MDO approach for a two-stage microlauncher

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Abstract*: The paper focuses on the multidisciplinary optimisation and preliminary design of a two-stage microlauncher capable of inserting a small 50 kg payload into Low Earth Orbit. The microlauncher is obtained using a MDO approach, where the lightest configuration capable of reaching the target orbit is considered to be optimal. For this paper, the propulsion system of the microlauncher is based on a non-cryogenic bipropellant combination.*

Key Words: *microlauncher, MDO, liquid propellant*

1. INTRODUCTION

The primary mission of a microlauncher is to insert a small payload into Low Earth Orbit. This payload can consist of one or more satellites; one category of satellites that stands out is the CubeSats, because of their reduced dimensions and weight.

A preliminary design for such a microlauncher is of interest. To successfully accomplish this task, a multidisciplinary approach must be used. This is often realised with the aid of a multidisciplinary design optimisation (MDO) algorithm.

The structure of the MDO algorithm can vary from a development team to another. The block scheme of the MDO algorithm used in this paper is based on the one presented in paper [1], and adapted for the needs of the study. It is shown in Figure 1.

The complexity of the MDO is given by the four main disciplines that must be integrated: Weights & Sizing, Propulsion, Aerodynamics and Trajectory. The last of the disciplines that must be integrated, in this particular case the trajectory, dictates the amount of data needed from the other disciplines.

Using a 3DOF approach is justified as it provides an accurate orbit injection, as seen in [2], but also significantly reduces the complexity of the other integrated modules.

This approach can be seen in other small launcher initiatives, as the one described in [3].

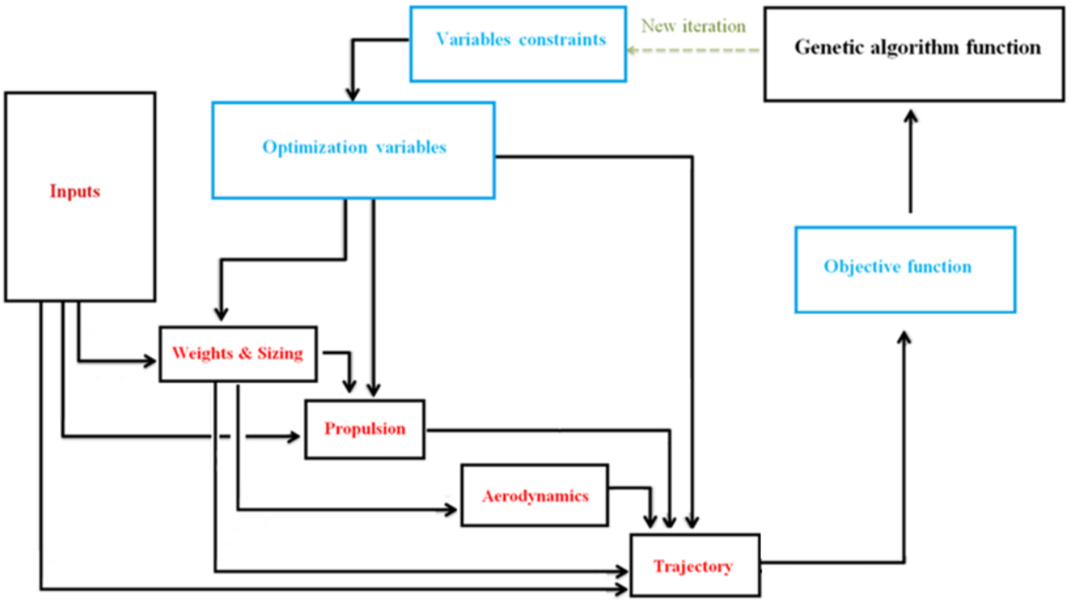

Figure 1 - Block scheme of the MDO algorithm

2. MDO BREAKDOWN

For the input module of the MDO algorithm the most important data are shown in Table 1. Other data are also declared in the input module, such as material properties, maximum load factors, fairing jettison condition, pumps and combustion efficiencies, acceptable expansion ratios and many more.

Table 1 - MDO inputs

Data	Value
Payload mass [kg]	50
Orbit altitude [km]	300
Orbit type	Circular, polar
Microlauncher architecture	Two stages, constant diameter First stage containing a fairing
Maximum stage diameter [m]	1
Propulsion system	$H_2O_2 + HC$
Launch site	Andøya Space Centre, Norway

The first step required in an iterative optimisation process is to define the optimisation variables vector. Based on this vector the preliminary design and performance analysis can be made.

Throughout the iteration process of the genetic algorithm (GA), the optimisation variables are updated based on an evaluation criterion, which in this specific case is the minimisation of a user-defined objective function.

