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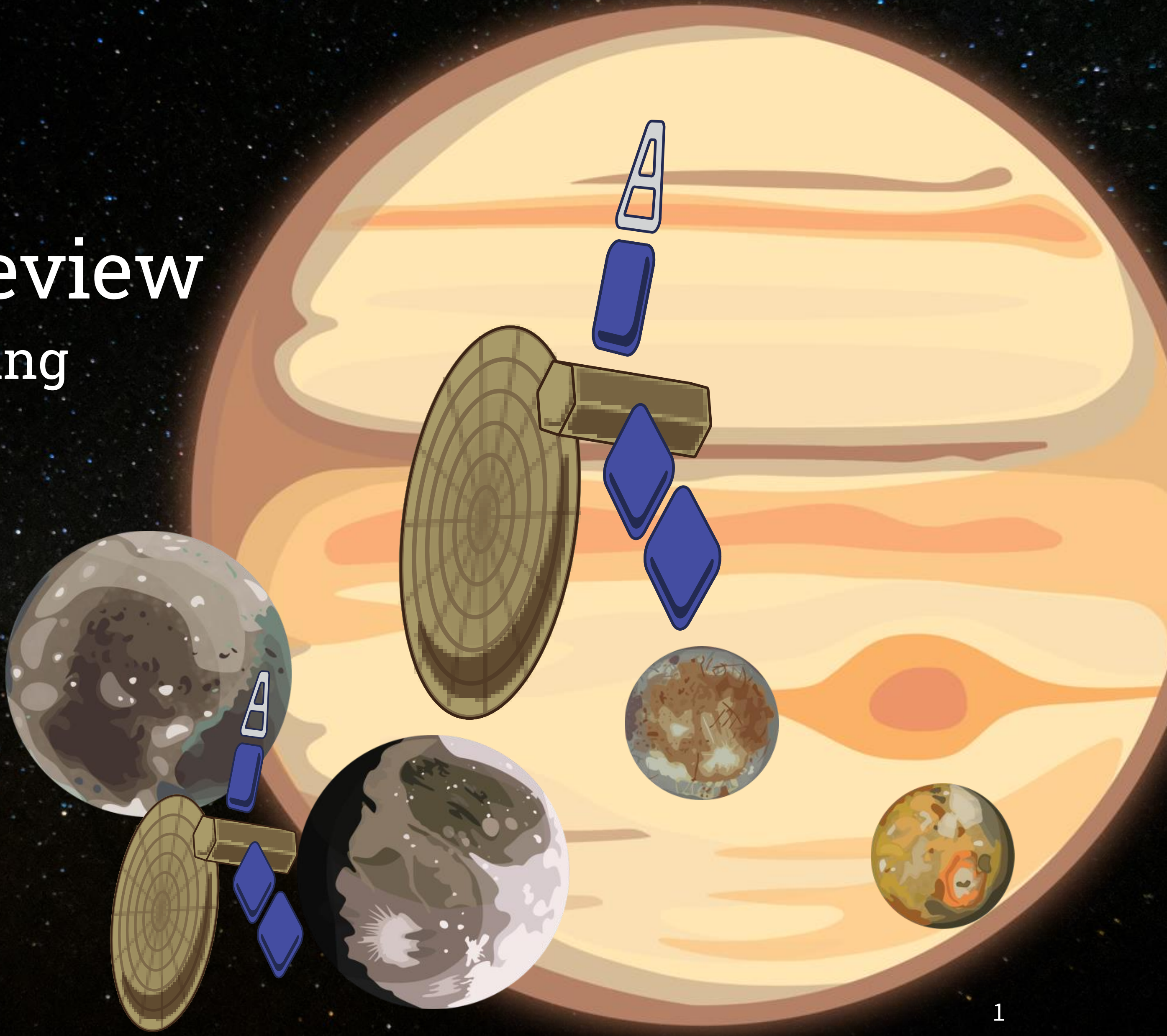
MOJO

Preliminary Design Review

Cal Poly Aerospace Engineering

Moons Of Jupiter Observation
Exploration Breeds Knowledge



Introduction

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Cal Poly Aerospace Engineering Astronautics Senior Design

- Spacecraft bus solicitation provided by Zeus Industries
  - Create a mission concept that supports Zeus' 4K optical payload, Zeus-1, allowing it to image the Galilean moons in the Jovian System
- MOJO – Moons of Jupiter Observation
  - 76 undergraduate and blended (B.S. + M.S.) program students
  - Broken up into 10 subsystem teams of about 5 – 10 students each

Agenda

Introduction

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1. Mission Overview
2. Concept of Operations
3. Trajectory
4. Propulsion
5. Communications
6. Command & Data Handling
7. Electrical Power System
8. Structures & Mechanisms
9. Attitude & Orbit Controls System
10. Thermal Control System
11. Project Management
12. Wrap Up

Please feel free to ask questions throughout the presentation.

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Mission Overview

Carlin Sherman-Shannon

Stakeholder Needs and Constraints

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Mission Overview

- System shall support the Zeus-1 optical payload to take detailed video of Jupiter's Galilean moons: Callisto, Europa, Ganymede, and Io
  - 4k video livestream of what the system is currently recording provided to Earth-based users - *successful pushback to intermittently transmit recorded video data per "Zeus-1 Mission Change Request" dated 3/6/2023*
  - 100% surface video coverage provided from 750-1000km above each moon
- Budget: $500 million
  - Up to $250 million extra per additional scientific payload with a maximum of 5 payloads
- Launch date by December 2025 – *successful pushback to 2026*
- Total system reliability greater than 98.5%
- Mission operations conducted within the Jovian system for 10-15 years
- Nuclear energy sources may not be utilized
- Compliance with all national and international guidelines, regulations, and policies

Zeus-1 Payload

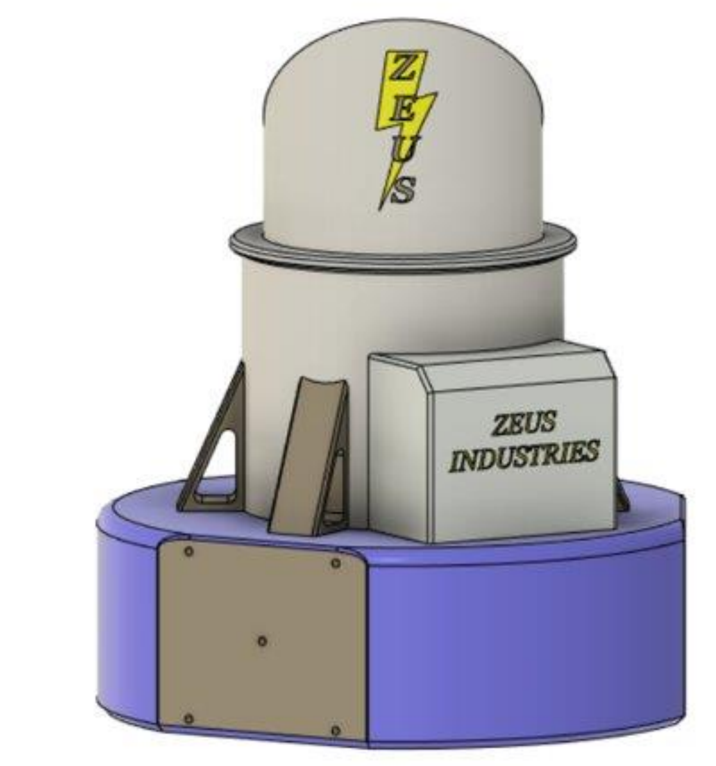
Mission Overview

Primary function:

- Captures videographic imagery of Galilean moons

Main Impacts:

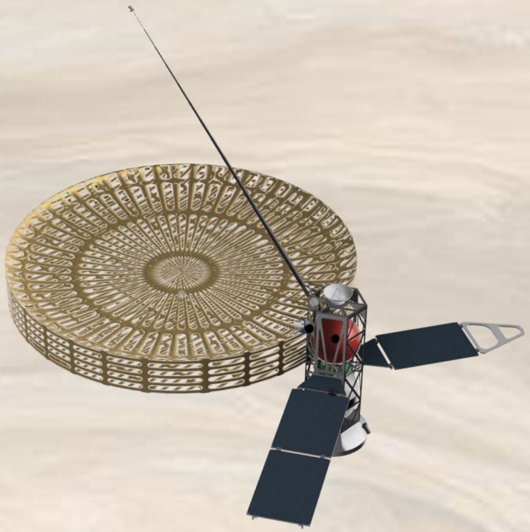
- Pointing Requirements:
  - 0.001° – 0.1° during collection
- Power Requirements:
  - 100 – 150 W
- Data handling and storage Requirements:
  - Data generation of 10s of Mbps at peak
- Communication Requirements:
  - S/C constrained to 225 kbps

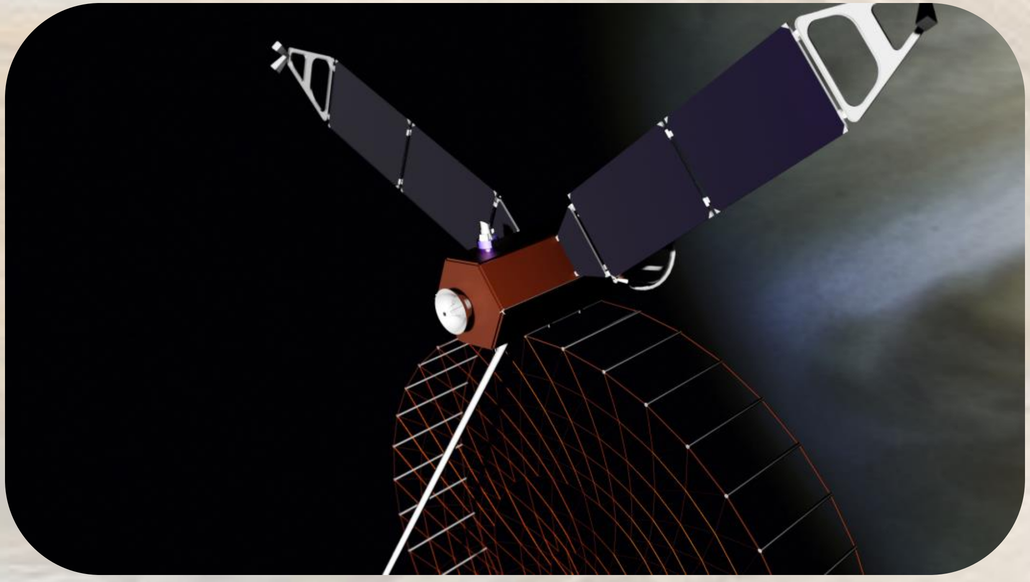


Model of Zeus-1 Payload provided by Zeus Industries

Spacecraft Overview

- Galilean Exploratory Craft (GEC)
- Twin spacecraft to cover different moons:
- The spacecraft’s fold-out solar arrays and large 13.5-meter mesh high gain antenna give it a signature shape
- Zeus-1 optical payload sits on a gimbal on the side of the spacecraft



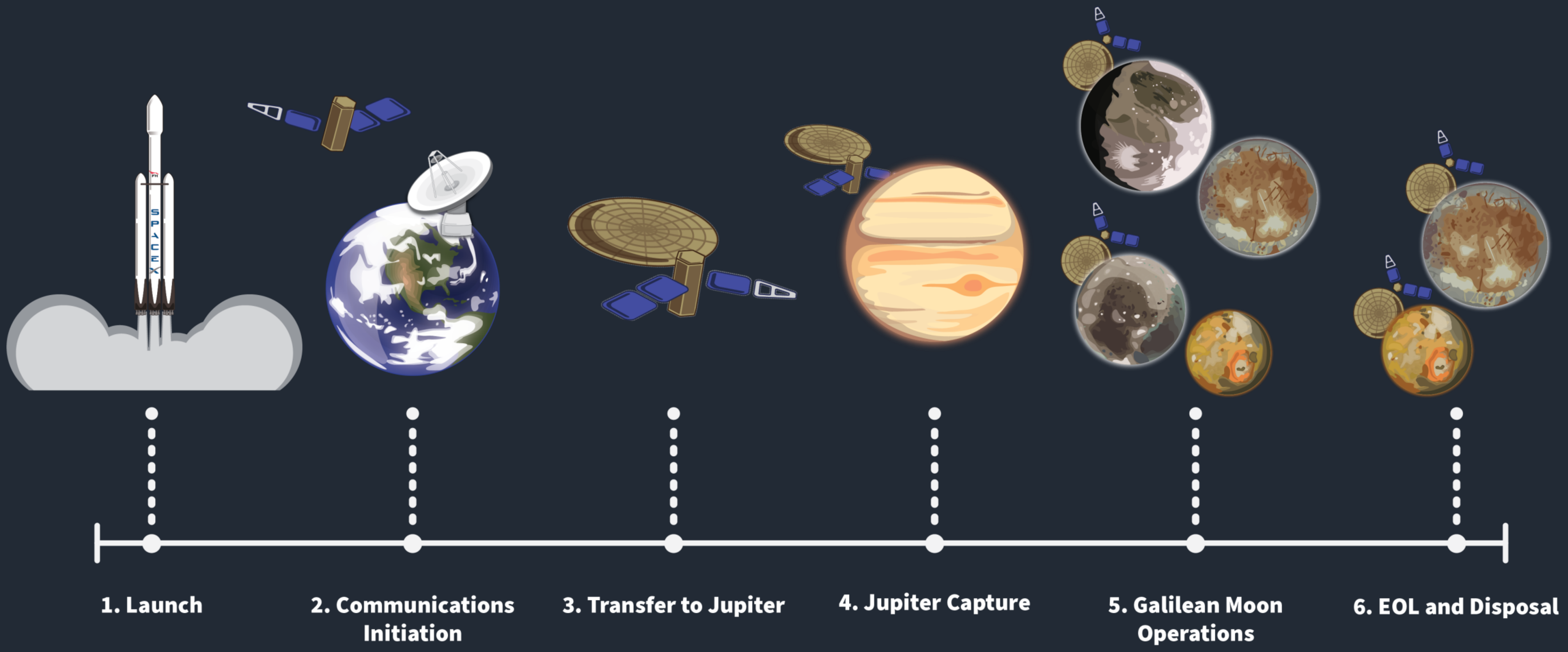


Concept of Operations

Presenter Name

Concept of Operations

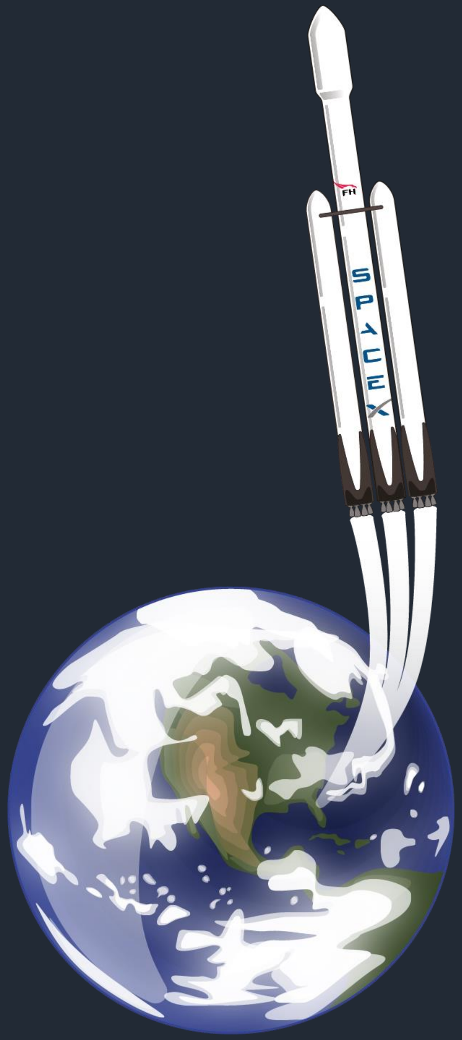
Phases Diagram



Phase 1: Launch

Concept of Operations

- First launch window: January 8th, 2026 – January 30th, 2026*
  - GEC – 1
- Second launch window: November 24th, 2026 – December 16th, 2026
  - GEC – 2
- GEC – 1 & GEC – 2 are integrated into individual Falcon Heavy Launch Vehicles at Cape Canaveral launch site
- Ground Segment prepares to initiate communication with the space segment



Phase 2: Communications Initiation

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Concept of Operations

- After their respective launches, GEC – 1 & GEC – 2 (Space Segment) detach from their launch vehicles
- Spacecraft powers on and an initial check of subsystem health and instrument status is performed
- Ground segment establishes a data link with space segment's low gain antenna and continues sending check commands to the Space Segment as it finishes stabilizing attitude
- Spacecraft deploy their high gain antennae at the end of the phase to confirm successful deployment before leaving Earth

Phase 3: Transfer to Jupiter

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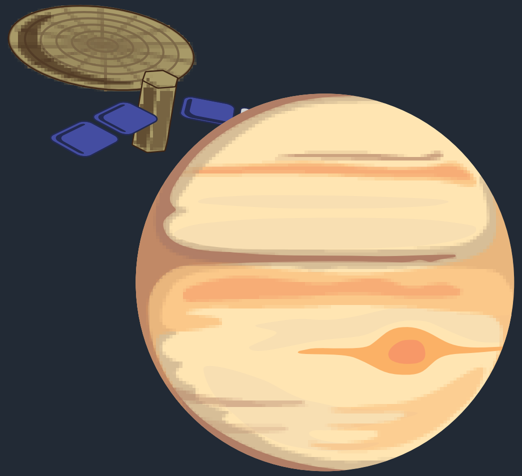
Concept of Operations

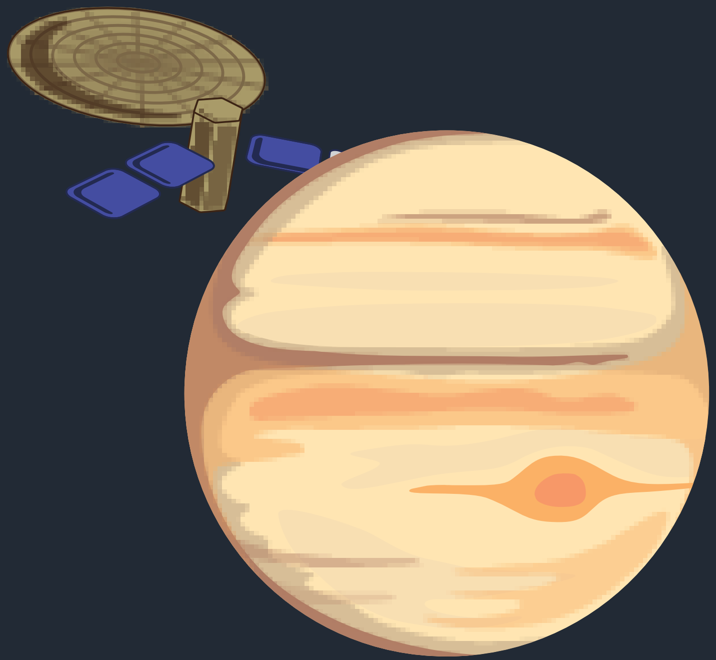
- Space Segment burns to exit Earth's Sphere of Influence and enters a heliocentric orbit
- Space Segment performs deep space maneuvers to prepare for the Earth Gravity Assist
- Space Segment performs an Earth Gravity Assist to enter a trajectory to rendezvous with Jupiter
  - GEC – 1 arrives on April 27th, 2031
  - GEC – 2 arrives on September 2nd , 2031



Phase 4: Jupiter Capture

Concept of Operations

- Space Segment burns and enters orbit around Jupiter
- Each spacecraft performs separate correction maneuvers to enter respective trajectories which will obtain video coverage of two moons
  - GEC – 1: Ganymede + Callisto
    - Moon centric orbits
  - GEC – 2: Io + Europa
    - Jupiter centric orbits



Phase 5: Galilean Moon Operations

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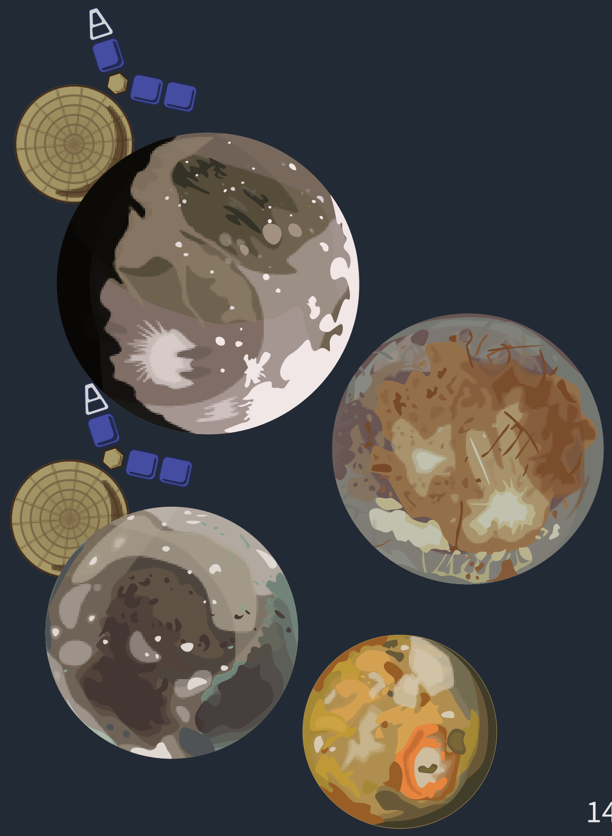
Concept of Operations

- Each spacecraft performs orbital corrections to remain on trajectory
- Each spacecraft transmits 4K video of their respective Galilean moons via high gain antenna
  - GEC – 1: orbit Callisto and Ganymede
  - GEC – 2: flybys of Europa and Io
- Each spacecraft transmits health and scientific data to the Ground Segment via its medium gain antenna
- Ground Segment processes received data



Phase 6: End of Life and Disposal

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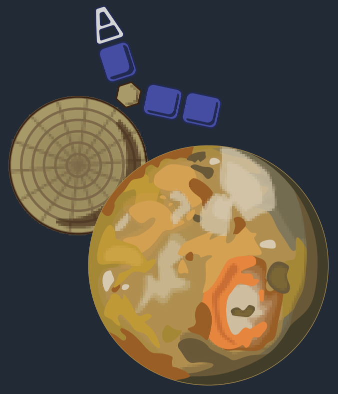
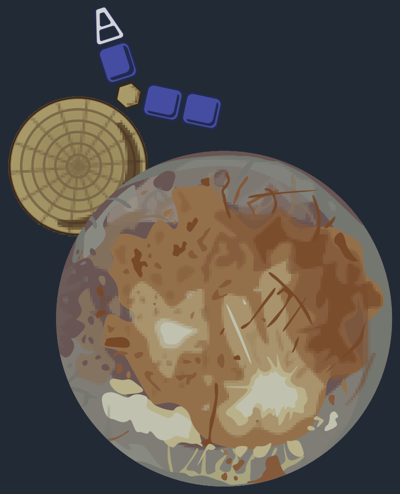
Concept of Operations

- Both spacecraft will perform end of life operations after (1) delivery of complete video coverage of all four Galilean moons to ground and (2) a minimum of 10 years has been spent in the Jovian system
- Both spacecraft perform maneuvers to enter trajectory to collide with two of the moons: GEC-1 with Callisto and GEC-2 with Io
- Both spacecraft continually transmit data back to Ground Station during de-orbit before collision with moons





Trajectory

Lindsey Langton and Johan Govaars

Mission Timeline

Trajectory – Overview

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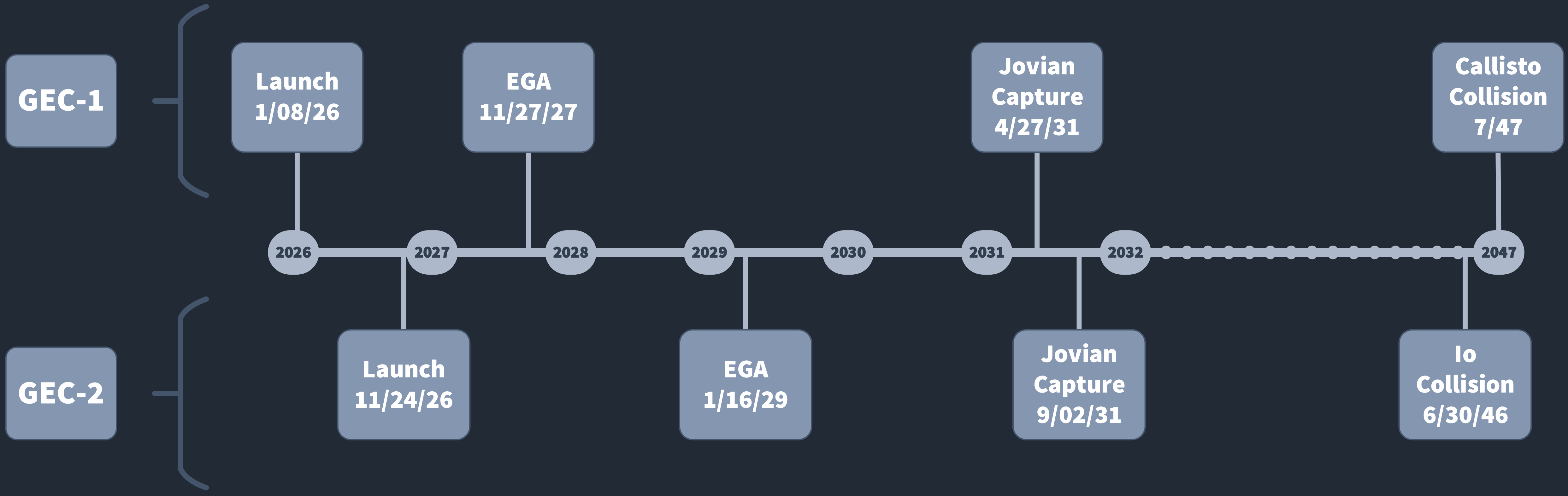

Interplanetary Transfer Methodology

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Trajectory: Interplanetary Transfer

- NASA Ames Trajectory Browser[1] used as starting point
  - Input desired launch year, max transfer time, max ΔV (injection and post-injection), and which to minimize (duration or ΔV)
  - Outputs calculated trajectories that fall within corresponding ΔV/time range
- Chose trajectories that fit requirements of mission and were most feasible
  - Started with details provided by browser and developed own trajectory model
  - Patched conics calculations and Lambert's solver
- Current Assumptions:
  - 2-Body System
    - N-body modeled but not found to be significant
  - Interplanetary propagation incorporates solar radiation pressure only
    - Drag and oblateness not significant for trajectories near Earth

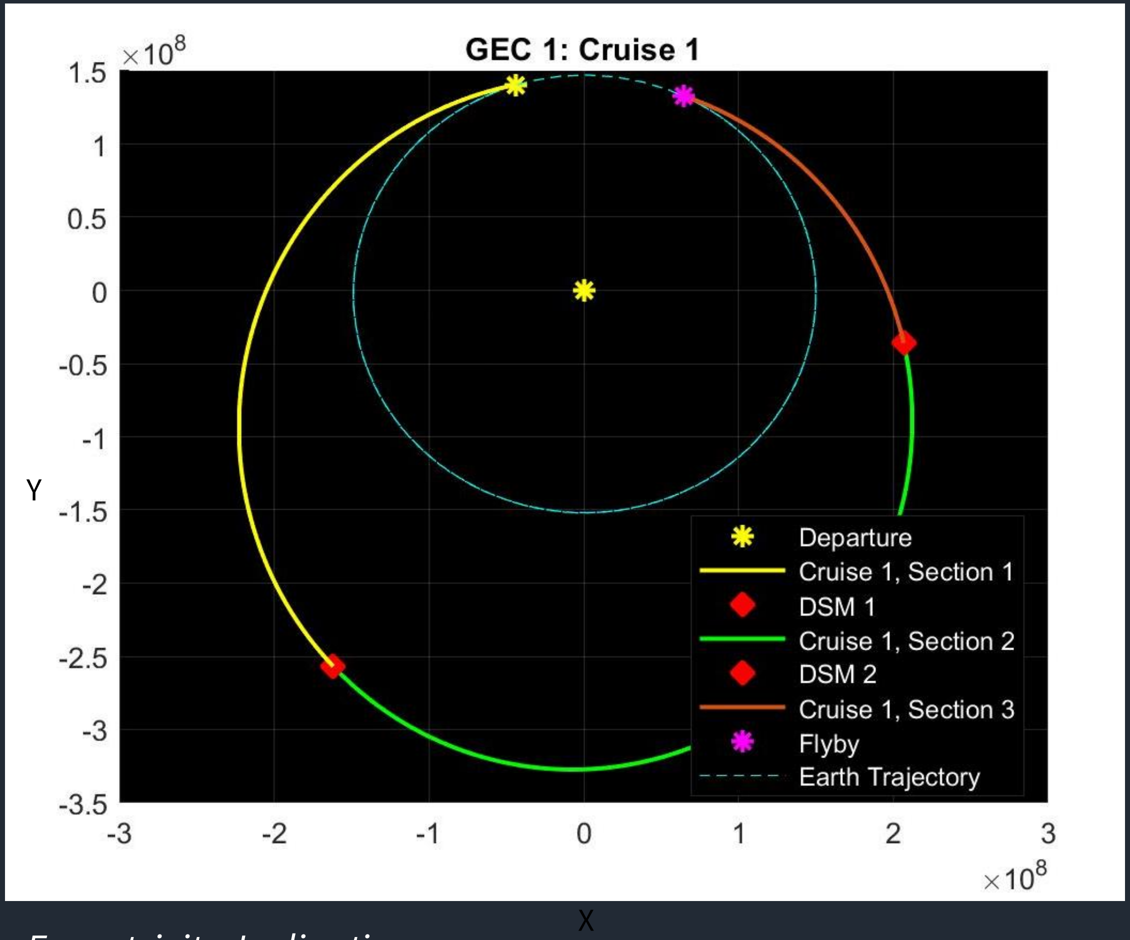
Cruise 1: GEC-1

Trajectory: Interplanetary Transfer

Maneuver	Start Date	Duration	ΔV [m/s]
Earth Departure	01/08/2026	N/A	N/A
Cruise Phase 1 – Section 1	01/08/2026	229 days	0
DSM 1	08/25/2026	35 seconds	6.7
Cruise Phase 1 – Section 2	08/25/2026	372 days	0
DSM 2	09/02/2027	42 seconds	8.1
Cruise Phase 1 – Section 3	09/02/2027	86 days	0
Total		**687 days** **1.9 years**	**14.8**

Note 1: 2 DSMs are required to counteract effects SRP has COEs

Note 2: Area to mass ratio assumed for SRP calculation



Eccentricity: Inclination:

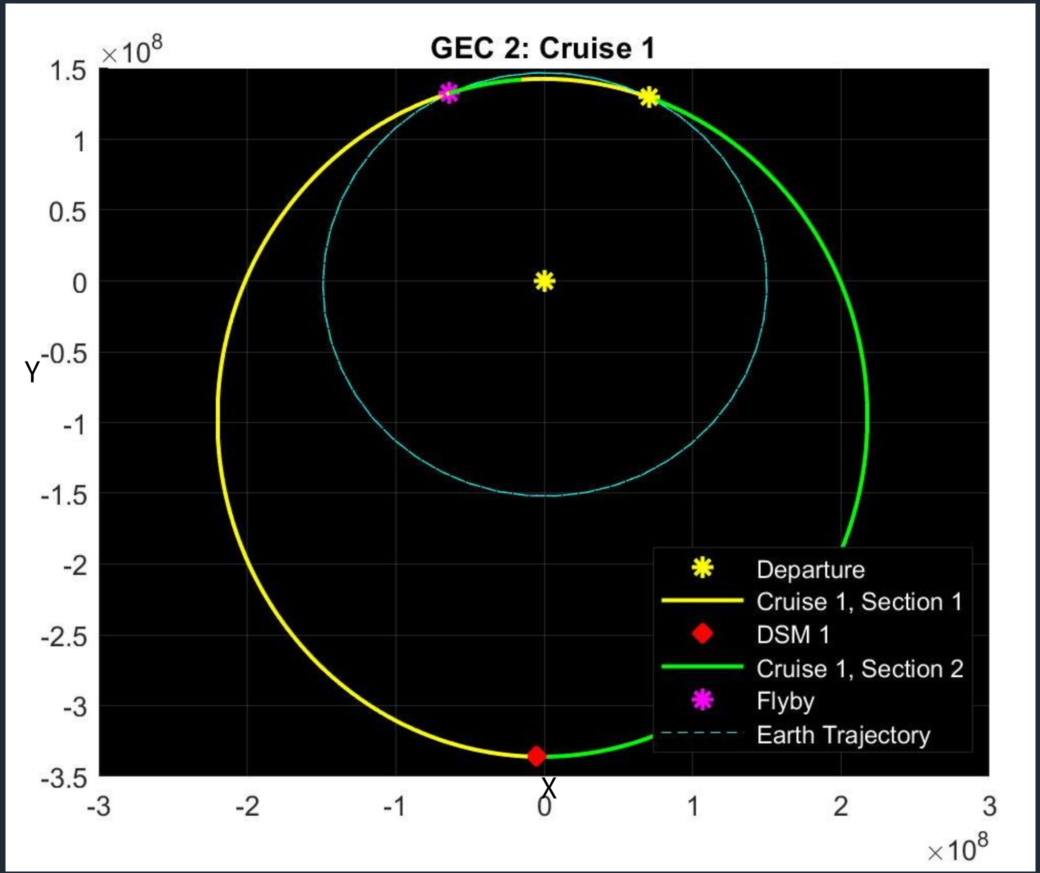
Cruise 1: GEC-2

Trajectory: Interplanetary Transfer

Maneuver	Start Date	Duration	ΔV [m/s]
Earth Departure	11/24/2026	N/A	N/A
Cruise Phase 1 – Section 1	11/24/2026	431 days	N/A
DSM 1	01/29/2028	26 seconds	5.01
Cruise Phase 1 – Section 2	01/29/2028	353 days	N/A
Total		**784 days 2.1 years**	**5.01**

Note 1: Only 1 DSM is required due to differing SRP levels on GEC-2 trajectory

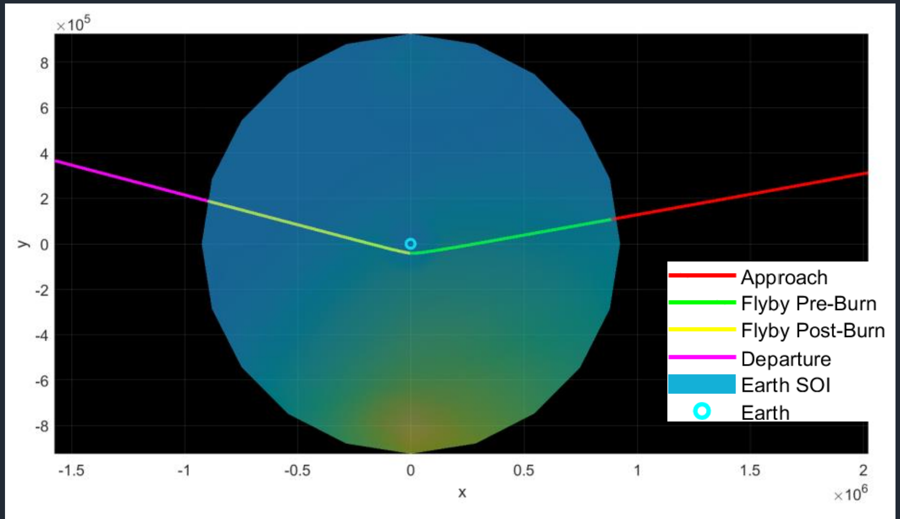
Note 2: Area to mass ratio assumed for SRP calculation

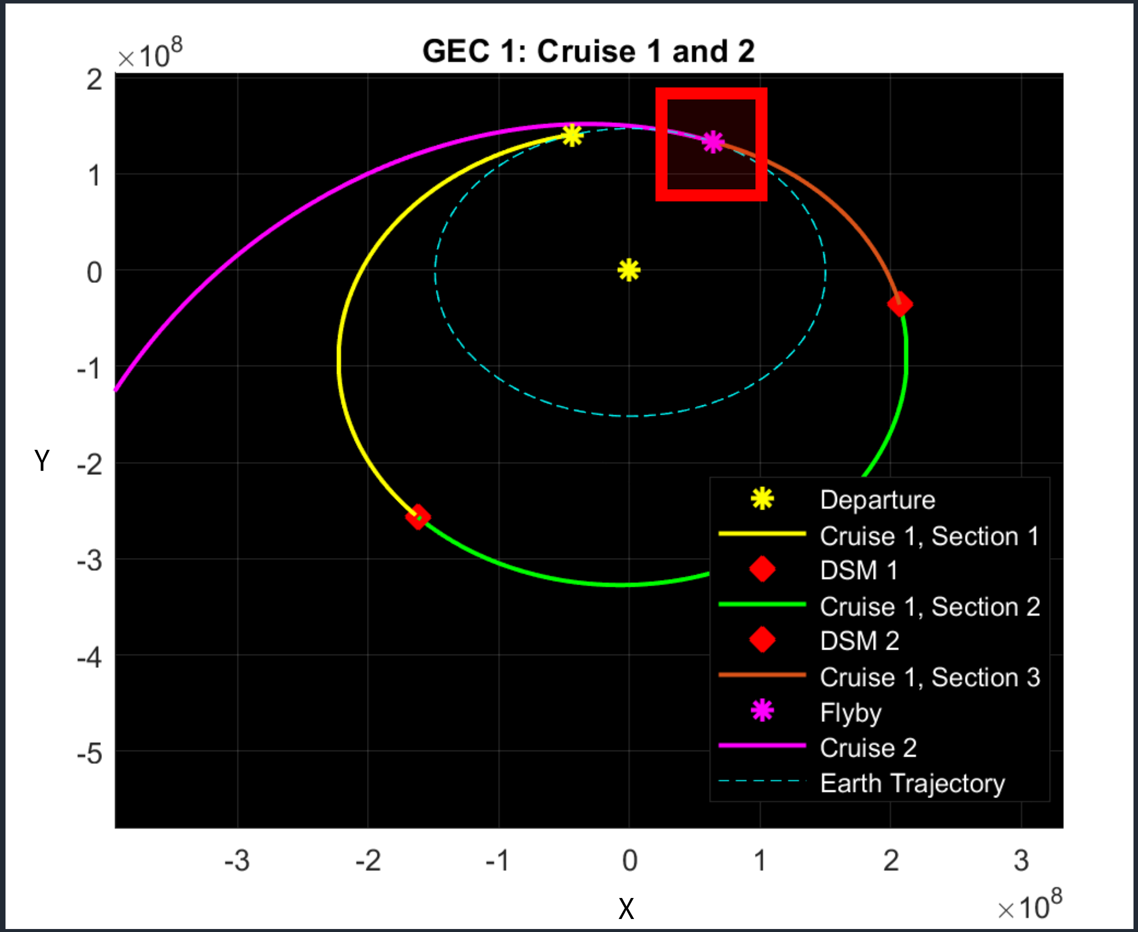


Eccentricity: Inclination:

Gravity Assist: GEC-1

Trajectory: Interplanetary transfer





Trailing-Edge, Powered Earth Flyby Summary		
Maneuver	Date	Heliocentric Speed [km/s]
Entrance into Earth's SOI	11/24/2027	35.1
Closest Earth Approach (4.09E4 km from Earth)	11/26/2027	37.0
Exit from Earth's SOI	11/28/2027	39.0

ΔV for Burn at Closest Approach - 0.25 [km/s]

Gravity Assist: GEC-2

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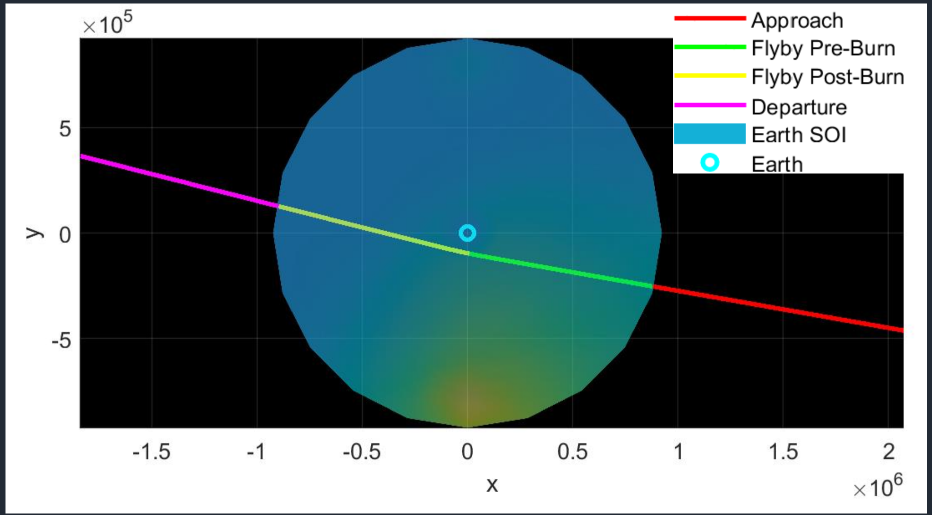
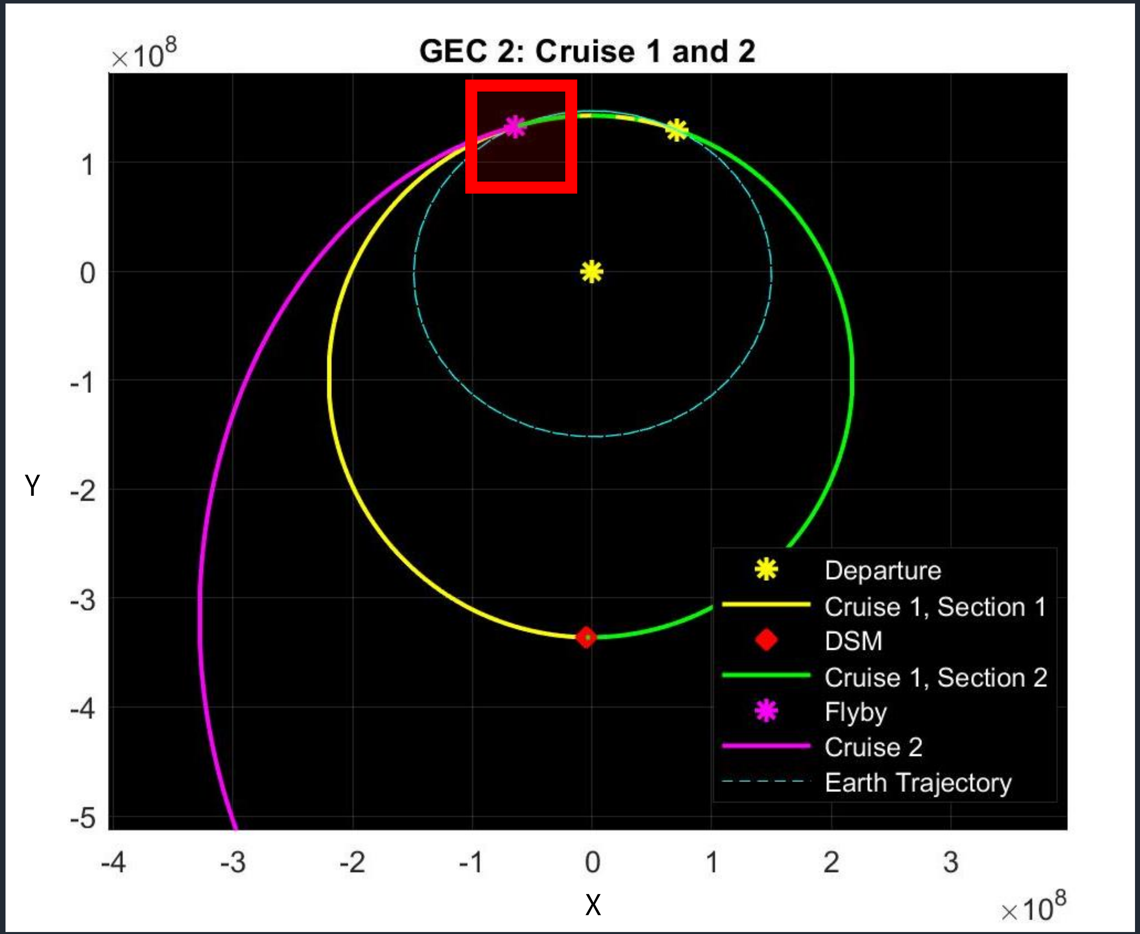
Trajectory: Interplanetary transfer





