

Article

Preliminary Definition of a Family of Partially Reusable Microlaunchers with the Aid of an MDO Approach

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Abstract

The topic of reusability in the launch vehicle sector is of current worldwide interest, as a shift from expendable to partially reusable configurations can be observed. Based on the work realized in a nationally funded Nucleu project, INCAS has developed a multidisciplinary optimization environment (MDO algorithm) capable of generating preliminary launcher concepts that also take into account the recovery process needed to reuse key major assemblies, such as the lower stage. The current paper analyzes a set of five key missions of interest (with different launch locations) and their influence on the preliminary definition of a family of partially reusable microlaunchers capable of inserting the same payload (100 kg) into different inclination, low Earth orbits (ranging from almost equatorial to Sun-synchronous orbits). The proposed microlauncher concepts are based on a two-stage constant-diameter architecture, where the first stage is recovered via a downrange, autonomous vertical landing mission, while the upper stage is expendable. The main scope of this paper is to quantify the impact of different key mission requirements on the characteristics of the reusable microlauncher that minimize its lift-off mass. This approach also correlates to the definition of reusable launch vehicle concepts that have reduced the associated costs of development, production, and operation.

Keywords: multidisciplinary optimization; MDO; reusable microlauncher; downrange recovery; vertical landing; mass minimization; LEO; equatorial orbit; polar orbit; SSO; methalox

1. Introduction

The space launch vehicle sector is one of great competitiveness, with the definition of almost optimal configurations being desired even from the preliminary phases of design, such that after the development period, the constructive solution is still economically feasible in the fast-paced satellite-insertion market. One great way to reduce the costs related to the exploitation of launch vehicles (regardless of size) is to reuse critical assemblies that have increased production costs [1].

This usually translates into the reuse of the first stage or, depending on the size of the launcher, its boosters. One key player that has demonstrated such capabilities is the American company SpaceX, which has recovered and reused the first stage of the Falcon 9 launcher numerous times [2]. This technological demonstration has initiated a ripple effect in the launcher sector, with more and more companies deciding to shift towards reusable launch vehicles from the standard expendable concepts.

In the European region, the level of maturity of reusable launchers is still low, as no flight-proven reusable vehicle exists. Many European demonstrators are in advanced



Academic Editor: Paolo Tortora

Received: 12 January 2026

Revised: 5 February 2026

Accepted: 6 February 2026

Published: 8 February 2026

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phases of development, with the most important being MIURA 5 [3], THEMIS [4], and CALLISTO [5].

One niche segment of launch vehicles is the segment associated with dedicated launchers for small satellites, known as microlaunchers. The current launcher market is defined by the existence of medium and large launch vehicles that can insert large satellites (up to several tons) anywhere from low Earth orbits (LEO) up to geostationary orbits (GEO). By using this kind of vehicle for low-mass satellites (mini- and microsatellites up to 100 kg [6]), the orbital insertion priority is low, as the small satellite is seen as a secondary payload (“piggyride” mission). Having a low orbital insertion priority, the small satellites usually must be re-adapted to better suit the main payload target orbit, which, of course, comes with some of its own performance loss.

Having all of these in mind, one project incorporated in the Romanian Nucleu program funding axis (known as PN 23 17 07 02) is intended to provide key insight into the definition, analysis, and selection of a partially reusable microlauncher concept that will be used as the first building block towards the development of such a vehicle in the European zone.

When addressing the feasibility of partially reusable launchers, it is necessary to discuss the method by which the lower stage is recovered. Many possible recovery solutions can be used [7–9], with the most important one being autonomous vertical landing (at the launch location or a secondary location). These two approaches have been studied in previous papers [10,11], with the outcome of paper [11] stating that implementing a return-to-launch-site recovery mission is not economically feasible for a reusable microlauncher concept, as the cost per launch is more than double compared to the secondary location recovery (downrange recovery). Paper [11] also shows that using a downrange recovery mission, the total cost per launch of a reusable microlauncher is lower than that of an expendable microlauncher [12], which validates the economic feasibility of developing, producing, and operating such a launch vehicle concept.

The current paper extends the study performed in [10], with several other missions of interest being the subjects of investigation. For the recovery mission, a downrange vertical landing recovery mission is envisioned, with the position of the landing platform not being imposed before microlauncher optimization, but rather as an output that maximizes the orbital performance of the launcher.

A total of five key missions will be studied, so that a preliminary quantification of the mission impact on the characteristics of the reusable microlauncher can be made. Several launch locations will be investigated together with different target low Earth orbit inclinations, ranging from almost equatorial to polar and Sun-synchronous ones (SSO). The payload analyzed is the same as the one used in papers [10,11], so that a clear and correct comparison can be made towards the selection of a baseline nominal reusable microlauncher concept.

Additionally, the setup of the optimization cases implemented during this paper’s elaboration will be as close as possible to that of [10–12]. This means that a two-stage, constant-diameter, microlauncher architecture is used, where the first stage is recovered and the second stage is expendable. The upper-stage de-orbitation and stage disintegration are not part of this study but can be analyzed with the aid of the mathematical models depicted in the literature. For future works, it is expected that after nominal microlauncher concept selection, several stage reentry studies will be performed as part of the currently funded Nucleu project.

In addition to the two-stage architecture, some aspects related to the propulsion side are also considered as input data, such as the number of rocket engines per stage (one) and the liquid propellant pair used (LOX/methane—known as methalox). Regarding the aerodynamics side, the fairing architecture is fixed, consisting of a 1.5 fineness ratio

von Kármán ogive generation profile, a profile which ensures minimal drag for a set base diameters [13].

2. Multidisciplinary Optimization Algorithm

Because of the high number of disciplines that must be assessed when analyzing a microlauncher (either expendable or reusable), one of the best ways of generating a potential concept is that of using a multidisciplinary optimization algorithm (MDO) [14–17]. The MDO is used when a trade-off between multiple disciplines must be realized in order to find a global optimum and not just the best concept for each of the disciplines of interest. When analyzing the performance of a microlauncher, one must carefully estimate some key parameters related to the following disciplines: Weights and sizing, aerodynamics, propulsion, trajectory propagation, and associated costs.

Such a multidisciplinary optimization algorithm has already been developed in INCAS, and it has the associated block scheme presented in Figure 1. The MDO was written in Matlab version R2024a [18] and has been used successfully in previous papers to generate two reusable microlauncher concepts [10,11]. The MDO will be used in this paper to quantify the impact of the main mission requirements on the microlauncher characteristics.

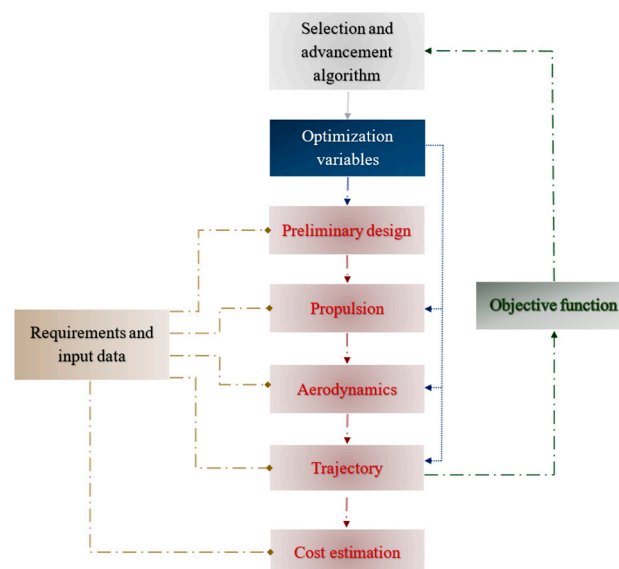


Figure 1. MDO algorithm block algorithm (the five main modules are depicted in red blocks, while the other four auxiliary modules are depicted in distinct color blocks; data workflow is dictated by arrows between modules).

The MDO algorithm is used to generate a partially reusable microlauncher concept that minimizes a set objective function. All of the launch vehicles' performances of interest are evaluated inside the five main modules (one for each of the main disciplines that must be assessed):

- Preliminary design;
- Propulsion;
- Aerodynamics;
- Trajectory;
- Cost estimation.

As seen in Figure 1, the iterative optimization loop contains only the four main modules mentioned before, the last one (cost estimation) being called only after solution convergence. In addition to the five main modules, another four auxiliary modules are necessary during solution optimization, with these being:

- Requirements and input data;
- Optimization variables;
- Objective function;
- Selection and advancement algorithm.

A short overview of the way in which the MDO works will be given in this paper; for more details regarding the full mathematical model used, one can consult paper [10].

The process of microlauncher optimization is equivalent to the optimization of a set number of variables known as optimization variables. All of the optimization variables are stored inside an optimization variable vector, which, for the current case, that of a partially reusable microlauncher for which the first stage is recovered from a secondary location, has a total of 20 variables [10]. These 20 optimization variables are split into four subsets:

- First-stage definition subset (five optimization variables):
 - Propellant mass: M_{p1} ;
 - Stage outer diameter: D_{e1} ;
 - Engine combustion chamber pressure: P_{c1} ;
 - Engine exhaust pressure: P_{e1} ;
 - Thrust-to-weight ratio at engine ignition: $\left(\frac{T}{W}\right)_1$.
- Second-stage definition subset (five optimization variables):
 - Propellant mass: M_{p2} ;
 - Stage outer diameter: D_{e2} ;
 - Engine combustion chamber pressure: P_{c2} ;
 - Engine exhaust pressure: P_{e2} ;
 - Thrust-to-weight ratio at engine ignition: $\left(\frac{T}{W}\right)_2$.
- Main mission definition subset (six optimization variables):
 - Primary active guidance phase desired flight path angle: γ_{d1} ;
 - Primary active guidance phase desired track angle: χ_{d1} ;
 - Vertical ascent duration: $t_{vertical}$;
 - Coast phase duration: t_{coast} ;
 - Ratio between the active guidance phase and total first-stage guidance phases: Δt_1 ;
 - Ratio between active guidance phase and total second-stage guidance phases: Δt_2 .
- Recovery mission definition subset (four optimization variables):
 - Propellant mass reserved for recovery mission: $M_{precovery}$;
 - Reentry burn altitude (at start of maneuver): $H_{reentry}$;
 - Ratio between the propellant reserved for the landing maneuver and recovery mission: $\Delta M_{planding}$;
 - Landing burn altitude (at start of maneuver): $H_{landing}$.

All of these optimization variables are used as input data in one or more of the main modules of the MDO. The first 10 are mainly used in the preliminary design, propulsion, and aerodynamics modules, while the last 10 are mainly used in the trajectory module.

Of course, in addition to the optimization variables, various input data are defined in the requirements and input data module. These include mission requirements such as target orbit (altitude, inclination, eccentricity, etc.), design architecture (number of stages, fairing generation profile, fairing jettison condition, propellant type, materials used, etc.), imposed constraints (max axial/normal load factors, maximum difference between stage diameters, min/max stage fineness ratios, etc.), launch site location, imposed solution search space (upper and lower numerical bounds), and so on.

As stated before, the optimal microlauncher concept is obtained at solution (optimization variable vector) convergence. The solution is advanced based on the values of a fitness score (objective function) by a selection and advancement algorithm. For the current and previous papers, a genetic algorithm was used to advance the solution until convergence, as it is one of the most robust algorithms that has provided good results in aerospace-related optimization complex problems [19,20]. For better controllability of solution convergence, a general-purpose pseudorandom number generator was implemented, in the form of the Mersenne Twister one [21].

Due to the inherent complexity of the microlauncher design space, achieving a mathematically verifiable global optimum is often computationally unfeasible. This study acknowledges that no optimization algorithm can guarantee an absolute best solution for such high-dimensional problems. Instead, the multidisciplinary design optimization framework is utilized to converge upon superior, near-optimal configurations that significantly enhance vehicle performance, despite the absence of a formal proof of optimality.

Probably one of the most important auxiliary modules is that associated with the objective function definition because the formulation used here will directly have an impact on the convergence process, as the function scores obtained by each of the individuals from the population used by the genetic algorithm are used to select which individuals become elites and which ones will suffer crossover and mutation processes [10]. The following objective function formulation is implemented inside the MDO:

$$f_{objective} = (M_{start} + I_{mission} + I_{recovery}) \cdot I_{constraints} \quad (1)$$

where M_{start} is the lift-off mass of the launcher (obtained in the preliminary design module), $I_{mission}$ is the main mission performance index, $I_{recovery}$ is the recovery mission performance index, and $I_{constraints}$ is the constraint index (for both missions).

It is of interest to showcase the formulation used for the main performance indices used in the objective function definition (1). The first one is the main mission performance index $I_{mission}$ and is evaluated using:

$$I_{mission} = \sqrt{w_a(a - a_t)^2 + w_V(V - V_t)^2 + w_\gamma(\gamma - \gamma_t)^2 + w_i(i - i_t)^2} \quad (2)$$

with $(w_a, w_V, w_\gamma, w_i)$ being parameter weights, a being the semi-major axis, V being the velocity (inertial), γ being the flight path angle, and i being the orbit inclination (all obtained in the trajectory module).

In Equation (2), the target values are expressed with t subscript, while the other parameters correspond to actual values obtained at the moment of satellite orbit insertion. Regarding the values used for the parameter's weights $(w_a, w_V, w_\gamma, w_i)$, it was observed from previous studies, including [12], that applying non-uniform weighting coefficients for the angle-related parameters (flight path angle γ and orbit inclination i) compared to the other two (semi-major axis a and inertial velocity V) accelerates the convergence process of the solution with minimal influence on the quality of the end result.

By using $w_a = 1$, $w_V = 1$, $w_\gamma = 10$, and $w_i = 10$ [10], the gradient of the main mission performance index $I_{mission}$ is heavily biased toward the terms with the higher order of magnitude. This disparity forces the optimization algorithm to prioritize the minimization of the flight path angle and orbit inclination insertion errors, effectively treating it as a pseudo-constraint. This is due to the fact that the changes in γ and i are harder to achieve compared to raising/lowering the altitude of the launcher (to modify a) and increasing/decreasing the rocket engine size or operating pressures (to modify V).

Overall, by aggressively reducing the largest source of "error" or cost first, the optimizer moves rapidly through the design space toward a region of feasibility. This behavior

results in a faster initial reduction in the global objective value $f_{objective}$, as the numerical “stiffness” of the problem directs the search path along the steepest descent of the dominant term before finer adjustments to the secondary term are performed.

The recovery mission performance index $I_{recovery}$ is computed using:

$$I_{recovery} = M_{p_{unused}} + E_{landing} \quad (3)$$

where $M_{p_{unused}}$ is the unused propellant mass with which the lower stage lands and $E_{landing}$ is a landing position error. As stated in [10], for a downrange recovery mission, $E_{landing} = 0$, as no initial conditions are imposed for the exact location of the landing platform/barge. After MDO convergence, the exact location that maximizes orbital performance will be given and is used as a reference for real-life operation.

