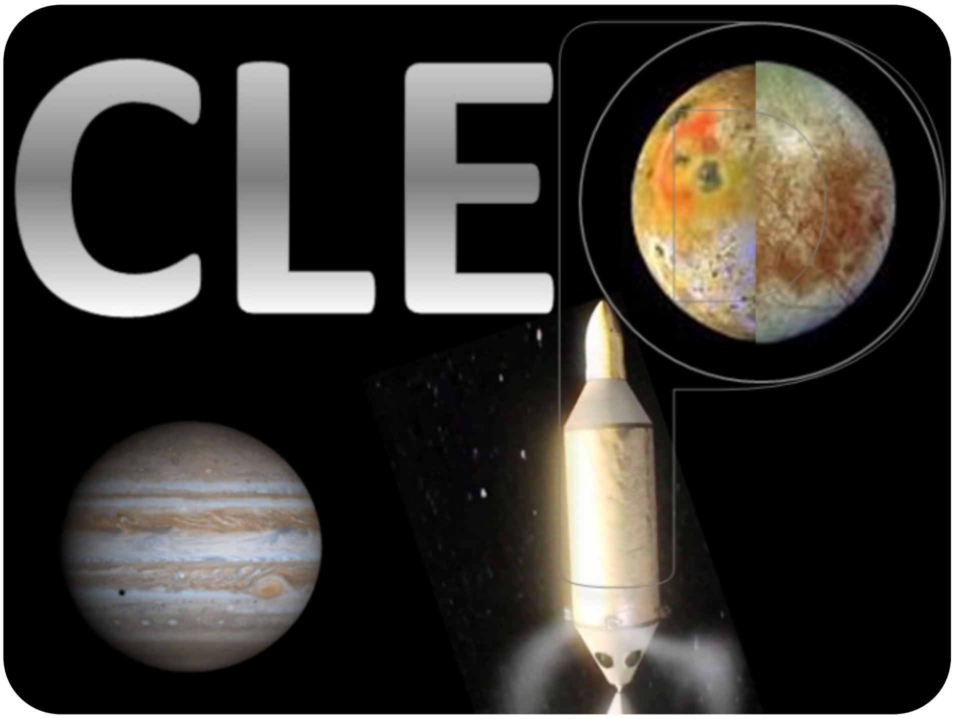
CDF Study Report

CLEO/P

Assessment of a Europa Penetrator Mission as Part of NASA Clipper Mission



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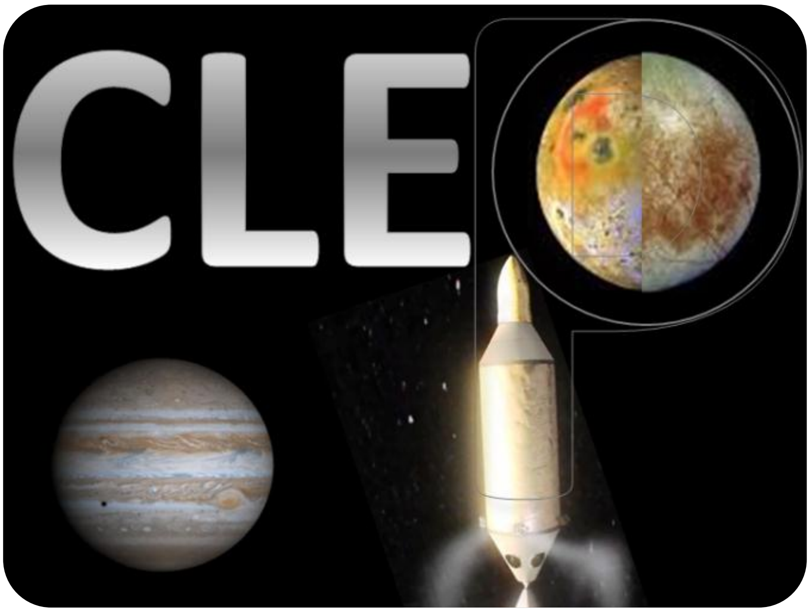

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Study Logo for the Penetrator Mission

STUDY TEAM

This study was performed in the ESTEC Concurrent Design Facility (CDF) by the following interdisciplinary team:

TEAM LEADER	R. Biesbroek, TEC-SYE		
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CONFIGURATION		PROGRAMMATICS/ AIV	
COST		PROPULSION	
RADIATION		RISK	
DATA HANDLING		SIMULATION	
GS&OPS		STRUCTURES	
MISSION ANALYSIS		SYSTEMS	
MECHANISMS		THERMAL	

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1 INTRODUCTION

1.1 Background

Following recent ESA-NASA bilateral discussions, mutual interest has been expressed for a potential contribution of ESA to the NASA Clipper Mission to the moons of Jupiter. As the junior partner to the Clipper mission ESA are considering a potential opportunity mission that could be considered by the science community in future mission proposals, to either carry out fly-bys of the Jupiter Moon Io or Europa or possibly to impact Europa. The study has been requested by ESA Science SRE-FM and financed by the General Studies Program (GSP)to be carried out in the CDF and has been nominated as CLEO/P: **CL**ipper **E**uropa ESA **O**rbiter or **P**enetrator (separate reports are produced for each case).

1.2 Scope

CLEO/P as the junior partner to the NASA Clipper mission will consist of a 250 (tbc) kg class element, attached to Clipper during launch and interplanetary transfer and released by Clipper after Jupiter Orbit Insertion (JOI) for close inspection and fly-bys of the Jupiter moon Io or possibly Europa, or an alternative mission to be a penetrator delivered to the surface of Europa.

The two concepts studied in the CDF were:

Concept 1: Minisat concept, providing close-up Io investigation and atmosphere in –situ measurements. Originally the mission was to be a Europa fly-by to investigate potential plumes identified on Europa, but the science argument for going to Io was greater, particularly when it is considered that the existence of Europa plumes have not been confirmed and that Clipper is anyway going to Europa. Europa was still to be considered as an option for this concept but more as a Delta to the Io mission. The minisat design was to take heritage from previous CDF studies (REIS, CRETE, JURA) and capitalising on JUICE developments and miniaturised and integrated technologies.

Concept 2: Penetrator concept, with high velocity impact with Europa and subsurface astrobiological and seismology investigation building on the Airbus industrial design originally performed in the context of the JUICE mission and updated in the context of the Clipper mission.

The purpose of the study was to design two different baselines, the Minisat concept and the Penetrator concept. Therefore the study consisted of 12 sessions including two internal final presentations, one at session 8 devoted to the minisat concept and one at session 12 for the penetrator. The study started with a Kick-off that was common to both baselines on the 10th February 2015 and ended with the penetrator internal final presentation on the 30th March 2015 and was carried out by a team of domain specialists from ESTEC and ESOC with involvement from NASA/JPL by teleconference to discuss interfaces with Clipper.

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2 EXECUTIVE SUMMARY

2.1 Study Flow

Requested by SRE-FM, the CLEP (penetrator Concept) study was performed in the Concurrent Design Facility (CDF) in four sessions, starting after a Final Presentation dedicated to the orbiter Concept, with a kick-off on 12 March 2015 and finishing with an internal final presentation on 30 March 2015. The sessions were supplemented with several splinter meetings to complete the design iteration in the very short time frame allocated to the Penetrator Study concept.

The assignment was to formulate a Penetrator concept (with high velocity impact on Europa and subsurface investigation, including a life detection experiment) for a possible ESA contribution to the NASA Clipper mission and to evaluate its feasibility.

2.2 Requirements and Design Drivers

The mission and systems requirements and design drivers for the CLEP study are provided in the systems chapter. As part of the outcome of an ESA contract, performed by AIRBUS in April 2014 under ESA contract #4000105327/NL/HB, was retained as starting point for the CDF assessment. It includes 2 stages for the S/C design : the penetrator itself, and a Penetrator Delivery System (PDS) carrying the penetrator and performing the main braking and penetrator targeting after release from Clipper.

2.3 Mission

Mission		
Launch date	May/June 2022	
Launcher	SLS direct to Jupiter in June 2022 is nominal plan; SLS direct to Jupiter in June 2023 is backup plan. *Atlas V 551 Earth-Venus-Earth-Earth Gravity Assist (EVEEGA) launching in May 2022 is alternate plan with Atlas V VEEGA launching in June 2023 as alternate backup.*	
Transfer Time	2.7 years	
CLIPPER Tour Modification	~ 14 months after JOI and after 7 high v-infinity nominal Europa fly-bys 24 additional perijove passages at Europa radius V infinity at release of the PDS" 1.68 km/s Additional mission duration: 150+45+150 = 345 days	
CLIPPER additional Fly-by Sequence characteristics	*EGA 1 altitude*	2870 km, crank-up
	Orbit	3:2
	EGA 2 altitude	90 km, crank-up
	Orbit	3:2
	EGA 3 altitude	2550 km
	Orbit	3:4
	EGA 4 altitude	456 km
	Orbit	1 revolution transfer to Ganymede

	Time from EGA1 to Ganymede	44 days
Landing Site	A3 See 5.3.2	
Ellipse landing dimensions	~ 300 x 300km	
PDS Release from CLIPPER	1.75 days before 2nd Europa fly-by, followed by a targeting manoeuvre at -1.5 days and Europa impact	
CLIPPER visibility from the penetrator	At impact for few minutes (2 min at 30 deg elevation) Post- impact, only after 10.5 days for data relay, and for 46 minutes	
PDS Targeting ΔV	10 m/s	
PDS Burn ΔV	2660 m/s	
Free-Fall height PDS+Penetrator	35 km	
Impact Velocity Penetrator	300 m/s	
Penetrator Design Concept	ForeBody + AfterBody Umbilical Cable assumed 10 m (TBC by test, Penetrator Equations from Sandia National Laboratories)	
ForeBody	Assumed as per AIRBUS Design	
AftBody	Textile antenna (40cmx40cm mounted on 4x1m tape springs), umbilical cord with comms and power lines	
Mass	Dry: 109.71kg incl 20% system margin Wet: 308.79kg incl 20% system margin Liquid Propellant Mass (Targeting + AOCS): SRM Propellant Mass (STAR 24):	
Radiation Shielding Mass	12.68 kg (spot shielding of sensitive equipment) required for Transfer phase. Radiation through the ice is negligible and no shielding is required. No radiation sensitive equipment will stay on Europa surface.	
Propulsion	Liquid (targeting, rate-dumping, spin-up, spin-down) - 1 x PEPT 230 tank, central axis - 3 x 20 N thrusters (main deltaV) - 2 x 20N + 2 x 20 N thrusters for spin/de-spin Solid (de-orbiting) - STAR 24: o 199.9 kg Solid propellant o 18.2 kg case o Isp: 282.9 s	
Penetrator Power	Energy req: 609 Wh incl 20% margin Primary Battery: Li-CFx 2.55 V per cell – 3s 6p 1377 Wh nameplate energy	
PDS Power	Energy req: 1980 Wh incl 20% margin Primary Battery: Li-CFx 2.55 V per cell – 5s 9p 3442 Wh nameplate energy	

	PCDU
Communications	1 Receiver; 1 Transmitter; 1 Diplexer (PDS) Deployable textile Antenna (penetrator) Fixed LGA in Penetrator ForeBody (penetrator) Umbilical RF Cable connecting Fore-body to antenna (penetrator)
Link Budget	Tx power: 1 W (penetrator) DataRate: 3 kbps (penetrator) Elevation > 30 deg (penetrator) Margin > 3 dB (penetrator) Achievable Data volume: 8 Mbit (penetrator)
AOCS/GNC	Micro STR (PDS) GYR on a chip (PDS) Capacitive MEMS ACC (PDS) [Optional] Altimeter on PDS for Penetrator/PDS Separation
Penetrator Thermal	2 "enclosures" architecture Energy Requirement considered: 20 Wh/day
PDS Thermal	MLI Kapton Foil Heaters Heating Power Consumption: 25 W
Penetrator Structures	Assumed same as AIRBUS industrial design
PDS + Penetrator Mechanisms	Clipper/CLEP Separation Mechanism: AIRBUS concept PDS/Penetrator Separation Mechanism: AIRBUS concept AftBody/ForeBody Separation Mechanism: pyro-mechanism triggering parachutes-like lines deployment Textile Antenna Deployment Mechanism (passive): 4 x 1 m tape springs, with antenna 40 cm centred at intersection point Cylinder with lid to be ejected by parachute deployment system

2.4 Technical Conclusions

The CDF Study has identified several points where further investigation is required, as - due to lack of time - several working assumptions have been made.

In particular, the following points will require further analysis:

1. Dispersions could be further reduced (inclusion of accelerometers brings improvements) through optimisation of the SRM burn
2. Separation Mechanism (Clipper-CLEP) Current reference design shall be assessed in detail
3. Separation Mechanism (PDS-Penetrator) Current reference design shall be revisited, induced error at separation has big impact on Penetrator Impact angle (including trigger strategy)

4. PDS/Penetrator Separation Triggering strategy (and required equipment – low TRL laser altimeter) shall be further explored
5. Ways to ensure that the antenna will not fall in the penetrator crater shall be further assessed; Dynamics induced by Aft-ForeBody separation shall be also investigated
6. Textile Antenna concept shall be studied in detail to achieve required maturity
7. Tape springs antenna deployment mechanism has only been sketched at CDF level, and shall be analysed in detail
8. Depth of penetration is computed based on empirical equations, valid under a set of assumptions. Length of the umbilical cord shall be determined based on representative test campaign
9. Umbilical folding strategy shall be optimised, as well as the required supporting structure/casing (conical structure + cylindrical case).
10. Clipper tour needs to be significantly modified compared to 13F7 reference tour including 24 additional perijove passages at Europa radius. The additional mission duration will be 345 days.

3 MISSION OBJECTIVES

3.1 Background

Following the successful GALILEO mission, a series of missions towards the Jovian system are currently in development : NASA's JUNO (on its way to Jupiter), NASA's CLIPPER (currently in phase A), and ESA's, JUICE (launch in 2022). While JUNO will focus on Jupiter system, CLIPPER will be dedicated to EUROPA and JUICE will mostly focus on GANYMEDE.

"Because of this ocean's potential suitability for life, Europa is one of the most important targets in all of planetary science" (NASA Space Studies Board 2011). As a potential piggy-back contribution to CLIPPER, a Europa penetrator mission would allow accessing Europa surface for the first time for in situ measurements.

3.2 Study Objectives

The main objectives of the study are the following:

- The Preliminary design of the CLEP Penetrator building on Airbus industrial design performed in the context of JUICE and updated in the context of Clipper
- To refine the science case and payload suite
- To identify the technology needs, risks and programmatics & cost aspects of CLEP and provide a preliminary risk register
- To iterate on the operational and interface requirements with NASA's Clipper mission.

3.3 Science Objectives

3.3.1 Europa Penetrator Mission (CLEP)

The focus for Europa should be on astrobiology and chemistry, supplemented by key measurements on geophysics.

The key science objectives of a Europa penetrator would be:

- Astrobiology of surface and sub-surface
- Chemical composition
- Geophysics: confirm existence of and determine ice depth to moon's ocean
- Geophysics: Characterise surface physical properties, and if possible their variation with depth
- Geophysics: determine additional constraints in interior structure.

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4 PAYLOAD

This chapter provides a summary of the individual elements of the model payload. The content is pure reflection of the output of the industrial study performed by Astrium UK in Phase 1 and Phase 2 (RD[4]).

The model payload selection followed a comprehensive scientific assessment of a penetrator mission to Europa buried in the top of the surface ice layer.

4.1 Baseline Design

From the early start, emphasis has been given to a resource budget sensible design omitting instruments that require large amounts of energy and very long operation time. Also the limited amount of transferrable data plays a crucial role in instrument design and operations.

Due to the nature of the mission, the scientific investigations are in-situ the icy layer which requires direct access to the outside of the probe. Thus the scientific payload requires a substantial support machinery to collect and process the sample material in proper format that can be analysed.

The probe contains a drill protruding from the probe into the adjacent ice. The collected ice is melted inside the probe and channelled to the individual scientific instruments.

4.2 List of Equipment

The scientific instrumentation is divided into two groups sharing different compartment within the penetrator.

The first group performs the in-situ analysis and is located in the so-called cold compartment in the penetrators head. The whole system is called E-PAC (Europa-Pentrator Astrobiology Complement). In here the sample acquisition and processing device is included. The scientific payload consists of the following instruments:

- Camera
- Habitability package
- Mass spectrometer.

The second group contains only one element, a seismometer, and is located in the so-called warm compartment that includes also the batteries, data processing and communication unit.

The overall TRL of the payload is rather low (2-3). Significant development steps towards an integrated design into the sample processing unit and shock resistivity has to be done in the future.

4.2.1 Camera

The camera will image the sample using differently coloured LEDs to determine the mineralogy, and search for potential bio-signatures.

A small detector assembly (256x256 pixel) provides a resolution of about 40 µm over the entire field of view of 10 mm. The overall mass of ~100 gram reflects the highly miniaturised design approach.

4.2.2 Habitability Package

A small fluid cell measure pH value, redox potential and electrical conductivity of a small liquid droplet (1 mm^3) extracted from the ice. The inside wall of the cell is covered with small electrodes immersed in the liquid sample. The mass of this instrument is estimated between 100 and 200 gram.

4.2.3 Mass Spectrometer

The mass spectrometer measures the chemical and isotopic composition of the volatile component of the collected sample. During the step-wise heating of up to 900 °C the volatiles are gradually released and directly analysed. After each measurement the sample cell is vented by an inert gas.

The current design favours a quadrupole ion trap mass spectrometer. This type of spectrometer can be build highly mass efficient. The whole system including the venting gas unit is estimated between 1 kg and 2 kg.

4.2.4 E-PAC

The whole E-PAC package has a mass of 2.255 kg, consumes 20.3 Wh during one full operational sequence (see Chapter 4.3) and creates of 3.048 Mbit of scientific and housekeeping data. A variable margin depending on the design maturity is included (up to 50%).

4.2.5 Micro Seismometer

The seismometer requires a minimum observation period of 3 days to have the chance to observe any kind seismic noise on the surface of Europa. The instrument is based on a 3-axis broadband MEMS device. The mass is assumed as 0.3 kg and would consume 2.86 Wh per operational day.

4.3 Timeline of Measurements

This timeline describes one full sequence of the scientific investigations on one collected sample. This forms the baseline science operation sequence. Only if resources allows, especially energy and data link budget

0 seconds	Switch on
4 seconds	The scientific investigation starts with an empty run of all instruments before inserting the sample material
36 seconds	The sample drilling procedure and sample sealing takes place
636 seconds	The sample is imaged and melted, habitability package measurement and first measurement by the mass spectrometer
724 seconds	Heating of sample up to 40 °C in steps by 10 °C followed by a mass spectrometer analysis at each step
849 seconds	1 image
1009 seconds	Heating of sample up to 100 °C in steps by 10 °C followed by a mass spectrometer analysis at each step
1024 seconds	Boil off all water
1579 seconds	1 image of residue

1589 seconds	Step combustion of residue material up to 900 °C in 50 °C steps followed by a mass spectrometer analysis at each step
2426 seconds	1 image of remaining material
2427 seconds	Switch off

During the scientific operations a total of 5 images, 29 mass spectra and a continuous measurement during the liquid state of the water by the habitability package are obtained.

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5 MISSION ANALYSIS

5.1 Requirements and Design Drivers

5.1.1 Requirements

MI-PE-000	The penetrator concept shall encompass the impact element (penetrator) and its carrier allowing to : cancel out the orbital velocity (braking manoeuvres(s)), to target the penetrator towards the targeted impact site, and to interface with Clipper during cruise.
MI-PE-005	The penetrator concept shall assume a release by CLIPPER S/C on a modified orbit wrt its nominal 4:1 resonant orbit with Europa. This modified CLIPPER orbit shall be 3:2 resonant with Europa (TBC) so as to allow lowering the Vinfinity at Europa to ~ 1650 m/s.
MI-PE-010	The penetrator shall impact Europa surface with a relative velocity of 300 m/s +/- TBD.
MI-PE-015	The penetrator shall impact the selected landing site with a dispersion ellipse of TBD*TBD km
MI-PE-020	Prior to the start of the landing sequence, the landing site shall be selected based on high resolution imaging of Europa surface. The landing site shall be such that : - Slope over a TBD m footprint shall be < TBD degrees (TBC) - Hazards with a height bigger than 0.5m are present with a probability lower than TBD % (TBC) - Visibility from Clipper within TBD days after impact shall be ensured.
MI-PE-030	The impact shall occur in visibility from Earth (TBC) and/or CLIPPER (TBC)

Table 5-1: Mission Analysis Requirements

5.1.2 Design Drivers

The following requirements and goals drive the Mission Analysis design:

1. Mass target of 250 kg
2. Precision of the targeting manoeuvre before landing
3. Visibility and choice of landing site
4. Radiation dose
5. Impact velocity range between 250-350ms^{-1}

The 1st point mainly drives the design of the transfer trajectory which aims at arriving at Europa with minimum infinite velocity. The 2nd and 3rd points play against each other and drive the design of the fly-by at which the PDM will be released from CLIPPER. The radiation dose drives the trajectory design towards spending as little time as possible in the vicinity of Jupiter.The allowable impact velocity dispersion drives the design of the whole trajectory design of the PDM post-release from Clipper.

5.2 Assumptions and Trade-Offs

5.2.1 CLIPPER Trajectory

The analysis was conducted assuming CLIPPER's arrival date is in April 2028. The corresponding Jupiter tour is 13F7 according to JPL nomenclature. This Jupiter tour consists of a total of 76 fly-bys with the Galilean Moons, 45 of which are with Europa. The tour is described in detail in RD[5]. It is subdivided into several phases each of which consists of 2-11 fly-bys:

1. Transition to Europa Science
2. COT-1 (Crank Over the Top 1)
3. COT-2
4. Petal Rotation
5. Crank-Up, Pump Down
6. Switch-Flip
7. Pump-Up, Avoid Sol. Conjunctions
8. COT-3
9. COT-4

In order to minimise the infinite velocity at release of the PDS, this tour has to be modified. The simplest scenario would be an insertion of an additional phase into the nominal CLIPPER tour. This additional phase would encompass:

1. Transfer to minimum V-infinity point
2. 3-4 Europa fly-bys at low V-infinity
3. Transfer back to initial orbital conditions

The point in the nominal tour where the additional phase is inserted is a trade-off between the accumulated radiation dose, impact on CLIPPER and the achieved knowledge of Europa ephemeris and characterisation of landing site prior to PDS separation. Ideally, the insertion of the additional phase should be as early as possible in the nominal tour. However, it should be guaranteed that at that point in the tour the ephemeris of Europa are known precisely enough for a safe landing and that the landing site has been characterised sufficiently.

5.2.2 Arrival V-infinity

The transfer trajectory is designed to minimise the infinite velocity at Europa arrival in order to minimise the SRM ΔV. However, the available sizes of SRMs do not always fit the required ΔV in relation to the dry mass. This constraint applies because there is a limit up to which propellant can be off-loaded from a SRM without risk of ignition failure. Therefore, it has been considered to increase the arrival infinite velocity beyond the minimum value to match the commercially available SRM sizes.

5.2.3 CLIPPER Fly-by Altitude at PDM Release

After the release of the PDM from CLIPPER, a targeting manoeuvre has to be executed in order to place the PDM on an impact trajectory with Europa and to arrive at the desired landing site. The size of this manoeuvre and also the dispersions connected to it are minimised if the fly-by altitude of CLIPPER is minimised. However, in order to have good visibility of CLIPPER after the impact, higher fly-by altitudes are favoured. During the CDF first high altitudes were considered. However, this resulted in unacceptably large dispersions of the impact velocity, therefore in order to mitigate these dispersions that originate from the targeting manoeuvre, a very low value for the fly-by altitude has been taken as the baseline at the cost of a reduced visibility window after impact.

5.3 Baseline Design

The baseline trajectory can be separated in several phases:

1. Interplanetary transfer.
2. JOI to end of COT-1 (12E6 is the last nominal fly-by), cf. RD[5].
3. Transfer to Minimum V-infinity point. I.e. CLIPPER's tour is modified using the fly-by sequence E-G-C-C-G-G-G-E.
4. Separation from CLIPPER and landing.
5. Additional Europa fly-bys of CLIPPER with Europa for data relay.
6. Transfer back to original orbital conditions.
7. Continuation of nominal CLIPPER tour.

Points 3-5 will be described in detail in the following.

5.3.1 Transfer to Minimum V-infinity Point

In the baseline design CLIPPER's tour is modified after COT-1, i.e. the transfer to the minimum V-infinity point is initiated shortly before fly-by 13E7. In order to minimise the radiation dose and the impact on CLIPPER an earlier transfer would have been advantageous. However, in order to guarantee a good knowledge of the landing site and Europa's ephemeris several Europa fly-bys must have been completed prior to release of the PDS. This would not have been the case for a transfer before COT-1.

The theoretical minimum of the infinite velocity w.r.t. Europa is around 1.6 km/s and is achieved in an orbit with the perijove at the Europa orbital radius and the apojove at the Ganymede orbital radius. This theoretical minimum cannot always be reached due to the phasing of the Moons. For the transfer after COT-1 two options are available: initiate the transfer shortly before 13E7 or before 14E8. A trajectory search has been conducted for both cases. The former leads to a solution lower arrival infinite velocity, so it has been chosen as a baseline. The baseline trajectory is shown in Figure 5-1. The initial orbit is the one with the highest apojove and the final one is the one with the lowest perijove. The corresponding evolution of the orbital radius is plotted in Figure 5-2, which also shows that the total duration of the transfer is around 150 days. This is quite long and is explained by an unfavourable phasing of the moons at that time: each arc has several revolutions around Jupiter until the spacecraft can encounter the next fly-by body.

Such a long transfer has a negative impact on the accumulated radiation dose of both CLIPPER and CLEP. If this turns out to be a driver, other transfer options have to be explored, e.g. using a pseudo-resonance with Europa prior to transfer that would change the relative phasing of the Moons.

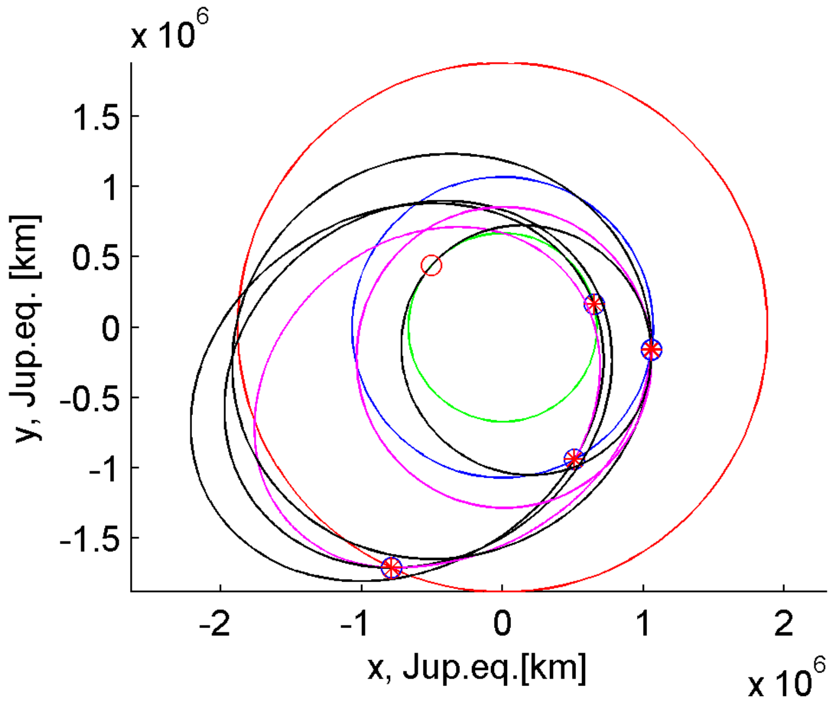


Figure 5-1: Transfer trajectory from after COT-1 to minimum V-infinity orbit.

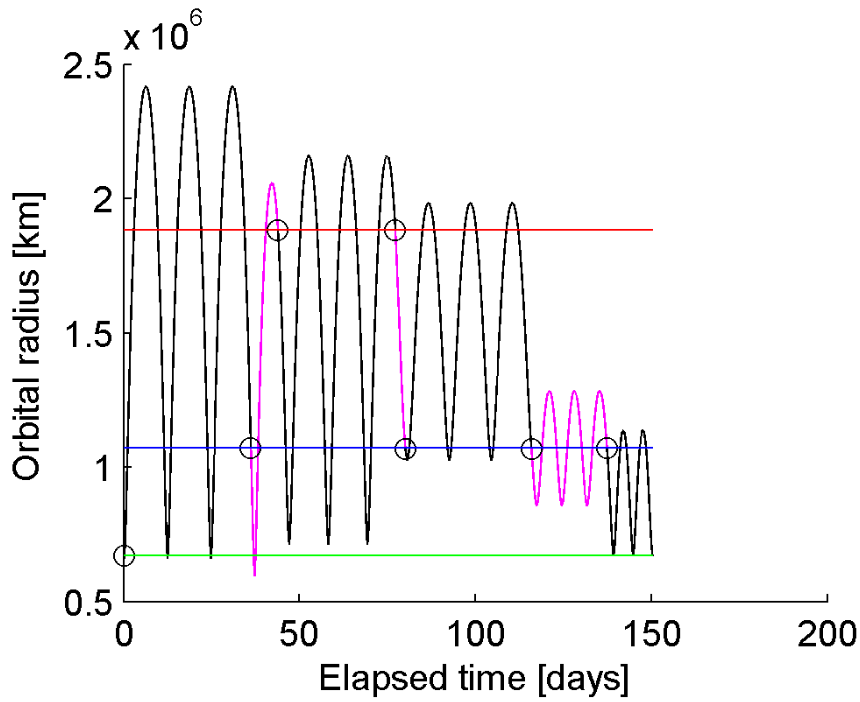


Figure 5-2: Evolution of the orbital radius of the baseline transfer trajectory. Europa, Ganymede and Calisto orbital radii are drawn as green, blue and red lines. Fly-bys are indicated as circles

The evolution of the distances to Galilean Moons is plotted in Figure 5-3. Fly-by epochs are indicated by green vertical lines. The Sun-Earth-CLEP geometry as plotted in Figure 5-4 indicates a superior conjunction shortly before arrival at Europa. This is indicated by both the Sun-CLEP-Earth and the Sun-Earth-CLEP angles going to 0°. It has yet to be analysed whether this superior conjunction is critical for communications between Earth and CLIPPER shortly before arrival.

For the sake of completeness also the Jupiter-CLEP-Earth and Sun-CLEP-Jupiter geometries are shown in Figure 5-5. Occultations by Jupiter occur if the Jupiter-CLEP-Earth angle is close to 0 deg. Eclipses by Jupiter occur if the Sun-CLEP-Jupiter angle is close to 0° and the Sun-Jupiter-CLEP angle is close to 180°. Figure 5-6 indicates that an almost continuous coverage could be obtained by using the Deep Space stations in NewNorcia and Malargue during the transfer phase. However, most likely operations during this phase will be done by NASA using their own antennas.

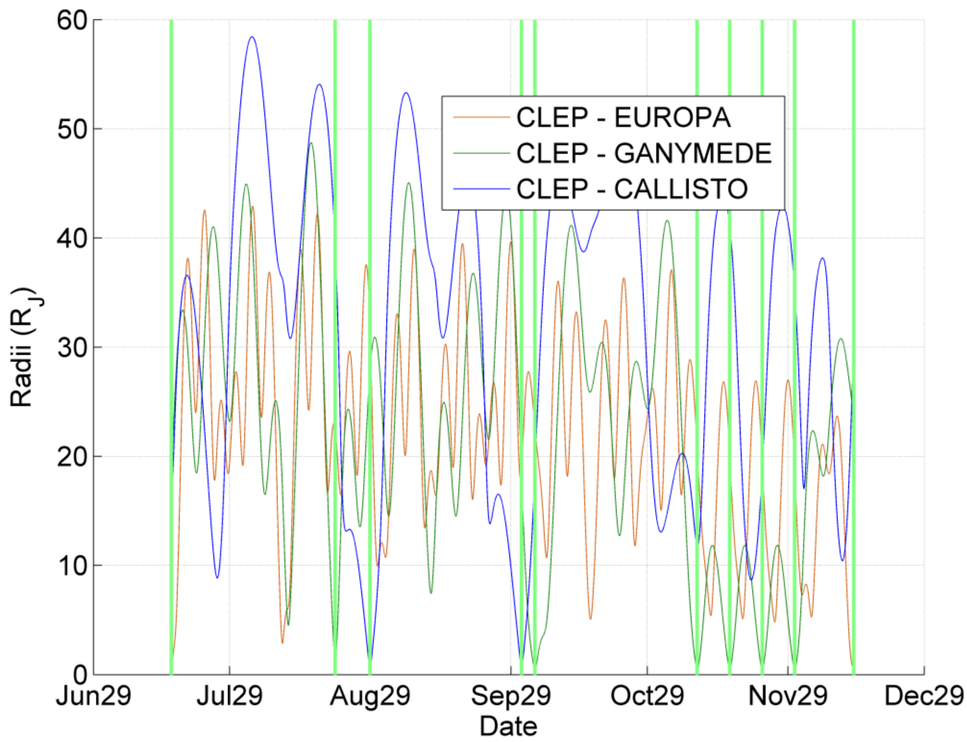


Figure 5-3: Evolution of distances to Galilean Moons for the baseline transfer trajectory to minimum V-infinity orbit

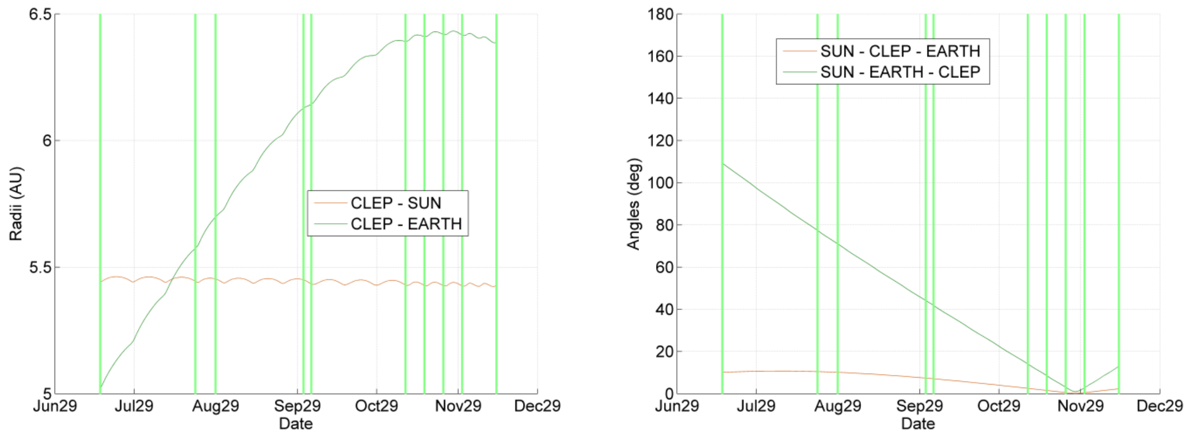


Figure 5-4: Evolution of the distances to the Sun and the Earth as well as the relevant angles for the baseline transfer trajectory from 13E7 to a minimum V-infinity orbit

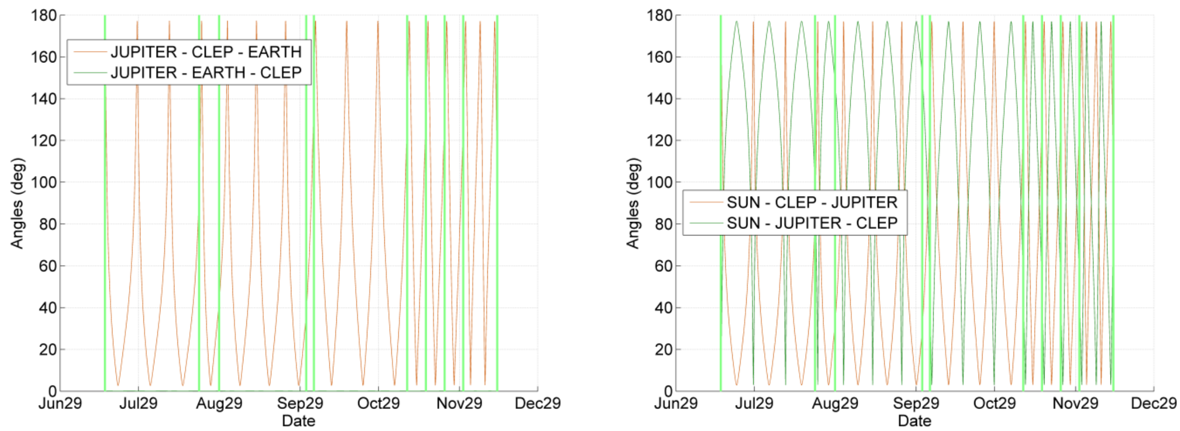


Figure 5-5: Evolution of Jupiter-CLEP-Earth and Sun-CLEP-Jupiter geometry for the baseline transfer to a minimum V-infinity orbit

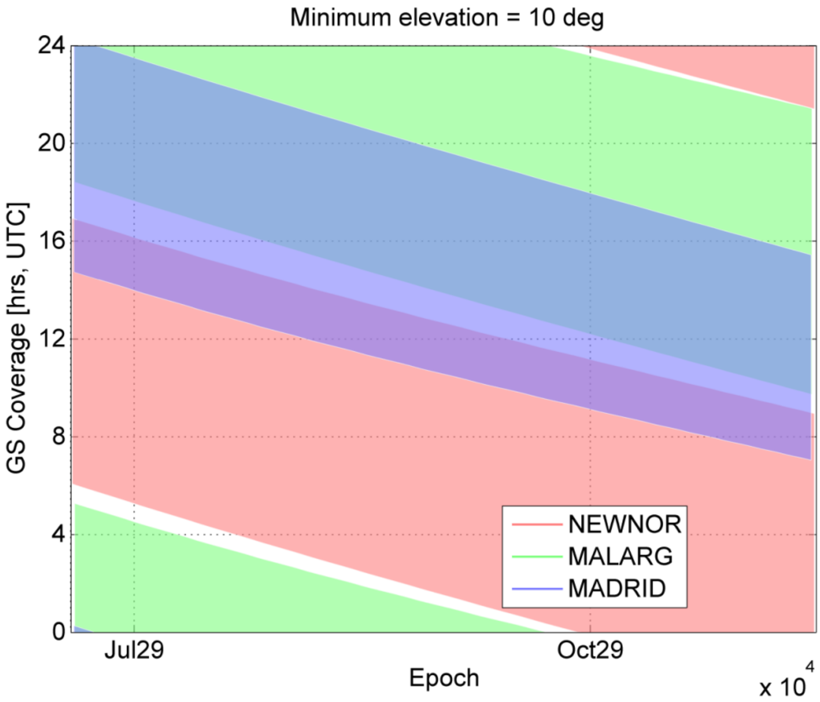


Figure 5-6: Ground station coverage for the baseline transfer trajectory

5.3.2 Landing Site Selection and Approach Strategy

In order to design the fly-by sequence after arrival, a landing site has to be selected. Actually the landing site selection process requires further thinking and should include Europa measurements from the Clipper orbiter before the Penetrator mission phase. For the purpose of this CDF study, possible landing sites have been proposed by the science team during the CDF, which are A3 or A1 (cf. Figure 5-7), with B1e, B1b and B1c as backup. A3 has been chosen as the baseline since it allowed for a better design of the ground tracks in the fly-by sequence with the given arrival conditions. The assumed coordinates for the centre of the landing site are shown in Table 5-2.

Latitude	-46.49°
Longitude	177.5°

Table 5-2: Assumed coordinates of baseline landing site A3 on Europa surface

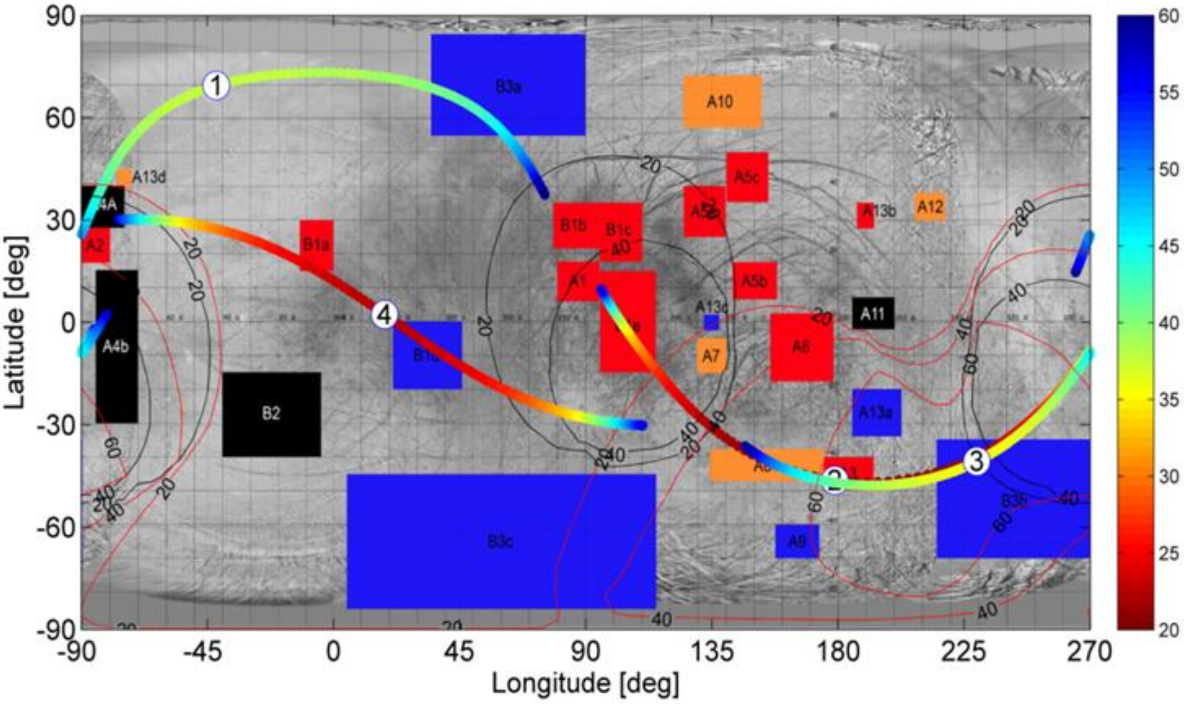


Figure 5-7: Location of candidate landing sites on the Europa surface as provided by the science team. A3 and A1 were proposed as primary sites and B1e, B1b and B1c as lower priority sites. The 0° longitude corresponds to the left edge of the figure

Figure 5-8 shows the arrival conditions at Europa in the B-plane. The B-plane is defined as the plane which contains the centre of the fly-by body and is perpendicular to the incoming velocity vector. The outgoing orbit of the spacecraft depends on which point in the B-plane is targeted during approach. Similarly, in the case of landing, a point in the B-plane corresponds to given latitude and longitude of the landing site as indicated in Figure 5-8. Note that the range of accessible landing sites covers more than one hemisphere. This is because the orbit of CLEP w.r.t. Europa is not a straight line, but a hyperbola. As a consequence also landing sites beyond the visible hemisphere are reachable. The AIRBUS design, RD[6], considers only two extreme cases of this approach:

1. Radial approach: This corresponds to a targeting of the centre of the B-plane in Figure 5-8 (i.e. the origin of the coordinate frame). This approach bears the advantage that the velocity of the PDS must not be reduced to zero by the SRM burn. Instead a lower altitude (e.g. 12 km) with some residual vertical velocity is targeted which implies some ΔV savings on the SRM. However, as the dispersion analysis later in this chapter will show, such a low targeting altitude implies an increased risk of collision with Europa even before the ignition of the SRM. Therefore, this ΔV saving strategy is not recommended at the current state of the study. Another disadvantage of the radial approach is that the landing site is entirely determined by the arrival conditions. A choice of landing site based on scientific objectives is highly restricted in such a scenario.
2. Tangential approach: This corresponds to targeting the edge of the light grey area in Figure 5-8. It implies that the SRM burn will always be close to the pericentre of the fly-by hyperbola. Again, it restricts the choice of landing site to a subset of available sites, although the restriction is not as severe as in the case of the radial

approach. Note that a purely tangential approach implies that the landing ellipse will be strongly elongated along the flight-path due to dispersions on the targeting manoeuvre.

