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**AEROFast: AEROcapture for Future
spAce tranSportation**

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AEROFast: AEROCapture for Future space tranSPorTation

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ABSTRACT

The aerocapture is a new technology for Solar System exploration that uses a single pass through a planetary atmosphere to decelerate the spacecraft and achieve a targeted orbit. Its main interest is that such manoeuvre saves a significant amount of mass with regard to a more conventional technique of insertion using propelled braking. An important step to allow for human expansion into the solar system is to develop advanced transportation systems to transport humans and cargo between GEO and LEO, and also returning them from the Moon or from Mars. Typically such a vehicle could rely on aerocapture to be mass effective: using atmospheric drag to slow space vehicles is regarded as one of the largest contributors to making Martian missions affordable.

Aerocapture on Mars is for the time being not currently studied by the space agencies. This is the reason why the AEROFast project is granted in the frame of the seventh European Community Framework Programme (FP7) first call, as a typical research and development project. Its aim is to prepare for such a mission demonstration and increase the TRL (Technology Readiness Level). Aerocapture technology is at TRL 2 to 3 in Europe. In order to use the aerocapture technology for operational missions while mitigating future development risks, the TRL of such a technology must reach the level 6. TRL 6 shall be reached through a flight demonstration study preferably performed with the Martian conditions in order to prepare for the future Mars Sample Return missions (first potential missions to use the aerocapture technology).

Because of the high heat flux, aerocapture requires a thermal protection system to shield the spacecraft from aerodynamic heating, as well as the use of a guidance system to assure that the spacecraft leaves the planetary atmosphere on the correct trajectory. The aim of this paper is to present the project and provide a status with respect to the different objectives of the project:

- OBJ1: Define a project of aerocapture demonstration (planet to be assessed).
- OBJ2: Make a significant progress in space transportation by increasing the TRL of the planetary relative navigation and the aerocapture algorithm up to 5.
- OBJ3: Build a breadboard to test in real time the pre-aerocapture and aerocapture GNC algorithms.
- OBJ4: Demonstrate/prototype the thermal protection system for such a mission.
- OBJ5: Define on-board instrumentation for aerocapture phase recovery.

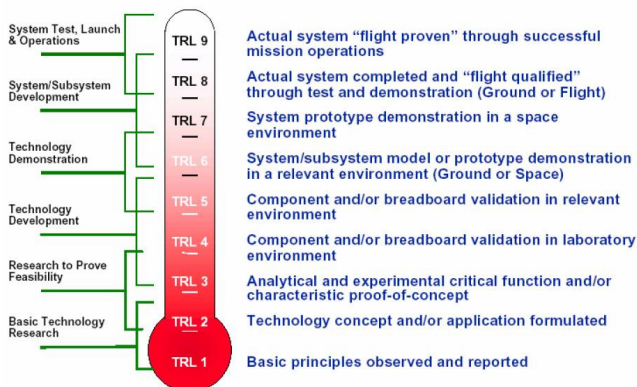
INTRODUCTION

The aerocapture is a new technology for Solar System exploration that uses a single pass through a planetary atmosphere to decelerate the spacecraft and achieve a targeted orbit. Its main interest is that such manoeuvre saves a significant amount of mass with regard to a more conventional technique of insertion using propelled braking. An important step to allow for human expansion into the solar system is to develop advanced transportation systems to transport humans and cargo between GEO and LEO, and also returning them from the Moon or from Mars. Typically such a vehicle could rely on aerocapture to be mass effective: using atmospheric drag to slow space vehicles is regarded as one of the largest contributors to making Martian missions affordable.

Aerocapture on Mars is for the time being not currently studied by the space agencies. AEROFAST project is granted by the European Community in the frame of the seventh European Community Framework Programme (FP7) first call. The industrial consortium relies on ASTRIUM SAS as coordinator and eleven partners (the list is given at the end of the paper).

AEROFAST is proposed in the frame of the “Space Transportation” theme as a typical research and development project in order to prepare for such a mission demonstration and to increase the TRL (Technology Readiness Level)

Aerocapture technology is at TRL 2 to 3 in Europe. In order to use the aerocapture technology for operational missions while mitigating future development risks, the TRL of such a technology must reach the level 6.