For each of the two stages the following optimisation variables will be used: propellant mass, outer diameter, combustion chamber pressure, exhaust pressure, thrust/weight ratio at the start of the burn and the TVC deflection angle, considered to be constant for the entire burn period. In addition to these 12 variables another 2 are used to define the flight sequences in the ascent phase.

These are: the vertical ascent time after lift-off and the coasting time between the first stage separation and the ignition of the second stage.

For the current study, a total of 14 optimisation variables have been used. A search space must be defined so that the genetic algorithm can select the population for the optimisation variables vector.

An advantage of using a genetic algorithm is that the search space can be very vast. For example, the first stage propellant mass is user bounded between 1 and 30 tons, at convergence, the MDO selecting its optimal value.

The optimisation variables constraints used in this study are shown in Table 2. To ensure a technological simple output design the chamber pressure has been limited to 60 atmospheres and the TVC deflection angle to 7 degrees.

Table 2 - Optimisation variables bounds

Lower bound	**Optimisation variable**	**Upper bound**
1	Stage 1 propellant mass [t]	30
0.8	Stage 1 diameter [m]	1
40	Stage 1 chamber pressure [atm]	60
0.3	Stage 1 exhaust pressure [atm]	1
0.1	Stage 1 Thrust/Weight ratio [-]	5
0	Stage 1 TVC deflection angle [deg]	7
0.1	Stage 2 propellant mass [t]	15
0.8	Stage 2 diameter [m]	1
40	Stage 2 chamber pressure [atm]	60
0.01	Stage 2 exhaust pressure [atm]	0.15
0.4	Stage 2 Thrust/Weight ratio [-]	1.5
0	Stage 2 TVC deflection angle [deg]	7
2	Vertical ascent time [s]	10
2	Coasting phase [s]	500

In the Weights & Sizing module the individual components of the launcher are defined and their masses and dimensions computed. For this module, analytical and semi-empirical models are used.

The inputs for this module are coming both from the optimisation variables vector and the input module.

Knowing the propellant mass and computing the structural mass, the entire stage mass can be obtained by adding up all of the individual components.

The mass breakdown scheme used in this paper is shown in Figure 2. Safety margins were used for both mass and length, the values being 5% for the structural mass and 10% for the stage length.

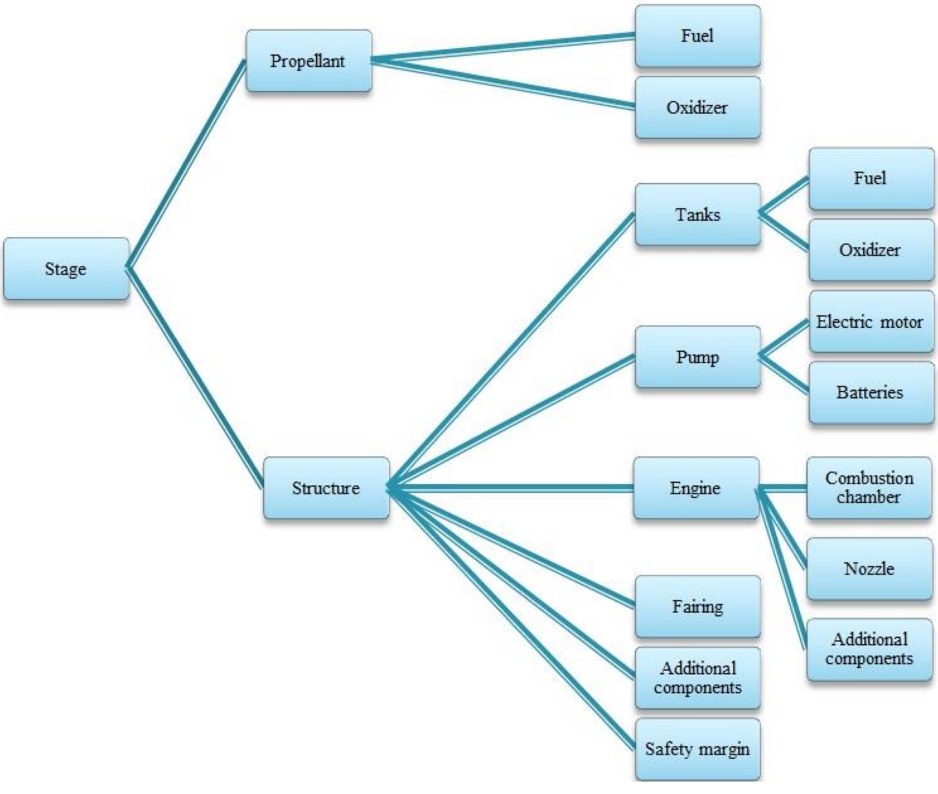

Figure 2 - Stage mass breakdown

In the Propulsion module, the thrust curve of the microlauncher is computed using the following formula:

$$T = q \cdot g_0 \cdot I_{sp} \tag{1}$$

The specific impulse I_{sp} is computed from:

$$I_{sp} = \eta_n \cdot \frac{C^*}{g_0}\left(\gamma\sqrt{\left(\frac{2}{\gamma-1}\right)\cdot\left(\frac{2}{\gamma+1}\right)^{\left(\frac{\gamma+1}{\gamma-1}\right)}\cdot\left(1-\left(\frac{P_e}{P_c}\right)^{\left(\frac{\gamma-1}{\gamma}\right)}\right)}+\frac{\varepsilon}{P_c}\left(P_e-P_a\right)\right) \tag{2}$$

Here: q, g_0 are the propellant mass flow rate and the gravitational acceleration at the sea level; η_n is the nozzle efficiency, C^* is the propellants characteristic velocity, γ is the isentropic coefficient at the throat and ε is the nozzle expansion ratio; P_c represents the chamber pressure, P_e represents the exhaust pressure and P_a represents the atmospheric pressure.

In the Aerodynamics module, the drag coefficient of the microlauncher is computed. This is the only coefficient needed because of the 3DOF model used in the Trajectory module.

The methods used to assess the aerodynamic performance are based on linearized models and therefore, the superposition principle can be applied.

Thus, the launcher can be broken down into individual components such as: fairing, stages, interstages and transitions.

To compute the total drag coefficient of the launcher, all individual contributions are scaled to the global reference area and then added.

$$C_{d_{launcher}} = \sum C_{d_{component}} \tag{3}$$

For the individual components drag coefficient, the model shown in [4] is used. Here, the aerodynamic drag is divided into 3 main contributions: body pressure drag, friction drag and base drag.

For each type, combinations of analytical and semi-empirical models are used to assess the drag coefficient for Mach numbers up to 20.

The Trajectory module is the last one to be assessed and here, the flight performance of the microlauncher are computed using a 3DOF model.

The following equations of motions are implemented:

$$\begin{aligned} M\dot{v} = (T\cos\delta_T - D) - Mg_c \sin\gamma + Mg_\delta \cos\gamma \cos A - \\ - M\omega^2 r \cos\delta(\cos\gamma\cos A \sin\delta - \sin\gamma\cos\delta) \end{aligned} \tag{4}$$

$$\begin{aligned} Mv\cos\gamma\dot{A} = M\frac{v^2}{r}\cos^2\gamma\sin A\tan\delta - Mg_\delta \sin A + M\omega^2 r\sin A\sin\delta\cos\delta - \\ - 2M\omega v(\sin\gamma\cos A\cos\delta - \cos\gamma\sin\delta) \end{aligned} \tag{5}$$

$$\begin{aligned} Mv\dot{\gamma} = T\sin\delta_T + M\frac{v^2}{r}\cos\gamma - Mg_c\cos\gamma - Mg_\delta\sin\gamma\cos A + \\ + M\omega^2 r\cos\delta(\sin\gamma\cos A\sin\delta + \cos\gamma\cos\delta) + 2M\omega v\sin A\cos\delta \end{aligned} \tag{6}$$

Here M, g_c, g_δ and ω are mass, gravitational acceleration (radial and tangential), and the Earth angular velocity; D represents the drag force, T represents the thrust, while δ_T represent the TVC angle in the vertical plane.

The kinematic equations described in [5] are used in the MDO and presented here:

$$\dot{r} = v\sin\gamma \tag{7}$$

$$\dot{\delta} = \frac{v}{r}\cos\gamma\cos A \tag{8}$$

$$\dot{\lambda} = \frac{v\cos\gamma\sin A}{r\cos\delta} \tag{9}$$

Kinematic trajectory equations are expressed using a set of spherical coordinates: r, δ, λ denoting the radius, geocentric latitude and longitude.

The velocity vector is expressed in terms of spherical coordinates: v, γ, A which represent the relative velocity magnitude, the flight path angle and the velocity azimuth, computed in the local horizon frame, ($Oxyz$).

For the trajectory computations, a planet centred, rotating frame ($SXYZ$) system is used and shown in Figure 3.

