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Design of fly-back reusable launch vehicle architecture using air-breathing propulsion

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Abstract

Partially reusable launch vehicle design has raised a great interest in order to minimize the costs of space transportation by recovering and refurbishing the first stage. Several configurations such as toss-back architectures are now operational. These concepts require additional rocket propellant in order to carry out the boost-back and landing maneuvers, that induce several losses in terms of performance for the ascent mission. In order to limit these losses, this paper focuses on the design of a winged fly-back configuration that performs the return-to-launch-site mission using several air-breathing engines located in the nose of the first stage. In this paper, the design of a reusability kit, allowing to provide the first stage with both expendable and reusable capabilities, is investigated. This kit is composed of the lifting surfaces, the nose including the air-breathing propulsive system, landing gears and additional avionics. This paper addresses the design of such a fly-back reusability kit and presents the mission specifications, the design process relying on Multidisciplinary Design Optimization techniques, the vehicle performance and the optimal trajectories. Different analyses (aero-propulsive, optimal control, operational considerations, *etc.*) are detailed.

1 Introduction

Reusable launch vehicle design has raised a great interest in the last decade in order to minimize the costs of space transportation by recovering and refurbishing the first stage. Toss-back configurations are operational and their efficiency is now well established. This type of concepts consists for the first stage in performing a Return To Launch Site (RTLS) mission by using the main rocket engine propulsion and landing vertically. To do so, this architecture has to carry out additional rocket propellant in order to perform the boost-back and landing burns. As a consequence, this induces several losses in terms of performance for the ascent mission and reduces the capability of injection. In order to limit these losses, different alternative configurations can be investigated. For example, it may be interesting to reduce the additional rocket propellant that has to be used for the RTLS mission. This can be partially performed using aeronautical technologies (wings, air-breathing propulsion, *etc.*) [1, 5, 18, 20]. This paper focuses on the design of a winged fly-back configuration that allows to totally remove the need of additional rocket propellant for the RTLS mission. The studied concept is operated similarly to classical expendable launch vehicles for the ascent mission and it performs the RTLS phase using air-breathing engines located in the nose of the first stage. After its separation with the second stage, the first stage carries out a ballistic phase followed by a high angle of attack reentry. Then, the turbojets are ignited to carry out a cruise flight and to land the vehicle horizontally. This study is a collaboration between the French Space Agency (Centre National d'Etudes Spatiales - CNES) and the French Aerospace Lab (Office National d'Etudes et de Recherches Aérospatiales - ONERA) on Reusable Launch Vehicles (RLV).

In this paper, the design of a reusability kit, allowing to provide the first stage with both expendable and reusable capabilities, is investigated. This kit, composed of the lifting surfaces, the nose including the air-breathing propulsive system and additional subsystems (*e.g.*, landing gears), can be mounted on the main core of the launch vehicle for performing several reusable missions, and then removed and installed on another first stage if the current one is used for a last expendable mission. In that way, the reusability kit may be used a large number of times allowing to lower the costs. The design is performed using multidisciplinary design optimization techniques allowing to assess the overall launch vehicle performance and to meet optimal compromises between the different disciplines (propulsion, aerodynamics, structure, trajectory, *etc.*) as well as the different flight phases (ascent, re-entry and return to launch site).

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This paper is organized as follows. In Section 2, the main specifications of the mission are detailed. In Section 3, the design process used to assess the performance of the vehicle is described. Then, in Section 4, the trade-off study between different fly-back architectures is presented. In this trade-off are balanced different technological choices such as the choice of the turbojets (type and number), the type of airfoil, the planform of the wings, *etc.* In Section 5, the best configuration selected in the trade-off study is refined in order to consolidate its performance assessment and investigate in depth several bottlenecks. Eventually, a conclusion and a synthesis are drawn about the defined concepts.

2 Mission specifications and reusability kit

This study deals with the design of a Two-Stage-To-Orbit launch vehicle with a reusable first stage. For that purpose, an expendable configuration is modified thanks to a reusability kit in order to provide it with reusability capability. This reusability kit is composed of lifting surfaces such as main wings, canards and a vertical plane, front and rear landing gears, and several turbojets located in the nose (Figure 1). The wings and rear landing gears are located in a case that is attached to the thrust frame of the main core. The turbojets, canards, aerodynamic nose and front gears are located in a reusability pack (composed of the nose and an additional skirt) that is positioned in the front of the stage. This reusability kit can be removed for expendable missions. The considered vehicle is a Vertical Take Off - Horizontal Landing architecture.

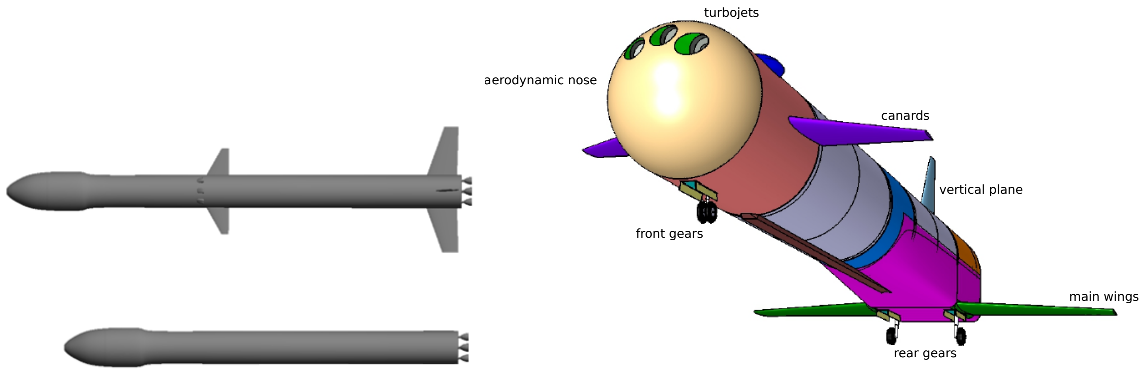

Figure 1: Left: illustration of the entire vehicle (reusable and expendable configurations). Right: details of the reusability kit

The ascent phase is performed classically like an expendable launch vehicle. After its separation, the first stage performs a ballistic phase followed by an atmospheric reentry. After the reentry, the vehicle uses its lifting surfaces to carry out a U-turn and a pull-up maneuvers in order to begin the RTLS mission. When the aerodynamic conditions are possible, the turbojets located in the nose are ignited and the vehicle begins its cruise flight phase until reaching the landing site (Figure 2).

For the main core, the Ariane-Next configuration [7] has been chosen as a baseline. The first stage is powered by 7 PROMETHEUS engines using LOx/LCH4 propellant. The second stage is also powered by one PROMETHEUS engine. The reference mission consists in injecting a 6 tons payload into a 800km circular SSO orbit. The launch site is the European Space Port in French Guiana. Additional constraints are considered in order to take operational conditions into account. Different thresholds concerning the loads (axial and transverse load factors, maximum dynamic pressure, maximal aerothermal flux) during both the ascent and return phases are considered. Moreover, visibility and safety specifications near the launch and landing sites are taken into account.

