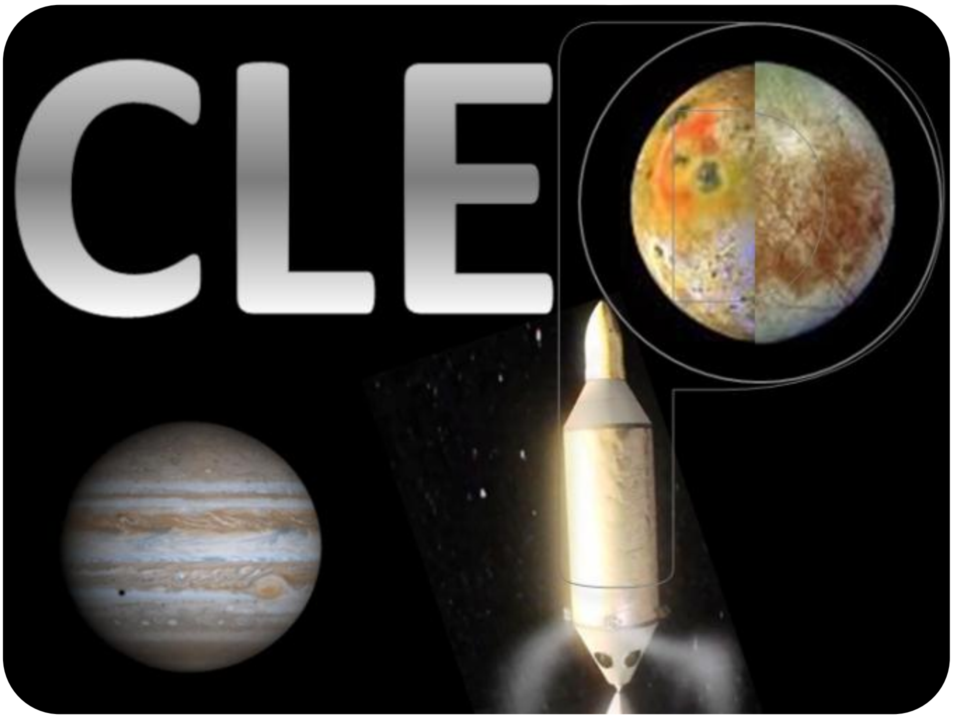
CDF STUDY REPORT

CLEO/P

Assessment of a Jovian Moon Flyby Mission

as Part of NASA Clipper Mission

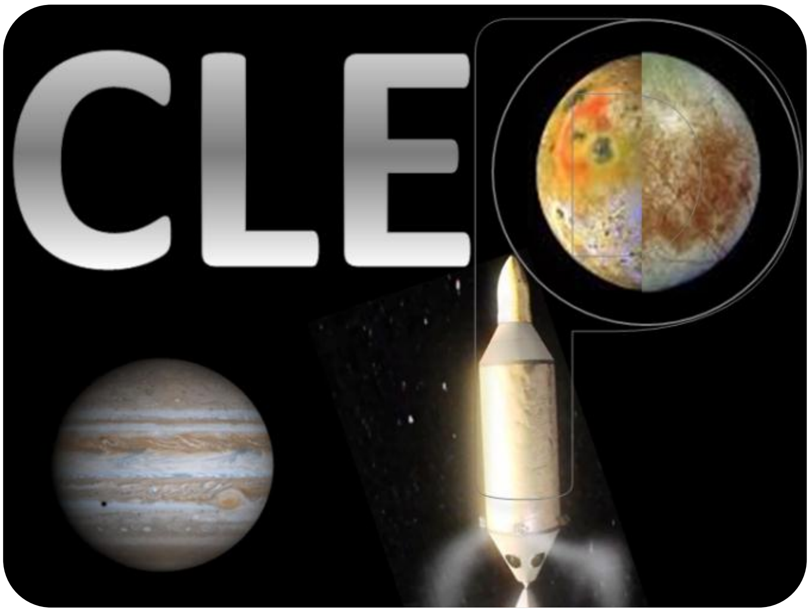


CDF-154(D)Public
April 2015

CDF Study Report

CLEO/P

Assessment of a Jovian Moon Flyby Mission as Part of NASA Clipper Mission



FRONT COVER

Study logo showing Orbiter and Jupiter with Moon

STUDY TEAM

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

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

Further information and/or additional copies of the report can be requested from:

T. Voirin
ESA/ESTEC/SRE-FMP
Postbus 299
2200 AG Noordwijk
The Netherlands
Tel: +31-(0)71-5653419
Fax: +31-(0)71-5654295
Thomas.Voirin@esa.int

For further information on the Concurrent Design Facility please contact:

M. Bandecchi
ESA/ESTEC/TEC-SYE
Postbus 299
2200 AG Noordwijk
The Netherlands
Tel: +31-(0)71-5653701
Fax: +31-(0)71-5656024
Massimo.Bandecchi@esa.int

TABLE OF CONTENTS

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**SA **O**rbiter or **P**enetrator (separate reports are produced covering 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 investigation (including a life detection experiment) building on the Airbus industrial design 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.

This Page Intentionally Blank

2 EXECUTIVE SUMMARY

2.1 Study Flow

The minisat concept study was performed in the Concurrent Design Facility (CDF) in eight sessions, starting with a kick-off on the 10th February 2015 and finishing with an internal final presentation on the 10th March 2015. The sessions were supplemented with several splinter meetings to complete the design iteration in the very short time frame allocated.

The assignment for the minisat concept was to formulate a small satellite concept (250 kg) that can become a junior partner of the NASA Clipper mission. The main premise was to have the minisat attached to Clipper during launch and interplanetary transfer and released by Clipper once it arrived at the Jovian system.

During the first sessions of the study the baseline mission design for the minisat concept converged towards the Jupiter moon Io. The original idea was to target Europa but, for this moon, the science case was less important than for Io. Io has not yet been explored by close remote exploration nor atmospheric in-situ measurements. The main science case related with Europa (that is not covered by Clipper itself) was possibility of exploring the Europa plumes, but this is physical phenomena is still to be confirmed. The baseline minisat concept design to Io is referred to as CLEO-I.

2.2 Requirements and Design Drivers

The mission and systems requirements and design drivers for the CLEO-I study are provided in the systems chapter.

The main overall drivers for the design were the mass allocation of 250 kg, the minimum mission lifetime of at least two close flybys with Io, and the science data downlink.

The mass, lifetime and data volume considerations led the study to focus on:

- Trading different mission options with different ΔVs, inclinations and transfer durations (see 5.4),
- Looking at lean spacecraft configurations where the redundancy is minimised to reduce mass and power (see 16.3),
- Trading different concepts to maximise the science download by changing either direct transmission to Earth or relay to Clipper (see 16.4),
- Looking at different operational strategies where the spacecraft would switch between different system modes in order to maximise the data downlink (see 6.3.2),
- Trading different shielding configurations: spot shielding, single vault or series of mini vaults, etc. (see 6.2.2.1) , and,
- Evaluating highly integrated configurations for the spacecraft avionics (see 6.2.2.2).

2.3 Mission

The baseline mission design is to separate from Clipper after Jupiter Orbit Insertion, after it performed the Perigee Reduction Manoeuvre. This mission profile has an orbit inclination close to 0 deg. This has the advantage of allowing a ΔV configuration (low propellant and propulsion system dry mass) but the disadvantage of imposing on the spacecraft higher radiation doses than for orbits with higher inclination.

The baseline design includes two flybys, the first one with a pericentre at 500 km, and a second one with a pericentre at 100 km. The spacecraft was designed to survive the radiation dose for these two flybys with a design radiation margin of 2. This approach might allow additional flybys at the end of life.

The study resulted in a CLEO-I spacecraft design with the following characteristics:

CLIPPER Esa Orbiter Io	
Launch Date	May/June 2022
Launcher	Nominal: SLS direct to Jupiter (June 2022)
	Backup: SLS direct to Jupiter (June 2023)
	Alternate: Atlas V 551 EVEEGA (May 2022)
	Alternate backup: Atlas V 551 VEEGA (June 2023)
Transfer time	2.7 years (Nominal), 7.2 years (Alternate)
Release from Clipper	After JOI, after PRM
From JOI to IGA1	1.5 year
Nr Flybys	2
Flyby 1 parameters	Period: 100 days (from Flyby 1 to Flyby 2)
	Near-equatorial (0.8 deg to Jupiter equator)
	Vinf 7 km/s
	Perijove: 5.9 Rj (~= Io orbital radius)
	Apojove: 160 Rj
	IGA C/A: 500 km Northern Hemisphere
Flyby 2 Parameters	Period: 190 days (from Flyby 2 to next flyby or impact)
	Near-equatorial (0.2 deg to Jupiter equator)
	Vinf 7 km/s
	Perijove: 5.8 Rj (~= Io orbital radius)
	Apojove: 260 Rj
	IGA C/A: 100 km Southern Hemisphere
Δ V	345.55 m/s (including margins)
Payload	Camera, Mag, MidIR, INMS; 14.82 kg, 51.6 W pPwr
Science Duration	300 mins per flyby (*note : Flux gate magnetometer is ON all along the orbit, in low resolution*)
Data Volume	7.22 Gb (2.14 Gb + 4.81 Gb) (to be shared between SCI & HK TM)
Mass	Dry mass (227.32 kg) (incl

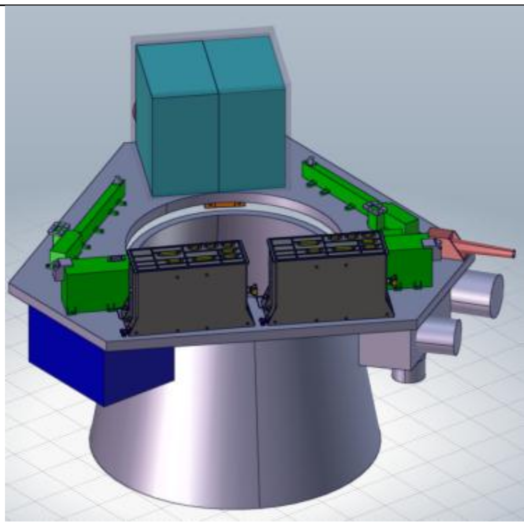
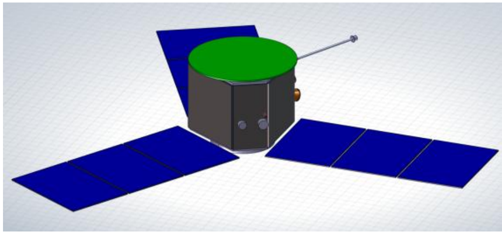
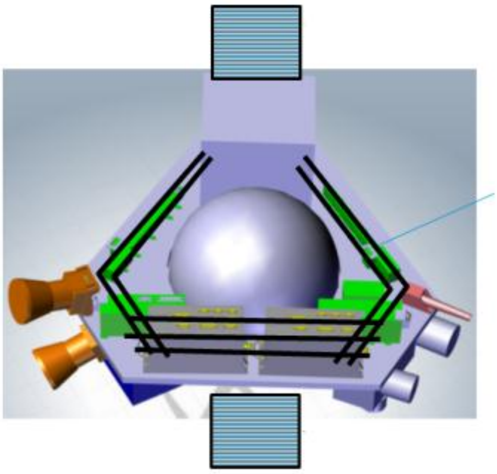
	DMM)	
	Propellant (39.93 kg) (incl 2% margin)	
	Wet (266.75 kg) (incl 20% system margin)	
Dimensions	Stowed: 1.2x1.2x0.8 triangular shape	
Structure	CFRP	
Shielding	Shielding Mass: 19.06 kg (5 mm Al Vaults + 10 mm MINIAvio + 3.5 kg Instruments)	
Mechanisms	Separation: Clamp band; SA hinges	
AOGNC	Cruise: Spin; Science: 3-axis stab with RCT. 2 GYROS, 2 STR, 8 SS;	
Propulsion	Monoprop System; 1 tank, 1x22N thruster(6Nom+6R)x1N RCT	
Power	6 m2 SA; MPPT; 4.9 kg Battery (690 Wh); Unregulated Bus	
Communication	X-Band HGA 1.1 m (tx) – 0.6 m (Rx); 2 LGA, RF pwr 65W; TM rate 3.5kbps; TC rate 1kbps (35 m GS)	
DHS	MINIAvio (OBC + PCDU + STR processing+ Gyros + Instrument processing)	
Thermal	Ext. MLI, Int. MLI, Instruments MLI, prop. MLI; 0.15 m2 2xLouvers; 6m heat-pipes; heaters; sensors. Propulsion heating power 25 W; platform Heating Power JC 5 W	

Table 2-1: CLEO-I baseline design

The baseline configuration slightly exceeds the 250 kg mass allocation (266.75 kg). In addition to the baseline configuration, two more options were evaluated at system level (see 6.7):

- A hyperbolic flyby option with much reduced ΔV requirements (CLEO-I hyper), and,
- An option with Europa flybys instead of Io (CLEO-E).

In the hyperbolic flyby option, the spacecraft would not be inserted in Jovian orbit and would remain in a heliocentric hyperbolic trajectory (only one Io flyby is then possible). It would separate from Clipper before JOI and only perform targeting manoeuvres estimated at around 40 m/s, which would lead to a much lower propulsion system (lower propellant mass and dry mass) and lower shielding mass due to the lower mission duration.

In the Europa option (CLEO-E), the spacecraft would target Europa instead of Io.

Both these options would allow meeting the 250 kg mass allocation, but with strong impact on the science return.

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 (currently in phase B, launch in 2022). While JUNO will focus on Jupiter system, CLIPPER will be dedicated to EUROPA and JUICE will mostly focus on GANYMEDE. As a potential piggy-back contribution to CLIPPER, a flyby mission dedicated to IO would offer a perfect complement to the other Jovian missions.

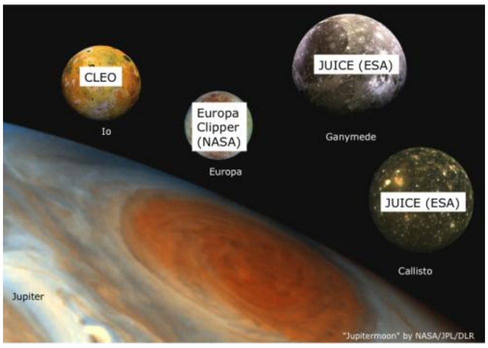


Figure 3-1: complementarity of CLEO-I, Juice and Clipper Jovian missions

3.2 Study Objectives

The main objectives of the study are the following:

- The preliminary design of the CLEO/I minisat building on past CDF studies (REIS, CRETE, JURA RD[1], RD[2] and RD[3]), capitalising on JUICE developments and miniaturised and integrated technologies (in particular avionics).
- To assess the applicability of the CLEO/I design concept to the CLEO/E mission, addressing the design deltas wrt Io flybys concept.
- To refine the science case and payload suite
- To identify the technology needs, risks and Programatics & cost aspects of CLEO 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 Io Flyby Mission (CLEO/I)

The Science objectives are:

Primary Science Objectives:

- Investigate Io's active volcanism and the nature and magnitude of heat loss
- Investigate the chemistry of Io
- Explore Io as a key element of the Jupiter system

Secondary Science objectives:

- Io interior structure
- Io mountains and tectonics
- Io atmosphere and ionosphere
- Io internal magnetic field
- Tidal heating

3.3.2 Europa Flyby Mission (CLEO/E)

To complement Europa Clipper and JUICE (Europa) science goals, CLEO/E would study the Europa atmosphere/exosphere and the dust environment with special emphasis on the plumes.

The key Science objectives to be addressed by CLEO/E are:

- Are there plumes on Europa?
- What are the constituents of Europa atmosphere/exosphere ?
- Are there dust particles present in Europa atmosphere/exosphere ?

4 PAYLOAD

This chapter describes the scientific instruments forming the model payload complement. The model payload is represented by instruments whose design is based on a previously flown model or shall be at least at an advanced level of development. The model payload serves to estimate reliably the resource requirements towards the spacecraft design and mission operations.

4.1 Requirements and Design Drivers

The instrument performance shall be capable to fulfil the science goals as formulated by the science advisory team. In Table 4-1 the science objectives are listed. The achievement of these goals drives the instrument design and the flyby geometry of the spacecraft.

Investigate Io's active volcanism and the nature and magnitude of heat loss	TIR imaging of volcanic thermal emission at better than 100 km/pixel spatial scale, absolute accuracy 2K, at silicate melt temperatures, over a range of temporal scales (e.g. hourly, daily, weekly, monthly). Desire better than 20 km/pixel spatial resolution	TIR
	Determine regional (and global?) heat flow by measuring surface thermal emission over active region at spatial resolution of 5 km/pixel to 10% radiometric accuracy for at least two wavelengths;	TIR
	Frequent multi-colour global mapping (minimum 3 colours) at better than or equal to 10 km/pix. Violet, green, NIR over a range of temporal scales (e.g. hourly, daily, weekly, monthly).	MAC
	High-resolution visible imaging (about 100 m spatial resolution) of selected volcanic features for change detection (e.g. with Galileo and Voyager data).	MAC
	VIS plume imaging: high phase angle plume monitoring (for dust and gas emissions) and low phase angle observations (for gas absorptions) over a range of temporal scales. Visible spatial resolution better than 20 km/pixel	MAC
	Global (>80%) monochromatic imaging at ~1 km/pixel spatial resolution at available opportunities.	MAC
Investigate the chemistry of Io	Explore roles and rates of sublimation, sputtering, and radiation darkening with thermal IR mapping with regional spatial resolution better than 10 km, including polar coverage.	TIR
	Global thermal observations at least two well-separated wavelengths with a spatial resolution of 100s of km/pixel over periods of days to weeks.	TIR
	Investigate column densities of atmospheric/plume species across the globe and document correlations with plumes, geologic features and local albedo variations from visible imaging in eclipse.	MAC
	In situ neutral mass spectroscopy measurements of Io's atmosphere. Mass resolution TBD	INMS
	Measure the chemical constituents of the atmosphere as an indicator of surface and subsurface composition. Measurements over a mass range better than 300 Daltons and mass resolution better than 500 (high sensitivity and sufficient mass resolution to determine stable isotope ratios are highly desirable).	INMS
Explore Io as a key element of the Jupiter system	Measure three-axis magnetic field components at 1 Hz near-continuously to characterise the properties of the inner magnetosphere and at 32 Hz within 20 Io radii. A sensitivity of 0.1 nT is expected.	FGM
	Measure three-axis magnetic field components at 1 Hz during C/A of Europa by US Clipper to provide simultaneous measurements	FGM

	Investigate plumes composition to better understand the composition of the Io tori, and the exogenic contribution of Io to Europa (Measure the volatile content of potential outgassing sources). Perform measurements over a mass range better than 300 Daltons and mass resolution better than 500 with sensitivity that allows the measurement of partial pressures as low as 10-17 mbar.	INMS
	Conduct a comprehensive search for embedded moons within the ring system via imaging, down to a limiting size of ~100 meters (~14th magnitude).	MAC
	Explore the rings' three-dimensional structure, including the vertical structure of the halo and gossamer rings, via imaging from a variety of viewing geometries. Requires complete mosaics of the system from Jupiter out to beyond the orbit of Thebe, with resolution of finer than 100 km/pixel globally and finer than 10 km/pixel on the main ring.	MAC

Table 4-1: The science goals of an Io flyby mission. The last column indicates the respective instrument to achieve the corresponding goal. TIR; thermal infrared imager, MAC; medium angle camera, INMS; ion and neutral mass spectrometer, FGM; magnetometer

The data volume accumulated of each flyby shall be transmitted. Possibly not all data can be transmitted immediately. A remaining volume can be stored for later transmission.

4.2 Assumptions

The goal was to identify scientific instruments of preferably European origin and technological heritage from a previous space mission. The starting point of the study with a limited spacecraft size indicates that also during the model payload selection preference should be given to low –resource instrumentation.

The mission operates in a harsh radiation environment yet the exact dose rates are subject to detailed study and very much dependent on the operational profile of the spacecraft. For the scientific instruments it is assumed that they withstand a total dose of 50 krad by design. For values in excess, additional shielding has to be provided. Specific caution must be paid to the instruments front ends, i.e. imaging sensors and ion optic. The required shielding is not only to protect the hardware against malfunctioning but also to reduce the background noise created by the massive abundance of charged particles.

Based on previous experience, also the data link budget was estimated as rather limited. Therefore a sensible approach for the collection of scientific data has been followed throughout the study.

4.3 List of Instruments

	Medium Angle Imager - MAC	**Thermal Imager**	**Neutral and Ion Mass Spectrometer - NIMS**	**Magnetometer – FGM**
	S/C interface			
accomm.	s/c panel	s/c panel	s/c panel	boom
electrical	28 V reg	28V reg	28V reg	28V reg
data	Spacewire	Spacewire	Spacewire	Tbd

	Medium Angle Imager - MAC	**Thermal Imager**	**Neutral and Ion Mass Spectrometer - NIMS**	**Magnetometer – FGM**
thermal	--	Uncooled sensor--	--	--
	Pointing			
direction	nadir	nadir	Ram and circumference	na
Field of view [°]	5.5	9	10 and 360	omnidirectional
Unobstructed field of view [°]	180	180	10 x 360	na
	Physical			
No. of unit	1	1	1	1 (boom + 2sensors)
Volume (hxwxl) [mm]	40x50x150	110x200x230	18Øx32	800 (boom)
Mass [kg]	2.75	5.75	3.0	0.85
*Backend electronicPCB only	0.25	0.5	0.5	0.25
**Frontend shielding	1.0	1.0	1.0	0.5
ΣMass [kg]	4.0	7.25	4.5	1.6
	Power [W]			
Operations	9	16.3	19.6	2.3
***Stand-by	-	-	-	-
	Temperature [C°]			
Min/max ops	-20 to 50	5 to 15	-20 to 50	-20 to 50
Min/max non ops	-30 to 60	-40 to 40	-30 to 60	-30 to 60
TRL	4	4	4	4

Table 4-2: Basic characteristics of the scientific instruments (no margins included)

* The mass of backend electronics has been added to the electronic vault (see 4.4.1)

** Shielding mass of the front end sensor

*** No stand-by power assumed. During flyby the instruments are switched on or off as needed.

4.3.1 Medium Angle Camera

The camera design is based on the AMIE camera flown on ESAs SMART-1 mission to the Moon RD[4]. This camera requires only relatively small resources due to its extremely optimised design. It satisfies the scientific requirements of the current mission study. The camera has a refractive optics with a focal length of 155 mm and a field of view of 5.3 degree. The optics are centred on a CCD with 1024x1024 pixel array and 14 µm pixel size (manufacturer THOMSON). The aperture is 15.5 mm. A complex filter pattern of different wavelength range is applied directly onto the CCD. For CLEO three generic filters are assumed. The filter concept, filter wheel vs. fixed mounting, has been not further specified.

In the CLEO case, the camera design would rather include an APS/CMOS sensor than a CCD. APS/CMOS sensors are better suited for radiation intensive environments due to the single pixel read out technology and extremely fast read out duration.

The flyby velocity in the mission baseline is 7.41 km/s at closest approach. This constrains the acquisition time per image. The maximum acceptable smearing is half a pixel ie 9m at the surface. This corresponds to an integration time of 1.2 ms which would be at the edge of the typical performance of an APS/CMOS detector.

The radiation environment and image integration/readout time are the most challenging drivers of the camera design.

4.3.2 Thermal Imager

The design of the thermal imager is based on the THERMAP design as proposed for the ESAs M3 candidate mission MarcoPolo-R RD[5]. The optical unit is based on a tri-mirror anastigmatic telescope with a focal length of 50 mm and 9 degree field of view. The instrument has an imaging channel and in an extension to the optical path, a slit spectrometer. Both units are using a separate microbolometer (manufacturer ULIS) of 640 by 480 pixels and a pixel size of 25 µm. The wavelength range between 8 and 16 µm is covered by the spectrometer part with a resolution of 0.3 µm. In addition the instrument requires another calibration channel pointing into deep space (Figure 4-1)

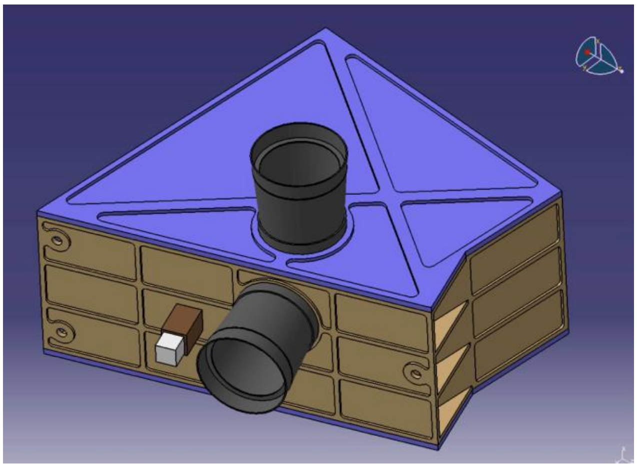


Figure 4-1: The THERMAP instrument as proposed for the MarcoPolo-R mission study (image reproduced from RD[5])

The principle design of the instrument takes strong heritage from the MERTIS instrument that has been build for Bepi Colombo mission RD[6]. This instrument combines an uncooled grating push-broom IR spectrometer with a radiometer rather than a second microbolometer for imaging as in THERMAP. In turn the mass is around 50% smaller.

Another example of a combined imager is the THEMIS instrument on Mars 2001 Odyssey mission RD[7]. THEMIS provides two imaging channels using the same optical unit but splitting the light beam to two uncooled micro-bolometers with filter in the visible/near-IR and mid-IR wavelength range.

In this study THERMAP properties are used for the model design case. In fact data of the spectrometer part are beyond the base scientific requirements. This is certainly an appreciated effect that merged instrumentation increases the scientific return while using similar resources. The drawback is a significant higher data volume that eventually has to be traded against the transmission of data from other instruments [section 4.5].

The overall instrument design is valid for the CLEO mission but certainly requires adaptation to the specific scientific target, instrument operations and radiation environment. A design driver for the detector is the large temperature range on Io surface ranging from 70K to 1700 K in small regions where magma is surfacing the moon's crust. Also the internal design, by using a dual acquisition channel with the same optics, requires a careful trade-off between a beam splitter and a flip mirror to channel the light to the corresponding detectors.

The design can be resource budget optimised by simply reducing the instrument to the imaging channel and applying at least two defined filter stripes to the detector.

4.3.3 Ion and Neutral Mass Spectrometer

Ion and neutral mass spectrometers play a crucial role by determining the chemistry of ions and gases lifted of a planetary surface, including the plasma environment. The current design takes heritage from an instrument which is currently built for the JUICE mission. The mass range is in the order of 1 to 1000 amu with a resolution of $m/\Delta m = 1100$.

The design is fully adaptable since the scientific targets and the environmental condition are comparable. The instrument has a circular view with an opening of 10 degree.

4.3.4 Magnetometer

The magnetometer is based on a design using magneto-resistive materials. The goal is a boom deployment. The current version of the spacecraft accommodated a spring deployed boom with a length of 80 cm. The boom design is flown on VenusExpress with a slightly larger length (1m).

Figure 4-2: The VeX magnetometer boom under test at ESTEC's test facility

4.4 Radiation Shielding

One major design driver for all sensitive hardware is the radiation environment. Two different parameters have to be taken into account for the instrument design approach. One is the pure survival of critical parts throughout the mission lifetime and specifically during operations. The other is the operation in a highly ionised environment where large quantities of charged particles deform sensor and front end electronic functionality.

4.4.1 Delocalised Backend Electronics

During the study it appeared as non-practical to place the whole instrument in a radiation resistant vault. The approach is to strip the backend electronics and place this in an isolated compartment together with all other system radiation sensitive equipment.

The definition of "back-end electronics" includes that part of the electronics that controls the instrument. All processing power required for data compression and storage is additional and has been transferred to the spacecraft's on-board data handling system.

The mass allocation of the backend electronics was based on the standardised, double mounted Eurocard size printed circuit board [Table 4-3]. The following assumptions have been taken.

Instrument	No. of PCBs	Mass [kg]
Camera	1	0.25
Thermal Imager	2	0.50
Neutral/ion MS	2	0.50
Magnetometer	1	0.25

Table 4-3: Backend electronics of the instruments

4.4.2 Front End Shielding

The shielding of the front end of the instruments that includes the sensor and front electronics, have to be addressed individually. This design exercise could be performed in the course of study but must be further addressed at a very early stage of the instrument development. The current mass allocation for an efficient shielding material is based on a rough estimate.

Instrument	Shielding mass (front end) [kg]
Camera	1
Thermal Imager	1
Neutral/ion MS	1
Magnetometer	0.5

Table 4-4: Allocated shielding mass at instrument front end

4.5 Payload Operations

The following fly-by scenario has been chosen to assess the instrument operations, performance and data volume production:

- Minimum flyby distance (from surface): 100 km
- Initial flyby velocity: 7 km/sec flyby velocity at closest approach: 7.41 km/sec

The instruments switch on/off sequence:

- Camera on/off +60.000 km/-60.000 km
- Thermal mapper on/off +60.000 km/-60.000 km
- Magnetometer high resolution on/off +36.000/-36.000 km
- Magnetometer low resolution on/off +60.000/+36.000 and -36.000/-60.000. (Note: in principle always on throughout the Jovian cruise)
- Neutral/ion MS on/off +12.600 km/-12.600 km

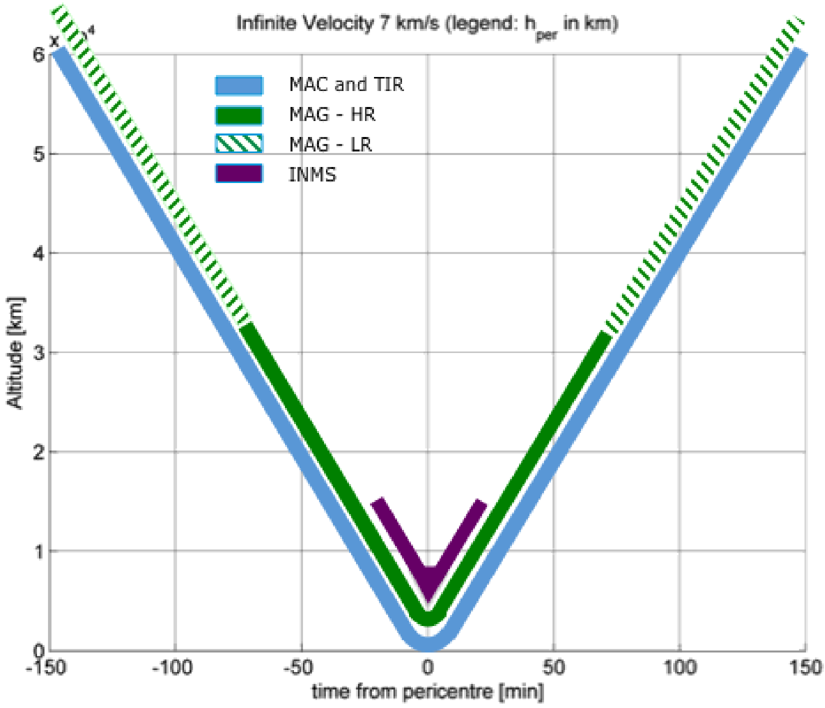


Figure 4-3: Science Operational sequence during flyby

Camera

- 60000 km to 100 km
- 143 minutes to closest approach
- Resolution (1 pixel) 2.21 km to 18 m (closest approach)
- Footprint 5554x5554 km to 18.5x18.5 km
- 286 images (1 per minute, 3 filter)
- Detector: 1024x1024x16 (compr. 1.8)
- = 8.0 Gb

Thermal Mapper

- 60000 km to 100 km
- Resolution (1 pixel) 30 km to 50 m (closest approach)
- Footprint 9444x9444 km to 31.5x31.5 km
- 143 images (1 image per 2 minutes)
- Detector: 315*315*16 (incl. 2 filter stripes) (compr. 1.5)
- =0.152 Gb + 10 % calibration data, =0.17 Gb
- Spectral information corresponding to 60 images above (not simultaneously)
- 60 "images"
- Detector: 315*315*16*40 (compr. 1.5)
- =2.54 Gb

NIMS

- Data rate 51.23 kb/s (incl. compression)
- Operation +- 30 minutes closest approach = 1 h total

- Distance ~±12600 km
- = 0.185 Gb

Magnetometer

- Data rate low resolution 0.128 kb/s
- Data rate high resolution 2.176 kb/s
- = 0.024 Gb

TOTAL data volume 8.38 Gbit + 2.54 Gbit spectral data

For the calculation of the total data volume accumulated during one flyby the spectral data were not to be taken into account because the spectral information is a goal and not a requirements for the Thermal Mapper instrument. The total data volume accounts to 8.38 Gbit science data and 0.62 Gbit housekeeping data. **The total data volume transmitted to Earth shall be 9 Gbit.**

After further iteration of the mission link budget the science data volume went under a strict revision to determine the minimum amount of data that must be returned. For that purpose, a higher compression factor of 7 has been applied to the camera and thermal mapper data. It has to be noted that this compression is not lossless as scientific information begins to vanish. **As a bare minimum, a science data volume in the order of 2.2 Gbit per flyby must be returned to Earth for a 2 flybys mission.**

4.6 Optional Instruments for a Europa Fly-By Mission

This section describes briefly an alternative payload selected for a fly-by mission at Europa. Since Europa is the main target of the CLIPPER mother spacecraft accumulating more than 40 fly-bys and in addition is a target of opportunity for the JUICE mission (2 fly-bys) the payload shall provide complementary measurements by the chosen instrumentation.

A complementary measurement may consist of:

- Higher spatial/spectral or mass resolution
- Extended dynamic range
- Larger target coverage
- Different instrumentation
- Different scientific measurement addressing different scientific theme.

The driving science case has been identified as the investigation of possibly existing plumes originating from Europa's surface. These plumes have been observed by the Hubble space telescope (RD[8]) , however, subsequent observations failed to confirm the initial data set. The plumes consist of gas and dust particles. Presumably the dust particles are not propelled into a higher orbit. They will remain in altitudes up to 25 km above the surface.

The following instruments have been selected as the model for a Europa flyby mission.

4.6.1 UV Spectrometer

The UV spectrometer characterises the exosphere of Europa. This includes the variety of different gas species but also limb observation on the dust plumes.

The UV spectrometer PHEBUS on BepiColombo is used as a design case. This double spectrometer covers the wavelength range between 55 nm -155 nm and 145 nm-315 nm with a resolution better than 1 nm.

The instrument has a mass of 7.6 kg and would fit into a volume of 500x400x400 mm (hxwxl). The average power consumption is 20 W. Per flyby it generates 10 Mbit of data.

4.6.2 Neutral and Ion Mass Spectrometer

This is the same instrument as for the Io flyby scenario (see section 4.3.3)

4.6.3 Dust Experiment

The possible presence of dust particles shall be addressed with an instrument that is unique on this spacecraft compared to CLIPPER and JUICE. Ideally suited is a combination of a dust counter with analytical capabilities. The impact velocity during the flyby is higher than 2.5 km/second thus the impacting particles disintegrate and will ionise to a large extent.

The resource envelope of the CLEO spacecraft foreseen for the payload is very limited. Thus a low mass/ low energy solution is required. Currently a complete unit that fits the requirements has not been developed. However, the combination of two existing instruments appears feasible. The Lunar Dust Experiment on the LADEE mission (RD[9]) serves as an example of a lightweight dust counter. The whole instrument weighs only 3 kg. An advanced breadboard of an extremely miniaturized time-of-flight mass spectrometer has been developed. It has a mass range from 1-300 amu with a resolution of m/Δm of 180. The instrument uses a laser ion source, which can be replaced by advanced ion optics, to channel the ions produced by impact ionisation to the mass spectrometer. The model including its electronics has an estimated mass of 0.5 kg. The overall combined instrument, of collector and analyser unit will be in the order of 4.5 kg considering several structural modification.

5 MISSION ANALYSIS

5.1 Requirements and Design Drivers

5.1.1 Requirements

The requirements (see details in Chapter 6.1) applicable to mission analysis are summarised below:

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
MI-GE-020	The CLEO mission design shall be compatible with CLIPPER mission baseline and back-up mission profiles : - Closest point to the Sun : 0.65 AU - 7.2 years interplanetary transfer with up to 1 VGA and 3 EGA . Note: This requirement is related to the arrival epoch and dynamical conditions of CLIPPER at Jupiter	
MI-GE-100	The CLEO S/C shall have the capability to use CLIPPER as relay for uploading science data to Earth. Note: This requirement drives the communication analysis (relay with CLIPPER or Direct To Earth (DTE))	
MI-GE-120	CLEO/P TT&C shall not foresee any data transfer to CLIPPER during Clipper flybys of EUROPA. Note: This requirements specifies that no relay is possible during CLIPPER's Europa Gravity Assist (EGA)	
MI-IO-010	The mission shall be able to perform IO flybys at altitude of 100 km.	
MI-IO-020	The mission shall perform at least 2 flybys of Io.	
MI-IO-030	The mission should target at least 2 flybys at opposite high magnetic latitudes.	

5.1.2 Design Drivers

The design drivers are:

- Minimise ΔV
- Maximise number of flybys for science
- Minimise radiation dose
- Provide sufficient time between flybys to allow transmitting back the science data.

These requirements exclude each other, e.g. performing more fly-bys will incur more radiation, or reducing the radiation dose will incur more ΔV.

Therefore several scenarios have been analysed to allow the team to perform a trade-off.

5.2 Assumptions and Trade-Offs

5.2.1 CLIPPER

The analysis was conducted assuming CLIPPER's arrival date is in April 2028 (the corresponding Jupiter tour is 13F7 according to JPL nomenclature).

For this interplanetary transfer, the infinite velocity w.r.t. Jupiter is 5.58 km/s and the declination w.r.t. Jupiter's equator is -4.6 deg.

After a first Ganymede Gravity Assist (GGA), the Jupiter Orbit Insertion (JOI) is performed such that CLIPPER is injected into a 200 days orbit.

At the first apojove, the Perijove Raising Manoeuvre (PRM) is performed, mainly to compensate the Sun gravity pull.

Then a sequence of GGA and Callisto GA (CGA) is used to reduce the energy, the inclination and the infinite velocity, see Figure 5-1 and Figure 5-2. Finally Callisto is used to reduce the perijove close to Europa orbital radius.

Then the Europa science begins. The first EGA is performed roughly one year after JOI.

The entire trajectory from JOI to EGA#1 (E1) is represented in Figure 5-3.

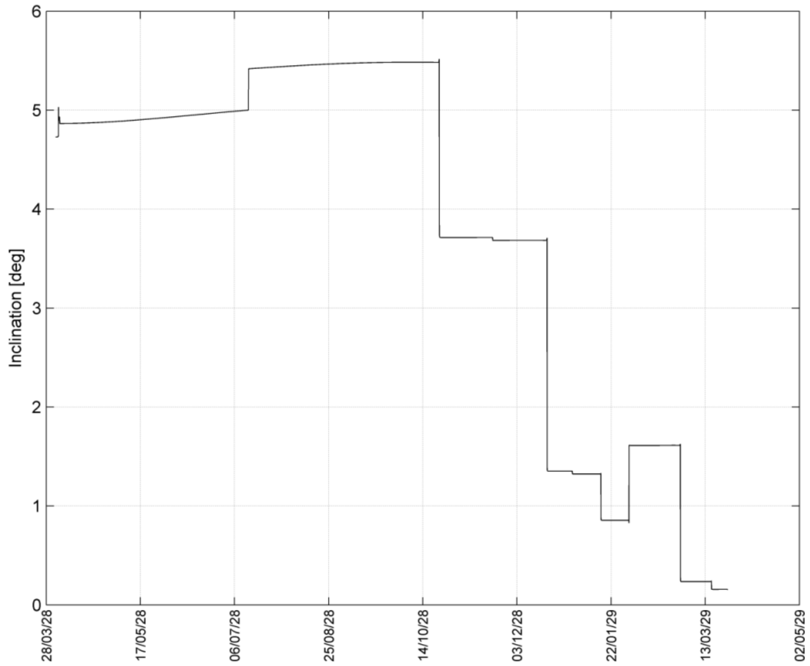


Figure 5-1: Evolution of CLIPPERS's inclination w.r.t. Jupiter's equator after JOI

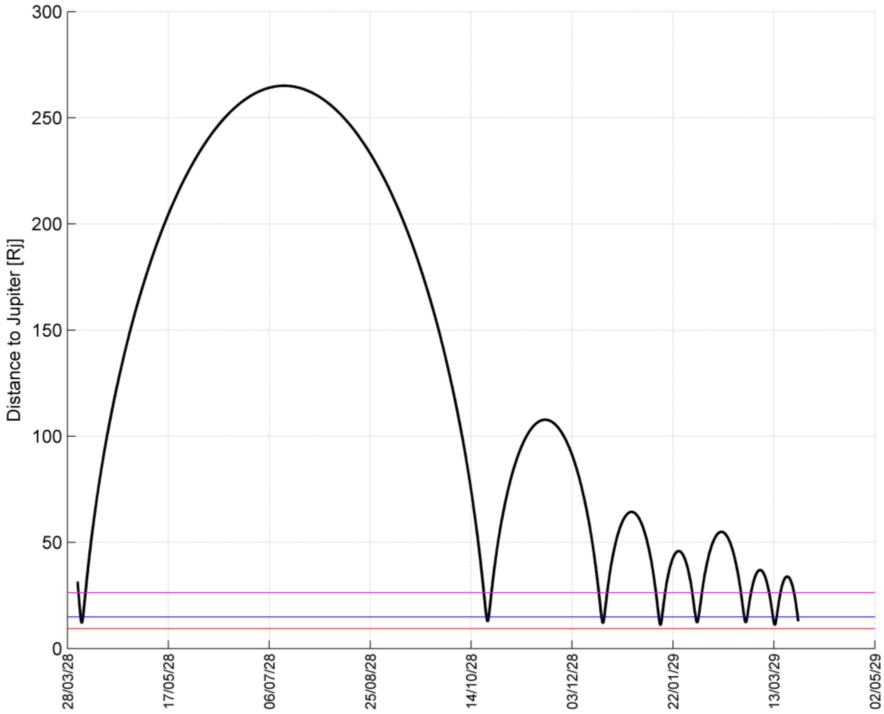


Figure 5-2: Evolution of the distance to Jupiter for CLIPPER after JOI

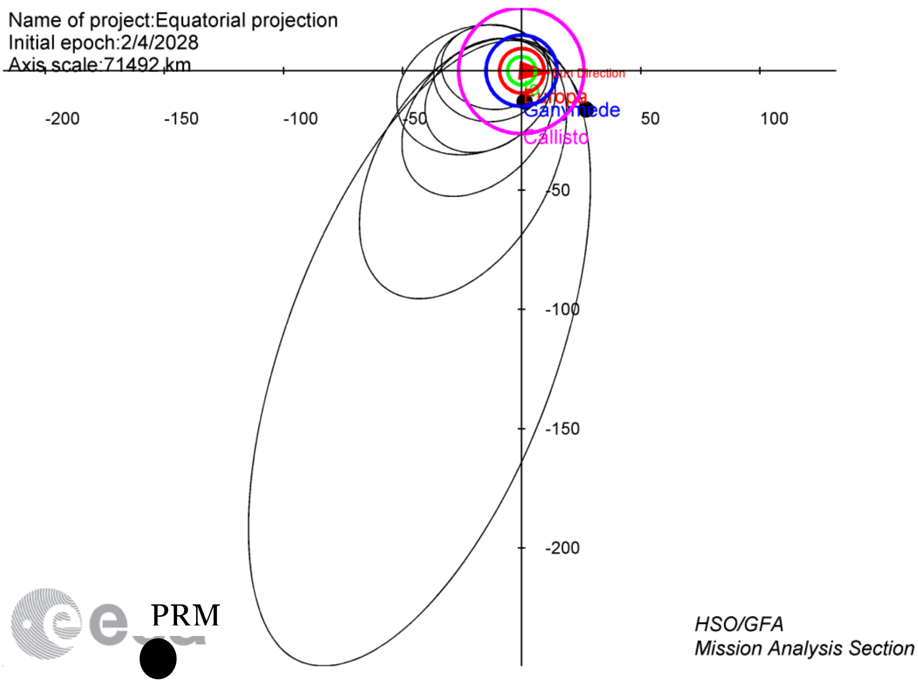


Figure 5-3: XY projection of CLIPPER's trajectory from JOI to E1 (7E1 in JPL's nomenclature) in Jupiter's equatorial of date (X-axis as the intersection of Jupiter's equator of date with Earth equator of date)

The CLIPPER Europa science is made of phases, where both the infinite velocity w.r.t Europa (around 4 km/s) and the orbital period (4:1 resonant with Europa, i.e. ~14.2 day) are rather constant.

The first phase is called Crank-Over-the-Top-1 (COT1) for a total of 6 EGA. It lasts roughly 3 months. The evolution of the distance to Jupiter is shown in Figure 5-4, while the trajectory is shown in Figure 5-5. The second phase, COT-2 finishes 1.5 year after JOI.

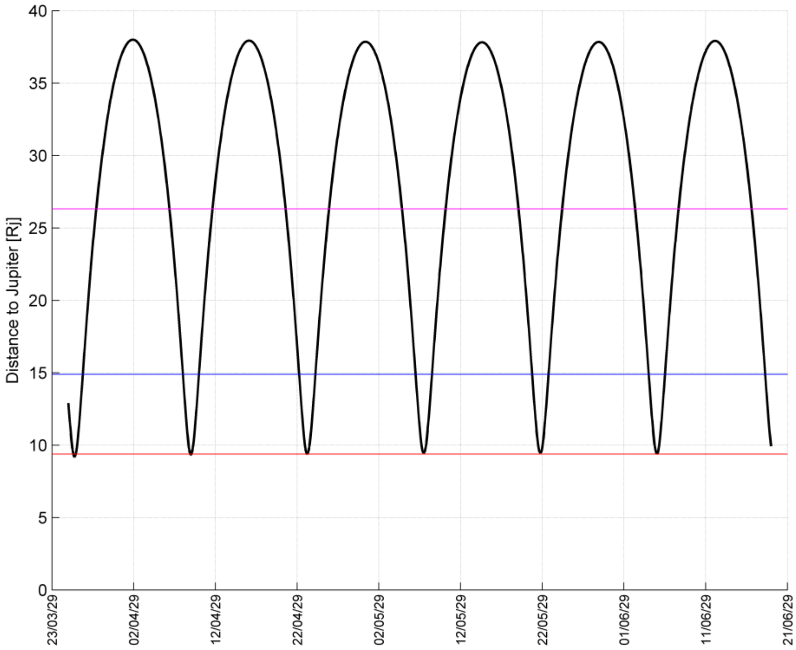


Figure 5-4: Evolution of the distance to Jupiter for CLIPPER during COT-1

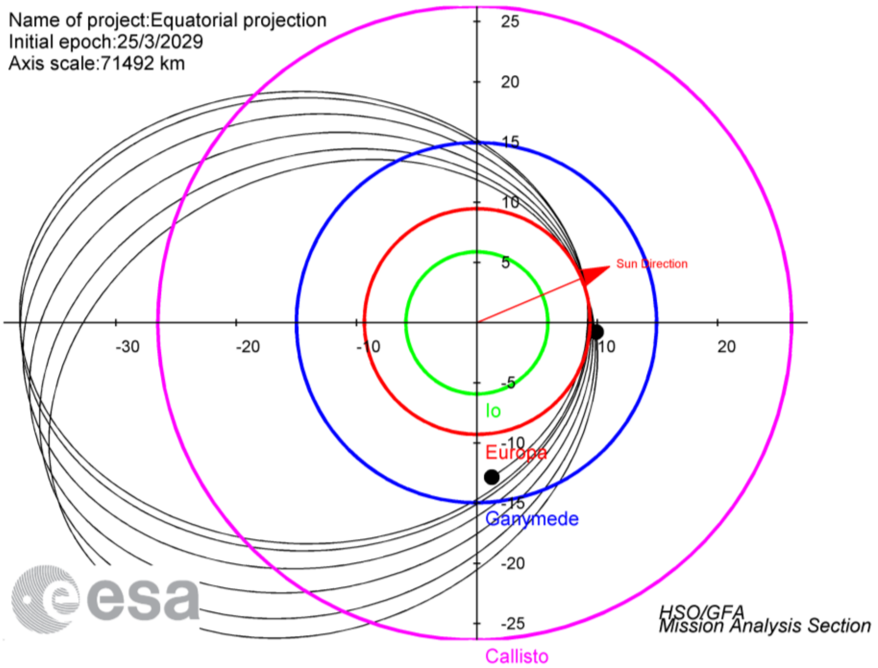


Figure 5-5: XY projection of CLIPPER's trajectory during COT-1

5.2.2 Io Science

Io orbit is near equatorial, near circular. Its orbital radius is ~420 000 km, i.e. ~5.9 Jovian radii (R_J). Io rotation is locked: the duration of a day is equal to the orbital period (1.78 day). The prime meridian is always pointing towards Jupiter, while the 270 deg East meridian is always aligned with Io velocity vector w.r.t. Jupiter.

A preliminary list of targets to fly-by during I1 and I2 is given in Figure 5-6 (this list was given by the CLEO science team).

