EXECUTIVE SUMMARY REPORT

1 INTRODUCTION

This is a summary of the main findings from the ESA *Aerocapture Demonstration Mission* (ADAM) contract.

An interplanetary orbital probe must decelerate from a hyperbolic interplanetary trajectory to an orbital trajectory when it arrives at its destination. Currently, this manoeuvre is achieved using chemical propulsion, which means that propellant makes up a significant fraction of the vehicle mass. Aerocapture is an alternative concept that instead uses aerodynamic drag from the planet's atmosphere to decelerate the arriving spacecraft.

So far, no mission has used aerocapture, mainly because of the lack of heritage. In this ESA study at Pre-Phase A level, concepts were developed for a Mars aerocapture demonstration mission. This would place a scientific payload in a 70° inclination, 1000 km circular orbit using single-event drag modulation aerocapture. Such a mission would increase the aerocapture TRL and heritage, enabling its consideration for future missions.

This summary report describes the analysis method, mission concepts and lessons learnt.

2 OVERVIEW OF THE AEROCAPTURE CONCEPT

In single-event drag modulation aerocapture (Figure 1), the vehicle enters the target atmosphere with a pre-deployed decelerator. Atmospheric drag decelerates the vehicle. Part-way through the atmospheric pass, the decelerator is jettisoned (or retracted) to increase the ballistic coefficient. This single-event change in ballistic coefficient gives the vehicle some control authority over its final orbital apoapsis. After the vehicle exits the atmosphere, it performs a periapsis raise manoeuvre to avoid a second atmospheric pass. The vehicle then establishes communications with Earth and manoeuvres to the target orbit.

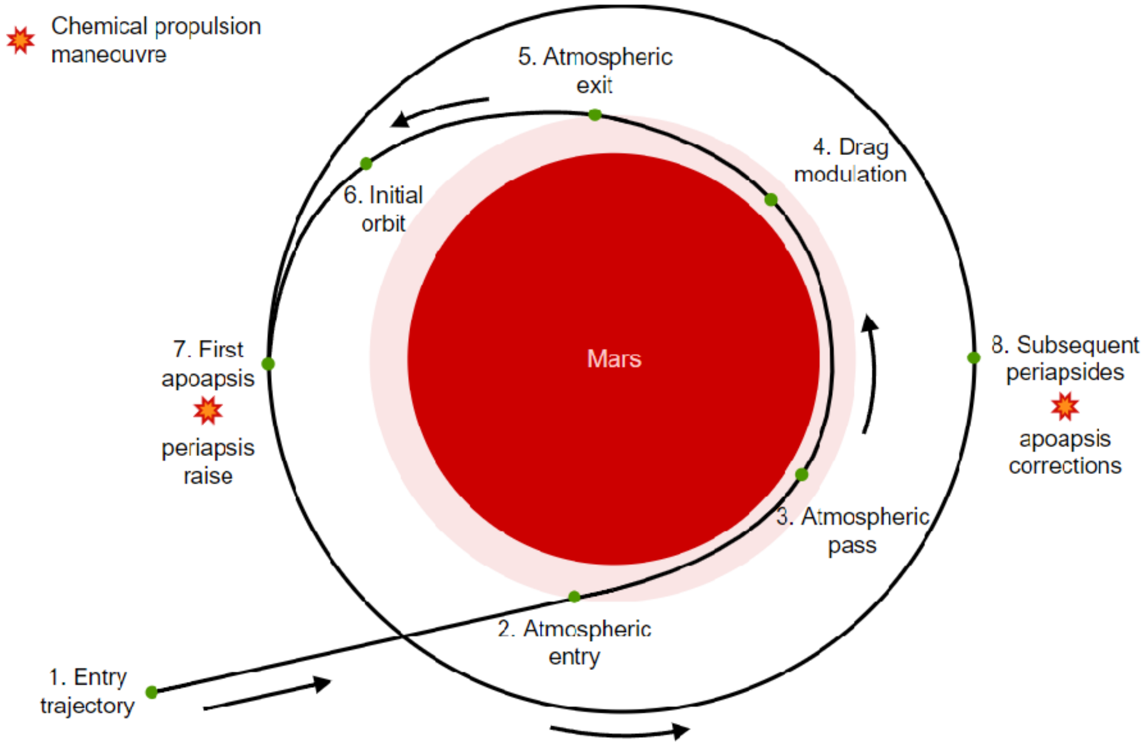

Figure 1: Mars aerocapture operational sequence

Two potential missions were considered: a piggy-back and a standalone, with masses of 90 kg and 400 kg respectively at atmospheric entry. The piggy-back mission is intended as a rapid demonstration of basic aerocapture technology that travels to Mars as a piggybacked onto a host mission. The standalone mission is a more advanced stepping-stone to larger missions. The designs are compatible with a V_{∞} of 3 km/s to 4 km/s by varying the entry flight path angle. The key characteristics of the two missions are detailed in Table 1.

