LuPoiNT

Final Presentation - 16.06.2022

Systems Engineering

Final Presentation - 16.06.22

Alexandre Frantz
Patrick Teyssier
Sumit Goski

Structure to follow

Introduction to your subsystem (including the team)

Requirements and Design Drivers

Design (including inputs/outputs. SE will show the entire N2 chart but for details you are free to talk about them)

Components

COMET model + Iterations

Trade-offs (on your subsystem level if any)

Conclusions / Assessment

Contribution Slide (so that after the presentation they can see who worked on which slide)

Ex: Alex → Slides 1,2, 5 - 8

\+ Anything else you might want to showcase

Objective of the Project and main deliverables

Objective:

To design a Positioning, Navigation, and Timing (PNT) system and mission for the Moon to ensure the safety and operational success of future lunar missions

Deliverables:

Spacecraft design and mission features

Name:

LuPoiNT

Lunar, Luxembourg… to the point

Role and Duties of System Engineers

SEs, a cross-functional team coordinating the design effort

Research and investigation of past & existing lunar / PNT missions

Define mission requirements, operations, trajectory, system overall requirements, budget allocations

Use SMAD, COMET, EXCEL, GMAT

Ensure project management, scheduling, consistency

Handle conflicts across subsystems, team progress, design iterations

Individual Responsibilities

Individual	Responsibilities
Alex	Overall Team Management CONOPs System Mass and Power Budgets
Patrick	System Mass and Power Budgets
Sumit	Trajectory Design (GMAT, Calculations) Orbit Design Configuration
All	Literature Review, Report Writing, N2 Chart, Requirements, Overall Mission Analysis, COMET, Spacecraft Iterations

Mission Analysis

Mission Objectives

LuPoiNT Mission Objectives
Primary Mission Objectives: 1. To establish an operational constellation of PNT satellites orbiting the Moon   2. To provide a PNT service covering the entire lunar surface, serving users, robotic or manned missions, on the Moon
Secondary Mission Objectives: 1. Demonstrate lifetime and suvivability of a lunar constellation of satellites   2. Support scientists in understanding the lunar geology and topography   3. Support the elaboration of lunar precise mapping   4. Support exploitation of in-situ resources

Mission Requirements

Critical Requirements:

- ➢ MREQ 01 (Coverage)
- ➢ MREQ 04 (Lifetime)
- ➢ MREQ 09 (Coverage)

ID	Requirement
MREQ 01	The mission shall provide a PNT service available over the entire Lunar surface
MREQ 02	The mission shall consist of 20–satellite constellation in two different orbital planes, that describe a circular inclined orbit.
MREQ 03	The two orbital planes shall have inclinations of 45° and 135°, a semi-major axis of 4000 km, and an orbital period 6 hours
MREQ 04	The mission lifetime shall be approximately 10 years
MREQ 05	When the mission's end of life has been reached, there shall be at least 18 satellites that are still operational – MTBF
MREQ 06	The entire mission operations shall not interfere with existing Lunar operations
MREQ 07	The launch vehicle shall place the spacecraft directly in a TLI.
MREQ 08	The PNT satellites shall be tracked through an existing Gateway station orbiting the Moon
MREQ 09	The Lunar ground segment elements shall have a minimum of 4 satellites in their FOV
MREQ 10	The mission shall maintain stable orbital parameters and attitude with respect to the Moon

Mission Constraints & Assumptions

→ No constraints have been imposed

→ Assumptions have been made to ensure feasibility and simplify the design process

ID	Assumption
MAS-REQ 01	The launch phase does not require an intermediate parking orbit, and that the launch vehicle is capable of inserting the constellation into the TLI
MAS-REQ 02	Technological maturity is reached at mission launch
MAS-REQ 03	There already exists a Ground Station infrastructure on the Moon that support our PNT operations. The same infrastructure is assumed to be further used as a relay station for communications between the constellation and Earth
MAS-REQ 04	The Navigation to the Moon is semi-autonomous therefore reducing communication power usage during Cruise to the Moon

Mission Risks

ID	Risk	Mitigation Strategy
MR 01	Launch vibrations causing structural failure	Design a stiff structure with damped mounting. Test and validate the design
MR 02	Launch vibrations causing combustion instability by increasing turbulence in the propellant tanks	Use a propellant tank designed against sloshing due to vibrations and motion using proven sloshing suppression methods
MR 03	Space Debris collision in LEO before TLI	Pre-mission analysis with Space Traffic Management Organisation to perform collision probability assessment
MR 04	Trans-lunar Injection accelerations cause a lot of stress on the structures	The TLI can be done in multiple small Delta-V increments instead of doing it in a single burn which will induce multiple smaller magnitudes of stress on the structure than one large high magnitude stress
MR 05	Failure of separation mechanisms to separate the spacecraft with the kick-stage	Use a Pyrotech bolt for redundancy
MR 06	Risk of collision of the spacecrafts among themselves during separation from kickstage	Planned operations where the aft-most spacecraft separates first and then the one in front of it after a few seconds of delay and so on, leading to the fore-most spacecraft separating at last
MR 07	Uneven cooling/heating of the spacecraft during Cruise due to the partial incidence of sunlight on the spacecraft body	Put the spacecraft in a slow spin during the cruise so that the entire surface of the spacecraft body is exposed to sunlight ensuring even heating/cooling with an active-passive thermal management system
MR 08	High braking forces of the spacecraft during Low Lunar Capture	Multiple burn capture where the first burn will put the spacecraft in an elliptical orbit and the consequent burns which will park the spacecraft in LLO
MR 09	Risk of collision of the spacecrafts among themselves in LLOI while performing orbit change maneuvers	Calculate the appropriate phasing parameter for the constellation using high fidelity simulation

Concept Of Operations (CONOPs)

Mission Phases

- Earth-Centric: events that occur in the vicinity of Earth
- Lunar Transfer: events that occur right after TLI initiation and during TLI
- Moon-Centric: events that occur in the vicinity of the Moon
- Disposal: sequence of events to dispose the spacecraft at EoL

Concept Of Operations (CONOPs)

Launch

Launch sequence from launch pad until orbit altitude to initiate TLI

TLI

Kickstage inserts constellation in TLI. Once Delta-V achieved, constellation and kickstage separate

Cruise

The constellation cruises in the desired trajectory path to reach the Moon. Solar panels are deployed, Trajectory corrections are performed when required

Constellation Formation

Spacecrafts enter LLOI, then LOI,before performing phasing and pointing manoeuvers to form a constellation. Testing is further performed with the Payload to ensure that services can be provided

PNT Operations

Commissioning, Station keeping and maintenance checks throughout lifetime

Decommissioning

EoL is reached, spacecrafts are sent to disposal

Concept Of Operations (CONOPs)

Each CONOPs phase is broken down into steps

Illustrates the flow of events throughout mission lifetime

LuPoiNT Mission Sequence	
Mission Phase	**Sequence**
Earth-Centric	1. Launch 2. TLI 2.1 Separation from Kickstage 2.2 Trajectory Corrections
Lunar Transfer	3. Cruise 3.1 Solar panel deployment 3.2 Cruise (free motion) 3.3 Trajectory corrections
Moon-Centric	4. Constellation Formation 4.1 LLOI 4.2 LOI 4.3 Satellite Phasing/Pointing 4.4 Testing 5. PNT Operations 5.1 Commissioning 5.2 Provide PNT 5.3 Orbit Corrections
Disposal	6. Decommissioning 6.1 Spacecraft EOL Disposal 6.2 Switch off Payload

Concept Of Operations (CONOPs)

Trade-offs were explored:

1. Separation from the Kickstage near Earth
2. Separation from the Kickstage near Moon

Reasoning:

- Save fuel for Trajectory Corrections during Cruise
- Kickstage inserts satellites in LLOI → further save fuel

Seperate from kickstage near Earth

LuPoiNT Concept of Operations (CONOPs)

Phase	Step	SM	POW	ADC	PRP	COM	DH	THM	P/L
Earth Centric Phase	1. Launch		x				x		
	2. TLI		x				x		
	2.1 Separation from Kickstage	x	x			x			
	2.2 Trakectory Corrections		x	x		x			
Lunar Transfer Phase	3. Cruise								
	3.1 Solar panel deployment	x	x	x			x	x	
	3.2 Cruise (free motion)		x	x		x	x	x	
	3.3 Trajectory Corrections		x	x	x	x	x	x	
Moon Centric Phase	4. Constellation Formation								
	4.1 LLOI		x	x	x	x		x	
	4.2 LOI		x	x	x	x		x	
	4.3 Satellite Phasing/Pointing		x	x	x				
	4.4 Testing					x	x		x
	5. PNT Operations								
	5.1 Commissioning		x			x	x		x
	5.2 Provide PNT	x	x	x		x	x	x	x
	5.3 Orbit Corrections		x		x	x			
Disposal Phase	6. Decomissioning								
	6.1 Spacecraft EOL Disposal				x				x
	6.2 Switch off Payload					x	x		x

Seperate from kickstage near Moon

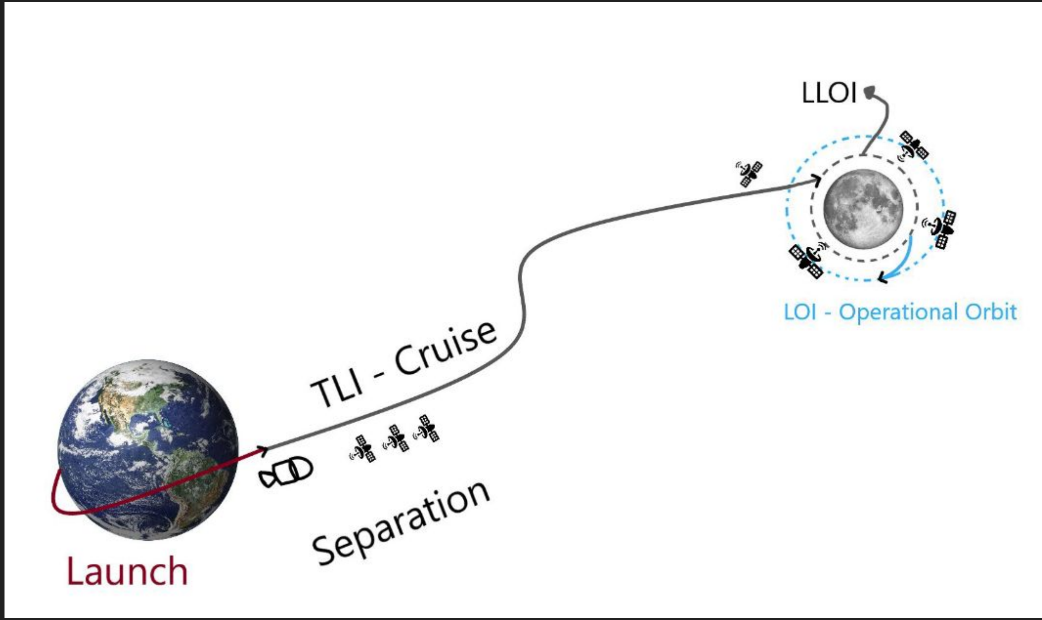
LuPoiNT Concept of Operations (CONOPs)

Phase	Step	SM	POW	ADC	PRP	COM	DH	THM	P/L
Earth Centric Phase	1. Launch		x				x		
	2. TLI		x				x		
Lunar Transfer Phase	3. Cruise								
	3.1 Solar panel deployment								
	3.2 Cruise (free motion)					x	x	x	
	3.3 Trajectory Corrections								
Moon Centric Phase	4. Constellation Formation								
	4.1 LLOI		x	x	x	x		x	
	4.2 Separation from Kickstage	x	x			x	x		
	4.3 Solar panel deployment	x	x	x			x	x	
	4.4 LOI		x	x	x	x		x	
	4.5 Satellite Phasing/Pointing		x	x	x				
	4.6 Testing					x	x		x
	5. PNT Operations								
	5.1 Commissioning		x			x	x		x
	5.2 Provide PNT	x	x	x		x	x	x	x
	5.3 Orbit Corrections		x		x	x			
Disposal Phase	6. Decomissioning								
	6.1 Spacecraft EOL Disposal				x				x
	6.2 Switch off Payload					x	x		x

Concept Of Operations (CONOPs)

Separation near Earth chosen

→ No kickstage to separate near Moon was found

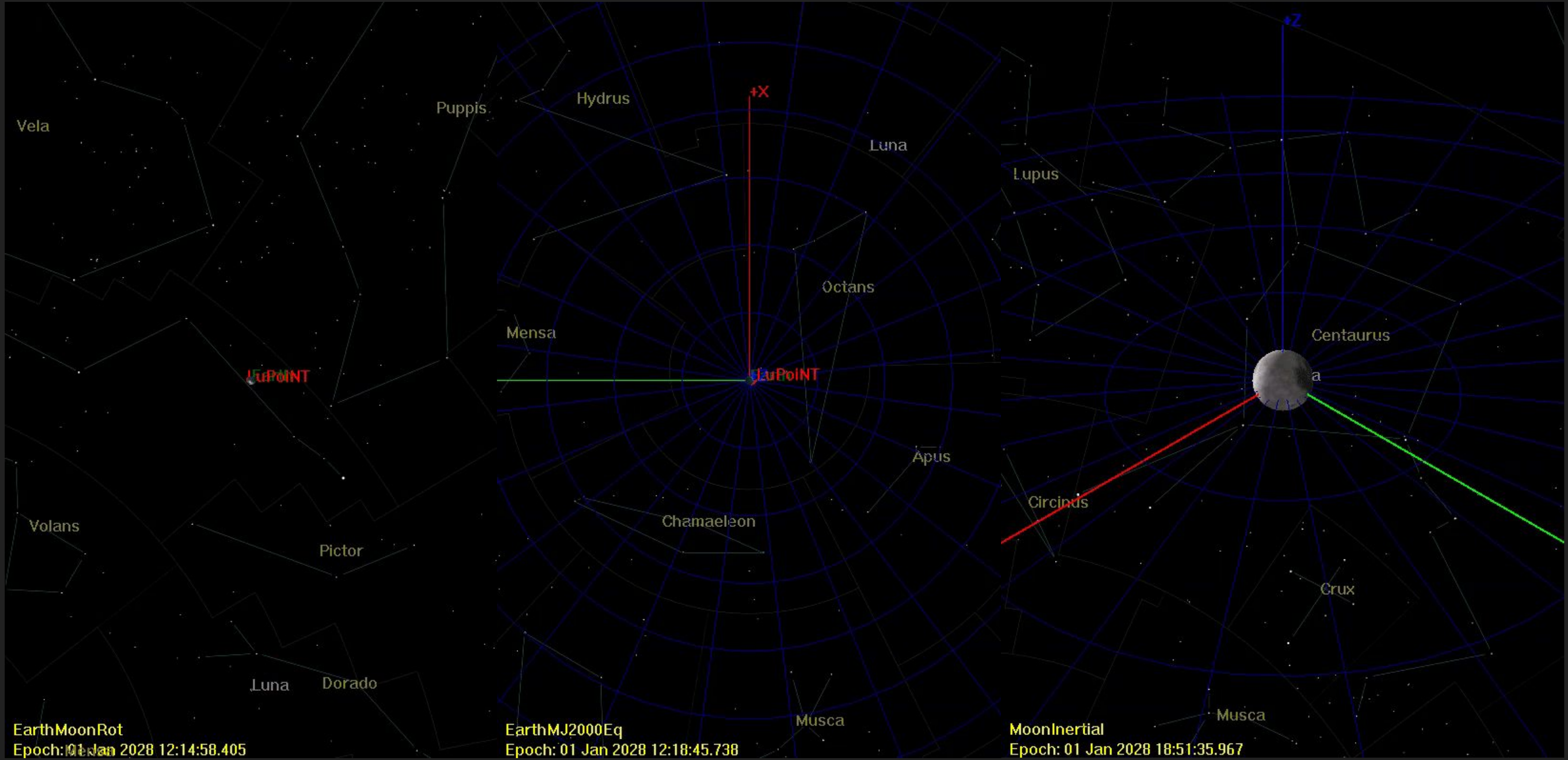


Trajectory - Earth to Moon

Many ways to do it with varying degrees of accuracy vs complexity

- Direct Hohmann transfer from Earth to Moon for a semi-major axis of 384,000 km
- B-plane targeting for the desired final orbit parameters
- Low energy transfer from Libration points and Free-return trajectories

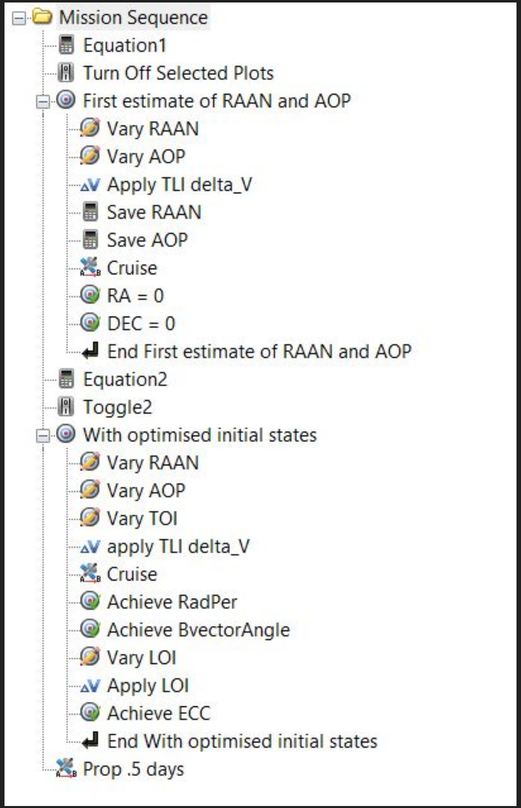
Trajectory - Earth to Moon



Trajectory - Earth to Moon

Process followed

- Vary TLI velocities and initial orbit parameters like RAAN, AOP and True anomaly to match the B-plane for a direct injection.
- Achieve B-vector angle of 90 deg which decides inclination of the LLO (Polar)
- Achieve Lunar radial distance of 4000 km.



