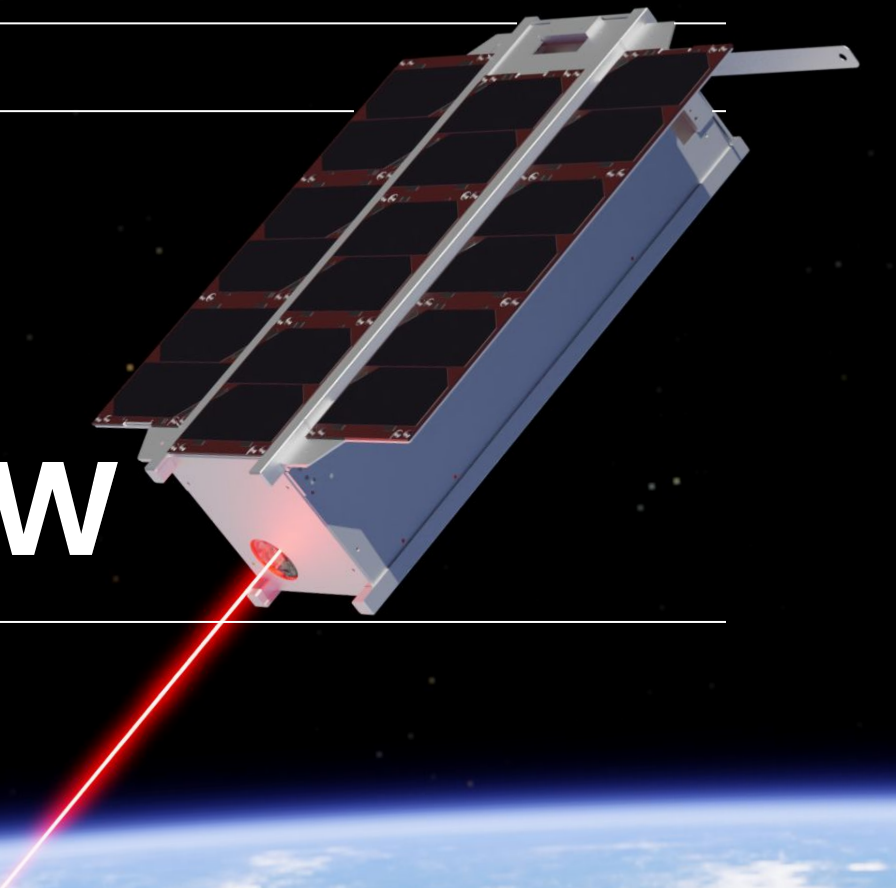
UCHICAGO SPACE PROGRAM

PULSE-A
PRELIMINARY DESIGN REVIEW

NOVEMBER 23 2024



Polarization-Mod**u**lated **L**aser **S**atellite **E**xperiment

PULSE-A

Preliminary Design Review

Introduction

Presenter Introductions

Logan Hanssler — Project Director

Seth Knights — Chief Engineer

Maya Shah McDaniel – Lead Payload Engineer

Graydon Schulze-Kalt — Lead Avionics Engineer

Robert Pitu – Lead Structural Engineer

Juan Prieto – Lead Ground Station Engineer

John Baird – Lead Systems Engineer

Catherine Todd – Flight Software Engineer

Spencer Shelton – Flight Software Engineer

University of Chicago Space Program (UCSP)

- The University of Chicago Space Program (UCSP) is a Registered Student Organization at the University of Chicago and a chapter of Students for the Exploration and Development of Space (SEDS)
- PULSE-A is UCSP's flagship project and the first mission of the organization's CubeSat Laboratory
- PULSE-A is the University of Chicago's first undergraduate-designed satellite and the sole opportunity for students interested in aerospace engineering to gain hands-on experience

A Brief History of UCSP

As the university's premier aerospace engineering organization, UCSP has grown from a small student group launching low-power rockets near campus to over 100 active members with a 66-student team working to develop PULSE-A.

Rockets Launched on Midway Plaisance Park (circa 2021)

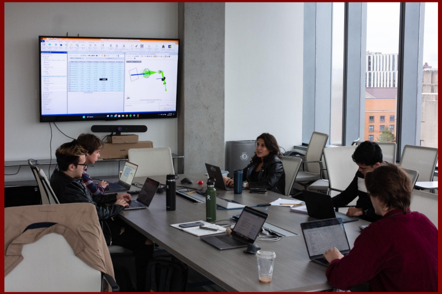
Hosted SpaceVision 2022 at the University of Chicago (Nov. 2022)

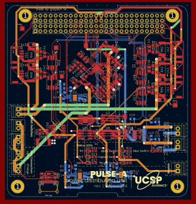
Developing PULSE-A (Development Apr. 2023 - Present, Team Photo from Oct. 2024)

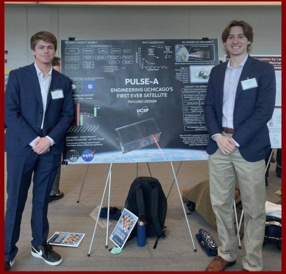
Continuing Higher-Caliber Rocketry Projects (Photo from May 2023)

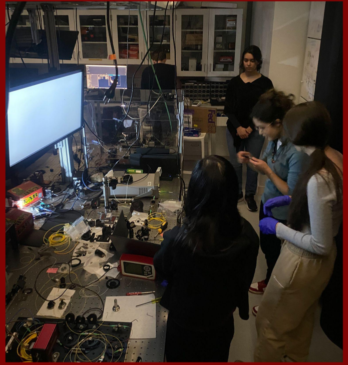
How the PULSE-A Team Got Here

- ❑ The project was founded in April 2023
- ❑ Accepted by Call 15 of NASA's CubeSat Launch Initiative in April 2024
- ❑ Completed System Requirements Review on June 2, 2024






PDR Overview

PDR Purpose

- The Preliminary Design Review (PDR) demonstrates that the preliminary design meets all system requirements with acceptable risk and within the program constraints and establishes the basis for proceeding with a detailed design.

- PDR provides evidence that the correct design options have been selected, interfaces have been identified, and verification methods have been described. It addresses and resolves critical, system-wide issues and shows that work can begin on the critical detailed design.

PDR Entrance Criteria

1. Successful completion of the SRR.

2. A preliminary PDR agenda and agreed upon success criteria by the technical team and project director prior to the PDR.

3. Availability of PDR technical products listed below for both hardware and software system elements to the cognizant participants prior to the review:
   a. Updated baselined documentation, including SRD TBDs.
   b. Systems Engineering Management Plan (SEMP).
   c. Preliminary subsystem design specifications for each configuration item, with supporting trade-off analyses and data.
   d. Updated risk assessment and mitigations.
   e. Assembly, integration and test logistics flow.
   f. Applicable technical plans.
   g. Applicable standards.
   h. Interface control drawing.

Not Included in the PDR

- ❑ Detailed cost estimates will be provided separately
- ❑ The following detailed design performance analyses will be provided at CDR
  - ❑ Detailed power generation and energy budget analysis
  - ❑ Detailed attitude control performance analysis
  - ❑ Structural loads analysis
  - ❑ Detailed thermal load analysis
  - ❑ Detailed communications analysis (after radio selection)
  - ❑ Detailed software architecture and performance
  - ❑ Integration and test flow logistics

PDR Agenda

Mission Overview

Principal Mission Objective

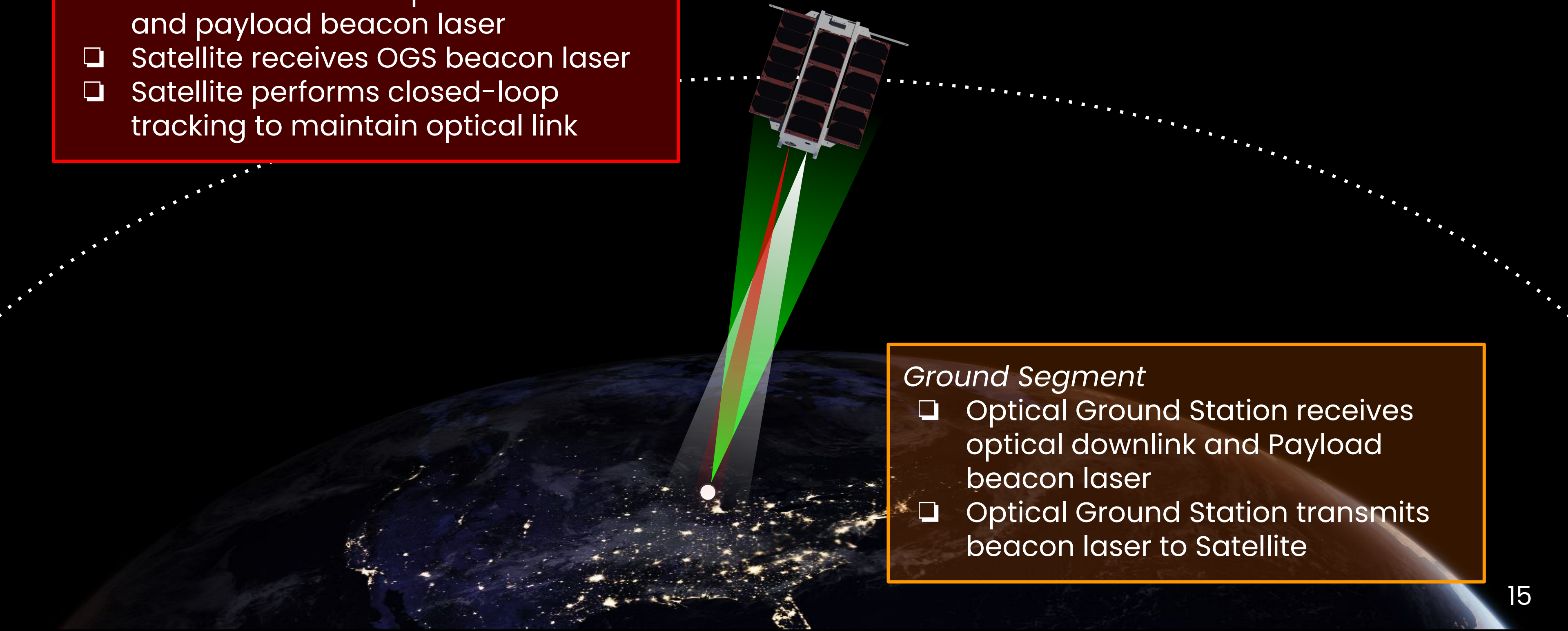
The mission shall execute Satellite-to-ground data transmission using circular polarization modulation, qualified by:

- ❑ An uncoded data rate of 1 to 10 megabits per second
- ❑ A consecutive transmission time of 5 to 150 seconds
- ❑ A coded bit error rate of 1e-9 to 1e-3.

Mission Concept

Space Segment:

- ❑ Satellite transmits optical downlink and payload beacon laser
- ❑ Satellite receives OGS beacon laser
- ❑ Satellite performs closed-loop tracking to maintain optical link



Ground Segment

- ❑ Optical Ground Station receives optical downlink and Payload beacon laser
- ❑ Optical Ground Station transmits beacon laser to Satellite

Follow-Up Mission Risk Reduction

PULSE-A functions as a risk reduction mission for a follow-up 3U CubeSat mission named PULSE-Q (Quantum). PULSE-Q will demonstrate space-to-ground quantum key distribution (QKD) using an optical downlink. The design for PULSE-A is intended to be repurposable with minimal modifications:

❑ Optical Payload: Must be modified for QKD transmission, but the foundation of an optical payload which modulates a laser transmission's polarization state remains the same.

❑ Bus: Repurposable after minor modifications for improved performance.

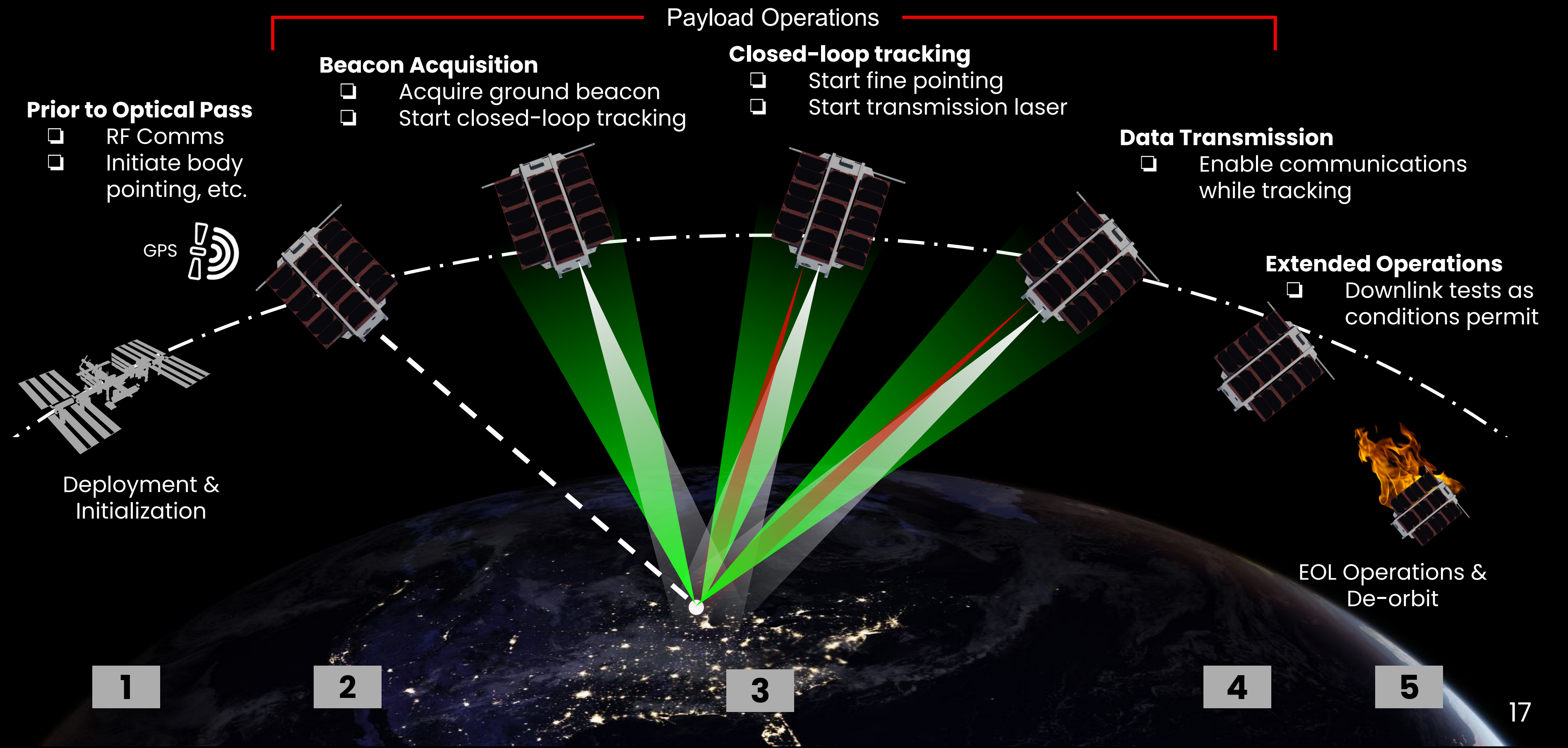
❑ Optical Ground Station (OGS): Must be modified to separate and detect 4 different polarization states, but the foundation of an OGS which tracks the CubeSat and interprets a laser downlink's polarization state remains the same.

❑ Radio Frequency Ground Station (RFGS): Repurposable as-is.

❑ Pointing, Acquisition, and Tracking (PAT) Sequence: Repurposable after minor modifications for improved performance.

The student knowledge base and documentation established during PULSE-A's development will significantly assist in the development of PULSE-Q and potential future missions.

Concept of Operations



Concept of Operations Considerations

1. Launch, Deployment, and Initialization

- ❑ Orbital parameters bounded by ~45 inclination through Sun Synchronous Orbit, ~90 deg inclination, to maximize launch opportunities.

2. Prior to Payload Operations

- ❑ GPS and RF communication must be established to enable accurate orbital propagation models for ground station tracking

3. Payload Operations

- ❑ Ideal launch timeline has payload operations primarily in summer (May-September) to maximize student availability and favorable weather conditions

4. Extended Operations

- ❑ Communications experiments are continued for as long as is feasible

5. End of Life Operations

- ❑ Natural orbit decay and re-entry, must happen within 5 years of mission end

Mission Success Status Determination

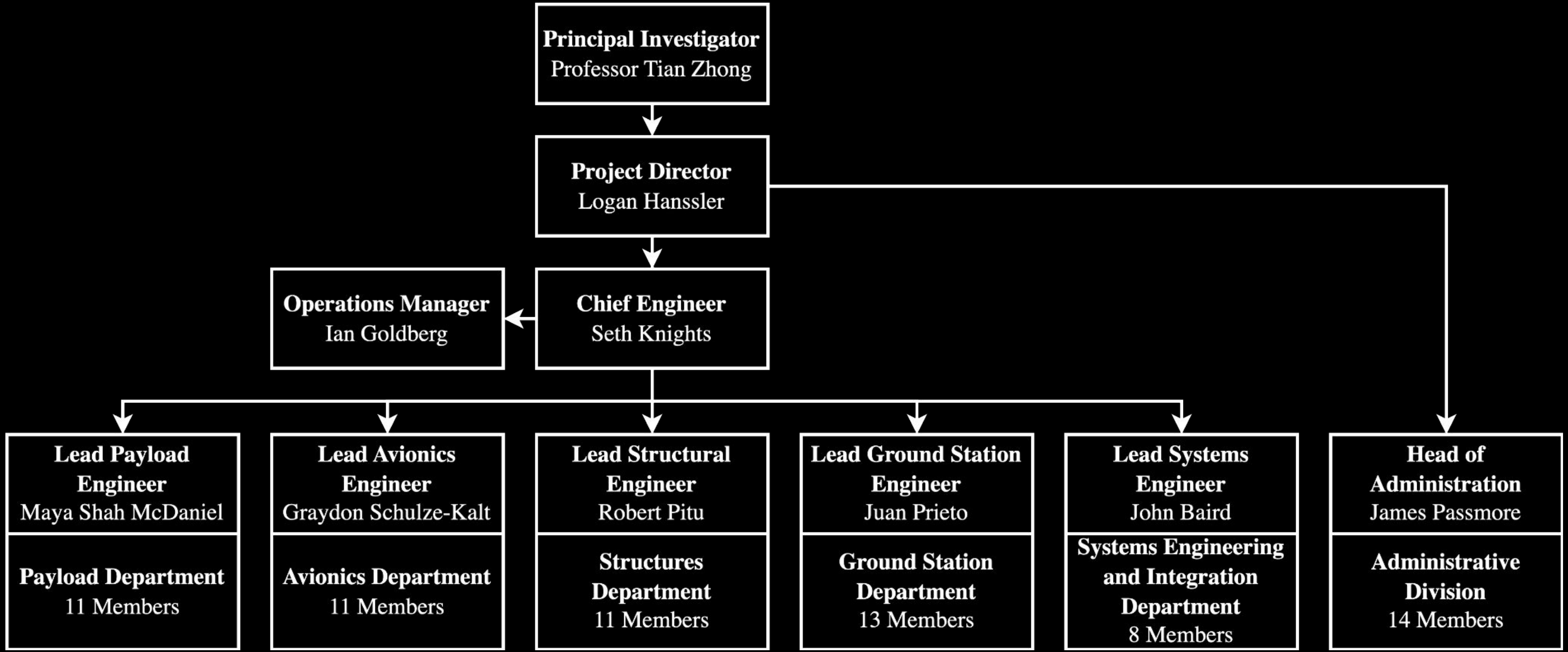
Success Status	Mission Result
Complete Success	Circular polarization-modulated downlink is successfully transmitted and received. All qualifications of mission success are achieved (raw data rate of 1 - 10 Mbps, link sustained for 5 - 150 seconds, 1e-3 - 1e-9 bit error rate).
Partial Success	Circular polarization-modulated downlink is successfully transmitted and received. At least one but not all qualifications of mission success are achieved (raw data rate of 1 - 10 Mbps, link sustained for 5 - 150 seconds, 1e-3 - 1e-9 bit error rate).
Limited Success	Circular polarization-modulated downlink is successfully transmitted and received, but no qualifications of mission success are achieved OR optical downlink is transmitted by the Satellite and detected by the OGS but cannot be interpreted.
Failure	Satellite is operational, but optical downlink cannot be transmitted by the Satellite OR Satellite is operational, but optical downlink cannot be received by the OGS.
Complete Failure	Satellite is not operational.

Secondary Project Objectives

1. Provide UChicago's undergraduates, who lack a traditional engineering program, an opportunity to design, build, assemble, and fly a fully-functioning CubeSat.
2. Prepare applied science students for careers in Aerospace, Electrical, Computer, and Mechanical engineering.
3. Design and build as much hardware in-house as possible.
4. Open-source all hardware, software, and relevant documentation.

Program Management

Organizational Chart Overview



Organizational Chart Responsibilities

Key Terms	Responsibilities
Principal Investigator	Overall mission management
Project Director, Chief Engineer, Operations Manager	Program management
Payload Department	Optical payload subsystem
Avionics Department	Electronic power, command and data handling, attitude determination and control, and RF communications subsystems; flight software
Structures Department	Structure and thermal subsystems; bus configuration; thermal analysis
Ground Station Department	OGS and RFGS subsystems; ground software
Systems Engineering and Integration Department	System requirements formulation, verification, and validation; AIT of satellite and ground station; risk assessment and mitigation
Administrative Division	Grant composition; procurement and accounting; outreach

Preliminary Design Phase Project Advisors

Dr. Tian Zhong
Principal Investigator and Primary Faculty Advisor
Assistant Professor of Molecular Engineering,
University of Chicago

Dr. Michael Lembeck
Systems Engineering Faculty Advisor
Chief Technical Officer,
StarSense Innovations

In addition to Dr. Zhong and Dr. Lembeck, PULSE-A's preliminary design was assisted by the team's 9 student advisors: 4 graduate students and 5 undergraduate upperclassmen.

Systems Engineering Management Plan (SEMP)

The SEMP is a document describing the activities to be performed by the the PULSE-A Team in support of the PULSE-A mission throughout the project's development phases.

The SEMP provides the technical approach to organizing the mission. This includes the people, products, and processes that put the project into operation within technical, cost, and schedule constraints.

The Project Director, Chief Engineer, and Lead Systems Engineer review and update the SEMP as necessary. The SEMP will be reviewed and revised at the end of each development phase at a minimum.

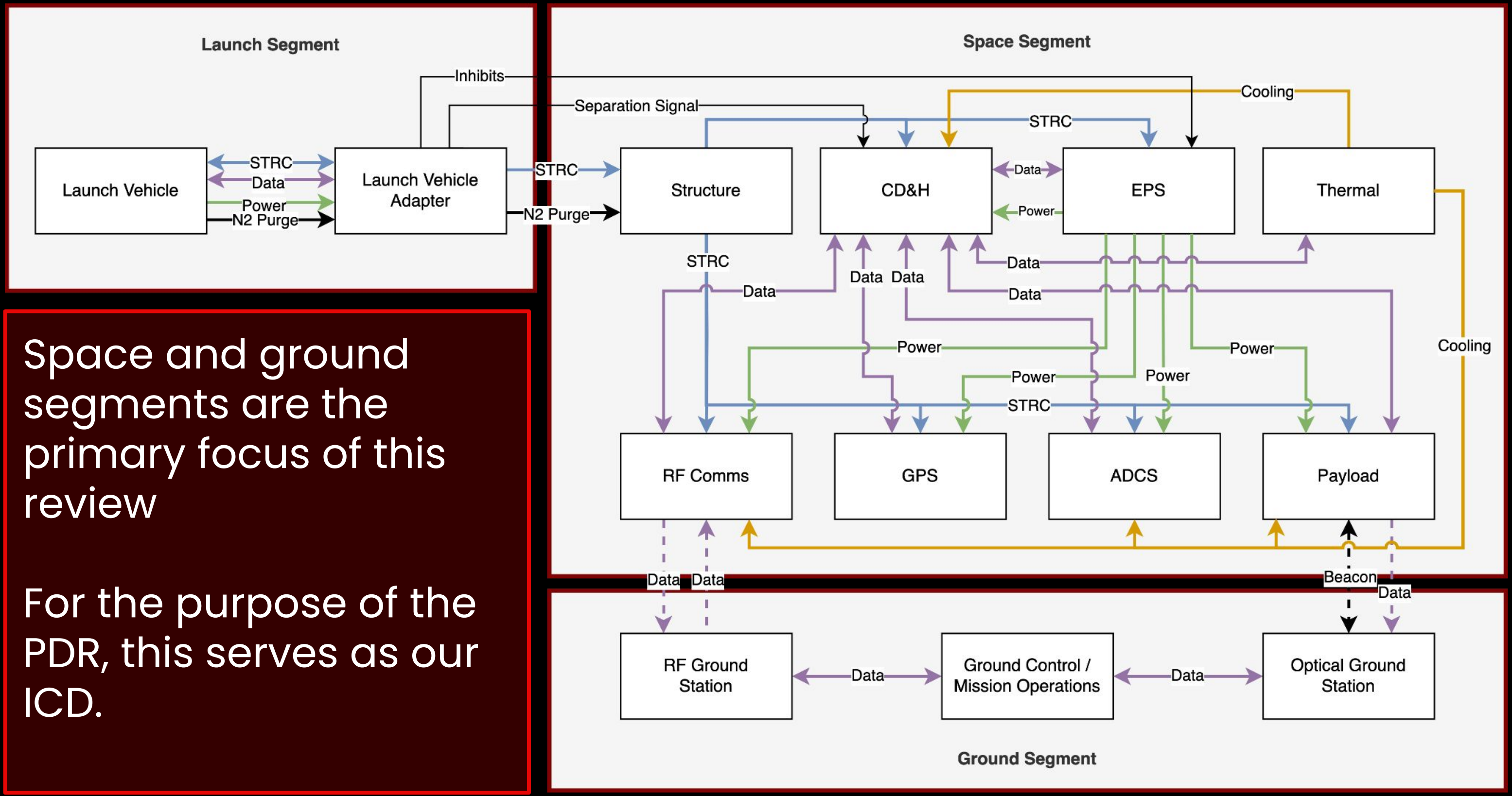
SUBJECT: PULSE-A Systems Engineering Management Plan	Effective Date 11-21-2024	Number UCSP-0001
	Page 1 of 22	Revision 1

PULSE-A
Systems Engineering Management Plan
University of Chicago Space Program

Satellite System Design:

Bus Components

System Architecture

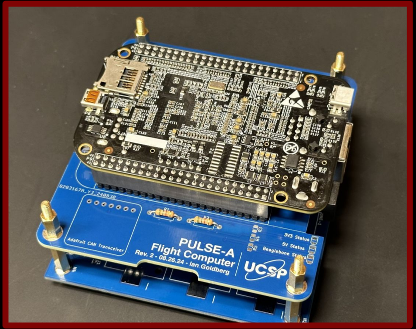


Space and ground segments are the primary focus of this review

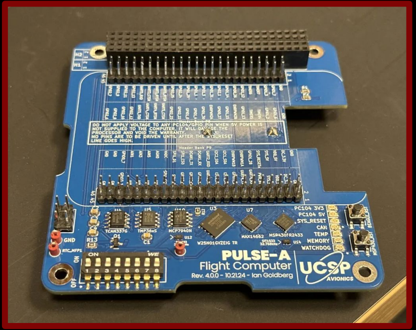
For the purpose of the PDR, this serves as our ICD.

Command & Data Handling (C&DH) (1/2)

Name	On Board Computer (OBC)	Payload Controller
Developed by	Beagleboard / PULSE-A	Same as OBC
Description	❑ Consists of Beaglebone Black mounted to a PC/104 cape with associated external hardware ❑ All hardware is rated to temp range of -40 to +80 ° C (or greater) ❑ Beaglebone Processor: 1GHz ARM® Cortex-A8 ❑ Radiation resistant External TI MSP430 Watchdog Processor with 15 KB FRAM ❑ 1 Gigabit external NAND memory IC ❑ Real-time-clock with battery backup (CR2032 lithium battery)	Same as OBC for the time being



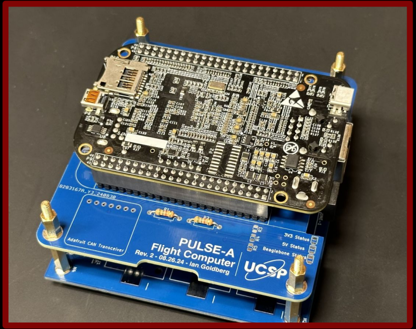
^ PULSE-A OBC



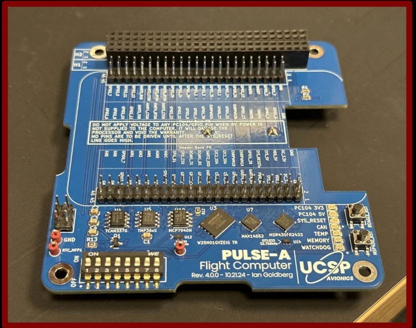
^ PULSE-A OBC

Command & Data Handling (C&DH) (2/2)

Name	On Board Computer (OBC)	Payload Controller
Description (cont.)	❏ Silicon MEMS oscillators for reduced clock drift at extreme temperatures. ❏ Onboard CAN transceiver ❏ Onboard watchdog processor ❏ Supports communication over CAN, I2C, and SPI (3k I2C pullup resistors)	Same as OBC for the time being
Readiness	Currently TRL 3, expecting TRL 6 before flight.	TRL 2 (TRL 6 before flight)



^ PULSE-A OBC

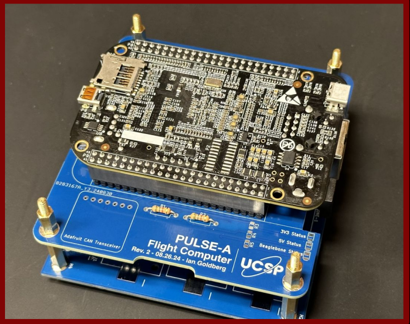


^ PULSE-A OBC

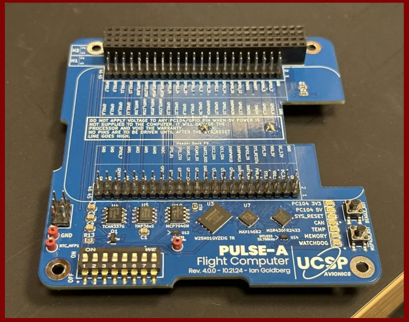
Command & Data Handling (C&DH) (2/2)

Name	On Board Computer (OBC)	Payload Controller
Description (cont.)	❏ Silicon MEMS oscillators for reduced clock drift at extreme temperatures. ❏ Onboard CAN transceiver ❏ Onboard watchdog processor ❏ Supports communication over CAN, I2C, and SPI (3k I2C pullup resistors)	Same as OBC for the time being
Readiness	Currently TRL 3, expecting TRL 6 before flight.	TRL 2 (TRL 6 before flight)

Trade study with COTS option from Pumpkin Space is given in Appendix



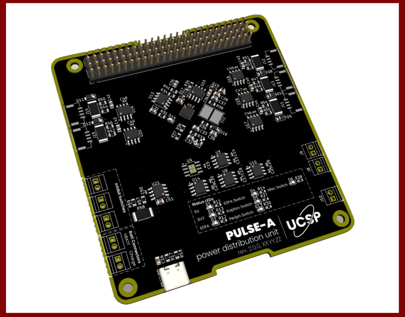
^ PULSE-A OBC



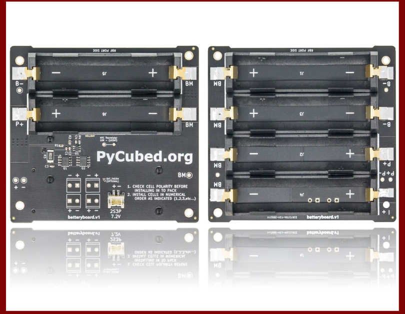
^ PULSE-A OBC

Electrical Power Subsystem (1/3)

Name	**Power Distribution Unit (PDU)**	**Battery Pack**
Developed by	PULSE-A / Hawaii SFL / Stanford	PULSE-A / Stanford
Development Description	❑ Custom-designed unit based on Univ. of Hawaii's Artemis Kit and Stanford's PyCubed kit ❑ Artemis kit developed with $500k from NASA, but has no flight heritage ❑ PyCubed has flight heritage ❑ PULSE-A's PDU has updated components, redundant systems, and systems tailored to powering the optical payload	❑ Pack developed by Stanford also used in the Artemis kit ❑ PULSE-A's pack is identical to Stanford's with small modifications to external interfaces
Maturity	Currently TRL 3, Expecting TRL 6 before flight.	TRL 4 (PyCubed is TRL 9)



^ Power Distribution Unit 2.0.0



^ PyCubed Battery Pack

Electrical Power Subsystem (1/3)

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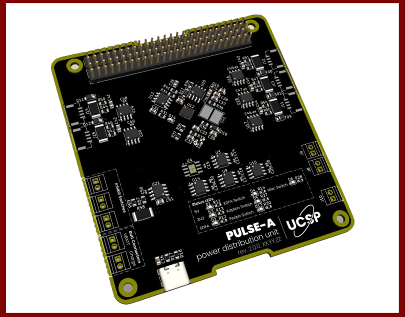
Developed by	PULSE-A / Hawaii SFL / Stanford	PULSE-A / Stanford
Development Description	❏ Custom-designed unit based on Univ. of Hawaii's Artemis Kit and Stanford's PyCubed kit ❏ Artemis kit developed with $500k from NASA, but has no flight heritage ❏ PyCubed has flight heritage the optical payload	❏ Pack developed by Stanford also used in the Artemis kit ❏ PULSE-A's pack is identical to Stanford's with es
Maturity	Currently TRL 3, Expecting TRL 6 before flight.	TRL 4 (PyCubed is TRL 9)

Although EPS is custom-designed, boards are rooted in existing designs with extensive test and flight heritage.



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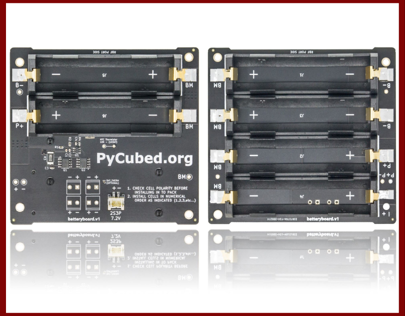

^ PyCubed Battery Pack

Electrical Power Subsystem (2/3)

Name	**Power Distribution Unit (PDU)**	**Battery Pack**
Statistics	❑ 3 LT8610 switched-mode voltage regulators supplying 5V/3V3 busses and a dedicated 5V line for high-power payload components (for max. power considerations) ❑ LTC1477 2A switches at several voltage levels for various components, including battery heaters ❑ INA219 current sensors for all solar panel boards & battery pack with adequate filtering ❑ Blocking diodes for each solar panel board ❑ 2 independent digital antenna deployment switches ❑ 2 high-side and 1 low-side mechanical inhibits ❑ RBFs to be implemented in 2.0.0 ❑ Controlled by OBC, with failsafes preventing components from driving any GPIO pins until board is booted ❑ Integrated battery charge circuit with MPPT (>90%)	❑ Li-Ion pack containing 6 Samsung 18650-35E cells ❑ 2s3p cell arrangement ❑ 7.2 V nominal Voltage ❑ 72.36 Wh total Energy (below soft cap of 80Wh set by deployment standards) ❑ ~24 A max continuous discharge current ❑ Heat provided by Kapton heaters ❑ Current revision uses terminal blocks to connect to PDU. Future revision likely to switch to locking molex connectors.

Electrical Power Subsystem (2/3)

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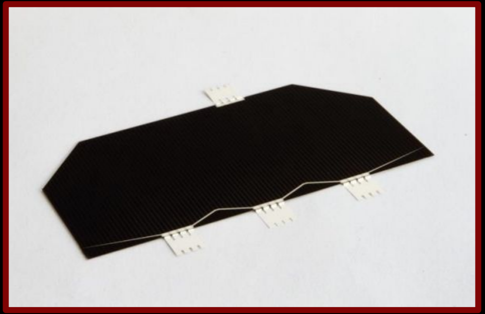
Electrical Power Subsystem (3/3)

Solar Cells

Name	**AzurSpace 3G30A**
Description	GaInP/GaAs/Ge triple junction
Efficiency at Beginning of Life (BOL)	29.6%
Efficiency at End of Life (EOL)	28%
Pros	Full 3U set gifted by UIUC
Cons	Mounted to existing panels, will be difficult to remount if necessary



^ AzurSpace 3G30A

Electrical Power Subsystem (3/3)

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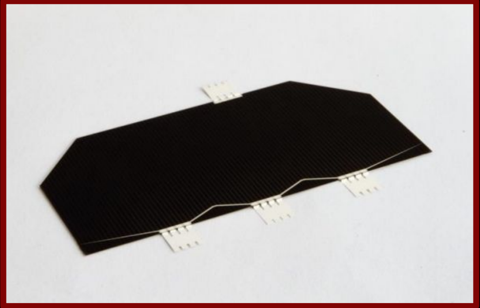


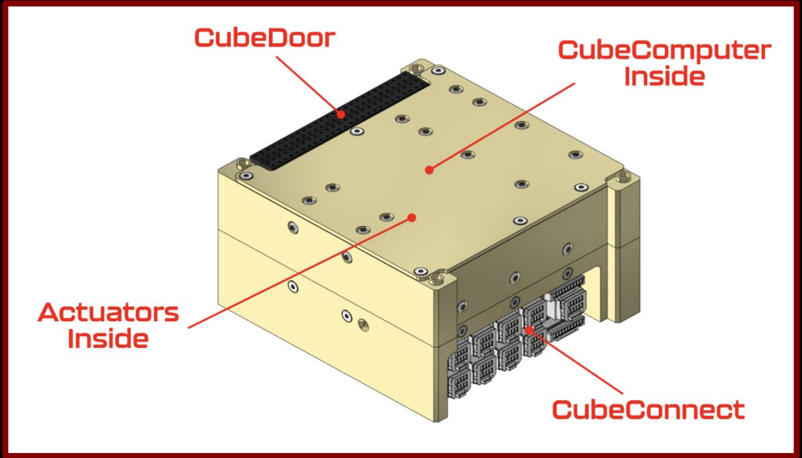
^ AzurSpace 3G30A

Need to resolve if current solar panels are usable, and if not, if it is feasible to remount cells to new board for deployment (potential for high risk)

ADCS Subsystem
Overview

PULSE-A has selected **CubeSpace Satellite Systems** to provide their Integrated Gen 2 Attitude Determination and Control System for the mission, **capable of <1° 3σ absolute pointing error.**

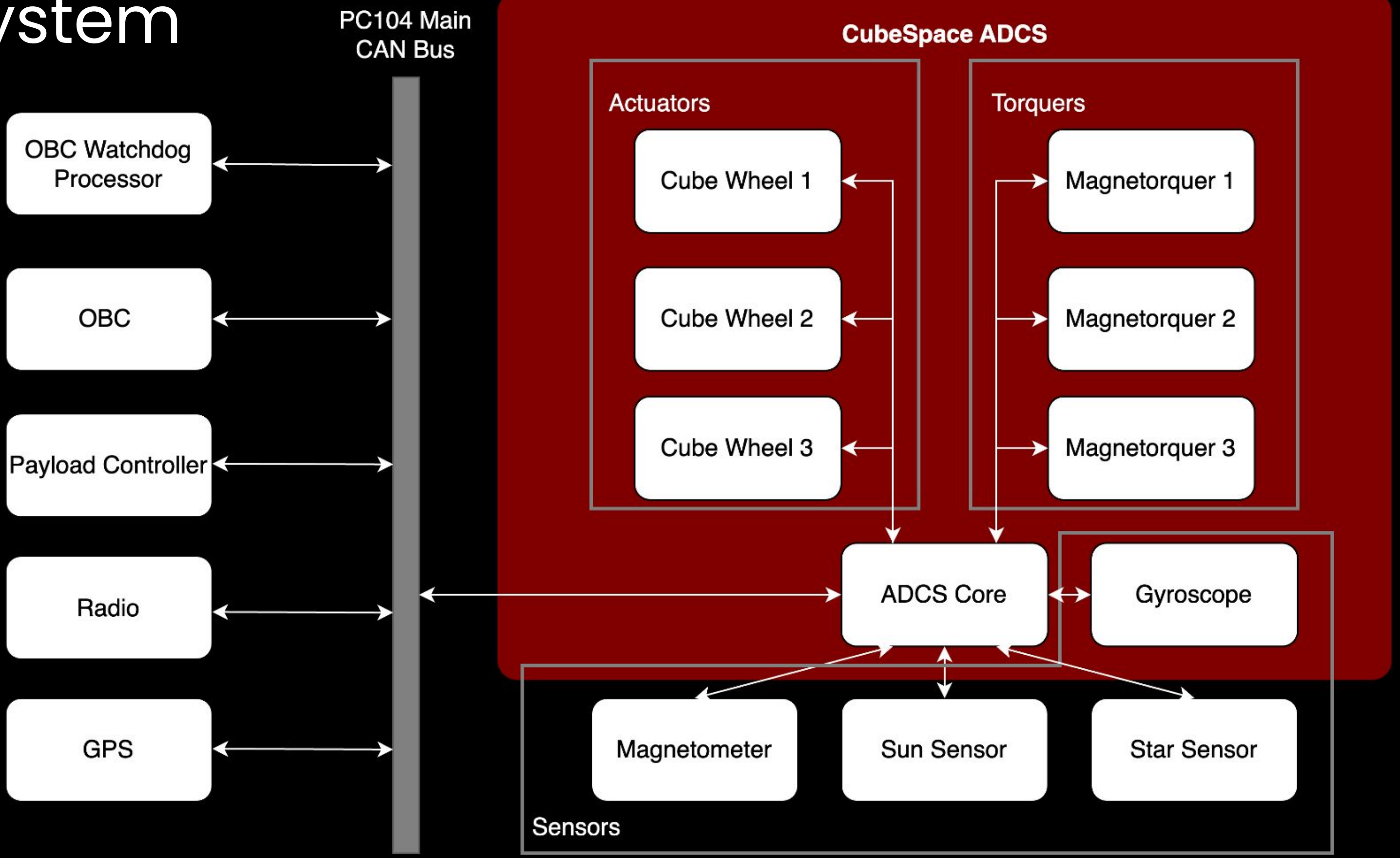
The Gen 2 kit comes preloaded with ground target and sun tracking algorithms, allowing for easier development of Pointing, Acquisition, and Tracking algorithms.



^ CubeSpace 3U Integrated Core System

ADCS Subsystem
Architecture

ADCS measures and controls satellite orientation to allow for accurate alignment during PAT, RF transmit, and solar panel charging

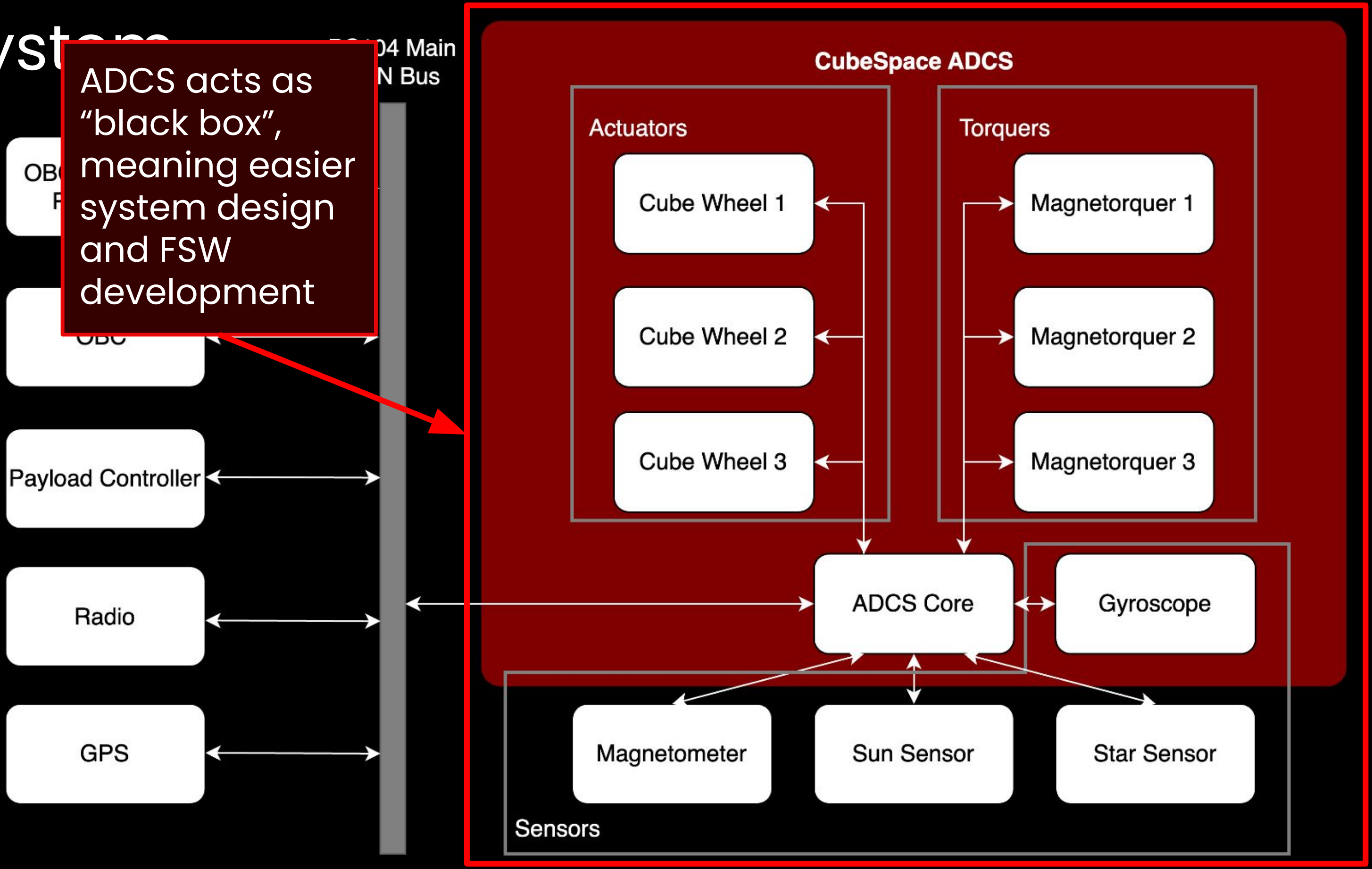


ADCS Subsystem

Architecture

ADCS measures and controls satellite orientation to allow for accurate alignment during PAT, RF transmit, and solar panel charging

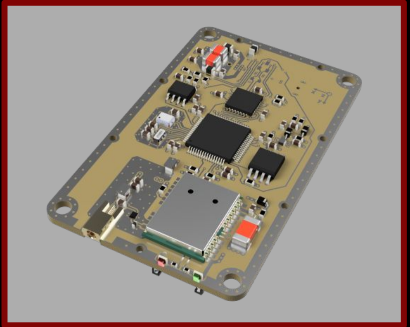
ADCS acts as “black box”, meaning easier system design and FSW development

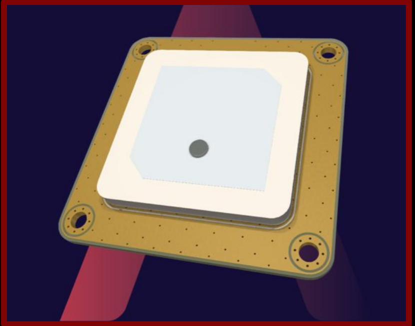


ADCS Subsystem
GPS and Patch Antenna

GPS	SpaceManic Celeste
Description	Provides orbital coordinates, rates, RTC sync
Accuracy	Position: 2.0 m Velocity: 0.1 m/s Time: 5 ns
Constellations	GPS/GLONASS/BeiDou/Galileo
Interfaces	I2C, RS485, UART, CAN
Average Power	100 mW
Mass	25 g
Dimensions	67 x 42 x 8 mm

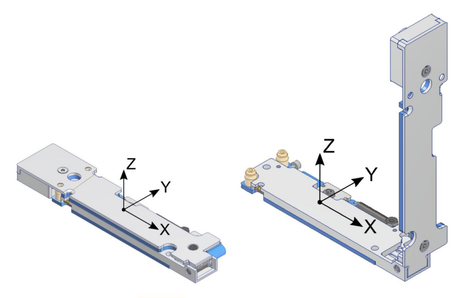
Antenna	Zenith Antenna
Description	Receiving antenna
Peak Gain	4 dBi
Impedance	50 ohm
Efficiency	70%
LNA Gain	15 dB
Power	50 mW
Mass	20g
Dimensions	50 x 50 x 8 mm


^ Celeste


^ Zenith Active Antenna

ADCS Subsystem
Sensors (1/2)

GPS	CubeSpace CubeMag Deployable Magnetometer
Description	Measures changes in the Earth's magnetic field
Power (steady/max)	50 mW, 230 mW
Measurement Range	-8 to 8 Gauss
Noise per Channel 3σ	50 nT
Linearity (full scale)	0.6%
Mass	16 g
Data Protocol	CAN/UART/RS-485



^ CubeMag Deployable

ADCS Subsystem
Sensors (2/2)

Item	CubeSpace CubeSense Sun	CubeSpace CubeStar
Description	CMOS-based fine sun sensor	High accuracy star tracker
Accuracy (dependent on slew)	0.2° 2σ (roll and elevation)	0.02° 3σ (cross-axis) 0.06° 3σ (roll)
Max Slew Rate	70°/s	0.3°/s
Mass	15 g	47 g
FoV (horiz./diag.)	166° / 176°	58x47° / 59.4°
Dimensions	35x22x24 mm	35x49x24 mm
Power (steady/peak)	100 mW / 174 mW	165 mw / 271 mW

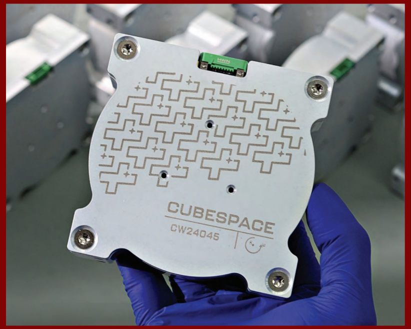
^ CubeSense Sun

^ CubeStar

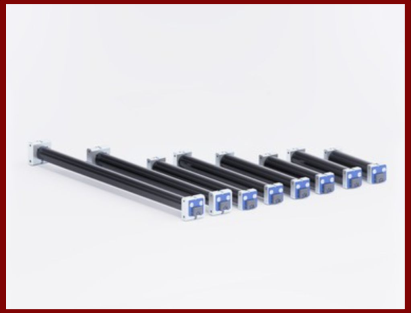
ADCS Subsystem
Actuators

Reaction Wheel	CubeSpace CW0017
Description	CubeWheel
Quantity	3
Max Speed	10000 rpm
Momentum @ 6000 RPM	1.77 mNms
Saturation Torque	0.23 mNm
Mass	60 g
Dimensions	28 x 26 x 28 mm
Power (steady/peak)	180 mW / 850 mW

Magnetorquer	CubeSpace CR0003
Description	CubeTorquer
Quantity	3
Minimum Magnetic Moment	0.3 Am^2
Mass	23 g
Dimensions	10.5 x 10.5 x 59 mm
Average Power	123 mW



^ CubeWheel


^ CubeTorquers

ADCS Required Capabilities

ID	Requirement	Notes	Parent	Verification	Compliance Timeline
ADCS-08	The ADCS shall provide three-axis pointing control with an accuracy of ≤ 1.2° 3σ.	The mirror lens in the payload system can provide 1.5° adjustment, so the ADCS should stay within 80% of that value	ADCS-01, PAY-15	Analysis	Compliant

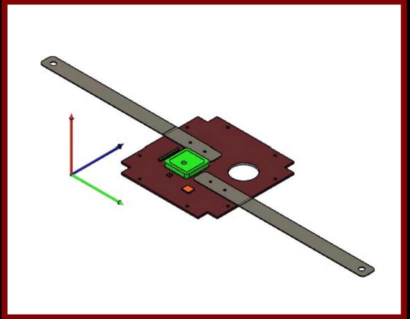
CubeSpace ADCS meets the necessary 1.2 deg body pointing error requirement with a 20% margin.

Communications Subsystem

Radio	GomSpace AX100-U
Description	Half-Duplex UHF Transceiver
Directionality	Half-Duplex
Data Protocol	CAN, I2C, or UART
Data Rate	0.1 to 38.4 kbps
Encoding	GMSK
Dimensions	65 x 40 x 6.5 mm
Rx Power (steady/peak)	180 mW / 400mW
Tx Power (steady/peak)	2.64 W / 2.81 W

Antennas	Custom Design
Description	Dual Monopole Antenna System
Dimensions	361 x 100 x 5.1 mm
Mass	74.4 g

^ GomSpace AX100-U



^ Dual Antenna System

Next Steps for Bus Components

- ❏ Detailed design of EPS and C&DH hardware, including ongoing prototype development and test.
- ❏ Payload Controller detailed design will begin once OBC hardware reaches adequate maturity level. Relevant payload control hardware is being developed by the Payload department.
- ❏ The PULSE-A Avionics department will host a **Detailed Electronics Design Review** (DEDR) in March. After concluding initial prototype development and testing, board schematics and layouts will be given to a panel to review. This review is hosted in part by the team’s sponsor, **Cadence Design Systems.**

Satellite System Design:

Optical Payload

Payload Driving Requirements (1/2)

ID	Requirement	Notes	Parent	Verification
PAY-03	The Payload shall modulate the transmission laser to encode data via circular polarization state.		PAY-01	Demonstration
PAY-06	The Payload shall be capable of distinguishing between signal from the OGS beacon and detector noise.	We have designed around an SNR of 6 dB.	PAY-05	Analysis
PAY-10	The Payload shall modulate the transmission laser at a frequency of 1 MHz ≤ Frequency ≤ 10 MHz.		PAY-03, MSN-02	Demonstration
PAY-11	The Payload shall output a transmission beam at a power of 250 mW ± 45 mW.	These values are derived in the Optical Tx Link Budget.	PAY-01	Analysis

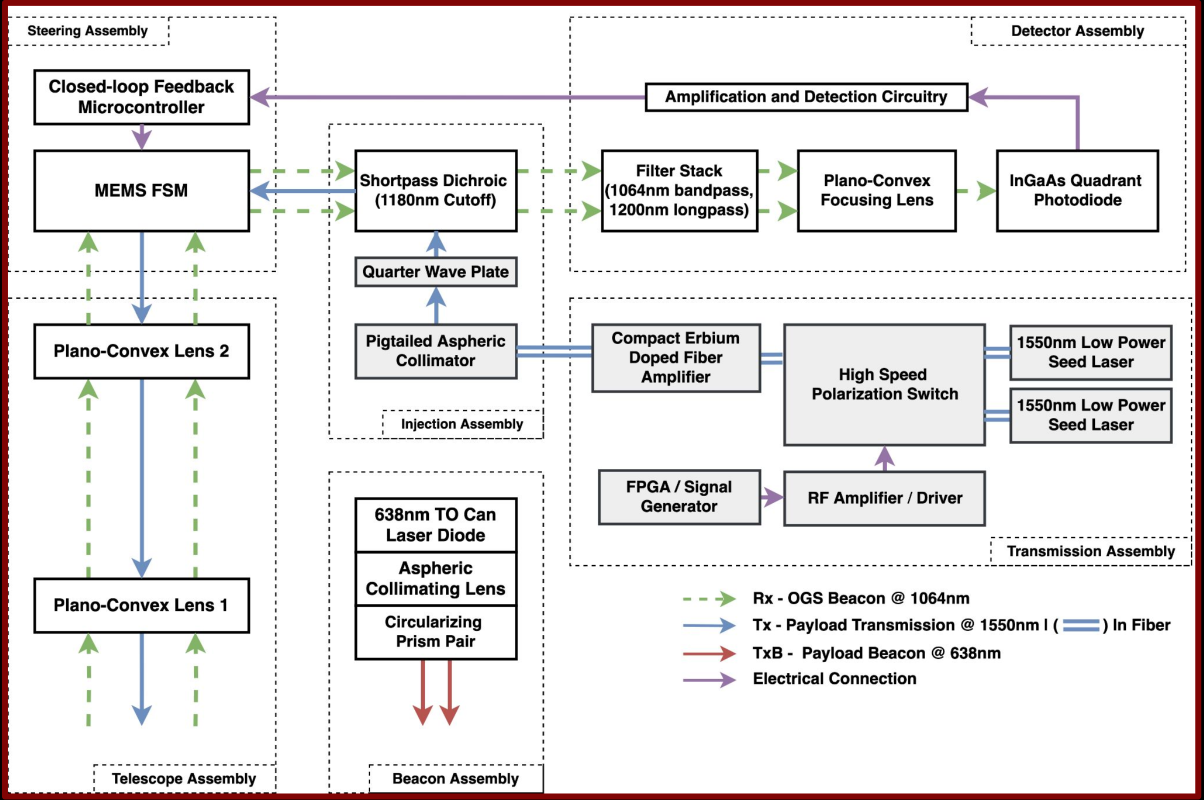
Legend: Compliant Compliant by CDR Compliant by TRR Compliant By FRR

Payload Driving Requirements (2/2)

ID	Requirement	Notes	Parent	Verification
PAY-12	The Payload shall output a laser beacon at a power of 200 mW ± TBD mW.	This value is derived in the Payload Beacon Link Budget; TBD tolerance depends on variance.	PAY-08	Analysis
PAY-15	The Payload shall provide fine pointing capabilities of 0.0429° ≤ Angle ≤ 1.5°.	LB derived from the Optical Tx Link Budget; TB from MEMS FSM.	PAY-07	Analysis
PAY-15, PAY-16	The Payload shall be capable of operating the optical transmission and laser beacon continuously for 150 s ≤ time ≤ 300 s.	150 s is the length of the typical pass, while there should never be a pass longer than 300 s.	PAY-01, PAY-08	Test

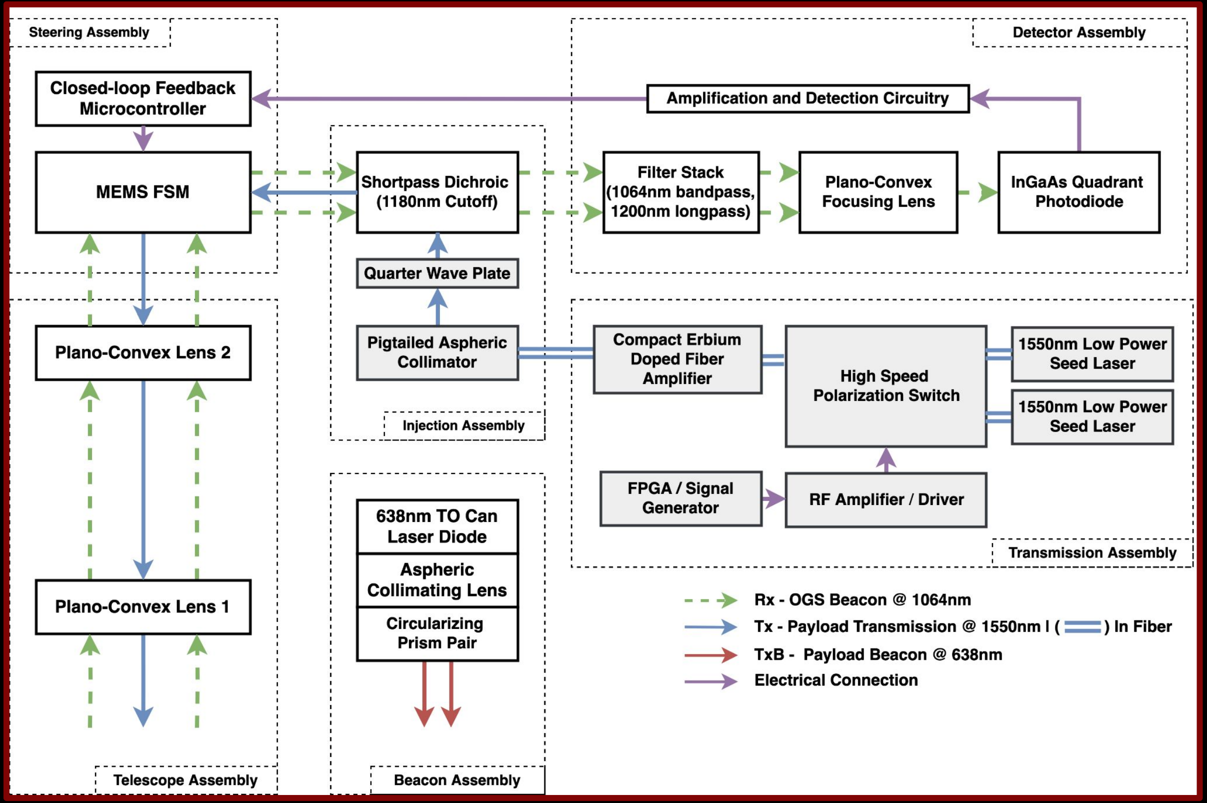
Legend: Compliant | Compliant by CDR | Compliant by TRR | Compliant By FRR

Payload Configuration



Ground beacon laser and transmission laser follow the same optical path, which enables fine pointing through a detection setup and FSM.

Payload Configuration



Data is encoded by: A linearly polarized seed laser is modulated using a polarization switch, amplified through an EDFA, and converted to circular polarization with a quarter waveplate.

Laser Wavelength and Beacon Justification

All wavelengths used have a high atmospheric transmittance (> 0.8).

- As a common telecom wavelength, components at 1550 nm are abundant.
- Thermal cameras required for detection of 1550 nm are outside of our budget
- 638 nm is detectable by a CMOS camera

Atmospheric Transmission

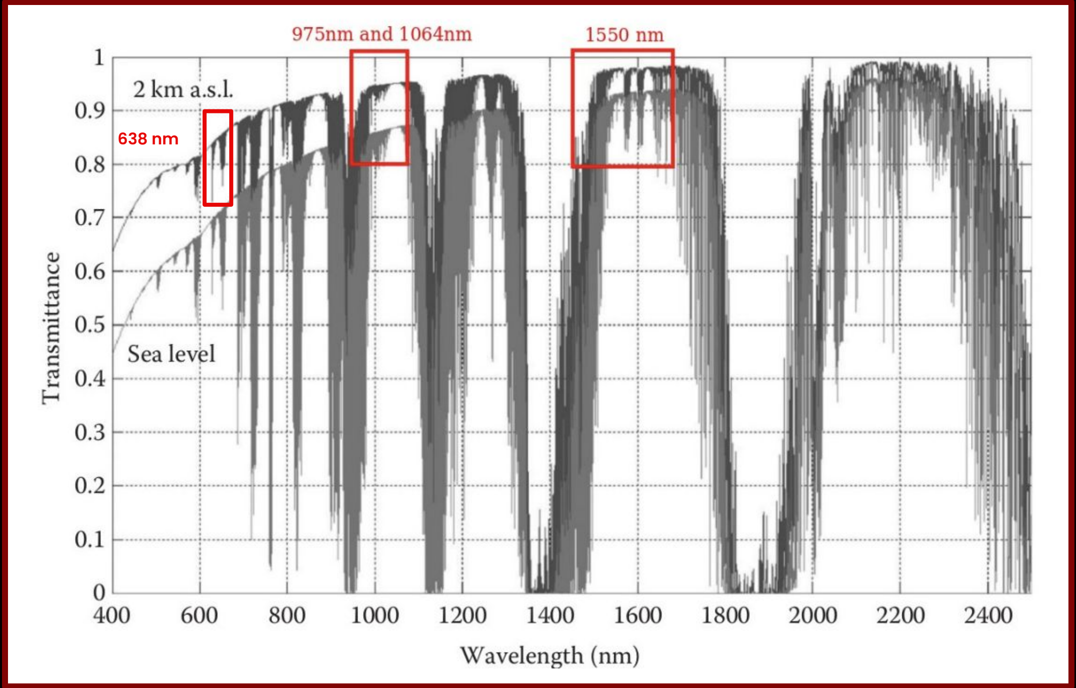


Figure adapted from Kingsbury, Optical Communication for Small Satellites, MIT 2015.

Beacon Subassembly: 638 nm Beacon Laser

Allows the OGS to locate and track the satellite using a CMOS detector. Followed by a collimating lens and a PS875-A-N-KZFS8 Mounted Prism Pair (adjusting beam profile and divergence).

Company	**THORLABS**	**Laser Tree (Ushio)**
Model	L638P200	HL63022DG
Wavelength (nm)	638	638
Power Output (mW)	200	200
Size (mm)	Ø5.6	Ø5.6
Operating Temp. (°C)	-10 to +40	-40 to +60
Price	$153.73	$28.00
Parallel Beam Divergence (°)	min=5; typical=8; max=11	min=5; typical=8; max=13
Perpendicular Beam Divergence (°)	min=10; typical=14; max=18	min=10; typical=14; max=18

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Perpendicular Beam Divergence (°)	min=10; typical=14; max=18	min=10; typical=14; max=18

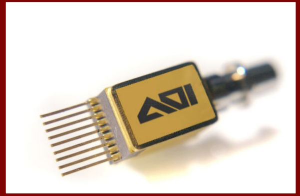
Better Max Parallel Beam Divergence

Better Operating Temperature and Price

Transmission Subassembly: 1550 nm Seed Laser

Transmission beam. Outputs a low power, 1550 nm linearly polarized laser beam to the polarization switch.

Company	**THORLABS**	**Laser Diode Source**	**AOI**
Model	L1550P5DFB	LD4B-1550-DFB-2.5G-20-C OAXB-2-SM1-FA-CW-1.0	DFB-BT-2.5-LC-476 TOSA
Wavelength (nm)	1550	1550	1550
Power Output (mW)	5	22	6mW
Size (mm)	Ø5.6	Ø2.1	1.7x0.5cm
Operating Temp. (°C)	-20 to +80	-40 to +85	-5 to +75
Price	$92.36	$565.00	~$1000
Rise/Fall Time (ns)	~0.1 ns	< 200 ps	< 200 PS
Polarized?	Needs polarizer	Can get PM fiber option	Needs polarizer



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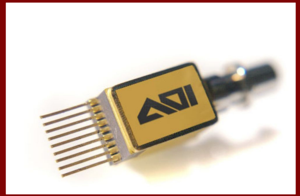
Transmission beam. Outputs a low power, 1550 nm linearly polarized laser beam to the polarization switch.