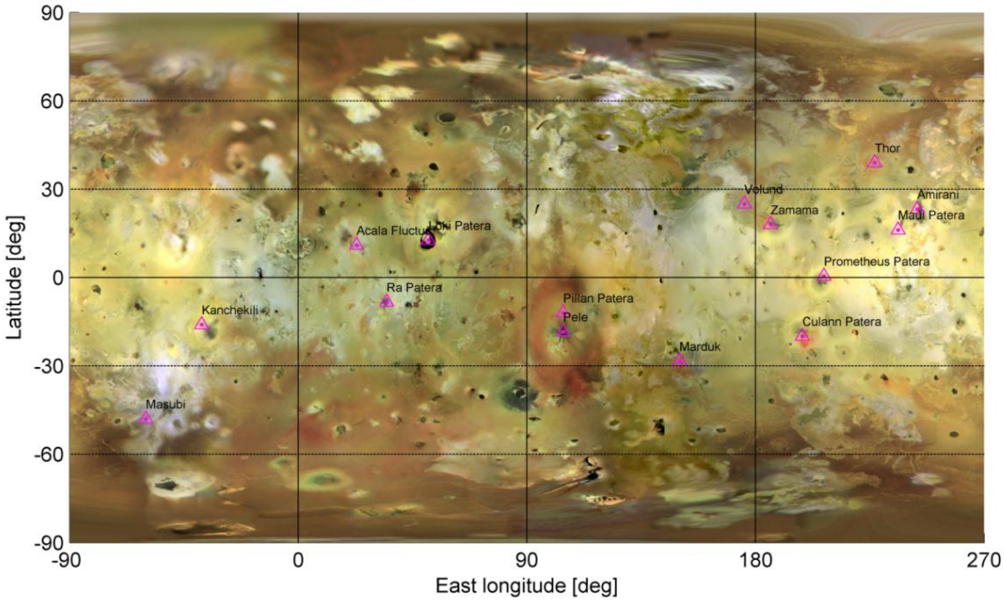


Figure 5-6: Preliminary list of targets for Io

5.2.3 Overview of the Scenarios

As explained in section 5.4, several scenarios have been proposed to the team. A qualitative comparison of the different scenarios is given in Table 5-1.

	S1	S2b	S3	S4	Europa
Separation	< JOI	> PRM	> PRM	> JOI	> PRM
V_∞ / moon	--	=	+	=	+
Inclination / Jupiter	free	~0	~0	low	~0
Wet mass	++	=	=	++	=
ΔV	--	=	+	++	+
Radiation dose	+	=	-	++	+
Science phase orbital period	=	=	--	N/A	--

Table 5-1: Qualitative comparison of the different scenarios. The baseline, S2b, is quoted with symbols '='. The other scenarios are compared w.r.t. the baseline

For details of the options refer to section 5.4. The main drawback of scenario CLEO-I S1 is the high infinite velocity w.r.t. Io, which is incompatible with the payload (camera smearing). Another drawback is the high ΔV budget (partially compensated by the higher wet mass). Its main advantage is the low radiation dose (no transfer to Io, high infinite velocity, inclined fly-bys).

The main drawback of CLEO-I S3 is the short orbital period (<20 day), which highly complicates the operations and science data download. Its main advantage is the low ΔV budget.

The main drawback of CLEO-I S4 is that only one IGA can be performed (at relatively high altitude because of Io initial ephemeris error). Its main advantage is the extremely small ∆V required.

The Europa scenario CLEO-E is very comparable in its design with S3, except that lower radiations are incurred (due to the higher orbital radius of Europa compared to Io).

5.3 Baseline Design

The baseline scenario is S2b.

5.3.1 From Separation to Io

CLIPPER's inclination w.r.t. Jupiter after JOI is 5.4 deg. Because a preliminary GGA is performed before JOI (named G0), the inclination shall be reduced to virtually 0 deg before transferring to Io (all Galilean moons are close to Jupiter's equator).

CLEO-I separates from CLIPPER shortly after the PRM (e.g. one week after the PRM Clean-Up manoeuvre (CU)) such that the related ∆V cost is saved. G1 B-plane is retargeted at low cost such that G1 correct as much inclination as possible. The infinite velocity at G1 is 6.2 km/s. Assuming a swing-by pericentre altitude of 100 km, the deflection is 9.8 deg. However the incoming infinite velocity vector declination is 12.3 deg. This means that G1 alone is not sufficient to be equatorial to further transfer to Io.

Therefore G2 is used to finish the inclination correction. At the apojove after G2, the perijove is reduced from the current value down to Io orbital radius. In order to minimise the size of this manoeuvre (the Perijove Lowering Manoeuvre (PLM)), the larger the apojove the better: this was obtained by keeping after G1 and G2 the same orbital period as after JOI: 200 days (depending on the ∆V needs, this value could easily be tuned in the future).

The PLM is implemented 3 months after G2 to reduce the perijove down to Io orbital radius (it is optimal from a radiation dose point of view not to go lower). Its value is ~250 m/s. Three months after the PLM, I1 is performed. It means that I1 takes place ~1.5 year after JOI, i.e. at the end of COT-2. The infinite velocity w.r.t. Io is 7 km/s.

Note that the amplitude of the PLM is driven by the insertion strategy of CLIPPER, which performs a JOI with a Ganymede flyby; would a JOI with a Io flyby be proven to be feasible for CLIPPER, then this would allow to drastically reduce the overall ∆V for this option.

5.3.2 Fly-by I1

For I1 the fly-by pericentre altitude is set to 500 km. This value is chosen to safely (collision risk) cover any Io ephemeris error (typically 99%). This leaves one free parameter to target the B-plane. A parametric representation is given in Figure 5-7.

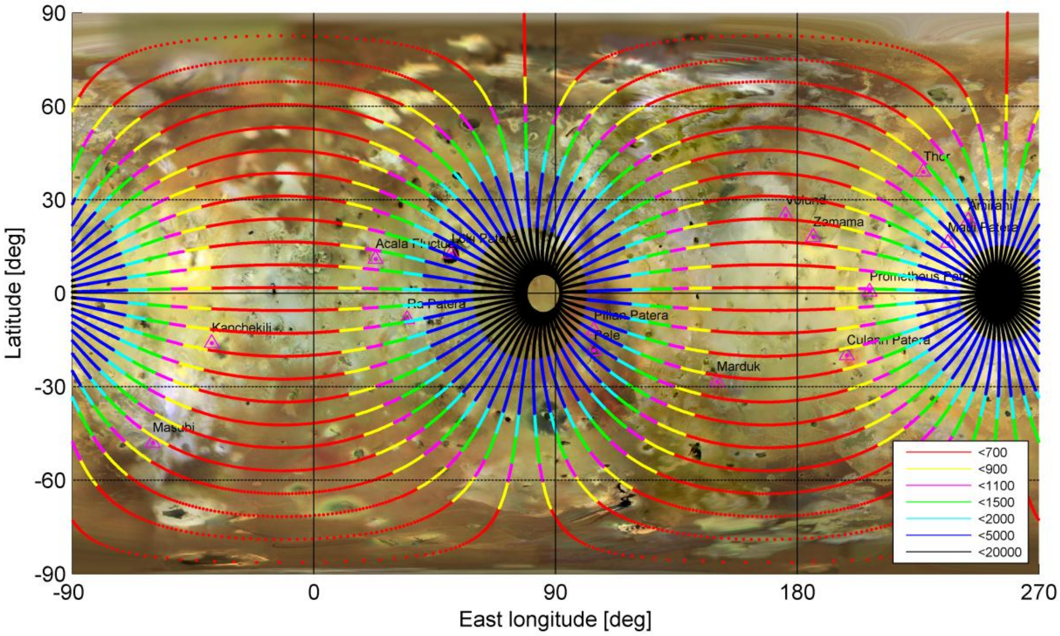


Figure 5-7: Parametric analysis of I1 for the baseline

Because the perijove is equal to Io orbital radius and also because CLEO-I orbit is near equatorial, the incoming infinite velocity vector direction is close to Io's equator and close to the 90 deg meridian, i.e. from "behind" Io. Therefore the C/A will always be close to the inner or outer meridians, while its latitude is free.

An interesting region was selected, where the groundtrack passes at C/A over Kanchekili on the inner meridian (C/A latitude @20 deg South), see Figure 5-8.

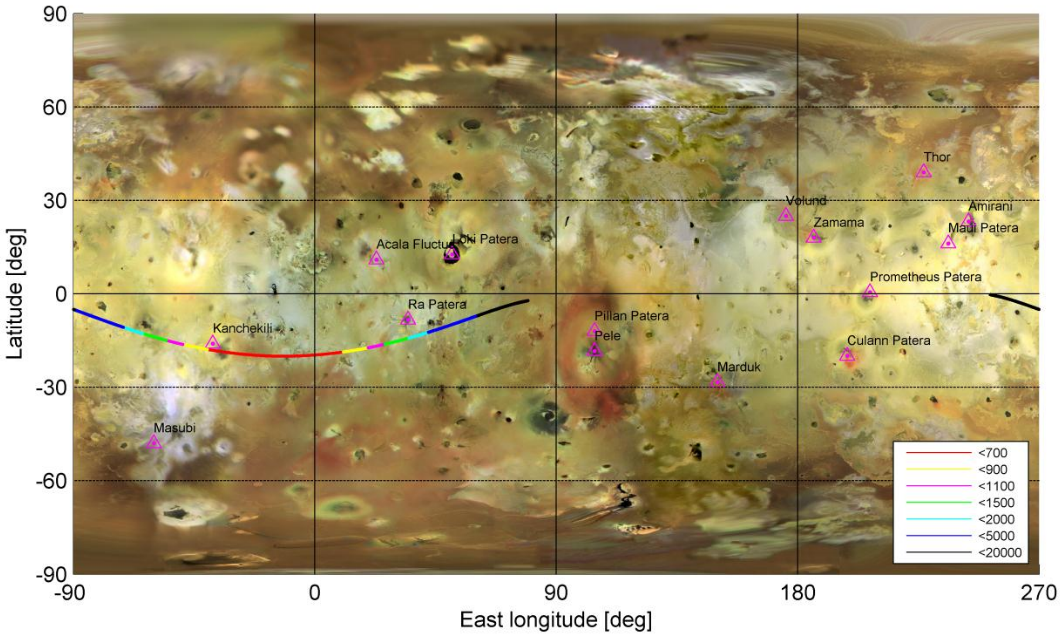


Figure 5-8: I1 solution

The main features of I1 are summarised in Table 5-2.

	V∞ / Io [km/s]	7
	Pericentre altitude [km]	500
Incoming	Pump [deg]	11.7
	Crank [deg]	0
	Resonance ratio	105:1
	Inclination / Jup. Eq. [deg]	0.0
	Perijove [R_J]	5.9
	Apojove [R_J]	257
	Orbital period [day]	186
Outgoing	Pump [deg]	17
	Crank [deg]	7
	Resonance ratio	61:1
	Inclination / Jup. Eq. [deg]	0.6
	Perijove [R_J]	5.9
	Apojove [R_J]	178
	Orbital period [day]	110

Table 5-2: I1 Summary

5.3.3 Fly-by I2

For I2 the fly-by pericentre altitude is set to 100 km. This value is chosen to take into account the improvement of Io ephemeris after I1. This leaves one free parameter to target the B-plane. A parametric representation is given in Figure 5-9.

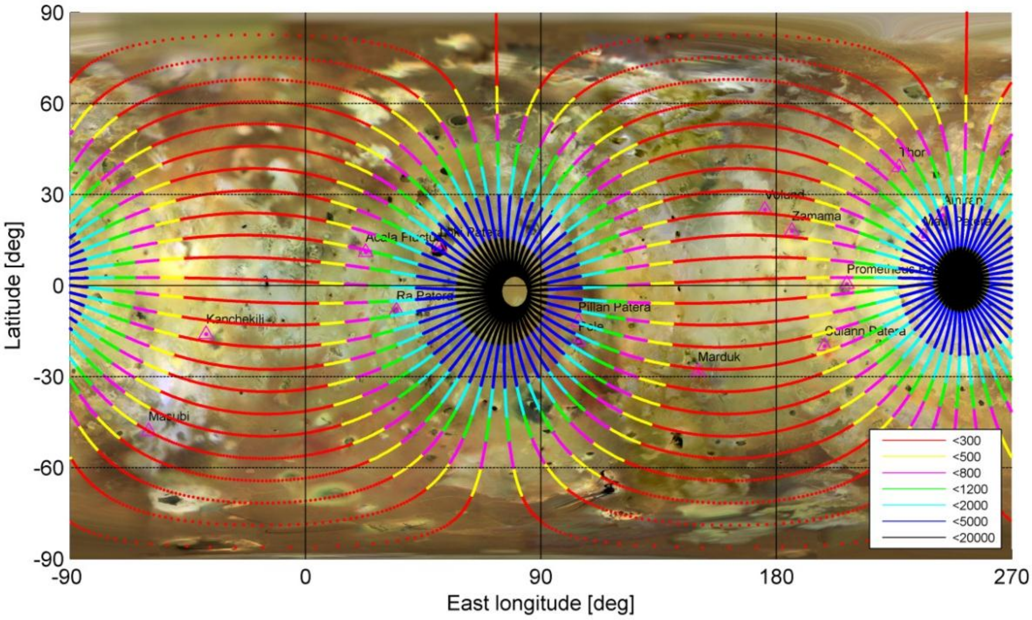


Figure 5-9: Parametric analysis of I2

The possible groundtracks are similar to I1, essentially because the deflection of I1 was small (6.8 deg).

An interesting region was selected, where the groundtrack passes at C/A close to Voluna and Zamama on the outer meridian (C/A latitude @20 deg North), see Figure 5-10.

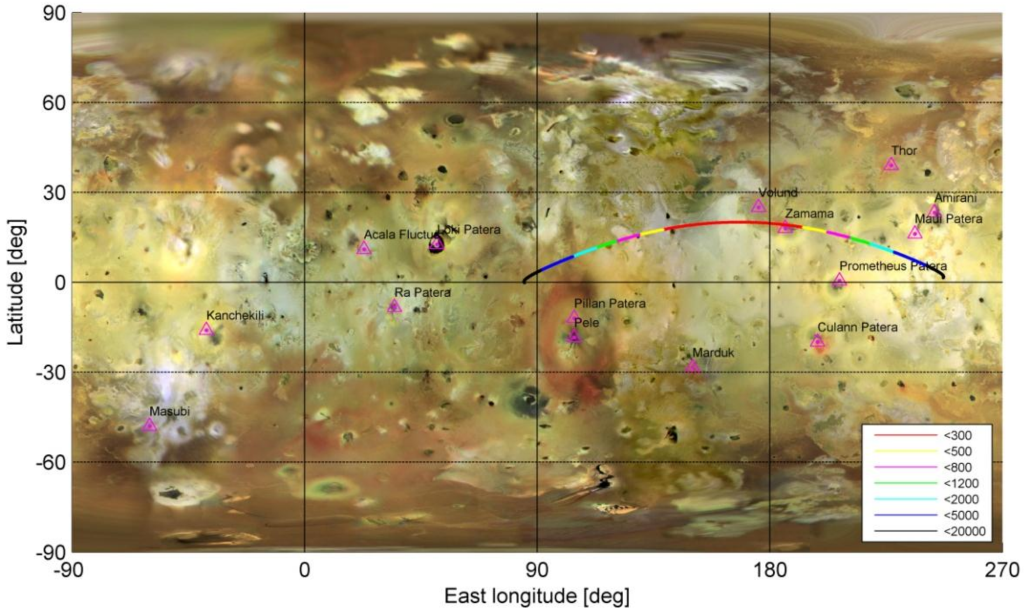


Figure 5-10: I2 solution

The main features of I2 are summarised in Table 5-3.

	V_∞ / Io [km/s]	7
	Pericentre altitude [km]	100
Incoming	Pump [deg]	17
	Crank [deg]	7
	Resonance ratio	61:1
	Inclination / Jup. Eq. [deg]	0.6
	Perijove [R_J]	5.9
	Apojove [R_J]	178
	Orbital period [day]	110
Outgoing	Pump [deg]	11
	Crank [deg]	-2
	Resonance ratio	115:1
	Inclination / Jup. Eq. [deg]	0.1
	Perijove [R_J]	5.9
	Apojove [R_J]	274
	Orbital period [day]	204

Table 5-3: I2 Summary

Mission extension is possible: I3 would take place ~6 months after I2 and would only require the B-plane retargeting.

Mission termination may consist of either an Impact at Jupiter or an impact at Io. This was not covered in detail during the CDF (only 15m/s was allocated for the disposal manoeuvre) and needs to be covered in a later phase of the mission.

5.3.4 Fly-bys Common Features

The two fly-bys have commonalities because the infinite velocity is the same while the pericentre altitude is close (500 km for I1, 100 km for I2). The following plots show various figures of merit helpful for sizing AOCS or instruments.

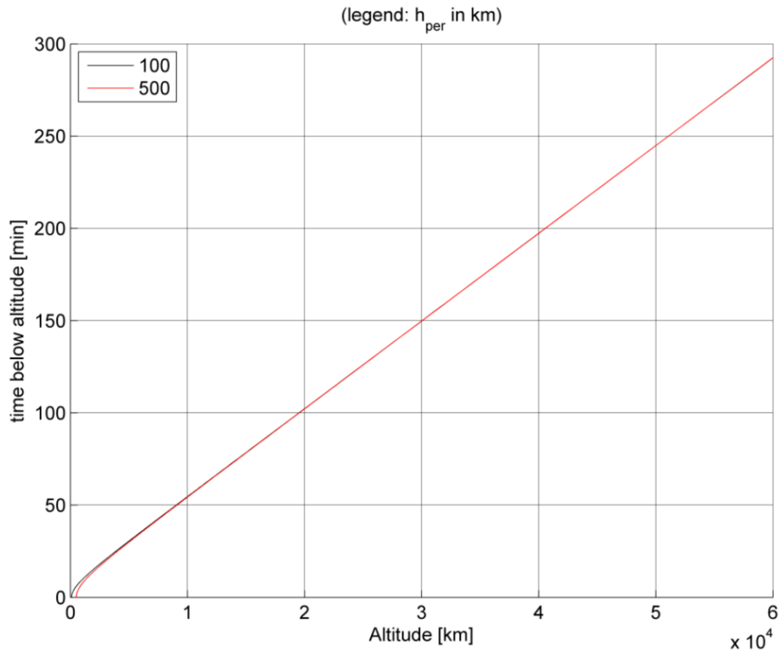


Figure 5-11: Time below altitude vs altitude for the baseline

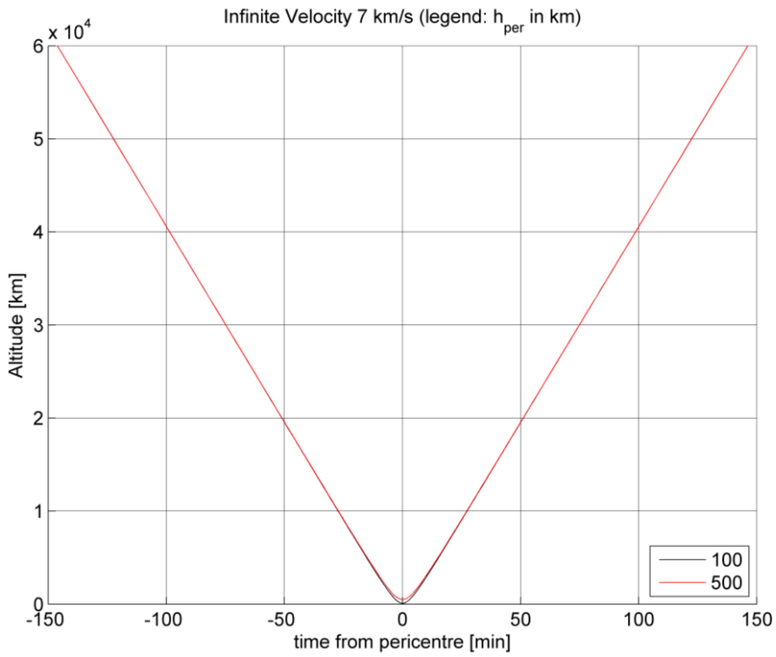


Figure 5-12: Altitude vs time from pericentre for the baseline

In Figure 5-13 the velocity tends by definition towards the infinite velocity when the time to pericentre increases (or equivalently the distance to the moon). It can be seen that the difference between the infinite velocity and the velocity at C/A is rather small (300-400 m/s) because the infinite velocity is large (7 km/s) and Io gravitational constant rather small (compared to e.g. that of Jupiter).

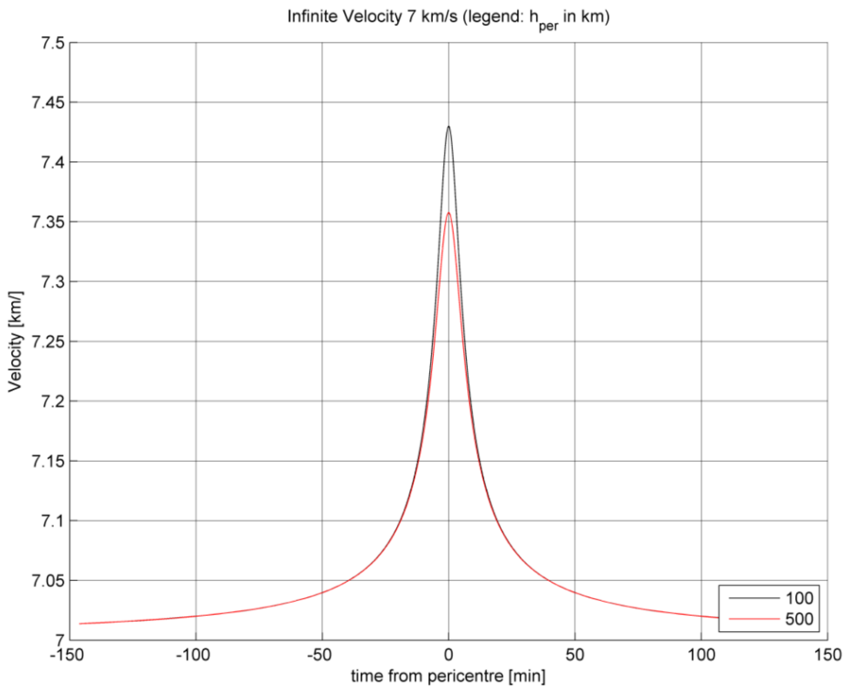


Figure 5-13: Velocity vs time from pericentre for the baseline

In Figure 5-14, the Flight Path Angle (FPA) is close to 90 deg when far from the moon: it reflects a radial approach. The FPA becomes zero by definition at pericentre.

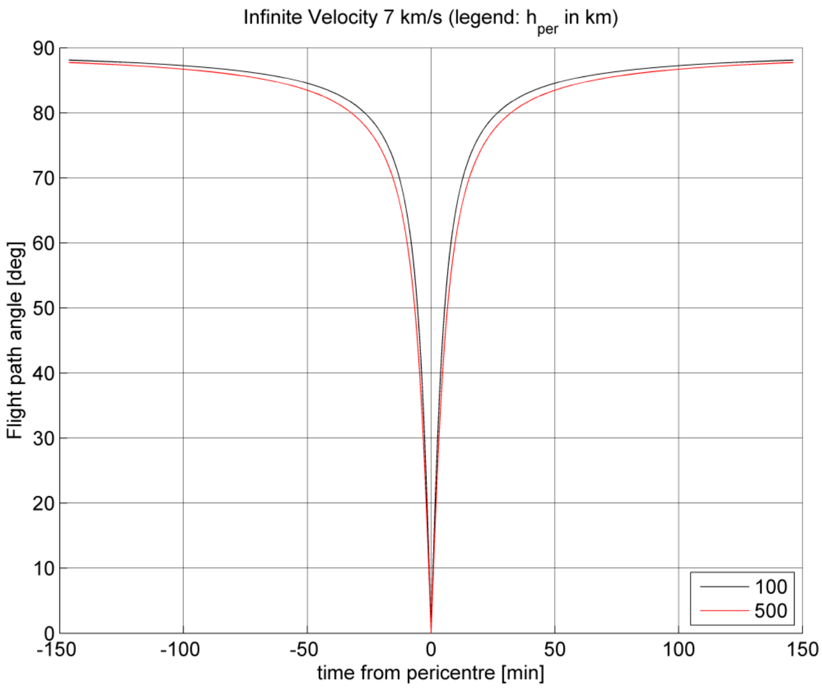


Figure 5-14: Flight path angle vs time from pericentre for the baseline

Figure 5-15 shows the variation of the angular velocity as a function of the altitude assuming nadir pointing. It can be seen that the maximum is ~200 mdeg/s at pericentre.

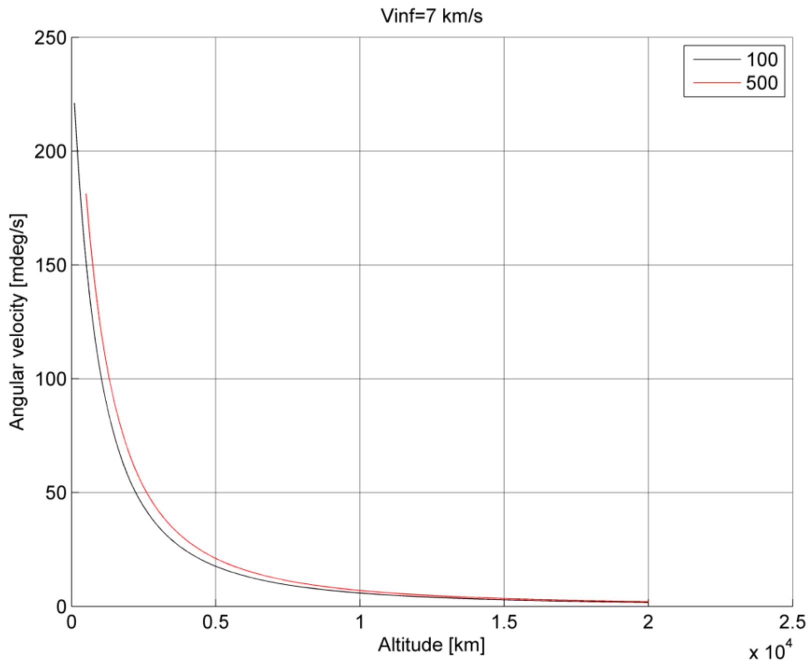


Figure 5-15: Angular velocity vs altitude for the baseline

Figure 5-16 shows the variation of the angular acceleration as a function of the altitude assuming nadir pointing. It can be seen that the maximum quickly evolve with the altitude: 350 µdeg/s² for a C/A @500 km and 550 µdeg/s² for a C/A @100 km. The angular acceleration is null at pericentre (the acceleration is a function of *sin(FPA)*) and at infinity (the acceleration is a function of *1/r*).

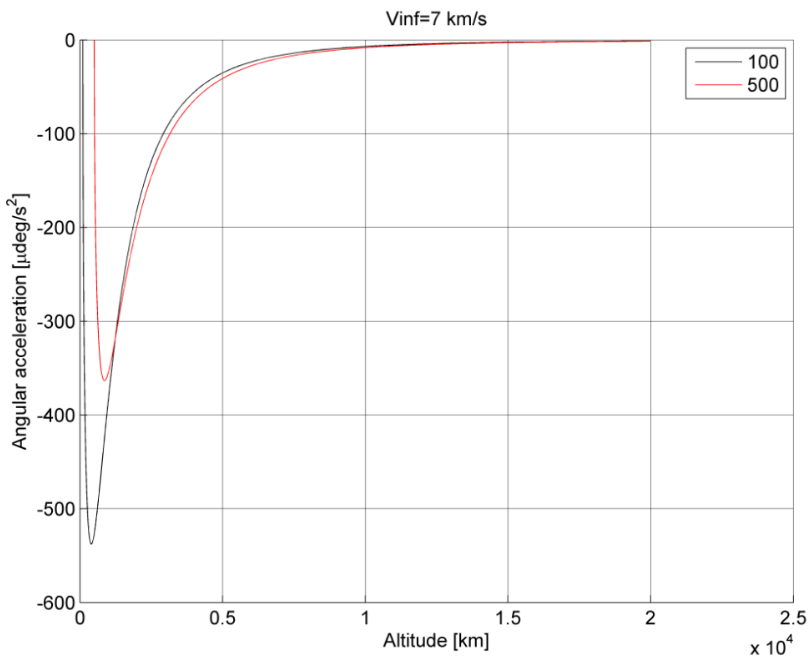


Figure 5-16: Angular acceleration vs altitude for the baseline

5.3.5 Communications

5.3.5.1 Relay with CLIPPER

The possible interest to use the CLIPPER S/C as communication relay has been analysed (but finally not retained as the baseline at this stage).

The trajectories of CLIPPER and CLEO-I are represented in Figure 5-17 for the phase starting at I1 and end at the hypothetical I3.

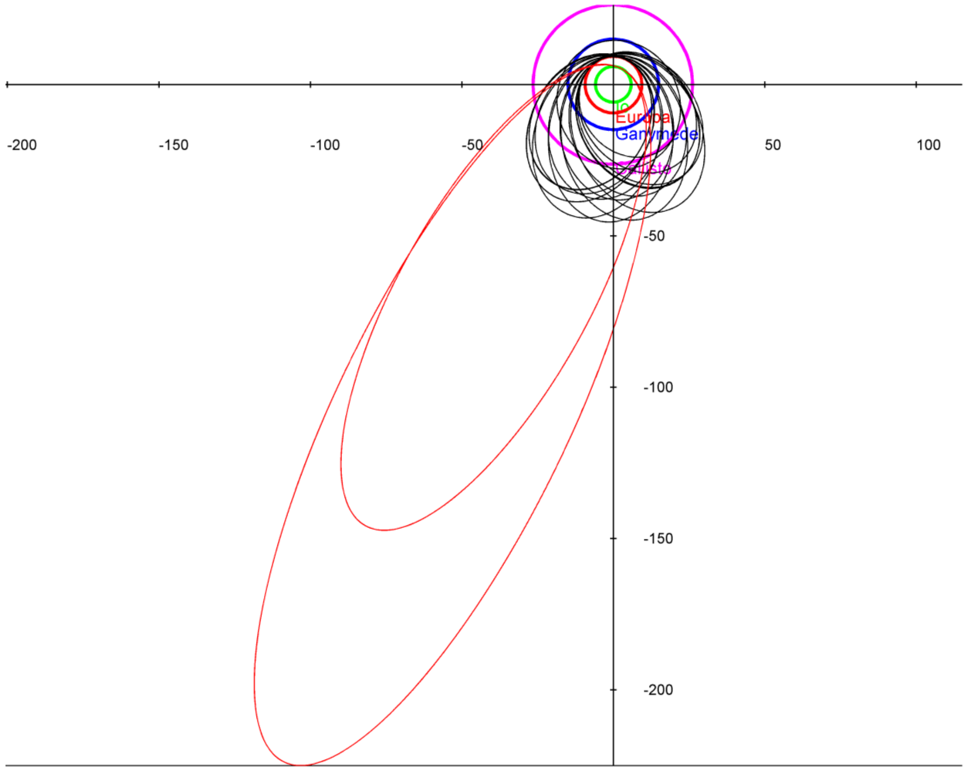


Figure 5-17: XY projection of CLIPPER's trajectory (in black) and CLEO-I (in red) for the phase starting at I1 and ending at the hypothetical I3. The axis unit is Jovian radius

It is clear that due to the eccentricity of CLEO-I, the distance quickly increases outside IGA. This is shown in Figure 5-18, where the evolution of the distance is given as a function of time.

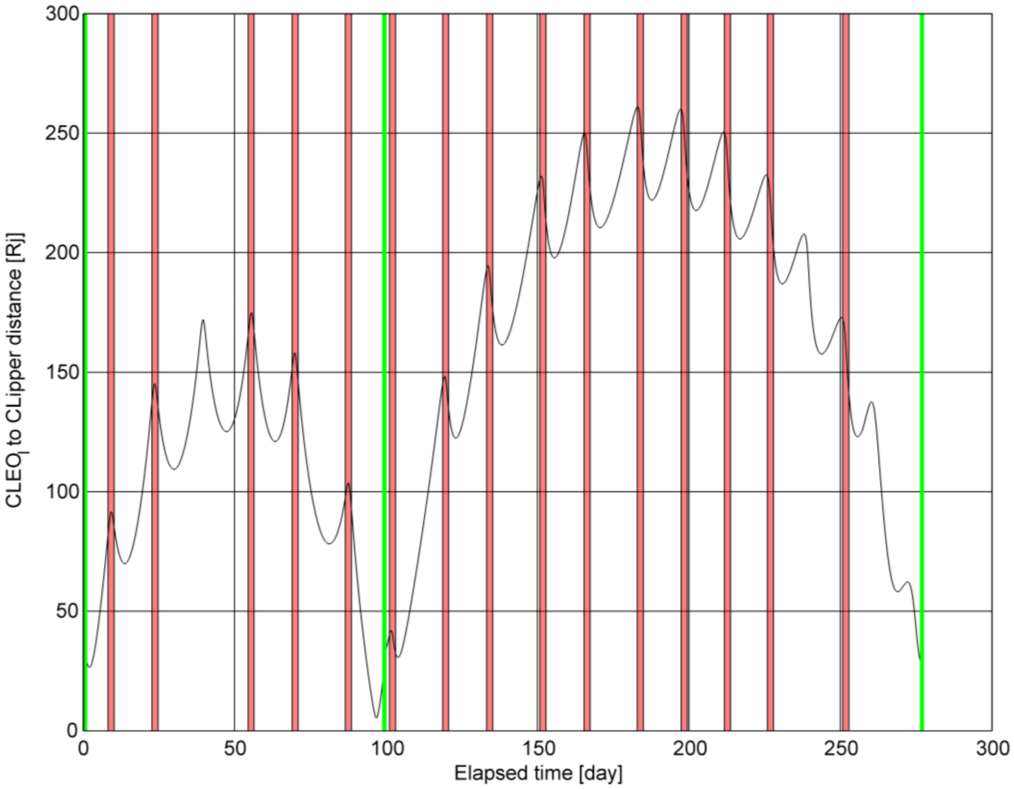


Figure 5-18: Evolution of the distance between CLEO-I and CLIPPER from I1 to "I3". The X-axis origin is the epoch of I1. The IGA are shown as green lines. CLIPPER's EGA are shown in red stripes (+/- 2 days around C/A)

Close to IGA, the distance is around 30-40 R_J. This has to be compared with CLIPPER's apojove during COT, which is equal to nearly 40 R_J. A perfect phasing of both spacecraft could permit to have less than 40 R_J for a reduced amount of time (typically one day), but it sounds more realistic to consider CLIPPER's apojove as a lower bound for radio link budget.

5.3.5.2 Direct To Earth (DTE) link

In the previous paragraph, the link budget was affected by the distance between CLEO-I and CLIPPER. In the case of DTE it is influenced by the distance from the CLEO-I to the Earth. It is given in Figure 5-19.

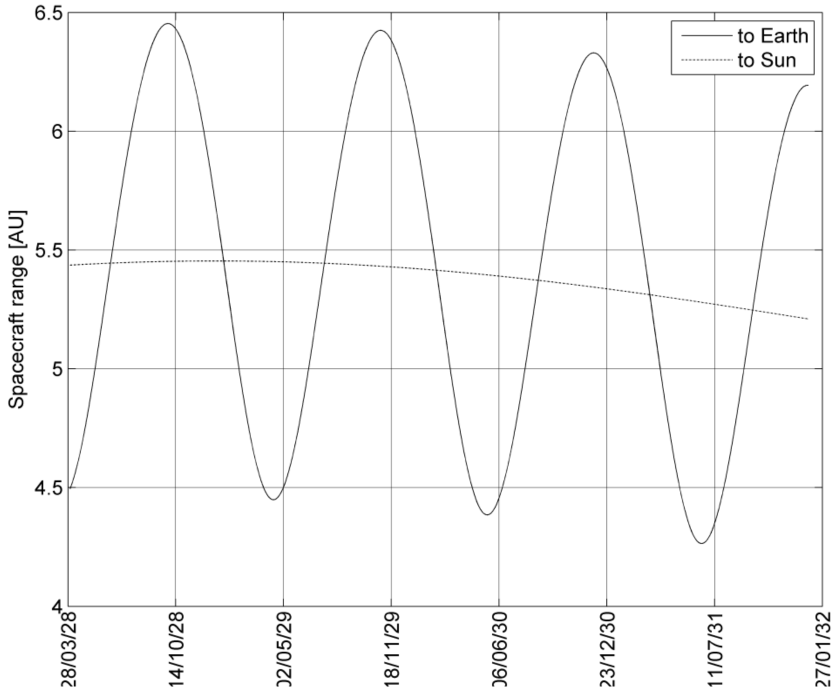


Figure 5-19: CLEO-I distance to the Sun and the Earth

The distance to the Sun slowly varies due to the eccentricity of Jupiter's orbit. At JOI it is around 5.5 AU. The distance to the Earth superimposes a yearly variation of +/- 1 AU. At I1 (Q3/2029), the distance is maximum, around 6.5 AU. At I2 (3 months later), it is back to 5.5 AU.

In the absence of a consolidated scenario, it is recommended to consider 6.5 AU as a sizing case for link budget.

The evolution of the maximum elevation as seen from ESA ground stations is given in Figure 5-20.

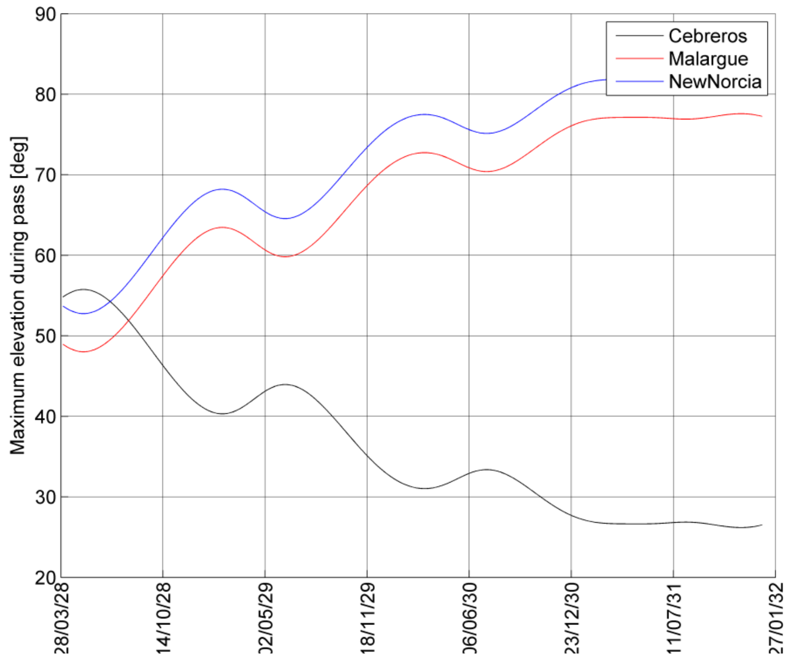


Figure 5-20: Daily maximum elevation vs time for ESA ground stations

With the current mission, southern hemisphere ground stations (New Norcia and Malargüe) have to be favoured: at epoch of I1, the maximum elevation is 35 deg for Cebreros while it is around 70 deg for the southern stations.

The corresponding duration of daily passes are given in Figure 5-21 (for a minimum elevation of 10 deg).

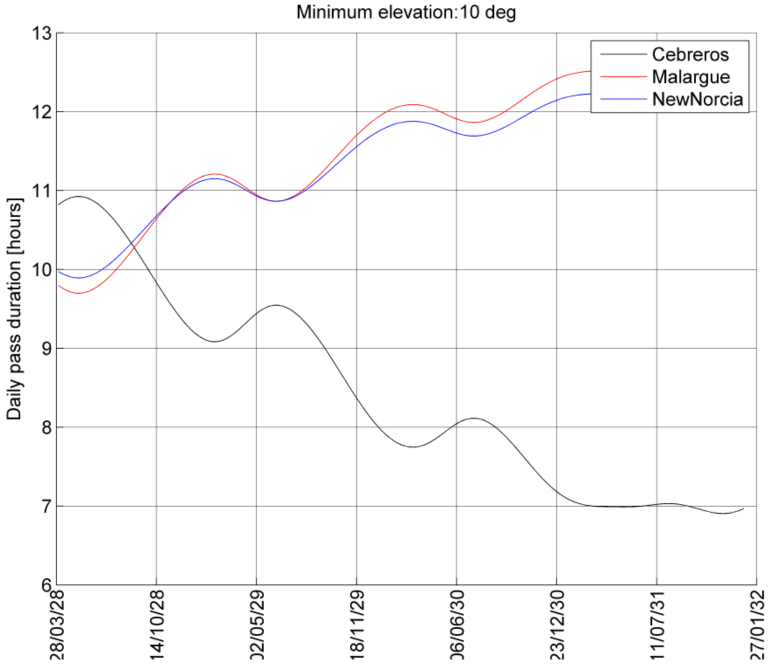


Figure 5-21: Daily pass duration vs time for ESA ground stations

For southern stations, more than 11 hours daily per station are guaranteed.

5.3.6 Navigation and Operational Concept

Only preliminary considerations have been addressed concerning Navigation and Operations in the frame of this CDF study.

5.3.6.1 Measurements

The baseline measurements are line of sight: range and Doppler. For other similar missions, a daily pass of 8 hours is assumed. For critical operations (e.g. JOI, moon-GA) a continuous coverage is also assumed for other missions.

For power reasons on S/C side, 8 hours continuous measurements cannot be envisaged. Calculating the minimum duration for acceptable Orbit Determination (OD) is not possible at CDF level, but a rule of thumb is that 4 hours is the bare minimum for a continuous measurement. Increasing the frequency of measurements is not very useful, as 1). they tend to be more and more affected by the same noise and 2). the measurements cover a smaller arc, thus with less orbital dynamics to observe.

In order to complement line of sight measurements, plane of sky measurements can be added, namely DDOR and opnav.

If the arc of line of sight measurements is large enough (typically 8 hours after GA), DDOR measurements are useless. On the other extreme, if the arc is small (typically 1-2 hours after GA), DDOR is a perfect complement.

Opnav measurements of the moons will help targeting the fly-bys (by reducing the moon ephemeris error). However it will not help reducing the Clean-Up (CU) manoeuvre, for which the OD wr.t. Jupiter is the driver. For the CU it might be helpful to perform opnav measurements of Jupiter.

As a baseline, 4 hours range + Doppler measurements are considered (spread along the orbit, prior to any CU and prior to any Retargeting Manoeuvre) For more information check the mission timeline in section 6.3.2 – a dual navigation / science camera is recommended to provide optical navigation measurements and science data during flybys. This camera would then be under ESA responsibility.

5.3.6.2 Guidance

A standard scenario is kept with three Trim Correction Manoeuvres (TCM):

- CU: typically 3-4 days after GA, purely stochastic, depends how well the GA was performed. This manoeuvre is the largest contributor to the navigation ΔV
- Apojove manoeuvre: this manoeuvre has a deterministic component to target next fly-by B-plane and a stochastic component to correct the CU dispersions
- Targeting: typically 2-3 days before GA, purely stochastic to correct the apojove manoeuvre dispersions.

Each TCM shall be performed after a ground process (measurements, Orbit Determination, manoeuvre computation, manoeuvre uplink).

5.3.7 Power

The computation of the maximum duration of an eclipse highly depends on a specific scenario. However there are commonalities:

- Eclipse by the Galilean moons: there might be eclipse by Ganymede and Io during fly-bys of typically less than 10 min
- Eclipse by Jupiter: long eclipse (typically 6-8 hours) could occur close to apojove. The baseline correspond to such a case: after JOI the Sun direction to CLEO-I perijove direction is about 45 deg. I1 is about 1.5 year after JOI, i.e. the Sun direction to CLEO-I perijove direction is close to 0 deg, meaning the apojove is close to Jupiter's shadow direction.

However there is enough flexibility in the design (epoch, inclination, perijove direction) to avoid such a case. In order to be conservative, it is recommended to keep 3 hours for worst case eclipse by Jupiter

5.4 Options

5.4.1 CLEO-I S1

S1 is intended to minimise the radiation dose per IGA. This is obtained by:

- Having inclined fly-bys (inclined w.r.t. Jupiter's equator)
- Having high infinite velocity at Io combined with the perijove at Io's orbital radius.

Such a scenario is obtained by separating from CLIPPER before JOI and retargeting a very low JOI, like for the Io Volcano Observer mission: 5000 km (above the reference 1 bar altitude). The selection of the inclination is free. It is taken equal to 45 deg. The JOI is performed by CLEO-I itself: 470 m/s to enter a 6 months period orbit.

The large PRM (to raise the perijove to Io and to counteract the Sun gravity pull) is 300 m/s.

Then the spacecraft flies directly towards I1. The infinite velocity w.r.t. Io is 17 km/s. It turned out to be incompatible with the payload requirements. AS an illustration a typical sequence of 4 IGA is shown in Figure 5-22.

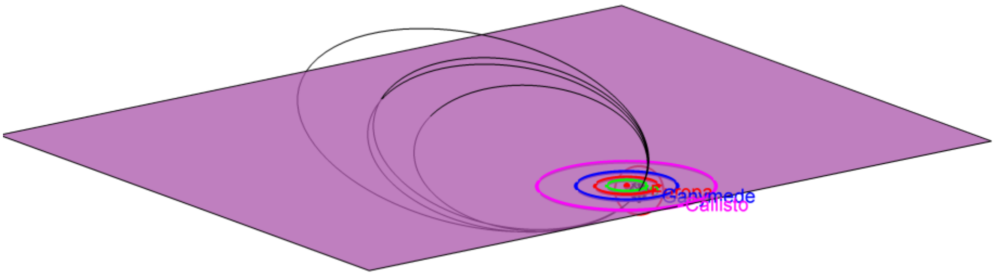


Figure 5-22: Illustration of 4 IGA for scenario S1. The 45 deg inclination is clearly visible

Moreover the ΔV budget for this scenario is very high. However it is partially compensated by the fact that the separation takes place before JOI: in the baseline scenario, the separation takes place after JOI, therefore CLIPPER needs more fuel because it has to carry the 250 kg wet mass of CLEO-I.

If the separation takes place before, the fuel mass necessary for the same ΔV (that of the JOI) is lower. It can be used to increase CLEO-I maximum wet mass.

By applying the rule that CLIPPER mass after separation and after JOI is the same for both cases, the wet mass increase is about 90 kg (assuming that CLIPPER's main engine specific impulse is 290 s).

After separation, CLEO-I will have to retarget Jupiter's B-plane: from low-inclined GGA to mid-inclined low altitude JOI. The cost of the Orbit Deflection Manoeuvre (ODM) varies with the time from separation to Jupiter's arrival. A low cost (20-30 m/s) is obtained by separating 1 year before JOI.

5.4.2 CLEO-I S3

S3 is intended to reduce the ΔV needs compared to the baseline. This is obtained by replacing the PLM by fly-bys.

The separation takes place after the PRM. A similar sequence of GGA is then performed to reduce the orbital period (from 200 days to a few weeks) by pumping the infinite velocity, but also to reduce the inclination w.r.t. Jupiter's equator (to transfer to Callisto) by cranking down the infinite velocity.

When the 6:1 with Ganymede is reached, CLEO-I is transferred to Callisto: the infinite velocity is ~6.5 km/s. The perijove radius is then reduced by using a 4:3 and a 1:1 resonant orbits with Callisto. The infinite velocity at Io is 5.9 km/s and the orbital period is ~14 days, close to the 8:1 resonant with Io.

I1 would take place roughly at the beginning of COT-1, i.e. one year after JOI.

By imposing that I1 C/A is 500 km, only four options are possible as shown in Figure 5-23.

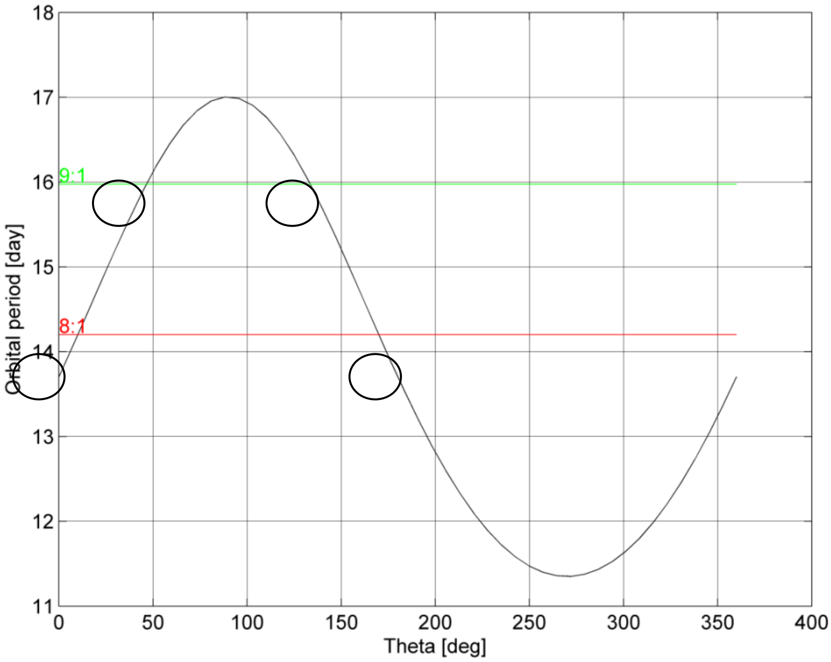


Figure 5-23: I1 parametric analysis. The orbital period after the fly-by vs Theta, the phase angle in the B-plane. Possible options are circled

Two of these options stay on the 8:1, i.e. only cranking, while the other two pump up to the 9:1 and crank up or down. The options with the 7:1 are not displayed because they incur a larger radiation dose (lower perijove).

The four options are shown in Figure 5-24. The same reasoning as for the baseline can be applied: choose the most promising groundtrack in terms of science and repeat the analysis for I2.