The last index needed for Equation (1) is the constraint index $I_{constraints}$, which has the following formulation:

$$I_{constraints} = \prod_{i=1}^{N_{constraints}} I_{constraint_i} \quad (4)$$

with $N_{constraints}$ being the total number of imposed constraints and $I_{constraint_i}$ being the individual performance index corresponding to the i^{th} constraint. A full list of possible constraints is given in [10], and the ones implemented in this paper will be further mentioned in Section 4. If the constraint is satisfied, then $I_{constraint_i} = 1$, otherwise we have the following:

$$I_{constraint_i} = 1 + \left| \frac{val - val_{imposed}}{val_{imposed}} \right| \cdot f_{penalty} \quad (5)$$

where $val_{imposed}$ is the imposed value of constraint i , val is the obtained value of the constraint parameter, and $f_{penalty}$ is a penalty factor considered as 100 for the main mission constraints and 1 for the recovery mission constraints.

With the aid of the data defined in the requirements and input data module and the data from the optimization variables vector, the entire reusable microlauncher concept (and performances) can be generated with the aid of the mathematical models implemented inside the five main modules.

The first main module is the preliminary design module, in which the launch vehicle is sized by means of both masses and dimensions. A bottom-up approach is used, and the breakdown scheme implemented is shown in Figure 2 (for a typical n-stage launcher).

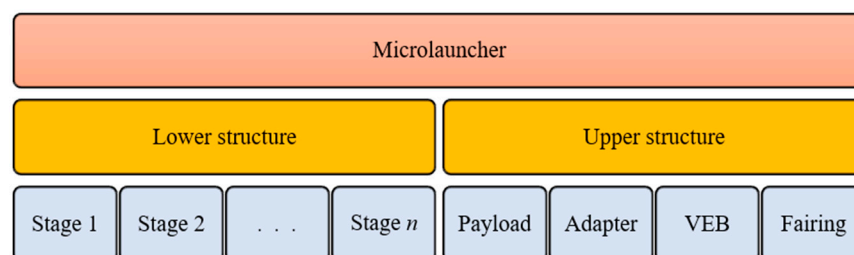


Figure 2. Breakdown scheme of a typical n-stage microlauncher.

As seen in Figure 2, the microlauncher architecture consists of a lower structure (two stages in our case) and an upper structure. The upper structure is the same regardless of the type of stage architecture (expendable or reusable) and contains the following individual components [22]:

- Payload (one or more mini/micro satellites);
- Payload adapter;

- Vehicle equipment bay (VEB) is the zone inside the payload adapter that includes the avionics and additional microlauncher electrical systems;
- Fairing (which houses the payload and is jettisoned at some point during the ascent).

When assessing the lower structure, the same bottom-up approach is used for every constituent stage. In the case of an expendable microlauncher concept, both stages would have the same breakdown scheme, but, because the first stage is now recovered, one must include an additional critical system in the first-stage breakdown scheme. The breakdown schemes implemented inside the preliminary design module are shown in Figure 3.

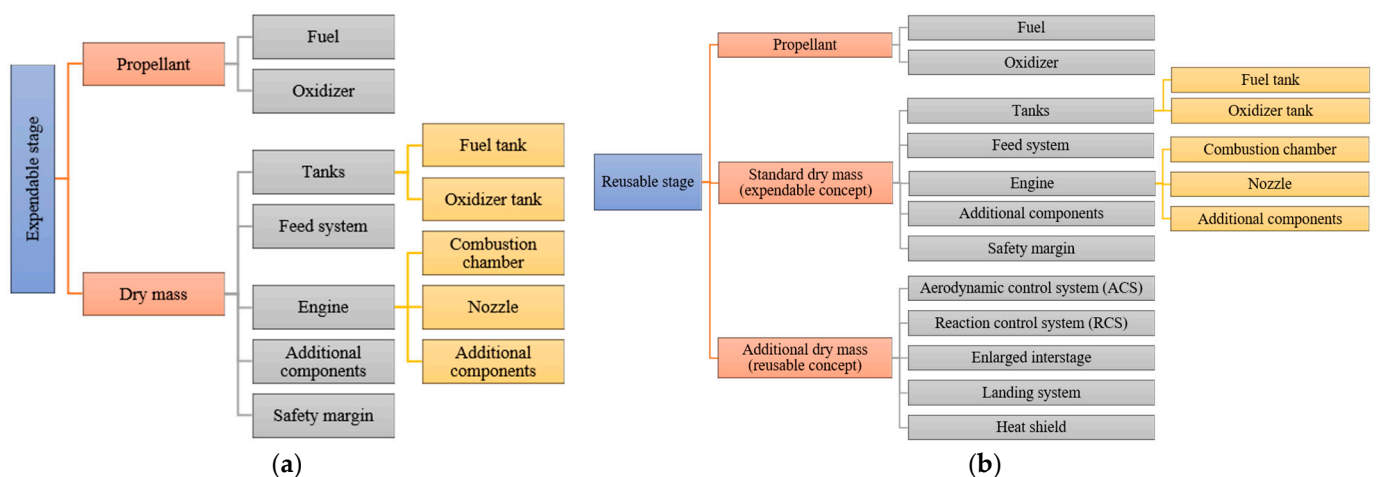


Figure 3. Breakdown scheme of a microlauncher stage: (a) expendable; and (b) reusable.

For the reusable stage concept, the following additional systems are accounted for: an aerodynamic control system (ACS), an extra-atmospheric control system (RCS based on cold gas thrusters), an enlarged interstage (to include the ACS and RCS), a foldable landing system, and a heat shield (to ensure that the atmospheric reentry phase does not structurally affect the integrity of the first stage).

All of the mathematical models implemented are based on analytical and semi-empirical, closed-relation models, as they provide a high-accuracy first estimation in record time (in the order of milliseconds). For example, both propellant tanks (material can be either a 2000/7000-series aluminum alloy or a titanium alloy such as Ti-6Al-4V) are modeled by determining critical parameters, including internal volume, operating pressure, wall thickness, and total mass [10]. The geometric configuration is determined by the vehicle's space allocation requirements, resulting in two distinct design outcomes. For stages requiring high volumetric capacity and elongation (typical of lower stages), the model yields a cylindrical geometry enclosed by hemispherical end-caps. Conversely, for configurations with a lower length-to-diameter ratio (characteristic of upper stages), the cylindrical section is omitted, and the model converges on a spherical geometry.

In the propulsion module, the performance of the liquid propellant rocket engines is assessed, both for the lower, reusable stage and for the upper, expendable stage. The main outputs of the module are the thrust curves, which are generated using:

$$T = \dot{m} \cdot g_0 \cdot I_{sp} \quad (6)$$

with $\dot{m}$ being the propellant mass flow rate (computed based on the optimization variables $\frac{T}{W}$), g_0 being the standard gravitational acceleration [23], and I_{sp} being the specific impulse.

During both the main and recovery mission of the microlauncher, the propellant mass flow is considered to be constant, thus implying a 100% throttle setting. The differences

between the thrust generated as time advances are dictated by the specific impulse, which is computed using [14,24,25]:

$$I_{sp} = \eta_n \cdot \frac{C^*}{g_0} \left(\gamma \sqrt{\left(\frac{2}{\gamma-1} \right) \cdot \left(\frac{2}{\gamma+1} \right)^{\frac{\gamma+1}{\gamma-1}} \cdot \left(1 - \left(\frac{P_e}{P_c} \right)^{\frac{\gamma-1}{\gamma}} \right)} + \frac{\varepsilon}{P_c} (P_e - P_a) \right) \quad (7)$$

with η_n being the nozzle efficiency (considered 98% [26]), C^* being the propellant characteristic velocity, γ being the exhaust gas isentropic coefficient, P_c being the combustion chamber, P_e being the exhaust pressure, P_a being the atmospheric pressure, and ε being the nozzle expansion ratio (obtained in the preliminary design module [10]).

The propellant characteristic velocity C^* is computed using:

$$C^* = \eta_c \cdot \frac{\sqrt{\gamma R T_f}}{\gamma \left(\frac{2}{\gamma+1} \right)^{\frac{\gamma+1}{2\gamma-2}}} \quad (8)$$

with η_c being the combustion efficiency (considered 98% for the methalox pair [26]), R being the exhaust gas constant, and T_f being the flame temperature.

To properly compute the specific impulse and, in the end, the rocket engine thrust curves, one must implement a mathematical model that estimates the optimal oxidizer/fuel ratio R_m , the flame temperature T_f , the gas molecular weight M_w , and the isentropic coefficient of the exhaust gases γ with great accuracy. These are obtained with the aid of an approximation function, following a thermochemical equilibrium analysis [26]:

$$f(x, y) = a + b \cdot x^c + d \cdot y^e \quad (9)$$

with $f = (R_m, T_f, M_w, \gamma)$, x being associated with the combustion chamber pressure (P_c), y being associated with either the exhaust pressure (P_e) or the optimal mixture ratio (R_m), and (a, b, c, d, e) being the model coefficients.

In order to obtain the coefficients (a, b, c, d, e) , several nonlinear regressions were performed in [26] and summarized in [10]. The model presented in Equation (9) is valid for combustion chamber pressures in the 10 to 250 atm range and exhaust pressures in the 0.1 to 1 atm range, which corresponds to most of the current rocket engines that are in use worldwide. For the LOX/methane pair, the model coefficients are shown in Table 1.

Table 1. Propulsion model coefficients, methalox pair [26].

Function	Variables		Model Coefficients				
f	x	y	a	b	c	d	e
R_m [-]	P_c [atm]	P_e [atm]	−0.88149	3.17521	0.03058	0.03777	−0.3413
M_w [-]	P_c [atm]	R_m [-]	−332.49854	372.70801	0.00045	−36.22978	−0.5195
T_f [K]	P_c [atm]	R_m [-]	−283,663.2214	200,294.10397	0.00045	84,639.97654	0.01976
γ [-]	P_c [atm]	R_m [-]	2.1738	−2.66329	0.00105	1.78059	−0.0395

In the aerodynamics module, all of the unique microlauncher and first-stage configurations are assessed, and multiple aerodatabases are generated that contain the force coefficients of interest needed in the trajectory module. Knowing that a three degrees of freedom (3DOF) dynamic model is implemented for the motion simulation part of the MDO, the following aerodynamic coefficients are computed:

- The axial force coefficient (C_A);
- The normal force coefficient (C_N);
- The drag coefficient (C_D);
- Lift coefficient (C_L).

During the aerodynamic assessments, a decision to split the configuration's aero-contribution into two key categories was made, the first one being related to the "clean configuration" where the microlauncher/first stage can be considered an axisymmetric body of revolution, and the latter one being related to all exterior components (a grid fin-based aerodynamic control system and landing system).

In this way, one can use a validated linearized model for the axisymmetric launcher contribution [10] and CFD results for the exterior components' contribution [11]. For a better understanding, the way in which the aerodatabase computation is split for one of the cases that appear during first-stage recovery is shown in Figure 4. More details can be found in paper [10].

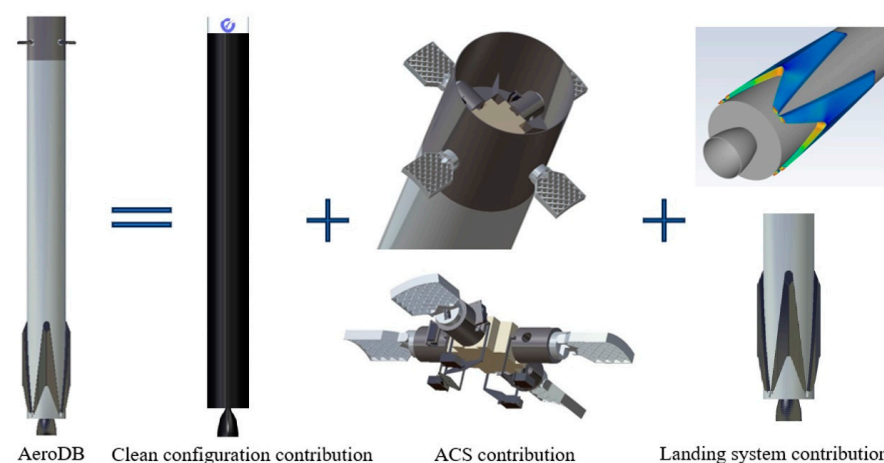


Figure 4. Aerodatabase breakdown of first stage (configuration with ACS deployed).

In the trajectory module, two different missions are simulated in a cascade fashion, the first one being the main one (payload insertion into orbit) and the second one being the recovery mission (of the first-stage assembly). To reduce computational time, the recovery mission is addressed only if the reference trajectory of the main mission ensures good stage separation. Because of the iterative nature of the multidisciplinary optimization process, the time at which key events occur during the mission constantly changes; thus, this module will have the most time-consuming contribution of all main modules.

Using a full six degrees of freedom (6DOF) simulation environment translates into using 21 differential equations for the case of a guided launcher [11]. If one uses a simplified three degrees of freedom (3DOF) dynamic model, this number drastically reduces to six [27]. This simplified model considers the microlauncher to be a point of variable mass and only describes its translational motion. Nevertheless, the motion simulation accuracy is quite high and is recommended to be used during preliminary design activities, where the amount of technical information regarding the launch vehicle is limited [14]. The following 3DOF set of equations is used (quasi-velocity frame) [27]:

$$\begin{aligned}
 \dot{V} &= \frac{N_x}{m} - g_r \sin \gamma - g_\omega (\cos \varphi \cos \chi \cos \gamma + \sin \varphi \sin \gamma) \\
 \dot{\gamma} &= \frac{N_y}{mV} - \frac{g_r}{V} \cos \gamma - \frac{g_\omega}{V} (-\cos \varphi \cos \chi \sin \gamma + \sin \varphi \cos \gamma) + \frac{V}{r} \cos \gamma - 2\Omega_p \cos \varphi \sin \chi \\
 \dot{\chi} &= -\frac{N_z}{mV \cos \gamma} + \frac{g_\omega \cos \varphi \sin \chi}{V \cos \gamma} + \frac{V}{r} \tan \varphi \sin \chi \cos \gamma + 2\Omega_p (\cos \varphi \cos \chi \tan \gamma - \sin \varphi) \\
 \dot{\varphi} &= \frac{V}{r} \cos \chi \cos \gamma; \quad \dot{\lambda} = -\frac{V \sin \chi \cos \gamma}{r \cos \varphi}; \quad \dot{r} = V \sin \gamma
 \end{aligned} \tag{10}$$

with V being the vehicle velocity (relative), γ being the flight path angle, χ being the track angle, φ being the geocentric latitude, λ being the geocentric longitude (relative), r being the distance between Earth and the vehicle center of mass, $[N_x, N_y, N_z]$ being the applied force components, m being the mass of the vehicle (instantaneous), $[g_r, g_\omega]$ being the gravitational acceleration components, and Ω_p being the Earth angular velocity.

A null bank angle μ dynamic model is used [27], where the aerodynamic angles (α and β^*) are seen as control parameters with which the trajectory of the launcher/lower stage can be controlled through the aid of feedback loop relations, such as [10]:

$$\alpha = -k_1(\gamma - \gamma_d); \quad \beta^* = -k_1(\chi - \chi_d). \quad (11)$$

with k_1 being a setting parameter, α being the angle of attack, β^* being the sideslip angle, γ being the flight path angle, and χ being the track angle.