Trajectory - Delta-V for Comet

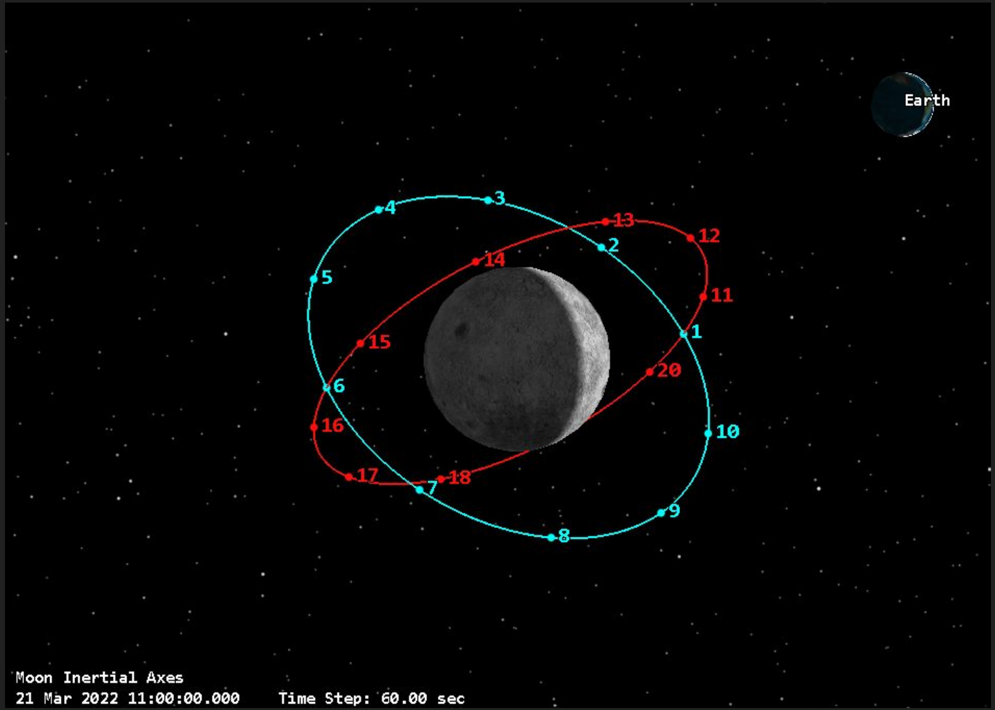
Method used - Patched Conics approximation

- TLI : Literature review and from Past missions - 1900 m/s
- LLOI : (Final Circular Orbit velocity - Hyperbolic excess velocity) - 500 m/s
- LOI : Hohmann transfer delta-V from LLO (100) to Operational Orbit altitude (4000) with inclination change ~ 650 m/s
- De-commissioning : Hohmann transfer delta-V from Operational Orbit (4000) to 50 km parking orbit ~ 500 m/s
- Station-keeping and Orbit Maintenance : Literature review from Past missions ~ 150 m/s

Trajectory - Orbit Design for PNT Constellation

Orbit selected

- Two Orbital planes at an inclination of 45 and 135 degrees respectively.
- Each plane has 10 satellites with a True anomaly difference of 36 degrees.
- Time period of each satellite is approximately 6.3 hours or 22700 seconds.



Mission Trade-offs

- Orbit Planes (numbers, inclination, etc)

- CONOPs

Systems

System Requirements

Critical Requirements:

- ➢ SREQ 01
- ➢ SREQ 02

ID	Requirement
SREQ 01	The satellite constellation shall consist at least of three communication links: Inter-satellite, Moon-Spacecraft, and Spacecraft-End user
SREQ 02	The spacecraft dry mass shall not surpass 1000 kg
SREQ 03	The spacecraft wet mass shall not surpass 1400 kg
SREQ 04	When direct line of sight with the Sun is possible, the spacecraft shall point the solar panels towards the Sun. The solar panels shall always be deployable and remain deployed throughout the mission
SREQ 05	When direct line of sight with the Sun is not possible, the spacecraft(s) shall operate using the on board batteries
SREQ 06	The peak power consumption of the spacecraft shall not exceed the 1550 W
SREQ 07	The peak power consumption of the payload shall not exceed the 800 W
SREQ 08	The pointing requirements shall be as follows: slew rate of 0.1 °/s, and pointing accuracy of 1°

System Assumptions

→ The payload is considered a black box and its specifications (mass, power, dimensions) are defined using the literature review

ID	Assumption
SAS-REQ 01	The constellation will be launched with the SLS launch vehicle
SAS-REQ 02	The payload is considered a black box
SAS-REQ 03	The CDH subsystem is considered a black box

System Drivers

- CONOPs
- Payload Dimensions
- Payload Mass
- Payload Power Consumption
- Specific placement of antennas and sensors
- Type of Propulsion system used

System Budgets

Dry mass
997 kg
Wet mass
1233 kg

Power
1668 W

No CONOPS optimization

SMAD analysis:
overall mass & power + subsystem breakdown

Initial budget allocations:
Nominal dry mass 850 kg + margins, wet mass 1400 kg
Total power with margins 1550 W

Initial power and mass budgets

Budgets after iterations

System Budgets – hybrid PRP - ESA margins

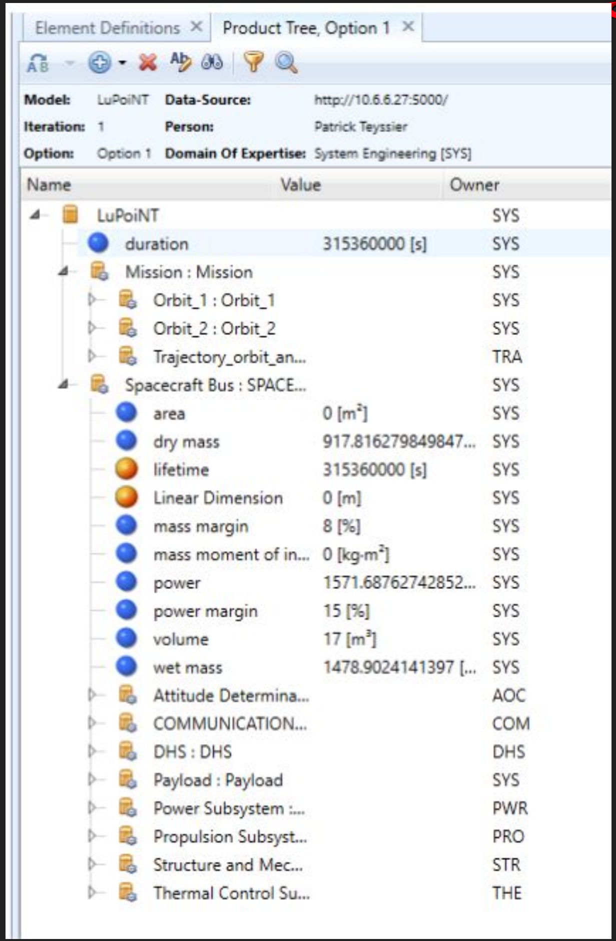
Subsystem	Mass in kg w/margin	Power in W w/margin	Margins (ESA)	Comment
Payload	210	840	5% / 5%	Black box
Propulsion PRP	74	**0**	5 / 5	Hybrid, below budget
Power POW	256	304	5 / 5	Above budget W
Structure S&M	158	0	5 / 5	Slightly above budget
Attitude Control ADC	29	87	5 / 5	
COM & CDH	40	107	10 / 10	Better electronics
Thermal TCS	26	53	5 / 5	
Propellant	236	N/A	2	
Harness	38	N/A	0	As per ESA philosophy
TOTAL dry/wet	**997 / 1233**	**1668**	20 / 20	With system margins

N2 Chart for LuPoiNT mission & systems

Inputs		Outputs
Payload Mass and Volume	**PAYLOAD**	-
Launch Vehicle, Dry mass, Mission parameters	**STRUCTURE**	Structure Volume and Mass
Mass budget, Power budget, dry mass, Delta-V budgets	**PROPULSION**	Propellant Mass
Power budget, Mass budget, Eclipse time, Orbit details	**POWER**	EOL Power available, Total Power
Spacecraft Mass, Surface area, Orbit details, Eclipse and Sunlight times	**THERMAL**	-
Satellite wet mass, Power budget, CONOPs, Orbit details	**ADC**	-
Ground Station access and passes, Intersat distance, Slant range, etc	**COM & CDH**	Frequency bands, Link budget

COMET Model

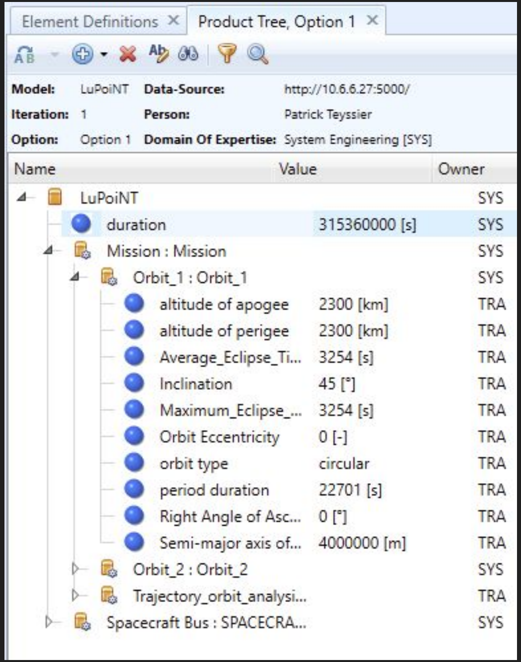
- LuPoiNT
  - Mission
    - Orbit 1
    - Orbit 2
    - Trajectory
  - Spacecraft Bus
    - ADC
    - COMMS
    - …



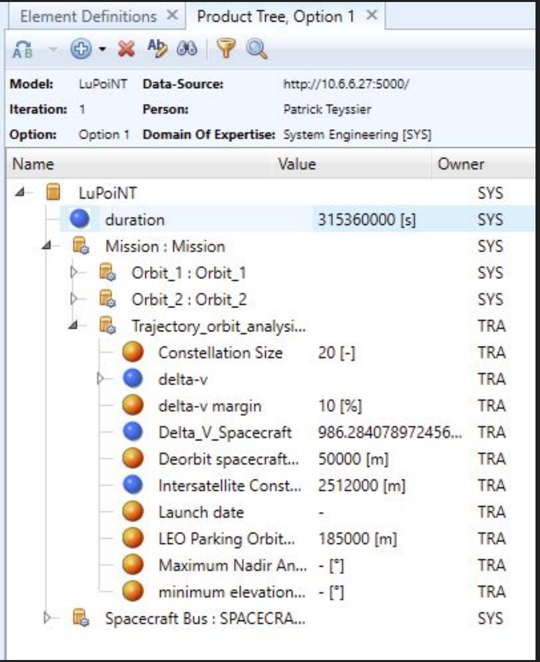
COMET Model

The Orbit Elements describe the mission orbits.

Identical values except for the inclination (45 and 135)



The Trajectory Element describes the constellation's trajectory and delta-v manoeuvers



Spacecraft Iterations

COMET was used for MBSE

Excel was used as our analytical tool for calculations and updating the model parameters

A total of three iterations have been analysed in the CDF:

1. Orbit altitude 2,300 km
2. Orbit altitude 5,000 km vs 2,300 km
3. Orbit altitude 2,300 km & Hybrid propulsion

Performance Metrics → Mass & Power

Iteration 1: Orbit Altitude 2,300 km

Orbit altitude of 2300 km with chemical propulsion

—

Results:

→ Overall budgets are below initial allocation

- Dry mass: 851 kg | 1000 kg
- Wet mass: 1212 kg | 1400 kg
- Power consumption: 1497 W | 1550 W
- Propellant weight: 360 kg | 400 kg

—

POWER Subsystem:

→ Over Budget in terms of mass (>25 kg) and power (>210 W)

Mass Budget			
Subsystem	Mass excl. Margin (kg)	Mass incl. Margin (kg)	Mass Margin (%)
S&M	150	165	10
POW	229	252	10
ADC	27	30	10
PRP	47	52	10
COMM	13	14	10
DH	25	28	10
THM	25	28	10
Payload	200	220	10
Total Dry Mass	717	851	8
Propellant		361	0
Total Wet Mass		1212	

Power Budget			
Subsystem	Power excl. Margin (W)	Power incl. Margin (W)	Power Margin (%)
S&M	0	0	0
POW	270	310	15
ADC	83	96	15
PRP	0	0	15
COMM	49	56	15
DH	50	58	15
THM	50	58	15
Payload	800	920	15
Total Power	1302	1497	

Iteration 2: Orbit Altitude 5,000 km

Comparison with iteration 1:

Dry mass **+ 8%** due to POW, PRP

Propellant mass **+ 55%** due to mass, ΔV

Mass & Power over budget

Mass Budget			
Subsystem	Mass excl. Margin (kg)	Mass incl. Margin (kg)	Mass Margin (%)
S&M	150	165	10
POW	268	295	10
ADC	27	30	10
PRP	64	70	10
COMM	13	14	10
DH	25	28	10
THM	25	28	10
Payload	200	220	10
Total Dry Mass	772	918	8
Propellant		561	0
Total Wet Mass		1479	

Power Budget			
Subsystem	Power excl. Margin (W)	Power incl. Margin (W)	Power Margin (%)
S&M	0	0	0
POW	308	354	15
ADC	83	96	15
PRP	0	0	15
COMM	49	56	15
DH	50	58	15
THM	50	58	15
Payload	800	920	15
Total Power	1340	1541	

Iteration 3: Hybrid Propulsion, alt. 2,300 km

Comparison with iteration 1:

Dry mass **+ 5%** due to POW, PRP

Propellant mass **- 35%** due to hybrid

Mass well below budget

Power slightly above, electrical propulsion used when other systems not in operation

Mass Budget			
Subsystem	Mass excl. Margin (kg)	Mass incl. Margin (kg)	Mass Margin (%)
S&M	150	165	10
POW	244	268	10
ADC	27	30	10
PRP	70	77	10
COMM	13	14	10
DH	25	28	10
THM	25	28	10
Payload	200	220	10
Total Dry Mass	755	896	8
Propellant		236	0
Total Wet Mass		1133	

Power Budget			
Subsystem	Power excl. Margin (W)	Power incl. Margin (W)	Power Margin (%)
S&M	0	0	0
POW	289	333	15
ADC	83	96	15
PRP	0	0	15
COMM	49	56	15
DH	50	58	15
THM	50	58	15
Payload	800	920	15
Total Power	1322	1520	

Iteration 3: Hybrid Propulsion / ESA margins

CONCLUSIVE REMARKS ON BUDGETS:

Mass 997 kg / 1233 kg power 1668 W

Subsystem margins reduced to 5% or 10% ⇐ TRL

Delta-V margins already conservative

System-level margins increased to 20%

Harness added

Power above budget by almost 200 W

Conclusions & Future Work

- Requirement analysis, validation and verification to be improved
- Further expand on the CONOPs.
- Use D&H actively in the design process

- Explore Low Energy Lunar transfer trajectories
- High fidelity trajectory simulation to get accurate launch window

- Detailed configurations (size and volume) and systems packaging studies for stowed and deployed states
- Mass budgets to be fine tuned / POW, PROP, include values for CDH
- Optimize power budget considering not all systems are in operation simultaneously. Consideration on BOL/EOL.

Slide Contribution

Sections	Individual
Mission Requirements CONOPs System Requirements System Assumptions COMET Model Spacecraft Iterations	Alex
Mission & system requirements Power and Mass budgets N2 diagram Spacecraft Iterations	Patrick
Trajectory Earth-Moon Orbit design Possible baseline configurations Risks	Sumit

<subsystem_name>

<abreviation>

Progress Report - 12.05.22

<Name1>
<Name2>

General Guidelines

- Each group should introduce themselves
- Each group has at most 7 minutes to cover all their slides
- Follow the template to keep consistency
- Asside from the structure that follows in the next slides, you can add anything you like as long as you respect the time limit. If you need more time, let everyone know and perhaps we can divide it differently
- Any improvements, suggestions etc.. for the presentation are as always welcome :)

Structure to follow

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Design (including inputs/outputs. SE will show the entire N2 chart but for details you are free to talk about them)

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COMET model + Iterations

Trade-offs (on your subsystem level if any)

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Ex: Alex → Slides 1,2, 5 - 8

\+ Anything else you might want to showcase

Structures and Mechanisms

SM

Final Presentation - 16.06.22

Aelyn Chong Castro

S&M: Introduction

The structure and mechanisms subsystems house the spacecraft subsystems during all their lifecycle. It attaches, and detaches from the launch vehicle and keep the subsystems safe from vibration, and environmental hazards.

For cislunar operations, the structure must endure a gravitation free environment. Meaning, that high loads, high acceleration, and severe vibrations are experienced by the spacecraft.

Choosing the right shape and material can improve not only the lifetime, but also the reliability of the spacecraft.