Company	**THORLABS**	**Laser Diode Source**	**AOI**
Model	L1550P5DFB	LD4B-1550-DFB-2.5G-20-COAXB-2-SM1-FA-CW-1.0	DFB-BT-2.5-LC-476 TOSA
Wavelength (nm)	1550	1550	1550
Pow		22	6mW
Size		Ø2.1	1.7x0.5cm
Ope (°C)	+80	-40 to +85	-5 to +75
Pric		$565.00	~$1000
Rise		< 200 ps	< 200 PS
Polarized?	Needs polarizer	Can get PM fiber option	Needs polarizer

Likely best option due to possibility of PM coupling and reasonable power output, though custom PM may increase price.



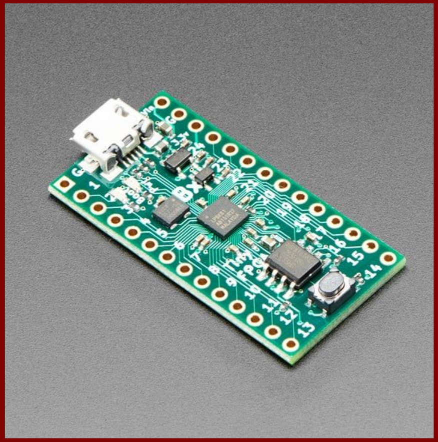
Transmission Subassembly: FPGA Signal Generator

Signal Generator

The **TinyFPGA BX** development board containing the **ICE40 FPGA** from Adafruit reads data from FLASH memory chip and sends it at the required frequency to the modulator.

Specs.

Mass (grams)	4.8
Size (mm)	18 x 36
Voltage (V)	3.3
Power (mW)	4
On-Board Clock (MHz)	16
Measured Rise time (ns)	15



Transmission Subassembly: FPGA Signal Generator

Signal Generator

The **TinyFPGA BX** development board containing the **ICE40 FPGA** from Adafruit reads data from FLASH memory chip and sends it at the required frequency to the modulator.

Chosen for its small form factor, power usage, cost, and reprogrammability.

Rise time and qualification to be further investigated.

Specs.

Mass (grams)	4.8
Size (mm)	18 x 36
Voltage (V)	3.3
Power (mW)	4
On-Board Clock (MHz)	16
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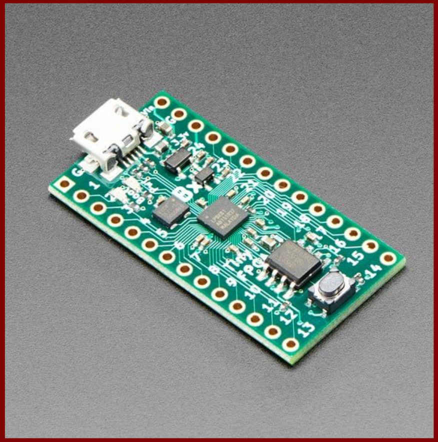

Allows easily for 10 MHz mod. rate, which is faster than target. Rise time causes only ~15% corruption of a bit when switching.

Transmission Subassembly: Polarization Switch

Encodes data transmitted from FPGA by modulating the phase of 1550 nm laser signal, by switching between two orthogonal linear polarizations.

Company	**iXblue**	**Agiltron**
Model	PSW-LN-0.1 Polarization Switch	Nano Speed Polarization Switch
Wavelength (nm)	1530 -1580	1260 - 1650
Switching Voltage	8-9V @150 MHz	5V
Insertion Loss (dB)	3.5 – 4.0	1.0 (Max)
Operating frequency (MHz)	150-200	1 or 10
Dimensions* (mm)	85 x 12 x 9.65	57.5 x 9.7 x 7.35

iXblue

Agiltron

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Dimensions* (mm)	85 x 12 x 9.65	57.5 x 9.7 x 7.35

Higher operating frequencies

Higher loss, dimensions* do not fit CAD
*Compact version **might** be available soon

iXblue

Agiltron

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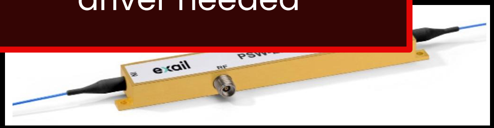
Wavelength (nm)	1530 -1580	1260 - 1650
Switching Voltage	8	5V
Insertion Loss (dB)	3	1.0 (Max)
Operating frequency (MHz)	1	1 or 10
Dimen	8	57.5 x 9.7 x 7.35

Selected because:
- ❑ Reasonable operating frequency, compact model, low loss
- ❑ Custom, low power driver needed



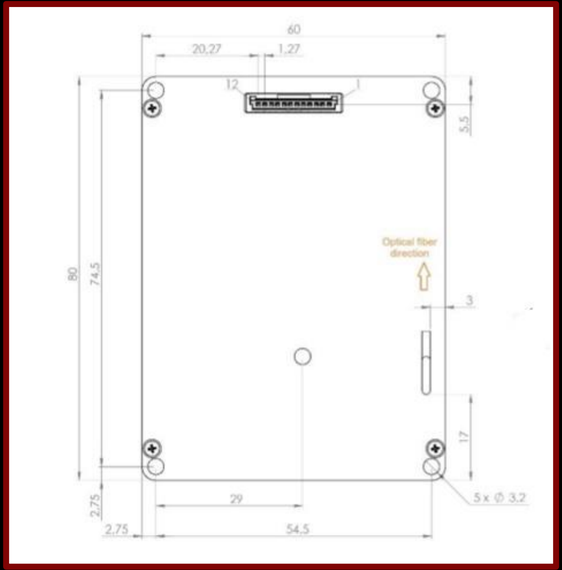
iXblue

Agiltron

Transmission Subassembly: EDFA

Amplifies power generated by seed laser to ~250 mW such that it can be detected by the APDs at the OGS.

Company	**BKtel Photonics**
Model	Compact Booster Amplifier Module
Wavelength (nm)	1540-1565
Power Output (dBm)	24
Polarization Dependent Gain (PDG)	3-5%
Noise figure (dBm)	8.5
Polarization	Linear
Return Loss (dBm)	40
Dimensions (mm)	60 x 80 x 12



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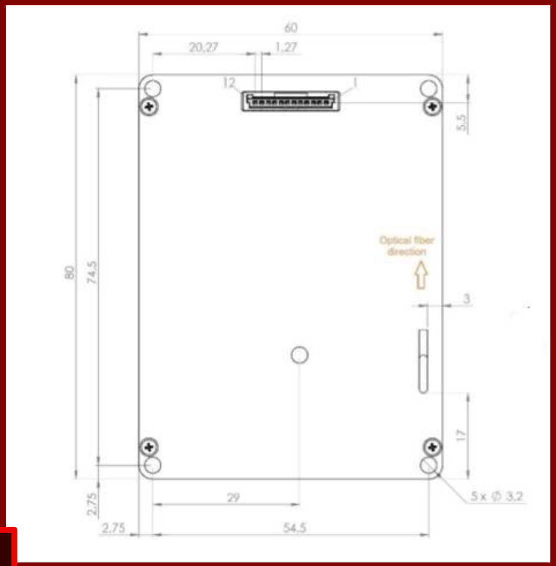
Noise figure (dBm)	8.5
Polarization	Linear
Return Loss (dBm)	40
Dimensions (mm)	60 x 80 x 12

High **power** output, minor polarization state distortion.

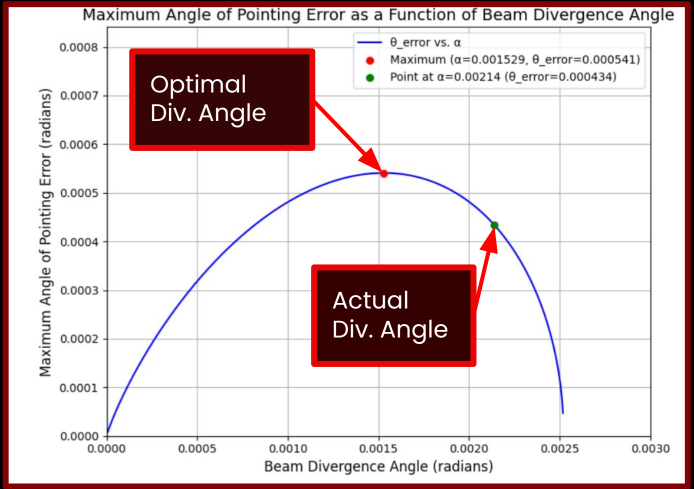


Transmission Subassembly: Collimator

Pigtailed PM Atmospheric Collimator:

Enables a fiber-to-free space transition along the transmission path. Collimates the transmission laser at the optimum divergence angle before directing the beam through a shortpass dichroic to the MEMS mirror.

Company	**Thorlabs**
Model/Item#	CFP5-1550A
Mass (g)	5.9
Housing outer diameter (mm)	5.7
Aperture size (mm)	3.0
Wavelength (nm)	1550
Divergence (mrad)	2.14 +0.17/-0.00

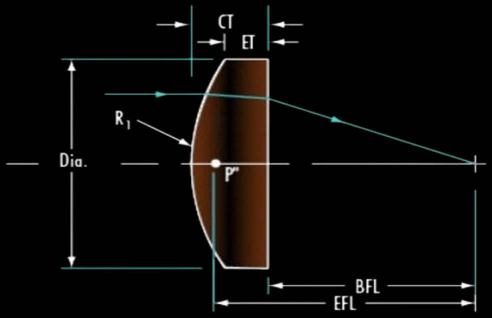


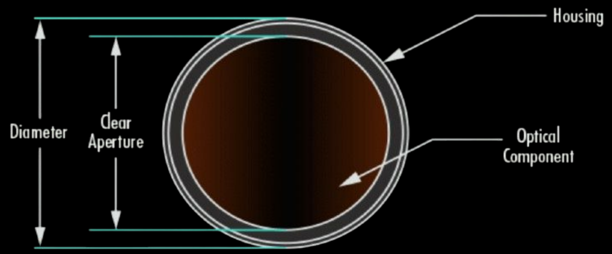
Plot derived from link budget. Closest divergence angle of all polarization maintaining collimator options, with a difference of 0.611 mrad.

Payload Optical Path Components: Lenses

There are three plano-convex lenses that constitute the telescope subassembly of the optical payload, with the third lens utilized as a focusing component of the detector subassembly for quadrant photodiode beam detection. Wavelength range for all three lenses is 350-2200 nm.

Lens	Diameter (mm)	E-Focal Length at 587nm (mm)	Numerical Aperture	Center Thickness (mm)	Substrate
EO #45-241	30.00	60.00	0.25	6.00 +/- 0.10	N-BK7
EO #28-952	8.00	8.00	0.50	2.90 +/- 0.05	N-SF11
EO #49-876	9.00	15.00	0.30	3.00 +/- 0.05	N-BK7





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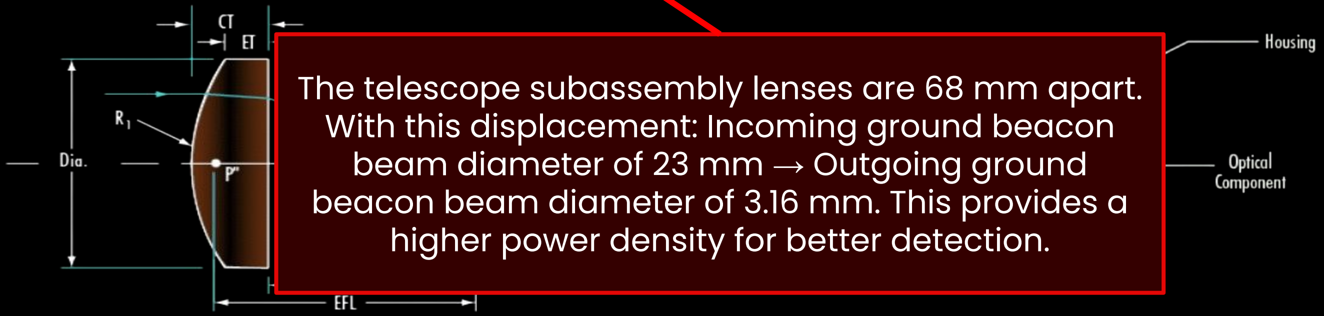

The telescope subassembly lenses are 68 mm apart. With this displacement: Incoming ground beacon beam diameter of 23 mm → Outgoing ground beacon beam diameter of 3.16 mm. This provides a higher power density for better detection.

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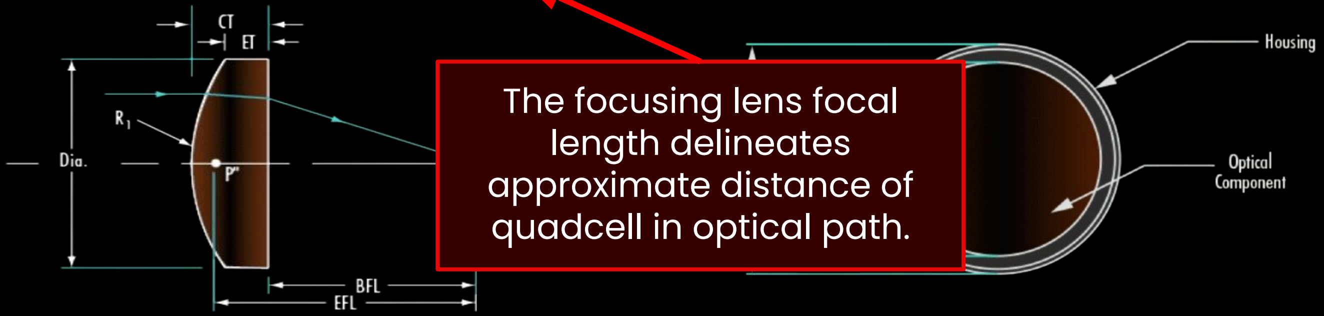


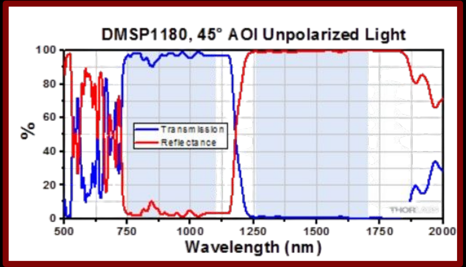
The focusing lens focal length delineates approximate distance of quadcell in optical path.

Payload Optical Path Components: Filters

Short Pass Dichroic Mirror

Dichroic mirror, DMSP1180T, serves to spectrally separate light by transmitting and reflecting light as a function of wavelength. It is highly transmissive below the cutoff wavelength and above the cut-on wavelength, while highly reflective between the cutoff and cut-on.

Company	Thorlabs
Cutoff Wavelength (nm)	1180
Transmission Band (nm)	750 - 1100
Reflection Band (nm)	1260 - 1700
Size (in)	Ø 1/2"
Clear Aperture (mm)	Ø 11.4
TWE	λ/4 (at 633 nm)
Incident Angle	45°



Payload Optical Path Components: Filters

Short Pass Dichroic Mirror

Dichroic mirror, DMSP1180T, serves to ... tr... ht a... t is hi... cu... the cu... nly re... and cut-on.

Directs incoming ground beam and outgoing transmission beam onto the same optical path. Passes the ground beam and reflects the transmission beam.

Company	Thorlabs
Cutoff Wavelength (nm)	1180
Transmission Band (nm)	750 - 1100
Reflection Band (nm)	1260 - 1700
Size (in)	Ø 1/2"
Clear Aperture (mm)	Ø 11.4
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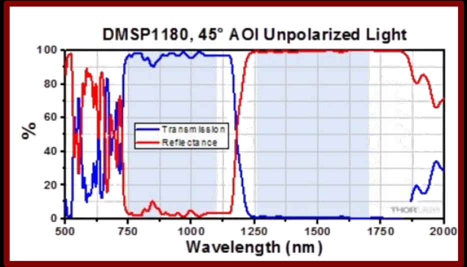
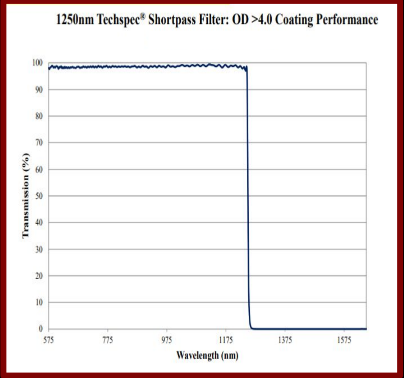


Payload Optical Path Components: Filters

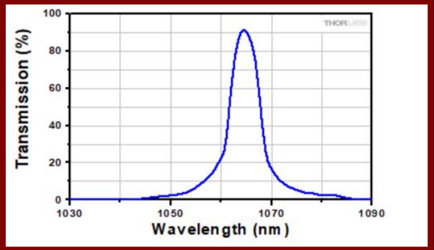
Bandpass Filter: FLH1064-8

Company	**Thorlabs**
Center Wavelength (nm)	1064 (T > 90%)
FWHM Bandwidth (nm)	8
Blocking Regions (nm)	200-1039, 1089-1200
TWE	λ/4
Outer Diameter (mm)	25.0
Surface Quality (SD)	60-40

Shortpass Transmission



Bandpass Transmission



Shortpass Filter: #89-671

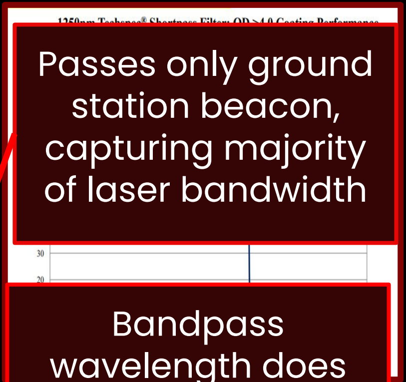
Company	**Edmund Optics**
Cutoff Wavelength (nm)	1250
Rejection Wavelength (nm)	1290-1650
Transmission Wavelength (>91%)	575-1235
Angle of Incidence	0°
Diameter (mm)	12.5 +/- 0.2
Surface Quality (SD)	60-40 SD==

Payload Optical Path Components: Filters

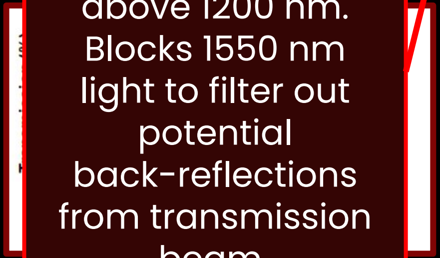
Bandpass Filter: FLH1064-8

Company	Thorlabs
Center Wavelength (nm)	1064 (T > 90%)
FWHM Bandwidth (nm)	8
Blocking Regions (nm)	200-1039, 1089-1200
TWE	λ/4
Outer Diameter (mm)	25.0
Surface Quality (SD)	60-40

Shortpass Transmission



Passes only ground station beacon, capturing majority of laser bandwidth



Bandpass wavelength does not extend to light above 1200 nm. Blocks 1550 nm light to filter out potential back-reflections from transmission beam.

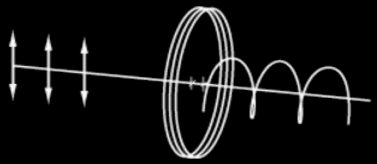
Shortpass Filter: #89-671

Company	Edmund Optics
Cutoff Wavelength (nm)	1250
Rejection Wavelength (nm)	1290-1650
Transmission Wavelength (>91%)	575-1235
Angle of Incidence	0°
Diameter (mm)	12.5 +/- 0.2
Surface Quality (SD)	60-40 SD==

Payload Optical Path Components: Filters

Zero-Order Quarter Wave Plate

The quarter wave plate, model WPQSM05-1550 designed for 1550nm, converts the polarization state from linear to circular, when incoming light is within the plate's operating range. This plate is built through the combination of two multi-order crystalline quartz wave plates.



Company	**Thorlabs**
Optic Diameter (mm)	12.7 +/- 0.10
Optic Thickness (mm)	2.00
Reflectance	< 0.25%
Retardance	λ/4 (accuracy λ/300)
Beam Deviation (arcsec)	< 10
Material	Crystalline Quartz
TWE	λ/8 (at 633 nm)

Payload Optical Path Components: Filters

Zero-Order Quarter Wave Plate

Tl el W 15 ation st when ir late's operating range. This plate is built through the combination of two m wave p

Describes the complete shifting of beam path (dissimilar to divergence), selected QWP does not cause significant deviation

Crystalline Quartz surpasses polymer QWP with higher damage threshold and retardation stability

Company	Thorlabs
Optic Diameter (mm)	12.7 +/- 0.10
Optic Thickness (mm)	2.00
Reflectance	< 0.25%
Retardance	λ/4 (accuracy λ/300)
Beam Deviation (arcsec)	< 10
Material	Crystalline Quartz
TWE	λ/8 (at 633 nm)

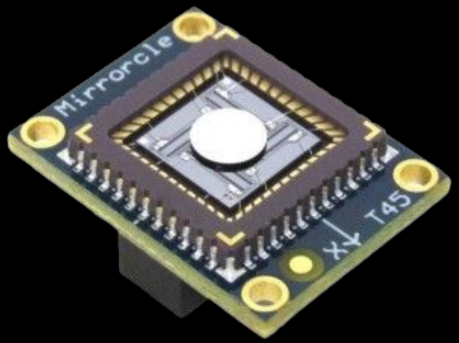
Steering Subassembly: Fast Steering Mirror

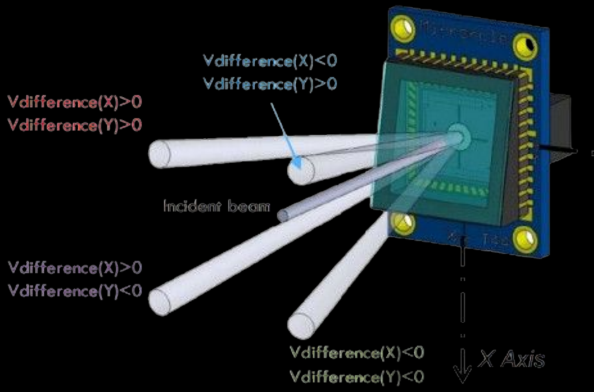
MEMS Fast-Steering-Mirror:

Dual-axis Quasistatic Microelectrical mechanical systems-Mirror, controls fine pointing. Used to point incoming ground station beacon laser at the center of quadrant photodiode.

Mirrorcle; Bonded Mirror A5L3.3(C2):

Mass (grams)	~250
Diameter (mm)	6.4
Max. Mech. Angle (coupled axes)	5.6°
Max. Mech. Angle (x axis)	+/- 4.4°
Max. Mech. Angle (y axis)	+/- 4.4°
Surface Roughness (nm rms)	<10
Mirror Coating	Gold (AU)
Optical Window (coating)(nm)	1040 - 1600
Mirror Reflectance (350-2000nm)	95%





Steering Subassembly: Fast Steering Mirror

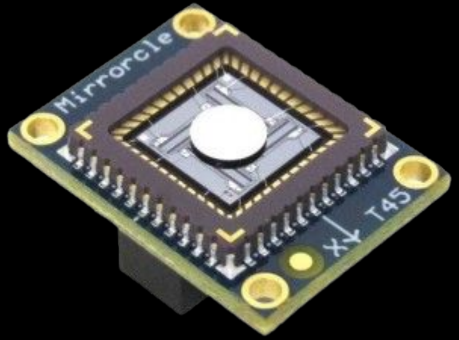
MEMS Fast-Steering-Mirror:

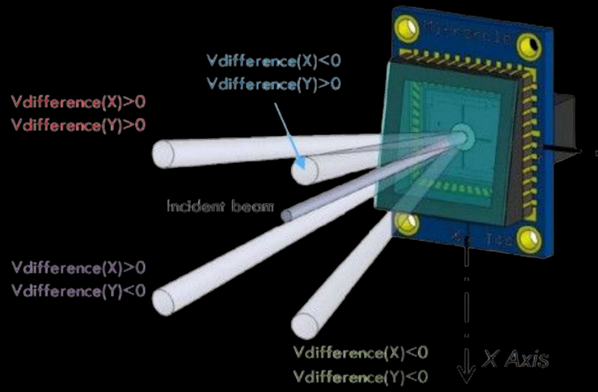
Dual-axis Quasistatic Microelectrical mechanical systems-Mirror, controls fine pointing. Used to point incoming ground station be er of

Allows enough angular tilt to maintain accurate pointing for laser detection at the OGS, as confirmed through simulation

Mirrorcle; Bonded Mirror A5L3.3(C2):

Mass (grams)	~250
Diameter (mm)	6.4
Max. Mech. Angle (coupled axes)	5.6°
Max. Mech. Angle (x axis)	+/- 4.4°
Max. Mech. Angle (y axis)	+/- 4.4°
Surface Roughness (nm rms)	<10
Mirror Coating	Gold (AU)
Optical Window (coating)(nm)	1040 - 1600
Mirror Reflectance (350-2000nm)	95%





Alignment Control: Fast Steering Mirror Driver

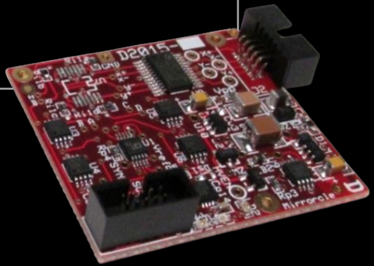
MEMS FSM Control System

Digital Input MEMS Driver

DR-10-056-00, from Mirrorcle Technologies, converts low-voltage user inputs into two differential pairs of high voltage analog outputs. With SPI digital inputs for X and Y axis drive, it is typically X200 Range, allowing use of the full voltage range since to set any V-bias and V-difference since all 4 output channels are set via software.

DR-10-056-00

Power Consumption (mW)	75-85
Signal Bandwidth (kHz)	< 25
Output Voltage (V)	0-200
Supply Voltage (V DC)	5



Alignment Control: Fast Steering Mirror Driver

MEMS FSM Control System

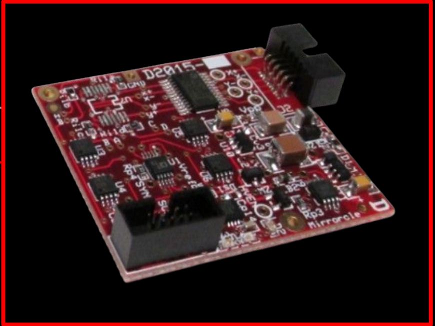
Digital Input MEMS Driver

DR-10-056-00, from Mirrorcle Technologies, converts low- … two … volta… SPI c… axis … Rang… volta… any … since all 4 output channels are set via software.

Manageable power consumption and relatively small footprint (~35x35x9mm) are not prohibitive, support Mirrorcle as meeting the necessary requirements.

DR-10-056-00

Power Consumption (mW)	75-85
Signal Bandwidth (kHz)	< 25
Output Voltage (V)	0-200
Supply Voltage (V DC)	5



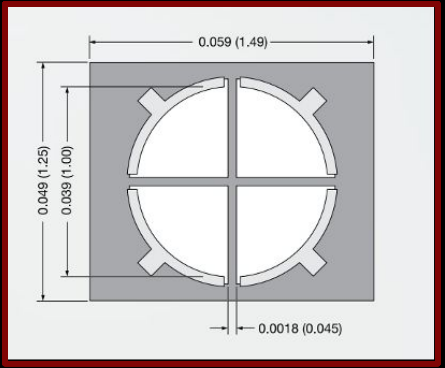
Detection Assembly: Quadcell

Quadrant Photodiode (Quadcell):

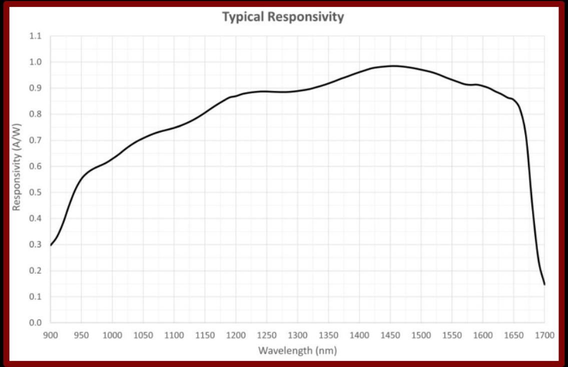
Four pixel InGaAs detector used for ground station beacon laser detection to achieve fine pointing. The center of the four pixels represents correct optical path alignment.

OSI; AFG91 FCI-INGAAS-Q1000:

Active Area Diameter	1000 µm = 1mm
Responsivity (1310nm)	0.90 A/W
Responsivity (1550nm)	0.95 A/W
Element Gap (mm)	0.045
Crosstalk (1550 nm, 5.0V)	1%
NEP (1550nm)	1.20E-14 W/√Hz



Quadcell Responsivity Graph



Detection Assembly: Quadcell

Quadrant Photodiode (Quadcell):

Four pixel InGaAs detector used for ground station

QC low crosstalk and noise equivalent power rates ensure signal accuracy, allowing for precise alignment.

OSI; AFG91 FCI-INGAAS-Q1000:

Active Area Diameter	1000 μm = 1mm
Responsivity (1310nm)	0.90 A/W
Responsivity (1550nm)	0.95 A/W
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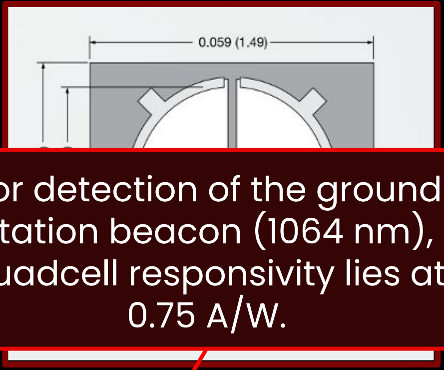
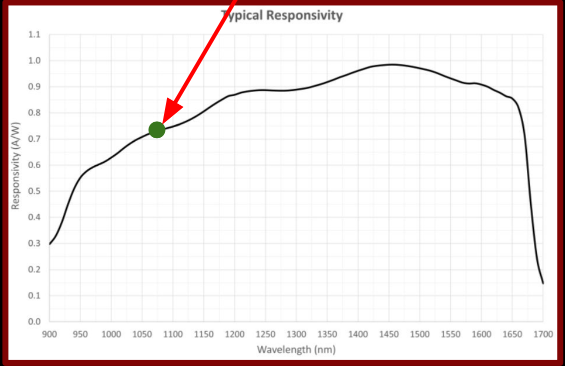


For detection of the ground station beacon (1064 nm), quadcell responsivity lies at 0.75 A/W.

Quadcell Responsivity Graph



Alignment Control: Quadcell

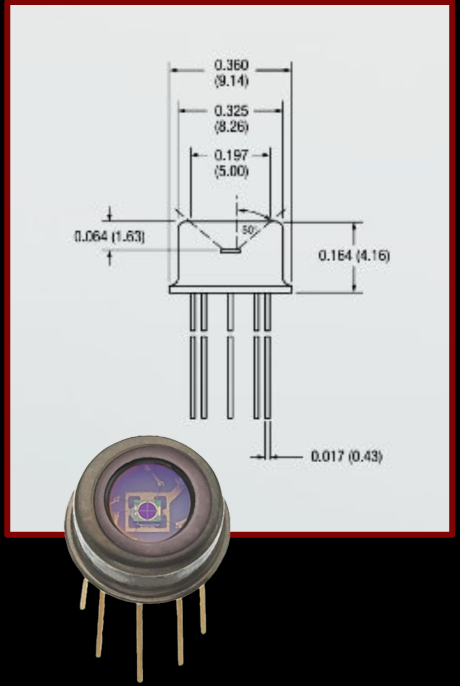
Quadrant Photodiode PSDS:

Quadcell data exhibits excellent responsivity from 1100 nm to 1620 nm, and remains stable over time and temperature variations. Electronic specifications for beam alignment applications are represented here.

OSI; AFG91 FCI-INGAAS-Q1000:

Active Area Diameter	1 mm
Dark Current (5.0V)	0.5 nA (15 nA Max)
Rise Time (5.0V, 50Ω, 10% to 90%)	3 ns
Max Reverse Voltage (V)	15
Operating Temp (°C)	-40 to 75

Quadcell Dimensions



Alignment Control: Quadcell

Quadrant Photodiode PSDS:

Quadcell data exhibits excellent responsivity from 1100 nm to 1620 nm, and remains stable over time and temperature variations.

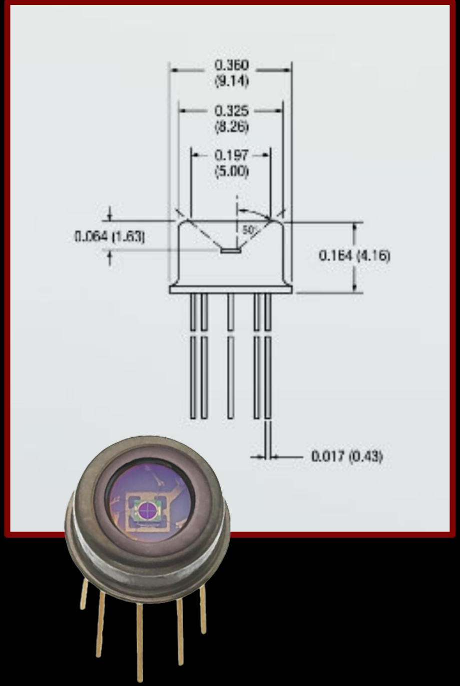
El n
al
re

- ❏ Tradeoff between cell diameter and dark current works out in our favor.
- ❏ Large enough to meet fine pointing requirements, sensitive enough to detect uplink beacon.

OSI; AFG91 FCI-INGAAS-Q1000:

Active Area Diameter	1 mm
Dark Current (5.0V)	0.5 nA (15 nA Max)
Rise Time (5.0V, 50Ω, 10% to 90%)	3 ns
Max Reverse Voltage (V)	15
Operating Temp (°C)	-40 to 75

Quadcell Dimensions



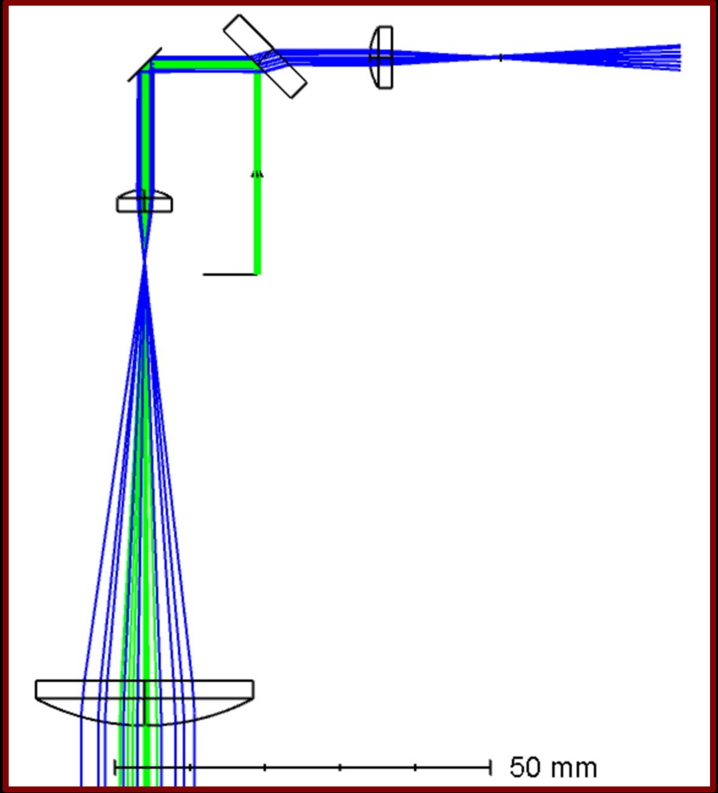
Zemax OpticStudio Payload Simulation Results (1/5)

We will use Zemax to:

- ❏ Determine optical component feasibility & performance
- ❏ Simulate optical loss and detection limits
- ❏ Optimize optical path

Components Currently Simulated:

- ❏ Plano-Convex Lens Assembly
  - ❏ Telescope lenses and detector lens
- ❏ FSM
- ❏ Shortpass dichroic
- ❏ Quadcell
- ❏ 1064 nm ground beacon laser (BLUE)
- ❏ 1550 nm Transmission Laser (GREEN)



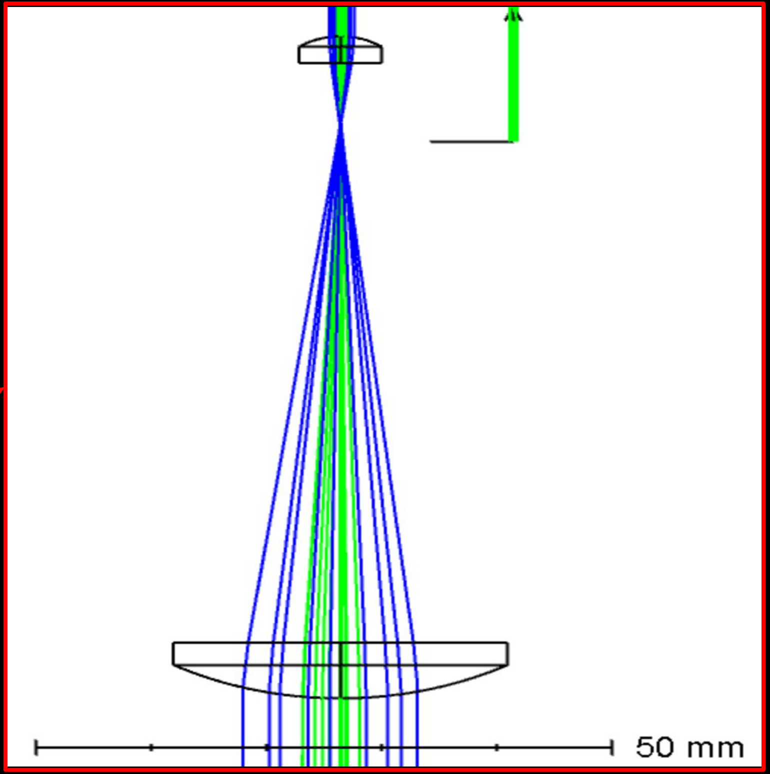
Zemax OpticStudio Payload Simulation Results (2/5)

Keplerian Beam Expander:

- 1064 nm ground beacon laser (BLUE)
- 1550 nm Transmission Laser (GREEN)

Modelling this sub-assembly allows us to:

- Model optimal beam diameter received by lens (avoiding edge aberrations).
- Measure power density and beam diameter changes through beam expander.

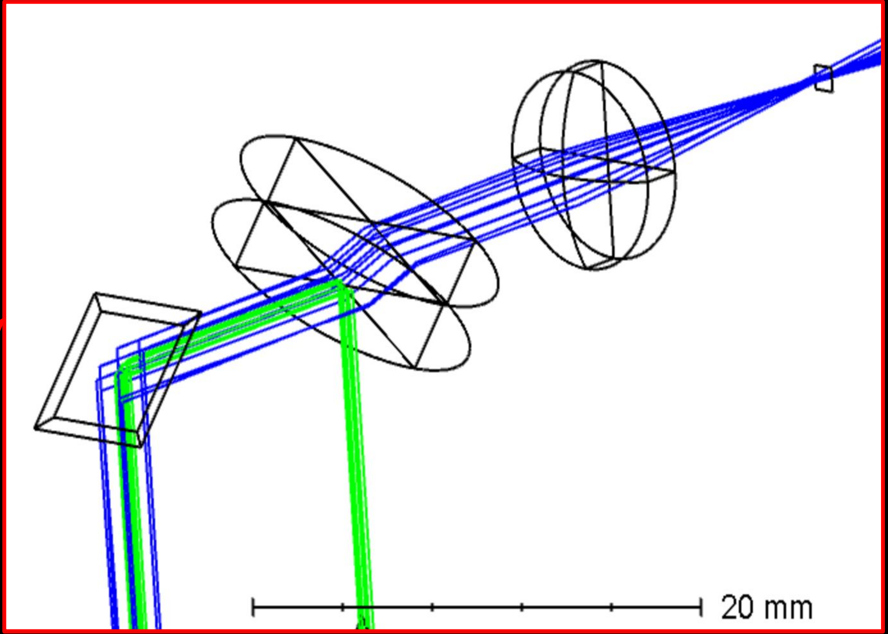


Zemax OpticStudio Payload Simulation Results (3/5)

Pointing and Detection:

- ❏ 1064 nm ground beacon laser (BLUE)
- ❏ 1550 nm Transmission Laser (GREEN)

FSM tilt between +/- 4.25 degrees gives us well over 1 degree of variation on our output beam (i.e., analysis shows we can correct for body mispointing up to ~1.5 degrees).



Zemax OpticStudio Payload Simulation Results (4/5)

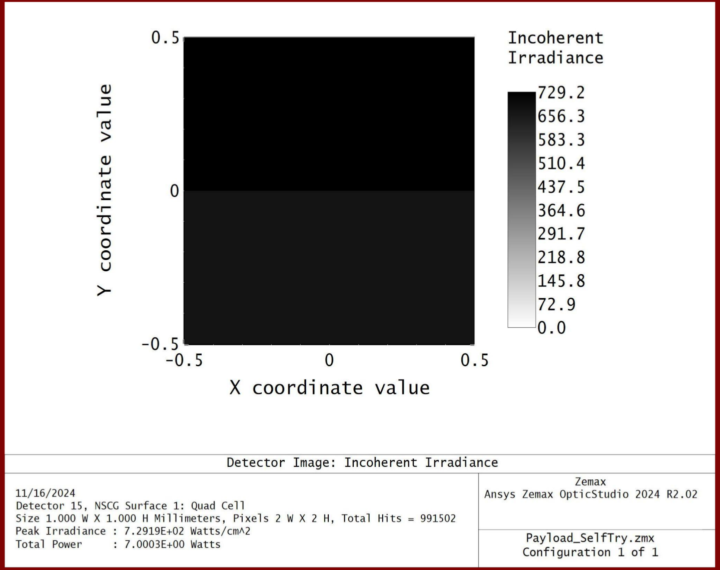
Quadcell Zemax Detector Simulation for Ground Beacon Laser:

- 2x2 pixels; uniform irradiance
- Total Power Loss: ~30%
- Peak Irradiance: 729 W/cm^2

Our predicted power loss is consistent with Zemax simulation, & we can use Zemax to help confirm predictions

Causes for Power Loss from substrate data sheets:

- Lens 1,3 (N-BK7): 92.2% transmittance
- Lens 2 (N-SF11): 87.3% transmittance
- Dichroic (UVFS): 94.9% transmittance
- Expected power loss from lenses: ~31%



Zemax Quadcell Detector Viewer; 4 pixels receiving equal power (ground beacon beam in center). Provides total power and power distributed across pixels.

Zemax OpticStudio Payload Simulation Results (5/5)

<u>Quadcell Detector Simulation Results:</u> Identify Pointing of Ground Beacon Laser

- Consider Small FSM Tilt angle: 0.500°
- Can estimate beam centroid from quadrant irradiance:

$$X_{pos} = \frac{(A+B)-(C+D)}{A+B+C+D}$$

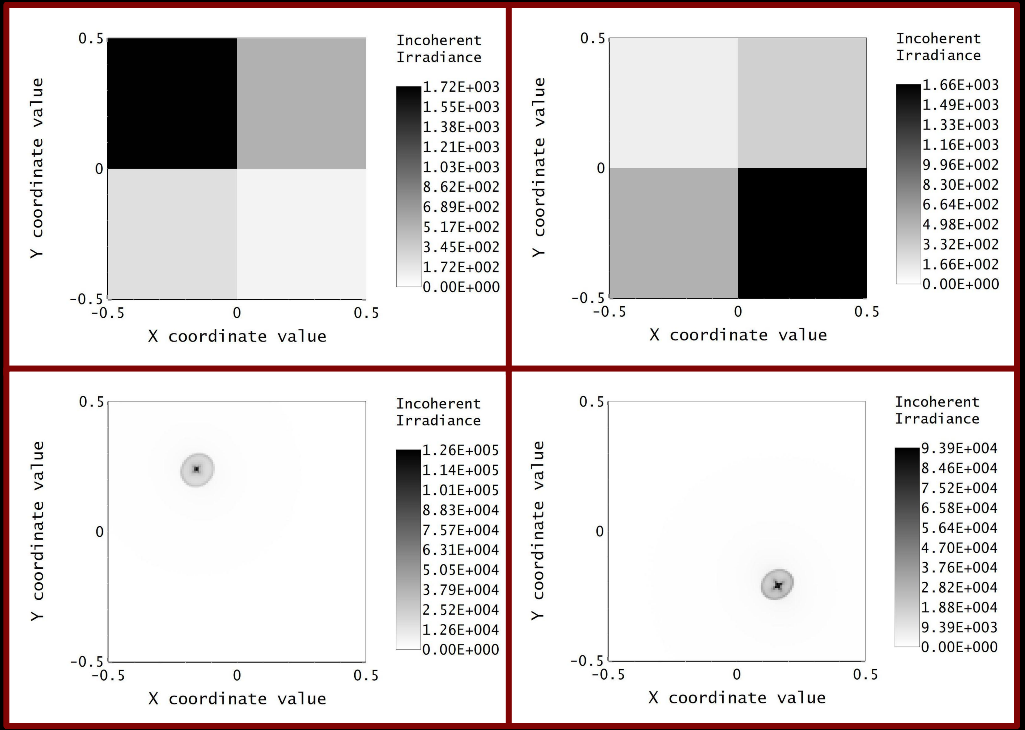
$$Y_{pos} = \frac{(B+C)-(A+D)}{A+B+C+D}$$

- A,B,C,D: Irradiances for
- each quadrant

Peak Irradiance Quadrant: (0,0) or "A" | (1,1) or "C"

Detected Light Intensity:

Actual Beam Position:



Zemax OpticStudio Payload Simulation Results (5/5)

<u>Quadcell Detector Simulation Results:</u> Identify Pointing of Ground Beacon Laser

- ❑ Consider Small FSM Tilt angle: 0.500°
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$$Y_{pos} = \frac{(B+C)-(A+D)}{A+B+C+D}$$

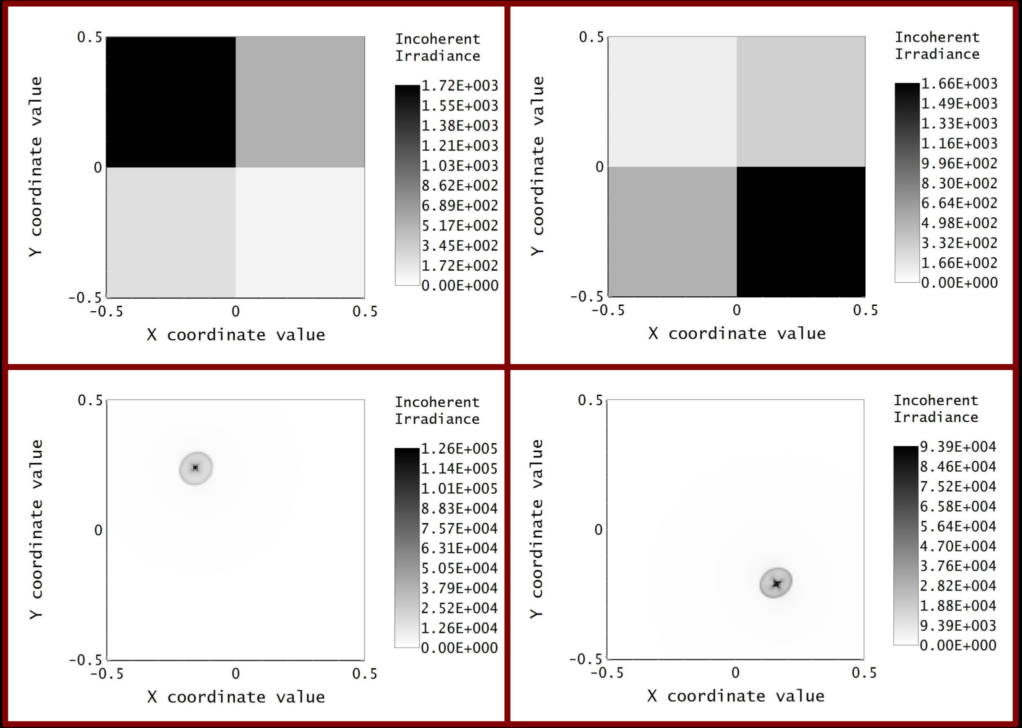
- ❑ A,B,C,D: Irradiances for
- ❑ each quadrant

Can Calculate Deviation from Center of Quadcell

Peak Irradiance Quadrant: (0,0) or "A" (1,1) or "C"

Detected Light Intensity:

Actual Beam Position:



Zemax OpticStudio Payload Simulation Results (5/5)

<u>Quadcell Detector Simulation Results:</u> Identify Pointing of Ground Beacon Laser

- Consider Small FSM Tilt angle: 0.500°
- Can estimate beam centroid from quadrant irradiance:

Comparing Detector Output & Actual Beam Position

$$Y_{pos} = \frac{(B+C)-(A+D)}{A+B+C+D}$$

- A,B,C,D: Irradiances for each quadrant

Peak Irradiance Quadrant:

(0,0) or "A"

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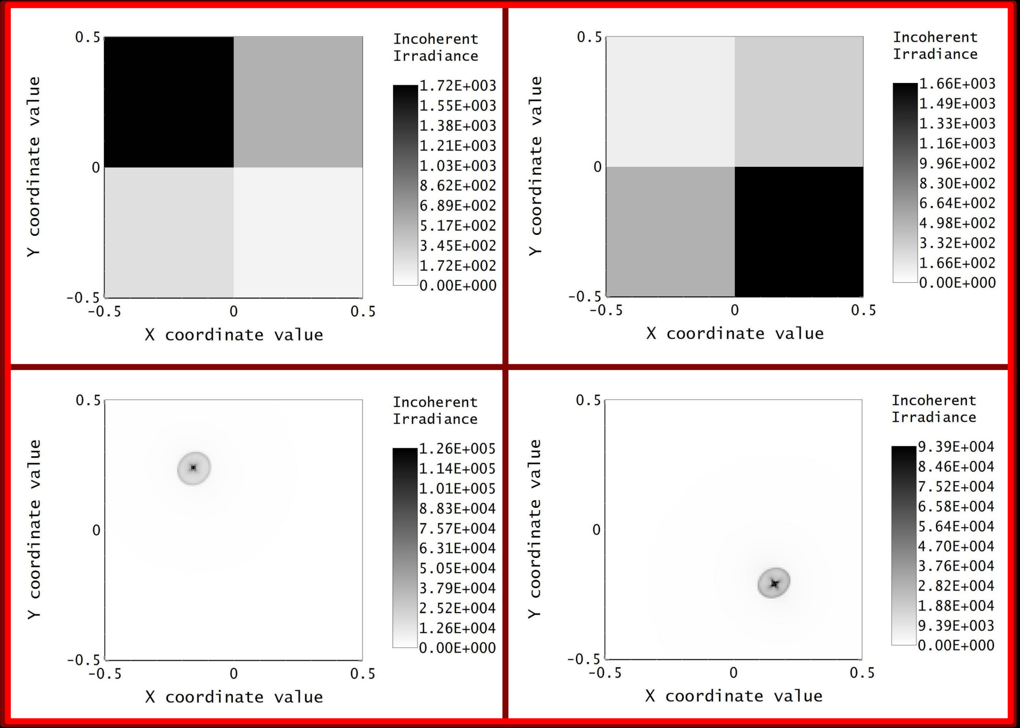
Detected Light Intensity:

Actual Beam Position:



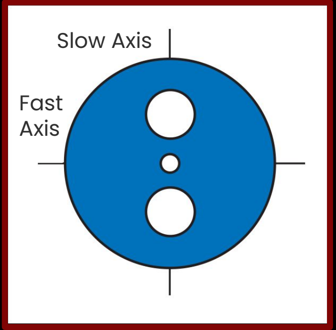
Polarization-Maintaining Components (1/2)

PANDA (polarization-maintaining and absorption-reducing) fiber

Stress conditions in PANDA have been studied extensively (Polarization effects abroad the Space Interferometry Mission, JPL). A second alternative is Bow-Tie fiber. Further testing is necessary to simulate our orbit conditions and test both fibers performances. The components that are fiber in-out will possibly arrive with standard PM PANDA fibers pre-coupled.

PM1550nm PANDA 1440 - 1625 nm, 0.125 NA, 10.1 µm MFD

Operating Wavelength (nm)	1270 – 1625
Beat Length	≤4.0 mm @ 1300 nm
Bend Radius (mm)	30
Attenuation	≤1.0 dB/km @ 1300 nm
Coating Material	UV Cured, Dual Acrylate
Operating Temperature (° c)	-40 to 85



The slow and fast axes of a PM are for two modes of polarization, but the fiber only maintains one at a time.

Polarization-Maintaining Components (1/2)

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PM fibers only maintain one polarization state as the input. Two polarization inputs will generate a slightly elliptical beam.

PM1550nm PANDA 1440 - 1625 nm, 0.125 NA, 10.1 µm MFD

Operating Wavelength (nm)	1270 – 1625
Beat Length	≤4.0 mm @ 1300 nm
Bend Radius (mm)	30
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Coating Material	UV Cured, Dual Acrylate
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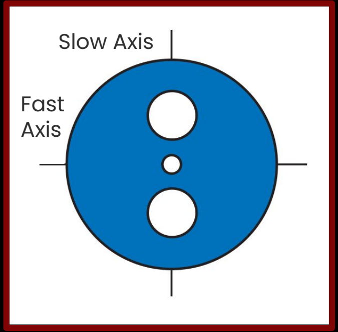

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Polarization-Maintaining Components (1/2)

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Changes in phase shift and amplitude difference between the orthogonal polarizations introduced by fiber and stress will make the beam elliptical at the output.

PM1550nm PANDA 1440 - 1625 nm, 0.125 NA, 10.1 µm MFD

Operating Wavelength (nm)	1270 – 1625
Beat Length	≤4.0 mm @ 1300 nm
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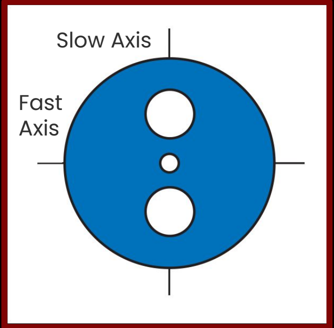


The slow and fast axes of a PM are for two modes of polarization, but the fiber only maintains one at a time.

Polarization-Maintaining Components (2/2)

EDFA (Erbium Doped Fiber Amplifier)

PDG	3 to 5%

Polarization dependent gain (PDG) is around 3 to 5%. Indicates de device's tendency to amplify the signal of one polarization slightly more than the other. This is small enough such that the EDFA is still considered a polarization-independent component.

Polarization Switch

SOP (degrees)	90 ± 0.5
Extinction Ratio (db)	18
Cross-Talk (dB)	-30 @1550 nm

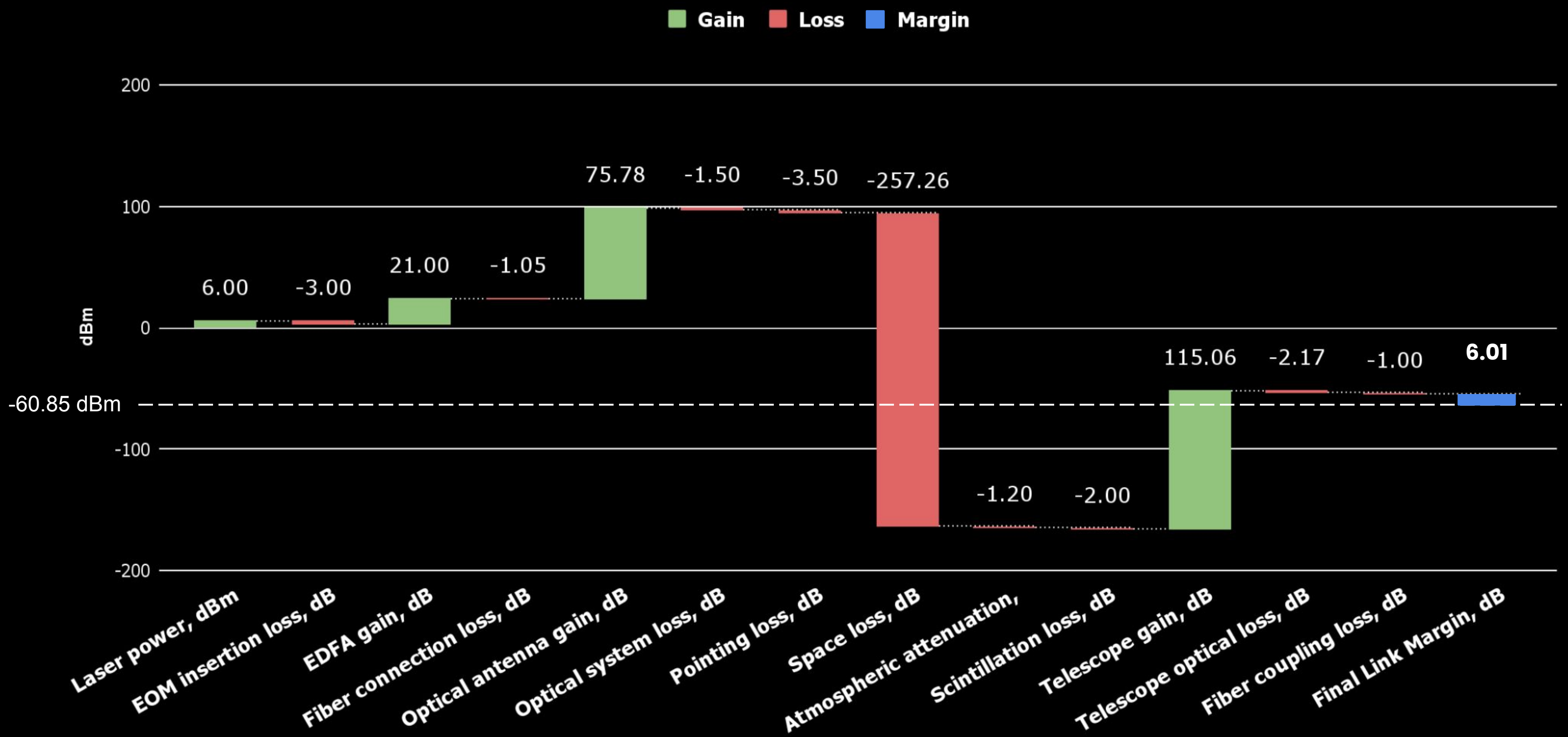
State of Polarization Tolerance (SOP) of a rotation is 90 ± 0.5 degrees, indicating precise polarization control.

Extinction Ratio of 18 dB, or approximately 98.4%. Is the ratio of the power in the intended polarization polarization to the power in the orthogonal state.

Cross Talk of -30 dB, or approximately 0.1% unwanted leakage power between orthogonal polarizations.

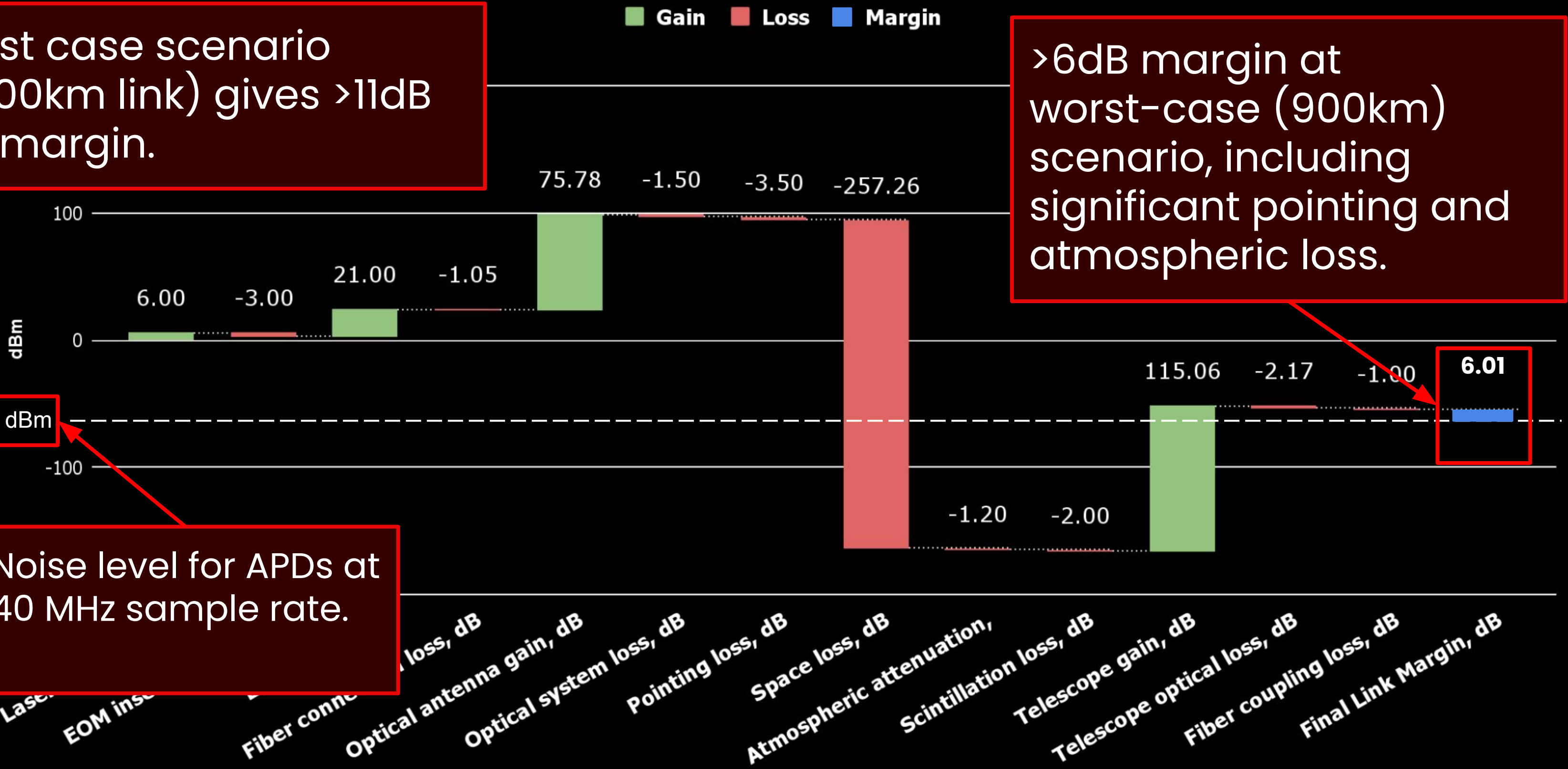
To a smaller degree, other components will introduce differences in amplitude between the two orthogonal polarizations.

Transmission Optical Link Budget Summary



Transmission Optical Link Budget Summary

Best case scenario (500km link) gives >11dB of margin.



>6dB margin at worst-case (900km) scenario, including significant pointing and atmospheric loss.

Noise level for APDs at 40 MHz sample rate.

Payload Beacon Optical Link Budget Summary

Operating Wavelength: 638nm

Beacon transmit power, dBm	23.00
Received power, dBm	-87.20
Equivalent received power, W	1.91E-12
Sirius App. Mag. (V)	-1.46
Flux of Sirius at Earth, W/m^2	1.00E-07
Flux of Beacon at Telescope, W/m^2	3.11E-11
Beacon Apparent Magnitude	7.31

Payload Beacon Optical Link Budget Summary

Operating Wavelength: 638nm

Beacon transmit power, dBm	23.00
Received power, dBm	-87.20
Equivalent received power, W	1.91E-12
	-1.46
	1.00E-07
m^2	3.11E-11
	7.31

Using ~200mW output power with large divergence angle, designed for ~9dB of loss at 1 degree of satellite mispointing.

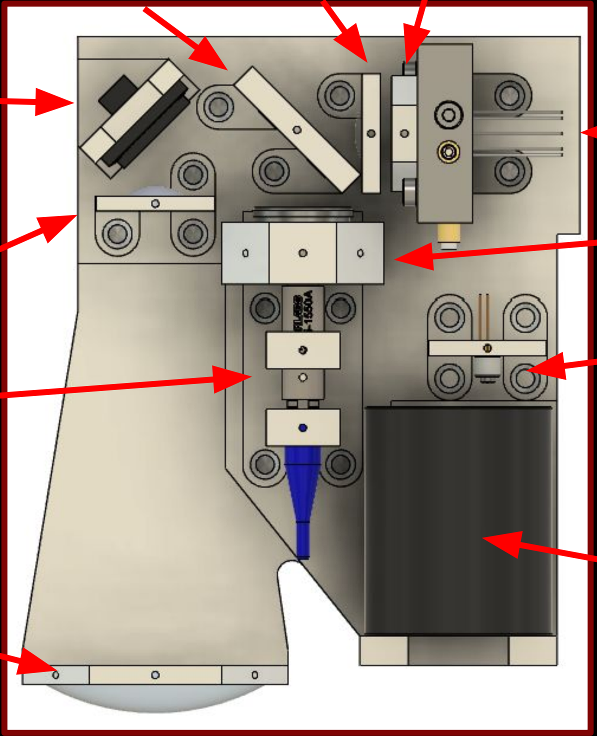
Payload Beacon Optical Link Budget Summary

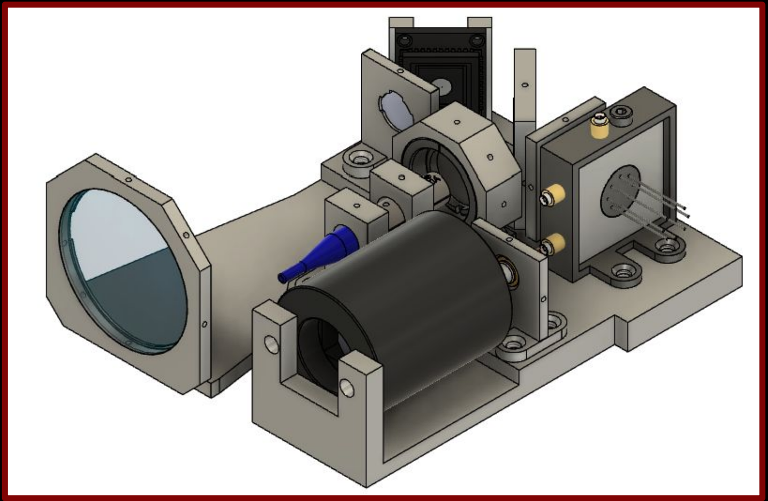
Operating Wavelength: 638nm

Link budgeting is more complex, but using flux calculations, we find worst-case ~7 visual magnitude (TBR). Integration time necessary on tracking camera will be experimentally determined.