The desired flight path angle γ_d and desired track angle χ_d needed for Equation (11) are two of the optimization variables used inside the MDO algorithm and will continuously change until solution convergence.

An additional constraint related to the maximum normal load is specifically implemented to safeguard the structural integrity of the microlauncher. This constraint ensures that even when the control laws (11) generate high lateral forces during specific maneuvers (active guidance phases), these remain within a range that does not necessitate excessive thrust deflection. While the 3DOF model does not explicitly integrate rotational rates, the resultant force vector accounts for the orientation of the thrust and the lift generated at non-zero angles of attack α and sideslip β^* . It is thus assumed that the vehicle's control system (via thrust vector control) provides sufficient restorative moments to maintain the commanded trim state and balance any generated aerodynamic moment. Of course, in a more detailed design phase, a full 6DOF simulation environment is required.

The projection of the applied forces $[N_x, N_y, N_z]$ are computed independently across propulsion, aerodynamic, and reactive sources within the body frame, as illustrated in Figure 5 for the first configurations of each mission. These components are subsequently aggregated and projected into the quasi-velocity frame using the corresponding transformation matrix [10,27]. More details can be obtained from paper [10].

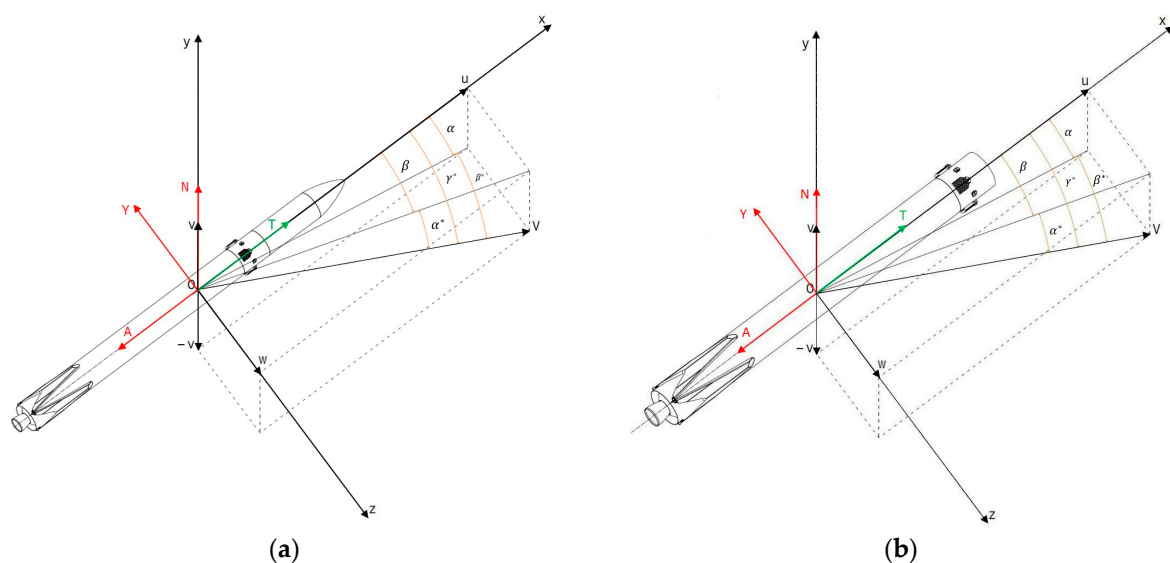


Figure 5. Body frame and aerodynamic angles towards the quasi-velocity frame: (a) microlauncher lift-off configuration; and (b) reusable first-stage configuration after microlauncher separation [10].

The last main module of the MDO is the estimated cost module, where cost estimation relations (CERs) are used to evaluate system-level costs associated with the development, production, and operation of the partially reusable launch vehicles. The mathematical models used are derived from the TransCost model [28], where the main unit of cost measurement is MYr (Man-Year). A typical CER implementation has the following formulation:

$$C = a \cdot M^x \quad (12)$$

where C is a cost, a is a system constant, M is the mass, and x is a sensitivity factor.

To fully understand all of the economic implications related to a specific microlauncher concept, the total cost per launch (C_{pL}) is computed and used as the main deciding factor when making a trade-off analysis for similar configurations. This total cost per launch is obtained by dividing the total development costs (C_{DEV}) by the number of estimated flights/missions ($N_{amortization}$) and adding the average microlauncher production costs (C_{PROD}), the operating costs (both direct C_{DOC} and indirect C_{IOC}), and the expected refurbishment costs (C_{RSC}). Thus, the final relation used to obtain the cost per launch is:

$$C_{pL} = \frac{C_{DEV}}{N_{amortization}} + C_{PROD} + C_{DOC} + C_{IOC} + C_{RSC} \quad (13)$$

3. Microlauncher Main Mission and First-Stage Downrange Recovery Mission Details

A decisive factor for the complexity of the trajectory module and of the entire MDO algorithm is the way in which the reference trajectories are generated. The evolution of the microlauncher and that of the first stage after separation are simulated with the aid of two mission profiles, one for the main mission and another for the recovery mission. The mission profiles integrated inside the MDO tool are shown in Figure 6.

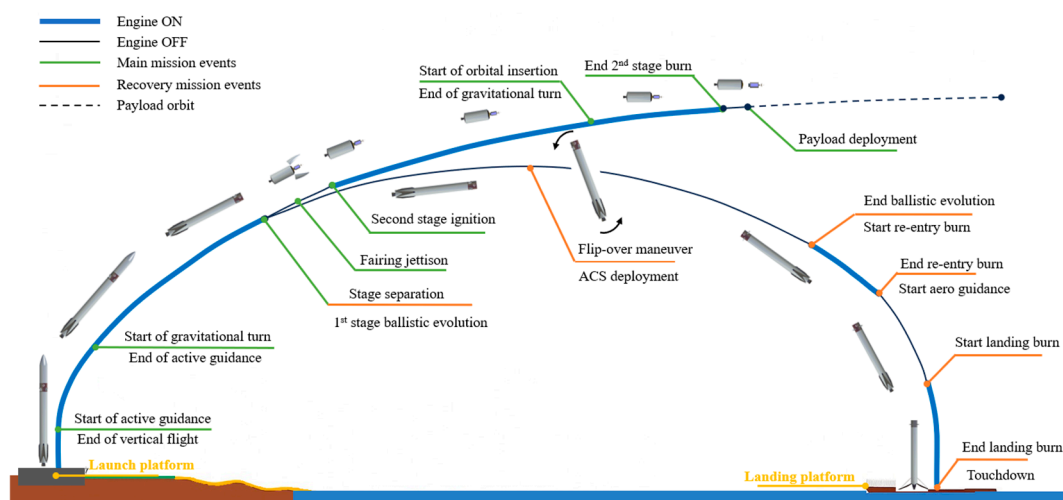


Figure 6. Partially reusable microlauncher mission profiles.

As seen in Figure 6, the main mission of the microlauncher (that of insertion of a payload into a target orbit) is realized with the aid of a direct ascent to orbit (DATO) trajectory. As the mass of the payload is very low (100 kg for the current study) and the altitude of the target orbit corresponds to a low Earth orbit (400 km), the use of a two-burn ascent trajectory is not needed. Also, by integrating a DATO main mission profile, the complexity of the upper stage is somewhat lower because the rocket engine does not need to be restartable (compared to a two-burn scenario). The full list of flight phases and key events that appear during the main mission is given in Table 2.

Table 2. Flight phase and key events for the main mission profile.

Flight Phase/Key Event	Description	Microlauncher Configuration	Engine
1	Vertical flight	Stage 1 + Stage 2 + Full upper structure (VEB + Adapter + Payload + Fairing)	On (first-stage engine)
2	Primary active guidance		On (first-stage engine)
3	Primary gravity turn		On (first-stage engine)
4	Lower-stage separation	Stage 2 + Full upper structure (VEB + Adapter + Payload + Fairing)	Off
5	Fairing jettison		Off
6	Second-stage ignition	Stage 2 + Upper structure (VEB + Adapter + Payload)	On (second-stage engine)
7	Secondary gravity turn		On (second-stage engine)
8	Secondary active guidance		On (second-stage engine)
9	Payload separation		Off

The trajectory module sequentially evaluates each individual flight phase in a cascade assessment. The equations of motion (10) are used regardless of the phase, but the applied forces $[N_x, N_y, N_z]$ are different based on the state of the engine and aero angles.

Even if there are a total of nine flight phases and key events for the main mission definition, the entire trajectory can be simulated with the aid of just six optimization variables (details in Section 2). Two of them are directly linked to the duration of the vertical flight phase (phase 1 from Table 2) and the coast (time between phases 4 and 6 from Table 2) in the form of $t_{vertical}$ and t_{coast} . Two of them quantify the duration of the active guidance phases (phases 2 and 8 from Table 2) in the form of Δt_1 and Δt_2 . The last two are linked to the control schemes associated with the primary active guidance in the form of two target parameters that are needed for Equation (11) in the form of γ_{d1} and χ_{d1} . For the secondary active guidance phase (phase 8), the aerodynamic angles (which are viewed as system-level control parameters) are computed based on an orbit-related reference command that minimizes the time needed for eccentricity decrease (details in [10]).

The recovery mission occurs after stage separation and is depicted in Figure 6. The stage separation event is modeled as an instantaneous state transition within the MDO framework. A downrange recovery has been used, as it was shown in [11] to be the most economically suitable recovery method for a microlauncher concept. The full list of flight phases and key events associated with the recovery mission is given in Table 3.

As can be seen from both Figure 6 and Table 3, two additional first-stage rocket engine restarts are needed to successfully recover the stage downrange in the form of a reentry burn and a landing burn. After stage separation, the stage has a ballistic trajectory, and at its apogee, the RCS is used for the flip-over stage, increasing the angle of attack α from 0° to 180° . This flip-over maneuver is considered to be achieved instantaneously inside the MDO. The ballistic trajectory continues up until reentry burn. After this phase, the ACS is active and, if needed, can alter the attitude of the first stage. Current ACS modeling omits deflection angles as no precise landing coordinates are mandated. Grid fins are utilized exclusively as high-drag aerodynamic braking devices during the recovery mission. This simplification allows the MDO to focus on the sensitivity of the aero-braking force without the computational overhead of additional active control laws. The last burn coincides with the landing burn, and after stage touchdown, the stage velocity and deviation from vertical position are extracted to quantify whether the recovery was done successfully or not.

Table 3. Flight phase and key events for the downrange recovery mission profile.

Flight Phase	Description	Engine	ACS	RCS	Landing System
1	Lower-stage separation	Off	Undeployed	Off	Folded
2	Ballistic evolution 1 ($\alpha = 0^\circ$)			On	
3	Stage flip-over			On	
4	ACS deployment		Deployed	On (minor corrections)	
5	Ballistic evolution 2 ($\alpha = 180^\circ$)				
6	Reentry burn	On		Active	
7	Aerodynamic guidance	Off			
8	Landing maneuver	On			
9	Stage touchdown and engine shutdown	Off	Deployed	Deployed	

Similar to the main mission, a total of nine flight phases and key events are simulated inside the MDO tool for the downrange recovery mission (Table 3) with the aid of just four optimization variables. At recovery mission T0 (stage separation), the propellant mass reserved for the first-stage recovery mission $M_{precovery}$ still existed onboard the tanks and was used for the two burns sequences (reentry and landing). The exact amount of fuel reserved for each maneuver is dictated by the variable $\Delta M_{planding}$. The altitudes at the start of the reentry and landing burns are given by the two optimization variables $H_{reentry}$ and $H_{landing}$. A heat shield is modeled inside the preliminary design module of the MDO such that the thermal loads do not destroy the integrity of the lower part of the first stage.

4. Studied Missions

In this study, a total of five different relevant missions are investigated to quantify their impact on the MDO-proposed reusable microlauncher concept. From these five missions, one has already been studied in [10], and thus, the other four will contain some similarities regarding the altitude of the orbit and the payload mass. The most important aspects studied will be the impact of the launch location and of the target orbit inclination. The main requirements imposed for the five studied missions are shown in Table 4.

Table 4. Main mission requirements.

Requirement	Specification				
	Mission 1	Mission 2	Mission 3	Mission 4	Mission 5
Launch location	Kourou	Omelek Island	Baikonur	Andøya	Andøya
Target orbit type	Quasi-equatorial	Quasi-equatorial	ISS specific	Polar	SSO
Orbit altitude	400 km				
Orbit inclination	~5.2	~9.1	51.6°	90°	97.03°
Payload mass	100 kg				

As can be seen from Table 4, there is a wide variety of launch locations and target orbits studied in this paper. From the five studied missions, two include quasi-equatorial target orbits (mission 1—launch from Kourou, Guiana Space Centre [29] and mission 2—launch from Omelek Island, USA [30]), and one is similar to the International Space Station's orbit [31] (and is launched from Baikonur, Kazakhstan [32]), while the last two include high

inclination orbits such as polar and an SSO (Sun-synchronous orbit) type (both launched from Andøya Space Centre, Norway [33]).

In the case of the Sun-synchronous orbit, the satellite passes over any given point on the planet's surface at the same local time, making it one of the most preferred orbits in imaging applications (Earth observation). The required inclination of the SSO orbit depends on its altitude [34], and in the case of mission 5 (for an altitude of 400 km), the associated SSO orbit has a target inclination of 97.03° . One interesting aspect related to the SSO is the fact that all such orbits have inclinations above 90° , and thus, SSO orbits are retrograde, having a rotation opposite to that of the Earth. This is expected to have a great impact on the propellant mass needed on board the microlauncher, as the launch vehicle must first negate the opposite inertial velocity due to the Earth's rotation.

Another set of important data is given in Table 5, presented here as the values of the target parameters that appear in the formulation (directly or indirectly) of the main mission performance index $I_{mission}$ from Equation (2).

Table 5. Main mission target parameters.

Parameter	Target Value				
	Mission 1	Mission 2	Mission 3	Mission 4	Mission 5
Semi-major axis	$a_{target} = 6778.137 \text{ km}$				
Eccentricity	$e_{target} = 0$				
Velocity (inertial)	$V_{target} = 7.6686 \text{ km/s}$				
Orbit inclination	-	-	51.6°	90°	97.03°
Flight path angle	$\gamma_{target} = 0^\circ$				

One can observe from Table 5 that most of the target parameters are the same across all five studied missions. One key difference that appears in the computation of $I_{mission}$ is the lack of an imposed orbit inclination for the quasi-equatorial missions (1 and 2). This aspect is implemented inside the MDO in the form of a null weight associated with its parameter ($w_i = 0$). The reason behind it can also be seen in the initial conditions depicted in Table 6. A -90° track angle is used for the first two missions to fully benefit from the angular velocity due to Earth's rotation. In the end, by using such a launch corridor (fully Eastward), the final orbit inclination is similar to the latitude of the launch location, data which is numerically presented in Table 6 for all five missions of interest. This will also impact the number of optimization variables, for the first two missions had no need for the exitance of the primary active guidance phase desired track angle χ_{d1} .

Table 6. Studied missions' initial conditions.

