Parametric analysis of a two-stage small launcher using a MDO approach

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***Abstract**: The paper presents the influence of the main design requirements on the optimal two-stage small launcher configuration. The influence of target orbit altitude, payload mass, typical missions, and the type of liquid propellant (oxidizer - fuel pair) used on the launcher characteristics is quantified. The launchers are obtained using a multidisciplinary design optimisation (MDO) algorithm, where the lowest mass configuration capable of accurately inserting the payload into the target orbit is considered to be optimal.*

***Key Words:** parametric analysis, small launcher, multidisciplinary design optimisation*

1. INTRODUCTION

The current context is favorable for the development of small satellite launchers, with a worldwide increase in the resources allocated to space programs (development of satellites and launch vehicles). Due to the miniaturization of components and systems, there is a constantly growing market for compact satellites under 250 kg, which attracts an increased demand for small launchers dedicated to them [1], [2]. One of the quickest and most efficient way to obtain an overall efficient launcher is by using a multidisciplinary design optimisation (MDO) approach. The tool used for this study is based on previous work elaborated in [3], [4], [5], [6], [7], [8] and [9], the block scheme of the MDO algorithm being shown in Figure 1. In addition to the preliminary design of the launcher, the mission profile is also optimized, imposing a reference trajectory for the launcher to follow.

The small launcher optimisation is done by obtaining an optimisation variables vector, following the use of a solution selection and advancement algorithm based on the evaluation of the objective function. The solution is considered optimal at the convergence of the MDO algorithm, when the objective function has not improved after a specified number of iterations. The choice of optimisation variables is made in accordance with the launcher requirements and its architecture. The launcher is then completely defined through them and the global input data using the mathematical models integrated inside the main MDO blocks/modules. The disciplinary analyses are done in a cascade sequence, the core of the program being made up of the following 4 main modules: *Weights and Sizing*, *Propulsion*, *Aerodynamics* and *Trajectory*. The mathematical models used for each of the main modules are independent of others, thus a total of 4 individual codes have been developed, which, after validation, were incorporated into the multidisciplinary optimization code. Along with the 4 main modules listed above, within the architecture of the multidisciplinary optimization algorithm, there are also the following secondary modules: *Requirements and input data*; *Optimization variables*; *Objective function*; *Selection algorithm*.

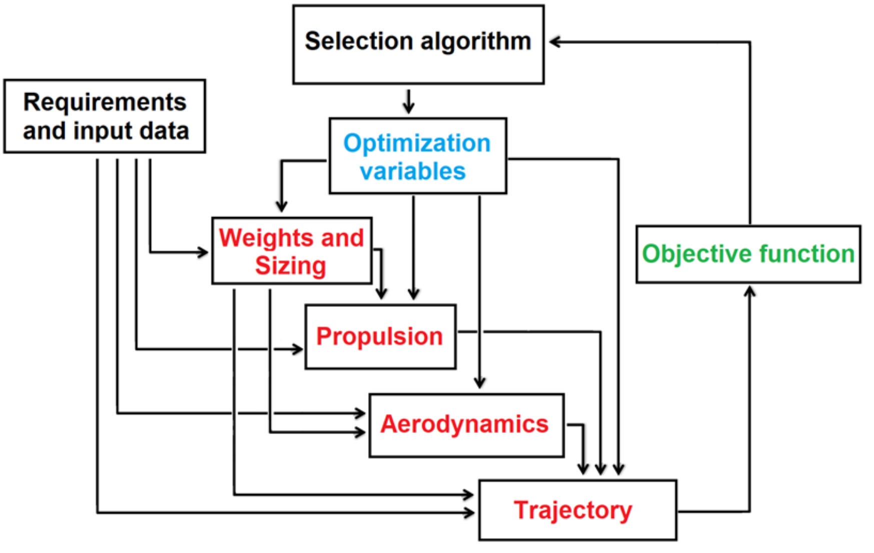


Figure 1 – Block scheme of small launcher MDO tool

The *Weights and Sizing* module implements a bottom-up approach, the dimensions and masses of the launcher components being individually calculated, as depicted in Figure 2 for a n-stage small launcher.

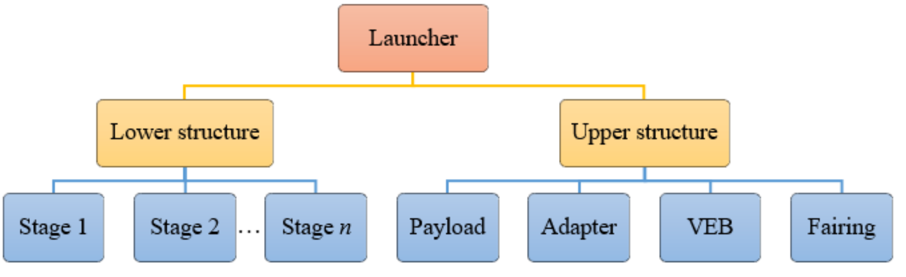


Figure 2 – Launcher breakdown scheme

For the upper structure, the component modelling is done as follows: the payload is considered input data; the adapter is estimated using a semi-empirical model [10]; the Vehicle Equipment Bay (VEB) is estimated using a semi-empirical model [11]; the fairing is modelled using the following formula:

$$M_{fairing} = 7.12 \cdot S_{lateral} \tag{1}$$

where $S_{lateral}$ is the fairing lateral surface area (measured in m^2).

For the lower structure modelling, the masses and dimensions of each stage are calculated individually, their contributions being latter summed up. For stages which incorporate a liquid propellant engine, the breakdown scheme implemented is shown in Figure 3, a visual representation of the main components (and interior planning) being observed in Figure 4.

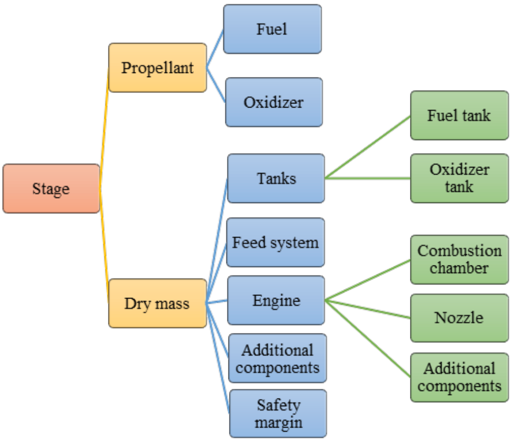


Figure 3 – Stage breakdown scheme (liquid propellant engine)

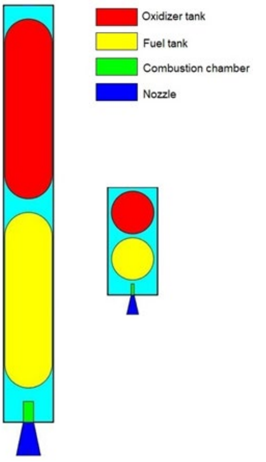


Figure 4 – Stage main components

Component modelling is done as follows: the propellant is considered an optimisation variable; the oxidizer and fuel are computed based on the mixture ratio [7]; the tanks are estimated using analytical models [11], [12], [13]; the feed system is based on mechanical turbopumps and is estimated using semi-empirical models [12], [13]; the combustion chamber is estimated using semi-empirical models [12], [13]; the nozzle is estimated using an analytical model [12]; the engine is estimated using semi-empirical models [12], [13]; the additional components are estimated using a semi-empirical model [12]. A dry mass safety margin of 5% and a length safety margin of 10% are also implemented to boost the overall confidence of the MDO assessment (it can be lowered or set to 0 in the detailed phase of launcher design).

