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AERODEM

AEROcapture DEmonstration mission on Mars

ESR

EXECUTIVE SUMMARY REPORT

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CONTRACT	Customer	Contract number			Programme
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	Contractual document	C.Work Unit	Item	Sub-item	CWS E-280145-27
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TITLE	AERODEM – ESR– EXECUTIVE SUMMARY REPORT

ABSTRACT

This document presents the ESR of AERODEM Pre-phase A activity.

KEYWORDS

AERODEM / MARS / AEROCAPTURE / FLEXIBLE INFLATABLE HEATSHIELD / AEROSHELL / MBSE

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Summary

1 INTRODUCTION AND OBJECTIVES

The AERODEM is an ESA demonstration mission that will implement an aerocapture phase in order to provide an atmospheric ΔV instead of a standard chemical one for orbiting a SpaceCraft around Mars. The aerocapture technique currently studied relies on a discrete event drag modulation technique that modifies only once the aerocapture module ballistic coefficient by separating / retracting the heatshield. In the perspective of future large missions to Mars, the mission will have to collect and transmit to Earth technology flight measurements that will be necessary for the implementation of the single drag modulation aerocapture technique.

A pre-phase A study has been conducted by a consortium led by ArianeGroup with Airbus Defense & Space – Madrid, RTech and Jehier-Hutchison as partners.

Two configurations are considered for AERODEM:

- The Standalone configuration
- The Piggy-Back configuration

2 ACRONYMS AND ABREVIATIONS

NAME	DESCRIPTION
AD	Applicable Document
AERODEM	AEROcapture DEmonstration mission on Mars
AGS	ArianeGroup SAS
AOCS	Attitude and Orbit Control System
AS	AeroShell
AU	Astronomical Unit
CoG	Center of Gravity (Center of Mass / CoM)
COTS	Commercial Off The Shelf
DST	Deep Space Transponder
EIP	Entry Interface Point
EPS	Electrical Power System
ESA	European Space Agency
FPA	Flight Path Angle
GNC	Guidance, Navigation and Control
HGA	High Gain Antenna
IMU	Inertial Measurement Unit
I/F	Interface
KS	Kick Stage
LGA	Low Gain Antenna
MGA	Medium Gain Antenna
MLI	Multi-Layer Insulation
PCDU	Power Control and Distribution Unit
RD	Reference Document
S/C	Spacecraft
SoW	Statement of Work
TCM	Trajectory Correction Maneuver
TN	Technical Note
TRL	Technology Readiness Level
TPS	Thermal Protection System

3 APPLICABLE AND REFERENCE DOCUMENTS

3.1 APPLICABLE DOCUMENTS

This paragraph lists the documents that are applicable for the considered activity in accordance with the applicability mentioned explicitly wherever relevant.

[AD1] ESA Contract No. 4000138143/22/NL/GLC/my with ArianeGroup SAS - AEROCAPTURE DEMONSTRATION MISSION F signed on 20/05/2022

3.2 REFERENCE DOCUMENTS

[RD-1] AERODEM-AGS-TN-01: AERODEM – Mission Architecture Concepts

[RD-2] AERODEM-AGS-TN-02: AERODEM - Architecture trades and baseline concept selection

[RD-3] AERODEM-AGS-TN-03: AERODEM – System Conceptual Design

[RD-4] AERODEM-AGS-TN-04: AERODEM MBSE Approach and Lessons Learned

[RD-5] AERODEM-AGS-C1: AERODEM Commercialisation Analysis

4 MISSION ANALYSIS

4.1 STANDALONE MISSION

Several different mission concepts were studied and basically three of them have been focused on as summarized here-below:

- **D.K.1**: This concept assumes a kick-stage (or transfer module) and a direct launch at Mars transfer orbit. The kick-stage is needed not so much for the delta-V needs but for supporting Orbiter during the transfer in terms of power and/or other subsystems.

- **G.K.1**: This concept assumes a launcher release in GTO (or GTO like orbit). In this case the AERODEM will be compound of a KS based on chemical propulsion.

- **H.K.1:** This concept is similar to concept G.K.1 but assuming a release in a HEO or *zero-departure-energy* orbit. One of the appeal of a release from C_3=0 compared to a GTO transfer is to check the compatibility with an A62 shared launch

Regarding the functional sharing, two different functional concepts for each mission architecture were considered as possible alternatives:

A- A solution to carry only elements that needs visibility in the KS is proposed. Therefore, comms antennas and/or OHs managed by the Orbiter transponder or STR EU, would be located in the KS, having enough visibility. These units would be switched to nominal ones in the Orbiter at the separation point between KS and the aerocapture module.

B- The fully autonomous KS solution is in principle the more expensive as the KS should be equipped with at least a CMDU with independent CSW, a complete power subsystem, with SA, battery and PCDU and deep space comms subsystem.

With these qualitative approach, the most suitable solution is found to be D.K.1.B direct transfer orbit with an autonomous configuration allowing for a very similar design for both Orbiter and Kick-Stage.

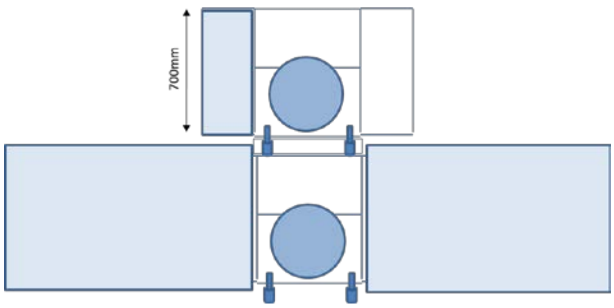


Figure 4-1: Proposed architectures of the auxiliary module for D.K.1 concept

4.2 PIGGY-BACK MISSION

Several mission concepts are considered for the Piggy-Back scenario in terms of two factors: the type of carrier for the piggyback spacecraft and the way to perform atmospheric insertion. Carrier options included fly-by (F), orbiter (O), and lander (L) missions. Atmospheric insertion options included: (1) the piggyback spacecraft performing a fully autonomous maneuver after release from the carrier, (2) the required delta-V for atmospheric entry being provided by the separation mechanism attached to the carrier, (3) the carrier performing a maneuver to leave the piggyback spacecraft on the required trajectory, (4) a combination of the two previous options, or (5) the piggyback spacecraft being released by a lander carrier upon entry. The full set of options considered are shown in Figure 4-2.

The trade-off performed resulted in the selection of O4 i-e the lander carrier releasing the piggyback S/C on entry.

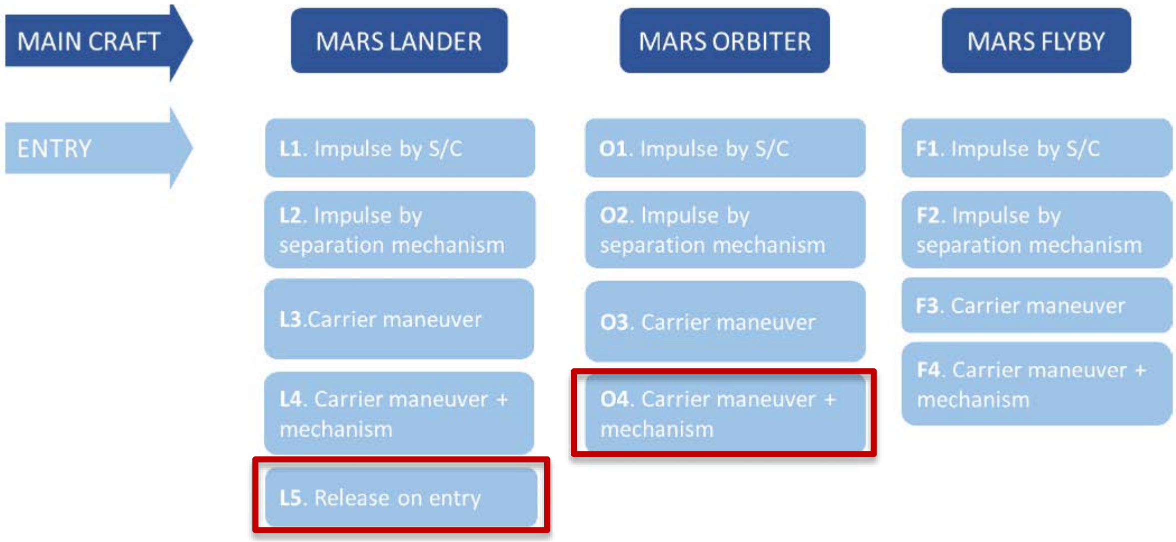


Figure 4-2: Preliminary mission concepts defined for the piggyback spacecraft

5 INTERPLANETARY TRANSFER & MARS APPROACH

The compatibility with the mission with A62, A64 and Vega-C were studied for different mission architectures. Vega-C launcher was discarded due to its low performance for an interplanetary mission. It was also discarded at that time electrical propulsion solution for the transfer phase.

Here after ballistic transfer windows are summarized:

Mission type	Earth departure	Transfer duration (days)	C_3 (km^2/s^2)	Excess speed Mars (km/s)	Declination (deg)
Type 1	28/1/2031	190	9.00	5.54	-34.2
Type 2	23/2/2031	320	8.24	5.53	1.02
Type 1	1/3/2031	210	17.89	3.78	-25.3
Type 2	13/12/2030	286	12.48	3.45	8.54
Type 1	6/4/2033	178	8.41	3.96	-54.9
Type 2	28/4/2033	274	7.78	4.38	-11.2
Type 1	20/4/2033	200	9.23	3.31	-53.2
Type 2	26/1/2033	264	17.78	3.83	-2.53

Table 5-1: Performances of ballistic transfer trajectories for selected date range [RD-2]

For the stand-alone mission, the first phase is to perform the Earth escape manoeuvres to achieve the patch conic trajectory to almost reach Mars. Usually and due to planetary protection the target orbit is designed to just miss Mars.

Next figure shows the relative interplanetary phase when launching at 6/April/2033. A very similar interplanetary phase is obtained when launching just 2 weeks after, having 22 days more of travel and need a higher characteristic energy, but providing the lowest excess speed at Mars, so making it very attractive for the aerocapture demonstration mission.

These orbits (Type I) allow a simpler communications system as the distance to Earth is limited. As it is shown in next figure, the distance with Earth is lower than one AU. It provides also an easy configuration for solar array and antenna. The attitude during the interplanetary transfer will be mostly aligned the X-Aerodem axis with the velocity vector, B axis pointing perpendicular and inside the Aerodem-Sun plane.

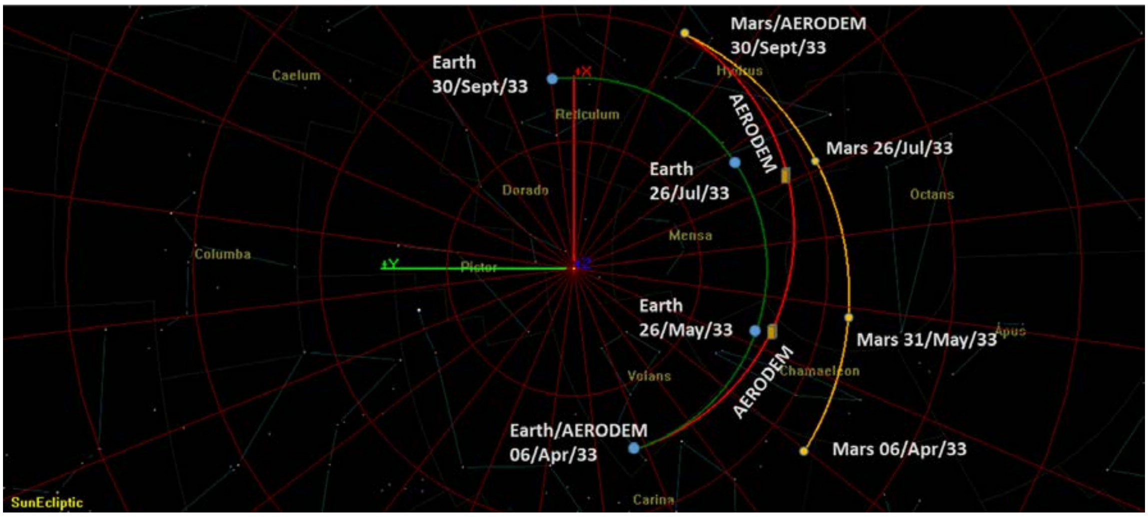


Figure 5-1: Mission timeline for a departure in April 2033

A dedicated analysis was performed in order to refine the expected performances of the interplanetary navigation; the orbit determination especially during the last weeks is of important relevance for the aerocapture success. This is the major contribution for the errors on the flight path angle at Entry Interface Point (EIP).

For that purpose, only ground based radiometric ranging, range-rate and Delta-DOR which are fundamental at high distance from Mars were considered.

The figure below shows the accumulated probability density of the FPA error, so that the 3-sigma value is about 0.23° providing an entry corridor width of 0.46°.

