AEROFAST: AEROCAPTURE FOR FUTURE SPACE TRANSPORTATION (IPPW-7)

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1. ABSTRACT / 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. Such manoeuvre saves a significant amount of mass with regard to a more conventional technique of insertion using propelled braking. Interest in developing aerocapture technology stems from the solar system exploration needs: sample return missions, in-situ missions and future manned missions require spacecraft to enter and manoeuvre in a planet's atmosphere in order to meet their mission objectives.

Aerocapture on Mars is for the time being not currently studied by the space agencies. This is the reason why the AEROFAST project was proposed in the frame of the European Community Framework n°7 (FP7), as a typical research and development project. Aerocapture technology is at Technology Readiness level (TRL) 2 to 3 in Europe. The objective is to increase the TRL to level 6 through a flight demonstration study performed with the Martian conditions regarding the future Mars Sample Return missions. In order to prepare for such a mission demonstration AEROFAST main objective is to reach a TRL 3 to 4 for this technology.

Aerocapture is a system level technology where disciplines such as system analysis and integrated vehicle design, aerodynamics, aerothermal environments, thermal protection systems (TPS), guidance, navigation and control (GN&C), instrumentation need to be integrated and optimized to meet mission specific requirements.

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 different objectives of the project are:

Obj1: Define a project of aerocapture demonstration

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.

AEROFAST project started in January 2009. The aim of this paper is to present the preliminary and achieved results. The frame of the mission has been defined: low cost mission, Soyuz-like Launch performance, optimized entry conditions at 120 km, main characteristics of the spacecraft (three modules vehicle concept).

The three main phases of the aerocapture manoeuvre were assessed:

- Pre-aerocapture phase: Challenge is to master the attitude/position of the S/C (spacecraft) before manoeuvre.
- Main aerocapture phase: Goal is to reach an elliptical orbit within a narrow corridor and challenge is to sustain important heat loads.
- Post aerocapture phase (transfer to a parking orbit): to target a quasi circular sun-synchronous orbit at a low altitude.

Emphasis will also be put on the spacecraft complete architecture and lay-out, including the front shield: aerodynamic shape and associated thermal protection system.

2. MISSION AND GNC

2.1 Mission Description

The analysis of the Earth to Mars opportunities has been performed for the whole decade 2020-2030. Considering communication visibility during aerocapture and navigation constraint, the set of 21 opportunities is depicted in Figure 1.

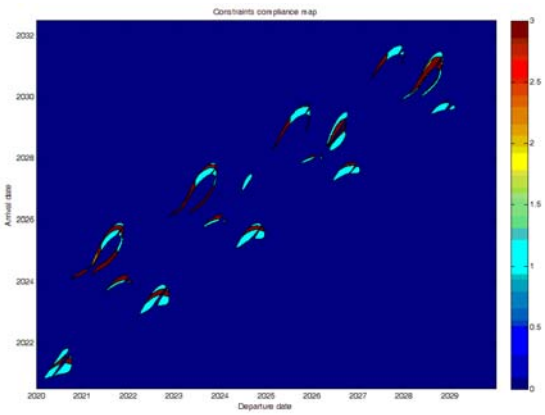

Figure 1: Constraints compliance map: no compliance (dark blue), arrival velocity < 4 km/s (light blue), arrival velocity SAA > 45º & EAA < 90º (orange), aerobraking velocity > 40º (brown)

Among them, the trajectory corresponding to a departure in May 2026 and to an arrival in June 2028 has been selected for detailed studies. The TCM (Trajectory Correction Maneuver) strategy enables the control of the planetary protection constraint while decreasing the dispersions