Parameter	Target Value				
	Mission 1	Mission 2	Mission 3	Mission 4	Mission 5
Launch location	Kourou	Omelek Island	Baikonur	Andøya	Andøya
Latitude $\varphi_{initial}$	5.21	9.05	45.6	69.29	69.29
Longitude $\lambda_{initial}$	-52.77	167.74	63.4	16.02	16.02
Flight path angle $\gamma_{initial}$	90°				
Track angle $\chi_{initial}$	-90	-90	-61.43	1.23	21.41

For the last three missions, the launch direction (given by the initial track angle $\chi_{initial}$) is computed based on a set of relations detailed in [10,35]:

$$\beta_{inertial} = \arcsin\left(\frac{\cos(i_{target})}{\cos(\varphi_{initial})}\right); \quad (14)$$

$$\beta_{launch} = \arctan\left(\frac{V_{target} \sin(\beta_{inertial}) - V_{eqrot} \cos(\varphi_{initial})}{V_{target} \cos(\beta_{inertial})}\right); \quad \chi_{initial} = -\beta_{launch}.$$

where V_{eqrot} is the rotation speed at the Earth's equator (approximately 465.1 m/s).

Regarding the reusable microlauncher architecture [10], Table 7 presents the main specifications implemented for the preliminary design of the constructive solution.

Table 7. Reusable microlauncher architecture for all missions.

Specification	Type
Number of stages	2
Recoverable stages	1 (first stage)
Expendable stages	1 (second stage)
Stages diameter	Constant
Propellant	LOX/Methane
Components length safety margin	10%
Dry mass safety margin	Stage 1: 0% Stage 2: 5%
Propellant mass safety margin:	Stage 1: 5% of recovery mission propellant mass Stage 2: 0%
Fairing separation condition	Dynamic pressure drops below 0.5 kPa [11]

A very important aspect that must be addressed is the imposed constraints implemented for the microlauncher concept and trajectory propagation (both for the main mission and for the recovery mission), as they will directly impact the value of the objective function by means of assessing the constraint index $I_{constraints}$ from Equations (4) and (5). To ensure a good comparison between the already accessible results [10] and the ones from this paper, the constraints are the ones shown in Table 8.

Table 8. Constraints used for microlauncher optimization (same as paper [10]).

Parameter	Constraint	Description
n_{axial}	≤ 11	Axial load factor under the value of 11
n_{normal}	≤ 0.75	Normal load factor under the value of 0.75
$\left \frac{D_{e1} - D_{e2}}{D_{e2}} \right $	≤ 0.0001	Small difference between consecutive stages dimeters ($\approx$ stages of the same diameter)
$\frac{L_{e1}}{D_{e1}}$	≤ 10	Maximum fineness ratio of first stage (computed without interstage) of 10
$\frac{L_{e2}}{D_{e2}}$	≤ 5	Maximum fineness ratio of second stage of 5
ε	$5 \leq \varepsilon \leq 90$	Nozzle expansion ratio between 5 and 90
γ_{d1}	$\subset \gamma_{active_guidance1}$	Achievement of the imposed trajectory flight path angle and track angle during primary active guidance phase
χ_{d1}	$\subset \chi_{active_guidance1}$	

Table 8. Cont.

Parameter	Constraint	Description
$\left \frac{\gamma_{d1} - \gamma_{final_guidance1}}{\Delta t_{\approx constant}} \right $	$\leq 0.25^\circ/\text{s}$	Deviation from the required flight path angle and track angle less than $0.25^\circ/\text{s}$
$\left \frac{\chi_{d1} - \chi_{final_active_guidance1}}{\Delta t_{\approx constant}} \right $	$\leq 0.25^\circ/\text{s}$	(For a short period of time, these angles are maintained quasi-constant such that the velocity vector aligns with the thrust vector)
$V_{landing}$	$\leq 3 \text{ m/s}$	Maximum velocity at stage touchdown of 3 m/s
$\Delta\gamma_{landing}$	$\leq 2^\circ$	Maximum deviation from vertical position at stage touchdown of 2°

Before running the MDO algorithm, one must define the limits of the solution search space. The implemented numerical values for all five missions are given in Table 9, being similar to those used in [10] (which studied a mission identical to number 4 of this paper).

Table 9. Solution search space used for microlauncher optimization (same as paper [10]).

Lower Bound	Optimization Variable	Upper Bound	Lower Bound	Optimization Variable	Upper Bound
5	M_{p1}	25	0.01	P_{e2}	0.15
1	D_{e1}	2	0.4	$\left(\frac{T}{W}\right)_2$	1.5
60	P_{c1}	80	2	$t_{vertical}$	100
0.3	P_{e1}	1	2	t_{coast}	100
1	$\left(\frac{T}{W}\right)_1$	5	0.01	Δt_1	0.5
50	γ_{d1}	85	0.01	Δt_2	0.5
$\chi_{initial} - 5$	χ_{d1}	$\chi_{initial} + 5$	0.1	$M_{precovery}$	1.0
0.5	M_{p2}	5	50	$H_{reentry}$	100
1	D_{e2}	2	0.01	$\Delta M_{p_{landing}}$	1
60	P_{c2}	80	0.01	$H_{landing}$	5

5. Results

5.1. MDO Convergence

The five missions presented in Section 4 were analyzed with the aid of the MDO algorithm developed so that a partially reusable microlauncher constructive solution that minimizes its lift-off mass can be generated. The solution (optimization variables vector) is considered converged when 100 consecutive generations have passed without objective function improvement. Some key details related to the MDO convergence process and final values of important parameters are given in Table 10.

The MDO convergence process is graphically presented in Figure 7, and the numerical values of the objective function are depicted as the genetic algorithm (ga) advances the solution. Because of the high values that occur during the first generations (as seen in Figure 7a), a zoom representation after 100 generations is also given in Figure 7b.

For all five cases analyzed, an initial population of 200 individuals was used (10 times the optimization variables vector size), with a 5% elite rate. The number of generations needed to achieve convergence ranged from 1875 (mission 3) to 4361 (mission 2), which corresponds to a total CPU time of between 50 min and 2 h. A dedicated workstation with an Intel i9-14900K processor was used (parallel computing with 24 workers).

Table 10. Studied missions' MDO details.

Solution Convergence Details	Mission 1	Mission 2	Mission 3	Mission 4	Mission 5
Number of ga generations	1888	4361	1875	3996	2065
Number of iterations	358,930	828,800	356,460	759,450	392,560
Computational time needed [min]	50	112	50	120	62
Objective function $f_{objective}$	10.3571	10.4909	12.4079	14.8309	15.4789
Lift-off mass M_{start} [t]	10.3566	10.4843	12.3923	14.8252	15.3758
Recovery mission propellant mass [kg]	131.9337	134.7215	165.1242	191.3285	199.7048
Main mission performance index $I_{mission}$	0.0006	0.0065	0.0132	0.0057	0.1031
Recovery mission performance index $I_{recovery}$	0	0	0.0024	0	0
Constraint index $I_{constraints}$	1	1	1	1	1

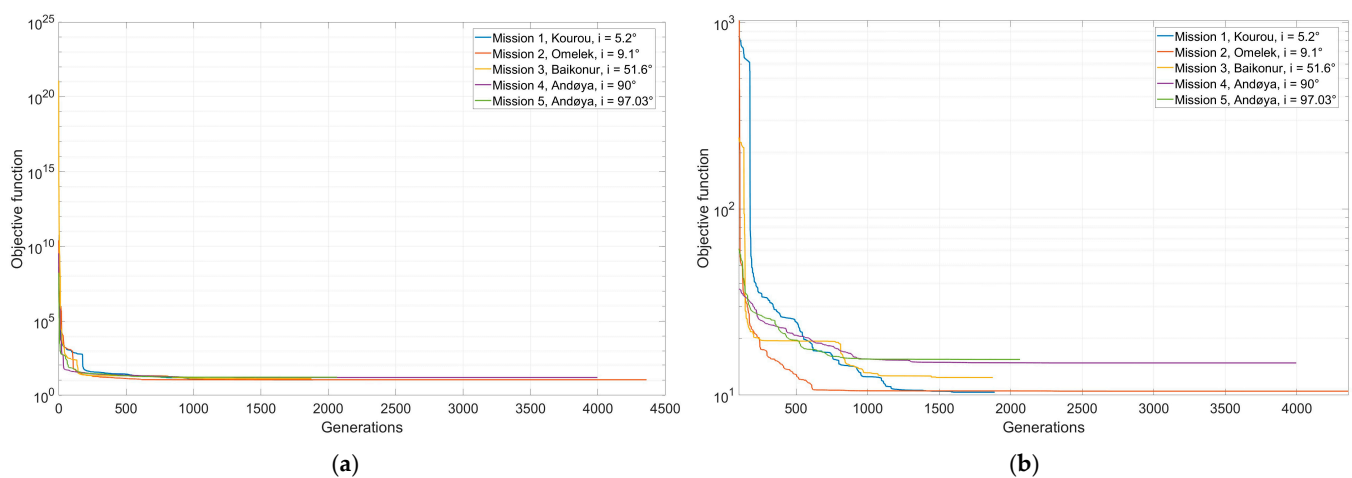
**Figure 7.** Objective function convergence (a) full data; (b) zoom after 100 generations.

Table 10 also presents the final values associated with the parameters that are used inside the objective function definition. It is also of interest to show the graphical evolution of these parameters with the genetic algorithm generations because it illustrates the way in which the optimized solution is obtained. This is realized in Figure 8.

As seen in both Figures 7 and 8, the microlauncher is optimized rapidly in the first hundred generations, and afterwards, a slow refining process occurs. Because of the multiplicative nature of the constraint index's influence on the objective function (see Equation (1)), this is the first parameter that the MDO reduces towards its minimal value ($I_{constraints} = 1$), as depicted in Figure 8d. The next major term that is minimized during multidisciplinary optimization is M_{start} , which corresponds to the lift-off mass of the microlauncher (including the 100 kg payload). One can see from Figure 8a that this process also occurs in the first couple of hundred generations. It can be seen that the MDO algorithm starts with low lift-off mass launch vehicles which cannot achieve an acceptable orbit insertion (the main mission performance index convergence is shown in Figure 8b) and steadily increases its mass towards the minimum value. This approach is beneficial to the overall convergence, as the risk of a solution plateauing into a local minimum is reduced.

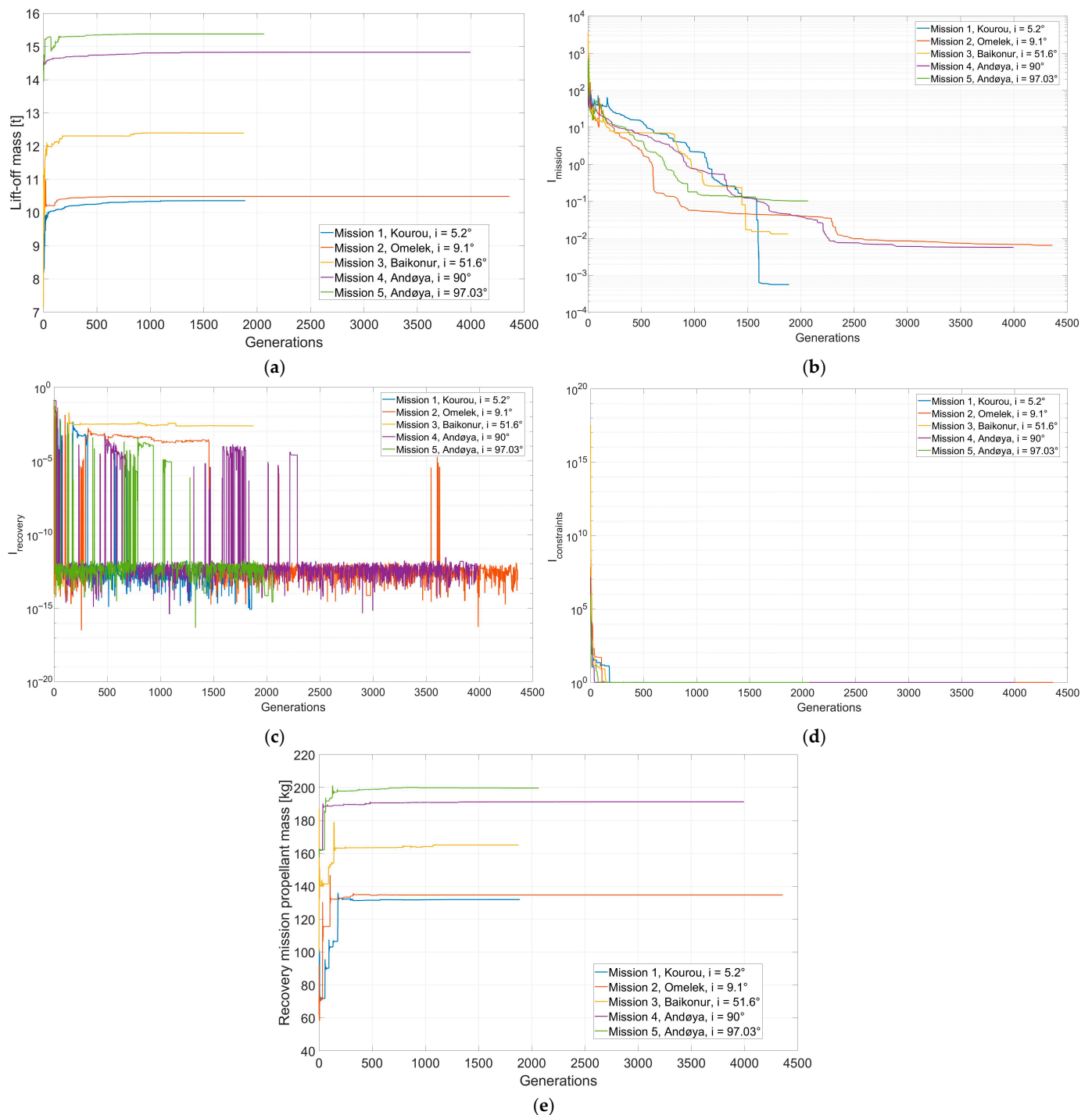


Figure 8. MDO convergence: (a) lift-off mass; (b) main mission performance index; (c) recovery mission performance index; (d) constraint index; and (e) recovery mission propellant mass.

The value of the recovery mission performance index $I_{recovery}$ depicted in Figure 8c is somewhat small compared to the others presented before, as it only consists of the extra propellant mass that is not used during the recovery mission (see Equation (3)). The impact of this term is much lower than the main mission performance index, but still has an impact on the value of the propellant mass reserved for the recovery mission (depicted in Figure 8e). The propellant mass needed for a successful recovery mission ranges from approximately 132 kg (mission 1) to almost 200 kg (mission 5) and is directly linked to the mass of the first stage and of the entire microlauncher concept.

The final numerical values of the 20 optimization variables used inside the MDO are presented in Table 11. As a reminder, the solution is considered converged when no objective function improvement has been obtained in 100 consecutive generations (which corresponds to ~19,000 distinct optimization variables vectors). A detailed comparison based on the data from Table 11 will be made in the next sections of this paper.

Table 11. Partially reusable microlaunchers' final optimization variables.