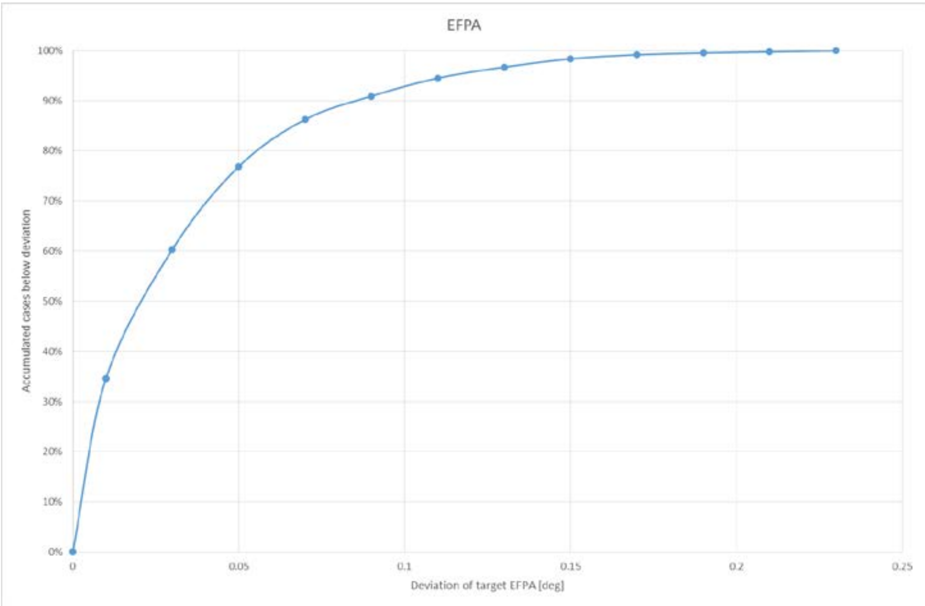


Figure 5-2: Entry FPA accumulated probability function

6 AEROCAPTURE PHASE

6.1 ENTRY CORRIDOR DETERMINATION

In this section is described the atmospheric leg mission analyses performed for the different configurations (Standalone, Piggyback) and accounting for the identified main constraints. The reference scenario is identified, providing the wider range in terms of flight path angle's corridor, i.e. define a nominal inbound periapsis and associated release time to reach a target apoapsis of 1000km. Consistently with the dispersion at EIP analysis results, an entry corridor width of +/- 0,24° is targeted.

A reference value for the hyperbolic excess velocity of 3.83 km/s was selected. Besides, the target operational orbit, following periapsis raise maneuver, shall have an inclination above 70°. A polar hyperbolic inbound trajectory has been considered as a first approach. The hyperbolic inbound trajectory parameters are synthetized hereafter:

Hyperbolic excess inertial speed	3.83 km/s
Periapsis	To be defined
True anomaly	-90°
Inclination	90°
Longitude of the ascending node	0°
Periapsis argument	0°

Table 6-1: Hyperbolic inbound trajectory parameters

For the graphs and associated parameters presented below, the following notations have been used:

- *slope_atm_entry* : Slope at atmospheric entry (°)
- *alt_flex_release* : Altitude at flexible release (m)
- *dur_atm_pass* : Duration from atmospheric entry to exit (s)
- *dur_atm_entry_flex_release* : Duration from atmospheric entry to flexible release (s)

A parameter of interest for the atmospheric leg mission analysis is the β_{ratio}, i.e. ballistic coefficient ratio, defined as the ratio between the ballistic coefficient computed after the flexible heat-shield release over the one defined before the release.

6.1.1 STANDALONE MISSION

Several heat shield designs have been studied, with varying half-cone angles from 45° up to 70°. In order to ensure sufficient stability margins at atmospheric exit, each design is associated to a minimal outer diameter for the rigid part of the heat shield. Based on preliminary mass assumptions, the mass of the rigid part of the heat shield for each design, the accessible mass for the flexible part of the heat shield is derived, together with its associated diameter. Hence, the β_{ratio} can also be computed. Based on the obtained flight path angle corridors, the 70° configuration was preferred.

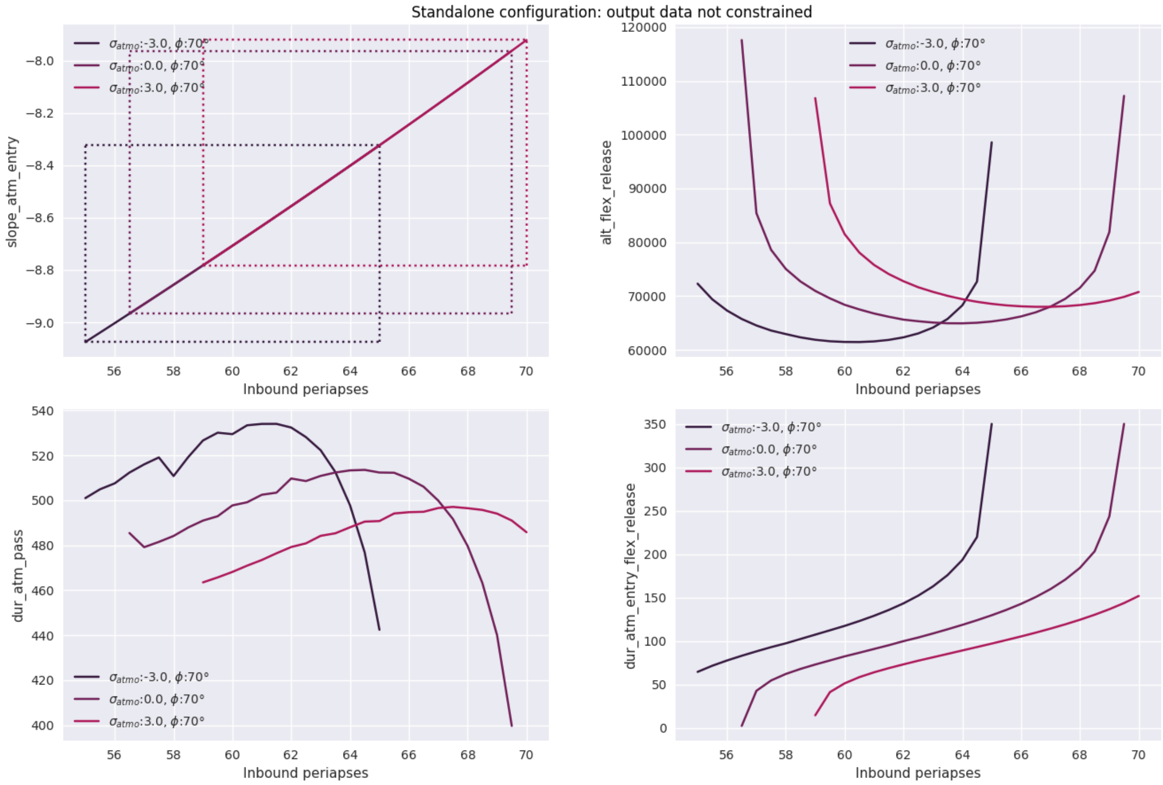


Figure 6-1: 70° half-cone angle, 470kg vehicle increased mass: output data not constrained as function of the inbound periapsis

In terms of flight path angle corridor, one has for a 62km reference periapsis:

- A positive range of +0.23°, limited by the -3σ case and a maximal accessible periapsis of 65km
- A negative range of -0.23°, limited by the +3σ case and a minimal accessible periapsis of 59km

NOTA. Increasing the flexible part of the heat shield has allowed to increase the range of slopes at atmospheric entry only by improving the low-density case. One is still limited on the +3σ case by the sizing of the rigid part of the heat shield.

6.1.2 PIGGYBACK MISSION

As done for the Standalone mission, several heat shield designs have been studied, with varying half-cone angles from 45° up to 70° with an expected optimal design at 70°. In order to ensure sufficient stability margins at atmospheric exit, each design is associated to a minimal outer diameter for the rigid part of the heat shield.

As it has been done for the Standalone configuration, the mass of the vehicle has been increased to target a β_{ratio} of 4.0, the latter being the value that provides a satisfying flight path angle for the Standalone configuration. The outer diameter of the flexible is increased from 1.99m up to 3.60m. Associated graphs are presented below:

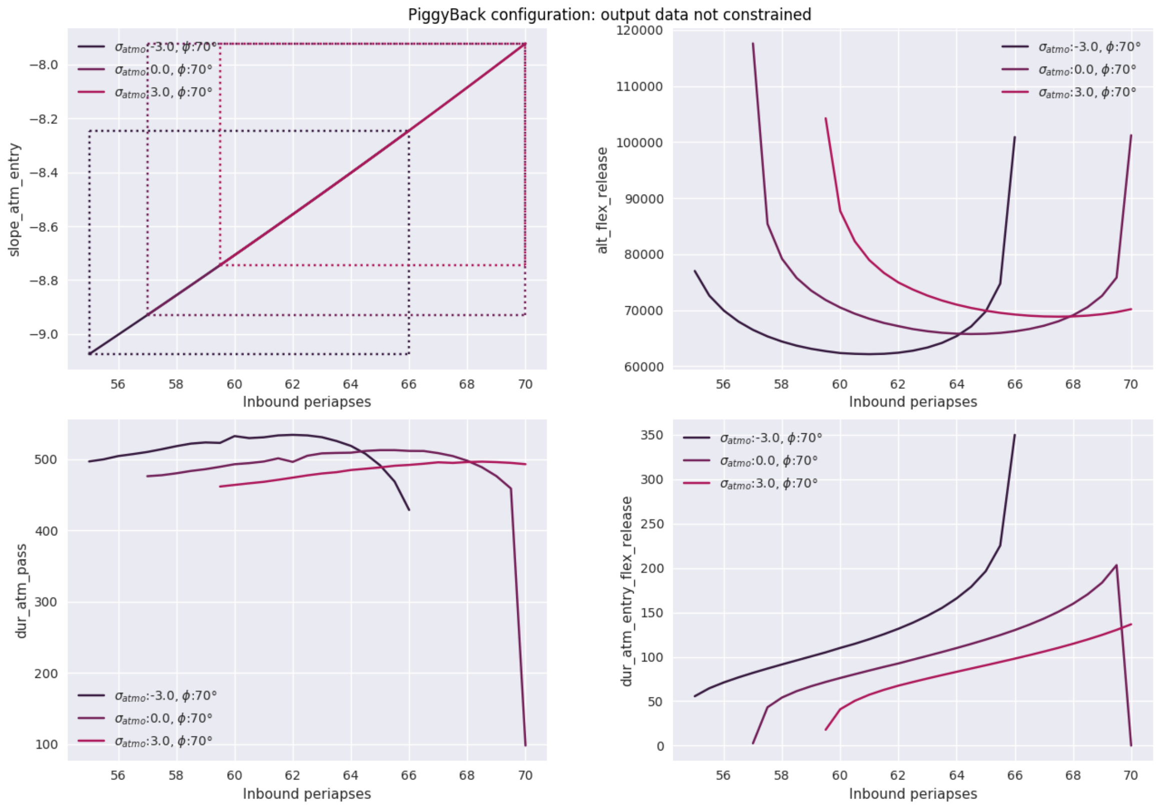


Figure 6-2: 70° half-cone angle, 155kg vehicle increased mass: output data not constrained as function of the inbound periapsis

In terms of flight path angle corridor, one has for a 62.75km reference periapsis:

- A positive range of +0.25°, limited by the -3σ case and a maximal accessible periapsis of 66km
- A negative range of -0.25°, limited by the +3σ case and a minimal accessible periapsis of 59.5km

NOTA. Increasing the flexible part of the heat shield has allowed to increase the range of slopes at atmospheric entry only by improving the low-density case. One is still limited on the +3σ case by the sizing of the rigid part of the heat shield.

6.2 GUIDANCE, NAVIGATION AND CONTROL

For the AERODEM standalone or piggy-back missions, the spacecraft designed to perform the aerocapture maneuver is only compliant with a single event drag modulation process. Thus, at a given time that is determined by the on-board guidance to meet a requirement on the apoapsis at exit, a flexible heatshield (inflatable or deployable) used to deplete the energy of the vehicle is jettisoned. By modifying the drag, this event is enough to drastically change the orbital parameters of the interplanetary incoming path such that the reached orbit at atmosphere’s exit is already elliptic.

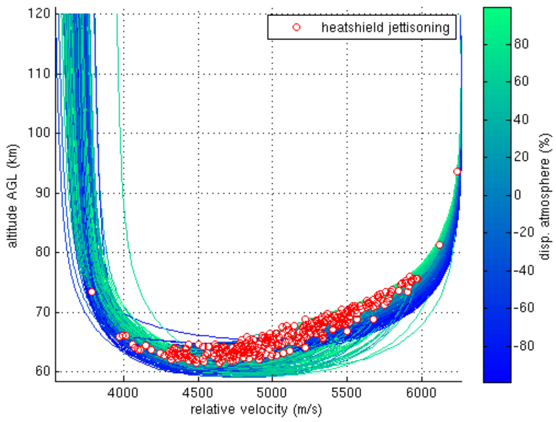
An algorithm solution relying on a numerical predictor-corrector technique has been eventually retained to design the guidance.

