Aerospace Laboratory for Innovative components S.C.a r.l.	**SMS** (Small Mars Satellite)

Doc. Type: Technical Note	*ESA Contract No. 4000115306/15/NL/LF/as*		*DRD N°:* EXS	
Doc. N°: **ALI EXS SMS ESA**	*Issue:* **1.1**	*Date* **12-05-2016**	*Page* **1** *Of* **23**	
Title : Executive Summary				

	Name & Function	*Signature*	*Date*
Prepared by:	F. Causa N. Cimminiello D. De La Torre E. Fantino L. Gramiccia M. Grassi C. Molfese P. Pasolini F. Punzo		
Approved by:	F. Punzo (PM)		
Application authorized by:	G. Squame		
Customer / Higher Level Contractor			
Accepted by:			
Approved by:			

DISTRIBUTION LIST	N	A	I
Internal			
R. Aurigemma P. Dell'Aversana E. Fantino L. Gramiccia M. Grassi F. Punzo C. Molfese P. Pasolini F. Causa D. De La Torre	1		X
External			
I. Roma (TEC-SYE) L. Fabrizi (IPL-PTE)	2		X
N=Number of copy A=Application I=Information			

Data Management: Francesco Punzo 12/05/2016 *Signature* *Date*	File: ALI EXS SMS ESA Issue 1.1 – Executive Summary

CHANGE RECORD			
ISSUE	*DATE (dd/mm/yyyy)*	*CHANGE AUTHORITY*	*REASON FOR CHANGE AND AFFECTED SECTIONS*
1	06/05/2016	F. Punzo	First issue of the document: all sections affected
1.1	12/05/2016	F. Punzo	Annex 2 added

Table of Contents

1. ACRONYMS

AD	Applicable Document
ALI	Aerospace Laboratory for Innovative components
ASI	Italian Space Agency
ATP	Authorization to Proceed
CLGA	Cruise Low Gain Antenna
COTS	Component Off The Shelf
DHS	Data Handling System
DII	Dipartimento di Ingegneria Industriale (Univ. “Federico II”- Naples)
DOD	Depth Of Discharge
EDAC	Error detection and correction
ESA	European Space Agency
ESTEC	ESA Technical Centre
G/S	Ground Station
GNC	Guidance, Navigation and Control
HGA	High Gain Antenna
HW	HardWare
IDM	Integrated Design Model
IF	Interface
IMU	Inertial Measurement Unit
LEO	Low Earth Orbit
LLGA	Low Gain Antenna
MEMS	Micro Electro Mechanical Systems
MGA	Medium Gain Antenna
MPU	Multi Purpose Unit
OCP	Over Current Protection
OVP	Over Voltage Protection
PA	Product Assurance
PL	PayLoad
RD	Reference Document
SDST	Small Deep Space Transponder
SoW	Statement of Work
SSMM	solid-state mass memory
SSPA	Solid State Power Amplifier
TBC	To Be Confirmed
TBD	To Be Defined
TN	Technical Note
TO	Technical Officer
TPS	Thermal Protection System
TRL	Technology Readiness Level
UHF	Ultra-High Frequencies
WP	Work Package (hereinafter used as a synonymous of “Task”)

2. APPLICABLE DOCUMENTS

AD 1: Request for Proposal (ESA/RFP/NC/IPL-PTE/LF/as/827.2015)

AD2: Tendering Conditions for Express Procurement Procedure (Appendix 3 to RFQ ESA/RFP/NC/IPL-PTE/LF/as/827.2015)

AD 3: General Conditions of Tender for ESA Contracts

AD 4: Statement of Work (Appendix 1 to RFQ ESA/RFP/NC/IPL-PTE/LF/as/827.2015)

AD 5: SMS (Small Mars Satellite) Proposal, 14/10/2015

AD 6: Contract ALI/ESA N. *4000115306/15/NL/LF* – SMS (Small Mars System)

AD 7: IRENE-TN-ALI-004 Issue 8, “Spacecraft Architecture”, 06-07-2011

3. REFERENCE DOCUMENTS

RD1: ALI TN1 SMS ESA “Characterization of the application scenario” Issue 4.1

RD2: ALI TN2 SMS ESA “Technical Requirements” Issue 3.1

RD3: ALI TN3 SMS ESA “Payload Preliminary Analysis” Issue 1

RD4: ALI SMS ESA TN4 "Preliminary System Design"

RD5: ALI TN5 SMS ESA “Development Plan” Issue 1

4. INTRODUCTION

4.1 SCOPE OF DOCUMENT

This document has been prepared in the framework of the ESA Express Procurement (EXPRO) on the activity “Small Mars System - SMS”. It summarizes the work carried out as part of the SMS project.

5. ASSESSMENT OF LAUNCH PROFILE, INTERPLANETARY TRAJECTORY AND MARS APPROACH

The activities carried out in the phase 0 deal with the mission concept and, in particular, with the interplanetary analysis and contributions to the mission requirements. The activities can be summarized as follows:

- study of the state of the art of Mars landers, with focus on issues such as the launcher, the interplanetary trajectory, the type of planetary approach, the design of the spacecraft, the layout, the mass and power budgets, the aerodynamic parameters, the entry, descent & landing strategy;
- study of the interplanetary trajectory (transfer orbit, ΔV budget) aiming at identifying the minimum-cost options and providing indications of launch dates, transfer times and arrival dates;
- evaluation of options for the launch vehicle: cost, available volume, launch configuration, launch profile and escape velocity;
- analysis of the characteristics of the arrival trajectory with definition of the entry point and velocity.

The study of the launch opportunities was carried out on the interval of dates 2020-2024 for the launch. By the analysis of the porkchop plots, the cheapest option was identified in the 2024 window. It is a direct trajectory of type II with launch on October 2, 2024 and arrival on September 1, 2025. The transfer takes 334 days and implies a transfer angle of 209.3° (Figure 1). The ΔV budget is given by a departure C_3 of 11.316km²/s² and a Mars v_∞ of 2.455km/s. This solution maximizes launch mass.

After evaluation of the performance of the three European launchers, considerations of cost and flexibility requirements (which favour the option of the individual launch), led to the identification of VEGA as the most suitable carrier rocket. Performance of VEGA for Earth escape requires the adoption of an *ad hoc* kick stage or propulsion module, as done by LISA PTF. This limits launch mass to approximately 320 kg at a C_3 = 11.3 km²/s².

Arrival at Mars on a direct hyperbolic trajectory in the middle of the Martian summer (solar longitude L_s=135°) identifies the equatorial regions as the best candidates for the landing site, as far as ground and atmospheric temperatures and weather conditions (specifically, dust storm risk) are concerned.