Optimization Variable	Value				
	Mission 1	Mission 2	Mission 3	Mission 4	Mission 5
M_{p1} [t]	7.5566	7.6644	8.9101	10.8690	11.2985
D_{e1} [m]	1.2290	1.2314	1.3090	1.3853	1.4060
P_{c1} [bar]	79.8920	79.8920	79.8675	79.9843	79.9428
P_{e1} [bar]	0.4810	0.5534	0.6478	0.5018	0.5150
$\left(\frac{T}{W}\right)_1$ [-]	2.2951	2.2136	2.5163	2.1168	2.3830
γ_{d1} [°]	55.2406	51.4510	56.0641	51.9744	53.4800
χ_{d1} [°]	-	-	-60.5106	2.4987	22.6400
M_{p2} [t]	1.0183	1.0352	1.3873	1.5391	1.5509
D_{e2} [m]	1.2289	1.2313	1.3083	1.3852	1.4058
P_{c2} [bar]	79.7122	79.7960	79.6337	79.9310	79.8289
P_{e2} [bar]	0.0580	0.0623	0.0784	0.0586	0.0667
$\left(\frac{T}{W}\right)_2$ [-]	0.6777	0.6674	0.7096	0.7045	0.7019
$t_{vertical}$ [s]	9.4346	12.5730	7.0997	13.7100	8.3180
t_{coast} [s]	8.0268	6.2200	8.5301	2.3009	5.9578
Δt_1 [-]	0.3048	0.3781	0.3095	0.3955	0.3361
Δt_2 [-]	0.3694	0.2978	0.4423	0.3308	0.3112
$M_{precovery}$ [t]	0.1319	0.1347	0.1651	0.1913	0.1997
$H_{reentry}$ [km]	52.7225	53.5996	51.2717	51.8140	51.6423
$\Delta M_{planding}$ [-]	0.9998	0.9942	0.9904	0.9978	0.9983
$H_{landing}$ [km]	0.1624	0.1715	0.1588	0.1848	0.1660

5.2. Reusable Microlauncher Concepts

Based on the values of the optimization variables presented in Table 11, the entire constructive solution of the reusable microlauncher can be generated according to the mathematical models implemented inside the preliminary design module. The key specifications of the family of reusable microlauncher concepts are given in Table 12.

Table 12. Reusable microlauncher specifications.

	Value				
	Mission 1	Mission 2	Mission 3	Mission 4	Mission 5
Lift-off mass [t]	10.36	10.48	12.39	14.83	15.38
Microlauncher length [m]	18.10	18.13	19.02	20.50	20.72
Exterior diameter [m]	1.23	1.23	1.31	1.39	1.41
Payload mass [kg]	100				
Propellant pair	O ₂ + CH ₄				
Upper structure mass [kg]	176.98	177.49	186.54	197.01	199.47
Upper structure length [m]	1.80	1.81	1.92	2.03	2.06

Table 12. Cont.

	Value				
	Mission 1	Mission 2	Mission 3	Mission 4	Mission 5
Second-stage mass [t]	1.16	1.18	1.57	1.75	1.76
Second-stage length [m]	3.13	3.13	3.43	3.62	3.60
Second-stage burn time [s]	392.24	398.42	384.74	392.34	392.62
Second-stage mean thrust [kN]	8.90	8.89	12.25	13.43	13.48
First-stage mass (+interstage) [t]	9.02	9.13	10.63	12.88	13.42
Recovery components mass [t]	0.52	0.52	0.60	0.70	0.73
Propellant mass reserved for recovery mission [t]	0.13	0.13	0.17	0.19	0.20
First-stage length (+interstage) [m]	13.60	13.61	14.13	15.38	15.57
First-stage burn time [s]	89.14	93.22	81.03	97.32	86.75
First-stage mean thrust [kN]	254.41	246.36	327.48	335.41	390.44

With the aid of the data presented in Table 12, one can compute the payload performance index using:

$$I_{payload} = \frac{m_{payload}}{M_{start}} \quad (15)$$

The main impact of the different requirements used for the five missions analyzed on the partially reusable microlauncher concept is presented in Table 13.

Table 13. Mission impact on microlauncher lift-off mass and payload performance index.

Mission Number	Orbit Altitude [km]	Payload Mass [kg]	Launch Location	Orbit Inclination	Microlauncher Lift-Off Mass	Payload Performance Index
1	400	100	Kourou	5.2°	10.36	0.97
2			Omelek	9.1°	10.48	0.95
3			Baikonur	51.6°	12.39	0.81
4			Andøya	90°	14.83	0.67
5			Andøya	97.03°	15.38	0.65

Based on the data depicted in Table 13, one can observe the fact that the inclination of target orbit is directly linked to the lift-off mass of the launcher (and payload performance index) for the same orbit altitude and payload mass, and it seems that the correlation is quasi-linear. This extends to other mass-related specifications, such as individual stage mass (Tables 14 and 15), but also to sizing aspects regarding the additional propellant mass needed (higher length, increased stage diameter, etc.) and propulsive parameters (higher thrust). Inside the preliminary design module of the MDO, a simplified “clean configuration” is also generated for the entire microlauncher (in Matlab), geometry which is presented in Figure 9, showing the entire family of reusable launcher concepts.

Based on the entire sizing output database that is provided by the MDO (at the individual component level, one can consult paper [10] for a full list of data), a more complex geometry is obtained after a CAD modeling process. The external CAD models realized for the family of five reusable microlauncher concepts are shown in Figure 10a, together with a representation of an average human (height = 1.8 m) for a better understanding of the scale of the launch vehicles. Figure 10b also presents an internal CAD view for one of the concepts, courtesy of paper [10].

Table 14. Reusable microlauncher family, first-stage mass breakdown.

Breakdown Scheme	Mass [kg]				
	Mission 1	Mission 2	Mission 3	Mission 4	Mission 5
Liquid-propellant	7556.63	7664.35	8910.08	10,869.00	11,298.51
Oxidizer (LOX)	5566.02	5644.16	6560.01	8005.40	8321.38
Fuel (Methane)	1990.61	2020.19	2350.08	2863.60	2977.14
Dry mass (expendable concept components)	942.27	941.82	1118.50	1307.77	1388.96
Oxidizer tank	97.52	98.64	114.88	132.17	135.42
Fuel tank	94.43	95.58	111.42	128.00	131.1
Oxidizer turbopump	52.19	51.30	61.05	62.01	67.91
Fuel turbopump	51.09	50.24	59.83	60.71	66.50
Rocket engine	118.07	109.57	147.61	164.05	197.05
Additional components	528.96	536.50	623.70	760.83	790.89
Additional dry mass (due to stage recovery)	519.12	520.47	603.14	704.64	729.22
Aerodynamic control system (ACS)	129.10	129.73	151.15	174.11	180.73
Reaction control system (RCS)	35.16	35.35	44.36	59.62	64.29
Enlarged interstage	96.98	96.31	108.94	126.89	127.70
Foldable landing system	208.38	209.40	243.98	281.09	291.72
Reentry heat shield	49.51	49.69	54.71	62.90	64.78
Total Stage 1 Mass	9018.03	9126.64	10,631.72	12,881.42	13,416.70

Table 15. Reusable microlauncher family, second-stage mass breakdown.

Breakdown Scheme	Mass [kg]				
	Mission 1	Mission 2	Mission 3	Mission 4	Mission 5
Liquid-propellant	1018.28	1035.21	1387.31	1539.07	1550.94
Oxidizer (LOX)	752.42	764.93	1025.09	1137.27	1146.02
Fuel (Methane)	265.87	270.28	362.23	401.81	404.92
Dry mass (expendable concept components)	143.27	145.00	186.72	207.66	208.69
Oxidizer tank	24.62	24.89	30.26	34.05	34.22
Fuel tank	23.86	24.12	29.32	32.99	33.16
Oxidizer turbopump	6.56	6.57	8.02	8.46	8.50
Fuel turbopump	6.37	6.38	7.80	8.22	8.26
Rocket engine	3.75	3.66	5.31	6.31	6.05
Additional components	71.28	72.46	97.11	107.33	108.57
Safety margin	6.82	6.90	8.89	9.89	9.94
Total Stage 2 Mass	1161.55	1180.21	1574.04	1746.74	1759.64

Based on the output data of the MDO algorithm, the mass breakdown of the first stage is shown in Table 14, while that of the second stage is depicted in Table 15. It can be clearly seen that most of the individual component masses of both stages are proportional to the target orbit inclination (which is higher as the mission number increases).

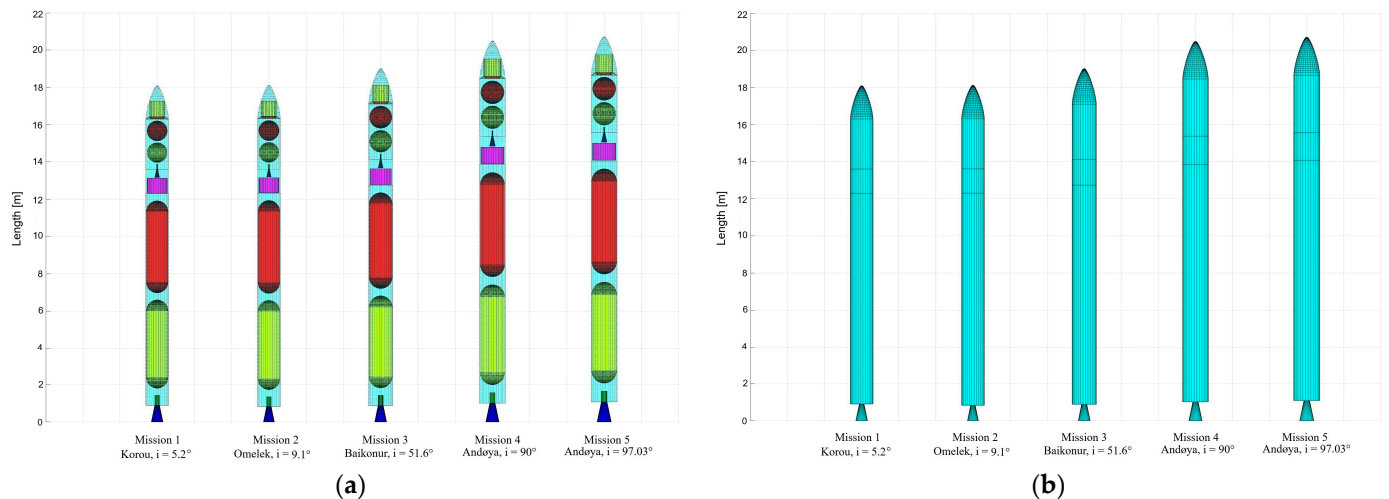


Figure 9. Preliminary design of “clean configurations”: (a) interior components; and (b) external view.

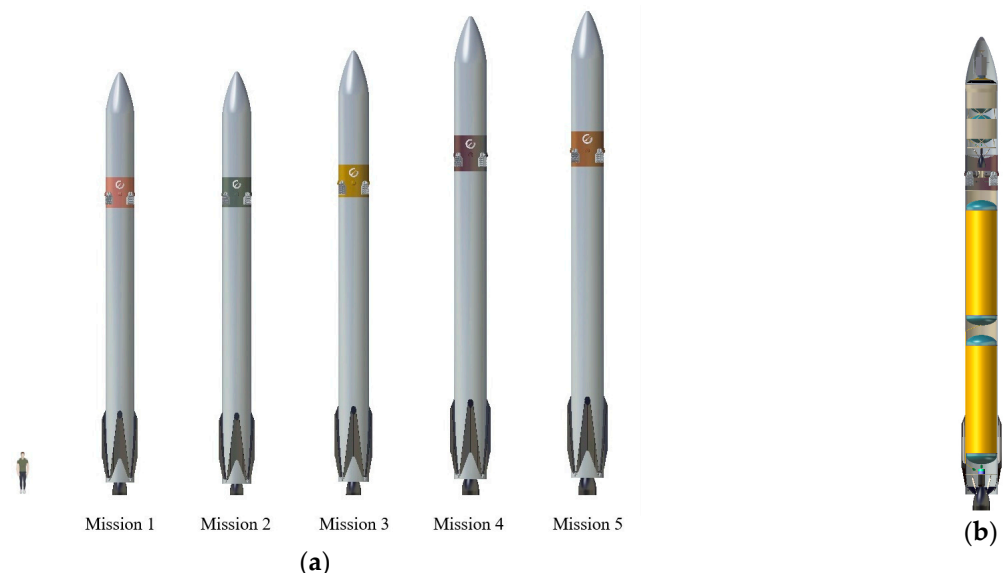


Figure 10. CAD model: (a) partially reusable microlauncher family, external view, and (b) interior components for mission 4 microlauncher [10].

5.3. Mission Analysis

After MDO convergence and microlauncher constructive solution generation, in the post-processing phase, several output databases related to the motion of the microlauncher and first stage are generated. Here, based on the reference trajectories that are optimized inside the trajectory module of the MDO, a complete mission analysis is realized for both the main mission (DATO trajectory) and the recovery mission (downrange autonomous vertical landing trajectory). These will now be briefly presented.

5.3.1. DATO Mission Profiles

The mission analysis of every microlauncher concept is presented in Tables 16–20. One can observe the fact that the duration of the microlaunchers' main mission is almost the same for all five studied cases (around 8 min), ranging from 474 to 497 s. Max q moment (maximum dynamic pressure) usually occurs at the 10–11 km mark and at Mach numbers around the value 2. The fairing jettison occurs during the coast period after lower-stage separation and before upper-stage engine ignitions to limit any dynamic instabilities. One can also observe the different final local altitudes at which the payload is separated from

the upper structure (ranging from 400.16 to 421.16 km), with this phenomenon appearing because the target altitude (global) is measured from the Equator.

Table 16. Mission 1 partially reusable microlauncher concept's main mission analysis.

Key Event	Start Reference Time [s]	Local Altitude [km]	Mach Number [-]	Microlauncher Mass [kg]
Microlauncher lift-off	0	0	0	10,356.55
Primary active guidance	9.43	0.60	0.38	9570.69
Primary gravity turn	33.73	7.49	1.64	7546.39
Max-Q	40.10	10.36	2.12	7016.57
Lower-stage separation	89.14	58.59	9.22	2931.85
Fairing jettison	94.29	67.36	9.60	1338.53
Second-stage ignition	97.16	72.15	9.83	1305.53
Orbital insertion maneuver	344.49	354.32	5.30	662.97
Payload separation	489.40	400.16	8.43	287.25

Table 17. Mission 2 partially reusable microlauncher concept's main mission analysis.

Key Event	Start Reference Time [s]	Local Altitude [km]	Mach Number [-]	Microlauncher Mass [kg]
Microlauncher lift-off	0	0	0	10,484.35
Primary active guidance	12.57	1.01	0.49	9468.77
Max-Q	42.14	10.93	2.11	7080.35
Primary gravity turn	43.06	11.47	2.19	6991.47
Lower-stage separation	93.22	60.92	9.28	2954.72
Fairing jettison	96.99	67.31	9.56	1357.70
Second-stage ignition	99.44	71.39	9.76	1324.58
Orbital insertion maneuver	379.21	371.18	5.71	592.99
Payload separation	497.86	400.47	8.43	289.36

Table 18. Mission 3 partially reusable microlauncher concept's main mission analysis.

