EAE 143B Critical Design Review | June 11th, 2026

Department of Mechanical and Aerospace Engineering

ODIN

Optical Detection and INspection

Inspection of Cooperative Target

TEAM
Organization

4 subteams

1. Systems : *Project Management, System Engineering*
2. Trajectory : *Trajectory, ADCS, Propulsion*
3. Mechanical : *Mechanical Structure, Thermal Engineering, Vehicle Design*
4. C&DH : *Payload, Communications, Power, C&DH*

MEMBERS

Toby Morrissey
Project Manager

Anderson Lin
System Engineer

Aurian Malhotra
Trajectory / GNC Engineer

Alex Zamora
ADCS Engineer

Shreya Chandra
Propulsion Engineer

Brian Maaliao
Thermal Engineer

Maya Bahnasy
Thermal Engineer

Rachel Farragher
Structural Engineer

Kai Hori
Comms / C&DH Engineer

Ethan Bintocan
Payload Engineer

Tim Siegal
Power Engineer

AGENDA

01. INTRODUCTION

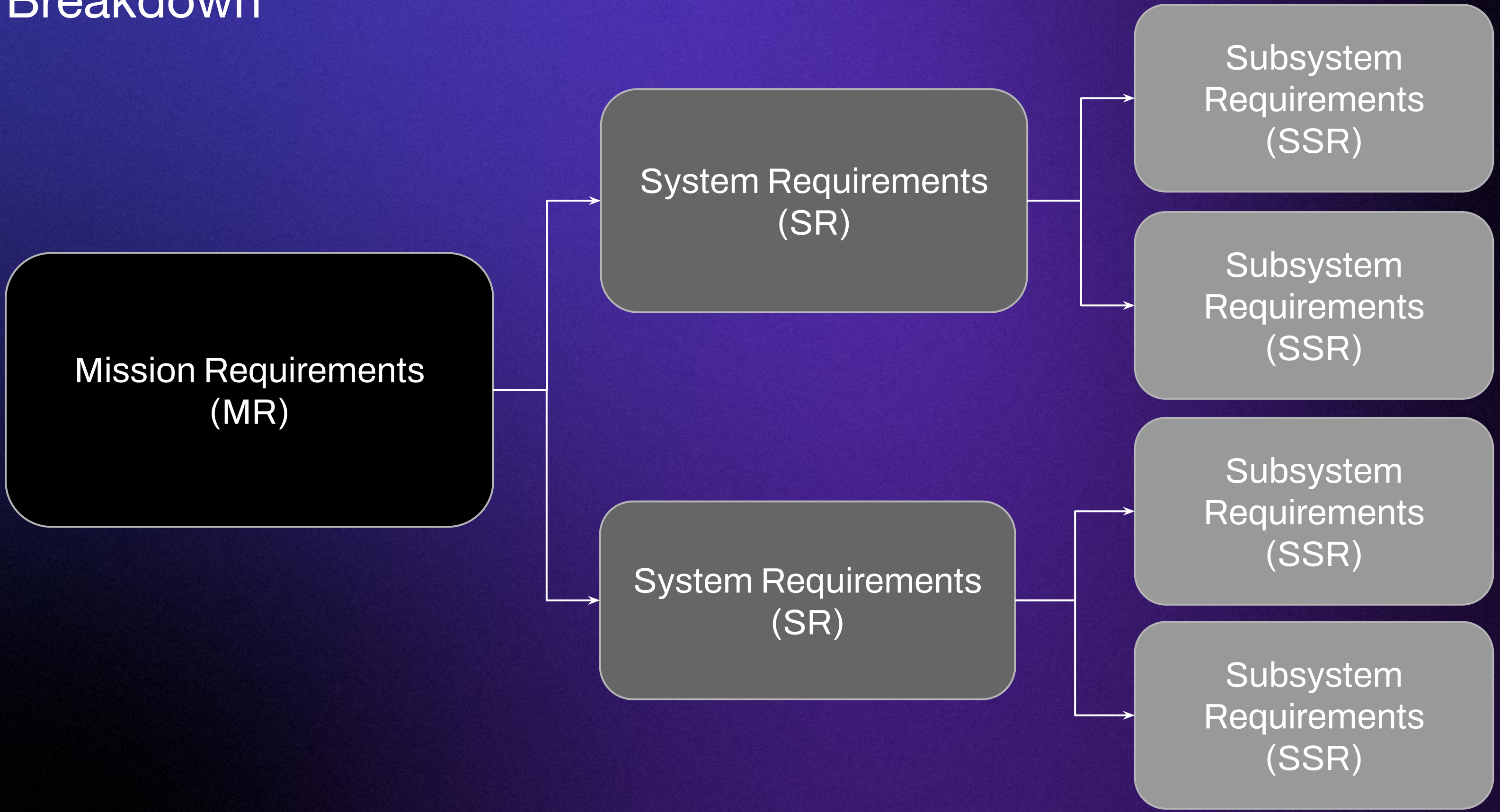
ODIN
Mission Overview

Objective:
Inspect the ESA Solar Orbiter in order to aid in the evaluation and feasibility of mission extension. Collect data to quantify degradation over a time period.

- Conduct a close-range inspection
  - Proximity Operations
- Highly elliptical heliocentric orbit
  - Fluctuating environment
  - Constant solar illumination

02. REQUIREMENTS

REQUIREMENTS
Breakdown



REQUIREMENTS

Mission Requirements

ID	Requirement
MR-1	The mission shall inspect one cooperative space object operating outside Earth orbit within 3 years of initial on-mission operations
MR-1-1	The target must be selected from one of the following: ● STEREO-A ● Solar Orbiter (SolO)
MR-1-2	The target spacecraft is assumed to be cooperative, with known ephemeris and attitude information available to support rendezvous and inspection planning
MR-1-3	The target spacecraft may execute limited, pre-coordinated attitude maneuvers to support inspection
MR-1-3-1	The inspection spacecraft shall define required target pointing constraints as part of the mission concept of operations
MR-1-3-2	No translational maneuvering or active proximity control by the target shall be assumed
MR-1-4	The inspection mission shall provide data sufficient to support assessment of the target spacecraft's current condition and potential for continued operations.
MR-2	The mission shall perform close-range inspection of the selected target without physical contact
MR-2-1	"Inspection" is defined as collecting sufficient data to: ● Identify major external features and spacecraft configuration ● Assess the apparent structural and operational condition of the target ● Detect any large-scale degradation, damage, or anomalies visible from external observation
MR-2-2	The target shall be inspected with sufficient resolution to detect surface features of 5 mm or larger on the target spacecraft

REQUIREMENTS

More Mission Requirements

MR-3	The spacecraft electrical systems shall provide ample power until mission EOL
MR-3-1	The spacecraft shall maintain consistent power generation throughout a wide range of solar distances (0.28-0.95 AU)
MR-3-2	The spacecraft must contain proper power storage to conduct inspection operations under unideal solar incidence angles
MR-3-3	The spacecraft must regulate power production and delivery to ensure safe operations
MR-4	The mission shall operate safely in proximity to the target spacecraft
MR-4-1	The spacecraft shall not make intentional physical contact with the target
MR-4-2	The mission shall include defined rendezvous, inspection, and departure phases
MR-4-3	The mission shall include provisions to minimize collision risk during all proximity operations
MR-4-4	Human-in-the-loop decision points shall be incorporated into the concept of operations
MR-5	The mission shall be capable of reaching and operating in the vicinity of the selected target using onboard propulsion
MR-5-1	All required propellant shall be carried onboard at launch
MR-5-2	No on-orbit refueling or assembly shall be required
MR-5-3	The mission architecture shall use a single inspection spacecraft
MR-6	The mission shall be compatible with a single-launch architecture
MR-6-1	The spacecraft shall fit within a single commercial 5-m class launch vehicle fairing
MR-6-2	Total spacecraft wet mass shall not exceed 1,500 kg at launch
MR-7	The mission shall support data return to Earth sufficient to enable post-inspection analysis

REQUIREMENTS

Even More Mission Requirements

Last Ones I Swear

MR-7-1	All inspection data shall be returned to Earth within the mission lifetime
MR-7-2	The mission shall define a communications architecture capable of supporting both cruise operations and inspection operations at the target mission distance
MR-8	The mission shall have a minimum operational lifetime sufficient to complete rendezvous, inspection operations, and data return for the selected target
MR-8-1	The mission shall support repeated observations of the target spacecraft over a time interval sufficient to assess changes in target condition over at least one target orbital period
MR-9	The mission shall comply with end-of-life disposal or passivation requirements appropriate to the selected operational environment
MR-9-1	The spacecraft shall include a defined end-of-life plan
MR-9-2	The end-of-life plan shall prevent the spacecraft from posing an undue long-term hazard to operational spacecraft or future missions in the relevant orbital environment

REQUIREMENTS
System Requirements

ID	Requirement	Traced To
SYS-1-1	The spacecraft shall rendezvous with and maintain controlled relative motion with the target using available ephemeris and attitude data.	MR-1, MR-1-2, MR-1-3
SYS-1-2	The spacecraft shall capture inspection imagery with a ground sampling distance ≤ 5 mm at the inspection range.	MR-2-1, MR-2-2
SYS-1-3	The spacecraft shall collect inspection data sufficient to: • Identify major external features • Visually assess structural condition • Detect visible anomalies ≥ 5 mm	MR-1-4, MR-2-1
SYS-2-1	The spacecraft shall execute distinct rendezvous, inspection, and departure phases.	MR-4, MR-4-2
SYS-2-2	The spacecraft shall maintain relative position knowledge to the target with uncertainty ≤ 2.5 m.	MR-4-3
SYS-2-3	The spacecraft shall maintain a separation distance ≥ 250 m with respect to the target during proximity operations.	MR-4-1, MR-4-3
SYS-2-4	The probability of collision during proximity operations shall be ≤ 0.01%.	MR-4-3
SYS-2-5	The mission shall include human-in-the-loop decision points throughout its lifetime (including during proximity operations).	MR-4-4
SYS-3-1	The solar array shall generate a surplus of power to maintain a “Safe” operation mode for the majority of the mission time even at aphelion.	MR-3

ID	Group Name
SYS-1	Target Interaction and Inspection Capability
SYS-2	ProxOps and Safety
SYS-3	Power
SYS-4	Propulsion
SYS-5	Comms and Data Return
SYS-6	Physical Constraints
SYS-7	Operations
SYS-8	End-Of-Life

REQUIREMENTS

More System Requirements

SYS-3-2	The solar array will be capable of power generation in thermal extremes through the use of material design and array inclination relative to the sun, up until mission EOL.	MR-3, MR-3-1
SYS-3-3	The spacecraft will utilize secondary Lithium Ion batteries to maintain power supply during inspection.	MR-3, MR-3-2
SYS-3-4	The Power Control and Distribution Unit (PCDU) must be able to regulate bus voltages at all load levels and survive any single event failure without mission loss.	MR-3-3
SYS-4-1	The spacecraft's onboard propulsion shall provide a total $\Delta V \geq 2$ km/s sufficient for transfer, rendezvous, proximity operations, and departure.	MR-5, MR-5-1, MR-4-2
SYS-4-2	At launch, the spacecraft shall be fully assembled, integrated, and fueled with all propellant carried onboard to support completion of all mission phases without on-orbit refueling or assembly.	MR-5-1, MR-5-2
SYS-4-3	The mission shall be executed using a single spacecraft.	MR-5-3
SYS-5-1	The spacecraft shall return all required inspection data and spacecraft telemetry to Earth over the mission lifetime, including stored data carried over between contact windows.	MR-7-1
SYS-5-2	The spacecraft shall maintain two-way communication with Earth ground stations at the maximum mission distance during cruise, proximity operations, safe mode, and downlink, except during planned communication outages.	MR-7-2, MR-4-4
SYS-5-3	Returned data shall be sufficient to assess the target spacecraft's condition, visible anomalies, and mission-extension viability.	MR-1-4

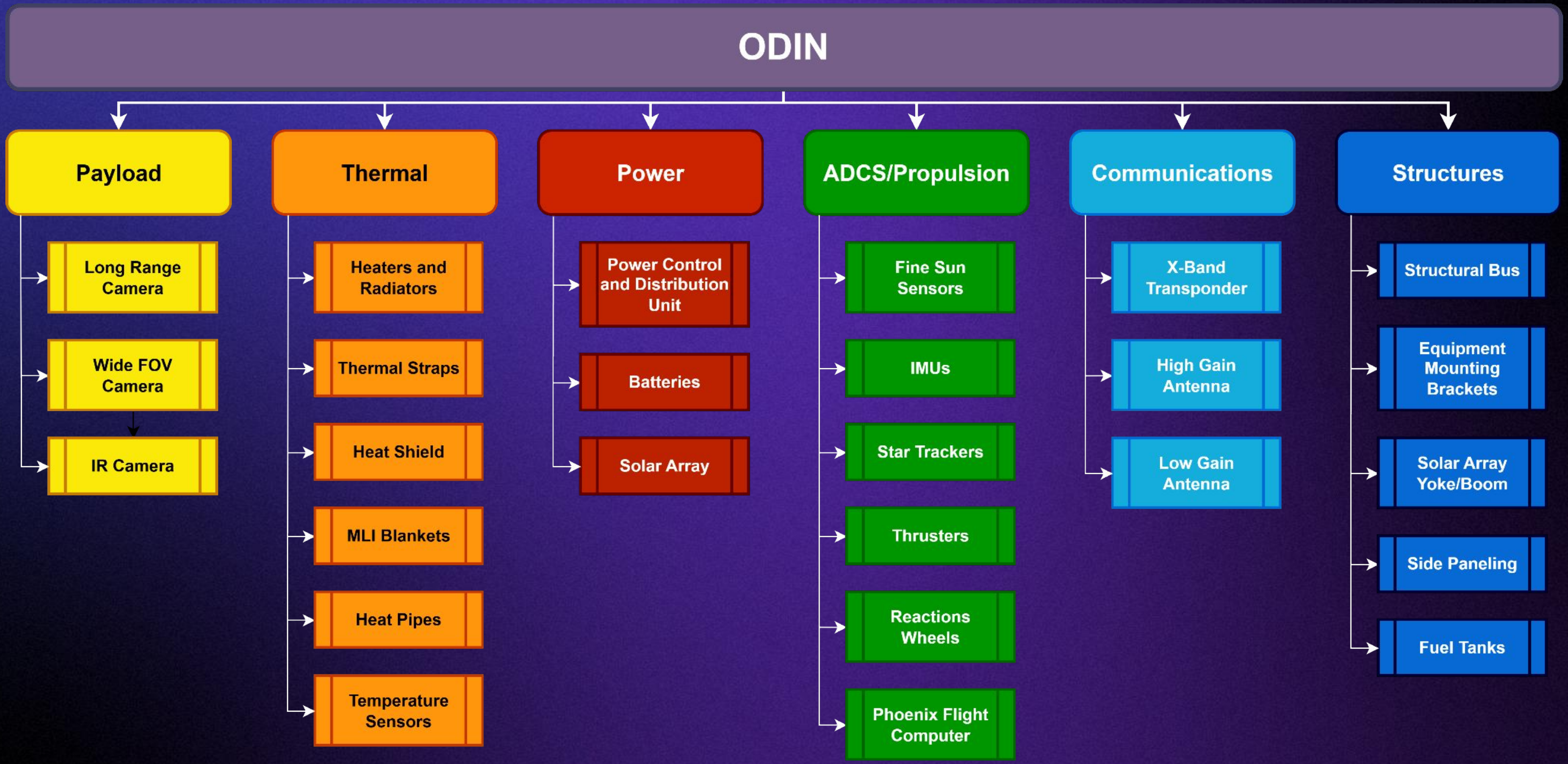
REQUIREMENTS

Last Few System Requirements

SYS-6-1	The spacecraft geometry shall be compatible with a single 5-m class commercial launch vehicle fairing.	MR-6-1
SYS-6-2	The spacecraft wet mass shall be ≤ 1500 kg at launch.	MR-6-2
SYS-7-1	The spacecraft shall support operations for ≥ 12 months- sufficient to complete all mission phases.	MR-8
SYS-7-2	The spacecraft shall support repeated observations of the target over a duration ≥ one target orbital period.	MR-8-1
SYS-8-1	The spacecraft shall implement a defined end-of-life disposal or passivation plan.	MR-9-1
SYS-8-2	The implemented end-of-life plan shall prevent the spacecraft from posing a long-term hazard to other missions.	MR-9-2

03. SYSTEM ARCHITECTURE

SYSTEM ARCHITECTURE
Product Breakdown



04. CONOPS

CONOPS
Mission Timeline

February 6, 2027 — December 7, 2027 — December 9, 2027

1. Departure

Leave Earth on Falcon Heavy into transfer orbit

2. Coast on Transfer

ODIN remains in Falcon Heavy during transfer

3. Injection Burn

Falcon Heavy performs injection burn and ODIN separates

4. Rendezvous

Rendezvous with Solar Orbiter and maintain a distance of 20km

5. Hibernation

Point solar shield towards sun and hibernate

February 16, 2028 — February 20, 2028 — March 2, 2028 — November 19, 2028 — November 23, 2028 — December 5, 2028

6. Pre-Inspection Check

Upon reaching 0.85 AU, get within 250 m of Solar Orbiter and being pre-inspection checks for inspection go-ahead from Earth

7. First Inspection and Downlink

Capture imagery data and downlink them to Earth

8. Hibernation and Ground Analysis

Back up to 20 km distance from Solar Orbiter and hibernate for another orbit. Ground analyzes data and create flags for secondary inspection

9. Pre-Inspection Check

At 0.85 AU, reach within 250m of Solar Orbiter and perform pre-inspection check

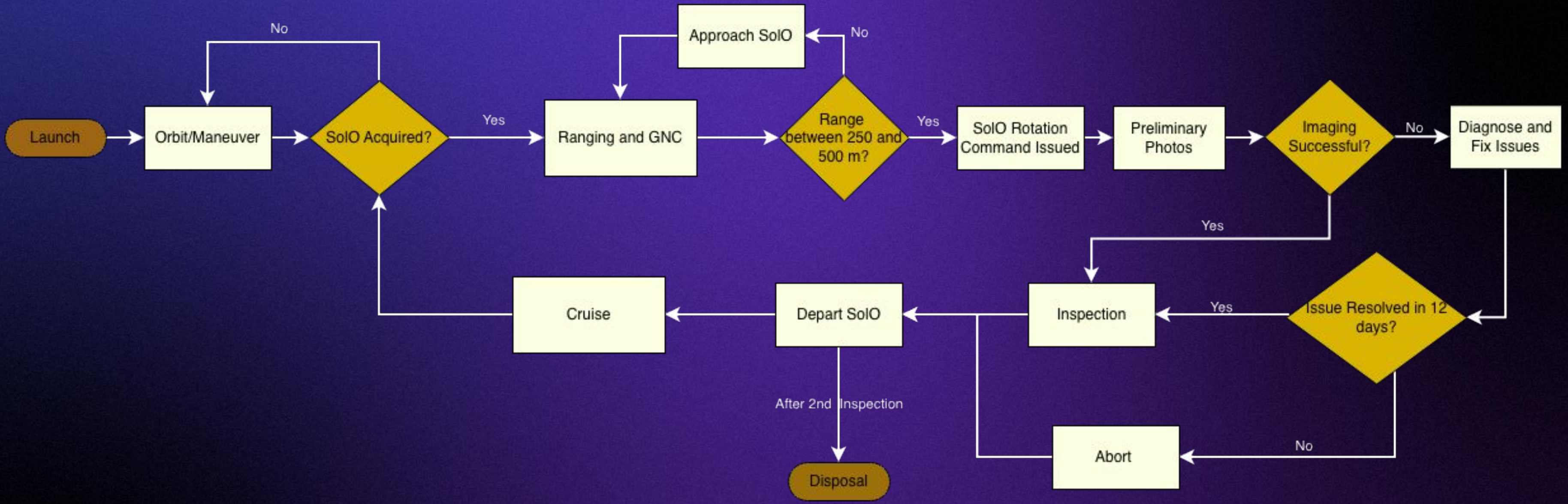
10. Second Inspection and Downlink

Capture imagery data on flags analyzed by ground and downlink

11. Disposal

Perform burn to slightly change orbital elements to minimize collision risk

CONOPS
Mission Logic



05. COST ESTIMATION

COST ESTIMATION
Scope Definition

Included in Estimate:

- Spacecraft fabrication, integration, and test
- Components
- Flight software development
- Launch services
- Mission operations (staff)
  - Cruise and human-in-the-loop phases separated
- Deep-space communications (DSN usage)

Excluded from Estimate:

- Concept development and design labor

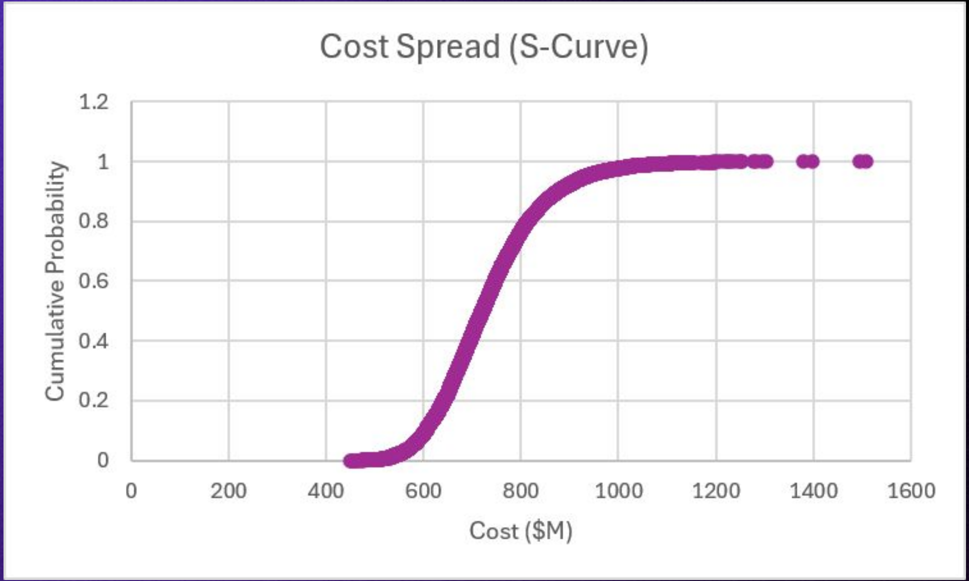
Assumptions:

- All numbers are in FY2026 US Dollars
  - No inflation estimate applied across life cycle
- Two payload cameras are only significant cost-driving components

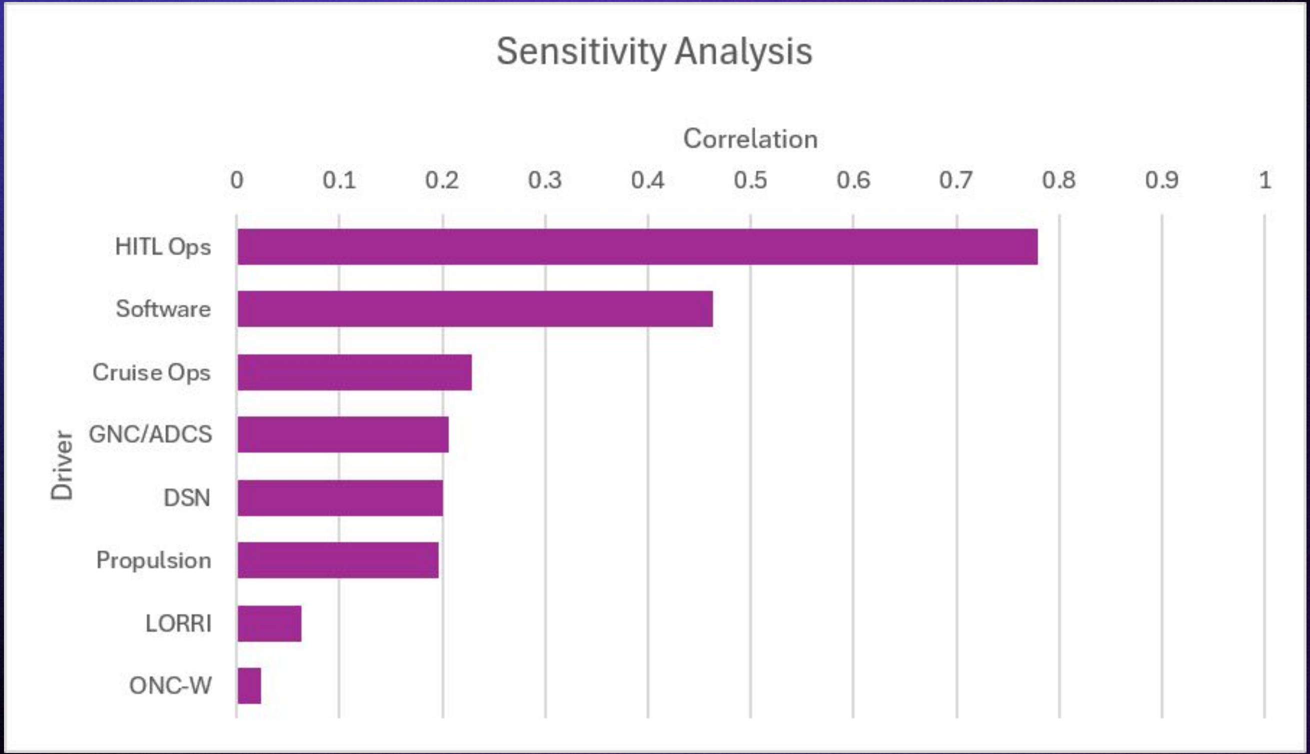
COST ESTIMATION
Estimates

- Main cost drivers assigned minimum, mode, and maximum values and distributions
  - Numbers from analogous missions and historical data
  - Distributions from uncertainty
- 10,000 missions simulated via Monte Carlo method

P50	$720.7 M
P70	$777.5 M
P80	$813.6 M
μ	$733.0 M
σ	$111.2 M

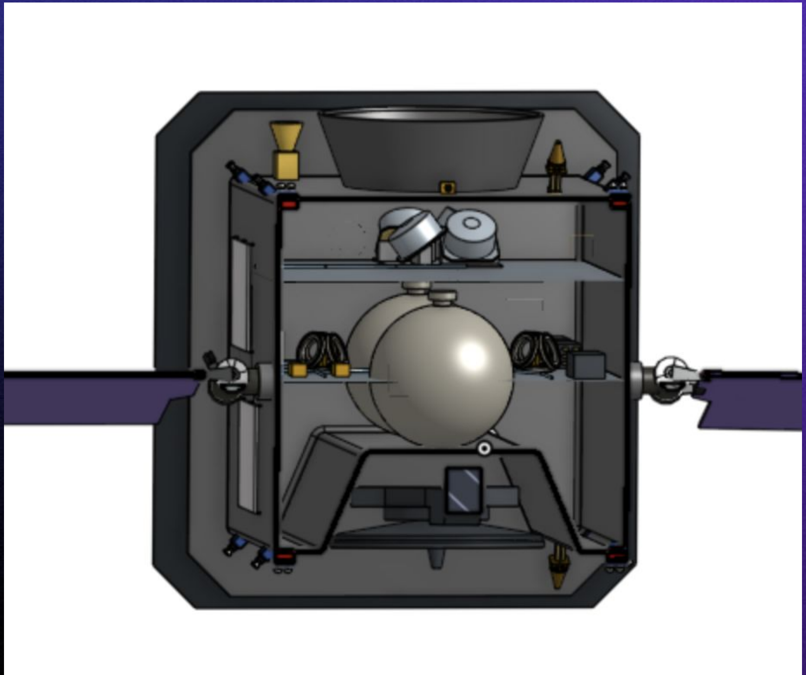


COST ESTIMATION
Driver Impact



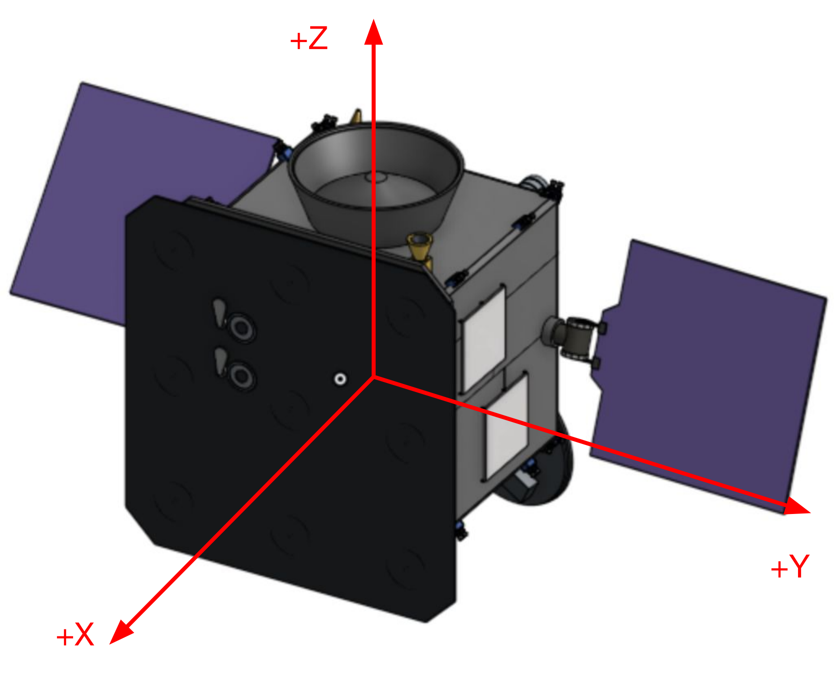
06. STRUCTURES

STRUCTURES
Design Overview

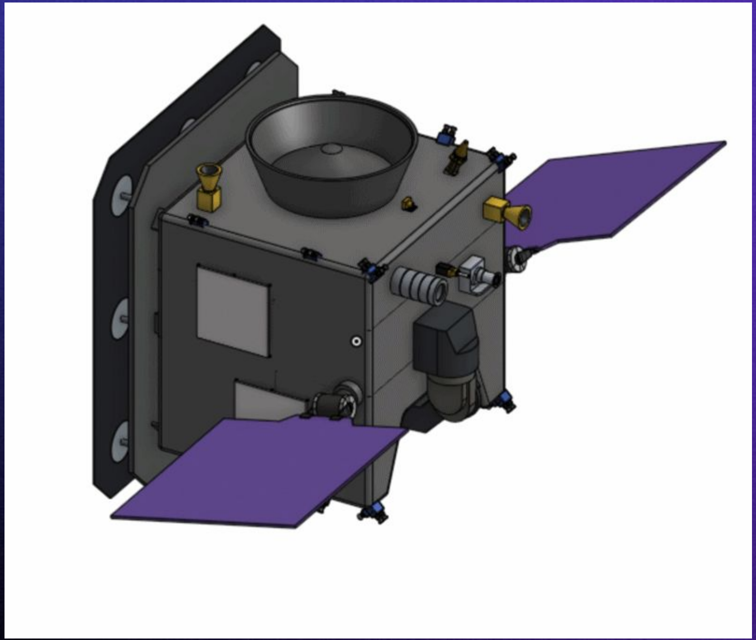


ODIN

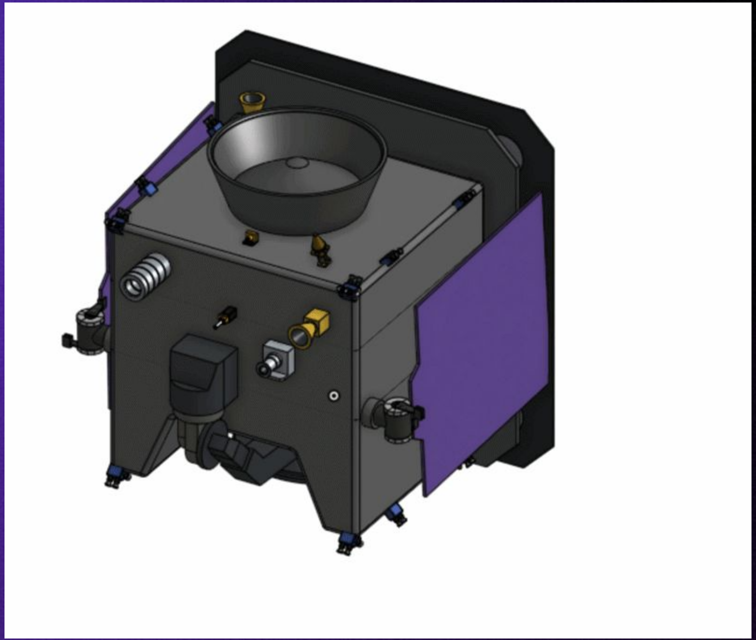
STRUCTURES
Reference System



STRUCTURES
Solar Arrays



Gimballing for thermal regulation and efficient sun pointing



Foldable for launch vehicle integration

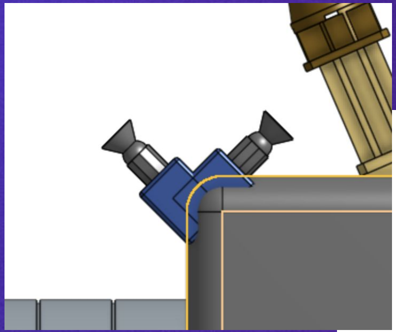
STRUCTURES
Foldable High Gain Antenna

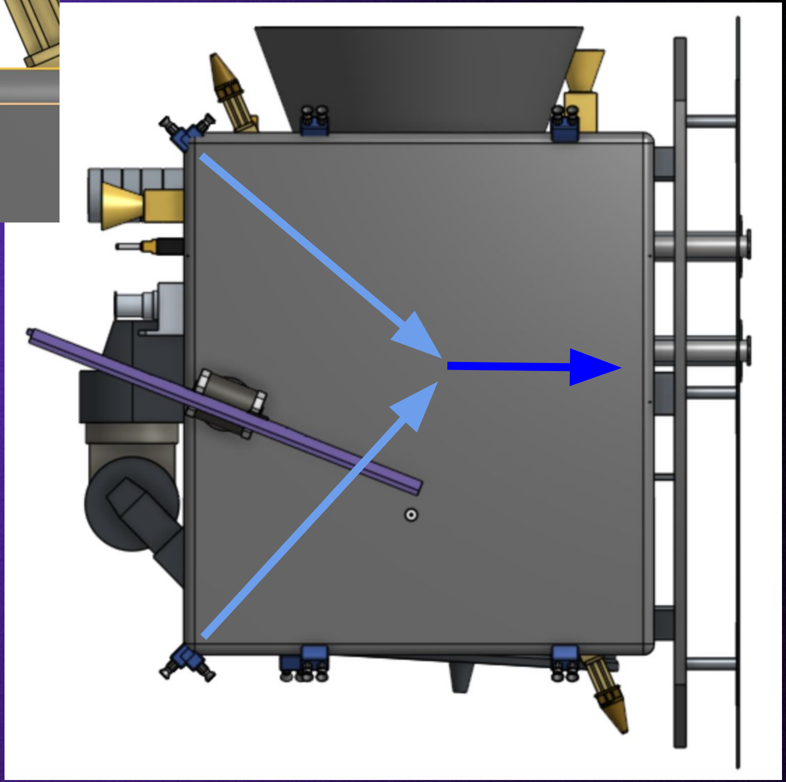
- Can fold to avoid long exposure to sun
- Can fold for integration into launch vehicle
- Can rotate for maximum signal strength

STRUCTURES
Thrusters

- Could not put thrusters on heat shield
- Put thrusters on the same edge but orthogonal
- The vertical components of direction vector cancel
- Satellite can move forward and backward
- Had to be aware of launch configuration

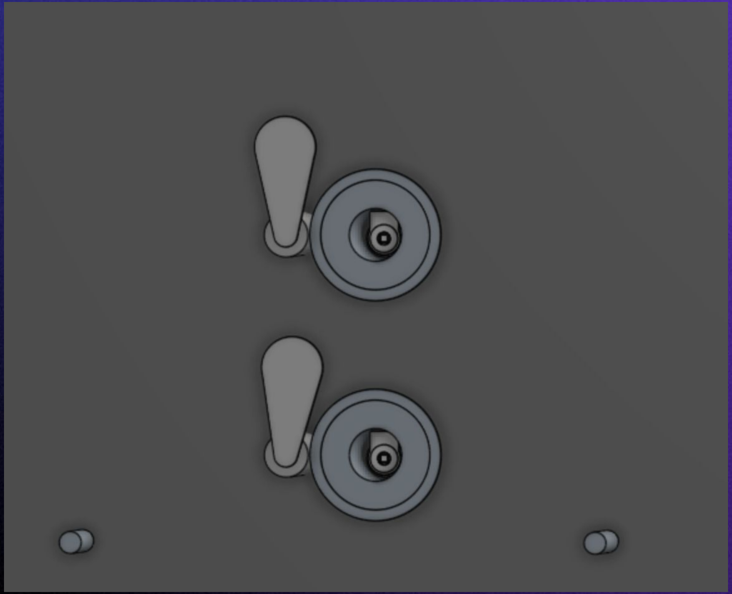
- 16 total thrusters
- Can achieve roll, pitch, and yaw

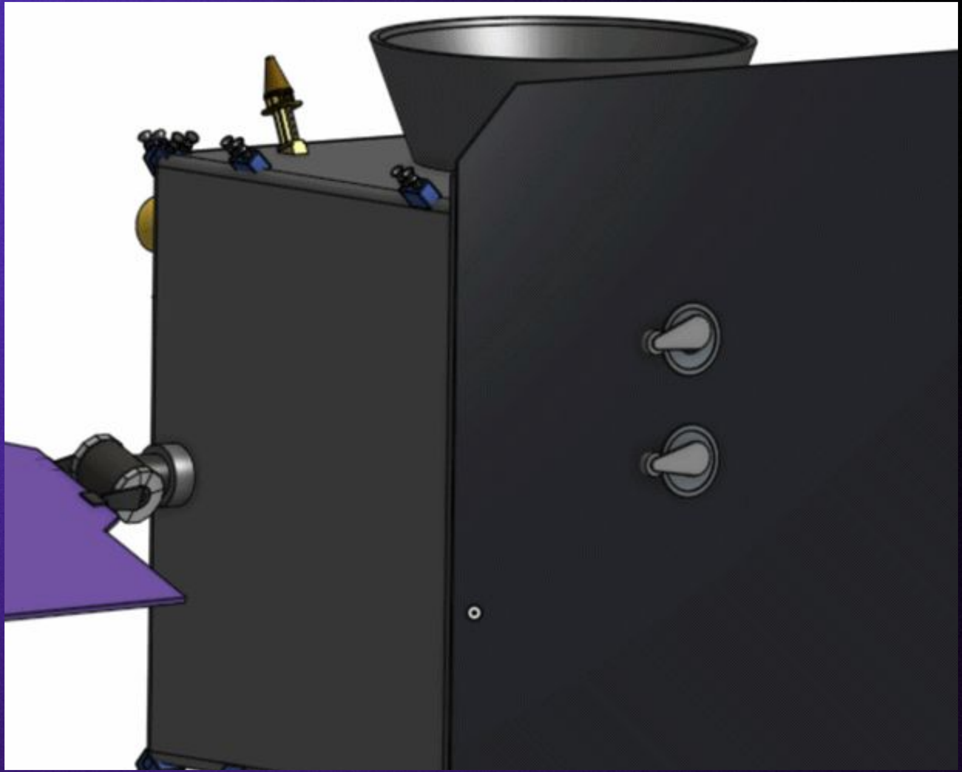




STRUCTURES
Sun Sensors

- Sensor Mounted on Bus
- Rotating Door for Limited sun exposure



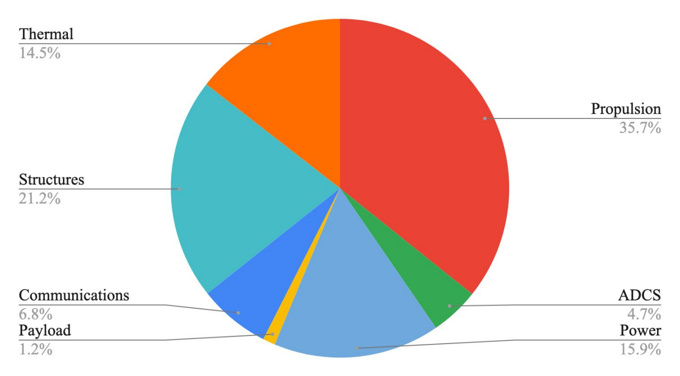


STRUCTURES
Mass Budget

Applied margin to each component and each subsystem

Dry mass: 1148 kg
Propellant Mass: 351 kg

Total Mass: 1499 kg (1kg under budget)



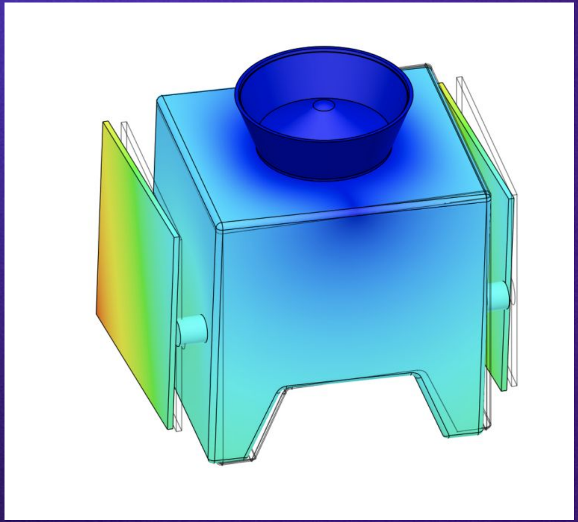
	COMPONENT	UNIT MASS	QUANTITY	TOTAL MASS	MARGIN	ALLOTTED MASS
Propulsion	Propellant	319.24	1	319.24	1.1	351.164
	Tanks	40	2	80	1.1	88
	Propulsion Sub-Total			487.164	1.1	535.8804
ADCS	Thrusters	3.4	8	27.2	1.1	29.92
	Reaction wheels	7.6	4	30.4	1.1	33.44
	Sensors	0.25	2	0.5	1.1	0.55
	ADCS Sub-Total			68.75	1.02	70.125
Power	Solar Arrays	50	2	100	1.1	110
	Battery	26.9	1	26.9	1.1	29.59
	PCDU	15	1	15	1.1	16.5
	Power Sub-Total			227.04	1.05	238.392
Payload	ONC Camera	2	1	2	1.1	2.2
	LLorri Camera	8	1	8	1.1	8.8
	Payload Sub-Total			17.6	1.01	17.776
Comms	X-Band Cables	5	-	5	1.1	5.5
	X-Band LGA	1	2	2	1.1	2.2
	X-Band HGA	60	1	60	1.1	66
	Comms Sub-Total			97.68	1.05	102.564
Structures	ODIN spacecraft bus	260	1	260	1.1	286
	Structures Sub-Total			302.5	1.05	317.625
Thermal	Heat Shield	150	1	150	1.1	165
	Radiators	15	1	15	1.1	16.5
	Thermal Sub-Total			207.02	1.05	217.371
						1499.7334

STRUCTURES
Vibrational Analysis

- Modal analysis on the primary structure
- Falcon Heavy requirement: 1st mode > 10Hz
- Only included main bus, solar panels, and fairing