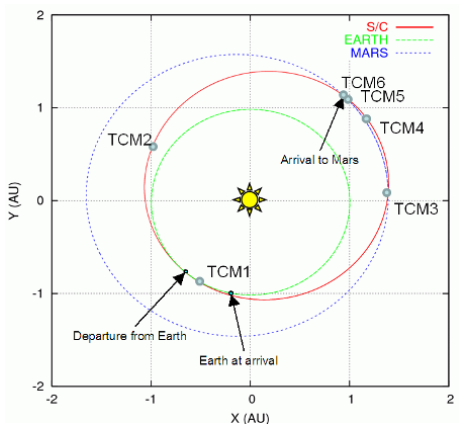

Figure 2: Proposed location of TCMs along the interplanetary trajectory

The initial conditions for the aerocapture are a V∞ of 4 km/s, an inclination of 94 degrees and a pericenter altitude of 28 km. Targeting an altitude of the apocenter of 316 km at the exit of the atmosphere, the spacecraft experiences a load factor less than 3 g and a heat flux peak of 160 kW/m^2, which are moderate values with regard to Earth reentry values.

The final orbit acquisition is obtained by a combination of successive maneuvers and by the exploitation of the natural nodal drift during a few months. The frozen sun synchronous orbit is characterized by:

- semi major axis 3713.87 km
- eccentricity 7.355 10-3
- inclination 92.7°
- argument of the pericenter 270°

The period of repetition is 1098 nodal periods and 85 nodal days for a swath of 20 km. The drag effect during the period of repetition requires a 3.6 m/s correction budget every 85 days.

2.2 GNC Concept

Interplanetary cruise phase

During this phase, the spacecraft is in Sun pointing mode. The only changes in orientation are required for the 6 TCM and for target acquisition by the optical navigation sensor during the last part of the trajectory.

The control is performed by reaction wheels well adapted to fine target pointing (see further the pointing accuracy results). Thrusters of the cruise module are used only for the desaturation of the wheels.

The attitude sensor is basically a star tracker and it is questionable to use the optical sensor also as an attitude sensor (eliminating the mounting error).

The navigation relies on DSN (Deep Space Network) transponder and on the optical sensor. The current status of the navigation analysis has not yet converged. Many points are investigated:

- optical characteristics of the sensor (focal length(s), number of pixels)
- kind of measurements (Mars Line Of Sight (LOS), Phobos and Deimos LOS, Mars range, Phobos and Deimos orbit) and the strategy of measurements in function of the distance taking into account DSN or not hybridation
- the influence of attitude error (mounting errors, thermal deviation), of the measurement frequency

A real time test bench, including a camera and a representative image scene, has been defined for the future validation.

Aerocapture phase

The navigation performance has been estimated for 3 different initial conditions corresponding to DSN measurements only (MER values) and 2 better assumptions. Using a Litton 200 Inertial Measurements Unit, the accuracy on the altitude is depicted Figure 3.

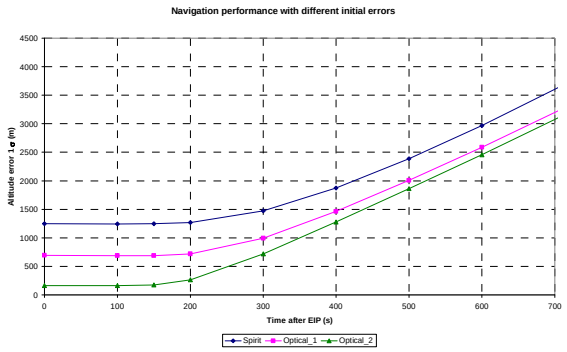

Figure 3: Altitude accuracy

The principle of the guidance is a numerical predictor-corrector targeting a given apocenter altitude at the exit of the atmosphere. The lateral control is performed by classical bank reversal according to a corridor around the nominal trajectory.

The estimated performances of the algorithm have been estimated by Monte-Carlo simulations and give the satisfactory following results.

Final constraints	mean value	std deviation	worst case
pericenter altitude	-60.5 km	33.7km	-176.2 km
apocenter altitude	335.3 km	48.6 km	202.2 km
inclination	94.23 deg	0.10 deg	93.89 deg
pericenter raising cost	99.3 m/s	9.1 m/s	131.1 m/s
apocenter correction cost	2.9 m/s	10.9 m/s	-56.7 m/s
inclination correction cost	-13.7 m/s	6.1 m/s	-26.7 m/s
ΔV correction cost	122.1 m/s	13.6 m/s	177.1 m/s

The control is performed at a frequency of 7 Hz, giving almost a factor 2 margin with regard to the 4 Hz frequency of the pulsation of the angle of attack.