Requirements

ID	Requirement	Parent ID
SMR 01	The spacecraft design shall have a safety factor of 2.	MREQ 03 ; MREQ 04
SMR 02	The structure design shall accommodate all the spacecraft subsystems.	MREQ 10
SMR 03	The spacecraft shall cover instruments to avoid contamination from burn, and structure out gassing.	MREQ 03
SMR 04	The spacecraft shall accommodate launch loads, and provide attenuation to impacts.	MREQ 03
SMR 05	The spacecraft mechanism shall use existing technologies to deploy and drive the antennas and solar panels.	–
SMR 06	The spacecraft shall fit in the launch vehicle.	MREQ 07; SREQ 09
SMR 07	The spacecraft structure and mechanism shall be able to survive the launching environment.	MREQ 03
SMR 08	The spacecraft shall separate from the launching vehicle during the cruise phase.	–
SMR 09	The spacecraft shall deploy the solar panels and antennas during the cruise phase.	–
SMR 10	The solar panels driven mechanism shall assure mobility to point towards the power source.	SREQ 07
SMR 11	The structure has a maximum volume of 1.7 m^3	SREQ 09

Design Drivers

Structural Mechanical

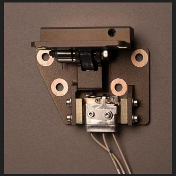
Inputs & Outputs - N2 Chart

Inputs		Outputs
Volume and mass	**PAYLOAD**	Structure volume and mass
Number of satellites in constellation	**TRAJECTORY AND ORBIT**	-
Size and location of engine and thruster, mass of propellant	**PROPULSION**	Dry Mass
Area of solar panels. Deployable and/or driven solar panels	**POWER**	Location of solar panels Spacecraft shape
Dimensions of thermal system. Mass and location	**THERMAL**	Structure material: Aluminum
Dimensions and mass	**ADC**	Size, shape, surface area, mass.
Deployable and/or driven antennas	**COM & CDH**	-

Mechanical Components

Phase	Launch	TLI	Cruise	Constellation Formation	PNT Operations	Decommissioning
Timeline	S/C escapes Eath’s gravity	**Separation from kickstage**, Trajectory corrections	**SP deployment,** Cruise (free motion),Trajectory corrections	LLOI, LOI, Satellite phasing/pointing, Testing	Commissioning, **Provide PNT**, Orbit corrections	Spacecraft EOL Disposal, Switch off payload
Mechanism		- Pyrotechnic bolt	- Hinge- and spring-based deployment mechanism		- ⅗ Solar Array Drive Assembly (SADA)	

Solar Panels Deployment

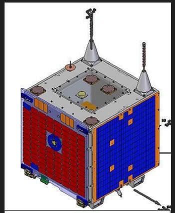


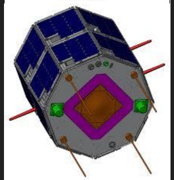
	Quantity	Dimension (mm)	Mass (g)	Power (W)	Temperature (°C)
Hinge- and spring-based deployment mechanism	8	68.5 x 45.5 x 14.0	560	100	-40 to 100

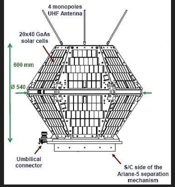
Solar Panels Driver

	Quantity	Dimension (mm)	Mass (g)	Power (W)	Temperature (°C)
⅗ Solar Array Drive Assembly (SADA)	2	25.4 x 13.36	4560	20	-30 to 60

Satellite Shape Trade-off



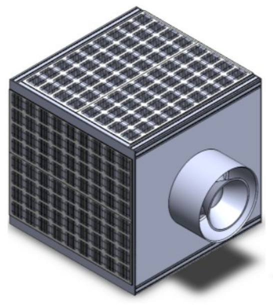


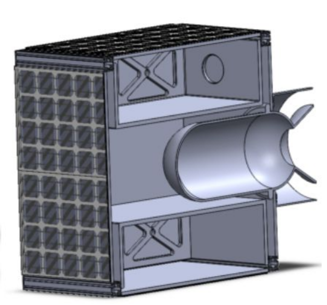


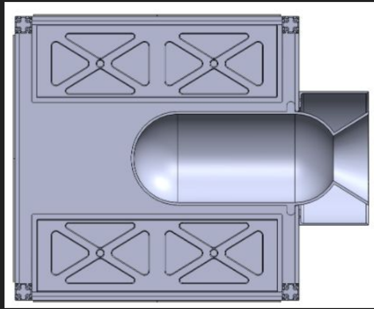
Shape	Advantages	Disadvantages
Cube / Rectangular Prism	- Easy to manufacturer - Less material - Less weight - Lower cost - Large surface area for the same volume - Easy heat dissipation - Less number of joints - Easy solar arrays integration	- Problem in arranging subsystems due to the sharp corners - Need for deployable solar panels for high power missions. This involves extra weight, and higher risk of failure. - Lesser surface
Sphere	- Large volume - Less material	- Difficult to manufacture - Difficult to utilize space due to curved surface - Difficult launch vehicle integration
Hexagonal Prism	- Larger volume than the cube for the same quantity of material. - Less sharp corners in comparison to the cubic shape satellite	- Angular constraints make it difficult to build - Too many joints increase weight - Affect panel folding
Octagonal	- Lower volume than the cubic shape for the same quantity of material	- More difficult to build - More number of joint, resulting in higher risk of failure

Spacecraft Structure Iterations

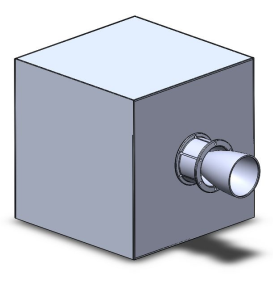
1st Iteration

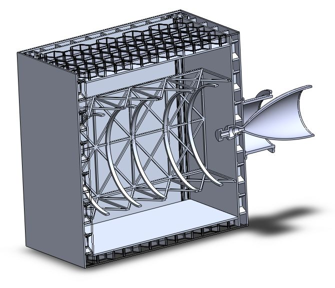


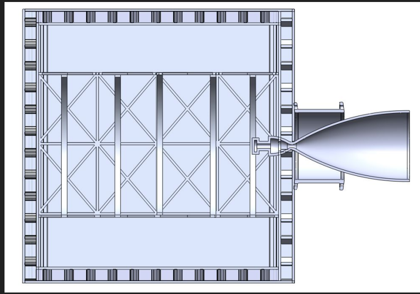




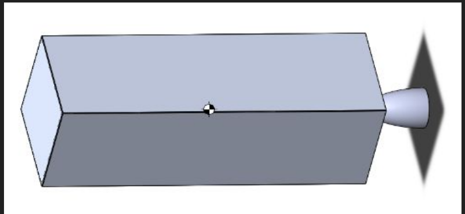
2nd Iteration

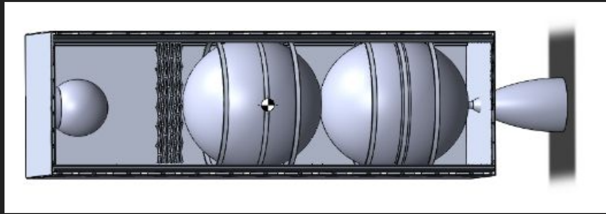




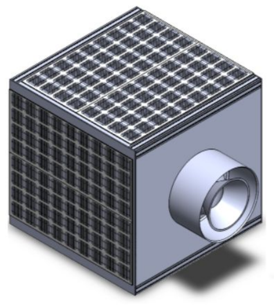


3rd Iteration

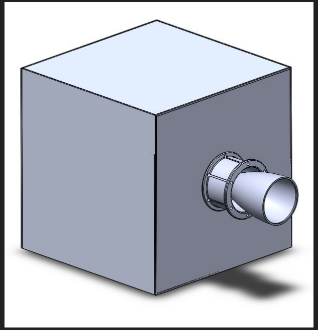




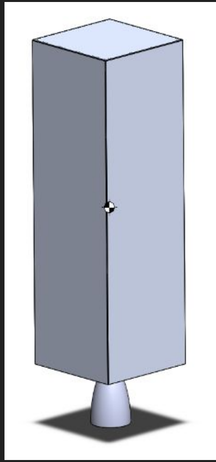
Spacecraft Structure Iterations



1st Iteration



2nd Iteration



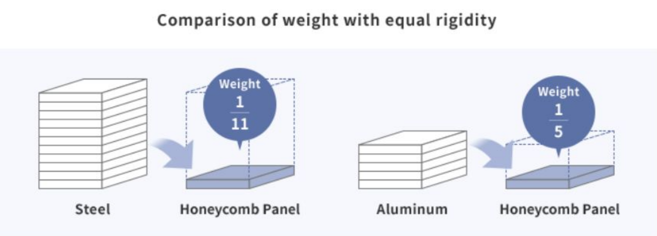
3rd Iteration

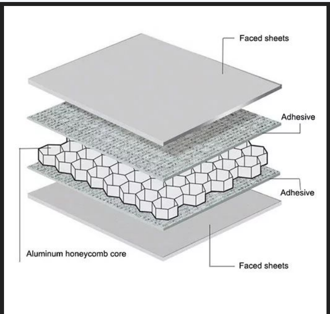
Shape	Cube	Cube	Rectangular Prism
Dimensions (m) (HxWxL)	1.2 x 1.2 x 1.2	1.2 x 1.2 x 1.2	2.4685 x 0.8317 x 0.8317
Mass (kg)	735	622.71	330.32
Volume (m^3)	1.72	1.72	1.71
Surface area (m^2)	8.24	8.24	9.6
Center of mass (m)	X = 0.70, Y = 0.25, Z = 1.10	X = 0.01, Y = 0.61, Z = 0.56	X = 0.45, Y = 1.63, Z = 0.56
Moments of Inertia ($kg*m^2$)	L_{xx} = 462.05 L_{yy} = 493.69 L_{zz} = 519.51	L_{xx} = 222.90 L_{yy} = 225.45 L_{zz} = 220.67	L_{xx} = 235.54 L_{yy} = 67.09 L_{zz} = 235.55

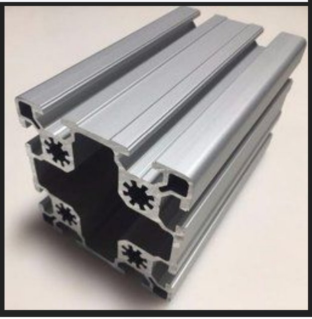
Satellite Material Trade-off

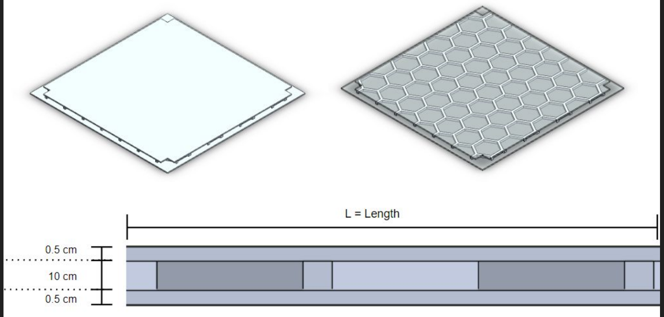
Material	Advantages	Disadvantages
Aluminum	- High strength vs weight - Ductile; tolerant of concentrated stress - Easy to machine - Low density; efficient in compression	- Relative low strength vs. volume. - Low hardness - High coefficient of thermal expansion
Steel	- High strength. - Wide range of strength, hardness, and ductility obtained by treatment.	- Not efficient for stability (high density) - Most are hard to machine - Magnetic
Titanium	- Low strength vs weight - Low coefficient of thermal expansion.	- Hard to machine - Poor fracture toughness if solution treated and aged
Composite	- Can be tailored for high stiffness, high strength, and extremely low coefficient of thermal expansion - Low density - Good in tension (e.g., pressurized tanks)	- Costly for low production volume; requires development program - Strength depends on workmanship; usually requires individual proof testing - Laminated composites are not as strong in compression - Brittle; can be hard to attach

Material: Al 6061



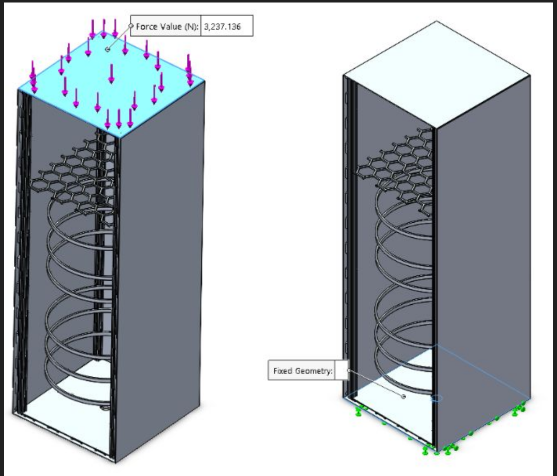






Spacecraft Structure FEM Analysis

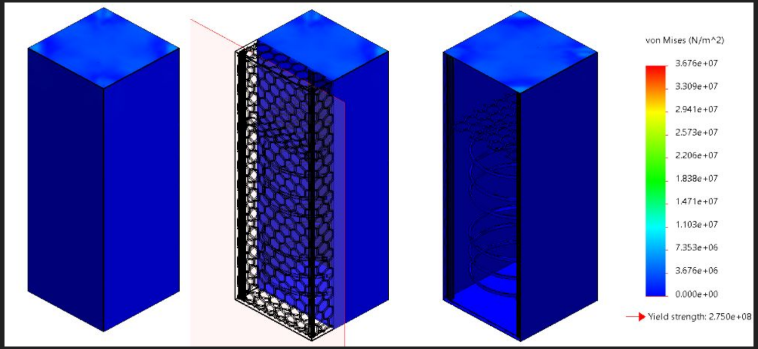
Force

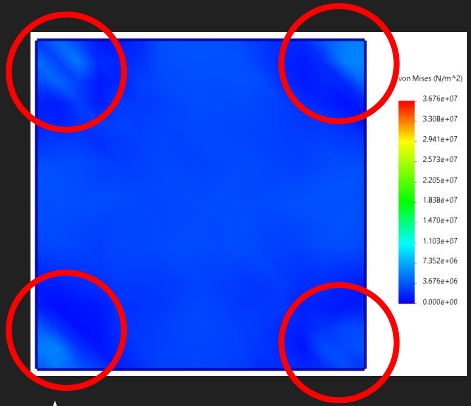


Since the spacecraft mass (m) is 330.32 kg, we multiply this by 9.8 N of gravity (g) resulting in 3 237.136 N force of gravity pulling it downwards,

Stress

Stress may cause deformation or fractures in materials.





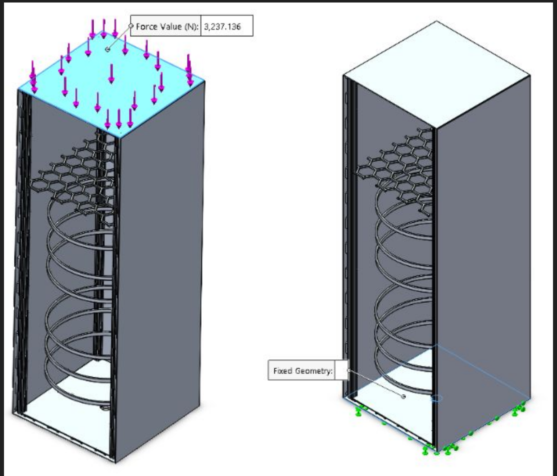
Al6061 T6:
Yield Strength = 275 000 000.9 N/m^2

>

Maximum stress is in the corners ~ 1.103e^7 N/m^2

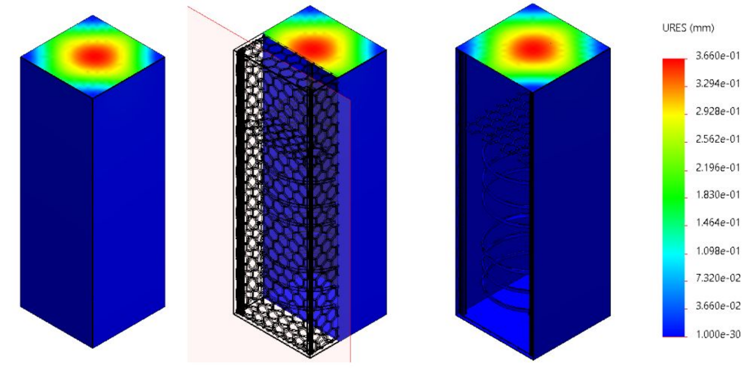
Spacecraft Structure FEM Analysis

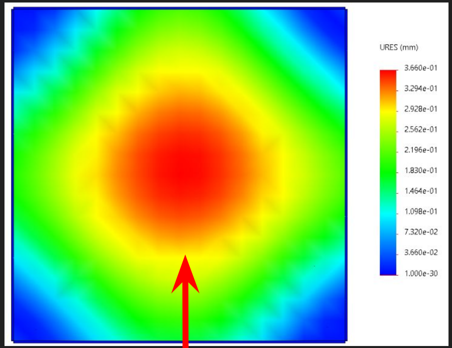
Force



Since the spacecraft mass (m) is 330.32 kg, we multiply this by 9.8 N of gravity (g) resulting in 3 237.136 N force of gravity pulling it downwards,

Displacement

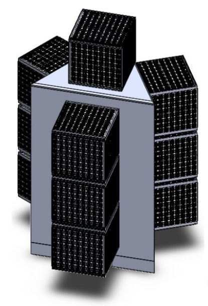


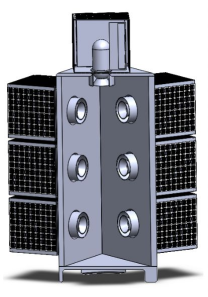


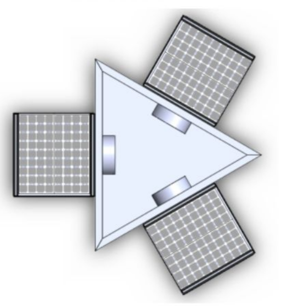
The biggest displacement happens at the top center of the spacecraft, moving 3.66 e^{-1} mm. The displacement propagates on a circular motion towards the edges, when the displacements can be neglected (1.00 e^{-30} mm) do to the support the aluminum profiles are providing