Quantity	Piggy-back mission	Standalone mission	Notes
Entry mass	75-100 kg (90 kg at Mars Entry)	400 kg	Includes mass of separation mechanism remaining on carrier for piggy-back mission
Diameter in launch configuration	< 1.6 m	<2.6 m	Diameters can be increased, if necessary.
Goal science payload in Mars orbit	6 kg	≥20 kg	
V_∞ Hyperbolic excess velocity	3.0 – 4.0 km/s		
Target operational orbit	Eccentricity: <0.05 Periapsis altitude: 500-1000 km Inclination: >70° Non sun-synchronous		

Table 1: Mission characteristics (AD-02)

Before atmospheric entry, the spacecraft ballistic coefficient is minimised by increasing the drag area. This could be achieved by a deployable, flexible, or rigid decelerator (Figure 2, left). During the atmospheric pass, the spacecraft loses energy through aerodynamic drag. When the desired amount of energy has been lost, the decelerator is either stowed or jettisoned, leaving a much smaller heat shield protecting the satellite for the remainder of the atmospheric pass (Figure 2, right). This reduces the drag area, increasing the spacecraft ballistic coefficient, ensuring that the rest of the atmospheric pass is completed without excessive energy loss.

Aerocapture does not eliminate propulsive manoeuvres (1), (2). Nevertheless, the necessary ΔV is significantly lower than that of pure propulsive orbital injection (3), on the order of 200 m/s.

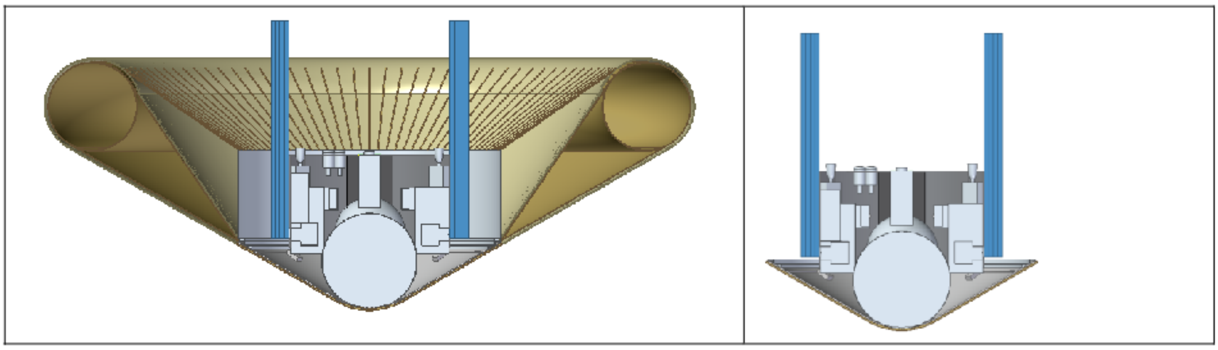

Figure 2: Decelerator concept in initial low ballistic coefficient configuration (left) and high ballistic coefficient configuration (right)

3 Mission Architecture concepts

For a given mission without variability or uncertainty in vehicle performance, trajectory or atmosphere, a single constant ballistic coefficient would achieve the target apoapsis without the need for drag modulation or any other form of control. However, uncertainties in entry parameters, atmospheric conditions and vehicle properties make this impossible. The vehicle must be able to modulate its lift/drag autonomously to compensate for these uncertainties and achieve orbital insertion as close as possible to the final targeted orbit.

Single-event drag modulation accomplishes this by jettisoning (or retracting) the decelerator during the atmospheric pass. The vehicle Guidance Navigation and Control (GNC) system chooses when to jettison and switch from the initial low ballistic coefficient to final high ballistic coefficient. The ratio of final to initial ballistic coefficient is a measure of the degree of control authority.