Trailing-Edge, Powered Earth Flyby Summary		
Maneuver	Date	Heliocentric Speed [km/s]
Entrance into Earth's SOI	01/14/2029	35.4
Closest Earth Approach (1.28E5 km from Earth)	01/16/2029	36.0
Exit from Earth's SOI	01/17/2029	38.9

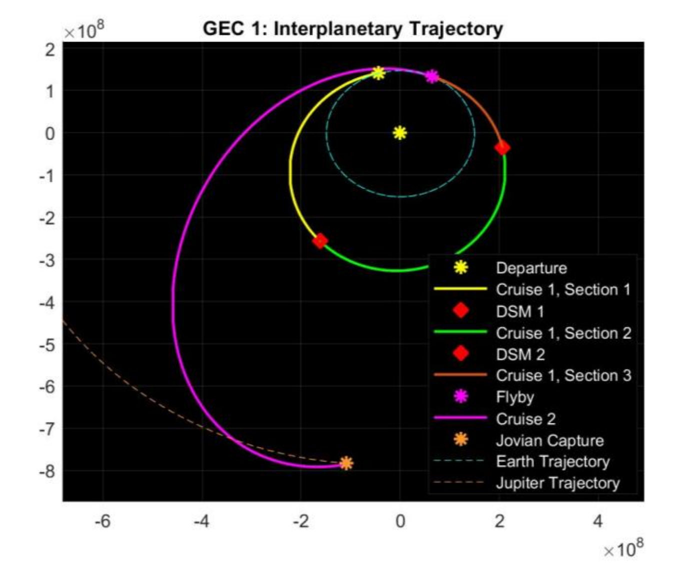
ΔV for Burn at Closest Approach - 0.39 [km/s]

Cruise 2: GEC-1 and GEC-2

Trajectory: Interplanetary transfer

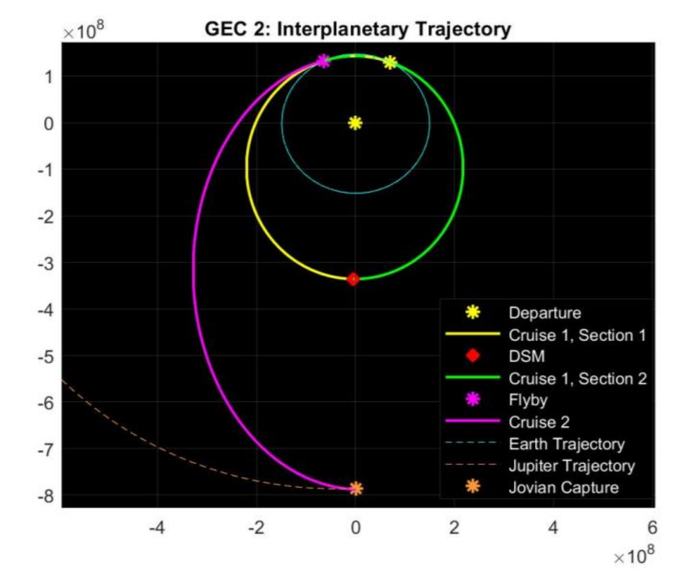
GEC-1			
Maneuver	Start Date	Duration [years]	ΔV [m/s]
Cruise Phase 2	11/28/2027	3.41	N/A
Jovian SOI Arrival	04/27/31	Instantaneous	N/A
Total		**3.41**	**0**

Eccentricity: 0.69468
Inclination: 1.3356°



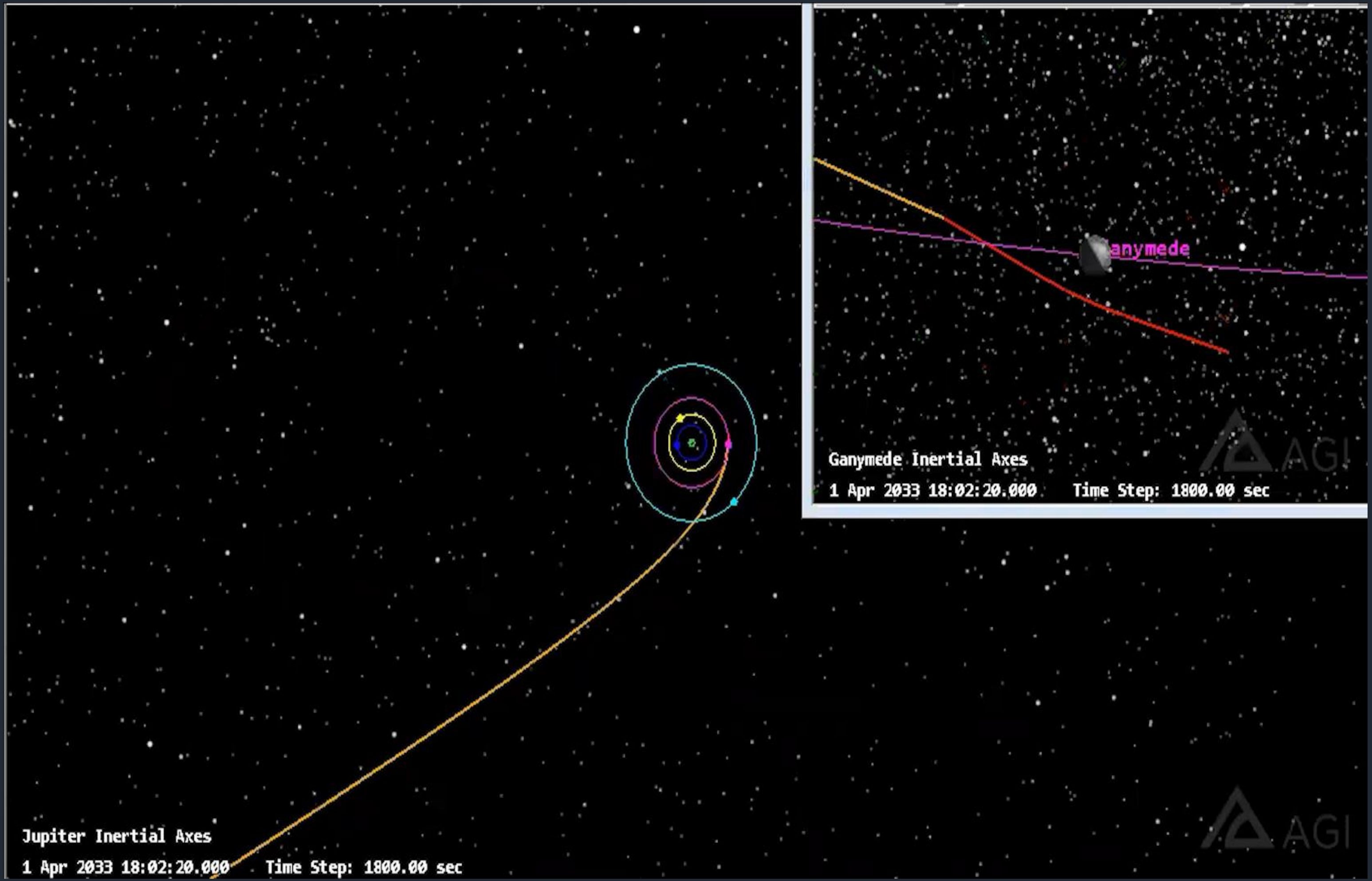
GEC-2			
Maneuver	Start Date	Duration [years]	ΔV [m/s]
Cruise Phase 2	01/17/2029	2.6	N/A
Jovian SOI Arrival	09/02/2031	Instantaneous	N/A
Total		**2.6**	**0**

Eccentricity: 0.695
Inclination: 0.55778°



Jovian System Entry: GEC-1

Trajectory: Interplanetary transfer



GEC-1 firs encounters Ganymede on May 1st, 2031. Final around Ganymede in an 875 km x 875 km orbit occurs on August 3, 2036

ΔV: 2.86 km/s

- 1.3 km/s for initial Jovian system entry, burn time : 85 minutes
- 0.160 m/s for 7 retargeting burns over the course of 22 leading-edge flyby gravity assists to decrease spacecraft energy
- 1.4 km/s for final Ganymede capture, burn time: 56 minutes

Total time from Jovian system entry to Ganymede capture: 5 years, 3 months

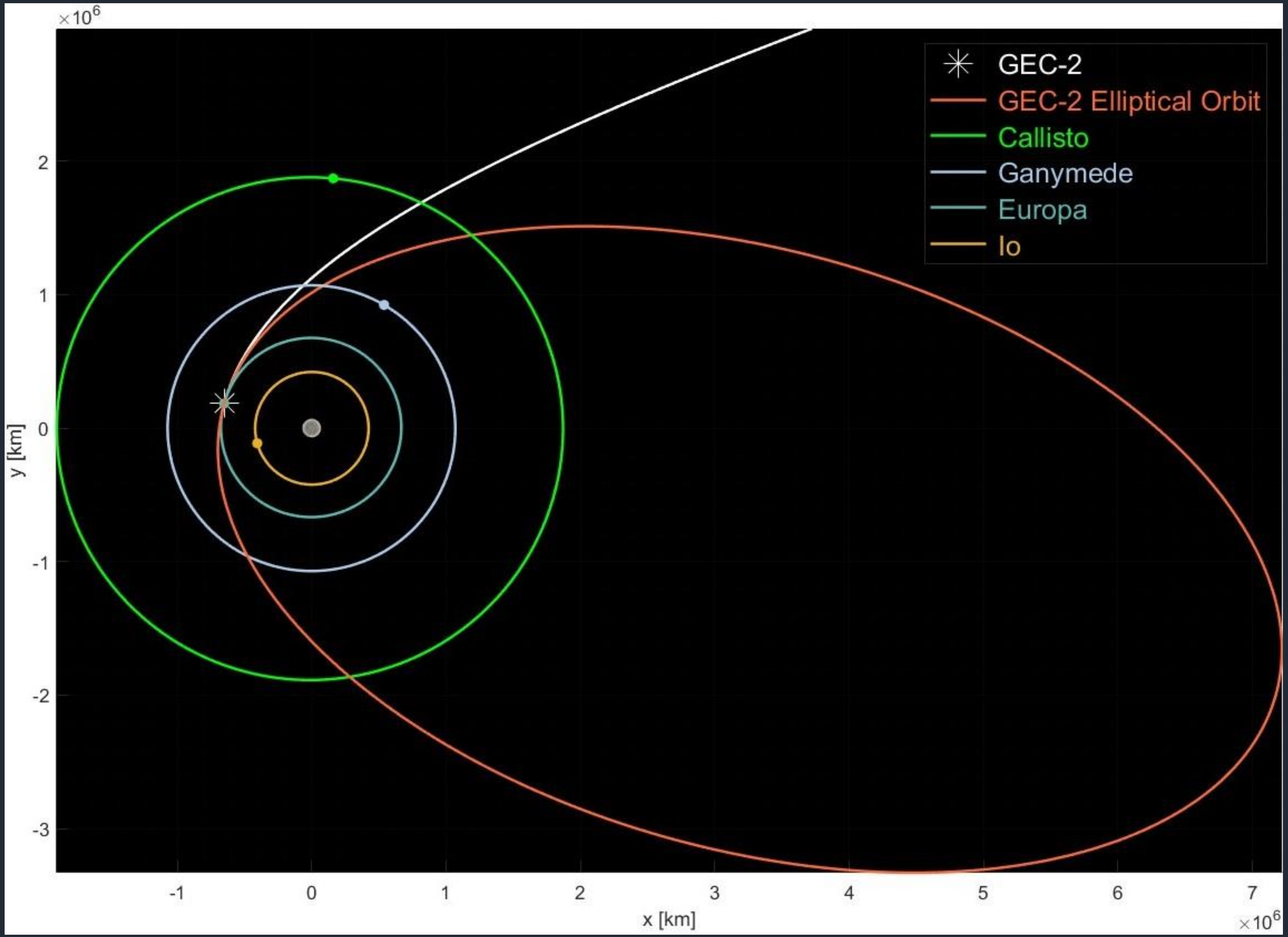
Jovian System Entry: GEC-2

Trajectory: Interplanetary transfer

GEC-2 arrives at Jupiter on September 2nd, 2031

ΔV: 1.68 km/s

- Burns into a highly elliptical orbit, 68,000 km x 7,500,000 km, around Jupiter with periapsis encountering Europa
- Burn time: 98 minutes



Summary Table: GEC-1

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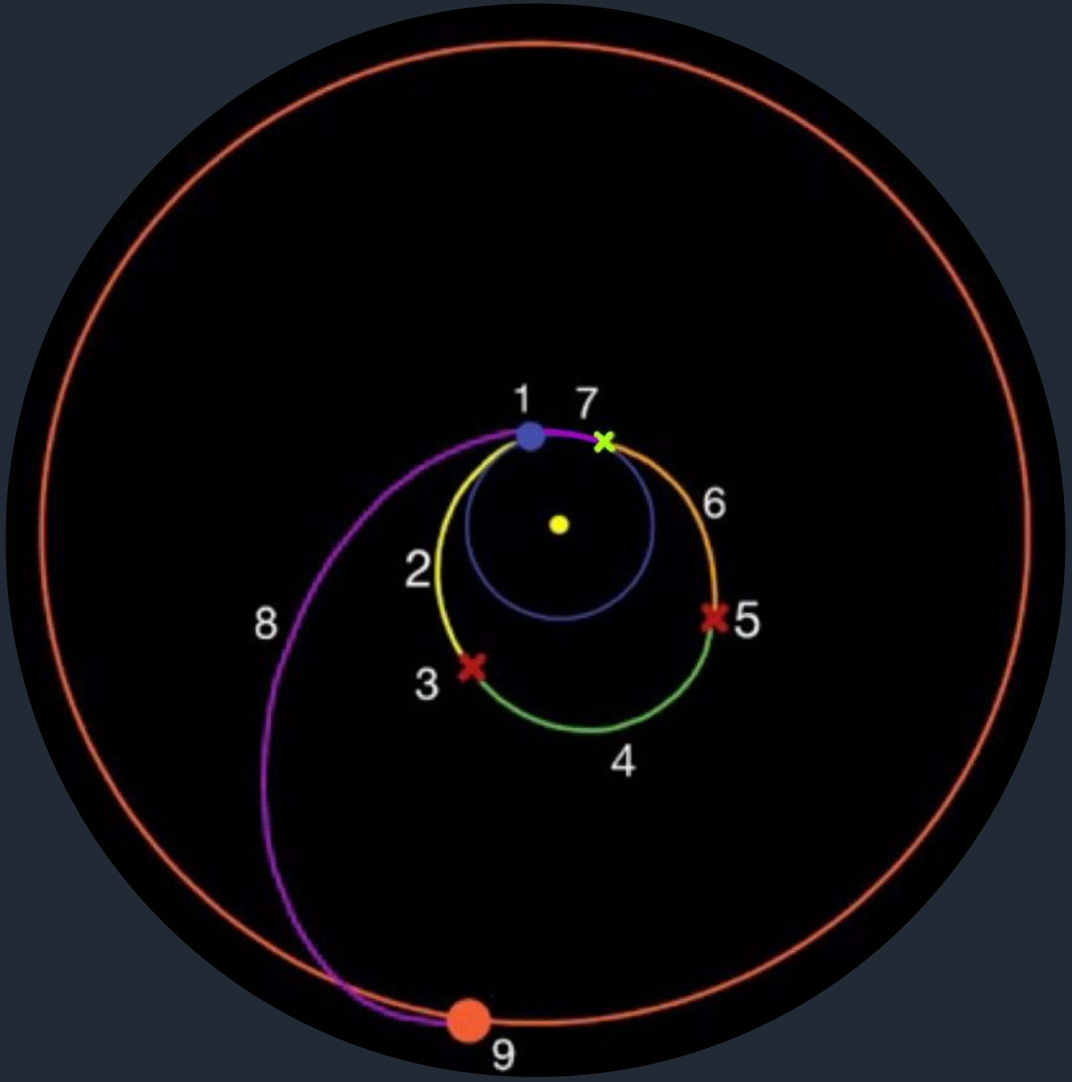
Trajectory: Interplanetary transfer

#	Maneuver	Start Date	Duration	ΔV [km/s]
1	Earth Departure	01/08/26	minutes	4.8 (LV)
2	Cruise Phase 1 – Section 1	01/08/26	229 days	0
3	DSM 1	8/25/26	35 seconds	0.007
4	Cruise Phase 1 – Section 2	8/25/26	1.02 years	0
5	DSM 2	9/02/27	42 seconds	0.008
6	Cruise Phase 1 – Section 3	9/02/27	86 days	0
7	Earth Gravity Assist	11/24/27	4 days	0.248
8	Cruise Phase 2	11/28/27	3.41 years	0
9	Jupiter SOI Entry - Ganymede Capture	04/27/31	5.26 years	2.86
Total			**10.56 years**	**4.8 (LV) + 3.12(S/C)**

Summary Figure: GEC-1

Trajectory: Interplanetary transfer



- Earth at departure
- Jupiter at rendezvous
- Burn locations
- Earth Gravity Assist

Summary Table: GEC-2

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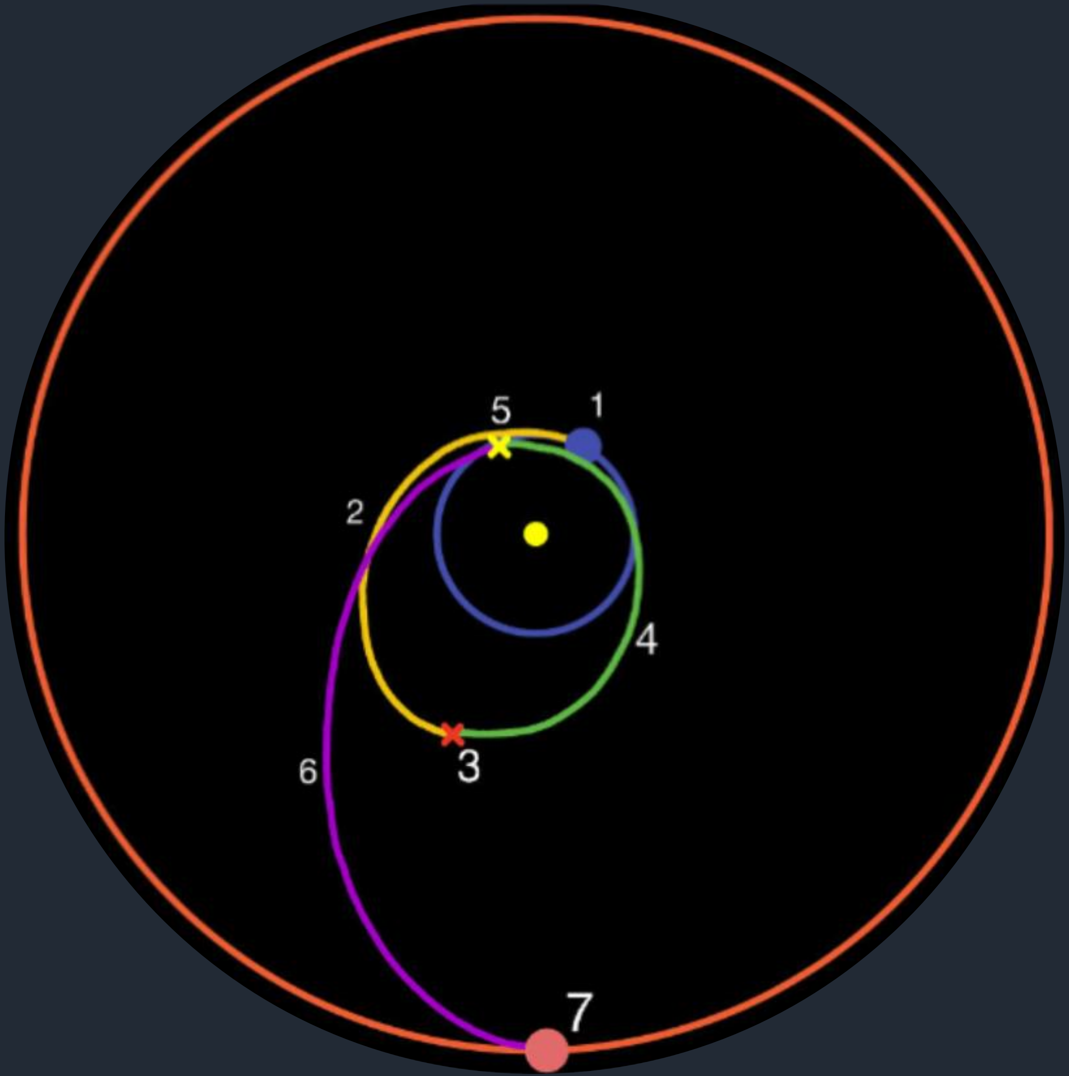
Trajectory: Interplanetary transfer

#	Maneuver	Start Date	Duration	ΔV [km/s]
1	Earth Departure	11/24/2026	minutes	4.88 (LV)
2	Cruise Phase 1 – Section 1	11/24/2026	431 days	0
3	DSM 1	01/29/2028	26 seconds	5.01e-3
4	Cruise Phase 1 – Section 2	01/29/2028	353 days	0
5	Earth Gravity Assist	01/16/2029	3 days	0.394
6	Cruise Phase 2	01/17/2029	2.62 years	0
7	Jovian Capture	09/02/2031	98 minutes	1.67
Total			**4.8 years**	**4.88 (LV) + 2.07(S/C)**

Summary Figure: GEC-2

Trajectory: Interplanetary transfer



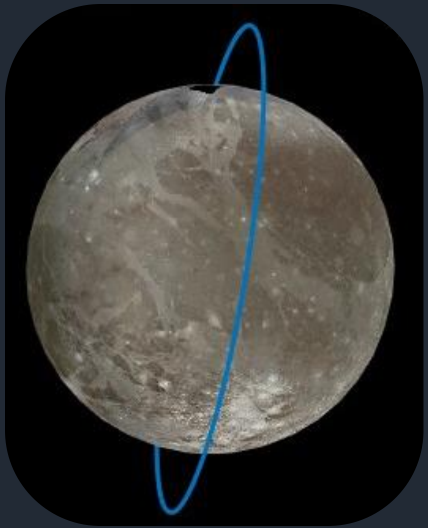
- Earth at departure
- Jupiter at rendezvous
- Burn locations
- Earth Gravity Assist

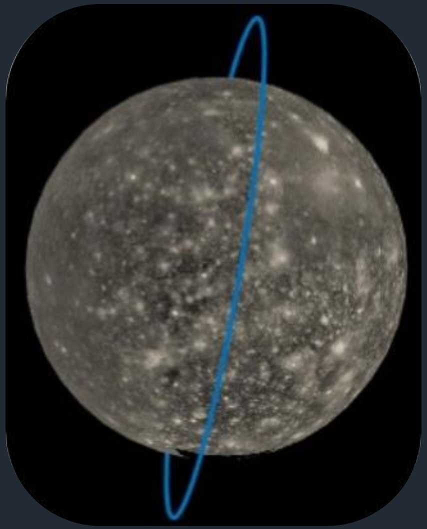
Jovian Operation Overview

Trajectory: Jovian Operation

GEC-1

- Direct orbits of Ganymede and Callisto
- Coverage estimates
- Jupiter's perturbational effects



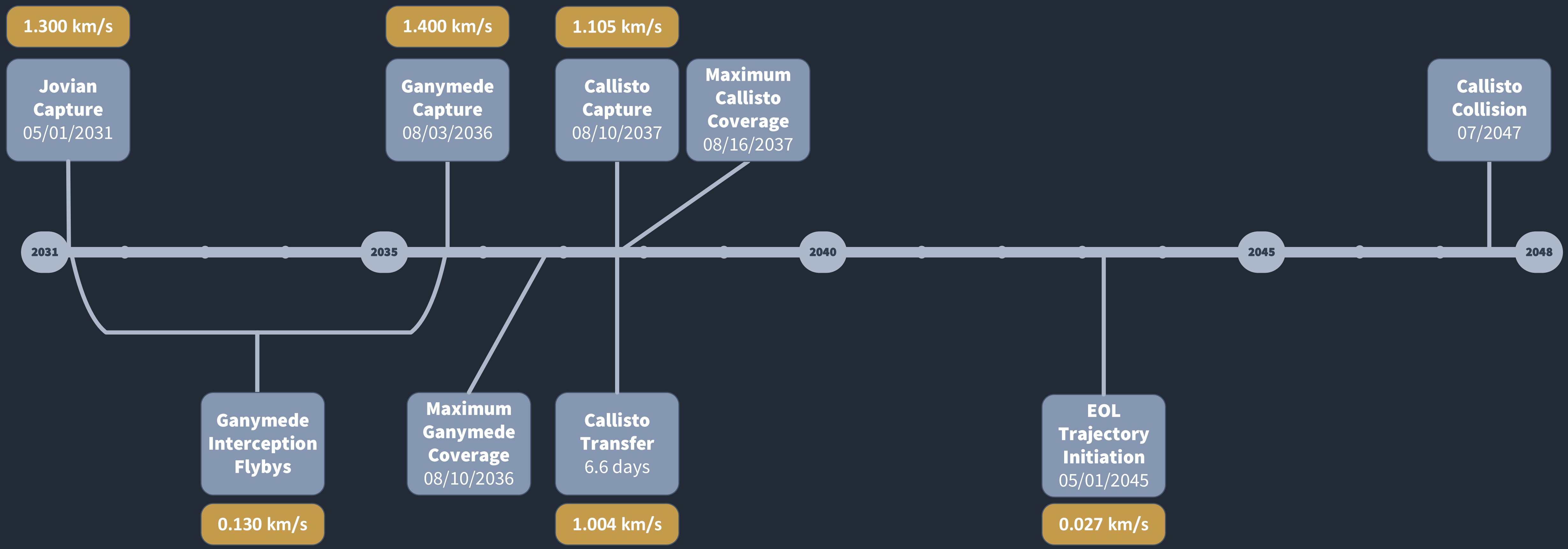


GEC-2

- Imaging flybys of Europa and Io
- Retargeting costs and optimization
- Coverage estimates

Jovian Operation Overview: GEC-1

Trajectory: Jovian Operation

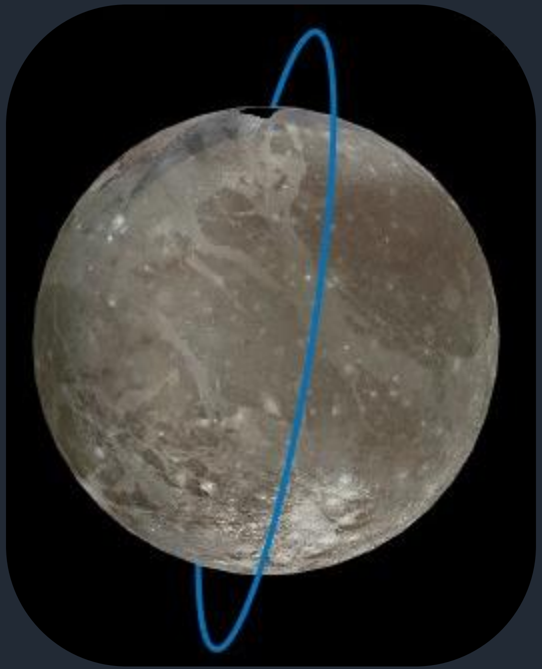


Jovian Operations: GEC-1

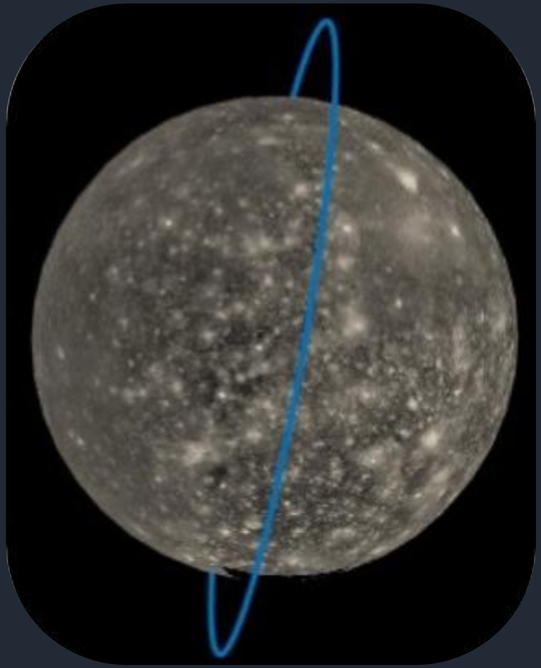
Trajectory: Jovian Operation

- Nominal operational orbits are 875 km altitude circular orbits at an 80° inclination
- Ganymede Period: 3.64 hr
- Callisto Period: 3.88 hr

Orbit about Ganymede

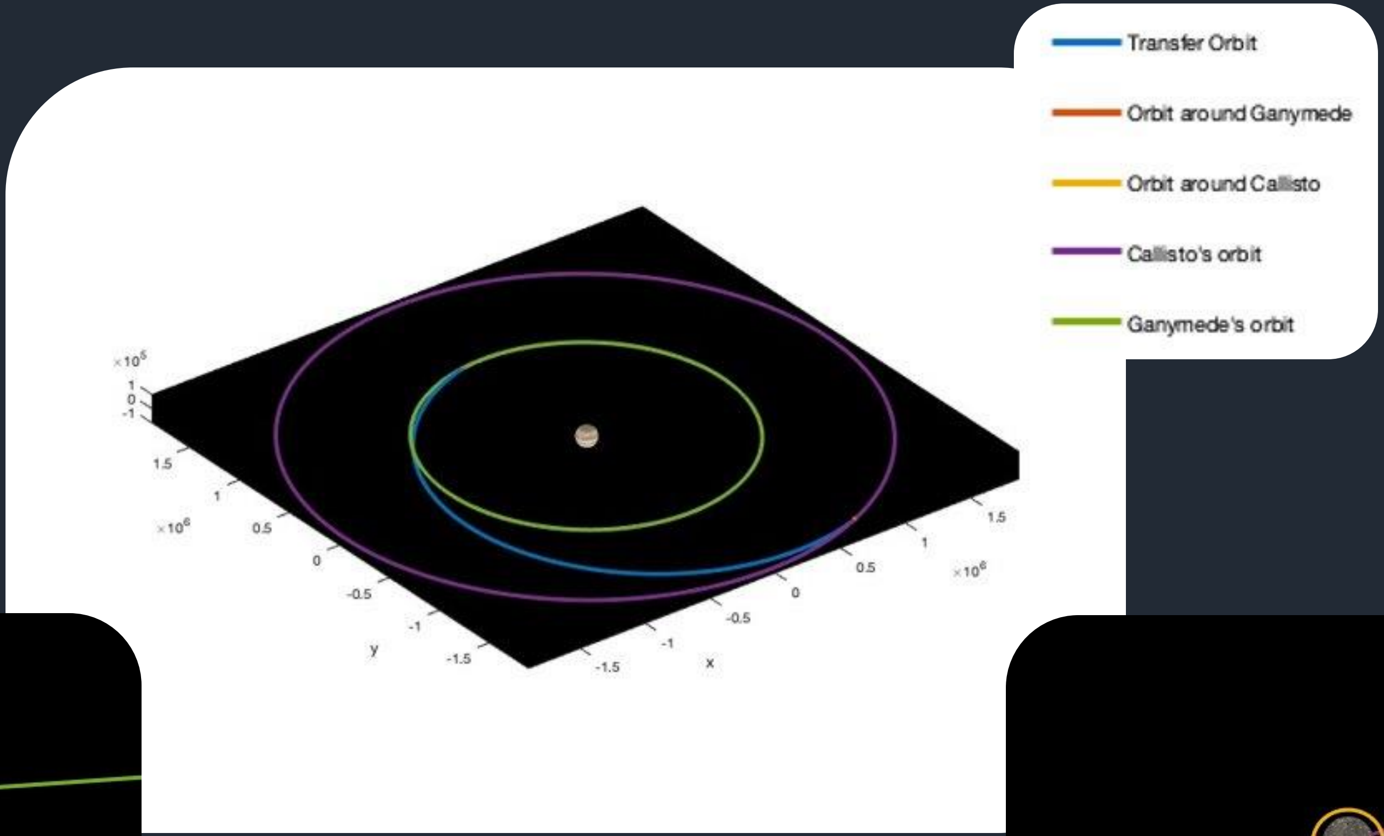


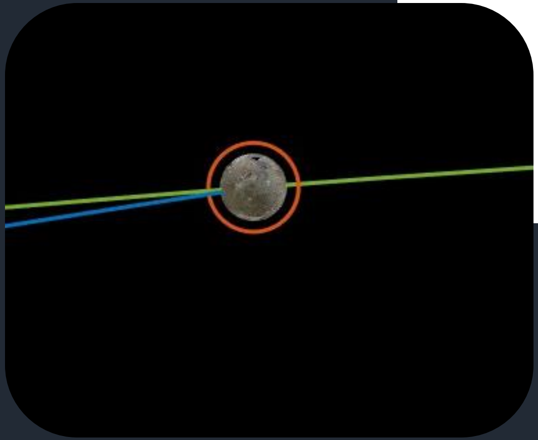
Orbit about Callisto



Transfer: GEC-1

Trajectory: Jovian Operation



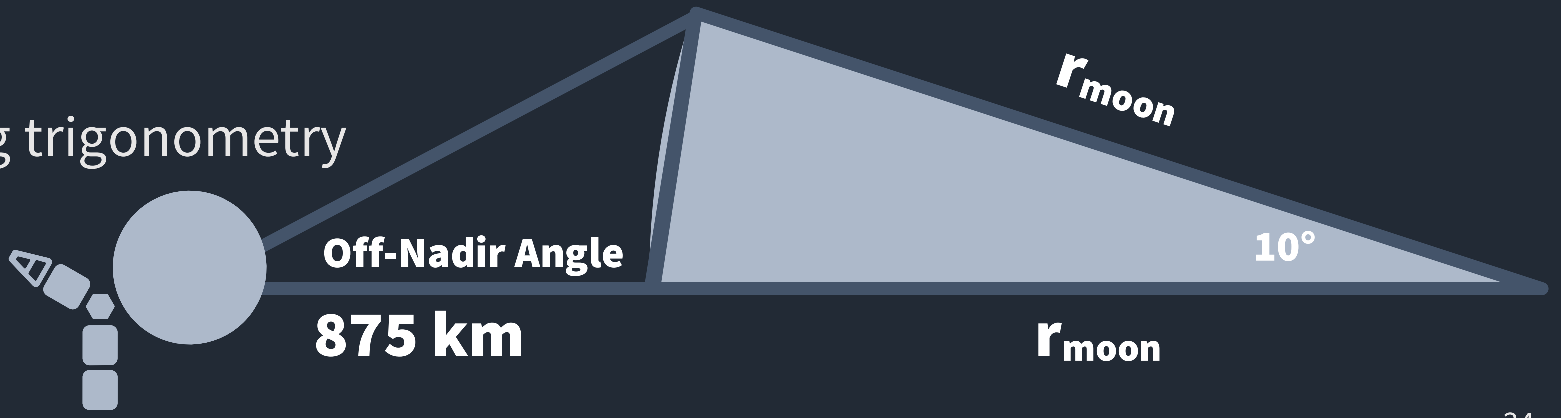


Galilean Moon Coverage: GEC-1

Trajectory: Jovian Operation

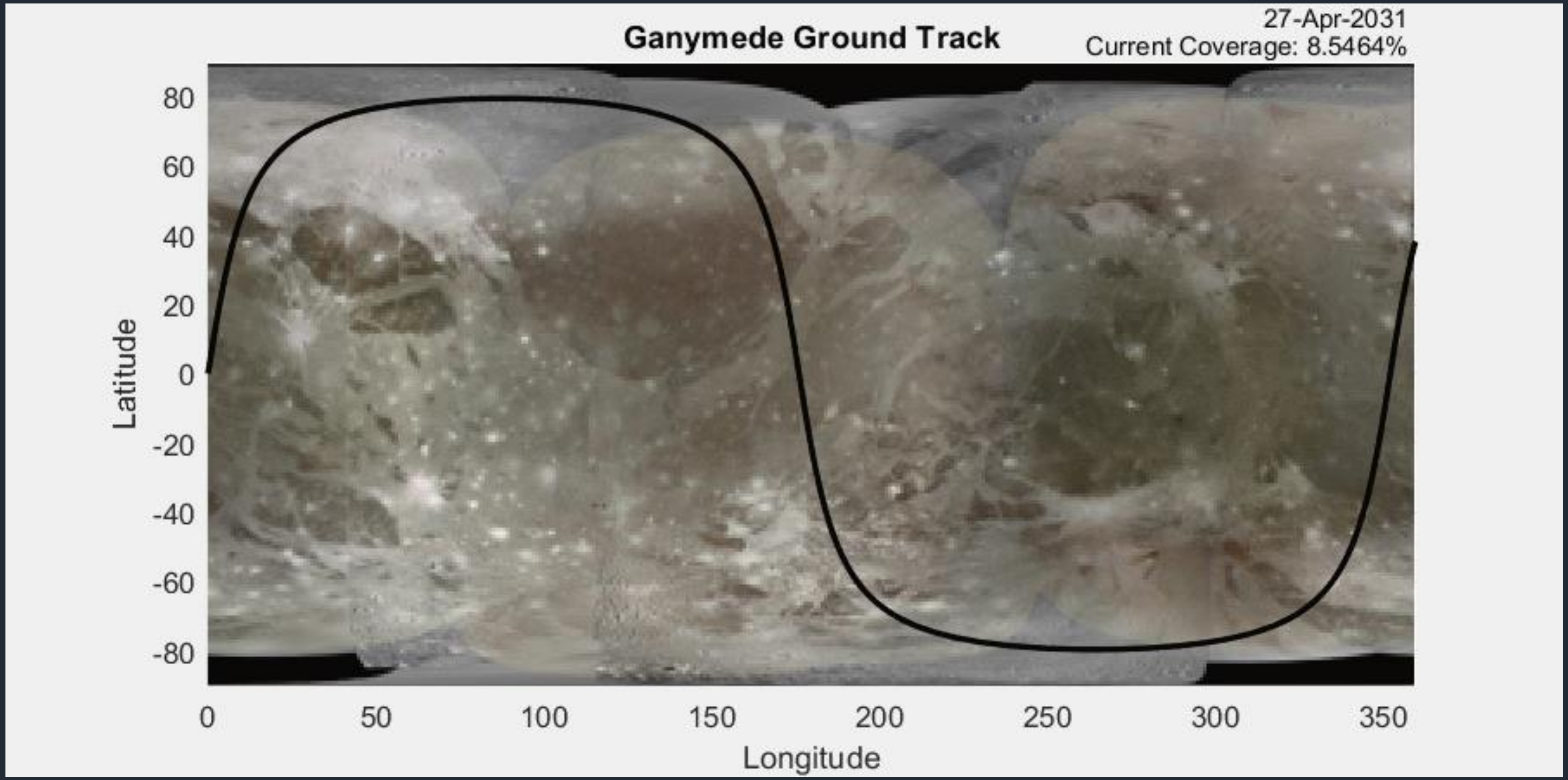
- 80° orbital inclination allows views of ± 80.2° latitude while pointing nadir
- Total coverage is possible (± 90°) with off-nadir pointing
  - Downside is slightly lower quality images
  - Off-nadir passes are saved until the end of the time around each moon to prioritize the higher quality images for transmission

- Off-Nadir pointing angle:
  - Ganymede: 27.5 °
  - Callisto: 25.6 °
  - First-order estimate using trigonometry