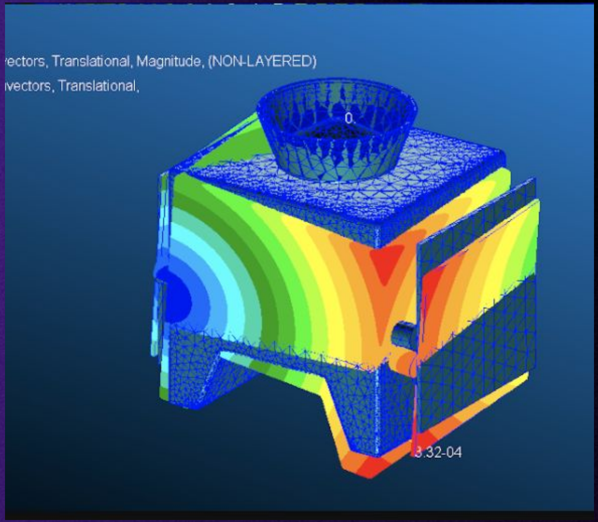
- Both models' results were too low
- This could mean the simulation is wrong
- Decrease mass or increase material stiffness
- Add structural supports

Autodesk



Mode 1: 6.987 Hz

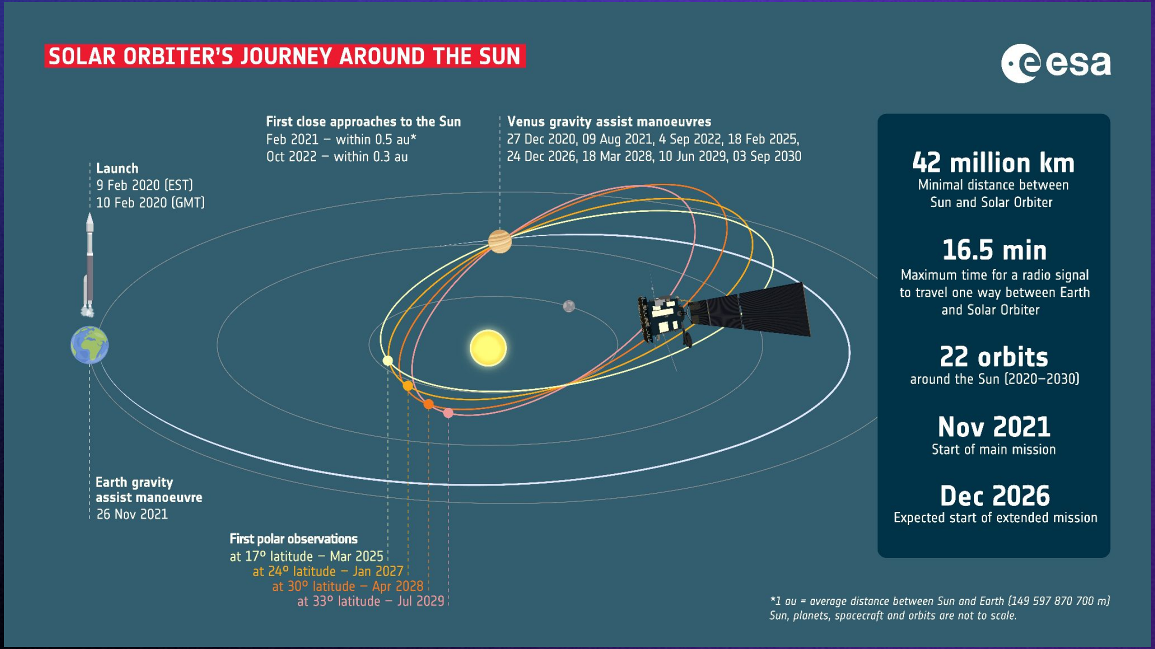
Patran



Mode 1: 1.403e-5 Hz

07. TRAJECTORY

Injection Trajectory
Problem



Problem

Solar Orbiter in a highly eccentric inclined orbit

Two- Step Approach

1. Match RAAN
2. Rendezvous in Plane

Injection Trajectory
Matching RAAN

Launching at descending node instead of ascending
- Arrive closer to inspection date
- Less secondary delta-V burn
  - Intercepting closer to perihelion of Solar Orbiter

Working in J2000 reference frame

Assumptions:
- Earth's Inclination is negligible
- Earth's Orbit is circular

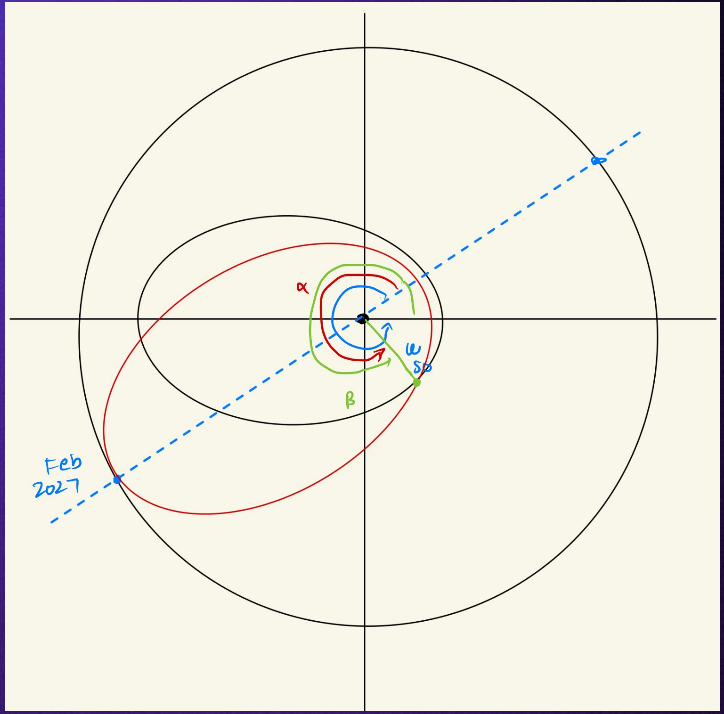
$$TA_{Earth,launch} = \Omega_{SO} - (\Omega_{Earth} + \omega_{Earth}) + 180^\circ$$

Launch Date: 6 February 2027 23:00 UTC

Injection Trajectory
In-Plane Problem Setup

Equations/Constraints

1. Same distance from sun at interception location
2. Difference of argument of perigee
3. Launching from Earth
4. Meeting at interception location at same time
5. Relationship between ODIN's location and time
6. Relationship between Solar Orbiter's location and time
7. Mean anomaly-true anomaly (ODIN)
8. Mean anomaly-true anomaly (Solar Orbiter)



Black ellipse- Solar Orbiter

Red Ellipse- Transfer Orbit

Black Circle- 1 AU from Sun

α=true anomaly of ODIN

β= true anomaly of Solar Orbiter

ω= argument of perigee of Solar Orbiter

Injection Trajectory

In-Plane Problem Setup

Equations/Constraints

1. Same distance from sun at interception location
2. Difference of argument of perigee
3. Launching from Earth
4. Meeting at interception location at same time
5. Relationship between ODIN's location and time
6. Relationship between Solar Orbiter's location and time
7. Mean anomaly-true anomaly (ODIN)
8. Mean anomaly-true anomaly (Solar Orbiter)
   a. 3rd order Taylor Expansion

$$r = \frac{a_t(1-e_t^2)}{1+e_t cos(\alpha)} = \frac{a_s(1-e_s^2)}{1+e_s cos(\beta)} \quad (1)$$

$$360^\circ - \omega_s = \beta - \alpha \quad (2)$$

$$r_{t,a} = a_t(1+e_t) \quad (3)$$

$$t_s = t_t \quad (4)$$

$$M_t - 180^\circ = \sqrt{\frac{\mu_{sun}}{a_t^3}}(t_t - t_{launch}) \quad (5)$$

$$M_s = \sqrt{\frac{\mu_{sun}}{a_s^3}}(t_t - t_{s,p}) \quad (6)$$

$$\nu = M + \left(2e - \frac{1}{4}e^3\right)\sin M + \frac{5}{4}e^2 \sin 2M + \frac{13}{12}e^3 \sin 3M + \mathcal{O}(e^4) \quad (7,8)$$

Injection Trajectory
In-Plane Results

Semi-Major Axis = 95.820 million km

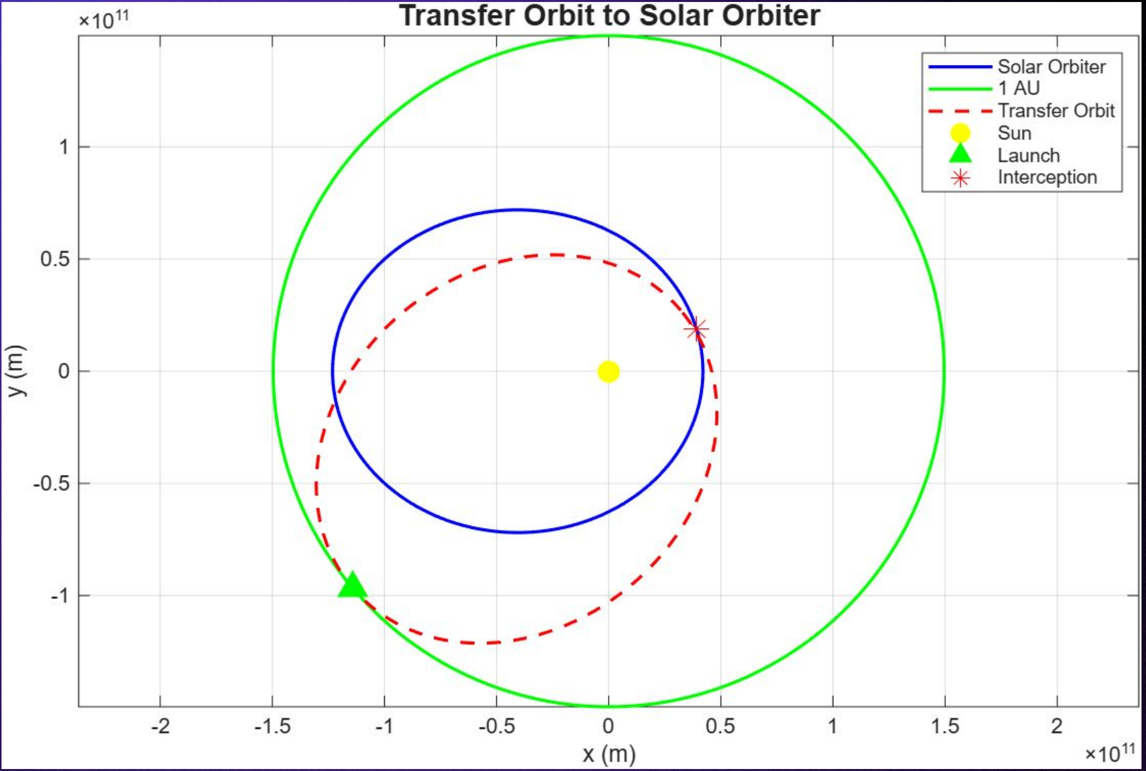
Eccentricity = 0.5612

Injection Date: 7 December 2027 01:00 UTC

Launch Date: 6 February 2027 23:00 UTC

β = 25.821°

M_s = 8.522°



Injection Trajectory
Validation

Semi-Major Axis = 95.820 million km

Eccentricity = 0.5612

Injection Date: 7 December 2027 01:00 UTC

Launch Date: 6 February 2027 23:00 UTC

β = 25.821°

MA_s = 8.522°

```
2461746.541666667 = A.D. 2027-Dec-07 01:00:00.0000 TDB
 EC= 4.912353865750836E-01 QR= 4.201215155360634E+07 IN= 1.928850473290181E+01
 OM= 3.178584159170751E+02 W = 3.198150087745678E+02 Tp=  2461743.010332241189
 N = 2.781579067686433E-05 MA= 8.486800633513008E+00 TA= 2.782164009795184E+01
 A = 8.257679572246140E+07 AD= 1.231414398913165E+08 PR= 1.294228893875839E+07
```

JPL Horizons System Ephemeris Data of Solar Orbiter

For further validation

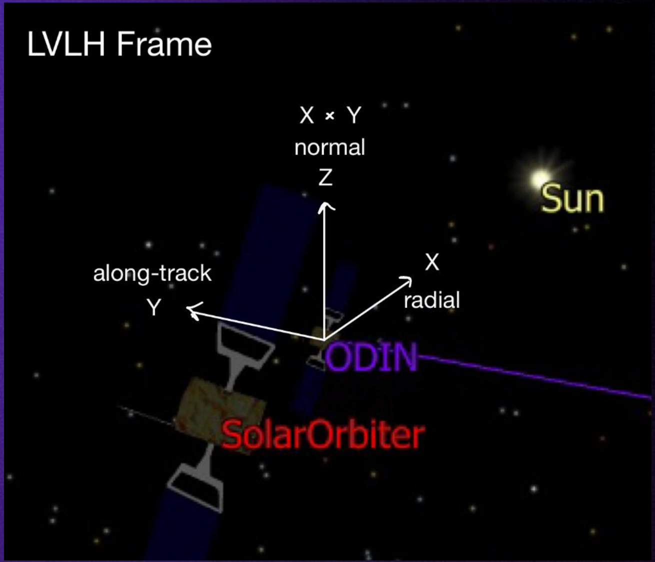
- Simulate trajectory in FreeFlyer
- Plug in values into Kepler’s equation of motion
  - The time of transfer matches
- Reduce Assumptions and redo calculations
  - Earth’s Orbit not completely circular
  - Iterate mean anomaly from true anomaly

Injection Trajectory
Delta-V Calculations

Burns	Delta-V (km/s)	Remarks
Ground to 200 km parking LEO	8.876	Losses ~1500 km/s
LEO to escape trajectory	3.325	
Escape to transfer orbit	12.926	Plane Change: 6.6108 km/s Speed Change: 10.0548 km/s
Injection Burn	1.753	Radial: 1212.32 m/s Transverse: 1266.73 m/s
Total	26.880	

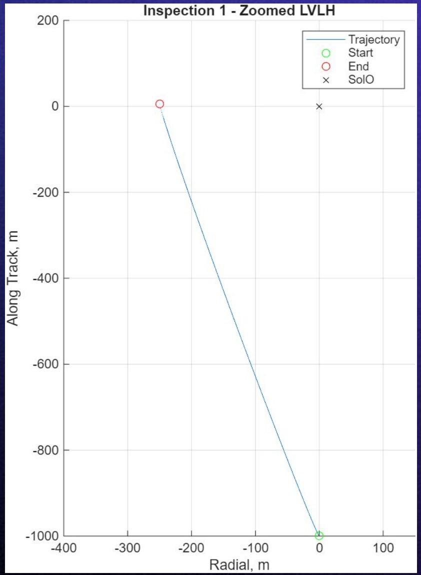
GNC Approach
CW Analysis & Control

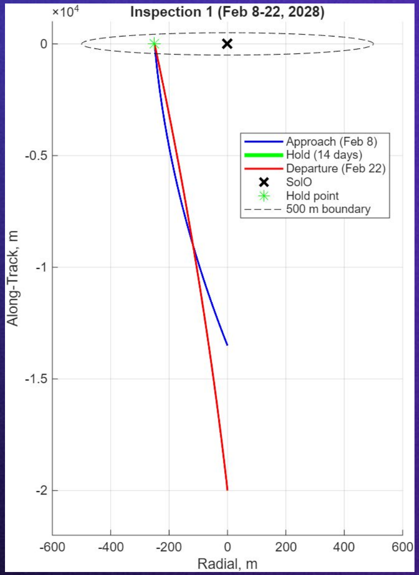
- Clohessy-Wiltshire equations used for two-impulse transfer design
  - Note: CW assumes circular reference orbit, results are approximate
  - State transition matrix solved analytically in MATLAB
  - Mean motion *n* evaluated locally at each maneuver epoch (approach, hold, departure)
  - Orbital elements from FreeFlyer ephemeris propagation (JPL Horizons)
- Hold point: 250 m radially inward, 500 m radius safety boundary
- Stationkeeping: PD controller simulation over 14-day window; ΔV estimate requires further numerical verification

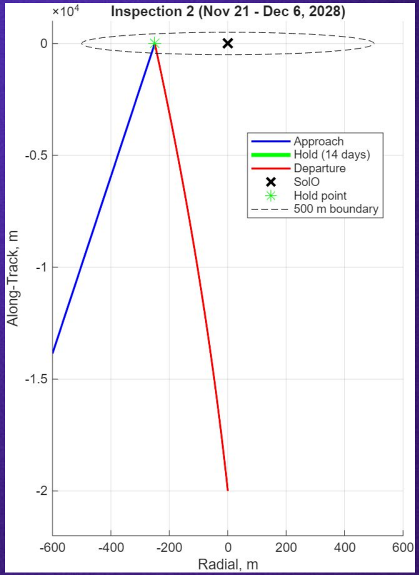


Proximity Operations
Results

along-Track begins -25,454 km







	Insp 1 (m/s)	Insp 2 (m/s)
Approach	0.63	589
Stationkeeping	0.016	0.016
Departure	2.78	2.78
Total ΔV	3.43	592

Budgeted ΔV	1000 m/s

Stationkeeping:
Hold for 14 days

Inspection 1:
12 hour approach
4 hour departure

Inspection 2:
24 hour approach
4 hour departure

Overview and Future Refinements
GNC

Confirmed Parameters

- Inspection 1: Feb 8-22, 2028 (Aphelion Feb 16)
- Inspection 2: Nov 21 - Dec 6, 2028 (Aphelion Nov 28)
- Hold point: 250 m radially inward, 500 m KoZ
- Total prox ops ΔV to propulsion: 1 km/s

Verification and Validation Items

- Saving ΔV by modifying approach method
- CW results are approximate; circular orbit assumption invalid at $e \cong 0.5$; high-fidelity numerical propagation required
- Stationkeeping ΔV requires numerical verification
- Simulation is 2D, out of plane dynamics unmodeled
- Perturbations not included (solar radiation pressure, solar wind, third-body effects)

08. ADCS

ADCS
Sub-System Requirements

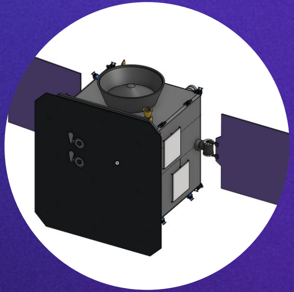
SSR-ID	Requirement Type	Description	Verified by	Rationale for requirement	Traces to
SSR-ADCS -01	Performance	Under nominal control conditions, ODIN shall maintain an angular error of <1 degree	Analysis (A) & Review (R)	For nominal control goals such as sun-pointing and earth-pointing communications.	SYS-1-3 SYS-3-1 SYS-2-1
SSR-ADCS -02	Performance	Under nominal control conditions, ODIN shall maintain an angular velocity error of <1deg/s	Analysis (A) & Review (R)	For nominal conditions, reduce tumbling and maintain aforementioned pointing goals.	SYS-1-3 SYS-3-1 SYS-2-1
SSR-ADCS -03	Performance	Under inspection control conditions, ODIN shall maintain an angular velocity error of <0.1 deg/s	Analysis (A) & Review (R)	To reduce the amount of pixel blur on the image taken from the payload.	SYS-1-3 SYS-2-1
SSR-ADCS -04	Performance	Under inspection control conditions, ODIN shall maintain an angular error of <0.1 degree	Analysis (A) & Review (R)	To keep images of Solar Orbiter within frame and have stable images throughout inspection.	SYS-1-3 SYS-2-1
SSR-ADCS -05	Performance	Under inspection conditions, ODIN shall desaturate reaction wheels with thrusters.	Analysis (A) & Review (R)	Rapid chance of angular set points can very easily saturate the reaction wheels	SYS-2-2 SYS-3-1

ADCS
Pointing Budget

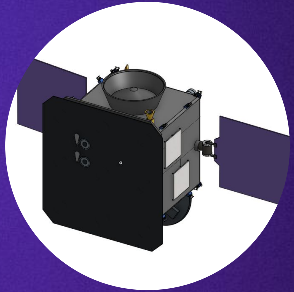
Subsystem	Component	Target	Alignment Axis	Accuracy	Precision	Function
Payload	Camera	Solar Orbiter	(-) X SC aligned with SolO	0.01°	0.001°	Needs Pointing
COMMs	Transmitter	Earth	(-) X SC aligned with Earth	0.01°	2 arcmin	Needs Pointing
	Receiver	Earth	(-) X SC aligned with Earth	0.01°	2 arcmin	Needs Pointing
ADCS	IMU x2	NA	NA	0.001°	0.001°	State Estimation
	Star Tracker x2	NA	NA	0.001°	0.001°	State Estimation
	Sun Sensor x2	NA	NA	1 arcmin	NA	State Estimation
	Reaction Wheels	NA	NA	0.01°	NA	Provides Control
	Thrusters	NA	NA	0.1°	NA	Provides Control
Thermal	Full Suite	NA	NA	NA	NA	NA
Power	Solar Panels	Sun	(+) X SC aligned with Sun	45°	45°	Needs Spin Preference or Pointing