The requirement of entering the atmosphere (125 km) with a flight path angle between -14° and -12° reflects into the requirement for a targeting maneuver (to be executed at the sphere of influence of the planet, or even further out) of magnitude between 33 and 50 m/s. Such maneuver delivers the spacecraft at the top of the atmosphere with an entry speed of 5.5 km/s. The targeting maneuver is the main item in the ΔV budget of the propulsion system of the cruise stage (i.e., the onboard propulsion). Such budget is estimated to be close to 100 m/s (including margins).

Mass and performance of the propulsion system of the cruise stage have been estimated based on a rescaling (found in the literature) of systems embarked on previous Mars landers. A 230s specific impulse monopropellant system would correspond to a mass ratio of 1.045. For a lander mass of 200kg and a cruise stage of 80kg, the resulting propellant budget would be of 13kg yielding a total mass of 293kg.

Data collected by previous missions during the interplanetary cruise and on the Martian ground indicate that the risk of impact by micrometeoroids is negligible, whereas little or no information have been found so far concerning the radiation environment during the cruise. Operations on the surface of the planet are limited to a few days, which makes the particle radiation issue irrelevant.

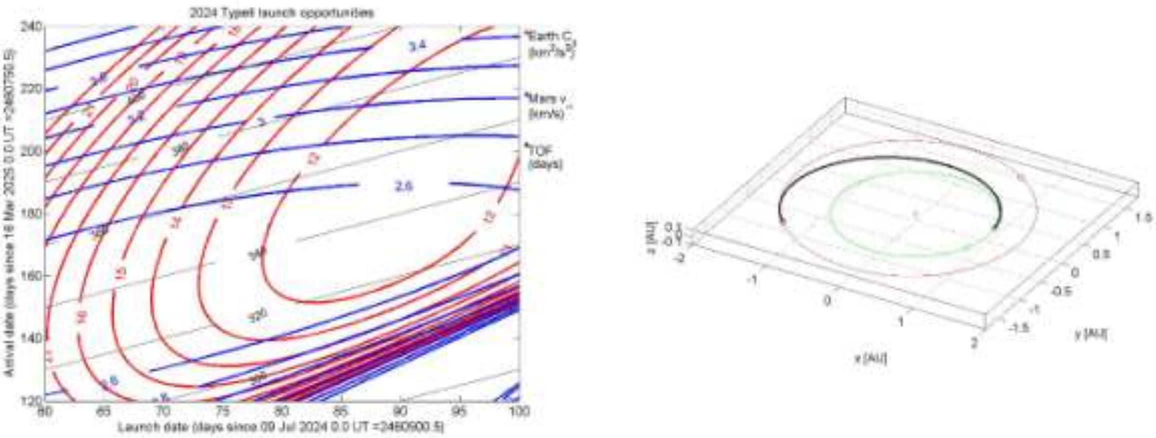


Figure 1 - The porkchop plot of the 2024 opportunity (left) and the trajectory with the lowest launch cost (right).

6. EDL

This part of the work has provided the definition of the capsule's entry trajectory and a preliminary assessment of the characteristics of the flight and the trajectory of the vehicle during the atmospheric entry until landing. Velocity, Mach number, deceleration and pressure profiles have been computed over the entry trajectory using 3 DOF models for entry trajectories evaluation. Heat fluxes have been estimated along the same trajectories using engineering formulations.

To this purpose, the EDL analysis phase can be summarized as follows:

- Definition of a 3 DOF model for the entry trajectory evaluation that takes into account the Martian atmosphere using the most advanced atmospheric models.
- Parametric analysis of the velocity, the Mach number, the deceleration, the stagnation point pressure and the heat flux profiles at Mars atmospheric entry, as functions of the ballistic parameter and flight path angle, aiming at assessing the maximum design parameters.
- Comparison between a baseline entry trajectory for SMS and those of past landers with respect to velocity, Mach number, pressure and heat flux profiles.
- Trade-off analysis aiming at the definition of the parachute dynamics.

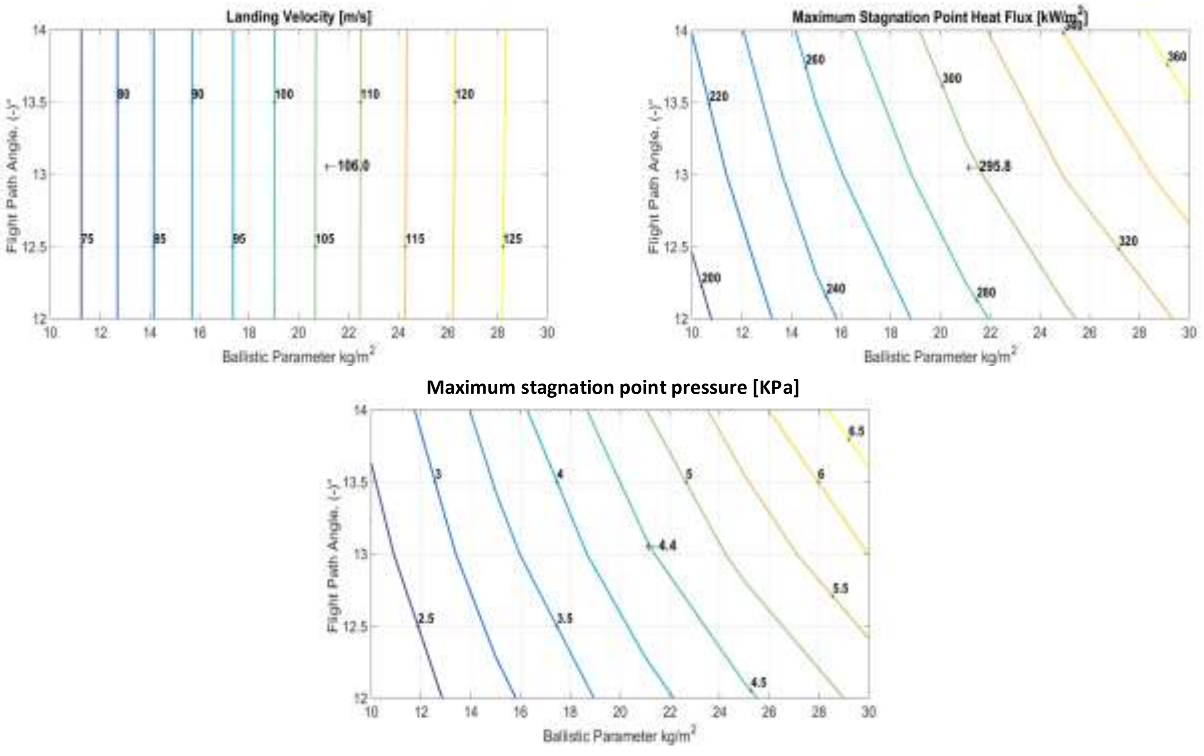


Figure 2 - Landing velocity, stagnation point heat flux and stagnation point pressure as functions of the ballistic parameter

It is a well-known fact (see also Figure 2) that the variation of the maximum stagnation point heat flux, the maximum stagnation point pressure and the landing velocity strongly depend on the variation of the ballistic parameter.