Constant ballistic coefficient trajectory simulations were conducted to determine the variation in required overall ballistic coefficient with atmosphere and entry flight path angle. Both GRAM v1.4 and MCD v5.3 Mars atmosphere models were used. A design tool was then developed to determine the ballistic coefficient ratio necessary to compensate for the various uncertainty sources.

Two launchers were assessed for the stand-alone mission: the Ariane 6 and Vega C. The Ariane has good capability for the mission while the Vega has insufficient capability to deliver the targeted standalone spacecraft mass of 400 kg at Mars arrival.

The piggy-back mission will use a host mission for interplanetary transfer.

A wide range of pre-deployed, deployable and inflatable decelerators was reviewed. These cover concepts from the Europe, the USA and elsewhere and encompass concepts with a wide range of existing TRLs.

The cruise functions required for spacecraft operation during the interplanetary transfer were assessed. Unlike an orbiter employing a traditional propulsive orbit insertion manoeuvre, the orbiter cannot survive aerocapture with unprotected appendages. Some degree of duplication is thus necessary using either a dedicated cruise stage or resources from a host mission.

Detailed concept designs were developed to deliver the required ballistic coefficient ratio. Valispace, a model-based systems engineering (MBSE) software tool, was used to generate decelerator and vehicle mass models. This tool semi-automatically produced vehicle and decelerator sizing from a given entry mass, arrival velocity and entry accuracy. The design parameters of all the subsystems, including decelerator, aeroshell and orbiter, were automatically recalculated after changing input parameters. This allowed rapid calculation of multiple vehicle designs. The orbiter mass for the candidate designs was maximised, while maintaining vehicle stability.

Aerocapture trajectories were analysed. The main causes of dispersion were identified as arrival trajectory uncertainty, atmosphere uncertainty and the ability of the controller to correct for these.

Trajectory analyses were conducted for each system design to determine apoapsis accuracy assuming a given aerocapture controller algorithm and to calculate the peak acceleration. Similar effects were seen with both GRAM and EMCD atmosphere models. It was found that the density profile on the outbound leg is different to that on the inbound leg due to the change in latitude as a consequence of the high inclination entry (Figure 3).

The GNC controller performance must be robust against density perturbations that vary with altitude and distance, a constant scale factor cannot be assumed.

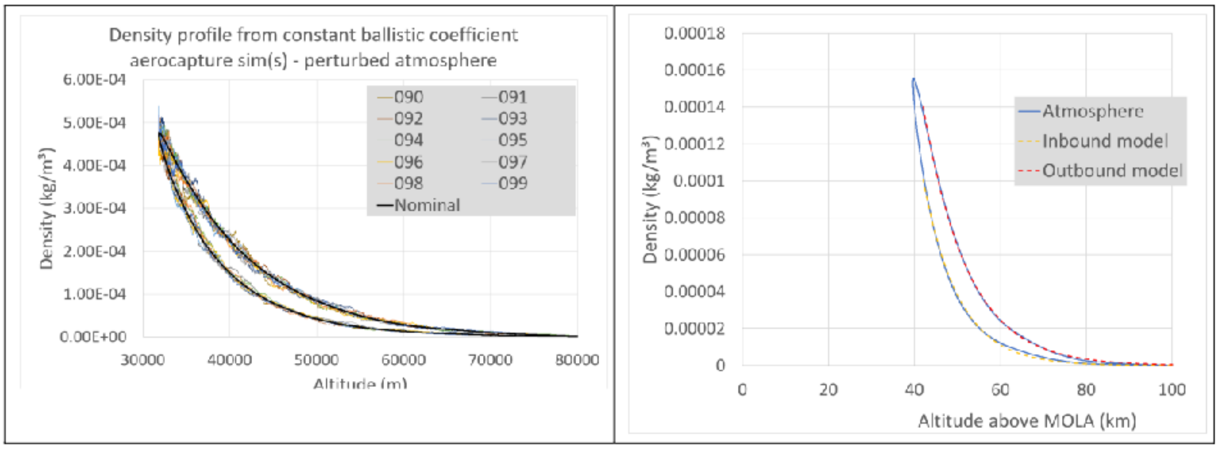

Figure 3: Density perturbations (left) and density model approach (right)

4 CONCEPTUAL SYSTEM DESIGN

4.1 Piggyback conceptual design

The piggyback mission uses a decelerator with a 60° sphere-cone front face made of aluminium honeycomb protected by a Thermal Protection System (TPS). This mission architecture was selected in TN-2 (4).

