Preliminary Design Review

C.O.S.M.I.C – Wednesday, June 2nd , 2021

CAL POLY

Introduction

C.O.S.M.I.C. (Celestial Object Sensing and Measuring Identification Campaign)

Our Team is a dedicated engineering group of 72 Cal Poly Spacecraft Design students working cooperatively in a virtual environment.

Our Mission is to provide space systems for interstellar exploration to further our understanding of the origins of the solar system through the study of interstellar objects and near-parabolic comets.

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Speaker: Natalia C.

PDR Entrance and Success Criteria

Entrance
The following primary products are ready for review: A preliminary design that can be shown to meet all technical requirements and performance measures.
Life-Cycle Cost and Integrated Master Schedule (IMS) that are ready to be baselined after review comments are incorporated.
Baseline operations concept
Updated risk assessment and mitigation

Success
The flow down of verifiable requirements is complete and proper, or, if not, an adequate plan exists for timely resolution of open items. Requirements are traceable to parent technical requirements and to mission goals and objectives
Preliminary analysis of the primary subsystems has been completed and summarized, highlighting performance and design margin challenges
TBD and TBR items are clearly identified with acceptable plans and schedule for their disposition
The preliminary design is expected to meet the requirements at an acceptable level of risk

Solicitation Breakdown

Goal: To collect data on the composition, morphology, and state of an ISO (InterStellar Object) or NPC (Near-Parabolic Comet)

Primary Objectives: The proposed system must be able to identify at least one ISO or NPC within 20 years of the system readiness date.

- Composition
- Morphology
- Angular Momentum

Secondary Objective Options: The proposed system must be able achieve at least one secondary objective.

- Impactor Science
- **Remote Observation Platform**
- **Advanced Object Definition ***
- Heliophysics Platform
- **Exoplanet Platform**
- Data Relay

Additional Points:

- The initial incoming ISO trajectory is **provided by the customer at 3 AU**
- **2 launches** are requested by the customer, within the 20-year mission period of 2030-2050

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Speaker: Natalia C.

Level 1 Requirements

Backup

Primary Objectives

ID: MP = Mission Primary

ID	Traceability	Requirement
MP1	Solicitation	The mission shall be ready to react to an ISO no later than 12/31/2030.
MP2	Solicitation	The mission shall have an 80% likelihood of reaching at least 1 object with the parameters specified in Table 1 within 20 years of its readiness date.
MP3	Solicitation A.2	The mission shall acquire visible imagery of 50% of the object's illuminated surface with a resolution of at least 5.0 meters per pixel.
MP4	Solicitation A.2	The mission shall acquire infrared imagery of 50% of the object's visible surface with a resolution of at least 10.0 meters per pixel.
MP5	Solicitation A.3, Solicitation B.1	The mission shall model 50% of the object's shape within +/- 10 meters using active measurement.
MP6	Solicitation A.3	The mission shall determine the object's mean dimension within +/- 10 meters.
MP7	Solicitation A.4	The mission shall determine the object's spin axis within +/- 1.0 degree.
MP8	Solicitation A.4, Customer Conversation 5/7/2021	The mission shall determine the object's rotation rate within 1%.
MP9	Customer Conversation 1/8/2021	The mission shall return data to the customer no later than 9 months post collection.
MP10	Customer Conversation 1/8/2021	The mission shall communicate with the deep space network.

Level 1 Requirements

Secondary Objectives

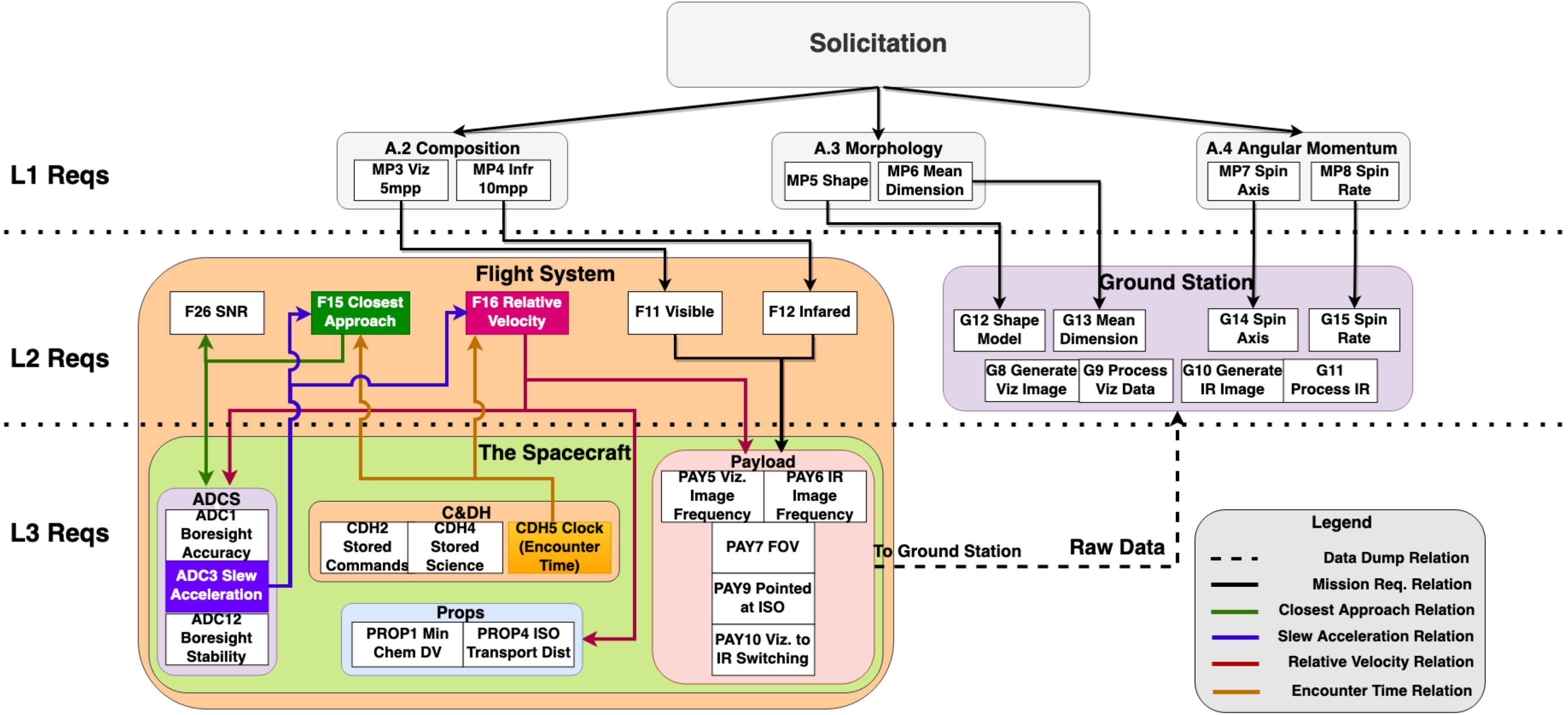
ID: MS = Mission Secondary

ID	Traceability	Requirement
MS1	Solicitation B.1	The mission shall measure the object's dielectric constant within +/- **TBD**.
MS2	Solicitation B.3, Solicitation B.5	The mission shall have a sky coverage of 0.15 % in the prepositioned orbit.
MS3	Solicitation B.3	The mission shall observe heliocentric orbiting bodies.
MS4	Solicitation B.5	The mission shall acquire exoplanet photometry of a minimum of 1 star system.

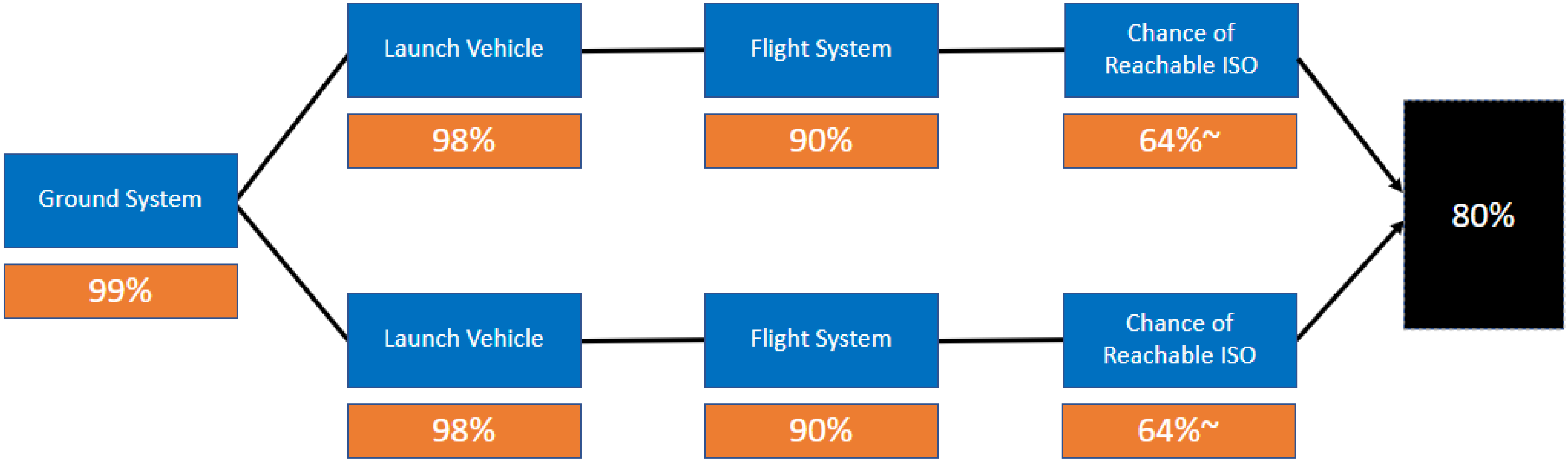
Design Challenges Overview

Speaker: Keilan R.

Encounter Justification



Probability Model

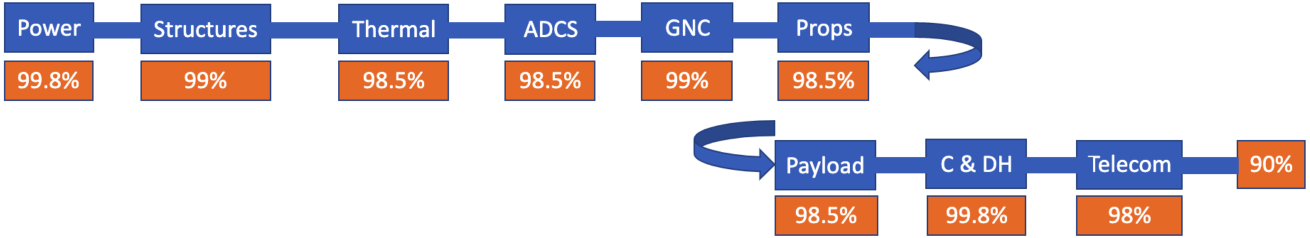


System Level (Level 2) Reliability Block Diagram

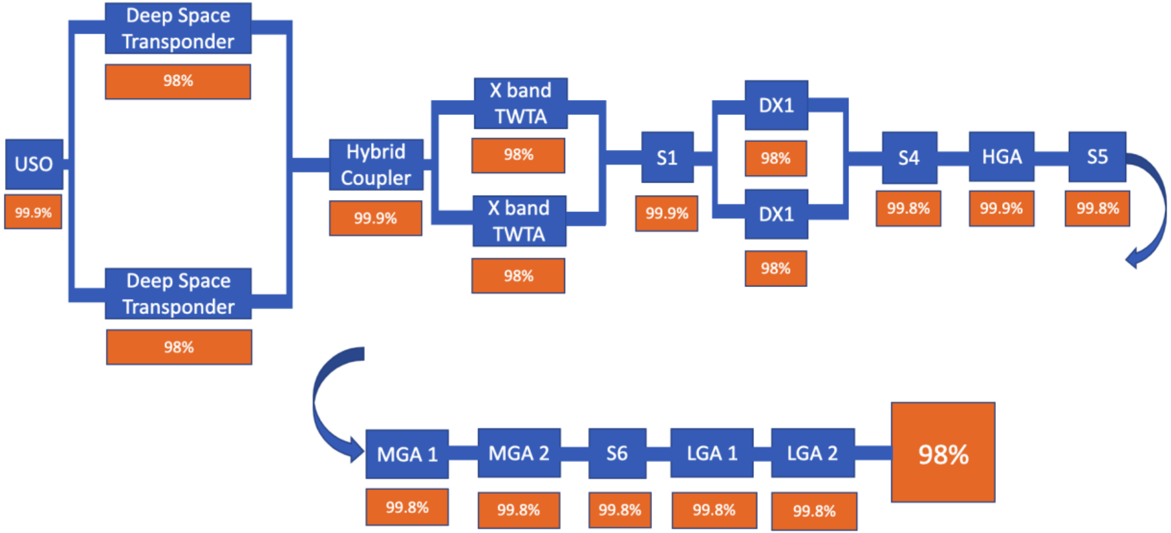
Note: Chance of Reachable ISO is 64%~ for each S/C, equivalent to an 87%~ chance of at least one Reachable ISO between both S/C

Flight System Reliability Model

Level 3 Reliability (Flight System)

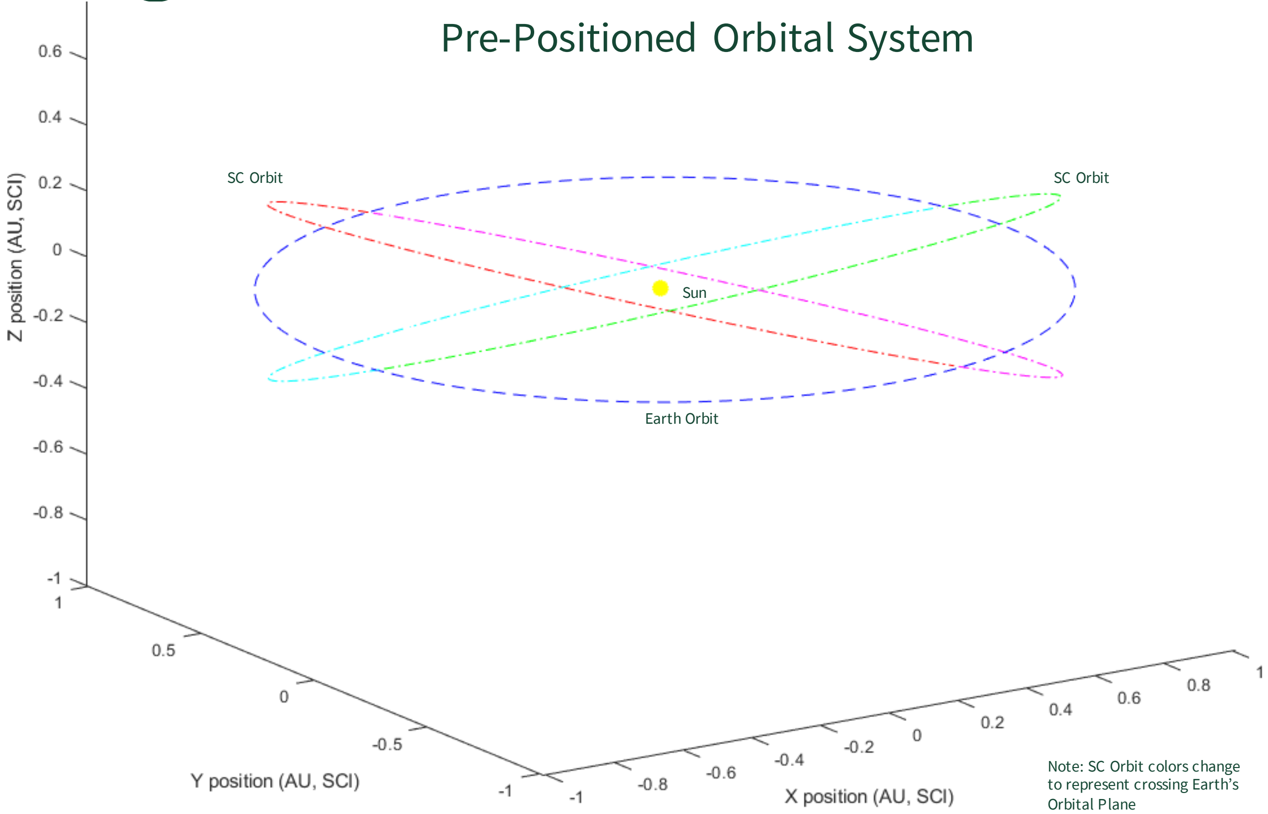


Level 4 Reliability (e.g., Comms)



Mission Design

- Require ISO Encounter within set parameters
- Require 87% chance of encountering ISO within 20 years
- Mission Design facilitates Encounter and ensures Coverage
  - Pre-positioned orbits chosen allow for use of Earth Gravity Assist in transfer to ISOs
- This Mission Design has an 87% chance of achieving specified ISO Encounter for a spacecraft carrying 4.5 km/s of onboard dV



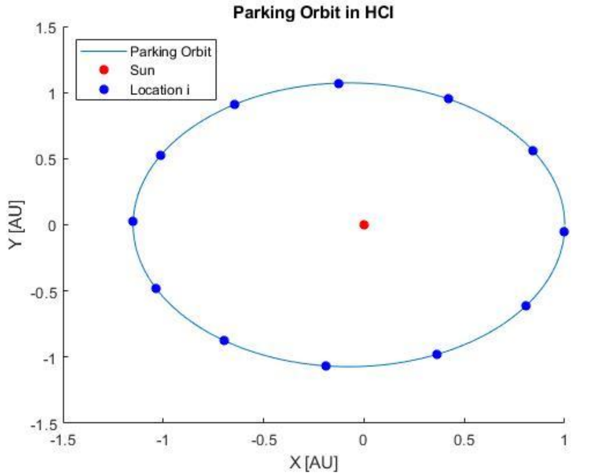
Note: SC Orbit colors change to represent crossing Earth's Orbital Plane

Speaker: Jordan W.

Problem Identification – E-Prop vs. Chem Prop

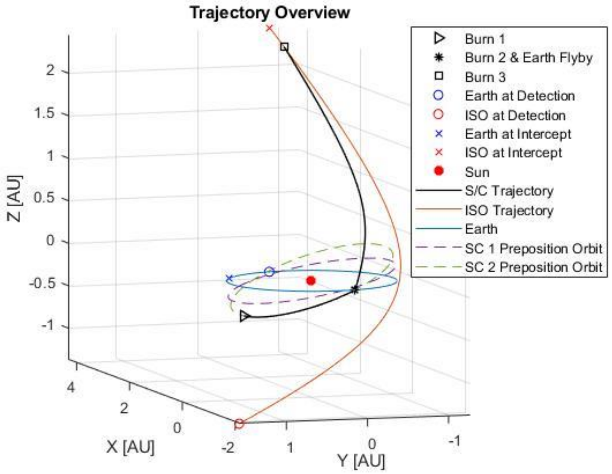
Preliminary Prepositioned System

- Large magnitudes of Delta-V (~22 km/s)
- Extended transfer durations
- Favored electric propulsion configuration



Inclined Gravity Assist Constellation

- Earth gravity assists became necessary to achieve the required high Delta-V
- Shorter transfer durations (as low as 27 days)
- Electric propulsion configuration **not beneficial**



Speaker: Jack K.

System Design

Speaker: Sean T.

Cal Poly Spacecraft Design 20-21

Mission Concept Of Operations

Phase 0
Pre-Launch

The launch services will transport and prepare each launch vehicle to be ready for a launch window around **October 9, 2029 and January 3, 2030.** The ground system (GS) will be prepared for launch by staffing the necessary personnel and creating software for the mission. Each S/C will also be integrated into each launch vehicle.

Phase 1
Launch

The mission will launch separately out of KSC using the launch services provided by SpaceX using **two Falcon Heavy Expendables**. During launch, the GS will monitor the telemetry of the S/C. This phase will conclude when the S/C detumbles after separation from the second stage of the launch vehicle.

Phase 2
Orbital Insertion

After separation from the launch vehicle, each S/C will detumble and begin communication with the ground system. The mission shall have **each S/C complete two gravity assists around Earth to increase inclination**. The final PP orbits of the spacecraft will be a circular, heliocentric orbit at **1 AU with 21 degrees of inclination, and 90 degree RAAN difference**.

Phase 3
PP Orbit Waiting Period

Each of the S/C will be commanded by the GS to complete secondary objective campaigns until an ISO is pursued. **The secondary objectives attempted in this phase include remote observation platform (B.3), and exoplanet platform (B.5)**. Each S/C will be in orbit and ready to conduct objectives before 12/31/2030.

Phase 4
Navigation to ISO

Upon detection of an ISO, **the S/C chosen to pursue will be commanded by the GS to leave its PP orbit on a trajectory towards an Earth gravity assist**. This S/C will then be designated as the Encounter S/C with the other S/C as the Secondary S/C. The mission will use OpNav on the way to the ISO and switch to Autonomous Optical Navigation at a distance of 55,000 km to the ISO.

Phase 5
ISO Flyby

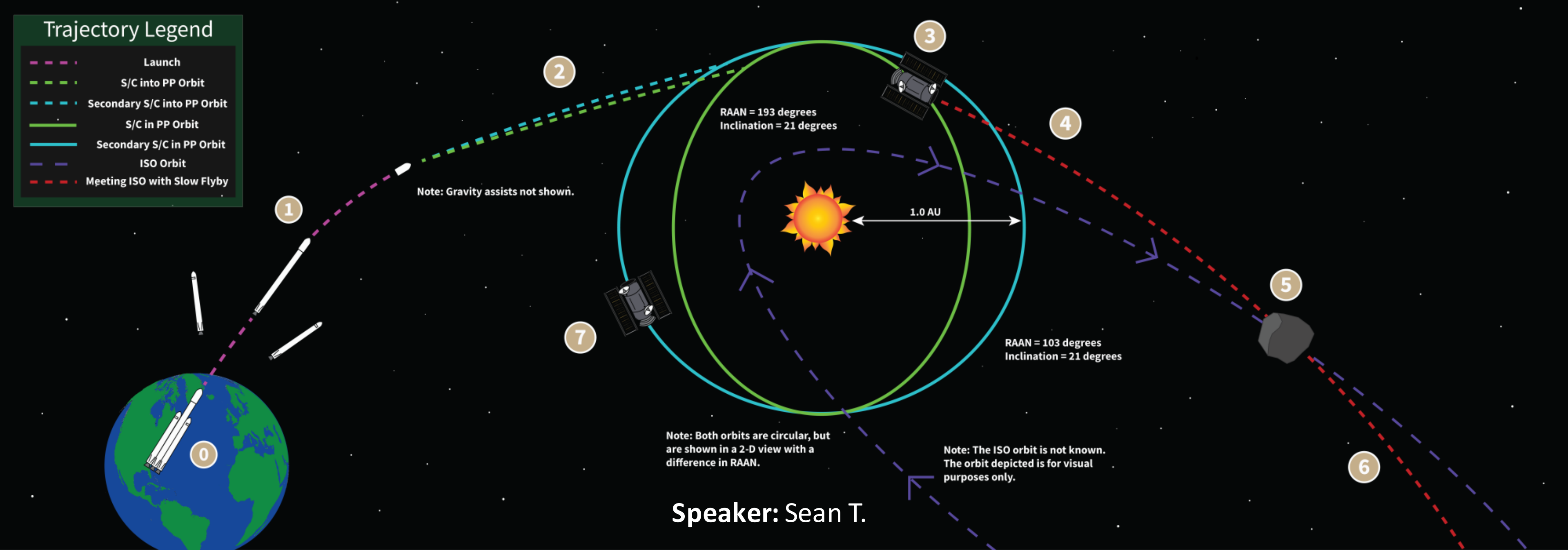
As Autonomous Optical Navigation begins, the flyby of the ISO will begin with a closest approach distance of **400 km and a maximum relative velocity of 13 km/s**. During this phase, the S/C will complete all primary objectives as well as **partial completion of the secondary objective of Advanced Object Definition (B.1).**

Phase 6
Downlink Data

Once the ISO is out of range of the S/C, **data taken during the encounter will begin to be downlinked to the GS.** All of the data must be downlinked before nine months after flyby of the ISO have elapsed. The GS will receive the data and begin processing. Once processed, the data will be made available to the customer.

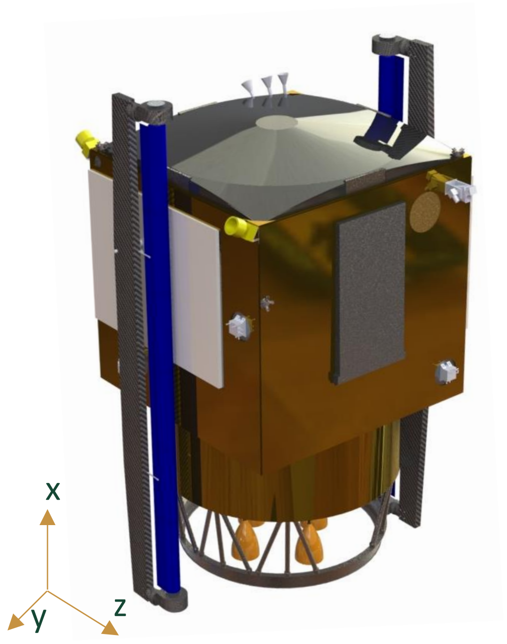
Phase 7
Decommission

After the primary science data has been downlinked back to the GS, the **Encounter S/C will enter the decommision phase** to be prepare for shutoff. During this time, the GS will monitor the Encounter S/C trajectory to ensure a safe graveyard orbit. The **Secondary S/C will remain in its PP orbit for a minimum of 20 years** since emplaced until decommission.

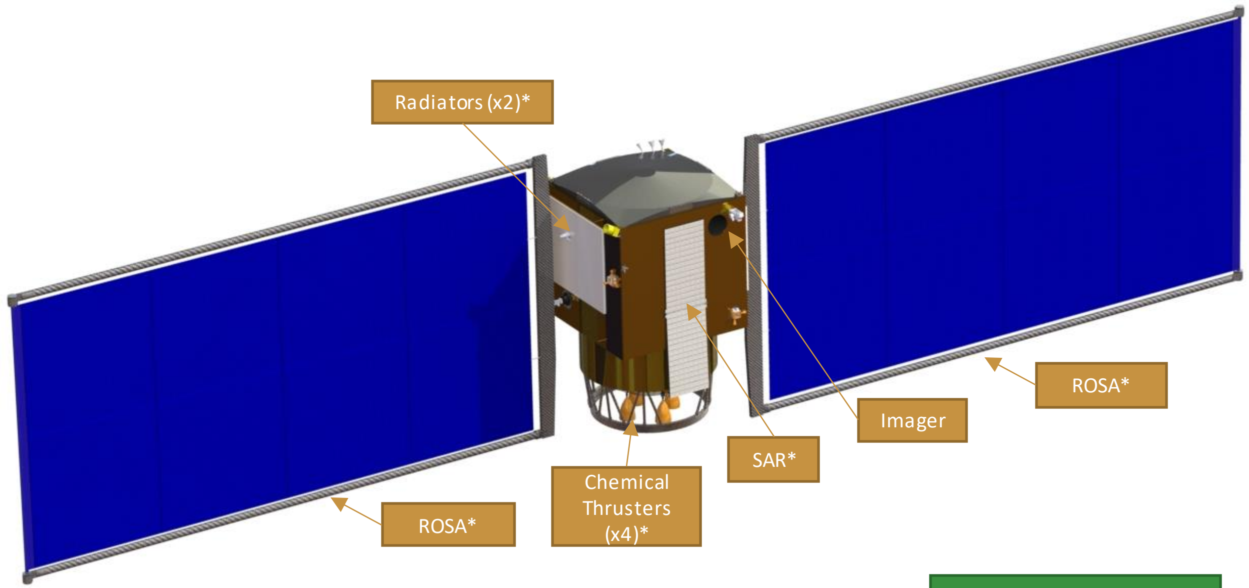


Speaker: Sean T.

The Flight System (1/3)



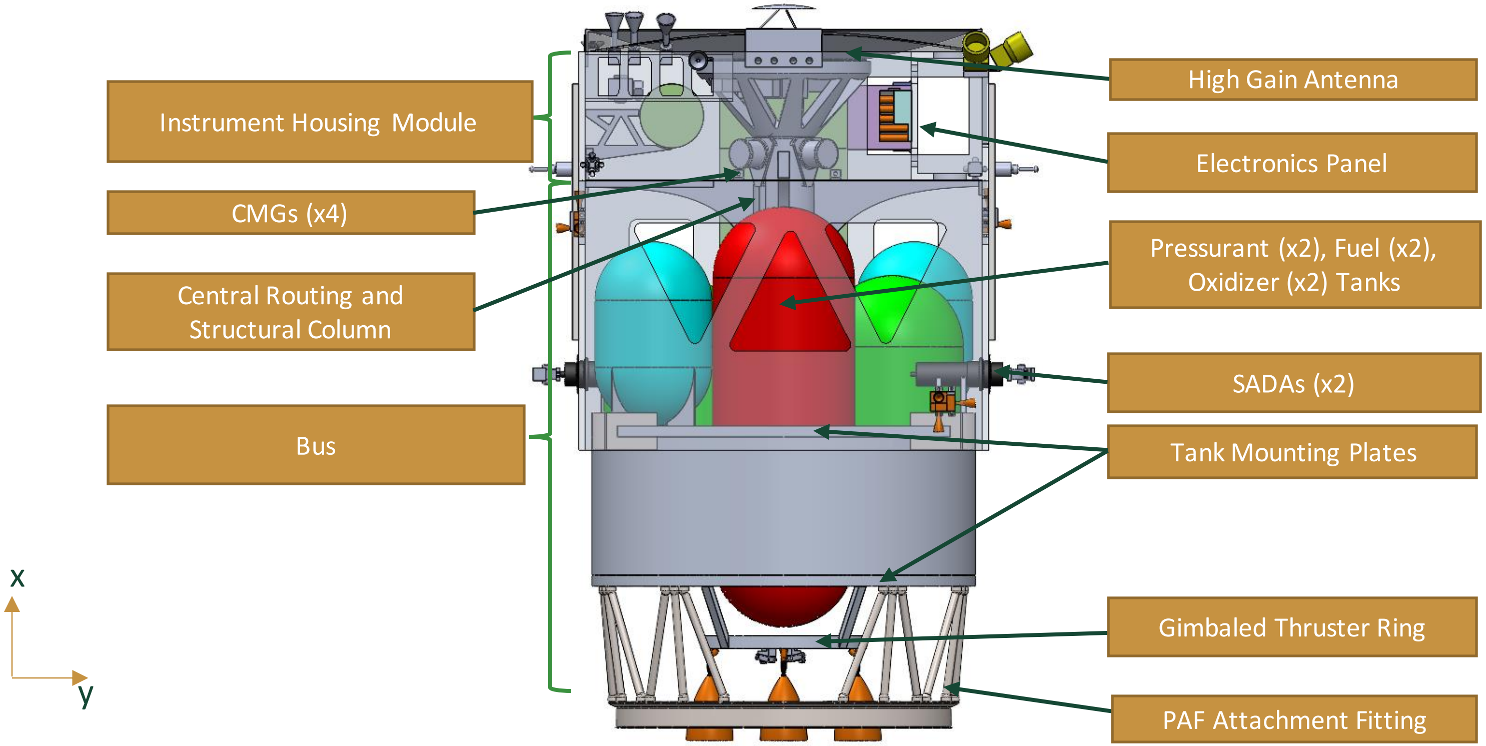
Stowed



Deployed

*Major Changes from SDR

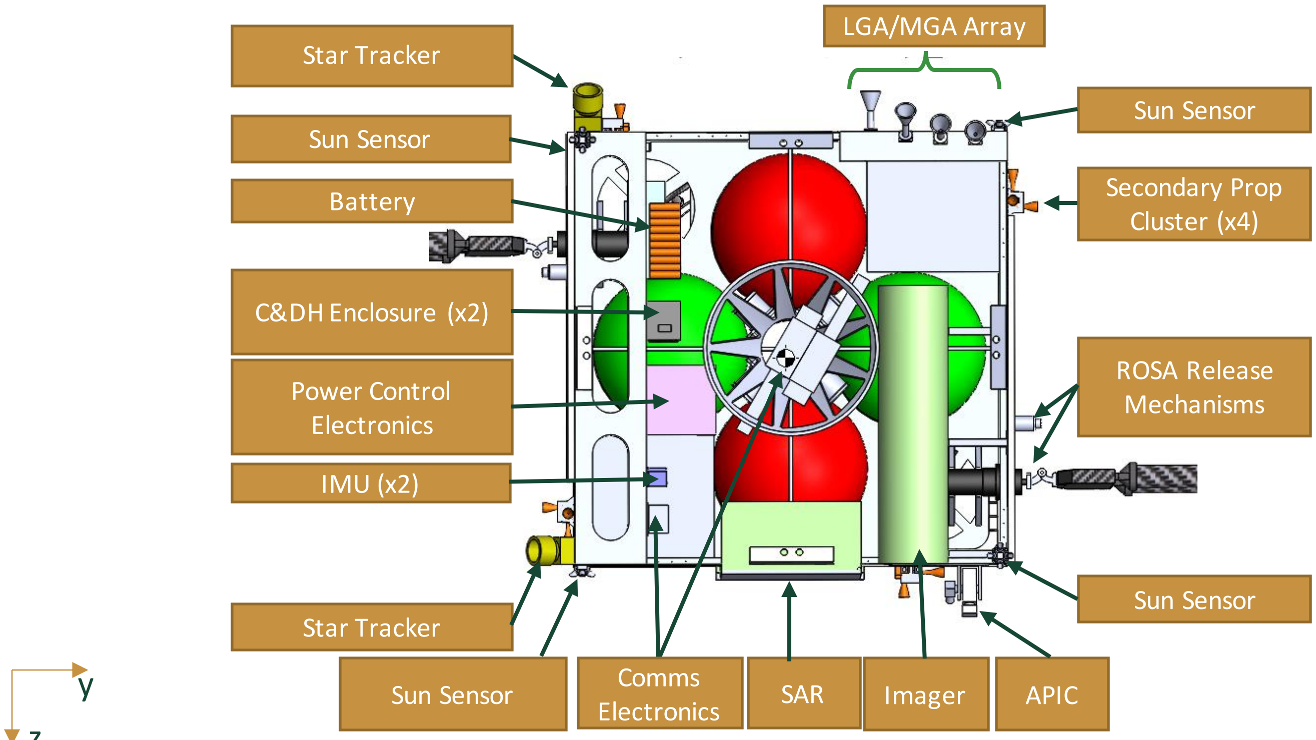
The Flight System (2/3)



Side View

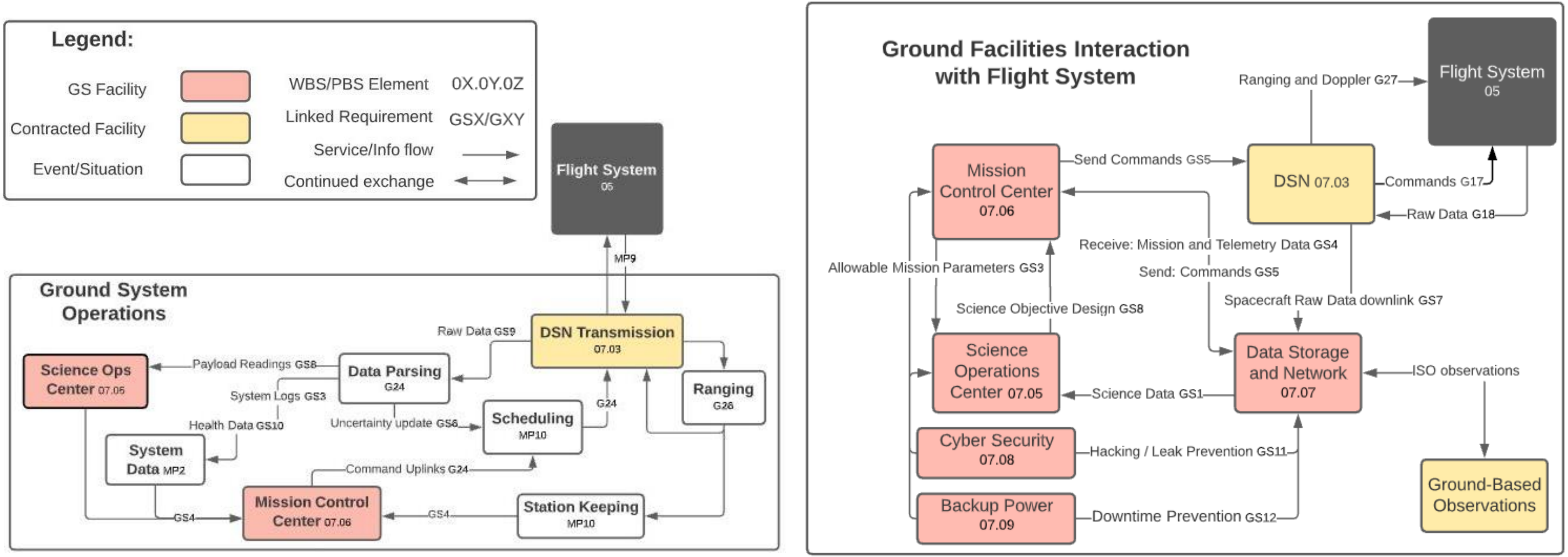
Speaker: Matthew S.

The Flight System(3/3)



Top View

Ground System Block Diagram



Mass Breakdown

Link to Backup

Subsystem	Basic Mass Estimate (kg)	Mass Growth Allowance[1]	Predicted Mass Estimate (kg)
Payload	155	30%	202
Propulsion	657	20%	788
Power	405	20%	486
Comms	101	15%	116
Thermal	224	30%	291
GNC	5	30%	7
ADCS	72	20%	86
Structures	909	20%	1,091
C&DH	60	15%	69
Propellant	10,818	N/A	10,818

Predicted System Mass	Allowable Mass	System Mass Margin
13,953 kg	14,000 kg	0.35%

[1] "Standard: Mass Properties Control for Space Systems" *ANSI/AIAA S-120A-2015)*, American Institute of Aeronautics and Astronautics, Inc., 2015, DOI:10.2514/4.103858.001

Phase 1

Speaker: Austin I.

Phase 1: Launch Overview

- **Launch Vehicle: SpaceX's Falcon Heavy Expendable**
- **First Spacecraft Launch: October 9, 2029 (2-week window)**
- **Second Spacecraft Launch: January 3, 2030 (2-week window)**
  - Launch into predetermined orbit and will separate from the second stage of the Falcon Heavy
  - Includes separation, detumble operations, and deployment order

Backup

Phase 1: Launch

Applicable Level 2 Requirements

ID: F = Flight System
ID: G = Ground System
ID: L = Launch System

ID	Requirement	Compliance (Y/N/M)
F4	The flight system shall survive the launch environment.	No
F6	The flight system shall have a maximum mass of 14,000 kg.	Yes
F7	The flight system shall fit within a volume of 175.15 m^3 with the dimensions specified in Figure 1.	Yes
F30	The flight system shall enter system fault mode in response to mission critical anomalies defined by Table 7.	Yes
G18	The ground system shall receive telemetry data from the launch vehicle.	Yes
L5	The launch system shall provide telemetry data to the ground system.	Yes
L1	The launch system shall be ready to launch no later than October 9th, 2029.	Maybe
L3	The launch system shall have a minimum reliability of 98% per launch vehicle.	Maybe

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Speaker: Bailey G.

Phase 1: Launch

Applicable Level 3 ***Structure Requirements***

ID	Requirement	Driving Phase	Compliance (Y/N/M)
STR1	The structure shall survive the launch environment as defined by the Falcon Heavy User Guide.	Phase 1	No
STR5	The structure shall survive the operational limit loads as defined by Table TBD.	N/A	Maybe
STR12	The structure shall have factors of safety defined by Table 4.3.	N/A	Maybe

Launch Load Cases and FOS

Loading	Scenario
6G Axial	**Launch**
2G Lateral	**Launch**
Modes up to 35Hz	**Launch**
Random Vibrations	**Launch**
Deployment	Operational
Thrusting	Operational
Control Torques	Operational

Factors of Safety

- Dependent on **materials** used
  - Metallic
  - Composite/Bonded
  - Glass/Ceramics
  - Bonds in Glass/Ceramics
  - Softgoods
  - Beryllium
- Dependent on **verification approach**
  - Prototype
  - Protoflight
- Dependent on **type** of design
  - Ultimate Design Factor
  - Yield Design Factor
  - Qualification Factor
  - Proof Test Factor

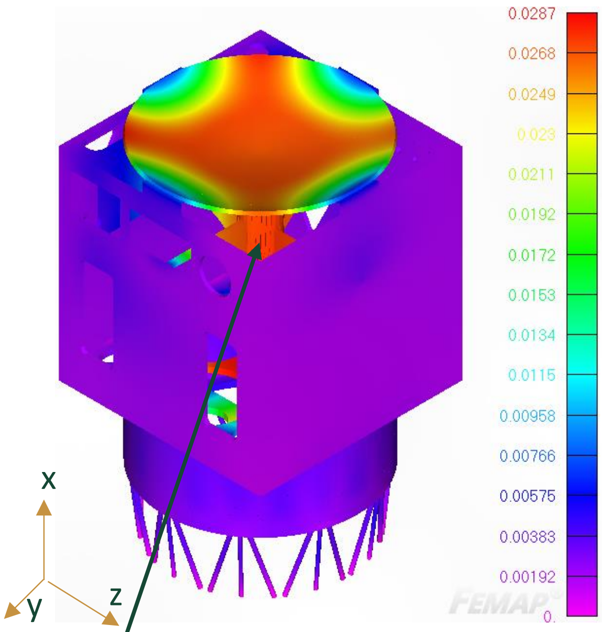
Ex.

AL-7075, Protoflight, Yield Design -> FOS = 1.25

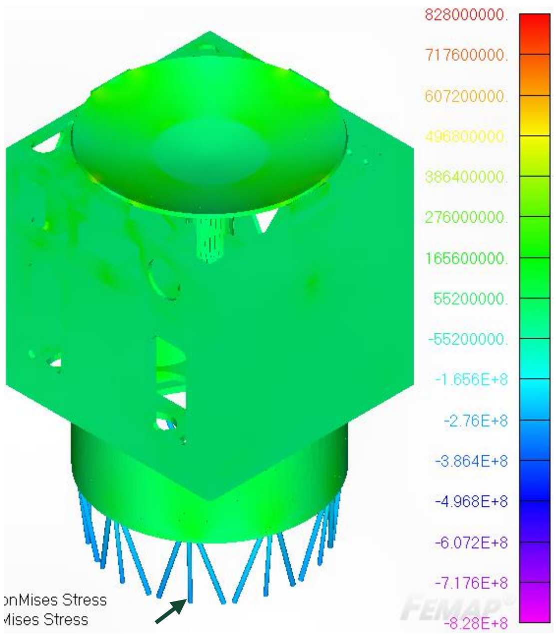
FOS Table

Launch Loading Results: 6G Axial

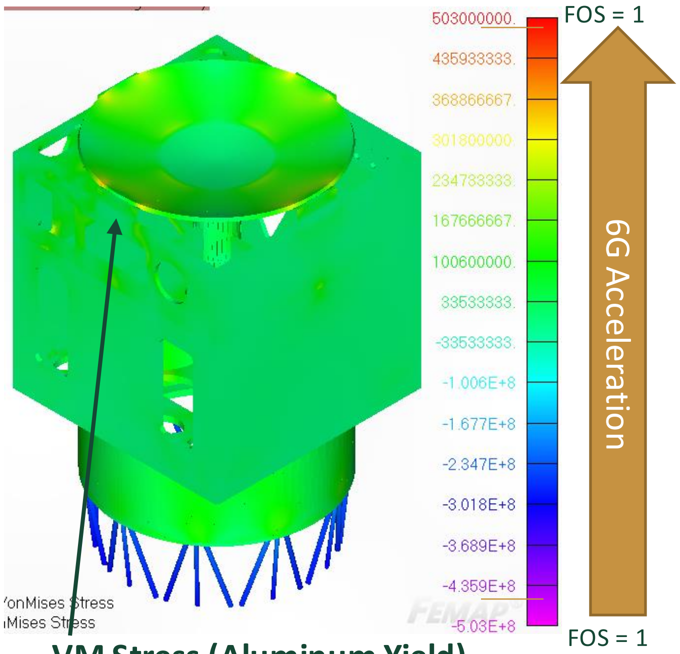
Link to Detailed Analysis Backup



Translation
Max: <3cm



VM Stress (Titanium Yield)
Min. FOS: 2.55

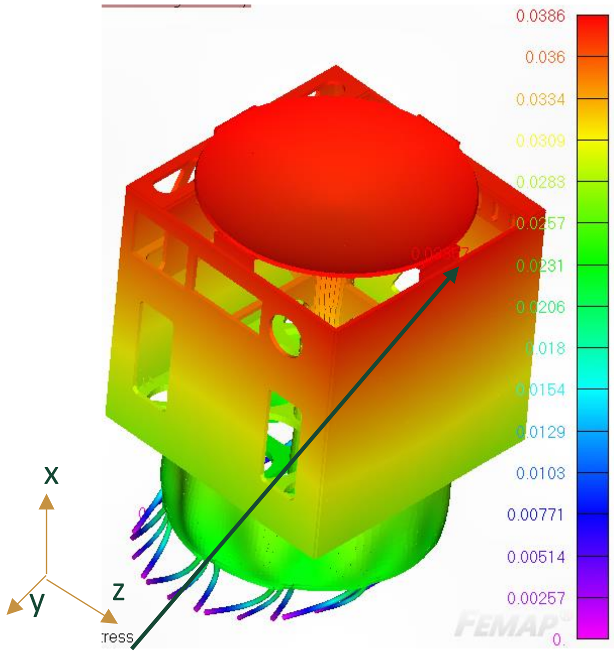


VM Stress (Aluminum Yield)
Min. FOS: <1 (Concentration on Internal Electronics Plate, every other location has high FOS)

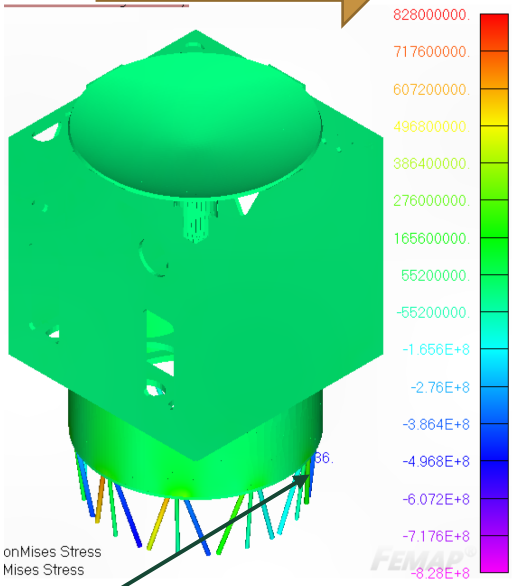
Launch Loading Results: 2G Lateral

Link to Detailed Analysis Backup

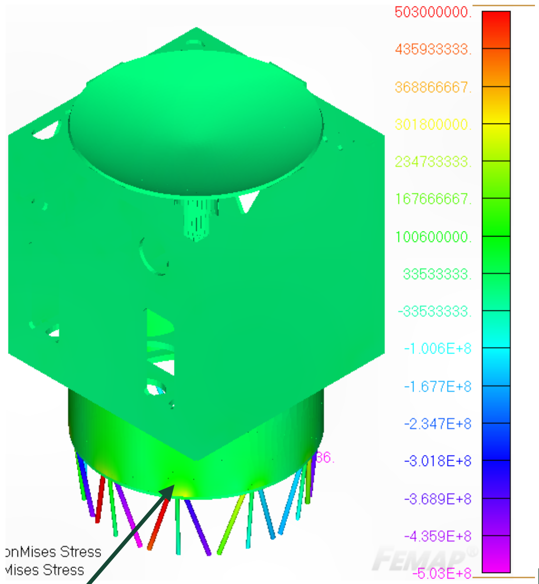
2G Acceleration



Translation
Max: 3.9cm
LV Limit: <11cm



VM Stress (Titanium)
Min. FOS: 1.25

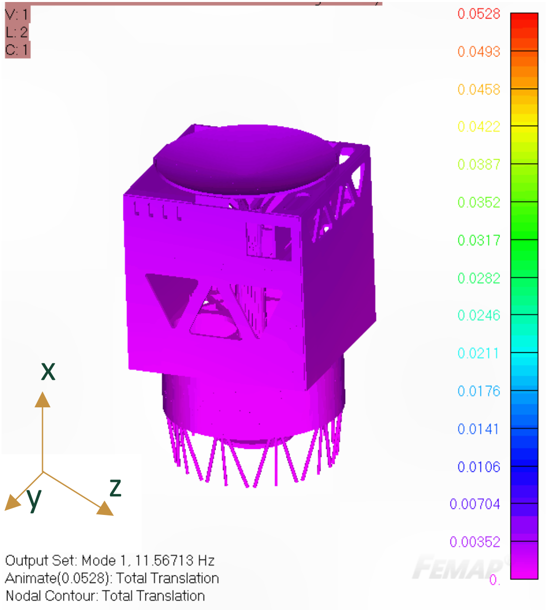


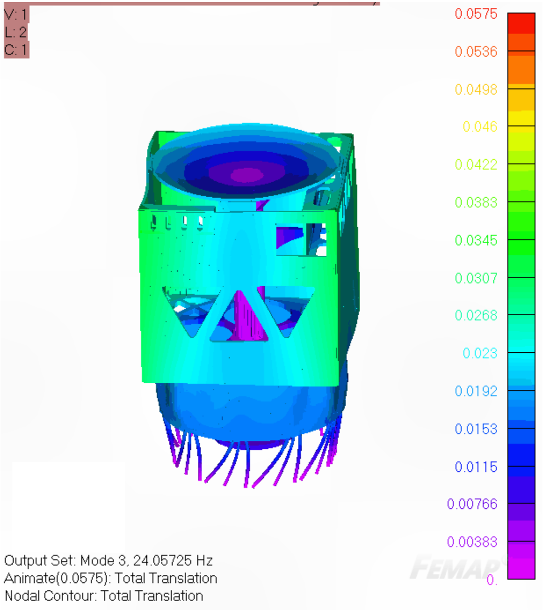
VM Stress (Aluminum)
Min. FOS: 2.26

Speaker: Matthew S.

Open Issue: Launch Modal Results

Link to Detailed Analysis Backup





Currently Not Meeting Launch Vehicle Requirement of under 35Hz

Current mass estimate: 910kg
Maximum mass allocation: 1100kg

Routes for Improvement

- Stiffer base beams or horizontal beams on base
- Added plate near base beams
- Internal gussets between corners of bus along plane of translation

Mode 1: 11.5Hz
Max Translation: 5.3cm
LV Limit: <7cm

Mode 3: 24.1Hz
Max Translation: 5.8cm
LV Limit: <7cm

Phase 1: Launch

Applicable Level 3 ***Structure Requirements***

ID	Requirement	Driving Phase	Compliance (Y/N/M)
POW2	The power system shall generate a minimum of 690 Watts + TBD Watts at end of life.	N/A	Yes
F7	The flight system shall fit within a volume of 175.15 m^3 [TBC] with the dimensions specified in Figure 1.	Phase 1	Yes
STR7	The structure shall sense the solar array's angular position within **+/- 0.02 degrees**.	N/A	Yes
STR8	The structure shall articulate the solar arrays **+/- 179 degrees** from the zero position.	N/A	Yes

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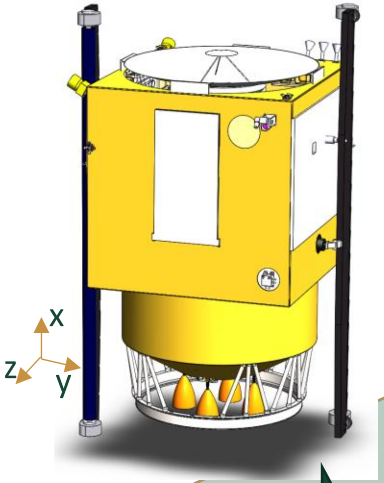
Speaker: Anthony G.

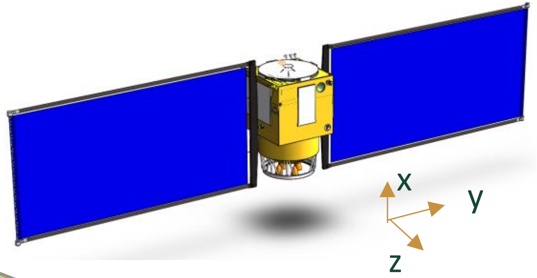
ROSA Deployment

Stowed



Deployment

Deployed



1. Stabilize Spin → 2. Non-Explosive Actuators release ROSA structure from top mount → 3. Pre-loaded hinges rotate ROSA structure 90 deg. → 4. ROSAs "Roll Out"

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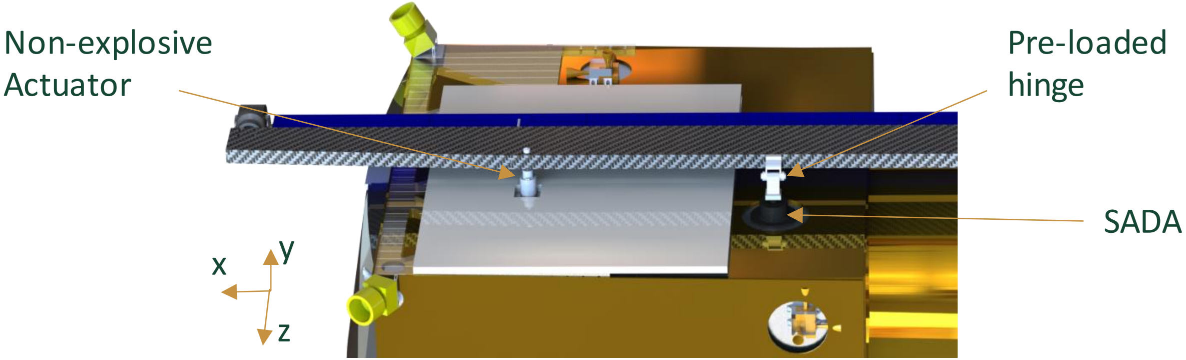
Speaker: Anthony G.