Payload Adapter Iterations

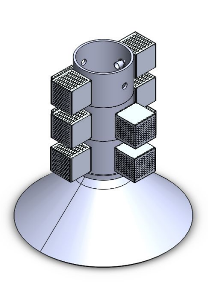
1st Iteration

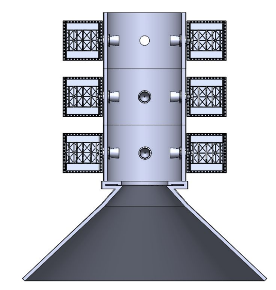


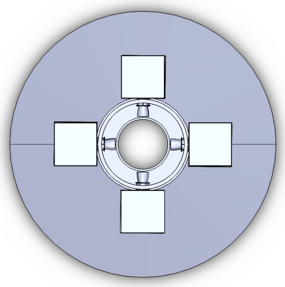




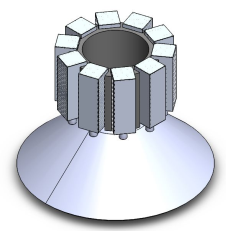
2nd Iteration

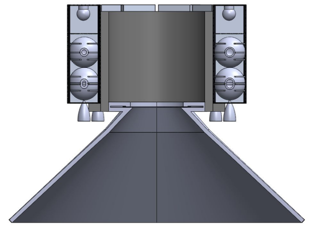


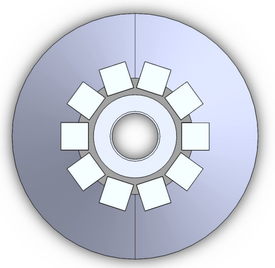




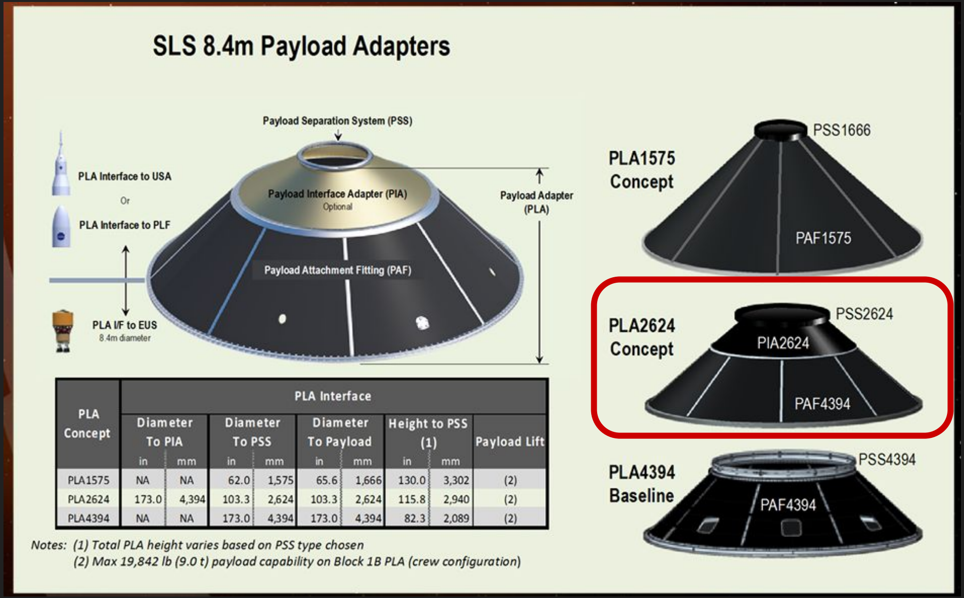
3rd Iteration





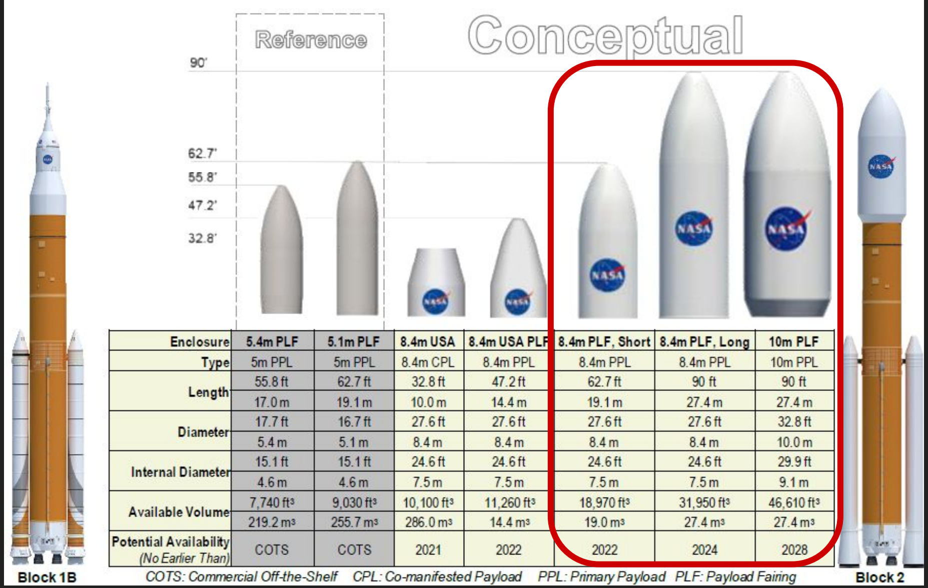


SLS Payload Adapter and Fairing



PLA Concept	PLA Interface								
	Diameter To PIA		Diameter To PSS		Diameter To Payload		Height to PSS (1)		Payload Lift
	in	mm	in	mm	in	mm	in	mm	
PLA1575	NA	NA	62.0	1,575	65.6	1,666	130.0	3,302	(2)
PLA2624	173.0	4,394	103.3	2,624	103.3	2,624	115.8	2,940	(2)
PLA4394	NA	NA	173.0	4,394	173.0	4,394	82.3	2,089	(2)

Notes: (1) Total PLA height varies based on PSS type chosen
(2) Max 19,842 lb (9.0 t) payload capability on Block 1B PLA (crew configuration)



Enclosure	5.4m PLF	5.1m PLF	8.4m USA	8.4m USA PLF	8.4m PLF, Short	8.4m PLF, Long	10m PLF
Type	5m PPL	5m PPL	8.4m CPL	8.4m PPL	8.4m PPL	8.4m PPL	10m PPL
Length	55.8 ft	62.7 ft	32.8 ft	47.2 ft	62.7 ft	90 ft	90 ft
	17.0 m	19.1 m	10.0 m	14.4 m	19.1 m	27.4 m	27.4 m
Diameter	17.7 ft	16.7 ft	27.6 ft	27.6 ft	27.6 ft	27.6 ft	32.8 ft
	5.4 m	5.1 m	8.4 m	8.4 m	8.4 m	8.4 m	10.0 m
Internal Diameter	15.1 ft	15.1 ft	24.6 ft	24.6 ft	24.6 ft	24.6 ft	29.9 ft
	4.6 m	4.6 m	7.5 m	7.5 m	7.5 m	7.5 m	9.1 m
Available Volume	7,740 ft³	9,030 ft³	10,100 ft³	11,260 ft³	18,970 ft³	31,950 ft³	46,610 ft³
	219.2 m³	255.7 m³	286.0 m³	14.4 m³	19.0 m³	27.4 m³	27.4 m³
Potential Availability (No Earlier Than)	COTS	COTS	2021	2022	2022	2024	2028

COTS: Commercial Off-the-Shelf CPL: Co-manifested Payload PPL: Primary Payload PLF: Payload Fairing

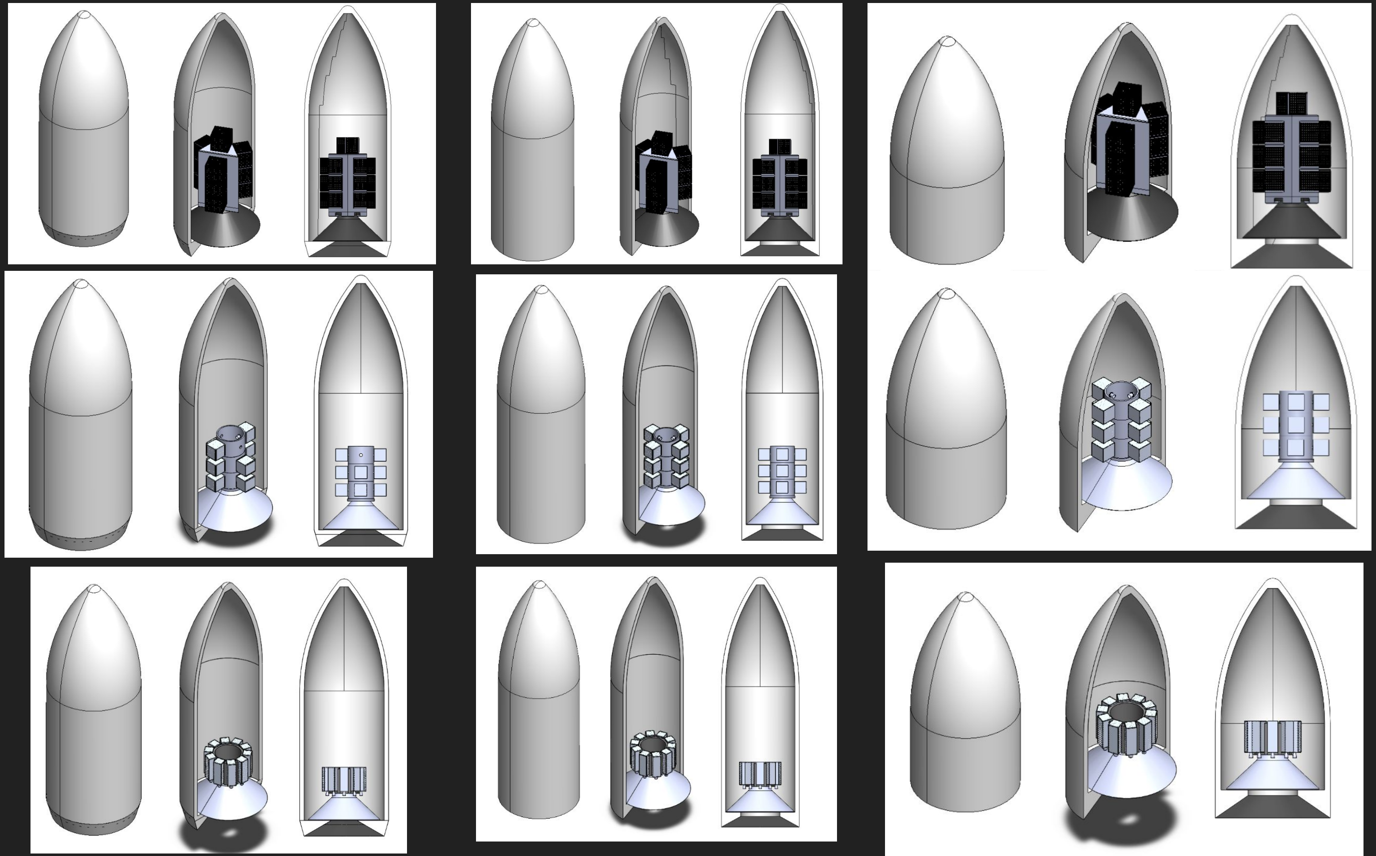
Payload Fairing Iterations

10 m | 8.4 m long | 8.4 m short

1st Iteration

2nd Iteration

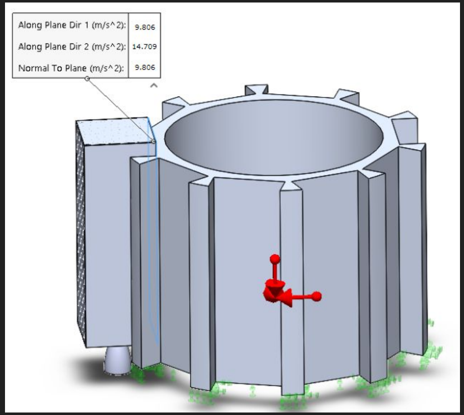
3rd Iteration

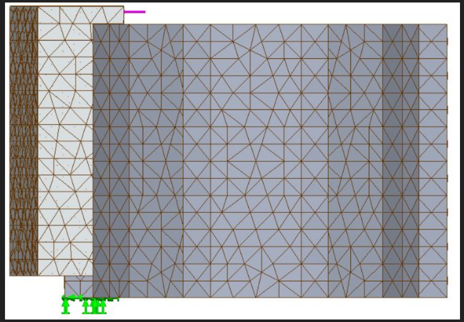


Quasi-static Loads Analysis

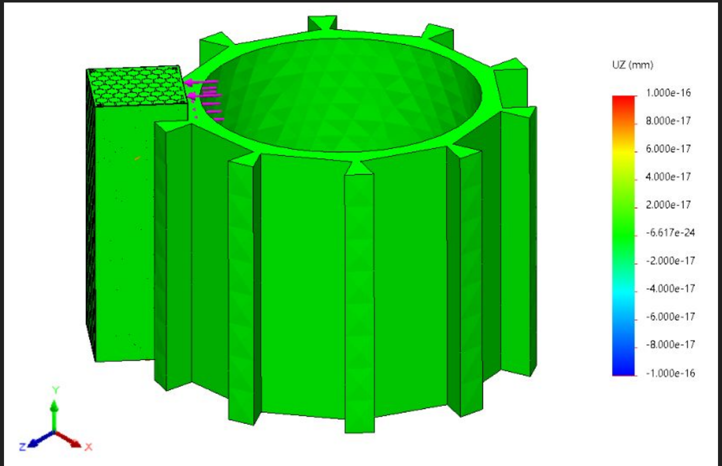
Block 1B PPL/CPL Combined Load Factors

CASE	Vehicle Axial (G)	Vehicle Axial (G)
Liftoff	1.0	+- 1.5
Liftoff / Transonic	-2.0	+-2.0
Max Q	-2.3	+-2.0
Core Stage Flight	-3.5	+-0.5
MECO	-4.1	+-0.5





Displacement z-axis



The spacecraft shifts between -6.617e^{-24} mm to -2 e^{-17} mm. Although, the simulation may not be correct since the load values were too heavy to be computed, and they had to be simplified, and an additional damping parameter needed to be added.

Conclusions / Assessment

Structural analysis engineers make sure your **spacecraft isn't going to fall apart**

Structural analysts do a lot of their work with the help of **finite element models**

Structural analysis: static and dynamic.

- **Static** means the environment around your spacecraft is not changing or it's slowly changing.

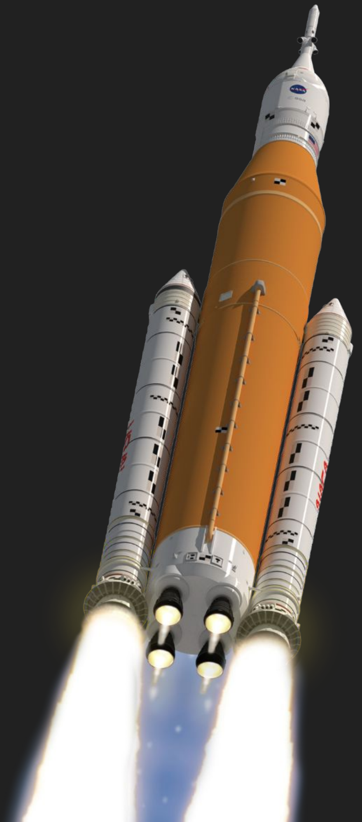
- **Dynamic** means the structure has quickly changing loads like vibration or loud noises.

Knowing which **materials** can take the loads, which ones won't melt in the heat of the sun, and which ones need special treatment.

A satellite worth hundreds of millions of dollars that took years to build is often **successful or fails completely because of how one mechanism does.**

Further iterations on the design need to be implemented.

Individual Responsibilities



Individual	Responsibilities for Structures and Mechanisms
	Literature Review
	CAD
	Component Selection / Trade off
	Structure Shape Selection / Trade off
	Material Selection / Trade off
Aelyn Chong C	COMET
	Reports
	Presentations
	Quasi-static Load Analysis
	Finite Element Analysis

Attitude Determination and Control

ADC

Final Presentation - 16.06.22

Lina Maria Amaya Mejia
Kevin Laurent Biewesch

Structure to follow

Introduction to your subsystem (including the team)

Requirements and Design Drivers

Design (including inputs/outputs. SE will show the entire N2 chart but for details you are free to talk about them)

Components

COMET model + Iterations

Trade-offs (on your subsystem level if any)

Conclusions / Assessment

Contribution Slide (so that after the presentation they can see who worked on which slide)

Ex: Alex → Slides 1,2, 5 - 8

\+ Anything else you might want to showcase

Outline

- Introduction to ADC
- Individual responsibilities
- Requirements
- Inputs and Outputs
- Operation Modes
- Disturbance Environment
- COMET model
- Iterations
- Trade-offs
- Final List of Equipment
- Conclusions

Introduction

ADCS:
Provides pointing accuracy and stability of the payloads and antennas as critical parts of the S/C operation and the mission success.

Attitude determination: combines available sensor inputs with knowledge of the spacecraft dynamics to provide an accurate and unique solution for the attitude state as a function of time.

Attitude control: is the combination of the prediction and reaction to a vehicle's rotational dynamics and environmental disturbances.

Individual Responsibilities

ADCS Responsibilities	Individual
Definition of requirements	All
Identification for ADC active phases	All
Definition of control modes	Lina
Quantification of disturbance environment	Lina
Select and size sensors and actuators	Lina
Select and size thrusters	Kevin
COMET Model (+ Excel sheets connection)	Kevin
Components trade-off	All
Report	All
Presentation	All

Requirements & Design Drivers

ID	Requirement	Parent ID
ADCR 01	The ADCS shall handle actions of fine pointing, spacecraft reorientation and maneuvering.	–
ADCR 02	The ADCS shall deal with distinct operation modes for the different mission phases	–
ADCR 03	The ADCS shall maintain the spacecraft's attitude despite the environment disturbances	–
ADCR 04	The ADCS shall be able to provide a maximum slew rate in PNT mode of less than 0.1 °/s	SREQ 8
ADCR 05	The ADCS shall provide 3-axis control during thrusting and while not thrusting	–
ADCR 06	The mass of the ADCS shall not exceed 50 kg	SREQ 2; SREQ 3
ADCR 07	The power used by the ADCS shall not exceed 50 W	SREQ 7

Inputs and Outputs

Subsystems	Inputs	Outputs
System Engineering	Power budget, mass budget	Hardware selection
Payload	Pointing accuracy, slew rate	Hardware selection (actuators)
Trajectory and Orbit	ConOps, Phases, Orbit characteristics	Operation modes definition, requirements by control mode, external disturbances torques
Structures	S/C mass, longest side, projected area, moment of inertia	Internal disturbances torques
Propulsion	/	/
Power	/	/
Thermal	/	/
Com & CDH	/	/

Design: Operation Modes

Mode	Purpose	System-level requirements	Equipment level requirements
Detumble mode	Attitude maintenance and momentum management when required.	3-axis control for attitude maintenance	Star trackers, IMU Thrusters
Sun safe mode	Initial attitude acquisition, and provide a sun-pointing, power and thermally safe attitude in the event of an anomaly.	The system momentum must be within the momentum capability of the RW	Sun sensors, IMU Reaction wheels
PNT mode	Provide nadir and inertial fine pointing for data measurements and for pointing the antenna to the ground station.	3-axis spacecraft attitude slewing Slew rate of 0.1 deg/sec	Star trackers, IMU Reaction wheels
Maneuvering mode	Maintain attitude pointing during cruise trajectory, insertion burns and lunar orbit maintenance.	3-axis control for attitude maintenance	Star trackers, IMU Thrusters

Design: Disturbance Environment

Since the mission takes place in the lunar orbit, there are only two significant external disturbances torques:

- **Gravity gradient torque:** caused when the spacecraft's center of gravity is not aligned with its center of mass.

$$T_g = \overbrace{\frac{3\mu}{2R^3}}^{\text{Environment}} \overbrace{|I_z - I_y| \sin(2\theta)}^{\text{Spacecraft}}$$

- **Solar radiation pressure torque (SRP):** Sunlight has momentum and it exerts pressure on those objects it strikes.

$$T_s = \overbrace{\frac{\phi}{c}}^{\text{Environment}} \overbrace{A_s(1+q)\cos(\psi)}^{\text{Spacecraft}}$$