ADCS
Simulation Breakdown

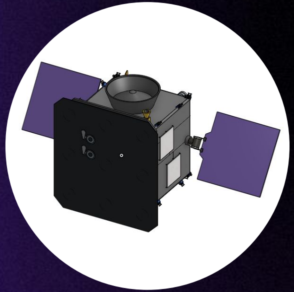
Fully Stowed



Deployed Solar Panel



Fully Deployed



Fully Deployed, Angled Solar Panel

Fully Stowed

I	X	Y	Z
X	597	-0.6	10
Y	-0.6	869	15
Z	10	15	838

Deployed Solar Panel

I	X	Y	Z
X	630.4	7.94	10.5
Y	7.94	1686	16.6
Z	10.5	16.6	1689

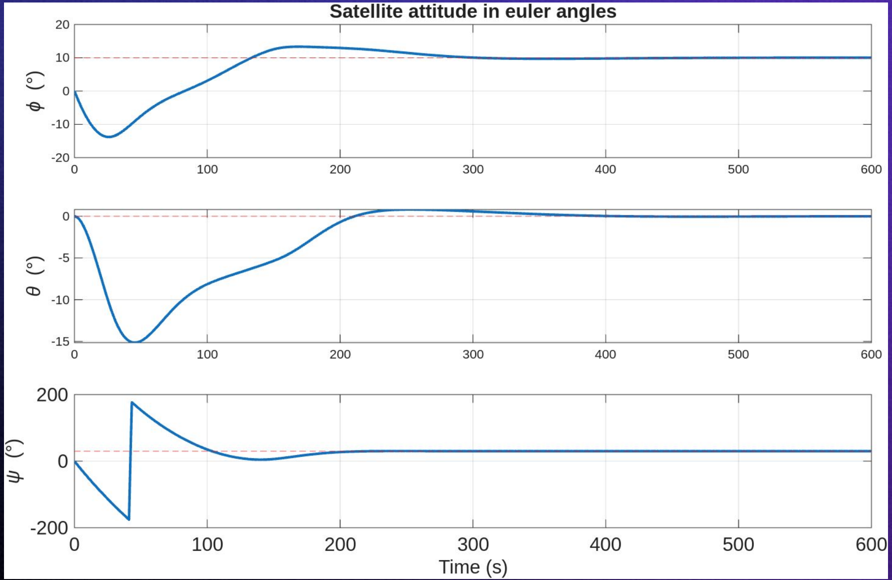
Fully Deployed

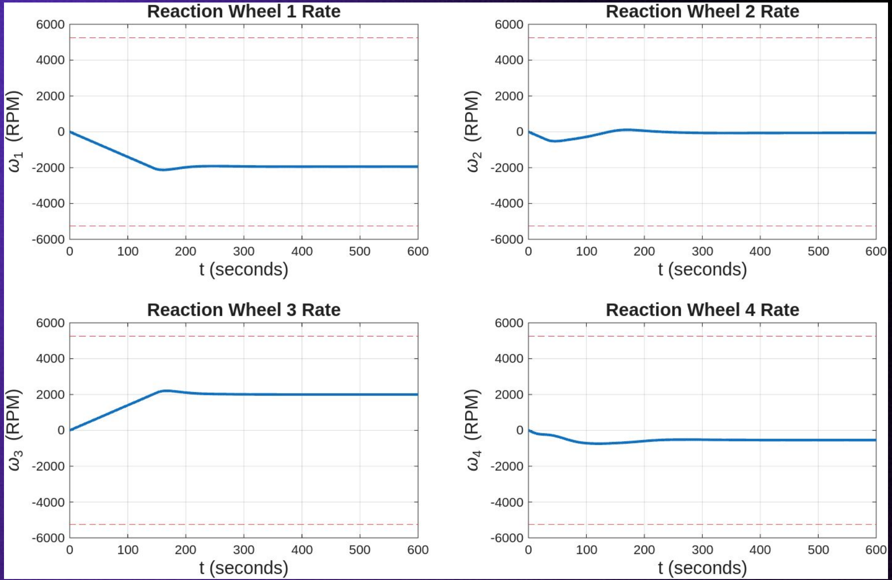
I	X	Y	Z
X	950.1	17.2	15.7
Y	17.2	1922	-189
Z	15.7	-189	1773

Fully Deployed, Angled Solar Panel

I	X	Y	Z
X	948	9.86	18.3
Y	9.86	1909	-171
Z	18.3	-171	1784

ADCS
Simulation Results Fully Stowed (Detumbling)

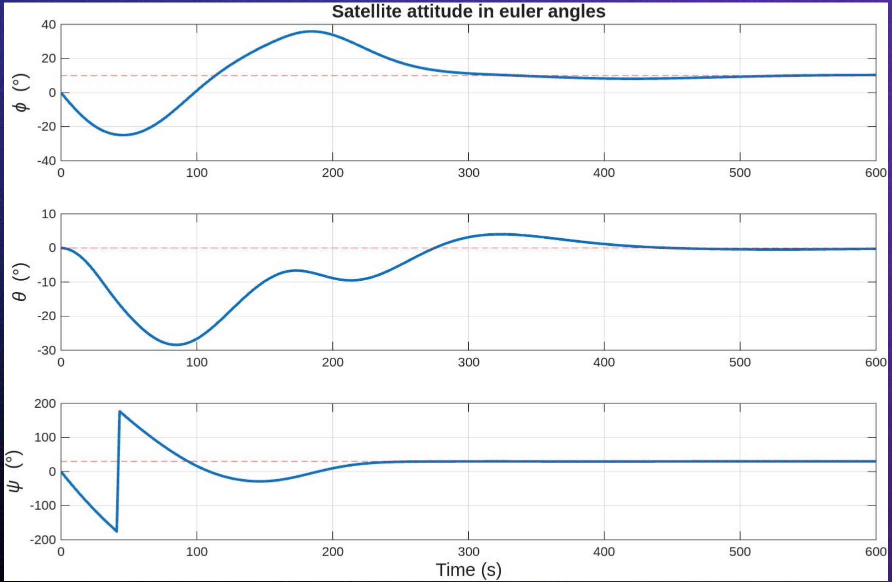




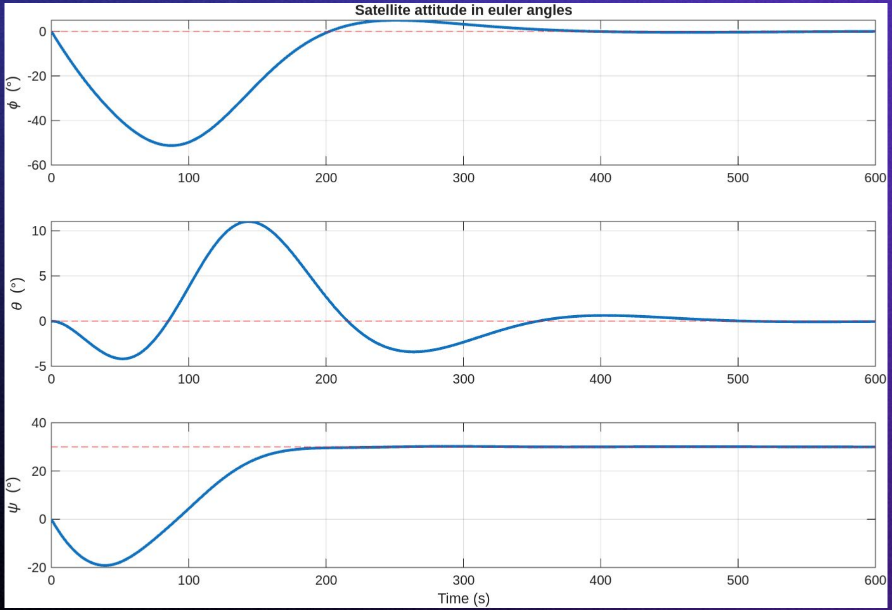
ODIN

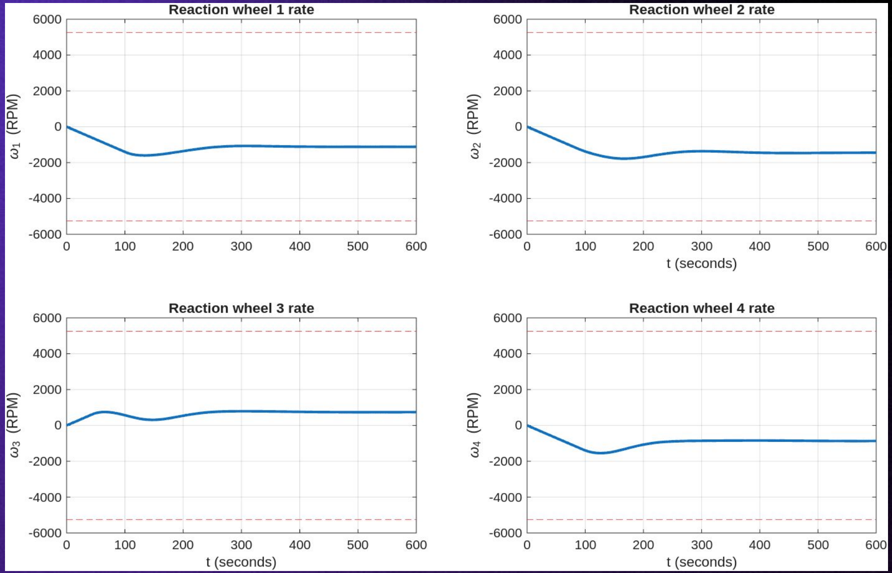
ADCS

Simulation Results Deployed Solar Panel (Sun Acquisition)



ADCS
Simulation Results Fully Deployed



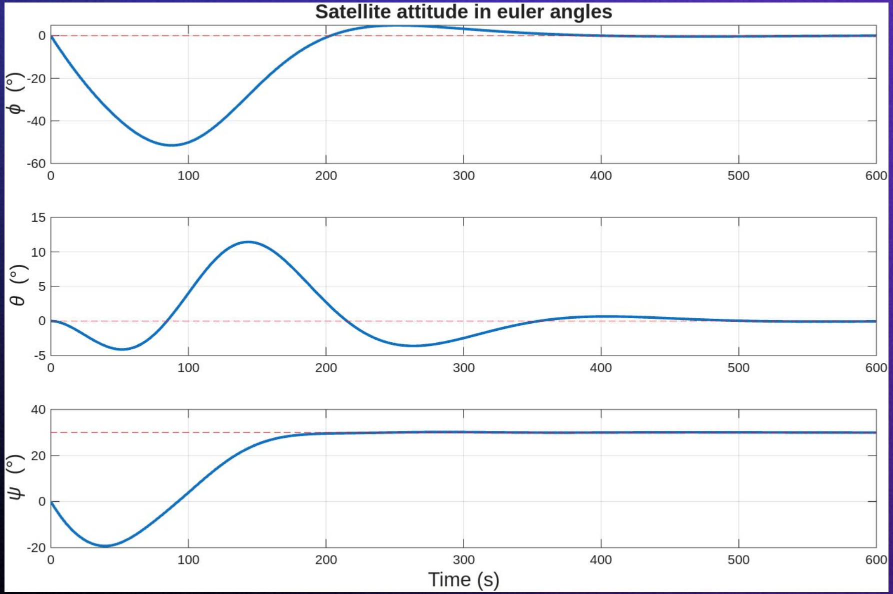


UC DAVIS MECHANICAL AND AEROSPACE ENGINEERING

UC DAVIS COLLEGE OF ENGINEERING

ADCS

Simulation Results Fully Deployed, Angled Solar Panel



ADCS
Verification & Validation

- Sensors
  - Sensor Testing and calibration of FSS, ST, IMU
  - Sensor characterization
- Actuators
  - Ensure location of pose sensitive components
    - IMU
    - Reaction Wheels
    - Thruster
  - Connector pin voltage testing for sensitive ADCS components
  - Loading connections for critical ADCS components
- Deployed Algorithms
  - Inspection of satellite rigid body dynamics
  - Reanalyze ADCS & GNC simulations with updated model
  - Hardware-In-The-Loop Testing for reaction wheels
  - Vibration testing for ADCS connections

ADCS
Risk Management

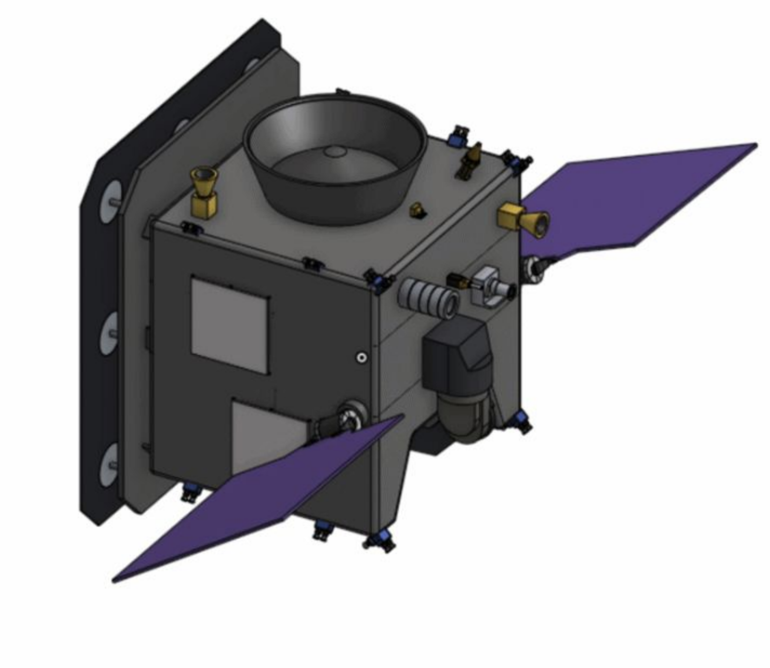
Code	Failure Name	Effect	Mitigation
A	Flight Computer Failure	Potential loss of control, lost control commands and state estimates	Secondary flight computer for control and state estimation redundancy.
B	Flight Software Failure	Loss of control	Testing and validation prior to launch to make sure flight software is deployable.
C	IMU Failure	Loss of state estimation	Use previous state vector before IMU failure. Primarily rely on Star tracker for state estimation or FSS.
D	Sun Sensor Failure	Loss of safety sensor	Use previous state estimate of sun from stellar-gyro. Redundancy from second sun-sensor.
E	Star Tracker Failure	Loss of state estimation	Use previous state estimate of sun from stellar-gyro. Redundancy from second sun-sensor.
F	Reaction Wheel Failure	Loss of control	Detumble with RCS, enter sun pointing with thrusters.
G	Thruster Failure	Limited availability of reaction wheel desaturation. Can lead to loss of control.	Detect which thrusters have failed. Determine our controllability.

Certain >80%					
Likely 60% - 70%					
Possible 40% - 59%					
Unlikely 20% - 39%		C,D,E	F		
Rare < 20%	A		B	G	
	1 Insignificant	**2** Minior	**3** Moderate	**4** Major	**5** Severe

09. PROPULSION

PROPULSION
Subsystem Architecture and Design Philosophy

- ***Single-Stage Profile*** → Now possible due to a delta-v budget of < 2 km/s
- ***All-Chemical Implementation*** → High-thrust hypergolic system removes high-voltage PPU mass penalties and multi-year radiation transit curves
- ***Unified Fluidic Infrastructure*** → Single non-cryogenic manifold architecture supplies both the main insertion engine and proximity RCS pods



PROPULSION
Requirements & Traceability

SSR-ID	Requirement Type	Description	Verified By	Rationale for Requirement	Traces To
SSR-PROP-01	Performance	Delta-v > 2 km/s (10% margin)	Analysis (FreeFlyer & Matlab)	Closes heliocentric transit and orbit-matching trajectory post-launch injection	SYS-4-1/MR-5
SSR-PROP-02	Performance	Minimum Impulse Bit (MIB) < 0.5 mNs (20 ms pulse)	Test (Software Bench)	Satisfied attitude deadbands to prevent camera jitter during close-range tracking	SYS-1-2/MR-2
SSR-PROP-03	Constraint	Subsystem Wet Mass < 536 kg	Inspection (CAD Model)	Enforces system compliance within the centralized spacecraft mass budget	SYS-6-2/MR-6
SSR-PROP-04	Geometric Constraint	Main Engine Cant Angle = 15° Outboard	Analysis (CAD Alignment)	Prevents exhaust plume deposition from contaminating Solar Orbiter's optical lenses	SYS-2-3/MR-4
SSR-PROP-05	Structural Performance	System MEOP = 17 Bar (FS > 2.0)	Test/Analysis (FEA)	Establishes structurally safe pressure boundaries for the propellant feed line manifolds	SYS-4-2/MR-6

Primary Actuator & Nozzle Geometry

Selected Primary Actuator: Aerojet Rocketdyne R-4D-11

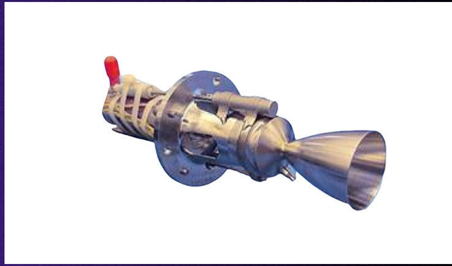
- ***Core Profile*** → Nominally rated for steady-state thrust = 490 N with a specific impulse of 312 s at a mixture ratio (O/F) of 1.76
- ***Ignition Mechanics*** → Spontaneous hypergolic liquid contact removes secondary high-voltage electrical ignition systems and turbopump complexity
- ***Thermal Mitigation*** → High-temperature silicide-coated columbium combustion chamber utilizes internal fuel-film cooling to endure solar loads at 0.28 AU

Parameter	Calculated Value
Throat Area	1.4909×10^{-5} m^2
Exit Area	2.2364×10^{-3} m^2
Throat Diameter	4.357 mm
Exit Diameter	53.361 mm
Total Mass Flow	0.006861 kg/s
Oxidizer Flow	0.004375 kg/s
Fuel Flow	0.002486 kg/s
Total Burn Life	51436.13 s

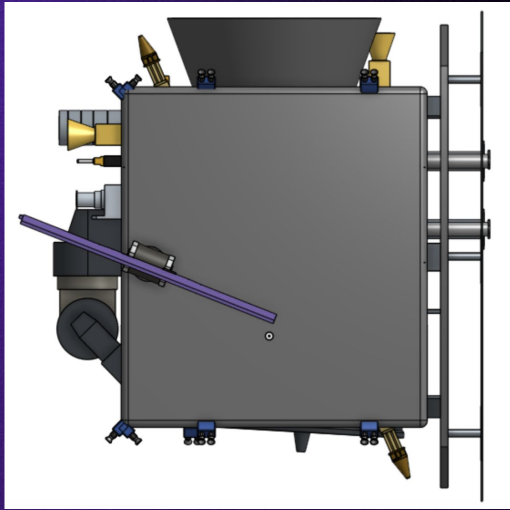
Reaction Control System (RCS)

Primary Actuators: Aerojet Rocketdyne MR-106L

- ***Spacecraft Layout*** → 16 actuators grouped into 4 corner quad-pods to deliver independent, decoupled 6-DOF translation and rotation
- ***Jitter Control*** → High-frequency valve mechanics support < 20 ms minimum pulse widths sub-millimeter LORRI camera stabilization
- ***Plume Impingement Control*** → Integrated 15° outboard cant limits exhaust deposition on Solar Orbiter optics at a cost of a 3.4% cosine thrust loss



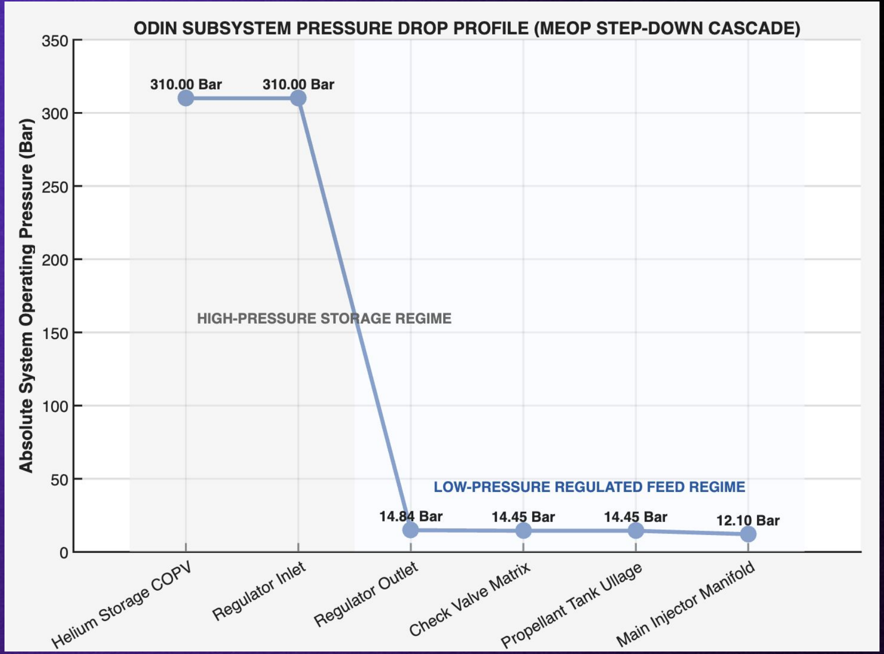
Retrieved From: https://www.satcatalog.com/component/mr-106l-22n/



Pressurization & Fluidic Layout

Primary Actuators: Aerojet Rocketdyne MR-106L

- ***Gas Pressurization*** → Cold helium gas isolated inside a titanium-lined carbon-fiber overwrapped pressure vessel (COPV) operating at 310 Bar
- ***Pressure Management*** → Series-redundant regulators step down storage pressure to a regulated, constant tank operating pressure of 14.84 Bar
- ***Microgravity Handling*** → Surface-tension metallic vane Propellant Management Devices (PMDs) maintain bubble-free liquid lines under zero-gravity conditions



PROPULSION
Material Compliance & Volumetric Sizing

- ***Tank Construction*** **→** Symmetrical cylindrical tank dual-assembly utilizing solution-treated and aged titanium alloy (Ti-6Al-4V)
- ***Ullage Boundary*** **→** Sized with a 4% volumetric ullage safety margin to absorb fluid thermal expansion at extreme solar perihelion

Element	Parameter Value	Allocation/Fluid Density
Oxidizer (N_2O_4) Mass	225.043 kg	Propellant Load ($\varrho \approx$ 1450 kg/m^3)
Exit Area	127.866 kg	Propellant Load ($\varrho \approx$ 880 kg/m^3)
Throat Diameter	352.909 kg	Integrated Burn Allocation
Oxidizer Flow	0.1563 m^3 (156.28 L)	Sized with 4% Ullage Margin Included
Fuel Flow	0.1453 m^3 (145.30 L)	Sized with 4% Ullage Margin Included
Total Burn Life	408.81 kg	Compliant (Subteam Mass Ceiling: 536.0 kg)