Due to the limitations they impose on the design, no restriction on these two scenarios has been assumed for the current study. Instead, any hybrid between the two extremes is considered possible. However, after several iterations in the CDF a tangential approach was chosen for the baseline design since it minimises the dispersions originating from the targeting manoeuvre. Figure 5-8 also shows the point that is targeted by CLIPPER which is very close to the chosen landing site corresponding to a tangential approach strategy.

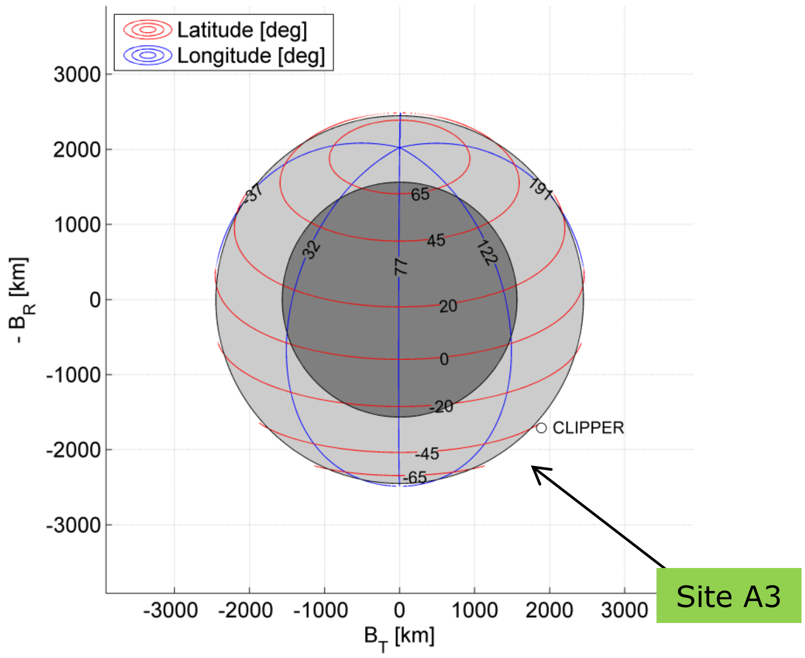


Figure 5-8: Arrival conditions at Europa as seen in the B-plane. The dark grey area indicates the radius of Europa, the light grey area all the points that will lead to an impact when targeted. The corresponding level lines of landing latitude and longitude are also shown

An overview of the final approach time line is shown in Figure 5-8. Separation from CLIPPER is assumed 1.75 days before CLIPPER pericentre. After 6 h for attitude acquisition and rate damping, the targeting manoeuvre is initiated, followed by a spin-up of the PDM. Based on the accelerometer measurement of the targeting manoeuvre, the time of SRM burn ignition will be updated on board during the following day. The SRM burn will be ignited such that the PDM becomes stationary w.r.t. the surface of Europa at an altitude of 35 km. In order to release the penetrator vertically to the surface of Europa, the PDM has to execute an attitude manoeuvre which is only possible after Spin-down. The penetrator will fall freely after separation from the PDS and impact at 300 m/s on the surface of Europa. A small deflection manoeuvre of the PDS will be necessary in order to avoid that the PDS and the penetrator land too closely together on the surface.

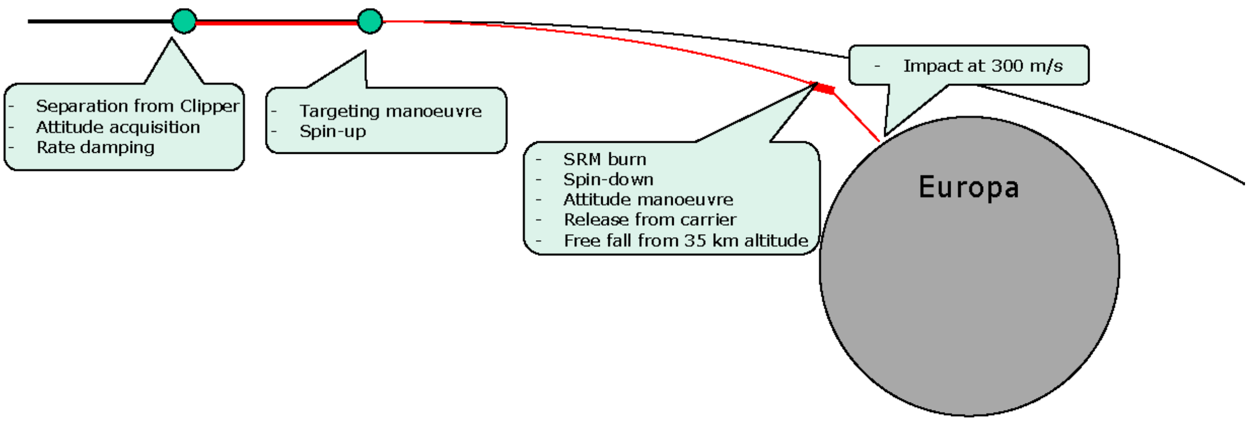


Figure 5-9: Time line overview of the CLEP final approach (schematic)

5.3.3 Fly-by Sequence and Visibility

Starting from arrival at Europa a fly-by sequence was designed for CLIPPER using the following requirements and assumptions:

1. The landing site shall be A3.
2. The landing occurs at the 2nd Europa encounter. The 1st fly-by is only used to tune the encounter conditions for the 2nd fly-by such that a tangential approach becomes possible for the chosen landing site.
3. It is assumed that the landing site has been sufficiently characterised during COT-1.
4. It is assumed that the ephemeris of Europa are sufficiently known after COT-1 to ensure a precise landing. No error on Europa ephemeris are regarded.
5. Landing has to occur in visibility of CLIPPER. Moreover, visibility with CLIPPER with minimum elevation of 30° is desired (but not required) after impact.
6. A 3rd pass of CLIPPER a few days later (duration to be minimised) has to occur in good visibility (minimum elevation of 30°) for several hours for download of the science data.
7. A 4th Europa fly-by has to be designed such that a Ganymede encounter is guaranteed initiating the transfer back to CLIPPER's original orbital conditions.

Table 5-3 summarises the baseline fly-by sequence that follows the transfer described in section 5.3.1. The corresponding ground tracks on the surface of Europa are depicted in Figure 5-10. The 1st Europa fly-by is solely designed in order to tune the arrival conditions for the 2nd fly-by to allow for a tangential approach. Landing occurs at the 2nd fly-by. The arrival conditions of the 2nd fly-by are those depicted in Figure 5-8. As can be seen from the ground track, the 2nd fly-by has its pericentre directly over the landing site A3. A third fly-by is needed for a download of the science data. Good visibility of the landing site can be achieved if the 3rd fly-by is chosen at 2550 km altitude with an outgoing resonance ratio of 4:3. The 4th Europa fly-by is solely designed to encounter Ganymede and initiate the transfer back to CLIPPER's original orbital conditions.

Europa GAM 1 altitude	2870 km, crank-up
Orbit	3:2
Europa GAM 2 altitude	90 km, crank-up

Orbit	3:2
Europa GAM 3 altitude	2550 km, crank-up
Orbit	4:3
Europa GAM 3 altitude	456 km
Orbit	1-revolutions transfer to Ganymede
Time from Europa GAM 1 to Ganymede	44 days

Table 5-3: Baseline fly-by sequence with Europa

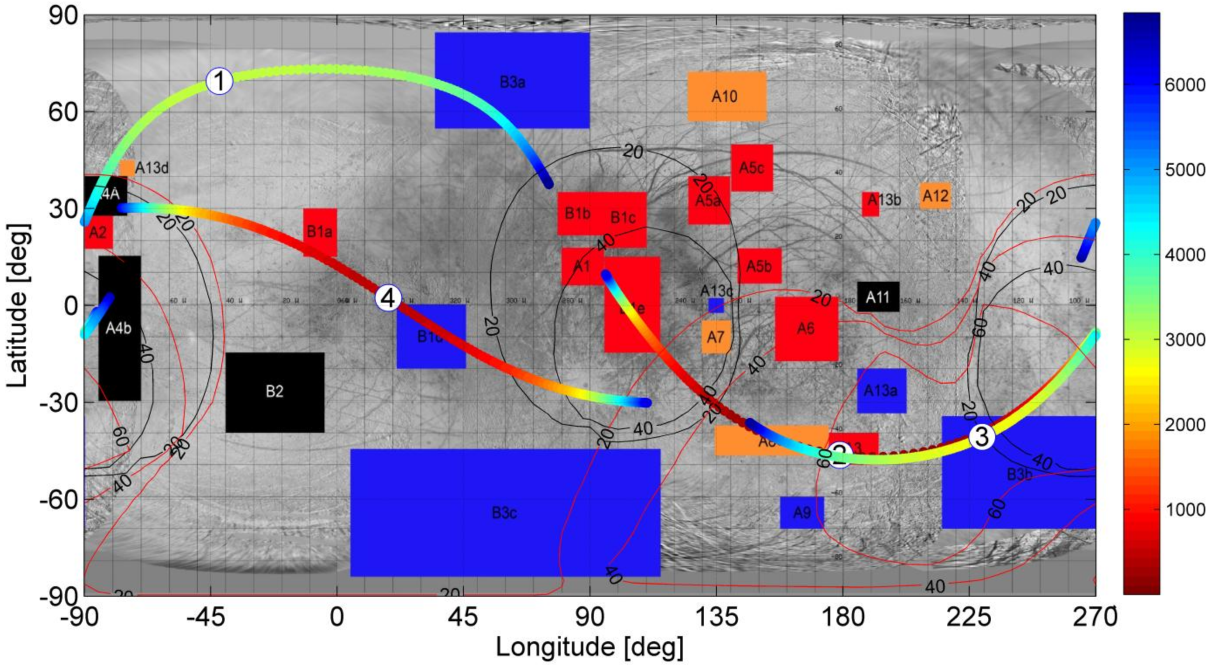


Figure 5-10: Ground tracks on the surface of Europa of the baseline fly-by sequence. The number labels indicate the pericentre of the fly-by. Only parts of the fly-by ±1h to CLIPPER pericentre are plotted. The altitude is indicated by the colorbar. Black contour lines enclose regions in which CLIPPER's elevation is higher than 30° for the indicated number of minutes at the 2nd fly-by. Red contours show the same for the 3rd fly-by

Figure 5-11 and Figure 5-12 shows the visibility plots for the 2nd and 3rd fly-bys. For the release (2nd) fly-by, the impact is at the same time as CLIPPER pericentre passage, and a moderate elevation of 30° is available for approximately 3 minutes. CLIPPER goes below horizon shortly after, and the next AOS at low range is expected to be achieved at the next fly-by, 10.5 days later.

The third fly-by visibility is better than the second one, as CLIPPER pericentre is much higher, and there is 1 hour of visibility at 45° elevation. Even more time is our disposal at lower elevations (~2 hours at 30°).

The fourth fly-by is used for setting up a transfer back to Ganymede, and as such, it is not vital to have good visibility, as all the transfer of scientific data is done during the third fly-by.

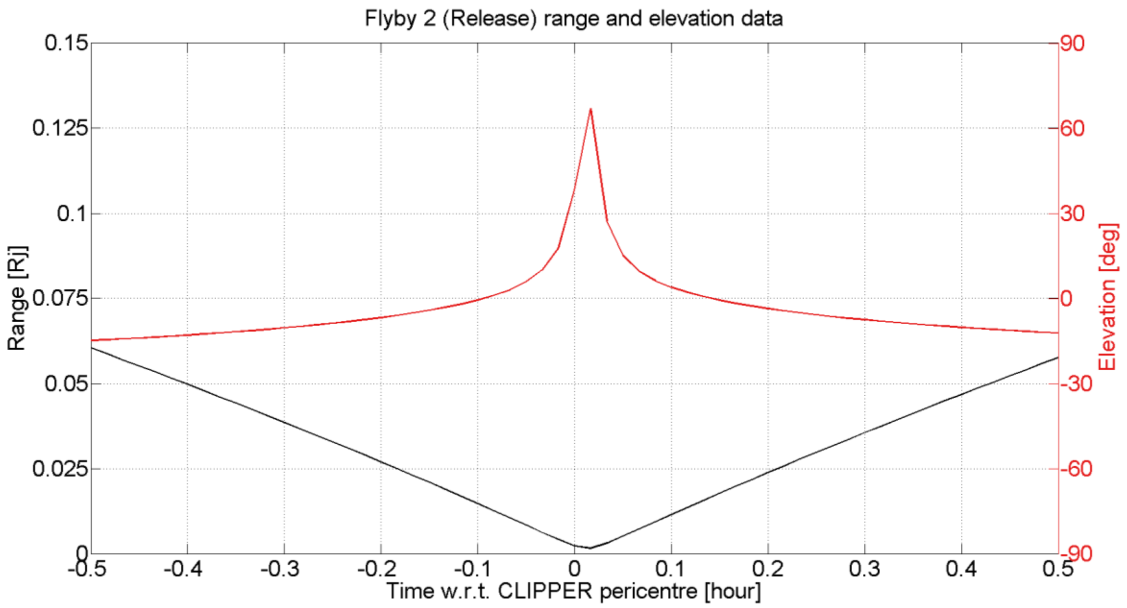


Figure 5-11 Range (black) and elevation (red) plots for the 1st fly-by

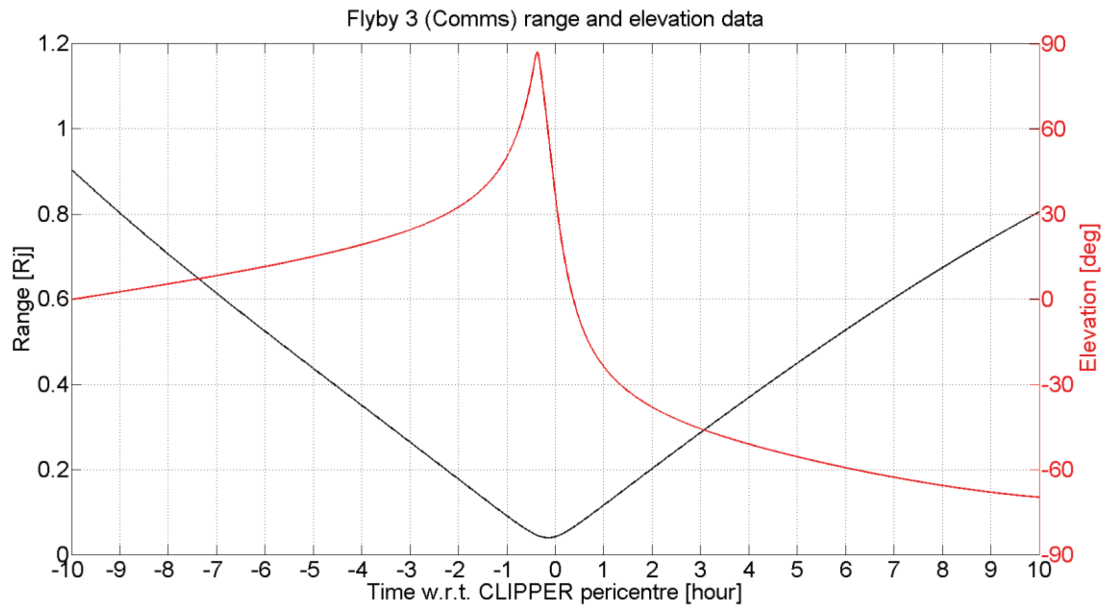


Figure 5-12 Range (black) and elevation (red) plots for the 2nd fly-by

5.3.4 Dispersion Analysis

Several sources of dispersion were investigated regarding the impact site and impact velocity of the penetrator at Europa, see the list composed below.

- Dispersion of CLIPPER state at separation
- Separation ΔV dispersion
- **Targeting manoeuvre**
- Spin-up
- **SRM burn**
- Spin-down

- Other parasitic forces
- Ephemeris error.

Two main contributors have been identified (in bold), the targeting and the SRM burn, with the rest having not significant effect compared to these, and thus have been neglected during the analysis.

The impact dispersion depends on two main properties of the manoeuvres: the ΔV (magnitude) of the manoeuvre, and the time of propagation between the execution of the manoeuvre and impact. The targeting burn is much lower than the SRM, but the time between is much longer; eventually this leads to a similar order of magnitude impact error from the two sources.

The targeting manoeuvre ΔV depends on several factors, first of all, the selected CLIPPER trajectory, and its corresponding Europa pericentre altitude. The higher the altitude, the more ΔV is required to put CLEP on a collision course with Europa. Other factors involve the targeting time, and as a free parameter while targeting a specific longitude/latitude, the arrival time. There exists an optimal ΔV guidance for the targeting manoeuvre, however, one may wish to specify the time of the impact, to be a within a certain time frame with respect to CLIPPER reaching Europa pericentre. Figure 5-13shows the contour plot of targeting ΔV magnitude in [m/s] for a specified targeting time and arrival time, while targeting the centre of Europa (radial case, targeting centre of the B-plane).

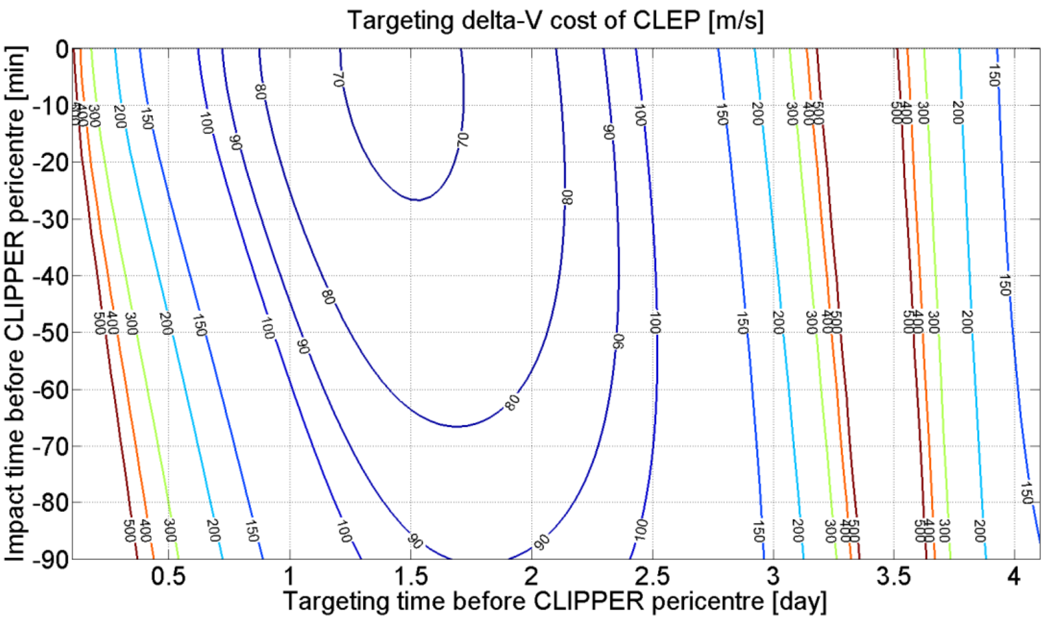


Figure 5-13 Targeting ΔV magnitude with different targeting and arrival times

The magnitude differs for each landing site, decreasing the ΔV as the landing site approaches the pericentric horizontal point, and increasing it as going "away" from the pericentre in the B-plane.

The baseline case assumes a targeting burn 1.5 days before, and landing at CLIPPER pericentre passage. This point refers to approximately 65 [m/s] targeting ΔV magnitude on Figure 5-13 (but with the specific assumptions of a radial approach, with Europa centre targeting and a high altitude targeting manoeuvre in order to maximise visibility of Clipper post-impact). The actual targeting manoeuvre corresponding to the selected landing site (A3) is much lower, 1.5 [m/s], as it is primarily selected in order to optimize the targeting burn magnitude which necessitates sacrificing the duration of visibility of Clipper post-impact. This is done to avoid large dispersions of the SRM ignition time, which would otherwise result in non feasible trajectories and impact velocities.

5.3.4.1 Targeting and SRM dispersion

The targeting manoeuvre is successfully reduced to a small value by choosing a specific fly-by sequence that puts CLIPPER pericentre right above the selected landing site (A3). Another contributor to further reduce dispersions is measuring the actual targeting ΔV magnitude and direction, which has also been investigated. The measurement confidence level has to be better than that of the manoeuvre to gain additional information. The information is used to recalculate the SRM ignition time w.r.t. the targeting burn epoch, to eliminate a good part of the impact velocity uncertainty resulting from the incorrect SRM ignition altitude.

The resulting impact velocity dispersions can be seen on Figure 5-14, with the following assumptions: SRM burn dispersions: 3% in magnitude, 1 deg in direction at 1 sigma. Targeting burn dispersions: 1% in magnitude, 1 deg in direction at 1 sigma. Accelerometer measurement 0.1% in magnitude, 0.1 deg in direction.

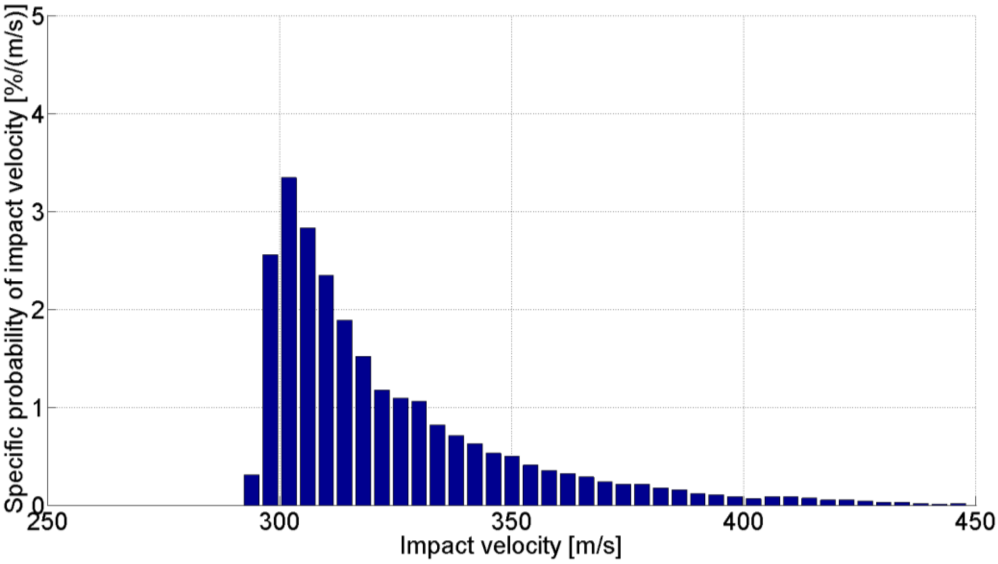


Figure 5-14 Total impact velocity dispersion with targeting measurement

A similar plot can be seen on Figure 5-15 regarding the impact flight path angle. A rather large range of impact angles are observable, and the grand majority comes from the SRM burn rather than the targeting manoeuvre. While the penetrator has been shown to survive an impact into ice at a high impact angle during a previous technology development activity, it is desirable to aim for low impact angles if possible to provide margin against the unknown surface slopes at the impact site as well. Thus, to reduce the range of possible angles, the SRM burn dispersion has to be optimized.

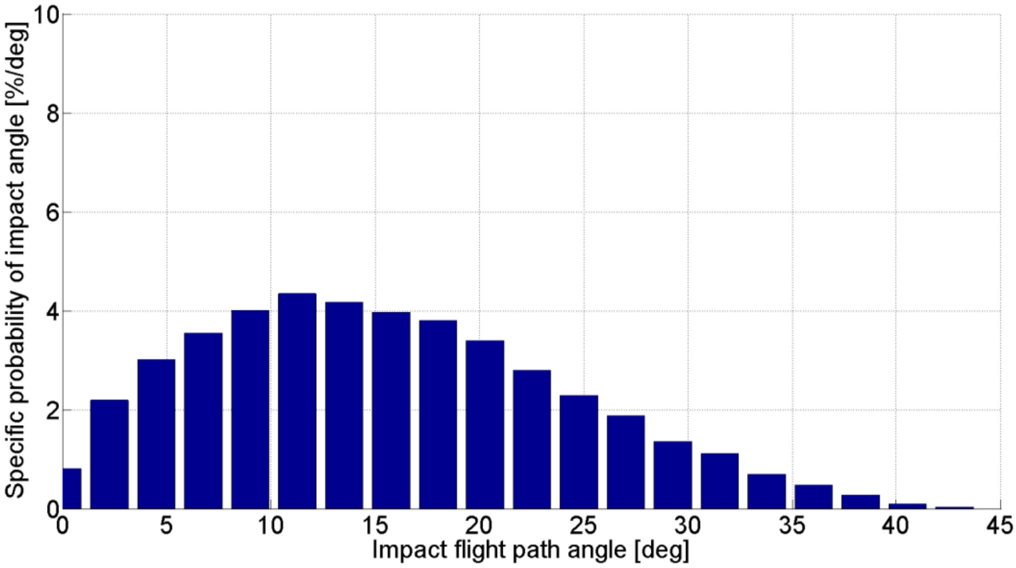


Figure 5-15 Flight path angle at impact, with targeting measurement

The resulting landing ellipse can be seen on Figure 5-16. It is elongated along the CLIPPER pericentre velocity vector, as the SRM burn is initiated horizontally w.r.t. Europa surface, and any error in the ΔV magnitude results in some remaining horizontal velocity.

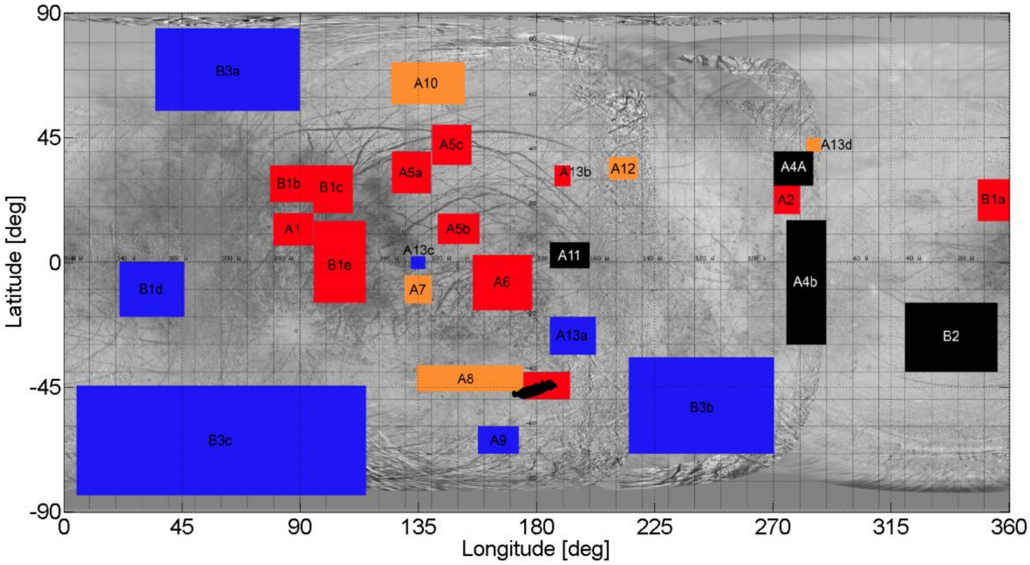


Figure 5-16 Landing sites on Europa surface and impact locations (black dots)

5.3.4.2 Dispersion conclusion

The impact velocity distribution at the current assumptions and baseline is manageable, the impact angle range is slightly larger than acceptable, but can still be adjusted by improving the SRM burn error.

The targeting ΔV has been reduced to a minimum achievable magnitude, such that the previously seen, fatal dispersion levels disappeared (e.g. impacting Europa with terminal velocity). The accelerometer measurement played a significant role at higher ΔV-s, however, at the current level they do not provide major improvement, nevertheless, they eliminate some of the error. Also, one question to be answered is that can we achieve the same measurement accuracy at 1.5 m/s as at 50 m/s.

The impact ellipse covers mostly the selected landing site, so that the impact location distribution is feasible as well.

5.4 ΔV Budget

The ΔV budget for CLEP is shown in Table 5-4. It includes gravity losses in case of the SRM burn. The actual targeting ΔV of the baseline design is 1.5 m/s. However, the sensitivity on the landing site selection and other parameters is rather strong therefore 10 m/s are allocated. No other margins are included here.

Manoeuvre	Size [m/s]
Targeting (liquid prop.)	10
De-orbit (SRM)	2600
TOTAL	2610

Table 5-4: Summary of the ΔV budget for CLEP

5.5 Options

5.5.1 Transfer before COT-1

In order to minimise the impact on CLIPPER and the radiation dose, initially a transfer before COT-1 was envisioned. However, the strategy has been dropped because no sufficient knowledge of Europa ephemeris and characterisation of the landing site can be assumed at Europa arrival, because there are no Europa fly-bys prior to COT-1 in the nominal CLIPPER tour. Nevertheless, this option shall be outlined here:

Five interface points with the nominal CLIPPER trajectory have been considered: 2G2, 3G3, 4C1, 5G4 and 6C2. For all of these a trajectory search has been conducted leading to similar arrival conditions at Europa. The infinite velocity at arrival is between 1.7 km/s and 1.8 km/s in all cases. Option 6C2 was chosen as the best case leading to the minimum infinite velocity of 1.75 km/s when using the fly-by sequence C-G-C-G-G-E. The duration from 6C2 to Europa arrival is 88 days. The corresponding trajectory is shown in Figure 5-17.

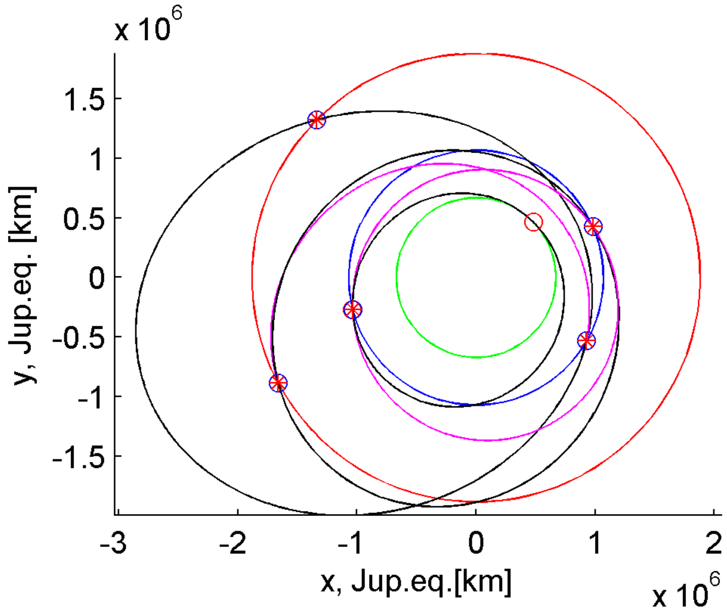


Figure 5-17: Transfer trajectory from 6C2 to a minimum infinite velocity w.r.t. Europa (option).

5.5.2 Tuning of the Arrival V-Infinity

As mentioned in section 5.2.2, the minimum arrival V-infinity that is obtained by the transfer does not necessarily fit the commercially available SRM sizes for a given dry mass. The arrival V-infinity and thus the size of the SRM manoeuvre can be tuned by simply shifting the arrival date at Europa by a few hours to earlier or later dates. Figure 5-18 shows how the infinite velocity and the SRM propellant mass are a function of the arrival date. For the calculation of the propellant mass an Isp of 283 s and a spacecraft mass before the burn of 240 kg have been assumed in the case shown. For a STAR-24 SRM the minimum fuel load is 160 kg. That corresponds to either an earlier arrival by 0.19 days or a later arrival of 0.38 days.

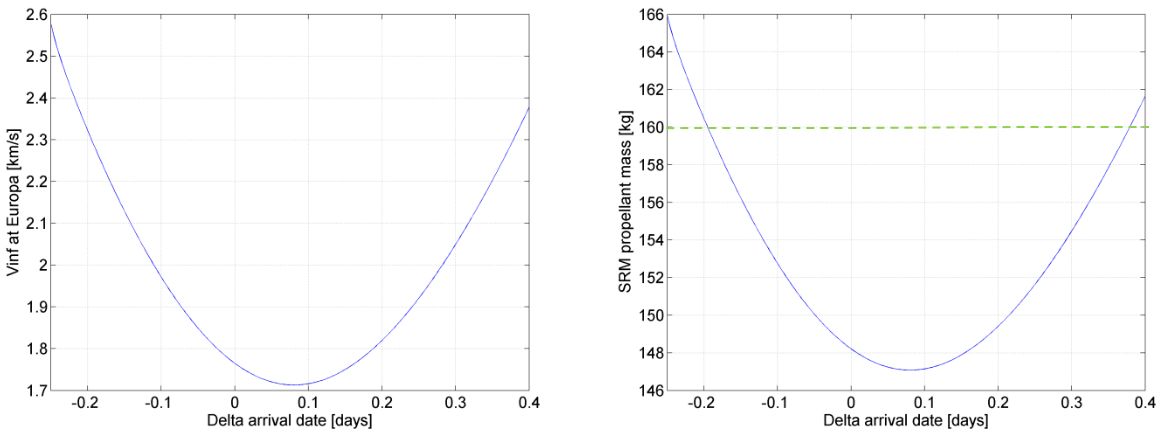


Figure 5-18: Arrival infinite velocity at Europa and corresponding SRM propellant mass as a function of the arrival date

Shifting the arrival dates not only affects the V-infinity magnitude, but also the direction as sketched out in Figure 5-19, which has an effect on the reachable landing sites. In the course of the study it turned out that a tuning of the arrival V-infinity is not needed for the current baseline scenario since the required propellant mass is in the range of allowed fuel loads of the STAR-24, i.e. the spacecraft dry mass is already rather high.

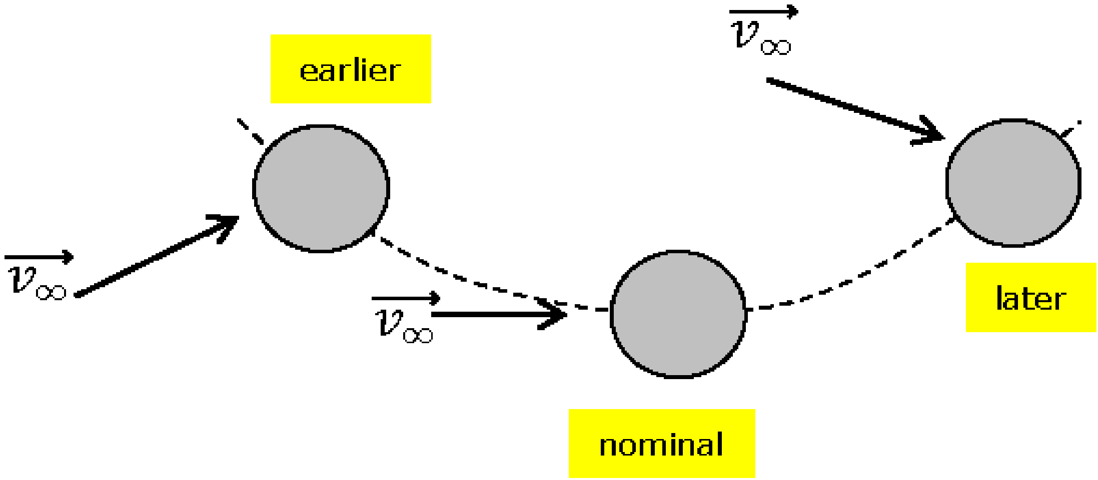


Figure 5-19: Effect on V-infinity direction and magnitude for earlier/later arrival. The dotted line indicates the orbit of Europa in the Jupiter equatorial frame

6 SYSTEMS

6.1 Requirements and Design Drivers

MI-PE-000	The penetrator concept shall encompass the impact element (penetrator) and its carrier allowing to: Cancel out the orbital velocity (braking manoeuvres(s)), to target the penetrator towards the targeted impact site, and to interface with Clipper during cruise. *C : this is the baseline concept definition from Airbus D&S Technical Note 15 PDS on board NASA Europa Clipper Mission Assessment PP3.ASU.TN.001 Jan 2015*
MI-PE-005	The penetrator concept shall assume a release by CLIPPER S/C on a modified orbit wrt its nominal 4:1 resonant orbit with Europa. This modified CLIPPER orbit shall be 3:2 resonant with Europa (TBC) so as to allow lowering the Vinfinity at Europa to ~ 1650 m/s. *C : this is the baseline mission concept definition from Airbus D&S Technical Note 15 PDS on board NASA Europa Clipper Mission Assessment PP3.ASU.TN.001 Jan 2015*
MI-PE-010	The penetrator shall impact Europa surface with a relative velocity of 300 m/s +/- 50 m/s. *C : as per Penetrator study – Airbus - data package*
MI-PE-015	The penetrator shall impact the selected landing site with a dispersion ellipse of TBD*TBD km *C: dispersion ellipse 300 km (TBC)*
MI-PE-020	Prior to the start of the landing sequence, the landing site shall be selected based on high resolution imaging of Europa surface. The landing site shall be such that : - Slope over a TBD m footprint shall be < TBD degrees (TBC) - Hazards with a height bigger than 0.5m are present with a probability lower than TBD % (TBC) - Visibility from Clipper within TBD days after impact shall be ensured. *C : TBD. Visibility from Clipper during the 2 minutes after impact is ensured and again for 1 hour after 10.5 days.*
MI-PE-030	The impact shall occur in visibility from Earth (TBC) and/or CLIPPER (TBC) *C : Impact occurs in visibility of CLIPPER*

Table 6-1: Mission & System requirements

The design drivers for the CLEP design can be summarised as follows:

- Minimise mass in order to meet the Not To Exceed allocation of 250 kg provided by NASA CLIPPER, by minimising:
  - ΔV (selecting appropriate trajectory and manoeuvre sequence)
  - Power (working on a mission timeline allowing to reduce the power required by the system)

  - Radiation (selecting appropriate trajectory and considering accommodation strategies so to minimise the required shielding mass, selecting high radiation tolerant equipment)
- Ensure Communication Robustness, ensuring both acknowledgement of successful landing and relay of scientific data to CLIPPER (specifying suitable requirements for CLIPPER tour modification)
- Minimise Landing Dispersions (by tuning manoeuvre and including adequate equipment in the baseline design so to reduce dispersion errors)
- Ensure Survival to harsh environment (shock at landing, extreme cold temperatures) while keeping in mind the mass constraint

6.2 System Assumptions and Trade-Offs

During the CLEO/P CDF study, four study sessions have been allocated to the Penetrator concept, considering the outcome of an ESA contract, performed by AIRBUS in April 2014 under ESA contract #4000105327/NL/HB,as a starting point for the assessment.

Airbus have developed a concept which the CDF Team adopted, aiming at bringing added value to the industrial baseline and consolidating the mission scenario.

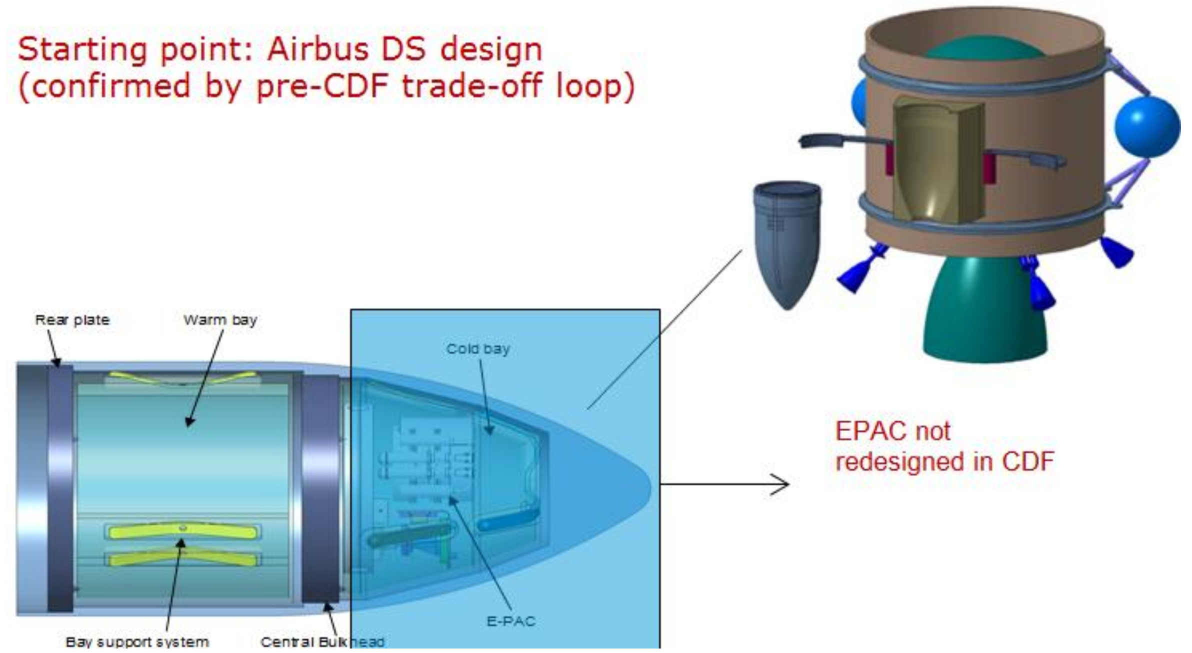


Figure 6-1: Airbus PDS & Penetrator Design (Courtesy of Airbus DS)

Main assumptions for the study were:

- Penetrator Delivery System (PDS) Design concept assumed to be the same as the one for the Airbus Design
- Penetrator main subsystems (E-PAC, Cold Bay, Warm bay, Rear Plate, Central Bulkhead, Bay support system) Design concepts assumed to be the same as the one for the Airbus Design.

As the AIRBUS Design had identified the risk associated with communication and relay of scientific data from the penetrator to CLIPPER through ice, the CDF design focussed on a concept which could reduce such risks by guaranteeing that the antenna stays on the surface.

The preliminary trade-off was dedicated to the analysis of the Penetrator Configuration:

- “Bullet-like” concept, with an expanded rear surface that would be used to brake at impact and prevent further penetration
- ForeBody and AftBody concept, based on the physical separation of the penetrator and the antenna, the former entering the surface and the latter staying on top of the surface after impact. Electrical and telecommunication connection would be implemented via an umbilical cable (single or redundant). Communication wireless options could also be possible, but have not been assessed in the frame of the CDF study. The biggest impact of the AftBody would be the necessity of a dedicated battery.

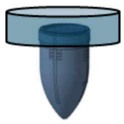
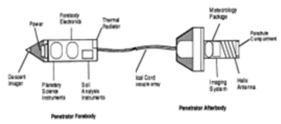
<table>
<tr><td>Configuration</td><td>Bullet-like with Braking Surface </td><td colspan="4">ForeBody+ AftBody </td></tr>
<tr><td>Fore/Aft Penetrator body separation</td><td>NA</td><td colspan="2">At impact</td><td colspan="2">Before impact (after PDS separation)</td></tr>
<tr><td>Fore/Aft resource sharing strategy</td><td>NA</td><td>Single Umbilical Cord</td><td colspan="2">Multiple Umbilical Cords</td><td>No Umbilical cord (dedicated battery on aft body + wireless comms fore/aft)</td></tr>
</table>

Table 6-2: Penetrator Configuration Trade-Off

Preliminary simulations clearly indicated that the idea of a braking surface would not be feasible: a huge surface would be required, with impact on mass and accommodation. Moreover, in the case of an oblique entry, the penetration depth would be reduced and in the best case would be 75 cm (the length of the Penetrator). This could jeopardise the scientific interest of the mission.

The ForeBody plus AftBody concept was selected as baseline.

Also the shape of the AftBody was subject to trade-off, and a conical rear body was investigated, though discarded for the accommodation complexity (and mass impact) that this shape would introduce at the interface with the PDS.