The vehicle and jettison sequence are shown in Figure 4. The overall aerocapture system sizing is presented in Table 2.

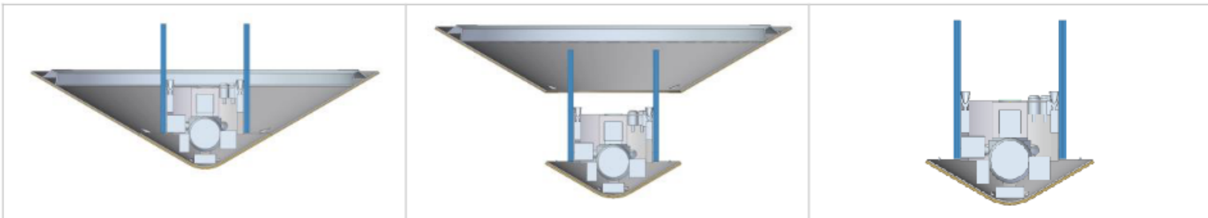

Figure 4: Rigid fixed decelerator architecture and sequence

Parameter	Value
Geometry	60° sphere-cone
Final ballistic coefficient (kg/m²)	89
Initial ballistic coefficient (kg/m²)	20
Ballistic coefficient ratio	4.44
Targeting accuracy (°)	0.2
Controller allowance (°)	0.2
Entry corridor ($V_\infty = 4$ km/s) (°)	-11.24 / -11.44
Entry corridor ($V_\infty = 3$ km/s) (°)	-10.74 / -10.54
Initial diameter (m)	1.93
Final diameter (m)	0.734
Nose radius (m)	0.184
Entry mass (kg)	72.6 (74.9)
System mass (with margin)	90
Decelerator mass (with margin) (kg)	15.0 (16.8)
Core vehicle mass allocation (kg)	58.2
Aeroshell mass (with margin) (kg)	57.6 (58.1)
Orbiter mass allocation (kg)	53.9

Table 2: Key parameters for the piggyback aerocapture mission configuration

The decelerator concept selected for the piggyback mission was a cold structure rigid fixed decelerator (4). This is composed of a set of rigid panels covered by a rigid TPS. The panels are manufactured from a sandwich structure made up of a core and two skins. This was analysed using finite element modelling (FEM) to understand the deformation modes and determine whether an additional stiffener was required towards the circumference (Figure 5). This stiffener was added to the design as shown inFigure 6.

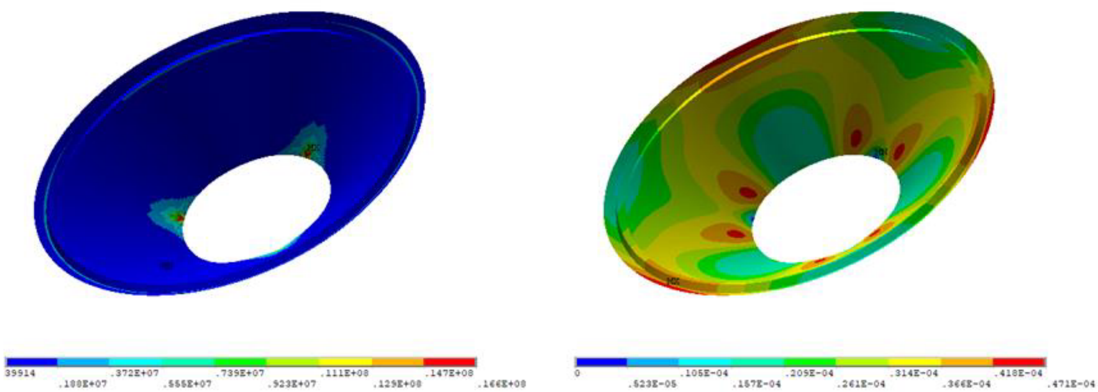

Figure 5: FEM results decelerator structure: left stress in Pa, right deformation in m

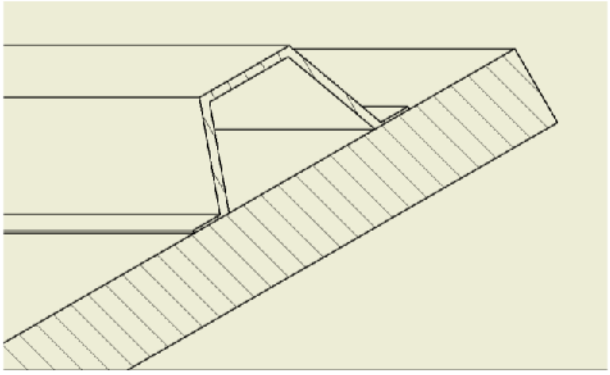

Figure 6: Local reinforcement

The decelerator and aeroshell are protected by Amorim P45 cork-phenolic ablator. It is important to use an ablator with low thermal conductivity such as cork-phenolic to minimise thermal conduction.