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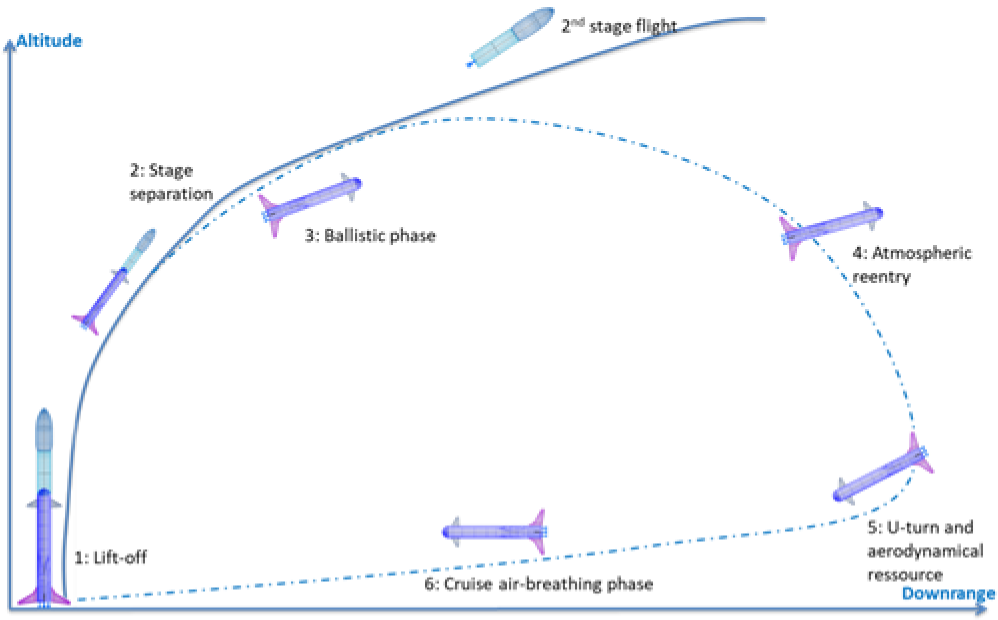

Figure 2: Flight phases of a fly-back architecture

3 Description of design process

3.1 Synthesis of technological choices

In this study, different aeronautical technological choices have been considered in order to find the global optimal architecture of the reusability kit. These choices concern mainly the turbojets and the lifting surfaces. Indeed, the overall optimal design results from compromises between the performance in the ascent phase versus the performance for the RTLS mission. For example, a very efficient airfoil profile for the cruise return phase at subsonic condition may induce additional drag for the ascent phase that may lead to poor overall performance. In this study, the technological choices that have been taken into account are :

- the airfoil profile: double-wedge and NACA (Figure 3) ;
- the planform of the main wings: 2 planforms have been considered (Figure 3) ;
- the type of low-bypass ratio turbojets: Snecma M88 [12] or Eurojet EJ200 [16] (Table 1);
- the number of turbojets: 2, 3 and 4.

Engines with low by-pass ratio have been chosen (M88 and EJ200) for the sake of integration into the reusability kit.

Table 1: Characteristics of engines (manufacturer data)

	M88	EJ200
Thrust at ground (kN)	60	60
Mass (kg)	990	897
Length (m)	4	3.54
Diameter (m)	0.74	0.69
Specific fuel consumption (cruise) (10^5 (kg/s)/N)	2.1	2.2

3.2 Disciplinary models

In this section, all the disciplines that have been included in the multidisciplinary design process are described.

3.2.1 Geometry and structural design

The geometry and structural design discipline is responsible for defining the overall geometry of all the stages, generating the meshes that will be used for aerodynamics assessment, and computing the mass budget of the

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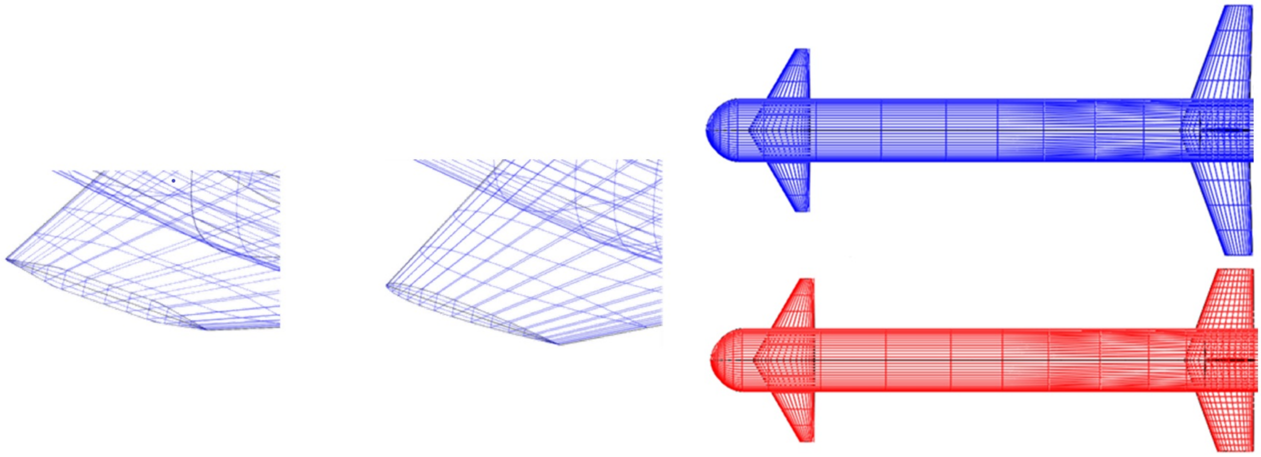

Figure 3: Airfoil profiles and main wing planforms

launch vehicle. This discipline uses a parametric geometric modeler to define all the vehicle from several decision variables such as the length of the tank, the wing planform, the diameter of the stage, *etc.* This geometry is used to build a preliminary CAD design that is integrated in the multidisciplinary process.
From the geometry data and the loads given by the trajectory, a structural design of the first stage is performed using a dedicated module developed at ONERA. The primary structures (wings, tanks, skirts, thrust frame, *etc.*) are designed using both analytical formula and Finite Element Analysis (Figure 4) to consider key elements such as the buckling, bending moment, wing deflection, *etc.* (Figure 5).