Ganymede Coverage: GEC-1

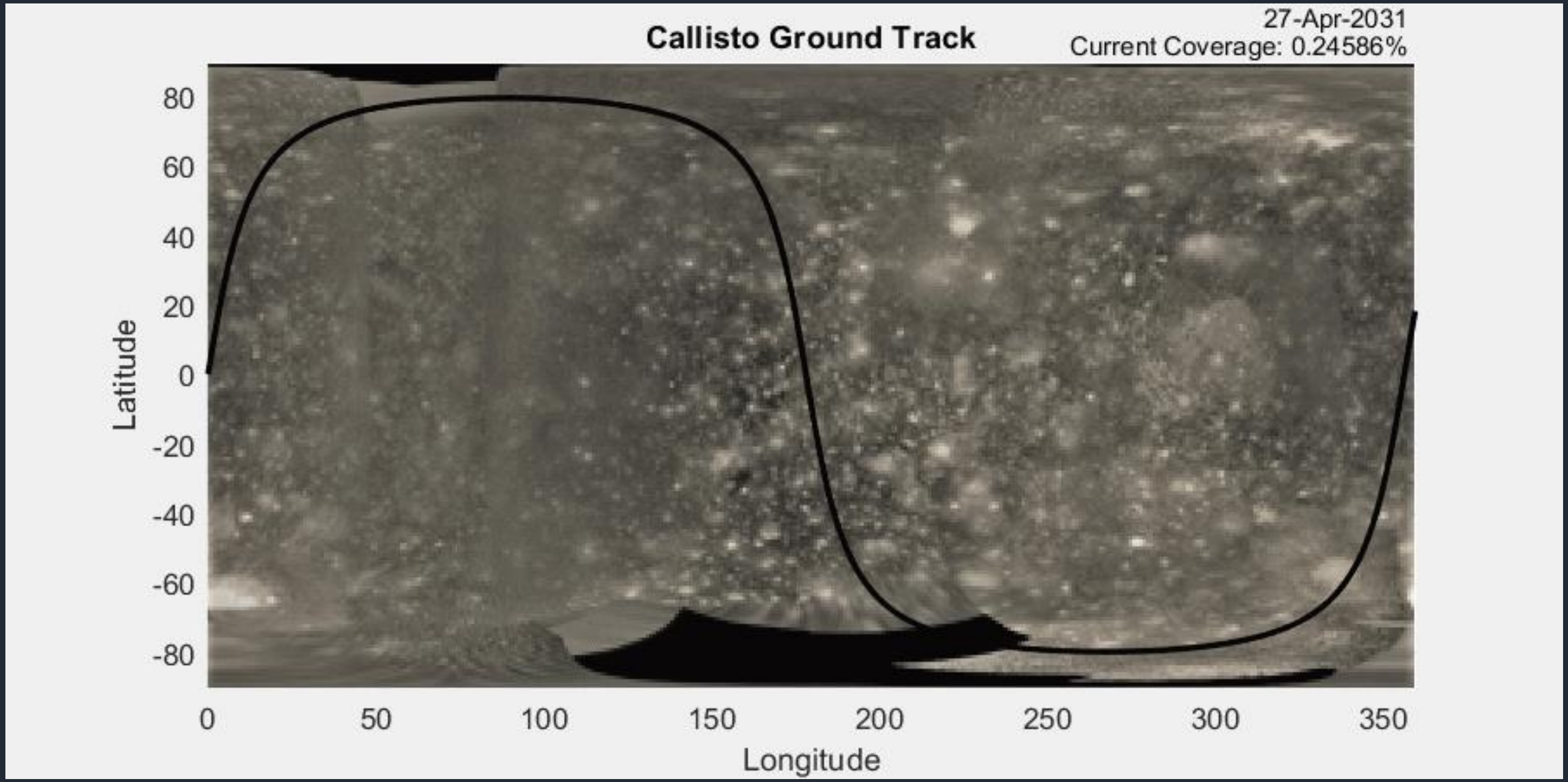
Trajectory: Jovian Operation



Total expected coverage: 100%

Callisto Coverage: GEC-1

Trajectory: Jovian Operation

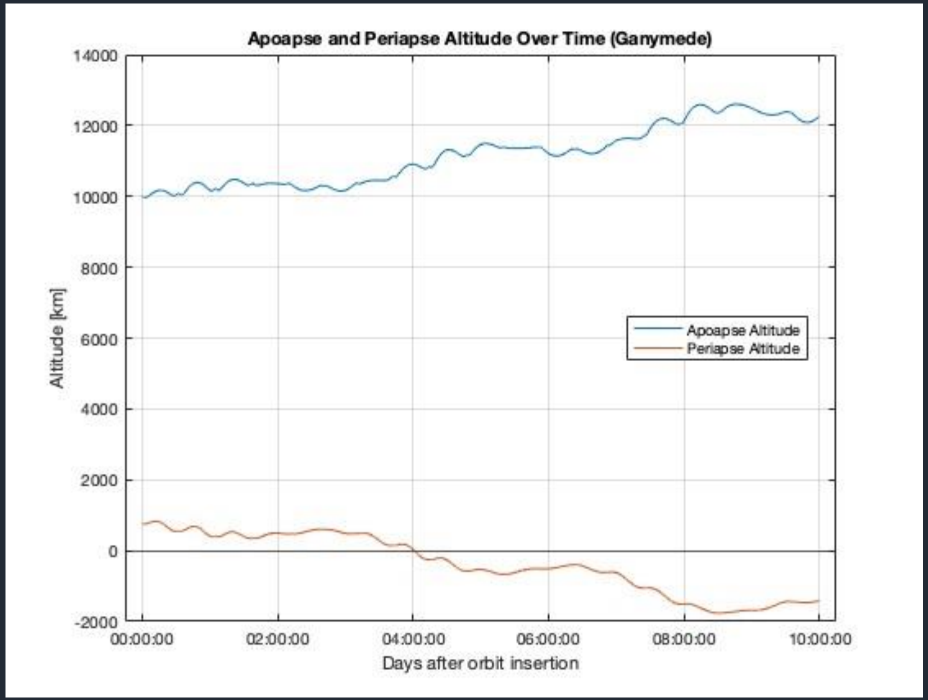


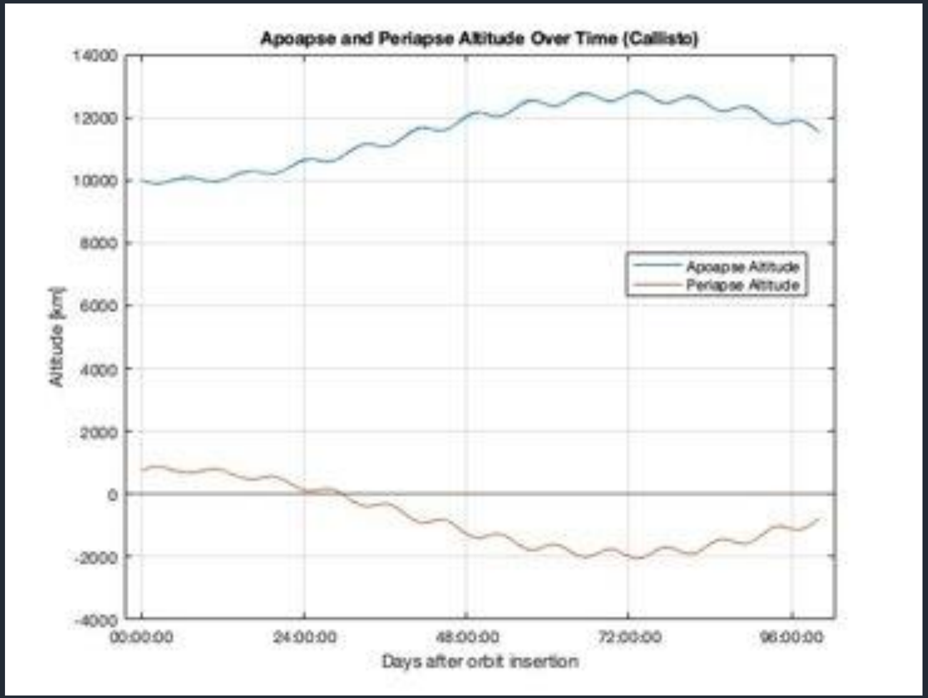
Total expected coverage:100%

Perturbations: GEC-1

Trajectory: Jovian Operation

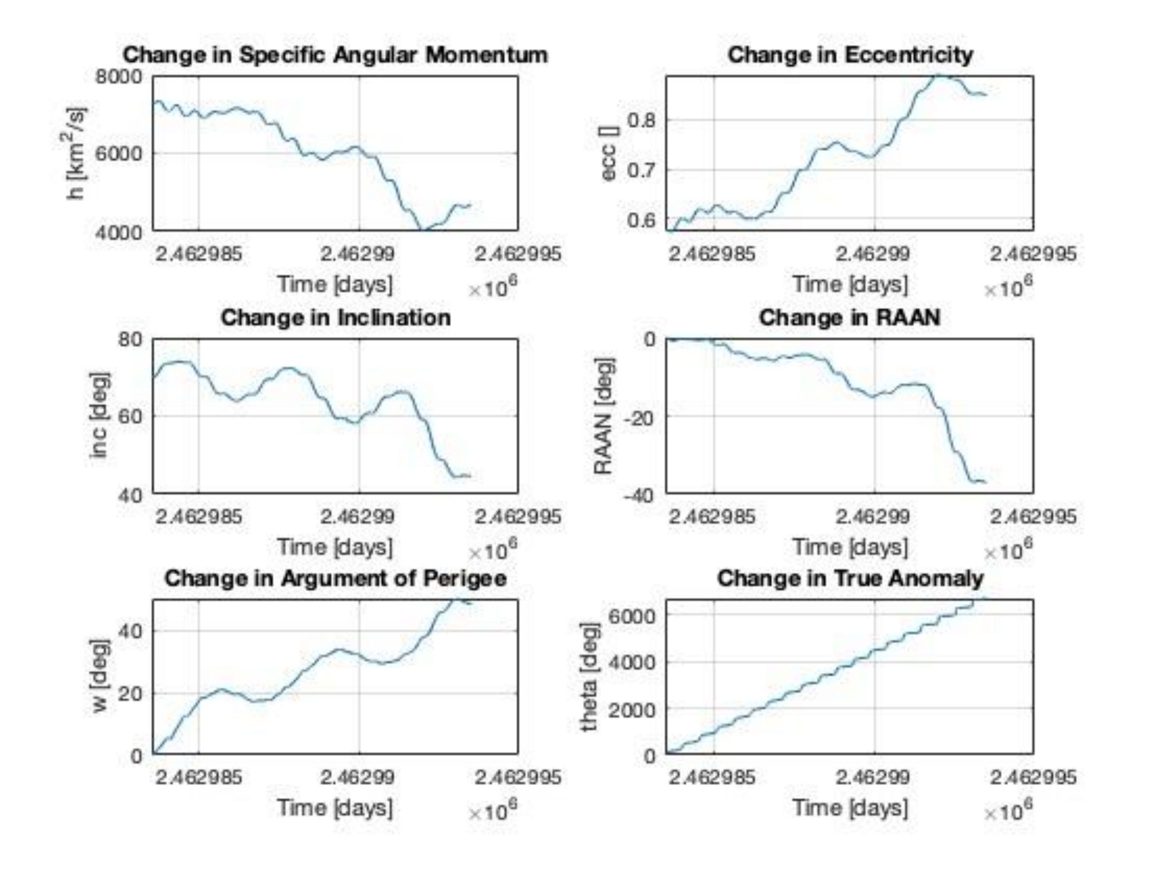
- Jupiter is by far the most influential perturbing body in the solar system, especially when operating around one of its secondary bodies within < $26*R_J$
    - One of the considered orbits around Ganymede (750x10,000 km) was so perturbed that it crashed into the moon after only 4 days
    - Callisto had a similar issue, causing GEC-1 to crash into the moon after 30 days



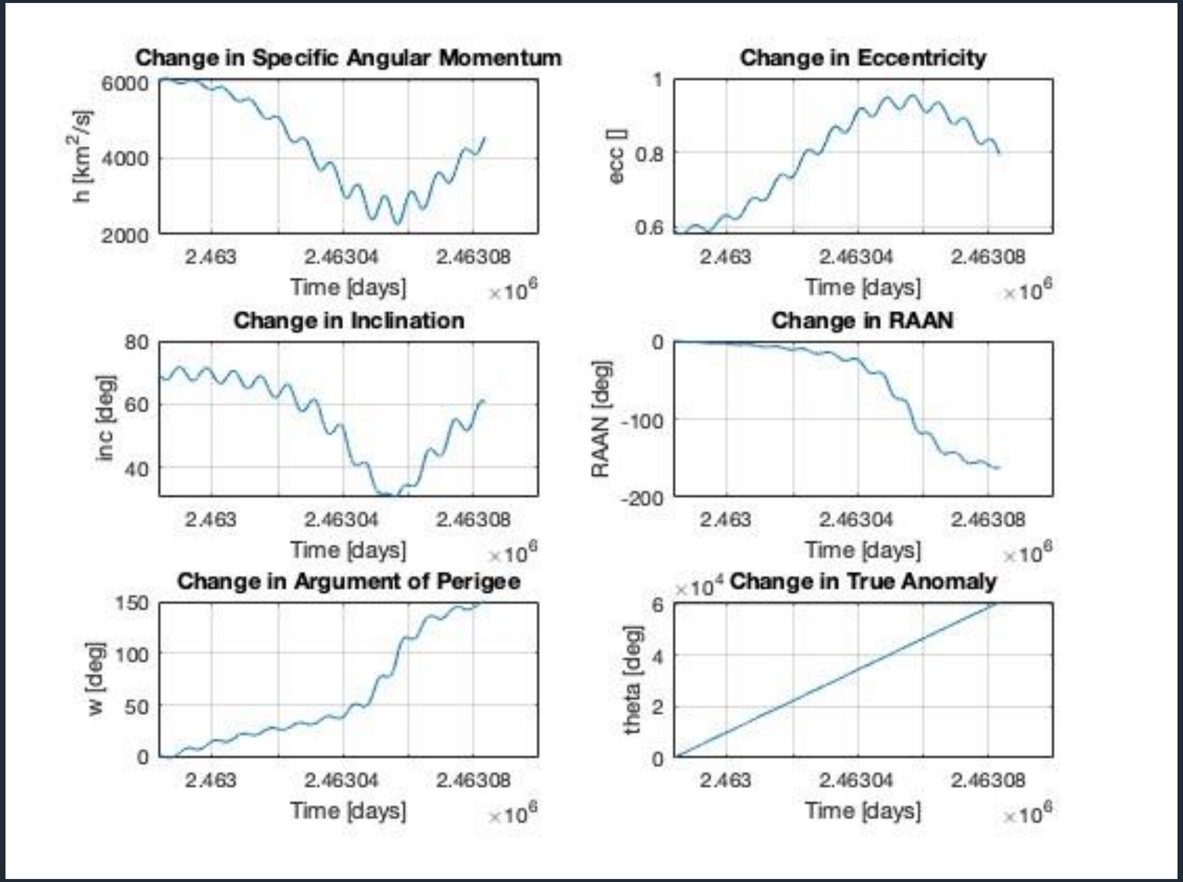


Perturbations: GEC-1

Trajectory: Jovian Operation



Ganymede

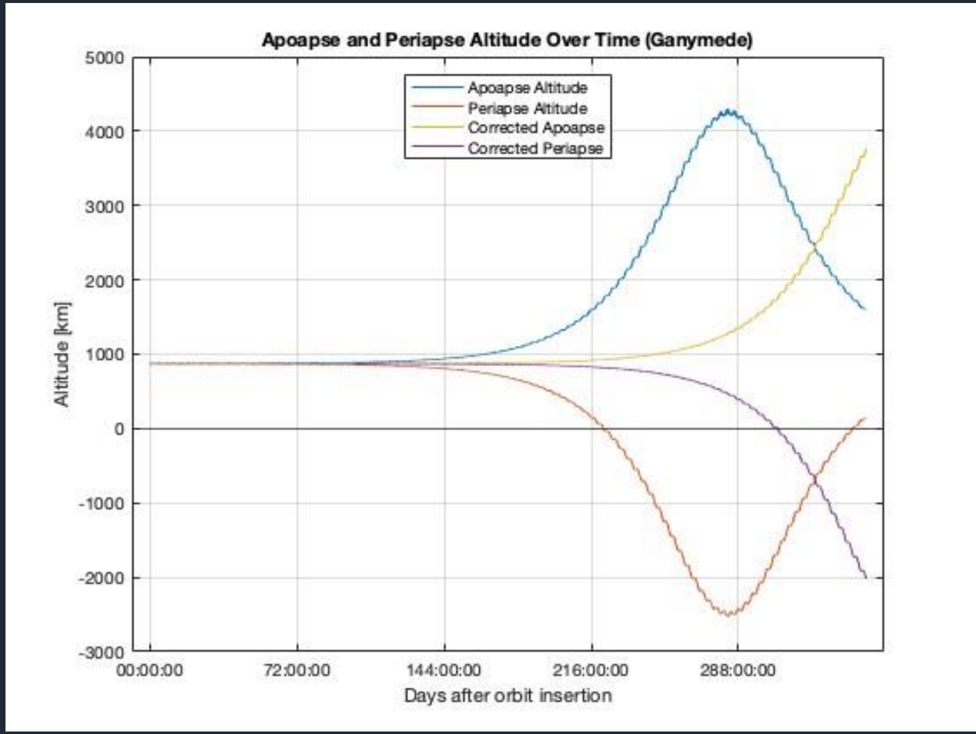


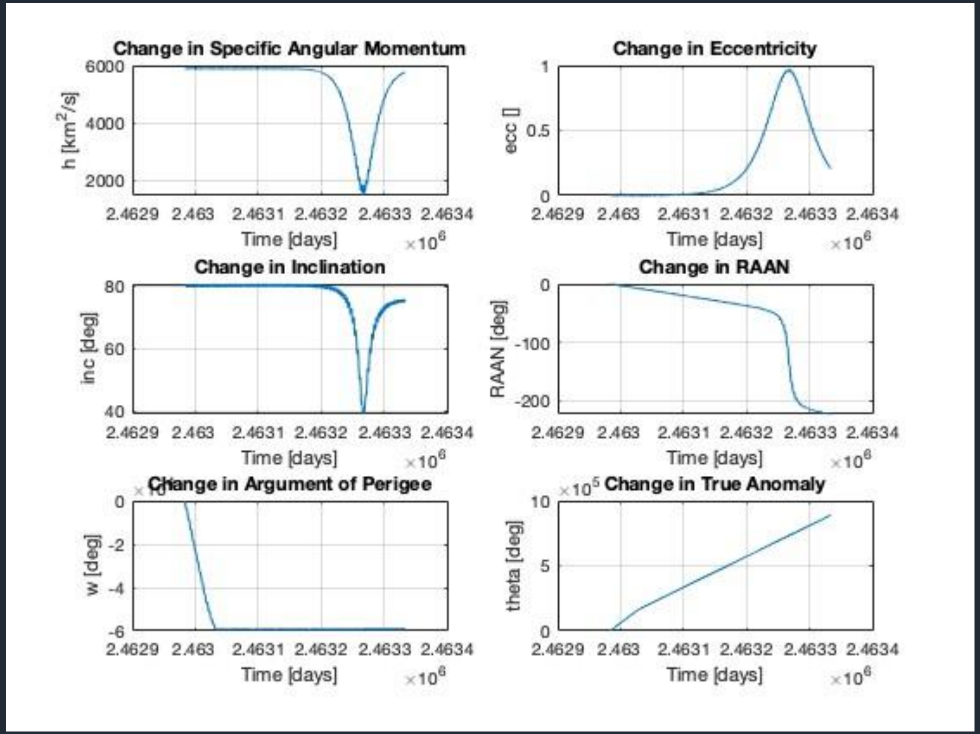
Callisto

Perturbations: GEC-1

Trajectory: Jovian Operation

- The final orbits (875 km altitude circular orbit) were chosen to minimize the n-body effects as much as possible. This is due to a much less asymmetric gravitational pull by Jupiter
- Fixes the problem of crashing prematurely on Callisto
- Ganymede still requires corrections every 100 days of about 4.6 m/s

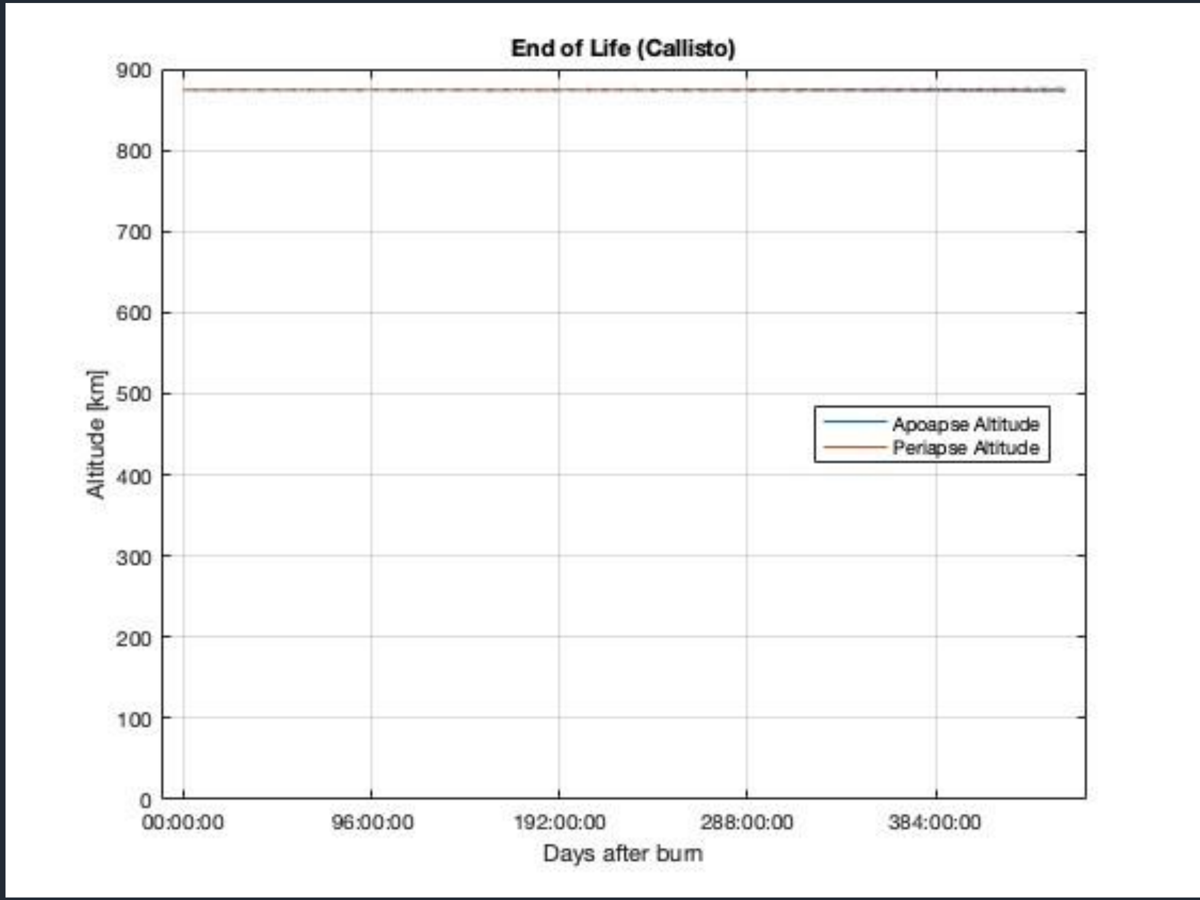


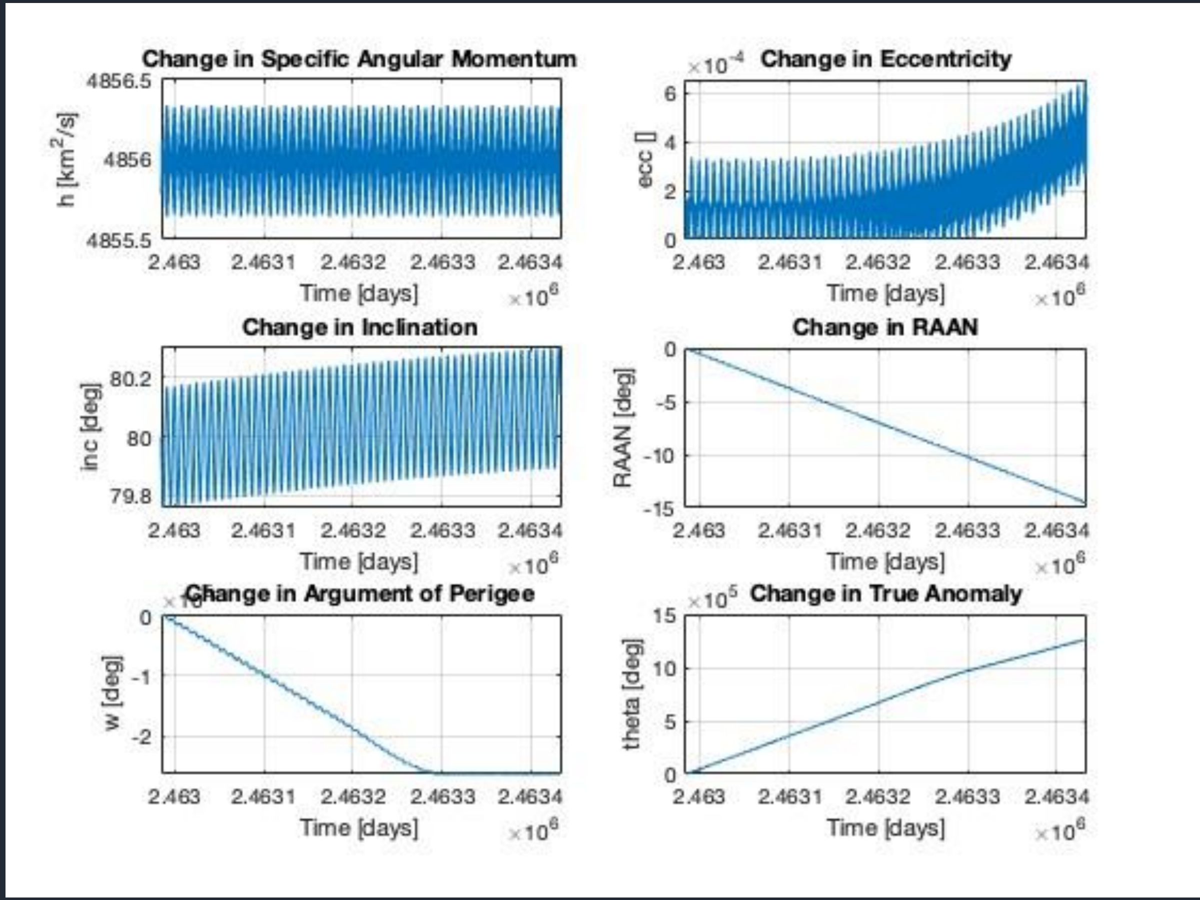


Perturbations: GEC-1

Trajectory: Jovian Operation

Callisto shows no significant change in at least 450 days due to perturbations



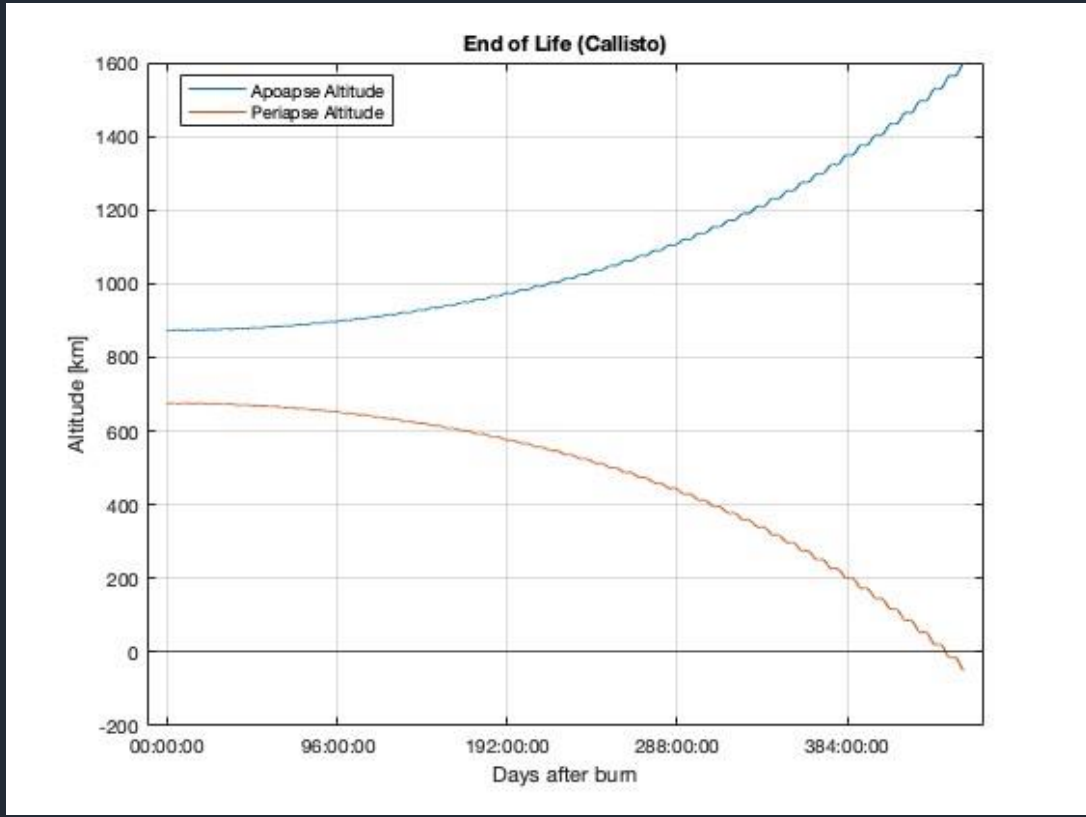


End of Life: GEC-1

Trajectory: Jovian Operation

- Collision with Callisto in July 2047 after coverage is achieved
- Initiate small 20 m/s ΔV burn to make orbit slightly more elliptical in April of 2046
  - Will be pulled into Callisto due to Jupiter n-body effects after about 440 days.

- Total Coverage Achieved:
  - Ganymede: 100 %
  - Callisto: 100 %



Jovian Operation Overview: GEC-2

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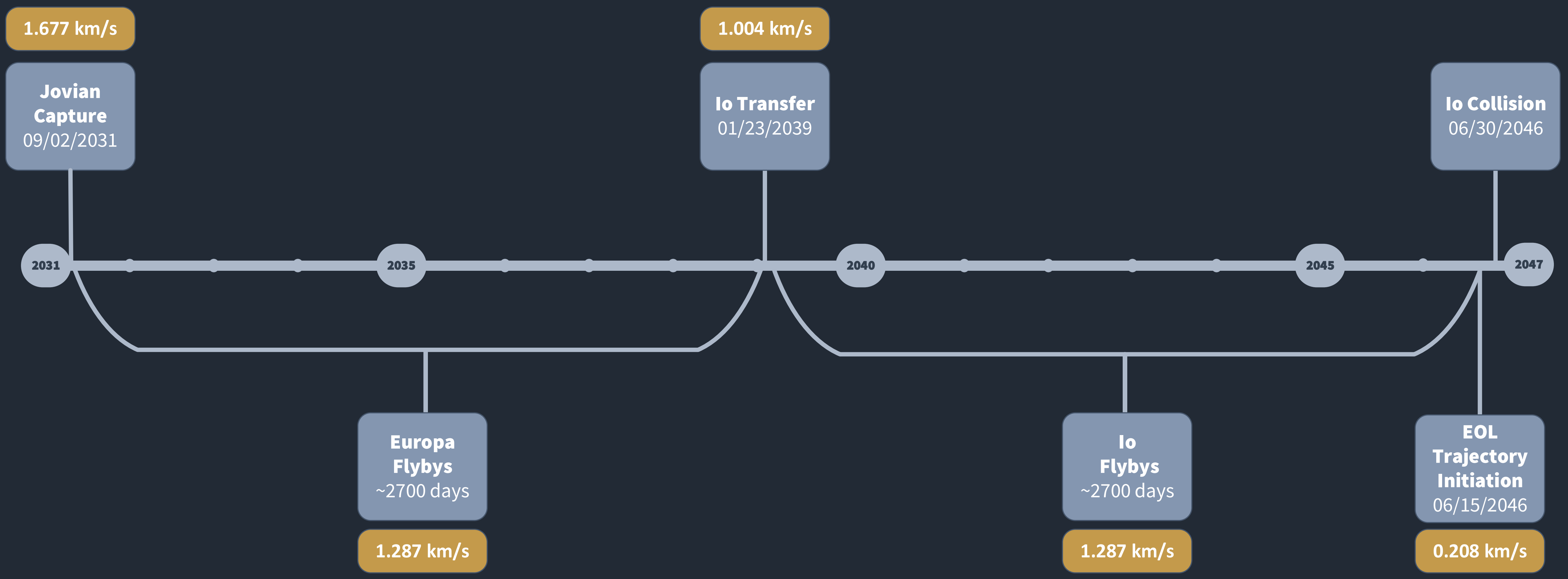
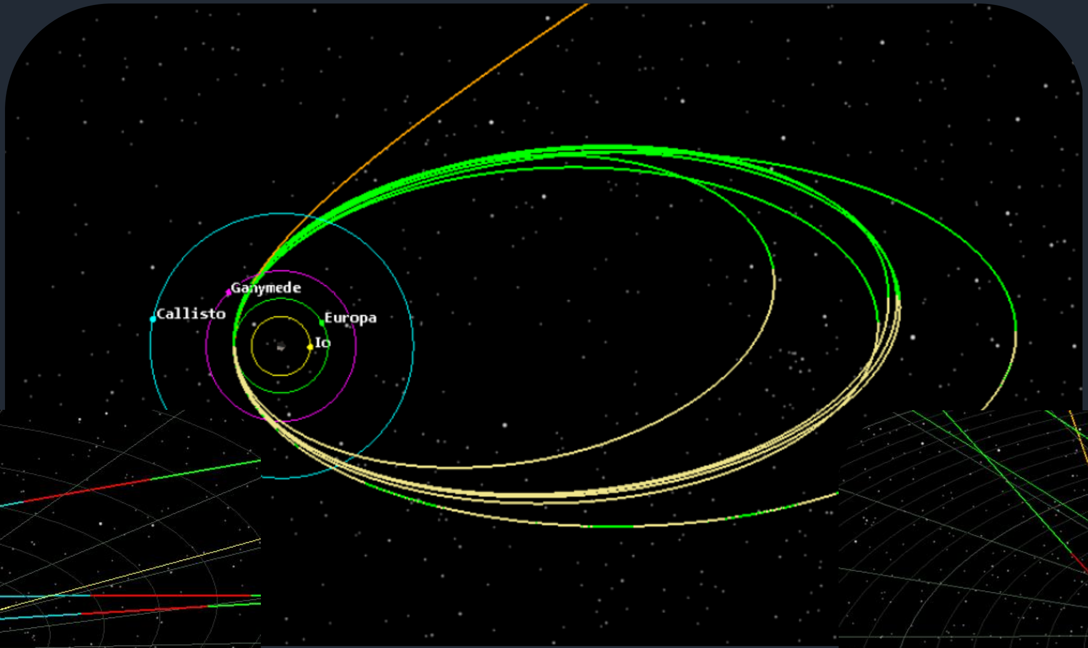
Trajectory: Jovian Operation

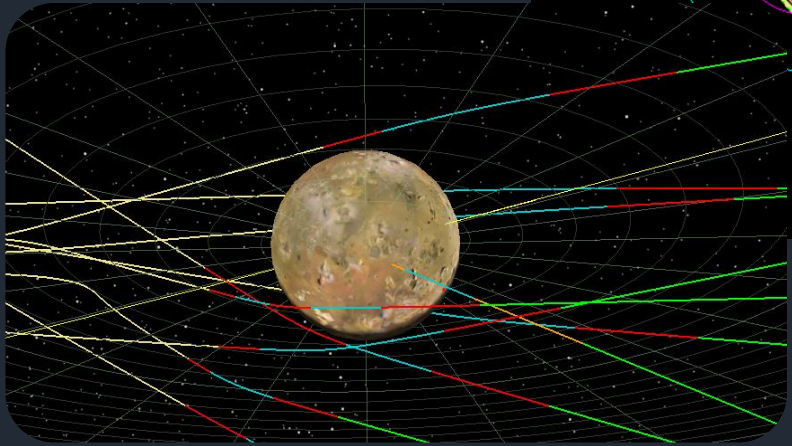


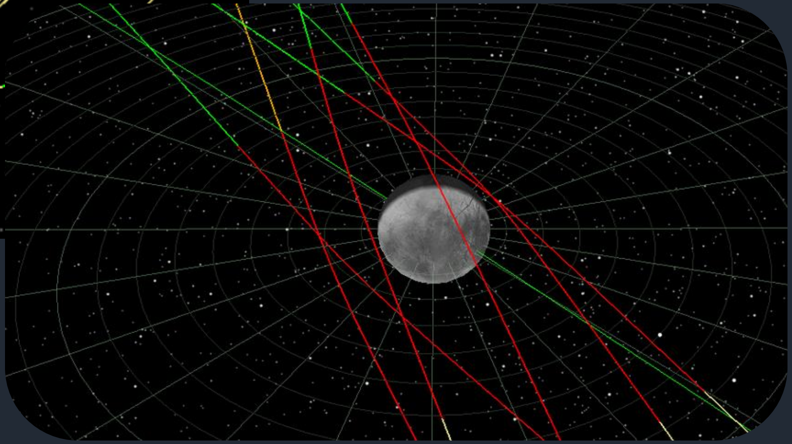
Jovian Operation Overview: GEC-2

Trajectory: Jovian Operation

- Performing flybys to bring Zeus-1 within imaging range while avoiding prolonged exposure to the most extreme Jovian radiation environments

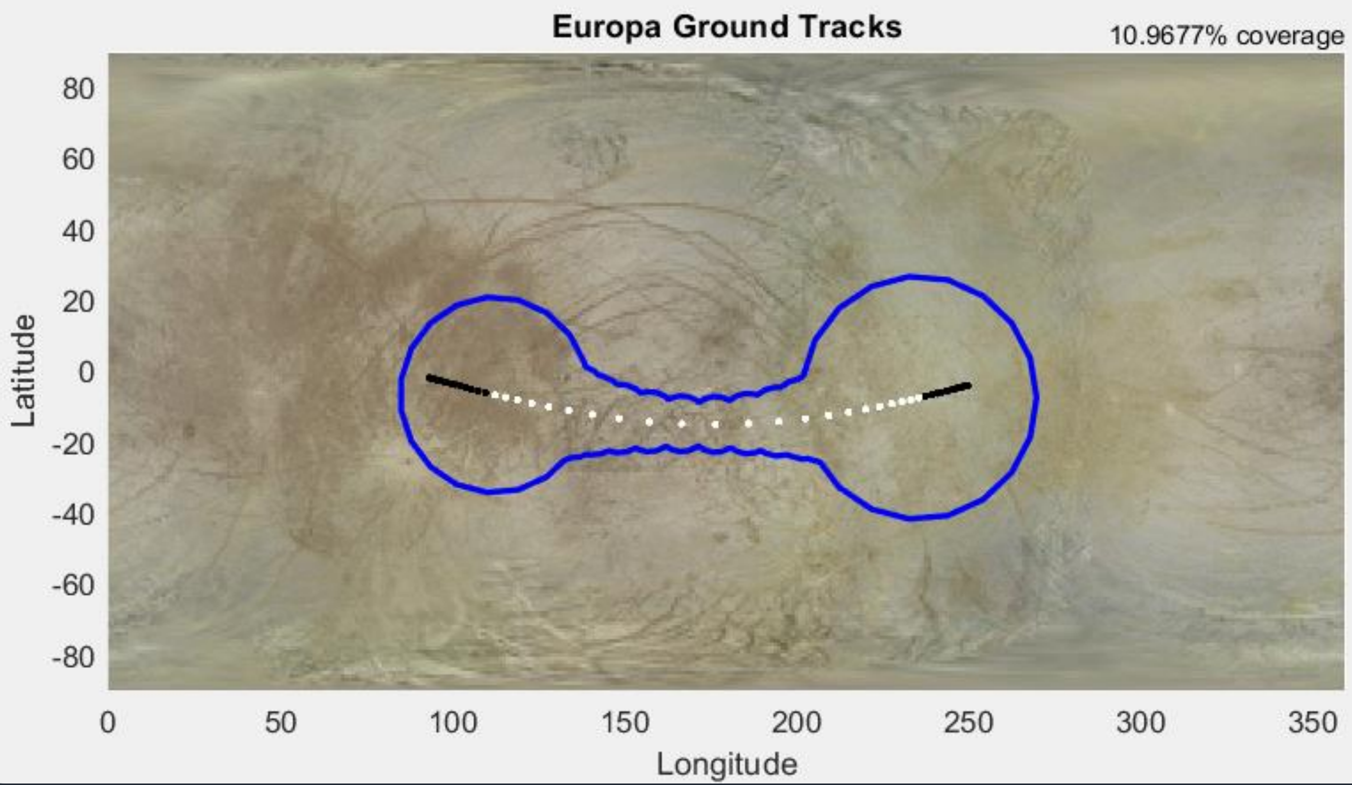






Europa Coverage: GEC-2

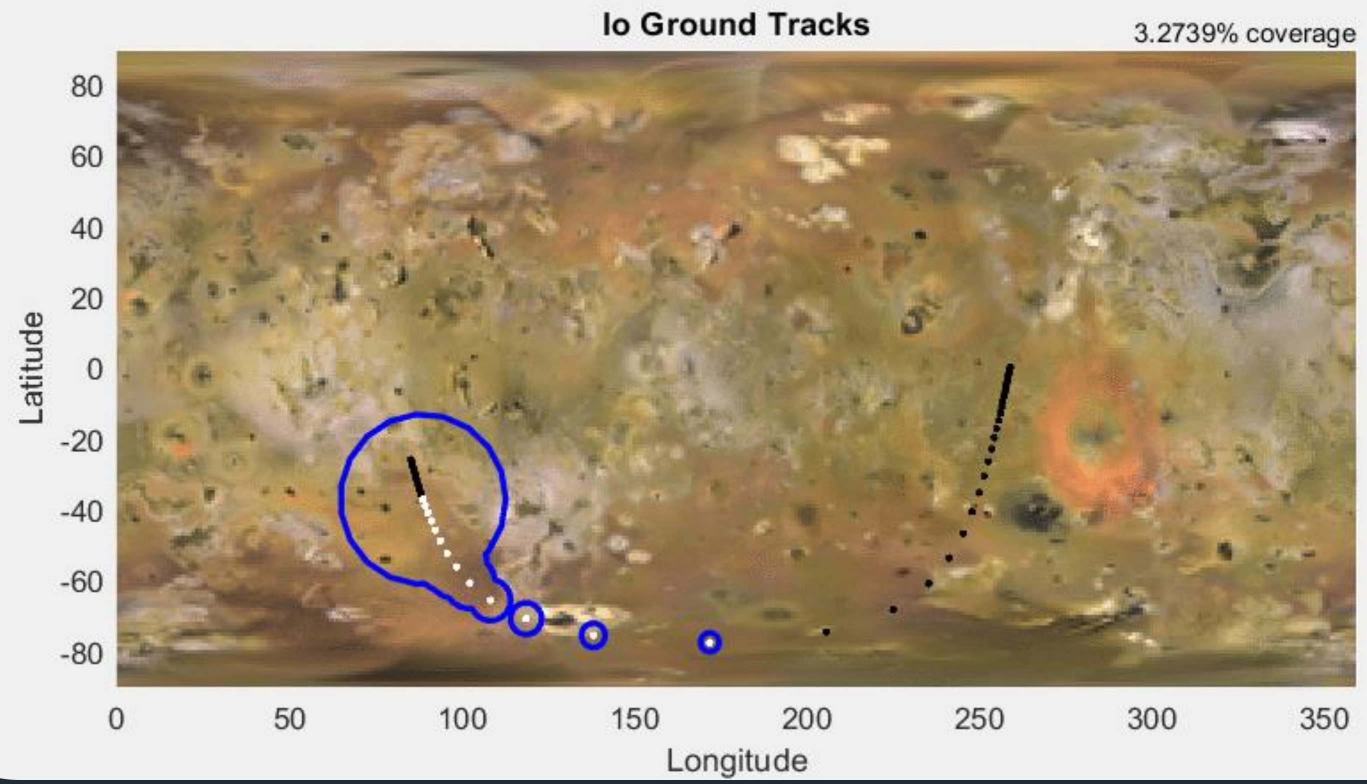
Trajectory: Jovian Operation



Total expected coverage: 100%

Io Coverage: GEC-2

Trajectory: Jovian Operation

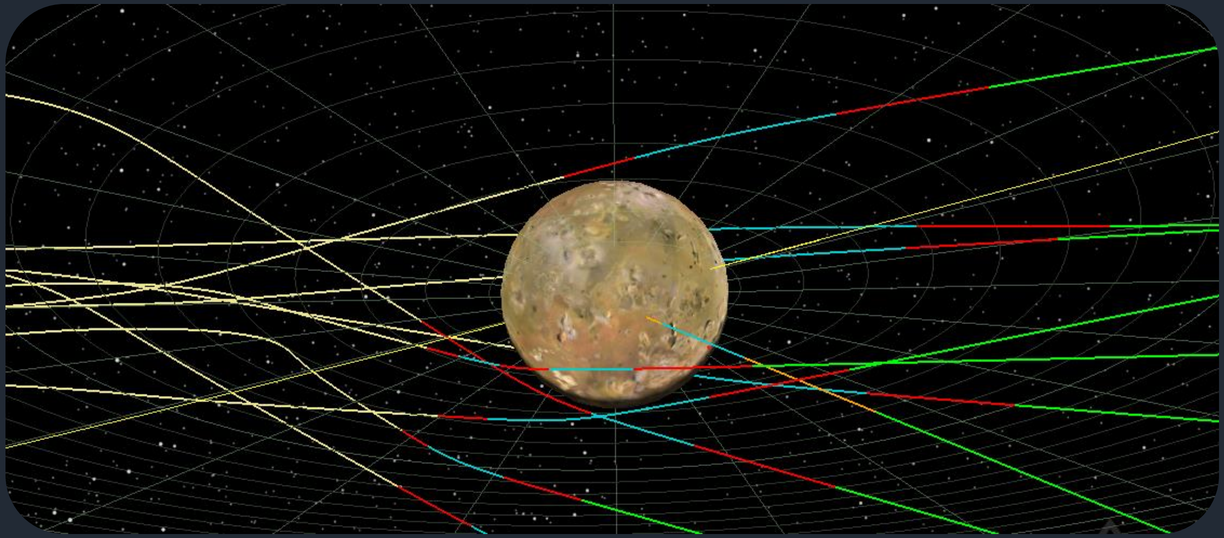


Total expected coverage: 100%

End of Life: GEC-2

Trajectory: Jovian Operation

- Major considerations
  - Propellant requirement
  - Jovian body contamination
- GEC-2 will initiate trajectory to collide with Io at the point of closest approach
- Occurring on 09/02/46 at end of the operating period in the Jovian system