4.2 Standalone conceptual design

The standalone mission uses a two-pressure-volume decelerator architecture where the jettisonable decelerator consists of an inflatable structure protected by a flexible TPS layup. This mission architecture was selected in TN-2 (4).

A conceptual outline of the aerocapture sequence is shown in Figure 7.

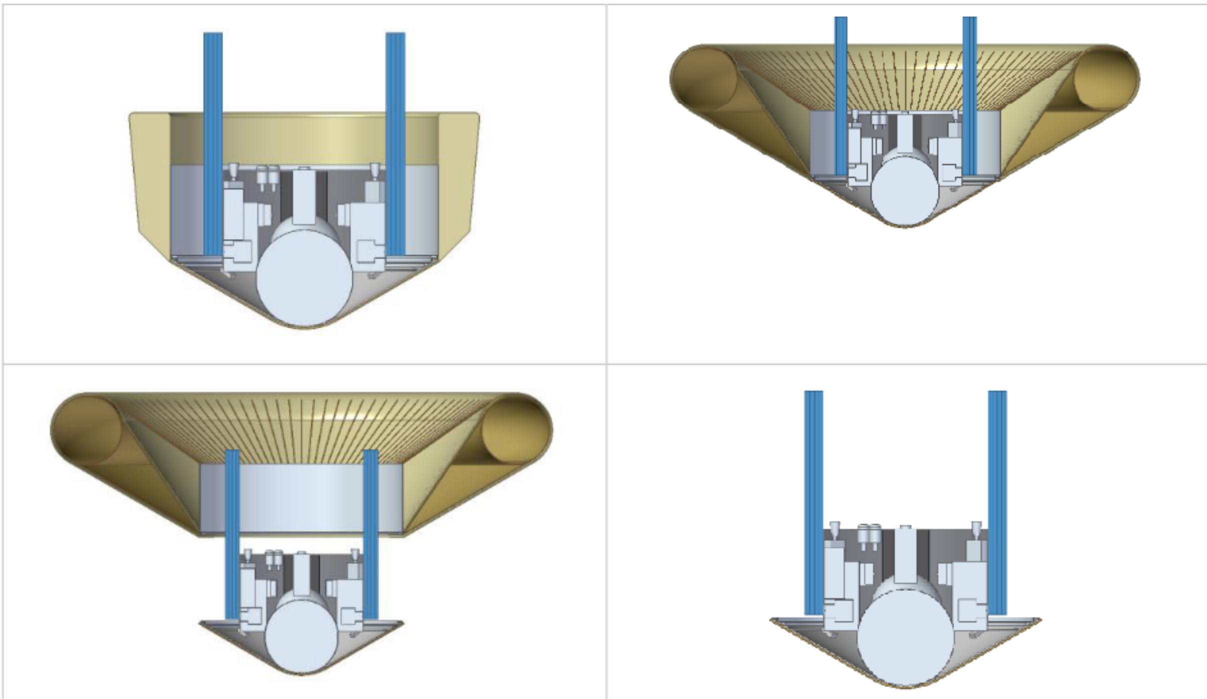

Figure 7: Two-volume decelerator architecture and sequence. Top left: Aerocapture vehicle with decelerator stowed. Top right: Decelerator inflated. Bottom left: Decelerator jettison. Bottom right: Core aerocapture vehicle.

The overall aerocapture system sizing is presented in Table 3.