Additional details regarding the *Weights and Sizing* module are presented in paper [9]. Here, the mathematical models used for weights and sizing assessment are validated with the aid of 8 launcher stages of interest reference data. An average error of 6.1% was found in stage dry mass estimation and 5.4% in stage length estimation, corresponding to a mean average model error of 5.8%.

The *Propulsion* module calculates the propulsive performance of the liquid propellant engines, the thrust curves for each constituent stage of the launcher being here generated. Thrust (T) assessment uses the classical, analytical formulation:

$$T = q \cdot g_0 \cdot I_{sp} \tag{2}$$

where q is the propellant mass flow rate, g_0 is the standard gravitational acceleration, and I_{sp} is the specific impulse.

In order to determine the thrust it is necessary to estimate the specific impulse, together with the expansion ratio, the characteristic velocity of the exhaust gases and their gas constant. To obtain the propulsive characteristics of the liquid propellant engines, 4 propulsive parameters need to be estimated: the optimal oxidizer/fuel mixture ratio (R_m), the flame temperature (T_f), the relative gas molecular weight (M_w) and the gas specific heat ratio at the throat (γ). For these 4 parameters, a nonlinear surface generation process has been done based on the two-variable power function formulation:

$$f(x, y) = a + b \cdot x^c + d \cdot y^e \tag{3}$$

where $f = (R_m, T_f, M_w, \gamma)$, $x = P_c$ (combustion chamber pressure), $y = P_e$ (exhaust pressure), and (a, b, c, d, e) are the model coefficients.

Additional details on the *Propulsion* module are presented in paper [7] where the model coefficients for the most common 4 pairs of liquid propellants (oxidizer/ fuel): Oxygen/Kerosene; Oxygen/Methane; Oxygen/Hydrogen; Oxygen/Ethanol are provided. The developed model is valid for P_c values in the range of 10-250 atm and P_e values in the range of 0.1-1 atm. The proposed mathematical model has a high degree of accuracy, the overall average error being about 1.65% (for specific impulse and thrust values at sea level and in vacuum). At the same time, using relations of type (3), the computational time is significantly reduced compared to the multidimensional interpolation of combustion charts.

Within the *Aerodynamics* module, the following aerodynamic characteristics of interest of the axisymmetric launch vehicle configurations are determined: axial force coefficient (C_A), normal force coefficient (C_N), drag coefficient (C_D) and lift coefficient (C_L).

It is practical to breakdown the launcher into simple components from a geometric point of view. A small launcher can be seen as an assembly consisting of the following components (shown in Figure 5): nose/tip/fairing (multiple geometries), cylindrical stages, positive transitions and negative transitions. The simplest launcher can consist of only two components, a nose and a cylindrical stage.

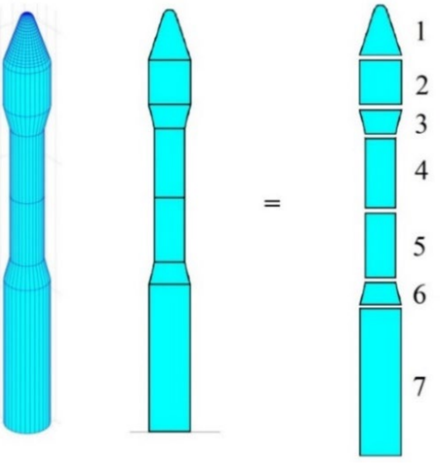


Figure 5 – Launcher breakdown into simple individual components [5], [6]

The mathematical modelling is centred around an accurate estimation of the launcher drag coefficient and normal force coefficient, the axial force coefficient and lift coefficient being afterwards computed. The drag coefficient is decomposed into a zero angle of attack drag coefficient C_{d_0} and an alpha drag coefficient C_{d_i}. Furthermore, drag is broken down into smaller components such as body pressure drag, friction drag and base drag, all of them being calculated for each individual component of the launcher and then added up to obtain the global aerodynamic characteristics. Summarizing papers [5] and [6], the mathematical modelling is done as follows: the body pressure drag coefficient is computed based on analytical [14], [15] and semi-empirical [16] models; the base drag coefficient is estimated using analytical [16] and semi-empirical models [17]; the friction drag coefficient is estimated using analytical and semi-empirical models [18], dependent on the flow regime (laminar, transitional, turbulent); the alpha drag coefficient is computed using semi-empirical models [19], [20].

For the normal force coefficient estimation, already existing models are used to compute the incompressible component (Barrowman model [21], together with the Galejs extension [22]), while for the compressibility factor F_{comp} the following model is proposed based on in-house results of axisymmetric configurations, CFD data [23] and experimental data [24], [25]:

$$F_{comp} = p_1 + p_2 M_c + p_3 \alpha + p_4 {M_c}^2 + p_5 M_c \alpha + p_6 \alpha^2 \tag{4}$$

where: α is the launcher angle of attack [°], $M_c = Msin(\alpha)$ is the crosswind Mach number and $P = (p_1, \dots, p_6)$ are the polynomial coefficients, being numerically given in [6].

The results obtained with the mathematical models integrated in the *Aerodynamics* module of the MDO tool were compared with those obtained after thoroughly CFD investigations, observing a very good data correlation, despite of the reduced calculation time (0.1s/case - developed model vs. 24h/case - CFD model). Details can be seen in [6].

In the *Trajectory* module, the primary objective is to integrate the equations of motion to accurately simulate the dynamic behaviour of the small launcher during its mission. The secondary objective includes the definition of a nominal optimal trajectory that maximizes imposed criteria (such as maximizing orbital performance and minimizing lift-off mass). Because of the high number if iteration needed for an MDO process convergence ([3], [4]), a fast mathematical model based on a null bank angle three degrees of freedom (3DOF) dynamic model was selected following the results of [26], the equations being written in the quasi-velocity frame, which describes only the translational motion of the launcher. Details about the kinematic and dynamic equations of the 3DOF model can be seen in [8].

The thrust orientation with respect to the velocity vector is described by the aerodynamic angles α and β^*, which can be seen as control parameters of the system with which the flight path angle γ, respectively track angle χ can be controlled using feedback relations such as:

$$\alpha = -k_1(\gamma - \gamma_d) \, ; \beta^* = -k_1(\chi - \chi_d) \tag{5}$$

where the reference (control desired) values are γ_d and χ_d, and k_1 is a setting parameter.