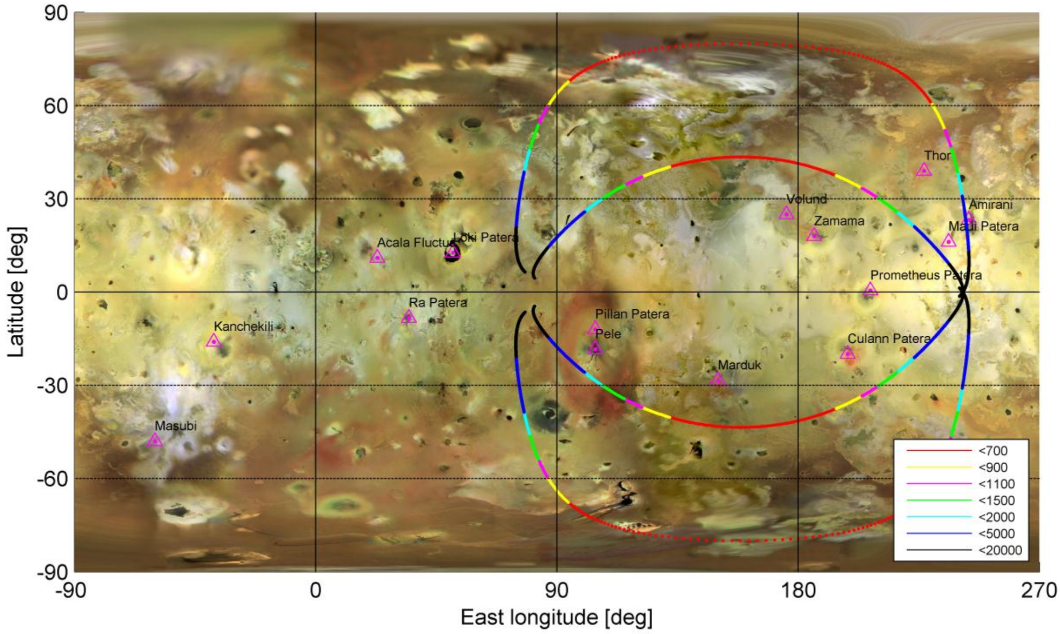


Figure 5-24: Possible groundtracks for I1 for S3

After I2, Callisto is used to pump up the perijove to place CLEO-I in a parking orbit with limited radiation dose accumulation. This parking orbit is then used to return the science data via DTE.

The evolution of the perijove during the mission is shown in Figure 5-25.

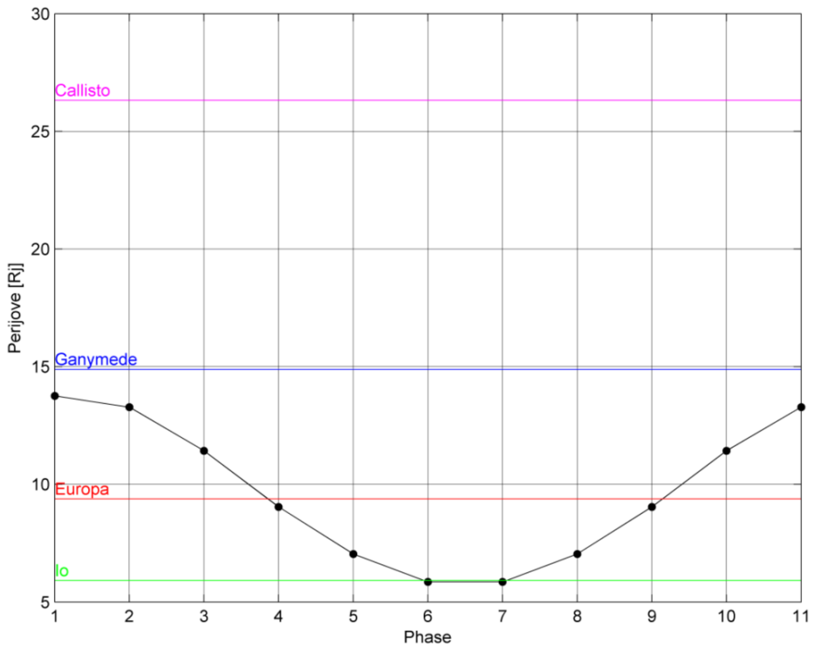


Figure 5-25: Perijove vs fly-by number for S3

The perijove is slowly reduced from Ganymede to Io via successive fly-bys. During this time, more radiation is accumulated compared to the baseline scenario. On the other hand, no PLM is needed, thus a lower ΔV.

The evolution of the apojove during the mission is shown in Figure 5-26.

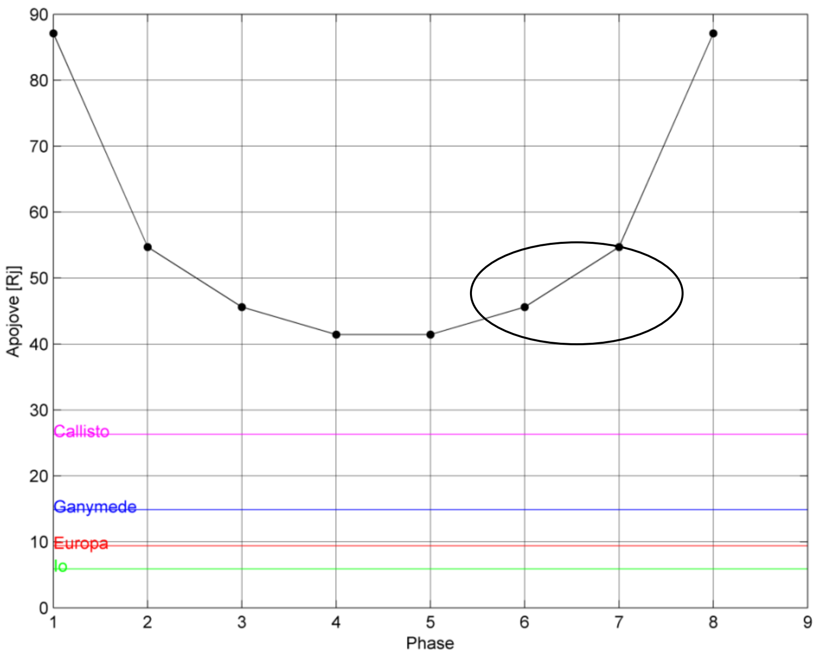


Figure 5-26: Apojove vs fy-by number for S3. Fly-bys #6 #7 could be used for relay via CLIPPER (circled)

The apojove in the parking orbit is of the same order as that of CLIPPER during COT. If the trajectory is designed such that the line of apsides of CLEO-I and CLIPPER are aligned, this would guarantee a short distance favourable for the relay (option of DTE for S3).

A drawback of S3 is the short amount of time from I1 to I2: 14.2 day. This is not sufficient to perform the data download, the battery recharge and the three TCM (measurements, OD, manoeuvre upload and realisation). Therefore it was decided to have only two TCM. Without being impossible to implement, it is more challenging than the baseline scenario.

5.4.3 CLEO-I S4

S4 is intended to minimise the ΔV with a single IGA. This is obtained by separating before the JOI (similar to S1), then retargeting the B-plane directly towards Io.

The B-plane retargeting is shown in Figure 5-27.

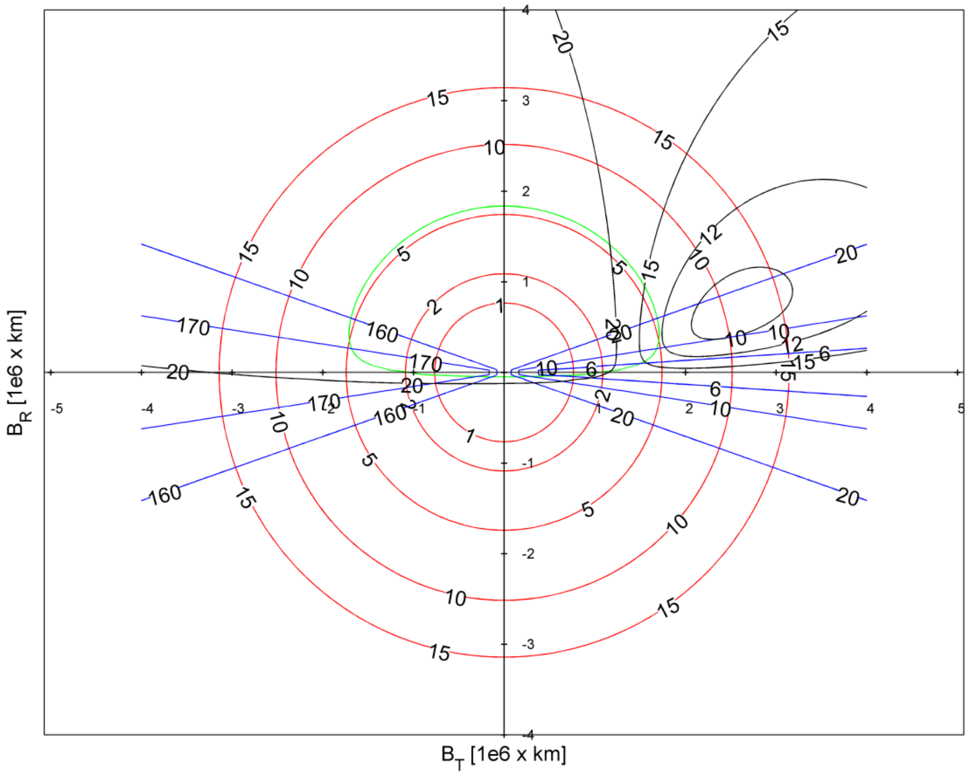


Figure 5-27: B-plane retargeting for S4. The red level lines show the perijove radius in Jovian radii. The blue level lines show the inclination w.r.t. Jupiter's equator in degrees. The black level lines show the difference between the velocity when crossing the Jupiter's equator and a fictitious moon having Io velocity in km/s. The green contour show the cases, where the radius when crossing the equator is equal to Io orbital radius

A parametric analysis can be done over the green contour. It gives the profile given in Figure 5-28.

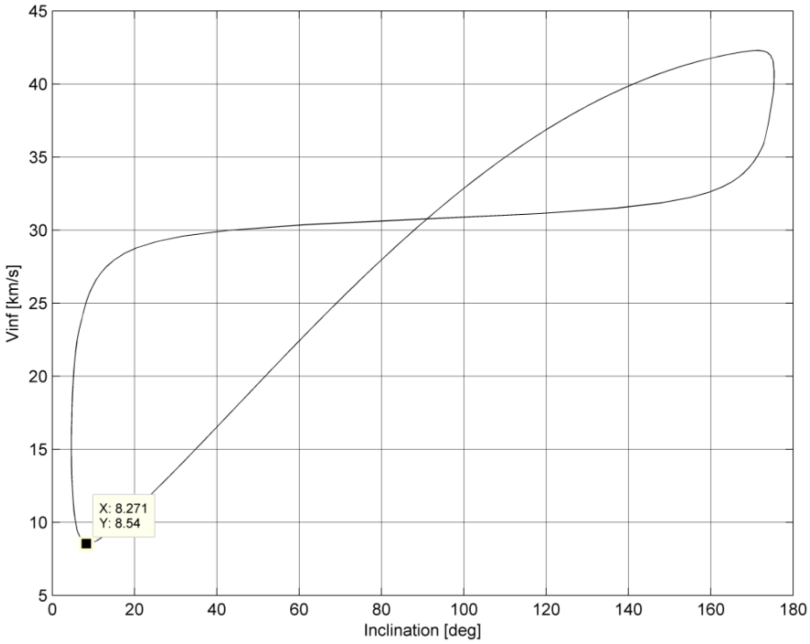


Figure 5-28: Infinite velocity vs inclination. This contour was obtained by scanning the green contour of the B-plane plot

In terms of payload, the minimal infinite velocity is sought. From the plot, it is 8.5 km/s (for an inclination of 8.3 deg). The corresponding pericentre velocity for a C/A @500 km is 8.8 km/s.

This point in the B-plane is:

- B_T = 3 100 000 km
- B_R = 150 000 km

CLIPPER's B-plane targeting is:

- B_T = 1 900 000 km
- B_R = 250 000 km

The estimated (linear) retargeting ΔV (ODM) after separation is 30 m/s one year before Jupiter's arrival.

This mission concept reduces significantly the ΔV for the mission but at the price of limited science return and higher risk (single Io flyby). Mission extension possibilities are almost non-existent and likely limited to a flyby of one asteroid of the main belt several years later.

5.4.4 CLEO-E

In its design, CLEO-E is very similar to CLEO-I S3: same separation, same energy and inclination reduction phase with GGA, same usage of Callisto to reduce the perijove down to Europa orbital radius.

Two options were analysed: one where the target orbit is 4:1 resonant with Europa, the other 6:1. In both cases, the perijove is close to Europa orbital radius to minimise the radiation dose.

For the 4:1, the infinite velocity is 4.1 km/s with an apojove of 38 R_J. For the 6:1 the infinite velocity is 4.7 km/s and the aopjove 53 R_J.

The 4:1 has the advantage to be phased with CLIPPER, which is good for relay (provided the trajectory design is made such that the line of apsides of CLEO-E and CLIPPER are roughly identical). However due to the short time between EGA, only 2 TCM are envisaged.

The 6:1 allow for three TCM. However CLEO-E has not the same orbital period as CLIPPER anymore: it is in a 3:2 resonant orbit with CLIPPER, thus with a 42 day cycle. The evolution of the distance between CLEO-E and CLIPPER was analysed over the cycle by varying the initial phasing. The optimal case, i.e. that minimising the distance, is shown in Figure 5-29.

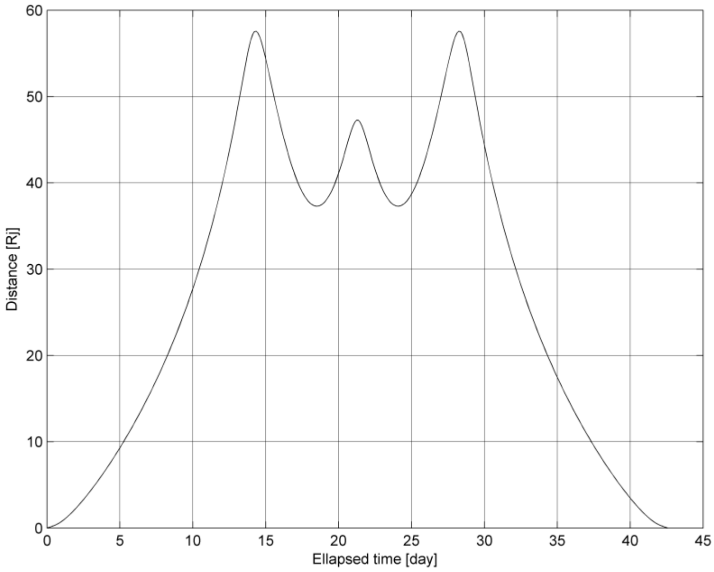


Figure 5-29: Distance from CLEO-E to CLIPPER as a function of time for the optimal phasing

The first EGA can be relayed immediately after the fly-by (day 0-5 to be below 10 R_J). The second EGA taking place at day 21.The third EGA can be relayed at the end of the cycle (day 37-42 to be below 10 R_J).

From a relay point of view, the 4:1 is favoured. In the end, this scenario, with a 4:1, essentially consists in sticking to CLIPPER. It is therefore recommended not to separate after PRM, but shortly before COT-1.

5.5 ΔV Budget

The ΔV budget for all cases is given in Table 5-4.

	CLEO/I				
Scenario	S1	S2b	S3	S4	CLEO/E
JOI (ODM for S4) [m/s]	470	0	0	30	0
PRM / PLM [m/s]	300	250	0	0	0
Nb fly-bys	2	4	15	1	10
Deterministic [m/s/flyby]	4	4	4	4	4
Deterministic [m/s]	8	16	60	4	40
JOI clean-up [m/s]	50	0	0	0	0
Stochastic [m/s/flyby]	10	10	10	10	10
Stochastic [m/s]	20	40	150	10	100
Margin 1st IGA [m/s]	10	10	10	0	0
Disposal [m/s]	15	15	15	0	15
Total [m/s]	873	331	235	44	155

Table 5-4: ΔV budget

Several remarks can be made relating to this table:

- All figures are given **without margin**.
- The deterministic cost per fly-by represents the B-plane retargeting from one fly-by to the next. It is assumed that the scenario is otherwise ballistic. The quoted value is based on experience with other missions
- The stochastic cost per fly-by represents the average sum of navigation manoeuvres (2 or 3 depending on the scenario), the CU being the dominant one.
- The special margin for I1 is intended to represent the higher cost due to initial uncertainties on Io ephemeris
- The disposal represents the extra cost after the science phase to e.g. retarget the spacecraft to an impact with Io or Jupiter
- The number of Gravity Assists for CLEO-E is 10. They are all EGA if a 4:1 is kept. Should another resonance be used, e.g. 6:1, only 4 EGA are assumed (separation after PRM + 6 GGA and CGA).

6 SYSTEMS

When referring to CLEO, reference is made to the S/C in general, whatever the concept. When referring to CLEO/I, CLEO/E or CLEO/P, reference is made to a mission concept in particular (respectively Io flyby concept, Europa flyby, Europa Penetrator).
Note: CLEO/P concept is described in a seperate Report.

6.1 Mission and System Requirements and Design Drivers

6.1.1 General

The following requirements are common to all mission concepts:

Mission & Systems Requirements	
Req ID	**Statement**
MI-GE-000	The CLEO S/C shall be carried as a piggy back on NASA Clipper S/C and released after Jovian Orbit Insertion *C: As per NASA/ESA initial discussions*
MI-GE-010	The CLEO mission design shall be compatible with following launcher environments : SLS, Atlas V 551 , DELTA IVH *C : SLS is the baseline launcher for Clipper while Atlas V and Delta IV are back-up solutions.*
MI-GE-020	The CLEO mission design shall be compatible with CLIPPER mission baseline and back-up mission profiles : - Closest point to the Sun : 0.65 AU - 7.2 years interplanetary transfer with up to 1 VGA and 3 EGA . *C : With Atlas or Delta transfer lasts up to 7.2 years EVEEGA for a launch in May 2022. 2.7 years with SLS for a direct transfer.*
MI-GE-030	The CLEO spacecraft shall not include any radioisotopic material for either the power generation or thermal control.
MI-GE-040:	The CLEO spacecraft shall accommodate, carry and operate the reference science payloads as described in the Payload Chapter.
MI-GE-050	The CLEO total mass shall not exceed 250kg including system margins and appropriate maturity margins *C: This includes any required separation mechanism or any shield required for thermal or radiation protection or any bio-barrier for planetary protection*
MI-GE-060:	The maximum volume allowable for CLEO in stowed configuration shall be less than 1 m × 1 m × 1 m (length/width/height) (TBC). *C: TBC by JPL. As a starting point, the volume allocated by Airbus for its carrier + penetrator concept has been considered (*Penetrator study – Airbus - datapackage*)*

MI-GE-070:	The CLEO mission shall consider as a reference scenario a launch in 2022 as per CLIPPER reference mission profile (Europa Clipper Science and Reconnaissance Payload proposal information package JPL D-92256 May 29, 2014) *C: Launch date programmatic feasibility is out of the scope of this study*
MI-GE-080:	The CLEO S/C should consider components qualified up to TID of 100 krad (TBC) and fluences up to TBD e-/m2 for the solar arrays *Rationale : space qualified equipments for higher radiation dose would require further development and qualification. It is preferred to use this qualification value and to add the required shielding to cope with the environment and mission profile.*
MI-GE-090:	The CLEO S/C shall conform to Category III Planetary Protection Requirements as per RD[33] for Europa concepts, and to Category I for Io concept. However Bio-burden requirements might be applied by CLIPPER, whatever the CLEO mission scenario. *C : CLIPPER is category III – RD[33]*
MI-GE-100:	The CLEO S/C shall have the capability to use CLIPPER as relay for uploading science data to Earth. *C: In order to maximise science data return.*
MI-GE-110	CLEO shall have Direct-to-Earth communications capability to be commandable from ESOC while allowing to retrieve the required housekeeping data and a minimum of science data to a level of TBD Gbit. *C:DTE is highly desirable if feasible to allow robustness and to allow for ESOC to control the S/C. As a minimum ESA ground station shall be able to send TC and retrieve HK data (Minimum required HK data volume TBC during the study) and a minimum of science data of TBD Gbit*
MI-GE-120	CLEO TT&C shall not foresee any data transfer to CLIPPER during Clipper flybys of EUROPA. *C: this could enter in conflict with Clipper own pointing requirements in such critical phases.*
MI-GE-130:	CLEO shall be designed with equipment compatible with TRL 5/6 by 2018. *C:Any deviation for this requirement can be discussed if deemed necessary. In the case of an opportunity mission, need date may be sooner*
MI-GE-140:	The Composite design shall comply to the margin philosophy described in RD[34]
MI-GE-150:	The composite shall be compatible with the Jupiter mission environment when applicable to CLEO *C : JUICE environment specification is the closest to CLEO at this stage RD[35]*
MI-GE-160:	Single-point failures shall be avoided in the CLEO spacecraft design. Retention of single-point failures in the design shall be declared with rationale and is subject to formal approval by ESA.
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.
MI-GE-180:	CLEO shall be able to perform the manoeuvres corresponding to the worst-case ΔV among the selected mission launch windows and trajectories, including contingencies
MI-GE-190	The mission shall be compatible with the science requirements defined in the Payload chapter

6.1.2 CLEO/I

MI-IO-010	The mission shall be able to perform at least one IO flyby at altitude of 100 km C : Minimal altitude to as per RD[36]
MI-IO-020	The mission shall perform at least 2 flybys of Io *C :The total number of flybys shall be maximised in order to maximise the science return. For minimal redundancy on science data return, the number of flybys shall be at least 2 [*Clipper ESA Contribution / Science Study Team Meeting – Minutes of Meeting – Ref ESA-SRE-F-ESTEC-MIN-2015-003 – 29/01/2015*]*
MI-IO-030	The mission should target at least 2 flybys at opposite high magnetic latitudes *C : This is required for magnetic measurements (for induction studies to test for magma ocean). This may have to be revisited if in conflict with other targets such as Volcanoes*
MI-IO-040	The CLEO/I pointing shall be such that : 1. MAC camera is oriented towards the target with an Absolute Pointing Error of less than 0.1 degree with a 95% confidence 2. Pointing of the S/C is maintained at better than 2 arcs over Ti, where Ti is the expected maximal integration time for the detector with a 95% confidence *C1 : 0.1 degree ensures the target is within the FoV (5 degree)* *C2 : 2 arcs stability ensures any S/C jitter limits the blur on camera measurement to ~ 0.1 pixel.*

6.1.3 CLEO/E

MI-EU-010	The mission shall be able to perform Europa close-by science measurements at flyby altitudes between 10 and 1000 km C : Minimal altitude to As per RD[37]
MI-EU-020	The mission shall perform at least 2 flybys of Europa *C :The total number of flybys shall be maximised in order to maximise the science return. For minimal redundancy on science data return, the number of flybys shall be at least 2*
MI-EU-030	The mission should allow for seeing Europa limb backlit by the Sun with phase angle > 150 deg. *C : This is required for UV limb measurements*

MI-EU-040	The mission should perform at least TBD flybys in the 10-500 km altitude range over the following regions of Europa : - South pole - large tidal stress (maximum at equator) - Large fissures and large scale lineaments *C : This is required for in-situ dust characterization*

6.1.4 Design Drivers

Design drivers are identified at subsystem level:

Domain of Expertise	Design drivers
Mission analysis	Minimise ΔV, maximise number of flybys, minimise radiation dose. Optimise flyby altitude wrt science objectives.
Propulsion	Minimise propulsion subsystem dry mass, minimise propellant mass
AOGNC	Comply with pointing requirements (SCI, DTE). Comply with 2 stabilisation strategies (Science 3-axis, JC spinning). Minimise mass and power (equip. selection, redundancy concepts). Highly autonomous safe mode.
Comms and DHS	Achieve DTE for TC and housekeeping data. Use of X-band for science data. Forbidden relay when Clipper is performing a flyby. Relay with Clipper analysed, DTE baselined
Power	Forbidden use of radio-isotopic power sources. High performance and light weight solar cells. Charging battery with low current. Minimise mass
Thermal	Consider Venus albedo for back-up. Minimise heating power required during non operational phases (e.g. Jovian Cruise)
Mechanisms	Minimise mass and complexity of separation mechanism. HDRM for Solar Panels. Boom design for magnetometer on CLEO options.
Structures and Configuration	Minimise mass, minimise volume, choose accommodation on Clipper, vault design for shielding.
Radiation	Design shielding concept (e.g. dedicated vault or equipment level shielding). Monitor TIDs of different equipment. Advise on configuration.
Cost	Mission of opportunity or potential M5 proposal. Take geo return constraints into account.
Programmatics	Align with project management timeline of Clipper. TRL 5/6 by 2018.
Risk	Provide risk register. Identify single point failures.
Planetary Protection	Align to NASA policy. Advise on how to minimize PP impact on CLEO/P design

Table 6-1: Design Drivers

COMPLIANCE MATRIX		
Req. ID	**STATEMENT**	**Req. Comments**
MI-GE-000	The CLEO S/C shall be carried as a **piggy back on NASA Clipper S/C** and released after **Jovian Orbit Insertion** C: As per NASA/ESA initial discussions	C – Baseline release after PRM
MI-GE-010	The CLEO mission design shall be compatible with following launcher environments : SLS, Atlas V 551, DELTA IVH SLS is the baseline launcher for Clipper while Atlas V and Delta IV are back-up solutions.	- under SLS Launcher environment assumed (unknown a.t.m.)
MI-GE-020	The CLEO mission design shall be compatible with CLIPPER mission baseline and back-up mission profiles : - Closest point to the Sun : 0.65 AU - 7.2 years interplanetary transfer with up to 1 VGA and 3 EGA . With Atlas or Delta transfer lasts up to 7.2 years EVEEGA for a launch in May 2022. 2.7 years with SLS for a direct transfer.	- Baseline - Back-up under assumption that CLEOP will not be exposed to direct sunlight for any significant duration inside of 1 AU. However, transient cases of up to 1 hour and albedo reflection from Venus should be considered (NASA answers)
MI-GE-030	The CLEO spacecraft shall not include any radioisotopic material for either the power generation or thermal control. C : NASA has removed such devices from CLIPPER baseline design.	C
MI-GE-040	The CLEO spacecraft shall accommodate, carry and operate the reference science payloads	C
MI-GE-050	The CLEO total mass shall not exceed 250kg including system margins and appropriate maturity margins This includes any required separation mechanism or any shield required for thermal or radiation protection	NC – Baseline Wet Mass incl. all margins (271.19 kg)
MI-GE-060	The maximum volume allowable for CLEO in stowed configuration shall be less than 1 m × 1 m × 1 m (length/width/height) (TBC). TBC by JPL. As a starting point, the volume allocated by Airbus for its carrier + penetrator concept has been considered	– triangular shape (base 1.2 m; height 0.8 m in radial direction)
MI-GE-070	The CLEO mission shall consider as a reference scenario a launch in 2022 as per CLIPPER reference mission profile	- under the AIT Approach assumed by Programmatics
MI-GE-080	The CLEO S/C should consider components qualified up to TID of 100 krad (TBC) and fluences up to TBD e-/m2 for the solar arrays	PC – components qualified for lower TID are shielded (assumed tolerances in BU slides)
MI-GE-100	CLEO shall have the capability to use CLIPPER as relay for uploading science data to Earth. C: In order to maximise science data return.	C – capability is guaranteed, but data volume is

		penalized wrt DTE
MI-GE-110	CLEO shall have Direct-to-Earth communications capability to be commandable from ESOC while allowing to retrieve the required housekeeping data and a minimum of science data to a level of TBD Gbit.	C – Total Data Volume in 2 fly-bys (baseline) is ~ 7.22 Gbit (SCI+HK TM)
MI-GE-120	CLEO TT&C shall not foresee any data transfer to CLIPPER during Clipper flybys of EUROPA. C: This could enter in conflict with Clipper own pointing requirements in such critical phases.	C
MI-IO-010	The mission shall be able to perform at least one IO flyby at altitude of 100 km	NC – IGA1 500 km (acceptable by science) C – IGA2 100 km
MI-IO-020	The mission shall perform at least 2 flybys of Io C : The total number of flybys shall be maximised in order to maximise the science return. For minimal redundancy on science data return, the number of flybys shall be at least 2	C – Baseline: 2 flybys
MI-IO-030	The mission should target at least 2 flybys at opposite high magnetic latitudes C : This is a goal to allow magnetic measurements (for induction studies to test for magma ocean)	C
MI-IO-040	The CLEO/I pointing shall be such that : 1. MAC camera is oriented towards the target with an Absolute Pointing Error of less than 0.1 degree with a 95% confidence 2. Pointing of the S/C is maintained at better than 2 arcs over Ti, where Ti is the expected maximal integration time for the detector with a 95% confidence (RPE) C1 : 0.1 degree ensures the target is within the FoV (5 degree) C2 : 2 arcs stability ensures any S/C jitter limits the blur on camera measurement to ~ 0.1 pixel.	C

Table 6-2: Compliance Matrix

6.2 System Assumptions and Trade-Offs

6.2.1 Assumptions

The following main assumptions have been adopted in the frame of the study:

- CLEO/I-E ejection after PRM
- Disposal ΔV allocation at end-of-mission of 15 m/s
- Distance to Clipper for relay link budget computation: 50 Rj (3m HGA on Clipper not pointed to CLEO/I-E, as discussed with NASA during dedicated TLC)
- Distance to Earth for link budget computations: 6 AU.
- Clipper unavailability for data relay:
    - at apojove
    - 3 days before flyby
    - 3 days after flyby
- SLS stiffness requirements: 60 Hz (unknown at the time of the study)

- Technology development of a low-mass integrated DHS-PCDU-AOGNC "MINIAvio", including:
  - Gyros (on a chip)
  - STR processing
  - PCDU functions
  - Instruments processing

  NOTE: 1 PCB = 18x23 cm inter-spaced by 2 cm

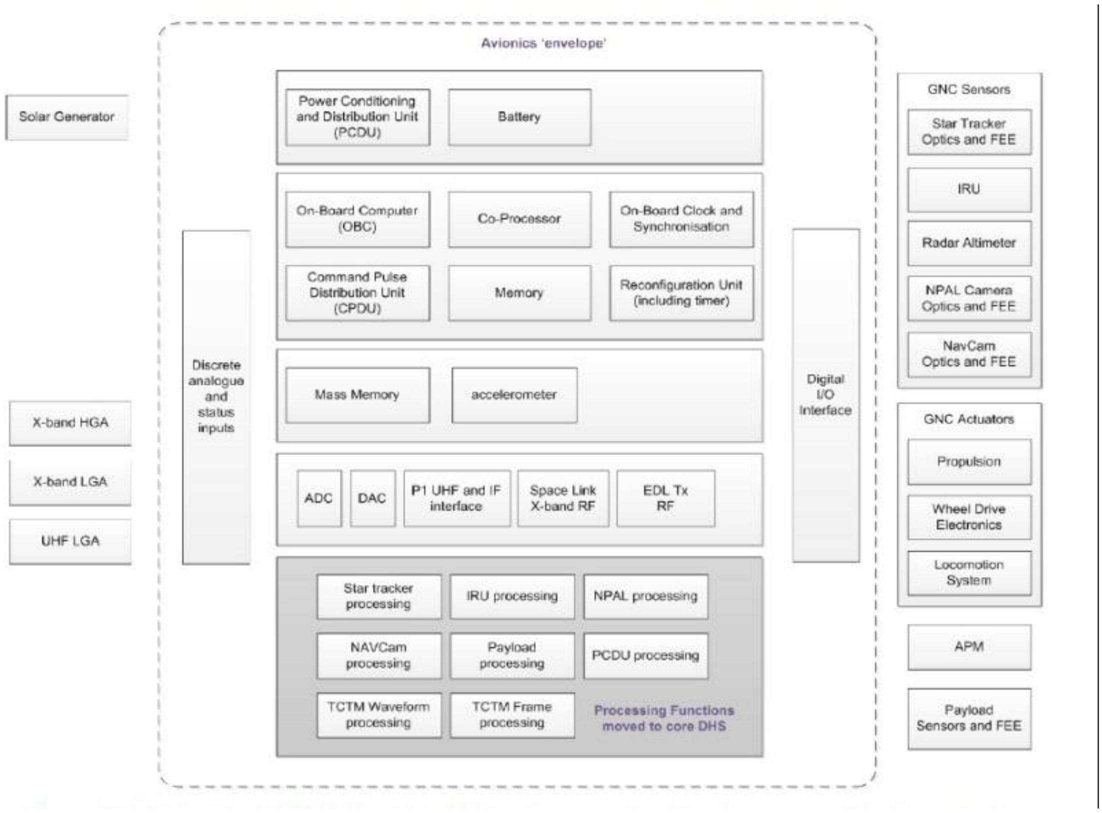


Figure 6-1: Integrated MINAVIO (Courtesy of RUAG Sweden)

- Units inside radiation shielding vault: Transponder, MINIAvio, EPC (for all other units radiation tolerance is retained good enough to withstand TIDs encountered during the mission.

6.2.2 Trade-Offs

The following table summarises the main trade-offs analysed during the CLEO/I-E study. Details are included in the relevant subsystem chapters, and are reported in this section in order to give an overview of the trade-space.

Subsystem	Options			
Mission Analysis	S1	S2a	S2b	S3
Comms Strategy	RELAY CLIPPER SCI DTE TM-HK		DTE SCI DTE TM-HK	
Propulsion	Monopropulsion	Bipropulsion	Green Propulsion	
AOGNC	Reaction Wheels		No Reaction Wheels	
	6 RCS		4 RCS	
Shielding Strategy/Conf/ Struct	SPOT	Shielding Structure	Mini Vaults	Single Vault
Miniaturization Integration	Avionics (OBC, AOCS Function, PCDU Functions) Metal based antenna (lower mass and volume)			

Table 6-3: Summary of main System trade-offs

6.2.2.1 Shielding Strategy

The selection of the shielding strategy was the outcome of a system-level trade-off, involving:

- All subsystems responsible for the equipment with high sensitivity to radiation
- Configuration advising on the most appropriate units accommodation in order to minimise the radiation shielding mass ("smart box positioning" indicates that 2 boxes share a common face, leading to the suppression of some shielding)
- Structures investigating on the possibility to give to the primary structure a shielding function (increasing Al in some areas of the primary structure, and accommodating sensitive units in correspondence of those areas)
- Radiation specialist specifying the Al shielding thickness required to guarantee an environment compatible with the units tolerance levels.

The following table shows 5 explored concepts:

1) Optimised structure: primary structure does not have shielding function, sensitive equipment shall be individually shielded
2) Al shielding columns: 10 mm Al columns are designed to host attachment points for the most sensitive equipment, to be individually shielded
3) Smart Box Positioning: 10 mm Al columns are designed to host attachment points for the most sensitive equipment. Boxes are placed close to each other, sharing as much as possible their surface in order to reduce shielding mass
4) Confined Vault: the upper (or bottom) part of the primary structure hosts an Al vault meant to contain sensitive equipment
5) Mini Vaults: primary structure does not have shielding function, sensitive equipment are shielded multiple vaults, with the number to be defined based on accommodation constraints.

Thermal considerations played a significant role in the selection of the best shielding strategy, as at 6 AU. the required heating power for the units shall be minimised placing them as close as possible to each other (benefiting from dissipation effects).

The idea of the vault appeared therefore the most attractive, however due to accommodation and volume constraints a single vault could not be baselined.

2 Mini Vaults have been placed into the CLEO/I spacecraft.

An advantage of having multiple vaults is the possibility to adapt the thickness to the sensitivity of the component included, avoiding mass waste.

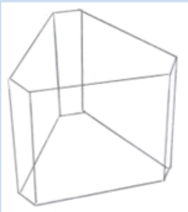
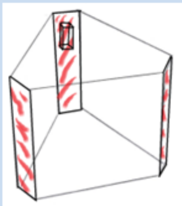
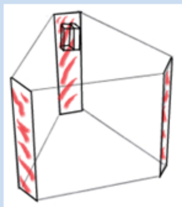
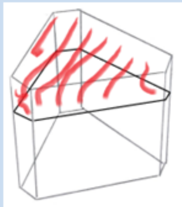
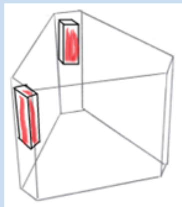
OPTIMISED STRUCTURE	Al SHIELDING COLUMNS	SMART BOX POSITIONING	CONFINED VAULT	MINI VAULTS
				
STRUCTURE MASS: 26.5 kg	STRUCTURE MASS: 33 kg Col 10 mm Al	STRUCTURE MASS: 33 kg Col 10 mm Al	STRUCTURE MASS: 25 kg (NC)	STRUCTURE MASS: 30 kg
SHIELDING MASS(12 mm AL): 74 kg (spot shielding): 42.6 kg	SHIELDING MASS (12 mm AL): 61 kg	SHIELDING MASS (12 mm AL): 58 kg	SHIELDING MASS (12 mm AL): 60 kg	SHIELDING MASS (12 mm AL): 73.5 kg
TOTAL: 72.6 kg	TOTAL: 94 kg	TOTAL: 91 kg	TOTAL: 85 kg	TOTAL: 103.5 kg

Table 6-4: Shielding Strategy Trade-Off

6.2.2.2 Miniaturisation and Integration

In the continuous attempt to save mass, a lot of effort has been dedicated to investigating the possibility for miniaturisation and integration, also capitalising from past CDF studies.

The outcome of this effort was:

- The selection of a holographic antenna
- The integration of OBC, AOCS (Gyros and STR), PCDU and Instrument processing functions into a low mass integrated Mini Avio

The table summarises the resolution of the trade-offs, with the CLEO/I-E baseline selection.

Subsystem	Options			
Mission Analysis	S1	S2a	S2b	S3
Comms Strategy	RELAY CLIPPER SCI DTE TM-HK		DTE SCI DTE TM-HK	
Propulsion	Monopropulsion	Bipropulsion	Green Propulsion	
AOGNC	Reaction Wheels		No Reaction Wheels	
	6 RCS		4 RCS	
Shielding Strategy/Conf/ Struct	SPOT	Shielding Structure	Mini Vaults	Single Vault
Miniaturization Integration	Avionics (OBC, AOCS Function, PCDU Functions) Metal based antenna (lower mass and volume)			

Table 6-5: Trade-off results

6.3 Mission System Architecture

6.3.1 Concept of Operations

The Io Flyby science lasts about 5 hours. The rest of the orbit is shared between sending back data to Earth and cruising. By designing a low power Jovian Cruise Mode (spinned and earth-pointed with only critical subsystems on) the Solar Array size can be minimised, while the S/C relies on battery for all other modes (DTE, SCI). The battery is sized by the Flyby Science phase.

This concept resulted in the repetition of cycles of 2.7 hours Comms sessions separated by 28 hours battery re-charging in Jovian cruise. To this pattern must be added the trim correction manoeuvres (Clean-Up, Correction Manoeuvre, Re-targeting)_

Orbital period needs to be long enough to allow for sufficient cycles to send back science data to Earth. In the baseline strategy, there are 100 days between the first and the second Io flybys and 190 days after the 2nd flyby.

6.3.2 Mission Timeline

A mission timeline has been defined combining the Modes described in 0.

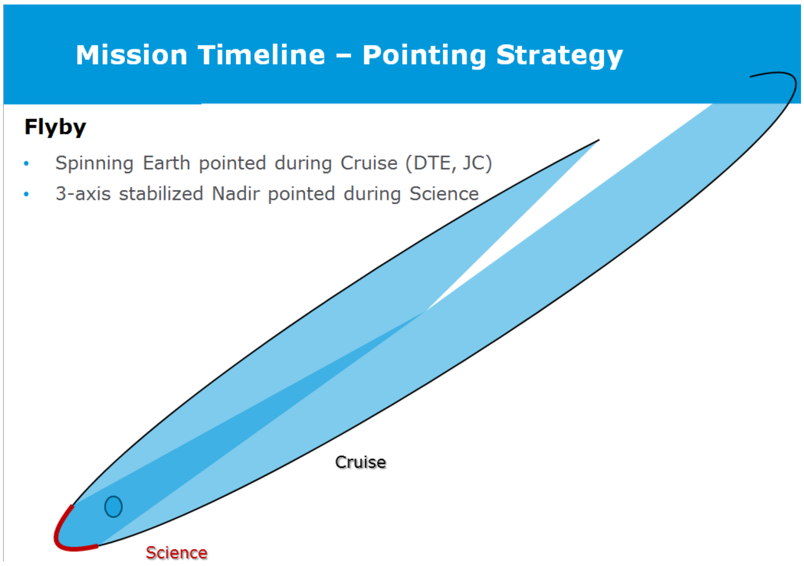


Figure 6-2: Mission timeline - pointing strategy

CLEO/I baseline foresees 2 flybys around Io, Science (red portion of the trajectory in the picture above) is performed during the time of the closest approach to the moon when the spacecraft is 3-axis stabilised and nadir pointed. A Jovian cruise follows the flyby, when the spacecraft is spinning and Earth pointed, with a maximum offset from the Sun of 11 degrees. Jovian Cruise (JC) Mode and Direct To Earth (DTE) communication Mode happen during this phase.

Mission Analysis has identified a trajectory with 100 days of Jovian cruise following the 1st flyby (at 500 km from Io surface), and 190 days following the 2nd flyby (at 100 km from Io surface).

The figures below illustrate the sequence of phases, and relative durations, occurring at each flyby.

6.3.3 Mission Timeline Assumptions

The CLEO/I mission timeline is valid under the following assumptions:

- **Orbit Determination (OD) DTE Sessions**: 2 OD sessions are assumed to be performed before flyby (for the FB targeting), and 2 OD sessions after flyby (FB Clean-Up), with a duration of 4 hours each *(this implies the capability to run up Orbit Determination before each manoeuvre, e.g. 2 times 4 hours range/Doppler measurements as baseline for this study ; or more frequent DDOR measurements with shorter duration. TBC by flight dynamic analysis out of scope in the CDF study). At the end of the second OD session it is assumed that the MAN command is uploaded.*
- **GA Related Manoeuvre** (targeting, clean up): 2 time slots of 3 hours duration each have been allocated in the timeline for targeting maneuver before the flyby, and clean-up manoeuvre after the flyby.
- **Potential Apojove Manoeuvre** This has not been calculated or taken into account during the CDF study and will need to be calculated (update mission timeline and data volume estimation) in a later phase.

Orbit Determination and Manoeuvres before and after flybys add up to ~9-10 days *not available for SCIENCE data download.*

- **Science**: assumed to last 300 minutes (based on camera parameters and spacecraft infinity velocity). CLEO/I battery is sized to cope with this phase (it is assumed that no illumination would come from SA, which is quite pessimistic).
- **Jovian Cruise (JC) is a Mode used for Recharging**: 28 hours (with 10W SA allocation) are needed to recharge CLEO/I battery, while only essential equipment are kept active, and most of the platform is switched off for optimisation purposes (Receivers are assumed to be OFF in JC for Comms, one would be awakened by timer)
- **Direct To Earth (DTE) communication**: duration is computed as 2.7 hours, relying on the battery sized for Science and on SA sized to recharge such battery during JC Mode

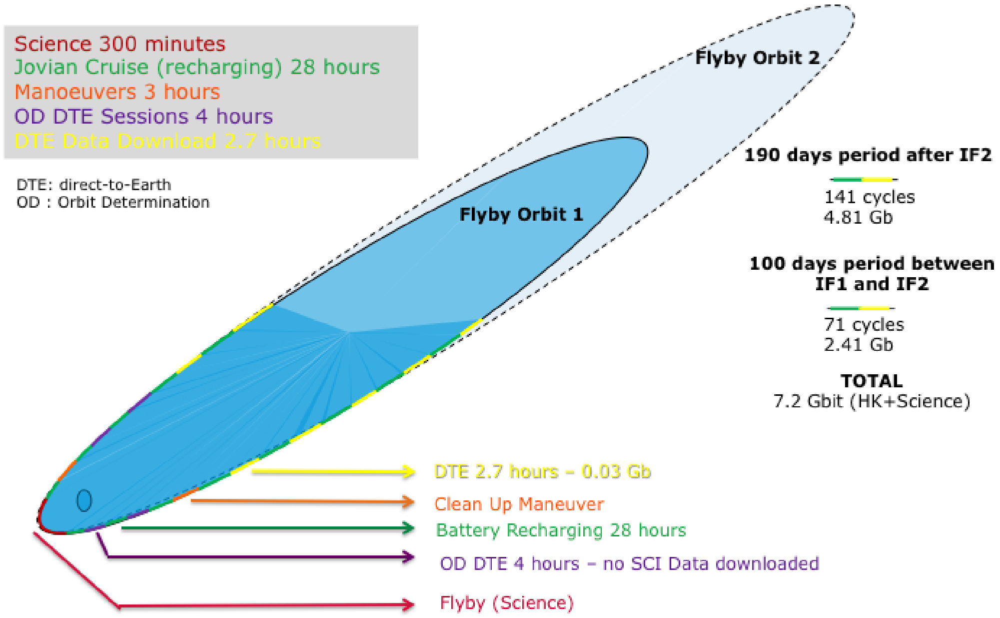


Figure 6-3: Mission timelines for flybys

6.3.4 Data Volume Download Capability (SCI+HK TM)

With the timeline described above, the Data Volume download capability (to be shared between science data and house-keeping telemetry) has been computed for both flybys, from the

- 100 days post FB1: 71 cycles DTE/JC following 1st fly-by→ 2.41 Gb Data Volume downloadable
- 190 days post FB2: 141 cycles DTE/JC following 2nd fly-by → 4.81 Gb Data Volume downloadable (7.22 Gbit total data volume).

The following Options have been also investigated during the study:

- **Longer JC Mode** (allowing for a smaller battery recharging time) to reduce SA area

  A "mathematical" case based on allocating 0 W for Battery Recharge (instead of the baseline: 10 W) has been run:

  - SA area required would be: 3.7 m², implying a mass of 16.4 kg (Baseline: 6 m²; 25.3 kg)
  - DTE duration would be 2.5 hours (Baseline: 2.7 hours)

- The system would not have any Recharging Capability (Baseline: 28 hours to recharge battery), as 0W are allocated for battery recharge.

10W charging power allows to recharge the battery in 28 hours, which means roughly that 1 DTE opportunity is possible per day. This was found to be an acceptable compromise from an operational point of view. Any solar array area reduction would allow to reduce slightly the dry mass but at the price of additional charging time meaning less DTE communications slots, and therefore less science data return. This case gave a clear indication for the fact that the system is very much optimised around a "low duty cycle" concept, and that SA are very close to the limit in terms of area and mass.

- **Low Power DTE Mode**

The possibility to implement a "low power DTE Mode" was also explored, splitting the 2.7 hours of DTE that the system is able to cope with in two chunks:

- 2 hours of full DTE (Tx + Rx + TWT on)
- 5 hours of low power DTE (Tx + Rx on), to be used for navigation only.

This mode may be beneficial because every time the S/C goes into DTE mode the orbit is reassessed. Another possibility is to have frequent short DTE windows just for DDOR and dedicated DTE's for data downlink.

Data Volume Download Capability (SCI + HK TM) would be negatively impacted by this split, as only 2 hours instead of 2.7 would be used to download Data to Earth. The capability would in fact be as follows:

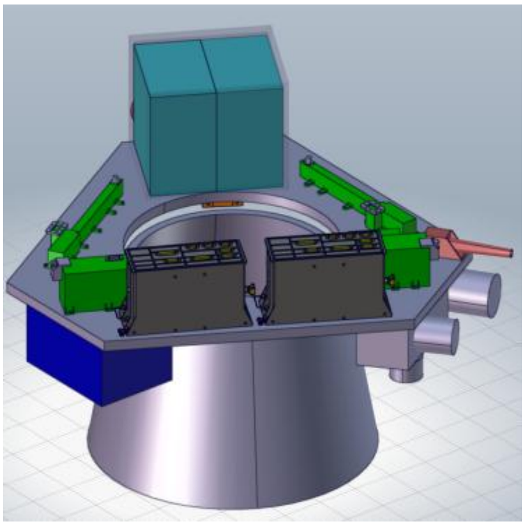
- 100 days post FB1: 62 cycles DTE/JC following 1st fly-by→ 1.57 Gb Data Volume downloadable
- 190 days post FB2: 124 cycles DTE/JC following 2nd fly-by → 3.12 Gb Data Volume downloadable (4.69 Gb total).