Technology Readiness Levels (TRL)

AEROFAST mission baseline is an aerocapture manoeuvre on Mars. The aim is to increase the TRL up to 3 – 4 and especially focus on the following aspects:

- Aerothermodynamic and thermal topics:
 - o Increase our knowledge on aerothermodynamics and flow behaviour,
 - o Improve and better assess the effect of ablation with respect to the centre of gravity,
 - o Demonstrate/prototype the thermal protection system for such a mission.
- Navigation aspects: Make a significant progress in space transportation by increasing the TRL of the planetary relative navigation and the aerocapture algorithm up to 5.

TRL 6 shall be reached through a flight demonstration study in order to prepare for the future Mars Sample Return missions (first potential missions to use the aerocapture technology).

This paper intends to present the AEROFAST project framework ... and the next steps.

JUSTIFICATION FOR A DEMONSTRATION ON MARS

The best way to improve aerocapture techniques around Mars is to focus on a aerocapture demonstration around Mars.

As can be seen in the table below, sizing a mission around Mars or around earth has consequences on navigation strategy and choices. Regarding the thermal aspects, for the alternative scenarios the vehicle will perform a high speed entry in an air atmosphere whereas for aerocapture around Mars the velocity is really lower in CO2. The thermal and mechanical environment is different and asks for other materials and strategies of design.

Criteria	Baseline (Mars aerocapture)	A/C around Earth (High)	A/C around Earth (low)
Main characteristics:			
V	5700 m/s	12700 m/s	9050 – 11500 m/s
FPA	-10°	-6.8 °	-5°
Peak stagnation convective heat flux	460 kW/m2	4050 kW/m2	873 kW/m2
Peak stagnation radiative heat flux	negligible	Expected to be high (TBD)	86 kW/m2
Max stagnation heat load (convective + radiative)	42 MJ/m2	212 MJ/m2 (convective part only)	86 MJ/m2
Aerothermodynamics	CO2 environment with winds & storms Trade-off of shape Need of S/C protection (TBC) Interaction with thrusters	Earth environment High speed re-entry Trade-off of shape Need of S/C protection Interaction with thrusters	Earth environment High speed re-entry Trade-off of shape Need of S/C protection Interaction with thrusters
GNC	Focus on relative navigation	Optical navigation to be adapted to Earth because of the planets sizes.	Only partial optical navigation: navigation on planet limbs. All the algorithms needed in an optical relative navigation to Mars would not be demonstrated.
TPS	Innovative ablation material based on cork	Material based on cork does not fit	Material based on cork does not fit

Because of the high heat flux, aerocapture requires a thermal protection system to shield the spacecraft from aerodynamic heating, as well as the use of a guidance system to assure that the spacecraft leaves the planetary atmosphere on the correct trajectory.

As a conclusion AEROFAST focuses on an aerocapture manoeuvre on MARS.

AEROFAST FRAMEWORK

A first iteration loop was performed in order to build the framework of the AEROFAST project.

The expected results of this iteration loop are:

- The mass characteristics of the vehicle (needed for future GNC activities),
- A preliminary aerocapture trajectory (including dispersions) needed to start the future thermal protection studies,
- A set of conditions to be applied to the future relative navigation activities.

In order to achieve these goals, a mission analysis study is performed, completed by an aerocapture corridor study aiming at defining the opportunities with their related aerocapture conditions and ΔV budget.

A preliminary design office study has been performed on this basis with the following assumptions:

- Three modules vehicle concept (final satellite, aerocapture module and cruise module),
- Apollo aerodynamic outer mold line,
- Hydrazine propulsion with existing 20 N engines,
- Scientific payload organised around a high resolution camera for a priori global amount of 200 kg requesting a HGA,
- A priori 1 failure tolerant architecture.

Preliminary mission analysis

The aim of the preliminary mission analysis is to provide input to the preliminary mass budget (ΔV budget), and to identify the main constraints to be taken into account during the study.

Note: The analysis started backward from the post-aerocapture down to the interplanetary trajectory in order to define the preliminary propellant budget.

1- Main characteristics of the different phases

Post-Aerocapture phase

The baseline of the scientific mission is Mars observation. In this preliminary phase, the targeted scientific orbit chosen is a frozen quasi circular sun-synchronous orbit at a low altitude with altitude and inclination characteristics of respectively 350 km and 92.8°.

These characteristics enable to estimate the necessary ΔV and propellant budget for the scientific satellite. The orbit will be refined during the study to get a correct coverage of Mars surface and repetitiveness of the observations. This will

depend on the precision and the field of vision of the instruments.

The preliminary ΔV budget estimated for the scientific payload is split up into:

- 175 m/s for circularization and dispersions recovery from aerocapture phase
- 100 m/s for orbit maintenance
- 75 m/s to modify the solar local time of the orbit.