The launcher can be controlled exclusively by feedback relations such as the one above both for the primary active guidance phase and for the orbital insertion phase. A better solution is to use a method based on optimal commands in the orbital injection phase, as shown in [26]:

$$\alpha = -k_2(\gamma - \delta_1) \, ; \beta^* = -k_3(i - i_d) \tag{6}$$

where i_d represents the target orbit inclination, k_2 and k_3 are setting parameters, and δ_1 is the command in the orbit-related reference system obtained by optimising the orbital injection manoeuvre (decrease of orbit eccentricity to zero in minimum time for a circular orbit).

For small launchers, predominantly, it is preferred the use of a direct trajectory (DATO - Direct Ascent To Orbit), because it does not involve successive stops and restarts of the upper stage engine. At the same time, the time required for the launcher mission is reduced. The key events and evolution phases specific to a typical small launcher mission (with a two-stage architecture) using a DATO trajectory are shown in Figure 6: Vertical evolution; Primary active guidance (non-zero aerodynamic angles); Primary gravitational turn (zero aerodynamic angles); Separation of the first stage; Separation of the fairing; Ignition of the second stage engine; Secondary gravitational turn; Final active guidance (orbital insertion); Separation of the satellite from the upper structure.

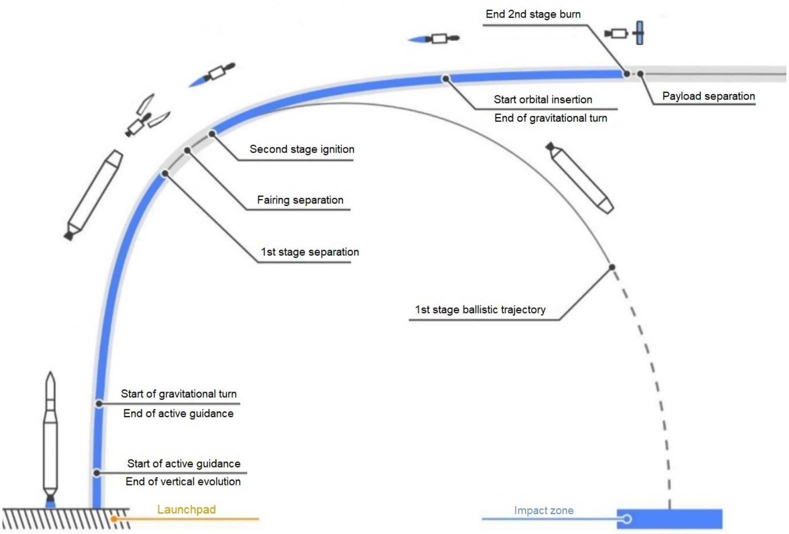


Figure 6 – Key events and evolution phases, two-stage launcher - DATO trajectory [8]

The solution selection and advancement algorithm chosen for the MDO tool is a hybrid one, which first uses a genetic algorithm (*@ga* from Matlab, population of 100 individuals) to significantly narrow the search field and then a gradient algorithm (*@fmincon* from Matlab with interior point method and BFGS Hessian).

The process of launcher global optimization is equivalent to the optimisation of several key parameters (optimisation variables), the launcher being then completely defined through them and the global input data. The optimisation variables are directly necessary for the mathematical models used in the disciplinary analyses. It was considered, where the implementation was convenient, a dimensionless formulation of these key parameters in order to reduce as much as possible the search space/ field of the optimal solution. A total of 16 optimisation variables is required for a preliminary design of an optimal constructive solution for a two-stage launcher. According to the optimisation vector structure detailed in Table 1, for a launcher with n stages ($n \geq 2$), in total $9n-2$ optimisation variables are required.

Table 1 – Optimisation variables needed for n-stage launcher

<table>
<tr><th>Type of optimisation variable</th><th>Number of optimisation variables</th><th>Total number of optimisation variables</th></tr>
<tr><td>M_p</td><td>n</td><td rowspan="10">$9n-2$</td></tr>
<tr><td>D_e</td><td>n</td></tr>
<tr><td>P_c</td><td>n</td></tr>
<tr><td>P_e</td><td>n</td></tr>
<tr><td>$\frac{T}{W}$</td><td>n</td></tr>
<tr><td>$t_{vertical}$</td><td>1</td></tr>
<tr><td>t_{coast}</td><td>$n-1$</td></tr>
<tr><td>γ_d</td><td>$n-1$</td></tr>
<tr><td>χ_d</td><td>$n-1$</td></tr>
<tr><td>Δt</td><td>n</td></tr>
</table>

where: M_p is the propellant mass (1 per stage), D_e is the outer diameter (1 per stage), P_c is the combustion chamber pressure (1 per stage), P_e is the exhaust pressure (1 per stage), $\frac{T}{W}$ is the thrust to weight ratio at the start of the burn (1 per stage), $t_{vertical}$ is the duration of vertical ascent after launch sequence, t_{coast} is the duration of coasting between two consecutive stages, γ_d is the desired flight path angle for the first stage control, χ_d is the desired track angle for the first stage control and Δt is the ratio between active guidance time and total guidance time (active + gravitational).

The current trend of launchers is that of miniaturization, so an objective function is implemented in which the dominant criteria is the minimum mass at the start of the mission. This is equivalent to maximizing the payload performance index (payload to lift-off mass ratio). Thus, the MDO tools developed minimises the following objective function:

$$f_{objective} = (M_{start} + I_{orbit}) \cdot I_{constraints} \tag{7}$$

where M_{start} is the launcher lift-off mass, I_{orbit} is the target orbit index, and $I_{constraints}$ is the imposed constraints index.

The target orbit index quantifies the quality of the orbit obtained compared to that imposed before launch. The formulation used to compute I_{orbit}, for a circular orbit of target inclination i_{target} and target altitude H_{target} is:

$$I_{orbit} = \sqrt{w_a(a - a_d)^2 + w_V(V - V_d)^2 + w_\gamma(\gamma - \gamma_d)^2 + w_i(i - i_d)^2} \tag{8}$$

where: w is a parameter weight, a is the semimajor axis, V is the velocity in ECI frame, γ is the flight path angle and i is the orbit inclination. Further details can be obtained from [8].

The imposed constraints index is used to quantify the validity of the obtained trajectory in relation to the imposed constraints and requirements. The formula used for $I_{constraints}$, with a number of $N_{constraints}$ is the following:

$$I_{constraints} = \prod_{i=1}^{N_{constraints}} I_{constraints_i} \tag{9}$$

If the constraints are not respected, the terms $I_{constraints_i}$ associated with these constraints would take over-unit values, which would increase the objective function numerical value, while the optimization process tries to minimise the objective function. If the constraint is met, then the term $I_{constraints_i}$ would take the value 1. In addition, a main constraint is implemented, having a value much higher than the others mentioned above, being used $I_{constraints_i} = 10^5$. This is only used if the trajectory is not computed until the end of the mission (for example the trajectory was sub-orbital, the launcher returning to Earth, $H < 0$) to remove the respective optimization vectors from future genetic algorithm populations. Typical constraints include (but are not limited to): maximum heat flux, maximum axial and normal load factors, maximum aerodynamic angles, desired nozzle expansions ratio, maximum stage fineness ratio, maximum deviation from imposed control parameters.