The needs in torque authority have been defined at 200 mN for roll and 100 mN for pitch and yaw with a MIB (Minimum Impulse Bit) duration of 25 ms.

The simple linear controller is derived from the flight proven ARD (Atmospheric Re-entry Demonstrator) controller. An example of the behavior is given in Figures 4 and 5 showing the angle of attack and side slip angle evolutions.

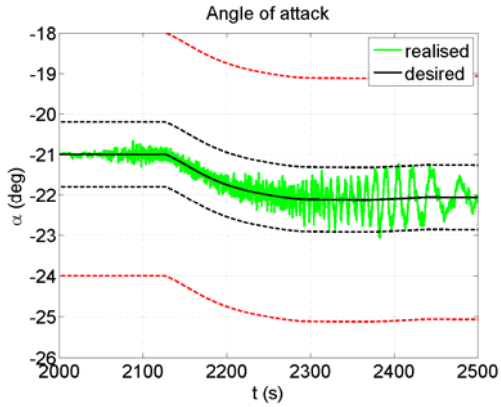

Figure 4: Angle of attack evolution

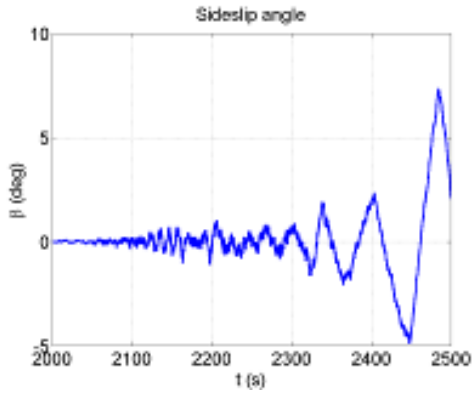

Figure 5: Slideslip angle evolution

The evolutions after T=2350 s are non significant since the dynamic pressure is lower than 50 Pa and that it would correspond to another setting designed for the orbital flight after exit of the atmosphere.

Post aerocapture phase

The main driver is the accuracy of the pointing when the target is locked. This constraint drives the selection of commercially available accurate gyro, star tracker and reaction wheels.

Simulations of long pointing phases illustrate the performances in pointing mode (Figure 6) with an average accuracy of 0.07 degrees in average and a standard deviation of 0.013 degrees compatible with a swath of 20 km with a resolution of 1 m.

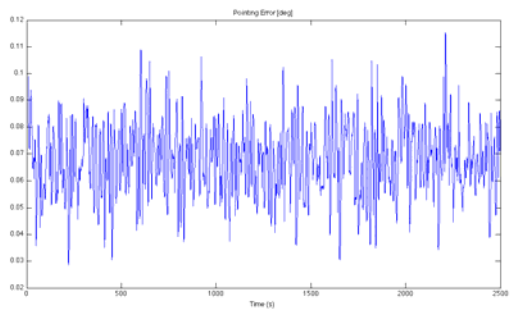

Figure 6: Performances in pointing mode

3. THERMAL PROTECTION SYSTEM

3.1 TPS Overview

One key objective of the AEROFAST project, referred as Obj4, is to Demonstrate/prototype the thermal protection system for such a mission.

In parallel, it is necessary to establish a preliminary definition of the TPS, as part of the spacecraft design (referred as Obj1: Define a project of aerocapture demonstration).

3.2 Development of Innovative Thermal Protection

The first step of the TPS development was obviously to setup an appropriate set of requirements, summarized in *Table 1* below.