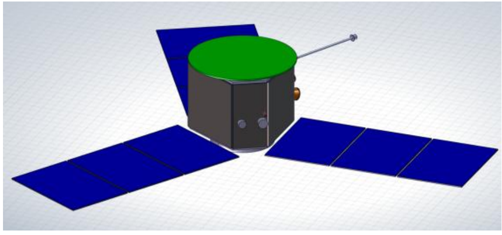
6.4 System Baseline Design

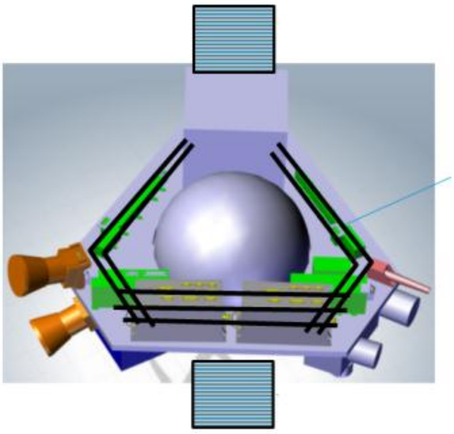
The characteristics of the baseline design are the following:

CLIPPER European Orbiter Io	
Launch Date	May/June 2022
Launcher	Nominal: SLS direct to Jupiter (June 2022)
	Backup: SLS direct to Jupiter (June 2023)
	Alternate: Atlas V 551 EVEEGA (May 2022)
	Alternate backup: Atlas V 551 VEEGA (June 2023)
Transfer time	2.7 years (Nominal), 7.2 years (Alternate)
Release from Clipper	After JOI, after PRM
From JOI to IGA1	1.5 year
Nr Flybys	2
Flyby 1	Period: 100 days (from Flyby 1 to Flyby 2)

parameters	Near-equatorial (0.8 deg to Jupiter equator)
	Vinf 7 km/s
	Perijove: 5.9 Rj (~= Io orbital radius)
	Apojove: 160 Rj
	IGA C/A: 500 km Northern Hemisphere
Flyby 2 Parameters	Period: 190 days (from Flyby 2 to next flyby or impact)
	Near-equatorial (0.2 deg to Jupiter equator)
	Vinf 7 km/s
	Perijove: 5.8 Rj (~= Io orbital radius)
	Apojove: 260 Rj
	IGA C/A: 100 km Southern Hemisphere
Δ V	345.55 m/s (including margins)
Payload	Camera, Mag, MidIR, INMS; 14.82 kg, 51.6 W pPwr
Science Duration	300 mins per flyby (*note : Flux gate magnetometer is ON all along the orbit, in low resolution*)
Data Volume	7.22 Gb (2.14 Gb + 4.81 Gb) (to be shared between SCI & HK TM)
Mass	Dry mass (227.32 kg) (incl DMM)
	Propellant (39.93 kg) (incl 2% margin)
	Wet (266.75 kg) (incl 20% system margin)
Dimensions	Stowed: 1.2x1.2x0.8 triangular shape
Structure	CFRP
Shielding	Shielding Mass: 19.06 kg (5 mm Al Vaults + 10 mm MINIAvio + 3.5 kg Instruments)
Mechanisms	Separation: Clamp band; SA hinges
AOGNC	Cruise: Spin; Science: 3-axis stab with RCT. 2 GYROS, 2 STR, 8 SS;
Propulsion	Monoprop System; 1 tank, 1x22N thruster(6Nom+6R)x1N RCT
Power	6 m2 SA; MPPT; 4.9 kg Battery (690 Wh); Unregulated Bus
Communication	X-Band HGA 1.1 m (tx) – 0.6 m (Rx); 2 LGA, RF pwr 65W; TM rate 3.5kbps; TC rate 1kbps (35 m GS)







DHS	MINIAvio (OBC + PCDU + STR processing+ Gyros + Instrument processing)	
Thermal	Ext. MLI, Int. MLI, Instruments MLI, prop. MLI; 0.15 m2 2xLouvers; 6m heat-pipes; heaters; sensors. Propulsion heating power 25 W; platform Heating Power JC 5 W	

Table 6-6: CLEO-I baseline design

6.4.1 Model Decomposition

The CLEO-I model has been decomposed into 2 first-tier products:

- The payload, containing the 4 instruments (and their shielding) with 18.33 kg (including DMM), and,
- The platform, containing all other equipment from all the other domains of expertise with 154.86 kg (including DMM).

6.4.2 Mass Budget

Domain	Mass (kg)	Margin (kg)	Mass Margin (%)	Mass (incl. DMM) (kg)
AOGNC	3.24	0.18	5.46	3.42
COM	22.20	2.55	11.49	24.75
CPROP	18.17	1.16	6.38	19.33
DH	4.50	0.90	20.00	5.40
INS*	12.35	2.47	20.00	14.82
MEC	11.07	1.11	10.00	12.18
PWR	34.69	6.94	20.00	41.63
RAD	19.06	0.00	0.00	19.06
STR	23.96	4.79	20.00	28.75
SYE	0.00	0.00	0.00	0.00
TC	9.71	0.97	10.00	10.68
Grand Total	158.95	21.06	13.25	180.01

Note*: Mass is without Back end electronics and shielding covered in DH and RAD respectively

Harness (%)	5
Harness (kg)	9.00
Total dry mass without margin (kg)	189.01
System margin (%)	20.00
Total dry mass (kg)	**226.81**
Total dry mass (for propellant calculation - excl. SDM Clipper) (kg)	221.14
Propellant mass (kg)	39.15
Propellant mass margin (2%) (kg)	0.78300
Total wet mass (kg)	266.75

Table 6-7: CLEO-I mass budget aggregated by domain of expertise

6.4.3 List of Equipment

	#	Mass (kg)	Mass Margin (%)	Mass + Margin (kg)
AOGNC		**3.24**	**5.46**	**3.42**
GYRO_Chip (GYRO on Chip MINAVIO)	2	0.05	20.00	0.06
STR_HydraOH (STR Sodern Hydra Optical Head)	2	1.37	5.00	1.44
SUN_MoogBrad_mFSS (SUN Moog Bradford Mini Fine Sun Sensor)	8	0.05	5.00	0.05
COM		**22.20**	**11.49**	**24.75**
EPC (Electronic Power Conditioning)	2	1.40	5.00	1.47
HGA (High Gain Antenna)	1	5.00	20.00	6.00
LGA (Low Gain Antenna)	2	0.30	10.00	0.33
RFDU (Radio Frequency Distribution Unit)	1	5.00	20.00	6.00
TRASP_Tx_MOD_Rx_DED (Transponder)	2	3.50	5.00	3.68
TWT (Traveling Wave Tube)	2	0.90	0.00	0.90
CPROP		**18.17**	**6.38**	**19.33**
FDV_Fuel (Fill Drain valve Fuel)	1	0.07	5.00	0.07
FDV_Pressurant (Fill Drain valve Pressurant)	1	0.05	5.00	0.05
FL (Feed line)	1	5.00	10.00	5.50
LV (Latch Valve)	3	0.55	5.00	0.58
NC_Pyro_Valve (NC Pyro Valve)	2	0.29	5.00	0.30
PF (Propellant Filter)	1	0.11	5.00	0.12
PropTank (Propellant Tank)	1	6.01	5.00	6.31
PRT (Pressure Transducer)	3	0.25	5.00	0.26
Thruster_AOCS (Small Thruster)	12	0.30	5.00	0.31
Thruster_LAE (Large Thruster)	1	0.40	5.00	0.41
DH		**4.50**	**20.00**	**5.40**
MINAVIO (Miniaturized Avionics)	1	4.50	20.00	5.40
INS		**12.35**	**20.00**	**14.82**
CamI (Camera)	1	2.75	20.00	3.30
MagI (Magnetometer)	1	0.85	20.00	1.02
MidIR_I (MidIR)	1	5.75	20.00	6.90
NIMS_I (Neutral/Ion spec)	1	3.00	20.00	3.60
MEC		**11.07**	**10.00**	**12.18**
HDRM (Solar Array HDRM)	9	0.35	10.00	0.39
SA_DH (SA Deployment Hinge)	18	0.09	10.00	0.10
SDM (Satellite Deployment Mechanism)	1	2.00	10.00	2.20
SDM_Clipper (Satellite Deployment Mechanism Clipper)	1	4.30	10.00	4.73
PWR		**34.69**	**20.00**	**41.63**
Bat (Battery_general)	1	4.90	20.00	5.88
MINAVIO (Miniaturized Avionics – PCDU boards)	1	4.50	20.00	5.40
SA (SolarArray)	3	8.43	20.00	10.12
RAD		**19.06**	**0.00**	**19.06**
Shield_CamI (Shielding Camera)	1	1.00	0.00	1.00

Shield_MagI (Shielding Magnetometer)	1	0.50	0.00	0.50
Shield_MidIR_I (Shielding MidIR)	1	1.01	0.00	1.01
Shield_MINAVIO (Shielding Miniaturized Avionics)	1	9.55	0.00	9.55
Shield_NIMS_I (Shielding Neutral/Ion spe)	1	1.00	0.00	1.00
Shield_TRASP_Tx (Shielding Transponder)	2	3.00	0.00	3.00
STR		**23.96**	**20.00**	**28.75**
Col (CLEO-I Columns)	1	1.03	20.00	1.24
Floor (CLEO-I Floor)	1	9.27	20.00	11.12
Floor_Rein (CLEO-I Floor Reinforcement)	1	0.85	20.00	1.02
Int_Adap (CLEO-I Interface Adapter)	1	5.26	20.00	6.31
Int_Floor (CLEO-I Intermediate Floor)	1	1.52	20.00	1.82
Lat_Pan (CLEO-I Lateral Panels)	1	2.07	20.00	2.48
Sun_Floor (CLEO-I Sun Floor)	1	2.07	20.00	2.48
Tank_Cone (CLEO-I Tank Cone)	1	1.89	20.00	2.27
TC		**9.71**	**10.00**	**10.60**
LVR (Louvre)	2	0.78	10.00	1.738
MLI (MLI)	1	5.85	10.00	6.435
Heat_P (Heat Pipes)	1	1.8	10.00	1.98
Misc (Miscellaneous)	1	0.5	10.00	0.55
Grand Total		**158.95**	**21.06**	**180.01**

Table 6-8: CLEO-I list of equipment

Comments:

- The separation mechanism is divided into two parts. One that is carried with CLEO-I (2.2 kg incl. DMM), and one that remains in CLIPPER (4.73 kg incl. DMM).
- There are a total of 12 1N Thrusters (6 nominal and 6 redundant)
- The allocation of the system harness (5%) is done afterwards and can be found in Table 6-7.

6.4.4 System Modes

During the sessions the following system modes were identified:

MODE	DESCRIPTION
Launch	S/C in Launch Configuration
Dormant Transfer	S/C attached to CLIPPER, dormant during transfer, HK checks possible
Commissioning	S/C performing commissioning activities of any kind, including instrument commissioning
Science	S/C during fly-by, full science capability (300 minutes)
Jovian Cruise	S/C in orbit around Jupiter, not in fly-by distance, reduced science possible
Eclipse	S/C in eclipse induced by Jupiter, Io, Europa, Ganymede or Callisto
DTE Comms	DTE communication for TM/TC and science data retrieval
Relay Comms with CLIPPER	Science data upload to CLIPPER (not baselined for CLEO-I)
Manoeuvre	S/C manoeuvering (e.g. orbit insertion, orbit maintenance, disposal), thrusters firing
Safe	For CLEO, no safe mode as such but safety is ensured by automatic contingency mode transitions between JC and DTE Comms sessions based on e.g. battery charging state monitoring

Table 6-9: System Modes

The table below identifies which equipment are switched on/off for each of the modes (red=off; yellow=on; green=on, highest values).

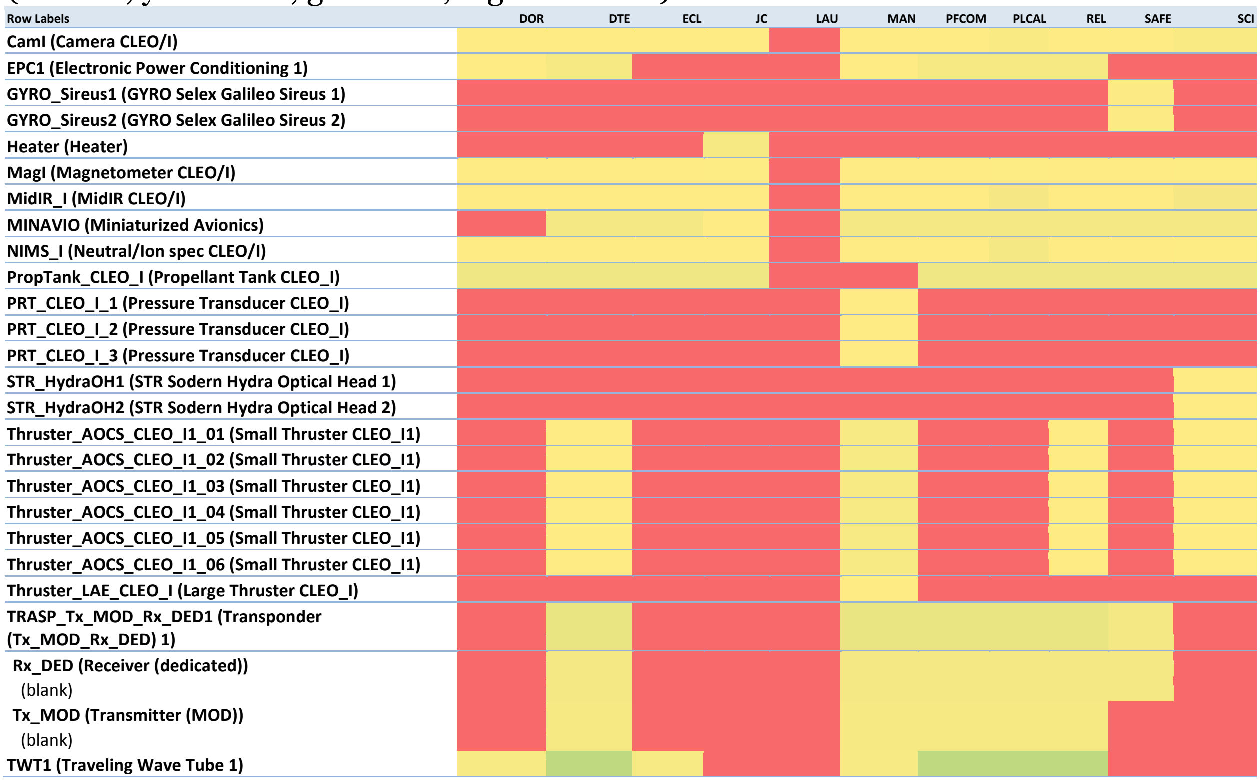


Table 6-10: Equipment switching per mode

It is likely that the Dormant Transfer and Commissioning modes will impact the CLIPPER design as the S/C will need the host to support it with power (in particular for propellant heating), and data transfer capabilities.

During nominal operation (in a flyby), CLEO-I will be switching between the "DTE", "Jovian Cruise" and "Science" modes using the strategy described in the earlier subchapters.

The "Jovian Cruise" mode, which can be described as an Earth pointing pseudo-hibernation mode during a flyby, drives the size of the solar arrays. Power consumption is minimised in this mode to minimise the battery charging time.

The "Science" mode refers to the part of the flyby closer to Io on which the instruments are in full operation. This mode drives the battery choice. The details of this sizing can be found in the power chapter.

The duration of the "DTE" mode is driven by the maximum energy storage of the battery (which is sized for the "Science" mode). The amount of science data that can be transferred is, therefore, driven by the maximum energy storage of the battery.

The duration of the "Eclipse" mode was assumed to be 3h worst case (Chapter 5.3.5.2) and not driving the system design.

The "Relay Comms with Clipper" mode was not taken as the baseline for communications because the large distance between CLEO-I and CLIPPER only allowed a small data rate in comparison to DTE communication (Chapter 16.4.1).

6.4.5 Safe Mode

The ambitious design approach which was taken to minimise the required resources (mass and power) requires a highly autonomous Safe mode compared to usual missions.

In fact, the power generation by the solar arrays is only sufficient to fill the battery during the Jovian Cruise mode. In all other modes, including the communication mode (DTE mode) the battery is depleting. A "standard" safe mode where S/C would be Sun pointed while continuously communicating with Earth would deplete the battery in a few hours and is not feasible with this approach, unless the solar panels are sized only for the safe mode, with a huge mass penalty.

During nominal operation, CLEO-I will be switching between the Science mode (SCI), Jovian Cruise mode (JC) and DTE communication mode (DTE).

To cope with the limited power available, a highly autonomous safe mode has to be defined, which is able to autonomously manage the transitions from JC to DTE as soon as the battery charging state allows, and switch back to JC as soon as the charging state goes below a predefined threshold. The current design assumes that the S/C always knows where Earth is through regular updates by ground. This allows the S/C to point to Earth with a coarse accuracy (1 degree) with its Sun Sensors and gyros (see chapter 13).

If there is a failure during the SCI Mode the S/C shall:

1. Achieve Sun pointing,
2. Spin the S/C to perform a scanning pattern (strobing) to restore Earth pointing to better than 2 degree accuracy, and point to Earth (with max SAA below 11deg).

3. Transition to DTE Mode– the battery is sized to allow for TBD minutes of DTE in the worst case situation where transition to safe mode would occur at the very end of the SCI mode.
4. At batteries depletion S/C go to JC mode (nominal Operation)

If there is a failure during JC mode the S/C shall:

1. Transition to DTE Mode. Nominal pointing to Earth; max Sun deviation ~11deg (no need to slew)
2. After HK TM to ground and at batteries depletion, go back to JC mode

If there is a failure during DTE mode, the S/C shall:

1. Use the OBC to go to JC mode to charge the batteries.
2. Transition to S/C in spin mode. Nominal pointing to Earth; max Sun deviation ~11deg (no slews).
3. At battery full switch back to DTE mode (HK TM to Ground)

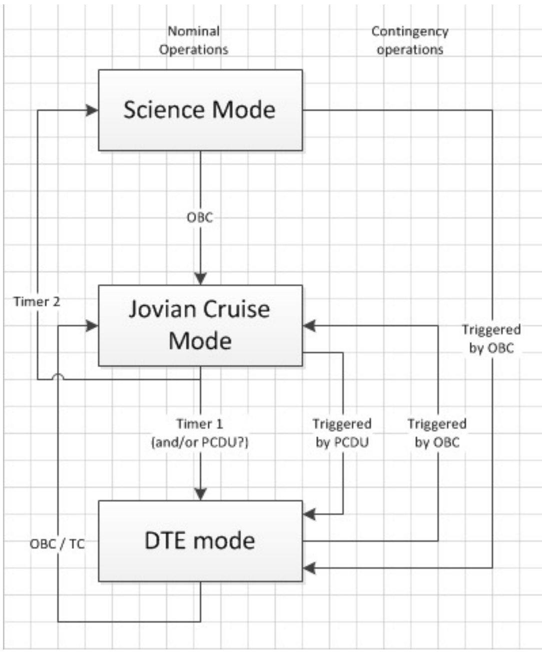


Figure 6-4: CLEO-I contingency strategy

Note: Although it has not been investigated in the CDF, a Sun Acquisition mode might be also required to improve the robustness of the design (case where context is lost, including Earth position)– communications with Earth would be very limited in this mode, but possible with its LGA with a very limited data rate (7.8 bps TC, 10 bps TM) and assuming the use of NASA's 70m Deep Space Antennas. The S/C would then need to scan for Earth (strobing) to acquire Earth and enter JC mode. The proposed baseline design could accommodate such a mode without additional sensors / actuators.

6.4.6 Power Budget

Details in Chapter 14.

6.4.7 Radiation Shielding Mass

Ray tracing analysis produced the following TID estimate for the Mission Analysis baseline scenario.

Values are reported, including a factor 2 according to the applied margin policy, for the most sensitive components, and confronted with their assumed sensitivity:

	S2b		Factor 2 Applied		TID Tolerance
	5mm	10 mm	5 mm	10 mm	
Transponder (Vac)	55	38	110	76	50
Transponder (Full)	22	16	44	32	50
EPC (Vac)	42	28	84	56	50
EPC (Full)	19	13	38	26	50
MiniAvionic (Vac)	52	36	104	72	50
MiniAvionic (Full)	2	2	4	4	50

Table 6-11: TID Estimates

TIDs are expected to be closer to "Full" case values rather than to "Vacuum" case values, therefore 5 mm Al shielding vaults have been selected as CLEO/I baseline.

Moreover, radiation analysis is considered quite conservative both due to high uncertainties in the radiation environment at Io and for the applied uncertainty margin (factor 2).

As a consequence, potentially more flybys could be performed for the allocated radiation shielding mass.

Table 6-11 lists TIDs considered at component level in the frame of the CLEO/P Study.

6.5 Margin Policy

The following margin policy is applicable to the CLEO/P Study:

- ΔV
  - 5% deterministic ΔV
  - 0% stochastic ΔV
- Propellant
  - 2% on MAN Propellant
  - 100% on AOGNC Propellant
- Mass
  - Maturity margins based on TRL (5, 10, 20 %)
  - System Margin 20%
- Harness
  - 5% of dry mass excl system margin
- Power
  - 20% on power budget
- Radiation
  - Factor 2 on the environment (TIDs)
- Volume
  - 20% on boxes volume

6.6 Interface to NASA CLIPPER

The following parameters are to be considered as main interface specifications to the NASA CLIPPER spacecraft:

- CLEO/I Wet Mass, including 20%: 266.75 kg
- Power required in DORMANT Mode: 32 W (incl. 20% margin), for thermal heating and periodic check-outs
- Separation Mechanism mass remaining on CLIPPER after CLEO Release: 5.91 kg (including maturity margins)
- CLEO/I shall be compatible with the ESA and NASA Deep Space ground stations. The use of the NASA 70 m dish would increase the data return (a link with 20 kbps can be achieved, instead of 3.5 kbps)
- Compatibility with CLIPPER Back-up transfer (7.2 years interplanetary transfer with up to 1 VGA and 3 EGA) assessed under assumption that CLEO/I will not be exposed to direct sunlight for any significant duration inside of 1 AU. Transient cases of up to 1 hour and albedo reflection from Venus are considered in the design.
- CLEO/I shall be mounted laterally on the CLIPPER by means of an I/F ring (24" I/F separation system)

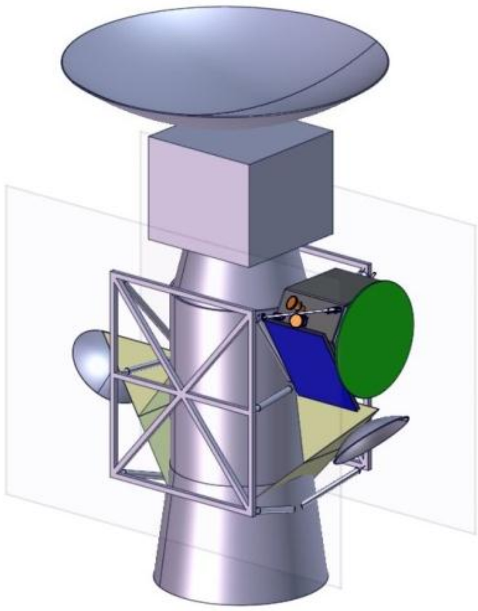


Figure 6-5: CLEO-I mounted on CLIPPER

6.7 System Options

In addition to the baseline configuration, two more options were evaluated at system level:

- A hyperbolic flyby option with much reduced ΔV requirements (CLEO-I hyper), and,
- An option with Europa flybys instead of Io (CLEO-E).

These options were not looked at with the same detail of the baseline. They were simply assessed at system level by establishing assumptions for the deltas w.r.t. to the baseline.

6.7.1 Hyperbolic Flyby (CLEO-I hyper)

One of the possibilities to reduce the mass of the CLEO-I mission is to simply make a hyperbolic flyby of Io. By separating from CLIPPER before JOI and performing targeting manoeuvres estimated at around 40 m/s it is possible to considerably reduce the required propellant mass and achieve a more lean configuration.

In this option, the S/C does not perform flybys around Jupiter but remains in a hyperbolic trajectory. Only a single passage near Io is achieved.

6.7.1.1 ΔV estimation

Assuming the following CLIPPER orbital characteristics:

- Vinf: 5.58 km/s
- Declination / Jupiter's equator: -4.6 deg

The estimated (linear) retargeting ΔV cost after separation is:

- Separation 2 months before JOI: 200 m/s
- Separation 6 months before JOI: 70 m/s
- Separation 1 year before JOI: 30 m/s

Assuming that the separation occurs one year before JOI and considering additional 10m/s for the fly-by targeting, the total estimated ΔV for the CLEO-I hyper option is 40 m/s.

6.7.1.2 Maximum flyby velocity constraint

The maximum flyby velocity is driven by the maximum allowable smearing in the camera that was chosen for the payload (AMIE camera identical to the SMART-1 mission).

Assuming a passage at 200 km altitude from Io, the spatial resolution of the camera (0.00576 deg/pixels), and the integration time of 1.2 ms, if the maximum allowed smearing is set 0.5 pixels, the maximum allowable velocity during the passage is 8.38 km/s.

The chosen trajectory for the CLEO-I hyper option has a Vinf of ~8.5 km/s and an estimated ground velocity for an altitude of 200 km w.r.t. to Io of ~8.8 km/s.

Even though this velocity is higher than the one estimated to limit the smearing to 0.5 pixels, it is very close and within error range.

Therefore, the CLEO-I hyper trajectory was deemed compatible with camera requirements.

6.7.1.3 Mass budget estimation

A preliminary mass budget was estimated by comparing the CLEO-I hyper option with the baseline option.

	Mass budget (with DMM) margin	
Domain	**S2b Total Mass (kg)**	**S4 Total Mass (kg)**
AOGNC	3.42	3.42
COM	24.75	18.38
CPROP	19.33	11.33
DH	5.40	5.40
INS	14.82	14.82
MEC	12.18	10.00
PWR	41.63	30.00
RAD	19.06	10.00
STR	28.75	23.74
SYE	8.66	7.22
TC	310.68	2.50
Total (kg)	**180.01**	**136.80**
System margin (%)	20.00	25.00
Total dry mass (kg)	**226.81**	**171.01**
ΔV (m/s)	345.55	40
Propellant (kg)	39.15	3.41
Propellant mass margin (2%) (kg)	0.78	0.07
Total wet mass (kg)	**266.75**	**174.48**
Allowable wet mass (kg)	**250.00**	**318**

Table 6-12: CLEO-I hyper option mass budget estimation

(differences w.r.t to baseline are highlighted in red)

The following assumptions were made:

- Same mass allocation for equipment performing the functions of attitude orbit control, guidance and navigation (AOGNC)
- Lower mass allocation for equipment performing the communications functions (COMMS) due to the removal of physical redundancy of transponders, EPCs and TWTAs
- Lower mass allocation for equipment performing the propulsion functions (CPROP) because the lower ΔV requires a smaller propellant tank, less thrusters (4 nominal + 4 redundant instead of 6 nominal + 6 redundant) and only 2 Latch valves
- Same mass allocation for equipment performing the functions of data handling (DHS)
- Same mass allocation for instruments (INS), still assuming the same 4 instruments, meaning no loss in terms of scientific return
- Lower mass allocation for equipment performing the functions of mechanisms (MEC), because on a first order analysis the mass of the mechanisms sizes with the overall dry mass
- Lower mass allocation for equipment performing the functions of power generation and storage (PWR), because the science phase should have a smaller duration leading the a smaller battery. Additional gains can also be achieved in terms of a lower mass PCDU, and possibly smaller solar arrays

- Lower mass allocation for radiation shielding (RAD), because the mission would have a lower overall duration (lower dose) and the removal of redundancy for the communication equipment leads to a reduced size of the vault to be shielded
- Lower mass allocation for the structural equipment (STR), because on a first order analysis the mass of the structures sizes with the overall dry mass
- Lower mass allocation for the harness (SYE), because the mass of the harness sizes with the overall dry mass
- Lower mass allocation for the thermal control equipment (TC), because a more compact (lower volume) configuration is expected.

6.7.2 CLEO/E

In addition to the CLEO concepts where the science target is Io, a delta-design was analysed during this CDF study to briefly assess the impact of orbiting the Jupiter moon Europa instead. A delta analysis w.r.t. payload, orbital transfer, subsystem design, system level mass budget as well as critical areas such as radiation shielding and planetary protection has been initiated and is described in this chapter.

6.7.2.1 Model Payload

The instruments for the CLEO/E payload were selected with the idea to conduct measurements that will complement the data gained during the CLIPPER and JUICE missions. Furthermore, the CLEO/E spacecraft provides the possibility to analyse and therefore to prove the existence of dust plumes originating from the surface of Europa.

Since these possibly existing plumes on Europa do not reach an altitude above 26 km above the ground, the flyby distance of the spacecraft needs to be lowered to 26 km above the Europa surface to be able to characterise the dust particles and their mass distribution. This is a key driver for the overall mission scenario.

Details regarding the instrumentation of the CLEO/E model payload can be found in chapter 4 (Payload) while an overview is depicted in Table 6-13 hereafter.

Instrument type and heritage	Instrument mass [kg] Dimensions [mm]	Power [W]	Data Rate [kbps]	Pointing
UV spectrometer (Phebus on BepiColombo)	7.6 500 x 400 x 400	20	29.4	Nadir
Neutral Gas and Ion Mass Spectrometer (NIMS from Particle Environment Package on JUICE)	3.5 (-0.5 backend electronics PCB) Sensor = 300 x 90 x 90 Electronics = 240 x 350 x 170	19.6	51.23 with com-pression	RAM, 360 deg FoV (10 deg opening angle)
Dust Experiment (Dust Detector from Lunar Dust Experiment – LDEX on LADEE + Time-of-Flight Mass Spectrometer from Laser Mass Spectrometer – Breadboard in SRE)	4.5 150 x 150 x 200	7	44	RAM Altitude: <26km
Total	**15.1**	**46.6**		

Table 6-13: CLEO/E model payload

6.7.2.2 Payload operations and data volume

The CLEO/E data volume generated per flyby is based upon the following assumptions:

- Flyby distance: < 26 km, due to maximum plume height
- Flyby velocity: 5.1 km/s (at least 2 km/s is required for Dust Experiment to ionize heavy ions).

The operational sequence per instrument is as follows:

- Dust Experiment on/off: ± 150 min to closest approach
- NIMS on/off: ± 60 min to closest approach
- UV Spectrometer on/off: ± 30 min to closest approach.

It should be noted that the Dust Experiment only generates science data in case of a dust detection event. The total science data depicted in Table 6-14 is therefore a worst case assumption.

Data Volume	Data rate [kbps]	Operations Time [h]	Data volume per flyby [Gb]
Dust Experiment	44.00	5	0.792
NIMS	51.23	2	0.369
UV Spectrometer	29.40	1	0.106
Total Instruments			**1.267**
Housekeeping and Calibration			0.3
Total per flyby			**1.6**

Table 6-14: CLEO/E data volume

In addition to the science data generated by the instruments, an allocation of 0.3 Gb has been added for payload housekeeping and calibration data. The total data volume generated by the payload per Europa flyby is therefore assumed to be 1.6 Gb.

6.7.2.3 Impact on communications subsystem: DTE vs. Relay

The CLEO/E payload data volume accumulated per flyby is about a tenth of the data generated by the CLEO/I payload. This offers the potential to establish a relay communications link with CLIPPER instead of direct-to-Earth (DTE) communications. Both options are discussed in this chapter.

For DTE communications, a worst case distance to Earth of 6 AU has been considered. Taking into account the same telemetry link set-up as for CLEO/I, a data rate of 3.5 kbps can be achieved.

Table 6-15 depicts the minimum orbital period based upon the durations required for each of the following modes: Science (SCI), Jovian Cruise (JC), Direct-to-Earth Communication (DTE), and Manoeuvre (MAN). The available power is a key driver for the orbital period. As for CLEO/I, 28 hours are needed to recharge the battery during JC. This is the case after 5 hours in Science mode, 2.7 hours in DTE mode or 3 hours in Manoeuvre mode.

For CLEO/E, three manoeuvres were considered per flyby: targeting, clean-up and PRM, each of them lasting 3 hours. Per manoeuvre, 2 DTE slots of each 2.7 hours are foreseen for Doppler and Ranging.

Modes	Duration	Unit
SCI	5	h
JC (after SCI)	28	h
DTE (for Science data download)	124.34	h
JC (after DTE for Science data download)	1289.46	h
MAN	9	h
JC (after MAN)	84	h
DTE (for Doppler and Ranging)	16.2	h
JC (after DTE for Doppler and Ranging)	168	h
Total	**71.83**	**d**

Table 6-15: CLEO/E orbit timeline for DTE option

Considering a total data volume of 1.6 Gb, a minimum orbital period of 72 days is required.

For relay communications, two options have been assessed: a worst case distance between the CLIPPER and CLEO/E of 40 Jovian radii and an optimised case where the maximum distance between the two spacecraft is 10 Jovian radii (R_J). It was confirmed by the mission analysis expert that the CLEO/E trajectory can be optimised to achieve the required time for relay communications within a distance of 10 R_J or less.

The data rate for both options was provided by the communications expert and is calculated to be 3.5 kbps at 40R_J and 60 kbps at 10 R_J.

The duration of one slot relay communications is however shorter than for one DTE slot. During DTE, the solar array is supporting the battery, while during relay communications this is not possible since the CLEO/E spacecraft is pointing to CLIPPER and not to the Earth (Sun direction). As for CLEO/I, one relay communications slot takes 2.1 hours.

Table 6-16 depicts the minimum orbital period based upon the durations required for Science, Jovian Cruise, Relay Communication (REL), Direct-to-Earth Communication for navigation, and Manoeuvre. Again, 28 hours are needed to recharge the battery during JC after 5 hours in Science mode, 2.7 hours in DTE mode, 3 hours in Manoeuvre mode, or 2.1 hours in REL mode.

Modes	Worst Case Duration	Optimised Case Duration	Unit
SCI	5	5	h
JC (after SCI)	28	28	h
REL (for Science data download)	126	7.25	h
JC (after REL for Science data download)	1680	96.71	h
MAN	9	9	h
JC (after MAN)	84	84	h
DTE (for Doppler and Ranging)	16.2	16.2	h
JC (after DTE for Doppler and Ranging)	168	168	h
Total [d]	**88.18**	**17.26**	**d**

Table 6-16: CLEO/E orbit timeline for relay communication option

Considering the same number of manoeuvres per flyby (targeting, clean-up and PRM) as for the DTE option as well as 2 DTE slots per manoeuvre for Doppler and Ranging, the minimum orbital period is 88 days at 40 R_J and 17 days at 10 R_J distance. It is clear that in case of 40 R_J distance, relay communications is not advantageous compared to DTE communications. However for the optimised case of 10 R_J distance, the minimum orbital period is only slightly above the targeted orbital period of 14 days (CLEO/E orbit phased with CLIPPER orbit). The following measures were identified to support a decrease of the orbital period:

- Improved gain on-board CLIPPER spacecraft: 50 dBi instead of the currently assumed 30 dBi
- Reduced number of manoeuvres per orbit: 2 instead of 3 as for CLEO/I
- Implementation of autonomous navigation to save one sequence of ground-based orbit determination (cf. currently on-going TDA on Innovative Autonomous Navigation Techniques (IANT))
- Increased solar array to reduce the time needed to recharge the battery
- Reduced overall data volume (science data plus housekeeping).

Further analyses are required to quantify the potential for each of the listed options or a combination hereof.

6.7.2.4 CLEO/E transfer and Europa flyby orbits

The CLEO/E transfer and orbit analysis has been conducted by the mission analysis expert and is described in detail in chapter 5 (Mission Analysis). As a result, three options were identified of which two support relay communications with CLIPPER and the third is suitable for DTE communications:

- 4:1 resonance with Europa: CLEO/E phased with CLIPPER, optimum for relay communications, 14 days cycle, infinite velocity: 4.1 km/s, apojove: 38 R_J
- 6:1 resonance with Europa: not phased with CLIPPER, less time for relay, 42 days cycle, infinite velocity: 4.7 km/s, apojove: 53 R_J
- Alternative: DTE communications, more than 60 days cycle.

The CLEO/E baseline selected at this stage is to be in an orbit phased with CLIPPER and 4:1 resonant with Europa. The key characteristics of the CLEO/E transfer to this orbit are listed hereafter:

- JOI and PRM with CLIPPER
- Separation after PRM
- G1 (>G1: 8:1, perijove around 12 R_J, 3.7 degree inclination, 7 km/s like CLIPPER)
- G2 (>G2: 5:1, perijove around 11 R_J, 1.3 degree inclination like CLIPPER)
- G3 (>G3: around 22-23 days, perijove around 10 R_J, 0 degree inclination): transfer to Callisto at 6.5 km/s
- C4: (>G4: around 20-21 days (--> high altitude C4), perijove around 9.2 R_J) transfer to Europa at 4.7 km/s (apojove around 37 R_J)
- Europa fly-bys: velocity at 200 km altitude: 5.1 km/s.

After the transfer to Europa, at least four more flybys are needed to obtain a flyby altitude below 26 km. To achieve this, the following manoeuvres need to be carried out:

- EGA1 at 400 km (first EGA at the beginning of COT-1, i.e. around 1 year after JOI)
- EGA2 at 200 km
- EGA3 at 50-100 km
- EGA4 at 25-50 km

After EGA4 two flybys at the required altitude of below 26 km above the Europa surface are foreseen to conduct the scientific measurements for CLEO/E.

6.7.2.5 ΔV budget

The ΔV budget for the CLEO/E baseline transfer and orbit is depicted in Table 6-17. The following considerations were taken when establishing the CLEO/E ΔV budget:

- JOI and PRM including the respective clean-up manoeuvres are carried out by the CLIPPER spacecraft and thus no ΔV allocation is needed for CLEO/E
- Separation of CLEO/E occurs after PRM
- 8 flybys with an altitude greater than 26 km above the Europa surface are planned: G1, G2, G3, C4, EGA1, EGA2, EGA3, and EGA4
- 2 flybys with an altitude below 26 km above the Europa surface are foreseen
- The deterministic ΔV assumed by the mission analysis expert is 4 m/s per flyby. The margin on the deterministic ΔV is 5% (derived from JUICE)
- The stochastic ΔV assumed by the mission analysis expert is 10 m/s per flyby. There is no margin applied to the stochastic ΔV
- An allocation of 15 m/s is given for the disposal of the spacecraft at EOL
- A margin of 5 % is applied to the overall ΔV (derived from JUICE).

ΔV Budget	**CLEO/E**	**Unit**
JOI	0	m/s
PRM	0	m/s
# flybys > 25 km	8	
# flybys <= 25km	2	
Deterministic ΔV per flyby	4	m/s/flyby
Deterministic ΔV	40	m/s
Margin on deterministic ΔV	5	%
Stochastic per flyby	10	m/s/flyby
Stochastic	100	m/s
Margin on stochastic ΔV	0	%
Disposal	15	m/s
Total without margin	**155**	**m/s**
Total incl. margin on det. and stoch. ΔV	157	m/s
Margin on total ΔV	5	%
Total incl. margin	**164.85**	**m/s**

Table 6-17: CLEO/E ΔV budget

The total ΔV needed for CLEO/E is 165 m/s. This includes the transfer to the target orbit as well as two flybys at an altitude below 26 km.

6.7.2.6 Radiation analysis

As part of this CDF study, an initial radiation analysis has also been performed for CLEO/E. The details of this analysis can be found in chapter 8 (Radiation) while the results of this analysis are summarised hereafter. In Table 6-18 the total ionising doses at the centre of each unit are depicted. To assess the required shielding thickness two different parameters have been adjusted:

- Shielding thickness of 5 and 10 mm aluminium
- Total ionising dose in vacuum and non-vacuum.

The critical components for which extra shielding has been identified to be required are identical with the ones of CLEO/I: two transponders, two EPCs and the MINAVIO.

TID	5mm	10 mm
Transponder (Vacuum)	108	43
Transponder (Full)	13	6
EPC (Vacuum)	77	30
EPC (Full)	11	5
MINAVIO (Vacuum)	111	45
MINAVIO (Full)	0.1	0.1

Table 6-18: CLEO/E Total Ionising Dose

As a result of the radiation analysis, the vaults have been conservatively designed with the same mass and dimensions as for CLEO/I (again it is recalled here that boxes are closer to the "full' than to the "vacuum" case). Considering a 5 mm wall thickness for both vaults, 9.55 kg shielding mass are required for the transponder and EPC and 6 kg are needed for the MINAVIO. In addition to that, 1 kg noise shielding is required per payload instrument.

6.7.2.7 Mass budget estimate

For CLEO/E the mass has been estimated on subsystem level in close collaboration with the respective domain experts. Within each domain of expertise, the impact of going to Europa instead of Io has been analysed w.r.t. potential consequences on the equipment selection and sizing. It has to be noted, that the CLEO/E design has not been established off scratch but rather derived as a delta from the CLEO/I baseline design. In this way the mass per subsystem could be estimated and the overall CLEO/E mass budget established in a very short timeframe. In the following paragraphs, all deltas w.r.t. the CLEO/I baseline design are addressed.

No changes in mass are foreseen for the following subsystems:

- AOGNC: same equipment as for CLEO/I
- Communications: no changes in mass, however for the relay link with CLIPPER the size of the HGA could potentially be decreased
- Data Handling: same equipment as for CLEO/I and the PCDU boards are included in the power subsystem mass
- Thermal subsystem: same mass allocation as for CLEO/I.

Minor changes have been reported for the following subsystems:

- Mechanisms: Due to the lower wet mass of the CLEO/E spacecraft, a slight reduction in mass has been considered for the satellite deployment mechanism between CLIPPER and CLEO/E.

- Power: As a first estimate, the power subsystem mass is assumed to be similar as for CLEO/I. It can be noted, that the battery mass could potentially be reduced by 0.5 kg maximum. The solar array on the other side could face a small increase in case of relay communications since the CLEO/E spacecraft needs to be pointing to CLIPPER. A more detailed assessment of these deviations from the CLEO/I power subsystem would be needed in case of further advancement of this study.
- Radiation: The mass of the two vaults remains the same but 0.5 kg is saved w.r.t. the noise shielding for the instruments.
- Structure: At this stage, the structure mass is assumed to be similar as for CLEO/I. However, due to the lower wet mass of the spacecraft, there is some potential to decrease the structural mass. Further analysis is required to optimise the structure for CLEO/E.
- Harness: 5 % of the spacecraft dry mass are allocated as harness mass (SYE). Due to the lower spacecraft dry mass, also the harness mass is slightly lower compared to CLEO/I.
- System margin: 20 % system margin is applied at spacecraft level. The lower dry mass compared to CLEO/I also impacts the mass of the system margin slightly.

Larger changes in mass were identified in the following areas:

- Chemical Propulsion: Due to the lower ΔV, a smaller tank could be selected, thus reducing the propulsion subsystem dry mass by 3.5 kg. The baseline design consisting of 12 x 1 N thrusters and 1 x 20 N thrusters remains identical. However, further analysis is required to assess whether the 20 N thruster can be removed from the CLEO/E design.
- Payload: Mass increase of 3.3 kg due to the different instrument suite for CLEO/E.
- Propellant: The most significant mass saving comes from the propellant mass. Approximately 20 kg less is needed for CLEO/E compared to the CLEO/I baseline scenario. The reason for this is the much lower ΔV required for the mission. 165 m/s (including margins) are required for the transfer to the target orbit and for performing two Europa flybys at an altitude below 26 km above the surface. In addition to that, AOGNC requires 0.31 kg of propellant (incl. 100 % margin). The propellant mass is currently calculated assuming a total ΔV of 170 m/s. Therefore the derived propellant mass is rather conservative and could be reduced even more after refined analyses.

	CLEO/I Baseline Design				CLEO/E
Row Labels	**Mass [kg]**	**Margin [kg]**	**Mass Margin [%]**	**Total Mass [kg]**	**Total Mass [kg]**
AOGNC	3.24	0.18	5.46	3.42	3.42
COM	22.20	2.55	11.49	24.75	24.75
CPROP	18.17	1.16	6.38	19.33	15.88
DH	4.50	0.90	20.00	5.40	5.40
INS	12.35	2.47	20.00	14.82	18.12
MEC	11.07	1.11	10.00	12.18	10.18
PWR	34.69	6.94	20.00	41.63	41.63
RAD				19.06	18.55

	CLEO/I Baseline Design				CLEO/E
Row Labels	**Mass [kg]**	**Margin [kg]**	**Mass Margin [%]**	**Total Mass [kg]**	**Total Mass [kg]**
STR	23.96	4.79	20.00	28.75	28.75
SYE				9.00	8.87
TC	9.71	0.97	10.00	10.68	10.68
Total Dry Mass	**139.89**	**21.85**	**12.32**	**189.01**	**186.22**
System Margin			20.00%	37.80	37.24
Total Dry Mass with Margin				**226.81**	**223.46**
Propellant				39.15	19.49
Propellant mass margin			2.00%	0.78	0.39
Total Wet Mass				**266.75**	**243.34**
Target mass				250.00	250.00
Above target mass				16.75	-6.66

Table 6-19: CLEO/E mass budget

Table 6-19 shows the CLEO/E mass budget in direct comparison to the CLEO/I mass budget. Taking into account the target mass of 250 kg, the total wet mass of the CLEO/E spacecraft remains below this target by 6.7 kg and is therefore compliant to the mass requirement.

6.7.2.8 Technical Conclusion

The first delta-design assessed for the CLEO/E spacecraft shows that a feasible design can be established within the required mass of 250 kg. However, it has to be noted that the CLEO/E design has only been done as a delta-design based on the CLEO/I baseline design. Therefore, further analyses are needed before deriving final conclusions.

In the following, those fields are listed where refined analyses and trade-offs need to be provided:

- Scientific return: A critical assessment of the CLEO/E benefits w.r.t. larger missions to Europa (JUICE, CLIPPER) should be performed, deriving the scientific objectives and the corresponding selection of instruments.
- Communications: Detailed analyses and trade-offs are required to optimise the communications concept for CLEO/E w.r.t. data volume for science and housekeeping data, communication windows for relay to CLIPPER, equipment selection and antenna sizing, as well as autonomous navigation possibilities.
- Power: A power budget based on the specific CLEO/E equipment and system modes needs to be provided to quantify conclusions related to the sizing of the CLEO/E battery and solar array.
- Planetary protection: For CLEO/E a detailed trade-off needs to be made between the implementation of an impact avoidance strategy (as for JUICE) or applying active bioburden control (as typical for landing systems). Both concepts have their advantages and disadvantages in terms of system mass, design complexity, as well as development and AIV cost. At this stage, a final recommendation for one or the other solution cannot be made. Both concepts of planetary protection implementation need to be refined, also keeping an eye on potential future changes to the planetary protection approach for CLIPPER.

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. In line with this category, the following planetary protection requirements of RD[11] are applicable to the CLEO/I and CLEO/E concepts:

Requirements	Note for CLEO/I	Note for CLEO/E
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 CLEO the analysis to be covered by NASA for Mars and Europa; post-release of CLEO the analysis for Europa has to be covered by ESA	Protected solar system bodies are Europa and Mars; prior to release of CLEO the analysis to be covered by NASA for Mars and Europa; post-release of CLEO the analysis for Europa has to be covered by ESA
5.3.2.1d	To be covered by NASA	To be covered by NASA
5.3.2.1e.1	To be covered by NASA	To be covered by NASA
5.3.3.2a, b	Suggest to focus on the probability of accidental impact on Europa for a time period until the most shielded parts of the spacecraft reach an ionizing radiation dose of at least 25 kGy	Due to flight profile and final disposition this would require substantial bioburden control, sterilisation and re-contamination protection
5.4		
5.5		
5.6a, b		
5.7		
Annex A, B, C, D, E, F (if applicable), and G		

Due to the current planetary protection approach of the NASA Clipper concept which is still under review (at least based on the information available in the SALMON-2), additional bioburden reduction and re-contamination control requirements might become applicable to CLEO. This could lead to a major cost increase for the CLEO/I concept and to a significant cost increase for the CLEO/E concept.

7.2 Design Drivers

Meeting the probability of contaminating Europa in the CLEO/I or CLEO/E concept should first focus on avoiding any impact on Europa (following the approach used for JUICE). Implementing an impact avoidance approach to meet the probability of impact to levels lower than $1x10^{-4}$ would require careful trajectory optimisation, very reliable hardware and a tailored FDIR strategy. In case the Europa impact avoidance cannot be demonstrated, active bioburden control measures would need to be applied to the CLEO/I and CLEO/E spacecraft.

Unlike for a lander with a capsule acting as a recontamination barrier (e.g., Viking) an orbiter spacecraft is essential an open system. So far, no orbiter spacecraft has been developed with the bioburden control, including recontamination protection, required for CLEO/E (or potentially CLEO/I).

The major design drivers for bioburden control of a spacecraft in general 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[12] 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[13] or room temperature hydrogen peroxide gas RD[14] would require the use of models that are similar to qualification models RD[15].

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 radiation analysis for CLEO/I was performed on the S2b trajectory. This trajectory was segmented into five different legs:

Leg 1: Ganymede 28:1, inclination 5.4°, 198 d

Leg 2: Ganymede 28:1, inclination 1.0°, 198 d

Leg 3: Ganymede 28:1, inclination 0.0°, 198 d

Leg 4: Io 56:1, inclination 0.8°, 99 d

Leg 5: Io 107:1, inclination 0.2°, 190 d

For CLEO/E, a set of gravity assist manoeuvres is done with the perijove and apojove data shown in Table 8-1:

Moon	**number**	**Perijove & Apojove**
G1	×1	12×53 Rj
G2	×1	9.2×38 Rj
G3	×1	9.2×38 Rj
C4	×1	9.2×37 Rj
EGA	×4	9.2×38 Rj
Europa Flyby	×2	9.5×37 Rj

Table 8-1: Jovian Moon gravity assists

As shown in Figure 8-1, the low inclination leads to traversal of the radiation belts.