The different inputs required for the MDO algorithm are defined prior to its execution, the most important being the requirements of the target orbit, the payload mass, the propellant used, the constraints to be applied, the problem analysis area limits (solution search space), the launch location, the fairing separation condition and the materials used, together with their mechanical and thermal properties. During the parametric analyses, only one of the key parameters was modified, the rest being kept identical for all the cases in that batch.

2. TARGET ORBIT ALTITUDE IMPACT

With the aid of the MDO tool briefly presented in chapter 1, the impact of the target orbit altitude on the small launcher configuration was studied, 5 missions being analysed, with altitudes of circular, polar, target orbits between 200 km and 600 km, with an increment of 100 km. The payload mass was considered 130 kg. The launcher setup was based on a two-stage, constant diameter architecture, the engines being based on a liquid oxygen / liquid methane mixture. The launch location used was the Andøya Space Centre platform (Norway), which is the preferred one for obtaining polar orbits due to its high launch latitude.

Typical MDO convergence occurs after 2000 generations (around 200k disciplinary analyses and 24 hours of runtime). Some of the general details regarding the specifications of the obtained constructive solutions are observed in Table 2, while the main components are represented graphically in Figure 7.

Table 2 – Small launchers general details, orbit altitudes 200 - 600 km missions

Specification	Value				
	H=200km	H=300km	H=400km	H=500km	H=600km
Lift-off mass [t]	**9.92**	**10.80**	**11.56**	**12.89**	**13.96**
Total length [m]	16.78	17.54	17.64	18.31	19.07
Outer diameter [m]	1.27	1.29	1.35	1.4	1.41
Payload mass [kg]	130	130	130	130	130
Propellant	$O_2 + CH_4$	$O_2 + CH_4$	$O_2 + CH_4$	$O_2 + CH_4$	$O_2 + CH_4$
First stage burn time [s]	117.64	129.17	112.68	102.09	96.96
First stage mean thrust [kN]	189.51	198.86	236.82	293.74	332.89
Second stage burn time [s]	328.69	401.62	386.57	439.57	490.8
Second stage mean thrust [kN]	12.2	7.75	10.97	10.14	11.2

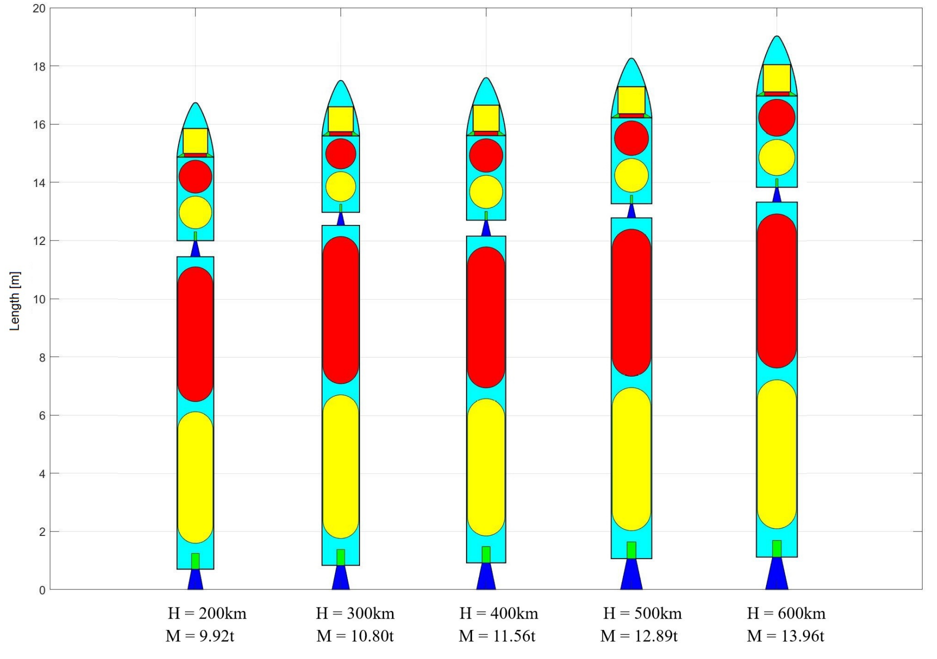


Figure 7 – Launcher main components, orbit altitudes 200 - 600 km missions

With the aid of the data from Table 2, one of the most important small launcher performance parameters can be calculated. This is the payload performance index $I_{payload}$ and it is obtained using:

$$I_{payload} = \frac{m_{payload}}{M_{start}} \tag{10}$$

Thus, the impact of the target orbit altitude on the small launcher can be quantified as in Table 3 and graphically represented in Figure 8 and Figure 9.

Table 3 – The influence of target orbit altitude on the small launchers

Payload mass [kg]	Circular polar orbit altitude [km]	Launcher lift-off mass [t]	Payload performance index [%]
130	200	9.92	1.31
130	300	10.80	1.20
130	400	11.56	1.12
130	500	12.89	1.01
130	600	13.96	0.93

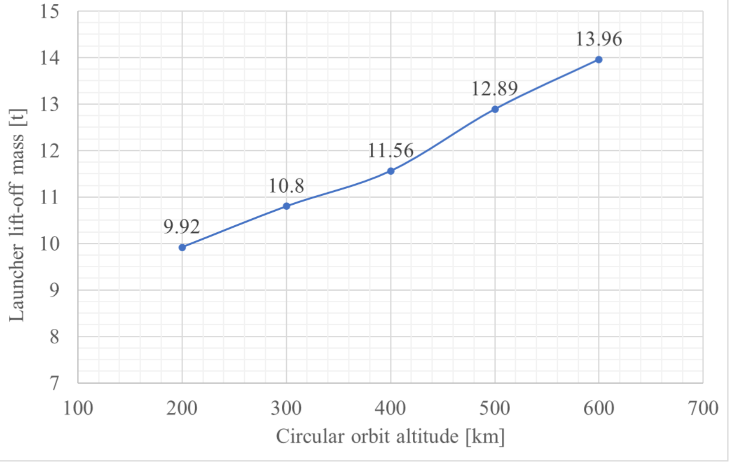


Figure 8 – Launcher lift-off mass vs. orbit altitude, 200 - 600 km missions

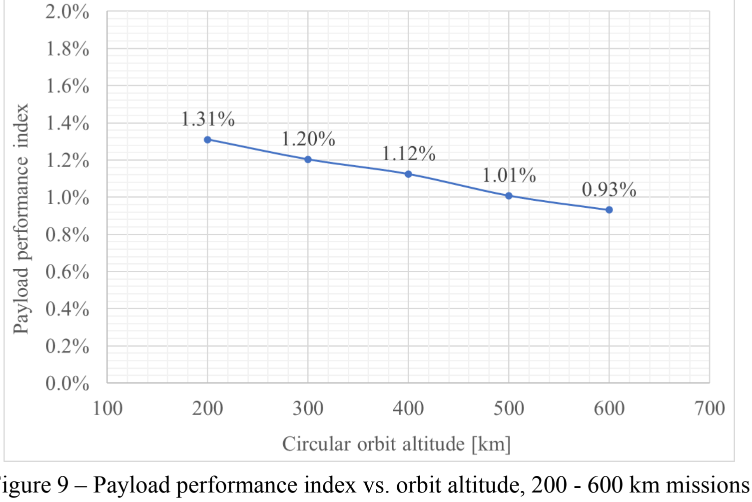


Figure 9 – Payload performance index vs. orbit altitude, 200 - 600 km missions

3. PAYLOAD MASS IMPACT

With the aid of the MDO tool briefly presented in chapter 1, the impact of the payload mass on the small launcher configuration was studied, 9 missions being analysed, with payload masses (m) between 10 and 250 kg, with an increment of 30 kg. The target orbit is a 400 km, circular, polar orbit. The launcher setup was identical to the one used in chapter 2, based on a two-stage, constant diameter architecture, the engines being based on a liquid oxygen / liquid methane mixture. The launch location used was the Andøya Space Centre platform (Norway), which is the preferred one for obtaining polar orbits due to its high launch latitude.