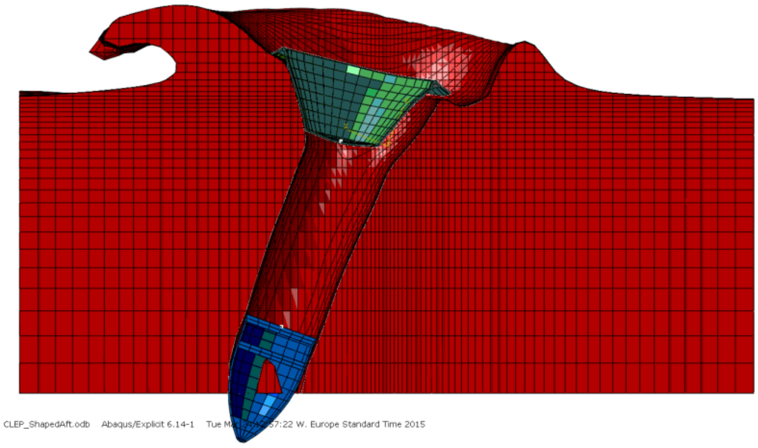


Figure 6-2: Preliminary simulations. Penetrator with conical aft body

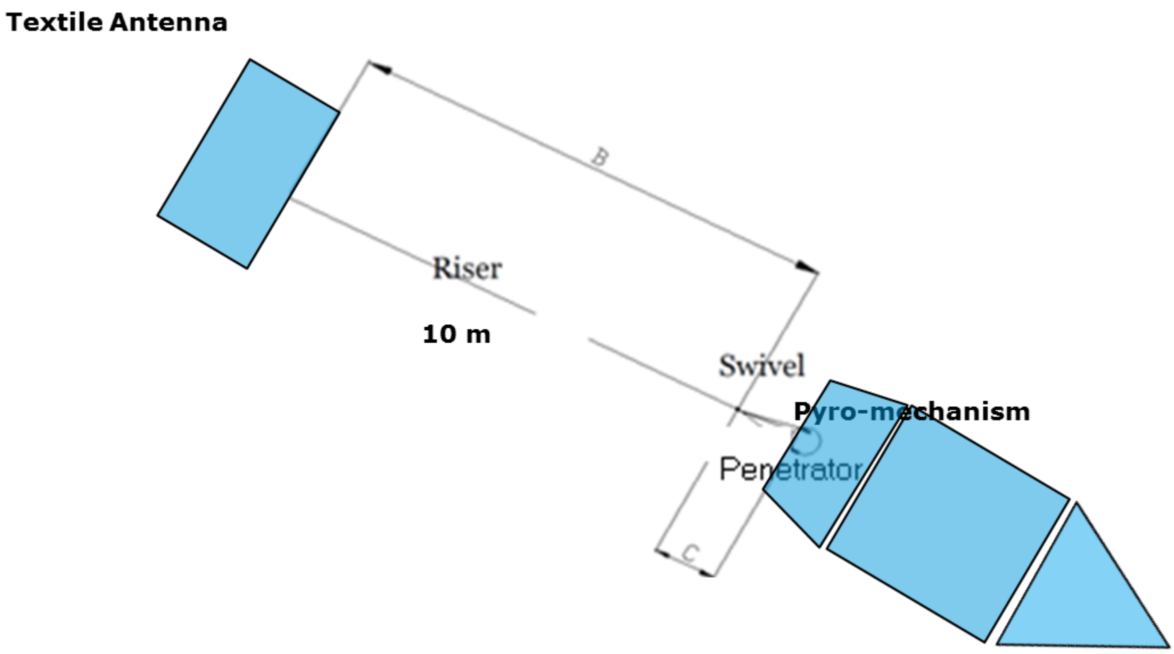
The two-bodies concept is based on the idea that the penetrator ForeBody is the Penetrator as per the Airbus design, while the Aftbody is simply constituted by an Antenna.

The Aftbody would be released before impact with a pyro-mechanism inherited from parachute deployment strategy and equipment.

Parachute lines would connect a low mass textile antenna folded into a cylinder with a lid, located in the rear plate of the penetrator.

The lines would be sized to take all the loads coming from the deployment, in order to protect a shorter umbilical cable, containing power and communication lines.

In folded configuration the umbilical would be wrapped around a conical support structure (aluminium made), located on the Penetrator rear end.



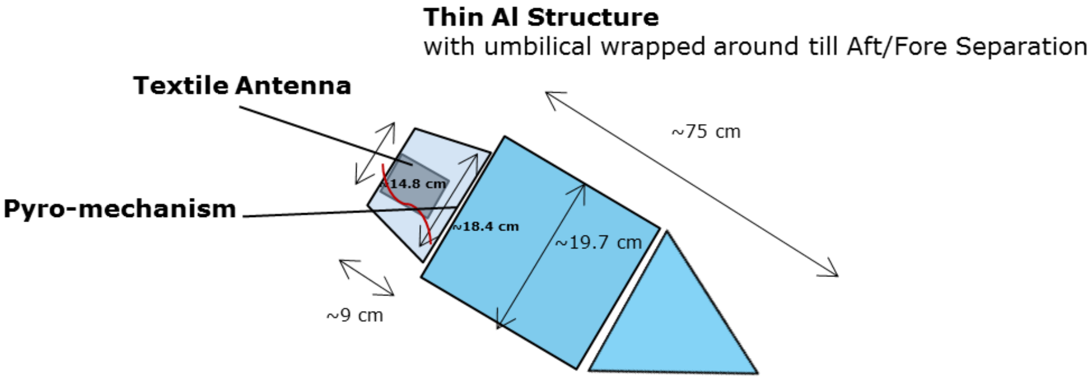


Figure 6-3: Penetrator Baseline: Fore and Aft Bodies

(deployed and stowed configuration)

Length of the umbilical cable and parachute lines has been assumed as 10 meters for CLEP.

A draft estimate of the penetration depth has been produced applying the Penetration Equations from Sandia National Laboratories (See Figure 6-4), and resulted in ~ 3 meters. However the uncertainty of the Europa surface characteristics, combined with the applicability boundaries for the empirical equations and the very preliminary maturity of the CLEO/P design suggested to apply substantial margins on the Sandia equations estimate.

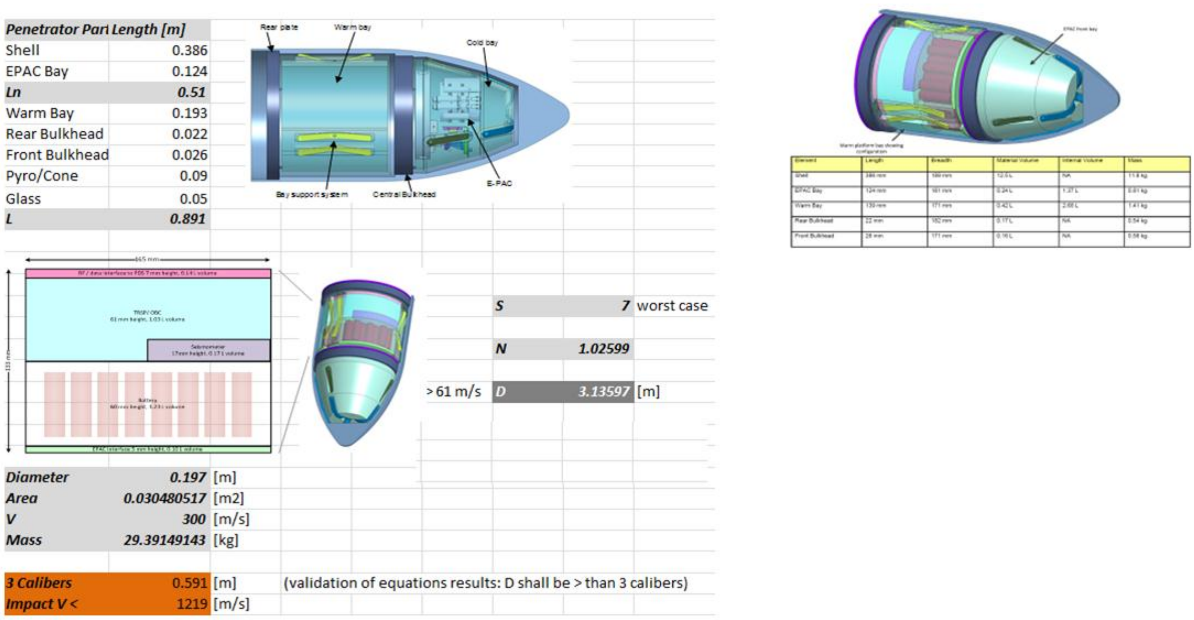


Figure 6-4: Sandia National Laboratories Equations Estimate (Courtesy of Airbus DS)

6.3 Product Tree and Function Tree

CLEP is the assembly of a Penetrator and a Penetrator Delivery System (PDS) which acts as a carrier. A list of subsystems and main functions of those elements is reported hereafter:

Penetrator:

- Penetrator (ForeBody)
  - Payload (EPAC) + Seismometer
  - Structure (incl I/F)
  - Thermal
  - Power (battery)
  - UHF
  - DHS
  - Harness
  - Pyro-mechanism (Fore-Aft Separation)
  - Lines
  - Mortar
  - Umbilical cord (comms/pwr)
- Penetrator (AftBody)
  - Textile Antenna
  - Antenna Deployment support Structure
- Functions:
  - Descent
  - Science (sub) surface ops
  - Comms

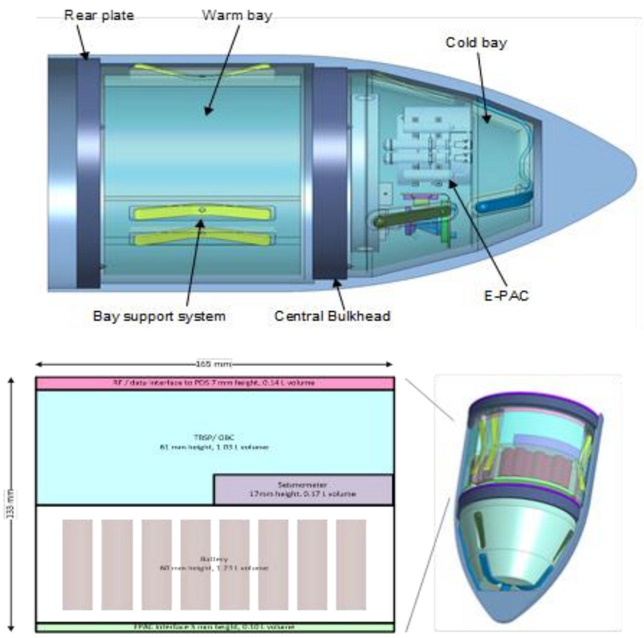


Figure 6-5: Penetrator (Courtesy of Airbus DS)

- PDS
    - Structure
    - Mechanisms
        - Clipper-CLEP
        - PDS-Penetrator
    - Thermal
    - Propulsion (SRM + Liquid)
    - AOCS
    - DHS
    - Harness
- Functions:
    - Maneuvers (targeting, release)
    - No comms

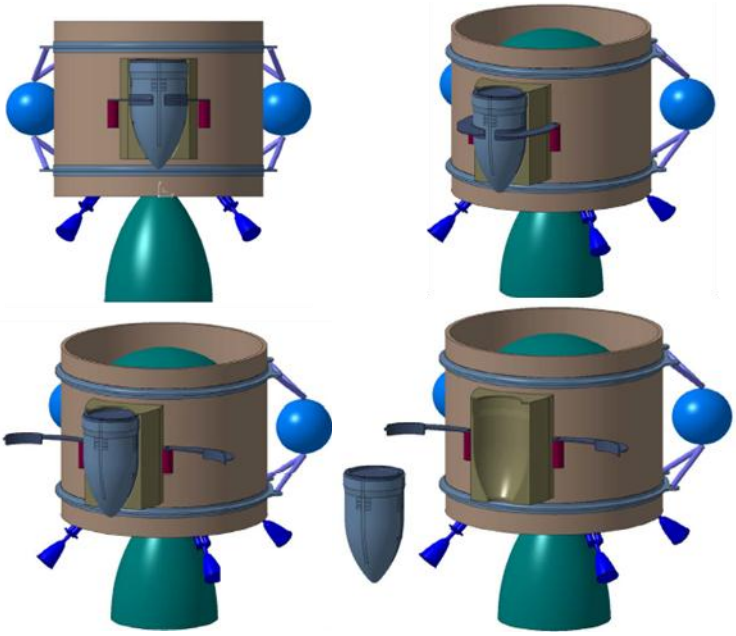


Figure 6-6: Views of the Penetrator Delivery System (Courtesy of Airbus DS)

6.4 Mission System Architecture

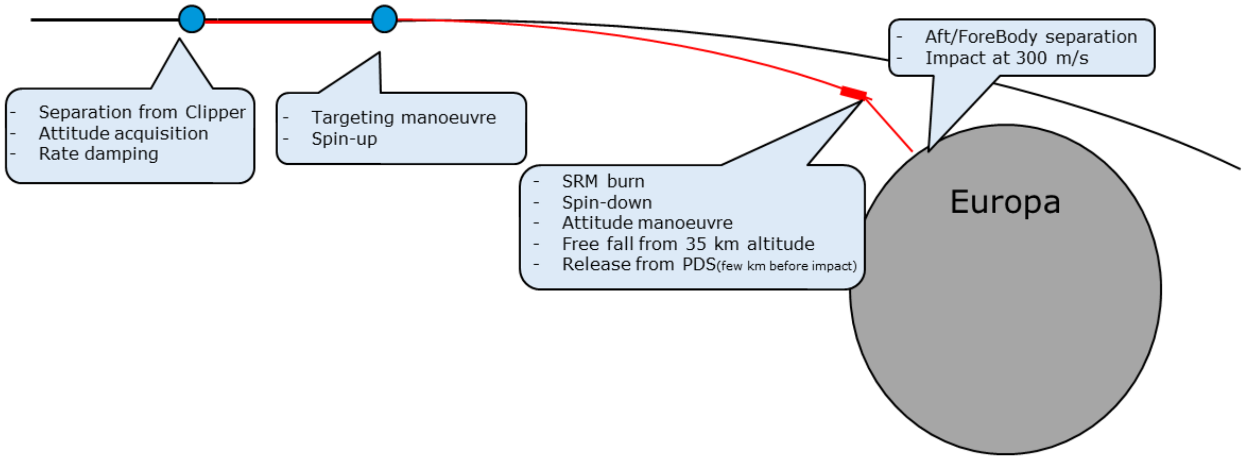


Figure 6-7: Mission Architecture

The mission concept is based on a separation from CLIPPER, followed by acquisition of inertial attitude using micro-STR and rate dumping and targeting manoeuvres implemented by liquid propulsion subsystem (ΔV=10m/s) using micro-STR, micro-GYR, micro-ACC (slews to point ΔV direction). A spin-up manoeuvre, still with liquid propulsion, is budgeted to guarantee efficiency during SRM Burn (De-Orbit SRM ΔV=2600 m/s). The stability of CLEO/P during the SRM burn is ensured by spinning the spacecraft, attempting at minimising the nutation angle amplitude, which reduces thrust efficiency. At the end of the SRM burn, CLEOP is at 35 km from Europa surface, with 0 m/s velocity (Altitude/Terrain relative Velocity acquisition could be performed with an altimeter, which is considered as an option in the CDF design).

At 35 km altitude the system PDS+Penetrator starts a free fall after SRM switch off. Stability is still ensured with spin, and the free fall duration is 231 seconds.

During the free fall, spin-down and attitude adjustment of the PDM occurs, prior to separation of the Penetrator from the PDS, which is triggered by a timer, or a measurement of the altitude via an altimeter or other technique (*Measurement-based alternative is optional).* Following the PDS/Penetrator separation, the AftBody/ForeBody separation is planned just before impact event followed by impact on Europa and acknowledgment of penetration being sent to CLIPPER.

Science is performed and the Penetrator shall survive 10.5 days: the time required to have CLIPPER back in visibility for relay of scientific data using the textile antenna left on the surface by design.

A rough timeline is displayed in the table below:

Manoeuvre	Size [m/s]	Thrust [N]	Thrust capability	Isp [s]	Propellant mass [kg]	Duration [s]	Altitude from Europa [km]
Separation CLIPPER PDM							
Navigation (CLIPPER)	12 x 8 = 96	20	3.46	210	11	327	
Rate Damping						21600	
Targeting	1.5	20	3.46	210	0.33	34	
Spin up		20	0.35	200	2.6	255	
	NA	NA	NA	NA	NA	129053	
De-orbit	2600	19660	1	282.9	200	28	42-35
Free Fall PDS+Penetrator	NA	NA	NA	NA	NA	60	32.6
Separation of Penetrator from PDS	NA	NA	NA	NA	NA	166	32.6 - 1.4 231 s
Fore-Aft Separation	NA	NA	NA	NA	NA	5	1.4
Impact	NA	NA	NA	NA	NA	0	0
Exact landing wrt peri Clipper	NA	NA	NA	NA	NA	0	
Science and Relay	NA	NA	NA	NA	NA	10.5	days
TOTAL						151201	seconds
						42.00	hours
						1.75	days

Table 6-3: CLEO/P Mission timeline

The assumptions resulting in the described timeline can be summarised as follows:

- CLIPPER tour is modified after COT-1, i.e. ~14 months after JOI and after 7 high v-infinity (nominal) Europa fly-bys
- The transfer uses the fly-by sequence: E-G-C-C-G-G-G-E
- The Carrier is released 1.75 days before the 2nd Europa fly-by, followed by a targeting manoeuvre (at -1.5 days) and Europa impact
- CLIPPER comes back for the 3rd Europa fly-by after 10.5 days for data relay
- A 4th Europa fly-by is assumed to transfer to Ganymede (no impact on CLEP)
- CLIPPER flies by Ganymede and continues with another mini-tour to return to a 4:1 resonance with Europa again
- Landing site is A3 (see *Mission Analysis chapter for details*)
- Visibility of CLIPPER at impact is required to acknowledge successful landing
- Visibility of CLIPPER post impact is not required, though desirable, to start data relay to CLIPPER
- Good visibility of CLIPPER at 3rd Europa fly-by is required for data relay.

Critical issues addressed but not resolved due to lack of time in the frame of the CDF study are the 2 separation events happening during the free fall:

- PDS/Penetrator Separation
- ForeBody/AftBody Separation.

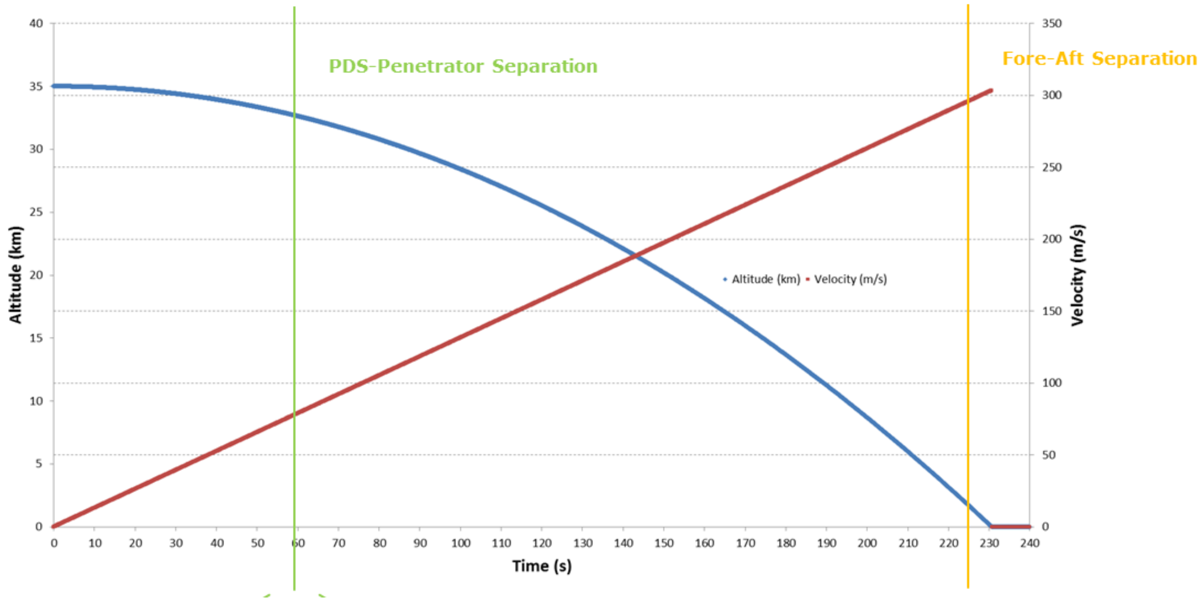


Figure 6-8: PDS & Penetrator Free Fall: separation events planning

The graph above illustrate the dependencies that shall be taken into account to determine the most adequate timing for the two events to occur.

In particular:

- PDS/Penetrator Separation:
    - Early separation reduces collision risk between PDS and Penetrator on the Europa surface
    - Late separation reduces the Penetrator off-set with respect to the vertical at impact.

The benefit of including an altimeter on the PDS to support this separation event has been identified. The altimeter would have the capability to work at ~ 2 km (7 s from surface) and could activate a timer both to separate PDS and Penetrator and also to separate Fore and Aft Body. Those separations would then be quite late in the free fall (driven by the laser altimeter working range) but very accurately triggered and compatible with the dispersions identified in the course of the study.

Drawbacks on including an altimeter in the CLEO/P baseline are the low TRL of this equipment and the ~ 1 kg mass impact on the PDS.

- ForeBody/AftBody Separation
    - Early separation guarantees deployment completion of the textile antenna
    - Late separation reduces disturbances in the Penetrator free fall dynamic.

6.4.1 Dispersions

6.4.1.1 Velocity

The dispersion issue is addressed in detail in the Mission Analysis chapter, however, for the sake of completeness, it is appropriate to report that impact velocity dispersions are rather well controlled with the last Baseline Mission Trajectory elaborated by the Mission Analysis. Moreover, they can be further decreased using accelerometer measurement: the scheduled targeting burn would be carried out, and the ΔV error would be measured during the manoeuvre. The SRM ignition time would be re-calculated according to this measurement.

The precision of the accelerometer has a threshold (reachable relative acc. is 0.52% for ΔV 10 m/s) at which the impact velocity dispersion is within ±20 m/s, which combined with the SRM dispersion, yields approx. ±50 m/s as total dispersion to be considered for the mission. This information has been used to refine CLEO/P trajectory.

The majority of the dispersions comes from the SRM burn, and the only way to further reduce such contribution is through optimisation of the SRM burn (introducing a bias, namely a burn bigger than required).

Impact angle dispersions are also reasonably well controlled, and could still be improved with a bias on the SRM burn. The landing accuracy capability of the Penetrator is estimated as ~300 km, including ephemeris error and CLEP initial state at release (this is the driver).

6.4.1.2 Angle

At the Penetrator separation the attitude has an offset from the Local Vertical, due to the following error contributions:

1. Nutation error during the SRM: this cannot be reduced because the GYROs is saturated at 600deg/s (useful range up to 100 deg/s). Therefore a Spin-Down manoeuvre is foreseen to reduce the spin rate to 10 rpm (i.e. 60deg/s) being able to measure the rotation and cancel the nutation angle with sequence of firings just before the penetrator release

   Note: *The reduction of the spin rate will also proportionally reduce the relative separation velocity of penetrator and PDS, with eventually smaller distance on surface at impact. The timing of the sequence shall be tuned to ensure sufficient margin for both aspects.*

2. NAVIGATION error: due to inaccuracies during the targeting manoeuvre. It is minimised but still existing

3. Error induced from mechanism at separation: this cannot be recovered because the penetrator is a passive element, however could be limited if separation occurred close to impact.

6.5 System Baseline Design

The baseline concept is illustrated in section 6.2, where the Fore and Aft Penetrator concept is introduced. In this section the main concept features are schematically recalled:

  - Fore body ~ 39 cm (Airbus Design)
  - Aft body ~ 9 cm for umbilical + 5 cm for textile antenna in glass (or ceramic) = 13 cm
  - Diameter ~ 19.7 cm (Airbus Design)

- Separation triggered by a timer; or by altimeter (TBC) on PDS + timer on the penetrator (later than 7 seconds from Surface = 2 km)
- Examples of separation timings are , AftBody separation at ~ 5 seconds at 1.4 km from surface (Alternatives: [4 s 1.1 km][3s 800 m] [1.7 s 50 m]), before impact

with parachute-like (no canopy) separation mechanism (pyro-mechanism, mortar, lines, umbilical containing power and comms lines)

- AftBody: Textile antenna, Umbilical cord (Comms and Power lines) ~ 5-10 m *(TBC, based on maximum penetration depth estimated with Penetrator Equations, Sandia National Laboratories)*
- Parachute lines shorter than umbilical cord (able to stand high loads in case of “pulling” of the AftBody from ForeBody)
- Textile Antenna 40 cm x 40 cm mounted on 4 x 1 m tape springs, rolled-up, contained and constrained by a cylinder with a lid that will be released by the parachute deployment system

Figure 6-9: Aft body

6.6 Mass Budget

		Column Labels				
		Function		Product		
Selection	Row Labels	Mass (kg)	Mass Margin (%)	Mass (kg)	Mass Margin (%)	Total (kg)
Product	AOGNC		0	0.61	14.05	0.69
Product	COM		0	1.90	20.00	2.28
Product	CPROP		0	8.11	6.66	8.65
Product	DH		0	2.20	20.00	2.64
Product	INS		0	2.26	20.00	2.71
Product	MEC		0	6.00	10.00	6.60
Product	PWR		0	9.40	12.13	10.54
Function	RAD	12.68	0		0.00	12.68
Function	STR	21.1	20		0.00	25.32
Product	SYE		0	0.74	0.00	0.74
Product	TC	0	0	2.22	2.88	2.28
	Grand Total	**33.78**	**12.49**	**33.43**	**11.07**	**75.13**

Harness	1.50%
Harness	1.13
System Margin	20
SRM Case	18.207
Total Dry Mass	109.71

Delta V Margin (%)	**5**
Propellant Margin	**2**

	DeltaV (m/	Isp	Prop (kg)	Prop with margin (kg)	Mass before manoeuvre (kg)	
SRM	**2691.718**	**282.3**	**195.024871**	**195.02**	**304.73**	
Targeting	**10.00**	**223.00**	**1.43**	**1.45**	**306.19**	
AOCS		**223.00**	**2.60**	**2.60**	**308.79**	**Total Wet mass (kg)**

Table 6-4: Mass budget

6.7 ΔV Budget

On the 1st Mission Analysis iteration, targeting ΔV (navigation, spin-up, spin-down) to be performed with liquid propulsion was 86 m/s, and de-orbiting ΔV to be performed with SRM burn was 2660 m/s. The early mass trade-offs indicated to adjust the Mission Analysis such that the SRM Burn ΔV could be "artificially" increased in order to use a full STAR 24 motor, without the need to offload it (implications on feasibility, delta qualification, and ultimately cost).

The result of such artificial increase was a targeting (liquid) ΔV of 51 m/s and a de-orbiting (solid) ΔV of 3175 m/s.

At IFP it became evident that with the consolidated mass budget and SRM Burn size of 3175 m/ a full STAR24 would be unfeasible. Moreover, targeting manoeuvre of 51 m/s would produce unacceptable levels of impactvelocity dispersions.

The refinement of the mass budget and the necessity to reduce the impact velocity dispersions, by reducing the size of the targeting manoeuvre, suggested to revisit the Mission Analysis strategy.

Manoeuvre	Size [m/s]
Navigation (CLIPPER)	12 x 8 = 96
Targeting	10
De-orbit	2600
TOTAL (for CLEP)	**2610**

Table 6-5: Mission Analysis Delta V Budget (Baseline selected after IFP)

The MA baseline, developed after IFP, foresees a SRM Burn of 2600 m/s and a targeting ΔV of 10 m/s, further optimised with respect to earlier iterations, in order to reduce velocity dispersions. As this refinement occurred only at IFP, the MA baseline was taken into account only at Propulsion and System Level, so that the selection of the most appropriate Solid Rocket Motor could be done, and reflected in the Final Mass Budget.

For all the other subsystems, Targeting ΔV of 51 m/s and SRM ΔV of 3175 m/s constitutes the baseline.

This gives a conservative design case (i.e. AOGNC is considering a canting angle for the thrusters of 15 deg, however with 10 m/s targeting ΔV, this canting angle might be reduced to 7-8 degrees).

The limited number of sessions allocated to the study did not allow to further flow down the new mission analysis strategy to all subsystems, however this should be done should further study be planned on the penetrator concept.

6.8 Power

Details are reported in the Power Chapter. Main Power subsystems features are shortly reported hereafter:

<u>Penetrator:</u>

- Energy requirement: 609 Wh, including 20% margin.
- Taking into consideration self-discharge, one redundant string, peak power capability and a DoD(energy) of ~70%, following battery is required:
  - Li-CFx, 2.55 V per cell. 3 cells in series x 6 strings (3s6p). 180 Ah nameplate capacity. At 7.65V = **1377 Wh nameplate energy**.
  - cells plus a small (4%) allowance for interconnection (no structure or epoxy) have a total mass of 2.0 kg (before mass margin)

PDS:

- Energy requirement: 1980 Wh, including 20% margin.
- Taking into consideration self-discharge, one redundant string, peak power capability and a DoD(energy) of ~70%, following battery is required:
  - Li-CFx, 2.55 V per cell. 5 cells in series x 9 strings (5s9p). 270 Ah nameplate capacity. At 12.75V = **3442 Wh nameplate energy**.
  - The cells plus a modest (15%) allowance for interconnection and battery structure, have a total mass of 5.4 kg (before mass margin)

6.9 Thermal

Details are reported in the Thermal Chapter. Main Power subsystems features are shortly reported hereafter:

Penetrator:

The CDF reviewed the Thermal general architecture proposed by AIRBUS, which is based on 2 enclosures, one cold that finish its operation when science is performed, and the other one decoupled focused on survivability.

Critical aspects of the design (performance after impact) have been evaluated by a Thermal Balance Test by the industrial contractor.

The necessary dissipation/heating to maintain the module above -20degC is ~4.75W.

Cool-down time to -20degC with 30degC starting temperature is ~ 15 hours.

The Energy Requirement considered in CDF design is 20 Wh/day, though this might be underestimated and would have to be more carefully assessed during later phases.

PDS:

PDS Thermal design is based on the usage of:

- MultiLayer Insulations and Kapton Foil Heaters. “Large 20 Layers MLI” (GL=0.0095W/m2K, GR=0.0075)
- Heating Power Consumption is 25 W, based on the following contributions:
  - Leakage Through MLI: **~11W** (surface considered: structure + external tanks = ~2m2).
  - Leakage Through Small Thrusters: **1.5W per Thruster**
    - Ref. Reduced Thermal Model from Lunar Lander B1.
  - Leakage Through Main Engine: **5W**
    - Ref. Reduced Thermal Model from Lunar Lander B1.
  - Budget allocation to the Penetrator: **3W**

6.10 Communications

Details are reported in the Communication Chapter. Main Comms subsystems features are shortly reported hereafter:

Based on the mission scenario:

- Fly-by 1 : No comms, no science
- Fly-by 2 : Release and impact - Impact will happen at the pericentre of this fly-by, with a very short communication slot to confirm successful impact.
- Fly-by 3 : 10.5 days after Fly-by 2 – **Communications**

The following Data generation is assumed for CLEO/P:

- E_PAC : the total data volume is 3.048Mbit. The whole science seq. is done in 2426s,
- MSEIS : a data volume of 0.731 Mbit/day is generated for 7 days since impact
- (Some housekeeping can be generated inside the penetrator (margin on top of this TM)

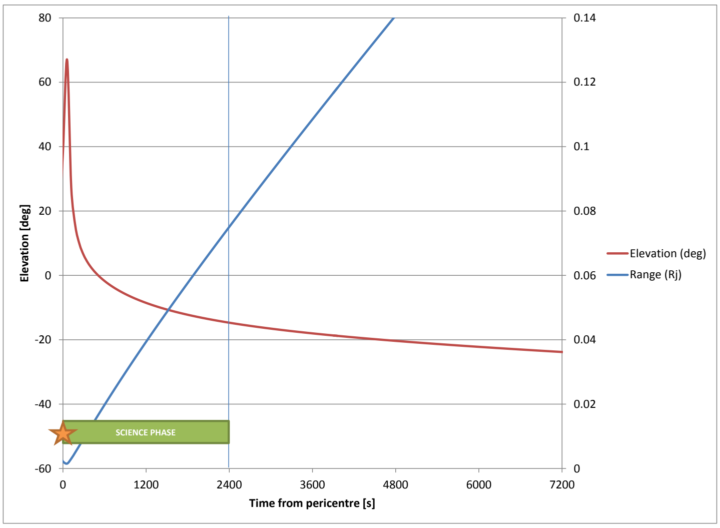


Figure 6-10: CLIPPER visibility Fly-by 2

CLIPPER visibility above the local horizon would end before initial science phase completes, therefore it is not possible to download science data during 2nd fly-by.

The short visibility can be used to download some initial TM and data to check status after impacts. Link budget is not critical for this phase as range is small and elevation is high.

During the 3rd Flyby, science data shall be relayed to CLIPPER.

- Contact is considered feasible when:
  - Margin > 3dB
  - Elevation > 30deg
- Following parameters are considered:
  - Data Rate : 3kbps
  - TX Power : 1W
- Resulting contact time is about 46min

- Achievable data volume is about 8Mbit
  - This is sufficient to download all E_PAC TM (in case first fly-by not successful) and the seismometer TM with good margin.

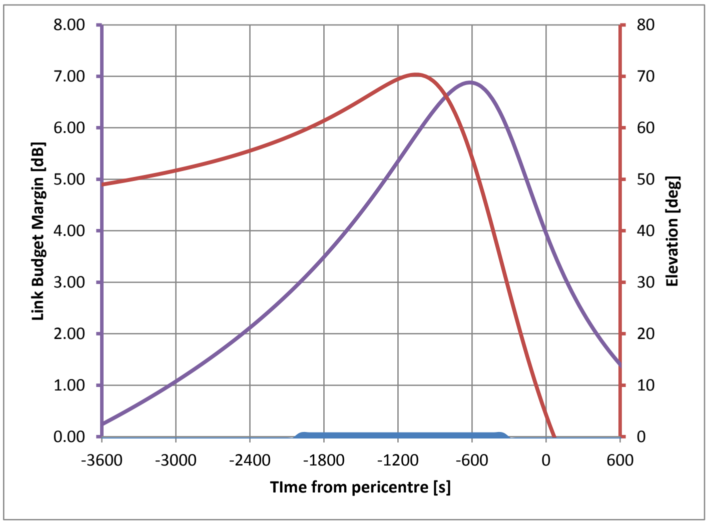


Figure 6-11: CLIPPER visibility Fly-by 3 - Link budget

6.11 Propulsion

CLEO/P has both a liquid and a solid propulsion system:

- Liquid (for targeting, rate dumping, spin-up, spin-down) is based on:
  - 1 PEPT 230 tank, positioned on central axis
  - 3 x 20 N thrusters for main ΔV
  - 2 x 20 N + 2 x 20 N thrusters for spin / de-spin (spin / de-spin propellant: 4.1 kg)

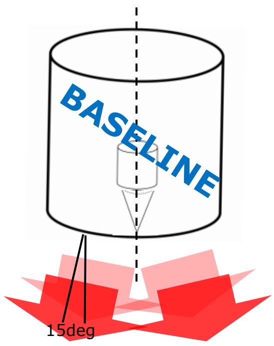


Figure 6-12: PDS Thrusters baseline architecture

- Solid (for de-orbiting) is based on:
  - STAR 24:
    - 199.9 kg Solid Propellant
    - 18.2 kg case
    - Isp 282.9 s

6.12 Radiation Analysis

6.12.1 Radiation During Transfer

6.12.1.1 Penetrator:

A sectoring analysis based on the following assumptions (CLIPPER shield effects not considered):

- case thickness 5 mm stainless steel
- additional 5mm Al casing for cold bay and warm bay

Indicates that Doses at the center of the Penetrator bays are ~50 krad(Si) [without factor 2 applied]

CDMU TID sensitivity is 50 krads therefore it requires further shielding, either with an Al case increase from current 3 mm to 11 mm (or equivalent stainless steel thickness) or by spot shielding for sensitive equipment only, which has been selected as baseline because less massive.

6.12.1.2 PDS:

TID levels require Al shielding for the following (most sensitive) items:

- OBC (37 mm Al; TIDs 50 krad)
- Gyro (31 mm Al TIDs 70 krad)
- STR (21 mm Al TIDs 150 krad)

This results in a radiation shielding mass of 12.68 kg, as indicated hereafter:

			thick	Mass
	Micro STR		21	0.9197874
	Micro STR		21	0.9197874
	GYRO		31	0.381016441
	GYRO		31	0.381016441
	GYRO		31	0.381016441
	GYRO		31	0.381016441
	ACCEL		37	0.2571426
	ACCEL		37	0.2571426
	ACCEL		37	0.2571426
	PDS OBC		37	7.030130376
	Penetrator CDMU		8	1.52002819
SHIELDING MASS TOTAL				**12.6852269**

Table 6-6: Shielding Mass Budget

6.12.2 Radiation on Europa Surface

A timeframe of ~ 10.5 days on Europa equates to:

- TID ~17 krad(Si) behind 12 mm Al
- TID ~12 krad(Si) behind 15 mm Al

However, no radiation sensitive components are left on Europa surface (only Textile Antenna) therefore no shielding is required.

6.12.3 Radiation in Ice

Under the assumption that ice's density is 1 g/cm^3, 50 cm ice are equivalent to an aluminum thickness of ~18 cm (at 2 m there would be radiation levels similar to Earth).

10 days period equates to a TID of ~150 rads(Si), which is negligible compared to the mission dose.

Thus no further radiation shielding is required for this mission segment.

6.13 Interface to Clipper

6.13.1 Accommodation

Clipper/PSD separation mechanism details are reported in the Mechanisms Chapter. Main subsystems features are shortly reported hereafter:

- 2 mounting points in ADS design, however 3 HDRMs could be needed
- NEA actuator based on a cup cone interface with push off spring
- Arquimea REACT selected as NEA device
- Approximate mass of 0.65 kg per HDRM.

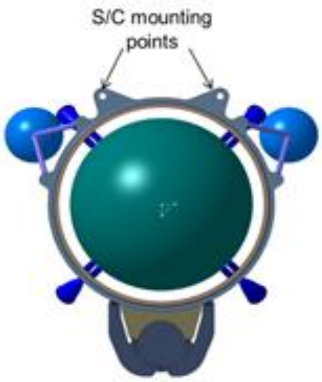


Figure 6-13: Mechanism I/F to CLIPPER (Courtesy of Airbus DS)

6.13.2 CLEO/P Impact on CLIPPER TOUR

Note that the Clipper tour modification feasibility is TBC on Clipper mission side.

Current Baseline:

- 24 additional perijove passages at Europa radius for the modified tour
- V-infinity at release of the PDS is 1.68 km/s
- The additional mission duration is 150+45+150=345 days

Best Case Scenario (TBC by analysis):

- Relative phasing of the Moons (i.e. the geometry) could be improved by staying for some time in the original orbit after COT-1 (if nothing else helps, NASA could consider shifting their tour by a few Europa revolutions)
- Transfer duration would go from 150 to 80 days in the best case: i.e. about 16 additional perijove passages at Europa altitude and a total duration of 80 + 45 + 80 = 205 additional days for Clipper

6.14 Penetrator Equations Annex

6.14.1 Penetrator Equations (October 1997)

Penetration Equations for Ice and Frozen Soil

For $V < 61$ m/s,

$$D = 0.00024\, S\, N\, (m/A)^{0.6} \ln(1 + 2.15V^2\, 10^{-4}) \ln(50 + 0.29m^2)$$

For $V \geq 61$ m/s,

$$D = 0.0000046\, S\, N\, (m/A)^{0.6} (V - 30.5) \ln(50 + 0.29m^2)$$

A	Cross sectional area, m^2
CRH	Caliber Radius Head, tangent ogive nose shape
d	Penetrator diameter, m
D	Penetration distance, m
L	Penetrator length, m
L_n	Penetrator nose length, m
m	Mass of penetrator, kg
m/A	Weight (mass) to Area ratio, kg/m^2
N	Nose performance coefficient
V	Impact velocity, m/s
S	Penetrability of target, S-number, dimensionless

Nose Performance Coefficient, N

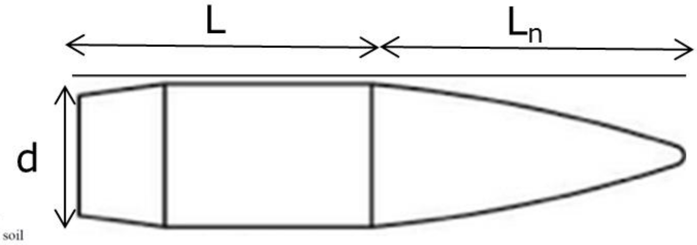
For tangent ogive nose shapes, either of the following two equations may be used:

$$N = 0.18\, L_n/d + .56$$

$$N = 0.18\, (CRH - .25)^{0.5} + .56$$

For conic nose shapes:

$$N = 0.25\, L_n/d + .56.$$



S-number for Ice/frozen soil

Both fresh water ice and sea ice will normally have an S-number of 4.5 ± 0.25. Completely frozen saturated soil will have an S-number of 2.75 ± 0.5. The S-number of partially frozen soil may be as high as 7.0, but the transition from partially frozen to unfrozen soil is not well defined.

6.14.1.1 Equations Applicability Boundaries

The following are assumptions or limitations which apply to all the penetration equations:

1. The penetrator remains intact during penetration.
2. The penetrator follows a basically stable trajectory. (No large changes in direction, and no tumbling or J-hook during penetration.)
3. The impact velocity is less than 4000 fps. In hard materials, the "intact penetrator" assumption probably governs the upper allowable impact velocity. In soft materials, there is no data at very high impact velocity for equation validation, so the upper limit on impact velocity is not known.
4. When the penetration depth is less than about 3 calibers (penetrator diameters), the equations may be questionable.
5. The equations are not valid for water or air penetration.
6. The equations are not applicable for armor penetration (eg, not for metals, ceramics or materials other than those specifically listed).
7. Minimum penetrator weight: about five pounds for soil and ten pounds for rock, concrete, ice and frozen soil.
8. The lower velocity limit of applicability has never been defined. In fact, limitation "4" above is likely the more realistic lower velocity limit in most targets.

Other limitations may be given as applicable to specific equations or techniques.

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7 PLANETARY PROTECTION

7.1 Requirements

Due to the Europa fly-bys, and potentially Mars gravity assist, the NASA Clipper mission would be a Planetary Protection Category III. The CLEO/P concept, however, intends to land/impact on Europa and would therefore be Planetary Protection Category IV. In line with this category, the following planetary protection requirements of RD[7] are applicable to the CLEO/P concept:

Requirements	**Note for CLEO/P**
5.1a, b, d, e, f	
5.2.1a	
5.2.2a	
5.2.3a, b	Protected solar system bodies are Europa and Mars; prior to release of the penetrator the analysis to be covered by NASA for Mars and Europa; post-release of the penetrator the analysis for Europa has to be covered by ESA for all elements that are not intended to impact on Europa
5.3.2.1d	To be covered by NASA
5.3.2.1e.1	To be covered by NASA
5.3.3.2a, b	Suggest to focus on terminal sterilization of the penetrator system (alternative is sub-system sterilization and aseptic assembly) and use of a bio-shield to protect from re-contamination; for all elements that are not intended to impact on Europa, assess the probability of accidental impact for a time period until the most shielded parts of the hardware reaches an ionizing radiation dose of at least 25 kGy
5.4	
5.5	
5.6a, b	
5.7	
Annex A, B, C, D, E, F (if applicable), and G	

7.2 Design Drivers

The major design drivers are:

1. Compatibility of the flight hardware to active sterilisation at the highest integration level
2. Recontamination protection of the flight hardware

Evaluating the compatibility of the flight hardware with sterilisation processes requires usually qualification at sub-system or system level to ensure that all aspects (e.g., different coefficients of thermal expansion) are covered. Although for most hardware a delta-qualification could be sufficient, some hardware might require dedicated developments.