Key Event	Start Reference Time [s]	Local Altitude [km]	Mach Number [-]	Microlauncher Mass [kg]
Microlauncher lift-off	0	0	0	12,392.29
Primary active guidance	7.10	0.39	0.33	11,626.10
Primary gravity turn	29.98	6.79	1.65	9153.54
Max-Q	38.10	10.67	2.33	8281.08
Lower-stage separation	81.03	55.28	8.74	3647.33
Fairing jettison	87.62	66.77	9.19	1760.57
Second-stage ignition	89.56	70.09	9.34	1723.17
Orbital insertion maneuver	304.15	343.67	4.87	939.59
Payload separation	474.30	412.77	8.59	335.86

Table 19. Mission 4 partially reusable microlauncher concept's main mission analysis.

Key Event	Start Reference Time [s]	Local Altitude [km]	Mach Number [-]	Microlauncher Mass [kg]
Microlauncher lift-off	0	0	0	14,825.17
Primary active guidance	13.71	1.11	0.49	13,321.01
Max-Q	43.34	10.96	2.06	10,070.64
Primary gravity turn	46.78	12.81	2.32	9670.74
Lower-stage separation	97.32	65.18	9.62	4147.50
Fairing jettison	98.67	67.55	9.73	1943.75
Second-stage ignition	99.63	69.21	9.81	1901.83
Orbital insertion maneuver	362.37	379.23	5.64	871.29
Payload separation	492.27	421.16	8.91	362.75

Table 20. Mission 5 partially reusable microlauncher concept's main mission analysis.

Key Event	Start Reference Time [s]	Local Altitude [km]	Mach Number [-]	Microlauncher Mass [kg]
Microlauncher lift-off	0	0	0	15,375.80
Primary active guidance	8.32	0.50	0.36	14,311.61
Primary gravity turn	34.68	8.43	1.87	10,907.67
Max-Q	40.01	10.94	2.32	10,256.77
Lower-stage separation	86.75	59.67	9.56	4276.99
Fairing jettison	91.34	67.84	9.94	1959.11
Second-stage ignition	92.71	70.24	10.06	1915.93
Orbital insertion maneuver	363.14	384.30	5.85	844.20
Payload separation	485.33	420.89	8.98	364.99

Another interesting set of data that must be provided is that of the payload insertion errors that occur (vs. the target orbit in Table 5) with the current optimized launch vehicle and reference trajectory. These are given in Table 21 for all five missions.

Table 21. Partially reusable microlauncher concepts' payload insertion errors.

Parameter	Payload Insertion Error				
	Mission 1	Mission 2	Mission 3	Mission 4	Mission 5
Semi-major axis (a_{target}) [km]	-2.50×10^{-4}	-4.17×10^{-3}	1.43×10^{-2}	-1.11×10^{-3}	-8.13×10^{-2}
Velocity (inertial) (V_{target}) [m/s]	-2.49×10^{-5}	1.71×10^{-4}	7.25×10^{-6}	5.99×10^{-5}	-4.76×10^{-4}
Flight path angle (γ_{target}) [°]	5.57×10^{-5}	6.12×10^{-4}	8.66×10^{-4}	3.17×10^{-5}	9.28×10^{-3}
Orbit inclination (i_{target}) [°]	-	-	7.01×10^{-4}	5.70×10^{-4}	1.95×10^{-3}
Eccentricity (e_{target}) [-]	9.10×10^{-7}	1.01×10^{-5}	1.45×10^{-5}	5.58×10^{-7}	1.63×10^{-4}

The deviations from target orbits are low enough to consider that the main mission of the microlauncher has been successfully achieved, with the biggest errors appearing for the orbit semi-major axis (80 m for mission 5). Next, Figure 11 depicts the variations in several parameters of interest with time for referencing ascent trajectories (all five missions). These parameters are the ones that directly appear in the objective function definition.

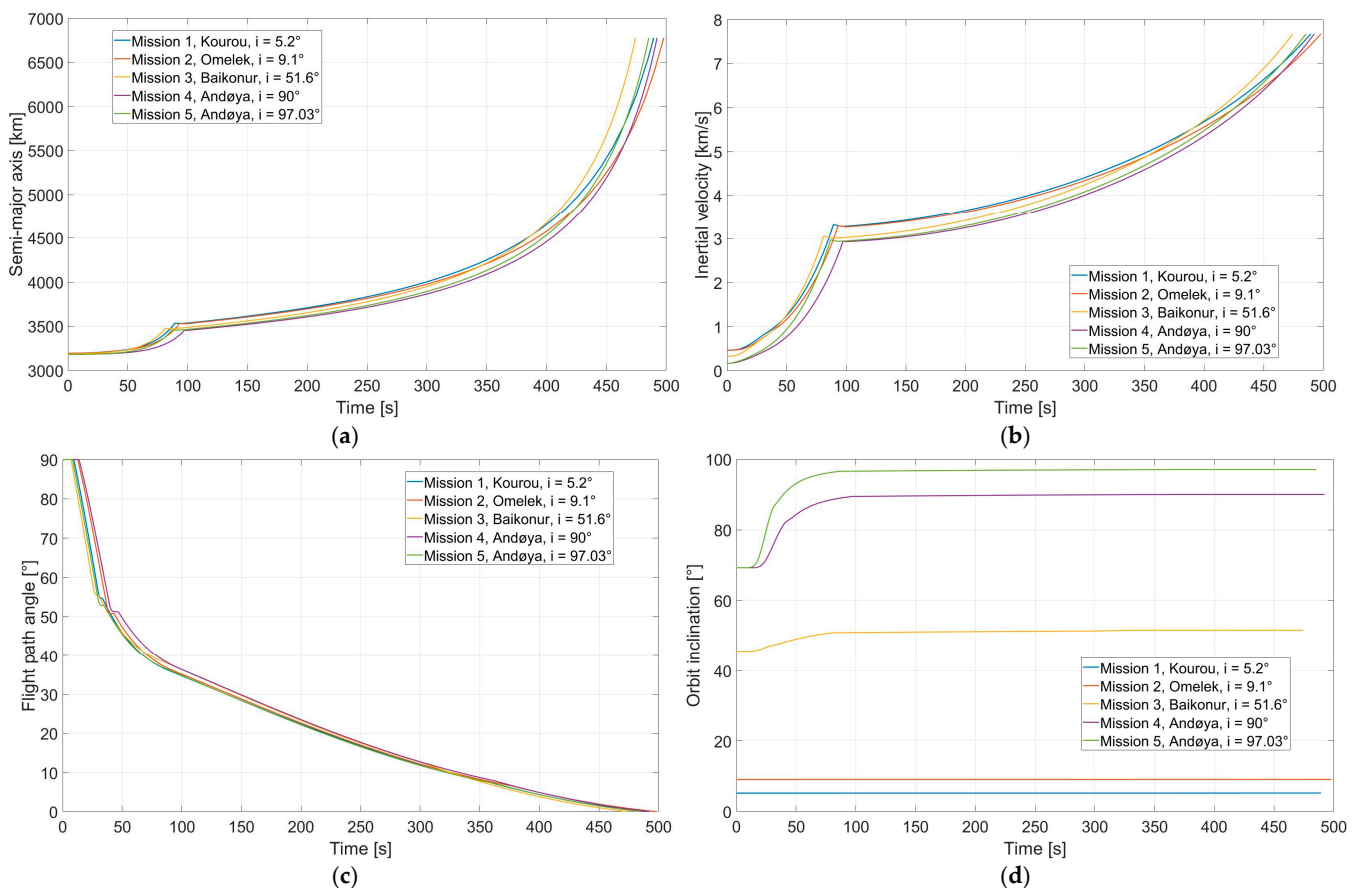


Figure 11. Reference main mission ascent trajectories: (a) semi-major axis vs. time; (b) inertial velocity vs. time; (c) flight path angle vs. time; and (d) orbit inclination vs. time.

It can be observed that the ascent trajectories are similar for all five studied missions, with this being due to the identical target orbit altitude and payload mass. Notable differences are, of course, seen in the orbital inclination graph (Figure 11d), and the main difference in the mass of the launchers generated by the MDO algorithm is given by the need to increase the target orbit inclination over the launch location latitude.

Another interesting set of plots is given in Figure 12 for all five nominal ascent trajectories. Figure 12a showcases the altitude variation with time, while Figure 12b presents the orbit eccentricity. As can be seen from these two figures (and correlated with the insertion errors of Table 21), all of the five optimized microlauncher solutions are capable of reaching a low circular Earth orbit.

As stated earlier, additional constraints related to the maximum load factors are specifically implemented to safeguard the structural integrity of the microlauncher (Table 8). This can be seen in Figure 12e,f. To be sure that the TVC can deflect the engine nozzle in a real 6DOF environment in such a way that the thrust vector is modeled correctly in our 3DOF environment, the control laws (11) have built-in maximum operating limits (as proposed in [10,27]). Thus, the aerodynamic angles are limited to maximum values of $\pm 7^\circ$, bounds which are clearly seen in Figure 12c,d.

The equations of motion (10) have been further integrated for the case of the payload (after separation) to showcase the Earth coverage obtained with each mission type. Figure 13 shows the Mercator projection of the satellite in the first 24 h. Having a higher surface coverage, especially for Earth observation satellites, could overcome the drawbacks of higher lift-off mass (and costs, see Section 5.4) needed to insert such satellites into LEO.

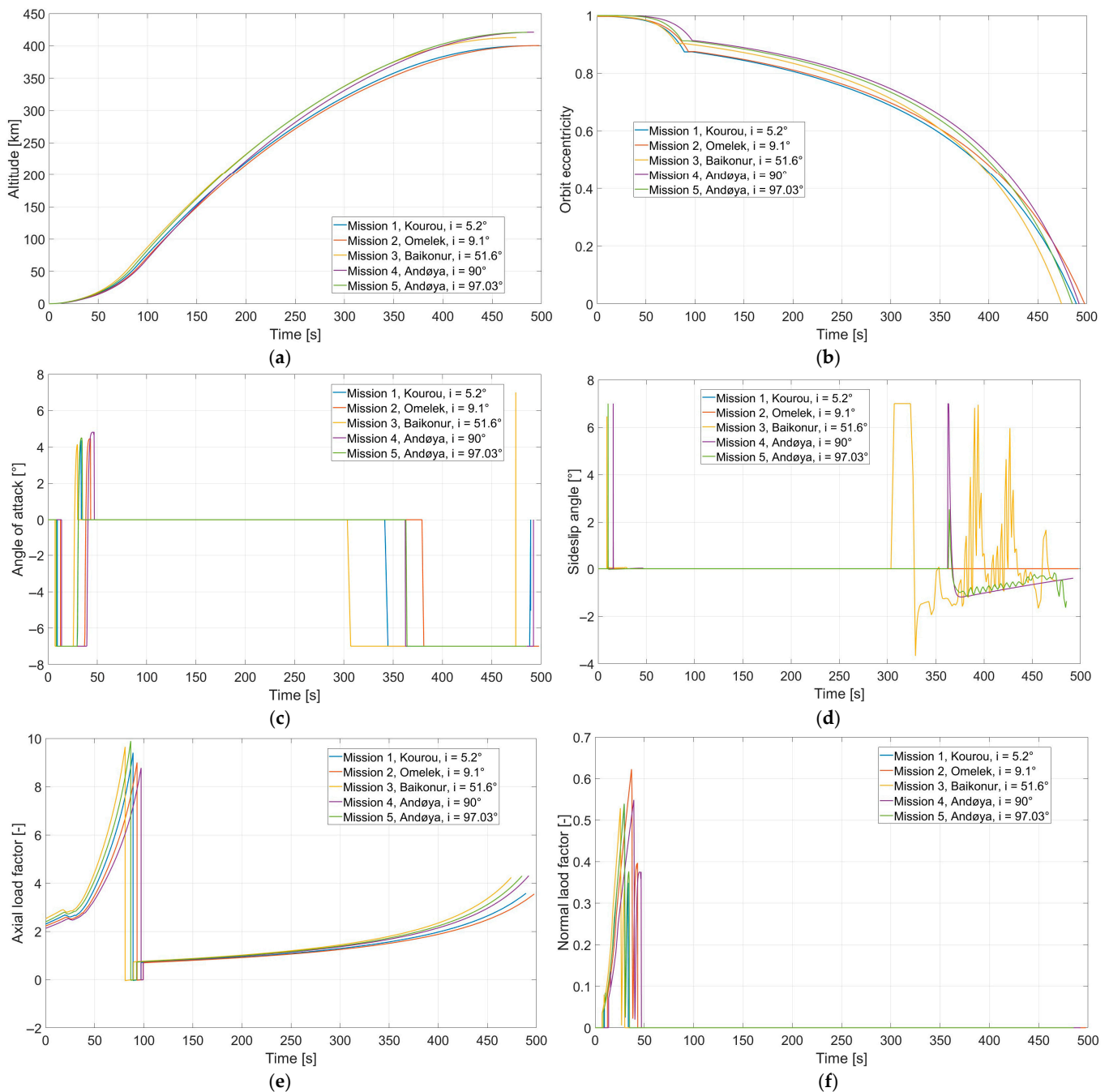


Figure 12. Reference main mission ascent trajectories: (a) altitude vs. time; (b) orbit eccentricity vs. time; (c) angle of attack vs. time; (d) sideslip angle vs. time; (e) axial load factor vs. time; and (f) normal load factor vs. time.

5.3.2. Recovery Mission Profiles

Similar to the main mission details of Section 5.3.1, the recovery mission analysis will now be presented. Because of the formulation used for the objective function, the recovery mission can be viewed as a separate optimization problem, in which a successful recovery is desired (the stage touchdown conditions must be under a velocity and deviation from vertical position threshold, mentioned in Table 8) that uses minimum propellant mass.

The recovery mission analysis of every microlauncher concept is presented in Tables 22–26. A somewhat similar recovery mission duration is observed across all five missions' launch vehicle concepts (around 8 min), which means that the time between microlauncher lift-off and reusable stage touchdown is around 9.5–10 min. One can also

observe the fact that similar stage separation conditions occur for all five missions investigated, with the altitude being around 60 km, the velocity being around 3 km/s, and the flight path angle being around 36° . At stage touchdown, the first-stage tanks still contain some propellant, which was previously imposed in Table 7 (safety margin of 5% from the total recovery mission propellant mass). For mission 3, an additional 2.4 kg of propellant mass was observed, which could later be refined in a more advanced phase of design.