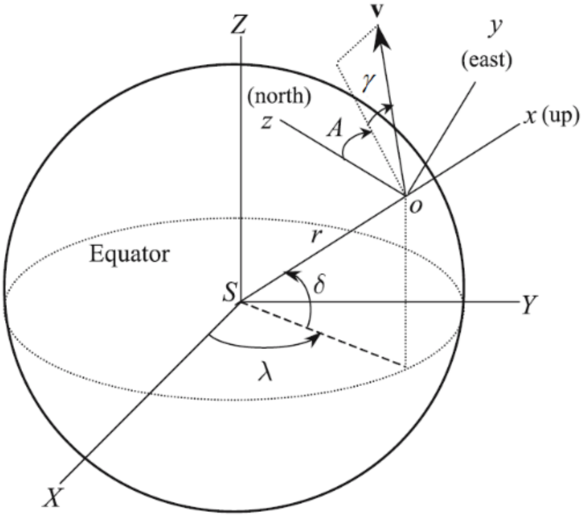

Figure 3 - Planet-fixed and local horizon frames

The MDO is optimising the launcher by means of minimising the objective function. This means that the objective function must be carefully constructed.

$$f_{obj} = (GLOW + I_{to}) \cdot I_{pc} \tag{10}$$

For this study we have used a simple objective function, consisting of 3 parameters. The first one is the GLOW which means the Gross Lift-Off Weight and is the mass of the launcher at lift-off.

The target orbit index measures the accuracy of the payload insertion by using the following function:

$$I_{to} = \sqrt{w_r(r - r_{target})^2 + w_v(v - v_{target})^2 + w_\gamma(\gamma - \gamma_{target})^2} \tag{11}$$

Here w_r, w_v, w_γ are the weights associated with the radius, velocity and flight path angle of the target orbit.

Another parameter which has been implemented in the objective function is the path constraint index, which indicates if the user imposed constraints are respected.

$$I_{pc} = \prod_{i=1}^{N_{pc}} I_{pc_i} \tag{12}$$

$$I_{pc_i} = \begin{cases} 1, \text{if constraint is respected} \\ > 1, \text{if constraint is violated} \end{cases} \tag{13}$$

Some examples of constraints that are implemented in the MDO are maximum load factors, maximum nozzle expansion ratios and no internal component clashes. At convergence, the objective function should be minimal and the GLOW should tend to its minimal value.

For an ideal insertion the target orbit index is equal to 0 and if all of the constraints are respected the path constraint index is 1.

3. RESULTS

It is of interest to start by presenting the convergence history. The objective function convergence is presented in Figure 4. It can be seen that the MDO converged in around 1250 generations. With a starting population of 200 individuals, the total number of iterations the algorithm needed was around 250 thousands. Also, from Figure 5 it can be seen that the target orbit index has decreased from 10^4 at the start to 10^{-5} at convergence, this meaning that the payload has successfully reached the target orbit.