Entry Environment	Heat Flux, Pressure, Atmosphere
Mass & Thermal Performance	Minimal TPS mass
Ablation Performance	Minimal, homogeneous, Predictable
Mechanical Requirements	Strength, Vibration, depressurization
Material Requirements	Outgassing, Space ageing, cleanliness
Interface	Structure, fixation, coating
Environmental Requirements	Vacuum, micrometeoroids, temperature
Physical Reqts	Emissivity, no transparency
Product Assurance	REACH, lifetime, flammability, Planetary Protection
Design Requirements	Thickness, curvature, dimensions
Programmatic Requirements	Availability, TRL, European, flexible process

Table 1 Set of requirements driving TPS elaboration

The selection of the most promising candidates will then be carried out. The ranking of requirements lead to the following key selection criteria

- Thermal Protection Performance
- Withstand heat flux, low ablation
- Process compatible with design requirements
- Long term availability

Relying on Astrium heritage, with the use of Norcoat-Liege for ARD, Beagle or ExoMars [5], it was deemed appropriate to assess advanced cork-based TPS.

In line with above-mentioned requirements and criteria, the main searched features are:

- Lighter solution wrt existing Norcoat-Liege
- More robust solution (limited ablation)
- Compliant with environment rules (REACH)

To achieve this enhancement of materials properties, the following steps are implemented

- Inclusion of appropriate fillers
- Assessment of production processes
- Evaluation and characterization

After a first series of satisfactory manufacturing and evaluation tests performed in the first quarter of 2010, complementary formulations are currently tested.

Figure 7: Cork based TPS with various formulations

3.3 TPS design for the Spacecraft

TPS is one of the subsystems to be defined in the framework of the overall spacecraft design, with the main objective to provide relevant inputs for system analyses. The preliminary design presented in [1] remains unchanged for the time being. It will need to be updated once aeroshape is selected and corresponding ATD data are established.

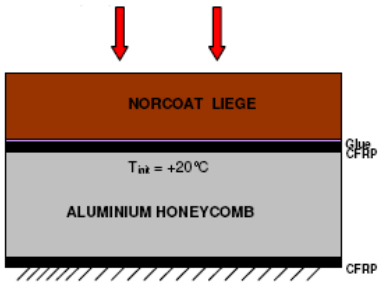

Figure 8: TPS concept

TPS concept is based on a classical sandwich structure covered with thermal protection (*Figure 8*). Norcoat-Liege is used as reference until sufficient data are available for enhanced formulations.

For the Front Shield the global thickness is ~ 45 mm (30 mm struct. + 14 mm TP), leading to an areal weight around 10 kg/m² (4 kg/m² for structure, 5.5 kg/m² for thermal protection).

For the Back Cover, which has to sustain quite lower mechanical and thermal loads, the first assumption is a surfacic mass of around 5 kg/m².

Beyond the TPS sizing itself, this part of the project offered the opportunity to develop (as a module of SAMCEF) a 3D thermal/pyrolysis/ablation code [4], which is indeed the most valuable outcome of this WP.

4. SPACECRAFT DESIGN

4.1 Aeroshape concepts

Aerocapture GNC requires an aeroshape that provides a lift-over-drag (L/D) ratio with sufficient provision ($L/D \geq 0.3$) and acceptable static stability performances (which drives the center of gravity (CoG) workable box definition). Additionally, the aeroshape must protect the payload (P/L) from a severe aerothermal environment during aerocapture while minimizing the required heat-shield mass. Last but not least, the aeroshape outer mold line (OML) shall enable a safe jettisoning of the heat-shield after aerocapture; to that respect, past studies concepts used truncated aeroshapes (see Figure 9), that simplified the post-aerocapture heat-shield jettisoning while offered ressources sharing capabilities with the carrier module during the cruise phase. The key issue then is the protection of the satellite from aeroheating during aerocapture. It was observed in the past that local severe overheating might occur because of flow re-attachment on the P/L (see Figure 10). When flow re-attachment occurs, specific protection is required; local TPS placed on critical zones or full encapsulation with a back-cover, which in turn makes more complex the post-aerocapture heat-shield jettisoning system design. Another important source of heating is the possible radiation of CO_2 molecule in the IR (low entry velocity, typically Ve<6 km/s) or UV (Ve>6 km/s) spectral range.

