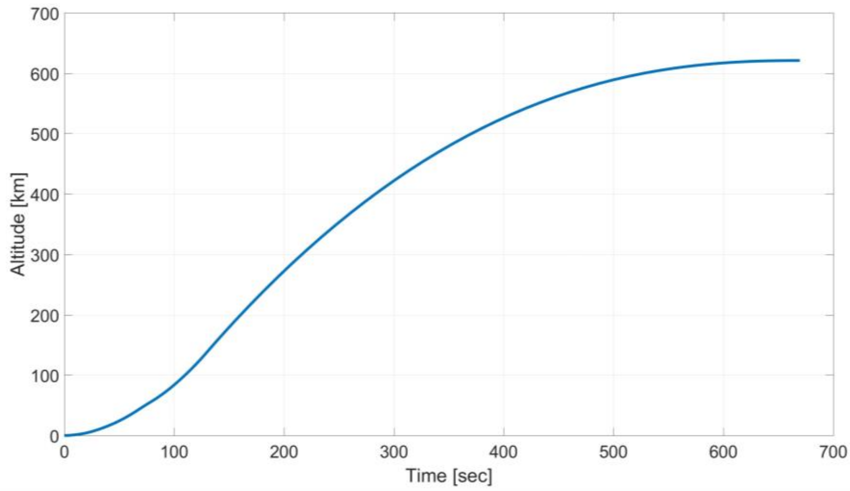

Figure 8 Baseline configuration - local altitude vs. time

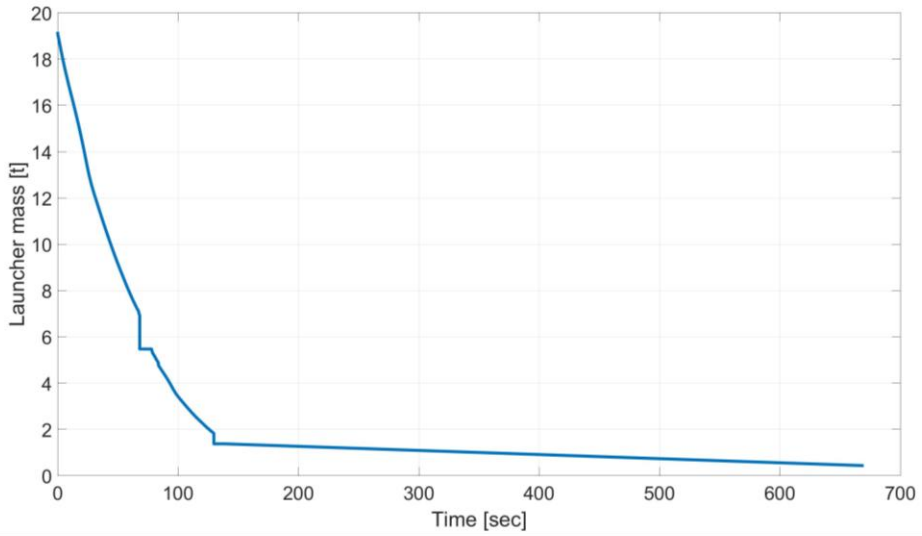

Figure 9 Baseline configuration launcher mass vs. time

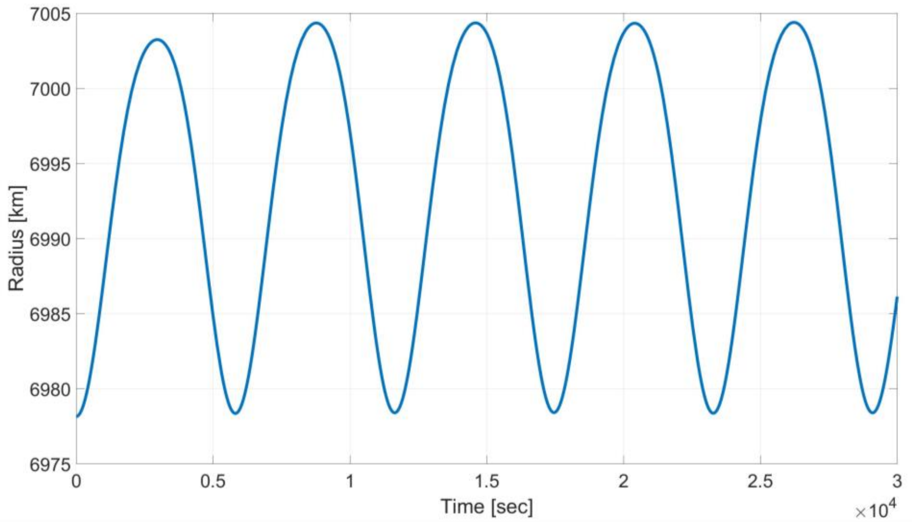

Figure 10 Baseline configuration - orbit propagation

4 REFINED CONFIGURATION

The output of the MDO tool is enough for a preliminary design of the launcher as in **Table 2**. To ensure that the solution obtained is a feasible one, detailed aspects must be addressed. Thus, a complex design process and stress analysis is performed. Solid motors for first and second stages are designed based on the MDO results. The more accurate thrust charts are included in the MDO tool as constraints, together with the updated mass breakdown