Mechanisms - Structures

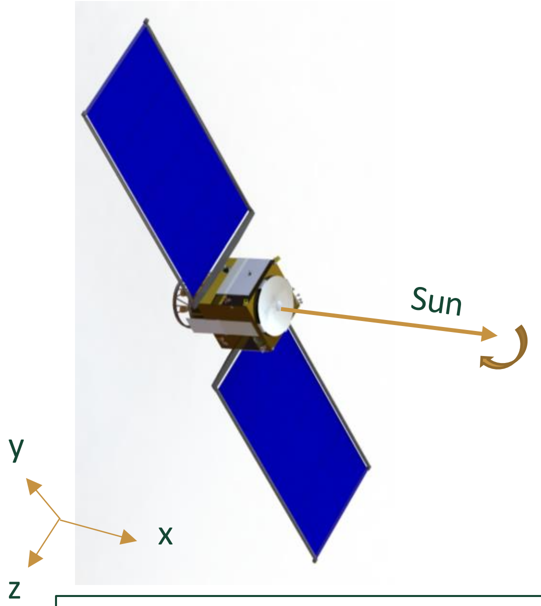
Mechanism	Source	Mass	Continuous Draw	Actuation Draw	Actuation Type	Range	Resolution	Operating Temperatures	Details
SADA (x2)	MOOG High Power Type 5	40kg	<20W	20W	Motor	+/-179 deg.	+/-0.02 deg.	-50 C to +70 C	[1]
Roll Out Solar Arrays (ROSAs, x2)	DSS ROSA	600 kg	N/A	N/A	Roll out	N/A	N/A	-65 C to +90 C	[3]
Non-Explosive Actuator for ROSA (x2)	EBAD NEA HDRM	4.3kg	250 mA	4 A (release current)	Hold Down and Release	N/A	N/A	-240 C to +135 C	[4]
Deployment Hinge for ROSA (x2)	Deployment System for Large Appendages	3kg	N/A	N/A	Spring driven	90-180 deg.	+/- .006 deg	-30C to +50 C (Survivable temperatures +/-150 C)	[5]



Detumble and System Fault Mode

Procedure (system fault mode):

1. System will turn off power for SAR, Imager, primary thrusters, Ka band travelling tube amplifiers, and APIC
2. IMUs will determine the spacecrafts inertial angular rates, using secondary thrusters to nullify angular rates
3. Once stabilized, the solar arrays will deploy
4. Sun sensors will locate the sun relative to the spacecraft
5. Solar arrays and HGA face will be positioned to point at the sun, then the solar array drives will lock the spacecraft into the sun pointing configuration
6. LGA will begin transmitting system level health data, and any secondary faults that were run
7. Secondary thrusters will rotate the spacecraft about the HGA boresight axis at 3 deg/min
8. System will await commands from ground



In this configuration the system will be **power positive, thermally stable** and **communication capable**.

Comm backup

Power backup

Thermal backup

System Fault Mode

Applicable Level 3 ***Telecom Requirements***

ID	Requirement	Driving Phase	Compliance (Y/N/M)
COM18	The communication system shall transmit telemetry at a minimum data rate of 20 bps [TBC] while in system fault mode.	Phase 6	Yes
COM19	The communication system shall receive commands at a minimum data rate of TBD while in system fault mode.	Phase 6	Maybe
COM20	The communication system shall be capable of continuous transmission.	N/A	Maybe
COM21	The communication system shall be capable of continuous reception.	N/A	Maybe

Speaker: Josh F.

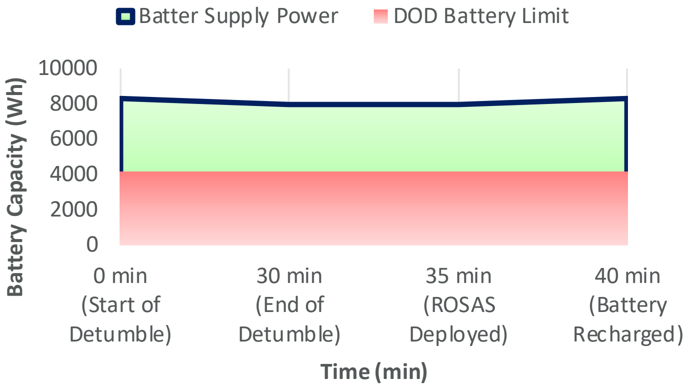
Phase 1: Launch

Applicable Level 3 ***Power Requirements***

ID	Requirement	Driving Phase	Compliance (Y/N/M)
POW1	The power system's battery shall be capable of supplying a minimum of **393 + TBD Whr** of power before solar array deployment.	Phase 6	Yes
POW5	The power system shall provide **305 +/- TBD Watts** during system fault mode.	N/A	Yes

Battery Schedule

Phase 1 Detumble Timeline



- Detumble phase is estimated to take 30 minutes at a respective system power draw of **548 W**. Given this power need the battery system is capable of detumbling for ~14 hours without array deployment.

Subsystem	Component	Power Draw (W)	Detumbling (30 mins)	ROSA deployment (5 mins)
Thermal	Chem Prop Heater	150	ON	ON
	Temp feedback devices	1	ON	ON
	Pumps for Cooling	50	ON	ON
ADCS	CMGs (x4)	148	OFF	ON
	A-STRs (x2)	18	ON	ON
	IMUs (x2)	20	ON	ON
C&DH	Flight Computer	95	ON	ON
	Health Sensors	5	ON	ON
Structures	ROSA Deployment	40	OFF	ON
Propulsions	Secondary Thrusters	194	ON	OFF
Comms	Deep Space Transponder	13	ON	ON
	Ultra Stable Oscillator	3	ON	ON
		Total (W):	**548**	**543**

Initial Battery Specifications

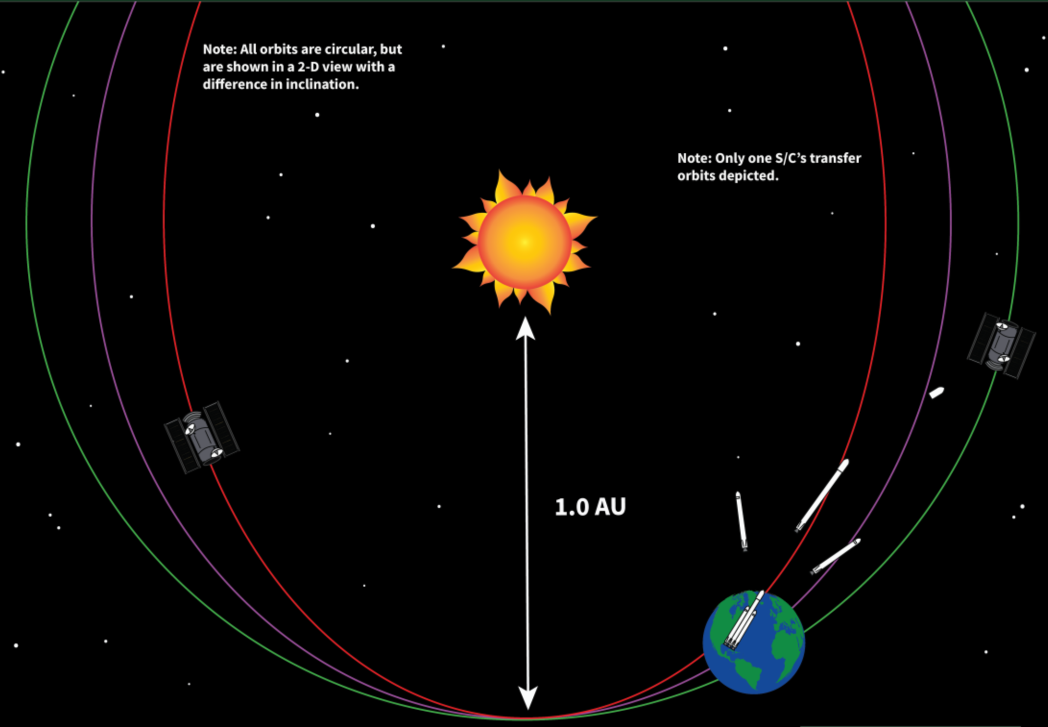
Battery (Li-Ion)	Size (m^3)	Mass (kg)	Capacity (Wh)	Cell Configuration
SAFT VL51ES	0.06892	80.4	10,452	6p10s

Phase 2

Speaker: Sean T.

Launch to Orbital Insertion: Phase 1 and 2

Mission Concept of Operations

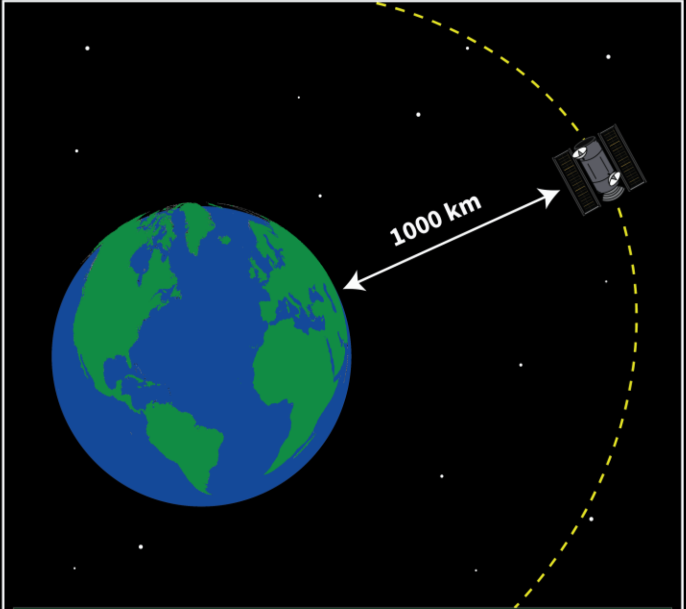


Phase 1: Launch

Two identical S/C will be launched out of KSC three months apart **using Space X's Falcon Heavy Expendable** each time. Each launch will **direct inject into a circular, heliocentric orbit at 1 AU and an inclination of 2.5 degree**s. The first S/C launched will have a **RAAN of 193 degrees and an Argument of Periapsis of 270 degrees while the second S/C launched will have a RAAN of 103 degrees and an Argument of Periapsis of 0 degrees.** This phase will conclude with the detumble of the S/C after separation.

Legend

- Final PP Orbit
- Transitional PP Orbit
- Initial PP Orbit
- Gravity Assist Trajectory



Phase 2: Gravity Assists

Each S/C will complete two gravity assists around Earth on the way to the final prepositioned orbits to increase inclination. The **gravity assists are bounded at a minimum altitude of 1000 km with each launch achieving a C3 of 2 km/s**.

Final PP Orbit Parameters

S/C	Semi-Major Axis	Eccentricity	RAAN	Inclination	Argument of Periapsis
A	1 AU	0.0167	193°	21°	270°
B	1 AU	0.0167	103°	21°	0°

Phase 2: Orbital Insertion

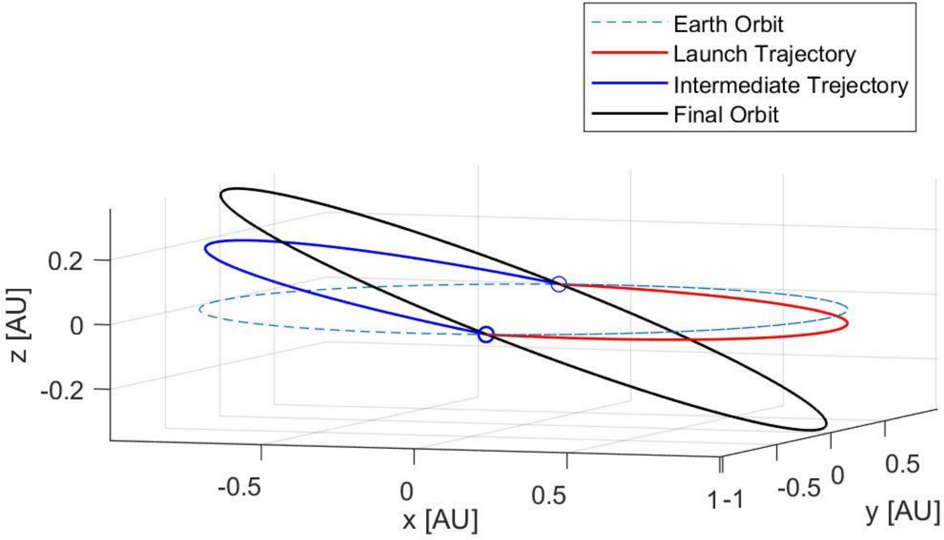
Applicable Level 2 Requirements

Backup

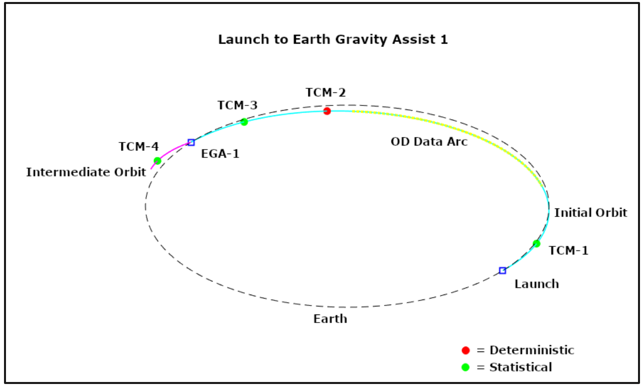
ID	Requirement	Compliance (Y/N/M)
L2	The launch system shall deliver the flight system to an orbit defined by Table 6.1 by 12/31/2030.	Yes
L4	The launch system shall be capable of delivering the mass and C3 combinations given in Figure 2.	Yes
F22	The flight system shall be compatible with the deep space network.	Yes

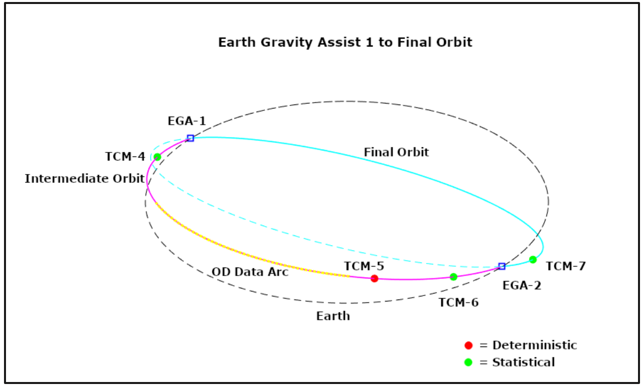
Gravity Assists - Orbits

Spacecraft	Gravity Assist/Launch Date	Orbit Inclination (deg)
A	Oct. 9, 2029	2.5
	Apr. 5, 2030	11
	Oct. 9, 2030	21
B	Jan. 3, 2030	2.5
	Jul. 8, 2030	11
	Jan. 4, 2031	21



Navigation Timeline - GNC





Maneuver	Date (COSMIC A | COSMIC B)	Purpose
TCM 1	July 25, 2029 | October 26, 2029	Separation Cleanup
TCM 2	November 20, 2029 | February 17, 2030	EGA-1 Maneuver
TCM 3	December 15, 2029 | March 14, 2030	TCM 2 Cleanup
TCM 4	January 24, 2030 | April 23, 2030	EGA-1 Cleanup
TCM 5	May 21, 2030 | August 23, 2030	EGA-2 Maneuver
TCM 6	June 15, 2030 | September 17, 2030	TCM 5 Cleanup
TCM 7	July 25, 2030 | October 27, 2030	EGA-2 Cleanup

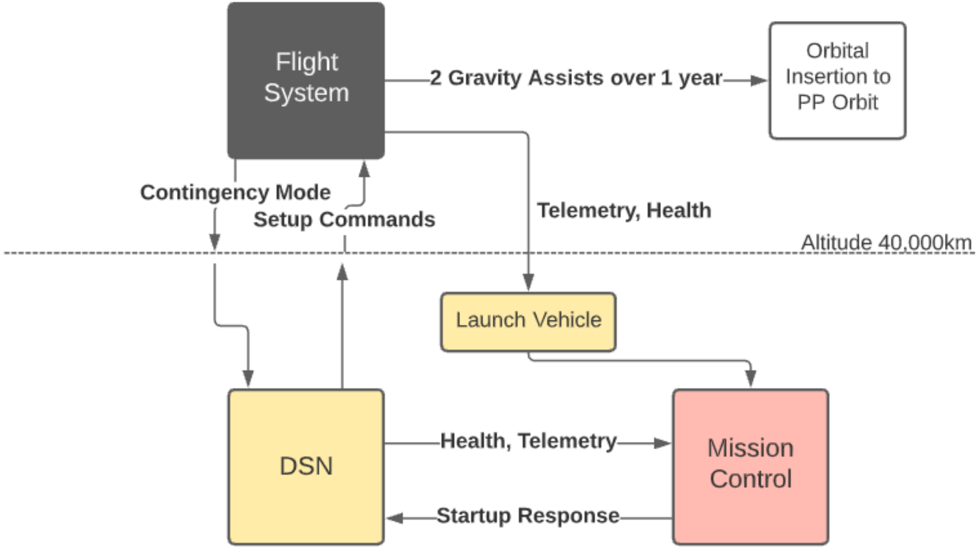
Phase 2: Orbital Insertion

Applicable Level 2 ***Ground System Requirements***

ID	Requirement	Compliance (Y/N/M)
G33	The ground system shall establish two-way communication sessions with the flight system on **average of three times a week for four hours per session** during the orbital insertion mission phase	Yes

Ground Operations – Orbital Insertion

- Launch Provider handles actions prior to separation, later sends data to Ground
- DSN Communication occurs at altitudes surpassing 40,000km for startup
  - Gravity assist within 1000km of Earth will be without transmission/reception
- Secondary science can begin after initial contact/setup, during emplacement
- Health checks
- Note: Startup Response procedures are the same to system fault Continency Operations



Phase 3

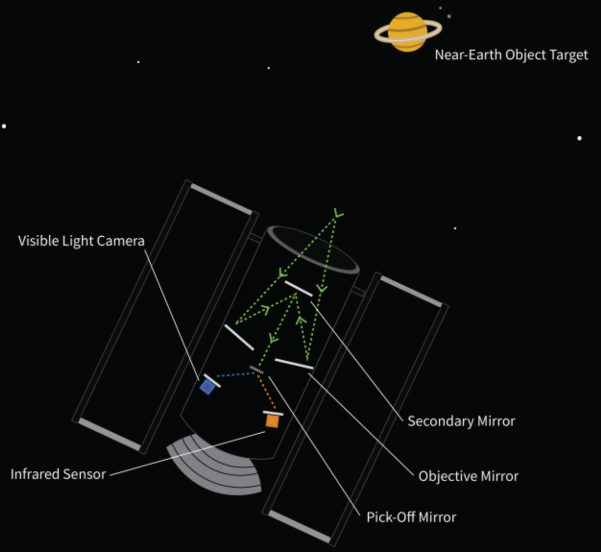
Speaker: Sean T.

Prepositioned Orbit Waiting Period: Phase 3

Mission Concept of Operations

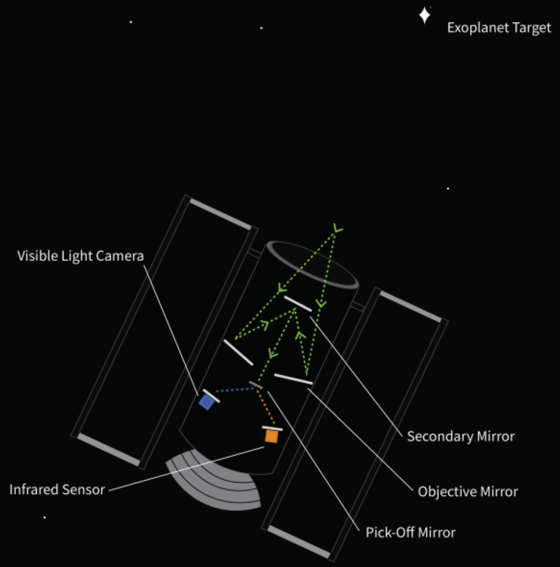
B.3 Remote Observation Platform

To meet B.3 Remote Observation Platform, the **S/C will observe objects for 2-4 weeks at a time.** These will be follow-up observations, and will alternate with B.5 Exoplanet Platform campaigns. Images will be acquired with the primary telescope so that **the target is observable with an SNR greater than or equal to 2.5**, based off of known near earth objects or heliocentric orbiting bodies. These observations will provide further knowledge of the targets to the scientific community, potentially used to derive spin rates or see how their orbits change over time. The campaigns pursued will be determined by the customer.



B.5 Exoplanet Platform

To complete B.5 Exoplanet Platform, each spacecraft will perform **follow-up science on known exoplanets for 2-4 weeks at a time, acquiring images every 30 minutes to build light curves** that will be analyzed on the ground to look for exoplanet transits. Targets will be selected based on known exoplanets with **apparent magnitude of their stars brighter than Magnitude 14.43 to ensure the transits are observable**. This follow-up science will provide greater knowledge of these exoplanets and their orbital characteristics. Again, the campaign pursued will be determined by the customer.



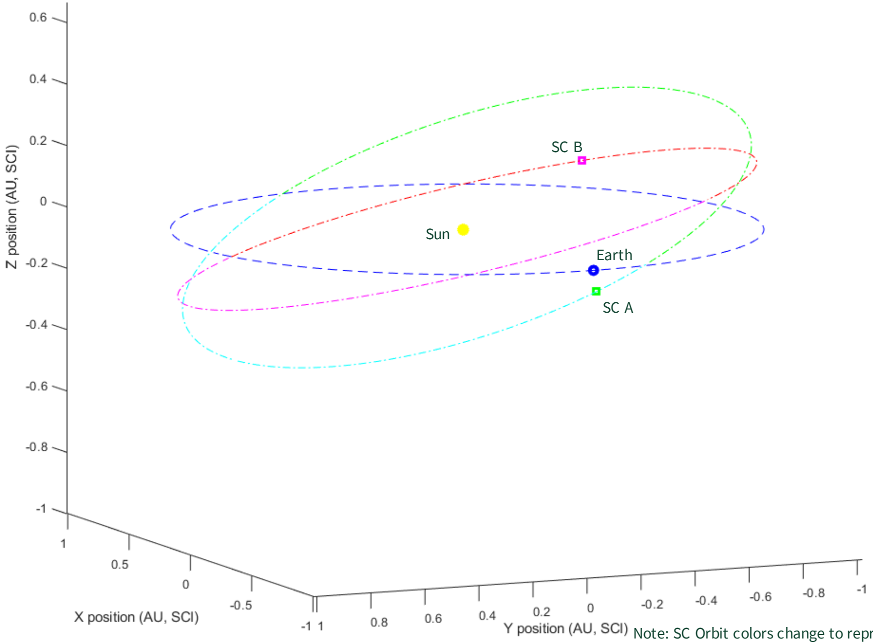
Pre-Positioned Orbit Model

- 2 spacecraft in the orbits described in Table and shown in Figure
  - Each spacecraft has 1 Earth approach every 6 months
- Advantages
  - Gravity Assists: the system collectively has 1 potential assist every 3 months
    - This **reduces onboard DV** required by **up to 7 km/s**
    - Potential to assist station-keeping
  - Low emplacement cost: Takes advantage of assists and launch vehicle
  - Stable Heliocentric Orbit: benefits various systems e.g. Power, Comms, Thermal, Secondary Science
- Coverage
  - Over 2000 20-year missions were simulated, with Monte Carlo style analysis run on the results
  - The system as designed has **at least an 87% chance** of encountering at least 1 ISO over a 20-year mission
    - This is tied to other design factors such as propulsive capacity and relative speed at encounter

Table: PP Orbit COE

	a (AU)	ecc	Inc (°)	Ω (°)	ω (°)
SC A	1	0.0167	21	193	270
SC B	1	0.0167	21	103	0

Figure: Prepositioned Orbits with Earth and Spacecraft positions on 3/30/31



Note: SC Orbit colors change to represent crossing Earth's Orbital Plane

Phase 3: Prepositioned Orbit

Applicable Level 2 Requirements

Backup

ID	Requirement
F23	The flight system shall stay in the prepositioned orbit defined by Table 6.2 for a minimum of 10 months.
F24	The flight system shall observe 0.0126% of sky per month.
F25	The flight system shall detect objects with a limiting apparent magnitude of 14.5 in the visible band.
F29	The flight system shall detect objects with a limiting apparent magnitude of 12.5 in the near-infrared band.
F26	The flight system shall acquire photometry data with a signal to noise ratio of at least 2.5
F27	The flight system shall monitor one star system for at least 14 days.

CAL POLY

Speaker: Bailey G.

Phase 3: Prepositioned Orbit

Applicable Level 3 ***Payload Requirements***

ID	Requirement	Driving Phase	Compliance (Y/N/M)
PAY4	The payload shall be capable of taking a visible image once every 200 milliseconds.	Phase 5	Y
PAY6	The payload shall be capable of taking a visible image with an exposure time of 13 milliseconds.	Phase 5	Y
PAY8	The payload shall have an FOV of 0.55 +/- TBD degrees.	Phase 5	Y

Science Collection – Payload

Secondary science to be conducted with Primary Imager

Remote Observation	Exoplanet Campaign
	6 total campaigns
6-month observation campaign	Each is a 2-4 week follow-up observation using transit method
Visual or IR images	Visual or IR images
Images taken every hour	Images taken every 30 minutes
~8 GB of memory required	~8 GB memory required

DSN Schedule	Available Data Rate	Link Margin	Antenna
4-8 hrs per s/c per week	3 Mbps	> 5 dB	HGA

Phase 3: Prepositioned Orbit

Applicable Level 3 ***ADCS Requirements***

ID	Requirement	Driving Phase	Compliance (Y/N/M)
ADC2	The ADCS shall point the optical boresight with an accuracy of 810 arcsec during secondary science acquisition.	Phase 3	Y
ADC4	The ADCS shall be capable of a maximum slew acceleration of **TBD** deg/s^2 during secondary science acquisition.	Phase 3	M
ADC5	The ADCS shall point the boresight of the high gain antenna with an accuracy of 360 arcsec during telecom.	All	Y
ADC6	The ADCS shall determine the direction of the Sun relative to the center of the solar array with an accuracy of **TBD** arcsec.	All	M
ADC8	The ADCS shall command the solar array drive assemblies to point at the Sun with an accuracy of 10 degrees.	All	Y
ADC10	The ADCS shall control the imager boresight stability to 11 arcsec/sec during secondary science acquisition.	Phase 3	Y

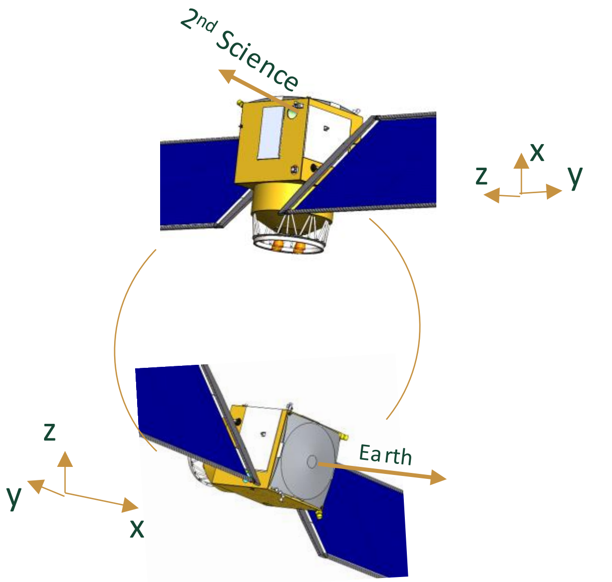
Science Collection - Pointing

Pointing Summary

HGA: Earth pointed for 4-8 hrs once a week to communicate with the DSN

Solar Arrays: Sun pointed

Imager: Will alternate between performing exoplanet campaign and remote observation



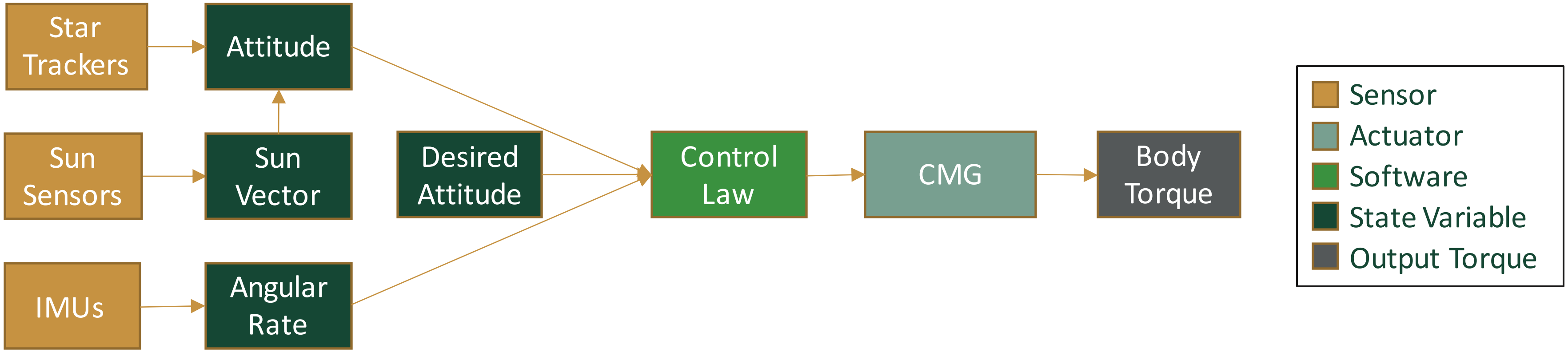
Phase 3: Prepositioned Pointing Cycle

Pointing Budget Description	Per-Axis Error Value (3 σ) [arcsec]	Radial Pointing Error (3 σ) [arcsec]	Radial Pointing Requirement [arcsec]	Stability Error (3 σ) [arcsec / s]	Stability Requirement [arcsec/s]
Preposition Science	189.69	201.58	810	10.81	10.81
Solar Arrays	260.30	468.54	36000	N/A	N/A
Downlink	188.30	213.66	360	N/A	N/A

CAL POLY

Speaker: Maya G.

ADCS Overview



Component	Quantity	Manufacturer	Mass (Each)	Power (Each)	Performance
Star Tracker	2	Leonardo A&SS	3.55 kg	8.9 - 13 W	2°/s tracking rate
Sun Sensor	4	Bradford Space	0.215 kg	0 W (Passive)	< ±1.5 accuracy on boresight
IMU	2	InnaLabs	2 kg	10 W	Range: ± 400°/s , ± 40g
CMG	4	Blue Canyon	15 kg	20 - 35 W	12 Nm Max Torque

Mass Memory Card Rationale

Applicable Level 3 ***C&DH Requirement***

ID	Requirement	Driving Phase	Compliance
CDH7	The C&DH system shall have a minimum storage capacity of 21 GB [TBC].	Phase 3	Yes

- PP Science data drives MMC (~16 GB), not Encounter data (~470 MB)
- Each will carry min. 21 GB
  - Additional ~5 GB health data system-wide
- Final campaign may differ
  - e.g, a 12-hour Remote Observation campaign at 5 sec cadence requires 16 GB
- Each C&DH system will have one additional MMC for redundancy (2x per system, 4 total)

Instrument, *Phase*	**Data Storage (MB)**
SAR	320.8
Vis. Imager	-
Remote Obs.	8078.4
*Exoplanet**	1344
Encounter	112.2
IR Imager	24

*one campaign; conceptual year-plan accounts for 6 campaigns

Backup

Phase 3: Prepositioned Orbit

*Applicable Level **3 Shielding Requirements***

ID	Requirement	Driving Phase	Compliance (Y/N/M)
STR4	The structure shall shield sensitive components from the space environment as defined by Table 4.2.	N/A	Y
THR1	The thermal system shall keep components that are turned on within their operational temperature ranges as defined by Table 3.1.	N/A	Y
THR2	The thermal system shall keep components that are turned off within their survivable temperature ranges as defined by Table 3.2.	N/A	Y
COM20	The communication system shall operate in the space environment for 22 years.	N/A	M

CAL POLY

Speaker: Anthony G.

Spacecraft Shielding - Structures

MLI

- The MLI shielding is a 15 mm thick, 30 layer blanket consisting of aluminized Mylar and Dacron
- Packing density of 30 layer/15 mm was chosen to reduce conductive shorting
- High gain antenna will be covered in a germanium coated polyimide, which is RF transparent

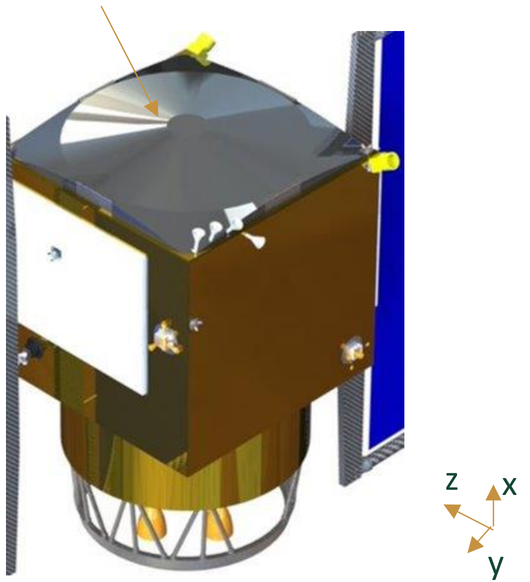
MMOD Shielding

- Shielding consists of honeycomb aluminum walls with 1 mm wall thickness on each side with a 5 cm honeycomb core.
- This shield can protect against particles with a critical diameter up to 1.5 cm

Link to Detailed MLI

Link to Detailed MMOD

Germanium coated kapton MLI for HGA



Preposition Risks - Structures

Given that...	There is a possibility of...	Adversely impacting...	Which can result in...	Likelihood (out of 5)	Severity (out of 5)	Sub team
The exposure periods are longer than expected	The shielding not being able to withstand the exposure	The payload and support bus	The shielding degrading	1	2	Structures

	0-20%	20-40%	40-60%	60-80%	80-100%
Likelihood	1	2	3	4	5

	Minimal impact on objectives	Minor impact on objectives	Unable to achieve a particular objective	Unable to achieve multiple objectives	Unable to achieve all objectives
Severity	1	2	3	4	5

CAL POLY

Speaker: Ismael C.

Phase 3: Prepositioned Orbit

Applicable Level 3 Propulsion Requirements

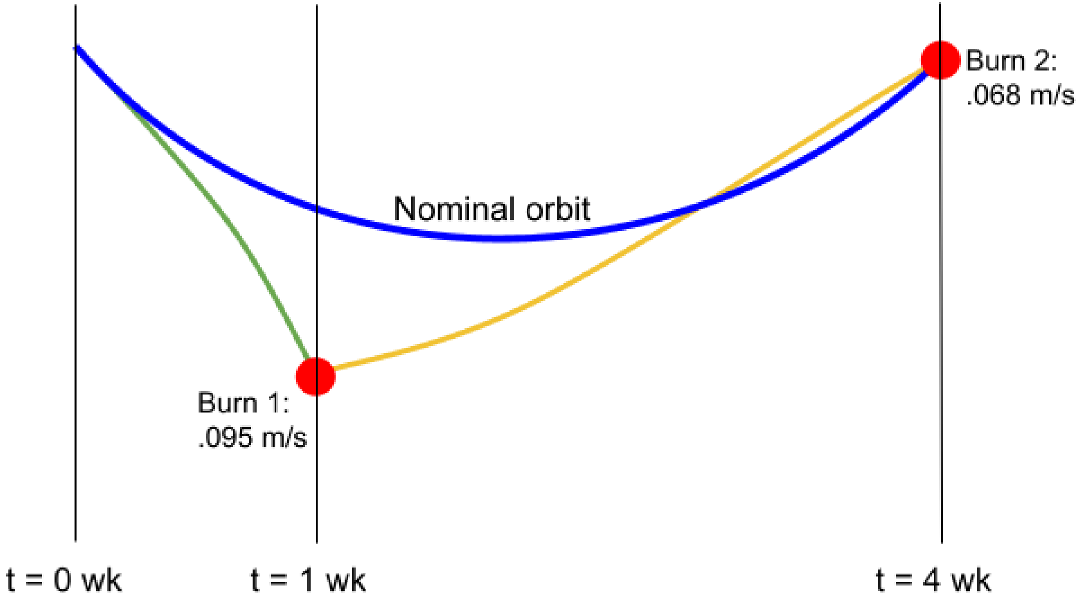
Subsystem	ID	Requirement	Driving Phase	Compliance
Propulsion	PROP2	The propulsion system shall provide the flight system with a dV of 60 +/- 15 m/s for station-keeping within the prepositioned orbit.	Phase 3	Y
Propulsion	PROP2	The propulsion system shall provide the flight system with a dV of 15 +/- 5 m/s for desaturation within the prepositioned orbit.	Phase 3	Y

Speaker: Chris L.

Station Keeping – Orbits & Propulsion

- Required to maintain the gravity assist schedule
- SRP (s/c only within Earth SOI for ~1.1% of orbit)
- **4wk cycle: 1wk wait, 3wk transfer**
- **0.16m/s of DV per 4wk cycle * 21 years of station keeping = 45 m/s** total DV needed for the 21 years of emplacement + prepositioned orbit

Station Keeping 4wk Cycle Overview



Phase 3: Prepositioned Orbit

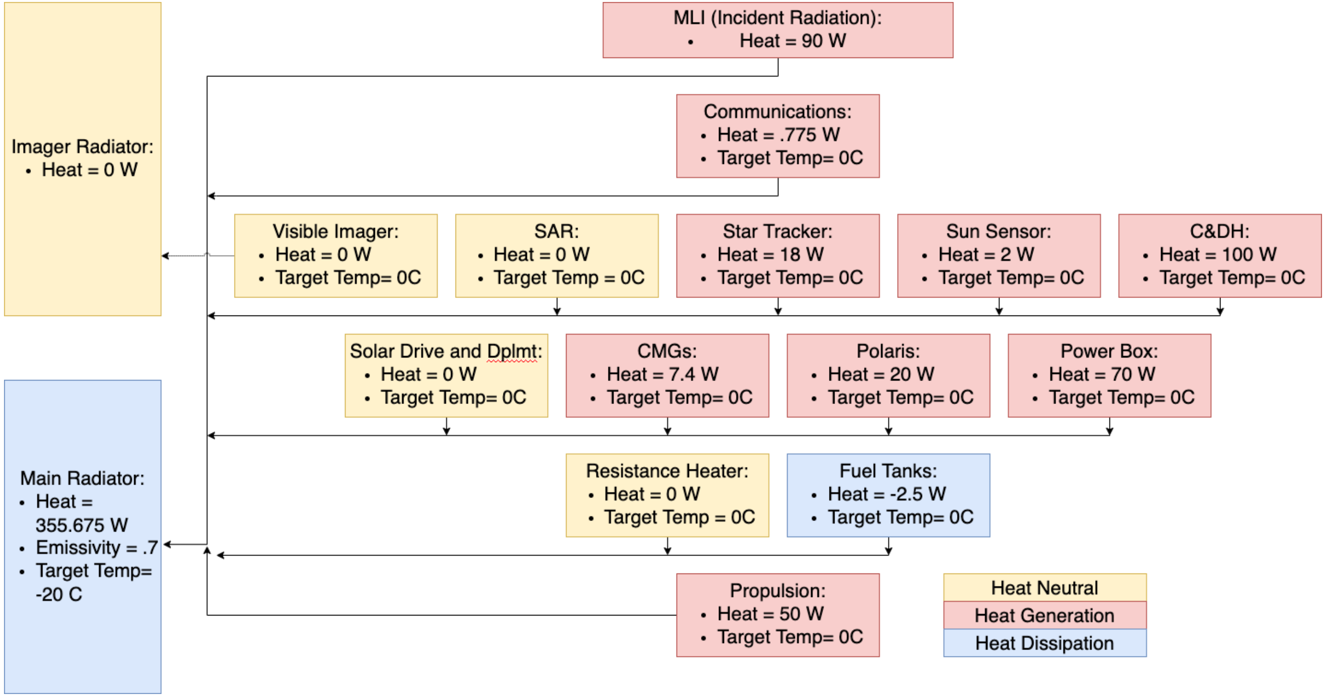
Applicable Level 3 Thermal Requirements

Subsystem	ID	Requirement	Driving Phase	Compliance (Y/N/M)
Thermal	THR1	The thermal system shall keep components that are turned on within their operational temperature ranges as defined by Table 3.1.	N/A	Y
Thermal	THR2	The thermal system shall keep components that are turned off within their survivable temperature ranges as defined by Table 3.2.	N/A	Y

CAL POLY

Speaker: Alec J.

Prepositioned - Thermal Design



Back Up Slides

Radiator Pointing Error Risk

Given that...	There is a possibility of...	Adversely impacting...	Which can result in...	Likelihood (out of 5)	Severity (out of 5)	Sub team
There are pointing errors which affect the orientation of the radiators	A reduction in heat rejection capability or excessive heat loads into the spacecraft	Any power consuming systems as waste heat will need to be reduced	A negative effect on the performance of the thermal system	1	3	Thermal

Likelihood	0-20%	20-40%	40-60%	60-80%	80-100%
	1	2	3	4	5

Severity	Minimal impact on objectives	Minor impact on objectives	Unable to achieve a particular objective	Unable to achieve multiple objectives	Unable to achieve all objectives
	1	2	3	4	5

Phase 3: Prepositioned Orbit

Applicable Level 3 Communication Requirements

Subsystem	ID	Requirement	Driving Phase	Compliance
Comms	COM15	The communication system shall transmit science data with a minimum data rate of 32.4 kbps [TBC].	Phase 6	YES
Comms	COM16	The communication system shall receive commands with a data rate of up to 2 kbps [TBC].	N/A	YES
Comms	COM18	The communication system shall transmit telemetry at a minimum data rate of 20 bps [TBC] while in system fault mode.	N/A	YES
Comms	COM19	The communication system shall receive commands at a data rate of 10 bps during system fault mode.	N/A	Maybe
Comms	COM22	The communication system shall have a high gain transmission EIRP of 80.89 [TBC] dBW	Phase 6	YES

Communication – Telecom Parameters Overview

Antenna	Quantity	Type	Frequency Band	Diameter [m]	Transmit Power [W]	Data Rate (7AU /PP)	Purpose
HGA	1	Parabolic Reflector	Ka / X	3	200 / 160	110 kbps to 3 Mbps	Science/telemetry d/l, command u/l ,ranging
MGA	2	RF Conical Horn	X	~0.15	160	25bps to 1.95 kbps	Safe Mode
LGA	2	Choked Horn	X	~0.1	160	N/A to 196 bps	Safe Mode

Backup

Phase 3: Prepositioned Orbit

ID	Requirement	Driving Phase	Compliance (Y/N/M)
G5	The ground system shall support the flight system for a minimum of 22 years [TBC].	N/A	Maybe
G23	The ground system centers shall interface with the deep space network.	N/A	Yes
G34	The ground system shall establish two-way communication with the flight system on average of **once a week for four-to-eight hours** per session during the preposition mission phase.	Phase 3	Yes
G25	The ground system shall provide the flight system with its trajectory +/- TBD km every 14 weeks.	Phase 3/4/6	Yes
G26	The ground system shall provide the flight system with a best-fit curve of Earth's position to +/- TBD km every 1 year.	Phase 3/4/6	Yes
G39	The ground system shall process science data to provide an exoplanet science package to the customer	Phase 3	Yes
G40	The grounds system shall process science data to provide a remote observation science package to the customer	Phase	Yes

Speaker: Alexi D.

Phase 3: Prepositioned Orbit

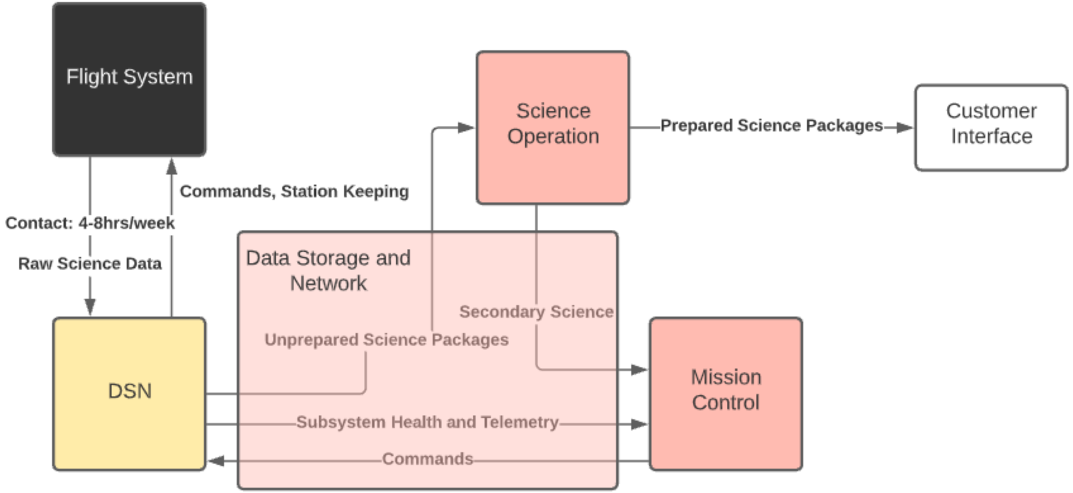
Applicable Level 3 ***Ground System Requirements***

ID	Requirement	Driving Phase	Compliance (Y/N/M)
GS5	The Mission Control Center shall interface indirectly with DSN via the Data Storage and Network to transmit or receive data/commands	N/A	Yes
GS3	The Mission Control Center shall design and execute commands to meet the mission demands	N/A	Yes
GS9	The Data Storage and Network shall store all data of all types locally	N/A	Maybe
GS1	The Science Operations Center shall analyze science data stored in the Data Storage and Network using science models	N/A	Maybe
GS2	The Science Operations Center shall provide science packages to the customer through the customer interface	N/A	Yes

Speaker: Alexi D.

Ground Operations – Preposition

- Secondary Science Data Product Generation
  - Downlink → Storage → Science Product Generation → Customer
- Determination of coverage requirements for ISO intercept confirmation
- Contact Schedule: Once per week per spacecraft for 4-8hrs
  - Ephemeris Information
    - Earth, Flight System
  - Data downlink for science production
- Routine Health Checks, station keeping, course correction



Future Steps:

- Further analysis to determine if the ground system facilities and interfaces have a life span of at least 22 years (G4)
- Further analysis to determine total science data size and storage facility capabilities (GS9)

Phase 4

Speaker: Austin I.

Phase 4: Navigation to ISO - Overview

Major Event	Encounter S/C Operations	Ground System Interactions
ISO Detection and Selection	ES waits in its PP orbit for an ISO to be selected.	After an ISO is detected by a third-party, the orbital data and trajectory is provided to GS. The GS team will verify that this is a desirable ISO.
End of ISO Waiting Period	ES receives ISO data and prepares for third GA.	Three days after ISO selection, GS will communicate to ES to tell it the ISO has been selected and where it is.
Beginning of Third Gravity Assist	ES uses a burn to leave its PP orbit and a second burn to begin its third GA.	ES Leaves PP Orbit and Begins Third Earth Gravity Assist.
End of Third Gravity Assist	ES finishes third GA.	GS updates ES on its location.
End of ISO Travel	ES uses a high magnitude burn to put it on a trajectory towards the ISO.	GS updates ES on ISO location and trajectory.
ES is 55,000 km Away from ISO	ES begins using automatic optical navigation.	GS stops communicating with ES until completion of flyby.

Phase 4: Navigation to ISO

Applicable Level 2 Flight System Requirements

ID: F = Flight System

ID	Requirement	Compliance (Y/N/M)
F8	The flight system shall carry a minimum onboard dV of 4.8 km/s [TBC].	Yes
F15	The flight system shall perform trajectory corrections to achieve a closest approach distance of 400 km +/- **TBD**.	Yes
F16	The flight system shall perform trajectory corrections to achieve a maximum relative velocity to the ISO of 13 km/s at closest approach.	Yes
F28	The flight system shall transmit radiometric position signals to the ground system.	Yes
F31	The flight system shall receive trajectory updates from the ground system every **TBD** days.	Maybe

Phase 4: Navigation to ISO

Applicable Level 2 ***Ground System Requirements***

ID	Requirement	Driving Phase	Compliance (Y/N/M)
G4	The ground system shall command the flight system to depart its prepositioned orbit.	Phase 4	Yes
G35	The ground system shall establish two-way communication sessions with the flight system **at least once every 36 hours for one hour per session** during the navigation mission phase	Phase 4	Yes
G36	The ground system shall establish **a minimum of 42 hours** of two-way communication with the flight system **within 2 weeks prior to the encounter**	Phase 4	Yes
G6	The ground system shall navigate the flight system to a closest approach distance of 400 km +/- **TBD** km.	Phase 4	Maybe
G26	The ground system shall provide the flight system with a best-fit curve of Earth's position to +/- TBD km every 1 year.	Phase 3/4/6	Yes
G25	The ground system shall provide the flight system with its trajectory +/- **TBD** km every 14 weeks.	Phase 3/4/6	Yes
G27	The ground system shall provide the flight system with the ISO ephemeris 24 hours prior to the beginning of autonomous operations.	Phase 3/4/6	Yes
G30	The ground system shall provide an ISO ephemeris with positional accuracy of +/- 20 km in all axes.	Phase 3/4/6	Maybe

CAL POLY

Speaker: Alexi D.

Phase 4: Navigation to ISO

Applicable Level 3 Ground System Requirements

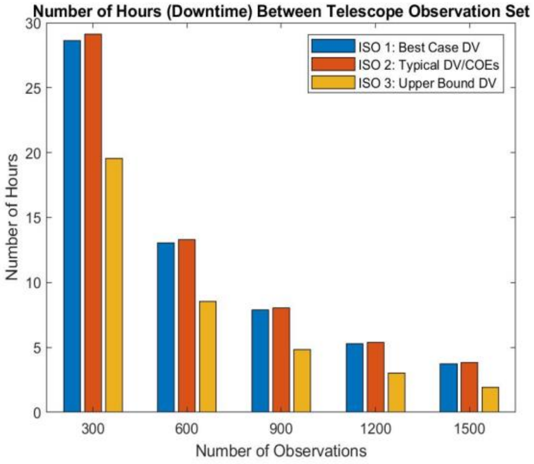
ID	Requirement	Driving Phase	Compliance (Y/N/M)
GS6	The Mission Control Center shall decide whether any customer provided ISO ephemeris meets the mission capabilities	N/A	Yes
GS3	The Mission Control Center shall design and execute commands to meet the mission demands	N/A	Yes
GS9	The Data Storage and Network shall store all data of all types locally	N/A	Maybe
GS1	The Science Operations Center shall analyze science data stored in the Data Storage and Network using science models	N/A	Yes
GS2	The Science Operations Center shall provide science packages to the customer through the customer interface	N/A	Yes

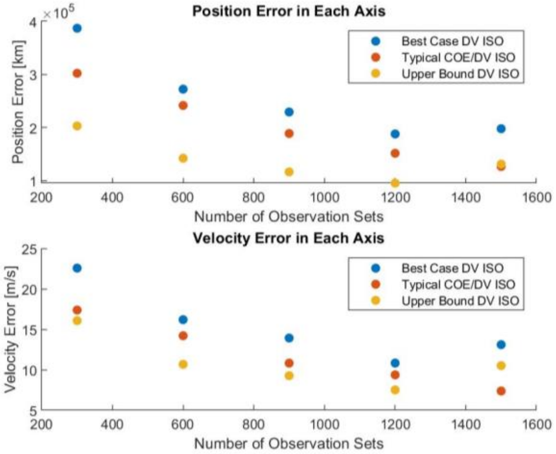
CAL POLY

Speaker: Alexi D.

Ground Based Observations – Ephemeris Uncertainties

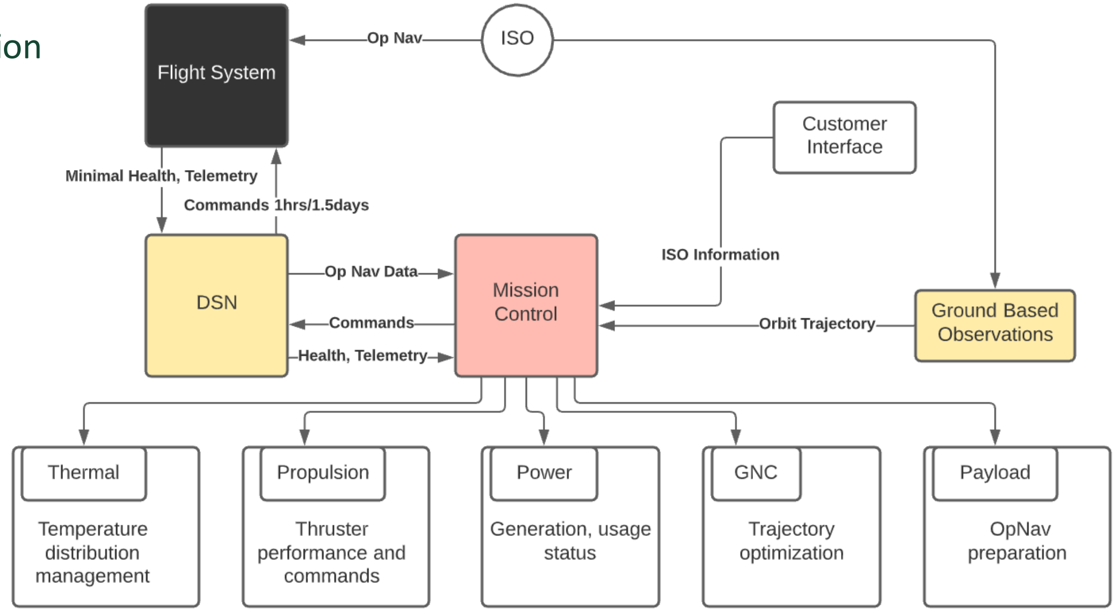
- Ground based telescopes will be used to take observations sets:
- 1200 observation sets are recommended
  - 2.5 hour observation sets (30 observations spaced by 5 min)
  - Require observations 5 hours apart from detection until the ISO leaves 3AU
  - Increasing the number of observations beyond 1200 decreases allowable downtime while not decreasing ephemeris uncertainty significantly





Ground Operations – Preposition Departure to Encounter

- New Contact Schedule starts upon ISO detection
  - **One hour every 36hours** to command departure and collect/update telemetry
  - **Once daily for 2-4hrs** weeks before AutoOpNav
- New Commands Designed on ground for
  - Ephemeris of Earth, Trajectory update
  - ISO imaging procedures for pointing
- Continued Data Ground-Downlink
  - Monitor flight system status
- Prepare for AutoOpNav and GNC
  - Design burn schedule
  - Assess possible gravity assists



Phase 4: Navigation to ISO

Applicable Level 3 GNC Requirements

ID	Requirement	Driving Phase	Compliance (Y/N/M)
GNC1	The GNC system shall propagate the heliocentric velocity of the flight system to +/- 10 m/s in all axes for 14 days	Phase 5	M
GNC2	The GNC system shall propagate the heliocentric position of the flight system to +/- 1000 km in all axes for 14 days	Phase 2,3,4,5,6	Y
GNC3	The GNC system shall receive the heliocentric position of the flight system to +/- **TBD** km in all axes every 14 days.	Phase 2,3,4,5,6	M
GNC4	The GNC system shall propagate the heliocentric position of Earth to +/- 1514 km in all axes for 1 year.	Phase 2,3,4,5,6	Y
GNC5	The GNC system shall receive the heliocentric position of Earth to +/- **TBD** km in all axes every 1 year.	Phase 2,3,4,5,6	M

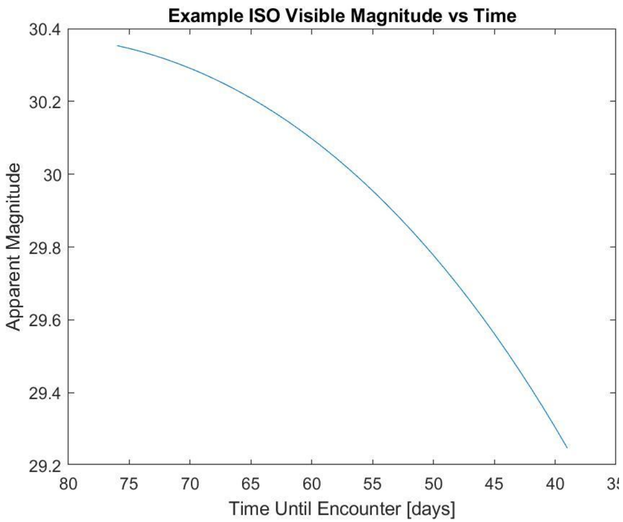
CAL POLY

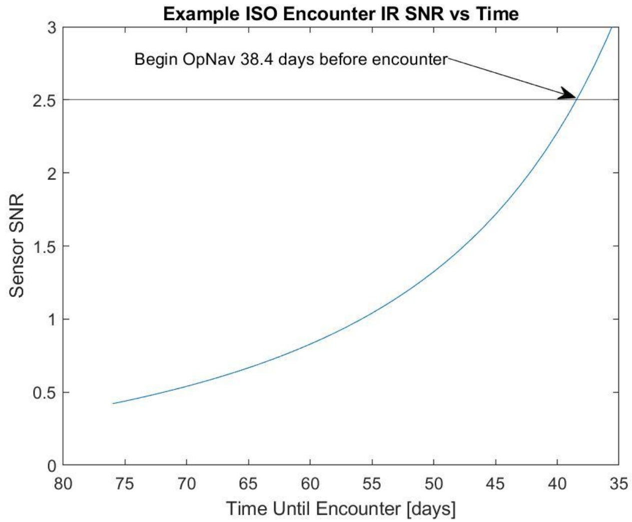
Speaker: Helen M.W.