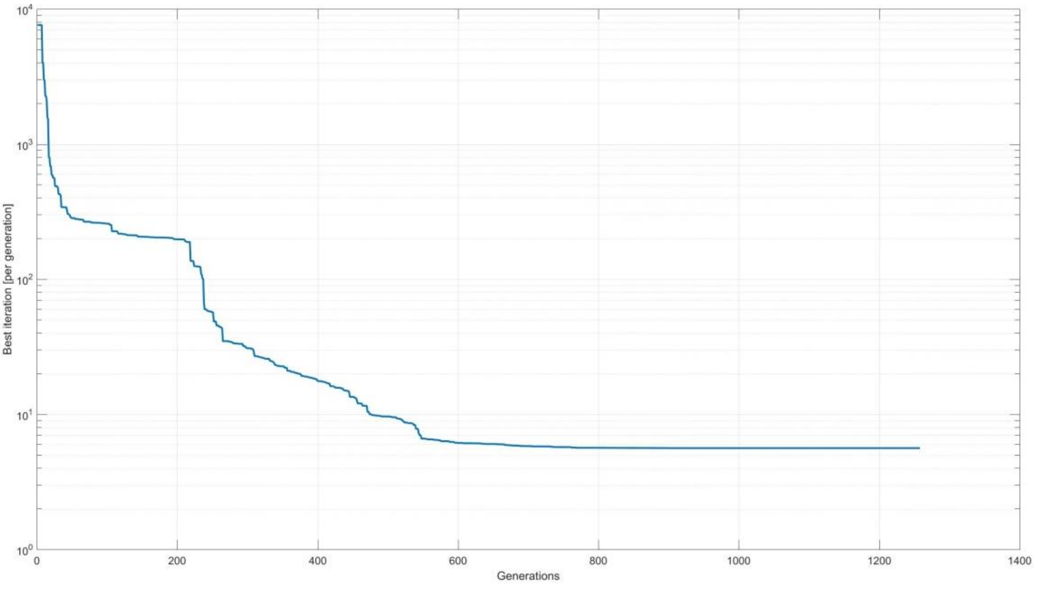

Figure 4 – Objective function convergence

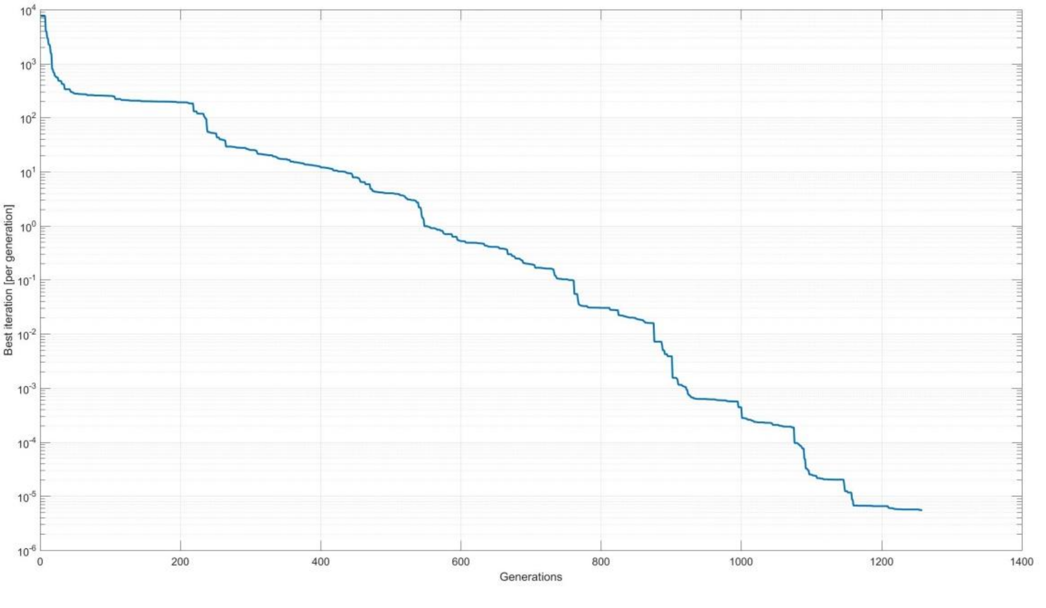

Figure 5 – Target orbit index convergence

The optimal values for the optimisation variables are shown in Table 3. The optimal diameter for the microlauncher capable of inserting the 50 kg payload in a 300 km LEO orbit is around 0.95 m.

For the first stage, 4.1 tons of propellant is used, while the second stage uses 830 kg of propellant.

The global characteristics of the microlauncher are shown in Table 4. The lift-off mass of the launcher is 5.62 t. This is broken down into a 4.62 t first stage, a 0.95 t second stage and a 50 kg payload. The total length of the microlauncher is 12.72 m with a constant outer diameter of 0.95 m.

Some of the propulsion system characteristics are shown in Table 5. It can be seen that the first stage has a lower burn time than the second one, but significantly more thrust being produced.

The mean specific impulse is lower for the first stage, mainly because of the dense atmosphere the launcher is flying during that phase. The second stage burn is in near vacuum conditions, the selected non-cryogenic liquid bipropellant combination producing a mean specific impulse of approximately 316 s.

Table 3 - Optimisation variables

Optimisation variable	**Value**
Stage 1 propellant mass [t]	4.10
Stage 1 diameter [m]	0.95
Stage 1 chamber pressure [atm]	60
Stage 1 exhaust pressure [atm]	0.54
Stage 1 Thrust/Weight ratio [-]	2.19
Stage 1 TVC deflection [deg]	2.15
Stage 2 propellant mass [t]	0.83
Stage 2 diameter [m]	0.95
Stage 2 chamber pressure [atm]	60
Stage 2 exhaust pressure [atm]	0.04
Stage 2 Thrust/Weight ratio [-]	0.77
Stage 2 TVC deflection angle [deg]	3.06
Vertical ascent time [s]	2.52
Coasting phase[s]	3.34

Table 4 – Global characteristics

Characteristic	**Value**
GLOW [t]	5.62
Stage 1 mass [t]	4.62
Stage 2 mass [t]	0.95
Length [m]	12.72
Diameter [m]	0.95
Payload [kg]	50

Table 5 – Propulsion characteristics

Characteristic	**Value**
Stage 1 burn time [s]	84.43
Stage 2 burn time [s]	339.14
Stage 1 mean thrust [kN]	131.51
Stage 2 mean thrust [kN]	7.64
Stage 1 mean I_{sp} [s]	275.83
Stage 2 mean I_{sp} [s]	316.5
Stage 1 Thrust/Weight at start [-]	2.19
Stage 2 Thrust/Weight at start [-]	0.77

The preliminary design of the microlauncher obtained from the MDO algorithm can be clearly seen in Figure 6.