The total ΔV budget is 350 m/s.

Aerocapture phase

The nominal aerocapture trajectory is computed with preliminary data for AEROFAST vehicle to fit at least within a Soyuz-class launcher:

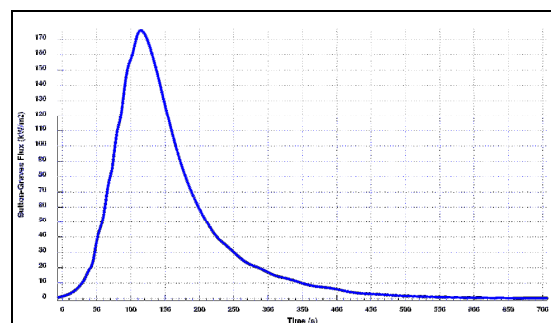
- Mass: 1100 kg,
- Diameter: 3.6 m,
- Aerodynamics coefficients: based on ARD vehicle.

As a starting point the nominal trajectory is computed with a constant bank angle of 90° all along the trajectory and without reverse manoeuvres. The initial infinite velocity at arrival is set to 4 km/s. The perigee altitude is optimized to obtain an apogee altitude of 350 km after the aerocapture phase.

The optimized entry conditions at 120 km are:

- Relative velocity: 6112 m/s
- Flight Path Angle: -10.59°
- Perigee altitude: 40.9 km

The aerodynamic phase lasts 690 seconds from the atmospheric entry at 120 km to the atmospheric exit at 120 km. The maximum heat flux value of 176 kW/m² is obtained 118 seconds after atmospheric entry.



Sutton-Graves heat flux

This reference trajectory will be used for the corridor analysis and the related extreme loads computation.

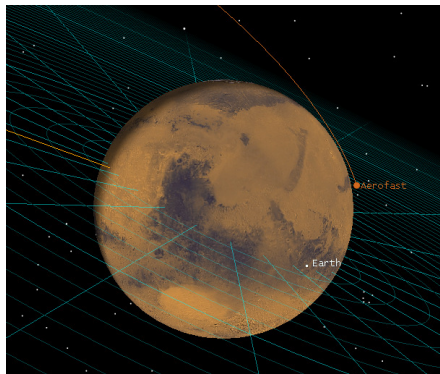
Pre-Aerocapture phase

In order to minimize the departure and arrival infinite velocities, the planetary transfer has to be predicted with the best departure and arrival dates.

In the case of AEROFAST mission, additional constraints must be considered:

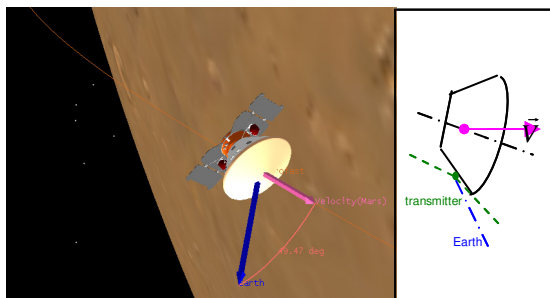
- Communication constraint

The main goal of the mission being to demonstrate aerocapture technology, the vehicle should be able to transmit data during aerocapture phase. As the lowest altitude of the trajectory is approximately 40 kilometres, the aerocapture has to take place on the side facing the Earth (V_{∞} directed toward Earth at ± 90 deg).



Aerofast arrival on Mars

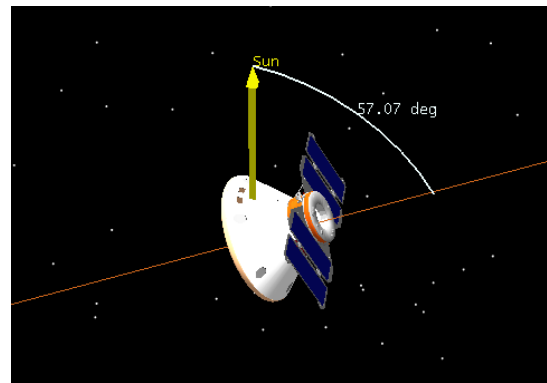
In a general way, a solution with visibility from Earth is guaranteed if the angle between the infinite velocity and the Earth direction is lower than 90° . Furthermore, as the transmitters cannot be placed on the shield side, the angle between the Earth direction and the vehicle velocity has to be greater than 40° during the whole aerocapture.