COMET: Inputs

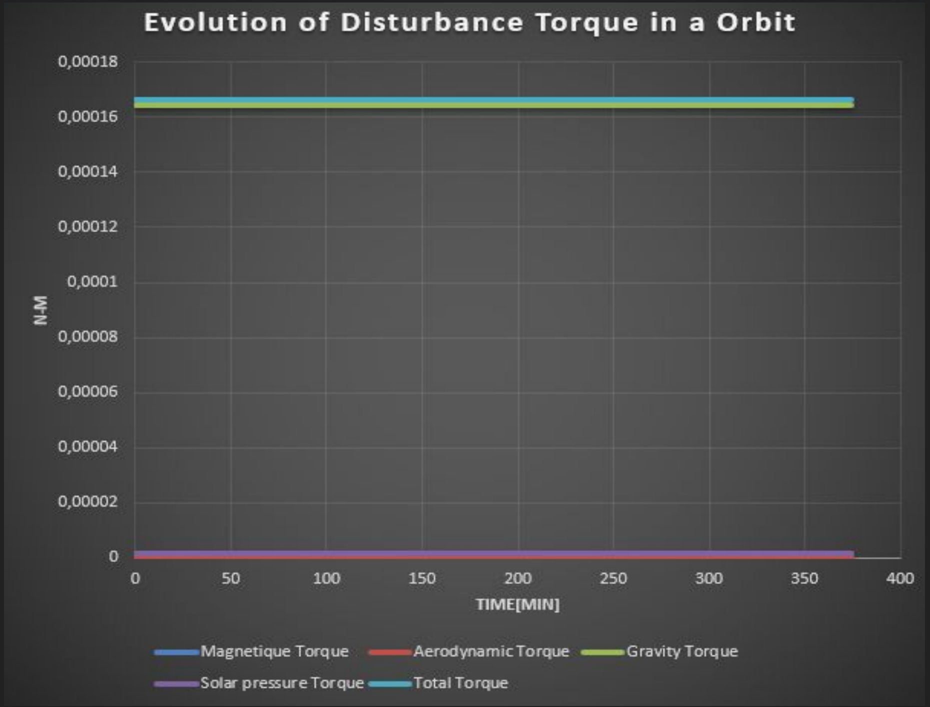
Orbit characteristics			
Inputs from OCDT			
Orbit Altitude (circular orbit)	[alt]	**2300**	**km**
Perigee Altitude (elliptical orbit)	[alt_peri]		**km**
Orbit Semi-Major Axis	[]	**4000**	**km**
Orbit eccentricity	[]	**0**	-
Orbit inclination	[inc]	**45**	
Orbit RAAN	[]	**0**	
Orbit Argument of Periapsis/Argument of Lati	[]		

Spacecraft characteristics			
Inputs from OCDT			
Expected Spacecraft Moment of Inertia	[MoI]	**1232,564883**	**kg*m2**
Expected Spacecraft Linear Dimension	[len]	**1,2**	**m**
Expected Spacecraft Volume	[V]	**1,728**	**m3**
Estimated Spacecraft Loaded Mass	[m_wet]	**1133,667964**	**kg**
Expected Spacecraft Projected Area	[area_proj]	**1,44**	**m2**
Drag Coefficient	[drag_coeff]	**0**	-
Spacecraft Surface Material	Aluminum606T6		

Slew characteristics			
Calculations			
Maximum slewing angle	*5,000*	**5,00**	**deg**
Minimum maneuver time	*60,000*	**60,00**	**s**
Slew rate (calculated)			**s**
Slew rate		0,083333333	deg/s

COMET: Outputs

Torques		
Central Body	Moon	
Calculations		
Environmental torques		
Gravity gradient	1,118E-04	N-m
Solar radiation	1,835E-06	N-m
Magnetic	0,000E+00	N-m
Aerodynamic	0,000E+00	N-m
Total (RSS)	1,118E-04	N-m
Slewing torque	1,195E-01	N-m
Evaluate Torque in a Orbit		
Torques and momentum in a orbit (Integration)		
Integrated angular momentum	6,22E-02	N-m-s
Momentum cyclical component	4,97E-02	N-m-s
Momentum secular component	1,24E-02	N-m-s
Max torque in a orbit	1,66E-04	N-m
Control algorithm approximation		
Required pointing accuracy	1,00E-01	deg



COMET: Propellant calculation

Parameters				
g		1,633333333	m/s^2	
I_sp		221	s	[SMAD]
L	Moment thruster arm (perpendicu	0,6	m	[SMAD]
t_tot	Time thruster is used	#¡REF!	s	[SMAD]
t_bt	Burn time for thruster	1	s	[SMAD]
dV	Delta-V	109,3142039	m/s	
mar	margin	20	%	
h_m	Moment for maneuvre	6,22E-02	Nms	
h_s	Moment for slew rate	#¡VALOR!	Nms	
h_max	Maximum momentum (not PNT)	40	Nms	[SMAD, p.374 PDF]
Thruster Propellant				
m	Propellant mass	#¡VALOR!	g	
	(per 180 deg slew)	#¡VALOR!	mg	
		#¡VALOR!	kg	
m	Propellant mass	#¡VALOR!	g	
	(whole mission)	#¡VALOR!	mg	
		#¡VALOR!	kg	
m	Propellant mass (delta-V)	1374,722172	g	
	(whole mission)	13747,22172	mg	
		13,74722172	kg	
Thruster Force				
F_max	for Maximum (not PNT)	66,66666667	N	
F_mom	for Storerd momentum (PNT)	0,103631115	N	
Fuel Oxidizer Mass / Volume				
Fuel	Fuel mass	13,74722172	kg	
	Fuel tank volume	16,36574014	dm^3	
Oxidizer	Oxidizer mass	0	kg	
	Oxidizer tank volume	0	dm^3	

$$m_{\text{propellant}} = \frac{m_{\text{dry}} + m_{\text{wet}}}{2}\left(e^{\frac{\Delta v}{I_{\text{sp}}\, g}} - 1\right)$$

$$V_{\text{propellant}} = \frac{m_{\text{propellant}}}{d_{\text{propellant}}(1-u)}$$

COMET: Model

Attitude Determination...

mass	27.254 [kg]
mass margin	10 [%]
peak consumed po...	83.14 [W]
power	83.14 [W]
power margin	15 [%]
Attitude Control Th...	
Attitude Control Th...	
Attitude Control Th...	
Attitude Control Th...	
Attitude Control Th...	
Attitude Control Th...	
Attitude Control Th...	
Attitude Control Th...	
Fuel ADC Tank : Fu...	
Inertial Measureme...	
Inertial Measureme...	
Propellant ADC Thr...	
Reaction Wheel : R...	
Reaction Wheel : R...	
Reaction Wheel : R...	
Reaction Wheel : R...	
Star Tracker : Star T...	
Star Tracker : Star T...	
Sun Sensor : Sun S...	
Sun Sensor : Sun S...	
Sun Sensor : Sun S...	
Sun Sensor : Sun S...	
Sun Sensor : Sun S...	
Sun Sensor : Sun S...	
Sun Sensor : Sun S...	
Sun Sensor : Sun S...	

Inertial Measurement...

height	0.0925 [m]
length	0.0888 [m]
mass	0.84 [kg]
maximum operatio...	71 [°C]
minimum operatio...	-40 [°C]
power	15 [W]
width	0.0888 [m]

Star Tracker : Star Tracker

height	0.093 [m]
length	0.06 [m]
mass	0.197 [kg]
maximum operational...	40 [°C]
minimum operational t...	-30 [°C]
power	1.35 [W]
width	0.056 [m]

Sun Sensor : Sun Sensor

height	0.027 [m]
length	0.08 [m]
mass	0.1 [kg]
maximum operatio...	85 [°C]
minimum operatio...	-40 [°C]
power	1.08 [W]
width	0.08 [m]

Reaction Wheel : Reaction...

height	0.235 [m]
length	0.235 [m]
mass	5 [kg]
maximum operational...	60 [°C]
minimum operational t...	-20 [°C]
momentum capacity	7.2 [N·m·s]
power	3 [W]
torque	0.19 [N·m]
width	0.073 [m]

Attitude Control Thrust...

mass	0.16 [kg]
specific impulse	221 [s]
total attitude contr...	500 [N·s]

Propellant ADC Thrusters :...

mass	13.7472217178754...
Fuel ADC Thrusters : Fu...	
Fuel Type	N2H2
mass	13.7472217178754...
volume	0.01636574014032...

Fuel ADC Tank : Fuel A...

dry mass	3.1 [kg]
height	0.1326 [m]
length	0.0528 [m]
maximum operatio...	50 [°C]
minimum operatio...	4 [°C]
volume	17.5 [m^3]
width	0.0528 [m]

First Iteration - Orbit 2,300 km

Inputs

Orbit

- Semi-major axis: 4,000 km
- Sun incidence angle: 45 deg
- Maximum deviation from local vertical: 45 deg

Spacecraft

- Wet mass: 1,400 kg
- Largest moment of inertia: 1752.051 kg*m^2
- Smallest moment of inertia: 320.856 kg*m^2
- Projected surface area: 1.440 m^2

Outputs

Torques

- Gravity gradient: 1.644E-04 Nm
- Solar radiation: 1.835E-06 Nm
- Slewing torque: 0.1699 Nm

Momentum

- Integrated angular momentum: 0.0622 Nms
- Required pointing accuracy: 0.1 deg

Propellant

- Propellant mass: 12.0491 kg
- Tank volume: 14.3442 L

Second Iteration - Orbit 5,000 km

Inputs

Orbit

- Semi-major axis: 6,700 km
- Sun incidence angle: 45 deg
- Maximum deviation from local vertical: 45 deg

Spacecraft

- Wet mass: 1,479 kg
- Largest moment of inertia: 1919.907 kg*m^2
- Smallest moment of inertia: 338.962 kg*m^2
- Projected surface area: 1.440 m^2

Outputs

Torques

- Gravity gradient: 3.865E-05 Nm
- Solar radiation: 1.835E-06 Nm
- Slewing torque: 0.1862 Nm

Momentum

- Integrated angular momentum: 0.0299 Nms
- Required pointing accuracy: 0.1 deg

Propellant

- Propellant mass: 12.1702 kg
- Tank volume: 14.4884 L

Third Iteration - Orbit 2,300 km + Hybrid Propulsion

Inputs

Orbit

- Semi-major axis: 4,000 km
- Sun incidence angle: 45 deg
- Maximum deviation from local vertical: 45 deg

Spacecraft

- Wet mass: 1,133.668 kg
- Largest moment of inertia: 1232.565 kg*m^2
- Smallest moment of inertia: 259.818 kg*m^2
- Projected surface area: 1.440 m^2

Outputs

Torques

- Gravity gradient: 1.118E-04 Nm
- Solar radiation: 1.835E-06 Nm
- Slewing torque: 0.1195 Nm

Momentum

- Integrated angular momentum: 0.0622 Nms
- Required pointing accuracy: 0.1 deg

Propellant

- Propellant mass: 13.7472 kg
- Tank volume: 16.365 L

Trade-Offs: Reaction Wheel

Name	Mass [kg]	Power [W]	Torque [Nm]	Ang. Mom. [Nms]
RW-10NM	5	3	0.19	7.2
SX-RW-50-2.0 (Sputnix)	2.5	40	0.05	2
Reaction Wheel Unit W18 (Bradford Space)	5.65	168	0.265	22

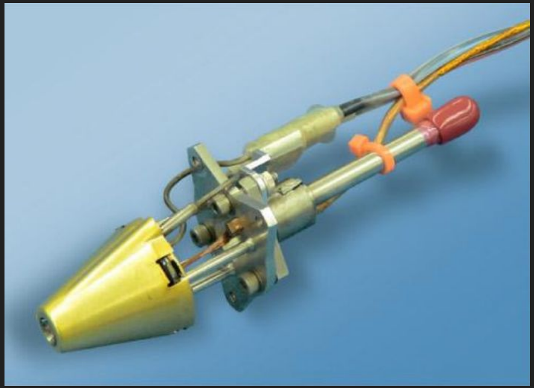
Amount	Justification
3+1	- Slewing torque (0.12 Nm) never exceeds RW torque - Angular integrated momentum (0.0622 Nms) never exceeds RW angular momentum - Low power consumption - Fine pointing manoeuvers (Component Justification) - Counter disturbance torques (Component Justification)

https://satsearch.co/products/spaceteq-reaction-wheel-rw-10nms

Trade-Offs: Thruster

Name	Mass [kg]	Power [W]	ISP [s]	Thrust [N]	Fuel	Oxidizer
Bipropellant Thruster (ArianeGroup)	0.659	??	292	10	MMH	N2O4, MON-1, MON-3
HPGP Thruster HPGP (Bradford ECAPS)	0.48	25	268.18	5	HPGP	–
MR-103M (Aerojet)	0.16	7.1	221	1	N2H2	–

Amount	Justification
8	- Difficult to estimate momentum to be generated - Thrust *indicates* time a manoeuvre should take - Thrust needed: $F = h / (L * t)$ - Time needed: $t = h / (F * L)$ - Momentum dumping (Component Justification) - Larger manoeuvers (Component Justification)



https://satsearch.co/products/aerojet-rocketdyne-holdings-inc-mr-107s

Trade-Offs: Tank

Name	Mass [kg]	Volume [dm^3]	Min Temp [°C]	Max Temp [°C]
PEPT-330	3.1	17.5	4	50
PEPT-420N	3.9	37.5	4	55
PEPT-260	1.7	9.3	10	50
PEPT-230	1.3	6	10	50

Amount	Justification
1	- Required fuel volume (16.4 L) needs to fit in the tank - Required fuel mass (13.7 kg) needs to fit in the tank

https://satsearch.co/products/rafael-pept-330

Trade-Offs: IMU

Name	Mass [kg]	Power [W]	Scale fact. [ppm]	Gyro range [°/s]
IMU 42, 42 CV 42XP	0.84	15	150	±2000
GS-IMU3000TA Fiber Optic Gyroscopes Inertial Measurement Unit (GranStal Solutions Ltd.)	1.9	12	100	±400
ASTRIX 1000 (Airbus Defence and Space)	4.5	13.5	300	±140

Amount	Justification
1+1	- High gyro range - Low mass - Angular Velocity (Component Justification) - Complementary to Star Tracker (Comp. Just.)

https://www.cielo-inertial.com/products/imu/imu42-imu42xp/

Trade-Offs: Star Tracker

Name	Mass [kg]	Power [W]	FOV [°]	Accuracy [°]	Slew rate [°/s]
ST Series Star Tracker (O.C.E. Technology)	0.13	1	12	±0.00083	2.5
ST (KAIROSPACE Co., Ltd.)	0.197	1.35	20	±0.00139	5
Star Tracker ST-1 (NanoAvionics)	0.108	1.2	21	±0.00278	1.5

Amount	Justification
1+1	- Wider FoV - Higher slew rate - High accuracy (Component Justification)

https://satsearch.co/products/kairospace-star-tracker

Trade-Offs: Sun Sensor

Name	Mass [kg]	Power [W]	FOV [°]	Accuracy [°]
Coarse Sun Sensor (Bradford Space)	0.215	0	180	±1.5
ISS-AX Solar (MEMS Technologies)	0.1	1.08	120	±1.5
COARSE SUN SENSOR PYRAMID (RedWire)	0.13	0	180	±5

Amount	Justification
8 + 2	- Low mass - High accuracy - Power and thermally safe sensor (Comp. Justif.)

https://satsearch.co/products/solarmems-iss-sun-sensor

Conclusion

- Torques did not change much for different iterations
  - Reaction wheels did not need to be changed
- No metric to assess sensors
- Mistake in propellant mass calculations!
  - 359.2 kg instead of 13.7 kg

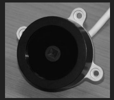
Apart from ADC thruster unit, our solution provides a feasible package

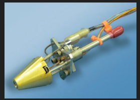
Contribution Slides

- Introduction [Lina]
- Individual responsibilities [Lina]
- Requirements [Kevin]
- Inputs and outputs [Lina]
- Operation modes [Lina]
- Disturbance environment [Lina]
- COMET [Kevin, Lina]
- Iterations [Lina]
- Trade-offs [Kevin, Lina]
- Conclusion [Kevin]

Components (Summary)

Name	Amount	Specification
		Sensors
Star trackers	2	• Provider: KAIRO SPACE • Pointing accuracy: < 5 arcsec • Field of View: 20 degrees • Acquisition time: <2 sec • Angular velocity: 5 °/s
Sun sensors	8	• Provider: SOLARMEMS • Field of View: 120° • Accuracy: < 10 %
IMU	2	• Provider: Cielo • Gyro scalable bias performance: 0.5 °/s to 0.1 °/s • Gyro scale factor: <150 ppm • Gyro dynamic range: ±2000 °/sec
		Actuators
Reaction wheels	4	• Provider: SpaceTeq • Max torque: 190 mNm • Angular momentum: 7.2 Nms • Speed range: -4300 to +4300 rpm • Speed control accuracy: 0.12 rpm
Thrusters	8	• Provider: Aerojet • Propellant: Hydrazine • Thrust: 1.02 - 0.22 N • Specific impulse: 224 - 209 sec
		Additional
Tank	1	• Provider: Rafael Space • Volume: 17.5 L • Maximum propellant mass: 15.4 kg


Propulsion System

PRP

Final Presentation - 16.06.22

Khushaldas BADHAN

Structure to follow

Introduction to your subsystem (including the team)

Requirements and Design Drivers

Design (including inputs/outputs. SE will show the entire N2 chart but for details you are free to talk about them)

Components

COMET model + Iterations

Trade-offs (on your subsystem level if any)

Conclusions / Assessment

Contribution Slide (so that after the presentation they can see who worked on which slide)

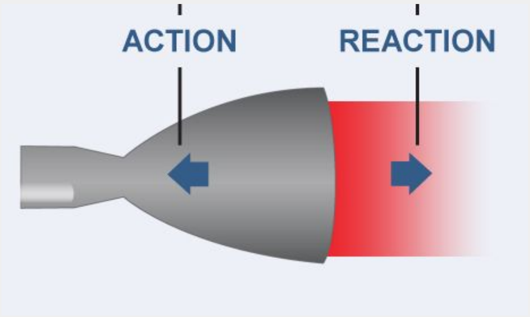
Ex: Alex → Slides 1,2, 5 - 8

\+ Anything else you might want to showcase

Propulsion Subsystem

Main Engine

Produces required forward thrust for various manoeuvres.

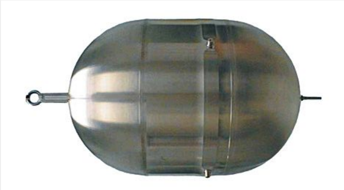


Propellant

Chemical to produce thrust
single chemical or two separate chemicals

Propellant Tanks

Tanks to store Propellant



Requirements

- # The propulsion system shall put the spacecraft in the required orbit.
- # The propulsion system shall provide thrust for station keeping of the spacecraft.
- # The propulsion system shall re-orbit/deorbit the spacecraft after the end of operational period.
- ~~# The propulsion system shall have components TRL no less than 6.~~
- # The propulsion system shall monitor anomalies and executes pre-defined counter measures during the lifetime.
- # The propulsion system shall comply with the mass budget and power
- # The thrust vector of the engine should pass through the center of mass of the spacecraft.