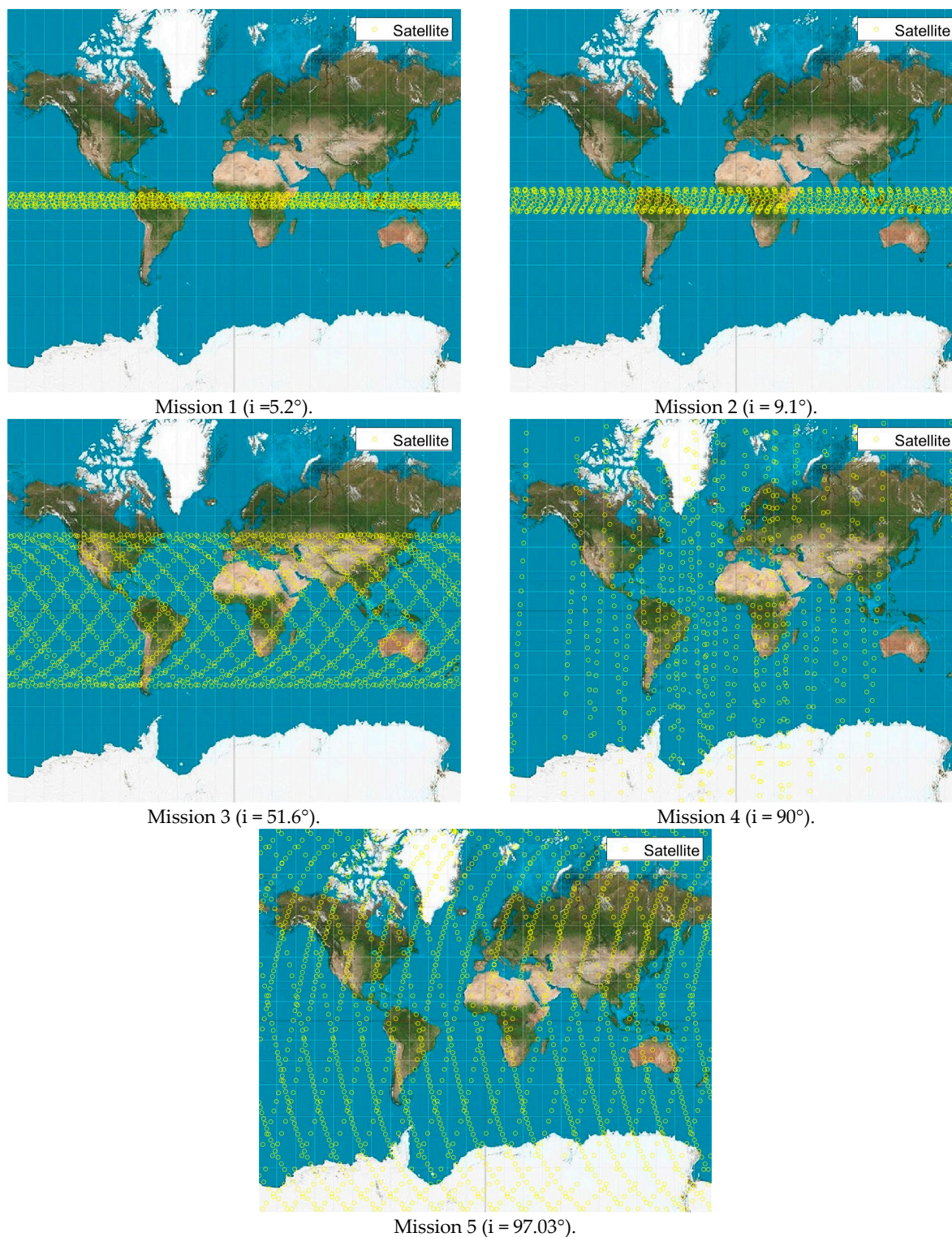


Figure 13. Payload trajectory from insertion up to 24 h afterwards: Mercator projection.

Table 22. Mission 1 partially reusable microlauncher concept: recovery mission analysis.

Key Event	Reference Time [s]	Stage Mass [kg]	Propellant Mass [kg]	Altitude [km]	Velocity [m/s]	Flight Path Angle [°]	Downrange [km]
First-stage separation	89.14	1593.33	131.93	58.59	2927.14	36.08	62.34
Ballistic evolution 1	300.92	1593.33	131.93	239.01	2267.94	0.00	535.76
Stage flip-over	300.92	1593.33	131.93	239.01	2267.94	0.00	535.76
ACS deployment	300.92	1593.33	131.93	239.01	2267.94	0.00	535.76
Ballistic evolution 2	516.12	1593.33	131.93	52.72	2928.30	−36.50	1017.11
Reentry burn	516.12	1593.30	131.90	52.72	2928.23	−36.50	1017.11
Aero guidance	583.19	1593.30	131.90	0.16	221.45	−75.48	1077.91
Landing maneuver	584.69	1467.99	6.60	0.00	0.54	−80.21	1077.95
Stage touchdown/ engine shutdown	584.91	1467.99	6.60	0.00	2.63	−88.01	1077.95

Table 23. Mission 2 partially reusable microlauncher concept: recovery mission analysis.

Key Event	Reference Time [s]	Stage Mass [kg]	Propellant Mass [kg]	Altitude [km]	Velocity [m/s]	Flight Path Angle [°]	Downrange [km]
First-stage separation	93.22	1597.01	134.72	60.92	2909.88	35.99	63.85
Ballistic evolution 1	303.23	1597.01	134.72	238.54	2260.70	0.00	531.69
Stage flip-over	303.23	1597.01	134.72	238.54	2260.70	0.00	531.69
ACS deployment	303.23	1597.01	134.72	238.54	2260.70	0.00	531.69
Ballistic evolution 2	517.50	1597.01	134.72	53.60	2920.11	−36.51	1009.40
Reentry burn	517.51	1596.23	133.94	53.58	2918.61	−36.51	1009.42
Aero guidance	582.76	1596.23	133.94	0.17	225.45	−73.92	1071.74
Landing maneuver	584.33	1469.03	6.74	0.00	0.08	−80.57	1071.79
Stage touchdown/ engine shutdown	584.60	1469.03	6.74	0.00	2.66	−89.71	1071.79

Table 24. Mission 3 partially reusable microlauncher concept: recovery mission analysis.

Key Event	Reference Time [s]	Stage Mass [kg]	Propellant Mass [kg]	Altitude [km]	Velocity [m/s]	Flight Path Angle [°]	Downrange [km]
First-stage separation	81.03	1886.76	165.12	55.28	2825.32	38.81	54.52
Ballistic evolution 1	291.10	1886.76	165.12	239.49	2111.48	0.00	492.34
Stage flip-over	291.10	1886.76	165.12	239.49	2111.48	0.00	492.34
ACS deployment	291.10	1886.76	165.12	239.49	2111.48	0.00	492.34
Ballistic evolution 2	504.12	1886.76	165.12	51.27	2817.59	−39.26	935.88
Reentry burn	504.14	1885.18	163.54	51.25	2815.01	−39.26	935.91
Aero guidance	565.78	1885.18	163.54	0.16	231.89	−73.78	990.30
Landing maneuver	567.20	1732.32	10.68	0.00	0.00	−82.17	990.34
Stage touchdown/ engine shutdown	567.48	1732.32	10.68	0.00	2.80	−89.81	990.34

Table 25. Mission 4 partially reusable microlauncher concept: recovery mission analysis.

Key Event	Reference Time [s]	Stage Mass [kg]	Propellant Mass [kg]	Altitude [km]	Velocity [m/s]	Flight Path Angle [°]	Downrange [km]
First-stage separation	97.32	2203.74	191.33	65.18	2943.14	36.71	66.38
Ballistic evolution 1	304.88	2203.74	191.33	246.86	2288.66	0.00	533.98
Stage flip-over	304.88	2203.74	191.33	246.86	2288.66	0.00	533.98
ACS deployment	304.88	2203.74	191.33	246.86	2288.66	0.00	533.98
Ballistic evolution 2	520.85	2203.74	191.33	51.81	2974.22	−37.86	1020.37
Reentry burn	520.85	2203.33	190.92	51.81	2973.64	−37.86	1020.38
Aero guidance	582.59	2203.33	190.92	0.18	232.06	−73.43	1078.25
Landing maneuver	584.25	2021.98	9.57	0.00	0.14	−80.07	1078.30
Stage touchdown/ engine shutdown	584.49	2021.98	9.57	0.00	2.56	−89.52	1078.30

Table 26. Mission 5 partially reusable microlauncher concept: recovery mission analysis.

Key Event	Reference Time [s]	Stage Mass [kg]	Propellant Mass [kg]	Altitude [km]	Velocity [m/s]	Flight Path Angle [°]	Downrange [km]
First-stage separation	86.75	2317.89	199.70	59.67	3019.03	36.52	63.11
Ballistic evolution 1	298.48	2317.89	199.70	248.57	2351.29	0.00	553.14
Stage flip-over	298.48	2317.89	199.70	248.57	2351.29	0.00	553.14
ACS deployment	298.48	2317.89	199.70	248.57	2351.29	0.00	553.14
Ballistic evolution 2	515.53	2317.89	199.70	51.64	3027.01	−37.26	1055.10
Reentry burn	515.53	2317.55	199.37	51.64	3026.56	−37.26	1055.10
Aero guidance	578.90	2317.55	199.37	0.17	231.51	−74.35	1113.64
Landing maneuver	580.38	2128.17	9.99	0.00	0.21	−79.88	1113.68
Stage touchdown/ engine shutdown	580.58	2128.17	9.99	0.00	2.22	−89.12	1113.68

The recovery mission used in this paper employs a downrange recovery (see Figure 6) from a separate landing location for which the exact coordinates are not imposed prior the optimization process, but rather it is an output of the MDO tool that maximizes the performance of the launch vehicle. To compute the downrange (distance between current stage position and launch location), the following relation is used [36]:

$$downrange_{1-2} = 2r \cdot \arcsin \left(\sqrt{\sin^2 \left(\frac{\Delta \varphi}{2} \right) + \cos \varphi_1 \cdot \cos \varphi_2 \cdot \sin^2 \left(\frac{\Delta \lambda}{2} \right)} \right) \quad (16)$$

where index 1 corresponds to the launchpad location (given in Table 6) and index 2 corresponds to first-stage configuration, r is Earth's radius, φ is the geocentric latitude, and λ is the geocentric longitude (relative).

Based on the data shown in Tables 22–26, it can be seen that the landing location is at a far location, with the downrange between the landing and launch location ranging from 990 km (mission 3) to 1113 km (mission 5). Nevertheless, the mass impact (and costs) associated with the stage recovery process will be greatly lower than that associated with a return to launch site recovery mission impact on the launch vehicle solution [11].

In order to minimize the propellant mass reserved for the recovery mission, it can be seen from both Tables 11 and 22, Tables 23–26 that most of the propellant is used during the landing burn (optimization variable $\Delta M_{p_{landing}}$ is close to 1). The rest is used during the reentry burn. To limit the aerothermal loads that the lower part of the first-stage assembly

is subjected to during the recovery mission, a heat shield is used which covers around 30–35% of the total surface area of the stage (details in [10]).

As no other constraints are imposed during the recovery mission, there is no reason for the necessity of a longer reentry burn (at least not quantifiable in the preliminary design phase, as no further data is available). While a reduced reentry velocity generally implies a lower total energy state for the vehicle, it is acknowledged that subcircular reentry trajectories can lead to steeper flight path angles and may result in higher peak thermal and dynamic loads as the vehicle encounters denser atmospheric layers more rapidly.

Nonetheless, within the scope of this preliminary design phase and based on the empirical data from SpaceX Starship Flight 11 [37] (which also employs a reentry burn skip phenomenon), the omission of a high-duration reentry burn remains a viable baseline solution for low-velocity configurations, provided the structural margins can accommodate the concentrated heat flux associated with these steeper entry vectors.

All five recovery missions have passed the safety check at stage touchdown, where a maximum velocity of 3 m/s and a maximum deviation from vertical of 2° was considered okay for a “soft touchdown”. An interesting output of the trajectory optimization is that, of the altitude at which the landing burn starts, a quasi-constant altitude is seen across all five missions (between 159 m and 185 m).

For a better grasp of the similarities of all five recovery trajectories, Figure 14 showcases the variation in several parameters of interest variation with time. Figure 14a–d depicts the plots of altitude, relative velocity, flight path angle, and downrange vs. time. Almost identical behavior can be seen across all missions as a direct result of the highly consistent stage separation conditions and the ballistic nature of the recovery trajectory. The apogee of a recovery mission is towards the 250 km altitude mark, while the maximum velocities before aerodynamic braking occurs (because of high air density) are around 3 km/s. Towards the end of the recovery mission, both the slope of altitude and velocity variation with time decrease, as the landing burn occurs.

Next, Figure 14e,f presents the plots of first-stage mass and generated thrust vs. time. In the context of the entire recovery mission, the engine restarts are reflected in the thrust profile as “spikes,” indicating short-duration, high-intensity thrust is used to reduce the vehicle’s descent speed. The propellant mass consumption seen is proportional to the burn duration, with a steep decline in stage mass being observed towards the end of the mission.

5.4. Cost Comparison

To better understand the implications of the imposed mission on the partially reusable microlauncher concept, it is best to look at the cost estimates for the entire operational lifetime. An output database is generated during the cost estimation module of the MDO, a module that is used only after MDO convergence (this way, the computational time needed for the microlauncher concept optimization is significantly reduced, as the MDO main loop has only four main modules and not five).

Details regarding the mathematical models and input data needed to generate the final cost estimates are given in [10,11]. For the operational lifetime of the microlauncher concept, a total of 20 years was envisioned, with a total of 12 launches per year. Thus, the development costs are amortized over 240 total launches (as seen in Table 27). The lower stage is reused a total of five times, and before each reflight mission, a refurbishment process is realized.

The costs are estimated inside the cost estimation module in terms of MYr (Man-Year) and are later assessed into EUR by using average Romanian salaries. The full cost estimates are given in Table 27 for all five microlauncher concepts studied in this paper.