and other elements in order the gain more confidence in the results. Short loop and full loop processes are realized so that the master mass breakdown can be obtained as seen in Fig.11. The master mass breakdown contains a more detailed view of all launchers systems and subsystems with realistic dimensions and masses.

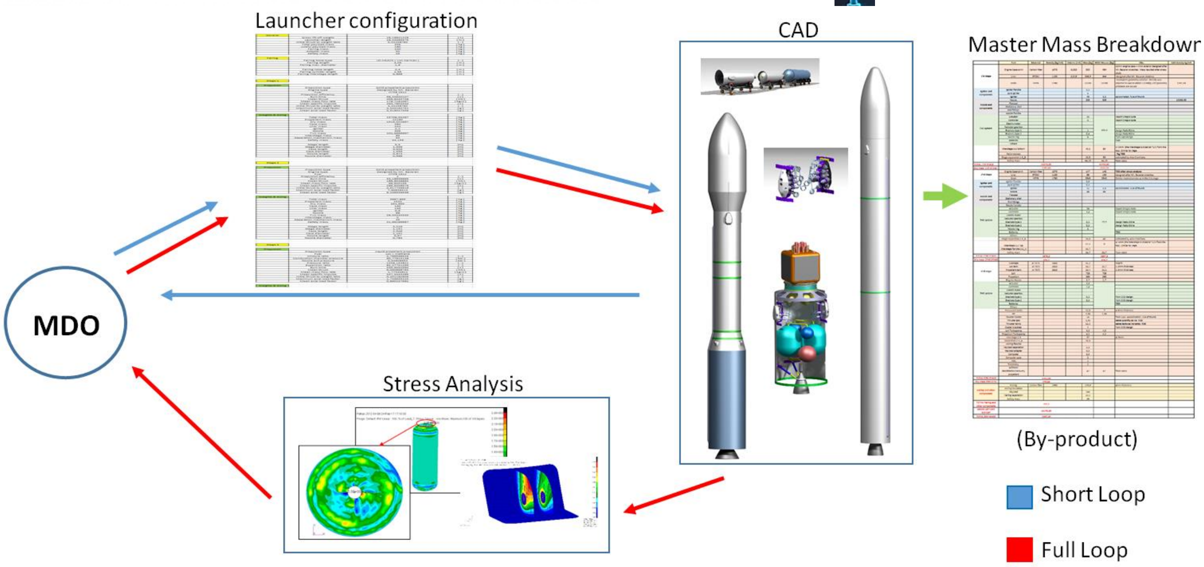

Figure 11 Configuration refinement process

Detailed 2D drawings of the refined launchers based on the MDO output are shown in Fig.12. Most of the launcher components have been designed in the MDO tool, but for some only a mass and dimension budget has been allocated and later defined in the full loop process. Also, for the baseline configuration the detailed CAD model can be observed in Fig.13.