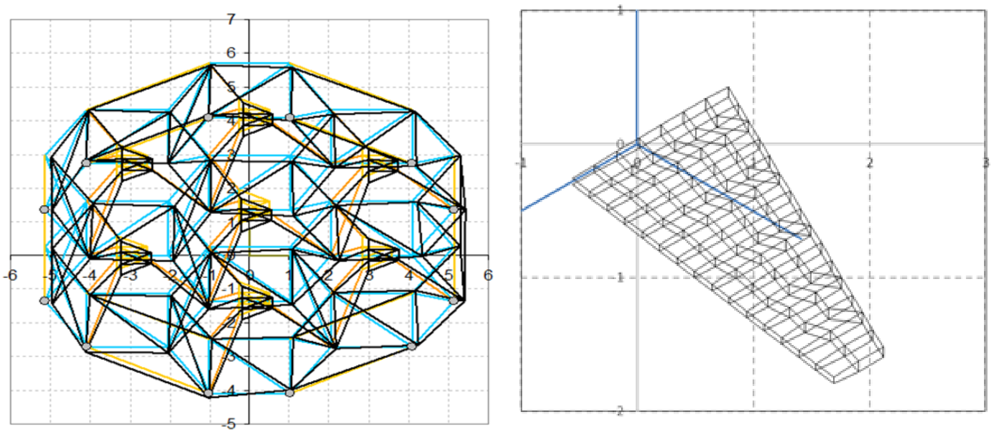

Figure 4: Thrust frame and wing design

All the loads for the different phases (ascent and return) are considered (coming from the aerodynamics discipline for instance). This allows to have a consistent structural design of the launch vehicle. The secondary structures (landing gears, equipments, turbojets, *etc.*) are either given by CNES for the ArianeNext configuration, manufacturers for the turbojets, or estimated using Mass Estimation Relationships (MER) formula. The second stage is a classical expendable one. Consequently, standard MERs have been used to determine its mass.

This module is included in the multidisciplinary design optimization process using OpenMDAO [11] framework as a parametric function allowing to reestimate the mass budget for different values of the design variables controlled by the optimization process.

3.2.2 Aerodynamics

In order to be able to perform global performance assessment, the aerodynamic coefficients of the vehicle have to be accurately computed for different architectures (total vehicle, first stage only, second stage only) and different flight phases (ascent phase: subsonic, supersonic, hypersonic, reentry phase: high angle of attack flight, return phase: subsonic, supersonic, powered phases or not). This makes the design of the aerodynamic module very challenging and dedicated tools of different fidelities have been combined in the aerodynamic component.

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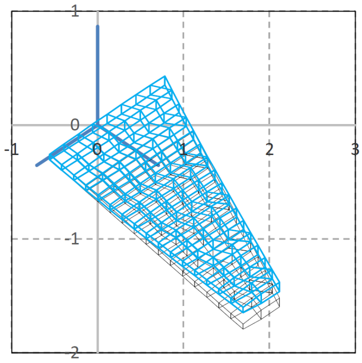

Figure 5: Wing deflection calculation

Furthermore, this module has to be evaluated a large number of times in the optimization process for different aerodynamic configurations. In that way, the computational cost is another key driver for aerodynamic assessment.

To assess the aerodynamics efficiency of the studied configuration with a mastered computational time, multi-fidelity techniques have been used [4] with Gaussian process based surrogate modeling (Figure 6). These multifidelity techniques allow to aggregate the responses of the different aerodynamic codes depending on the flight phase and the Mach number regime while accounting for their accuracy and uncertainty level.

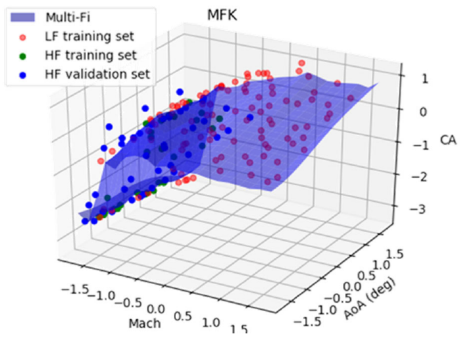

Figure 6: Multi-fidelity techniques

Four levels of fidelity have been used in the aerodynamics discipline. The first one is a semi-empirical code named MISSILE [9] that allows to compute the aerodynamics characteristics for Mach 0 to Mach 10. This code involves analytical formula calibrated with wind tunnels results. The second code is an in-house code named SHAMAN which is based on Local Surface Inclination methods and allows to determine the aerodynamics coefficients for supersonic and hypersonic regimes at a reduced computational cost (Figure 7).

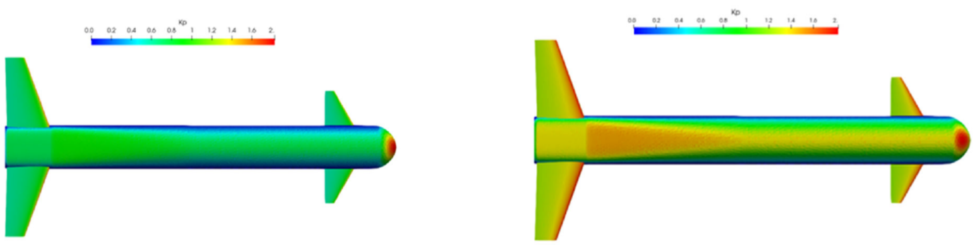

Figure 7: Results from SHAMAN estimation

To improve the accuracy of the aerodynamic coefficients estimation, Euler CFD calculations have been also used thanks to an aerodynamic module based on the ONERA CANOE [8] module relying on SU2 [15]. Eventually, a RANS CFD calculations campaign has been performed (Figure 8) to improve the accuracy of the coefficients estimates for specific flight domains such as the reentry phase. These models have been validated on classical launch vehicle configurations.

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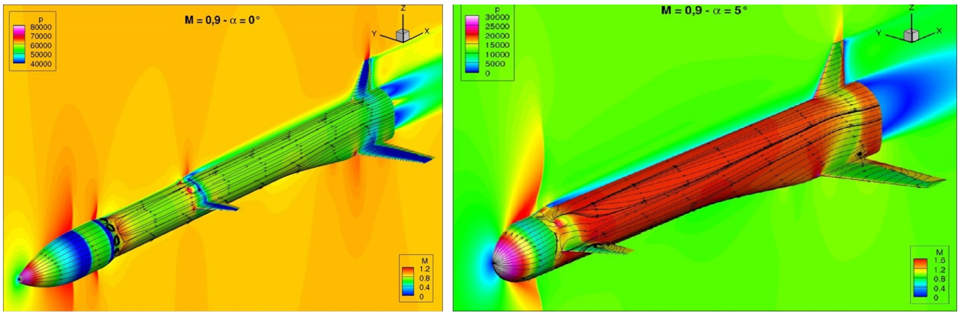

Figure 8: Examples of CFD RANS calculations (left: global vehicle in ascent phase, right: first stage in RTLS phase)

3.2.3 Propulsion

Concerning the rocket propulsion, the two stages are powered by PROMETHEUS engines using LOx/LCH4 propellants [14]. The first stage involves 7 engines and the second stage uses one engine. The data of the engine have been provided by CNES for the study. These engines are throttable and re-ignitable. The thrust varies from 300kN to 1000kN. The throttling level is optimized during the trajectory in order to optimize the performance and meet the design specifications (*e.g.*, maximal dynamic pressure).

Concerning the air-breathing propulsion, semi-empirical formula taken from [17] have been implemented in order to determine for the different considered engines (M88 and EJ200) the thrust and specific fuel consumption as functions of the Mach number and the altitude (Figure 9) from the main characteristics of the engine (nominal thrust, pressures ratio, by-pass ratio, *etc.*). This model has been calibrated on the manufacturer data.