Some of the general details regarding the specifications of the obtained constructive solutions are observed in Table 4, while the main components are represented graphically in Figure 10.

Table 4 – Small launchers general details, payload mass 10-250 kg missions

Specification	Value				
	m=10kg	m=70kg	m=130kg	m=190kg	m=250kg
Lift-off mass [t]	**4.97**	**8.43**	**11.56**	**15.79**	**18.93**
Total length [m]	10.59	14.35	17.64	20.34	21.72
Outer diameter [m]	1.35	1.37	1.35	1.43	1.51
Payload mass [kg]	10	70	130	190	250
Propellant	$O_2 + CH_4$	$O_2 + CH_4$	$O_2 + CH_4$	$O_2 + CH_4$	$O_2 + CH_4$
First stage burn time [s]	102.64	97.5	112.68	119.64	103.14
First stage mean thrust [kN]	109.71	196.71	236.82	303.46	414.45
Second stage burn time [s]	382.67	392.69	386.57	404.76	382.61
Second stage mean thrust [kN]	4.28	8.07	10.97	14.53	20.84

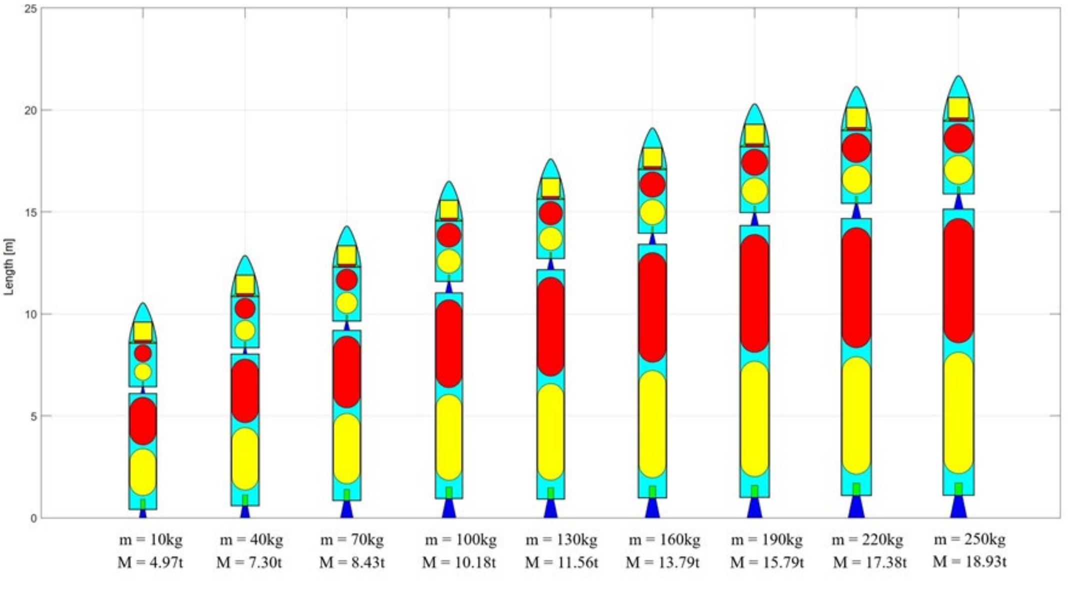


Figure 10 – Launcher main components, payload mass 10-250 kg missions

With the aid of the data from Table 4, the impact of the payload mass on the small launcher can be quantified as in Table 5 and graphically represented in Figure 11 and Figure 12.

Table 5 – The influence of payload mass on the small launchers

Payload mass [kg]	Circular polar orbit altitude [km]	Launcher lift-off mass [t]	Payload performance index [%]
10	400	4.97	0.20
40	400	7.30	0.55
70	400	8.43	0.83
100	400	10.18	0.98
130	400	11.56	1.12
160	400	13.79	1.16
190	400	15.79	1.20
220	400	17.38	1.27
250	400	18.93	1.32