RISK MANAGEMENT

Propulsion

Code	Failure Name	Effect	Mitigation
A	Propellant Vaporization/Boil-Off	Extreme solar flux at 0.28 AU over-pressurizes lines, causing fuel boil-off and breaking the 352.91 kg trajectory mass budget	Embed fluid feedlines directly into high-conductivity carbon-fiber face sheets and optimize localized MLI wrap thickness
B	Plume Deposition on Target Optics	Chemical exhaust from RCS or primary engine coats the lens of the target platform, corrupting critical tracking and optical data	Enforce a 15° outboard thruster cant angle, accepting a minor 3.4% cosine thrust loss to clear the 65 m safety envelope
C	Hypergolic Reverse Vapor Mitigation	Volatile fuel/oxidizer vapors travel upstream into common pressurization lines, creating a hypergolic detonation hazard	Integrate series-parallel, quad-redundant check valve strings immediately downstream of the 14.84 Bar gas regulators
D	Nozzle Throat Thermal Erosion	Extreme core temperatures over the continues 51436.13 s total burn life melt or erode the throat, destroying the nominal chamber pressure	Specify a high-integrity silicide-coated columbium combustion chamber utilizing continuous internal fuel-film boundary cooling
E	Valve Corrosive Sticking/Leakage	Long-term deep space chemical exposure degrades valve seats, preventing clean < 20 ms minimum pulse bits and causing attitude jitter	Utilize dual-seat, torque-motor-actuated pilot valves on the primary R-4F-11 engine and all 16 RCS lines for zero-leakage

Certain >80%					
Likely 60% - 70%					
Possible 40% - 59%					
Unlikely 20% - 39%					
Rare < 20%			E	A, B,D	C
	1 Insignificant	**2** Minor	**3** Moderate	**4** Major	**5** Severe

10. POWER

POWER
Sub-System Requirements

SSR-ID	Requirement Type	Description	Verified by	Rationale for requirement	Traces to
SSR-POW-01	Constraint	ODIN must keep Solar Array (SA) Temperatures below 215°C through a mixture of angling relative to the sun and intermixing of Optical Solar Reflectors (OSRs) on the front surface.	Analysis & Testing (A&T)	The SA must be kept within operational temperature ranges in order to maintain function. Solders can't handle 215°C and above	SYS-3-1 & SYS-3-2
SSR-POW-02	Constraint	The SA's power must be sized to the power generation values at aphelion + EOL degradation.	Analysis & Testing (A&T)	The SA will produce much lower power values near aphelion.	SYS-3-1
SSR-POW-03	Performance	The Power Control and Distribution Unit (PCDU) will regulate bus voltages to within ±4% of nominal values.	Analysis & Testing (A&T)	Voltage spikes must be mitigated to reduce possible damage towards subsystems.	SYS-3-4
SSR-POW-04	Operational	The PCDU will perform Max Peak Power Tracking (MPPT) along with Sequential Switching Shunt Regulation (S3R) to modulate SA power production.	Analysis (A)	With the large distance near aphelion and increased incidence angle near perihelion, MPPT will be required to maintain power array power regulation. S3R will also be required to dissipate any excess power from the system.	SYS-3-1 & SYS-3-4
SSR-POW-05	Operational	The PCDU will have redundant systems running hot to reduce any single failure point.	Testing (T)	Due to the communications lag between ground control and the spacecraft, any redundant systems must be able to quickly and autonomously swap.	SYS-3-4
SSR-POW-06	Operational	Secondary Lithium Ion batteries shall be used to supply power during inspection operations.	Analysis & Testing (A&T)	Inspection represents the highest loads on the spacecraft's electrical bus, with SA power generation being at unideal angles.	SYS-3-3

POWER
Power Budget

Quantity	Component	Total Power Usage (W)	Standby	Safe	Inspection
		Comms			
x1	Comms Array	Mode Specific	38 W	112 W	233 W
		C&DH			
x1	Phoenix Flight Computer	20 W	ON	ON	ON
x3	SSMM	~10 W	OFF	OFF	ON
x1	Ultra Stable Oscillator	6.5 W	ON	ON	ON
		Power			
x1	Command and Monitoring Module	~15 W	ON	ON	ON
		Thermal			
NA	Heaters	400 W	OFF	ON	ON
		ADCS/GNC			
x2	ASTRIX 200 IMU	44 W	OFF	ON	ON
x2	AA-STR Star Tracker	27 W	OFF	OFF	ON
x2	Jena FSS	14.4 W	OFF	ON	ON
x4	Reaction Wheels	80 W	OFF	ON	ON
x1	LiDAR Altimeter	18 W	OFF	OFF	ON
		Payload			
x1	ONC-W HK	15 W	OFF	OFF	ON
x1	LORRI HK	14.9 W	OFF	OFF	ON
x1	IR Camera HK	7 W	OFF	OFF	ON
	Sum Per Mode (W):		79.5 W	691.9 W	900.2 W

Results:

- Three power modes were chosen based off ConOps and equipment requirements
  - Standby
    - Low power/pre-separation mode
    - 79.5 W Requirement
  - Safe
    - Housekeeping/transit mode
    - 691.9 W Requirement
  - Inspection
    - Mission centered mode
    - All instruments on
    - 900.2 W Requirement

POWER
Battery Sizing

Sum Per Mode (W):	79.5 W	691.9 W	900.2 W
Max time per Mode during Inspection (hours):	0	2 Hrs	5.64 Hrs
Battery Requirements During Inspection (Watt*hours):	0	1383.8 Wh	5077.2 Wh

Assumptions

- 0.91 AU | 2.25 m^2 solar cell | Max 60° Inclination
  - 2882.2 Wh generated over inspection period
- +30% Final Wh for Battery DOD

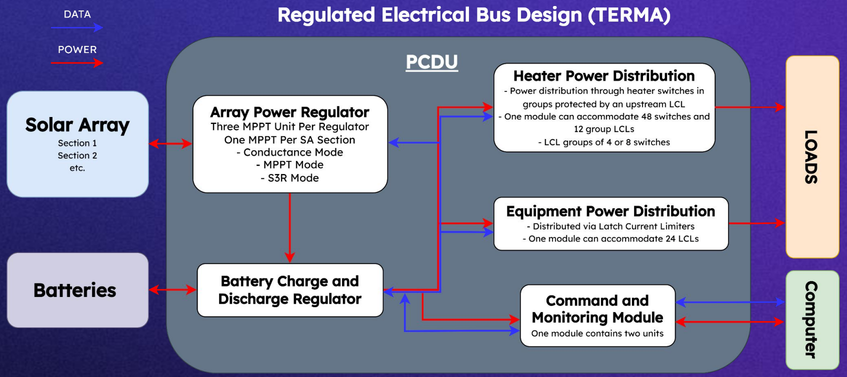
Results

- 5582.8 Wh required minimum

Battery Choice

- EnerSys ABSL 18650 Li-Ion Rechargeable Battery
  - Requires custom configuration 8s58p 28V pack
  - 5684 Wh
  - ~30 kg

POWER
Power Control and Distribution Unit



#	Equipment	OP Temp Range (°C)	Input Voltage Range	Output Voltage Range	Power BOL/EOL (W)	Notes
ADCS						
2	Fine Sun Sensors (FSS) from TNO	-30°C / +65°C	± 13/15 V	± 5 V	0.2	Power Requirements Per Channel
2	Inertial Meas. Units / Astrix 200	-10°C / +50°C	± 22/50 V	-	5.5 / 7.5	-
2	Star Trackers / AA-STR from Leonardo	-30°C / +60°C	± 20/110 V	-	5.6 to 12.6	Dependent on Temperature (review Data Sheet)
4	Reaction Wheel Assembly	-20°C / +50°C	± 24/51 V	-	20	Power draw dependent on mode (review Data Sheet)
1	LiDAR Altimeter	10ºC / +40ºC	-	-	18	Temp range is for operation. Storage temp range is -30ºC / +60ºC.
2	Phoenix Flight Computer	-30°C / +65°C	-	-	20	Can provide power to peripherals (IMU,FSS, AA-STR, etc.)
Communications						
#	**Equipment**	**OP Temp Range (°C)**	**Input Voltage Range**	**Output Voltage Range**	**Power Modes (W)**	**Notes**
1	Communications Assembly	-	± 28 V (Solar Orbiter)	N/A (elecrical load only)	Standby: 38 Safe: 112 Inspection: 233	One high gain antenna, multiple low gain
Payload						
#	**Equipment**	**OP Temp Range (°C)**	**Input Voltage Range**	**Output Voltage Range**	**Power BOL/EOL (W)**	**Notes**
1	LORRI from New Horizons	-125°C / +40°C	± 30±1 V	-	5 + 10	Camera Electrical Power + Camera Heater Power (Requires ±2.5/6/15 V internally)
1	ONC-W from Hayabusa2	-30°C / +40°C	-	-	14.9	Not power consumption for single ONC-W camera - value is for camera suite including electronics and 2 ONC-W cameras. No available input voltage data.
1	MDA Long Wave IR Camera	-35ºC / + 50ºC	± 24-34 V	-	7	-
Heaters						
#	**Equipment**	**OP Temp Range (°C)**	**Input Voltage Range**	**Output Voltage Range**	**Power BOL/EOL (W)**	**Notes**
NA	Electric Heaters / Minco Thermofoil	-200°C / +150°C	-	-	360 W	-

Specifications:

- TERMA manufactured PCDU
- 28 V Electrical Bus
- Array Power Regulator
  - Sequential Switching Shunt Regulation
  - Maximum Power Point Tracking

Reasoning

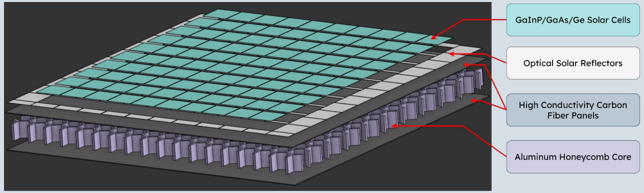
- Long flight heritage
  - 100+ years of inflight services on multiple modules
  - Modularity and willingness to work with consumers
- In-built hot redundancy systems

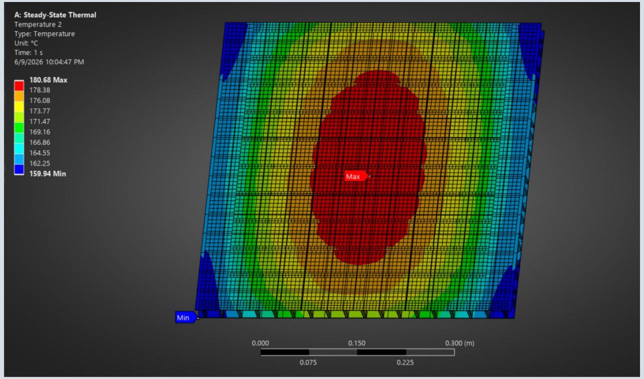
POWER
Solar Arrays Thermal Analysis

Solar Array Design
- Triple junction Gallium Arsenide solar cells
- Metalized Quartz Optical Solar Reflectors (OSR)
- 7075-T6 Aluminum Honeycomb Core
- Pitch carbon fiber with cyanate ester resin

Refined Thermal Analysis Results
- 180°C Thermal Maximum

Possible Improvements
- Larger model area





Array Material Properties	x	y	z					
	Length	Width	Thickness	Density (kg/m^3)	Cp (J/kg*K)	k (W/m*K)	abs	emis
Panel Size:	0.5 m	0.5 m	0.015 m	-	-	-	-	-
GaAS Cell Size:	40 mm	40 mm	0.15 mm	5416.666	330	50	0.91	0.85
OSR Size:	40 mm	40 mm	0.15 mm	2600	740	1.38	0.09	0.825
7075-T6 Aluminum Honeycomb Size:	0.5 m	0.5 m	0.01 m	2810	960	130	-	-
Carbon Fiber Sheet Size:	0.5 m	0.5 m	0.001 m	1720	1450	241	-	0.85

POWER

Solar Array Sizing

Degradation	Chosen	Range
Design and Assembly	0.85	0.77 - 0.90
Temperature of Array	0.8	0.8 - 0.98
Cell Shadowing	1	0.8 - 1.0
Total Inherent Degradation	0.68	0.49 - 0.88

Solar Energy	
Solar Luminosity (W)	3.83E+26
SC Min Distance (m)	1.36E+11
Energy Intensity (W/m^2)	1,643.82

Panel Power (III GaA)	
Panel Efficiency	30.00%
Inclination (Degrees)	0
Power Output (W/m^2)	335.34

Sizing Requirements

- Capable of sustaining ‘Safe’ mode without the use of battery power throughout mission lifetime
  - Sizing based off aphelion distance from the sun

Sizing Assumptions

- 0.91 AU
- 0° solar inclination
- 30% cell efficiency
- 32% degradation

Results

- 335.34 W/m^2
- ‘Safe’ mode requires 691.9 W + 15%
- Total solar cell area of ~2.4 m^2 (750 cells)
- 60% Solar Cell / 40% OSR
- Total panel area of 4 m^2

Verification & Validation
Power

Main Electrical Bus / PCDU

- Flatsat power performance tests
  - Verify power requirements and averages
  - Validate system architecture and system redundancy

Battery Systems

- Constant charge degradation analysis
  - Near constant excess power keeps batteries at full charge
- Environmental degradation analysis
  - Radiation exposure

Solar Arrays

- Degradation analysis
  - Thermal cycling
  - Radiation exposure

Risk Management

Power

Code	Failure Name	Effect	Mitigation
A	Solar Array Pointing Failure	Possibly causing critical lack of power generation at aphelion; solar array eventually overheats to above 215°C; likely a complete loss of power generation.	Solar Array Drive with inbuilt redundancies; dual-motor windings, redundant bearings, launch locks (launch vibrational damage.
B	Solar Array Degradation	As the spacecraft experiences extreme thermal and radiation fluctuations, the solar array might degrade more than expected causing critically low power generation at aphelion.	The solar cells chosen are capable of withstanding high radiation environments and the arrays themselves are double their necessary size ($8m^2$)
C	PDCU Voltage Regulation Failure	Possible shorting of critical mission systems; could result in total loss of mission if critical ADCS portions fail.	Voltage regulation for incoming/outgoing systems, solar array regulation, battery regulation, and harness shielding.
D	PDCU Event Failure	If a single failure event is capable of propagating beyond its system, it could cascade requiring ground help or lead to failure of critical systems.	Inbuilt hot redundant systems capable of handling single event failures, survival circuitry bypassing PDCU to allow sunpointing and solar array control during the event of a failure.
E	Battery Degradation	Battery degradation caused by constant full charge might be greater than expected, possibly leading to critical battery DOD.	With ground testing of battery systems based off lots used, the already assumed 30% excess for DOD, 20% on top for degradation, extreme max power draw assumptions, double the necessary SA area, and the ability to orientate inspection profiles to maximize SA availability during inspection.

Certain >80%					
Likely 60% - 70%					
Possible 40% - 59%					
Unlikely 20% - 39%		E			
Rare < 20%			D	B, C	A
	1 Insignificant	**2** Minor	**3** Moderate	**4** Major	**5** Severe

11. COMMAND & DATA HANDLING

COMMAND & DATA HANDLING

Sub-System Requirement

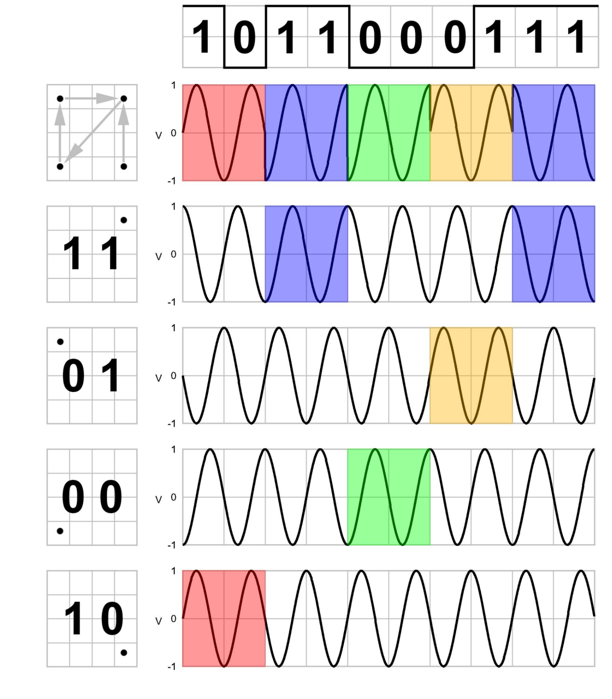
SSR-ID	Requirement Type	Description	Verified by	Rationale for requirement	Traces to
SSR-CDH-01	Constraint	C&DH shall store payload data and HK telemetry onboard during blackouts, including 30 days of HK storage.	Analysis, Test	Prevents data loss during heliocentric communication gaps and bounds SSMM sizing.	SYS-5-1
SSR-CDH-02	Operational	C&DH shall packetize telemetry using CCSDS TM Space Data Link and Space Packet protocols.	Analysis, Test	Ensures compatibility with ESTRACK/DSN ground processing and efficient telemetry handling.	SYS-5-1, SYS-5-2
SSR-CDH-03	Operational	C&DH shall receive, authenticate, decode, and route uplink commands over MIL-STD-1553B Bus-A and Bus-B.	Test	Provides reliable command handling for operations, proximity approval, and contingencies.	SYS-5-2
SSR-CDH-04	Operational	C&DH shall autonomously initiate FDIR responses within 60 seconds of anomaly detection.	Analysis, Test	Enables fast fault response when ground intervention is delayed or unavailable.	SYS-5-1, SYS-5-2
SSR-CDH-05	Operational	C&DH shall maintain mission time synchronization using the USO within ±10 μs over 24 hr and ±1 μs after ranging.	Analysis, Test	Supports accurate telemetry correlation, inspection timestamping, and blackout operations.	SYS-5-1, SYS-5-3

COMMAND & DATA HANDLING

How do we guarantee the signals are not lost during downlink?

Layered Downlink Protection: Margin + Coding + Detection + Retransmission

1. **Link margin**
   - HGA and LGA cases close with ≥ 6 dB margin
2. **Robust modulation and coding**
   - QPSK used for efficient deep-space downlink
   - CCSDS rate-1/3 channel coding lowers required (E_b/N_0)
3. **Error detection**
   - Downlink data is packetized into CCSDS transfer frames
   - Frame CRC/FEC checks reject corrupted frames before image
4. **Store-and-retransmit**
   - Inspection imagery stays in the **96 GiB SSMM** until ground receipt is confirmed
   - Dropped or corrupted frames can be re-requested on later ESTRACK passes reconstruction

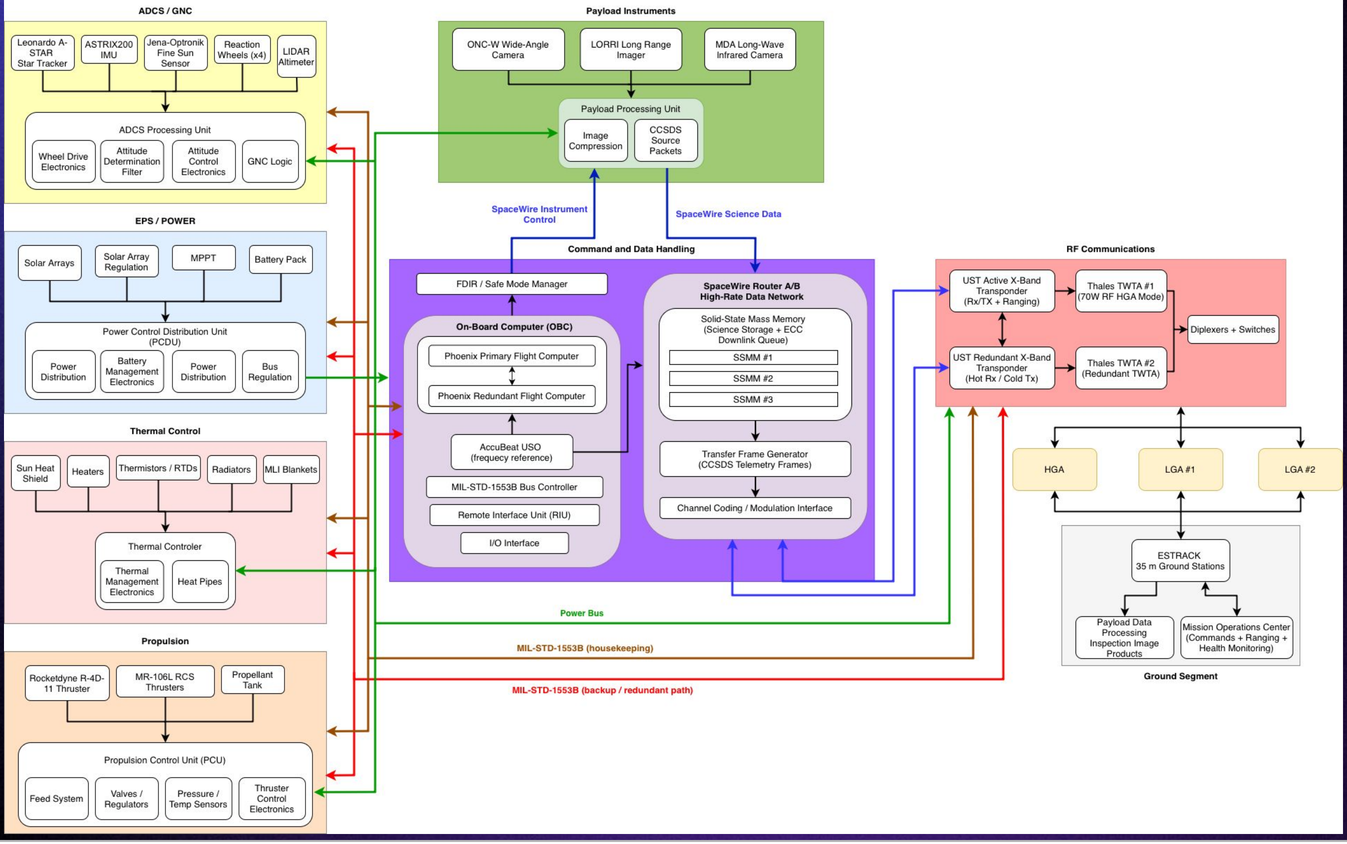


https://www.etti.unibw.de/labalive/experiment/qpsksignalgeneration/

ODIN does not depend on a perfect pass. If a frame is corrupted or missed, it is detected, retained onboard, and retransmitted later.