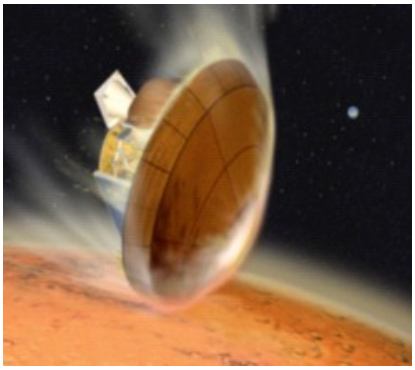

Figure 9: NASA-CNES MSRO project

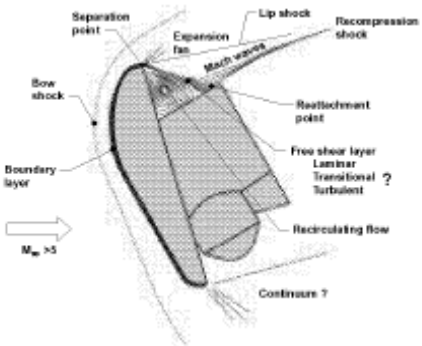

Figure 10: NASA-CNES MSRO Flow Re-attachment

In the frame of AEROFAST, two configurations have been considered (see Figure 11):

- A blunt body capsule, derived from Apollo-ARD concept,
- A lifting body dealing with a truncated biconic shape.

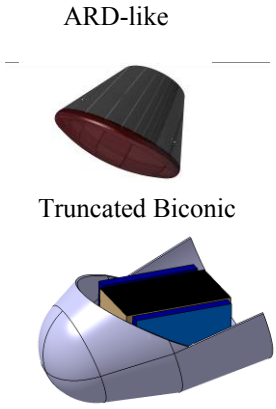

Figure 11: Candidate aeroshapes

<u>Apollo/ARD-like shape</u>

The shape has the front-shield used for the Apollo and ARD programs. High risk of P/L flow impingement with likely local convective overheating requires a payload full encapsulation. The back-cover is slightly larger when compared to Apollo-ARD (20° instead of 33° rear-cone half angle) for P/L accommodation purposes.

Main advantages of this aeroshape are:

- A high P/L protection efficiency with respect to aerothermal heating,
- A heat-shield manufacturing simplicity,
- A flight-proven aeroshape that mitigates development risks.

Main drawback is dealing with:

- A possible mass penalty due to the additional back-cover and a minimisation of ressources/sensor sharing with the carrier module during cruise,
- A possible more complex post-aerocapture heat-shield jettisoning

Initial aerodynamic data are taken on the ARD AEDB (AErodynamic Data Base) data and then correspond to an Earth-entry. Figure 12 shows the required trim incidence to achieve the desired L/D of 0.3 during the aerocapture maximum braking phase. The AEROFAST study will enable to determine the modifications on the coefficients for a Mars-entry (CO_2 atmosphere). The CO_2 effects are currently addressed by Astrium-ST with the support from Institute of Aviation (IoA - Poland).

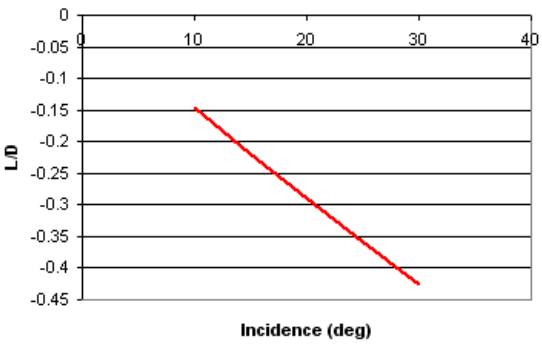

Figure 12: *L/D versus AoA at Pdyn max*

<u>Truncated Biconic shape</u>

This concept has been considered in past Astrium studies (ESA, CNES…). It is very promising in terms of payload protection so that a back-cover may not be needed, while L/D requirements can be easily achieved (see Figure 13)
As for the Apollo-like shape, the Biconic complies with lay-out requirements and protection with respect to aerothermal heating.
The shape is currently under study by Astrium-ST with the support of ONERA. In particular, the need of a back-cover that protects the payload from the aerothermal environment will be addressed by ONERA through dedicated CFD and radiation computations along the aerocapture phase.