Furthermore, for the case of ballistic parameter of 21 kg/m², flight path angle (FPA) of -13°, entry velocity of 5.5 km/s, entry altitude in the Mars atmosphere of 125 m, the main data of the Mars entry trajectory have been evaluated and compared with those of previous Mars missions. As shown in Figure 3, the stagnation point pressure for the SMS system is more than 2.5 times smaller than other EDL systems that performed Mars atmospheric entry.

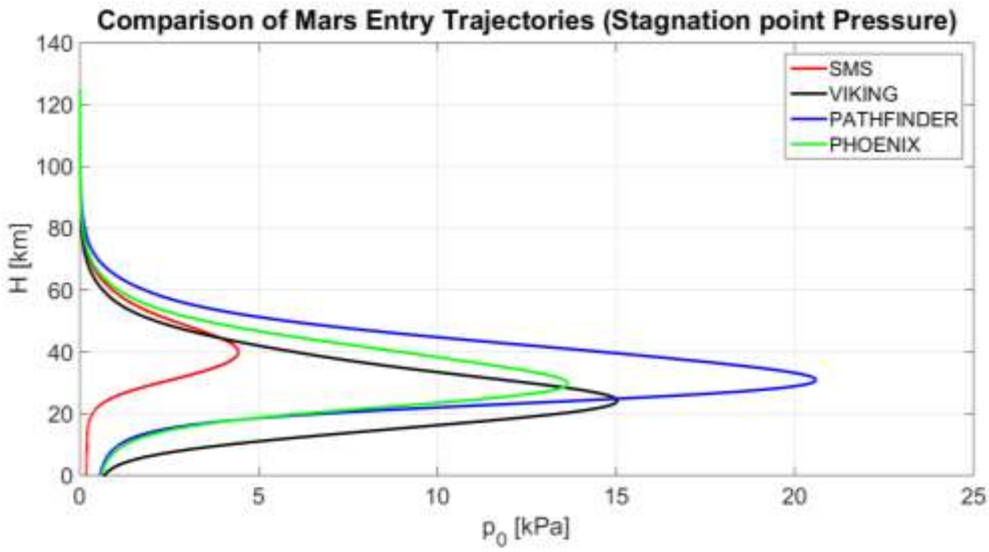


Figure 3 - Comparison of Mars entry trajectory stagnation point pressures

Finally, a parachute analysis has been carried out in order to assess the SMS performances in terms of landing capability, using a single subsonic parachute. In order to take into account the possibility of the TPS ejection during the entry trajectory, after the parachute deployment, an analysis of the variation of the landing speed as a function of the lander mass, considering four diameters, has been developed (Figure 4).

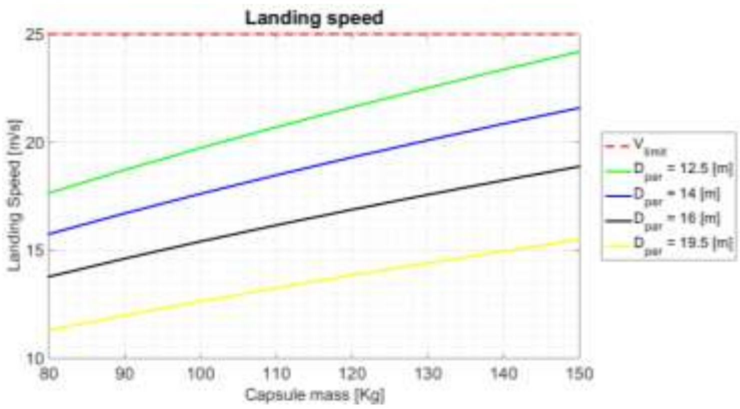


Figure 4 - Landing speed

7. AEROTHERMODINAMICS ANALYSIS

In the framework of the "Phase 0" of the SMS project, the University of Naples carried out the aerothermodynamic analysis for the selected capsule configuration [AD7]. The scope of this part of the work is to evaluate the aerodynamic and thermal field around the entry capsule and, in particular, on the flexible (deployable) and rigid elements of the thermal protection surface.

The analysis conducted can be summarized as follows:

- Development of the CFD and DSMC 2D fluid dynamic models taking into account the SMS geometry, the Martian atmosphere and the entry path.
- Collecting CFD simulation results at critical points along the entry trajectory in transitional and continuum regime.
- Collecting DSMC simulation results at critical points along the entry trajectory in transitional and rarefied regime.
- Development of the CFD 3D fluid dynamic models taking into account the SMS geometry, the Mars atmosphere and the entry path at maximum stagnation point pressure and maximum stagnation point heat flux conditions.
- Collecting CFD simulation 3D results at critical points along the entry trajectory in order to evaluate the pressure and temperature distribution on the capsule surface.

State of the art computer codes have been selected for the different flow regimes. In continuous regime, they provide the solution of the classical Navier-Stokes and energy equations. In rarefied regime, however, the study of the aerodynamic characteristics require usage of a "Direct Simulation Monte Carlo (DSMC) method".

In particular, two-dimensional axisymmetric analyses have allowed to estimate the distribution of the thermal and mechanical loads on the surface of the capsule, in order to compare the results obtained by the EDL phase. The three-dimensional analyses have allowed to calculate the values of temperature and pressure distribution on the three-dimensional capsule baseline geometry [AD7] at the most severe conditions along the capsule entry trajectory.

As shown in Figure 5, there is good agreement in terms of stagnation point pressure prediction. This means that the Newton theory is a valuable tool to predict the stagnation point pressure for a blunt body in hypersonic flow field.

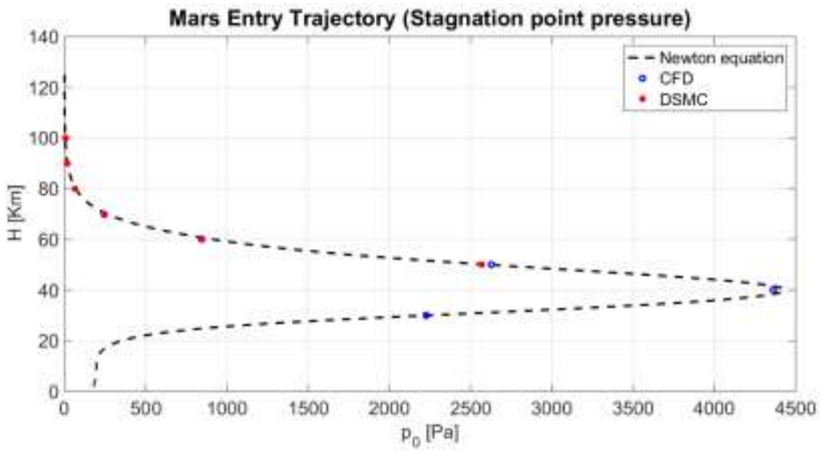


Figure 5 - CFD, DSMC and Newton theory comparison results

Furthermore, Figure 6 shows the pressure and temperature distribution, respectively, at maximum stagnation point pressure condition and maximum stagnation point heat flux condition along the capsule entry trajectory.