Recontamination barriers are mostly simple sub-systems for ground and flight operations.

7.3 Resources for Implementation

Bioburden control for a spacecraft requires some dedicated infrastructure (i.e. bioburden controlled cleanrooms, microbiological laboratory, sterilisation equipment), development of re-contamination barriers, and additional personnel to develop, implement and monitor the bioburden control throughout the project phases. See RD[8] for more information.

All these aspects have been developed in Europe in the frame of the ExoMars program.

7.4 Technology Requirements

To test the compliance of flight hardware or sub-systems with active sterilisation processes like dry heat RD[9] or room temperature hydrogen peroxide gas RD[10] would require the use of models that are similar to qualification models RD[11].

Application of active sterilisation processes could reduce the TRL level of the individual hardware or sub-system.

8 RADIATION

8.1 Assumptions and Trade-Offs

The orbit used for the calculation of the radiation dose analysis consists of two parts; the firs leg is referred to as the '6C2' trajectory, whereas the second leg is the CLEP resonant phase 02. The altitude as function of time is shown in Figure 8-1. This analysis assumes a release of CLEP after COT-1 which is worst case in terms of radiation dose. An earlier release (during COT-1) would significantly decrease the accumulated dose.

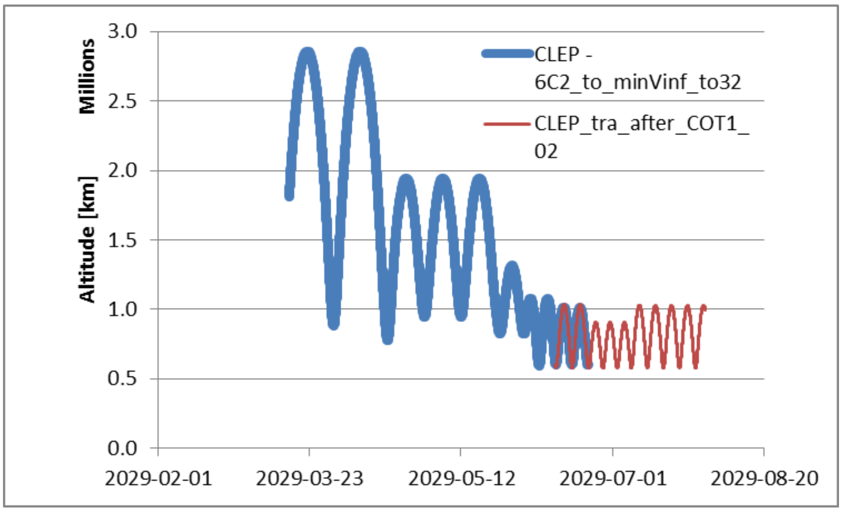

Figure 8-1: CLEP altitude as function of time

8.2 Radiation Dose Analysis

Based on the previous trajectory, the radiation dose is calculated assuming a spherical spacecraft, as function of shielding thickness. Figure 8-2 shows these results.

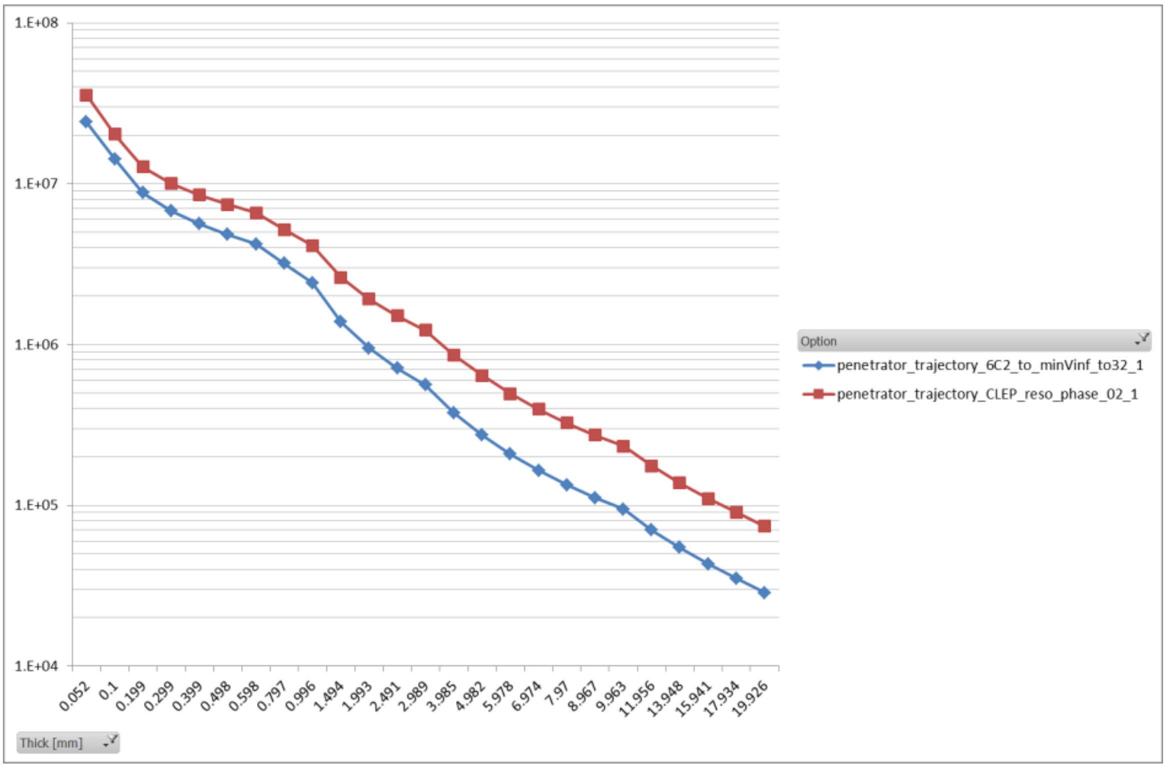

Figure 8-2: CLEP dose as function of shielding thickness

8.3 Solar Cell Degradation

To support the power subsystem design, the solar cell degradation is computed and shown in Figure 8-3.

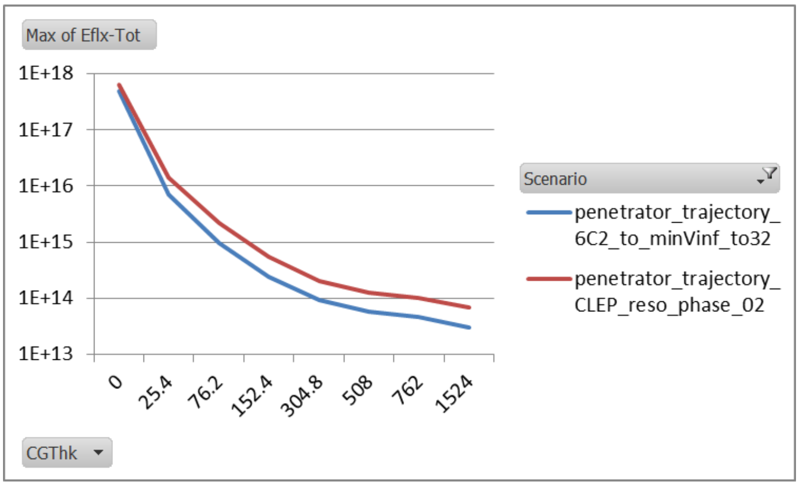

Figure 8-3: CLEP solar cell degradation

8.4 Sector Analysis

A ray tracing model was created by modelling the penetrator (see Figure 8-4) as:

- 5 mm thick Stainless steel case
- 3 mm thick Aluminium instrument boxes
- Using target locations at centre of cold bay and warm bay.

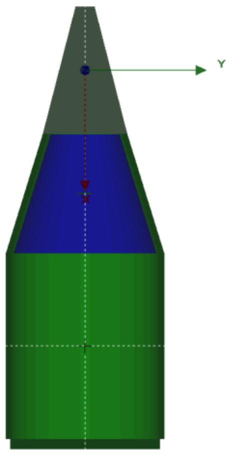

Figure 8-4: CLEP penetrator model for ray tracing

The doses calculated at the bays are:

- Cold bay: ~21 krad(Si)
- Warm bay: ~21 krad(Si)

Doses for the equipment are shown in Table 8-1 below:

AOGNC	50 krad(Si)	> 20 mm
OBC	50 krad(Si)	> 20 mm
Micro STR	150 krad(Si)	12 mm
Gyros	70 krad(Si)	18 mm

Table 8-1: TID for CLEP equipment

8.5 Sub-Surface Dose

The final analysis was to determine the average and maximum dose per day at sub-surface conditions, i.e. the penetrator buried within the ice, which provides shielding. The results of this analysis is shown in Figure 8-5.

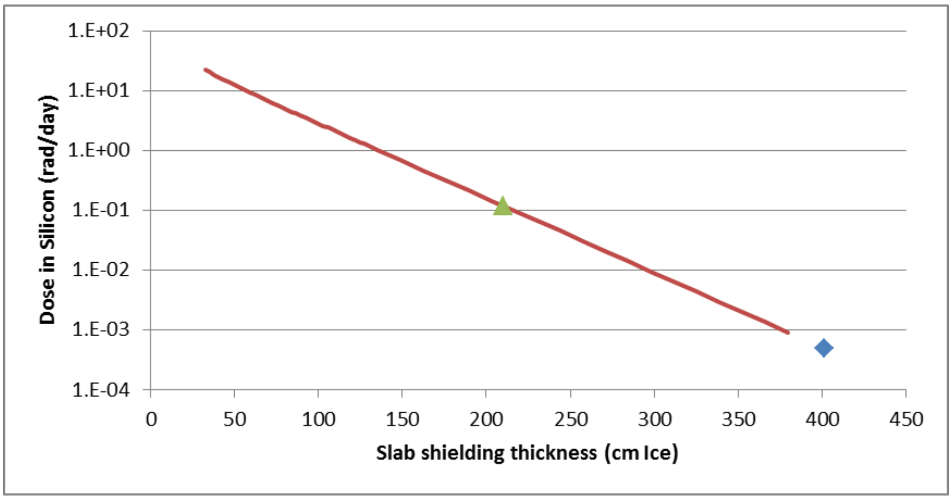

Figure 8-5: CLEP penetrator sub-surface dose

9 CONFIGURATION

9.1 Requirements and Design Drivers

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
CFG-010	CLEP spacecraft shall interface with CLIPPER laterally on tbd interface location	
CFG-020	CLEP is an assembly of a Penetrator and a Penetrator Delivery System. PDS has a function to deliver the penetrator to Europa at 35 km altitude. Airbus design concept shall be used for the configuration	
CFG-030	An aftbody structure that contains a textile antenna is added to the original Airbus Penetrator design (Forebody)	
CFG-040		

9.2 Assumptions and Trade-Offs

PDS and penetrator design is derived from the Airbus design concept.

9.3 Baseline Design

Figure 9-1 shows a possible interface location of CLEP on the CLIPPER S/C

Assembly of CLEP spacecraft consists of

- Penetrator Delivery System
- Penetrator

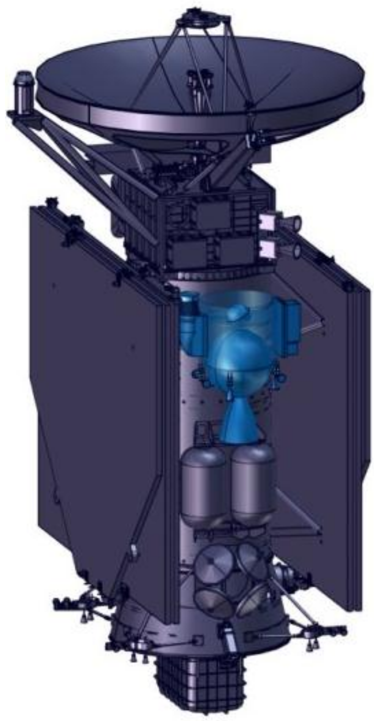

Figure 9-1: CLEP on CLIPPER (Courtesy of Airbus DS)

9.3.1 Penetrator Deliver System

PDS cylindrical structure of 730mm diameter accommodates mainly the propulsion subsystem equipment:

- Propellant tank PEPT 230 with 230mm diameter
- Solid Rocket Motor STAR24 with 622mm diameter

Height of the cylindrical structure is driven then by the total length of PEPT230 and STAR24. This gives overall dimension of cylinder as shown in the

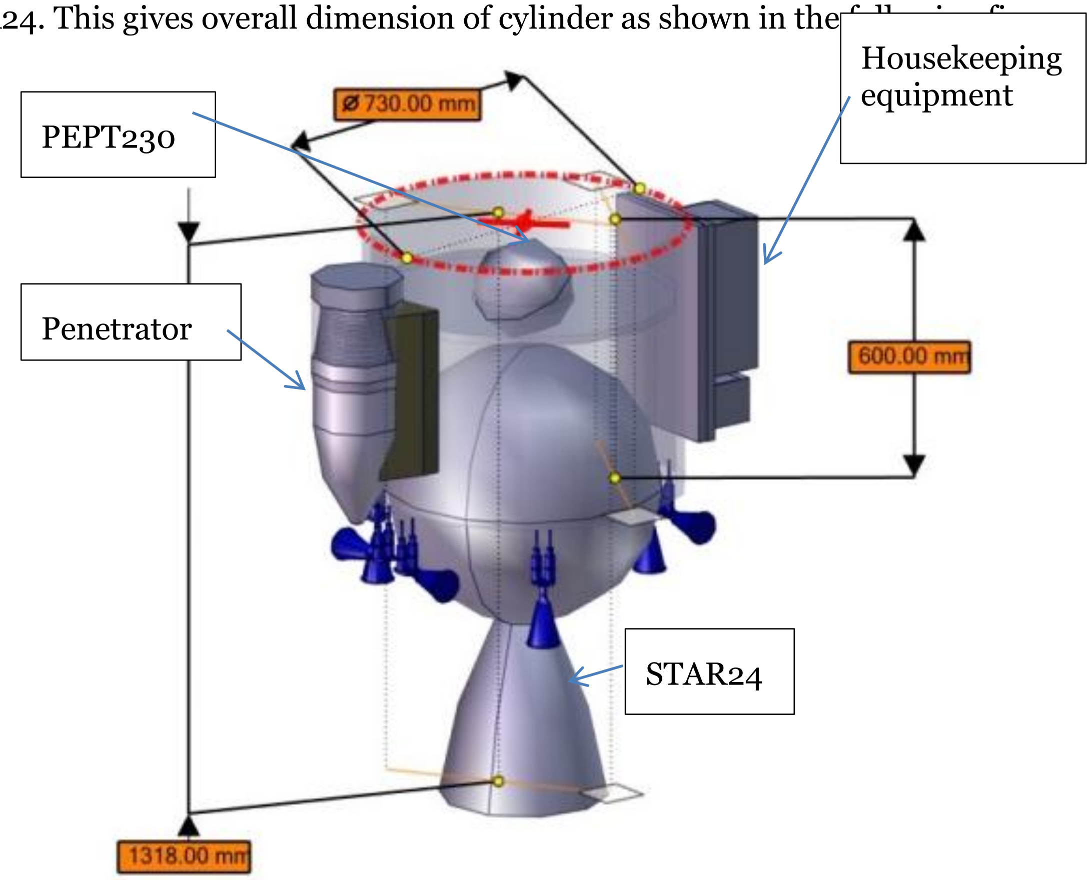

Figure 9-2: Penetrator on PDS

PDS accommodates the penetrator laterally on one side. On the opposite side all required housekeeping equipment for POWER, GNC and Data Handling will be mounted on a common platform as shown in Figure 9-2 and Figure 9-3.

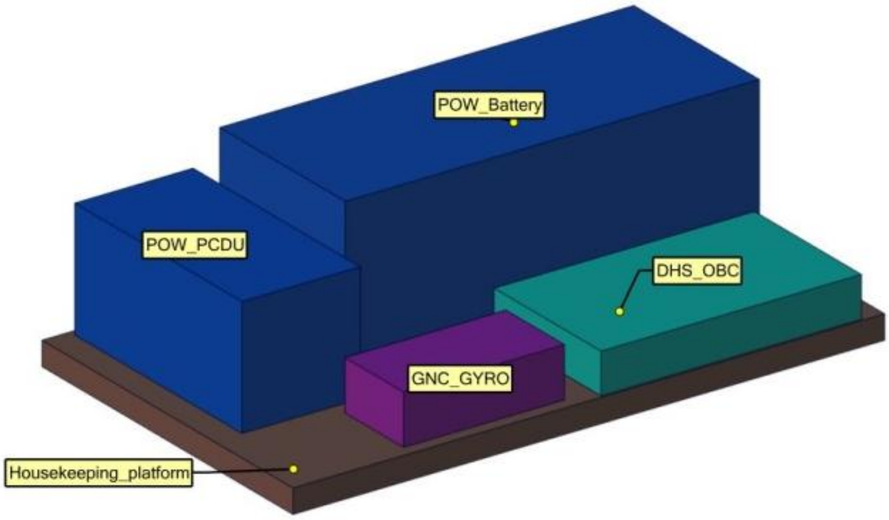

Figure 9-3: CLEP - Housekeeping equipment

No detailed design of support structures were done during the study.

9.3.2 Penetrator

Penetrator Frontbody is taken from Airbus design as shown in Figure 9-4.

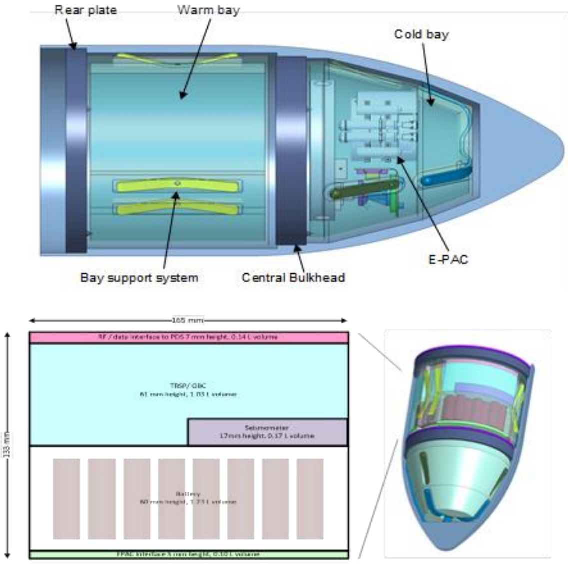

Figure 9-4: Airbus Penetrator design (Courtesy of Airbus DS)

Penetrator aftbody accommodates the textile antenna. Final aftbody shape needs to be studied further. Figure 9-5 illustrates design concept of the Penetrator including the aftbody

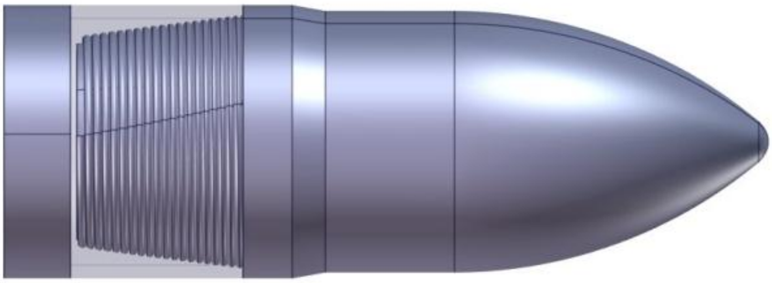

Figure 9-5: Penetrator design including the aftbody

9.4 Overall Dimensions

Following figures show the overall dimension of the PDS and the Penetrator.

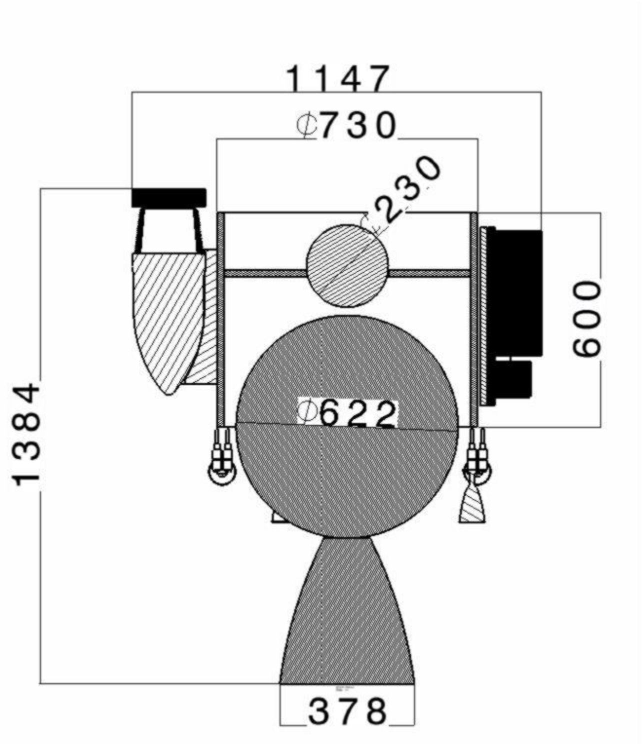

Figure 9-6: CLEP overall dimension

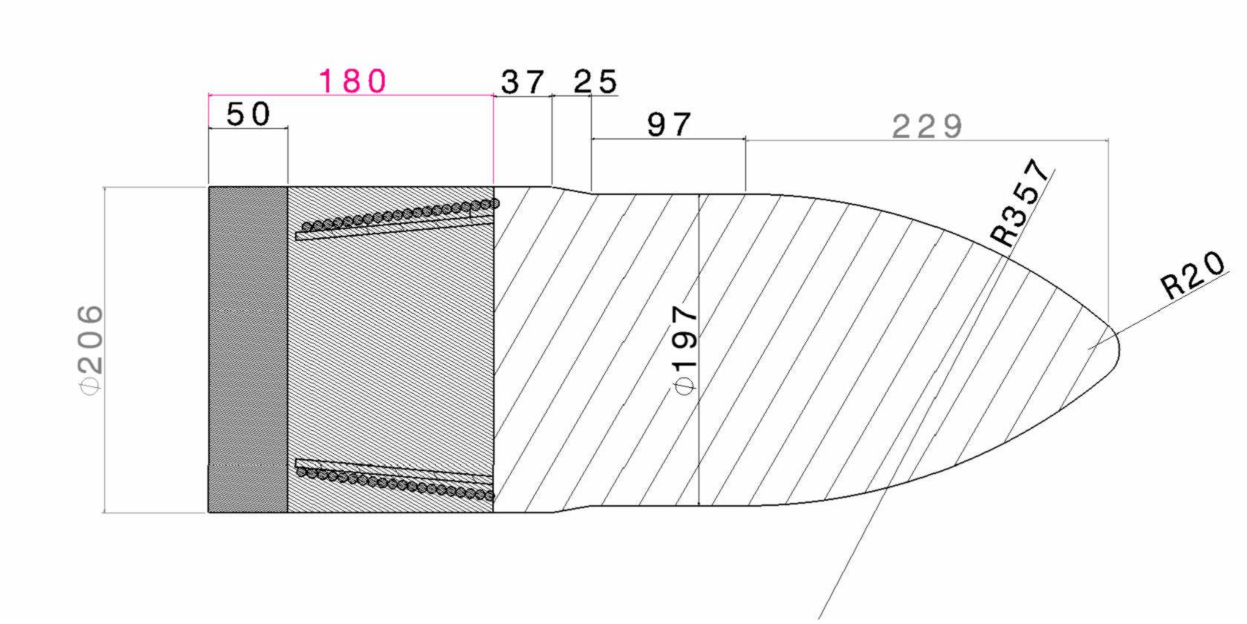

Figure 9-7: Penetrator overall dimension

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10 STRUCTURES

10.1 Requirements and Design Drivers

CLEP is composed of two separate bodies, the penetrator and the penetrator delivery system. As an assembly, the system shall be compatible with launch and orbit environments. Therefore, the applicable requirements to the structure design, similar to those of the orbiter configuration, are the following:

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
STR-010	The first axial and lateral frequency in stowed configuration shall be above TBD Hz	
STR-020	The spacecraft shall be compatible with the payload allocated volume as applicable.	
STR-030	The spacecraft shall be compatible with the Clipper environment (TBD), as applicable, at any stage of the mission.	
STR-040	The spacecraft shall be compatible with Clipper interface adapter (TBD).	

In addition, the penetrator itself shall meet the following requirements:

SubSystem requirements		
Req. ID	STATEMENT	Parent ID
STR-050	The penetrator shall survive the mechanical environment at impact	
STR-060	The penetrator shall travel more than 500mm under the surface of Europa.	
STR-070	The crater generated at impact shall not endanger the communications between the penetrator and Clipper	

The penetrator delivery system and the initial penetrator designs are based on the configuration from Airbus contract. Therefore, for this study, the structures evaluation has focused on the analysis of the penetrator impact in order:

- To evaluate the possible dispersions in the crater size;
- To determine possible penetration depths, and
- To assess the effectiveness of different after body release concepts.

These items are considered the most critical for the mission feasibility and for the overall mission concept. The structural feasibility of the penetrator delivery system and for the penetrator architecture is considered covered by the Airbus study.

10.2 Assumptions and Trade-Offs

10.2.1 Assumptions

In order to evaluate the consequences of the impact, a non-linear explicit finite element analysis has been performed. This model has been compared with the Airbus penetrator full scale test results performed in the frame of the penetrator contract.

The initial penetrator model has been based on the Airbus design, with modified after body shapes and impact speeds of 300m/s. The Europa terrain models are based on validated terrestrial ice non-linear models for different applications and reported in RD[12] and RD[13].

10.2.2 Sensitivity Analysis

The following cases have been performed in the sensitivity analysis:

- Vertical impact analysis
- 22.5 degrees with 4 degrees/sec yaw speed (Full scale penetrator case)
- 22.5 with conical after body (Spin stabilised penetrator impact).

For the full scale penetrator case the following material models have been evaluated:

- Equation of State model with shear stiffness and rate dependent plastic failure (ductile model)
- Elastic model with Mohr failure criteria (brittle model).

10.2.2.1 Vertical impact with ductile material model

The vertical impact with a ductile failure model shows a moderate penetration depth which would be compatible with a permanently fixed antenna in the after body.

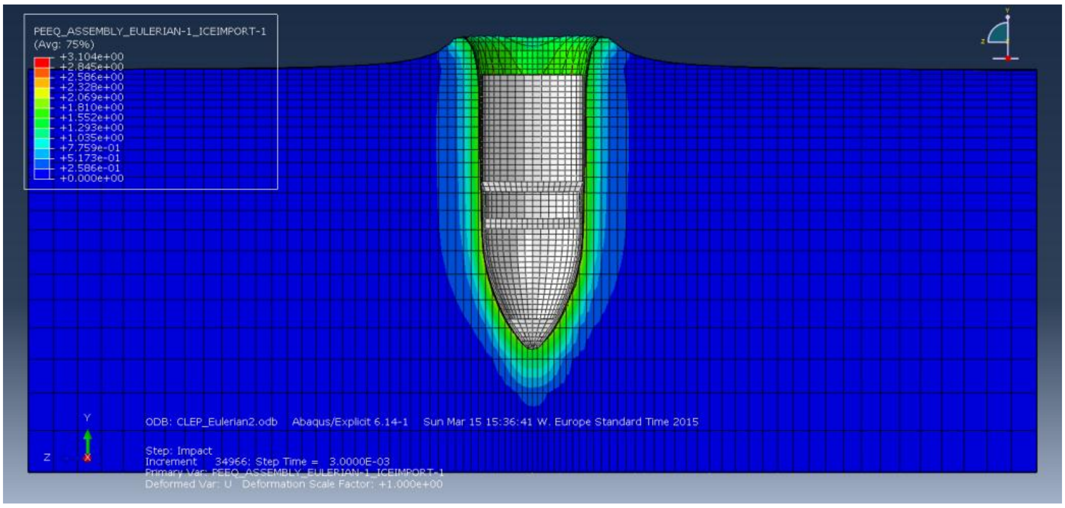


Figure 10-1: Vertical impact penetration with ductile ice model

This ice model is based on hail impact tests RD[12] and shows a ductile behaviour of the material which effectively reduces the speed of the penetrator.

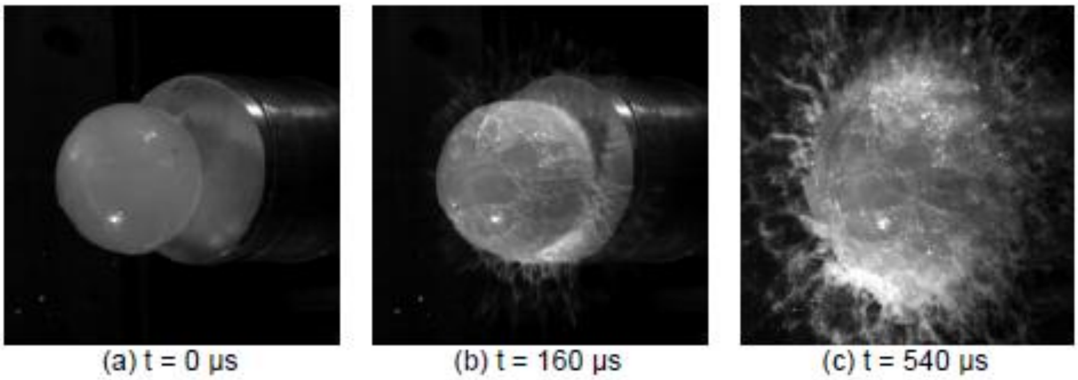


Figure 10-2: Hail Impact test results RD[12]

Such situation, while being desirable, contrasts with the results of the penetrator full scale results performed within the Airbus contract. Therefore, the material model is not further considered in the assessment. Nevertheless, it is recommended to perform full scale test on terrestrial ice terrain to verify the validity of the simulation.

10.2.2.2 Inclined impact with brittle material model

A second analysis has been performed with a brittle material model which degrades the material shear stiffness once a certain level of stress is reached (Nisja, 2013). This simulation considers the initial conditions of the Airbus full scale test; i.e. inclination of 22.5 degrees and 14.3 rad/s rotation.

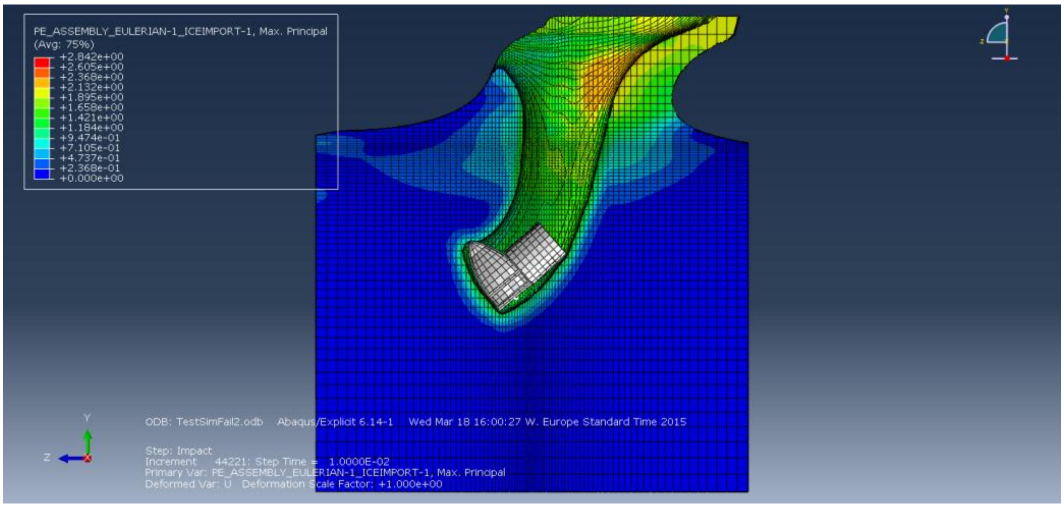


Figure 10-3: Oblique impact with detachable after body and brittle ice model

The results show a better agreement with the observed behaviour of the full scale tests, although still showing a higher resistance to the penetration as in the full scale tests.

This analysis shows that the crater size and plume could be several times the diameter of the penetrator and the penetrator can travel several meters under the ice. Such scenario would set several constraints on the communication system and therefore requires further investigation.

10.2.2.3 Inclined impact with shaped after body

Using the above model, an analysis of a spin stabilised penetration with a shaped detachable after body has been performed.

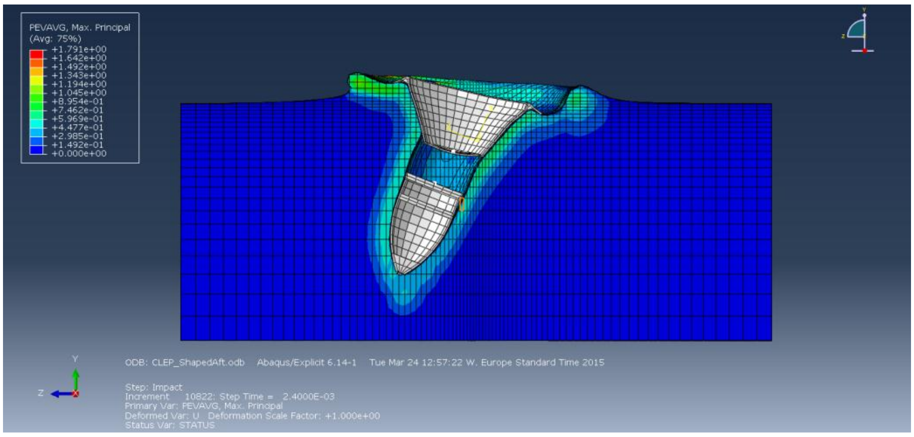


Figure 10-4: Oblique spin stabilised impact with conical after body after 2.4ms

The results show that such a shaped after body would effectively detach from the fore body, as shown in the figure. However, the crater and debris cloud are still several times the diameter of the penetrator and cannot ensure that the antenna field of view is free after impact. Also, this analysis shows that the spin of the penetrator is effective in maintaining the attitude of the penetrator during penetration despite the oblique impact.

10.2.3 Impact Analysis Conclusion

From the sensitivity analysis the following conclusions have been drawn:

- The material model assumptions are key in the impact analysis. Brittle and ductile material models lead to very different crater size and penetration depth. Therefore, considering the unknown behavior of Europa soil, the final penetrator design must be robust to any of the soils conditions.
- In addition, it will be necessary to perform test with several impacts on ice in different conditions (on terrestrial ice terrain) to determine either brittle or ductile ice behavior under these impact conditions.
- It is also observed that the penetrator spin stabilises the impact and avoids rotations during the penetration. This is beneficial in case oblique impacts.

10.3 Structures Mass Budget

The structural subsystem is assumed to be the one baselined in the Airbus Data Package, and no new design has been proposed during the 4 CDF Sessions dedicated to the penetrator. Only the AfterBody structural mass has been estimated as 0.2 kg before maturity margin.

The contribution of the Structures subsystem to the Mass Budget is 25.32 kg, resulting from:

- AftBody Structure: 0.2 kg + 20% maturity margin = 0.24 kg
- ForeBody Structure: 13.9 kg + 20% maturity margin = 16.68 kg
- PDS Structure: 7 + 20% = 8.4 kg

11 MECHANISMS

11.1 Requirements and Design Drivers

The main design drivers taken in consideration for the dimensioning of the mechanisms are:

- Mass optimisation
- Impact velocity of penetrator ~ 300m/s
- Penetrator configuration on one side of the PDS and the release time of the penetrator from the PDS
- Release of the antenna prior to impact.

11.2 Assumptions and Trade-Offs

The following assumptions have been taken in consideration:

- Separation velocity of the penetrator from the PDS~ 5m/s
- Separation time of the penetrator from the PDS for a correct deployment ~ 10 ms.

For the Fore-Aft Penetrator separation mechanism a two solutions has been considered:

- Parachute Deployment System
- Mechanism composed on a pyro nut and a spring

As baseline it has been considered the Parachute Deployment System because of the possibility of this solution to accommodate a less complex antenna deployment mechanism.

11.3 Baseline Design

11.3.1 Clipper – PDS Separation Mechanism

For the separation mechanism of the PDS from Clipper the design proposed by ADS (Airbus Defence and Space) has been considered as baseline. ADS design only foresees two mounting points (Figure 11-1), however, it is considered that and extra point could be needed due to the shear loads in the holding points. This should be analysed more in detail in a later phase.

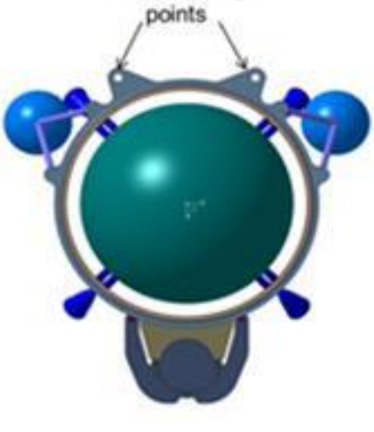


Figure 11-1: PDS to Clipper mounting points (Courtesy of Airbus DS)

The separation mechanism selected is a Hold Down and Release Mechanism (HDRM) based on a cup- cone interface and push off springs in each of the mounting points.

The Non Explosive Actuator selected is the “REACT” developed by Arquimea.

Figure 11-2: REACT Actuator

The configuration of the mounting points, where the HDRM’s are mounted, shall be designed according to the correct separation trajectory of the PDS after release, so the push off springs can give the PDS the delta velocity necessary in the correct direction.

The specifications of the REACT 35 KN can be summarised as follows:

- Maximum Preload: 35 KN
- Operating Temperature [°C]: -40 +70
- Power consumption [W]: 30@4.1A and 23 °C
- Envelope (Ø x L) [mm]: 78 x 78.5
- Mass [g]: 412

11.3.2 Penetrator-PDS Separation Mechanism

The design proposed by ADS comprises two actuated clamps as shown in Figure 11-3. However, it is considered that this mechanism, that seems to rely only on friction to hold the penetrator, can have disadvantages in terms of deployment synchronisation and interference of the clamps with the penetrator while deploying due to the spinning of the PDS.

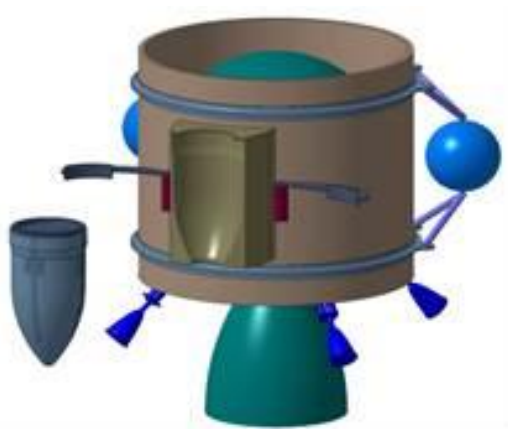


Figure 11-3: Separation Mechanism (Courtesy of Airbus DS)

In order to deploy the penetrator without having any interference with the clamps, as the Penetrator will be ejected from the PDS with a tangential velocity of 5m/s, It has been assumed that an approximate release time of 10ms will be needed.

The fact that the penetrator is mounted on one side of the PDS implies having a more complex mechanism compared to a configuration with the penetrator mounted on the rotation axis of the PDS.

Instead of selecting the design proposed by ADS, a new concept is proposed as baseline.

The Penetrator is mounted on the cradle for stability during the ejection and is held by a single HDRM based on a pyrotechnic device as shown in Figure 11-4. The HDRM consists of a cup-cone interface with the pyro nut on the PDS side. On the HDRM side only the interface with the HDRM is mounted. After release, this part will remain on the penetrator, for that reason a symmetric part has to be attached to the other side of the penetrator.

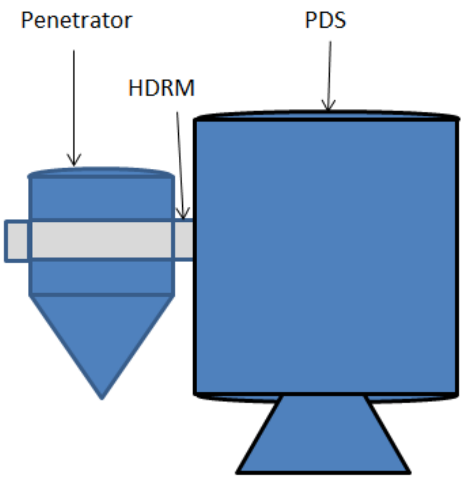


Figure 11-4: Penetrator HDRM

To reduce the impact on the shape of the penetrator of the part remaining after the release, this part can be integrated in the shell of the penetrator.

11.3.3 Fore-Aft Penetrator Separation Mechanism

The separation of the after body from the fore body is considered as the separation of the textile antenna from the fore body, that needs to be released before impacting the ice.

As baseline, a parachute deployment system has been selected. Figure 11-5 shows the typical configuration of a Parachute Deployment System.

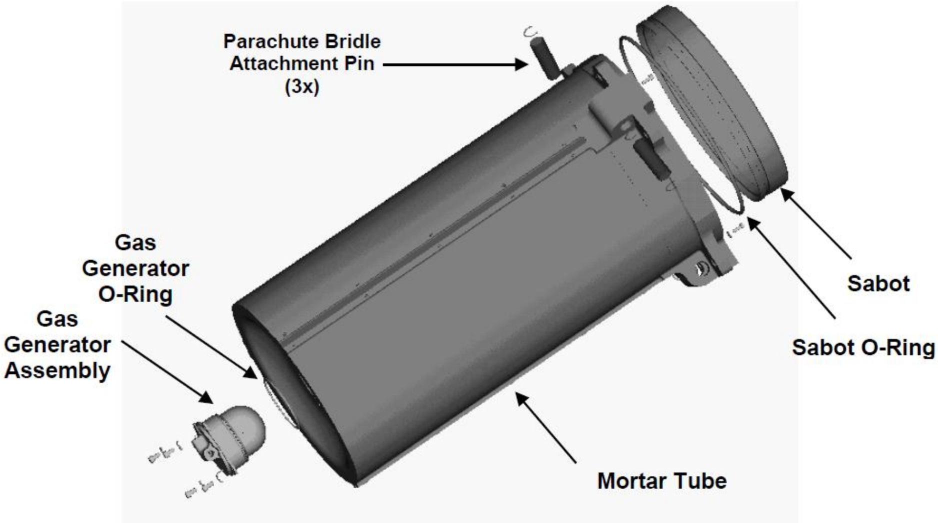


Figure 11-5: Typical configuration of a Parachute deployment system

The deployment system is based on a gas generator with a pyro initiator, which is very common in space missions and can be easily scalable. The proposed baseline is a Parachute Deployment Device based on a gas generator (Figure 11-6) being currently developed by Aerospace Propulsion Products B.V & Vorticity Ltd.

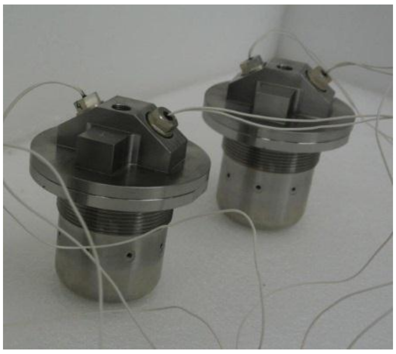


Figure 11-6: Gas generator (APP & Voticity)

Parachute deployment systems normally deploy parachutes with a velocity up to 30 m/s, however for this application the velocity has to be reduced to approximately 1 m/s. The release time can be approximately 15 ms, however the deployment of the antenna after release will be in the order of several seconds. As the release velocity has to be reduced the release time can increase from this value.

11.3.4 Antenna Deployment Mechanism

To assure the proper deployment of the textile antenna, 4 tape springs in cross configuration will be accommodated with the textile antenna giving also enough stiffness to the antenna after deployment. The textile antenna needs to be rolled inside the cylinder together with the tape springs.

The four tape springs need to have a length of 1 m each to assure that the textile antenna will stay in the surface of the ice and will not follow the penetrator into the penetrated ice. The tape springs need to dimensioned to have the correct stiffness and to be able to deploy the antenna after separation. An example of tape springs developed by SSTL for the solar sails deployment mechanisms is shown in Figure 11-7.