Parameter	Value
Geometry	60° sphere-cone
Final ballistic coefficient (kg/m²)	157
Initial ballistic coefficient (kg/m²)	40.4
Ballistic coefficient ratio	3.89
Targeting accuracy (°)	0.2
Controller allowance (°)	0.2
Entry corridor ($V_\infty = 4$ km/s) (°)	-11.55 / -11.75
Entry corridor ($V_\infty = 3$ km/s) (°)	-10.84 / -11.04
Initial diameter (m)	2.86
Final diameter (m)	1.17
Nose radius (m)	0.293
Entry mass (kg)	400
System mass (with margin)	315 (333)
Decelerator mass (with margin) (kg)	60.1 (72.1)
Core vehicle mass allocation (kg)	261
Aeroshell mass (with margin) (kg)	29.6 (35.9)
Orbiter mass allocation (kg)	225

Table 3: Key parameters for the standalone aerocapture mission configuration

The mission uses a two-pressure volume inflatable structure protected by flexible TPS. It has two separate inflatable components: an annular torus and a conical volume as presented in Figure 8. These parts are inflated at different pressures. The annular torus (1) is the key inflatable element that gives shape to the decelerator. For increased stability and rigidity of the decelerator with respect to the satellite, avoiding relative displacement between the decelerator and the spacecraft, the annular torus is connected to the satellite main body through radial tendons. The conical volume (2) inflatable section is defined to mitigate the creation of concave shapes during the atmospheric flight, avoiding localized heating in the inflatable structure. More detail is shown in Figure 9.

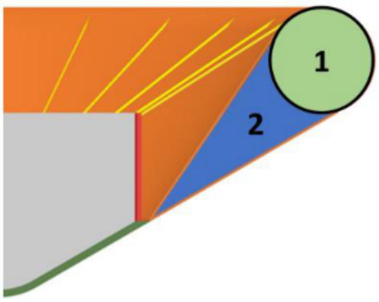

Figure 8: Inflatable volumes

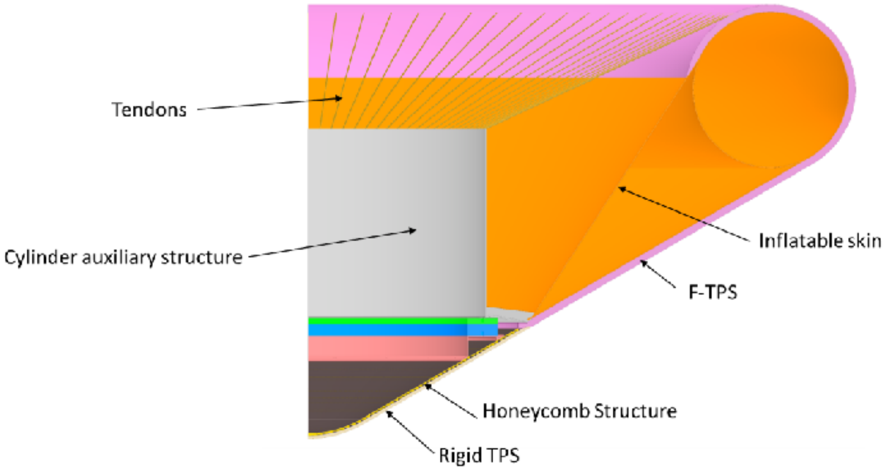

Figure 9: Inflatable decelerator and aeroshell overview

The flexible TPS protects the inflatable structure from the high temperatures and heat fluxes that the vehicle is exposed to during entry. This is a lay-up of three different materials. The outer layer of silicon carbide (SiC) satin cloth provides protection from the hot gasses during re-entry, the middle layer of Sigratherm provides insulation and the inner layer of silicone coated Kevlar is a gas barrier which separates the atmosphere and any ablation products from the inflatable structure (Figure 10).

.

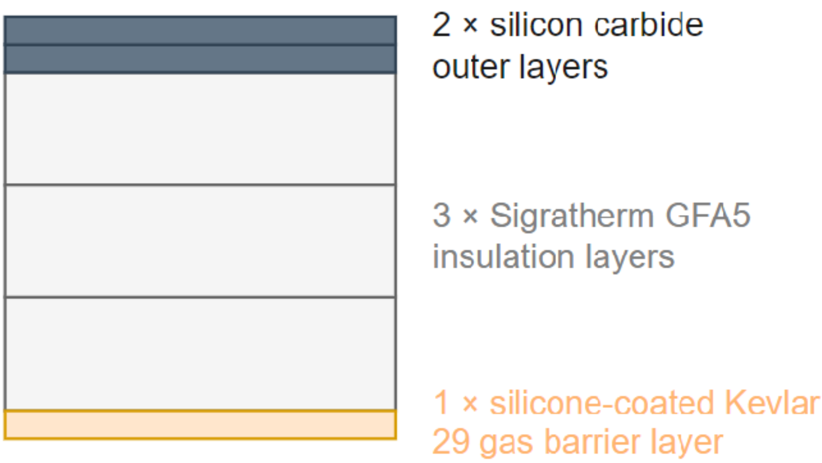

Figure 10: Flexible TPS layup

4.3 RoadMap

4.3.1 Introduction

The roadmap development approach covers ground testing and qualification for the development of the technology elements that support the design and verification activities.