COMMAND & DATA HANDLING

System Architecture



ODIN

VERIFICATION & VALIDATION
Command & Data Handling

Storage integrity

- SSMM capacity, ECC protection, and store-and-forward operation verified through write/read-back, error injection, and blackout data-storage tests.

Protocol compatibility

- OBC telemetry packetization verified against **CCSDS** packet structure for all housekeeping and mission-mode data.

Command reliability

- Uplink commands verified for authentication, decoding, and routing over both **MIL-STD-1553B**.

Autonomous fault response

- FDIR and OBC redundancy tested through injected faults, including computer lockup, SSMM ECC errors, UST failure, and power-limit events.

High-rate science path

- SpaceWire throughput tested at peak inspection imaging load from payloads to SSMM.

End-to-end validation

- Integrated test verifies the full chain: **ground command → UST → OBC → SSMM → packetization → RF downlink → ground.**

RISK MANAGEMENT
Command & Data Handling

Code	Failure Name	Effect	Mitigation
A	Software memory error / bit flip	Flight computer malfunctions; incorrect commanding.	Dual Phoenix MCUs monitor each other; faulted unit is reset/reprogrammed. EDAC protects internal flash.
B	Processor lockup / hang	No command execution; satellite unresponsive.	Hardware watchdog reset, safe-mode entry, and beacon transmission for recovery.
C	Chip / hardware failure	Loss of primary flight computer; potential mission failure.	Hot-standby redundant Phoenix unit with automatic switchover.
D	Single bit error	Stored payload data inaccurate or corrupted.	24-hr data scrubbing; EDAC corrects single-bit errors.
E	SSMM chip failure	Mass memory unavailable; no payload data storage.	Phoenix falls back to internal flash for housekeeping and beacon data.
F	Multi-bit / uncorrectable error	Payload data block lost; partial science data loss.	Corrupted blocks flagged/excluded from downlink; re-observation if timeline permits.

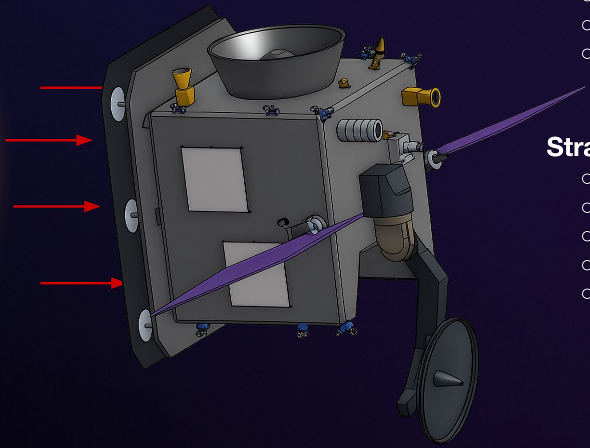
Certain >80%					
Likely 60% - 70%					
Possible 40% - 59%		D	A, B		
Unlikely 20% - 39%			F		
Rare < 20%		E			C
	1 Insignificant	**2** Minior	**3** Moderate	**4** Major	**5** Severe

12. THERMAL

THERMAL
Subsystem Requirements

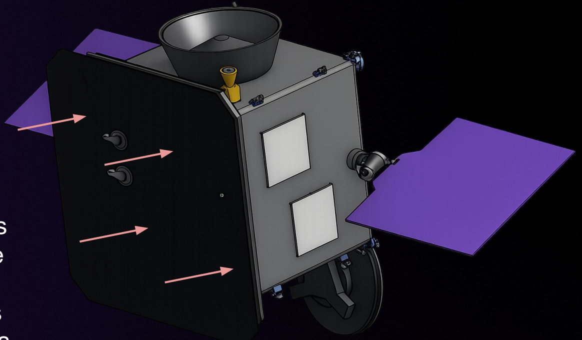
SSR-ID	Requirement Type	Description	Verification	Rationale	Traceability
SSR-THM-01	Functional	The TCS shall maintain all spacecraft hardware within qualified survival temperature limits throughout all mission phases between 0.28 AU and 1.0 AU.	Analysis Test	ODIN will experience extreme thermal cycling between perihelion and launch conditions that may expose spacecraft components to temperatures outside their qualified survival limits.	SYS-6-1 SYS-6-2
SSR-THM-02	Performance	The TCS shall maintain all mission=critical hardware within qualified operational temperature limits during all nominal mission operations between 0.28 AU and 1.0 AU.	Analysis Test	Components operating at perihelion may require more limiting temperature ranges than survival ranges to ensure operation.	SYS-1-1 SYS-1-2 SYS-1-3 SYS-2-1 SYS-4-2
SSR-THM-03	Functional	The TPS shall protect the spacecraft bus within a conical shadow from a solar flux of 17,500 W/m^2 encountered at 0.28 AU.	Analysis	Components operating at aphelion may require more limiting temperature ranges than survival ranges to ensure operation.	SYS-1-1 SYS-3-1 SYS-6-1
SSR-THM-04	Performance	The active thermal control system shall provide thermostatically controlled heating sufficient to maintain all heater-equipped hardware within operational temperature limits during the cold-case mission environment.	Test	The spacecraft will encounter extreme solar flux at 0.28 AU that may damage spacecraft structures, electronics, and externally mounted hardware without adequate thermal protection.	SYS-2-1 SYS-3-1 SYS-6-1
SSR-THM-05	Performance	The passive TCS shall reject internally generated and environmental heat loads sufficient to maintain spacecraft hardware within qualified operational temperature limits during hot-case mission conditions.	Analysis Test	Certain spacecraft components may require supplemental heating to maintain operational or survival temperatures during cold mission phases.	SYS-1-1 SYS-3-1 SYS-5-2 SYS-6-1
SSR-THM-06	Constraint	The thermal subsystem mass and power consumption shall not exceed the allocations defined by the spacecraft system design.	Inspection	Spacecraft internal and external heat loads during perihelion conditions may exceed allowable spacecraft thermal limits without passive heat rejection capability.	SYS-5-2

THERMAL
Thermal Environment



Challenges
- 10.4 : 1 solar flux ratio
- Solar events
- Radiation environment (TID)
- Extreme temperatures
- Low temperatures

Strategies
- Modeling to predict temperatures
- Shield from direct solar exposure
- Coatings to protect materials
- Heaters for cold-case operations
- Radiators for hot-case operations



Sun

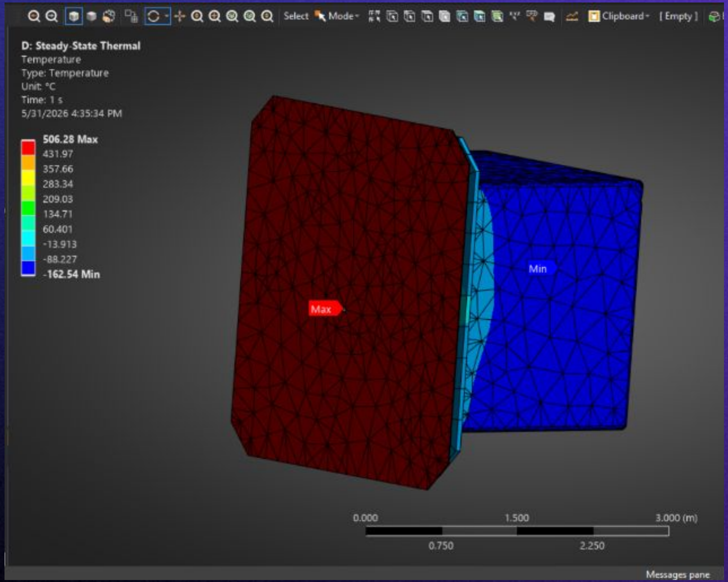
Perihelion = 0.28 AU

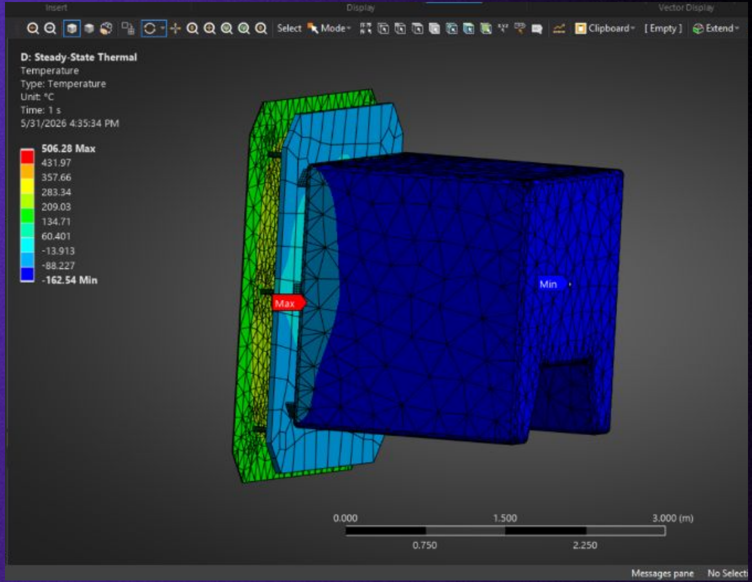
17,500 W/m^2

Aphelion = 0.90 AU

1,680 W/m^2

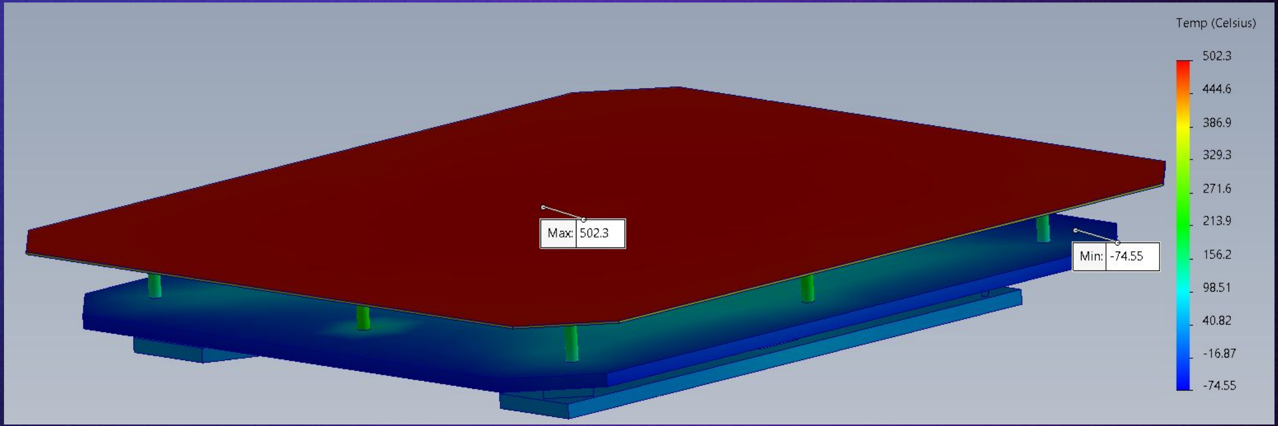
THERMAL
Models: Ansys





Bus panel temperatures are too cold

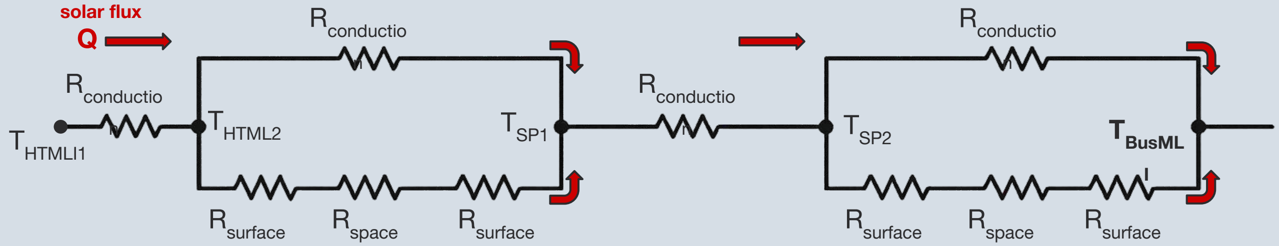
THERMAL
Models: Solidworks



Surface	ODIN Temp °C	SolO Temp °C
HTMLI Top	502	505
HTMLI Bottom	389	280
SP Top	116	200
SP Bottom	86	20
SC MLI	69	55

THERMAL
Model Methods

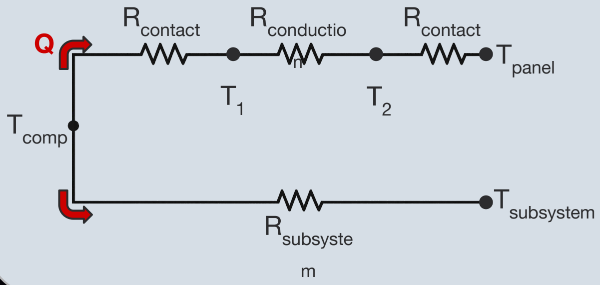
Heat Shield



Assumptions

Heat Shield:
- Steady-state
- 1D through-thickness
- Effective HTMLI conductivity
- Diffuse-gray radiation
- Edge effects neglected
- Constant thermo-optical properties

Components:
- Steady-state
- 1D through-thickness
- Lumped subsystem temp node
- $R_{subsytem}$ is a lumped resistance
- Mounted with baseplate to panel
- Fasteners modeled as $R_{contact}$



Components

$$R_{cond} = \frac{L}{kA}$$

$$R_{contact} = \frac{R''_c}{A}$$

$$R_{surface} = \frac{1-\varepsilon}{\varepsilon A}$$

$$A_1 F_{12} = A_2 F_{21}$$

(View Factor Reciprocity)

$$R_{space} = \frac{1}{AF_{12}}$$

Terms

ODIN

THERMAL

Model Methods (cont.)

Multi - Nodal Model

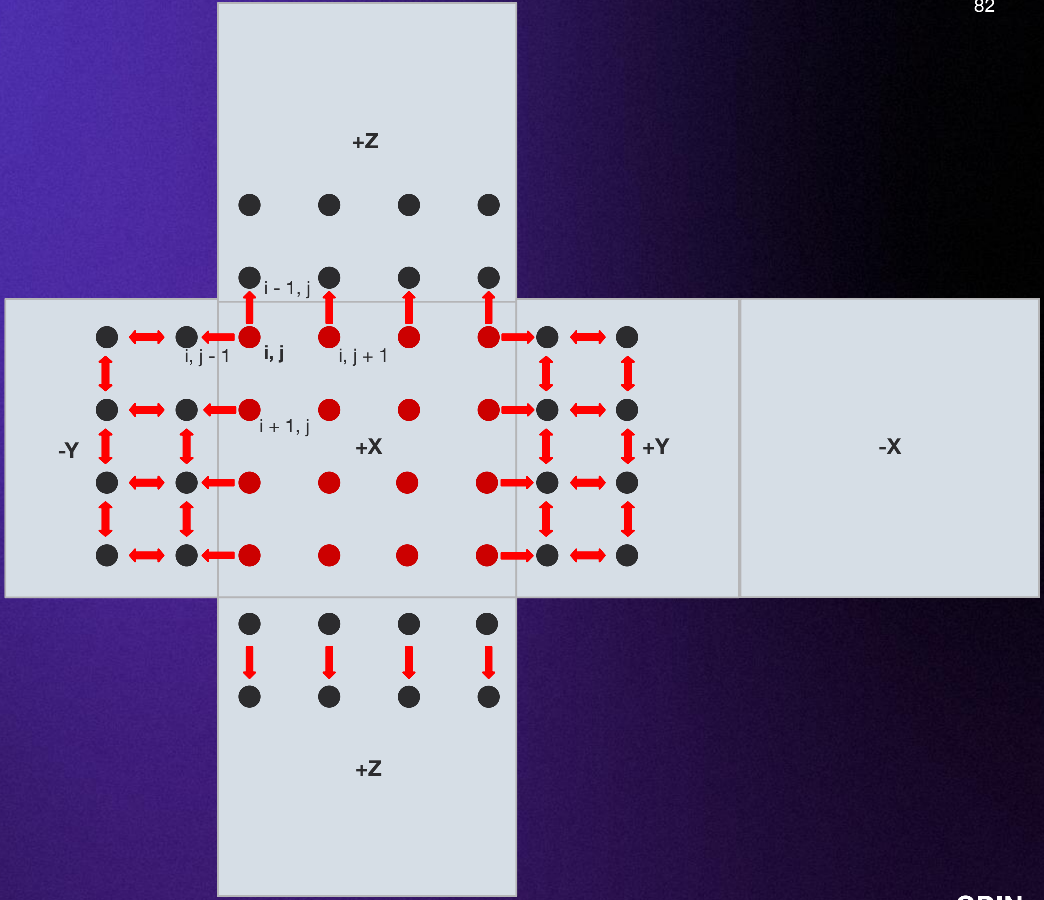
$$q_{i-1,j\to i,j}+q_{i+1,j\to i,j}+q_{i,j-1\to i,j}+q_{i,j+1\to i,j}+q_{int\to i,j}+q_{rad\to i,j}+\dot{q}_{gen}=0$$

$$kA_y\frac{T_{i-1,j}-T_{i,j}}{\Delta y}+kA_y\frac{T_{i+1,j}-T_{i,j}}{\Delta y}+kA_x\frac{T_{i,j-1}-T_{i,j}}{\Delta x}+kA_x\frac{T_{i,j+1}-T_{i,j}}{\Delta x}+\frac{T_{int}-T_{i,j}}{R_{int}}+h_rA_{rad}(T_{space}-T_{i,j})+\dot{q}_{gen}=0$$

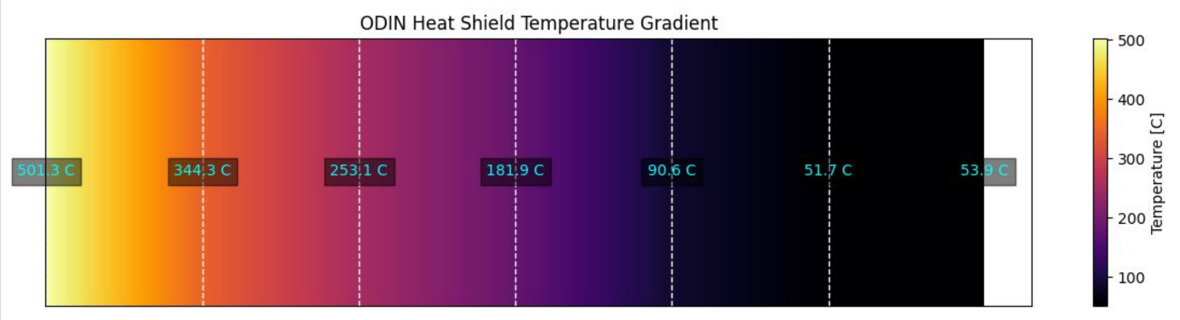
$$T_{i,j}=\frac{\frac{kA_y}{\Delta y}T_{i-1,j}+\frac{kA_y}{\Delta y}T_{i+1,j}+\frac{kA_x}{\Delta x}T_{i,j-1}+\frac{kA_x}{\Delta x}T_{i,j+1}+\frac{T_{int}}{R_{int}}+h_rA_{rad}T_{space}+\dot{q}_{gen}}{\frac{2kA_y}{\Delta y}+\frac{2kA_x}{\Delta x}+\frac{1}{R_{int}}+h_rA_{rad}}$$

$$\Delta x=\frac{L_{bus}}{N-1} \qquad h_r=\varepsilon\sigma(T_{i,j}+T_{space})(T_{i,j}^2+T_{space}^2)$$

$$\Delta y=\frac{L_{bus}}{N-1}$$

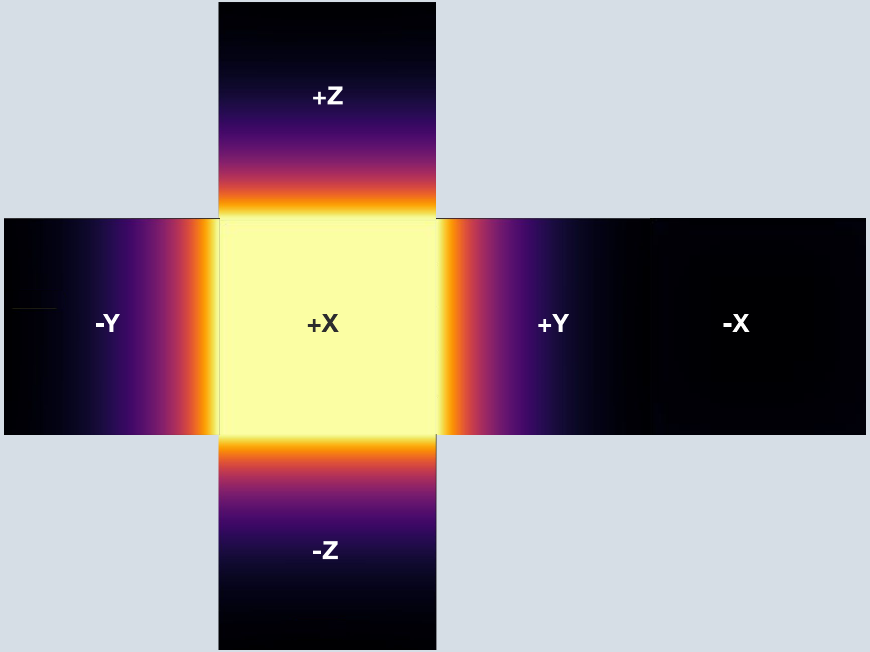


THERMAL
Model: Python



Surface	Perihelion T °C	Aphelion T °C	0.59 AU T °C
HTMLI Outer	501	160	258
HTMLI Inner	344	123	195
SP Outer	253	38	107
SP Inner	182	21	78
Bus MLI	91	-39	7
Bus Panel	**52**	**-78**	**-32**

THERMAL
Model: Python (cont.)



