Mission Analysis for radio science measure on asteroid

Complement of Thesis

IMPACTOR SPACECRAFT DESIGN

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1. PURPOSE

This document is dedicated to the Impactor spacecraft design.

2. SYSTEM TRADE OFF APPROACH

System options are based on the following launchers due to the fact the different class of launchers in term of mass capability, fairing envelope and cost (Table 2-1) are covered by the selected one's:

1) Soyuz direct launch
2) Dnepr
3) Vega

These launchers span the whole spectrum of options if are completed with the following ideas:

1) Soyuz has the higher cost but we can use Exomars carrier as Impactor S/C. The idea is to check if Exomars carrier can be used with no changes (even using the same antenna). This means a) save the cost of development of one of the two S/C and see if savings can be reached on orbiter trying to partially derive it from Exomars carrier as well, b) avoid chemical propulsion module and hence save cost both because of missing hardware and of simplified operations.

2) Dnepr is an intermediate option both by the point of view of performances and costs. In this case it is not obvious that the Impactor can be derived from Exomars carrier (taking also into account update information from Exomars study). A cost optimisation can be obtained trying to build a common Chemical Propulsion System using existing hardware (either other existing CPM or pieces taken from Exomars carrier).

3) Vega has the lower cost but the lower mass capability. In this case commonalities between Orbiter and Impactor (same chemical propulsion module) will be considered if and only if they do not jeopardise Vega option (the bigger save is to use Vega).

Launcher	Expected performance [kg]	Fairing usable room [mm]				QSL envelope [g's]		Minimum Eigenfrequency [Hz]		Launch Cost [M€]
		Φ_1	H_1	Φ_2	H_2	Lateral	Axial	Lateral	Axial	
Soyuz	>5000	3800	5060	2070.9	3343	1.8	5.0	15	35	40
Dnepr	3700	2700	1880	1930	1530	1.0	8.3	10	20	15
Vega	2160	2380	3515	1060	2000	0.9	5.0	15	20÷45	TBD

Table 2-1: Launchers performances comparison

2.1 Analysis of the major SC design drivers

A first analysis of major DQ Impactor requirements is performed to drive the system solutions retained for final trade off.

Several major drivers shall be considered to perform a first approach of Impactor design:

- the propellant mass compatible to both targets (2002AT4 and 1989ML) [RM-3320]
- the final mass and velocity to impact the asteroid with sufficient energy [RG-1130]
- the ESA margins

One simple exercise consists to calculate the SC dry mass considering these drivers (Table 2-2):

1) first input is the delta V provided by mission analysis for one impact opportunity
2) apply the 5% ESA margin [RM-3140] to this delta V
3) the max launcher capability (minus the adapter mass)
4) SC total mass @ launch considered as the max launcher capability minus the 10% % ESA margin [RM-3150]
5) Calculation of the SC dry mass considering the ISP of 500N engine (325N)
6) the resulting SC nominal mass is the calculated dry mass minus the 20% ESA margin on system mass [RM-3120]

In the first example (Table 2-3) corresponding to 1989ML impact with a direct Dnepr launch, the resulting dry SC mass (623kg) is not compatible with the SC mass requirement @ impact (823kg for 108m change on semi major axis) due to the launcher capability.

In the second example (Table 2-4) corresponding to 1989ML impact with an indirect Vega launch, the resulting dry SC mass (387kg) is not compatible with the high delta V requirement (4200m/s id est 1.4T of propellant) considering the minimal mass for structure and propulsion module.

Considering some ESA margins (very stringent in particular the 20% applied to SC dry mass before propellant mass calculation), **some opportunities with high delta V and an important impact mass could not be compatible to Dnepr capability. So, only indirect launch with Dnepr is possible to impact 1989ML**.

Concerning Vega launcher, the limitation is directly dependant to high delta V requirement. **So impact of 1989ML is not possible thanks to Vega launch due high velocity required to move from more than 100m its semi major axis.**

If some ESA margins are relaxed (in particular the system margin applied before propellant calculation or launcher capability margin applied to qualified one's), then some interesting impact opportunities could be studied. For instance, the opportunity to impact 1989ML with an indirect Vega launch is feasible (Table 2-5).

SC max mass top down calculation		
1. Delta V mission	DV	m/s
2. Delta V with margin ESA	DV_ESA=DV*1,05	m/s
3. Total Mass (launcher capacity)	CL	kg
4. Total Mass with margin ESA	CL_ESA=CL*0,9	kg
5. Resulting SC max dry mass	MSC=CL_ESA/EXP(DV_ESA/(ISP*9,82))	kg
6. SC dry mass considering ESA margin	**MSC_ESA=MSC/1,2**	kg

Table 2-2: Top down SC mass calculation

This calculation first approach allows to determine the Impactor resulting dry mass with as inputs the required Delta V and the full launcher capacity considering ESA margins.

SC max mass top down calculation		
Delta V mission	4400,00	m/s
Delta V with margin ESA	4620,00	m/s
Total Mass (**Dnepr** capacity without adapter)	3650,00	kg
Total Mass with margin ESA	3285,00	kg
Resulting SC max dry mass	768,98	kg
SC dry mass considering ESA margin	**640,81**	kg

Table 2-3: Direct trajectory to impact 1989ML on Dnepr

The resulting SC max total mass is not compatible to the SC impact mass required by mission (762kg to change the semi major from 100m!)

SC max mass top down calculation		
Delta V mission	4200,00	m/s
Delta V with margin ESA	4410,00	m/s
Total Mass (**Vega** capacity without adapter)	2100,00	kg
Total Mass with margin ESA	1890,00	kg
Resulting SC max dry mass	464,49	kg
SC dry mass considering ESA margin	**387,08**	kg

Table 2-4: Indirect escape (fly by Venus) to impact 1989ML on Vega

The resulting SC max total mass is not compatible to the mass of a propulsion module with 1.4t of fuel capacity and associated structure.

SC max mass top down calculation		
Delta V mission	4200,00	m/s
Total Mass (**Vega** capacity without adapter)	2100,00	kg
Resulting SC max dry mass	551,77	kg
SC dry mass considering ESA margin	551,77	kg

Table 2-5: Indirect escape (fly by Venus) to impact 1989ML on Vega without ESA margins

The resulting SC max total mass is compatible to the mass of a propulsion module with 1.4t of fuel capacity and associated structure.

2.2 Soyuz launch

To compensate the launcher cost, the SC shall have a high recurrence with an existing one. Due to its features, the re-use of Exomars design is proposed. As explained hereafter, Exomars carrier not requires a lot of modifications to answer to Impactor requirements with Soyuz launcher.

2.2.1 Exomars main features

The Exomars option retained is the carrier one (Figure 2-1) without the probe due to the cost and mass of the complete option.

2.2.2 Exomars modifications for Impactor re-use

Of course the Exomars carrier dimensions shall not be changed due to the fact it is designed for the Soyuz fairing. But due the mass impact requirement it should be necessary, depending on trajectory chose, to add ballast mass.

Not any problem concerning the propellant budget considering a direct escape from Earth is possible (then not need of a lot of fuel due to ballistic like fly). The only points to be check are:

- the release of not necessary tanks (only 2 external bus one's should be sufficient),
- the number and the configuration of the thrusters,
- the compatibility of the fixed SA and the one axis motion MGA with the new mission objectives,
- the possibility to integrate the platform equipments (id est CDMU, PCDU, batteries, AOCS, TTC RF) in place of the probe with a box like or directly fixed on the central bus.

Due to the fact the SA and MGA dimensions are close to the Impactor needs it is not a challenging exercise to adapt the Exomars carrier design to DQ mission. This work could be perform later and corresponding budgets could be provided in case this solution is interesting at system level.

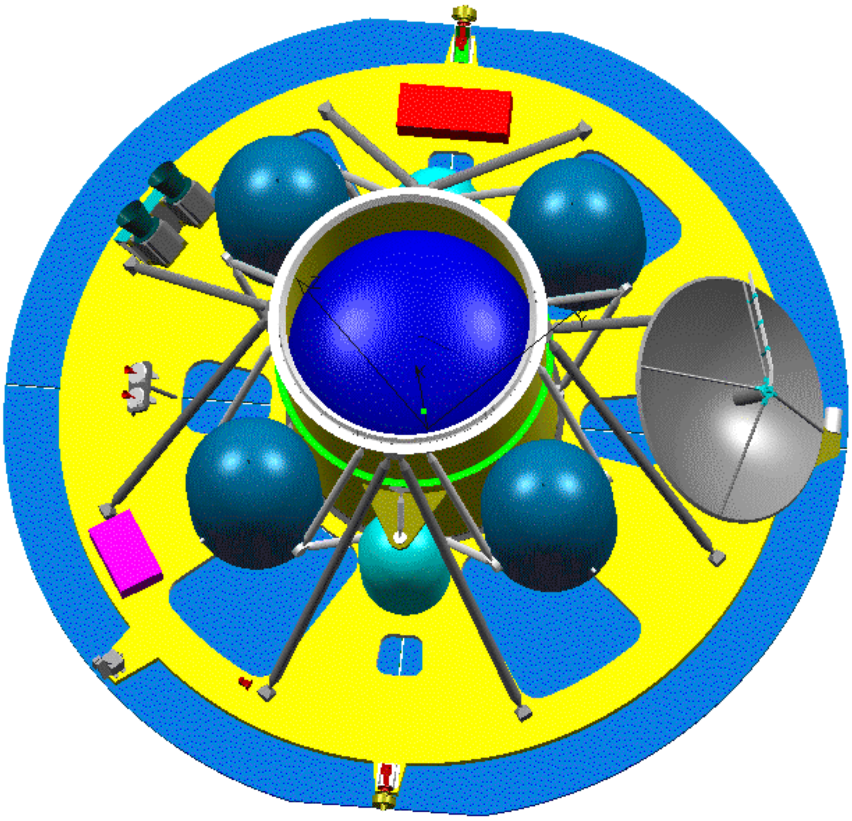


Figure 2-1: Exomars carrier without the probe on the top

The main features are a fixed 6m2 SA and a one axis deployment MGA with a propulsion module max capability of 1.5T.

	Unit	SOYOUZ S	SOYOUZ ST	ROCKOT	VEGA	DNEPR	PSLV
simplified volume (cylinder, without conical top volume)							
cylinder diameter	mm	3395	3800	2220	2380	2700	2900
cylinder length	mm	2364	5070	3711	3515	1880	2900
conical length	mm	3193	4448	2424	2000	3280	2540
adapateur 1		1194-SF (p182)		CASA CRSS 1194	937B		
hauteur	mm	230		740	1100		
diamètre IF SL	mm						
compatibilité lanceur / SL : margins							
vs diameter		737,73	1142,73	-437,27	-277,27	42,73	242,73
vs heigth		1184,00	3890,00	1791,00	2335,00	700,00	1720,00
				not possible	not possible		

DON QUICHOTTE impactor based on Exomars carrier propulsion module

diameter	mm	2657,27	
heigth (cylindric part)	mm	1180	! without antenna to fit inside conical volume

Table 2-6: Exomars propulsion module compatibility with the different launchers envelope

The Exomars propulsion module is compatible with Dnepr fairing width but not with Vega (and Rockot)

2.3 Dnepr Launch

The top down approach considering the mission requirements and the Dnepr capability is used to define the main features of the Impactor (the dry mass and its associated fuel consumption). Thanks to these data, a design based on an existing propulsion module is proposed considering also the CPM will be compatible to Orbiter fuel budget. Finally the Impactor "platform box" will be fixed on the top of the CPM as the orbitor.

2.3.1 Top down approach

As explained in 2.1 (Analysis of the major SC design drivers), the mass required @ impact and the associated delta V are not compatible of a direct trajectory on Dnepr so only indirect trajectory with fly by Venus is studied. The same first approach allows estimating the SC mass target and associated propellant mass.
Considering the max mass capability of Dnepr the SC dry mass is 849kg what answers to the mission requirement: 682kg to change the semi major from 100m.
The propellant mass required to perform the required delta V (4100m/s) is 1956kg considering the dry mass required at impact (682kg). So the Orbiter fuel budget (2400kg) will be considered cause it is higher than Impactor one.

2.3.2 Exomars re-use on Dnepr launcher

Clearly the width of Exomars SA is not compatible to Dnepr but it is interesting to re-use at least the propulsion module. So it is necessary to check the compatibility of Exomars propulsion module with the Dnepr fairing dimensions and with the mission fuel requirements: the external dimension of Exomars propulsion module is compatible with Dnepr envelope (Table 2-6). However, the propellant mass capability (1.5T) is low in regard to Impactor fuel necessary to perform the required delta by mission analysis to impact 1989ML so the tanks volume shall be increased (considering qualified one's).

Finally, the Impactor "platform box" will be fixed in place of the probe.

So the Exomars propulsion module design could be re-used as DQ common CPM for Dnepr laucher but the tanks capability shall be increased.

2.3.3 AAS Space Bus alternative solution

To reduce the CPM cost a flight proven SB propulsion module could be used. The SB configuration corresponding to the fuel budget required is unfortunately not compatible to the Dnepr fairing with the Orbiter due to the resulting height.

SC max mass top down calculation		
Delta V mission	4100,00	m/s
Delta V with margin ESA	4305,00	m/s
Total Mass (**Dnepr** capacity without adapter)	3650,00	kg
Total Mass with margin ESA	3285,00	kg
Resulting SC max dry mass	849,00	kg
SC dry mass considering ESA margin	**707,50**	kg

Table 2-7: Fly by Venus trajectory to impact 1989ML on Dnepr

The resulting SC max total mass is compatible to the SC impact mass required by mission (682kg to change the semi major from 100m!)

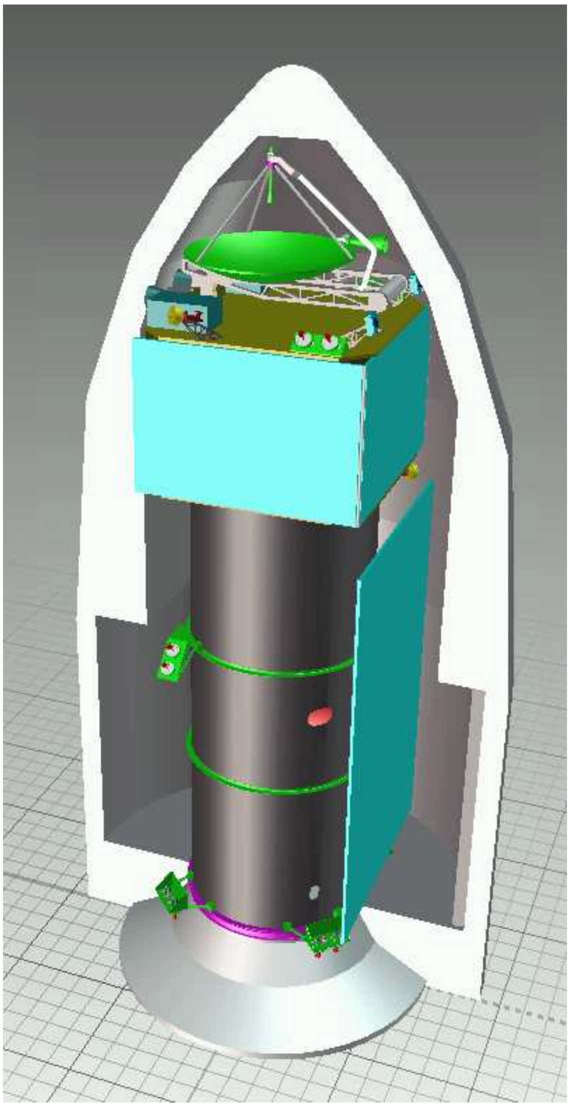


Figure 2-2: View of Impactor with SB 2.4T CPM

The resulting height with Orbitor fixed on SB CPM 2.4T is not compatible to Dnepr envelope.

2.4 Vega Launch

The top down approach considering the mission requirements and the Vega capability is used to define the main features of the Impactor (the dry mass and its associated fuel consumption). Thanks to these data, a design based on an existing propulsion module is proposed considering in this case that the commonality with Orbiter is not a main driver.

2.4.1 Top down approach

As explained in 2.1 (Analysis of the major SC design drivers), the resulting SC max total mass (387kg) for impacting 1989ML is not compatible to the mass of a propulsion module and its associated structure so only 2002AT4 impact is studied. The same first approach allows estimating the SC mass target and its associated propellant mass.
Considering the max mass capability of Vega the SC dry mass is 465kg what is an objective compatible to the Impactor mission (not any payload).
The propellant mass required to perform the required delta V (3650m/s) is 1331kg considering the max dry mass (465kg) possible due to Vega capability.

2.4.2 Exomars re-use with Vega launch

Due to Vega fairing width (2380mm), the Exomars propulsion module cannot be re-used (Table 2-6) because it is not possible to arrange 4 fuel tanks with sufficient volume around its central bus (1200mm of diameter). So some major changes should be performed on Exomars propulsion module (more tanks around or narrow one's but not qualified) what is not compatible with low cost criteria.

2.4.3 AAS Space Bus alternative solution

To reduce the CPM cost SB propulsion module could be used. One SB CPM flight proven configuration corresponding to the fuel budget exists with a fuel max capability of 1542kg. Its width is fully compatible to Vega due the fact the MMh and MOM tanks are integrated inside the central tube. No problem concerning the height also considering the Impactor as a "platform box" to fix upper the CPM (SB propulsion modules are qualified with higher load on the top).

So the use of SB 1.4T propulsion module as Impactor CPM for Vega launch is a great opportunity considering the high level of qualification and of recurrences.

SC max mass top down calculation		
Delta V mission	3650,00	m/s
Delta V with margin ESA	3832,50	m/s
Total Mass (**Dnepr** capacity without adapter)	2100,00	kg
Total Mass with margin ESA	1890,00	kg
Resulting SC max dry mass	558,20	kg
SC dry mass considering ESA margin	**465,17**	kg

Table 2-8: 2002AT4 impact trajectory on Vega

The resulting SC max total mass is compatible to the mass of a propulsion module and its associated structure.

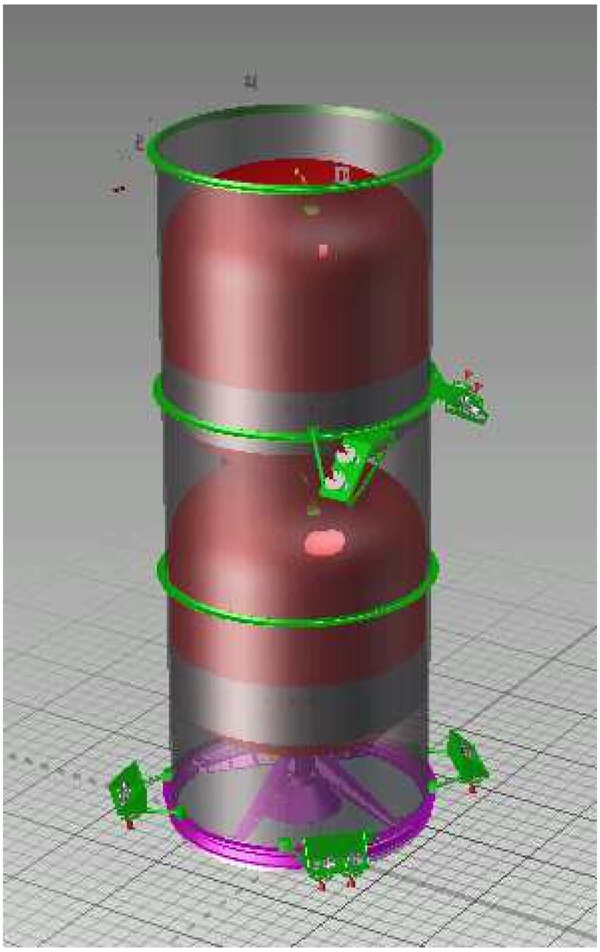


Figure 2-3: View of SB propulsion module

The SB CPM is fully compatible to Vega envelope.

2.5 Impactor options synthesis

The retained Impactor solutions thanks to first design approach are:

- Soyuz direct escape: Exomars carrier with lower modifications to be compliant with Impactor mission

- Dnepr indirect trajectory: CPM of Exomars carrier with extended tanks (to be compatible to Orbiter)

- Vega (only 2002AT4 target): flight proven AAS Space Bus module propulsion

Considering the existing Exomars study, the first option is not challenging in regard to Soyz mass capability. **So the next steps will focus on the study of the solutions compatible with Dnepr and Vega**.

Concerning the platform equipments (id est CDMU, PCDU, batteries, AOCS, TTC RF), a short trade off is performed in the next § between 2 solutions

- to fix an Impactor “platform box” on the top of the CPM (in place of the Exomars probe)
- to integrate the equipments on the central bus.

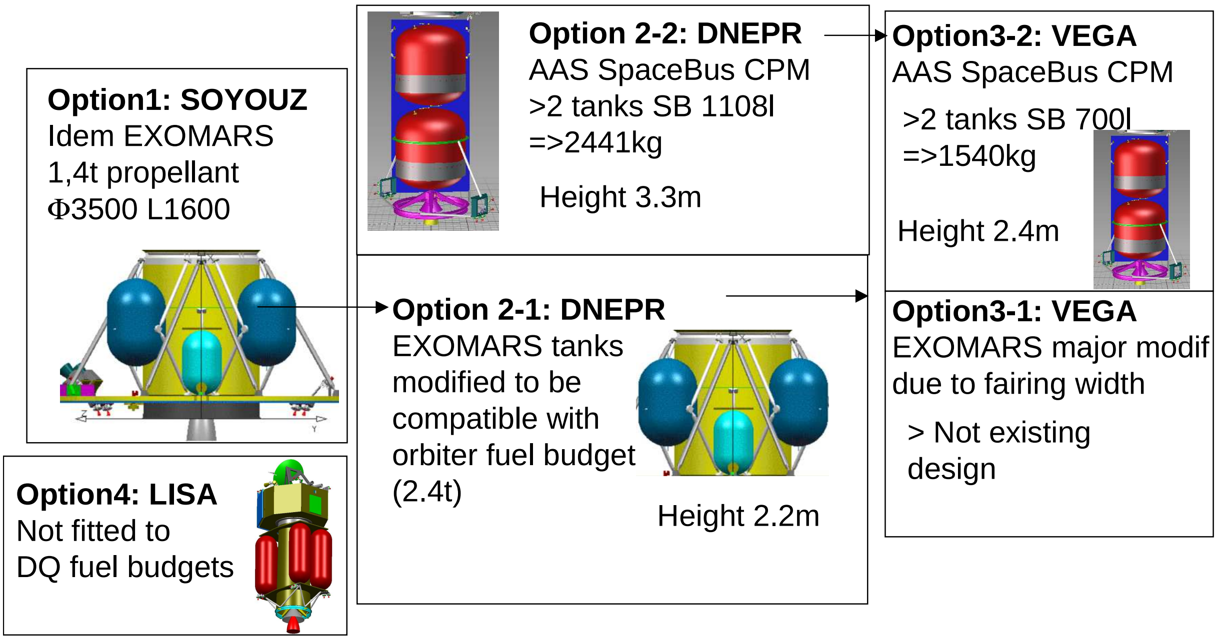


Figure 2-4: Different Impactor options synthesis

The options 2.1 and 3.2 shall be validated thanks to precise dimensioning.

3. PROMISING OPTIONS DIMENSIONNING

This part constitutes a deeper analysis of the most promising solutions identified during system trade off.
Firstly, the platform design (common to both CPM options) is performed, considering the worst-case missions constraints allow designing the platform subsystems (anyway are the propulsion module and associated structure).
Then the two complete (including CPM) SC options (one option corresponding to 1989ML target with Dnepr and the other one to 2002AT4 with Vega) are studied to verify its compliance with the mission requirements.

3.1 Platform design

3.1.1 Spacecraft Configuration

Thanks to the system modes definition (Table 3-1), it could be possible to define an Impactor configuration respecting the different mission constraints.

3.1.1.1 System modes presentation

Spacecraft general configuration and flight attitude is driven by the mission geometry in the different mission phases (only dimensioning phase are studied considering other one's mainly as sub-phases).

<u>LEOP (Launch Earth orbit phase</u>)

In this phase the spacecraft attitude is driven by the power needs (solar array illumination) and the thrust vector requirement during the manoeuvres. Communications are not a driver in this phase as at such close range, low gain antennas can be used and they will have omni-directional coverage. During the manoeuvres, it is possible that the solar array illumination is not optimal (due to the thrust direction requirement), therefore the battery might be used as a complement. Should eclipses be present in this phase they will also be taken into account in the power subsystem sizing.

<u>Cruise phase</u>

During the cruise phase, as no specific operations are planned except for periodic telemetry and status downlink, and ranging, a nominal sun-pointing attitude is foreseen. A slow spin around the sun vector during routine cruise (outside communication sessions) will allow to average the solar radiation pressure torques and thus minimise the actuation needs (AOCS propellant, reaction wheels if any).
In terms of communications: low gain antennas will be marginally usable for telemetry at large distance, thus at least a medium gain antenna with low beamwidth (ie sufficient gain) will be needed to download the necessary spacecraft telemetry.
The spacecraft geometry during the cruise phase will therefore be driven by the Sun-Spacecraft-Earth angle evolution along the cruise, as both power generation (sun on solar array) and communications must be ensured.

<u>Approach and impact phase</u>

The pointing axis of the optical navigation camera towards the target asteroid will drive the attitude of the impactor spacecraft during the final guidance. One degree of the rotation around the navigation camera line of sight shall be sufficient to answer to the other mission constraints.
The geometry of the approach and impact phase for both potential target asteroids provides drivers for mainly the power and TTC subsystems.

System Modes	**Description**	**AOCS modes**	**Notes**	**TT&C**	**Notes**
Pre Launch	On launch pad	Stand-by		Launch	units warming up
Launch	From launch to separation				
Sun Acquisition	After separation	SAM	CSS, GYRO, Thrusters	Safe	X band
Leop	Earth escape phase encompass orbits & chemical raise manoeuvres	ORM	STR, GYRO, Thrusters		
Correction Manoeuvres	Correction manoeuvres, eventually fly-by	OCM_C	STR, RWA, Thrusters	Cruise	X band
Cruise	Cruise	Cruise	STR, RWA, NAVCAM		
Final Targeting	Close approach to NEO, navigation, payload & proximity UHF link	Autonav	STR, GYRO, RWA, Thrusters, NAVCAM	Final Targeting	X band UHF band (link Orbiter)
Safe		SAM	CSS, GYRO, Thrusters	Safe	X band

Table 3-1:System modes definition

Mission Phase	**Spacecraft attitude**
Sun acquisition / Safe	SA axis pointing towards Sun with slow spin
LEOP	Main engine driven by required thrust direction SA axis pointing towards Sun
Cruise	SA axis pointing towards Sun TTC antenna axis pointing towards Earth (when required)
Correction manoeuvres	SA axis pointing towards Sun Main engine (4 thrusters on the same panel in fact) driven by required thrust direction
Final targeting	Navigation camera sight axis aligned with target line of sight TTC antenna axis pointing towards Earth (and UHF towards Orbiter)

Table 3-2: Impactor Mission Phases and related S/C Attitude

3.1.1.2 Synthesis of mission inputs:

The mission opportunities retained by system trade off for SC design are:

- 1989ML target with Dnepr launch on October 2013
- 2002AT4 target with Vega launch on October 2015

The corresponding constraints for SC design are given hereafter:

	Launch date	ΔVesc km/s	Δa km	Mass kg	Velocity km/s	Sun min AU	Sun max AU	Earth max AU	Earth Impact Angle	Sun Impact Angle
2002AT4	2015/10/08	3.65	2.8	606	13.2	1.00	1.35	1.74	18 deg	17 deg
1989ML	2013/10/28	4.1	0.132	901	8.1	0.69	1.52	2.16	89 deg	34.4 deg

Considering the above opportunities to study, the main subsystem drivers are:

- max Earth distance 2.16 AU (TTC major input) corresponding to 1989 ML target
- max Sun distance 1.52 AU (Power major input) corresponding to 1989 ML target
- min Sun distance 0.69 AU (Thermal major input) corresponding to 1989 ML target
- Earth angle range @ impact 18 to 89 Deg (Configuration major input)
- Sun angle range @ impact 17.0 to 34.4 Deg (Configuration major input)

So the 1989ML target shall be the most dimensioning for platform subsystems. Moreover the large range of Earth angle @ impact will be a major constraint to configure the MGA in a position compliant with both targets considering the different SUN angles @ impact.

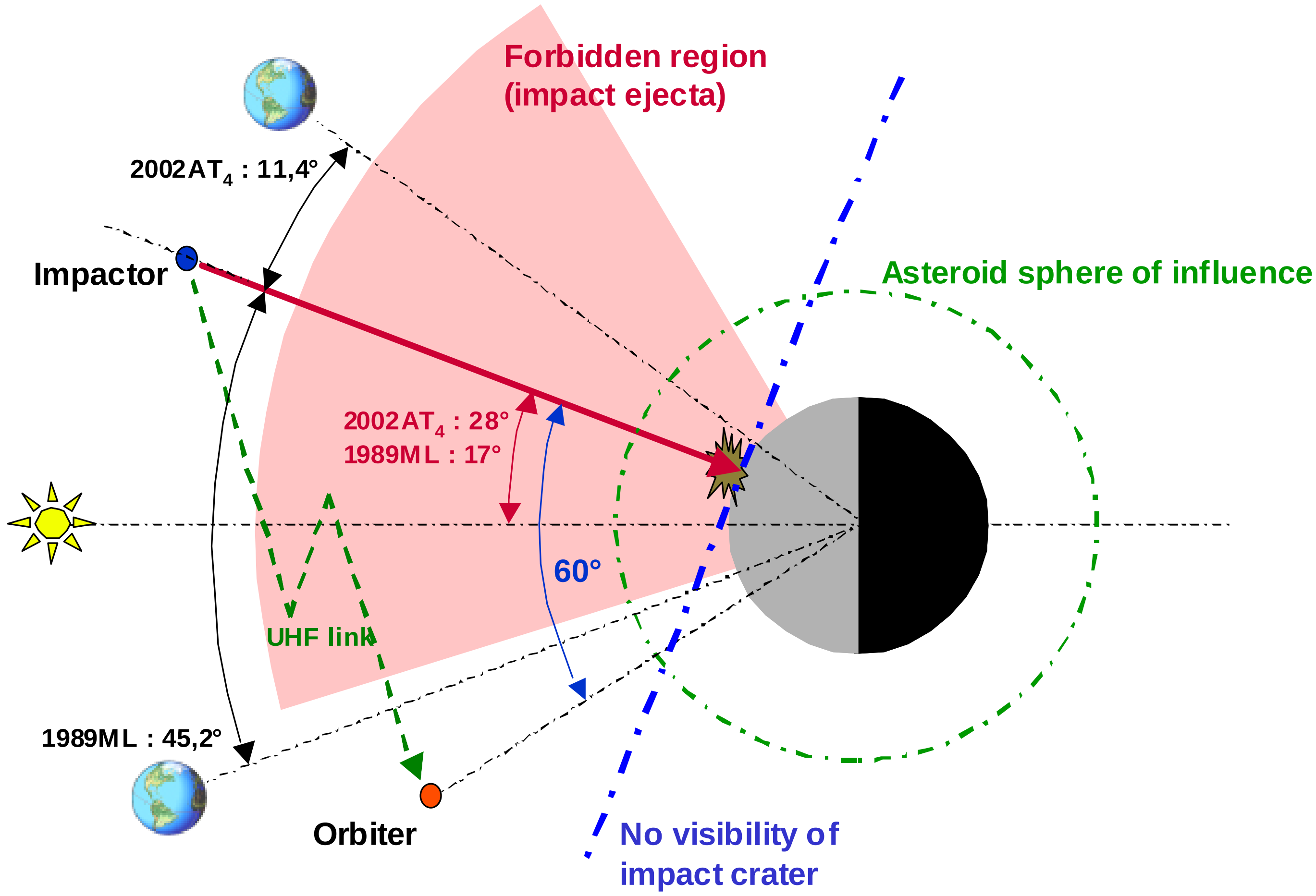


Figure 3-1: Geometry of the approach and impact phase

3.1.1.3 SC configuration trades off:

The envisaged possibilities are :

(a) Fixed body-mounted solar array and fixed antenna

In that case, pointing the antenna towards the Earth requires turning the spacecraft body, meaning that depending on the Earth position, power generation might not be ensured. Therefore the communications sessions will have to be powered by batteries, which could be a sizing case (depending on power deficit due to SA depointing and communication pass duration).

Due to the two targets shall be raised thanks to different trajectories, this solution is not feasible (even if only one target is retained then the back-up trajectory shall be considered in case of launch delay).

(b) Orientable solar array and fixed antenna

By implementing a steerable solar array, it will be possible to keep the solar array optimally sun-pointed while turning the spacecraft body to ensure correct Earth pointing for the antenna.

Considering the resulting cost and risk, this solution is not the preferred one.

(c) Fixed solar array and steerable antenna

Thanks to this configuration, the spacecraft can be kept in a sun-pointing attitude during communication sessions.

With low Sun angle @ impact compensated by an over sizing of the SA, the Earth pointing for the antenna can be achieved by (1) a rotation of the spacecraft around the sun vector (thus not modifying the solar aspect angle on the solar panel), and (2) a 1degree of freedom steering mechanism. For this mechanism, a 90° range with a rotation axis perpendicular to the sun direction (until 180 Deg) is sufficient to ensure a whole sky coverage.

With higher Sun angle @ impact, if the resulting SA over sizing should not be compatible with the mass and the fairing limitations, then a 2 axes motorisation shall be implanted for MGA.

So due to its flexibility and reduced risk associated (in case of mechanism failure it could be always possible to use the batteries during the Cruise phase and the Orbiter TTC link during Impact phase), this last solution is considered hereafter.

#	Option	Rationale	Comments
		Fixed Solar Array	**Steerable Solar Array**
	Fixed Antenna	•1 S/C body depointing for communications •2 Power generation not guaranteed during communications •3 Limited sun incidence on SA during final impact phase	•4 S/C body depointing for communications •5 Requires SADM
	Steerable Antenna	•1 Optimal power generation during communications in cruise •2 Limited sun incidence on SA during final impact phase •3 Requires 1-axis pointing mechanism	•4 Optimal power generation during communications in cruise and final guidance •5 Requires APM and SADM

Table 3-3: Synthesis of configuration options and preliminary baseline

3.1.2 Mechanical design

3.1.2.1 Integrated or "box like" platform trade off :

The "box like" solution presents several advantages in regard to the integration of the platform equipments on the CPM:

- the "box like" (thanks to a 1200mn IF) is compatible to all proposed CPM (Exomars or SpaceBus)
- the CPM Assembly, Integration and tests sequence is not modified by new equipments to be fixed
- the "box like" qualification based on enlarged environmental constraints is applicable to any launcher
- once flight proven the "box like" (in particular its avionics) is so re-usable to another asteroid impact (only the CPM fuel capability and the total mass thanks to ballast should be modified if necessary)

The only drawback in regard to an integrated solution should be the mass of the platform box but it is not evident due to the fact:

- the equipment fixing supports are anyway necessary with a platform integrated solution
- the external panel are required also for a platform integrated solution
- the harness mass saved in particular on tanks thermal lines is lost cause to the different boxes are farther one to the other when placed around the central bus

So the platform box like design is retain due to its flexibility in regard to the different options and due to its large heritage in case of further missions.

3.1.2.2 Main platform box drivers:

Here after are listed the main SerVice Module design drivers :

- Architecture based on a separated CPM / PLM, to facilitate manufacturing, integration and tests operation
- compatible IF with SpaceBus central bus (1200mm diameter)
- fixed solar array with max size (3x1,7m2)
- integration of all platform equipments (Table 3-4)
- low dimension to be compatible to top fairing of Dnepr and Vega.

A view of the resulting SerVice Module is provided Figure 3-2.

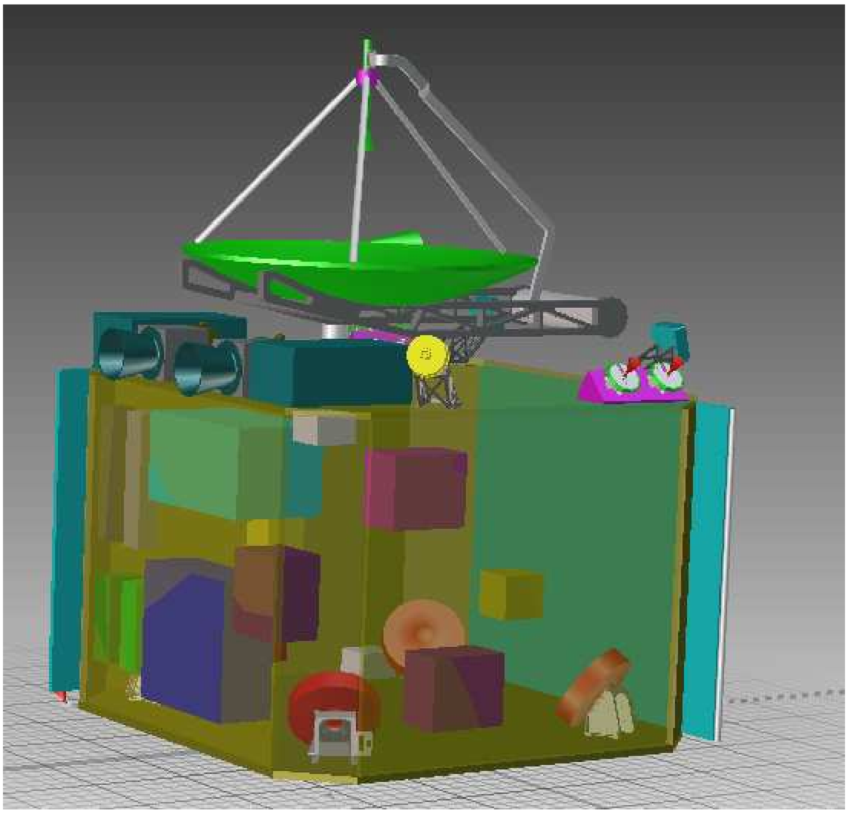
3.1.2.3 Definition of spacecraft axes:

To clarify the drawings provided this is the SC axes definition:

1. +X = longitudinal axis, positive upwards (launcher in vertical position)
2. +Y = sun direction during cruise (towards solar panels)
3. +Z = completing the right-handed orthogonal triad

That means for instance:

- main engine @ -X
- SA @ +Y



Main characteristics :
1380 x 1380 x 980

Structure mass ~43kg,
+ margins 12%

Solar array : 3,8 m2
(1,3 + 2x1,28)

Figure 3-2: Platform box like (SerViceModule) CAD

Instrument	Mass Kg	*mass reference*	Power W	*power reference*	dimensions mm	*dimensions reference*
CDMU	19	*22/9/06*	35	*11/9/06*	470x270x250	*22/9/06*
XPND-RX	3	*11/9/06*	28	*11/9/06*	200x230x145	*11/9/06*
XPND-TX	3	*11/9/06*	9	*11/9/06*	200x230x145	*11/9/06*
XPND-UHF	0,6	*11/9/06*	33	*11/9/06*	200x230x145	*11/9/06*
TWTA1 X	1,5	*11/9/06*	128	*11/9/06*	385x63x82	*11/9/06*
EPC1 X	0,8	*11/9/06*	see note	*11/9/06*	250x85x105	*11/9/06*
TWTA2 X	1,5	*11/9/06*	128	*11/9/06*	385x63x82	*11/9/06*
EPC2 X	0,8	*11/9/06*	see note	*11/9/06*	250x85x105	*11/9/06*
RFDN X-Band	2,66	*11/9/06*				
RFDN UHF-Band	0,9	*11/9/06*				
LGA1 X band	0,3	*11/9/06*				
LGA2 X band	0,3	*11/9/06*				
MGA X band	3,1	*11/9/06*				
HGA X band	7,4	*11/9/06*			1m	*11/9/06*
LGA1 UHF	1,5	*11/9/06*				
LGA2 UHF	1,5	*11/9/06*				
GYR1	2	*11/9/06*	7,5	*11/9/06*	0,125x0,125x0,12	*11/9/06*
GYR1	2	*11/9/06*	7,5	*11/9/06*	0,125x0,125x0,12	*11/9/06*
CSS1	0,14	*11/9/06*	0	*11/9/06*		*11/9/06*
CSS2	0,14	*11/9/06*	0	*11/9/06*		*11/9/06*
RW1	3,7	*11/9/06*	20	*11/9/06*	∅ 225x86	*11/9/06*
RW2	3,7	*11/9/06*	20	*11/9/06*	∅ 225x86	*11/9/06*
RW3	3,7	*11/9/06*	20	*11/9/06*	∅ 225x86	*11/9/06*
RW4	3,7	*11/9/06*	20	*11/9/06*	∅ 225x86	*11/9/06*
STR1	1,9	*11/9/06*	7,5	*11/9/06*	165x165x198	*11/9/06*
STR2	1,9	*11/9/06*	7,5	*11/9/06*	165x165x198	*11/9/06*
Nav Cam	6	*11/9/06*	7	*11/9/06*	165x165x198	*11/9/06*
Nav Cam	6	*11/9/06*	7	*11/9/06*	165x165x198	*11/9/06*
PCDU	10	*11/9/06*	280	*11/9/06*	390x430x160	*11/9/06*
BATTERY	8	*11/9/06*		*11/9/06*	321x205x116	*11/9/06*

Table 3-4: Equipments dimensions inputs for SVM CAD

3.1.3 Guidance Navigation and Control Subsystem sizing

First, the system useful data concerning the AOCS equipments is re-called (the details are in the GNC design report). Then a quick presentation of the different AOCS modes for Impactor and the synoptic to describe the modes chaining are provided.

3.1.3.1 GNC equipments overview

Two Actuator/Sensor layouts have been designed, to handle the different mission scenarios:
- a "0°-configuration", to handle impact angles from 0° to 22° ;
- a "45°-configuration", to handle impact angles from 22° to 68°.

Impact angles greater than 68° have been excluded during mission analysis in order to avoid adverse effects from self-shadowing during targeting.

The configurations differ in the angle between the camera boresight (impacting direction) and the body-mounted solar array. Once the camera is pointed toward the asteroid during the final targeting phase, only small rotations around the camera boresight direction are permitted. Sun rays will reach the solar array with an angle no greater than 23°. This angle represents a decrease of the solar flux of 8%.Solar arrays must be sized accordingly.

Three-axis pointing is required during the final targeting phase. Therefore the High-Gain Antenna must be mounted so as to point towards the Earth during this phase.

3.1.3.1.1 Navigation camera

A narrow angle camera with an Active Pixel Sensor array is used for the autonomous navigation phase. The precise sizing of the camera is on-going. A trade-off will address the best way to deal with the large range of magnitude during the targeting phase (attenuation, variation of exposure time, use of an additional Wide Angle Camera for the final phase).

The following characteristics are foressen (TBC):
- FoV 0.7°
- Pixel accuracy 10 microrad [3σ]
- Magnitude 12 to –2
- Long focal length TBD ; a small defocus will enable sub-pixel accuracy at long range.
- Rate 1 to 0.2 Hz
- Mass < 7 kg
- Power < 8 W

To minimize angular drift between the Star Trackers and the Navigation Camera during the targeting phase, all the instruments will be accommodated on an optical bench on the +X panel. The camera boresight is along the –Y direction (anti-sun).

One-for-one cold redundancy is baselined. If the warm-up time is not compatible with the image acquisition rate in the final impact phase (last manoeuvre, around 1000 to 100s), warm redundancy will be considered.

Element 1	-		MASS [kg]			
Unit	**Unit Name** Click on button above to **insert new unit**	**Quantity**	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	Coarse Sun Sensor	2	0,3	Fully developed	5	0,6
2	Star Tracker	2	1,9	To be modified	10	4,1
3	Coarse Gyro	2	2,0	Fully developed	5	4,2
4	Navigation Camera	2	6,0	To be developed	20	14,4
5	Reaction Wheel	4	3,7	Fully developed	5	15,5
-	Click on button below to insert new unit		0,0	To be developed	20	0,0
SUBSYSTEM TOTAL		**5**	**35,0**		**10,7**	**38,8**

Table 3-5: AOCS equipments mass budget

3.1.3.1.2 Star trackers

Attitude determination will rely mainly on a high quality Star Tracker. The APS-based Autonomous Star Tracker (AA-STR) from Laben (originally BepiColombo TDA) is baselined. The characteristics of this sensor are presented inTable 3-6. The 3σ accuracy in the boresight plane is 36 microrad (quasi-inertial pointing).

The nominal and redundant Star Trackers are accommodated on +X the optical bench in the –Y direction, with a small offset to avoid stray light from the asteroid at close range.

Cold redundancy is considered compatible with the Line-Of-Sight measurement rate for this equipment.

Even though this STR has been designed to withstand large radiation levels (> 100 Krad), a large solar flare during the 2 days of the targeting phase would compromise the mission success. However, the probability of this event (on the order of 10^{-3}) is deemed acceptable in view of 95% success requirement.

3.1.3.1.3 Coarse sun sensor

Two 2π-steradian sun sensors (Officine Galileo, used on Italsat programs) are mounted on the sun side, to acquire sun after launcher separation and in safe mode. Characteristics are summarized in Table 3-7. The same equipment is used for the orbiter.

3.1.3.1.4 Coarse rate gyro

The coarse rate gyro (TRIS from Laben) serves several purposes:

- Back-up equipment for Safe mode
- Increase robustness by STR/Gyro hybridising during manoeuvres (Leop)
- Detect STR or NavCam failure by cross-equipment monitoring (level 1.b failure detection).

However, the gyro is not sufficiently accurate (1° error after 100s typically) to be used to improve the pointing knowledge during the targeting phase. The main characteristics of the equipment are summarized in Table 3-8.