N2 Chart

2		Total delta-V budget Delta-V margin Operational Lunar Altitude Operational Orbit eccentricity ~~Launch date~~	Launch Vehicle Spacecraft dry mass Mission duration	Spacecraft Lifetime Satellite dry mass Mass budget Power budget Propellant mass allocation	Spacecraft Lifetime Power Budget Spacecraft Dry Mass	Spacecraft mass Spacecraft body area	Satellite wet mass Power budget ~~Drag coefficient~~	Ground station requirements Duty cycle Communications requirements ~~Minimum Elevation angle~~ ~~Maximum nadir angle~~
3	**Payload**		Payload volume and mass		Avg Power consumption		Pointing Accuracy Slew rate	~~Payload Datarate~~ ~~Instrument data interface~~
4		**Trajectory & Orbit**	Acceleration at each Delta-V manoeuver Number of satellites in constellation	Spacecraft Delta-V PNT Operations (Station keeping delta-V)	Average Eclipse time per orbit Maximum Eclipse time Avg. Sunlight time per orbit Operational Orbit altitude	Operational Orbit altitude Average Eclipse time per orbit Maximum eclipse time Avg. Sunlight time per orbit	ConOps / Phases (Maneuvers / Trajectory) Orbit characteristics - perigee - altitude	Intersatellite Distance Orbit Period Minimum Elevation Range Slant Range Maximum Nadir Angle
5	Structure dimensions		**Structure**	Dry Mass Maximum permissible acceleration	Spacecraft size Spacecraft shape Solar panels configuration	Structure material	Spacecraft size Spacecraft shape Spacecraft volume Center of gravity	
6			Size and location of engine, of thrusters Volume and mass of fuel tank Mass of propellant Propulsion system mass Maximum acceleration	**Propulsion**	Total Power required per main engine Number of engines ~~Total Power required per attitude thrusters~~ ~~Number of attitude thrusters~~	Choice of propellant Temperature range of propellant		
7			Areas of solar panels Dimension of batteries		**Power**	Power generation (eclipse) Power generation (sunlight) Area of solar array		
8			Dimensions of thermal system Mass of thermal system Location of thermal system		TCS Power Consumption	**Thermal**		
9			Mass, dimension, and location of thrusters, star tracker, electronics	Amount of fuel Total time * average torque (note: +10% of fuel for ADC)	ADC Power required (reaction wheel) ADC Power required (sensors) ADC Power required (thrusters)		**ADC**	
10			Mass, dimension, and location of electronics, antennas Are antennas deployable and driven?		COM Power required, depending on data range and bandwith			**Com & CDH**

N2 Inputs and Outputs

Inputs

Spacecraft Lifetime

Satellite Dry Mass

Mass Budget

Power Budget

Propellant Mass Budget

Spacecraft Delta V

PNT operations and Station Keeping Delta V

Maximum Permissible Acceleration

Output

Spacecraft Wet Mass

Propellant Mass

Size and location of engine, of thrusters

Volume and mass of fuel tank

Mass of propellant

Propulsion system mass

Maximum acceleration

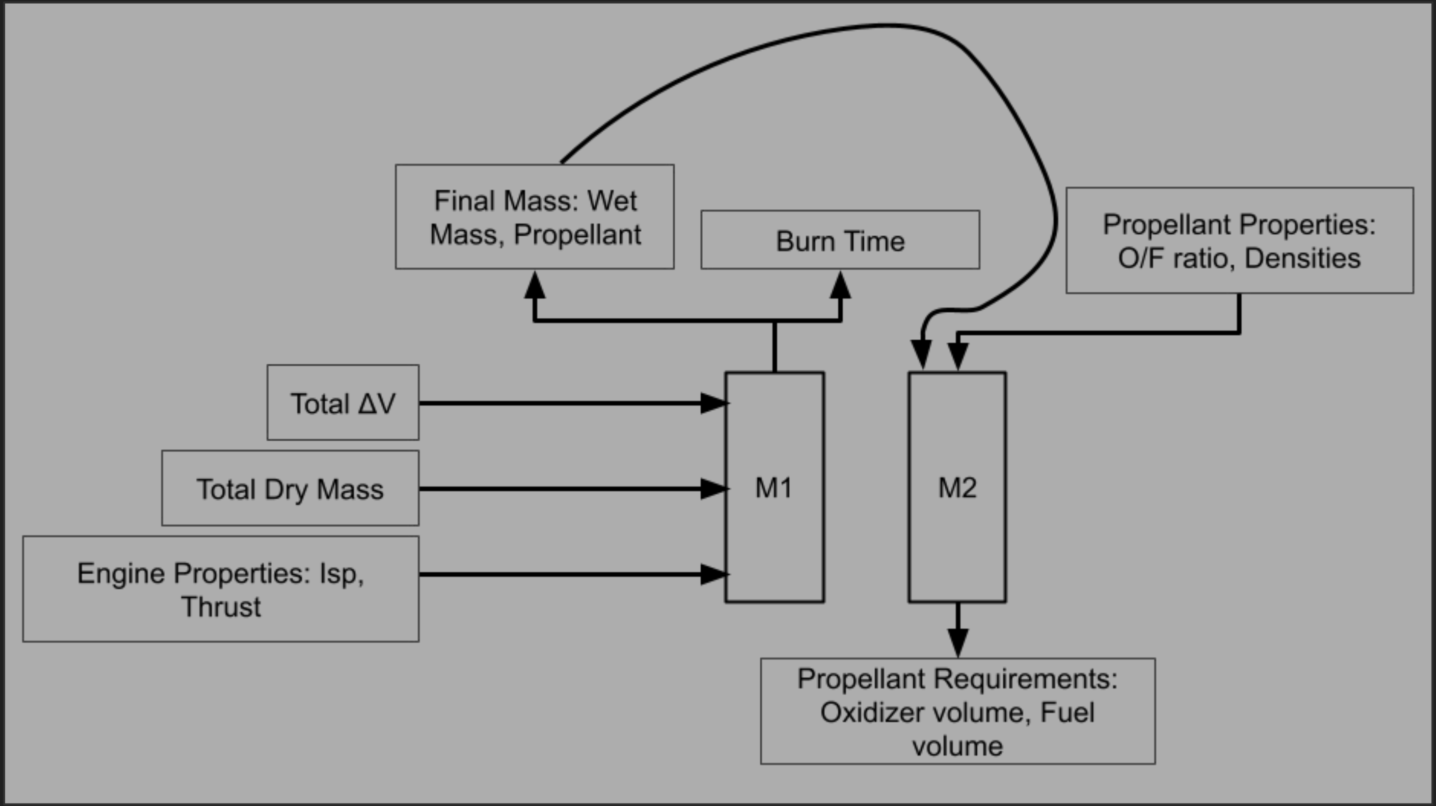
Total Power required per main engine

Number of engines

Choice of propellant

Temperature range of propellant

Design Process and Comet Model



Design Process and Comet Model

For calculating initial mass of the spacecraft:

$$m_{initial} = m_{final} \times e^{\frac{\Delta V}{Isp \times g_0}}$$

For burn time:

$$t_{burn} = \frac{m_{initial} \times \Delta V}{Thrust}$$

For propellant mass:

$$m_{propellant} = m_{initial} - m_{final}$$

Propellant mass with margin:

$$m_{\text{safe propellant}} = m_{propellant} \times 1.1$$

Safe Wet Mass :

$$m_{\text{wet safe}} = m_{\text{safe propellant}} + m_{dry}$$

Design Process and Comet Model

Fuel Mass:

$$m_{fuel} = \frac{m_{propellant}}{1 + ratio_{Ox:Fuel}}$$

Fuel Volume:

$$V_{fuel} = m_{fuel} \times \rho_{fuel}$$

Oxidizer Mass:

$$m_{Oxidizer} = m_{propellant} \times \frac{ratio_{Ox:Fuel}}{ratio_{Ox:Fuel} + 1}$$

Oxidizer volume:

$$V_{Oxidizer} = m_{Oxidizer} \times \rho_{Oxidizer}$$

Iterations

561 kgs

361 kgs

245 kgs

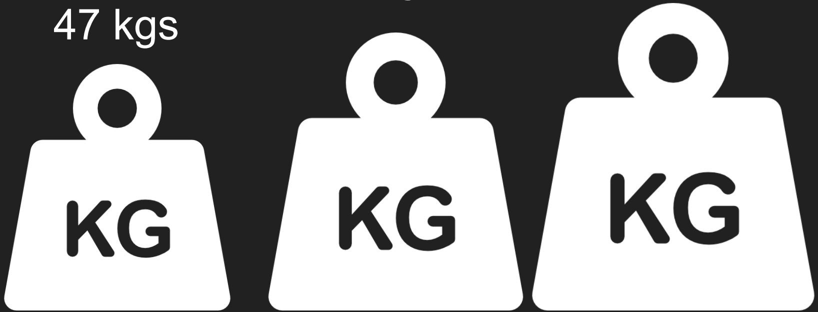
1 2 3

Total Propellant Mass

70 kgs

64 kgs

47 kgs



1 2 3

Total Propulsion Subsystem Dry Mass

Iterations 1 & 2

Rocketdyne AMBR 556 Bi-propellant

Ariane Space Model T11-0

Ariane Space Model OST 21/0

1

Antrix Corporation 390 L

Ariane Space Model OST 25/0

2

Iteration 1

Delta_V_Spacecraft: 986.28 m/s
mass_dry: 917.81 kgs
Engine Selected: Rocketdyne AMBR 556 Bi-propellant
Fuel: Monomethylhydrazine (MMH)
Oxidizer: Dinitrogen tetroxide (NTO)
Oxidizer to fuel ratio: 1.3
Thrust: 556 N
Isp: 329 s
Engine Mass: 4.9 kgs
Valve Peak Power: 70 W
Quantity of engines: 1
Propellant Mass with margin = 361 kgs
Total Burn Time = 2210 s
Fuel Mass = 136.22 kgs
Fuel Volume = 0.194 m^3

Oxidizer Mass = 224.77 kgs
Oxidizer Volume = 0.222 m^3
Fuel Tank Selected: Ariane Space Model T11-0
Fuel Tank Volume: 235 l
Oxidizer Tank Selected: Ariane Space Model OST 21/0
Oxidizer Tank Volume: 218 l
Total Propulsion System Mass = 46.9 kgs

Iteration 2

Delta_V_Spacecraft: 2964 m/s
Engine Selected: Rocketdyne AMBR 556
Bi-propellant
Fuel: Monomethylhydrazine (MMH)
Oxidizer: Mixed oxides of nitrogen (MON-3)
Oxidizer to fuel ratio: 2.27
Thrust: 556 N
Isp: 329 s
Engine Mass: 4.9 kgs
Valve Peak Power: 70 W
Quantity of engines: 1
Propellant Mass with margin = 561 kgs
Total Burn Time = 2210 s

Fuel Mass = 171.6 kgs
Fuel Volume = 0.245 m^3
Oxidizer Mass = 389.44 kgs
Oxidizer Volume = 0.386 m^3
Fuel Tank Selected: Ariane Space OST 25/0
Fuel Tank Volume: 282 l
Oxidizer Tank Selected: Antrix Corporation 390 L
Oxidizer Tank Volume: 390 l
Total Propulsion System Mass = 46.9 kgs

Iterations 3

Rocketdyne AMBR 556 Bi-propellant

Ariane RIT 10 EVO

Ariane Space Model DT180

MT Aerospace AG XS-XTA/7

Iteration 3

Delta_V_for Electric Propulsion of the Spacecraft: 300 m/s
Delta_V_ for Chemical Propulsion of the Spacecraft: 686.28 m/s
mass_dry: 861.45 kgs
Chemical Engine Selected: Rocketdyne AMBR 556 Bi-propellant
Chemical Engine Fuel: Monomethylhydrazine (MMH)
Chemical Engine Oxidizer: Dinitrogen tetroxide (NTO)
Chemical Engine Oxidizer to fuel ratio: 1.3
Chemical Engine Thrust: 556 N
Chemical Engine Isp: 329 s
Chemical Engine Engine Mass: 4.9 kgs

Chemical Engine Valve Peak Power: 70 W
Quantity of chemical engines: 1
Electrical Engine Selected: Ariane RIT 10 EVO
Electrical Engine Propellant: Xenon gas (Xe)
Electrical Engine Thrust: 0.025 N
Electrical Engine Isp: 3400 s
Electrical Engine Engine Mass: 1.8 kgs
Electrical Engine Peak Power: 760 W
Quantity of electric engines: 1
Chemical Propellant Mass with margin = 236.26 kgs
Total Burn Time = 1909 s
Fuel Mass = 102.75 kgs
Fuel Volume = 0.146 m^3
Oxidizer Mass = 133.54 kgs

Iteration 3

Oxidizer Volume = 0.132 m3

Fuel Tank Selected: Ariane Space Model DT180

Fuel Tank Volume: 180 l

Oxidizer Tank Selected: Ariane Space Model DT180

Oxidizer Tank Volume: 180 l

Electric Propellant Mass = 8.56 kgs

Electric Propellant Volume = 5.71 m^3

Propellant Tank Selected: MT Aerospace AG

XS-XTA/7

Propellant Tank Volume: 7 l

Total Propulsion System Mass = 70.4 kgs

Conclusion

- Propellant required for 5000 kms altitude > Propellant required for 2300 kms altitude
- Hybrid propulsion can reduce mass of the subsystem but makes the system complex
- To implement hybrid propulsion configuration approximately 5 hours of downtime for each satellite is required each month along with a slow implementation of end of life manoeuvre which in exchange can provide weight savings of 80 to 120 kgs per satellite by reducing the chemical propellant required for orbital correction.

Contributions

Individual	Responsibilities for Structures and Mechanisms
Khushaldas Badhan	Literature Review
	Design Process
	Numerical Model
	Component Selection / Trade off
	COMET
	Reports
	Presentations

Power SubSystem

Final Presentation - 16.06.22

Laura Chikosman
Carla Muttoni

Tabla de contenido (7 minutos)

Contenido	Descripción	Responsable	Duración
1.Introduction al subsistema de power	Describir lo que es el subsistema de power : - El procesos de power - Los componentes - Y cómo interactúa con el reto de los sistemas	Carla	2 minutos
2. Design Process	- Requerimientos - Methodology -	Carla	2 minutos
3. Proceso	- Energy storage - Formulas	Laura	
3. Model	- Parametros	Carla	2 minutos
4. Entregar resultados (2 opciones)	- Trade off - Solar arrays - Baterias - Regulator	Laura	2 minutos

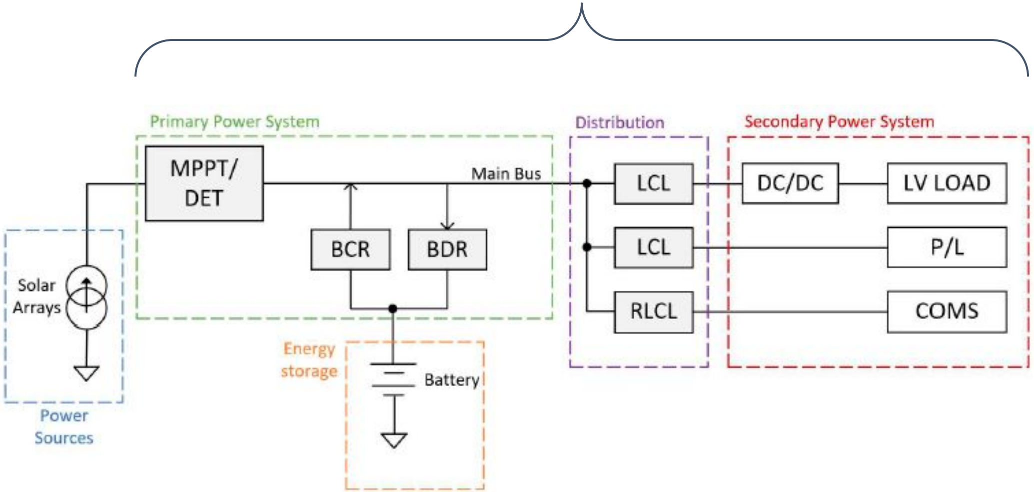
Introduction

Individual	Responsibilities
Laura, Carla	Definition of requirements
Carla	Determine solar panel array requirements, calculate amount of power, select types of cells.
Laura	Determine batteries requirements, select type of batteries and calculate size of batteries.
Laura	Define distribution and regulator subsystem
	Report
	Trade off
	Comet

Introduction to Power Subsystem

The power subsystem must:

- Produce, store and distribute power to the rest of the spacecraft during the different mission modes.
- Be autonomous of the rest of the satellite
- Be robust (free of “single points of failure”)
- Protect the loads from over current events
- Manage the energy available



Ref: An Introduction to Satellite Power Systems – Quentin Mannes

GOAL

Main goal:
Design a power subsystem that produces, stores and distributes power to the loads in all phases of the mission (sunlight and eclipse).

- Define a detailed time power consumption and an orbital average power.
- Identify, select and size the elements of the power subsystem.
- Interconnect power elements considering their electrical specifications and limitations.
- Ensure that there is always enough power available for the loads even if the satellite was at the eclipsed phase of the orbit.
- Regulate the power delivered to the loads.
- Estimate the behaviour of the power taking into account degradation, path losses, etc.

Method

- Top level requirements
- Mission type
- Mission Life
- Configuration
- Payload

Identify Requirements

- Time power consumption
- Orbital average Power

Estimate power consumption

- Trade off solar panels
- Configuration
- Power produced
- Power available

Select and size power sources

- Trade off batteries
- Load current needs
- DoD
- Charge and Discharge

Select and size energy storage

- Trade off distribution system
- Protection of loads
- Converters

Select and size distribution and regulation

- Are requirement met?

Validation

COMET Tool

Power Subsystem Requirements

ID	Requirement	Parent ID
POWR 01	Power shall be enough to fulfil the mission	–
POWR 02	Power shall provide power to subsystems and payload	–
POWR 03	Power shall have a robust safe mode	–
POWR 04	Battery must provide sufficient power during eclipse operations	–
POWR 05	When direct line of sight with the Sun is not possible, the spacecraft(s) shall operate using the batteries	–
POWR 06	Power fault detection	–
POWR 07	Solar panels are deployed right after separation with launch vehicle	–
POWR 08	The peak power consumption of the payload shall not exceed the assumed power consumption	–
POWR 09	Power shall not surpass 1.5 kW	–

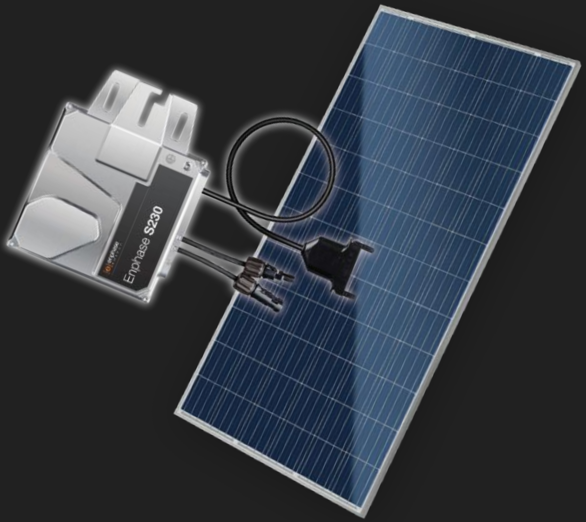
“Design is based on requirements. There's no justification for designing something one bit "better" than the requirements dictate”.