ΔV Mission Breakdown - Timeline

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Mission Phase	Segment	Maneuver	ΔV Estimate [km/s]
Launch	Falcon Heavy 1	Earth Departure	4.8
	Falcon Heavy 2	Earth Departure	4.88
Interplanetary	GEC-1	Deep Space Maneuver 1 & 2	0.015
		Earth Flyby	0.248
		Capture at Ganymede	2.860
	GEC-2	Deep Space Maneuver 1	0.005
		Earth Flyby	0.394
		Capture at Jupiter	1.677
Jovian	GEC-1	Ganymede Departure	1.018
		Correction Burns (x3)	0.014
		Capture at Callisto	1.105
		Callisto Collision	0.027
	GEC-2	Europa Coverage Burns	0.123
		Europa -> Io Transfer	0.128
		Io Coverage Burns	0.123
		Io Collision	0.208

ΔV Mission Breakdown - Stages

Trajectory

Stage	Estimated Total ΔV Requirements [km/s]
Falcon Heavy 1	4.80
Falcon Heavy 2	4.88
GEC-1	5.29
GEC-2	2.66

Current Assumptions:

- Instantaneous impulsive burns
- Interplanetary propagation incorporates solar radiation pressure only
  - N-body modeled but not found to be significant for interplanetary
  - Drag and non-spherical earth not significant for trajectories outside of Earth's SOI
- GEC-1 Inter-lunar propagation is assumed to be only two-body effects
- GEC-1 lunar orbits incorporates Jupiter N-Body effects and lunar oblateness
  - Lunar oblateness insignificant compared to Jupiter N-Body effects

Propulsion

Grace Garmire

Launch Vehicle Selection

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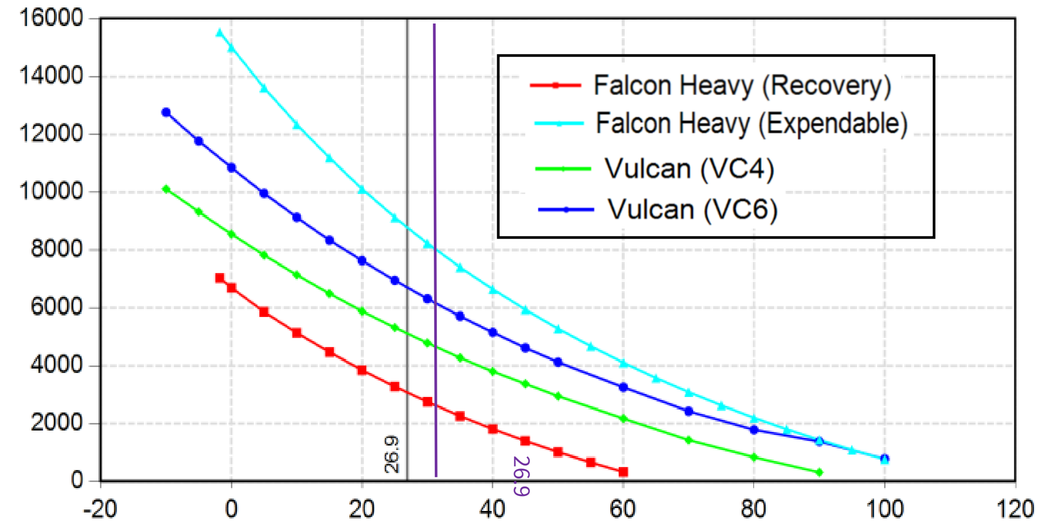
Propulsion

- Major considerations informing launch vehicle choice
  - Characteristic Energy - C_3
  - Maximum payload mass
  - Launch site latitude
  - Fairing size
- Launch vehicle carrying GEC-1
  - Date: 1/08/2026
  - Required C_3=**24.7 km²/s²**
- Launch vehicle carrying GEC-2
  - Date: 11/24/2026
  - Requires C_3=**26.9 km²/s²**
- Higher energy requirement lowers payload mass ceiling

Launch Vehicle	Max payload mass at desired C_3 [kg]	
	GEC-1	GEC-2
Vulcan Centaur 4	5350	5115
Vulcan Centaur 6	6990	6705
Falcon Heavy (Recovery)	3305	3070
Falcon Heavy (Expend)	9185	8785

Table values and plot from NASA's Launch Vehicle Performance tool[2]



Launch Vehicle Selection

Propulsion

Fairing Size

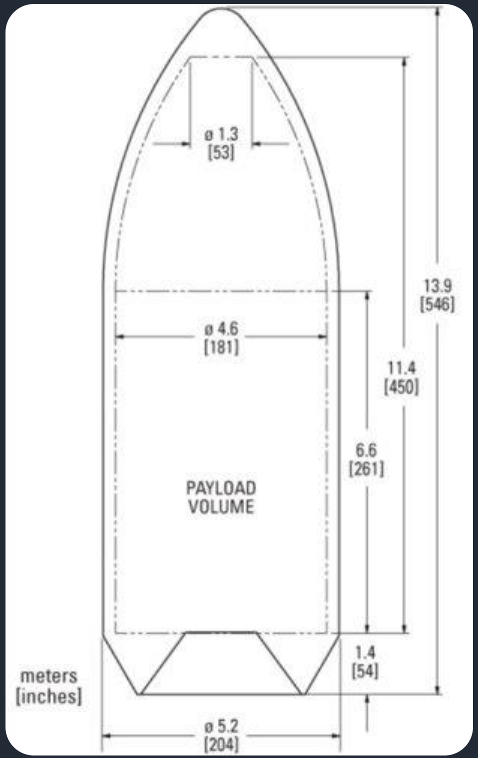
- Vulcan Centaur has a larger payload envelope
    - Internal diameters of both fairings ~ 4.6m
    - Vulcan has an extra 4 m length

Payload Mass

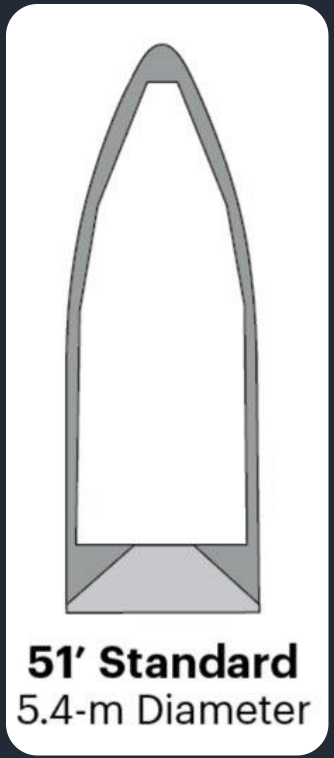
- The extra 2000 kg afforded with Falcon Heavy outweighs extra space available with Vulcan due to high ΔV requirements after launch

Selection: One Falcon Heavy launch for each spacecraft

- Both spacecraft will be designed to the lower payload ceiling of 8785 kg from the previous slide, because they're identical



Falcon Heavy[3]



Vulcan [4]

Vehicle	Fairing Internal Diameter [m]	Fairing Length [m]
Vulcan Centaur 6	4.6	15.5
Falcon Heavy	4.6	11.5

Propulsion System Requirements

Propulsion

The desired mission profile requires that the spacecraft propulsion system must:

- Provide **5.29 km/sec of ΔV** for each spacecraft
- Ignite 60+ times in space

To achieve these goals, the propulsion system must:

- Maintain functionality across long periods of inactivity
- Reliably reignite in space
- Remain stable when subject to:
  - Temperature fluctuations
  - Pressure fluctuations

The system that we chose has extensive flight heritage demonstrating these capabilities. We have implemented pressure and temperature sensors to ensure optimal operation.

Propulsion System Type Selection

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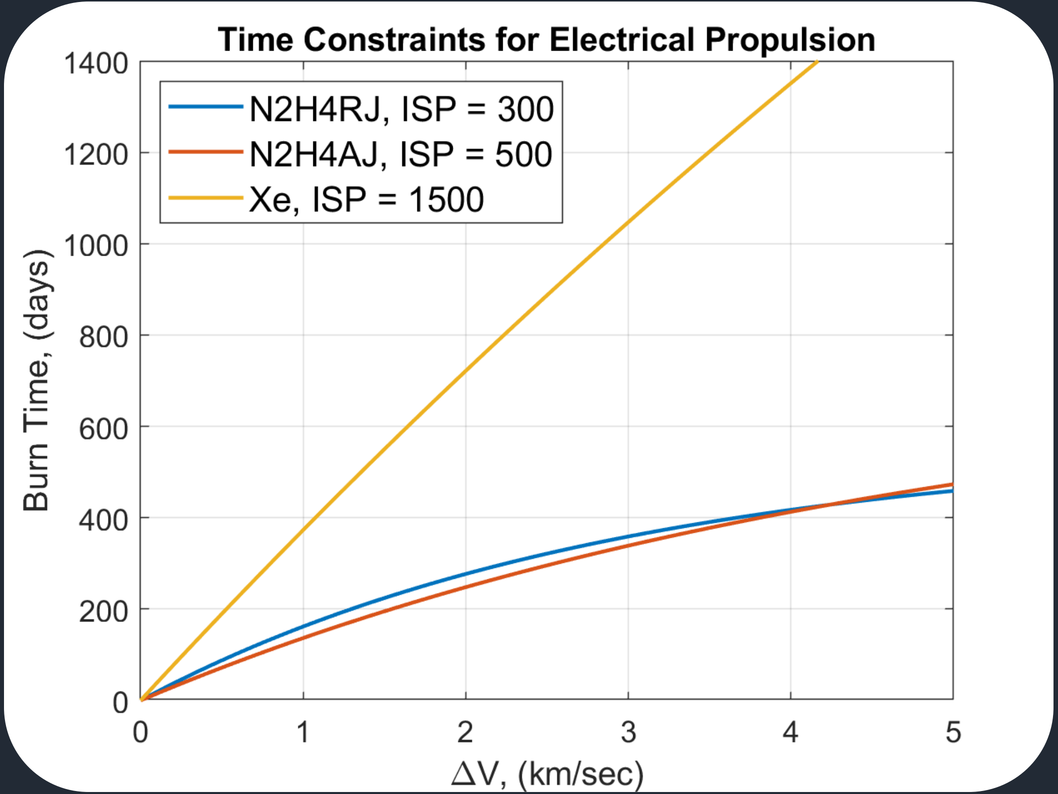
Propulsion

- Chemical and Electric Propulsion (EP) were considered and traded
- Both propulsion types can provide the required ΔV
- EP Limitations
  - Power required necessitates excessively large solar panels and batteries
  - Burn time is excessively lengthened
- Chemical Propulsion was selected for use for both the interplanetary and Jovian segments of the mission



Chemical Propellant Selection

Propulsion

Factors influencing chemical propellant combinations:

- Isp, mass, complexity, flight heritage, and compatibility with deep space missions

High Isp is necessary because of the high ΔV requirement, leading to the choice of a bipropellant system.

Cryogenically stored fuel (e.g. LOX, LH2, LCH4) necessitates:

- Large tank volume, due to low propellant density
- Strenuous thermal requirements
- Storage for long durations is challenging due to boil-off

Selected: hypergolic combination of MMH and MON/NTO

- Extensive interplanetary flight heritage
- Stable, storable, reliable, and meets mission requirements

Engine Definition and Selection

Propulsion

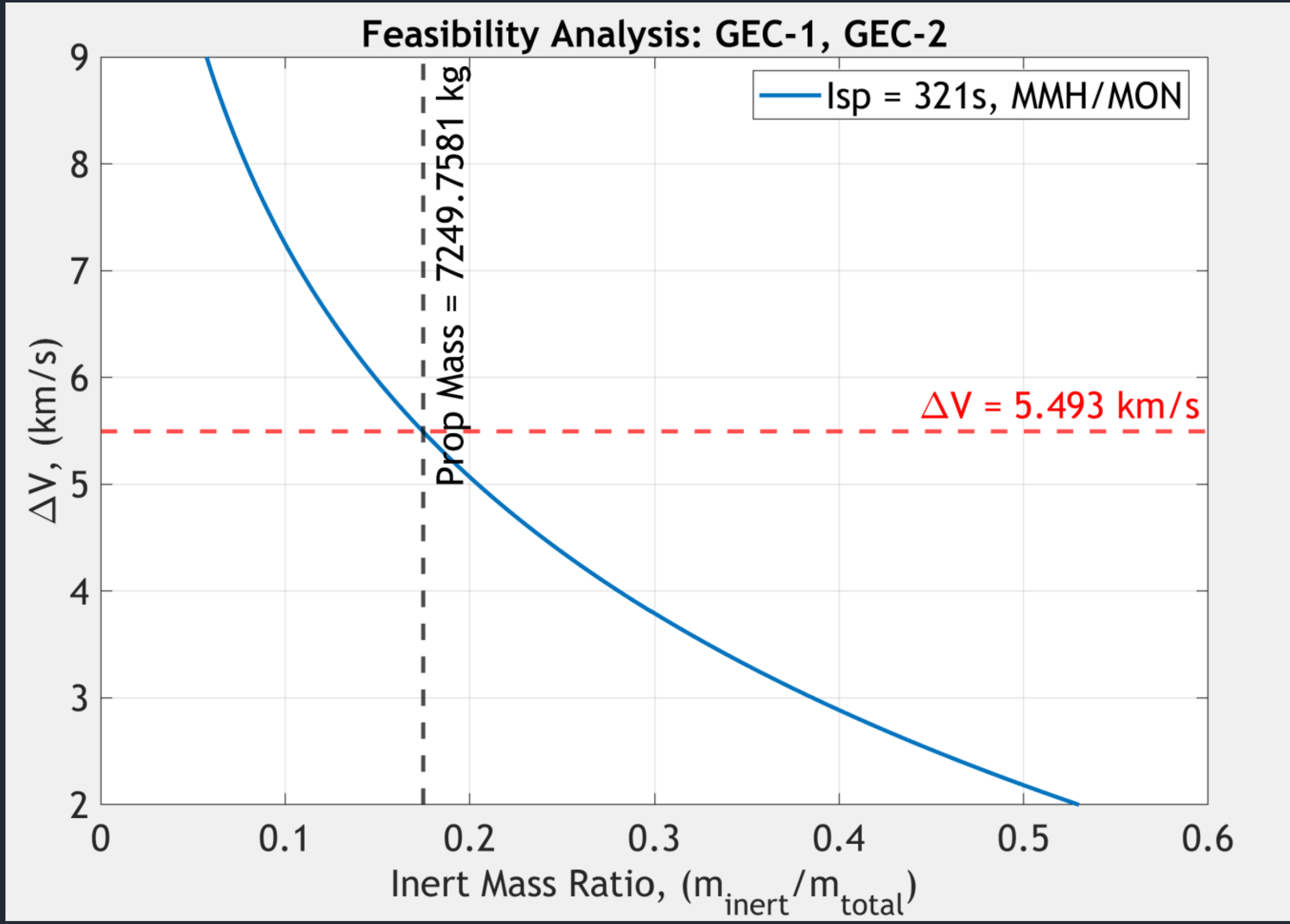
- The propulsion system's main engine will be bought from a commercial vendor, and thus needs to be selected
  - It will be integrated into the spacecraft's overall propellant feed system
- Selected engine: S400-15 (ArianeGroup)
  - Thrust: 425 N
  - Isp: 321 s
  - Propellants: MMH and NTO/MON
  - Single Burn Life – 1.85 hr
    - 4-engine configuration selected to ensure no single burn lasts more than this value
  - Total Burn Life – 8.5 hr

400 N Bipropellant Apogee Motors (space-propulsion.com)

GEC-1 & GEC-2 Propulsion System

Propulsion

Total Mission Propulsion Capability	
Inert Mass Ratio	0.175
Total Initial Wet Mass (kg)	8785
Total Inert Mass (kg)	1535
Total Propellant Mass (kg)	7250
Oxidizer Volume (m^3)	3.441
Fuel Volume (m^3)	2.613
GEC-1 & GEC-2 ΔV (km/s)	5.493



GEC-1 Maneuver Breakdown

Propulsion

Maneuver	ΔV [m/s]	Burn Time [s]	Δm [kg]
Deep Space Maneuver 1	7	34.7	18.8
Deep Space Maneuver 2	8	41.8	22.6
Earth Flyby	248	1224.9	661.2
Capture at Jupiter	1320	5118.5	2763.2
Capture at Ganymede	1540	3802.0	2052.5
Ganymede Departure	1018	1665.0	898.9
Capture at Callisto	1119	1304.8	704.4
Callisto Collision	27	26.2	14.2
Total	**5287**	**13218 (3.67 hr)**	**7135.7**

GEC-2 Maneuver Breakdown

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Propulsion

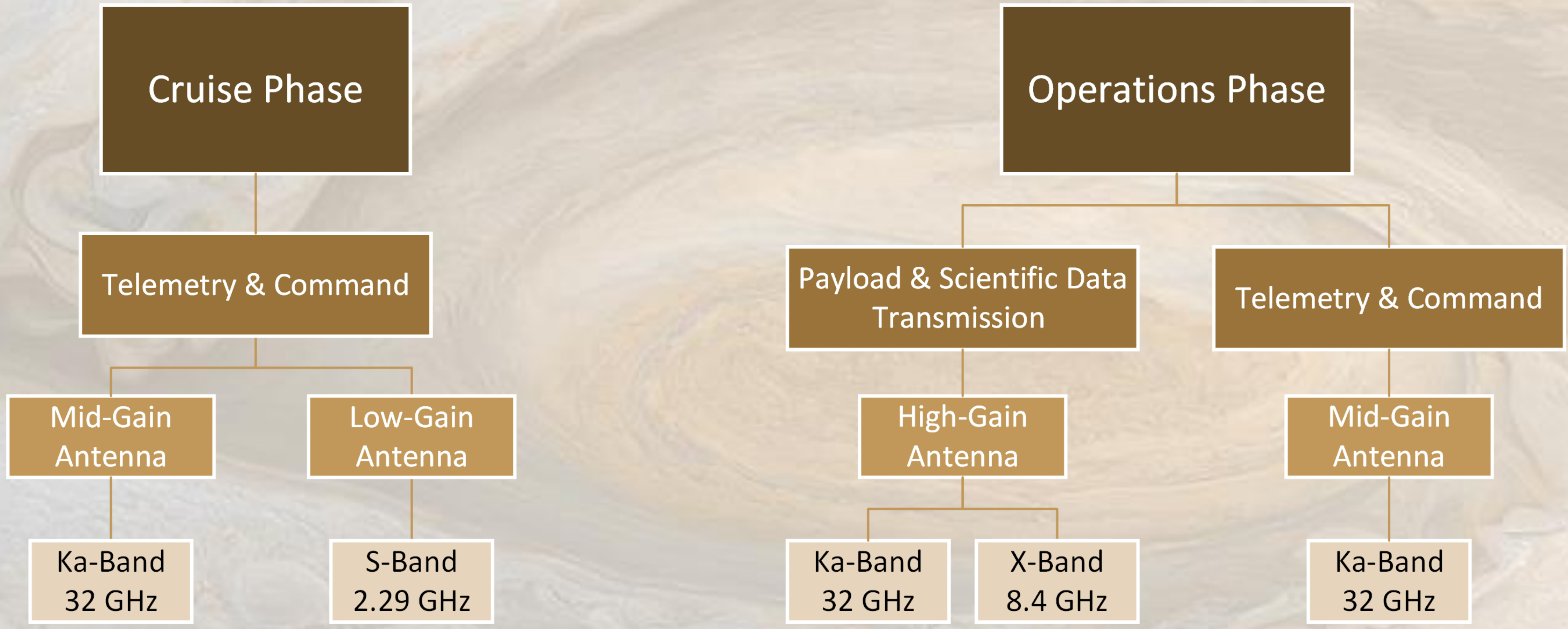
Maneuver	ΔV [m/s]	Burn Time [s]	Δm [kg]
Deep Space Maneuver 1	5	25.8	13.9
Earth Flyby	394	1908.1	1030.1
Capture at Jupiter	1677	5910.9	3191.0
Europa Coverage Burns	123	321.97	173.8
Europa -> Io Transfer	128	321.97	173.8
Io Coverage Burns	123	297.3	160.5
Io Collision	208	477.1	257.6
Total	**2658**	**9263.1 (2.57 hrs)**	**5000.7**

Communications

Andy Cristales, Abigail Outcalt

Comms System Overview

Communications



Link Budget Overview

Communications

$$\frac{E_b}{N_0} = EIRP + L_{pr} + L_s + L_a + G_r + 228.6 - T_s - 10\log(R)$$ [1]

Variables

- E_b/N_0 - energy per bit
- EIRP – effective isotropic radiative power of the transmitter

Fixed Values

- L_{pr} – pointing loss (dB)
- L_s – free space loss (dB)
- L_a – atmospheric attenuation (dB)
- G_r – receiving antenna gain (dB)
- T_s – system noise temperature (dB·K)
- R – data rate (bps)

Process

- Select target E_b/N_0 for the link
  - Equal to the desired SNR
- Determine the value of the fixed quantities based on the mission parameters (all quantities in dB)
- Increase transmitter EIRP until the RHS of the equation is greater than or equal to the target E_b/N_0

Link Budget Process - SNR

Communications

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College of Engineering

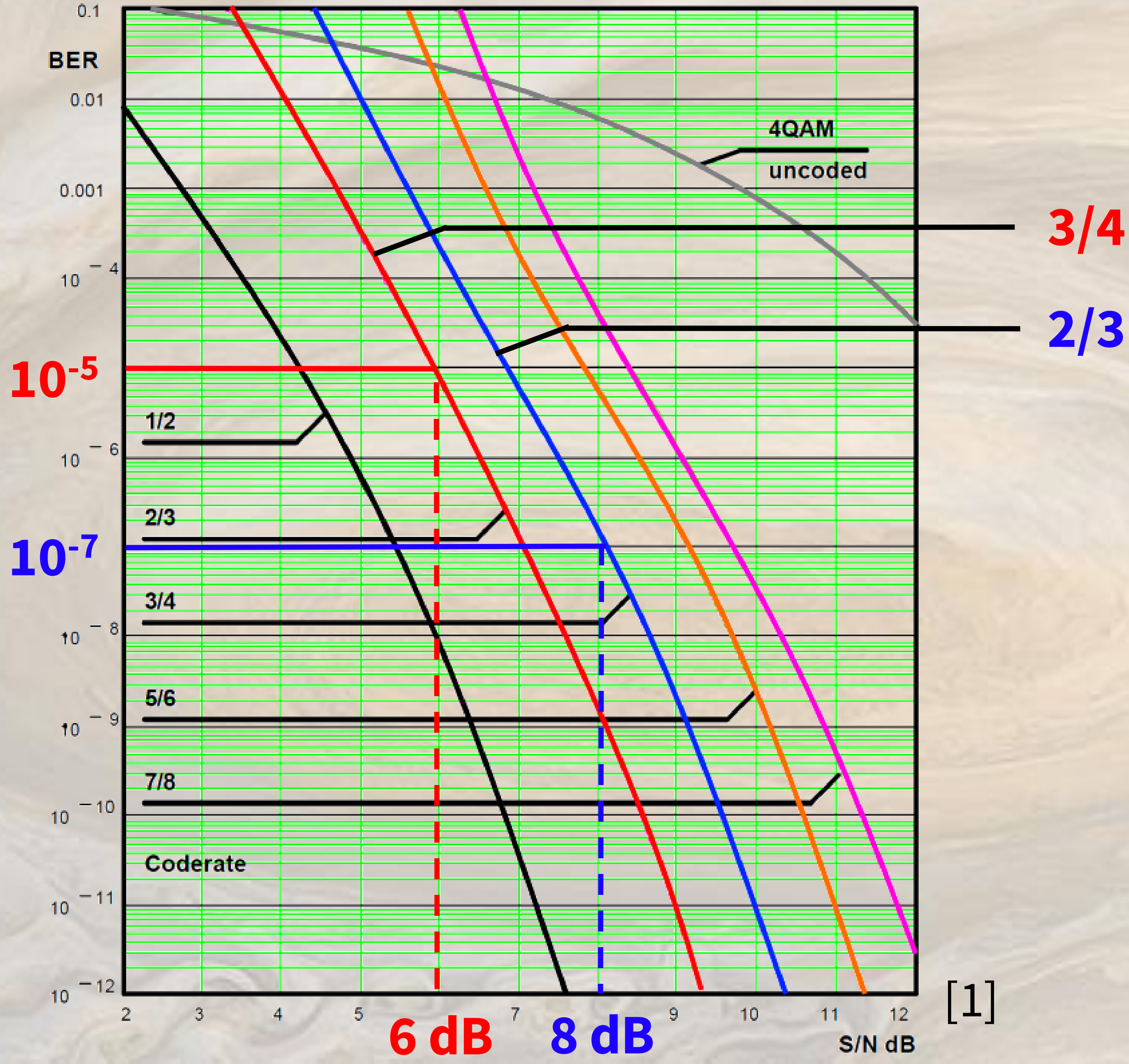
Selecting a target Signal-to-Noise Ratio (SNR) for tolerable Bit Error Rate (BER):

- Telemetry and health data
  - Tolerable BER of 10^{-7}
    - Ensures telemetry data and commands have minimal to no errors during transmission
  - 3/4 encoding scheme
    - Can also be combined with more advanced error correcting codes like Reed-Solomon

- Payload video data
  - BER of 10^{-5} – acceptable error rate for typical video transmission [2]
  - 2/3 encoding scheme
    - High data rate limits the encoding scheme since sending multiple bits to check a single video data bit will multiply the payload data rate, which is already in the 10s of Mbps

Link Budget Process - SNR

Communications



[1]

Link Budget Process - Losses

Communications

- L_s – free space loss
  - Varies by frequency and distance from ground station
  - Distances divided into best and worst case
    - X-band & Ka-band – Jupiter orbital radii
    - S-band – Earth flyby periapsis and SOI radius

$$L_s = 20 \log \left(\frac{\lambda}{4\pi S} \right)$$

Frequency	X-band		Ka-band		S-band	
Case	Best	Worst	Best	Worst	Best	Worst
λ (m)	0.009		0.036		0.131	
S (km)	7.15e8	9.68e8	7.15e8	9.68e8	150	9.29e5
L_s (dB)	-288.0	-290.6	-299.6	-302.3	-143.1	-219.0

Link Budget Process - Losses

Communications

- L_a – Atmospheric attenuation
  - Varies by frequency
  - X-band – 0.06 dB
    - Taken from the Juno mission link budget
  - Ka-band, S-band – 2 dB
    - Estimated based on typical values
- L_{pr} – pointing loss
  - Assumed to be 2 dB

- T_s - System noise temperature
  - Not the literal component temperatures, but instead the equivalent temperature that would produce the same amount of random noise as the system by Brownian motion
  - Assume low noise amplifier and at least one of the ground station antennae pointed at the night sky
    - Low noise amplifier – 12 K
    - Ground station antenna – 278 K
  - Total system noise temperature: 290 K
    - Convert to dB: 10 log(290 K) = 24.6 dB·K

Link Budget Process – Ground Station

Communications

- G_r – gain of the receiving antenna
  - 74 dB receiver gain based on typical values for the DSN's 34 m diameter ground antennae
  - Ground station locations spread out in longitude so one is always visible from Jupiter
  - Transmission frequencies chosen to conform to the FCC 'Space Research' designation

DSN 34-m Antenna[3]

Link Budget Process – Ground Station

Communications

Ground Station Options:

JPL Deep Space Network

- Three existing ground stations in
  - Goldstone, California, USA
  - Madrid, Spain
  - Canberra, Australia
- Reliable option with decades of heritage
- May be unwilling to allow commercial use

Zeus Industries Custom Ground Station

- Three newly constructed ground stations in
  - Virginia, USA
  - Spain
  - Australia
- Allows more flexibility for scheduling but is more expensive and will take several years to construct

Contract with another third-party ground station provider (new or existing)

Link Budget Process – Data Rate (GEC-1)

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Communications

- GEC-1 will record 4K video of the lunar surface for a portion of the orbit, then stop recording and spend the subsequent orbits transmitting the stored video data
- Minimum data rate determined by lunar orbit period and chosen recording duration
- Current assumptions:
  - Nominal video imaging window of 1 hour during the recording orbit
    - Imaging windows could be a few minutes to over 24 hours depending on trajectory planning needs
  - 40 Mbps payload data rate → 60 Mbps total video data rate with 2/3 encoding
  - Ground contacts make up 20% of the time spent in the lunar orbits

Moon	Recording Duration (min)	Orbit Period (hr)	Orbits per Transmission	Transmission Time per Window (days)	Video Data Size per Window (GB)	Total Video Data Size (GB)	Min Data Rate (kb/s)
Ganymede	60	3.88	356	57.5	27	810	217
Callisto	60	3.60	414	62.9	27	810	199

Link Budget Process – Data Rate (GEC-2)

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Communications

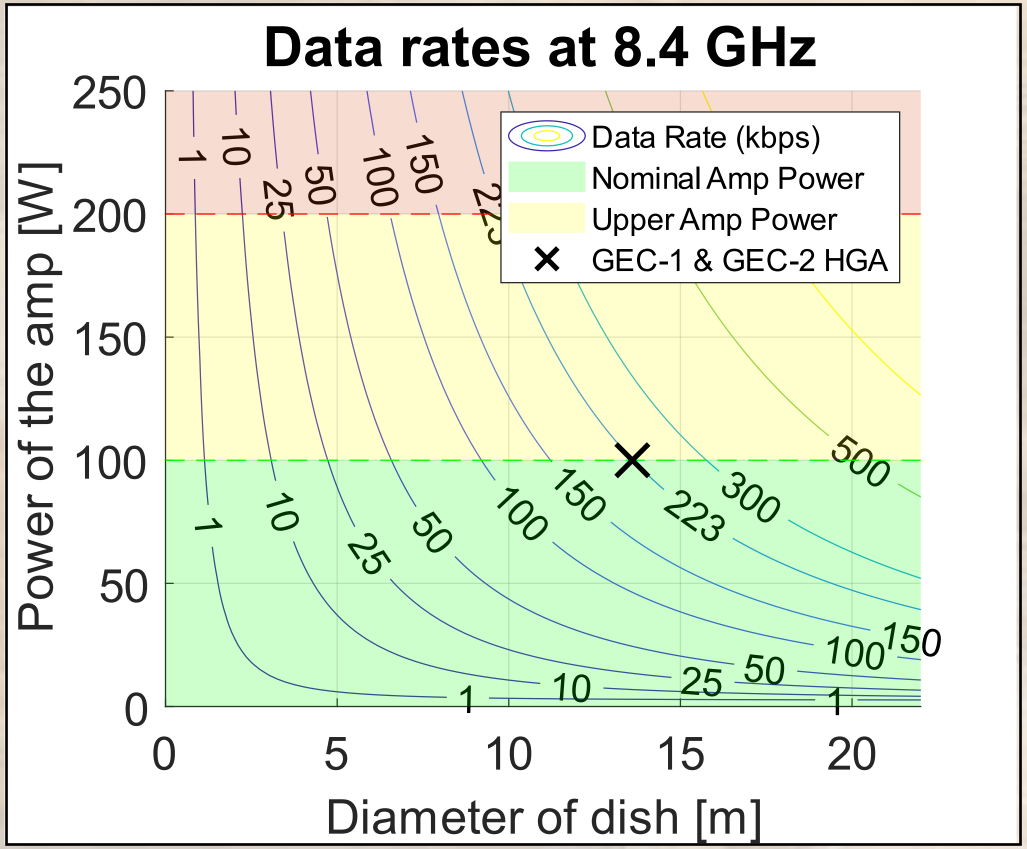
- GEC-2 will record 4K video of the lunar surface during each flyby pass, then transmit the stored video data back to Earth during the rest of the orbit around Jupiter
- Minimum necessary data rate determined by flyby velocity and Jovian orbit period
- Current assumptions:
  - Video imaging only when within allowed altitude of the moons during each flyby
    - Flyby imaging time is determined by the orbit and will be up to 30 minutes long
  - 40 Mbps payload data rate → 60 Mbps total video data rate with 2/3 encoding
  - Ground contacts make up 20% of the time spent in the Jovian orbit

Moon	Flyby Duration (min)	Flyby Orbit Period (days)	Video Data Size per Flyby (GB)	Total Video Data Size (GB)	Min Data Rate (kb/s)
Io	24	28	10.8	540	179
Europa	30	28	13.5	675	223

Link Budget Results – HGA

Communications

- Target SNR used in the calculation: 8 dB
  - 6 dB minimum from video data BER
  - 2 dB development and implementation margin
- EIRP trade space generated by varying reflector diameter and amplifier power at worst case distance
  - Max data rate backed out from the link budget equation
- Nominal amplifier power - 100 W
  - Modeled after the Mars Recon Orbiter TWTAs
- Reflector efficiency: 60%
  - Typical candidate product efficiency + EOL margin
- Antenna diameter determined from required data rate
  - GEC-1: 199-**217** kbps → **13.5 m** diameter
  - GEC-2: 179-**223** kbps → **13.5 m** diameter



Link Budget Results – MGA/LGA

Communications

- MGA data rate fixed to send telemetry & health data (~1.5 kb) in 6 minutes occurring every 30 minutes
  - Reflector diameter is then selected to minimize amplifier power during cruise phase
- LGA is used for near-Earth communication, with the same fixed data rate as the MGA
  - Omnidirectional with low gain and wide beam to remove pointing requirements during launch and Earth gravity assists
- Main consideration is free space loss between best-case closest and worst-case farthest distance

LGA / MGA Downlink						
Constraints	**Symbol**	**LGA Best (perigee)**	**LGA Worst (SOI edge)**	**MGA Best Case**	**MGA Worst Case**	**units**
Target S/N ratio	S/N	8	8	8	8	dB
Antenna Diameter	d	0.084	0.084	0.8	0.8	meters
Antenna gain	G	0	0	46.69	46.69	dB
Data Rate	R	Up to 10,000	Up to 10,000	34	34	bps
Amplifier Power	P	3	3	10	10	Watts
Antenna EIRP at edge	EIRP	1.77	1.77	53.69	53.69	dB
Free Space Loss	Ls	-143.16	-219.00	-299.63	-302.26	dB
Final Calculation		**LGA Best Case**	**LGA Worst Case**	**MGA Best Case**	**MGA Worst Case**	**units**
E_b/N_0 (Signal to Noise Ratio)		91.59	15.75	12.72	10.09	dB
Development and Implementation Margin		2	2	2	2	dB
Surplus Gain		81.59	5.75	2.72	0.09	dB

Antenna Choice – HGA

Communications

The current best candidates are **Northrop Grumman's Astro Mesh** parabolic mesh reflectors

- **AM-1 model** - current design
  - Sizes vary from 6-22m
  - Extensive flight heritage
- **AM-2 model** - possible improvement
  - Sizes vary from 6m-22m, optimized for 18m and above
  - Lighter than AM-1 model

Pointing requirement equivalent to the beamwidth approximated by BW = 70 * λ / d

For the proposed 13.5 m diameter:

Ka-band: 0.05° X-band: 0.18°

AM-1 Class[4]

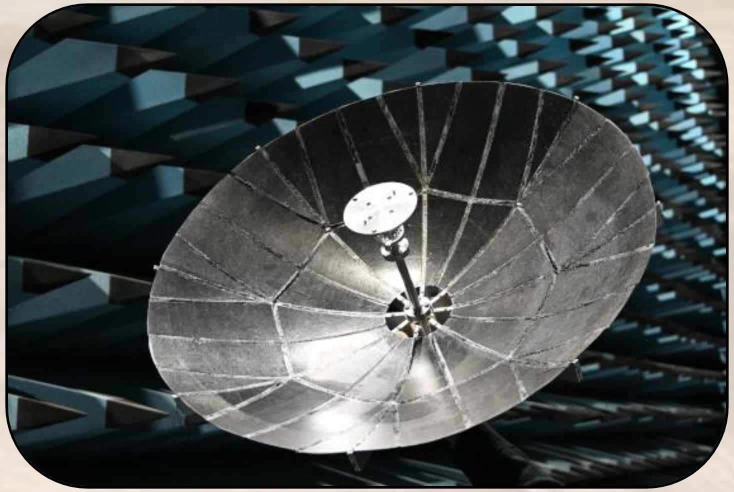
AM-2 Class[4]

Antenna Choice – MGA/LGA

Communications

MGA Current Selection:

- **Oxford Space Systems Hinged Rib**
  - 0.6m to 1.6m diameter
  - High gain of 46 dB - necessary to overcome free space loss
  - Low complexity – higher reliability
  - Used for telemetry, health & command during cruise to Jupiter and Jovian operations



Hinged Rib[5]

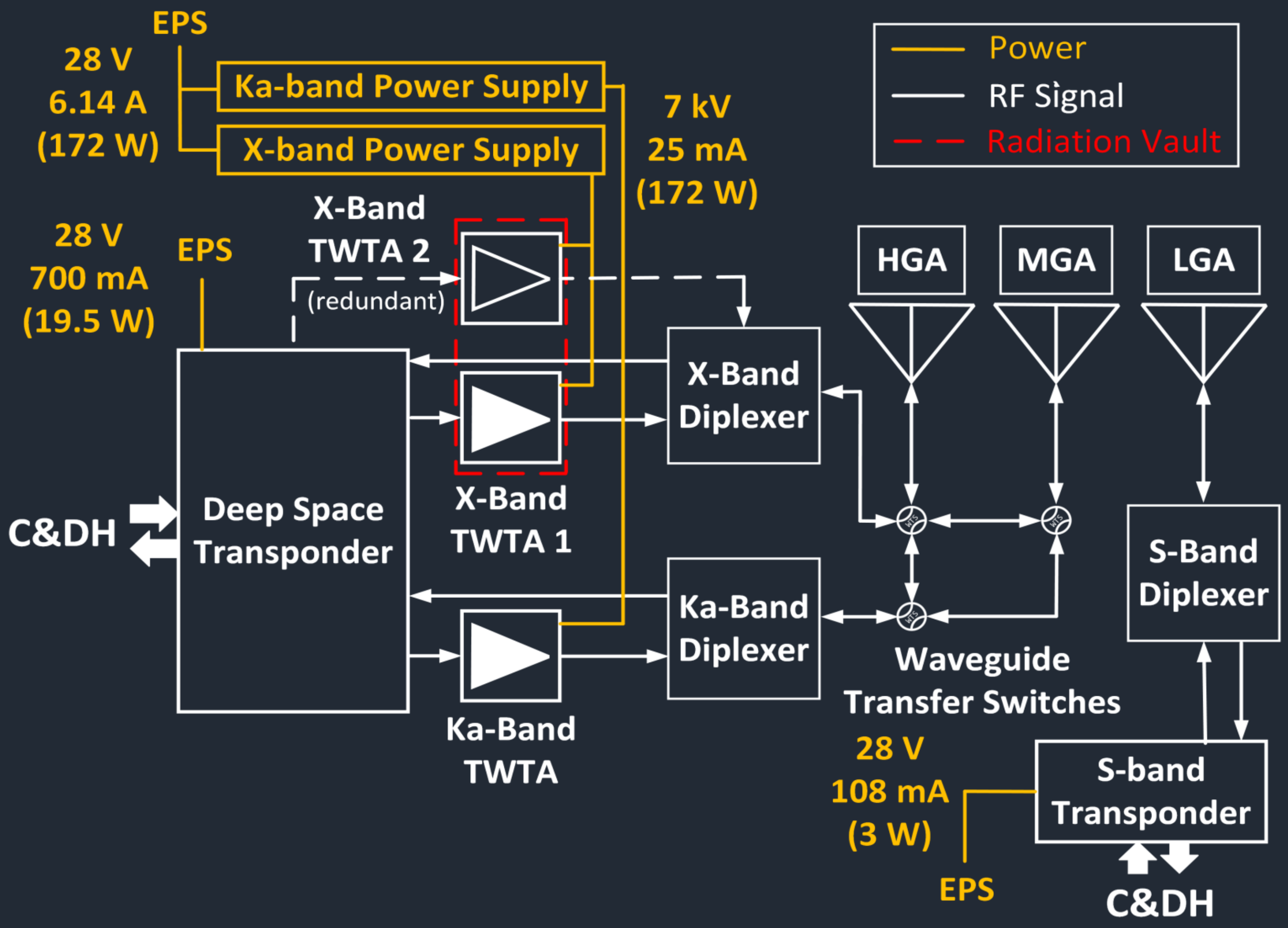
LGA Current Selection:

- **ANYWAVES S-Band TT&C Antenna**
  - 6.5 dB gain at boresight
  - >90° Beamwidth
  - Full coverage for our purposes
  - Used for telemetry, health & command while near Earth

S-Band Antenna[6]

Communications Hardware

Communications



Communications Hardware

Communications

- Transponders
  - General Dynamics Small Deep Space Transponder
    - Used on almost every deep space mission
    - Transmits and receives signals at X- and Ka-band
  - S-band Transponder
    - Manufacturer TBD, assuming typical specs
- Amplifiers
  - TESAT custom-built traveling wave tube amplifiers
    - Flight heritage from the Mars Recon Orbiter
  - 2 redundant X-band amps for reliability, 1 Ka-band
- Diplexers
  - Splits the transmitted and received signals
  - Allows one antenna to be used for both Tx and Rx
- Waveguide Transfer Switches
  - Routes the RF signals between components

Small Deep Space Transponder[7]

Typical Space-Rated TWTA and Power Supply[8]

Communications Hardware

Communications

- Feed Horn
  - Aaronia PowerLOG 40400
  - Spreads out the RF signal onto the surface of the HGA parabolic reflector
  - Dual ridge design allows for transmission in both X- and Ka-bands
- Telescoping Boom
  - Northrop Grumman Telescopic Tube Mast
  - 1 m stowed length, 10 m deployed length
  - 8" diameter hollow aluminum segments
  - Sized to position the feed horn directly above the center of the reflector
    - 5.4 m distance between feed horn and plane of reflector → 0.4 f/D ratio

PowerLOG 40400
Feed Horn [9]

Northrop Grumman
Telescopic Tube Mast[10]

Line of Sight Considerations

Communications

- Line of sight obstruction from Jupiter and the Galilean moons varies by spacecraft
  - GEC-1: Circular orbit → 50% obstruction time at most
  - GEC-2: Flybys → ~10% obstruction time
- Payload data transmission ratio is 20% to reduce necessary DSN usage
  - Both craft will have ample time to transmit during their visibility windows
- Telemetry, health, and command data: obstruction by moons is not negligible, autonomous operation necessary for both short and long-term durations
  - Obstruction from the sun during solar conjunction will cause a blackout for 4½ days every ~13 months
- Ground station locations arranged approximately 120° in longitude to avoid Earth surface obstruction

DSN 120° Skewed Ground Station[11]

Command & Data Handling

Adam Martin

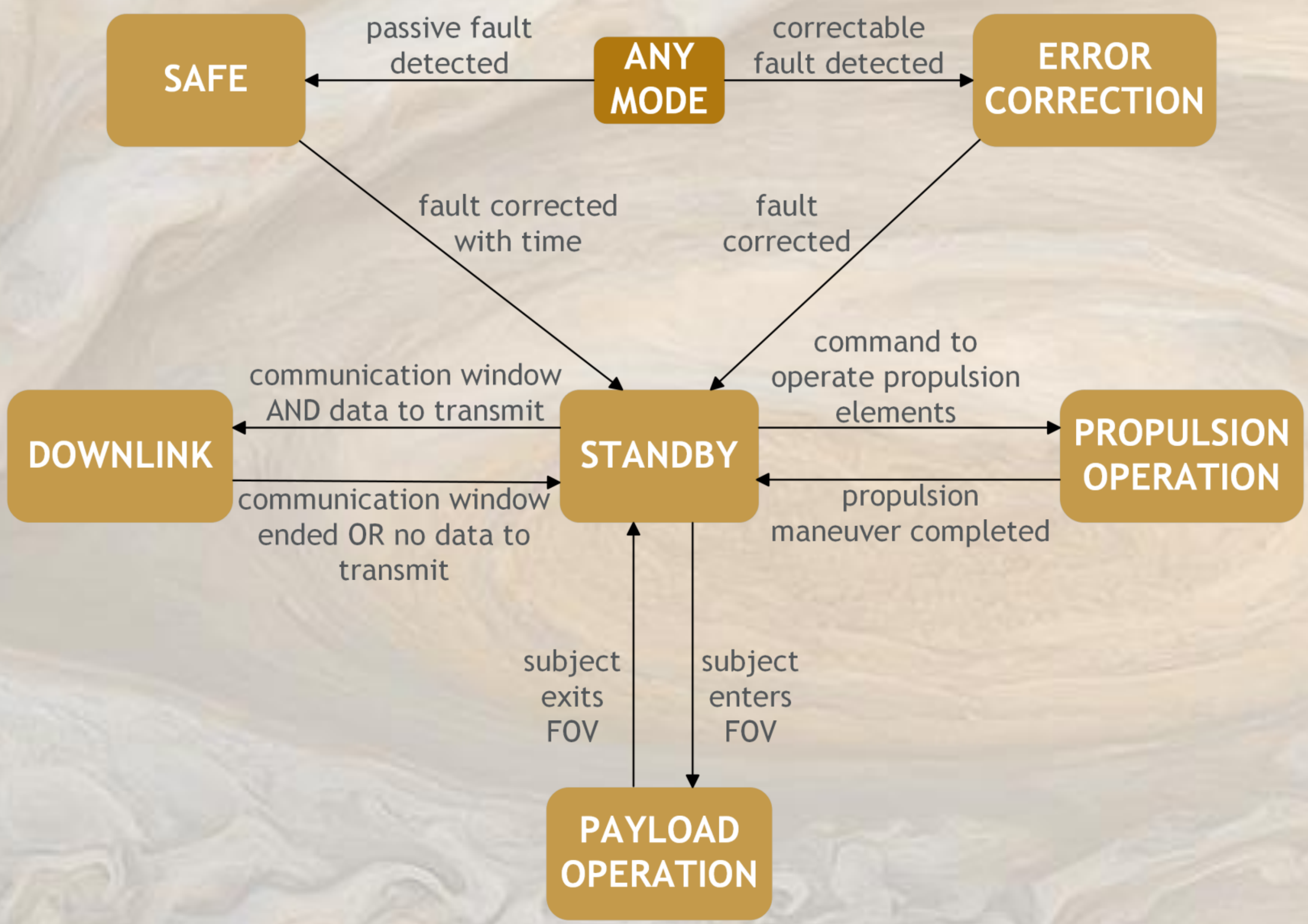
Spacecraft Modes

Command & Data Handling

Mode	Definition
Standby	Nominal mode where commands from ground can be received, no high-power operations or intentionally low power draw.
Payload Operation	Payload is on and collecting data to fulfill mission requirements.
Propulsion Operation	Propulsion system has been provided power for control of attitude or orbital maneuvers.
Downlink	High powered mode to transmit data to ground.
Safe	Low power draw, TCS and EPS remain active to support spacecraft health.
Error Correction	Non-essential systems (payload/scientific instrument operation) disabled in the event of a detected error, spacecraft to correct error.