3.1.3.1.5 Accelerometer

Errors in the realization of ΔV can affect the performance of the autonomous targeting phase. A trade-off study is on-going to determine the need for a precise accelerometer to reduce these errors. The impact on the overall mass/power/bulk budget is very low (250g, 1W). However it is not obvious whether the high precision required (better than 30 μg to reach 1% precision on the ΔV) can be met by existing devices.

Parameters	Value
FOV	20 ° full cone
Attitude accuracy (random contribution) Pitch, yaw Roll	 5 arcsec (2σ) 40.6 arcsec (2σ)
Star magnitude limit	Mi=5.5
Prob. of attitude determination within 20 seconds	> 99.8 % @ 0 SEU/frame > 99 % @ 1400 SEU/frame
Update rate	10 Hz
Outputs	Quaternion Angular rate Star unit vectors and index
Mass	1.85 Kg (including baffle with 40° exclusion angle)
Size	165 x 165 x 198 mm with 40 deg balle
Power consumption	6.2 W at 0°C 8.4 W at 40 °C
Input voltage	16 V to 40 V
Data I/F	MIL 1553B
Operating temperature	-20°C to +50°C
Non operating temperature	-30°C to +60°C
Random Vibration	17 g rms
EMC	Compliant MIL-STD-4161C

Table 3-6: AA-STR characteristics.

ITEM DESCRIPTION	DESIGN SPECIFICATION
Dimensions	86 x 86 x 59 mm without baffles
Mass	< 0.270 Kg
FOV	2 π sterad without baffling
Accuracy	0.5° or 10% which is bigger
Power consumption	N.A. (passive sensor)
Alignment	Mechanical by mounting plane
Operating temperature	-30°C + 60°C
Failure rate	3.42 x 10^{-9}
Outputs	Two wires for each cell

Table 3-7:Coarse sun sensor characteristics.

3.1.3.1.6 Reaction Wheels

Four reaction wheels (Teldix 4-75/60) in a pyramidal configuration with 4:3 (cold) redundancy are used for attitude control during cruise and final targeting. A capacity of 4 Nms is sufficient to compensate Sun Radiation Pressure perturbations during final targeting (with a margin of 100%) and allows a reasonable off-loading rate during cruise (of the order of once per 4 days)..

3.1.3.1.7 Reaction Control Thrusters

The RCT is composed of 2 (4+4) sets:

- A set of (4+4) 22 N thrusters on the –X panel to control disturbances of the 500N main engine (up to 15 Nm considered) during LEOP.
- 2 clusters of (2+2) 10 N thrusters to perform lateral ΔVs during final targeting.

Two configurations have been designed (0° and 45° configurations) with comparable characteristics.

Force-free torques can be delivered along all axes with a reasonable efficiency. Care has been taken to optimize the +/Z axis torque capacity, in order to improve an efficient unloading of reaction wheels during cruise. Indeed, the Solar Radiation Pressure will mostly affect the momentum around this axis.

Torque-free thrusts are obtained by coupling two thrusters on opposite ends of the spacecraft, in such a way that the failure of one thruster can be detected by attitude monitoring.

In the course of the study, an alternative configuration using only 10 N thrusters will be considered and a trade-off will be performed.

3.1.3.1.8 Other devices managed by the AOCS

The AOCS will manage the High-Gain Antenna pointing mechanism (1 degree of freedom hinge).

POWER	
Input Power Bus Voltage	+20V ÷ +55V or +90V ÷ +150V
Input Power	<5.5 W
PERFORMANCE	
Angular Rate Range	Up to ± 10°/s
Null Bias O.T.R. [1]	0.01 °/sec (5mV)
Null Bias around 22°C	0.002°/sec (1mV)
Null Bias Stability [2] over 72 hrs	<0.014°/sec (7mV)
Switch On to Switch On	0.0024°/sec (3σ)
Nominal Unit Scale Factor (for ± 10°/s)	0.5V/°/sec
Unit Scale Factor Linearity	±1%
Scale Factor calibration begin of life	< 0.1% or negligible
Scale Factor temperature sensitivity	Negligible
Bandwidth	Selectable (0 ÷ 5 Hz baseline)
O/P Noise PSD (single sided)	flat <0.005°/sec/√Hz in bandwidth 0-5Hz
Threshold / Resolution	Negligible
Mechanical Misalignment[3]	<0.8°
ELECTRICAL OUTPUT I/Fs	
Analog Angular Rate O/P range [4]	±5.0V
Analog Angular Rate Monitor O/P range	0 ÷ 5V
Bottom Plate Temperature Monitor	Based on thermistor YSI44908
ELECTRICAL INPUT I/Fs	
Angular Rate Stimulation range	±10V
ENVIRONMENT	
Operating Temperature (bottom plate)	-25°/C to +50°/C
Non-Operating Temperature	-40°/C to +70°/C
Radiation	≤100 Krad, Latch-up free
Random Vibrations	Up to 15 g_{rms} (2 min.)
Shock	Up to 1500 g (from 2000Hz)
EMI/EMC	MIL-STD 461-462
Lifetime	>10 years
RELIABILITY [5]	
Single Unit	0,970859052
Two Units, one in cold redundancy	0,999528802
MTBF	1.48 x 10^6 hr
WEIGHT and DIMENSIONS	
Weight	1500 gr.
Dimensions (LxWxH)	125x125x120 mm
Optical Alignment device	Yes

Table 3-8:Coarse rate sensor characteristics.

Main Technical Data	RSI 4-75/60			RSI 12-75/60x	
Angular Momentum at Nominal Speed	4	Nms		12	Nms
Operational Speed Range	± 6000	rpm		± 6000	rpm
Speed Limiter (EMF)	< 7500	rpm		< 7500	rpm
Motor Torque over full Speed Range	± 75	mNm		± 75	mNm
Loss Torque (max.)	< 20	mNm		< 20	mNm
Dimensions:					
- Diameter	222	mm		247	mm
- Height	85	mm		85	mm
- Mass	< 3.7	kg		< 4.85	kg
Power Consumption:			< 20 W		
- Steady State at Nominal Speed			< 90 W		
- Max. Torque at Nominal Speed					

Table 3-9:Reaction Wheels characteristics.

3.1.3.2 GNC modes overview

3.1.3.2.1 Sun Acquisition mode

This mode is used at launcher separation and as a safe mode. The AOCS first reduces the spin rate if necessary (e.g. after separation). Two successive slew manoeuvres around the X and Z axis (orthogonal to the sun sensors boresight) are performed to acquire the sun direction with a precision of 1°. A slow spin around this axis is then established and the AOCS points the antenna towards the Earth.

3.1.3.2.2 Orbit Raising Manoeuvre mode

This is the control mode used during LEOP. It takes advantage of the large capacity of the RCT configuration. The total ΔV of the Don Quijote mission being quite large, it is important to reduce AOCS fuel cost to a minimum during this phase. The controller bandwidth will be estimated, taking into account the matrix of inertia of the spacecraft and the required pointing performance.

3.1.3.2.3 Cruise mode

Inertial sun pointing is considered during cruise. 2-degree of freedom pointing of the High-Gain Antenna is assured thanks to the hinge mechanism of the antenna and yaw steering around the sun axis. A Reaction Wheels unloading frequency of once every 4 days is foreseen, making it possible to rely on ground-commanded unloadings.

3.1.3.2.4 Orbit control mode

The Orbit Control Mode performs ground commanded manoeuvres with the 22/10 N thrusters set. This mode is used during fly-by and for final trajectory adjustment before the start of the autonomous navigation phase. Thrusts can be performed in all directions after a small slew manoeuvre around the X axis.

3.1.3.2.5 Autonomous Navigation

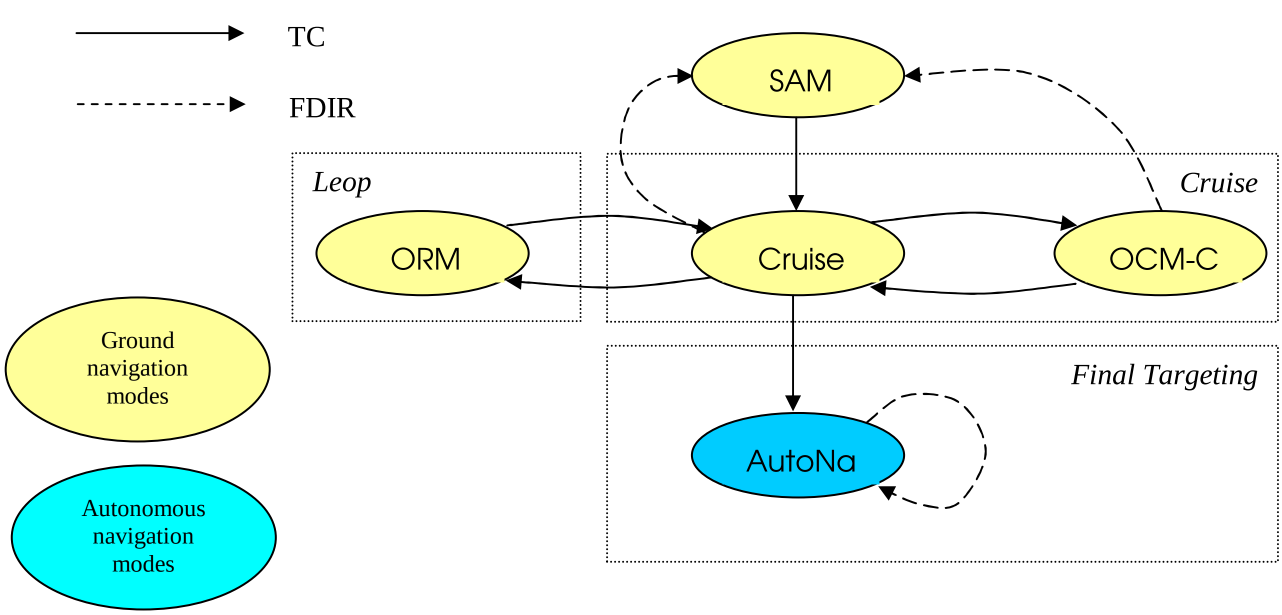
This mode is of course the crucial mode of the Don Quijote mission. A preliminary analysis including numerical simulations has confirmed the findings of the ESA CDF study concerning the number and the timing of the manoeuvres, while stressing the overwhelming importance of measurements filtering.

A number of trade-offs are still on-going:

- STR/NavCam versus NavCam only estimation of the Line of Sight
- Choice of the filtering algorithm (batch, LDO, Extended Kalman) and parameters (duration, frequency)
- Targeting strategy: fixed schedule, fixed error reduction factor, fixed cost of manoeuvres, etc.

A crucial point is to ensure that the time interval between the last two manoeuvres is large enough to perform an efficient filtering of LoS measurements, while leaving enough time to execute the computed manoeuvre. The relevance of warm redundancy for some equipment during this lapse of time will be studied.

Requirements for the Navigation Camera and for the OBDH system will be derived from this study.



	CSS	Gyro	STR	NavCam	RW	RCT
SAM	●	●				●
Cruise			●		●	
ORM		●	●			●
OCM-C		●	●			●
AutoNav		❍	●	●	●	●

Table 3-10: Impactor GNC mode overview

3.1.4 Thermal Control Subsystem sizing

3.1.4.1 Understanding of mission requirements and sizing cases

The design of the thermal control subsystem will depend on the envelope of sizing cases, according to the trajectory design corresponding to the selected potential targets.

Among the important factors is the trajectory design, and the resulting evolution of the sun distance vs. time. The maximum and minimum distance during the cruise phase (for all potential targets) will determine the hot and cold design cases.
The conjunction of spacecraft operating modes (and thus internal dissipation) with the environmental conditions will determine the sizing cases.

The main identified design cases are listed herebelow :

Case	**Conditions**	**Comments**
Earth orbit	1AU Terrestrial IR and albedo	Classical case. Eclipse to be taken into account if any (mission analysis)
Cruise (cold case)	1,39AU	For 2002AT4 trajectory
Cruise (hot case)	0,69AU	For 1989ML trajectory
Final guidance	Dependent on target	Spacecraft in final guidance mode (additional navigation equipment + hot redundancies if any)

Table 3-11: Impactor main identified design cases

3.1.4.2 General design principles

The main purpose of the thermal control is to provide a thermal design which guarantees the requested temperature ranges for all the equipment throughout all the mission phases and in the different operational modes.

The thermal control subsystem will be designed taking into account the following main guidelines:

- Use of well-proven design solutions.
- Minimize cost and budget (mass, power, etc.)
- Meet the thermal requirements

The thermal design will be based on proven solutions like:

- MLI
- Radiators
- Paints and Tape
- Heaters, thermistors and thermostats

Thermal control materials selection will be based on our assessment on the worst case spacecraft environment, and will be based on the knowledge built on other missions with similar requirements.

No use of heat-pipes is foreseen for the moment. However they eventually could be considered later in the study in deemed interesting. For instance solutions using variable conductance heat pipes (VCHP) have been studied in the frame of previous Mars missions (MSR and Mars Premier for CNES). Their main advantage was

- To allow variable coupling of the spacecraft interior to a dedicated radiative surface, thus minimising the required heating power in cold case
- To allow coupling of the attitude control thrusters with the spacecraft body, further reducing the heating power.

Such a solution, from the point of view of the thermal control subsystem, is obviously more complex and costly, but its merit actually depends on the respective weights of the different trade-off criteria, among which mass & cost.

As the primary focus is at the moment on the cost, while the mass of the impactor is not currently considered an issue (mass = kinetic energy, as long as the launcher capability is sufficient), this solution is not investigated further in this proposal.

<u>Case of trajectory with Venus flyby :</u>
The solar array temperature will have to be looked at in detail during the study. Already three possible solutions to this problem can be envisaged :

- Classical solar array technology, tilting the array away from the sun to control its temperature. This could be done either by with a body-mounted solar array by turning the spacecraft, or with a deployable solar array
- High temperature solar array, based on the BepiColombo technology development activities (currently ongoing)

Option	Principle	Pros	Cons
Body mounted low temperature SA	SA temperature controlled by turning the spacecraft	Low cost SA No mechanism	Complexity on AOCS side (including sun-pointing safe mode)
Body mounted high temperature SA	SA can survive normal sun incidence in worst hot case	Simple attitude control No mechanism Robust solution	Cost
Deployable low temperature SA	SA temperature controlled by SA rotation, while S/C stays sun-pointing	Simple attitude control	Additional SADM (cost/risk) Compatibility with attitude control during final guidance?

The option of a high temperature deployable solar array (Venus Express type) will be avoided as it combines the drawbacks of having a costly solar array with the added cost and risk of a drive mechanism.

The solution assumed for the first design iteration of this proposal is the first one (body-mounted SA, low temperature). Preliminary calculations show that a sun incidence of less than 60° would be sufficient to keep the temperature of a body-mounted SA below 140°C when closest to the sun.

3.1.4.3 Heater power budget

The heater power budget has been shown in table AAA and it has been calculated considering:

- the power dissipation reported in the table XXX and
- the temperature requirement reported in table YYY.

Moreover, the external environment has been taken in account assuming entering into module a further heat flux deriving from MLI and external appendixes. It has been estimated in 65 W in Cruise, 15 W in LEOP and zero in the other cases.

Equipment	LEOP CASE [W]	Correction Manouvres CASE [W]	CRUISE CASE [W]	FINAL TARGETING CASE [W]	SAFE CASE [W]
EPC1 X	9,2	9,2	9,2	9,2	9,2
TWTA X	67,2	67,2	67,2	67,2	67,2
GYR	5,775	5,775	5,775	5,775	5,775
RW1	11,55	11,55	11,55	11,55	11,55
RW2	11,55	11,55	11,55	11,55	11,55
RW3	11,55	11,55	11,55	11,55	11,55
RW4	0	0	0	0	0
Nav Cam	0	0	0	0,9	0
CDMU	38,5	38,5	38,5	38,5	38,5
XPND1-RX	14,7	14,7	14,7	14,7	14,7
XPND2-RX	14,7	14,7	14,7	14,7	14,7
XPND-TX	6,3	6,3	6,3	6,3	6,3
XPND-UHF	0	0	0	34,65	0
PCDU	52	52	52	55	51
STR	8,4	8,4	8,4	8,4	0
STR	0	0	0	0	0
Thrusters	32	32	32	32	32
TANK	36	36	36	36	36
RCS	10	10	10	10	10
EXT. Items	20	20	20	20	20
battery	5	5	5	5	5

Table XXX: Electronic Equipment Preliminary Power Dissipation

Equipment	Tmin Op. [°C]	Tmax Op. [°C]	Tmin non Op. [°C]	Tmax non Op. [°C]
TWTA1 X	-20	60	-30	70
EPC1 X	-20	60	-30	70
TWTA X	-20	60	-30	70
EPC X	-20	60	-30	70
RFDN X-Band	-25	55	-35	65
RFDN UHF-Band	-25	55	-35	65
GYR	-15	65	-25	65
RW1	0	55	-10	65
RW2	0	55	-10	65
RW3	0	55	-10	65
RW4	0	55	-10	65
CDMU	-10	50	-20	60
XPND1-RX	-10	45	-20	55
XPND2-RX	-10	45	-20	55
XPND-TX	-10	45	-20	55
XPND-UHF	-10	45	-20	55
PCDU	-10	50	-20	60
STR	-20	40	-30	50
STR	-20	40	-30	50
battery	0	35	0	35

Table YYY: Electronic Equipment Temperature requirements

DESCRIPTION (*)	LEOP CASE [W]	Correction Manouvres CASE [W]	CRUISE CASE [W]	FINAL TARGETING CASE [W]	SAFE CASE [W]
SVM	5	5	5	5	9
FCV thrusters	32	32	32	32	32
TANKS	36	36	36	36	36
Propulsion tubing & valves	10	10	10	10	10
EXT. ITEMS	20	20	30	30	30
BATTERY	5	5	5	5	5
TOTAL	108	108	108	108	112

Table AAA: Heater power budget without margin

(*) In the preliminary assessment for the trusters, tanks, propulsion, battery and external items a fixed value has been assumed.

3.1.4.4 Mass budget

The thermal control mass budget, without margin, is reported in the table below.

Items	Mass [Kg]
Thermal blankets	10.9
Paint/tape	4
Heaters/temp. sensor	2.1
miscellanea	2.4
Total	19.4

3.1.5 On Board Data Handling Subsystem sizing

3.1.5.1 Subsystem drivers

In the SRD only general mission requirements have been indicated. In particular two aspects are underlined for the Impactor:
-RM1150 The Impactor trajectory shall allow for the use of autonomous optical navigation i.e. the target illumination conditions should be such that visual target acquisition can be performed at least 2 days before impact.
-RM1160 The Impactor shall relay images of the target either directly to the GS or through the Orbiter spacecraft.

In particular the RM1160 requirement is addressed during the last mission phase of the Impactor when the optical guide navigation shall be activated to allow at the Impactor to operate as much as possible in autonomous mode.
Because during the last mission phase the Orbiter and Impactor cameras are both activated at the maximum rates and resolution the above last requirement, concerning the autonomy, presents some impacts in terms of mass memory capacity and computational effort.

In additions both the Data Handling System shall be in charge:
- Controlling and managing the payloads on board;
- Acquiring the data incoming from the payloads;
- Storing the scientific and the housekeeping data for a long period;
- Guaranteeing the functional support for the GNC function.

3.1.5.2 Subsystem brief description

Due to commonality objective to reduce the cost, the Data Handling Subsystem is more detailed in the Orbiter design report.
As remind, it is based on distributed architecture, which is based on a common bus with all equipment or payloads connected to the bus. Usually the bus chosen uses a command-response protocol. This architecture is more flexible because it is quite easy to add or remove equipment from the common bus.

On the basis of the ESA information on board of the Impactor the scientific equipment is limited to only the camera and the autonomous guide navigation elaborated autonomously by the GNC. The overall average traffic data rate between the payloads and the CDMU is quite low of the order of tenths kbps. However these low data rate figures does not need an apposite point to point connection between payloads and DHS except camera connection. A common serial bus like 1553B, largely used in the past scientific missions, is in charge to sustain all these data rates without any problem. For the camera, it is forecast a SpaceWire connection (TBC).

3.1.5.3 Image acquisition during the impact phase

Concerning the time and the maximum data rate transferred via RF as a first guess to evaluate the overall amount of data transmitted by the Impactor to the Orbiter has done integrating the maximum data rate obtainable along the impact time. The result obtained is 56.44 Mbits for a maximum integration time of 12 minutes. The minimum data rate is close to 1 kbps increments up to 2 Mbps. It is under investigation the possibility to rise up to 4 Mbps during the transmission but it is necessary to evaluate in more detail the switching time occurring to increase the transmission bandwidth. Anyway it is possible to assume a coarse value of 100 Mbits for mass memory size on board the Impactor.
The images are sent directly via RF to the Orbiter without any data compression to avoid the reduction of data frame acquisition on the last seconds of the mission.

The image data shall be stored internally to the Orbiter and transmitted toward the Earth at the end of the impact phase.
It has been uppsoe that the camera is based on a previous ESA project (IRIS camera) characterised by a12.5 Mbytes/s.This means a maximum of 12 frames full size (1k x 1k) at second. This estimated value could be not true because it depends by the integration time of the image acquired. In particular it has to point out that the asteroid image size is depending by its distance from the spacecraft. In particular, assuming a dimension of the asteroid of 300 m we obtain:

- distance 1865 Km size: 2 pixels
- distance 900 km size: 4 pixels
- distance 3.73 Km size: 1024 pixels

Therefore meaning images are obtained only in the last minute of trajectory.

This resulting microprocessor choice strictly depends by re-usability policy adopted for this project. Nowadays there are two possible choices: the adoption of the well known ERC32 microchip or the usage of a new board generation based on the new LEON2 microchip. Both the processors are valid and the data traffic is limited for both the spacecraft because the image processing is partially elaborated in the DPU. Because the mission is forecast for the 2011 it appears as more suitable choice the LEON II microprocessor in terms of computational power and software tools availability.

3.1.5.4 Impactor DHS Budgets

The overall estimated mass of the Command Data Handling Unit will not exceed the 19 kg.

Dimensions
The preliminary CDMU estimated dimensions are:

Length:	470 mm
Height:	270 mm
Width:	250 mm

In the dimensions are included feet mounting. A margin of 5% is assumed because the box already exists.

3.1.5.5 Power consumption

The CDMU average power consumption will not exceed the 35 W of average power with a peak of 60 W during the switch on phase.

3.1.6 Telemetry Tracking & Command Subsystem sizing

For detailed subsystem design please refer to Annex 1.

3.1.6.1 Link architecture and operative modes

The overall DQ link architecture is shown in Figure 3-1. Three kind of links have been identified:

- An **X band link**, used for communications with Ground Stations. It includes telecommands, housekeeping telemetry, scientific telemetry, data relay forwarding, ranging operations and support to RSE operations. It is used both on Impactor and Orbiter.
- A **Ka band link**, whose only purpose is to perform high resolution RSE operations. It is provided by the Orbiter only.
- An **UHF link**, based on CCSDS Proximity-1 protocol, for low range communications between Orbiter – Impactor and Orbiter-ASP.

For the Impactor, the list of operations vs frequency band and mission phases are resumed in the following table.

System Modes	Description	TT&C Mode	Notes
Pre Launch	On launch pad (30 min)	Launch	units warming up
Launch	From launch to separation (30 min)		
Sun Acquisition	After separation (120min assumed)		
Leop	Earth escape phase encompass orbits & chemical raise manoeuvres. (Not present if Suyuz launch)	Safe	X band
Correction manoeuvres	Correction manoeuvres, eventually fly-by	Cruise	X band
Cruise	Cruise phase		
Final targeting	Close approach to NEO, navigation, payload & proximity UHF link	Final targeting	X band UHF band
Safe	Safe Mode	Safe	X band

Table 3.1.6-1: Impactor operations by mission phase

3.1.6.2 Mass Budget

ACRONYM	TT&C UNIT NAME	Nominal Mass [kg]	Uncertainty: Development Stat. New	Der	Mod	Exis	Mar	Maximum mass [kg]
DST1 X/X	Transponder X/X	3,00			1		10%	3,30
DST2 X/X	Transponder X/X	3,00			1		10%	3,30
X - TWTA1	TWT Amplifier 65W - X Band	1,50				1	5%	1,58
X - EPC1	Electric Power Conditioner for TWTA1	0,80				1	5%	0,84
X - TWTA2	TWT Amplifier 65W - X Band	1,50				1	5%	1,58
X - EPC2	Electric Power Conditioner for TWTA2	0,80				1	5%	0,84
RFDN - X Band	Radio Frequency Distribution Network	3,06						3,21
	X Band - 4-Port Switch 4 0,32	1,28				1	5%	1,34
	X Band - Transfer Switch 2 0,32	0,64				1	5%	0,67
	Isolators 2 0,2	0,40				1	5%	0,42
	X Band - Diplexer 2 0,35	0,70				1	5%	0,74
	X Band - 3dB Hybrid 1 0,035	0,04				1	5%	0,04
External RFDN - X Band	Radio Frequency Distribution Network	3,00						3,60
	2m Waveguide to HGA I/F (250g/m)	0,50	1				20%	0,60
	2m Waveguide to MGA (250g/m)	0,50	1				20%	0,60
	3m Waveguide to LGA1 (250g/m)	0,75	1				20%	0,90
	3m Waveguide to LGA2 (250g/m)	0,75	1				20%	0,90
	2m Waveguide to HGA I/F (200g/m)	0,50	1				20%	0,60
LGA1 - X band	Low Gain antenna	0,30				1	5%	0,32
LGA2 - X band	Low Gain antenna	0,30				1	5%	0,32
MGA - X band	Medium Gain Antenna	3,10						3,72
	MGA X band	0,60	1				20%	0,72
	MGA pointing mechanism -1 dof	2,00	1				20%	2,40
	MGA support boom	0,50	1				20%	0,60
HGA - X Band	High Gain Antenna (1m)	7,40						8,14
	HGA Antenna Reflector Assembly	4,20			1		10%	4,62
	HGA RF parts	2,20			1		10%	2,42
	HGA S/C I/F parts	1,00			1		10%	1,10
TOTAL X-Ka Band TT&C SUBSYSTEM		27,76						30,73
UHF XPND1	Transponder UHF Band1	3,00				1	5%	3,15
UHF XPND2	Transponder UHF Band2	3,00				1	5%	3,15
LGA1 - UHF	Low Gain antenna	1,00			1		10%	1,10
LGA2 - UHF	Low Gain antenna	1,00			1		10%	1,10
RFDN - UHF	Radio Frequency Distribution Network	0,90						0,95
	UHF Band - 4-Port Switch 2 0,2	0,40				1	5%	0,42
	Harness 1 0,5	0,50				1	5%	0,53
TOTAL UHF TT&C SUBSYSTEM		8,90						9,45
GRAND TOTAL TT&C		**36,66**						**40,17**

3.1.6.3 Power Budget

DQ **TT&C Power Consumtion and Dissipation - Impactor**

Unit	LAUNCH			Cruise			Safe Mode			Final Targetting		
		CONSUMPTION	DISSIPATION		CONSUMPTION	DISSIPATION		CONSUMPTION	DISSIPATION		CONSUMPTION	DISSIPATION
XPND1 RX	ON	14	14	ON	14	14	ON	14	14	ON	14	14
XPND1 TX -X	OFF			TX ON	6	3	TX ON	6	3	TX ON	6	3
Total XPND1		14	14		20	17		20	17		20	17
XPND2 RX	ON	14	14	ON	14	14	ON	14	14	ON	14	14
XPND2 TX	OFF			OFF			OFF			OFF		
Total XPND1		14	14		14	14		14	14		14	14
EPC1	Pre Heat	8,8	5,5	Saturation	128	11	Saturation	128	11	Saturation	128	11
X - TWT1	Pre Heat		3,3	TWT ON		52	TWT ON		52	TWT ON		52
Total TWTA1		8,8	8,8		128	63		128	63		128	63
EPC2	OFF	0	0	OFF	0	0	OFF	0	0	OFF	0	0
X - TWT2	OFF	0	0	OFF	0	0	OFF	0	0	OFF	0	0
Total TWTA2		0	0		0	0		0	0		0	0
RFDN	N/A	0	0	N/A	0	12	N/A	0	12	N/A	0	12
Ext. WGs	N/A	0	0	N/A	0	5	N/A	0	5	N/A	0	5
Total TWTA2		0	0		0	17		0	17		0	17
XPND1 UHF RX	OFF			OFF			OFF			OFF		
XPND1 UHF TX	OFF			OFF			OFF			OFF		
Total XPND UHF		0	0		0	0		0	0		0	0
XPND2 UHF RX	OFF			OFF			OFF			ON	18,5	18,5
XPND2 UHF TX	OFF			OFF			OFF			ON	42	34
Total XPND UHF		0	0		0	0		0	0		60,5	52,5
TOTAL		36,8	36,8		162	111		162	111		222,5	163,5

3.1.6.4 RF Loss Budget

The following tables provide the expected losses for the RFDN. Some of the parameters considered are worse with respect to the data provided above as given by the foreseen supplier. This approach has been considered in order to consider some margin for possible degradation of the performances in the integrated system.

UPLINK DATA – X Band

HGA to XPND

	Path		Loss per item NOM	ADV	FAV		Subtotal NOM	ADV	FAV	
Coaxial Cable	1,00	m	0,60	0,66	0,54	dB/m	0,60	0,66	0,54	dB
Switch	4,00	units	0,05	0,06	0,05	dB	0,20	0,22	0,18	dB
Int. Waveguide	3,00	m	0,08	0,09	0,07	dB/m	0,24	0,26	0,22	dB
Rotary Joint	0,00	units	0,40	0,44	0,36	dB	0,00	0,00	0,00	dB
Ext. Waveguide	3,00	m	0,08	0,09	0,07	dB/m	0,24	0,26	0,22	dB
Total RFDN							1,28	1,41	1,15	dB
Total Ext. Waveguide							0,24	0,26	0,22	dB
Diplexer							0,30	0,33	0,27	dB
TOTAL							**1,82**	**2,00**	**1,64**	**dB**

DOWNLINK DATA – X Band

TWTA to HGA

	Path		Loss per item NOM	ADV	FAV		Subtotal NOM	ADV	FAV	
Isolator	1,00	units	0,20	0,22	0,18	dB	0,20	0,22	0,18	dB
Switch	4,00	units	0,05	0,06	0,05	dB	0,20	0,22	0,18	dB
Int. Waveguide	3,00	m	0,08	0,09	0,07	dB/m	0,24	0,26	0,22	dB
Rotary Joint	0,00	units	0,40	0,44	0,36	dB	0,00	0,00	0,00	dB
Ext. Waveguide	3,00	m	0,08	0,09	0,07	dB/m	0,24	0,26	0,22	dB
Total RFDN							0,88	0,97	0,79	dB
Total Ext. Waveguide							0,24	0,26	0,22	dB
Diplexer							0,30	0,33	0,27	dB
TOTAL							**1,42**	**1,56**	**1,28**	**dB**

Table 3.1.6-2: X-Band RFDN Parameters.

3.1.6.5 Link Budgets

3.1.6.5.1 X band link

On the basis of system requirements, cdf study and the design drivers the following assumptions have been considered for the link budgets calculations:

- Orbiter max distance from earth is 2,5 AU, while 1.7 AU is considered as worst-case reference distance for the impact. Minimum G/S elevation is 10°.

- The G/S of Cebreros has been assumed as nominal for the cruise and radio-science phases. The support of 70m NASA DSN Ground stations can be requested for the impact phase to increase the achievable data rate.

- Average data rate required for the radio science phase: 30kbps @ 1.7 AU (TBC). HK data rate: 5kbps.

- TM Encoding Scheme: Turbo code ¼ or standard Concatenated (RS+convolutional) code.

- Spacecraft losses and antenna gains are derived from BepiColombo (see Antenna and RFDN section). In particular:
    - HGA peak gain: 33dBi uplink , 36 dBi downlink
    - MGA peak gain: 23dBi downlink, 21 dBi uplink
    - LGA minimum gain: -3 dBi (hemispherical coverage)
    - RFDN RF losses: 2 dBi uplink, 1,5 dBi downllink

- Due to the very high distance from Earth at opposition, two 65W X Band TWTAs from Mars Express have been chosen as main RF signal amplificators.

Based on the link budgets results, the current configuration provides a data rate 30kbps @ 1.80 AU(impact worst case distance). With the support of a DSN 70m G/S, the data rate can be increased up to about 100kbps.

In case of loss of functionality of the HGA, the MGA can guarantee communications at lower data rates. However a G/S with a low receiver loop bandwidth and high performances (like a 70m DSN G/S) is needed.

During LEOP, only LGAs are available for transmission. Link budgets at the reference distance of 0,01AU are reported.

The link budget resume is shown in the next table.

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DQ IMPACTOR X Band LINK BUDGET RESUME TABLE

DQ UPLINK TABLE (X Band)			LEOP		Cruise		Impact		Safe Mode
			TC + RNG	TC + RNG	TC + RNG	TC + RNG	TC+RNG	TC+RNG	TC+RNG
			27-set-06	28-set-06	29-set-06	29-set-06	29-set-06	29-set-06	27-set-06
			LGA 1-2	CLGA 1-2	HGA 1m	HGA 1m	HGA 1m	HGA 1m	MGA
			NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM
		Ground Station:	Kourou	Perth 35m	Cebreros 35m	DSN70m	New Norcia	DSN70m	Cebreros
distance	1000 km		350	350	380000	380000,00	270000	270000	560000,00
	AU		0,002	0,002	2,540	2,540	1,805	1,805	3,743
TC Bit Rate	**kbps**		**2,000**	**2,000**	**2,000**	**2,000**	**2,000**	**2,000**	**0,200**
TC Mod Index	rad/pk		1,0	1,0	1,0	1,0	1,0	1,0	1,0
RNG Mod Index	rad/pk		0,6	0,6	0,7	0,7	0,7	0,6	0,7
Antenna Gain	dBi		-3,00	-3,00	34,20	34,20	34,20	34,20	21,00
RFDN & WG Losses	dB		1,52	1,52	1,52	1,52	1,52	1,52	1,52
Carr. Power Margin (dB)	**Nom.**	**ESA Margin = 3dB**	**25,52**	**51,56**	**13,13**	**13,13**	**16,09**	**21,95**	**13,56**
	Mean - 3 Sigma	ESA Margin = 0dB	24,32	50,57	12,14	12,14	15,10	21,06	12,57
	Marg - WC-RSS	ESA Margin = 0dB	24,51	50,83	12,35	12,35	15,32	21,28	12,79
Carr. Recov. Margin (dB)	**Nom.**	**ESA Margin = 3dB**	**29,28**	**55,32**	**33,59**	**33,59**	**36,56**	**42,71**	**17,02**
	Mean - 3 Sigma	ESA Margin = 0dB	27,60	53,89	32,02	32,02	34,99	41,30	15,45
	Marg - WC-RSS	ESA Margin = 0dB	27,92	54,16	32,39	32,39	35,36	41,59	15,82
TC Recov. Margin (dB)	**Nom.**	**ESA Margin = 3dB**	**6,88**	**32,93**	**11,19**	**11,19**	**14,16**	**20,31**	**4,62**
	Mean - 3 Sigma	ESA Margin = 0dB	5,53	31,79	9,99	9,99	12,96	19,25	3,42
	Marg - WC-RSS	ESA Margin = 0dB	5,76	32,05	10,26	10,26	13,23	19,49	3,69

DQ DOWNLINK TABLE (X Band)			LEOP		Cruise		Impact		Safe Mode
			TM + RNG	TM + RNG	TM + RNG	TM + RNG	TM+RNG	TM+RNG	TM
			27-set-06	28-set-06	29-set-06	29-set-06	29-set-06	29-set-06	27-set-06
			LGA 1-2	CLGA 1-2	HGA 1m	HGA 1m	HGA 1m	HGA 1m	MGA
			NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM
		Ground Station:	Kourou	Perth 35m	DSN 70m	DSN70m	New Norcia	DSN70m	DSN70m
distance	1000 km		350,00	350,00	380000,00	380000,00	270000,00	270000,00	560000,00
	AU		0,002	0,002	2,540	2,540	1,805	1,805	3,743
TM Bit Rate	**kbps**		**50,00**	**50,00**	**8,00**	**15,00**	**30,000**	**100,00**	**0,200**
Encoding			Concatenated	Concatenated	Concatenated	Turbo 1/4	Turbo 1/4	Turbo 1/4	Concatenated
TM Mod Index	rad/pk		1,20	1,20	1,20	1,20	1,20	1,20	1,20
RNG Mod Index	rad/pk		0,50	0,50	0,50	0,50	0,50	0,50	0,00
Antenna Gain	dBi		-3,00	-3,00	35,60	35,60	35,60	35,60	23,00
RFDN & WG Losses	dB		1,54	1,54	1,54	1,54	1,54	1,54	1,54
Carr. Recov. Margin (dB)	**Nom.**	**ESA Margin = 3dB**	**35,76**	**47,66**	**11,40**	**11,40**	**14,37**	**42,01**	**11,20**
	Mean - 3 Sigma	ESA Margin = 0dB	31,48	43,25	7,19	7,19	10,12	37,49	7,19
	WC-RSS	ESA Margin = 0dB	32,12	43,84	7,82	7,82	10,75	38,16	7,78
TM Recov. Margin (dB)	**Nom.**	**ESA Margin = 3dB**	**5,65**	**17,55**	**3,97**	**3,64**	**3,60**	**4,42**	**4,57**
	Mean - 3 Sigma	ESA Margin = 0dB	4,82	16,96	3,37	3,04	3,01	3,51	3,85
	WC-RSS	ESA Margin = 0dB	4,64	16,99	3,41	3,08	3,05	3,40	3,94
RNG Recov. Margin (dB)	**Nom.**	**ESA Margin = 3dB**	**26,67**	**53,75**	**14,08**	**14,08**	**19,70**	**37,57**	**No RG**
	Mean - 3 Sigma	ESA Margin = 0dB	18,11	43,44	5,64	5,64	11,14	28,14	No RG
	WC-RSS	ESA Margin = 0dB	19,57	45,06	7,04	7,04	12,55	29,75	No RG

3.1.6.5.2 *UHF link*

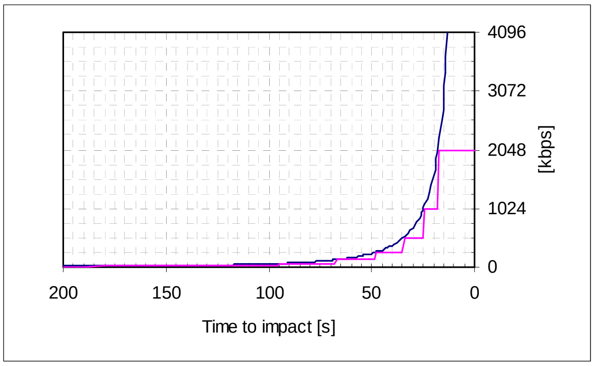
As already stated, the UHF link on the impactor is used to relay data to the G/S passing through the DQ Orbiter, before the impact.

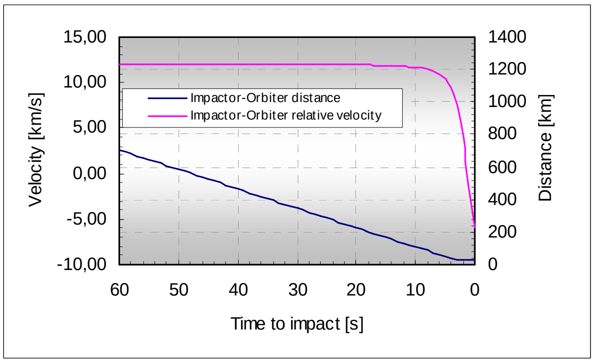
Based on mission analisys and SRD, the communication with the orbiter should start after the last correction manouver has been performed. This happens about 5 minutes before the impact, with the Impactor being at about 5000Km away from the Orbiter.

At that distance a low data rate is assumed, as the onboard camera, which is the main data rate driver during this phase, cannot "see" the target asteroid. The camera can distinguish the asteroid shape at a distance of 900 Km away from the asteroid itself, which happens about 90secs before impact).

The approach taken into the UHF section during the proposal design is to adapt the data rate during the approaching phase. This allows to have lower data rates at higher distances, growing up when the Impactor approaches to the asteroid.

During the proposal a preliminary data rate change plan was presented in the following figure.





However this preliminary approach did not take into account the time needed by the onboard transceivers to handshake the new data rate. This has been estimated with the order of magnitude of about 8-10 seconds. Considering also the short time window available for communication and the very high velocity of the impactor, these few seconds can imply a great data volume losses at the higher data rates.
Also, it is safer for reliability reason to reduce the number of the data rate changes, as one failure during this phase cannot be recovered in time.

3.1.7 Power Subsystem sizing

For detailed subsystem design please refer to AAS-I document "Impactor Power Subsystem".

The equipment power consumption reported in the NEO power Budget depends on the maturity status of the equipment.

The following equipment contingency has been considered in relation to its level of development:

1. 5% for the off-the-shelf equipment (e.g. Battery, XPND, TWTA, AOCS equipment etc);
2. 10% for the item to be modified (e.g. CDMU, PCDU etc);
3. 15% for the item to be developed.

A system margin of 10% has been applied to the NEO Impactor satellite to take into account the uncertainties of the model used to determinate the power budget at system level.

In addition 1 string has been subtracted to SA and Battery total string to take into account the potential failure.

In order to verify the necessary power margin for all possible orbits relevant to the Impactor mission, an analysis tool has been refined using an Excel spreadsheet.

For using this tool it is necessary to define the boundary condition (i.e. Orbit parameters, Solar Panel Sun Aspect Angle, Solar Panel Temperature, Sun Distance, Failure, Life and Degradation) and to select an EPS architecture (i.e. power bus regulated, BCR/BDC DC/DC step-up, SA MPPT regulator)

The Excel tool developed by AAS-I has been used to:

1. Sizing the Battery in term of number of cells (series and parallel) to supply, during the eclipse phase, the Impactor electrical load maintaining, in worst conditions the bus voltage within the range 28 Vdc +/- 0,5 % along the orbit or phase

2. Sizing the Solar Array in term of number of cells (series and parallel) to supply, during the sun phase, the Impactor electrical load and also the recharging of the Battery maintaining, in worst conditions the bus voltage within the range 28 Vdc +/- 0,5 % along the orbit or phase

3. Determination along the orbit or operational phase the State of Charge (SoC) of the Battery

4. Determination of the power required to SA for recharging of the Battery during the charge phase (sun phase)

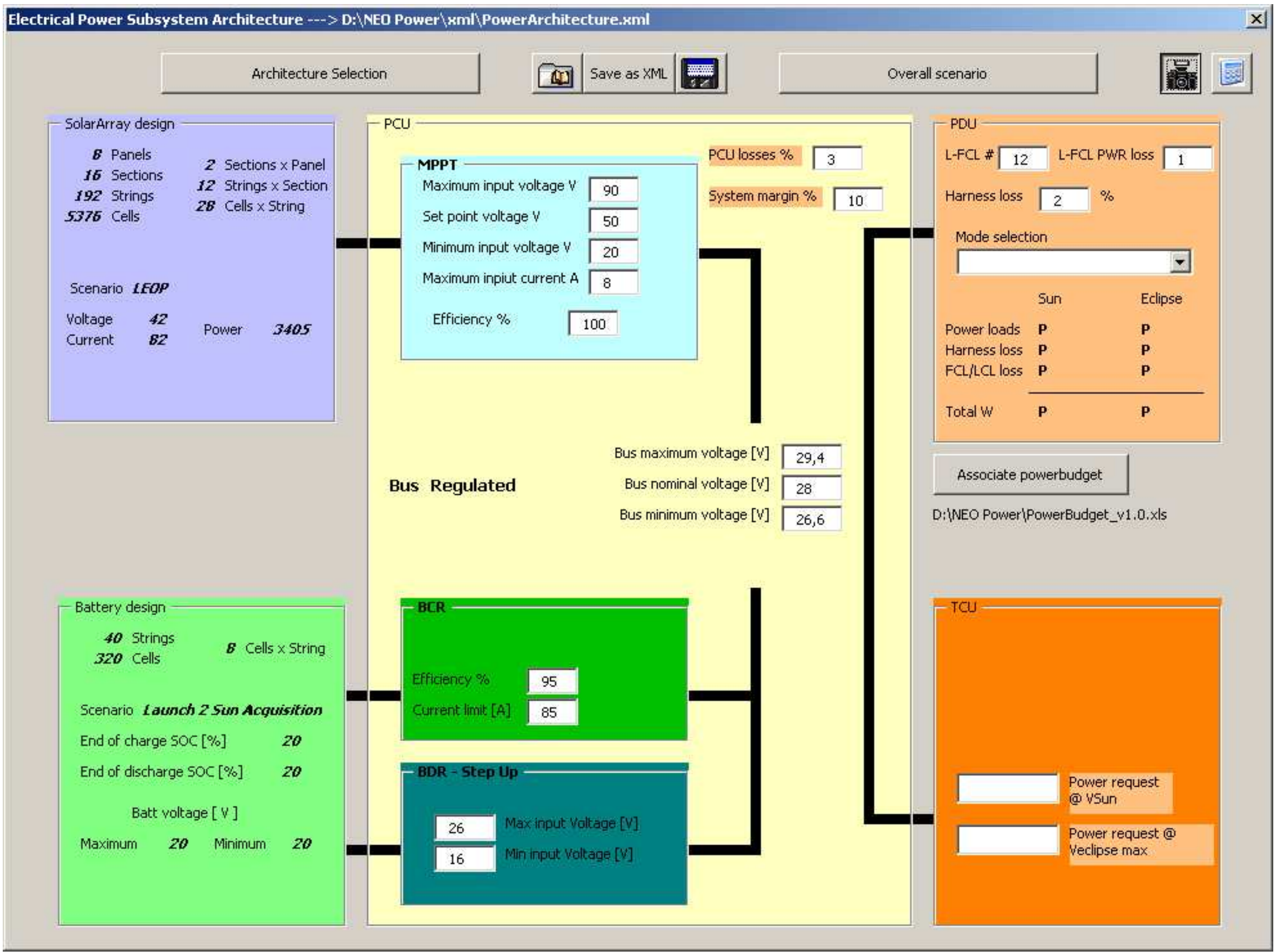
5. Determination of the power dissipation of the PCDU due to the power regulation, conversion and distribution

6. Determination of the mass of the main part of the EPS : SA, BATT and PCDU

7. Determination of the Power dissipation of the EPS BATT and PCDU

8. Determination of the power margin for each operation phase and mode to determinate the boundary condition needed to define the suitable nominal operation.