	23.00
	-87.20
	1.91E-12
Sirius App. Mag. (V)	-1.46
Flux of Sirius at Earth, W/m^2	1.00E-07
Flux of Beacon at Telescope, W/m^2	3.11E-11
Beacon Apparent Magnitude	7.31

Payload Optical Mounting Overview

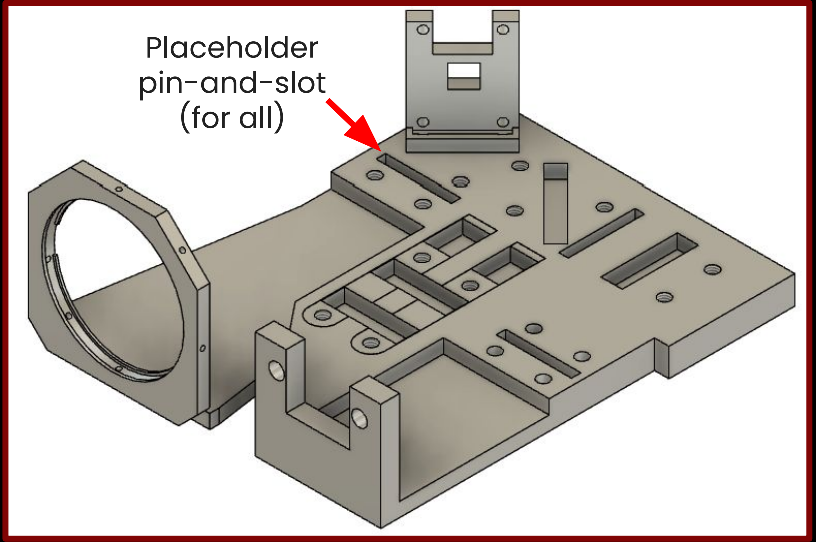




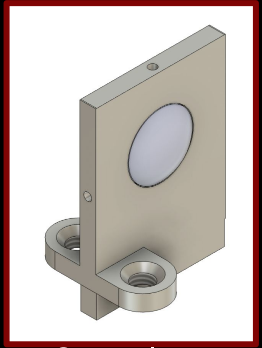
Using aluminum to make the mounts and breadboard has benefits:

- ❑ Lightweight
- ❑ Easy to manufacture and iterate
- ❑ Conductivity, heat capacity

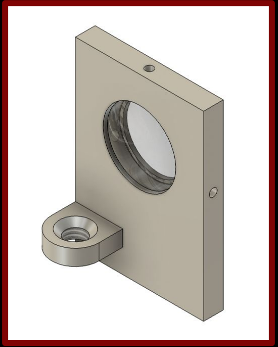
Breadboard and Basic Mounts



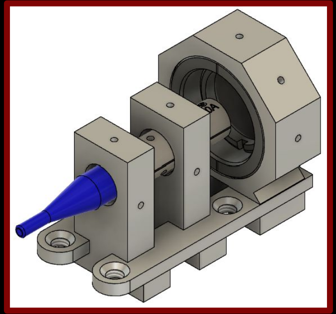
Pulse-A Optical Breadboard



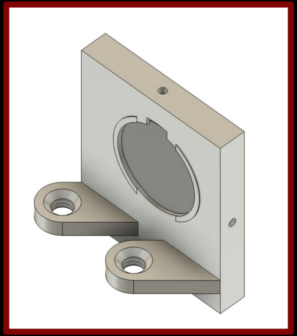
8 mm lens mount



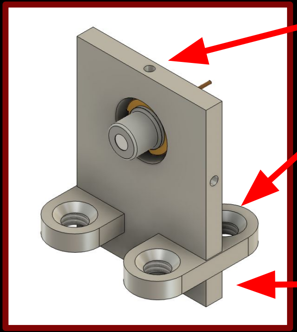
9 mm lens mount



Collimator + quarter waveplate mount



Dichroic mount



Beacon laser mount

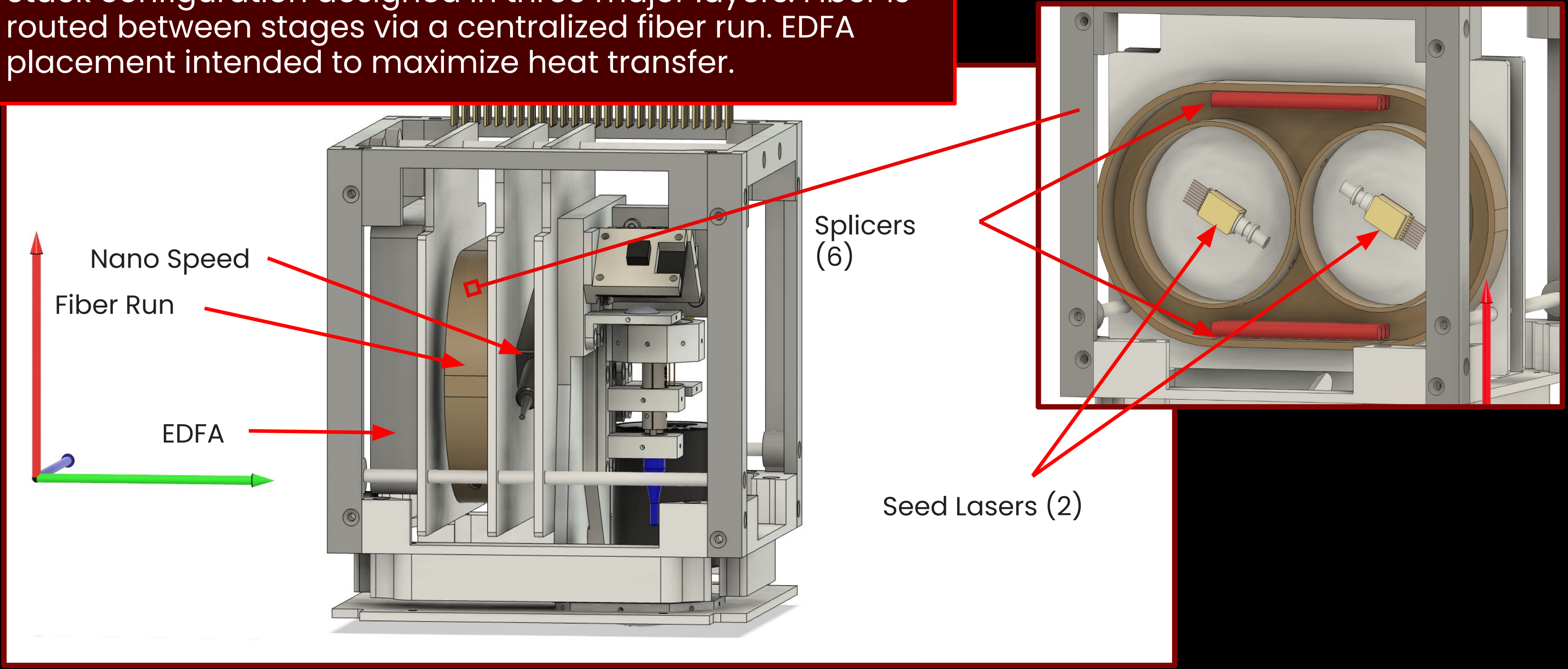
RTV injection ports

Holes for screwing mounts into breadboard

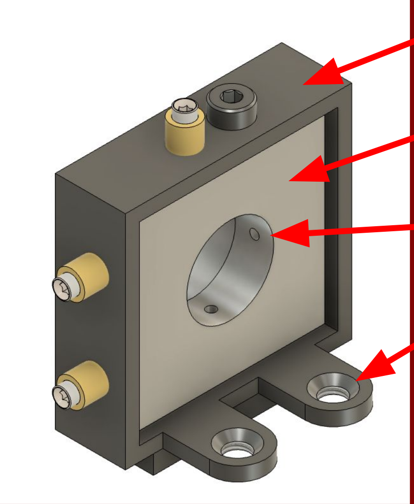
Extended base locates mounts on optical bench

Payload Stack Configuration

Stack configuration designed in three major layers. Fiber is routed between stages via a centralized fiber run. EDFA placement intended to maximize heat transfer.



Adjustable Mounts

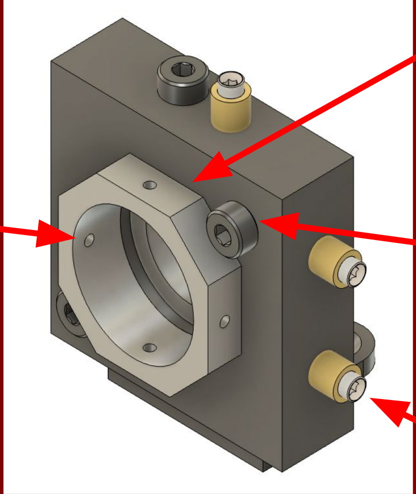


Mount support

Lens mount

RTV injection ports

Screw holes for attaching mount support to breadboard



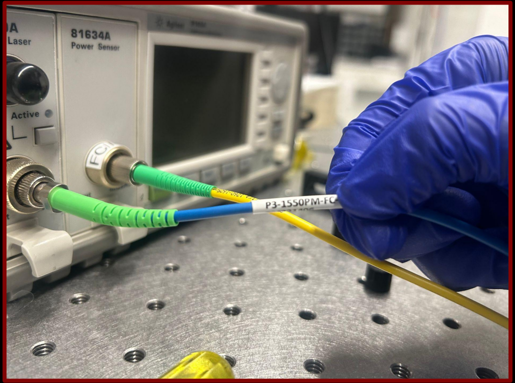
BP filter mount is threaded to secure in mount support

Bolts will have Belleville washers to preload mount

Adjustment screws for fine positioning

- Moves normal to the direction of light
- Adjustable so that we can assure perfect alignment
- Adjustable mount will be implemented for other lenses, this is just an example

Next Steps for Payload Design



- ❑ Finalize simulations & math for component selection
  - ❑ Optical path pointing for FSM diameter
  - ❑ Divergence angle for collimators
- ❑ Test to confirm final components
  - ❑ Polarization switch
  - ❑ Collimators
  - ❑ Lasers
- ❑ Purchase models for lab testing
  - ❑ Test PM fibers for performance under stress conditions
  - ❑ Test the beam size and power flux in the QD Cell for feedback control
  - ❑ Test the FPGA signal to Polarization Switch for rate of data acquisition
  - ❑ Test prototype of set of first lenses with relative distances defined in Zemax
  - ❑ Test prototype of payload using FSM control
- ❑ Determine degree of space qualification needed for: Modulator, EDFA, FSM

Satellite System Design:

Structure and Bus Configuration

Structure Driving Requirements

ID	Requirement	Notes	Parent	Verification
STR-01, STR-02, STR-03	The Structure shall be compliant with the most up-to-date revision of the CubeSat Design Specification (CDS), the Launch Provider's requirements, and the Deployer Provider's requirements.	The most current CDS is revision 14.1. Launch and Deployer Providers are TBD until manifested by NASA.	SAT-01, SAT-02, SAT-03	Inspection
STR-04	The Structure shall provide an external aperture for the OGS laser beacon's reception and downlink laser transmission.		PAY-01, PAY-04	Inspection
STR-08	The Structure shall secure any appendages prior to deployment.		STR-01, STR-02, STR-03	Inspection
STR-12, STR-13	The Structure shall maintain the Payload's optical path lens alignment within 0 mm ± TBD mm after launch/during operations.	Launch refers to vibration effects, operations refer to thermal effects. TBD depends on PAY-18.	PAY-18	Test

Legend: Compliant Compliant by CDR Compliant by TRR Compliant By FRR

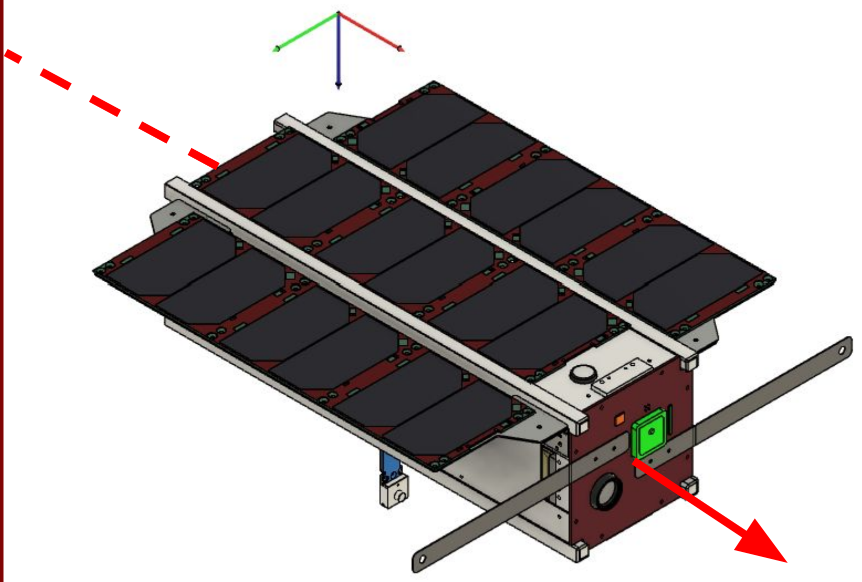
Notes on Nomenclature

During coasting, which is the primary orientation regime, the x-axis vector is collinear with the spacecraft's velocity vector. Z-axis is defined as nadir-pointing, leaving solar panels as zenith-pointing.

In this section, we define the terms:

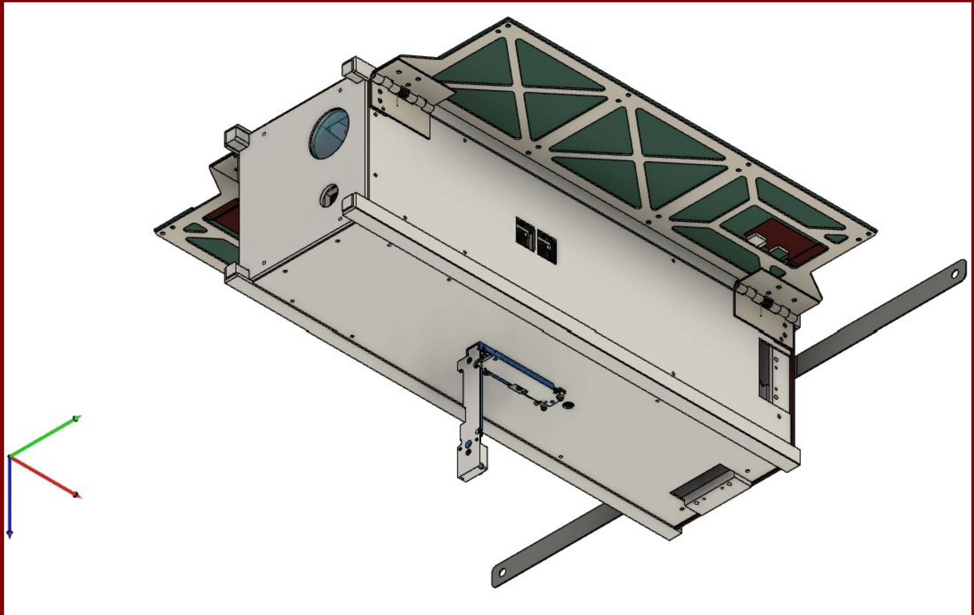
- "Front" - face with deployed solar panels
- "Top" - face with antennas, star tracker
- "Bottom"- face with payload lens in/out
- "Back"- face with magnetometer

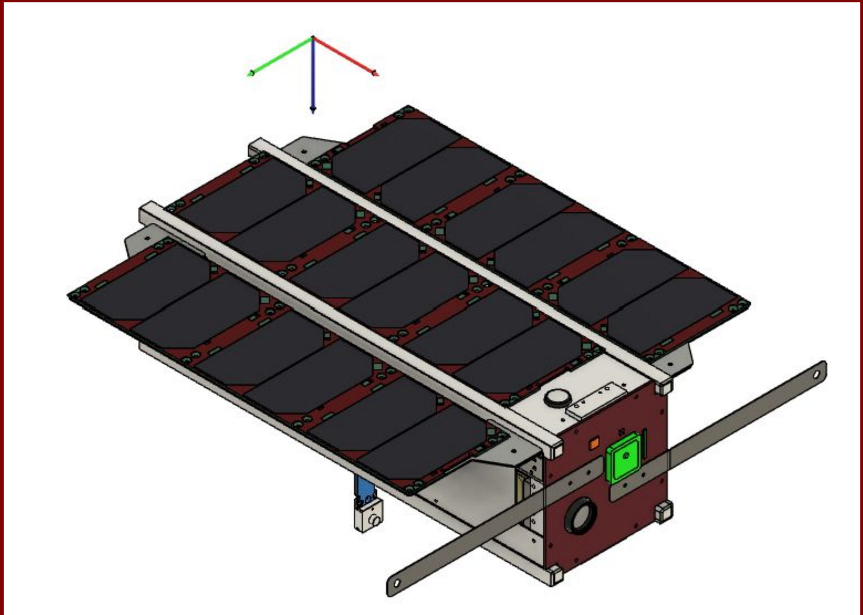
This bottom-top definition aligns with PC-104 standard.



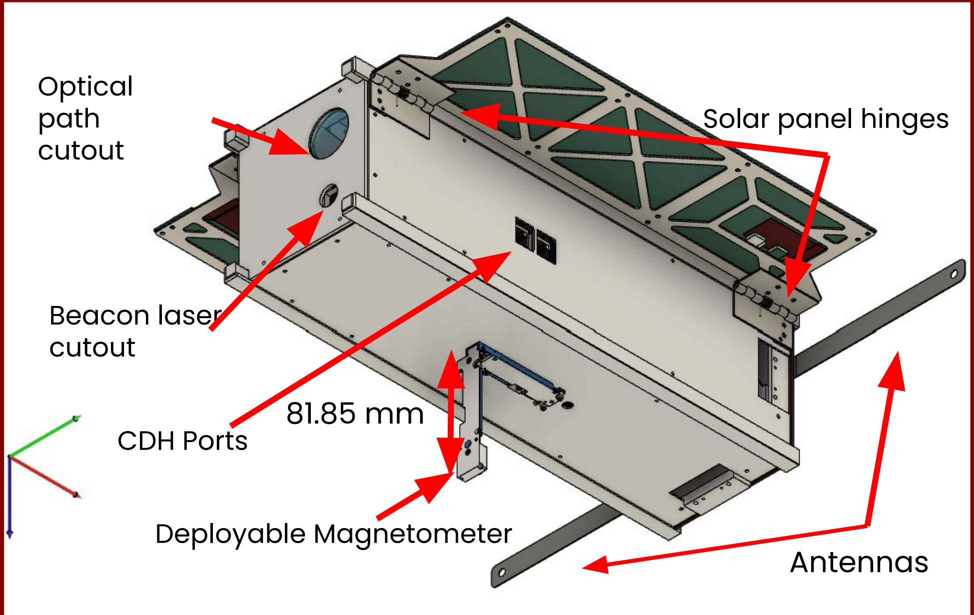
Orbital velocity vector (red), collinear with x-axis

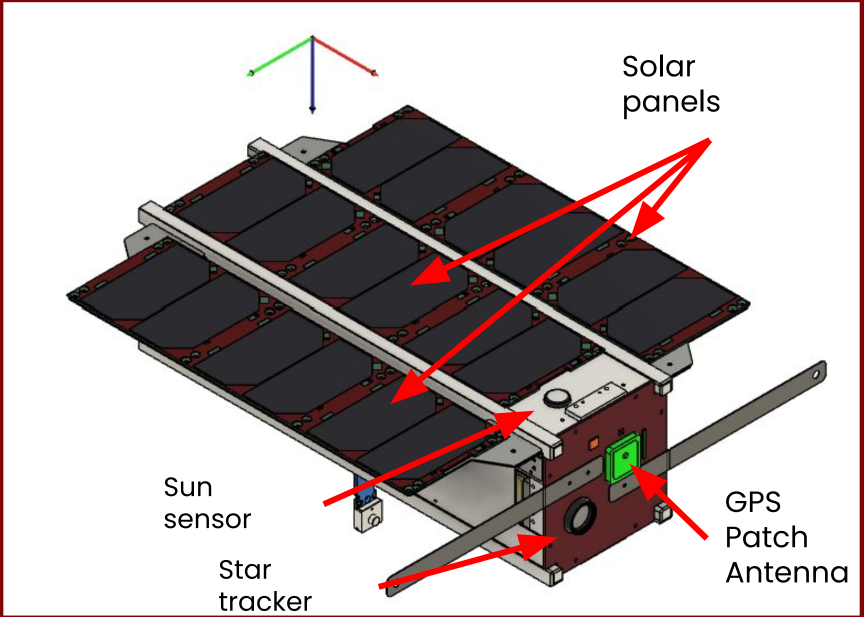
External Configuration (1/4)



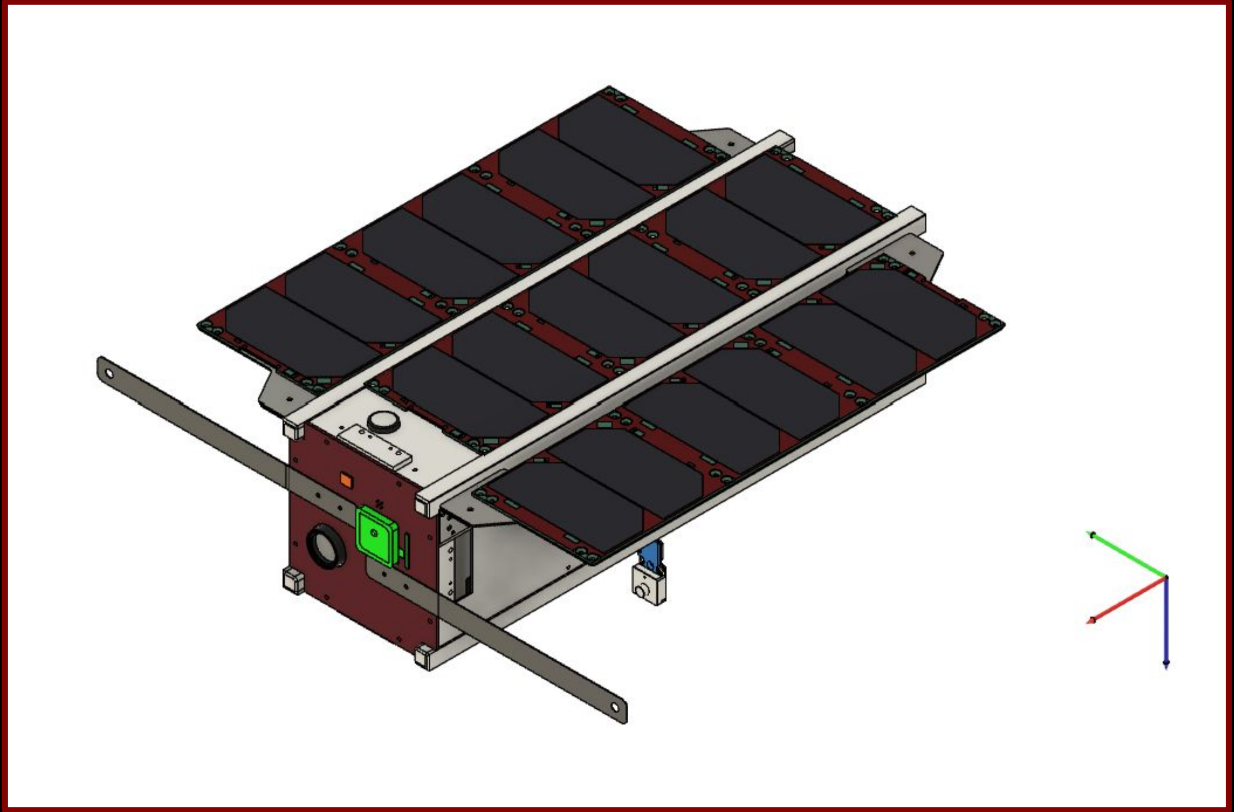


External Configuration (1/4)

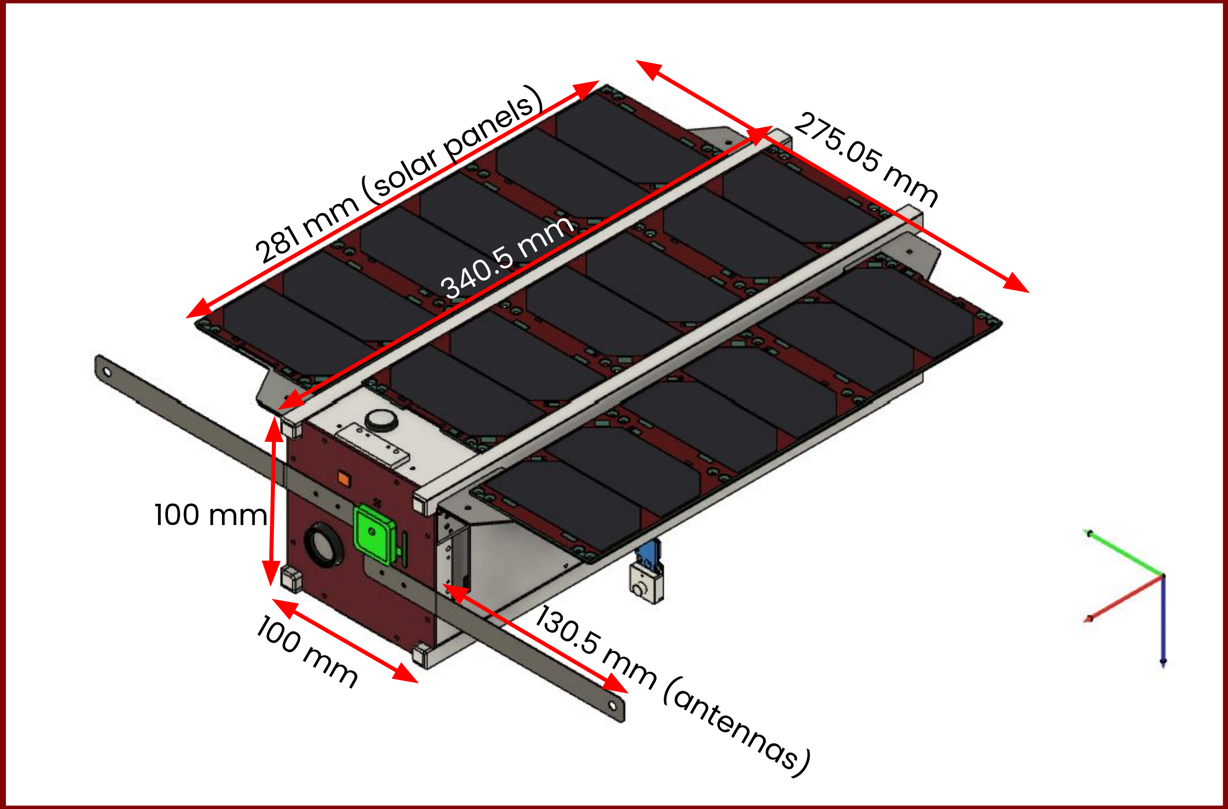




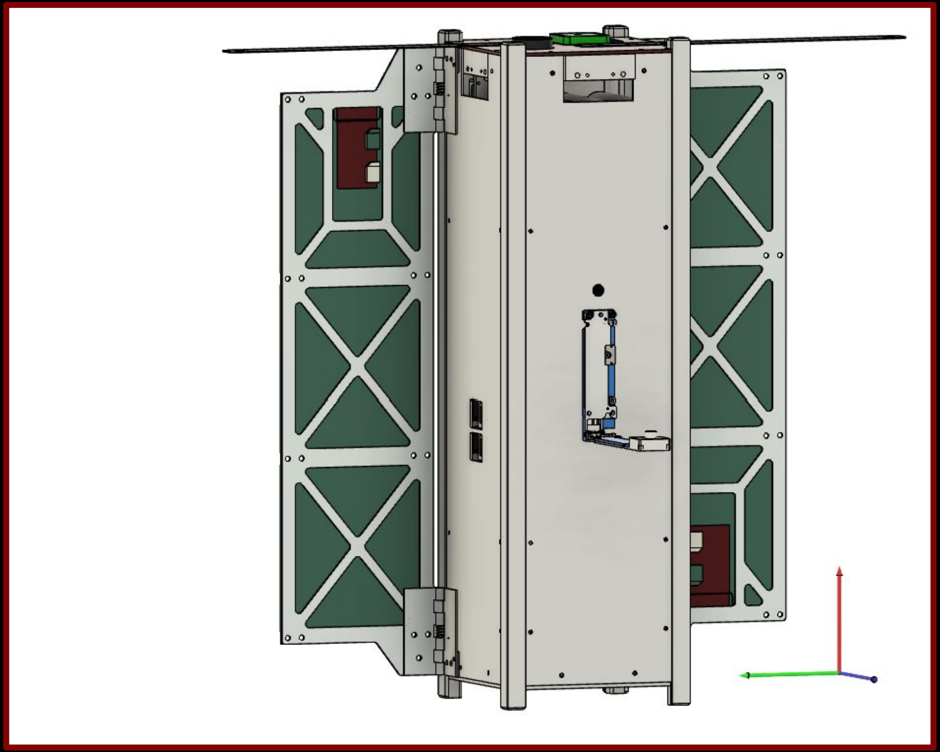
External Configuration (2/4)

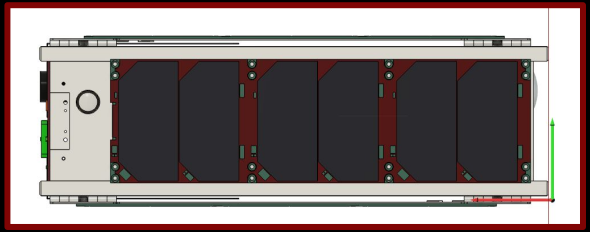


External Configuration (2/4)

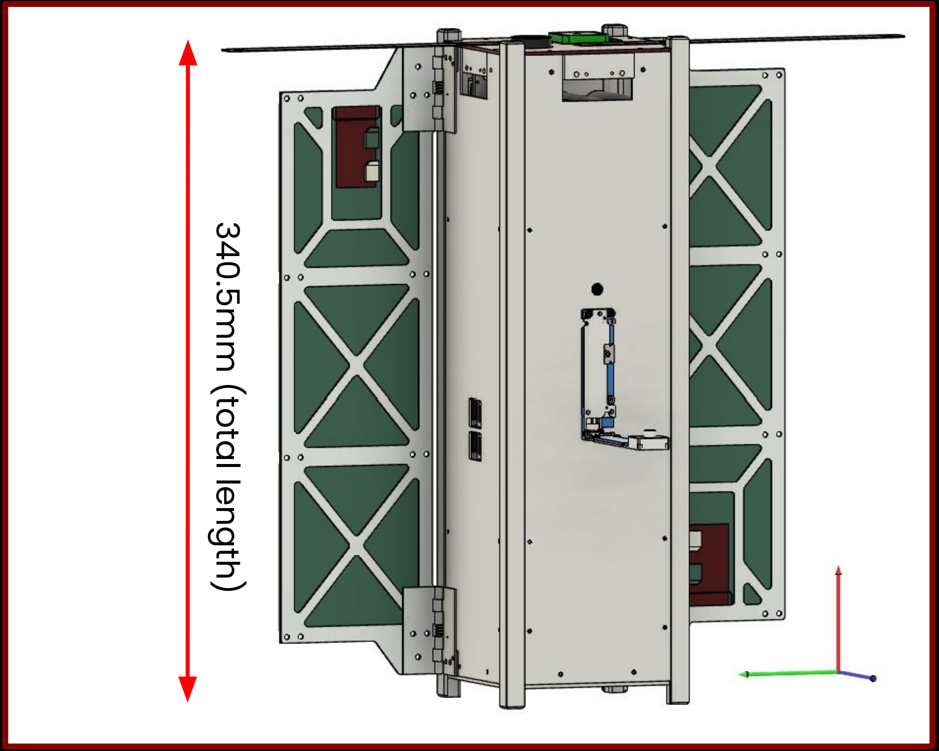


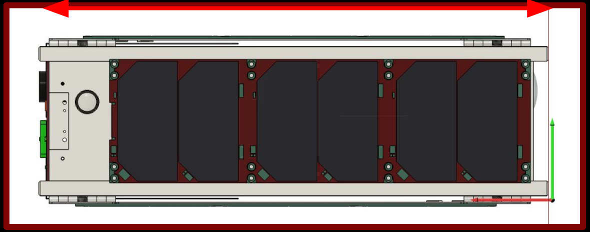
External Configuration (3/4)



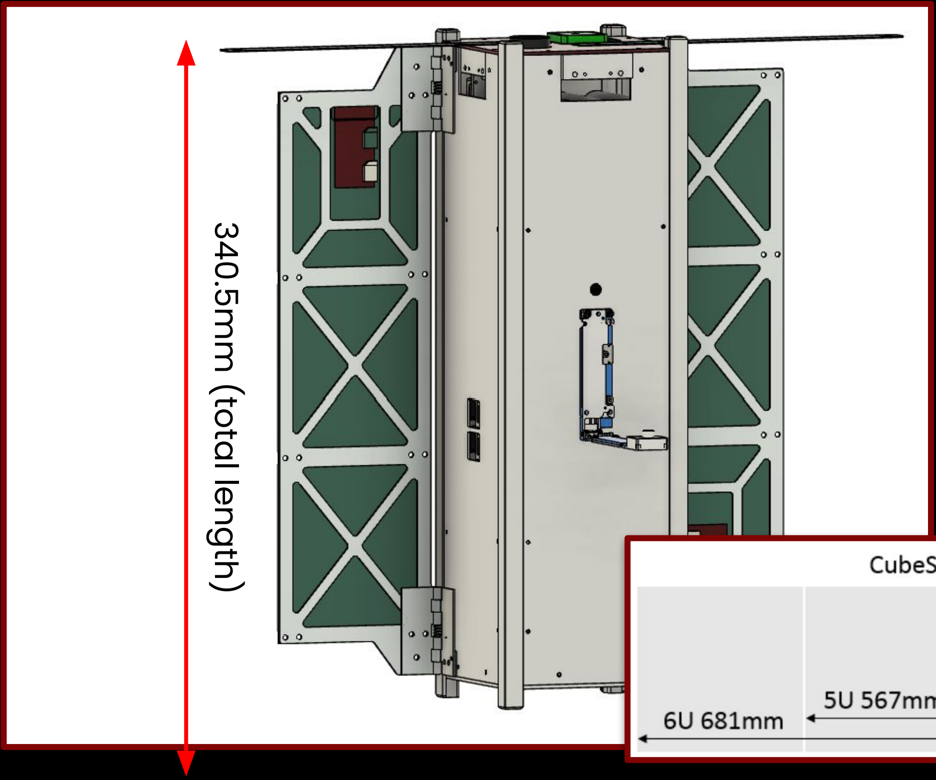


External Configuration (3/4)

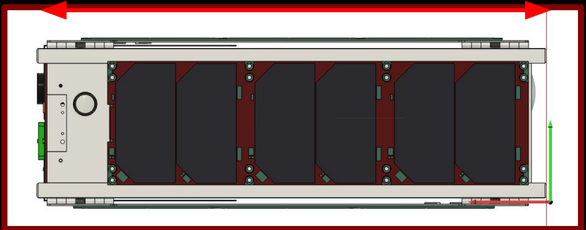


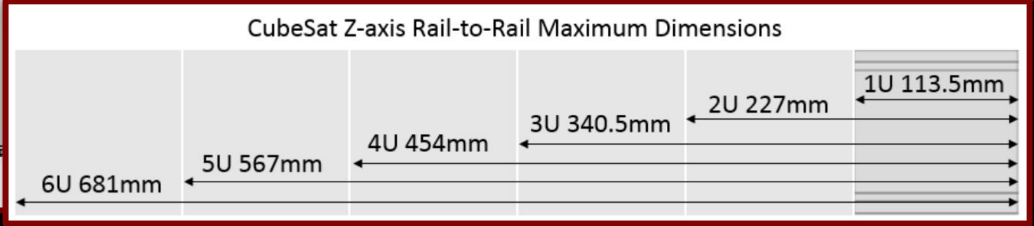


External Configuration (3/4)



340.5mm (total length)





External Configuration (4/4)

Min. clearance on rails is 6mm, we offer 6.239mm (meets specification requirements)

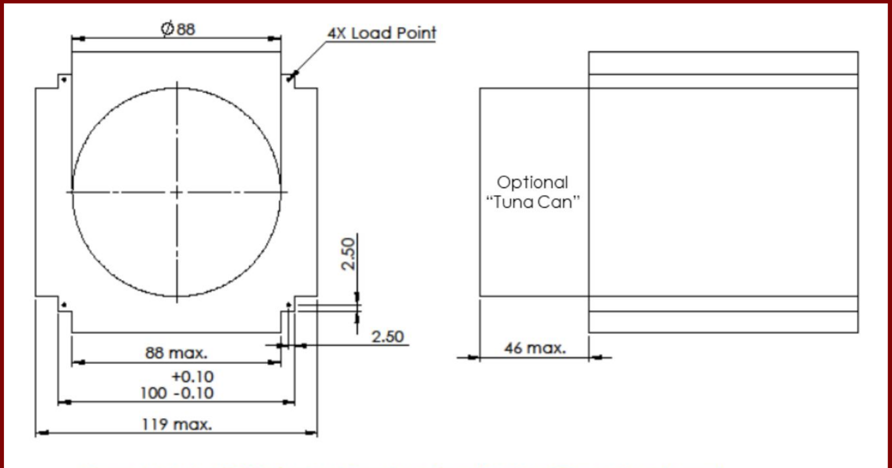
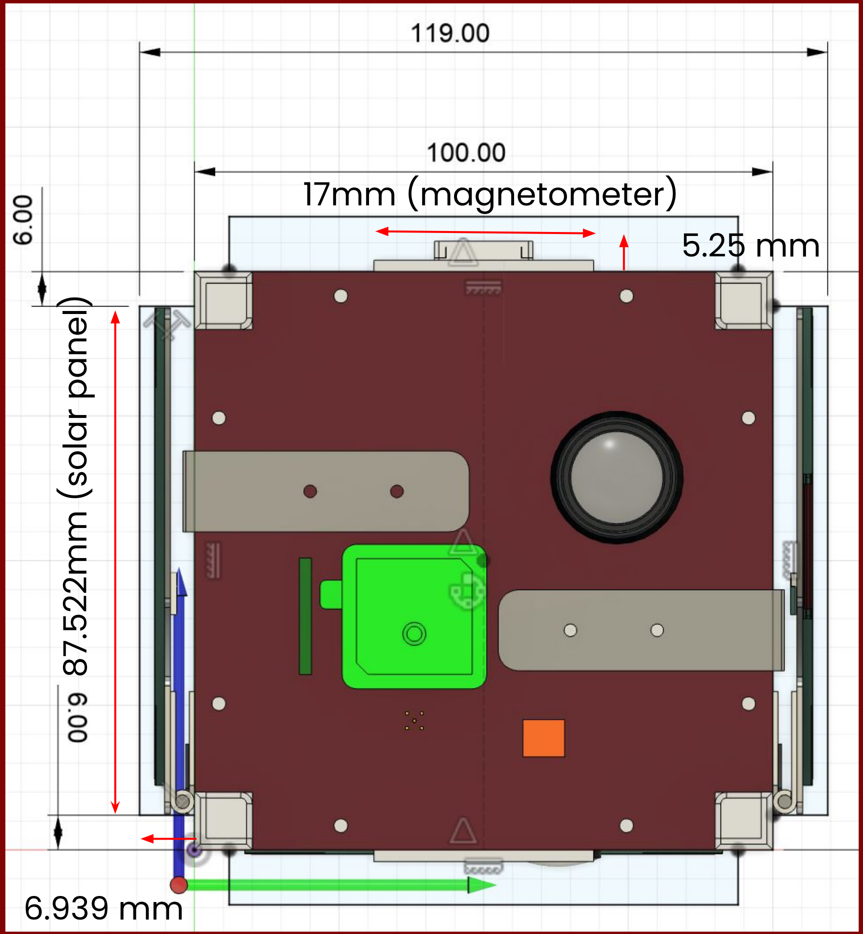
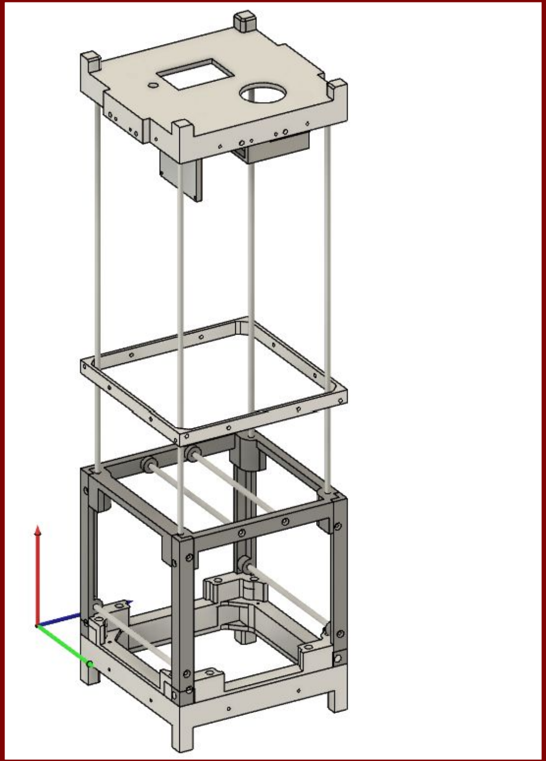
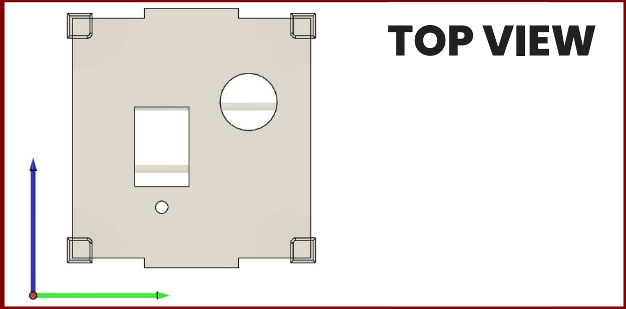


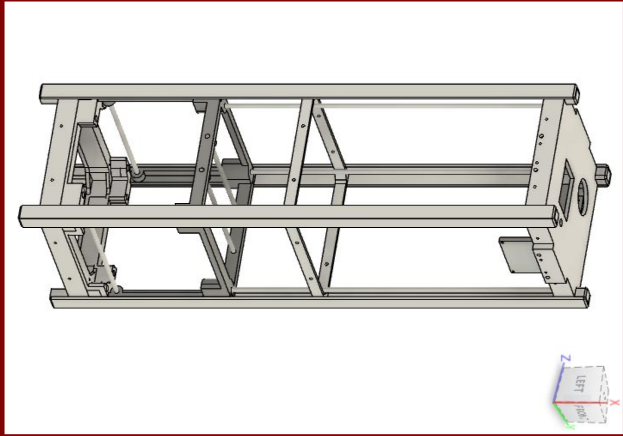
Figure 4.1.1-1: NRCSD Payload Envelope Specification (Dimensions in mm)

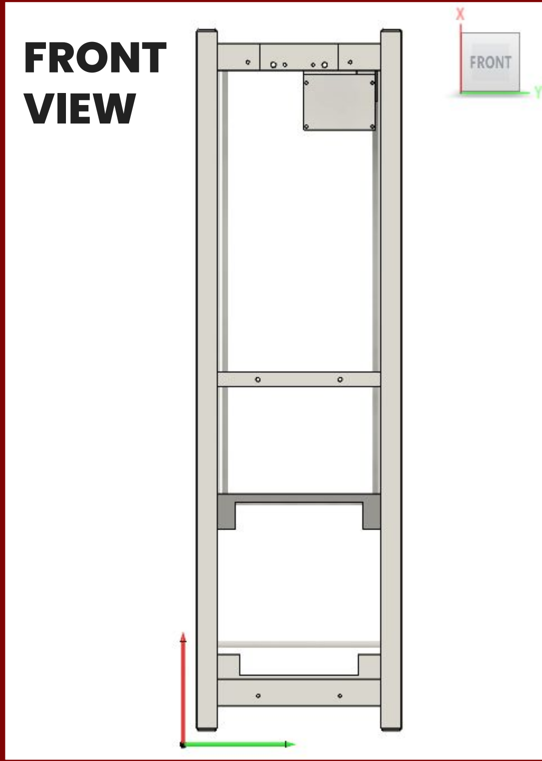


Frame Configuration

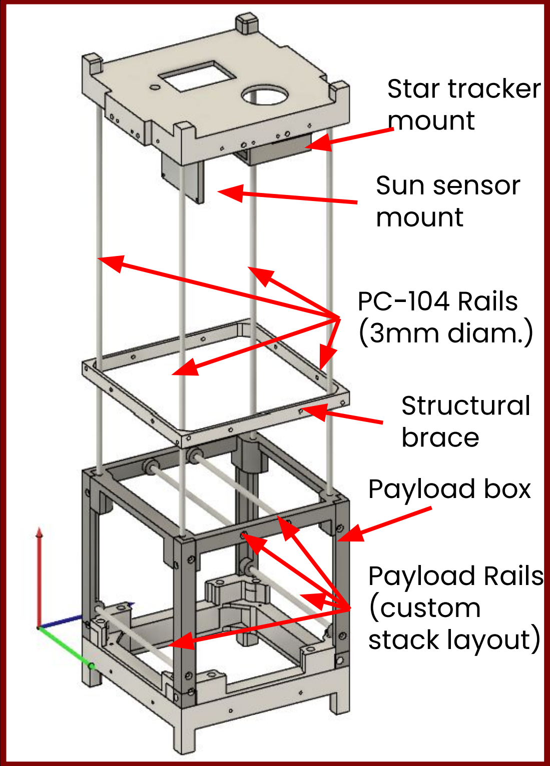


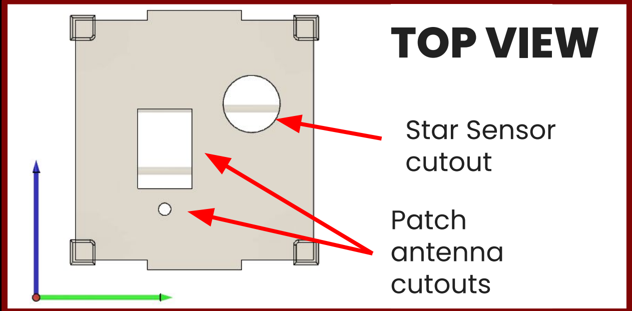


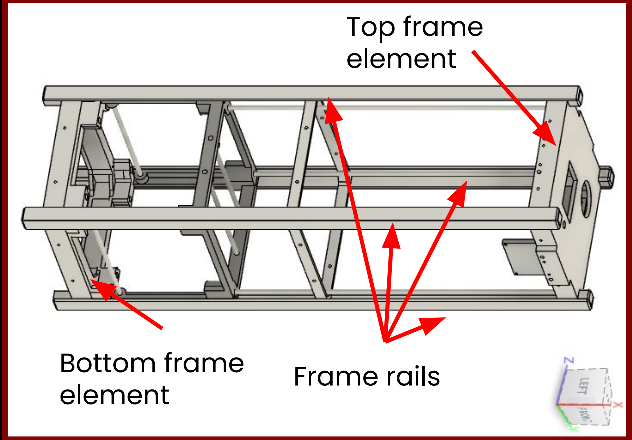


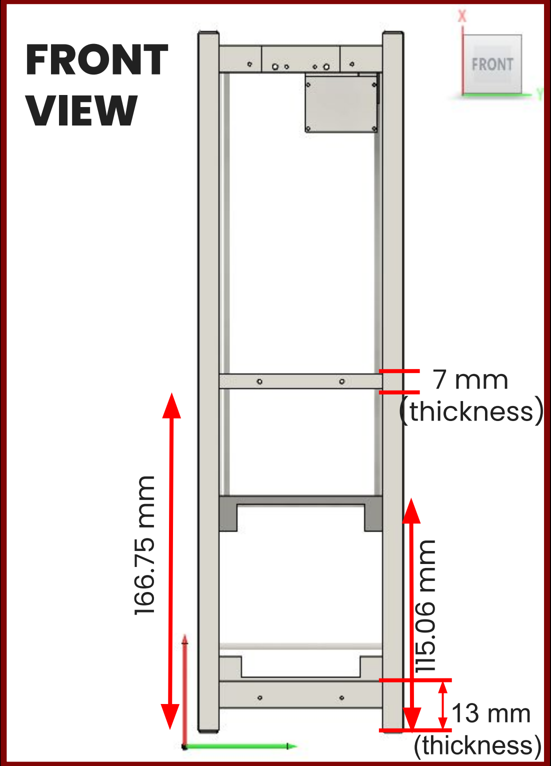


Frame Configuration







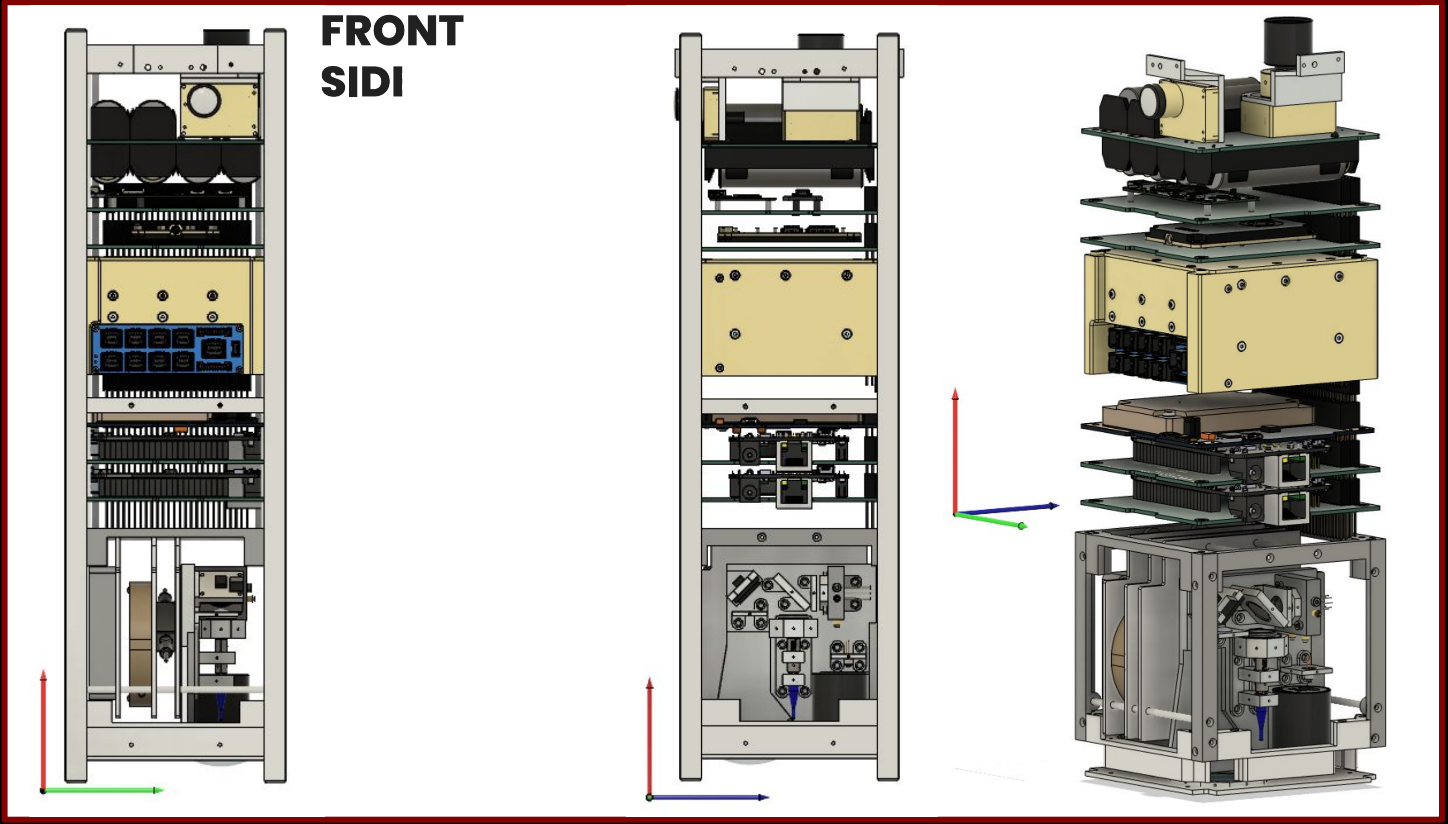


Internal Configuration

Full stack shown.

All boards connected and spaced with PC-104 standard, SSQ-104 pin connections.

Boards fit on rails with teflon tube spacers (TBD).

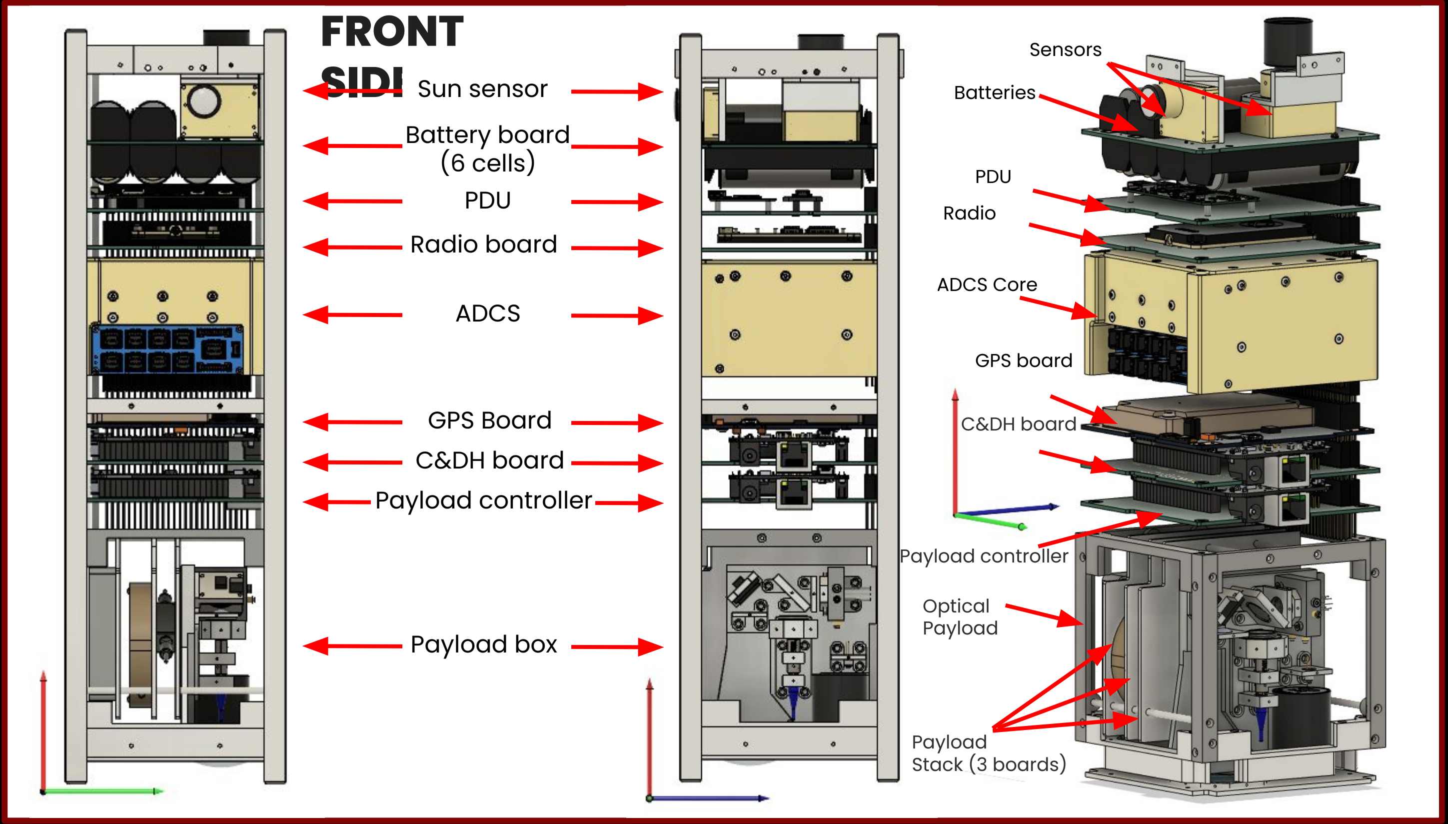


Internal Configuration

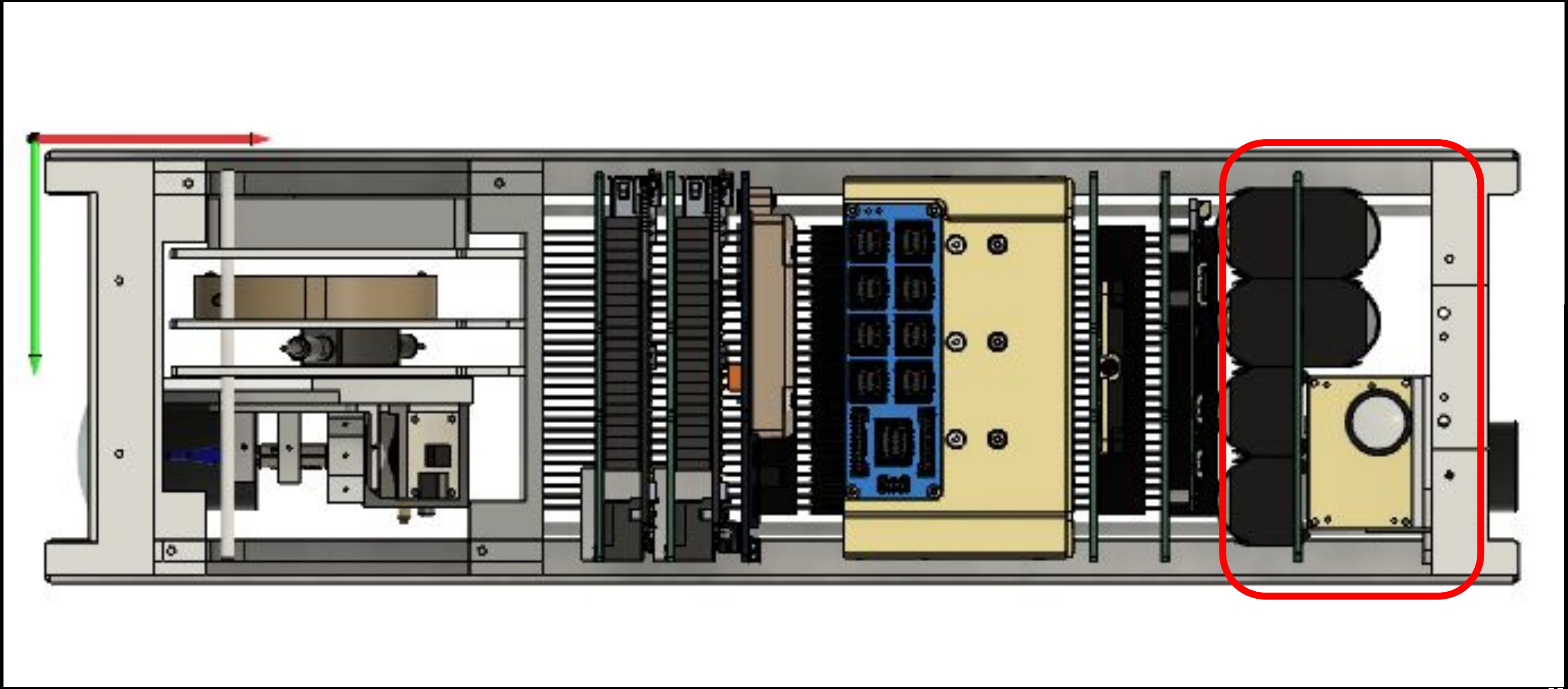
Full stack shown.

All boards connected and spaced with PC-104 standard, SSQ-104 pin connections.

Boards fit on rails with teflon tube spacers (TBD).



Internal Configuration



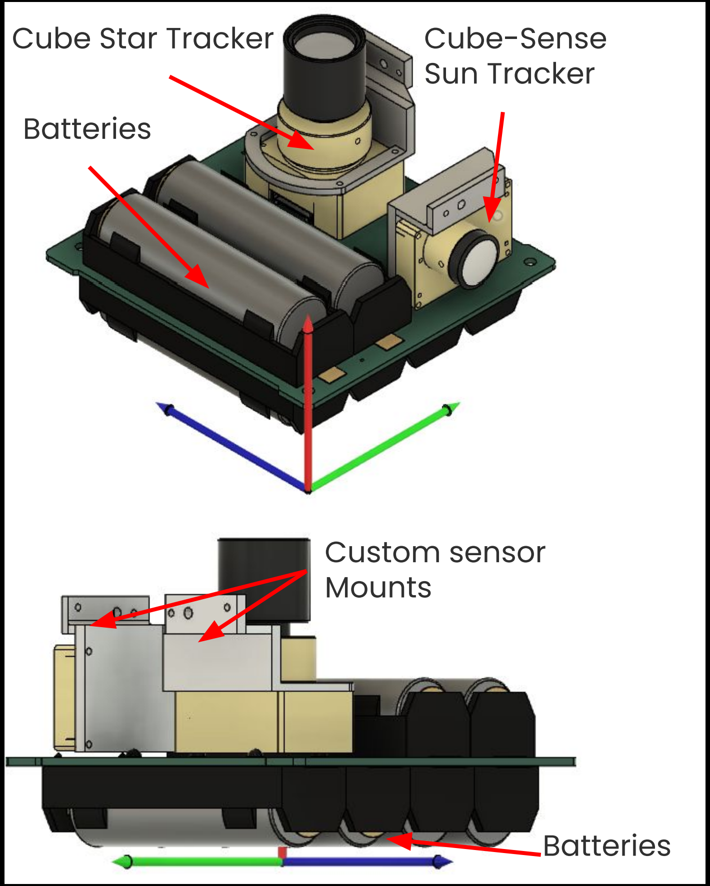
Internal Configuration

Battery Board and Sensors - A Closer Look:

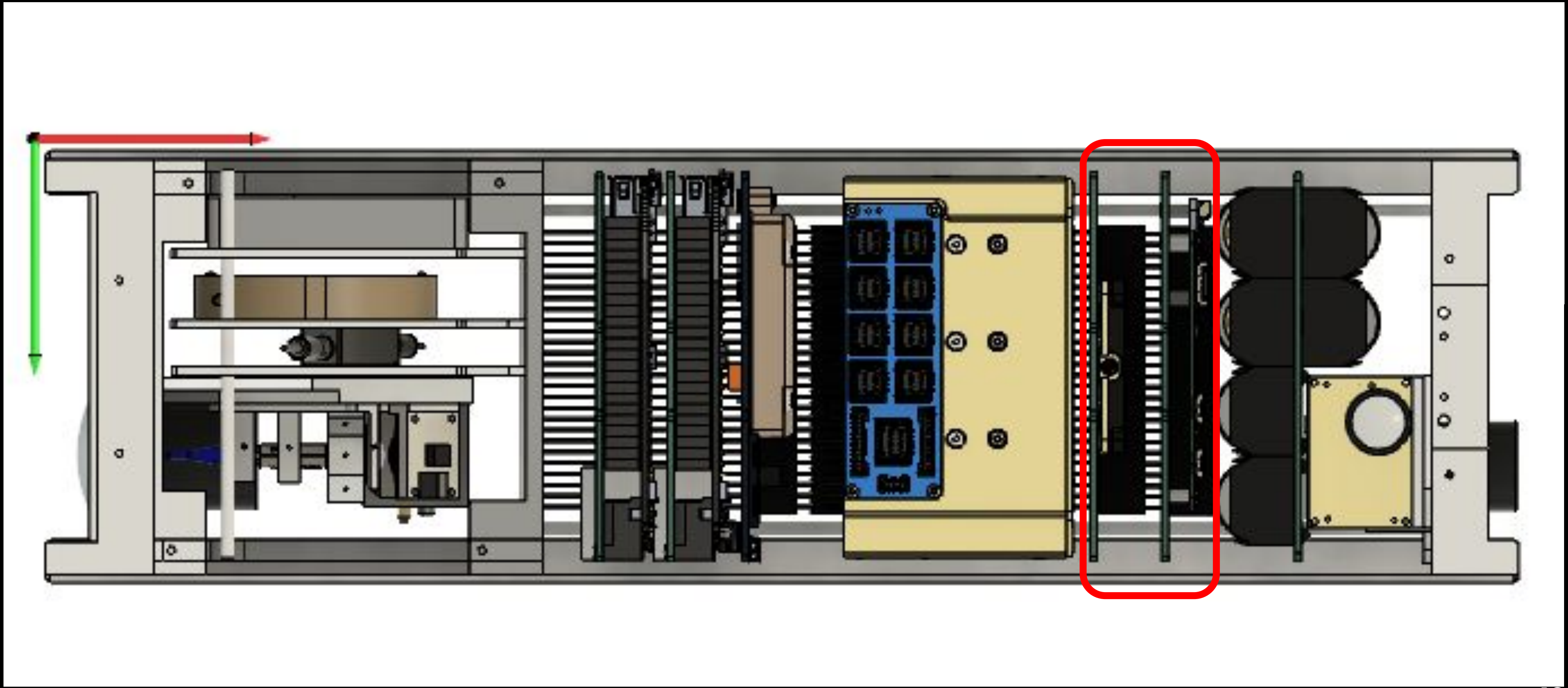
For correct functioning of our sensors, they have to be placed at the top of our stack.

Volume constraints require us to fit them as shown, overlapped with the bounding box of our battery board.

The sensors are mounted to the top element of our frame using custom Aluminum 6061 mounts.