Spacecraft Mode Transitions

Command & Data Handling



Spacecraft Autonomy

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Command & Data Handling

- ECSS standards define autonomy in mission execution, data storage, and FDIR
- As a deep space mission, our spacecraft requires a high level of autonomy

Type	ECSS Level	Description	Functions [1]
Mission Execution	E3	Execution of adaptive mission operations on-board	Event-based autonomous operations Execution of on-board operations control procedures
Data Management	D2	Storage on-board of all mission data	Storage and retrieval of event reports Storage management Storage and retrieval of all mission data
FDIR	D2	Re-establish nominal mission operations following an on-board failure	Identify anomalies and report to ground segment Reconfigure on-board systems to isolate failed equipment or functions Place space segment in a safe state Reconfigure to a nominal operational configuration Resume execution of nominal operations Resume generation of mission products

Autonomous Transition Examples

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Command & Data Handling

Problem	Detection Method	Entered Mode	Autonomous	Exit Criteria
Component overheating	Temperature reading is outside nominal/expected range	Safe	Yes	Component has been turned off or power lowered, cooled
In eclipse, batteries close to max DOD	~0 voltage from solar panels, battery status	Safe	Yes	Eclipse is exited, solar panels are producing voltage
Dangerous environment caused by the sun	Time stamped command from ground with safe mode duration	Safe	No	Safe mode duration passed
Faulty reaction wheel	Measured speed not equal to commanded speed	Error correction	Yes	AOCS algorithm switched to compensate for faulty wheel
Ground not receiving transmissions	No check response from ground received	Error correction	Yes	Error corrected and check from ground received

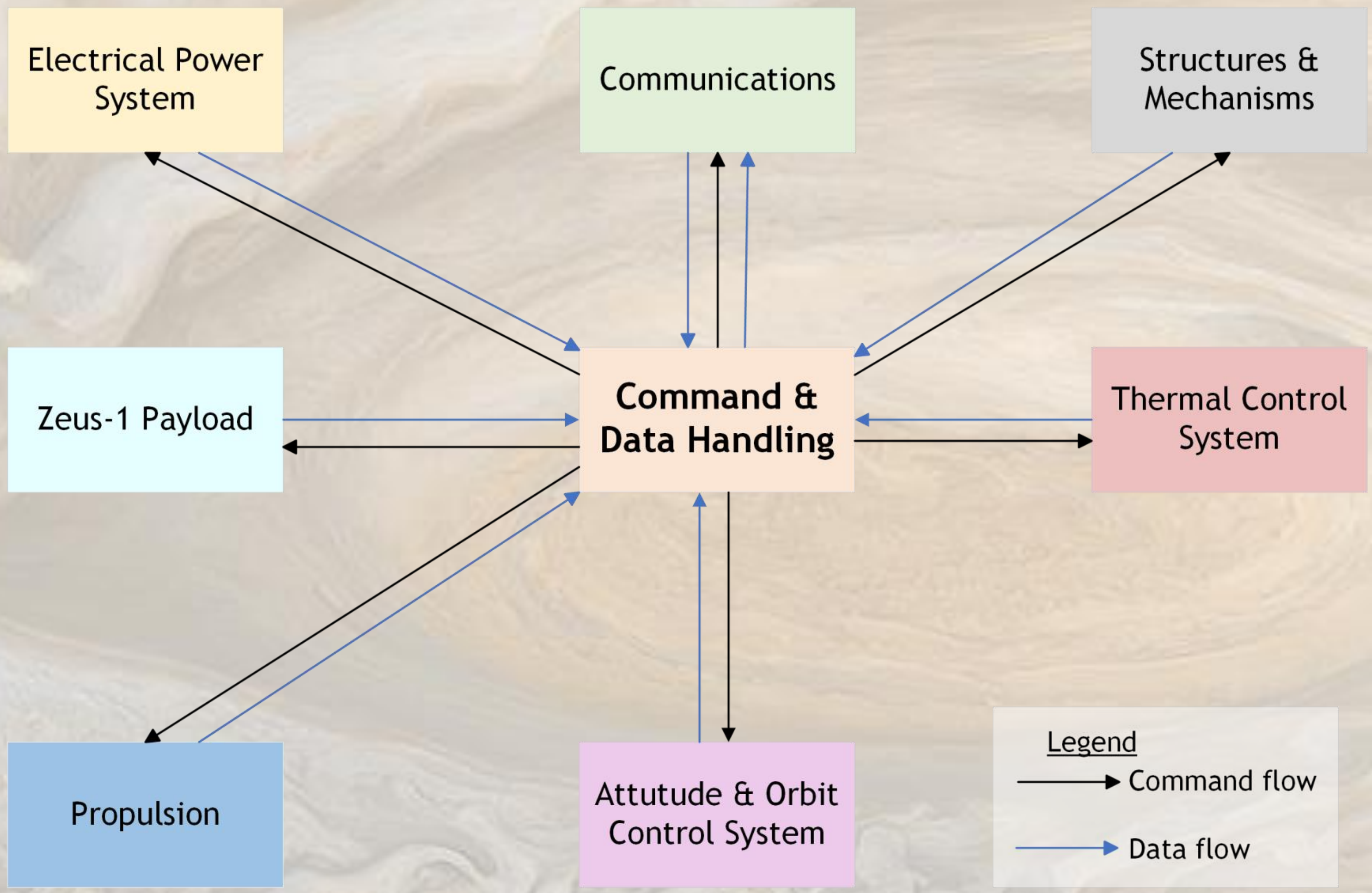
Continuous Monitoring

Command & Data Handling

Subsystem	Data Monitored	Components	
EPS	Temperature, Voltage, Current, Board Status	• Solar Arrays • Power Management System • Battery Management System	• Voltage Converter • Switcher Board
COMM	Temperature	• Transponder • Amplifier	• High Gain Antenna • Medium Gain Antenna
Thermal	Temperature	• Radiation Vault (19 values) • All External Components (60 values)	
Prop	Pressure, Temperature, Flow Rate, Voltage	• Tanks • Chamber • Feed System • Pump	• Thruster Clusters • Gas Generator • Solenoids
AOCS	Sensor Data, Momentum	• Star Trackers • Sun Sensors • Magnetometer Clusters • Accelerometers	• Horizon Sensors • Laser Range Finder • Reaction Wheels

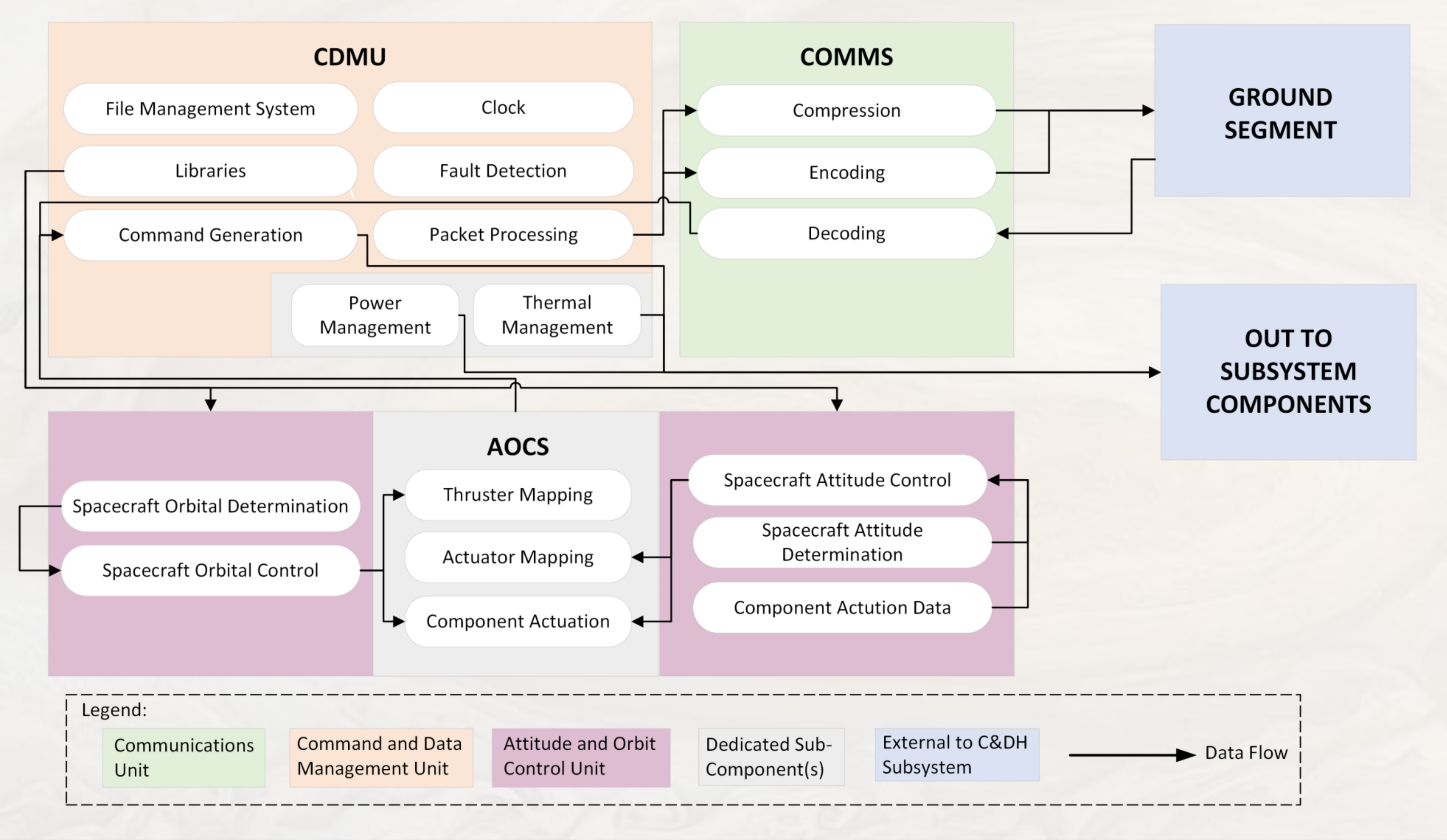
Centralized System Architecture

Command & Data Handling



Processing Functional Flow

Command & Data Handling



C&DH Subsystem Processes

Command & Data Handling

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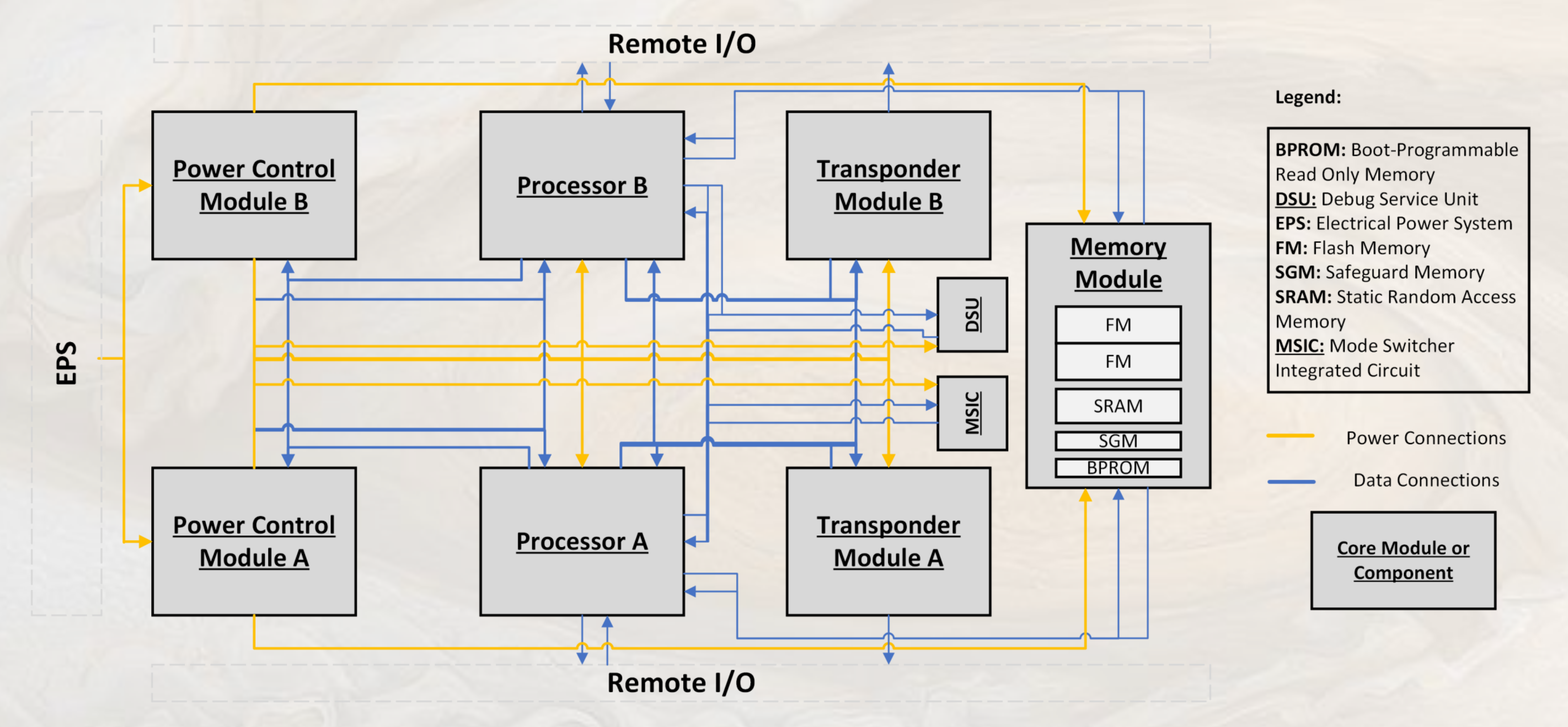
Process	Definition
Fault Detection, Isolation, and Recovery (FDIR)	Top level system health monitoring and recovery action control
Telemetry Processing (TM)	Acquisition, control, and storage of spacecraft telemetry packets
Attitude and Orbit Control (AOC)	Sensor data handling and computation for spacecraft attitude and orbit control
Payload Control	Payload actuation sequencing and data control
Thermal and Power Control	Active spacecraft thermal and power management (if applicable)

Cross-Strapped Core Block Diagram

Command & Data Handling



Onboard Memory Sizing

Command & Data Handling

Memory Type	Onboard Memory Allocation
Safe Guarded Memory (SGM)	256 kB
Boot Programmable Read-Only Memory (Boot PROM)	64 kB
Flash or Electrically Erasable Programmable Read-Only Memory (EEPROM)	256 GB (Zeus-1 video storage drive) + 64 GB (spacecraft health and other associated data)
Static Random Access Memory (SRAM)	1 GB
Total	321 GB

CCSDS

Command & Data Handling

- Standardized satellite communication protocol named after the **Consultative Committee for Space Data Systems**
- Defines telecommand (TC) and telemetry (TM) packet layout
- CCSDS Space Link Extension (SLE) Reference Model eliminates blocking protocols and provides increased security measures
- Although significant overhead is required, errors are minimized, and commands can be properly routed through OBC
- Over 1000 missions have been or are being flown using CCSDS, including upcoming deep space missions Europa Clipper and JUICE

ESA's JUICE [2]

CLTU Composition

Command & Data Handling

CLTU - Command Link Transmission Unit		
Start Sequence	TC Transfer Frame	Tail Sequence
2 byte	<= (248*n) + 8 byte	8 byte

TC Transfer Frame		
Frame Header	TC Frame Data Unit	TC Frame Error Control
5 byte	<= (248*n) + 1 byte	2 byte

TC Segment			
Segment Header	Segment Data Field		
	Packet #1	...	Packet #n
1 byte	<= 248*n byte		

TC Source Packet			
Packet Header	Packet Data Field		
	Data Field Header	Command Data	Packet Error Control
6 byte	4 byte	<= 236 byte	2 byte

CADU Composition

Command & Data Handling

CADU - Channel Access Data Units		
Attached Sync. Marker	Data Space	Check Symbols
4 byte	1115 byte	373 byte

TM Transfer Frame		
Frame Header	Frame Data Field	Frame Trailer
6 byte	1105 byte	4 byte

TM Source Packet			
Packet Header	Packet Data Field		
	Data Field Header	Telemetry Data	Packet Error Control
6 byte	12 byte	<= 1085 byte	2 byte

Telemetry Source Packet Layout

Command & Data Handling

- Computer can be commanded to change packet layout if necessary
- Error Correction Mode contents are based on Standby Mode, with greater allocation given to the subsystem that experienced the error

Telemetry Data - Standby Mode					
C&DH	COMM	EPS	THERM	ADCS	PROP
10-130 B	8-65 B	8-60 B	8-80 B	64-600 B	18-150 B
<= 1085 byte					

Telemetry Data - Downlink Mode					
C&DH	COMM	EPS	THERM	ADCS	PROP
10-150 B	13-85 B	8-100 B	8-80 B	64-600 B	13-80 B
<= 1085 byte					

Telemetry Data - Payload Mode					
C&DH	COMM	EPS	THERM	ADCS	PROP
10-130 B	8-65 B	8-60 B	8-80 B	64-600 B	18-150 B
<= 1085 byte					

Telemetry Data - Propulsion Mode					
C&DH	C	E	T	ADCS	PROP
10-131 B	8 B	8 B	8 B	64-600 B	18-330 B
<= 1085 byte					

Telemetry Data Contents

Command & Data Handling

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Spacecraft Level Mode	Source Data Contents
Standby	Regular polling of spacecraft sensors and component status
Payload Operation	Standby contents
Propulsion Operation	Standby contents + detailed status of propulsion system components
Downlink	Standby contents + additional power and communications information
Safe	N/A
Error Correction	Standby contents + polling of sensors from subsystem that caused fault

Video Data Packets

Command & Data Handling

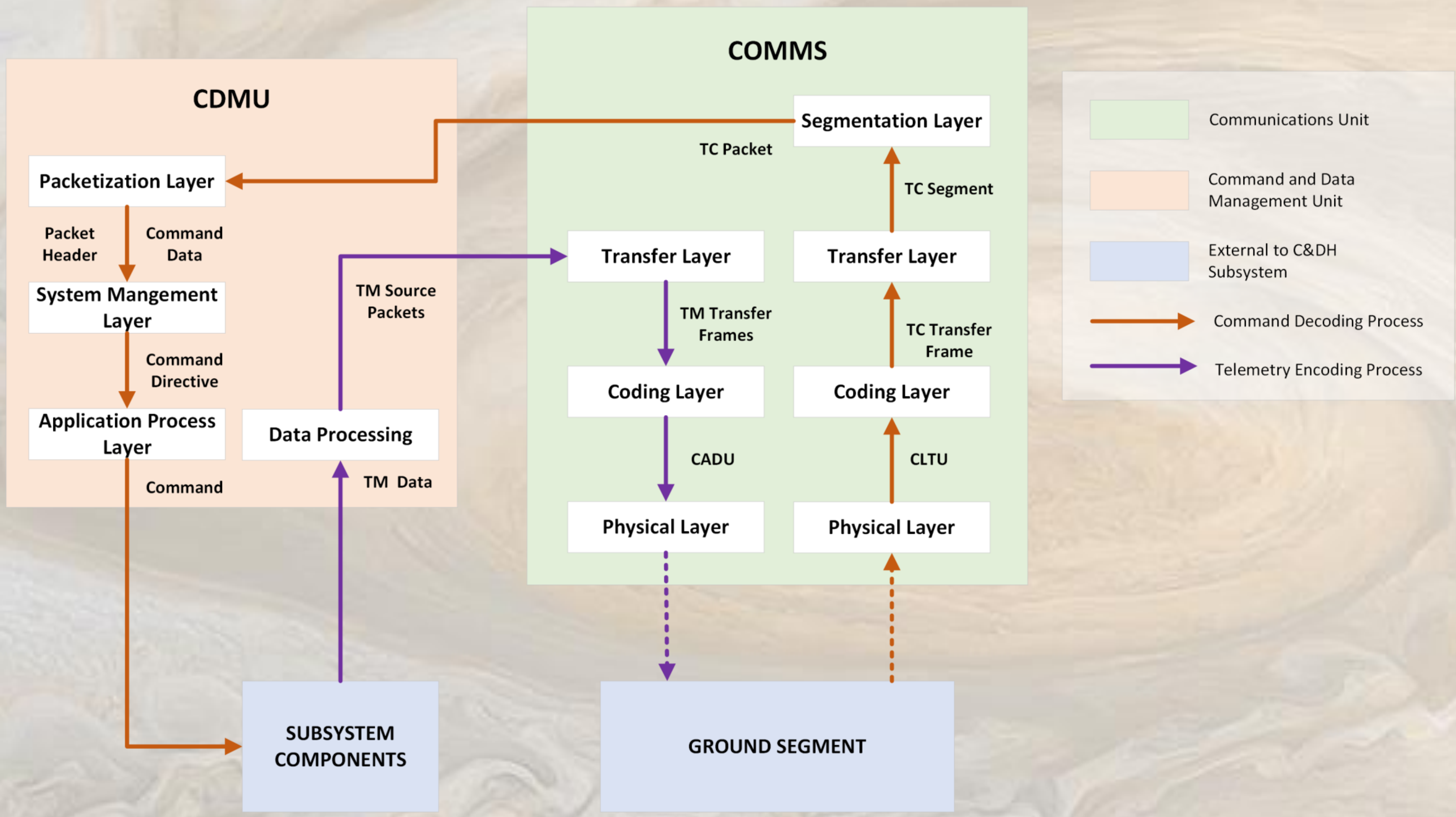
VDU - Video Data Unit		
Attached Sync. Marker	Data Space	Check Symbols
4 byte	1530 byte	767 byte

TM Transfer Frame		
Frame Header	Frame Data Field	Frame Trailer
6 byte	1520 byte	4 byte

TM Source Packet			
Packet Header	Packet Data Field		
	Data Field Header	Imager Data	Packet Error Control
6 byte	12 byte	1500 byte	2 byte

Encoding/Decoding Process

Command & Data Handling



Electrical Power System

Josh Zielinski & Marissa Harrison

Payload Data Acquisition Power

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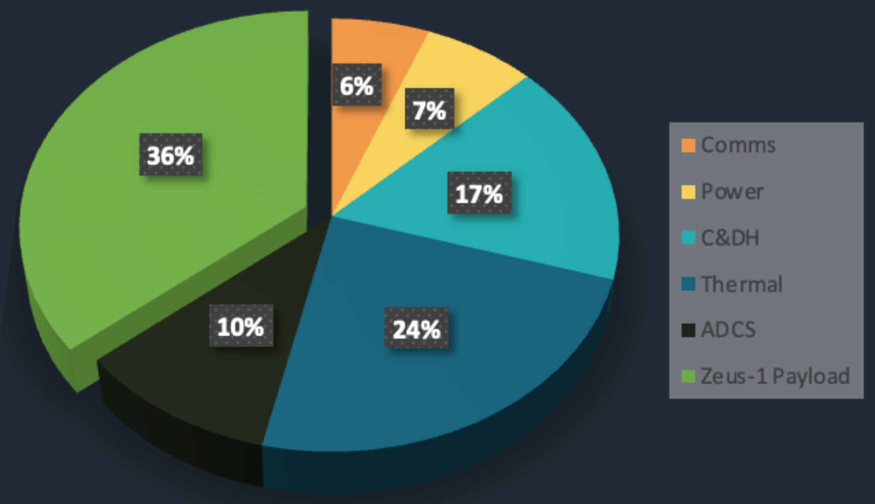
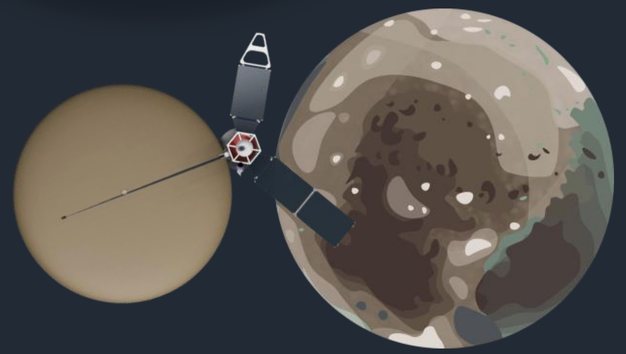
Electrical Power System

Moon:	Callisto	Ganymede	Europa	Io
Minutes in range:	38	35	5	4
Comm Required Power (W):	25	25	25	25
EPS Required Power (W):	29	29	32	32
C&DH Required Power (W):	70	70	70	70
TCS Required Power (W):	100	100	90	90
AOCS Required Power (W):	44	44	74	74
Zeus-1 Required Power (W):	150	150	150	150
Total Power + 20% (W):	**502**	**502**	**530**	**530**
Total Power + 20% (W-hr):	**311**	**287**	**45**	**36**
BOL RECHARGE TIME (Min):	**32**	**29**	**6**	**5**
EOL RECHARGE TIME (Min):	**167**	**154**	**26**	**21**

*Batteries will fully recharge before entering downlink

Power Distribution During Payload Data Acquisition (PDA)



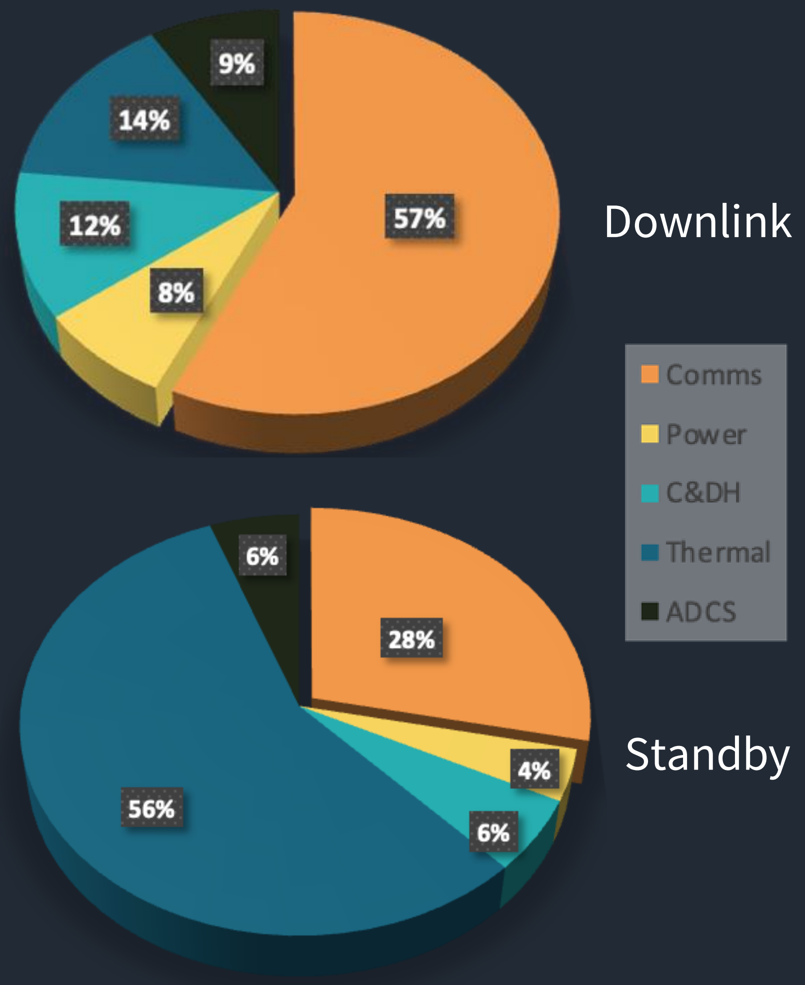


Transmission Cycle Power

Electrical Power System

Modes:	Downlink (20%)	Standby (80%)
Comm (W)	195	50
Power (W)	27	7
C&DH (W)	40	10
Thermal (W)	50	100
AOCS (W)	30	10
Zeus-1 Payload (W)	0	0
Total (W)	342	177
TOTAL + 20% (W):	**410**	**213**

Total Average Power Consumption + 20% Margin: **252 W**



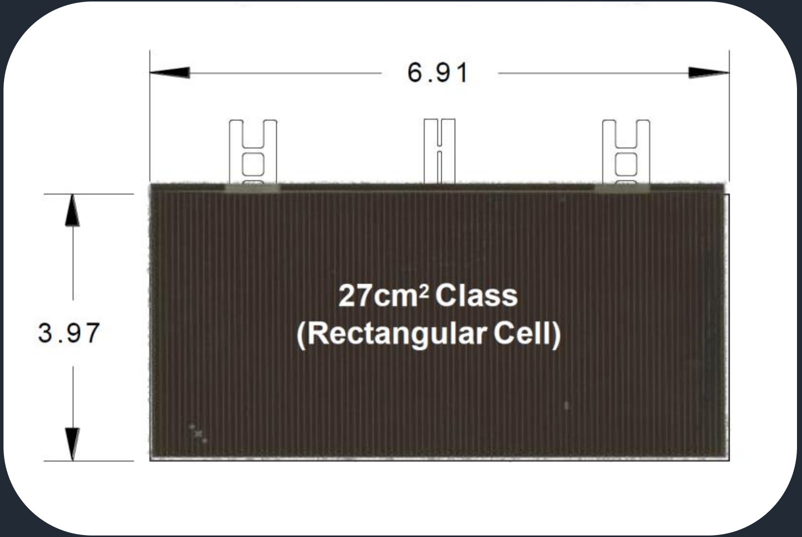
Solar Array Sizing

Electrical Power System

Solar Cell Efficiency at BOL (%)	37.0
Efficiency Degradation Rate (%/yr)	2.0
EOL Power Generation Efficiency (%)	26.5

$$A_{array} = \frac{P_{avg}}{flux \times Eff_{cell} \times Eff_{line} \times Eff_{packing}}$$

Total Solar Array Area (m^2)	20.75
Total Solar Array Mass (kg)	98
Total BOL Power Output (W)	371
Total EOL Power Output (W)	266



SpectroLab XTE-LILT Solar Cell[1]

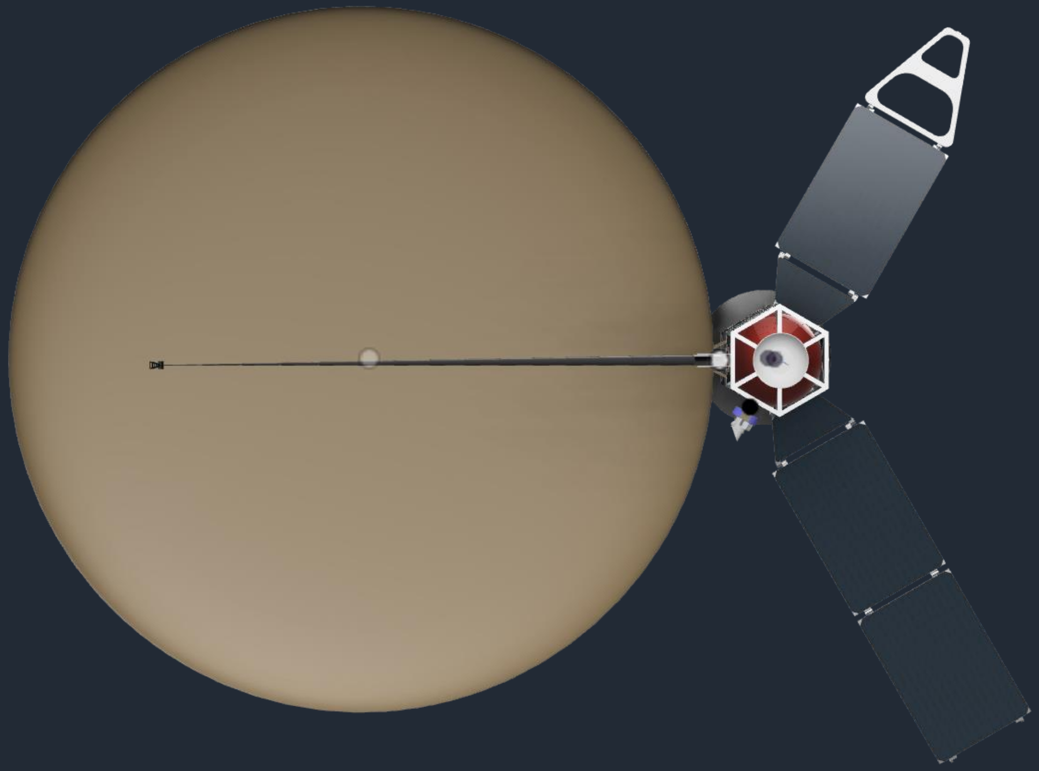
Sizing based on Fold-Out *SpectroLab XTE-LILT* (Gallium Arsenide)

- High power density of 18.6 W/m^2 at BOL
- Advanced radiation hardening
- Optimized for low light, low temperature conditions

Solar Array Configurations

Electrical Power System

Arrays Stowed for Launch



Deployed Solar Arrays

Solar Array Configuration:	Deployed	Stowed
Total Solar Array Area (m^2)	20.75	5.21
Total BOL Power Output	370	93
Total EOL Power Output	265	67

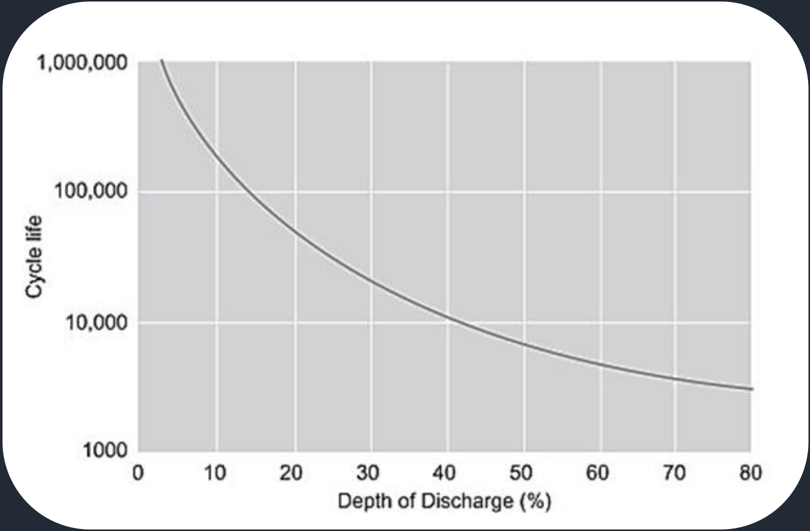
Mission-Oriented Battery Selection

Electrical Power System

Ideal Battery Properties
- High energy density
- Capable of >5500 cycles
- Long charge shelf-life
- Spaceflight heritage

Lithium-Ion Battery
- >100 W-hr/kg energy density
- Capable of >10,000 cycles when DoD is limited
- Only ~1-2% per month self-discharge rate
- Near-Earth and interplanetary heritage beginning in 2004

EaglePicher LP 33165 Lithium-Ion Battery[2]

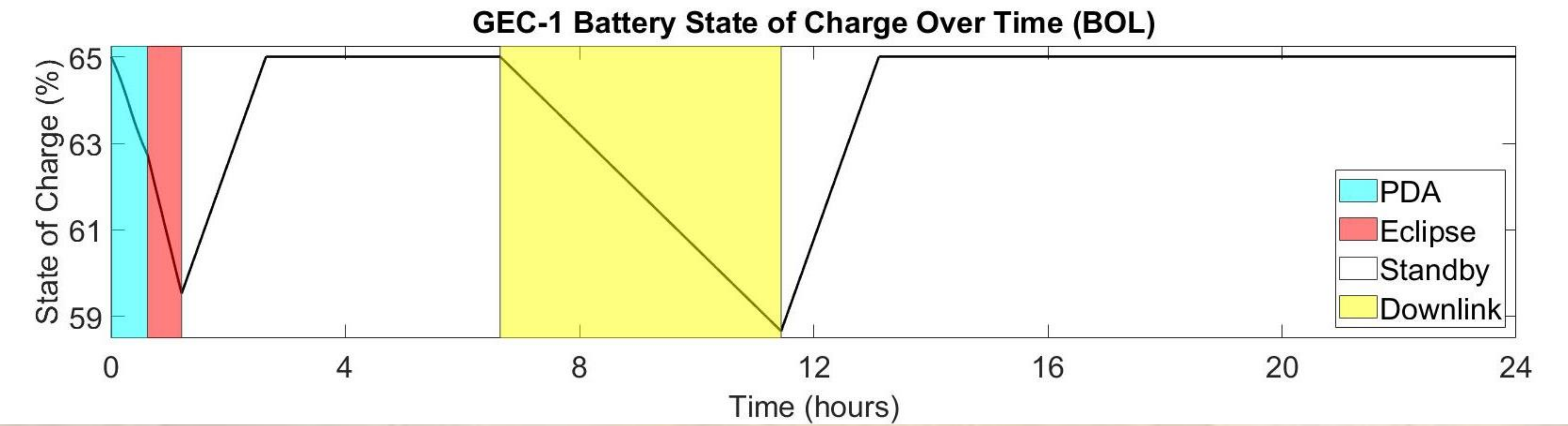


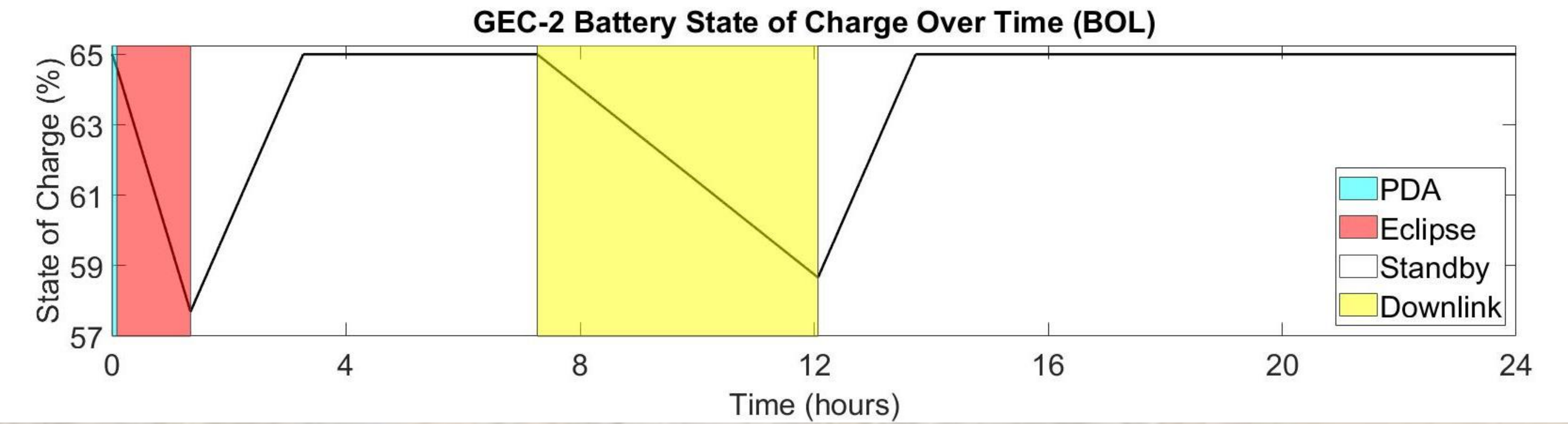
Cycle Life vs. Depth of Discharge for a Typical Lithium-Ion Battery[3]