4.3.2 Piggyback mission

A roadmap for the piggyback mission is shown in Figure 11.

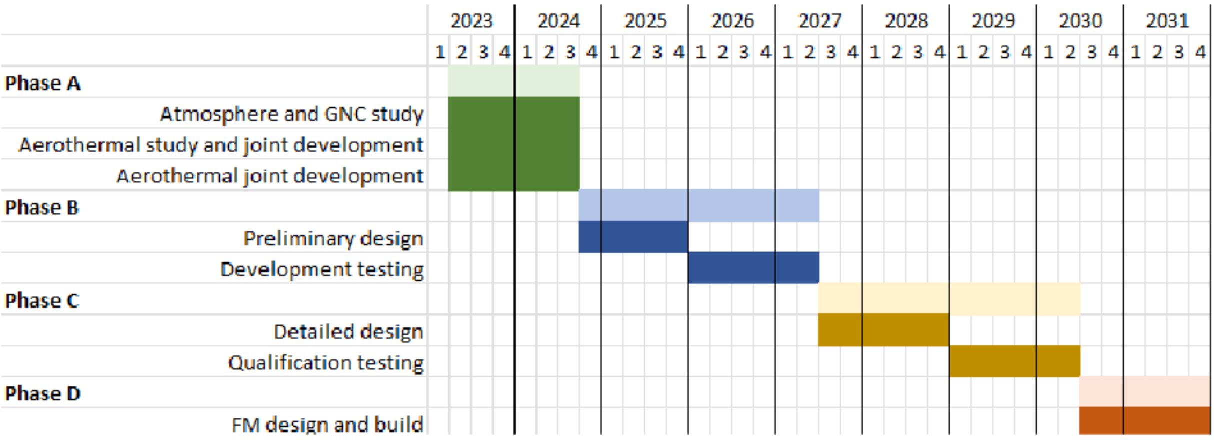

Figure 11: piggyback mission roadmap

The intent for the piggyback mission is to have the simplest possible Mars aerocapture demonstration mission with a target launch window of 2031.

For this mission, the only critical technologies that require further developments to reach TRL 5 are the atmosphere modelling approach, the GNC system and the joint between the jettisonable panels and retained aeroshell. Further aerothermal studies to better understand the thermal environment on the aft of the decelerator and the aft and sides of the retained aeroshell will also be worthwhile as part of Phase A.

For the piggyback mission it is necessary to identify the host mission early in the development, as it will influence some aspects of the mission, the most important ones being interfaces, available volume and mass budget. This will ensure that the piggyback can be successfully incorporated into the mission and will help to prevent any potential delays or issues that could arise if the selection is made at a later stage.

The host mission should be ideally identified by Phase B.