In the following figure the main window of the excel spreadsheet tool used to perform the power budget and calculate the power margins is shown.


Electrical Power Subsystem Architecture ---> D:\NEO Power\xml\PowerArchitecture.xml
Architecture Selection
Save as XML
Overall scenario
SolarArray design
8 Panels
16 Sections
192 Strings
5376 Cells
2 Sections x Panel
12 Strings x Section
28 Cells x String
Scenario LEOP
Voltage 42
Current 82
Power 3405
PCU
MPPT
Maximum input voltage V 90
Set point voltage V 50
Minimum input voltage V 20
Maximum inpiut current A 8
Efficiency % 100
PCU losses % 3
System margin % 10
Bus Regulated
Bus maximum voltage [V] 29,4
Bus nominal voltage [V] 28
Bus minimum voltage [V] 26,6
PDU
L-FCL # 12
L-FCL PWR loss 1
Harness loss 2 %
Mode selection
Sun
Eclipse
Power loads P P
Harness loss P P
FCL/LCL loss P P
Total W P P
Associate powerbudget
D:\NEO Power\PowerBudget_v1.0.xls
Battery design
40 Strings
320 Cells
8 Cells x String
Scenario Launch 2 Sun Acquisition
End of charge SOC [%] 20
End of discharge SOC [%] 20
Batt voltage [V]
Maximum 20
Minimum 20
BCR
Efficiency % 95
Current limit [A] 85
BDR - Step Up
26 Max input Voltage [V]
16 Min input Voltage [V]
TCU
Power request @ VSun
Power request @ Veclipse max

The power margin values have been calculated to provide an overview of the Impactor power situation for each case as result from the simulation case:

- Negative value means that the EPS can not be able to provide sufficient power/energy to supply the specified loads (SoC of the battery at the beginning of the orbit is higher than the SoC at the end). This means that the power load demand needs to be reduced by this amount in order to get stable condition.
- Positive value means that the EPS is able to provide sufficient power/energy to supply the loads Power Load demand (SoC of the battery at the beginning of the orbit is equal to the SoC at the end). This means that the power load demand can be increased by this amount in order to get stable condition.
- Zero values means that the EPS is able to provide the sufficient power/energy to supply the loads Power Load demand (SoC of the battery at the beginning of the orbit is equal to the SoC at the end). This means that the power load demand can not be increased at all.

To provide the power budget of the Impactor the following main assumptions (very conservative) have been considered:

<u>For SA</u>:

1. For LEOP phase only solar flux (1375 W/m2@ 1 AU) has been considered to calculate the SA power generation. No contribution of Earth Albedo factor has been considered.
2. For the other phases/modes (Cruise, Final Target) the determination of the solar flux has been based on the formula :

   Solar Flux @ Distance = Solar Flux @ 1AU/Distance^2

3. For the SA cells an End of Life (EoL) degradation of 1e+15 [Mev]
4. A failure on one string
5. The maximum temperature of 100°C @1AU and 10°C @2,7AU
6. The fill factor is 0,85

<u>For Battery</u>:

1. The temperature constant of 40°C
2. The cells degradation (EoL) of 0,98 considering 100 cycles of charge/discharge
3. A failure on one string
4. The maximum DoD of 88% at the beginning of LEOP phase (1 time) considering that this phase start with a maximum duration of eclipse (36,6 minutes) and minimum duration of the sun phase (55,8 minutes)
5. A initial SoC of 98% has been considered taking into account the initial capacity degradation due to the storage and pre-launch test.

For PCDU:

1 It is assumed a power consumption of 49,5 W (average) in eclipse phase due to the power consumption of the internal power conversion and internal electronic.

2 It is assumed a power consumption of 110 W (average) in sun phase due to the power consumption of the internal power conversion, internal electronic and MPPT control electronic power efficiency.

3 Power distribution Loss of 3% due to the LCL/FCL power loss that is function of the power requests.

4 Harness losses of 3% due to the power dissipation along the harness from SA to PCDU and between PCDU and Loads.

For Loads:

5 The power consumption of electronic unit has been considered taking into account the average figure with a dedicated uncertatains figure depending of the maturity of the design

6 The power consumption of the thermal control has been considered taking into account the figure coming from the thermal analysis

8 The power consumption of the AOCS Thrusters has been considered in the power budget

9 The ABM has been used during the LEOP. In the power budget a power consumption of **38,8 W** has been taking into account considering a contingency of 5%..

10 Harness losses of 3% due to the power dissipation along the harness from SA to PCDU and between PCDU and Loads.

A system margin of 10 % is added on estimated load to cover any uncertainties of the analysis performed with the Excel spreadsheet tool.

The following modes and conditions have been used to size the Impactor SA :

IMPACTOR	Pwr Bdgt for IMPACTOR - DNEPR/SOUZ Launcher - SA Sizing Cases			Solar Array : 3 body panels (each 2 sections x 8 strings x 28 cells		Data : 27/09/06
	Modes					
	SUN ACQUISITION	CRUISE		FINAL TARGET		
Study Case		case 1	case 2	case 1	case 2	case 3
Sun Distance [AU]	1	1,39	0,7	1,23	1,23	1,23
SAA [°]	30°	10°	60°	0°	0°	0°
Temperature [degC]	100 °C	50 °C	130 °C	50°C	50°C	50°C
Degradation	BOL	EOL	EOL	EOL	EOL	EOL
SA Failure	1 string	1 string	1 string	1 string	1 string	1 string
SA configuration	28cx8strx2secx3BPs	28cx8strx2secx3BPs	28cx8strx2secx3BPs	28cx8strx2secx3BPs	28cx9strx2secx3BPs	28cx10strx2secx3BPs
SA area [m2]	5,1	5,1	5,1	5,1	5,7	6,35
SA mass [Kg]	18,0	18,0	18,0	18,0	20,5	22,7
Battery configuration	6sx55p	6sx55p	6sx55p	6sx55p	6sx55p	6sx55p
Battery mass [Kg]	12	12	12	12	12	12
Battery Failure	1 string	1 string	1 string	1 string	1 string	1 string
Eclipse/Sun time	no eclipse	no eclipse	no eclipse	no eclipse	no eclipse	no eclipse
S/C Load [W]	334,6	583,3	583,3	745,5	745,5	745,5
Battery Recharging Load [W]	0	0	0	0	0	0
Power from Battery [W]	0	0	0	70,5	0,0	0,0
Total Power Required [W]	334,6	583,3	583,3	745,5	745,5	745,5
SA Power generation [W]	858	523	758	675	759	844
Power Margins [W]	**523,4**	**-60,3**	**174,7**	**0,0**	**13,5**	**98,5**
					1 day of battery autonomy	

On the basis of the estimated power margin calculated by the Excel spreadsheet tools confirms that the designed SA with an area of 5 m^2 equipped with 1344 triple junction cells (GAGET 2/160-8040) is sufficient to support any operation and relevant Impactor mode.

The following table summarize the calculation of the power margins estimated by the Excel spreadsheet tools for the others IMPACT modes :

IMPACTOR	Modes			Date : 28/09/06
Parameters	**caso 1**	caso 2		
Modes	LEOP		SAFE	Correction Manouvre
Sun Distance [AU]	1	1	1	1
SAA [°]	0°	0°	0°	0°
Temperature [degC]	100 °C	100 °C	100 °C	100 °C
Degradation	BOL	BOL	EOL	EOL
SA Failure	1 string	1 string	1 string	1 string
SA	28cx8strx2secx3panels	28cx8strx2secx3panels	28cx8strx2secx3panels	28cx8strx2secx3panels
Battery	6sx50p	6sx50p	6sx50p	6sx50p
Batter Failure	1 string	1 string	1 string	1 string
Eclipse/Sun time	36,6 min/55,8 min	40 min/50 min	-	-
S/C Load [W]	479,5	479,5	560,5	831,5
Battery Recharging Load [W]	400	500	0	1
Total Power Required [W]	879,5	979,5	560,5	660,4
SA Power generation [W]	987	987	840,0	840,0
Power Margins [W]	**107,5**	**7,5**	**279,5**	**179,6**

3.1.8 Harness

At this study step, the harness mass, to be considered in further SC budgets, will be estimated thanks to a statistical tool based on flight programs.

But this approach shall be balanced by the specific Impactor configuration: a “separate like” CPM but not release after Earth escape due to the need of high final dry mass (and of chemical propulsion for further phases). Effectively the harness is well concentrated inside the platform box (all avionics equipments are on this part) but some cabling is necessary towards the CPM (in particular for thrusters valves and tanks heaters powering).

That is why the assessment of the Impactor harness mass is based on the average approach between:

- the total SC dry mass that provides around 26kg without margin,
- the total CPM surface considered (as cylinder like) that provides around 30kg without margin,

So the resulting harness mass in first approach is 28kg without margins (20% more considering the assessment accuracy!)

3.2 Option 2.1 design: Impactor for Dnepr launch

This option corresponds to a common extended Exomars propulsion module (2.4T) with the SVM platform fixed on the top. The launch vehicle is Dnepr with 1986ML worst-case target.

3.2.1 Configuration

One stowed view under Dnepr fairing (Figure 3-3) and one deployed view (Figure 3-4) are provided.

The stowed configuration under Dnepr shows the margins in regard to the fairing envelope:

- it seems possible to install a MGA deployed (if rotation mechanism not necessary)
- it is possible to increase only the SerVice Module SA

The deployed configuration is close to the impact one: @ impact, the MGA shall be moved by only 1 Deg from the stowed position to point the Earth (the Earth aspect angle is 89 Deg) considering the asteroid impacting direction is uncluded in the plan (X,Y) with 34 Deg around Z.

Due to Sun angle of 34 Deg, the SA shall not be aligned towards Sun axis at impact so the SA surface required by power calculations is increased consequently (considering Sun distance @ impact) but other alternative solutions exist to release the SA deployment mechanisms:

- to add a motion mechanical axis to the MGA,
- to tilt the 10N thrusters with 45 Deg around X axis.

However to add an antenna motorisation axis adds cost, mass and risk.

And if the thrusters are tilted, then the MGA support shall be integrated with a fixed bias or another antenna motorisation axis shall be added, because in this case the rotation around impact axis is not possible (the impact axis is not collinear with the SA axis).

During the cruise phase, to perform TTC link, it is not a problem to rotate the Impactor around the Sun axis (the only constraint is the SA pointing towards the Sun).

So, if retained by ESA as a promising solution, the Impactor configuration compliant with Dnepr should be deeply studied to define the best trade between SA surface increased, MGA new motorisation axis and thrusters tilt (considering other target and back trajectories).

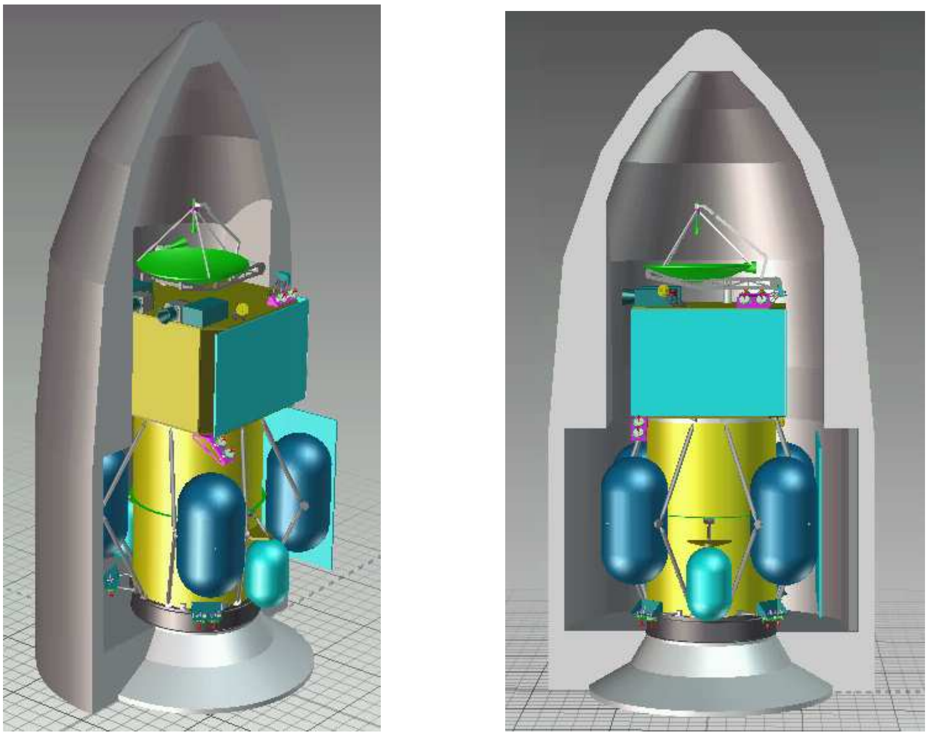


Figure 3-3: Impactor option 2.1 under Dnepr fairing

To be provided

Figure 3-4: Impactor option 2.1 in deployed configuration

3.2.2 *Mechanical*

The structure mass budgets is separated in 3 main parts (Table 3-3):

- the SVM or platform box
- the mechanisms (only one axis for MGA)
- the propulsion module (used separately by Orbiter)

The total platform structure mass (including the mechanisms) is around 55kg with 12% margins and the CPM structure mass is around 115kg with 10% margins considering new tanks supports.

COMPONENT	Reference		Qty	Unit mass (kg)	Class (1 or 2)	Total mass (kg)	STATUS E (%)	C (%)	Q (%)	W (%)	Uncert. (g)	Disp. (g)	Max. mass (kg)
structure prop module						**106,87**							**113,86**
tube central + accessoires (inserts, fixation tank	exomars		1,9	21,50	1	40,85	50	50	0	0	3064	0	43,91
IF ring	exomars		1	20,00	1	20,00	0	100	0	0	1000	0	21,00
DM IF ring pour SVM (Φ1194)	exomars		1	8,00	1	8,00	0	100	0	0	400	0	8,40
struts support GS + thrusters	RTM + alu	treillis barres	12	0,19	2	2,23	100	0	0	0	445	0	2,67
10N thruster support	alu		4	0,50	2	2,00	100	0	0	0	400	0	2,40
MMH Tanks Struts			16	1,22	1	19,49	0	100	0	0	974	0	20,46
MMH Tanks Monopod			8	1,00	1	8,00	0	100	0	0	400	0	8,40
Helium Tanks Support	exomars		4	0,80	1	3,20	0	100	0	0	160	0	3,36
Main Engine support	exomars		1	3,10	1	3,10	0	100	0	0	155	0	3,26
structure SVM (1380x1380x980)						**42,95**							**48,16**
panneaux latéraux + inserts			4	3,93	1	15,71	50	50	0	0	1178	0	16,89
panneaux top/bottom + inserts + IF tube central			2	7,12	1	14,25	100	0	0	0	1425	0	15,67
éléments de liaisons	62kg sur TURSAT pour 350kg CU		1	4,00	2	4,00	100	0	0	0	800	0	4,80
structure porteuse antenne			1	3,00	2	3,00	100	0	0	0	600	0	3,60
support RW			4	1,00	2	4,00	100	0	0	0	800	0	4,80
support SST		monobloc	2	1,00	2	2,00	100	0	0	0	400	0	2,40
Mécanismes (dans SVM)						**5,85**							**6,16**
mécanisme antenne (déploiement motorisé cor	ADPM TURKSAT 12kg solar array depl. = 6,9kg		1	3,50	2	3,50	0	0	100	0	0	70	3,57
stowing mechanism (pour antenne)	TURKSAT		1	2,35	2	2,35	0	100	0	0	235	0	2,59
						155,7	**33**	**58**	**0**	**10**	12811	0	**168,2**

Table 3-3: Option Dnepr Impactor structure mass budget

3.2.3 *Propulsion module*

The proposed Propulsion Subsystem for the common chemical propulsion modules is a bi-propellant subsystem using MON and MMH as propellant and gaseous Nitrogen as pressurant. Boosts are performed with a 500N ABM in pressurised mode. Eight 10N thrusters and eight 22N are used for the orbit raising and orbit control by providing thrust and torque to the spacecraft and can be easily used in pressurised mode as in blow down. The design is axed around a high level of reliability to assure success to the mission with the best confidence in the performances of the propulsion of the spacecraft.
The general configuration of the propulsion subsystem is widely inherited from EXOMARS configuration of which first launch window is foreseen in 2011.

Item	Performances	Status	Heritage
MON Tank	1 central MON tank OST 22/X (1108L)	Space qualified	AMC9
MMH Tank	4 external MMH Tanks OST 06/X (267L)	Space qualified	INSAT
He Tank	2 HE tank PSI 80400-1	Space qualified	A2100
ABM	1 EADS EAM 500N	In development for @bus	
10N Thrusters	8 10N thrusters S10-18	Space qualified	SPACEBUS family
22N Thrusters	8 22N thrsuters S22-02		

A synoptic of the Chemical propulsion Module is presented Figure 3-5, considering the main engine is insulated after the Earth escape so the fly by Venus is performed thanks to 22N thrusters.

The resulting total mass (Table 3-4) of the Dnepr CPM with a max capacity of 2407kg is around 180kg with 10% subsystem margins included due to the maturity level of this propulsion module.

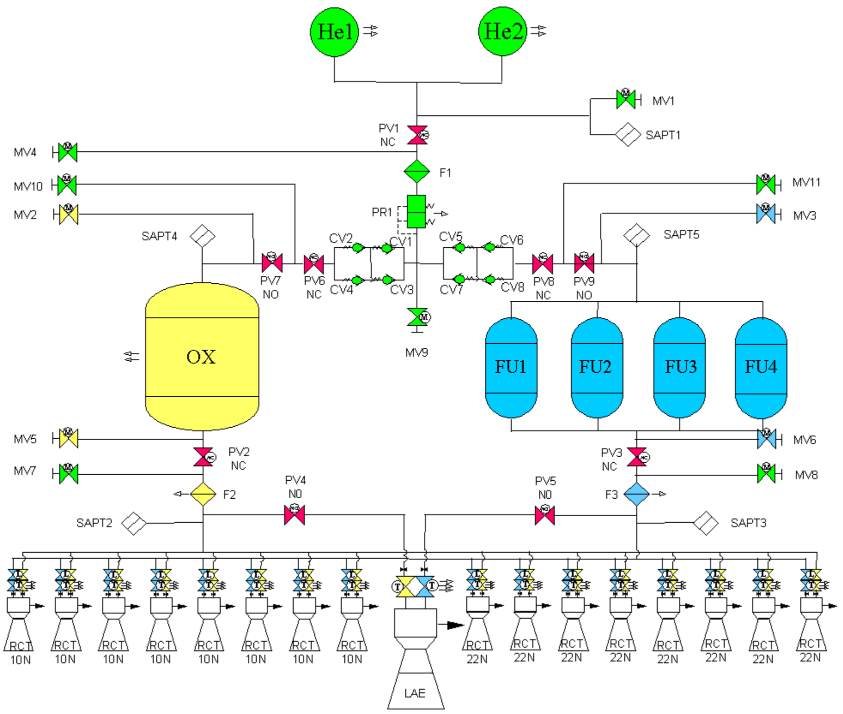


Figure 3-5: Option Dnepr CPM synoptic

FUNCTIONAL SUBSYSTEM	No	Mass (kg) per unit	Total Mass (kg)	Margin (%)	Margin (kg)	Mass with Margin (kg)
PROPULSION			167,92	6,1	10,31	178,22
MON Tank	1	49,00	49,00	5,0	2,45	51,45
MMH Tanks	4	16,00	64,00	5,0	3,20	67,20
He Tank	2	9,98	19,96	5,0	1,00	20,96
500N engine with support	1	9,50	9,50	10,0	0,95	10,45
THR 10N	8	0,65	5,20	5,0	0,26	5,46
THR 22N	8	0,65	5,20	5,0	0,26	5,46
Miscellaneous			15,06	14,6	2,19	17,25
Tubing and Fitting	1	8,00	8,00	20,0	1,60	9,60
Pressure regulator	1	1,18	1,18	5,0	0,06	1,24
Check Valves	8	0,09	0,72	5,0	0,04	0,76
Propellant Fill and Drain Valves	4	0,05	0,19	5,0	0,01	0,20
Pressurant FdV or Test Port	7	0,05	0,34	5,0	0,02	0,35
Pressurant Filter	1	0,08	0,08	5,0	0,00	0,08
Propellant Filter	2	0,25	0,50	5,0	0,03	0,53
Pyro Valve N/C	5	0,15	0,74	5,0	0,04	0,78
Pyro Valve N/O	4	0,15	0,61	5,0	0,03	0,64
Propellant Latch Valve	0	0,34	0,00	5,0	0,00	0,00
Pressurant Latch Valve	0	0,35	0,00	5,0	0,00	0,00
Pressure Transducer	5	0,22	1,11	5,0	0,06	1,16
Module 2 alignement set	1	0,47	0,47	20,0	0,09	0,56
10N thruster alignement set	1	0,22	0,22	20,0	0,04	0,26
Supports + screw (FdV, PR, PV, SAPT)	1	0,90	0,90	20,0	0,18	1,08

Table 3-4: Option Dnepr propulsion module mass budget

3.2.4 System Budgets

The mass budget of the Impactor impacting 1989ML with Dnepr launcher (Table 3-5) shows:

- a consequent margin (around 16%) in regard to launcher capacity
- the total fuel capacity of the CPM (2407kg) compatible with propellant budget (1983kg)
- a ballast mass (5kg) to raise the total mass at impact

Due to the margins of the CPM and launcher capacities, it could be interesting instead to add ballast mass to complete the fuel tanks as security propellant in case of launcher under performances.

NB: to optimize the calculation, the propellant budget was separated in 2 parts:

- the fuel mass (Table 3-7) necessary for orbit correction during the cruise phase and during impact phase calculated on the basis of the SC dry mass with ESA margins
- the fuel mass (Table 3-17) necessary for Earth escape (the attitude correction is introduced as a reduction of the main engine ISP) calculated on the basis of the total launch mass (considering the cruise and impact fuel as dry mass)

The option 2.1 (1989ML target with Dnepr) Impactor mass budget with ESA margins is 682kg including 5kg of ballast to respect the impact mass. The resulting propellant mass is 1983kg, including 24kg for the cruise and impact phases so its is compliant to CPM max capacity (2407kg).

Element 1	Impactor						
			Target Spacecraft Mass at Launch			3300,00	kg
			Below Mass Target by:			583,51	kg
Input Mass	Input Margin	Dry mass contributions	Without Margin	Margin %	Margin kg	Total kg	% of Total
EL			Structure	155,67 kg	8,03	12,51	168,18	6,19
EL			Thermal Control	19,40 kg	20,00	3,88	23,28	0,86
EL			Communications	36,66 kg	10,01	3,67	40,33	1,48
EL			Data Handling	19,00 kg	5,00	0,95	19,95	0,73
EL			AOCS	35,04 kg	10,66	3,74	38,78	1,43
EL			Propulsion	167,92 kg	6,14	10,31	178,22	6,56
EL			Power	52,00 kg	20,00	10,40	62,40	2,30
DI	28,00	20,00	Harness	28,00 kg	20,00	5,60	33,60	1,24
			Total Dry(excl.adapter)	513,69			564,74 kg	
			System margin (excl.adapter)		20,00 %		112,95 kg	
			Total Dry with margin (excl.adapter)				677,69 kg	
			Other contributions					
DI	5,00	0,00	Ballast (add. fuel)	5,00 kg	0,00	0,00	5,00	0,18
			Wet mass contributions					
DI	1983,80	0,00	Propellant	1983,80 kg	0,00	0,00	1983,80	73,03
			Adapter mass (including sep. mech.), kg	50,00 kg	0,00	0,00	50,00	0,02
			Total wet mass (excl.adapter)				2666,49 kg	
			Launch mass (including adapter)				2716,49 kg	

Table 3-5: Mass budget of the Dnepr option with 1989ML target

PROPELLANT BUDGET	Isp	DV Nom	Margin	DV Max
Escape Manœuvre	324	4100,0 m/s	5%	4 305,0
Trajectory Manœuvre	290	0,0 m/s	100%	0,0
Attitude Control	290	0,0 m/s	100%	0,0
TOTAL DV	**324**	**4 100,0**		**4 305,0**
TOTAL Propellant Mass		**1349,8kg**		**1959,4kg**

Table 3-17: Mass fuel budget for Earth escape on Dnepr with 1989ML target

PROPELLANT BUDGET	Isp	DV Nom	Margin	DV Max
Escape Manœuvre	324	0,0 m/s	5%	0,0
Trajectory Manœuvre	290	40,0 m/s	100%	80,0
Attitude Control	290	10,0 m/s	100%	20,0
TOTAL DV	**290**	**50,0**		**100,0**
TOTAL Propellant Mass		**9,1kg**		**24,4kg**

Table 3-7: Mass fuel budget for cruise and impact of 1989ML target

3.3 Option 3.2 design: Impactor for Vega launch

This option corresponds to a SpaceBus flight proven propulsion module (1.4T) with the SVM platform fixed on the top. The launch vehicle is Vega with 2002AT4 target (due to Delta V and impact mass required, Vega is not compatible to 1989ML target).

3.3.1 Configuration

One stowed view under Vega fairing (Figure 3-6) and one deployed view (Figure 3-7) are provided.

The stowed configuration under Vega shows the margins in regard to the fairing envelope:

- it seems possible to install a MGA deployed (if rotation mechanism not necessary)
- it is possible to increase only the CPM SA

The deployed configuration is close to the impact one: @ impact, the MGA shall be moved by 72 Deg from the stowed position to point the Earth (the Earth aspect angle is 18 Deg) considering the asteroid impacting direction is included in the plan (X,Y) with 17 Deg around Z @ impact.

Due to Sun angle of 17 Deg, the SA shall not be aligned towards Sun axis at impact so the SA surface required by power calculations is increased consequently (considering Sun distance @ impact) but depending on resulting mass (SA+mechanisms) other alternative solutions could be studied as addind a motion mechanical axis to the MGA.
The tilt of the 10N thrusters impacts largely the design but could be retained in regard to the range of the Sun angles @ impact (if both targets and back-up trajectory are considered).

During the cruise phase, to perform TTC link, it is not a problem to rotate the Impactor around the Sun axis (the only constraint is the SA pointing towards the Sun).

So, if retained by ESA as a promising solution, the Impactor configuration compliant with Dnepr should be deeply studied to define the best trade between SA surface increased, MGA new motorisation axis and thrusters tilt (considering other target and back trajectories).

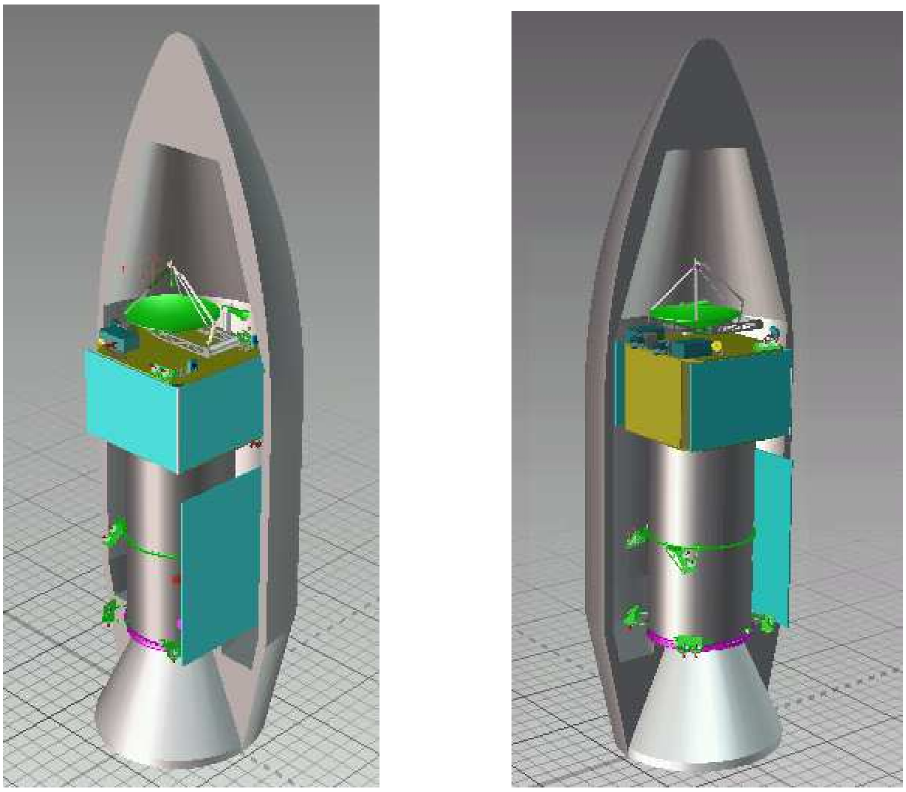


Figure 3-6: Impactor option 3.2 under Vega fairing

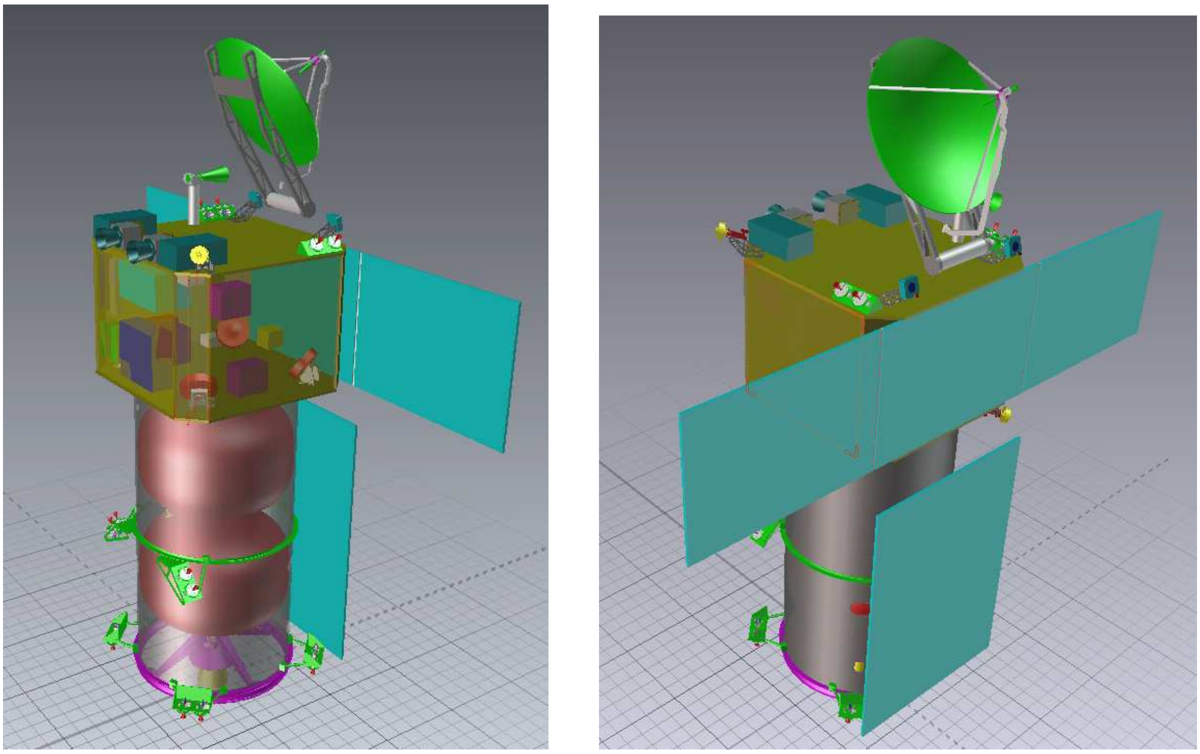


Figure 3-7: Impactor option 3.2 in deployed configuration

3.3.2 Mechanical

The structure mass budget is separated in 3 main parts (Table 3-19):

- the SVM (or platform box)
- the mechanisms (only one axis for MGA)
- the propulsion module (used separately by Orbiter)

The total platform structure mass (including the mechanisms) is around 55kg with 12% margins and the CPM structure mass is around 93kg with 7% margins considering flight proven supports.

COMPONENT	Reference	Qty	Unit mass (kg)	Class (1 or 2)	Total mass (kg)	STATUS E (%)	C (%)	Q (%)	W (%)	Uncert. (g)	Disp. (g)	Max. mass (kg)
structure prop module					87,43							93,62
tube central + accessoires	exomars	2,2	21,50	1	47,30	50	50	0	0	3548	0	50,85
IF ring	exomars	1	20,00	1	20,00	0	100	0	0	1000	0	21,00
DM IF ring Cannes	exomars	1	8,00	1	8,00	0	100	0	0	400	0	8,40
struts support GS + thrusters	RTM + alu treillis barres	12	0,19	2	2,23	100	0	0	0	445	0	2,67
10N thruster support	alu	4	0,50	2	2,00	100	0	0	0	400	0	2,40
MMH Tanks Struts	exomars	0	1,22	1	0,00	0	100	0	0	0	0	0,00
MMH Tanks Monopod	exomars	0	1,00	1	0,00	0	100	0	0	0	0	0,00
Helium Tanks Support	exomars	2	0,80	1	1,60	0	100	0	0	80	0	1,68
ME support	exomars	1	3,10	1	3,10	0	100	0	0	155	0	3,26
Helium Tanks Support		4	0,80	1	3,20	0	100	0	0	160	0	3,36
structure SVM (1380x1380x980)					42,95							48,16
panneaux latéraux + inserts		4	3,93	1	15,71	50	50	0	0	1178	0	16,89
panneaux top/bottom + inserts + IF tube central		2	7,12	1	14,25	100	0	0	0	1425	0	15,67
éléments de liaisons	62kg sur TURSAT pour 350kg CU	1	4,00	2	4,00	100	0	0	0	800	0	4,80
structure porteuse antenne		1	3,00	2	3,00	100	0	0	0	600	0	3,60
support RW		4	1,00	2	4,00	100	0	0	0	800	0	4,80
support SST	monobloc	2	1,00	2	2,00	100	0	0	0	400	0	2,40
Mécanismes (dans SVM)					5,85							6,16
mécanisme antenne (déploiement motorisé cor	ADPM TURKSAT 12kg solar array depl. = 6,9kg	1	3,50	2	3,50	0	0	100	0	0	70	3,57
stowing mechanism (pour antenne)	TURKSAT	1	2,35	2	2,35	0	100	0	0	235	0	2,59
					136,2	**40**	**49**	**0**	**11**	11626	0	**147,9**

Table 3-19: Option Vega Impactor structure mass budget

3.3.3 Propulsion

The proposed Propulsion Subsystem for the common chemical propulsion modules is a bi-propellant subsystem using MON and MMH as propellant and gaseous Nitrogen as pressurant. Boosts are performed with a 400N ABM in pressurised mode. Six-teen 10N thrusters are used for the orbit raising and orbit control by providing thrust and torque to the spacecraft and can be easily used in pressurised mode as in blowdown. The design is axed around a high level of reliability to assure success to the mission with the best confidence in the performances of the propulsion of the spacecraft.
The general configuration of the propulsion subsystem is identical to SPACEBUS configuration. Only orientation of the thrusters have been modified in order to optimize their effect for this specified mission.

Item	Performances	Status	Heritage
MON Tank	1 central MON tank OST 22/X (700L)	Space qualified	Artemis, Arabsat
MMH Tank	1 central MON tank OST 22/X (700L)	Space qualified	Artemis, Arabsat
He Tank	2 EADS He tank (89.5L)	Qualified	Implemented on CIEL 2
ABM	1 EADS S400-15 424N	Space qualified	SPACEBUS Family
10N Thrusters	16 10N thrusters S10-18	Space qualified	SPACEBUS family

A synoptic of the Chemical propulsion Module is presented Figure 3-8, considering the main engine is insulated after the Earth escape.

The resulting total mass (Table 3-20) of the Vega CPM with a max capacity of 1542kg is around 135kg with 8% subsystem margins included due to the high maturity level of this flight proven propulsion module.

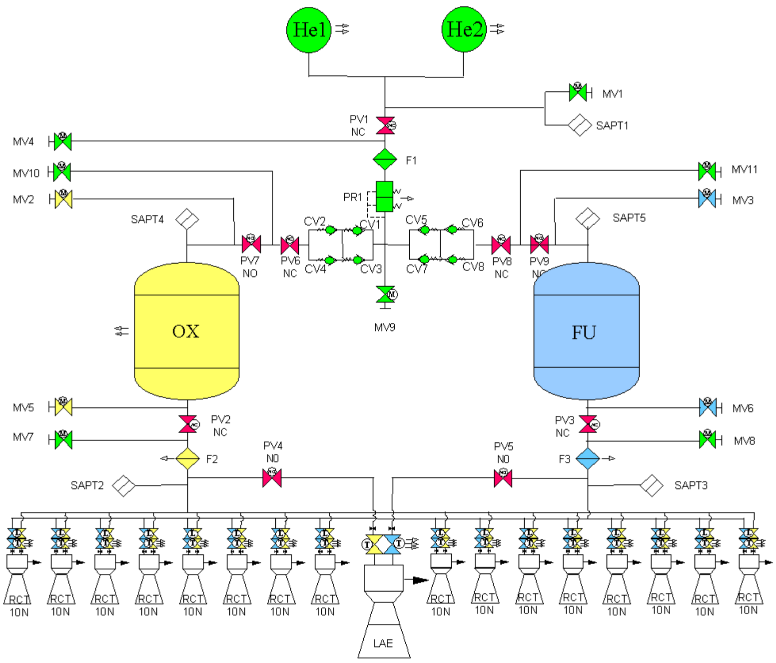


Figure 3-8: Option Vega CPM synoptic

FUNCTIONAL SUBSYSTEM	No	Mass (kg) per unit	Total Mass (kg)	Margin (%)	Margin (kg)	Mass with Margin (kg)
PROPULSION			126,92	6,5	8,26	135,17
MON Tank	1	36,00	36,00	5,0	1,80	37,80
MMH Tanks	1	36,00	36,00	5,0	1,80	37,80
He Tank	2	9,98	19,96	5,0	1,00	20,96
400N engine with support	1	9,50	9,50	10,0	0,95	10,45
THR 10N	16	0,65	10,40	5,0	0,52	10,92
Miscellaneous			15,06	14,6	2,19	17,25
Tubing and Fitting	1	8,00	8,00	20,0	1,60	9,60
Pressure regulator	1	1,18	1,18	5,0	0,06	1,24
Check Valves	8	0,09	0,72	5,0	0,04	0,76
Propellant Fill and Drain Valves	4	0,05	0,19	5,0	0,01	0,20
Pressurant FdV or Test Port	7	0,05	0,34	5,0	0,02	0,35
Pressurant Filter	1	0,08	0,08	5,0	0,00	0,08
Propellant Filter	2	0,25	0,50	5,0	0,03	0,53
Pyro Valve N/C	5	0,15	0,74	5,0	0,04	0,78
Pyro Valve N/O	4	0,15	0,61	5,0	0,03	0,64
Propellant Latch Valve	0	0,34	0,00	5,0	0,00	0,00
Pressurant Latch Valve	0	0,35	0,00	5,0	0,00	0,00
Pressure Transducer	5	0,22	1,11	5,0	0,06	1,16
Module 2 alignement set	1	0,47	0,47	20,0	0,09	0,56
10N thruster alignement set	1	0,22	0,22	20,0	0,04	0,26
Supports + screw (FdV, PR, PV, SAPT)	1	0,90	0,90	20,0	0,18	1,08

Table 3-20: Option Vega propulsion module mass budget

3.3.4 System Mass Budgets

The mass budget of the Impactor impacting 2002AT4 with Vega launcher (Table 3-21) shows:

- a negative margin (around -7%) in regard to launcher capacity with ESA margins
- the total fuel capacity of the CPM (1542kg) compatible with propellant budget (1440kg)

But the mass target @ launch is not very difficult to raise considering the actual negative margin (-143kg) : a reduction of the Impactor dry mass from 30kg is sufficient (due to its impact on propellant budget). So due to the fact the TTC, power and thermal subsystems have been dimensioned considering the worst-case mission inputs corresponding to 1989ML, the 30kg could be saved only by taking into account the 2002AT4 mission inputs (but the Impactor will not be compatible to both targets in this case). Another reduction axis is the margins ESA (in particular the 20% of system margins applied before fuel calculation).

However the Vega capability is based on its requirements and not on its final results (TBD).

NB: to optimize the calculation, the propellant budget was separated in 2 parts:

- the fuel mass (Table 3-23) necessary for orbit correction during the cruise phase and during impact phase calculated on the basis of the SC dry mass with ESA margins
- the fuel mass (Table 3-22) necessary for Earth escape (the attitude correction is introduced as a reduction of the main engine ISP) calculated on the basis of the total launch mass (considering the cruise and impact fuel as dry mass)

The option 3.2 (2002AT4L target with Vega) Impactor mass budget (602kg with ESA margins) exceeds the Vega capability, but the sizing of the platform with 2002AT4 mission inputs should be enough to raise the right dry mass (30kg be saved). The resulting propellant mass is 1440kg, including 13kg for the cruise and impact phases so it is compliant to CPM max capacity (1542kg).

Element 1	Impactor							
			Target Spacecraft Mass at Launch			1944,00 kg		
			ABOVE MASS TARGET BY:			-143,23 kg		
	Input Mass	Input Margin	Dry mass contributions	Without Margin	Margin %	Margin kg	Total kg	% of Total
EL			Structure	136,23 kg	8,59	11,70	147,93	7,09
EL			Thermal Control	19,40 kg	20,00	3,88	23,28	1,12
EL			Communications	36,66 kg	10,01	3,67	40,33	1,93
EL			Data Handling	19,00 kg	5,00	0,95	19,95	0,96
EL			AOCS	35,04 kg	10,66	3,74	38,78	1,86
EL			Propulsion	126,92 kg	6,51	8,26	135,17	6,48
EL			Power	52,00 kg	20,00	10,40	62,40	2,99
DI	28,00	20,00	Harness	28,00 kg	20,00	5,60	33,60	1,61
			Total Dry(excl.adapter)	453,25			501,44 kg	
			System margin (excl.adapter)		20,00 %		100,29 kg	
			Total Dry with margin (excl.adapter)				601,73 kg	
			Other contributions					
			Wet mass contributions					
DI	1440,50	0,00	Propellant	1440,50 kg	0,00	0,00	1440,50	69,01
			Adapter mass (including sep. mech.), kg	45,00 kg	0,00	0,00	45,00	0,02
			Total wet mass (excl.adapter)				2042,23 kg	
			Launch mass (including adapter)				2087,23 kg	

Table 3-21: Mass budget of the Vega option with 2002AT4 target

PROPELLANT BUDGET	Isp	DV Nom	Margin	DV Max
Escape Manœuvre	320	3650,0 m/s	5%	3 832,5
Trajectory Manœuvre	286	0,0 m/s	100%	0,0
Attitude Control	286	0,0 m/s	100%	0,0
TOTAL DV	**320**	**3 650,0**		**3 832,5**
TOTAL Propellant Mass		**994,9kg**		**1435,3kg**

Table 3-22: Mass fuel budget for Earth escape on Vega with 2002AT4 target

PROPELLANT BUDGET	Isp	DV Nom	Margin	DV Max
Escape Manœuvre	320	0,0 m/s	5%	0,0
Trajectory Manœuvre	286	20,0 m/s	100%	40,0
Attitude Control	286	10,0 m/s	100%	20,0
TOTAL DV	**286**	**30,0**		**60,0**
TOTAL Propellant Mass		**4,9kg**		**13,0kg**

Table 3-23: Mass fuel budget for cruise and impact of 2002AT4 target

3.4 First Impactor study conclusions

Using Dnepr, the Impactor design, based on Exomars enlarged CPM common with Orbiter, complies with both targets 2002AT4 and 1989ML that is the worst-case used to perform the dimensioning. The main point to study deeply is the configuration (SA, MGA and impact thrusters) to be compliant of nominal and back-up trajectories for both asteroids.

Using Vega, due to ESA margins the Impactor, design based on flight proven SpaceBus CPM, complies only with 2002AT4 target (due to final mass required for 1989ML impact considering GNC limited velocity). At this step, the proposed design overpass the launcher mass capacity but several axes of improvement exist (in particular at platform level due to subsystems sized on worst-case input but also considering the configuration optimization).

The best point is the compatibility of the platform box like with all launchers and with several CPM in particular the large SpaceBus family (so further re-use with CPM fitted to the new delta V mission is possible).

4. ANNEX 1: TTC SUBSYSTEM DETAILLED DATA

4.1 Telemetry Tracking & Command

The onboard TT&C subsystem handle all the communications links required by the SRD for the Impactor.
The approach taken into the design of this subsystem is to provide an overall solution for every launcher and asteroid options, using a minimum number of equipments and maximizing their use. A special care has been taken to ensure the whole subsystem to be single point failure tolerant.

4.1.1 Design Drivers

4.1.1.1 Link architecture and operative modes

The overall DQ link architecture is shown in Figure 3-1. Three kind of links have been identified:

- An **X band link**, used for communications with Ground Stations. It includes telecommands, housekeeping telemetry, scientific telemetry, data relay forwarding, ranging operations and support to RSE operations. It is used both on Impactor and Orbiter.
- A **Ka band link**, whose only purpose is to perform high resolution RSE operations. It is provided by the Orbiter only.
- An **UHF link**, based on CCSDS Proximity-1 protocol, for low range communications between Orbiter – Impactor and Orbiter-ASP.

For the Impactor, the list of operations vs frequency band and mission phases are resumed in the following table.