External Surface	Perihelion T °C	Aphelion T °C
Front Face (+X)	52	-77
Lateral Faces	44 to 17	-78 to -87
Back Face (-X)	16	-88

THERMAL
Heating Budget

Component	Perihelion Heating (W)	Aphelion Heating (W)
IMU	0	14
RWs	0	0
ADCS Processing Unit	0	30
Maximum Power Point Tracker	0	0
Batteries	0	61
Power Control Distribution Unit	0	34
Thermal Controller	0	0
Propulsion Control Unit	0	0
Payload Processing Unit	0	43

Component	Perihelion Heating (W)	Aphelion Heating (W)
Phoenix Flight Computer	0	0
Solid State Mass Memory	0	31
Ultra Stable Oscillator	0	25
SpaceWire Router	0	25
X-Band Transponder	0	20
TWTA	0	0
Propellant Tanks	0	47

Max Heating (W)	344 (~400 w/ 15% margin)
Max Cooling (W)	340

$$Q_{rad} = \varepsilon \sigma A \left(T_{rad}^4 - T_{space}^4\right)$$

$$A = \frac{Q_{rad}}{\varepsilon \sigma \left(T_{rad}^4 - T_{space}^4\right)}$$

$A_{radiator} = 1.1\ m^2$

ODIN

THERMAL
Coatings/Multi-Layer Insulation

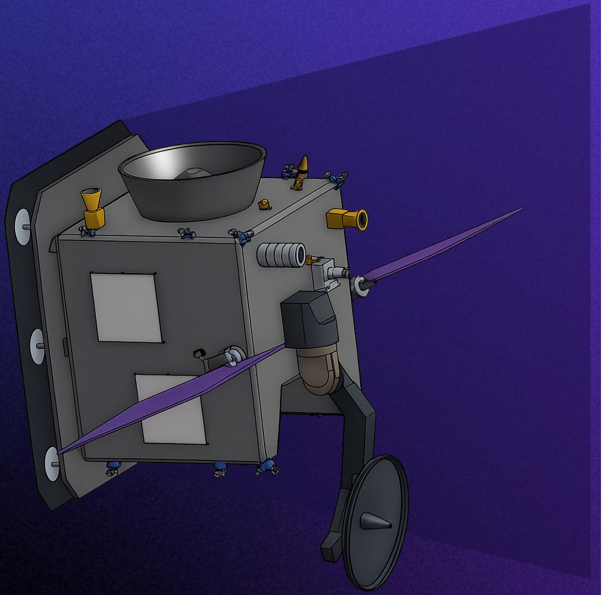
Component	MLI/Coating	α	ε
Bus	Upilex/Aluminized Kapton	0.35	0.60
Heat Shield	SolarBlack	0.96	0.78
Radiators	SolarWhite	0.24	0.83
Sun Sensors	SolarBlack	0.96	0.78
Star Trackers	Aeroglaze Z306	0.95	0.9
HGA	SolarWhite	0.24	0.83
LORRI	Upilex/Germanium Black Kapton	0.92	0.90
ONC	Upilex/Germanium Black Kapton	0.92	0.90
Long Wave IR Camera	Upilex/Germanium Black Kapton	0.92	0.90
LIDAR	Upilex/Germanium Black Kapton	0.92	0.90
Thrusters	Titanium-Upilex/SolarBlack	0.96	0.78

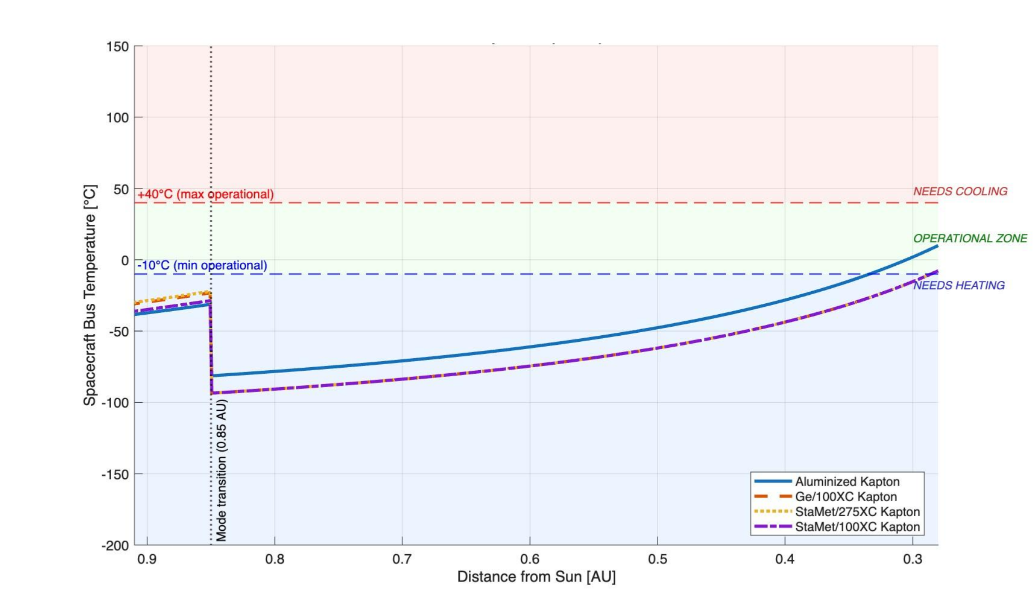
Considerations

- Absorptivity to emissivity ratio to understand radiative heat transfer
- Electrical conductivity to prevent charge buildup
- Long-term stability across mission orbit

THERMAL
Bus Coatings/MLI

13 Layer Upilex MLI
Aluminized Kapton Coating





ODIN

THERMAL
SolarBlack and SolarWhite

SolarBlack

- Heat Shield coating
- Calcium Phosphate based coating
- Stable thermo-optical properties

Solar Orbiter Testing		BOT	EOT
UV (26000 Equivalent Sun Hours) VUV (19,500 Equivalent Sun Hours) Electron (234 hrs at 60 KeV and 0.5nA/cm^2) Proton (2.5 hrs at 60 KeV and 1 nA/cm^2)	Absorption (α_s)	0.96	0.96
	Emissivity (ε)	0.78	0.78
Thermal Endurance (960 hrs at 600°C) Hot (20 cycles between 40°C and 550°C) Cold (20 cycles between -196°C and 50°C)	Absorption (α_s)	0.96	0.96
	Emissivity (ε)	0.78	0.78

SolarWhite

- Radiator coating
- Stable thermo-optical properties
- Low α/ε

Testing		BOT	EOT
VUV (7444 Equivalent Sun Hours – 3 years GEO) Electron (equivalent 1 year 400 KeV) Proton (equivalent I yr hr at 240 and 45 KeV)	Absorption (α_s)	0.24	0.28
	Emissivity (ε)	0.83	0.83
Thermal cycling under vacuum: 100 cycles between +180ºC and -180ºC	Absorption (α_s)	0.18	0.18
	Emissivity (ε)	0.84	0.84

VERIFICATION & VALIDATION
Thermal

Heat shield performance

- Heat shield thermal-vacuum testing verifies bus temperatures remain within allowable limits under representative high-flux conditions

Thermal Model Correlation

- Mathematical model correlated against thermal balance test data to predict spacecraft temperatures, heater requirements, and thermal margins through the mission

Thermal Cycling

- Spacecraft components subjects to thermal cycling to verify survival and functionality across mission temperature extremes

RISK MANAGEMENT

Thermal

Code	Failure Name	Effect	Mitigation
A	Heat Shield Damage	Excessive heat leakage into the bus causing overheating and physical damage	Test the heat shield's structural durability to evaluate if stronger materials are needed.
B	Heat Shield Degradation	Higher than predicted bus temperatures over mission life	Material degradation testing
C	Heater Failure	Components fall below operational/survival temperatures ranges at aphelion	Redundant heaters
D	Bus MLI Damage	Increased heat loss requiring higher heater demand	MLI inspection, careful installation, thermal margin with heating demand
E	Prolonged Side Panel Exposure to Solar Flux	Solar flux to bus surface and components leading to temperatures exceeding limits	Bus and component MLI and coatings maximize emission and minimize absorption and conduction to inside bus

Certain >80%					
Likely 60% - 70%					
Possible 40% - 59%					
Unlikely 20% - 39%			D, E		
Rare < 20%			B, C	A	
	1 Insignificant	**2** Minior	**3** Moderate	**4** Major	**5** Severe

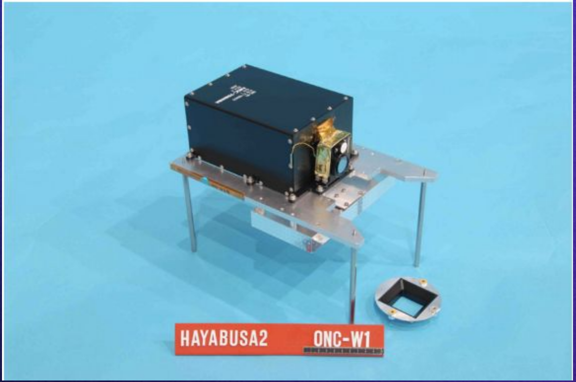
13. PAYLOAD

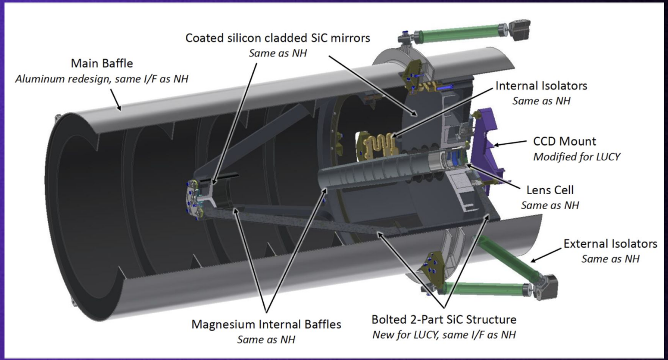
PAYLOAD

Requirements and Traceability

SSR-ID	Requirement Type	Description	Verified by	Rationale for requirement	Traces to
SSR-PAY-01	Functional	Payload shall acquire and frame Solar Orbiter using ONC-W before detailed inspection imaging	Testing and Simulation	Ensures target is identified before close-range imaging begins.	SYS-1-1 SYS-1-3
SSR-PAY-02	Performance	LORRI shall capture visible inspection imagery and identify surface features as small as 5mm at inspection range.	Analysis and Testing	Enables detection of visible surface damage and external anomalies.	SYS-1-2 SYS-1-3
SSR-PAY-03	Performance	LWIR camera shall collect thermal imagery sufficient to identify thermal anomalies on Solar Orbiter.	Analysis	Adds thermal condition assessment beyond visible imagery.	SYS-1-3 SYS-5-3
SSR-PAY-04	Performance / Safety	Laser Altimeter shall provide repeated range measurements during target acquisition and inspection hold points.	Testing and Analysis	Confirms standoff distance, supports framing, and reduces proximity risk.	SYS-2-2 SYS-2-3
SSR-PAY-05	Interface / Data	Payload shall store inspection image and range data in a format that can be returned to Earth within mission data limits.	Analysis	Ensures all required inspection data can be saved and downlinked.	SYS-5-1 SYS-5-3 SYS-6-2

PAYLOAD
Camera Suite Components and Capabilities





Wide Optical Navigation Camera (ONC-W)
- Designed to support navigation
- Wide Angle FOV of 65.24° x 65.24°
- Capable of acquiring Solar Orbiter at a standoff distance of 1.51km

Long Range Reconnaissance Imager
- Fixed Focus Telescopic Imager
- Narrow Angle FOV of 0.29° x 0.29°
- Can resolve required minimum size surface features at a standoff distance of 505m

PAYLOAD
Camera Suite Components and Capabilities

MDA Long Wave Infrared Camera

- Designed for performance in space applications
- 45° x 33° FOV with 1024 x 768 resolution
- Capable of detecting heat patterns across surface of Solar Orbiter within our inspection range

Laser Altimeter

- Measures slant range by firing laser pulses and timing the return
- Range of 30m to 25km
- Originally made for asteroid surface ranging in deep space

PAYLOAD
Inspection Concept of Operations

1. Pre-Inspection

Checkout phase before close range imaging begins.

Confirm cameras and laser altimeter are ready before committing to inspection hold point.

ONC-W powered on for wide field target acquisition, other cameras on standby.

LA performs initial range checks.

2. Target Acquisition and Framing

ONC-W captures wide angle images of Solar Orbiter.

Solar Orbiter location, orientation, and configuration is confirmed.

LA verifies intended inspection distance, pointing refined.

3. Inspection Hold Point

Once Solar Orbiter is acquired, stability is confirmed at inspection hold point.

Relative motion, pointing jitter, and blur risk are checked before detailed imaging.

4. Wide Area Survey Imaging

Perform initial survey pass using ONC-W and LWIR.

ONC-W to capture broad visual images of heat shield, solar arrays, antennas, and Solar Orbiter body.

LA continuously recording range values to tie to image timestamps.

5. Region of Interest Selection

Ground contact decides which regions deserve higher resolution imaging, directing LORRI's pointing.

Looking for heat shield discoloration, scratches or dents on exposed surfaces, panel swelling or deformation, and thermal anomalies.

PAYLOAD
Inspection Concept of Operations

6. Narrow Angle Feature Identification

LORRI becomes the main inspection camera.

ONC-W keeps target area framed at a wider scale, LA continuously confirming standoff distance.

High resolution images of surface features are captured.

7. Thermal Anomaly Follow Up

After visible feature imaging, LWIR to follow up on regions that show thermal degradation.

Thermal images are compared against visible images, hot or cold spots are flagged for review.

8. New Angle / Repeat Imaging

Repeated imaging of the same face is done at this stage if inspection of face is incomplete.

Included as a precaution for unintended viewing geometry changes.

9. Data Validation

After image collection, checks for whether data is usable before leaving inspection geometry.

Images are checked from excessive blur, missed framing, poor exposure, etc.

Once a single face is fully inspected and validated, viewing geometry transitions to next face.

10. Downlink and Assessment

Upon completion of inspection, stored data is to be sent to ground.

Ground can then decide if even more imaging is required.

PAYLOAD

Complete Inspection Data

Camera	Data Generated Per Image	Images Taken Per Face	Total Images Per Inspection	Total Data Generated Per Inspection
ONC-W	0.8853 MB/image	6	36	31.87 MB
LORRI	1.1531 MB/image	6	36	41.51 MB
LWIR	0.8400 MB/image	44	264	221.76 MB

Total camera data generation comes out to about 300 MB per inspection.

VERIFICATION AND VALIDATION
Payload

End-to-End Payload Operations

- Inspection sequence simulation
  - Validate target acquisition, range confirmation, and multi-angle imaging
  - Verify Payload data storage and downlink readiness
- Environmental capability analysis
  - Radiation exposure and payload degradation risk

Camera Functionality and Practical Image Quality Testing

- Verify wide field target acquisition
- Verify high resolution inspection imaging
- Validate exposure and image clarity

Laser Altimeter

- Range measurement performance testing
  - Verify accurate distance measurements
- Sensor Cross-Checking
  - Compare laser range with camera-based target size estimates

RISK MANAGEMENT
Payload

Code	Failure Name	Effect	Mitigation
A	Target Not Properly Acquired by ONC-W	Solar Orbiter is not centered or identified, inspection sequence cannot begin reliably.	Use ONC-W wide-field imaging first. Verify target location before switching to detailed imaging.
B	LORRI Target Misalignment / Narrow FOV Miss	High-resolution images may miss Solar Orbiter or only capture partial surfaces.	Require inspection hold point before LORRI imaging. Use ONC-W framing and laser altimeter range data to confirm pointing.
C	Spacecraft Jitter / Motion Blur During Imaging	Images may be blurred, reducing ability to identify surface damage or anomalies.	Coordinate imaging with ADCS pointing stability. Also capable of retaking images if quality check fails.
D	Laser Altimeter Range Dropout / Incorrect Ranging	Incorrect standoff distance may affect framing, scaling, and inspection safety.	Take repeated range measurements and reject outliers. Cross-check range using apparent target size in camera images.
E	LWIR Calibration Error / Thermal Saturation	Thermal images may produce false hot or cold spots, or even miss real thermal anomalies.	Perform pre-inspection calibration, and compare LWIR results with expected thermal models and visible imagery.
F	Payload Data Volume Exceeds Storage or Downlink Margin	Some inspection images may be delayed or lost.	Prioritize critical images, ensure proper image compression, stagger image collection and downlink.

Certain >80%					
Likely 60% - 70%					
Possible 40% - 59%			C F	A B	
Unlikely 20% - 39%		E	D		
Rare < 20%					
	1 Insignificant	**2** Minior	**3** Moderate	**4** Major	**5** Severe

ODIN

14. COMMUNICATIONS

COMMUNICATIONS
Sub-System Requirements

SSR-ID	Requirement Type	Description	Verified by	Rationale for requirement	Traces to
SSR-COMM -01	Constraint	ODIN shall use X-band communication for uplink and downlink.	Analysis, Test	X-band supports deep-space range performance and compatibility with ESTRACK 35 m DSAs.	SYS-5-2, SYS-5-1
SSR-COMM -02	Constraint	The ground system shall receive and process ODIN X-band telemetry.	Review	Ensures ground station compatibility with ODIN's communication link.	SYS-5-2, SYS-5-1
SSR-COMM -03	Performance	The link budget shall maintain ≥6 dB margin from 0.7 AU to 2.0 AU.	Analysis	Provides margin for pointing loss, atmospheric effects, and range variation.	SYS-5-2
SSR-COMM -04	Operational	ODIN shall downlink inspection payload data within available ESTRACK contact windows.	Analysis, Test	Ensures inspection imagery and proximity data can be returned during daily ESTRACK contacts.on.	SYS-5-1, SYS-5-3
SSR-COMM -05	Operational	ODIN shall downlink spacecraft housekeeping telemetry for health monitoring.	Test	Ensures spacecraft health is continuously monitored through HK telemetry.	SYS-5-1, SYS-5-3
SSR-COMM -06	Operational	ODIN shall receive and route uplink commands to onboard subsystems.	Test	Required for mission planning, proximity operations, and contingency response.	SYS-5-2

COMMUNICATIONS

Sub-System Requirements (Operations Ground Segment)

SSR-ID	Requirement Type	Description	Verified by	Rationale for requirement	Traces to
SSR-MC-01	Constraint	ODIN MOC shall use ESA ESTRACK 35 m DSAs as the primary ground network.	Review	ESTRACK provides X-band compatibility, multi-station coverage, and reduced interface risk.	SYS-5-2
SSR-MC-02	Constraint	MOC shall schedule ≥1 daily contact: 6 hr/day during cruise and 8 hr/day during inspection.	Analysis	Supports spacecraft tracking, command uplink, and timely anomaly response.	SYS-5-2, SYS-5-1
SSR-MC-03	Performance	MOC shall process and distribute telemetry within 4 hr after end-of-pass.	Test	Enables timely engineering review before inspection GO/NO-GO decisions.	SYS-5-1, SYS-5-3
SSR-MC-04	Operational	MOC shall generate, validate, and uplink command sequences for nominal, TCM, and contingency ops.	Test	Required for ground-planned maneuvers, payload activation, and inspection sequencing.	SYS-5-2
SSR-MC-05	Operational	MOC shall archive telemetry and payload data for the mission duration plus ≥5 years.	Review	Supports post-mission analysis and lessons learned for future inspection missions.	SYS-5-1, SYS-5-3
SSR-MC-06	Failure Tolerance	MOC shall maintain backup operations so commanding is not interrupted for more than 24 hr.	Analysis	Reduces single-point failure risk during deep-space and proximity operations.	SYS-5-2

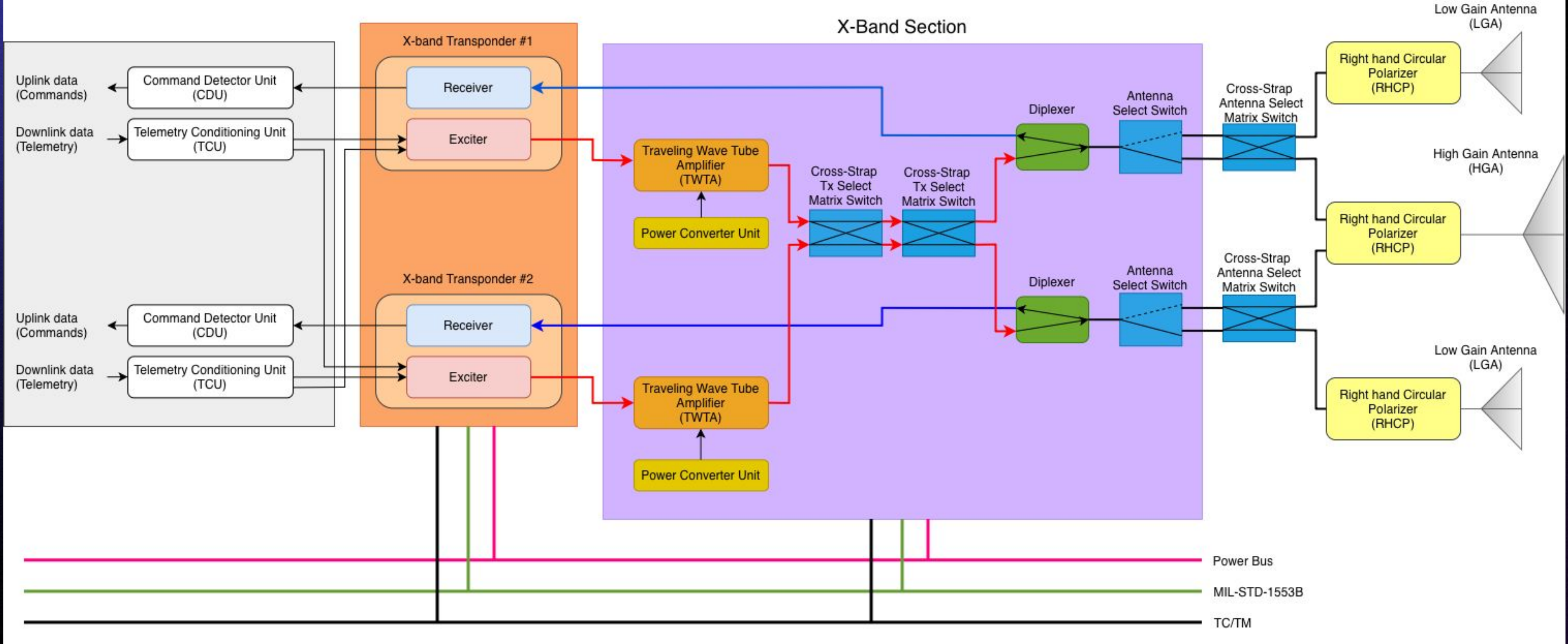
COMMUNICATIONS
Ground Segment

- 35 m parabolic tracking antenna, X-band uplink (7.17 GHz) / downlink (8.42 GHz), compatible with ESA ESTRACK and Solar Orbiter.
- Single ground station baseline; multi-station coverage for extended contact windows/backups depending on the location of ODIN relative to the Earth. (ESA ESTRACK + NASA DSN)
- New Norcia as our Mission Operation Center (Solar Orbiter Heritage)
- ~ 8 hours/day of communication