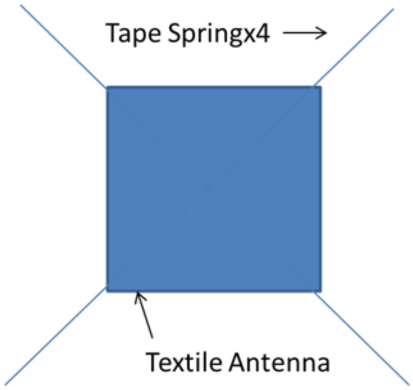


Figure 11-7: Tape spring(left) Textile antenna deployed (right)

11.4 List of Equipment

	mass (kg)	mass margin (%)	mass incl. margin (kg)
⊞ADM (Antenna Deployment Mechanism)	0.30	10.00	0.33
⊞CPSM_1 (Clipper-PDS Separation Mechanism)	0.70	10.00	0.77
⊞CPSM_2 (Clipper-PDS Separation Mechanism)	0.70	10.00	0.77
⊞FAPSM (Fore-Aft Penetrator Separation Mechanism)	1.30	10.00	1.43
⊞PPSM (Penetrator-PDS Separation Mechanism)	3.00	10.00	3.30
Grand Total	6.00	10.00	6.60

Table 11-1: List of Equipment

11.5 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed
- Technologies available within European non-space sector(s)
- Technologies identified as coming from outside ESA member states.

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
Antenna Deployment Mechanism	Tape Spring	3		
Clipper/PSD Separation Mechanism	SMA	6		
Penetrator-PDS Separation Mechanism	Pyro-nut	9		
Fore-Aft Body Separation Mechanism	Gas generator	5		TRL 3 for the application with the antenna deployment

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
				mechanism and the reduced velocity

12 PROPULSION

12.1 Requirements and Design Drivers

CLEO/P mission concept has been iterated three times during the course of the study, producing varying ΔV requirements to be satisfied by the propulsion subsystem.

On the 1st Mission Analysis iteration, targeting ΔV (navigation, spin-up, spin-down) to be performed with liquid propulsion was 86 m/s, and de-orbiting ΔV to be performed with SRM burn was 2660 m/s. The early mass trade-offs indicated to adjust the Mission Analysis such that the SRM Burn ΔV could be "artificially" increased in order to use a full START 24 motor, without the need to offload it (implications on feasibility, delta qualification, and ultimately cost). The result of such artificial increase was a targeting (liquid ΔV of 51 m/s and a de-orbiting ΔV of 3175 m/s. The refinement of the mass budget and the necessity to reduce the velocity dispersions, by reducing the size of the targeting manoeuvre, suggested to revisit the Mission Analysis strategy.

At IFP it became evident that with the consolidated mass budget and SRM Burn size of 3175 m/ a full STAR24 would be unfeasible. Therefore Mission Analysis strategy was revisited. Manoeuvre of 51 m/s would produce unacceptable levels of velocity.

The MA baseline foresees SRM Burn of 2600 m/s and a targeting ΔV of 10 m/s, further optimised with respect to earlier iterations, in order to reduce velocity dispersions. As this refinement occurred only at IFP, the MA baseline was taken into account only at Propulsion and System Level, so that the selection of the most appropriate Solid Rocket Motor could be done, and reflected in the Final Mass Budget.

For all the other subsystems, Targeting ΔV of 51 m/s and SRM ΔV of 3175 m/s constitutes the baseline.

This gives a conservative case (i.e. AOGNC is considering a canting angle for the thrusters of 15 deg), however with 10 m/s targeting ΔV, this canting angle might be reduced to 7-8 degrees.

The limited number of sessions allocated to the study did not allow to further flow down the new mission analysis strategy to all subsystems, however this should be done should further study be planned on the penetrator concept.

This chapter describes the technical solution to the latest Mission Analysis scenario, developed after the IFP (see Systems Chapter for further details), which foresees:

- Small re-start-able propulsion system for navigation and pointing of 10 m/s
- Large solid propulsion system to deliver a main ΔV of 2600 m/s

The following requirements have been considered in the propulsion architecture design:

Dry mass: 117 kg (latest System iteration is 108.97 kg including system margins 20%, which has a minor impact on the proposed architecture, and leaves more design margins)

- Hydrazine system shall provide 10 m/s velocity increment
- Solid propellant system shall provide 2600 m/s velocity increment.

12.2 Assumptions and Trade-Offs

During the course of the study, many different propulsion concepts have been investigated to provide the required velocity increments. Many of these resulted in solutions that were not feasible or that were less feasible than other alternatives.

This paragraph describes the major options that were investigated. It shall be noted that during the course of the study the requirements on propulsion have changed substantially. This concerns the dry mass of the system as well as the to be generated velocity increments. The proposed design is based on the following assumptions:

For the hydrazine propulsion system (liquid), following assumptions have been taken into account:

- Isp: 210 s
- Steering losses: 5 % for hydrazine manoeuvres
- Residuals and reserve (sliver for SRM): 2%

For solid propellant propulsion system the following assumptions have been taken into account:

- Isp solid: 282.9 s
- For solid no steering losses have been assumed
- Nutation angle is not larger than 8 degrees (in AOGNC Chapter 15 degrees are mentioned, which are computed based on the 2nd mission analysis iteration, giving a SMR Burn of 3175 m/s)
- Sliver fraction (residual propellant that is not used) for SRM assumed at 2%

The following options have been investigated:

1. Mono propellant hydrazine system with a single commercial off the shelf solid propellant rocket motor with operational concept of liquid burns – solid burn
2. Mono propellant hydrazine system with a single commercial off the shelf solid propellant rocket motor with operational concept of liquid burns – solid burn – liquid burns
3. Mono propellant hydrazine system with a cluster of smaller commercial off the shelf solid propellant rocket motor with operational concept of liquid burns – solid burn
4. Mono propellant hydrazine system with a cluster of smaller commercial off the shelf solid propellant rocket motor with operational concept of liquid burns – solid burn – liquid burns
5. Bipropellant MON –MMH system

Problems that were encountered were that either the rocket motors were too small, or too large and that off loading of propellant could not occur to levels that the study required. Usually offloading can occur to about 20%. If offloading would have to occur below these levels then there exists a risk that the motor can not be ignited anymore (under certain conditions). This might be solved by redesigning the ignition system, but this has the drawback of additional qualification efforts / costs.

As expected, analysis of a bipropellant system proved that such a system had too high mass.

For the clustered motors mentioned at 3 and 4 different cluster concepts of different solid propellant rocket motors have been investigated.

Many different cluster motor concepts have been investigated. Many motors of which reference data was obtained turned out to be out of production or had reached only the level of conceptual study. Smaller motors that are still in production have been listed in the table below.

Type	Case mass	Propellant mass	Specific Impulse
Star 12 GV	9 kg	33 kg	282
Star 13 B	5.8 kg	41.2 kg	285 s
STAR 15 G	14.11	79.6 kg	282 s
Star 17 A	13.4 kg	112.3 kg	287 s
Star 24 (for comparison and as a single motor)	17 kg	200 kg	283 s

Table 12-1: Overview of investigated smaller to be clustered SRM's that are still in production.

<u>New motor design:</u>

For a single newly developed SRM a solution to the problem was found using an Isp for the SRM of 285 s and a case mass of 17 kg. The propellant load was slightly below 140 kg. (based on mass and velocity increments applicable at 1st design iteration). When selecting a different propulsion concept, one may conclude from the numbers above that as soon as the case mass reaches 19 kg (17+2) or above, less ΔV can be generated (assuming that the Isp does not change).

<u>Cluster of smaller motors:</u>

If 140 kg of solid propellant is required in a SRM with case mass of 17 kg, when applying clusters of smaller motors this would result in the following:

For a cluster of Star 12 GV motors:

140 kg of required propellant /33 kg of propellant per motor = 4.24, hence more than 4 motors. Case mass of 4 motors is already 36 kg, so this does not provide a solution.

For a cluster of Star 13 B motors:

140/41.2=3.4, hence more than 3 motors. Case mass of 3 motors is already 17.4 kg. So there is no advantage in dry mass and even the propellant load is not sufficient, i.e. an additional motor is required. So also this does not provide a solution. (4 slightly offloaded motors would have a combined case mass of 23.2 kg).

For a cluster of STAR 15 G motors:

140/79.6=1.75 motors, hence 2 motor cases and offloaded propellant load would lead to a solution. However each motor case mass is 14.11 kg, so the total dry mass will increase to 28.2 kg. Compared to the 17 kg case mass that was mentioned before, this does not generate a solution.

For a cluster of Star 17 A motors:

140/112.3 =1.24 motors required. Hence 2 motors with the same drawback as above.

Clustering existing motors therefore does not provide advantages over the use of a single dedicated motor. Either existing (offloaded and slightly modified) or newly developed.

As could be expected clustering these types of motors usually results in larger dry mass. The simplified calculations above confirm this.

It shall be noted that some of these motors have a relatively large case mass. This is probably due to their design age. New developments using fibre overwrapped cases could have better mass ratios, hence lower dry mass.

Furthermore it shall be noted that a single motor may have a large expansion ratio. With a cluster of motors the expansion ratio per motor probably goes down due to physical size limitations of the nozzles in a cluster configuration.

However, if the thrust of a new developed motor could be kept low, the throat could be small and the expansion ratio could therefore be larger. This would lead to a higher specific impulse.

Clustering does have an advantage related to the configuration of the vehicle. If a cluster is used, the configuration could change to a configuration of SRM surrounding the penetrator, which will allow the penetrator to be placed on the centre line.

Combinations of hydrazine burns for targeting and navigation, SRM burn for providing main ΔV and additional hydrazine burns to perform the residual ΔV has been investigated as well. This did not lead to viable design solutions.

It shall be noted that when the hydrazine burn takes place after a solid rocket motor burn with an offloaded SRM, that then propellant that delivers a high specific impulse (~ 283 s) is replaced with propellant that delivers a low specific impulse (~ 210 s). Therefore this is not a logical choice. Calculations proved this.

Bipropellant propulsion system

At some point even a bi-prop system has been considered since ΔV and thrusters firing sequence demanded flexibility. Within the mass constraints that were applicable at that time, such a system was considered as too massive. The mass would have been at least 66 kg.

Monopropellant hydrazine system

This paragraph describes the small monopropellant system that provides attitude and pointing ΔV and that is complemented by a large solid rocket motor.

Based on experience, this small system shall be hydrazine based, since such systems deliver the best solution between dry mass and performance, leading to the lowest overall mass.

- Hydrazine Isp assumed at: 223 s (average with CHT-20 thruster)
- Average thrust is 19.5 N

This high thrust value is considered feasible since a relatively large tank has been baselined which results in a low blow down ratio i.e. maintaining high Isp and thrust values.

Solid Rocket Motor

The following values were assumed for the large single Solid Rocket Motor to be complemented by the hydrazine system described above. These values come from the ATK catalogue.

	STAR 24	STAR 24C (assessed and not selected)
Case mass (prior to firing)	18.3 kg	19.7 kg
Propellant mass (max)	199.9 kg	219.5 kg
Specific Impulse	282.9 s	282.3 s
Thrust	19660 N	21350N
Length	1029 mm	1067 mm
Diameter	622 mm	622 mm

Table 12-2: SRM data of Star 24 and Star 24 C

A small hydrazine system in combination with large SRM STAR 24 was baselined.

12.3 Baseline Design

The baselined design consists of a hydrazine system with one PEPT 230 tank. Initially a larger tank had been baselined. A large tank has the advantage that it leads to a favourable blow down ratio, which is favourable for maintaining high pressure levels with high thruster and high Isp as a consequence. In a smaller tank the blow down ratio is larger and therefore final thrust is lower as well as final Isp.

For the provision of the large velocity increment a STAR 24 solid propellant rocket motor has been baselined.

With the reduction of the SRM Burn, according to the Mission Analysis baseline, selected after the IFP of the study, nutation angle estimated as 15 deg with SRM ΔV 3175 m/s would go down to 7-8 deg with SRM ΔV 2600 m/s. This gives confidence that the selection of the STAR 24 is adequate to cover the mission needs. Characteristics of STAR 24 C are also reported in this chapter for completeness, as they were assessed as potential SRM candidate, should the STAR 24 provide insufficient propellant for the mission needs.

12.3.1 Hydrazine Propulsion System

The hydrazine propulsion system architecture is straight forward as can be seen in the architecture drawing below.

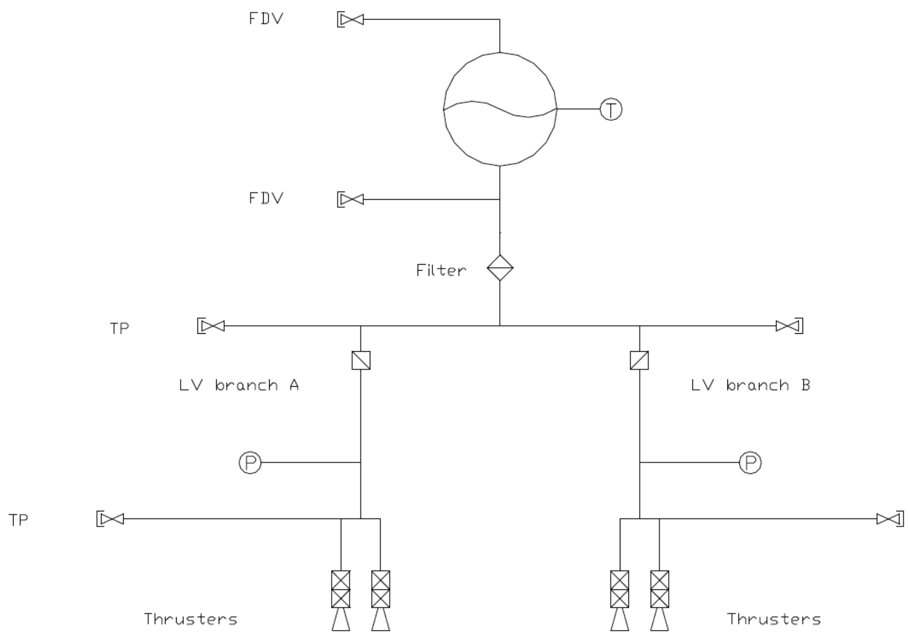


Figure 12-1: Hydrazine baselined propulsion system architecture

Data on the 20 N thrusters can be found in the table below.

Property	Value
Propellant	Hydrazine
Nom. Thrust (Range min/max)	20 (7.5 - 24) N
Nom. Specific impulse (Range min/max)	218 (210 - 228) s
Nom. Inlet Pressure (Range min/max)	22 (5.5 - 24) bar
Nom. Mass flow (Range min/max)	9.4 (3.6 - 11) g/s
Minimum Impulse Bit (Range min/max)	0.212 (0.132 -) Ns
Nozzle Area Ratio	60:1
Mass (Thruster with Valve)	0.372 kg
Catalyst	Haynes Alloy 25 (L605)
Catalyst Bed Heater Power:	3.05W @ 28VDC 20°C W
Valve:	20-32 V DC, Power: 13W @28VDC/60°C W

Table 12-3: CHT-20 data

The picture below shows the 20 N thrusters.

Figure 12-2: Selected 20 N thruster

The figure below shows the selected propellant tank.

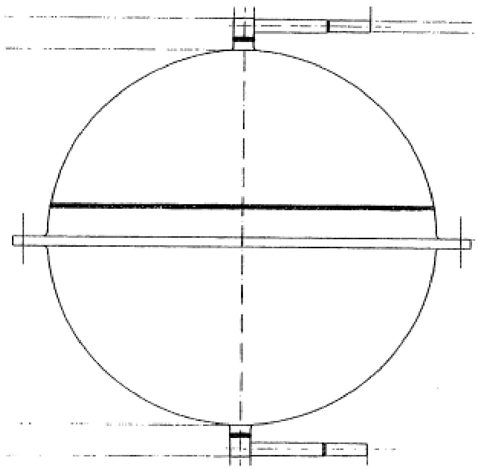


Figure 12-3: PEPT 230: Selected propellant tank

The tables below show propellant tank characteristics.

Property	Value
Propellant	Hydrazine
MEOP	24 bar
Proof Pressure	36 bar
Burst Pressure	> 75 bar
Volume	6 l
Usable Volume	4.5 l
Mass	1.25 kg
Pressurant	Helium, Nitrogen

Table 12-4: PEPT 230 tank main characteristics

Property	Value
Tank Type	Diaphragm
Mount	4 "90-degrees-spaced" Tabs
Mount Location	Equatorial
Shape	Spherical
Outer Diameter	230 mm
Length	268 (including fluid ports) mm
Min. Wall Thickness	0.6 mm

Table 12-5: PEPT 230 tank size and shape data

12.3.2 Solid Propellant Propulsion System

The selected motor is the STAR 24 with minor offload to 195 kg propellant.

	STAR 24
Case mass (prior to firing)	18.3 kg
Propellant mass (max)	199.9 kg
Specific Impulse	282.9 s
Thrust	19660 N
Length	1029 mm
Diameter	622 mm

Table 12-6: SRM data of Star 24

Figure 12-4 shows the STAR 24 Solid Propellant Rocket Motor.

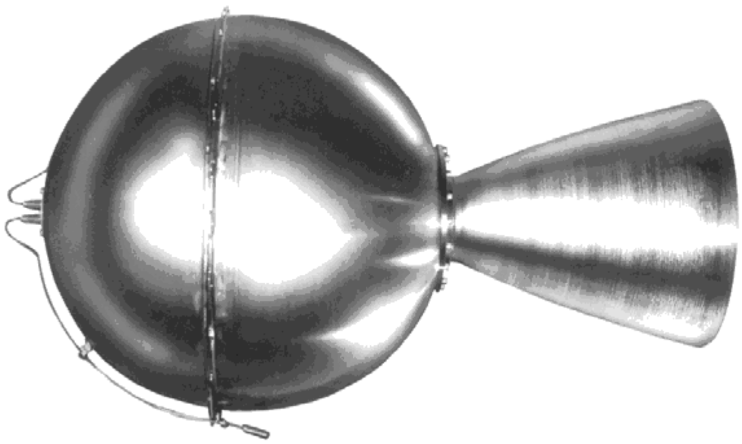


Figure 12-4: STAR 24 Solid Propellant Rocket Motor

Figure 12-5 shows the Safe and Arm device in order to ignite the STAR 24 Solid Propellant Rocket Motor as well as a schematic of the ignition train.

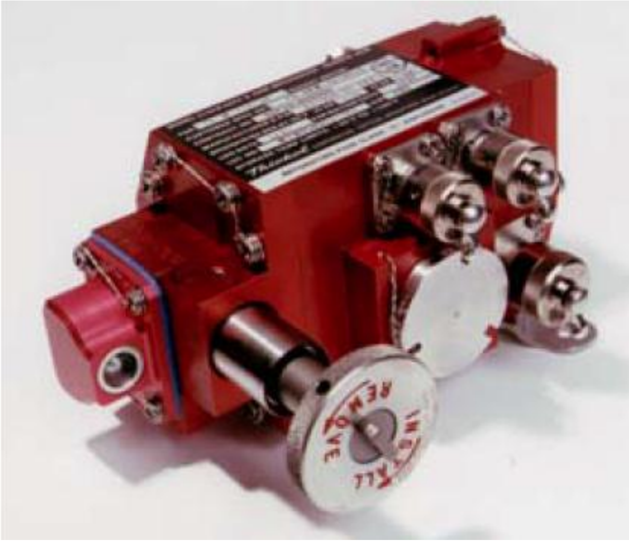


Figure 12-5: Safe and Arm device

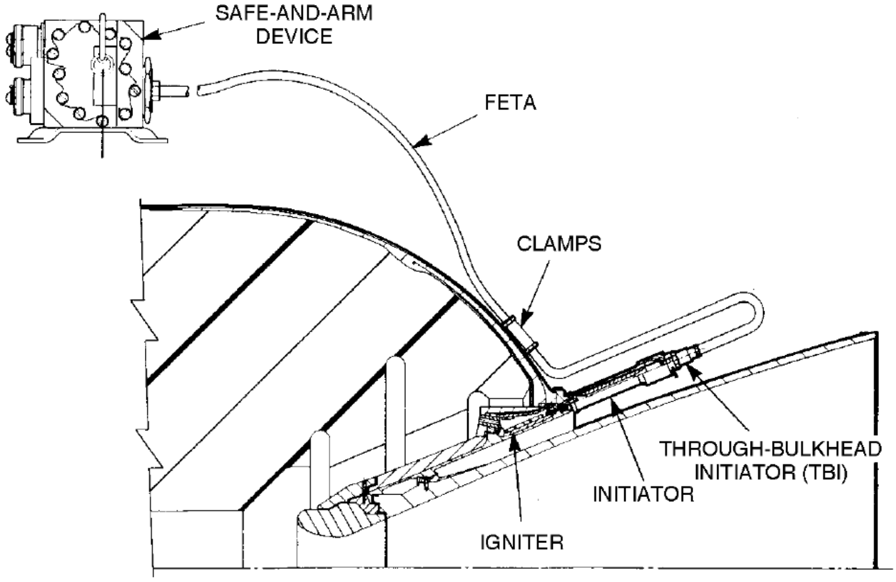


Figure 12-6: Typical STAR solid propellant rocket motor ignition train

12.4 List of Equipment

Table 12-7 shows the equipment list of the liquid propulsion system.

-				MASS [kg]							
Unit Name	Part of custom subsystem		Quantity		Mass per quantity		Maturity Level	cell name	Margin		Total Mass incl. margin
20N thruster			7		0.4				5		2.9
Propellant tank PEPT 230 with diaphragm			1		1.250				5		1.3
Propellant filter			1		0.100				5		0.1
Latching valve			2		0.250				5		0.5
Pressure transducer (same as bepi Colombo)			3		0.125				5		0.4
Fill and Drain valve / Vent valve (propellant)			2		0.070				5		0.1
Fill and Drain valve / Vent valve (pressurant)			4		0.070				5		0.3
Piping (incl fittings)			1		0.500				20		0.6
Stand-off			20		0.010				20		0.2
Mounting screws			20		0.005				20		0.1
Miscellaneous			1		0.100				20		0.1
Pressurant			1		0.250				5		0.3
											0.0
Click on button below to insert new unit											
SUBSYSTEM TOTAL			12		6.6				7.1		7

Table 12-7: Equipment list of hydrazine system

Table 12-8 shows the equipment list of the solid propulsion system.

Unit Name	Part of custom subsystem		Quantity		Mass per quantity		Maturity Level	cell name	Margin		Total Mass incl. margin
SRM			1		18.3				5		19.2
S&A device			1		1.5				5		1.6
											0.0
Click on button below to insert new unit											
SUBSYSTEM TOTAL			2		19.8				5.0		21

Table 12-8: Equipment list of solid propellant propulsion system

12.5 Options

No further options have been identified.

12.6 Technology Requirements

In principle no new technologies need to be developed since an existing SRM is chosen.

13 ATTITUDE CONTROL SYSTEM

13.1 Requirements and Design Drivers

13.1.1 Functional Requirements

The AOGNC tasks for the CLEP mission are the following:

- Spacecraft stabilisation after separation from CLIPPER (rate dumping)
- Acquisition of inertial attitude (slew)
- Targeting manoeuvre (ΔV)
- Spin-up (stabilisation during Solid Rocket Motor (SRM) burn)

Spin-down before Nutation cancelation manoeuvre and Penetrator separation

These tasks are performed by the Penetrator Delivery System (PDS) between separation from CLIPPER and release of the penetrator.

After this event the Penetrator will passively free fall until touchdown and entry into the planet surface, while the PDS will crash on the surface away from the penetrator. No additional tasks are required from AOGNC.

13.1.2 Performance Requirements

The main driver in terms of AOGNC performance is the accuracy of the Targeting Manoeuvre in terms of ΔV amplitude and direction.

The timer for SRM actuation is set according to the estimated ΔV and minimum deviation wrt the actual one can eventually lead to high dispersion in the surface impact velocity.

In order to achieve good on-board estimation accuracy the system will be equipped with accelerometers which directly measure on-board the ΔV. The pointing before the manoeuvre is ensured by miniaturised GYR/Star Tracker filter.

Another key driver for AOGNC is the stability during the SRM burn. The system is spin stabilised during the firing but the actual nutation will end in loss of efficiency of the burn, causing dispersion on the final velocity and eventually on impact velocity of the penetrator.

In this phase the AOGNC is passive, but selection of spin rate and accurate evaluation of the induced nutation is provided in next section to ensure the correct sizing of the SRM.

Finally the last key event where AOGNC accuracy is important, is the separation of the penetrator from PDS. In order to ensure limited angle of impact of the penetrator on the surface, the residual nutation shall be cancelled before separation. At the same time the spin rate shall be sufficient to ensure enough separation at landing between PDS and Penetrator.

Concerning the mission requirements that drive the configuration, the most critical are the mass and the power consumption.

13.2 Assumptions and Trade-Offs

The design of the AOGNC for this mission is mainly based on previous study results. However following the identified driver requirements presented in previous section, some analyses and trade-off have been performed, in particular:

- Thruster layout configuration
- Accuracy of ΔV measurement with accelerometers
- Spin stability during SRM burn.

13.2.1 Assumptions

The PDS will have cylindrical shape with overall wet mass of ≈315 kg at separation from CLIPPER. The mass assumed at the end of SRM is 115 kg (≈108 kg dry mass plus propellant residuals).

The inertia properties of the PDS are:

- At separation from CLIPPER: Iz=25kg*m², Iy=58kg*m², Ix=45kg*m²
- At the end of SRM burn: Iz=9kg*m², Iy=21kg*m², Ix=16kg*m²

The requested ΔV for targeting manoeuvre has been assumed 51m/s, while for the SRM the STAR24 has been assumed with thrust capacity of ≈19kN, delivering 3175m/s.

13.2.2 Trade-Off: Thruster Configuration

The configuration of the thrusters is traded-off between two possible solutions, looking at the overall efficiency in terms of mass and duration of targeting ΔV and spin-up/spin/down.

The thrusters used for the trade-off are always the same type, namely the Airbus CHT-20N thrusters in baseline design, which have mass less than 0.5 kg each, Isp=220s.

13.2.2.1 4 thrusters with cant angle

This solution aims to minimise the number of thrusters, using the same set to perform both the ΔV and the spin-up/spin-down manoeuvres.

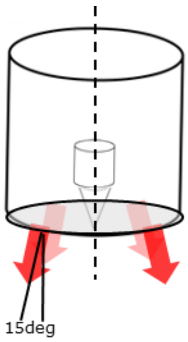


Figure 13-1: 4 thrusters layout

The configuration as proposed includes 4 THR x 20N mounted along symmetry axis with 15deg cant angle, providing the following performance:

- ≈77N Force for ΔV (duration of manoeuvre 158s)
- ≈5.2Nm Torque for Spin-UP (duration of manoeuvre 48s with parasitic ΔV generated ≈12m/s)
- ≈4.68Nm Torque for Spin-DOWN (duration of manoeuvre 18s with parasitic ΔV generated ≈11m/s).

It is assumed that parasitic ΔV generated during Spin-Up and Spin-Down manoeuvres are part of the calculated ΔV. This complicates operations since part of ΔV is performed with 4THR and part with only 2THR (during Spin-UP).

This configuration is inefficient during ΔV because of cant angle. The value of 15 deg has been chosen as optimum compromise between having limited losses during ΔV burn and sufficient torque for Spin-Up/Down.

It shall be noted that after IFP, Mission Analysis refined the assessment on the targeting ΔV magnitude, reducing ΔV from 51 m/s to 10 m/s. This is not reflected in this chapter, due to lack of time, and would allow for increasing thrusters canting angle from 15 deg to 30 degrees (more efficiency during spin-up/down).

The overall mass of this configuration is 12.56 kg, including:

- Propellant mass for ΔV(*) ≈ 6.12kg
- Propellant mass for Spin-Up ≈ 1.67 kg (+100% margin) ≈ 3.34 kg
- Propellant for Spin-Down ≈ 0.55 kg (+100% margin) ≈ 1.1 kg
- Thrusters dry mass ≈ 2 kg

(*)ΔV considered is 51m/s required minus the 12m/s generated during Spin-Up. The parasitic ΔV generated during Spin-Down shall be counted for the SRM since the manoeuvre is performed at the end of descent phase.

13.2.2.2 7 thrusters: 3 dedicated to ΔV and 2+2 for Spin-Up Spin-Down

This solution aims to maximise the efficiency in performing the manoeuvre, optimising the angle for each manoeuvre.

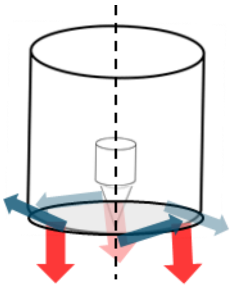


Figure 13-2: 7 thrusters layout

The configuration includes 2 THR x 20N mounted along symmetry axis dedicated to ΔV, 2 THR x 20N mounted orthogonally for the Spin-Up manoeuvre and 2 THR x 20N mounted orthogonally providing torque for the Spin-Down manoeuvre, providing the following performance:

- ≈60N Force for ΔV (duration of manoeuvre 265s)
- ≈20Nm Torque for Spin-UP (duration of manoeuvre 13s with no parasitic ΔV)
- ≈18Nm Torque for Spin-DOWN (duration of manoeuvre 5s with no parasitic ΔV)

The configuration provides simplification of operations, being that each set of thrusters is dedicated to specific manoeuvres at specific moments in time.

The overall mass of this configuration is 12.61 kg, including:

- Propellant mass for ΔV(*) ≈ 7.977 kg
- Propellant mass for Spin-Up ≈ 0.43 kg (+100% margin) ≈ 0.86 kg
- Propellant for Spin-Down ≈ 0.14 kg (+100% margin) ≈ 0.28 kg
- Thrusters dry mass ≈ 3.5 kg

13.2.2.3 Conclusion

The results provide an equivalent result for both configurations in terms of mass. However the configuration with 7 thrusters is baselined, considering the considerable saving in terms of complexity of operations and therefore risk.

It shall be noted that after IFP, Mission Analysis refined the assessment on the targeting ΔV magnitude, reducing ΔV from 51 m/s to 10 m/s. This is not reflected in this chapter, due to lack of time, and shall be re-assessed, should future studies be dedicated to the Penetrator concept.

13.2.3 Analysis: Accuracy of ΔV Measurement with Accelerometers

The requirement on high accuracy in determination and implementation of targeting manoeuvre not only in terms of direction but as well in terms of magnitude, leads to the need of having on-board accelerometers.

The selected units are ESA-developed Colybris SA0120, 1g sensor, MEMS accelerometers designed for space applications. The main performance (provided by manufacturer) is summarised below (1σ):

- Full scale range: ± 1g
- Bias calibration: <2mg
- Bias stability (1h): 0.025mg
- Scale factor stability: 300ppm
- Non linearity: <0.5% FS
- Noise in band @10Hz: 5µg/√Hz

These sensors are very small as they are set of 2 integrated in a single chip together with ASIC electronics.

Assuming on-board calibration of the sensor bias, to be performed one hour just before the manoeuvre, the residual bias is considered as 5% of total uncalibrated value.

The figure below shows the error for several cases of ΔV performed with the same configuration of thrusters as presented in previous section.

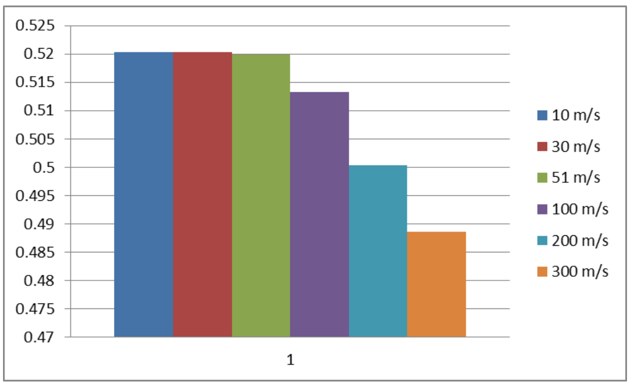


Figure 13-3: ΔV measurement accuracy

The reachable relative accuracy for the case under study (ΔV=51m/s) is ≈0.519%.

This value improves a lot the overall dispersion on impact velocity and it is therefore considered in the AOGNC baseline design.

13.2.4 Analysis: Spin Stability During SRM Burn

The SRM burn happens at the end of the descent phase, to cancel the terrain relative velocity of the PDS at 35 km altitude above the surface, then starts the free fall to terrain impact.

The selected Solid Rocket is the STAR24 which provides roughly 20kN of thrust for about 30s. Nominally the actual thrust direction is aligned with the CoG.

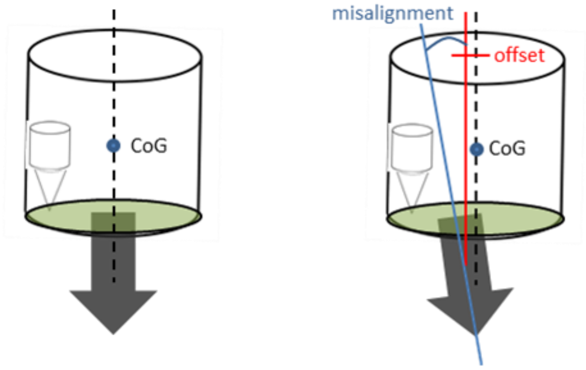


Figure 13-4: Nominal and actual SRM thrust direction

The misalignment and the offset of mounting wrt the CoG creates disturbance torques during the burn that cannot be compensated with the other thrusters, as they are not sized for that.

The stability shall therefore be ensured by spin. Given the system inertia properties, the nutation angle of the spacecraft during the spin decreases with respect to the increase of spin rate and is linked to the magnitude of disturbance torque, which depends on motor alignment and offset mounting.

The higher is the nutation angle the lower is the efficiency of the burn, because in average the thrust direction is not towards the velocity vector but misaligned with an angle correspondent to nutation. For a nutation angle of 10deg the efficiency is 98.5%, that means ≈47m/s over the total of 3175m/s.

The SRM burn size baselined at system level after IFP is 2600 m/s, leading to a better picture in terms of absolute ΔV losses: smaller manoeuvre size, ≈39m/s for 2600m/s assuming the same thrust level (STAR24 at 20kN).

The maximum value of the spin rate is driven by the motor qualification, which has been tested up to 100 rpm (i.e. 600deg/s). This value shall be therefore assumed as bound.

The next plots shows how the nutation angle varies wrt the offset (colours) and misalignment (ordinates). Figure 13-5 shows results with spin rate of 50 rpm and Figure 13-6 shows results at spin rate of 100 rpm.

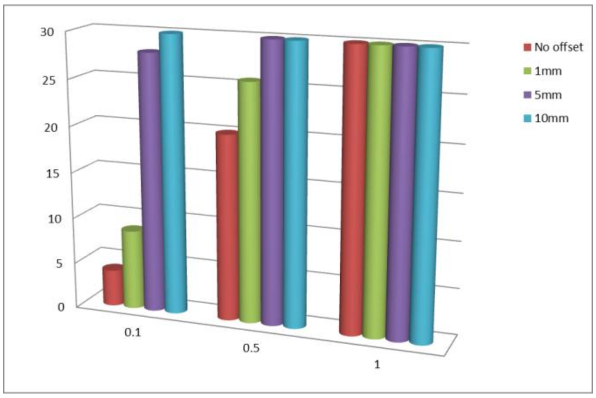


Figure 13-5: Nutation angle wrt misalignment and offset at 50 rpm

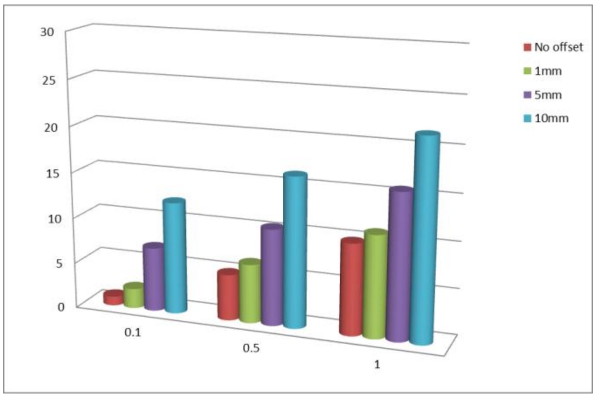


Figure 13-6: Nutation angle wrt misalignment and offset at 100 rpm

The results shows that the only way to keep burn efficiency under control is to maximise the spin rate. Therefore the maximum possible has been selected, as 100 rpm.

13.3 Baseline Design

The baseline AOGNC implements the following operative modes:

- Targeting Mode (TGT) starts at separation from CLIPPER and includes the following sub-modes:
  - Rate Dumping of residual separation rate
  - Calibration of accelerometers
  - Targeting manoeuvre burn (ΔV)
- SPIN Mode (SPN) starts when the ΔV has been completed and foresees open loop time-tagged thrusting to spin up the PDS up to 100 rpm
- Descent Mode (DSC) starts when the spin-up thrust is completed and lasts until the SRM burn is completed. During this phase the AOGNC is passive, ensuring stabilisation by spin.
- DESPIN Mode (DSPN) starts when SRM burn is completed and foresees open loop time-tagged thrusting to spin-down the PDS until the spin rate is reached (measured with GYR) and the following thrust sequence to cancel the residual nutation.

The modes are sequential and they are triggered on events. Due to the nature of the mission it is not foreseen to return on previous operative mode and there is no Safe Mode defined (Autonomous Fail Operational mission).

13.3.1 Nutation Cancellation Before Penetrator Release

After the SRM burn the PDS starts its free fall. The PDS will still be in spin mode with the nutation angle as accumulated during the burn.

This angle, if not corrected, translates into contribution to the angle of impact of the penetrator on the surface. One of the possibilities to cancel this nutation would be the use of thrusters in combination with GYR to estimate the correct thrust instant.

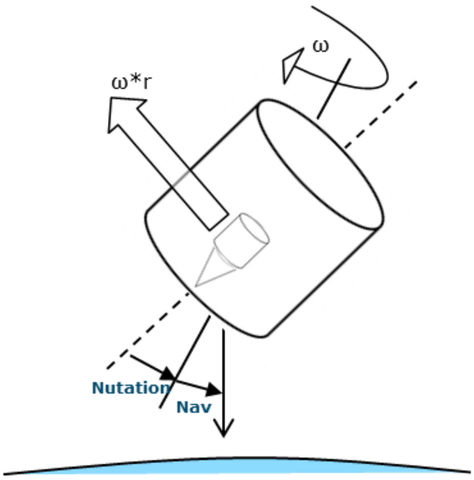


Figure 13-7: Separation of penetrator during free-fall

The selected GYR, however, are saturated at 600deg/s, as their useful range is up to 100deg/s. Therefore, after the end of SRM burn, a Spin-Down manoeuvre is foreseen to reduce the spin rate to 10 rpm (i.e. 60deg/s) being able to measure the rotation and cancel the nutation angle with sequence of firings just before the penetrator release.

The reduction of spin rate will also reduce proportionally the relative separation velocity of penetrator and PDS, with eventually smaller distance on surface at impact. The timing of the sequence shall be tuned to ensure sufficient margin for both aspects.

13.4 List of Equipment

The list of baseline AOGNC equipment includes only sensors, as the actuation is based only on thrusters and their relevant description is detailed in the RCS section.

The selection of the sensors has been driven by the need to minimise mass and power consumption. As a consequence the selected sensors are all based on MEMS technology.

13.4.1 Micro - Star Tracker

The selected STR is the micro-STR from SELEX-ES which includes sensor, processor, and interface electronics on the same chip.

It is expected that the star tracker could provide the accuracy of ≈15 arcsec within a mass of 175 grams, power consumption of 0.72W, and volume of 42mmx37mmx83mm.

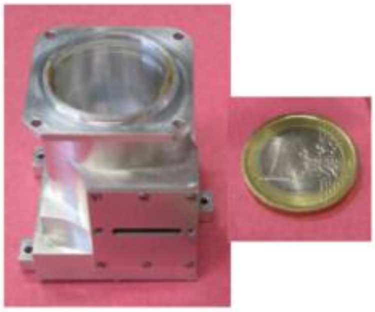


Figure 13-8: micro-STR SELEX-ES

The STR is used in TGT Mode for the inertial attitude acquisition before and during the ΔV manoeuvre. It will not be used in any other mission phase.

Two units will be mounted working in hot redundancy.

13.4.2 GYR on a Chip

The solution selected for the GYR is the sensor on a chip, where all the acquisition and processing is performed by the spacecraft OBC. The unit will be used in all operative modes but the DSC where the output is saturated due to the high spin rate.

The selected unit is a medium class, based on MEMS technology manufactured by Systron Donner and flown already as part of Quartz Rate Sensor (QRS).

Figure 13-9: MEMS GYR chip

IRS QRS11 Quartz MEMS technology providing a solid-state gyro, typical performance: range ±100°/sec, Short Term Bias Stability (100 sec at const. temp) < 0.01°/sec, output noise (DC to 100 Hz) < 0.01°/√Hz. Mass of 60 grams (per axis), power 0.4W each, dimensions d=42mm h=13.5mm.

One set of 4 sensors (mounted on bracket in skewed configuration) are foreseen for redundancy.

13.4.3 ACC on a Chip

The solution selected for the ACC is the MEMS Accelerometers produced by SAFRAN Colibrys. The technology foresees 2 channels on same chip together with their ASIC electronics.

The unit will be used during TGT mode to measure accurately the realised ΔV.

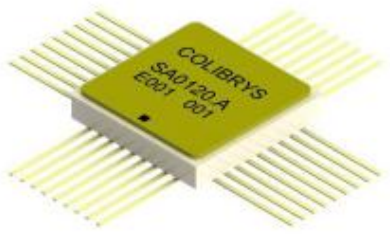


Figure 13-10: MEMS ACC chip

Capacitive MEMS accelerometers providing dual ranges Radhard sensors on single chip, typical performance: range ±1g, Bias <2mg, Bias Stability (1h) < 0.025mg, Scale Factor <300ppm, resolution @1Hz =0.05mg, Noise (@10Hz) <5µg/√Hz. Mass of 5 grams (per axis), power 10mW each, dimensions 33mmx33mmx3.5mm.

Three chips shall be mounted, each being already internally redundant with 2 channels.

13.5 Options

13.5.1 Navigation Camera

During the study it came out that measurement of relative distance from the target planet would be beneficial for the dispersion in the impact velocity, being a driver in resetting the timer for the SRM burn ignition start (accurate triggering is required for the events subsequent to the release – a simple time propagation from Clipper release would result in huge dispersions, calling for some events-based timer reset) .

One option is to mount a Navigation Camera on board the PDS to measure the distance wrt the planet estimating the diameter of the planet.

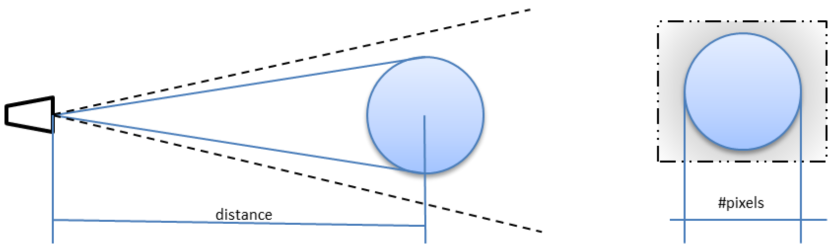


Figure 13-11: NAV CAM measurement

The Figure 13-12 below report preliminary estimation of reachable accuracy with NAV CAM assuming the following:

- Camera image is square, 1024x1024 pixels
- Europa occupies the 90% of the Camera FoV when the image is taken (i.e. ≈920pixels over 1024pixels)
- Europa diameter is known with accuracy of 0.05%, i.e. 3121.6km ±1.56km

The resulting error on distance measurement are plot for different FoV's varying from 10deg to 180deg (note that distance from planet is adjusted to cope with the assumption of 90% and varies accordingly between 40000km and 500km) and linked to camera resolution in pixels (subpixels).