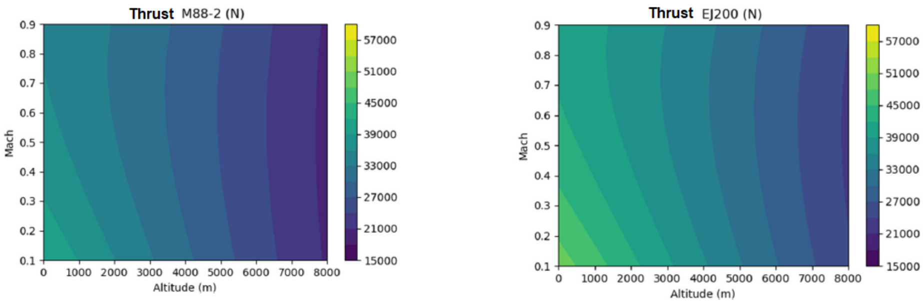

Figure 9: Example of thrust profiles as functions of the Mach number and the altitude for the M88 and EJ200 engines

3.2.4 Trajectory

The trajectory discipline is responsible for the assessment of the overall performance of the launch vehicle. To do so, all the flight phases are simulated. The ascent phase is decomposed into a vertical flight, following by a pitch over maneuver and a gravity turn phase during the atmospheric flight. When the exo-atmospheric conditions are met, a controlled phase is involved to reach the orbit conditions. All the parameters of the different phases are optimized during the design process. The second stage performs a coast-phase and a circularization burn is finally carried out at the apogee of the transfer orbit in order to inject the payload on the SSO orbit. During the ascent phase, the pitch and yaw angles profiles are optimized as well as the throttling of the vehicle.
The RTLS phase is decomposed into 6 phases (Figure 10), namely the ballistic flight, the reentry, the pull up and U-turn maneuvers, the cruise flight and the landing. For all the different phases, specific parametric control laws have been implemented and are optimized along with the design variables during the MDAO process with a direct single-shooting strategy [3]. To model the trajectory, an ordinary differential equation system composed of the standard three degrees of freedom equations of motion is integrated using the 5^{th} order Runge-Kutta method [10].

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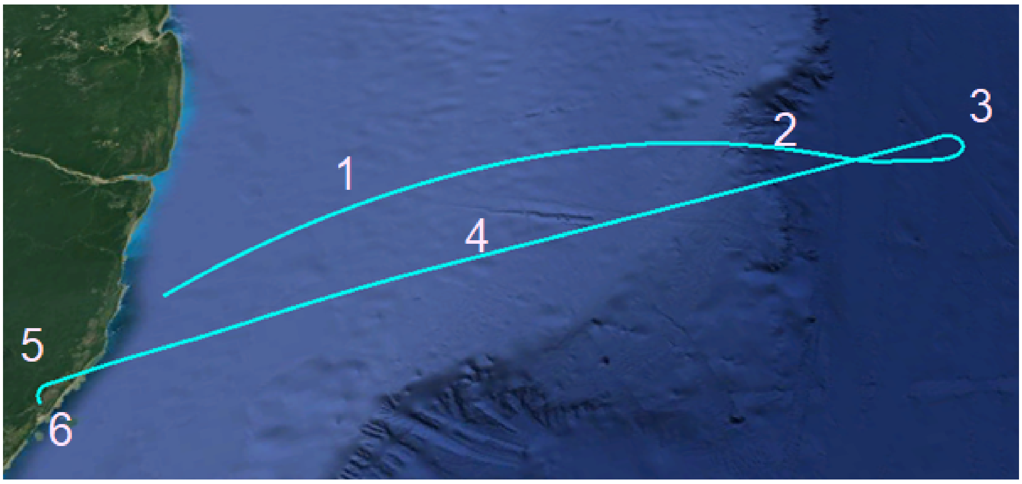

Figure 10: Simulation of RTLS trajectory with: 1 - ballistic flight, 2 - high angle of attack reentry ; 3 - pull up and U-turn maneuvers; 4 : cruise flight ; 5 - final U-turn maneuver ; 6 - landing

3.3 Multidisciplinary design process

The disciplines described above have been integrated into an MDAO process (Figure 11) using the open source OpenMDAO framework developed at NASA [11]. This allows to assess the overall performance of the vehicle and let the optimizer make the optimal compromise between the performance of the RTLS phase and induced penalties for the ascent phase. The objective function that has to be minimized is the Gross Lift Off Weight. The decision variables are the propellant masses of the two stages, the characteristics of the reusability kit (*e.g.* wings and canards planforms, amount of kerosene), the different control laws associated to the optimal control problem of the trajectory discipline. A MultiDiscipline Feasible formulation has been used with a Gauss-Seidel Fixed Point Iteration between the different disciplines [6]. For the optimization, because the optimization problem is non-linear with a large number of local optima, the Covariance Matrix Adaptation - Evolution Strategy is used [13].

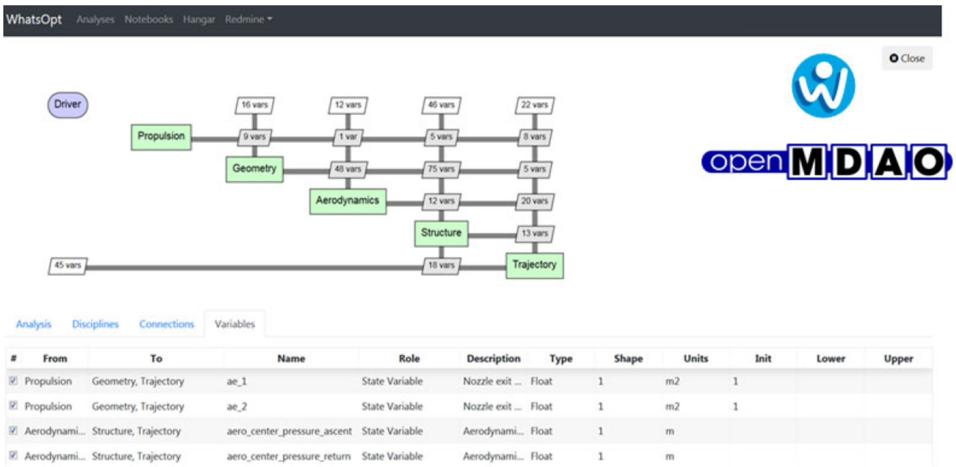

Figure 11: N2 chart of the design process

4 Preliminary trade-offs

4.1 Impact of the number of turbojet engines

4.1.1 Nominal mission

In order to assess the trade-off concerning the number of turbojets that have to be included in the reusability kit, three configurations using EJ200 turbojets are studied: architectures with 2, 3 and 4 turbojets. The number of turbojets has an influence on the quality of the cruise flight. With more thrust, the RTLS cruise flight has more

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flexibility to fly at a higher altitude limiting additional drag and consequently limiting the required kerosene, at the price of complexity of integration and more dry mass (additional engines) that is a direct loss for the ascent phase. The different simulated trajectories are described in Figure 12.