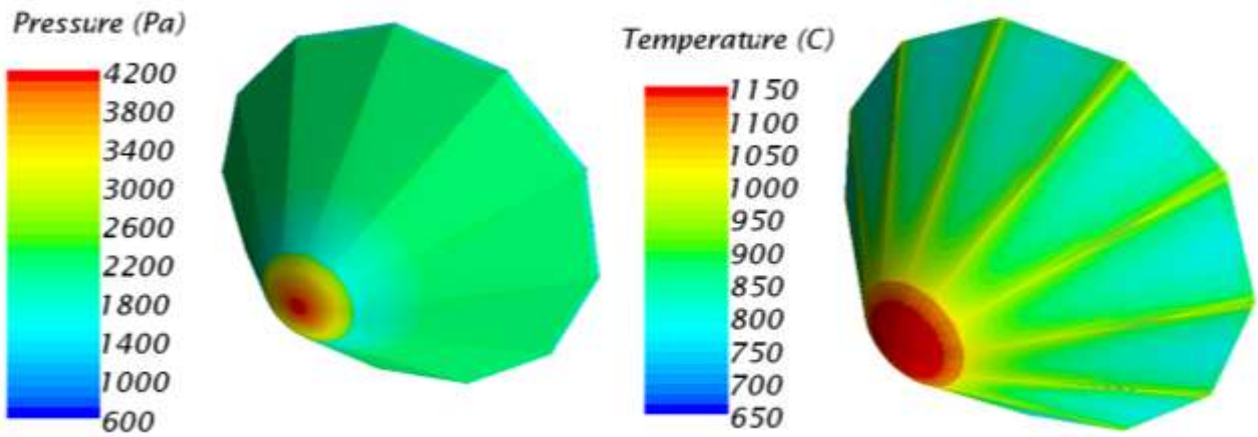


Figure 6 - Pressure distribution at maximum stagnation point pressure condition (at 40 Km-left) and temperature distribution at maximum stagnation point heat flux condition (at 50 Km-right)

8. Electronic Subsystem

TTC&C subsystem & Link Budget

The competing power and telecom needs are major factors in the design trades in cruise stage orientation that must be made. The telecom and power/thermal constraints are linked by the Sun–Probe–Earth (SPE) angle. Figure 7 shows the spacecraft-Earth range (d) on the left vertical axis and the SPE angle on the right vertical axis. The type II trajectory initially has very high SPE angles, large SPE angles are a challenge for telecom, even at relatively short ranges. Power output and thermal considerations require the solar array to be pointed within an optimum range of angles from the Sun. While Sun-Earth distance remains short, the solar array should be pointed not too close to the Sun, to avoid overheating the solar panels and losing efficiency. During cruise, communications are guaranteed via the Cruise LGA (CLGA) or the Medium-Gain Antenna MGA. The CLGA served for the first few weeks after launch and for some TCMs, while the MGA provided added capability as the Earth-to-Mars distance increases.
During entry, descent, and landing (EDL), the cruise stage is jettisoned; the SMS lander continued to communicate via an X-band downlink to the DSN, and it can initiate a UHF return link to an existent orbiter (to define).

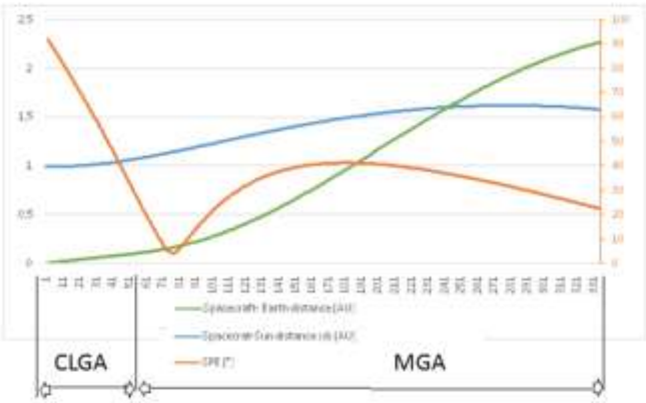


Figure 7 - Communication strategy during cruise phase.

Communication during EDL is required to provide information to help reconstruct a fault should one occur. The LGAs available during EDL accommodated wide variations in orientation. During EDL, the X-band system transmits multiple-frequency shift-keying (M-FSK) tones or semaphores, indicating the spacecraft state and completion of major EDL phases.
During the primary and extended surface missions, the X-band transponder has been supported by either an HGA or the LLGA. The LLGA has provided near omnidirectional coverage for both command and low rate telemetry data. The HGA is a steerable, flat-panel, phased array, providing high-rate reception of command and transmission of telemetry data. During the surface missions, the uplink and downlink rate-capability via the HGA has depended on the Mars-Earth distance. In addition to the X-band system, the UHF system can be also used for the portion of EDL where the lander was suspended on the bridle. On Mars' surface, instead, UHF system is used to communicate with drone during its mission and as backup communication system (in the case of X-band communication is not available).

A preliminary link budget was performed for X-Band subsystem, considering as ground station the Deep Space Network and following combinations:

- 2,26 UA Spacecraft-Earth distance for LLGA, HGA MGA
- 0,1 UA Spacecraft-Earth distance for CLGA

Link was closed in all combination, with required margin.

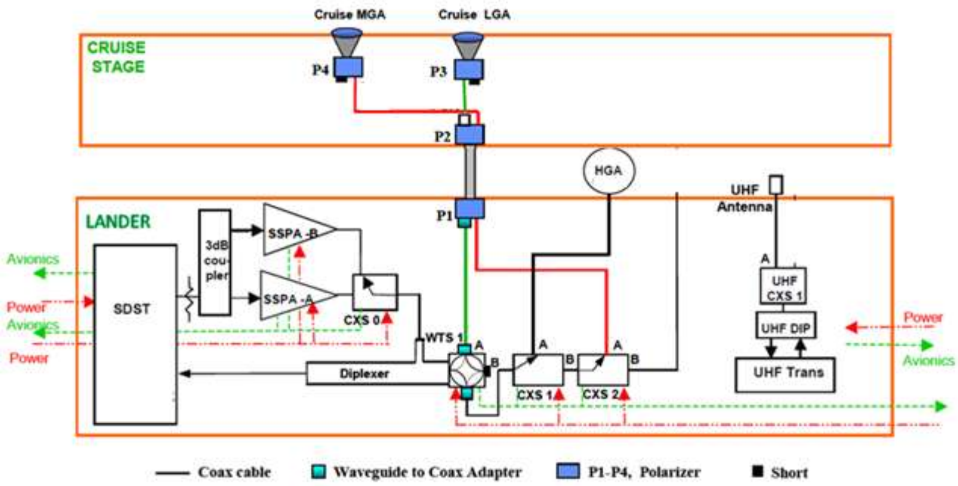


Figure 8 - TT&C system preliminary layout

Preliminary Electronic subsystem layout

The OBDH represents the core element of the system. It interfaces all the other flight and ground subsystems. Preliminary design of the OBDH has been carried on by taking into account a number of guidelines which are partly dictated by the mission requirements and partly by the adherence to recognized standards/guidelines issued by the European Space Agency, as, for example, the Reference Architecture for Space Avionics. Hence the main design drivers can be summarized under the following areas:

- Accommodation inside a reduced volume with a reduced mass and power available
- High reliability to cope with the external environment and the long mission duration
- Performances in line with the requirements of the data handling processes and algorithms foreseen for all the phases of the mission
- Adoption of a flight proven software infrastructure in terms of Board Support Package (BSP), Real-Time Operating System (RTOS) and qualified development and testing tools for the realization of the flight software

Due to the reduced resources available, it is proposed an advanced OBDH that would integrate the GNC functionalities together with the classical functions of a space OBDH. Such a solution is in line with the current trend in the space systems to adopt **Integrated Modular Avionics** (IMA) architectures, which are commonly used in civil avionics. This was made possible, in particular, by the availability of space qualified components, e.g. FPGA, that allow to integrate several functions into a single device thus permitting a drastic reduction of volumes and mass without penalizing, rather increasing, the overall performances. The OBDH proposed will make use, as indicated in the following, of several space grade reprogrammable devices that integrate the selected space processor, the communication interfaces and can serve as hardware accelerators for particularly intensive navigation tasks. Several mitigation techniques will be adopted for the design of the OBDH:

- Redundancy of all crucial circuits and interfaces like the TM/TC I/F
- Memory protected with EDAC mechanisms
- Latch-up protection systems
- Distributed functionality

The following block diagram shows the decomposition of units with the indication of the main interfaces and data and control paths.

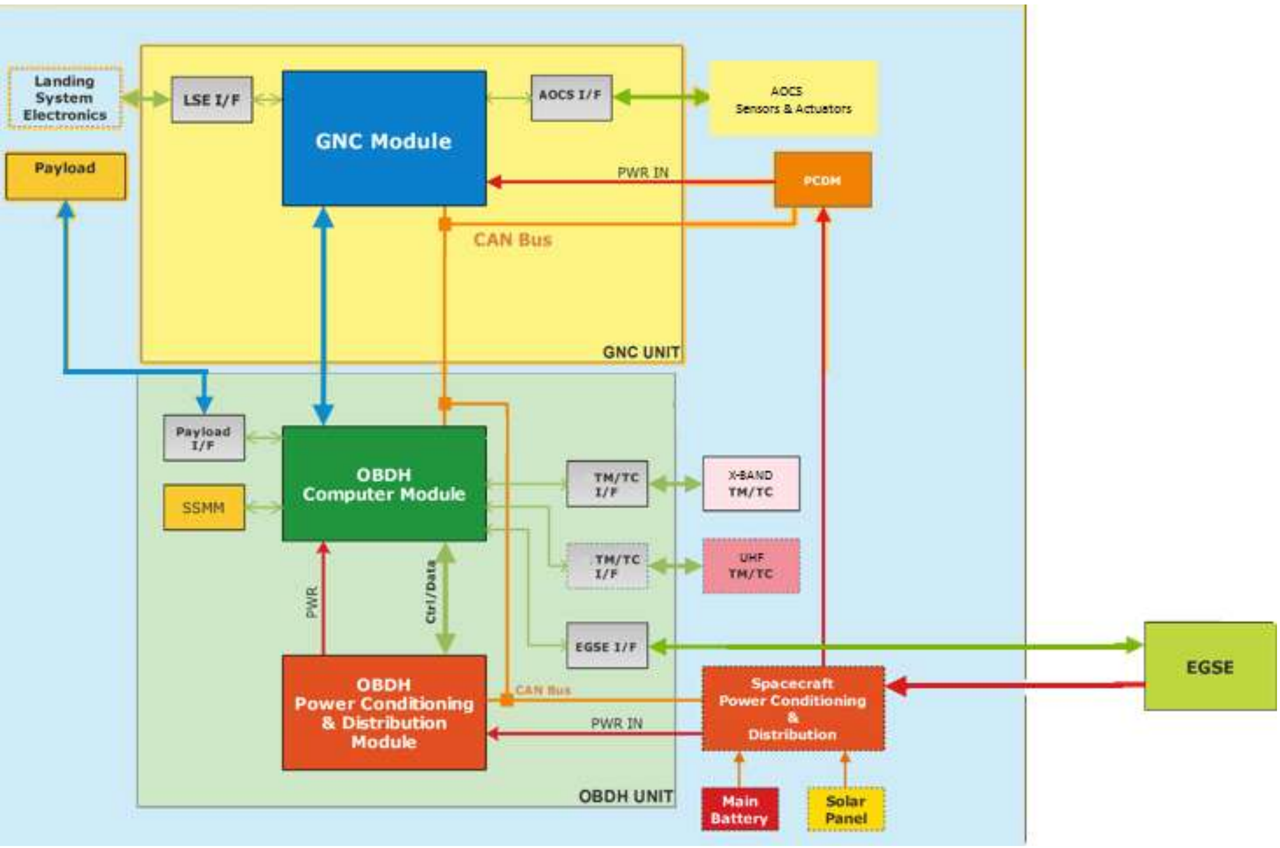


Figure 9 – Electronic main function

9. PAYLOAD SYSTEM 1: DRONE

The possibility of embarking an aerial drone, aiming at demonstrating the capability of flight through the Martian thin atmosphere, was evaluated. Specifically, the following drone objectives were established:

- Demonstrating the feasibility of atmospheric flight on Mars
- Demonstrating the capability of collecting low-altitude, high-resolution images of the Martian surface
- Demonstrating the capability of using the lander as a ground control station for drone operation
- Demonstrating the drone multi-mission capability.

Based on these objectives, following a comprehensive analysis of past aerial drone projects and of SMS mission and system requirements, the best drone configuration was selected. The proposed configuration consists of an electrically powered rotorcraft (see Figure 11), since it offers a vertical take-off and landing capability, being thus compliant with the multi-mission requirement. Specifically, the capability of vertical landing at the Lander to recharge the on-board battery was investigated. To this end, a charging PAD exploiting inductive technology (typically used for terrestrial drone recharging) embedded in the upper surface of the Lander is foreseen.