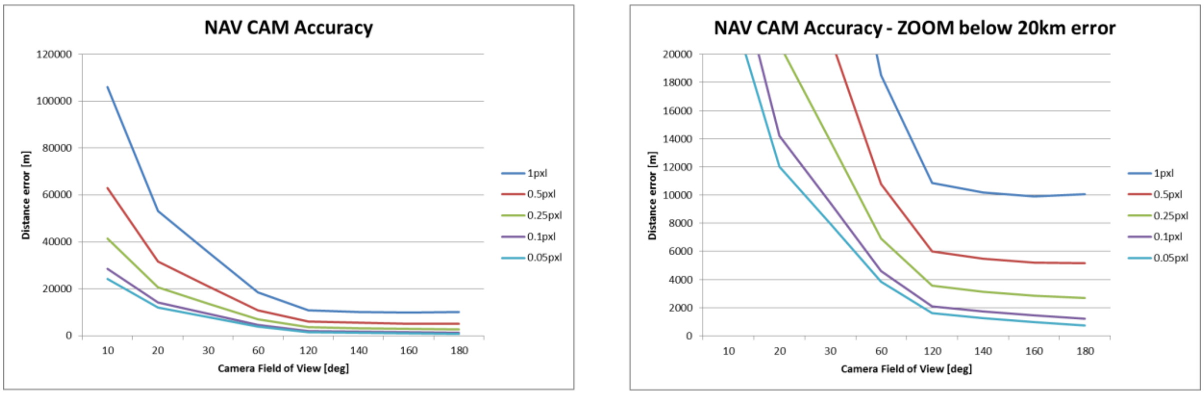


Figure 13-12: Preliminary NAV CAM accuracy

The optimum in terms of FoV is 120deg: wider FoV's do not improve the result and narrower FoV's are worst because in terms of absolute measurement because the distance is higher.

Assuming 120deg the optimal distance where to take the image is at 2500 km and the accuracy is 3.5 km assuming the resolution of 0.1pixels.

Provided results are very preliminary and it is recommended for future studies to further investigate this option wrt existing technology and possibly new developments.

13.5.2 Altimeter

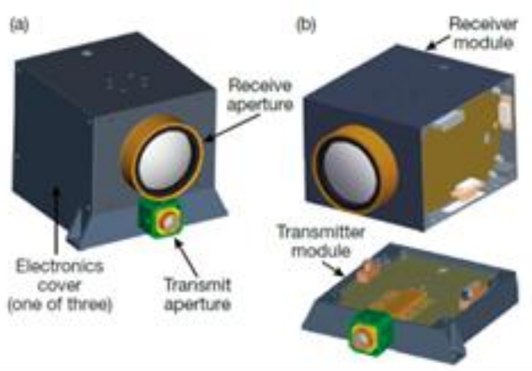
The Penetrator separation from PDS is triggered by timer activated at the end of targeting manoeuvre (last available updated of estimated timeline).

On one side the separation shall happen as late as possible to prevent increase of angular error due to residual angular rate from separation mechanism.

On the other side, the separation shall be anticipated to cope with velocity dispersion.

Another option to improve the triggering is to use an altimeter on the PDS instead of time tagged triggers. This option will provide precise trigger relative to terrain, in terms of velocity and distance.

The drawback is that currently existing (low TRL) altimeters have limited measuring range (<2 km). This will imply late release and limited distance between PDS and penetrator at impact.



Parameter	goals
Maximum range (km)	1–2
Albedo (reflectivity)	≥0.02
Range accuracy (m)	≤0.03 (closest approach)
Range resolution (m)	≤0.03 (closest approach)
Horizontal accuracy (footprint)	≤10 m at 1 km, ≤1 cm at 1 m
Measurement update rate (Hz)	1000–5000
Averaging	100× (nominal)
Mass (kg)	≤1.5
Power (W)	≤7 (over temperature)
Required operational period	≤1 month (continuous)

Figure 13-13: Preliminary altimeter accuracy

It is therefore recommended for future studies to look for solutions where miniaturised altimeter can increase the operative range (with relaxation of performance) and be used at higher altitude to measure both terrain relative altitude and velocity.

	mass (kg)	mass margin (%)	mass incl. margin (kg)
ACC_1 (Accelerometer SA0120)	0.01	20.00	0.01
ACC_2 (Accelerometer SA0120)	0.01	20.00	0.01
ACC_3 (Accelerometer SA0120)	0.01	20.00	0.01
GYRO_QRS11_1 (GYRO on Chip QRS11)	0.06	5.00	0.06
GYRO_QRS11_2 (GYRO on Chip QRS11)	0.06	5.00	0.06
GYRO_QRS11_3 (GYRO on Chip QRS11)	0.06	5.00	0.06
GYRO_QRS11_4 (GYRO on Chip QRS11)	0.06	5.00	0.06
STR_micro_1 (STR Selex Micro Star Tracker)	0.18	20.00	0.21
STR_micro_2 (STR Selex Micro Star Tracker)	0.18	20.00	0.21
Grand Total	**0.61**	**14.05**	**0.69**

Table 13-1: AOGNC Equipment list

Power (W)		
	P_on	P_stby
ACC_1 (Accelerometer SA0120)	0.02	0.00
ACC_2 (Accelerometer SA0120)	0.02	0.00
ACC_3 (Accelerometer SA0120)	0.02	0.00
GYRO_QRS11_1 (GYRO on Chip QRS11)	0.40	0.00
GYRO_QRS11_2 (GYRO on Chip QRS11)	0.40	0.00
GYRO_QRS11_3 (GYRO on Chip QRS11)	0.40	0.00
GYRO_QRS11_4 (GYRO on Chip QRS11)	0.40	0.00
STR_micro_1 (STR Selex Micro Star Tracker)	0.72	0.00
STR_micro_2 (STR Selex Micro Star Tracker)	0.72	0.00
Grand Total	**3.10**	**0.00**

Table 13-2: AOGNC Power budget

13.6 Technology Requirements

As identified in the previous section, the following technologies would be beneficial to this mission:

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
NAV CAM	Measurement of planet diameter	TRL=5		To be further improved on HW and SW sides
Miniaturized Altimeter	High range, medium accuracy			Existing technologies looking for extended ranges

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14 POWER

14.1 Requirements and Design Drivers

- There is very weak sunlight in the Jovian system, especially around the foreseen arrival time of the years 2025-2030 (Jupiter's aphelion). The solar flux at this time will be 46 W/m², as compared to 56 W/m² at perihelion. (The solar flux at Earth is ~ 1367 W/m²)
- Very low mass target for the spacecraft (as a passenger of CLIPPER)
- After separation from CLIPPER, the CLEP power system must provide power/energy to support all platform and payload requirements
- Time from CLEP separation from CLIPPER separation to Europa descent and impact = 1.75 days
- Europa impact to end of surface mission = 10.5 days (arising from the orbital period of CLIPPER, which will receive the science data from CLEP).

14.1.1 Penetrator Power Budget (Consumptions)

The core information for the penetrator power/energy budget is taken from RD[14], Page 125, Table 12-6. This information pertains to a 7 day surface mission, so must be adjusted for the CLEP timeline (for the warm bay only). The relevant tables are reproduced here as Table 14-1 and Table 14-2

	Item	Daily energy usage (Whrs)							TOTAL	Comment
		Day 1	Day 2	Day 3	Day 4	Day 5	Day 6	Day 7		
COLD BAY	Drill	0.75	0.00	0.00	0.00	0.00	0.00	0.00	**0.75**	See TN3.1
	Sample Container	9.20	0.00	0.00	0.00	0.00	0.00	0.00	**9.20**	See TN3.1
	Pyro / Gas Valve	0.47	0.00	0.00	0.00	0.00	0.00	0.00	**0.47**	See TN3.1
	Common Electronics	0.65	0.00	0.00	0.00	0.00	0.00	0.00	**0.65**	See TN3.1
	BMS	1.98	0.00	0.00	0.00	0.00	0.00	0.00	**1.98**	See TN3.1
	Sample Imager	0.00	0.00	0.00	0.00	0.00	0.00	0.00	**0.00**	See TN3.1
	Habitability Package	0.23	0.00	0.00	0.00	0.00	0.00	0.00	**0.23**	See TN3.1
	PCDU efficiency	98%	98%	98%	98%	98%	98%	98%		Assume 98% efficiency
	Interface efficiency	80%	80%	80%	80%	80%	80%	80%		worst case efficiency
	Total exc margin	16.91	0.00	0.00	0.00	0.00	0.00	0.00	16.91	
	System margin	20%	20%	20%	20%	20%	20%	20%		
	TOTAL	**20.30**	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	**0.00**	**20.30**	

Table 14-1: Cold bay equipment budget from RD[14]. This equipment runs one operational sequence, so the energy is independent of mission length, and the data can be used directly for CLEP

WARM BAY	Item	Daily energy usage (Whrs)							TOTAL	Comment
		Day 1	Day 2	Day 3	Day 4	Day 5	Day 6	Day 7		
	Microseismometer	2.86	2.86	2.86	2.86	2.86	2.86	2.86	**20.00**	20Whrs for a week
	Comms and OBDH	19.20	19.20	19.20	19.20	19.20	19.20	19.20	**134.40**	Assume 1W receive
	PCDU Quiescent	2.40	2.40	2.40	2.40	2.40	2.40	2.40	**16.80**	0.1W overhead
	Heater power	20.00	20.00	20.00	20.00	20.00	20.00	20.00	**140.00**	
	PCDU efficiency	98%	98%	98%	98%	98%	98%	98%		Assume 98% efficiency
	Total exc margin	45.36	45.36	45.36	45.36	45.36	45.36	45.36	**317.55**	
	System margin	20%	20%	20%	20%	20%	20%	20%		
	TOTAL	**54.44**	**54.44**	**54.44**	**54.44**	**54.44**	**54.44**	**54.44**	**381.06**	
	AVERAGE POWER	**2.27**	**2.27**	**2.27**	**2.27**	**2.27**	**2.27**	**2.27**		

Table 14-2: Warm bay equipment budget from RD[14]. This equipment runs for the full surface mission, so the energy requirement must be adjusted accordingly for CLEP

14.1.2 PDS Power Budget (Consumptions)

The core information for the PDS power/energy budget is taken from RD[14], Page 97, Tables 9-2 and 9-3. This information pertains to a 1.2 hour separated cruise, so must be adjusted for the CLEP case of 1.75 days. The relevant tables are reproduced here as Table 14-3 and Table 14-4.

Contributing Sub-System	**Peak Power (W)**
OBC (LEON MCC)	5
Inertial Rate Sensors (QRS11) 4-off @ 0.4W each	2
Micro- Star Tracker (Sensor on a chip SOAC-TN-GA-003)	0.25
AOCS Interface Unit	3
Flow Control Valves (8 each x 4-off +55.3 main engine)	87
Memory Unit	1
Temperature sensors (0.25W ea x 4-off)	1
Communications (powered by PDS via Penetrator)	10
Equipment Total Power	**109**
Margin at 20%	22
Equipment Total Power with Margin	**130**
Power Control Unit Overhead (85% efficient)	20
Total Power at Battery	**150**

Table 14-3: PDS peak power budget from RD[14]

These durations/ energy values are adjusted for 1.75 days

These durations/ energy values are NOT adjusted

Contributing Sub-System	Inst. Power (W)	Duration (Hr)	Energy (WHrs)
OBC (LEON MCC)	5	1.2	6.0
Inertial Rate Sensors (QRS11) 4-off @ 0.4W each	1.6	1.2	1.9
Micro- Star Tracker (Sensor on a chip SOAC-TN-GA-003)	0.25	1.2	0.3
AOCS Interface Unit	2.5	0.1	0.3
Flow Control Valves (8 each x 4-off +55.3 main engine)	87.3	0.1	8.7
Memory Unit	1	1.2	1.2
Temperature sensors (0.25W ea x 4-off)	1	1.2	1.2
Communications (powered by PDS via Penetrator)	10	0.4	4.0
Equipment Total Energy			**23.6**
Margin at 20%			4.7
Equipment Total Energy with Margin			**28.3**
Power Control Unit Overhead (85% efficient)			4.2
Total Energy at Battery			**32.6**

Table 14-4: PDS energy budget from RD[14], with annotations to explain how the data is applied to the CLEP case

In the CLEP case, the longer separated cruise leads to an additional requirement for heating that is not present in the Astrium Phase 2 power/energy budget tables.

However, in the Airbus Penetrator Phase 3 Technical Note 15 (RD[15], Table 7-5 on page 28), an estimate of Penetrator and PDS heater demand during cruise is presented. The total heat loss (assuming some use of insulation) is estimated to be 23 W. Therefore, for the CLEP case, 23W (constant) is added to the power budget.

14.2 Assumptions and Trade-Offs

CLEP will make the trip from Earth to the Jovian system as a passenger of CLIPPER.

It is assumed that CLEP will take some power from its host for:

- Battery top up / self discharge compensation (only if CLEP uses a secondary battery, and will be a negligible amount of energy in any case)
- Periodic check-outs & housekeeping tasks (negligible energy if performed infrequently)
- Thermal control (possibly significant energy, e.g. 25 W constant for propulsion system heating).

14.2.1 Selection of Battery Cell Technology for Penetrator

As a "one-shot" device with a short mission and no practicable possibility for solar cell employment, the penetrator battery is clearly best formed from primary (non-rechargeable) cells. Furthermore, the task of assessing and evaluating the possible primary cell technologies has been performed by Airbus under ESA contract, and is described in RD[16]. The document selects the QinetiQ M1 Li-CF_X pouch cell as optimum for the penetrator application.

Whilst the technical information on the cell is incomplete (e.g. "shock: not declared"), the selection of this cell as baseline is justified, pending confirmation of detailed specifications. In particular, the M1 cell has a very high mass-specific and volume-specific energy, far exceeding any of the other candidates. A selection of relevant details is reproduced here as Figure 14-1

Dimensions : 137mm x 92mm x 7.5mm.

Commercially available: No

Flight heritage: Currently military only, with some applications up to TRL9. Intellectual property rights are owned by QinetiQ

Shock: Not declared

Acceleration: Not declared

Nominal voltage: 2.55V to 1.5V at cell level.

Capacity: 30Ah

C rating: 0.14/h (C/7) at 20degC but depends very much on thermal design of the pack

Energy density: 728Wh/kg

Mass density: 1177 / 728 x 1000 = 1617kg/m3

Mass: 0.105kg

Self discharge: 0.5-1% per year at room temperature

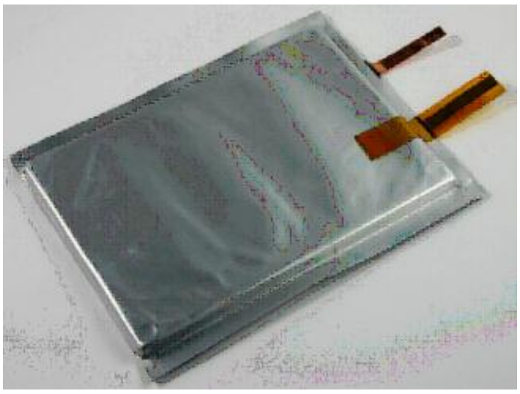

Figure 14-1: Images and data of the QinetiQ M1 lithium carbon monofluoride pouch cell. Reproduced from RD[16]

14.2.2 Selection of Power Source for PDS

At first consideration, the PDS power system could be supplied by various energy sources. The possibilities and a brief trade-off are detailed in Table 14-5.

Power source	Comments
Solar Array	At Jupiter (at aphelion), < 3 W/kg. So, over the 1.75 day cruise < **126 Wh/kg**
Secondary (rechargeable) battery [Airbus design baseline]	Space Li-ion batteries : < **170 Wh/kg** Requires circuitry to top-up charge from clipper before separation, then is used as one-shot, like a primary battery.
Primary (non-rechargeable) battery	Established lithium primary cells, eg. Li-$SOCl_2$: ~**200 to 400 Wh/kg**, after 8 years self-discharge, depending on discharge current. QinetiQ M1 Li-CF_x pouch cell ~**670 Wh/kg** after 8 years self-discharge.
Combination, eg, SA + secondary batt.	Complicated. Not necessary unless none of the above options can fulfil requirements.

Table 14-5: PDS power source trade-off

Given the unsurpassed energy density of the M1 Li-CF_x pouch cell, and considering the attractive synergy of developing/using the same technology for PDS and penetrator, QinetiQ's lithium carbon monofluoride primary cell is selected as power source for the PDS.

14.3 Baseline Design

14.3.1 Battery Sizing

The batteries of the penetrator and PDS were sized using the performance and mass/volume characteristics of the M1 Li-CF_x pouch cell, assuming a depth of discharge of approximately 70%. Extracts from the spreadsheets used, with annotations highlighting important assumptions and results, are presented below as Figure 14-2 and Figure 14-3.

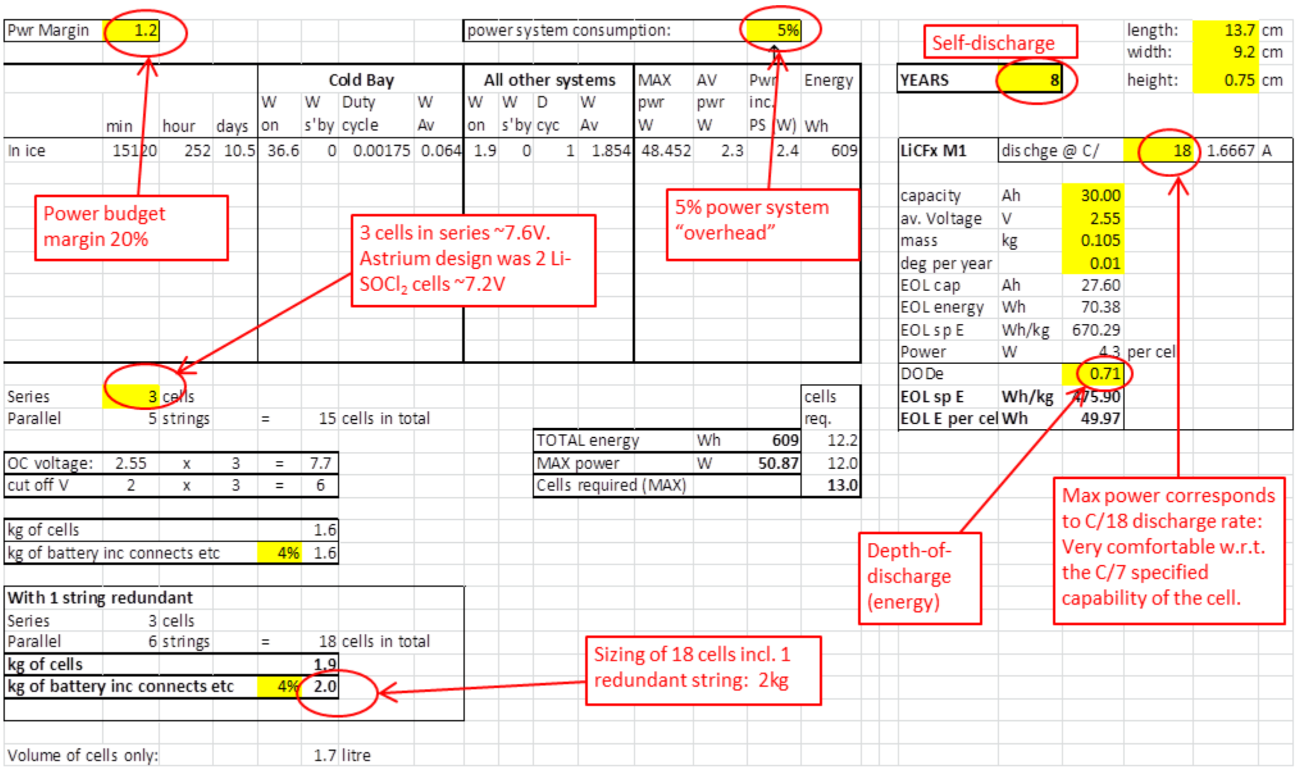

Figure 14-2: Annotated battery sizing spreadsheet (Penetrator)

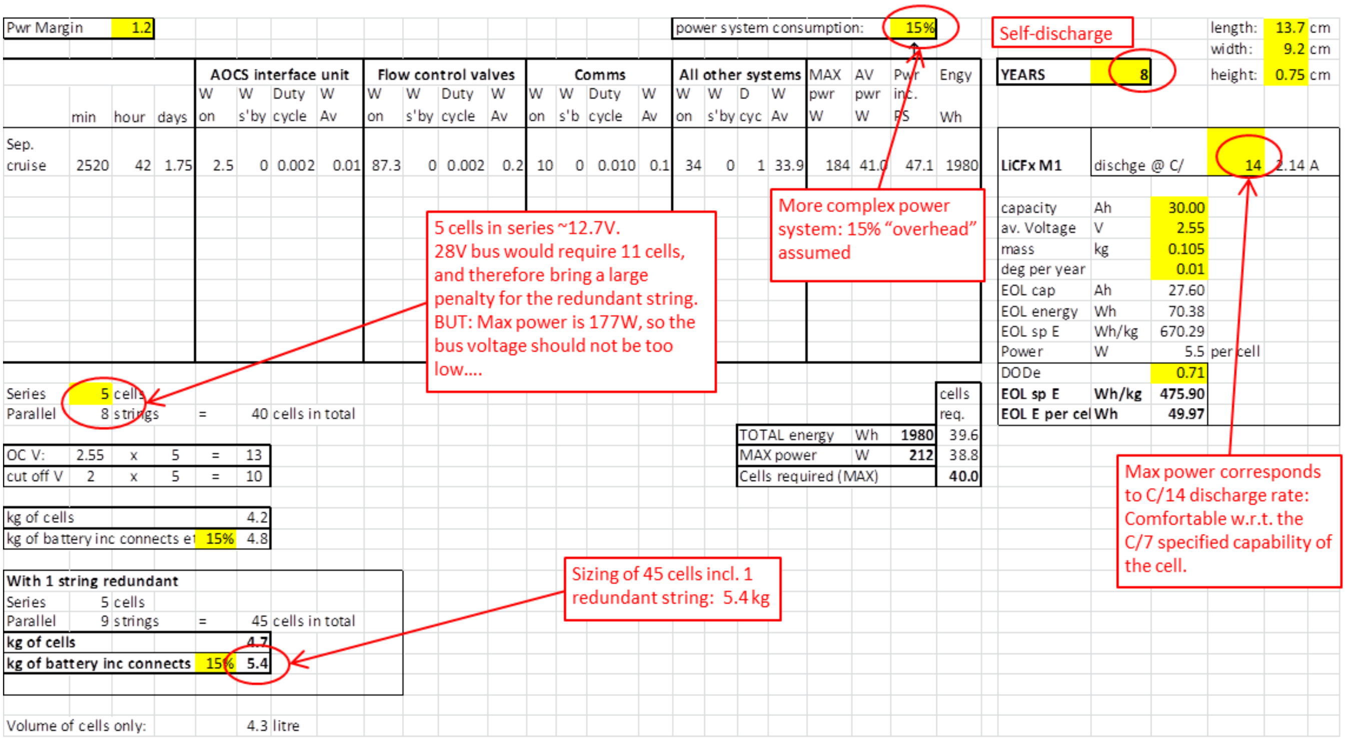

Figure 14-3: Annotated battery sizing spreadsheet (PDS)

14.3.2 PCDU Sizing

For the penetrator, the Airbus design approach of integrated avionics is followed. So no separate PCDU mass is accounted for.

For the PDS, considering the minimum functionality needed: some DC-DC converters; interface to CLIPPER's power bus; distribution lines (LCLs, heater switches, pyro lines), and assuming integration of power boards in a combined avionics box, the PCDU electronics is estimated to require three circuit boards with a mass of approximately 2 kg.

14.4 List of Equipment

Product/Function	Product
Owner	PWR
Parameter	m

	mass (kg)	mass margin (%)	mass incl. margin (kg)
Clipper Europa Penetrator (CLEP)	**9.40**	**12.13**	**10.54**
PDS (Penetrator Delivery System CLEP)	**7.40**	**12.70**	**8.34**
BatPrim2 (Battery_Primary 2)	5.40	10.00	5.94
PCDU2 (Power Conditioning & Distribution Unit 2)	2.00	20.00	2.40
FPEN (Fore Penetrator CLEP)	**2.00**	**10.00**	**2.20**
BatPrim (Battery_Primary)	2.00	10.00	2.20
Grand Total	**9.40**	**12.13**	**10.54**

Table 14-6: Power system list of equipment

14.5 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed
- Technologies available within European non-space sector(s)
- Technologies identified as coming from outside ESA member states.

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
QinetiQ M1 Li-CF_x cell RD[16].	Primary battery		QinetiQ. TRL 9 for military (non-space) applications	Fundamental shock capability is believed to be sufficient, but formal verification is required

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15 DATA HANDLING

15.1 Requirements and Design Drivers

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
DH-010	Low mass and low power consumption	
DH-020	Science data acquisition and storage on the penetrator	
DH-030	Wireless communication between penetrator warm and cold bay	

15.2 Assumptions and Trade-Offs

The mission contains two DHS units, one on the PDS and another one in the Penetrator Warm bay.

Due to the extremely reduced mass and power budget and thanks to the short mission duration, the data handling design could be based on a non-redundant architecture with multiple single-points-of-failure. A detailed reliability analysis, and special emphasis on component selection in early phases of the mission will be required.

The FDIR mechanisms could only trigger the reboot or power cycle of the units. Thanks to latch-up protections, watchdogs, non-volatile safeguard memories and so on, some failures could be recovered with limited downtime. However, permanent failures on critical components would lead to loss of the mission.

Even with a non-redundant architecture, the reliability of the data handling unit during the short time of operation would be relatively high and no major impact on the overall mission reliability is expected.

The processing requirements of the units are relatively small, especially in the case of the penetrator. Both units could highly benefit from the use of small, low-power and low-performance microcontrollers. There has been significant effort in ESA and industry to come up with devices with reduced functionality based on SPARC or ARM with low power consumption, low pin count and small PCB footprint. The development and qualification of those devices is still on-going and they will be available on the market in the following years.

The communication between penetrator warm bay, where the penetrator OBC is located and the cold bay, where some of the scientific instruments are, is achieved by means of wireless data communication with relatively low data throughput (~kbps).

The alternatives for this wireless data link are the following:

- RF communications based on commercial standards such Bluetooth LE or low-power wireless transceivers. Although the terrestrial heritage of those standards is huge, none of those has ever been qualified or used in space. They were developed to cope with completely different scenarios and they're probably too complex for this specific situation
- Communication based on inductive coupling (NFC). Assuming that the cold bay does not have a battery and it has to be powered remotely, the same coils used to

transfer the power could be used for low speed data traffic. Some commercial implementations of remote energy transfer already send limited amounts of data to identify the devices or control the battery charge.

- Wireless optical communications. Optical Wireless Intra-Spacecraft Communications (OWLS) is a promising technology that has already been flown on OPTOS, an optical nanosatellite with CAN bus implemented over an optical network built with qualified LEDs and photodiodes. The technology could be adapted and re-qualified to implement a point to point communication between warm and cold bay of the penetrator.

15.3 Baseline Design

The mission includes two non-redundant and highly miniaturized data handling units, one on the PDS platform and another one in the warm bay of the penetrator.

The PDS OBC is in charge of the PDS platform control and GNC algorithms. The processing requirements for the processor are rather limited, there is no science data and very little platform telemetry. It is assumed that there is no need for a dedicated mass memory and all data is stored in the processor RAM. As mention in the assumptions and trade-offs section, for further mass and power reduction, the design may be microcontroller based.

The mass of the unit is 0.5 kg without housing and the power consumption 5W. The volume around 0.5l, which can be shaped with quite some freedom to fit the configuration needs of the PDS.

The Penetrator CDMU, which is located on the Warm Bay, is in charge of the penetrator platform control. It has to acquire and maybe process and compress the scientific payload data before transfer to ground. Depending on the amount of scientific data, it may be OK to store it in the processor RAM; otherwise a small dedicated mass memory may be needed. The Penetrator CDMU is also in charge of the wireless communication with the Cold bay electronics.

The mass of the unit is 1 kg without housing and the power consumption 8W. The volume is around 1l, which can be shaped with quite some freedom to fit the configuration need of the penetrator.

	mass (kg)	mass margin (%)	mass incl. margin (kg)
PDS_OBC (PDS OBC)	**0.50**	**20.00**	**0.60**
(blank)	0.50	20.00	0.60
PEN_CDMU (Penetrator CDMU)	**1.00**	**20.00**	**1.20**
(blank)	1.00	20.00	1.20
Grand Total	**1.50**	**20.00**	**1.80**

Table 15-1: Data handling mass budget

Power (W)	P_on
PDS_OBC (PDS OBC)	**5.00**
(blank)	5.00
PEN_CDMU (Penetrator CDMU)	**8.00**
(blank)	8.00
Grand Total	**13.00**

Table 15-2: Data handling power budget

15.4 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed
- Technologies available within European non-space sector(s)
- Technologies identified as coming from outside ESA member states.

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
Wireless transceivers		4	Yes	Technology widely available on commercial sector.
Rad-hard microcontroller		4	Yes	Technology widely available on commercial sector

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16 TELECOMMUNICATIONS

16.1 Requirements and Design Drivers

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
COM-010	The communication S/S shall provide TM transmission function to penetrator for housekeeping and scientific data transmission to orbiter during on-site activity.	
COM-020	The communication S/S shall provide TM transmission and TC reception functions to PDS for commanding via orbiter and for housekeeping data and images transmission to orbiter during the descent, till penetrator separation.	
COM-030	Antenna selection and accommodation shall be done in order to maximise link performances.	
COM-040	A minimum elevation angle of 30deg shall be considered for penetrator to orbiter communications, with respect to local horizon. The true minimum elevation angle shall be defined considering antenna characteristic and pointing.	
COM-050	The data return link from penetrator shall allow transmission of all scientific data generated during the post-impact activity plus some housekeeping data.	
COM-060	The data return link from PDS shall allow transmission of the housekeeping data and the landing site images acquired during the descent.	
COM-070	Landing site images are assumed to be acquired starting at an altitude of 35 km and have to be transmitted before impact.	

16.2 Assumptions and Trade-Offs

16.2.1 Assumptions

The following assumptions are made concerning communication windows availability (as per Mission Analysis, see section 5):

- Fly-by 1 : This is a transfer fly-by. No communication or science is foreseen
- Fly-by 2 : Release and impact fly-by. Impact will happen at the pericentre of this fly-by. Communication window is short, centred on pericentre
- Fly-by 3 : 10.5 days after Fly-by 2 – Communications fly-by.

On-board data generation is assumed to be as follow:

- E_PAC : the total data volume generated is 3.048Mbit. The whole science sequence is done in 2426s after impact and data transmission can be started only immediately after. Since orbiter visibility above local horizon ends before this time (see Figure 16-1), it is not possible to download science data during 2nd fly-by. Anyway the short visibility can be used to download some initial TM and data to check penetrator status after impact.
- MSEIS : a data volume of 0.731 Mbit/day is generated continuously for 7 days after impact.

- Some housekeeping is generated inside the penetrator. This is accounted as a margin on top of the scientific data volume mentioned above.

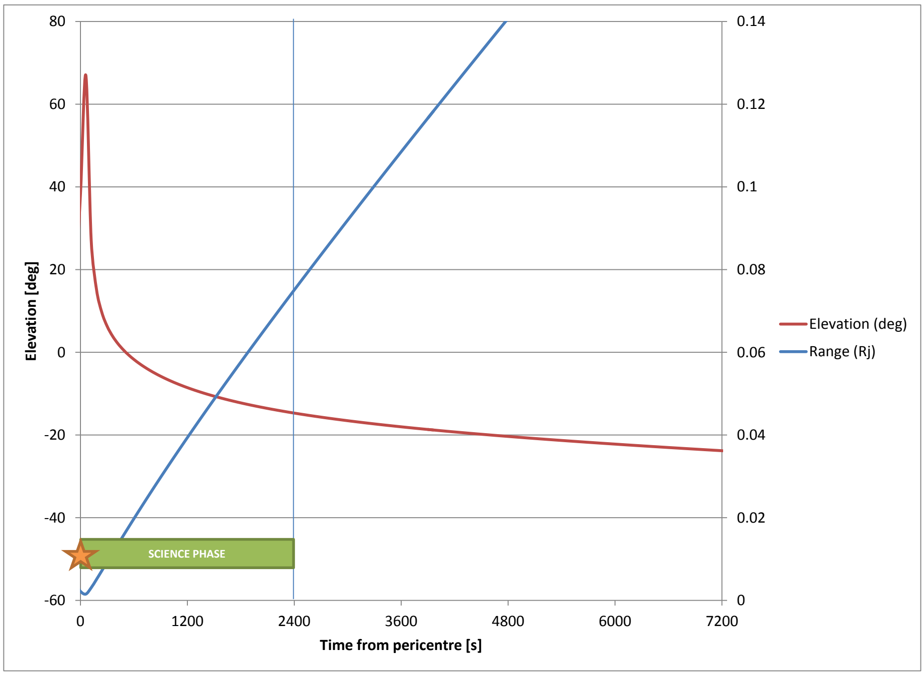

Figure 16-1: Visibility vs data generation during second fly-by

The following are further assumptions considered in the design:

- Redundancy is not required for communication S/S, neither on the PDS nor on the penetrator
- The characteristics of Europa ice are considered unpredictable from RF point of view. In this sense, the performances of a RF link through the ice (i.e. from below the surface) are assumed to be significantly degraded. As the value of this degradation cannot be assessed, a link performed through ice is considered highly risky. Results of studies performed on artic ice can be found e.g. in RD[23] or RD[24]
- The attitude of the penetrator after impact cannot be predicted precisely a-priory. The same applies for any antenna rigidly mounted on the penetrator, thus leading to an unpredictable coverage in terms of minimum elevation angle.

16.2.2 Communication Subsystem Architecture Trade-Off

Two main options have been considered concerning communication subsystem architecture and accommodation of units on PDS and penetrator:

16.2.2.1 Communication s/s on penetrator only

A full communication subsystem is installed on the penetrator. The subsystem will allow commanding and monitoring the penetrator during its mission, but it shall also allow commanding and monitoring the PDS during the descent till penetrator separation. To this purpose, a data link is required between PDS and penetrator.

Furthermore, an additional antenna may be required to be installed on the PDS, on the side facing the orbiter to improve coverage during the descent. Thus, an RF connection will be needed from penetrator to PDS to feed this antenna.

These additional functionalities will increase the complexity of the subsystem in terms of:

- Data processing: the penetrator on-board computer has to exchange information with PDS data handling subsystem to allow TM/TC operation during the descent
- Interfaces between penetrator and PDS: an RF link and data link shall be implemented with relevant separation system

The main advantage of this approach is in terms of mass and power consumption saving.

16.2.2.2 Communication s/s on PDS and data transmitter on penetrator

In this case two separate communication systems are considered.

One full Rx/Tx system is installed on the PDS. This will allow two-way communication between orbiter and PDS during the descent phase. Return link data rate can be optimised considering the specific scenario for images transmission.

A simple Tx only system is installed on the penetrator to allow transmitting the generated scientific data to the orbiter. Since the penetrator does not need to be commanded, a receiver is not needed. Transmission slots (during orbiter fly-by's) can be pre-programmed in the on-board timeline and a simple continuous cyclic transmission of all on-board stored data can be implemented (to cover for the need of re-transmission in case some data get lost).

16.2.2.3 Trade-off result

Considering the complexity of the first solution, the completely different requirements in terms of data return between PDS and penetrator and the uncertainties in the implementation of the connections between PDS and penetrator (data exchange, RF), the first solution is discarded in favour of the second one.

16.2.3 Penetrator antenna trade-offs

Two main options exist for the accommodation of the transmitting antenna on the penetrator:

- An antenna mounted on the structure of the penetrator (in principle on the back panel)
- A deployable antenna to be unfolded on the surface of the ice and connected to the penetrator through an umbilical RF cable.

From communication point of view, the main advantages and disadvantages of the two solutions are shown in the following table:

	Pros	Cons
Fixed	• High stiffness and robustness to shock • High reliability • Low RF losses from Tx due to short connection	• Radiation pattern is affected by surrounding ice (in particular if penetrator remains completely covered by ice). • Visibility to orbiter unpredictable (depends on penetration depth and orientation)
Deployable	• Good visibility to orbiter	• Risk due to deployment (failure of mechanism, unpredictable behaviour during unfolding or landing on the ice layer) • Umbilical cable damage at impact • High RF losses from Tx due to umbilical

Table 16-1: Antenna Configuration Trade-Off Summary

As both solutions present some pros but are not exempt from risks, a mixed approach is considered for the baseline design, where both a deployable antenna and a fixed one are installed. The nominal antenna for communication is the deployable one, as it provides the better performances in terms of link budget. The fixed antenna is used as back-up, in case of failure of the deployment system or of the umbilical cable. As indicated above, its performances will be worse in terms of link budget, therefore a lower data rate is expected in case it will have to be used.

16.2.4 Penetrator to Orbiter Link Trade-Offs

The penetrator communication subsystem performances have been traded-off vs the required minimum data volume to be downloaded. As indicated above in paragraph 16.2.1, 3.048Mbit are generated by E-PAC payloads plus 0.731 Mbit/day for 7 days are generated by seismometer. This leads to an overall data volume of about 8,165 Mbit to be downloaded.

As a result of the trade off, a minimum data rate of 3kbps and a transmitter RF output power of 1W were found to be necessary on-board to meet the requirements. The following Figure 16-2 shows the minimum orbiter elevation angle with respect to penetrator local horizon and computed link budget margin. Considering the assumptions in paragraph 16.2.1 on minimum elevation angle and margin, the link is feasible in the time windows highlighted in blue in the graph. This is about 46min, and allows downloading 8.28Mbit of data, in line with the required data return.

Slight improvement can be obtained by considering a lower elevation angle or slightly higher Tx power.

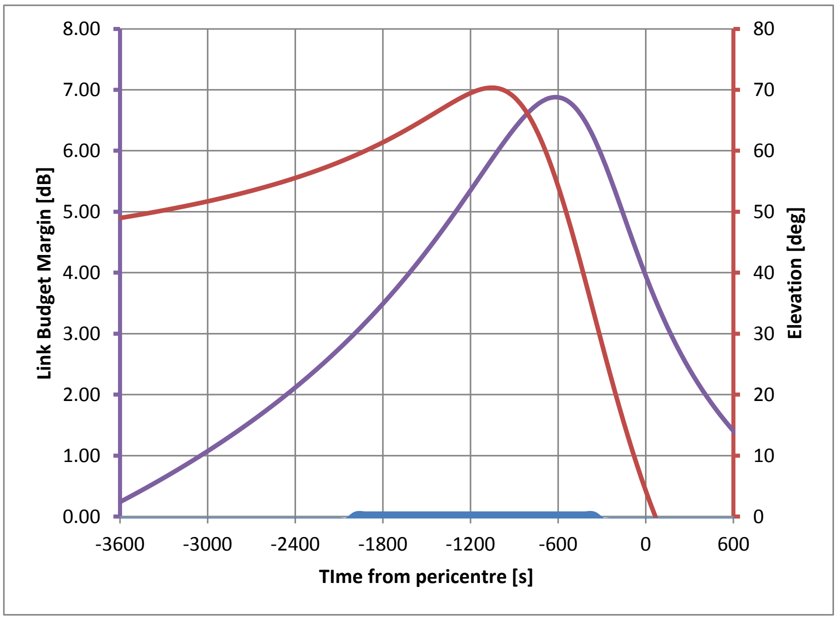

Figure 16-2: 3rd Fly-By visibility and link budget margin analysis

16.2.5 PDS to Orbiter Link Trade-Offs

In order to assess the feasibility of a link from PDS to Orbiter during last phase of the descent, for landing site images transmission before impacts, the visibility and distance between PDS and Orbiter have been analysed.

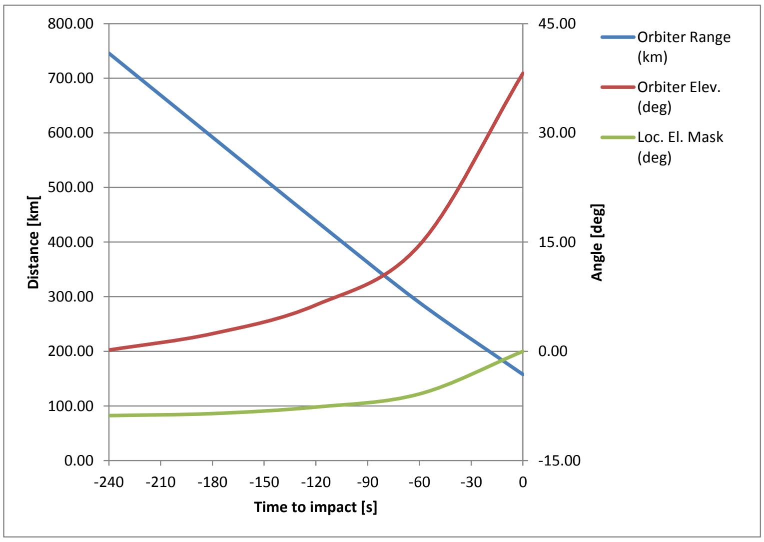

Figure 16-3: Orbiter visibility from PDS during last phase of descent

The distance ranges from 750 to 150 km while elevation is from 0deg to 40deg with a local horizon mask always lower (5deg as a minimum). Therefore link is in principle feasible. Link budget has been sized for maximum distance (see Table 16-5) and a data rate in the order of 100kbps has been found to be achievable. Thanks to this high data rate, a maximum downloadable data volume of 24Mbit (3Mbyte) could be feasible, which should be sufficient to download some pictures acquired just before impact. Data rate can be further increased, by increasing the Tx power which has been fixed to 1W to use same Tx configuration as on penetrator.

16.3 Baseline Design

As described in section 16.2.2, the communication subsystem is split in two independent parts: one allocated in the PDS and one in the penetrator.

Both subsystems will operate in UHF band, in order to minimise the propagation losses. The use of higher frequencies does not give any advantage on the link budget as omnidirectional coverage antennas are required on both sides of the link. In principle both transmitters can operate on the same frequency as they will not be on simultaneously: the PDS Tx will be operated till impact while the penetrator one will be operated after impact.

In terms of communication protocols, there is not the need to follow the proximity-1 standard, since dedicated transmitter and receivers will be used for the whole mission. Thus the protocol can be optimised considering the specific mission needs (e.g. no Rx capabilities on the penetrator).

16.3.1 Penetrator Subsystem

The penetrator communication subsystem is composed by:

- The TM transmitter
- The fixed low gain antenna
- The deployable low gain antenna connected via an umbilical RF cable to the penetrator.

The switch shown in the picture may be part of the transmitter board (in this case two outputs will be available) or can be and external device.

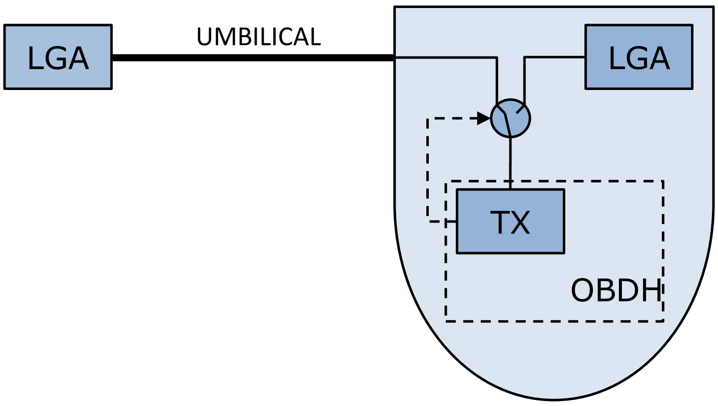

Figure 16-4: Penetrator Communication S/S

As indicated in section 16.2.4, the transmitter shall provide 1W RF power and support a data rate of 3kbps. An additional data rate shall be implemented to support degraded communication via fixed low gain antenna. The selection of the antenna to be used (and consequently of the data-rate) is done by the transmitter itself, based on RF power measurements: in case of failure of the deployable antenna resulting in a lower than expected irradiated power, switching to fixed antenna shall be performed automatically. As baseline, the transmitter will be incorporated in the on-board data handling enclosure, as an additional board. This will allow minimising the mass and occupied volume.