It has a two-stage, constant diameter architecture, where the first stage can be used as an independent launcher.

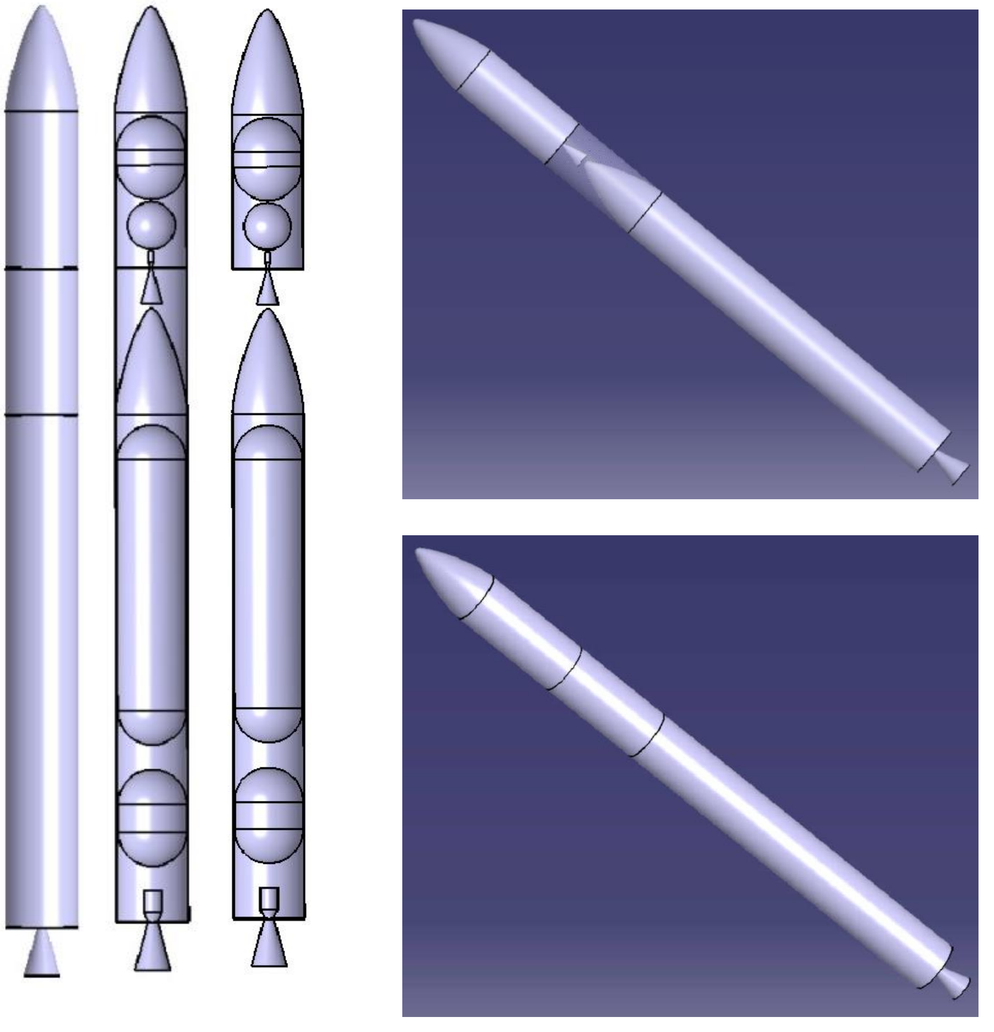

Figure 6 – Microlauncher geometry

The paper now presents trajectory output plots of interest. In Figure 7, the distance between Earths centre and the launcher position is plotted with respect to the flight time of the mission. It can be seen that the duration of the mission is close to 7 minutes.

Figure 8 shows the velocity of the launcher. Here, the first stage separation can be clearly seen, because for a short period of time the velocity of the launcher does not increase, in fact, it slightly decreases because of the drag and gravitational force acting on the launcher. The optimal duration of the coasting phase between the first stage separation and second stage ignition is 3.3 seconds.

In Figure 9 the flight path angle of the microlauncher can be seen throughout its entire mission. At lift-off and for the first 2.5 seconds the launcher is flying vertically, but after this phase the constant 2.15° TVC angle is used up until stage separation.

At the end of the ascent phase, the flight path angle has a value of $5 \cdot 10^{-7}$, corresponding to an almost perfect circular orbit.