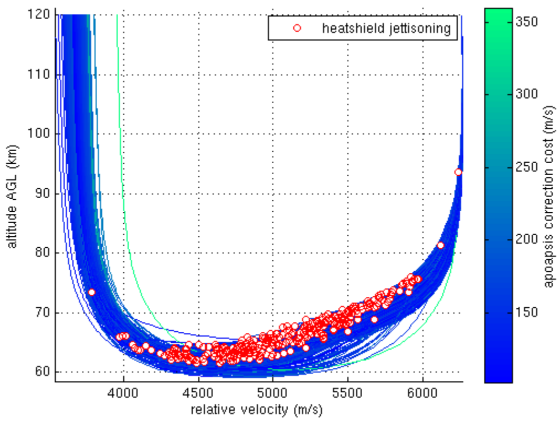
The performance assessment of the guidance is achieved through Monte-Carlo simulations considering dispersions at EIP resulting from the last TCM and a performance model for the atmospheric Navigation.

6.2.1 Standalone Mission

Considering a total mass of 470 kg at entry, we observe a satisfactory performance obtained with the Navigation performance model, as illustrated on next figures.

The apoapsis at exit meet the mission requirement in 97 % of the simulated cases with 2 cases below 500 km AGL (403 and 490 km AGL) and 7 cases over 1500 km AGL (1 case at 2900 km AGL and 6 cases ranging from 1517 to 1762 km AGL). Reaching the parking orbit with an eccentricity below 0.05 is thus obtained with a correction cost below 300 m/s in 99.7 % of the simulated cases, see next figures.





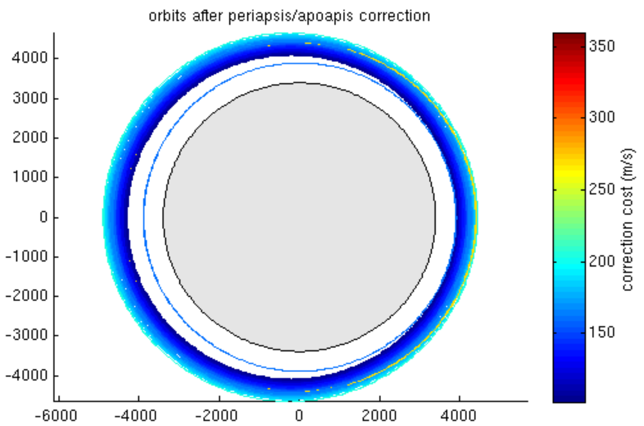


Figure 6-3: Improvement of the performance obtained with the AEROFAST Navigation performance model

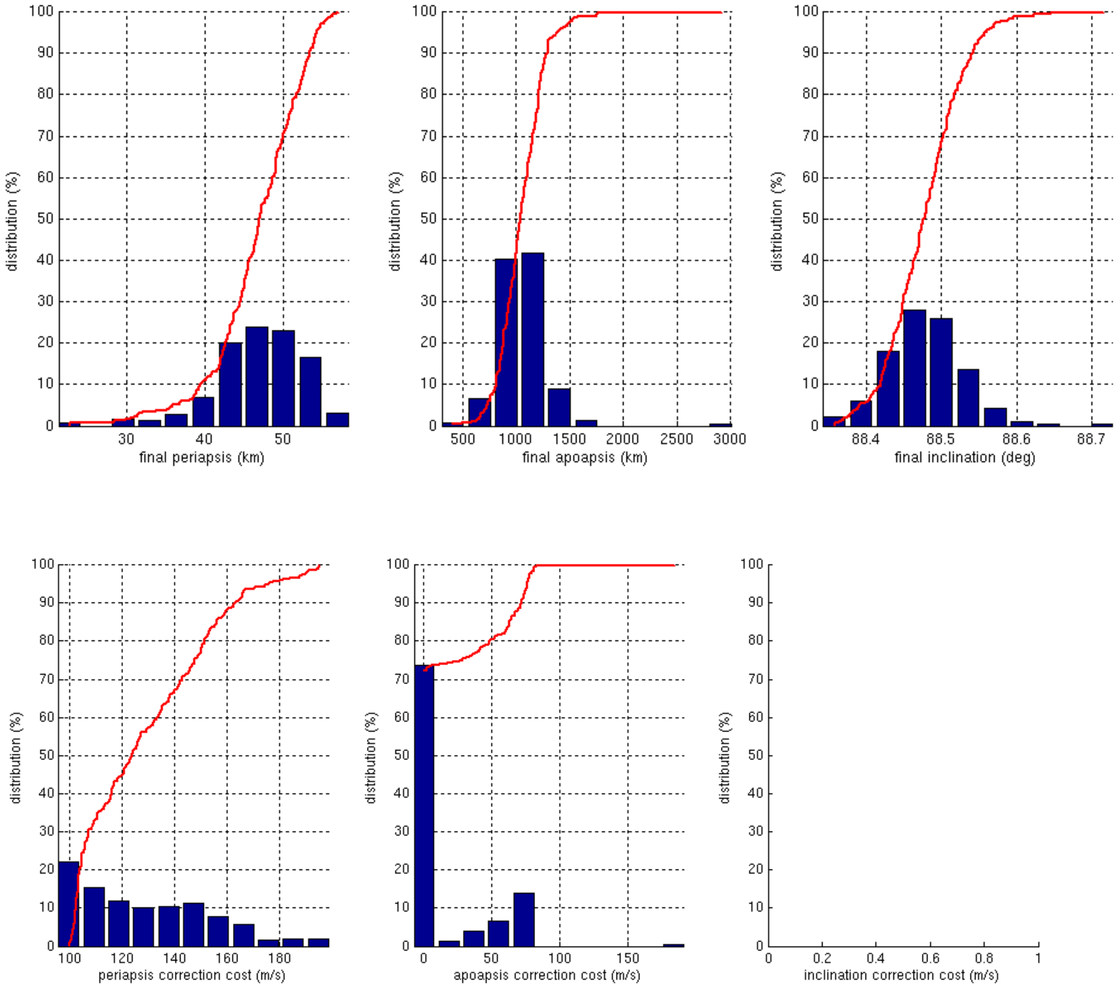


Figure 6-4: Reaching the parking orbit with an eccentricity below 0.05 is thus obtained with a correction cost below 300 m/s in 99.7 % of the simulated cases

6.2.2 Piggy-back Mission

Running a Monte-Carlo simulation over the same of off-nominal flight conditions and with the Navigation performance model yields a 97 % fulfillment rate of the mission requirement on the apoapsis at exit. That is similar to the standalone mission results but with limited overshoots, the correction cost to reach the parking orbit remaining below 270 m/s in all the cases, see next figure.

Cases over the requirement on the apoapsis at exit are limited to 9 with 8 cases ranging from 1511 up to 1738 km AGL and 1 case ending at 399 km AGL, see next figure. It has to be noted that the simulation cases beyond the apoapsis requirement are the same as for the standalone mission (same off-nominal flight conditions with same Navigation performance model).

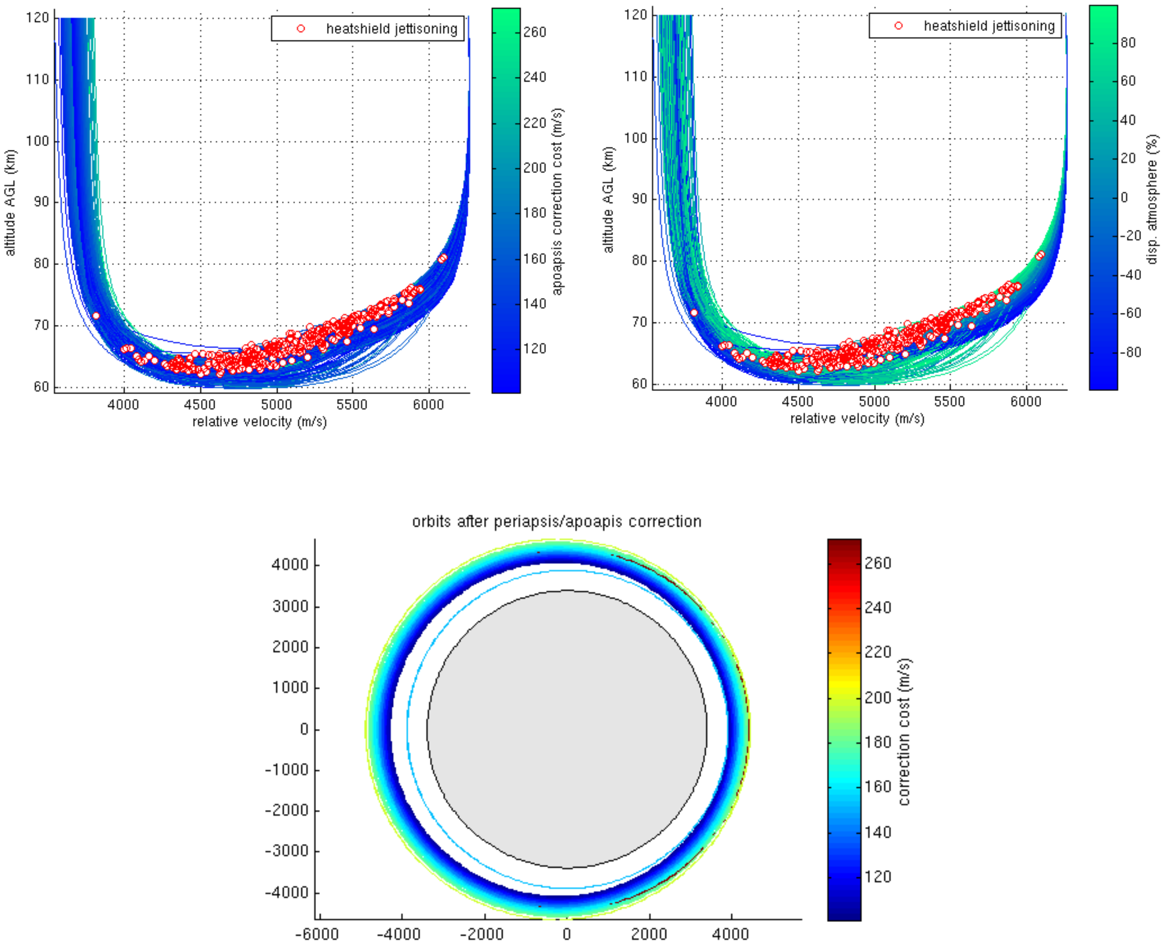


Figure 6-5: Monte-carlo simulation – Piggy Back mission

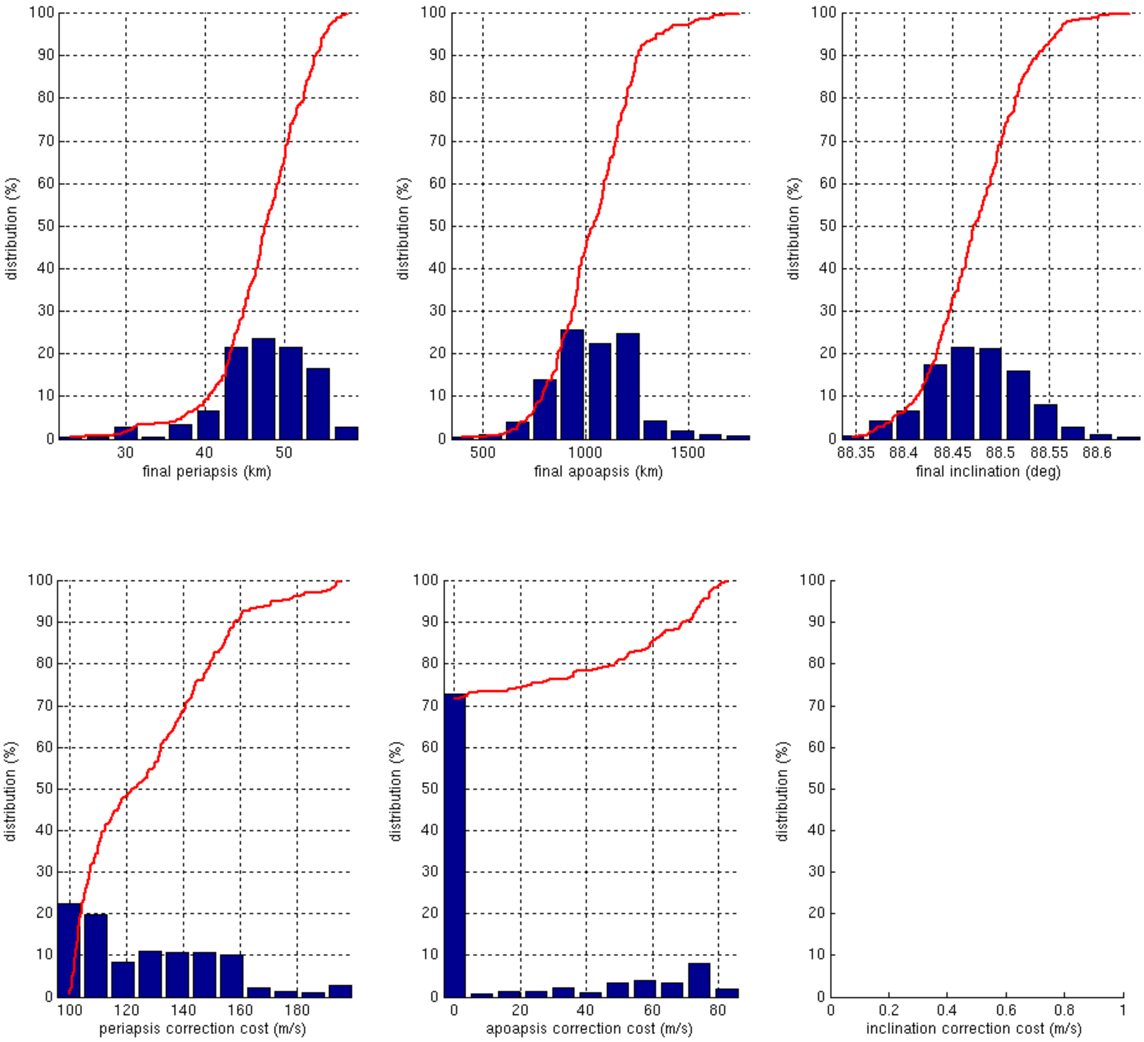


Figure 6-6: Final parameters – Piggy Back mission

7 AEROCAPTURE TECHNOLOGIES TRADE-OFF

The trade-off analysis for the flexible heatshield focused on 5 different axisymmetric concepts referenced as:

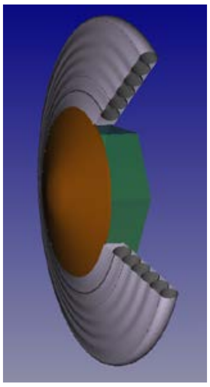
- **Stacked of Inflatable toroid with double shell encapsulation: Double shell toroid**. The aeroshell is comprised of structural and TPS components. The structure is constructed from a series of stacked inflatable torus tied to each other and to the vehicle with encapsulating shell.
- **Tension shell with a unique shoulder inflatable torus**: The Tension shell use an inflatable structure of a trailing torus shape to support a conical membrane. The concave curvature of the membrane depends on the conical angle, the external loads as well as the meridian tension in the membrane cannot be strictly conical. Therefore, the drag loading is essentially supported by the membrane.
- **Inflatable Dual volume – EFESTO**: The aeroshell is comprised of structural and TPS components. The structure is constructed from a unique large inflatable torus to the vehicle at the edge of the rigid heat shield substructure. The inflatable structure is fully encapsulated in an external shell. The initial concept coming from EFESTO project is design to sustain large aerothermal and mechanical loads. The 45° cone angle is a consequence of this aerodynamic load constraints. As for a wider cone angle (70° e.g.) the section of the main torus cannot be a circle anymore but must be flattened and of elliptic shape.
- **Deployable shell with telescopic poles**: The concept is based on the ADEPT technology which is a flexible multilayer fabric that forms a semi-rigid membrane when pre-tensioned by deployment of supporting ribs. This multi-layered fabric must transfer aerodynamic loads to the support structure while operating at high temperatures due to aeroheating. The bottom layers of the cloth carry the aerodynamic load while the top layers manage the thermal and chemical energy of the plasma.
- **Deployable thermostructural rigid panels**: The concept of deployable rigid panels relies on the capacity to manufactured lightweight stiff hot structural petals. Once deployed, these petals form an integral unit acting as a pressure barrier to atmospheric pass aerodynamic forces. An outer smooth profile can be obtained by the design of meridional overlapping from one panel to its neighbor one. In a simplicity and economical purpose, all the panels are identical.

The following list of criteria were considered:

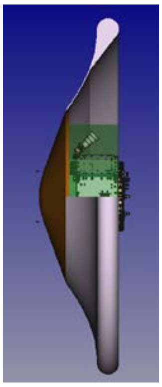
- Mass budget
- Stowed compacity
- TPS TRL
- Actuator TRL
- Retractability
- Manufacturability
- Impact on S/C design
- Scalability (geometric)
- Scalability (mission/planet)
- Geometric stability (impact on aerodynamics/aeroheating)

Regarding the proposed criteria, the torus stack type concepts appeared to have the worst ranking while the tension shell family won the trade-off for the typical specifications of an aerocapture on Mars. The torus stack finally presents few differences with the dual volume

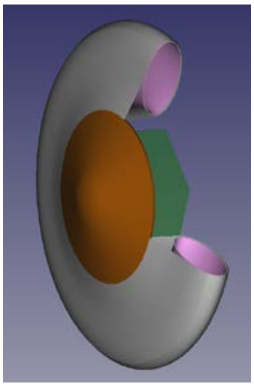
but this latest offer the advantages of less complexity in manufacturing the inner volume and could offer a better spacecraft protection.
It is worth noticing that the gap between the inflatable tension shell and the deployable umbrella like was small and more detailed analysis could challenge the inflatable solution versus the performance of the deployable concept. It is however believed that the deployable concept is not easily scalable to very large Orbiter and thus the inflatable tension shell concept was retained; the inflatable Tension Shell configuration was finally preferred for the study.



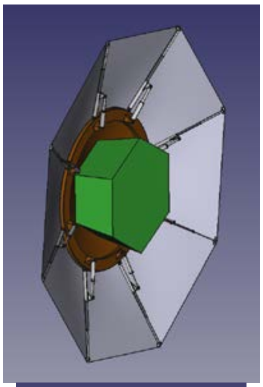
Double Shell Toroid

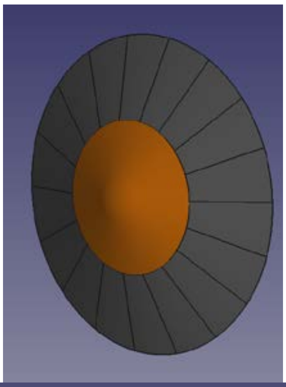


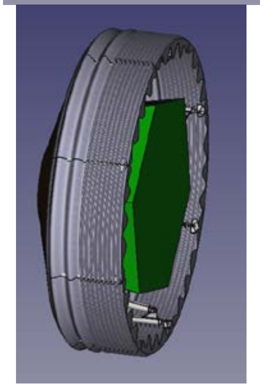
Tension Shell



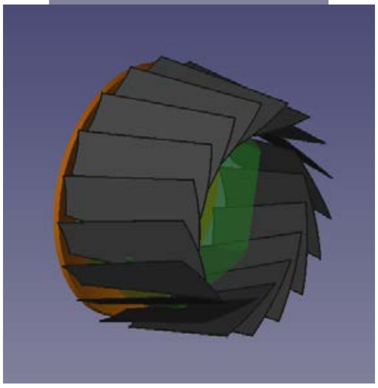
Inflatable Dual Volume (EFESTO)







Deployable Shell with Telescopic Poles (ADEPT)



Deployable Thermo-Structural Rigid Panels

8 CFD CAMPAIGN

DSMC computations have been afforded in order to assess the drag and the stability during the rarefied flow regime. Laminar Navier-Stokes solutions have been also computed in order to assess the aeroheating during the most critical part of the trajectory at minimum altitude.
From the CFD campaign, following conclusions can be drawn:

- Figure 8-1 displays a comparison between the inflatable heat-shield configurations CFD and DSMC results against the ExoMars AEDB coefficient as used in the course of the present study. The drag coefficients remains in the considered uncertainty bands which validate the trajectory computations performed in atmospheric leg mission analysis and guided trajectories & dispersion analysis (summed up in §6). The figure also confirms the centre-of-pressure to be considered for computing the minimum rigid heat-shield diameter ensuring a minimum static margin of 1%,
- Heat-shield surface heat flux are found to remain significantly below both the rigid and flexible limit heat flux (1.8 and 0.35 MW/m² respectively),
- Flow re-attachment once the inflatable / deployable heat-shield is jettisoned (rigid heat-shield alone configuration) is not observed at 1° angle-of-attack, but still marginal so that in case of an angle-of-attack excursion, flow re-attachment may occur. Attention should be paid on the possibility to get a flow re-attachment which is highly not desirable on the Orbiter, with possible consequences on the minimal rigid heat-shield diameter definition which is currently only driven by the minimum static margin at aerocapture exit.
- Orbiter Surface heat flux (free of flow re-attachment) are found to be higher than the Solar Array limit heat flux (163°C translated into 1750 W/m²) and marginal with respect to MLI (385°C translated into 9140 W/m²). This might have some consequences on the selection of the MLI technology, and require specific actions for protecting the Solar Array with possibly the addition of a back-shell.

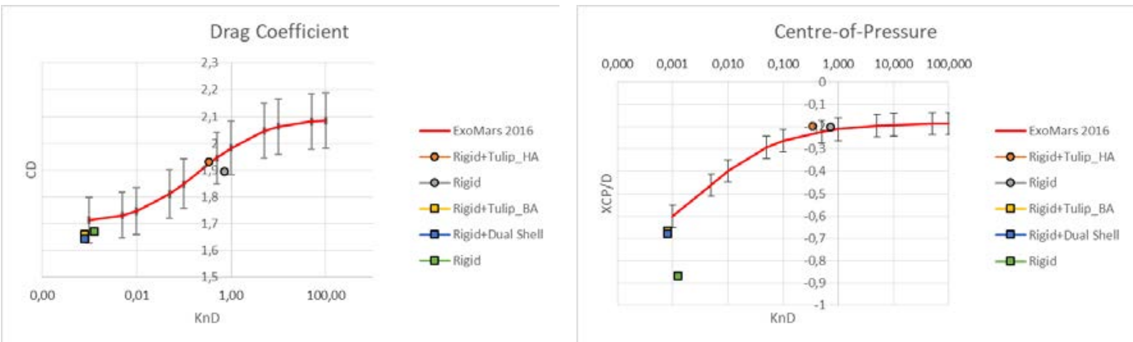
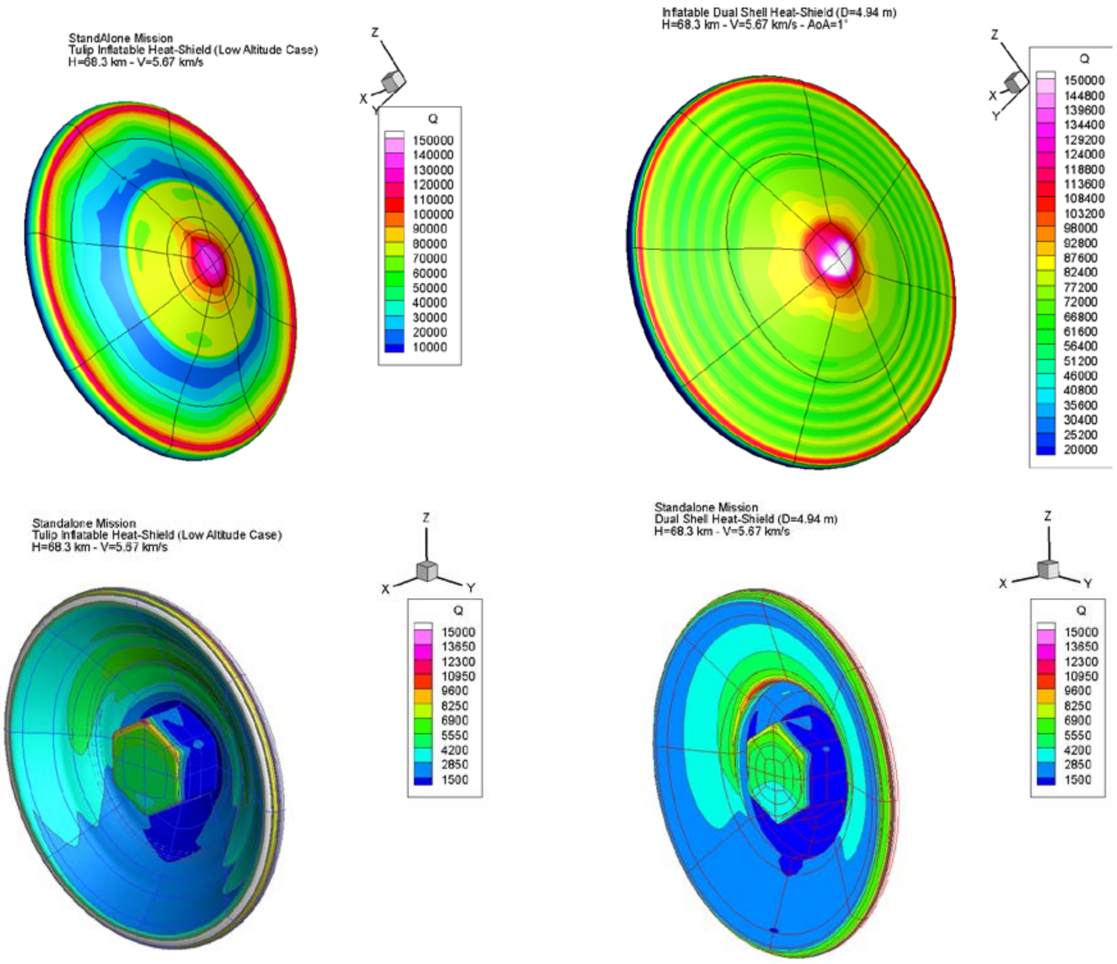