Optical Navigation- IR vs. Visual

- Due to small ISO size, large heliocentric range, and large phase angles (angle between the ISO 2 Sun Vector and the ISO 2 SC vector), the IR sensor on the primary science imager will be used to detect the ISO at the beginning of the optical navigation campaign.

- Using IR allows us to start optical navigation earlier than if we used the visual sensor.





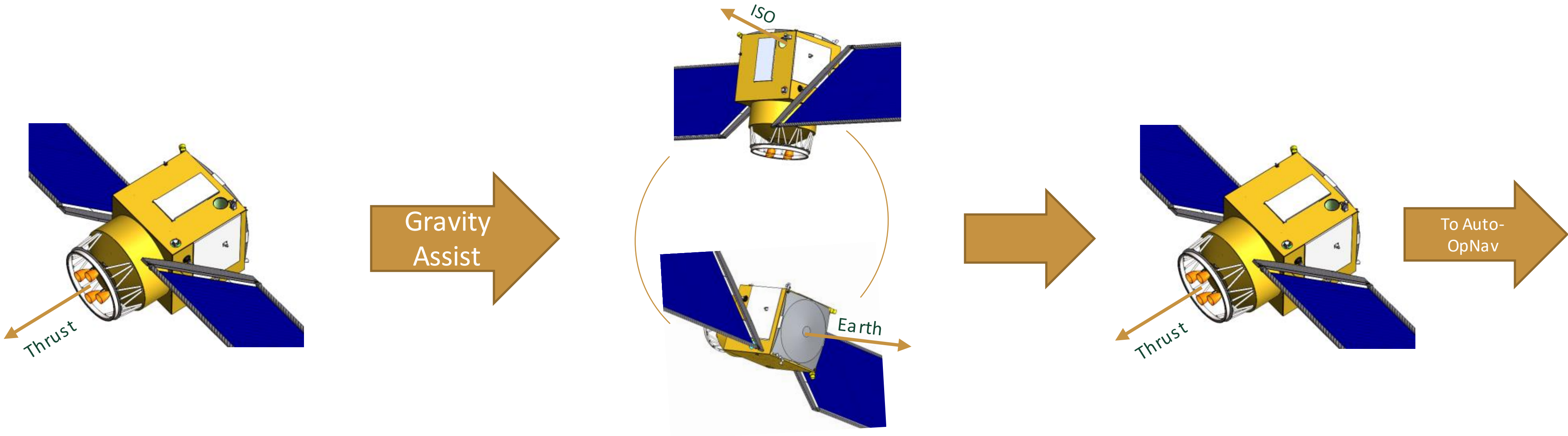
Future Work: Create model of fuel slosh during navigation to ISO, potentially add Diaphragm-style positive expulsion Pressure Management Device to fuel tanks to minimize fuel slosh

Phase 4 Navigation Timeline

- TCM 8 and 9 compose our third Earth Gravity Assist and set us on a trajectory towards the ISO
- TCM 10-13 are additional maneuvers taken based on optical navigation data from the spacecraft to correct for uncertainty in the ISO ephemeris knowledge generated by ground systems

Event	Date (Days before Encounter)	Data Arc (#Days)
TCM 8	N/A	N/A
TCM 9	N/A	N/A
Op Nav Begins	T – 38	N/A
TCM 10	T – 18	20
TCM 11	T – 6	12
TCM 12	T – 2	4
TCM 13	T – 1	1

Navigation Pointing Directions



Start: Thrusters @ Burn Direction 2

Cycle: OpNav @ ISO / HGA @ Earth

Break: Thrusters @ Burn Direction 3

Burn Order

Burn 1

- 3-90 days prior to selected Earth gravity assist
- Departs from prepositioned orbit

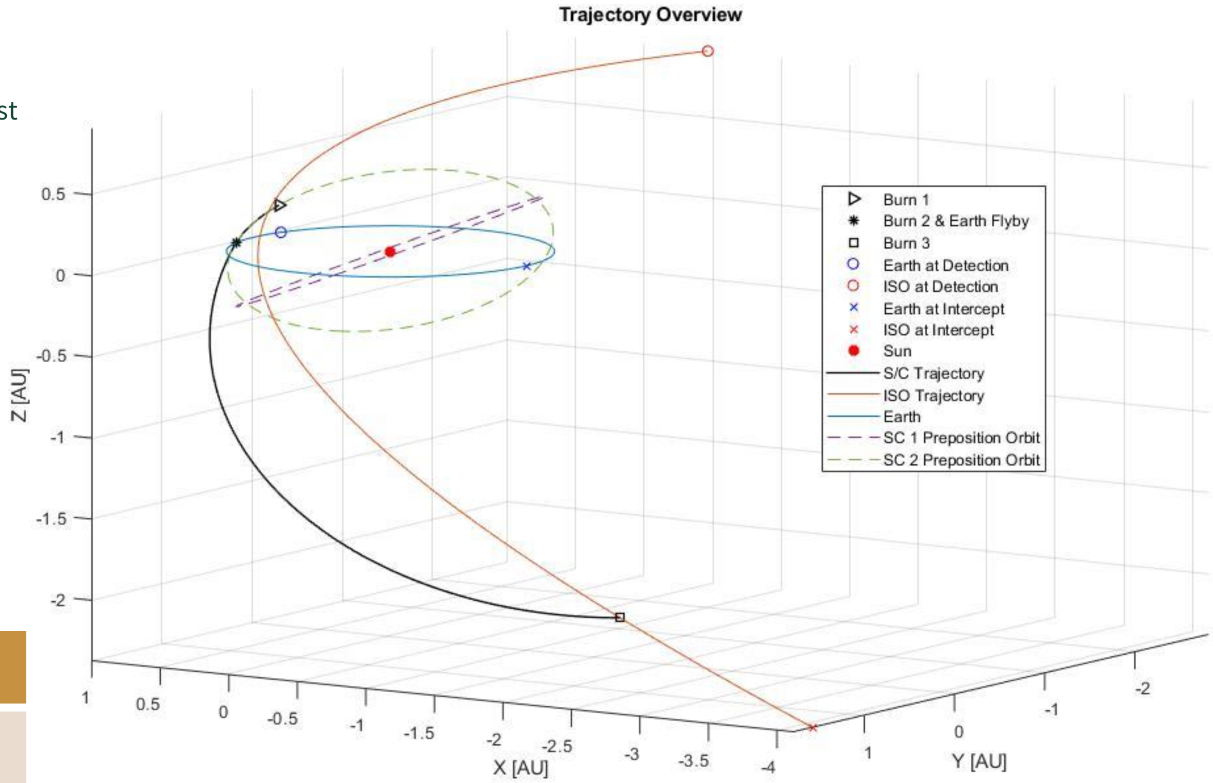
Burn 2

- Occurs at Earth gravity assist
- Corrects velocity vector into assist

Burn 3

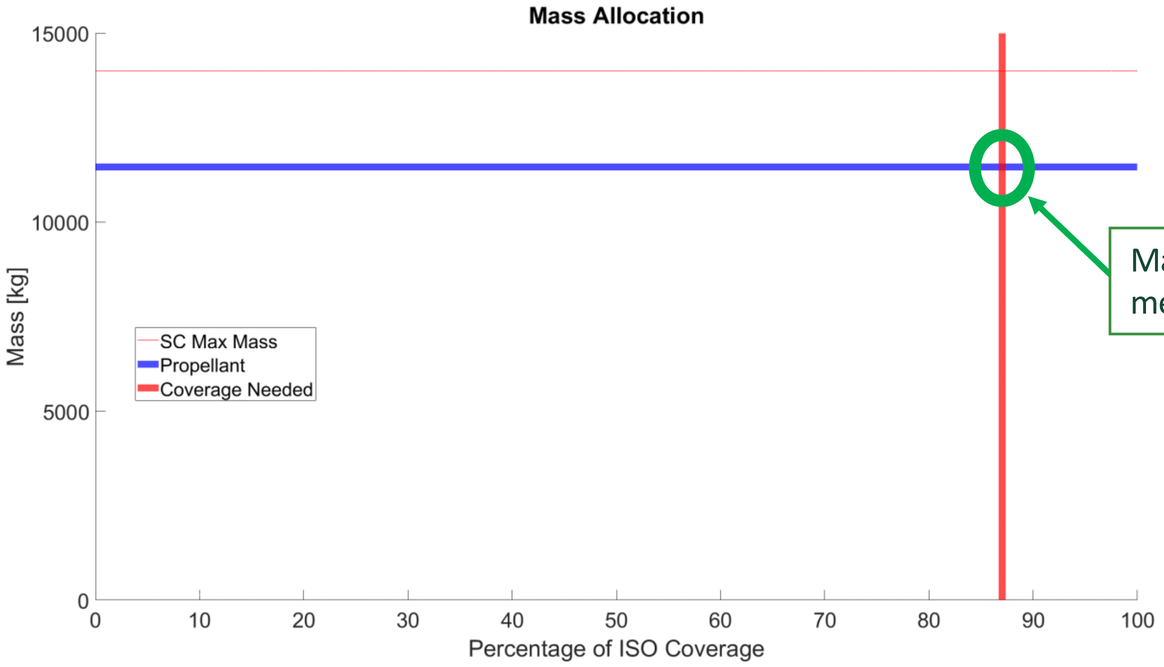
- Occurs at ISO intercept
- Reduces SC-ISO relative velocity to less than 13 km/s
- Uses remainder of propellant to make ISO intercept as slow as possible

[km/s]	Burn 1	Burn 2	Burn 3
Mean	0.10	0.10	3.86
STD	0.12	0.13	0.25



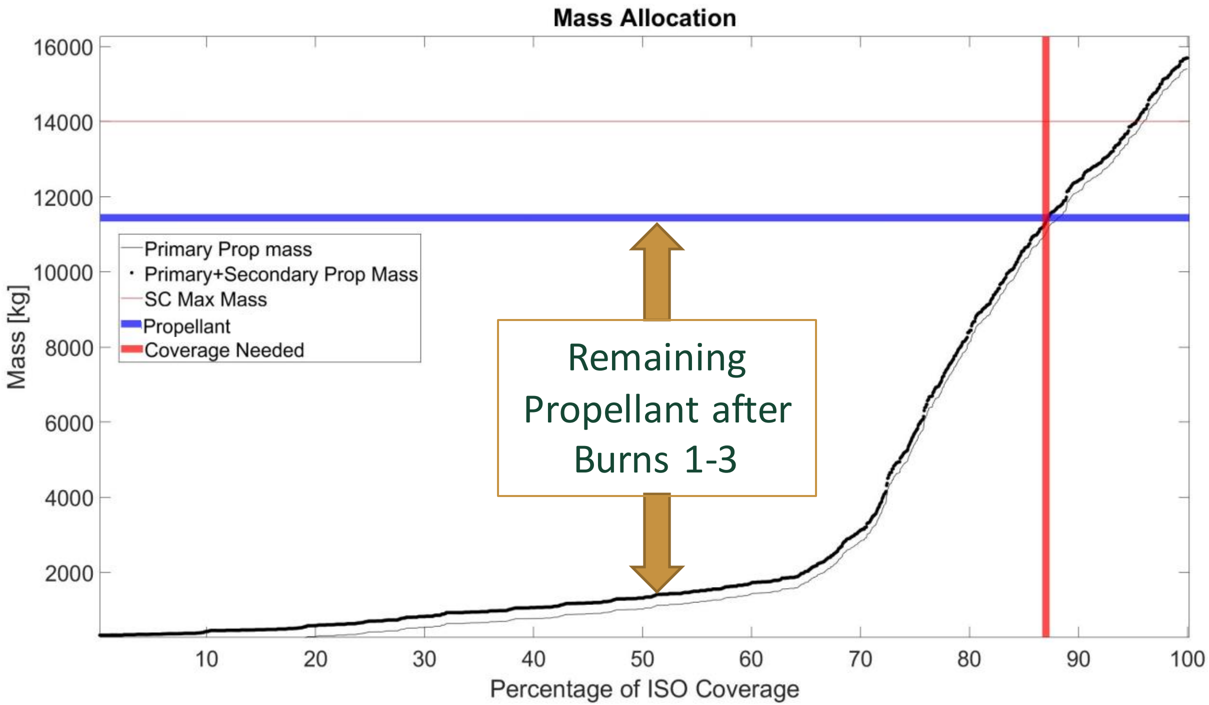
Backup

Mass Sizing



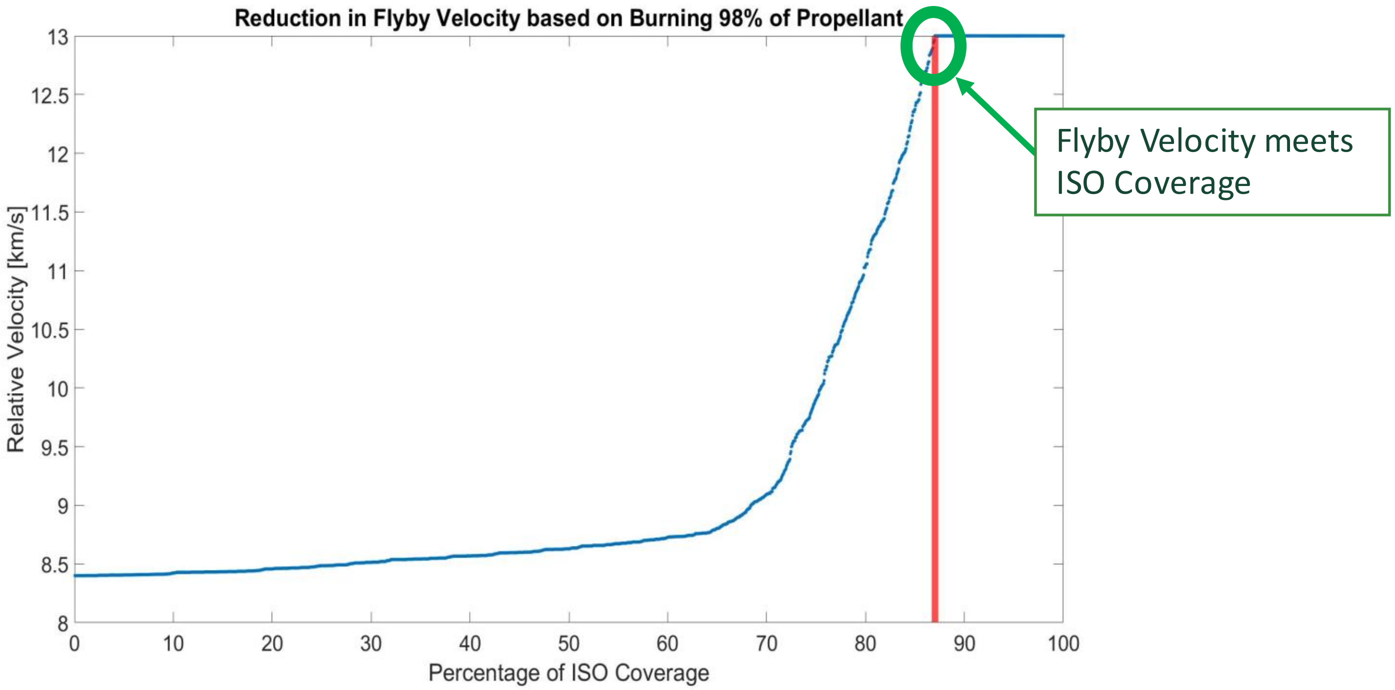
Mass of propellant meets ISO Coverage

Mass Sizing



Mass Allocation
16000
14000
12000
10000
8000
6000
4000
2000
Mass [kg]
10
20
30
40
50
60
70
80
90
100
Percentage of ISO Coverage
Primary Prop mass
Primary+Secondary Prop Mass
SC Max Mass
Propellant
Coverage Needed
Remaining Propellant after Burns 1-3

Flyby Velocity Reduction



ISO Flyby Risks -

Given that…	There is a possibility of…	Adversely impacting…	Which can result in…	Likelihood (out of 5)	Severity (out of 5)	Sub team
There are uncertainties about the ISO state	The ISO not being at the distance or relative velocity that was predicted	The propulsion system's dV capability and payload subsystem	The spacecraft navigating to a location from which we can no longer complete the mission	2	3	Orbits

Likelihood	1	2	3	4	5
	0-20%	20-40%	40-60%	60-80%	80-100%

Severity	1	2	3	4	5
	Minimal impact on objectives	Minor impact on objectives	Unable to achieve a particular objective	Unable to achieve multiple objectives	Unable to achieve all objectives

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Speaker: Jack K.

Phase 4: Navigation to ISO

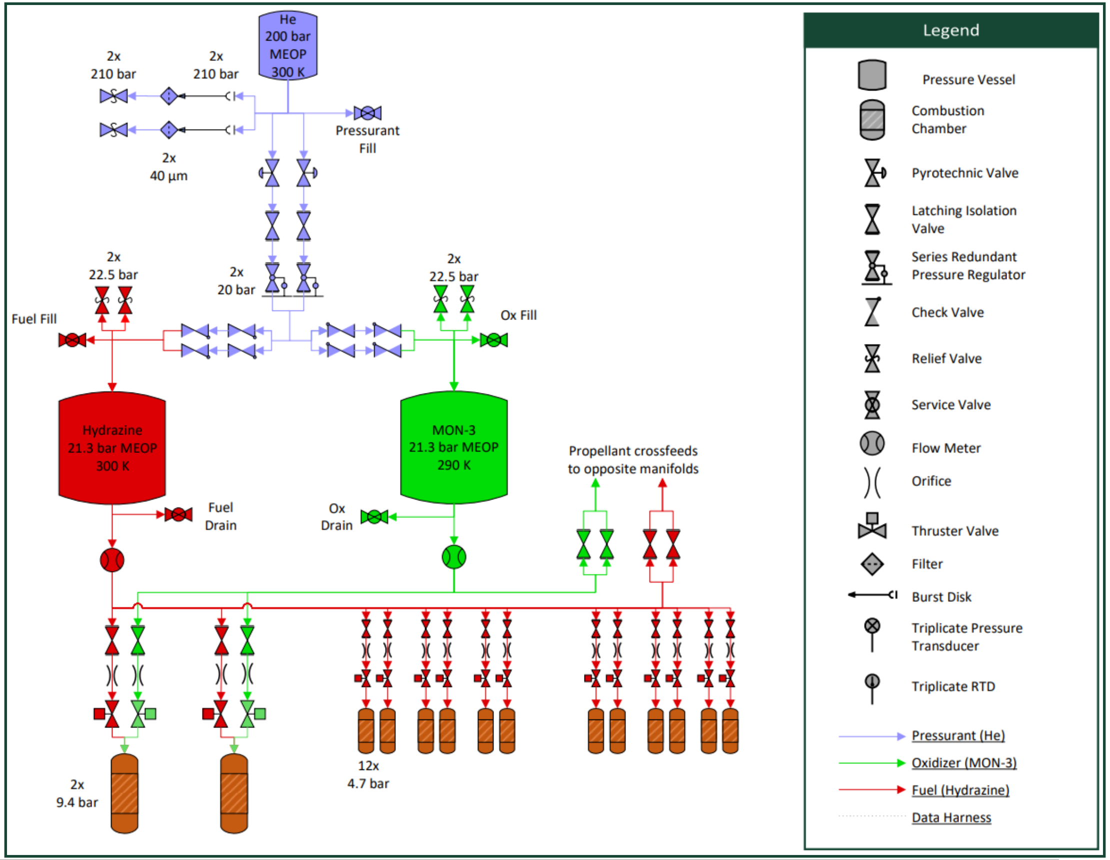
Applicable Level 3 ***Propulsion Requirements***

ID	Requirement	Driving Phase	Compliance
PROP1	The propulsion system shall provide the flight system with a dV of 4.5 +/- 0.5 km/s for transport to the ISO .	4	Yes
PROP4	The propulsion system shall have a minimum reliability of 98.5%	All	Maybe
PROP5	The propulsion system shall have a mass of 11481 kg +/- 300 kg	All	Yes
PROP6	The propulsion system shall have a volume of 15 m^3 +/- 3 m^3	All	Yes
PROP7	The propulsion system shall operate in the space environment for 22 years.	All	Yes
PROP8	The propulsion system shall operate at a heliocentric range of 0.5-7 AU	All	Yes

Speaker: Chris L.

Propulsion System

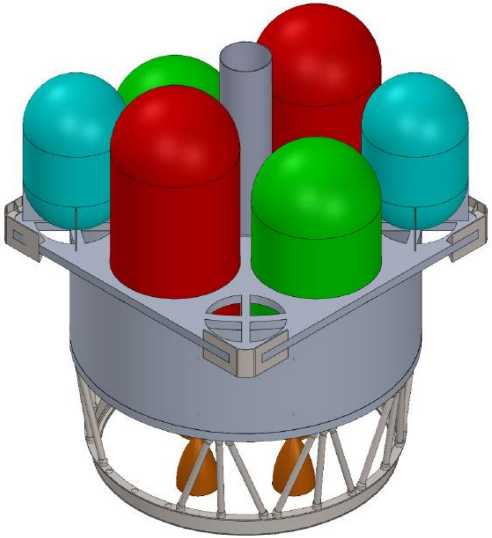
- Propulsion system is composed of:
  - **Primary propulsion**, which will be used for large-dV maneuvers such as navigating to the ISO
  - **Secondary propulsion**, which is used for prepositioning burns, stationkeeping, and ADCS CMG singularity-avoidance
- Propulsion system designed to be one-fault tolerant to ensure we're always able to get to the ISO
  - Most valves have single-redundant backups
  - All thrusters a single-redundant backup
  - Pressure/temperature measurements are taken in triplicate, using the median value



Backup

Propulsion System Components

- Tank sizing was calculated using propellant mass outputs from the Mass Model.
  - The tanks are designed as homogenous 6Al-4V titanium
  - Tanks will use a surface-tension style propellant management device (PMD) to ensure proper propellant flowrate in zero-G conditions
  - Extra ullage space in the fuel tank allows blowdown functionality for small burns prior to pyro valve actuation.
  - A propellant utilization (PU) algorithm will be used to ensure maximum dV for a given propellant split.



Thruster	Quantity	Propellants	Thrust [N]	Isp [s]	Mixture Ratio
R-4D-15	4	Hydrazine/MON-3	445	329	0.70-1.33 1.0 Nominal
MR-111G	24	Monopropellant Hydrazine	4	229-219	N/A

Backup

Phase 4: Navigation to ISO

Applicable Level 3 ***Structure Requirements***

ID	Requirement	Driving Phase	Compliance (Y/N/M)
STR6	The structure shall sense the thruster's angular position within **0.1 +/- 0.01 degrees**.	Phase 4	Yes
STR9	The structure shall articulate the thrusters **+/- 8 degrees** from the zero position.	Phase 4	Yes

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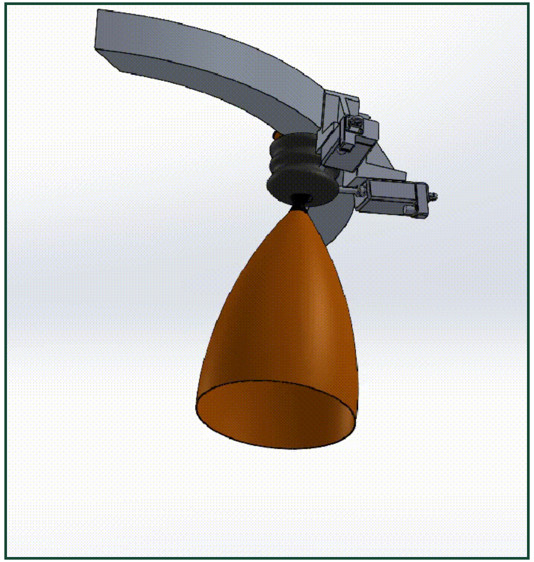
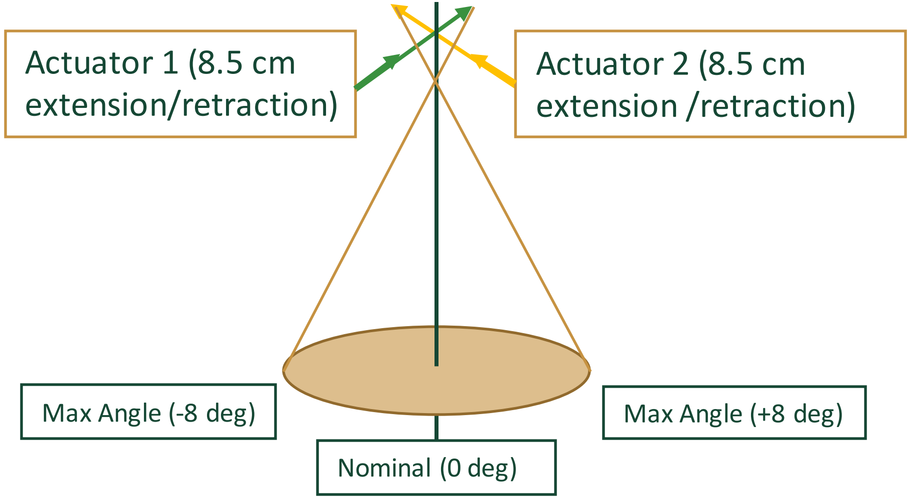
Speaker: Ricardo C.

Mechanisms - Structures

Mechanism	Source	Total Mass	Continuous Draw	Actuation Draw	Actuation Type	Range	Resolution	Operating Temperatures	Details
Thruster Ring Actuator (x8)	MOOG 310	16 kg	<28 V DC	28 V DC	Motion translation (rotational to linear)	17 cm stroke	0.025 mm	-50°C to 80°C	[2]



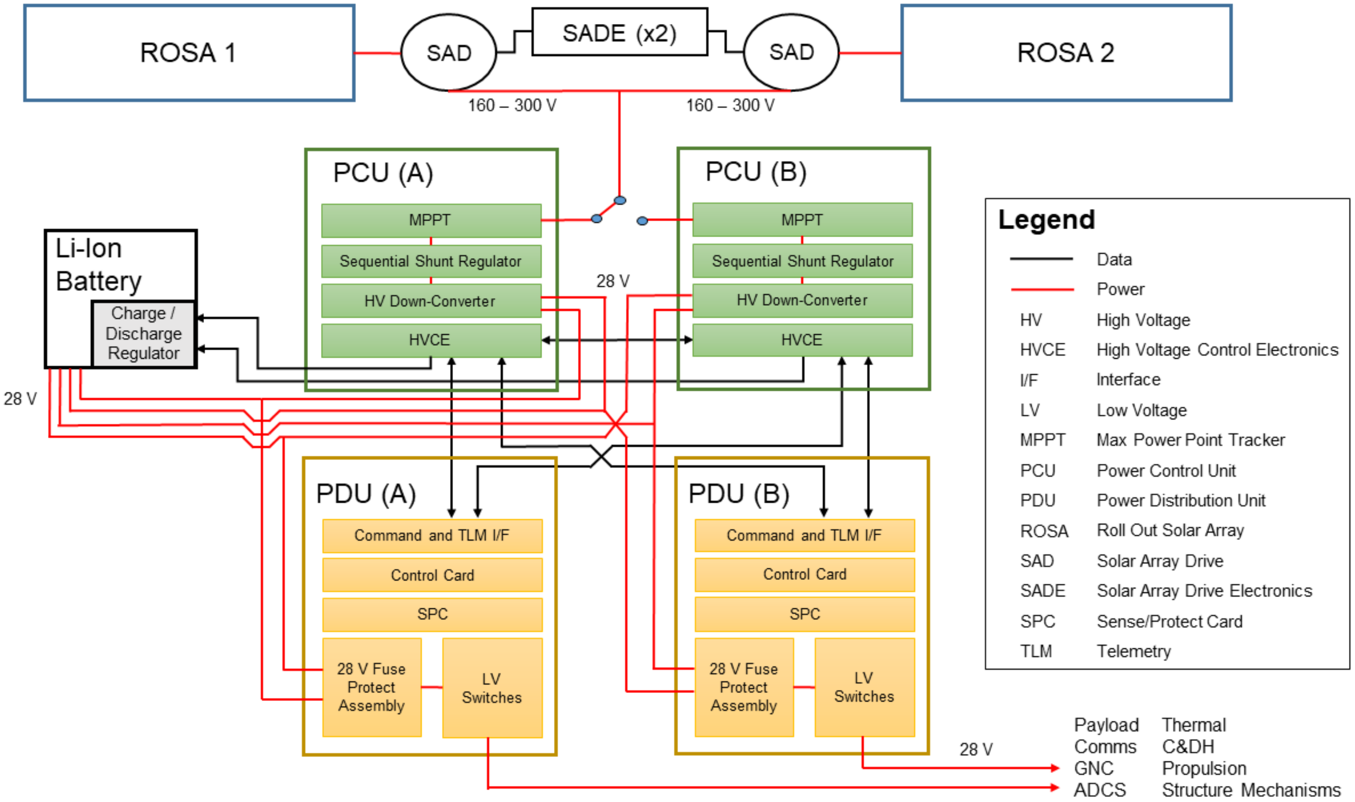


Phase 4: Navigation to ISO

Applicable Level 3 Power Requirements

ID	Requirement	Driving Phase	Compliance
POW2	The power system's solar panels shall generate a minimum of **690 + TBD** Watts at end of life.	Phase 6	Yes
POW3	The power system shall allocate power for each subsystem as specified by **Table 2.0**.	N/A	Yes
POW4	The power system shall provide **28 +/- TBD** Volts to all components.	N/A	Yes
POW5	The power system shall provide **305 +/- TBD** Watts during system fault mode.	N/A	Yes

Power Block Diagram



Thank you!
End of Day 1

Preliminary Design Review – Day 2

C.O.S.M.I.C – Friday, June 4th , 2021

CAL POLY

Introduction

Speaker:

C.O.S.M.I.C. (Celestial Object Sensing and Measuring Identification Campaign)

Our Team is a dedicated engineering group of 72 Cal Poly Spacecraft Design students working cooperatively in a virtual environment.

Our Mission is to provide space systems for interstellar exploration to further our understanding of the origins of the solar system through the study of interstellar objects and near-parabolic comets.

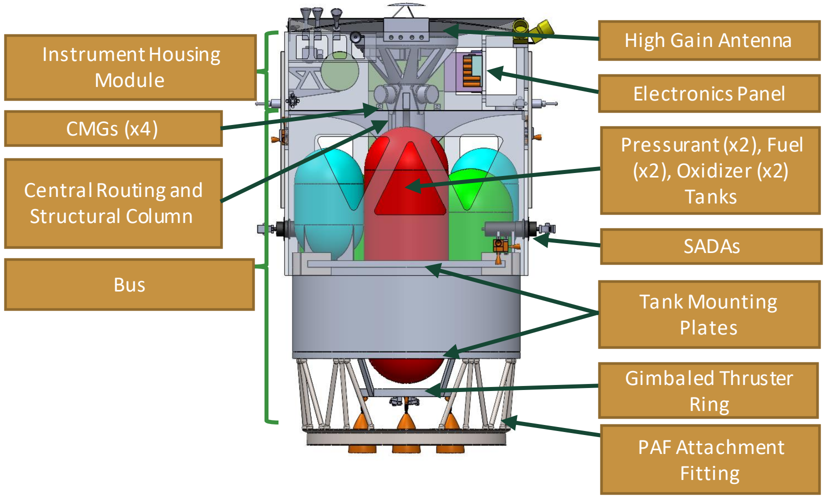
Table of Contents

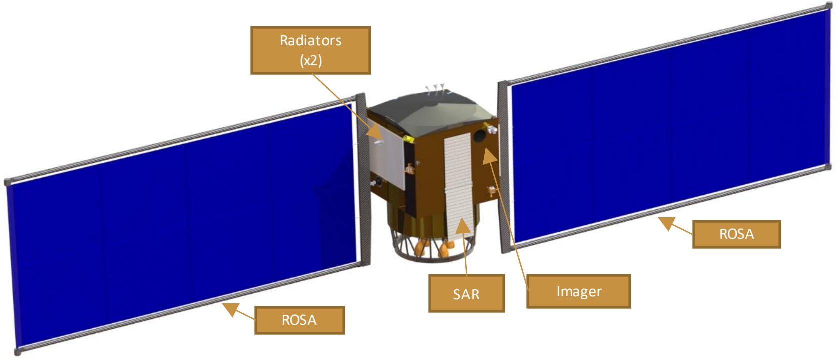
Day 1 Summary

- Introduced the Solicitation
  - Composition
  - Morphology
  - Angular Momentum
  - Advanced Object Definition
  - Remote Observation
  - Exoplanet

- Decomposition of System Design

- Phase 1: Launch to Detumble
- Phase 2: Orbital Insertion into Preposition
- Phase 3: Prepositioned
- Phase 4: Navigation





Phase 5

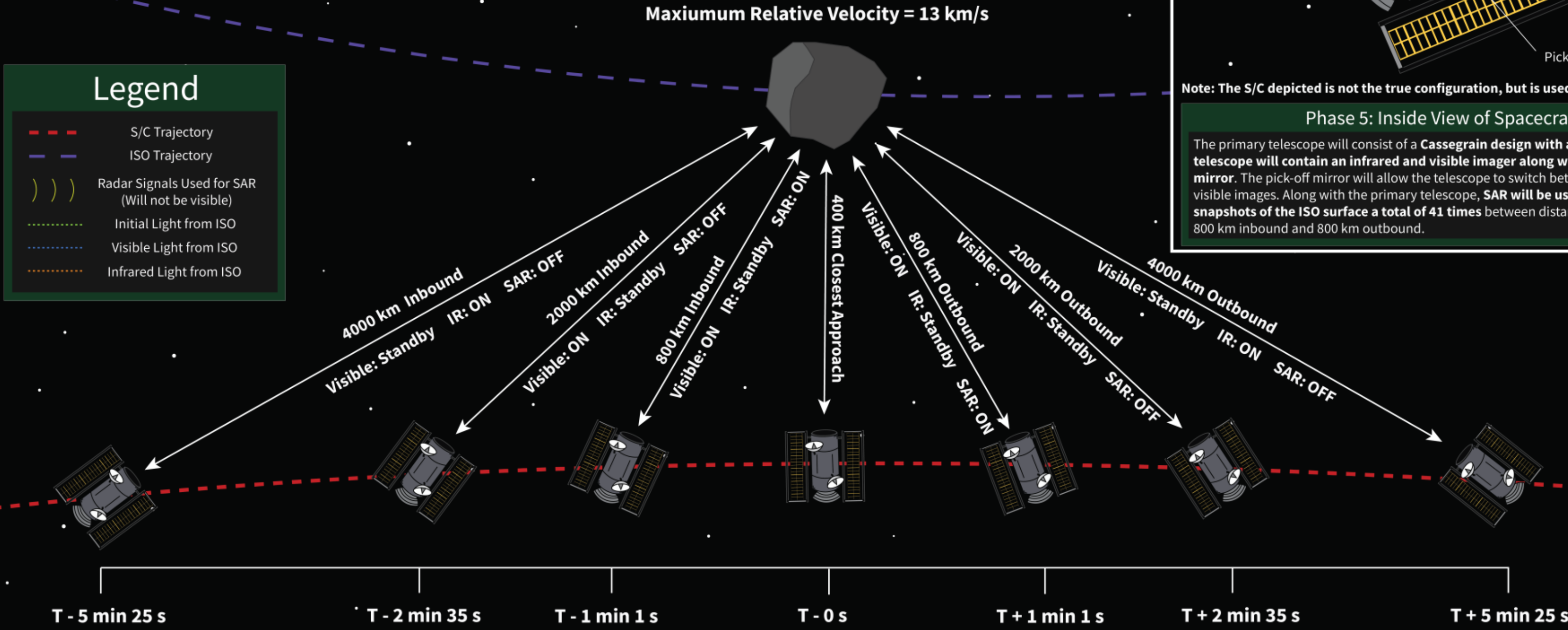
Speaker: Sean T.

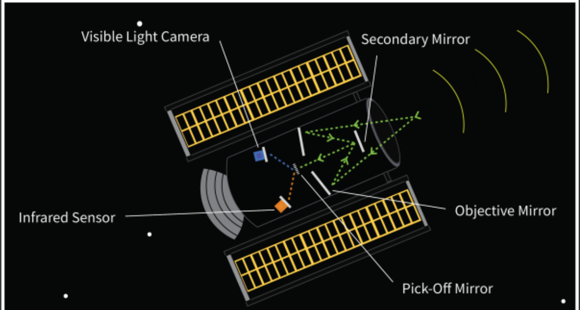
Encounter Strategy: Phase 5

Mission Concept of Operations

Phase 5

The Encounter S/C will begin Autonomous Optical Navigation at 55,000 km away from the ISO on the inbound trajectory. At **4000 km away from the ISO, the IR imager will begin taking images** of the ISO surface. Once the **S/C is 2000 km away from the ISO, visible imaging will begin. At 800 km away, the SAR system will begin** taking snapshots. The **closest approach will be 400 km** away. The **Encounter S/C will stop collecting primary science data at 4000 km** away on the outbound trajectory.





Note: The S/C depicted is not the true configuration, but is used for visual purposes.

Phase 5: Inside View of Spacecraft

The primary telescope will consist of a **Cassegrain design with an FOV of 0.55**. The **telescope will contain an infrared and visible imager along with a pick-off mirror**. The pick-off mirror will allow the telescope to switch between infrared and visible images. Along with the primary telescope, **SAR will be used to take snapshots of the ISO surface a total of 41 times** between distance ranges of 800 km inbound and 800 km outbound.

Phase 5: Encounter

Applicable Level 2 Flight System Requirements

ID	Requirement	Compliance (Y/N/M)
F9	The flight system shall operate autonomously within 24 hours of closest approach.	Y
F11	The flight system shall acquire visible imagery of the object at a resolution of 5 mpp.	Y
F12	The flight system shall acquire infrared imagery of the object at a resolution of 10 mpp.	Y
F13	The flight system shall acquire radar data of the object with a maximum doppler shift precision of 0.0515 Hz at closest approach to the ISO.	Y
F14	The flight system shall image the object's orthogonal axes.	Y
F15	The flight system shall perform trajectory corrections to achieve a closest approach distance of 400 km +/- TBD.	Y
F16	The flight system shall perform trajectory corrections to achieve a maximum relative velocity to the ISO of 13 km/s at closest approach.	Y
F32	The flight system shall support a total data volume of up to 21 GB.	Y
F34	The flight system shall acquire radar data of the object with a maximum slant range uncertainty of 1.5 meters at closest approach.	Y
F35	The flight system shall acquire radar data every 3 degrees of angular position between the ISO and the spacecraft.	Y

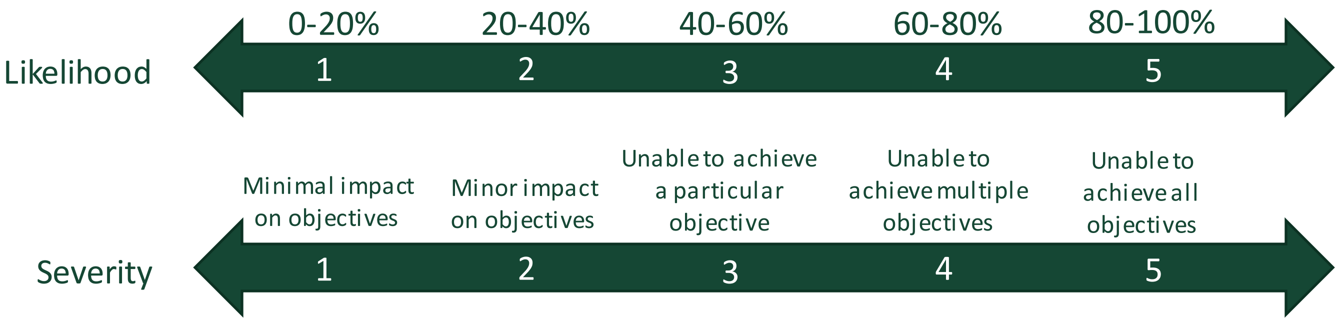
Encounter Models

Effect of ISO Rotation Rate on Primary Objectives

	Spin Axis	Rotation Rate	Shape Model Percent Coverage		
ISO Rotational Period	Spin Axis Error	Confidence Within +/-1.0%	+/- 10 m resolution	+/- 20 m resolution	+/- 30 m resolution
8 min	≤0.18°	H	50.3%	67.7%	73.5%
10 min	≤0.23°	H	48.2%	66.5%	73.0%
42 min	≤1°	H	22.7%	42.2%	49.5%
1 hr	1.4°	H	14.9%	36.7%	45.8%
2 hr	2.8°	H	0%	20.7%	34.1%
4 hr	6.2°	H	0%	0%	12.5%
6 hr	12.2°	H	0%	0%	0%
5 days	>50°	H	0%	0%	0%
10 days	>50°	M	0%	0%	0%
20 days	>50°	L	0%	0%	0%

ISO Flyby Risk

Given that …	There is a possibility of…	Adversely impacting…	Which can result in…	Likelihood (out of 5)	Severity (out of 5)	Sub team
There are unknowns about the ISO	The angular velocity of the ISO being too slow	The post-processing of the mission data necessary to generate the shape mode and determine the spin axis	The radar data not being able to produce a shape model with the required resolution and the images taken not being able to produce spin axis to the required error	4 *unknown because not enough data on ISOs	3-4 Not meeting some primary objectives to the required error but still getting data	Sci Tech



Rotation Rate Method Overview and Assumptions

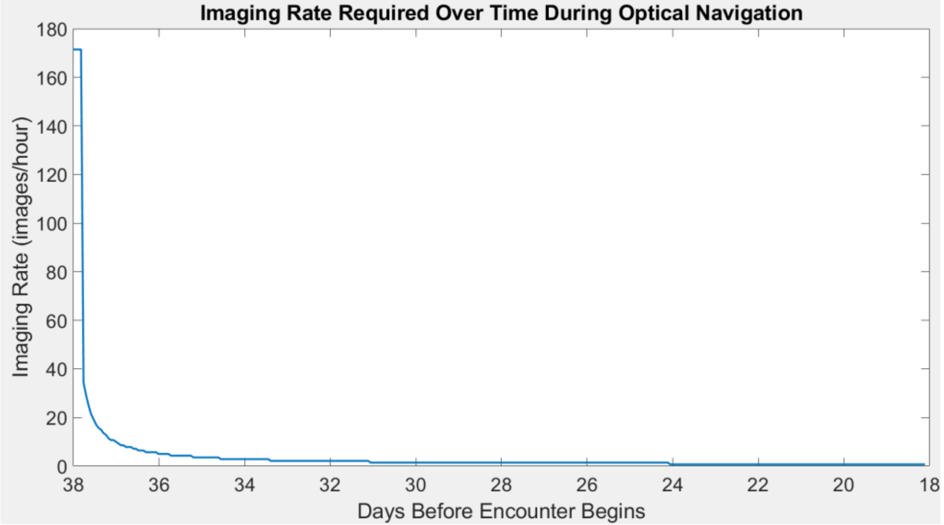
- Method Overview
  - IR light curves will be used to determine the rotation rate of the ISO
  - Images will be taken for ~20 days during OpNav
  - Data will be downlinked before auto navigation begins
- Assumptions (taken from Rosetta flyby of asteroid Steins)
  - The rotation rate can be determined to within 1% with:
    - An imaging rate of 60 images per period
    - An imaging duration between 3 and 4 rotational periods
  - A comet coma will have negligible effects on the light curves

Rotation Rate Method Results

Level 1 Reqs

MP8	The mission shall determine the object's rotation rate within 1%.	✓ Period ≤ 5 days

ISO Rotation Period	Imaging Duration (periods)	Imaging Rate (images/period)
< 0.35 hours	> 4	< 60
0.35 hours to 5 days	3 - 4	≥ 60
5 days to 10 days	2 - 3	≥ 60
10 days to 20 days	1 - 2	≥ 60

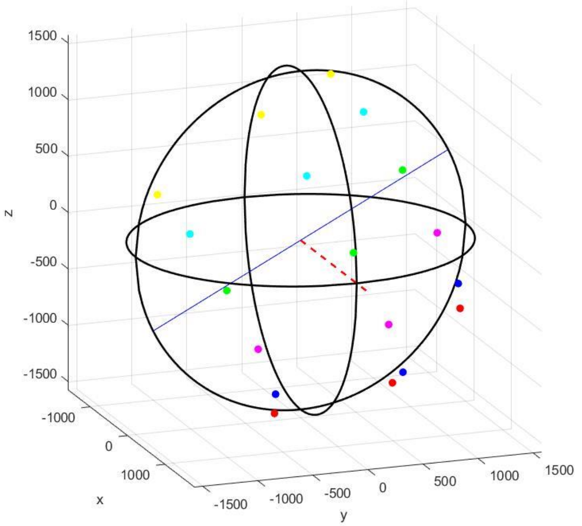


- Requires access to the DSN for 46.2 hours during the 2.2 weeks prior to the encounter.
- Using light curves, the requirement is met for rotation periods between 0.35 hours and 5 days.
- Other radar and image data can be used to determine the rotation rate within 1% for periods less than 0.35 hours.

Spin Axis Model

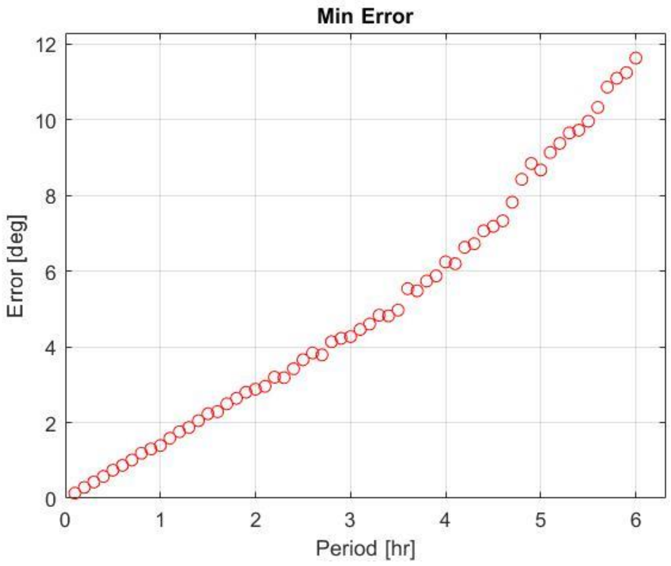
MP7	The mission shall determine the object's spin axis within +/- 1.0 degree.	✓ Period ≤0.7 hr

- Model Assumptions
  - There are identifiable features that can be tracked over multiple images
  - Spin axis orientation problems ignored because of multiple viewing angles during encounter
  - ISO can be seen with IR and visible imaging (may have problems with comet comas)
- Model Overview
  - Spin axis can be determined by tracking points on a set of images
  - Spin axis knowledge error is calculated by varying time intervals of data



Spin Axis Model

- Strong correlation between amount of rotation seen and spin axis knowledge error
- Longer imaging intervals required for longer period ISOs



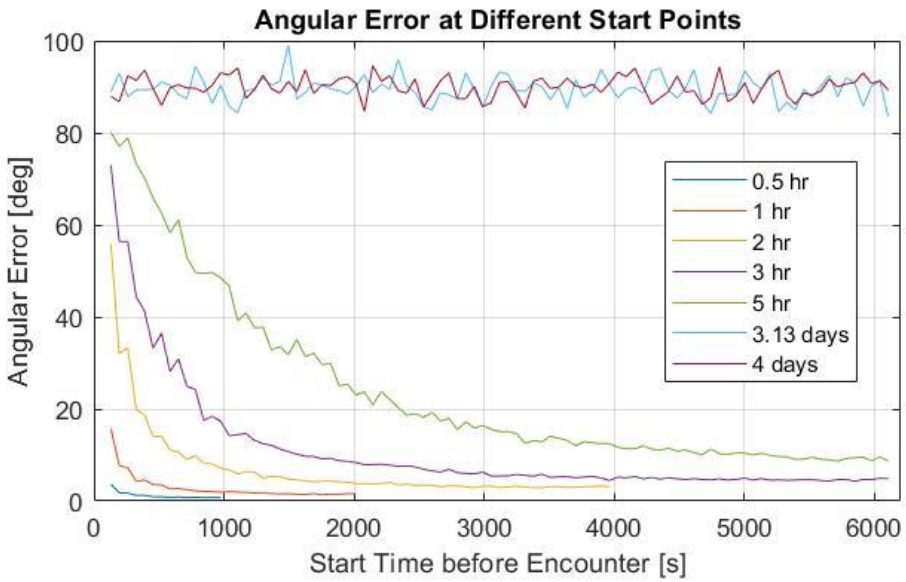


Image Quality Model (Smear + Jitter)

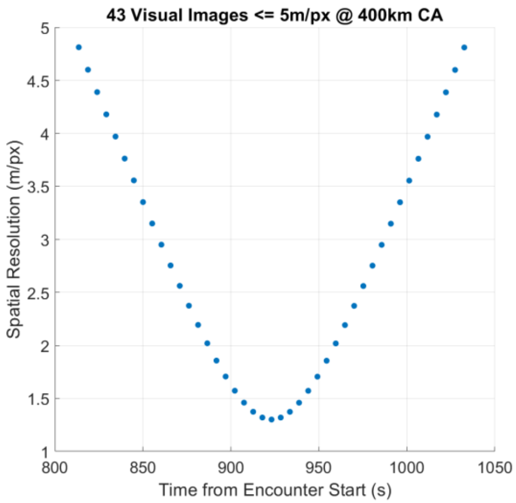
Jitter:

- Can obtain required resolution for IR/Visual with Jitter = 1^{-4} rad/s
- Exceeds current estimated Jitter by 190%

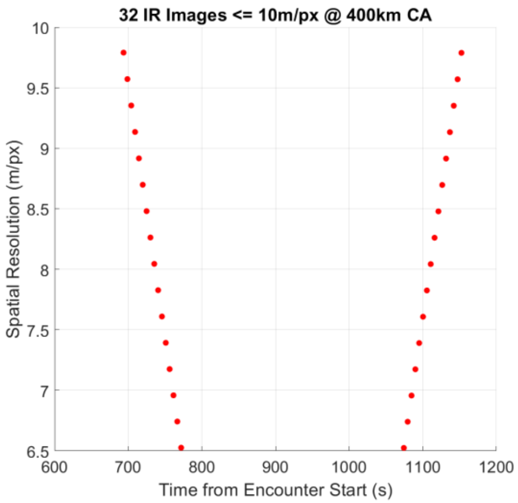
Smear:

- Cross-Track Smear is fully corrected by slew profile.
- Along-Track (Radial) Smear is given by: iFOV $\times \Delta t_{exp} \times v_{relative}$

Req: F11 ✓



Req: F12 ✓

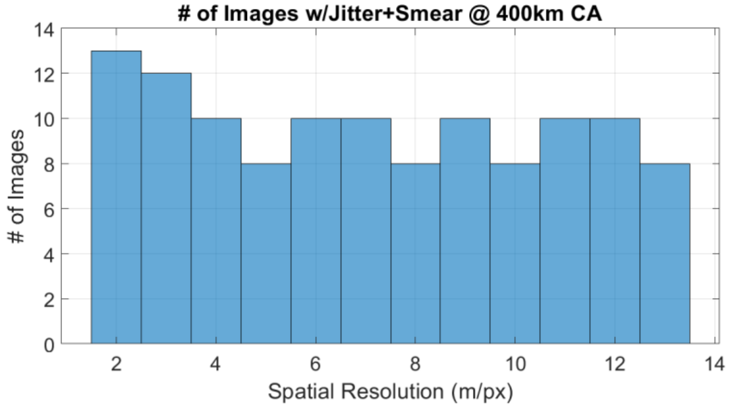


Level 1 Reqs

Visual and IR Surface Coverage

Assumptions:

- IR Range = [4000, 2000] km
- Visual Range = [2000, 400] km
- Min. Exposure Time = 15ms
- Max. Framerate = 5 FPS
- Best Image stored per 5s interval



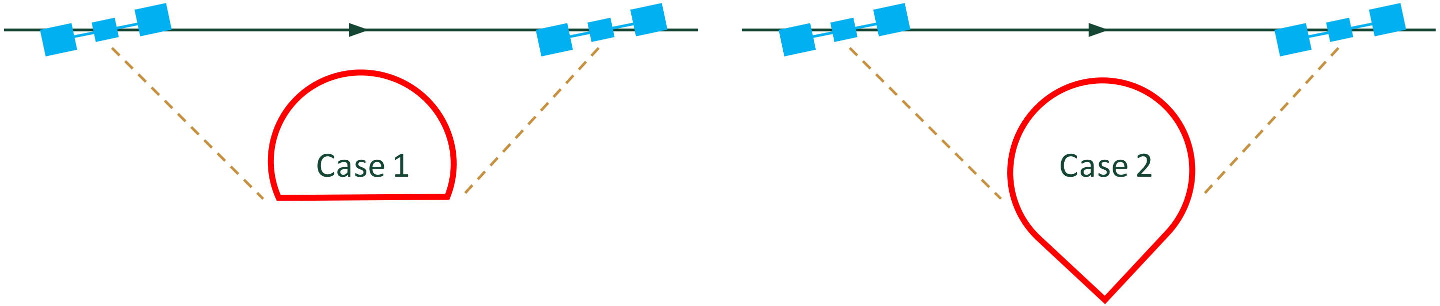
Req: MP4 ✓ Req: MP3 ✓

Stats from ISO Cases A, B, C	Visual (Lit) Coverage %	IR (Unlit) Coverage %
Mean	48.75	97.83
1σ	1.00	0.73

Mean Dimension Model

MP6	The mission shall determine the object's mean dimension within +/- 10 meters.	✓ Period >12 s

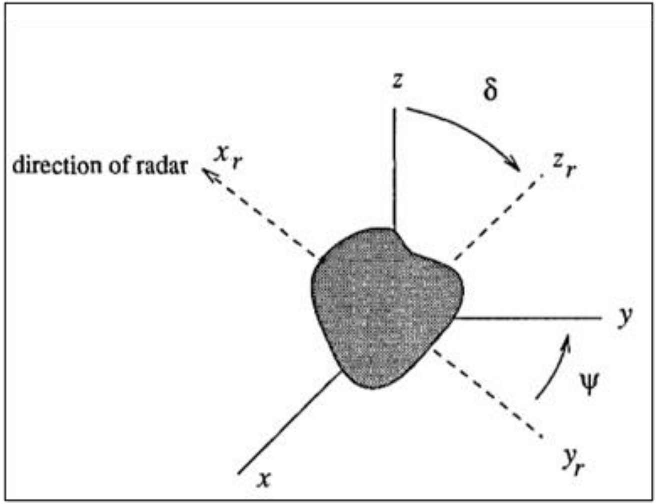
- Model Assumptions
  - Spherical ISO
  - Non viewing side of ISO is not concave
- Conclusion
  - Opposite cases within +/- 10 m requirement for 160° viewing angle
  - Mean dimension requirement will be met from data already collected for other primary objectives



Speaker: Lauren F.