Mars Velocity - Earth direction angle during aerocapture

- Navigation constraints

To be able to use optical measurements on Mars moons, the spacecraft trajectory has to respect lighting requirements in the last 20 days before entry. Star trackers are usually designed to achieve measurements to star line-of-sights within 40° of the sun line-of-sight. Therefore the trajectory has to be designed to have an angle between the sun, the spacecraft and the Martian moons larger than 40° . The navigation system studied in [1] rejects all the measurements with line-of-sights within 45° of the sun line-of-sight.



Sun - spacecraft - Mars angle during final approach

Another constraint on the trajectory due to the use of optical navigation concerns the moons phasing in the last days before encounter: the sighted moon and the inbound trajectory have to be correctly phased to give optimal geometry information.

The navigation constraints can be translated into a constraint on the infinite velocity. During the final approach, the Martian moon direction could be approximated by the infinite velocity direction as the vehicle flies forward Mars with a velocity close to the infinite velocity. The constraint could be stated as the angle between the infinite velocity at arrival and the Sun direction must be greater than 45° .

The constraints to be applied on the launch opportunity can be summarized as follows:

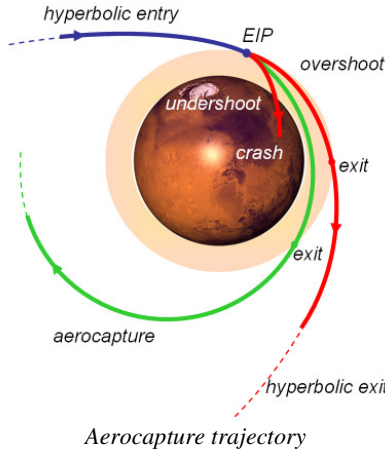
- the infinite velocity at Earth departure has to be minimized,
- for the infinite velocity at Mars arrival:
 - o the module has to be lower than 4 km/s,
 - o the angle with Earth direction has to be lower than 90° ,
 - o the angle with Sun direction has to be greater than 45° .
- the angle between the Earth direction and the vehicle velocity has to be greater than 40° during the whole aerocapture.

2- Definition and sizing of the aerocapture corridor

The aerocapture trajectories are bounded by overshoot and undershoot trajectories defining the so-called aerocapture corridor. These extreme trajectories are defined as follows:

- the overshoot trajectory that leads to a hyperbolic exit corresponds to a full lift-down manoeuvre (bank angle forced at 180 deg all along the atmospheric path) for the shallowest entry conditions at Entry Interface Point, or EIP, that is defined at 120 km AGL;
- the undershoot trajectory that leads to a crash on the planet corresponds to a full lift-up manoeuvre

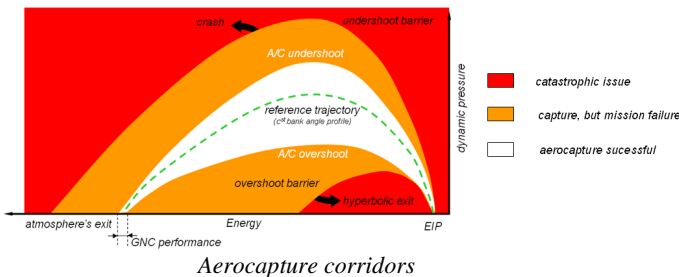
(bank angle forced at 0 deg all along the atmospheric path) for the steepest entry conditions at EIP.



Aerocapture trajectory

Interest of sizing the aerocapture corridor is to define the expected/requested performance of the pre-aerocapture phase whose main contributors will be the Navigation performance expressed with respect to flight path angle (or FPA) errors and the S/C release errors and their propagation till EIP.

Next figure displays the different definitions of the aerocapture corridor in the Energy plane vs. dynamic pressure.

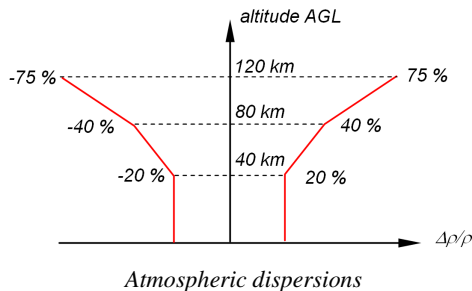


Aerocapture corridors

Input data

The sizing of the aerocapture corridor depends on:

- the nominal EIP conditions provided by Mission Analysis results;
- the atmospheric dispersions that are derived from Kass-Schofield model (used for defining MER entry conditions) as follows:



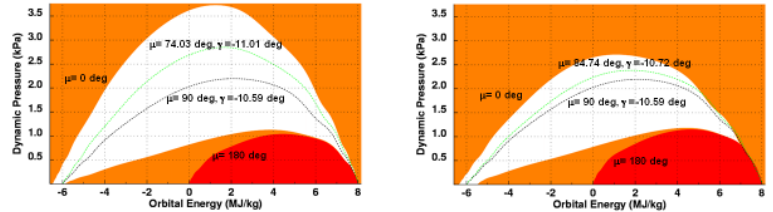
Atmospheric dispersions

- the uncertainties on drag and lift coefficients (defined in body axes;

- the targeted apoapsis altitude is set at 362 km AGL what yields a 350 km value when reaching the apoapsis due to J2 effects;
- the GNC accuracy on the targeted apoapsis is set at ± 100 km.

Aerocapture corridors

The next figure displays the aerocapture corridors obtained when considering ± 15 % (case 1, left) or ± 30 % (case 2, right) uncertainties on drag and lift coefficients.



Aerocapture corridors with uncertainties

These estimates lead to widths of corridor from ± 1.06 deg (case 1) to ± 0.79 deg (case 2) when expressed wrt median EIP flight path angle (resp. -11.01 deg or -10.72 deg). Thus, pre-aerocapture Navigation performance and propagation of S/C released down to EIP must be such that FPA errors at entry point are below ± 1.06 deg (case 1) or ± 0.79 deg (case 2).

The corridor's barriers are defined with respect to the periapses.

trajectory	Case 1	Case 2
overshoot	50.1 km	50.4 km
undershoot	17.4 km	26.5 km

Corridor barriers defined by periapses

Eventually, according to lessons learned from past ESA-sponsored studies (ATPE), the nominal arrival conditions should be such that the atmospheric path is in the middle of the aerocapture corridor with respect to the flight path angle and not to the bank angle as it is done mainly for simplification purpose of the mission analysis task. Main impact of such baseline is to increase the structural but also thermal loads on the capsule, the EIP flight path angle leading to steeper entry conditions. For instance, following values are obtained on reference trajectories corresponding to mid-corridor with respect to bank angle (optimal case from Mission Analysis, case 1) or to flight path angle (nominal case for GNC studies, case 2).

trajectory	Parameter	Case 1	Case 2
optimal	heat flux peak	177.9 kW/m ²	177.9 kW/m ²
(centred wrt bank angle)	g-load peak	2.72 g	2.72 g
	dyn. pressure peak	2.19 kPa	2.19 kPa
nominal	heat flux peak	194.7 kW/m ²	183.5 kW/m ²
(centred wrt FPA)	g-load peak	3.53 g	2.96 g
	dyn. pressure peak	2.85 kPa	2.38 kPa

Corridor barriers for reference trajectories (mid-corridor wrt. bank angle or FPA)

One observes that thermal and structural loads on the nominal trajectory are lower on case 2, but this is achieved at the expense of a narrower corridor.

Heat loads and heat flux

1- Stagnation point heat flux

An engineering correlation (Sutton & Graves) is used for the calculation of the cold wall heat flux at the stagnation point.

2- Max Heat Flux Point

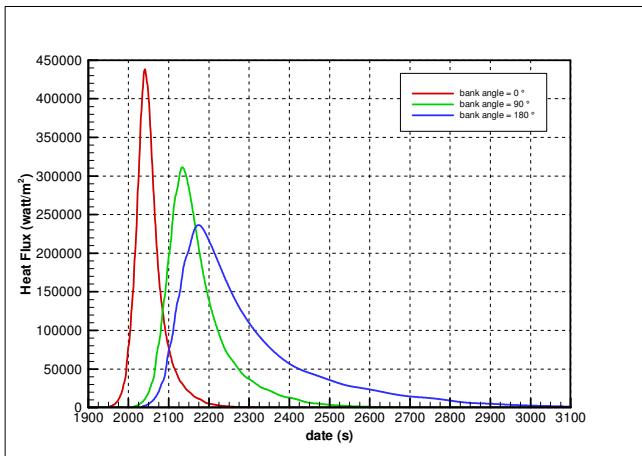
For the max heat flux assessment on the heatshied, a calibration factor derived from ARD studies is used.

The max heat flux point should be located at the toroidal part and the calibration factor is chosen from Euler + Boundary Layer computation in perfect gaz at Mach 10 for angle of attack value of 25°.