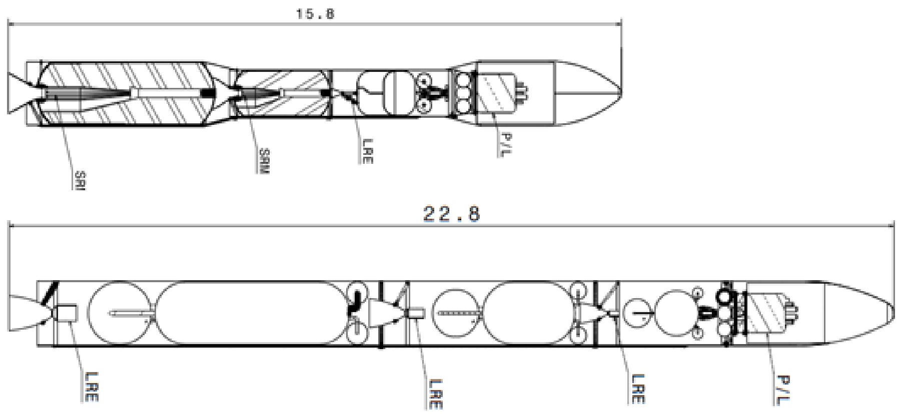

Figure 12 Baseline and backup refined configurations

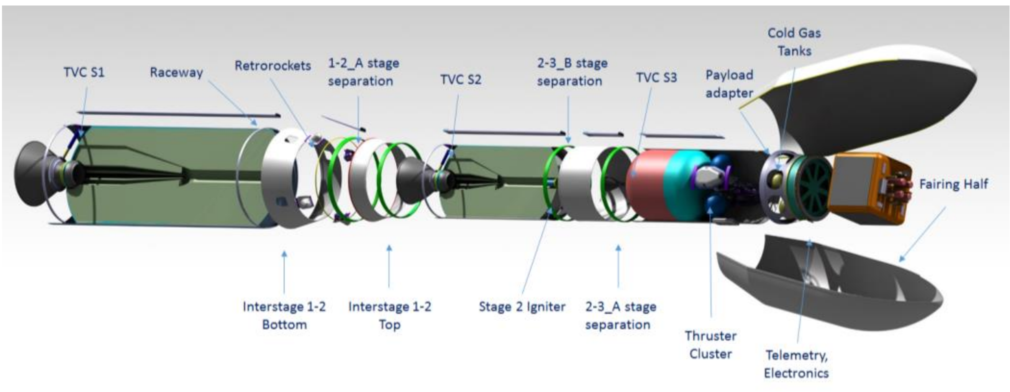

Figure 13 Baseline configuration detailed CAD

5 CONCLUSIONS

The paper presents an overview of the process of obtaining a suitable microlauncher for the imposed mission. The best way of designing the launcher was by using an MDO approach. An in-house tool has been developed and used to generate launcher configurations that respect the imposed mission.
The final eccentricity of the orbit has a 0.001 eccentricity which corresponds to a very good orbital insertion. For both the ascending phase and the orbital propagation a 3DOF dynamic models has been used.
The two best solutions were refined and used as baseline and a backup configuration. For the detailed design, a full loop process was used in which complex design and stress analyses were performed. The baseline launcher offers the advantage of a shorter development roadmap, important in getting a market share earlier, while the backup has more potential as mature product, requiring a higher investment and time in the development phase. However the upper stage, payload adaptor, avionics and actuators are planned to be common.
Future work is going to consider adding more parameters to the fitness function, especially number of stages, plus more advanced physical models for standard atmosphere and gravity. Also adding a lateral control is of interest for expanding the current MDO tool.

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