Internal Configuration

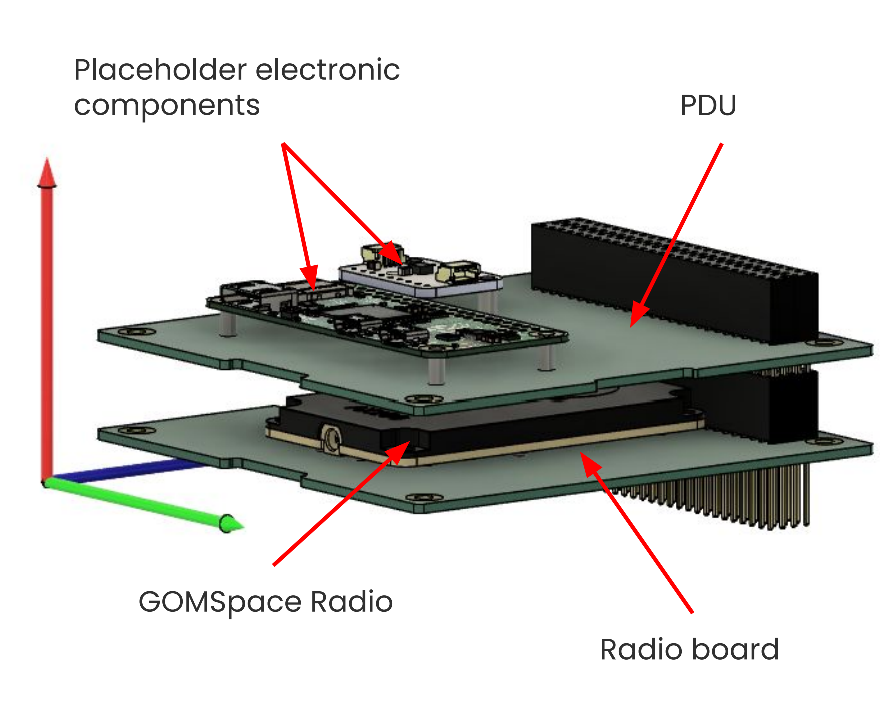


Internal Configuration

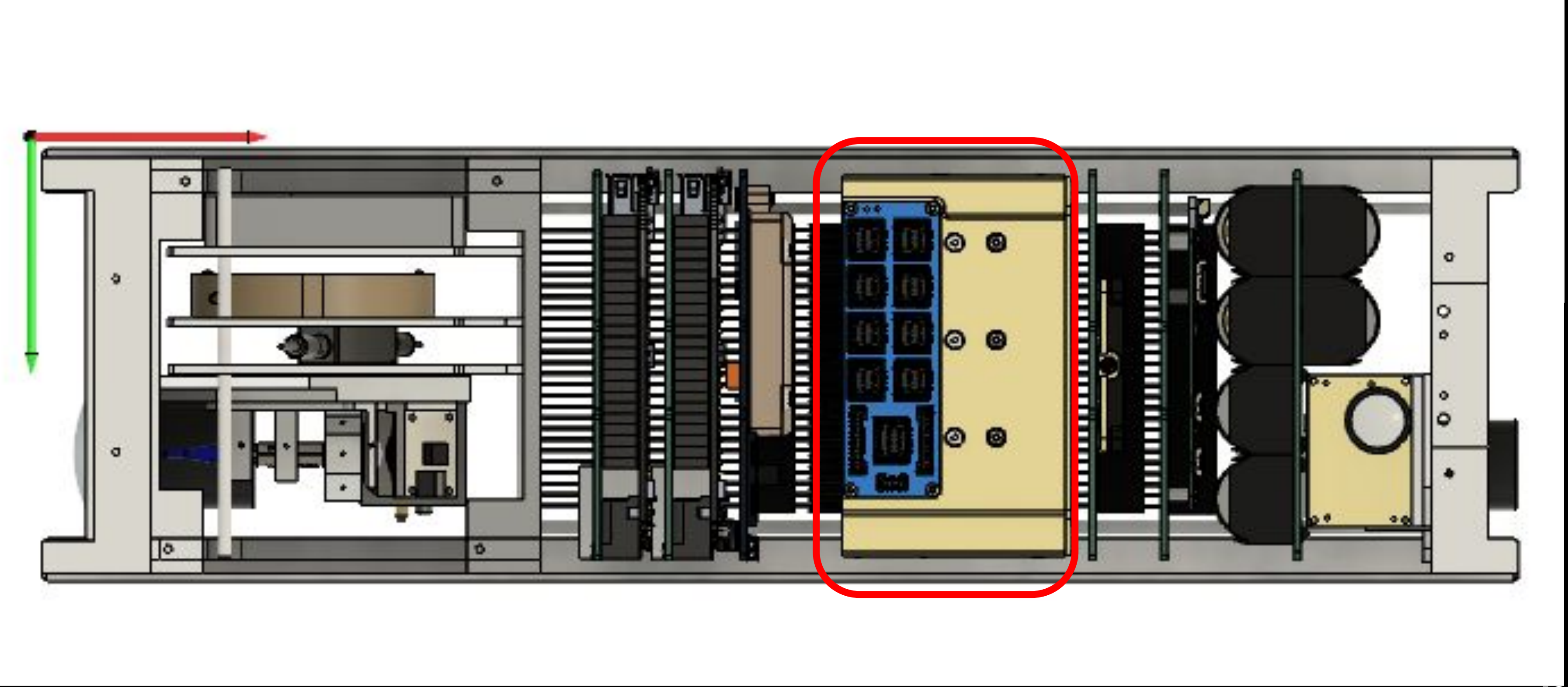
Power Distribution Unit (PDU) and Radio Board:

With Battery board selected as only component with enough volume allowance for sensor placement, PDU positioning below is required. Electronic components are placeholders.

Radio board follows after, which includes the actual GOMspace radio element we are using.



Internal Configuration

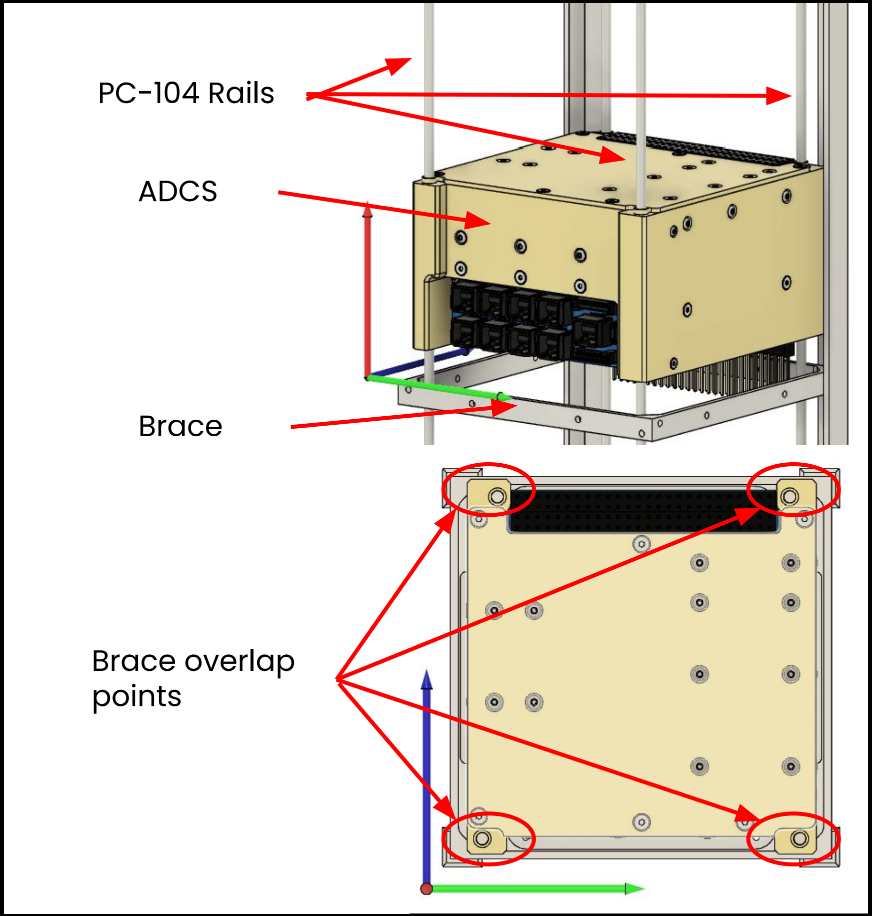


Internal Configuration

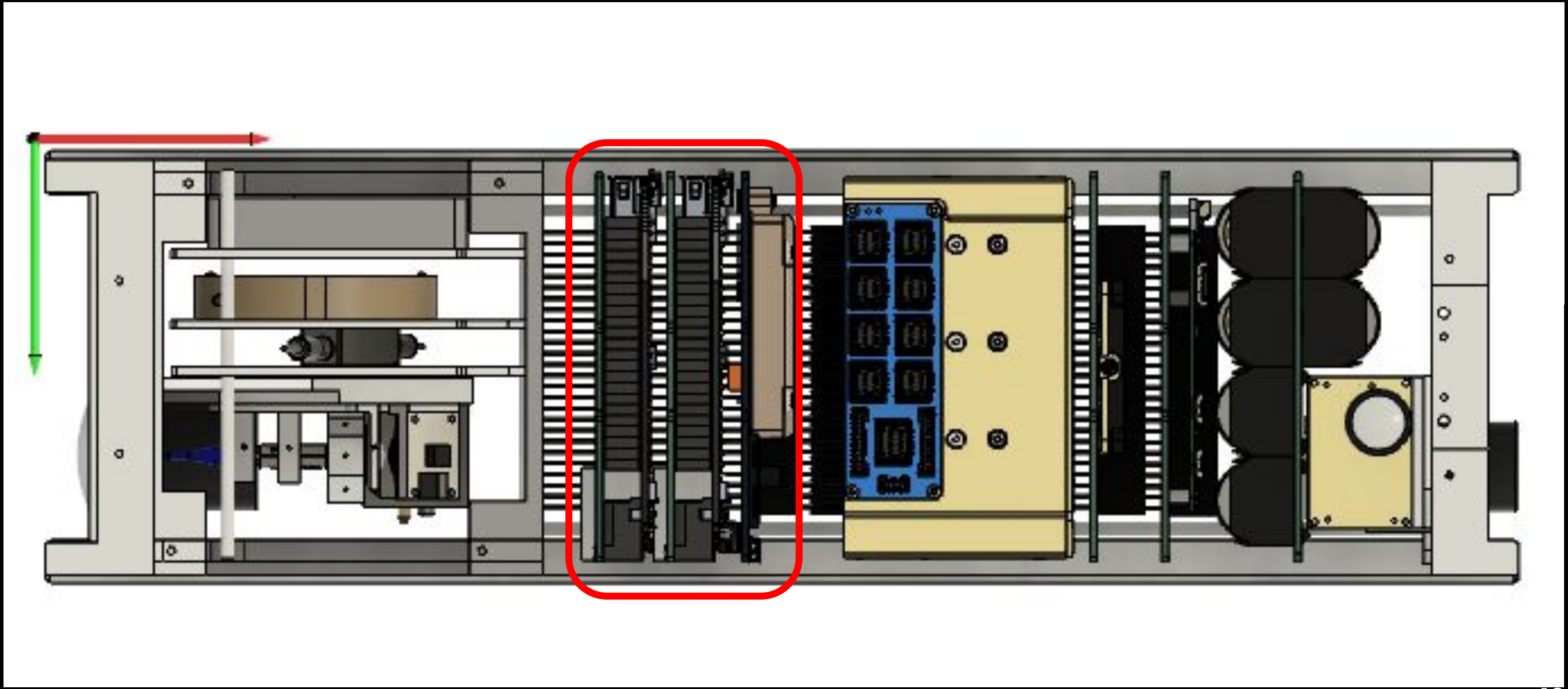
ADCS Placement Consideration:

Our frame has to include a central structural brace, which disallows placement of any PC-104 board within its profile.

This dictates the placement of our ADCS, and creates a vacancy slot in our PCB stack. This vacancy is bridged with an additional header connector.



Internal Configuration

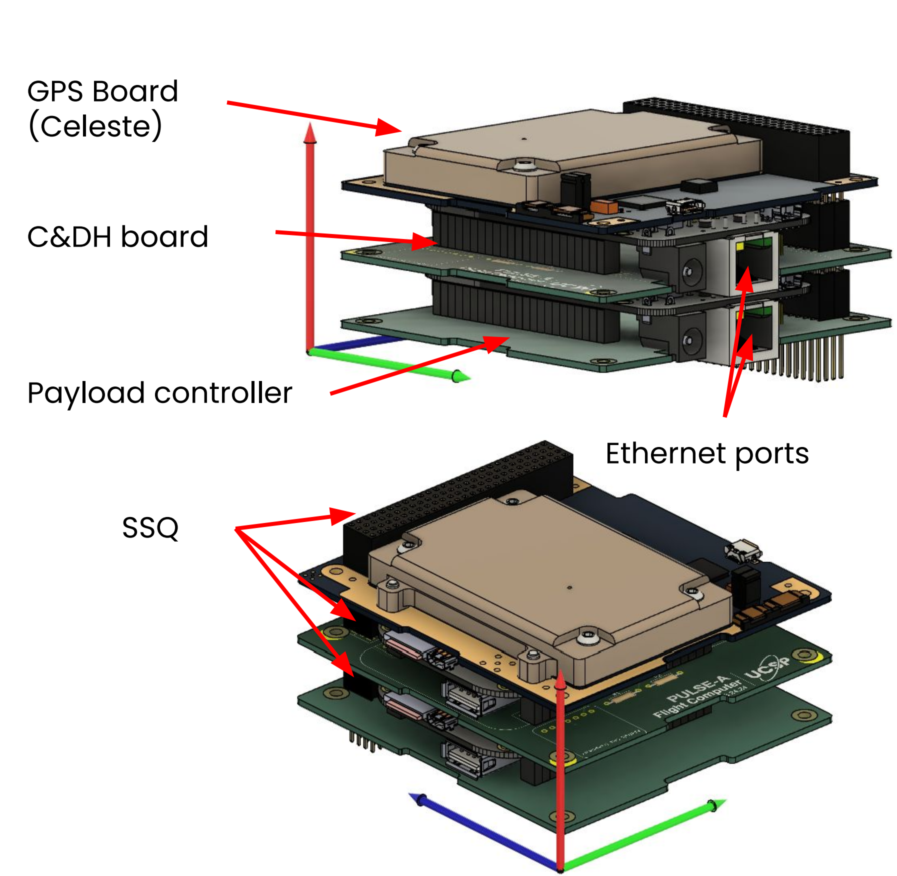


Internal Configuration

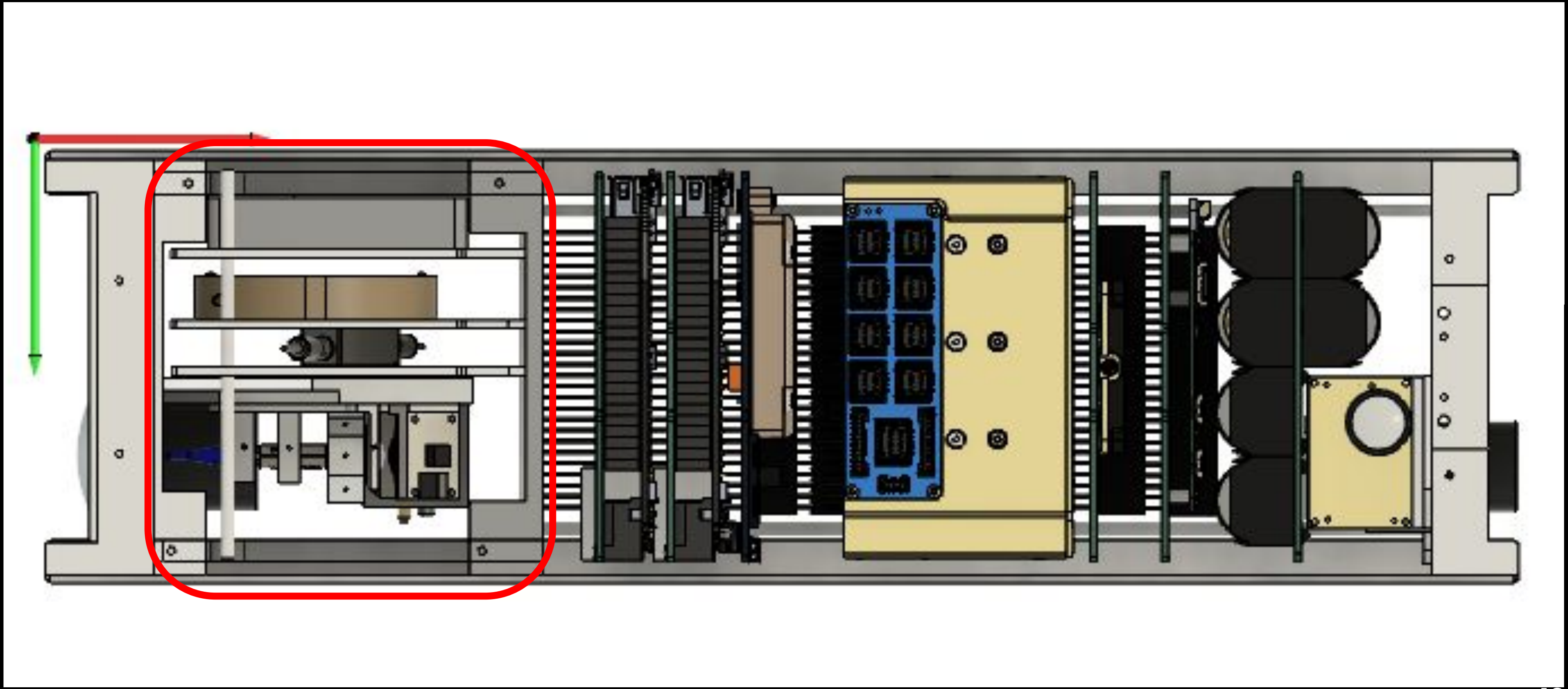
<u>GPS, OBC Board and Payload Controller</u>:

The GPS board uses Celeste, with additional placeholder electronics.

Beaglebone is used on both the C&DH board and payload controller.



Internal Configuration



Internal Configuration - Optical Payload

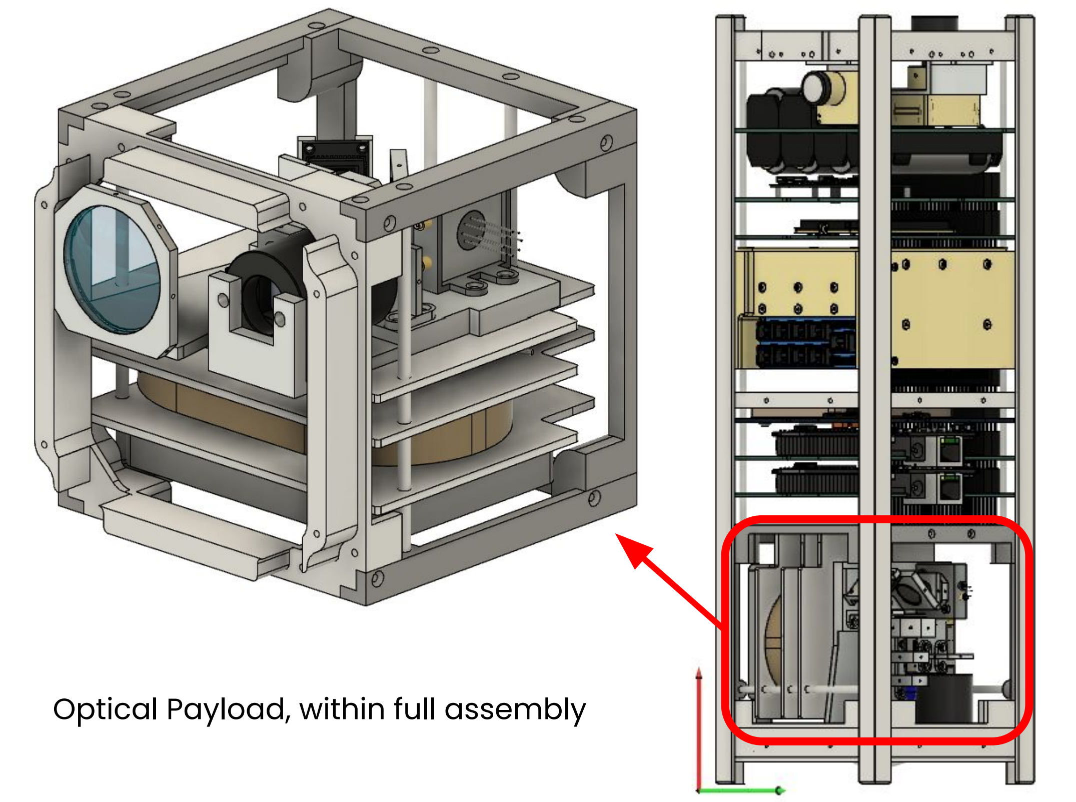
Some Design Considerations

- Tight tolerances required for pointing accuracy maintenance of our satellite means that the design of the optical payload, and especially that of the optomechanical layout, have to be very well thermally regulated
- Resistance to launch environment vibration requirements
- Bend radii of polarization-maintaining fiber require forethought of component layout and significantly influences overall design
- Tight Volume margins within our Bus dimensionally bounds payload to be as compact as possible
- Modularity and ease of assembly inform primarily the design of the Payload Frame, but also lens mounts

Internal Configuration - Optical Payload

The Optical Payload is split into:

- Payload box
  - 2 type A L-beams
  - 2 type B L-beams
  - Top plate
  - Bottom plate
  - 4 poles
  - Shield (not shown)
- Payload Stack
  - Optomechanical layer
  - Generation layer
  - EDFA layer

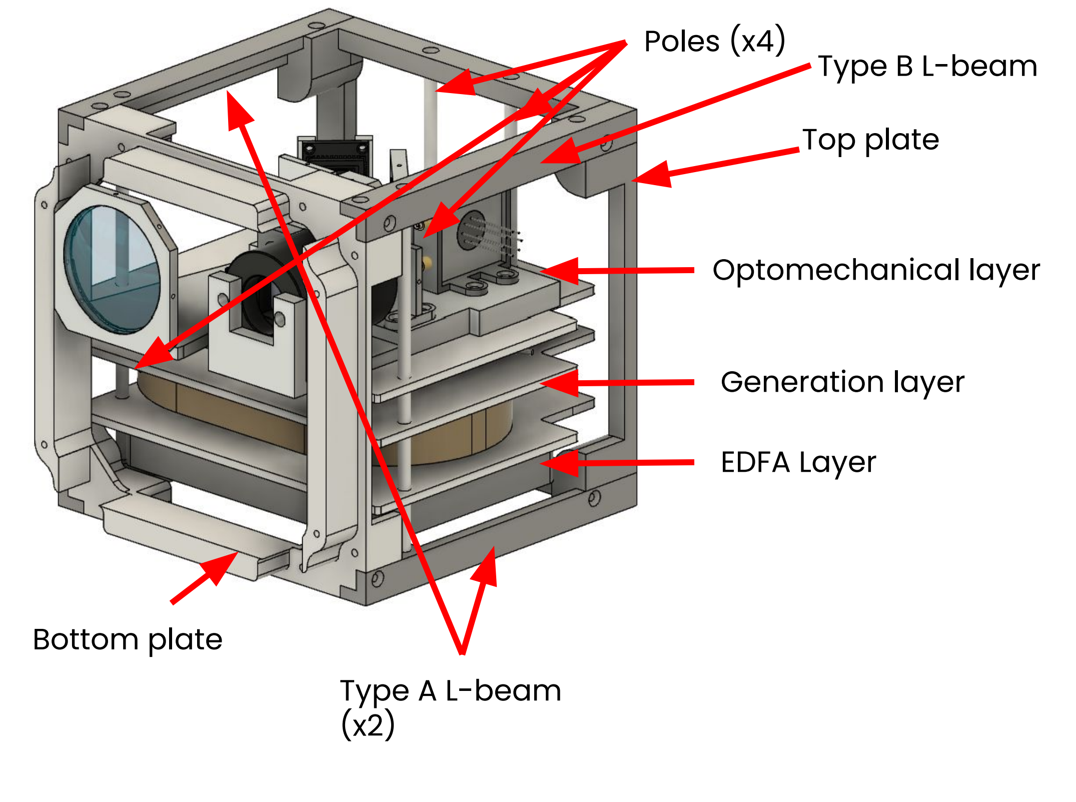


Optical Payload, within full assembly

Internal Configuration - Optical Payload

The Optical Payload is split into:

- Payload box
  - 2 type A L-beams
  - 2 type B L-beams
  - Top plate
  - Bottom plate
  - 4 poles
  - Shield (not shown)
- Payload Stack
  - Optomechanical layer
  - Generation layer
  - EDFA layer



Internal Configuration - Payload Box interface

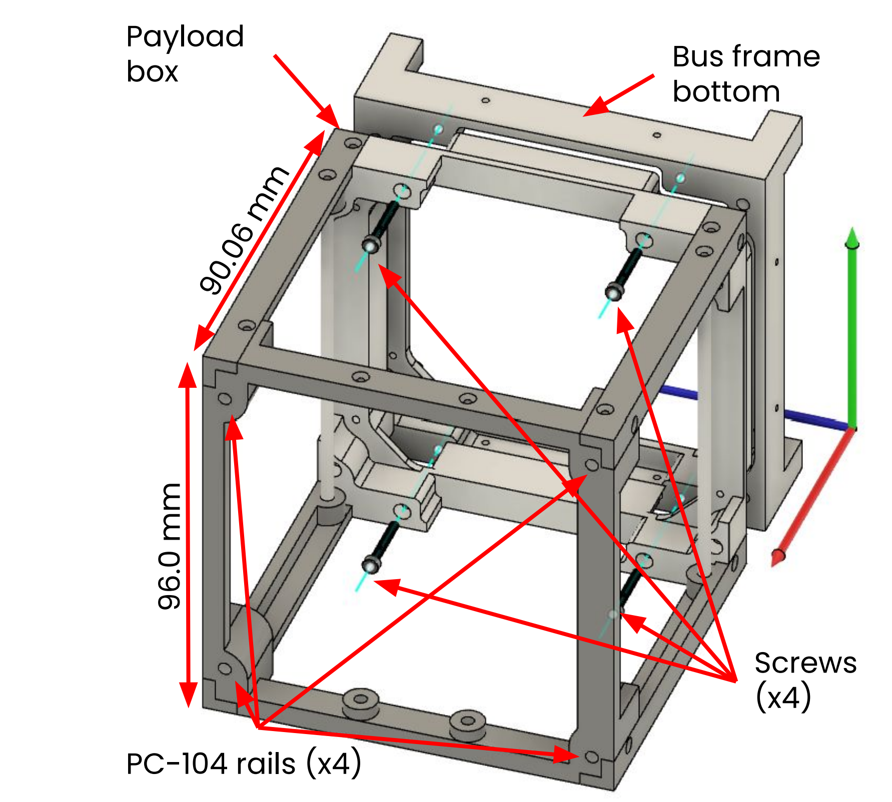
Material: Aluminum 6061

The box has the primary goal of orienting the optomechanical layout in the x-axis and fixing support hardware.

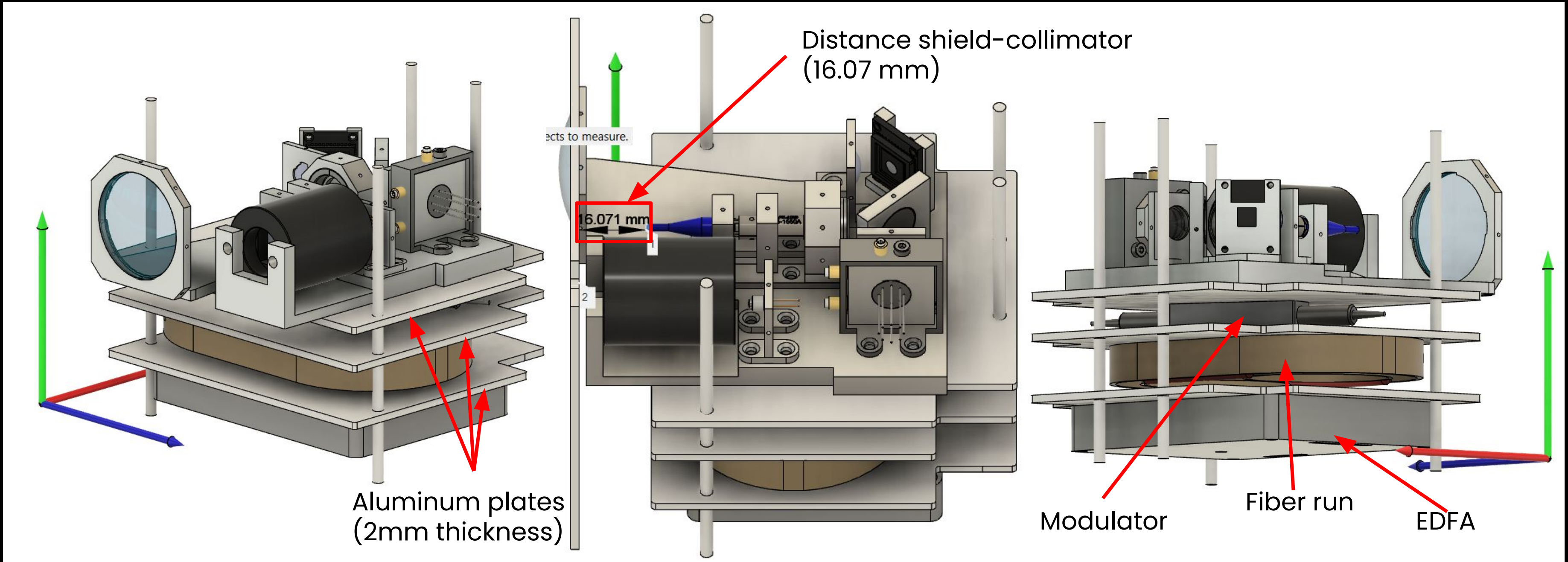
It connects to the bus via 8 screws to the bottom element of the bus sets.

On it’s top, it has 4 holes which are the start of the rails used in the PC-104 standard.

Shield is mounted externally, to the bottom of the Bus frame

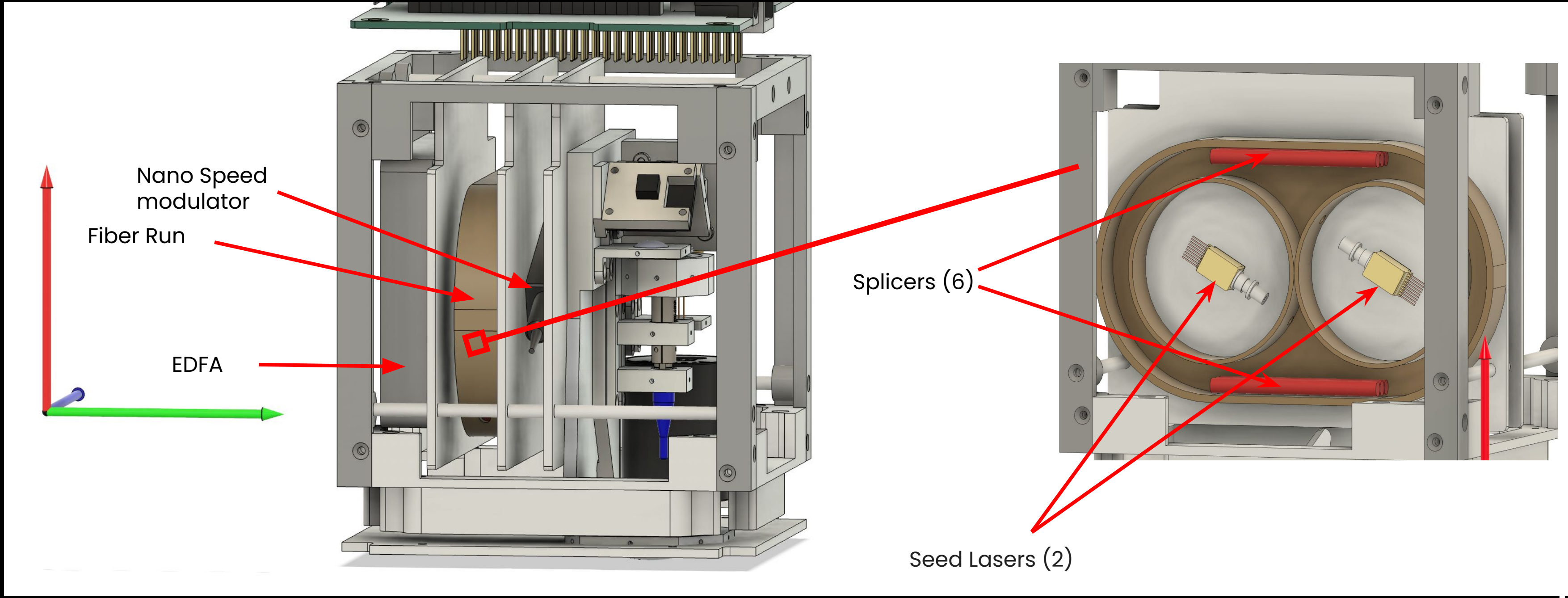


Internal Configuration - Optical Stack



Internal Configuration - Optical Stack

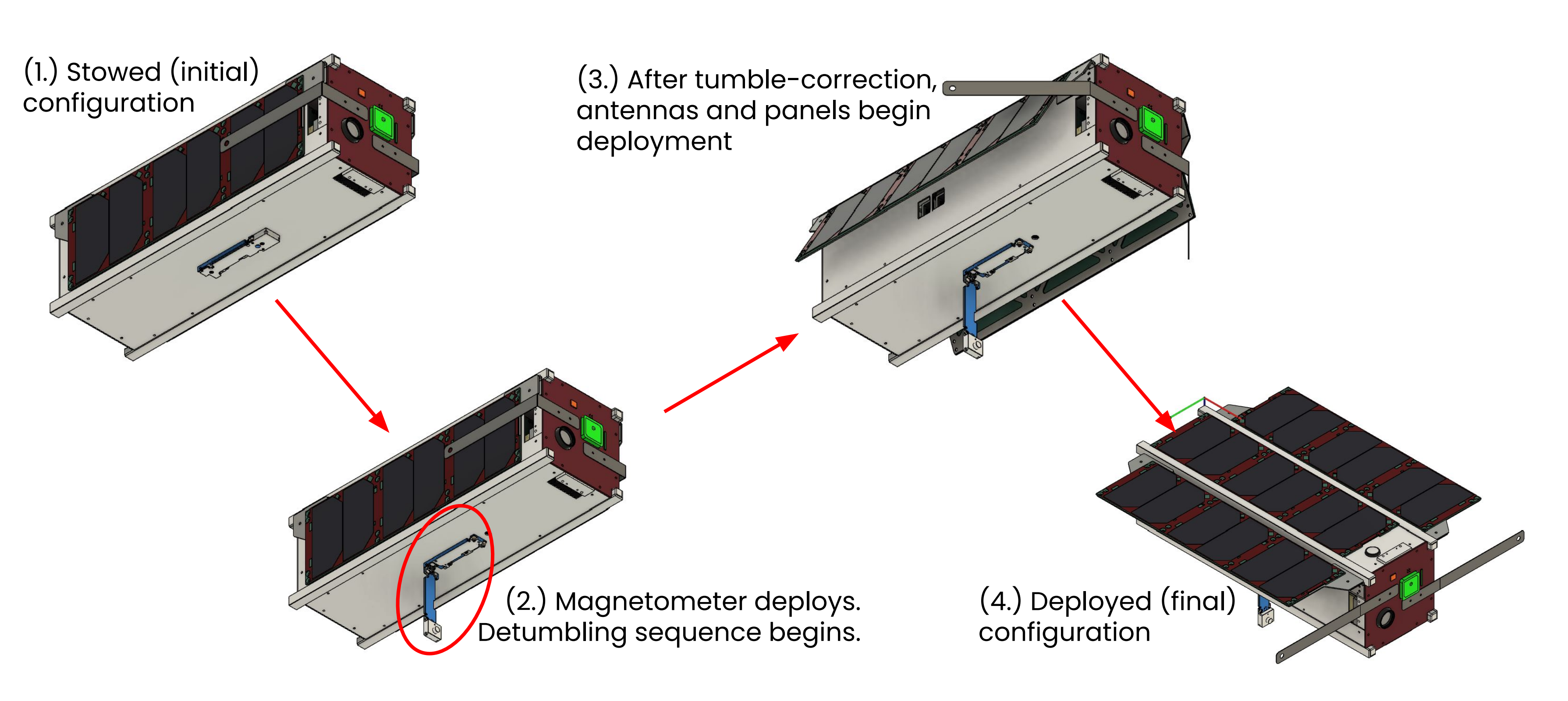
Optical Stack (x-axis pointing)



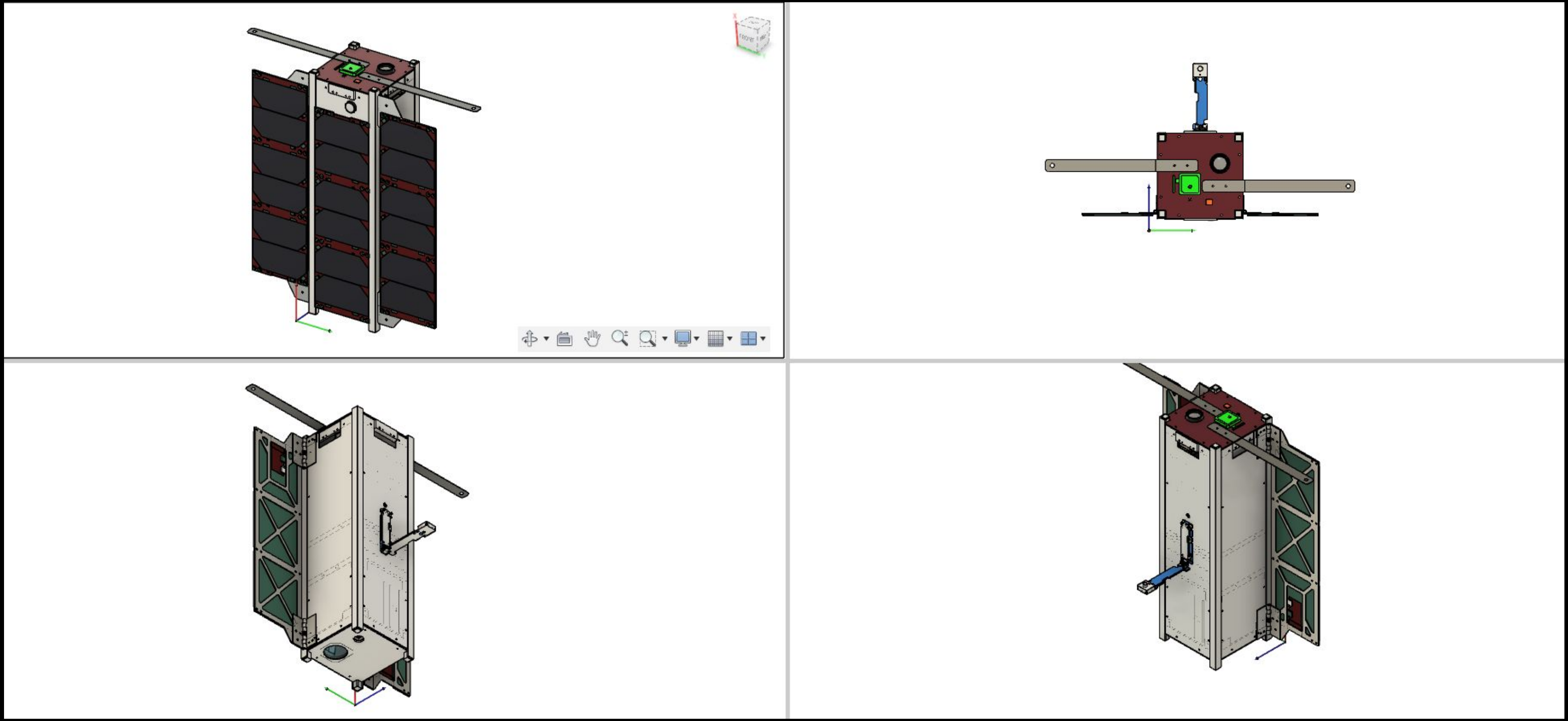
Deployment Sequence

- 2-step: magnetometer, then antennas + solar panels
- Mechanisms:
  - For magnetometer, it is integrated into COTS package. Potential energy stored in spring, also within package
  - For antennas + solar panels, held down using fishing wire, wrapped around the outside of the CubeSat, finally passed through a 2-nichrome burn wire redundant system, at the back of the satellite (not yet modeled). For antennas, potential energy stored in natural elasticity of antenna. For solar panels, hinges with springs are used.

Deployment Sequence



Deployment Sequence- Final configuration



Next Steps for Bus Configuration

- ❏ Optical fiber wiring
  - ❏ Greater consideration towards bend radii and volume
- ❏ Electrical wiring
  - ❏ Greater consideration towards bend radii and volume
- ❏ Prototyping
  - ❏ Though we have prototyped some optical payload mounting layouts, more sophisticated prototyping will be a priority in the coming months.
  - ❏ 3D-printed, as well as metal prototypes. We are actively looking into on-site fabrication, which would give us the opportunity to experiment much more easily

Satellite System Analysis:

Structure and Bus Configuration

Volume Budget

Component	Subsystem	Source	Quantity	Total Volume (cm^3)
Battery Board	Power	CAD	1	239.63
PDU	Power	CAD	1	140.07
Radio Board	Comms	CAD	1	140.07
OBC	C&DH	CAD	1	140.07
Payload Controller	C&DH	CAD	1	140.07
Star Tracker Mount	ADCS	CAD	1	33.32
GPS Board	ADCS	CAD	1	140.07
ADCS Assembly	ADCS	CAD	1	473.08
CubeSense Sun + Mount	ADCS	CAD	1	21.21
CubeStar	ADCS	CAD	1	16.46
Payload Box	Payload	CAD	1	938.00
Baffle PC104 Adapter	Structures	CAD	1	8.56
			Available Volume	2,920.56
			Used Volume	2,430.61
			Margin Volume	489.95
			Margin %	16.78%

Volume Budget

Component	Subsystem	Source	Quantity	Total Volume (cm^3)
Battery Board	Power	CAD	1	239.63
[illegible]	Power	CAD	1	140.07
[illegible]	Comms	CAD	1	140.07
[illegible]	C&DH	CAD	1	140.07
[illegible]	C&DH	CAD	1	140.07
Star Tracker Mount	ADCS	CAD	1	33.32
GPS Board	ADCS	CAD	1	140.07
[illegible]	ADCS	CAD	1	473.08
[illegible]	ADCS	CAD	1	21.21
[illegible]	ADCS	CAD	1	16.46
[illegible]	Payload	CAD	1	938.00
[illegible]	Structures	CAD	1	8.56

Available Volume	2,920.56
Used Volume	2,430.61
Margin Volume	489.95
Margin %	16.78%

Margin with current payload layout is below desired margin at PDR.

However, within Payload bounding box, there is almost 0.5 U of empty space due to layout inefficiencies (leading to almost 1U spacecraft margin).

Mass Budget (1/3)

Component	Subsystem	Mass per unit (g)	Source	Quantity	Net Mass (g)	Margin (%)	Mass Margin (g)	Total Mass with Margin (g)
PDU board	CDH	90	CAD	1	90	25	22.5	112.5
Flight Computer Board	CDH	86	CAD	1	86	25	21.5	107.5
Payload Controller	CDH	86	CAD	1	86	25	21.5	107.5
Gran Systems 3U Modified Frame	Structure	605	Manufacturer Data Sheet	1	605	10	60.5	665.5
Fixed Solar Panel	Structure	75	CAD	1	75	10	7.5	82.5
Deployable Solar Panel	Structure	119	CAD	2	238	30	71.4	309.4
Payload box	Structure	80	CAD	1	80	30	24	104
Payload stack plate	Structure	42	CAD	3	126	30	37.8	163.8
Batteries	Power	50	Manufacturer Data Sheet	6	300	10	30	330
Battery Board	Power	100	CAD	1	100	25	25	125
ADCS (Reference: 3-6U)	ADCS	743	Manufacturer ICD	1	743	10	74.3	817.3
Deployable Magnetometer	ADCS	16	Manufacturer ICD	1	0.016	10	0.0016	0.0176
Star Tracker	ADCS	47	Manufacturer ICD	1	47	10	4.7	51.7
Sun Sensor	ADCS	15	Manufacturer ICD	1	15	10	1.5	16.5
Star Tracker Mount	ADCS	8.309	CAD	1	8.309	30	2.4927	10.8017
Sun Tracker Mount	ADCS	11.195	CAD	1	11.195	30	3.3585	14.5535
Spacemanic Celeste	ADCS	25	Manufacturer Data Sheet	1	25	10	2.5	27.5
Antenna board (Patch, 2x Deployable, PCB)	Comms	74	CAD	1	74	10	7.4	81.4
GOMSpace AX100	Comms	24.5	Manufacturer Data Sheet	1	24.5	10	2.45	26.95
Payload Aluminum Breadboard	Structure	68.83	CAD	1	68.83	30	20.649	89.479
Beam Expander 8mm Lens Mount	Structure	2.22	CAD	1	2.22	30	0.666	2.886
Beacon Laser Mount	Structure	2.294	CAD	1	2.294	30	0.6882	2.9822

Mass Budget (2/3)

Component	Subsystem	Mass per unit (g)	Source	Quantity	Net Mass (g)	Margin (%)	Mass Margin (g)	Total Mass with Margin (g)
Dichroic Mount	Structure	4.354	CAD	1	4.354	30	1.3062	5.6602
9 mm Lens Mount	Structure	2.297	CAD	1	2.297	30	0.6891	2.9861
Quadcell Mount	Structure	22.304	CAD	1	22.304	30	6.6912	28.9952
1/4 Wave and Fiber Collimator Mount	Structure	11.761	CAD	1	11.761	30	3.5283	15.2893
Beam Expander 30mm Lens	Payload	12.24699	Communication with Manufacturer	1	12.24699	10	1.224699	13.471689
Beam Expander 8mm Lens	Payload	4.53592	Communication with Manufacturer	1	4.53592	10	0.453592	4.989512
Quadcell 9mm Lens	Payload	4.989516	Communication with Manufacturer	1	4.989516	10	0.4989516	5.4884676
Prism Pair	Payload	130	Manufacturer Data Sheet	1	130	10	13	143
Payload Filter	Payload	1	Manufacturer Data Sheet	3	3	10	0.3	3.3
Payload Quadcell	Payload	2	Estimate	1	2	10	0.2	2.2
Payload Circuitry	Payload	90	CAD	1	90	10	9	99
Payload FSM	Payload	150	Rough	1	150	10	15	165
Payload FSM Driver	Payload	24	Estimate	1	24	10	2.4	26.4
Payload Modulator Driver	Payload	24	Estimate	1	24	10	2.4	26.4
Payload EDFA	Payload	100	Manufacturer Data Sheet	1	100	10	10	110
Payload Modulator	Payload	200	Estimate	1	200	10	20	220
Payload FPGA	Payload	10	Estimate	1	10	10	1	11
Payload Beacon Laser	Payload	0.3	Manufacturer Data Sheet	1	0.3	10	0.03	0.33
Beacon Laser Collimator Lens	Payload	0.01	Manufacturer Data Sheet	1	0.01	30	0.003	0.013
Payload Seed Laser	Payload	0.3	Manufacturer Data Sheet	2	0.6	10	0.06	0.66
Payload Quarter Wave Plate	Payload	4	Manufacturer Data Sheet	1	4	10	0.4	4.4
Payload Beam Splitter	Payload	1	Manufacturer Data Sheet	1	1	10	0.1	1.1

Mass Budget (3/3)

Component	Subsystem	Mass per unit (g)	Source	Quantity	Net Mass (g)	Margin (%)	Mass Margin (g)	Total Mass with Margin (g)
SSQ	Payload	12	Estimate	1	12	30	3.6	15.6
Payload Collimator	Payload	5.6	Manufacturer Data Sheet	1	5.6	10	0.56	6.16

Total Mass (g):	3626.362426
Total Margin (g):	534.8510426
Total Mass with Margin (g):	4161.213469

Total mass with margin falls within allowable mass budget

Margins Assigned:
- ❑ 10% for COTS Unchanged
- ❑ 25% for custom PCBs
- ❑ 30% for custom structural components

Satellite System Analysis:

Thermal Performance

Thermal Analysis Introduction

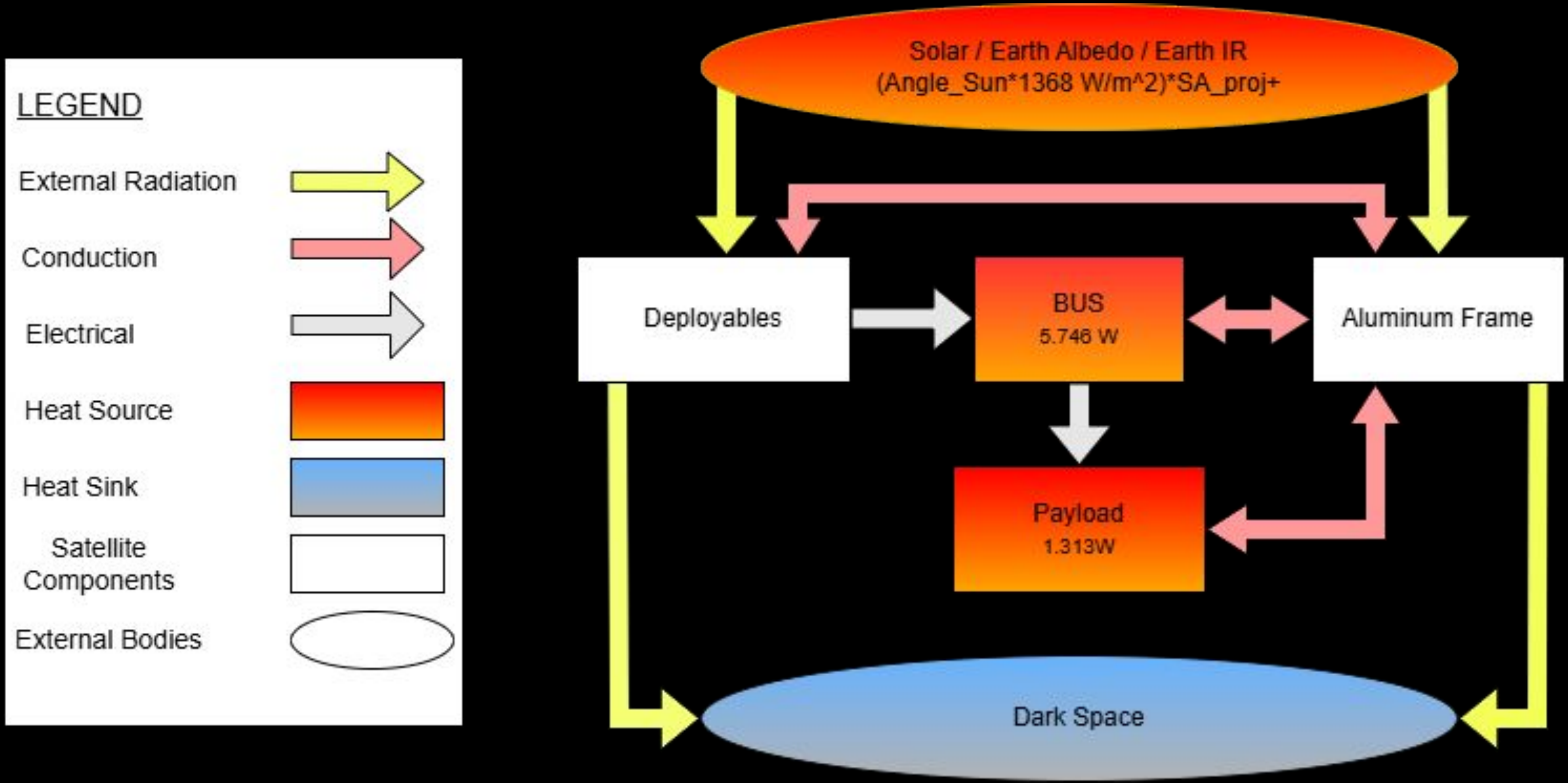
- Goal: Assess the thermal response of various systems exposed to the ISS orbital environment. This analysis informs critical decisions about material selection, thermal strap placement and insulation requirements for optimal performance and longevity.
- Design Process: Using Ansys Thermal Desktop to model designs made by the Structures Department, including the outer frame, the payload area, ADCS, C&DH, Power, and Comms, while using appropriate material approximations.
- Results: Using Thermal Desktop we have developed graphs depicting how different components of the satellite experience temperature change as they orbit the Earth. These results were checked via comparison to literature data.

Thermal Driving Requirements

ID	Requirement	Notes	Parent	Verification
THRM-01	The Thermal Subsystem shall maintain components within their survival temperature ranges during Satellite operations.		SAT-05	Analysis
THRM-02	The Thermal Subsystem shall maintain all operating components within their operational temperature ranges.	Due to nearly constant operations of EPS, CDH, ADCS, and Comms., components in these subsystems must always be within their operational temperature ranges.	SAT-05	Analysis

Legend: Compliant Compliant by CDR Compliant by TRR Compliant By FRR

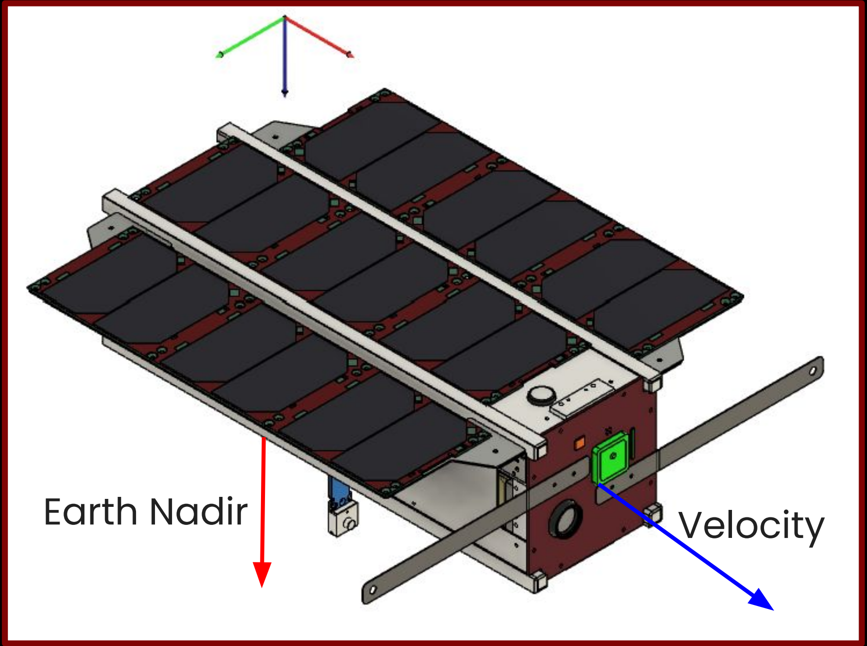
Thermal Model Overview



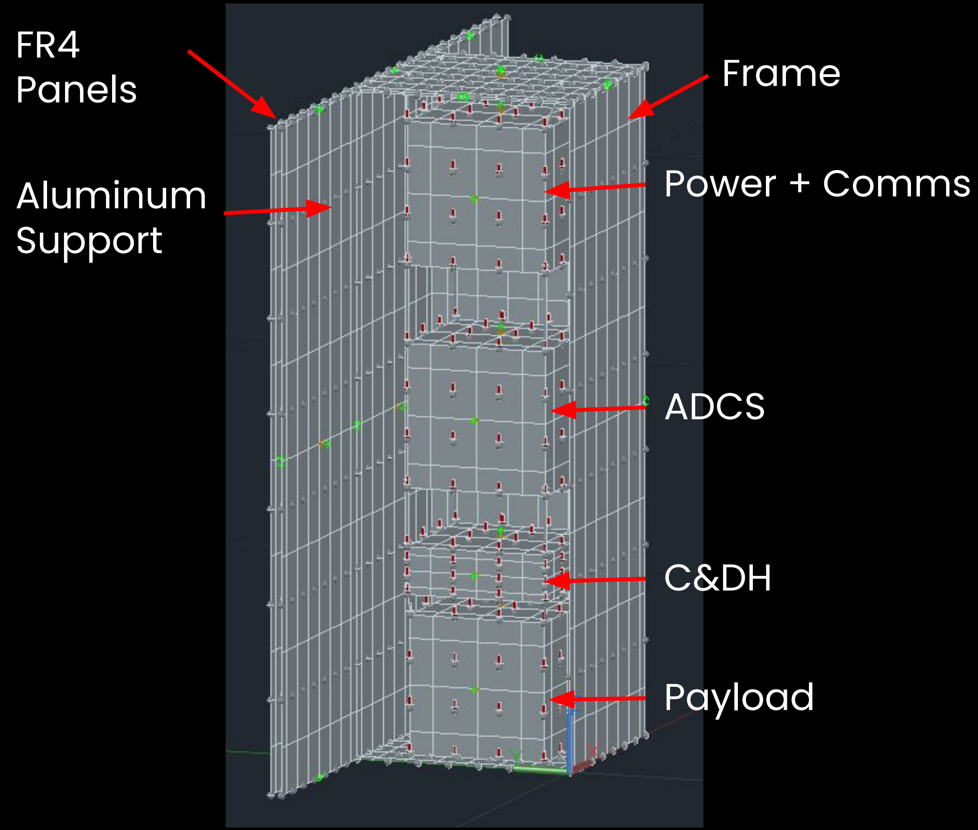
Orbital Heating Simulation Parameters

Thermal Desktop model configured with the Zarya orbit parameters. Modeled to have z axis Earth Nadir, with solar panels facing away from Earth's surface.

- Inclination: 51.6416
- Eccentricity: 0.00072
- RA ascension: 18.0268
- Arg. Perigee: 233.0708
- Solar Flux: 1354 W/m^2
- Albedo: 0.35
- IR Planetshine: 250K



Internal Applied Heat Loads



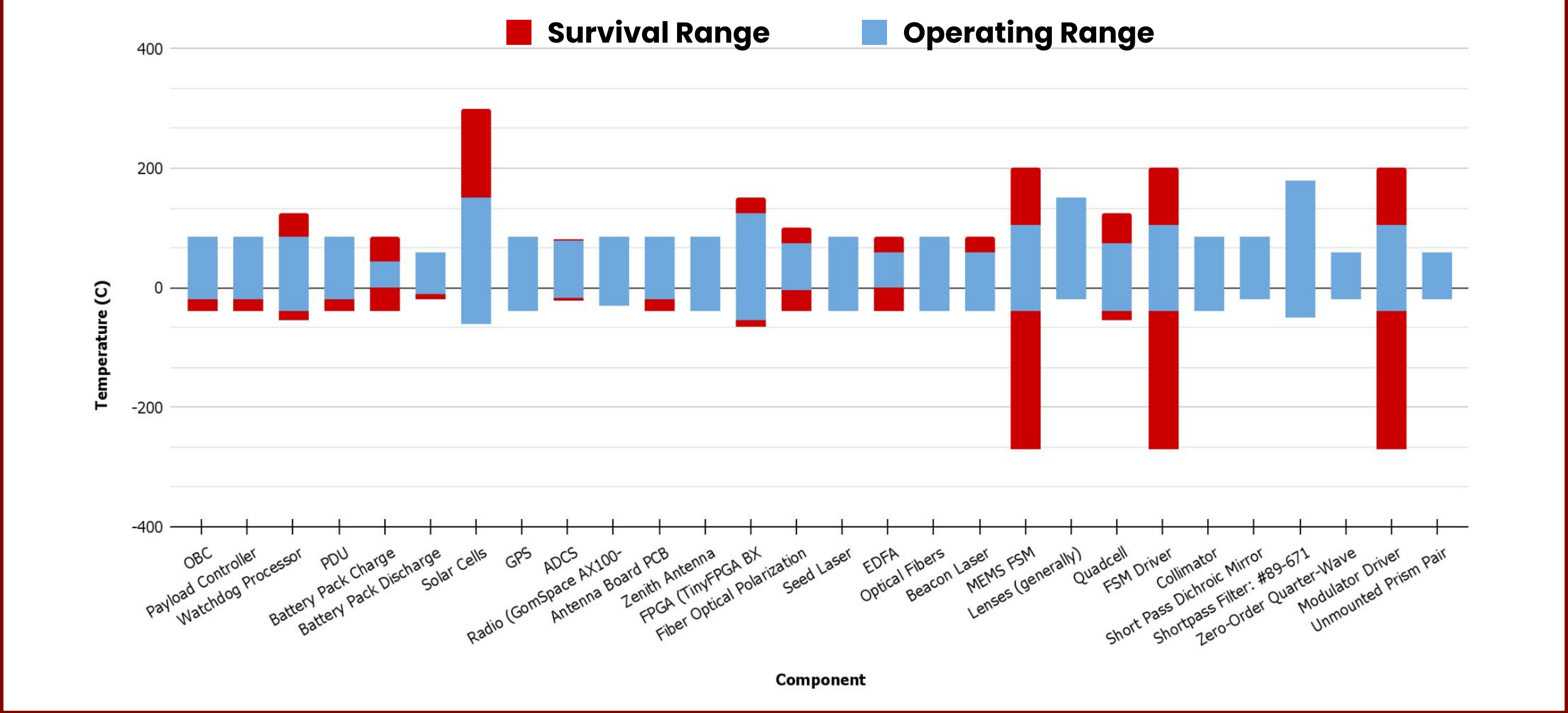
External Surface	Absorptivity	Emissivity
Frame	0.14	0.84
Solar Panels	0.92	0.85
Panel Support	0.14	0.84
System	Transmit	Nominal
Payload	15.71W	0W
ADCS	1.49W	1.46W
C&DH	3.66W	3.66W
Power + Comms	3.43W	3.42W

Internal systems grouped for simplicity of model. For this reason, they may not perfectly reflect CAD model. Internal power data modeled from peaks.

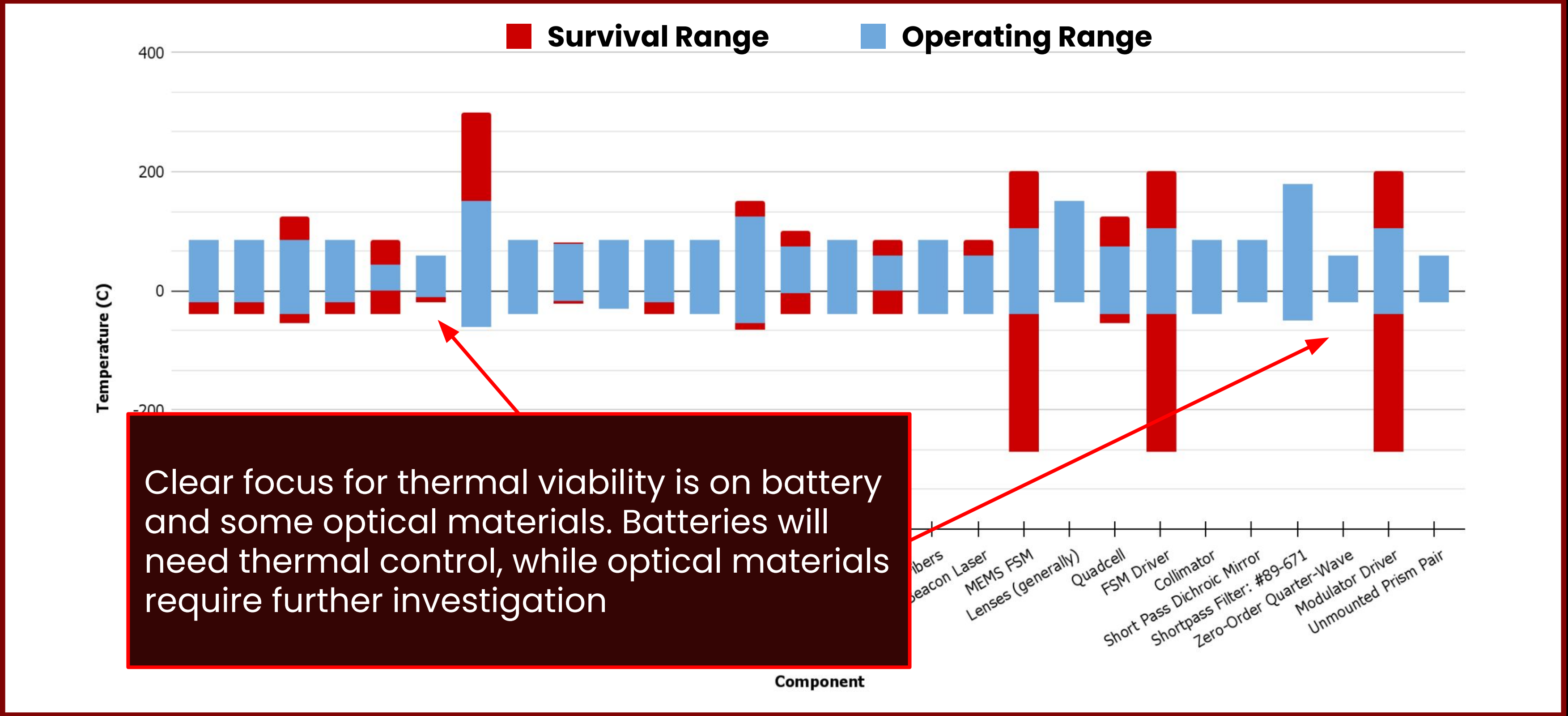
Internal Heat Loads and External Surface Properties

- External Bus Surface:
  - Anodized Aluminum 6061 T6. Absorptivity of 0.07, emissivity of 0.84 with an a/e ratio of 0.17.
- External Solar Panel Surface:
  - Panels made of FR4. Absorptivity of 0.85, emissivity of 0.92, a/e of 1.08
  - Supported by 6061 Aluminum structure
- Transmission Orbit:
  - Orbit average power of 10.29 W, peak power of 24.29 W. Greatest loads are the EDFA - 9.955 W.
  - Peak power applied for 10 minutes every orbit.
- Nominal Orbit:
  - Orbit Average Power of 8.54 W.
  - Data not covered in depth in presentation with exception of Payload as results do not vary greatly for other systems.