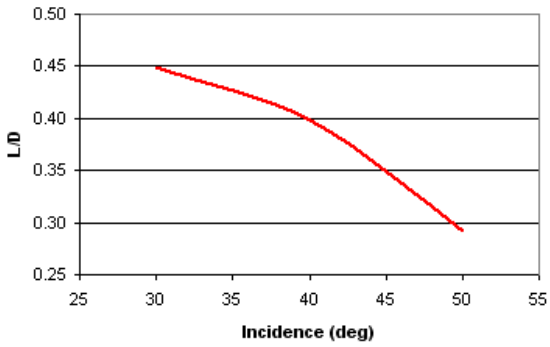

Figure 13: L/D versus AoA at Pdyn max

4.2 Spacecraft architecture

The AEROFAST spacecraft is made up of three elements:

- the interplanetary Cruise Stage (ICS),
- the Aerocapture System (AS)
- and the Mars Orbiter (MO)

One of the basic design drivers is the aerodynamic behavior during aerocapture. Currently two different AS shapes are under evaluation. They are presented within Figure 14.

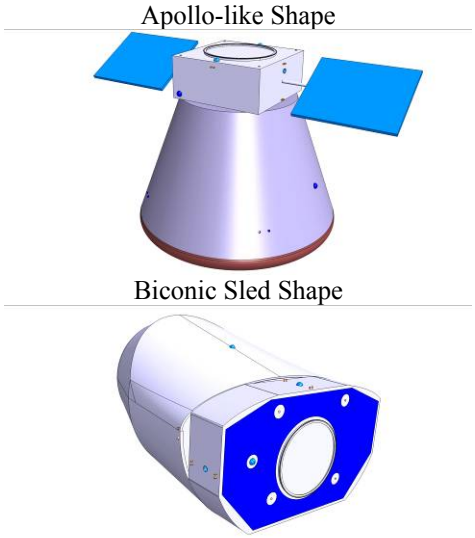

Figure 14: External Shape Alternatives of the AEROFAST spacecraft

Both shapes are depicted in their cruise configuration. The design of the individual mission elements is still on-going and hence the ICS design is not settled (e.g. body fixed solar cells or cells on a panel).

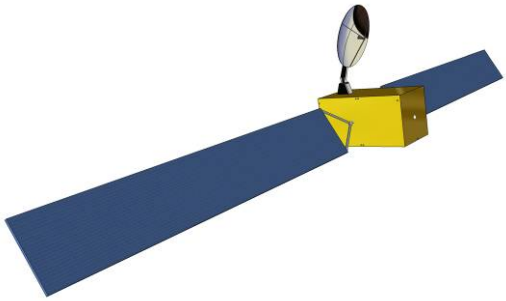

Figure 15: Mars Orbiter

The Mars Orbiter is the central element of the AEROFAST spacecraft. It serves as bus for the scientific payloads during the science mission and provides essential services, such as:

- attitude and orbit determination and control,
- telecommunication,
- data handling,
- thermal control,
- power generation, storage and distribution

Some of the functionalities are solely active during orbital phase, others are also used to support cruise and aerocapture. The functional layout of the whole system is depicted within Figure 16.

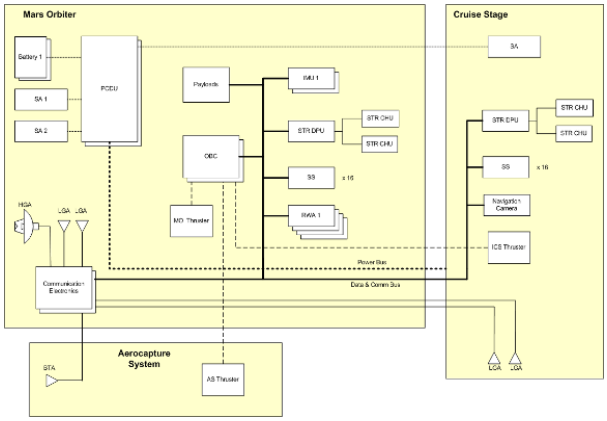

Figure 16: Functional Diagram

The MO is encapsulated inside the AS during interplanetary cruise and aerocapture. The AS provides thermal protection and attitude control capabilities during aerocapture. Dedicated instruments are monitoring the performance of the heatshield during aerocapture.