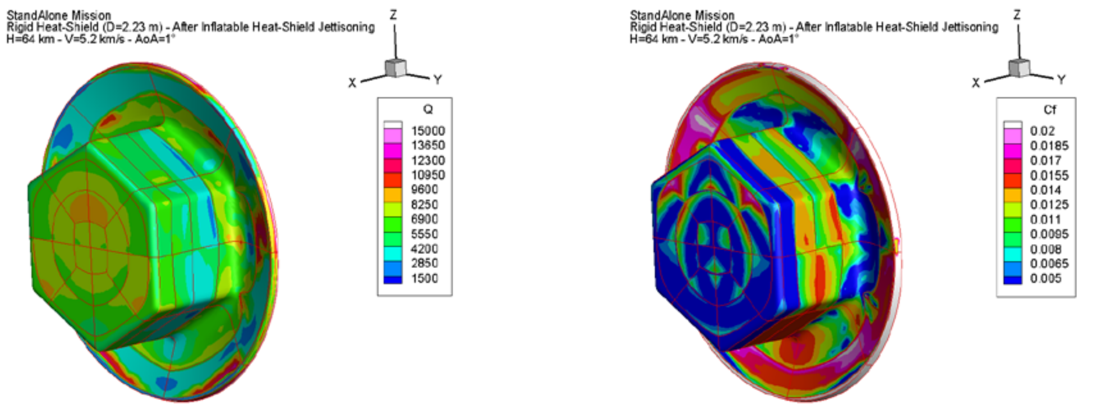
Figure 8-1 : Drag force coefficient and Centre-of-pressure



Tension-Shell

Dual Shell

Figure 8-2: Inflatable Heat-Shield Aeroheating



Heat-Flux

Skin Friction

Figure 8-3 : Post Flexible Heat-Shield Jettisoning Heat-Flux and Skin Friction Orbiter Distribution

9 FLIGHT MEASUREMENT PLAN

As part of the technology demonstration, the instrumentation plan will address following items:

- Aeroshell:
    - Aerodynamic drag and stability (IMU, accelerometers)
    - Orbiter Thermal Protection efficiency (Internal Thermocouples)
- Inflatable heat-shield:
    - Inflation system performance (pressure, temperature and mass flow rate sensors)
    - Kinematic of deployment (HR video)
    - Aerothermal environment (heat flux sensors, thermal plugs, radiometers, surface thermocouples, optics fiber, IR camera)
    - Mechanical environment (strain gauges, load cells)
- GNC performance (IMU, accelerometers, star sensors)

10 SPACECRAFT DESIGN

A general product Breakdown Structure of the AERODEM Spacecraft is presented below for the standalone configuration. For the Piggy-Back configuration, the Transfer Module has to be removed while keeping the Aerocapture Module.

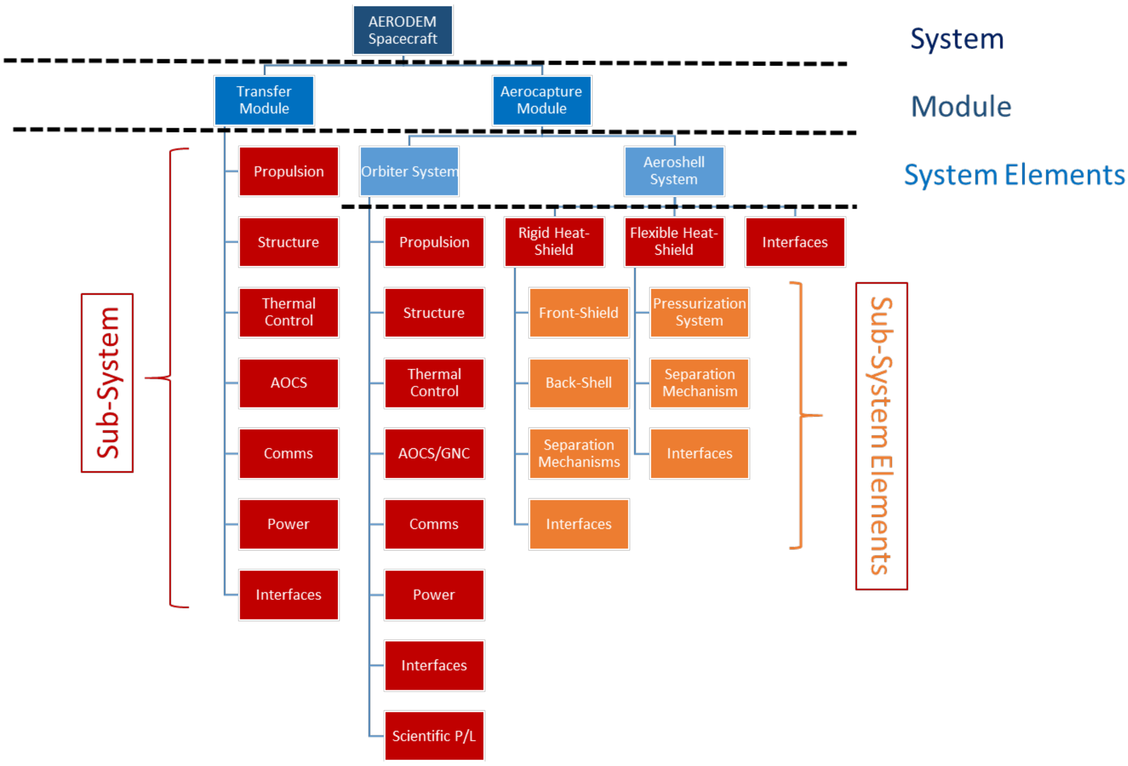


Figure 10-1 : StandAlone Configuration Product Breakdown Structure

A schematic description of the piggyback spacecraft's systems is presented in Figure 10-2. The AERODEM system can be divided in two main elements: the Aeroshell, consisting of a rigid and flexible part and an interface between them, and the Orbiter. The latter can be divided into the Orbiter platform, a Science payload yet to be determined, and the interfaces between them.

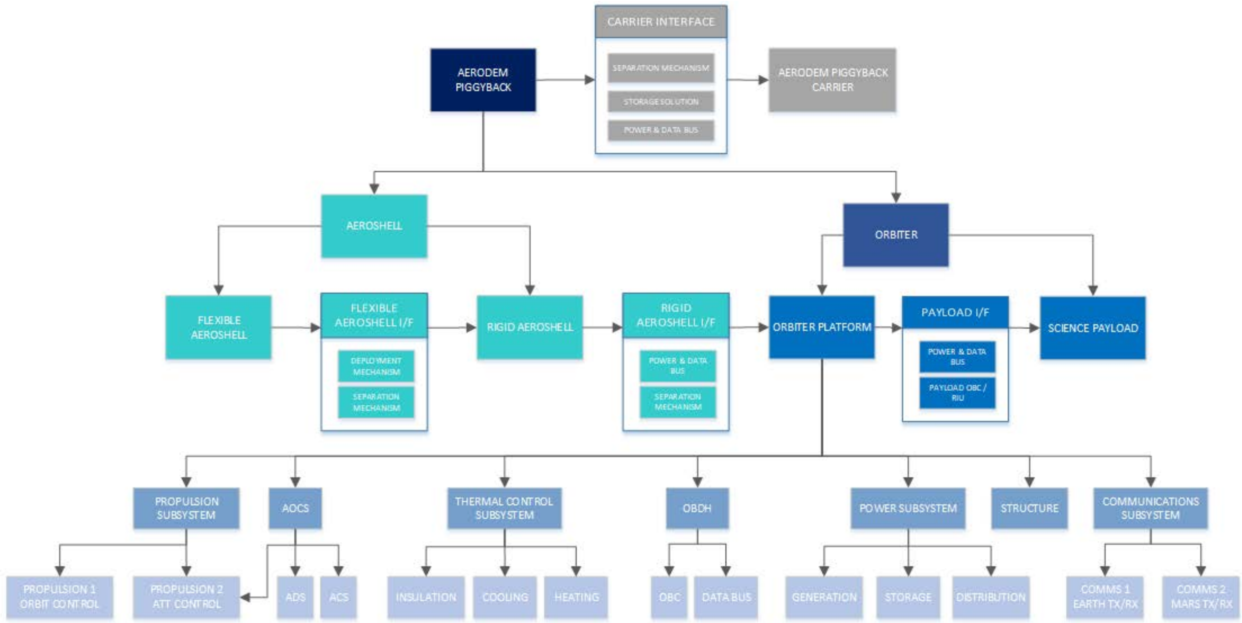


Figure 10-2: Piggyback spacecraft system structure

10.1 TRANSFER MODULE DESIGN

The Transfer Module provides support for the Orbiter during the interplanetary phase for the Standalone configuration. The final design of the Transfer Module has been driven by cost factors. The units needs have been simplified to keep the minimum number of functionalities needed in this module. The accommodation of Solar arrays and HGA were selected in vertical position during launch to reduce the loads reduce mechanical complexity.

As the Orbiter has to be deployed to go through the atmospheric leg, the Transfer Module provides it the needed power during the cruise phase. For cost reduction also, the solar arrays have been selected to be fixed, so there is no need for deployment devices.

Finally, the mechanical design of the transfer module is considered a cylindrical cone that would provide the best structural design to transfer the loads between the launcher and the Orbiter.

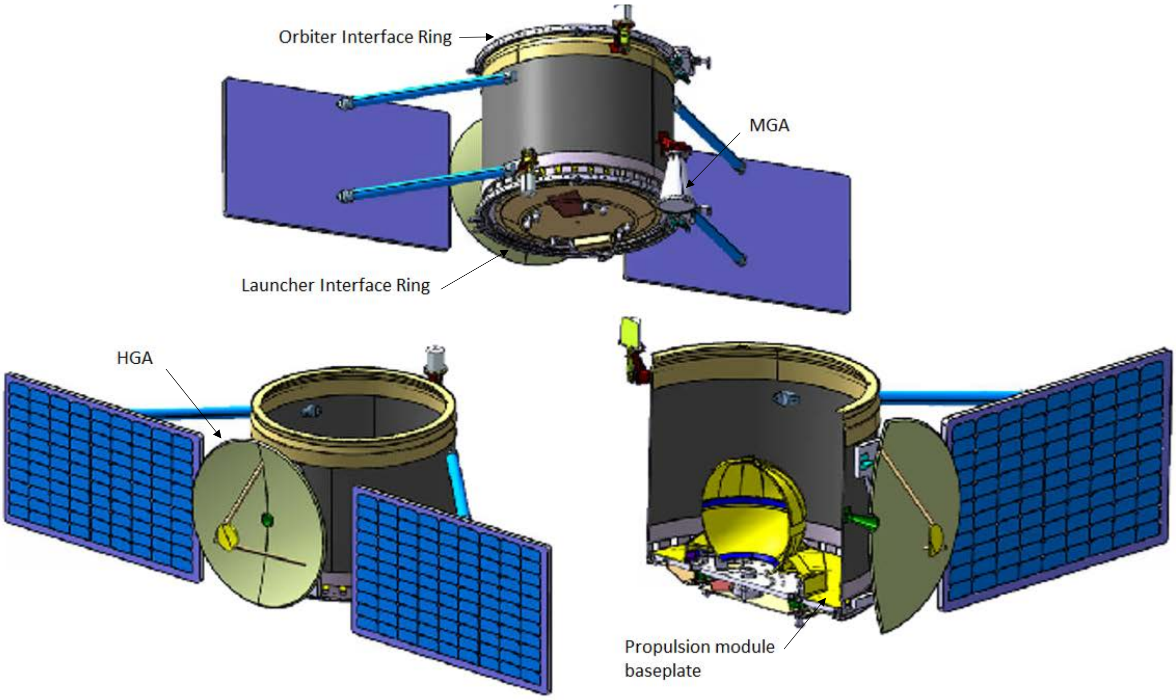


Figure 10-3 : Different views of the Transfer Module

10.2 STANDALONE ORBITER DESIGN

Most of the equipment that contribute to the Orbiter are off-the-shelf and related to high TRL.

Propulsion Subsystem

The Orbiter propulsion subsystem is a N2H4 mono-propellant system pressurized with He and operated in blow-down mode. In order to exploit commonalities, the same propulsion module is envisaged for the Transfer Module.

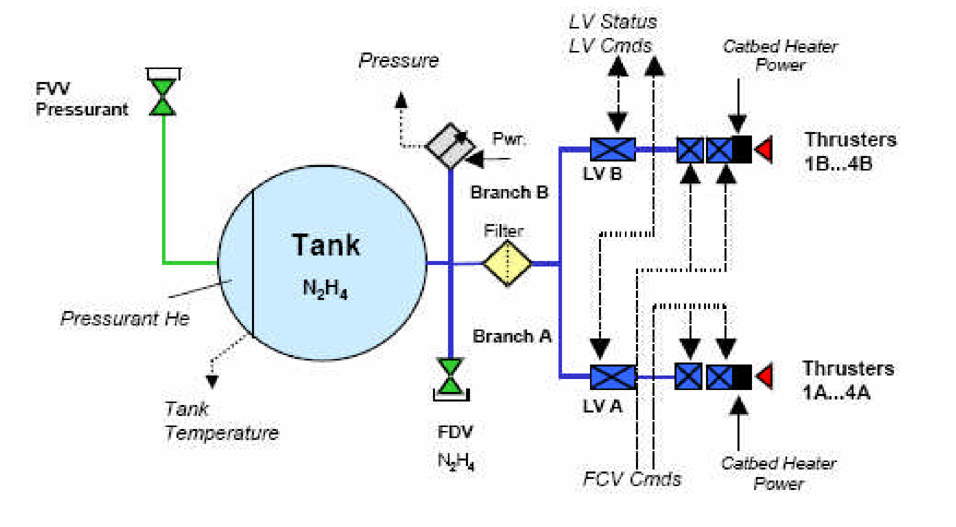


Figure 10-4 : Propulsion module architecture

Mechanical architecture

The primary structure is made of two hexagonal floors, six corner beams, six side panels and a Transfer module interface ring (LIR) compatible with 937 clamp band. This primary structure profits from strong heritage from CHEOPS satellite.