SAR Shape Model Assumptions

- Spherical ISO shape with radius of 750 meters
- The uncertainties in the following values have a negligible effect on the final shape model:
  - Radar orientation in inertial space during encounter
  - Time stamps on all radar and orientation data
  - Backscatter power measurement (24-bit samples)
  - Angular velocity of the ISO
- Backscatter from the entire surface visible to the radar is collected by the receiver
- Assume the sub-radar latitude (delta) is 30 degrees



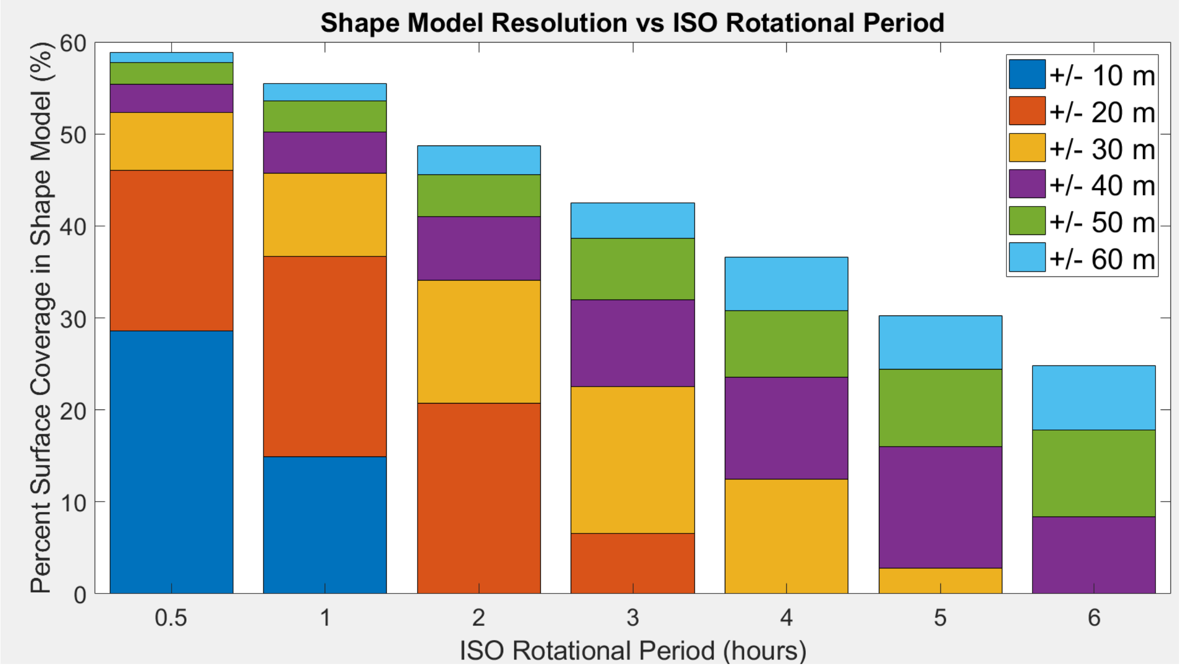
Scott Hudson (1994) Three-dimensional reconstruction of asteroids from radar observations, Remote Sensing Reviews, 8:1-3, 195-203, DOI: 10.1080/02757259309532195

SAR Shape Model Results

Level 1 Reqs

MP5	The mission shall model 50% [TBC] of the object's shape within +/- 10 meters using active measurement.	✓ Period ≤ 8 minutes

- Encounter Parameters
  - Relative Velocity: 13 km/s
  - Slant Range to ISO when Data Collection Begins and Ends: 800 km
  - Closest Approach Distance to ISO: 400 km
  - SAR coherent processing interval: 1.63 seconds
  - Pulse every 3 degrees in angular position to ISO

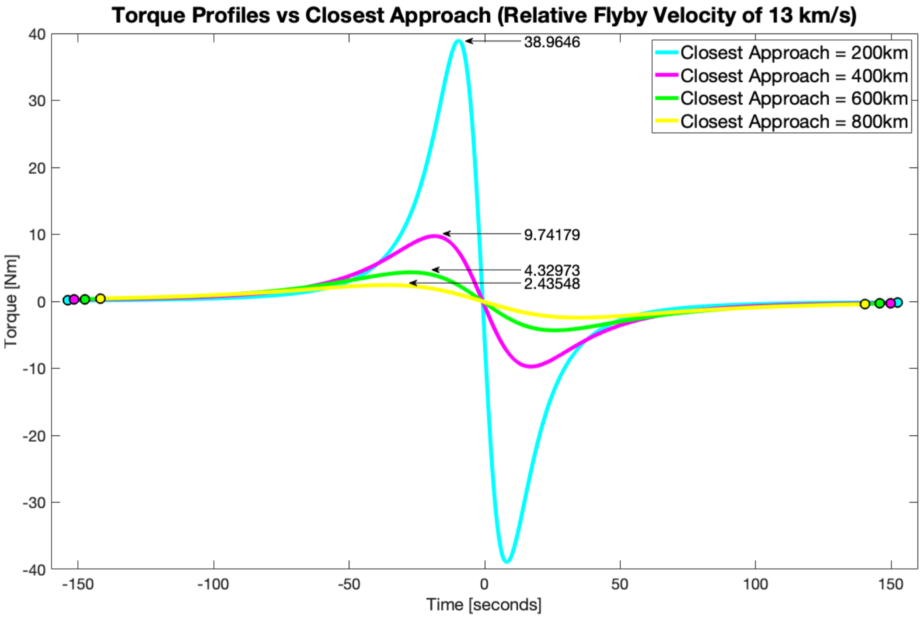


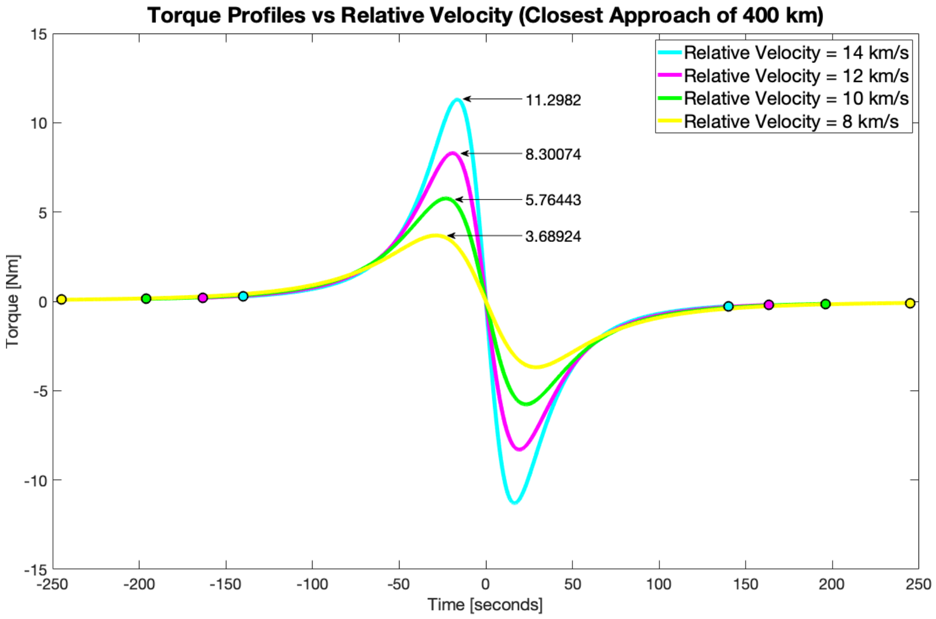
Shape Model – Additional Data

- The visual images taken during the flyby can also be used to generate a shape model.
- This data can supplement the radar data to develop a better shape model.
- All visual images are expected to capture around 48.75% of the total ISO surface area.

Resolution (meters per pixel)	Number of Images
≤ 5	43
5 - 10	32 (IR)
10 - 15	46
15 - 20	46
20 - 25	46
25 - 30	44

Torque vs. Encounter Variables





Speaker: Scott P.

Encounter Summary

Parameter	Range	Driving Considerations
Maximum Encounter S/C Relative Velocity to ISO	13 km/s	Decision to use Chemical propulsion system
Distance from ISO AON Begins	55000 km	Distance that ISO resolves to > 1 pixel using APIC
Primary Telescope Begins Taking IR Images	4000 km away from ISO on inbound trajectory	10 mpp spatial resolution is achieved at 4000 km
Primary Telescope Begins Taking Visible Images	2000 km away from ISO on inbound trajectory	5 mpp spatial resoluton is achieved at 2000 km
SAR Begins	800 km away from ISO on inbound trajectory	The SNR for the SAR is too low past 800 km
Closest Approach Distance	400 km	The SNR for the SAR is sufficient at 400 km
SAR Ends	800 km away from ISO on outbound trajectory	The SNR for the SAR is too low past 800 km
Primary Telescope Stops Taking Visible Images	2000 km away from ISO on outbound trajectory	No longer satisfy solicitation req at this distance
Primary Telescope Stops Taking IR Images	4000 km away from ISO on outbound trajectory	No longer satisfy solicitation req at this distance
Encounter Duration	11 min and 14 sec +/- 3 min and 30 sec	Function of closest/furthest approach distance and relative velocity
Maximum Distance from Sun at Closest Approach	5 AU	Beyond this we won't have enough sunlight for solar power
Number of SAR bursts	41	Provides data redundancy and prevents doppler smearing
Maximum Slew Acceleration Rate	3.93×10^{-2} rad/s^2	Maintain pointing for ISO within FOV
Total Amount of Data Taken	469.57 MB	Total encounter data collected

Encounter System Design

Speaker: Helen

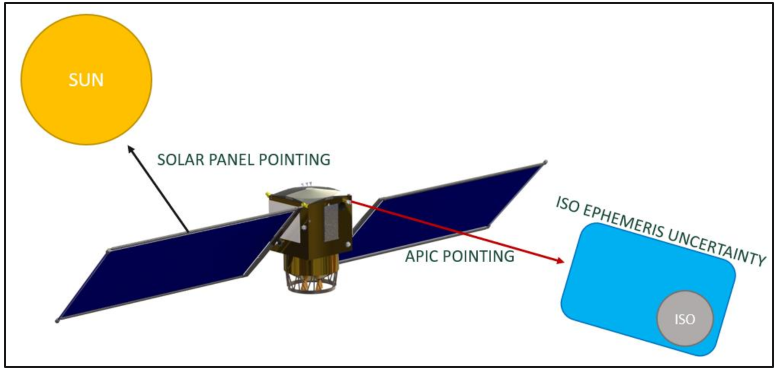
Phase 5: Encounter

Applicable Level 3 GNC Requirements

ID	Requirement	Driving Phase	Compliance (Y/N/M)
GNC 6	The GNC system shall propagate the spacecraft relative position of the ISO to +/- 100 m in the imager cross track axis from T-3 minutes to T+3 minutes.	Phase 5	Maybe
GNC 7	The GNC system shall propagate the spacecraft relative position of the ISO to +/- 1.2 km in the imager down track axis from T-3 minutes to T+3 minutes.	Phase 5	Maybe
GNC 8	The GNC system shall propagate the spacecraft relative velocity of the ISO to +/- 50 cm/s in the imager down track axis from T-3 minutes to T+3 minutes.	Phase 5	Maybe
GNC 9	The GNC system shall propagate the ISO ephemeris from 24 hours prior to the beginning of autonomous operations to 24 hours after closest approach.	Phase 5	Yes

Auto-OpNav

Method	Time	Down-Track Error	Cross-Track Error
OpNav	T - 24 hrs	9.0 km	9.0 km
Auto OpNav	T - 0 s	1.2 km	0.1 km



- Utilize APIC system to image ISO and star field simultaneously
- APIC resolution becomes **<1 km/pixel** at 55,000 km from the ISO
- ISO ephemeris will be updated every 15 seconds
- No trajectory correction maneuvers after autonomous cutoff

Primary Pointing

- APIC – Pointed at expected ISO position
- Solar Array – Pointed to Sun

Mechanisms in use

- APIC
- Solar Drive Mechanism

Mechanism	Source	Mass	Continuous Draw	Actuation Draw	Actuation Type	Range	Resolution	Operating Temperatures	Details
APIC Gimbal	JPL	<5 kg	<12 W	<12 W	Elevation Actuator	+/-90 deg	19 microrad	Actively controlled	[6]

Phase 5: Encounter

Applicable Level 4 C&DH Requirements

ID	Requirement	Driving Phase	Compliance (Y/N/M)
CDH-C11	The processor card shall have at least 5 MB of RAM.	Phase 5	Y
CDH-C12	The processor card shall handle at least 85.21 kHz bandwidth.	Phase 5	Y
CDH-C13	The C&DH subsystem shall have a redundant copy of itself.	Phase 5	Y
CDH-C14	Each C&DH subsystem shall have a redundant mass memory card.	Phase 5	Y

Backup

OBC Design – C&DH

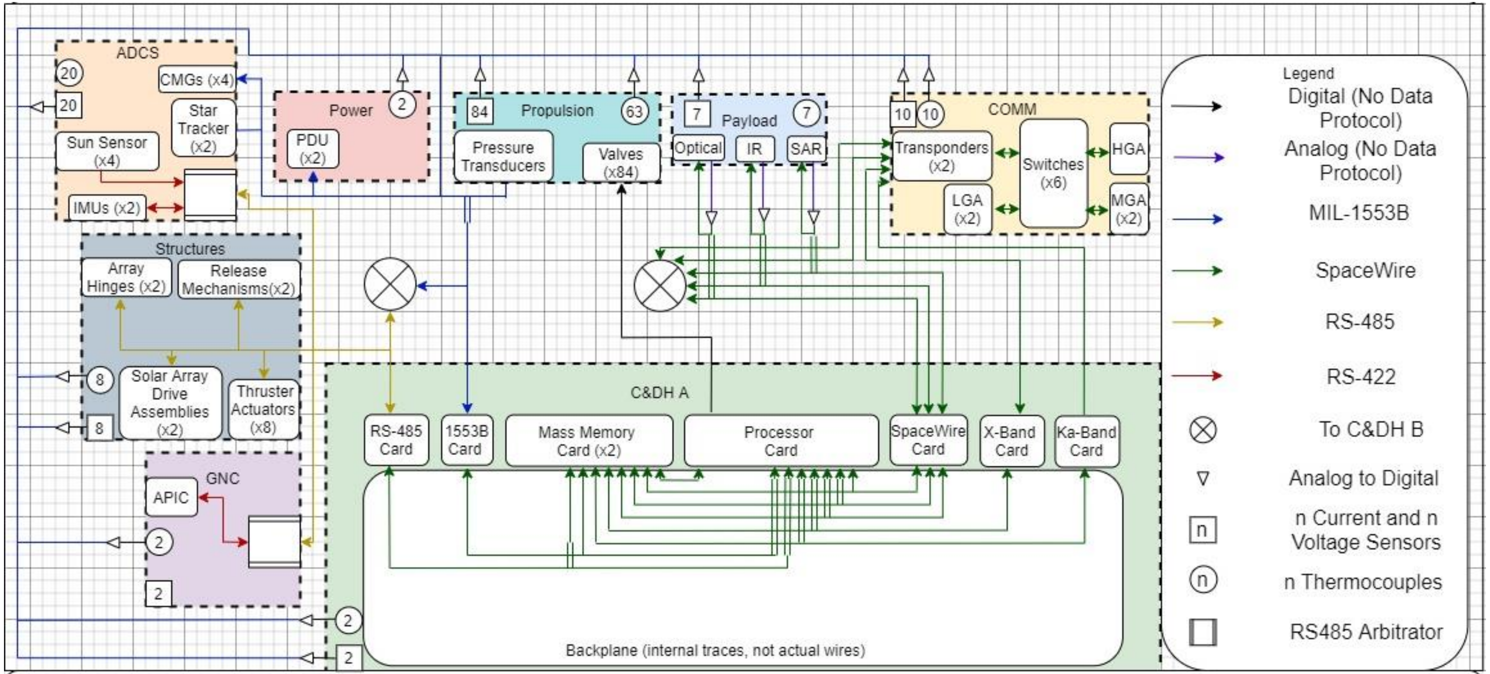
ID	Requirement	Driving Phase	Compliance (Y/N/M)
CDH-C11	The processor card shall have at least 5 MB of RAM.	Phase 5	Y
CDH-C12	The processor card shall handle at least 85.21 kHz bandwidth.	Phase 5	Y

- Current minimum required RAM is 5 MB
    - Informed via APIC autoNav requirements
- Each OBC will have **8 MB RAM**
- Processing speed 85.21 (kHz, direct calc from required)
- OBC will have **100 kHz** (inc. 1.2 F.S)
- Dimensions: 0.4 x 0.3 x 0.25 m (**0.03 m^3 vol.)**
- Mass: **30 kg**
- Power varies by phase, max **95 W**
- Will have an additional C&DH system, one processor card per system

Operation	Processing Speed (kHz)
Secondary Science	69.61
Encounter	85.21

Backup

Data Harness – C&DH



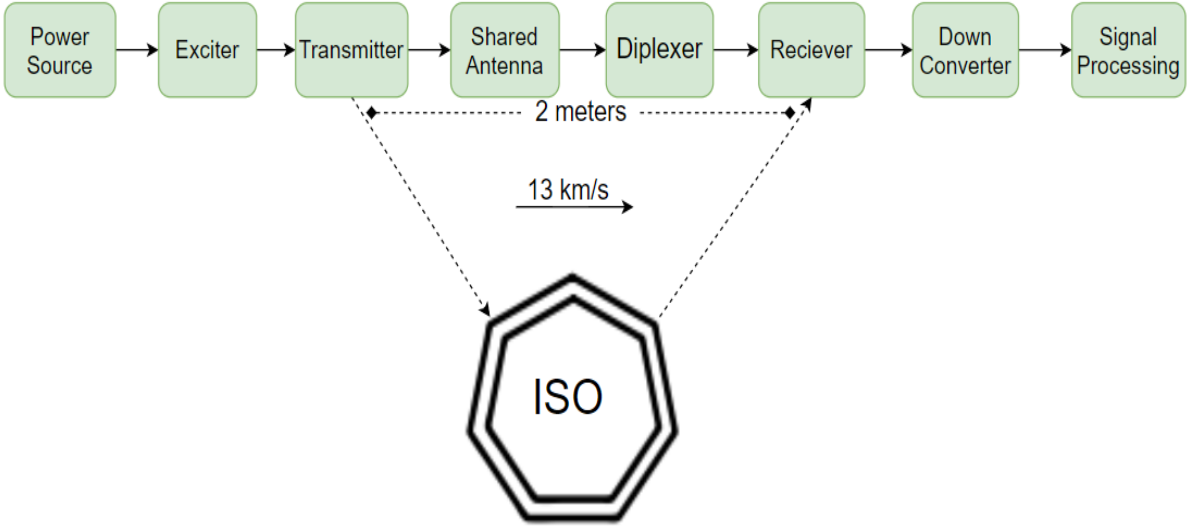
*Not all components of the flight system are shown in this diagram; only components that have a direct data link

Backup

Design Specifications - SAR

ID	Requirement	Driving Phase	Compliance (Y/N/M)
PAY-C15	The SAR shall have a gain of 56 dB.	Phase 5	Yes
PAY-C16	The SAR shall acquire radar data of the object with a doppler uncertainty of less than or equal to 0.32 Hz at closest approach.	Phase 5	Yes

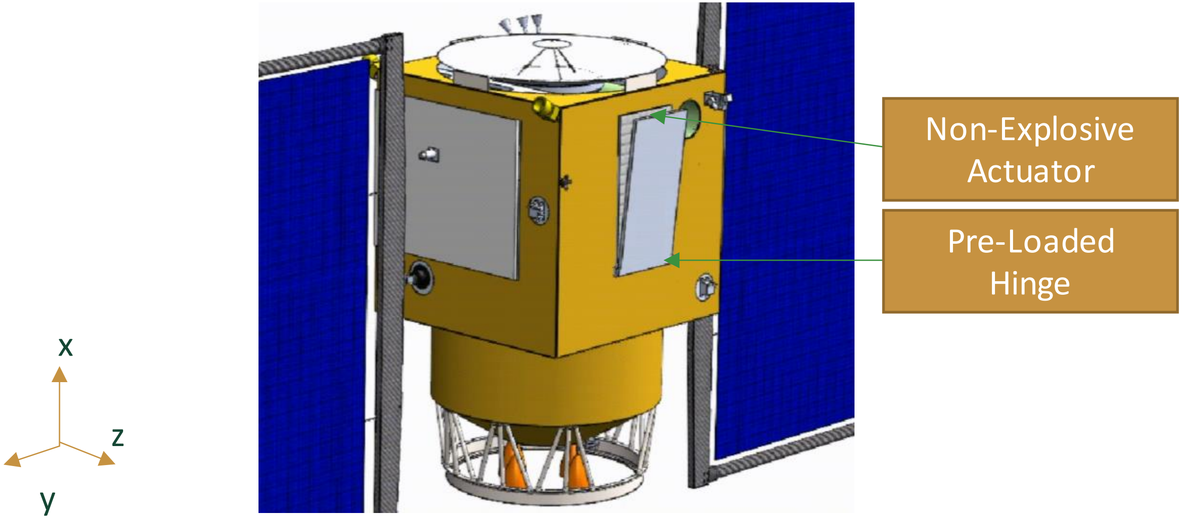
SAR Parameter	Value
Bandwidth	0.1 GHz
Central Frequency	26.5 GHz
Power Average	16 W
Power Peak	246 kW
Antenna Size	4m x 1m
PRF	6500 Hz



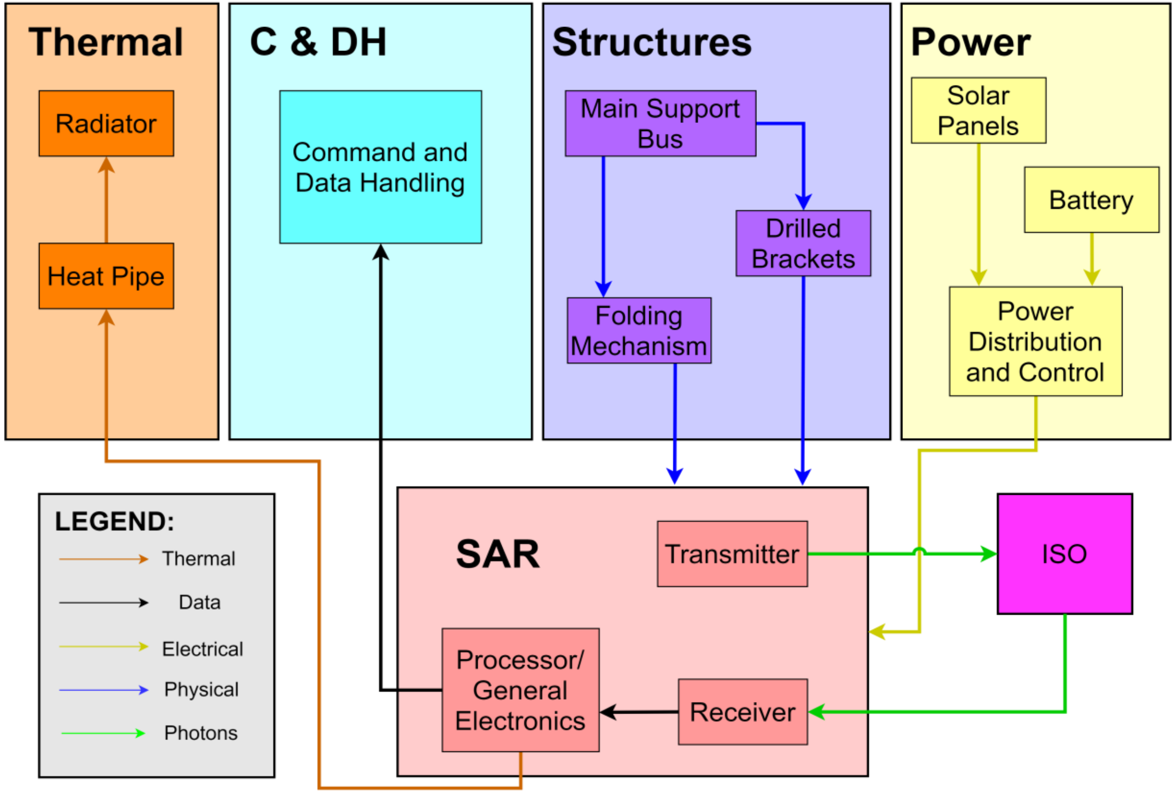
To SAR Model Verification:

SAR Mechanisms - Structures

Mechanism	Source	Mass	Continuous Draw	Actuation Draw	Actuation Type	Range	Resolution	Operating Temperatures	Details
Non-Explosive Actuator for SAR	NEA Model 9100	0.7 kg	250 mA	4 A	Hold Down and Release	N/A	N/A	-135C to +135C	[4]
Deployment Hinge for SAR	Deployment System for Large Appendages	1.5 kg	N/A	N/A	Spring driven	90-180 deg.	+/- 0.006 deg.	-30C to +50 C (Survivable temperatures +/-150 C)	[5]



Bus Connections – Payload (SAR)

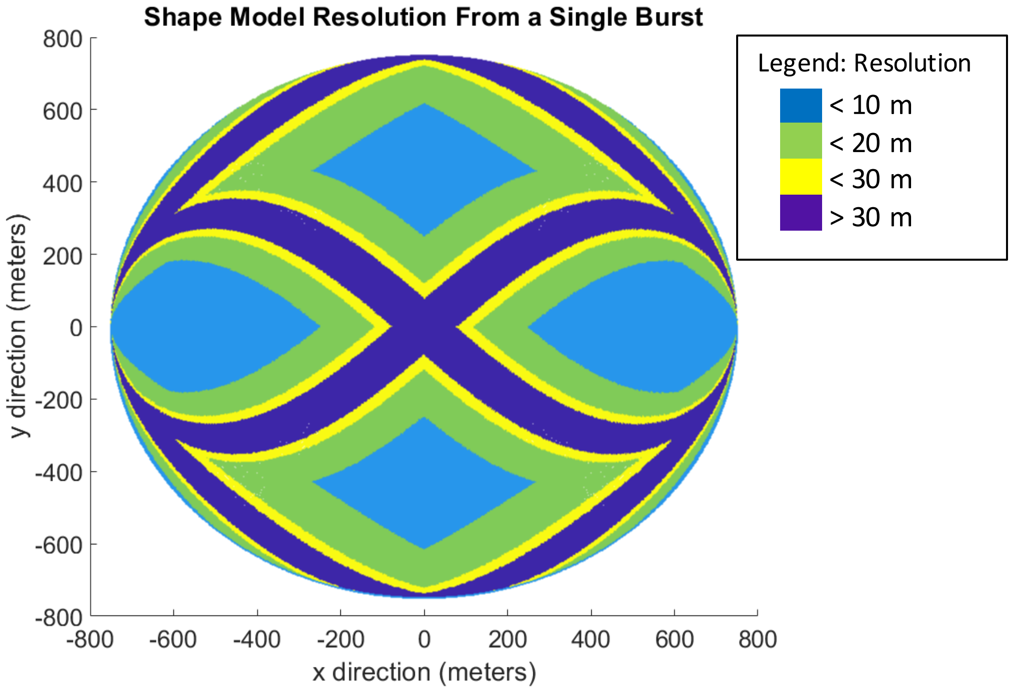


Advanced Object Definition - SAR

- Active measurement of shape and range
  - Achieved since we are using the SAR for our morphology model
- Active measurement of surface dielectric properties
  - SAR can determine dielectric constant by computing the co-polarized radar cross section for VV and HH polarizations.

$$\frac{\sigma_{HH}}{\sigma_{VV}} = \frac{COS(\theta_i) - \sqrt{\varepsilon_r - \sin^2(\theta_i)}\left(COS(\theta_i) + \sqrt{\varepsilon_r - \sin^2(\theta_i)}\right)^{-1}}{(\varepsilon_r - 1)\left[\sin^2(\theta_i) - \varepsilon_r - \varepsilon_r \sin^2(\theta_i)\right]\left[\varepsilon_r \cos(\theta_i) + \sqrt{\varepsilon_r - \sin^2(\theta_i)}\right]^{-2}}$$

Equation: Marghany, Maged. "Synthetic Aperture Radar Imaging Mechanisms for Oil Spills" 2020



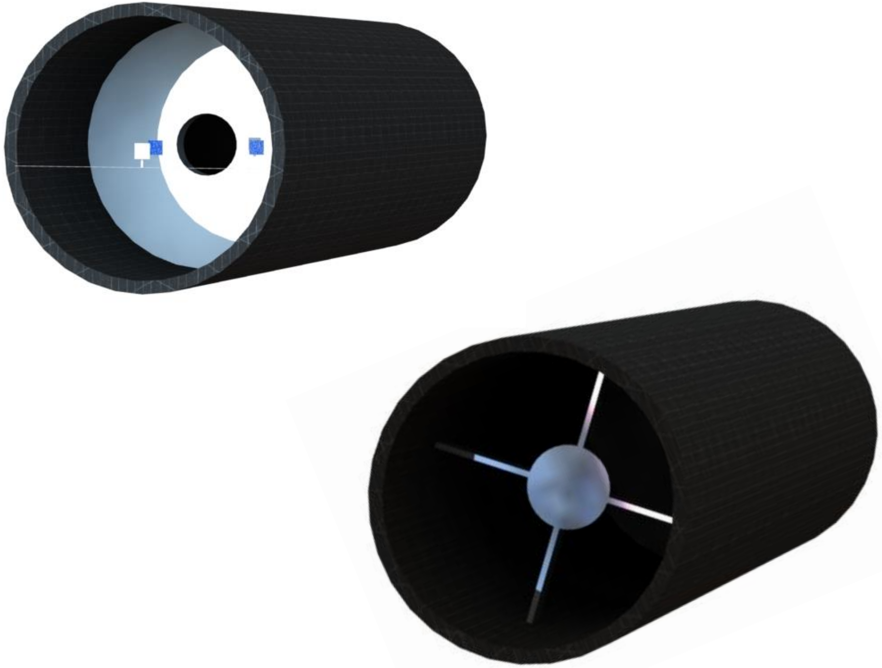
Phase 5: Encounter - Imager

Applicable Level 3 Payload Requirements

ID	Requirement	Driving Phase	Compliance (Y/N/M)
PAY4	The payload shall be capable of taking a visible image once every 200 milliseconds.	5	Y
PAY5	The payload shall be capable of taking an infrared image every 200 milliseconds.	5	Y
PAY6	The payload shall be capable of taking a visible image with an exposure time of 13 milliseconds.	5	Y
PAY7	The payload shall be capable of taking an infrared image with an exposure time of 13 milliseconds.	5	Y
PAY8	The payload shall have an FOV of 0.55 degrees.	5	Y
PAY9	The payload shall articulate between the visible and infrared sensors.	5	Y
PAY12	The visible CCD board shall collect the visible imagery signal.	5	Y
PAY13	The infrared CCD board shall collect the infrared imagery signal.	5	Y

Design Specifications - Imager

- Imaging system designed to meet 10 meter per pixel IR resolution and 5 meter per pixel visual resolution
- Cassegrain telescope
- CCD boards used for visual and IR sensors
- Pick off mirror to direct light into correct sensor
- 15 ms exposure time

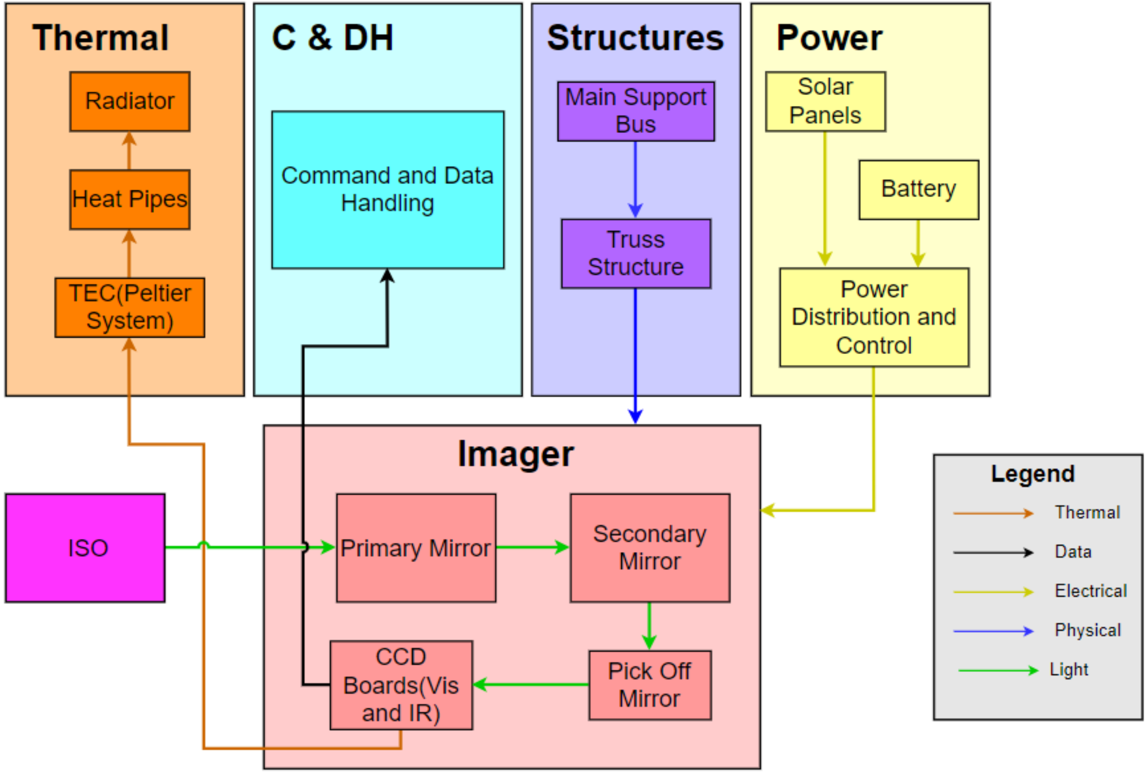


Design Specifications - Imager

Imager Parameter	Value
Total Length	2 m
Primary Aperture	0.500 m
Secondary Aperture	0.125 m
System Focal Length	6 m
FOV (visual and IR)	0.55 deg

Sensor Parameter	Value
Pixel Count	2715 x 2715
Pixel Size	0.015 mm x 0.015 mm
Full Well Depth	300000 e
Pixel Data Rate	3-6 MHz
Readout Rate	20 ms
Image Size	1.87 MB

Bus Connections – Payload (Imager)



ISO Related Risks

Given that...	There is a possibility of...	Adversely impacting...	Which can result in...	Likelihood (out of 5)	Severity (out of 5)	Sub team
There are uncertainties about the ISO	The coma, temperature, albedo, or position of the ISO are vastly different from what we expect	The encounter	Us not being able to obtain the images that we expect	3	3	Payload

Likelihood	1	2	3	4	5
	0-20%	20-40%	40-60%	60-80%	80-100%

Severity	1	2	3	4	5
	Minimal impact on objectives	Minor impact on objectives	Unable to achieve a particular objective	Unable to achieve multiple objectives	Unable to achieve all objectives

ISO Flyby Risk - Structure

Given that...	There is a possibility of...	Adversely impacting...	Which can result in...	Likelihood (out of 5)	Severity (out of 5)	Sub team
Debris that is larger than the critical diameter of the shielding or has a very high velocity impacts the spacecraft	The payload and critical flight components being damaged	The payload and support bus shielding	Catastrophic failure	1	5	Structures

Likelihood

0-20%	20-40%	40-60%	60-80%	80-100%
1	2	3	4	5

Severity

Minimal impact on objectives	Minor impact on objectives	Unable to achieve a particular objective	Unable to achieve multiple objectives	Unable to achieve all objectives
1	2	3	4	5

Phase 5: Encounter

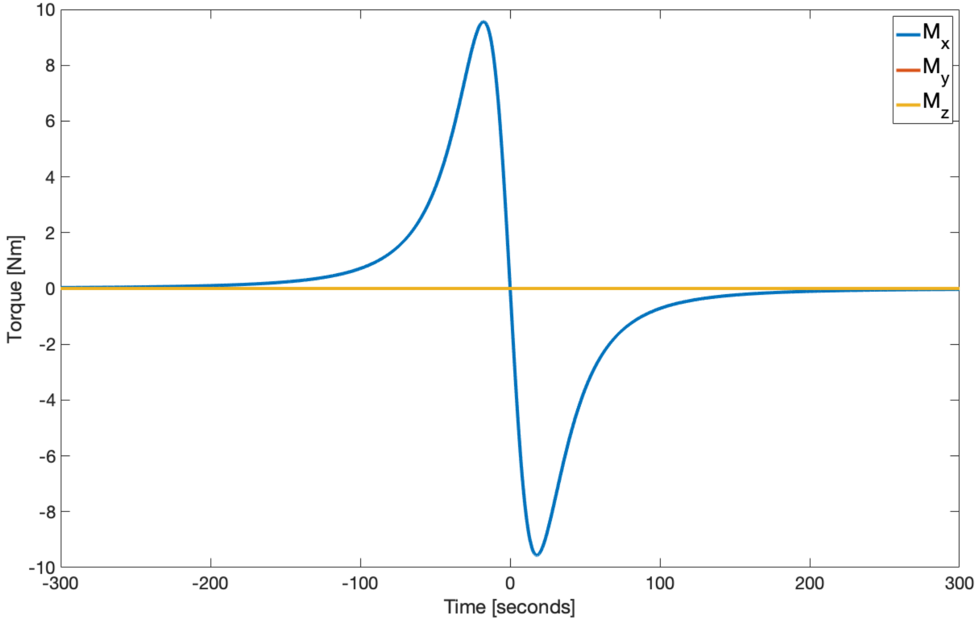
Applicable Level 3 Pointing & ADCS Requirements

ID	Requirement	Driving Phase	Compliance (Y/N/M)
ADC1	The ADCS shall point the optical boresight with an accuracy of 742 arcsec during primary science acquisition	Phase 5	Y
ADC3	The ADCS shall be capable of a maximum slew acceleration of 0.0393 deg/s^2 during primary science acquisition.	Phase 5	Y
ADC11	The ADCS shall control the imager boresight stability to 21 arcsec/sec during primary science acquisition.	Phase 5	M

Encounter Torques-ADCS

- Max Torque during Encounter of 9.5 Nm
- Each of our four CMGs selected can deliver 12 Nm of torque
- Models assumes a relative velocity at encounter of 13 km/s, closest approach of 400 km, and a single axis slew about x-axis
- Individual Wheel Torques and Gimbal Angles are within capability of the 12 Nm CMGs

Backup



Required MED Torques during Encounter

Pointing Budget - ADCS

Pointing Budget Description	Per-Axis Error Value (3 σ) [arcsec]	Radial Pointing Error (3 σ) [arcsec]	Radial Pointing Requirement [arcsec]	Stability Error [arcsec / s]	Stability Requirement [arcsec/s]
Encounter	580.19	741.60	741.60	12.79*	20.63

- For encounter, ADCS meets the accuracy requirements as clarified by the payload team.
- Stability error does not account for the slew performance error related to the angular velocity.
    - *Required performance error to meet requirement: < 0.03% of maximum slew rate

Mass, CM, and Inertia Matrix

Mass Matrix (Total, Empty):

Moments of inertia: (kilograms * square meters)
Taken at the center of mass and aligned with the output coordinate system.

Lxx = 63000, Lyy = 70600, Lzz = 14200

Mass Matrix (Total, Full):

Moments of inertia: (kilograms * square meters)
Taken at the center of mass and aligned with the output coordinate system.

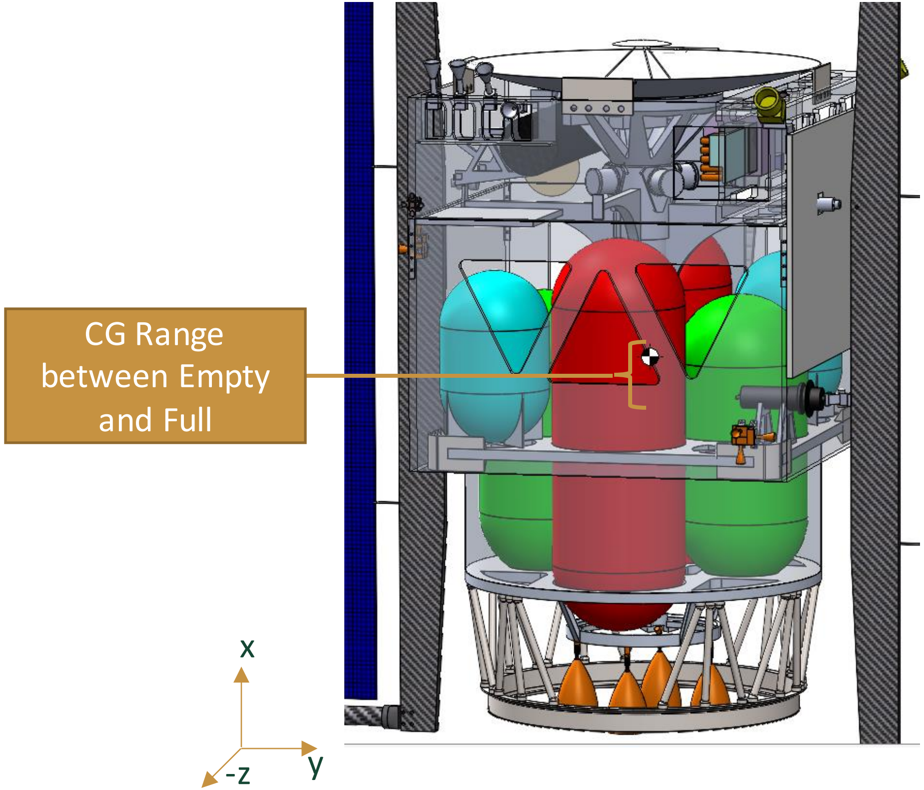
Lxx = 67100, Lyy = 78300, Lzz = 22000

Mass Matrix (Arrays):

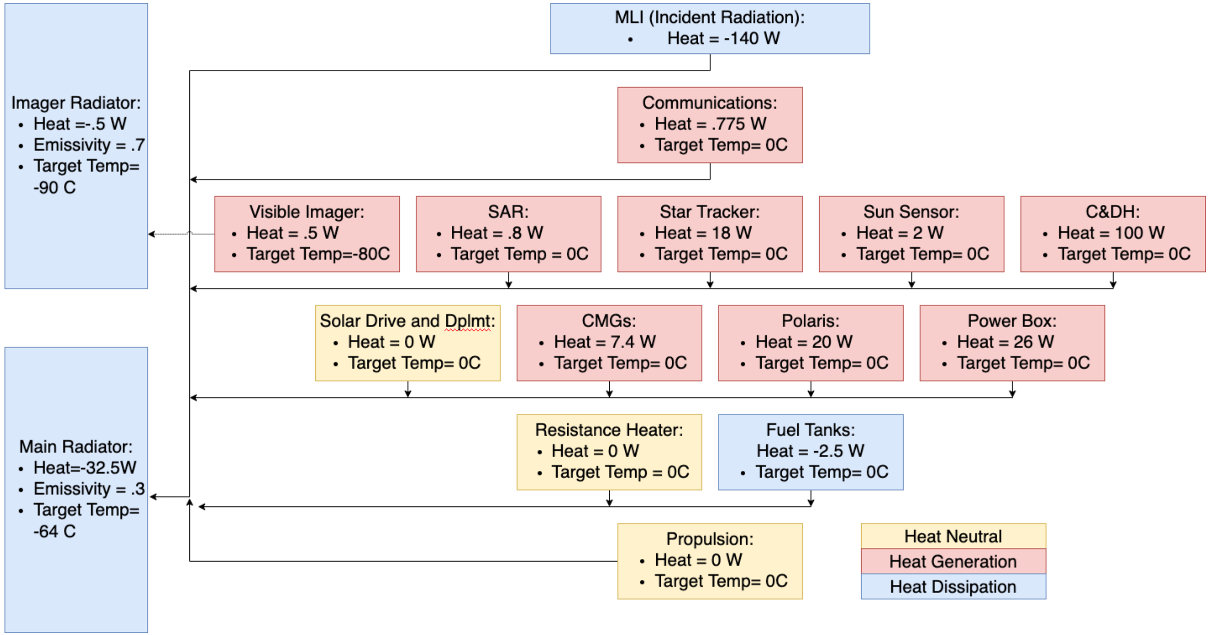
Moments of inertia: (kilograms * square meters)
Taken at the center of mass and aligned with the output coordinate system.

Lxx = 60000, Lyy = 64300, Lzz = 5000

(Full) CG from PAF (x = 2.5m)



Thermal Control– Science Instruments



Phase 6

Speaker: Austin I

Phase 6: Data Downlink

Applicable Level 1 Requirements

ID	Requirement
MP9	The mission shall return data to the customer no later than 9 months post collection.

Applicable Level 2 Requirements

ID	Requirement
F17	The flight system shall downlink data during scheduled passes.
F18	The flight system shall transmit data to the ground system with a minimum data rate of 32.4 kbps during primary science data downlink.
F21	The flight system shall be capable of retransmitting data to the ground system.

Phase 6: Data Downlink

Applicable Level 2 ***Ground System Requirements***

ID	Requirement
G17	The ground system shall receive data from the flight system with a data rate of at least **32.4 kbps.**
G19	The ground system shall verify that all requested data packets were received within **TBD** hours of receipt.
G37	The ground system shall establish two-way communication sessions with the flight system **on average of once a day for four-to-eight hours per session** during the post-encounter downlink mission phase.

Speaker: Alexi D.

Phase 6: Data Downlink

*Applicable Level 3 **Telecommunication Requirements***

ID	Requirement	Driving Phase	Compliance
COM15	The communication system shall transmit science data with a minimum data rate of TBD bps.	Phase 6	Maybe
COM16	The communication system shall receive commands with a data rate of up to 2 kbps [TBC].	N/A	YES
COM22	The communication system shall have a high gain transmission EIRP of 80.89 dBW [TBC].	Phase 6	YES

Ka-Band Downlink Budget

Spacecraft Parameter	Value
Antenna	HGA [Ka Band]
Boresight Gain [dBi]	57.88
3 dB Beamwidth [deg]	.217
Centering Frequency [GHz]	32.083
Transmission Power [W]	200
Max Pointing Error [deg]	0.10
Required BER	1E-6
Modulation Scheme	Direct BPSK
Coding	Turbo (R=½, l=5)
Required Eb/No [dB]	1.25
Nominal Link Margin [dB]	3.00

DSN Parameter	Value
Antenna	34m BWG
Boresight Gain [dBi]	79.30
Max Pointing Error [deg]	0.077
Required BER	1E-6
Modulation Scheme	Direct BPSK
Coding	Turbo (R=½, l=5)
Min Elevation Angle [deg]	10.5
Atmospheric Loss [dB]	2.00

Backup

Phase 6: Data Downlink

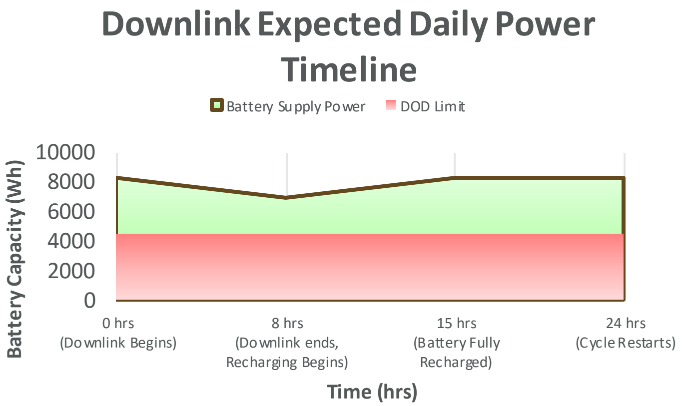
Applicable Level 3 Power Requirements

ID	Requirement	Driving Phase	Compliance
POW1	The power system's battery shall be capable of supplying a minimum of **1407 + TBD Wh** at end of life.	Phase 6	Yes
POW2	The power system's solar panels shall generate a minimum of **690 + TBD** Watts at end of life.	Phase 6	Yes
POW5	The power system shall provide **305 +/- TBD** Watts during system fault mode.	N/A	Yes

Downlink Power Budget

- End of life battery system capability assumed to degrade a total of 8% over 22 years.
  - Daily power demands during downlink were taken from the operational timeline to verify battery's capability to supply power for the power deficit (~197 W) during downlinking at a max range of 7AU

Subsystem	Component	Power Draw (W)	During Downlink (8 hours)	Recharging (16 hours)
C&DH	C&DH Components*	100	ON	ON
Comms	Deep Space Transponder	13	ON	ON
	Travelling Wave Tube Amplifier K-band	335	ON	OFF
	Ultra Stable Oscillator	3	ON	ON
	Antenna	50	ON	OFF
Thermal	Thermal Components*	201	ON	ON
ADCS	ADCS Components*	186	ON	ON
		Total (W):	**887**	**503**

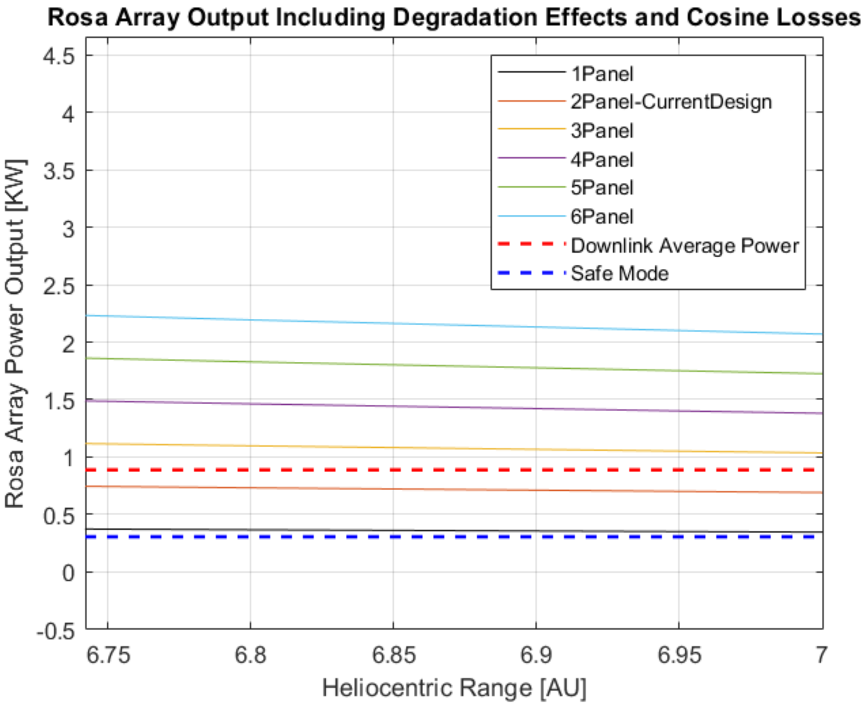


Downlink Power Budget

Table 1. ROSA Array Specifications

Size Deployed (m2)	Size Stowed (m^3)	Mass (kg)	Specific Power (W/kg)	BOL Power @ 1 AU (kW)
156.25	~1.14	407.59	112.3	45.776

- End of life array power generation taken to be reduced by ~1.25% yearly from exposure to the operating environment
  - Assumed average cosine loss is 10 degrees throughout
- Array sized for worst case scenario during most power sensitive portions of the mission
  - Initial model limit taken to be 5 AU given additional panel mass and power generation trade would be not optimal due to solar irradiance fall off.



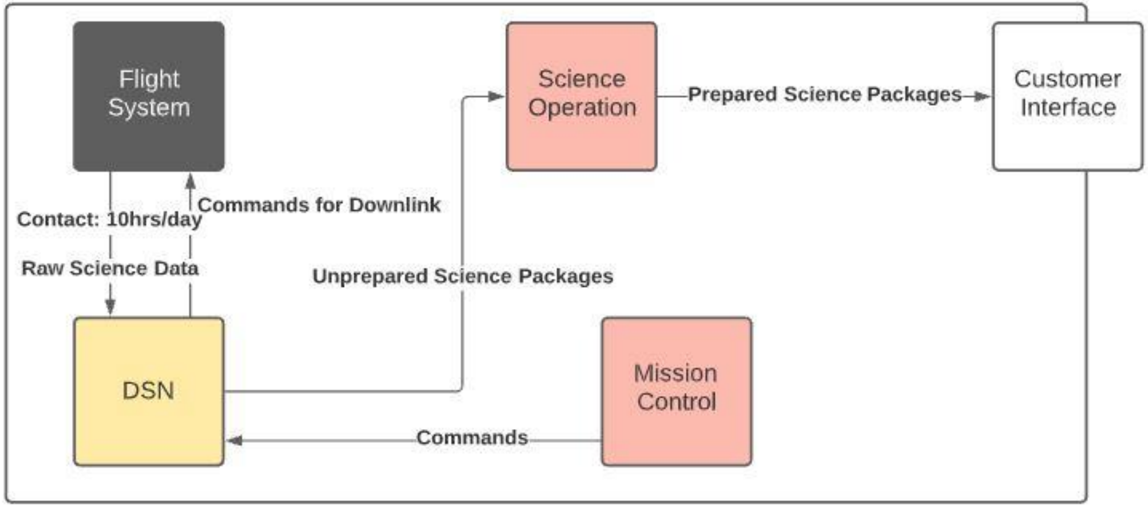
Phase 6: Data Downlink

Applicable Level 3 ***Ground System Requirements***

ID	Requirement	Driving Phase	Compliance (Y/N/M)
GS6	The Mission Control Center shall decide whether any customer provided ISO ephemeris meets the mission capabilities	N/A	Yes
GS3	The Mission Control Center shall design and execute commands to meet the mission demands	N/A	Yes
GS9	The Data Storage and Network shall store all data of all types locally	N/A	Maybe
GS1	The Science Operations Center shall analyze science data stored in the Data Storage and Network using science models	N/A	Yes
GS2	The Science Operations Center shall provide science packages to the customer through the customer interface	N/A	Yes

Ground Operations – Post Encounter Downlink

- Data Storage
  - Daily ground contacts for up to 10hrs
  - Data immediately transfers to GS facilities for storage and processing
- Science Product Generation
  - Science models are assembled from raw data packages
  - Complete products are generated for analysis
- Customer Interface
  - Science products packaged for customer according to accessibility requirements
  - Once packaged, products made available via required transmission methods



Phase 7

Speaker: Austin I

Individual Spacecraft Decommission

Encounter Spacecraft (ES)

- Customer can either choose to have it attempt one of the extended mission scenarios or move directly into decommission.
- ES will be in deep space and in a heliocentric orbit (could also be a hyperbolic trajectory)
- Will just follow the decommission procedure outlined in the next slide

Secondary Spacecraft (SES)

- Customer can either choose to have it attempt one of the extended mission scenarios or move directly into decommission.
    - If SES pursues an ISO, then it will follow ES decommission process
- After 20 years, SES health and telemetry will be analyzed and presented to customer
    - Good condition: customer can choose to decommission or outsource SES
    - Bad condition: decommission
- If at any point during 20-year span the SES is in bad condition, it will be decommissioned

Backup

Decommission Procedure

Order	Decommission Activity
1	Dump remaining propellant from tanks and lines.
2	Vent pressurant from tanks and chambers to safe levels.
3	Turn off payload and science instrumentation.
4	Turn off ADCS and GNC equipment.
5	Shut down thermal subsystem.
6	Turn off communications subsystem.
7	Discharge batteries and power subsystem.
8	Turn off solar arrays.