Due to the high angle of attack value, the location of the stagnation point is not so far from the max heat flux point located on the toroidal part. For this reason, no turbulent effect is taken into account.

3- Preliminary Heat flux time histories

The max heat flux time history (cold wall case) is determined for 3 trajectories (bank angle value of 0°, 90° and 180°). It is illustrated on the following figure.

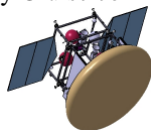


Max heat flux trajectory

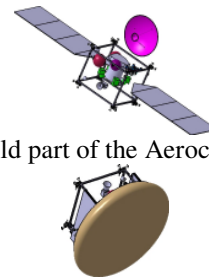
Design and lay-out

For this preliminary study, the Spacecraft composite is assumed to be composed of:

- the Interplanetary Cruise configuration,



- the Mars orbiter (hosted in the Aerocapture system),



- the front shield part of the Aerocapture system.

At this step of the study, the Spacecraft design is not refined; for instance:

- the structures have been roughly assessed and their mass is a preliminary evaluation (based on their main dimensions);
- equipments are laid out, but none of their specific supports is shown;

However this general design allows to check that all the needed elements can be accommodated inside rough volumes – compatible with a launcher fairing or first assumptions on the aerocapture composite aerothermodynamic behaviour – and gives a base for mass budget evaluation.

1- Avionic architecture

Avionic system can be split in several sub-systems each one made of a set of equipment as listed below:

ORBITER

- DHS – Data Handling System for Mission & Vehicle Management and GNC (S/W)
- NAVIGATION
- POWER
- CONTROL
- COMMUNICATIONS

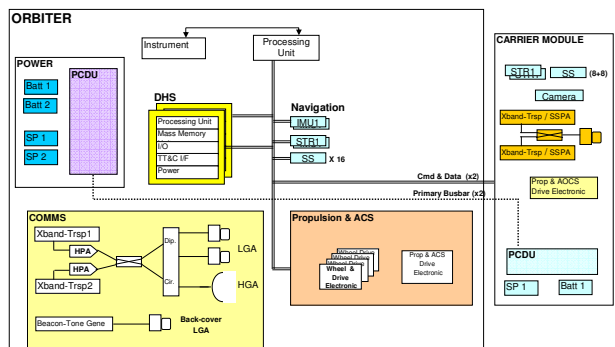
ORBITER PAYLOAD

- Dedicated processing unit
- Dedicated data bus to DHS Mass Memory

CRUISE MODULE

Some units from the Orbiter may be used by the Cruise Module during the cruise phase

- NAVIGATION
- POWER
- CONTROL
- COMMUNICATIONS



Orbiter avionic architecture

2- Structure and thermal protection system

Structure and thermal control aspects for the Interplanetary Cruise module and the Mars Orbiter are not yet detailed: the assumption is that classical satellite technologies will be used, such as sandwich panels to build the main structures and a combination of radiative areas and heat loops piping for the thermal control.

Front Shield structure

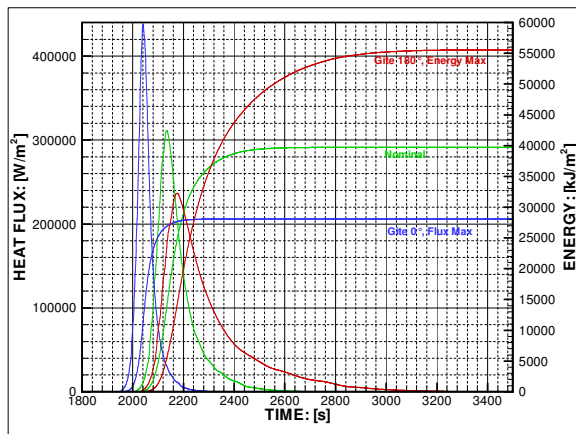
The Front Shield structure main function is to keep a defined external shape so that the vehicle has aerodynamic coefficients along the Aerocapture phase of the mission coherent with the parameters used by the flight control software. Even if the mechanical loads during the aerocapture phase are relatively low (large dimensions), the Front Shield has to be designed as a rather “stiff” structure.

One good way to answer this requirement is the use of a sandwich structure, using external Carbon Fibre Reinforced Polymer skins and a light alloy honeycomb core.