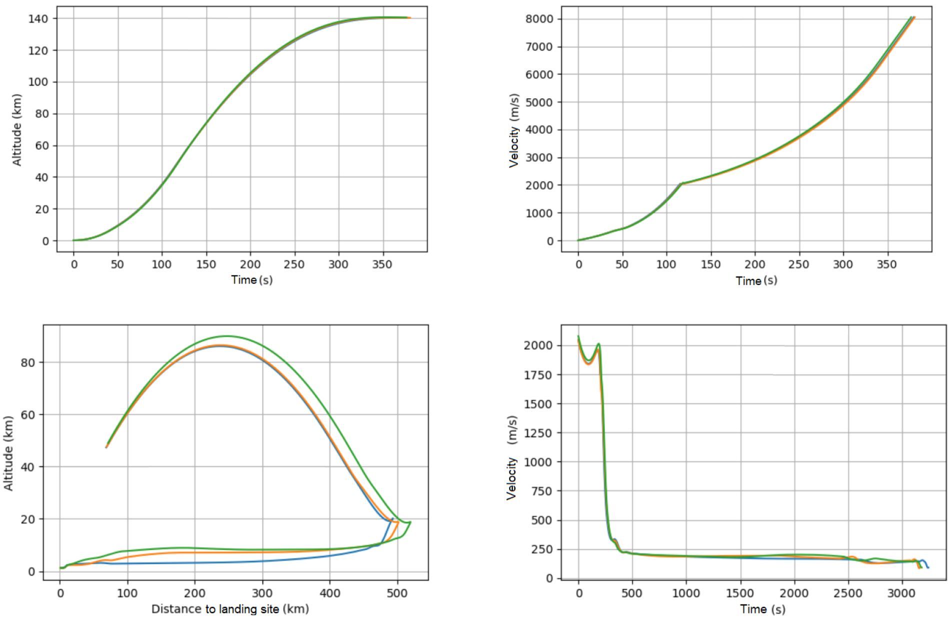

Figure 12: Ascent phase profile (top) and RTLS mission (bottom) for architectures with 2 engines (blue), 3 engines (orange) and 4 engines (green)

While the ascent phase is similar for the different architectures, the RTLS phase is quite different, both in terms altitude of the cruise flight and the maximal distance to landing site. Indeed, the configuration with 4 engines (green) has the maximal distance to return to the landing site. Moreover, as expected, the more the thrust, the more the cruise altitude (9km for 4 engines, 7km for 3 engines and 3.2km for 2 engines). Considering the Gross Lift Off Weight for this configuration, both the configurations with 2 and 3 engines provide quite similar results (few tons of difference) whereas the architecture with 4 engines is 8 tons heavier (comprising 500kg of additional kerosene with respect to the other configurations), with a baseline about 405 tons.

4.1.2 Abort scenario

As the configurations with 2 and 3 engines are very close in terms of performance (GLOW), an additional study has been conducted in order to identify whether the configuration with 3 engines could provide more robustness with respect to an engine failure. For that purpose, a simulation of a specific scenario has been carried out considering that only 2 of the 3 engines are ignited at the beginning of the cruise flight. To do so, the control law for this flight phase (profile of angle of attack and bank angle, throttling of turbojets) has been reoptimized considering the loss of one engine. The resulting RTLS trajectory is described in Figure 13. As it can be seen on this figure, the trajectory with one turbojet out of order remains feasible. Consequently, the architecture with 3 turbojets seems to present the best trade-off between the reliability of the configuration (redundancy of one turbojet) and performance (GLOW).

4.2 Impact of engine type

In order to assess the trade-off considering the type of engine, both configurations with 3 M88 and 3 EJ200 have been designed. For this technological choice, the trade-off involves the compactness of the propulsive system, for the sake of integration into the reusability kit, and the overall performance of the reusable launch vehicle. Both architectures with 3 M88 or 3 EJ200 have been designed using the MDO process. The resulting trajectories are described in Figure 14.

The results of the two optimizations show that the two configurations are quite close in terms of GLOW (the configuration with 3 M88 is 1.5 tons lighter than the other). This is due to the fact that the specific fuel consumption of the M88 is a little bit less than the one of the EJ200. This allows to save around 100 kg of kerosene for the return mission, that leads to 1.5 tons of savings considering the impact on the ascent flight. Furthermore, considering the compactness of the two types of engines, the EJ200 presents advantages both in terms of length and diameter, and unitary mass (Table 1). Consequently, from the two characteristics

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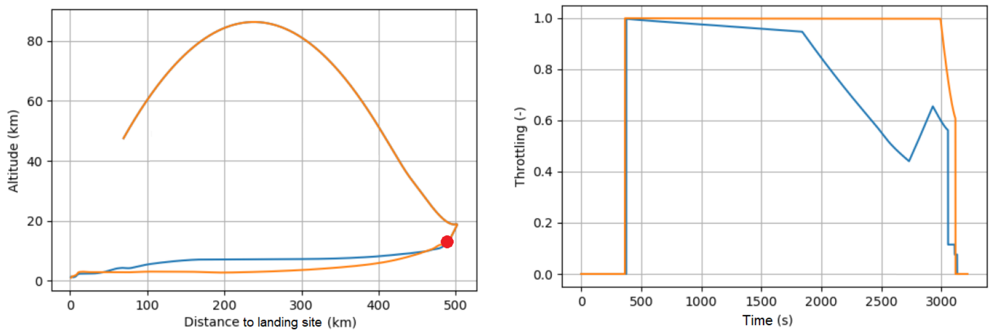

Figure 13: Abort scenario, at left: altitude profile as a function of the distance to the landing site, at right: throttling of the turbojets as a function of the time. In blue: nominal trajectory, in orange: abort trajectory, in red: failure point (loss of one turbojet)

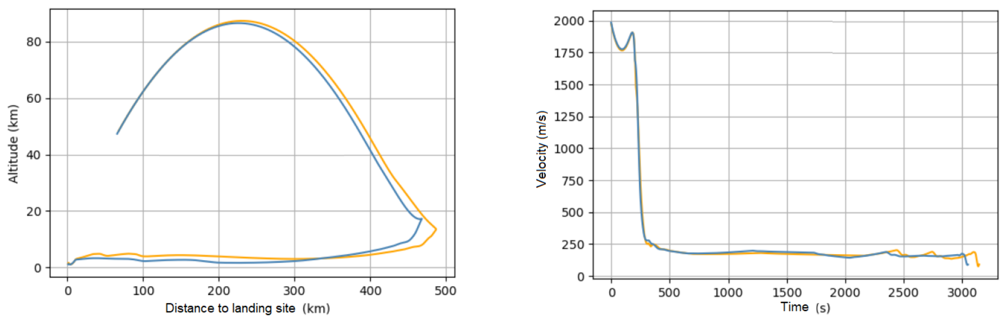

Figure 14: Return trajectories for different turbojet types, at left: altitude profile as a function of the distance to landing site, at right: velocity as a function of time. In blue: RLV architecture with 3 M88, in orange: RLV architecture with 3 EJ200

(performance and integration of propulsive system), it has been chosen to select the EJ200 engine for the following of the study.