Phase 7: Decommission

Applicable Level 2 Flight System Requirements

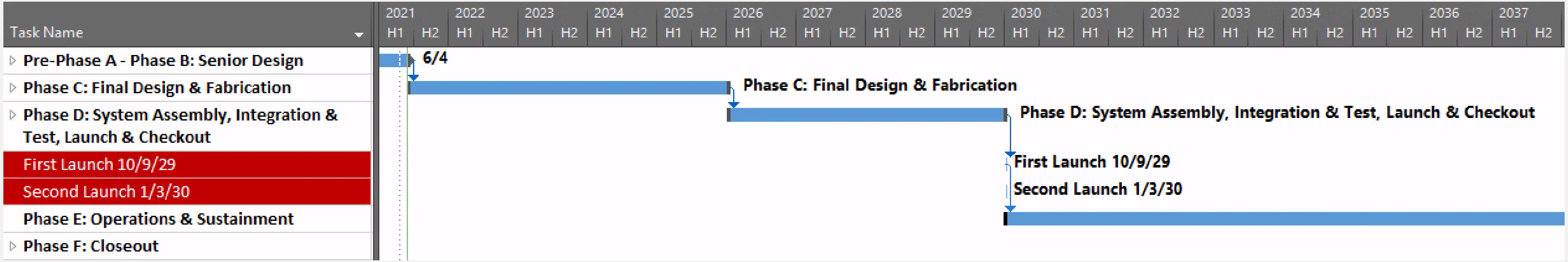
ID	Requirement
F5	The flight system shall operate in the space environment for a minimum of 22 years.
F33	The flight system shall follow decommission protocol as commanded by ground.
G5	The ground system shall support the flight system for a minimum of 22 years.
G29	The ground system shall command the flight system to decommission.

Project Life Cycle, System Integration, and Testing

Speaker: Luke

Backup

COSMIC Project Life Cycle



Starting Point: NASA Program/Project Life Cycle

- NASA outlines 8 Phases, we have 5 Phases
- Pre-Phase A – Phase B: Senior Design
- Phase C: Final Design and Fabrication
- **Phase D: System Assembly, Integration & Test, Launch & Checkout**
- Phase E: Operations & Sustainment
- Phase F: Closeout

System I&T Schedule

- Timeline overall flow
- Assumption:
  - IHM and bus modules arrive fully integrated and tested
- Any major integration will be followed by a functional test
- 20% margin built into each task duration
- Shipping and handling is built into the duration of tasks

IHM/Bus Integration

TVAC/EMI

Solar Panels

Vibe/Acoustic

Final Tasks

Launch Activates

Phase D: System Assembly, Integration & Test, Launch & Checkout	1011 days	Wed 2/18/26	Wed 1/2/30
Cosmic - A: Proto/Qualification	929 days	Wed 2/18/26	Mon 9/10/29
Integrate IHM & Bus	70 days	Wed 2/18/26	Tue 5/26/26
IHM & Bus Functional Test	56 days	Wed 5/27/26	Wed 8/12/26
Integrate Antennas	21 days	Thu 8/13/26	Thu 9/10/26
Communication System Functional Test	21 days	Fri 9/11/26	Fri 10/9/26
Integrate Radiators	21 days	Mon 10/12/26	Mon 11/9/26
Thermal System Functional Test	21 days	Tue 11/10/26	Tue 12/8/26
Full Functional Test	49 days	Wed 12/9/26	Mon 2/15/27
Thermal Vacuum Qualification Test	63 days	Tue 2/16/27	Thu 5/13/27
Functional Test	49 days	Fri 5/14/27	Wed 7/21/27
EMI/EMC Qualification Test	49 days	Thu 7/22/27	Tue 9/28/27
Functional Test	49 days	Wed 9/29/27	Mon 12/6/27
Integrate Solar Panels	14 days	Tue 12/7/27	Fri 12/24/27
Power Functional Test	14 days	Mon 12/27/27	Thu 1/13/28
Functional Test	35 days	Fri 1/14/28	Thu 3/2/28
Vibration Qualification Test	42 days	Fri 3/3/28	Mon 5/1/28
Functional Test	49 days	Tue 5/2/28	Fri 7/7/28
Acoustic Qualification Test	42 days	Mon 7/10/28	Tue 9/5/28
Solar Panel First Motion Deployment Test	21 days	Wed 9/6/28	Wed 10/4/28
Full Functional Test	49 days	Thu 10/5/28	Tue 12/12/28
Flight System Mass Properties Evaluation	35 days	Wed 12/13/28	Tue 1/30/29
Finalization	21 days	Wed 1/31/29	Wed 2/28/29
Operational Readiness Review	0 days	Wed 2/28/29	Wed 2/28/29
Margin	56 days	Thu 3/1/29	Thu 5/17/29
Chemical Propellant Loading	35 days	Fri 5/18/29	Thu 7/5/29
Deliver and Unpack at PPF	7 days	Fri 7/6/29	Mon 7/16/29
Functional Test	28 days	Tue 7/17/29	Thu 8/23/29
Flight Readiness Review	0 days	Fri 8/24/29	Fri 8/24/29
Integration with Fairing	7 days	Fri 8/24/29	Mon 9/3/29
SpaceX Activities	5 days	Tue 9/4/29	Mon 9/10/29
Storage COSMIC A	20 days	Tue 9/11/29	Mon 10/8/29
Cosmic - B: Proto/Qualification	929 days	Wed 5/13/26	Mon 12/3/29
Storage COSMIC B	22 days	Tue 12/4/29	Wed 1/2/30
First Launch	1 day	Tue 10/9/29	Tue 10/9/29
Second Launch	1 day	Thu 1/3/30	Thu 1/3/30

Environmental Testing

	TVAC	EMI/ECI	VIBRATION	ACOUSTIC
C.O.S.M.I.C.-A Protoflight Qualification Standards	• T-Vac: +/- **10°C** Max./min. of predicted levels • Thermal Cycling: +/- **25°C** Max./min. of predicted levels • Pressure: Venting analyses performed	• As specified for mission • Generate electromagnetic compatibility report	• Quasi-Static Loads: **1.25** x Limit Load for 5 cycles of 30 sec at full level per axis • Random Vibration: MEFL **+ 3 dB** for 1 min/axis • Sine Vibration: **1.25** x MEFL for 4 oct/min • Minimum component vibe test workmanship test: $6.8g_{rms}$	• MEFL **+ 3dB** for 1 minute (w/ minimum of 138 dB)
C.O.S.M.I.C.-B Acceptance Standards	• T-Vac: +/- 5°C Max./min. of predicted levels • Thermal Cycling: +/- 20°C Max./min. of predicted levels • Pressure: Venting analyses performed	• As specified for mission • Generate electromagnetic compatibility report	• Quasi-Static Loads: Limit Load for 5 cycles of 30 sec at full level per axis • Random Vibration: MEFL for 1 min/axis • Sine Vibration: MEFL for 4 oct/min • Minimum component vibe test workmanship test: $6.8g_{rms}$	• MEFL for 1 minute (w/ minimum of 138 dB)

Source: NASA GEVS, NASA-STD-7001, NASA-STD-7002

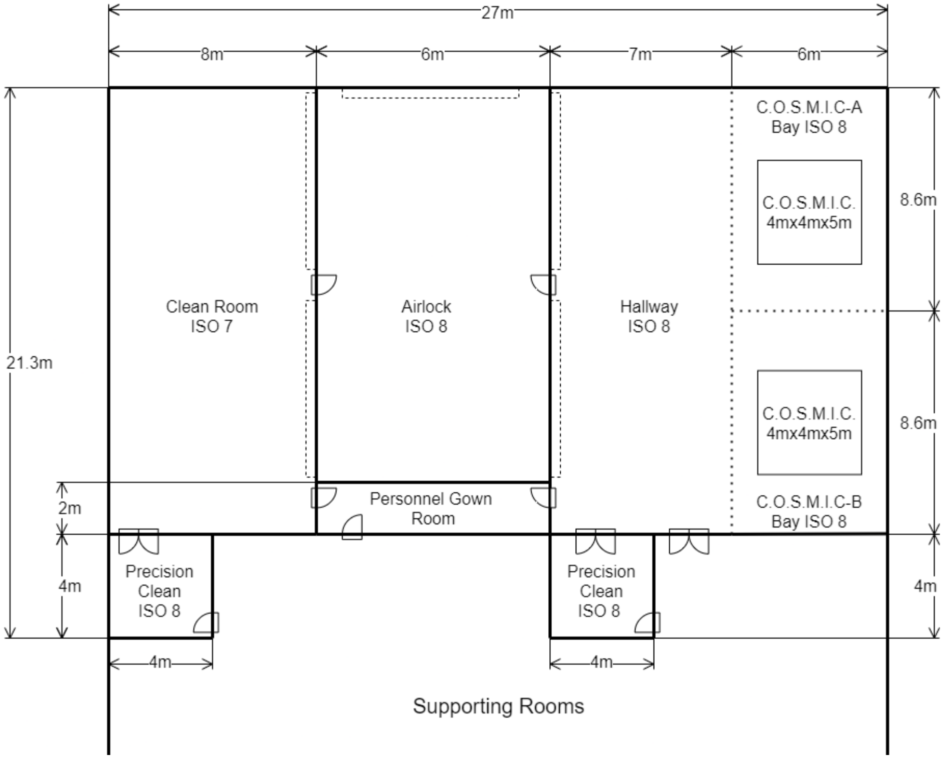
I&T – Personnel & Facilities

Personnel – 1 Shift Per Day

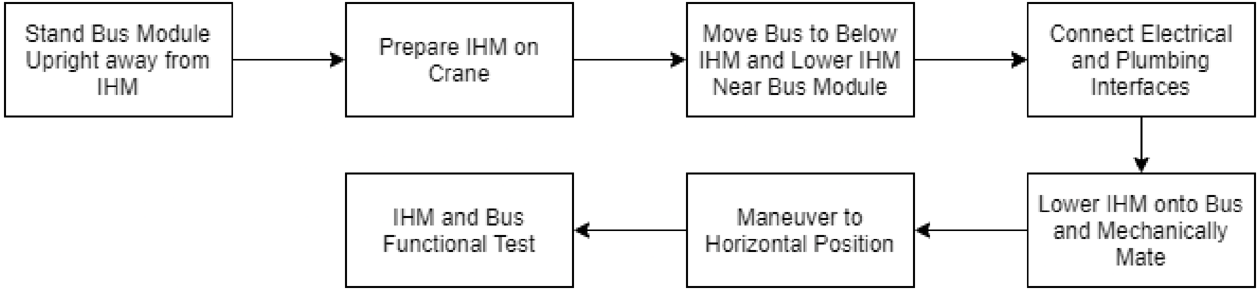
Teams	C.O.S.M.I.C.-A	C.O.S.M.I.C.-B
Engineers	8	8
Technicians	8	8
Total Personnel	16	16

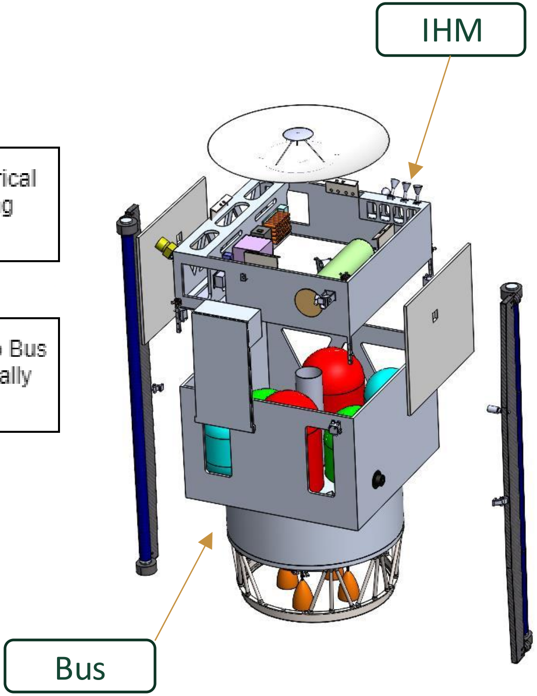
Facilities

- High-bay doors are 6.7 m wide
- Cranes and other lifting equipment in all three large rooms
- Supporting rooms for preparation and smaller assemblies below



I&T Task – IHM and Bus Module Integration Block Diagram





- Task occurs within the C.O.S.M.I.C. facility
- Process activities may be refined after integration of C.O.S.M.I.C.-A to optimize integration of C.O.S.M.I.C.-B

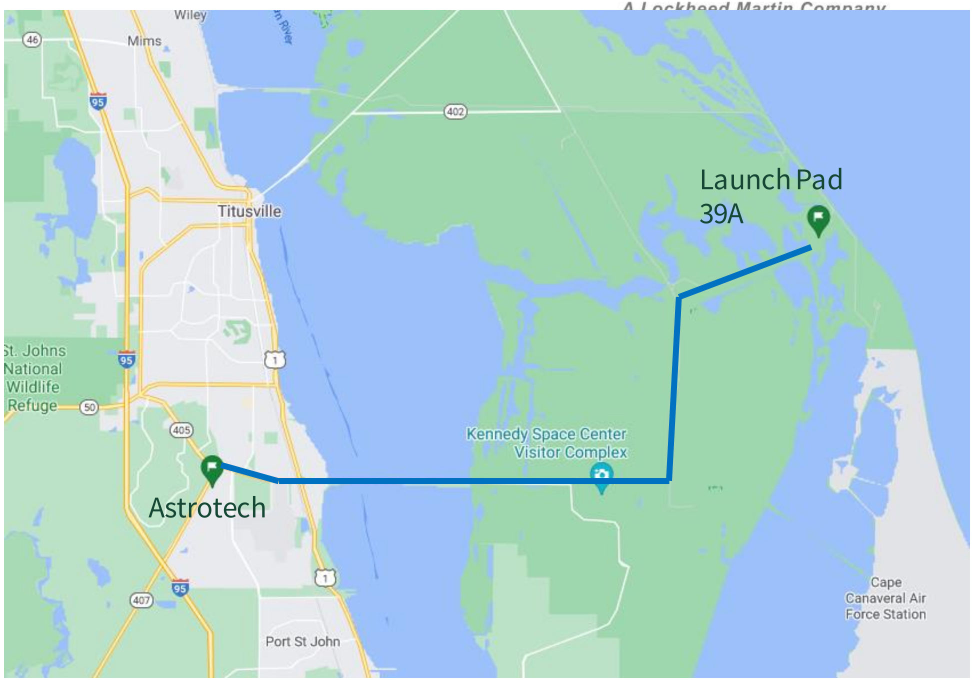
Pre-Launch Overview

Task	Begins Days Before Launch	Description
Chemical Propellant Loading	82	• Performed at Astrotech approximately 25 miles away from launch pad • Operations will occur over 21 work days
Delivery and Unpacking at PPF	47	• Non-standard service for additional time in the PPF will be utilized to perform a functional test to same standards as functional tests during I&T
Functional Test	40	• Electrical checkout to check health of subsystems before launch
SpaceX Activities	7	• SpaceX takes over operations • SpaceX integrates the spacecraft into the fairing • System Checks • Rollout to pad
Launch	0	

3rd Party Fueling

- Fueling service is about 20 miles (32 km) away from launch site
- 68 days before launch
  - Spacecraft arrives at fueling facility
- 47 days before launch
  - Spacecraft leaves fueling facility and is delivered directly to the SpaceX payload processing facility

Propellant	Amount per Spacecraft
Hydrazine	7200 kg
MON-3	7200 kg
He	$10\ m^3$

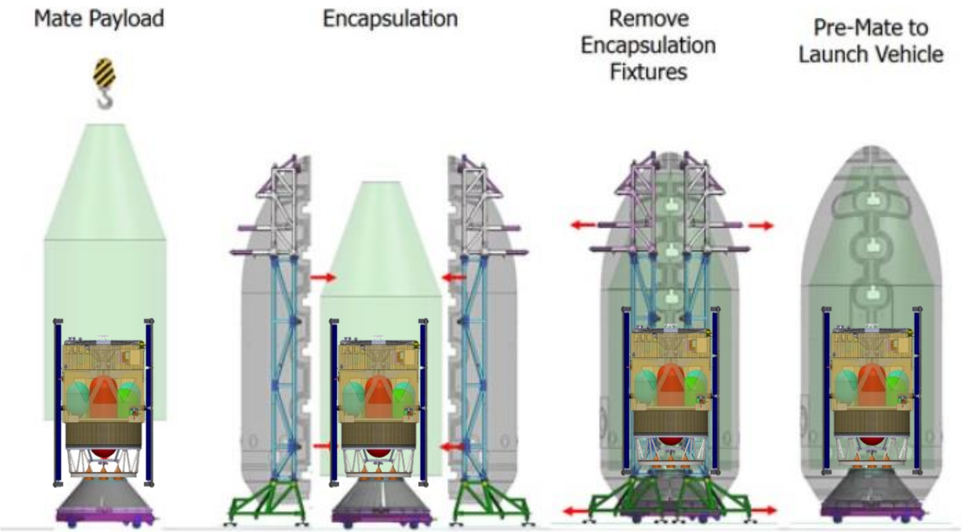


Launch Vehicle Integration

- C.O.S.M.I.C. will first undergo integration to the PAF within SpaceX's PPF

- C.O.S.M.I.C. will then be integrated by SpaceX to the fairing in the vertical orientation

- The fairing will be horizontally integrated into Falcon Heavy inside the SpaceX hangar facility.



Source: SpaceX

- Once integrated onto FH, the hangar facility HVAC system is connected via a fairing air conditioning duct. Also, electrical ground support equipment is reconnected, and electrical interfaces are verified.

Roll Out to Launch Pad

Image: SpaceX

- On the same day of launch, the flight vehicle will be rolled out to launch pad LC-39A, then erected into the vertical orientation.
- During rollout, HVAC & Electrical will be disconnected until the vehicle is at the pad.
  - However, if needed a mobile electrical unit can be used.
- Electrical Ground Support Equipment (EGSE) and HVAC connections are restored once the vehicle reaches the pad (before erection)

Pre-Launch Risks – Launch Window

Given that…	There is a possibility of…	Adversely impacting…	Which can result in…	Likelihood (out of 5)	Severity (out of 5)	Sub team
the launch window is subject to weather or technical issues	a delay in the launch date or time	the entire mission	a delay in the readiness date	2	1 to 5	Launch Services

	1	2	3	4	5
Likelihood	0-20%	20-40%	40-60%	60-80%	80-100%
Severity	Minimal impact on schedule	Delay on some tasks not impacting overall schedule	Delay on some tasks minimally impacting schedule	Major slip in overall schedule	Critical milestones cannot be met

Mission Budget

Speaker: Andres MA

Budget Breakdown

System Life Cycle Phase	Direct Labor ($ in million)	Benefits & Indirect Cost ($ in million)	Material/Cost ($ in million)	Total Cost per Phase ($ in million)
Pre- Phase A to B: Senior Design	0	0	0	0
Phase C: Final Design & Fabrication	41	82	TBD	123
Phase D: System Assembly, Integration & Test, Launch & Checkout	61	122	313	496
Phase E: Operations & Sustainment	131	262	0	393
Phase F: Closeout	0.24	0.48	TBD	0.72
Total Cost per Category	233.24	466.48	313	**1,012.72**
			Profit (7%):	**70.89**
			Total Budget:	**1,084**

Assumptions

- **Direct Labor Calculation:** Used salary databases to estimate average wage for the respective position in California
  - For aerospace engineers in CA, the average pay was $60 per hour
- **Benefits & Indirect costs Calculation:** ~2 x Direct Labor
- **Material:** Due to time restraints, the team focused on large material costs such as the launch vehicle and facility cost.
- **Formula for Budget Cost:** Direct Labor + ~2 x Direct Labor + Materials + 7% Profit

- **As a result, we can expect COSMIC's cost to increase as the WBS & Project schedule are refined and material costs are added.**

Summary

Speaker: Keilan R

Design and Success Summary

System Summary

- Two Flight Systems
- Falcon Heavy Expendable chosen as the Launch Vehicle for both Flight Systems
- Ground Station which interfaces with the Deep Space Network to communicate with the Flight System

Mission Summary

- Purpose of the mission is to collect science data from an Interstellar Object
- Both spacecraft in pre-positioned orbits (Phase 3) for up to 20 years completing secondary objectives and waiting for a suitable ISO
- One spacecraft will fly by an ISO (Phase 5) to complete primary objectives and ISO associated secondary objectives [40% chance both spacecraft could fly to different ISOs]

Success
The flow down of verifiable requirements is complete and proper, or, if not, an adequate plan exists for timely resolution of open items. Requirements are traceable to parent technical requirements and to mission goals and objectives
Preliminary analysis of the primary subsystems has been completed and summarized, highlighting performance and design margin challenges
TBD and TBR items are clearly identified with acceptable plans and schedule for their disposition
The preliminary design is expected to meet the requirements at an acceptable level of risk

ISO Science Requirements Summary

ID: MP = Mission Primary, MS = Mission Secondary

ID	Traceability	Requirement	Completion*	Discussion
MP3	Solicitation A.2	The mission shall acquire visible imagery of 50% of the object's illuminated surface with a resolution of at least 5.0 meters per pixel.	Yes	Slide 107
MP4	Solicitation A.2	The mission shall acquire infrared imagery of 50% of the object's visible surface with a resolution of at least 10.0 meters per pixel.	Yes	Slide 107
MP5	Solicitation A.3, Solicitation B.1	The mission shall model 50% of the object's shape within +/- 10 meters using active measurement.	Some cases	Slide 109-111, Slide 123
MP6	Solicitation A.3	The mission shall determine the object's mean dimension within +/- 10 meters.	Yes	Slide 108
MP7	Solicitation A.4	The mission shall determine the object's spin axis within +/- 1.0 degree.	Some cases	Slide 104-105
MP8	Solicitation A.4, Customer Conversation 5/7/2021	The mission shall determine the object's rotation rate within 1%.	Yes	Slide 103
MS1	Solicitation B.1	The mission shall measure the object's dielectric constant within +/- **TBD**.	Expected	Slide 123

CAL POLY

Speaker: Keilan R

*Completion for ISO periods of 0.5 to 50 hours, based on similarly sized asteroids

Prepositioned Science Requirements Summary

ID: MS = Mission Secondary

ID	Traceability	Requirement	Completion	Discussion
MS2	Solicitation B.3, Solicitation B.5	The mission shall have a sky coverage of 0.15 % in the pre-positioned orbit.	Yes	Slides 45-66
MS3	Solicitation B.3	The mission shall observe heliocentric orbiting bodies.	Yes	Slides 45-66
MS4	Solicitation B.5	The mission shall acquire exoplanet photometry of a minimum of 1 star system.	Yes	Slides 45-66

Additional Mission Requirements Summary

ID: MP = Mission Primary

ID	Traceability	Requirement	Completion	Discussion
MP1	Solicitation	The mission shall be ready to react to an ISO no later than 12/31/2030.	Yes	150-151
MP2	Solicitation	The mission shall have an 80% likelihood of reaching at least 1 object with the parameters specified in Table 1.0 within 20 years of its readiness date.	Expected	11-12, 46
MP9	Customer Conversation 1/8/2021	The mission shall return data to the customer no later than 9 months post collection.	Yes	136-144
MP10	Customer Conversation 1/8/2021	The mission shall communicate with the deep space network.	Yes	20, 66, 136-144

Thank you!
End of Day 2

Support Slides

Project Management


David Schreiber

Keilan Ramirez


Luke Kutz


Natalia Cieply

Andres Mendoza Arteaga

System Engineering

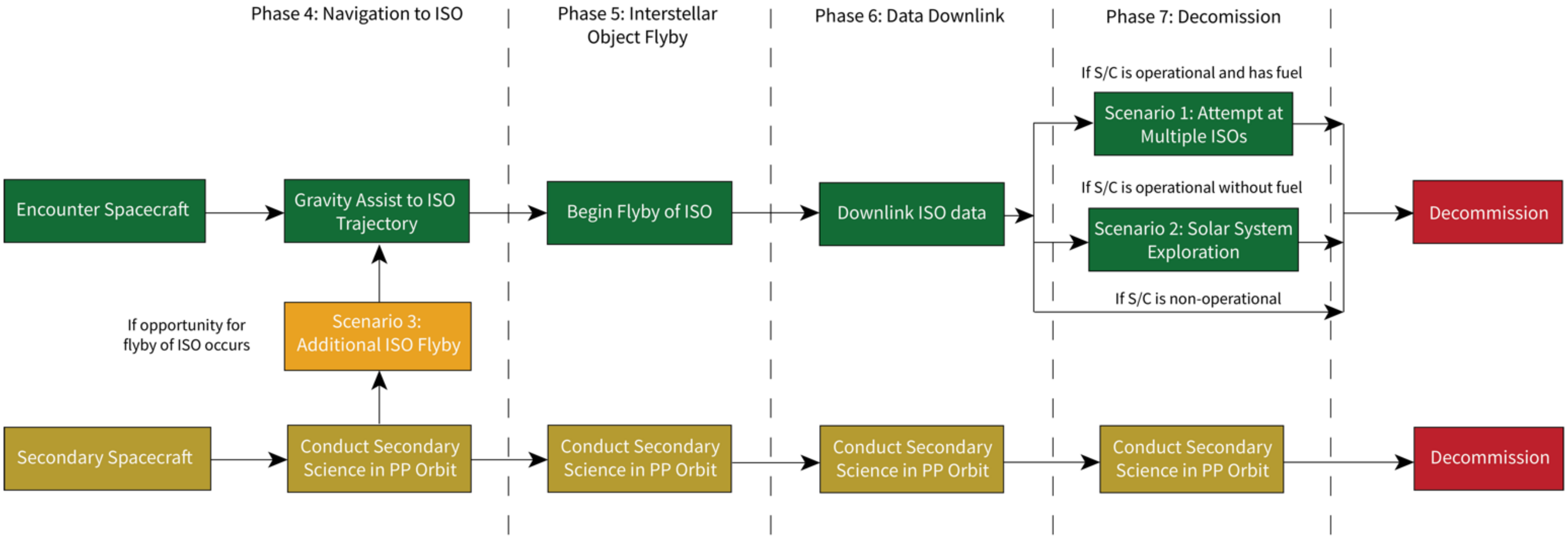
Subsystem Breakdown

Back to Mass Breakdown

Subsystem	Components
Payload	SAR, Imager System
Propulsion	R-4D-15 HiPAT™ 445 N (4 thrusters), Propellant Tanks, Plumbing & valves, Propellant, Pressurant
Power	ROSA Panels (x2) , Battery, Solar Drive Assemblies (x2), Solar Array Drive Assembly Electronics, PCE (Power Control Electronics) - Power Box, Wiring Harness
Comms	High Gain Antenna, Medium Gain Antenna (x2), Low Gain Antenna (x2), Deep Space Transponder (x2), Ka-Band Travelling Wave Tube (x2), X-band Travelling Wave Tube (x2), Ultra Stable Oscillator, Diplexer (x2), Cabling/Waveguide, Switches, Hybrid Coupler
Thermal	Radiators, Chem Prop heaters, Heat Pipes, Thermal Switches, Thermal Isolators, MLI Blankets, Louvers, Pump Module, Louvers, thermoelectric coolers
GNC	APIC (OPNAV Camera)
ADCS	CMGs (x4) (Blue Canyon CMG-12) - baseline CMGs, A-STR (Star Tracker), CSS (Course Sun Sensor x4), IMU (Polaris x2)
Structures	Spacecraft Attachment Fitting, Hook Integration Mount, Crane Integration Mount, Solar Array Drive Assemblies (x2), MROSA Deployment Structure (x2), Lower Structure Walls, Upper Structure Walls, Routing Platforms, Electronics Plate, Electronics Floor, Imager Mounts, Center Column, Upper and Lower Tank Plate, Thruster Structure, Brackets, Launch Support Structure
C&DH	Combined Enclosure

Note: sizing of the propellant includes a 10% margin and so it doesn't include a specific MGA percentage

Extended Mission Scenarios



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Risk Statement

A concise description of an individual risk that can be understood and acted upon

“Given that [CONDITION] there is a possibility of [DEPARTURE] adversely impacting [ASSET], which can result in [CONSEQUENCE] ”

- Condition: a single phrase that describes the current key fact-based situation or environment that is causing concern, doubt, anxiety, or uneasiness
- Departure: describes a possible change from the (agency, program, project, or activity) baseline project plan. It is an undesired event that is made credible or more likely as a result of the CONDITION
- Asset: an element of the organizational unit portfolio (OUP) (analogous to a WBS). It represents the primary source that is affected by the individual risk.
- Consequence: a single phrase that describes the foreseeable, credible negative impact(s) on the organizational unit’s ability to meet its performance requirements

Risk Likelihood and Severity

Severity \ Likelihood	1	2	3	4	5
5	5	10	15	20	25
4	4	8	12	16	20
3	3	6	9	12	15
2	2	4	6	8	10
1	1	2	3	4	5

Severity Scale

- Impact on performance
  1. Minimal impact on goals
  2. Minor impact on goals
  3. Unable to achieve a particular goal
  4. Unable to achieve multiple goals
  5. Unable to achieve the overall goal
- Impact on schedule
  1. Minimal impact on schedule
  2. Delay on some tasks not impacting overall schedule
  3. Delay on some tasks minimally impacting overall schedule
  4. Major slip in overall schedule
  5. Critical milestones cannot be met

Likelihood Scale

1. Not likely (under 20% probability of happening)
2. Not very likely (between 20% and 40% probability of happening)
3. Likely (between 40% and 60% probability of happening)
4. Highly likely (between 60% and 80% probability of happening)
5. Near certainty (over 80% probability of happening)

Concept of Operations

Austin Iannitti

Sean Thompson

Requirements

Bailey Garrett

Sydney Retzlaff

Back to main!

Table 1: ISO Characteristics

ISO Characteristic	Parameters
Inbound Distance	3AU
Detection	Uniform Distribution
Eccentricity	0.99-3.50
Perihelion	0.3 AU - 2 AU
Inclination	0-180 deg
Argument of Perihelion	0.05 or higher
Geometric Albedo	1.0-1.5 km
Mean Dimension	+/- 10 m

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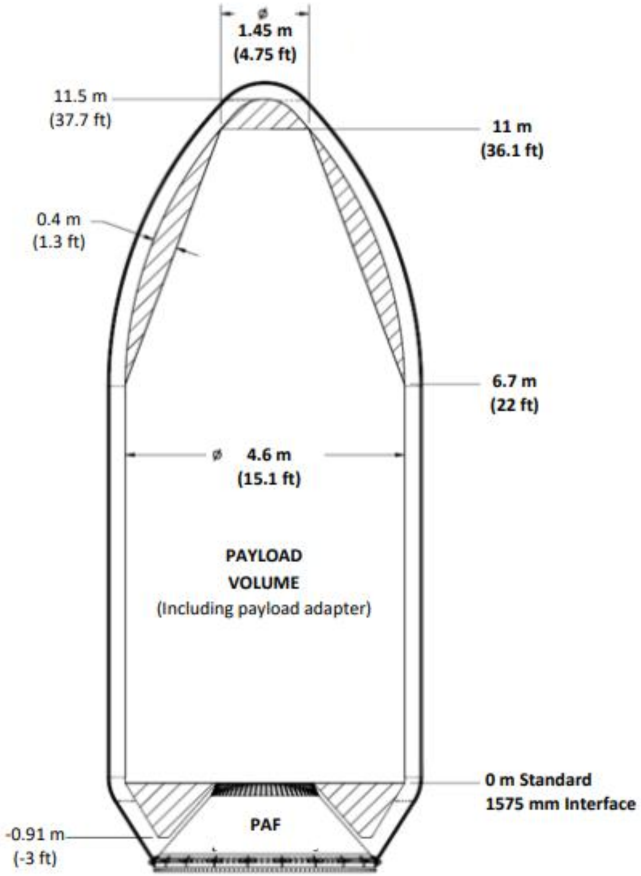
Figure 1: Falcon Heavy Launch Vehicle Fairing



Source: Falcon Heavy User Guide Figure 5-1
https://www.spacex.com/media/falcon_users_guide_042020.pdf

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Table 7: Fault Mode Causes

Fault Mode Causes
C&DH does not receive confirmation that a mission critical command was performed.
C&DH does not receive conformation on any attitude commands.
C&DH does not receive conformation on any positioning commands.
C&DH reboots for any unexpected reason.
For power cycle issues.
If a health sensor is outside of survival range.
Following separation from the launch vehicle.

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Figure 2: Mass of Payload vs. C3 Values

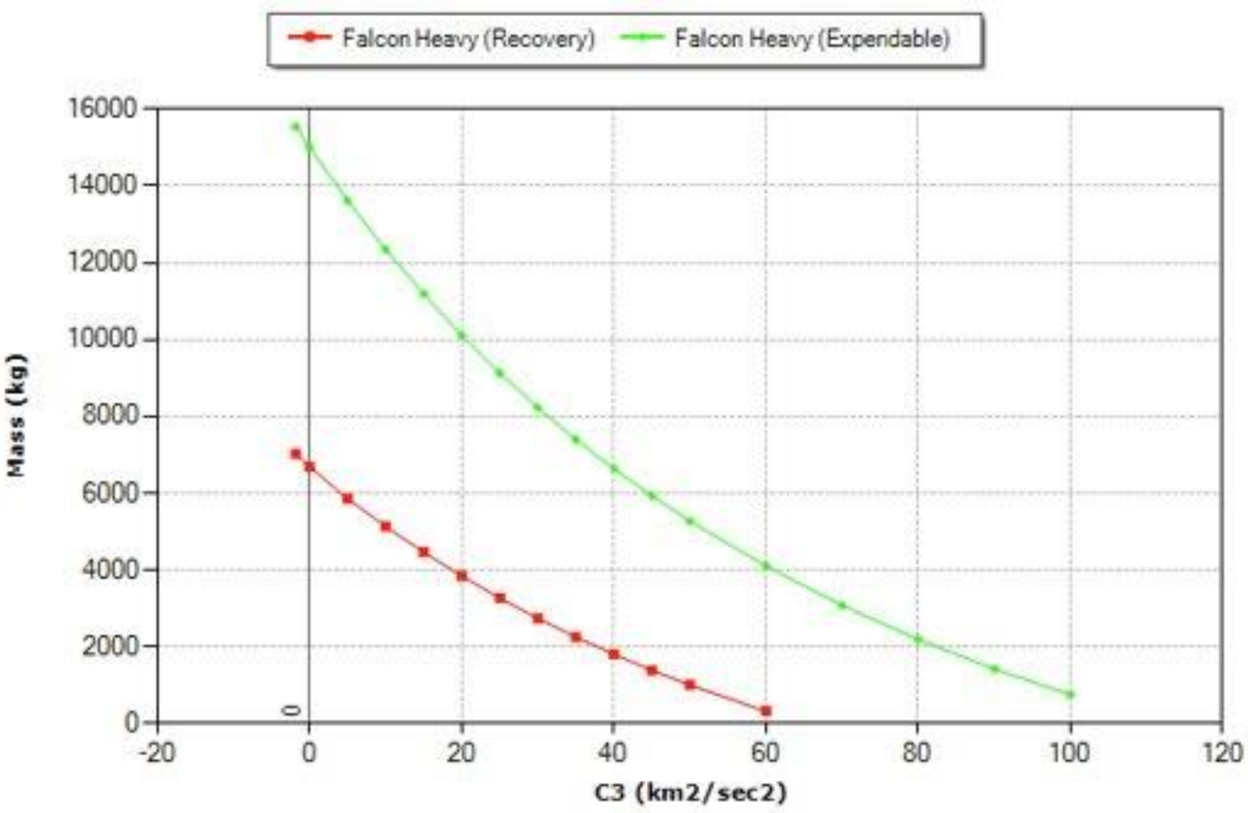

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Table 6.1: Initial Emplacement Orbit Definition

Initial Emplacement Orbital Elements by Spacecraft					
	Semi-Major Axis (AU)	**Eccentricity**	**Inclination (°)**	**Right Ascension of the Ascending Node (°)**	**Argument of Periapsis (°)**
Spacecraft 1	1	0.0167	2.5	103	0
Spacecraft 2	1	0.0167	2.5	193	270

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Table 6.2: Pre-Positioned Orbit Definition

Prepositioned Orbital Elements by Spacecraft					
	Semi-Major Axis (AU)	Eccentricity	Inclination (°)	Right Ascension of the Ascending Node (°)	Argument of Periapsis (°)
Spacecraft 1	1	0.0167	21	103	0
Spacecraft 2	1	0.0167	21	193	270

Orbits

Christian Fuller

Ryan Meisberger

Jack Kelly

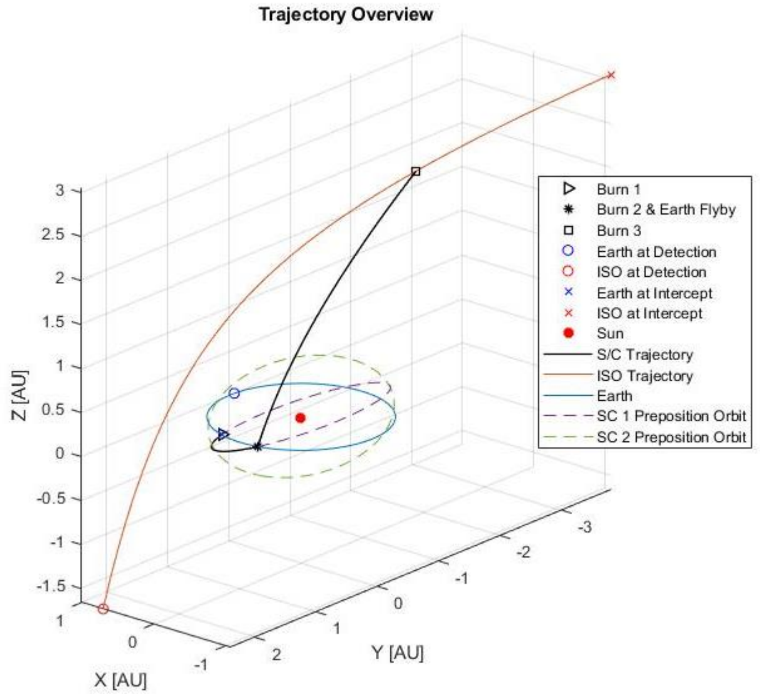
Jordan Watt

Evan Agarwal

Dominick Bologna

Orbit Case Studies: ISO A

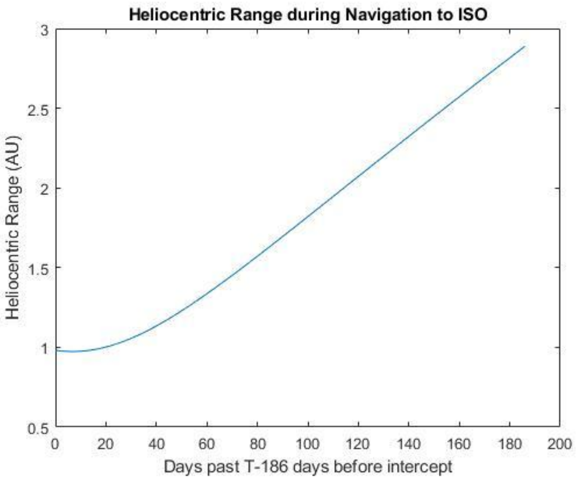
- ISO COEs
  - Eccentricity: 1.65
  - Radius of Perihelion: 1.50 AU
  - Inclination: 93.94 deg
  - RAAN: 71.11 deg
  - Argument of Perihelion: 45.03 deg
- Delta-V
  - Total DV: 4.494 km/s
  - Burn 1: 0.017 km/s
  - Burn 2: 0.018 km/s
  - Burn 3: 4.459 km/s
- Encounter
  - Relative Velocity to ISO: 8.541 km/s
- Flyby
  - Earth Flyby Altitude: 14848 km

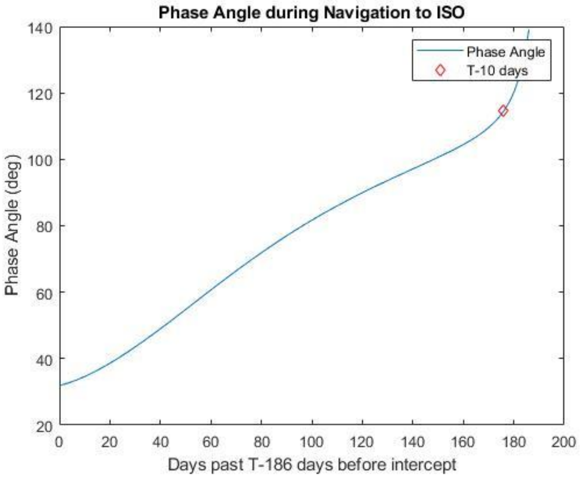


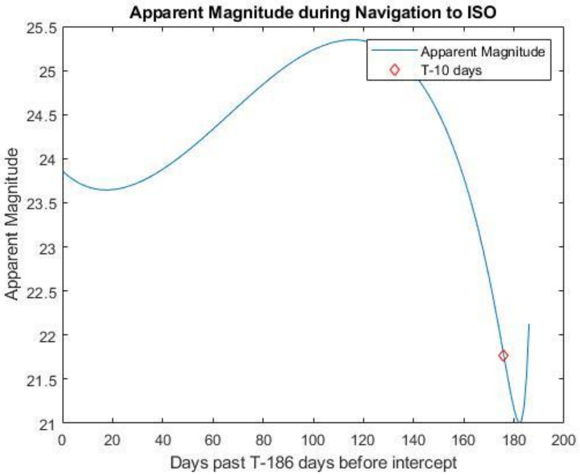
Orbit Case Studies: ISO A

Transfer Characteristics
- Day of ISO Detection: Sept 19, 2044
- Transfer Duration: 186 days

10 Days Before Intercept
- Apparent Magnitude: 21.77
- Phase Angle: 114.48 deg

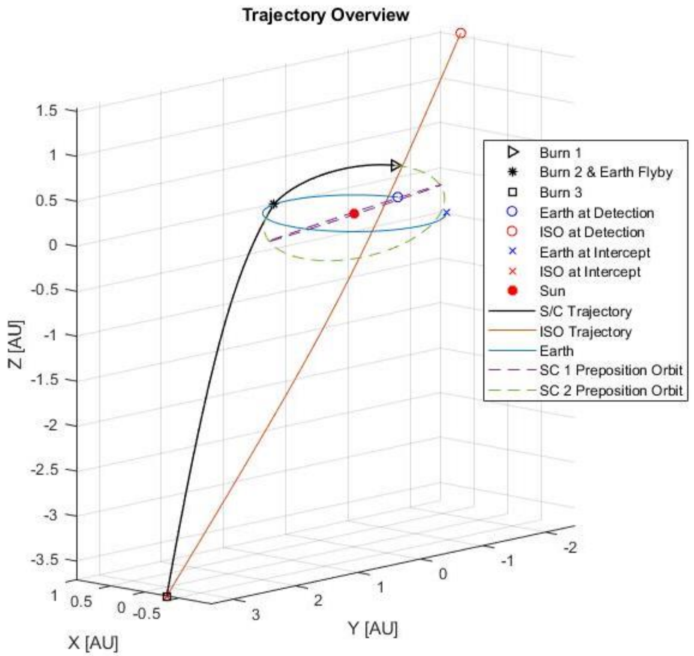






Orbit Case Studies: ISO B

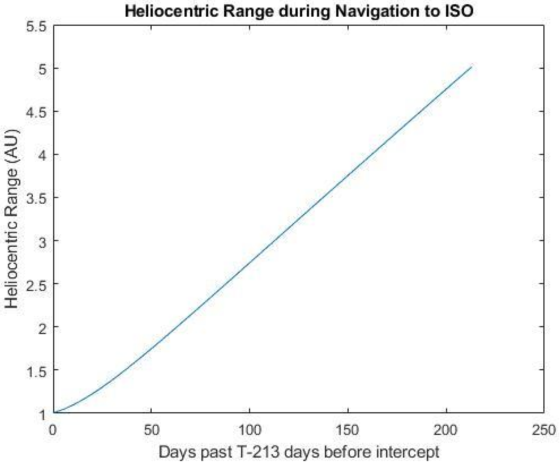
- ISO COEs
  - Eccentricity: 3.07
  - Radius of Perihelion: 1.38 AU
  - Inclination: 71.26 deg
  - RAAN: 117.47 deg
  - Argument of Perihelion: 220.86 deg
- Delta-V
  - Total DV: 4.387 km/s
  - Burn 1: 0.073 km/s
  - Burn 2: 0.073 km/s
  - Burn 3: 4.241 km/s
- Encounter
  - Relative Velocity to ISO: 10.239 km/s
- Flyby
  - Earth Flyby Altitude: 7353 km

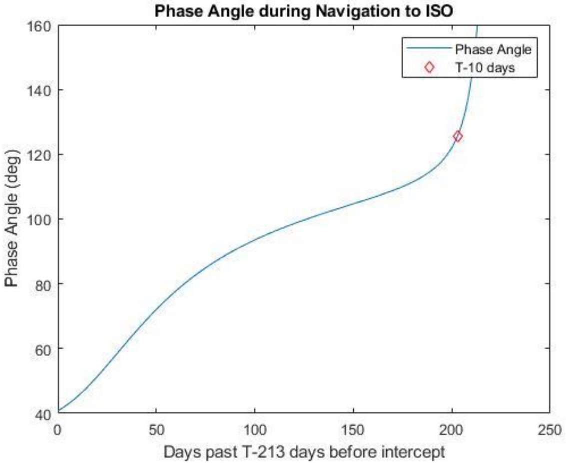


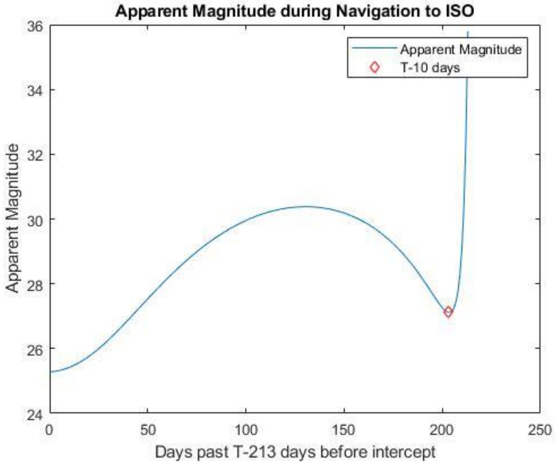
Orbit Case Studies: ISO B

Transfer Characteristics
- Day of ISO Detection: Dec 29, 2032
- Transfer Duration: 213 days

10 Days Before Intercept
- Apparent Magnitude: 27.12
- Phase Angle: 125.71 deg

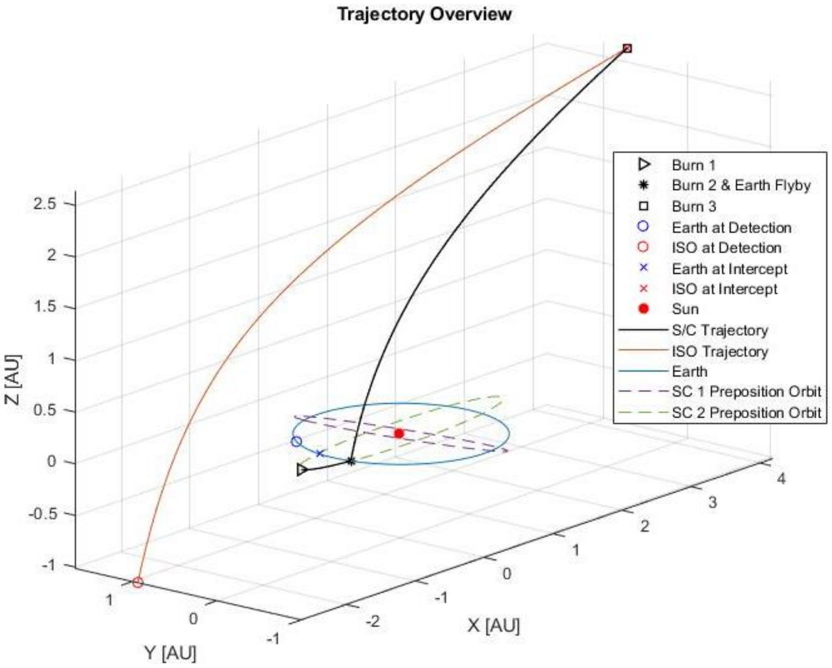






Orbit Case Studies: ISO C

- ISO COEs
  - Eccentricity: 2.18
  - Radius of Perihelion: 1.81 AU
  - Inclination: 144.04 deg
  - RAAN: 131.07 deg
  - Argument of Perihelion: 30.20 deg
- Delta-V
  - Total DV: 4.108 km/s
  - Burn 1: 0.310 km/s
  - Burn 2: 0.312 km/s
  - Burn 3: 3.486 km/s
- Encounter
  - Relative Velocity to ISO: 12.710 km/s
- Flyby
  - Earth Flyby Altitude: 6172 km



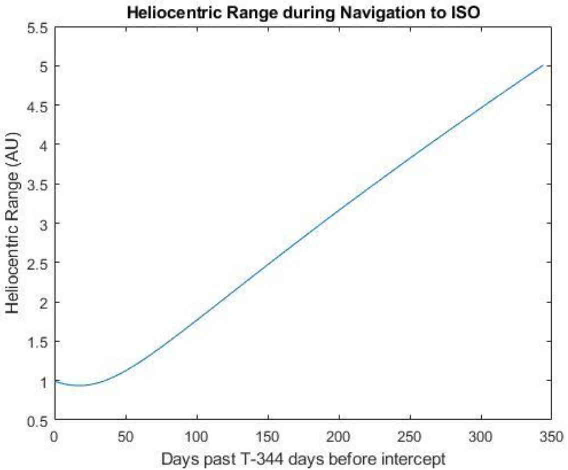
Orbit Case Studies: ISO C

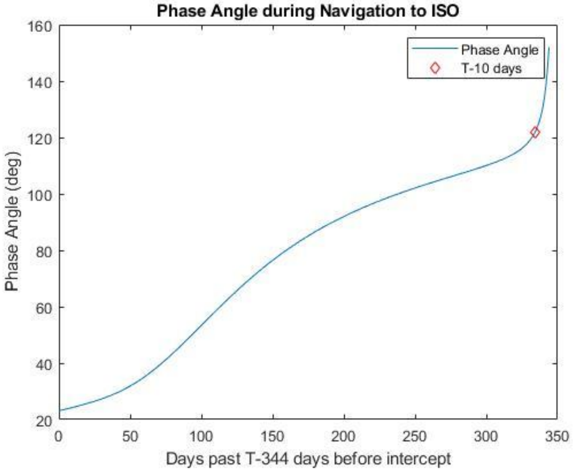
Transfer Characteristics

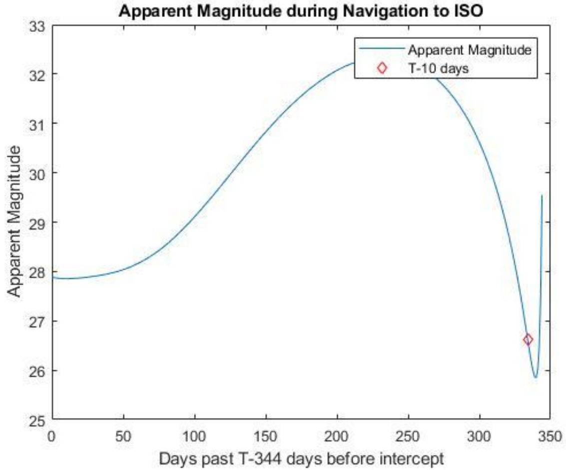
- Day of ISO Detection: Feb 15, 2030
- Transfer Duration: 344 days

10 Days Before Intercept

- Apparent Magnitude: 26.62
- Phase Angle: 121.65 deg







Orbital Simulation Assumptions

Assumption	Reasoning
Only 1 chance at ISO detection per day	Daily checks is sufficiently fine resolution given mission length (up to 20 years). Evaluating more frequently would strain computational resources for diminishing returns.
Probability of ISO detection averages to 1/year	Solicitation states 1/year ISO detection rate
ISO detected at 3 AU heliocentric inbound	Solicitation states customer will identify ISO at 3 AU
Spacecraft are in constant inclination orbits	The final orbital details are to be determined and may involve small changes (<2°) to inclination due to high-altitude Earth flybys. The variation is expected to be minor, but this remains an open action moving forward.
Perturbations are neglected	Perturbations from Solar Radiation Pressure and N-Body effects are expected to be minor. As such they are neglected in the simulation for the sake of processing speed
Impulsive burns	Given the time scales and chemical propulsion system, impulsive burns are suitable for modeling at his level.
Earth modeled as a point mass	The distances involved sufficiently dwarf Earth's Sphere of Influence and the gravity assist functions operate independently of this assumption. The difference is expected to be minor, but this remains an open action moving forward

Orbital Simulation Inputs

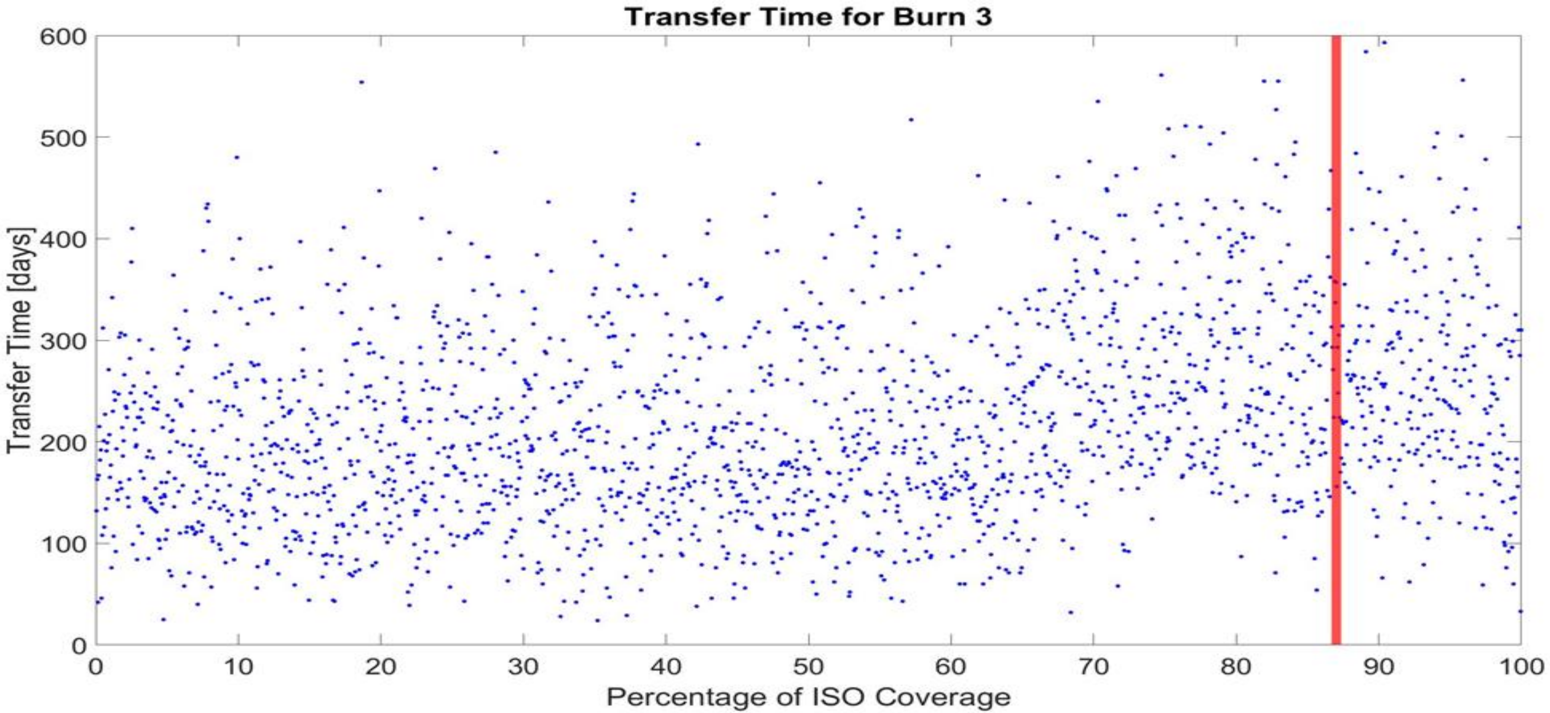
Input Categories	Values	Reasoning
Mission Start Date	01/01/2031	Solicitation directs mission readiness by 12/31/2030
Mission Length	7305 days	Solicitation directs mission completion by 12/31/2050
Propagation Time Step	1 day	For time scales involved, 1 day is sufficiently fine resolution
Transfer Evaluation Time Step	1 day	Generates more data for evaluation
Min Spacecraft Distance from Sun	0.5 AU	Preliminary value set by structures
Max Heliocentric Encounter Distance	5 AU	Set by GNC, driven by restrictions in Optical Navigation
Max Relative Speed at Encounter	13 km/s	Set by Encounter lead, driven by Payload and ADCS needs
Min Time From ISO Detection to 1st Burn	3 days	Arbitrary, to be revisited with input from Ground Systems
Max Time From 1st Burn to Gravity Assist	91 days	Arbitrary, to be revisited in light of station-keeping needs
Min Gravity Assist Altitude	1000 km	Chosen to avoid atmospheric effects, especially drag, as well as concentrations of Low Earth Orbit satellites

Orbital Simulation Process

1. Step through each mission day and determine if an ISO is detected
    1. Once an ISO is detected, generate its orbital elements randomly in accordance with the solicitation
    2. Determine the ISO's initial state and propagate it until it reaches the outer bound set for Encounter
    3. Determine the next gravity assist opportunity from day of detection
        1. For each transfer timestep calculate the trajectory from Earth to the ISO
            1. Use the difference in velocities at the ISO to determine the final burn needed to achieve the relative encounter speed
            2. Use the required trajectory velocity at Earth as the outbound velocity of the gravity assist and determine the necessary inbound velocity required to achieve it
            3. Determine the spacecraft trajectory from its pre-positioned orbit leaving location to the gravity assist
            4. Determine burns required to enter trajectory to gravity assist and to begin gravity assist once at Earth SOI
            5. Sum DV magnitudes of all three burns as total cost of achieving the given encounter
        2. Repeat 1.3.1 until ISO is passed the outer bound set for encounter
        3. Store lowest total DV cost for the given gravity assist
    4. Repeat 1.3 for each gravity assist opportunity while the ISO is within the outer bound set for encounter
    5. Store lowest total DV cost for the given ISO
2. Repeat 1 until mission length is reached
3. Store lowest DV cost for the mission

Note: All generation, propagation, and evaluation data generated by the simulator is saved and available for post-processing and analysis.