Typical dimensions could be 0.5 mm thick skins with a 30 mm height honeycomb core. With the mass of the bonding between the skins and the core, it leads to an apparent mass of 3.4 kg/m² for the “current” point of the structure. Taking into account the needs of local reinforcement for interfaces (attachment points to the Orbiter structure, attachment of thrusters, tanks ...), a global equivalent mass of 4 kg/m² was assumed.

Front Shield thermal protection

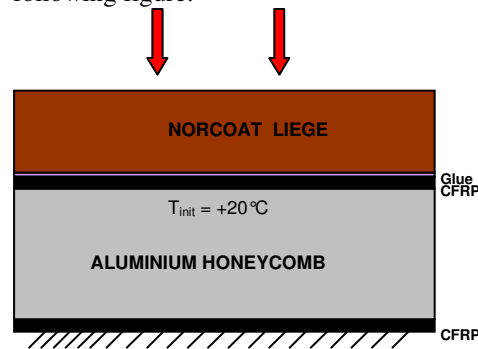
Analyse hypothesis: The used thermal loads refer to the maximum energy trajectory which is characterized by a bank angle of 180°. This trajectory presents a maximum heat flux of 240 kW/m² reached at an altitude between 40 and 45 km. The maximum energy value is around 55 MJ/m² at the highest layer of atmosphere.



Heat flux and energy trajectory history as a function of time for 3 bank angles

Material: Based on the global thermal loads of the mission, and on previous experience on TPS sizing for Mars re-entry mission, the chosen thermal protection system material is the Norcoat-Liege in order to allow start the assessment of TPS. The Norcoat liege is a flight proven ablator material made of cork powder and phenolic resin matrix.

The whole TPS design is summarized in the following figure:

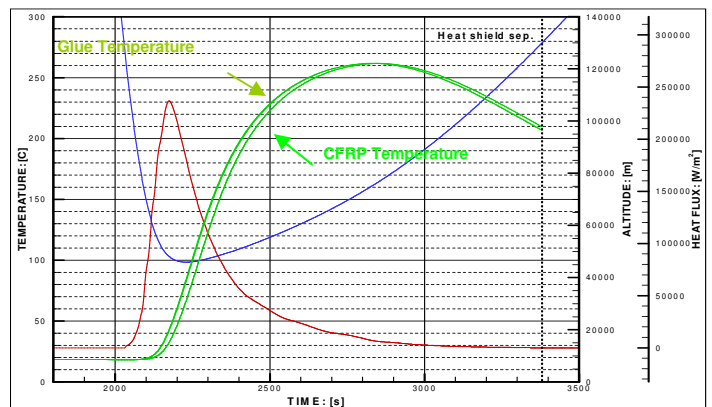


For this preliminary sizing, the effects on maximum structure temperature of only 2 parameters have been investigated:

- Norcoat-liege thickness
- Glue thickness

Results

The following figure represents a typical temperature evolution of the aeroshell for a Norcoat –Liege thickness of 8 mm.



Structure temperature history for a 8 mm Norcoat-Liege layer thickness

The maximum structure temperature is reach before the aeroshell jettisoning (at least above 70 km). Atmospheric thermal load cannot be neglected.

The aerial mass defined previously takes only into account the Norcoat-Liege and glue density. For example, the structure temperature criterion (180°C) is reached for a maximum Norcoat liege

thickness of 11.5 mm corresponding to a minimum glue thickness and an aerial mass around 5.5 kg/m². At a 14 mm design thickness of Norcoat, increasing the TP of 1 mm would decrease the maximal temperature of the structure of around 12°. As expected, the sensitivity of the structural temperature decreases with larger thermal protection thickness.

Conclusion for Front Shield

The Front Shield has a global thickness around 45 mm and can be assumed to weight around 10 kg/m² (4 kg/m² for structure, 5.5 kg/m² for thermal protection).

Assumptions for Back Cover

In the proposed design, the Back Cover has to sustain quite lower mechanical and thermal loads than the Front Shield. The first assumption will then be a mass of around 5 kg/m².

3- Product tree

The AEROFAST spacecraft in its launch configuration includes three main components as shown in next table

4- Mass budget

The establishment of the mass budget is based on:

- literature data for “out of the shelf” components,
- preliminary evaluation for specific components (for instance through main dimensions for the structure).

A first assessment is as follows:

Aerofast Spacecraft composite	1260 kg
Interplanetary Cruise Module	270 kg
Aerocapture Composite	990 kg
Mars Orbiter	730 kg
Aerocapture System	260 kg

NEXT STEPS

The AEROFAST project will last 2 and a half years. The project activities have started beginning of 2009. During the first semester, a preliminary design of the aerocapture and mission was performed. The next months will focus on the detailed definition of the system requirements.