EaglePicher LP 33165 Lithium-Ion Battery
- 109 W-hr/kg energy density, mass of 18 kg each
- Rated for >40,000 cycles at 40% DoD
- Jovian flight heritage (Juno and MAVEN)

Battery State of Charge (BOL)

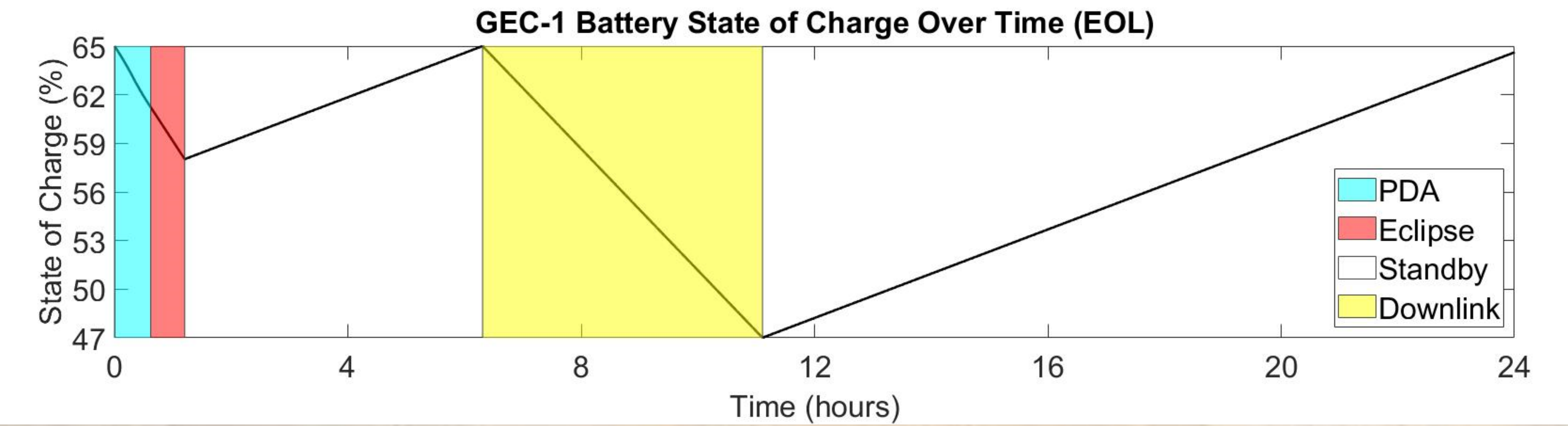
Electrical Power System

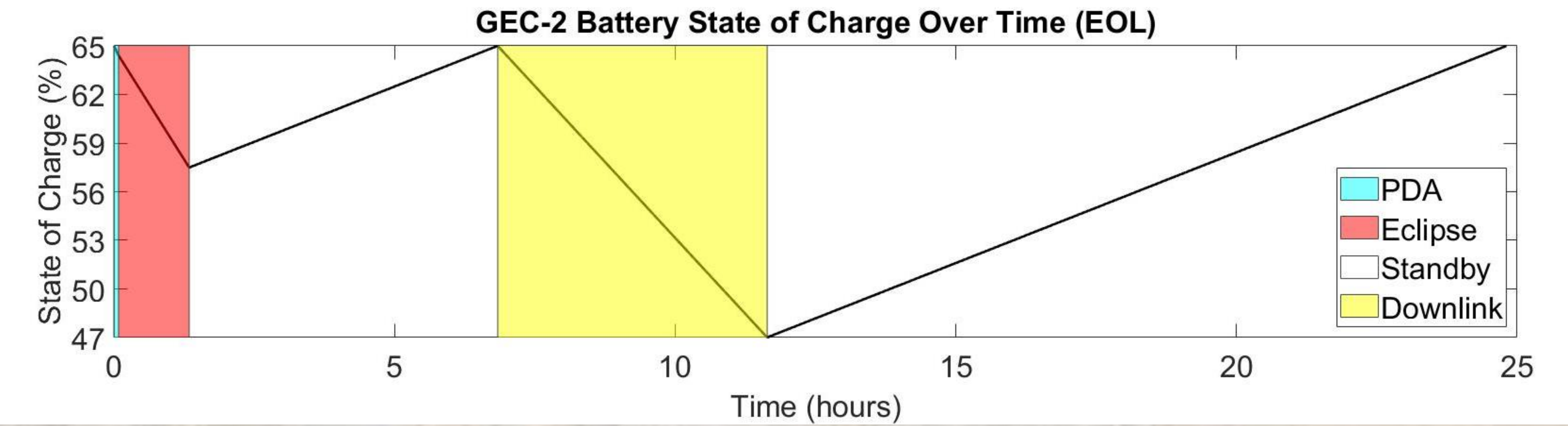




Battery State of Charge (EOL)

Electrical Power System





Downlink Operations System Load

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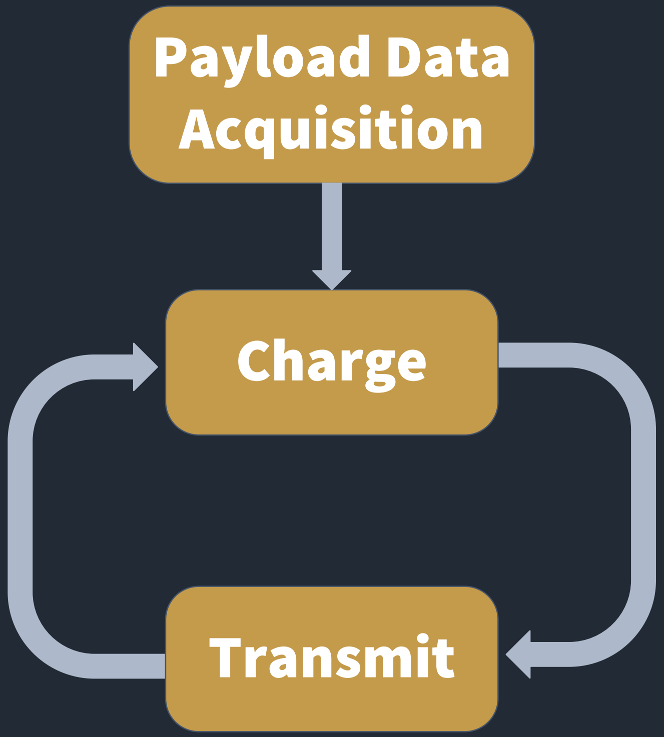
Electrical Power System

Mode:	Downlink	Standby
Duty Cycle (% Per Day):	20	80
Power Used (W-hr):	1968	4079
Power Charged (BOL) (W-hr):	1777	7105
Power Charged (EOL) (W-hr):	1273	5091
Total Power Change (BOL) (W-hr):	-191	3027
Total Power Change (EOL) (W-hr):	-695	1013
POWER REMAINING (W-hr):	447	1513



The spacecraft's power system must comfortably cycle a total power flux of **~1038 Watt-Hours** per day for 15 years (**~5500 cycles**)

Battery Sizing

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Electrical Power System

Target DoD	30	%
Total Line Losses	5	%
Total Energy Capacity Required	695	W-hr
Capacity Required (100% DOD)	2315	W-hr
Mass Per Battery	18	kg
Energy Density	109	W-hr/kg
Capacity Per Battery (100% DoD)	1962	W-hr
Total Battery Capacity (100% DoD)	3924	W-hr
Usable Battery Capacity (Target DoD)	1177	W-hr

Batteries: **2**

Recharge Time (BOL-EOL):

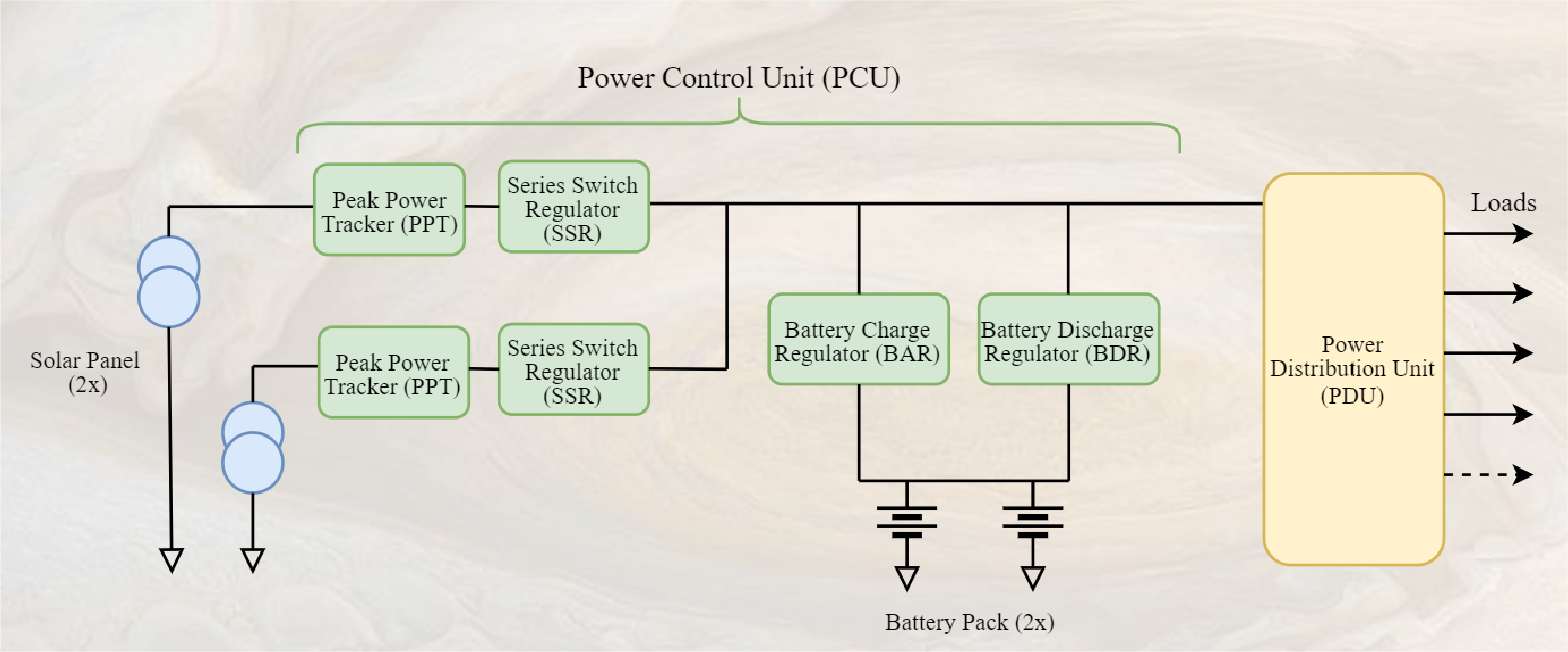
- **4.4-13.2 Hours**

Reserve Power After Target DoD:

- **3230 Watt-Hours**

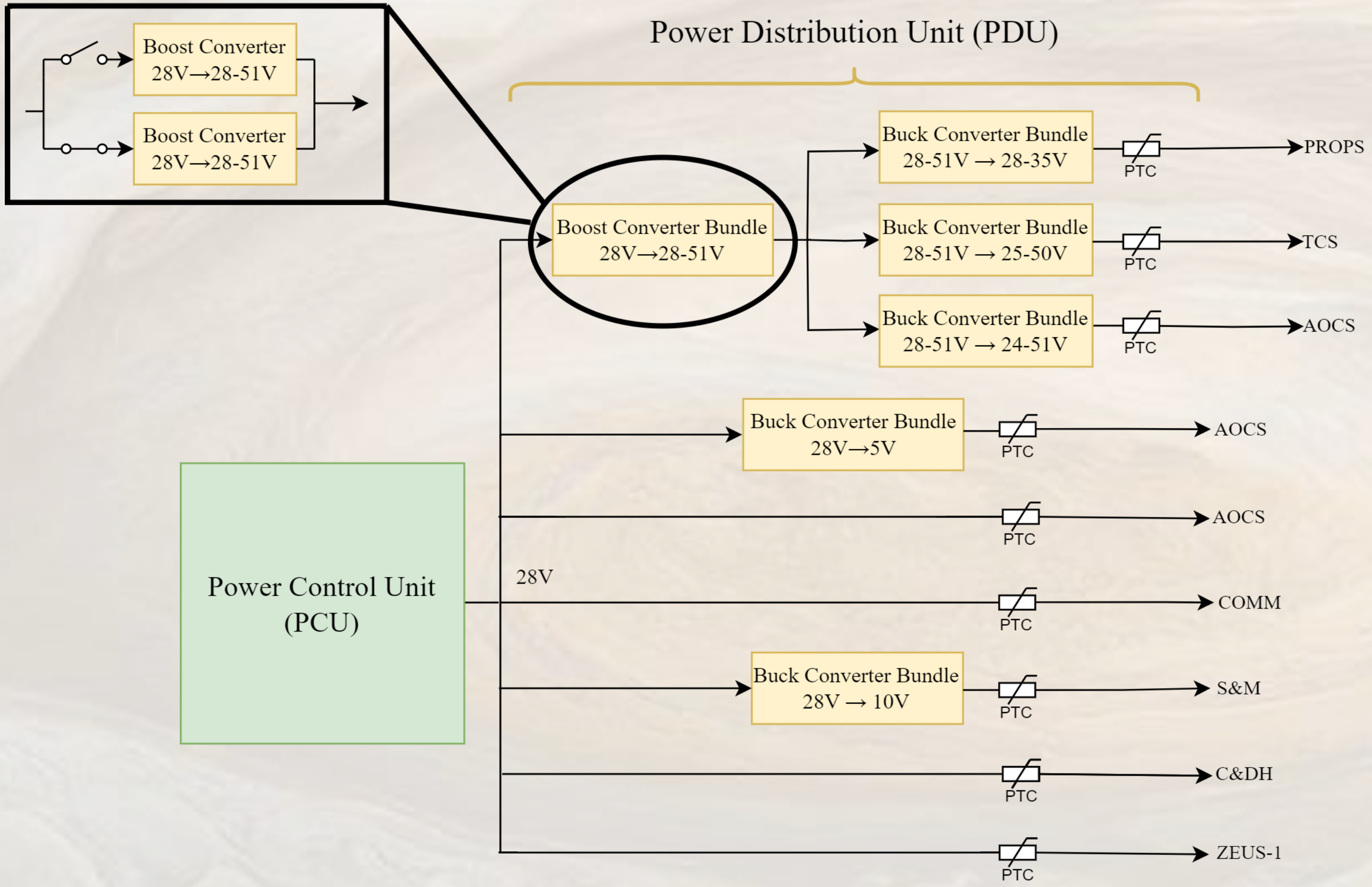
Power Regulation and Distribution

Electrical Power System



Power Distribution System

Electrical Power System



Structures & Mechanisms

Roman Rosser

Driving Components

Structures & Mechanisms

Driving Component	Approximate Size	Mass (kg)	Location Consideration
Zeus-1 payload	0.25 m^3	20	Moon-facing during imaging windows (gimbaled)
Solar arrays[1] (2)	10 m^2 each	94	Sun-facing
High gain reflector[2]	⌀13.5 m	53	Earth-facing during relevant coms windows
Medium gain antenna	⌀ 1 m	20	Earth-facing during relevant coms windows (gimbaled)
Spherical/Capsule-shaped Propellant tanks[4] (3)	⌀ 1.76 m	305	Located on center axis to minimize effects of change in mass
Radiation vault	47 x 37 x 24 cm	41	In between prop tanks for thermal control

[1]Mass reflects total mass of both arrays and cover glass; size is deployed area

[2]Size is deployed area

[3]Mass reflects total mass of two battery packs

[4]Mass reflects total mass of fuel, oxidizer, and inert gas tanks

Material Considerations

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Structures & Mechanisms

Component	Materials Considered	Material Selection	Reasoning
Bus Frame	Al, Al alloy, composites	Al-7075	Mass and durability considerations, flight heritage
Solar arrays (2)	Gallium Arsenide (GaAs), Si	GaAs	Flight heritage (Juno), efficiency and lower degradation rate
High gain reflector	AstroMesh	AstroMesh	Best option for HGA, material selected by manufacturer
Medium gain antenna	Gold-plated tungsten wire	Gold-plated tungsten wire	Best option for MGA, material selected by manufacturer
Battery packs (2)	Lithium-ion	Lithium-ion	Flight heritage (Juno), able to withstand radiation environment
Propellant tanks (2)	Ti alloy	Ti-6Al-4V	Mass and durability considerations
Radiation vault	Ti, Al alloy	Ti	Flight heritage (Juno)

Mass Considerations for GEC-2

Structures & Mechanisms

Subsystem	Mass Estimate (kg)	MGA (%)	MGA (kg)	Mass Estimate + MGA (kg)
Payloads	20.0	5	1.0	21.0
S&M	520.7	5	26.0	546.8
TCS	120.3	5	6.0	126.4
EPS**	182.9	5	9.1	192.0
Coms	135.3	5	6.8	142.1
C&DH	11.0	5	0.6	11.6
AOCS	94.3	5	4.7	99.1
Prop	376.0	5	18.8	394.8
Propellant	7250.0	--	--	7250.0
Wet Mass	**8710.5**	**--**	73.0	**8783.8**

** EPS mass includes all subsystem harnessing

We are designing to GEC-2's requirements to minimize the overall spacecraft mass

CoM and Moment of Inertia

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Structures & Mechanisms

- The Center of Mass and Moment of Inertia were calculated in Fusion360 using the full spacecraft assembly. These values are accurate for the spacecraft in the deployed configuration with the maximum propellant mass. These values will change throughout the mission as propellant is consumed.

CoM Location [m]

(0.029, -0.022, 2.858)

*relative to the contact plane between the launch vehicle and S/C

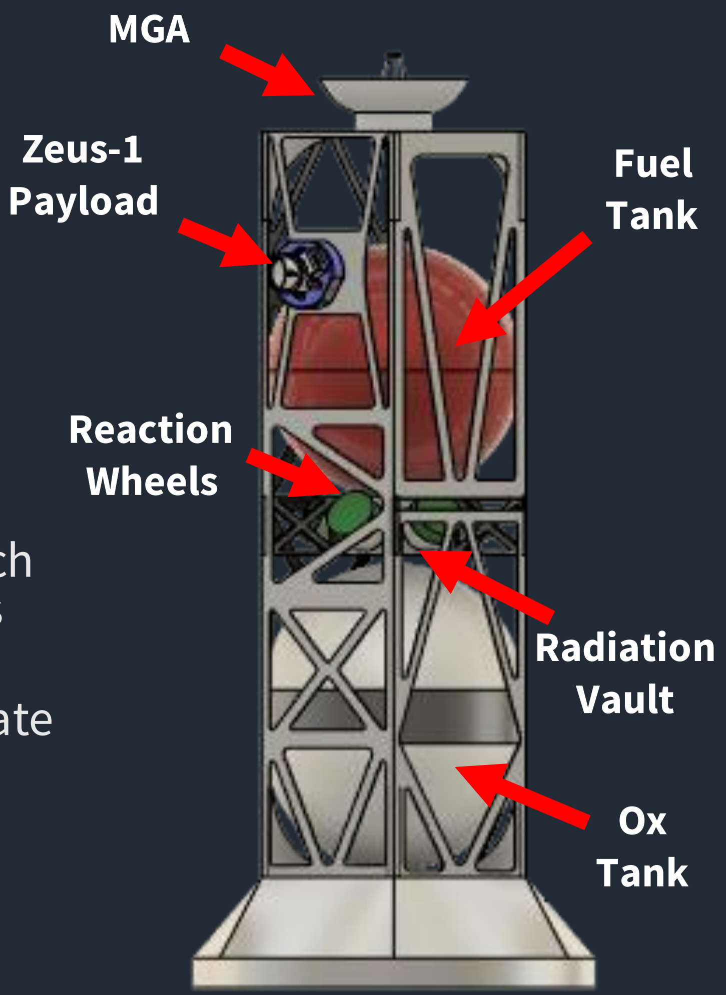
Inertia Matrix [kg*m^2]

$2.077*10^4$	$1.146*10^3$	$-3.854*10^2$
$1.146*10^3$	$2.071*10^4$	$1.146*10^3$
$-3.854*10^2$	$1.146*10^3$	$1.296*10^4$

Component Placement

Structures & Mechanisms

- Zeus-1 Payload
  - Cannot be on the same face as thrusters due to outgassing and contamination
  - Moon-facing as much as possible, particularly during imaging windows
  - On a bi-axial gimbal to allow for free movement and control of pointing
- Propulsion
  - Thrusters must be placed about central vertical-axis of the spacecraft such that a burn does not create any moments due to changing center of mass from emptying propellant tanks
  - Radiation vault will be housed between propellant tanks for most adequate thermal control options
- Medium Gain Antenna (MGA)
  - Must be Earth-pointing during relevant communication windows
  - On bi-axial gimbal to allow for the solar array to be sun-pointing



Component Placement

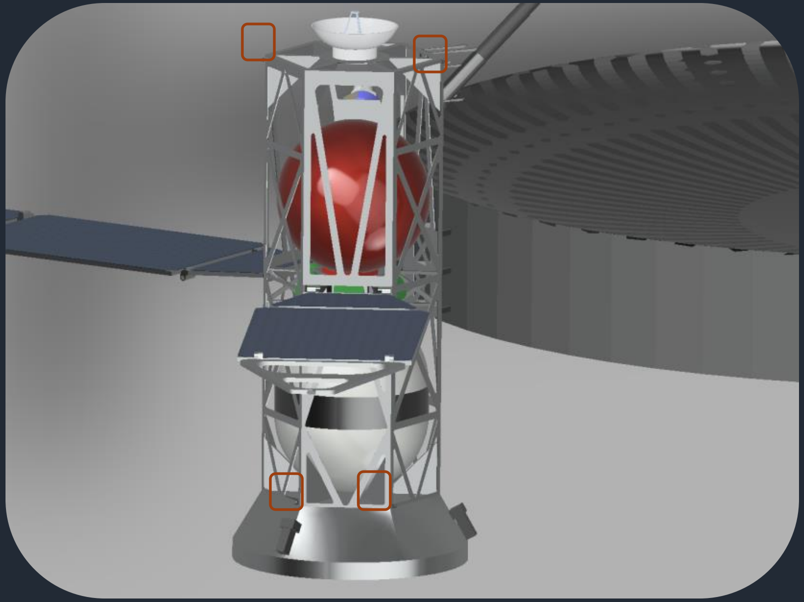
Structures & Mechanisms

- High Gain Antenna (HGA)
  - Must be earth-pointing during relevant communication windows
  - 0.05° pointing accuracy requirement from communications link budget
  - AOCS will be capable of adequately pointing the reflector during communications windows – the reflector will be rigidly mounted to spacecraft structure
- On Board Computers (OBC)
  - Must be shielded from radiation and charged particle environment of the Jovian System
  - OBC and other sensitive components will be located inside the radiation vault for maximum possible protection
- Solar Array
  - Must be sun-pointing and in a location with minimal outgassing risk
  - Will deploy via pin-release
  - Does not require gimbaling – the fixed HGA will often be pointed towards Earth, so the solar panels would be facing the sun with a maximum of 12° incidence

Component Placement

Structures & Mechanisms

- Thruster Cluster
  - Placed in corners of the spacecraft to maximize torque.
  - Each cluster has a set of three thrusters
- Star Tracker
  - Need to face opposite of the sun.
  - Since the antennas and solar arrays will mostly be sun-facing, the star trackers are facing in the opposite directions.
- Reaction Wheel System
  - Must be close to the center of mass, so it was placed between the oxidizer and fuel.
  - 6 wheels on a flat plane facing 30 degrees upward



Thruster Cluster Locations

Launch Environment

Structures & Mechanisms

- The NASA GEVS Test Factors/Durations were used to determine test loads for the Protoflight Qualification Level based on Falcon Heavy's User Guide
- SpaceX recommends that payloads have a primary lateral frequency of >10 Hz and a primary axial frequency of >25 Hz in order to avoid interaction with launch vehicle dynamics[1]
  - Secondary structures should have minimum resonant frequencies > 35 Hz

Load Type	Falcon Heavy Limit Level [4]	Protoflight Test Level	Test Duration or Sweep Rate
Acoustic	132.2 dB	135.2 dB	1 min
Random Vibration	5.13 G_{RMS}	7.26 G_{RMS}	1 min/axis
Sine Vibration	0.5-0.9 g's axially, 0.5-0.6 g's laterally	1.125 g's axially, 0.75 g's laterally	4 oct/min
Shock	30-1000 g's from 100-10000 Hz	1400 g's	2 actuations 1 x each axis

Table 2.2-2
Test Factors/Durations

Test	Prototype Qualification	Protoflight Qualification	Acceptance
Structural Loads[1] Level	1.25 x Limit Load	1.25 x Limit Load	1.0 x Limit Load
Duration Centrifuge/Static Load[6]	1 minute	30 seconds	30 seconds
Sine Burst	5 cycles @ full level per axis	5 cycles @ full level per axis	5 cycles @ full level per axis
Acoustics Level[2] Duration	Limit Level + 3dB 2 minutes	Limit Level + 3dB 1 minute	Limit Level 1 minute
Random Vibration Level[2] Duration	Limit Level + 3dB 2 minutes/axis	Limit Level + 3dB 1 minute/axis	Limit Level 1 minute/axis
Sine Vibration[3] Level Sweep Rate	1.25 x Limit Level 2 oct/min	1.25 x Limit Level 4 oct/min	Limit Level 4 oct/min
Mechanical Shock Actual Device Simulated	2 actuations 1.4 x Limit Level 2 x Each Axis	2 actuations 1.4 x Limit Level 1 x Each Axis	1 actuations Limit Level 1 x Each Axis
Thermal-Vacuum	Max./min. predict. ± 10°C	Max./min. predict. ± 10°C	Max./min. predict. ± 5°C
Thermal Cycling[4,5]	Max./min. predict. ± 25°C	Max./min. predict. ± 25°C	Max./min. predict. ± 20°C
EMC & Magnetics	As Specified for Mission	Same	Same

NASA GEVS Test Factors/Durations [1]

Propellant Tank Sizing

Structures & Mechanisms

- The propellant tanks must contain 7250 kg of propellant
- Fuel tank has spherical shape and Oxidizer tank is pill shaped with the same radius as the fuel tank.
- Used a factor of safety of 2.0 for buckling. [1]
- Assuming an ullage margin of 10% for tank volume [2]
- Propellant tanks are separated to reduce change in CoM throughout mission, therefor a common bulkhead could not be used.
- Material selected is Titanium (Ti-6Al-4V), which has a high ratio of Ultimate Strength to density and is compatible with the propellants selected. Metallic Materials Properties Development and Standardization was used to compare options.

Propellant Tank Sizing

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Structures & Mechanisms

	Oxidizer Tank	Fuel Tank
Propellant Type	NTO	MMH
Propellant Mass [kg]	4955	2294
Tank Material	Ti-6Al-4V	Ti-6Al-4V
Tank Pressure [MPa]	4955	4955
Tank Thickness [mm]	2.20	2.20
Tank Diameter [m]	0.88	0.88
Tank Height [m]	(Spherical)	2.143
Tank Structural Mass [kg]	162	142

Attitude & Orbit Control System

Josh Zolkewitz & Amogha Sarang

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AOCS Performance Overview

Attitude & Orbit Control System

General

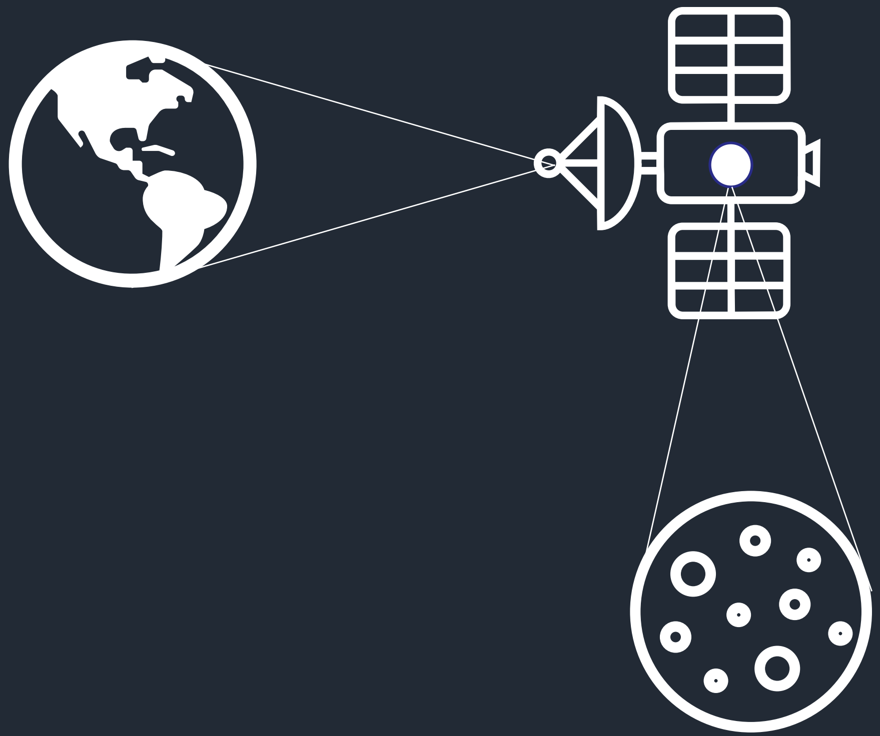
- GEC-1 & GEC-2 will have nominal pointing for each stage
- Autonomous operations only occur when no connection to Earth
  - DSMs, Jovian Tour Burns, Errors, and Eclipse

Interplanetary

- Default spacecraft pointing is at Sun
- Spacecraft pointing changes for burns and MGA
- MGA will use gimbal for finer connection to Earth

Jovian System

- Spacecrafts will switch to Earth-pointing except for burns
  - Attitude rotation to point HGA towards Earth
- Gimbal will point Zeus-1 payload

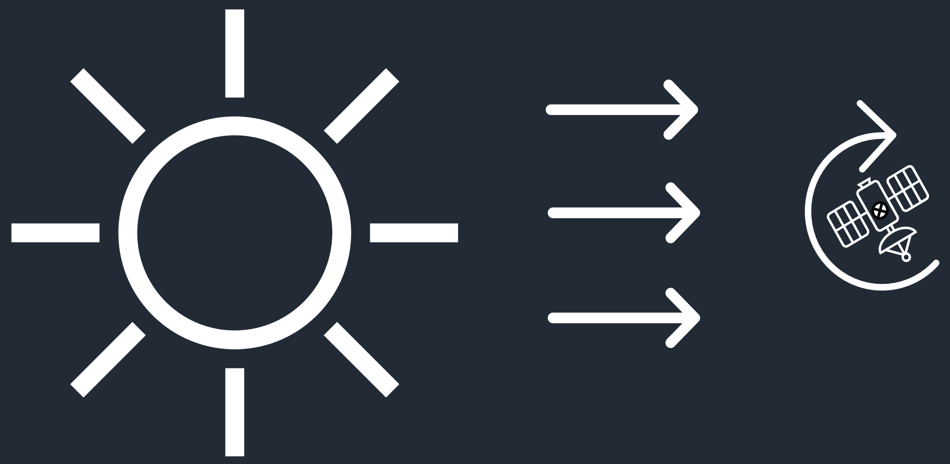


Interplanetary Disturbances

Attitude & Orbit Control System

Solar Radiation Pressure

- Peak Magnitude: **~5 x 10^{-3} N-m**
- Occurs at near Earth operations



- All other torques peak briefly during the gravity assist and are negligible due to closest approach altitude being **40,900 km**

Jovian System Disturbances

Attitude & Orbit Control System

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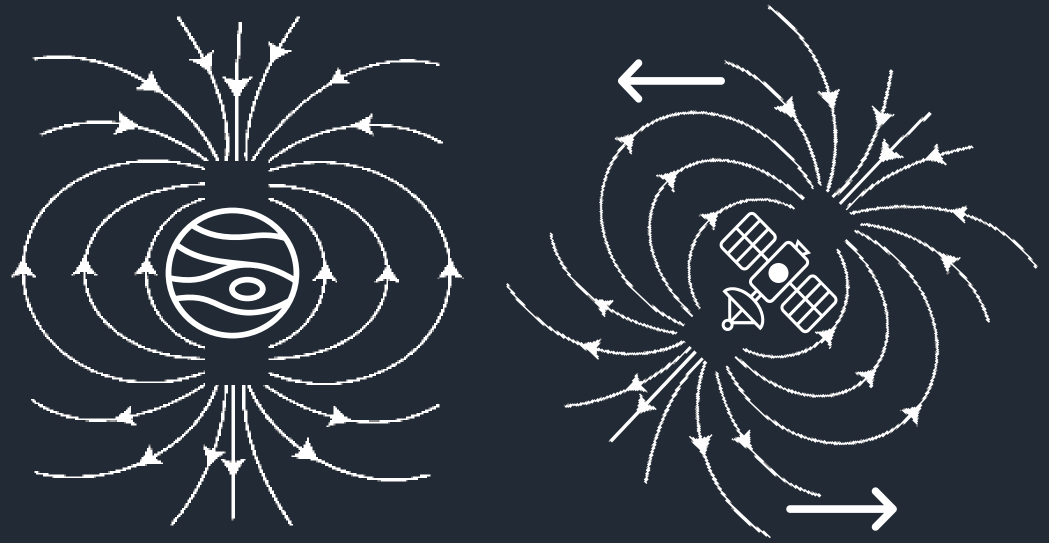
Magnetic

- Peak magnitude: $\sim\mathbf{6.0 \times 10^{-3}\ N-m}$
- Simulated using the VIP4 model
- Current design has very large conductive areas, residual magnetic moment needs to be minimized



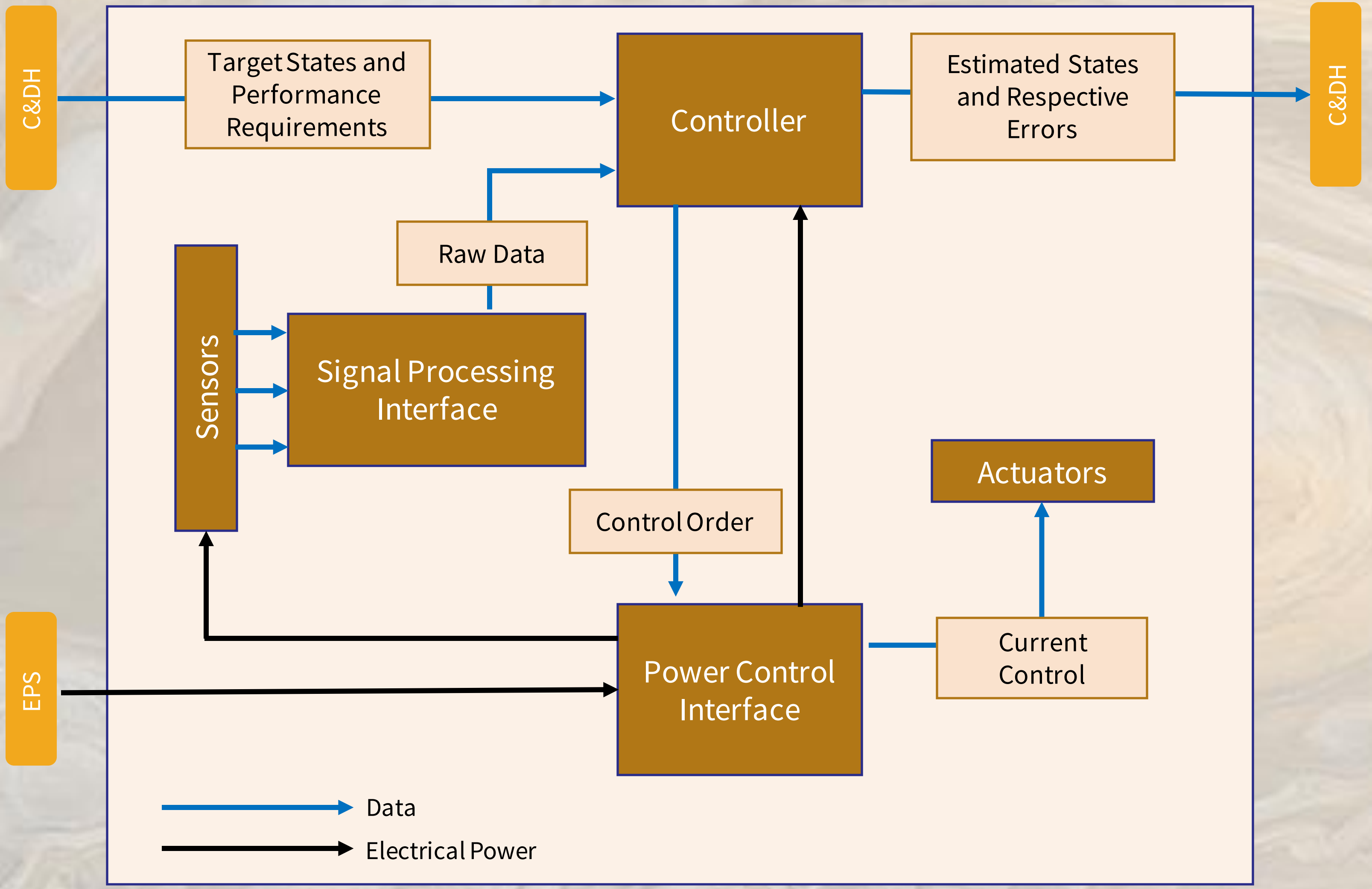
Gravity gradient

- Peak magnitude: $\sim\mathbf{2.0 \times 10^{-3}\ N-m}$
- Increased due to large spacecraft size
- Reduced by large distance from planetary masses

The estimated average disturbance torque for the mission is **1.4 mN-m.**

AOCS Subsystem Interaction

Attitude & Orbit Control System



Pointing Needs

Attitude & Orbit Control System

Component	Pointing Angle	Actuation Method
High Gain Antenna	±0.05°	Spacecraft Attitude
Medium Gain Antenna	±0.7°	Gimbal
Zeus-1 Payload *	±0.001-0.1°	Spacecraft Attitude & Gimbal
Main Engines **	±0.05°	Spacecraft Attitude
Solar Panels	±12°	Spacecraft Attitude

- Maximum Attitude Pointing Need: **HGA (±0.05°)**
- Maximum Gimbal Pointing Need: **Zeus-1 Payload (±0.001-0.1°)**

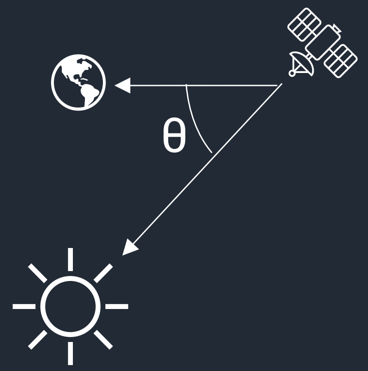
*As our performance will show, we can perform within this range, but not the lower end

**It's assumed that the pointing need is the same as our strictest orientation need (HGA)

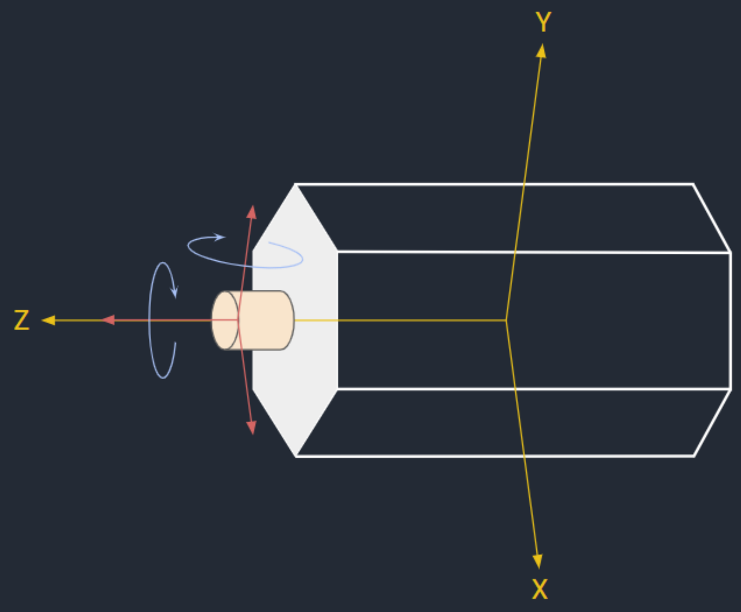
Pointing Geometry (Interplanetary)

Attitude & Orbit Control System

- Angle Between Earth and Sun will not exceed **90°**
  - We can point at Sun primarily and still maintain connection with Earth



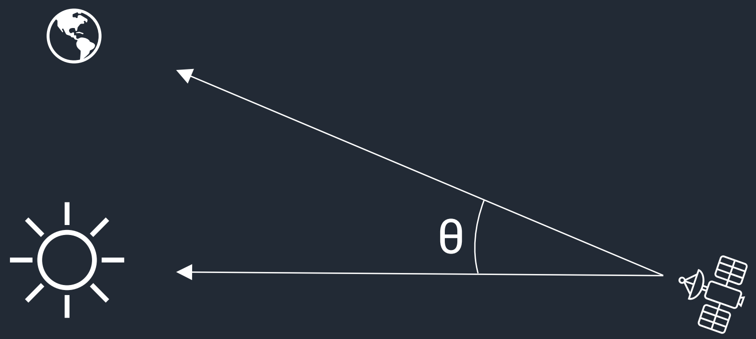
- Gimballed antenna can achieve hemispherical range of motion
  - Can point up to ~90° off of z-axis