Two drone configurations have been designed, the last being a scaled version of the first configuration. This was done in response to the need of reducing the overall system launch mass to comply with the maximum capability of VEGA for interplanetary missions.

While the first configuration was able to perform 2 5-min flight around the landing site and required about 7 sols to recharge the batteries, the scaled configuration allows only 7-min flights and takes about 1 sol to recharge the battery. Despite the shorter flight duration, the mission goals can be demonstrated also with the scaled configuration. The maximum altitude that can be reached by the drone is 100 m.

Drone avionics design was made with an aim at reducing mass and power requirements and performing autonomous flight. To this end, the avionics include MEMS inertial sensors, a downward looking micro camera, micro solar sensors, a processing unit with sufficient data storage, and a miniaturized RF module with antennas. All these components are COTS developed for terrestrial drones but capable of operating over extended temperature ranges.

The definition of the requirements in terms of mass, power, flight duration and mission profile (see Figure 12), as well as the accommodation solution, were defined at a preliminary level to guarantee the correct operation of the drone. The release sequence was also investigated.

The drone conceptual design was carried out with attention to the following issues:

- overall configuration (blades and main body sizing)
- mass budget
- avionics architecture

- thermal control budget
- Accommodation envelope
- power and energy budget (based on mission profile)
- TT&C with the Lander

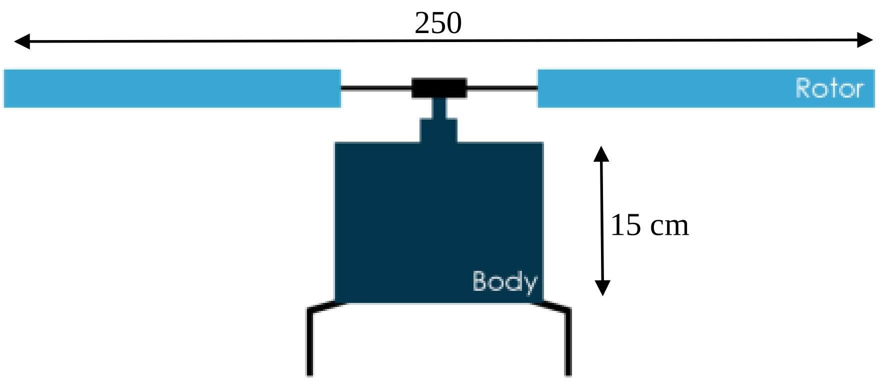


Figure 10: Drone concept (not to scale).

Parameter	Mass, kg
Fuselage & landing Gear	1.2
Drive & Hub	1.0
Power Supply	0.5
Blades	2.5
Avionics, Payload & TMS	1.8
Total Mass	7.0

Table 1: Drone mass breakdown

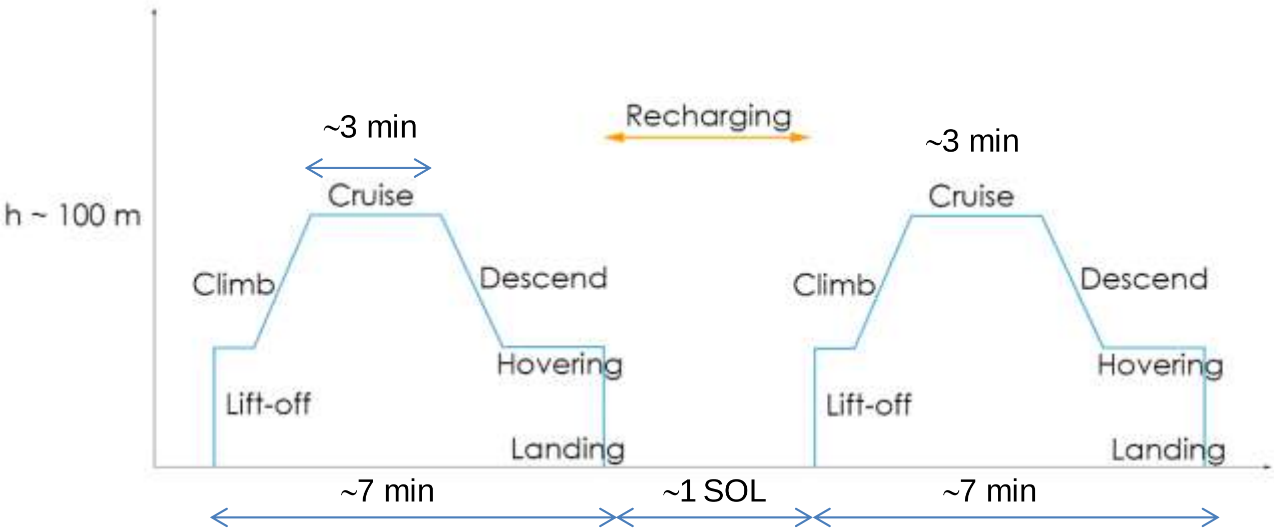


Figure 11: Drone Mission Profile

10. Payload System 2: Dust particle analyser concept design

The activities of evaluating the possibility to embark a scientific instrument for dust analysis in the Martian atmosphere were carried out.

The scientific objectives for the measurement of dust in the Martian atmosphere were given. Based on these objectives, several instruments, working with different measurement principles, were studied with the aim to identify the most suitable to achieve the scientific goals. The possibility to measure the dust abundance before landing (i.e., during the descendent phase of SMS) and after landing was taken into account as well.

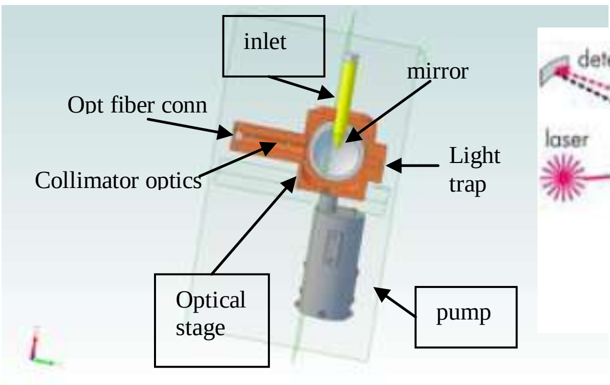
The possibility to accommodate the chosen instrument, i.e. a Dust Particle Analyser (DPA), onboard the drone was evaluated, but discarded for the poor scientific return compared to the complexity and criticalities of this solution.

The definition of the requirements for mass, power, measurement performance of the DPA and the definition of the accommodation requirements, to be fulfilled in order to guarantee the correct operations for the instrument, were made.

The mission scenario for the DPA was analysed and reported.