New Norcia Ground Station (MOC)

https://www.esa.int/Enabling_Support/Operations/ESA_Ground_Stations/Estrack_ESA_s_global_ground_station_network

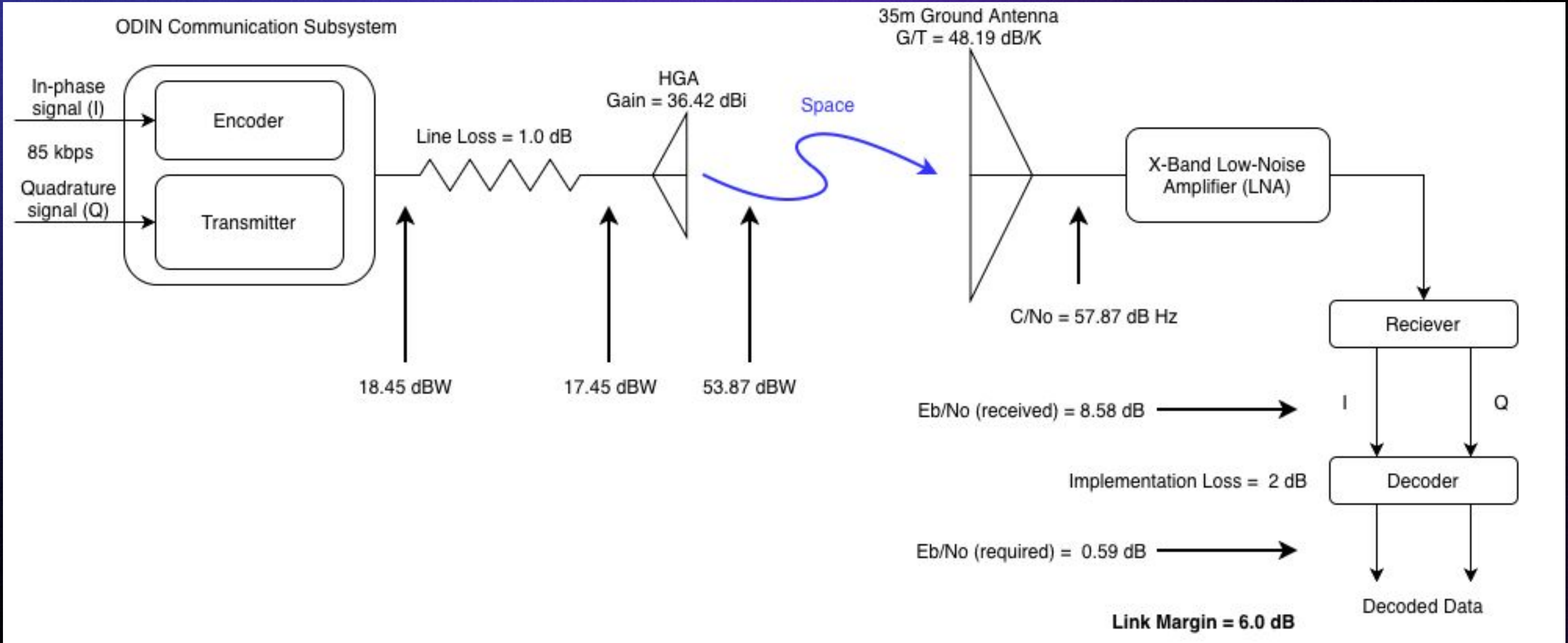
COMMUNICATIONS
System Diagram for Redundant X-Band HGA & LGA Suite



ODIN

COMMUNICATIONS

Link Budget Calculation Scheme for HGA Downlink



COMMUNICATIONS
Link Budget

Parameter	Units	HGA Far-Range Case	HGA Nominal Case	Inspection Case 1	Inspection Case 2	LGA Far-Range Case	LGA Nominal Case
Mission Mode		Reduced-rate HGA downlink	Nominal HGA downlink	Inspection image downlink	Inspection image downlink	Safe-mode TT&C only	Safe-mode TT&C only
Earth-ODIN Range	km	299,195,741.4	149,597,870.7	114,000,000	109,370,000	299,195,741.4	149,597,870.7
Approximated Range	AU	2.0	1.0	0.76	0.73	2.0	1.0
Downlink Net Path Loss	dBW	-280.98	-274.95	-272.59	-272.23	-280.98	-274.95
Downlink Received Carrier Power	dBW	-160.11	-154.09	-151.73	-151.37	-194.06	-188.04
Downlink Receiver C/N_0	dB-Hz	49.49	55.51	57.87	58.23	15.54	21.56
Available Downlink Eb/N_0	dB	8.63	8.61	8.58	8.74	8.55	8.55
Implementation Loss	dB	2.00	2.00	2.00	2.00	2.00	2.00
Final Link Margin	dB	6.0	6.0	6.0	6.1	6.0	6.0
Total Capacity	**kbps**	**12.2**	**49.0**	**85.0**	**89.0**	**5**	**20**

COMMUNICATIONS

Mode-Wise Data Generation and Downlink Activity

Component	Data Rate (kbps)	Standby	Cruise	Pre-Inspection	Approach	Inspection	Downlink	Safe	Conjunction
Comms									
HGA Imagery Downlink	12.2 - 89	OFF	OFF	OFF	OFF	OFF	ON	OFF	OFF
HGA Cruise Downlink	50	OFF	ON	OFF	OFF	OFF	OFF	OFF	OFF
LGA Telemetry	0.35	OFF	OFF	OFF	OFF	OFF	OFF	ON	OFF
UST Receiver (Uplink)	0.5	ON	ON	ON	ON	ON	ON	ON	OFF
Thales TWTA	0.096	OFF	ON	OFF	OFF	OFF	ON	ON	OFF
APME Gimbal	0.064	ON	ON	ON	ON	ON	ON	OFF	OFF
Power + Thermal + C&DH (Always ON) & Payload (ON during Inspection) & ADCS + Propulsion (ON/OFF)									
Total (kbps)		**1.38**	**55.94**	**241.86**	**242.23**	**694.50**	**90.83 – 94.83**	**5.86**	**5.17**

COMMUNICATIONS
Housekeeping Data Rate

Subsystem	Measurements Included	HK Data Rate
C&DH	Flight computer health, memory, FDIR status, SSMM utilization/ECC errors, Time Keeping	160 bps
Thermal	Component temperature, heat shield temperature, heater status, solar array temperature	244 bps
Power	Bus voltage/current, battery voltage/temp, MPPT regulator telemetry	399 bps
ADCS	IMU rates/accelerations, star tracker quaternion, sun sensor angle, reaction wheel telemetry	4,352 bps
Propulsion	Thruster chamber pressure, valve status, pulse count, tank pressure/temp, propellant flow	381 bps
Payload	Camera housekeeping, IR camera status, laser altimeter status	64 bps
Communications	UST RF status, TWTA telemetry, HGA gimbal position/current/status	288 bps
Total		**5,888 bps**

COMMUNICATIONS
Comms ConOps

Before inspection:
Verify SEP angle, ground contact, and approach geometry.

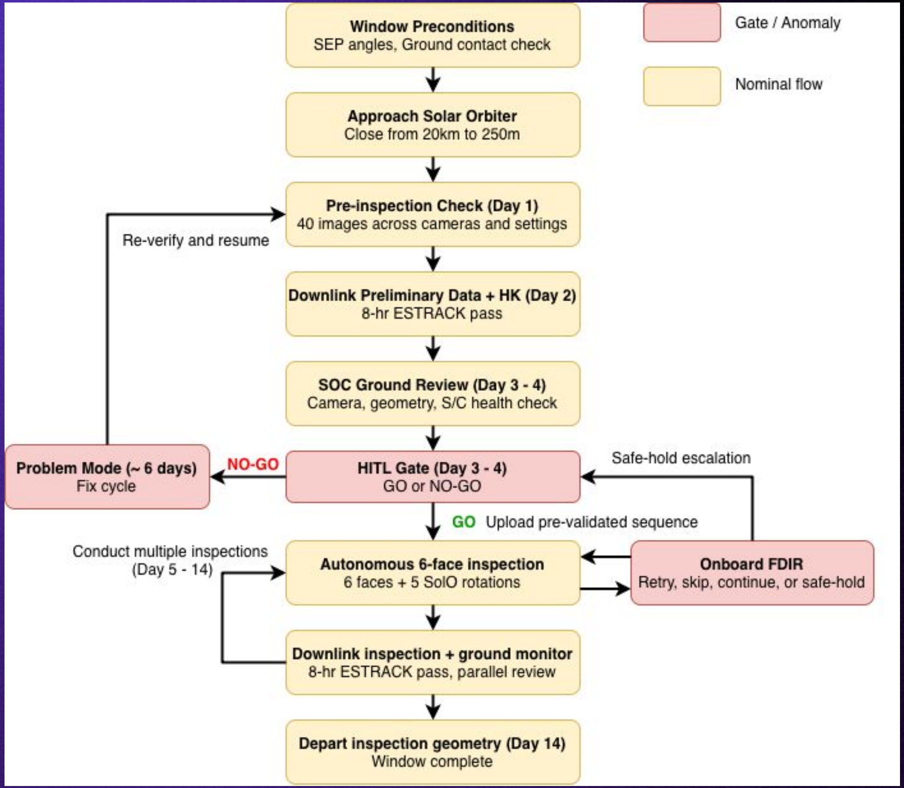
Day 1–2:
Collect calibration imagery and downlink preliminary data + HK through ESTRACK.

Day 3–4:
Ground review and HITL GO/NO-GO decision.

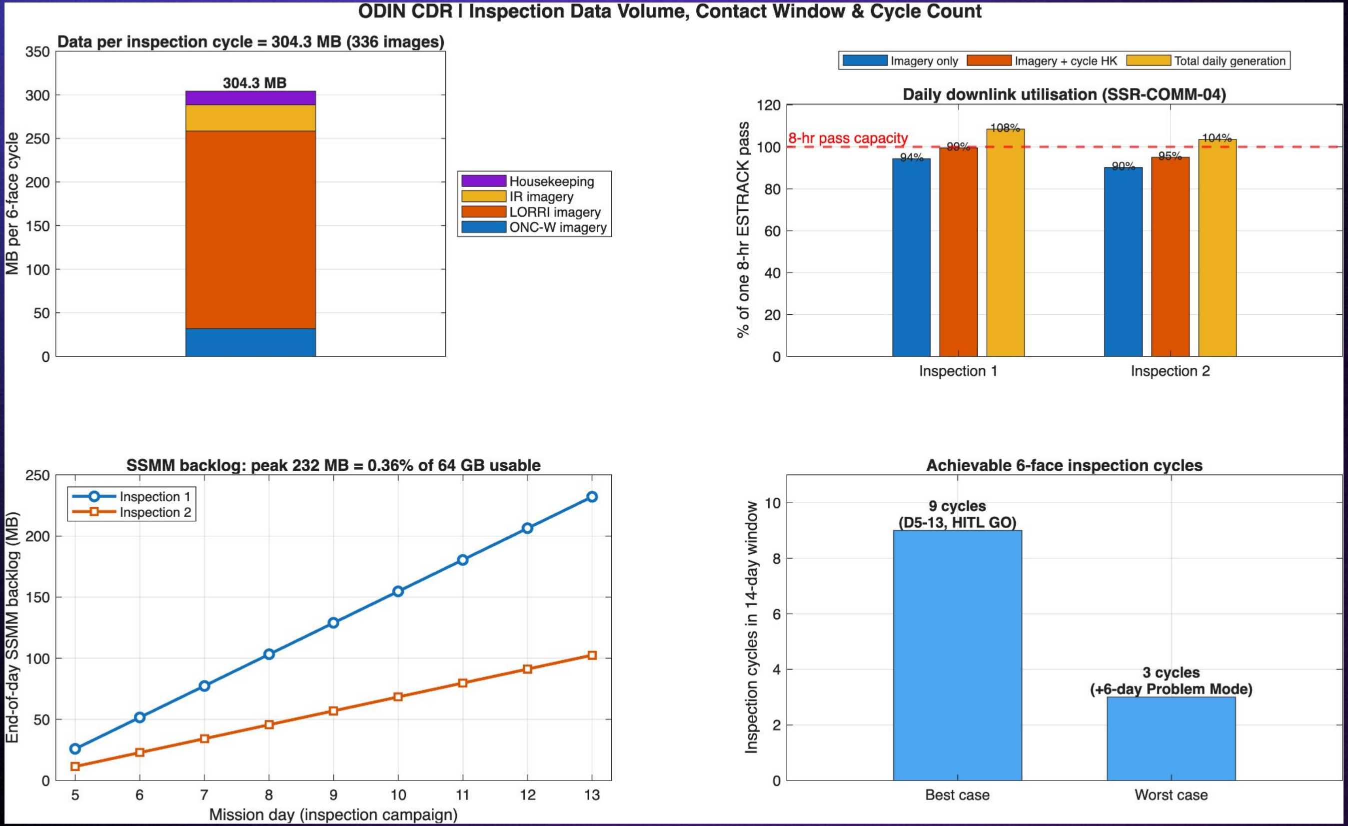
Inspection phase:
Upload pre-validated sequence, perform autonomous 6-face inspection, and store data onboard.

After each cycle:
Downlink inspection data during 8-hour ESTRACK contacts for parallel ground review.

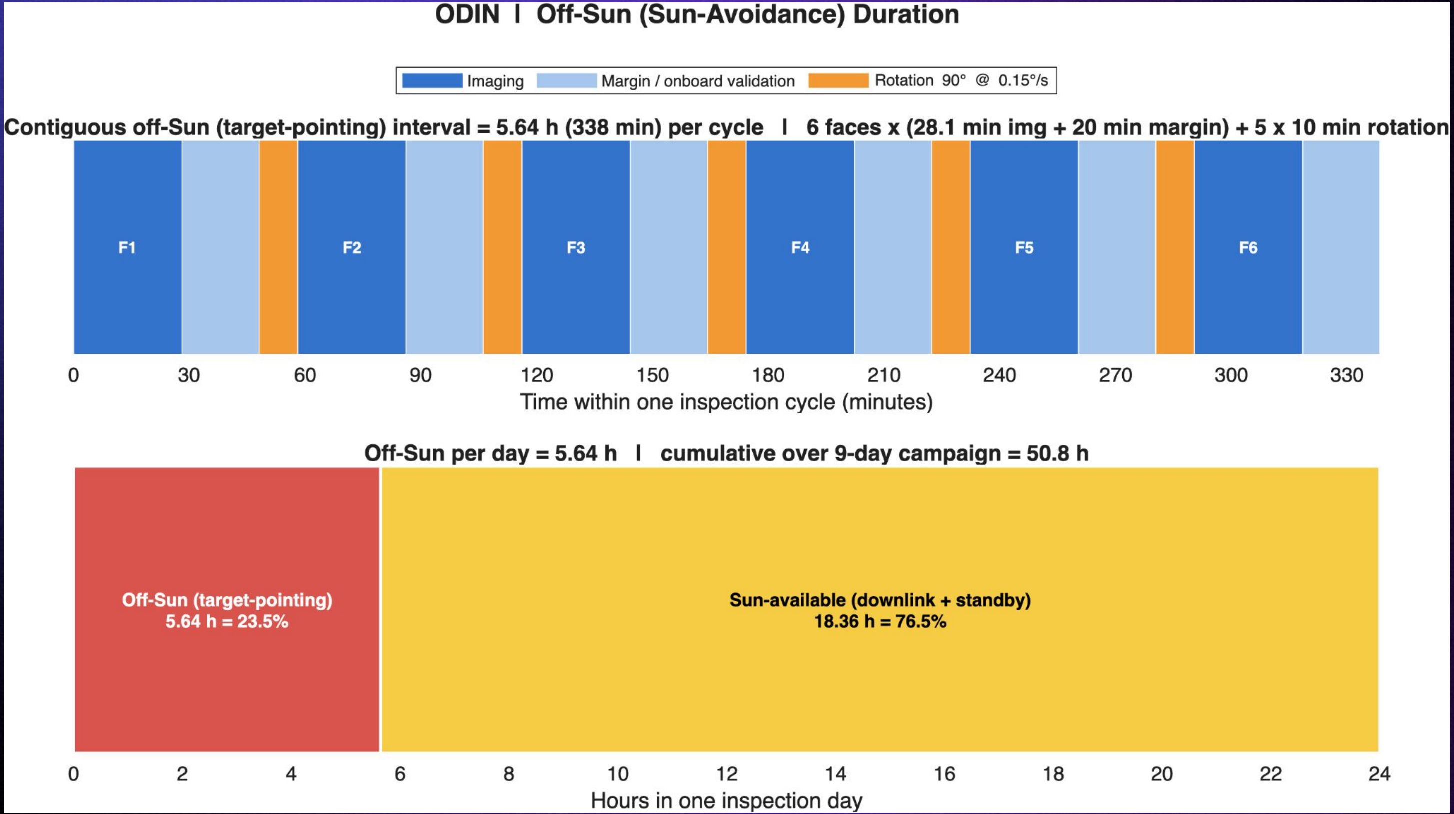
Contingency path:
Onboard FDIR handles local faults; NO-GO cases enter Problem Mode before re-verification.



COMMUNICATIONS
Contact Window Analysis



COMMUNICATIONS
Inspection Duty Cycle



ODIN

VERIFICATION & VALIDATION

Communications

Link performance by analysis

- HGA inspection, HGA cruise, and LGA safe-mode links verified to close with **≥6 dB margin.**

Inspection data return

- Contact-window analysis verifies inspection data clears within planned **8-hr ESTRACK passes**.

Ground compatibility

- ESTRACK X-band compatibility reviewed for **7.17 GHz uplink / 8.42 GHz downlink**.

End-to-end RF tests

- UST, TWTA, HGA gimbal, RF switch matrix, and payload-data downlink tested at representative rates.

Fault tolerance

- Safe-mode test verifies ODIN can switch to **LGA-based TT&C** after HGA failure.

RISK MANAGEMENT
Communications

Code	Failure Name	Effect	Mitigation
A	Transponder power failure	Loss of uplink/downlink.	Hot-standby redundant transceiver with automatic switchover.
B	Firmware crash	Transceiver becomes unresponsive.	OBC watchdog reset and backup firmware image.
C	Oscillator drift	Frequency offset and possible signal reacquisition.	AFC compensation; ground re-acquisition if needed.
D	TWTA overheating	HGA downlink loss or reduced throughput.	Thermal monitoring, duty-cycle limits, power reduction, LGA recovery link.
E	TWTA HVPS failure	Loss of RF amplification.	Fallback to low-power UST/LGA housekeeping and beacon downlink.
F	TWTA cathode degradation	Gradual loss of link margin.	Link margin allocation and reduced data-rate mode.
G	HGA deployment failure	Loss of high-rate downlink.	Redundant deployment mechanism, secondary attempt, LGA emergency link.
H	HGA pointing error	Reduced link margin and data rate.	ADCS repointing and updated ground pointing solution.
I	LGA chip failure	Loss of emergency omni link redundancy.	Use HGA in degraded mode; assess single-point risk.

Certain >80%					
Likely 60% - 70%		F			
Possible 40% - 59%			H		
Unlikely 20% - 39%		C	D		
Rare < 20%		I	B, E		A, G
	1 Insignificant	**2** Minior	**3** Moderate	**4** Major	**5** Severe

ODIN

15. CONCLUSION

CONCLUSION
Reflection

- Ambitious and expensive mission, but possible
- Largest unknowns:
  - Trajectory/GNC (Falcon Heavy)
  - Propulsion
  - Budget/Funding
- Near flagship-level support required

Non-Mission Remarks:

- Massive learning experience (both technical and otherwise)
- Huge thanks to our professors and instructors :)

THANK YOU


041
Inspection of Cooperative Target: ODIN
NAVIGATION & SYSTEM ENGINEERING
ADCS, PAYLOAD, POWER, & COMMUNICATIONS
THERMAL, STRUCTURES, & PROPULSION
06/04/2025

THANK YOU


043
041
Inspection of Cooperative Target: ODIN
NAVIGATION & SYSTEM ENGINEERING
ADCS, PAYLOAD, POWER, & COMMUNICATIONS
THERMAL, STRUCTURES, & PROPULSION
06/04/2025