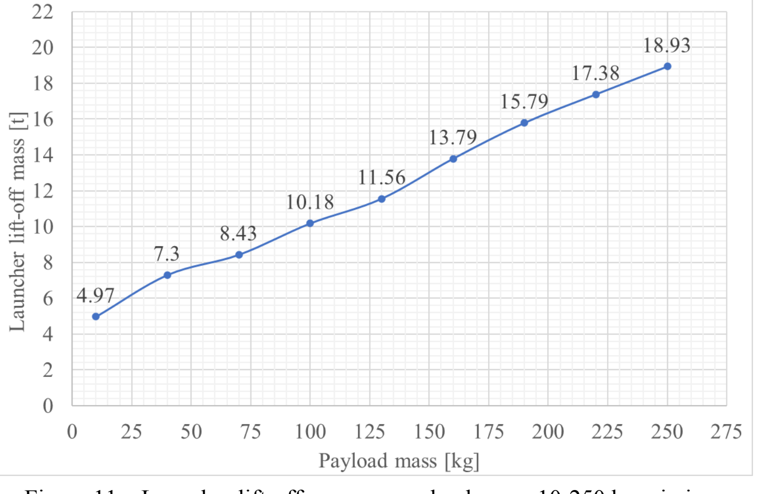


Figure 11 – Launcher lift-off mass vs. payload mass, 10-250 kg missions

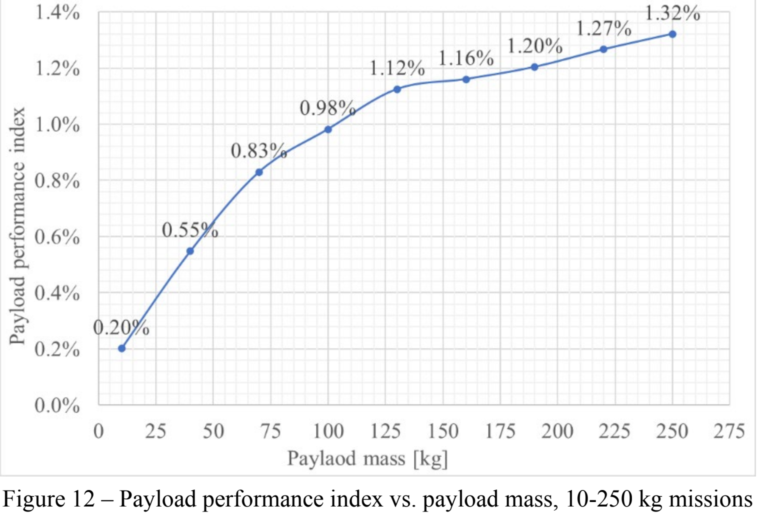


Figure 12 – Payload performance index vs. payload mass, 10-250 kg missions

4. TYPICAL MISSIONS IMPACT

With the aid of the MDO tool briefly presented in chapter 1, the impact of the typical missions on the small launcher was studied, 5 missions being analysed, with different launch locations and target orbit inclinations, specific to the approximately equatorial orbits, the International Space Station (ISS) orbit, the polar orbit, but also Sun-synchronous orbit (SSO). An altitude of 400 km was considered for all studied missions. The payload considered was 130 kg. The launcher setup was identical to the one used in chapter 2 and 3, based on a two-stage, constant diameter architecture, the engines being based on a liquid oxygen / liquid methane mixture.

Some of the general details regarding the specifications of the obtained constructive solutions are observed in Table 6, while the main components are represented graphically in Figure 13.

Table 6 – Small launchers general details, typical missions

Specification	Value				
	Kourou i = 5.3°	Omelek i = 9.1°	Baikonur i = 51.6°	Andøya i = 90°	Andøya i = 97.03°
Lift-off mass [t]	**8.36**	**8.52**	**9.73**	**11.56**	**12.06**
Total length [m]	16.06	16.17	16.33	17.64	18.13
Outer diameter [m]	1.18	1.19	1.29	1.35	1.35
Payload mass [kg]	130	130	130	130	130
Propellant	$O_2 + CH_4$	$O_2 + CH_4$	$O_2 + CH_4$	$O_2 + CH_4$	$O_2 + CH_4$
First stage burn time [s]	104.67	103.68	115.31	112.68	118.34
First stage mean thrust [kN]	185.5	192.35	199.86	236.82	242.69
Second stage burn time [s]	405.18	401.96	401.18	386.57	392.47
Second stage mean thrust [kN]	6.54	6.32	6.89	10.97	8.80

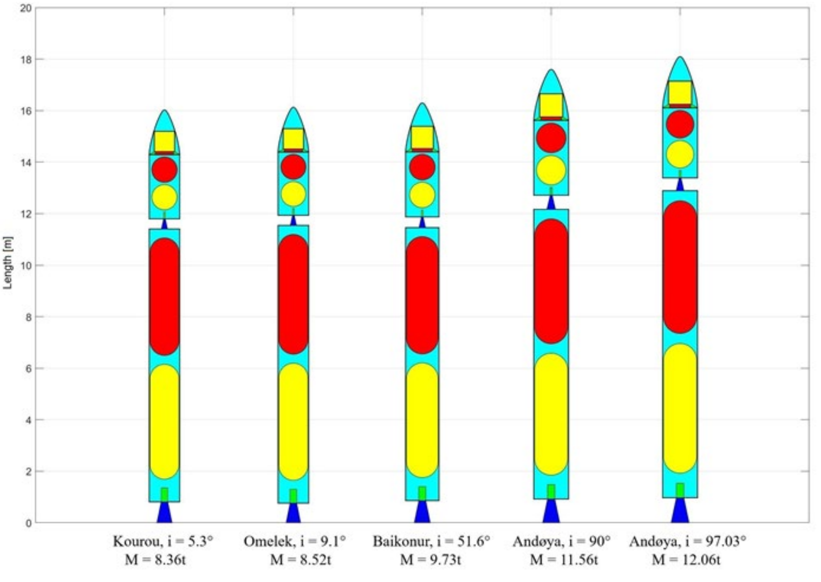


Figure 13 – Launcher main components, typical missions

With the aid of the data from Table 6, the impact of the typical missions on the small launcher can be quantified as in Table 7 and graphically presented in Figure 14 and Figure 15.

Table 7 – The influence of typical missions on the small launchers

Payload mass [kg]	Circular orbit altitude [km]	Launch location and target orbit inclination	Launcher lift-off mass [t]	Payload performance index [%]
130	400	Kourou, i = 5.3°	8.36	1.56
130	400	Omelek, i = 9.1°	8.52	1.53
130	400	Baikonur, i = 51.6°	9.73	1.34
130	400	Andøya, i = 90°	11.56	1.12
130	400	Andøya, i = 97.03°	12.06	1.08