Akin's Laws.

Inputs & Outputs

Inputs

- Eclipse times
- Estimate power C
- Spacecraft Dry Mass
- Orbit Altitude
- Mission Duration
- Orbital Period
- Sunlight Time



Outputs

- Areas of solar panels
- Dimension of batteries
- EOL power available
- Total Power
- Power generation (eclipse)
- Power generation (Sunlight)
- Solar Cell Degradation per year

Design Process: Inputs

Eclipse Time

Estimated Spacecraft Average Power Consumption

Inputs from OCDT			
Orbital Period	378.35	**378.35**	**Min**
Eclipse Time	54.23333333	**54.23**	**Min**
Sunlight		**324.12**	**Min**
Subsytems Peak Power			
ADCS	83.14	**83.1**	**Watt**
C&DH	100	**100.0**	**Watt**
Power	308.5	**308.5**	**Watt**
Propulsion	760	**760.0**	**Watt**
Structure	0	**0.0**	**Watt**
Thermal	0	**0.0**	**Watt**
TT&C	100	**100.0**	**Watt**
Payload	800	**800.0**	**Watt**
Total		**2151.7**	**Watt**

Spacecraft Dry Mass

Inputs from OCDT			
Estimated Spacecraft Dry Mass	[m]	1016.119	**kg**

Orbit Altitude

Mission Duration

Inputs from OCDT			
Estimated Spacecraft Average Power Consumption	[P_mean]	1686.0999	**W**
Orbit Altitude (circular orbit)	[alt]	2300	**km**
Perigee Altitude (elliptical orbit)	[alt_peri]		**km**
Mission Duration	[dur]	10	**yr**

Source: ESA excel sheets

Design Process: Inputs

Eclipse Time

Estimated Spacecraft Average Power Consumption

Spacecraft Dry Mass

Orbit Altitude

Mission Duration

Inputs from OCDT			
Orbital Period	378.35	378.35	Min
Eclipse Time	54.23333333	54.23	Min
Sunlight		324.12	Min
Subsytems Peak Power			
ADCS	83.14	83.1	Watt
C&DH	100	100.0	Watt
Power	308.5	308.5	Watt
Propulsion	760	760.0	Watt
Structure	0	0.0	Watt
Thermal	0	0.0	Watt
TT&C	100	100.0	Watt
Payload	800	800.0	Watt
Total		2151.7	Watt

Inputs from OCDT			
Estimated Spacecraft Dry Mass	[m]	1016.119	kg

Inputs from OCDT			
Estimated Spacecraft Average Power Consumption	[P_mean]	1686.0999	W
Orbit Altitude (circular orbit)	[alt]	2300	km
Perigee Altitude (elliptical orbit)	[alt_peri]		km
Mission Duration	[dur]	10	yr

Source: ESA excel sheets

Process: Formulas

Solar Array

Eq.1 $Area = \boxed{P_{\mathrm{SA}}}/\boxed{P_{\mathrm{EOL}}}$

Eq.2 $P_{\mathrm{SA}} = (P_{\mathrm{sun}} + P_{\mathrm{CHARGE}})$

$P_{\mathrm{CHARGE}} = (T_{\mathrm{eclip}} * P_{\mathrm{eclip}})/T_{\mathrm{Sun}} * \eta_{\mathrm{batteries}}$

$P_{\mathrm{sun}} = 1.15 \cdot P_{\mathrm{TOTAL}}$

$P_{\mathrm{eclip}} = 1 * P_{\mathrm{sun}}$

$T_{\mathrm{Sun}} = T - T_{\mathrm{eclip}}$

Eq.3 $P_{\mathrm{EOL}} = (P_{\mathrm{BOL}} * (1 - \gamma)^{\mathrm{YEARS}}$

$P_{\mathrm{BOL}} = S * \eta * cos\theta * \eta_{\mathrm{st}}$

Battery

Eq.1 $m_{\mathrm{battery}} = \boxed{E}/\boxed{e}$

Eq.2 $E = V_{\mathrm{av}} * Chg$

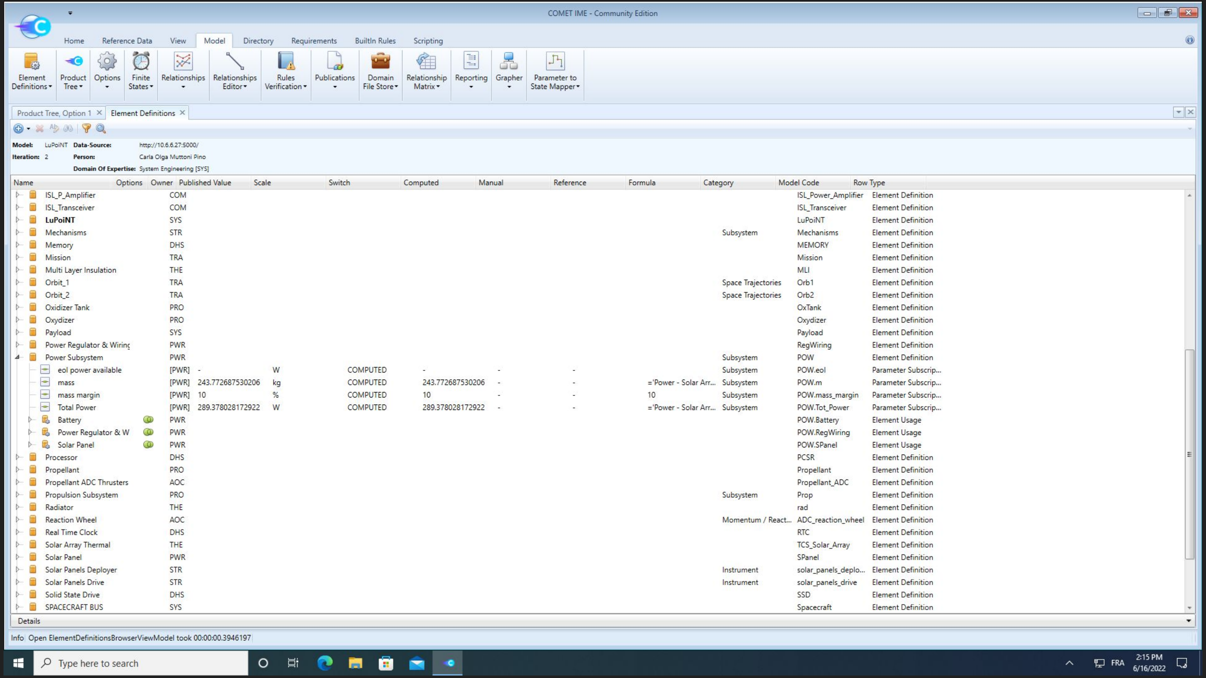
$Chg = P_{\mathrm{eclip}} * T_{\mathrm{load}}/V_{\mathrm{av}} * \%DODmax$

e = energy density (type battery)

High %DoD → smaller cycle

Source: Study of a lunar satellite navigation system. Gemma Saura Carretero

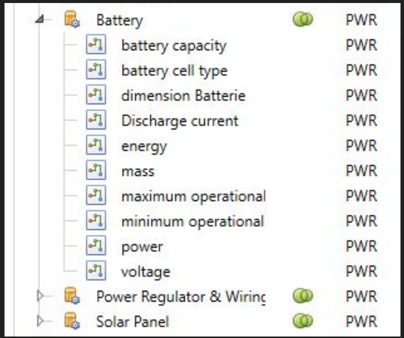
COMET (Model)

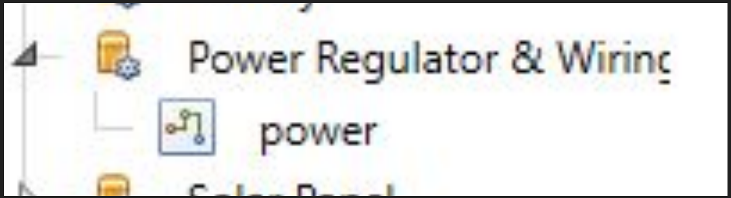


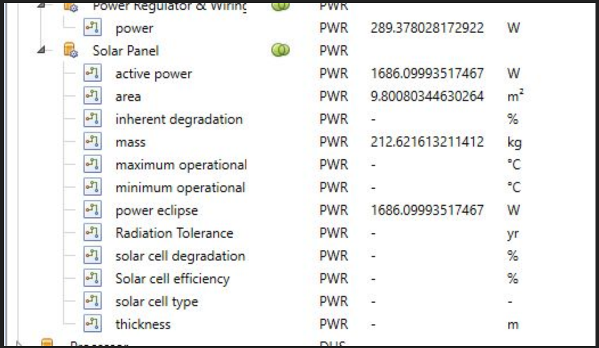
Name	Options	Owner	Published Value	Scale	Switch	Computed	Manual	Reference	Formula	Category	Model Code	Row Type
ISL_P_Amplifier		COM									ISL_Power_Amplifier	Element Definition
ISL_Transceiver		COM									ISL_Transceiver	Element Definition
LuPoiNT		SYS									LuPoiNT	Element Definition
Mechanisms		STR								Subsystem	Mechanisms	Element Definition
Memory		DHS									MEMORY	Element Definition
Mission		TRA									Mission	Element Definition
Multi Layer Insulation		THE									MLI	Element Definition
Orbit_1		TRA								Space Trajectories	Orb1	Element Definition
Orbit_2		TRA								Space Trajectories	Orb2	Element Definition
Oxidizer Tank		PRO									OxTank	Element Definition
Oxydizer		PRO									Oxydizer	Element Definition
Payload		SYS									Payload	Element Definition
Power Regulator & Wiring		PWR									RegWiring	Element Definition
Power Subsystem		PWR								Subsystem	POW	Element Definition
eol power available		[PWR]	-	W	COMPUTED	-	-	-		Subsystem	POW.eol	Parameter Subscrip...
mass		[PWR]	243.772687530206	kg	COMPUTED	243.772687530206	-	-	='Power - Solar Arr...	Subsystem	POW.m	Parameter Subscrip...
mass margin		[PWR]	10	%	COMPUTED	10	-	-	10	Subsystem	POW.mass_margin	Parameter Subscrip...
Total Power		[PWR]	289.378028172922	W	COMPUTED	289.378028172922	-	-	='Power - Solar Arr...	Subsystem	POW.Tot_Power	Parameter Subscrip...
Battery		PWR									POW.Battery	Element Usage
Power Regulator & W		PWR									POW.RegWiring	Element Usage
Solar Panel		PWR									POW.SPanel	Element Usage
Processor		DHS									PCSR	Element Definition
Propellant		PRO									Propellant	Element Definition
Propellant ADC Thrusters		AOC									Propellant_ADC	Element Definition
Propulsion Subsystem		PRO								Subsystem	Prop	Element Definition
Radiator		THE									rad	Element Definition
Reaction Wheel		AOC								Momentum / React...	ADC_reaction_wheel	Element Definition
Real Time Clock		DHS									RTC	Element Definition
Solar Array Thermal		THE									TCS_Solar_Array	Element Definition
Solar Panel		PWR									SPanel	Element Definition
Solar Panels Deployer		STR								Instrument	solar_panels_deplo...	Element Definition
Solar Panels Drive		STR								Instrument	solar_panels_drive	Element Definition
Solid State Drive		DHS									SSD	Element Definition
SPACECRAFT BUS		SYS									Spacecraft	Element Definition

In this section we see the COMET model which allowed us to work co-design, co-simulation, and co-verification with the others disciplines.

Model


Battery PWR
battery capacity PWR
battery cell type PWR
dimension Batterie PWR
Discharge current PWR
energy PWR
mass PWR
maximum operational PWR
minimum operational PWR
power PWR
voltage PWR
Power Regulator & Wiring PWR
Solar Panel PWR


Power Regulator & Wiring
power


power PWR 289.378028172922 W
Solar Panel PWR
active power PWR 1686.09993517467 W
area PWR 9.80080344630264 m²
inherent degradation PWR - %
mass PWR 212.621613211412 kg
maximum operational PWR - °C
minimum operational PWR - °C
power eclipse PWR 1686.09993517467 W
Radiation Tolerance PWR - yr
solar cell degradation PWR - %
Solar cell efficiency PWR - %
solar cell type PWR - -
thickness PWR - m

First Results : ITERATION 1 - Orbit: 2,300 km

INPUTS

- ★ Average Power Consumption = 1,571.7 W
- ★ **Orbit altitude = 2,300 km**
- ★ Mission Duration = 10 years
- ★ Eclipse Time = 54.23 min

OUTPUTS

- → Solar array area = 9.14 m^2
- → P_{BOL} = 258.7 W/m^2
- → P_{EOL} = 246.05 W/m^2

- → Battery capacity = 3,629.7 Wh

- → Power consp Reg & Wiring = 269.7 W
- → Power subsystem mass = 226.05 kg

First Results : ITERATION 2 - Orbit: 5,000 km

INPUTS

- ★ Average Power Consumption = 1947.87W
- ★ Orbit altitude = 5,000 km
- ★ Mission Duration = 10 years
- ★ Eclipse Time = 67.9 min

OUTPUTS

- → Solar array area = 10.42 m^2
- → P_{BOL} = 258.7 W/m^2
- → P_{EOL} = 246.05 W/m^2

- → Battery capacity = 5685.7 Wh

- → Power consp Reg & Wiring = 307.5W
- → Power subsystem mass = 264,3 kg

First Results : ITERATION 3 - Orbit 2,300 km + Hybrid Propulsion system

INPUTS

- ★ Average Power Consumption = 1686.1 W
- ★ Orbit altitude = 2,300 km
- ★ Mission Duration = 10 years
- ★ Eclipse Time = 54.23 min
- ★ Hybrid propulsion system

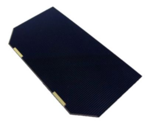
OUTPUTS

- ➔ Solar array area = 10.45 m^2
- ➔ P_{BOL} = 258.7 W/m^2
- ➔ P_{EOL} = 246.05 W/m^2
- ➔ Battery capacity = 3893.9 Wh
- ➔ Power consp Reg & Wiring = 308.5W
- ➔ Power subsystem mass = 255,2 kg

First Results : 3rd Iteration - Trade-off

COMPONENTS TRADE-OFF - Solar Panels

Material	Area (m^2)	P_{BOL} (W/m^2)	P_{EOL} (W/m^2)	Mass (kg)
Silicon	27.12	130.31	88.92	96.5
High Efficiency Silicon	20.83	160.24	115.75	83.2
Single junction GaAs	17.38	183.40	138.77	77.8
Dual junction III-V	12.18	212.36	197.96	70.9
Triple-junction III-V	9.80	258.70	246.05	63.5



Source: Sunket, AzureSpace, OCE technology

First Results : 3rd Iteration - Trade-off

COMPONENTS TRADE OFF - Battery

Material	Capacity (W hr)	Voltage (V)	Mass (kg)
Lithium-Ion	3893.9	28	31.2
Nickel-Cadmium	10600	28	353.3
Nickel-Hydrogen	3634.3	28	60.6

Source: OCE technology

First Results : 3rd Iteration - Trade-off

COMPONENTS TRADE OFF - Regulators & Wiring

Component	Weight (kg)	Power (W)	Comments
Solar Arrays	0.04 *P*	—	× π for cylindrical body-mounted × 4 for omnidirectional body mounted
Batteries	$C/35$ (NiCd) $C/45$ (NiH_2)	—	C = capacity in W·hrs
Power Control Unit	0.02 *P*	—	*P* = controlled power
Regulator/Converter	0.025 *P*	0.2 *P*	*P* = converted power
Wiring	0.01–0.04 M_{dry}	0.02–0.05 *P*	M_{dry} = spacecraft dry weight

Source: SMAD, Chpt.10.4, page 334,

	PCU		Regulator		Wiring	
	DET	PPT	DET	PPT	DET	PPT
Power (W)	-	-	180	168.8	128.6	120.6
Mass (kg)	51.4	48.2	64.3	60.3	40.6	40.6

2% Ptotal

2.5% Ptotal

4% Dry mass

2.5% Dry Mass

DET =Direct Energy Transfer. Dissipates unneeded power. Usually systems less than 100W
PPT = Peak Power Tracking. Extract the real power required from Solar Array. Needs 4%-7% Ptotal.
Source: Spacecraft Power Systems. David W. Miller. John Keesee

First Results : 3rd Iteration - Trade-off

COMPONENTS TRADE OFF - Regulators & Wiring

Component	Weight (kg)	Power (W)	Comments
Solar Arrays	0.04 P	—	× : for cylindrical body-mounted × : for omnidirectional body mounted
Batteries	$C/35$ (NiCd) $C/45$ (NiH_2)	—	C = capacity in W·hrs
Power Control Unit	0.02 P	—	P = controlled power
Regulator/Converters	0.025 P	0.2 P	P = converted power
Wiring	0.01–0.04 M_{dry}	0.02–0.05 P	M_{dry} = spacecraft dry weight

Source: SMAD, Chpt.10.4, page 334,

	PCU		Regulator		Wiring	
	DET	PPT	DET	PPT	DET	PPT
Power (W)	-	-	180	168.8	128.6	120.6
Mass (kg)	51.4	48.2	64.3	60.3	40.6	40.6

20% → 7% Ptotal

5% Ptotal

2.5% Dry Mass

DET =Direct Energy Transfer. Dissipates unneeded power. Usually systems less than 100W
PPT = Peak Power Tracking. Extract the real power required from Solar Array. Needs 4%-7% Ptotal.
Source: Spacecraft Power Systems. David W. Miller. John Keesee

Conclusion

- The creation and use of power was evaluated linearly and at its maximum during the entire mission, in other to simplify the analysis.
- The power produced is always higher than the power consumed positive power budget.
- The batteries have the adequate capacity to supply the loads needs (current, voltage, power).
- The discharge time is compensated by the charge time.
- The distribution system interfaces the power source and storage elements with the load, protecting the main bus and the loads.

Future steps would be adjust power design modifying parameters shown before: wiring and regulators power and mass, %DoD, inherent degradation, etc..