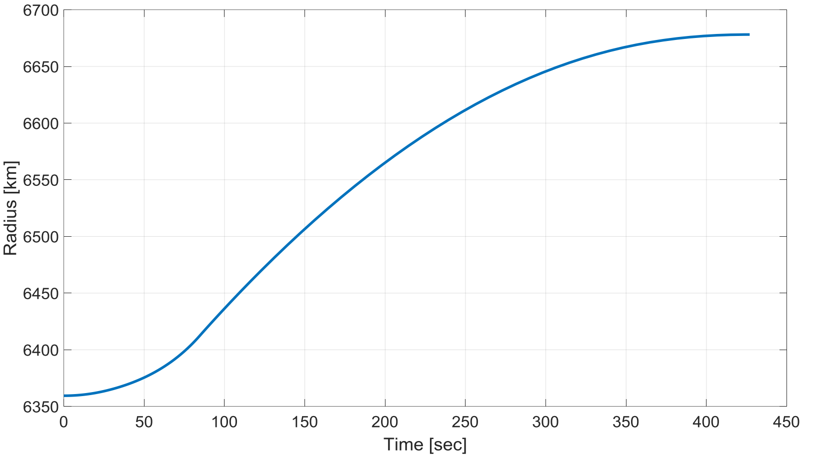

Figure 7 – Microlauncher radius vs. time

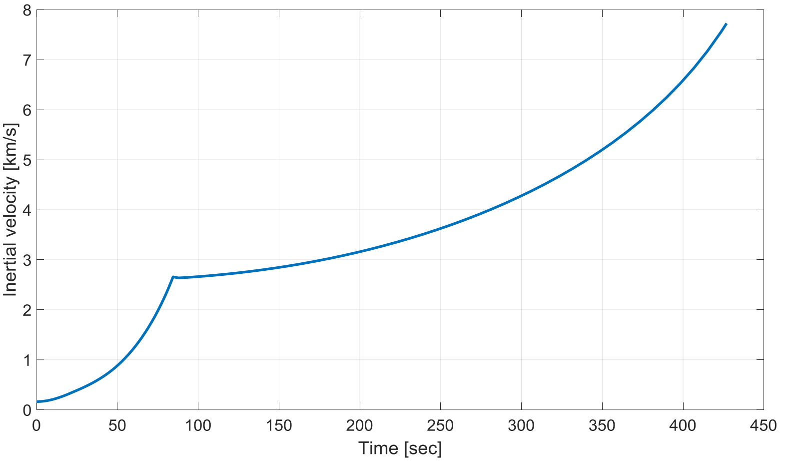

Figure 8 – Microlauncher velocity vs. time

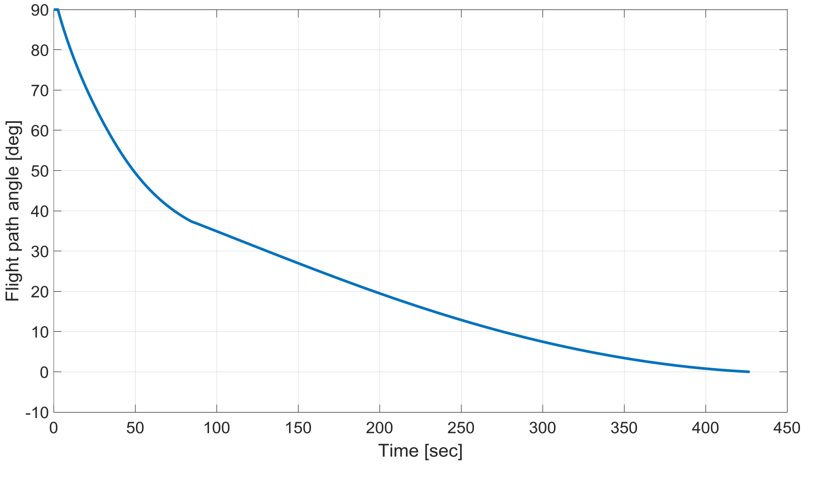

Figure 9 – Microlauncher flight path angle vs. time

The mass of the microlauncher can be seen in Figure 10. It can be clearly seen that for the first stage the propellant mass flow is significantly greater than for the second stage. The first stage burns 4.1 tons of propellant in 84 seconds, while the second stage burns 830 kg of propellant in 339 seconds. The first stage separation is at 52 km altitude.