The conceptual design of the DPA was carried out attention to the following technical areas:

- Mechanical design and interfaces
- electrical design and interfaces
- External harness and connectors
- Data handling design and interfaces
- Heritage and maturity
- Accommodation envelope
- Mass budget
- Power and energy budget
- Data management
- Telemetry and commands



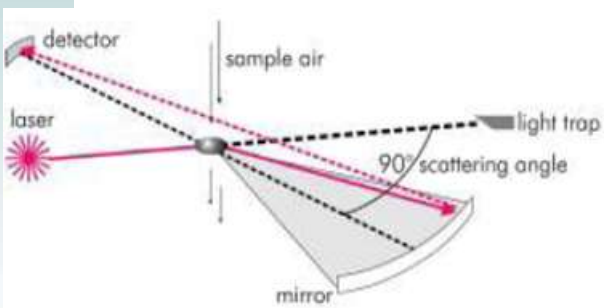


Figure 12: (a) Dust Particle Analyser concept; (b) measurement principle.

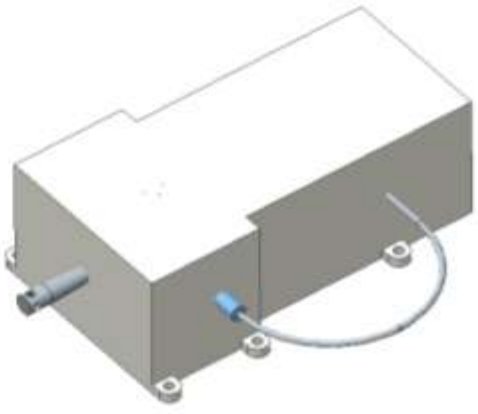


Figure 13: Instrument assembly

11. SMS PRELIMINARY DESIGN

The preliminary design of the overall satellite system was carried out. Particular attention was devoted to the Mars entry protection system, i.e. the satellite structural and thermal subsystem crucial for Mars atmosphere entry, descent and landing. In addition, technical solutions were outlined to embark and deploy the identified payloads.

Starting from the characteristics of the reference mission the following tasks were executed:

- A reference lay-out of the capsule;
- The kinematic for the deployment system;
- The initial forms of the flexible structure;
- Aerobraking system simulations
- Loads on structure.
- Preliminary definition of the capsule mechanical structure;
- Positioning of the main components inside the capsule

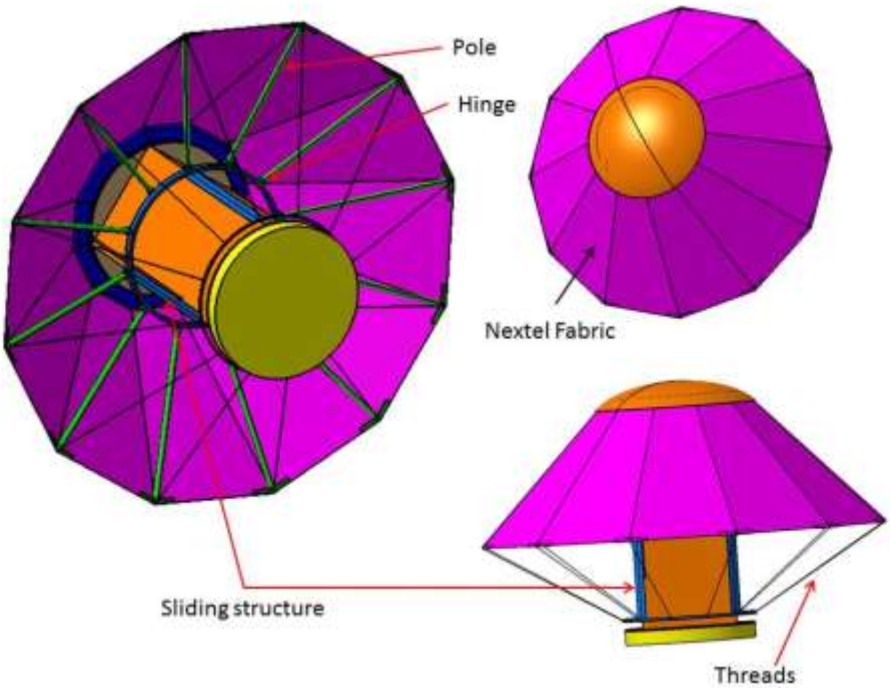


Figure 14 – Aerobraking system (deployed configuration)