Next figure shows the layout of units in the different surfaces of the hexagonal body:

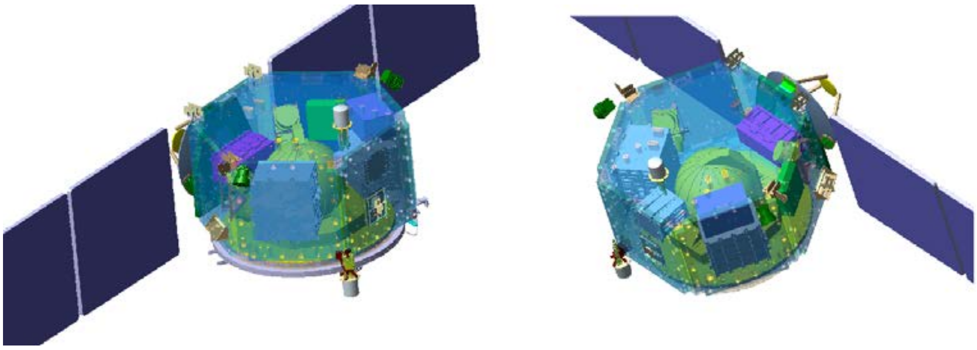


Figure 10-5 : Satellite units layout in the hexagonal body surfaces

Power Subsystem

The main equipment's of the EPS are:

- Solar Array, generating the electrical power
- PCDU, accommodating and distributing the electrical power
- Battery, storing the excess of energy during sunlight, and providing power during eclipse

The Solar Array is made of two wings with two deployable panels with a total of 8 strings with 20 serial super-size XTJ cells. These panels are reused from Astrobus-S generic platform.

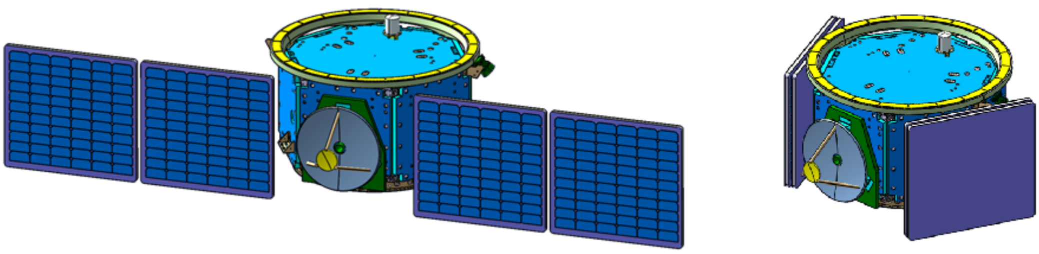


Figure 10-6 : Obiter views with solar arrays deployed (left) and folded (right)

Communications Subsystem

For communications right after launch, in the Earth´s orbit, we use two Low Gain Antennas (LGA). Considering AERODEM main mission (aerocapture demonstrator) a HGA is not strictly needed in the Orbiter but it is in the Transfer Module to provide DDOR measurements at 1 AU. To ensure the main mission of AERODEM, a MGA is also considered in the Transfer Module. The MGA is used if the satellite attitude is such that the HGA cannot point the Earth (for instance during TCMs maneuvers) during the outbound transfer and for safe mode when the pointing is not accurate enough to use the HGA. For the Orbiter and for the sake of cost reduction no Medium Gain Antennas (MGAs) are considered as backup for the HGA and for special events such as safe mode operations two LGAs will be used with omnispherical coverage.

As the cost of DSTs are quite high, the DST are only considered in the Orbiter while LGAs and MGA are duplicated in both Orbiter and Transfer Module and managed by Orbiter DST.

10.3 PIGGY-BACK ORBITER DESIGN

The preliminary architecture for the piggyback spacecraft has attempted to use commercially available components with a high TRL whenever possible, to minimize cost and maximize the reliability of mission equipment. Taking this into account, it is possible to divide the Orbiter subsystems in four main groups:

- Subsystems which can be sourced directly with COTS: Part of the AOCS/GNC, insulation and heating components of the thermal control subsystem, OBC and avionics, power storage, power distribution, data buses, and the transponders for both communication subsystems.
- Subsystems which can be sourced from COTS, but which may require some modifications: Secondary propulsion system, part of the AOCS/GNC system, antennas for the main and secondary communications system, power generation subsystem.
- Subsystems which can be sourced from COTS but will require modifications: Main propulsion system, cooling equipment for the thermal control subsystem, structure, carrier separation mechanism depending on stabilization strategy chosen for aerocapture.
- New developments: Aeroshell, Aeroshell internal interfaces, Aeroshell external interfaces including data and power links with the Orbiter platform.

10.4 AEROSHELL DESIGN

The dimensions that have been considered are given here-below:

Table 10-1: AERODEM front-shield dimensions

	Standalone	Piggy-Back
Nose Radius	1 m	0.3 m
Half cone angle	70°	70°
Maximum diameter (rigid heat-shield)	2.23 m	1.09 m
Maximum diameter w/ deployed flexible heat-shield	4.94 m	3.6 m

The layering of the rigid front-shield is currently the same on both standalone and piggyback configurations:

- a structural sandwich (made of CFRP and aluminum honeycomb)
- a RTV like bonding adhesive.
- a thermal protection material (Norcoat® Liege)

The configurations also include a fixed open rear backshell that has a thermal protection role and can provide a mechanical support for the flexible heat shield. Heat flux should be less important than the ones on the frontshield: thus Prosial TP material (Huygens-like) has been considered.

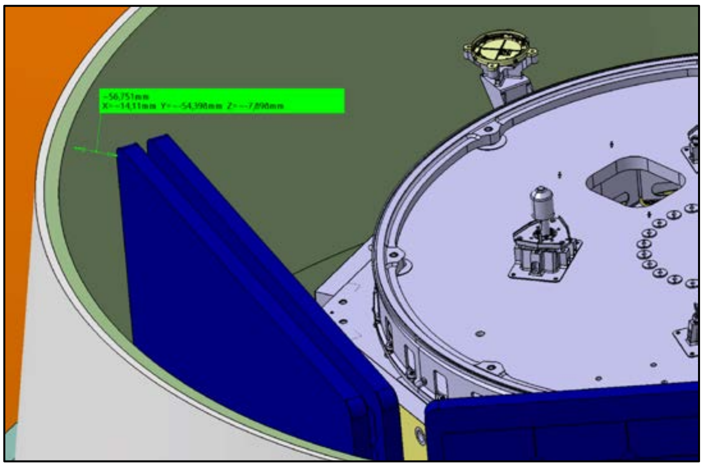


Figure 10-7: Standalone back-shell

For the flexible heat-shield, the same layering has been considered by JEHIER for both standalone and piggy-back configurations for this preliminary design. The layer thickness is also considered to be constant all along the flexible shield. The flexible shield is equipped with anti-torque system: the Kevlar structure is bounded from the torus to the middle shell.

Once the flexible heat shield is deployed, it is mandatory to preserve the aeroshape profile between the rigid and the flexible parts for aerodynamic purpose. In order not to design a backward facing step at the interfacing, an adaptation flange has been designed (Figure

10.10). The flange is currently made of aluminum, its weight is nearly 40kg for the standalone configuration. Replacing its material by a lighter one could save mass. ALM process would also offer potential mass savings.

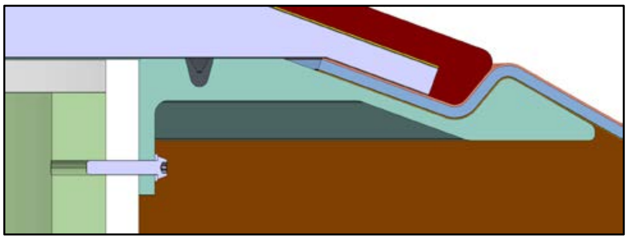
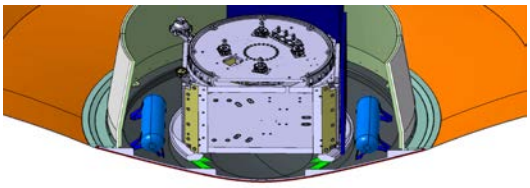
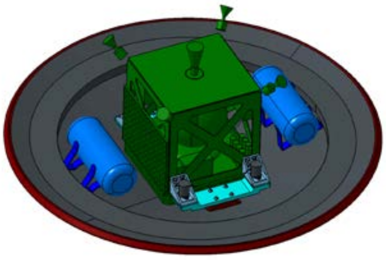


Figure 10-8: illustration of a breakable flange system

A minimum pressure has to be maintained into the torus to ensure the flexible structure not to collapse. The minimum pressure is around 5 bars for a structure without anti-torque system. Anti-torque system allows the minimum pressure to be around 2 bars. The minimum pressure to be reached in the torus is set to 5bars as a conservative approach. Two tanks configuration has been preferred for mass centering.



Standalone configuration



Piggy-Back configuration

Figure 10-9 : Pressurized tanks location

11 PROGRAMMATICS

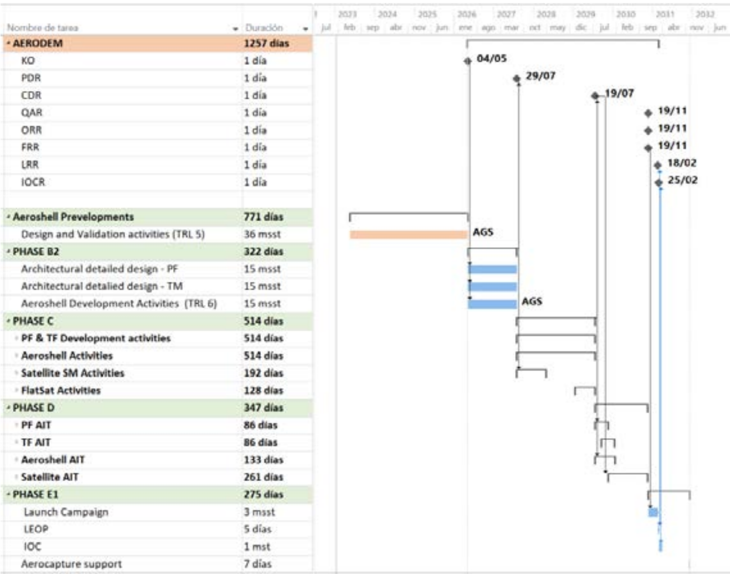
11.1 STANDALONE SOLUTION

Development activities

- It is assumed T0 in May 2026
- Before T0, it is proposed predevelopment activities running in parallel to phase A/B1 in order to achieve TRL 5 maturity for the Aeroshell inflatable elements
- From KO, a (15 months) Phase B2 is executed, finally freezing design and interfaces and concluding in the system PDR. Aeroshell foresees to achieve TRL 6 by System PDR
- Launch in 2031 appears technologically and programmatically feasible →Assuming phase A/B1 and Aeroshell predevelopment start in 2023 and further budget granted for a B2CDE1 KO in 2026.