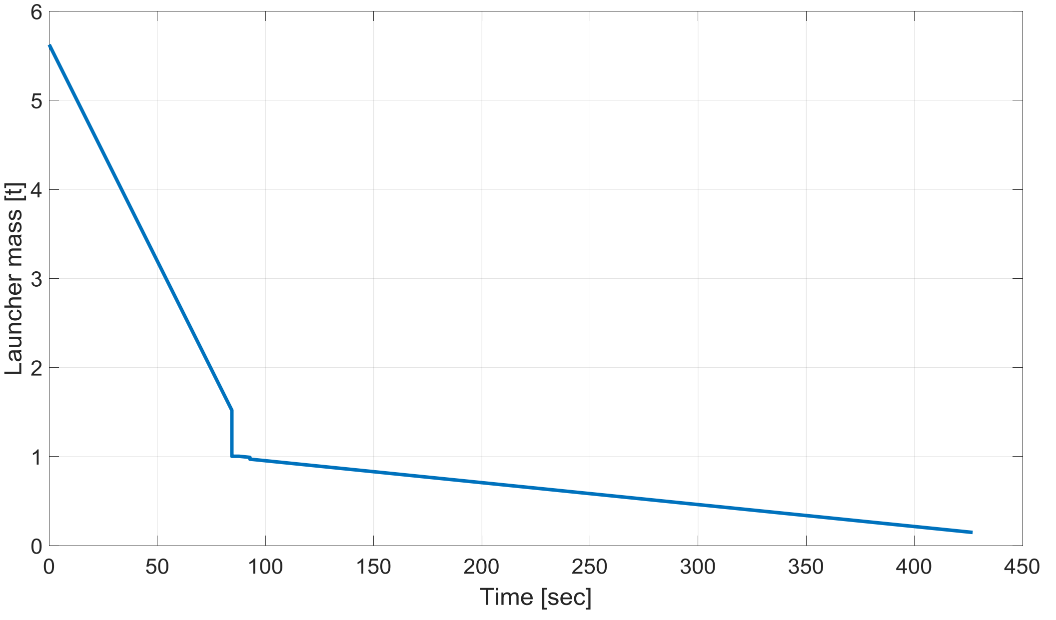

Figure 10 – Microlauncher mass vs. time

In Figure 11, the microlauncher acceleration is plotted. The mean acceleration for the first stage is around 3 g, while the maximum is 8.3 g at the end of the first stage burn. This is explained by the fact that the launcher mass is decreasing with time, while the thrust is not changing drastically during the first stage burn phase.

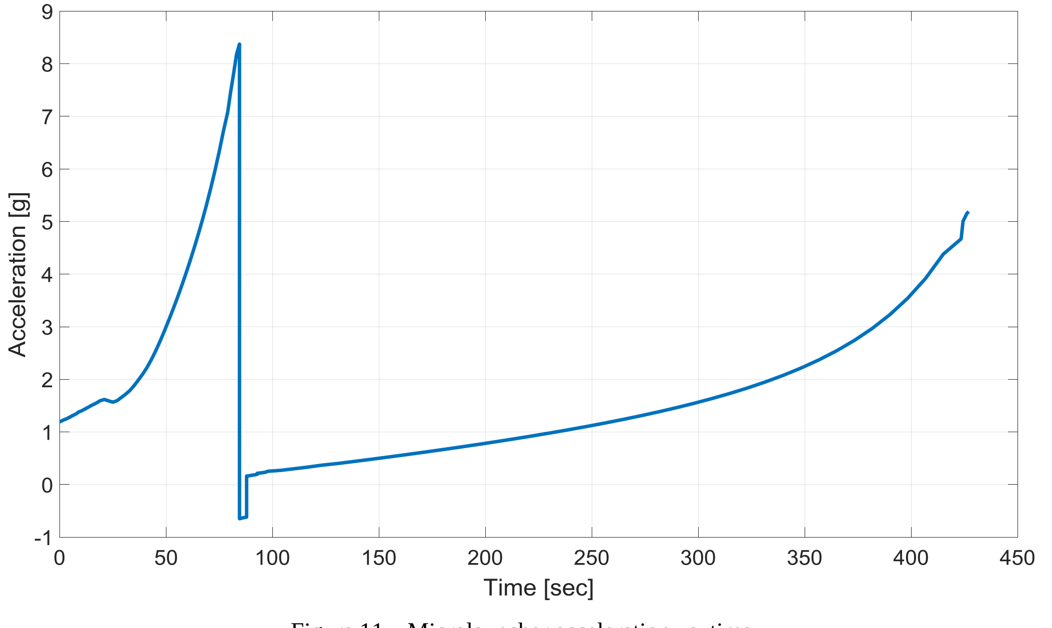

Figure 11 – Microlauncher acceleration vs. time

4. CONCLUSIONS

The paper presents the multidisciplinary optimisation and preliminary design of a two-stage microlauncher capable of inserting a small 50 kg payload into a 300 km altitude Low Earth Orbit. The structure of the algorithm used has been presented and the major elements have been detailed. The optimal solution obtained is a 5.62 t microlauncher. The architecture of the microlauncher allows the utilisation of the first stage as an independent suborbital launcher.

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