4.3 Impact of wing planform

Two wing planforms have been studied in this work. The first is a reduced wingspan (with an half wingspan equal to the diameter of the main core) and an extended wingspan (with an half wingspan equal to 1.5 $\times$ the main core diameter), see Figure 3. Indeed, a larger wingspan can present advantages for the RTLS cruise flight by delivering more lift at the price of additional drag for the ascent phase. Consequently, a global trade-off needs to be assessed. To evaluate this trade-off, configurations with 3 EJ200 and the different wingspans have been designed. As it can be seen on Figure 15, the wingspan has a great effect on the return trajectory but also on the ascent velocity profile. Indeed, the different wingspans cause different drag effects that can be noticed in the ascent velocity profiles. On the performance point of view, the configuration with the longer wingspan presents better characteristics in terms of GLOW (7 tons of difference). Indeed, the smaller wingspan induces a penalization of about 1 points in terms of lift-over-drag ratio, that causes 1.6 tons of additional kerosene to be put on the reusability kit to perform the RTLS mission. Consequently, this mass induces additional rocket propellant for the ascent phase that overall generates a difference of 7 tons on the global launch vehicle. Consequently, it appears that the largest wingspan configuration is more suitable for the RTLS mission. This configuration has been selected for the rest of the study.

4.4 Synthesis of the trade-off

The different trade-off studies described in this section involved the choice of the engine, the choice of the engines number and the wing configuration. To assess the trade-offs, multiple multidisciplinary designs have been performed in order to take the compromises between the ascent and return phases into account. Furthermore, several other aspects such as the redundancy or the integration have also been studied. From all these investigations, the configuration with 3 EJ200 turbojets and the larger wingspan configuration has been selected as providing the best compromises between the overall performance (GLOW), the reliability of the RTLS mission

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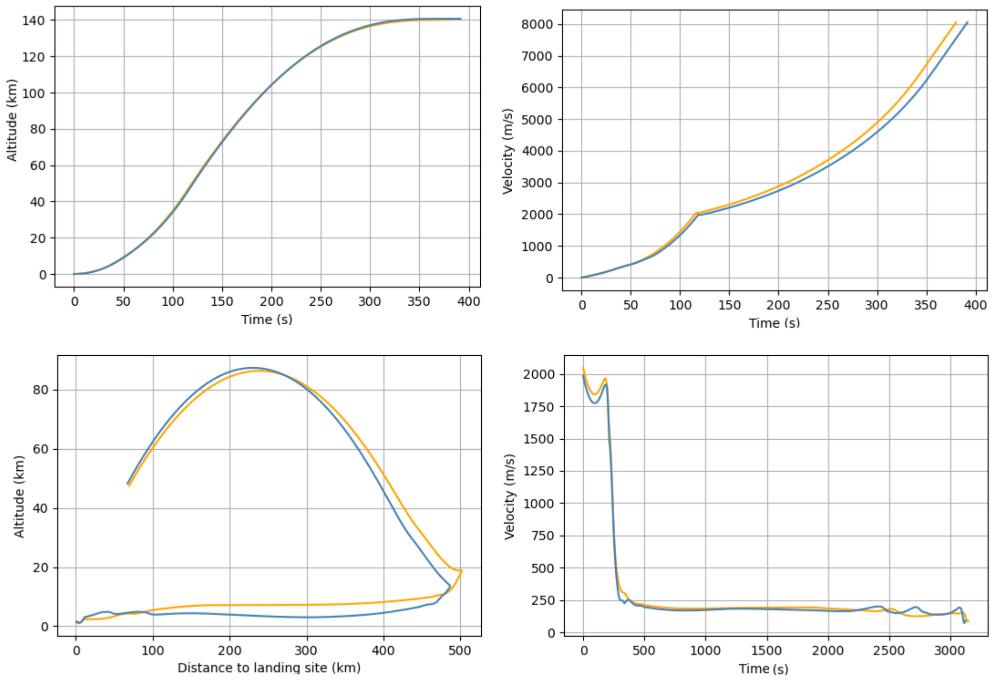

Figure 15: At the top: ascent trajectory profiles. At the bottom: RTLS mission, with in blue: short wingspan, and in orange: large wingspan

(trajectory with the loss of one engine is feasible) and the integration (EJ200 is more compact than M88). In order to analyze in-depth this configuration, several model refinements have been made to address specific keypoints of the design. These refinements are described in the following section.

5 Refinement of the selected configuration

Different model improvements have been carried out in order to refine the performance assessment of the flyback configuration. These refinements are detailed in the next sections.

5.1 Aerodynamics

Concerning the aerodynamics performance of the launch vehicle, RANS high-fidelity calculations have been performed on several key flight points (Figure 16).

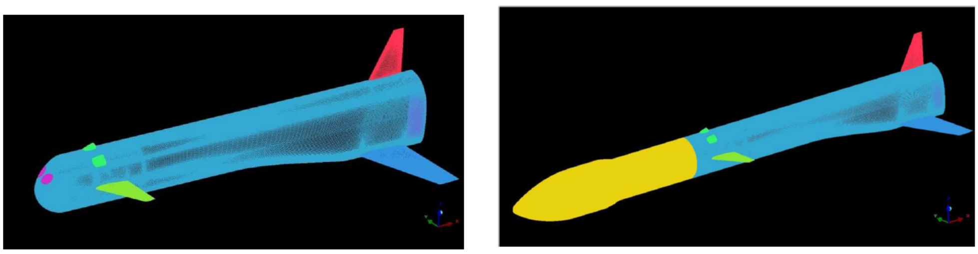

Figure 16: Aerodynamical meshes for composite and first stage configurations

The first critical key-point concerns the reentry at a high angle of attack. The CFD calculations allow to evaluate the position of the shock at different flight configurations to evaluate the feasibility of the reentry (Figure 17).

The second critical key-point is the perturbation of the flow due to the ignition of the turbojets. To evaluate this aspect, different RANS calculations have been performed with turbojets on and off in order to identify the impact in terms of lift decrease (Figure 18) and to refine the aerodynamics coefficients estimation for the powered RTLS cruise flight.