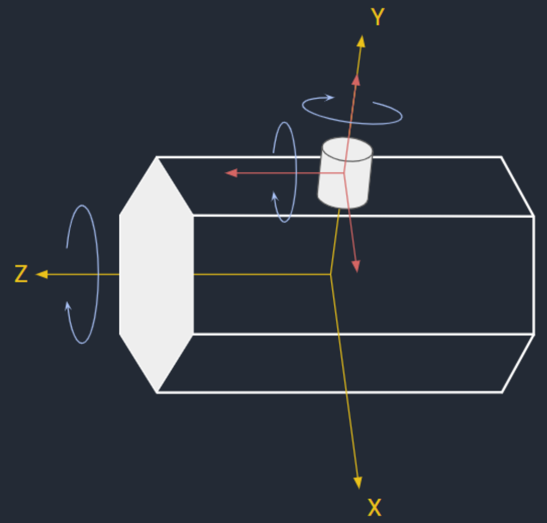
Pointing Geometry (Jovian System)

Attitude & Orbit Control System

- Max Angle Between Earth and Sun: **~10.9°**
  - We can point at Earth all the time and still meet power requirements



- Gimballed payload has near-omnidirectional field of view
  - Can't point where spacecraft is in the way
  - Spacecraft rotates about axis connecting HGA to Earth to ensure spacecraft body does not obstruct payload



Sensors Suite

Attitude & Orbit Control System

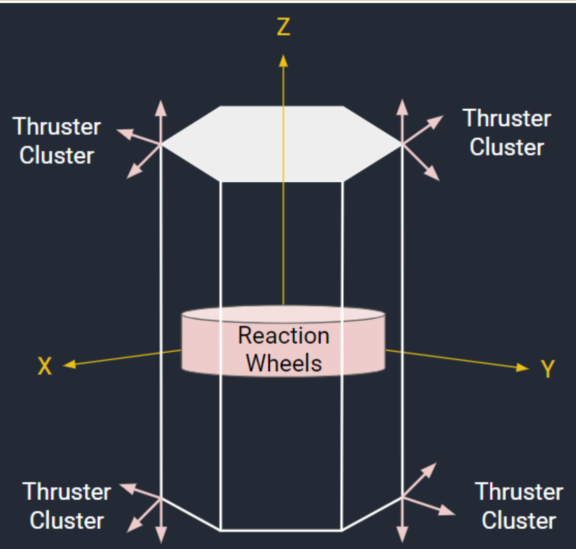
Sensor Name	# of Sensors	Subject for collecting Attitude/Trajectory Data	Accuracy
Star Tracker	3	Extrasolar Stars	11 Arcsec
Sun Sensors	12	Sun	<0.5°
Accelerometers	6	Internal mechanism measuring lateral acceleration	0.1 $mV/m/s^2$
Gyroscopes	6	Internal mechanism measuring rotational acceleration	<0.0015°/hr
Magnetometer Clusters	6	Magnetic Fields around Earth and Jupiter	<8 nt
Horizon Sensors	2	Celestial Body edges/horizons	0.015°
Radionavigation	-	Predictive calculations computed at ground station	-

Actuator Suite

Attitude & Orbit Control System

Attitude Actuation System	# of Actuators	Peak Torque	Momentum Storage
Reaction Wheels	6	0.8 N-m	61.25 Nms (per wheel)
Monopropellant Thrusters	12	2.4 N-m	N/A
Magnetorquers	6	0.2 N-m	N/A

- Also includes the 2 gimbals for finer pointing of components
- Reaction wheels are primary form of attitude control
- Magnetorquers are primary source of desaturation
- Cold Gas Thrusters are for detumble, back desaturation, and backup attitude control



Reaction Wheels

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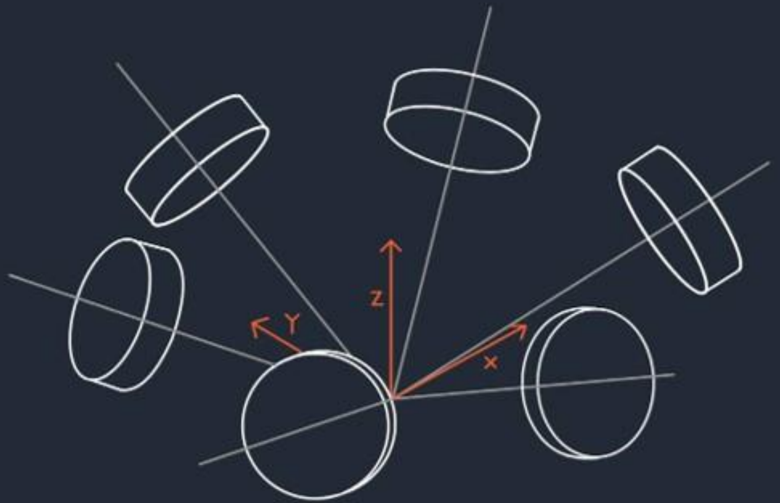
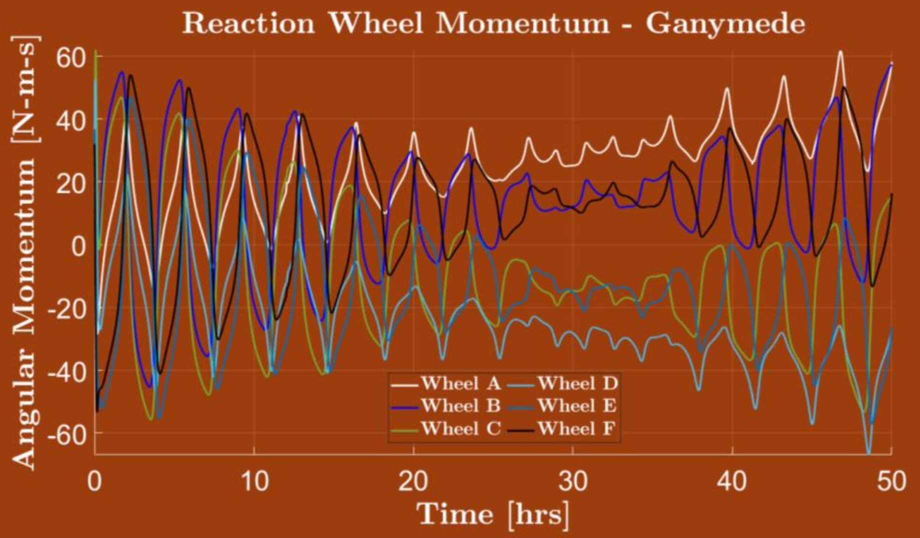
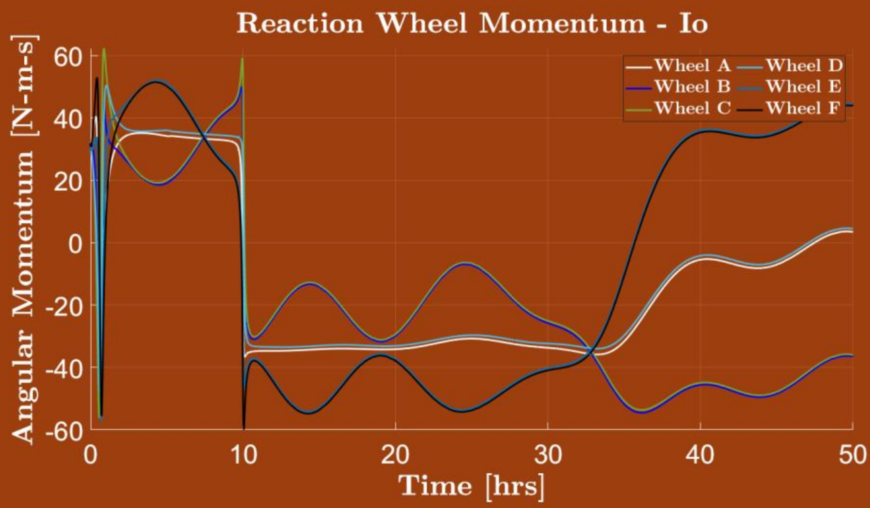
Attitude & Orbit Control System

- 6 Reaction Wheels in total
- Mass: **8.5 kg**
- Diameter: **0.35 m**
- 35.26° angle between Reaction Wheels







Gimbals

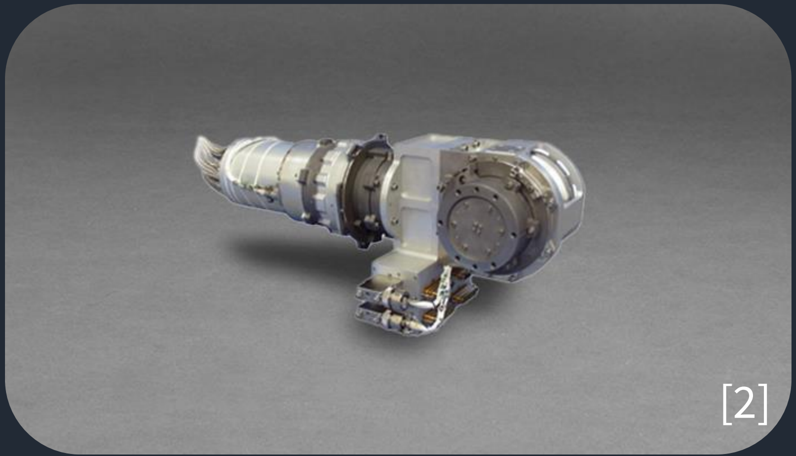
Attitude & Orbit Control System

Large Gimbal (Modelled after MOOG Type 55)

- Used for Payload
- Smaller step size (0.0075°)
- More Accurate
- Heavier Model (7.5 kg)
- Biaxial

Small Gimbal (Modelled after MOOG Type 33)

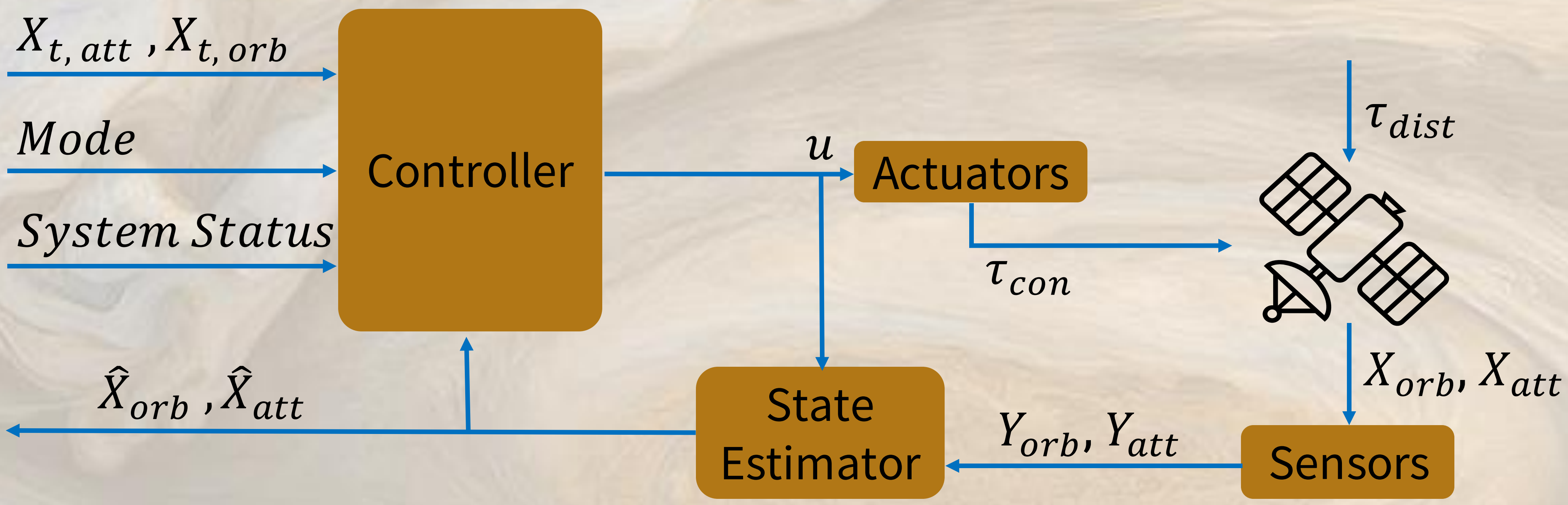
- Used for MGA
- Larger step size (0.009375°)
- Lighter Model (4.4 kg)
- Biaxial



[1] https://www.moog.com/products/space-mechanisms/antenna-pointing-mechanisms/type-55.html
[2] https://www.moog.com/products/space-mechanisms/antenna-pointing-mechanisms/type-33.html

AOCS Algorithm Block Diagram

Attitude & Orbit Control System



Legend

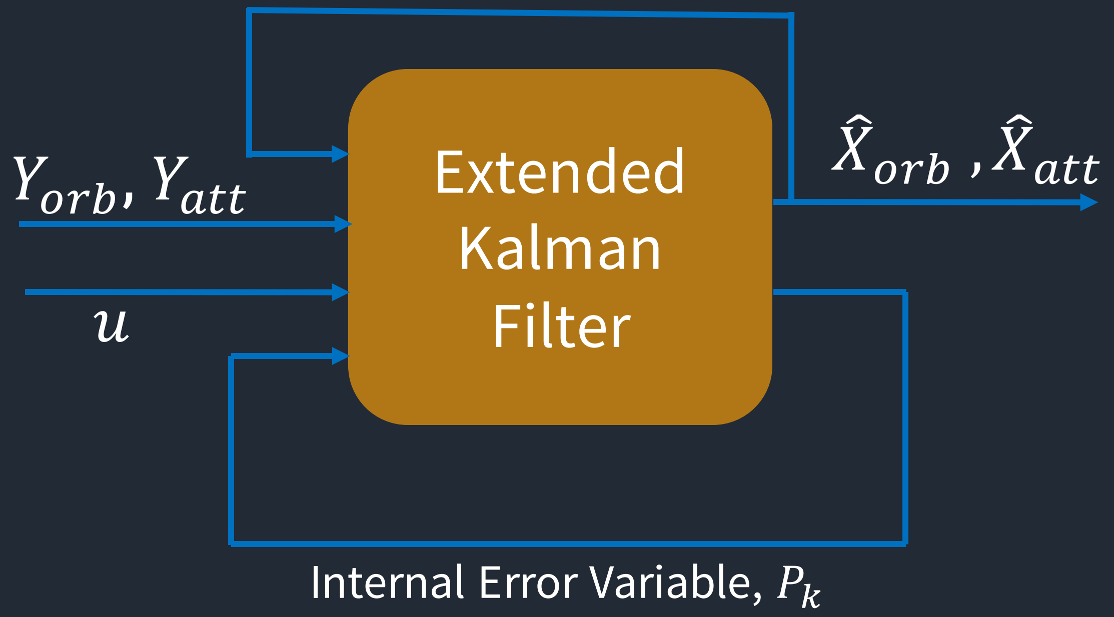
X_t — target state
$Mode$ — S/C operation mode
u — control input
τ_{con} — control torque

τ_{dist} — disturbance torque
X — true state
Y — measured state
$\hat{X}$ — estimated state

Algorithm State Estimator

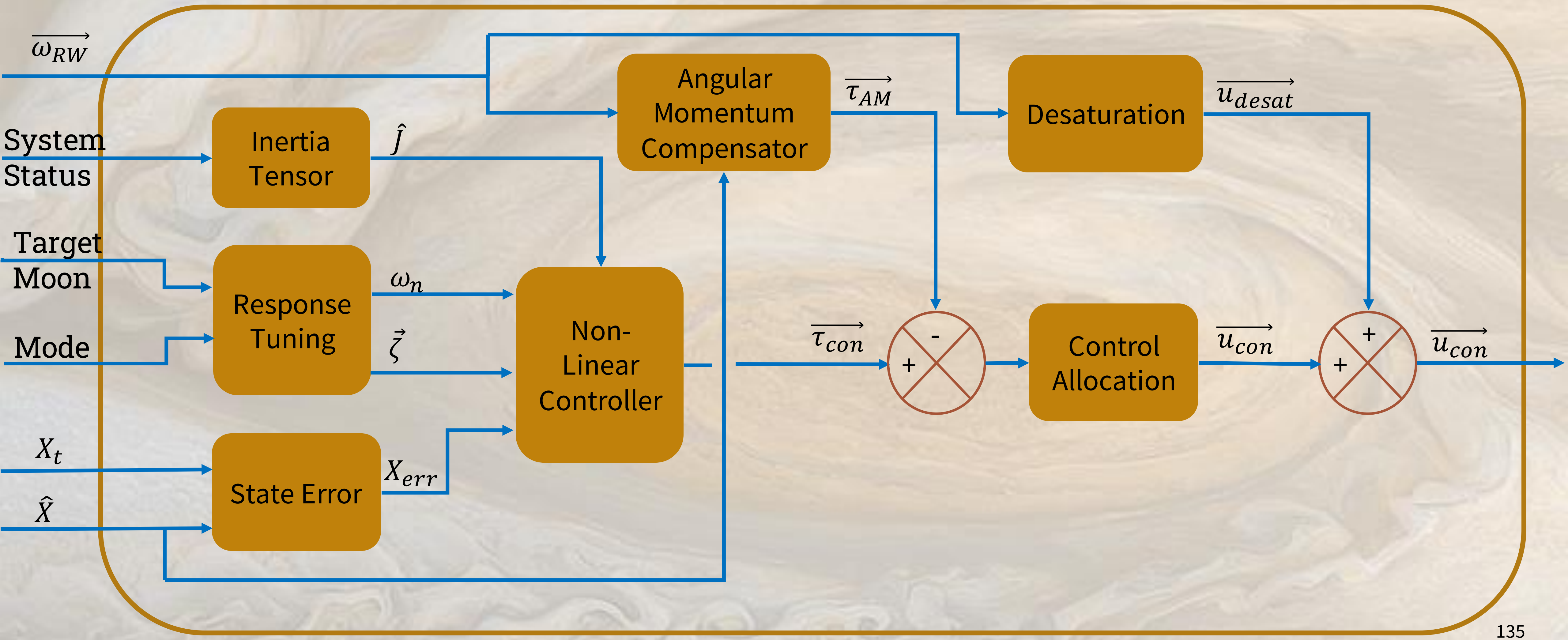
Attitude & Orbit Control System

- Current Simulink model uses actual state
- Real sensors have noise, drift, and error which need to be compensated for
- Algorithm Scheme: Extended Kalman Filter
  - Able to estimate non-linear models with noise by using prediction and correction to model the real system without noise



Controller Block Diagram

Attitude & Orbit Control System



Algorithm Overview

Attitude & Orbit Control System

$$\tau_c = -\omega^x J\omega \; - D(\omega - w_t) \; - K\;(q_{err}) + J\alpha_t \; [3]$$

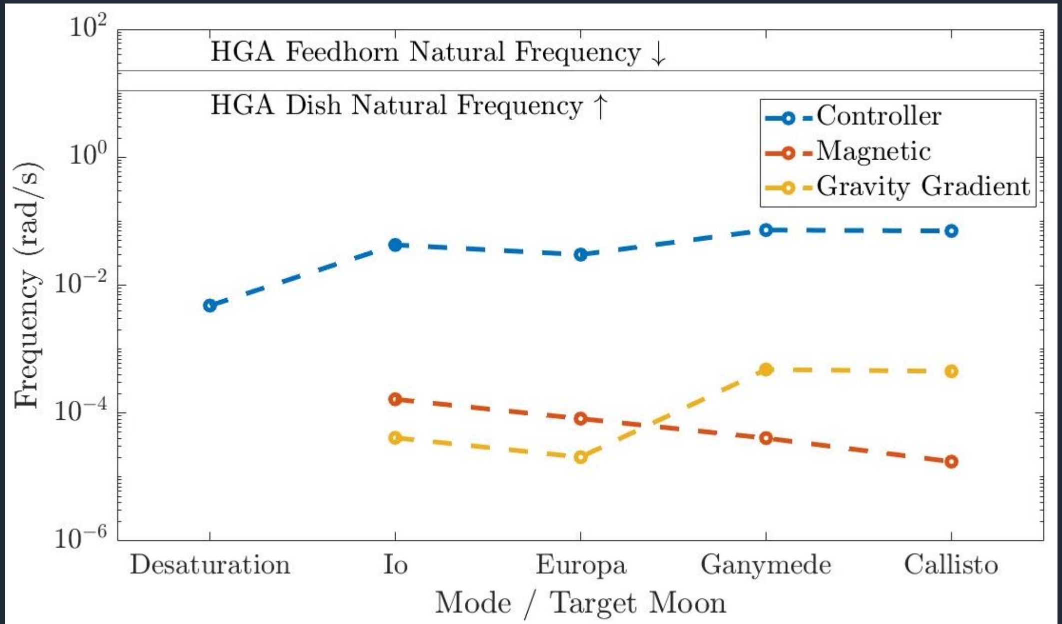
- Scheduled Non-linear controller
  - LQR broke down during fast slew due to assumptions used for linearization no longer being valid
- Drives spacecraft to target attitude, angular velocity, and angular acceleration
- Tuned by specifying a natural frequency and damping ratio for the system
- D and K are feedback matrices which are defined by user specified natural frequencies and damping ratios

Control Algorithm Tuning

Attitude & Orbit Control System

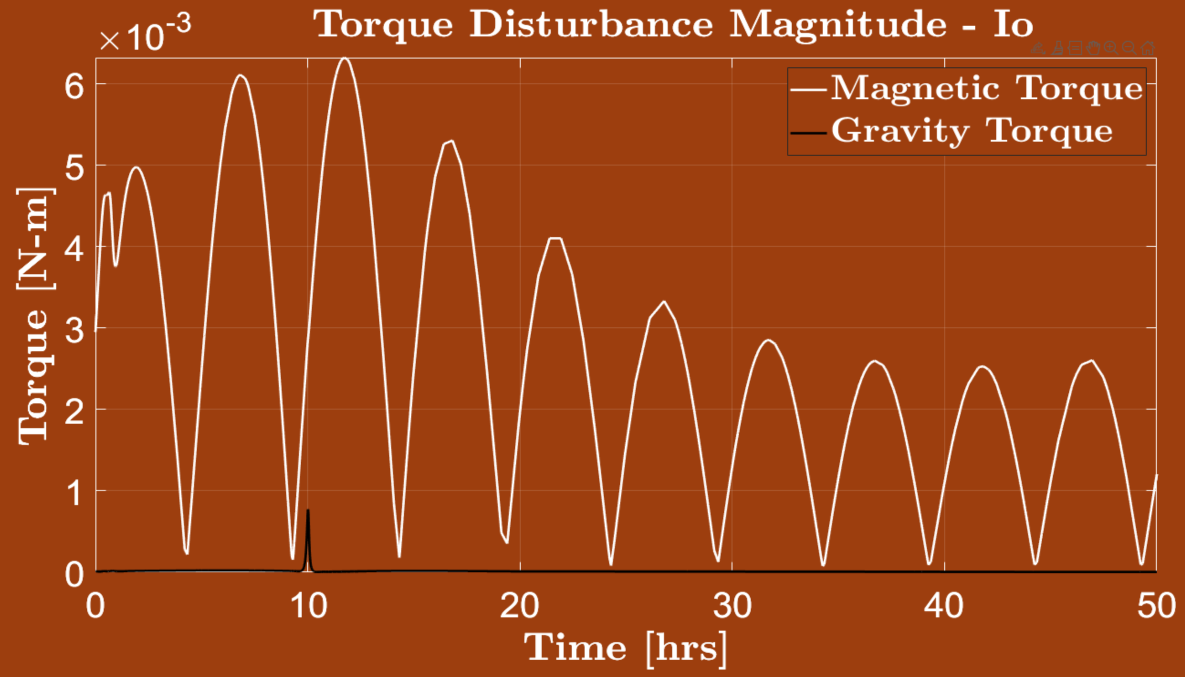
- Attitude Controller must have significantly slower frequency than the structural modes and significantly faster frequency than the disturbances.
- Controller frequency set equal to logarithmic mean of fastest disturbance and slowest structural mode as a baseline

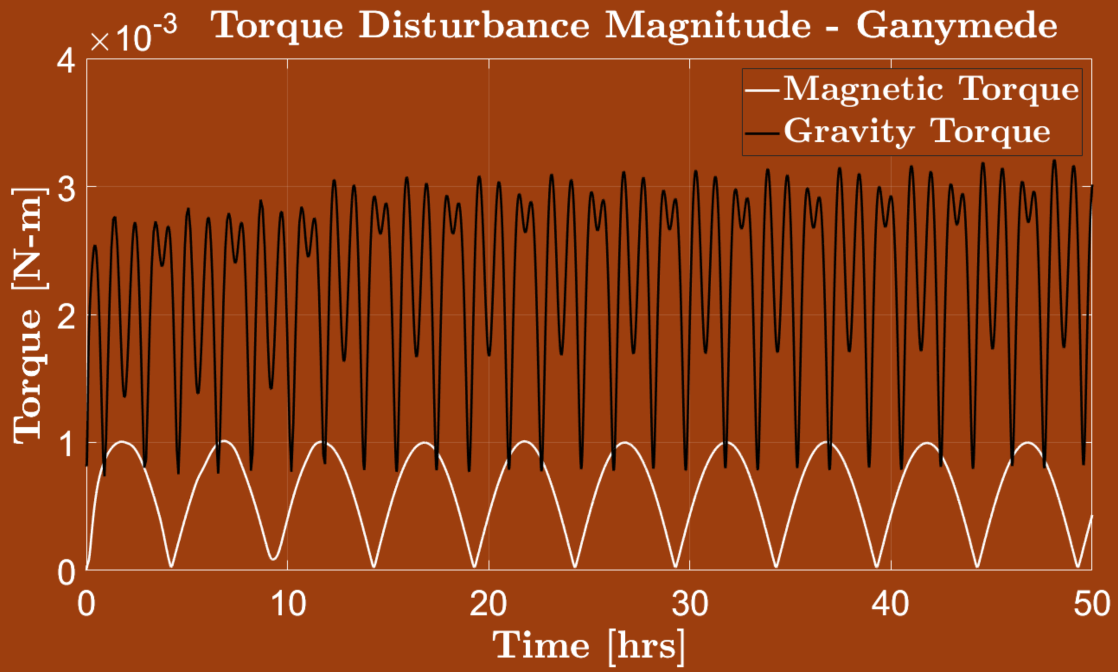
Mode	Damping Ratio
Detumble	2
Data Acquisition	$\sqrt{2}/2$
Downlink	$\sqrt{2}$



Simulated Disturbance Torque

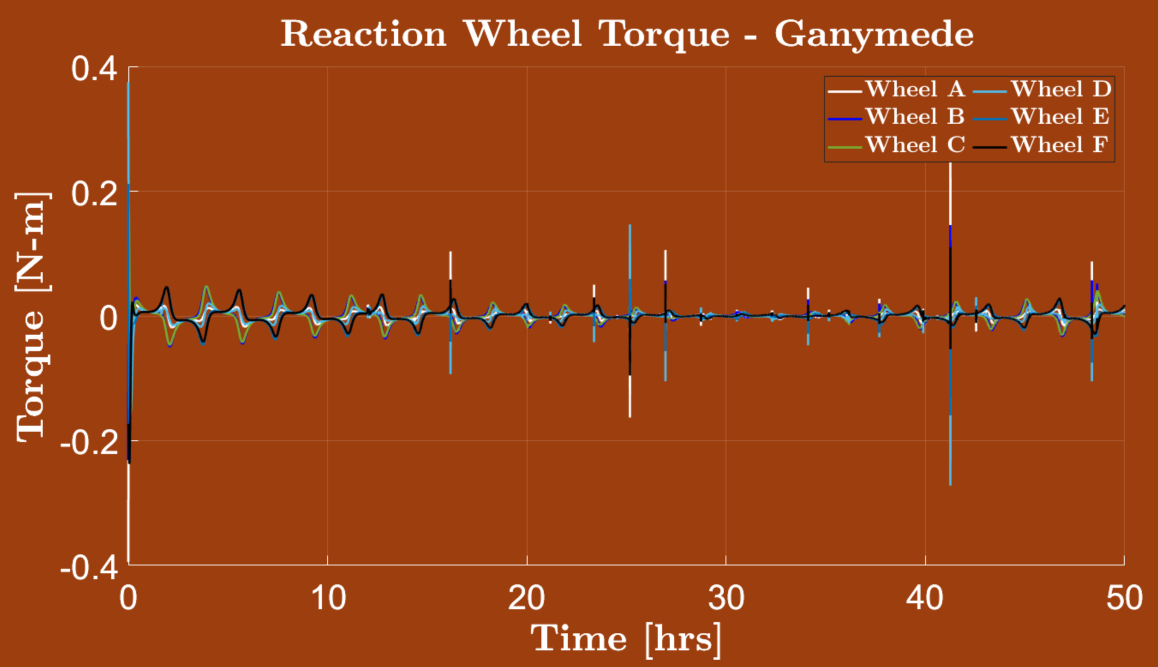
Attitude & Orbit Control System

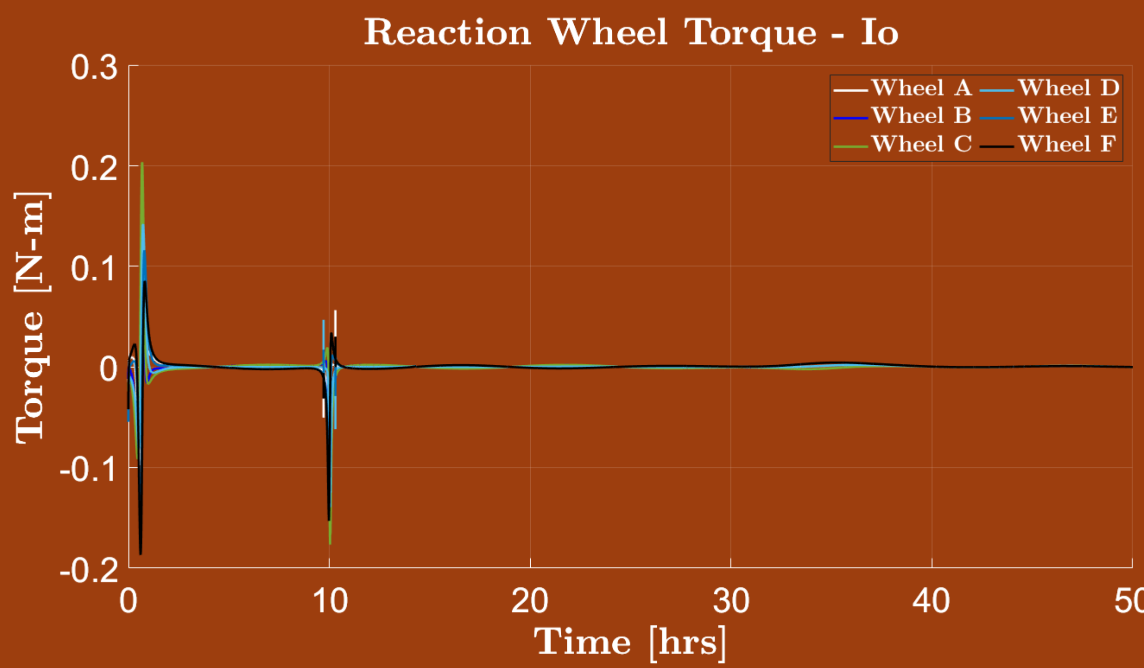




Reaction Wheel Torque Performance

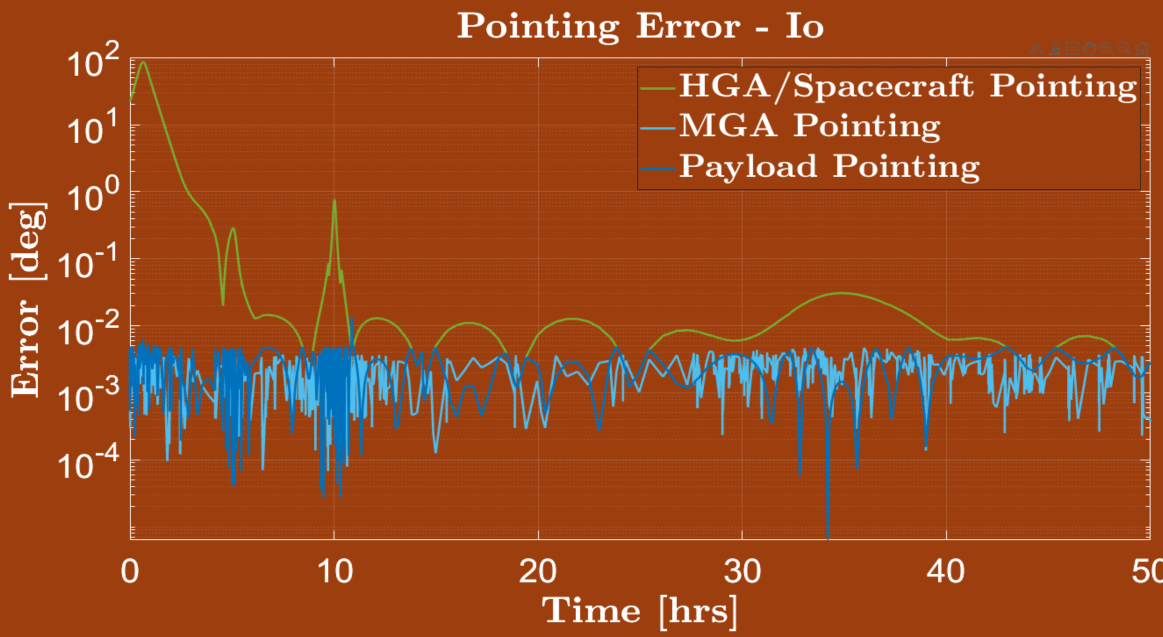
Attitude & Orbit Control System

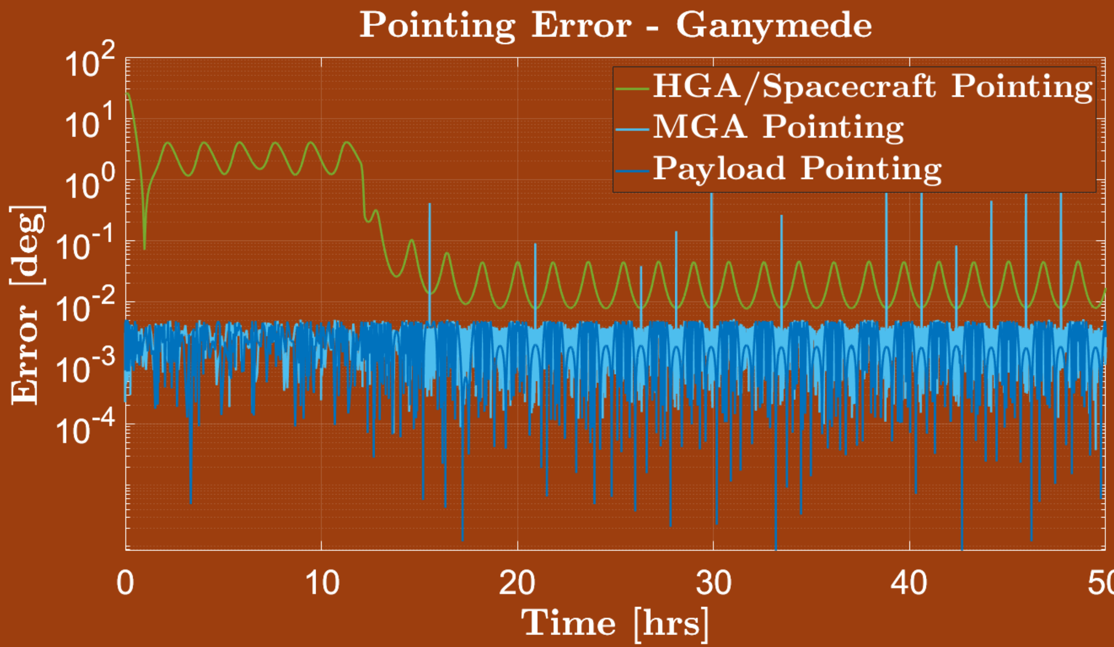




Pointing Performance

Attitude & Orbit Control System





Component	Pointing Angle Requirement
High Gain Antenna	±0.05°
Medium Gain Antenna	±0.7°
Zues-1 Payload	±0.001-0.01°

Thermal Control System

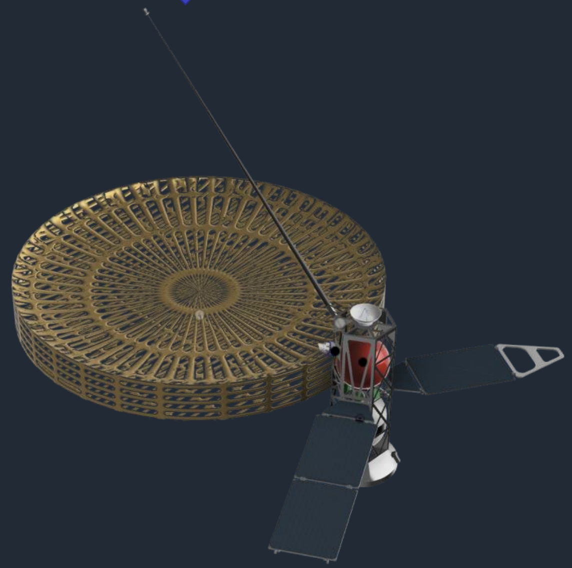
Summer Tietje

Overview

Thermal Control System

- Nodal Thermal Analysis
  - Worst Case Hot and Cold Temperatures
  - Thermal Desktop Model
- Thermal Design
  - Passive and Active Components
  - Exterior Protectants
- Radiation Analysis
- Radiation Vault

Heat Inputs: Generated On-Board Heat, Solar Radiation, Albedo and IR from Jupiter, Earth, and Moons



Heat Outputs:
Heat Radiated to Space

Single Node Analysis

Thermal Control System

Worst Case Hot: 330 K or 57 °C

- Exposure to Earth environment creates highest solar effect
- Location of s/c: 320 km altitude during Earth flyby

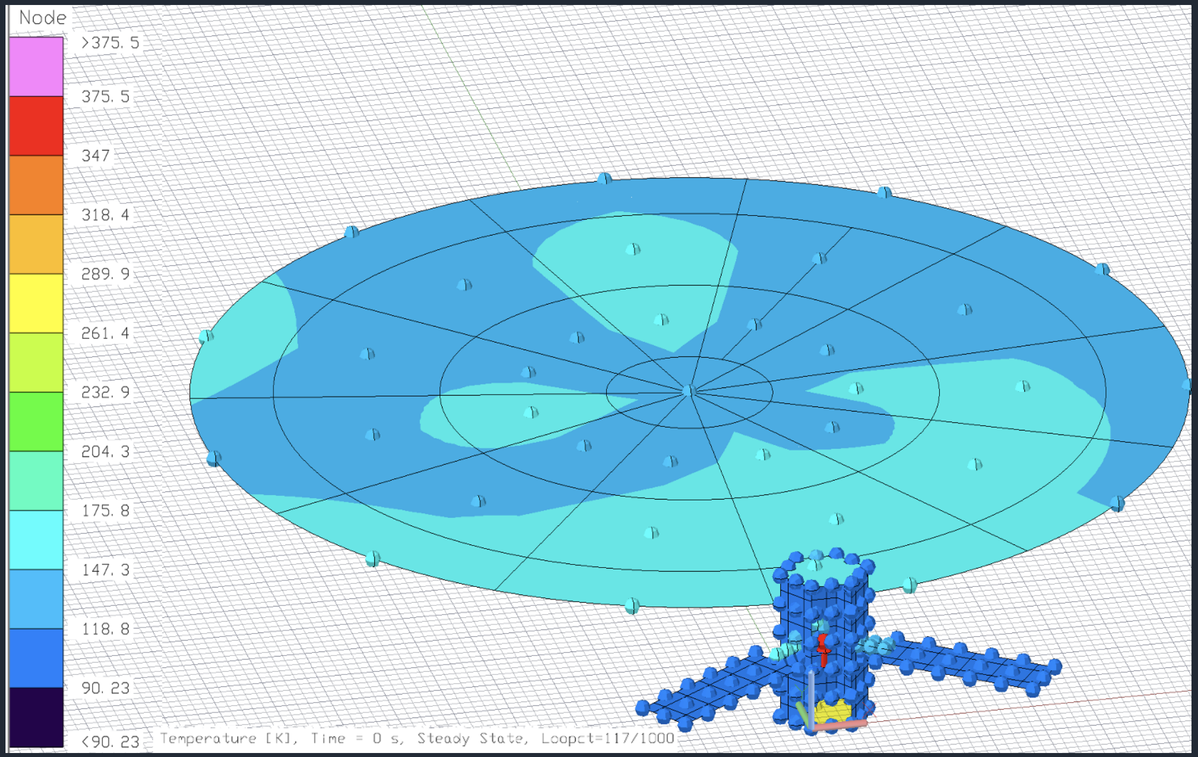
Worst Case Cold: 116 K or -157 °C

- Lack of solar influence while in eclipse and lowest internal heat generation of standby mode will create worst case cold scenario
- Location of s/c: Eclipse 1000 km from Callisto's far side from the sun
- *Note:* time in eclipse and transient cooling effects are not considered
- ***Passive System Design Considerations:*** Coatings, Multi-Layer Insulation (MLI), Radiators, Heat Pipes
- ***Active System Design Considerations:*** Heaters

Thermal Desktop Analysis

Thermal Control System

- Fly-by orbit of Europa
- Assumptions
  - Mesh antenna as thin steel disk for simpler modeling
  - Exterior protectants are MLI and black Kapton
  - Interior coatings are black Chemglaze Z306
  - GaAs solar cells
  - Titanium radiation vault with a constant heat load reflecting Standby Mode
- Modeled through an orbit that doesn't enter eclipse and without thermal controls – reflects steady state as close as possible
- Verifies active heating is necessary and active cooling is unnecessary



Temperature Ranges Outside the Vault

Thermal Control System

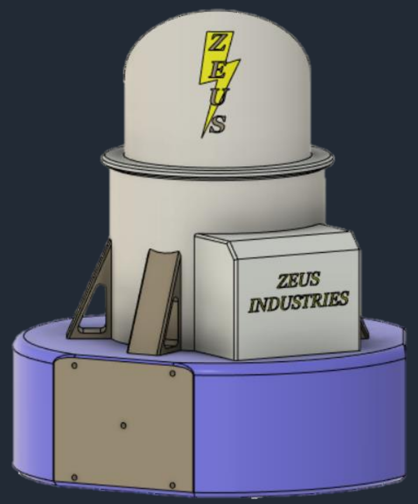
Subsystem	Component	Operational (°C)		Survivable (°C)	
		Min	Max	Min	Max
EPS	Batteries	10	30	0	40
	Solar Arrays	-100	120	-100	120
S&M	Deployment Mechanisms	-40	55	-55	80
Comm	HGA	-30	140	-120	150
	MGA	-100	200	-120	220
	LGA	-40	70	-100	80
	Diplexer	-40	70	-40	70
AOCS	Star Trackers	10	40	-40	70
	Sun Sensors	-40	38	-67	120
	Horizon Sensors	-20	60	-20	60
Payload	Zeus-1	0	40	-35	50