Transfer Time



CAL POLY

Propulsion

Speaker: C. Larkin

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Tank Sizing

Fluid	Total Volume [m3]	MEOP [bar]	Temp [K]	Fluid Mass [kg]	Tank Mass [kg]
He	1.2	200	300	18	189
Hydrazine	5.8	21.3	300	5926	216
MON-3	3.7	21.3	290	5408	130

- Tanks were sized based on R. Meisberger's mass model
- 1.5x burst factor used per AFSPCMAN 91-710
- Propellant tanks:
  - 21.3 bar MEOP
    - Driven by primary prop max inlet pressure
  - 21.3 bar x 1.5 = 34.5 bar MAWP
  - 300 K Hydrazine/290 K MON-3
- Pressurant tanks:
  - 200 bar MEOP
    - Driven by need to keep tanks at 21.3 bar at burnout
  - 200 bar x 1.5 = 300 bar MAWP
  - 300 K Helium

12.3.3. Flight Hardware Metallic Pressure Vessels with Brittle Fracture or Hazardous LBB Failure Mode.

12.3.3.1. Flight Hardware Metallic Pressure Vessels with Brittle Fracture or Hazardous LBB Failure Mode Factor of Safety Requirements.

12.3.3.1.1. Safe-life design methodology based on fracture mechanics techniques shall be used to establish the appropriate design factor of safety and the associated proof factor for metallic pressure vessels that exhibit brittle fracture or hazardous LBB failure mode.

12.3.3.1.2. The loading spectra, material strengths, fracture toughness, and flaw growth rates of the parent material and weldments, test program requirements, stress levels, and the compatibility of the structural materials with the thermal and chemical environments expected in service shall be taken into consideration.

12.3.3.1.3. Nominal values of fracture toughness and flaw growth rate data corresponding to each alloy system, temper, and product form shall be used along with a life factor of 4 on specified service life in establishing the design factor of safety and the associated proof factor.

12.3.3.1.4. Unless otherwise specified, the minimum burst factor shall be 1.5.

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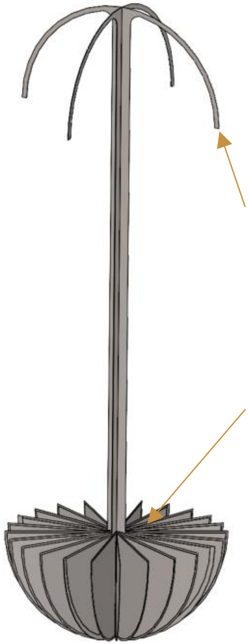
Zero-G Propellant Management

- Propellant management is achieved using a surface-tension style propellant management device (PMD)
  - Using a surface-tension style PMD over a positive-expulsion device such as a diaphragm allows multiple pressure cycles to be performed on the propellant tank without having to worry about damage the diaphragm. Piston-style devices allow multiple pressure cycles but are heavier.
  - A combination of vanes and a sponge is used to ensure that we always have propellant ready to go
  - Mass flow rates are low for the secondary propulsion system, so it is unlikely to deplete the sponge's propellant capabilities during ADCS "desaturation" burns. If larger burns are required, a settling burn may be performed to allow the sponge to resaturate with propellant.



Vanes help propellant passively commute from opposite side of tank.

Sponge helps guarantee propellant is always available to tank outlet at bottom of tank.

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Ullage Calculations

- Because the high-pressure helium system is sealed off using a pyrotechnic valve prior to Phase 4, the Helium tank isn't able to be used to pressurize the propellant tanks.
- In order to allow propellant tanks to function during orbital adjustments/stationkeeping, extra ullage space (~10%) was added to both tanks to allow the propulsion system to perform as a blowdown system prior to Phase 4.
- Ullage calculations were based upon secondary propulsion, but similar sizing was added to the oxidizer tank as well to allow high-thrust contingencies.

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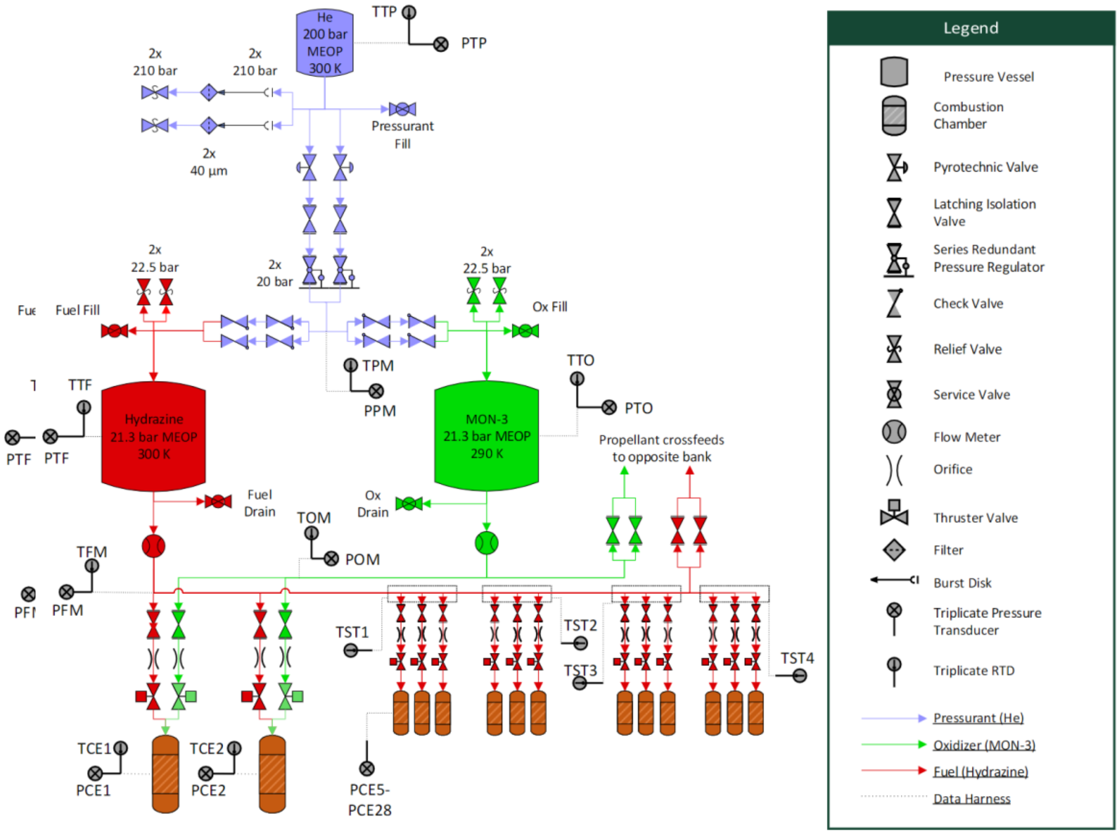
Leak Rates

- Helium leakage rates were calculated to verify pressurant mass margin over the 20+ year mission.
- Welded joints were analyzed as leaking 1e-6 scc/s GHe, threaded joints (AN) were analyzed as leaking 1e-4 scc/s Ghe
- Initial findings found leakages of upwards of 7 kg over 21 years, which exceeded out mass margin of 6 kg. Pyrotechnic valves were added to isolate the high-pressure Helium system prior to Phase 4, bringing leakage down to ~.5 kg in the same time span.
- Joints will be tested per NASA-STD-7012 to verify that they are within the leak rates above.
- The system level leakage will also be calculated by taking the sum of all joint leakages after manufacturing, and will be confirmed to provide positive margin at the end of mission.

Pressure/Temperature Data

Back to main!

- Pressure and temperature data will be taken using triplicated pressure transducers and resistance temperature detectors (RTD).
  - RTDs chosen over thermocouples for superior precision, at the cost of reduced responsiveness.
- Pressure and temperature data will be taken or the tanks, critical junctions of plumbing, and the thrusters.
  - Instead of taking temperature data on each of the secondary thrusters, temperature data will be taken on the mounting location of the thruster packs t allow better knowledge of the heat going into the system. This could alternatively be achieved by using thermopiles, at the cost of direct temperature measurement.



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Pressure/Temperature Data

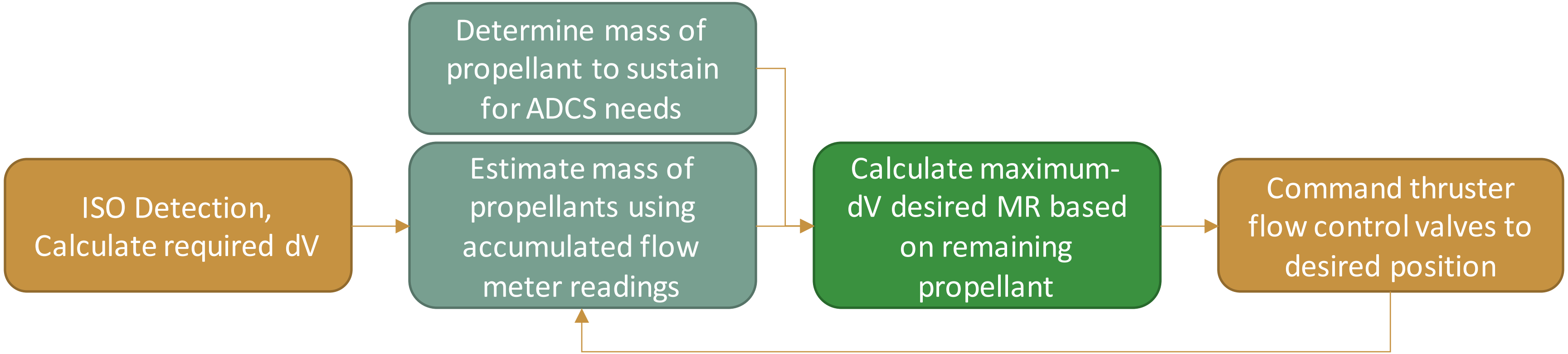
- Data Channels will be named as
  - First letter: P/T for pressure/temperature
  - Next letters: Component designator
    - TO: Tank, Oxidizer
    - TF: Tank, Fuel
    - TP: Tank, Pressurant
    - OM: Oxidizer Manifold
    - FM: Fuel Manifold
    - PM: Pressurant manifold
    - CE: Chamber, Engine
    - ST: Secondary Thruster mounting plate
  - Numbers at the end designate multiples of components to help differeniate
  - Not shown at right, will be A/B/C following reference designators for triplicate-redundant sensors.
  - EX: PCE4B : Pressure, Chamber, Engine 4, Channel B

Reference Designator	Description	Data Rate [Hz]	MEOP/T [bar, K]	Driven by
PTO	Ox tank pressure	1000	22.5	PMO
PTF	Fuel Tank Pressure	1000	22.5	PMF
PTP	Pressurant Tank Pressure	1000	205	PTO/PTF after blowdown
POM	Ox manifold pressure	1000	21.5	PCPE
PFM	Fuel manifold pressure	1000	21.5	PCPE
PPM	Pressurant Manifold Pressure	1000	21.5	PTO/PTF
PCE1	Primary Engine 1 Pc	1000	9.4	Aerojet
PCE2	Primary Engine 2 Pc	1000	9.4	Aerojet
PCE3	Primary Engine 3 Pc	1000	9.4	Aerojet
PCE4	Primary Engine 4 Pc	1000	9.4	Aerojet
PCE5	Secondary Engine 1 Pc	1000	6.9	Aerojet
PCE6	Secondary Engine 2 Pc	1000	6.9	Aerojet
PCE7	Secondary Engine 3 Pc	1000	6.9	Aerojet
PCE8	Secondary Engine 4 Pc	1000	6.9	Aerojet
PCE9	Secondary Engine 5 Pc	1000	6.9	Aerojet
PCE10	Secondary Engine 6 Pc	1000	6.9	Aerojet
PCE11	Secondary Engine 7 Pc	1000	6.9	Aerojet
PCE12	Secondary Engine 8 Pc	1000	6.9	Aerojet
PCE13	Secondary Engine 9 Pc	1000	6.9	Aerojet
PCE14	Secondary Engine 10 Pc	1000	6.9	Aerojet
TTO	Ox tank temperature	10	290	Ox Boiling
TTF	Fuel tank temperature	10	300	~Ox boiling
TTP	Pressurant tank temperature	10	300	~Ox boiling
TOM	Ox manifold temperature	10	290	~Ox boiling
TFM	Fuel manifold temperature	10	300	~Ox boiling
TPM				
TCE1	Primary engine 1 Tc	10	3000	PE Tc
TCE2	Primary engine 2 Tc	10	3000	PE Tc
TCE3	Primary engine 3 Tc	10	3000	PE Tc
TCE4	Primary engine 4 Tc	10	3000	PE Tc
TST1	Thruster Plate 1 Temp	10	400	SE Flange temp
TST2	Thruster Plate 2 Temp	10	400	SE Flange temp
TST3	Thruster Plate 3 Temp	10	400	SE Flange temp
TST4	Thruster Plate 4 Temp	10	400	SE Flange temp

Propellant Utilization (PU) Algorithm

Back to main!

A propellant utilization (PU) algorithm will be used to maximize the dV while the primary propulsion system is active. This will allow the vehicle to burn at lower or higher mixture ratios in order to guarantee that the spacecraft has a minimal dry mass. Thruster flow control valves on the primary propulsion engines can skew the MR from 0.70 - 1.33 and will be used to achieve bulk MR corrections. The PU algorithm has not been explicitly written, but a general control scheme can be seen below.



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Propulsion Team Risks

- Given that the propellant lines freeze there is a possibility of the necessity of using resistive heaters to keep the propellant lines with acceptable temperatures adversely impacting the propulsion plumbing system, which can result in the spacecraft leaking propellant.

  Likelihood: 1
  Severity: 5

- Given that the gimbal mounting fails or there is degradation of the mounting component over time due to the environment there is a possibility of the components that utilize these mounts not deploying, adversely impacting the secondary thruster system and structural gimbal mounts, which can result in a reduction in attitude control effectiveness, and increased difficulty or destruction, and a reduction of the translational speed control.

  Likelihood: 1
  Severity: 3 or 4

- Given that the secondary thrusters go out there is a possibility of the propulsion of the spacecraft not being sufficient to complete the mission adversely impacting the secondary thruster systems and propulsion plumbing system, which can result in a reduction of attitude control effectiveness and a reduction in translational motion control.

  Likelihood: 2
  Severity: 3 or 4

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Secondary Propulsion Thrusters

Link

Thrust: 4 N each

Isp: 219-229 s

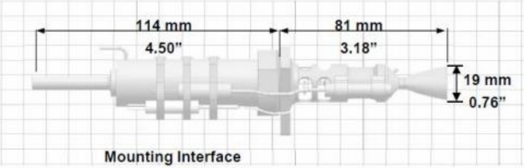
Qty: 12 pairs, total quantity 24

Total propellant consumed: (760 kg Hydrazine)

Monopropellant chosen to provide simple secondary system. Thrust level semi-arbitrary based on ADCS unknowns, upsizing to higher thrust levels is still an option.

MR-111G 4N (1.0 lbf) Rocket Engine Assembly

AEROJET ROCKETDYNE



Design Characteristics

- Propellant........ Hydrazine
- Catalyst........ S-405
- Thrust/Steady State........ 4.9 - 1.8 N (1.1 - 0.4 lbf)
- Feed Pressure........ 24.1 - 6.7 bar (350 - 100 psia)
- Chamber Pressure........ 10.0 - 3.7 bar (145 - 54 psia)
- Expansion Ratio........ 74:1
- Flow Rate........ 2.0 - 0.77 g/sec (0.0044 - 0.0017 lbm/sec)
- Valve........ Dual Seat
- Valve Power........ 8.25 Watts Max @ 28 Vdc & 21°C
- Valve Heater Power... 1.54 Watts Max @ 28 Vdc & 21°C
- Cat. Bed Heater Power....6.32 Watts Max @ 28 Vdc & 21°C
- Mass........ 0.37 kg (0.82 lbm)
  - Engine........ 0.11 kg (0.24 lbm)
  - Valve........ 0.20 kg (0.44 lbm)
  - Heaters........0.065 kg (0.14 lbm)

Performance

- Specific Impulse........ 229 - 219 sec (lbf-sec/lbm)
- Total Impulse........ 262,000 N-sec (59,000 lbf-sec)
- Total Pulses........ 420,000
- Min Impulse Bit........ 0.076 N-sec @ 15.5 bar & 20 ms ON
- ....(0.017 lbf-sec @ 225 psia & 20 ms ON)
- Steady State Firing..... 10,000 sec demonstrated - Single Firing

Status

- Flight Proven
- Currently in Production

Reference

- AIAA-2012-3817

11411 139th Place NE • Redmond, WA 98052
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High Pressure Regulator

Link

Capable of .003 lbm/sec Ghe =1.35e-3 kg/s Ghe

System requirement: 8.47e-4 kgs GHe

Helium Flow Rate during Primary Firing (assuming constant nomiunal thrust, ideal Isp)

Force	890	N			
Isp	329	s			
mdot prop	0.275756083	kg/s			
mdot (Ox/Fu)	0.137878041	0.137878041	kg/s		
Volumetric Flow Rate (Ox/Fu)	9.50883E-05	0.000135042	m3/s		
Mass Flow Rate (He)	3.43E-04	5.04E-04	kg/s		
	7.55E-04	2.29E-04	lbm/s		
Total Mass Flow Rate (He)		8.47E-04	kg/s		

VACCO SPACE PRODUCTS

HELIUM REGULATOR / SYSTEM FILTER
V1E10776-01

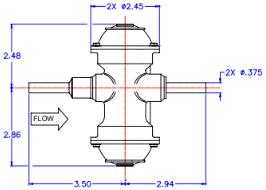
DESCRIPTION

VACCO Industries offers a compact, pre-integrated system filter and Helium regulator to precisely control the pressure to propellant feed systems. The series-redundant regulator is qualified for use in spacecraft and missile dynamic environments. Dynamic stability and contamination protection has been demonstrated under all operating conditions, including rapid application of inlet pressure from a pyrotechnic valve.

The series-redundant pressure regulators are completely protected from external contamination by filters in the inlet, outlet, and sensing ports. For added protection; there is a separate inlet filter for the redundant regulator.

FEATURES

- Combination System Filter and Regulator
- Series Redundant CRES Regulators
- Regulation Accuracy within ± 2.5%
- Inlet Pressure Range from 4500 to 400 psig
- Internal Leakage is less than 20 sccm
- Rapid Transient Recovery
- 25 Micron (absolute) Filtration
- Etched Disc Titanium Filters
- Clog-Proof Titanium Sensing Restrictor



OPERATING PARAMETERS

Operating Pressure Range	4500 to 400 psig	External Leakage	1×10^{-6} scch GH
Proof Pressure	6750 psig	Mass	< 2.30 lbm
Burst Pressure	11,250 psig	Operating Temperature	0° to 100° F
Flow	0.003 lbm/sec GHe	Back Pressure Relief	350 psid
Internal Leakage	< 20 sccm GHe	Filter Rating	25 micron

Performance characteristics are based upon customer requirements, as such, are not representative of component capabilities or limitations

Low Pressure Relief Valve

Link

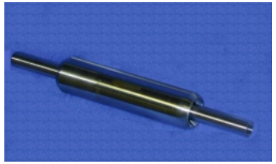
VACCO SPACE PRODUCTS

LOW PRESSURE RELIEF VALVE
V1D10879-01

DESCRIPTION

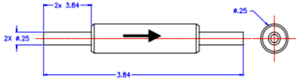
VACCO Aerospace Products produces a relief valve for designed exclusively to satisfy the needs of the space industry.

A unique advantage of VACCO's relief valve is that the seat/seal configuration is flight qualified and has demonstrated successful flight heritage for over 10 years. The relief valve incorporates a teflon seat design used for low leakage over a wide range of temperatures.



FEATURES

- Seat/Seal Configuration Flight Qualified
- All Welded External Construction
- Relief Pressure: 335 psid max.
- Reset Pressure: 280 psid min.
- Weight: .75 lbm (340 grams)
- Machined Body to Assure Structural Rigidity
- Optimized for High Flow Stable Operation



OPERATING PARAMETERS

Operating Pressure Range	355 psid*
Proof Pressure	650 psig
Burst Pressure	1,300 psig
Flow	100 scfm GHe**
Pressure	335 psid max.
Reseat Pressure	280 psid min.
External Leakage	1 X 10^{-6} sccs GHe***
Internal Leakage	1 X 10^{-3} sccs GHe****
Operating Temperature	-170° to +45° C
Non-operating Temperature	-40° to +45° C
Weight	340 grams
Shock	2000 G's
Life	200 cycles

*Note: Based on Actual Test Data
**Note: Based on Actual Test Data, 100 scfm @ 355 psid
***Note: External Leakage measured at 270 ± 5 psia for 5 minutes
****Note: Internal Leakage measured from 0-280 psia

Performance characteristics are based upon customer requirements, as such, are not representative of component capabilities or limitations

Back to main!

High Pressure Latching Iso Valve

Link

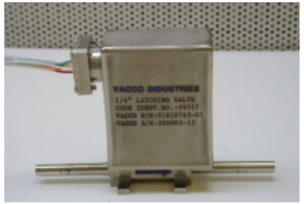
VACCO
SPACE PRODUCTS

¼" HIGH PRESSURE LATCH VALVE
V1E10763-01

DESCRIPTION

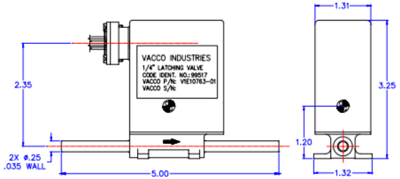
VACCO Industries maintains a product line of titanium latching valves designed to meet industry's demand for high reliability, tight leakage, and quick response capabilities.

The VACCO 5000 psi ¼" latch valve utilizes proven torque tube technology. The all titanium construction helps reduce weight while bolstering reliability; qualified to a minimum 13,000 life cycles while maintaining performance. The valve is fully qualified and is scheduled for flight in 2002.



FEATURES

- 5,000 psig Operating Pressure
- 13,000 Operating Cycles
- Ti Body with CRES Interface
- Low Weight – 0.75 LB (340 Grams)
- Integral Microswitch
- 40 Micron Absolute Etched Disc Inlet Filter



OPERATING PARAMETERS

Operating Pressure .... 5000 psig
Proof Pressure .... 7500 psig
Burst Pressure .... 12,500 psig
Flow/Pressure Drop .... <10 psid @ 1.06
Leakage:
Internal .... < 5.0 scch GHe @ 5000 psid
External .... $1x10^{-6}$ sccs GHe @ 5000 psid
Back Pressure Relief .... 1500 ± 500 psid

Cycle Life .... > 13,000 Cycle
Operating Temp. .... 14 - 140°F)
Response Time: ....
Opening: .... 20 msec @ 29 VDC @ 5000 psid
Closing: .... 20 msec @ 29 VDC, 5000 psid
Opening Voltage .... 24 - 32 VDC.
Power Consumption .. 20 W max @ 29 VDC and 76°F
Dielectric Strength ...... 500 VAC RMS @ 60 Hz–1 min

Back to main!

Low Pressure Latching Iso Valve

Link

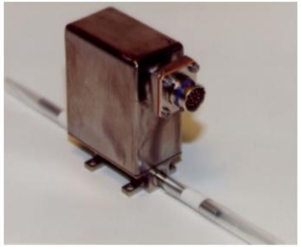
VACCO SPACE PRODUCTS

LOW PRESSURE ¼" LATCH VALVE
V1E10405-04

DESCRIPTION

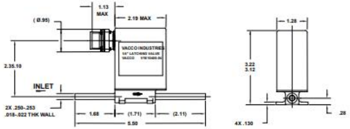
VACCO Industries maintains a product line of titanium latching valves designed to meet industry's demand for high reliability, tight leakage, and quick response capabilities.

The 1/4" low-pressure latch valve utilizes VACCO's proven torque tube technology. The all titanium construction helps reduce weight while bolstering reliability; qualified to a minimum 20,000 life cycles without deteriorating performance. The valve is fully flight-qualified and holds extensive heritage.



FEATURES

- 400 psi Operating Pressure
- Flight Qualified
- All Titanium Construction
- Low Weight – 340g/0.75 lbm
- Integral Microswitch
- Bi-Directional Flow
- 40µ Absolute Inlet Filter



OPERATING PARAMETERS

- Operating Pressure Range .... 400 psig
- Proof Pressure .... 900 psig
- Burst Pressure .... 2400 psig
- Flow .... 0.025 lbm/sec H_2O
- Pressure Drop .... <15 psid
- Internal Leakage .... <5.0 scch GHe
- External Leakage .... <1 X 10^{-6} sccs GHe
- Weight .... 0.75 lbm (340 grams)

- Operating Temperature .... +30° to 150° F
- Response Time
  - Opening .... 50 msec @ 20 VDC
  - Closing .... 50 msec @ 20 VDC
- Back Pressure Relief .... 350 psid
- Inlet Filter Rating .... 40 microns absolute
- Operating Voltage .... 20 – 50 VDC
- Dielectric Strength .... 500 VAC RMS @ 60 Hz – 1 min

[Back to main!]

Low Pressure Check Valve

Link

Would need to get check valves manufactured to ¼”(6.35 mm) tube instead of 3/8” tubes.

VACCO SPACE PRODUCTS

3/8” Low Pressure Check Valve V1D10856-02

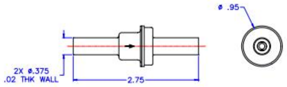
Description

VACCO Industries maintains a product line of stainless and titanium check valves to meet industry's demand for high reliability and tight leakage.

The 3/8” check valve developed for the Mars Ascent program is a titanium welded unit capable of very low external leakage and wide range of temperature extremes (-170° to 45° C). The valve contains an angled-machined seating surface, different from the poppet sealing surface allowing an increase in seat stress. The increase in seat stress improves sealing capability of the valve. This type of design has been in used in other VACCO valves for many years.

Features

- Temperature Range: -170° to 45° C
- Seat/Seal Configuration Flight Qualified
- All Welded External Construction
- Operating Pressure: 550 psia
- Weight: 20 grams
- 10 Years of Successful Flight Heritage
- 5,000 Poppet Cycles



Operating Parameters

Operating Pressure Range	550 psia	Reseat Pressure	1.0 psid
Proof Pressure	825 psia	External Leakage	<1 X 10^{-6} sccs GHee
Burst Pressure	1,100 psia	Internal Leakage	100 scch GHe @ 5 psid 0
Flow	.7 grams/second GHe	Operating Temperature	-170° to +45° C
Pressure Drop	10 psid	Non-operating Temperature	-40° to +45° C
Cracking Pressure	7.5 psid	Weight	20 grams

Back to main!

Pyrotechnic Valve

Link

Pyrovalve Key Technical Characteristics	
Initiators	Redundant ESA Standard Initiators
Design	All-welded Titanium design
Fluid Compability	Helium, Argon, Xenon, Nitrogen, MON, MMH, Hydrazine, Deionized Water, IPA
Response Time (Mechanical)	< 7ms
Mass	< 0.160 kg (depending on type)
Qualified Operating Temperature	-90°C ≤ T ≤ 100°C
Qualified Operating Pressure (MEOP)	310 bar
Proof Pressure	1.5 x MEOP (465 bar)
Burst Pressure (NO and NC)	
Pre firing	> 4x MEOP (rupture pressure: > 1240 bar)
Post firing	> 2.5x MEOP (rupture pressure: > 775 bar)
Leakage	
Normally Open	Internal leak after firing: < 1x10^{-6} scc/s (GHe) External leak before/after firing: < 1x10^{-6} scc/s (GHe)
Normally Closed	Internal leak before firing: < 1x10^{-6} scc/s (GHe) External leak before/after firing: < 1x10^{-6} scc/s (GHe)

[Back to main!]

Primary Propulsion Engine

Link

Thrust: 445 N each

Isp: 329 s @ ε = 375

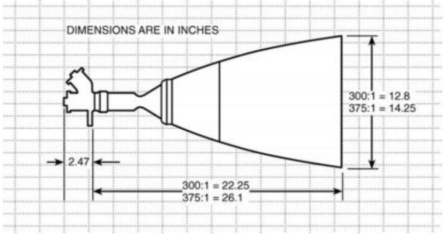
Qty: 4 total

Total propellant consumed: 10816 kg (5408 kg MON-3, 5408 kg Hydrazine)

Chosen due to highest Isp, reasonable thrust, and capability to use Hydrazine for monopropellant secondary.

R-4D-15 HiPAT™ 445 N (100 lbf) Dual Mode High Performance Rocket Engine

AEROJET ROCKETDYNE



Design Characteristics

- Propellant . . . . .Hydrazine/NTO (MON-3)
- Nominal Thrust (steady state) . . . . 445 N (100 lbf)
- Thrust Range (steady state) . . . . .329 – 556 N (70-125 lbf)
- Chamber Pressure* . . . . .9.4 bar (137 psia)
- Inlet Pressure* . . . . >16.2 bar (235 psia)
- Inlet Pressure Range . . . . .21.4 – 15.2 bar (310 – 220 psia)
- Valve . . . . .Aerojet Rocketdyne, Dual Coil, Single Seat
- Expansion Ratio . . . . 300:1 or 375:1
- Nominal Mixture Ratio (O/F) 1.0
- Mixture Ratio Range (O/F) . . . . 0.70 – 1.33
- Mass . . . . . 300:1= 5.2 kg (11.5 lbm), 375:1 = 5.44 kg (12.0 lbm)

Performance

- Specific Impulse @ 70°F and MR = 1.0. . . . . 300:1= 326 sec, 375:1 = 329 sec
- Total Impulse Qualified...> 9.55 X10[6] N-sec (2.15 X 10[6] lbf-sec)
- Minimum Impulse Bit . . . . 35.6 N-s (8 lbf-s)
- Demonstrated Steady State Firing Duration ....1,800 sec
- Total Number of Pulses Qualified . . . . 672 starts

. . . . 345 thermal cycles

Status

- Qualified
- Currently in Production

References

- AIAA-2003-4775

* At nominal Thrust

11411 139th Place NE • Redmond, WA 98052
(425) 885-5000 FAX (425) 882-5747

Payload

Speaker:

Move to Backup

Operational Timeline - Payload

Payload Operation Timelines

- At 4000 km inbound approach pick off mirror will be directing light to IR sensor
- At 2000 km inbound approach pick off mirror will switch to direct light to visual sensor
- At 800 km inbound approach SAR system will begin to collect data
- At 800 km outbound approach SAR will end data collection
- At 2000 km outbound approach pick off mirror will switch back to IR sensor
- Primary science collection will be finished at 4000 km outbound approach

SAR Model Verification

Inputs:

SAR Parameter	Look Angle	SC Height	Bandwidth	Central Frequency	Antenna Size
ERS-1	23 deg	785 km	15.55 MHz	5.3 GHz	10m x 1m

Comparing Outputs vs Actual Performance:

SAR Performance	Model	ERS-1
Cross-track Footprint	96.5 km	100 km
Cross-track Resolution	24.7 m	26.3 m
Slant Range Resolution	9.64 m	10 m

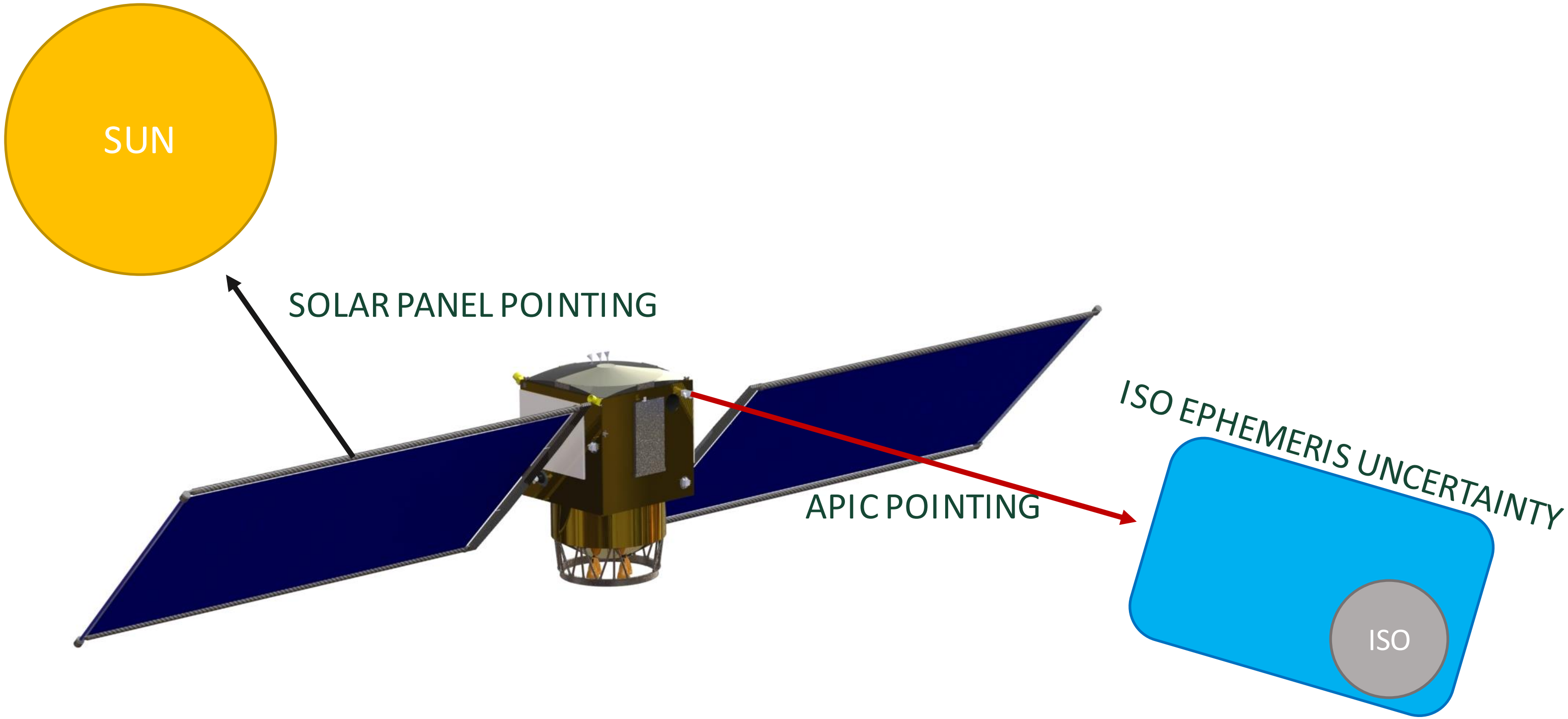
Back to Design Specifications-SAR

Payload Risks

- Given that the imager isn't aligned properly there is a possibility of a need to abandon primary and secondary sciences on one of the spacecraft adversely impacting the payload and primary and secondary science subsystems, which can result in an ability to only complete half of the required primary and secondary science data, limiting us to only using one spacecraft to rendezvous with the ISO

GNC

Speaker: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SOLAR PANEL POINTING
APIC POINTING
ISO EPHEMERIS UNCERTAINTY
ISO

Phase 5: ISO Flyby

Applicable Level 3 GNC Requirements

ID	Requirement	Parent Requirement
GNC7	The GNC system shall be capable of optically navigating to the ISO autonomously.	
GNC8	The GNC system shall calculate the relative position between the flight system and the ISO +/- TBD km within a distance of TBD km from the ISO.	
GNC9	The GNC system calculate the relative velocity between the flight system and the ISO +/- TBD km/s within a distance of TBD km from the ISO.	
GNC12	The GNC system shall propagate the position of ISO with an accuracy of +/- TBD km throughout autonomous operations.	
GNC13	The GNC system shall propagate the velocity of the ISO with an accuracy of +/- TBD km/s throughout autonomous operations.	

Navigation Model

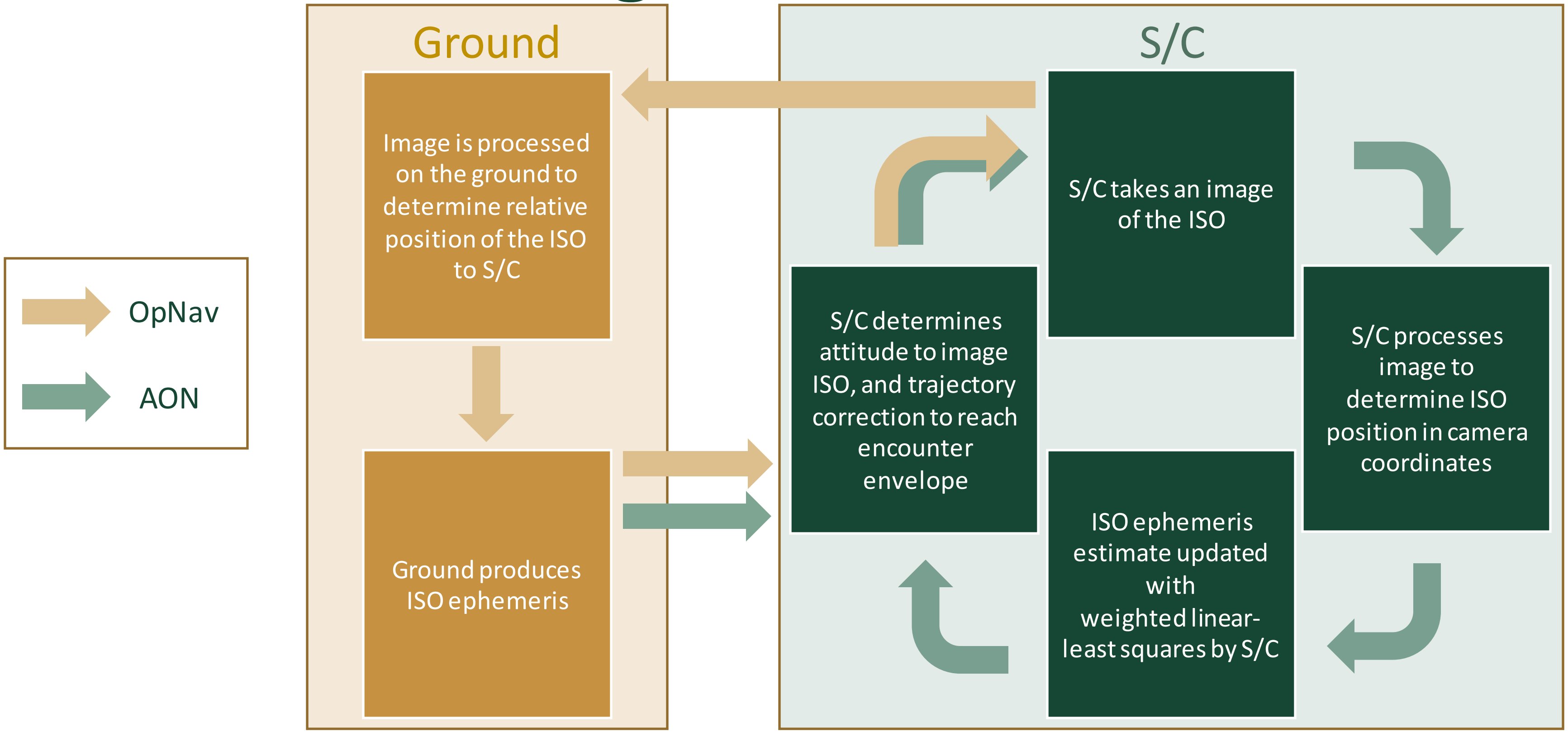


Image Processing Methods

Iso <= 10 Pixels

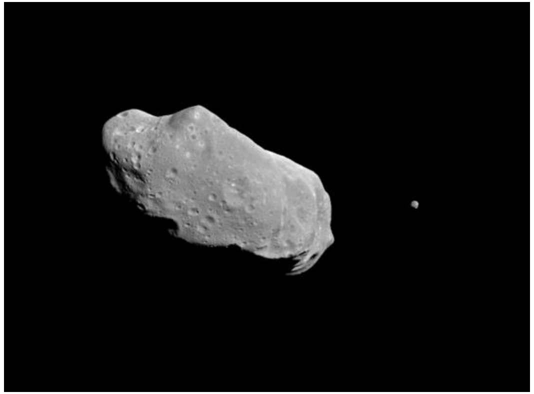
- Analytics Function Fitting

Iso > 10 Pixels

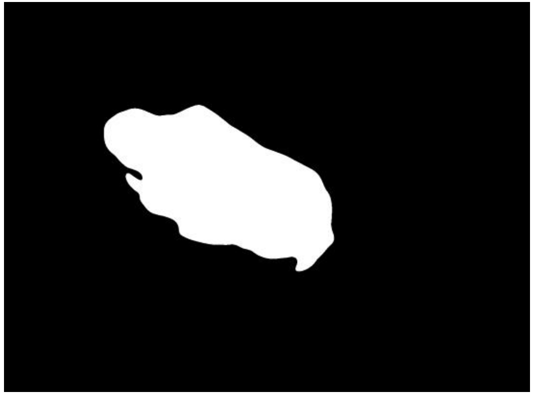
- Threshold Segmentation
- Edge Detection
- Ellipse Approximation via least squares

Input	Output
Image of the ISO taken by the Spacecraft	ISO center in Pixel Coordinates

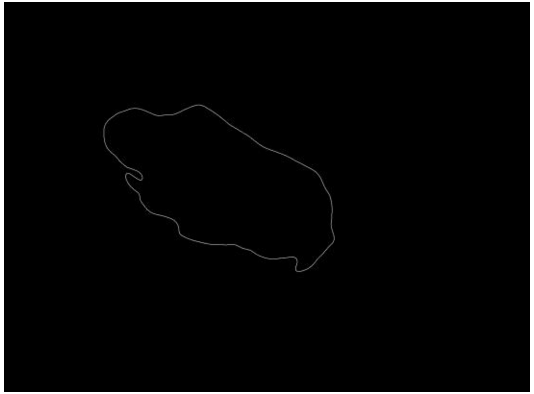
Image Processing Model



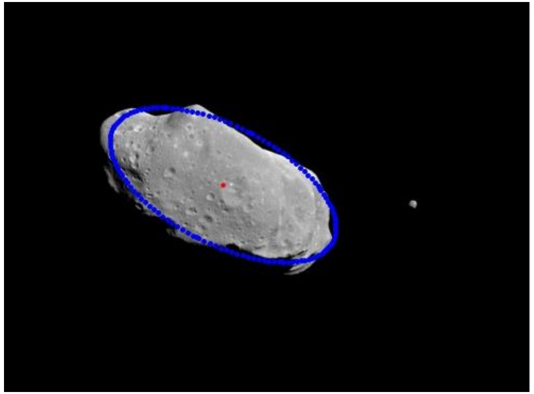
Base Image



Threshold Segmentation



Edge Detection



Ellipse Approximation with Center

GNC Risks

- Given that the ISO shape and shadow can affect ephemeris updating of the autoNav system leading to false knowledge of the ISO position there is a possibility of an inability to properly image the ISO since we will be at a different range than expected adversely impacting the payload components, which can result in an inability to acquire images and science information from the ISO.

Likelihood: 3

Severity: 1 or 2

Thermal

Thermal Models – SolidWorks

Overview

- Simplified geometry
- Steady-state thermal studies within SolidWorks
- Multiple scenarios studied varying different parameters

Outcomes

- Verify behavior of combined system
- Provide recommendations for component placement, mounting, and surface finishes

Inputs

- Geometry
- Material and surface properties
- Component contact resistance
- Mission scenarios

  Component heat loads
  Environmental heating
  Louver and heater settings

Outputs

- Temperature distribution
- Heat flux through different surfaces

Thermal Models – SolidWorks

Scenario Name	Tank Fill	Component Activity	Pointing Orientation	Distance from Sun	Engine Activity
1AU Standard	Full	On	Earth	1 AU	Off
5AU Standard	Full	On	Earth	5 AU	Off
1AU Everything Off	Full	Off	Earth	1 AU	Off
5AU Everything Off	Full	Off	Earth	5 AU	Off
1AU Everything On	Full	On	Earth	1 AU	On
5AU Everything On	Full	On	Earth	5 AU	On
5AU No Prop	Empty	Off	Earth	5 AU	Off
Phase 4 -OpNav	Full	Phase 4.7	Earth	5 AU	On
Phase 7 - EOL	Full	Phase 7.1	Earth	5 AU+	On
Emergency 1	Half	Off	Sun	1 AU	On
Emergency 2	Empty	Off	Sun	5 AU	On

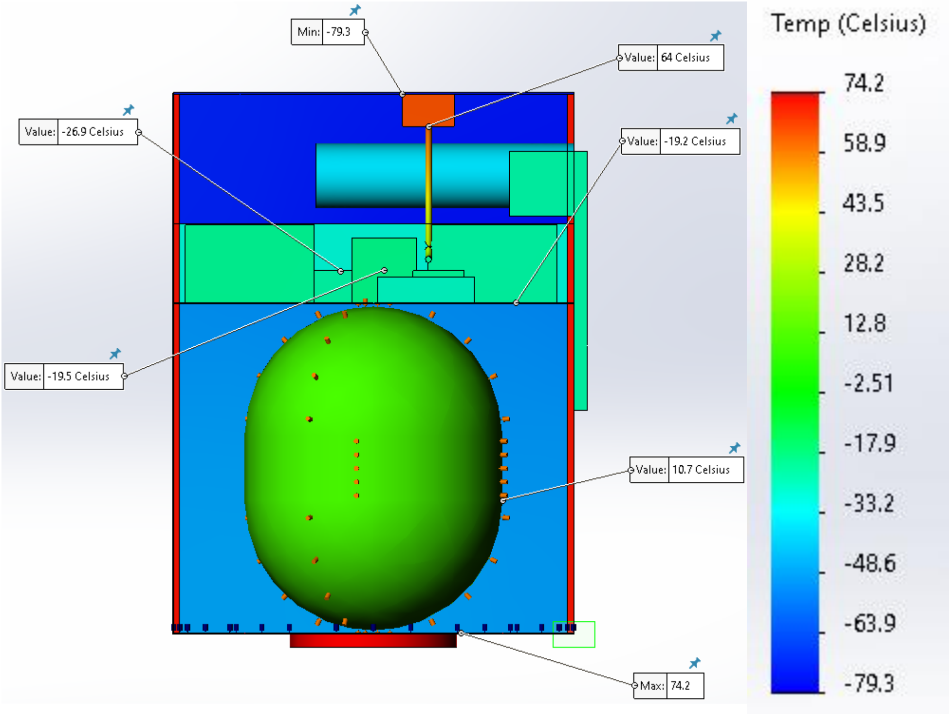
CAL POLY

Detumble

Thermal Models – SolidWorks

Scenario Name	Temperatures (Deg C)			Louver Value (Effective ε)	Heater Value (W)	Notes and Issues
	Max	Min	Baseplate			
1AU Standard	78	-67	-3	0.3	150	Too cold, excess heat leak
5AU Standard	65	-80	-20	0.3	150	Too cold, excess heat leak
1AU Everything Off	65	-124	-83	0.1	N/A	Too cold, expected
5AU Everything Off	-120	-205	-187	0.1	N/A	Too cold, expected
1AU Everything On	N/A	-68	-6	0.3	50	Engines highly insulated
5AU Everything On	N/A	-71	-9	0.3	100	Engines highly insulated
5AU No Prop	-120	-205	-187	0.1	N/A	See 5 AU Everything Off
Phase 4 -OpNav	N/A	-71	-9	0.3	100	See 5 AU Everything On
Phase 7 - EOL	N/A	-71	-9	0.3	100	See 5 AU Everything On
Emergency 1	76	-64	2	0.3	250	Too cold, excess heat leak
Emergency 2	76	-64	2	0.3	250	Too cold, excess heat leak

Thermal Models - SolidWorks



1 AU – Components on, Engines off

- Baseplate ties all components together, keeps them within 5 degrees of same temperature
- Future work: give each component specialized conduction to radiator using heat pipes
- Outer MLI exposed to sun has highest temperatures, not exposed has lowest. Neither significantly affect nearby components
- Internal radiation causes heating to telescope as well as heat leak not via radiator

Thermal Models – SolidWorks

Assumptions

- Geometry is close enough to provide +- 20 deg accuracy
- Radiation between internal components
- External panels approximate MLI via material and surface properties
- All components operate with constant power use and waste heat generation
- Solar heating modeled by applying internal heat load to inside of MLI sheets

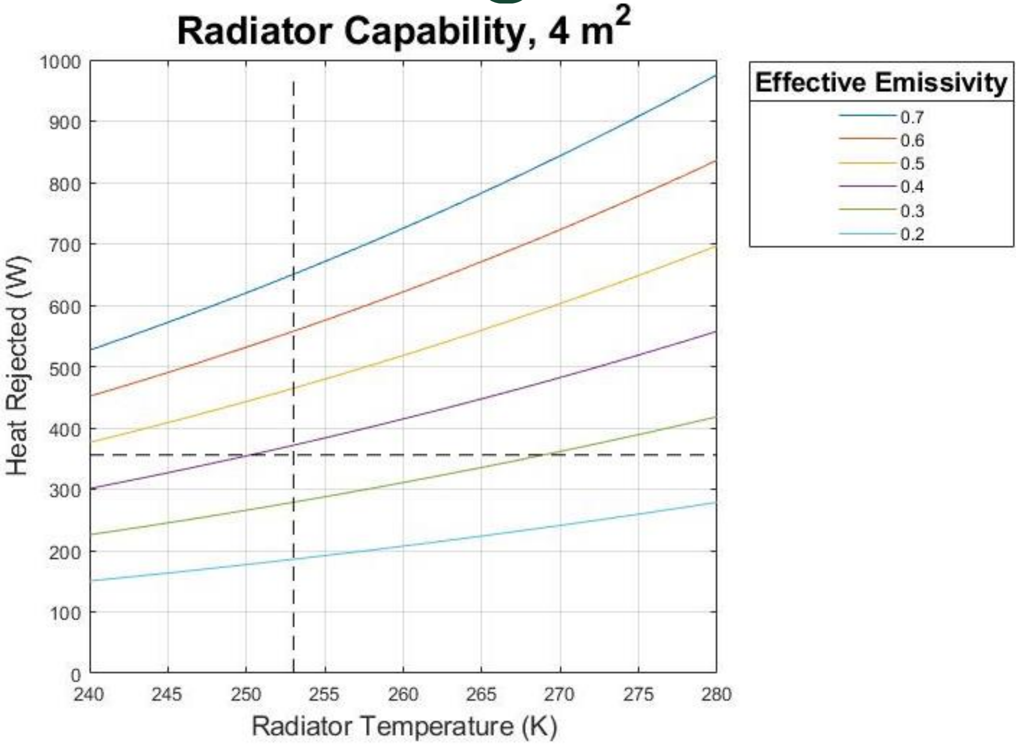
Takeaways

- Heater balance – Power positive until later phases. Can run heaters as needed for most of mission, but need enough insulation for final phases
- Heat Leak – more heat than anticipated currently leaking from spacecraft, need proper insulation for thermal system to behave as intended
- Conductance – high conductance heat paths reduce component temperatures excessively, more design needed to correctly control temperature differentials

Thermal Radiator Sizing

- Total Area = 4 m^2
- Target Heat Rejection = 356 W
- Target Temperature = -20 °C
- Louver Control

  Effective emissivity range: 0.2- 0.7
- 180 – 650 W range at –20 °C



Possible heat rejection of 4m2 Radiators
Dashed lines are target values

CAL POLY

Thermal Heat Pipe Design and Sizing

- Aluminum was chosen for the main heat pipe material as it has a high heat transfer coefficient and a low density compared to other heat pipe material options considered (Copper and Stainless 316).
- Ammonia was chosen as the operating fluid, as the temperature ranges of ammonia best fit the operating temperature ranges and thermal conductivity that our system components required.
- A grooved aluminum pipe was chosen for the wick design, as it would accomplish the desired capillary action based on our orbital parameters and thermal requirements for ammonia as a working fluid.