System Modes	Description	TT&C Mode	Notes
Pre Launch	On launch pad (30 min)	Launch	units warming up
Launch	From launch to separation (30 min)		
Sun Acquisition	After separation (120min assumed)		
Leop	Earth escape phase encompass orbits & chemical raise manoeuvres. (Not present if Suyuz launch)	Safe	X band
Correction manoeuvres	Correction manoeuvres, eventually fly-by	Cruise	X band
Cruise	Cruise phase		
Final targeting	Close approach to NEO, navigation, payload & proximity UHF link	Final targeting	X band UHF band
Safe	Safe Mode	Safe	X band

Table 4.1.1-1: Impactor operations by mission phase

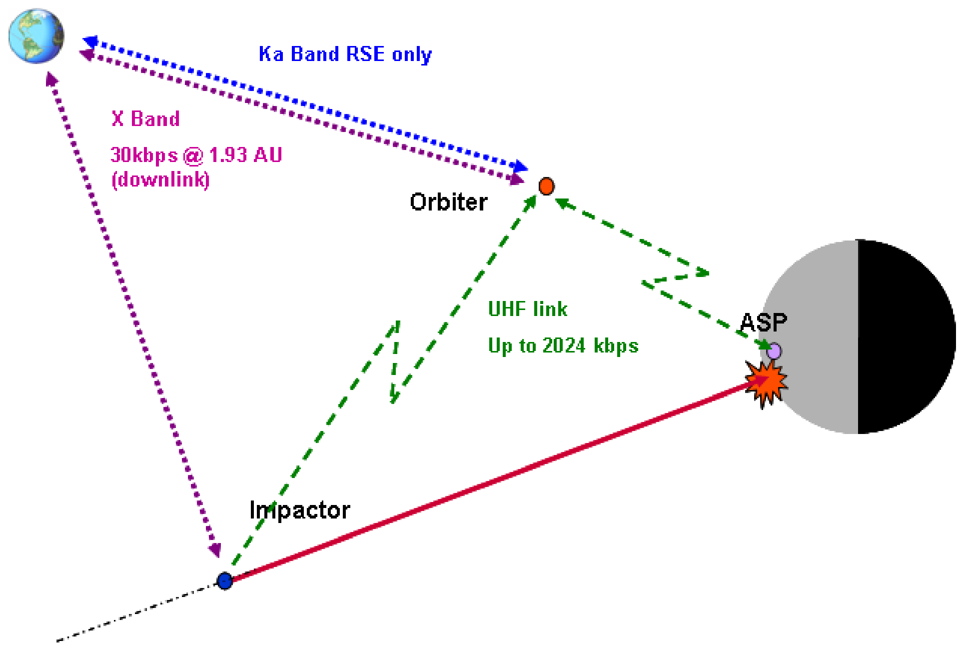


Figure 4-1: Overall DQ communication link overview

Links	Uplink Frequency (MHz)	Downlink frequency (MHz)	Orbiter	Impactor	ASP
X/X	X Band 7145-7190 MHz	X Band 8400-8450 MHz	S/C Operations and TM Ranging (standard and DDOR)	S/C Operations and TM Ranging (standard and DDOR)	-
X/Ka		Ka Band 31800-32300 MHz	Ranging operations supporting RSE (DDOR and WBRS)	-	-
Ka/Ka	Ka Band 34200-34700 MHz	Ka Band (31800-32300 MHz)	RSE	-	-
UHF	TBD UHF Forward (*) Link Band (400-450 MHz)	TBD UHF Return (**) Link Band (390-400 MHz)	Link WITH Impactor/ASP Data Relay	Link With Orbiter Backup TM Before Impact	Link with Orbiter ASP Operations and TM

Table 4.1.1-2: DQ TT&C operations by frequency band

(*) Forward Link = Orbiter – Impactor/ASP
(**) Return Link = Impactor/ASP – Orbiter Link

4.1.1.2 Frequency, Modulation and Encoding Plan

The following table resume the frequency, modulation and encoding plan

	Uplink Frequency (MHz)	Downlink frequency (MHz)	Turnaround ratio	TM Modulation	Encoding	Comments
Orbiter						
X/X	TBD X Band 7145-7190 MHz	TBD X Band 8400-8450 MHz	749/880	PCM/NRZ/BPSK (SQUARE)/PM Or PCM-SPL/PM	Concatenated or Turbo Code ½ - ¼	From Bepicolombo
X/Ka		TBD Ka Band 31800-32300 MHz	739/3344	PCMNRZ/BPSK (SQUARE)/PM Or PCM-SPL/PM		
UHF band	TBD UHF Forward (*) Link Band (400-450 MHz)	TBD UHF Return (**) Link Band (390-400 MHz)	13113/38*39	BPSK/PM	Convolutional k=7, r =1/2	Proximity-1 channel 4
Impactor						
X/X	TBD X Band 7145-7190 MHz	TBD X Band 8400-8450 MHz	749/880	PCM/NRZ/BPSK (SQUARE)/PM Or PCM-SPL/PM	Concatenated or Turbo Code ½ - ¼	
UHF band	TBD UHF Forward (*) Link Band (400-450 MHz)	TBD UHF Return (**) Link Band (390-400 MHz)	13113/38*39	BPSK/PM	Convolutional k=7, r =1/2	Proximity-1 channel 4
ASP						
UHF band	TBD UHF Forward (*) Link Band (400-450 MHz)	TBD UHF Return (**) Link Band (390-400 MHz)	13113/38*39	BPSK/PM	Convolutional k=7	Proximity-1 channel 4

Table 4.1.1-3: Proposed DQ Frequency, Modulation and encoding plan Proposed DQ Ground station assignmentsponder

4.1.1.3 Modulation Schemes

The onboard transponder shall be capable to switch between a set of predetermined data rates, and the whole TT&C subsystem shall be capable to provide a set of minimum data rate at different MPO-Earth distances.

Data rates will be considered supported by the current design only if the following performances requirements and margins are guaranteed:

Object	Value	Remarks
Bit Error Rate (BER)	$< 10^{-5}$ (X Band)	For Uplink TC (X Band)
	$< 10^{-6}$ (UHF Band)	For Both Forward and return link (UHF)
Frame Error Rate (FER)	$< 10^{-5}$ (X Band)	For Downlink TM (X Band)
	$< 10^{-3}$ (UHF Band)	For Both Forward and Return Link (UHF)
Link Budgets margins	> 3 dB	Design parameters only
	> 0 dB	Design – worst case RSS
	> 0 dB	Mean - 3σ

Table 4.1.1-4: End-to-end link performances requirement

4.1.1.4 Encoding Schemes

The coding and decoding of data streams is handled by the CDMU therefore coding requirements does not directly apply to the TT&C subsystem. The impacts on the TT&C subsystem derive from the ratio between the bit rate fb and the symbol rate fs (with reference to Figure 4.1-2 below) related to coding. In fact, the TT&C subsystem shall be able to handle the data stream at the fs rate but the data volume computation and link budget computation (in particular for TM) shall be done on the basis of the effective fb bit rate.

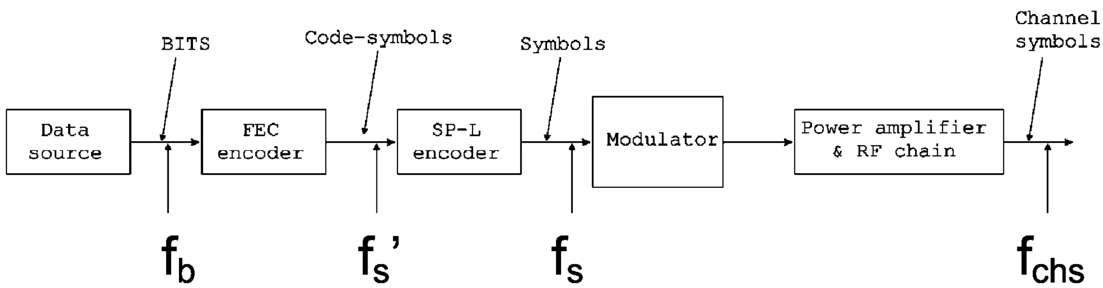


Figure 4.1-2: Data Rates definitions

As far as telecommand bit rates BCH (Bose-Chaudhuri-Hocquenghem) coding is assumed as baseline as per "TC Synchronization and Channel Coding. Blue Book", [SD-33]. In addition, a TC Pseudo Randomization (scrambling) can be used to ensure a minimum bit transition density.

For TM the coding schemes to be considered (in particular for link budget and data volume computation) are the following.

Coding Scheme	Description
Concatenated Encoding	Reed Solomon (255,223), Interleaving depth I=5 + Convolutional Code (rate ½, constraint length k=7)
Turbo Codes ½	Information block length k = 8920 bits (=223 x 5 octets)
Turbo Codes ¼	Information block length k = 8920 bits (=223 x 5 octets)

Table 4.1.1-5: Encoding schemes to be supported by the onboard Transponder

Depending on the selected coding scheme and based on the feasible symbol rates shown in Table 7.11-8, different data rates can be achieved before encoding (fb with reference to Figure 4.1-2 above). These will be the effective TM download data rate to be considered for data volume analysis and link budget computation. The following table provides the ratio between the bit rate fb and the symbol rate fs for the coding schemes above.

Encoding Scheme	Symbol Rate to Bit Rate ratio
Concatenated Encoding	1,1471
Turbo Codes ½	2,0081
Turbo Codes ¼	4,0161

Table 4.1.1-6: Encoding schemes to be supported by the onboard Transponder

4.1.2 Ground Stations

The following ground stations has been identified as reference for the DQ operations, on different mission phases. All the link budgets has been calculated on the basis of the following table.

Station	LEOP	Cruise	Cruise: Critical Phases	Impact	Data Relay Operations
Kourou/Vilspa 15m	Yes	No	No	No	No
New Norcia/ Cebreros 35m	Yes	Yes	Yes	Yes	Yes
NASA DSN 34m	No	Potential back-up for Cebreros/New Norcia			
NASA DSN 70m	No	Potential emergency back-up only			

Table 4.1.2-1: Proposed DQ Ground station assignmentsponder

4.1.3 Orbit design drivers

The orbit design drivers for the TT&C are derived from the mission analysis and provided for the two target asteroids, 2002AT4 and 1989ML and different vectors.

	Max Earth	Earth@impact	Max Sun	Min Sun	Sun@impact
2002AT4					
Dnepr_backup	1,74	1,74	1,39	1,01	1,17
Dnepr_baseline	1,74	1,74	1,35	1,00	1,17
Vega_backup	1,74	1,74	1,39	1,01	1,17
Vega_baseline	1,74	1,74	1,35	1,00	1,17
Worst case	**1,74**	**1,74**	**1,39**	**1,00**	**1,17**
1989ML					
Dnepr	1,11	1,04	1,11	0,80	1,11
Mars_Dnepr	2,33	1,48	1,61	0,56	1,24
Venus_Dnepr	2,16	1,53	1,52	0,69	1,17
Worst case	**2,33**	**1,53**	**1,61**	**0,56**	**1,24**

Table 4.1.3-1: Impactor reference distance wrt target asteroid and vector launcher

4.1.3.1 Final Targetting

The Orbiter-Impactor relative velocity and distance during final targeting phase is shown in the next figure (from proposal). Negative velocity means that the two S/C are approaching.

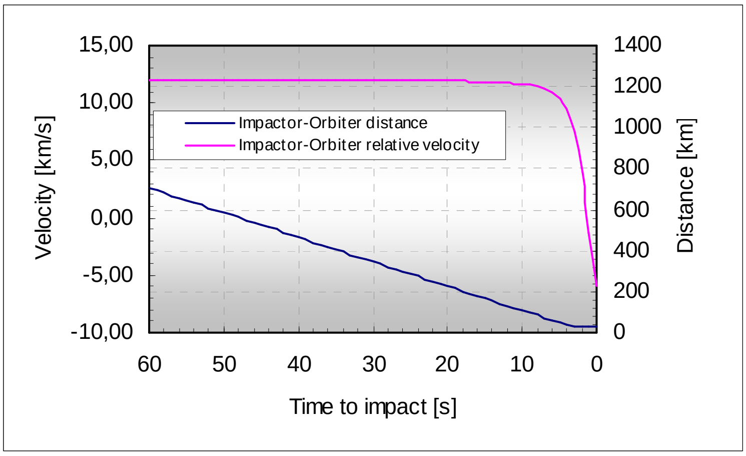


Figure 4.1-3: Impactor-Orbiter distance and velocity during last minute before impact (data from proposal)

4.1.4 Data relay link considerations

The Orbiter acts as a data relayer, with a store-and-forward method. Data coming from the Impactor and/or ASP) is kept in the Orbiter mass memory until the next G/S pass, when data is downloaded to Earth. The link design can therefore be divided into two parts:

- Direct-to-Earth link, which is in charge to handle communications with the Ground Stations. The standard X band link will perform this task.

- An Impactor -to-Orbiter (or ASP-to_orbiter) link, provided in UHF band using CCSDS Proximity-1 protocol. This kind of protocol was already used in several NASA Mars missions and also planned for EXOMARS. It includes a full duplex link to download data from the Impactor and upload/download data to/from ASP.

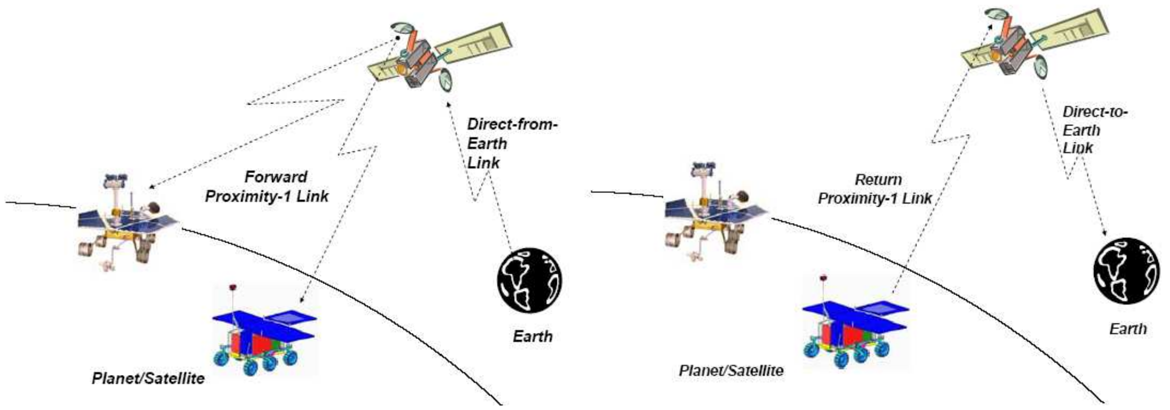


Figure 4.1-4: Typical data relay link using UHF Proximity-1.

The UHF link between ASP/impactor and the orbiter will be full duplex and in accordance with the CCSDS 211-B Recommendation for the Proximity-1 Space Link Protocol.
This kind of protocol was intended for short-range, bidirectional links, generally used to communicate among fixed probes, landers, rovers, orbiting constellations and orbiting relays. These links are characterized by short time delays, moderate (not weak) signals, and short/independent sessions.
Data rates usually vary between 64kbps and 512kbps, even though the protocol can support data rates up to 4Mbits.
For the Impactor link, an adaptative data rate adjustment system is foreseen to maximize the data volume that can be transmitted during the period in which the two spacecrafts are in sight. Therefore, at higher distances the data rate will be initially low, and grows up till impact. The expected time to handle a bit rate change is in the order of seconds (about 10 seconds). Refer to the subsystem budgets section for a detailed analysis.

4.1.5 Ranging design drivers

In the following a summary of the main characteristics of the ranging signals that are proposed as baseline. The main reference is BepiColombo mission.

The **Code Ranging** signals to be supported is in accordance both to ESA Ranging Standard [SD-20] (when the link is performed toward ESA G/S) and to NASA applicable standards (when the link is performed toward NASA DSN).As far as the ESA standard is concerned, a code with a nominal length of N=14 shall be considered. For contingency, any code length from 14 to 20 can be also used. The selected ranging tone frequency has to be chosen to reduce the interference with the telemetry signal.

In addition **Pseudo Noise Code** signals is supported by the TT&C subsystem according to CCSDS standard (the standard is still under discussion). In this case the TT&C subsystem shall implement a regenerative retransmission of the PN Codes, in order to improve the performances.

The **Delta-DOR tones** shall be square waves, modulating in phase the down-link carriers in X band only non-coherent mode. No TM or RSE signal shall be transmitted during DDOR operations. Delta-DOR ranging is also supported by the Orbiter, at higher frequency. On the Impactor it is limited to X band only, therefore a less ranging accurancy is foreseen. According to the BepiColombo reference, the tones are coherent with downlink carrier frequency values according to the following ratios:

- for the X-Band case $f_{X1} = f_{down,X} / 8800$ (~ 1 MHz). The peak modulation indices are TBD

4.1.6 Doppler and Doppler Rate

4.1.6.1 X band link

Doppler and Doppler rate define the frequency shift and frequency shift variation due to the Impactor to ground relative velocity and acceleration. This value affects the transponder capabilities to lock and maintain the X band carrier signal.
The Impactor is characterized by an high relative velocity wrt the earth, with a peak of 12 km/s before impact. The corresponding Doppler shift (one-way) has been estimated and shown in the next table. Doppler shifts can be easily recovered on the Ground stations (i.e. for downlinks), but not on the onboard transponder (i.e. for uplinks), which has a limited PLL bandwidth to reduce the RF signal noise introduced on the transparent channel. This may limit the maximum uplink data rate.

As far as the Doppler rate concerns, the relative velocity is almost constant before the impact. Therefore, Doppler rate contribution is negligible before the impact. A “standard” value coming from Herschel Planck program has been assumed as reference.

Doppler Initial reference X Band		
Doppler shift	+/- 290 kHz	One way shift For a velocity of about 12Km/s
Doppler Rate	+/- 250 Hz/s	From H/P program

Table 4.1.6-1: Impactor X band Maximum Doppler and Doppler rates

4.1.6.2 UHF band link

The contributions for the Impactor-Orbiter link in UHF band are evaluated in this paragraph.
The Doppler shift and Doppler rate figures has been calculated on the basis of the Impactor-Orbiter relative velocity profile shown above.

Results are shown in the next table and into the following figures. Even if the Impactor-Orbiter relative velocity is high (about 12Km/s), the frequency band is much lower than X band, and the resulting Doppler and Doppler rate impact is negligible.

Doppler Initial Reference UHF band		
Doppler shift	+/- 15 Hz	One way
Doppler Rate	+/- 10 Hz/s	

Table 4.1.6-2: Impactor UHF band Maximum Doppler and Doppler rates

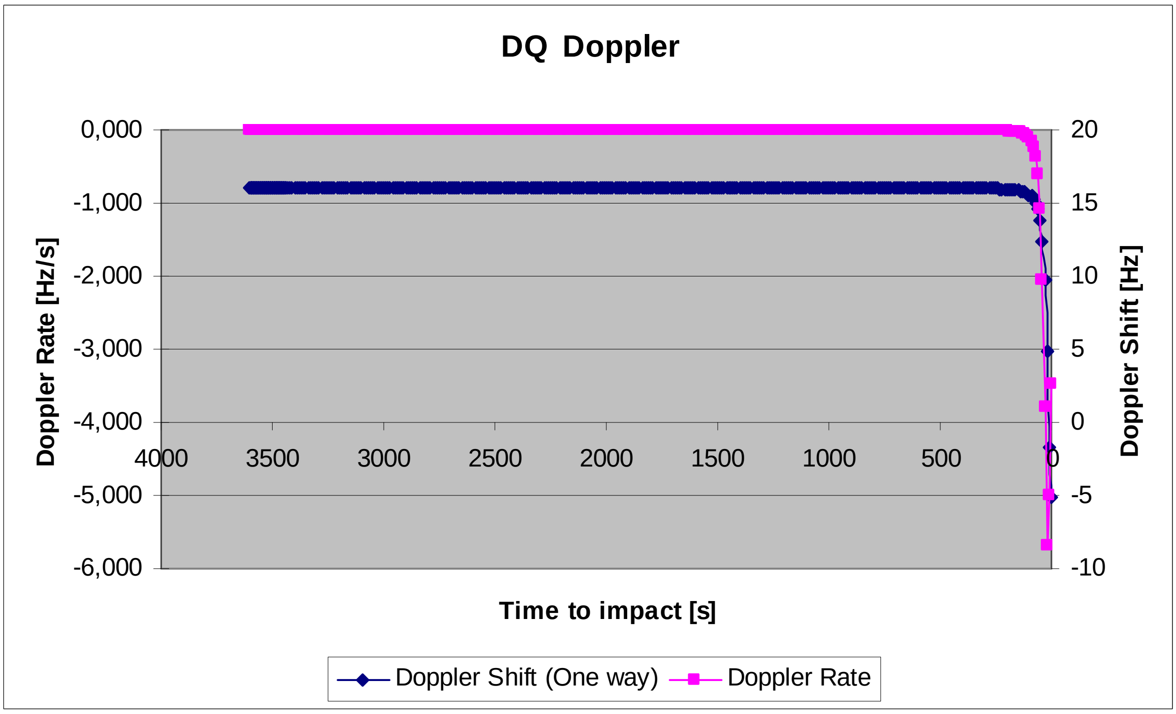


Figure 4.1-5: UHF doppler and Doppler rate (final targeting phase)

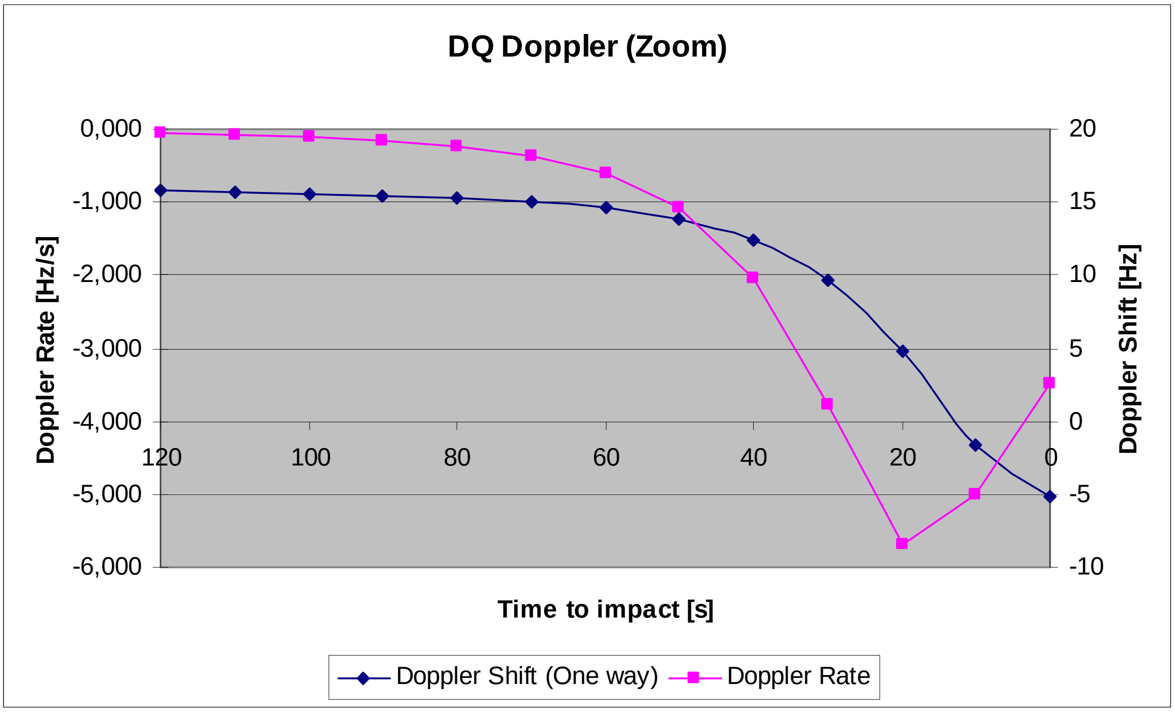


Figure 4.1-6: UHF doppler and Doppler rate (final targeting phase, last two minutes before impact)

4.1.7 Data Volume

The rationale of the required data volume and corresponding total data rate has been developed into the data handling and payload section. A resume is reported here for reference, for each phase:

System Modes	Description	TT&C Mode	Required Data rate
Pre Launch	On launch pad (30 min)	Launch	(No TX)
Launch	From launch to separation (30 min)		
Sun Acquisition	After separation (120min assumed)	Safe	5 kbps (S/C Housekeeping)
Leop	Earth escape phase encompass orbits & chemical raise manoeuvres.		
Correction manoeuvres	Correction manoeuvres, eventually fly-by	Cruise	5 kbps (S/C Housekeeping)
Cruise	Cruise phase		
Final Targeting	Close approach to NEO, navigation, payload & proximity UHF link	Impact UHF Link with Orbiter	30kbps (TBC) Variable From 4kbps to 1Mbps
Safe	Safe Mode	Safe	As high as possible

The data rate figures above are based on the assumption of 8h of daily contact.

4.1.8 *Impactor TT&C Subsystem Architecture*

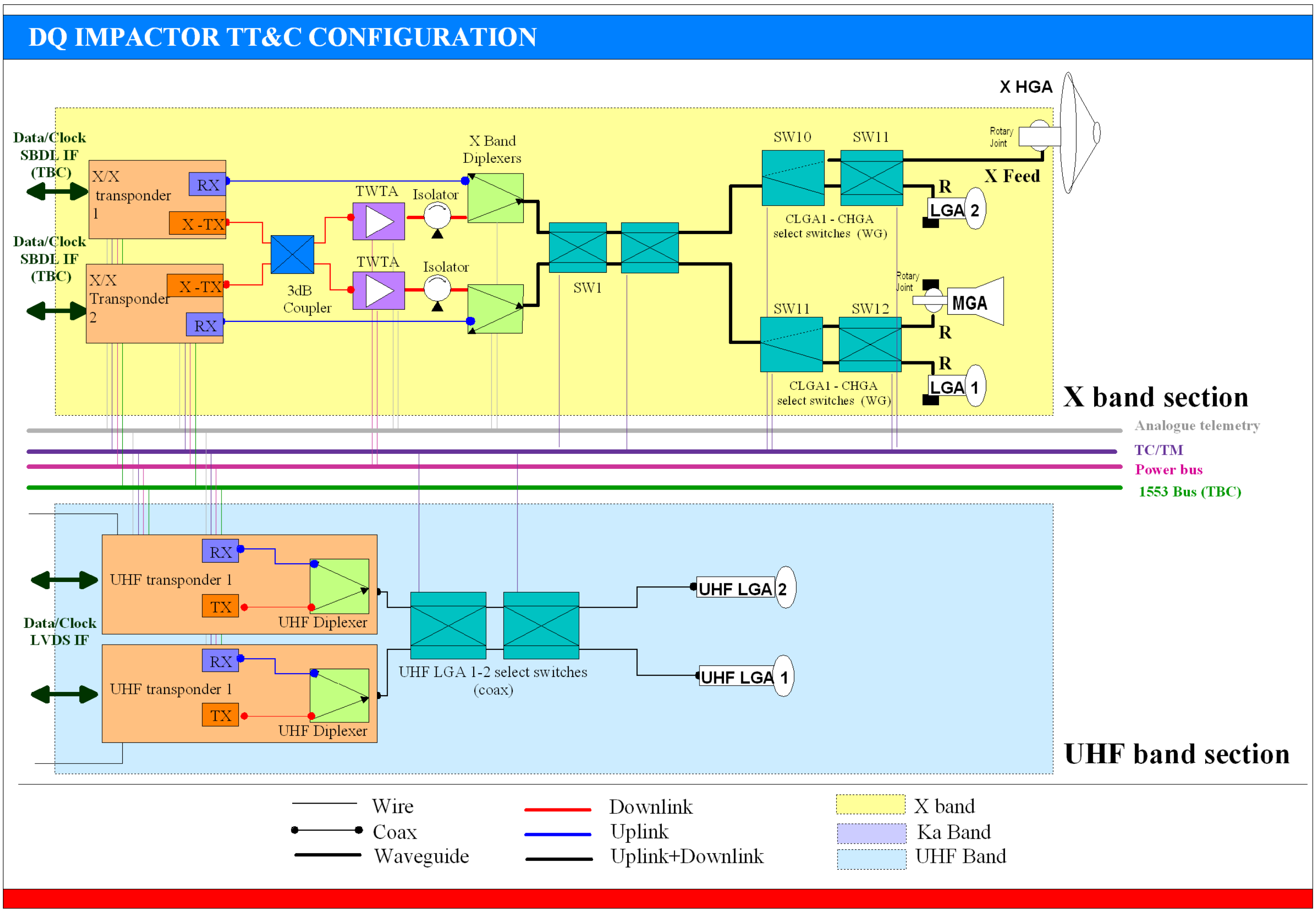


Figure 4.1-7: DQ Impactor TT&C Architecture

The TTC function thus comprises:

- **One X Band High Gain Antenna (HGA)**, used to perform main S/C communications with the G/S. The antenna is provided with a 2-DOF pointing mechanism.

- **Two Telemetry Encoders/Decoders** allocated in the data management unit operated in hot redundancy for the receiving part and cold redundancy for the Tx one. In particular the TM encoders are in charge of the TM stream generation and the required bit encoding.

- At the same time the CDMU includes two pairs of **TC Decoders** that receive the digital TC signal from XPNDs TC Demodulators, decode it and provide the Telecommands to the data management unit.

- **Two X/X Band Transponders (XPND)** operated in hot redundancy for the receiving part and cold redundancy for the Tx one. The Transmitting part accepts the Transmitted Symbol Rate from the TM Encoders and generates all the required Modulations.

- An **X Band section**, including:
    - **Two 65 W X band TWTAs** that provide the necessary RF amplification to the signal coming from the Transponders.
    - **One X-band Medium Gain Antenna (MGA)** with a 2-DOF pointing mechanism, used as backup of HGA during safe mode. It allows communications at a lower bit rate. The option of a 1-DOF pointing mechanism (as in the Impactor TT&C) can be evaluated in the next phase.
    - **Two Omnidirectional X-Band Low Gain Antennas (LGAs)**, used when spacecraft attitude is not known (during LEOP or in case of loss of attitude).

- An **UHF section**, which comprises:
    - Two dedicated UHF Transceivers, which implement the CCSDS Proximity-1 protocol used for communications between Orbiter and Impactor and between Orbiter and ASP (the latter is a DQ+ option only). This includes encoding/deconding, modulation/demodulation and signal amplification and filtering.
    - Two UHF antennas, one used for Orbiter-Impactor link and one for Orbiter-ASP link. Depending on the Orbiter attitude, a single antenna may be sufficient. This will be confirmed during phase A.

The TTC architecture also comprises the suitable set of diplexers, hybrids and switches integrated in the so-called RFDN, one for each band section, that guarantee a connection to each antenna with a one point failure tolerance.

4.1.8.1 X band section functional description

X band signals are routed in different paths to the HGA; each Transponder-antenna path is one point failure tolerant

The core of the TT&C subsystem is represented by the DST. In the proposed design, two DST are foreseen for redundancy reasons. The RX section of both are always powered, while the TX sections (both X and Ka ones) are in cold redundancy i.e. only one of the two DST is powered at a time.
The DST will be in charge of data reception and transmission from/to ground and of ranging signal handing. The foreseen uplink and downlink operative modes are shown in Table 4.1.8-1 below. In addition, in case of presence of the uplink carrier the DST can be set in coherent mode allowing G/S to perform Doppler measurements. In case of absence of the uplink signal (or by telecommand) the XPND is able to operate in non-coherent mode locking the downlink carrier on internal reference or external USO.

Uplink (X band only)	Downlink (X and Ka Band)
Carrier Only	Carrier Only
Carrier + Telecommand	Carrier + Telemetry
Carrier + Telecommand + STD Ranging	Carrier + Telemetry + STD Ranging
Carrier + Telecommand +PN Ranging	Carrier + Telemetry + PN Ranging
Carrier + STD Ranging	Carrier + STD Ranging
Carrier + PN ranging	Carrier + PN Ranging
	Carrier + DDOR Ranging
	Carrier + Beacon Tone

Table 4.1.8-1: TT&C Subsystem Operative Modes

The outputs of the two DSTs are coupled together with a 3dB hybrid, which provides the required cross-strapping between DSTs and TWTAs. This approach is more reliable than using switches because it is passive only (no moving mechanical parts). The RF losses are not relevant because the RF signal will be amplified by the TWTAs. With the proposed configuration each X band output of either DST1 or DST2 is always connected to each X-TWTA, and each Ka band output of either DST1 or DST2 is always connected to the Ka-TWTA. In this way it is possible to power-on whichever TWTA without any constraints on the DST being in used.
The correct path from the powered TWTA to the antenna to be used will be selectable thanks to the RFDN whose main functions are to provide the required switching capabilities to support this function and to provide up-link and down-link signals separations to be routed on different paths. It has to be highlighted that the cases in which both TWTAs are powered at the same time will be avoided by software implementation however the RFDN will be able to sustain the increased power without damage.

The RFDN switching capabilities are achieved by means of 4 ports switches (4PS) realized in waveguide technology in X band. Transfer switches (in the diagram above) are in practice 4 port switches with one port left open. Each switch is redunded by unit duplication, to avoid any loss of functionality if one switch gets stuck in a position. Couplers are not used as the increased RF losses directly affect the link performances.

The RF uplink and downlink signal separation capabilities of the RFDN are achieved by the means of diplexers.
With the proposed RFDN design, the X-Band TM signal amplified by the TWTA passes through an isolator to protect the TWTA from mismatches, then through the Diplexer and two 4-ports switches. Thanks to these 4PS it is possible to feed the LGA1/MGA couple or the LGA2/HGA couple. The selection between LGA or MGA/HGA is then performed through two further 4-ports switches.

The X-Band TC signal is routed directly from the diplexer to DST. This approach does not foresee cross-strapping between diplexers and DSTs. However, in case of failure on one of the two chains this can be recovered in any case with a DST switchover. This approach has been selected in order to reduce the losses on the receiving path, where the weak signals are more affected by noise.

4.1.8.2 UHF section functional description

The UHF link, based on CCSDS 211-B (proximity-1) protocol is used to establish a communication with Impactor during the last 2 minutes before the impact. The same link can also be used to communicate with the ASP on the asteroid surface.

For short range links, uplink and downlinks terms are no more applicable. The Orbiter-to device link is usually indicated as forward link, while the device-to-Orbiter link is indicated as return link. However, it depends on the device who starts the communication.

A session is usually initiated by sending a string of “hail” data packets while looking for a response from the specific lander identified in the hail packet. This standard operating procedure can be reversed, that is, lander-initiated relay sessions are possible. The hail includes information describing the session operating mode for both the forward and return link directions. This includes operating frequency, data rate, and channel coding mode, to name a few important things.
Once a session is initiated, Prox-1 transfer frames are sent on both the forward link and on the return link using the Prox-1 protocol link management in either reliable (retransmission) or expedited (no retransmission) mode.

In Prox-1 reliable mode, data frames with bit errors are automatically detected and retransmitted via a standard Go-Back-N protocol scheme.
In Prox-1 expedited mode, data frames with bit errors are discarded on the receive end. All that remains is a record of the data frame number missing from the frame sequence accounting.

The radio stack of the protocol also foresees a number of commandable timer settings that allow it to flywheel over short link drop outs or that force automatic link reacquisition after longer signal drop out periods. This is usually used to maximize data return in a relay link environment with variable link performance.
Prox-1 sessions are terminated by timed sequenced command or by the time out of a dropped signal count down timer.

The protocol also include other services to improve the overall flexibility:

- **Timing service**, for time-stamping and correlation calculations.
- **Radiometric service**, like phase and signal power measurement that can be used to obtain information about Doppler.
- **Raw data service**, in which the Prox-1 packet management is off. This allows to use a dedicated transmission protocol, maintaining the Prox-1 radio and physical layer.

All these operations are handled by the onboard transceivers, which are provided with an internal diplexer and power amplifier. A cold redundancy is used, due to fact that UHF links are time-short. Two switches (redounded) provide the connection with the two UHF antennas. The first one is used to communicate with the Impactor, the other one to communicate with the ASP.

4.1.8.3 Ranging Operations

The TT&C Ranging function is implemented by the DST. Its design will allow complying with the MSRD requirements and in particular it will implement:

- **Standard Ranging codes** retransmission on a transparent channel (codes that will be supported will both the ESA MTPS and NASA ones);
- **Pseudo-Noise codes** retransmission, with regenerative technique to improve the ranging performances;
- **Delta-DOR** signal generation.

All the modes listed above are supported on X band only.

As indicated above, the standard ranging signal transponding is performed in the classical way, i.e. through a transparent video channel having a bandwidth in the order of 5 MHz. This allows the usage of any kind of code with a band occupation compatible with the DST video channel, in particular the required ESA MPTS codes and NASA codes.

As far as Pseudo-Noise Ranging is concerned, this is a signal generated using a logical combination of a ranging clock and several component PN codes. Received by the spacecraft, the ranging signal is demodulated by the spacecraft transponder, and the ranging code is acquired. The spacecraft then regenerates a local representation of the ranging code coherently with the received one, and phase modulates the downlink carrier with this newly generated version, not affected by the uplink noise. Back at the ground station, the station receiver demodulates the downlink and correlates the received ranging signal with a local model of the range clock and component PN codes to determine the roundtrip time.

The Delta-DOR mode can be set on the DST by dedicated TC. In this mode no TM is transmitted, and the DST can operate also in non-coherent mode. The Delta-DOR subcarriers are generated by the DST itself coherently with downlink carrier frequency, according to the requirements. These signals will phase modulate the carrier with presettable modulation indices. All these operations will be telecommanded through the CDMU bus.

4.1.8.4 Tone Beacon Mode

The DST currently supports also a Tone Beacon Mode, which can be useful to monitor S/C status during cruise without taking too much resources from the Ground stations (i.e. with a improved cost impact). A brief description is provided in this paragraph.

The design of this TT&C system operative mode is similar to the one developed for the NASA Deep Space One mission.

In beacon monitor operations, an on-board data summarization system routinely determines the overall spacecraft health. A state keeps the information about spacecraft health conditions. Then, depending on the health state, it selects one of the available radio tones to send to Earth to indicate how urgently it needs contact with the ground station.
Tones are generated by phase-modulating the RF carrier by a square-wave subcarrier with a TBD modulation index. The selected approach (different from the NASA one) is not to completely suppress the carrier. In this way it will be easier to recover the signal on ground, and it will be also possible to perform Doppler measurements if necessary. The sub-carrier can be generated by the DST upon command from the CDMU, or can be generated by the CDMU itself, as a stream of 0 and 1 bits at the proper rate.

The resulting downlink spectrum will consist of tones at odd multiples of the sub-carrier frequency, above and below the carrier signal (see
Figure **4.1-8**). It has to be highlighted that with this modulation approach, when Tone Beacon Mode is selected, no telemetry can be sent to ground. However, this is the purpose of this mode, i.e. to avoid to continuously send housekeeping TM to ground by transmitting simply a status signal.

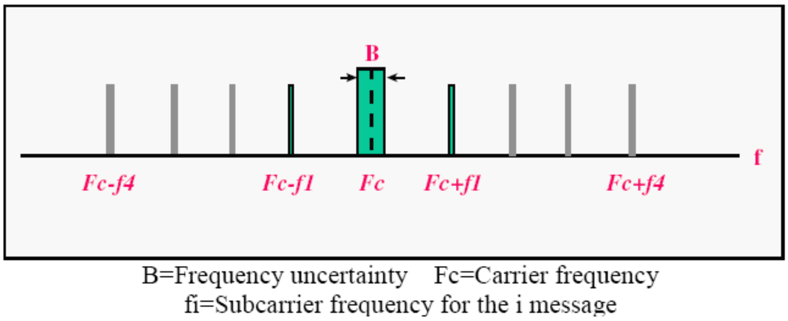


Figure 4.1-8: Tone Signal structure

An example with 4 tones is shown in the next figure:

	Subcarrier Freq.	S/C state	Description	Comments
Tone 1	**F_c ± 20 KHz**	**Nominal ("Green")**	Spacecraft is nominal. All functions are performing as expected. No need to downlink engineering telemetry.	
Tone 2	**F_c ± 25 KHz**	**Alert ("Yellow")**	An interesting and non-urgent event has occurred on the spacecraft. Establish communication with the ground when convenient.	e.g. device reset to clear error caused by Single Event Upset (SEU), other transient events.
Tone 3	**F_c ± 30 KHz**	**Medium Alert ("Orange")**	Communication with the ground needs to be achieved within a certain time or the spacecraft state could deteriorate and/or critical data could be lost.	e.g. memory near full, non-critical hardware failure
Tone 4	**F_c ± 35 KHz**	**Serious Alert ("Red")**	Spacecraft emergency. A critical component of the spacecraft has failed. The spacecraft cannot autonomously recover and ground intervention is required immediately.	e.g. CDMU failure, Thrusters failure.

Table 4.1.8-2: Tone Beacon description

The beacon signal detection at ground is not challenging to be implemented. A possible functional block diagram is shown in Figure 4.1-9. The signal coming from the S/C passes through four tone detectors, one for each message. To ensure proper signal detection, the bandwidth of each tone detector must be sufficiently large to accommodate the frequency uncertainty and frequency drift of the downlink frequency.
An FFT (Fast Fourier Transform) technique can be employed to compute the energy of all spectral pairs having spacing corresponding to the four Beacon signals.
The maximum of the outputs of the four tone detectors is then selected and compared against a pre-determined threshold to determine which message has been received.

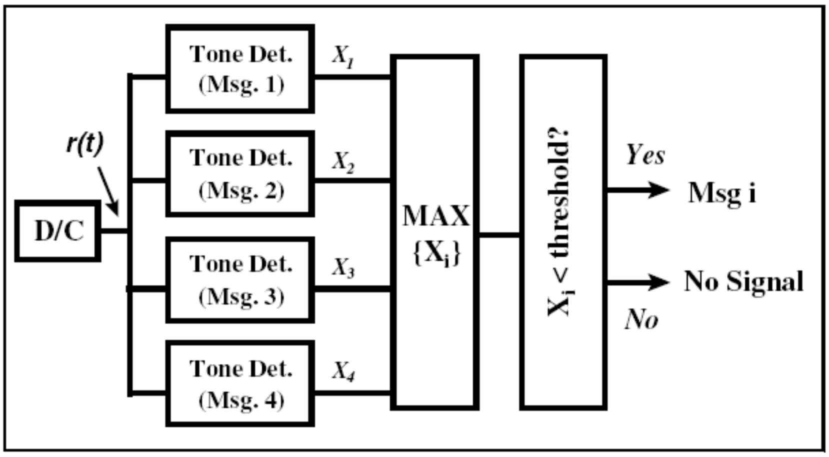


Figure 4.1-9: Tone detection functional diagram

These tones can be easily detected with low cost receivers and small ground antennas, so monitoring a spacecraft that uses this technology is not very resource-demanding.

4.1.8.5 Antenna configuration

The antenna selection table with different mission phases is provided below, in accordance to SRD requirements.