[8]

Zeus-1 Payload Consideration

Thermal Control System

- Zeus-1 thermal requirements:
  - Operational temperature range: 0°C – 40°C
  - Survival temperature range: -35°C – 50°C
  - Waste heat generation: 36W at 1000km, 24W at 750km, 0W otherwise
- Zeus-1 can keep itself thermally regulated
  - When actively collecting data, the imager will output waste heat at a rate equivalent to heat lost due to surface radiation
- Patch heaters will be needed for contingency operations as well as keeping Zeus-1 within survival range while not operating
- Expected to receive 4150 kRad, will work closely with Zeus to ensure payload will survive the environment
  - Can be radiation hardened, coated, or enclosed

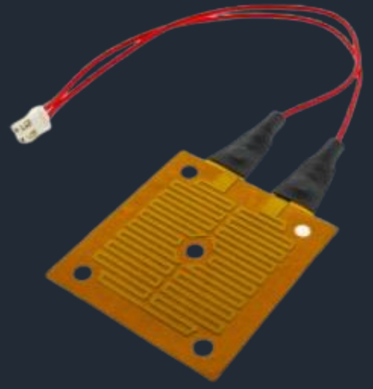


Model of Zeus-1 Payload provided by Zeus Industries

Active Component Analysis

Thermal Control System

- **Active cooling deemed unnecessary**
  - Maximum operational temperature for most sensitive component in vault is ~40 °C
  - Worst case hot condition is ~ 57 °C
    - ~17 °C difference can be accommodated passively
  - Very high power requirements to include active cooling
- **Active heating deemed necessary**
  - Minimum operational temperature for most sensitive component in vault is ~ -50 °C
  - Worst case cold condition is ~ -157 °C
    - All components outside the radiation vault will need heaters attached
      - There is no vault thermally isolating these components
      - Heaters can keep components above survivable temperature range during contingency operations
      - Propulsion tanks, batteries, and Zeus-1 payload with special attention

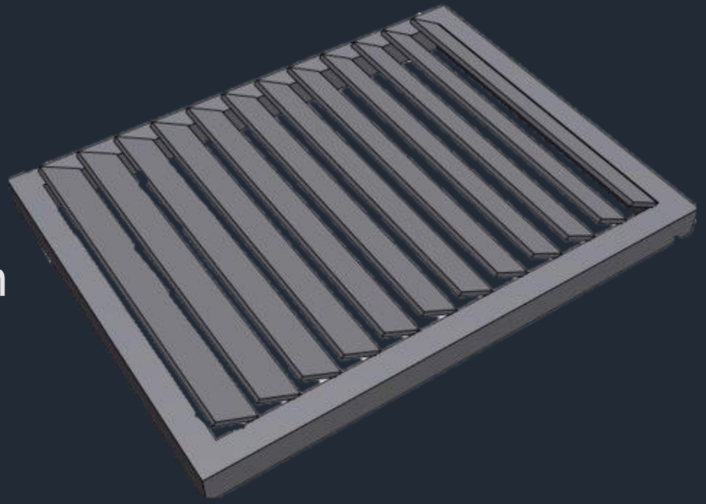


Patch heater [1]

Exterior Radiators

Thermal Control System

- The interior of the spacecraft will need to be kept between 10-30°C due to the temperature tolerances of the components
- Both GEC-1 and GEC-2 will employ two 1 x 0.75 m louvers
  - 1.5 m^2 of radiative surface area
  - 10°C open/close differential, from 15°C fully closed to 25°C fully open
  - Directly behind solar panels to minimize solar flux
  - 12 individually actuated aluminum fins to maximize reliability
- The required radiative surface parameters to balance the thermal system of the spacecraft
  - Polished magnesium radiators
  - ~1m^2 of surface area
  - 0.49 emissivity
- The louvers will never fully open in nominal operation, which leads to additional margin for emergencies

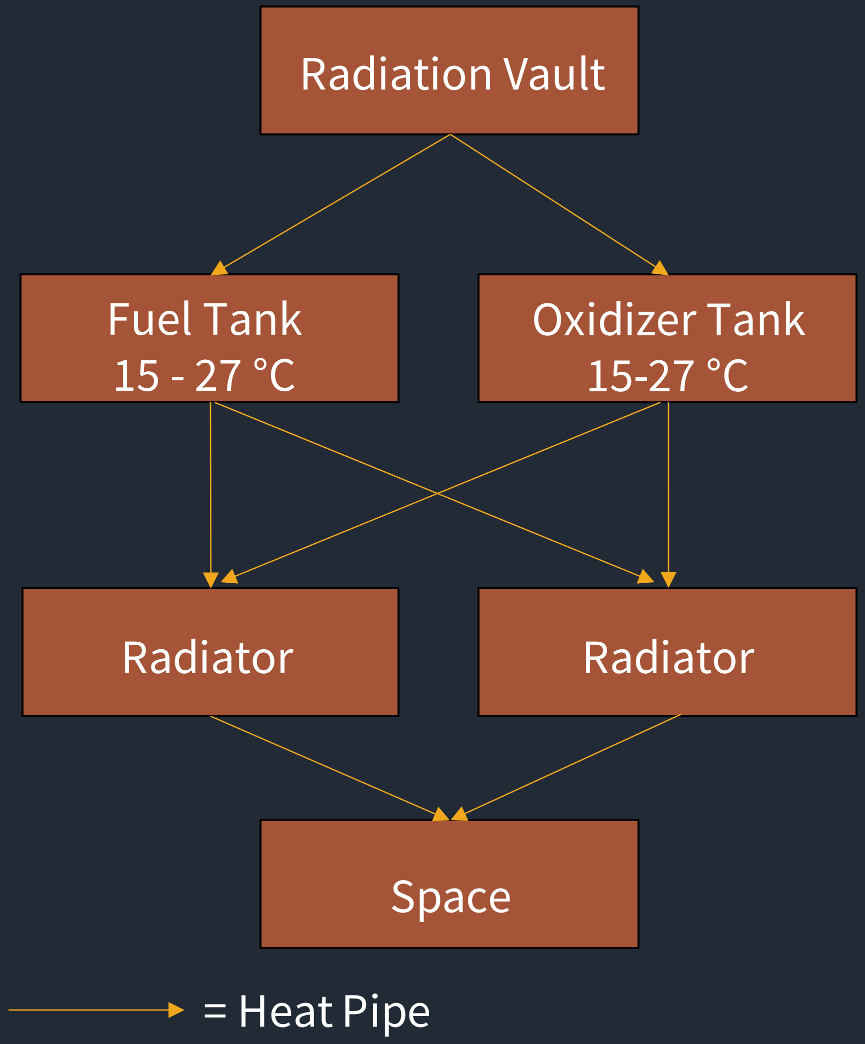


Proposed louver design

Heat Pipes

Thermal Control System

- The heat pipe system is represented as a series of nodes in the diagram
- There will be one pipe between transferring up to 250 W Each
  - Constantly Conducting Heat Pipes between the Radiation Vault and Propellant Takes
  - Variable Conducting Heat Pipes between the Fuel
  - 31 cm long, with a diameter of about 0.6 cm
- The thermal switch will be capable of controlling the heat flux between the propellant tanks and the radiators
  - Will maintain required propellant tank temperatures

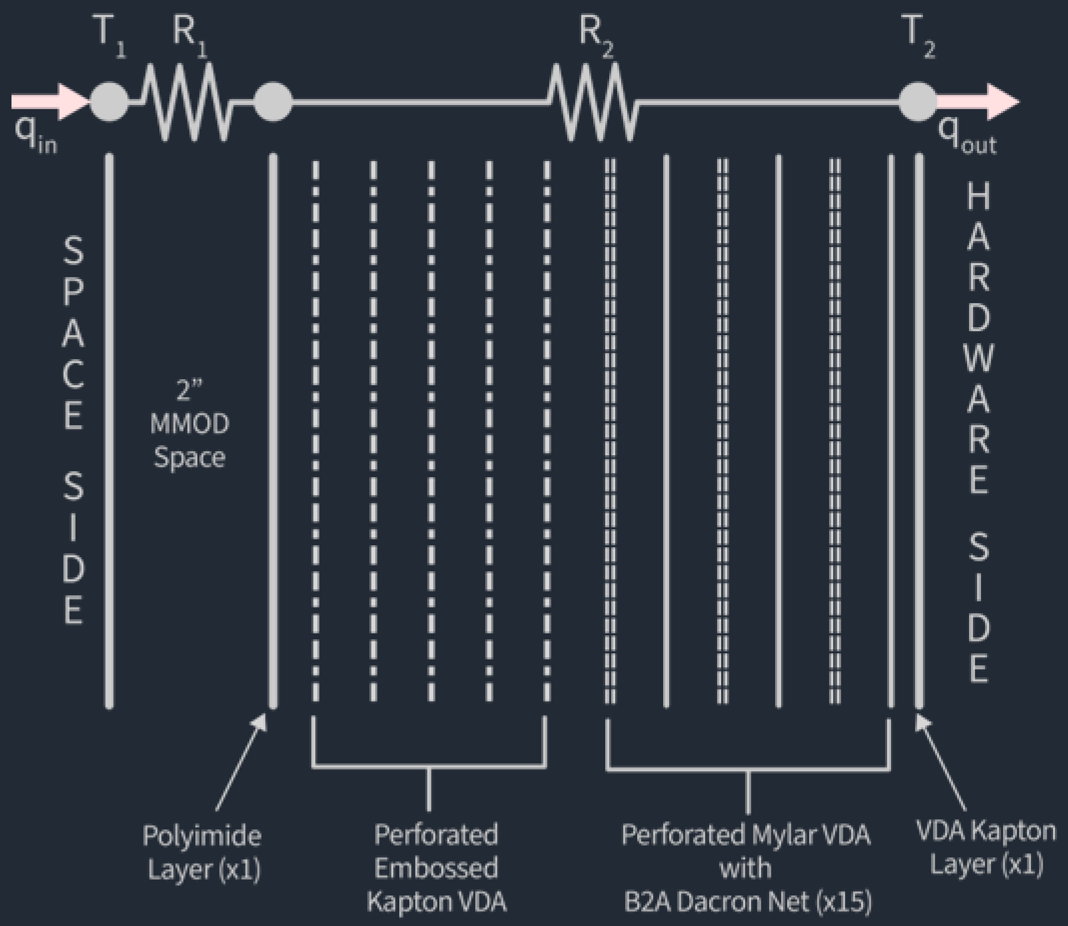


Standard MLI Blanket

Thermal Control System

- Inner 15 layers
  - VDA coated perforated mylar with dacron netting spacers
- Outer 5 layers
  - VDA perforated embossed Kapton
  - StaMet coated black Kapton
- Emissivity of 0.02
- Serviceable up to 250°C
- Density of 900 g/m^2

Standard Blanket Layup

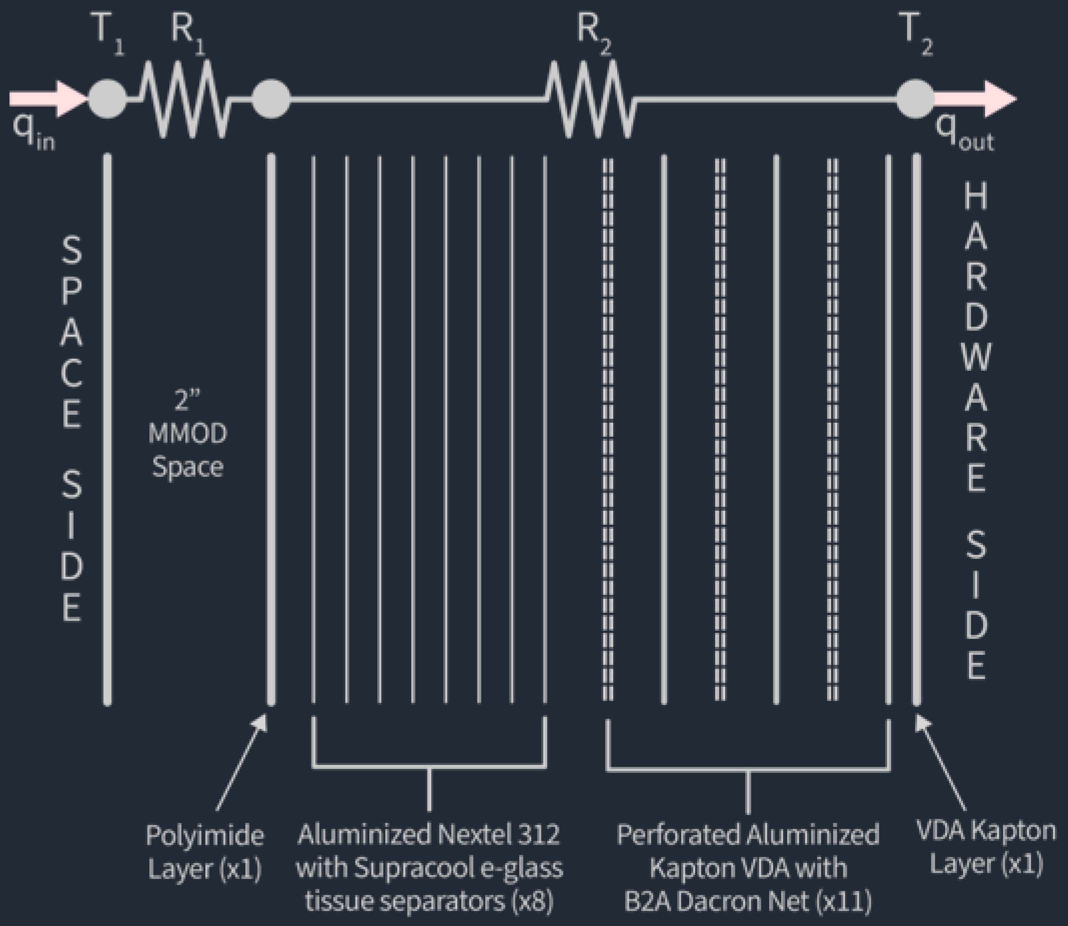


High Temperature MLI Blanket

Thermal Control System

- Inner 11 layers
  - Aluminized Kapton VDA with dacron netting spacers
- Outer 8 layers
  - NextelTM 312 fabric with ceramic glass separator layers
  - Final outer layer is aluminum foil
- Emissivity of 0.02
- Serviceable up to 1000°C
- Density of 1600 g/m^2

High-Temp Blanket Layup



Exterior Protectants

Thermal Control System

- Standard MLI will be used to insulate most of both GEC-1 and GEC-2
- High temperature MLI will be used near propulsion thrusters
  - Minimized impact of thruster operation on thermal control of spacecraft
- Vapor-deposited gold will be used on Zeus-1 payload and other sensitive equipment
  - Reflects radiation and other noises for optimal data collection
- Vapor-deposited aluminum coating used on louver fins
  - Absorptivity: 0.08, Emissivity: 0.02 [7]
  - Reflects most heat to isolate radiative surface area underneath
- Black Chemglaze Z306 used within spacecraft bus
  - Maximizes heat radiation between vault and spacecraft exterior

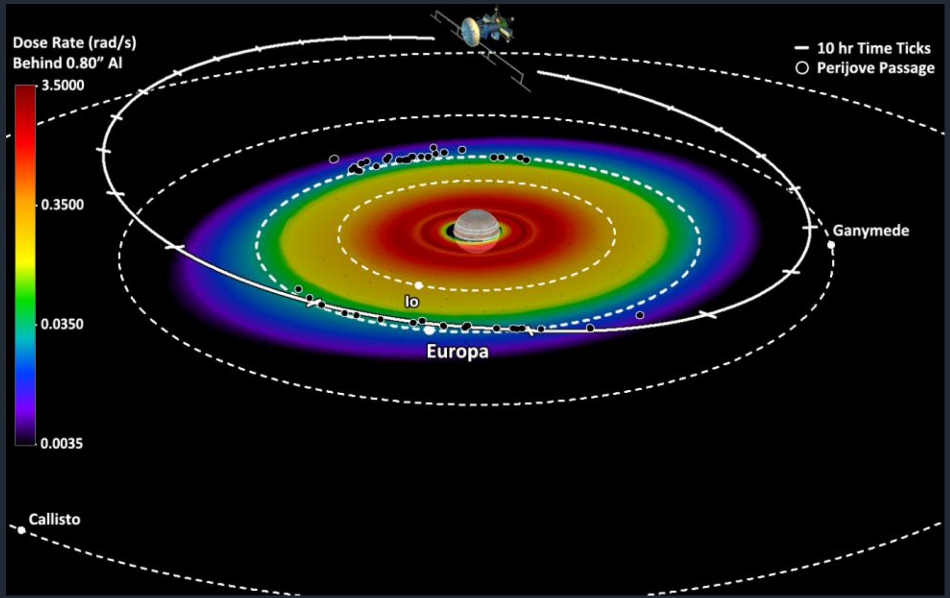
[2] CHEMGLAZE Z306 Black Polyurethane Coating 5 Gallon Pail

GEC-1 Flyby Radiation Analysis

Thermal Control System

- Lifetime radiation dosage experienced by GEC-1 was modeled using dosage values based on Europa Clipper radiation analysis [3]
- Callisto and Ganymede's radiation environments are similar, and with shielding the absorbed radiation for components have the same rate
- **Worst case lifetime radiation dosage is 165.56 kRads**

Satellite Location	Time	Radiation Dosage (kRad)
Orbit 750-1000km above Ganymede and Calisto	1 day	0.03
	4 years (2 years each moon)	44.15
	15 years (7.5 years each moon)	165.56



Europa Clipper predicted radiation mitigation [3]

GEC-2 Radiation Analysis

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Thermal Control System

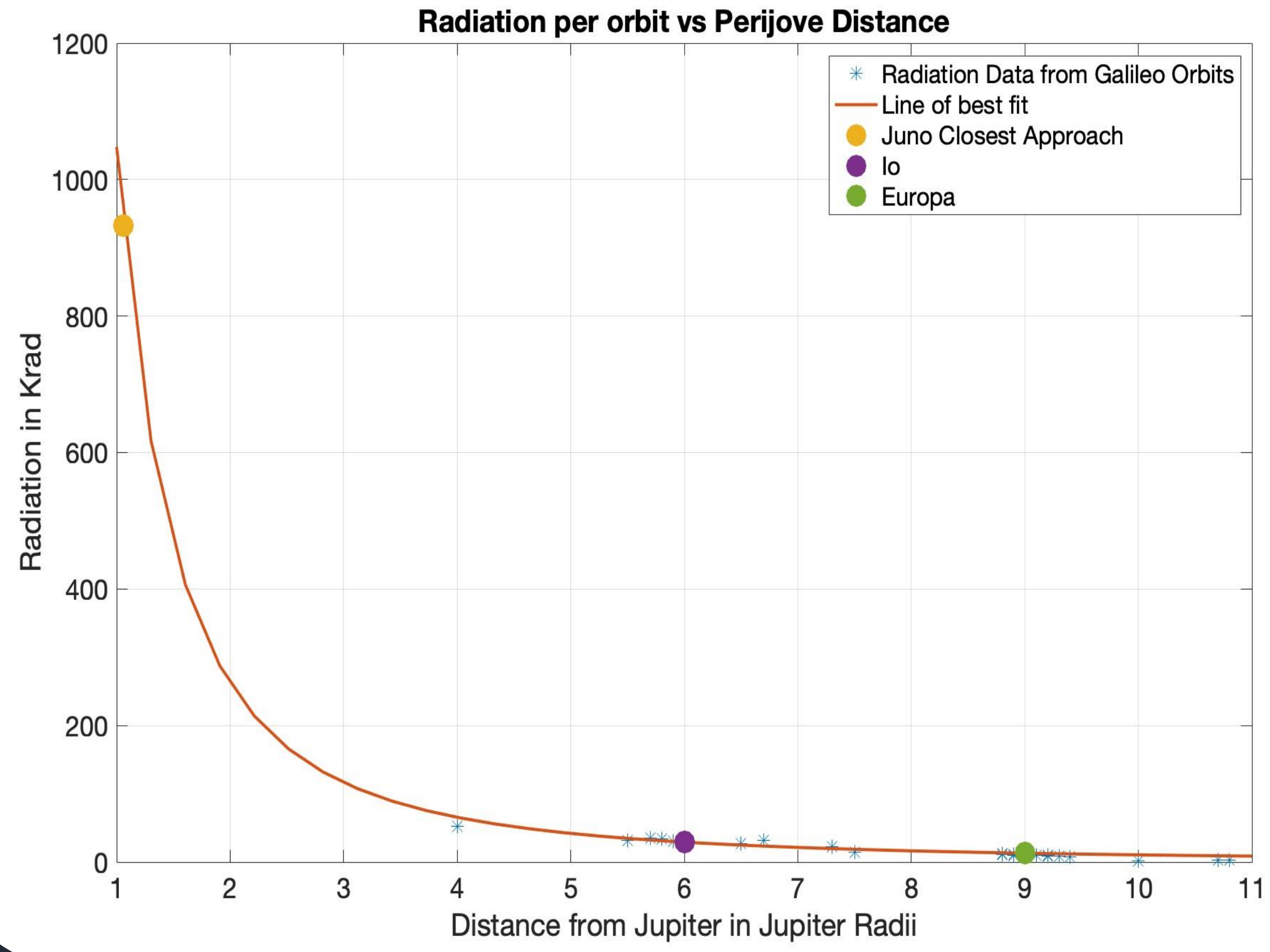
- Dosage calculation estimated using Galileo mission radiation dosage predictions
  - Galileo's 24th orbit around Io and 4th orbit around Europa are most similar to the GEC-2 orbits
  - Radiation analysis predicts lifetime dosage based on closest approach (perigee) to Jupiter
- GEC-2 orbits Io with a period 31 days and Europa with a period of 24 days
- Mission requirements the closest approach of both orbits will be in the 750-1000km range
- Using the Galileo radiation data for a 15-year mission of 100 flybys of each moon results in a **lifetime radiation of 4,150 Krads for GEC-2**
- Designing both spacecraft for worst case environment on GEC-2

Galileo Orbit	Closest Approach (km)	Radiation Per Orbit (kRad)	Orbital Period (days)
Io	611	32.0	44
Europa	688	9.5	35

[4]

Radiation Visualization

Thermal Control System

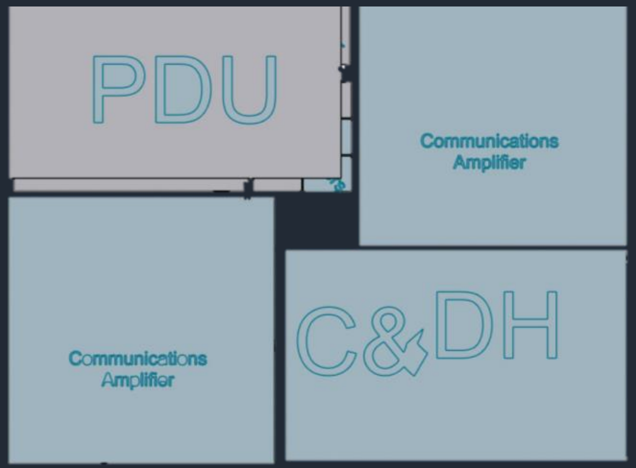


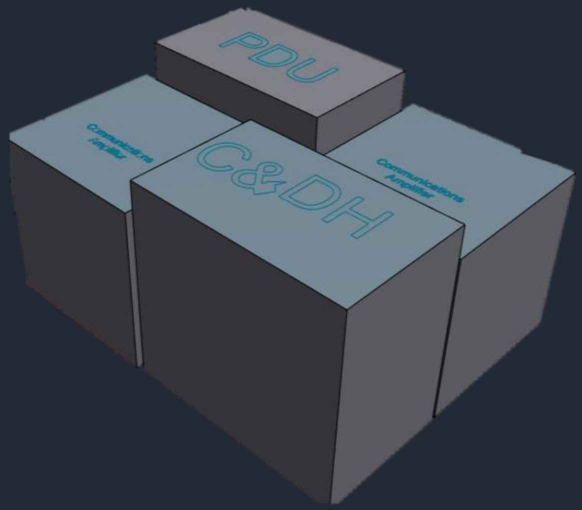
[4]

Titanium Radiation Vault

Thermal Control System

- Juno's design
  - Expected to experience 20,000 kRad over lifetime
    - Extremely high lifetime radiation due to closeness of orbit perigee
  - Components in vault experienced 25 kRad
    - Thickness decreases radiation by factor of 800
- Spacecraft vault design
  - Both spacecraft will have 50x40x30cm radiation vaults
    - Vault thickness is 0.875cm as it has flight heritage with Juno and has proven to survive extended Jovian missions
  - GEC-1 is expected to experience 165.56 kRads after 15 years
  - GEC-2 is expected to experience 4,150 kRads after 15 years





Project Management

Meghan Aerick

Risk Analysis

Project Management

- A preliminary risk analysis was performed, and 5 risks were identified to be critical to the overall mission.

Probability \ Impact	Very low	Low	Moderate	High	Very high
Very low				3	
Low					1, 2, 4
Moderate					
High			5		
Very high					

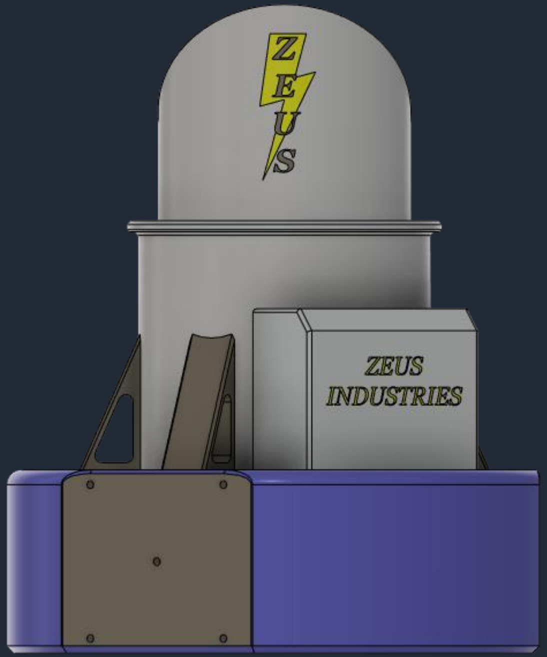
Top Mission Risks

1. Zeus-1 Imager Failure
2. High Gain Antenna Failure
3. Solar Array Deployment Failure
4. Loss of Attitude Control
5. Radiation Exposure

Zeus-1 Imager Failure

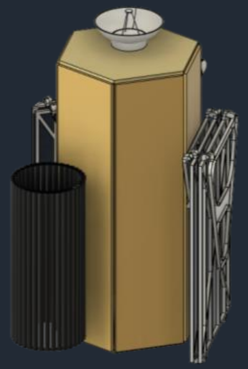
Project Management

- Possible causes:
  - MMOD impact
  - Outgassing onto lens causing poor quality video
  - Failure to meet power and thermal requirements
- Possible impacts:
  - Poor video quality
  - Inability to fulfill stakeholder needs
- Mitigation Efforts:
  - Extensive thermal vacuum and vibe testing of payload
  - Proper protection around payload when not in use
  - Careful placement of payload in relation to thrusters and materials that are susceptible to outgassing

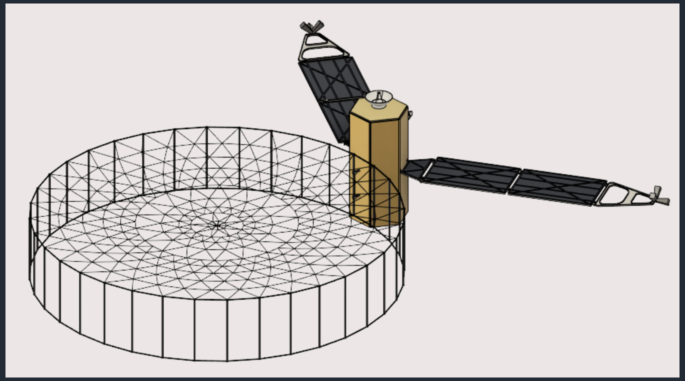


High Gain Antenna Failure

Project Management



- Possible causes:
  - Failure of deployment mechanism
  - Damage to antenna from MMOD
- Possible impacts:
  - Loss of ability to transmit 4k video data to ground station
  - Failure to meet stakeholder needs

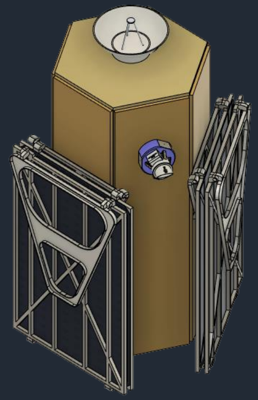


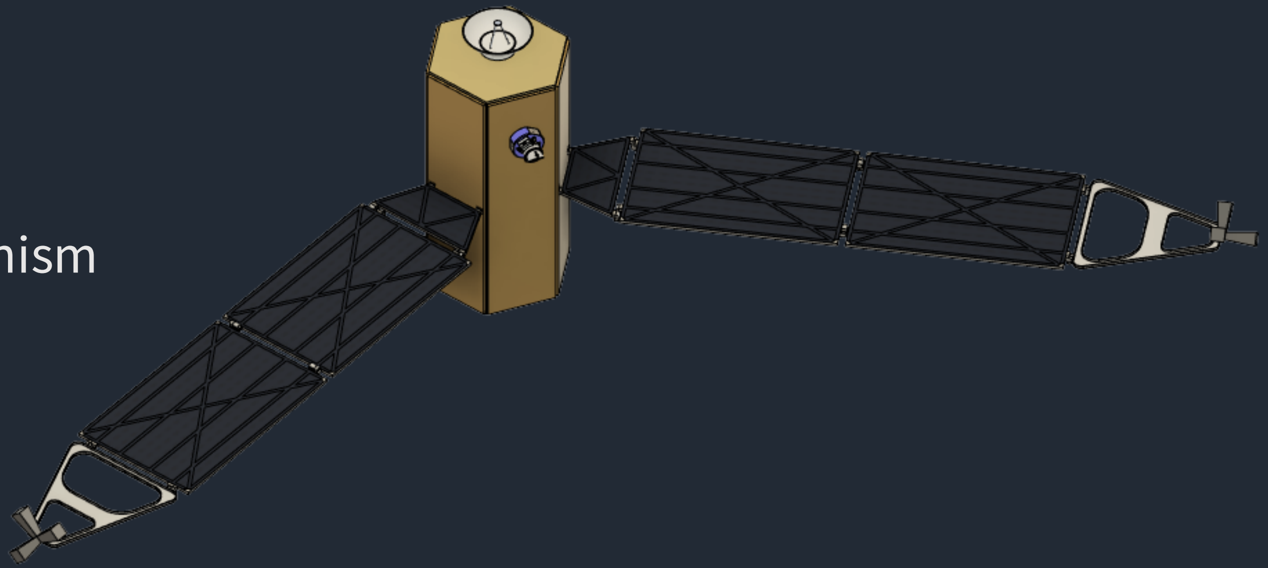
- Mitigation Efforts:
  - Thorough testing of antenna deployment mechanism
  - Deployment of the HGA before leaving Earth orbit
  - Last-ditch efforts to shake the mechanism free by RCS or engine fire
  - Medium gain antenna included to at least be able to communicate with ground station from Jovian system for telemetry and command
    - Video footage could still be transmitted but at an extremely slow rate

Solar Array Deployment Failure

Project Management

- Possible causes:
  - Failure of deployment mechanism
- Possible impacts:
  - Significant loss of power generation
    - Only the outward facing panels in the stowed configuration would be generating power
  - Lower lifespan of the spacecraft
  - Inability to meet mission requirements
- Mitigation Efforts:
  - Rigorous testing of deployment mechanism
  - Use a mechanism with high TRL





Loss of Attitude Control

Project Management

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- Possible causes:
  - Sign errors
  - Reversal of polarity in magnetorquer caused by strong magnetosphere in Jovian system
  - Gyro, reaction wheel, or star tracker failures/errors
- Possible impacts:
  - Could be catastrophic if solar arrays and HGA are deployed as they could snap off
  - Loss of ability to control spacecraft
- Mitigation Efforts:
  - Redundancy in magnetorquers, sun sensors, reaction wheels, star trackers, gyros, etc.
  - Built-in fault detection, isolation, and recovery system within C&DH

Radiation Exposure

Project Management

- Possible causes:
  - Jovian system has a highly radiative environment, especially around Io (closest to Jupiter)
  - Inadequate protection of components from radiation
- Possible impacts:
  - Single event upsets (SEUs)
  - Major electronic malfunction
  - Solar panel degradation
    - Decreased mission duration
- Mitigation Efforts:
  - Radiation vault for electronics
  - Cover glass on solar panels
  - Radiation hardening for selected components

Risk Analysis

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- With the mitigation efforts, the probability and some of the impacts of these risks can be reduced.

Probability	Impact: Very low	Low	Moderate	High	Very high
Very low			3	1	
Low		4	2		
Moderate		5			
High					
Very high					

Top Mission Risks

1. Zeus-1 Imager Failure
2. High Gain Antenna Failure
3. Solar Array Deployment Failure
4. Loss of Attitude Control
5. Radiation Exposure

Schedule

Project Management

With the first scheduled launch in January 2026, this schedule from concept development to launch is loosely based on other interplanetary missions of this magnitude.

1. **Conceptual Design** — **September 2022 – December 2023**
   - Define mission objectives and requirements from solicitation
   - Identify key technologies and subsystems needed
   - Evaluate different spacecraft and mission design options (GEICOS, JOLT, MOJO, MoonTube)
   - Stakeholder pushbacks
2. **Preliminary Design** — **January 2023 – June 2023**
   - Develop initial spacecraft design based on selected concept (MOJO)
   - Conduct trade studies and feasibility assessments
   - Define spacecraft architecture and subsystem requirements
3. **Critical Design** — **July 2023 – March 2024**
   - Finalize spacecraft design and subsystem specifications
   - Perform detailed analyses and simulations
   - Conduct design reviews and obtain necessary approvals

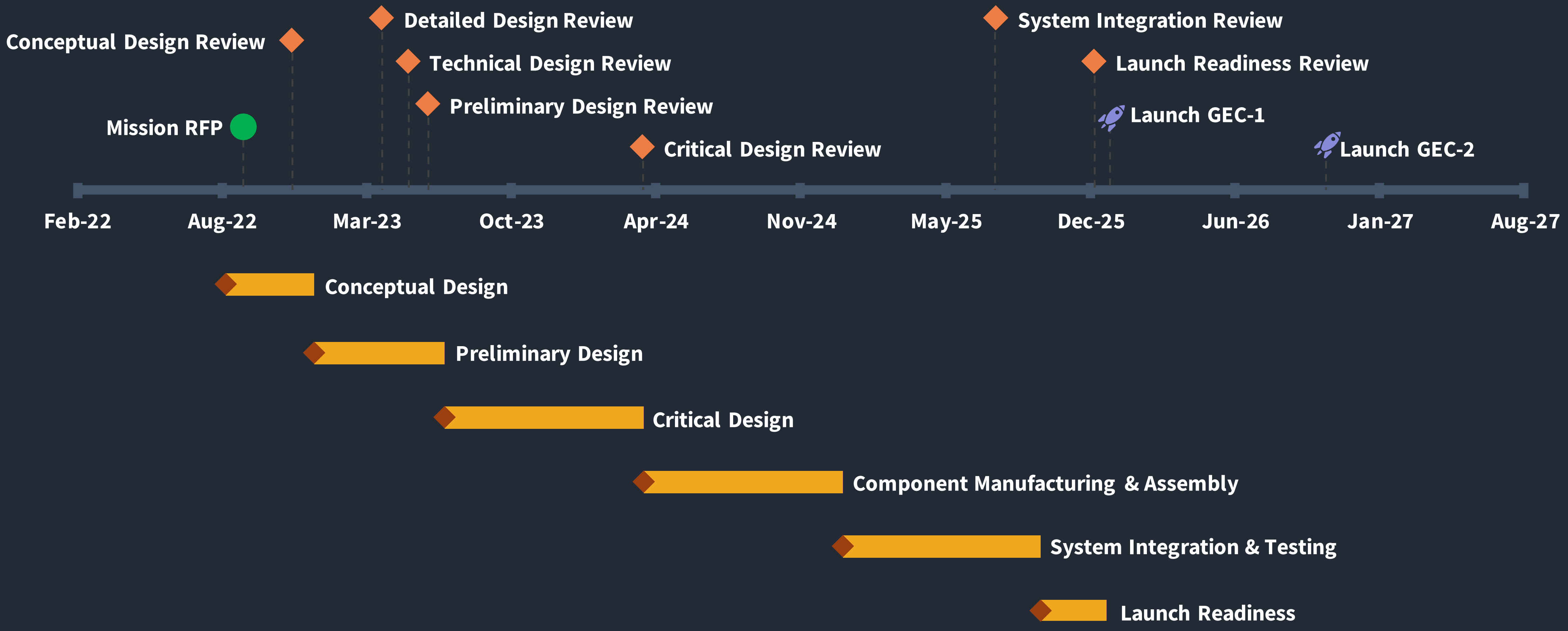
Schedule

Project Management

4. **Component Manufacturing & Assembly** — **April 2024 – December 2024**
   - Purchase or manufacture all components in PBS
   - Assemble bus structure and integrate subsystem components
   - Perform functional and integration testing
5. **System Integration & Testing** — **January 2025 – September 2025**
   - Conduct system-level tests to verify performance and validate requirements are met
   - Perform environmental testing (vibration, thermal, vacuum, etc.)
   - Integrate the Zeus-1 payload
   - Ensure compatibility with Falcon Heavy launch vehicle
6. **Launch Readiness** — **October 2025 – December 2025**
   - Conduct final system tests and functional checks
   - Prepare spacecraft for launch and obtain necessary clearances
   - Integrate spacecraft with launch vehicle

Schedule

Project Management



Development Budget

Project Management

- Cost Estimate Models provided by SME
- Our current budget of **$500 million** cannot cover the estimated cost of the bus and launches
- The inclusion of additional payloads will only cover the cost of their inclusion and cannot be relied upon to pay for the mission
- The budget provided by Zeus Industries will have to be increased significantly
- Future work: break down the costs to the component level and work to reduce the largest costs

Cost Estimate Model	Cost of One Spacecraft	Cost of Two Craft and Two Launches
Subsystem ROM	76.4 M	553 M
USCM8	756 M	2.64 B
NASA QuickCost	496 M	1.39 B

Operations Budget

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- Deep Space Network - $490 million
  - Quoted from the DSN estimating worksheet using 20% contact time and 34-m antennae
- Custom Ground Station - $895 million
  - Estimated from JPL contracts and average regional land values

Custom Ground Station Initial Development Costs (million USD)				
Antenna Location	Virginia, USA	Australia	Spain	Total
Land Acquisition	30.3	0.7	25.4	56.4
Antenna Construction	96.7	96.7	96.7	290.1
Initial Development Total	127.0	97.4	122.1	346.5
Custom Ground Station Recurring Operations Costs (million USD)				
	Yearly	10 years	15 years	Total (24 y)
Per Antenna	7.6	76	114	182.4
Total	22.8	228	342	547.2

Wrap Up

Meghan Aerick

Success Criterion

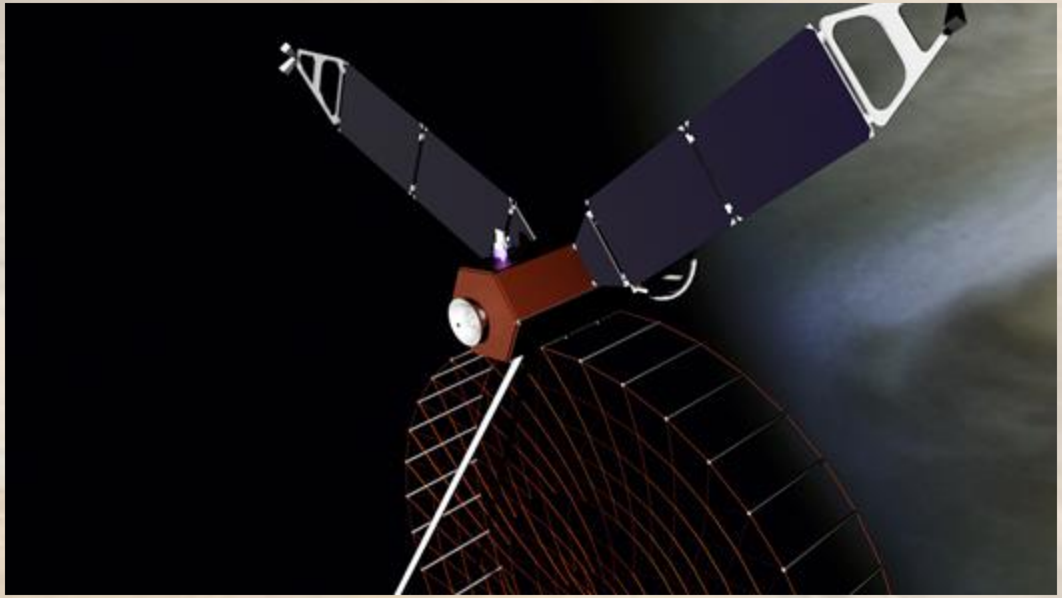
Wrap Up

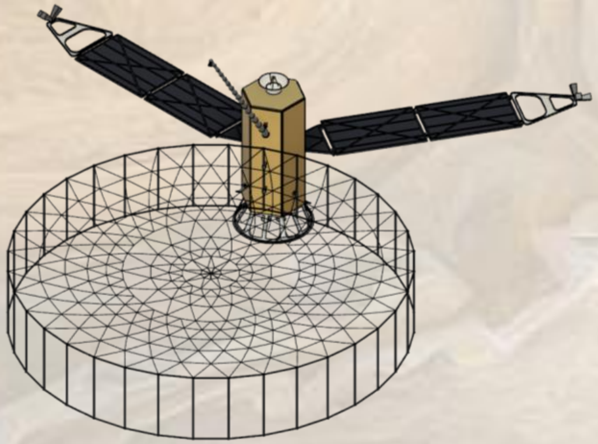
- In order for the mission to proceed, the following is necessary:
    - Increased development budget
    - Reliability analysis to ensure a total mission reliability of 98.5% is achievable
    - Adequate budget for ground station and mission operations
    - Decision for how and where ground station operations will take place
- In the current state of the mission, MOJO will be able to fulfill all of the stakeholder needs & constraints except the budget and reliability, which will require further analysis and/or pushbacks with Zeus Industries

Major Accomplishments

Wrap Up

- Subsystem trade studies completed, and optimal components selected
- Launch vehicle and launch dates selected
- Major structural components designed and modeled
- Working flight software setup
- Payload selection narrowed down to just the Zeus-1 optical payload
- Preliminary thermal and structural analyses run
- Attitude and orbit control implemented in simulation





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Wrap up

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Questions?

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