Pipe Materials	Pipe Wall Thickness	Pipe Diameter	Pipe Length Estimate	Ammonia and Mercury Mass Estimates	Heat Pipe Total Mass Estimate
Aluminum	0.00089m	0.008m	11.5m < L < 34.5m	2-4kg of Ammonia	90kg – 270kg

CAL POLY

MLI Spotheating Risk

Given that	There is a possibility of	Adversely impacting	Which can result in	Likelihood (out of 5)	Severity (out of 5)	Sub team
There is spot heating on the MLI blankets	An unintended reduction in MLI performance leading to excess heat loads or cooling of the spacecraft	The thermal system	The degradation of MLI which will reduce the ability of the thermal system to operate correctly for the entire mission	3	4	Thermal

Loss of Control Risk

Given that	There is a possibility of	Adversely impacting	Which can result in	Likelihood (out of 5)	Severity (out of 5)	Sub team
There are vibrations of launch and the 20-year operation length	The components tied to the thermal system losing necessary control	The thermal system	A hinderance in overall performance of the thermal system, potentially up of loss of the spacecraft	3	2	Thermal

Thermal Team Risks Continued

Given that	There is a possibility of	Adversely impacting	Which can result in	Likelihood (out of 5)	Severity (out of 5)	Sub team
The temperature sensors fail and that the team does not currently know a way to calibrate a temperature sensor in space	Component damage due to improper thermal man agement	The proper management and reliability of any component on the spacecraft where temperatu re knowledge is lost	The reduction of longevity, espe cially of the electronics	2	2	Thermal

Thermal Team Risks Continued

Given that	There is a possibility of	Adversely impacting	Which can result in	Likelihood (out of 5)	Severity (out of 5)	Sub team
Thermal models cannot 100% match real world conditions or represent every interaction within the spacecraft	The expected behavior not being fully achieved leading to incorrect control	The thermal system	Out of bounds temperatures of some components	2	3	Thermal

Thermal Team Risks Continued

Given that	There is a possibility of	Adversely impacting	Which can result in	Likelihood (out of 5)	Severity (out of 5)	Sub team
Steady state conditions are much easier to analyze, plan for, and model	Unforeseeable dynamic situations not being modeled or analyzed beforehand	The startup of electronic components as thermal control may lag	Issues quickly adapting to changes in the environment or spacecraft state meaning some timelines mat not be executed as desired			Thermal

Thermal Team Risks Continued

Given that	There is a possibility of	Adversely impacting	Which can result in	Likelihood (out of 5)	Severity (out of 5)	Sub team
There is inadequate heat power	The heaters failing to keep components warm, particularly the propulsion tanks	The propulsion fuel	The propulsion failing during navigation to the ISO	2	3	Thermal

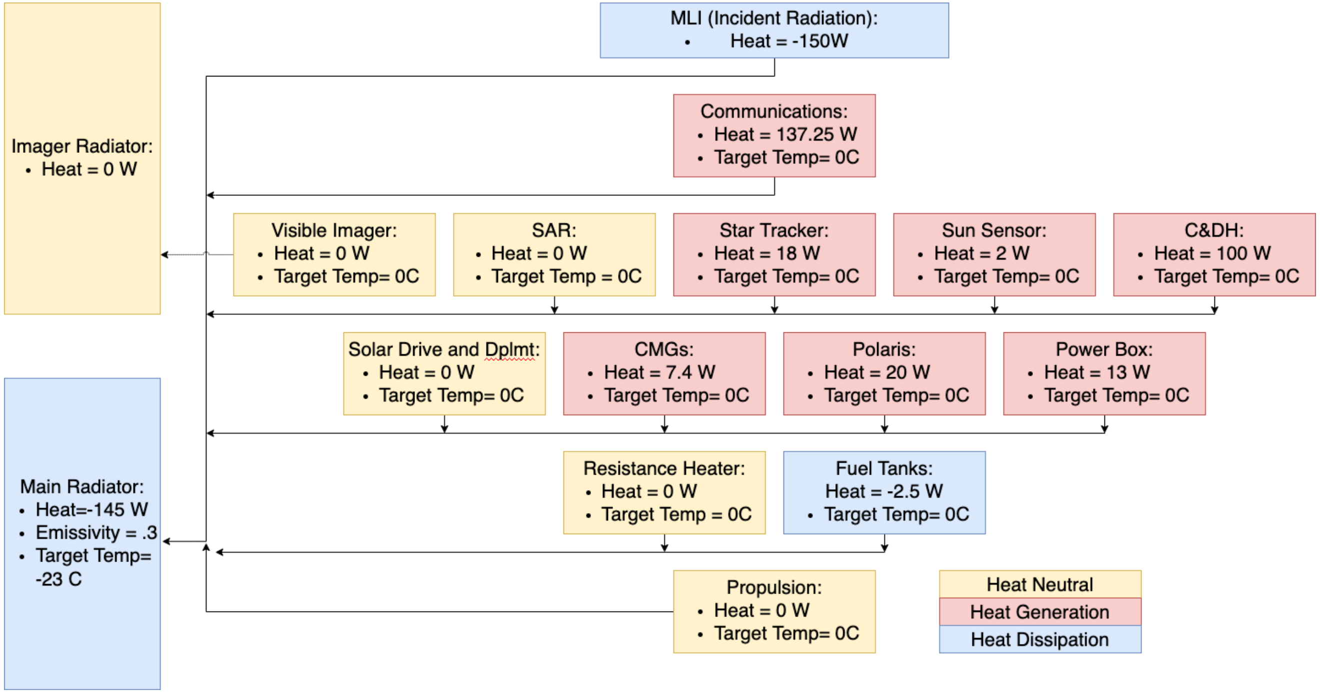
Thermal Team Risks Continued

Given that	There is a possibility of	Adversely impacting	Which can result in	Likelihood (out of 5)	Severity (out of 5)	Sub team
There is unanticipated degradation of the radiators	The radiators emitting less waste heat than necessary to maintain nominal onboard temperatures	The power system and any power consuming systems	The reduced performance of the thermal system	2	3	Thermal

Thermal Team Risks Continued

Given that	There is a possibility of	Adversely impacting	Which can result in	Likelihood (out of 5)	Severity (out of 5)	Sub team
There is external heat load coming into the spacecraft due to solar irradiation, albedo from Earth, and albedo from the ISO	The radiators not being sized to emit enough waste heat from the spacecraft	The entire thermal management procedures	The spacecraft heating up more than anticipated and thermal control no longer being able to maintain temperature ranges	2	2	Thermal

Phase 6 – Support Bus Downlink

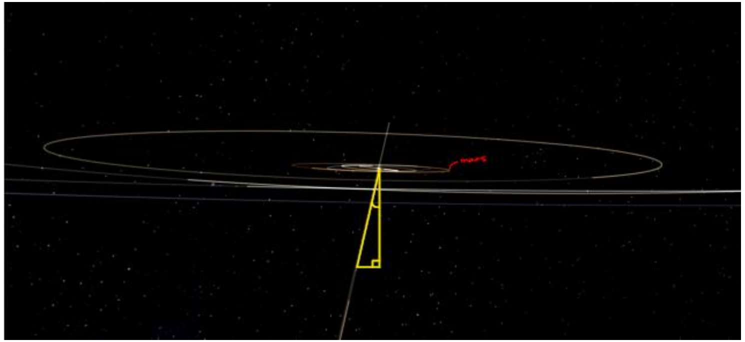


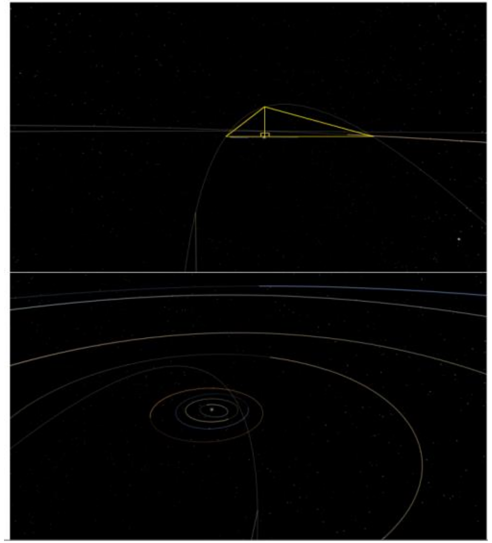
Ground System

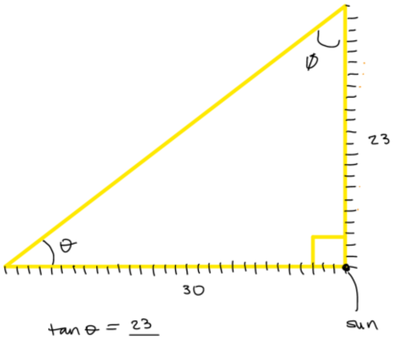
Speaker:

Ground System/Spacecraft Geometry

- ISO entry case





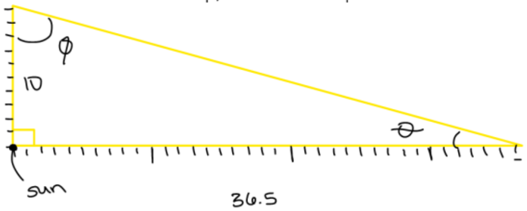


$\tan\theta = \frac{23}{30}$

$\boxed{\theta = 37.476^\circ}$

$\phi = 90 - \theta$

$\boxed{\phi = 52.524}$



$\tan\theta = \frac{10}{36.5}$

$\boxed{\theta = 15.3215^\circ}$

$\phi = 90 - \theta$

$\boxed{\phi = 74.678^\circ}$

Ground System Risks

- Given that there are unknows about the ISO's trajectory there is a possibility of only being able to contact the spacecraft irregularly during mission critical times adversely impacting the mission control's ability to communicate with the spacecraft, which can result in uncertain orbits and GNC knowledge.

Likelihood: 4

Severity: decreases over time

Mission Operations and Facilities

Speaker: Alexi or Berenice

COSMIC Operation Centers – move to backup

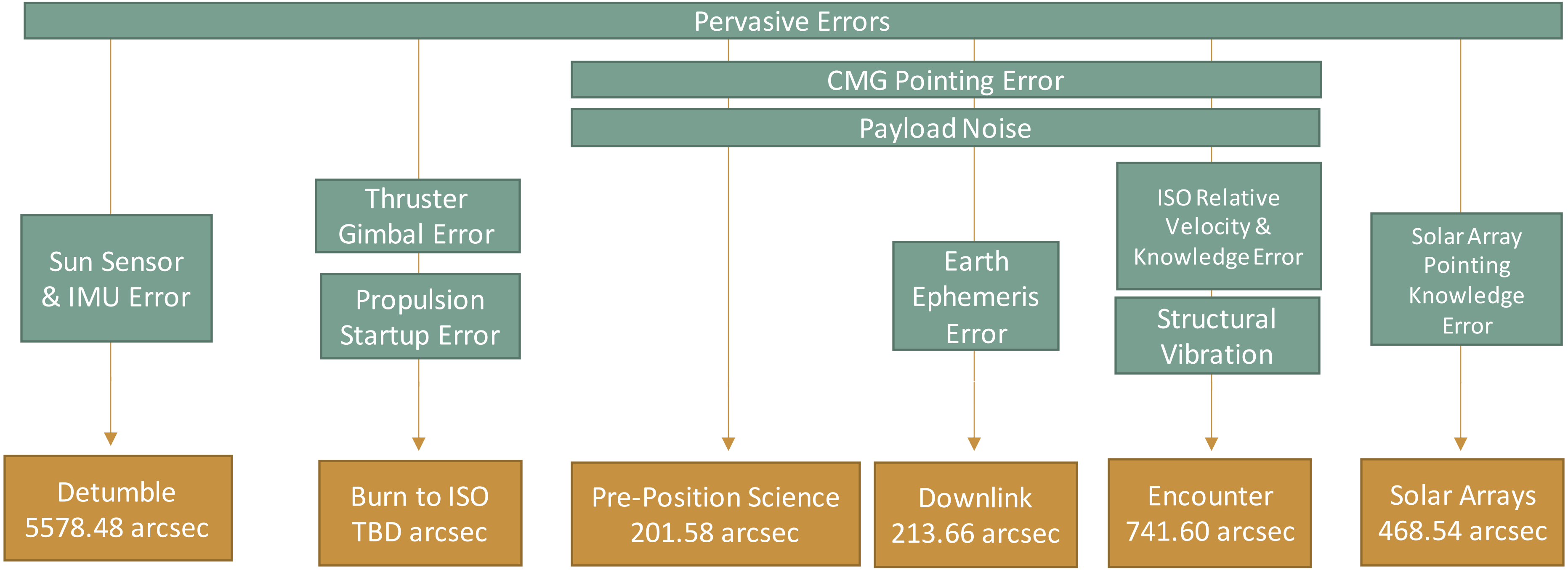
ID	GS Facility	Rationale	GS Operation
MP3:8 MS1:4	Science Operation Center	Processes and meets all primary and secondary science-based objectives	
MP1,2 MP9	Mission Control Center	Processes and meets all command, navigation, and monitoring based requirements	
MP10	Network and Data Storage	Supports all operations of the Science and Mission Centers	
MP10	NASA DSN	Contracted Element for spacecraft contact	

Operation Centers Interfaces

ID	Requirement	Rationale	GS Facility	Number of Personnel Assignment Per Spacecraft (x2)
G2, MP2, G26 G27, G28	Orbit design and control	Orbit/trajectory related station keeping, ISO departure determination and command design, and ISO encounter trajectory management	Mission Control Center	3 (ex: ephemeris management)
MP3, MP4, MP5 MP6, MP7, MP8 G25	Science processing	Received science packages must be unloaded to determine progress towards goals	Science Operations Center	2 (ex: dielectric, rotation, dimension)
MS2, MS3, MS4	Science command design	Science teams will want to point at celestial objects which competes with solar and station keeping pointing	Science Operations Center	4 (ex: exoplanet, sky coverage)
G3	Station keeping, subsystem health monitoring, response and command design	ADCS, propulsion, power keeping demands for pointing, thermal management, C&DH updates or patches	Mission Control Center	+4 (fuel management, trajectory upkeep, subsystem liaisons)
G20, G21, G22 G24,	Mission command packaging	The last step before COSMIC commands are sent to the DSN for flight system uplink	Mission Control Center	2

ADCS

Pointing Capability Model



Preposition Science Pointing Budget

Accuracy Error Description	3 σ Error [arcsec]
CMG Pointing Error During Quasi-Static	20.626
Sensor Misalignment	0.017
Payload Misalignment	0.017
Star Tracker Misalignment	0.017
Star Tracker Noise	21.000
Thermal Deviations	135.520
Star Tracker Error	11.100
Per Axis Total	**188.296**

Stability Error Description	3 σ Error [arcsec / s]
MED Vibration	0.5
Propellant Sloshing	10.31324031
Total	**10.81324031**

Downlink Pointing Budget

Accuracy Error Description	3 σ Error [arcsec]
CMG Pointing Error During Quasi-Static	20.626
Sensor Misalignment	0.017
Payload Misalignment	0.017
Star Tracker Misalignment	0.017
Star Tracker Noise	21.000
Thermal Deviations	135.520
Star Tracker Error	11.100
Per Axis Total	**188.296**

Encounter Pointing Budget

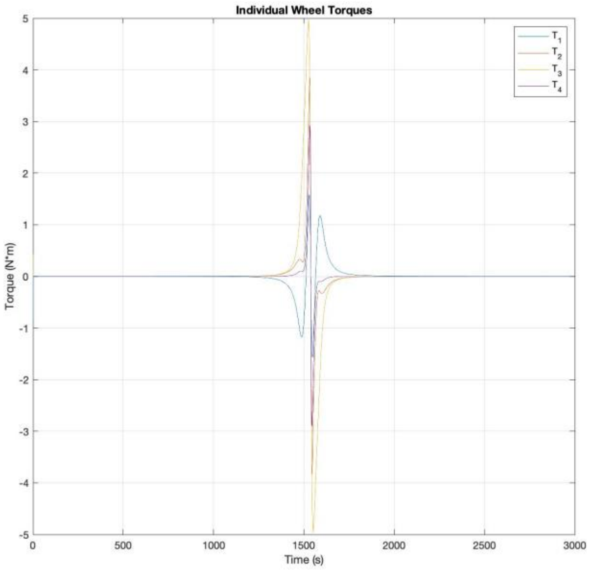
Accuracy Error Description	3 σ Error [arcsec]
CMG Pointing Error During Large Maneuver	412.530
Sensor Misalignment	0.017
Payload Misalignment	0.017
Star Tracker Misalignment	0.017
Star Tracker Noise	21.000
Thermal Deviations	135.520
Star Tracker Error	11.100
Per Axis Total	**580.199**

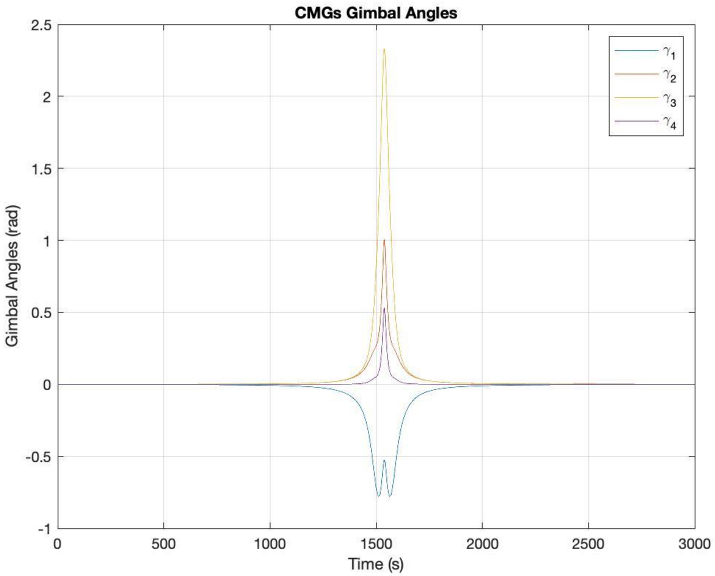
Stability Error Description	3 σ Error [arcsec / s]
MED Vibration	0.500
Propellant Sloshing	10.313
Structural Vibration During Large Maneuver	0.003
APIC Measurement Error	1.974
Total	**12.790**

Solar Array Pointing Budget

Accuracy Error Description	3 σ Error [arcsec]
CMG Pointing Error During Quasi-Static	20.626
Antenna Misalignment	0.017
Payload Misalignment	0.017
Star Tracker Misalignment	0.017
Star Tracker Noise	21.000
Thermal Deviations	135.520
Star Tracker Error	11.100
Solar Array Drive Pointing Knowledge error	72
Per Axis Total	260.296

CMG Wheel Torques/ Gimbal Angles (return)





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CMG Preposition Desaturation due to Disturbance Torques

- Solar Radiation Pressure is the highest magnitude disturbance torque acting on the spacecraft during the prepositioned phase
- Attitude of spacecraft will point high gain antenna at Earth during most of phase, Solar Panels will have ~ 80-degree inclination to sun vector.
  - Near constant torque and accumulation of momentum
- **Torque values over time**: 0.1336 - 0.1430 Nm
- **Desaturation Timeline with 48 Nms of Momentum Storage**: 93.24 - 99.5 hours
- Desaturation will be performed by calculating the angular impulse vector required and using the secondary thrusters to achieve necessary impulse.
- Required mass of 76 kg for 20-year mission period

ADCS Risks

- Given that the MEDs fail over the long mission life there is a possibility of an inability to perform the slew for encounter adversely impacting the resolution and our ability to find the ISO, which can result in an inability to achieve objectives.

Comms

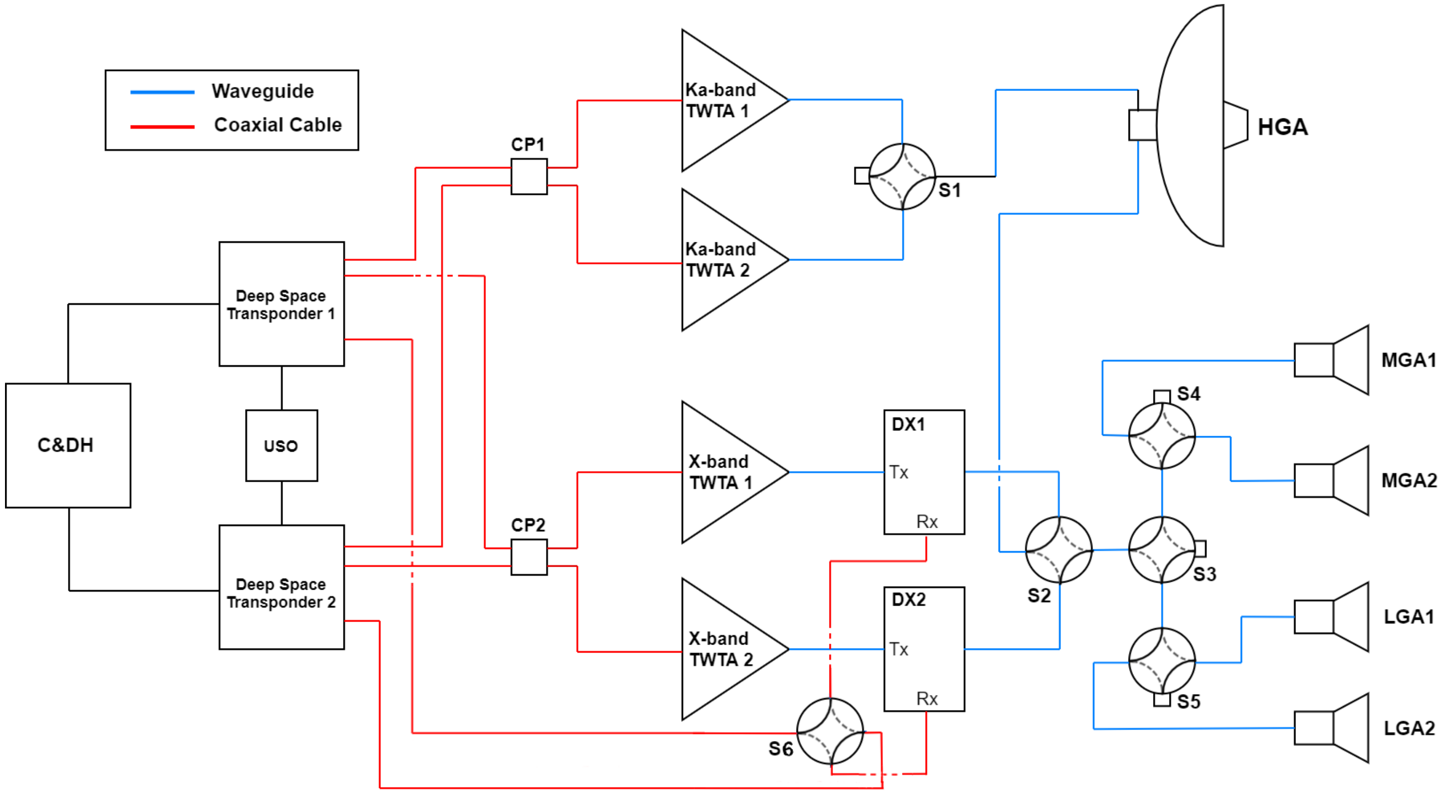
Communication – Downlink Budget Supporting Values

Antenna	Supplier	Boresight Gain [dBi]	3dB Beamwidth [deg]	Centering Frequency [GHz]	Max Pointing Error [deg]	Required BER (u/d)	Modulation Scheme	Coding	Required Eb/N0 [dB]	Nominal Link Margin [dB]
HGA [Ka / X]	In House	57.88 / 44.81	0.217 / 0.979	32.2 / 7.153	0.1	1e-6	Direct BPSK	Turbo (R=½, l =5)	1.25	3
MGA	JPL	18.8	20	7.153	0.1	1e-6	Direct BPSK	Turbo (R=½, l =5)	1.25	3
LGA	JPL	7.7	80	7.153	0.1	1e-6	Direct BPSK	Turbo (R=½, l =5)	1.25	3

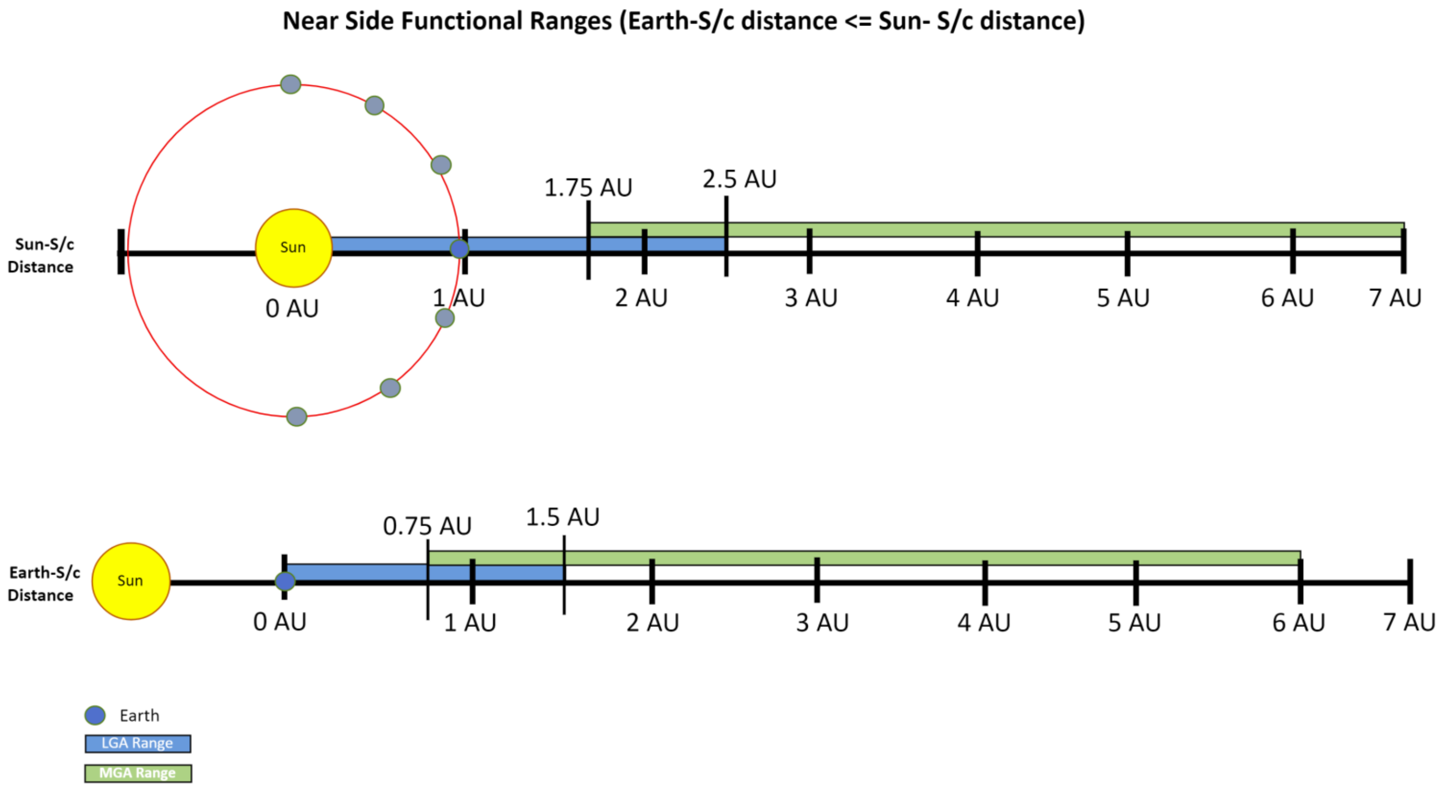
RETURN TO 138

Backup

Telecom – Block Diagram

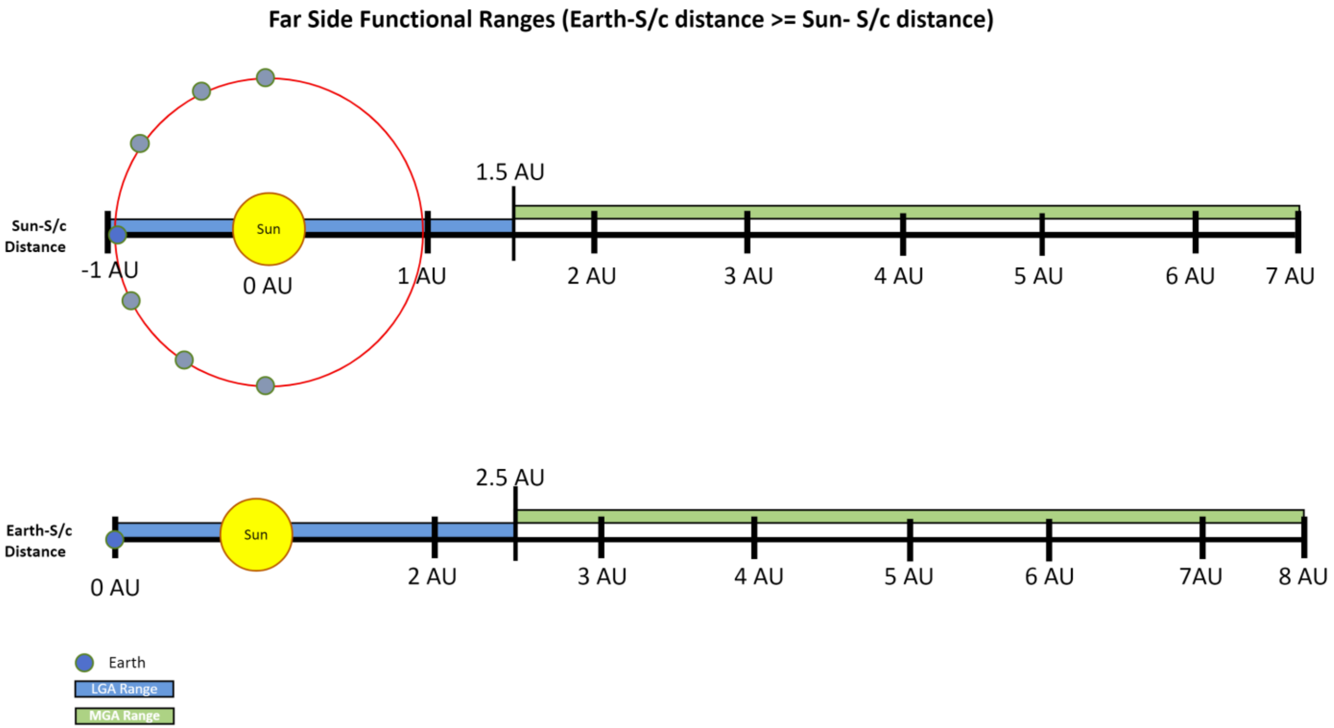


Safe Mode Diagrams



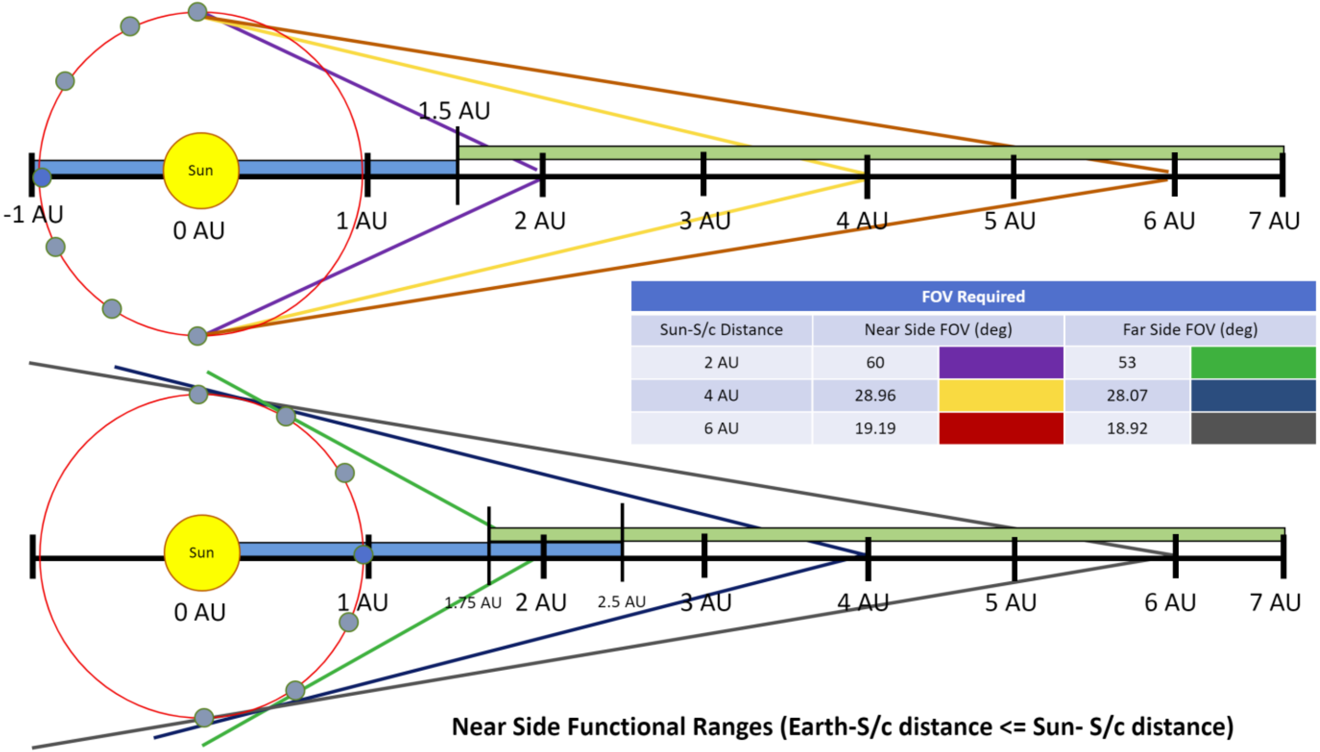
Detumble

Safe Mode Diagrams



Safe Mode Diagrams

Far Side Functional Ranges (Earth-S/c distance >= Sun- S/c distance)



FOV Required				
Sun-S/c Distance	Near Side FOV (deg)		Far Side FOV (deg)	
2 AU	60		53	
4 AU	28.96		28.07	
6 AU	19.19		18.92	

Near Side Functional Ranges (Earth-S/c distance <= Sun- S/c distance)

Safe Mode Diagrams

With s/c rotating at 3 deg/min minimum communication period with Earth every 2 hours

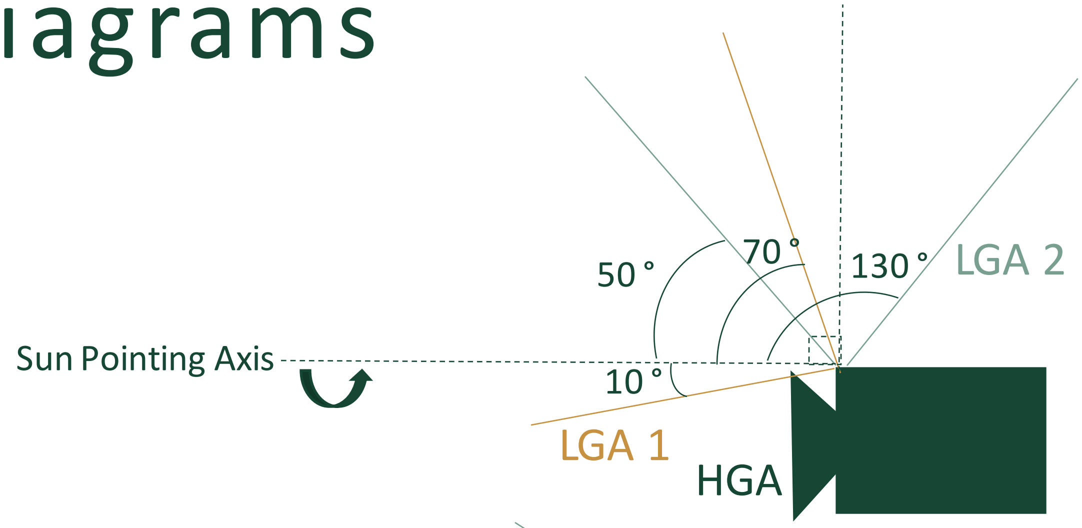
Using LGAs

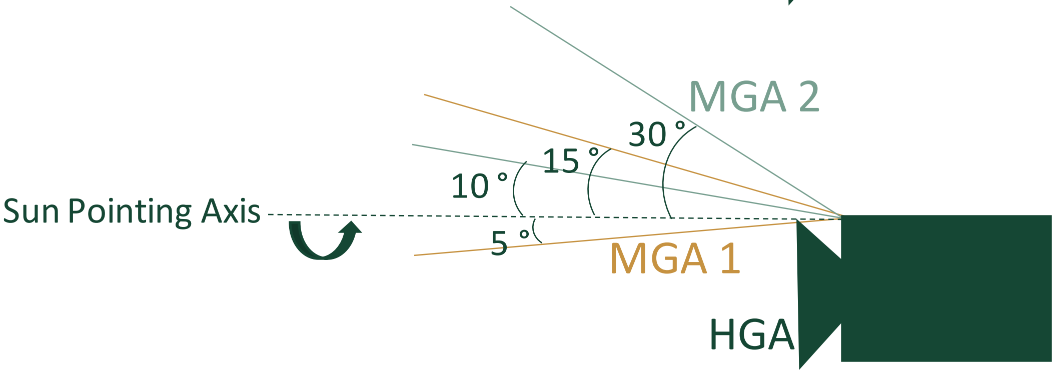
- PP: 26 min
- 1AU: 13 min

Using MGAs

- 2AU: 10 min
- 4AU: 20 min
- 7AU: 44 min

*Earth relative distance





Telecom Level 3 Requirements

ID	Requirement	Verification Method
COM1	The communication system shall transmit science data over a high gain downlink.	
COM2	The communication system shall transmit telemetry over a high gain downlink.	
COM3	The communication system shall be capable of transmitting telemetry over a medium gain downlink.	
COM4	The communication system shall be capable of transmitting telemetry over a low gain downlink.	
COM5	The communication system shall support ranging over a high gain link.	
COM6	The communication system shall support Doppler tracking over a high gain link.	
COM7	The communication system shall receive commands over a high gain uplink.	
COM8	The communication system shall be capable of receiving commands over a medium gain uplink.	
COM9	The communication system shall be capable of receiving commands over a low gain uplink.	
COM9	The communication system shall be capable of receiving commands over a low gain uplink.	
COM10	The communication system shall transmit science data at Ka-band.	
COM10	The communication system shall transmit science data at Ka-band.	
COM11	The communication system shall transmit telemetry at X-band.	
COM11	The communication system shall transmit telemetry at X-band.	
COM12	The communication system shall support ranging at X-band.	
COM12	The communication system shall support ranging at X-band.	

ST: ST: 0:00-0:00

Telecom Level 3 Requirements

ID	Requirement
COM15	The communication system shall transmit science data with a minimum data rate of 32.4 kbps.
COM16	The communication system shall receive commands at a data rate of up to 2 kbps [TBC] during normal operations.
COM17	The communication system shall transmit telemetry with a data rate of TBD bps during normal operations.
COM18	The communication system shall transmit telemetry at a minimum data rate of 20 bps [TBC] while in system fault mode.
COM19	The communication system shall receive commands with a data rate of TBD bps during system fault mode.
COM20	The communication system shall be capable of continuous transmission.
COM21	The communication system shall be capable of continuous reception.
COM22	The communication system shall have a high gain transmission EIRP of 80.89 dBW [TBC].
COM23	The communication system shall have a medium gain transmission EIRP of TBD dBW.
COM24	The communication system shall have a low gain transmission EIRP of TBD dBW.
COM25	The communication system shall comply with CCSDS format to communicate with the DSN.
COM26	The communication system shall have a minimum reliability of 98%.
COM27	The communication system shall have a mass of 117 kg +/- TBD kg.
COM28	The communication system shall have a volume of 0.072 m^2 +/- TBD m^3.
COM29	The communication system shall operate in the space environment for 22 years.
COM20	The communication system shall operate at a heliocentric range of 0.5-7 AU.

C&DH

Jonathan Hood

Samuel Westrick

Solomon Dovinh

Juan Carlos Flagg

Processing, Memory Calculations

Yellow: Assumptions
Blue: Raw analog output
Orange: Based on default values

Team	Data Source	Data Format Protocol (type)	Data rate (MB/s)	Data storage (MB)	Frequency (Hz)	Operation Time & Phase
FS-Payload	SAR	8-bit analog, no protocol, Spacewire (can handle data rate, does point to point, no limit on packet size)	4.8	320.784	15600	*(ASSUMPTION) On for full encounter at max data rate (4/28 Matt Jones run of SAR model, bounded by resolution morphology constraints)
	Visual Imager					
	Remote Obs.	14-bit analog, SpaceWire	10.00	8078.4	5.347593583	*Remote Obs. (ASSUMPTION, see Editable Variables) Latest readout rate from Payload, 6 months campaign, 1 hr cadence, all data stored onboard for full campaign, Visual only
	Exoplanet	""	""	1344		*Exoplanet (ASSUMPTION, see Editable Variables) Latest readout rate from Payload, 4-week campaign, 30 min cadence, all data onboard for full campaign, Visual only
	OpNav/AutoNav	""	""	0		*ASSUMPTION: no opnav/autonav images stored onboard; transmitted or processed directly.
	Encounter	""	""	112.2		*(ASSUMPTION) taking 60 images during encounter at 1.87 MB/image
	IR Imager	14-bit analog, SpaceWire	**2.00**	24		*(ASSUMPTION) taking 60 images during encounter at 0.4 MB/image
	Health Sensors	Analog voltage for temp sensing (1553B)	1.87E-04	0.8064	4.67	*(ASSUMPTION: stored health data is downlinked weekly) Every phase, worst case 1/3 Hz frequency check

Return

Processing, Memory Calculations

FS-Propulsion	Chem Prop Sensors	Floats (4 bytes)		9.91872		
	Thermocouples	1553B A	0.00252		630	
	Check Frequency (Hz)	10				
	Channels	63				
	Pressure Transducers	1553B A	0.24		60000	
	Check Frequency (Hz)	1000				
	Channels	60				
	Current/voltage Sensors:	1553B A	0.002	0.8064	4200	
	~~Electric Prop sensors~~	~~Floats (4 bytes)~~	~~0.000204~~	~~8.22528~~		~~*All prop sensors; may send down health data only occasionally or when requested~~
	~~Thermocouples~~				~~51~~	
	~~*Check Frequency (Hz)*~~	~~1~~				
	~~*Channels*~~	~~51~~				
	~~Pressure Transducers~~				~~510~~	
	~~*Check Frequency (Hz)*~~	~~10~~				
	~~*Channels*~~	~~51~~				
	~~PPU~~		~~0.004~~	~~0.08064~~	~~1000~~	
	~~*Check Frequency (Hz)*~~	~~1000~~				
	~~*Channels*~~	~~1~~				
	Thrust Control	Digital PWM control	0			

Yellow: Assumptions Blue: Raw analog output Orange: Based on default values

Return

Processing, Memory Calculations

FS-Power	Power Distribution Unit(s)	MIL-STD-1553	1.25	0	As needed	*(ASSUMPTION) using entire available data rate
	Health Sensors	1553B, A	0.002	0.8064	150	*two thermocouples and the PDU for current/voltage sensing
FS-Comms	Transponder	Spacewire	0	0		
	Redundant Transponder	Spacewire	0	0		
	Health Sensors	1553B, A	0.002	0.8064	1000	
~~**FS-Thermal**~~	~~Thermostat~~	~~readings, output analog volt (assuming 1553B due to required health data)~~	~~0.002~~	~~0.8064~~	~~50~~	~~Constantly operating; C&DH controls, rate unknown.~~
FS-GNC	Health Sensors	1553B, A	0.002	0.8064	200	
	APIC	RS-422	**X kbps**	**How Much is Stored?**	Readout rate, exp time?	Encounter Time, Phase 5
ADCS	Health Sensors	1553B , output A	0.002	0.8064	2000	
	Star Tracker (2x)	1553B	**X kbps**	**How Much is Stored?**	20	Constant, 10Hz, 4Hz
	IMU(s) (2x)	RS-422 converted to 485	**X kbps**	**How Much is Stored?**	400	Constant, 200 Hz
	CMGs x4	1553B	**X kbps**	**How Much is Stored?**	As needed	Constant
	Sun Sensor (x4)	RS-422 converted to 485	**X kbps**	**How Much is Stored?**	N/A	Constant

Yellow: Assumptions Blue: Raw analog output Orange: Based on default values

Return

Processing, Memory Calculations

FS-Structures	Solar Array Drive Assemblies x2	RS-485	0	0		Command 0-48000 to turn to specific angle
	~~Electric Thruster Gimbal~~	~~RS-485~~	~~0~~	~~0~~		~~Command 0-64000 to turn to specific angle~~
	Thruster actuators x8	RS-485	~~0~~	~~0~~		
	x2 release mechanisms	RS-485	0	0		
	Health Sensors	1553B	0.002	0.8064	800	
	x2 array hinges	RS-485	0	0		
FS-C&DH	Comm cards-->Transponders	Spacewire (no limit packet size, high data rate)	400	0		Every Phase, downlink/uplink duration
	~~A2D Unit~~	~~Matches input and output~~	~~0~~	~~0~~		~~Every Phase, continuous~~
	Types	Sigma-Delta	0	0	1.00E+05	24-bit resolution
		Successive Approx.	0	0	1.00E+06	16-bit resolution
		Flash	0	0	1.00E+08	12-bit resolution
	Interface Cards	Inputs Spacewire,RS422, 1553B, RS485	0	0		Every Phase, continuous
	Health Sensors	1553B	0.002	0.8064	200	
	Storage Unit	Backplane trace	**X kbps**	**File storage protocol size**		Every Phase, continuous. File structure may impact available storage
	Processing Units	Backplane trace	0	**X flight software size**	0	Every Phase, continuous. OS may impact available storage. (-55, 125 C operating temp, 10^1 kg mass for 1 OBC+mass memory)

Yellow: Assumptions Blue: Raw analog output Orange: Based on default values

Return

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Processing, Memory Calculations

	Phase 1					*No Payload use, negligible storage
	Phase 2: C&DH		13.51	20342.04365	69610.01	*STORAGE: Includes 1 full year of data following the conceptual PP outline (full year of health data too)
	Phase 2 Comms: 1 Week			643.884		*TELECOM: Maximum new data onboard over the course of 1 week in Phase 2 assuming Exoplanet campaign
Total	Phase 3					*Looking for ISO, negligible storage
	Phase 4					*OpNav, negligible storage
	Phase 5: C&DH		18.31	563.478912	85210.01	*STORAGE: full encounter data + FOS
	Phase 5: Comms			469.56576		*TELECOM: full encounter data
	Phase 6:					*N/A
	System-wide health, 1 week:			16.36992		
	System-wide health:			0.00081200		
				**includes C&DH factor of safety, used to size onboard storage		
				*does not include C&DH factor of safety, used to size Comms		

Yellow: Assumptions Blue: Raw analog output Orange: Based on default values

Return

Processing Speed Assumptions

- Highest required during encounter: 85.24 kHz
- Includes health sensors from each subsystem (usually 2 per component)
  - C&DH : 50Hz
    - 2x voltage/current
    - 2x thermocouples
  - Structures : 50Hz
    - 8x voltage/current
    - 8x thermocouples
  - ADCS : 50Hz
    - 20x voltage/current
    - 20x thermocouples
  - GNC : 50Hz
    - 2x voltage/current
    - 2x thermocouples
  - Comms : 50Hz
    - 10x voltage/current
    - 10x thermocouple
  - Power : 50Hz
    - 2x thermocouple
    - 1x PDU output (volt/current all subcomponents)
  - Propulsion : Variable Hz
    - 84x voltage/current (valve checking, 50 Hz)
    - 63x thermocouples (10 Hz)
    - 60x pressure transducers (1000 Hz)
  - Payload : 0.3Hz
    - 7x voltage/current
    - 7x thermocouples

Return

Processing Speed Assumptions

- Additional bandwidth comes from other subsystems during operation
- Payload:
  - SAR: 15.6 kHz
  - Visual/IR Imager: 35.34 Hz
- ADCS:
  - Star Tracker x2: 10 Hz
  - IMU x2: 200 Hz

Return

Wire Protocol Characteristics

Potential Protocols	Multidrop?	Data Rate (MB/s)	Standard cable	Mass per unit length (kg/m)	Cable diameter (in)	Operating Temperature Range (°C)	Waste heat/ other thermal properties
RS-422	No	1.25	EIA Industrial RS-422 PLTC/CM	0.036	0.23	-40 to 75	Negligible
SpaceWire	No	400	Full-duplex	0.085	0.281	-100 to 180	Negligible
1553B	Yes	1	Twinax	0.061	0.244	-20 to 60	Negligible
RS-485	Yes	1.25	EIA Industrial RS-485 PLTC/CM	0.143	0.03	-20 to 60	Negligible

Return

Wire Protocol Rationale

- ADCS
  - CMGs: 1553B, required
  - Star Tracker: 1553B, required
  - Sun Sensors & IMUs: RS-485, modified.
    - 6 discrete RS-422 connections required; included an RS-485 Arbitrator to convert RS-422 connections to one multidrop-capable RS-485 connection to simplify physical integration
- Structures
  - Thruster Actuators (x8), SADAs (x2), Array Hinges (x2), Release Mechanisms (x2): RS-485, selected.
    - Capable of most protocols except 1553B; selected the other multidrop protocol, RS-485, to reduce incoming wire connections from 14 discrete to one combined
- Power
  - PDU (x2): 1553B, required

Return

Wire Protocol Rationale

- <u>Propulsion</u>
  - Pressure Transducers (x60): 1553B, selected.
    - Each transducer is equipped with a miniature ADC to produces a digital signal. To avoid 60 discrete connections, the signals are routed into a 1553B connection, reading each transducer at 1000 Hz intervals (requested by Propulsion)
  - Fuel Valves (x?): no protocol, digital, required.
    - Valves operate via PWM; C&DH sets a digital duty cycle and, when "HIGH", the valves open, allowing fuel flow and thrust, and when "LOW", close the valves.
- <u>Health Sensors</u>, 1553B, selected.
    - All sensors are analog signals individually connected to ADCs
    - All sensors except thermal sensors on Payload, ADCS, GNC are routed to C&DH via 1553B to take advantage of low data rate and multidrop capabilities
    - Thermal sensors on Payload, ADCS, and GNC are looped internally and controlled via subsystem-specific preexisting processors already in use. Thermal data is included in science/regular data downstream. Mechanical thermostat keeps them in survival temperature range when not in use

Return

Wire Protocol Rationale

- Payload
  - Optical: SpaceWire, modified.
  - IR: SpaceWire, modified.
  - SAR: SpaceWire, modified.
    - All payload instruments output analog signals of varying packet size and data rates. Each is equipped with an ADC and then passed into a SpaceWire formatted wire, capable of very high (400 MB/s) data rates and of variable packet sizes. SpaceWire is a discrete connection.
- Telecom
  - Transponder (x2): SpaceWire, selected.
    - To avoid wire data rates becoming a limiting factor to uplink/downlinking, SpaceWire was selected (400 MB/s). SpaceWire is also capable of varying its packet size, and can follow any desired Telecom turbocode encoding scheme.