Component Temperature Limits



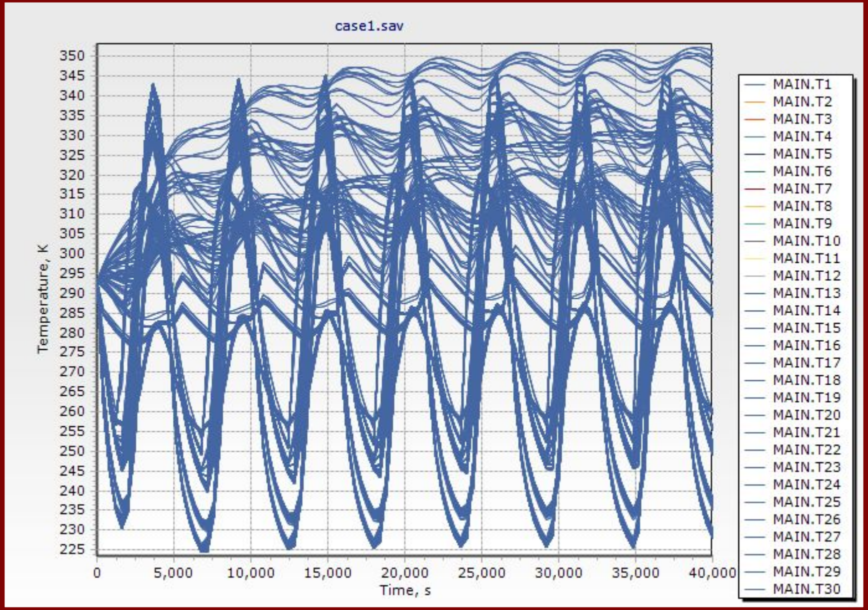
Component Temperature Limits

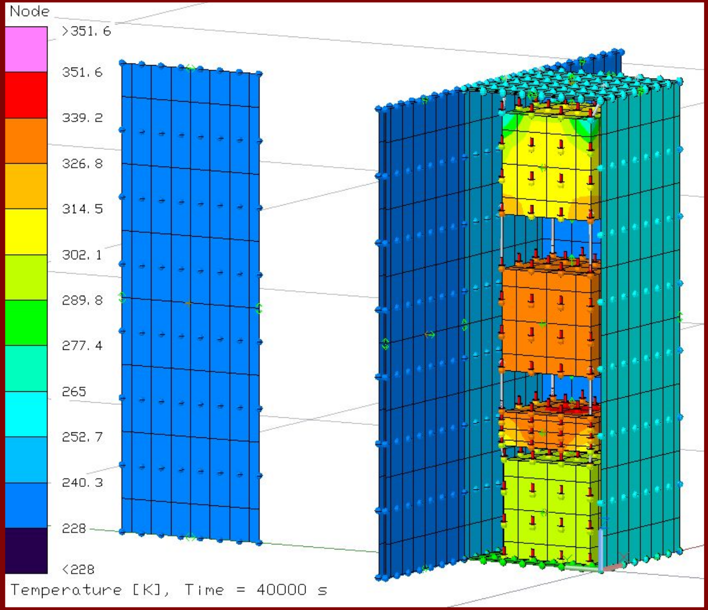


Clear focus for thermal viability is on battery and some optical materials. Batteries will need thermal control, while optical materials require further investigation

Transmission Orbit

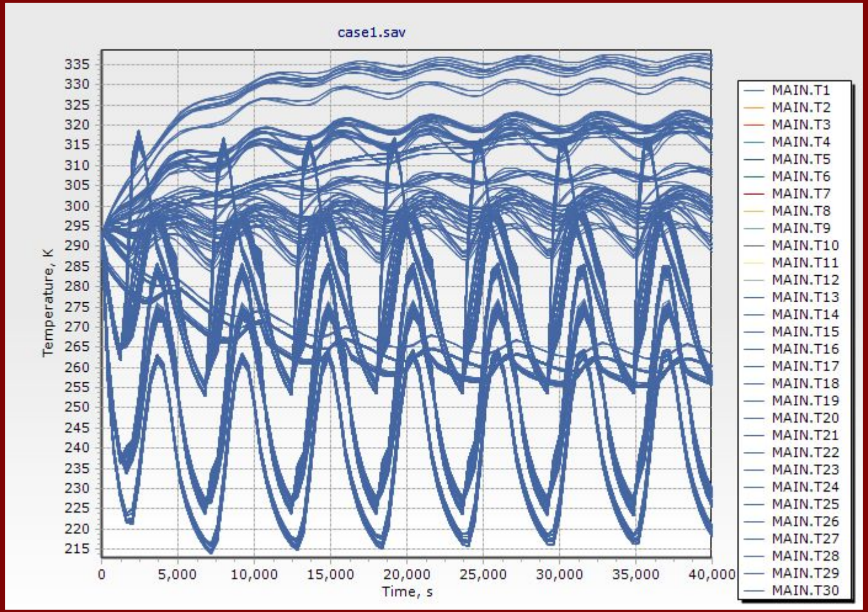
40,000s transmit simulation
Max temp: 350 K
Min temp: 225 K

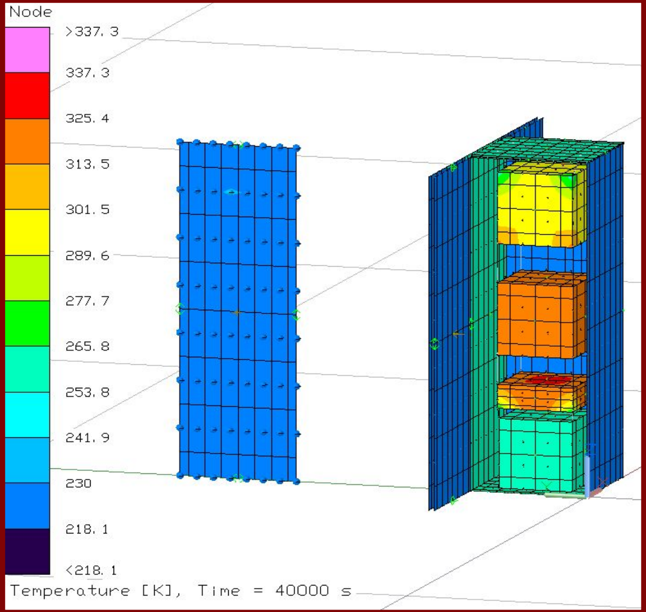




Nominal Orbit

40,000s transmit simulation
Max temp: 337 K
Min temp: 215 K





ADCS Data (Transmission)

Model Design

- Thermally coupled to C&DH and Power + Comms.
- Node to node contacts to simulate rails. Aluminum conductivity.
- Polished Aluminum a/e = 3.0.

Temperature Range

- Max: 332 K
- Min: 293 K

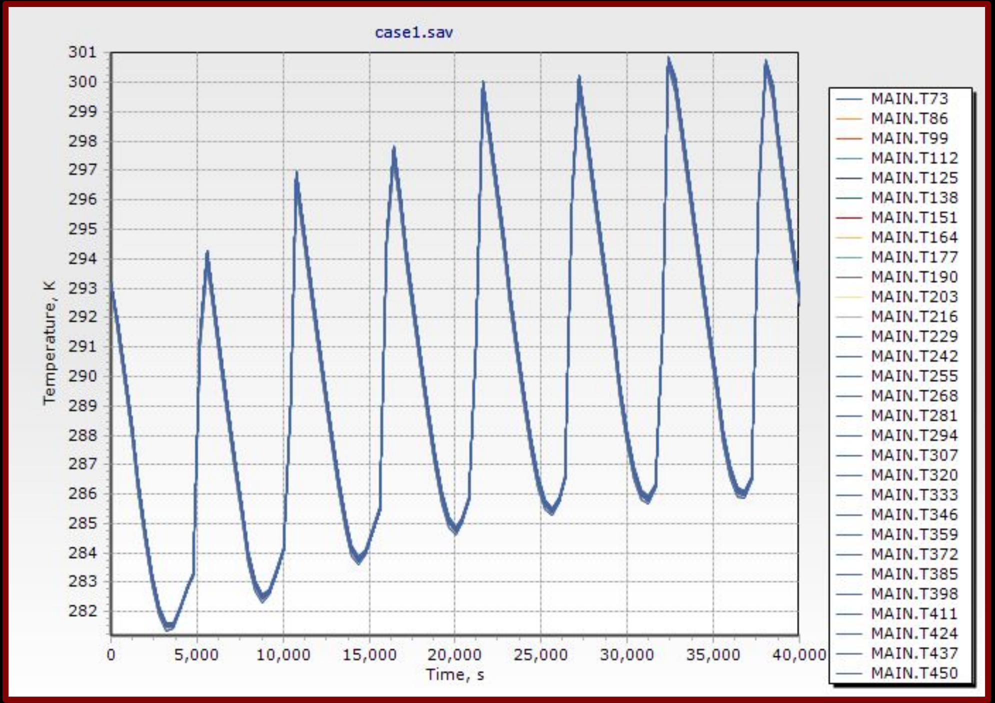
Payload Data (Transmission)

Model Design

- Thermally coupled to Frame and C&DH
- Node to node contacts to simulate rails. Aluminum conductivity.
- Polished Aluminum a/e = 3.0.

Temperature Range

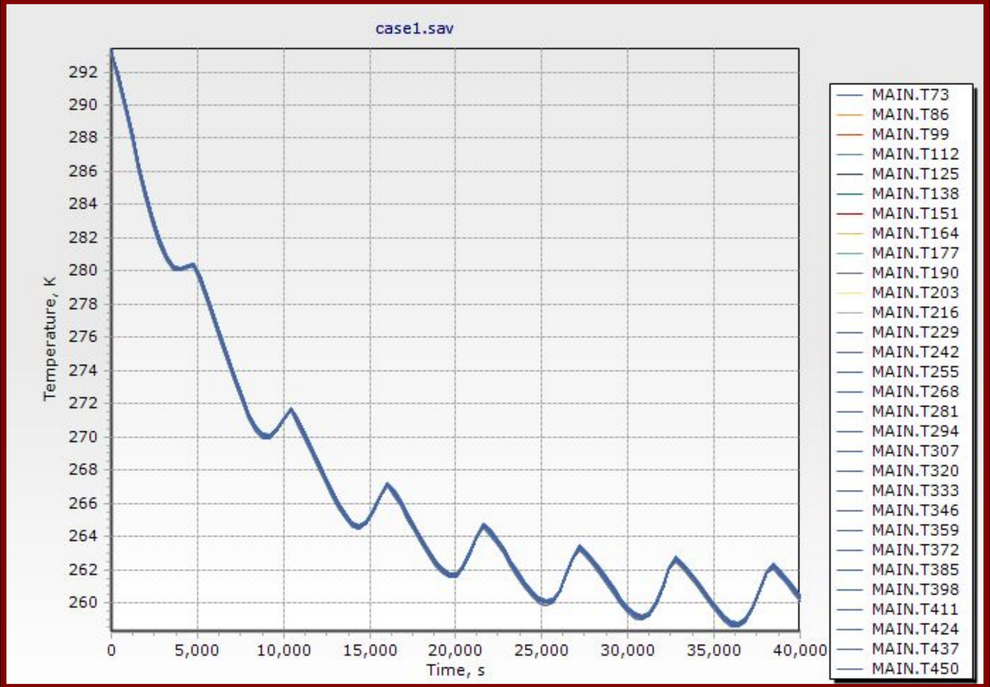
- Max temp: 301 K
- Min temp: 281 K



Payload Data (Nominal)

Model Design
- Thermally coupled to Frame and C&DH
- Node to node contacts to simulate rails. Aluminum conductivity.
- Polished Aluminum a/e = 3.0.

Temperature Range
- Max temp: 293 K
- Min temp: 258 K



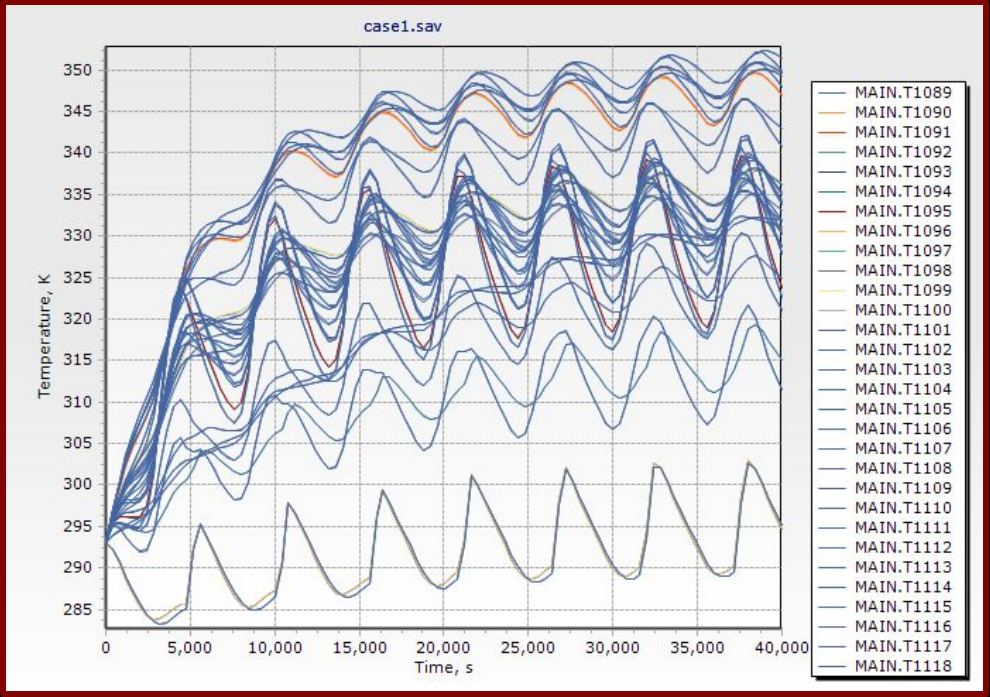
C&DH Data (Transmission)

Model Design

- ❑ Thermally coupled to Payload and ADCS
- ❑ Node to node contacts to simulate rails. Aluminum conductivity.
- ❑ FR4 a/e = 1.08.

Temperature Range

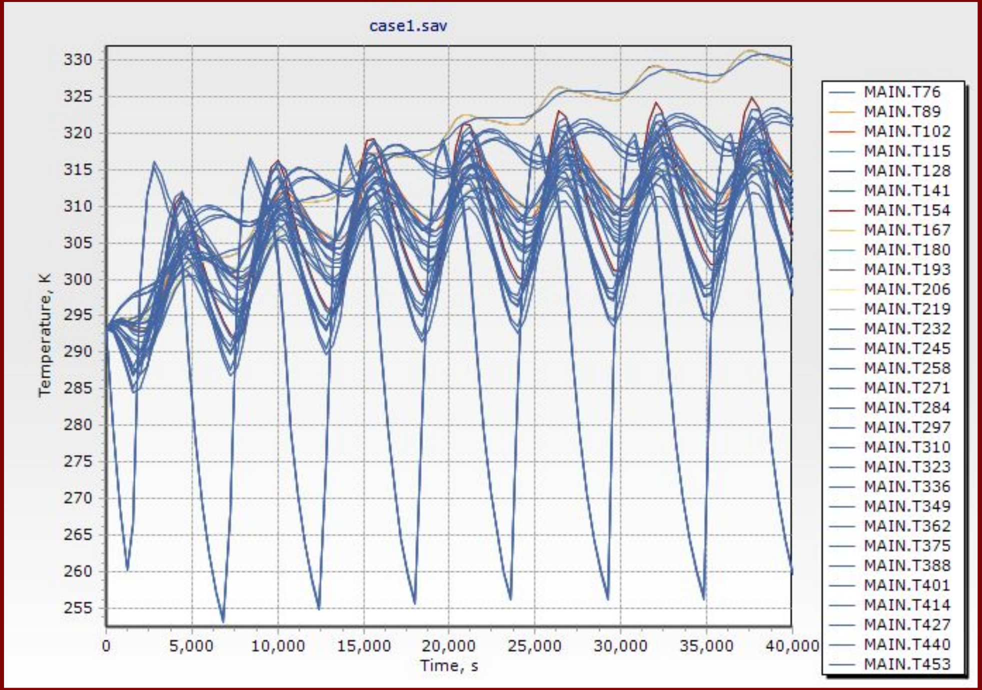
- ❑ Max temp: 350 K
- ❑ Min temp: 283 K



Power + Comms (Transmission)

Model Design
- ❏ Thermally coupled to Frame and ADCS
- ❏ Node to node contacts to simulate rails. Aluminum conductivity.
- ❏ FR4 a/e = 1.08.

Temperature Range
- ❏ Max temp: 330 K
- ❏ Min temp: 255 K



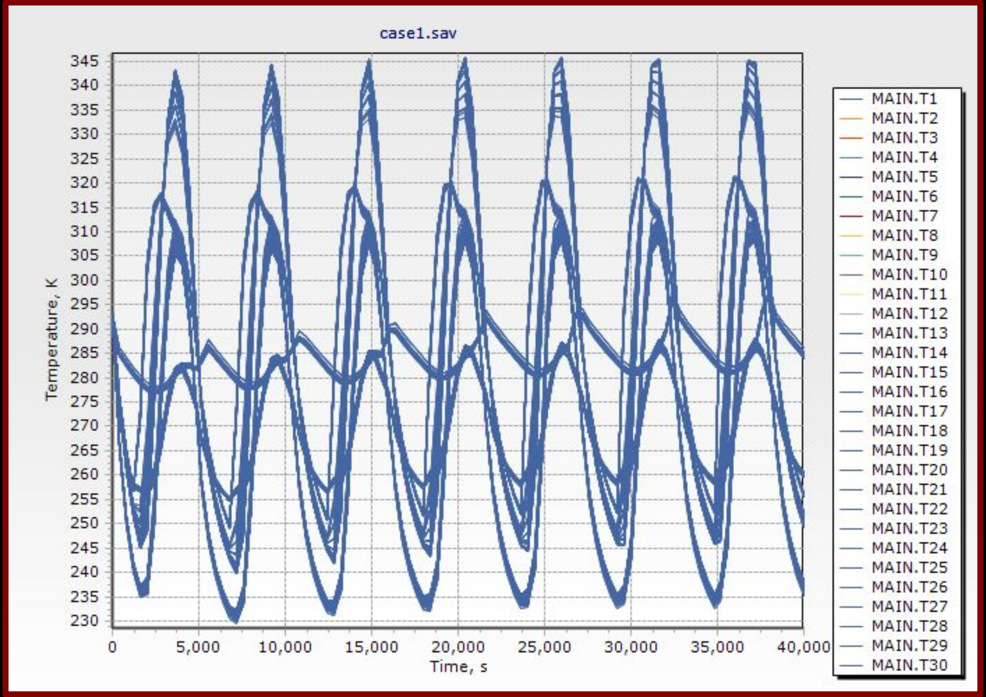
Frame (Transmission)

Model Design

- Thermally couple to Panels, Payload, and Power + Comms
- 1mm edge contacts, aluminum 6061.
- Properties of anodized aluminum 6061, a/e = .167

Temperature Range

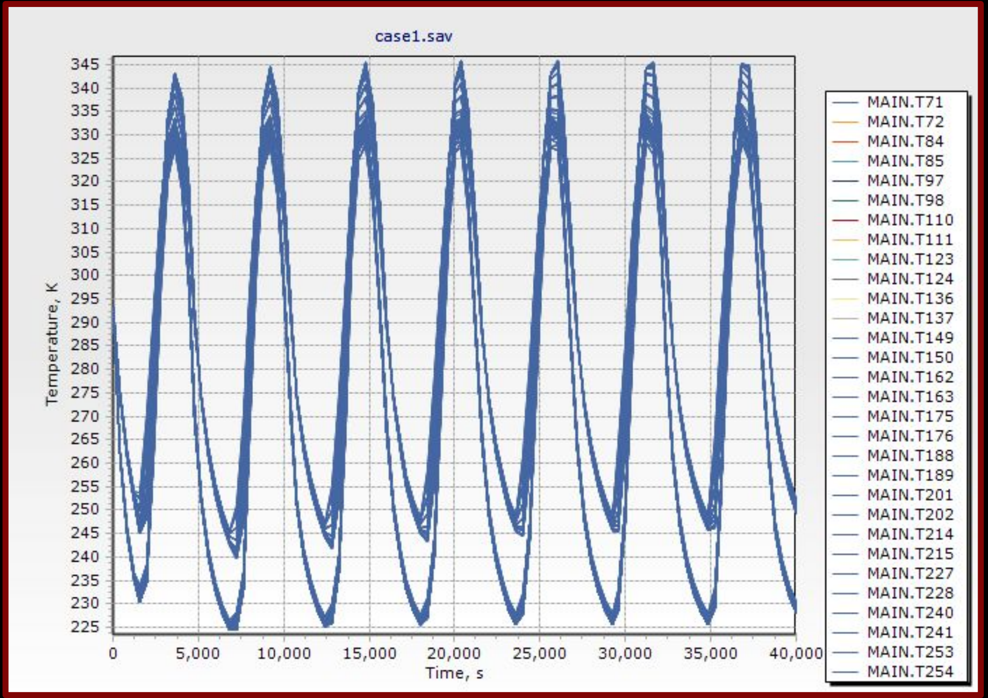
- 230 K
- 345 K



Solar Panels (Transmission)

Model Design
- ❏ Thermally coupled to Frame
- ❏ 1mm aluminum edge and face contacts.
- ❏ Properties of FR4 on face, a/e = 1.08, anodized aluminum on back, a/e = .167.

Temperature Range
- ❏ Max temp: 345 K
- ❏ Min temp: 225 K



ADCS Temperature Considerations (Transmission)

Temperature Issues
- ❏ Maximum does not reach equilibrium

Potential issues with model
- ❏ Conduction coefficient between ADCS and railes rough estimate.

Solution
- ❏ Improvement of model detail and accuracy
- ❏ Thermal strap to frame if equilibrium not observed.

Min Operational	255 K
Max Operational	351 K
Min Observed	293 K
Max Observed	-

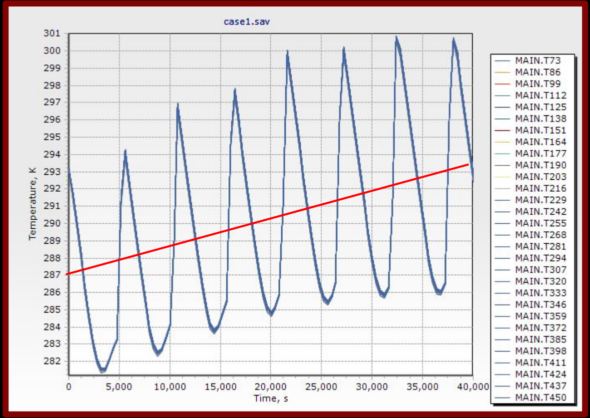
Payload Temperature Considerations (Transmission)

Temperature Issues
- ❑ Maximum equilibrium not reached.

Potential issues with model
- ❑ Low detail within payload - does not account for internal heat buildups.
- ❑ Contact to frame over-simplified.

Solutions
- ❑ Improve detail and accuracy of model
- ❑ Thermal strap if equilibrium not observed.

Min Survivable	253 K
Min Operational	273 K
Max Survivable	333 K
Max Operational	333 K
Min Observed	281 K
Max Observed	-



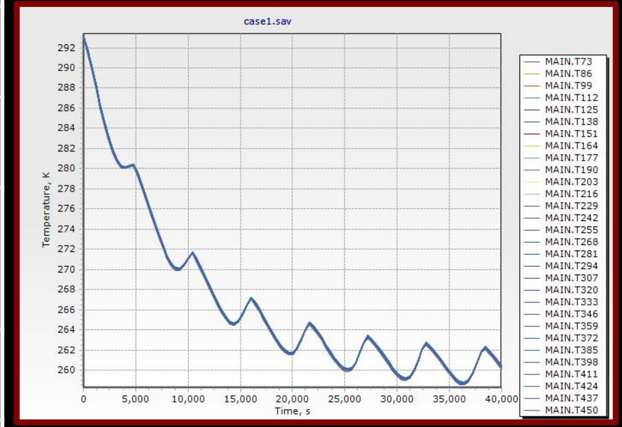
Payload Temperature Considerations (Nominal)

Temperature Issues
- ❑ None for non operation.

Potential issues with model
- ❑ Low detail within payload - does not account for internal component temperatures.
- ❑ Contact to frame over-simplified.

Solutions
- ❑ Improve detail and accuracy of model
- ❑ Heater for payload if minimum temperature decreases below survivable.

Min Survivable	253 K
Min Operational	273 K
Max Survivable	333 K
Max Operational	333 K
Min Observed	258 K
Max Observed	293



C & DH Temperature Considerations (Transmission)

Temperature Issues

- ❑ Maximum equilibrium not reached.

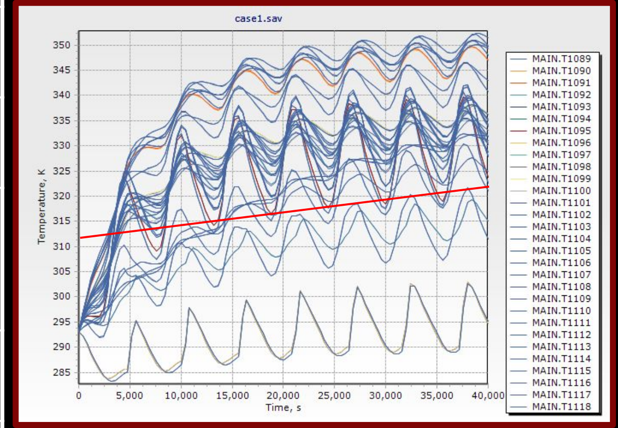
Potential issues with model

- ❑ Low detail within C&DH - does not account for well for internal heat buildups.

Solutions

- ❑ Improve detail and accuracy of model
- ❑ Increase conductivity to rails if equilibrium is not observed.

Min Survivable	233 K
Min Operational	253 K
Max Survivable	358 K
Max Operational	358 K
Min Observed	283 K
Max Observed	~350 K



Power+Comms Temperature Considerations (Transmission)

Temperature Issues

- ❑ Maximum equilibrium likely not reached.

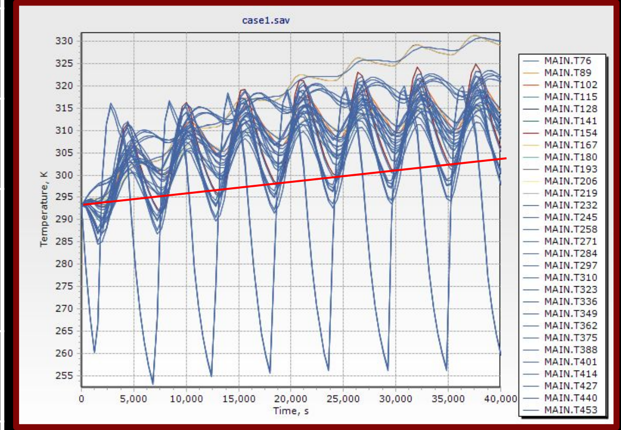
Potential issues with model

- ❑ Comms and Power combines - does not account for well for internal heat buildups within either system.

Solutions

- ❑ Seperate Power and Comms, improve model detail
- ❑ Increase conductivity to rails if equilibrium is not observed, potential thermal strap

Min Survivable	253 K
Min Operational	273 K
Max Survivable	358 K
Max Operational	318 K
Min Observed	255 K
Max Observed	~330 K



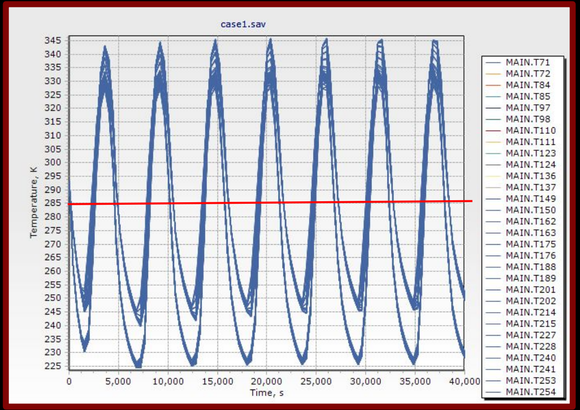
Solar Panel Temperature Considerations (Transmission)

Temperature Issues
- ❏ None observed

Potential issues with model
- ❏ Solar cell absorptivity not accounted for.
- ❏ Conductance with 6061 support approximated

Solutions
- ❏ Add solar cells to model

Min Survivable	223 K
Min Operational	223 K
Max Survivable	423 K
Max Operational	423 K
Min Observed	225 K
Max Observed	345 K



Next Steps for Thermal Analysis

- Improvement of Simulation Structure Parameters:
  - Improve accuracy of all contacts by calculating exact conductivity.
  - Improve detail of internal heat loads, primarily Payload, C&DH, and Power + Comms.
  - Simulate during different times of the year, with different Beta angles.
- Improvement of Orbit Parameters:
  - Import orbit from STK giving more accurate orientation and solar flux values.
- Practical Testing
  - Test thermal efficiency of components. Use these values, rather than nominal power draw.
  - Test thermal conductivity between components. Use these values to improve simulation.
  - Test temperature ranges of components. These may very in testing from those defined on spec sheets.

Satellite System Analysis:

Power Performance

Power Driving Requirements (1/2)

ID	Requirement	Notes	Parent	Verification
EPS-08	The EPS shall generate 11 Wh < Energy < 80 Wh (TBR) of energy during each orbit.	Energy generation requirement is determined from the satellite's Orbit Average Power and orbital period. Maximum is driven by maximum capacity of energy storage.	EPS-02	Test
EPS-09	The EPS shall provide 13 Wh (TBR) < Capacity < 80 Wh of energy storage.	Minimum requirement is based on the maximum Depth of Discharge of the battery pack and the Satellite power budget. Maximum is driven by NanoRacks standards for ISS deployment.	EPS-03	Test
EPS-10	The EPS shall provide 7 W ± 2 W (TBR) average power during entire mission operations after initial boot-up.	Average power is driven by the satellite's Orbit Average Power.	EPS-04	Test

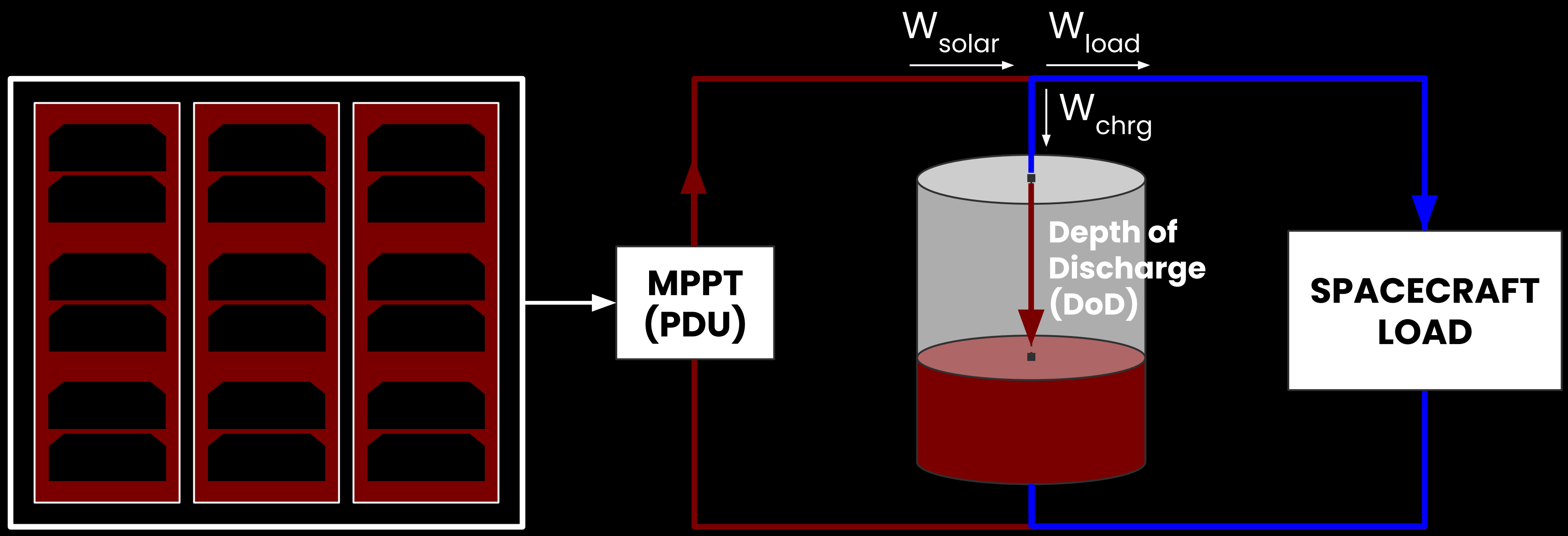
Legend: Compliant Compliant by CDR Compliant by TRR Compliant By FRR

Power Driving Requirements (2/2)

ID	Requirement	Notes	Parent	Verification
EPS-11	The EPS shall be designed to provide 32 W ± 5 W (TBR) maximum continuous power during eclipse.	This requirement is driven by the maximum draw of the power budget during transmission.	EPS-04, PAY-11, PAY-12	Test
EPS-12	The EPS shall be designed to provide 50 W ± 10 W (TBR) peak power draw during deployment of external components.	This requirement is driven by the maximum draw of the power budget before payload operations plus deployment values.	EPS-04	Test

Legend: Compliant Compliant by CDR Compliant by TRR Compliant By FRR

Power Subsystem Model



Array Power Required = Charging Power + Load Power

Power Budget (1/2)

Component	Subsystem	Current (A)	Voltage (V)	Quantity	Power (W)	Margin (%)	Design Power (W)	Duty Cycle (%)	Consumption (W)
On Board Computer	C&DH	0.300 A	5.0 V	1	1.500 W	20%	1.800 W	100%	1.800 W
Watchdog Processor	C&DH	0.008 A	3.3 V	2	0.053 W	5%	0.055 W	100%	0.055 W
Payload Controller	C&DH	0.300 A	5.0 V	1	1.500 W	20%	1.800 W	50%	0.900 W
Power Distribution Unit	Power	0.015 A	7.2 V	1	0.108 W	20%	0.130 W	100%	0.130 W
Battery Heater	Power	0.300 A	5.0 V	2	3.000 W	20%	3.600 W	5%	0.180 W
Magnetorquers	ADCS	0.051 A	7.2 V	3	1.110 W	5%	1.166 W	10%	0.117 W
CubeMag	ADCS	0.007 A	7.2 V	1	0.050 W	5%	0.053 W	100%	0.053 W
Reaction Wheels (Peak)	ADCS	0.118 A	7.2 V	3	2.550 W	5%	2.678 W	5%	0.134 W
Reaction Wheels (Nominal)	ADCS	0.025 A	7.2 V	3	0.540 W	5%	0.567 W	95%	0.539 W
ADCS Core	ADCS	0.127 A	7.2 V	1	0.918 W	5%	0.964 W	100%	0.964 W
Sun Sensor (Nominal)	ADCS	0.030 A	3.3 V	1	0.100 W	5%	0.105 W	98%	0.103 W
Sun Sensor (Peak)	ADCS	0.053 A	3.3 V	1	0.175 W	5%	0.184 W	2%	0.004 W
Star Tracker (Nominal)	ADCS	0.050 A	3.3 V	1	0.165 W	5%	0.173 W	98%	0.170 W
Star Tracker (Peak)	ADCS	0.082 A	3.3 V	1	0.271 W	5%	0.284 W	2%	0.006 W
GPS	ADCS	0.020 A	5.0 V	1	0.100 W	5%	0.105 W	100%	0.105 W
Active Antenna	Comms	0.010 A	5.0 V	1	0.050 W	5%	0.053 W	100%	0.053 W
Radio Rx	Comms	0.055 A	3.3 V	1	0.182 W	5%	0.191 W	100%	0.191 W
Radio Tx	Comms	0.800 A	3.3 V	1	2.640 W	5%	2.772 W	16%	0.444 W
Thermal Sensors	Thermal	0.0001 A	5.0 V	8	0.002 W	5%	0.002 W	100%	0.002 W

Power Budget (2/2)

Component	Subsystem	Current (A)	Voltage (V)	Quantity	Power (W)	Margin (%)	Design Power (W)	Duty Cycle (%)	Consumption (W)
FPGA	Payload	0.150 A	3.30 V	1	0.495 W	20%	0.594 W	6%	0.036 W
EDFA	Payload	1.810 A	5.0 V	1	9.050 W	20%	10.860 W	6%	0.652 W
Seed Laser	Payload	0.080 A	1.90 V	2	0.304 W	20%	0.365 W	6%	0.022 W
Beacon Laser	Payload	0.330 A	3.00 V	1	0.990 W	20%	1.188 W	8%	0.095 W
FSM Driver	Payload	0.250 A	5.00 V	1	1.250 W	20%	1.500 W	8%	0.120 W
Modulator Driver	Payload	0.083 A	12.00 V	1	1.000 W	20%	1.200 W	6%	0.072 W
								OAP	**6.94 W**
								Peak	**31.54 W**
								Energy	**10.41 Wh**
								Energy in Eclipse	**4.03 Wh**

Power Budget (2/2)

Component	Subsystem	Current (A)	Voltage (V)	Quantity	Power (W)	Margin (%)	Design Power (W)	Duty Cycle (%)	Consumption (W)
FPGA	Payload	0.150 A	3.30 V	1	0.495 W	20%	0.594 W	6%	0.036 W
EDFA	Payload	1.810 A	5.0 V	1	9.050 W	20%	10.860 W	6%	0.652 W
Seed Laser	Payload	0.080 A	1.90 V	2	0.304 W	20%	0.365 W	6%	0.022 W
Beacon Laser	Payload	0.330 A	3.00 V	1	0.990 W	20%	1.188 W	8%	0.095 W
FSM Driver	Payload	0.250 A	5.00 V	1	1.250 W	20%	1.500 W	8%	0.120 W
Modulator Driver	Payload	0.083 A	12.00 V	1	1.000 W	20%	1.200 W	6%	0.072 W

OAP	**6.94 W**
Peak	**31.54 W**
Energy	**10.41 Wh**
Energy in Eclipse	**4.03 Wh**

OAP = Orbit Average Power

Energy = OAP × Orbital Period

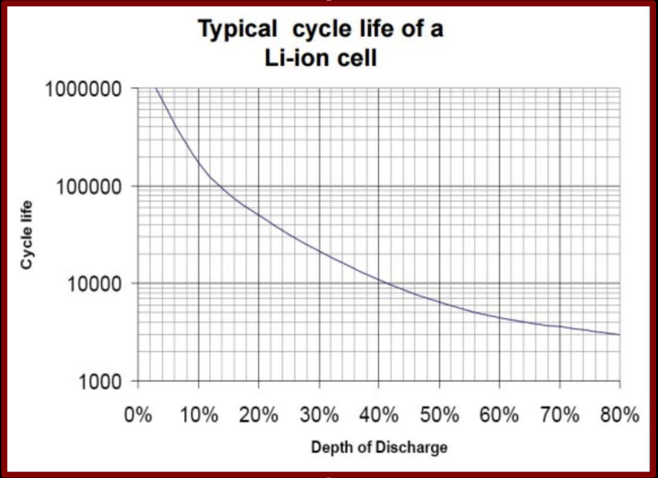
Eclipse Energy Consumption = OAP × Time in Eclipse

Desired Maximum DoD

Maximum Depth of Discharge (DoD) is derived from cycle count and 18650 Li-Ion battery life.

Result: Max DoD

Estimated Mission Duration	2.0 years
Number of Cycles	11680



Max Depth of Discharge	**33.00%**

Battery Sizing

Batteries are sized based on eclipse energy usage and desired DoD from previous slide.

Result: DoD

Battery Energy Used in Eclipse	4.03 Wh
Single Cell Capacity	12.06 Wh
Desired DoD	33%
PDU Efficiency	95.00%

Required Capacity = Energy / (DoD × PDUe)

Required Battery Capacity	12.84 Wh

cells = CEILING(required capacity / single capacity)

Cells Needed	2
Actual DoD (with 6 Cells)	**5.56%**

Battery Sizing

Batteries are sized based on eclipse energy usage and desired DoD from previous slide.

Result: DoD

Battery Energy Used in Eclipse	4.03 Wh
Single Cell Capacity	12.06 Wh
Desired DoD	33%
PDU Efficiency	95.00%

Required Capacity = Energy / (DoD × PDUe)

Required Battery Capacity	12.84 Wh

\# cells = CEILING(required capacity / single capacity)

Cells Needed	2
Actual DoD (with 6 Cells)	**5.56%**

Delta between Actual and Desired DoD covers for changes in Payload design & on times

Derived Battery Charging Power

Batteries are charged from excess array power in sunlight. Required charge power is derived from battery energy consumption in eclipse and time in sunlight per orbit.

Result: Required Charge Power

Eclipse Energy Usage	4.03 Wh
Sunlight Duration (ISS)	0.92 h

Charge power = Eclipse energy / Sunlight duration

Required Charge Power	**4.38 W**

Solar Array Power Required

Solar Array Power is derived from OAP and Battery Charging Power.

Result: Required Array Power

OAP	6.94 W
MPPT Efficiency	90%
Distribution Efficiency (estimated)	95%
Adjusted OAP	7.31 W
Adjusted Battery Charging Power	4.38

$$W_{array} = W_{load} + W_{charge}$$

Array Power Required	**11.68 W**

Solar Array Power Required

Solar Array Power is derived from OAP and Battery Charging Power.

Result: Required Array Power

OAP	6.94 W
MPPT Efficiency	90%
[illegible]nated)	95%
Adjusted OAP	7.31 W
Adjusted Battery Charging Power	4.38

Adjusted powers account for MPPT and distribution efficiencies

$$W_{array} = W_{load} + W_{charge}$$

Array Power Required	**11.68 W**

Array Area Required

Solar Array area is derived from required power and cell efficiency.

Result: Required Array Area

Efficiency at EOL	28.0%
Power / cm^2 at EOL	0.0377
Array Power Required	11.68 W
Array Area Required	**310.0 cm^2**

Array Area Required

Solar Array area is derived from required power and cell efficiency.

Result: Required Array Area

Estimates 5E14 e/cm² fluence for 1 MeV protons over mission lifetime

Efficiency at EOL	28.0%
Power / cm^2 at EOL	0.0377
Array Power Required	11.68 W
Array Area Required	**310.0 cm^2**

Solar Array Deployment Scheme

Configuration	No Deployables	Shuttlecock (4 dep.)	Butterfly (2. dep)
Cell Count	~20-22 (depending on Payload config)	24	18
Power delivered (W)	~8	~26.9	~20.2
Solar Cell Area (cm^2)	~190.25	724.32	543.24
Pros	Easier engineering	Most power delivered	High power delivery, low cell count
Cons	High cell count for drastically lower power delivered	Decreased mission lifetime (drag) and high MoI	Not natively offered by GranSystems (but they are willing to accommodate)

Solar Array Deployment Scheme (cont.)

Configuration	No Deployables	Shuttlecock (4 dep.)	Butterfly (2. dep)
Cell Count	~20-22 (depending on Payload config)	24	18
Power delivered (W)	~8	~26.9	~20.2
Solar Cell Area (cm^2)	~190.25	724.32	543.24
Pros	Easier engineering	Most power delivered	High power delivery, low cell count
Cons	[illegible] power delivered	[illegible] high MoI	Not natively offered by GranSystems (but they are willing to accommodate)

Does not satisfy EOL area requirement

Solar Array Deployment Scheme (cont.)

Configuration	**No Deployables**	**Shuttlecock (4 dep.)**	**Butterfly (2. dep)**
Cell Count	~20-22 (depending on Payload config)	24	18
Power delivered (W)	~8	~26.9	~20.2
Solar Cell Area (cm^2)	~190.25	724.32	543.24
Pros	Easier engineering	Most power delivered	High power delivery, low cell count
Cons	[illegible] power delivered	[illegible] high MoI	Not natively offered by GranSystems (but they are willing to accommodate)

Does not satisfy EOL area requirement

Butterfly deployment selected due to design feasibility and reduced drag

EOL & BOL Solar Power Margins

Solar margins come from required solar power, panel sizing, and cell efficiency.

Result: EOL & BOL Solar Power Margins

End of Life

Efficiency at EOL	28.0%
Solar Array Area	543.2 cm^2
Solar Power at EOL	20.48 W
Efficiency-Adjusted Solar Power at EOL	17.51 W
Solar Power Margin at EOL	**33.27%**

Beginning of Life

Efficiency at BOL	29.6%
Solar Array Area	543.2 cm^2
Solar Power at BOL	21.81 W
Efficiency-Adjusted Solar Power at BOL	18.64 W
Solar Power Margin at BOL	**37.33%**

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Efficiency-Adjusted Solar Power at BOL	18.64 W
Solar Power Margin at BOL	**37.33%**

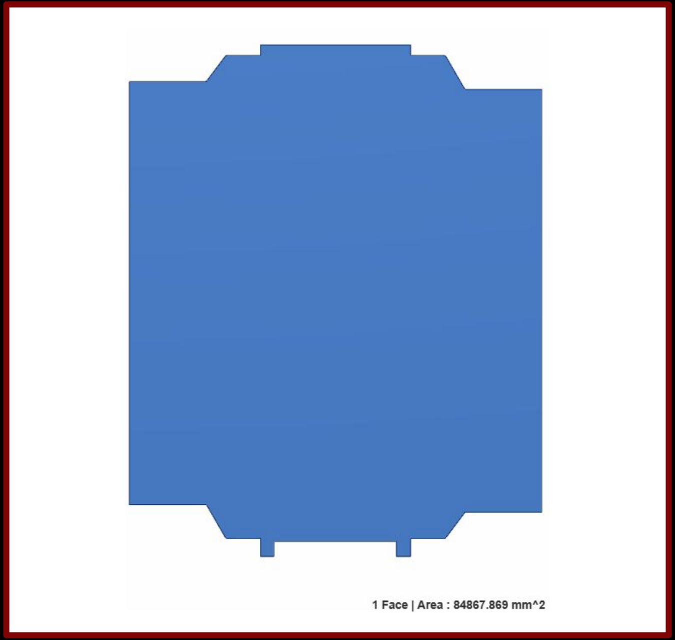
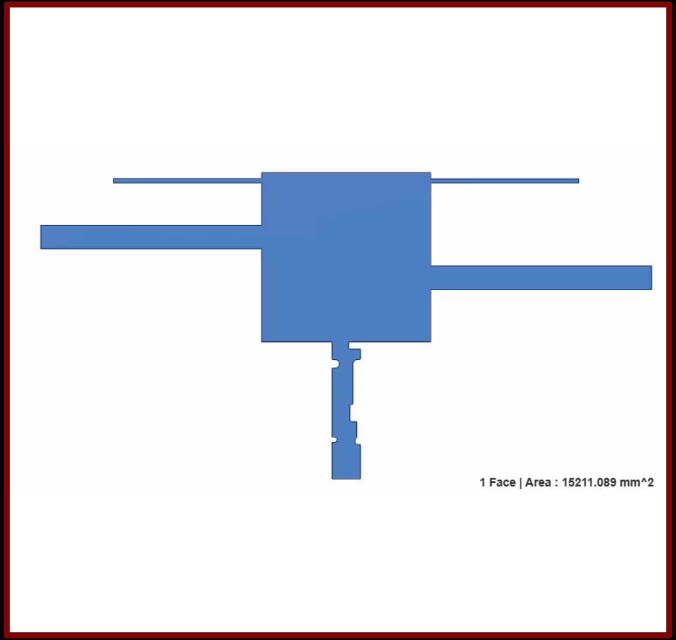
Excess margin at EOL and BOL allows for peak power draws through mission lifetime

Satellite System Analysis:
Orbital Model

Orbital Analysis Introduction

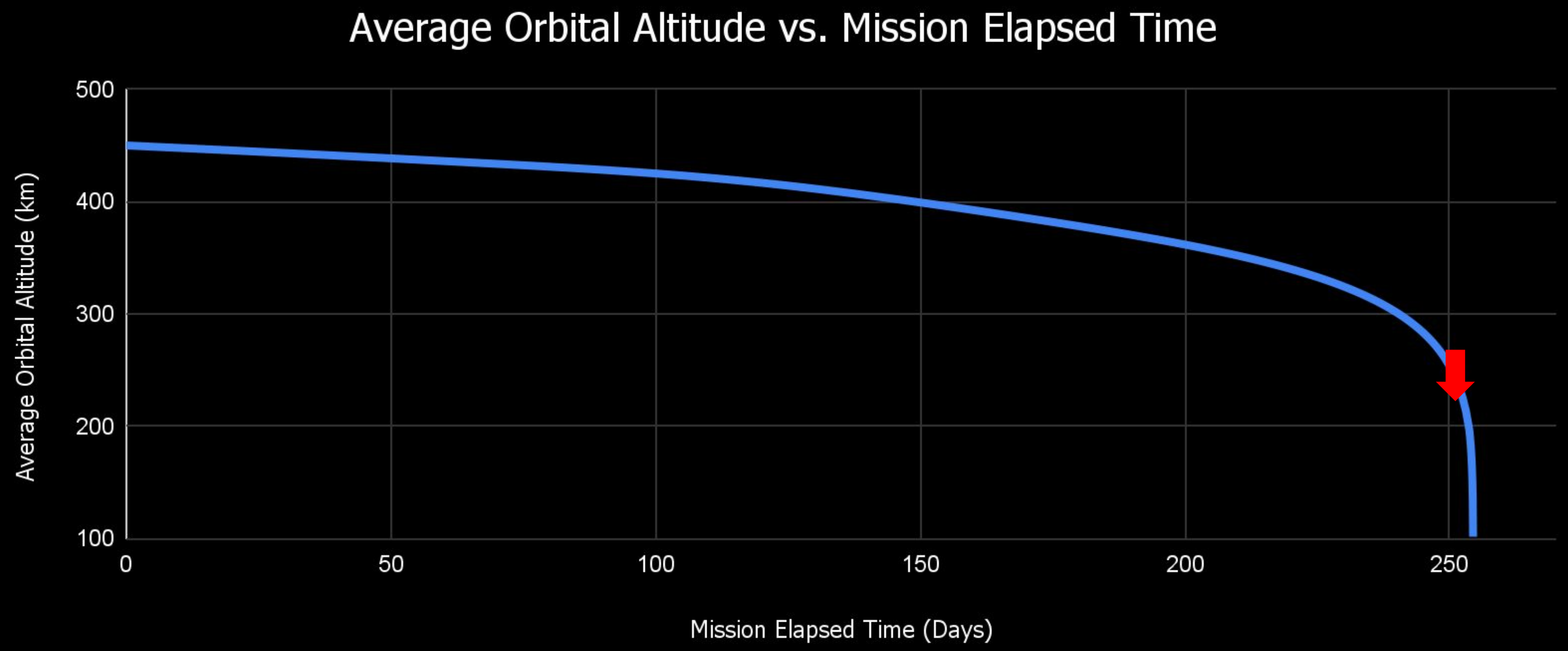
Goal	Orbital analysis undertaken to confirm major aspects related to mission design and lifetime, most importantly altitude specifications and drag implications
Design Process	Satellite replicated as 3D model in Ansys STK. Orbital inclination assumed to be near-ISS (51.6 deg). Mass assumed to be ~3 kg.
Worst-case	9U of Drag Surface (butterfly back profile)
Best-case	1U of Drag Surface (+X/-X profile)

Orbital Drag Model

Worst-case (~9U drag area)	Best-case (~1U drag area)
	

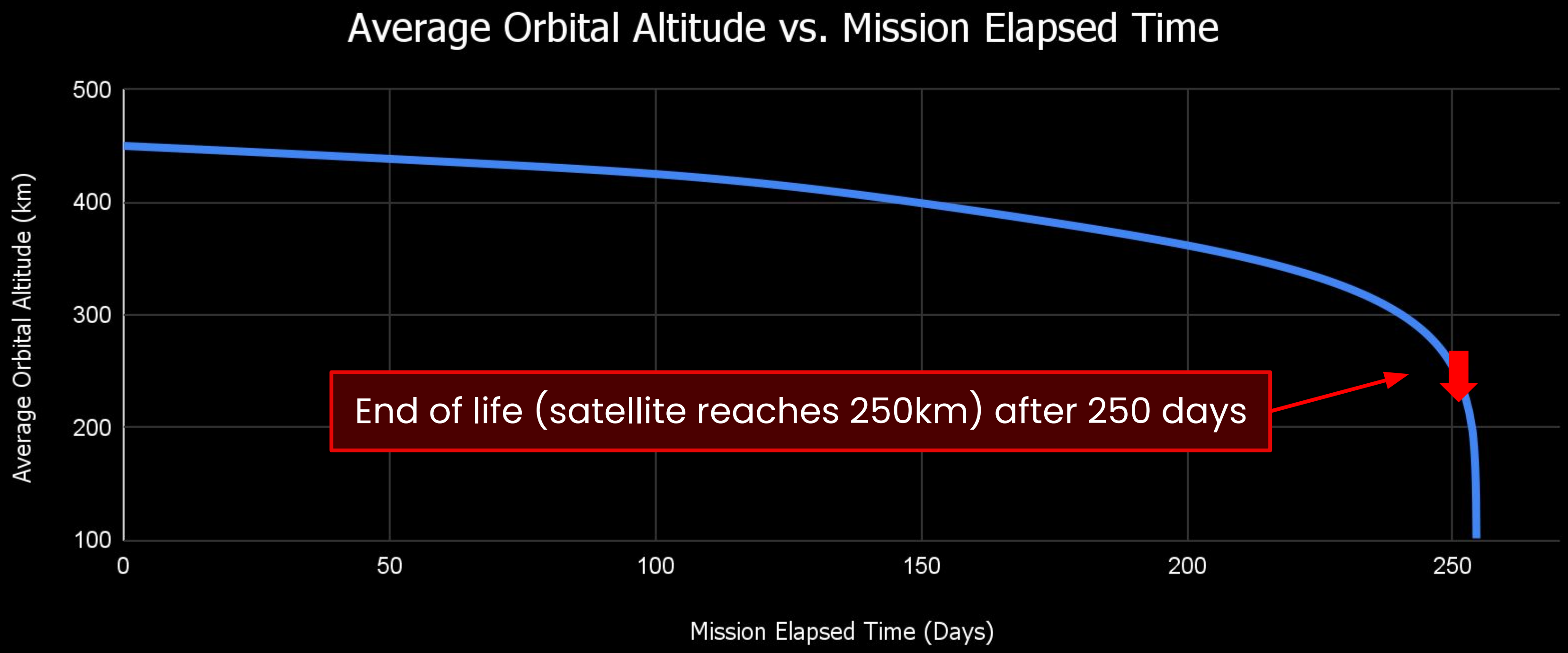
Orbital Analysis Results (1/2)

WORST CASE DECAY: 9U of Drag Surface Area



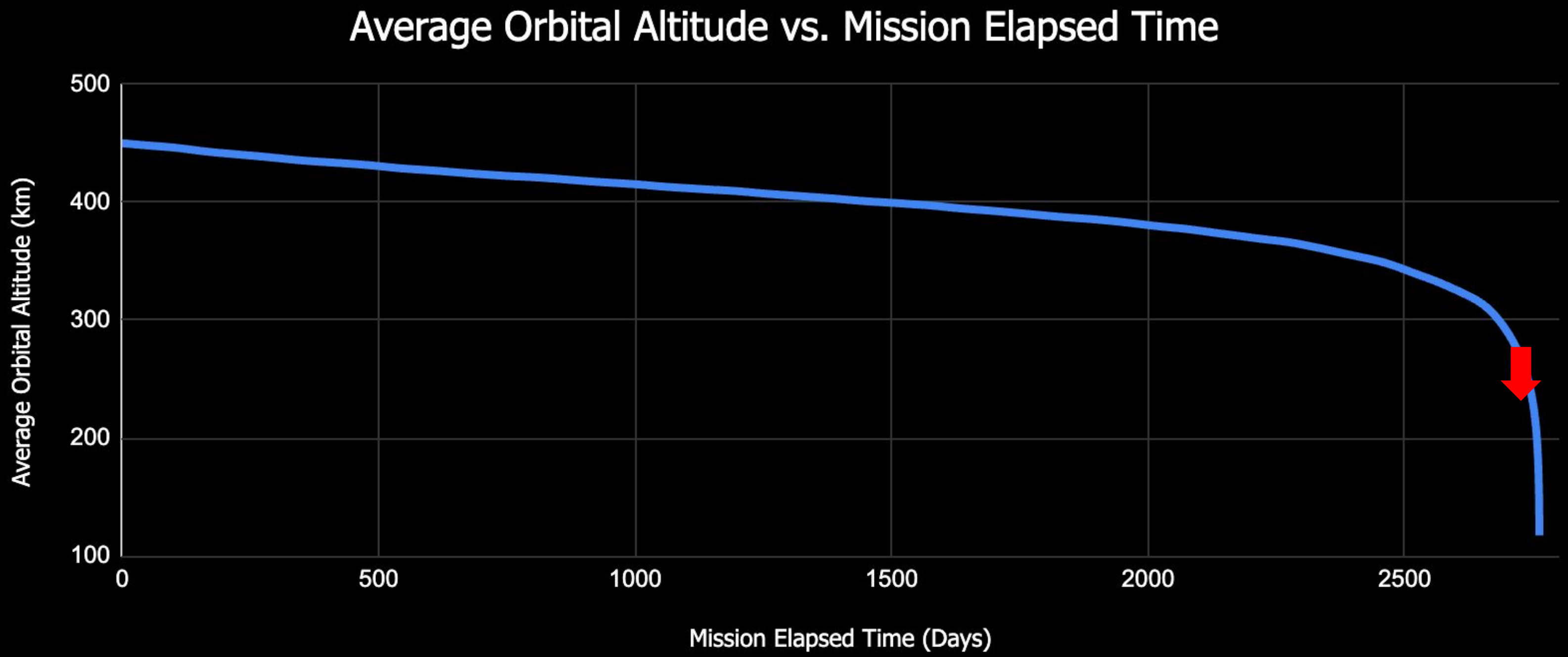
Orbital Analysis Results (1/2)

WORST CASE DECAY: 9U of Drag Surface Area



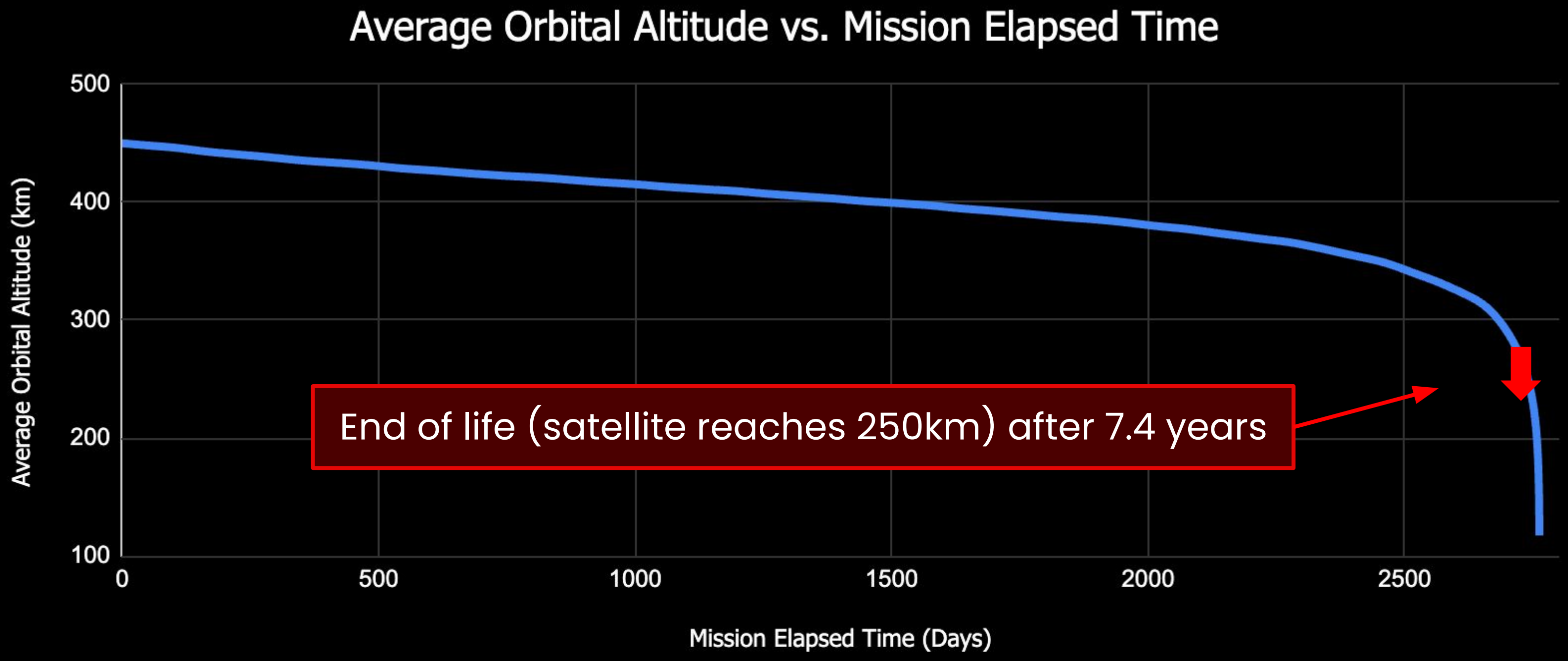
Orbital Analysis Results (2/2)

BEST CASE DECAY: 1U of Drag Surface Area

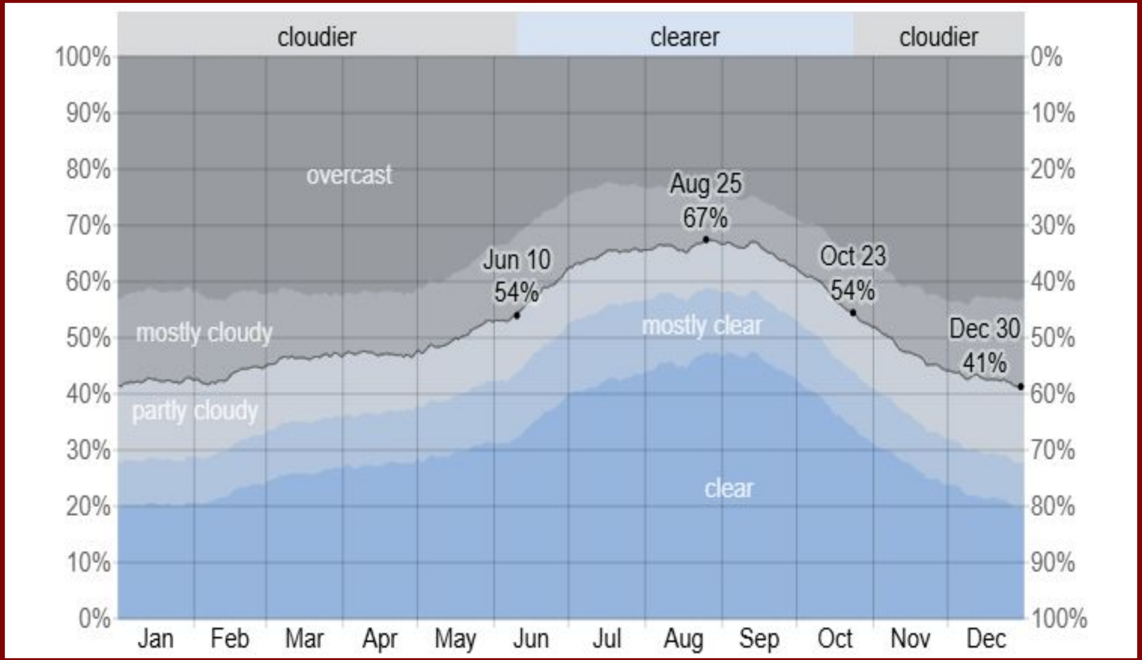


Orbital Analysis Results (2/2)

BEST CASE DECAY: 1U of Drag Surface Area



Yerkes Observatory Weather Analysis



Weather Spark, Nov. 17th 2024, https://weatherspark.com/y/13420/Average-Weather-in-Williams-Bay-Wisconsin-United-States-Year-Round#Sections-Clouds

PULSE-A launch window should be spring/early summer to maximize early optically viable passes while allowing for commissioning phase. Benefits of this include:

- Student availability over summer
- Maximized passes for early development / fixes
- Clear skies for transmissibility

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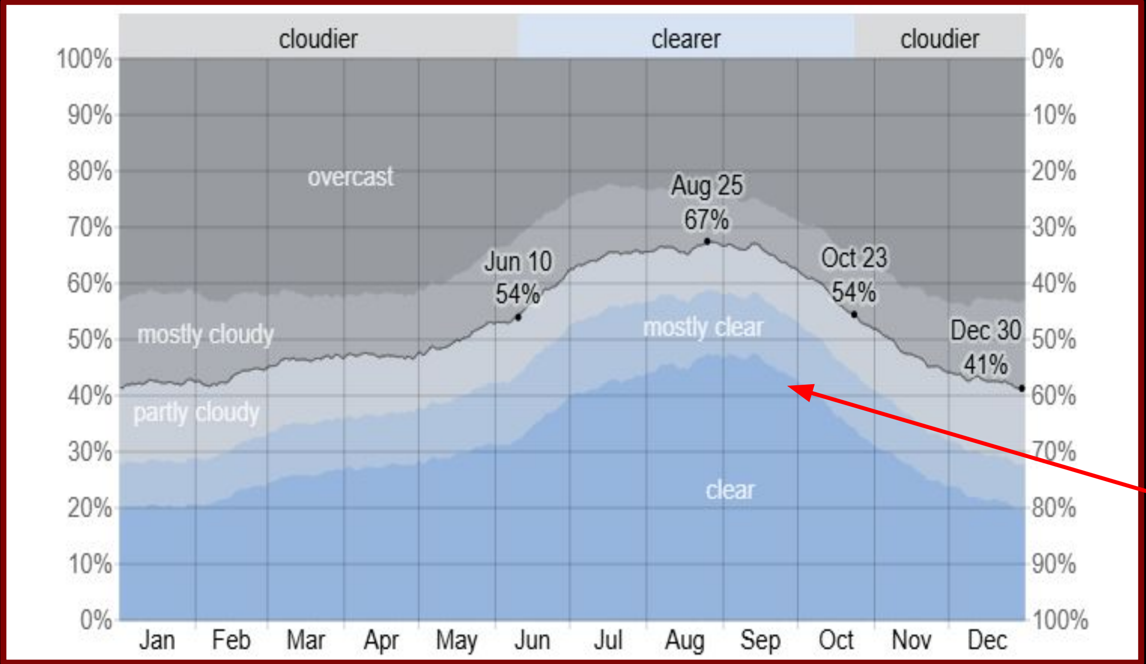

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- Student availability over summer
- Maximized passes for early development / fixes
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Takeaways from Orbital & Weather Analysis

Orbital decay analysis is preliminary, but shows that launch window is flexible.