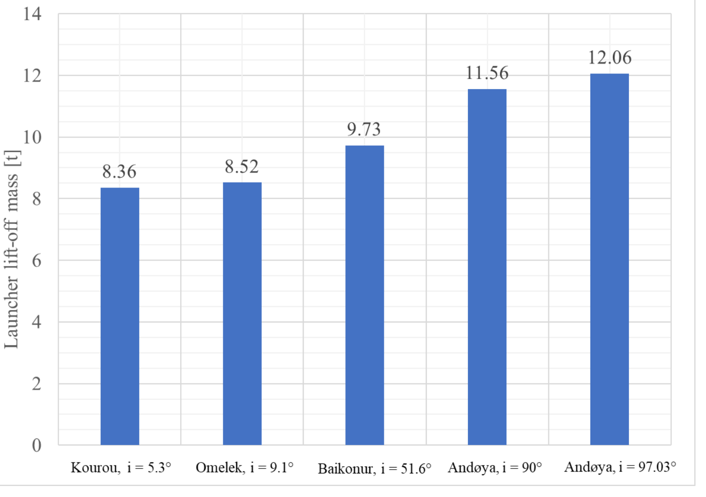


Figure 14 – Launcher lift-off mass vs. typical missions

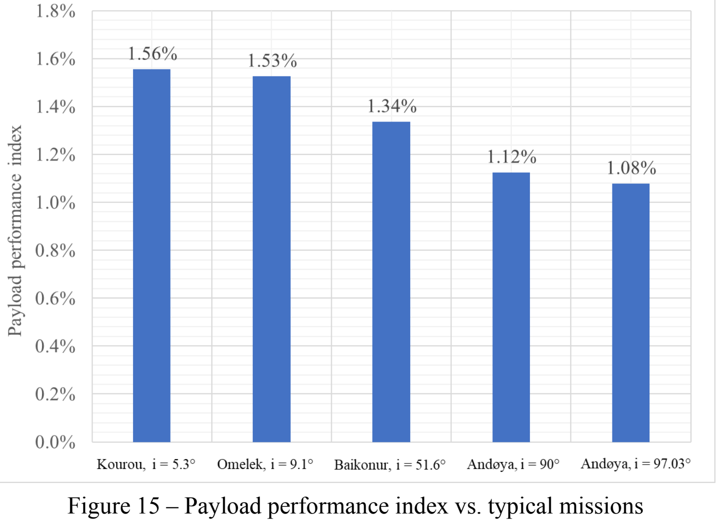
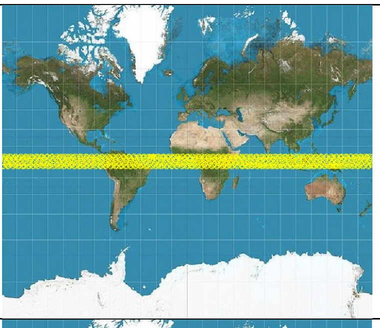
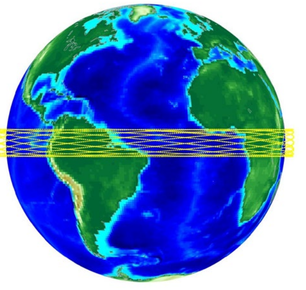
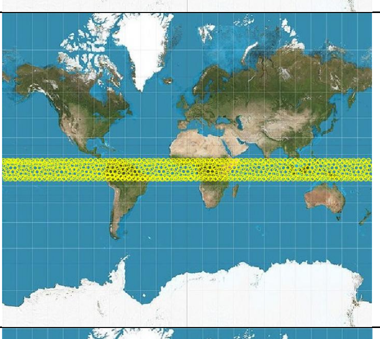
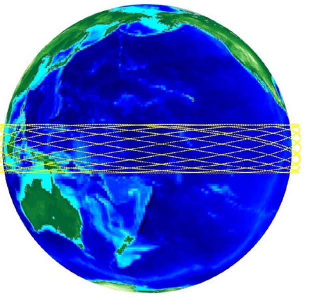
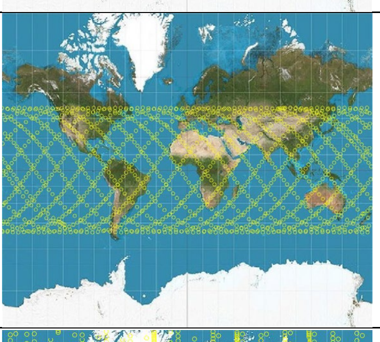
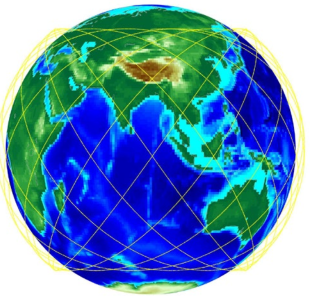
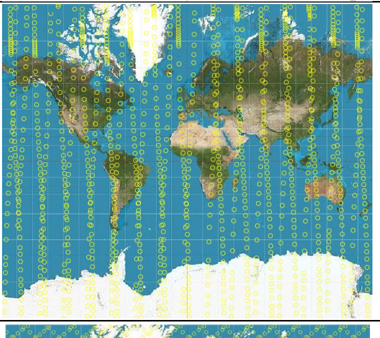
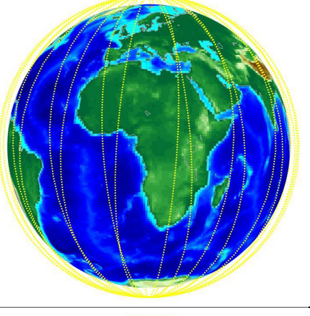
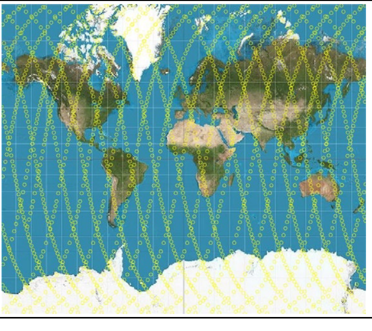
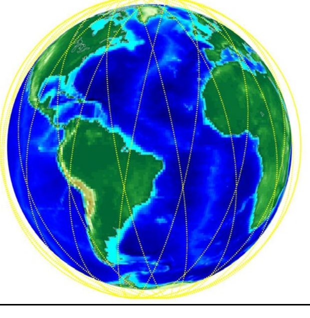


Figure 15 – Payload performance index vs. typical missions

The different inclinations of the target orbit allow the satellite inserted in it to view more or less extensive areas of the Earth. This aspect is strongly visible if one uses a graphical representation of the satellite orbital trajectory on a Mercator projection (2D) and one on the globe (3D), according to those presented in Table 8.

Table 8 – Satellite trajectories, first 24h after insertion, typical missions (2D and 3D representations)

		Mission 1, Kourou launch, Inclination = 5.3°
		Mission 2, Omelek launch, Inclination = 9.1°
		Mission 3, Baikonur launch, Inclination = 51.6°
		Mission 4, Andøya launch, Inclination = 90°
		Mission 5, Andøya launch, Inclination = 97.03°

5. USED PROPELLANT IMPACT

With the aid of the MDO tool briefly presented in chapter 1, the impact of the used propellant type on the small launcher configuration was studied, being analysed a total of 4 liquid bipropellants, having as an oxidizer liquid oxygen and as liquid fuel hydrogen (Launcher H-H), methane (Launcher M-M), kerosene (Launcher K-K) and ethanol (Launcher E-E). The target orbit was a circular, polar orbit of 400 km altitude, the payload mass being 130 kg. The launcher setup was identical to the one used in previous chapters, based on a two-stage, constant diameter architecture. The launch location used was the Andøya Space Centre.

Some of the general details regarding the specifications of the obtained constructive solutions are observed in Table 9, while the main components are represented graphically in Figure 16.

Table 9 – Small launchers general details, different used propellant

Specification	Value			
	Launcher H-H	Launcher M-M	Launcher K-K	Launcher E-E
Lift-off mass [t]	**6.51**	**11.56**	**15.07**	**26.96**
Total length [m]	15.99	17.64	18.61	23.26
Outer diameter [m]	1.58	1.35	1.30	1.64
Payload mass [kg]	130	130	130	130
Fuel	Hydrogen	Methane	Kerosene	Ethanol
First stage burn time [s]	123.18	112.68	111.55	104.04
First stage mean thrust [kN]	138.31	236.82	293.20	552.19
Second stage burn time [s]	379.26	386.57	372.33	377.22
Second stage mean thrust [kN]	9.74	10.97	15.72	19.05

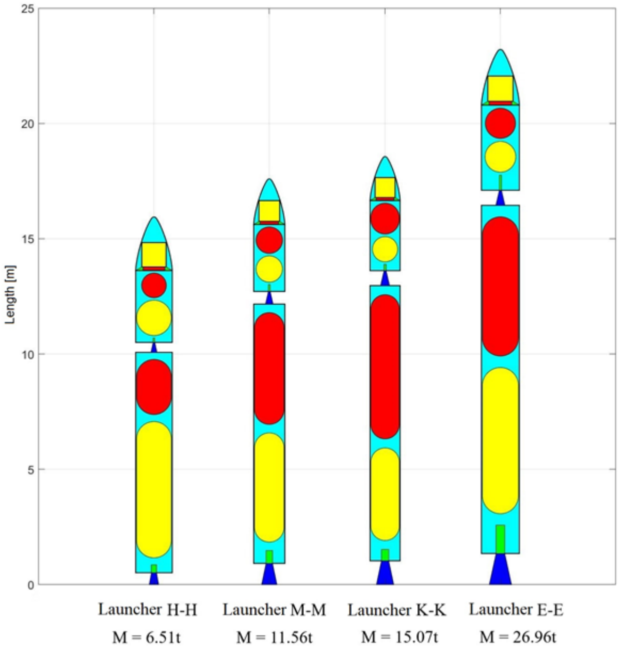


Figure 16 – Launcher main components, different used propellant

With the aid of the data from Table 9, the impact of the propellant type on the small launcher can be quantified as in Table 10 and graphically represented in Figure 17 and Figure 18.