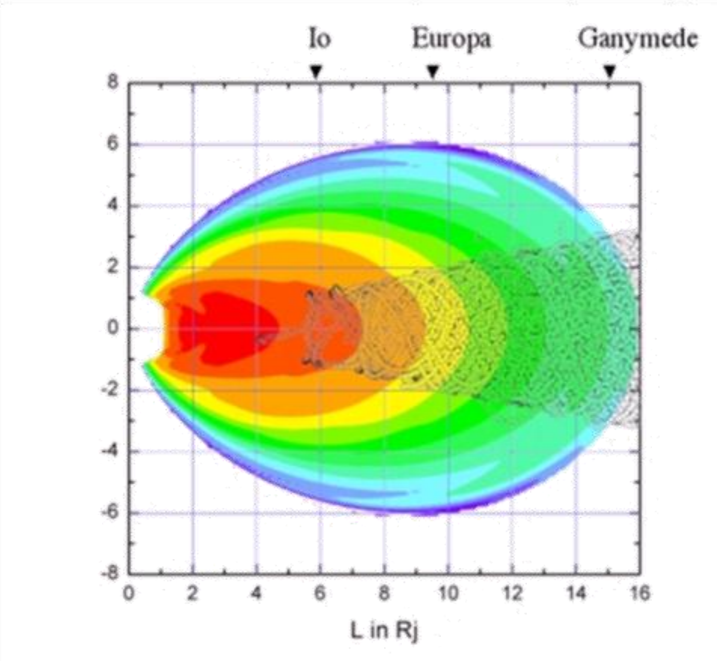


Figure 8-1: Overview of Jovian radiation belts

8.2 Radiation Dose Analysis

Note that other aspects of the radiation environment could be critical for the mission, in particular the transfer of charged particles and flux of heavy ions. This would need a careful assessment in any subsequent phase, in particular for the instruments. Table 8-2 gives an overview of the radiation dose, as function of the shielding thickness (mm aluminium), for CLEO/I. It should be noted that these doses are calculated excluding margin.

Shielding	Leg 1	Leg 2	Leg 3	Leg 4	Leg 5	Total
1.00	8.91E+04	1.15E+05	1.16E+05	1.60E+05	1.59E+05	6.38E+05
1.50	4.50E+04	6.20E+04	6.25E+04	1.01E+05	9.97E+04	3.70E+05
2.00	2.78E+04	4.03E+04	4.06E+04	7.60E+04	7.47E+04	2.59E+05
2.50	1.93E+04	2.90E+04	2.93E+04	6.25E+04	6.17E+04	2.02E+05
3.00	1.43E+04	2.21E+04	2.23E+04	5.39E+04	5.35E+04	1.66E+05
4.00	8.91E+03	1.41E+04	1.43E+04	4.35E+04	4.38E+04	1.25E+05
5.00	6.16E+03	9.91E+03	1.00E+04	3.75E+04	3.83E+04	1.02E+05
6.00	4.57E+03	7.38E+03	7.47E+03	3.37E+04	3.50E+04	8.81E+04
7.00	3.54E+03	5.73E+03	5.80E+03	3.11E+04	3.27E+04	7.89E+04
8.00	2.83E+03	4.59E+03	4.65E+03	2.92E+04	3.10E+04	7.24E+04
9.00	2.32E+03	3.76E+03	3.81E+03	2.78E+04	2.97E+04	6.74E+04
10.00	1.93E+03	3.13E+03	3.18E+03	2.66E+04	2.87E+04	6.35E+04
12.00	1.40E+03	2.27E+03	2.31E+03	2.46E+04	2.67E+04	5.73E+04
14.00	1.06E+03	1.72E+03	1.75E+03	2.28E+04	2.50E+04	5.23E+04
16.00	8.23E+02	1.34E+03	1.36E+03	2.10E+04	2.32E+04	4.76E+04
18.00	6.57E+02	1.07E+03	1.09E+03	1.93E+04	2.15E+04	4.36E+04
20.00	5.23E+02	8.50E+02	8.68E+02	1.73E+04	1.95E+04	3.91E+04

Table 8-2: Mission doses [rad(Si)] excluding margin

The total ionisation doses for the CLEO/I S2b trajectory and CLEO/E trajectory are shown in Figure 8-2.

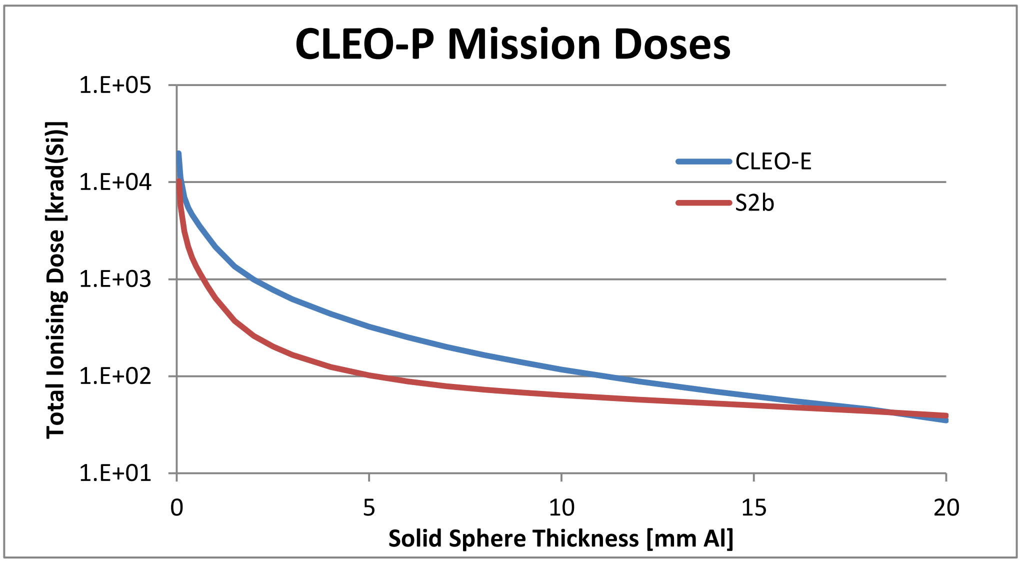


Figure 8-2: Total ionisation dose

8.3 Sector Analysis

While the previous section only focussed on the total dose experienced by the spacecraft within the orbit, a sector analysis was done to determine the dose per equipment. The approach taken was to insert two vaults (as shown in Figure 8-3 covering the most radiation sensitive units. As at this stage in the design no specific information is available on the internal accommodation of the units, each unit box density is adjusted to provide a homogenous shielding consistent with the total unit mass.

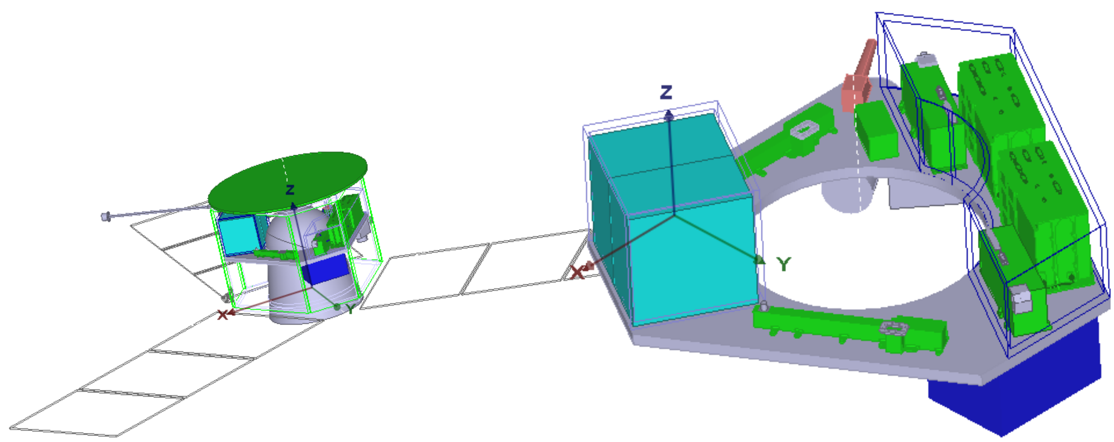


Figure 8-3: Definition of vaults for sectoring analysis

The dose analysis was then used to determine the required vault thickness. The results are shown in Table 8-3, showing the CLEO/E and CLEO/I (S2b) results, again excluding margins.

	CLEO-E		S2b	
	5mm	10 mm	5mm	10 mm
Transponder (Vac)	108	43	55	38
Transponder	13	6	22	16
EPC (Vac)	77	30	42	28
EPC	11	5	19	13
MiniAvionic	0.1	0.1	2	2
MiniAvionic (Vac)	111	45	52	36

Table 8-3: Unit level doses [rad(Si)] excluding margin. Vac indicates the dose assuming vacuum

For a 5 mm vault mass, the mass of the two vaults are: 9.55 kg and 6 kg respectively. For a 10 mm vault mass, they are 19.1 kg and 12 kg respectively.

In order to verify compliance with the TID tolerance of the sensitive equipment (i.e. equipment with 50 krad TID tolerance), the following approach was taken:

1. The TID is calculated assuming that the boxes are empty (i.e. contain 'vacuum'); these are the first numbers in Table 8-3. This is a very conservative number as the unit boxes are typically filled up with equipment.
2. The TID is also calculated assuming that the boxes are solid, but with a density scaled to match the unit's total mass. This is an optimistic approach as unit boxes

typically do contain some empty space inside, however it is deemed more realistic than the 'vacuum' approach.

3. An average is taken between these numbers. This is still considered conservative as the unit boxes would normally contain more than 50% equipment; only a small portion is vacuum.
4. Finally, a factor 2 margin is applied.

The results are shown in Table 8-4. It can be seen that in particular the transponder is at the limit of the TID. If required spot shielding could be applied to the transponder, which is estimated at 2.3 kg spot-shielding mass per transponder, or 4.6 kg in total.

	S2b		Factor 2 Applied		TID Tolerance
	5mm	10 mm	5 mm	10 mm	
Transponder (Vac)	55	38	110	76	
Transponder	22	16	44	32	
Transponder Average	38.5	27	***77***	54	50
EPC (Vac)	42	28	84	56	
EPC	19	13	38	26	
EPC Average	30.5	20.5	61	41	50
MiniAvionic	2	2	4	4	
MiniAvionic (Vac)	52	36	104	72	
MiniAvionic Average	27	19	54	38	50

Table 8-4: Unit level doses [rad(Si)] including margin

8.4 Solar Cell Degradation Fluences

In support of solar panel sizing, an analysis was done on solar cell degradation fluences for both options (Io and Europa) and the results are shown in Table 8-5..

	Cover glass Thickness	0	25.4	76.2	152.4	304.8	508
Option S2b	LEG1	2.4E+16	2.0E+14	2.1E+13	5.6E+12	2.3E+12	1.5E+12
	LEG2	2.6E+16	3.0E+14	3.4E+13	9.0E+12	3.5E+12	2.2E+12
	LEG3	2.6E+16	3.1E+14	3.5E+13	9.1E+12	3.5E+12	2.3E+12
	LEG4	4.6E+16	3.6E+15	1.2E+15	5.5E+14	2.0E+14	8.1E+13
	LEG5	4.6E+16	3.9E+15	1.4E+15	6.3E+14	2.3E+14	9.5E+13
	Total	**1.7E+17**	**8.3E+15**	**2.7E+15**	**1.2E+15**	**4.4E+14**	**1.8E+14**
CLEOP-E	G1	3.1E+16	3.7E+14	4.3E+13	1.1E+13	4.3E+12	2.8E+12
	G2	3.6E+16	5.4E+14	7.2E+13	1.8E+13	6.8E+12	4.3E+12
	G3	3.5E+16	6.0E+14	8.9E+13	2.3E+13	8.3E+12	5.2E+12
	C4	4.0E+16	7.3E+14	1.2E+14	3.0E+13	1.1E+13	7.0E+12
	EGAn	1.6E+17	2.9E+15	4.6E+14	1.2E+14	4.3E+13	2.8E+13
	EuropaFlyBy	8.0E+16	1.5E+15	2.3E+14	5.8E+13	2.1E+13	1.3E+13
	Total	**3.8E+17**	**6.6E+15**	**1.0E+15**	**2.6E+14**	**9.5E+13**	**6.0E+13**

Table 8-5: Solar cell degradation fluences (e/m²)based on cover glass thickness, excluding margin

9 CONFIGURATION

9.1 Requirements and Design Drivers

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
CFG-010	CLEO S/C shall be mounted on NASA Clipper S/C	
CFG-020	CLEO shall not interfere with NASA Clipper units.	
CFG-030	CLEO S/C shall be equipped with all units from other subsystem according to their requirement i.e. pointing direction, unobstructed field of view, structural and thermal stiffness.	
CFG-040	Maximum volume allocation of 1m x 1m x 1m as starting point to design the CLEO S/C.	

9.2 Assumptions and Trade-Offs

CLIPPER solar power configuration shown in Figure 9-1 dictates the possible location of the CLEO. Thus CLEO will then be mounted laterally on the CLIPPER. This can be done by means of a 24 inch I/F ring. A trade-off to determine the best configuration was carried out in an early stage of the study. A triangular shape S/C body is chosen to minimise volume and eventually the mass. .

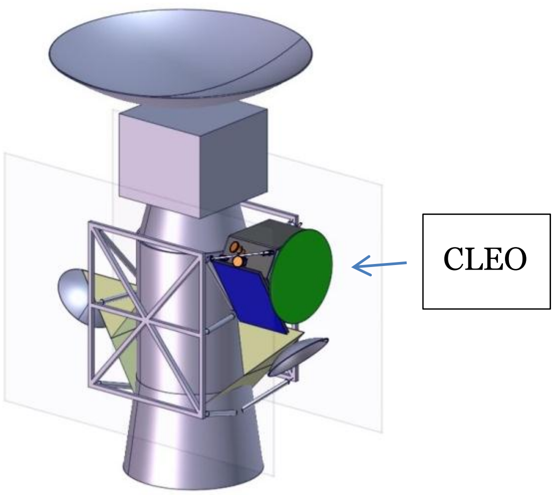


Figure 9-1: CLEO on CLIPPER

9.3 Baseline Design

CLEO spacecraft design has a triangular shape body with cut-out on each of the three corners to accommodate payload that need certain pointing direction and unobstructed FoV. Central cone of diameter 650mm at the bottom will interface with the CLIPPER. The top part of the central cone supports the main propulsion tank of 484mm diameter and supports also the middle platform. The middle platform will give enough mounting surface area for communication, power and data handling equipment. The HGA of 1.1m diameter is mounted on the top panel.

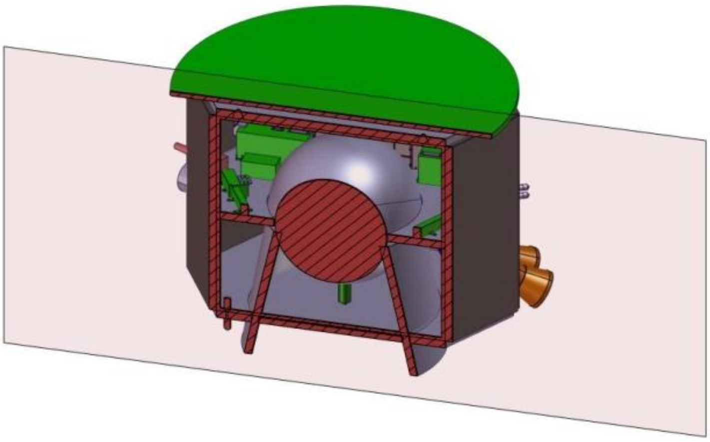


Figure 9-2: CLEO cross section

Bottom panel accommodates mainly the thrusters: 1 x 20N thruster at the centre and four sets of 2x1N-thrusters around the central cone. The other two sets of 2x1N thrusters are accommodated on the S/C body corners as shown in Figure 9-3.

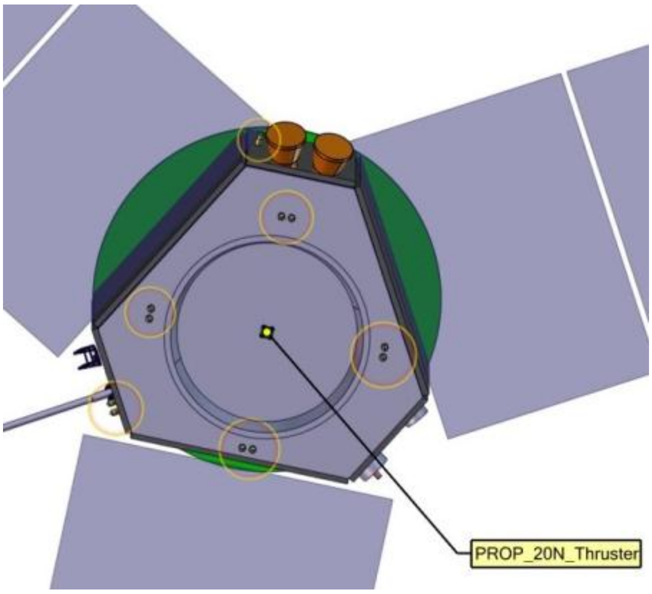


Figure 9-3: Thrusters location

The required surface area of the solar panels is 6m². This can be achieved by having 9 panels of about 0.7m² each. The final dimension of each of solar panel is 1.1m by 0.7m. One stack of solar panels that contains 3 solar panels is mounted on three side panels. Figure 9-4 shows the stowed configuration of the orbiter.

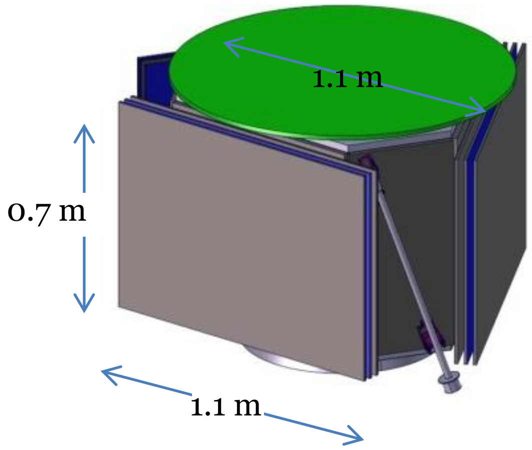


Figure 9-4: CLEO stowed configuration

Figure 9-5 - Figure 9-7 show the exploded view of the orbiter, equipment accommodation and instrument field of view.

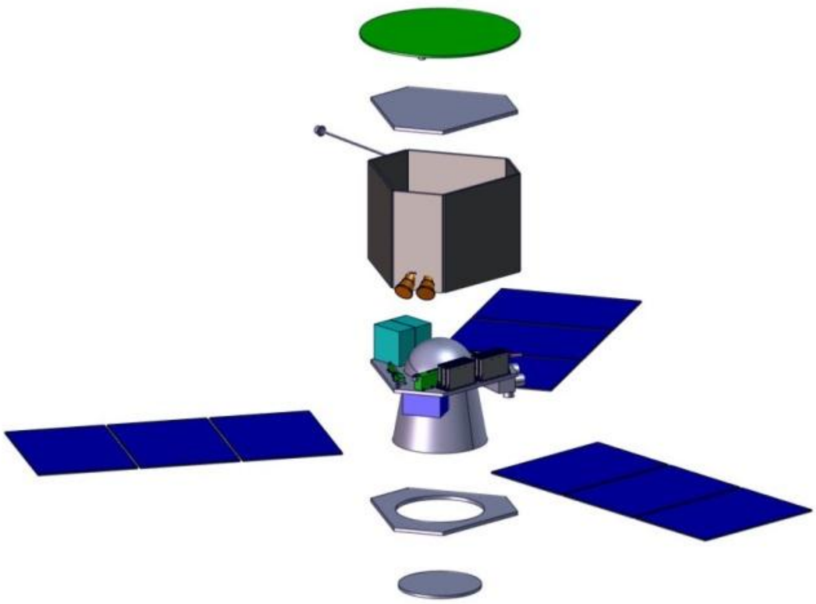


Figure 9-5: Exploded view of CLEO S/C

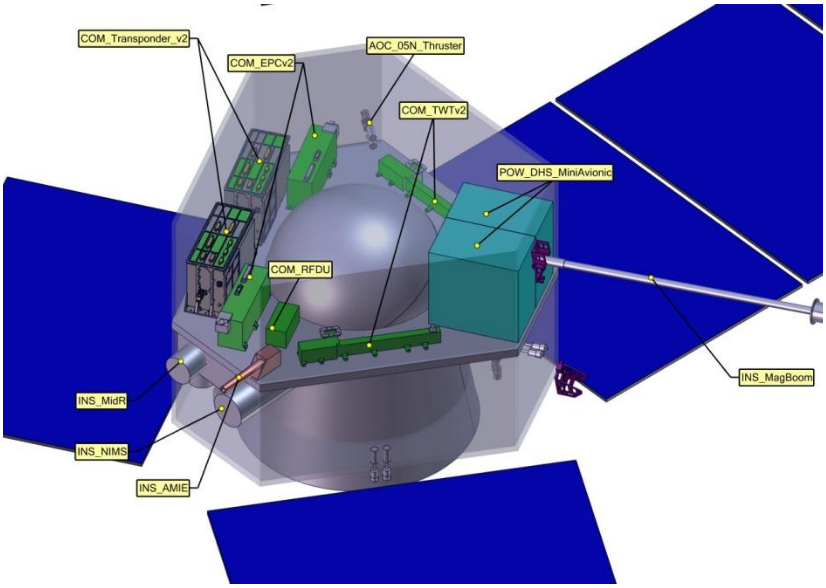


Figure 9-6: CLEO accommodation

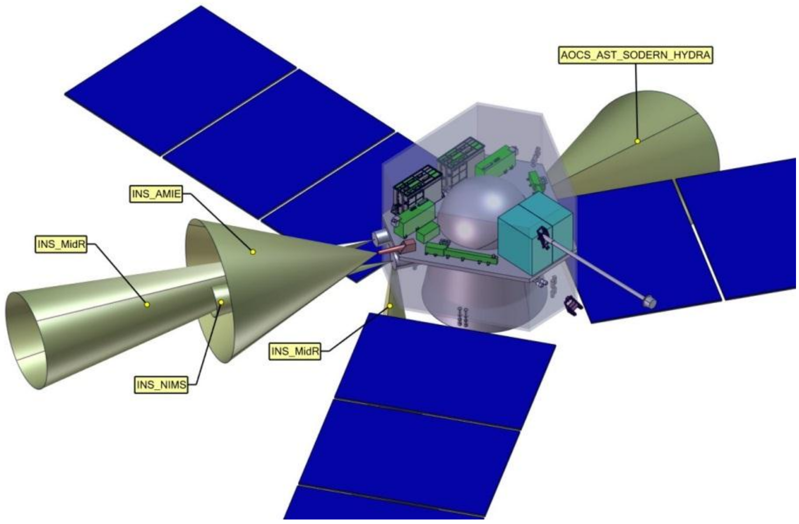


Figure 9-7: Instruments field of view

9.4 Overall Dimensions

Overall dimensions of the stowed and deployed configuration are shown in the following figures

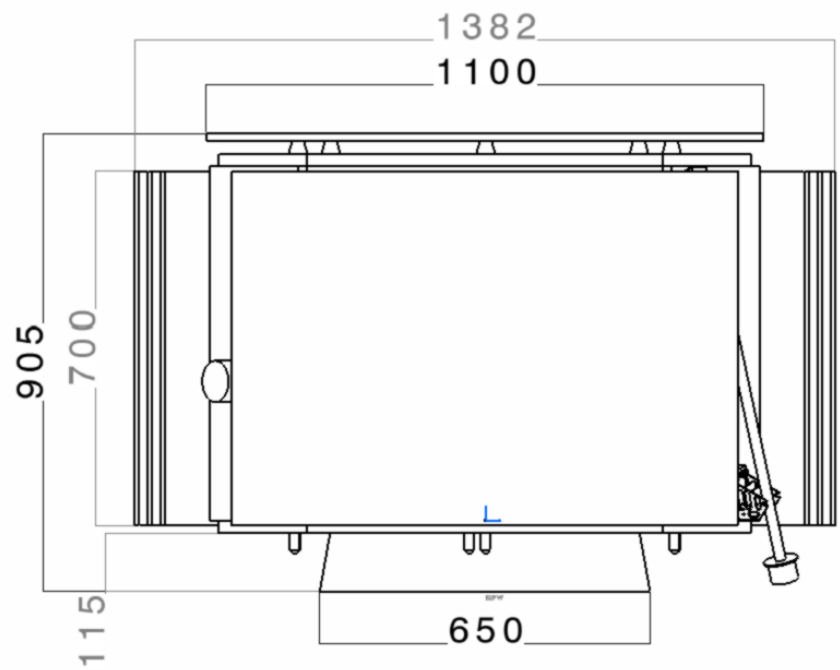


Figure 9-8: CLEO stowed configuration– side view dimension

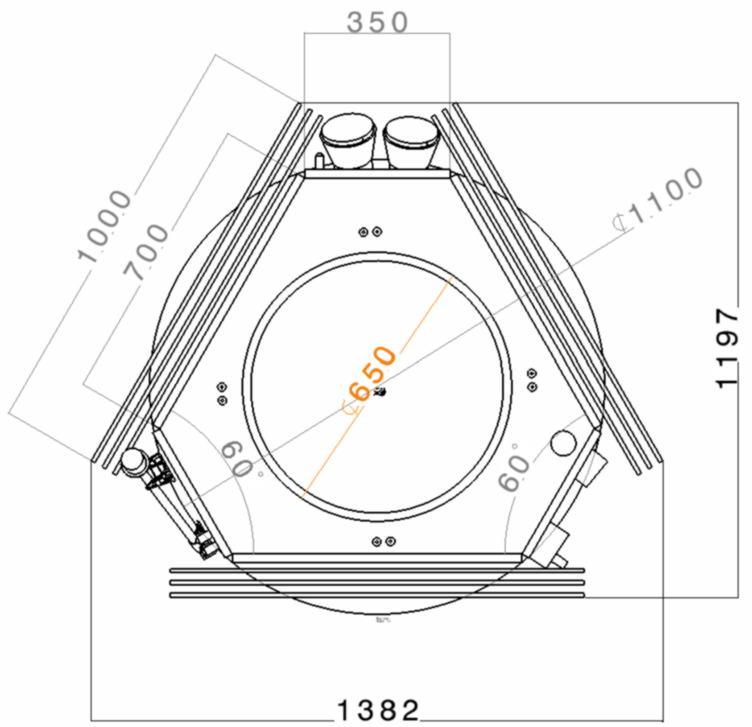


Figure 9-9: CLEO stowed configuration – bottom view dimension

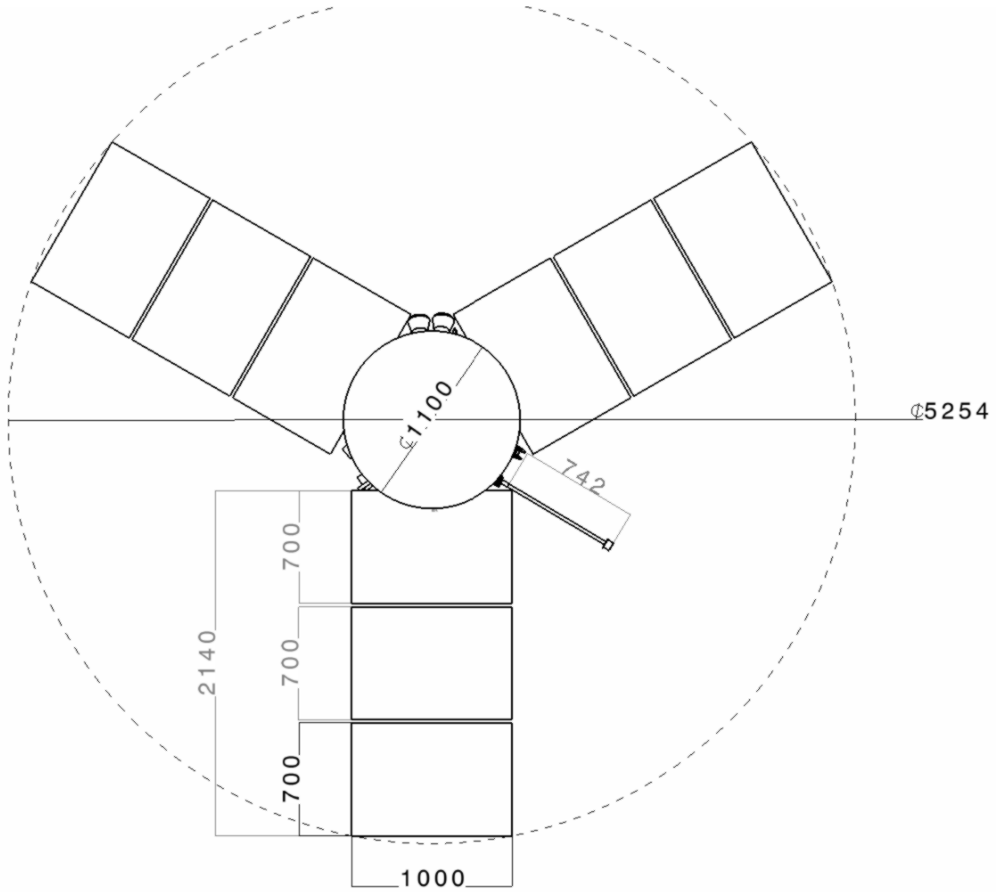


Figure 9-10: CLEO deployed configuration – top view dimension

10 STRUCTURES

10.1 Requirements and Design Drivers

The main requirements applicable to the structure design are stated as follows:

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 before and during LEOP.	
STR-040	The spacecraft shall be compatible with Clipper interface adapter (TBD).	

10.2 Assumptions and Trade-Offs

10.2.1 Assumptions

In order to perform the feasibility study of the structure for CLEOP, some assumptions have been made on the most relevant requirements.

Based on the frequency requirements of the target launcher range, it has been considered necessary to have a first axial and lateral eigenfrequency above 60 Hz. Also, it has been assumed that the allowable volume is restricted to $1m^3$. In addition, given the mass constraints for the satellite wet mass, the structure design will be optimised to be as light as possible while trying to maximise the radiation shielding.

As Clipper dynamic environment is not yet defined, the structure design has been developed based on robust heritage designs. This ensures the feasibility to sustain the environment that will be specified by Clipper with only local reinforcements in the design.

With respect to the interfaces, the design considers the use of COTS clamp band payload adapters.

10.2.2 Shielding Concept Structural Trade Off

Low mass and radiation shielding requirements are often contradictory. Therefore, in order to identify the best shielding strategy, a trade-off has been performed considering the following structural design concepts:

a) Full CFRP sandwich structure with shielding implemented in the E-boxes
b) CFRP sandwich structure with structural vault located under the high gain antenna
c) CFRP sandwich structure including small vault volumes in the columns of the S/C
d) Aluminium sandwich structure with aluminium cast columns to serve as shielding and structural support

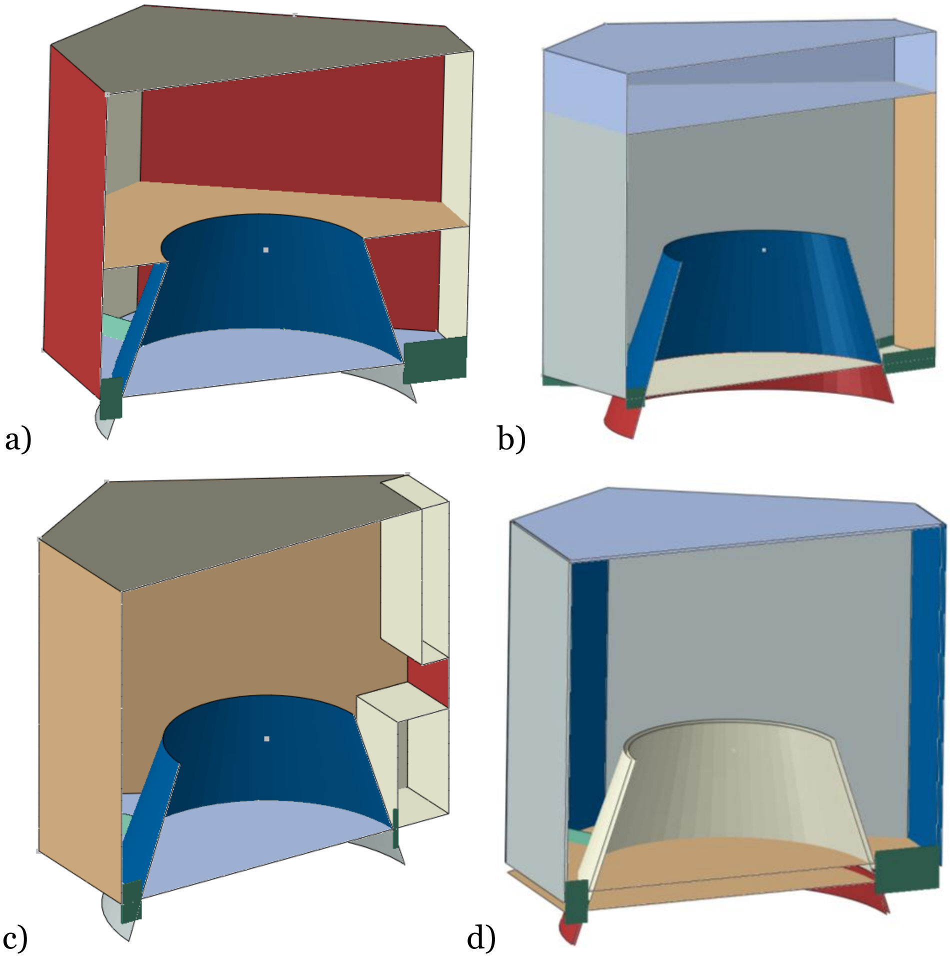


Figure 10-1: Structural concepts evaluated in the trade off

	Full CFRP	Full Vault	Mini Vaults	Al Columns
Columns	1.03	2.69	1.72	27.22
Floor	9.27	9.27	5.94	8.33
Floor Reinforcement	0.848	0.703	0.656	0.848
I/F-Adapter	5.26	4.11	3.94	5.26
Lateral Panels	2.07	4.11	4.78	6.62
Shields 12mm	0	59.99	24.55	0
Sun Floor	2.07	0	2	2.72
Tank Cone	1.89	2.58	1.89	2.58
Intermediate Floor	1.52	0	0	0
Total FEM Mass	**23.958**	**83.453**	**45.476**	**53.578**
Additional Shield mass	15.42	0	48.71	58.56
Total Structural + Shield	39.378	83.453	94.186	112.138
Total Analysis Mass	260.342	250.586	249.4714	247.9557
1st Lat Freq	**112.91**	**61.961**	**78.192**	**65.44**

Table 10-1: Trade-off table of structural concepts

From the table above it is clear that the best solution is to develop a light weight structure using CFRP sandwich technology and optimise the shielding at equipment level.

10.3 Baseline Design

10.3.1 Structure Baseline

The baseline design is composed of an interface ring attached to a stiff baseplate reinforced with radial ribs. The interface ring provides a direct load path to the propellant tank I/F cone and to the equipment intermediate panel. The lateral panels serve as a secondary load path through the baseplate to the intermediate and top panels.

This solution provides a simple and efficient load transfer from the S/C interface to the payloads.

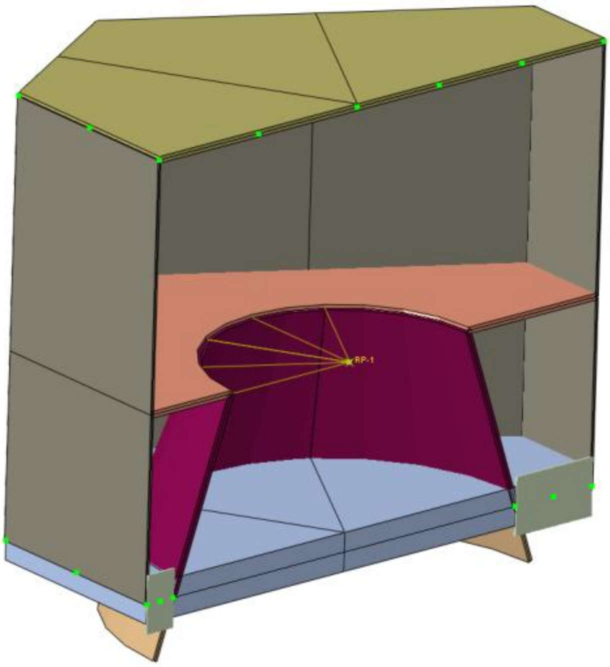


Figure 10-2: Baseline design concept

	mass (kg)	mass margin (%)	mass incl. margin (kg)
CLEO_I_Col (CLEO-I Columns)	1.03	20.00	1.24
CLEO_I_Floor (CLEO-I Floor)	9.27	20.00	11.12
CLEO_I_Floor_Rein (CLEO-I Floor Reinforcement)	0.85	20.00	1.02
CLEO_I_Int_Adap (CLEO-I Interface Adapter)	5.26	20.00	6.31
CLEO_I_Int_Floor (CLEO-I Intermediate Floor)	1.52	20.00	1.82
CLEO_I_Lat_Pan (CLEO-I Lateral Panels)	2.07	20.00	2.48
CLEO_I_Sun_Floor (CLEO-I Sun Floor)	2.07	20.00	2.48
CLEO_I_Tank_Cone (CLEO-I Tank Cone)	1.89	20.00	2.27
Grand Total	**23.96**	**20.00**	**28.75**

	Mass [kg]	Properties
Columns	1.03	0.3mm CFRP / 5mm HC
Floor	9.27	3mm CFRP / 60 mm HC
Floor Reinforcement	0.848	Al Profile [100mm x 5mm]
I/F-Adapter	5.26	8mm Al
Lateral Panels	2.07	0.3mm CFRP / 5mm HC
Intermediate Floor	1.52	0.7mm CFRP / 10 mm HC
Sun Floor	2.07	0.7mm CFRP / 10 mm HC
Tank Cone	1.89	0.7mm CFRP / 5mm HC
Total FEM mass	**23.958**	
Equipment and shielding	236.384	
Total Analysis Mass	260.342	
1st Lat Freq	**112.91**	

Table 10-2: Mass budget and properties

The figures below represent the main structural modes from the simplified structural FE model. Note that the equipment mass, 236 kg in total, are either distributed in the relevant panels as non-structural mass or concentrated in representative locations.

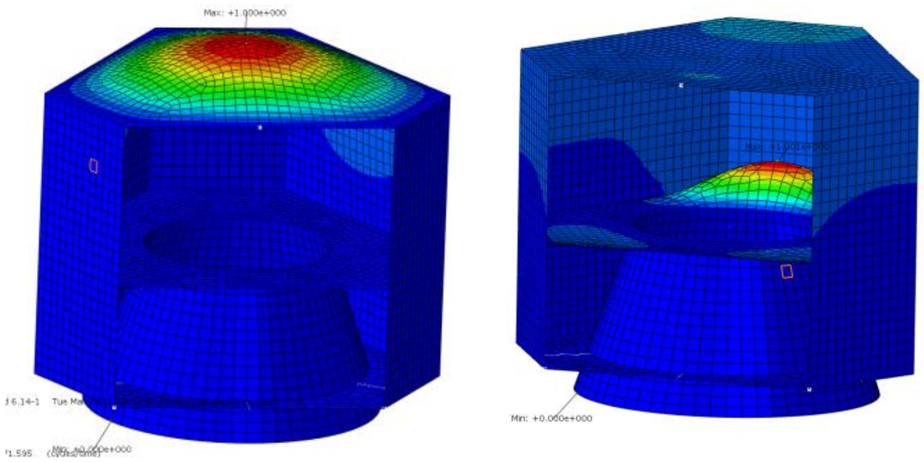


Figure 10-3: First axial and lateral mode shapes

Mode	Freq [Hz]	Type
1	71.5	Z Axial mode - Top Floor
2	105.8	Y lateral mode
3	112.9	X lateral mode

Table 10-3: Mode frequencies

10.3.2 Solar Array Attachment Points

In order to define the required number of hold down points for the solar array panels, a simplified model has been developed considering standard CFRP sandwich panels as solar cell structural support.

In this evaluation, three HDRM points has been located so that the first mode of the stack in stowed configuration is above 60Hz. Note that this analysis considers infinitely rigid interfaces. The figure above shows the definition of the stack and the first torsional mode at 64.3Hz.

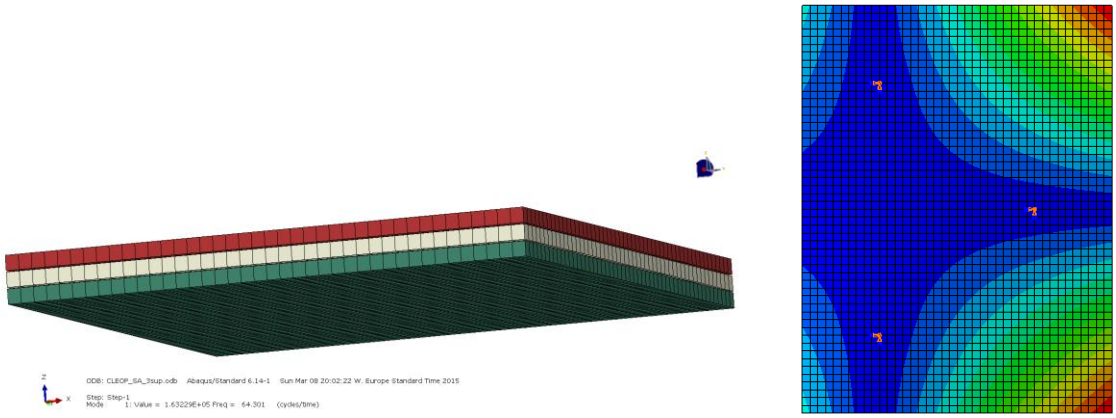


Figure 10-4: Solar array FE model and torsional mode shape

This Page Intentionally Blank

11 MECHANISMS

11.1 Requirements and Design Drivers

The main design drivers for the mechanisms are:

- Clipper-CLEO/I separation mechanism:
  - Velocity after separation (assumed 0.5 m/s)
  - Accuracy of the separation velocity
  - Mass reduction
  - Lateral position of the mechanism, perpendicular to the lunch loads, leading to bending moments at the separation plane
- Solar panel deployment mechanism:
  - Deployment of 3 solar panels per solar array wing
  - Mass reduction
- Solar array hold down and release mechanism:
  - Stowed configuration fundamental frequency
  - Mass reduction.

11.2 Assumptions and Trade-Offs

The separation delta velocity of CLEO-I from Clipper is assumed to be approximately 0.5m/s assuming the estimated mass separated for the calculations is 280 kg.

For the Hold-Down and Release mechanisms dimensioning, it has been considered that the first natural frequency in stowed configuration of the solar panels shall be higher than 60 Hz.

11.3 Baseline Design

11.3.1 Clipper-CLEO-I Separation Mechanism

The separation mechanism selected is a clamp-band of 24” from Ruag (ESS 610S).

The advantage of this mechanism is the mass reduction.

The clamp band is suitable for payloads up to 350 kg. It provides 8 springs of 4,7J energy each, however, the spring energy can be reduced to suit for the application.

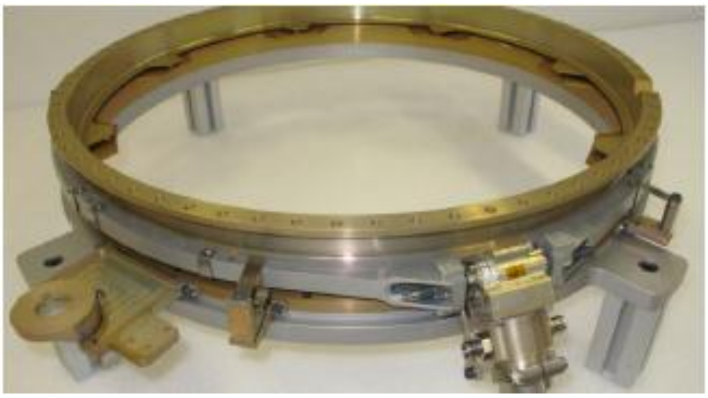


Figure 11-1: Ruag ESS 610S Clamp-band

Taking into consideration the conservation of energy and momentum for the two spacecrafts, and with m1 being the mass of Clipper, m2 the mass of CLEO-I and V1 and V2 their respective velocities, we have:

(1) $\frac{1}{2}\text{m1V1} + \frac{1}{2}\text{m2V2} = \text{N}\left[\frac{1}{2}K(\Delta L)^2\right]$

(2) m1V1 = m2V2

N is the number of springs and $\frac{1}{2}K(\Delta L)^2$ is the potential energy of the springs.

Solving equations (1) and (2) we obtain a ΔV equal to 0.5 m/s.

The mass of the clamp-band is 6.3 kg, including the spacecraft interface ring. From this mass, 4.3 kg have been considered to remain with Clipper after separation and 2 kg with CLEO/I.

11.3.2 Solar Panel Deployment Mechanism

For the deployment of the solar panels (3 solar panels per solar wing) in order to save mass, a tape spring hinge (Maeva Hinge) has been selected. This hinge has already flight heritage on Myriade satellites.

The hinge is composed of three Carpentier curved elastic strips (or tape springs). The hinge is self-actuating and self-locking, however it takes several oscillations of the tape springs to achieve the final steady state deployed position.

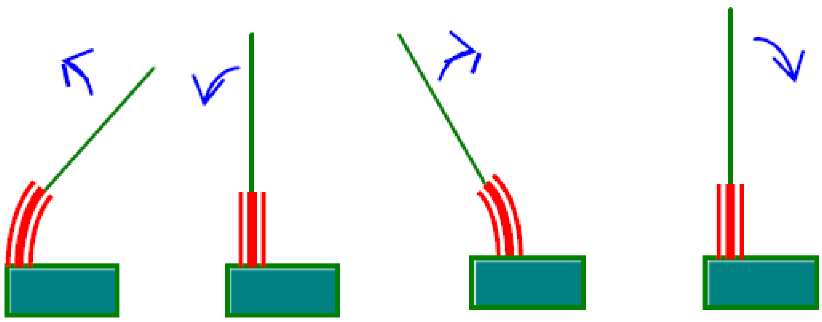


Figure 11-2: Deployment Kinematics

As there are three panels to be deployed, the solar panel configuration has to be taken into account to ensure correct opening with no interference between each of the solar panels of the solar wing.

As there are three panels to be deployed per solar array wing, and this has not yet been tested (only configurations with one or two panels has been tested), in order to avoid synchronisation problems during deployment a delta development will be needed to test and obtain the best configuration. It is then considered that the configuration of the three solar panels with Maeva hinges has a TRL 4.

The main Characteristics of the hinge can be summarised as follows:

- Mass: 90 g
- Dimensions: 0.3x0.02x0.02 m
- Temp.: -75 to 105degC

- Power: none.
- Angular position accuracy : < 1°
- Driving torque : > 0.15 N.m
- Open stiffness: 1000 N.m/rad
- Holding torque : > 4.5 N.m

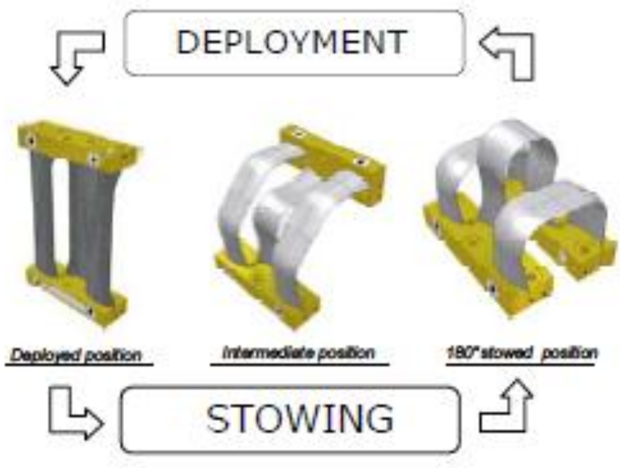


Figure 11-3: Deployment sequence

Two hinges have been considered per panel, as there are three panels per solar array wing, and there are three solar array wings, this makes a total of eighteen hinges. The reduced mass of each Maeva hinge implies a considerable reduction in mass compared with other standard spring driven hinges.

11.3.3 Solar Array Hold Down and Release mechanism

The objective of the Hold Down and Release Mechanism is to provide a stiff interface between the spacecraft and the solar array panels during launch.

A Frangibolt FC4 (Non Explosive actuator) based on SMA, has been selected as baseline. The Frangibolt will be mounted with standard cup and cone interfaces on the solar panels.

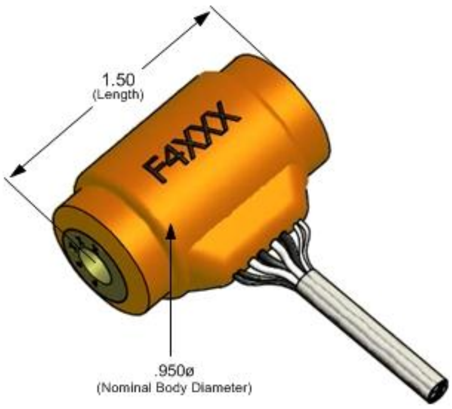


Figure 11-4: Standard FC4 Frangibolt

The Frangibolt actuator comprises a cylinder of Nitinol (Nickel-Titanium) SMA and a specially designed (integrated) heater (28 Vdc, 80 W). By heating, the SMA cylinder elongates to fracture a bolt element. At minimum temperature (-65°C) and minimum voltage of 21.5 Vdc the Frangibolt will actuate in 250 seconds. At normal voltage (28 Vdc) and –60°C the Frangibolt actuates in 150 seconds.