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Figure 17: RANS calculation of reentry phase

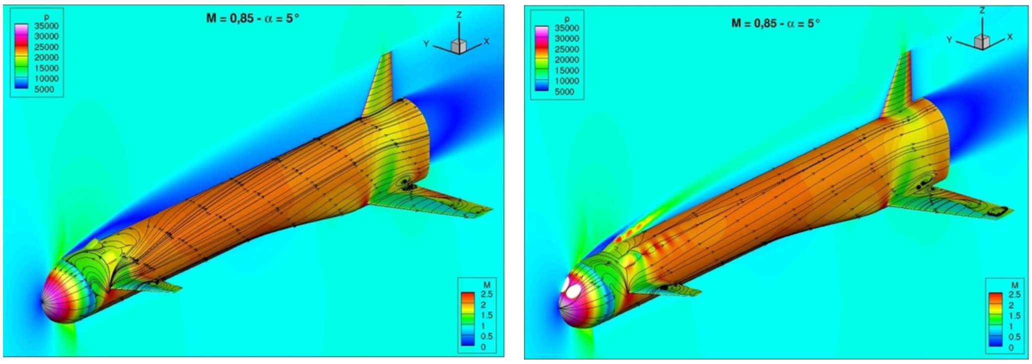

Figure 18: RANS calculations of RTLS cruise flight with at left: turbojets off and at right: turbojets on

5.2 Air-breathing propulsion

The air-breathing engine performance have also been refined by using an in-house thermodynamic cycle simulation code in order to simulate the thrust and specific fuel consumption on the obtained reference trajectory. To do so, an off-design phase has been conducted to simulate the EJ200 engine characteristics. Then, an on-design study has been carried out to evaluate the performance of the engine and compare with the preliminary results obtained in the trade-off assessment phase. Whereas the thrust profile directly corresponds to the one used in the preliminary phase, the specific fuel consumption differs by 10%. This penalization has been included for refining the performance of the vehicle.

5.3 Performance assessment

From the aerodynamic and propulsion refined models, the design of the candidate architecture has been updated. To do so, an additional multidisciplinary optimization has been conducted taking the into account the new aerodynamic coefficients estimations and the new air-breathing engine data. The updated trajectories are described in Figures 19 and 20. Several differences can be noted on these trajectories, both for the ascent phase (especially on the velocity profile as seen on the pitch angle profile) and the return phase (especially concerning the maximal distance to the landing site). Consequently, the overall performance using refined models has grown of approximately 5% in terms of GLOW.

The first stage of the final flyback architecture with updated performance is illustrated in Figure 21.

5.4 Robustness analysis

In order to consolidate the design and refine the design margins (given by CNES), a robustness analysis has been performed with respect to model uncertainty (*e.g.*, specific impulse, subsystems masses, drag coefficient estimation). Because this type of architectures has never flown, it is quite important to evaluate the dispersion of the performance with respect to model uncertainties. For that purpose, in collaboration with CNES, 8 uncertainties have been considered: on the specific impulse of both the first and the second stages, on the mass flow rate of the engines, on the thrust and specific fuel consumption of the air-breathing engines, on the dry mass of

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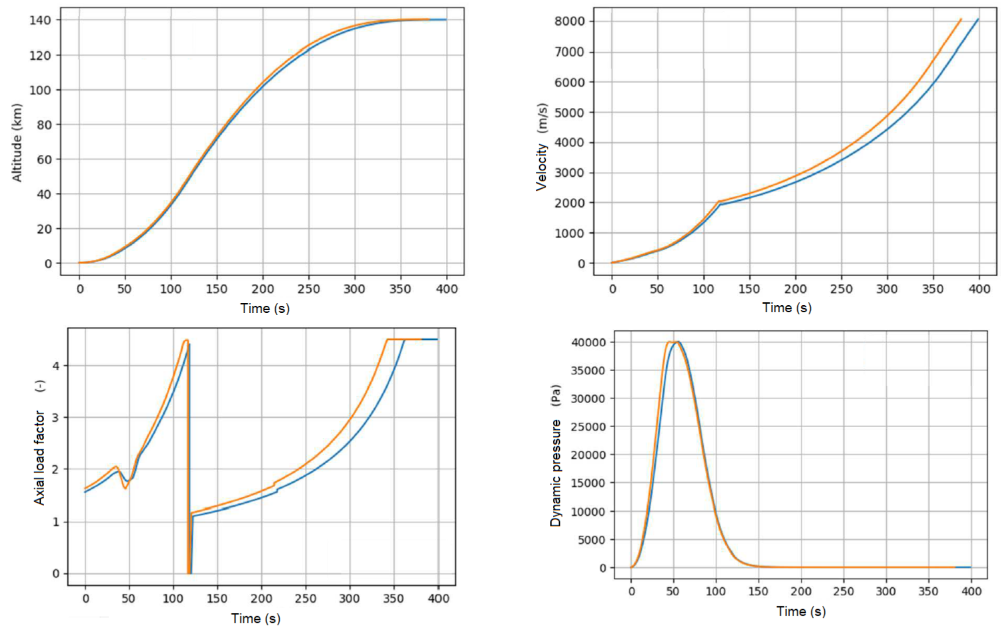

Figure 19: Ascent phase trajectory profiles, with in orange: the reference trajectory and in blue: the trajectory obtained with refined models