As its name imply, the cruise stage delivers the AS and the MO to Mars. It provides basic services such as power generation and attitude control. Low gain antennas are installed to provide a telecommunication link to Earth ground stations.

Payload and Instrumentation

The primary payload is the Imaging Multispectral Sensor which provides a maximum spatial resolution of 1 m (panchromatic). Space for additional sensors is allocated. At least a wide angle camera needs to be on-board in order to give a context for the high resolution imaging.

During aerocapture sensors are active which measure the performance of the atmospheric pass.

Communication Subsystem

Data downlink during science phase is provided by a high gain communication link. Currently a 1.8 m diameter two-axis gimbaled high gain antenna is foreseen for transmitting scientific data to Earth. The transponder will operate in Ka-Band, providing a minimum data rate of 440 kbit per second when Mars is at the maximum distance from Earth. A traveling wave-tube amplifier boosts the radio frequency power to 150 Watts.

The Ka-Band was only recently demonstrated to be a practical selection for data transfer on interplanetary missions. The successful Ka-Band demonstration on the Mars Reconnaissance Orbiter mission led to the implementation of a Ka-Band system as primary communication means within the Kepler mission. An alternative, and more traditional option, for the communication system would be the implementation of an X-Band communication link. However, the minimum data rate would be 102 kbit/s. The total transferable data volume over the complete mission (10 years) would decrease from 377 to 110 Tbit.

Power Subsystem

Power is provided by means of solar arrays in a Peak Power Tracking architecture. Two batteries are installed inside the Mars Orbiter which are recharged during daylight by two solar arrays, having a total size of 13 m² providing 460 Watts of electrical power at the end of life. A trade off still needs to be made in order to define whether the solar array will be rotated by a solar array drive mechanism to optimize the sun incidence angle. Alternatively, the size of the arrays could be increased in order to counteract lower efficiency when the incidence angle is not optimal. The benefit would be that the solar array drive mechanisms would not be needed.

During cruise phase a direct energy transfer power architecture is implemented. A solar array of 3.5 m² provides 420 Watts when arriving at Mars. The batteries from the Mars Orbiter will be charged by this array in order to use as much available components as possible. The solar cells will either be mounted directly on the surface of the ICS or attached on an external solar panel.

Attitude and Orbit Determination and Control

Analysis of different navigation system options is still on-going. Within the current design attitude determination during cruise and orbital phase will be performed by means of Star Trackers. Because the Star Trackers need to have a free sight, dedicated sensors are needed for both spacecraft elements. In contrast to this a single redundant set of Inertial Measurement Units provides attitude and acceleration knowledge during propelled or highly dynamical phases (trajectory corrections during cruise, aerocapture, science orbit acquisition and maintenance). The IMUs are placed within the Orbiter. Cruise phase navigation is performed by Deep Space network measurements and approach navigation is supported by an optical navigation camera.

Attitude control is performed by means of reaction wheels during all mission phases except aerocapture and trajectory maneuvers. The wheels are implemented inside the orbiter in a tetraeder configuration.

Dedicated thrusters on each spacecraft element provide reaction wheel desaturation and trajectory control capabilities. The AS thrusters are used for trajectory control during the atmospheric pass.

Thermal Control System

Work is still on-going and the design of the thermal control system has not yet started. However, system complexity should be as low as possible and a passive system is targeted.

Redundancy Concept

It is important for the success of the mission that a failure in one subsystem component does not lead to a failure of the subsystem, especially in critical phases of the mission.