MISSION PHASE		NOMINAL ANTENNA	OPERATIONS
LAUNCH		LGA 1-2	Omnidirectional Coverage
CRUISE		HGA/MGA	MGA or HGA are used for payload calibration and TC/TM link (depending on the distance)
FINAL TARGETING	X BAND LINK	HGA	Primary TC/TM Link
	LINK WITH ORBITER	UHF LGA1-2	Link with Orbiter
CONTINGENCY	HGA FAILURE	MGA	Secondary TC/TM
	SAFE MODE	MGA	Carrier Recovery
	ATTITUDE LOST/LEOP	LGA1-2	Omnidirectional Coverage

Table 4.1.8-3: Antenna selection table

4.1.9 Subsystem Budgets

4.1.9.1 Mass Budget

ACRONYM	TT&C UNIT NAME	Nominal Mass [kg]	Uncertainty					Maximum mass [kg]
			Development Stat.				Mar	
			New	Der	Mod	Exis		
DST1 X/X	**Transponder X/X**	**3,00**			**1**		**10%**	**3,30**
DST2 X/X	**Transponder X/X**	**3,00**			**1**		**10%**	**3,30**
X - TWTA1	**TWT Amplifier 65W - X Band**	**1,50**				**1**	**5%**	**1,58**
X - EPC1	**Electric Power Conditioner for TWTA1**	**0,80**				**1**	**5%**	**0,84**
X - TWTA2	**TWT Amplifier 65W - X Band**	**1,50**				**1**	**5%**	**1,58**
X - EPC2	**Electric Power Conditioner for TWTA2**	**0,80**				**1**	**5%**	**0,84**
RFDN - X Band	**Radio Frequency Distribution Network**	**3,06**						**3,21**
	X Band - 4-Port Switch 4 0,32	1,28				1	5%	1,34
	X Band - Transfer Switch 2 0,32	0,64				1	5%	0,67
	Isolators 2 0,2	0,40				1	5%	0,42
	X Band - Diplexer 2 0,35	0,70				1	5%	0,74
	X Band - 3dB Hybrid 1 0,035	0,04				1	5%	0,04
External RFDN - X Band	**Radio Frequency Distribution Network**	**3,00**						**3,60**
	2m Waveguide to HGA I/F (250g/m)	0,50	1				20%	0,60
	2m Waveguide to MGA (250g/m)	0,50	1				20%	0,60
	3m Waveguide to LGA1 (250g/m)	0,75	1				20%	0,90
	3m Waveguide to LGA2 (250g/m)	0,75	1				20%	0,90
	2m Waveguide to HGA I/F (200g/m)	0,50	1				20%	0,60
LGA1 - X band	**Low Gain antenna**	**0,30**				**1**	**5%**	**0,32**
LGA2 - X band	**Low Gain antenna**	**0,30**				**1**	**5%**	**0,32**
MGA - X band	**Medium Gain Antenna**	**3,10**						**3,72**
	MGA X band	0,60	1				20%	0,72
	MGA pointing mechanism -1 dof	2,00	1				20%	2,40
	MGA support boom	0,50	1				20%	0,60
HGA - X Band	**High Gain Antenna (1m)**	**7,40**						**8,14**
	HGA Antenna Reflector Assembly	4,20			1		10%	4,62
	HGA RF parts	2,20			1		10%	2,42
	HGA S/C I/F parts	1,00			1		10%	1,10
TOTAL X-Ka Band TT&C SUBSYSTEM		**27,76**						**30,73**
UHF XPND1	**Transponder UHF Band1**	**3,00**				**1**	**5%**	**3,15**
UHF XPND2	**Transponder UHF Band2**	**3,00**				**1**	**5%**	**3,15**
LGA1 - UHF	**Low Gain antenna**	**1,00**			**1**		**10%**	**1,10**
LGA2 - UHF	**Low Gain antenna**	**1,00**			**1**		**10%**	**1,10**
RFDN - UHF	**Radio Frequency Distribution Network**	**0,90**						**0,95**
	UHF Band - 4-Port Switch 2 0,2	0,40				1	5%	0,42
	Harness 1 0,5	0,50				1	5%	0,53
TOTAL UHF TT&C SUBSYSTEM		**8,90**						**9,45**
GRAND TOTAL TT&C		**36,66**						**40,17**

4.1.9.2 Power Budget

DQ	TT&C Power Consumtion and Dissipation - Impactor											
Unit	LAUNCH			Cruise			Safe Mode			Final Targetting		
		CONSUMPTION	DISSIPATION		CONSUMPTION	DISSIPATION		CONSUMPTION	DISSIPATION		CONSUMPTION	DISSIPATION
XPND1 RX	ON	14	14	ON	14	14	ON	14	14	ON	14	14
XPND1 TX -X	OFF			TX ON	6	3	TX ON	6	3	TX ON	6	3
Total XPND1		14	14		20	17		20	17		20	17
XPND2 RX	ON	14	14	ON	14	14	ON	14	14	ON	14	14
XPND2 TX	OFF			OFF			OFF			OFF		
Total XPND1		14	14		14	14		14	14		14	14
EPC1	Pre Heat	8,8	5,5	Saturation	128	11	Saturation	128	11	Saturation	128	11
X - TWT1	Pre Heat		3,3	TWT ON		52	TWT ON		52	TWT ON		52
Total TWTA1		8,8	8,8		128	63		128	63		128	63
EPC2	OFF	0	0	OFF	0	0	OFF	0	0	OFF	0	0
X - TWT2	OFF	0	0	OFF	0	0	OFF	0	0	OFF	0	0
Total TWTA2		0	0		0	0		0	0		0	0
RFDN	N/A	0	0	N/A	0	12	N/A	0	12	N/A	0	12
Ext. WGs	N/A	0	0	N/A	0	5	N/A	0	5	N/A	0	5
Total TWTA2		0	0		0	17		0	17		0	17
XPND1 UHF RX	OFF			OFF			OFF			OFF		
XPND1 UHF TX	OFF			OFF			OFF			OFF		
Total XPND UHF		0	0		0	0		0	0		0	0
XPND2 UHF RX	OFF			OFF			OFF			ON	18,5	18,5
XPND2 UHF TX	OFF			OFF			OFF			ON	42	34
Total XPND UHF		0	0		0	0		0	0		60,5	52,5
TOTAL		36,8	36,8		162	111		162	111		222,5	163,5

4.1.9.3 RF Loss Budget

The following tables provide the expected losses for the RFDN. Some of the parameters considered are worse with respect to the data provided above as given by the foreseen supplier. This approach has been considered in order to consider some margin for possible degradation of the performances in the integrated system.

UPLINK DATA – X Band

HGA to XPND

	Path		Loss per item NOM	ADV	FAV		Subtotal NOM	ADV	FAV	
Coaxial Cable	1,00	m	0,60	0,66	0,54	dB/m	0,60	0,66	0,54	dB
Switch	4,00	units	0,05	0,06	0,05	dB	0,20	0,22	0,18	dB
Int. Waveguide	3,00	m	0,08	0,09	0,07	dB/m	0,24	0,26	0,22	dB
Rotary Joint	0,00	units	0,40	0,44	0,36	dB	0,00	0,00	0,00	dB
Ext. Waveguide	3,00	m	0,08	0,09	0,07	dB/m	0,24	0,26	0,22	dB
Total RFDN							1,28	1,41	1,15	dB
Total Ext. Waveguide							0,24	0,26	0,22	dB
Diplexer							0,30	0,33	0,27	dB
TOTAL							**1,82**	**2,00**	**1,64**	**dB**

DOWNLINK DATA – X Band

TWTA to HGA

	Path		Loss per item NOM	ADV	FAV		Subtotal NOM	ADV	FAV	
Isolator	1,00	units	0,20	0,22	0,18	dB	0,20	0,22	0,18	dB
Switch	4,00	units	0,05	0,06	0,05	dB	0,20	0,22	0,18	dB
Int. Waveguide	3,00	m	0,08	0,09	0,07	dB/m	0,24	0,26	0,22	dB
Rotary Joint	0,00	units	0,40	0,44	0,36	dB	0,00	0,00	0,00	dB
Ext. Waveguide	3,00	m	0,08	0,09	0,07	dB/m	0,24	0,26	0,22	dB
Total RFDN							0,88	0,97	0,79	dB
Total Ext. Waveguide							0,24	0,26	0,22	dB
Diplexer							0,30	0,33	0,27	dB
TOTAL							**1,42**	**1,56**	**1,28**	**dB**

Table 4.1.9-1: X-Band RFDN Parameters.

4.1.9.4 Link Budgets

4.1.9.4.1 X band link

On the basis of system requirements, cdf study and the design drivers the following assumptions have been considered for the link budgets calculations:

- Orbiter max distance from earth is 2,5 AU, while 1.7 AU is considered as worst-case reference distance for the impact. Minimum G/S elevation is 10°.
- The G/S of Cebreros has been assumed as nominal for the cruise and radio-science phases. The support of 70m NASA DSN Ground stations can be requested for the impact phase to increase the achievable data rate.
- Average data rate required for the radio science phase: 30kbps @ 1.7 AU (TBC). HK data rate: 5kbps.
- TM Encoding Scheme: Turbo code ¼ or standard Concatenated (RS+convolutional) code.
- Spacecraft losses and antenna gains are derived from BepiColombo (see Antenna and RFDN section). In particular:
    - HGA peak gain: 33dBi uplink , 36 dBi downlink
    - MGA peak gain: 23dBi downlink, 21 dBi uplink
    - LGA minimum gain: -3 dBi (hemispherical coverage)
    - RFDN RF losses: 2 dBi uplink, 1,5 dBi downllink
- Due to the very high distance from Earth at opposition, two 65W X Band TWTAs from Mars Express have been chosen as main RF signal amplificators.

Based on the link budgets results, the current configuration provides a data rate 30kbps @ 1.80 AU(impact worst case distance). With the support of a DSN 70m G/S, the data rate can be increased up to about 100kbps.

In case of loss of functionality of the HGA, the MGA can guarantee communications at lower data rates. However a G/S with a low receiver loop bandwidth and high performances (like a 70m DSN G/S) is needed.

During LEOP, only LGAs are available for transmission. Link budgets at the reference distance of 0,01AU are reported.

The link budget resume is shown in the next table.

UPDATED: 25/09/06 18.31

DQ IMPACTOR X Band LINK BUDGET RESUME TABLE

DQ UPLINK TABLE (X Band)			LEOP		Cruise		Impact		Safe Mode
			TC + RNG	*TC + RNG*	*TC + RNG*	*TC + RNG*	*TC+RNG*	*TC+RNG*	*TC+RNG*
			27-set-06	*28-set-06*	*29-set-06*	*29-set-06*	*29-set-06*	*29-set-06*	*27-set-06*
			LGA 1-2	***CLGA 1-2***	***HGA 1m***	***HGA 1m***	***HGA 1m***	***HGA 1m***	***MGA***
			NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM
		Ground Station:	Kourou	Perth 35m	Cebreros 35m	DSN70m	New Norcia	DSN70m	Cebreros
distance	1000 km		**350**	**350**	**380000**	**380000,00**	**270000**	**270000**	**560000,00**
	AU		0,002	0,002	2,540	2,540	1,805	1,805	3,743
TC Bit Rate	**kbps**		**2,000**	**2,000**	**2,000**	**2,000**	**2,000**	**2,000**	**0,200**
TC Mod Index	rad/pk		1,0	1,0	1,0	1,0	1,0	1,0	1,0
RNG Mod Index	rad/pk		0,6	0,6	0,7	0,7	0,7	0,6	0,7
Antenna Gain	dBi		-3,00	-3,00	34,20	34,20	34,20	34,20	21,00
RFDN & WG Losses	dB		1,52	1,52	1,52	1,52	1,52	1,52	1,52
Carr. Power Margin (dB)	**Nom.**	**ESA Margin = 3dB**	**25,52**	**51,56**	**13,13**	**13,13**	**16,09**	**21,95**	**13,56**
	Mean - 3 Sigma	ESA Margin = 0dB	24,32	50,57	12,14	12,14	15,10	21,06	12,57
	Marg - WC-RSS	ESA Margin = 0dB	24,51	50,83	12,35	12,35	15,32	21,28	12,79
Carr. Recov. Margin (dB)	**Nom.**	**ESA Margin = 3dB**	**29,28**	**55,32**	**33,59**	**33,59**	**36,56**	**42,71**	**17,02**
	Mean - 3 Sigma	ESA Margin = 0dB	27,60	53,89	32,02	32,02	34,99	41,30	15,45
	Marg - WC-RSS	ESA Margin = 0dB	27,92	54,16	32,39	32,39	35,36	41,59	15,82
TC Recov. Margin (dB)	**Nom.**	**ESA Margin = 3dB**	**6,88**	**32,93**	**11,19**	**11,19**	**14,16**	**20,31**	**4,62**
	Mean - 3 Sigma	ESA Margin = 0dB	5,53	31,79	9,99	9,99	12,96	19,25	3,42
	Marg - WC-RSS	ESA Margin = 0dB	5,76	32,05	10,26	10,26	13,23	19,49	3,69

DQ DOWNLINK TABLE (X Band)			LEOP		Cruise		Impact		Safe Mode
			TM + RNG	*TM + RNG*	*TM + RNG*	*TM + RNG*	*TM+RNG*	*TM+RNG*	*TM*
			27-set-06	*28-set-06*	*29-set-06*	*29-set-06*	*29-set-06*	*29-set-06*	*27-set-06*
			LGA 1-2	***CLGA 1-2***	***HGA 1m***	***HGA 1m***	***HGA 1m***	***HGA 1m***	***MGA***
			NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM	NRZ-L/BPSK/PM
		Ground Station:	Kourou	Perth 35m	DSN 70m	DSN70m	New Norcia	DSN70m	DSN70m
distance	1000 km		**350,00**	**350,00**	**380000,00**	**380000,00**	**270000,00**	**270000,00**	**560000,00**
	AU		0,002	0,002	2,540	2,540	1,805	1,805	3,743
TM Bit Rate	**kbps**		**50,00**	**50,00**	**8,00**	**15,00**	**30,000**	**100,00**	**0,200**
Encoding			**Concatenated**	**Concatenated**	**Concatenated**	**Turbo 1/4**	**Turbo 1/4**	**Turbo 1/4**	**Concatenated**
TM Mod Index	rad/pk		1,20	1,20	1,20	1,20	1,20	1,20	1,20
RNG Mod Index	rad/pk		0,50	0,50	0,50	0,50	0,50	0,50	0,00
Antenna Gain	dBi		-3,00	-3,00	35,60	35,60	35,60	35,60	23,00
RFDN & WG Losses	dB		1,54	1,54	1,54	1,54	1,54	1,54	1,54
Carr. Recov. Margin (dB)	**Nom.**	**ESA Margin = 3dB**	**35,76**	**47,66**	**11,40**	**11,40**	**14,37**	**42,01**	**11,20**
	Mean - 3 Sigma	ESA Margin = 0dB	31,48	43,25	7,19	7,19	10,12	37,49	7,19
	WC-RSS	ESA Margin = 0dB	32,12	43,84	7,82	7,82	10,75	38,16	7,78
TM Recov. Margin (dB)	**Nom.**	**ESA Margin = 3dB**	**5,65**	**17,55**	**3,97**	**3,64**	**3,60**	**4,42**	**4,57**
	Mean - 3 Sigma	ESA Margin = 0dB	4,82	16,96	3,37	3,04	3,01	3,51	3,85
	WC-RSS	ESA Margin = 0dB	4,64	16,99	3,41	3,08	3,05	3,40	3,94
RNG Recov. Margin (dB)	**Nom.**	**ESA Margin = 3dB**	**26,67**	**53,75**	**14,08**	**14,08**	**19,70**	**37,57**	**No RG**
	Mean - 3 Sigma	ESA Margin = 0dB	18,11	43,44	5,64	5,64	11,14	28,14	No RG
	WC-RSS	ESA Margin = 0dB	19,57	45,06	7,04	7,04	12,55	29,75	No RG

4.1.9.4.2 UHF link

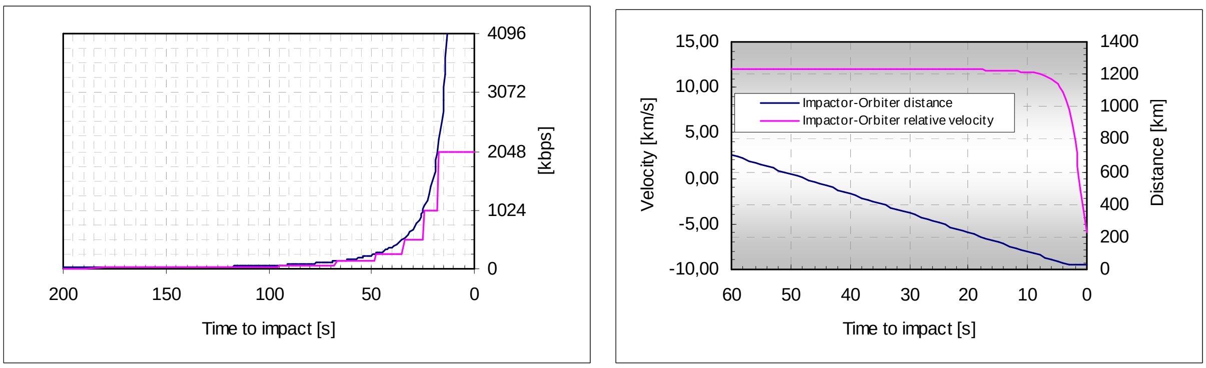
As already stated, the UHF link on the impactor is used to relay data to the G/S passing through the DQ Orbiter, before the impact.

Based on mission analisys and SRD, the communication with the orbiter should start after the last correction manouver has been performed. This happens about 5 minutes before the impact, with the Impactor being at about 5000Km away from the Orbiter.

At that distance a low data rate is assumed, as the onboard camera, which is the main data rate driver during this phase, cannot "see" the target asteroid. The camera can distinguish the asteroid shape at a distance of 900 Km away from the asteroid itself, which happens about 90secs before impact).

The approach taken into the UHF section during the proposal design is to adapt the data rate during the approaching phase. This allows to have lower data rates at higher distances, growing up when the Impactor approaches to the asteroid.

During the proposal a preliminary data rate change plan was presented in the following figure.



However this preliminary approach did not take into account the time needed by the onboard transceivers to handshake the new data rate. This has been estimated with the order of magnitude of about 8-10 seconds. Considering also the short time window available for communication and the very high velocity of the impactor, these few seconds can imply a great data volume losses at the higher data rates.
Also, it is safer for reliability reason to reduce the number of the data rate changes, as one failure during this phase cannot be recovered in time.

A new data rate change plan is proposed below (TBC during phase A):

Distance to Asteroid	Data rate	Duration	Time to impact
5000Km	4kbps	200 seconds	5minutes
910Km	Data rate change	10 seconds	100secs
900Km	256kbps	30seconds	90secs
410Km	Data rate change	10 seconds	60secs
400Km	1024kbps	50 seconds	50sec

The total data volume that can be transmitted this way is about 59 Mbit.

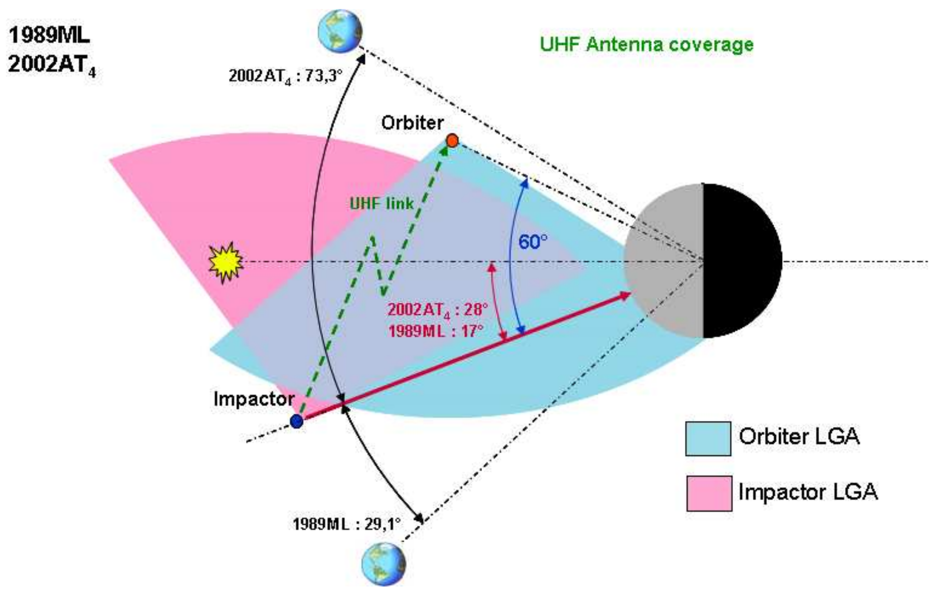


Table 4.1.9-2: UHF foreseen antenna coverage (from proposal, TBC)

4.1.10 TT&C EQUIPMENTS

4.1.10.1 Heritage

As general guideline, maximum reuse of of-the-shelf unit has been considered in defining the TT&C subsystem architecture. In particular the Impactor X Band requirements and mission characteristics are very similar to another space programs, i.e. Hershel/Planck. Commonality will be implemented to the maximum extent, taking into consideration the envelope of the requirements.

This solution allow to:

- Reuse flight spares when possible, i.e. lower cost
- Simplify the validation process

The following considerations apply:

- Frequency allocation is the same as H/P (X Band section).
- A trade off is on for the X Band HGA (X). Several alternatives are available from Venus Express, Mars Express. A recurrency from the orbiter (with a different feed) can also be studied.
- The LGAs can be considered as recurrent from the Hershel/Planck program at least for what concern the RF Design.
- the MGA RF design concept is derived from the Hershel/Planck one. However higher gain is required with respect to the HP MGA, therefore a partial redesign is foreseen at least for what concern the horn length.
- TWTA power is increased wrt Herschel/Planck to match the increased distance from the Earth. However, the X band TWTAs are considered fully recurrent units from Mars Express.
- the RFDN (at least for the X-Band part) is composed by units already used in previous programs, in particular the switches, couplers, isolator and diplexer. Obviously the overall design has to be tailored to DQ and for this reason the RFDN can not be considered as recurrent unit.
- No major commonalities (beside Bepicolombo) are foreseen for the HGA, the DST, the Ka-TWTA and the MGA pointing mechanism.
- As far as the HGA and DST are concerned a minor heritage can be considered since similar units have been developed in the frame of the Cassini mission, which provides robust guidelines for the DQ implementation.

4.1.10.2 Equipment Description

In the following pages a list of possible equipments related to DQ are described in detail.

4.1.10.2.1 Deep Space Transponder

The current state-of-art design for deep space transponders is the Alenia DST, developed by AAS-I Rome on the basis of the Rosetta, Mars Express and Venus Express heritage.
The DST is capable to acquire, lock and demodulate the RX signal in X-Band and to synchronize, generate and modulate the TX signal in both X and Ka bands.

The actual design is based on a digital architecture, i.e. only the signal acquisition and signal/frequency generation are on analogue technology. This solution allows meeting the functional requirements with the following advantages:

- Receiver reconfigurability according to the received signal input power;
- Easy implementation of narrow loop bandwidths;
- Inclusion of data demodulation capability (subcarrier tracking circuit, bit-synchronizer);
- Data rate flexibility with easy matched filtering implementation;
- Interface optimization based on MLC and DS16;
- Design flexibility due to software tuning of signal processing algorithms;
- Direct digital frequency synthesis (DDFS);
- Digital modulation capabilities

A Top Level block diagram of the DST is shown in Figure 4.1-11.

Figure 4.1-10: Alenia Rome DST Transponder

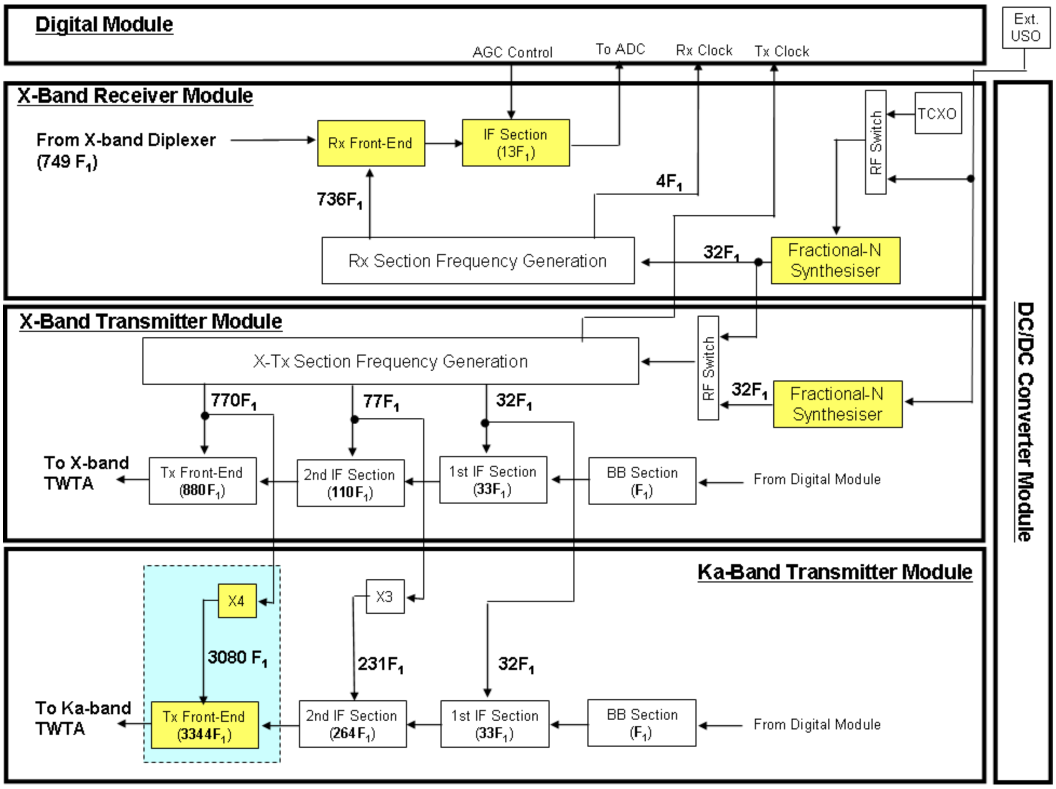


Figure 4.1-11: DST Top level block diagram with frequency distribution

The DST architecture is composed by a Digital Module, a Receiver (analogue) Module, an X Band Transmitter (analogue) Module and a Ka Band Transmitter (analogue) Module. **For the Impactor, the Ka module will not be implemented or modified accordingly.**

The Digital Module includes all the functional TX and RX features in a single board (with internal power redundancy) such as:

- All-digital modulation/demodulation capabilities including advanced format such as GMSK, SRRC-OQPSK and the traditional residual carriers modulation scheme (i.e. PM/BPSK/NRZ, PM/SP-L).
- Convolutional and Turbo-Coding coding/decoding support (if not already included in the CDMU);
- Transparent and Regenerative Ranging operations (for X/X and X/Ka links);
- Digital Signal Shape Filtering
- The digital architecture allows supporting the following advanced features:
- Short-Loop (based on Phase Rotator) and Long-Loop based architecture;
- Autonomous carrier acquisition by local sweep;
- Advanced tracking capabilities (suppressed and residual carrier, 2nd and 3rd-order loops);
- Advanced data demodulation (for NRZ and SP-L) with lower implementation losses;
- Embedded Micro-controller working as a sequencer in order to manage the transponder configuration (no external Microprocessor is needed);
- Possibility to change sampling frequency according to the mission phase in order to optimise the performance and reduce the power consumption;

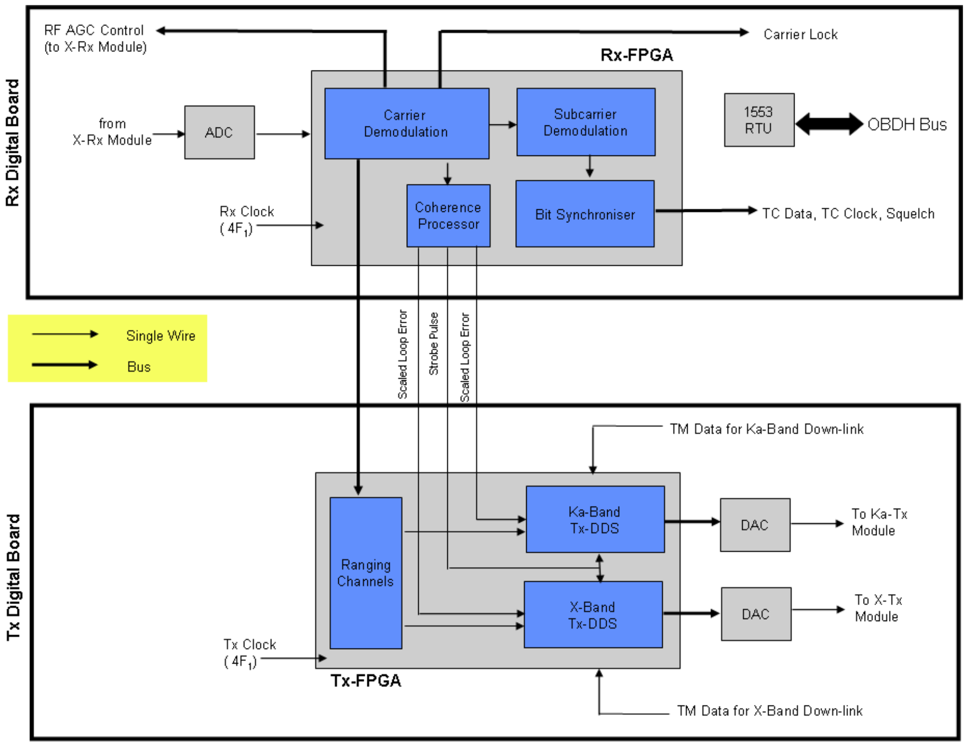


Figure 4.1-12: Digital Module block diagram

The Receiver (analogue) Module converts the incoming X band signal into an RF signal at IF frequency with an Automatic Gain Control (AGC) system. The IF signal will be then A/D converted for further processing by the digital module. A TCXO and a digital frequency generation block provide the uplink RF frequency.
For coherent operations, the receiver will lock on the uplink frequency carrier coming from the ground station, and uses the onboard generated uplink frequency to accelerate the locking operations.

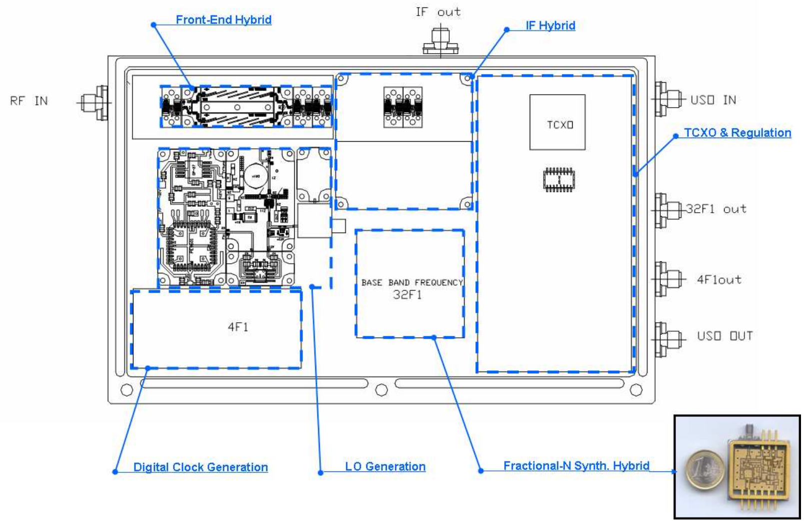


Figure 4.1-13: Receiver Module Board schematic

The X Band Transmitter (analogue) Module modulates the digital signal D/A converted by the digital module into an X Band signal according to the chosen modulation scheme, bit rate and ranging. For coherent operations, the TX signal frequency is generated from the uplink one. The X Band Transmitter also provides an IF signal for the Ka Band Transmitter.

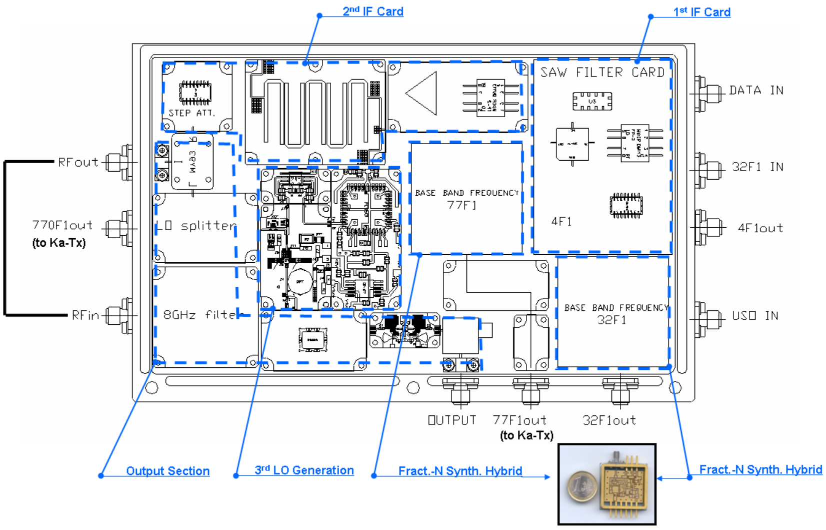


Figure 4.1-14: X Band Transmitter Module board schematic

The Ka Band Transmitter (analogue) Module generates the Ka Band signal according to the chosen modulation scheme, bit rate and ranging. The Ka signal is derived from the X band one through several conversions at different IF frequencies. The module also includes the DDOR block for DDOR ranging support.

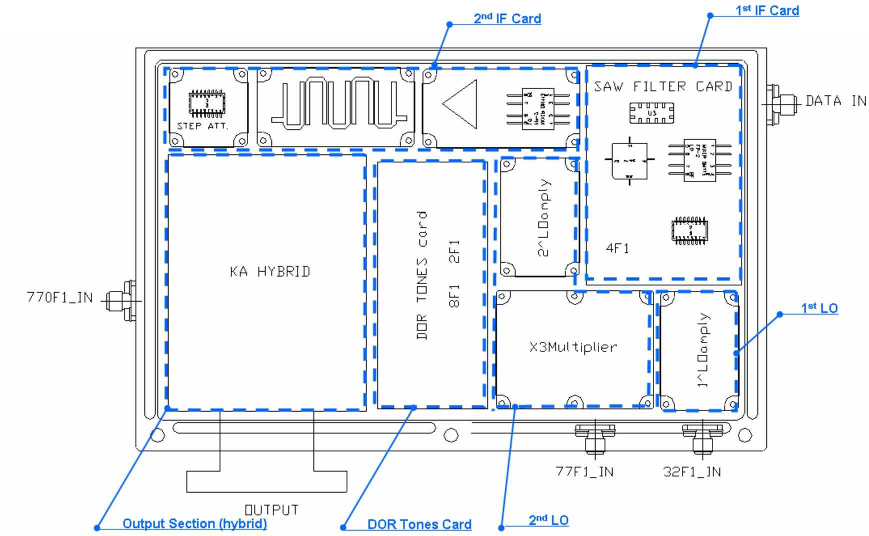


Figure 4.1-15: Ka Transmitter Module Board schematic

The following table provides a summary of the main parameters of the DST.

Parameter	Value X-Band	Value Ka-Band
Uplink frequency allocations	7145-7190 MHz	N/A
Downlink frequency allocations	8400-8450 MHz	31.8-32.3 GHz
Frequency translation ratios (in coherent mode)	880/749	3344/749
DST parameters UPLINK		
Noise Figure at receiver input	2 dB	N/A
Carrier Acquisition threshold	-143 dBm	N/A
AGC Input Bandwidth	3 kHz	N/A
PLL Bandwidth (2Blo)	30Hz	N/A
PLL Threshold	10 dB	N/A
PLL Damping Factor	0,71	N/A
Carrier Recovery Implementation Losses	1 dB	N/A
Required C/N_0 in PLL bandwidth	10 dB	N/A
Telecommand Recovery Implementation Losses	2 dB	N/A
Ranging Channel Bandwidth	5 MHz	TBD
Ranging Channel Impl. Losses	1 dB	TBD
Ranging Channel Regenerative ranging	Capable	Capable
Ranging Channel RNG Waveform	Sine	TBD
DST parameters DOWNLINK		
TM Modulation index	0.2 – 1.25	0.2 – 1.25
RNG Modulation index	0.2 – 0.7	0.2 – 0.7
Power Consumption		
RX	11 W TBC	
RX + TX (X Band)	17 W TBC	
RX + TX (X Band) + TX (Ka Band)	20 W TBC	
Size (Envelope)	215x176x125 mm	
Mass	3.3Kg	

Table 4.1.10-1: DST parameters

4.1.10.2.2 High Power Amplifiers

The proposed design foresees both for X-Band and Ka-Band TT&C signals amplification the usage of Travelling Wave Tube Amplifiers. The X band one shall provide at least 65W RF output power at saturation, while the Ka band output power will be 35W (TBC). This solution has been selected since TWTAs provides a better power efficiency with respect to a solid state power amplifier, allowing significantly reducing the consumption.
The TWTAs are made up by an Electrical Power Conditioning (EPC) and by the Travelling Wabe Tube itself. The first provides the high voltage necessary to drive the tube, which actually amplifies the RF signal. As far as the tubes are concerned the only European manufacturer on the market its Thales, which provides the tube for almost all European satellites. The EPC are manufactured by several companies in Europe (e.g. AAS-Belgique, Galileo Avionica, Tesat), therefore it is not considered a technological issue.
As far as the X-Band is concerned, the foreseen solution is to use the off-the-shelf Thales tube, with an EPC manufactured by AAS-Belgique. This is the same solution used in the frame of the Mars Express program, which gives a higher confidence on the achievable results. The following table provides the main characteristics of the X-Band amplifier. AAS-Belgique has also recently developed a low mass EPC which can be considered an alternative solution for mass saving during phase B.

Parameter	Value	Notes
Operating Frequency Range	8400 ÷ 8450 MHz	
Saturated Output Power	65 W	
Power Consumption @ P_{OUT} = P_{SAT}	140W	(about 50% power efficiency)
Size (envelope)	TBD	
Mass	2,88 Kg	

Table 4.1.10-2: X Band TWTA parameters

Thanks the proposed TT&C configuration, where just one signal has to be amplified by each TWTA, the preferred approach is to drive the tube at saturation in order to provide the maximum available output power minimising at the same time the power consumption. However, the X band exciters inside the X DST will include an Automatic Level Control circuits capable to control the TWTAs input power over 10 dB of dynamic range. In this way, it will be possible to select the TWTA operating point even during flight, in order to tailor it to the mission needs.

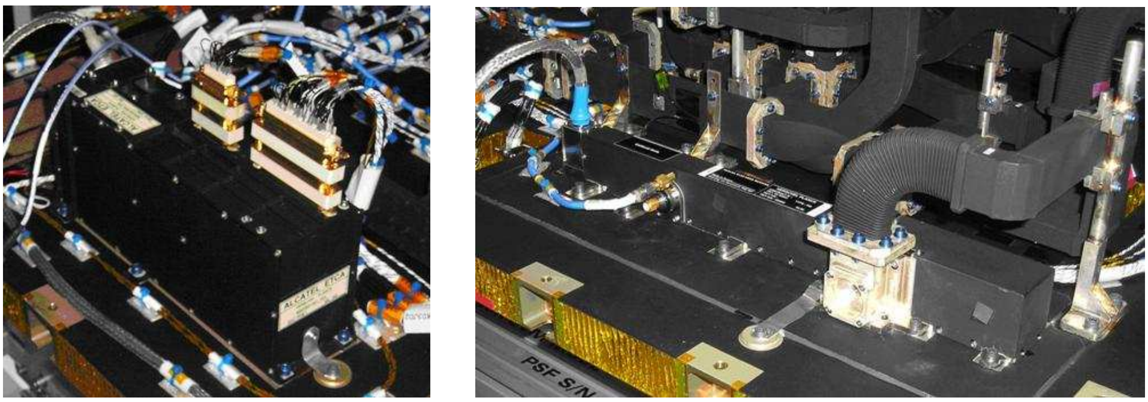


Figure 4.1-16: X Band EPC (left) and TWT (right) from Herschel Planck TT&C integration phase

4.1.10.2.3 RFDN

As indicated above, the Radio Frequency Distribution Network (RFDN) is in charge of connecting the Transponders to the antennas.

In the following, a short description of the main units composing the RFDN is provided (i.e. 4-Port Switches, Isolators, X-Band Diplexer, Ka-Band Diplexer). A summary of the overall RFDN characteristics is also provided. The **4-Ports Switches** are (both X and Ka band) are vented magnetically latching devices. TESAT hi-rel parts are the baseline for this proposal. They connect two of the associated waveguides according to the functional diagram in the following Figure 4.1-17. The numbering of RF ports and an external view of the switches are also depicted.

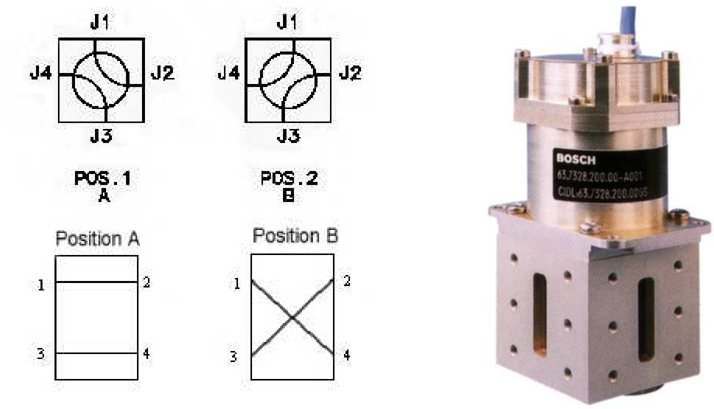


Figure 4.1-17: X Band 4-Port Switches

Typical characteristics of these devices are presented in the following tables for X and Ka band respectively.

Parameters	X-Band
RF PERFORMANCE	
Frequency Range	6580 - 10000 MHz
Insertion Loss (max)	0.05 dB
Return Loss (min)	30 dB
Isolation (min)	70 dB
Power handling (avg)	500 w
Power Handling (peak)	1000 w
DRIVE CHARACTERISTICS	
Type	2 Coil Random Access
Voltage	24 – 32 V
Current (max)	500 mA (typ. 350 mA)
Coil Resistance	70 – 100 Ω
Pulse Length	600 – 1000 ms
Switching time	500 ms
MECHANICAL CHARACTERISTICS	
Dimensionsion	50 x 50x 120 mm
Mass	320 g
DC Interface	HD 15 Pin Sub-D
Waveguide interface	WR112
Telemetry	Reed switches
Operating Temperature Range	-40 to +85 ºC
Operational Life Time (min)	15 year
Switching cycles (min)	100 000

Table 4.1.10-3: 4-Port Switches parameters

Isolators exhibiting high isolation combined with low insertion losses exist with suitable characteristics for this application both for X-Band and Ka-Band. As far as the X-Band is concerned, isolators will be procured from TEMEX. A picture of this kind of isolator is shown in Figure 4.1-18 below. In Ka-Band, Sierra hi-rel parts have been selected.

Figure 4.1-18: X Band Isolator

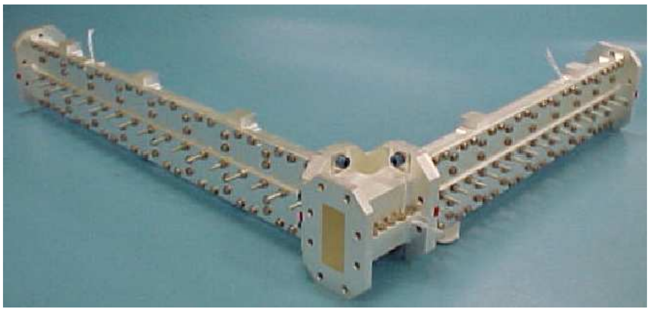
The main performances of the X-Band and Ka-Band Isolators are shown in below.

Parameters	X-Band
RF PERFORMANCE	
Operating Frequency	7000-9000 MHz
Useful bandwidth	150 MHz
Insertion Loss (max)	0.15 dB
Return Loss (min)	25 dB
Isolation (min)	25 dB
Power handling (avg)	65 w
MECHANICAL CHARACTERISTICS	
Dimensions	143 x 54 x 35 mm
Mass	270 g.
Waveguide interface	WR112
ENVIRONMENTAL CHARACTERISTICS	
Operating Temperature Range	-40 to +85 ºC
Non Operating Temperature Range	-40 to +90 ºC
Operational Life Time (min)	15 year

Table 4.1.10-4: Isolators parameters

The design of the **X-Band Diplexer** is based on the one used for Rosetta, Mars Express, Venus Express and Herschel/Planck programs (the same electrical response), developed and manufactured by AAS-Espana.
It consist of an E or H plane divider or bifurcation connecting two bandpass filters performs the separation between Uplink and Downlink frequencies. Interfaces will be standard WR112 flange or SMA for interfacing with DST's.

SMA interfaces will be implemented in the last cavity of the bandpass Rx filter in order to avoid a waveguide to coaxial transition, thus saving mass and volume.
The diplexer will be made in silver plated aluminium and black painted. 3D views of the Mars-Express and HP devices are shown in Figure 4.1-19 below.



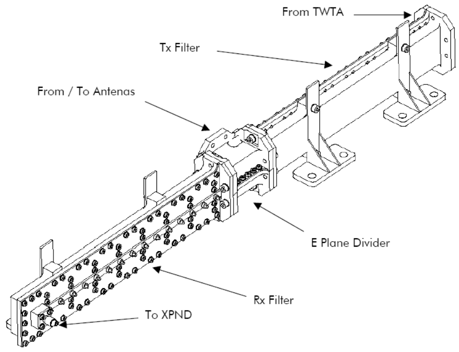


Figure 4.1-19: X-Band Diplexer 3D views

The performances of the diplexer can be summarised in the following table.

	Tx Path	Rx Path
RF PERFORMANCES		
Passband	8 200 to 8 650 MHz	7 100 to 7 350 MHz
I / O Return Loss	20 dB	20 dB
Insertion Loss	0.25 dB	0.25 dB
Rejection	70 dB up to 7 850 MHz	70 dB up to 6800 MHz
	85 dB over Rx band	30 dB over (6.8 – 6.9) GHz
	30 dB over (7.85 – 8.05) GHz	85 dB over Tx band
	30 dB over (8.85 – 9.15) GHz	60 dB over (8.48 – 10) GHz
	60 dB over (9.15 – 11.8) GHz	30 dB over (12.2 – 13.5) GHz
	30 dB over (16.55 – 17.55) GHz	
OTHERS		
Mass	350 g	
RF interfaces	TBD	TBD
Temperature	-30ºC / + 85ºC	

Table 4.1.10-5: Diplexer parameters

4.1.10.2.4 HGA

Due to long Earth-S/C distances involved in DQ project, an High Gain Antenna is needed to establish the X band communication between G/S and Orbiter/Impactor. While Orbiter HGA constraints are well defined by radio science requirements, the HGA design on Impactor has more degrees of freedom. A trade off between several solutions is performed, based on the following design drivers.

- The HGA gain shall guarantee the required TM data rate @ impact range.
- Ka band is not needed on the Impactor, unless a more precise DDOR ranging is required.
- Overall mass shall be minimized.
- Heritage or cost saving solutions shall be preferred.

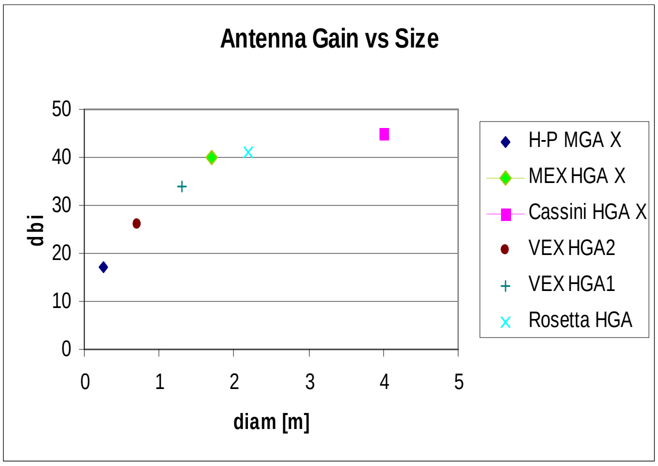
Three possible solutions has been identified:

- A dual band X/Ka 1.0m HGA, recurrent from Orbiter design. Ka band is not used, but is available if needed for precise DDOR ranging measurements. The overall gain is enough to ensure the required data rate. Costs are high due to a new development, but they can be lowered thanks to the recurrency from the Orbiter.
- A dual band X/S 1.6m HGA, recurrent from MarsExpress. Again, only X band capability is used. Gain is higher wrt first solution (+4dB) at the cost of higher mass/envelope (+3-4Kg). This kind of antenna has already been developed/tested for MarsExpress, so overall costs are expected to be lower than first solution.
- A dual band X/S 1.3m HGA, recurrent from Venus Express. This is a good compromise between the 2 previous solutions.