Mass:	50 g
Power:	80 W @ 28 VDC
Operational Voltage:	22 - 34 VDC
Current Draw:	3.0 A @ 28 VDC
Resistance:	9.7 ± 0.5 Ω
Bolt Tensile Strength:	Typical 22,241 N
Max Load Support and Release:	11,120 N
Function Time:	Typical 35 sec. @ 28 VDC
Reusable:	By Re-Compressing Actuator
Life:	60 Cycles MIN
Operational:	-65° C to +80° C

Table 11-1: Frangibolt FC4 Specifications

It has been assumed that the first eigen-frequency of the solar panels in stowed configuration shall be higher than 60Hz. To achieve this value, three Hold Down and Release Mechanisms will be needed in the configuration shown in Figure 11-5. For the analysis it has been rigid Hold Down and Release Mechanisms and no additional stiffness from the hinges.

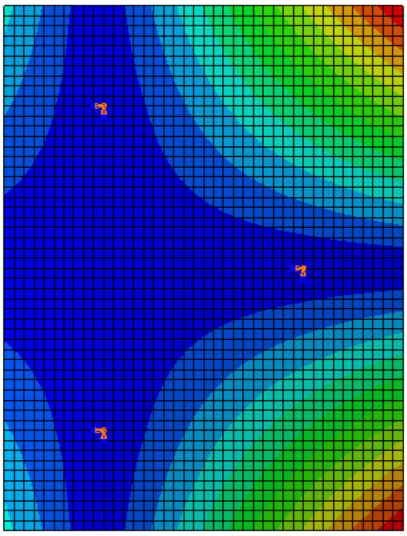


Figure 11-5: Solar panel analysis

The first mode obtain in the analysis is 64,3 Hz and the second mode 144,83 Hz.

11.4 List of Equipment

	mass (kg)	mass margin (%)	mass incl. margin (kg)
⊞ HDRM_1 (Solar Array HDRM)	0.35	10.00	0.39
⊞ HDRM_2 (Solar Array HDRM)	0.35	10.00	0.39
⊞ HDRM_3 (Solar Array HDRM)	0.35	10.00	0.39
⊞ HDRM_4 (Solar Array HDRM)	0.35	10.00	0.39
⊞ HDRM_5 (Solar Array HDRM)	0.35	10.00	0.39
⊞ HDRM_6 (Solar Array HDRM)	0.35	10.00	0.39
⊞ HDRM_7 (Solar Array HDRM)	0.35	10.00	0.39
⊞ HDRM_8 (Solar Array HDRM)	0.35	10.00	0.39
⊞ HDRM_9 (Solar Array HDRM)	0.35	10.00	0.39
⊞ SA_DH_01 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_02 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_03 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_04 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_05 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_06 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_07 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_08 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_09 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_10 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_11 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_12 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_13 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_14 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_15 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_16 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_17 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SA_DH_18 (SA Deployment Hinge)	0.09	10.00	0.10
⊞ SDM (Satellite Deployment Mechanism)	2.00	10.00	2.20
⊞ SDM_Clipper (Satellite Deployment Mechanism Clipper)	4.30	10.00	4.73
Grand Total	11.07	10.00	12.18

Table 11-2: List of Equipment

11.5 Options

As an option, a Solar Array Drive Mechanism (SADM) has been considered to rotate the panels and thus keep the solar panels sun pointing during JC mode. The SADM will be attached to the spacecraft through the deployment hinges and on the other side to the first solar panel.

A Septa 41 from Ruag, with a maximum power transfer of 600W, has been considered for this option, with three units needed (one SADM per solar wing).

The mass of each unit is 1.7 kg, the total mass of the 3 units 5.1 kg.

The power consumption of each unit is 4.1W.

Figure 11-6: SADM Septa 41

11.6 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
Clamp-Band	Fast-Acting Shockless Separation Nut	TRL6		
Maeva Hinge	Tape springs	TRL 9		TRL 4 for the configuration of three solar panels
Hold Down and Release Mechanism	Frangibolt (SMA)	TRL 9		TRL 6 for the Solar Panel configuration
SADM (option)	Electric motors	TRL 9		

12 PROPULSION

12.1 Requirements and Design Drivers

The following requirements are applicable to the CLEO/I baseline mission (Scenario 2b). Additional requirements for alternative scenarios are included below the table.

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
PROP-010	Provide Δv for main manoeuvres and AOCS requirements	
PROP-020	Include redundant functionality for all thrust commands	
PROP-030	AOGNC needs a thruster class of 1N to fulfil the mission needs	
PROP-040	Radiation influence on the propellant must be known	
PROP-050	Smallest tank size as possible to save mass	
PROP-060	Scenario 2b requirements shall be considered for designing the propulsion subsystem (Δv, AOCS, flyby's,…)	

12.1.1 Additional Requirements for the Different Options

Option 1: Europa flyby:

The requirements for the Europa flyby mission are:

- Same AOCS requirements and same propellant needed as for scenario 2b
- Same stabilising mode of the satellite during the different mission phases
- Lower Δv requirement but same thruster configuration.

Option 2: Io flyby Backup mission (Scenario 4 see 6.7.1)

- Same AOCS requirements and same propellant needed for this mission
- Same stabilising mode of the satellite during the different mission phases
- Lower Δv requirement, only 1N thrusters assumed.

12.2 Assumptions and Trade-Offs

The assumptions for the calculations are:

- Hydrazine as monopropellant system
- Propellant density set to 1.01kg/l
- Linear decrease of Isp in relation to decrease of tank pressure
- Large Δv manoeuvre done by "big" engine (20N). The redundancy is achieved by means of the AOCS thrusters
- Calculation of wet mass using the given dry mass for the satellite
- 4 large Δv manoeuvres, in between the manoeuvres where AOCS is needed
- Diaphragm tanks for the propellant
- Propellant is kept within the nominal range of temperature for usage
- No equipment for draining of the propellant at end of life, only passivation using pyrovalves included.

Within this study, the following trade-offs have been performed:

- Comparing bipropellant system (MON/MMH with MR of 1.65) in comparison to hydrazine. Assuming an overall mass of 250 kg of the satellite and a Δv requirement of 415m/s, the bipropellant wet mass was higher than the monopropellant system
- Using ITAR-thruster (Aerojet MR-103G and MR-106E 22N thruster). The detailed results are shown within the Options section.

12.3 Baseline Design

The baseline consists of a monopropellant system based on hydrazine. The pressurant gas for the diaphragm tank is assumed to be helium. The tank pressure is observed by means of pressure transducer with a redundant transducer in the hydrazine part. This second transducer can also be placed downstream to enable an observation of the pressure upstream of the main engine.

Prior to the mission, the complete propulsion system is passivated by using a normally closed pyrovalve. The mission starts by opening this valve. Each scenario needs to open the corresponding latch valve for propellant flow. After this, the corresponding flow control valves at each thruster are used for firing. The time itself for firing is assumed during the different mission phases. This results in a power estimation for the different mission phases by means of dividing the firing time by the overall time.

At end of life, the residual propellant will stay within the tank and the normally open pyrovalve will be fired. The last burn is planned to bring the satellite in a stable position (orbit or deorbiting on Europa or Io) and therefore the residual propellants are assumed to be very small.

Within the propellant branch, the only filter for the propellant exists after the fill & drain valve. The filling and draining of the propulsion system is done through the filter directly in front of the fill and drain valves. The influences on the propellant (particles,…) has to be assessed in a proper manner but the mass saving here has been considered as necessary.

The following figure demonstrates the propulsion system layout of the baseline design. The only difference within the lower Δv-requirement mission (Scenario 4) is that the 20N thruster will not be used.

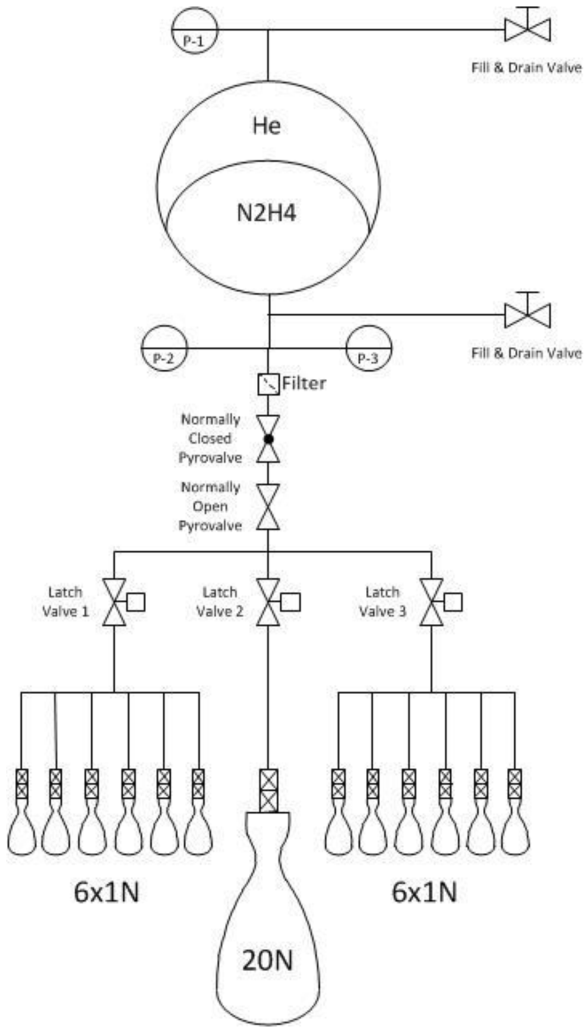


Figure 12-1: Propulsion system

12.4 List of Equipment

The equipment for the baseline is summarised in Table 12-1. Additionally, the mass margin and the corresponding masses are shown.

	mass (kg)	mass margin (%)	mass incl. margin (kg)
Fill & Drain valve Fuel	0.07	5.00	0.07
Fill & Drain valve Pressurant	0.05	5.00	0.05
Feed line	5.00	10.00	5.50
Latch Valve #1	0.55	5.00	0.58
Latch Valve #2	0.55	5.00	0.58
Latch Valve #3	0.55	5.00	0.58
NC Pyro Valve	0.29	5.00	0.30
NO Pyro Valve	0.32	5.00	0.33
Propellant Filter	0.11	5.00	0.12
Propellant Tank (ATK DS512)	6.01	5.00	6.31
Pressure Transducer #1	0.25	5.00	0.26
Pressure Transducer #2	0.25	5.00	0.26
Pressure Transducer #3	0.25	5.00	0.26
Airbus CHT-1 #1	0.30	5.00	0.31

Airbus CHT-1 #2	**0.30**	**5.00**	**0.31**
Airbus CHT-1 #3	**0.30**	**5.00**	**0.31**
Airbus CHT-1 #4	**0.30**	**5.00**	**0.31**
Airbus CHT-1 #5	**0.30**	**5.00**	**0.31**
Airbus CHT-1 #6	**0.30**	**5.00**	**0.31**
Airbus CHT-1 #7	**0.30**	**5.00**	**0.31**
Airbus CHT-1 #8	**0.30**	**5.00**	**0.31**
Airbus CHT-1 #9	**0.30**	**5.00**	**0.31**
Airbus CHT-1 #10	**0.30**	**5.00**	**0.31**
Airbus CHT-1 #11	**0.30**	**5.00**	**0.31**
Airbus CHT-1 #12	**0.30**	**5.00**	**0.31**
Airbus CHT-20	**0.40**	**5.00**	**0.41**
Grand Total	**18.17**	**6.38**	**19.33**

Table 12-1: Equipment summary

Using the values for the Δv and the following assumption for the AOCS and the corresponding manoeuvres, the propellant can be calculated. Within the following table, the manoeuvres, the mass at begin of the manoeuvre, the mass at the end of the manoeuvre, the velocity increment, the propellant mass needed for this manoeuvre, the tank pressure at the beginning of the manoeuvre and the corresponding firing time using one engine is shown. The given values for Δv and propellant mass (AOCS manoeuvres) are underlined. The values for the velocity increment of the AOCS manoeuvre are only for information since this kind of firing is not assumed to achieve a Δv on the satellite.

Manoeuvre	mass begin [kg]	mass end [kg]	velocity increment [m/s]	propellant mass [kg]	tank pressure [bar]	Firing time [s]
1	265.2	235.4	262.50	29.86	34.5	4146
2	235.4	235.2	1.24	0.14	14.4	394
3	235.2	232.3	27.30	2.98	14.4	433
4	232.3	232.1	1.25	0.14	13.6	414
5	232.1	227.8	40.00	4.32	13.5	666
6	227.8	227.8	0.27	0.03	12.6	95
7	227.8	226.1	15.75	1.68	12.5	265
Summation			**348.31**	**39.15**	**12.2**	**6412**

Table 12-2: Summary of propellant usage

Within this calculation, the pressurant mass of 0.123 kg helium was not mentioned.

Due to the redundancy concept for the 20N thruster, an additional calculation using only the 1N thruster has been performed. This leads to the following mass budget:

manoeuvre	mass begin [kg]	mass end [kg]	velocity increment [m/s]	propellant mass [kg]	tank pressure [bar]	Firing time [s]

1	266.4	235.7	262.50	30.69	34.5	89899
2	235.7	235.6	1.23	0.14	13.7	411
3	235.6	232.5	27.30	3.09	13.7	9521
4	232.5	232.3	1.24	0.14	12.9	432
5	232.3	227.9	40.00	4.48	12.9	14763
6	227.9	227.8	0.27	0.03	11.9	99
7	227.8	226.1	15.75	1.75	11.9	5902
Summation			**348.29**	**40.32**	**11.53**	**121026**

Table 12-3: Summary of propellant usage (redundant branch)

The comparison shows that the propellant needed to fulfil the mission is 1.2 kg more with the redundancy concept for the large thruster.

Additionally, the overall firing time of the thruster is quite large in comparison to the baseline design. The constraints for the 1N thruster of Airbus (CHT-1) is currently a single burn of 12 hours (43200s). Assuming two thruster, the longest duration for one single fire burn is 86400s and therefore lower than the firing time needed for manoeuvre 1. This would need a delta qualification for this purpose or the split of this manoeuvre into two with a non-firing time in between. But the overall firing time of one thruster is currently no limitation because the thruster is qualified up to 50hrs of firing (180000s).

Due to the possible need of an additional delta-qualification of the 1N thruster for the redundancy concept, a second calculation after the study with a redundant 20N thruster was performed. This leads to a minimum increase of 1 kg of dry mass (the increase of the mass for the feeding line was assumed to be zero, mass of thruster plus latch valve). This leads to the following mass for the design with a redundant 20 N thruster:

Manoeuvre	mass begin [kg]	mass end [kg]	velocity increment [m/s]	propellant mass [kg]	tank pressure [bar]	Firing time [s]
1	266.4	236.4	262.50	30.00	34.5	4191
2	236.4	236.3	1.23	0.14	14.3	397
3	236.3	233.3	27.30	3.00	14.3	438
4	233.3	233.1	1.24	0.14	13.5	416
5	233.1	228.8	40.00	4.34	13.4	673
6	228.8	228.8	0.27	0.03	12.4	95
7	228.8	227.1	15.75	1.69	12.4	268
Summation			**348.30**	**39.33**	**12.09**	**6479**

Table 12-4: Summary of propellant usage (redundant branch of 20 N Thruster)

12.5 Options

There have been several options which have been investigated. The first one is the shift to ITAR components which have a higher Isp value. Using the ITAR-thruster from Aerojet (MR-103Gand MR-106E 22N thruster) will lead to a higher dry mass of the system due to higher masses of the thruster in comparison to the Airbus thruster, but they have a slightly better performance. Therefore, the following table summarises the options and the corresponding system masses. The redundant concept of using the available 1N thruster to compensate a malfunction of the 20N thruster is leading to a higher propellant mass because the Isp of these thrusters are lower. Using the concept of a redundant 20 N thruster is increasing dry mass, but the Isp of the thruster is higher and therefore the wet mass of both systems are comparable (using the 20 N thruster is theoretically 30g heavier).

	Propellant mass	Subsystem mass	Dry Mass	Wet mass
CHT-20 + CHT-1	39.1	18.2	226.1	265.2
-redundant	40.3	18.2	226.1	266.4
-redundant CHT-20	39.3	19.2	227.1	266.4
MR-106E 22N+MR103G	37.3	18.9	226.8	264.1
-redundant	40.8	18.9	226.8	267.6
MR-106E 22N + CHT-1	37.2	18.4	226.3	263.5
-redundant	40.4	18.4	226.3	266.6
CHT-20 + MR-103G	39.2	18.7	226.5	265.7
-redundant	40.8	18.7	226.5	267.3

Table 12-5: Summary of masses for the different options

The results for the wet mass are shown within Figure 12-2 graphically. It can be seen that the option using the 22N thruster of Aerojet could decrease the overall mass about 1.6kg, but the redundant option is 0.3 kg heavier than using the European thruster. Nevertheless, the baseline was chosen due to the preferred usage of non-ITAR components.

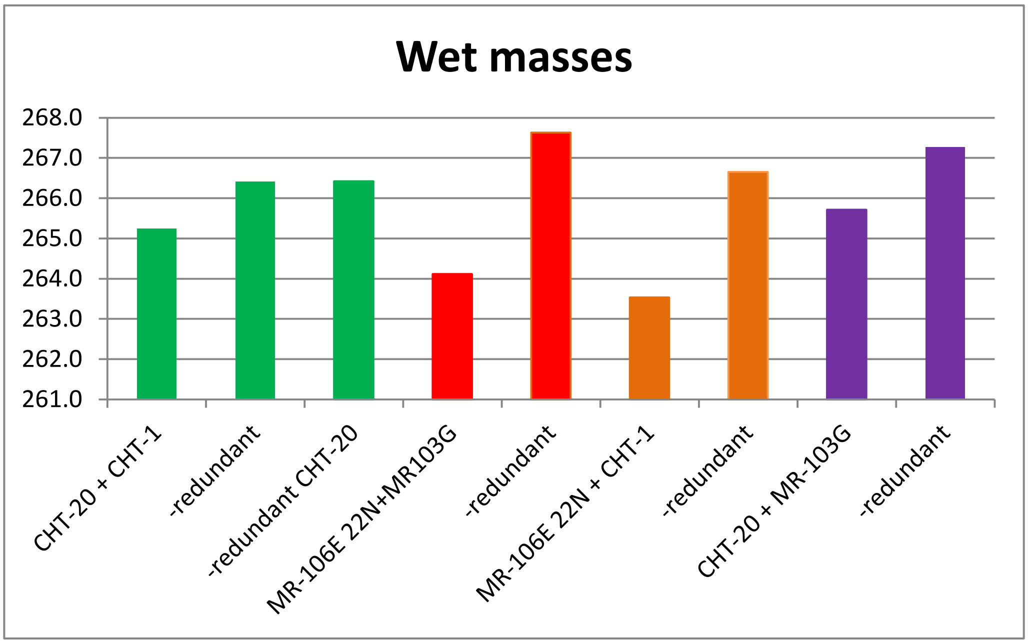


Figure 12-2: Comparison of the different options

Europa mission:

For the Europa fly-by mission, the requirement for the Δv was lower (170m/s for the main manoeuvres and AOCS). Given the lower Δv, the tank size was able to be reduced and the resulting dry mass of the propulsion system was reduced to 15.88 kg. The following table summarises the main results for the different manoeuvres again:

Manoeuvre	mass begin [kg]	mass end [kg]	velocity increment [m/s]	propellant mass [kg]	tank pressure [bar]	Firing time [s]
1	250.0	238.4	102.41	11.62	22.1	1990
2	238.4	238.2	1.20	0.14	10.9	496
3	238.2	233.6	40.96	4.60	10.8	903
4	233.6	233.5	1.20	0.14	9.0	578
5	233.5	231.2	20.48	2.28	9.0	474
6	231.2	231.2	0.26	0.03	8.3	132
7	231.2	230.5	6.14	0.68	8.3	144
Summation			**172.65**	**19.49**	**8.13**	**4717**

Table 12-6: Summary of propellant usage for Europa mission

Using the redundant branch for the 20N thruster, the increase of mass is calculated to be 0.65 kg. Therefore, neglecting the 20N thruster and the corresponding latch valve, the mass saving is about 0.945 kg. In a detailed study with the calculation of a target mass instead of assuming the initial mass, it could be mass saving not to have the 20N thruster for this mission. This needs then a further assessment.

Scenario 4 with a low Δv requirement for Io:

The Δv requirement for this option has been set to be 40m/s for the main manoeuvres. The AOCS mass was as before 0.31 kg split into 0.14, 0.14 and 0.03 kg. Due to the lower Δv, the tank chosen (ATK DS222) and the discarding of the 20N thruster leads to a dry mass of 11.33 kg. All other data are again presented within the following table:

Manoeuvre	mass begin [kg]	mass end [kg]	velocity increment [m/s]	propellant mass [kg]	tank pressure [bar]	Firing time [s]
1	163.4	161.5	25.00	1.87	27.6	4398
2	161.5	161.4	1.84	0.14	17.9	333
3	161.4	160.8	8.00	0.61	17.4	1598
4	160.8	160.6	1.83	0.14	15.7	370
5	160.6	160.2	6.00	0.46	15.3	1304
6	160.2	160.2	0.39	0.03	14.3	85
7	160.2	160.0	2.00	0.15	14.2	446
Summation			**45.06**	**3.41**	**13.90**	**8535**

Table 12-7: Summary of propellant usage for Scenario 4

12.6 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
Propellant (option)	High Performance Green Propellant	ECAPS, TRL 9 (1N) and 5-6 (20N)		This could enhance the propulsion system due to a higher Isp of the propulsion systems.
Propellant (option)	AF-M315E	Aerojet, TRL ?		This is a green propellant monopropulsion system and has a higher Isp than currently hydrazine. This could enhance the propulsion system.

13 ATTITUDE CONTROL SYSTEM

13.1 Requirements and Design Drivers

13.1.1 Functional Requirements

The following functions are required from the AOGNC subsystem:

- Fine Attitude pointing 3 axes stabilised during fly-by (duration 300 min) – science
- Coarse Attitude pointing during the Cruise phase (duration max 190 days) – communication
- Perform Orbit Control Manoeuvre ΔV after separation
- Implement redundancy and reliable AOGNC safe mode.

13.1.2 Performance Requirements

Performance requirements are slightly different during the orbit phases.

1. During the Science Mode (SCM) the pointing requirement is driven by the pointing accuracy of the camera. The Orbiter symmetry axis shall be kept Nadir pointed with APE < 10' (arcmin).
2. During the Communication Mode (DTE) the pointing requirement is driven by antenna pointing to Earth for communication. Antenna boresight shall be pointed to Earth with APE < 0.1deg.

In both cases the AOGNC is requested to be inertial pointed to target defined by Ground.

13.1.3 Understanding of Requirements

The AOGNC requirements do not suggest the need for a highly accurate pointing system, and the main design drivers are the minimisation of mass and power consumption together with reliability in harsh radiation environment.

Furthermore, the duration of science acquisition is much shorter than the cruise phase, meaning that the fine attitude pointing is only required for limited time periods. This leads to the consideration of two options for the design:

- Reaction Wheels based science/cruise mode and RCS based Orbit Control Mode
- Full RCS based AOGNC covering OCM, science and cruise. For this option the case of spin stabilised cruise shall also be considered.

The trade-off among the different options is reported below.

13.2 Assumptions and Trade-Offs

The major trade-off at AOGNC level has the objective to minimise mass and power consumption. The trades are focused on two aspects:

- Actuator architecture: Reaction Wheels (RWL) vs Thrusters (THR)
  - The selection will also consider solution of 3-axes stabilised and spin stabilised spacecraft for the cruise phase, outside science acquisition
- THR layout: Attitude Control THR's + Main THR (ΔV) vs AOC THR's performing also ΔV

- The first option will also consider the number of AOC thrusters, with different solutions between 8, 6 or 4 AOC thrusters (0.5N or 1N) in addition to the main engine (20N).

13.2.1 Assumptions for the Trade-Off

13.2.1.1 Physical properties

The spacecraft has a triangular shape with dimensions included in a volume of cylinder 1m diameter by 1m height. The total mass of the spacecraft is in the order of 270kg.

13.2.1.2 Environmental Disturbance torques

The disturbance torques in the designed orbit have been estimated using the AOGNC_Workbook tool and the total contribution from magnetic residual, solar pressure and gravity gradient has been estimated being in the order of 1E-7 Nm.

13.2.1.3 Mission timeline

The manoeuvres to be performed by the spacecraft during its lifetime are summarised below, they are considered in the trade-off to derive the sizing of actuators and required propellant mass.

The orbit is split into two main parts:

- Fly-by: lasting 300 min where science is performed and fine pointing is required. This phase shall be 3 axes stabilised, nadir pointed following the rotation about the surface. In the case of the thrusters, only the control strategy is assumed as PD with limit cycle to keep the pointed axis at Nadir within required range of 10 arcmin. 2 fly-by are foreseen during lifetime.
- Cruise phase: lasting 100 days for the first one and 190 days the second one. During the cruise phase the spacecraft shall implement two operative modes: DTE (Direct to Earth) communication where coarse inertial pointing is required and JC (Jovian Cruise) where Sun pointing is required. Being that communication is the objective of this phase, the spacecraft can be either 3 axes stabilised or spin stabilised, with possible saving of propellant to hold the pointing.

13.2.1.4 Sun – spacecraft – Earth angle evolution

Another player in the trade-off is the consideration about the number of manoeuvres to be performed during the Cruise phase, in order to keep the spacecraft Earth pointed during DTE and Sun pointed during the JC.

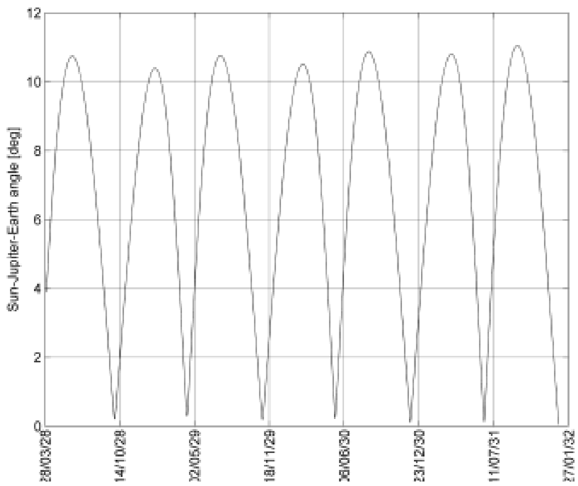


Figure 13-1: Sun/Jupiter(≈Spacecraft)/Earth angle

The drift of the Earth in case of inertially fixed pointing can be estimated from Figure 13-1 as about 0.12deg/day. This is the amplitude of the slew to be performed during cruise between two consecutive DTE slots.

In addition to this slew the system shall compensate for the external disturbance torque, assumed as worst case always in the wrong direction during the entire cruise phase.

13.2.1.5 Spin rate

In the case of spin stabilised spacecraft, the contribution of external disturbance torques on the pointing drift will have the effect of angular momentum drift H_{drift}, that depends on the spin rate. The acceptable drift shall be such that during the DTE (2.8h) the pointing to Earth (0.1deg) is not lost. With spin rate of 1rpm the angular momentum drift is contained within 0.1deg /day. This value ensures to passively keep the pointing during DTE and at the same time does not require high amounts of propellant to spin-up/spin-down.

Therefore in the case of spin stabilised spacecraft during cruise, the slews required to perform the communication with Earth will be in the order of 0.25deg/day considering both disturbance effects and Sun-S/C-Earth angle evolution (0.12deg/day as per section 13.2.1.4). This number will be considered in the propellant budget.

The spin-up/spin-down manoeuvre shall be performed 4 times during the spacecraft lifetime, i.e. at each entry exit from science mode during the fly-by.

13.2.2 Actuator Architecture Trade-Off

13.2.2.1 RWL

Reaction wheels sizing is derived by the estimated disturbance torques and slew needs during science mode. With the assumptions reported in sections above, the following sizing case is obtained:

TORQUE ≈ 0.5mNm, MOMENTUM ≈ 0.28Nms

Several solutions have been identified among those available on the market.

	TRL	Mom	Torque	mass	power
MSCI MW1000	Flying	1.1Nms	30mNm	1.4kg	9W @max speed
SINCLAIR	(FF 2014)	1Nms	100mNm	0.97kg	?
SSTL 100SP	(FF 2014)	1.5Nms	110mNm	2.6kg	10W @5000rpm
ASTRO RW150	Not flying	1Nms	30nMm	1.5kg	5W @constant

Figure 13-2: RWL solutions

The main advantage of the RWL is that no propellant is required to hold the attitude pointing or to perform the slews; some propellant is however needed for wheels off-loading at the end of the fly-by. The drawbacks are the mass (minimum configuration of 4 RWL's requires at least 5.6 kg) and the power consumption, where during active phases the wheels requires up to 36W continuously to be operated.

13.2.2.2 THR

The solution based on thrusters does not imply any additional dry mass, since the thrusters are present also in the RWL based science mode. Different layouts have been considered in the trade-off, looking in two different aspects:

1. Main engine for ΔV vs AOC thrusters only
2. 6 AOC thrusters vs 4 AOC thrusters

13.2.2.2.1 Main engine vs AOC thrusters based ΔV

The parameter on which the trade-off is based is the specific impulse Isp. The available options are:

- Main engine: 20N thruster (in addition to AOC thrusters), mounted aligned with CoG nominally providing torque-free force with efficiency of 100%, Isp=230s, additional dry mass ≈ 0.6kg.
- AOC 1N thrusters only, mounted symmetrically aligned with CoG (requires 6 AOC thrusters) nominally providing torque-free force with efficiency of 100%, Isp=200s.

The budget allocated to ΔV is 331m/s. Looking at the two possible solutions (only wrt mass impact), the propellant required is:

- Propellant using 20N THR ≈ 38.7kg
- Propellant using 1N THR's ≈ 39.8kg

As the difference in required propellant mass is bigger than the mass of the thruster itself, the baseline selected foresees the presence of the main engine 20N thruster.

Note that the budget does not consider additional propellant to compensate for 20N thruster misalignment disturbance, assuming AOC thrusters mounted such to provide desired torque with forces in the direction of ΔV (requires 6 AOC thrusters).

Additional advantage comes from the duration of the manoeuvre that is 5 times less with main engine wrt the AOC thrusters solution because of different thrust level.

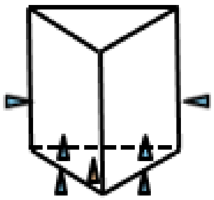
13.2.2.2.2 6 AOC thrusters vs 4 AOC thrusters

Having selected the baseline with the main engine, AOC thrusters are then used to hold pointing during fly-by, to perform slews during Cruise and losses due to compensation for disturbance torques generated by main engine during ΔV (case of 4 AOC thrusters only).

The two solutions with relevant budgets are reported below.

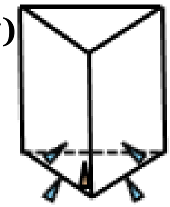
1. **(6m+6r)x1N ACS THR and 1x20N DV THR**
   a. AOC RCS mass: 3.6kg
   b. Propellant for AOCS: 0.32kg
   c. ΔV disturbance comp. 0kg

   TOTAL mass (AOC RCS): 3.92kg



2. **(4m+4r)x1N ACS THR and 1x20N DV THR(*)**
   a. AOC RCS mass: 2.4kg
   b. Propellant for AOCS: 1.022kg
   c. ΔV disturbance comp. 0.77kg

   TOTAL mass (AOC RCS): 4.20kg



(*) better alignment required to have enough authority to compensate during ΔV

The solution 1 is therefore the most favourable in terms of overall mass and provides additional margin wrt the acceptable misalignment of main engine and associated disturbance torque.

13.2.2.3 Conclusion

The trade-off between Reaction Wheels and Thruster based AOGNC during the fly-by, considering mass and power as driver requirements, led to the selection of a thrusters only based architecture, comparing the required propellant mass of 0.32 kg including margin and the RWL mass of 5.6 kg for RWLs.

13.3 Baseline Design

The AOGNC foresees the following Operative Modes, during the various phases of the mission:

- Science Mode (SCM): implemented during the fly-by. The spacecraft is 3-axes stabilised, keeping the Nadir pointing to the planet with an accuracy of APE=10'. This mode uses the AOC thrusters as actuators performing a limit cycle PD control within the required band. The SCM includes also two sub-modes, the Spin-down before starting science fine pointing and the Spin-up at the end of science fine pointing. At the end of fly-by, the AOGNC switch to Cruise (DTE/JC) mode (spin stabilised).
- Direct to Earth (DTE) mode: during communication windows the AOGNC is spin stabilised. At the beginning of DTE the AOGNC performs slew to point Earth with accuracy of APE=0.1deg before starting communication. The slew is performed during the spin, as detailed in section 13.3.1 below.
- Jovian Cruise (JC) mode: during battery charging the spacecraft is spin stabilised and the AOGNC does not perform any manoeuvre.
- SAFE Mode: in the case of major failure leading to Safe Mode, the AOGNC shall switch on redundant branches (both actuators and sensors) and enter in coarse Earth pointing safe attitude (from AOGNC it is equivalent to DTE with redundant units and with relaxed pointing requirement of 1deg). In the case the Safe Mode triggers during Science, a slew/spin-up sequence shall be performed, otherwise, if the Safe is triggered during Cruise, the AOGNC will not perform any manoeuvre.

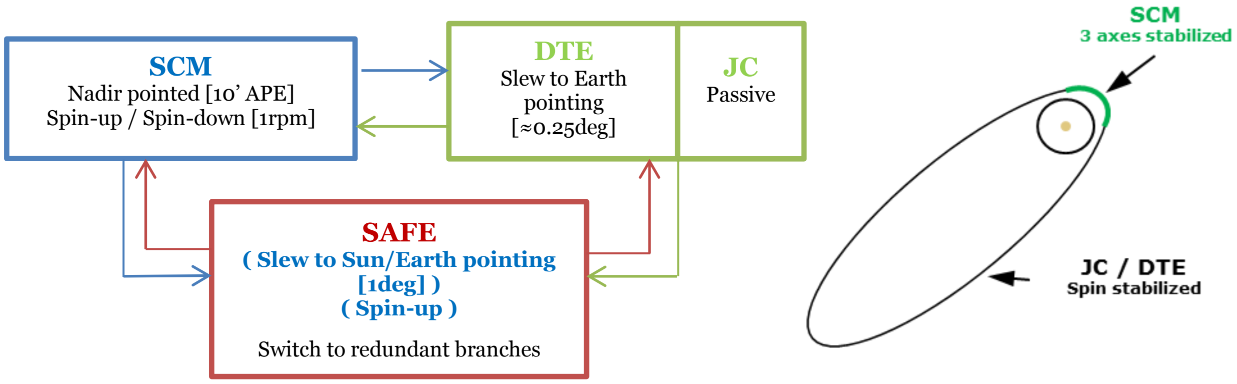


Figure 13-3: AOGNC Modes during orbit phases

CLEO/I pointing strategy assumed being always Earth pointed during Cruise (DTE/JC).

The expected Earth and Sun vectors evolution wrt on-board time during the mission will be pre-loaded on-board from Ground (and eventually updated during the lifetime) such that at any time it will be possible to use Sun Sensor and GYR to point the spin axis to the expected Earth direction (equivalent to expected angles wrt the Sun).

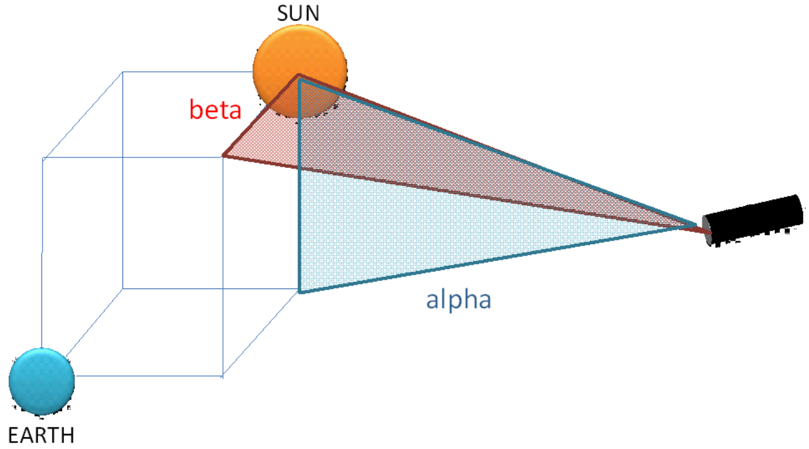


Figure 13-4: Sun Spacecraft Earth Angle

13.3.1 Strategy for Slew During Spin Stabilised Cruise

The selected AOGNC strategy during cruise, when the spacecraft is spin stabilised is based on angular momentum re-pointing. The system aligns the Spacecraft H vector with the target vector and the spin vector in two steps with 2 thrusts during one revolution period.

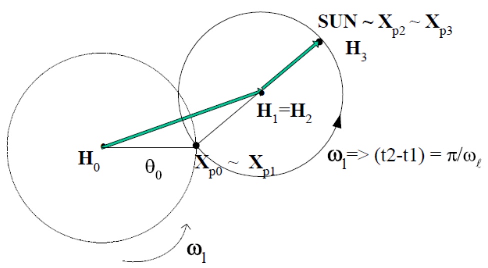


Figure 13-5: Slew strategy in spin

The first thrust moves the H vector in the middle between the current pointing and the target vector. Then, during revolution of symmetry axis around the H vector, when the two vectors (symmetry and target) are aligned, the second thrust will move the H vector such that the pointing is reached and nutation is cancelled.

Such strategy has been used on Planck spacecraft and it minimised the usage of propellant.

If the slew is larger (i.e. to point Earth in Safe Mode during Science) it is completed in a sequence of smaller steps, performed with the thruster.

13.4 List of Equipment

The list of baseline AOGNC equipment includes only sensors, as the actuation is based only on thrusters (as per trade-off results) and relevant description 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 external processing, i.e. with the spacecraft OBC acting as sensor data processing unit. This led to mass saving for sensors' electronics.

13.4.1 Star Tracker

The selected STR is the SODERN HYDRA STR Optical Head. Two OH are mounted for redundancy reason and the processing function (algorithms for attitude determination) is performed by the spacecraft OBC.

The unit main performance are: bias=11arcsec, NEA=2arcsec/√Hz(XY)/15arcsec/√Hz (Z) that are fully in line with the pointing requirements of this mission. Note : these values assume only 1 OH works at the same time.

Figure 13-6: STR OH SODERN

The mass of a single Optical Head is 1.4kg, while the power consumption when operative is 2.5W.

The STR is used for the inertial pointing during the science mode and for the attitude measurement during slews in cruise mode. This unit is able to operate with angular rates up to 8deg/s, compatible with the selected spin rate of 6deg/s.

Star Trackers in general (due to the degradation of the detector performance) are quite sensitive to harsh radiation environment but detailed investigations and analyses performed for the JUICE mission have allowed optimizing the design of this STR for such environments – only limited changes in the environment are expected for CLEO/I compared to Europa flybys (higher dose). *The impact of single event effects is also a major aspect, in order for the Star Tracker to be able to reach the main mission mode (tracking) even if case of very important flux of particles (highly energetic electrons).*

Details and suggestions on this aspect have been reported below.

13.4.1.1 STR in harsh Radiative Environment

The following sections provide an overview of the radiation effects in imaging detectors, for further details refer to RD[25].

13.4.1.1.1 Ionizing dose effects:

Ionization damage: One of the major effects of radiation on MOS devices is threshold voltage shift, which can result in improper bias conditions or degraded noise margin and increased power consumption. Sufficiently large threshold can result in functional failure. This effect can vary enormously from one manufacture to another and even from one lot to another.

The second effect is the increase of surface generated dark current, which might exceed the signals to be detected, resulting in functional failure. Even if less than the signal, it contributes seriously to noise of component.

Displacement damage: this is a problem of the CCD where it can delay the shifting of charges from one pixel to another, reducing the charge transfer efficiency (CTE).

These effects shall be considered for bias and temperature, both during and after irradiation. It is recommended to characterise these effects (with specific tests and assuming sufficient margin) for the selected manufacturer, possibly on one unit of the same lot as the flight unit.

For the detector degradation, it is recommended to implement in the on-board algorithms a system to keep trace of bright pixels (permanent spikes) and to make use of attitude information by GYR to propagate frame to frame attitudes and compare the spots found on one frame to others.

13.4.1.1.2 Particles impinging on the detector/optics:

In the vicinity of proton hit it is impossible to determine a star centroid, while for electron hit, noise will be added to the star centroid position estimation.

Darkening of glass: Ordinary glass darkens due to irradiation. Typically glasses begin to darken after a few krad of irradiation, proportionally to the number of certain impurities present.

The particles impinging on optics result in a 'glow' creating luminescence and radiation, that shall be included in the design considerations.

The use of refractive design can provide large field of view, allowing the use of brighter stars. However, refractive elements comes with luminescence and radiation adding background signal, proportional to the image exposure time. With increase of background the pixel to pixel response variation becomes an even larger contributor. One way to reduce this effect would be to calibrate the individual pixel response or to provide real-time background estimates on a pixel by pixel basis. The possibility of limiting the exposure time shall also be investigated.

Specific shielding shall be included surrounding the detector, to limit the number of proton hits that causes measurement and detection impossible. Behind this shielding the number of events shall be much limited. The hit of electron flux is expected to be the main problem. It can cause a centroid bias or detection failure. Different measures shall be taken to limit this effect: use only bright stars in selected fields, or smaller pixels with smaller star image. Algorithms to determine if a signal caused by impinging charged particle shall also be implemented (e.g. looking at charge accumulation rate).

13.4.1.1.3 Conclusion and recommendations

Finally, there might be short periods of time where the flux becomes too high; in those cases the attitude can be propagated by GYR measurements. The careful estimation of duration of such transients could eventually lead to selection of higher performance class GYR, accounting for impacts on mass and power.

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, while the processing (detector drive and sense) is performed by a mixed signal ASIC, placed as close as possible to the detector (ideally co-packaged)..

The selected unit is a medium class GYR (bias stability≈10deg/h), based on MEMS technology manufactured by UTAS and mounted as part of GYR assembly together with electronics. Such a gyro allows to compensate for short duration outages of the STR during flybys, of maximum several minutes. If maintaining accurate pointing during longer duration without STR (e.g. flyby duration, 6 hours) would become mandatory, then highly accurate gyroscope would be mandatory (FOG), with significant impact on the mass.

Figure 13-7: 3-axes MEMS GYR assembly

Two set of 3 sensors are foreseen for redundancy reason.

The mass and power consumption in this case are very limited, being based on MEMS technology. The mass of one unit is below 0.1kg, while the power consumption when operative is 0.3W.

Sensibility of the detector to radiation is minimised by implementing the unit in the OBC, which is the most shielded part within CLEO/I S/C.

13.4.3 Sun Sensor

The selected Sun Sensor is the mini-FSS from MOOG Bradford. The unit is a very light Sun Sensor with medium/high accuracy (0.2deg with on-board calibration table).

The sensor nominal FoV is 128° x 128° (i.e. ± 64° x ± 64°) and therefore four sensors are needed to cover the entire celestial sphere. Two sets of 4 units (total of 8) are implemented for redundancy.

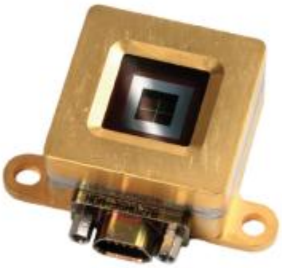


Figure 13-8: Mini-Fine Sun Sensor

The mass of a single unit is less than 50g and as the unit is fully passive, no power is needed.

The unit provides the 4 photo-diode currents, where their calibration and processing to obtain the Sun vector angles is performed by the spacecraft OBC.

Row Labels	Values mass (kg)	mass margin (%)	mass incl. margin (kg)
GYRO_Chip (GYRO on Chip MINAVIO)	0.05	20.00	0.06
GYRO_Chip2 (GYRO on Chip MINAVIO 2)	0.05	20.00	0.06
STR_HydraOH1 (STR Sodern Hydra Optical Head 1)	1.37	5.00	1.44
STR_HydraOH2 (STR Sodern Hydra Optical Head 2)	1.37	5.00	1.44
SUN_MoogBrad_mFSS1 (SUN Moog Bradford Mini Fine Sun Sensor 1)	0.05	5.00	0.05
SUN_MoogBrad_mFSS2 (SUN Moog Bradford Mini Fine Sun Sensor 2)	0.05	5.00	0.05
SUN_MoogBrad_mFSS3 (SUN Moog Bradford Mini Fine Sun Sensor 3)	0.05	5.00	0.05
SUN_MoogBrad_mFSS4 (SUN Moog Bradford Mini Fine Sun Sensor 4)	0.05	5.00	0.05
SUN_MoogBrad_mFSS5 (SUN Moog Bradford Mini Fine Sun Sensor 5)	0.05	5.00	0.05
SUN_MoogBrad_mFSS6 (SUN Moog Bradford Mini Fine Sun Sensor 6)	0.05	5.00	0.05
SUN_MoogBrad_mFSS7 (SUN Moog Bradford Mini Fine Sun Sensor 7)	0.05	5.00	0.05
SUN_MoogBrad_mFSS8 (SUN Moog Bradford Mini Fine Sun Sensor 8)	0.05	5.00	0.05
Grand Total	**3.24**	**5.46**	**3.42**

Table 13-1: AOGNC Equipment list

13.5 Options – Europa Fly-By

In order to evaluate the impacts on AOGNC subsystem in case of Europa fly-by, the main differences are listed below:

Europa mission differences:

- Duration of cruise: 20days
- Slew to communicate with CLIPPER (2x90deg at each fly-by)

Assumptions:

- Same Earth-S/C-Sun angle
- Same disturbances
- Same configuration (RCS 6-1NxAOC + 1-20NxDV)
- ΔV 170m/s

With the above considerations, the AOGNC subsystem remains unchanged in terms of sensor/actuator configuration and the only difference is in the propellant budget, which can be reduced thanks to the shorter duration of cruise phase. The total propellant budget becomes ≈0.151kg.

13.6 Technology Requirements

Following considerations listed in section 13.4.1.1 the effects of radiation on STR unit could be candidate as further technology development, based on existing units to further study and mitigate degradations.

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
STR	Harsh Radiation environment	STR manufacturers		To be further investigated
Gyro on a chip	Integrated gyrsoscope in OBC			

This Page Intentionally Blank

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)
- There shall be very high radiation dose during the Jovian moon flybys. This degrades solar cell performance significantly, and is accounted for in the modelling calculations, including a x2 margin factor in the 1MeV electron-equivalent figures, which were provided as a function of cover glass thickness, for the 2 flyby case
- High TID to electronics. For the PCDU electronics, it should be quite possible to achieve a TID sensitivity of >150 krad with appropriate choice of components
- Earth direction and Sun direction are always within 11° or less, so comms and power pointing requirements are compatible (cos 11° = 0.98). The payload pointing requirements are different, but only for ~5 hours during the moon flyby
- The highly elliptical proposed Jovian orbit gives a long cruise (months) between science flybys of Io.

14.1.1 Power Budget (Consumptions)

The power requirements of the spacecraft platform and payload are derived from the power consumption data of the individual equipment element definitions in the OCDT CLEO model. The "ON" power, "STANDBY" power, and the mode-specific duty cycles of the equipment elements are used to derive mode-average power consumptions. These are shown in Table 14-1. A maximum power consumption (per mode) can also be determined by simple addition of the equipment "ON" power values, but of course this is a crude worst case, and so should be used carefully.

A safe mode (SAFE) is listed in Table 14-1. However, in the final analysis, no standard safe mode can be afforded by the power system sizing, instead an intelligent safe mode strategy has been proposed (see 6.4.5). The average load is more than the solar array generation.

The average power requirement for dormant cruise during attachment to Clipper (DOR mode) is somewhat higher than the 20W value that was preliminarily stated by NASA as acceptable. Of course, at this stage, the acceptable value is far from definite, but the point should be flagged for analysis and consolidation in any further iterations.