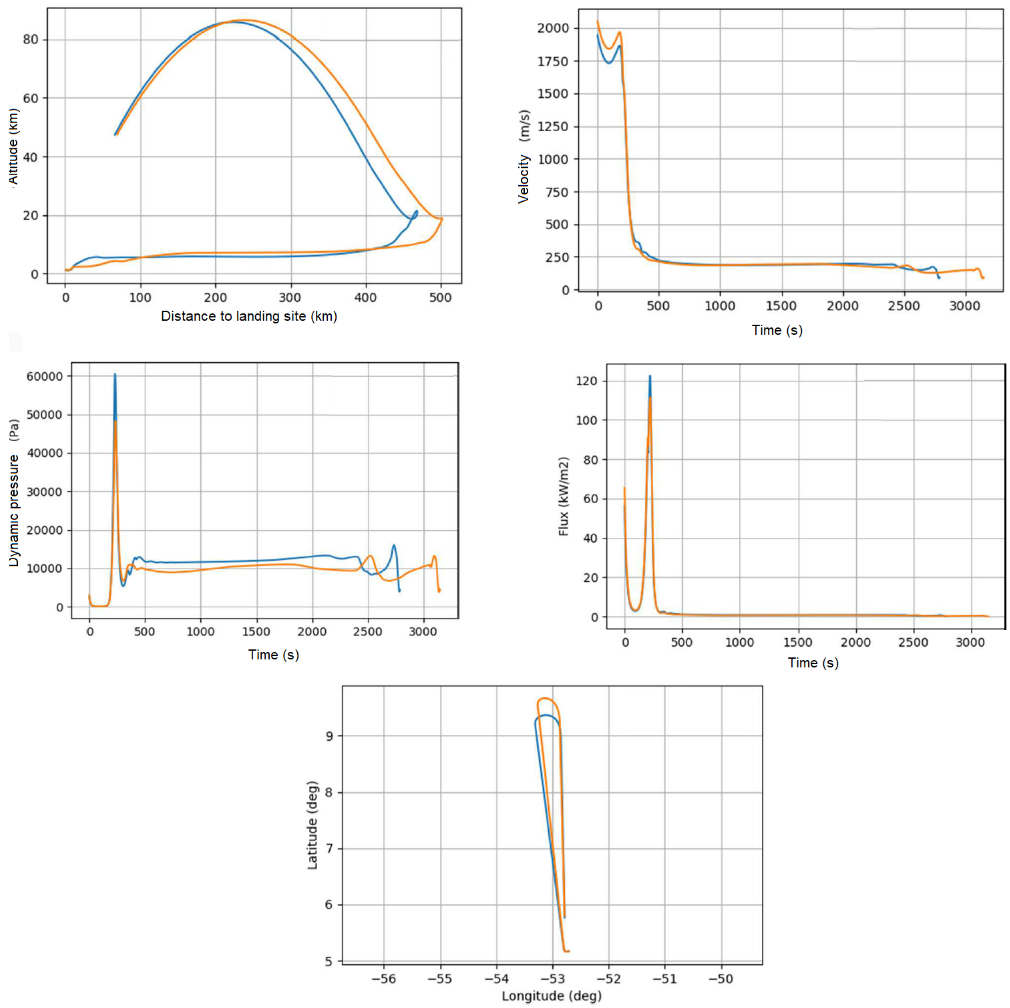

Figure 20: RTLS trajectory profiles, with in orange: the reference trajectory and in blue: the trajectory obtained with refined models

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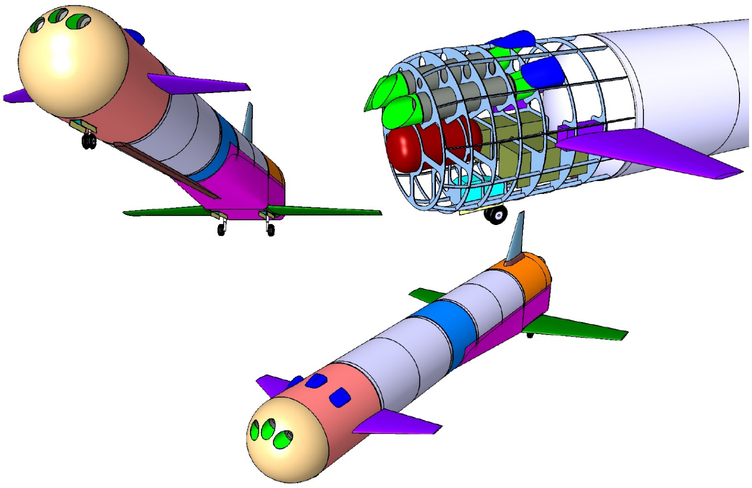

Figure 21: Illustrations of the first stage with the reusability kit

the first stage and on the base drag of the first stage. OpenTURNS [2] library has been used for the uncertainty quantification study, using Monte-Carlo method. This allows to identify flight envelops for both ascent and RTLS phase, and especially for the final phase in order to evaluate the safety of the return trajectories (Figure 22).

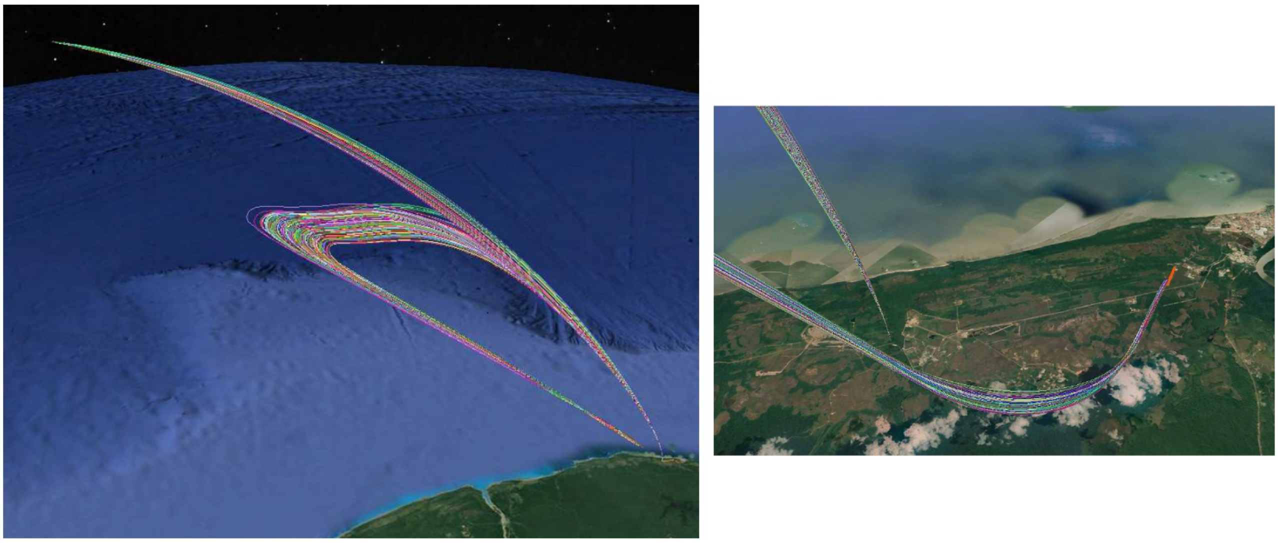

Figure 22: Dispersion of trajectories for both ascent and return phases

Finally, a sensitivity analysis has been conducted in order to identify which uncertainty has the greatest impact on the overall performance (Figure 23). This has been achieved by computing Sobol' indices from chaos polynomial built on the dispersed trajectories data [19]. It appears that the uncertainty on the specific impulse of the second stage engine has the most effect of the dispersion of the performance of the vehicle, following by the uncertainty on the specific impulse of the first stage. This study allows to identify which model uncertainty (in this case, the uncertainty about the PROMETHEUS specific impulse) has to be refined in order to reduce the uncertainty of the RLV performance.

6 Conclusions

This paper explores a new type of architectures for reusable first stage of launch vehicle that uses aeronautics technologies such as lifting surfaces and turbojets. All the different key disciplines that allow to assess trade-off between different flyback configurations are described. Aeropropulsive analyses and trajectory optimizations (for both ascent and return phases) have been carried out to find optimal configurations and select

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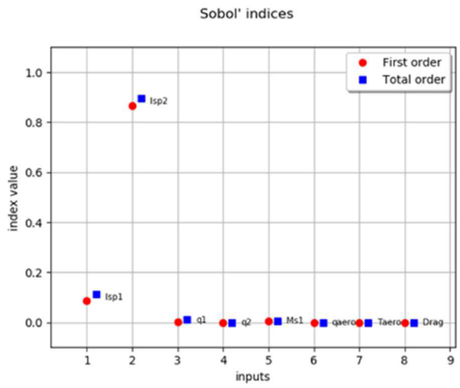

Figure 23: Sensitivity analysis on the overall performance for the different uncertainties (Isp: specific impulse, q: mass flow rate, Ms: dry mass, T: thrust)

the most appropriate turbojet type for the return-to-launch-site mission. Eventually, high-fidelity aerodynamics computations and robustness analyses have been performed to refine the designed configuration. This study has illustrated that fly-back architectures are competitive with respect to more classical configurations (*e.g.* toss-back) in terms of ascent mission performance and could be an interesting alternative in the future.

Acknowledgments

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