12. ANNEX 1 "MASS BUDGET"

SMS MASS BUDGET			Quantity	TRL	Mass [Kg]	Mass x quantity	Design Maturity Margin	Total mass (margin included)	References
CRUISE MODULE	PROPULSION SYSTEM		1	4 o 5	6,5	6,5	1,2	7,8	MOOG Monopropellant Thrusters datasheet Rev 0613
	ATTITUDE DETERMINATION SYSTEM		1	8 o 9	0,6	0,6	1,05	0,6	Star tracker RT 2015a datasheet Sun sensor 2011a datasheet
	TELEMETRY, TRACKING & COMMAND SYSTEM		1	8 o 9	1	1	1,05	1,1	RF3 - pag 34: [illegible]
	THERMAL SYSTEM		1	4 o 5	9	9	1,2	10,8	RF1 - pag.6 [illegible]
	POWER SYSTEM		1	4 o 5	6	6	1,2	7,2	RF1 - pag.6
	STRUCTURE & MECHANISMS		1	4 o 5	36	36	1,2	43,2	RF1 - pag.6
	SUBTOTAL 1					59,1		70,7	With subsystem margin
	HARNESS		1	4 o 5	3,53	3,53		3,53	(5% of total dry mass with margins)
	CRUISE MODULE DRY MASS					62,6		74,2	
LANDER	TT&C X-BAND SYSTEM	SDST (small deep space	1	8 o 9	3,20	3,20	1,05	3,4	X-band_SDST_DsS-813-12 datasheet
		SSPA (solid state power amplifier)	2	8 o 9	1,30	2,60	1,05	2,7	RF3 - pag 34
		HGA (high gain antenna)	1	8 o 9	1,10	1,10	1,05	1,2	RF3 - pag 34
		LLGA (high gain antenna)	1	8 o 9	0,78	0,78	1,05	0,8	RF3 - pag 34
		other passive comp.	1	8 o 9	1,54	1,54	1,05	1,6	RF3 - pag 34
	TT&C UHF-BAND SYSTEM	UHF Antenna	1	8 o 9	0,10	0,10	1,05	0,1	RF3 - pag 34
		other passive comp.	1	8 o 9	0,78	0,78	1,05	0,8	RF3 - pag 34
		UHF transceiver	1	8 o 9	2,00	2,00	1,05	2,1	C/TT-505 CMC UHF-transreceiver
	THERMAL SYSTEM		1	4 o 5	6,68	6,68	1,15	7,7	RF1 - pag.11
	POWER SYSTEM		1	4 o 5	14,03	14,03	1,2	16,8	See Power budget
	COMMAND & DATA HANDLING SYSTEM		1	4 o 5	2,60	2,60	1,2	3,1	RF1 - pag.11
	GNC	sun sensor	2	8 o 9	0,04	0,07	1,05	0,1	Sun sensor 2011a datasheet
		IMU	1	4 o 5	0,75	0,75	1,05	0,8	LN-2005_IMU_RAD_HARD Northrop Grumman datasheet
		RADAR ALTIMETER	1	8 o 9	1,40	1,40	1,05	1,5	HG8500_radar_altimeter_Honeywell datasheet
		CAMERA	3	8 o 9	0,30	0,90	1,05	0,9	RF1 - pag.33
	STRUCTURE & MECHANISMS		1	4 o 5	19,00	19,00	1,2	22,8	Rif. ALI SMS TN4 + antenna gimbal assemblies MSL
		Drone	1	4 o 5	6,10	6,10	1,2	7,3	RF4 - TN4
		Dust analyzer	1	4 o 5	0,48	0,48	1,2	0,6	Dust particle analyzer mass budget
	PARACHUTE SYSTEM		1	4 o 5	8,70	8,70	1,2	10,4	Rif. email ESA 19 February 2016. Value estimated as 7% Entry Mass.
	SOFT LANDING SYSTEM		1	4 o 5	16,50	16,50	1,2	19,8	RF2 - pag.15 [illegible]
	SUBTOTAL 1					89,3		104,5	With subsystem margin
	HARNESS		1	4 o 5	5,2	5,2		5,2	(5% of total lander mass with margins)
	LANDER MASS					94,5		109,8	
SHIELD	STRUCTURE		1	4 o 5	30,58	30,58	1,2	36,7	RF4. ALI SMS TN4 "Preliminary System Design"
	NOSECONE		1	4 o 5	8,5	8,5	1,2	10,2	RF4. ALI SMS TN4
	TPS (thermal protection system)		1	4 o 5	8,5	8,5	1,2	10,2	RF4. ALI SMS TN4
	SHIELD MASS					47,58		57,1	With subsystem margin
SUBTOTAL						204,7		241,1	
SYSTEM MARGIN 20%								48,2	
SYSTEM DRY MASS								289,3	
Propellant						14,31	1,03	14,7	[illegible]
LAUNCH MASS								304,0	

Launcher Capacity: 320 Kg.

Launcher Margin: 5 %

13. ANNEX 2 “POWER BUDGET”

Power budget was performed according following hypothesis:

- Mission phases in which is possible to use solar energy are orbit raising, cruise and Martian operations. For this phase an evaluation of solar irradiance was performed.
- System is composed by two solar array: Cruise stage’s solar array, Lander’s solar array, working on Mars surface, while is folded during other mission phases.
- Lander’s solar array sizing is performed considering a part of the four deployed panel’s surface (1m x 0,6m).
- Cruise’s stage solar is designed to satisfy spacecraft power requirement at minimum of estimated solar flux during cruise.
- In the first two phases only cruise stage's solar array can work, while lander's solar array is folded and covered by umbrella's fabric.
- During EDL cruise stage is released, and only lander's solar array is available on planet surface.
- Cruise stage can use lander's battery before separation.
- As baseline we considered worst power consuming scenario for communication (Earth link via X-Band subsystem), while UHF subsystem is used only during drone mission.
- During “night survival" and launch lander’s batteries are the only energy source.

Margin policy is applied according "Margin philosophy for science assessment studies" document.

In following table is shown the result of performed power budget analysis for the first three phase of mission.

		Lander [Wh]							cruise stage [Wh]							
		TT&C	thermal	POWER	OBDH	GNC	Payolad	landing system	Engine + RCS	AOCS	Thermal	TT&C	POWER	Structure	*tot [Wh]*	*Margin*
Launch	*Energy requirements*	23,10	60,00	10,00	11,00	24,00	0,00	0,00	0,00	2,40	0,00	0,00	0,00	0,00	131	576%
	Available Energy														882 (*)	
orbit raising	*Energy requirements*	7522,48	5040,00	12600,00	924,00	2016,00	0,00	0,00	0,00	734,27	147,84	0,00	0,00	0,00	28985	47%
	Available Energy														42582 (**)	
cruise	*Energy requirements*	387858,24	261360,00	105930,00	47916,00	104544,00	0,00	0,00	94089,60	37635,84	344995,20	0,00	0,00	0,00	1384329	93%
	Available Energy														2678600	

In following figure, instead, is shown a detailed energy balance for EDL and Martian surface operations.

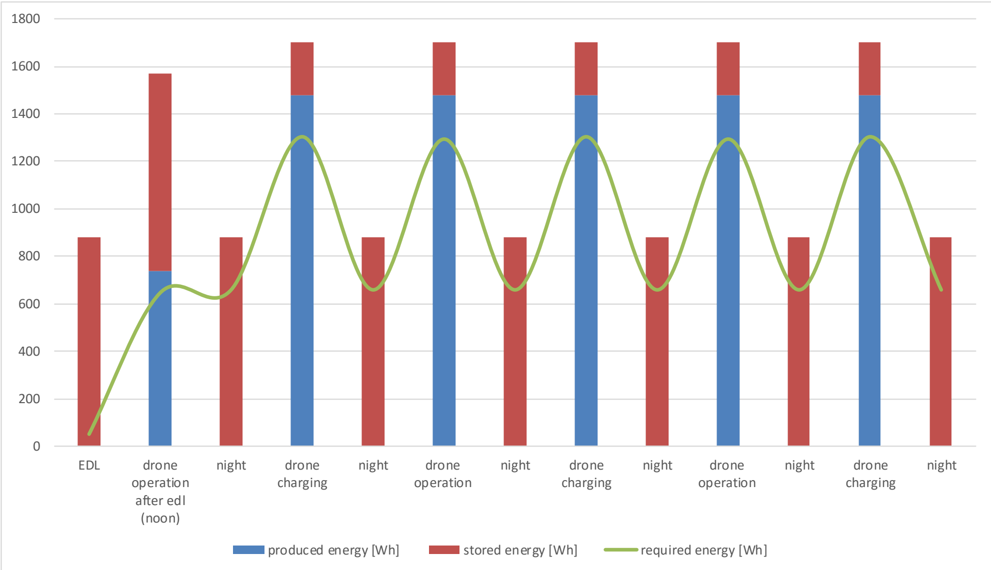


Figure 15 Detailed energy balance for EDL and Martian surface operations.

This preliminary analysis shows the system is able to satisfy all power requirement (applying ESA's required margins) in all mission phases.