Contribution Slides

- Introduction [Carla]
- Power subsystem Design Process [Carla]
- Process [Laura]
- Model [Carla]
- Final Results [Laura]
- Conclusion [Laura & Carla]

CDH and Communication SubSystem

Final Presentation - 16.06.22

Ahmed Baahmed

Introduction

Responsible for:

- Transmission and reception of data from and to the satellite.
- Telemetry, Tracking and Control (TTC) between Spacecraft and ground.

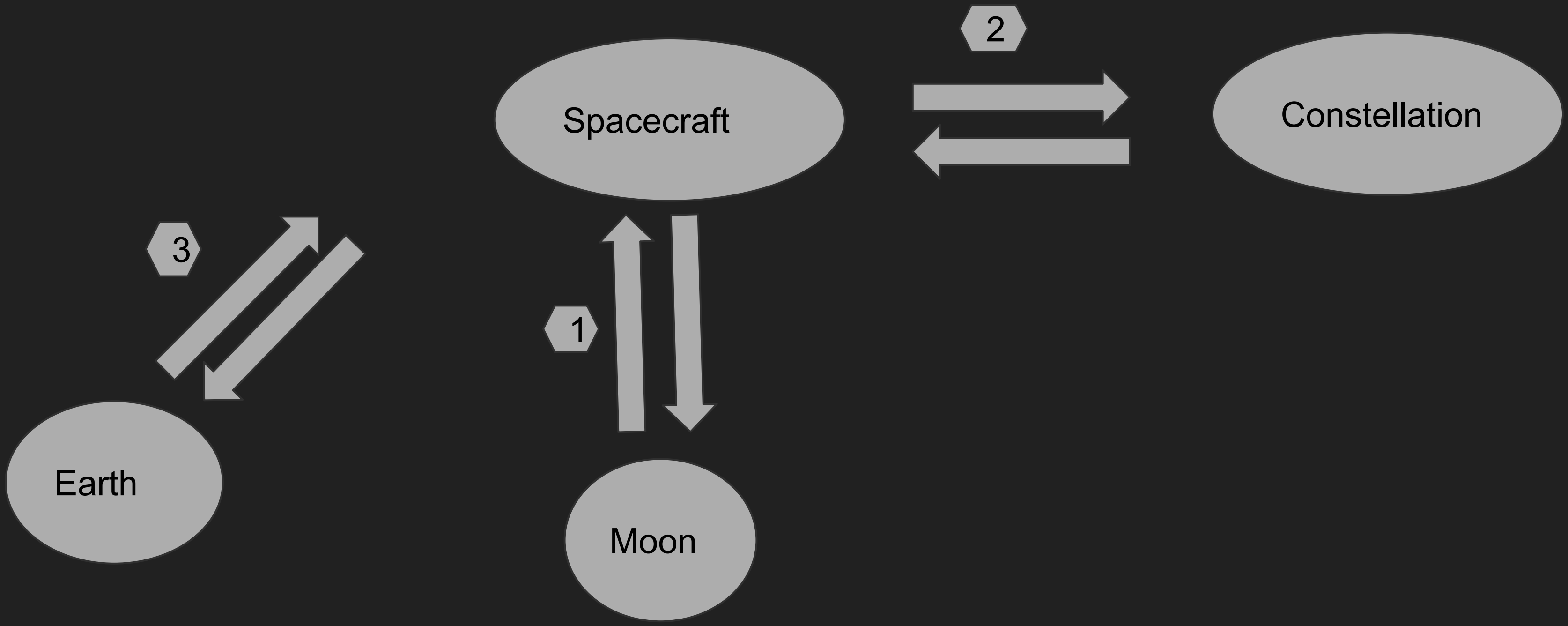
Elements selection

- COST components reduce new development risk and launch time.
- Enough for the data rate required by DHS.
- Shared Transceiver and Power Amplifier to save weight, volume, and power.
- L-band high gain broadband space antenna
- s-band patch antenna (2)
- x-band 4x4 Patch array antenna

Requirements

- COMMs subsystem should perform carrier tracking and lock onto the GS.
- COMMs subsystem should receive, process and transmit ranging signal in order to determine the Satellite’s position.
- Telemetry modulation and transmission should be performed.
- Data rates to be detected between the Spacecraft and the Moon, Inter-Satellites, and Spacecraft and Earth.
- Frequency, Beamwidth, EIRP TBD.
- The uplink system must be able to achieve maximum access time between Spacecraft and GS.
- The COMMs part must comply with the Satellite standard.

Design



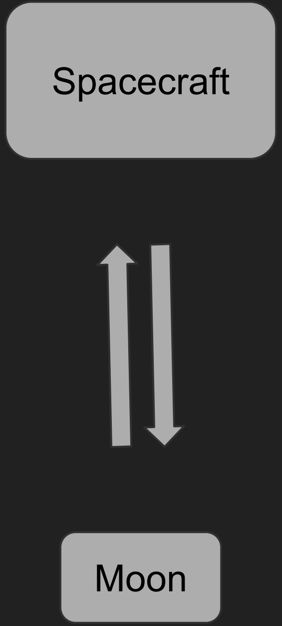
Communication with the Moon

Requirements:

- The transmission frequency shall be 1575.42 MHz, the same as in GPS L1.
- The transmission bandwidth shall be 20 MHz (similar to the GPS bandwidth).
- The minimum power guaranteed in reception shall be -160 dBW (L1 GPS standard).

  A 4.5 dBW margin is added to cover the uncertainty. The final power in reception shall be -155.5 dBW.

- The satellite range is 2300 km.
- The transmission beamwidth (BW) is 20°.
- Transmission and reception losses (Lt and Lr) are both set in 3 dB



Results

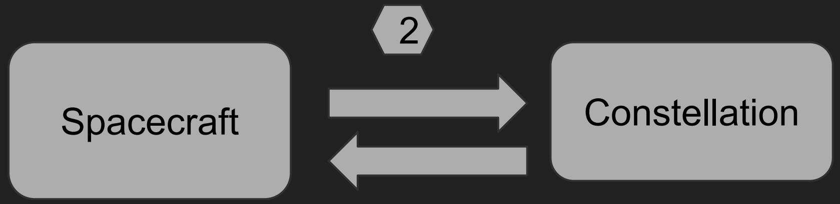
- The Energy per bit to Noise rate (Eb/N0) will be 67.28 dB which is high enough to assure a good link quality.
- For a gain (G/T) value of 6.11 dB, the output power required is 12 W
- The minimum Effective Isotropic Radiated Power (EIRP = 24.01 dBM)
- The link quality, The Margin=57.28 dB, is good for the communication system

Parameter	Downlink	Uplink
EIRP (dBW)	24.01	37
G/T (dB/k)	6.11	13
Eb/No (dB)	67.28	97.1
Margin (dB)	57.28	15

Inter-Satellite Link Communication

<u>Requirements:</u>

- The Satellite range is 2512 km.
- Frequency band is S-band.
- The transmission beamwidth (BW) will be 20°.
- Transmission and reception losses (Lt and Lr) are both set in 3 dB
- Data rate is 124 kbits.



Link Budget

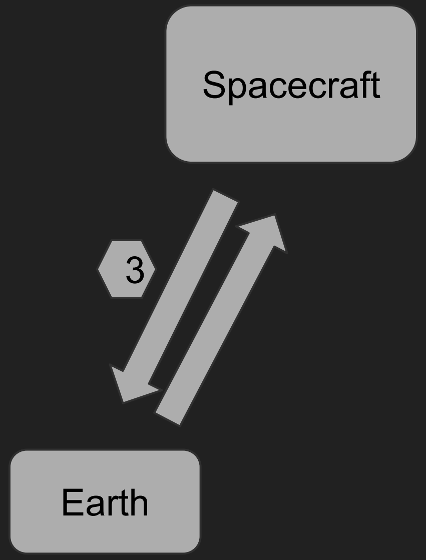
- The Energy per bit to Noise rate (Eb/N0) will be 44.46 dB which is high enough to assure a good link quality.
- For a gain (G/T) value of 9.58 dB, the output power required is 12 W
- The minimum Effective Isotropic Radiated Power (EIRP = 25.78 dBM)
- The link quality, The Margin=31.96 dB, is good for the communication system

Parameter	Downlink	Uplink
Distance (KM)	2512	2512
EIRP (dBW)	25.78	25.78
G/T (dB/k)	9.58	9.58
Eb/No	44.46	44.46
Margin	31.96	31.96

Communication with the Earth

Requirements

- Satellite range is 384400 km.
- The transmission frequency will be 8.4 GHz (X-band).
- The transmission beamwidth (BW) will be 20°.
- Transmission and reception losses (Lt and Lr) are both set in 3 dB.
- The download data rate will be 50 bps (the same as GPS).



Link Budget

- The Energy per bit to Noise rate (Eb/N0) will be 22.849 dB which is high enough to assure a good link quality.
- For a gain (G/T) value of 6.11 dB, the output power required is 12 W
- The minimum Effective Isotropic Radiated Power (EIRP = 24.01 dBM)
- The link quality, The Margin=23.849 dB, is perfect for the communication system.

Parameter	Downlink	Uplink
Distance (KM)	384400	/
EIRP (dBW)	24.01	/
G/T (dB/k)	6.11	/
Eb/No	22.849	/
Margin	23.84	/

COMET

- First iteration: 2300 km.
- Second iteration: 5000 km.
- Third iteration: 2300 km Hybrid Propulsion.

COMMUNICATION	COM			
mass	COM	13.16	kg	COMPUTED
mass margin	COM	10	%	MANUAL
power	COM	49	W	COMPUTED
ISL_Antenna	COM			
ISL_Antenna_1	COM			
ISL_Antenna_2	COM			
ISL_Antenna_3	COM			
ISL_P_Amplifier	COM			
ISL_Transceiver	COM			

Conclusion

- It is possible to transfer the information for the mission.
- The link budget is closed with a good margin.
- The components selected fall within the power and mass budget.

Thermal Control Subsystem

TCS

Final Presentation - 16.06.22

Misbah Rashid

Introduction

Thermal system ensures to maintain all spacecraft and payload components and subsystems within their required temperature limits for each mission phase

Operational limits - components must remain within the range while operating
Survival limits - components must remain within at all times, even when not powered

In lunar missions, variations in solar flux and lunar eclipse may result in extreme cold and hot temperatures

Requirements and Design Drivers

ID	Requirement	Parent ID
TCSR 01	The spacecraft thermal system shall verify or regulate that all components are within an acceptable thermally operational range 5° to 35°.	–
TCSR 02	The thermal control system shall meet performance requirements for the full 10 years duration of the mission.	MREQ 04
TCSR 03	The active thermal system shall be designed to maintain all components within the survival temperature limits during cruise.	–
TCSR 04	The thermal control shall dissipate heat produced by power subsystem to satisfy temperature requirements of components.	–
TCSR 05	The thermal system shall have components with TRL no less than 6.	–

- Spacecraft material, mass and dimensions
- Absorbed radiation and albedo
- Temperature limits of spacecraft and its subsystems
- Solar array configuration

Inputs & Outputs - N2 Chart

SUBSYSTEM	INPUTS (FROM OTHER SUBSYSTEMS)	OUTPUTS
PAYLOAD	–	–
TRAJECTORY AND ORBIT	Operational orbit altitude, perigee, eclipse time, mission duration	
STRUCTURE	Spacecraft structure material, area, mass, configuration	Mass, dimension and location of components
PROPULSION	-	-
POWER	Power generation (eclipse and sunlight), area of solar array, cell efficiency, inherent degradation, solar cell degradation per year	Power generation
ADC	-	-
COMS & CDH	-	-

Components

	COMPONENT (+PROPERTIES)	MISSION PHASE	JUSTIFICATION
Passive	**MULTI-LAYER INSULATION**	All mission phases	Propellant tanks, propellent lines
Passive	**RADIATOR**	LLOI, LOI, PNT, EOL	Batteries and high solar loads
Passive	**HEAT PIPES**	Cruise, LLOI, LOI, PNT	Solar arrays
Passive	**SURFACE FINISH - WHITE PAINT**	All mission phases	Limit min-max temperatures of heater-power requirements
Passive	**TEMPERATURE SENSORS**	All mission phases	Monitor component temperatures
Active	**HEATING SYSTEM**	Cruise	S&M, PROP, POW

Surface Finish Trade-off

Surface Finish	Absorptivity (BOL)	Emissivity
White Paint - Chemglaze A276	0.22 - 0.28	0.88
Black Velvet - 3M	0.97	0.87
Aluminized Kapton - ½ mil	0.34	0.55

COMET Model

Thermal Control Subsystem	THE								Subsystem	TCS
area	THE	-	m^2	MANUAL	-	-	-	-	Subsystem	TCS.area
mass	THE	25	kg	MANUAL	-	25	-	-	Subsystem	TCS.m
mass margin	THE	10	%	MANUAL	-	10	-	-	Subsystem	TCS.mass_margin
power	THE	50	W	MANUAL	-	50	-	-	Subsystem	TCS.P
power margin	THE	10	%	MANUAL	-	10	-	-	Subsystem	TCS.pamargin
Heat Pipe	THE									TCS.Heat_Pipe
mass	THE	5.37	kg	MANUAL	-	5.37	-	-		Heat_Pipe.m
Heater	THE									TCS.Heater_THR
mass	THE	-	kg	MANUAL	-	-	-	-		Heater_THR.m
power	THE	-	W	MANUAL	-	-	-	-		Heater_THR.P
Multi Layer Insulation	THE									TCS.MLI
mass	THE	5.329	kg	MANUAL	-	5.329	-	-		MLI.m
Radiator	THE									TCS.rad
area	THE	3.474	m^2	MANUAL	-	3.474	-	-		rad.area
mass	THE	2.53	kg	MANUAL	-	2.53	-	-		rad.m
volume	THE	-	m^3	MANUAL	-	-	-	-		rad.V
Surface Finish	THE									TCS.TCS_Surface_Fi...
absorptance	THE	0.28	-	MANUAL	-	0.28	-	-		TCS_Surface_Finish....
emissivity	THE	0.88	-	MANUAL	-	0.88	-	-		TCS_Surface_Finish....
mass	THE	12.88	kg	MANUAL	-	12.88	-	-		TCS_Surface_Finish.m
Thermal Sensor	THE									TCS.TCS_Thermal_S...
mass	THE	0.25	kg	MANUAL	-	0.25	-	-		TCS_Thermal_Sens...

Design

Thermal Control - Spacecraft

Inputs from OCDT							
Orbit Altitude (circular orbit)	[Alt]	**2300**	**km**				
Perigee Altitude (elliptical orbit)	[Alt]		**km**	Structure Material		**Aluminium 6061**	
Eclipse Time	[Ecl]	**54.3**	**Min**	Absorptivity	abs	**0.16**	
Expected Spacecraft Body Area	[Area]	**9.6**	**m2**	Emissivity	eps	**0.78**	
Spacecraft Power Consumption (sunlight)	[P_mean]	**1686.1**	**W**	Thermal Conductivity	k	**180**	**[W/mK]**
Spacecraft Power Consumption (eclipse)	[P_mean]	**1686.1**	**W**	Specific Heat Capacity	C	**896**	**[J/kgK]**
Spacecraft Mass	[m]	**1016.11**	**Kg**	Central Body		**Moon**	
Spacecraft Configuration	**Deployable SA**			Solar Flux Intensity		**Minimum**	

Thermal Control - Solar Array

Inputs from OCDT				
Orbit Altitude (circular orbit)	[alt]	**2300**	**km**	Worst-case Sun Incidence Angle
Perigee Altitude (elliptical orbit)	[alt]		**km**	
Mission Duration	[dur]	**10**	**yrs**	
Solar Array Area	[Area]	**10.45**	**m2**	
Ideal Solar Cell Efficiency	[Eff]	**26.8**	**%**	
Inherent Degradation	[degrad]	**77**	**%**	
Solar Cell Degradation per year	[degr_solar_ce	**0.5**	**%/year**	

Source: ESA Excel Calculation Sheets

First Iteration - Orbit 2,300 km

Inputs

- Eclipse time: 54.23 minutes
- Spacecraft power consumption (sunlight): 1,571 W
- Spacecraft mass: 1,133 kg
- Spacecraft area: 6.79 m^2
- Mission duration: 10 years
- Solar array area: 9.14 m^2
- Ideal solar cell efficiency: 26.8%
- Inherent degradation: 77%
- Solar cell degradation per year: 0.5%/year

Outputs

Temperature

- Maximum equilibrium temperature: 21°C
- Minimum equilibrium temperature: 7.4°C
- Actual minimum temperature after eclipse: 14.5°C

Radiator

- Radiator area: 3.5 m^2

Solar array temperature

- Maximum equilibrium temperature: 46.61°C
- Minimum equilibrium temperature: -218.59°C

Second Iteration - Orbit 5,000 km

Inputs

- Eclipse time: 68.7 minutes
- Spacecraft power consumption (sunlight): 1,947 W
- Spacecraft mass: 900.6 kg
- Mission duration: 10 years
- Solar array area: 10.42 m^2
- Ideal solar cell efficiency: 26.8%
- Inherent degradation: 77%
- Solar cell degradation per year: 0.5%/year

Outputs

Temperature

- Maximum equilibrium temperature: 26.9°C
- Minimum equilibrium temperature: 22.8°C
- Actual minimum temperature after eclipse: 7°C

Radiator

- Radiator area: 4.3 m^2

Solar array temperature

- Maximum equilibrium temperature: 40,52°C
- Minimum equilibrium temperature: -230.88°C

Third Iteration - Orbit 2,300 km + hybrid propulsion

Inputs

- Eclipse time: 54.23 minutes
- Spacecraft power consumption (sunlight): 1,686 W
- Spacecraft mass: 1,233 kg
- Spacecraft area: 9.6 m^2
- Mission duration: 10 years
- Solar array area: 10.42 m^2
- Ideal solar cell efficiency: 26.8%
- Inherent degradation: 77%
- Solar cell degradation per year: 0.5%/year

Outputs

Temperature

- Maximum equilibrium temperature: 9.8°C
- Minimum equilibrium temperature: -11.2°C
- Actual minimum temperature after eclipse: 14.5°C

Radiator

- Radiator area: 3.75 m^2

Solar array temperature

- Maximum equilibrium temperature: 46.61°C
- Minimum equilibrium temperature: -218.6°C

Conclusions/Assessments

- Thermal control is critical to ensuring the performance and survival of spacecraft and payload components.
- Radiators and surface finish will compensate for the internal heat generated
- Passive over active control meets the requirements and constraints at an acceptable mass, cost, and risk.

Individual Responsibilities

Individual	Responsibilities
Misbah	Literature Review
	Thermal Requirements
	Thermal Control Components selection
	Determine radiator and heater size and power
	TCS mass and power
	COMET
	Presentation
	Report

Thank you!

Questions?