Antenna	Peak Gain	Costs	Mass	Envelope impact
X/Ka from Orbiter design	About 36 dBi	High, recurrency	About 8Kg	Average (1.0m diameter)
X/S MarsExpress	About 40 dBi	Average, recurrency	About 11Kg	Very High (1.6m diameter)
X/S Venus Express	About 38dBi	Average, recurrency	About 10Kg	High (1.3m diameter)

Table 4-6. Impactor HGA Tradeoff summary

The next two graphs show how HGA reflector mass and gain varies wrt antenna diameter. The shape is parabolic, so there will be an optimal point which maximize gain/mass figure.



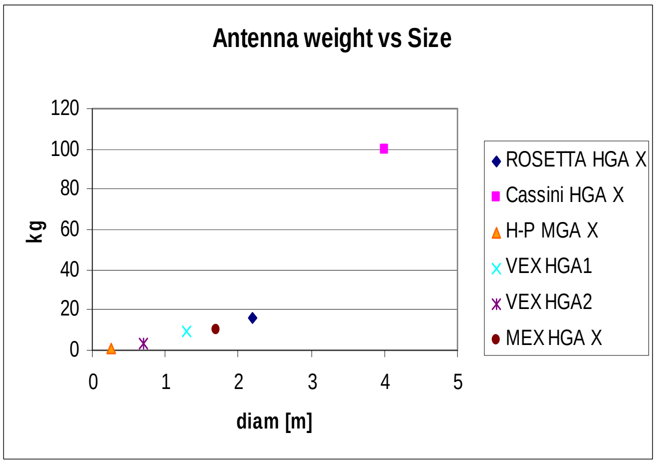


Table 4-7. HGAs Gain and weight vs size from several space missions. Except Cassini, the mass figures are relative to the reflector only (without pointing mechanisms)

As a side note, a custom X band only antenna (to be developed) can be considered as 4th choice. However, this solution seems not worth the expected cost, as an already available dual band antenna can be easily used in one band by simply removing an illuminator.

Thus, the final choice is tied to two factors, i.e. costs and mass/envelope impact on the spacecraft. The third solution, based on Venus Express heritage, seems to be a good gain/envelope compromise. However, problems due to limited satellite envelope can arise.
Currently, a worst performance case has been taken as reference and a 1.0m HGA has been assumed for all link budgets.

Parameter		Value X-Band	Comments
Frequency	Uplink	7149-7189 MHz	
	Downlink	8400-8450 MHz	
Antenna Diameter		1.0m	
Bandwidth		20 MHz	TBC
Gain	Uplink	33 dBi	
	Downlink	36 dBi	
Axial Ratio		0,2 dB	
Polarisation		RHCP/LHCP	
VSWR		1,4:1	

Table 4.1.10-8: HGA parameters

4.1.10.2.5 *MGA*

The MGA antenna proposed design is an X-Band dual-flared horn with global coverage and circular polarization. Its main characteristics are reported in Table 4.1.10-9 below.

Parameter		Value X-Band	Comments
Frequency	Uplink	7149-7189 MHz	
	Downlink	8400-8450 MHz	
Bandwidth		20 MHz	
Peak Gain	Uplink	23 dBi	
	Downlink	23 dBi	
Axial Ratio		0,5 dB	
Polarisation		RHCP/LHCP	
VSWR		1,25:1	
Dimentions		Ø < 300 mm L < 500 mm	Including test cap attachment
Mass		< 700g	

Table 4.1.10-9: MGA parameters

The foresee MGA supplier is Rymsa that already provides a similar antenna in the frame of the Hershel/Planck program. The following figures provides the foreseen MGA mechanical design and RF performances (gain pattern) as obtained by Rymsa by analysis. As can be seen the obtainable performance should be better than the minimum required gain. However, since these have been obtained only by analysis, a margin is still considered.

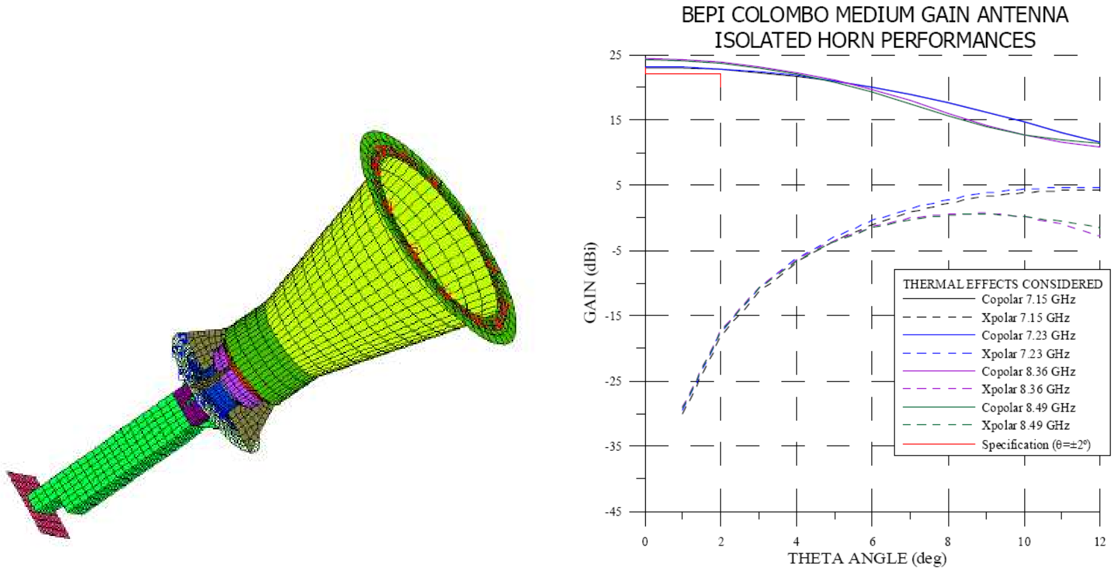


Figure 4.1-20: Expected MGA design and pattern from RYMSA

As in the case of the HGA, also the MGA includes a rotary joint allowing a two-degree pointing, and a boom to provide the necessary distance between antenna and S/C body. Considering that the MGA rotary joint accuracy is worse to that of the HGA, an overall pointing accuracy of 1° both during cruise and on-orbit phases can be considered. This is equivalent to a pointing loss about 0,1dB. In addition, since the MGA is used also during safe-mode, when the S/C attitude accuracy is about 5°, a pointing loss of 1dB has also to be considered. The following tables summarize these performances.

Pointing Accuracy	Pointing Losses (dB)	Notes
1°	0,1	Reference value for Cruise and On-orbit
5°	1	Reference value for Safe Mode

Table 4.1.10-10: MGA pointing losses

4.1.10.2.6 MGA pointing mechanism

A possible implementation for the MGA pointing mechanism has already been studied by SENER in the frame of BepiColombo proposal and may be applicable to the DQ.

The mechanism is supported on a composite boom which acts as a mast. The azimuth motion is created in the lower part of the boom in contact with the spacecraft and the elevation movement in the upper part.

The azimuth axis motion system is located in the lower part in the bracket interfacing the spacecraft. The actuator is hollow to house the wave guide and the rotary joint for the azimuth rotation. Other possibility is having the rotary joint some distance over the actuator.

The actuation system is based in a stepper motor with gearhead and a final stage pinion-wheel.

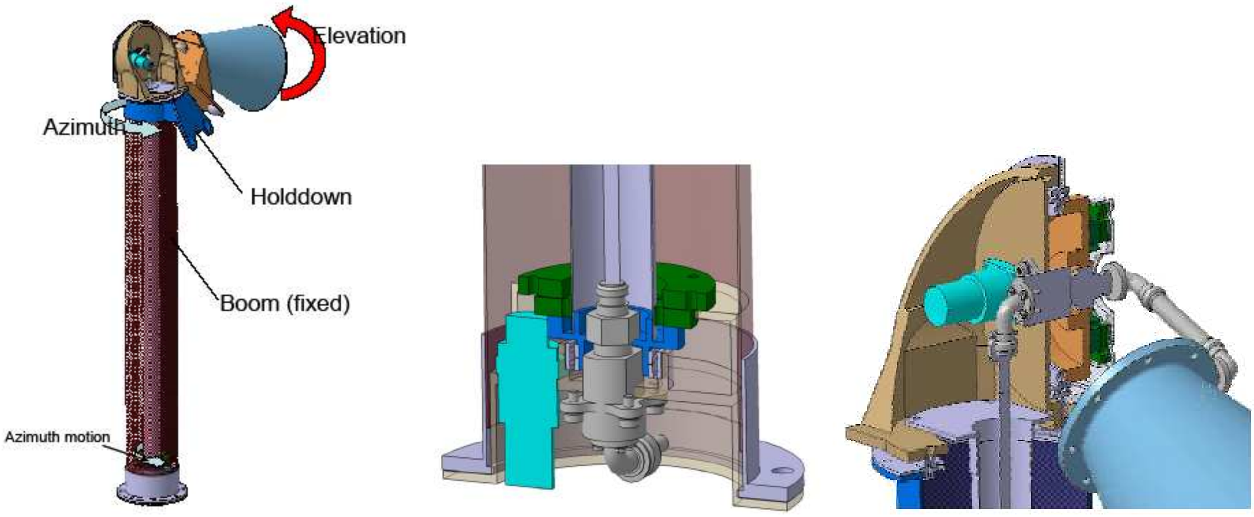


Figure 4-21 MGA pointing mechanism design (left). Azimuth (center) and elevation (right) actuation systems

The upper bracket contains all the connections to the boom and the hold-down mechanism. Spheres or similar features will couple the output bracket with the upper fitting, creating firm support for the antenna and output bracket.
The holddown will be released by redundant pyrotechnic device. Pin pullers or other devices may also be used, the holding force to be released is quite low while the temperatures are the critical requirement for component selection.

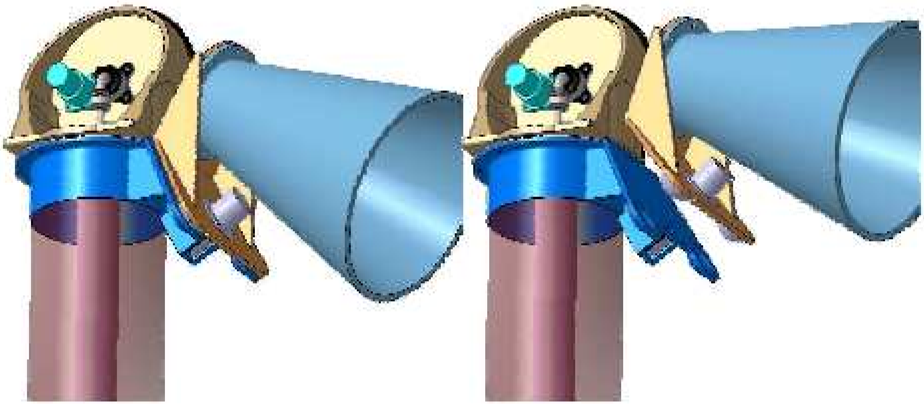


Figure 4-22 Hold down and release mechanism

The whole mechanism accomplishes the full range of travel I from -120º to 90º in elevation axis and -90º+90º, +90º +270º in azimuth axis. Multiple options exist to perform the end stops of each degree of freedom and the limit switch implementation.

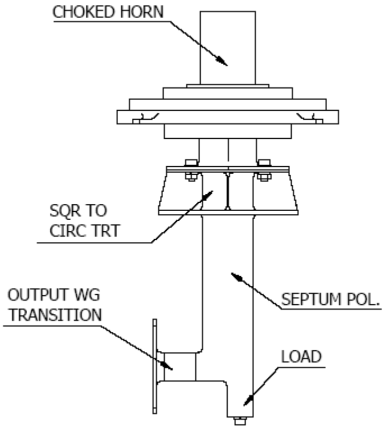
Stepper Step size	30	deg
Gear Head ratio	100	
Wheel pinion ratio	3	
Step size	0.1	deg
Stepper torque	15.7	Nmm
Rate	240	PPS
Max Speed	24	deg/seg
Max Gearhead Torque	1.57	Nm
Max Output torque	4.239	Nm

Figure 4-23 Different position and actuator performances

The expected mass of the pointing mechanism (excluding boom) is about 3 Kg.

4.1.10.2.7 LGA

The objective of the LGAs is to guarantee an omnidirectional coverage with a minimum gain of –3dB. Such assumption needs to be verified when a detailed configuration will be frozen and a GTD analysis will be performed, but in general this is the common approach followed to define the omnidirectionl coverage from the RF point of view.
The baseline LGA design, based on the HP LGA manufactured by RYMSA, consists in a radiating element (a chocked horn) connected to the RFDN through a septum polarizer and a square-to-circular waveguide transition, which converts the linear polarization of the waveguide into an RHCP/LHCP through two different ports.



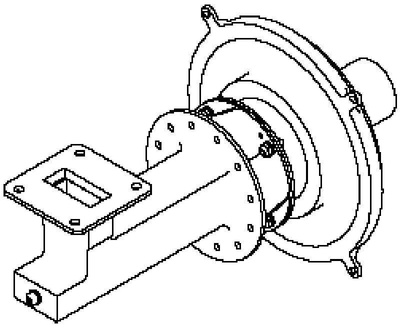


Figure 4.1-24 Herschel-Planck LGA design

The following table shows what the main antenna characteristics are. It shall be highlighted that in case of LGA, the pointing losses are assumed to be zero, since the antenna does not need to be pointed, and the –3dB gain takes into account each possible signal direction of arrival.

Parameter		Value X-Band	Comments
Frequency	Uplink	7149-7189 MHz	
	Downlink	8400-8450 MHz	
Bandwidth		20 MHz	
Gain	Uplink Downlink	-3 dBi	
Axial Ratio		4 dB	
Polarisation		RHCP/LHCP	
VSWR		1,25:1	
Antenna Noise Temp		TBD K	
Power Handling		35 W	
Dimensions		Ø < 135 mm L < 210 mm	Including test cap attachment
Mass		320g	

Table 4.1.10-11: LGA parameters

The following figures provides a picture of the Herschel/Planck antenna and its RF performances (gain pattern) as obtained by Rymsa. These can be considered fully representative of the DQ ones.

Figure 4.1-25: LGA from RYMSA

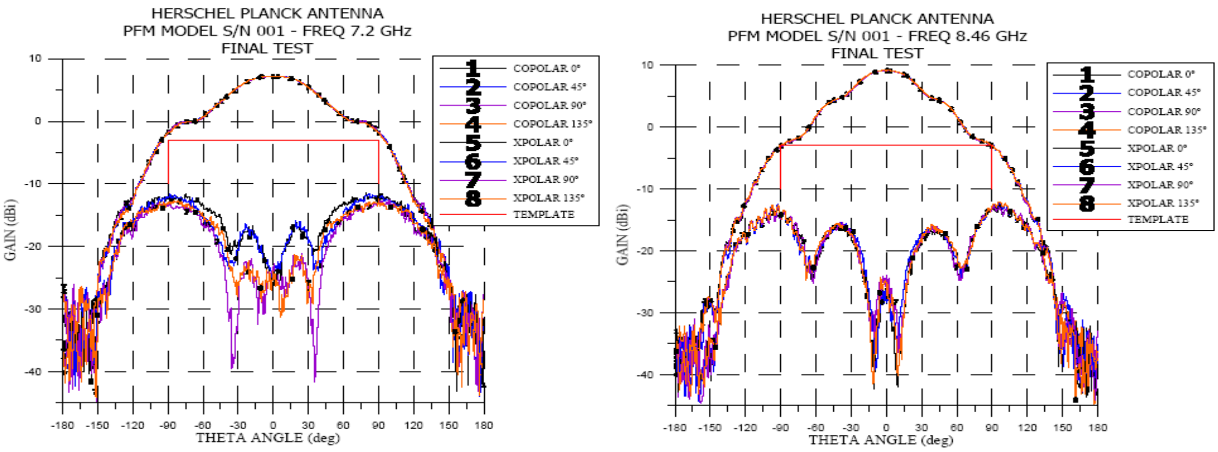


Figure 4.1-26: Herschel-Planck FM LGA gain pattern

4.1.10.2.8 UHF Transceiver

The current market of the UHF transceivers proposes two possible choiches:

- The American ELECTRA transceiver (EUT), produced by L3 Communications, which has successfully used in several NASA missions like MER and MRO. Recently, a new model called ELECTRA LITE has been developed, with improved mass and power consumptions wrt the original one.

- The French SMART-UHF, produced by ELTA, which was developed under CNES contract and represents a good european alternative in the framework of the future martian missions.

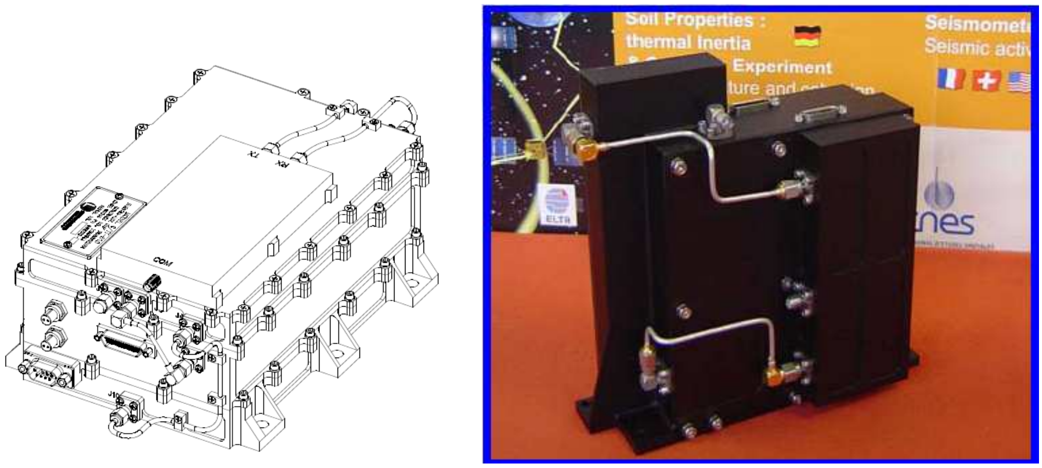


Figure 4.1-27: NASA ELECTRA Lite (left) and SMART (right) UHF Transceivers

A brief description of the EUT is reported below.

The EUT assembly consists of five modular slices. From top to bottom, the slices are:

- Half-duplex overlay (HDO) receiver filter and UHF diplexer
- Filtering and switch unit (FSU)
- UHF radio frequency module (RFM, the receiver and transmitter)
- Baseband processor module (BPM)
- Power supply module (PSM) with integral power amplifier module

The FSU slice in the EUT consists of a high-isolation diplexer, the HDO receive/transmit (R/T) switch, and the coaxial transfer switch. The BPM slice can be interfaced directly with the CDMU, a solid state recorder, the USO, and the modules that comprise the EUT.
The RFM slice consists of a single-channel UHF transmitter and receiver.
The PSM slice consists of the power supply and the driver/power amplifier. The PSM provides power to the BPM and, under BPM control, to the elements of the RFM. The PSM slice also includes a power amplifier that amplifies the modulated signal to the appropriate RF output level.
The BPM performs all signal processing, provides overall EUT control, and services the external spacecraft interfaces.
The functionality of the modem processor (MP) portion of the BPM is summarized in block diagram form in the next figure.

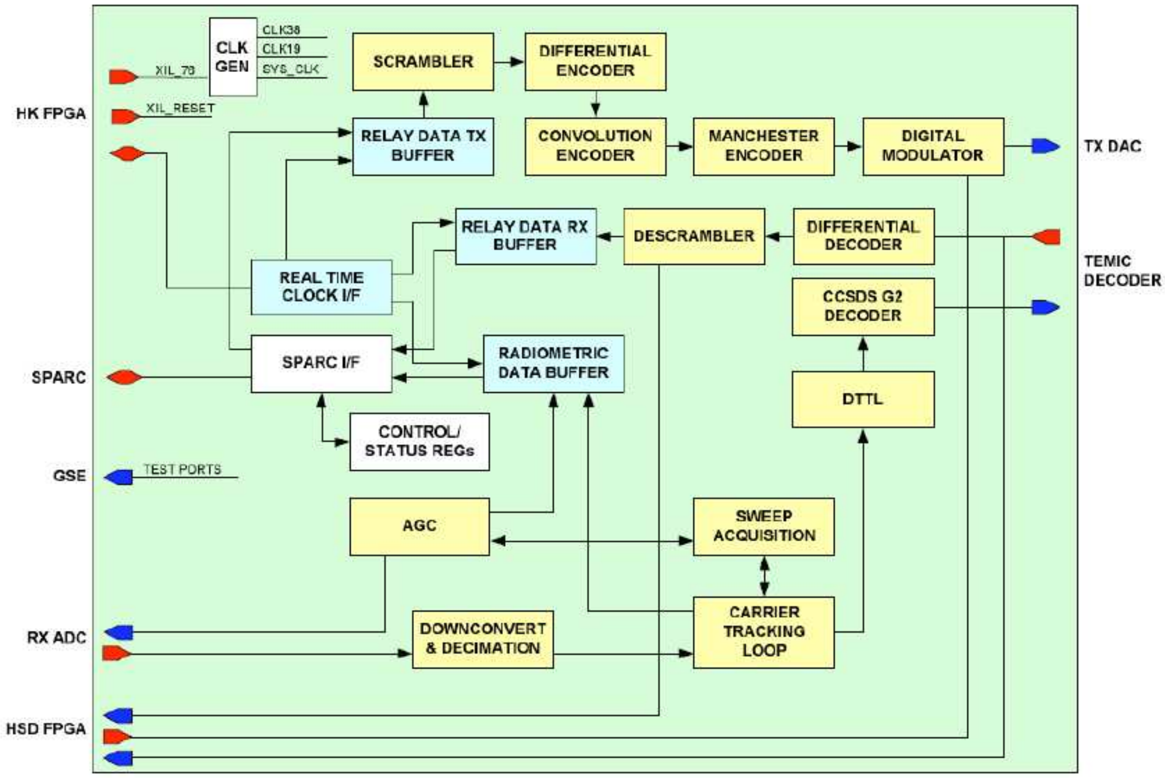


Figure 4.1-28: ELECTRA Modem Processor

The BPM consists of a 32-bit microprocessor, two radiation-hardened program-once field programmable gate arrays (FPGAs), and a large (~1Mgate) reprogrammable FPGA, along with a substantial amount of dynamic and static memory. The reprogrammable FPGA contains the
modem functions and is reprogrammable post-launch. The 32-bit microprocessor manages the
EUT and the relay Prox-1 protocol.

In concept, one side of the BPM handles the spacecraft interfaces. A dedicated 1553 transceiver chip supports the command and telemetry interface to the host CDMU. An LVDS interface supports high-rate relay and radiometric data transfers through the high-speed data (HSD) FPGA. The other side of the BPM handles the EUT, with the housekeeper (HK) FPGA managing control and telemetry signals to and from the EUT front end, and the MP FPGA.

The main functions of the MP FPGA include:

- Coding and decoding
- Modulation and demodulation
- Carrier, symbol, and decoder synchronization
- Prox-1 frame synchronization detection
- Prox-1 transmit (Tx) and receive (Rx) user data and control data buffering
- Receive signal level management, automatic gain control (AGC) Radiometric Doppler and open-loop record functions Clock (CLK) and timestamp functions Implementation of the physical layer of the communication link from baseband to an intermediate frequency (IF).

The MRO Electra does not have an internal clock. The clocks for the BPM FPGAs, including bit, symbol, and sample rate clocks, are derived from the external USO.
MRO Electra implements frequency agility and swappable transmit and receive bands.

Parameter	Electra Lite
TX Frequency	390 to 405 MHz
RX Frequency	435 to 450 MHz
Duplex	Half OR Full
Operational Modes	FD: Rx, Rx/Tx
TX/RX Rate	1,2,4,8…4096 Ksps
Modulation	Manchester, NRZ-L, BPSK, QPSK (TBD), Mod Index 60 & 90
Coding	Reed Solomon, K=7, R=1/2 Conv Encode/Decode
Spectrum Record	Open Loop Signal Sampling < 100 KSPS, 1-8 bits/sample
RX Noise Figure	FD = 4.5 dB, HD=3.9 dB
RF TX Power	FD: 8.5 W; HD: 10.5W
Protocols	Proximity-1
Reconfigurability	Yes
Doppler Obs	1-way OR 2-way (HD, FD)
Mass	3000 gms (w/Diplexer)
Volume	~2872 cm^3
Dimensions (l,w,h)	20.3 cm x 13.1 cm x 10.8 cm
DC Power - RX Mode	18.5 (WC, EOL)
DC Power - TX/RX Mode	60.2 (WC, EOL)
Parts Grade	B+
TID	6.5 Krad (MSL Req't)

Figure 4.1-29: ELECTRA LITE Features

4.1.10.2.9 UHF Antenna

The chosen antenna for both impactor and orbiter is a quadrifilar helix in axial mode.
This antenna consists of four wire radiators curved symmetrically around the antenna axis. The axial mode has its radiation peaked along its axis and can be made relatively small, it can provide a relatively broad beam with good CP radiation. It can be designed to achieve approximately 140 deg (+-70 deg) of beamwidth and a peak gain of about 4dBi. A circular ground plane with a diameter of approximately 25 cm is needed for the axial-mode antenna. The total height of the antenna above the ground plane is about 40 cm, and overall mass is about 1Kg.

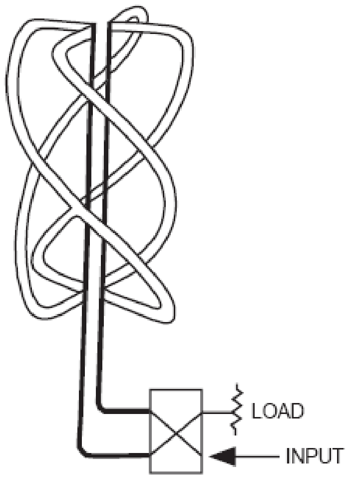


Figure 4-12: UHF LGA Design

Parameter		Value X-Band	Comments
Frequency	UHF Band	400-450MHz	
Bandwidth		7MHz	TBC
Gain	Uplink	4.5 dBi peak	
	Downlink		
Axial Ratio		4 dB	TBC
Polarisation		RHCP/LHCP	
VSWR		1,25:1	TBC

Table 4-13: UHF - LGA Parameters.

The gain pattern from UHF helix antenna from MRO mission is shown in the next figure as reference.

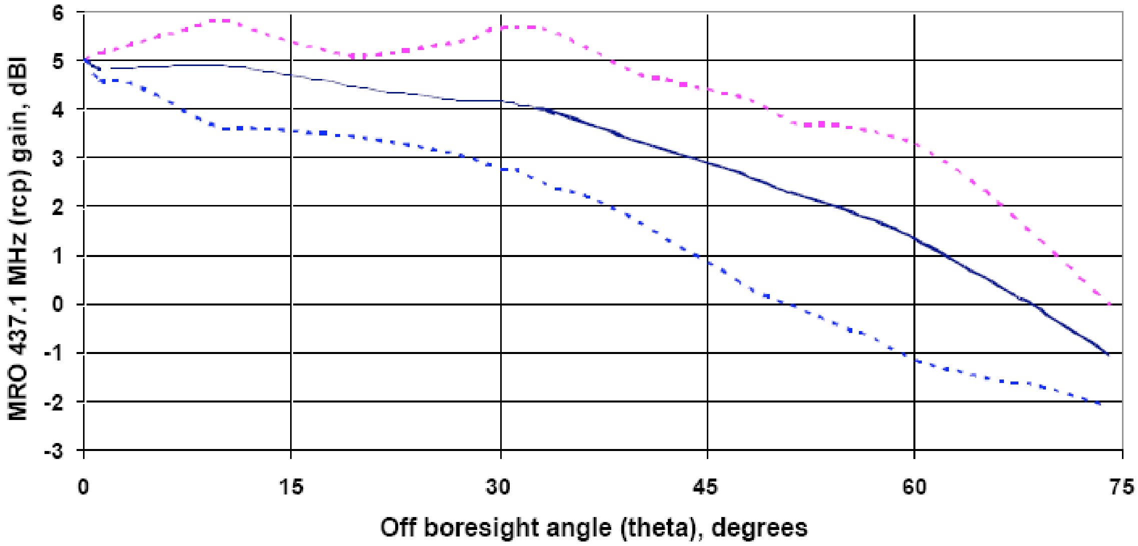


Figure 4.1-30: MRO UHF LGA Gain pattern (437.1 MHz) – NASA

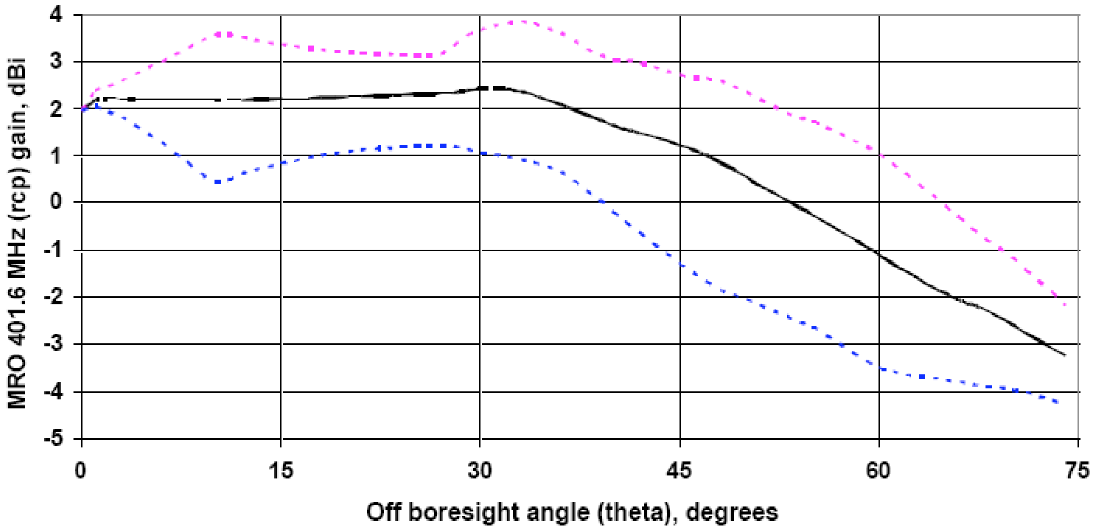


Figure 4.1-31: MRO UHF LGA Gain pattern (401.6 MHz) – NASA

5. ANNEX 2: POWER SUBSYSTEM DETAILED DATA

5.1.1 Impactor Electric Power Subsystem (EPS)

The Impactor Electrical Power Subsystem (EPS) provides the following main functions:

- Generation of electrical power by means of a Solar Array (SA)
- Power control, storage and distribution of electrical power to/via a main bus
- Management of battery charge/discharge
- Provision of regulated main bus power to the unit at 28 Vdc
- Provision of status monitoring and telecommand interfaces for subsystem operation and performance
- Provision of adequate redundancy and protection circuitry to avoid failure propagation and to ensure recovery from any malfunction within the subsystem and/or load failure

Since, maximun commonality between Orbiter and Impactor have been implemented. The Impactor PCDU uses the same design of Orbiter including less power regulators to be compatible with the SA power generation capability and Battery capacity.

The EPS consist of the following equipment:

- Solar Array
- PCDU
- Battery
- Interconnecting Cables

The NEO Impactor EPS driving points are the following:

- Power demand up to 800 W during final target phase;
- Solar Generator (SA) consisting of 3 body mounted panels equipped with Triple Junction GaAs solar cells for a total area of 5m².
- SA power regulation : MPPT;
- Battery supplying the Satellite during Pre-Launch, Launch, Orbit injection and separation from the launcher up to sun acquisition (about 180 minutes)
- Battery cells technology : Li-Ion ;
- SA cells technology: Triple junction GaAs, providing an efficiency figure of at least 26%;

The Solar Array consist of:

- 1 Body mounted panel made by a sandwich of aluminium honeycomb with Carbon Fibre Reinforced Plastic skin and equipped with SCA assembly.

- 2 deployable panels made by a sandwich of aluminium honeycomb with Carbon Fibre Reinforced Plastic skin and equipped with SCA assembly

- Interconnecting harness and Temperature sensors

The Solar Cells Assembly (SCA) includes the following components :

- Triple junction GaAs Solar cells
- Coverglass
- Coverglass adhesive
- Interconnector
- By-pass diode

The figure shows the SCA components:

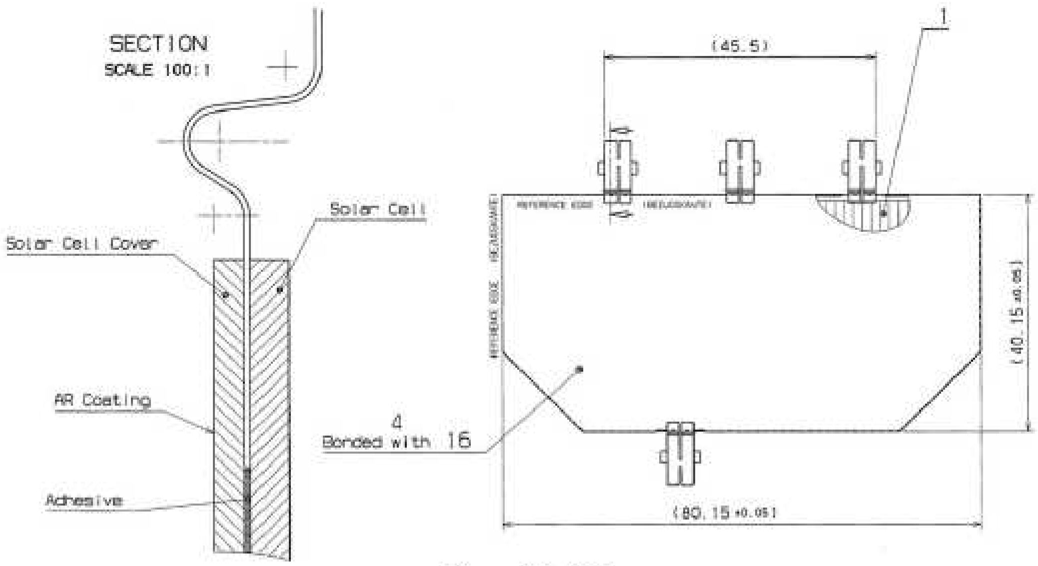
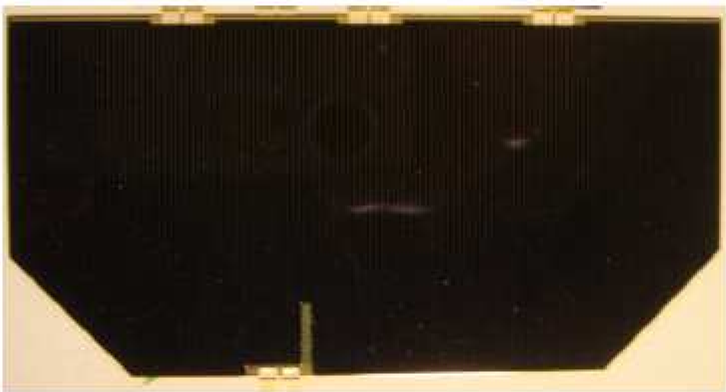
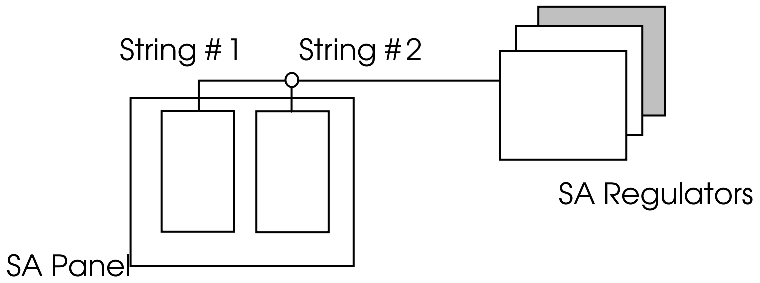


Figure 3-1: SCA



The PCDU consists of :

- 9 Solar Array Regulators (SAR) modules (V_{SAin} = 27V - 96V, I_{SAin} < 7,5A, Pwreff. 95%) configured as 3x3 to guarantee 1 single point failure free autonomous MPPT regulators;



- 1 bus module containing three voted Main Error Amplifier (MEA), DNEL Electronic and power and signal backplane;
- 2 nominal plus 2 redundant Battery Charge Discharge Regulators (BCDR) modules;
- 1 capacitor bank split in different modules;
- 4 Nominal plus 4 redundant Power Distribution modules providing :
    - o 24 nominal LCLs;
    - o 24 redundant LCLs
- 1 Nominal plus 1 redundant Power Distribution modules providing :
    - o 6 nominal FCLs (every 6 FCL are protected by LCL)
    - o 6 redundant FCLs (every 6 FCL are protected by LCL)
- 2 Heater modules providing :
    - o 12 nominal Heater switching (HSW) lines;
    - o 12 redundant Heater switching (HSW) lines
- 1 nominal plus 1 redundant Pyro modules providing :
    - o 16 nominal pyro firing lines
    - o 16 redundant pyro firing lines

- Two Command and monitoring modules
- Internal Harness, Voltage, Current and Temperature sensors
- Case and electrical connectors

In the following table there is the mass budget of the PCDU.

PCDU Items	n° Board	Mass Board [Kg]	Thick [mm]	Tot mass	Notes
SA MPPT reg	9	0,5	24	4,5	2 SA sections per 3 regulators
BCDR	4	0,55	24	2,2	
TMTC	2	0,35	24	0,7	
PDU	10	0,55	23	5,5	
Heater module	2	0,625	23	1,25	
bus capacitor	1	2	23	2	
case	1	2,5		2,5	
pyro actuator	2	1,25	23	2,5	
backplane+mea	1	1	24	1	
Contingency	20%			4,43	
Total mass [Kg]				**26,58**	

On the basis of the internal configuration the estimated dimensions of PCDU are the following :

With = 158 mm

Height = 67 mm

Length = 750 mm

The figure shows an example of the PCDU configuration and envelope

The Battery consist of:

- Lithium-Ion battery ; Cell type :"Sony 18650";
- Heaters;
- Harness and Temperature sensors
- Case and electrical connectors

In the following table there is the mass budget of the NEO Impactor Battery.

Battery Items	Number of Items	Mass [Kg]
Li-Ion battery cell	300	10
Heaters	6	0,1
Case + Electrical connectors	1	1,7
Interconnection Harness	1	0,2
Contingency 20 %		2,4
Total with contingency		14,4

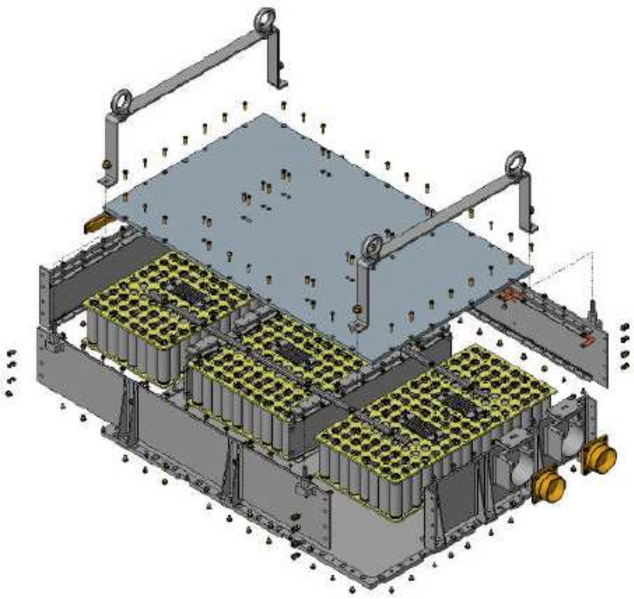
On the basis of the internal configuration the estimated dimensions of Battery are the following:

With = 354 mm

Height = 102 mm

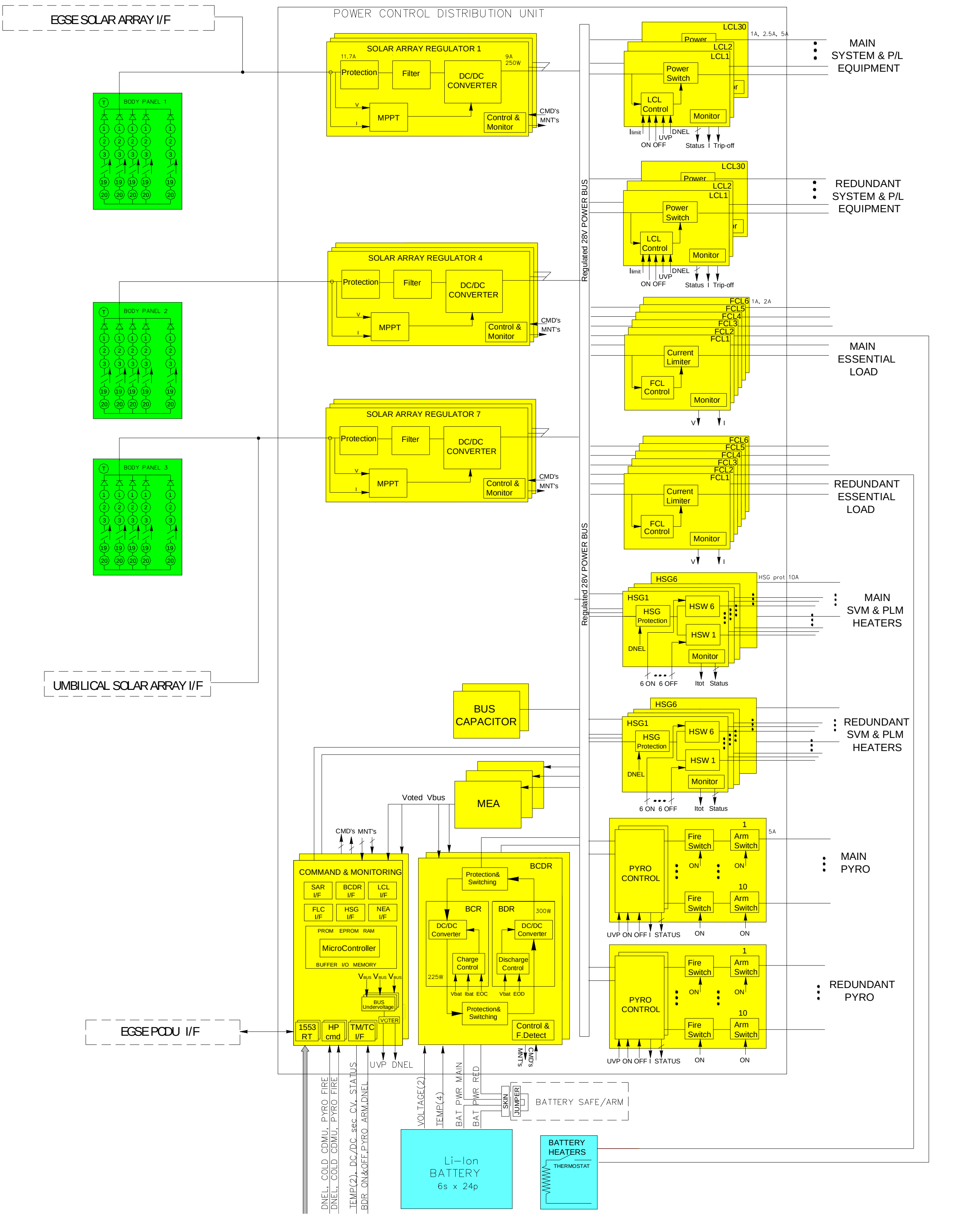
Length = 600 mm

The figure below shows an example of the battery configuration and envelope.



Interconnecting Cables consisting of power cables connecting the SA with PCDU, Battery with the PCDU, signal harness interfacing the PCDU with the distributed temperature sensors and Power Bus cables connecting the PCDU with the Electronic Units, Electrical devices and Heaters

The figure below shows the Impactor Electric Power Architecture.



EGSE SOLAR ARRAY I/F
POWER CONTROL DISTRIBUTION UNIT
BODY PANEL 1
BODY PANEL 2
BODY PANEL 3
SOLAR ARRAY REGULATOR 1
SOLAR ARRAY REGULATOR 4
SOLAR ARRAY REGULATOR 7
Protection
Filter
DC/DC CONVERTER
MPPT
Control & Monitor
CMD's
MNT's
Regulated 28V POWER BUS
LCL30
LCL2
LCL1
Power Switch
LCL Control
Monitor
MAIN SYSTEM & P/L EQUIPMENT
REDUNDANT SYSTEM & P/L EQUIPMENT
FCL6
FCL5
FCL4
FCL3
FCL2
FCL1
Current Limiter
FCL Control
MAIN ESSENTIAL LOAD
REDUNDANT ESSENTIAL LOAD
HSG6
HSG1
HSG Protection
HSW 6
HSW 1
MAIN SVM & PLM HEATERS
REDUNDANT SVM & PLM HEATERS
UMBILICAL SOLAR ARRAY I/F
BUS CAPACITOR
MEA
Voted Vbus
COMMAND & MONITORING
MicroController
BCDR
BCR
BDR
Protection& Switching
Charge Control
Discharge Control
Control & F.Detect
PYRO CONTROL
Fire Switch
Arm Switch
MAIN PYRO
REDUNDANT PYRO
EGSE PCDU I/F
UVP DNEL
BATTERY SAFE/ARM
Li-Ion BATTERY
6s x 24p
BATTERY HEATERS
THERMOSTAT

Figure Impactor Electric Power Architecture

The primary source of power is the Solar Array (SA). The SA is composed by 3 body mounted Power from SA is transferred to the Power Control and Distribution Unit (PCDU) and distributed to users via a 28Vdc regulated bus. Power management and regulation is performed in PCDU according to a three-domain regulation scheme. Maximum Power Point Tracker (MPPT) technology is used to avoid solar array oversize.