With a worst-case mission duration of 250 days, an autumn launch is not recommended as due to lack of summer transmission season and unideal winter weather conditions.

Worst and best-case estimates are unrealistic and mission duration is expected to be in the 2-4 year range. Further orbital analysis is necessary to confirm ideal launch availability.

Satellite System Analysis:

RF Communications

Communications Driving Requirements

ID	Requirement	Notes	Parent	Verification
COMM-05	The communications subsystem shall be capable of transmitting data to the RFGS in a downlink at a rate of 3.2 kbps ≤ Rate ≤ 50 kbps.	3.2 kbps is derived from uplinking a 200 KB update file or downlinking a 200 KB log file over 10 minutes with a 20% margin (10 minutes is taken from an RF pass taking approximately 15 minutes). 50 kbps is an approximate upper bound for COTS UHF radios.	COMM-01	Analysis
COMM-06	The communications subsystem shall be capable of receiving uplinked data from the RFGS at a rate of 3.2 kbps ≤ Rate ≤ 50 kbps.		COMM-02	Analysis

Legend: Compliant | Compliant by CDR | Compliant by TRR | Compliant By FRR

Downlink Data Budget

Subsystems	Nominal Data (Bytes)	PAT Data (Bytes)
Power	35	9
OBC	22	18
Temperature	30	30
Radio	50	20
ADCS	126	126
GPS	33	33
Payload	39	33
Housekeeping	67	14
Total	**398**	**283**
Total with Margin	**478**	**340**

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Temperature	30	30
Radio	50	20
ADCS		
GPS		
Payload		
Housekeeping	67	14
Total	**398**	**283**
Total with Margin	**478**	**340**

Sufficient data is allocated to ensure the capability to complete PAT sequence and ensure general satellite health

Radio Frequency Link Budget Summary (GomSpace)

Downlink Budget

Parameter	Value
Data Rate	9600 bps
Transmit Power	1 W
Frequency	
Modulation	GFSK
Channel	25 kHz
Link Margin	**19.1 dB**

Uplink Budget

Parameter	Value
Data Rate	9600 bps
Transmit power	15W
Frequency	
Modulation	GFSK
Channel	25 kHz
Link Margin	**23.8 dB**

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Uplink Budget

Parameter	Value
Data Rate	9600 bps
Transmit power	15W
Frequency	
Modulation	GFSK
Channel	25 kHz
Link Margin	**23.8 dB**

Link margin is sufficient to establish secure comms at data rate acceptable within systems and telemetry requirements

Next Steps for Communications

- ❑ Review and confirm telemetry budget
- ❑ Review and confirm decision between full vs half duplex
- ❑ Determine telemetry packet structure
    - ❑ Investigate further into CCSDS standards
- ❑ Detailed design of radio integration into the bus

Ground Station System Design:

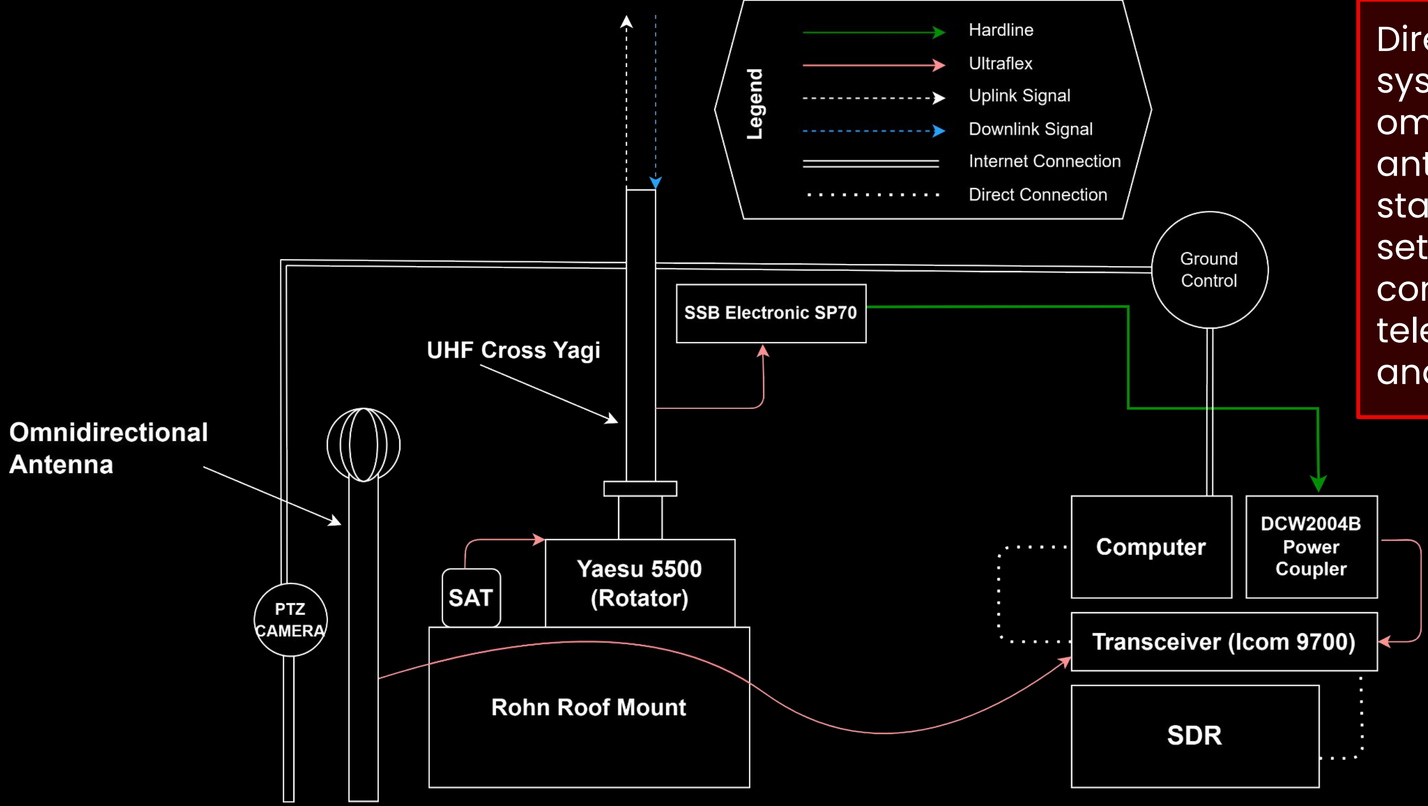
RF Ground Station (RFGS)

RFGS Driving Requirements

ID	Requirement	Notes	Parent	Verification
RFGS-06	The RFGS shall point to the Satellite in orbit.	Use an orbital model updated with GPS data.	RFGS-01, RFGS-02, RFGS-03	Demonstration
RFGS-07	The RFGS shall accept downlinked GPS data from the Satellite every TBD sec ≤ Time ≤ TBD sec.	Necessary to point to the Satellite. Roughly estimate every 1 sec.	RFGS-06, OGS-04	Demonstration
RFGS-08	The RFGS shall be capable of receiving downlinked data from the Satellite at a rate of 3.2 kpbs ≤ Rate ≤ 50 kbps.	The same boundaries as parent COMM requirement.	RFGS-03, COMM-05	Demonstration
RFGS-09	The RFGS shall be capable of uplinking data at a rate of 3.2 kbps ≤ Rate ≤ 50 kbps.	The same boundaries as parent COMM requirement.	RFGS-02, COMM-06	Demonstration

Legend: Compliant | Compliant by CDR | Compliant by TRR | Compliant By FRR

RFGS Configuration and Components



Directional antenna system with backup omnidirectional antenna, connected to standard transceiver setup and to ground control for data and telemetry use and analysis.

RFGS Configuration and Components

Component Name	Model	Function
Transceiver	Icom 9700	High-performance transceiver for amateur radio operators.
SDR	Ettus USRP B205mini-i: 1x1, 70 MHz-6 GHz SDR/Cognitive Radio	SDR helps with demodulation, filtering, and frequency tuning through an embedded system.
Omnidirectional Antenna	EB-432/RK70cm, 400-470 MHz	Receives and transmits radio signals equally in all directions, without requiring precise aiming. **Backup for failure of yagi rotators. (8dB Uplink Link Margin with 15W) (9.6dB Downlink Link Margin)**
Pan-Tilt-Zoom Camera	MTZ5-IR-FC-AD-WIFI	Used for monitoring and visual alignment of the antenna setup.

RFGS Configuration and Components

Component Name	Model	Function
Roof Mount	Rohn JRM23810	Sable, non-penetrating base for antenna installation.
Rotator / Controller	Yaesu 5500	Rotates the antenna to the desired azimuth and elevation, allowing precise aiming for tracking satellites or specific signal sources.
Cross Yagi UHF Antenna	M2 2MCP14 (7x7 Element) Circular Polarized Yagi	Directional antenna for UHF signals, for transmission and reception.
1/2" Hardline Coaxial Cable	LDF4-50A	Low-loss coaxial cable for component connection.
Power coupler	DCW 2004 B	Facilitates the transmission of both RF signals and DC power over the same coaxial line.

RFGS Configuration and Components

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Rotator / Controller	Yaesu 5500	Rotates the antenna to the desired azimuth and elevation, allowing precise aiming for tracking satellites or specific signal sources.
Cross Yagi UHF Antenna		HF signals, ption.
1/2" Hardline Coaxial Cable		
Power coupler	DCW 2004 B	Facilitates the transmission of both RF signals and DC power over the same coaxial line.

Directional antenna and supplementary components enable higher link margin with attainable pointing requirements (25 degrees maximum pointing error)

RFGS Configuration and Components

Component Name	Model	Function
Coaxial Cable	LMR 400 Ultraflex	Flexible, low-loss coaxial cable, ideal for connecting equipment with minimized signal loss.
Preamplifier	SSB Electronic SP70	Preamplifier for UHF signals, improving weak signal reception by amplifying UHF frequencies before they reach the receiver.
Rotator Controller	Self-Contained Antenna Tracker (S.A.T)	A self contained antenna rotator and radio controller. Natively controls Icom radios, Yaesu rotators and can interface with PSTRotator.

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Rotator Controller	Self-Contained Antenna Tracker (S.A.T)	A self contained antenna rotator and radio controller. Natively controls Icom radios, Yaesu rotators and can interface with PSTRotator.

Pointing software outsourced unlike PAT sequence software for both OGS and Satellite, making it significantly lower risk.

RFGS Pointing Scheme

Cross Yagi antennas must be pointed within 25° of satellite’s azimuth for proper RF communication. Therefore, we need a software that will assist in automatically pointing the antennas to the satellite as soon as satellites crosses above the horizon.

Source: DX Engineering Customer Review

SAT

Pros:

- ❏ Native support for Yaesu 5500 rotator
- ❏ API for easy integration with Ground Control
- ❏ Simple interface and more visualizations

Cons:

- ❏ Less modular and compatible with fewer 3rd party applications
- ❏ Is an additional point/source of failure/error

SATPC32

Pros:

- ❏ Compatible with 3rd party applications that we may utilize later on + more advanced functions
- ❏ Much more information provided about status of rotor, antenna, etc.

Cons:

- ❏ Software only, so cannot control rotator by itself; needs 3rd party rotor interface/controllers
- ❏ No API; Ground Control must read data from older DDE interface

RFGS Next Steps

- ❑ Amateur/Experimental frequency licensing from IARU, FCC
- ❑ RFGS placement - expand
- ❑ Communication Control simulation
- ❑ Ground Control Interfacing
- ❑ Radio qualification testing through in-lab simulation
  - ❑ Communicating with our radios
  - ❑ Collecting telemetry from other LEO Cubesats

Ground Station System Design:

Optical Ground Station (OGS)

OGS Driving Requirements (1/2)

ID	Requirement	Notes	Parent	Verification
OGS-02	The OGS shall transmit the ground station laser beacon to the Satellite.		SAT-07	Demonstration
OGS-09	The OGS shall detect the downlink transmission laser's wave packets as bits based on separated polarization states.		OGS-01	Demonstration
OGS-11	The OGS shall digitalize data at transmission rate of 1 Mbps ≤ Data Rate ≤ 10 Mbps.	Transmission rate refers to the Payload's uncoded data rate.	MSN-02, OGS-01	Demonstration
OGS-13	The OGS shall be capable of maintaining the telescope pointed at the Satellite during the optical pass for 150 seconds ≤ Time ≤ 300 seconds.	150 s is the length of the typical pass. There should never be a pass longer than 300 s.	PAY-16, PAY-17	Analysis

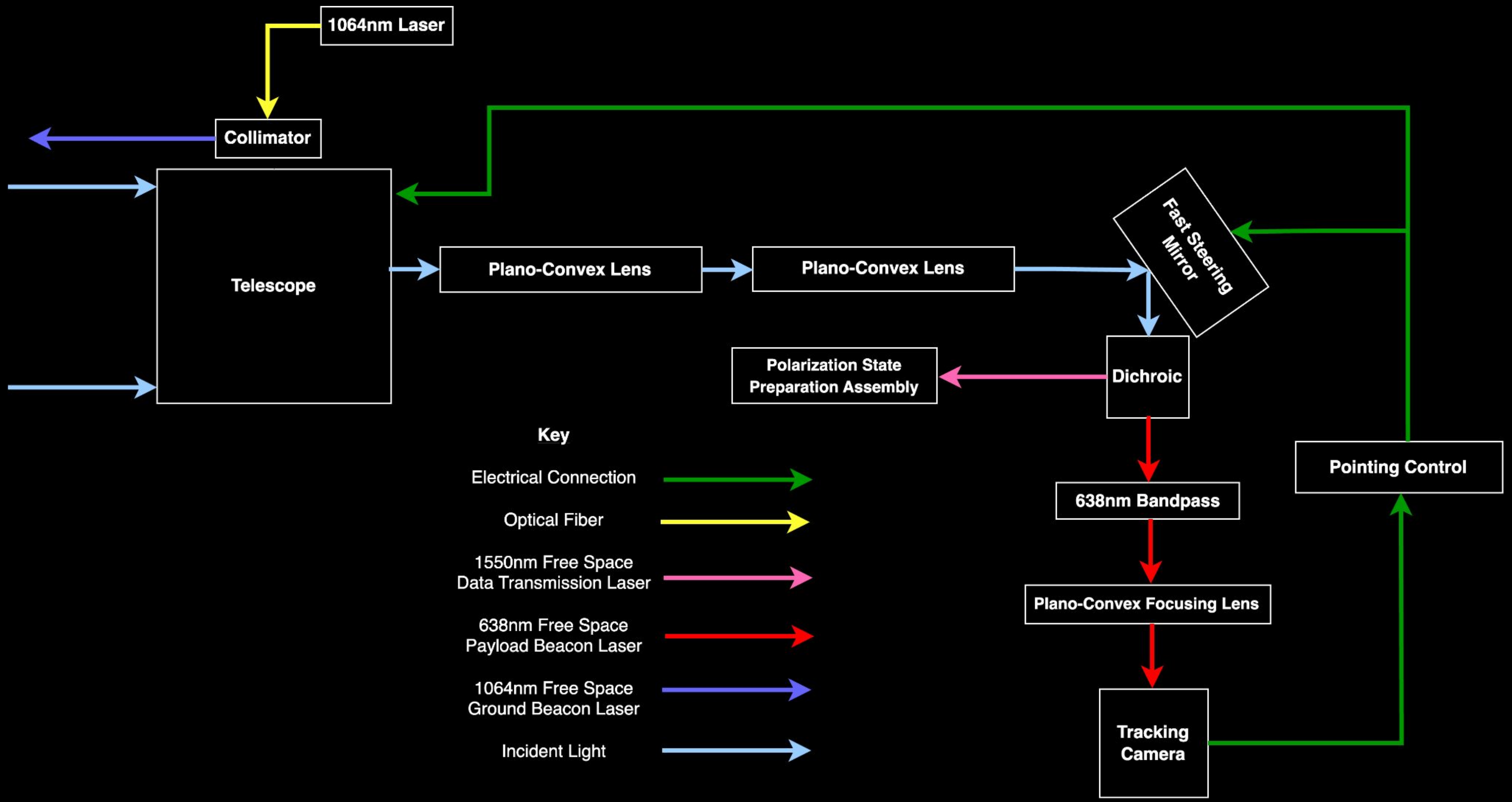
Legend: Compliant Compliant by CDR Compliant by TRR Compliant By FRR

OGS Driving Requirements (2/2)

ID	Requirement	Notes	Parent	Verification
OGS-15	The OGS shall be capable of detecting the downlink laser transmission at a link margin of 3 dB ≤ Margin ≤ 6 dB.	These values are taken from the Optical Tx Link Budget.	OGS-09, OGS-12	Analysis
OGS-16	The OGS shall be capable of detecting the Payload beacon laser at a link margin of 3 dB ≤ Margin ≤ 6 dB.	These values are taken from the Payload Beacon Link Budget.	OGS-06	Analysis

Legend: Compliant Compliant by CDR Compliant by TRR Compliant By FRR

OGS Configuration - Tracking Assembly



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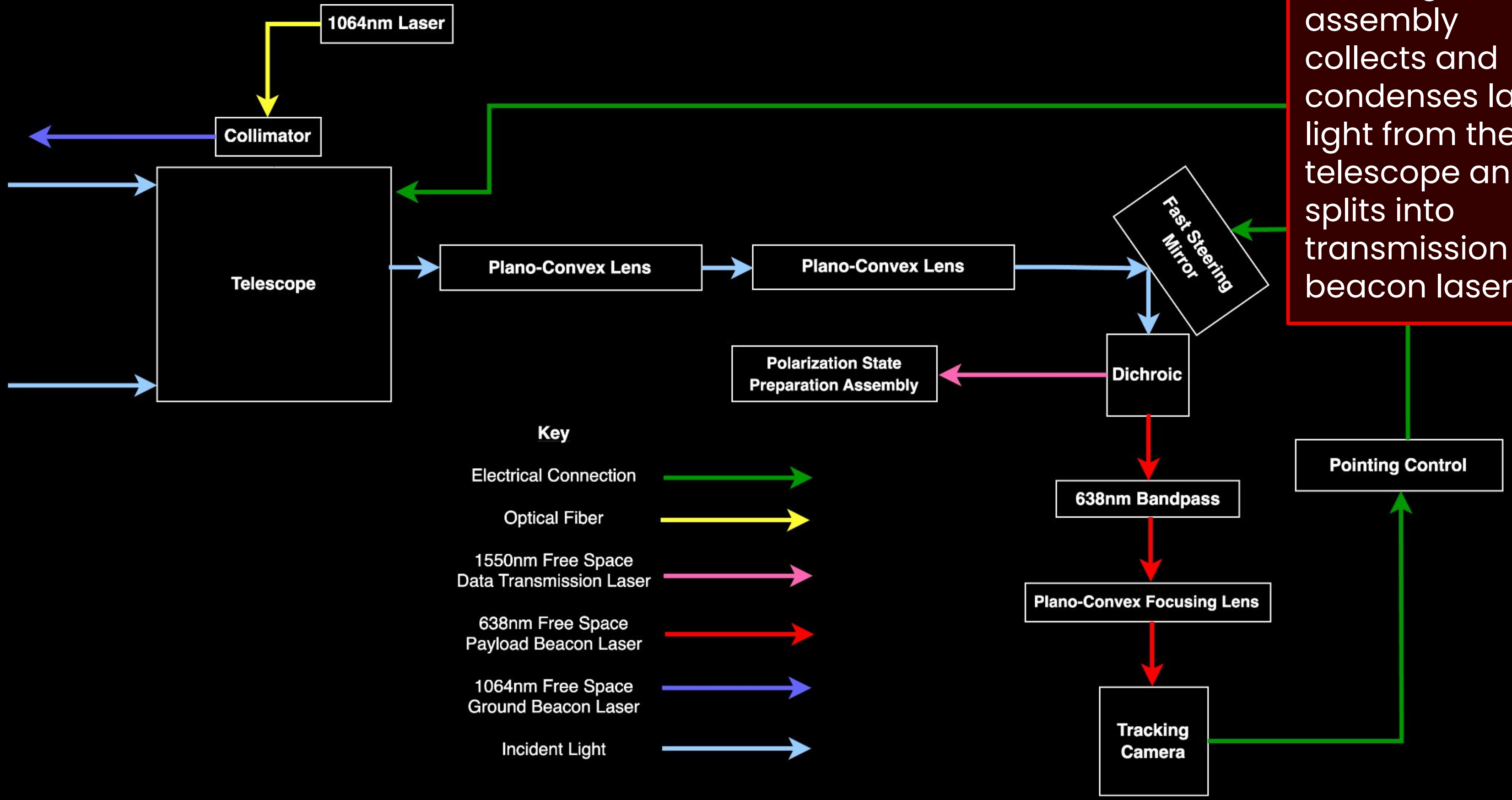
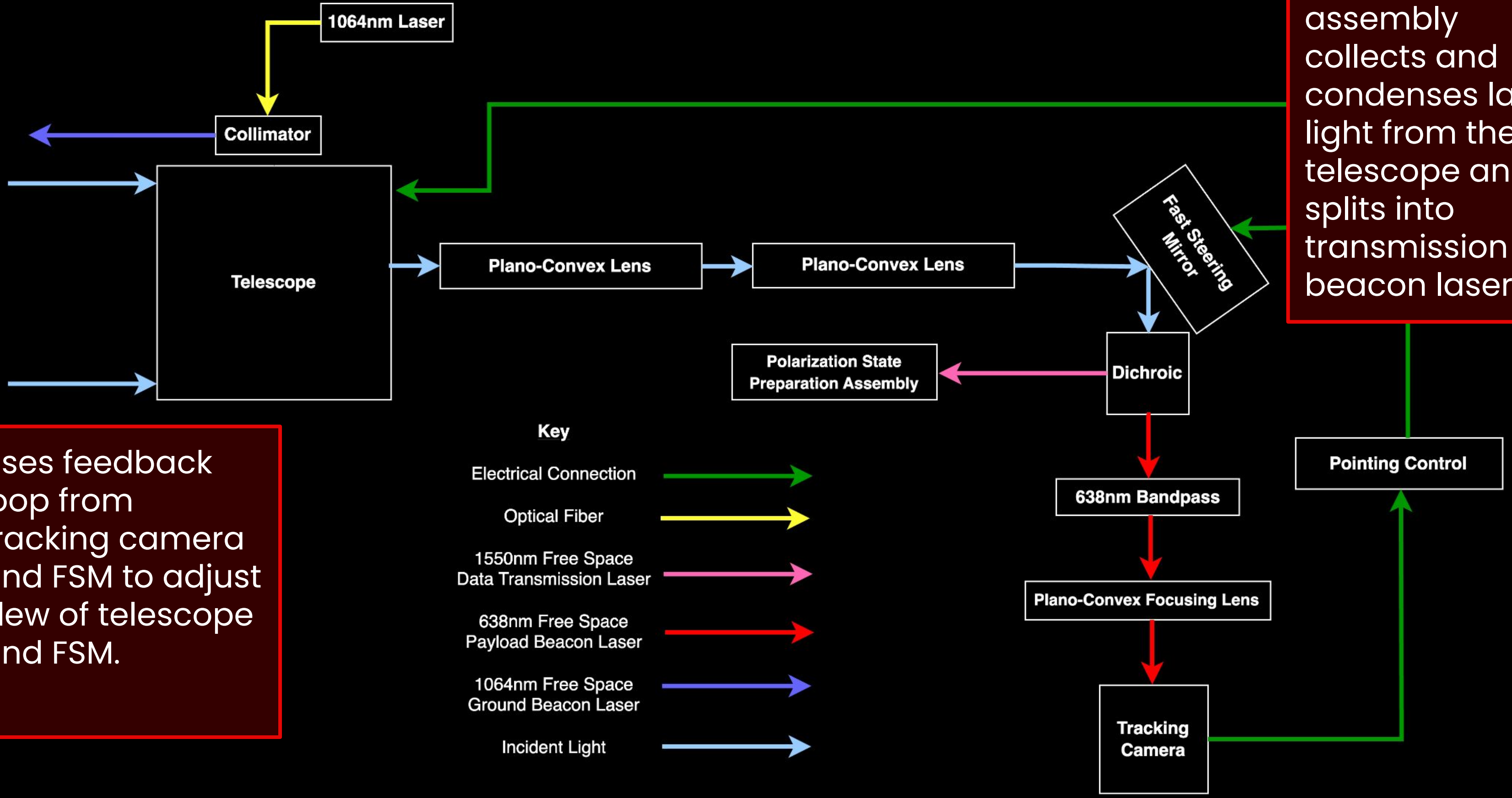


Tracking assembly collects and condenses laser light from the telescope and splits into transmission and beacon lasers.

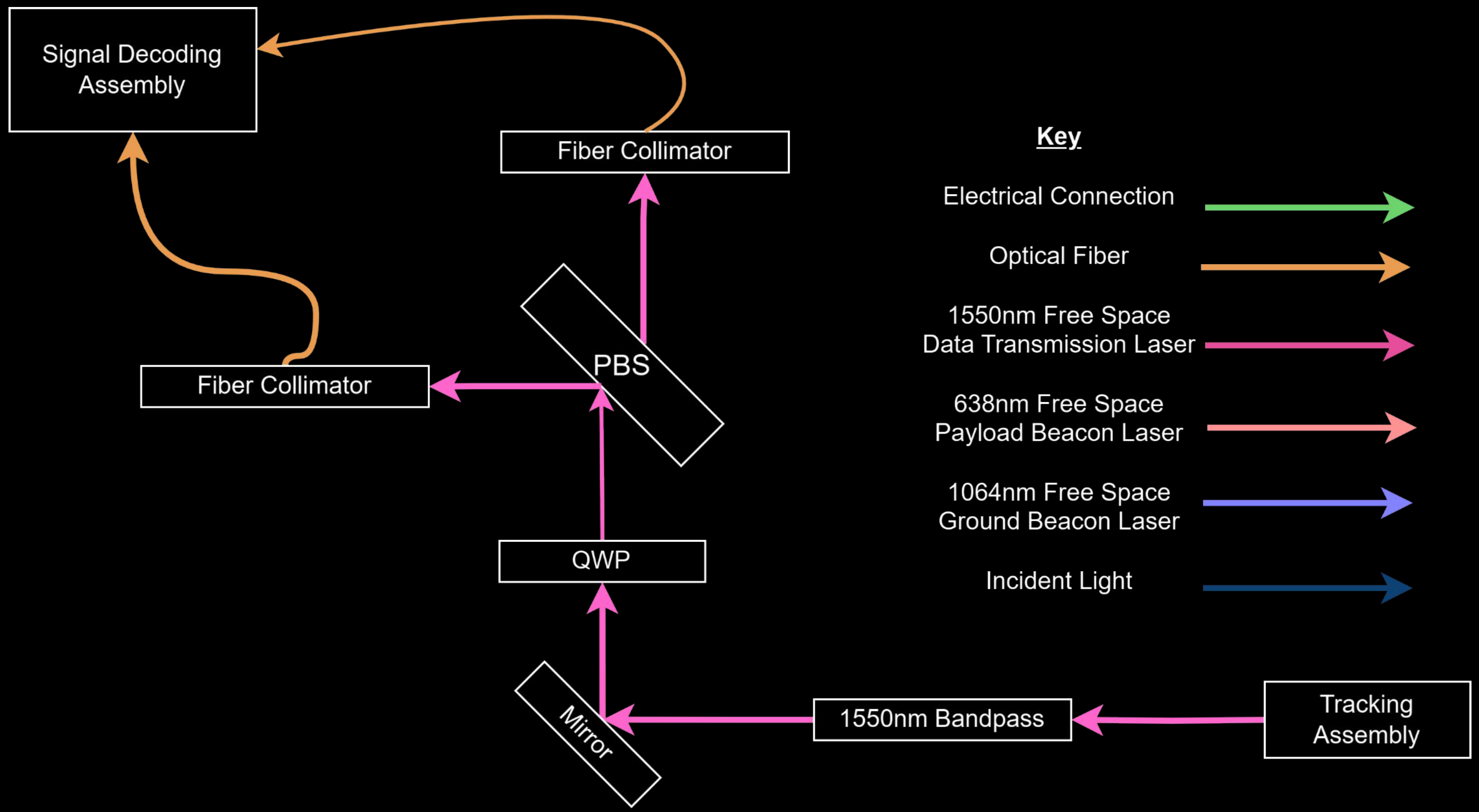
OGS Configuration - Tracking Assembly



Tracking assembly collects and condenses laser light from the telescope and splits into transmission and beacon lasers.

Uses feedback loop from tracking camera and FSM to adjust slew of telescope and FSM.

OGS Configuration - Polarization State Preparation Assembly



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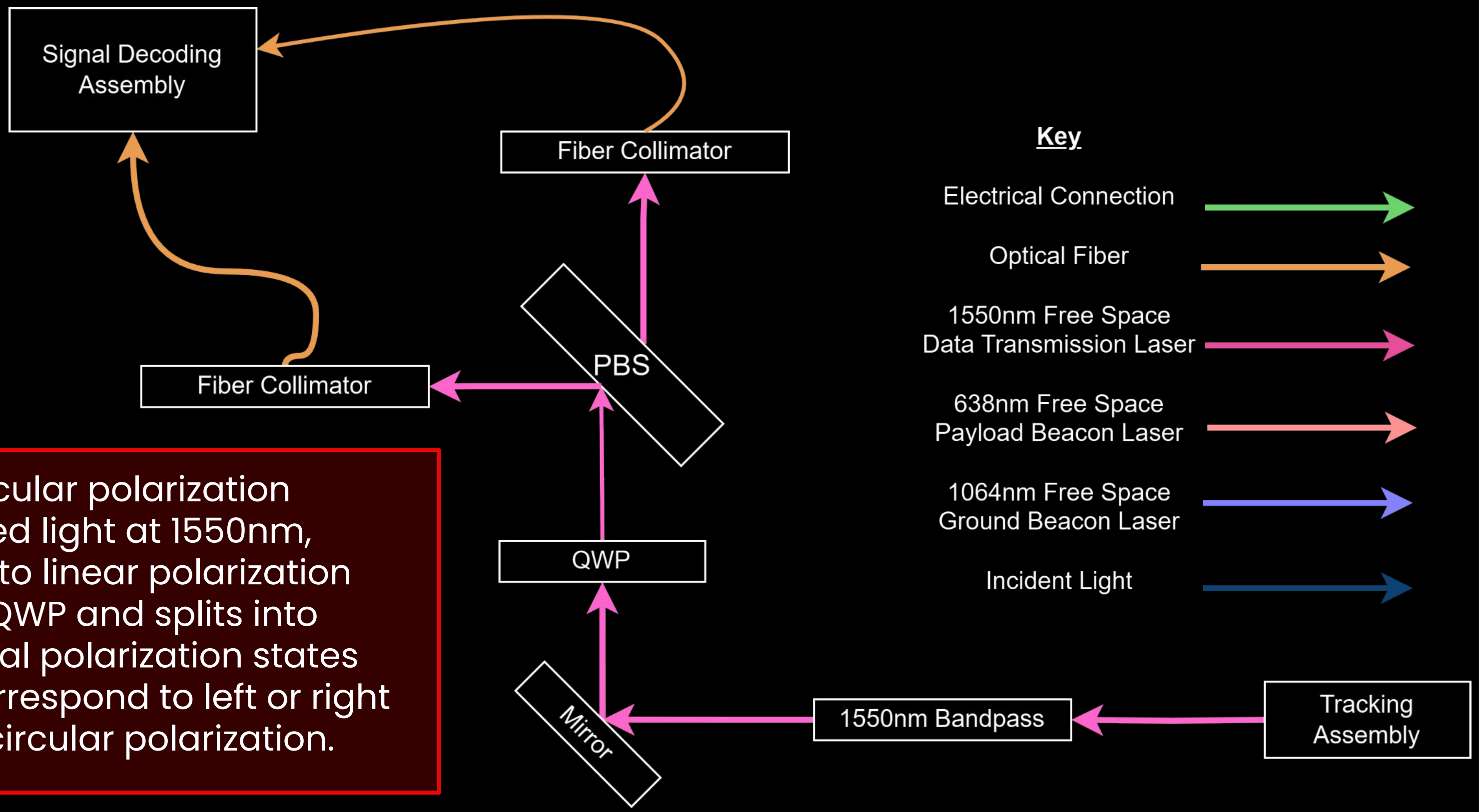
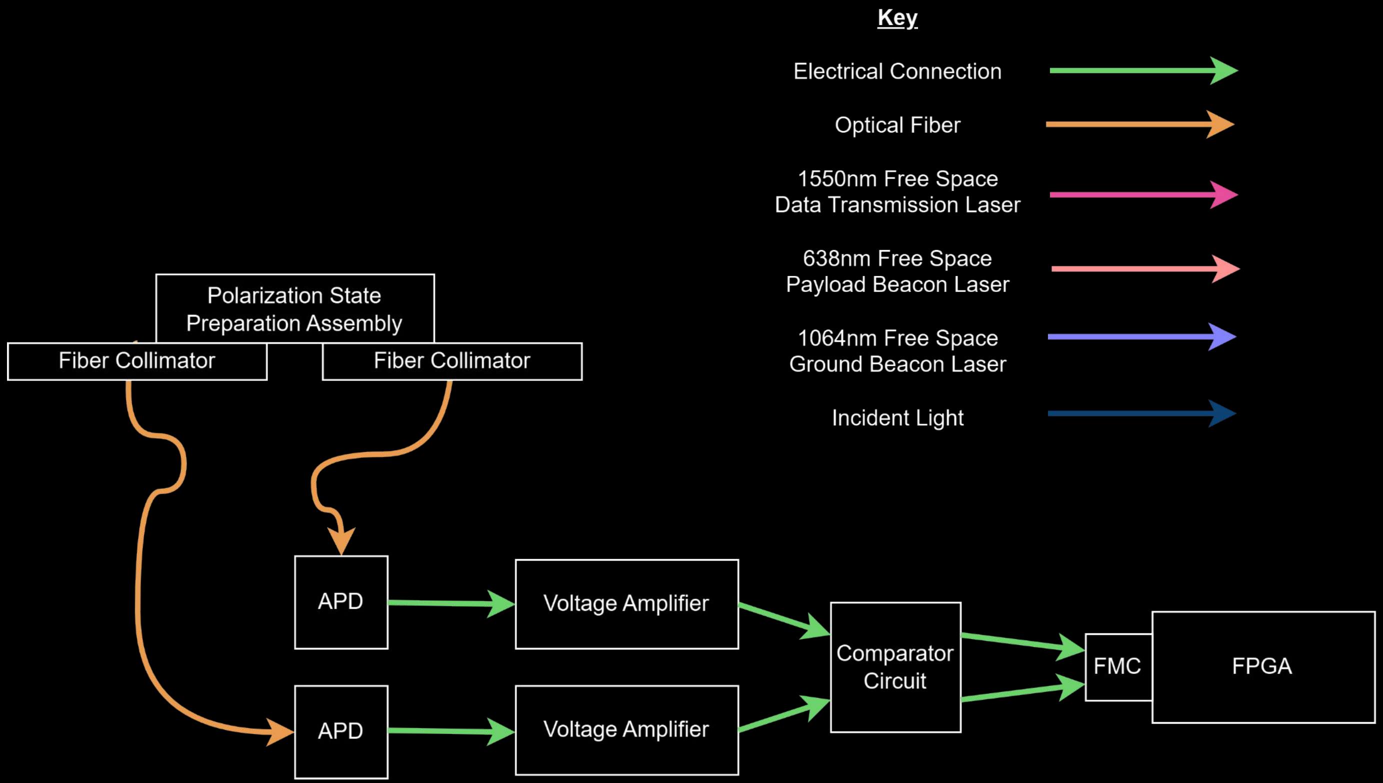


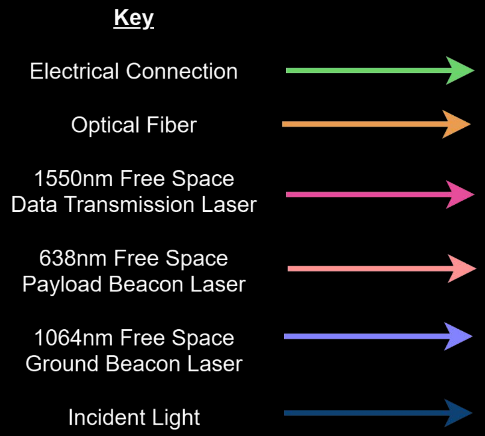
Takes circular polarization modulated light at 1550nm, converts to linear polarization through QWP and splits into orthogonal polarization states which correspond to left or right handed circular polarization.

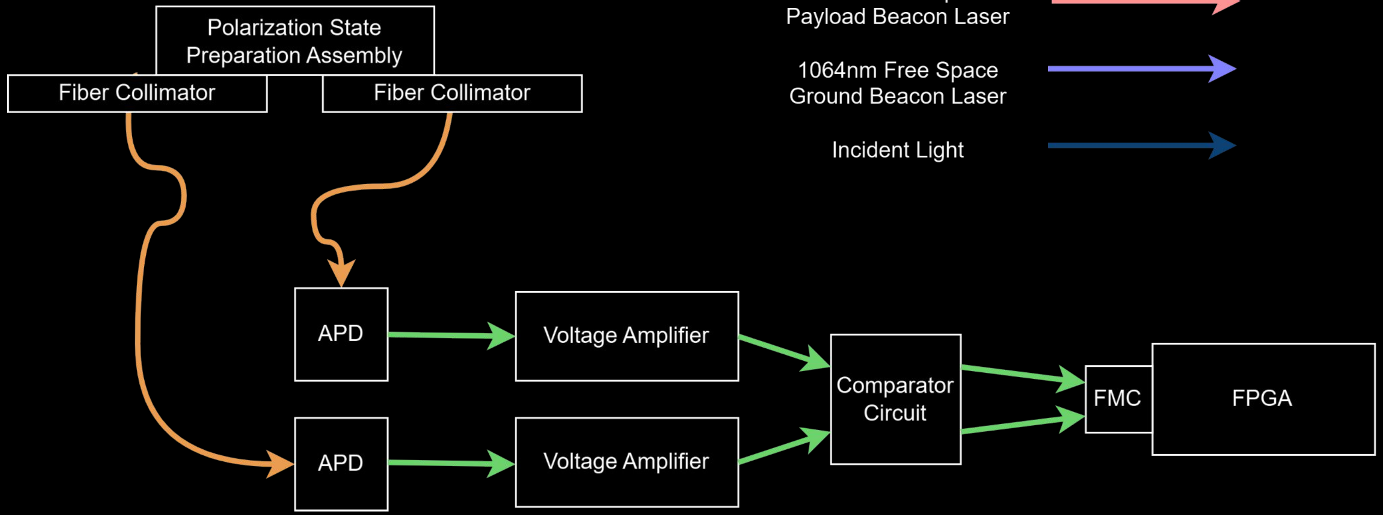
OGS Configuration - Signal Decoding Assembly



OGS Configuration - Signal Decoding Assembly

Avalanche photodiodes and voltage amplifiers convert photon signals into readable voltages. Signals are compared and digitized within 1 to 10 MHz as specified by mission requirements.





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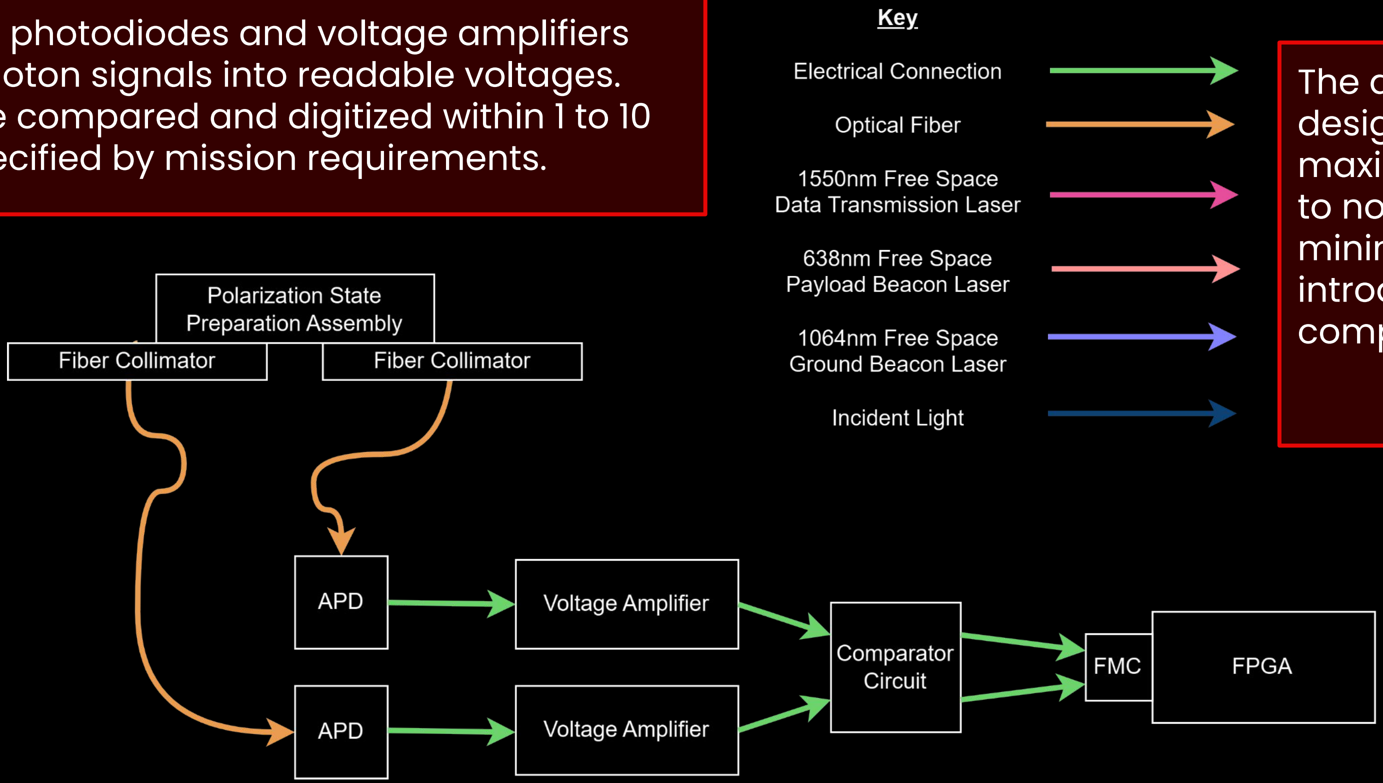

The assembly is designed to maximize signal to noise ratio, by minimizing noise introduced by components.

Fast Steering Mirror (FSM)

The fast steering mirror directs the downlink beam utilizing a reflective mirror mounted on rotary structures with motion-tracking sensors. This allows the FSM to quickly and accurately redirect incoming signal beam.

Model	Mass (kg)	Mirror head size (mm)	Clear Aperture (ø mm)	Angular range (degrees)	Bandwidth (Hz)	Angular Resolution (urad)
OIM5001	0.08	40.64 x 40.64 x 41.402	23.876	+/- 1.5 to +/- 3.0	>875	<0.6 to <1.2

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OIM5001	0.08	40.64 x 40.64 x 41.402	23.876	+/- 1.5 to +/- 3.0	>875	<0.6 to <1.2

Large clear aperture allows for large beamwidth out of telescope. Large bandwidth and small angular resolution mean high precision pointing and correction which will ultimately be limited by our software implementation.

Tracking Camera

The CMOS tracking camera receives information from the downlink laser. It is in a closed loop with both the FSM and the telescope in order to accurately aim the uplink beacon towards the satellite.

Model	Mass (kg)	Dimensions (mm)	Clear Sensor Aperture (mm)	Resolution	Exposure Range
ASI294MM-P	0.41	73.5 x Ø 78	L x W:19.1 x 13.0 Diagonal: 23.1	4144 x 2822	32 us to 2000 s

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Model	Mass (kg)	Dimensions (mm)	Clear Sensor Aperture (mm)	Resolution	Exposure Range
ASI294MM-P	0.41	72.5 x Ø 78	L x W: 19.1 [illegible] 23.1	4144 x 2822	32 us to 2000 s

Large exposure range allows for adaptable sampling rate to fit according to SNR requirements.

Optical Components

Component	Model	Mass (kg)	Outer Diameter (ø mm)	Clear Aperture (ø mm)	Wavelength Specifications (nm)	Focal Length (mm)
PCX Lens	**LA4904**	0.05	50.8	N/A	N/A	150.5
PCX Lens	**LA4306**	0.02	25.4	N/A	N/A	40.1
Dichroic	**DMLP900**	0.02	25.4	22.9	900 (Cut-On)	N/A
Silver Mirror	**PF10-03-P01**	0.05	25.4	22.9	450 - 20,000	N/A
PCX Lens	**LA4647-A**	0.05	12.7	11.43	350 - 700	20

Optical Components

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PCX Lens	**LA4904**	0.05	50.8	N/A	N/A	150.5
PCX Lens	**LA4306**	0.02	25.4	N/A	N/A	40.1
Dichroic	**DMLP900**	0.02	25.4	[illegible]	[illegible]	[illegible]
Silver Mirror	**PF10-03-P01**	0.05	25.4	[illegible]	[illegible]	[illegible]
PCX Lens	**LA4647-A**	0.05	12.7	[illegible]	[illegible]	[illegible]

Large diameters (approx 1 inch) for optics to accommodate for large beamwidth out of second PCX Lens (collimating setup).

Optical Components

Component	Model	Mass (kg)	Outer Diameter (ø mm)	Clear Aperture (ø mm)	Wavelength Specifications (nm)
1550 Bandpass	FBH1550-12	0.05	25	21.1	1550 (T ≥ 90%)
640 Bandpass	FBH640-10	0.05	25	21.1	635 (T ≥ 90%)
Quarter Waveplate	WPQSM05-1550	0.05	25.4	22.6	1550
Polarizing Beamsplitter	PBSW-1550	0.03	25.4	22.86	1550
Fiber Collimator	F810FC-1550	0.07	24	N/A	1550

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1550 Bandpass	FBH1550-12	0.05	25	21.1	1550 (T ≥ 90%)
640 Bandpass	FBH640-10	0.05	25	21.1	635 (T ≥ 90%)
Quarter Waveplate	WPQSM05-1550	0.05	25.4	22.6	1550
Polarizing Beamsplitter	PBSW-1550	0.03	25.4	22.86	1550
Fiber Collimator	F810FC-1550	0.07	24	N/A	1550

Wavelength specific components for good transmission (needs to be verified on Zemax simulation). FWHM of bandpass filters is 12 nm. Particularly, the 640 Bandpass FWHM is within the range of the 638 nm beacon laser.

Telescope - CPC 1100 Celestron

Optical Design	Schmidt-Cassegrain
Aperture	279.4mm (11”)
Focal Length	2800mm (110”)
Obstruction by Secondary Mirror	95mm (3.75”, 12% of aperture area)
Slew Speeds	Variable rates, max 5° / second

- ❑ Automated tracking by connecting a computer to the NexStar hand controller with a USB cable (Communication with telescope is in serial)
- ❑ Python script translated to serial commands

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- ❑ Python script translated to serial commands

11 inch diameter is consistent with link budget for transmission and beacon lasers. Slew speeds higher than required for PAT.

OGS Pointing Capabilities I: Information Inputs

The OGS pointing routine receives inputs from both its own pointing and the satellite's position and outputs slew rates for the telescope and FSM.

Satellite Position Inputs:

- ❏ GPS information from the RFGS
- ❏ Two-line elements (possibly from NORAD)

Pointing Inputs:

- ❏ Telescope angular position at any given interval
- ❏ Offset of satellite downlink beacon from center of tracking camera (offset_predict.c++)
- ❏ FSM angular position at any given interval

OGS Pointing Capabilities II: Outputs and Limitations

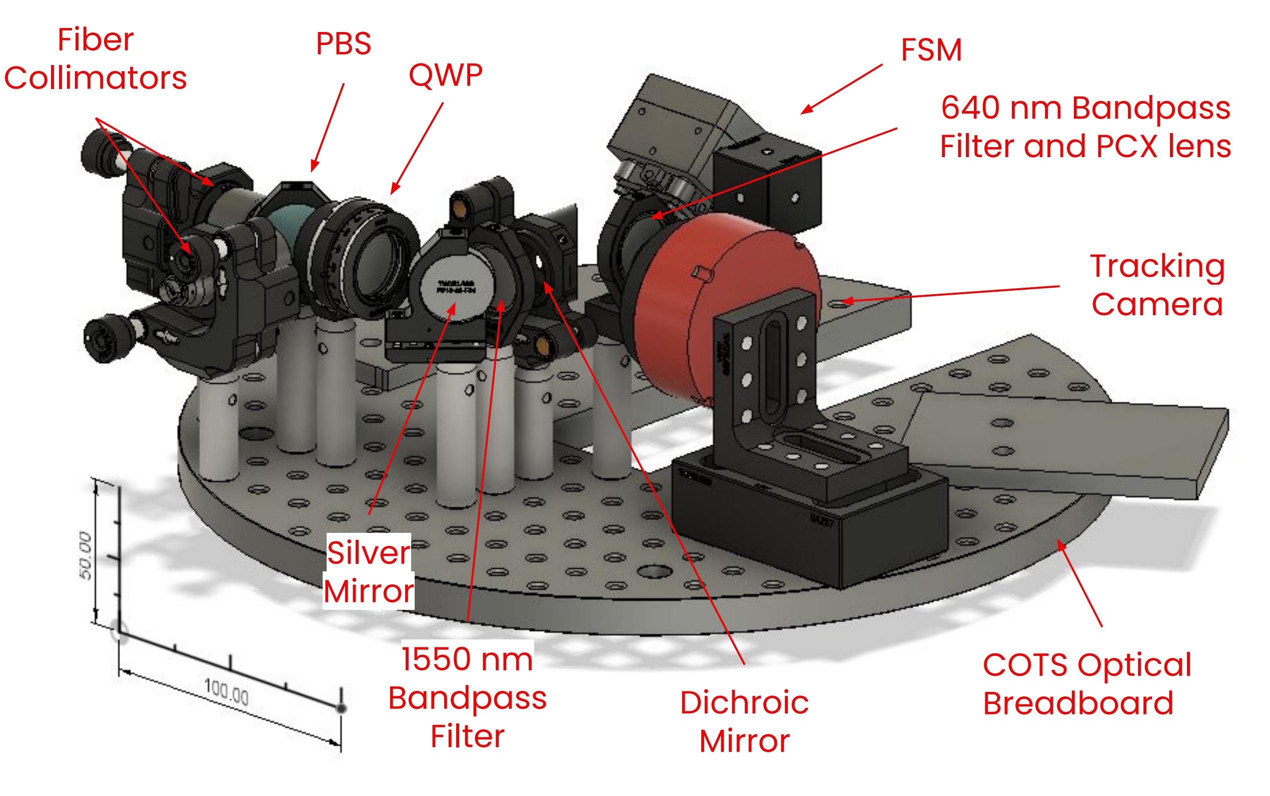
Output:

- ❑ Satellite position inputs are converted to a predicted orbital path which is combined with telescope pointing inputs to output angular slew rates for the telescope and FSM.

Limitations:

- ❑ Predictive pointing must be precise to the FOV of the tracking camera (TBR >2.3 mrad) in order for more precise pointing scheme to come into play
- ❑ The maximum pointing error for the ground station beacon is 1.4 mrad at 1000 km (with ideal divergence angle of 4.17 mrad)

OGS Optical Mounting



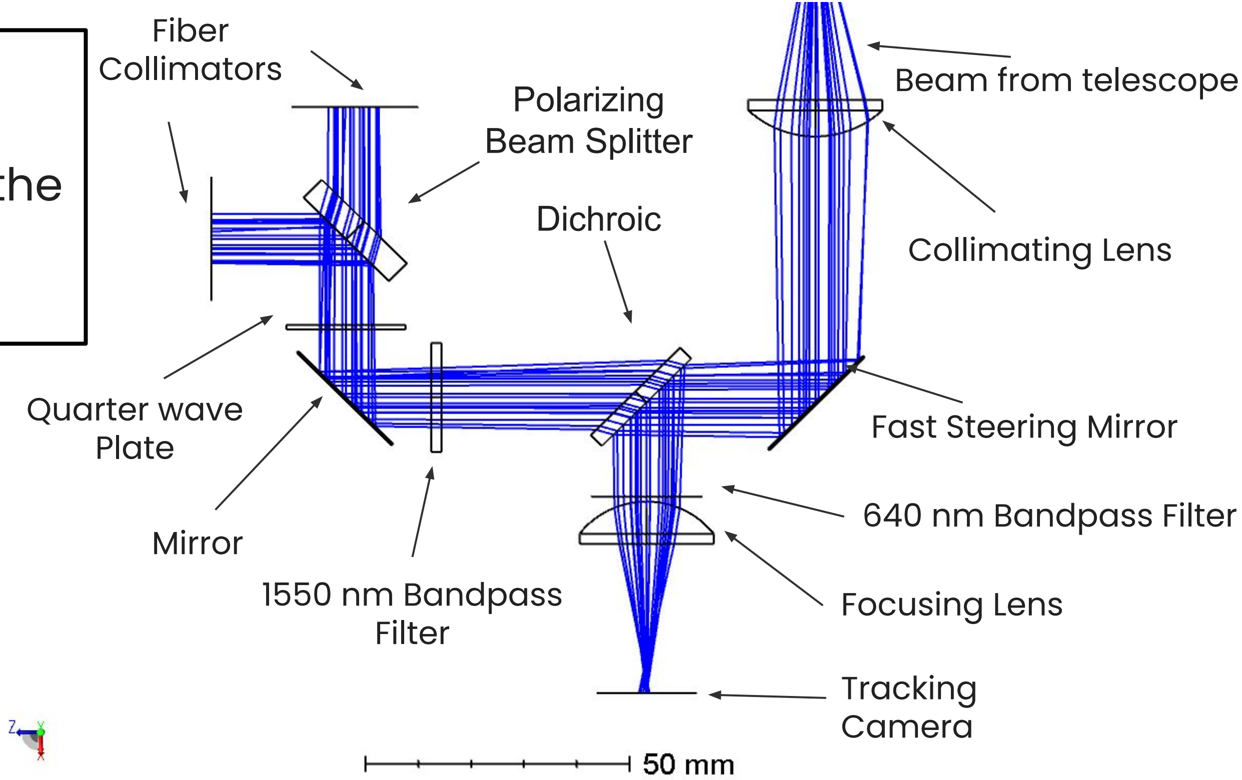
Total Mass:
3.64 kg with breadboard

1.29 kg without breadboard

Entire layout is mounted behind CPC 1100 celestron telescope.

Zemax OpticStudio OGS Simulation Results

View of optical system on the back of the telescope



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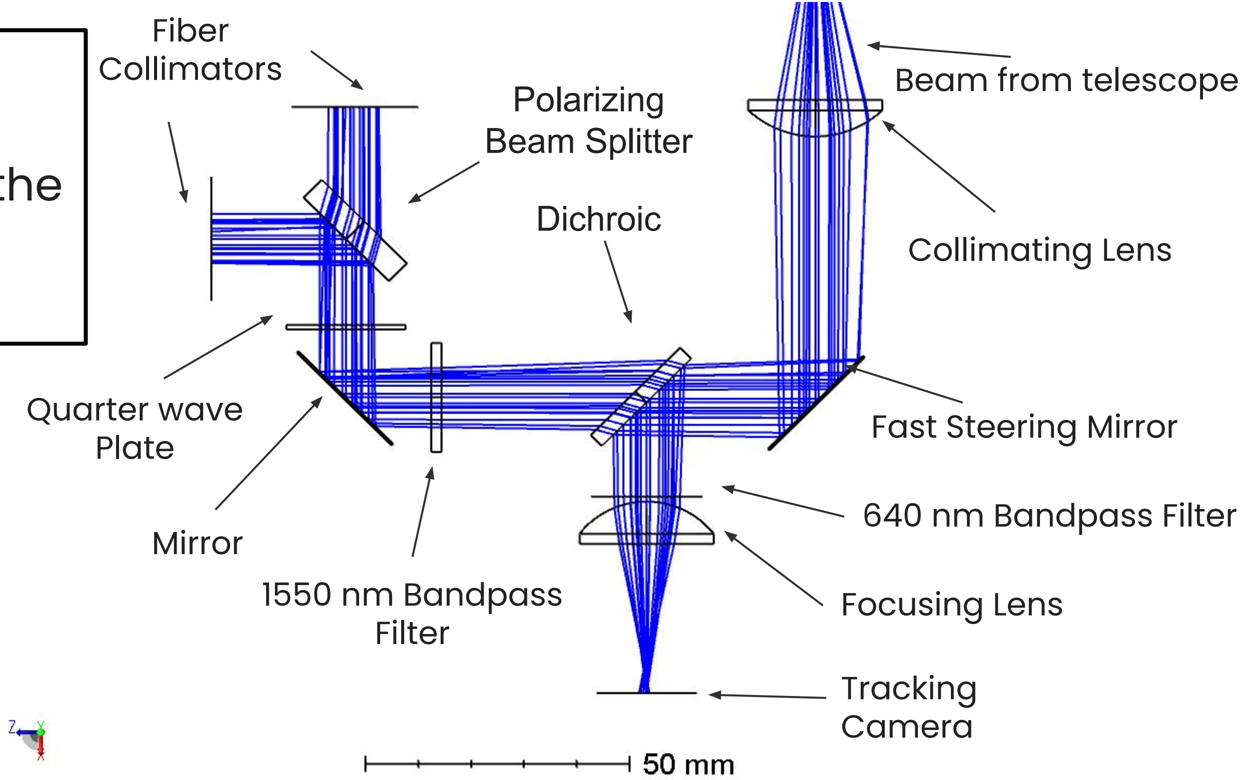
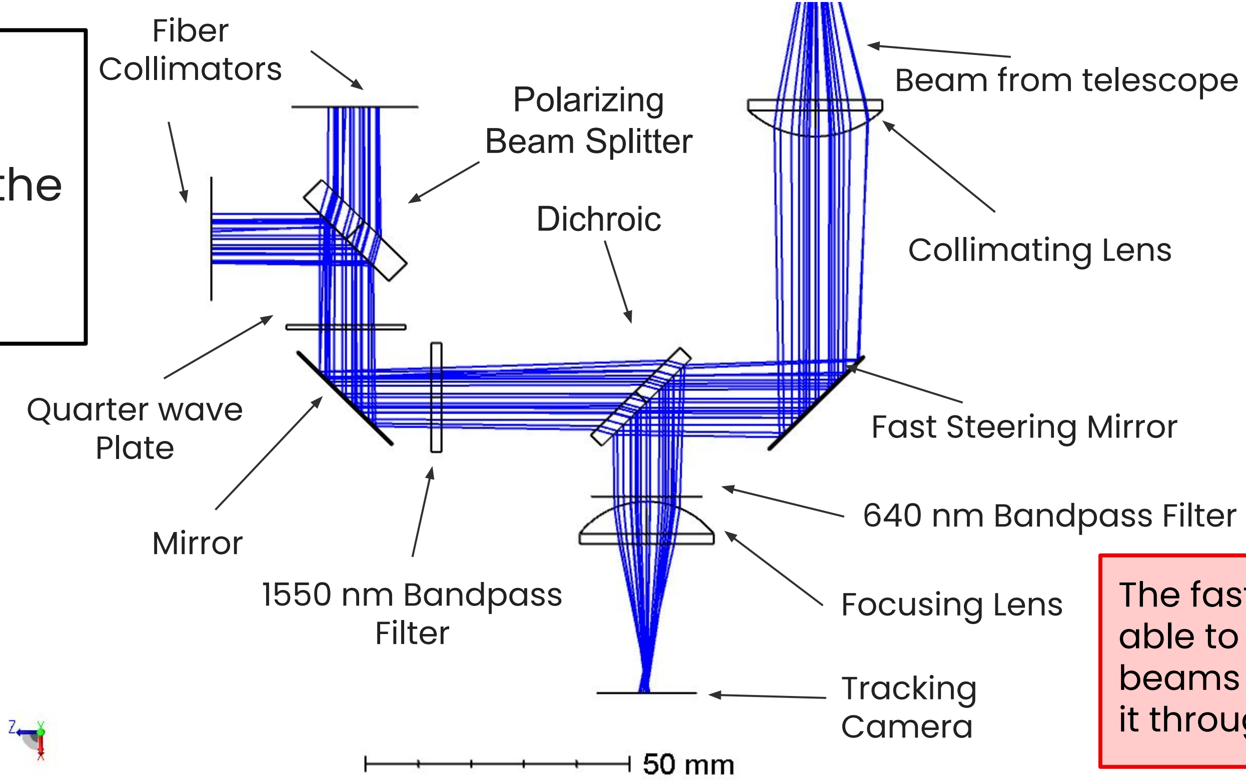
View of optical system on the back of the telescope



The optical system works for small shifts in the angle of incidence of the incoming beam from the telescope

Zemax OpticStudio OGS Simulation Results

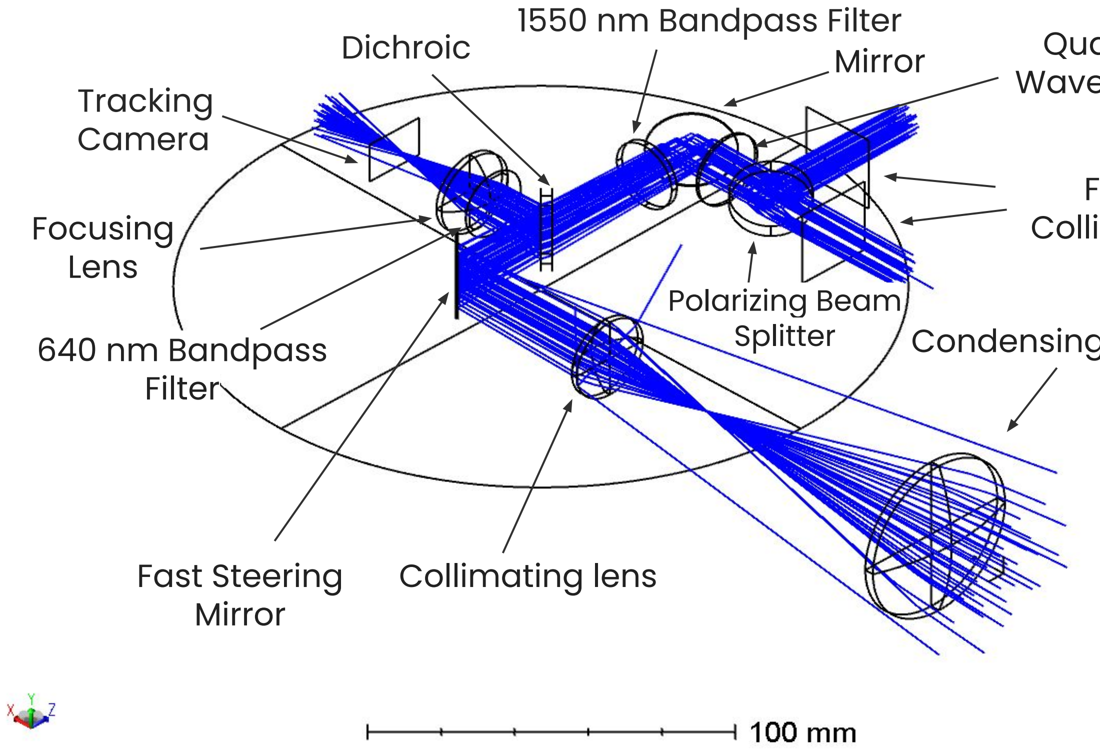
View of optical system on the back of the telescope



The optical system works for small shifts in the angle of incidence of the incoming beam from the telescope

The fast steering mirror is able to adjust off-axis beams so they will make it through the system

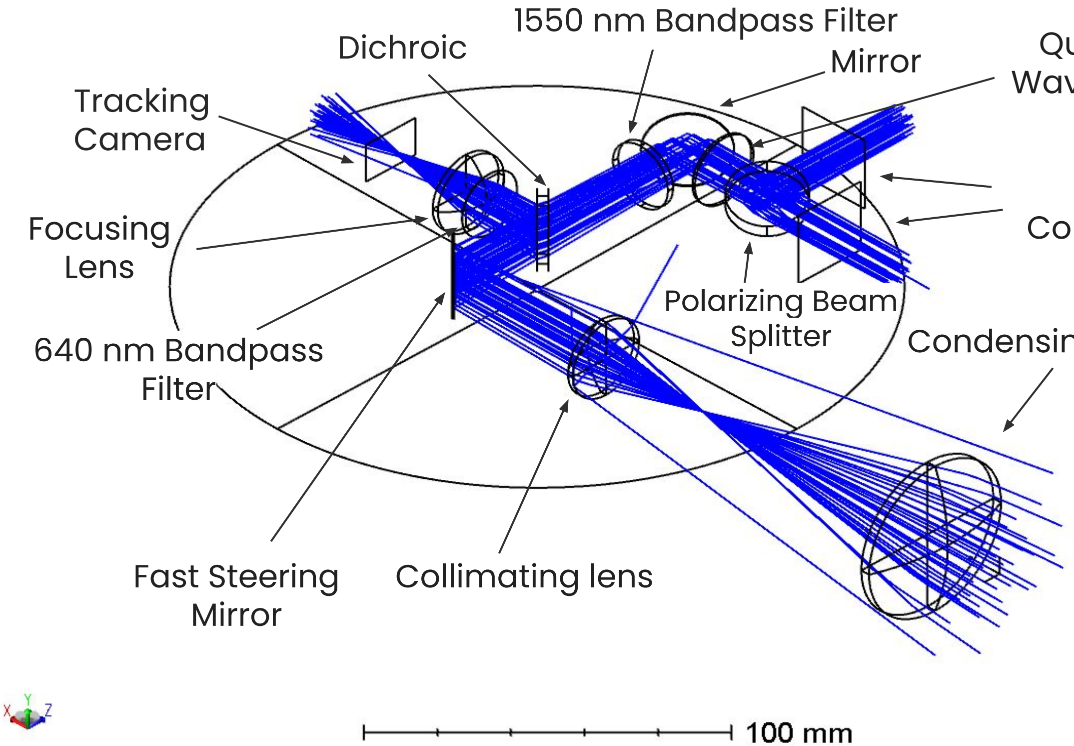
Zemax OpticStudio OGS Simulation Results



Alternate view of optical system on the back of the telescope

Zemax OpticStudio OGS Simulation Results

The optical system can correct for the incident beam (out of telescope) being 1° off axis (TBR)



Alternate view of optical system on the back of the telescope

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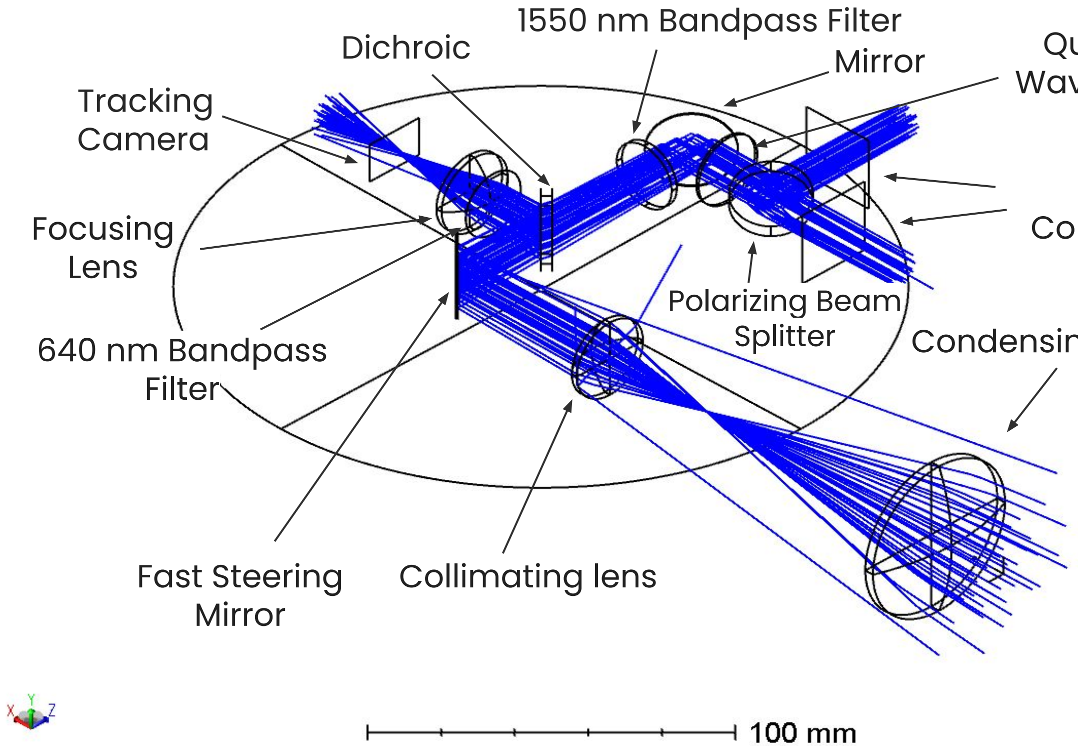

The system can support a relatively large-diameter of beam size.