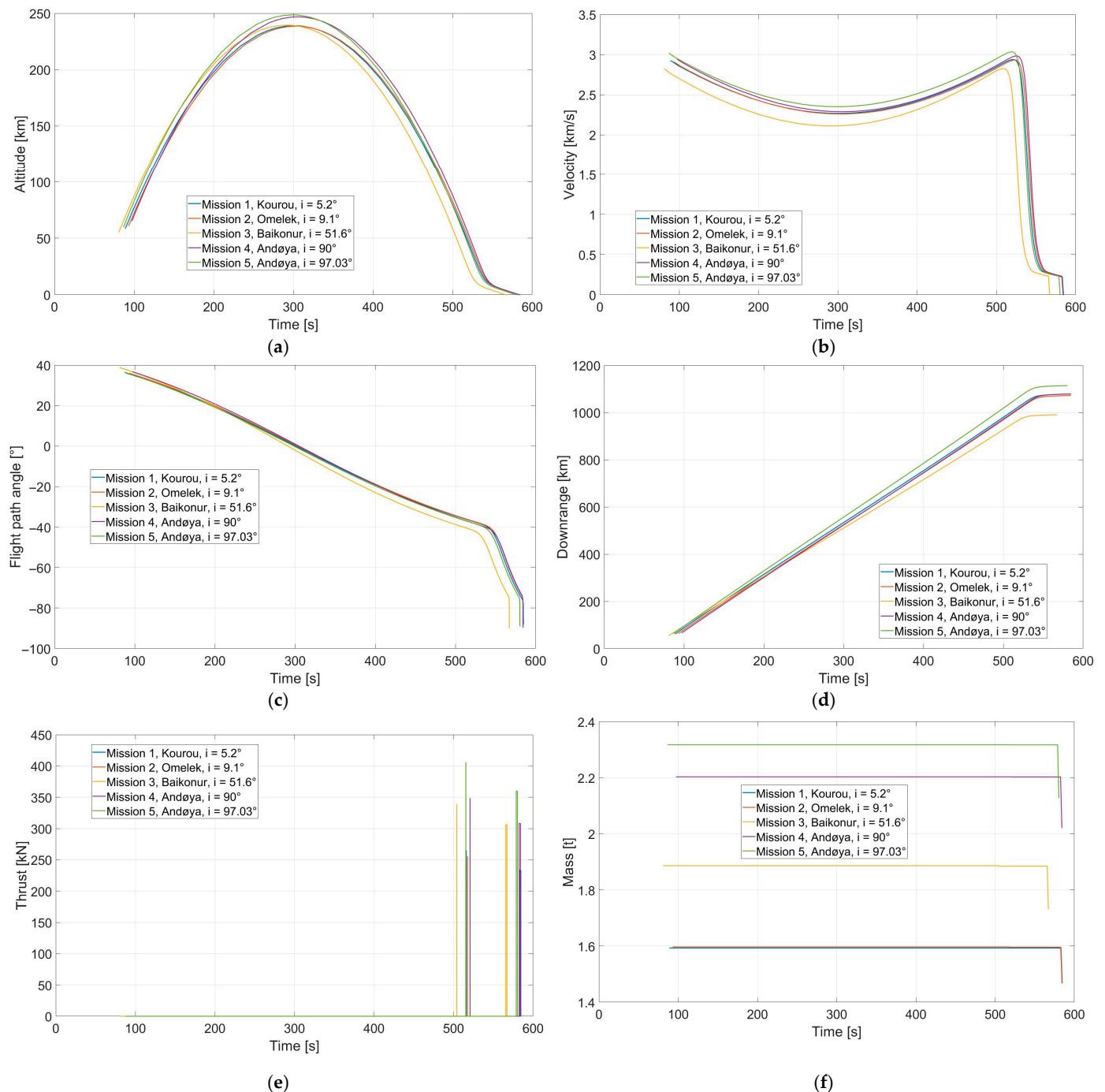


Figure 14. Reference recovery mission trajectories: (a) altitude vs. time; (b) relative velocity vs. time; (c) flight path angle vs. time; (d) downrange vs. time; (e) thrust vs. time; and (f) first-stage mass vs. time.

The most important costs are given by launch vehicle development, which ranges from EUR 545.28 M to EUR 646.39 M. The difference between the development costs associated with the launcher generated for mission 5, compared to the launcher generated for mission 1, is approximately 18%. By using a total number of 240 launches (1 per month over 20 years), the amortization of these development costs becomes acceptable, being approximately EUR 2.27–2.69 M per launch, depending on the mission chosen as reference (and the use of the microlauncher concept dedicated to this mission).

The production costs show a similar behavior to the development costs, ranging from EUR 1.43 M for the concept dedicated to mission 3 to EUR 1.83 M for the concept dedicated to mission 5. In percentage terms, this difference is approximately 27%. Direct and indirect

operational costs are quasi-constant regardless of the reference mission, totaling between EUR 500 and 530 k per launch. The refurbishment process varies between EUR 190 and 240 k.

Table 27. Reusable microlauncher family: cost breakdown.

Cost Breakdown	Value [M EUR]				
Development	547.04	545.28	593.36	630.15	646.39
Estimated lifetime launches					240
R&D amortization per launch	2.28	2.27	2.47	2.63	2.69
Production (average)	1.44	1.43	1.65	1.79	1.83
First stage (5 reuses)	3.60	3.56	4.02	4.36	4.61
Second stage	0.72	0.72	0.85	0.92	0.91
Direct Operations	0.29	0.29	0.31	0.32	0.32
Ground operations	0.05	0.05	0.06	0.07	0.07
Materials and propellants	0.02	0.02	0.03	0.03	0.03
Launch, Flight and Mission Operations	0.11	0.11	0.11	0.11	0.11
Transport and Recovery	0.11	0.11	0.11	0.11	0.11
Indirect Operations	0.21	0.21	0.21	0.21	0.21
First-stage refurbishment	0.19	0.19	0.21	0.23	0.24
Total cost per launch	4.41	4.39	4.86	5.17	5.30
Price per launch (with 30% profit margin)	5.73	5.71	6.31	6.73	6.89

Summing up all of the cost contributions mentioned earlier, the total cost per launch for each concept generated in this paper can be obtained. One can observe the fact that the costs associated with the mission five microlauncher concept are around 20% higher than the ones associated with the mission 1 microlauncher. The minimum costs were observed for the mission 2 launcher, totaling EUR 4.39 M per launch (average value considering a five times first-stage reuse). Since the payload considered for insertion has a value of 100 kg, the numerical values presented in Table 27 translate into a total cost ranging between EUR 44 and 53 k per kilogram inserted into a LEO orbit of 400 km altitude.

Of course, to these costs will be added the considered profit margin; for this project, a profit margin of 30% is implemented, and thus, the final price per launch paid by the customer is 5.73 M€ for the case of an approximate equatorial orbit with a launch from Korou and 6.89 M€ for an SSO orbit with launch from Andøya.

6. Conclusions

This paper proposes a family of partially reusable microlaunchers based on a multidisciplinary optimization investigation. The optimization was realized considering the minimization of lift-off mass, together with the successful accomplishment of both the main mission (insertion of a 100 kg payload into LEO) and the downrange recovery mission.

An analysis was made regarding the influence of five key operational missions of interest (with different launch locations and target orbit types) on the optimal microlauncher concepts. The final configurations and specifications presented throughout the study can be viewed as preliminary building blocks towards the development of a locally based launch vehicle.

The paper concludes with an estimated cost comparison between the five microlauncher concepts proposed. It was discovered that the main economic roadblock is given by the development costs, which are in the order of 600 million euros. By operating such a reusable launch vehicle for 20 years, the amortization becomes much more tolerable, being

approximately EUR 2.5 M per launch. In the end, the final cost per launch was observed to be in the region of EUR 50 k per kg of payload, which seems to be on par with other expendable European launch vehicles.

Funding: The work presented in this study has received funding from the Romanian Nucleu Program, project PN 23 17 07 02.

Data Availability Statement: The original contributions presented in the study are included in the article, and further inquiries can be directed to the corresponding author.

Acknowledgments: The author would like to thank Tudorel-Petronel Afilipoae, Oana Popescu, Georgiana Ichim, Alexandra Popescu, and Ionuț Bunescu for their continuous aid, which was instrumental in shaping the functionality of the MDO tool.

Conflicts of Interest: The author declares no conflicts of interest.

References

1. De Concini, A.; Toth, J. *The Future of the European Space Sector How to Leverage Europe's Technological Leadership and Boost Investments for Space Ventures*; report for the European Commission by the European Investment Bank; European Investment Bank: Luxembourg, 2019. [CrossRef]
2. SpaceX. Falcon 9—First Orbital Class Rocket Capable of Reflight. Available online: <https://www.spacex.com/vehicles/falcon-9/> (accessed on 12 January 2026).
3. Gallego, P. Miura 5 The European and Reusable Microlauncher for CubeSats and Small Satellites. In Proceedings of the Small Satellites Conference, Logan, UT, USA, 1–6 August 2020.
4. Vila, J.; Hassin, J. Technology Acceleration Process for the THEMIS Low Cost and Reusable Prototype. In Proceedings of the 8th European Conference for Aeronautics and Space Sciences, Madrid, Spain, 1–4 July 2019. [CrossRef]
5. Frenoy, O.; Hiraiwa, T. Concept of Operations—CALLISTO Demonstrator. In Proceedings of the 8th European Conference for Aeronautics and Space Sciences, Madrid, Spain, 1–4 July 2019. [CrossRef]
6. Gao, S.; Clark, K.; Unwin, M.; Zackrisson, J.; Shiroma, W.A.; Akagi, J.M.; Maynard, K.; Garner, P.; Boccia, L.; Amendola, G.; et al. Antennas for Modern Small Satellites. *IEEE Antennas Propag. Mag.* **2009**, *51*, 40–56. [CrossRef]
7. Dietlein, I.; Bussler, L.; Stappert, S.; Wilken, J.; Sippel, M. Overview of system study on recovery methods for reusable first stages of future European launchers. *CEAS Space J.* **2025**, *17*, 71–88. [CrossRef]
8. Bussler, L.; Dietlein, I.; Sippel, M. Comparative analyses of European horizontal-landing reusable first stage concepts. *CEAS Space J.* **2025**, *17*, 131–154. [CrossRef]
9. Wilken, J.; Herberhold, M.; Sippel, M. Options for future European reusable booster stages: Evaluation and comparison of VTHL and VTVL costs. *CEAS Space J.* **2025**, *17*, 177–198. [CrossRef]
10. Onel, A.-I.; Afilipoae, T.-P.; Popescu, O.-I.; Ichim, G.; Popescu, A. Multidisciplinary Design Optimization of a Two-Stage LOX/Methane Partially Reusable Microlauncher. *Aerospace* **2025**, *12*, 1030. [CrossRef]
11. Onel, A.-I.; Afilipoae, T.-P.; Popescu, O.-I.; Ichim, G.; Popescu, A.; Bunescu, I. Feasibility Analysis of a Return to Launch Site Partially Reusable Microlauncher via Multidisciplinary Optimization. *Aerospace* **2026**, *13*, 66. [CrossRef]
12. Onel, A.-I.; Afilipoae, T.-P.; Popescu, O.-I.; Popescu, A.; Pricop, M.-V.; Bunescu, I.; Hothazie, M.-V. Multidisciplinary design optimization of a two-stage LOX/methane microlauncher. *AIP Conf. Proc.* **2025**, *3315*, 190006. [CrossRef]
13. Kumar Shah, V.; Ateeb, K.; Razzaq, M.; Varun, C.; Avinash, B. Determination of the optimum nose cone geometrical shape for supersonic missile. *Mater. Today Proc.* **2022**, *64*, 749–754. [CrossRef]
14. Balesdent, M. Multidisciplinary Design Optimization of Launch Vehicles. Ph.D. Thesis, Ecole Centrale de Nantes, Nantes, France, 2011.
15. Dresia, K.; Jentsch, S.; Waxenegger-Wilfing, G.; Dos Santos Hahn, R.; Deeken, J.; Oschwald, M.; Mota, F. Multidisciplinary Design Optimization of Reusable Launch Vehicles for Different Propellants and Objectives. *J. Spacecr. Rocket.* **2021**, *58*, 1017–1029. [CrossRef]
16. Brevault, L.; Balesdent, M.; Hebbal, A. Multi-objective Multidisciplinary Design Optimization approach for partially Reusable Launch Vehicle design. *J. Spacecr. Rocket.* **2020**, *57*, 373–390. [CrossRef]
17. Brevault, L.; Balesdent, M.; Hebbal, A.; Patureau de Mirand, A. Surrogate Model-Based Multi-Objective MDO Approach for Partially Reusable Launch Vehicle Design. In Proceedings of the AIAA Science and Technology Forum and Exposition, San Diego, CA, USA, 7–11 January 2019. [CrossRef]
18. MathWorks. MATLAB Documentation. Available online: <https://www.mathworks.com/help/matlab/index.html> (accessed on 12 January 2026).

19. Holland, J.H. *Adaptation in Natural and Artificial Systems. An Introductory Analysis with Applications to Biology, Control, and Artificial Intelligence*; Michigan Press: Cambridge, MA, USA, 1992. [CrossRef]
20. Bäck, T. *Evolutionary Algorithms in Theory and Practice. Evolution Strategies, Evolutionary Programming, Genetic Algorithms*; Oxford University Press: Oxford, NY, USA, 1996. [CrossRef]
21. Matsumoto, M.; Nishimura, T. Mersenne Twister: A 623-Dimensionally Equidistributed Uniform Pseudo-Random Number Generator. *ACM Trans. Model. Comput. Simul.* **1998**, *8*, 3–30. [CrossRef]
22. Castellini, F. Multidisciplinary Design Optimization for Expendable Launch Vehicles. Ph.D. Thesis, Politecnico Di Milano, Milano, Italy, 2012.
23. Organisation intergouvernementale de la Convention du Metre. *The International System of Units*, 8th ed.; Stedi Media: Paris, France, 2006.
24. Humble, R.; Henry, G.; Larson, W. *Space Propulsion Analysis and Design*; Mc-Graw Hill: New York, NY, USA, 1995.
25. Sutton, G.P.; Biblarz, O. *Rocket Propulsion Elements*, 7th ed.; Wiley-Interscience: New York, NY, USA, 2000.
26. Onel, A.I.; Popescu, O.I.; Neculăescu, A.M.; Afilipoae, T.P.; Chelaru, T.V. Liquid Rocket Engine Performance Assessment in the context of Small Launcher Optimisation. *INCAS Bull.* **2019**, *11*, 135–145. [CrossRef]
27. Onel, A.I.; Chelaru, T.V. Trajectory Assessment and Optimisation in the Context of Small Launcher Design. *INCAS Bull.* **2020**, *12*, 117–132. [CrossRef]
28. Dietrich, E.K. *Handbook of Cost Engineering and Design of Space Transportation Systems*; TransCostSystems: Ottobrunn, Germany, 2013.
29. Centre Spatial Guyanais (CSG). Europe's Spaceport. Available online: <https://centrespatialguyanais.cnes.fr/en> (accessed on 12 January 2026).
30. SpaceX. Installations Omelek Island. Available online: http://capcomespace.net/dossiers/espace_privé/SpaceX/installations_omelek_island.htm (accessed on 12 January 2026).
31. NASA. International Space Station. Available online: <https://www.nasa.gov/reference/international-space-station/> (accessed on 12 January 2026).
32. NTI. Baikonur Cosmodrome. Available online: <https://www.nti.org/education-center/facilities/baikonur-cosmodrome/> (accessed on 12 January 2026).
33. Andoya Space. Orbital Launch Services. Available online: <https://andoyaspace.no/spaceport/> (accessed on 12 January 2026).
34. Small Satellites. Sun-Synchronous Circular Orbit, Inclination vs Altitude. Available online: <https://parkinresearch.com/engineering-inference-engine/3d-orbit-calculator/> (accessed on 12 January 2026).
35. OrbiterWiki. Launch Azimuth. Available online: https://www.orbiterwiki.org/wiki/Launch_Azimuth (accessed on 12 January 2026).
36. Selvaraj, S.; Sabarish, B.A. Analysis of Distance Measures in Spatial Trajectory Data Clustering. In Proceedings of the Annual International Conference on Emerging Research Areas on “Computing & Communication Systems for a Fourth Industrial Revolution”, Kanjirapally, India, 14–16 December 2020. [CrossRef]
37. SpaceX. Starship's Eleventh Flight Test. Available online: <https://www.spacex.com/launches/starship-flight-11> (accessed on 12 January 2026).

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