Table 10 – The influence of propellant on the small launchers

Payload mass [kg]	Circular polar orbit altitude [km]	Launcher configuration	Launcher lift-off mass [t]	Payload performance index [%]
130	400	Launcher H-H	6.51	2.00
130	400	Launcher M-M	11.56	1.12
130	400	Launcher K-K	15.07	0.86
130	400	Launcher E-E	26.96	0.48

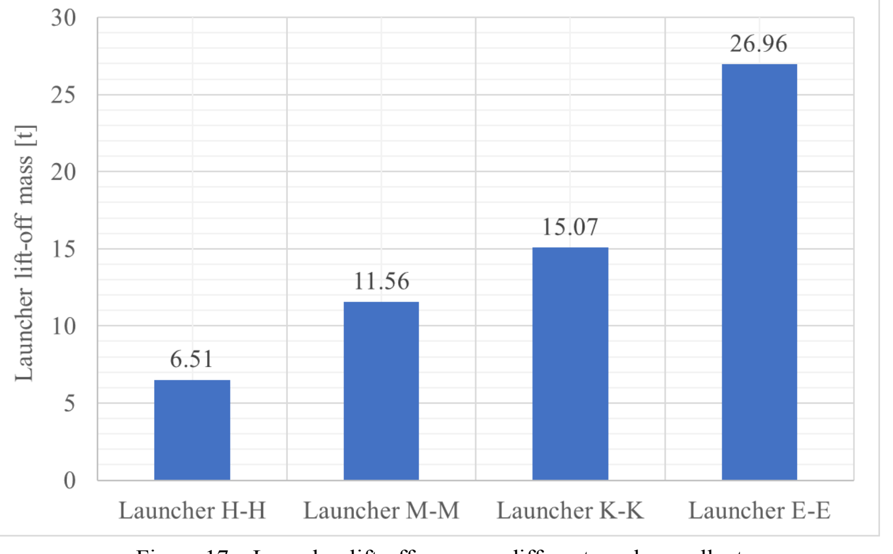


Figure 17 – Launcher lift-off mass vs. different used propellant

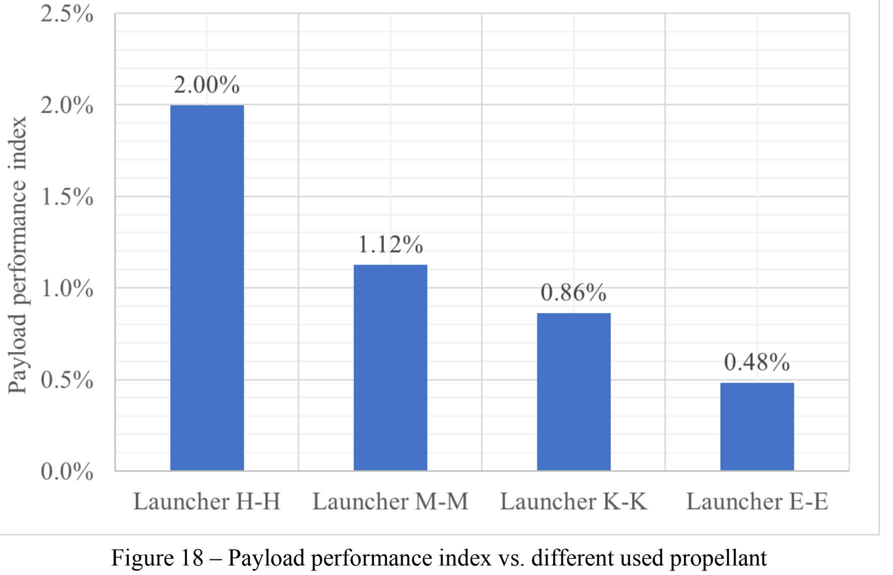


Figure 18 – Payload performance index vs. different used propellant

6. CONCLUSIONS

The objective of this paper was to present the influence of the main design requirements on the characteristics of the optimal small launcher. The launchers are obtained using a multidisciplinary design optimisation (MDO) algorithm, where the lowest mass configuration capable of accurately inserting the payload into the target orbit is considered to be optimal. An overview of the MDO tool developed is given in the first chapter, but details about all of the building blocks can be seen in previous works such as [3], [4], [5], [6], [7], [8] and [9].

In chapter 2 the impact of the target orbit altitude on the small launcher was studied, being analysed 5 missions, with altitudes of the circular, polar, target orbit between 200 and 600 km, with an increment of 100 km. A quasi-linearity relationship was observed between the orbit altitude and the lift-off mass of the launcher, obtaining a minimum value for the 200 km altitude case (Lift-off mass = 9.92 t) and a maximum value in the case of a 600 km altitude orbit (Lift-off mass = 13.96 t).

In chapter 3 the impact of the payload mass on the launcher was studied, being analysed 9 missions, with payloads between 10 and 250 kg, with an increment of 30 kg. A quasi-linearity relationship was observed between the payload mass and the lift-off mass of the launcher, a sharp decrease in the payload performance index for low payload masses appearing. Values of payload performance index higher than 1% are observed for payloads of more than 100 kg.

In chapter 4 the impact of the typical missions on the small launcher was studied, being analysed 5 missions, with different launch locations and target orbit inclinations, specific to the approximately equatorial orbits, the International Space Station (ISS) orbit, the polar orbit, and also Sun-synchronous orbit (SSO). A significantly lower lift-off mass is observed at low orbit inclinations, launched from a location with similar latitude to the target orbit (Kourou and Omelek). The most penalizing orbit from launcher lift-off mass point of view is the SSO, having an inclination specific to the 400 km altitude of 97.03°, the lift-off mass being approximately 45% higher than in the case of an 5.3° inclination orbit. In addition, a comparison of the satellites Earth's coverage is presented, maximum coverage being seen for the polar orbit.

In chapter 5 the impact of the propellant used on the small launcher was studied, being analysed a total of 4 pairs of liquid propellants, having as oxidizer liquid oxygen and as liquid fuel hydrogen (Launcher H-H), methane (Launcher M-M), kerosene (Launcher K-K) and ethanol (Launcher E-E). The target orbit was considered the circular, polar one of 400 km altitude, the payload mass being 130 kg. A very low lift-off mass is observed in the case of the H-H type launcher, due to its extremely high propulsive performances. The use of ethanol is not justified due to the very high lift-off mass associated. The simplicity of the technical solution required to use methane or kerosene as fuel (compared to hydrogen) justifies their use in the detriment of the optimal solution of the H-H launcher.

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