Alternate view of optical system on the back of the telescope

OGS Beacon Subassembly

Subassembly consists of a low intensity noise, fiber laser at 1064 nm, consisting of a collimated fiber output.

Model	Dimensions (mm)	Spectral Bandwidth (nm)	Output Power (W)	Output Divergence Angle (mrad)
FA-SF-1064-10-CW	240 x 300 x 76	±10	10	2.26

OGS Beacon Subassembly

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FA-SF-1064-10-CW	240 x 300 x 76	±10	10	2.26

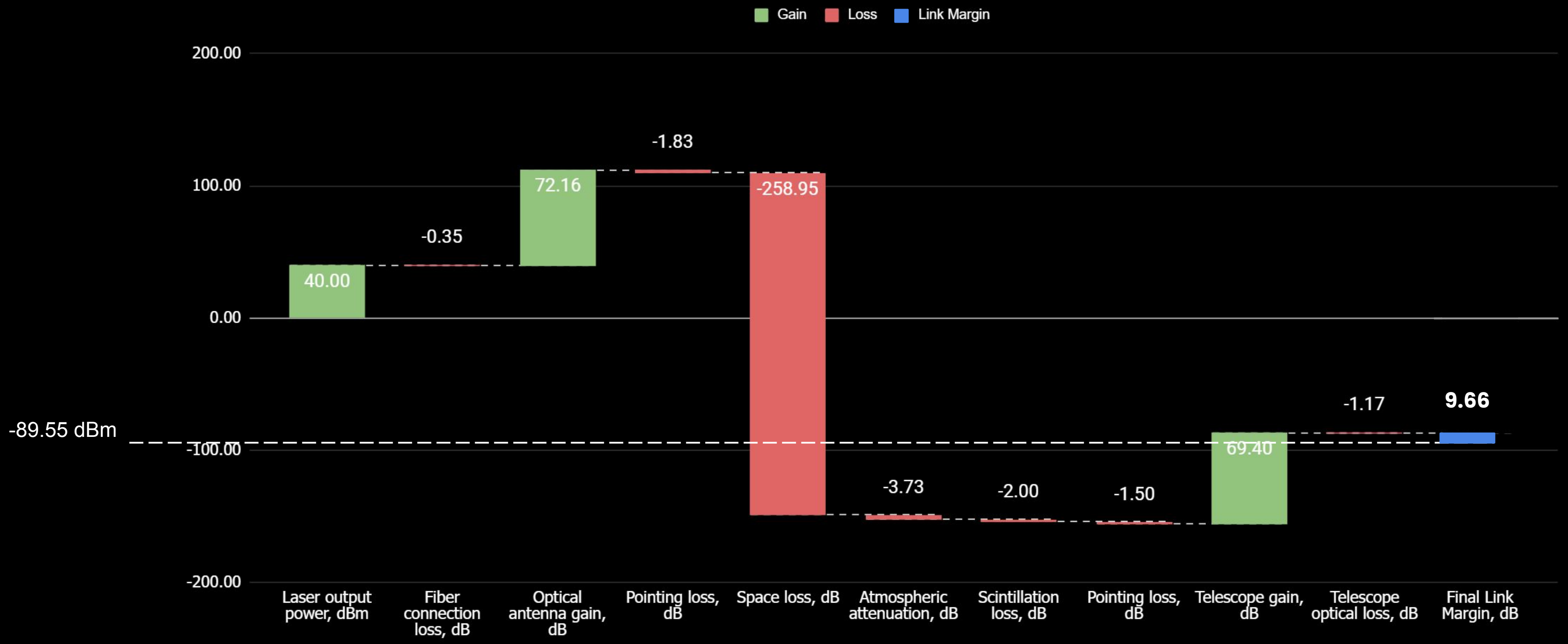
Full divergence angle is not optimal divergence angle found in determining pointing error at worst case scenario. Can potentially be optimal divergence angle outside of worst case scenario. Output power is sufficient to maintain a good SNR at the Payload.

OGS Beacon Optical Link Budget Summary

- High transmit power means visibility (weather) is not a harsh limiting condition on transmission
- SNR significantly above 3dB requirement even with small payload optical antenna (telescope assembly aperture)
- SNR values given at worst case 900 km link

Transmit Power:	10W / 40dBm
Uplink beacon wavelength:	1064 nm
SNR (dB):	9.66
SNR (Gain):	9.24

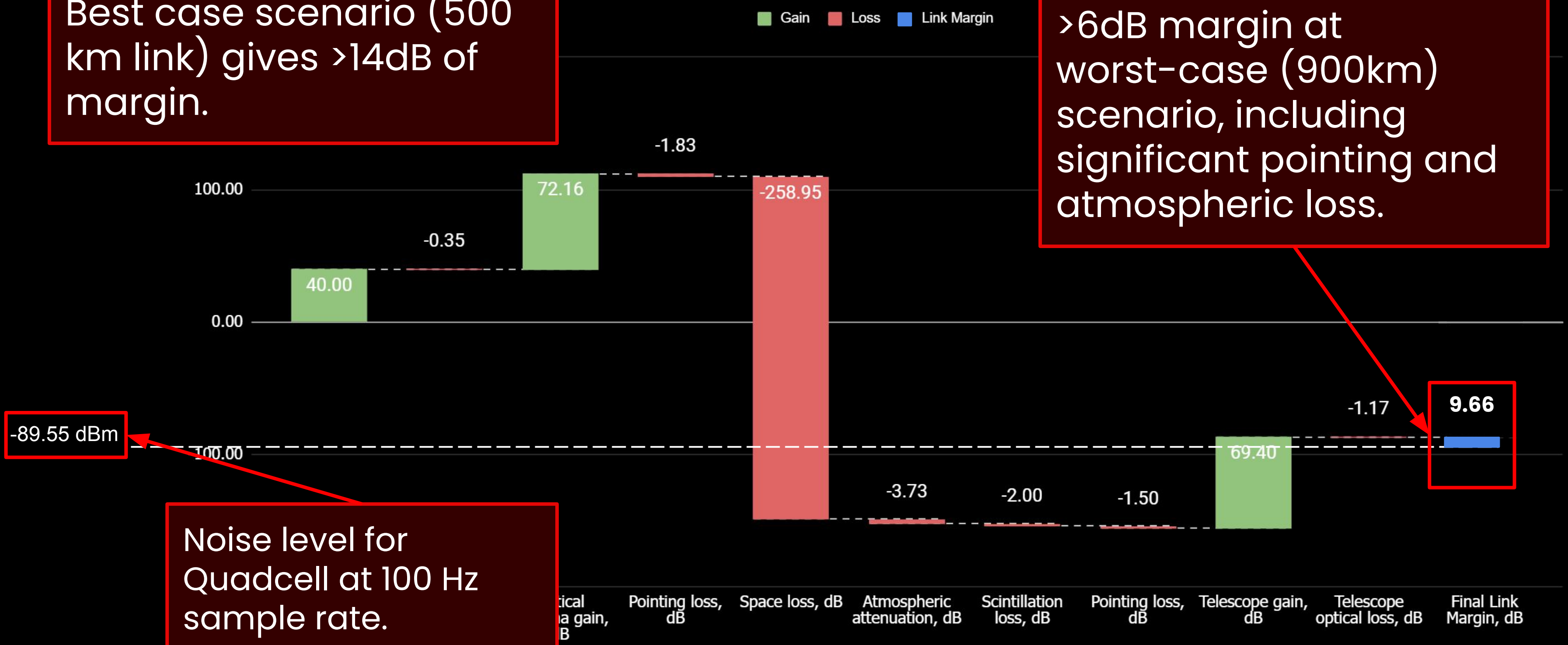
OGS Beacon Optical Link Budget Summary



OGS Beacon Optical Link Budget Summary

Best case scenario (500 km link) gives >14dB of margin.

>6dB margin at worst-case (900km) scenario, including significant pointing and atmospheric loss.



Noise level for Quadcell at 100 Hz sample rate.

Avalanche Photodiode (APD)

The two APD units capture the laser output from the collimators.

Model	Nominal Bandwidth	Noise Equivalent Power (NEP)	Responsivity at 1550 nm	Design Wavelength (nm)
LLAM-1550-R2 AH	50 MHz	130 fW/√Hz	90 kV/W	1550

Avalanche Photodiode (APD)

The two APD units capture the laser output from the collimators.

Low NEP and high responsivity supported by signal to noise ratio calculation. Nominal bandwidth within the requirement for modulation, and allows for faster than modulation sampling.

Model	Nominal Bandwidth	Noise Equivalent Power (NEP)	Responsivity at 1550 nm	Design Wavelength (nm)
LLAM-1550-R2 AH	50 MHz	130 fW/√Hz	90 kV/W	1550

Voltage Amplification Trade Study

Must amplify voltage to be within detection range of comparators / FPGA, while maintaining an appropriate signal to noise ratio.

Model	Gain	Noise Type	Noise	Cost per unit	Bandwidth
Pasternack PE15A1013	50 dB	Noise Figure	1.1 dB	$730	10 MHz to 1 GHz
Femto HLVA-100	80 dB	Noise Equivalent Voltage	2 nV/Hz$^{1/2}$	$4100	100 MHz

Voltage Amplification SNR Comparison

Model	Pasternack PE15A1013	Femto HLVA-100
Input dBm	-54.84	-54.84
SNR (Gain)	2.162	3.435
SNR (dB)	3.348	5.359

- Input dBm determined from transmission link budget
  - Link budget is determined as worst case scenario
- SNR here is likely near minimum possible SNR, meaning both systems will meet requirements
  - Pasternack option only marginally above 3dB requirement (but for worst case)
  - 2 dB is not worth $5000
- Pasternack gain at 50 MHz expected lower than 50 dB (gain bandwidth product)

$$\frac{F(x)}{P(x)} = 1.233$$

→ SNR ratio, Femto 23% higher

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Pasternack meets requirements at a lower cost, hence more favorable although having a worse signal to noise ratio.

$$\frac{F(x)}{P(x)} = 1.233$$

SNR ratio, Femto 23% higher

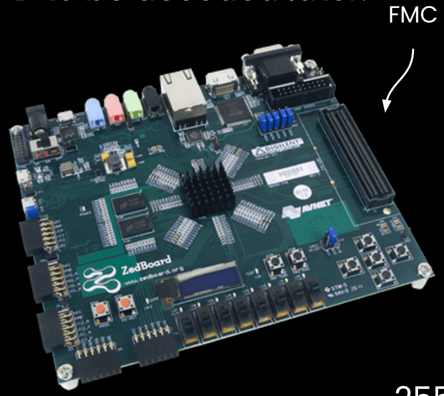
Signal Collection Interface

- Using the Zedboard w/ Zynq-7000 SoC used for high-speed recording of incoming APD signals from a comparator circuit

Amplified and Compared APD Signal $\xrightarrow{\text{FMC Port}}$ FPGA Processor $\xrightarrow{\text{TCP or USB}}$ Error Correction Computer

- The FPGA has a theoretical maximum processing bandwidth of *100MHz*
- Considering TCP or USB protocols for outputting collected data to a computer
  - Feasibility of each protocol needs to be explored but no functional difference
- *No signal processing performed on FPGA.* Directly collects signal from APD to be decoded later.

Model	**FMC Connector Bandwidth**	**Oscillators Frequencies**	**Ethernet Bandwidth**
ZedBoard Zynq-7000	2 GHz	33.333MHz (PS)/ 100MHz (PL)	1Ghz

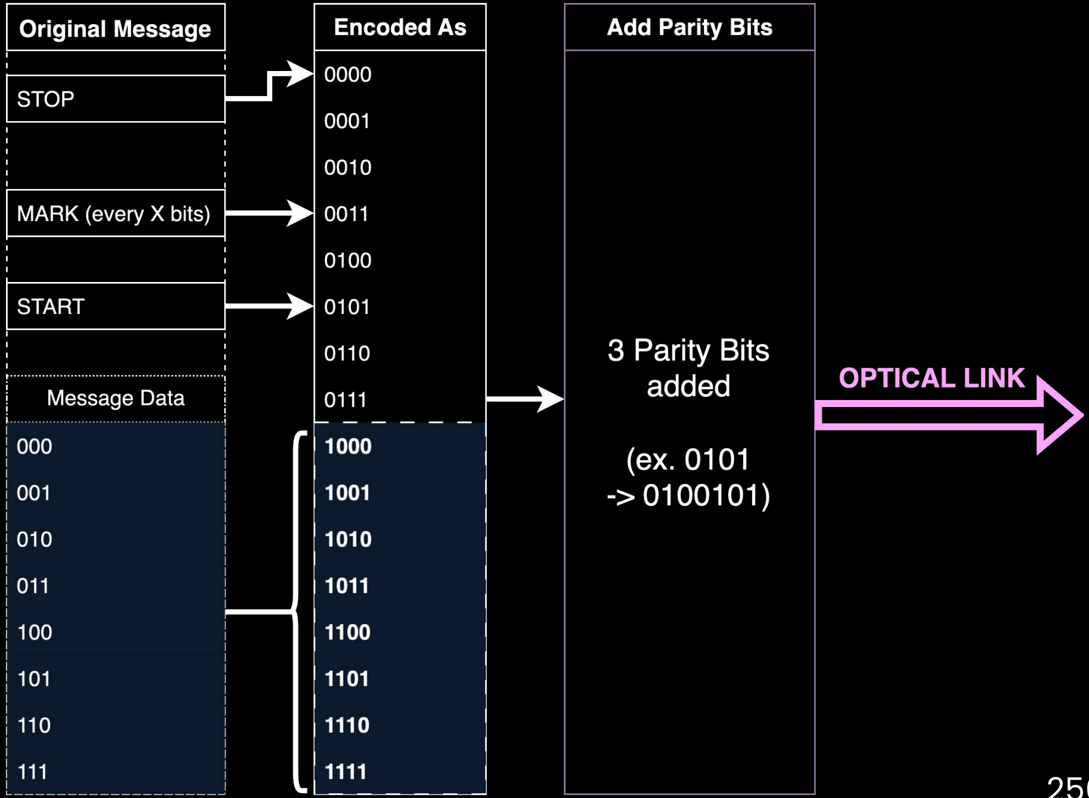


Data Loss Protection & Error Correction - Payload

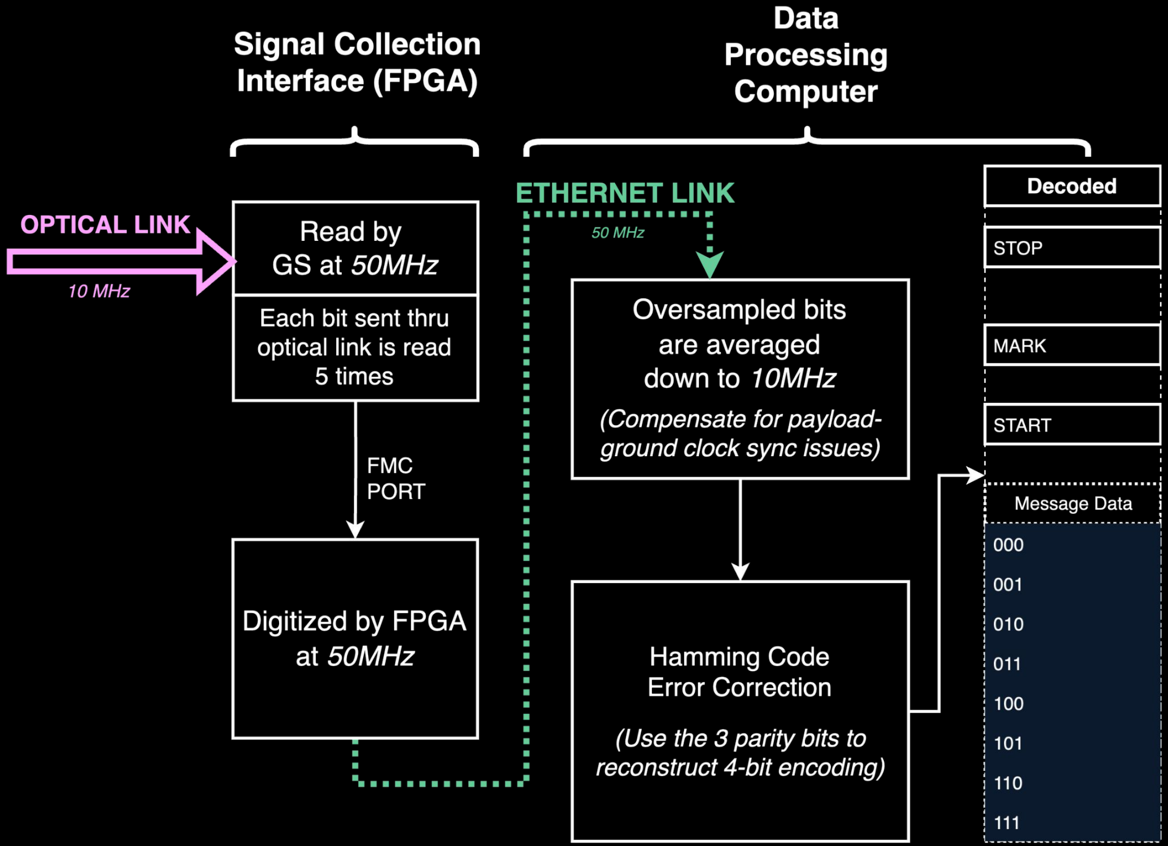
Forward error correction (FEC) to protect against atmospheric/external interference.

A [7,4] Hamming code used as an example of FEC

Can encode for metadata (strings starting with 0) such as start, stop and mark bits. Bit data chunks encoded separately (strings starting with 1).



Data Loss Protection & Error Correction - Ground



The theoretical maximum data rate lower than the modulation rate (3/7 of modulation rate with [7, 4] Hamming code example)

Next Steps for the OGS

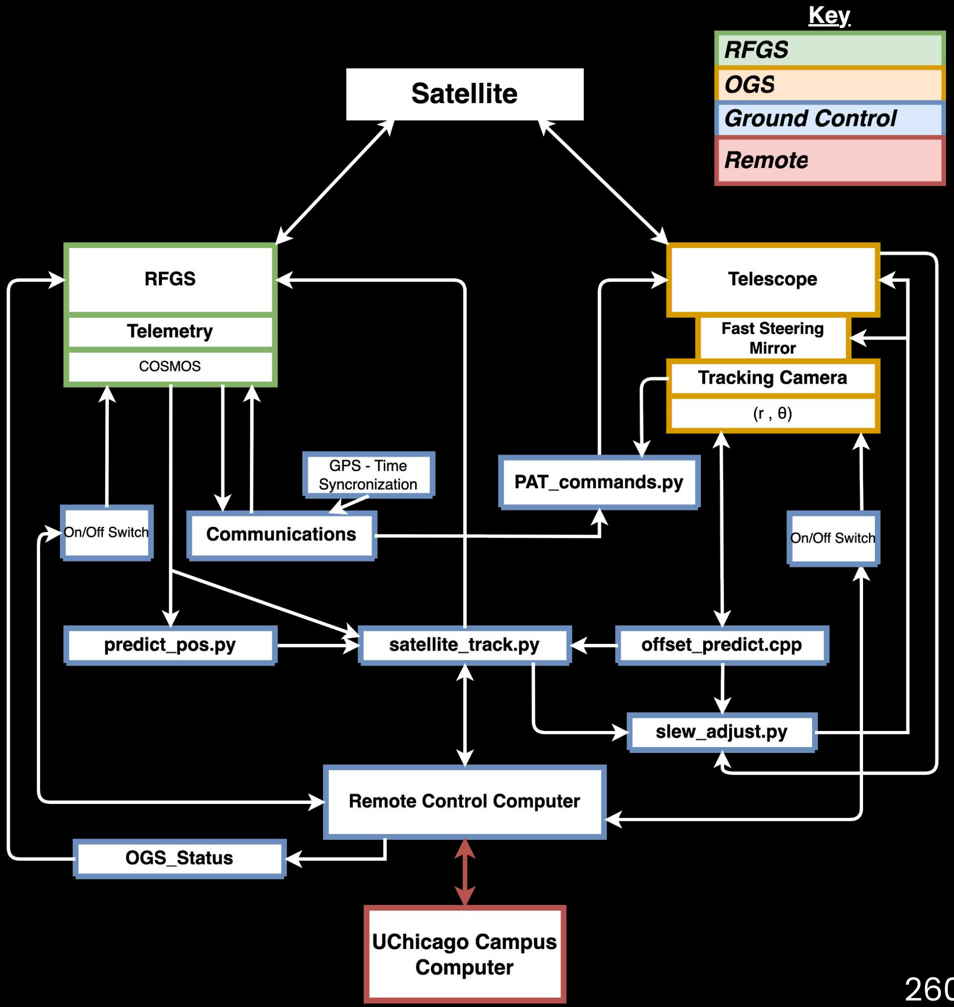
- ❑ Telescope control and pointing testing
  - ❑ Integrate the tracking camera to the telescope
  - ❑ Command through serial connections specific paths across the sky
  - ❑ Track the International Space Station
- ❑ Measure probability distribution of different types of error on circular polarization through the atmosphere
- ❑ Optical bench setup
- ❑ Complete, detailed Zemax simulation
- ❑ Complete mathematical analysis of ground station geometry (to be verified by zemax)

Software Architecture:

Ground Software

Mission Operations

- ❑ Ground Control - OGS interface:
  - ❑ PAT tracking algorithms implemented to update slew and outputting telemetry to be uplinked.
- ❑ Ground Control - RFGS interface:
  - ❑ Uplink and downlink communications and rotator mount slew adjustment.
- ❑ OGS - RFGS interface:
  - ❑ PAT telemetry from both uplink and downlink exchanged between subsystems
- ❑ Ground Control - Campus interface:
  - ❑ Ground control information flow from remote control computer to on-campus mission control.



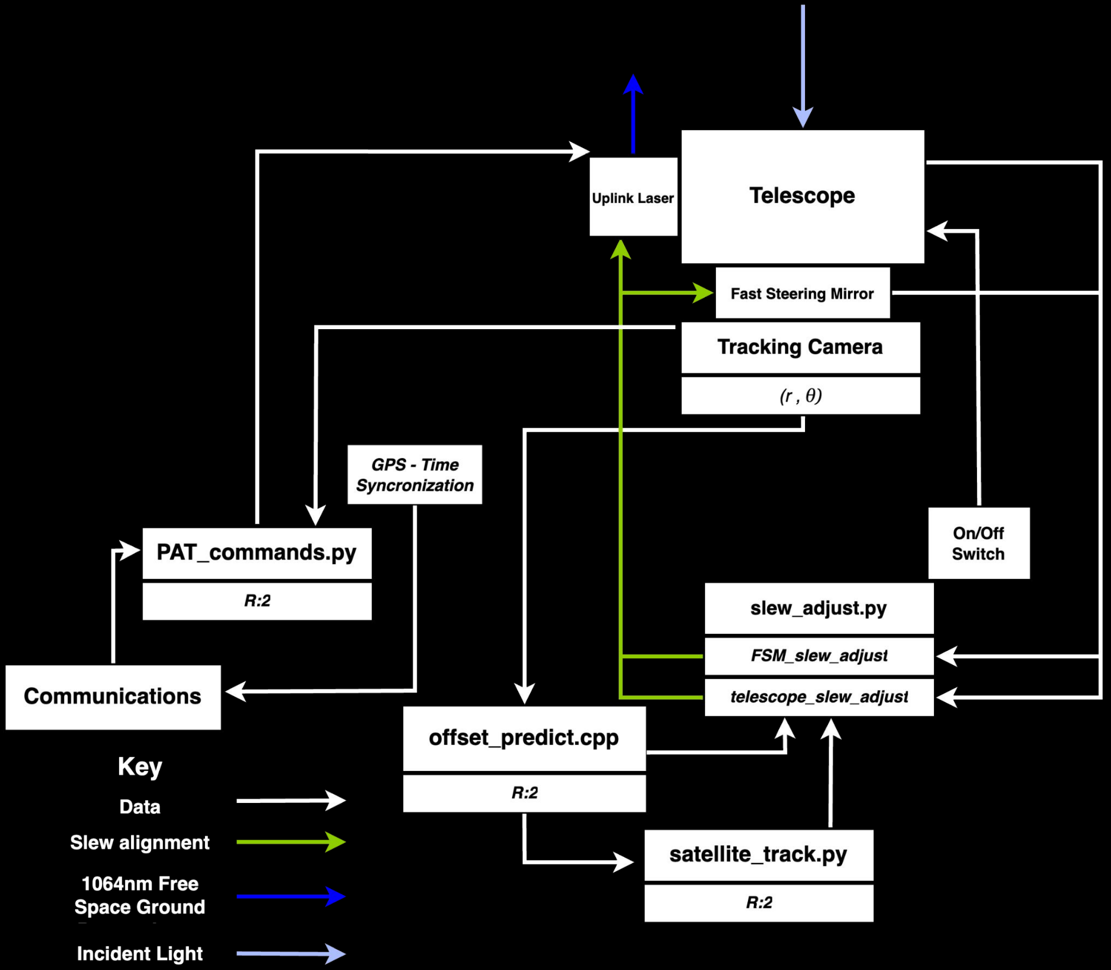
Ground Control - OGS

Communications:

- PAT sequence communications (including OGS beacon enable) are facilitated via RFGS communication and GPS time sync.

Tracking:

- Slew adjustment algorithms are fed data inputs from FSM, Tracking Camera and Telescope and feed commands to the telescope and the FSM accordingly.



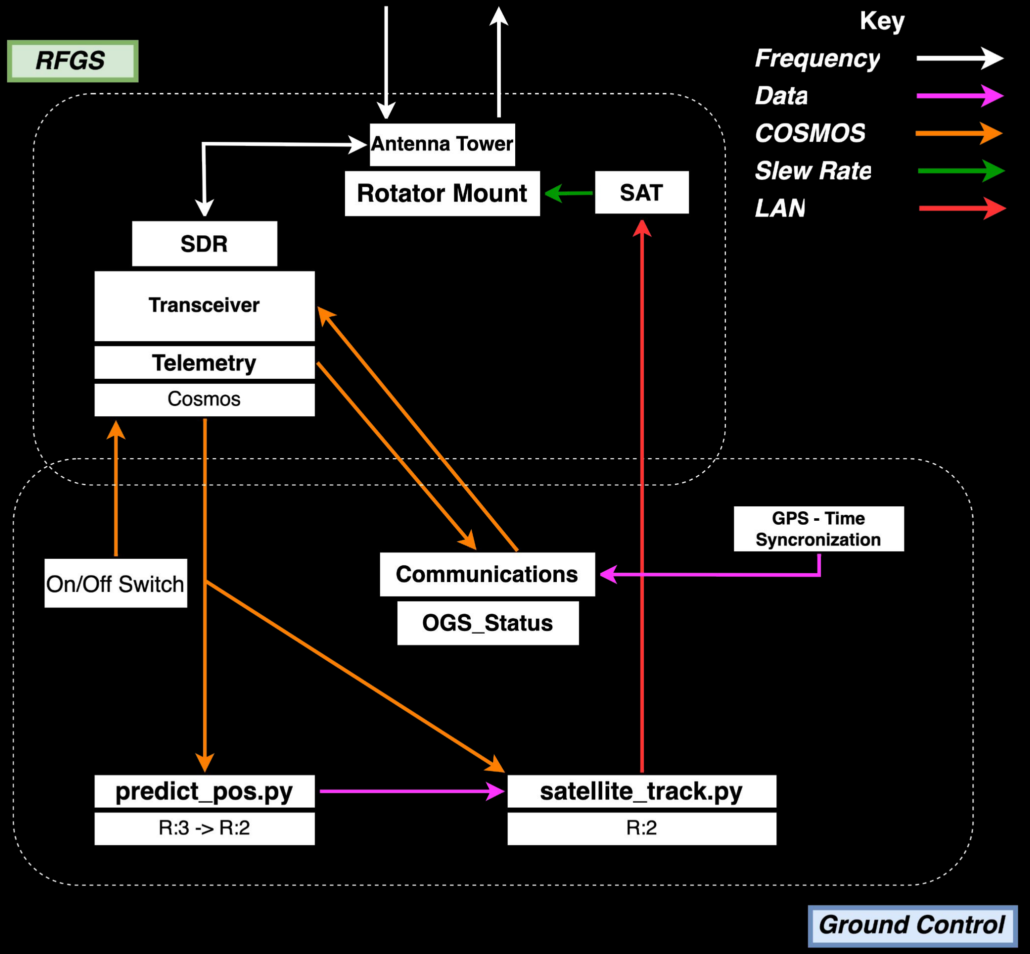
Ground Control - RFGS

Communications:

- GPS used for time synchronization of communications, and uplink and downlink telemetry is exchanged.

Tracking:

- GPS telemetry in two line element format is used to track the satellite (using a predictive / orbit propagation algorithm) and adjust the slew of the RFGS rotator mount.



Next Steps for Ground Software

- ❏ Develop ground control API
  - ❏ Including separate OGS and RFGS interfaces
- ❏ Determine ground control logistics and component level requirements
- ❏ Determine transmission interfaces from ground control to RFGS and OGS
- ❏ Determine data types for data interfaces to subsystems

Software Architecture:

Flight Software

CDH Driving Requirements

ID	Requirement	Notes	Parent	Verification
CDH-01	The CDH shall be capable of processing data from other subsystems.		SAT-04	Demonstration
CDH-02	The CDH shall be capable of sending data to other subsystems.		SAT-04	Demonstration
CDH-03	The CDH shall be capable of commanding other subsystems.		SAT-04	Demonstration
CDH-04	The CDH will be able to detect system faults and take corrective actions.		SAT-04	Demonstration
CDH-08	The CDH shall perform computations on stored data.	Includes processes that store, compress, or alter data.	SAT-04	Analysis
CDH-09	The CDH shall execute the FSW.		SAT-04	Demonstration

Legend: Compliant Compliant by CDR Compliant by TRR Compliant By FRR

FSW Driving Requirements

ID	Requirement	Notes	Parent	Verification
FSW-01	The FSW shall implement a portable software framework.	cFS has been selected as the software framework.	SAT-04	Demonstration
FSW-03	The FSW shall provide a stable environment for software modules to run.		SAT-04	Demonstration
FSW-04	The FSW modules shall operate over a common software bus.	A CAN bus has been selected.	SAT-04	Inspection
FSW-07	The FSW watchdog processor shall take corrective action when process faults are detected		CDH-04	Demonstration
FSW-08	The FSW shall synchronize the real-time clock with a time standard.		CDH-12	Demonstration
FSW-09	The FSW shall be re-programmable by access ports and over-the-air updates.		SAT-04, SAT-13	Demonstration

Legend: Compliant Compliant by CDR Compliant by TRR Compliant By FRR

cFS vs F Prime

Framework Name	Core Flight System	F Prime
Developed By	NASA Goddard	JPL
Documentation	❏ Longer reference documentation ❏ cFS-basecamp tutorial guides	❏ More integrated documentation ❏ More early stage guides
Contacts	❏ David McComas (34 years FSW Engineer NASA Goddard) ❏ Met with David multiple times	❏ No ongoing contacts, but have been connected with Michael Starch (JPL)
CCSDS Support	❏ Built-in CCSDS SPP and CFDP integration	❏ Some support of CCSDS protocols but not full integration currently

cFS vs F Prime

Framework Name	Core Flight System	F Prime
Complexity Level	❑ Considered by FSW community to be more complex	❑ Considered to be simpler, easier to use, and sufficient for most university CubeSat missions
Programming Language	❑ Uses C ❑ High knowledge of C at University of Chicago	❑ Uses C++ ❑ C++ not included in core CS classes
Ground Station Software	❑ Integration with COSMOS and AIT	❑ Integration with F Prime Ground Data System
Development Environment	❑ Requires Linux environment	❑ Can be developed on MacOS

cFS vs F Prime

Framework Name	Core Flight System	F Prime
Module Communication	❏ Pub-Sub software bus	❏ 1 to 1 communication
Platform Approach	❏ Full OS abstraction through OSAL ❏ Easy abstraction process	❏ Relies on modular design and cross-compilation ❏ May require low level changes
Testing	❏ cFS Test Framework	❏ Code generation tool for unit test creation
Software Patching	❏ Supports hot patching	❏ Typically requires restart to patch software ❏ Possible to develop hot patching support

cFS vs F Prime

Framework Name	Core Flight System	F Prime
Module Communication	❏ Pub-Sub software bus	❏ 1 to 1 communication
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Core Flight System was chosen while acknowledging both frameworks have key benefits because we felt our relationship with Dave and the strong knowledge of C compared to C++ were the most important criteria.

RTEMS vs VxWorks vs FreeRTOS vs Embedded Linux

OS Name	RTEMS	VxWorks	FreeRTOS	Embedded Linux
Real Time	❏ Native	❏ Native	❏ Native	❏ Can be added with PREEMPT_RT
File System	❏ RFS, IMFS, and FAT natively supported	❏ DOSFS, HRFS, and NFS natively supported	❏ No native support ❏ FAT support can be added	❏ ext2/ext3/ext4, FAT32m and many more supported natively
License	❏ Open Source GPL	❏ Not open source ❏ Free educational license	❏ Open Source GPL	❏ Open Source GPL

RTEMS vs VxWorks vs FreeRTOS vs Embedded Linux

OS Name	RTEMS	VxWorks	FreeRTOS	Embedded Linux
Documentation	❏ Known cFS, BeagleBone Black, and RTEMS demo	❏ Does not have BeagleBone Black documentation	❏ Has BeagleBone Black documentation	❏ Has BeagleBone Black documentation
Memory	❏ Low memory usage	❏ Moderate memory usage	❏ Low memory usage	❏ High memory footprint
Development Environment	❏ Basic development tools	❏ Strong development tools	❏ Basic IDE support	❏ Robust development tools widely supported

RTEMS vs VxWorks vs FreeRTOS vs Embedded Linux

OS Name	RTEMS	VxWorks	FreeRTOS	Embedded Linux
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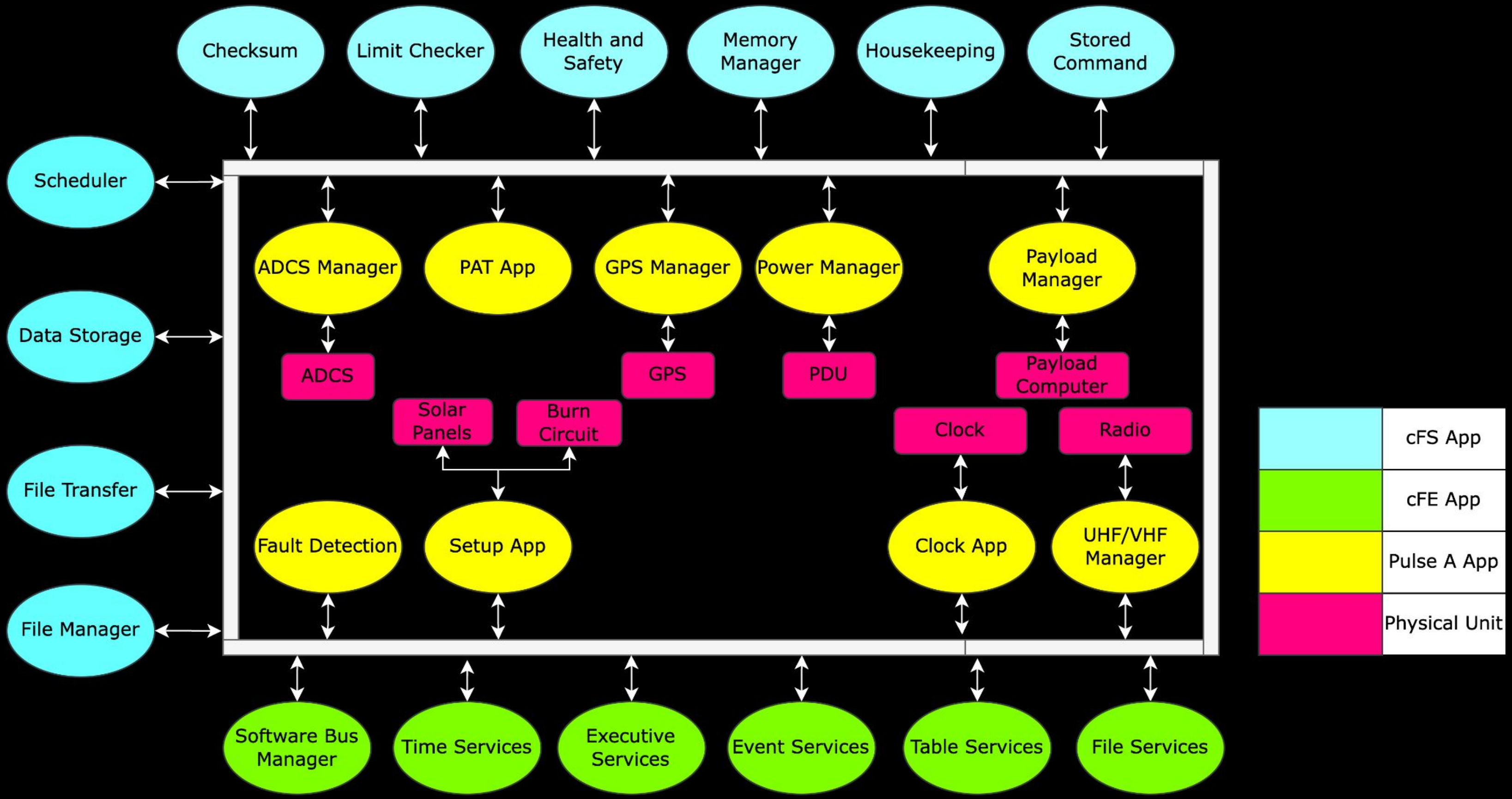
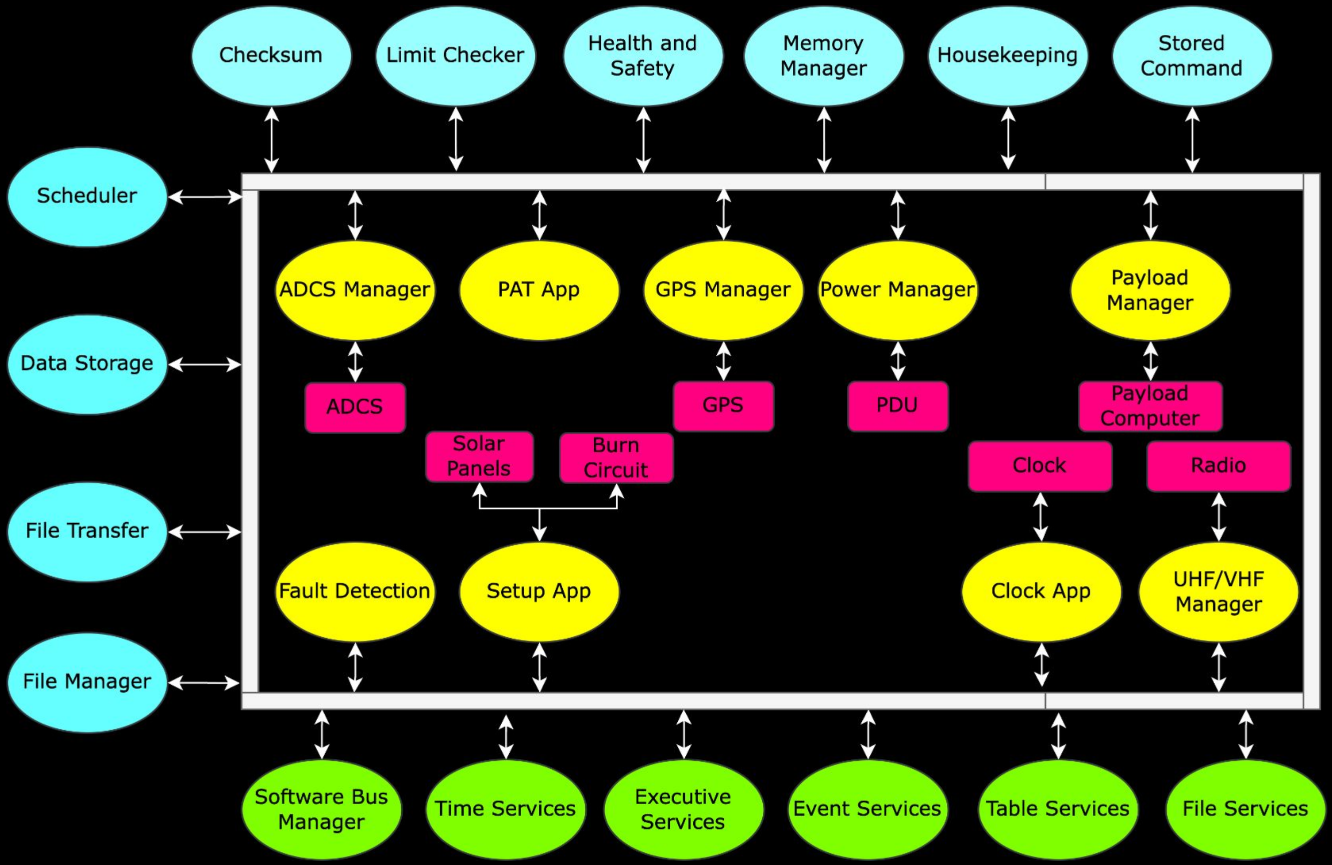
Memory	❏ Low memory usage	❏ Moderate memory usage	❏ Low memory usage	❏ High memory footprint
Development Environment	❏ Basic development tools	❏ Strong development tools	❏ Basic IDE support	❏ Robust development tools widely supported

RTEMS was chosen for being lightweight with real time features and having documentation for our specific hardware configuration (Beaglebone Black).

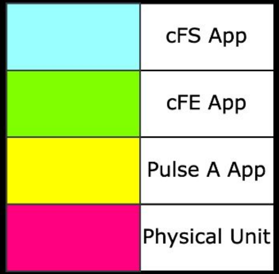
OBC Software Architecture



OBC Software Architecture



Planned apps to use and develop for the OBC running on a common software bus to manage general operations of the satellite

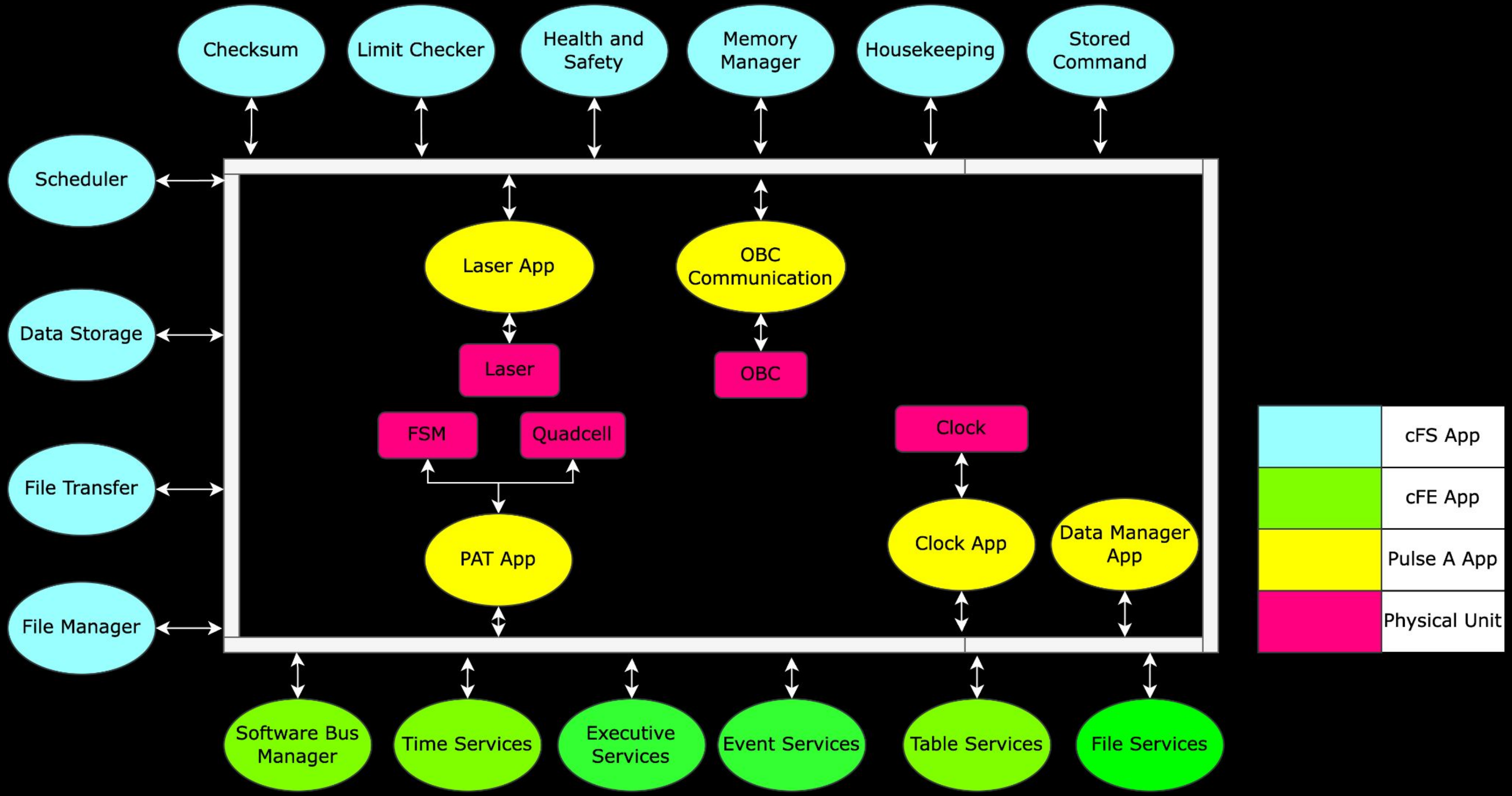


Encryption/Decryption and Authentication

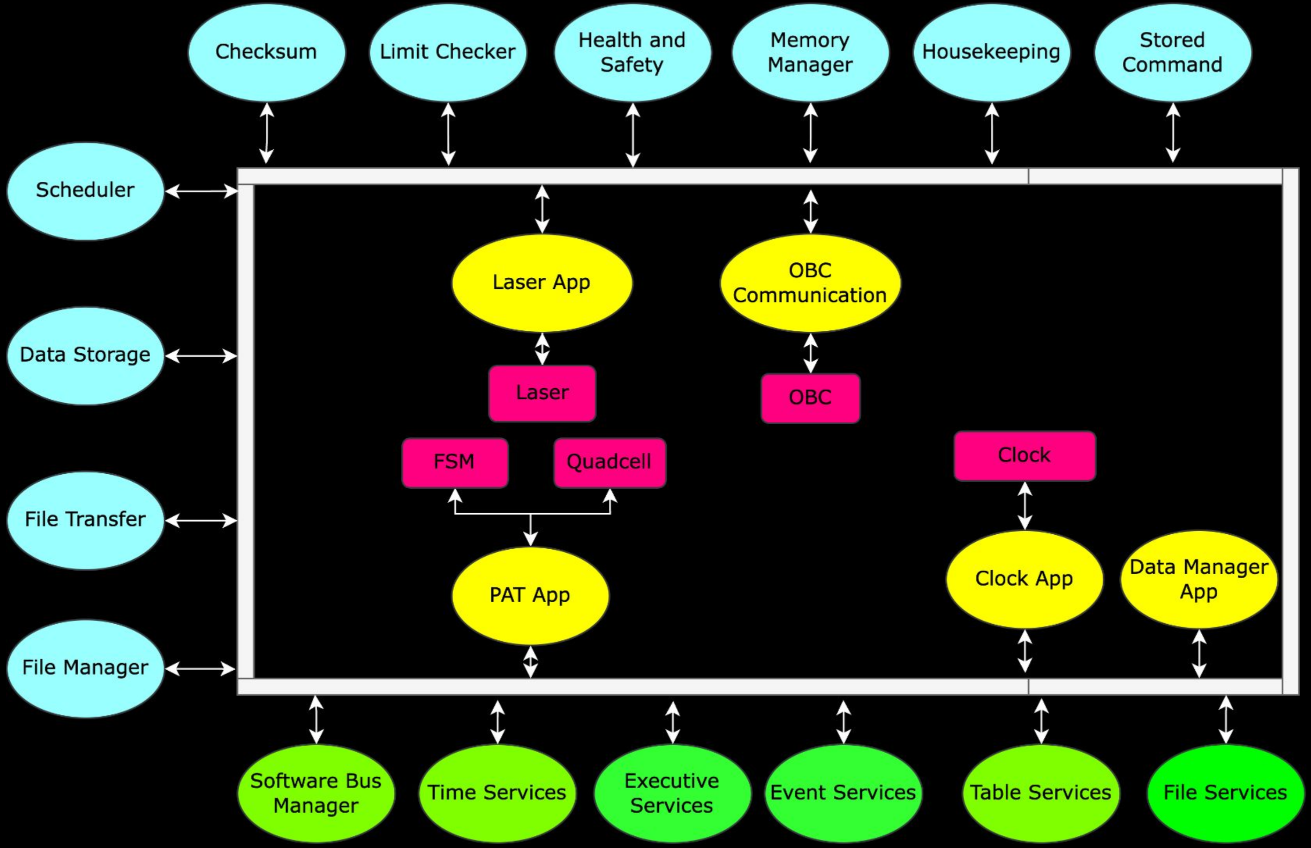
- Important to implement due to laser onboard satellite and to prevent general pirating of satellite
- Adhere to CCSDS Cryptographic Algorithms Space Data Link Security Protocol recommendations
- Implement encryption and authentication with CryptoLib integrated into cFS and the ground station software
- Authenticated Encryption Scheme Galois/Counter Mode, O(n) time complexity
- CryptoLib implements mechanisms to ensure protection against replay attacks

CryptoLib

Payload Controller Software Architecture



Payload Controller Software Architecture

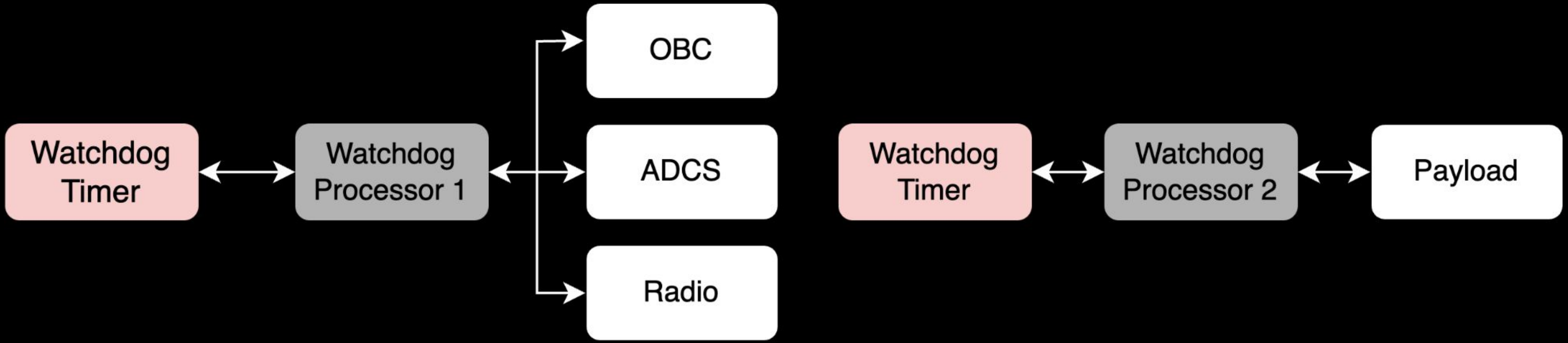


Planned apps to use and develop for the Payload Controller running on a common software bus to manage the payload of the satellite

(cyan)	cFS App
(green)	cFE App
(yellow)	Pulse A App
(magenta)	Physical Unit

Watchdog Requirement Implementation

- Each Watchdog Processor (MSP430FR2433) contains an internal Watchdog Timer.
- Each Watchdog device maintains a counter incremented every CPU cycle. Child devices reset this counter to 0 with heartbeat signals. Upon this counter overflow, a reboot command is issued to the child device.
- Watchdog Processor will reboot all devices periodically at a variable time interval.

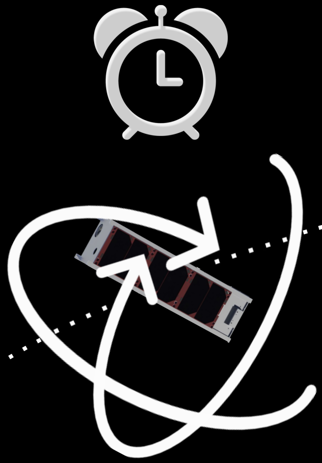


Software States

1. Initial Startup
2. Detumble
3. Deployment
4. Sun Tracking
5. RF Receive
6. RF Transmit
7. Restart
8. Safe Mode
9. PAT Sequence

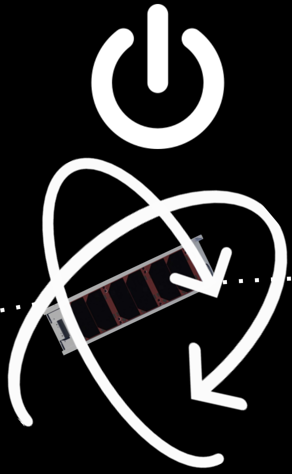
Detailed State Diagrams

Initial Startup State



- ❑ Satellite exits deployer tumbling
- ❑ Satellite waits for time defined by launch provider

Detumble State

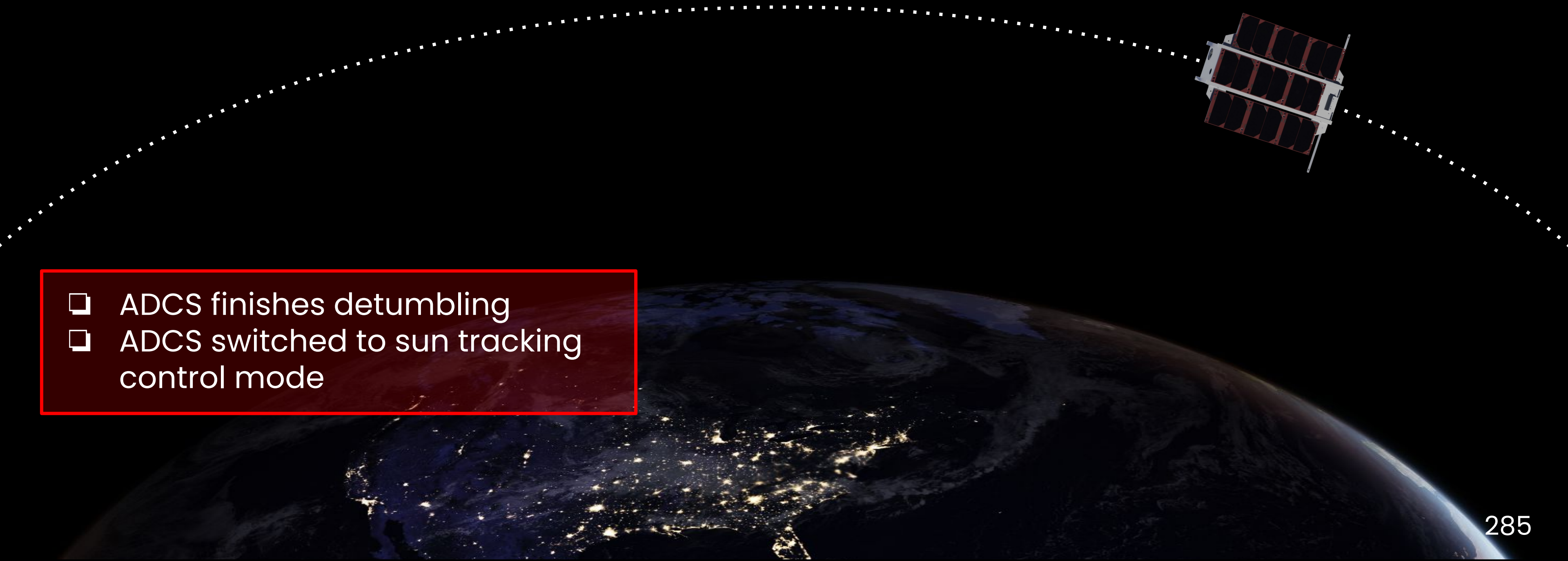


- After time defined by launch provider satellite powers on
- ADCS changes control mode to detumble

Detumble State

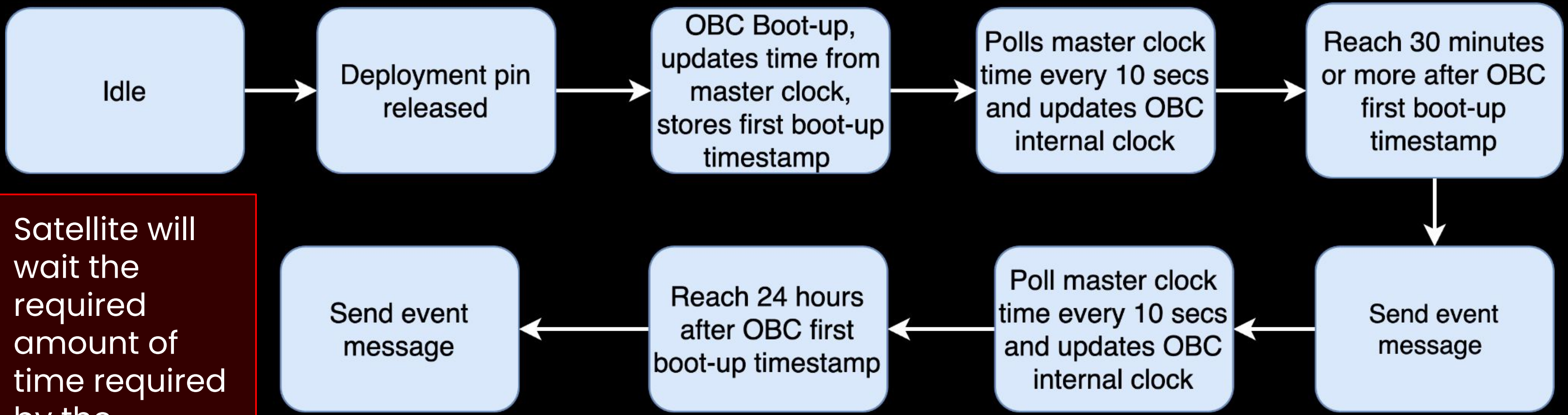
- ❑ ADCS finishes detumbling
- ❑ ADCS switched to sun tracking control mode

Deployment State



- ❑ ADCS finishes detumbling
- ❑ ADCS switched to sun tracking control mode

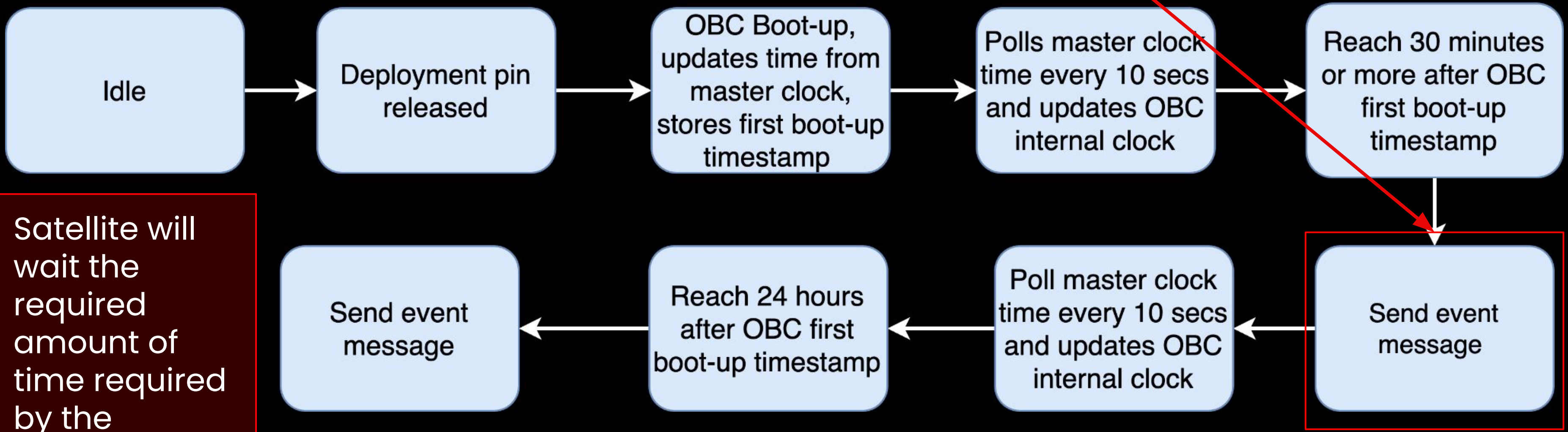
Initial Startup State



Satellite will wait the required amount of time required by the deployer before beginning operations