Requirements & Cost Drivers	Compliance Assessment
Techincal Requirements (as reported in slide 3)	Solution proposed Compliant with Technical requirements for Aerocapture demo
Aeroshell development	Inflatable elements of Aeroshell are a new development Verification approach complies with ESA ECSS → models 1 SM, 1 EM, 1 PFM Predevelopments required to achieve TRL 5 by KO of phase B2CDE1 (05/2026)
Mission / Payload requirements	Assumed Payload provided as CFI. If complex Mission / Payload the platform may need to be tailored with potentially increased cost. Assumed simple mission / payload with objectives focused on Aerocapture demonstration.
Communications Subsystem	Driver for mission cost → Therefore, baseline accounts with shared elements (DST, TWTA, RF waveguide table) of the communication subsystem between Transfer Module and Orbiter. In addition, - No pointing mechanism for antennae included - Assumed UHF transceiver provided as CFI ESA and Airbus to study alternative solutions to reduce cost → potential areas: COTS components, simplified communications baseline (TBC)
Verification approach	Assumed SM activities for mechanical de-risk during phase C and SM refurbishment for PFM during phase D Next period to investigate if further simplification with verification approach directly to PFM would incur in cost savings

Risk	Description / Mitigation
Aeroshell development	Aeroshell including inflatable technology constitutes a new development in Europe, therefore the development of this technology represents the highest risk for Aerodem mission. However, mitigation has been proposed by predevelopments activities starting in 2023 to achieve TRL 5 for KO in 05/2026 and TRL 6 for PDR in 07/2027
Mission and payload definition	Mission and payload have nor been defined and platform complexity could be increased if payload with stringent requirements was foreseen for the aerocapture demonstrator
Transfer Module and Orbiter development	Considered low risk mitigated by the re-use of existent and qualified platforms and parallelization of design between both elements (Transfer Module and Orbiter) The cost driver for the platform appears to be the Communications Subsystem compatible with Deep space



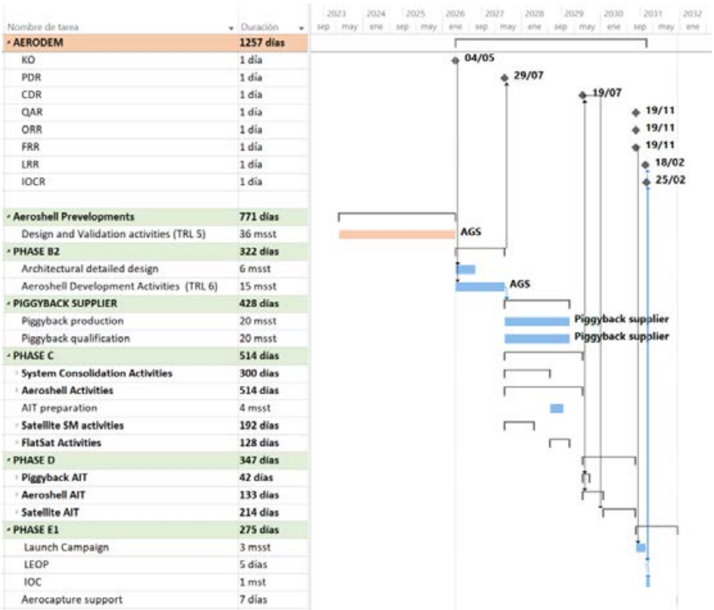
11.2 PIGGYBACK SOLUTION

Development activities

- It is assumed T0 in May 2026
- Before T0, it is proposed predevelopment activities running in parallel to phase A/B1 in order to achieve TRL 5 maturity for the Aeroshell inflatable elements
- From KO, a (15 months) Phase B2 is executed, finally freezing design and interfaces and concluding in the system PDR. Aeroshell foresees to achieve TRL 6 by System PDR
- **Launch in 2031 appears technologically and programmatically feasible →** Assuming phase A/B1 and Aeroshell predevelopment start in 2023 and further budget granted for a B2CDE1 KO in 2026.

Requirements & Cost Drivers	Compliance Assessment
Techincal Requirements (as reported in slide 3)	Solution proposed Compliant with Technical requirements
Aeroshell development	Inflatable elements of Aeroshell are a new development Verification approach complies with ESA ECSS → models 1 SM, 1 EM, 1 PFM Predevelopments required to achieve TRL 5 by KO of phase B2CDE1
Mission / Orbiter requirements	Assumed Payload provided as CFI If unforeseen elements for Orbiter interface adaptation could increase complexity and cost in future design iterations
Piggyback adaptations for aerocapture and deep space	Piggyback to communicate with orbiter during aerocapture phase and with Earth during nominal phase → Communications compatible with UHF and X-band Structure in line with aerocapture stability needs and able to fit the required tank size Need of chemical propulsion to execute Pericenter raising Manoeuvre after aerocapture
Verification approach	Assumed SM activities for mechanical de-risk during phase C and SM refurbishment for PFM during phase D Next period to investigate if further simplification with verification approach directly to PFM would incur in cost savings

Risk	Description / Mitigation
Aeroshell development	Aeroshell including inflatable technology constitutes a new development in Europe, therefore the development of this technology represents the highest risk for Aerodem mission. However, mitigation has been proposed by predevelopments activities starting in 2023 to achieve TRL 5 for KO in 05/2026 and TRL 6 for PDR in 07/2027
Mission / Orbiter definition	Mission and Orbiter have nor been defined and unforeseen elements for Orbiter interface adaptation could increase complexity in future design iterations
Piggyback development	Risk due to the development of cubesat adapted for deep space and aerocapture as adaptations of existent platforms will be required For aerocapture: specific structure compatible with stability conditions and able to fit the required tank size Propulsion subsystem needs to be compatible with deltaV needs Communication subsystem / strategy adapted to deep space



12 FLEXIBLE HEAT-SHIELD TRL DEVELOPMENT PLAN

The following table gathers the critical inflatable heat shield related technologies, their TRL associated to environments from a launch up to atmospheric entry. The efforts and a preliminary development/demonstration plan is also proposed in order to reach TRL6 for the next decennia.

Technologies	TRL	Needed development
Rigid CFRP – aluminium – Honeycomb structure	9	The structure technology in itself is well known but the qualification of the interface with the inflatable heat shield required fatigue static ground test to demonstrate compatibility with the flexible skin
Thermal protection material (Norcoat Liege)	9	As far as Norcoat Liege is used no specific effort are required
Tension shell outer fabric	3-4	Considering Alumino silicate fibers based fabric, the following effort must be envisaged: - Securing procurement and increase quality, homogeneity of the fabric for large gores. - Small scale test for panels assembly, gluing, sewing procedures - Small scale test for interfacing the fabric with the insulative material and bladder - Aging test and effect on flexibility, strength Full scale ground test (gravity assisted) for storing, inflation reduced scale flight demonstration
	2	For SIC, development plan would request a lot of effort to bring it to the aluminosilicate fiber and is therefore considered as a show stopper.
Insulative layer	4	Recent work at material level are promising and continuation of effort at a thermal system level are required: Interfacing with the bladder an outer layer, Deployment, inflation test with and without neighbor layers
Bladder	5	The tension shell solution used a unique toroidal inflated structure pressurized at relatively high pressure. - Large diameter bladder, potentially made of several circumferential compartments required a controlled leakage rate experimentally characterized - Bonding and closing technology TRL need to be risen in the context of deep space and reentry environment
Gas barrier	6	No specific effort as expected as far as high temperature thin polyimide film are well known for representative environment
Separation mechanism	3	Separation mechanism for rigid structure present obviously higher TRL, but for flexible shell and heat shield new and concept and associated development for smooth, lightweight, symmetrical and robust separation could be interesting
Pressurization system	4	High pressure spherical and spherocylindrical vessels are already employed for space environment and have already high TRL, but specific shape (toroidal) might be need for layout constraints. Flexible tubing, connector, pressure valves exist but miniaturizing effort could be also required

Table 12-1 : Critical technologies and TRL

13 CONCLUDING REMARKS

13.1 MAIN ACHIEVEMENTS

Two configurations are considered for AERODEM:

- The Standalone configuration
- The Piggy-Back configuration

For the Standalone configuration, the AERODEM Spacecraft consists in 2 different modules: The Transfer Module or Kick-Stage and the Aerocapture Module to be launched with a European launcher Ariane 62 in 2031 or 2033 according to a dedicated mission. While for the Piggy-Back, it will deal with the Aerocapture Module only; the Carrier being part of another, to-be-defined mission. The different modules and elements constituting the spacecraft for the Standalone configuration are briefly recalled hereafter. A preliminary mission timeline is also presented.

- The Transfer Module will enable the Earth escape maneuver and provides the interplanetary navigation and the trajectory corrections (TCM) during the Earth-to-Mars cruise and Mars approach, prior to the aerocapture entry phase. Typically, about 5 TCMs will be performed after launcher release. The baseline sequence for the stand-alone mission is to use the transfer module capability to perform all of the TCMs. The transfer module provides the thrust to perform attitude control maneuvers and to provide capability of low thrust for performing the TCMs. It also provides the capability for communications during the interplanetary phase. This is very important not only to provide TC capability and housekeeping data but also for providing navigation measurement for achieving the strict requirements of orbit accuracy when entering Mars atmosphere.
  During the Mars final approach and before the Orbiter jettisoning, the flexible heat-shield is deployed (see *Figure 12-2*). The subsequent disturbances are damped by the Transfer Module RCS, the SpaceCraft is 3-axis stabilized but is spun for ensuring gyroscopic stability to the Aerocapture Module prior to EIP.

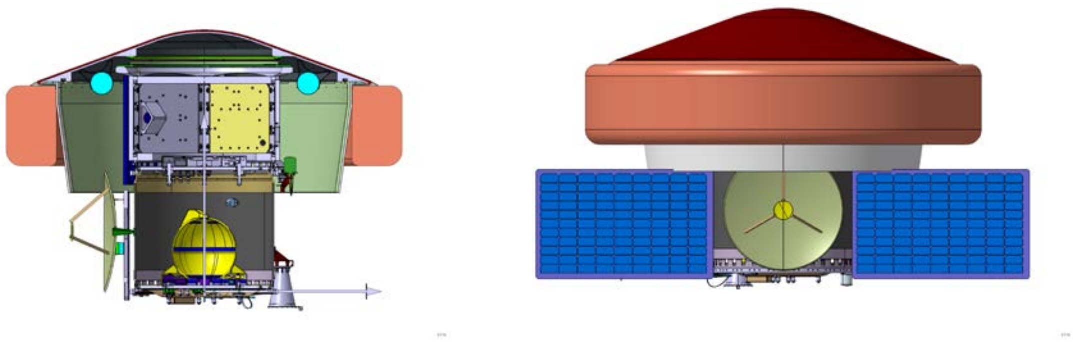


Figure 12-1 : AERODEM SpaceCraft – Configuration during Launch & Interplanetary Transfer phases

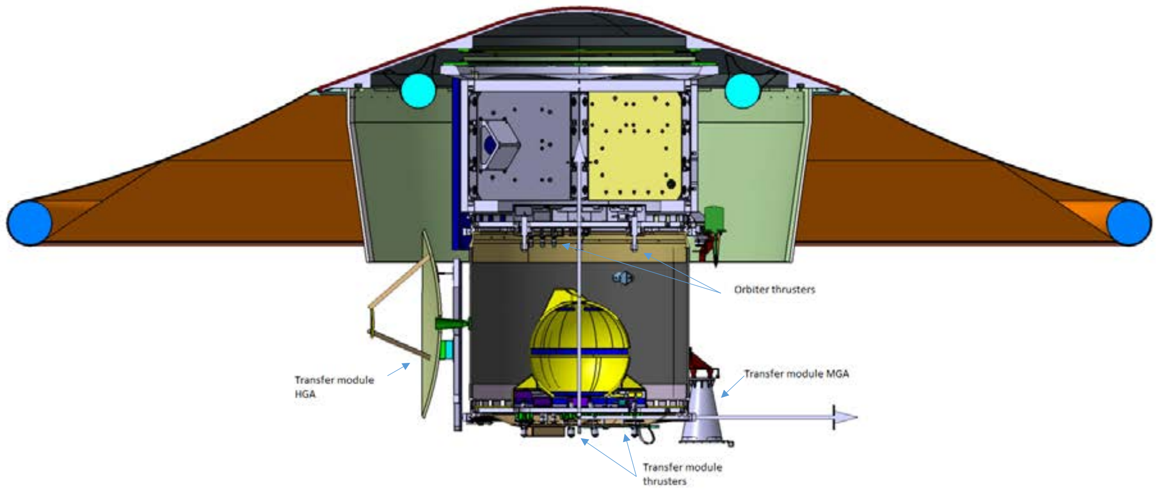


Figure 12-2 : AERODEM SpaceCraft configuration with Inflated Heat-Shield prior Aerocapture Module / Transfer Module separation

- The Aerocapture Module (AM) that enters the Mars atmosphere and realizes the drag modulation command and maneuver. Its stability is gyroscopically ensured by an initial spin rate provided by the Transfer Module before EIP and aerodynamically by the aerodynamic center of pressure backward position (positive static margin). It places the Orbiter into its capture orbit. It consists of the Mars Orbiter and the AeroShell System elements:

  - The AeroShell (AS) element that provides the atmospheric drag deceleration and modulation while ensuring aerodynamic stability and protecting the Mars Orbiter from the aero-heating loads along the atmospheric path. It carries some technology measurements sensors. The AeroShell is divided into the following sub-system elements:
    - The rigid heat-shield which consists of two parts: the front-shield and the back-shell. A back-shell has been added since the fragile equipment on the Orbiter (Solar Array, MLI, optical device,…) were finally found to be not compliant with the computed aeroheating environment; it relies on Huygens back-shell design. The rigid front-shield part relies on ExoMars front-shield design and carries the flexible heat-shield sub-system. It is jettisoned from the Orbiter at the aerocapture exit after the drag modulation instant. In order to mitigate the heat to be conducted from the heat-shield towards the Orbiter, the instant for rigid heat-shield separation will be fixed so as the Orbiter equipment thermal limits will be compliant with the remaining aeroheating environment,
    - The flexible heat-shield consists of an inflatable tension shell system of nearly 5-m diameter. It is inflated during the Mars final approach and prior to the Spacecraft spinning command and the separation of the Aerocapture Module from the Transfer Module. It is jettisoned during the atmospheric leg upon a GNC command in order to provide the required drag modulation. As long as the mission requirements on exit conditions are not too stressing, a single event drag modulation aerocapture process can be considered to reach a parking orbit (with an eccentricity below 0.05) from

hyperbolic entry conditions. This can be achieved using a simple and light guidance scheme relying on a full numerical predictor-corrector scheme whose objective is just to compute the jettisoning time of a flexible heatshield. The GNC performance assessments carried out on the standalone (and piggy-back) missions yield preliminary acceptable results for a ±0.2 deg (0.4° width) corridor at entry with correction cost roughly below 350 m/s to reach the parking orbit.