	Cruise	Aerocapture & orbit acquisition	Orbit
AOCS	***Medium***	***High***	***Medium***
Propulsion	***Medium***	***High***	***Low***
Power	***Medium***	***High***	***Medium***
OBDH	***Medium***	***High***	***Medium***
Communication	***Low***	***Low***	***Medium***

Table 2 Criticality of spacecraft functions to recoverable failures

Table 2 gives an overview about the criticality of spacecraft functions within the different mission phases. The failures are assumed to be recoverable within a limited time span, but not immediately. As can be seen from the table different redundancy requirements can be derived from each phase. It is required that any medium or high critical function is being one-failure tolerant.

In order to guarantee the required one-failure tolerance some components are implemented redundantly. Figure 16: shows that some components are implemented two times. A second option is to implement them internally redundant. Within highly critical situations the components will work in hot redundant mode, while in medium and low critical situations cold redundancy is acceptable.

Mass Breakdown

The resulting mass breakdown of the AEROFAST spacecraft is given in *Table 3* for both system options.

	Total Mass [kg]	
	Apollo-like Design	Bi-conic Design
Interplanetary Cruise Module	228	228
Aerocapture System	336	357
Mars Orbiter	598	598
Total	1162	1183
Resulting system margin [%]	15.3	13.3

Table 3 Mass breakdown of the AEROFAST spacecraft

The Soyuz-Fregat launch vehicle has the capability of injecting 1450 kg into Mars transfer orbit. This includes the launch vehicle adapter of the cruise stage weighing 110 kg. Hence the total AEROFAST spacecraft is allowed to have a mass of 1340 kg including system margin.

5. NEXT STEPS

The AEROFAST project will last 2 and a half years. The project activities have started beginning of 2009. During the first year, a detailed definition of the system requirements was performed.

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 lay-out. 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.

Next steps linked to Obj1 will include the selection of the aeroshape, the detailed spacecraft architecture.

Regarding the objectives Obj2 and Obj3 the following steps are currently being performed:

- 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 next steps will be the selection of the most promising TPS candidates.

Objective Obj5 will progress with the identification and definition of the sensors necessary to perform measurements during the aerocapture flight.

6. 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.

The frame of the mission has been defined: low cost mission, Soyuz-like Launch performance, optimized entry conditions at 120 km, main characteristics of the spacecraft (three modules vehicle concept).

The three main phases of the aerocapture maneuver have been assessed:

- Pre-aerocapture phase: Challenge is to master the attitude/position of the S/C before maneuver.
- Main aerocapture phase: Goal is to reach an elliptical orbit within a narrow corridor and challenge is to sustain important heat loads.
- Post aerocapture phase (transfer to a parking orbit): to target a quasi circular sun-synchronous orbit at a low altitude.

The expected results of the AEROFAST project are:

- Further assessment of the design of the composite:
  - Best aerodynamic shape for aerocapture,
  - Need of a protection for the rear part of the composite to protect the equipment behind the heat shield,
  - Overall design and lay-out.
- Identification of the instrumentation needed during the aerocapture phase: What phenomena shall be observed in the aerocapture path? Add scientific objectives when the spacecraft is on the final orbit
- Identification of innovative thermal protection systems (TPS): low density while sustaining high heat loads:
  - Assessment of TPS options based on Cork (New high temperature resins, Possible fillers to use, Moulding possibilities, Surface treatments),
  - Assessment of non-ablative TPS.
- Improvement of the navigation sub-system for the pre-aerocapture phase and test of all algorithms through real time and non-real time breadboards.

AEROFAST CONSORTIUM

The AEROFAST consortium is composed of the following partners:

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

Organisation of the tasks:

WORKPACKAGES	WP leader	Participants
WP1 Aerocapture overall design	AST-F	SAMTECH DEIMOS FCUL UNIROME IoA ONERA
WP2 Spacecraft design	AST-D	AST-F DEIMOS UNIROME
WP3 Innovative thermal protections	AMORIM	AST-F AST-D
WP4 O/B GNC function development & validation	DEIMOS	FCUL AST-F
WP5 On-board instrumentation	UNIROME	STIL-BAS SRC-PAS
WP6 Overall coordination & management, and dissemination	AST-F + KYBERTEC + Work package leaders	

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