When the power demand exceeds SA power output or there is a presence of eclipse the power is provided by one Lithium-ion (Li-Ion) battery. The Li-Ion battery combines a low battery mass with high charge efficiency and facilitates the battery implementation in terms of thermal control and battery management.

Two Battery Charge and Discharge Regulators (BCDR) each managing 300 W are required for battery charging and discharging management and operations .

5.1.1.1 Power Storage and Generation

5.1.1.1.1 Power Storage

NEO Satellite Orbiter is equipped by one battery, which provide energy from the launch up to sun acquisition and during LEO eclipse.

The energy required to the battery during the Pre-launch, Launch and Sun Acquisition and LEOP is used to size the battery.

The Battery capacity is calculated under the following conditions:

Modes	Time [min]	Pload [W]
Pre-Launch	30	117,3
Launch	30	149,4
Sun Acquisition	120	334,6
LEOP (eclipse)	36,6	479,5
LEOP (sun)	55,8	550,1

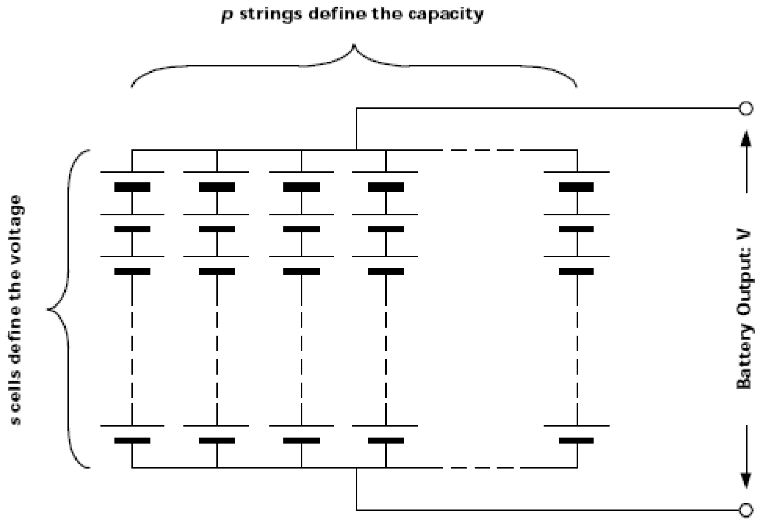
The energy required to Battery during this phase is 1500 Wh.

Li-Ion technology is used due to its efficiency, simple monitoring (the voltage is sufficient to get exactly the charge level), and lack of memory effect. SONY 18650HC cells are used for NEO batteries.

Electrically, the cells are connected as parallel strings, each containing the cells in series. Strings are wired in parallel to produce the specific battery capacity. The power to/from PCDU is provided via nominal and redundant connectors.

The following figure shows the basic electrical concept for the modular battery units. They consist of an array of cells connected in series strings to achieve the correct operating voltage range. Multiple strings are then wired in parallel to produce a specific battery capacity. This series then parallel arrangement of cells is known as ‘*s-p*’topology. Due to the low cell capacity of the Sony mass produced cells, the capacity of the battery can therefore be incremented in small stages to optimise a battery performance to suit specific applications.

Battery module electrical concept – ‘s-p’ topology

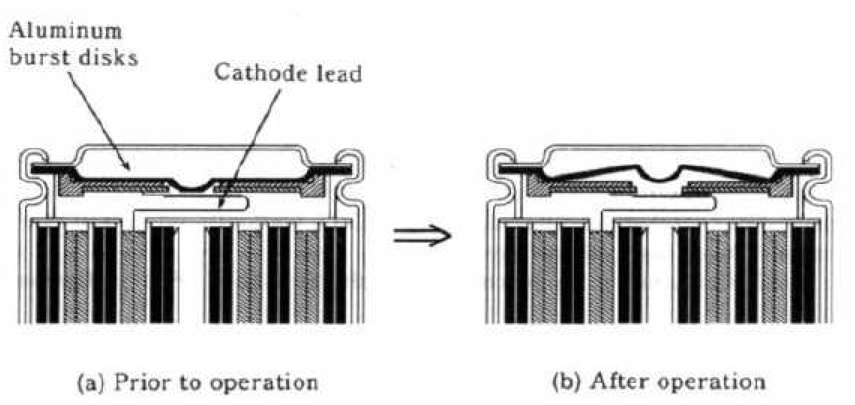


The SONY Lithium-ion cell contains built-in safety mechanisms that prevent hazardous results from severs battery abuse, or cell failure.

The safety mechanisms consist of:

- overcharge disconnect device
- pressure release valve and
- Polyswitch Positive Temperature Coefficient (PTC) mechanism

The PTC mechanisms will permanent open the cell in case of failure. The open circuit failure of the cell results in open circuit failure of the string that reduces the battery capacity by 1/p, where p is the number of string in the battery.
The figure below shows the PTC mechanism operation.



Battery charge occurs at the maximum rate available from the SA until EOC voltage is reached.
At this point the charge power is tapered down towards zero as the battery voltage is clamped at the EOC level (taper charge). The above method of operation is to maximise the charge efficiency of the battery

The major characteristic of the battery are here below summarised:

- Maximum voltage = 16 V;
- Minimum voltage = 26 V;
- Energy density =100Wh/Kg;
- Worst case DOD ~ 85% during LEOP eclipse;
- Cycle numbers ~ 150.

5.1.1.1.2 Impactor Solar Array

NEO Impactor power is generated by 3 body mounted panels.

When the NEO is in stowed configuration, the panels are facing to space.

The SA design major key aspects are:

- Sun distance between 0.7AU to 1.39AU
- Temperature between 50°C@1,39 AU and 130°C @0,7AU
- LEO orbit eclipse of 36,6 minutes (No eclipse during interplanary flight);
- LEO orbit sun of 55,8 minutes
- Maximum Impactor Load of 800 W
- SAA from 0° to 30°;
- SA lifetime of 3.5 years;
- EOL power margin for critical phases
- One cell in short circuit (the interested string has Ns=Ns-1);
- One cell in open circuit (loss of one string);
- SA cell operative temperature : 100°C @1AU;
- SA cell maximum temperature : 130°C @0.68AU (with SAA ~60°);
- Cell packing density : ~0.85
- String Blocking diode : 1V;
- SA to PCDU harness losses : 0.5V.

The solar cell used for NEO Impactor are the same as per Orbiter, it is the triple junction cells (GAGET 2/160-8040) .

In the following table is reported the power generated by SA during different mission phases. The 3 Body panels are connected to 9 Solar Array Regulators (SAR). The 9 SAR are arranged in 3 groups of 3 SAR in parallel. Two of the three SAR are sufficient for the regulation of two sections; the third one is for redundancy. In this way all the power regulators groups have in input the same amount of power and this configuration form four single point failure free.

The following figure shows the power generation capability of SA with a surface of 5 m^2 equipped by Triple junction GaAs in different mission wc conditions:

	IMPACTOR SATELLITE MODES		
	Initial Sun Acquisition	Cruise	Final Target
Orbit Attitude	1AU	1.39AU	0,7AU
SAA	30°	10°	60°
SA POWER 1 string in failure [W]	858	523	758

Power generated by SA in the different mission phases

5.1.1.2 Impactor Power Control and Distribution

The PCDU design is common (more spare outlets are implemented with respect to Orbiter PCDU).

The power conditioning, control, protection and distribution to spacecraft equipment and payload instruments, battery management, heaters control and power pulse generation for solar array deployment are performed by the Power Control and Distribution Unit (PCDU).

The PCDU receives power from the body mounted and deployable SA panels and/or the battery and provide:

- regulation of the electrical power generated by the SA;
- regulation of the energy stored in the battery when required;
- battery charge/discharge control;
- control, monitoring and health management of the EPS;
- switching to distribute power to the scientific instruments and spacecraft equipment;
- switching to distribute power to the heaters, thermal cutters and to pyros;
- switching to distribute power to electrical propulsion;
- protection from external faults and prevention of failure propagation;
- interfacing the system data bus and exchanging TC/TM ;
- Interfacing the EGSE for AIV and Launch support.

5.1.1.2.1 Power Conditioning and Control

For NEO Impactor EPS a **regulated power bus** has been implemented taking into account the following considerations:

- There is no significant difference in term of electrical efficiency between the a regulated and a unregulated Bus solutions.
- Optimisation of Battery Design
- Users simplified interfaces for off-the-shelf equipment
- Better EMC performances (small voltage transient during users switch on/off)
- The unregulated option has a slight PCDU mass benefit (some Kg)
- Use off-the-shelf equipment for unregulated bus is not always guaranteed. Delta qualification, cost and schedule risk must be taken into account.
- For the Payload a stable power supply is provided w/o requiring of DC/DC converter at unit level
- Degradation of the Battery performance with an unregulated bus is more
- Power Bus voltage is dependent of SoC level of the Battery

The **regulated voltage is +28.14 V ±0.5%** at the regulation point, which is located at the Bus Capacitor.

The Power Supply Subsystem management is ensured by a conventional three-domain control system.

In the PCDU is implemented the power regulation based on **MPPT** regulation approach.

The MPPT has been chosen for the following major point:

- its flexibility to different power generation conditions (e.g. different solar flux);
- SA mass and dimension optimization;
- flexibility design in term of user needs;
- one failure tolerant design (no system degradation if 1 MPPT fail);
- modules design commonality with Rosetta and Mars Express.

For these reasons the MPPT with respect to S3R/S4R appears suitable for NEO scenario. The main bus regulation is performed by a conventional three-domain control system, based on one common, reliable Main Error Amplifier (MEA) signal.

A bank of capacitor is distributed along the main bus bar to ensure a low impedance mask for all connected functions. The design ensures compliance of the main bus with the voltage regulation, power quality and impedance requirements of ECSS-E-20A. When the available array power exceeds the total power demand from the PCDU, including the battery recharge power, the Array Power Regulator (APR) will perform the main bus regulation based on the MEA control line signal. The regulation function is a buck type switched regulator, which will leave the surplus energy on the array by increasing its input impedance. An MPPT function will automatically take over the regulation control of the regulator when the MEA signal enters the BCR or BDR control domain. The MPPT monitors the array voltage and current and controls the regulator to provide that specific input impedance, which will derive the maximum electrical power available on the array. The MPPT functions find the maximum power point by oscillating the APR input impedance slightly around the impedance providing the maximum power. The APR function comprises 9 individual Array Power Regulators, configured as 3 sets of 3 APRs with 4 hot redundant regulators. The active regulators share equally the requested power transfer to the main bus.

5.1.1.2.2 Power Management

Power management is supported by an adequate measurement of the power parameters within the PCDU. This includes:

- array current and voltage
- BCDR output current
- battery charge and discharge currents
- battery voltage
- LCL/FLC output current
- LCL status
- Heater line output current and status
- PCDU housekeeping data
- Main Error Amplifier (MEA) voltage.

When the SA power is no sufficient to satisfy the bus power consumption (i.e. during the launch phase and during LEO eclipse), the batteries via two BDRs in hot redundancy delivers the power. The function of the MEA is to manage the available energy sources, in order to guarantee a regulated bus. This circuit is designed to be one failure tolerant. Its reliability is obtained by using three MEA channels in hot redundancy followed by majority voting. The MEA senses the bus voltage at the regulation point and pilot the SAR sections and the BDRs to ensure the bus regulation. The power management is ensured by a conventional three-domain control system. Battery Discharge is controlled by the BDRs, which are conventional voltage boost regulators with overcurrent protection. Battery Charge is controlled by the BCRs, which are conventional step down current. The transition between these modes is automatic and lead to negligible transient of the main bus.

The PCDU start automatically whenever there is sufficient SA power available. Priority is given first to regulate SA power and then use the battery. At start-up the PCDU sets the electrical power distribution configuration into a known deterministic and reproducible state. All automatic protections are reset at switch-ON or by a dedicated High priority (HP) command.

The power to "Electrical Propulsion" is provided via 5 (TBC) nominal plus 5 (TBC) redundant LCL class IV.

The PCDU is able to manage and distribute up to 800W-output power.

5.1.1.2.3 Battery Management

The Battery Charge and Discharge Regulator (BCDR) is in charge to monitor the battery voltage, maintain the operating conditions and manage the charge/discharge. There are 2 BCDR modules (including redundancy) so that loss of one BCDR still allows the remaining module to satisfy the power needs.

The battery is charged by the BCR at constant regulated current, until a selected voltage limit is reached. The BCR will then maintain the battery at this voltage level (DC Taper Charging). The BCR is able to charge the battery with a rate from1 to 10A.

The BCDR is also in charge to prevent un-authorised discharging of the battery.

<u>CHARGE MANAGEMENT</u>

An automatic End of Charge (EOC) control function maintains the battery at the End of Charge voltage (EOC).

The EOC control function is single failure tolerant and maintain the battery within its V_{EOCMAX} (25.2V ±20mV).

In nominal case the battery is charged at 3.6 A ± 5% up to EOC voltage. After reach the EOC voltage the battery is maintained in DC Taper Charging.

In case of emergency the battery can be charged at 9A.

<u>DISCHARGE MANAGEMENT</u>

The BDR inhibits battery discharge operation if the battery voltage falls below V_{EODMAX} (16V ±20mV).

In the event that a faulty BDR is not inhibited and continues to discharge the battery, the PCDU shall isolate the battery if the battery voltage reduces below V_{EODMIN} (15V ±20mV).

Following either EOD events, the BDRs shall remain OFF until the battery SoC has been increased to a level equivalent to $V_{EODRESET}$ (under charge mode not discharge mode).

After EOD reset, the PCDU will automatically reconnect the BDRs and permit battery discharging.

It is possible to override each automatic discharge inhibits by HP command. The battery discharge current for each BDR is in the range from 0 to 12A.

5.1.1.2.4 DNEL and UVP

Two reliable under-voltage bus detection signals (Disconnect Not Essential Load (DNEL) and Under Voltage Protection (UVP)) are generated inside the PCDU if the bus voltage drops below a predefined limit. All not essential loads are connected to the high-level threshold DNEL signal. Heaters are considered as not essential loads. When DNEL signal is reached for a time more than 100us (TBC), all the concerned LCLs and HSGs protections are switched-OFF. All essential loads are connected to the low-level threshold UVP signal. When UVP signal is reached for a time more than 100us (TBC), all LCLs and HSGs protection are switched-OFF. Once LCLs and HSGs have been switched-OFF by UVP signal they can only switched-ON if commanded and if the bus voltage is above DNEL threshold. Note that CDMU, RX XPND, TWTA equipment are powered by FCL and they stay always ON at least above 20V bus voltage.

5.1.1.2.5 Power Protection and Distribution

The power protection and distribution policy is based on a centralised scheme. Each PCDU output power line is switched/protected by means of:

- Latch Current Limiter (LCL)
- Protected LCL
- Fold-back Current Limiter (FCL)
- Heater Switch Group (HSG)
- Thermal cutter and PYRO.

Each type of protection device shall prevent disruption of normal power bus operation if the device is commanded ON or OFF (even in the presence of a direct short-circuit at the output) or if an external fault occurs in any power distribution output. All LCLs, protected LCLs, HSG and NEA are controlled via the 1553 data bus system.

5.1.1.2.5.1 LCL

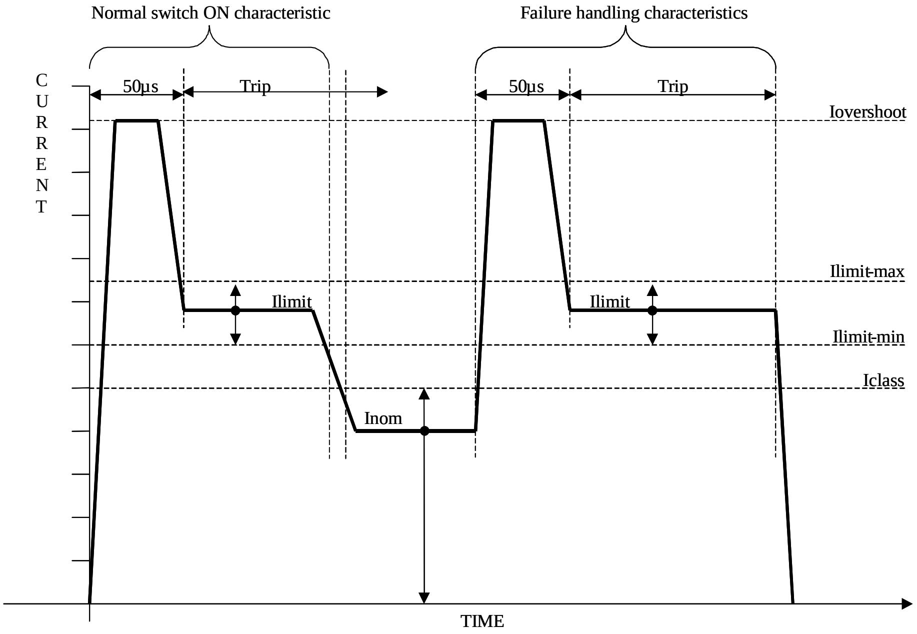
The function of the LCL switch is to act as a telecommand-operated switch with a limited current source capability. The switch is automatically tripped OFF in case the current limitation period exceeds a maximum time limit, or the bus voltage drops below one of the predefined limits Disconnect Non Essential Load (DNEL) or Under Voltage Protection (UVP). Three LCLs classes are defined on NEO to cope with a wide range of nominal current while ensuring an efficient protection. All LCLs types of protection device shall be in the OFF state at start-up and shall be reset when commanded OFF. The voltage drop across any protection device in the ON state is less than 0.28 Volts. The LCL is designed in accordance with the following requirements reported in following figure. If I_{LIMIT} is maintained for a period equal to T_{TRIP}, the LCL shall latch-OFF.Moreover, LCLs shall be switch-OFF and shall stay OFF until programmed back ON if the bus voltage falls below one of the two voltage thresholds:

- at 25.5±0.5 volts all the LCLs identified as "DNEL" (Disconnect Not Essential Load);
- at 21.5±0.5 volts ALL the LCLs shall be switch OFF "UVP" (Under Voltage Protection).

In particular, the DNEL and UVP shall not operate prematurely. A reliable alarm interface Telemetry signal will indicate if a Bus "DNEL" or "UVP" has switched OFF the LCL. It is possible to determine the status of each LCL via the 1553 data bus, including ON/OFF condition and output current (accuracy ±5%).

5.1.1.2.5.2 FCL

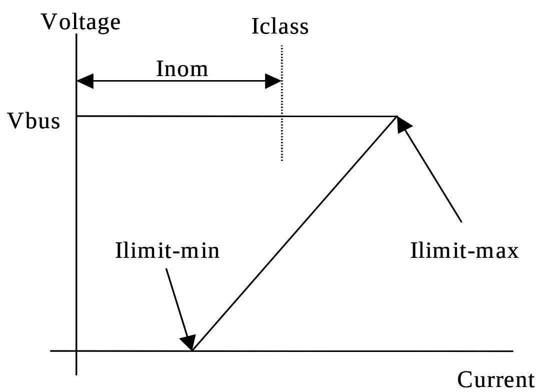
The function of the FCL is to act as a limited current source when main bus voltage is applied to the PCDU. In case of overload, these current limners will enter in a fold-back mode and the voltage will decrease. When the output voltage reaches 0V, the current load is = I_{SC}. The minimum FCL short circuit current (I_{LIMIT} min) is reported in the following figure. If the fault clears, the FCL circuit will return immediately back to normal operation.



LCL Characteristics

LCL Type	I_{class}	$Ilimit_{min}$=1.2xIclass	$Ilimit_{max}$=1.5xIclass	$I_{overshoot}$	$T_{tripmin}$	T_{tripma}x
Class I	1A	1.2A	1.5A	2.25A	10ms	12ms
Class II	2.5A	3A	3.75A	5.63A	10ms	12ms
Class III	5A	6A	7.5A	11.25A	10ms	12ms
Class IV	10A	12A	15A	22.25A	10ms	12ms

LCL classes summary



FCL Current-Voltage Characteristics

Type	I_{CLASS}	$I_{LIMITmin}$	$I_{LIMITmax}$	$I_{OVERSHOOT}$
FCL	1A	0.25A	1.5A	2.25A
FCL	2.5A	3A	3.75A	5.63A

FCL I_{LIMIT} Characteristics

The FCL is automatically switched-ON at bus start-up and after a bus recovery. The role of the FCL is to distribute the Main Power Bus to the essential loads in a protected, continuos and safe way. FCLs have the same constraints and requirements as the standard LCL except for the following:

- The FCL is NOT switched OFF until the bus voltage is less or equal to 18V;
- The FCL is NOT re-switched ON until the bus voltage is equal to 20V;
- It is NOT possible to switch/latch OFF an FCL;

In the following figure is reported the FCL characteristic. The FCLs I_{LIMIT} characteristics are reported in the following table:

It is possible to determine the output voltage and output current of each FCL via the 1553 data bus.

Six FCLs nominal and six FCLs redundant provides power to nominal and redundant TT&C and CDMU equipment and Battery heaters.

5.1.1.2.5.3 HSG

The Heater Switching Groups (HSGs) are in charge to control, protect and distribute power to the SVM and P/L heaters. One HSG protection plus six Heater Switching (HSW) compose each HSG. The HSG protection has the same characteristics as a standard LCL except for the following parameters:

- have a continuous current rating, "class IV", of 10 A;
- have a maximum trip current "I_{TRIP}" threshold of 12 to 15 A;
- If I_{LIMIT} is maintained for a period of more than 3ms the HSG protection is latched OFF.

Each HSW can be switched ON and OFF individually under the control of the 1553 bus. The HSG provide the status of each HSW and total output current via the 1553 data bus. Each HSW is able to manage up to 3.5 A. At PCDU start up, all HSWs are switched OFF and will stay in OFF until receiving an ON command. Since the Heaters are defined as NON-ESSENTIAL loads, after a bus voltage falls "DNEL" threshold (25.5 ±0.5V) the HSG protection will be switch-OFF and shall stay OFF state until receiving ON command. The HSW are controlled by the CDMU.

5.1.1.2.5.4 Thermal cutter and PYRO

The power needed for the SA deployment is given by a dedicated section of the PCDU. The thermal cutter and the pyro are fully redundant set at both actuation and initiator level. The PCDU provide prime and redundant drivers for thermal cutter and pyro. The driver can be activated by an ARM and then FIRE commands. The SA deployment function is enabled by a select CDMU command which power the current limiter used for control the thermal cutter. CDMU ARM commands allow the generation of the command pulse of any line. Finally, the CDMU HP FIRE command triggers the command pulse generation.

The PCDU provide 5 + 5 thermal cutter (nominal + redundant) drivers plus 5 (TBC) + 5 (TBC) Pyro (nominal +redundant) each one with the following characteristics:

- Actuation Circuit: 5A @28V;
- Actuation Time: 30ms (TBC).

The drivers are able to withstand a permanent short circuit of the output terminals. If the bus voltage falls 25.5 ±0.5V "DNEL", the NEA procedure cannot start. An external ARMING PLUG, as part of a safety procedure at AIV level, isolates the NEAs.

5.1.1.2.5.5 UMBILICAL EGSE and SAFE ARM I/F

The power to the NEO satellite before S/C lift-off will be provided via PCDU solar array umbilical input. During the integration phases the power to the NEO satellite will be provided via PCDU body and deployable solar array EGSE input. A safe and arm battery jumper connector is foreseen between the battery and the PCDU. This connector will be removed during on-ground NEO integration. The connector will be installed before flight and during NEO test.

5.1.1.3 Impactor Power budget

The equipment power consumption reported in the NEO power Budget depends on the maturity status of the equipment.

The following equipment contingency has been considered in relation to its level of development:

- 5% for the off-the-shelf equipment (e.g. Battery, XPND, TWTA, AOCS equipment etc);
- 10% for the item to be modified (e.g. CDMU, PCDU etc);
- 15% for the item to be developed.

A system margin of 10% has been applied to the NEO Impactor satellite to take into account the uncertainties of the model used to determinate the power budget at system level.

In addition 1 string has been subtracted to SA and Battery total string to take into account the potential failure.

In order to verify the necessary power margin for all possible orbits relevant to the Impactor mission, an analysis tool has been refined using an Excel spreadsheet.

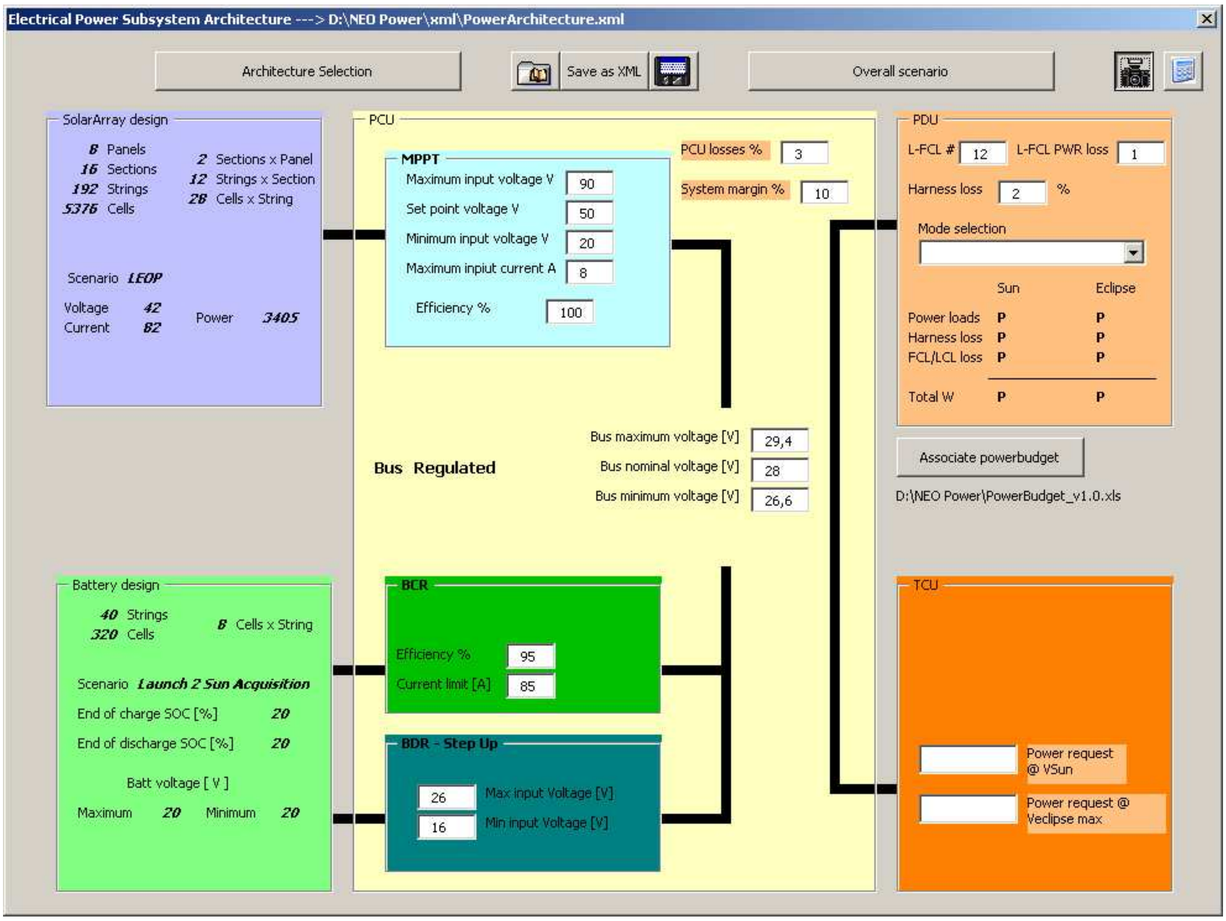
For using this tool it is necessary to define the boundary condition (i.e. Orbit parameters, Solar Panel Sun Aspect Angle, Solar Panel Temperature, Sun Distance, Failure, Life and Degradation) and to select an EPS architecture (i.e. power bus regulated, BCR/BDC DC/DC step-up, SA MPPT regulator)

The Excel tool developed by AAS-I has been used to:

- Sizing the Battery in term of number of cells (series and parallel) to supply, during the eclipse phase, the Impactor electrical load maintaining, in worst conditions the bus voltage within the range 28 Vdc +/- 0,5 % along the orbit or phase
- Sizing the Solar Array in term of number of cells (series and parallel) to supply, during the sun phase, the Impactor electrical load and also the recharging of the Battery maintaining, in worst conditions the bus voltage within the range 28 Vdc +/- 0,5 % along the orbit or phase
- Determination along the orbit or operational phase the State of Charge (SoC) of the Battery
- Determination of the power required to SA for recharging of the Battery during the charge phase (sun phase)
- Determination of the power dissipation of the PCDU due to the power regulation, conversion and distribution
- Determination of the mass of the main part of the EPS : SA, BATT and PCDU
- Determination of the Power dissipation of the EPS BATT and PCDU

- Determination of the power margin for each operation phase and mode to determinate the boundary condition needed to define the suitable nominal operation.

In the following figure the main window of the excel spreadsheet tool used to perform the power budget and calculate the power margins is shown.


Electrical Power Subsystem Architecture ---> D:\NEO Power\xml\PowerArchitecture.xml
Architecture Selection
Save as XML
Overall scenario
SolarArray design
8 Panels
16 Sections
192 Strings
5376 Cells
2 Sections x Panel
12 Strings x Section
28 Cells x String
Scenario LEOP
Voltage 42
Current 82
Power 3405
PCU
MPPT
Maximum input voltage V 90
Set point voltage V 50
Minimum input voltage V 20
Maximum inpiut current A 8
Efficiency % 100
PCU losses % 3
System margin % 10
Bus Regulated
Bus maximum voltage [V] 29,4
Bus nominal voltage [V] 28
Bus minimum voltage [V] 26,6
BCR
Efficiency % 95
Current limit [A] 85
BDR - Step Up
26 Max input Voltage [V]
16 Min input Voltage [V]
Battery design
40 Strings
320 Cells
8 Cells x String
Scenario Launch 2 Sun Acquisition
End of charge SOC [%] 20
End of discharge SOC [%] 20
Batt voltage [V]
Maximum 20 Minimum 20
PDU
L-FCL # 12 L-FCL PWR loss 1
Harness loss 2 %
Mode selection
Sun Eclipse
Power loads P P
Harness loss P P
FCL/LCL loss P P
Total W P P
Associate powerbudget
D:\NEO Power\PowerBudget_v1.0.xls
TCU
Power request @ VSun
Power request @ Veclipse max

The power margin values have been calculated to provide an overview of the Impactor power situation for each case as result from the simulation case:

- Negative value means that the EPS can not be able to provide sufficient power/energy to supply the specified loads (SoC of the battery at the beginning of the orbit is higher than the SoC at the end). This means that the power load demand needs to be reduced by this amount in order to get stable condition.
- Positive value means that the EPS is able to provide sufficient power/energy to supply the loads Power Load demand (SoC of the battery at the beginning of the orbit is equal to the SoC at the end). This means that the power load demand can be increased by this amount in order to get stable condition.
- Zero values means that the EPS is able to provide the sufficient power/energy to supply the loads Power Load demand (SoC of the battery at the beginning of the orbit is equal to the SoC at the end). This means that the power load demand can not be increased at all.

To provide the power budget of the Impactor the following main assumptions (very conservative) have been considered:

For SA:

- For LEOP phase only solar flux (1375 W/m2@ 1 AU) has been considered to calculate the SA power generation. No contribution of Earth Albedo factor has been considered.
- For the other phases/modes (Cruise, Final Target) the determination of the solar flux has been based on the formula :

  Solar Flux @ Distance = Solar Flux @ 1AU/Distance^2

- For the SA cells an End of Life (EoL) degradation of 1e+15 (Mev)
- A failure on one string
- The maximum temperature of 100°C @1AU and 10°C @2,7AU
- The fill factor is 0,85

For Battery:

- The temperature constant of 40°C
- The cells degradation (EoL) of 0,98 considering 100 cycles of charge/discharge
- A failure on one string
- The maximum DoD of 88% at the beginning of LEOP phase (1 time) considering that this phase start with a maximum duration of eclipse (36,6 minutes) and minimum duration of the sun phase (55,8 minutes)
- A initial SoC of 98% has been considered taking into account the initial capacity degradation due to the storage and pre-launch test.

For PCDU:

- It is assumed a power consumption of 49,5 W (average) in eclipse phase due to the power consumption of the internal power conversion and internal electronic.
- It is assumed a power consumption of 110 W (average) in sun phase due to the power consumption of the internal power conversion, internal electronic and MPPT control electronic power efficiency.
- Power distribution Loss of 3% due to the LCL/FCL power loss that is function of the power requests.
- Harness losses of 3% due to the power dissipation along the harness from SA to PCDU and between PCDU and Loads.

For Loads:

- The power consumption of electronic unit has been considered taking into account the average figure with a dedicated uncertatains figure depending of the maturity of the design
- The power consumption of the thermal control has been considered taking into account the figure coming from the thermal analysis
- The power consumption of the AOCS Thrusters has been considered in the power budget taking into account the following data and assumption:

Power consumption of 22N thrust				
Items	**Impactor Modes**			
	LEOP	Cruise	Correction Manouvre	Final Target
Valve (W)	15	15	15	15
Heater (W)	9,5	9,5	9,5	9,5
N° of active thrust	4	2	2	2
Duty-cycle	0,6	0,4	0,9	0,9
Contingency	**5%**			
Total Power (W)	**61,7**	**20,5**	**46,3**	**46,3**

Power consumption of 10N thrust				
Items	**Impactor Modes**			
	LEOP	Cruise	Correction Manouvre	Final Target
Valve (W)	15	15	15	15
Heater (W)	9,5	9,5	9,5	9,5
N° of active thrust	0	2	2	2
Duty-cycle	0	0,9	0,9	0,9
Contingency	**5%**			
Total Power (W)	**0**	**46,3**	**46,3**	**46,3**

- The ABM has been used during the LEOP. In the power budget a power consumption of **38,8 W** has been taking into account considering a contingency of 5%..

- Harness losses of 3% due to the power dissipation along the harness from SA to PCDU and between PCDU and Loads.

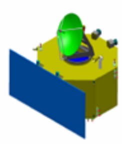
A system margin of 10 % is added on estimated load to cover any uncertainties of the analysis performed with the Excel spreadsheet tool.

The following NEO satellite modes have been considered to determine the electrical Load to be included in the power budget for the Impactor:

- **Pre-Launch Mode;**
- **Launch Mode**
- **Sun Acquisition Mode**
- **LEOP Mode**
- **Cruise Mode**
- **Correction Manouvre**
- **Final Target Mode**
- **Safe Mode**

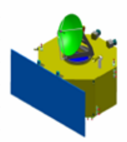
Pre-Launch Mode : This mode start 30 minutes before the Launch phase. During this mode the power to the Loads is provided by the Battery via BDR.The following table reports the relevant estimated power consumption in Pre-Launch mode of the Impactor equipments to be considered in the power budget.Power Consumption of Impactor for Pre-Launch Mode:

IMPACTOR	Date : 27/09/2006				
Subsystem : TT&C	IMPACTOR : Launch				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/C Modes**
RF band section					
XPND1 RX	1	14	10	15,4	Launch
XPND1 TX -X Band	1	0	10	0	
				0	
XPND2 RX	1	14	10	15,4	
XPND2 TX	1	0	10	0	
				0	
EPC+ TWT1 (X band)	1	8,8	5	9,24	
				0	
EPC+ TWT2 (X Band)	1	0	5	0	
UHF band section					
XPND1 UHF RX	1	0	5	0	
XPND1 UHF TX	1	0	5	0	
XPND2 UHF RX	1	0	5	0	
XPND2 UHF TX	1	0	5	0	
Total				**40,0**	
Subsystem : EPS	IMPACTOR : Launch				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/C Modes**
PCDU Electronic	1	23	10	25,3	Launch
Total				**25,3**	
Subsystem : C&DH	IMPACTOR : Launch				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
CDMU	1	*32*	10	35,2	Launch
Total				**35,2**	
Subsystem : CPM Thrusts	ORBITER : LEOP				
Units	**Thrust level [N]**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
22 N Thruster	22	*32*	5	33,6	4 thrust active at each time
Total				**0**	
Subsystem : TCS	IMPACTOR : Launch				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
P/L Heaters	1	*0,00*	10	0	Launch
S/C Heaters	1	*0,00*	10	0	
Total				**0**	
			Total for S/C in Launch	**100,5**	
P/L : DQ+	IMPACTOR : Launch				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
none	0	*0,00*	15	0	Launch
Total				**0**	
			Total Load for P/L in Launch	**0**	
			Total Load for ORBITER [W]	**100,5**	Launch
Harness Losses		**3%**		**3,0**	
PCDU power Losses		**3%**		**3,1**	
System Margins		**10%**		**10,7**	
			Total Load for ORBITER with loss and margins [W]	**117,3**	



Launch Mode : This mode starts at launch (vehicle lift-off) with a duration of 30 minutes. During this mode the power to the Loads is provided by the Battery via BDR.The following table reports the relevant estimated power consumption in Launch mode of the Impactor equipments to be considered in the power budget.Power Consumption of Impactor for Launch Mode:

IMPACTOR	Date : 27/09/2006				
Subsystem : TT&C	IMPACTOR : Launch				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Modes**
RF band section					
XPND1 RX	1	14	10	15,4	Launch
XPND1 TX -X Band	1	0	10	0	
				0	
XPND2 RX	1	14	10	15,4	
XPND2 TX	1	0	10	0	
				0	
EPC+ TWT1 (X band)	1	8,8	5	9,24	
EPC+ TWT2 (X Band)	1	0	5	0	
UHF band section					
XPND1 UHF RX	1	0	5	0	
XPND1 UHF TX	1	0	5	0	
XPND2 UHF RX	1	0	5	0	
XPND2 UHF TX	1	0	5	0	
Total				**40,0**	
Subsystem : EPS	IMPACTOR : Launch				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Modes**
PCDU Electronic	1	45	10	49,5	Launch
Total				**49,5**	
Subsystem : C&DH	IMPACTOR : Launch				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
CDMU	1	35	10	38,5	Launch
Total				**38,5**	
Subsystem : CPM Thrusts	ORBITER : LEOP				
Units	**Thrust level [N]**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
22 N Thruster	22	32	5	33,6	4 thrust active at each time
Total				**0**	
Subsystem : TCS	IMPACTOR : Launch				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
P/L Heaters	1	0,00	10	0	
S/C Heaters	1	0,00	10	0	Launch
Total				**0**	
Total for S/C in Launch				**128,0**	
P/L : DQ+	IMPACTOR : Launch				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
Total				**0**	
Total for P/L in Launch				**0**	
Total Load for ORBITER [W]				**128,0**	Launch
Harness Losses	3%			3,8	
PCDU power Losses	3%			4,0	
System Margins	10%			13,6	
Total for IMPACTOR in Launch				**149,4**	



Sun Acquisition Mode : This mode starts after the Launch up to the Orbiter acquire the Sun in stable condition (Sun Acquisition). It has been assumed to take about 120 minutes. During this mode the power to the Loads has been assumed to be done by the Battery via BDR. This is a worst case condition because no contribution of the sun has been considered after the rate dumping phase where the Solar Array is able to catch partially the Sun allowing a partial battery recharging. Power to Loads is provided by Solar Array and the Battery during eclipse. The battery is recharged mode during sun and it is discharge mode during eclipse. The following table reports the relevant estimated power consumption in Sun Acquisition mode of the Impactor equipments to be considered in the power budget.

Power Consumption of Impactor in Sun Acquisition mode:

IMPACTOR	Date : 27/09/2006				
Subsystem : TT&C	IMPACTOR Mode : Sun Acquisition				
Units	Number	Power consumpt.[W]	Contingency [%]	Power with cont. [W]	S/C Modes
RF band section					
XPND1 RX	1	14	10	15,4	Safe
XPND1 TX -X Band	1	6	10	6,6	
				0	
XPND2 RX	1	14	10	15,4	
XPND2 TX	1	0	10	0	
				0	
EPC+ TWT1 (X band)	1	128	5	134,4	
				0	
EPC+ TWT2 (X Band)	1	0	5	0	
UHF band section					
XPND1 UHF RX	1	0	5	0	
XPND1 UHF TX	1	0	5	0	
XPND2 UHF RX	1	0	5	0	
XPND2 UHF TX	1	0	5	0	
Total				171,8	

Subsystem : EPS	IMPACTOR Mode : Sun Acquisition				
Units	Number	Power consumpt.[W]	Contingency [%]	Power with cont. [W]	S/C Modes
PCDU Electronic	1	45	10	49,5	Nominal
Total				49,5	

Subsystem : AOCS sensors	ORBITER Mode : Sun Acquisition				
Units	Number	Power consumpt.[W]	Contingency [%]	Power with cont. [W]	S/S Modes
GYR1	**main**	7,5	5	7,9	SUN Acquisition
GYR2	cold	0	5	0,0	
RW1	**main**	0	5	0,0	
RW2	**main**	0	5	0,0	
RW3	**main**	0	5	0,0	
RW4	cold	0	5	0,0	
STR1	**main**	0	5	0,0	
STR2	cold	0	5	0,0	
NAVCAM 1	**main**	0	5	0,0	
NAVCAM 2	cold	0	5	0,0	
Total				7,9	

Subsystem : C&DH	IMPACTOR Mode : Sun Acquisition				
Units	Number	Power consumpt.[W]	Contingency [%]	Power with cont. [W]	Mode
CDMU	1	35	10	38,5	Nominal
Total				38,5	

Subsystem : AOCS actuators	IMPACTOR Mode : Sun Acquisition				
Units	Thrust level [N]	Power consumpt.[W]	Contingency [%]	Power with cont. [W]	Mode
10 N Thruster	10	98	10	107,8	Sun Acquisition
Total				26,95	

Subsystem : CPM Thrusts	ORBITER : LEOP				
Units	Thrust level [N]	Power consumpt.[W]	Contingency [%]	Power with cont. [W]	Mode
22 N Thruster	22	0	5	0	4 thrust active at each time
Total				0	

Subsystem : TCS	IMPACTOR Mode : Sun Acquisition				
Units	Number	Power [W]	Contingency [%]	Power with cont. [W]	Mode
S/C Heaters	1	0,00	10	0	Survival
P/L Heaters	1	0,00	10	0	
Total				0	
			Total for S/C in Sun Acquisition	286,8	

P/L : DQ+	IMPACTOR Mode : Sun Acquisition				
Units	Number	Power [W]	Contingency [%]	Power with cont. [W]	Mode
Total				0	Off
			Total for P/L in Sun Acquisition	0	
			Total Load for ORBITER [W]	286,8	Sun Acquisition

Harness Losses	3%	8,6	
PCDU power Losses	3%	8,9	
System Margins	10%	30,4	
	Total for IMPACTOR in Sun Acquisition	334,6	

LEOP Mode : This mode start with a specific command. During this mode the power to the Loads has been assumed to be done by the SA during in sun period (55,8 minutes) and by the Battery in eclipse phase (36,6 minutes) via BDR. In the power budget it is assumed that this mode start with an eclipse. This is a worst case condition because no contribution of the sun has been considered after the Sun acquisition The battery is recharged mode during sun and it is discharge mode during eclipse. The following table reports the relevant estimated power consumption in LEOP mode of the Impactor equipments to be considered in the power budget.

Power Consumption of Impactor for LEOP case

IMPACTOR	Date : 27/09/2006				
Subsystem : TT&C	IMPACTOR : LEOP				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/C Modes**
RF band section					
XPND1 RX	1	14	10	15,4	safe
XPND1 TX -X Band		0	10	0	
XPND2 RX	1	14	10	15,4	
XPND2 TX	1	0	10	0	
EPC+ TWT1 (X band)	1	128	5	134,4	
EPC+ TWT2 (X Band)	1	0	5	0	
UHF band section					
XPND1 UHF RX	1	0	5	0	
XPND1 UHF TX	1	0	5	0	
XPND2 UHF RX	1	0	5	0	
XPND2 UHF TX	1	0	5	0	
Total				**165,2**	

Subsystem : EPS	IMPACTOR : LEOP				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	
PCDU Electronic	1	45	10	49,5	IN ECLIPSE
		100	10	110	IN SUN
Total					

Subsystem : AOCS sensors	ORBITER : Sun Acquisition				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/S Modes**
GYR1	**main**	7,5	5	7,9	SUN Acquisition
GYR2	cold	*0*	5	0,0	
RW1	**main**	*0*	5	0,0	
RW2	**main**	*0*	5	0,0	
RW3	**main**	*0*	5	0,0	
RW4	cold	*0*	5	0,0	
STR1	**main**	*7,3*	5	7,7	
STR2	cold	*7,3*	5	7,7	
NAVCAM 1	**main**	*0*	5	0,0	
NAVCAM 2	cold	*0*	5	0,0	
Total				**23,2**	

Subsystem : C&DH	IMPACTOR : LEOP				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
CDMU	1	*35*	10	38,5	LEOP
Total				**38,5**	

Subsystem : AOCS Thrusts	IMPACTOR : LEOP				
Total	**Thrust level [N]**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
ABM		37	5	38,85	LEOP
10 N Thruster	10	*0*	5	0	
Total				**38,85**	

Subsystem : CPM Thrusts	IMPACTOR : LEOP				
Total	**Thrust level [N]**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
22 N Thruster	22	*32*	5	33,6	4 thrust active at each time
Total				**20,16**	

Subsystem : TCS	ORBITER : LEOP				
Units	**Power [W]**		**Contingency [%]**	**Power with cont. [W]**	
	IN SUN	**IN ECLIPSE**		**IN SUN**	**IN ECLIPSE**
S/C Heaters	108	*108,00*	10	118,8	118,8
P/L Heaters	0	*0,00*	10	0	0
Total				**118,8**	**118,8**
				IN SUN	IN ECLIPSE
			Total for S/C	**471,4**	**410,9**

P/L : DQ+	IMPACTOR : LEOP				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
none	1	*0,00*	15	0	Survival
Total				**0**	
			Total for P/L	**0**	**0**

			IN SUN	IN ECLIPSE
Harness Losses	3%		**14,1**	**12,3**
PCDU power Losses	3%		**14,6**	**12,7**
System Margins	10%		**50,0**	**43,6**
			IN SUN	IN ECLIPSE
		Total for IMPACTOR in LEOP	**550,1**	**479,5**

Cruise Mode : This mode starts with a specific command. Two different orbit conditions affecting the SA power generation capability have been considered taking into account the minimum and maximum distance of the Impactor to the Sun:

- Cruise case when the Sun distance of the Impactor is 0,7AU
- Cruise Cold case when the Sun distance of the Impactor is 1,39AU

Power to PCDU is provided by solar array. The battery is maintained in DC taper charge.