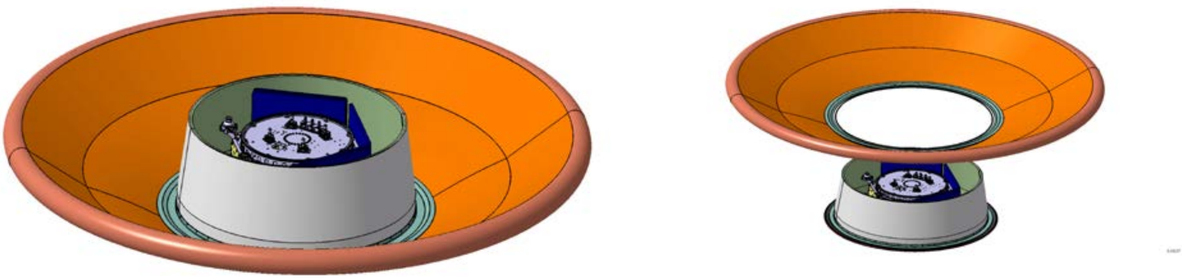


Figure 12-3 : Aerocapture Module Configuration before & after drag modulation

- o The Mars Orbiter (MO) element that carries the scientific payload and will be deployed around Mars. During the atmospheric leg, it ensures the GNC function according to an inertial navigation system where gyro-stellar estimation is used prior to the Aerocapture module spinning and separation with the Transfer module according to the interplanetary navigation system (DDOR, Star Tracker). After the atmospheric leg, once the Aeroshell has been jettisoned, and again triggered by timeline schedule, the Orbiter enters in a three-axis stabilization mode in 3 steps:
  - Angular rate (spin mode) compensation by use of gyro measurements and orbit thrusters,
  - Attitude estimation function is autonomously changed to gyro-stellar estimation when triggered by a small angular rate estimation from the gyro.
  - After stabilization, the solar arrays are deployed.

  After the atmospheric leg, once the Aeroshell has been jettisoned, it performs the maneuvers for apoapsis correction and periapsis raising for insertion into the target orbit. A total delta-V of about 195 m/s is needed to increase the periapsis value. This delta-V can be split in several maneuvers, the first one being critical to ensure a minimum value of following periapsis. The minimum considered delta-V to be provided at first apoapsis is 23m/s that would raise the initial periapsis of 60km to 150km avoiding therefore a second critical pass through the atmosphere. This maneuver lasts less than 20minutes.

  The Orbiter also records the flight and technology measurements and transmits the data to the Earth after the atmospheric leg once inserted into its final Mars Orbit. However, essential flight measurement data will be transmitted to a data relay Orbiter before and after the black-out phase. An interleaving approach can be envisaged to retransmit after the blackout data acquired during the blackout (was not done for Schiaparelli but is envisaged for ExM22/28).

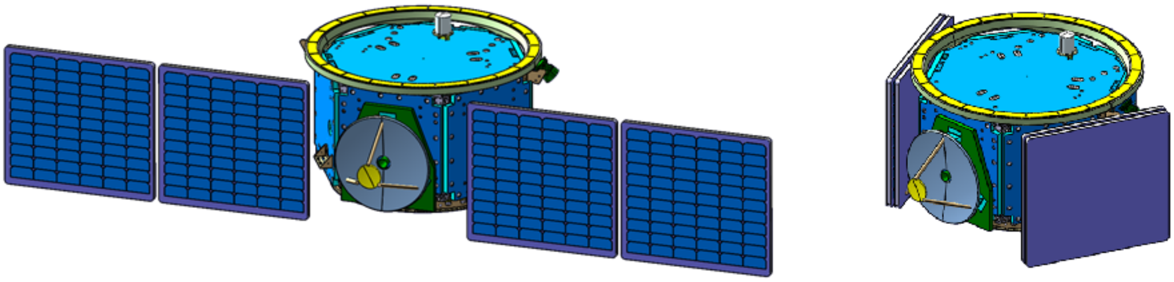


Figure 12-4 : Mars Orbiter – Unfolded and Folded configuration

Launch	06/04/2033 20:26				Baseline launch date
Satellite release	06/04/2033 21:26	60,00	Minutes	From Launch	Earth escape trajectory; delta wrt Mars injection
TCM-1	11/04/2033 20:26	5,00	Days	From Launch	Objective to modify the trajectory orbit to reach Mars
TCM-2	10/06/2033 20:26	65,00	Days	From Launch	Goal to correct the error in the execution of the first control manoeuvre.
TCM-3	12/06/2033 20:26	67,00	Days	From Launch	These manoeuvers are to fine-tune the trajectory, and should be very small if needed.
TCM-4	20/09/2033 12:52	10,00	Days	To EIP	
TCM-5	28/09/2033 12:52	2,00	Days	To EIP	
Flex shield deploy.	30/09/2033 12:36	0,01	Minutes	To EIP	The subsequent disturbances are damped by the Transfer Module RCS, the SpaceCraft is 3-axis stabilized
Spin acquisition	30/09/2033 12:37	0,01	Minutes	To EIP	0,5rad/s. Given by transfer module in 5 min
Orbiter release	30/09/2033 12:42	0,01	Minutes	To EIP	Aerocapture module release from Transfer Module
Atmospheric entry	30/09/2033 12:52	0,00	Minutes	EIP	120km altitude
Flex shield jettison	30/09/2033 12:55	1,50	Minutes	From EIP	Jettisoned during the atmospheric pass triggered by a GNC command in order to provide the required drag modulation.
Atmospheric exit & rigid heat-shield jettison	30/09/2033 13:01	8,33	Minutes	From EIP	500s
Detumbling start	30/09/2033 13:03	2,00	Minutes	From Atmos. Exit	
Detumbling achieved	30/09/2033 13:06	5,00	Minutes	From Atmos. Exit	
Solar panel deployment	30/09/2033 13:08	7,00	Minutes	From Atmos. Exit	
Catbed Heating	30/09/2033 13:10	9,17	Minutes	From Atmos. Exit	30min prior the thrust start
Thrust start	30/09/2033 13:40	39,17	Minutes	From Atmos. Exit	half duration of mano from apoapsis
Apoapsis	30/09/2033 13:50	49,17	Minutes	From Atmos. Exit	Apoapsis from periapsis 55-62 min
Thrust stop	30/09/2033 14:00	59,17	Minutes	From Atmos. Exit	20 minutes of manoevure
2nd PRM	02/10/2033 14:00	2,00	Days	From PRM	After orbit determination after PRM

Table 12-1 Mission timeline

The scalability of the system to larger missions, i-e larger orbiter is ensured both with the Standalone and the Piggy-back configurations. The key elements that must be scalable consists on the Aeroshell; to that respect we do not identify any risks on the Orbiter part. The Aeroshell consists in a rigid and a flexible part:

- For the rigid part: its maximum diameter is driven by the allocated volume under fairing. Along current launchers capabilities, maximum rigid heat-shield diameter would be about 4.5 m,
- For the flexible part: assuming the minimum beta ratio of 3.8 requirement found in the current study to be applicable. A crude estimation of the flexible heat-shield diameter for an orbiter mass of 1500 kg would be about 10 m. No showstopper has been identified so far for the selected tension shell technology.

13.2 OPEN POINTS

The standalone configuration mass budget assessment led to exceed the SOW requirements values:

- Total launch mass (all margins included): 892 kg. This is compliant with an A62 dedicated launch and a direct injection to Mars transfer orbit which is the selected launch option for AERODEM i.e. 2100 kg.

- Entry mass (all margins included): 660 kg = 252 kg (Orbiter) + 408 kg (AeroShell). This exceeds the required 400 kg. About 50 kg being due to the addition of a back-shell which is found to be required since the computed aeroheating environment (convective and radiative) is not compliant with the Orbiter Solar Array temperature limit. This has also some impacts on the achieved ballistic coefficient ratio; a minimum targeted ratio of 3.8 has to be achieved so as to ensure a minimum entry corridor FPA width of 0.46° compliant with the predicted Mars approach navigation performance. Additionally, the centre-of-gravity has shifted backward making the Aerocapture module after the drag modulation instant (flexible heat-shield is jettisoned), marginally aerodynamically unstable, so that 55 kg of ballast has been added for ensuring a positive static margin. The up-dated mass budget results in a ballistic coefficient ratio of 3.6 which leads to a degradation of the entry FPA corridor width to 0.42° (see Table 12-2 and *Figure 12-5*).

The Piggy-back configuration mass budget also leads to exceed the SOW requirement:

- Entry mass (all margins included): 188 kg = 46 kg (Orbiter) + 142 kg (AeroShell) compared to the 75 to 100 kg required mass range. Aerodynamic stability at atmosphere exit is achieved by adding 16.9 kg ballast which results in a large ballistic coefficient ratio of 5.78 (see Table 12-3 below) and an entry corridor width of 0.75° (+/- 0.375°) close to the corridor width of 0.8° as required in the SOW, while ensuring aerodynamic stability at atmosphere exit, which sounds acceptable with respect to the expected Carrier navigation performances (which are currently unknown).

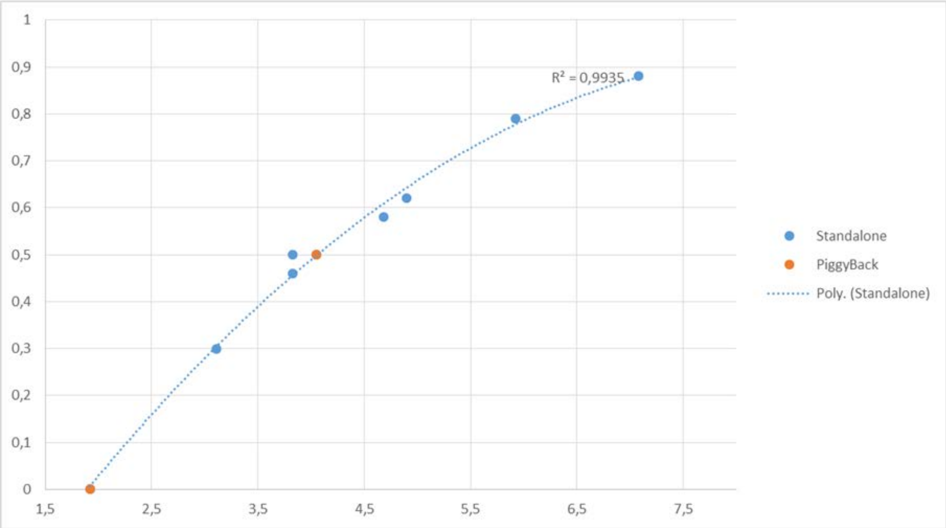


Figure 12-5 : Corridor width (°) w.r.t. β_{ratio}

	Entry	Exit
Mass [-kg]	553.17	406.36
Diameter [-m]	4.94	2.233
Sref [-m²]	19.167	3.916
Cx	1.68	1.68
Beta [-kg/m²]	17.18	61.76
Beta Ratio	-	**3.60**

Table 12-2 : StandAlone configuration

	Entry	Exit
Mass [-kg]	157.45	83.48
Diameter [-m	3.6	1.09
Sref [-m²]	10.179	0.933
Cx	1.68	1.68
Beta [-kg/m²]	9.21	53.25
Beta Ratio	-	**5.78**

Table 12-3 : Piggy-Back configuration

Further changes for both configurations need then to be examined in a next phase at the Aerocapture Module level for correcting these issues:

- Mass budget consolidation:
  - Refine the back-shell mass budget by dedicated thermal and mechanical analyses. This has a strong mass impact in particular for the standalone configuration.
  - Refine pressurization system tanks design,
  - A flange has been designed to prevent a backward facing step flow over the flexible shield. It currently deals with a significant part of

the aeroshell mass located close to the rear of the front-shield. Reconsidering this whole flange, its material or its shape may decrease the mass budget significantly,

- The intermediate plate is quite heavy as well and it has not currently a real function since the pressurization system has been removed from the front-shield dome; available volume was not large enough for the targeted pressurization system.

o Aerodynamic stability improvement: an optimization sizing loop has to be conducted between an additional ballast mass and a rigid front-shield diameter increase. The optimization has also to account for the necessity to increase the flexible heat-shield diameter for achieving the minimum ballistic coefficient ratio. In order to increase the aerodynamic stability avoiding addition of ballast mass or increasing the front-shield diameter, a dedicated design of the orbiter structure aligned with the front shield design is recommended in terms of having the CoM as front as possible. This coordinated design could focus on an orbiter design that maximizes the available volume close to the front shield or even a more aggressive solution, using the front-shield-structure to directly support the orbiter units. The drawback of this solution is the reduction of platform heritage.