4.3.3 Standalone mission

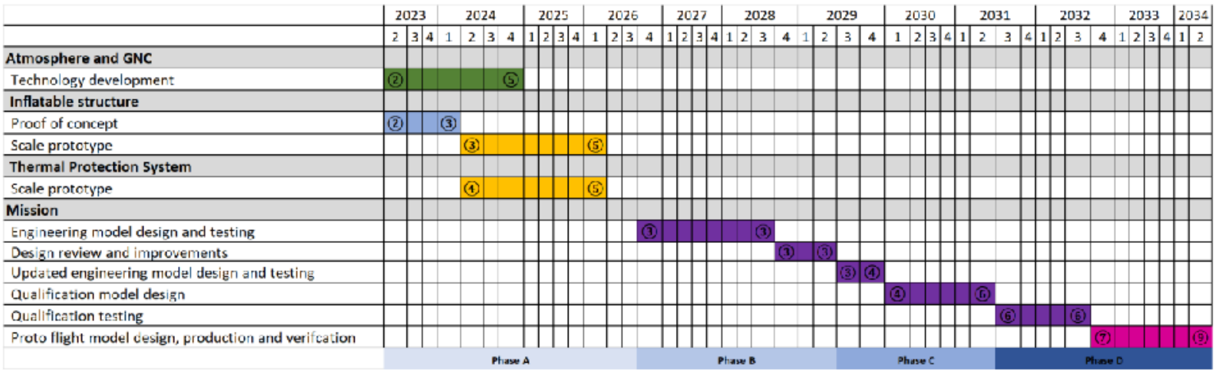

Figure 12: Roadmap standalone mission

The intent for the standalone mission is to develop a scalable aerocapture Mars aerocapture demonstration system with a target launch window of 2035.

The initial activities aim to develop the critical technologies to TRL 5. These are an atmosphere study and GNC development, the inflatable structure (including its jettison mechanism), the flexible TPS and the thermal interface between the rigid aeroshell and the jettisonable flexible TPS and inflatable structure.

The inflatable structure and the F-TPS have many shared interfaces and are highly interrelated regarding temperature and packing requirements requiring that the TRL of these technologies grow hand in hand. For the inflatable structure it is proposed to perform 2 tasks; a task to develop an initial proof of concept and a subsequent task to develop the scale prototype. In general, the focus is to demonstrate that the inflatable and flexible technologies are feasible and sufficiently mature to be considered on a mission.

5 Commercialisation

The commercialization analysis was divided into three parts. The first part examined the launch commercialization process, including the steps to acquire a launch service, the typical timeline, and key milestones. The goal was to provide a clear understanding of the process and highlight important activities and critical factors that may impact mission design. In the second part, the business case of Mars aerocapture mission was explored, providing a detailed explanation of the operating principle of the technology, discussing its advantages and disadvantages when compared to other alternatives and competitors in the market. A market assessment was conducted to understand the commercial potential of the aerocapture mission by identifying potential clients for the technology and estimating the required investments for the mission. The last part of the commercialization analysis focussed on identifying other areas where the technology development can be applied and exploring alternative business models.

6 Conclusion

Aerocapture is only useful if it can deliver a higher orbiter mass than existing chemical propulsion techniques. It is critical to maximise useful orbiter mass during the design process. The main driver is the required ballistic coefficient ratio: larger ratios require larger, heavier decelerators. The required ballistic coefficient ratio can be reduced by minimising the variabilities/uncertainties which drive the need for control: entry parameters, atmospheric conditions, vehicle conditions and GNC performance.

In particular, the required decelerator size is strongly driven by the assumed atmospheric model. Accurate atmospheric modelling, common standards and full understanding of model uncertainty are essential.

Provided that uncertainties can be reduced sufficiently, this study showed that the relatively simple single-event drag modulation aerocapture could deliver increased payload masses into Mars orbit.

7 REFERENCES

1. *Drag-Modulation Flight-Control System Options for Planetary Aerocapture.* **Putnam, Zachary R and Braun, Robert D.** 1, s.l. : American Institute of Aeronautics and Astronautics, 2014, Journal of Spacecraft and Rockets, Vol. 51. https://doi.org/10.2514/1.A32589.

2. *Drag modulation flight control for aerocapture.* **Putnam, Zachary R, Clark, Ian G and Braun, Robert D.** s.l. : IEEE, 2012. 2012 IEEE Aerospace Conference. pp. 1-10. https://doi.org/10.1109/AERO.2012.6186999.

3. *Assessment of Aerocapture for Orbit Insertion of Small Satellites at Mars.* **Falcone, G, Williams, J W and Putnam, Z R.** 6, s.l. : American Institute of Aeronautics and Astronautics, 2019, Journal of Spacecraft and Rockets, Vol. 56. https://doi.org/10.2514/1.A34444.

4. **Overend, S.J.** *Architecture trades and reference architectures selection.* Chalgrove, UK : Vorticity Ltd., 2022. VOR-TN-2210.

5. *Mission Design and Performance Analysis of a Smallsat Aerocapture Flight Test.* **Werner, Michael S and Braun, Robert D.** 6, s.l. : American Institute of Aeronautics and Astronautics, 2019, Journal of Spacecraft and Rockets, Vol. 56. https://doi.org/10.2514/1.A33997.

6. **Overend, S.J.** *Mission Architecture Concepts.* Chalgrove, UK : Vorticity Ltd,, 2022. VOR-TN-2208.