The following table reports the relevant estimated power consumption of the Impactor equipments in Cruise Cold conditions to be considered in the power budget calculation.

Power Consumption of Impactor for Cruise case:

IMPACTOR	Date : 27/09/2006				
Subsystem : TT&C	IMPACTOR : Cruise				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/C Modes**
RF band section					
XPND1 RX	1	14	10	15,4	Cruise
XPND1 TX -X Band	1	0	10	0	
XPND2 RX	1	14	10	15,4	
XPND2 TX	1	0	10	0	
EPC+ TWT1 (X band)	1	128	5	134,4	
EPC+ TWT2 (X Band)	1	0	5	0	
UHF band section					
XPND1 UHF RX	1	0	5	0	
XPND1 UHF TX	1	0	5	0	
XPND2 UHF RX	1	0	5	0	
XPND2 UHF TX	1	0	5	0	
Total				**165,2**	

Subsystem : EPS	IMPACTOR : Cruise				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/C Modes**
PCDU Electronic	1	100	10	110	nominal
Total				**110**	

Subsystem : AOCS sensors	ORBITER : Sun Acquisition				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/S Modes**
GYR1	**main**	0,0	5	0,0	SUN Acquisition
GYR2	cold	0,0	5	0,0	
RW1	**main**	**20,0**	5	21,0	
RW2	**main**	**20,0**	5	21,0	
RW3	**main**	**20,0**	5	21,0	
RW4	cold	0,0	5	0,0	
STR1	**main**	**7,3**	5	7,7	
STR2	cold	**7,3**	5	7,7	
NAVCAM 1	**main**	0,0	5	0,0	
NAVCAM 2	cold	0,0	5	0,0	
Total				**78,3**	

Subsystem : C&DH	IMPACTOR : Cruise				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
CDMU	1	*35*	10	38,5	nominal
Total				**38,5**	

Subsystem : AOCS Actuators	IMPACTOR : Cruise				
Units	**Thrust level [mN]**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [kW]**	**Mode**
10 N Thruster	10,0	*49*	5	51,45	Cruise
Total				**46,3**	

Subsystem : CPM thrusters	IMPACTOR : Cruise				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
22 N Thruster	22	*16*	5	16,8	2 thrust active at each time
Total				**6,72**	

Subsystem : TCS	IMPACTOR : Cruise				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
P/L Heaters	1	*0,00*	10	0	Cruise hot case
S/C Heaters	1	*108,00*	10	118,8	
Total				**118,8**	
			Total for S/C in Cruise Case (1 AU)	**499,8**	

P/L : DQ+	IMPACTOR : Cruise				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
none	0	*0,00*	0	0	
Total				**0**	
			Total for P/L in Cruise Case (1 AU)	**0**	
			Total Load for ORBITER [W]	**499,8**	**Cruise Hot case**
Harness Losses	3%			**15,0**	
PCDU power Losses	3%			**15,4**	
System Margins	10%			**53,0**	
			Total for ORBITER in Cruise Hot Case (1 AU)	**583,3**	

Correction Manoeuvre Mode : This mode starts with a specific command. Power to PCDU is provided by solar array. The battery is maintained in DC taper charge. The following table reports the relevant estimated power consumption of the Impactor equipments in Correction Manoeuvre conditions to be considered in the power budget calculation.

<u>Power Consumption of Correction Manouvre case</u>:

IMPACTOR	Date : 27/09/2006				

Subsystem : TT&C	IMPACTOR : Correction Manouvre				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/C Modes**
RF band section					Cruise
XPND1 RX	1	14	10	15,4	
XPND1 TX -X Band	1	6	10	6,6	
XPND2 RX	1	14	10	15,4	
XPND2 TX	1	0	10	0	
EPC+ TWT1 (X band)	1	128	5	134,4	
EPC+ TWT2 (X Band)	1	0	5	0	
UHF band section					
XPND1 UHF RX	1	0	5	0	
XPND1 UHF TX	1	0	5	0	
XPND2 UHF RX	1	0	5	0	
XPND2 UHF TX	1	0	5	0	
Total				**171,8**	

Subsystem : EPS	IMPACTOR : Correction Manouvre				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/C Modes**
PCDU Electronic	1	100	10	110	Correction manouvre
Total				**110**	

Subsystem : AOCS sensors	ORBITER : Sun Acquisition				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/S Modes**
GYR1	**main**	0,0	5	0,0	SUN Acquisition
GYR2	cold	0,0	5	0,0	
RW1	**main**	**20,0**	5	21,0	
RW2	**main**	**20,0**	5	21,0	
RW3	**main**	**20,0**	5	21,0	
RW4	cold	0,0	5	0,0	
STR1	**main**	**7,3**	5	7,7	
STR2	cold	**7,3**	5	7,7	
NAVCAM 1	**main**	0,0	5	0,0	
NAVCAM 2	cold	0,0	5	0,0	
Total				**78,3**	

Subsystem : C&DH	IMPACTOR : Correction Manouvre				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
CDMU	1	*35*	10	38,5	Correction Manouvre
Total				**38,5**	

Subsystem : AOCS actuators	IMPACTOR : Correction Manouvre				
Units	**Thrust level [N]**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
10 N Thruster	10	*49*	10	53,9	Correction Manouvre
Total				**48,51**	

Subsystem : CPM thrusters	IMPACTOR : Cruise				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
22 N Thruster	22	*16*	5	16,8	Cruise hot case
Total				**15,12**	

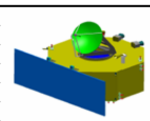
Subsystem : TCS	IMPACTOR : Correction Manouvre				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
P/L Heaters	1	*0,00*	10	0	Correction Manouvre
S/C Heaters	1	*108,00*	10	118,8	
Total				**118,8**	
			Total for S/C	**565,9**	

P/L : DQ+	IMPACTOR : Correction Manouvre				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
Total				**0**	
			Total for P/L	**0**	
			Total Load for ORBITER [W]	**565,9**	**Correction Manouvre**
Harness Losses	**3%**			**17,0**	
PCDU power Losses	**3%**			**17,5**	
System Margins	**10%**			**60,0**	
			Total for IMPACTOR	**660,4**	

Final Target Mode : This mode starts with a specific command. During this mode the power to PCDU is provided by solar array. The battery is maintained in DC taper charge. The following table reports the relevant estimated power consumption of the Impactor equipments in Final Target Mode to be considered in the power budget calculation.

Power Consumption of Impactor for Final Target Mode:

IMPACTOR	Date : 27/09/2006				
Subsystem : TT&C	IMPACTOR : Final Target				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/C Modes**
RF band section					Final Target
XPND1 RX	1	14	10	15,4	
XPND1 TX -X Band	1	6	10	6,6	
XPND2 RX	1	14	10	15,4	
XPND2 TX	1	0	10	0	
EPC+ TWT1 (X band)	1	128	5	134,4	
EPC+ TWT2 (X Band)	1	0	5	0	
UHF band section					
XPND1 UHF RX	1	18,5	5	19,4	
XPND1 UHF TX	1	42	5	44,1	
XPND2 UHF RX	1	0	5	0	
XPND2 UHF TX	1	0	5	0	
Total				**235,3**	
Subsystem : EPS	IMPACTOR : Final Target				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/C Modes**
PCDU Electronic	1	100	10	110	Nominal
Total				**110**	
Subsystem : AOCS sensors	IMPACTOR : Final Target				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/S Modes**
GYR1	**main**	**7,5**	5	7,9	SUN Acquisition
GYR2	cold	0,0	5	0,0	
RW1	**main**	**20,0**	5	21,0	
RW2	**main**	**20,0**	5	21,0	
RW3	**main**	**20,0**	5	21,0	
RW4	cold	0,0	5	0,0	
STR1	**main**	**7,3**	5	7,7	
STR2	cold	**7,3**	5	7,7	
NAVCAM 1	**main**	**7,0**	5	7,4	
NAVCAM 2	cold	**7,0**	5	7,4	
Total				**100,9**	
Subsystem : C&DH	IMPACTOR : Final Target				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
CDMU	1	*35*	10	38,5	Nominal
Total				**38,5**	
Subsystem : AOCS Actuators	IMPACTOR : Final Target				
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [kW]**	**Mode**
10 N Thruster	8	*49*	5	51,45	2 thruster active
Total				**46,3**	
Subsystem : CPM thrusters	IMPACTOR : Cruise				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
22 N Thruster	22	*16*	5	16,8	2 thruster active
Total				**15,12**	
Subsystem : TCS	IMPACTOR : Final Target				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
P/L Heaters	1	*0,00*	10	0,0	
S/C Heaters	1	*108,00*	10	118,8	
Total				**106,92**	
Total Load for S/C in Final target Case				**638,0**	
P/L : DQ+	IMPACTOR : Final Target				
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
NAC (Telecamera)	1	*0,75*	20	0,9	
Total				**0,9**	
Total Load for P/L in Final Target Case				**0,9**	
Total Load for ORBITER [W]				**638,9**	**FINAL Target**
Harness Losses	3%			**19,2**	
PCDU power Losses	3%			**19,7**	
System Margins	10%			**67,8**	
Total Load for IMPACTOR in Final Target case				**745,5**	



Safe Mode : This mode is initiated when a safeguard detecting a dangerous NEO condition is triggered. Power to PCDU is provided by solar array and battery if required. The following table reports the relevant estimated power consumption of the Impactor equipments in Safe Mode to be considered in the power budget calculation.

Power Consumption of Impactor for Safe Mode:

IMPACTOR	Date : 27/09/2006				
Subsystem : TT&C		IMPACTOR : Safe			
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Modes**
RF band section					
XPND1 RX	1	14	10	15,4	Safe
XPND1 TX -X Band	1	6	10	6,6	
XPND2 RX	1	14	10	15,4	
XPND2 TX	1	0	10	0	
EPC+ TWT1 (X band)	1	128	5	134,4	
EPC+ TWT2 (X Band)	1	0	5	0	
UHF band section					
XPND1 UHF RX	1	0	5	0	
XPND1 UHF TX	1	0	5	0	
XPND2 UHF RX	1	0	5	0	
XPND2 UHF TX	1	0	5	0	
Total				**171,8**	

Subsystem : EPS		IMPACTOR : Safe			
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Modes**
PCDU Electronic	1	100	10	110	Nominal
Total				**110**	

Subsystem : AOCS sensors		IMPACTOR : Safe			
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**S/S Modes**
GYR1	**main**	**7,5**	5	7,9	SUN Acquisition
GYR2	cold	0,0	5	0,0	
RW1	**main**	0,0	5	0,0	
RW2	**main**	0,0	5	0,0	
RW3	**main**	0,0	5	0,0	
RW4	cold	0,0	5	0,0	
STR1	**main**	0,0	5	0,0	
STR2	cold	0,0	5	0,0	
NAVCAM 1	**main**	0,0	5	0,0	
NAVCAM 2	cold	0,0	5	0,0	
Total				**7,9**	

Subsystem : C&DH		IMPACTOR : Safe			
Units	**Number**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
CDMU	1	*22*	10	24,2	Safe
Total				**24,2**	

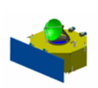
Subsystem : AOCS Actuators		IMPACTOR : Safe			
Units	**Thrust level [mN]**	**Power consumpt.[W]**	**Contingency [%]**	**Power with cont. [kW]**	**Mode**
1 N Thruster	10	*39,3*	10	43,23	Sun Acquisition
Total				**43,2**	

Subsystem : CPM thrusters		IMPACTOR : Cruise			
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
22 N Thruster	22	*32*	5	33,6	Cruise hot case
Total				**0**	

Subsystem : TCS		IMPACTOR : Safe			
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
P/L Heaters	1	*0,00*	10	0	Safe
S/C Heaters	1	*112,00*	10	123,2	
Total				**123,2**	
			Total for S/C in Cruise Case (1 AU)	**480,3**	

P/L : DQ+		IMPACTOR : Safe			
Units	**Number**	**Power [W]**	**Contingency [%]**	**Power with cont. [W]**	**Mode**
Total				0	
			Total for P/L in Cruise Case (1 AU)	**0**	
			Total Load for ORBITER [W]	**480,3**	**Safe**

Harness Losses	3%	**14,4**	
PCDU power Losses	3%	**14,8**	
System Margins	10%	**51,0**	
	Total for IMPACTOR in Cruise	**560,5**	



Hereafter are reported all the electrical characteristics of the SA and Battery cells used in the models included in the Excel spreadsheet.

Cell junction cells GAGET 2/160-8040 characteristic of NEO Impactor Solar Array

Design and Mechanical Data

Base Material	GaInP2/GaAs/Ge on Ge substrate
AR-coating	TiO_x/Al_2O_3
Dimensions	40 x 80mm ± 0.1mm
Cell Area	30.18 cm²
Average Weight	≤ 86 mg/cm²
Epi-Wafer Thickness	150 ± 20 µm
Ag - Thickness	4 – 10 µm
Grid Design	Grid system with 3 contact pads
Shadow Protection	Integral by pass diode protecting the adjacent cell $V_{reverse}$ (1.2 I_{SC}) ≤ 2.5 V; T= 25 °C ± 3°C

Electrical Data

		BOL	5E13	1E14	3E14	1E15
Average Open Circuit V_{OC}	[mV]	2575	0.950	0.942	0.927	0.904
Average Short Circuit I_{SC}	[mA/cm²]	16.9	0.999	0.999	0.997	0.989
Voltage at max. Power V_{pmax}	[mV]	2275	0.962	0.953	0.935	0.908
Current at max. Power I_{pmax}	[mA/cm²]	15.95	1.000	1.000	0.998	0.976
Maximal Power P_{max}	[mW/cm²]	36.29	0.962	0.953	0.933	0.886
Average Efficiency η_{bare}	[%]	26.8	0.962	0.953	0.933	0.886

Test Conditions: AM0 Spectrum ; Light Intensity E = 135.3 mW/cm²; Cell Temperature T_C = 28°C
Standard: SPL 2002-10-08 (set of 3 isotype cells and 1 triple cell)

Temperature Gradients

Voltage dV_{oc}/dT	[mV/°C]	- 6.0	- 6.0	- 6.3	- 6.5	- 6.6
Short Circuit dI_{sc}/dT	[µA/cm²/C°]	0.009	0.010	0.008	0.009	0.010
Voltage dV_{pmax}/dT	[mV/°C]	-6.4	- 6.5	- 6.8	- 6.8	- 7.0
Current dI_{mpp}/dT	mA/cm²/°C]	0.004	0.007	0.007	0.007	0.008

Threshold Values

Absorptivity	≤ 0,91 (with CMX 100 AR)
Pulltest	> 1.6 N at 45 ° welding test (with 12.5 µm Ag stripes)
Development Status	Qualified

SONY 18650HC cells characteristics of NEO Impactor Battery

- ***18 mm diameter by 65 mm high***
- ***Mass 40 grams***
- ***1.5 Ah total capacity***
- ***5.4 Wh nameplate energy***
- ***Cell energy density 133 Wh/kg***
- ***4.2 V end of charge limit***
- ***2.5 V end of discharge limit***

The calculation of the power margins have been performed to :

- o Size the Battery and Sola Array in term of number of cells (in series and parallel)
- o Determination of the conditions and constraints to the nominal operations of Impactor in term of SAA, Sun Distance and Load Power consumption

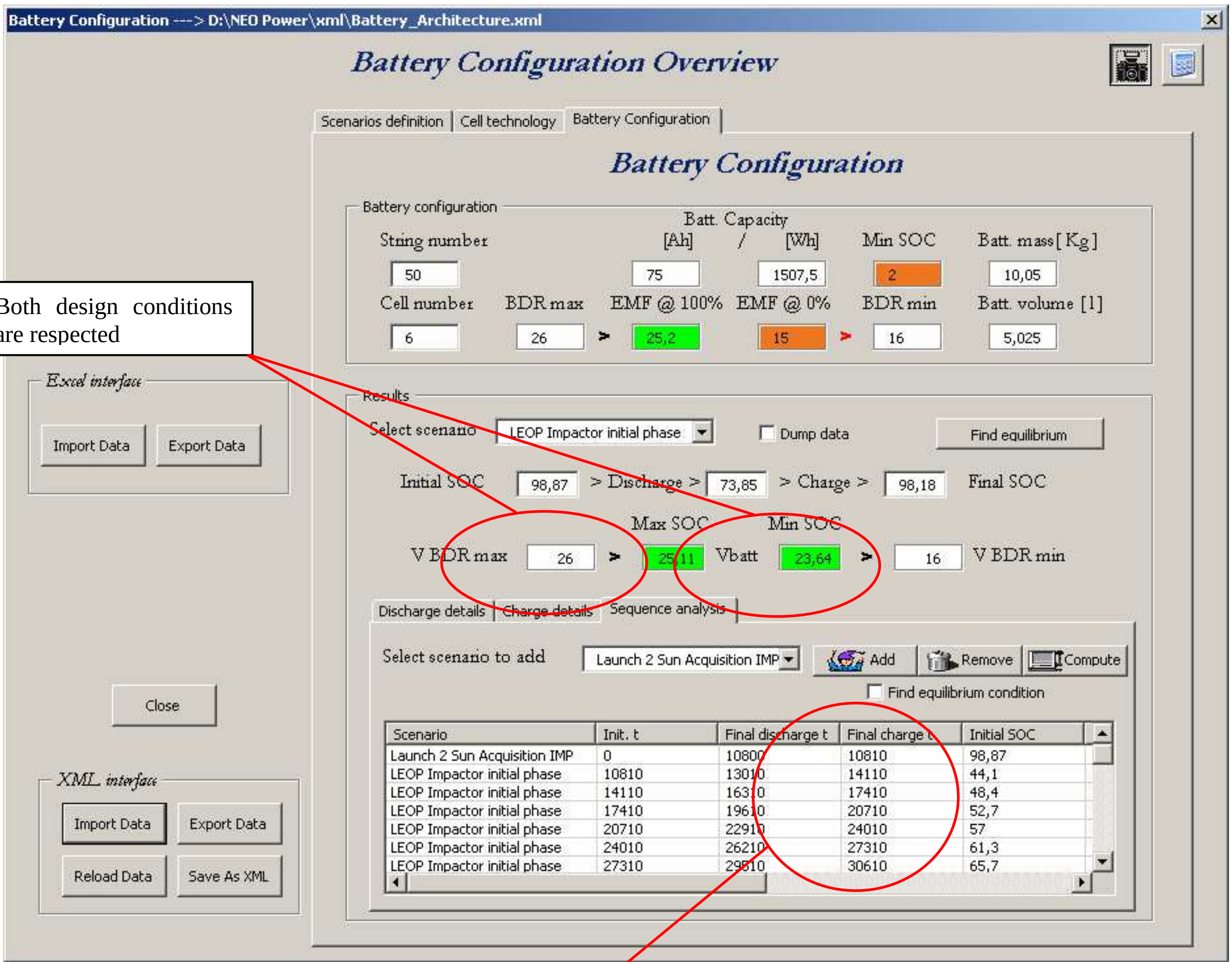
The following modes and worst case conditions have been used to size the Battery of Impactor :

Modes	Time [min]	Pload [W]
Pre-Launch	30	117,3
Launch	30	149,4
Sun Acquisition	120	334,6
LEOP (eclipse)	36,6	479,5
LEOP (sun)	55,8	550,1

In the following figures are shown the data outputs generated by the Excel spreadsheet tool used to perform the Impactor power budget using a Battery of 75 Ah (6 cells x 50 strings):

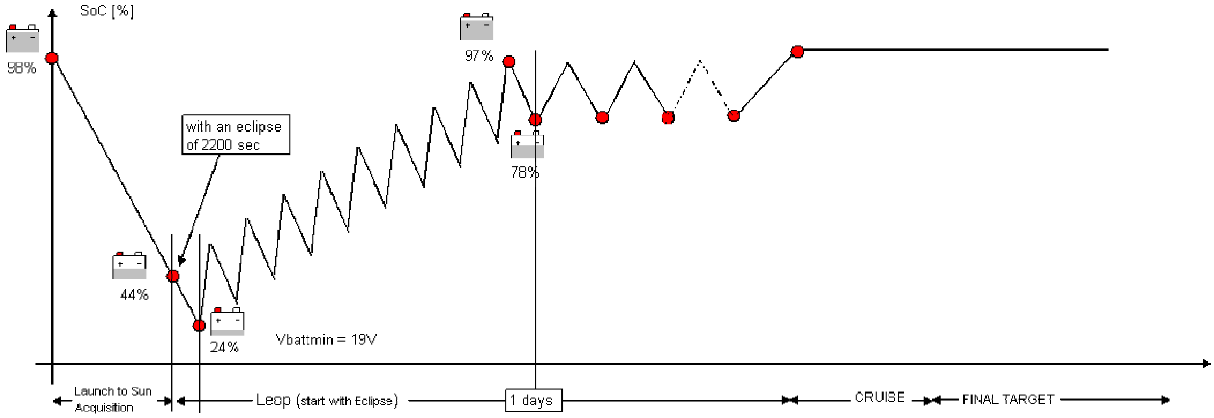
SoC of the Battery along Pre-Launch, Launch and Sun Acquisition phases :

Both design conditions are respected



This is the estimated SoC profile of Battery along the entire phase

This figure shows the SoC profile of the Battery along more mission phases to verify the capability of the Battery to supply the Loads.



It is important to note that also considering a worst case condition where the LEOP start with an eclipse the minimum SoC of the battery will be greater than 24%.

Considering that the minimum Vbatt is equal to 19V, this remains higher than the min voltage input of the BDR. On the basis of these results it is possible to conclude that the Battery selected can be considered sufficient to sustain the complete all the operation phases.

An estimation of the time required to maintain the SoC of the battery within a stable range of 98% and 78% range has been done.

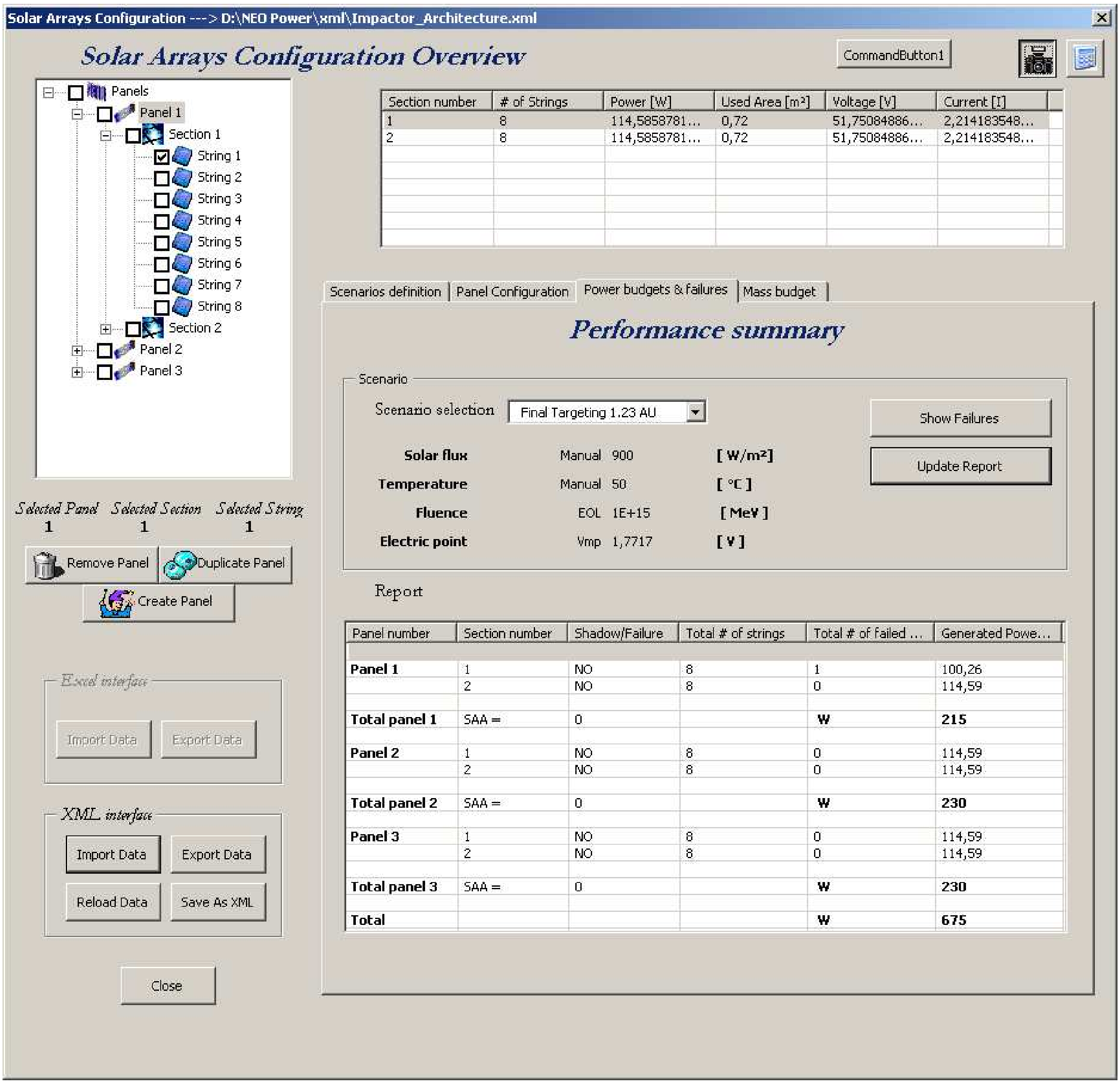
On the basis of the results after 1 day the SoC of the battery will be maintained within the estimated range.

The following modes and conditions have been used to size the Impactor SA :

IMPACTOR	Pwr Bdgt for IMPACTOR - DNEPR/SOUZ Launcher - SA Sizing Cases		Solar Array : 3 body panels (each 2 sections x 8 strings x 28 cells			Data : 27/09/06
	Modes					
	SUN ACQUISITION	CRUISE		FINAL TARGET		
Study Case		case 1	case 2	case 1	case 2	case 3
Sun Distance [AU]	1	1,39	0,7	1,23	1,23	1,23
SAA [°]	30°	10°	60°	0°	0°	0°
Temperature [degC]	100 °C	50 °C	130 °C	50°C	50°C	50°C
Degradation	BOL	EOL	EOL	EOL	EOL	EOL
SA Failure	1 string	1 string	1 string	1 string	1 string	1 string
SA configuration	28cx8strx2secx3BPs	28cx8strx2secx3BPs	28cx8strx2secx3BPs	28cx8strx2secx3BPs	28cx9strx2secx3BPs	28cx10strx2secx3BPs
SA area [m2]	5,1	5,1	5,1	5,1	5,7	6,35
SA mass [Kg]	18,0	18,0	18,0	18,0	20,5	22,7
Battery configuration	6sx55p	6sx55p	6sx55p	6sx55p	6sx55p	6sx55p
Battery mass [Kg]	12	12	12	12	12	12
Battery Failure	1 string	1 string	1 string	1 string	1 string	1 string
Eclipse/Sun time	no eclipse	no eclipse	no eclipse	no eclipse	no eclipse	no eclipse
S/C Load [W]	334,6	583,3	583,3	745,5	745,5	745,5
Battery Recharging Load [W]	0	0	0	0	0	0
Power from Battery [W]	0	0	0	70,5	0,0	0,0
Total Power Required [W]	334,6	583,3	583,3	745,5	745,5	745,5
SA Power generation [W]	858	523	758	675	759	844
Power Margins [W]	523,4	-60,3	174,7	0,0	13,5	98,5
					1 day of battery autonomy	

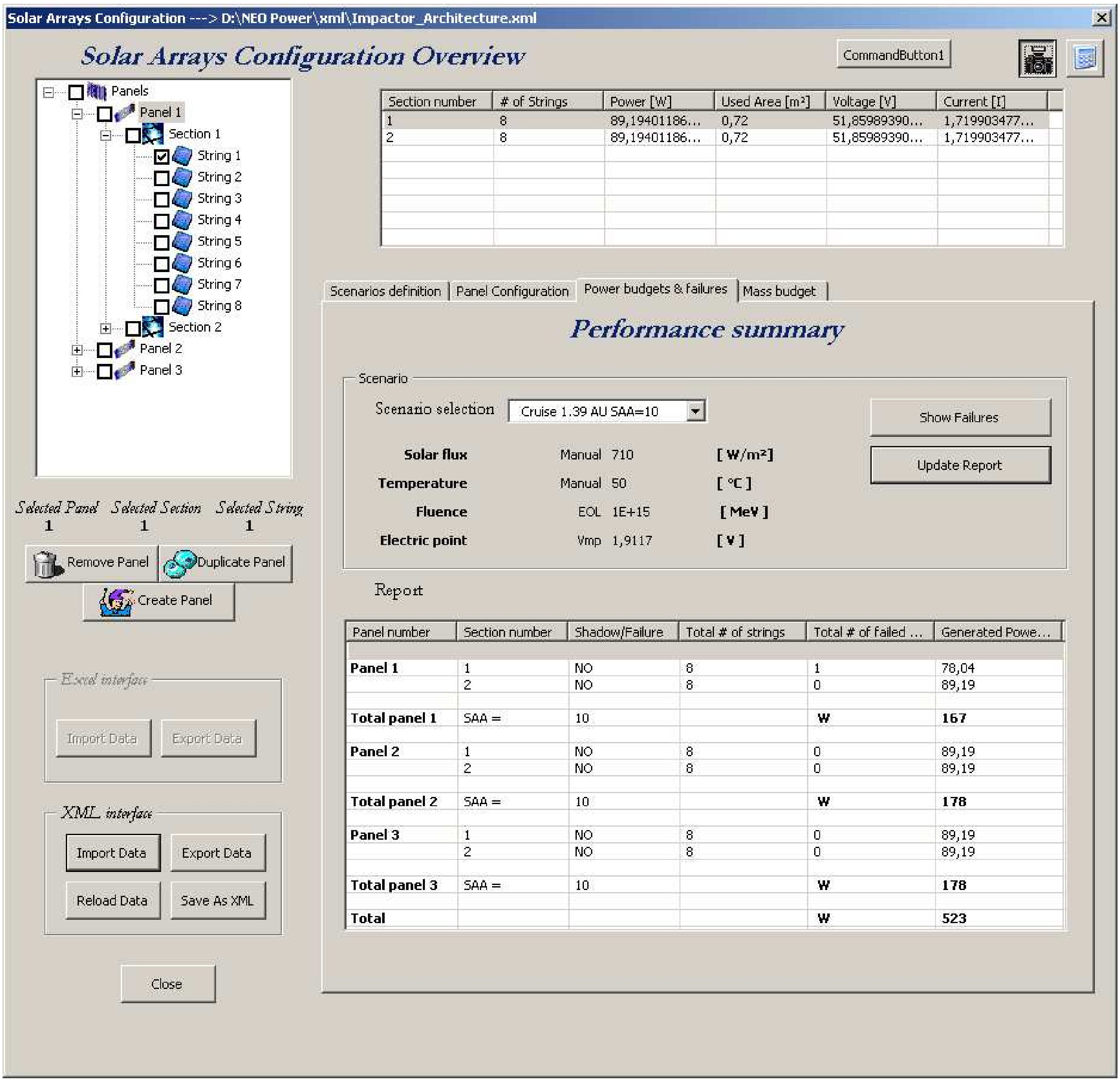
In the following figures are shown the data outputs generated by the Excel spreadsheet tool used to perform the Impactor power budget using a 3 SA body panels each one configured in 2 sections with 8 strings in parallel and 28 cells in series (total 1344 cells) . The comparison between the SA power generation level wrt the power demand from the Load and to recharge, if any, the battery allows to verify if the power margin is positive or negative.

<u>SA power generation capability for Final Target phase - case1:</u>

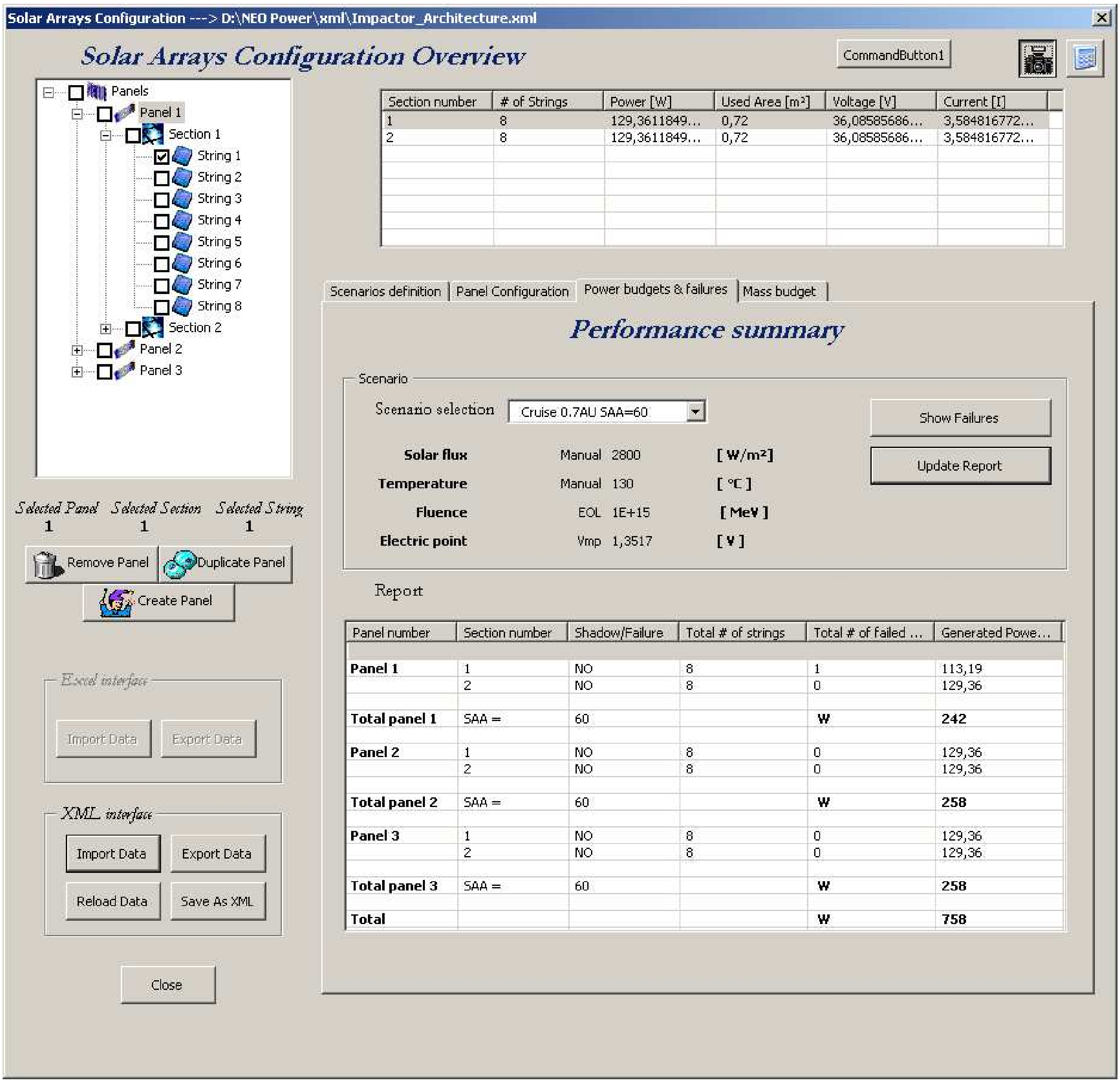


The Peak power consumption of 70,5 W exceeding the SA power generation capability will be covered by the Battery discharge. On the basis of the available data the Battery will be available to support this peak for one day. The Battery discharge time is compatible with the duration of this phase.

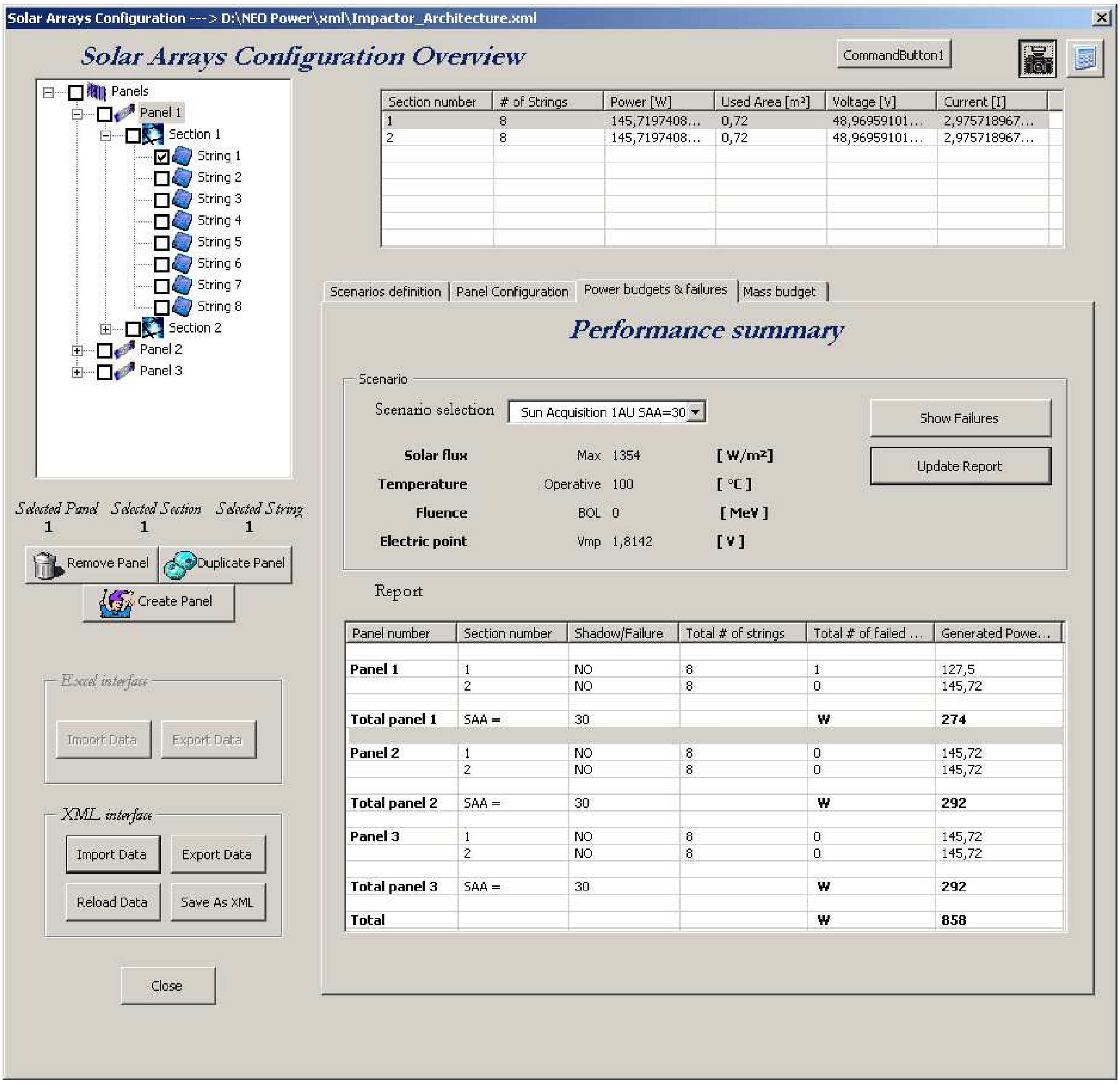
SA power generation capability for Cruise Mode - case1:



SA power generation capability for Cruise Mode - case2:



<u>SA power generation capability for Sun Acquisition Mode :</u>



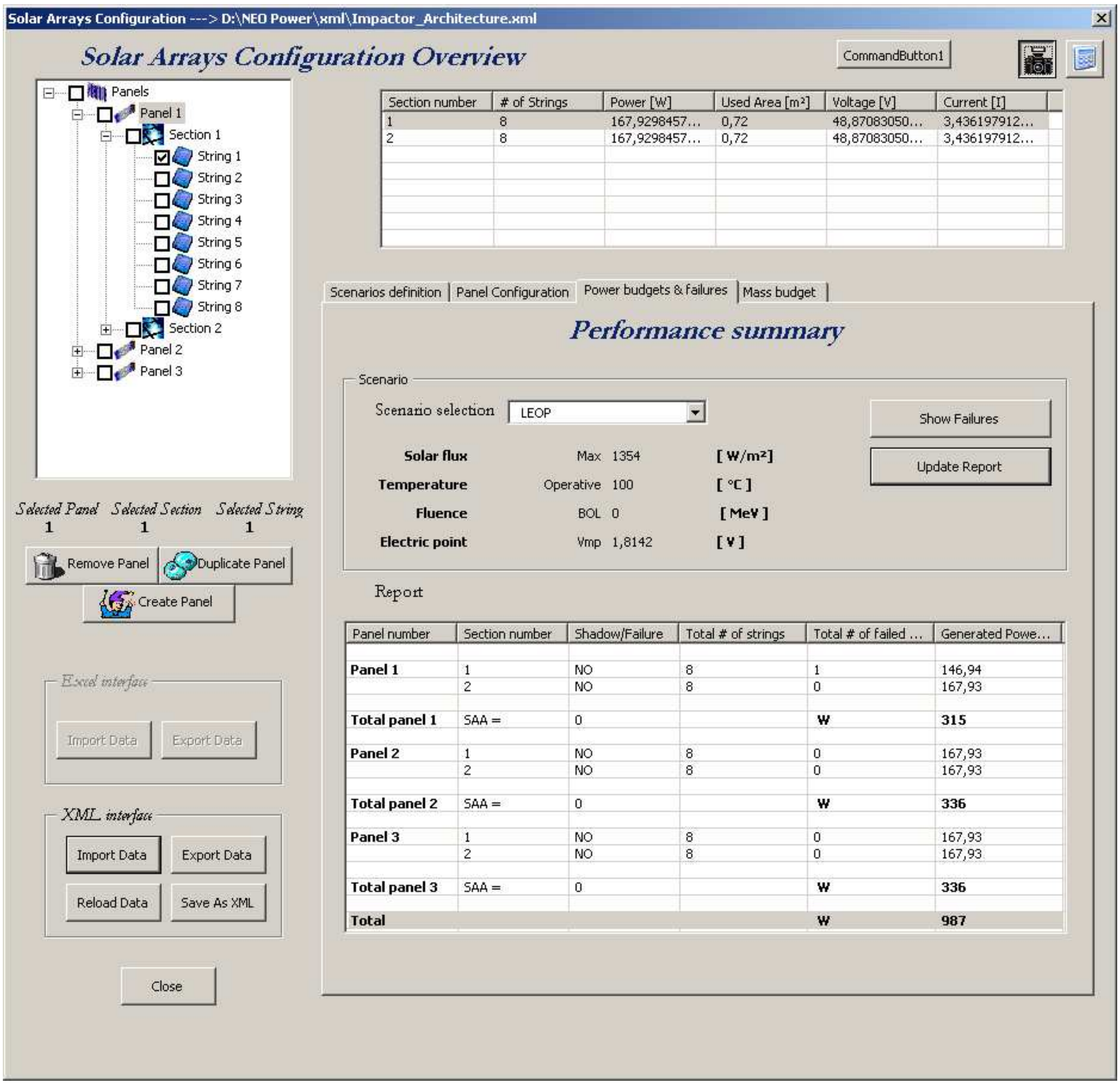
On the basis of the estimated power margin calculated by the Excel spreadsheet tools confirms that the designed SA with an area of 5 m^2 equipped with 1344 triple junction cells (GAGET 2/160-8040) is sufficient to support any operation and relevant Impactor mode.

The following table summarize the calculation of the power margins estimated by the Excel spreadsheet tools for the others IMPACT modes

IMPACTOR	Modes			Date : 28/09/06
Parameters	**caso 1**	caso 2		
Modes	LEOP		SAFE	Correction Manouvre
Sun Distance [AU]	1	1	1	1
SAA [°]	0°	0°	0°	0°
Temperature [degC]	100 °C	100 °C	100 °C	100 °C
Degradation	BOL	BOL	EOL	EOL
SA Failure	1 string	1 string	1 string	1 string
SA	28cx8strx2secx3panels	28cx8strx2secx3panels	28cx8strx2secx3panels	28cx8strx2secx3panels
Battery	6sx50p	6sx50p	6sx50p	6sx50p
Batter Failure	1 string	1 string	1 string	1 string
Eclipse/Sun time	36,6 min/55,8 min	40 min/50 min	-	-
S/C Load [W]	479,5	479,5	560,5	831,5
Battery Recharging Load [W]	400	500	0	1
Total Power Required [W]	879,5	979,5	560,5	660,4
SA Power generation [W]	987	987	840,0	840,0
Power Margins [W]	**107,5**	**7,5**	**279,5**	**179,6**

In the following figures are shown the data outputs generated by the Excel spreadsheet tool used to perform the Impactor power budget.

SA power generation capability for LEOP Mode :



SA power generation capability for SAFE and Correction Manouvre Mode :