Various options have been considered for the external low gain antenna (e.g. patch, helix, dipoles/monopoles). Among them, a textile antenna solution has been identified as a good candidate for this mission. The antenna is basically a patch built on a fabric support. The advantage of this solution is that the antenna itself can be folded inside a crushable "capsule" which can be released from the penetrator before impact. The antenna will autonomously unfurl on the surface of the ice, when the capsule breaks as the consequence of the impact.

This kind of antenna was studied already in the frame of ESA contracts for use on ground (Figure 16-5). Further development is required to manufacture an antenna able to withstand the harsh environment of Europa.

Figure 16-5: Textile Antenna (L-Band version)

The antenna radiation patter is very similar to that of a patch antenna, as shown in Figure 16-6 below. At 30deg elevation angle above horizon (that is 60deg from boresight in the figure) a gain in the order of 0dB can be expected.

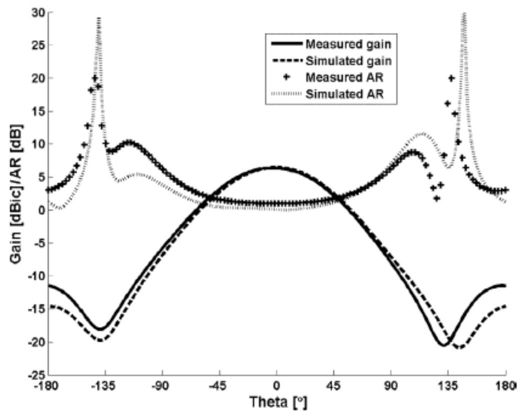

Fig. 13. Measured and simulated gain and axial ratio in XZ-plane at 1.605 GHz.

Figure 16-6: Textile Antenna Radiation Pattern

The back-up LGA can be in principle a patch antenna, to be installed inside the penetrator. As the back side of the penetrator is not usable for installation of the antenna due to the presence of the releasable capsule and umbilical, this back-up antenna will have to be "wrapped" on the penetrator lateral surface. As the antenna has to be installed inside the penetrator, proper slots shall be foreseen in the structure to allow radiating the signal in all directions. As a consequence of this accommodation, the coverage will not be optimal toward the zenith (assuming the penetrator will have a vertical position inside the ice). In any case, further analyses are needed to define the best accommodation and resulting pattern for this antenna, also considering the surrounding ice.

The umbilical cable will be a standard RF cable qualified for low temperatures which shall be reinforced to withstand the impact. As the cable will have to unwrap from the supporting structure on the penetrator in very short time, it shall be quite flexible. This implies that it shall be quite small in diameter and its conductive core shall not be made of solid copper. As a result, the RF losses of such kind of cable will be higher than a typically used low-loss space qualified RF cable. A value in the order of 2dB has for a 10m long cable has been estimated, based on off-the-shelf cables data sheet. Note that RF cables for space applications are qualified up to -180degC. This gives good confidence about their applicability in this specific environment.

16.3.2 PDS Subsystem

The PDS communication subsystem is composed by:

- The TM transmitter
- The TC receiver
- A diplexer
- A fixed low gain antenna.

As shown in figure below, the subsystem is quite simple. The transmitter and receiver are connected together to the LGA through a diplexer, which allow full-duplex operations.

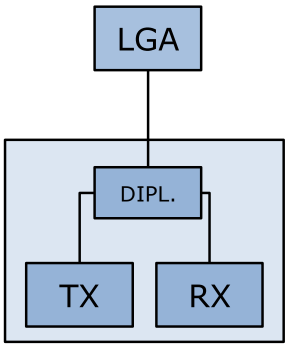

Figure 16-7: PDS Communication S/S

As in the case of the penetrator, it is assumed that both Tx and Rx are two boards incorporated in the same enclosure of the on-board computer. As there is not a stringent requirement on mass and power consumption of the PDS, the Tx and Rx can also be independent units. This alternative approach will increase slightly the mass but will reduce the development risks. As indicated in section 16.2.4, the transmitter shall provide 1W RF power and support a data rate of 100kbps during last phase of the descent. An additional data rate (e.g. 3kbps in line with penetrator) could be implemented to support, with improved margins, nominal communication from orbiter separation till 35 km altitude point where high data rate will be enabled.

The LGA can be a classical patch installed on the face of PDS facing the orbiter during the descent.

16.4 Link Budgets

A preliminary budget for the penetrator to orbiter TM link has been established and results are reported in Table 16-2 and Table 16-3, including also the estimation of orbiter receiver G/T in Table 16-4. Main assumptions concerning on-board parameters can be found in the table. Note that antenna gain is fixed to the worst case value of 0dB and it is not adjusted considering the true elevation.

Worst case results for maximum distance and minimum elevation angle during 3rd fly-by are provided. Results for other ranges have been derived scaling the margin obtained in these two cases.

As can be seen, with the selected on-board power the required minimum margin of about 3dB is obtained.

PARAMETER	VAL.	Notes
ELEVATION ANGLE [deg]	62.0	
RANGE [km]	4824.9	
FREQUENCY [MHz]	450	
MAX BIT RATE [kbps]	3.00	
MAX BIT RATE [dBHz]	34.77	
TX POWER [W]	1.00	
TX LOSSES [dB]	2.05	Preliminary Estimated Value
TX EIRP [dBW]	-0.82	Calculated
PATH LOSSES [dB]	159.18	Calculated
ATMOSPHERE LOSS [dB]	0.00	
RX G/T [dBK]	-20.90	
DEMOD. LOSS [dB]	3.00	
MOD. LOSS [dB]	0.00	
REQIRED Eb/No [dB]	6.80	
MINIMUM MARGIN [dB]	3.14	

Table 16-2: Link Budget at Maximum Distance

PARAMETER	VAL.	Notes
ELEVATION ANGLE [deg]	31.8	
RANGE [km]	3083.2	
FREQUENCY [MHz]	450	
MAX BIT RATE [kbps]	3.00	
MAX BIT RATE [dBHz]	34.77	
TX POWER [W]	1.00	
TX LOSSES [dB]	2.05	Preliminary Estimated Value
TX EIRP [dBW]	-0.82	Calculated
PATH LOSSES [dB]	155.29	Calculated
ATMOSPHERE LOSS [dB]	0.00	
RX G/T [dBK]	-20.90	
DEMOD. LOSS [dB]	3.00	
MOD. LOSS [dB]	0.00	
REQIRED Eb/No [dB]	6.80	
MINIMUM MARGIN [dB]	7.03	

Table 16-3: Link Budget at Minimum Elevation Angle

PARAMETER	VAL.	
S/C RX ANT GAIN [dBi]	5.0	Wide angle antenna - pointed
ANTENNA NOISE TEMP [K]	100.0	TBD
RFDN PHYSICAL TEMP [K]	290.0	Assumption
RFDN LOSS [dB]	1.0	Assumption
Rx NOISE FIGURE [dB]	2.0	Typical value
RX SYSTEM TEMP [K]	308.7	Calculated
RX SYSTEM TEMP [dBK]	24.9	Calculated
NOISE FLOOR [dBm/Hz]	-173.7	Calculated
S/C RX G/T [dB/K]	-20.9	Calculated

Table 16-4: Orbiter G/T Estimation

The link budget for PDS to orbiter link has been also established and worst case results provided in Table 16-5 below, for maximum orbiter to PDS distance at 35 km altitude with maximum data rate.

PARAMETER	VAL.	Notes
ELEVATION ANGLE [deg]	0.2	
RANGE [km]	745.6	
FREQUENCY [MHz]	450	
MAX BIT RATE [kbps]	100.00	
MAX BIT RATE [dBHz]	50.00	
TX POWER [W]	1.00	
TX LOSSES [dB]	2.05	Preliminary Estimated Value
TX EIRP [dBW]	-0.82	Calculated
PATH LOSSES [dB]	142.96	Calculated
ATMOSPHERE LOSS [dB]	0.00	
RX G/T [dBK]	-20.90	
DEMOD. LOSS [dB]	3.00	
MOD. LOSS [dB]	0.00	
REQIRED Eb/No [dB]	6.80	
MINIMUM MARGIN [dB]	4.13	

Table 16-5: Link Budget PDS to Orbiter

16.5 List of Equipment

The following tables provide the mass budget and power consumption for the units composing the communication subsystem, as obtained from OCDT model. The units are allocated into Aft Penetrator (LGA1), Fore Penetrator (Transmitter, LGA2 and relevant harness), PDS (Transmitter, Receiver, LGA and relevant harness) and umbilical.

	mass (kg)	mass margin (%)	mass incl. margin (kg)
Clipper Europa Penetrator (CLEP)	**2.89**	**20.00**	**3.46**
APEN (Aft Penetrator CLEP)	**0.30**	**20.00**	**0.36**
LGA_P_DEPL (Low Gain Antenna Deployable CLEP)	0.30	20.00	0.36
FPEN (Fore Penetrator CLEP)	**0.50**	**20.00**	**0.60**
LGA_FP (Low Gain On Fore Penetrator)	0.30	20.00	0.36
RF_Harness_CLEP (RF Harness CLEP)	0.10	20.00	0.12
Tx_MOD_CLEP (Transmitter CLEP)	0.10	20.00	0.12
PDS (Penetrator Delivery System CLEP)	**1.10**	**20.00**	**1.32**
Rx_CLEP_PDS (Receiver CLEP PDS)	0.35	20.00	0.42
Tx_MOD_CLEP_PDS (Transmitter CLEP PDS)	0.35	20.00	0.42
RF_Harness_CLEP_PDS (RF Harness CLEP PDS)	0.10	20.00	0.12
LGA_PDS (Low Gain Antenna On PDS)	0.30	20.00	0.36
Umbilical_CLEP (Umbilical Cord)	**0.99**	**20.00**	**1.18**
Grand Total	2.89	20.00	3.46

Table 16-6: Mass Budget

Power (W)	P_on	P_stby
Clipper Europa Penetrator (CLEP)	**8.05**	**0.05**
APEN (Aft Penetrator CLEP)	**0.00**	**0.00**
LGA_P_DEPL (Low Gain Antenna Deployable CLEP)	0.00	0.00
FPEN (Fore Penetrator CLEP)	**4.00**	**0.00**
LGA_FP (Low Gain On Fore Penetrator)	0.00	0.00
RF_Harness_CLEP (RF Harness CLEP)	0.00	0.00
Tx_MOD_CLEP (Transmitter CLEP)	4.00	0.00
PDS (Penetrator Delivery System CLEP)	**4.05**	**0.05**
Rx_CLEP_PDS (Receiver CLEP PDS)	0.05	0.05
Tx_MOD_CLEP_PDS (Transmitter CLEP PDS)	4.00	0.00
RF_Harness_CLEP_PDS (RF Harness CLEP PDS)	0.00	0.00
LGA_PDS (Low Gain Antenna On PDS)	0.00	0.00
Umbilical_CLEP (Umbilical Cord)	**0.00**	**0.00**
Grand Total	8.05	0.05

Table 16-7: Power Budget

16.6 Options

No options have been considered in addition to the presented baseline.

16.7 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed
- Technologies available within European non-space sector(s)
- Technologies identified as coming from outside ESA member states.

Equipment and Text Reference	**Technology**	**Suppliers and TRL Level**	**Technology from Non-Space Sectors**	**Additional Information**
LGA (Sect. 16.3.1)	Textile Antenna	N/A	Prototype for ground use (SAR) developed by Patria Aviation Oy, under ARTES 5.1 (see RD[25])	
Transmitter Receiver (Sect. 16.3.1)	Tx & Rx boards, embedded in OBDH	TRL-5 Various suppliers.		Technology already available in space. Main issue is the development of a board compatible to OBDH from EMC point of view and qualification of whole assembly.

17 THERMAL

17.1 Requirements and Design Drivers

17.1.1 PDS

Concerning the PDS, no thermal requirement was explicitly stated. Consequently the requirements that were used to drive the design are either classical, or derived from other subsystems:

- Maintain the units (namely the Propulsion Subsystem – tanks, lines, thrusters) in their operational temperature range during the mission lifetime
- Maintain an acceptable interface temperature for the Penetrator.

17.1.2 Penetrator

The main design driver for to the thermal subsystem for the Penetrator is to ensure the survival of units for an extended period of time (10.4 days) in a cold environment (80K) while limiting the power consumption (on battery only).

This design driver is aggravated by the fact that the thermal architecture chosen shall survive the impact. Mechanical robustness is therefore required, which is often contradictory with the objective of thermally decoupling an enclosure.

17.2 Assumptions and Trade-Offs

17.2.1 PDS

For the PDS, the main assumptions for the evaluation of the design are the following:

- *Thermal Environment:* a worst case of No External Fluxes is considered
- *Temperature Range:* The whole structure and the tanks shall be maintained above 0degC (minimum temperature acceptable by the Propulsion elements)
- *Temperature Margin:* A margin of 10degC is applied (i.e. calculations are performed in order to guarantee a minimal temperature of 10degC).
- *Configuration and dimensions:* The configuration is supposed to be similar to the one selected by ADS for the CLIPPER study with 1 main engine and 4 small thrusters:

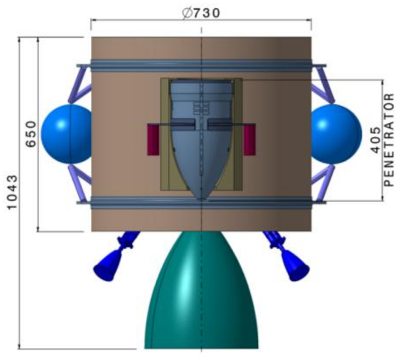

Figure 17-1: PDS Configuration for the CLIPPER study (Courtesy of Airbus DS)

17.2.2 Penetrator

No significant assumption has been taken for the Penetrator (cf. 17.3.2 for more information).

17.3 Baseline Design

17.3.1 PDS

The thermal control of the PDS relies on classical and simple solutions: multi-layers insulation and Kapton foil heating lines. The heating lines can be controlled either by thermistors or thermostat. The MLI has been considered (in performance and mass budget) to be 20 layers.

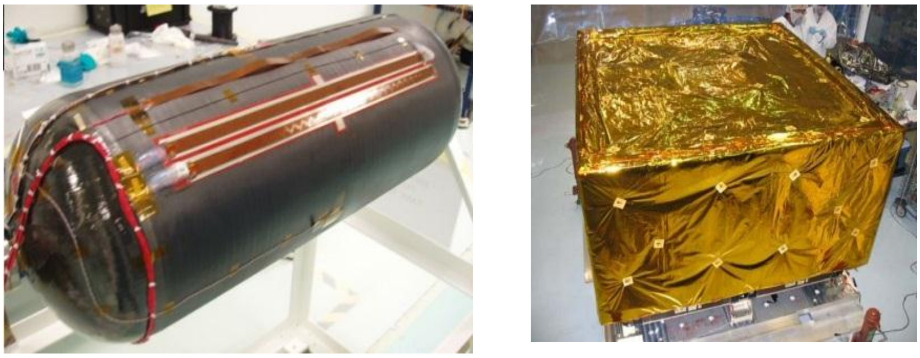

Figure 17-2: Examples of items used for the TCS of the PDS – Left Kapton Heaters, Right MLI

In order to evaluate the power consumed by the active thermal control, one must evaluate the heat leaking to space assuming an inner enclosure at 10degC.

- Leakage through MLI: **~11W** (surface considered: structure + external tanks = ~2m2).
- Leakage through small thrusters: **1.5W per Thruster (4 of them)**
  - Comes from Reduced Thermal Model from Lunar Lander B1.

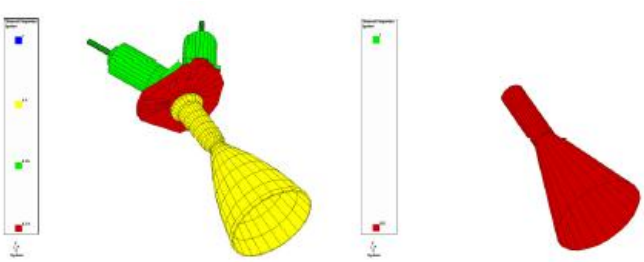

Figure 17-3: Detailed and Reduced Thermal Model of Lunar Lander small Thruster (Courtesy ADS)

- Leakage through main engine: **5W**
  - Comes from Reduced Thermal Model from Lunar Lander B1.

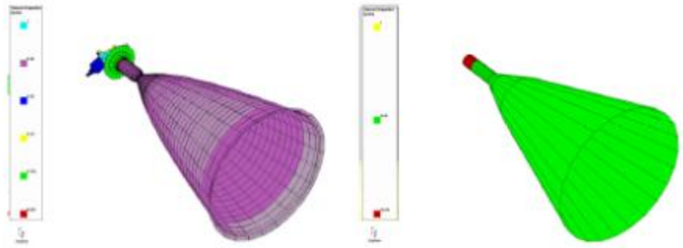

Figure 17-4: Detailed and Reduced Thermal Model of Lunar Lander Main Engine (Courtesy ADS)

  - Budget allocation to the Penetrator: **3W**

→ Total Average Power Consumption: 25W

17.3.2 Penetrator

The design of the Penetrator as conceived by ADS presents multiple advantages:

- The general architecture is sound: 2 enclosures, one cold that dies relatively quickly after the samples collection, and one "warm" that is decoupled and focused on survivability.

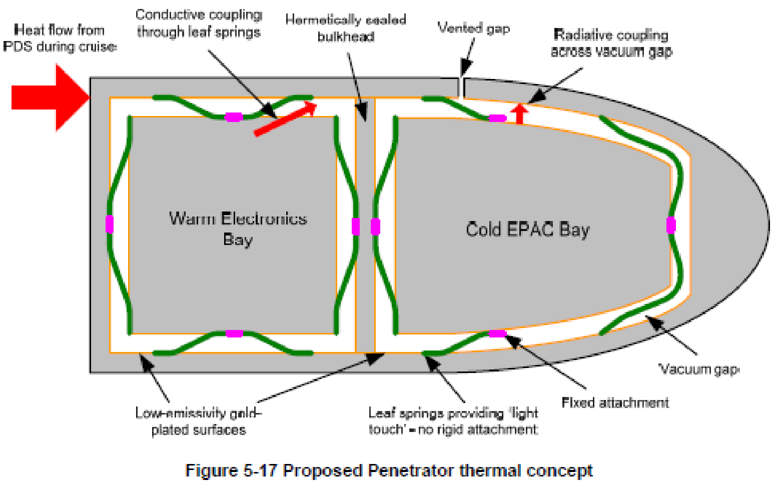

Figure 17-5: Thermal Architecture proposed by ADS for the penetrator (Courtesy of Airbus DS)

- The technical solutions chosen (namely Torlon springs for conductive decoupling) are good.

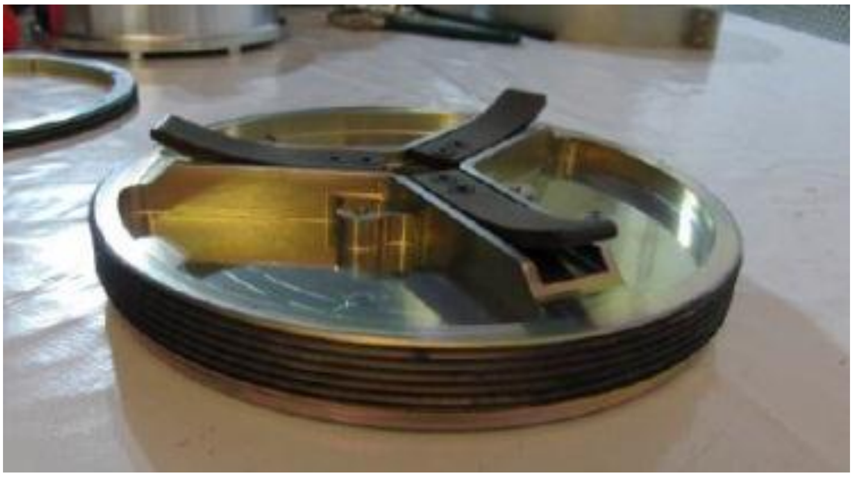

Figure 17-6: Example of Torlon Springs used in the design (Courtesy of Airbus DS)

- Moreover, the critical aspects of the design (survivability of the impact, and part of the thermal performances) have been evaluated by test.

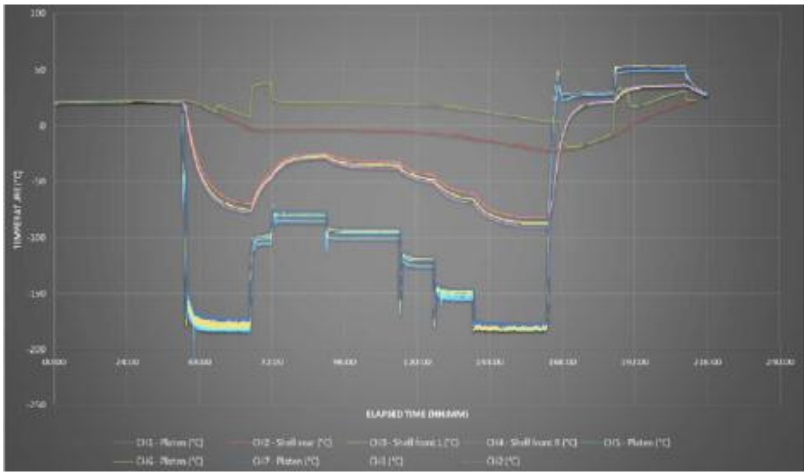

Figure 17-7: Screen capture of the post-impact TBT results (Courtesy of Airbus DS)

Considering the above, the focus of the Thermal activities during this CDF for what concerns the penetrator was put on the in-house evaluation of the thermal performances of the design during the post-impact mission time in order to provide the Power Subsystem with the necessary energy to survive 10.4 days.

17.3.3 Interpretation of the Thermal Balance Test Results

The Thermal Balance test was meant to verify the conductive performance of the Penetrator after the Impact test. The radiative decoupling (low emissivity surface) was not representative (bare metal was used), but the Torlon springs system is comparable to the flight design.

The first task is therefore to 'pseudo-correlate' the Thermal Balance Test using realistic MCp, couplings through the torlon springs as predicted by analysis (total conductance ~5.3mW/K) and emissivity in the range of bare metal (0.2). The results are depicted hereafter:

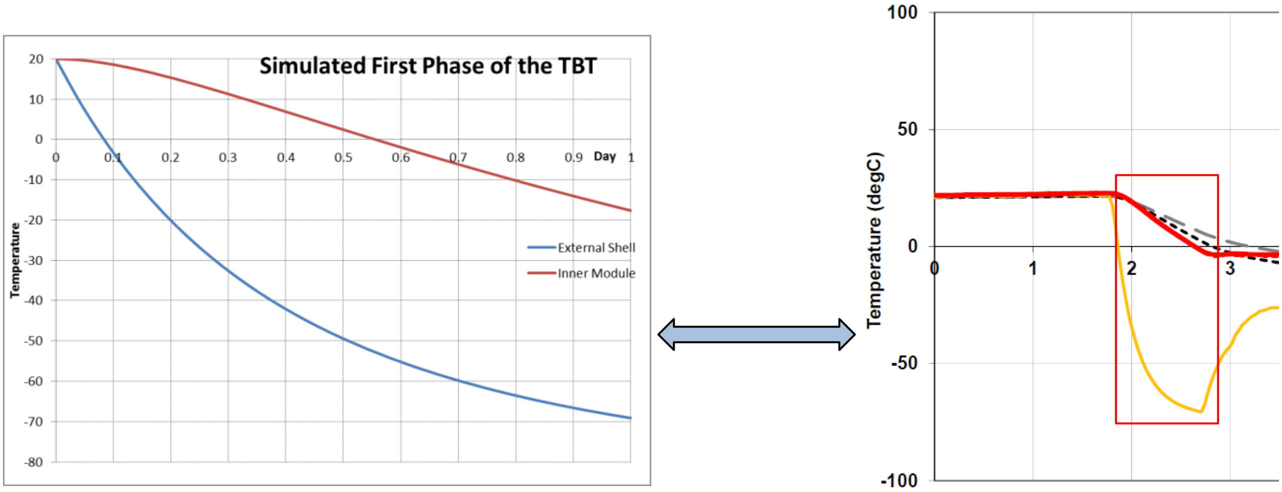

Figure 17-8: Left – Results of the simplified thermal Model, Right – Excerpt of the Thermal Test Report

The 'pseudo-correlation' being satisfactory, it is now possible to predict the necessary power to survive in an 80K environment. The critical parameter for this evaluation is the radiative decoupling between the inner module and the external shell. ASD assumed a low emissivity of 0.03 which assumes a perfectly polished, non-contaminated, non-disturbed (by screws etc.) Vacuum Gold deposited surface.

This assumption is considered pretty optimistic, and a non-contaminated emissivity of 0.05 is considered for both the Inner module and the External Shell:

- Necessary dissipation/heating to maintain the module above 20degC without contamination: 2.5W

Considering that after the impact, the External Shell goes down in temperature below 180K, it is possible that contaminants trapped in the gap (due to outgassing of Torlon for example) deteriorate the emissivity of the External Shell. In order to account for this phenomenon, a calculation has been performed assuming +0.1 emissivity (0.15) on the External Shell:

- Necessary dissipation/heating to maintain the module above 20degC with contamination: 3.2W

In terms of Battery sizing, it means that:

- The Necessary energy to survive 10.4 days ranges between 624Wh and 798Wh depending on the contamination hypothesis.

17.4 List of Equipment

17.4.1 PDS

Element	Quantity	Mass per unit	Total Mass	Power Consumption
MLI	$3.8m^2$	0.5 kg/m2	1.9 kg	N/A
Heaters (+Misc)	N/A	N/A	0.15 kg	25W average 34W peak 75% DC in all modes

17.4.2 Penetrator

Not Applicable (cf. ADS design).

17.5 Options

17.5.1 Penetrator

17.5.1.1 Low Temperature Inner Module

If the internal temperature of the inner module is allowed to go down to -20degC (instead of +20degC), it will affect the necessary heating power in steady state (between 1.7 and 2W depending on the contamination) and it will allow the Penetrator to benefit from the thermal drift between +20degC and -20degC (1.5 days).

- The Necessary energy to survive 10.4 days at -20degC ranges between 367Wh and 432Wh depending on the contamination hypothesis.

18 GROUND SEGMENT AND OPERATIONS

18.1 Requirements and Design Drivers

Launch is in 2022 (MI-GE-070) with a 2.7 years (optionally 7.2 years) (MI-GE-020) Interplanetary Transfer phase plus an 18 months Jovian phase as a hosted payload on the CLIPPER spacecraft before separation and descent. CLEP only has a UHF comms package for the purpose of a relay link with CLIPPER once separated so, for the entire mission, all communications between ESOC and CLEP will be via the CLIPPER MOC.

CLEP is a composite of the Penetrator Delivery System (PDS) and the Penetrator (mounted on the side of the PDS parallel to its spin axis). Separation from CLIPPER will be 1.75 days before CLIPPER's perijove to allow an optimum visibility of the impact at perijove. As specified in the Mission Analysis chapter, "after 6 h for attitude acquisition and rate damping, the targeting manoeuvre is initiated [via pre-loaded, time-tagged commands], followed by a spin-up of the PDS. Based on the accelerometer measurement of the targeting manoeuvre, the time of SRM burn ignition will be updated on board during the following day." The purpose of the accelerometer driven update is to reduce the targeting dispersions caused by uncertainty in the targeting ΔV. There will be approximately 34 hours between the targeting burn and the SRM firing (to bring the PDS to a "stationary point" 35 km above the surface of Europa for the Penetrator release), so there should (shall) be the option to have the update commanding done from ground via a UHF relay link with CLIPPER or let it be done autonomously on-board.

18.2 Assumptions and Trade-Offs

The transfer to Jupiter is assumed to be a "free-ride" in that NASA does not require support (other than possibly ground station support) for CLIPPER operations (including transfer, JOI and PRM), but routine periods will be available for check-out/characterisation of CLEP.

Assuming that there will be a permanent UHF link between CLEP and CLIPPER from the time of separation up to the predicted LOS after the impact depends on the TT&C design of the composite as a whole. In the Astrium design, the PDS has no comms package of its own and must use that of the Penetrator which, from this CDF, will have a fixed LGA in its body for pre-impact comms and one on a deployable umbilical to remain at the surface following impact. In addition, the AOCS baseline assumes the spin-up manoeuvre to be part of the targeting manoeuvre and a spin-rate of 100 RPM. So, can the fixed LGA of the Penetrator on the side of the PDS spinning at 100 RPM support a useable command link with CLIPPER? Probably not, and if the spin-up is part of the targeting manoeuvre, then the updating of the SRM timer cannot be done from ground. Mounting a pair of LGAs on the "top and bottom" of the PDS (the faces perpendicular to the spin axis), for the Penetrator's TT&C subsystem to use whilst attached, would overcome this limitation.

In the nominal case, it is assumed that there will be no need to command CLEP following separation (e.g. the update of the SRM burn timing is done autonomously on-board and monitored from ground) but, as stated previously, there are 6 hours between separation and the targeting burn, and 34 hours between the targeting burn and the SRM firing. In theory, there could be an anomaly during this period that can be recovered from ground, and, if a reliable TT&C link were available, preparations for this would be made. Reliable communications with CLEP via a continuous UHF link with CLIPPER from the point of separation from CLIPPER up until the loss of visibility that occurs after landing, will be assumed.

On the other hand, it is also assumed that should an anomaly cause the timely activation of the targeting or SRM burns to be missed, then the science mission cannot be recovered and that disposal of the composite within the remaining lifetime of the battery is the only other operation to perform.

In effect, the entire composite will be treated as a single planetary probe that has limited commanding possibilities:

- The only TT&C sub-system is on the Penetrator (with supporting antennas on the PDS (see above)) and, once the Penetrator separates, the fate of the PDS is unknown unless it can be observed by CLIPPER,
- Battery-only Power sub-systems on the PDS and Penetrator
- Independent OBC and Data handling sub-systems but each with a very restricted purpose
- AOCS and Propulsion on the PDS only (commanded from ground either via the umbilical with CLIPPER directly or via the Penetrator's TT&C sub-system) and with a very restricted purpose.

There will be a cut-off point before the SRM (final descent) burn after which no more commanding of either module will be attempted.

It is assumed that NASA will be responsible for the design of the CLIPPER orbits to support CLEP's mission and that ESA (ESOC Mission Analysis and Flight Dynamics) will be responsible for the design and implementation of the separation and descent sequence.

18.3 Baseline Design

CLEP will be operated from ESOC in the Solar and Planetary Family of Missions with as much reuse as possible of the mission facilities and data systems infrastructure (deviations from the accepted ECSS standards in the design of the spacecraft increases the cost of operations preparation).

Phase B2 starts in Q3/2017, delivery of the PFM to NASA is in Q1/2021 and launch is by the end of Q2/2022. Post-launch checkout activities to be performed during the CLIPPER Commissioning phase will be agreed with NASA.

The operations of CLEP during the Transfer phase depend a lot on its own design and what is imposed (or not) by CLIPPER. At the very least CLEP will be in hibernation interspersed with routine system and sub-system checkouts/maintenance and instrument characterisation (measurement of dark currents etc.) via OBC automated sequences for data collection and transmission via CLIPPER every few months. As CLEP will be spending up to 2 days separated from CLIPPER and performing its own manoeuvres, however, it is preferred that it be switched on for as much time as possible during the Transfer so as to collect the maximum amount of in-flight data, even in its hosted state.

The FCT will consist of a full-time SOM supported by the equivalent of 2 full-time system engineers on average for the life-time of the mission (manpower may well be shared with other missions depending on the state of other missions at that time).

CLEP's final switch-on will be no later than at separation minus 10 days. The final phase will be operated under LEOP conditions for which a B-team of engineering manpower shared from other missions will be trained up to support the separation and descent activities. The ESTRACK Deep Space ground stations will support the NASA DSN to ensure full-time redundant coverage of CLIPPER at this time.

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19 RISK ASSESSMENT

19.1 Reliability and Fault Management Requirements

The following reliability and fault management requirements were proposed for the CLEO mission

ID	Requirement
CLEO and 'orbiter'	
MI-GE-NEW	The overall reliability of the CLEO mission shall be ≥ 85% at end of life as defined in MI-GE-170. (TBD*)
MI-GE-160a	Single-point failures with a severity of catastrophic or critical (as defined in ECSS-Q-ST-30C/40C) shall be eliminated or prevented by design.
MI-GE-160b	Retention in the design of single-point failures of any severity rating is subject to formal approval by ESA on a case-by-case basis with a detailed retention rationale.
MI-GE-NEW	A failure of one component (unit level) shall not cause failure of, or damage to, another component or subsystem within CLEO or across the interface to the CLIPPER S/C.
MI-GE-NEW	The failure of an instrument shall not lead to a safe mode of the S/C.
MI-GE-NEW	The design shall allow the identification of on-board failures and their recovery by autonomously switching to a redundant functional path. Where this can be accomplished without risk to spacecraft and instrument safety, such switching shall enable the continuity of the mission timeline and performance.
MI-GE-NEW	Where redundancy is employed, the design shall allow operation and verification of the redundant item/function, independent of nominal use.
MI-GE-170	The lifetime of CLEO shall be compatible with the longest mission duration resulting from the mission trajectories selected, including contingencies, and including the phases where CLEO is attached to CLIPPER.
CLEO penetrator	
MI-GE-160c	Single-point failures shall be avoided in the spacecraft design.
MI-GE-160b	Retention of single-point failures in the design shall be declared with rationale and is subject to formal approval by ESA.

**** To Be Discussed***

Table 19-1: Reliability and Fault Management Requirements

The requirements were reviewed during the course of the study and found to be adequate for CLEOP orbiter and CLEOP penetrator.

The suitability of a quantitative requirement related to 'reliability' for a robotic exploration mission was questioned and will be discussed in a follow up phase of the study.

19.2 Risk Management Process

Risk management is an organised, systematic decision making process that efficiently identifies, analyses, plans, tracks, controls, communicates, and documents risk in order to increase the likelihood of achieving the project goals. The procedure comprises four fundamental steps RD[26]:

- Step 1: Definition of the risk management policy which includes the project success criteria, the severity & likelihood categorisations, and the actions to be taken on risks
- Step 2: Identification and assessment of risks in terms of likelihood and severity
- Step 3: Decision and action (risk acceptance or implementation of mitigating actions)

- Step 4: Communication and documentation

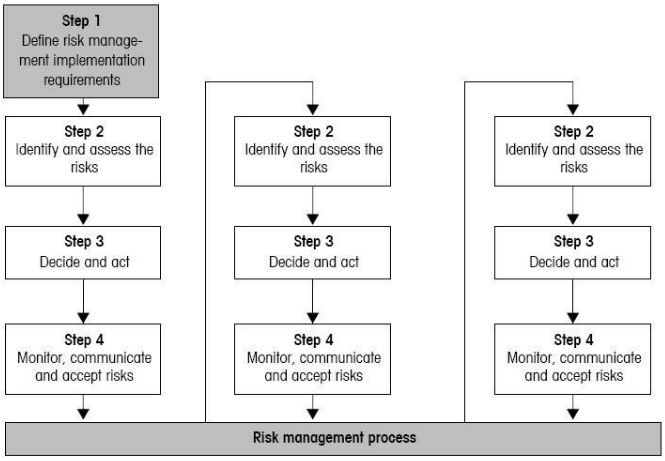

Table 19-2: ECSS-M-ST-80C, 2008 Risk Management Process

Hence the study is still pre-mature the results all 4 steps has to be seen as preliminary as well and a full documentation of the Risk assessment was waived.

19.3 Risk Management Policy

The CDF risk management policy for CLEO aims at handling risks which may cause serious science, technical, schedule and/or cost impact on the project.

19.3.1 Success Criteria

The success criteria with respect to the science, technical, schedule, and cost objectives are presented in Table 19-3:

Domain	Success Criteria
Science + Technical	SCI1. The mission accomplishes the key science goals (Exploration of Io, its surface including geological activities - Io flyby's) TEC1. The SC operates successfully over the designated mission lifetime. TEC2. No performance degradation owing to SPF, and no failure propagation. TEC3. A reliability of >85% at the end of mission as defined in MI-GE-170. (TBD)
Planetary protection	PRO1. The mission is compliant with the ESA Planetary Protection Requirements
Schedule	SCH01 The mission schedule is compatible with the expected launch date (launch is no later than 2022) SCH2. Achieve TRL ≥ 5 at the time of mission adoption (end 2018) SCH3. Low development risk during Phase B2/C-D.
Cost	COS01-The mission is compatible with the ESA M5 CaC boundary

Table 19-3: Success Criteria

The applicability of reliability-related mission success criteria TEC03 has still to be discussed in connection with adequate requirement (MI-GE-170; see para 1.1).

As shown in para. 1.5 the available time for CLEO orbiter and CLEO penetrator (6 years till Clipper launch scheduled by NASA) appears to be very short. The ESA CaC boundaries might have to be re-defined depending on from NASA's position regarding a possible shift of the launch date.

19.3.2 Severity and Likelihood Categorisations

The risk scenarios are classified according to their domains of impact. The consequential severity level of the risks scenarios is defined according to the worst case potential effect with respect to science objectives, technical performance objectives, schedule objectives and/or cost objectives.

In addition, identified risks that may jeopardise and/or compromise the CLEO orbiter and CLEO penetrator mission will be ranked in terms of likelihood of occurrence and severity of consequence.

The scoring scheme with respect to the severity of consequence on a scale of 1 to 5 is established inTable 19-4, and the likelihood of occurrence is normalised on a scale of A to E in Table 19-5.

Score	Severity	Science	Technical / Protection	Schedule	Cost
5	Catastro-phic	Failure leading to the impossibility of fulfilling the mission's scientific objectives	Safety: Loss of life, life-threatening or permanently disabling injury or occupational illness; Severe detrimental environmental effects. Loss of CLIPPER system *, launcher or launch facilities Protection: violation of planetary protection	Delay results in project cancellation	Cost increase result in project cancellation
4	Critical	Failure results in a major reduction (70-90%) of mission's science return	Safety: Major damage to flight systems, major damage to ground facilities; Major damage to public or private property; Temporarily disabling but not life- threatening injury, or temporary occupational illness; Major detrimental environmental effects Dependability: Loss of mission	Critical launch delay (24-48 months)	Critical increase in estimated cost (100-150 M€)
3	Major	Failure results in an important reduction (30-70%) of the mission's science return	Safety: Minor injury, minor disability, minor occupational illness. Minor system or environmental damage Dependability: Major degradation of the system	Major launch delay (6-24 months)	Major increase in estimated cost (50-100 M€)
2	Significant	Failure results in a substantial reduction (10-30%) of the mission's science return	Dependability: Minor degradation of system (e.g.: system is still able to control the consequences) Safety: Impact less than minor	Significant launch delay (3-6 months)	Significant increase in estimated cost (10-50 M€)
1	Minimum	No/ minimal consequences (<10% impact)	No/ minimal consequences	No/ minimal consequences (1-3 month delay)	No/ minimal consequences (<10 M€)

** the severity classification of the consequences has to be aligned with NASA*

Table 19-4: Severity Categorisation

Score	Likelihood	Definition
E	Maximum	Certain to occur, will occur once or more times per project.
D	High	Will occur **frequently**, about 1 in 10 projects
C	Medium	Will occur **sometimes**, about 1 in 100 projects
B	Low	Will occur **seldom**, about 1 in 1000 projects
A	Minimum	Will **almost never** occur, 1 in 10000 projects

Table 19-5: Likelihood Categorisation

The severity classification of the loss of the Clipper-mission due to failure in CLEO orbiter/ penetrator has to be aligned with NASA.

19.3.3 Risk Index & Acceptance Policy

The risk index is the combination of the likelihood of occurrence and the severity of consequences of a given risk item.

The CLEP is an exploration mission with an inherently higher risk potential. Accordingly the generic Risk Index was adapted and a wider range of risk is considered acceptable (adapted Risk Index).

The generic risk ratings (see Table 19-6b) of
* very low risk (green),
* low risk (yellow),
* medium risk (orange),
* high risk (red), and
* very high risk (dark red)
were adapted as follow:
* very low risk (green),
* low/ medium risk (yellow),
* high risk (orange), and
* very high risk (dark red)
assigned based on the criteria of the adapted risk index scheme (see Table 19-7b).

The level of criticality of a risk item is denoted by the analysis of the adapted risk index. By policy very high risks are not acceptable and must be reduced (see Table 19-8).