	LAU	MAN	SAFE	DOR	SCI	JC	ECL	DTE	REL	PFCOM	PLCAL
⊟PF (Platform)	0	224.97	63.6	25.15	50.155	32	58	193.2	193.15	193	193
⊞EPC1 (Electronic Power Conditioning 1)	0	0.15	0	0.15	0	0	0	15	15	15	15
⊞MINAVIO (Miniaturized Avionics)	0	20	20	0	20	2	20	20	20	20	20
⊞PropTank_CLEO_I (Propellant Tank CLEO_I)	0	0	25	25	25	25	25	25	25	25	25
⊞PRT_CLEO_I_1 (Pressure Transducer CLEO_I)	0	0.8	0	0	0	0	0	0	0	0	0
⊞PRT_CLEO_I_2 (Pressure Transducer CLEO_I)	0	0.8	0	0	0	0	0	0	0	0	0
⊞PRT_CLEO_I_3 (Pressure Transducer CLEO_I)	0	0.8	0	0	0	0	0	0	0	0	0
⊞STR_HydraOH1 (STR Sodern Hydra Optical Head 1)	0	0	0	0	2.5	0	0	0.025	0	0	0
⊞STR_HydraOH2 (STR Sodern Hydra Optical Head 2)	0	0	0	0	2.5	0	0	0.025	0	0	0
⊞Thruster_AOCS_CLEO_I1_01 (Small Thruster CLEO_I1)	0	12.9	0	0	0.01	0	0	0.01	0.01	0	0
⊞Thruster_AOCS_CLEO_I1_02 (Small Thruster CLEO_I1)	0	12.9	0	0	0.01	0	0	0.01	0.01	0	0
⊞Thruster_AOCS_CLEO_I1_03 (Small Thruster CLEO_I1)	0	12.9	0	0	0.01	0	0	0.01	0.01	0	0
⊞Thruster_AOCS_CLEO_I1_04 (Small Thruster CLEO_I1)	0	12.9	0	0	0.01	0	0	0.01	0.01	0	0
⊞Thruster_AOCS_CLEO_I1_05 (Small Thruster CLEO_I1)	0	12.9	0	0	0.01	0	0	0.01	0.01	0	0
⊞Thruster_AOCS_CLEO_I1_06 (Small Thruster CLEO_I1)	0	12.9	0	0	0.01	0	0	0.01	0.01	0	0
⊞Thruster_AOCS_CLEO_I1_07 (Small Thruster CLEO_I1)	0	12.9	0	0	0.01	0	0	0.01	0.01	0	0
⊞Thruster_AOCS_CLEO_I1_08 (Small Thruster CLEO_I1)	0	12.9	0	0	0.01	0	0	0.01	0.01	0	0
⊞Thruster_AOCS_CLEO_I1_09 (Small Thruster CLEO_I1)	0	12.9	0	0	0.01	0	0	0.01	0.01	0	0
⊞Thruster_AOCS_CLEO_I1_10 (Small Thruster CLEO_I1)	0	12.9	0	0	0.01	0	0	0.01	0.01	0	0
⊞Thruster_AOCS_CLEO_I1_11 (Small Thruster CLEO_I1)	0	12.9	0	0	0.01	0	0	0.01	0.01	0	0
⊞Thruster_AOCS_CLEO_I1_12 (Small Thruster CLEO_I1)	0	12.9	0	0	0.01	0	0	0.01	0.01	0	0
⊞Thruster_LAE_CLEO_I (Large Thruster CLEO_I)	0	0.75	0	0	0	0	0	0	0	0	0
⊞TRASP_Tx_MOD_Rx_DED1 (Transponder (Tx_MOD_Rx_DED) 1)	0	33	18	0	0	0	0	33	33	33	33
⊞TWT1 (Traveling Wave Tube 1)	0	13.9	0	0	0	0	13	100	100	100	100
⊞Heater (Heater)	0	0	0	0	0	5	0	0	0	0	0
⊞GYRO_Chip (GYRO on Chip MINAVIO)	0	0	0.3	0	0	0	0	0	0	0	0
⊞GYRO_Chip2 (GYRO on Chip MINAVIO 2)	0	0	0.3	0	0	0	0	0	0	0	0
⊟PL (Payload)	0	3.5	3.5	1.91	26.8	8.1	3.5	3.5	3.5	3.5	42.5
⊞MagI (Magnetometer CLEO/I)	0	1	1	0.02	3.3	3.3	1	1	1	1	3.3
⊞MidIR_I (MidIR CLEO/I)	0	0.5	0.5	0.66	16.3	0.5	0.5	0.5	0.5	0.5	16.3
⊞NIMS_I (Neutral/Ion spec CLEO/I)	0	1	1	1.20	3.92	1	1	1	1	1	19.6
⊞Dust_Exp (Dust Experiment)	0	1	1	0.02	3.3	3.3	1	1	1	1	3.3
Grand Total	**0**	**228**	**67**	**27**	**77**	**40**	**62**	**197**	**197**	**197**	**236**
Including 20% power budget margin	**0**	**274**	**81**	**32**	**92**	**48**	**74**	**236**	**236**	**236**	**283**

Table 14-1: Mode-averaged power requirements (values in Watts)

14.2 Assumptions and Trade-Offs

CLEO-I will make the trip from Earth to the Jovian system as a passenger of CLIPPER. During this time the CLEO-I solar array will be folded and cannot generate power.

CLEO-I will take some power from its host for:

- Battery top up / self discharge compensation (negligible energy)
- Periodic check-outs & housekeeping tasks (negligible energy if performed infrequently)
- Thermal control (Significant energy, e.g. 25 W constant for propulsion heating).

After separation, the CLEO-I power system must provide power/energy to support all platform and payload requirements for 2 flybys of Io, and the communication of the associated data.

14.2.1 Power Bus Topology

A regulated bus requires a battery charge regulator (BCR) and battery discharge regulator (BDR). These increase the mass of the PCDU.

A regulated bus introduces greater losses (lower efficiency) for all energy that passes to the users via the battery. CLEOP will rely on battery-stored energy for all of the “active” modes, so this is an important factor.

A regulated bus can be an optimum solution when most of the electrical loads require a single, stable voltage. However, in CLEO, a large proportion of the energy goes to heaters that can use variable voltage without problems.

Unregulated (battery) bus is therefore the baseline.

14.2.2 Solar Array Regulation

The main trade-off to be considered for the solar array power regulation is whether to use maximum power point tracking (MPPT) or direct energy transfer (DET). In situations where the voltage of the solar array and the main power bus are stable and predictable, DET is most efficient (close to 100%). For CLEO, the solar array is relatively thermally stable, so temperature-linked array voltage variations should not be a large factor.

However, the previous trade off selected an unregulated power bus, in which the bus voltage varies according to the state-of-charge of the battery. Furthermore, the CLEO mission requires only very few charge/discharge cycles of the battery, which means that deep battery discharges can safely be used to minimise the battery sizing. Therefore, the bus voltage will vary considerably, adversely affecting the effectiveness of DET.

Also, the extreme radiation environment will cause degradation of the solar cells, leading to some change in their voltage output during the mission.

Considering all factors, MPPT is selected as optimum – mainly because of the deep discharges on the battery bus. It is also a lower risk option (in the sense that the accuracy of our thermal and radiation modelling in the “exotic” environment of Jupiter is less important when we are predicting the power system performance).

14.2.3 Battery Sizing vs. Solar Array Sizing

In simplistic terms, the CLEO power system could either:

- Have a solar array big enough to support the active modes (SCI, DTE), allowing the use of a very small battery
- Or support the active modes with a bigger battery and have a smaller array.

This is an easy trade-off: A battery performs equally well at Jupiter as at Earth, but the solar arrays can only provide less than 4% of the performance per kg or m^2 than they can at Earth (approx. 10W per m^2).

So, the optimum approach is to minimise the solar array size by running the "active" modes from the battery.

NOTE: DTE mode is Earth-pointing so is supported by solar power as well as battery. In REL mode, the antenna is pointed to Clipper, so zero solar power is assumed as a conservative worst case.

SCI mode is nadir pointed, so zero solar power is assumed in the baseline case. In reality, some power, at 25-35° off-pointing, could be available. This could allow reduction of the battery size, but this would have knock-on effect on the maximum duration of DTE and other active modes.

14.2.4 Solar Cell Coverglass Thickness

Coverglass thickness is optimised by considering the maximum end-of-life specific power [W/kg] of the solar array, including the mass of the coverglass.

Data from the JURA study [Figure 14-1], (for 3×10^{15} cm^{-2} 1MeV electron-equivalent at zero coverglass), shows a very broad peak in EOL specific power at a coverglass thickness of ~200 μm, but the relationship is relatively insensitive between 100 and 300 μm.

However, analysis of the CLEO-I mission radiation environment predicted higher fluences than the JURA case (in the region of 2×10^{17} cm^{-2} 1MeV electron-equivalent at zero coverglass). So, a CLEO-specific sensitivity study was performed using the CLEO-I 2-pass radiation fluence data, and the PEPS power system model. The results are presented in Figure 14-2, and show a 300 μm coverglass to be the optimum thickness. Therefore, 300 μm is considered as the design baseline in the sizing calculations.

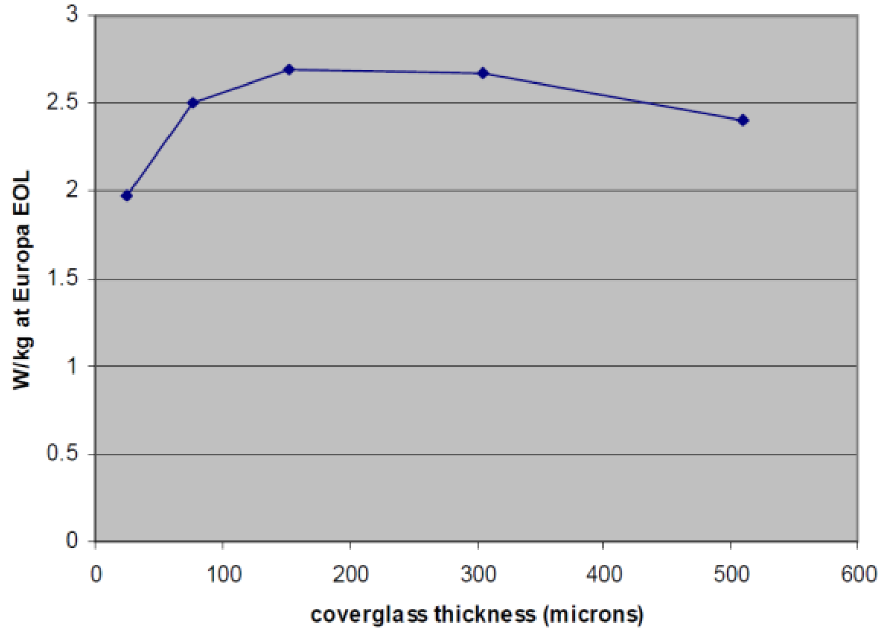


Figure 14-1: From the JURA study (Error! Reference source not found.**): Illustration of he EOL mass-specific power of a solar array as a function of coverglass**

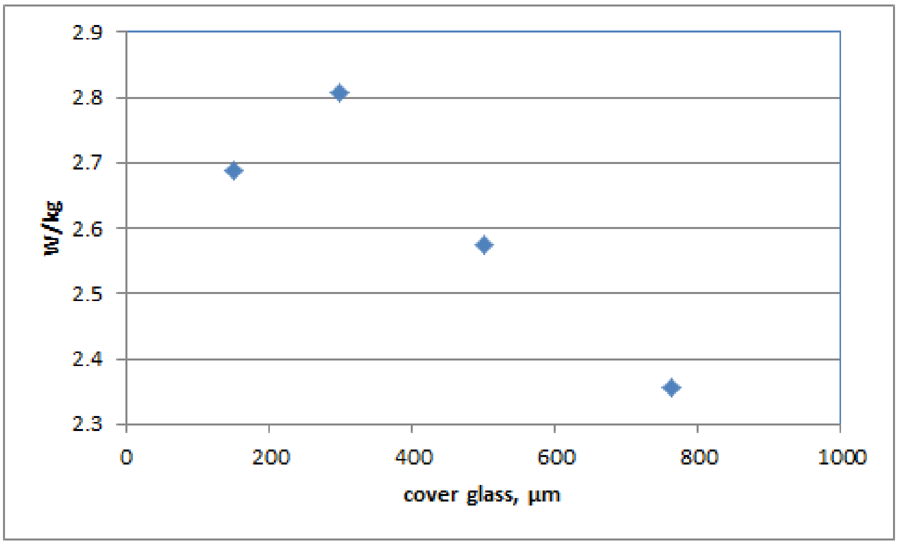


Figure 14-2: EOL mass-specific power of a solar array as a function of coverglass, for the CLEO-I 2-pass case

14.2.5 Array Size vs. Battery Charging Time

As explained above, the battery is sized to provide the energy requirement of the "active spacecraft system modes. Specifically, SCI mode (5 hours) is the battery sizing case, and requires a 4.9 kg (690Wh) battery.

In this approach, the solar array is sized to provide the battery charging time required by the mission timeline and operational logistics.

The baseline case for JC mode duration changed after the IFP from 23hours to 28 hours. The following analysis was done using the 23hour baseline and therefore needs to be recalculated in the next phase of the study.

Taking as an input the JC mode power consumption of 39 W (including margin), the baseline case of 23h battery charge time corresponds to a 5.8 m², 25.3 kg array. The maximum supportable duration of DTE mode is 2.7 hours in this case.

When considering how far the solar array size can be minimised, it is interesting to consider the theoretical limiting case of "infinite" battery charge time (i.e. the array provides enough power to prevent further battery discharge during JC mode, but does not provide any charge current. This case corresponds to a 3.7 m², 16.4 kg array. The maximum DTE duration is reduced to 2.5 hours in this case.

The concurrent design approach revealed that the communication strategy was the driving factor for the required battery charging time. Specifically, on the assumption that 222 hours of DTE mode is required per flyby:

- For a 100 day orbit, a 28 hours recharge time is appropriate
- For a 150 day orbit, a 40 hours recharge time is appropriate.

The relationship between array area, mass and battery charging time is illustrated in Figure 14-3. The charging times plotted are 28 hours, 40 hours, 60 hours and 10,000 hours (i.e. "infinite").

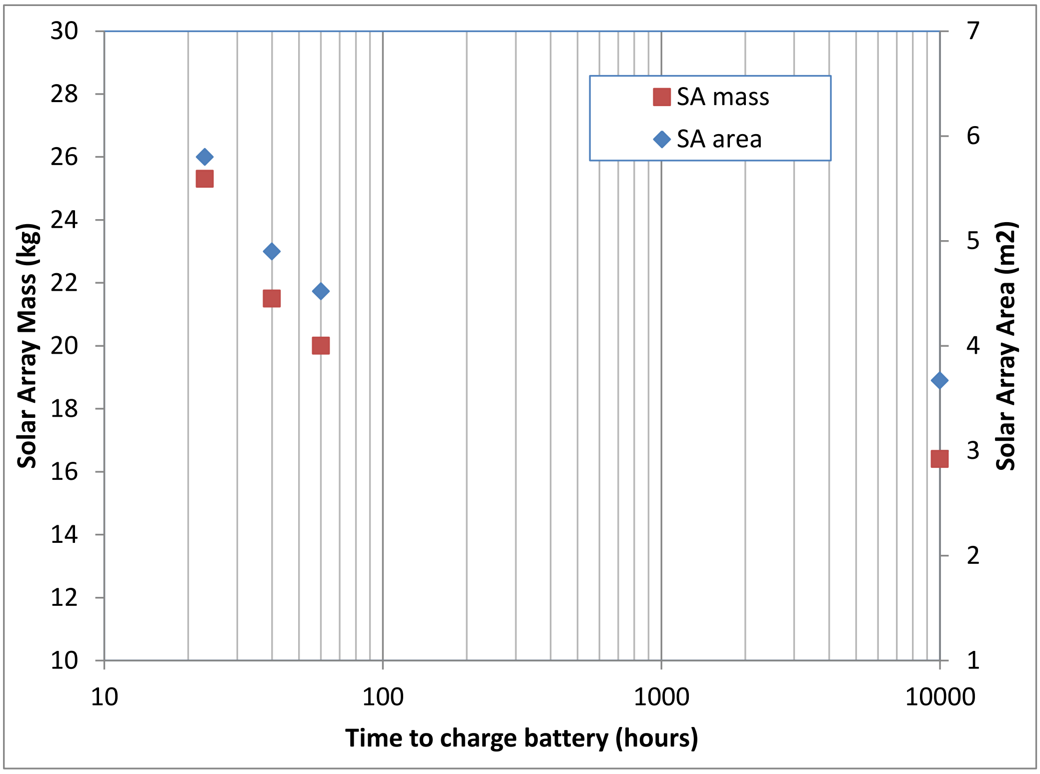


Figure 14-3: Solar array mass & area as a function of battery charging time using 23 hour baseline

14.2.6 Array Size vs. Battery Charging Time (Low Power DTE Option)

One communication strategy option that was considered was to use full-power DTE mode for 2 hours to transmit data, then switching to a "low power DTE" mode for flight dynamics trajectory determination only (TWT & EPC off). In low power DTE, the COMMS system requirement would be 113 W less than full power DTE. (135 W lower including margin).

In this scenario, the baseline power system (690 Wh, 4.9 kg battery & 5.8 m², 25.3 kg array) would support 2 hours of full DTE, followed by approx. 5 hours of low power DTE.

14.3 Baseline Design

The design tool used for power system analysis and sizing was the ESA TEC-EP PEPS tool. The graphical interface of the tool is used to illustrate the power system baseline design [Figure 14-4 and Figure 14-5]. An example of the model output is shown in Figure 14-6: the modelled timeline begins with a stabilisation period, followed by SCI, JC, DTE, JC, REL and JC modes in representation of a part of the mission timeline. The battery SoC, plotted in red, can be seen to remain above approximately 30% at all times.

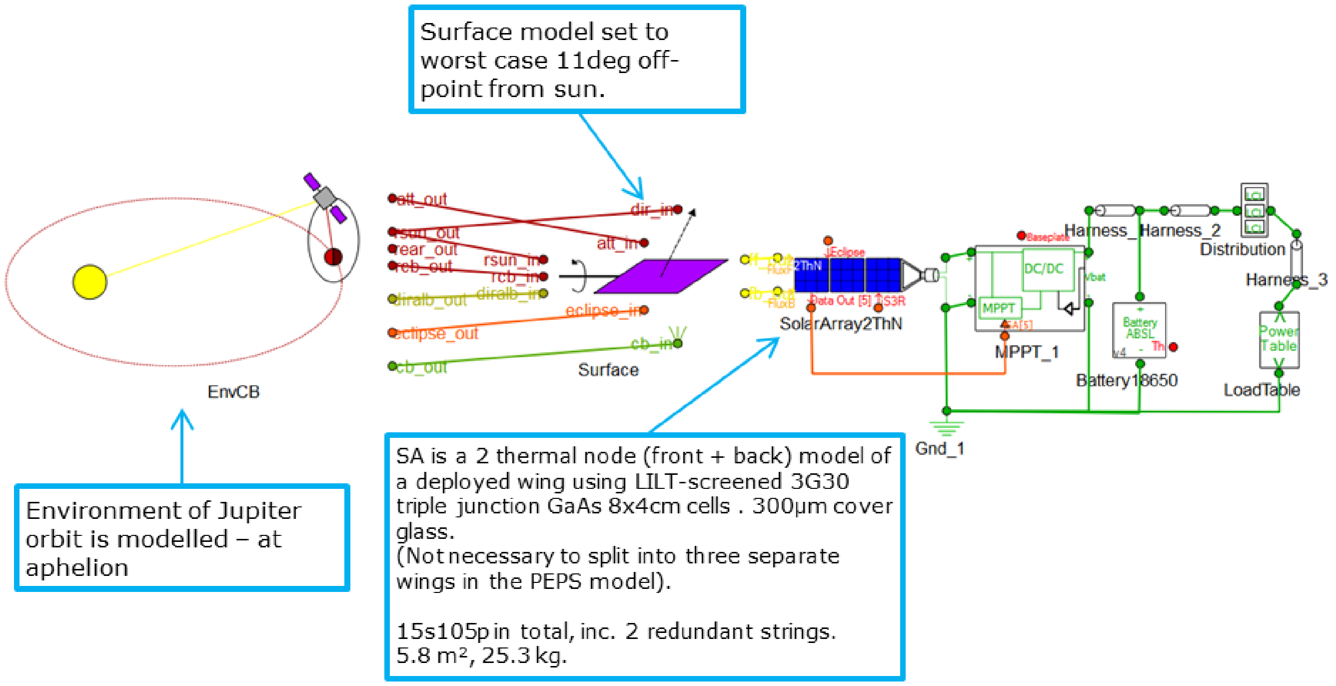


Figure 14-4: CLEO-I power system baseline design

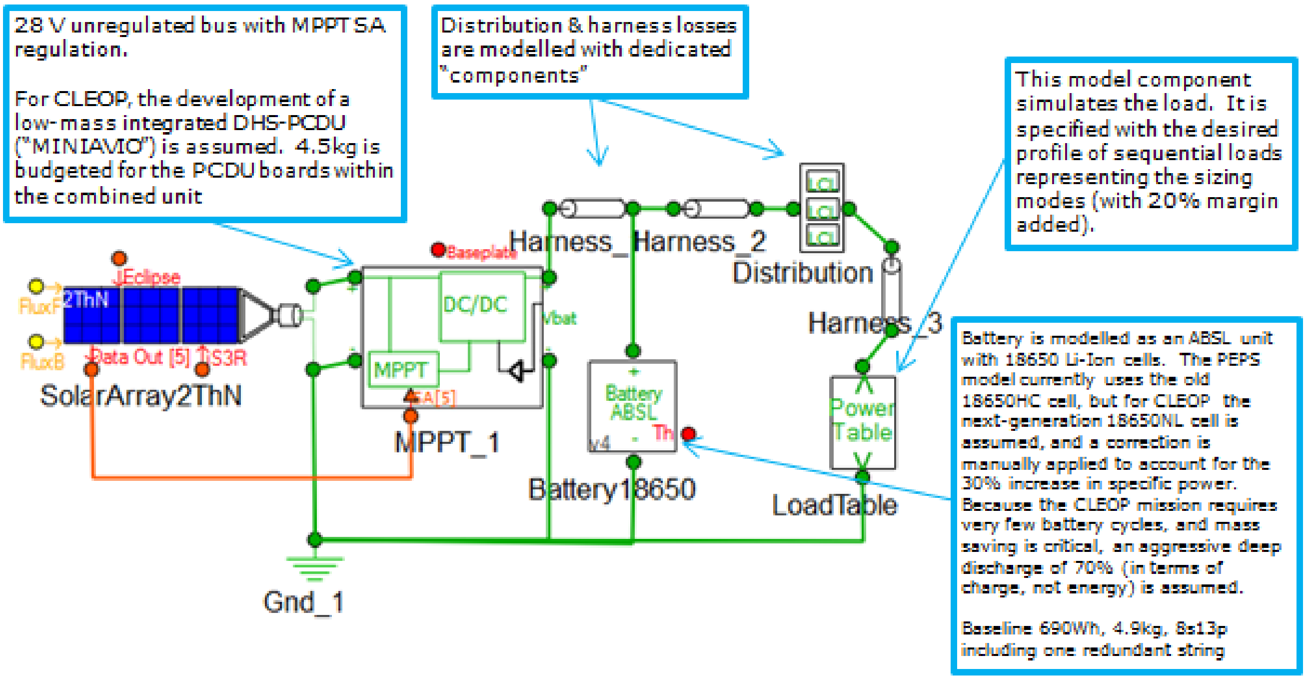
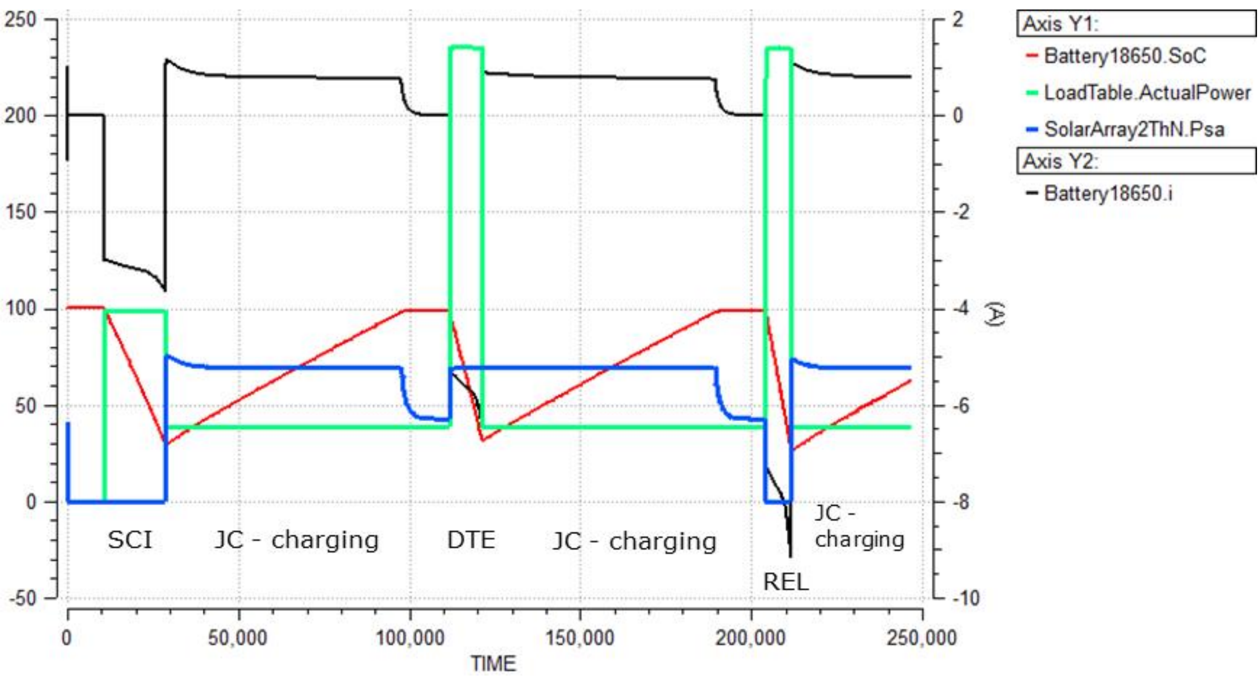


Figure 14-5: CLEO-I power system baseline design



LoadTable.ActualPower	Power consumption of the loads (W)
SolarArray2ThN.Psa	Total power output from the solar array.
Battery18650.SoC	Battery state of charge % (i.e. 1 – DoD)
Battery18650.i	Battery current (A). (charge = positive, discharge = negative)

Figure 14-6: CLEO-I power system model result

14.4 List of Equipment

	mass (kg)	mass margin (%)	mass incl. margin (kg)
CLEO_I (CLEO Io)	**34.69**	**20.00**	**41.63**
PF (Platform)	**34.69**	**20.00**	**41.63**
Bat (Battery_general)	4.90	20.00	5.88
MINAVIO (Miniaturized Avionics)	4.50	20.00	5.40
PCDU (Power Conditioning & Distribution Unit)	**4.50**	**20.00**	**5.40**
SA (SolarArray)	8.43	20.00	10.12
SA2 (SolarArray 2)	8.43	20.00	10.12
SA3 (SolarArray 3)	8.43	20.00	10.12
Grand Total	**34.69**	**20.00**	**41.63**

Table 14-2: Power subsystem list of equipment

14.5 Options

14.5.1 Europa Orbiter

The Europa payload has a lower average power requirement in SCI mode than the Io orbiter (see Table 14-3).

	PowerON	Duty Cycle	Av. Power
Dust Expt	7	1	7
NIMS-I	19.6	0.4	7.84
UV Spec	20	0.2	4
Total			18.84
Instr. heating (estimate)			3
Total incl. heating			**21.84**

Table 14-3: Europa Orbiter payload SCI mode power requirements (values in Watts)

The payload average power requirement of 21.8 W is 10.7 W lower than the Io case of 32.5 W. The duration of SCI mode is the same.

Hence the total spacecraft power consumption for SCI mode will be (82.7 - 10.7) = 72 W (86.4 W incl. margin). This is 87% of the Io case.

In the best case, this could result in a proportional reduction of the battery mass to 4.3 kg (saving 630 g w.r.t. the Io case). However, the battery reduction would impact also on the achievable duration of the other “active” modes, e.g. DTE, REL.

This Page Intentionally Blank

15 DATA HANDLING

15.1 Requirements and Design Drivers

The design of the data handling subsystem was carried out to provide sufficient control and data storage space to meet the mission requirements. To this end the following design drivers were used:

- Reduced mass and dimensions of the data handling system components to meet the stringent mass and volume restrictions of the S/C
- Reduced power consumption and a low power mode for the Jovian Cruise phase
- Harsh radiation environment in the Jovian system
- AOCS integration (IMU). Real time processing of AOCS payloads data (star tracker, navigation camera)
- Science data processing and compression requirements.

15.2 Assumptions and Trade-Offs

The main design driver of the overall avionics of the spacecraft is the limited mass and power consumption. In order to minimise the mass, a miniaturised and integrated avionics solution, MINAVIO, with resource sharing among the different subsystems, is proposed.

The idea is to integrate most of the avionics in the same mechanical box, power it from the same power conditioning circuitry and share the processing capabilities (processors and FPGAs). Additionally, it could also be considered to include a common back-end electronic design for all the scientific units.

The possible outcomes of the integration of the data handling unit with power, comms, AOCS and science in MINAVIO are the following:

- PCDU integration: The DHS unit could be powered directly from the PCDU auxiliary voltages without the need for extra power conditioning. The DHS wake-up lines, such as timers, voltage levels… could be implemented on the PCDU side, which is always on.
- Comm integration: On one hand, packetization, framing and encoding could be easily integrated in the DHS. Most processor chips, such as, SCOC3, GR712, COLE, Epica Next already include this functionality. The Analog part of the RF chain, on the other hand, would not be so easy to integrate. Most likely, there will be EMC issues, with digital noise interfering with the RF signals. Independent power conditioning, with isolated ground and power planes would be needed and not much gain would be achieved.
- AOCS integration: The image processing of the Star Tracker and the Navigation Camera could be performed on the main processor. These types of applications are typically implemented in low-performance processors such as the ERC32. Modern multi-core processors, such as the GR712, have enough resources to handle all these tasks simultaneously. This would, however, increase the SW complexity. Time and space partitioning SW architectures, such as the Integrated Modular Avionics (IMA) would be needed to assure fault detection and containment in the different applications. The integrated solution could also include MEMS gyros to provide IMU functionality.

- Science backend and processing integration: The scientific instrument backend electronics could also be integrated. The mission would highly benefit if the same electronic circuitry could be used for all payloads. The science data processing and compression could also be implemented in MINAVIO, either on real time, as data is acquired, or "offline" during cruise phase before communication.

15.3 Baseline Design

The baseline design is based on a single unit, MINAVIO that integrates power, part of AOCS and science data processing. The proposed design is based on Eurocard 6U PCB format with common backplane for all the units.

For the DHS part, the design contains 6 boards:

- 1 + 1 PM boards including processor module, reconfiguration unit and wake-up timers. The board should be based on GR712 or another processor with enough performance
- 1 + 1 MM boards including memory modules C&C and data interfaces. The board should contain a small non-volatile mass memory based on Flash technology
- 1 + 1 RTU boards including discrete telemetries, mechanisms and propulsion interfaces as well as MEMS gyros.

For the science backend electronics, the design contains 3 extra boards with an estimated mass of 1.5 kg.

For the power conditioning, the design includes 9 extra boards, although their mass is allocated on the power chapter of the report.

All the equipment is allocated in a common housing that shields the electronics from the radiation environment. Its mass is also allocated at system level.

	mass (kg)	mass margin (%)	mass incl. margin (kg)
MINAVIO (Miniaturized Avionics)	**4.50**	**20.00**	**5.40**
Grand Total	**4.50**	**20.00**	**5.40**

Table 15-1: DHS mass budget

The estimated power consumption of the DHS part is 20W while ON and 2W on low power mode, during which no processing capabilities are required.

The power consumption of the rest of the functionality can be found on their respective chapters of the report.

Power (W)	P_on	P_stby
MINAVIO (Miniaturized Avionics)	**20.00**	**2.00**
Grand Total	**20.00**	**2.00**

Table 15-2: DHS 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
MINAVIO	Integrated avionics unit	New design based on existing components from several suppliers		A similar unit is being developed in the frame of the “Miniaturized Avionics for Martian Landers (MINAVIO, RUAG Sweden)”

This Page Intentionally Blank

16 TELECOMMUNICATIONS

16.1 Requirements and Design Drivers

The objectives of the study includes the design of the communications subsystem capable of transmitting the payload data with a direct to Earth link and a relay link via Clipper.

The major design driver is the optimisation of the mass and power resources.

The TTC subsystem main requirements are the following:

- It shall receive and demodulate the uplink signal and transmit the commands to the on-board data handling
- It shall modulate and transmit the generated telemetry (housekeeping and science data)
- It shall provide navigation capabilities.

An initial estimate of 12 Gbits of compressed data has been provided as requirement to size the TTC subsystem for the two flybys.

The mission analysis expert has confirmed that the use of ΔDOR is not needed for the orbit determination requirements of the mission and therefore it has not been considered in the design. It shall also be noted that ΔDOR is not compatible with the transmission of telemetry.

Orbit determination is performed with ranging and Doppler measurements.

In order to optimise the mass, the reuse of the equipment for both links; the DTE and Clipper relay will be considered.

16.2 Assumptions and Trade-Offs

16.2.1 Frequency Selection

There are allocations for Deep Space missions in the S, X and Ka bands.

The X-band allocation has been selected for compatibility with the ground station network (currently the ESA Deep Space Network implements X-band uplink and downlinks in all stations and Ka-band reception in DSA2 and DSA3) and the availability of fully developed X-band hardware.

The S-band allocation is not selected since this band is not available in the ESA Ground stations. The use of the Ka-band will provide a higher data rate return however it will also imply additional equipment and increase in mass.

For the relay link, there is no allocation provided by ITU however after consulting the frequency management office and in an effort to reduce the equipment on board, the X-band allocation adjacent to the DS band has been selected (8400-8450 MHz).

In order to implement then the two links, two solutions are possible : A modification of the existing hardware to be able to select in flight the downlink frequency by telecommand (with a frequency step in the range of 20 to 50 MHz) or, to embark a dedicated set of redundant transmitters dedicated to the relay link.

16.2.2 NASA Inputs

The Clipper satellite can provide relay communications capabilities in different frequency bands UHF, S or X-band relay. NASA has confirmed that a two-way communications link is also possible.

The relay link is based on a store and forward capability.

An initial gain on-board the Clipper satellite of 30dBi has been assumed for the antenna gain but later 50 dBi has also been considered feasible.

16.2.3 Ground Station Assumptions

The CLEO satellite shall be compatible with the ESA and NASA Deep Space ground stations.

The availability of the NASA 70 m dishes for the mission nominal operations cannot be confirmed at this stage however it is known that NASA is investigating other means of providing similar capabilities to the 70 m antennas by arraying antennas.

16.2.4 Radiation

The components that are part of the EPC and transponder electronic boxes are sensitive to radiation. Different solutions are possible to overcome this problem: At system level the equipment can be put in a vault or, at equipment level, the walls can be thickened to provide additional shielding or spot shielding can be applied to protect specific components after performing the dedicated analysis.

The Travelling Wave Tubes (TWT) and Radio Frequency Distribution Units (RFDU) are not sensitive to radiation. The radiation effect on the switches shall be further investigated.

16.2.5 Antenna Trade-Offs

The high gain antenna is based on the metal antenna technology. The advantage of this antenna is its low mass compared to the parabolic reflector.

This antenna is currently at a very low TRL level however no criticalities in the design are foreseen.

Additionally it shall be noted that there is no current development that includes both receiving and transmitting capabilities.

16.3 Baseline Design

The TT&C subsystem is composed of:

Deep Space transponders: Two transponders are considered for redundancy, however the transponders will not be operated in a traditional way. Due to major power constraints, the transponder receivers will be OFF and switched ON by time-tagged commands or timers.

Travelling Waveguide Tube Amplifier: Two amplifiers are required to ensure the overall subsystem reliability figure. 65W of RF output power are confirmed from the link budgets and selected as a good compromise between needed RF power and power consumption.

Each TWTA is composed of a TWT (Travelling Wave Tube) and an EPC (Electronic Power Conditioning).

Radio Frequency Distribution Network: The RFDN provides all connecting elements between the output of the transponder and amplifiers to the antennas.

The RFDN will contain a 3dB coupler to provide the cross-strapping between the transponder transmitter and amplifiers. The diplexer filter will provide the separation between transmit and receive frequencies and provide the filtering to ensure compliance to the emissions and ensure RF auto-compatibility. Waveguides switches and waveguides will also be included to interconnect the transponders/TWTAs to the antennas.

Low Gain Antennas: Two low gain antennas are considered to provide almost omni-coverage. These antennas will be used in case of attitude loss. These antennas implement transmit and receive capabilities. Right hand circular polarisation is baselined.

The LGA baselined are based on the GAIA low gain antennas.

High Gain Antenna: A 1 meter antenna is considered. The high gain antenna will need very accurate pointing, a pointing accuracy of 0.2 deg it is assumed for the link budget calculations. The pointing will be achieved by pointing the spacecraft to Earth, thus the use of a pointing mechanism will not be required.

In safe mode, at max distance, and considering the NASA 70 m ground station performance, the LGA cannot support the communications link unless some pointing is ensured (around +/- 5degrees). In this case a limited TC rate of around 7.8bps could be achieved, for the TM link, the recovery of the carrier presents a challenge while a TM rate of 10 bps seems feasible. It is preferred that in the safe mode the HGA is pointed with an accuracy around 2 deg. The use of an MGA could also be considered to achieve a higher data rate.

Link Baseline: The single direct-to-Earth link has been selected as baseline since the relay link under the assumptions above provide a limited data rate.

A communications window of 2.8 hours is available (due power consumption constraints), however it is noted that this window is too short to acquire sufficient measurements for navigation. In order to extend this window, it has been decided to allocate communications windows of 2 hours for telemetry transmission, followed by extended 4 hours of TC and ranging/Doppler in low power mode (TWTA OFF).

Turbo codes with rate ¼ and 1/6 are baselined. Turbo codes with rate 1/6 are already available in the NASA Deep Space Network and will be available at the ESA Deep Space Antennas from 2016.

16.4 Link Budget

Link budget margins shall comply with the ECSS-E-ST-50-05C RF and Modulation Standard. For the nominal case the margin shall be higher than 3 dB.

16.4.1 DTE (Direct to Earth) Link

The degradation due to the Sun conjunction has not been considered in the link budget assessment. The communications windows shall ensure a minimum SES angle of 5 degrees.

The HGA will be pointed with a very good accuracy (+/-0.2 deg). A data rate of 1 kbps has been considered for the telecommand. For the receiving antenna a 0.5 m size has been considered. This needs to be confirmed, otherwise a bigger receiver antenna shall be implemented.

For the DTE link budgets a maximum distance to Earth of 6AU has been considered.

For the telemetry link a data rate of 3.5 kbps has been calculated. The telemetry data rate shall be shared between the scientific and housekeeping data.

The link budget margins are very marginal and should be carefully judged.

The use of the NASA 70 m dish has also been considered to increase the data return; a link with 20 kbps can be achieved.

16.4.2 Clipper-CLEO Telemetry Link

For the sizing of the link it is assumed that the CLEO will point to the Clipper satellite with its HGA and that Clipper will provide 30 dBi.

The data rate depends on the distance between the two spacecrafts, as an example for a distance of 50 R_j the telemetry data rate calculated is 2.8 kbps.

During the sessions the communication opportunities have not been defined, operational constraints need to be taken into account.

16.5 List of Equipment

16.5.1 Power Budget

Power (W)	P_on	P_stby
⊞ HGA (High Gain Antenna)	0.00	0.00
⊞ LGA1 (Low Gain Antenna 1)	0.00	0.00
⊞ LGA2 (Low Gain Antenna 2)	0.00	0.00
⊞ RFDU (Radio Frequency Distribution Unit)	0.00	0.00
⊟ TRASP_Main (Transponder Main)	33.00	0.00
⊞ Rx_DED (Receiver (dedicated))	18.00	0.00
⊞ Tx_MOD (Transmitter (MOD))	15.00	0.00
⊞ (blank)	0.00	0.00
⊟ TRASP_Slave (Transponder Slave)	0.00	0.00
⊞ Rx_DED (Receiver (dedicated))	0.00	0.00
⊞ Tx_MOD (Transmitter (MOD))	0.00	0.00
⊞ (blank)	0.00	0.00
⊞ TWT_Main (Traveling Wave Tube Main)	100.00	13.00
⊞ TWT_Slave (Traveling Wave Tube Slave)	0.00	0.00
⊞ EPC_Main (Electronic Power Conditioning Main)	15.00	0.00
⊞ EPC_Slave (Electronic Power Conditioning Slave)	0.00	0.00
Grand Total	148.00	13.00

Table 16-1: CLEO Telecom Power Budget

Note: See operational constraints to understand the power budgets.

The TWTA efficiency needs to be confirmed.

16.5.2 Equipment List

	mass (kg)	mass margin (%)	mass incl. margin (kg)
EPC1 (Electronic Power Conditioning 1)	1.40	5.00	1.47
EPC2 (Electronic Power Conditioning 2)	1.40	5.00	1.47
HGA (High Gain Antenna)	5.00	20.00	6.00
LGA1 (Low Gain Antenna 1)	0.30	10.00	0.33
LGA2 (Low Gain Antenna 2)	0.30	10.00	0.33
RFDU (Radio Frequency Distribution Unit)	5.00	20.00	6.00
TRASP_Tx_MOD_Rx_DED1 (Transponder (Tx_MOD_Rx_DED) 1)	3.50	5.00	3.68
TRASP_Tx_MOD_Rx_DED12 (Transponder (Tx_MOD_Rx_DED)1 2)	3.50	5.00	3.68
TWT1 (Traveling Wave Tube 1)	0.90	0.00	0.90
TWT2 (Traveling Wave Tube 2)	0.90	0.00	0.90
Grand Total	**22.20**	**11.49**	**24.75**

Table 16-2: Equipment list

16.6 Operational Constraints

Since the power consumption is a major constraint, the following operational constraints have been identified and shall be carefully analysed:

- The receivers are OFF and only awaken by a time-tagged command or an on-board timer
- The current implementation of the transponders is such that in order to transmit the receiver must be ON.
- The current baseline is based on ONE receiver ON at any given time, in case of a failure of the unit there shall be a software function capable of detecting and reconfiguring the satellite to the redundant unit.
- The communications windows are limited to 2.7 hours (due to on board power constraints), and the one-way trip time is around 50 min (@6AU), 41 min (@5AU), this means that most of the commanding will be “in the blind” since the acknowledgement will arrive outside the comms window. The time between passes is 28 hours.

16.7 Options

Options have been identified in the different sections.

16.8 Technology Requirements

The following new technologies have been identified:

- Metal antenna, current TRL level is 3

This Page Intentionally Blank

17 THERMAL

17.1 Assumptions and Trade-Offs

17.1.1 Identification of Worst Hot / Cold Cases

Sizing the Thermal Control Subsystem begins with the identification of the worst hot case and the worst cold case.

The worst hot case is the DTE mode as it features the highest simultaneous dissipation. This worst hot case occurs in Jupiter environment. Because of the satellite configuration and attitude (Solar Arrays and High Gain Antenna oriented towards the inner solar system), the Sun does not illuminate any of two radiators. The hot case directly drives the minimum needed radiator surface in combination with the thermal coupling between the dissipative units and the radiators.

The worst cold case occurs during Jupiter Cruise, when most of the units are switched off. This cold case drives the minimum heating power needed to maintain the units within their specified temperature range. This case occurs in Jupiter environment and it is assumed the radiators do not receive any external flux (assuming the solar arrays are pointed towards the Sun).

Note: Using the alternate launcher (Atlas 5) will alter the hot case and require the TCS to be sized to survive the Venus flyby (EVEEGA).

17.1.2 Units Temperature Limits and Dissipation

Table 17-1 below summarises the main unit temperature limits and dissipation, as taken into account for the thermal analysis.

	Design minimum temperature	Design maximum temperature	Dissipation in cold case	Dissipation in hot case (DTE)
Mini AVIO	-20°C	+40°C	2 W	20 W
Instrument electronics	-20°C	+40°C	5.3 W	2 W
TWT (2)	-20°C	+75°C	0	45 W
EPC (2)	-20°C	+50°C	0	15 W
STRE	-20°C	+50°C	0	0
Battery	0°C	+30°C	0	0
TRSP (2)	-20°C	+50°C	0	33 W
		TOTAL:	**7.3 W**	**116 W**

Table 17-1: Units temperature limits and dissipation

17.2 Baseline Design

17.2.1 Basic Principles

The spacecraft thermal control is mainly passive, essentially based on thermal insulation, thermal finishes, heat pipes to increase thermal coupling between units and supplementary heaters which are controlled by thermistors or thermostats. In order to fulfil the demanding extreme variations of thermal environment and internal dissipation, the thermal control design uses 2 louvered radiators.

The main principles of the thermal control consist of:

- Separating the different modules (platform, instruments, propulsion module, appendages) so as they are thermally independent
- Maximising thermal insulation from external environment, with:
  - Extensive use of MLI
  - Conductive decoupling
  - Closure of radiators with louvers
- Sharing the heat between all platform units with a network of heat-pipes
- Taking benefit of the thermal inertia of the structure to slow down cooling down of the units (particularly during Jupiter cruise mode).

The external surfaces that are not used as radiators (e.g. antenna, hold-down points...) are thermally insulated to the maximum extent possible by means of MLI blankets and low emissive coating.

17.2.2 Platform Units Thermal Control

All the platform units and the Instruments electronic boxes are collectively controlled to share their heat and keep all of them within their specified temperature with a minimum of heating power. This collective thermal control requires a complex network of standard ammonia heat-pipes which geometry is adapted to the structure and the unit footprint.

Each unit is thermally coupled to 2 heat-pipes to ensure redundancy. Heat-pipes are embedded in the structure or surface mounted to make possible inter-connections. On one side, Heat pipes are coupled:

- To a radiator on the one side of the Spacecraft
- To the mini-Avio box (on the opposite side) which drives the heat through its thick walls and act as a radiator (with the appropriate coating and radiating surface).

Both radiators consist simply of a white-painted area on the structure or directly on the mini-Avio box, which are thick enough to drive and spread the heat. Both radiators are equipped with louvers, to limit heat loss when the units are non-operating or in minimal dissipative mode.

Figure 17-1 presents a simplified overview of the heat pipes and louvers accommodation.

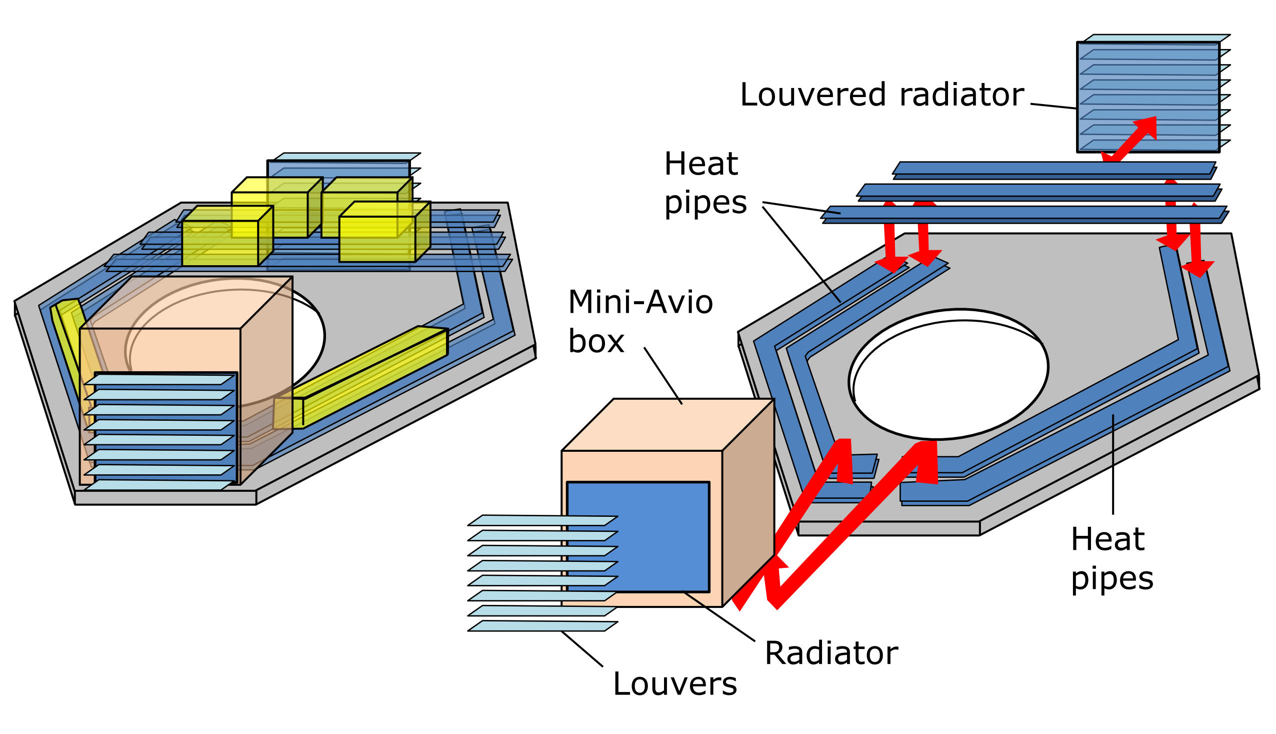


Figure 17-1: Accommodation of heat pipes and louvers

17.2.3 MLI

The Multi-Layer Insulation (MLI) is the key element to insulate the overall spacecraft but also different modules (propulsion, antennas) or pieces of equipment (piping, valves...). The external MLI is composed of 20 layers to minimise heat leaks towards cold Space. This kind of MLI benefits from a good heritage thanks to many missions such as Rosetta, GAIA, Herschel-Planck... Internal parts that require individual insulation to minimise heating power are wrapped with standard MLI (10 layers). Figure 17-2 shows an example of external MLI (star trackers MLI).

Figure 17-2: MLI

17.2.4 Heat pipes

Heat pipes are very common heat transport systems that ensure a high thermal coupling between several units and an external radiator. Made of stainless steel, the shape and inner design is adapted to the spacecraft configuration and dissipated power, as illustrated in the Figure 17-3.

Heat pipes are widely used on many spacecrafts, and particularly on Telecom satellite, with many heat pipes interconnections

Figure 17-3: Heat pipes

17.2.5 Louvers

Louvers are passive mechanisms mounted in front of a radiator and based on 8 independent bi-metallic actuators. Louvers vary the angle of their blades to provide thermal control by changing the effective emissivity of a covered surface. They are positioned by bi-metallic strips similar to those in a thermostat. They directly force the louvers open when internal temperatures are high, permitting heat to radiate into space. Cold internal temperatures cause the louvers to drive closed to reflect back and retain heat.