Return

Duration Budget: PP Orbit

PP Orbit (1 Year Plan)		
Within Duration Budget		
Arrival Date:	12/31/2030	
DSN contact frequency (/week):	1	
DSN contact duration (hours)*:	0.5	
Num. of remote obs. campaigns:	1	
Remote obs. Image cadence (/hr):	1	
Duration of Remote Obs. Campaign (months):	6	*GS, 4/19, Alexi D.
Num. of exoplanet obs. campaigns:	6	*GS, 4/19, Alexi D.
Duration of Exoplanet Obs. Campaign (weeks):	4	*SS, Jon H. 4/19
Exoplanet Image cadence (/hr):	2	*SS, Jon H. 4/19
Default Health Sensor Frequency (Hz):	50	*SS, Jon H. 4/19
Default # of Channels (x2 redundancy, health sensor):	10	*Adjusted to fill remaining time
Default Health Data Size (bytes):	4	*SS, Jon H. 4/19
Default Data Rate (MB/s):	0.002	*SS, Jon H. 4/19
Default Storage Rate (Hz):	0.033333333	*Jon H 4/26, for sizing
Stored Health Data (#):	20160	*Jon H 4/26, for sizing
		*Float, bytes assumption 4/26
Total Duration (days):	349.0833333	*Calculated from above
Available time (days):	15.91666667	*Assumed stored health data every 30 seconds, Jon H 4/26
*NOTE: duration is purely communication downlink, does NOT		*assumed # of health data points to downlink for each system if health data is stored every 30 seconds

Return

Duration Budget: Encounter

Encounter Timeline		
Within Duration Budget		
Total Allowed Encounter Duration (min):	9.2674	*5/17, Liam
Visual Imager Time (min):	0.19	* calc from below
IR Imager Time (min):	0.20	* calc from below
SAR Operation Duration (min):	1.7324	*5/17, Liam
Required Duration (min):	0.39	
Imager Operation Duration (min):	0.39	*calculated from below assuming image taken does not overlap with image readout
Time in IR Imager Operational Range (min):	4.67	*5/17, Liam
Time in Visual Imager Operational Range (min):	2.87	*5/17, Liam
Remaining Visual Imager Available Duration (min, total):	2.68	
Remaining IR Imager Available Duration (min, total):	4.47	
Remaining Available Duration (min, total):	8.8804	

Return

CAL POLY

Duration Budget: Assumptions

SAR # of Pulses:	41	***Matt Jones, 5/21**
Pulse Duration (sec):	1.63	***Matt Jones, 5/21**
Visual Image Num During Encounter:	60.00	*Lauren F, worst case from Encounter Meeting 4/26
IR Image Num During Encounter:	60.00	*Lauren F, worst case from Encounter Meeting 4/26
Visual Smallest Exp. Time (sec):	0.013	***PDR slides 5/24, formerly 0.023**
IR Smallest Exp. Time (sec):	0.00	***Matt Jones, 5/5**
Visual Readout Rate (sec):	0.1870	***PDR slides 5/24; 200 ms total, so -exposure time for readout rate. Formerly 0.0283**
IR Readout Rate (sec):	0.2000	***PDR slides 5/24**
IR Image size:	0.40	*MB, 4/19 Colleen
Visual Image Size:	1.87	*MB, 4/19 Colleen
C&DH Data Factor of Safety:	1.20	*4/19 Solomon D., value over expected maximum storage required

Return

C&DH Requirements: Level 3

Level	ID	Parent	Requirement	Rationale
3	CDH1	F1, F30	The C&DH system shall be capable of responding to commands received from ground.	The C&DH subsystem will be responsible for interpreting commands from Earth and controlling the necessary functions of each subsystem as a result. The system must be capable of receiving and immediately implementing commands for situations where immediate action is necessary.
3	CDH2	F9, F11, F12, F13, F14, F15, F16	The C&DH system shall be capable of storing commands for delayed execution.	The C&DH subsystem will be responsible for interpreting commands from Earth and controlling the necessary functions of each subsystem as a result. The system must be capable of storing these received commands and executing a sequence at a designated later time (like imaging campaign).
3	CDH3	F17, F20	The C&DH system shall store data until receipt is confirmed by the ground system.	The flight system will need to verify that data has been received by ground before deleting from storage.
3	CDH4	F3, F5, F15, F16, F20, F31, F33	The C&DH system shall decode data received from the ground system.	C&DH Comms cards will be responsible for decoding data received from ground and turning it into command form that can be sent to C&DH.
3	CDH5	F3, F5, F17, F18, F19, F21, F28	The C&DH system shall encode data prior to transmission to the ground system.	C&DH Comms cards will be responsible for encoding data sent to ground.

C&DH Requirements: Level 3

Level	ID	Parent	Requirement	Rationale
3	CDH6	F17	The C&DH system shall track elapsed time with an accuracy of +/- 1 millisecond.	Commands will need to be executed at a designated time. Additionally, the spacecraft will need to propagate position data onboard. In order to this, the C&DH system will need to accurately track elapsed time. The accuracy of +/- 1 millisecond was estimated with information from the deep space communications lecture.
3	CDH7	F11, F12, F13, F14	The C&DH system shall have a storage capacity of 21 GB [TBC].	The C&DH system needs to store science and telemetry data collected until there is a communication opportunity. The data storage was sized to accommodate several varieties of pre-positioned orbit observational campaigns
3	CDH8	F9, F11-F21, F28, F30, F31, F33	The C&DH system shall support the ADCS subsystem flight algorithms.	The C&DH subsystem will be responsible for supporting the ADCS components. The specific components are outlined in the Data Definition Presentation.
3	CDH9	F3, F5, F9, F11-F21, F28, F30, F31, F33	The C&DH system shall support of the ADCS subsystem file format protocols.	
3	CDH10	F10, F17-F22, F28, F30, F31, F33	The C&DH system shall support the Communications subsystem flight algorithms.	The C&DH subsystem will be responsible for supporting the Telecom components. The specific components are outlined in the Data Definition Presentation.

C&DH Requirements: Level 3

Level	ID	Parent	Requirement	Rationale
3	CDH11	F10, F17-F22, F28, F30, F31, F33	The C&DH system shall support the Communications subsystem file format protocols.	
3	CDH12	F9, F15, F16	The C&DH system shall support the GNC subsystem flight algorithms.	The C&DH subsystem will be responsible for supporting the GNC components. The specific components are outlined in the Data Definition Presentation.
3	CDH13	F9, F15, F16	The C&DH system shall support the GNC subsystem file format protocols.	
3	CDH14	F9, F11-F14, F24, F25, F27, F29	The C&DH system shall support the Payload subsystem flight algorithms.	The C&DH subsystem will be responsible for supporting the Payload components. The specific components are outlined in the Data Definition Presentation.
3	CDH15	F9, F11-F14, F24, F25, F27, F29	The C&DH system shall support the Payload subsystem file format protocols.	

C&DH Requirements: Level 3

Level	ID	Parent	Requirement	Rationale
3	CDH16	F5	The C&DH system shall support the Power subsystem flight algorithms.	The C&DH subsystem will be responsible for supporting the Power components. Power is required for the spacecraft to operate in space for 22 years. The specific components are outlined in the Data Definition Presentation.
3	CDH17	F5	The C&DH system shall support the Power subsystem file format protocols.	
3	CDH18	F15, F16	The C&DH system shall support the Propulsion subsystem flight algorithms.	The C&DH subsystem will be responsible for supporting the Propulsion components. The specific components are outlined in the Data Definition Presentation.
3	CDH19	F15, F16	The C&DH system shall support the Propulsion subsystem file format protocols.	
3	CDH20	F5	The C&DH system shall support the Structures subsystem flight algorithms.	The C&DH subsystem will be responsible for supporting the Structures components. The specific components are outlined in the Data Definition Presentation.

C&DH Requirements: Level 3

Level	ID	Parent	Requirement	Rationale
3	CDH21	F5	The C&DH system shall support the Structures subsystem file format protocols.	The C&DH subsystem will be responsible for supporting the Structures components. The specific components are outlined in the Data Definition Presentation.
3	CDH22	F3	The C&DH system shall have a minimum reliability of 99.8%.	In order to acheive an 80% likelihood of reaching and identifying at least one ISO in 20 years, the flight system must have an overall reliability of at least 90%. The breakdown of this reliability into the individual reliabilities of subsystems is described here.
3	CDH23	F5	The C&DH system shall operate in the space environment for 22 years.	The flight system will be operating in the space environment for at least 22 years, therefore the C&DH subsystem must operate in this environment for the listed time period.
3	CDH24	F10	The C&DH system shall operate at a heliocentric range of 0.5-7 AU.	The flight system will be at a range of 0.5 to 7AU from the Sun, therefore the C&DH subsystem must be able to operate in this environment.

C&DH Requirements: Level 4

Level	ID	Parent	Requirement	Rationale
4	CDH-C1	CDH3	The data storage shall store science data collected from the ISO.	Long-term hard storage holds flight system science data for downlink to ground
4	CDH-C2	CDH3	The data storage shall store flight system telemetry data.	Long-term hard storage holds flight system telemetry for downlink to ground
4	CDH-C3	CDH2	The data storage shall store commands from the ground system.	To hold commands for long-term delayed execution, they are stored in the MMC instead of in flash memory
4	CDH-C4	CDH1, CHD2	The processors shall employ flight software capable of handling ground system commands.	Created FS for OBC interprets received GS commands
4	CDH-C5	CDH20	The 1553B interface card shall provide wire protocol to SpaceWire backplane protocol.	Provide wire protocol to backplane protocol interface for each required protocol

C&DH Requirements: Level 4

Level	ID	Parent	Requirement	Rationale
4	CDH-C6	CDH9, CDH13	The RS-485 interface card shall provide wire protocol to SpaceWire backplane protocol.	Provide wire protocol to backplane protocol interface for each required protocol
4	CDH-C7	CDH15	The SpaceWire interface card shall provide wire protocol to SpaceWire backplane protocol.	Provide wire protocol to backplane protocol interface for each required protocol
4	CDH-C8	CDH4	The X-band communication card shall decode uplinked commands.	DSN uplinks in X-band to the HGA and requires turbo-decoding
4	CDH-C9	CDH5	The X-band and Ka-band communications cards shall encode downlinked commands/telemetry.	DSN downlinks in X and Ka-bands and requires turbo encoding
4	CDH-C10	CHD27	The power card shall regulate power into the C&DH subsystem.	Power regulation is necessary for efficient OBC operation.

C&DH Requirements: Level 4

Level	ID	Parent	Requirement	Rationale
4	CDH-C11	CDH12	The processor card shall have at least 5 MB of RAM.	GNC's APIC optical autoNav processing card requires 5 MB of RAM to run.
4	CDH-C12	CDH8-21	The processor card shall handle at least 85.21 kHz bandwidth.	Full system requirement
4	CDH-C13	CDH22	The C&DH subsystem shall have a redundant copy of itself.	To meet reliability requirements, there will be two separate C&DH subsystems capable of taking over for the other in the event of a fault.
4	CDH-C14	CDH22	Each C&DH subsystem shall have a redundant mass memory card.	To meet reliability requirements, there will be two separate mass memory cards in each C&DH system. When storing data, data will be written to each card and checked for errors when downlinking.

C&DH Risks

- Given that there is an unknown distribution and direction of energetic particles from a solar particle event or CME there is a possibility of the flight system being impacted by energetic particles adversely impacting the memory storage device and processor card, which can result in a loss or change in data stored in the memory device and the processor card being unable to properly process data.

Structures

Matthew Slymen   Ismael Castorena   Ricardo Contreras   Anthony Garcia   Saul Portillo   Edgar Yanez

Structures Risks

- Given that the mechanism to deploy the instrumentation does not deploy or gets jammed there is a possibility of the mechanisms not deploying adversely impacting any component that is gimbaled or deployed, which could result in the spacecraft not collecting data, power, or not directing its thrust.

- Given that we can come across unanticipated launch loads or launch environments there is a possibility of the spacecraft shaking more than anticipated adversely impacting the entire spacecraft, which can result in a structural integrity compromise or component failure.

  Likelihood: 1

  Severity: 2

Mechanisms - Structures

Mechanism	Source	Mass	Continuous Draw	Actuation Draw	Actuation Type	Range	Resolution	Operating Temperatures
SADA (x2)	MOOG High Power Type 5	40kg	<20W	20W	Motor	+/-179 deg.	+/-0.02 deg.	-50 C to +70 C
Thruster Ring Actuator (x8)	MOOG 310	16kg	<28VDC	28VDC	Motion translation (rotational to linear)	2cm stroke	Future Work	-50 C to 80 C
Roll Out Solar Arrays (ROSAs, x2)	DSS ROSA	600 kg	N/A	N/A	Roll out	N/A	N/A	-65 C to +90 C
Non-Explosive Actuator for ROSA (x2)	EBAD NEA HDRM	4.3kg	250 mA	4 A (release current)	Hold Down and Release	N/A	N/A	-240 C to +135 C
Deployment Hinge for ROSA (x2)	Deployment System for Large Appendages	3kg	N/A	N/A	Spring driven	90-180 deg.	+/- .006 deg	-30C to +50 C (Survivable temperatures +/-150 C)
Non-Explosive Actuator for SAR	NEA Model 9100	0.7kg	250 mA	4A	Hold Down and Release	N/A	N/A	-135C to +135C
Deployment Hinge for SAR	Deployment System for Large Appendages	1.5kg	N/A	N/A	Spring driven	90-180 deg.	+/- .006 deg	-30C to +50 C (Survivable temperatures +/-150 C)
APIC Gimbal	JPL	<5kg	<12W	<12W	Elevation Actuator	+/-90	19 microrad	Actively controlled
Primary Imager Lens Cap	Future Work							

Total Mechanisms: approx. 20

SADAs

Model Number: MOOG High Power Type 5

Source: MOOG High Power Type 5

Notes:

- Suited for 9kW through the system, scaled SMaP values x2 to accommodate for further mechanism development
- Mechanisms – Structures

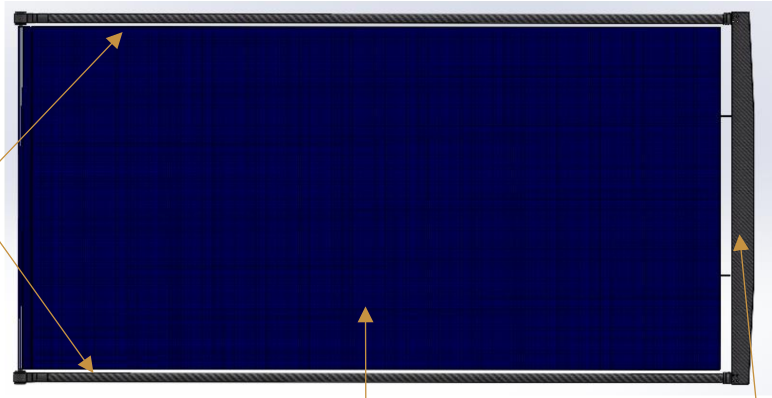
ROSAs

Model Number: Deployable Space Systems ROSA

Source: Deployable Space Systems ROSA
Notes:

- Roll out mechanism is the carbon fiber slit tubes on both sides (uses the tubes stored strain energy)

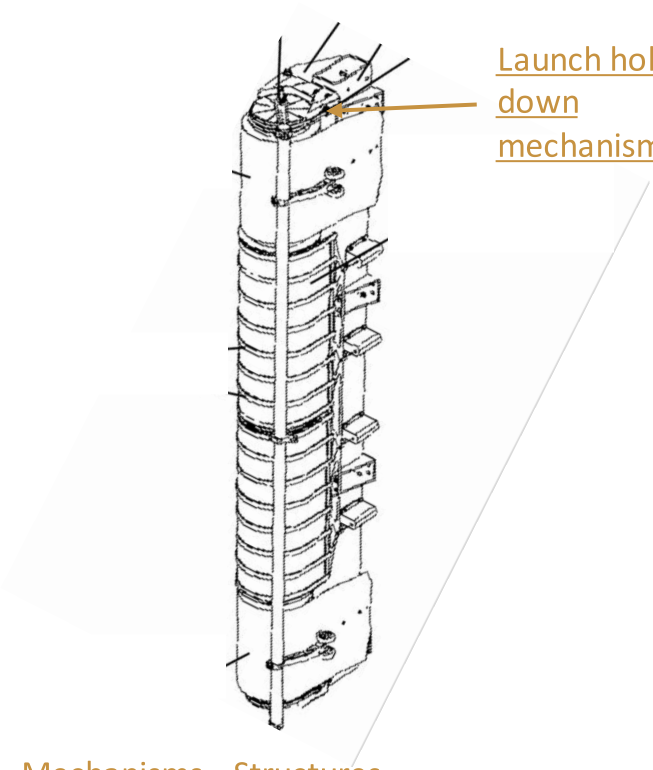
Carbon Fiber slit tubes



Solar Array

Spacecraft attachment

Launch hold-down mechanism



Mechanisms - Structures

Thruster Ring Linear Actuators

Model Number: MOOG 310 Linear Actuator

Source: MOOG 310

Notes:

- Two actuators needed for control of each of four thrusters for total of eight actuators
- Mechanisms - Structures

Non-Explosive Actuator for ROSA: Hold Down and Release Mechanism

Model Number: EBAD NEA HDRM

Source: EBADNEAHDRM

Notes:

- Allows for 6 degree cone of misalignment
- <50 ms release time
- 195 kN release load rating
- Mechanisms - Structures

Deployment Hinge for ROSA

Model Number: DESY

Source: DESY(Deployment System for Large Appendages)

Notes:

- Spring Driven, High Position accuracy and stiffness in deployed configuration
- Used for deployment of large appendages such as solar arrays & antennas
    - Mechanisms - Structures

Non-Explosive Actuator for ISAR

Model Number: EBAD NEA 9100 HDRM

Source: EBAD NEA HDRM

Notes:

- Allows for 6 degree cone of misalignment
- <50 ms release time
- 7.6 kN release load rating
- Mechanisms - Structures

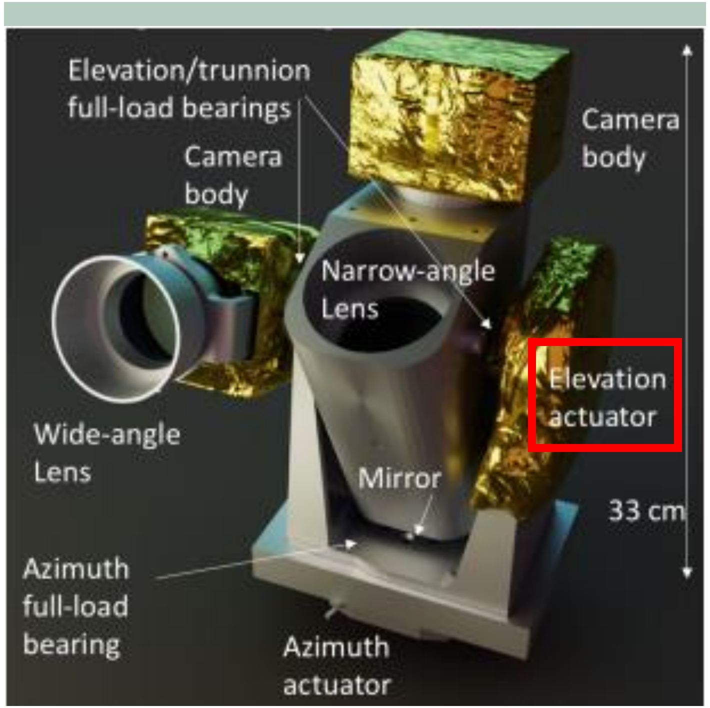
APIC Mechanism

Model Number: Advanced Pointing Imaging Camera Concept

Source: APIC

Notes:

- Comes pre-installed in APIC assembly, not a mechanism we need to design
- See GNC for more information
- Mechanisms - Structures

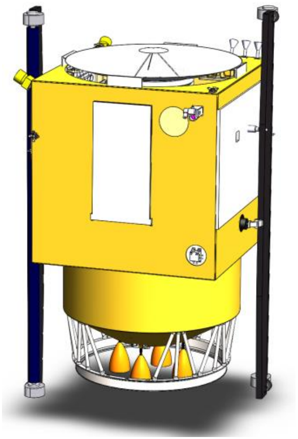


Driving Configuration Considerations

1. Adaptability
   1. Spacecraft must be able to point in a variety of directions for different ISO cases
2. Minimum Structural Mass
   1. Configuration must allow for small structural mass for maximum fuel
3. Large Tanks and High Gain Antenna
4. High Power Requirement

Phase Timeline 0 thru 1

Phase 0: PreLaunch



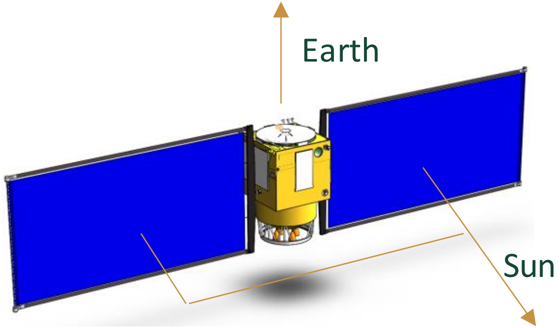
Primary Points

- N/A

Mechanisms in use

- N/A

Phase 1: Launch to Separation



Primary Points

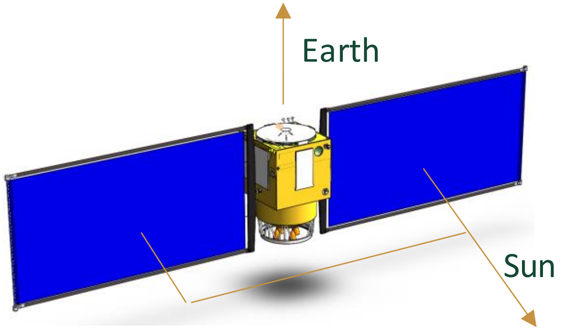
- HGA – Pointed to Earth
- Solar Array – Pointed to Sun

Mechanisms in use

- Solar Drive Mechanism

Phase Timeline 6 thru 7

Phase 6: Downlink Data



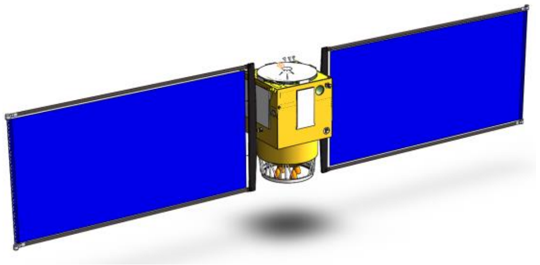
Primary Points

- HGA – Pointed to Earth
- MROSAs – Pointed to Sun

Mechanisms in use

- Solar Drive Mechanism

Phase 7: EOL



Primary Points

- N/A

Mechanisms in use

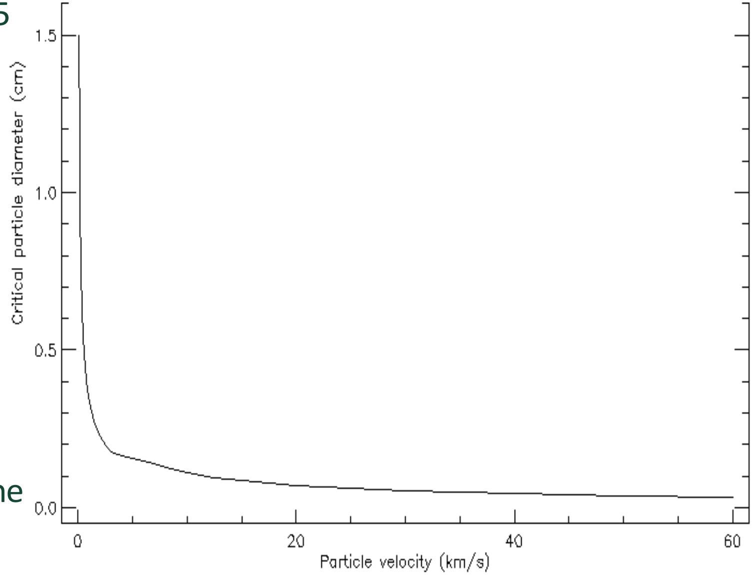
- N/A

MMOD Shielding Study

Link to Main

MMOD shielding consist of aluminum honeycomb sandwich panel and MLI shielding.

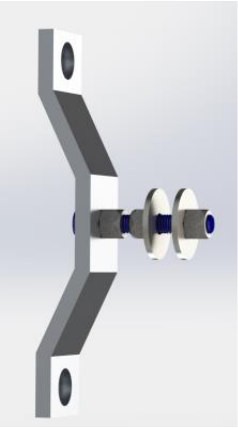
- The shield dimensions are 1 mm wall thickness on each side with a 5 cm honeycomb core. This shield can protect against particles with a critical diameter up to 1.5 cm.
- Grunmodel and MEM3 was used to determine MMOD flux environment. That then determines the probability of particles impacting that could cause failure.
- At the 1 AU prepositioned orbit we have a 98.38% probability of avoiding impacts of particles with a diameter of +0.2 cm
- During an Earth fly by we have a 99.34% probability of avoiding impacts of particles or orbital debris with diameters of +0.032 cm
- To determine if shielding can protect against the coma or tail of an ISO, model information on 2I/Borisov was used. For most studies, the particles diameters ranged from 1E-3 cm to .4 cm with the smaller particles reaching velocities up to 50 m/s



Link to Main

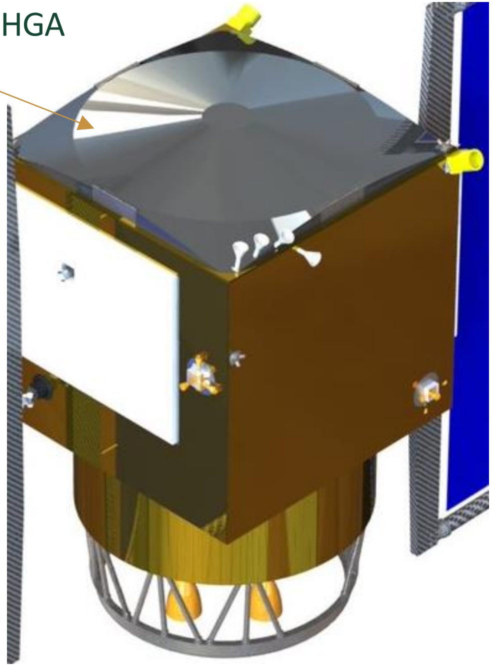
MLI Shielding Study

- The MLI shielding is a 15 mm thick, 30 layer blanket consisting of aluminized Mylar and tulle
  - The packing density chosen is 30 layers/15 mm due to the decrease in conductive shorting with less dense packing
  - The tulle is used as a spacer to prevent shorting as well
  - The decision to use this configuration is based on experimental studies in which number of layers and packing density were varied and a NASA study which saw diminishing returns past 20 layers
  - The heat flux through the 30-layer blanket was about .07 W/m^2 less than the 20-layer blanket
- Due to the spacecraft's varied orientation, the entire spacecraft bus will be covered in MLI including the HGA which will be covered in a Germanium coated polyimide, which is RF transparent



MLI mounting bracket

Germanium coated kapton MLI for HGA



Structural Factor of Safety Table

Return to Main

Table S1

Factor of Safety per Material						
Material Type	Verification Approach	Additional Considerations	Ultimate Design Factor	Yield Design Factor	Qualification Test Factor	Proof Test Factor
Metallic	Prototype	N/A	1.4	1.0[1]	1.4	N/A or 1.05[2]
	Protoflight	N/A	1.4	1.25	1.2	N/A or 1.05[2]
Composite/Bonded	Prototype	Discontinuity Area Geometry	2.0[3]	N/A	1.4	1.05
		Uniform Geometry	1.4	N/A	1.4	1.05
	Protoflight	Discontinuity Area Geometry	2.0[3]	N/A	1.2	1.2
		Uniform Geometry	1.5	N/A	1.2	1.2
Glass/Ceramics	Proof Test	Nonpressurized Loading Condition	3	N/A	N/A	1.2
		Pressurized Loading Condition	3	N/A	N/A	2
	Analysis Only[4]	Nonpressurized Loading Condition	5	N/A	N/A	N/A
Bonds in Glass/Ceramics	Proof Test	Nonpressurized Loading Condition	1.5	N/A	N/A	1.2
		Pressurized Loading Condition	3	N/A	N/A	2
Softgoods	Prototype and Proof Tests	Loss of Life or Vehicle Criticality	4	N/A	4	1.2
		Any Other Criticality	2	N/A	2	1.2
Beryllium	Proof Test	None	1.6	1.4	N/A	1.2

[1]Structure has to be assessed to prevent detrimental yielding during its design service life, acceptance, or proof testing.

[2]Propellant tanks and SRM cases only.

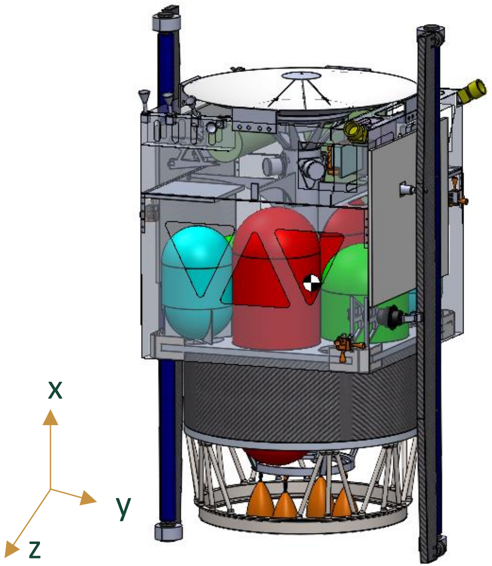
[3]Factor applies to concentrated stresses. For nonsafety-critical applications, this factor may be reduced to 1.4 for prototype structures and 1.5 for protoflight structures.

[4]Not applicable to ceramic structures.

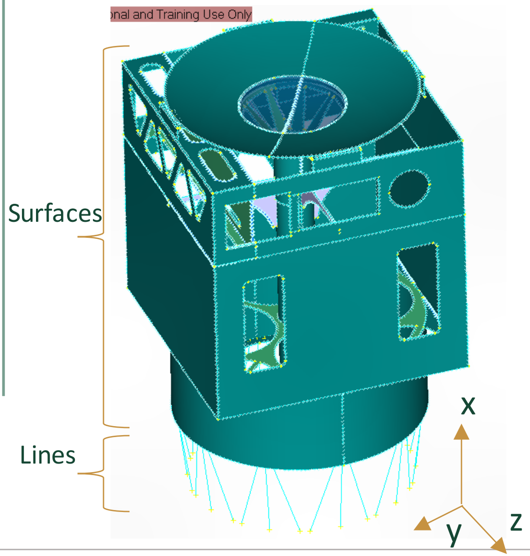
Link to Main Analysis Slides

Structures Flow

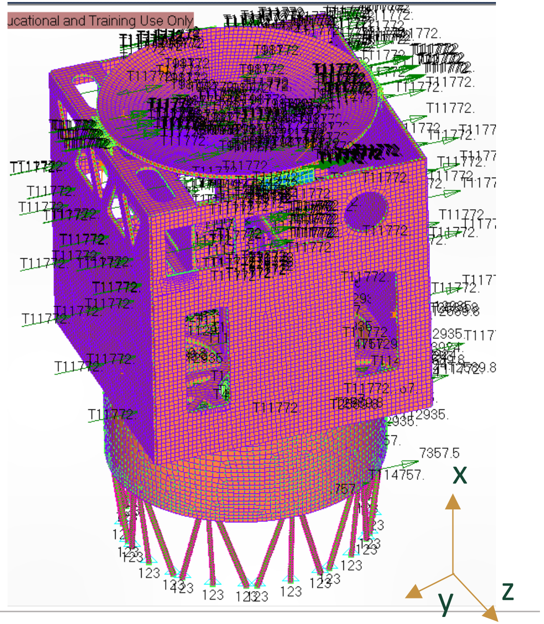
CAD (Solid Features)



Geometry Pre-Processing

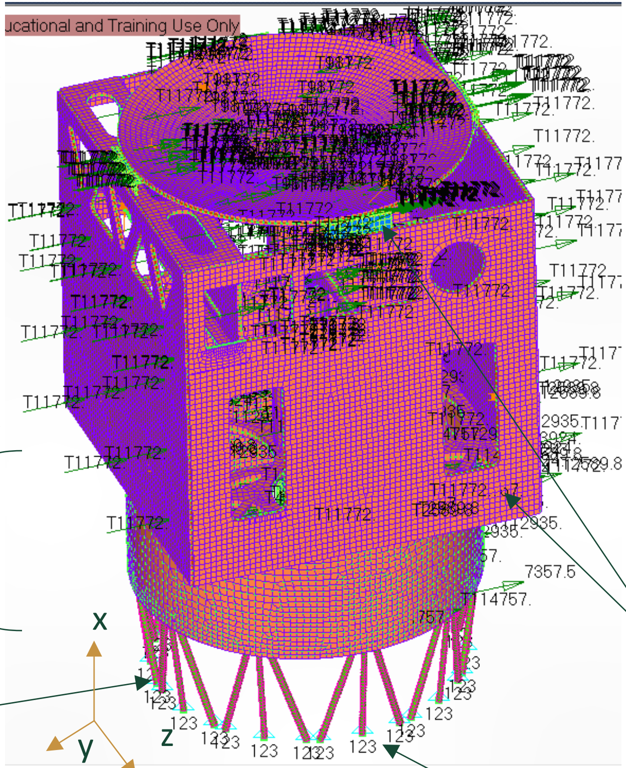


Meshing and BC



Link to Main Analysis Slides

FEM



Upper Nonmanifold Body, shared edge nodes

Lower Nonmanifold Body, shared edge nodes

Beam Elements

Pinned Base

Bracketry to connect regions with contact (glue) segments

Materials:

- Various thicknesses of 7075Al Facesheets/Hexcel Al Honeycomb Core
- Various thickness of 7075Al Plates
- Various Thicknesses of Ti-6Al-4V (Bracketry and Beams)

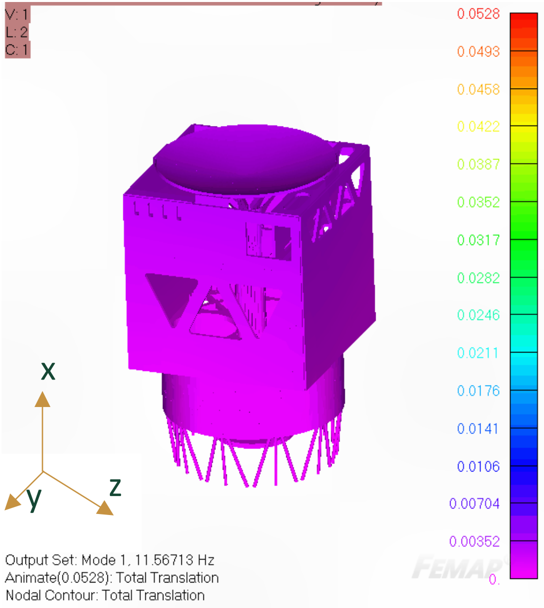
Loads:

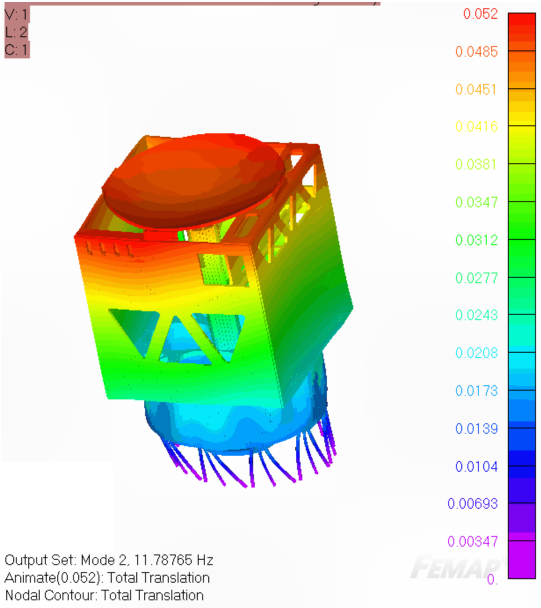
Load Definitions
- 1..HGA
- 2..InstrumentPlate
- 3..Thrusters
- 4..ROSAs
- 5..UpperTankPlate
- 6..FuelTanks
- 7..OxTanks
- 8..PressurantTanks
- 9..LowerTankPlate
- 10..UpperStructureMass (Approx)

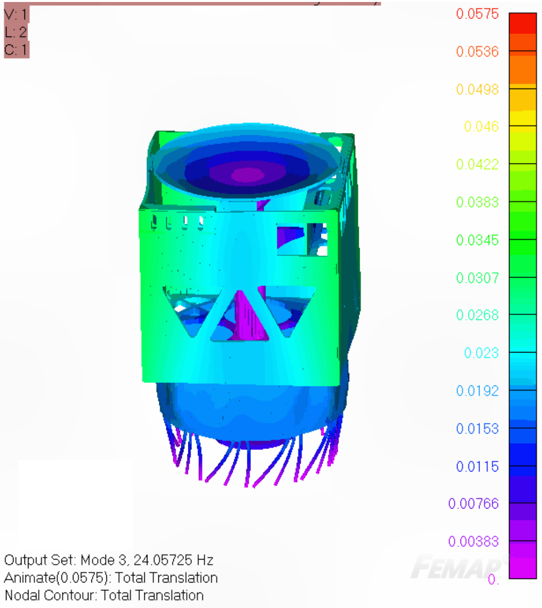
Remaining "Unaccounted for" mass
(Primarily Structural mass)

Link to Main Analysis Slides

Modal Results (2) <35Hz (Open Issue)







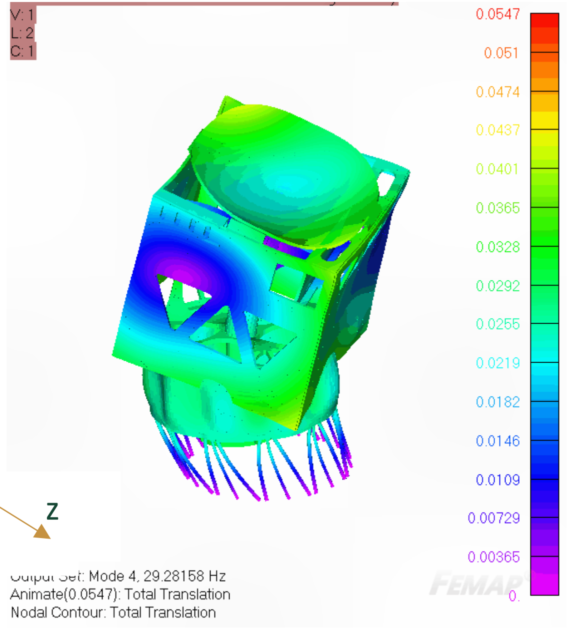
Mode 1: 11.5Hz
Max Translation: 5.3cm
LV Limit: <7cm

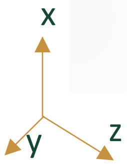
Mode 2: 11.8Hz
Max Translation: 5.2cm
LV Limit: <7cm

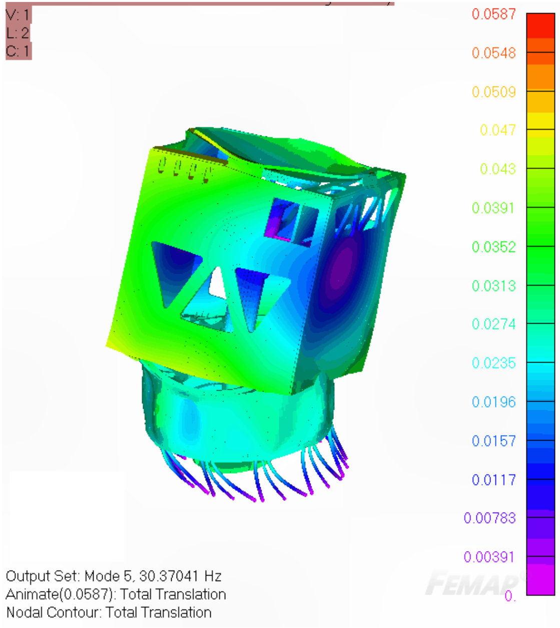
Mode 3: 24.1Hz
Max Translation: 5.8cm
LV Limit: <7cm

Link to Main Analysis Slides

Modal Results (3) <35Hz (Open Issue)







Mode 4: 29.3Hz
Max Translation: 5.5cm
Required: 7cm?

Mode 5: 30.4Hz
Max Translation: 5.9cm
Required: 7cm?

Link to Main Analysis Slides

Analysis Next Steps

Post-PDR

- Finalize Modal Analysis
- Random Vibration
- Structural Optimization
- Further Bracketry for instrumentation
- Vibration dampening systems for subsystems
- Operational Loading Conditions
  1. Thrusters On
  2. Deployment Loads
  3. Control Torques

Link to Main Analysis Slides

Analysis Summary

Static Loading

Load Case	Min/Max	Location
6G Axial Stress	FOS = 1	Upper Payload Box
6G Axial Translation	Deflection = 2.9cm	Center Column, upward
2G Lateral Stress	FOS = 1.25	LV Attachment Tubes
2G Lateral Translation	Deflection = 3.9cm	Top Corner, sideways

Ready for Refinement

Modal Analysis

Mode	Resonant Frequency	Max Translation
1	11.5Hz	5.3cm
2	11.8Hz	5.2cm
3	24.1Hz	5.8cm
4	29.3Hz	5.5cm
5	30.4Hz	5.9cm

Further Design Required

Power

Karen Llacsa

Joseph Poncini

Ethan Tran

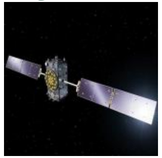
Power Risks

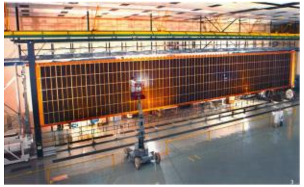
- Given that electronic arcing occurs there is a possibility of electric failure adversely impacting the number of solar cells we have in series, which can result in reduced power availability.

  Likelihood: 2

  Severity: 2

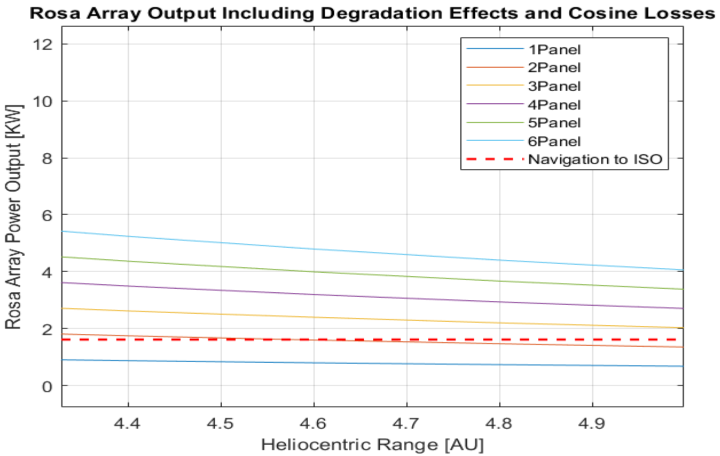
Solar Array and Batteries





Solar Array Design:

- Sizing for ROSA panel dependent on most power sensitive portions of the power budget. Model is slightly over conservative on degradation rate (~1.25% yr) for expected 23.75 yr lifetime and average 10 deg cosine loss.
- Shifted panel design to a different heavier variant of the ROSA solar panels. More accurate density and improved cell efficiency compared to previous ROSA panel.
  - For full chem prop system investigated panel solutions between 2-6 panels.
- Temperature degradation assumed most impactful for 1 AU prepositioned phase.
  - Investigating panel orientation during preposition to reduced sun facing area to aid station keeping and thermal loads



Battery Design:

- Time from initial launch separation to array deployment is taken to be about 2 hours (This includes detumble and ROSA roll out)

  At ~28 V bus voltage thus expecting a min battery capacity of 64.035 Ah

- Previous optimized battery had about 255 Ah capability and 5p10s configuration. Need for larger sized battery constrained by science collection phase and downlink at Au ranges past 4.

  ~8324 Wh new battery capacity after applying 8% degradation factor and assumption that 1 open parallel cell leaves 10 cells inactive.

Table 1. New ROSA Variant Array Specifications

Size Deployed (m2)	Size Stowed (m^3)	Mass (kg)	Specific Power (W/kg)	BOL Power @ 1 AU (kW)
156.25	~1.14	167.59	273.14	45.776

Table 2. Initial Battery Specifications

Battery (Li-Ion)	Size (m^3)	Mass (kg)	Capacity (Wh)	Cell Configuration
SAFT VL51ES	0.06892	80.4	10,452	6p10s

Max Power per Phase

Phase	Maximum Power Draw (W)	Distance from the Sun (AU)	Max Solar Panel Power (W)	Max Angle from sun, θ (°)
2	753	1	34835	88
3	519	1	34835	88
4	602	5	1393	64
5	753	5	1393	65
6	887	7	710	0
7	514	7	710	36
SF	304	1-7	> 710	< 65

Phase 2: Power Budget Summary

Subphases	Duration (min)	AU	Power Draw (W)	ROSAS Deployed	Solar Power (W)	Initial Battery Power (W-hr)	Final Battery Power (W-hr)
2.1	1	1	316	0	0	8324	8319
2.2	30	1	696	0	0	8319	7971
2.3	5	1	543	0	0	7971	7926
2.4	5	1	583	2	34836	7926	8324
2.5	120	1	503	2	34836	8324	8324
2.6	60	1	753	2	34836	8324	8324
2.7	1051200	1	503	2	34836	8324	8324
2.8	60	1	753	2	34836	8324	8324

Phase 3: Power Budget Summary

Subphases	Duration (min)	AU	Power Draw (W)	ROSAS Deployed	Solar Power (W)	Initial Battery Power (W-hr)	Final Battery Power (W-hr)
3.1	10	1	503	2	34836	8324	8324
3.2	5256000	1	519	2	34836	8324	8324
3.3	30	1	519	2	34836	8324	8324

Phase 4: Power Budget Summary

Subphases	Duration (min)	AU	Power Draw (W)	ROSAS Deployed	Solar Power (W)	Initial Battery Power (W-hr)	Final Battery Power (W-hr)
4.1	60	1	603	2	34836	8324	8324
4.2	60	3	603	2	3871	8324	8324
4.3	259200	3	503	2	3871	8324	8324
4.4	60	5	753	2	1394	8324	8324
4.5	60	5	519	2	1394	8324	8324
4.6	30	5	519	2	1394	8324	8324
4.7	60	5	531	2	1394	8324	8324

Phase 5: Power Budget Summary

Subphases	Duration (min)	AU	Power Draw (W)	ROSAS Deployed	Solar Power (W)	Initial Battery Power (W-hr)	Final Battery Power (W-hr)
5.1	3	5	530	2	1393	8324	8324
5.2	5	5	503	2	1393	8324	8324
5.3	5	5	503	2	1393	8324	8324

Phase 6: Power Budget Summary

Subphases	Duration (min)	AU	Power Draw (W)	ROSAS Deployed	Solar Power (W)	Initial Battery Power (W-hr)	Final Battery Power (W-hr)
6.1	10	7	503	2	710	8324	8324
6.2	480	7	887	2	710	8324	6917
6.3	960	7	503	2	710	6917	8324

Phase 7: Power Budget Summary

Subphases	Duration (min)	AU	Power Draw (W)	ROSAS Deployed	Solar Power (W)	Initial Battery Power (W-hr)	Final Battery Power (W-hr)
7.1	7200	7	514	2	710	8324	8324

Fault Mode: Power Budget Summary

Duration (min)	AU	Power Draw (W)	ROSAS Deployed	Solar Power (W)	Initial Battery Power (W-hr)	Final Battery Power (W-hr)
14400	7	304	2	710	8324	8324

Detumble

Table 2.0 Power Table

Phase	2	3	4	5	6	7	FM
Subsystem	**Allocated Power (W)**						
Thermal	201	201	201	201	201	201	201
ADCS	186	186	186	186	186	186	38
C&DH	100	100	100	100	100	100	0
Comms	16	16	16	16	400	16	16
GNC	0	0	12	12	12	12	0
Structures	164	84	84	84	84	0	0
Propulsion	200	0	200	0	0	0	0
Payload	0	25	25	25	0	0	0
Total:	867	612	824	624	983	515	255

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Sci Tech

Matthew Jones

Lauren Fukaye

Michael Limotta

Sci Tech Risks

- Given that there are uncertainties about the backscatter power there is a possibility of the shape model not considering the measurement uncertainty or measurement precision in the backscatter power adversely impacting the post-processing necessary to generate the shape model, which can result in the shape model not being able to be known to the required resolution if the backscatter power is not able to be measured with a high enough precision and low enough certainly.

- Given that there are unknowns about the ISO's shape there is a possibility of the ISO not being spherical as predicted adversely impacting the post-processing of the data and the shape model, which can result in an ISO shape which could reduce the coverage that we could get on our model, leading to an unmet requirement.

- Given that the shape model resolution may decrease due to time delay measurement uncertainty there is a possibility of the model predicting higher resolutions than are possible adversely impacting the post-processing necessary to generate the shape model, which can result in the final shape model not meeting the resolution requirement.

SAR Shape Model – Governing Equations

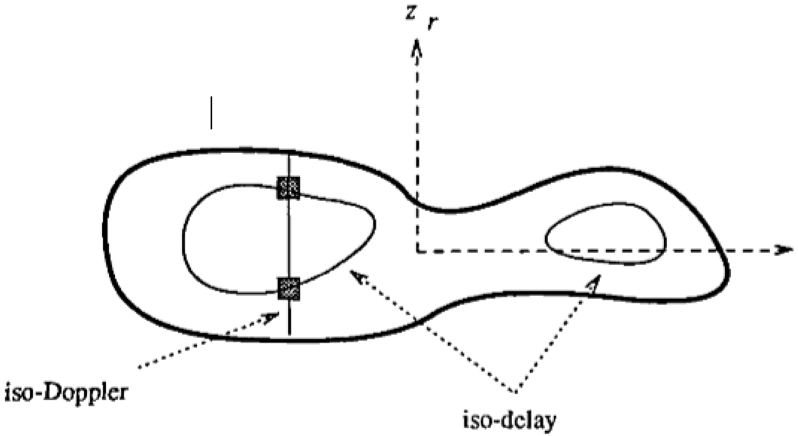
Doppler Equation:

$$\nu = -(2/\lambda)(dR/dt)$$

Delay Equation:

$$\tau = 2R/c$$

- R is the distance from the radar to a point on the ISO surface
- dR/dt is the slant range velocity between the radar and a point on the ISO surface
- Lambda is the radar signal wavelength
- c is the speed of light



Equations and Image From: Scott Hudson (1994) Three-dimensional reconstruction of asteroids from radar observations, Remote Sensing Reviews, 8:1-3, 195-203, DOI: 10.1080/02757259309532195

Model Assumptions

SAR Shape Model – General Process

- Choose a point on the ISO surface
- Calculate the expected delay and doppler values for a signal that came from that point
- Move to points 10 meters away in two specified directions (e.g., north and west)
- Calculate the expected delay and doppler values for a signal that came from those points
- Take the difference between the original delay and doppler values and those of the nearby points
- Calculate the total ISO surface area from which the data returns (assumed 50%)
- If the uncertainty in the doppler measurements is larger than the difference in doppler between either of the two points, then the point is unable to be discerned to the +/- 10 meter resolution. The associated ISO surface area is subtracted from the total area represented in the return signal.
- Given the measurement uncertainty, if the radar is unable to discern the difference in doppler required to maintain the +/- 10 meter resolution, the associated surface area is subtracted from the total area represented in the return signal.
- The remaining ISO surface area can be discerned to +/- 10 meters

Model Assumptions

Launch Services

Ryan Antoff **Chris Wang**

Electrical Connections to GSE on Launch Pad

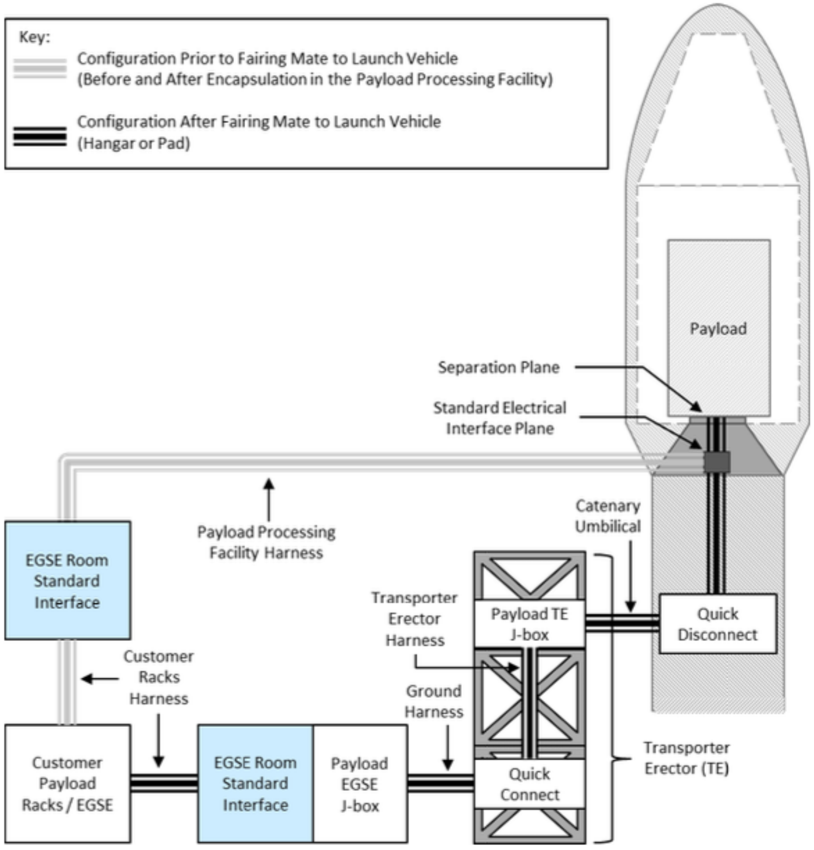


Figure 5-2: On-pad electrical interfaces

Source: Falcon Heavy Users Guide

Pre-Launch Risks – Fueling

Given that...	There is a possibility of...	Which can result in...	Likelihood (out of 5)	Severity (out of 5)	Sub team
The 3rd party fueling is unable to provide their services	No propellant being loaded into the spacecraft immediately before launch	A significant delay in the launch and lead to a delayed reaction readiness date	2	1 or 4	Launch Services

	1	2	3	4	5
Likelihood	0-20%	20-40%	40-60%	60-80%	80-100%
Severity	Minimal impact on schedule	Delay on some tasks not impacting overall schedule	Delay on some tasks minimally impacting schedule	Major slip in overall schedule	Critical milestones cannot be met

Customer Clarifications

Customer Clarifications

Assumptions Clarified with Customer:

- The number of follow up trajectory updates and timeliness needed to keep the ISO uncertainty low will be defined by our ephemeris uncertainty team, but **updates may be provided by the customer**.
- Our team is required to design the function and performance of ground system but may utilize existing ground assets that match the mission needs, e.g. DSN
- A space-base system will react to incoming ISOs determined by the customer:
  - ISO determination will be based on orbital analysis of detected ISOs which fall within a predefined on-board dV value and other factors deemed most important by our team upon further research.
  - Our team will define these values based on orbit models that assume a constant, average rate of **1 ISO detection per year for a 20-year period**, and with ISO orbital parameters given by the solicitation.

Schedule Breakdown

System Realization Schedule

Main Slide

Phase C: Final Design & Fabrication	**1199 days**
Final Design	113 days
Technology Research & Development	**113 days**
Critical Design Review	5 days
Develop Plans & Procedures	**857 days**
Development of the Instrument Housing Module (IHM)	**641 days**
COSMIC A IHM Realization	**618 days**
COSMIC B IHC Realization	**585 days**
Development of the Spacecraft Bus	**277 days**
Development of the Ground System	**110 days**
Phase C Margin	166 days
Systems Integration Review	5 days
Phase D: System Assembly, Integration & Test, Launch & Checkout	**1021 days**

Development of the Instrument Housing Module (IHM)	**641 days**
COSMIC A IHM Realization	**618 days**
IHM Structure Subsystem Realization	**80 days**
Power Subsystem Realization	**56 days**
Guidance, Navigation, and Control (GNC) Subsystem Realization	**53 days**
Attitude Determination and Control (ADCS) Subsystem Realization	**82 days**
Payload Subsystem Realization	**111 days**
Communication Subsystem Realization	**93 days**
Command and Data Handling (C&DH) Subsystem Realizatio	**60 days**
Thermal Subsystem Realization	**23 days**
COSMIC A IHM Transition 1	14 days
COSMIC A IHM Functional Test	49 days
COSMIC A IHM Transition 2	14 days
COSMIC A IHM Environmental Testing	49 days
COSMIC A IHM Transition 3	14 days
COSMIC B IHC Realization	**585 days**