The work break down structure of AEROFAST project is shared out of five work packages in order to fulfill the objectives of the project. The first one is dedicated to the overall mission definition and set of requirements, as well as to the different system analyses. The second one concentrates on the vehicle itself and its architecture, budget and layout. The 3rd work package intends to find out attractive solutions for thermal protection systems. The 4th ensures the design and establishment of the real time test bench with focus on relative navigation. The 5th work package deals with on-board instrumentation during the aerocapture phase. A sixth one deals with management dissemination and exploitation.

Regarding the objectives OBJ2 and OBJ3 the following steps will be successively:

- Definition of prototype algorithms,
- Definition of the camera requirements,
- Validation of the algorithms with a non real time numerical simulator,
- Final validation with a real time simulator, including a navigation camera breadboard.

For objective OBJ4, the following steps will be:

- Identification of potential thermal protection solutions,
- Characterization of the two or three selected solutions thanks to two loop study involving wind tunnel tests and computation.

Aerofast Spacecraft composite	nb	Comments
Interplanetary Cruise Module		
Avionics		The Cruise Module uses the Mars Orbiter Data Handling System
Power		The power system of the Cruise Module uses the Mars orbiter batteries and PCDU
Solar Arrays		
Sensors		
Star Trackers	2	
Sun sensors	8	
Camera	1	
Communications		
Low gain antennas	4	covering all space, for safe mode
AOCS Actuators		The Cruise Module uses the Mars Orbiter wheels and the Aerocapture System thrusters for its AOCS
Main propulsion		
Structure		
Thermal Control		
Aerocapture Composite separation system		
Aerocapture Composite		
Mars Orbiter		
Scientific payloads		
Avionics		
Power		
Solar Arrays		
Batteries		
PCDU	1	(failure tolerant design)
Sensors		
Sun sensors	16	for full redundancy, including safe mode
Star Trackers	2	redundancy
IMUs	2	redundancy
AOCS Actuators		
Inertia wheels	4	redundancy
Thrusters	12	3 x 4 (pure torques in nominal configuration, failure tolerant)
Communications		
High gain antenna	1	
Low gain antennas	4	covering all space, for safe mode
Transponders		
Data Handling		
On Board Computer	1	(failure tolerant design)
Mass Memory	1	(failure tolerant design)
Main propulsion		
Thrusters	4	
Propergol Tank	1	
Pressurisation tank	1	
Valves and piping	X	
Structure		
Thermal Control		
Radiators		
Heaters		
Piping		
MLI		
Aerocapture System		
Aerocapture Front Shield		
Back Cover		
Actuators		
Thrusters	8	The Aerocapture System will rely on the Orbiter batteries for power and use the Orbiter wheels as
Propergol Tank	1	actuators in combination with its own thrusters
Pressurisation Tank	1	
Valves and piping	X	
Aerocapture System jettisoning system		

Objective OBJ5 will progress with the identification and definition of the sensors necessary to perform measurements during the aerocapture flight.

CONCLUSION

Aerocapture on Mars is for the time being not currently studied by the space agencies. This is the reason why this project is proposed in the frame of the FP7, this project is a typical one for research and development. AEROFAST end goal is not to perform a phase A of the mission demonstration but is to prepare for such a mission demonstration and increase the TRL level: AEROFAST intends in the frame of the FP7 to reach a TRL 3 to 4 for this technology. Moreover the AEROFAST partners have been chosen in accordance with these goals. Main focus of AEROFAST is put on relative navigation and thermal aspects.

A preliminary mission scenario and concept was drafted in order to provide inputs to the main objective. The preliminary study encloses the necessary inputs to frame the project and provide inputs to the different work packages of the project. Preliminary computations have been performed to define the mission scenario and the aerocapture corridor and assess the thermal loads. A preliminary design is built and preliminary assessment of mass budget is provided. These are the inputs that will be used by the consortium as a point of departure.

AEROFAST CONSORTIUM

The AEROFAST consortium is composed of the following partners:

Astrium-ST - France and Germany,
DEIMOS - Portugal,
AMORIM – Portugal,
SAMTECH – Belgium,
UNIROME – Italy,
INETI - Portugal,
STIL-BAS – Bulgaria
IoA - Poland,
SRC-PAS - Poland,
ONERA - France,
KYBERTEC - Czech.

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