Figure 17-4: Rosetta louver

These louvers were extensively used in many deep space American probes (Pioneer, Voyager, Cassini...) and are still envisaged on current or future missions. European louvers fly on Rosetta. The Spanish company SENER has designed, developed, and qualified louvers. More than 10 flight-models have been manufactured for Rosetta, which one of them is shown in Figure 17-4.

When fully open: the equivalent emissivity is 0.71.

When closed, the equivalent emissivity is 0.11

17.2.6 Active Heating Control With Heaters When Needed

The use of active heating control with heaters is reduced as much as possible thanks to the passive thermal design as described above.

Heaters will mainly be used to maintain within their temperature range:

- The main electronic units above the minimum temperature limit during long periods of stand-by in Jupiter Cruise
- The Instrument sensors (which are not coupled to the main units)
- The propulsion module.

The internal propulsion parts (tanks, fluid lines, valves, pressure sensors) and pipes are radiatively and conductively insulated from the structure and provided with their own thermal control, including electrical heaters when needed.

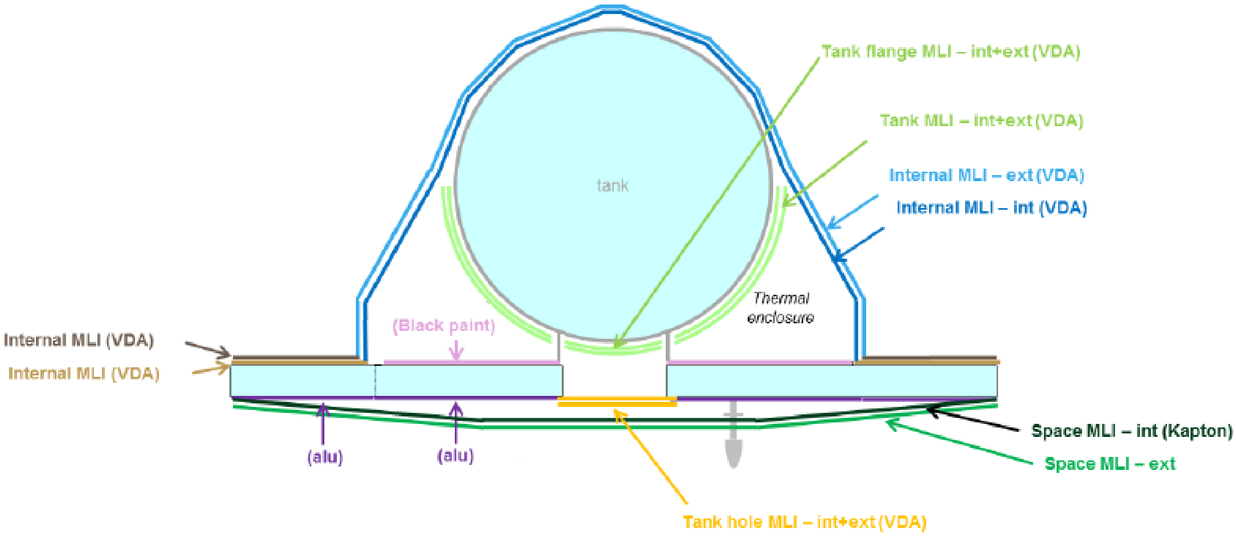


Figure 17-5: Propulsion module thermal control

All lines and components are individually controlled in order to decouple as much as possible the propulsion module thermal control from the spacecraft and to reduce the heating power budget. They are conductively decoupled from the structure by means of low conductive stand-offs and covered with a VDA tape (single pipe) or an MLI (several pipes) to reduce radiative exchanges. Each pipe is covered with Chofoil and equipped with individual spiral heaters when needed. The heaters are locally attached to the pipes by VDA tapes and covered by another layer of Chofoil, as illustrated in the Figure 17-6.

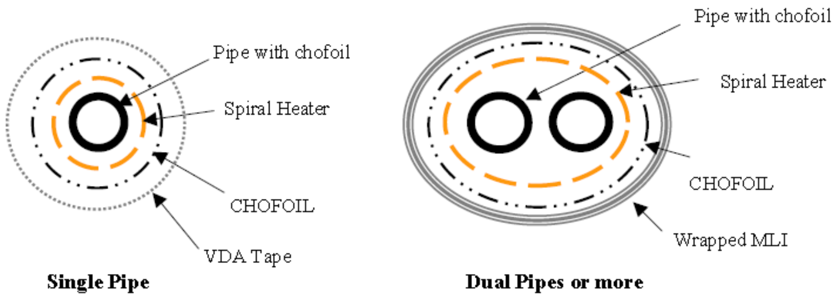


Figure 17-6: Examples of propulsion pipes thermal control

17.3 Thermal Analysis

17.3.1 Hot Case

Hot case is performed as a steady case.

- Overall dissipation: 116 W
- Limit temperature: +30°C (which corresponds to the battery maximum design temperature limit decreased by 10°C to account for uncertainty and model simplification)
- Radiator equivalent emissivity: 0.71 (open louvers).

The main output is the total radiators surface: 0.30 m² (2 times 0.15 m²)

In addition, the propulsion module consumes 25 W heating power to maintain all its components within their temperature range (Myriade Evolution Propulsion Subsystem thermal prediction in coldest case –no external flux- is used as reference because of the similarities).

17.3.2 Cold Case

The cold case is performed as a transient case.

- Overall dissipation: 7.3 W
- Unavoidable heat leaks (MLI, uncovered external surfaces…): assumed equal to overall dissipation (-7.3 W)
- Limit temperature: +10°C (which corresponds to the battery minimum design temperature limit increased by 10°C to account for uncertainty and model simplification)
- Radiator equivalent emissivity: 0.10 (closed louvers)
- Initial temperature: +30°C (hot case steady state)
- Overall mass of units + heat pipes + coupled structure: assumed equal to half of the overall Spacecraft mass = 41.1 kg.

With this set of assumptions, no heating power is needed for the units during the first 16 hours. Then, the overall assembly of units is maintained at +10°C with heating power, which consumes an average heating power of 4.9 W over a full period of 30 hours (from t=0 to t=108000 s, as illustrated in Figure 17-7.

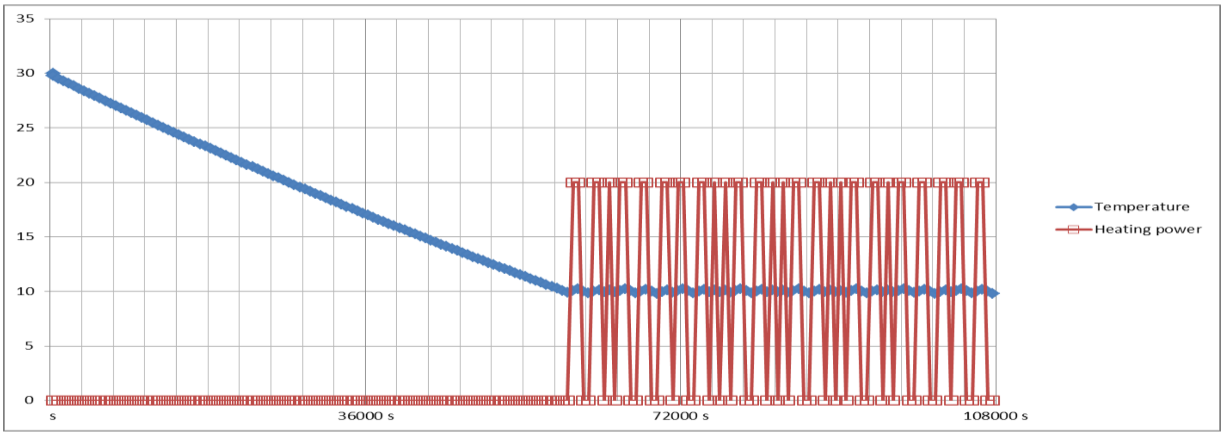


Figure 17-7: Transient cold case thermal prediction (temperature in blue, heating power in red)

As for the hot case, the propulsion module consumes 25 W heating power to maintain all its components within their temperature range.

17.4 List of Equipment

MLI

	surface	mass
External MLI	4.1 m²	2.05 kg
Internal MLI (units)	3.2 m²	1.6 kg
4 instrument MLI	2 m²	1 kg
Propulsion MLI (tank, pipings)	3 m²	1.5 kg
TOTAL MLI	**10.7 m²**	**5.85 kg**

MLI mass encompass the MLI itself, the attachment devices (stand-off, Velcro…) and the electrical grounding.

Heat Pipes

Total length: 6 m

Total mass: 1.8 kg (0.3 kg/m)

Louvers

Number: 2

Mass: 1.56 kg (0.78 kg each)

Miscellaneous (thermal)

Miscellaneous thermal devices: 0.5 kg.

This encompasses heaters, thermal sensors, thermal doublers or thermal strap (when needed), radiator coating.

	mass (kg)	mass margin (%)	mass incl. margin (kg)
LVR (Louvre)	0.79	10.00	0.86
MLI (MLI)	3.00	0.00	3.00
Grand Total	**3.79**	**2.07**	**3.86**

17.5 Technology Requirements

The following technologies are required for the thermal control:

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
	MLI	Many in Europe (raw material is usually from the US) TRL 9		Fly on many missions
	Heaters	RICA (Italy) Minco (US) Clayborn (US) Tayco (US) TRL 9		Fly on many missions
	Louvers	Sener (Spain) TRL 8-9		Flew on Rosetta
	Heat-pipes	EHP (Belgium) IberEspacio (Spain) TRL 8-9		Fly on many missions

This Page Intentionally Blank

18 GROUND SEGMENT AND OPERATIONS

18.1 Requirements and Design Drivers

Launch is in 2022 with a 2.7 years (optionally 7.2 years) Interplanetary Transfer phase as a hosted payload on the CLIPPER spacecraft. During this period, all communications between ESOC and CLEO will be via the CLIPPER MOC. There is also the possibility of direct to Earth communications around the time of the CLIPPER Earth fly-by.

The Commissioning of CLEO will be limited while attached to CLIPPER although a Huygens-like approach of periodic, open-loop checkouts during the transfer will be assumed.

Separation from the host will be performed 10 days after the clean-up manoeuvre of CLIPPER's PRM following which CLEO will have to perform two Ganymede Gravity Assists and a PLM before entering into its Science phase. CLEO's first Io GA will occur 15 months after separation. The baseline is two fly-bys but this can be extended without the need of an intermediate operations phase, i.e. the periodic fly-bys of Io will continue with the same routine of TCMs at apojove and pre- and post-fly-by.

The severe power constraints result in a routine operations period that repeats every ~31 hours: a ground station pass of 2.7 hours (Direct-to-Earth (DTE) mode) followed by 28 hours of battery recharging whilst in a hibernation state (Jovian Cruise (JC) mode). This atypical routine is further perturbed by:

- The fly-by phase of 300 minutes at the perijove
- The four 4 hours long passes (two pre- and two post-fly-by) that are dedicated to radiometric tracking for the single targeting manoeuvre and the single clean-up manoeuvre. These extended periods of contact are enabled by not powering the TWTA at the cost of the link margin for data transmission.

The pre- and post-fly-by manoeuvres and the apojove manoeuvre will be performed during what would otherwise be a DTE period (i.e. a period of full power after a JC/recharge period). This implies that they may not be done at the optimal time, but adjusting the DTE/JC cycle to put the DTE period at the correct time for the respective manoeuvre would require cutting one or more of the previous DTE periods short and, hence, reducing the science downlink time.

The relevant mission requirements are:

- MI-GE-070 specifies a launch in 2022
- MI-GE-110 specifies a direct to Earth link with monitoring and control performed from ESOC.

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
GS-010	Dual Ground Station coverage shall be provided for the "critical" passes. *CLEO is an offline mission but the communication periods are so short and infrequent that, once the science phase begins, every pass is critical.*	

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).

18.2.1 Limited Visibility

The Earth separation distance during the active (Jupiter) phase will be 4.5-6.5 AU which equates to a OWLT of approximately 37-54 minutes which, with the short 2.7 hours communication periods and the intervening 28 hours enforced hibernation, means that monitoring and control (especially closed-loop) will be even more limited than in a comparable deep space mission. Given this comms profile, the following sections apply.

18.2.1.1 Spacecraft operability

It is assumed that the spacecraft is simple to operate and simple to recover from an anomaly (see below).

Following input from the Rosetta mission, 1kbps of the downlink is assumed as a reasonable budget for HKTM (payload HKTM is part of the science data TM budget) which should include the selected real-time TM from the regular DTE passes and the selected recorded TM from the Science modes (Io fly-bys) and the special tracking DTE passes in which TM cannot be transmitted due to the TWTA being off.

CLEO cannot have a traditional Safe Mode (it cannot point at the Sun and, at the same time, transmit to Earth) so it is assumed to be robust against all but the most severe anomalies. In the event of a severe anomaly (an event that causes the spacecraft to abandon its routine operations), it is assumed that the entire bandwidth can be used to dump the on-board TM stores (i.e. the platform takes priority). In order to be able to confidently continue with the mission, the ground operations and Industry teams will need a suitable snapshot of the anomaly events and state of the spacecraft around the time of the anomaly and, the longer it takes to dump this data, the more science data is lost. Likewise, given the long loop, recovery actions cannot be complex.

For 28 hours out of every 31 the spacecraft is in JC mode (effectively switched off) during which there is no TM generation although the System design does foresee a reaction to an anomaly encountered during these periods. The action would be to point and transmit to Earth for the life of the existing battery power and then return to JC mode. These events would, however, kick the spacecraft out of the contact cycle that the ground is expecting and require permanent ground station coverage for at least the next 34hrs (i.e. the 3hrs of the expected pass, plus 28hrs, plus 3hrs of the subsequent pass to cover the possibility of the spacecraft having returned to JC mode just before the start of the planned pass). It is, therefore, assumed that the spacecraft will maintain the ground contact cycle even in the event of a severe anomaly.

18.2.1.2 Manoeuvres

The data cut-off for manoeuvre planning must be at the previous pass to that of the manoeuvre itself (giving over 24hrs for manoeuvre planning and command generation) with the manoeuvre commands initially uplinked in the blind even before the spacecraft is due to be in the target DTE mode during which the manoeuvre will take place. Note that this is not a recommended sequence for manoeuvre commanding but seems to be the only option at this stage For example, with a OWLT of 50mins:

1. DTE – 40mins: uplink commands and request a dump of the on-board timeline,

2. DTE + 10mins: receive commands and dump MTL,
3. DTE + 60mins: receive MTL and confirm its validity, if it is corrupted, send the abort command,
4. DTE + 110mins: possible reception of abort command,
5. DTE + 130mins: TCM,
6. DTE + 160mins: end of DTE, transition to JC mode.

18.2.1.3 Critical passes

It is a usual requirement that dual ground station coverage is required for "critical" passes to remove the risk of a ground station outage at the time of the pass. As it turns out for CLEO, most of the passes can be considered as critical:

- Every pass with Science data to downlink (i.e. every pass following the first Io fly-by): the actual data return is already much less than what is recorded during the fly-by (prioritisation of the science data for downlink is a science centre issue) so missing a contact period due to a ground station outage is to be avoided.
- Every navigation and TM/TC pass considered essential for manoeuvres/fly-bys: given the previous assertion, this really only applies to the period between CLIPPER separation and the first IGA orbit in which there are two Ganymede gravity assists. The four passes before and after the TCM/fly-by will be critical.
- Every fly-by and manoeuvre: the FDIR response to an unrecoverable anomaly during any non-Earth pointing activity is to point the HGA to Earth for the remainder of the battery power.
- The Commissioning period following separation: it is assumed that the initial telemetry link will be via an LGA relay with CLIPPER until the auto-sequence (attitude acquisition, solar array deployment, slew to Earth) has been completed and the (HGA) comms link to Earth has been established. Operations will be keen to demonstrate the DTE link whilst on the initial battery charge and then continue with guaranteed coverage for the free-flying commissioning of CLEO.

The planning of dual station coverage for so many passes at irregular periods (out-of-sync with the 24 hours of an Earth day) may not be 100% achievable in practice although cooperation with NASA DSN is assumed.

18.2.2 Orbit Determination

Section 5.3.6.1 in the Mission Analysis chapter suitably covers the issue of the limited time available for Radiometric Tracking. To summarise here,

- Range and Doppler is much more useful than DDOR in a bound planetary orbit
- RARR data should be collected over long tracking arcs to contribute to a good quality orbit determination (8hrs per 24hrs is a good baseline)
- DDOR and optical navigation (plane-of-sky measurements) are valuable additions when the line-of-sight tracking arcs are unavoidably short.

Of course, DDOR precludes TM/TC and requires a minimum of 1 hour ground station pass time, whilst opnav has a similar penalty in the CLEO case as it consumes some of the limited science bandwidth for the downlink of the navigation image.

Whilst RARR is typically in parallel with TM and TC, a DTE mode dedicated to radiometric tracking has been defined for CLEO in which the TWTA is switched off in order to save power and, thus, extend the pass duration to 4 hours (the assumed minimum for useable RARR tracking arcs for manoeuvre planning), but without TM (a severely questionable practice in the run-up to a manoeuvre and something for which the spacecraft would have to be specifically designed). Such a pass and DDOR are mutually exclusive leaving DDOR seemingly impractical for CLEO. The topic of orbit determination is a major issue for the mission and requires significant analysis with Flight Dynamics expertise that is beyond the scope of this study.

18.2.2.1 Alternative techniques

With dual-station coverage for the majority of passes, differenced-Doppler measurements (the coherent Doppler signal is received by two stations in parallel) offer an improvement in accuracy over standard Doppler.

The presence of CLIPPER in the Jovian system at the same time as CLEO opens the door to another orbit determination technique that has not been used by ESA before but has been demonstrated a number of times by NASA. Assuming that CLIPPER's orbit is very well known, we could use it to improve our orbit knowledge with Same Beam Interferometry (SBI).

SBI was first used by NASA with Pioneer 12 and Magellan around Venus and, at the time, provided an order of magnitude improvement over Doppler-only (improved VLBI systems since then have improved this further). The technique would require CLIPPER to be in DTE at the same time as CLEO and both be visible by two stations in parallel. In a similar way that DDOR uses the positional knowledge of quasars to determine the plane-of-sky position of a spacecraft, SBI uses the orbital knowledge of another spacecraft. And, significantly, SBI can be done in parallel with RARR, TM and TC. The availability of CLIPPER to support SBI would have to be analysed in advance and routinely assessed during the mission, but it is assumed that NASA would be willing to support this.

A further option that would also require active NASA support is Spacecraft-to-Spacecraft tracking with CLIPPER. This would require a laser terminal on CLIPPER, one or more laser retro-reflectors on CLEO and, most significantly, could be performed whilst CLEO is in JC mode (i.e. hibernation).

18.3 Baseline Design

CLEO will be operated from ESOC as part of 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 Q1/2018 with launch by the end of Q2/2022. An initial check-out/Commissioning will be performed via CLIPPER during the CLIPPER Commissioning phase and, in addition, a direct to Earth communications check with the CLEO HGA and general LGA communications as far as geometry and CLIPPER operations allow.

During the Transfer Phase it is preferred to keep CLEO switched on as much as possible, if not permanently: given the extremely limited visibility of the spacecraft once it is separated, the collection of as much in-flight data as possible, even in the hosted state, would be extremely valuable to operations around Jupiter.

Once separated, because of the short amount of time available per pass and the long duration between each one, CLEO's operations need to be extremely well planned out in advance: no time can be wasted. At the same time, there are only a few unique elements to CLEO's operations and lots of repetition so, once the initial development of the scheduling processes and tools has been made, spacecraft operations for the FCT should be of a low intensity. Planning for the minimum, however, would be a bad decision given the absence of flexibility that is possible in the mission operations design, and, besides, the ground station scheduling activities will be demanding. Given the questionable state of the orbit determination possibilities and the mission to be flown, it is expected that Flight Dynamics activities will also be demanding.

A small FCT will be built up to its core complement 6 months before the shipment of CLEO to NASA with the focus of the first SVTs on the commandability of the spacecraft directly by the FOS. Later SVTs will involve the commanding path via CLIPPER. This alternative path will be essential during the Transfer phase which will be used for the implementation and validation of the procedures, processes and interfaces that will govern the Jupiter phase. In time for the separation from CLIPPER, additional part-time engineering manpower will be brought in from other SP missions to support the first year of free-flying operations (see below).

Note that planning to share the members of an FCT with other missions is always a huge assumption about the state of other missions at that time.

Time to next event	Event	Activities
	CLEO switch-on (if it was hibernated previously, e.g. for the CLIPPER JOI)	- Sub-system check-outs, - routine monitoring, - DTE tests with the HGA when possible
10d	CLIPPER PRM	
10d	Separation and initial acquisition	- Assume CLEO inherits good orbit knowledge from CLIPPER - LGA comms relay via CLIPPER - SA deployment, spin-up and stabilisation, DTE comms by the end of the initial battery charge - Dual GS coverage
+ 3m	GGA1	- Targeting ΔVs at -3w, -1w, -3d - Clean-up at +2d - Platform Commissioning, calibrations etc. to be done before the first ΔV - Dual GS coverage for the first week of passes and then for the tracking passes and the manoeuvres
+6m	GGA2	- Apojove, targeting and CU manoeuvres - 3-axis stabilised (AOCS Science mode) entry and exit tests during coverage with dual GS support - Payload Commissioning - The fly-by will be used as a test Science run for the Io fly-bys - Dual GS coverage for the tracking passes and the manoeuvres
+3m	PLM	- 2x targeting, 1x CU manoeuvres - Dual GS coverage for the tracking passes and the manoeuvres - Routine monitoring/operations
+3m	IGA1	- Apojove, targeting and CU manoeuvres - Dual GS coverage for the rest of the mission from the time of the first fly-by - Routine operations
+100d	IGA2	
+190d	IGAx	

Table 18-1: Jupiter Phase Operations

18.4 Options

Ka-band communication links would improve the quality of the radiometric tracking data as well as the downlink bandwidth. The susceptibility of these links to bad weather at the site of the ground station is somewhat compensated for by the dual ground station coverage by globally separated ground stations.

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[27]:

- 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

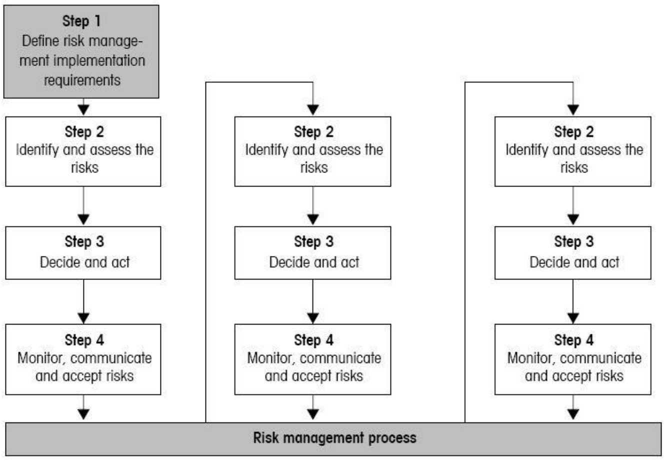


Table 19-2: ECSS-M-ST-80C, 2008 Risk Management Process

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

Score	Likelihood	Definition
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 CLEO 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 Tab. 1-6b) 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	

<table>
<tr><td>CLEO
/I + /E</td><td>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;

<u>basel.: RISK:</u>
Science - loss of mission before finishing of complete science program
likel.: high / sev. critical -> high risk

* AOCS/ TM house keeping</td><td><u>basel. MITIG.:</u>
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%

<u>remain. RISK:</u>
a+b+c) Science - loss of mission before finishing of complete science program
likel.: medium / sev. critical -> medium risk

* AOCS/ TM house keeping</td><td>basel. RISK:
R14 -> D4sc
remain. RISK:
R14 -> C4sc</td><td></td></tr>
<tr><td>CLEO
/I</td><td>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)

<u>basel. RISK:</u>
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</td><td><u>basel. MITIG.:</u>
a) trajectory should not pass over active areas
b) increase of fly-by attitude > app. 500km

<u>remain. RISK:</u>
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
*</td><td>basel. RISK:
R15 -> D4sc
remain. RISK:
R15a -> C4sc
R15b -> C3sc</td><td></td></tr>
<tr><td>CLEO
/I + /E</td><td>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)

<u>basel. RISK:</u>
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
*</td><td><u>basel. MITIG.:</u>
consideration planetary protection (ESA) requirements, resulting into e.g. sterilisation (/E) **

<u>remain. RISK:</u>
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</td><td>basel. RISK:
R16.2 -> E5po
remain. RISK:
R16.2 -> E3c, C5po</td><td></td></tr>
</table>

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					
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	
	A (min.)	B (low)	C (medium)	D (medi.)	E (max.)
					Likelihood

* not applicable for CLEO/E

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

Severity					
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
	A (min.)	B (low)	C (medium)	D (medi.)	E (max.)
					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.

… and 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 CLEO project from a programmatics point of view are:

- The CLEO S/C shall be carried as a piggy back on NASA Clipper S/C and released after Jovian Orbit Insertion
- The CLEO 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 CLEO S/C total mass shall not exceed 250 kg
- The CLEO S/C shall conform to Category III Planetary Protection Requirements (significant chance of biological contamination, potentially higher for CLEO/E compared to CLEO/I).
- The schedule needs to be aligned with project management timeline of Clipper
- TRL 6 required by 2018
- CLEO 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 CLEO, 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	CLEO Io	26		
Elements	SYE	Payload CLEO	1		
Elements	SYE	Platform CLEO	1		
Components	SYE	Shielding	24		
Subsystems	AOGNC	Attitude, Orbit, Guidance, Navigation Control Subsystem	33		
Equipment	AOGNC	SUN Moog Bradford Mini Fine Sun Sensor	8	Box	9
Equipment	AOGNC	STR Sodern Hydra Optical Head	2	Cylinder	9
Equipment	AOGNC	GYRO on Chip MINAVIO	2	Box	6
Subsystems	COM	Communications Subsystem	1		
Equipment	COM	Electronic Power Conditioning	2	Box	-
Equipment	COM	High Gain Antenna	1	-	-
Equipment	COM	Low Gain Antenna	2	-	-
Equipment	COM	Modulator	6	-	-
Equipment	COM	Radio Frequency Distribution Unit	1	Box	-
Equipment	COM	Receiver (dedicated)	2	-	-
Equipment	COM	Transmitter (MOD)	2	-	-
Equipment	COM	Transponder (Tx_MOD_Rx_DED) Master	1	-	-
Equipment	COM	Transponder (Tx_MOD_Rx_DED) Slave	1	-	-
Equipment	COM	Traveling Wave Tube	2	Box	-
Subsystems	CPROP	Chemical Propulsion Subsystem	26		
Equipment	CPROP	Feed line CLEO_I	1	-	9
Equipment	CPROP	Fill Drain valve Fuel CLEO_I	1	-	9
Equipment	CPROP	Fill Drain valve Pressurant CLEO_I1	1	-	9
Equipment	CPROP	Large Thruster CLEO_I	1	-	9
Equipment	CPROP	Latch Valve CLEO_I	3	-	-
Equipment	CPROP	NC Pyro Valve CLEO_I	1	-	-
Equipment	CPROP	NO Pyro Valve CLEO_I	1	-	-
Equipment	CPROP	Pressure Transducer CLEO_I	3	-	9
Equipment	CPROP	Propellant Filter CLEO_I	1	-	9
Equipment	CPROP	Propellant Tank CLEO_I	1	Sphere	9
Equipment	CPROP	Small Thruster CLEO_I1	12	-	9

Consumables	CPROP	Propellant			
Subsystems	DH	Data-Handling Subsystem	1		
Components	DH	Miniaturized Avionics	1	-	3
Subsystems	INS	Instruments Subsystem	5		
Equipment	INS	Camera CLEO/I	1	-	4
Equipment	INS	Mag Boom CLEO/I	1	-	6
Equipment	INS	Magnetometer CLEO/I	1	-	6
Equipment	INS	MidIR CLEO/I	1	-	4
Equipment	INS	Neutral/Ion spec CLEO/I	1	-	5
Subsystems	MEC	Mechanisms Subsystem	31		
Equipment	MEC	Clipper-PDS Separation Mechanism	2	-	-
Equipment	MEC	SA Deployment Hinge	18	-	9
Equipment	MEC	Satellite Deployment Mechanism CLEO	1	-	-
Equipment	MEC	Satellite Deployment Mechanism Clipper	1	-	6
Equipment	MEC	CLEO/P Solar Array HDRM	9	-	6
Subsystems	PWR	Power Subsystem	5		
Equipment	PWR	Battery_general	1	Box	5
Equipment	PWR	Power Conditioning & Distribution Unit	1	Other	4
Equipment	PWR	SolarArray	3	Other	8
Subsystems	RAD	Radiation Subsystem	1		
Subsystems	STR	Structures Subsystem	8		
Equipment	STR	CLEO-I Columns	1	-	-
Equipment	STR	CLEO-I Floor	1	-	-
Equipment	STR	CLEO-I Floor Reinforcement	1	-	-
Equipment	STR	CLEO-I Interface Adapter	1	-	-
Equipment	STR	CLEO-I Intermediate Floor	1	-	-
Equipment	STR	CLEO-I Lateral Panels	1	-	-
Equipment	STR	CLEO-I Sun Floor	1	-	-
Equipment	STR	CLEO-I Tank Cone	1	-	-
Subsystems	TC	Thermal Control Subsystem	9		
Equipment	TC	Louvre	1	-	-
Equipment	TC	MLI	1		
Components	TC	Heater	7		

Table 20-1: CLEO product tree

Note:

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

The TRL definitions from RD[28] 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

Although a general statement is made, that only technology sufficiently advanced (TRL) to start the Implementation Phase will be proposed, there are TRL as low as 3, 4 and 5 identified.

Table 20-3 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 20-3: 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 model philosophy proposed at orbiter level for CLEO/I 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).

The feasibility of this approach depends on the currently undefined planetary protection approach for Clipper.

For CLEO/E it is likely that in addition a QM is needed, because it is unlikely that CLEO/E can meet the required low impact probability and therefore requires bioburden control. If sterilisation at orbiter level is required, again this depends on the Clipper approach, then most likely a QM will be needed.

At equipment level, a model philosophy depending on equipment heritage is foreseen:

- For new developments
  - EM (for EFM), QM and FM
- For recurrent equipment
  - EM and FM

Note:

Any hardware going through a sterilisation process will need a QM.

Note:

NASA identified for their spacecraft, the Europa Clipper spacecraft the instrument hardware delivery schedule identified in Table 20-4.

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 20-4: Europa Clipper instrument hardware delivery schedule

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, as shown later, such a conservative approach is not compatible with the target launch date, a more success oriented or "Proba-approach" was also investigated, which is an approach tailored to in-orbit demonstration . 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. Accordingly less time is allocated to project phases, reviews and the interruptions for approval of the next contract phase.

20.6.1 Test Matrix

The system level test matrix is the same for both development approaches. Table 20-5 shows the test matrix with tests on orbiter level (CLEO S/C) and the joint tests with Clipper denoted as "Composite" in the table.

20.7 Schedule

When comparing the schedule in Figure 20-2 with the schedule in Figure 20-1, which is based on typical phase durations for small to medium size spacecraft, note:

- Phase A is reduced from typically 12 month to 10 month
- Phase B1 is reduced from typically 12 month to 8 month
- In both cases 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.

Test Description	CLEO STM	CLEO AVM	CLEO 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 / Separation	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 20-5: CLEO system level test matrix

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**	**1695 days**	**Tue 01/09/15**	**Mon 28/02/22**
10			Kick-off	0 days	Tue 01/09/15	Tue 01/09/15
11			PRR	40 days	Thu 07/07/16	Wed 31/08/16
12			SRR	40 days	Fri 07/07/17	Thu 31/08/17
13			PDR	50 days	Fri 21/12/18	Thu 28/02/19
14			CDR	60 days	Fri 28/02/20	Thu 21/05/20
15			QR	44 days	Mon 02/11/20	Thu 31/12/20
16			STM delivery to NASA	0 days	Thu 31/12/20	Thu 31/12/20
17			AR	41 days	Mon 03/01/22	Mon 28/02/22
18			PFM delivery to NASA	0 days	Mon 28/02/22	Mon 28/02/22
19			Phase A (12 month)	262 days	Tue 01/09/15	Wed 31/08/16
20			Phase B1 (12 month)	261 days	Thu 01/09/16	Thu 31/08/17
21			Intermediate phase (6 month)	129 days	Fri 01/09/17	Wed 28/02/18
22			Phase B2 (12 month)	261 days	Thu 01/03/18	Thu 28/02/19
23			Phase C/D (36 month)	782 days	Fri 01/03/19	Mon 28/02/22
24			STM procurement	261 days	Mon 02/09/19	Mon 31/08/20
25			STM environmental test campaign	44 days	Tue 01/09/20	Fri 30/10/20
26			AVM procurement	130 days	Mon 02/09/19	Fri 28/02/20
27			AVM integration	175 days	Mon 02/03/20	Fri 30/10/20
28			AVM testing	346 days	Mon 02/11/20	Mon 28/02/22
29			PFM procurement	261 days	Fri 01/05/20	Fri 30/04/21
30			PFM integration	175 days	Mon 03/05/21	Fri 31/12/21
31			PFM environmental test campain	41 days	Mon 03/01/22	Mon 28/02/22

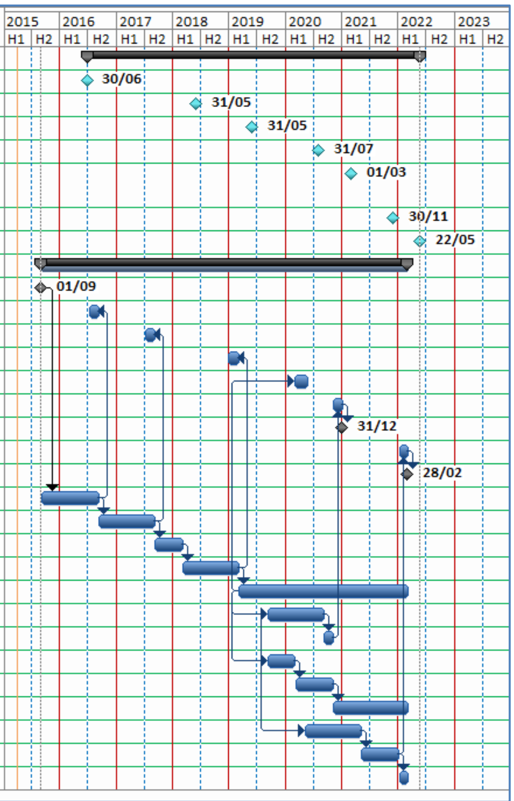


Figure 20-1: CLEO schedule – conservative approach

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

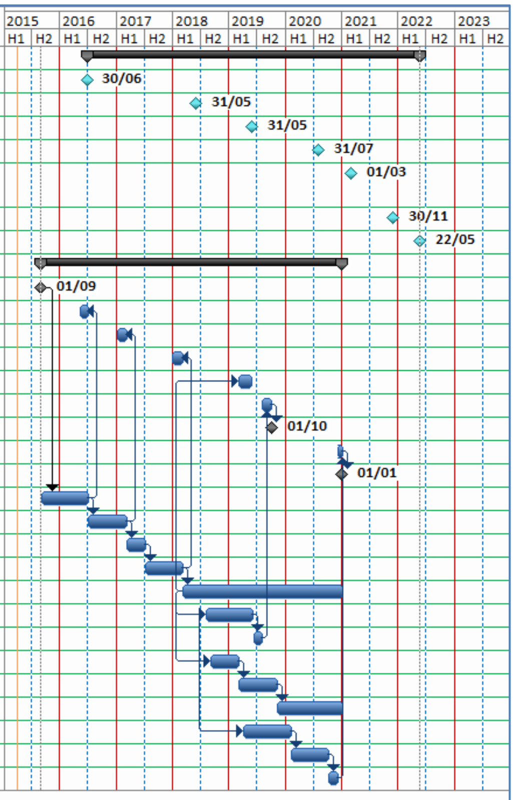


Figure 20-2: CLEO schedule – "Proba-approach"

20.8 Summary and Recommendation

- A conventional model philosophy is proposed for CLEO with at system level STM, AVM and PFM
- Environmental test campaigns at CLEO 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 2022) 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 risky to expect technology with a TRL lower than 5 to 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.

This Page Intentionally Blank

21 COST

The contents of this chapter has been removed from this version of the report.

This Page Intentionally Blank

22 CONCLUSIONS

22.1 Satisfaction of Requirements

Preliminary designs of the CLEO/I and CLEO/E orbiters have 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/I design has a baseline design that performs minimum 2 Io fly-bys. This is a completely independent satellite orbiting Jupiter and is designed for maximum science return while constraining to very tough mass limit of 250 kg for a completely independent Jovian orbiter. The final wet mass is 266.75 kg based on a low power design (e.g. 56W for re-charge), a challenging timeline with frequent switching off of communications system.

A low-mass option was identified that does comply with the 250 kg mass limit, at the expense of performing only 1 swing-by (strong impact on science return). This makes for a very compact spacecraft design with low propellant. Also, all redundancy was removed. For this option, Clipper ejects CLEO/I before the JOI which in turn also increases the mass accommodation on Clipper as Clipper can perform the JOI without extra CLEO/I mass. The wet mass of CLEO/I would then be 175 kg.

Finally, a Europa orbiter (CLEO/E) was studied. For this case there is no Io augmented science for Clipper. CLEO/E needs to stay attached longer to Clipper (after PRM) which reduces propellant and brings the wet mass down to 243 kg. The communication trade-off DTE (higher fly-by velocity) versus relay (higher TID) could not be concluded in this CDF study and would need to be optimised in a subsequent phase.

22.2 Compliance Matrix

Preliminary design of the CLEO/I minisat building on past CDF studies (REIS, CRETE, JURA), capitalizing on JUICE developments and miniaturised and integrated technologies (in particular for avionics)	Completed. Compact spacecraft, 267 kg. Integrated avionics applied. Small solar panels & battery, minimised number of mechanisms.
Optimise the mission profile, orbits strategy & associated ΔVs	Completed. Five mission profiles traded. Option '2b' selected as baseline which is a compromise of ΔV, fly-by velocity, operational strategy and radiation.
Identify the key design drivers and the operational challenges of the mission (in particular linked to Io environment)	Completed. Key drivers are strong mass constraint, low power available at Jupiter, large distance to Earth (6 AU) and radiation environment
Trade-off different subsystem design options focusing on mass, power, radiation tolerance, shielding strategy, duty cycle optimisation as to minimise the power mass.	Completed. Many subsystem trade-offs performed (optimisation battery versus solar array power, reaction wheels versus thrusters, amount of thrusters, type of propulsion system, several structures evaluated with

	optimised shielding design), instruments housing etc.
Optimise the TT&C subsystem to cope with DTE needs while maximising science data return via Clipper	Completed. TT&C subsystem is one of the large drivers for this mission. DTE versus relay was traded, as well as redundancy approach. Low mass HGA antenna selected.
To assess the applicability of the CLEO/I design concept to the CLEO/E mission, addressing the impact with respect to Io flybys concepts in terms of: i. ΔV ii. Radiation Shielding iii. Spacecraft design (e.g. power, thermal, propulsion s/s, etc.) iv. Number of flybys Planetary Protection constraints	Completed. First assessment shows a slight reduction in mass while keeping 2 Europa swing-bys. Planetary protection proposal to be reported.
Propose and define a Science case and payload suite for both concepts	Completed. Two instrument suites selected (Io & Europa case)
Identify technological needs, and associated Programmatics, Risk and Cost aspects of CLEO/E, 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

- The iteration presented in this report includes a power subsystem sized for a "battery charging" mode (Jovian Cruise) duration of 23 hours instead of the final 28 hours. A small dry mass change (possibly a reduction) is therefore expected on future assessments
- 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
- It should be investigated if star tracker shielding is necessary, and verify field of view with respect to the magnetic boom
- Heating power is to be optimised
- Safe mode should be further investigated in subsequent phases, considering need for Sun Acquisition mode, possibility to use LGA while Sun pointed, or analysing the need for MGA .

22.4 Final Considerations

Many options on system and subsystem level were considered and traded during this CDF study. The baseline design is based on maximum science return with a goal of minimising wet mass. Yet, the 250 kg mass constraint is a hard constraint and not reached with baseline design by 17 kg. The Minimised mass design leads to more risky operational scheme (e.g. no hot redundancy on TT&C system, frequent switching on/off of the transponder, highly autonomous safe mode). The CDF team had to make some assumption on Clipper as not all information (e.g. bending frequency, available mass if ejection before JOI, schedule issues such as the delivery of STM & PFM) was available.

Several alternative system options were identified, such as the 1-swingby option which does comply with the 250 kg mass constraint, and the Europa option.

This Page Intentionally Blank

23 REFERENCES

RD[1] CDF-R-11, CDF Review of Europa Industrial Study, April 2006

RD[2] CDF-48(A), CDF Study of Jovian Minisat Explorer, July 2006

RD[3] CDF-61(A), CDF Study of Jupiter Radiation, July 2007

RD[4] The advanced Moon micro-imager experiment (AMIE) on SMART-1: Scientific goals and expected results, Pient et al., Planetary and Space Science 53 (2005) 1309-1318

RD[5] MarcoPolo-R Yellow Book, ESA/SRE (2013) 4, December 2013

RD[6] The Mercury Radiometer and Thermal Infrared Spectrometer (MERTIS) for the BepiColombo mission, Hiesinger et al., Planetary and Space Science 58 (2010), 144-165

RD[7] The thermal emission imaging system (THEMIS) for the Mars 2001 Odyssey mission, P. Christensen et al., Space Science Reviews 110, 85-130, 2004.

RD[8] Transient Water Vapor at Europa's South Pole, Roth et al., Sciencexpress, 10.1126, science.1247051, 12 December 2013.

RD[9] The lunar dust experiment (LDEX) for the lunar atmosphere and dust environment explorer (LADEE) mission, Horanyi et al., 40th LPSC, 1741.pdf, 2009.

RD[10] A highly minituarised laser ablation time-of-flight mass spectrometer for planetary exploration, Rohner et al. 2004, 35th LPSC, 7003.pdf

RD[11] ESA Planetary Protection Requirements, ESSB-ST-U-001

RD[12] Space product assurance - Bioburden control of cleanrooms, ECSS-Q-ST-70-58C

RD[13] Space product assurance - Dry heat bioburden reduction for flight hardware, ECSS-Q-ST-70-57C

RD[14] Space product assurance - Hydrogen peroxide bioburden reduction for flight hardware, ECSS-Q-ST-70-56C

RD[15] Space product assurance - Materials and hardware compatibility tests for sterilization processes, ECSS-Q-ST-70-53C

RD[16] Application of "MAEVA" hinge to MYRIADE microsatellite deployments needs, J. Sicre, D. Givois, A. Emerit, ESMATS 2005 - 11th European Space Mechanisms andTribology Symposium.

RD[17] Deployment of appendices using “Maeva” hinges:progress in dynamic models, Michel LATHUILIERE, Jacques SICRE, Fabrice GERMAIN, ESMATS 2009 - 13th European Space Mechanisms and Tribology Symposium.

RD[18] Airbus CHT-1: http://cs.astrium.eads.net/sp/spacecraft-propulsion/hydrazine-thrusters/1n-thruster.html

RD[19] Airbus CHT-20 : http://cs.astrium.eads.net/sp/spacecraft-propulsion/hydrazine-thrusters/20n-thruster.html

RD[20] Aerojet MR-103G : http://cs.astrium.eads.net/sp/spacecraft-propulsion/hydrazine-thrusters/20n-thruster.html

RD[21] Aerojet MR-106E: http://cs.astrium.eads.net/sp/spacecraft-propulsion/hydrazine-thrusters/20n-thruster.html

RD[22] ATK DS512: http://www.psi-pci.com/Data_Sheets_Library/DS512.pdf

RD[23] ATK DS468: http://www.psi-pci.com/Data_Sheets_Library/DS468.pdf

RD[24] ATK DS222: http://www.psi-pci.com/Data_Sheets_Library/DS222.pdf

RD[25] Abstract Star Tracker Design Considerations for the Europa Orbiter Mission

RD[26] Miniaturised Avionics for Planetary Landers (MINAVIO)

RD[27] Space Project Management, Risk Management, ECSS-M-ST-80C, 31 July 2008.

RD[28] "Guidelines for the use of TRLs in ESA programmes", ESSB-HB-E-002, Issue 1, Revision 0, 21 August 2013.

RD[29] "Europa Clipper Science and Reconnaissance Payload Proposal Information Package", JPL D-92256, 29 May

RD[30] TEC-SYC Cost Risk Procedure, TEC-SYC/5/2010/PRO/HJ, February 2010

RD[31] ESA Cost Engineering Charter of Services, Issue 4, TEC-SYC/12/2009/GRE/HJ

RD[32] Technology Readiness Levels Handbook for Space Applications, TEC-SHS/5551/MG/ap, Issue 1 Revision 6, Dated September 2008

RD[33] Planetary Protection Categorization for CLEO/P Mission concepts study – TEC-QI/15-204 20/01/2015

RD[34] Margin philosophy for science assessment studies SRE-PA/2011.097/ Ed 2.0 23/12/2014

RD[35] JUICE Environmental Specification Issue 5 Rev 3 Ref JS-14-09, 17/10/2014

RD[36] Clipper ESA Contribution / Science Study Team Meeting – Minutes of Meeting – Ref ESA-SRE-F-ESTEC-MIN-2015-003 – 29/01/2015

RD[37] Clipper ESA contribution / NASA-ESA Meeting #2 – ESA-SRE-F-ESTEC-MIN-2015-002 21/01/2015

24 ACRONYMS

Acronym	Definition
AIT/V	Assembly, Integration and Test/Verification
AIVT	Assembly, Integration, Verification and Test
Al	Aluminium
AOCS	Attitude and Orbit Control System
AOGNC	Attitude and Orbit Guidance Navigation and Control
AU	Astronomical Unit
AVM	Avionics Verification Model
BCR	Battery charge regulator
BDR	Battery discharge regulator
CaC	Cost at Completion
CCD	Charge Coupled Device
CER	Cost Estimation Relationship
CGA	Callisto Gravity Assist
CLEO	Clipper Europa Orbiter
CLEO/E	CLEO option Europa fly-by
CLEO/I	CLEO option Io fly-by
CLEO/P	CLEO option Europa penetrator
CLEP	Clipper Europa Penetrator
CMA	Cost Model Accuracy
COT	Crank Over the Top
CTE	Charge Transfer Efficiency
CU	Clean-Up (manoeuvre)
DET	Direct Energy Transfer
DHS	Data Handling System
DMM	Design Maturity Margin
DOA	Degree of Adequacy of the cost model
DoD	Depth of Discharge
DOR	Differential One-way Ranging
DSA	Deep Space Antenna

Acronym	Definition
DTE	Direct To Earth
ECSS	European Cooperation for Space Standardisation (Standards)
EFM	Electrical Functional Model
EGA	Europa gravity Assist
EIRP	Equivalent Isotropic Radiated Power
EM	Engineering Model
EMC	Electro-Magnetic Compatibility
EPE	External Project Events
EQM	Engineering and Qualification Model
ESA	European Space Agency
FM	Flight Model
FPA	Flight Path Angle
FPGA	Field Programmable Gate Array
FSS	Fine Sun Sensor
GA	Gravity Assist
GGA	Ganymede Gravity Assist
GSE	Ground Support Equipment
GSP	General Studies Program
GYR	Gyroscope
HGA	High Gain Antenna
IGA	Io Gravity Assist
IMU	Inertial Measurement Unit
IQM	Inherent Quality of the cost Model
IR	infrared
JC	Jovian Cruise
JOI	Jupiter Orbit Insertion
kGy	Kilo Gray
LGA	Low Gain Antenna
LoS	Line of Sight
LOS	Loss of Signal
MAG	Magnetometer

Acronym	Definition
MAIT	Manufacturing Assembling Integrating Testing
MDR	Mission Definition Review
MEMS	Micro Electro-Mechanical Systems
MLI	Multi-Layer Insulation
MOC	Mission Operations Centre
MPPT	Maximum power point tracking
MS	Mass Spectrometer
NC	Normally Closed
NIMS	Neutral and Ion Mass Spectrometer
NO	Normally Open
OCM	Orbit Control Mode
OD	Orbit Determination
ODM	Orbit Deflection Manoeuvre
OH	Optical Head
PCB	Printed Circuit Board
PCDU	Power Conditioning and Distribution Unit
PFM	Protoflight Model
PI	Principal Investigator
PLM	Perijove Lowering Manoeuvre
PLM	Payload Module
POE	Project Owned Events
PRM	Perijove Raising Manoeuvre
QIV	Quality of the Input Values
QM	Qualification Model
RARR	Range and Range-Rate
RCS	Reaction Control Subsystem
RF	Radio Frequency
R_J	Jovian radius (~71400 km)
RWL	Reaction Wheels
S/C	Spacecraft
STM	Structural Thermal Model

Acronym	Definition
SADM	Solar Array Drive Mechanism
SBI	Same Beam Interferometry
SCM	Science Mode
SFT	System Functional Test
SMA	Shape Memory Alloy
SoC	State of Charge
STM	Structural Thermal Model
STR	Star Tracker
SVM	Service Module
SVT	System Validation Test
TBC	To be confirmed
TBD	To be defined
TC	TeleCommand
TCM	Trim Correction Manoeuvres
TM	TeleMetry
TRL	Technology Readiness Level
TT&C	Tracking, Telemetry and Command
TWTA	Travelling Wave Tube Amplifier
UV	Ultra violet
VLBI	Very Long Baseline Interferometry
α	Solar absorptivity
ε	Infrared emessivity