Severity					
5 (catastr.)	A5	B5	C5	D5	E5
4 (critical)	A4	B4	C4	D4	E4
3 (major)	A3	B3	C3	D3	E3
2 (signif.)	A2	B2	C2	D2	E2
1 (minor)	A1	B1	C1	D1	E1
	A (min.)	B (low)	C (medium)	D (medi.)	E (max.)
					Likelihood

Table 19-6a: generic Risk Index

Severity					
5 (catastr.)	A5	B5	C5	D5	E5
4 (critical)	A4	B4	C4	D4	E4
3 (major)	A3	B3	C3	D3	E3
2 (signif.)	A2	B2	C2	D2	E2
1 (minor)	A1	B1	C1	D1	E1
	A (min.)	B (low)	C (medium)	D (medi.)	E (max.)
					Likelihood

Table 19-7b: adapted Risk Index

adopted Risk Index	Risk Magnitude	Proposed Actions (during assessment phase)
E4, E5, D5	Very High Risk	**Unacceptable risk:** implement mitigation action(s) - either likelihood reduction or severity reduction through new baseline
E3, D4, C5	High Risk	**Unacceptable risk:** implement mitigation action(s) with responsible
E2, D3, C4, B5	Medium Risk	**Acceptable risk:** control, monitor
E1, D1, D2, C2, C3, B3, B4, A5	Low Risk	
C1, B1, A1, B2, A2, A3, A4	Very Low Risk	**Acceptable risk:** *see above*

Table 19-8: Proposed Actions

19.4 Risk Drivers

The following risk drivers have been considered in the identification of specific risk items:

- New technologies
- Environmental factors
- Design challenges
- Reliability issues (TBD), single point failures (SPFs)
- Major mission events
- Programmatic factors

19.5 Top Risk Log (preliminary)

Top risk items have been preliminary identified at the mission (ESA) levels. Please refer to Table 19-9a, b for a complete list of preliminary identified top risks and their corresponding suggested mitigating actions. Risk index results are summarised in Table 19-10a, b.

appl. option	Identified risks and preliminary risk assessment; RISK policy (TBD): [red - > uncond. unacceptable], [orange -> cond. unaccep.),	risk mitigation and prelimenary assessment [yellow/ green -> acceptable]	generic / adapted (Tab. 1-6a / 1-6b) RISK index
CLEO /I + /E + /P	**R1 - Unrealistic schedule** (design/ factory/ qualification of CLEOP) launch date of 2022 of CLIPPER mission specified by NASA; the project schedule has to be in line with the launch date **basel. RISK:** **Schedule**(program.) - design/ realisation/ qualification needs more time than available till 2022 likel.: **max.** / sev.: **catastr.*** -> **very high risk** ** ESA-payload rejected by NASA*	basel. MITIG.: negotiationt with NASA to adopt launch date * no agreement: b1) smaller probe (proba-like) * agreement: b2) change into a M class mission **remain. RISK**: b1) **Cost** - increase (10-40 Mill. EUR ??) likel.: **max.** / sev.: **significant** -> **medium risk** b2) **risk removed**	basel. RISK: R1 -> E5sh-p remain. RISK: R1a -> E2c R1b -> removed
CLEO /I + /E + /P	**R2 - Launcher uncertainties** (2 launcher possibilities: SLS vs. Atlas 5/Delta IV heavy) * design life time of CLEO has to be in line with transfer time to Jupiter incl. 1,5a Jupiter orbit before separation of CLEO ; travel time specified by NASA: 2.7 years direct transfer (SLS) vs. 7.2 years transfer * CLEO design has to be aligned with the launcher environment which is different for the both launcher possibilities **basel. RISK:** **Schedule** - uncertainty for design baseline/ start likel.: **max.** / sev.: **catastr.** * (with respect to unrealistic. schedule -> R1) -> **very high risk** ** ESA-payload rejected by NASA*	basel. MITIG.: a) design for worst case life time and has to cover environment of both launcher b) project schedule for worst case life time and has to cover environment of both launcher **remain. RISK:** a) **Cost** - increase of cost due to worst case life time/ launcher environment (10-40 Mill. EUR) likel.: **max.** sev.: **major** -> **high risk** b) **Schedule** (program.) -> extension of proj. schedule due to worst case life time/ launcher environment (TBC) -> further decision in connection with (**R1**)	basel. RISK: R2 -> E5sh remain. RISK: R2a -> E3c R2b -> R1a/b (see R1)
CLEO /I + /E	**R4.1 - mass budget** (>275kg) mass budget specified by NASA (250kg) **basel. RISK:** **Schedule**(program.) .. classic sat. design is exceeding given size specification likel.: **max.** / sev.: **catastr.*** / -> **very high risk** ** ESA-payload rejected by NASA*	basel. MITIG.: negotiation of available mass budget with NASA * no agreement: a1) reduction of science payload/ return a2) reduction of fly-bys to one * agreement: b) no further mitigation needed **remain. RISK:** a1) **Science** - reduction of science return likel.: **max.** / sev.: **signif.** -> **medium risk** a2) **Science** - reduction of science return likel.: **max.** / sev.: **signif.** -> **medium risk** b) **risk removed** ** ESA-payload rejected by NASA*	basel. RISK: R4.1 -> E5sp remain. RISK: R4.1a -> E2sc R4.1b -> removed

CLEO /I + /E + /P (TBC for /E + /P)	**R4.2 - power budget (DOR)** (~33W) power budget (20W) specified by NASA **basel. RISK**: **Schedule**(program.) .. power budget (DOR) is exceeding given power budget likel.: **max.** / sev.: **catastr.*** /-> **very high risk** * *ESA-payload rejected by NASA*	basel. MITIG.: negotiation of available power budget with NASA * no agreement: a) re-design of propulsion/ mission (main consumer of power) * agreement: b) no further mitigation needed remain. RISK: a) **Science** - reduction of science return likel.: **max.** / sev. **signif.** -> **medium risk** b) **risk removed** *	basel. RISK: R4.2 -> E5sh-p remain. RISK: R4.2a -> E2sc R4.2b -> removed	
CLEO /I /+E	**R5 - Radiation impact** (100krad) expected radiation level requires consideration of shielding options, rad-hard components, trajectory limitation etc.); 'rad'-sensitive components are e.g. transponders, giros, ... **basel. RISK:** **Schedule** (program.) due to additional mass for shielding -> see RISK related to mass budget likel.: **max.** / sev.: **catastr.*** -> **very high risk** * *ESA-payload rejected by NASA*	basel. MITIG.: negotiation of available mass budget with NASA * no agreement: -> R4.1 a1) reduction of science payload/ return -> R4.1 a2) reduction of fly-bys to one * agreement: -> R4.1 b) no further mitigation needed **remain. RISK:** a1) -> **R4.1** a1) -> **medium risk** a2) -> **R4.1** a2) -> **medium risk** b) -> **R4.1 b)** -> **risk removed** *	basel. RISK: R5 -> E5sh-p remain. RISK: R5 -> E2sc R5 -> removed	
CLEO /I + /E	**R6.1 - Hibernation strategies** for Jovian cruise limitation of power (see also **R2.1**) demands hibernation strategy for Jovian cruise basel. RISK: **Science/ Tech.** - Jovian cruise - loss of mission due to failure of wake up after hibernation likel.: **high** / sev.: **catastr.** -> **very high risk** *	basel. MITIG.: Jovian cruise - carefully selection of hibernation strategy **remain. RISK.:** **Science** - loss of mission likel.: **medium** / sev.: **catastr.** -> **high risk** *	basel. RISK: R6.1 -> E5sc-t remain. RISK: R6.1 -> C5sc Risk addressed: **MI_02**	
CLEO /I + /E	**R6.2 - limited communication redundancy** during fly by limitation of power (see also **R2.1**) demands cold redundancy for communication receiver during fly-bys **basel. RISK:** **Science/ Tech.** - loss of mission due to lim. com.-redund./ science return (no communication) likel.: **high** / sev.: **catastr.** -> **very high risk** *	basel. MITIG.: fly-by - carefully design of automatic contingency procedure for activation of cold redund. remain. RISK.: **Science** - total loss of science return likel.: **medium** / sev.: **catastr.** -> **high risk** *	basel. RISK: R6.2 -> E5sc-t remain. RISK: R6.2 -> C5sc	

CLEO /I + /E	**R14 - active fly-by strategy** alternating science phase/ data transmission(to Clipper)/ re-positioning manoeuvre/ power recharge-mode including hibernation phase (1 day) and DTE* modes (2 hours) is a **very active fly-by strategy** incl. low margin in delta-V/ propellant and limited battery capacity; **basel.: RISK:** **Science** - loss of mission before finishing of complete science program likel.: **high** / sev. **critical** -> **high risk** ** AOCS/ TM house keeping*	**basel. MITIG.:** a) carefully panning of fly-by strategy (increase of time/ capacity margins) b) autonomous contingency strategy c) limitation of lower level of battery discharge by 20% **remain. RISK:** a+b+c) **Science** - loss of mission before finishing of complete science program likel.: **medium** / sev. **critical** -> **medium risk** ** AOCS/ TM house keeping*	basel. RISK: R14 -> D4sc remain. RISK: R14 -> C4sc	
CLEO /I	**R15 - PL contamination/ impact during fly-by** geological activities (e.g. active volcanos) of Io before and during fly over (flyby distance 200km; 7.41km/s; pyroclastic material up to 500 kilometres into space above Io) basel. RISK: **Science** - damaging of platform (solar areas) and instruments (camera), impact on trajectory incl. increased friction with impact on trajectory likel.: **high** / sev.: **critical** -> **high risk**	**basel. MITIG.:** a) trajectory should not pass over active areas b) increase of fly-by attitude > app. 500km **remain. RISK:** a) **Science** - damaging of platform (solar areas) and instruments (camera), impact on trajectory likel.: **medium** / sev.: **critical** -> **medium risk** b) **Science** - lower data quality due to increased monitoring distance likel.: **medium** / sev.: **signif. .. major** -> **low risk** *	basel. RISK: R15 -> D4sc remain. RISK: R15a -> C4sc R15b -> C3sc	
CLEO /I + /E	**R16.2 - Planetary protection** during fly-by (of CLIPPER & CLEO/I + /E) compliance to COSPAR cat. III (fly-by; baseline is for CLIPPER) requirements (de-contamination) (overall requ. Poc < 10-4 (prob. of bio contamination) **basel. RISK:** **Planetary protection*** - CLEO/I and CLEO/E is the currently undefined planetary protection approach for the CLIPPER (at least as described in the NASA SALMON-2); NASA might require some bioburden control for the orbiters to protect CLIPPER from recontaminatio; the consequence of that is a major cost increase for CLEO/I and significant for CLEO/E. - CLEO/I+E could cause biological planetary contamination during fly-by (/I) and penetration (/E) - violation of planetary protection likel.: **high** / sev.: **catast.** -> **very high risk** **** the currently undefined planetary protection approach for the CLIPPER (at least as described in the NASA SALMON-2) must be seen as a major risk for CLEO/I and CLEO/E*** *	basel. MITIG.: consideration planetary protection (ESA) requirements, resulting into e.g. sterilisation (/E) ** **remain. RISK:** **Cost** - increased cost for /I (0-10Mil. EUR)*** likel.: **max.** / sev.: **major** -> **high risk** **Protection** - CLEO/I and CLEO/E do not to meet the planetary protection requir. (nobody has so far built a bioburden controlled orbiter and ensured re-contamination protection) likel.: **med.** / sev.: **catast** -> **high risk**	basel. RISK: R16.2 -> E5po remain. RISK: R16.2 -> E3c, C5po	

CLEO /I + /E + /P	**R18** - Cost uncertainty by **industrial geo-return** the financing strategy: * small supplier -> lower cost; impact on geo return * bigger industrial supplier -> higher overhead cost however better geo return impacts the overall cost and project acceptance basel. RISK: **Cost** - cost increase due to consideration of geo-return by choosing bigger industrial partner* likel.: **50/50 -> high** / sev.: **minor** -> ..	basel. MITIG.: no further mitigation needed	basel. RISK: R18 -> D1c	

Table 19-9a: Risk Log applicable for CLEO orbiter + penetrator

appl. option	**Identified risks and preliminary risk assessment; RISK policy** (TBD): **[red - > uncond. unacceptable], [orange -> cond. unaccep.),**	**risk mitigation and prelimenary assessment** **[yellow/ green -> acceptable]**	**generic / adapted** (Tab. 1-6a / 1-6b) **RISK index**
CLEO /P	**R7** - Uncertainties in **material properties of the ice** (e.g. hardness) and its **testability (unknown) environm. conditions** final position of penetration and use of instruments **basel.: RISK:** **Science/ Tech.** - final position of penetr.: * on ice surface or not deep enough -> instruments do not work; limited science return (on surface organic mat. destroyed by rad.) * to deep (communication limited or not possible due to ice contamination) see also **R11** likel.: **high** / sev.: **catastroph.** - **very high risk** *	**basel. MITIG.:** a) impact angle and design (material, shaping) of penetrator ~~b) comprehensive test programme for penetrator~~ **remain. RISK:** a) **Science** - final position of penetrator on ice, not deep enough or to deep likel.: **high** / sev.: **critical** - **high risk** ~~b) **Schedule/ Cost** - increased time/ costs for test, develop. and design (100-200Mill. EUR)~~ ~~likel.: **max.** / sev.: **critical** - **very high risk**~~	basel. RISK: R7 -> D5sc remain. RISK: R7 -> D4sc
CLEO /P	**R9.1** - **high impact load** on penetrator* (> 300m/s) the expected impact speed is exceeding the specified value of 300m/s **basel. RISK:** **Science/ Tech.** - damage of penetrator likel.: **max.** / sev. **catastr.** -> **very high risk** ** Remark: several penetrator concept - even realisation - are available; however not of the penetrator could demonstrate finally its suitability because of failure/problems in mission outside of penetrator.(however NASA lost Deep Space penetrator 1999)*	**basel. MITIG.:** adequate propulsion system/ strategy; carefully selection of trajectory/ landing side **remain. RISK:** **Schedule** (program.) - mass budget) due to additional propulsion -> **R4.1 mass budget;** however so far spec. 250kg not exceeded for /E-opt. likel.: **med.**/ sev.: **sign.** -> **low risk**	basel. RISK: R9.1 -> E5sc-t remain. RISK: R9.1 -> C2sh
CLEO /P	**R9.2**- **high impact load** on instrument* (>300m/s) the expected impact speed is exceeding the specified value (300m/s) **basel. RISK:** **Science/ Tech.** - damage of instruments likel.: **max.** / sev. **catastr.** -> **very high risk** ** Remark: several penetrator concept - even realisation - are available; however not of the penetrator could demonstrate finally its suitability because of failure/problems in mission outside of penetrator.(however NASA lost contact to Deep Space 2 penetrator 1999)*	**basel. MITIG.:** use of NAV cam/ accelerator measurement/ attitude indication **remain. RISK:** **Science/ Tech.** - damage of instruments (however decreased likel.) likel.: **med.** / sev.: **catastr.** -> **high risk** **Cost** - additional equipment likel.: **max.** / sev. **minor** -> **low risk** ** Remark: several penetrator concept - even realisation - are available; however not of the penetrator could demonstrate finally its suitability because of failure/problems in mission outside of penetrator.(however NASA lost contact to Deep Space 2 penetrator 1999)*	basel. RISK: R9.2 -> E5sc-t remain. RISK: R9.2 -> C5sc-t E1c

CLEO /P	**R10 - Limited survival time of the penetrator's** - unknown effect of cold ice (heat transfer) in "cold bay" instruments are hosted, however very low temperatures impact the life time of instruments basel.: RISK: **Science** - underestimation of heat transfer leads to lower temp. in cold bay as designed; early failure of instruments and could reduce science return likel.: **high** / sev.: **major** -> **medium risk** *	basel. MITIG.: design with margins to ensure sufficient contingency time to extract and analyse sample. remain. RISK: **Cost** - increased costs for design (10-40 Mill EUR??) likel.: **max.** / sev.: **signif.** **-> medium risk**	basel. RISK: R10 -> D3sc remain. RISK: R10 -> E2c		
CLEO /P	**R11** - **Ice RF transparency** issues (ice contaminations), penetrator attitude in the ice impact on communications link (need for omnidirectional coverage?) and impact of ice contamination on RF transparency **basel. RISK:** **Science** - uncertainties in the science return due to limitation/ failure of communication likel.: **max./** sev.: **catast.** -> **very high risk**	basel. MITIG.: a.) choice of most suitable frequency option b.) buoyant antenna separates from penetrator which should remains near to surface + umbilical cord + speed breaking shape of penetrator **remain. RISK**: a) **Science** - uncertainty of depth of penetration/ kind of ice contamination remains high likel.: **high** / sev.: **catast.** -> **very high risk** b) **Science\Tech** - separation triggering/ mechanism, position of antenna, stability of antenna connection, uncertainty of depth of penetration; uncertainty of penetrator integrity, depth of penetration/ radiation likel.: **high** / sev.: **critical** **->** **high risk** *	basel. RISK: R11 -> E5sc remain. RISK: R11a -> D5sc. R11b -> D4sc-t		
CLEO /P	**R12** - **Communication** window for science return communication will be possible only during the orbiter fly-by (waiting time 14 days is impacting data return success) **basel.: RISK:** **Science** - limitation of science return due to limited communication window (limited battery cap.) likel.: **high** / sev.: **major** -> **medium risk**	basel. MITIG.: increase of battery size/ capacity **remain. RISK:** **Cost** - increase of cost due to additional equipment (0-10Mill. EUR) likel.: **max.** / sev.: **minor** -> **low risk**	basel. RISK: R12 -> D3sc remain. RISK: R12 -> E1c		
CLEO /P	**R16.1** - **Propulsion module impact** on Europa after separation from penetrator. Planetary protection is impacted by propulsion module of penetrator in case it is not cleaned accordingly **basel. RISK:** **planet. protection** - violation of planetary protection likel.: **max.** / sev.: >**minor** -> **medium risk** .. **very high risk** *	basel. MITIG.: cleaning according COSPAR cat. IVb **remain. RISK:** **Cost** - increased cost due to COSPAR IV requirements for propulsion module (1-10 Mill. EUR) likel.: **max.** / sev.: **minor** -> **low risk**	basel. RISK: R16.1 -> E5po remain. RISK: R16.1 -> E1c		

CLEO /P	**R17.1 - Planetary protection** during impact. (penetrator) compliance to COSPAR cat. IVb (landing/ penetration) (overall requ. Poc < 10-4 (prob. of bio contamination) **basel. RISK:** **Planetary protection** - CLEO/P could cause biological planetary contaminate during fly-by (/I) and penetration (/E) - violation of planetary protection likel.: **max.** / sev.: **>minor** -> **medium risk** .. **very high risk**	**basel. MITIG.:** consideration planetary protection (ESA) requirements resulting into e.g. sterilisation + bio shield toward CLIPPER ** **remain. RISK:** **Cost** - increased cost for /I (0-10Mil. EUR)* likel.: **max.** / sev.:**sign** -> **medium risk** **Protection** - CLEO/P does not to meet the planetary protection requirements likel.: **low.** / sev.: **catast** -> **medium risk** * *assumed PL(ESA) costs:* *** planetary protection requirements are already part of the baseline; however they are mentioned here for formal demonstration of risk mitigation* *	basel. RISK: R17.1-> E5po remain. RISK: R17.1 -> E2c, B5po		
CLEO /P	**R17.2 - Planetary protection** during impact (propulsion module of penetrator - PDS) Propulsion module impact on Europa after separation from penetrator. (overall requ. Poc < 10-4 (prob. of bio contamination) **basel. RISK:** **planet. protection** - violation of planetary protection by impact of propulsion module of penetrator in case it is not cleaned accordingly likel.: **max.** / sev.: **>minor** -> **medium risk** .. **very high risk**	**basel. MITIG.:** a) cleaning according COSPAR cat. IVb + bio shield toward CLIPPER ~~b) deflection manoeuvre~~ **remain. RISK:** a) **Cost** - increased cost due to COSPAR IV requirements for propulsion module (1-10 Mill. EUR) likel.: **max.** / sev.: **minor** -> **low risk** ~~b) **Schedule** (program.)/ **Cost** due to additional propulsion/ increased complexity of design (10-40Mil.. EUR)~~ ~~likel.: **max.** / sev.: **signific.** -> **medium risk**~~	basel. RISK: R17.2 -> E5p remain. RISK: R17.2 -> E1c R ->		

Table 19-9b: Risk Log applicable for CLEO penetrator only

Severity	A (min.)	B (low)	C (medium)	D (medi.)	E (max.)
5 (catastr.)			R6.1sc, R6.2sc, R16.2po		
4 (critical)			R14sc, R15a sc*		
3 (major)			R15b sc		R2c
2 (signif.)					R1c, R4.1sc, R4.2sc, R5sc, R16.2c
1 (minor)				R18c	
					Likelihood

* not applicable for CLEO/E

Table 19-10a: Top Risk Index Chart applicable for CLEO orbiter + penetrator

Severity	A (min.)	B (low)	C (medium)	D (medi.)	E (max.)
5 (catastr.)		R17.1po	R9.2sc/t,	R11a-sc,	
4 (critical)				R7sc, R11b-sc/t	
3 (major)					R2c
2 (signif.)			R9.1sc,		R1c, R4.2sc, R10c, R17.1c
1 (minor)				R18c	R9.2c, R12c, R16.1c, R17.2c
					Likelihood

Table 19-10b: Top Risk Index Chart applicable for CLEO penetrator only

19.5.1 Risk Log General Conclusions

- Very high risks and high risks are typical of a phase A project. Areas with lack of definition or little previous experience pose a priori more risk to the mission and therefore are the ones with more risk reduction potential
- Experience shows that all risk items with a critical risk index (red, orange area) must be analyzed and proposals for risk treatment actions elaborated
- In the end, ideally all risk items should reach a level of justifiable acceptance
- The risk management process should be further developed during the project definition phase in order to refine the risk identification/analysis and provide evidence that all the risks have been effectively controlled.

19.6 Risk Log Specific Conclusions and Recommendations

The CLEO is an exploration mission with an inherently higher risk potential. Accordingly the Risk Index was adapted and a wider range of risk is considered acceptable.

However for both, CLEO orbiter (Io and Europa fly-by) and penetrator (Europa), it is recommended to mitigate/ discuss further the following risks intensively:

- Launcher uncertainty (R2) with respect to design-life-time and qualification

- Hibernation strategy for Jovian cruise (R6.1) with respect to wake-up failure
- Limited communication redundancy (R6.2) with respect to the cold redundancy concept
- Aspects of planetary protection whereby the currently undefined planetary protection approach for the CLIPPER (at least as described in the NASA SALMON-2) must be seen as a major risk for CLEO/I and CLEO/E.

For the CLEO penetrator a higher risk potential was identified in comparison to the CLEO orbiter. The following risk has to be mitigated before this option becomes acceptable from risk viewpoint:

- Ice RF transparency (R11) with respect to a robust option to guarantee the uplink of the research data independently from the depth of the penetrator, its position in the ice and the ice contamination.

It is recommended to mitigate/ discuss further the following penetrator specific risks intensively:

- Uncertainties due to unknown ice properties (R7) with respect to test coverage of the worst case conditions of ice on Europa surface
- High impact load on instruments (R9.1) with respect to the robustness of instruments
- Launcher uncertainty (R2) with respect to design-life-time and qualification.

Further more it is recommended to discuss with NASA the possibility of an earlier separation of CLEO which is at the moment foreseen after the Jupiter orbit insertion. This would:

- Reduce the design life time by more than 1a
- Eliminate the risk ‘Hibernation strategy for Jupiter orbit insertion (R6.1)
- give more freedom in the design specially of ‘CLEO orbiter’-options from a mass viewpoint (the reduced mass for propulsion could be used for CLEO platform or payload or fly-by planning).

20 PROGRAMMATICS/AIV

20.1 Requirements and Design Drivers

The main requirements and design drivers for the CLEP project from a programmatics point of view are:

- The CLEP S/C shall be carried as a piggy back on NASA Clipper S/C and released after Jovian Orbit Insertion
- The CLEP S/C shall be compatible with SLS as the baseline launcher for Clipper and with Atlas V and Delta IV as back-up solutions
- Earliest launch date in May 2022
- Nominal 2.7 years transfer duration, but up to 7.2 years for back-up launcher
- The CLEP S/C total mass shall not exceed 250 kg
- The CLEP S/C stowed envelope shall be less than 1m x 1m x 1m
- The CLEP S/C shall conform to Category IV Planetary Protection Requirements
- The schedule needs to be aligned with project management timeline of Clipper
- TRL 6 required by 2018
- CLEP S/C structural model and FM are to be delivered to NASA.

20.2 Assumptions and Trade-Offs

- For system level qualification ESA should deliver a STM for structural and thermal qualification
- The FM will possibly undergo protoflight levels during NASA system level acceptance tests, thus it is considered to be a PFM
- No AVM will be required by NASA, if requested a simulator could be delivered
- System Level tests of the composite of Clipper and CEO will include at least: Random Vibration, Acoustics, Pyro Shock, Thermal Vacuum, Solar Exposure, Electromagnetic Emission and Conduction
- STM and FM will be environmentally tested before delivery to NASA
- Environmental test levels and durations to be applied in Europe and at NASA will need to be specified early in the program
- FM delivery to NASA is expected to be required 18 month before launch
- STM delivery to NASA is expected to be required 12 month before the FM.

20.3 Options

No options were considered for the programmatics assessment.

20.4 Technology Requirements

The Technology Readiness Levels (TRL) present a systematic measure, supporting the assessments of the maturity of a technology of interest and enabling a consistent comparison in terms of development status between different technologies.

The product tree for CLEP, as established in the CDF workbooks, is shown in Table 20-1. It identifies for each subsystem the associated equipment, some times components, their quantity and their TRL as far as available.

Category	Owner	Name	n_items	shape	TRL
Elements	SYE	Aft Penetrator CLEP	5		
Elements	SYE	Clipper Europa Penetrator	1		
Elements	SYE	Fore Penetrator CLEP	1		
Elements	SYE	Penetrator Delivery System CLEP	1		
Equipment	SYE	Epoxy	1		
Equipment	SYE	Harness	1		
Subsystems	AOGNC	Attitude, Orbit, Guidance, Navigation Control Subsystem	9		
Equipment	AOGNC	STR Selex Micro Star Tracker	2	Box	5
Equipment	AOGNC	GYRO on Chip QRS11	4	Cylinder	9
Equipment	AOGNC	Accelerometer SA0120	3	Box	5
Subsystems	COM	Communications Subsystem	11		
Equipment	COM	Low Gain Antenna Deployable CLEP	1	-	-
Equipment	COM	Low Gain Antenna On PDS	1	-	-
Equipment	COM	Low Gain Antenna On Fore Penetrator	1	-	-
Equipment	COM	Modulator	2	-	-
Equipment	COM	Receiver CLEP PDS	1	-	-
Equipment	COM	RF Harness CLEP Penetrator	1	-	-
Equipment	COM	RF Harness CLEP PDS	1	-	-
Equipment	COM	Transmitter CLEP Penetrator	1	-	-
Equipment	COM	Transmitter CLEP PDS	1	-	-
Equipment	COM	Umbilical Cord	1	-	-
Subsystems	CPROP	Chemical Propulsion Subsystem	64		
Equipment	CPROP	20N Thruster CLEP	7	-	9
Equipment	CPROP	Fill and Drain Valve / Vent Valve (Pressurant) CLEP	4	-	9
Equipment	CPROP	Fill and Drain Valve / Vent Valve (Propellant) CLEP	2	-	9
Equipment	CPROP	Latching Valve CLEP	2	-	-
Equipment	CPROP	Miscellaneous CLEP	1	-	9
Equipment	CPROP	Mounting Screws CLEP	20	-	9
Equipment	CPROP	Piping (including fittings) CLEP	1	-	9
Equipment	CPROP	Pressurant CLEP	1	-	9
Equipment	CPROP	Pressure Transducer CLEP	3	-	9
Equipment	CPROP	Propellant Filter CLEP	1	-	9
Equipment	CPROP	Propellant Tank PEPT 230 with Diaphragm	1	Sphere	9
Equipment	CPROP	Safe and Arm Device Model 2134B	1	-	9

Category	Owner	Name	n_items	shape	TRL
Equipment	CPROP	Stand-off CLEP	20	-	9
Subsystems	DH	Data-Handling Subsystem	2		
Components	DH	PDS OBC	1	-	3
Components	DH	Penetrator CDMU	1	-	3
Subsystems	INS	Instruments Subsystem	1		
Equipment	INS	E_PAC CLEP	1	-	4
Subsystems	MEC	Mechanisms Subsystem	7		
Equipment	MEC	Antenna Deployment Mechanism	1	Box	-
Equipment	MEC	Clipper-PDS Separation Mechanism	4	-	-
Equipment	MEC	Fore-Aft Penetrator Separation Mechanism	1	-	-
Equipment	MEC	Penetrator-PDS Separation Mechanism	1	-	-
Subsystems	PWR	Power Subsystem	3		
Equipment	PWR	Battery_Primary	1	Box	8
Equipment	PWR	Battery_Primary 2	1	Box	8
Equipment	PWR	Power Conditioning & Distribution Unit 2	1	Box	6
Subsystems	RAD	Radiation Subsystem	1		
Subsystems	STR	Structures Subsystem	3		
Subsystems	TC	Thermal Control Subsystem	4		
Equipment	TC	MLI PDS	1		
Components	TC	Heater	1		
Components	TC	Heater PDS	1		
Components	TC	Thermal Equipment Penetrator	1		

Table 20-1: CLEO product tree

Note:

Most of the hardware on CLEO/P will need to be exposed to sterilisation processes (e.g., ECSS-Q-ST-70-57C, ECSS-Q-ST-70-56C). This could lower the TRL level of the respective hardware and might require dedicated developments.

The TRL definitions from RD[27] are shown in Table 20-2:

TRL	ISO Definition	Associated Model
1	Basic principles observed and reported	Not applicable
2	Technology concept and/or application formulated	Not applicable
3	Analytical and experimental critical function and/or characteristic proof-of concept	Mathematical models, supported e.g. by sample tests
4	Component and/or breadboard validation in laboratory environment	Breadboard

5	Component and/or breadboard critical function verification in a relevant environment	Scaled EM for the critical functions
6	Model demonstrating the critical functions of the element in a relevant environment	Full scale EM, representative for critical functions
7	Model demonstrating the element performance for the operational environment	QM
8	Actual system completed and "flight qualified" through test and demonstration	FM acceptance tested, integrated in the final system
9	Actual system completed and accepted for flight ("flight qualified")	FM, flight proven

Table 20-2: TRL scale

For the instruments and related equipment the presently achieved TRL levels are identified in Table 20-3. TRL as low as 2, 3 and 4 are identified.

Penetrator Instrument	TRL
Drill / sample collection	2
Sample container	2
Common electronics	2
Instrument 1: Mass spectrometer	4
Instrument 2: Sample imager	2
Instrument 3: Habitability package	3

Note: Mass Spectrometer may only be at TRL 2/3 for high g (except for certain components)

Table 20-3: CLEP instrument TRL

Table 20-4 shows an indication of the development time depending on the current TRL. According to the European Space Technology Master Plan, to prepare the contractual basis for multi-annual programs it takes about 18 months to reach political agreement on financial ceiling. This has also been included in the table.

TRL	Duration
5-6	4 years + 1.5 year
4-5	6 years + 1.5 year
3-4	8 years + 1.5 year
2-3	10 years + 1.5 year
1-2	12 years + 1.5 year

Table 20-4: TRL – development duration

Assuming, that the development of technology at TRL lower than 6 is already approved and on-going, we can expect that we need another 2 years before the implementation phase can start for technologies at TRL 4 and another 4 years for technologies at TRL 3 unless very special effort is made to speed up the development.

20.5 Model Philosophy

The CLEP S/C is also called Penetrator Descent Module (PDM) and consists of the Penetrator Delivery System (PDS) and the Penetrator itself.

The model philosophy proposed at PDM level is similar to the model philosophy of the ESA Huygens project:

- Structural Thermal Model (STM)
- Protoflight Model (PFM)
- Electrical Functional Model (EFM also known as ATB or AVM).

At Penetrator level the proposed model philosophy is:

- Flight Model (FM)
- Qualification Model (QM)
- Electrical Functional Model (EFM also known as ATB or AVM)
- At least 2 Development Models (DM).

Penetrator FM and EFM will be integrated into the respective models at PDM level.

Only a penetrator mass dummy will be included in the PDM STM.

The amount of tests with the penetrator QM is still TBD. The development models are expected to be representative for specific aspects, e.g. structure, instruments, mechanisms, and the use of several such models will allow parallel advancement of related designs in an efficient manner.

Note:

NASA identified for their spacecraft, the Europa Clipper spacecraft the instrument hardware delivery schedule identified in table Table 20-5.

Deliverable Item	Due Date
Engineering Model and GSE	I-CDR + 4 months
Flight Model and GSE	SIR + 3 months
Flight Spare	SIR + 3 months

Table 20-5: Europa Clipper instrument hardware delivery schedule

The Penetrator Descent Module can be considered as “super instrument” at the level of the Clipper S/C.

20.6 Development Approach

The typical scientific development approach shows following steps:

- Phase A
- Phase B1
- Intermediate Phase
- Phase B2/C/D (implementation Phase)
- Agency contingency

Because from the CLEO study it is known that such a conservative approach is not compatible with the target launch date and, because the PDM is a rather simple satellite, a more success oriented or "Proba-approach" , which is an approach tailored to in-orbit demonstration is proposed. Its characteristics are:

- Reduced Phase A and B1
- Short intermediate phase (quick approval for opportunity mission)
- Implementation phase well below 4 years
- Increased risk (experimental mission)

The difference is in a higher integration of the manufacturing with the prime contractor, i.e. less sub contractors, geographical distribution only to a few participating states, streamlined documentation possible due to the reduced number of contractual interfaces. According less time is allocated to project phases, reviews and the interruptions for approval of the next contract phase.

20.6.1 Test Matrix

Table 20-6 shows the test matrix with tests on PDS level (CLEP S/C) and the joint tests with Clipper denoted as "Composite" in the table.

Test Description	CLEP STM	CLEP EFM	CLEP PFM	Composite QM	Composite FM
Mech. Interface	R, T		R, T		
Mass Property	A, T		A, T		
Electr. Performance		T	T		
Functional Test		T	T		
Propulsion Test		T	T		
Thruster Lifetime Test					
Deployment Test	A, T		A, T		
Telecom. Link		T	A, T		
Alignment	A, T		T		
Strength / Load	A, T		T		
Shock / Seperation	T		T (tbd)	T	T (tbd)
Sine Vibration	A, T		T		
Random Vibration	T		T	T	T
Modal Survey	A				
Acoustic	T		T	T	T
Outgassing			I (T)		
Thermal Balance	T (tbc)		A, T	T (tbc)	
Thermal Vacuum			T	T (with sun)	T (with sun)

Micro Vibration					
Grounding / Bonding			R, T		
Radiation Testing			A		
EMC Conductive Interf.			T	T (tbc)	T
EMC Radiative Interf.			T	T (tbc)	T
DC Magnetic Testing					
RF Testing			T		

Abbreviations: I: Inspection, A: Analysis, R: Review, T: Test

Table 20-6: CLEP system level test matrix

20.7 Schedule

The schedule for the proposed success oriented approach has following characteristics:

- Phase A is reduced from typically 12 month to 10 month
- Phase B1 is reduced from typically 12 month to 8 month
- No intermediate phase is included after PRR for Phase B 1 ITT, proposal evaluation and negotiation. Phase A and B1 are assumed to be covered by one contract
- The intermediate phase after SRR for mission adoption, ITT and Phase B2/C/D proposal evaluation and negotiation of typically at least 6 month is reduced to 4 month
- Phase B2 is reduced from 12 month to 8 month
- Phase C & D is reduced from 36 month to 34 month (typical are 30 to 48 month)
- No ESA contingency is included
- The above phase durations do include PRR, SRR, PDR, CDR and QR and the review durations are based on the average review durations.

ID	Task Mode	Task Name	Duration	Start	Finish
1		**Nasa Milestones**	**1537 days**	**Thu 30/06/16**	**Sun 22/05/22**
2		Clipper SRR/MDR	0 days	Thu 30/06/16	Thu 30/06/16
3		Clipper Project PDR	0 days	Thu 31/05/18	Thu 31/05/18
4		Clipper Project CDR	0 days	Fri 31/05/19	Fri 31/05/19
5		Clipper System Integration Readiness	0 days	Fri 31/07/20	Fri 31/07/20
6		Clipper Environmental Test Readiness (estimate)	0 days	Mon 01/03/21	Mon 01/03/21
7		Clipper Pre-Ship Review (PSR)	0 days	Tue 30/11/21	Tue 30/11/21
8		Clipper Launch	0 days	Sun 22/05/22	Sun 22/05/22
9		**CLEO Schedule**	**1394 days**	**Tue 01/09/15**	**Fri 01/01/21**
10		Kick-off	0 days	Tue 01/09/15	Tue 01/09/15
11		PRR	40 days	Fri 06/05/16	Thu 30/06/16
12		SRR	40 days	Wed 04/01/17	Tue 28/02/17
13		PDR	50 days	Thu 21/12/17	Wed 28/02/18
14		CDR	60 days	Thu 28/02/19	Wed 22/05/19
15		QR	43 days	Thu 01/08/19	Mon 30/09/19
16		STM delivery to NASA	0 days	Tue 01/10/19	Tue 01/10/19
17		AR	23 days	Tue 01/12/20	Thu 31/12/20
18		PFM delivery to NASA	0 days	Fri 01/01/21	Fri 01/01/21
19		Phase A (10 month)	218 days	Tue 01/09/15	Thu 30/06/16
20		Phase B1 (8 month)	173 days	Fri 01/07/16	Tue 28/02/17
21		Intermediate phase (4 month)	88 days	Wed 01/03/17	Fri 30/06/17
22		Phase B2 (8 month)	173 days	Mon 03/07/17	Wed 28/02/18
23		Phase C/D (34 month)	741 days	Thu 01/03/18	Thu 31/12/20
24		STM procurement	218 days	Wed 01/08/18	Fri 31/05/19
25		STM environmental test campaign	43 days	Mon 03/06/19	Wed 31/07/19
26		AVM procurement	129 days	Mon 03/09/18	Thu 28/02/19
27		AVM integration	175 days	Fri 01/03/19	Thu 31/10/19
28		AVM testing	305 days	Fri 01/11/19	Thu 31/12/20
29		PFM procurement	220 days	Mon 01/04/19	Fri 31/01/20
30		PFM integration	173 days	Mon 03/02/20	Wed 30/09/20
31		PFM environmental test campain	43 days	Thu 01/10/20	Mon 30/11/20

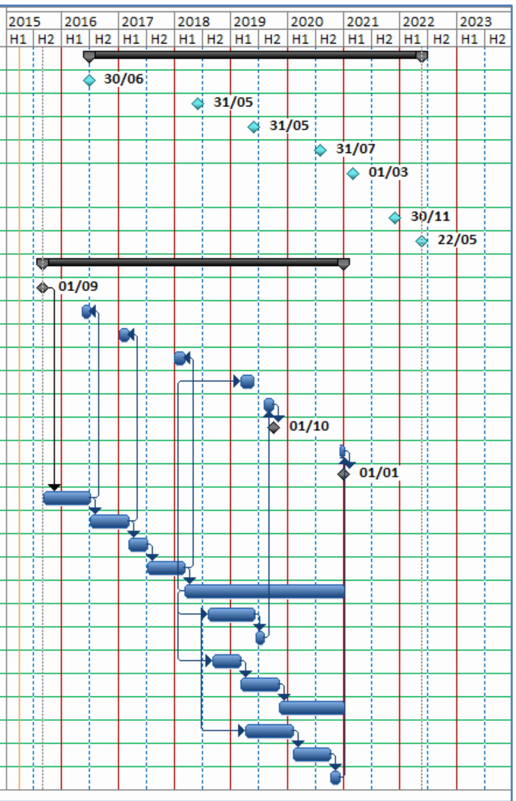

Figure 20-1: CLEP schedule – "Proba-approach"

20.8 Summary and Recommendation

- A conventional model philosophy is proposed for CLEP with at system level STM, AVM and PFM
- Environmental test campaigns at CLEP level (in Europe) are proposed to be performed before delivery to NASA for the composite level environmental test campaigns
- Accurate agreements deliveries and on test levels and durations for all test campaigns need to be established early in the program
- From the proposed first launch date for clipper (May 22) we derive the need for STM delivery by November 2019 (this might be too late for NASA) and for PFM delivery by November 2020
- A conservative schedule will lead to STM delivery end 2020 and to PFM delivery end February 2022. Only a success-oriented "Proba-approach" could lead to a STM delivery at the estimated need date and a PFM delivery beginning 2021
- However this approach requires the start of the implementation phase by July 2017 at the latest and it is very unlikely, that technologies with a TRL significantly lower than 5, can achieve TRL 6 by then
- Consequently for technology at TRL below 5 a specific development plan up to demonstrating TRL 6 should be elaborated and at the same time back-up solutions should be identified.

21 COST

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22 CONCLUSIONS

22.1 Satisfaction of Requirements

A preliminary design of the CLEP penetrator has been done building on past CDF studies such as REIS, CRETE and JURA, as well as JUICE developments and miniaturised and integrated technologies.

The CLEO/P design was based on the Airbus design with a modified retro-burn engine, trajectory, and a textile antenna design that remains on the surface rather than penetrates it. The trajectory design is such that two communication windows exist during the 10 days. A fore-body plus aft-body type of design was selected for the penetrator.

22.2 Compliance Matrix

Preliminary design of the CLEP Penetrator building on Airbus industrial design performed in the context of JUICE and updated in the context of Clipper	Completed. Compact spacecraft based on a solid STAR24 PDS and a penetrator consisting of a forebody and aftbody. The wet mass of over 300 kg however exceeds the mass target of 250 kg.
Optimise the mission profile including the braking strategy performed by the penetrator carrier	Uncompleted. The CDF study has highlighted the high dispersions that could be the result of the SRM burn. This is an aspect for which possible solutions have been identified, but the issue is not resolved and needs further study.
Identify the key design drivers and the operational challenges of the mission	Completed. Key drivers are strong mass constraint, the SRM burn dispersion, large distance to Earth (6 AU), the communication link from the penetrator to the orbiter, and radiation environment
Identify mass reduction options to meet the stringent 250 kg mass allocation for CLEO/P.	Solutions have been found by means of a textile low-weight antenna, and the use of an SRM. Nevertheless the mass constraint of 250 kg is not met, with a large difference.
Propose and define a Science case and payload suite for both concepts	Completed. See payload chapter.
Identify technological needs, and associated Programmatics, Risk and Cost aspects of CLEO/P, incl. geographical return impacts, and provide a preliminary risk register	Completed. See cost/risk/programmatics chapters
Iterate on the operational and interface requirements with NASA's Clipper mission	Completed. Telecon with NASA held during the study, with questionnaire by CDF team answered.

22.3 Further Study Areas

- Ranging/Doppler versus delta-DOR is to be further assessed (possibly in dedicated study)
- Planetary protection implementation is to be consolidated for Io case, in cooperation with Clipper project
- Optimisation of shielding of specific components and mass should be done
- The overall communications strategy should be further investigated through extensive testing.
- The textile antenna should be further designed
- The navigation for the SRM burn is to be investigated and optimised
- Mature the design of separation mechanisms and triggering strategies
- Improve modelling of penetration and depth calculations
- Optimise umbilical folding strategy
- Assess and minimise the impact on the CLIPPER tour.

22.4 Final Considerations

The CDF study for the penetrator had a reduced set of sessions (four) including final presentation, and therefore a proof of concept was not found. Nevertheless a concept was identified where the antenna remains at the surface of the moon, rather than penetrate inside the ice.

If the Europa Penetrator option is taken further, then discussions with NASA are required regarding the optimum Jovian cruise of CLIPPER prior to the Penetrator release.

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24 ACRONYMS

Acronym	Definition
ΔV	Delta-V
ACC	Accelerometer
ADS	Airbus Defence and Space
AIT/V	Assembly, Integration and Test/Verification
AIV	Assembly, Integration and Verification
AIVT	Assembly, Integration, Verification and Test
Al	Aluminium
AOGNC	Attitude and Orbit Guidance Navigation and Control
AOS	Acquisition Of Signal
AVM	Avionics Verification Model
BCR	Battery Charge Regulator
BDR	Battery Discharge Regulator
BER	Bit Error Rate
CaC	Cost at Completion
CCD	Charge Coupled Device
CAN	Controller Area Network
CDMU	Command and Data Management Unit
CER	Cost Estimation Relationship
CLEO	Clipper Europa Orbiter
CLEP	Clipper Europa Penetrator
CMA	Cost Model Accuracy
COT	Crank Over the Top
CTE	Charge Transfer Efficiency
DHS	Data handling Subsystem
DMM	Design Maturity Margin
DOA	Degree of Adequacy of the cost model
DoD	Depth of Discharge
DSC	Descent Mode
DSN	Deep Space Network

Acronym	Definition
DSPN	De-spin Mode
ECSS	European Cooperation for Space Standardisation
EFM	Electrical Functional Model
EIRP	Equivalent Isotropic Radiated Power
EM	Engineering Model
EMC	Electromagnetic Compatibility
E-PAC	Europa-Penetrator Astrobiology Complement
EPE	External Project Events
EQM	Engineering Qualification Model
ESA	European Space Agency
ESTRACK	ESA Tracking Network
FCT	Flight Control Team
FDIR	Failure Detection Isolation and Recovery
FER	Frame Error Rate
FM	Flight Model
FoV	Field of View
G/S	Ground Station
GAM	Gravity Assist Manoeuvre
GNC	Guidance Navigation and Control
GSE	Ground Support Equipment
GYR	Gyroscope
HDRM	Hold Down and Release Mechanism
HK	Housekeeping data
IFP	Internal Final Presentation
IQM	Inherent Quality of the cost Model
JC	Jovian Cruise
JOI	Jupiter Orbit Insertion
kGy	Kilo Gray
LED	Light Emitting Diode
LEOP	Launch and Early Operations Phase
LGA	Low Gain Antenna

Acronym	Definition
Li-CFx	Lithium-carbon monoflouride (type of primary battery cell)
Li-$SOCl_2$	Lithium thionyl chloride (type of primary battery cell)
LoS	Line of Sight
LOS	Loss of Signal
MAIT	Manufacturing Assembling Integrating Testing
MDR	Mission Definition Review
MEMS	Micro Electro-Mechanical Systems
MOC	Mission Operations Centre
NFC	Near Field Communication
OBC	On Board Computer
OBDH	On-Board Data Handling
PCB	Printed Circuit Board
PDM	Penetrator Descent Module
PDS	Penetrator Delivery System
PFM	Protoflight Model
PI	Principal Investigator
PLM	Payload Module
POE	Project Owned Events
PSD	Penetrator Separation Device
QIV	Quality of the Input Values
QM	Qualification Model
RAM	Random Access Memory
RF	Radio Frequency
RX	Receiver / Reception
S/C	Spacecraft
S/S	Subsystem
SAR	Search And Rescue
SFT	System Functional Test
SOM	Spacecraft Operations Manager
SPN	Spin Mode
SRM	Solid Rocket Motor

Acronym	**Definition**
STM	Structural Thermal Model
STR	Star Tracker
SVM	Service Module
SVT	System Validation Test
TBC	To be confirmed
TBD	To be defined
TC	Telecommand
TGT	Targeting Mode
TM	Telemetry
TRL	Technology Readiness Level
TT&C	Tracking, Telemetry and Command
TX	Transmitter / Transmission