Initial Startup State

First event message to start detumbling after 30 minutes



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Initial Startup State

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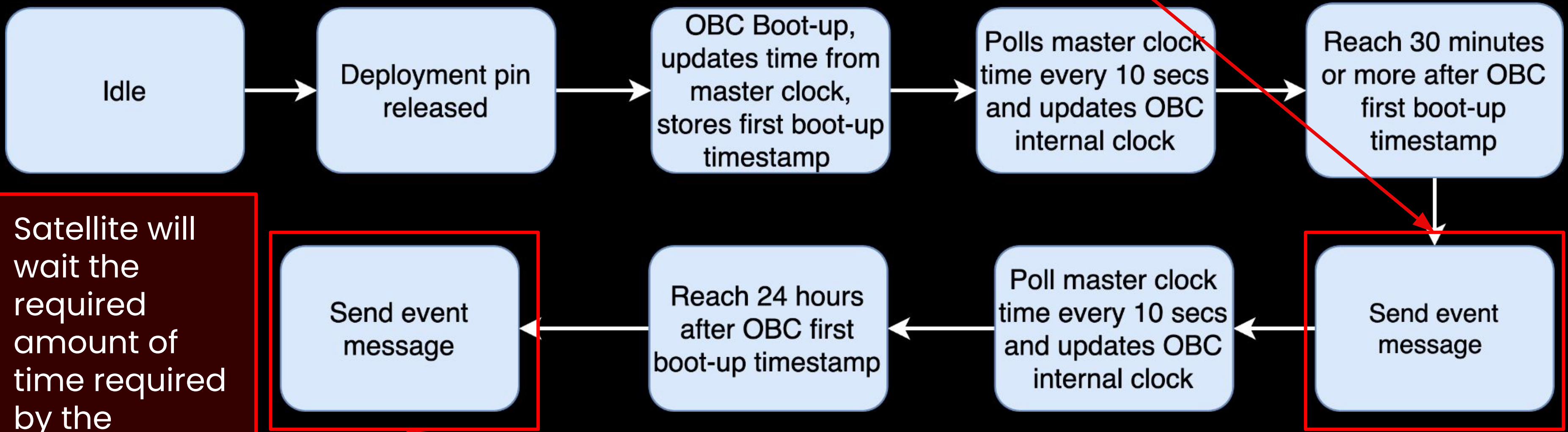
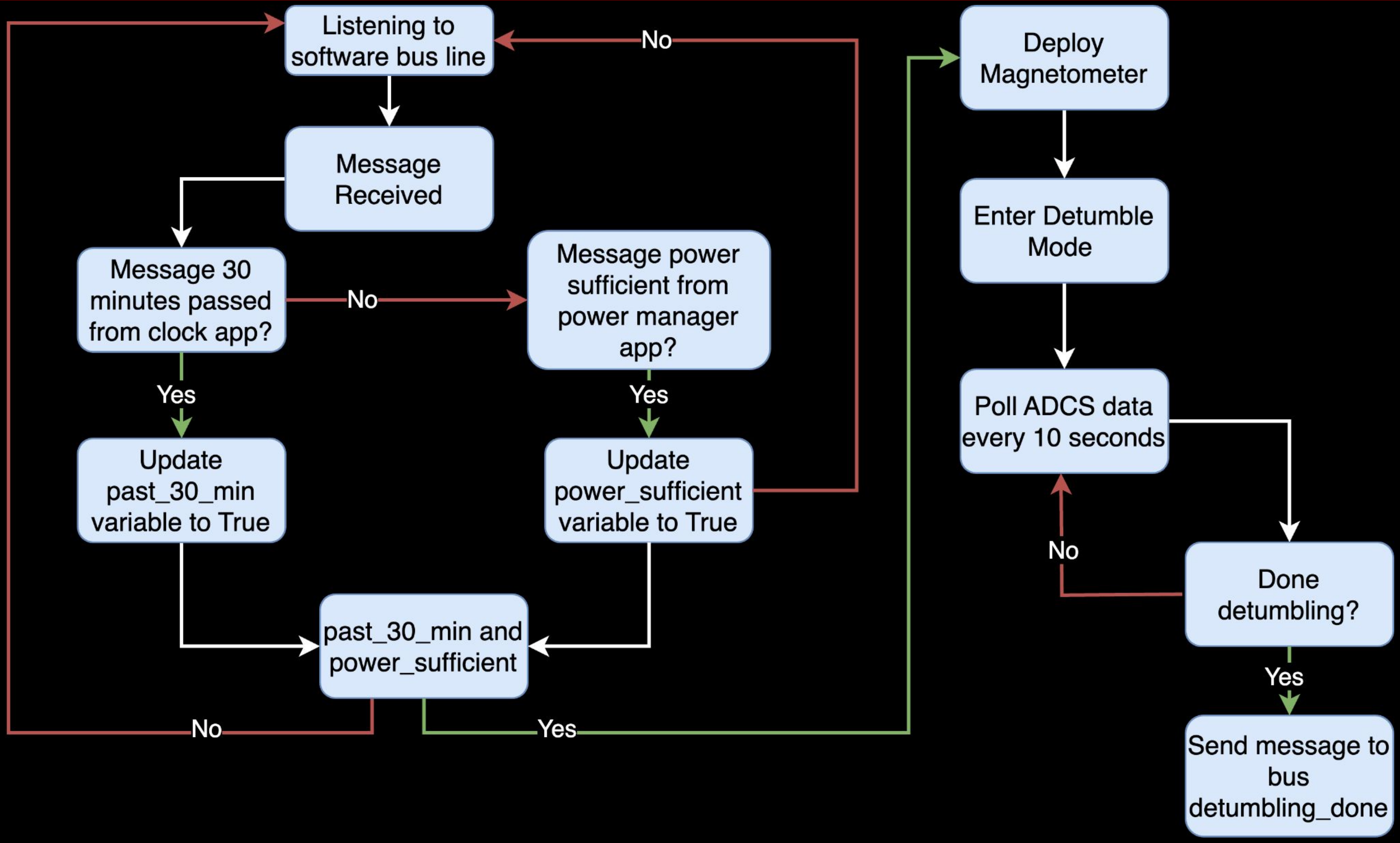


Satellite will wait the required amount of time required by the deployer before beginning operations

Backup message after 24 hours to ensure deployment in case of detumble failure

Detumble State

Satellite will start detumbling once at least 30 minutes have passed and the satellite has sufficient power



Detumble State

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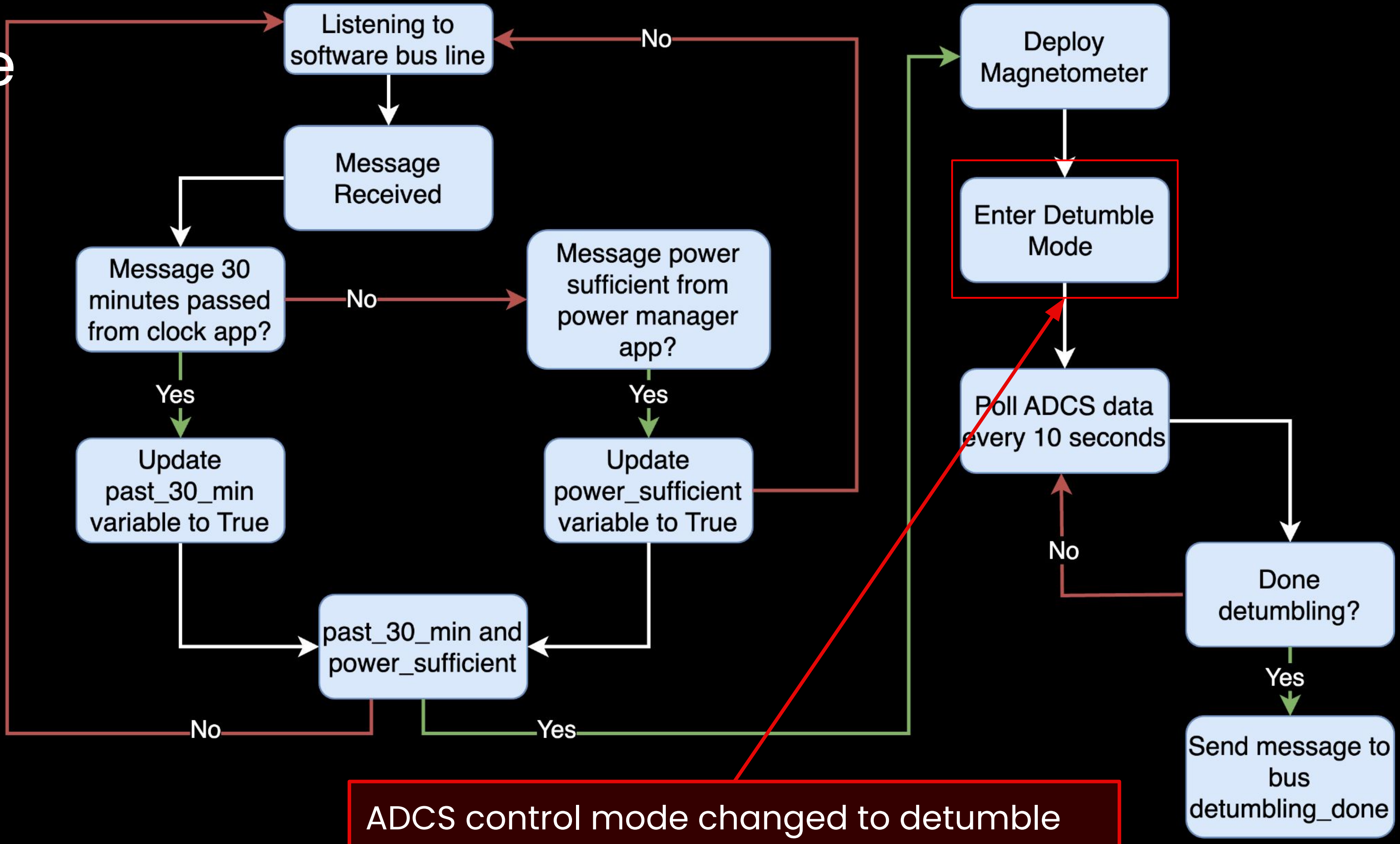
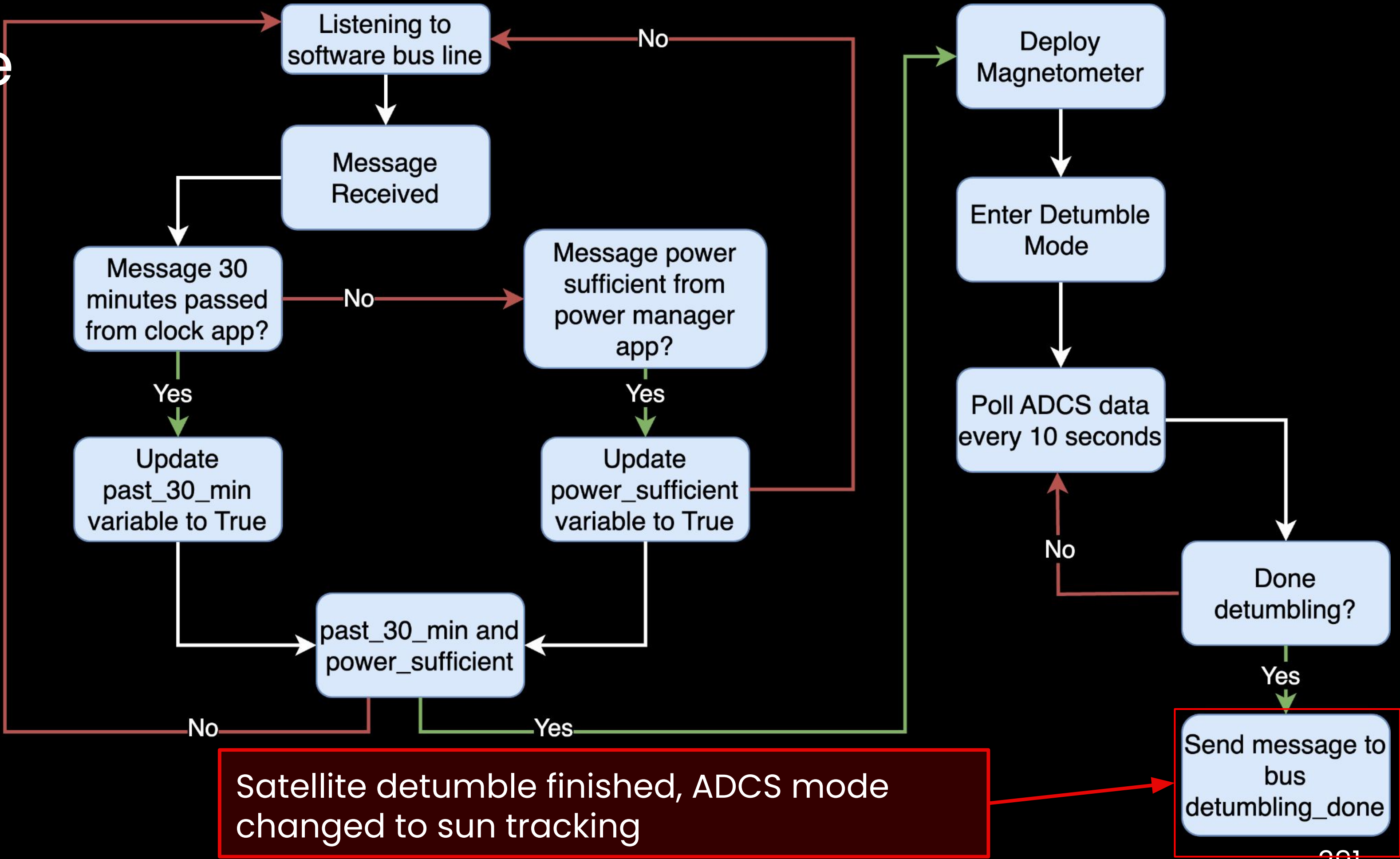


ADCS control mode changed to detumble

Detumble State

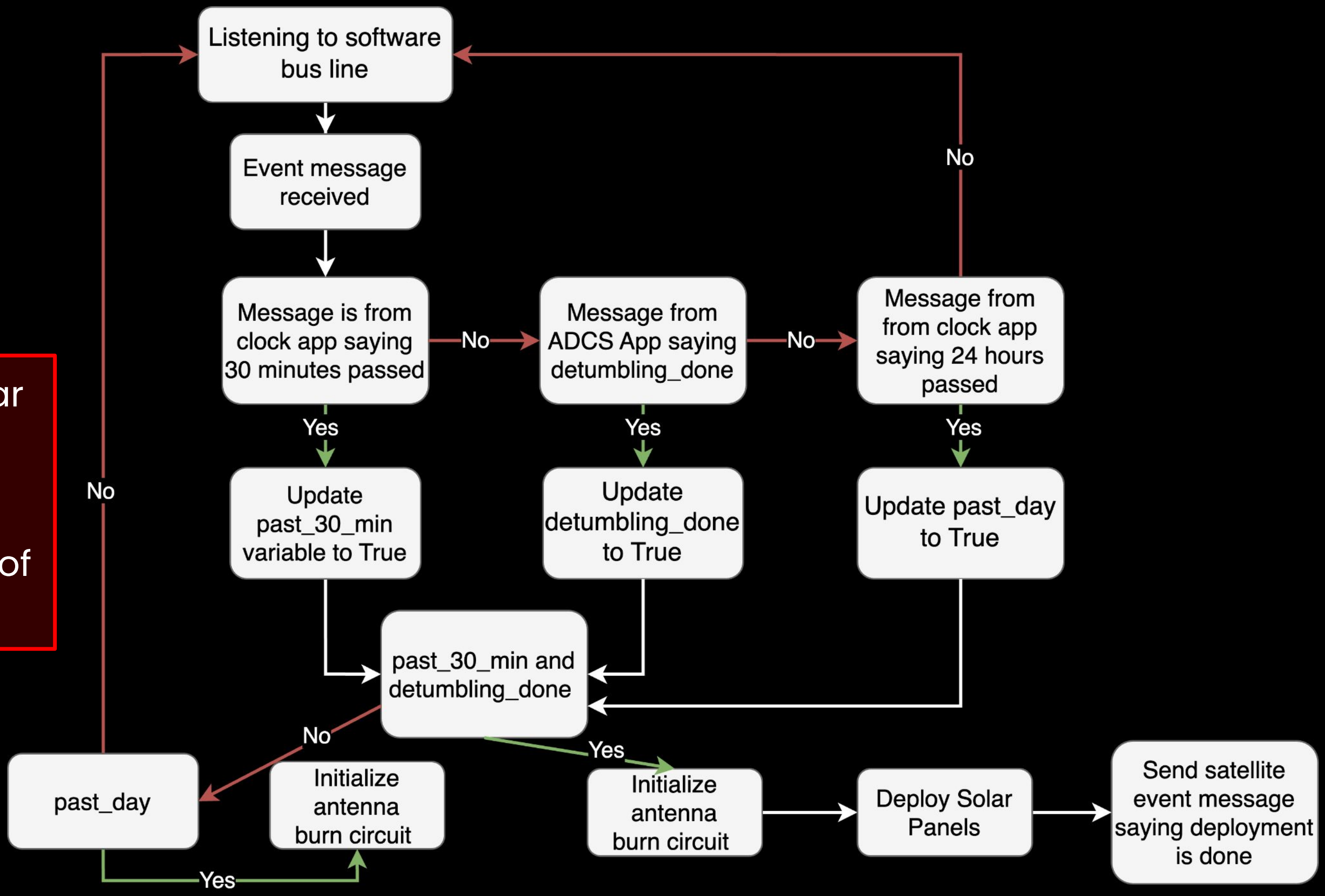
Satellite will start detumbling once at least 30 minutes have passed and the satellite has sufficient power



Satellite detumble finished, ADCS mode changed to sun tracking

Deployment State

Satellite will deploy solar panels, and initiate all antenna burn circuits following the end of detumbling or the end of 24 hours



Nominal State

- ❑ ADCS in sun tracking control mode
- ❑ Satellite approaches RF Pass

RF Receive State

- ❑ Satellite enters RF Pass
- ❑ ADCS switches to ground tracking control mode

RF Receive State

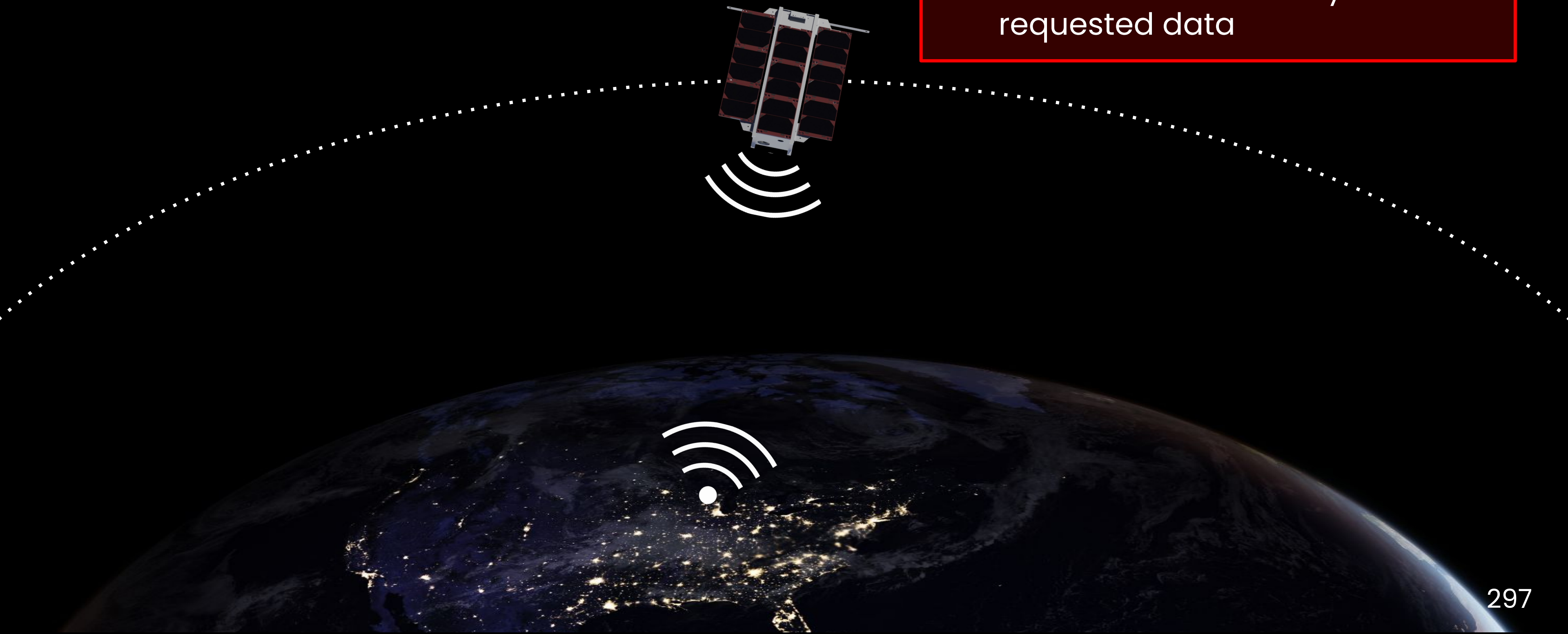
- ADCS reorients satellite in the direction of the ground station

RF Receive State

- ❑ Satellite receives transmission from RFGS

RF Transmit State

- ❏ Satellite processes uplinked data
- ❏ Satellite downlinks telemetry
- ❏ Satellite downlinks any requested data



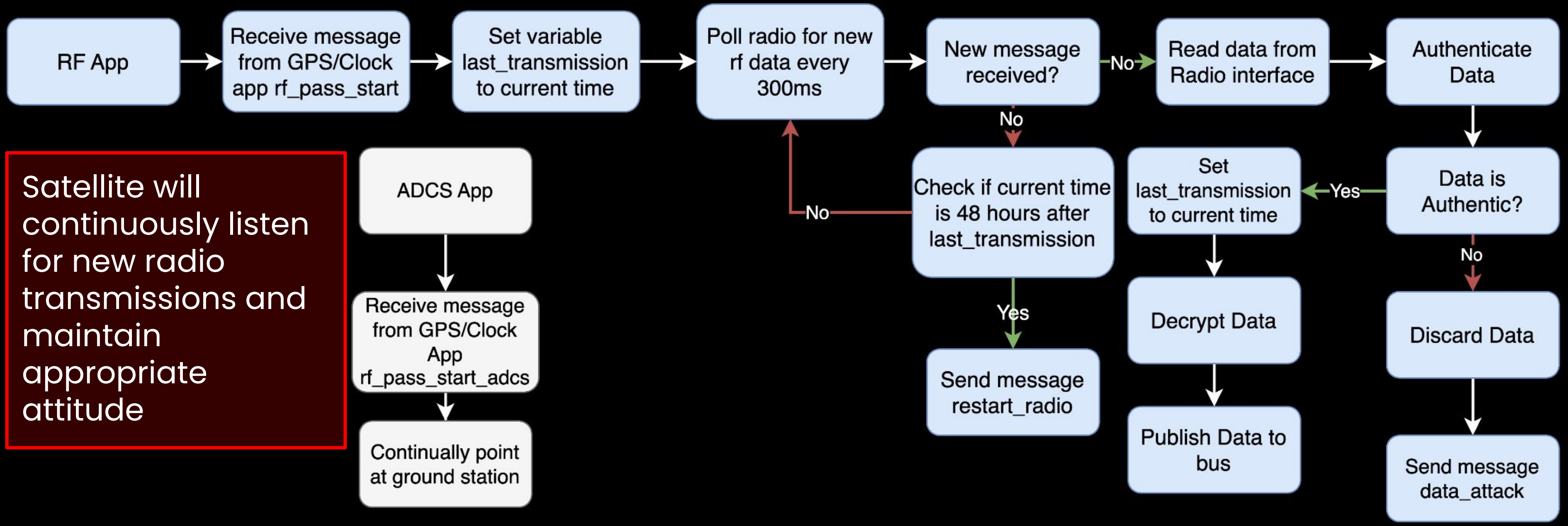
RF Receive State

- Satellite approaches end of RF Pass
- ADCS control mode switches to sun tracking

Nominal State

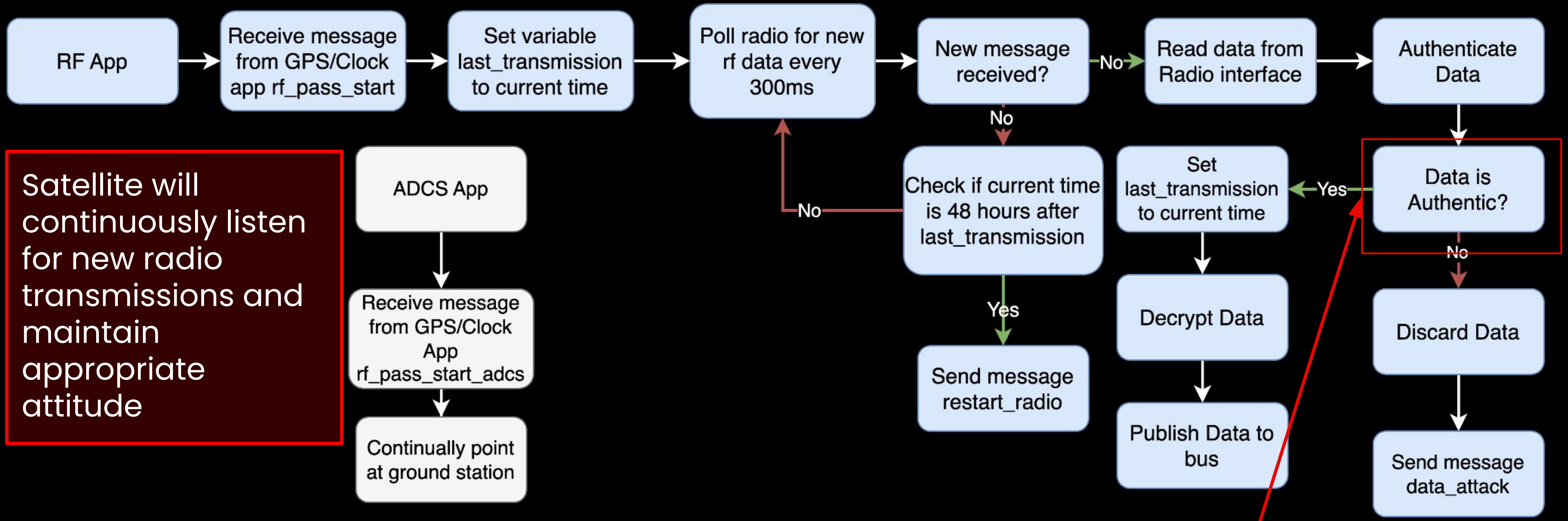
- ❏ Satellite exits RF Pass
- ❏ ADCS in sun tracking control mode

RF Receive State



Satellite will continuously listen for new radio transmissions and maintain appropriate attitude

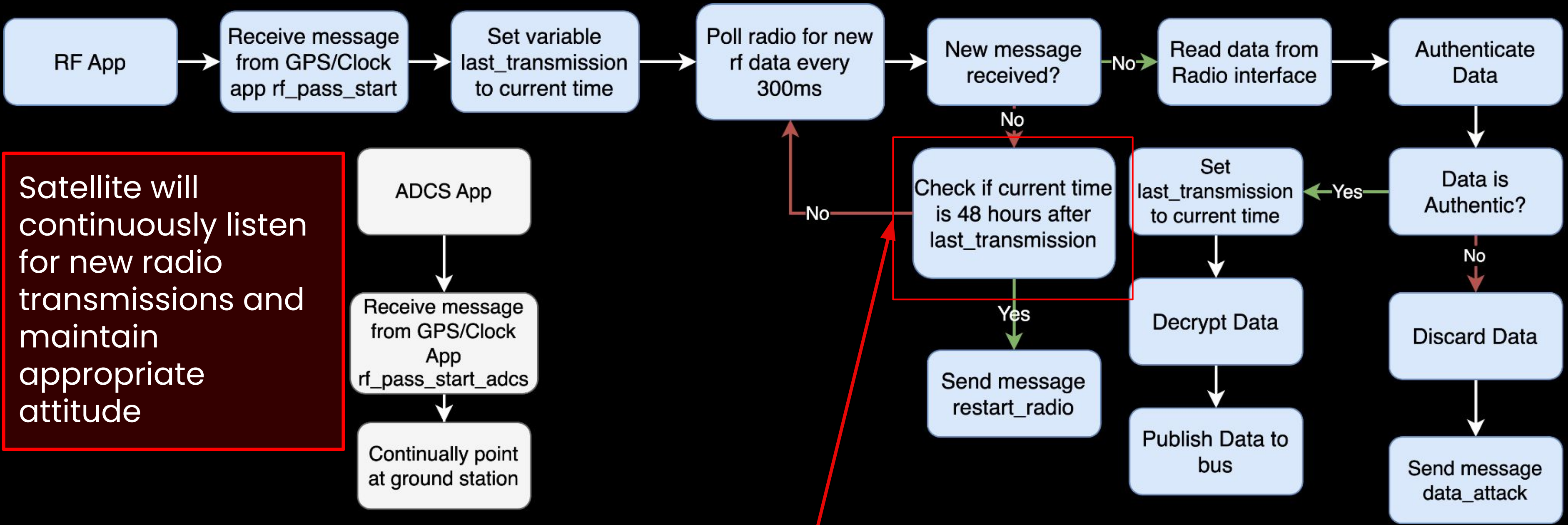
RF Receive State



Satellite will continuously listen for new radio transmissions and maintain appropriate attitude

Satellite authenticates data before passing on to software bus

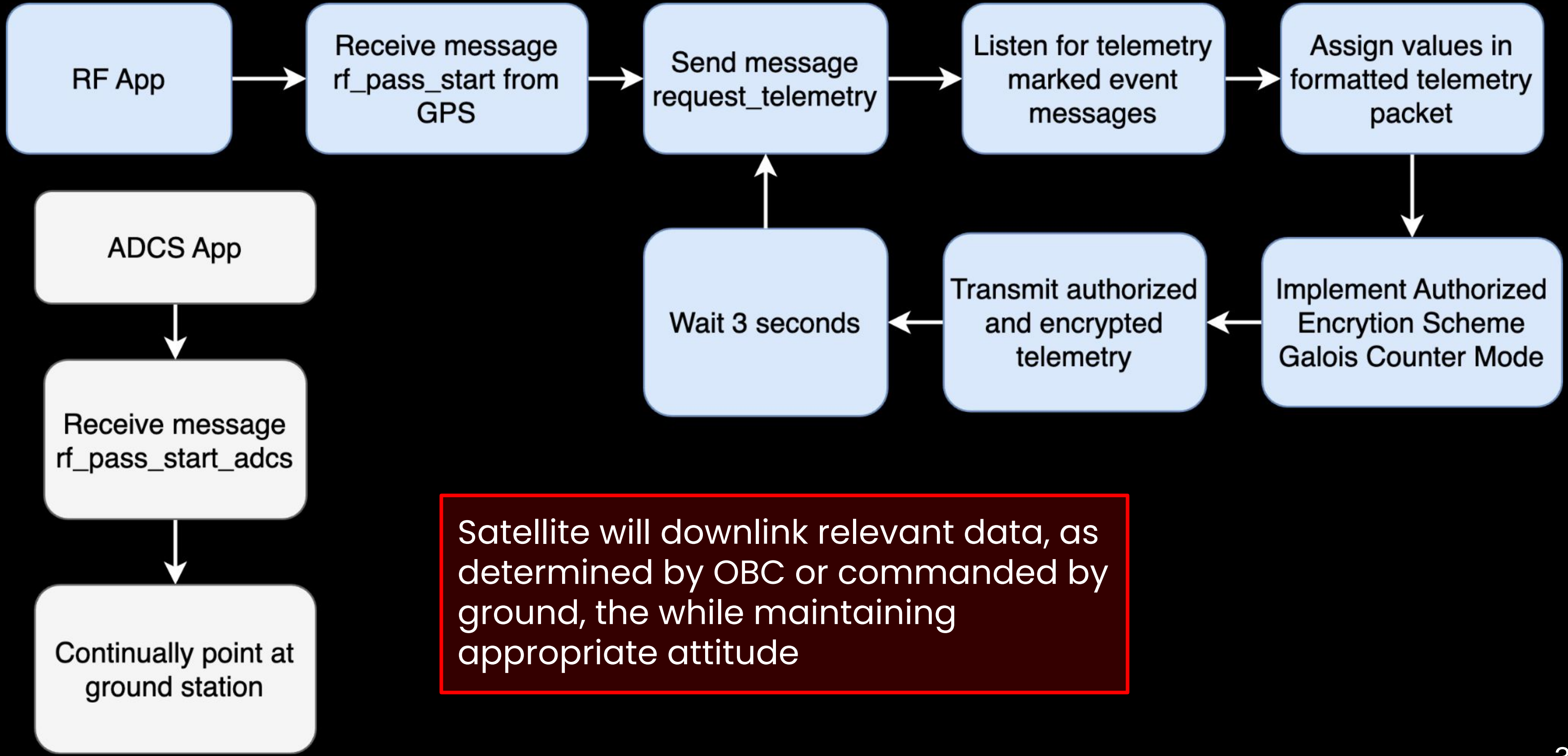
RF Receive State



Satellite will continuously listen for new radio transmissions and maintain appropriate attitude

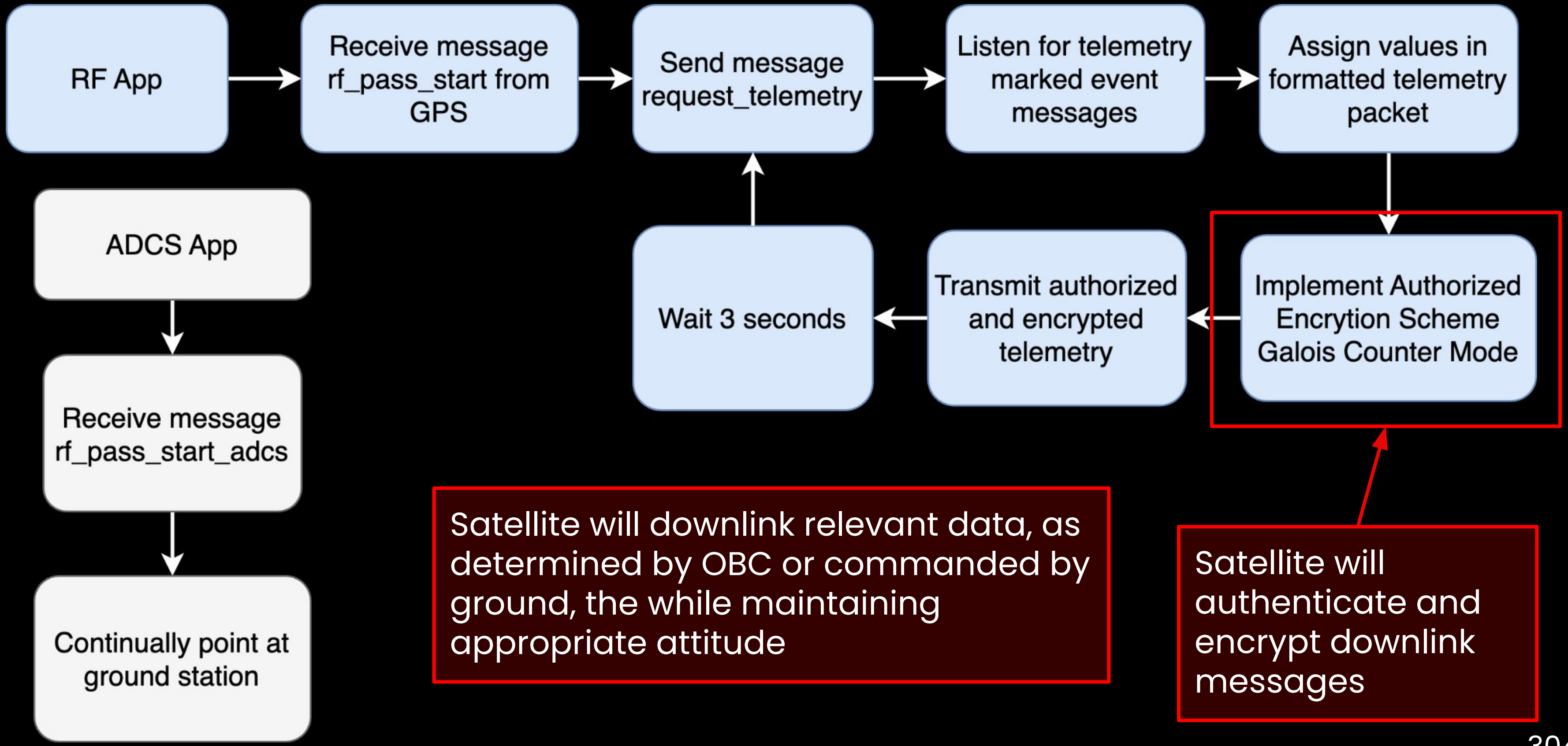
If no communication has occurred for 48 hours the radio is reset

RF Transmit State



Satellite will downlink relevant data, as determined by OBC or commanded by ground, the while maintaining appropriate attitude

RF Transmit State



Next Steps for Avionics Software

- Build OBC and Payload controller cFS instances using pre-built and custom made apps
- Develop hot patching protocol
  - Ensure robust rollback functionality in case a patch introduces new issues
  - Simulate patches on a ground-based environment identical to the satellite hardware/software stack
- Finalize safe mode operations
  - Clearly define the conditions under which the satellite enters safe mode
  - Specify the steps to be taken when safe mode is activated
- Develop failure states
  - Assign specific responses to each failure category, such as restarting subsystems, notifying the ground station, etc.
  - Conduct a fault tree analysis

Pointing, Acquisition, and Tracking (PAT)

PAT Sequence Introduction

- ❑ Achieve alignment of the satellite lasers and ground station such that our principle mission objectives can be fulfilled
- ❑ Maintain alignment throughout optical pass
- ❑ Log and timestamp data throughout optical pass to allow for potential hardware and software changes in the future

PAT Sequence

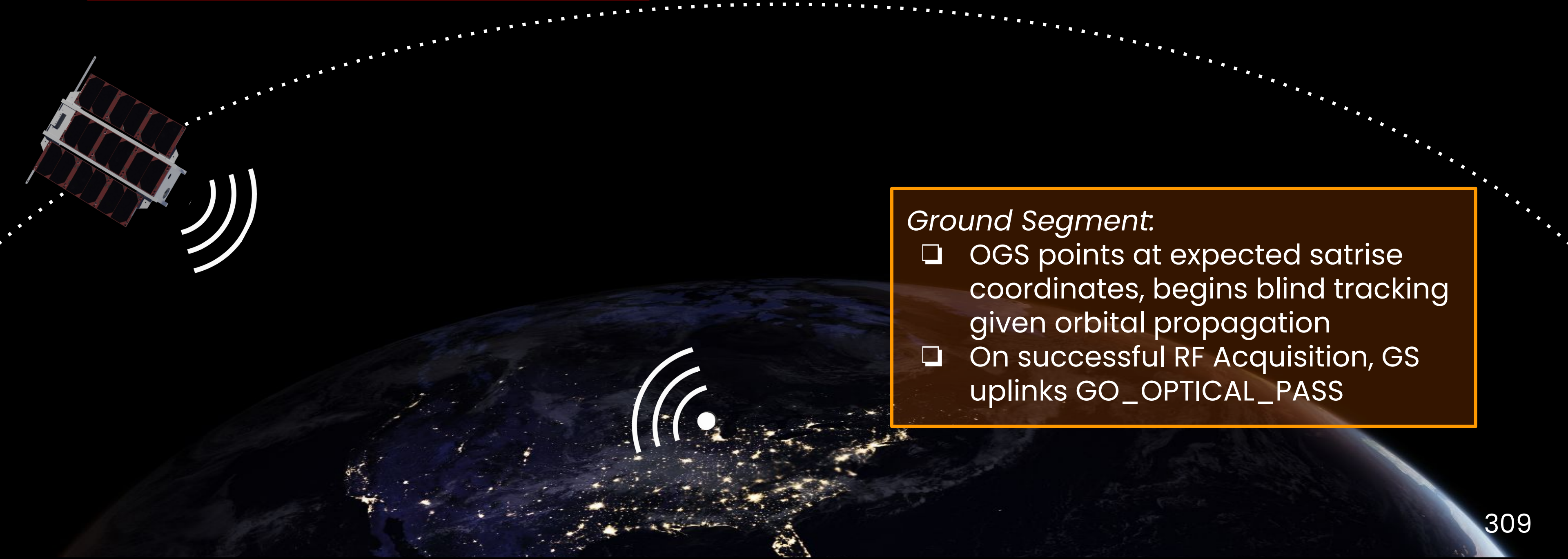
Setup Pass

- ❏ Optical pass viability is confirmed (orbit, weather, OGS preparedness)
- ❏ On successful RF Link, GS uplinks GO_NEXT_OPTICAL_PASS command

PAT Sequence

Space Segment:

- ❑ Before satrise, satellite enters ground tracking regime



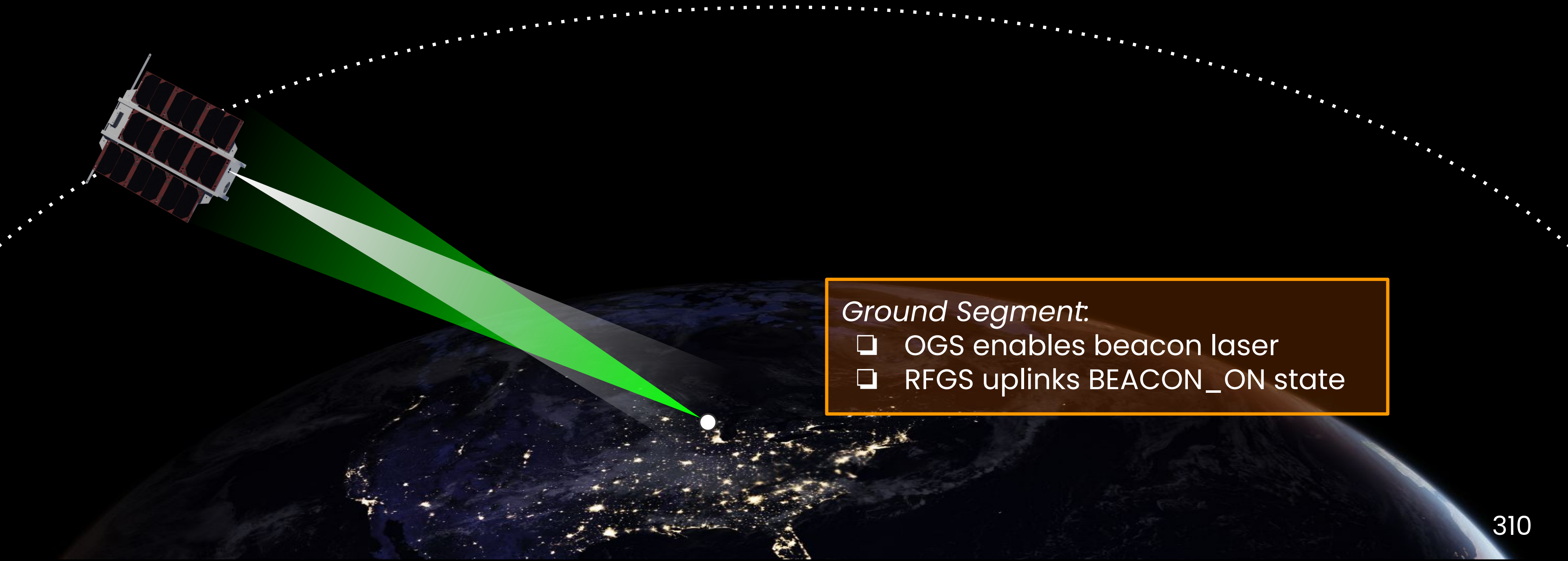
Ground Segment:

- ❑ OGS points at expected satrise coordinates, begins blind tracking given orbital propagation
- ❑ On successful RF Acquisition, GS uplinks GO_OPTICAL_PASS

PAT Sequence

Space Segment:

- ❑ Satellite receives GO_OPTICAL_PASS
- ❑ Satellite enables beacon laser
- ❑ Satellite downlinks BEACON_ON state



Ground Segment:

- ❑ OGS enables beacon laser
- ❑ RFGS uplinks BEACON_ON state

PAT Sequence

Space Segment:

- Satellite begins FSM spiral search pattern

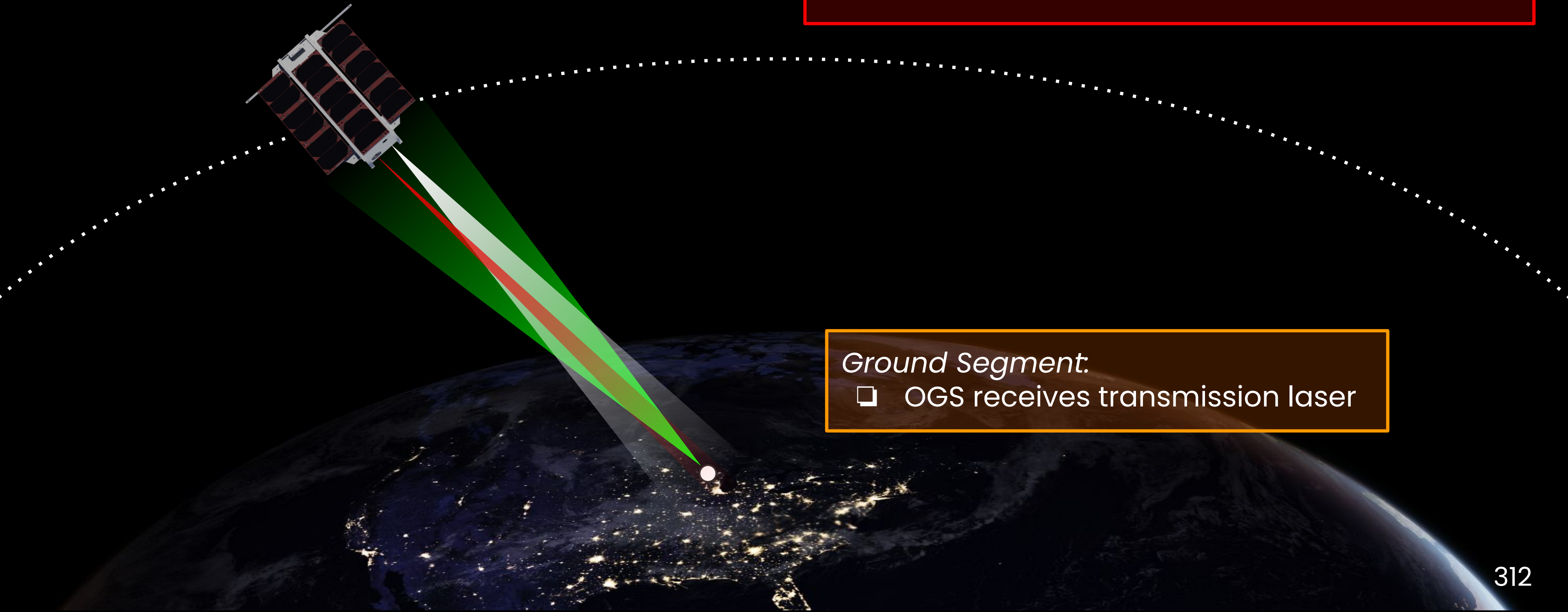
Ground Segment:

- OGS begins acquisition timeout

PAT Sequence

Space Segment:

- ❏ Satellite quadcell acquires OGS beacon
- ❏ FSM aligns beacon to center of quadcell (aligns transmission laser)
- ❏ Transmission laser enabled



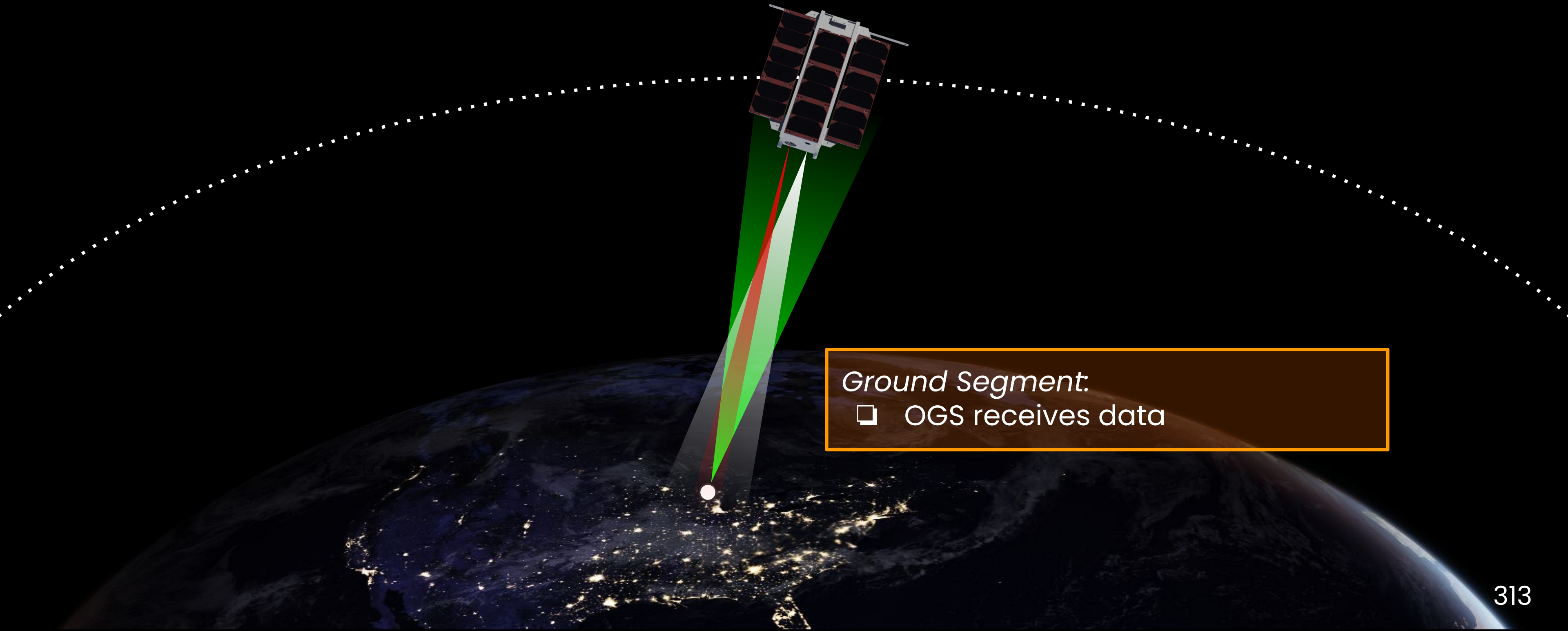
Ground Segment:

- ❏ OGS receives transmission laser

PAT Sequence

Space Segment:

- ❏ Satellite transmits data



Ground Segment:

- ❏ OGS receives data

PAT Sequence

Space Segment:
- ❏ Satellite disables beacon laser
- ❏ Satellite disables transmit laser

Ground Segment:
- ❏ OGS disables beacon laser

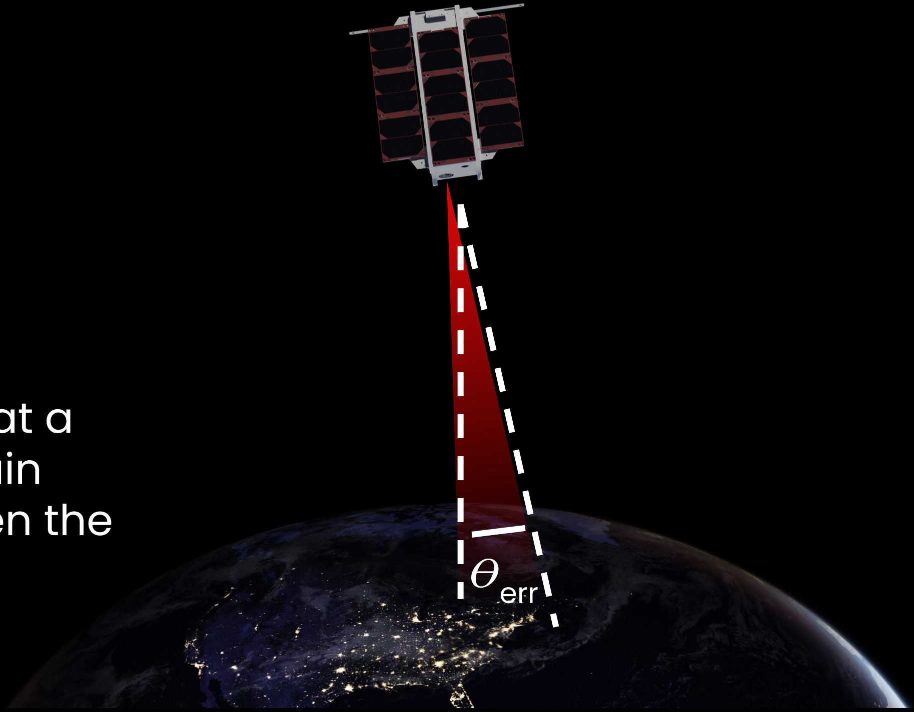
Satellite Coupled Pointing Capabilities

Maximum Allowable Pointing Error:

Assuming a max allowed pointing loss of 3.5dB, divergence angle of downlink beam ~1 mrad:

θ_{err} = 0.317 mrad = 0.0182 degrees

Interpretation: To ensure detection of the downlink transmission beam by the OGS, at a distance of 1000 km or less, we must remain within this angle of a straight path between the satellite and OGS.



OGS Coupled Pointing Capabilities

$$\theta_{err} = \frac{\alpha}{2\sqrt{2}}\sqrt{\ln\left(\frac{8P\tau A_{detector}}{Nk\pi\alpha^2 D_{sat}^2}\right)}$$

α: Full angle divergence

$A_{detector}$: Area of OGS telescope

P: Power of downlink transmission laser

k: SNR (Signal-to-Noise Ratio)

D_{sat}: Distance from satellite to OGS

N: Average noise power

$1-\tau$: Atmospheric loss of light intensity (where $0<\tau<1$)

Maximum Allowable Pointing Angle of Error of the Beacon:

Assuming SNR = 5:

Ideal divergence angle:

θ_{div} = 0.00417 radians = 0.239 degrees

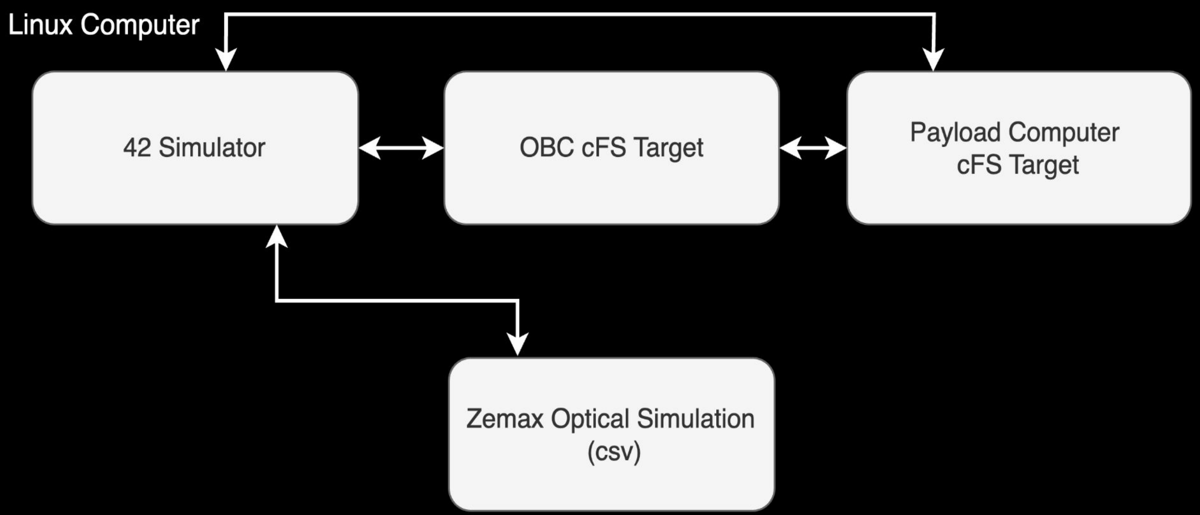
Maximum allowable pointing error of the beacon:

θ_{err} = 0.0014 radians = 0.0802 degrees

Interpretation: To ensure detection of the uplink transmission beam by the Payload, at a distance of 1000 km or less, we must remain within this angle of a straight path between the OGS and satellite.

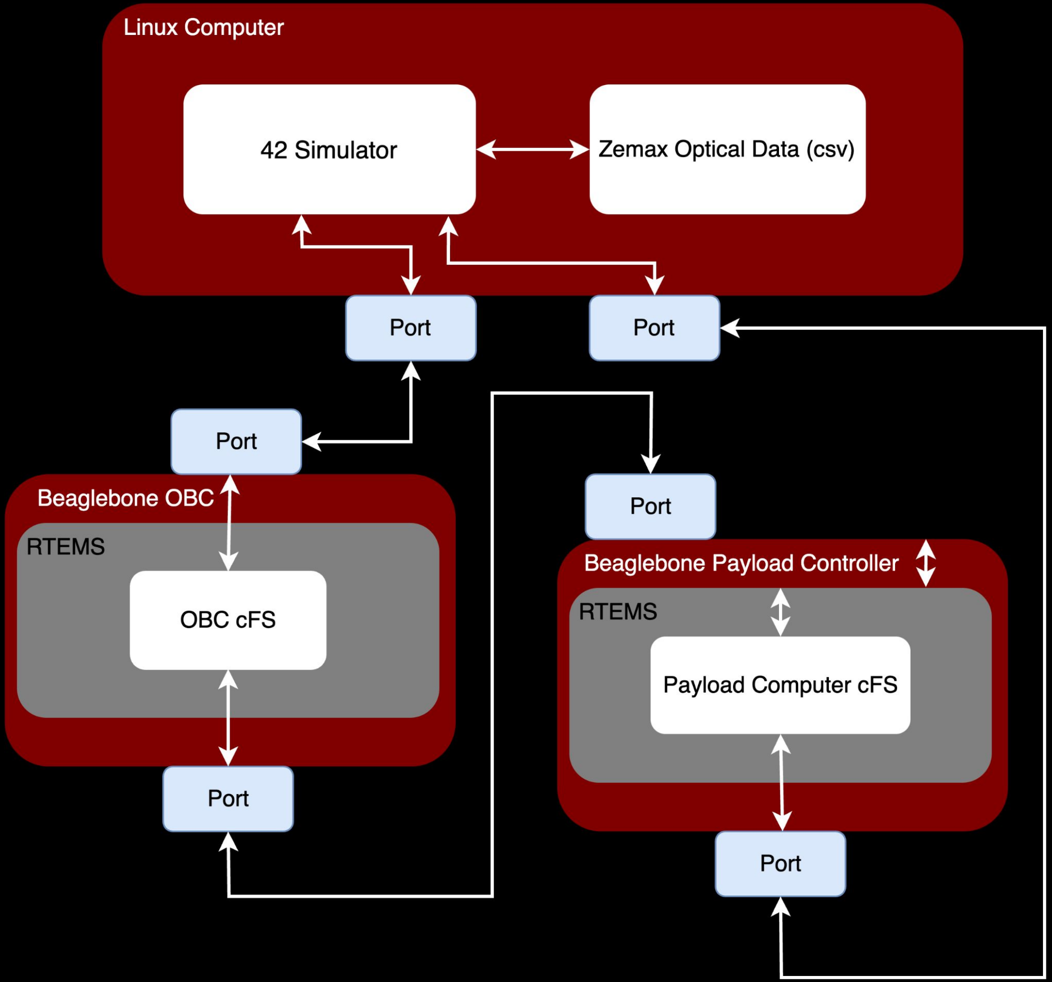
Software Simulation

This is fully virtual simulation using 42 and Zemax which allows for the testing of software in a virtual environment



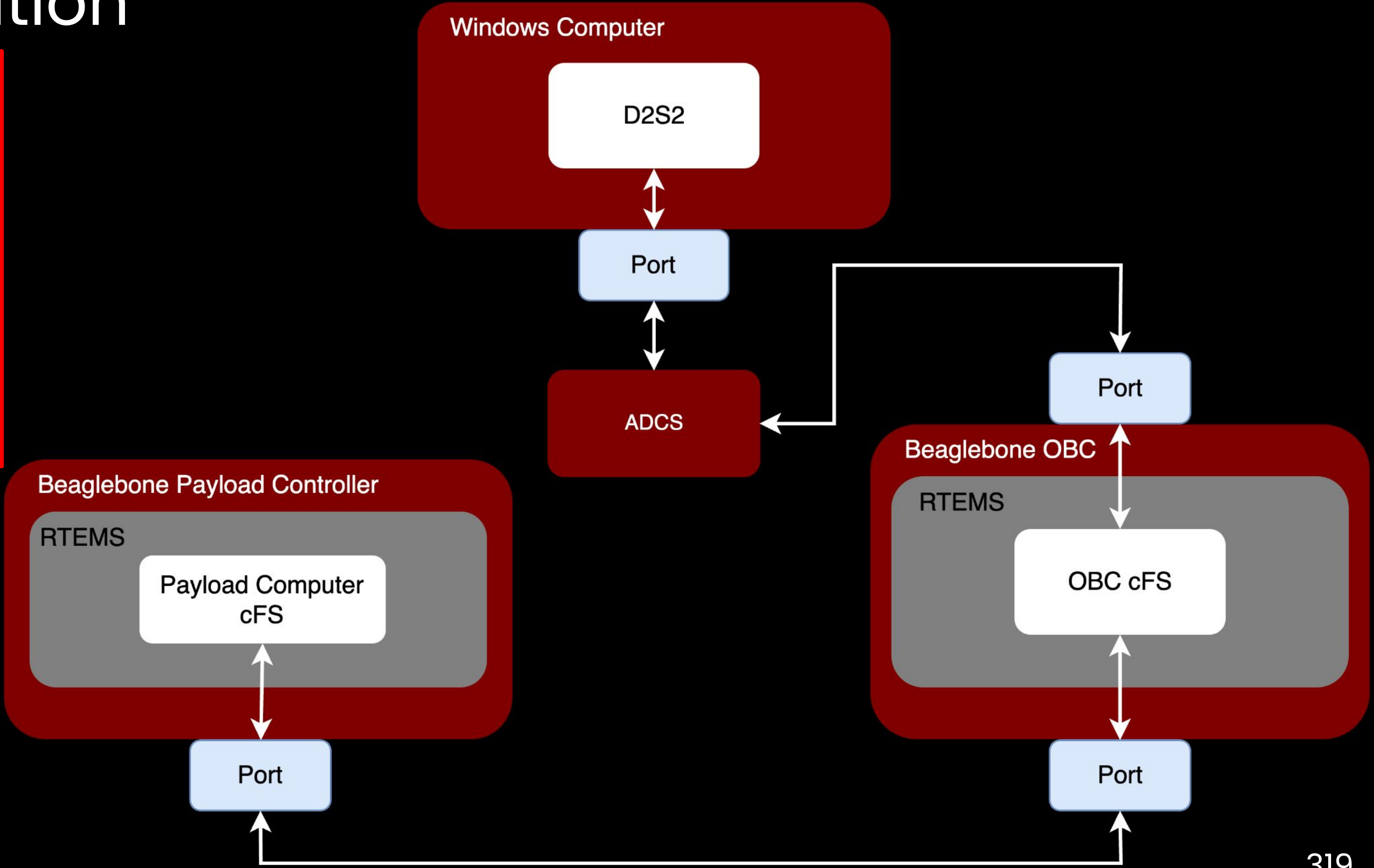
42 PAT Sequence Simulation

This simulation uses the open source 42 simulator from NASA while interfacing with Zemax optical data. It interfaces with the OBC and Payload Controller to simulate the Pointing, Acquisition, and Tracking



D2S2 Simulation

This simulation uses the D2S2 simulator provided by the ADCS manufacturer to test and verify the ADCS pointing performance



Next Steps for PAT Sequence

- ❑ Build and begin software and HIL simulations and assess potential changes to sequence as necessary
- ❑ Build cFS PAT app and other key systems and test through simulation
- ❑ Update key telemetry requirements for uplink and downlink and optimize transmission
- ❑ Test FSM spiral search on the ground
- ❑ Build validation metrics and monitor throughout testing process
- ❑ Incorporate ground station visibility and line-of-sight predictions

Assembly, Integration, and Test (AIT)

AIT Driving Requirements (1/2)

ID	Requirement	Notes	Parent	Verification
AIT-01	The Satellite shall be integrated in a clean room that complies with the highest classification required by component vendors.	Clean room specifications are dependent on final design.	MSN-05	Inspection
AIT-02	AIT shall ensure that the Satellite complies with all requirements in the most recent version of the CubeSat Design Specification (CDS).	The most recent version of the CDS is Revision 14.1.	SAT-01	Inspection
AIT-03	AIT shall ensure that the Satellite complies with the most up-to-date revision of the NASA Launch Services Program: Program Level Dispenser and CubeSat Requirements Document.	The most recent version of the Program Level Dispenser and CubeSat Requirements Document is LSP-REQ-317.01 B.	AIT-02	Inspection

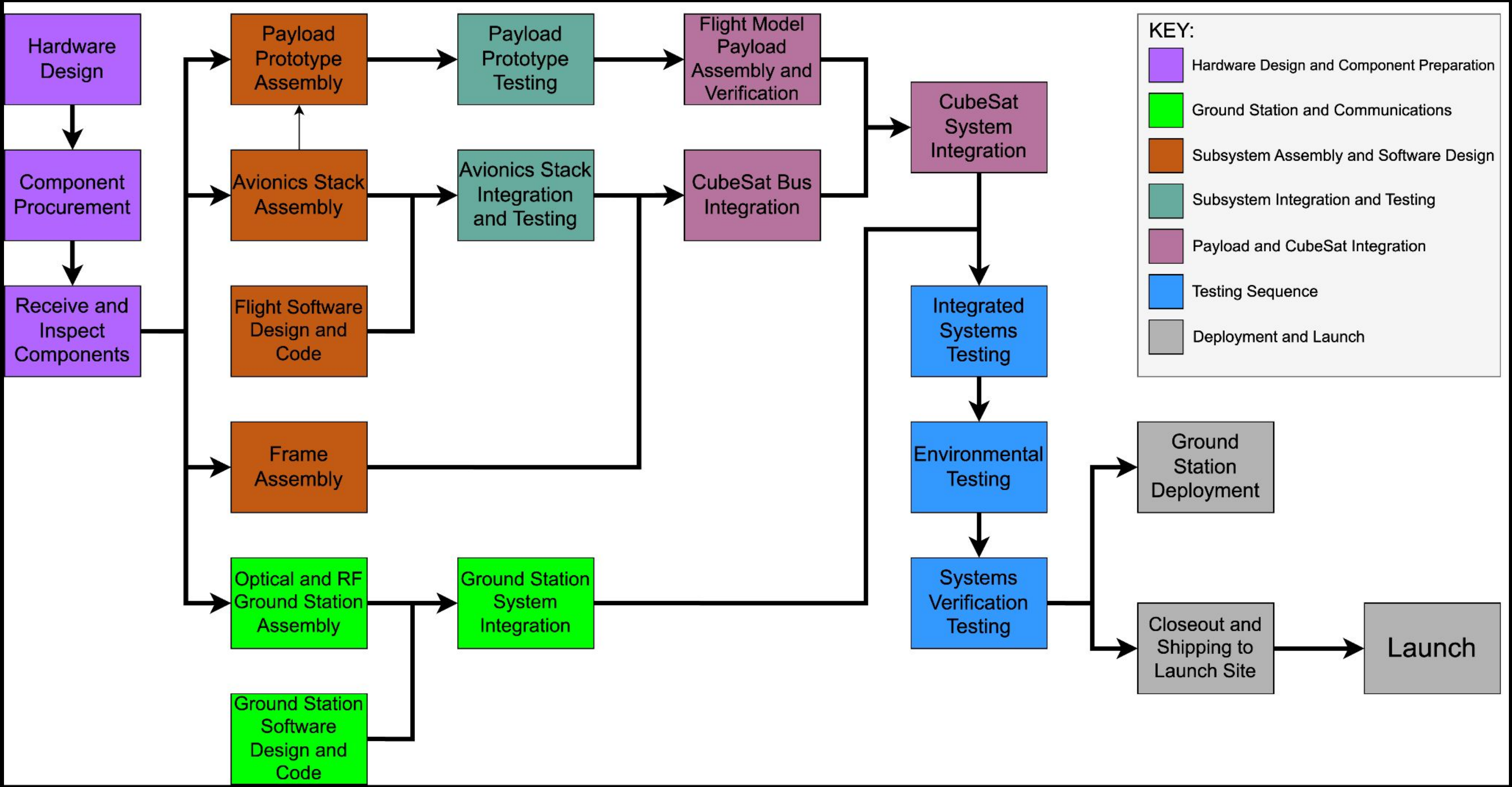
Legend: Compliant Compliant by CDR Compliant by TRR Compliant By FRR

AIT Driving Requirements (2/2)

ID	Requirement	Notes	Parent	Verification
AIT-04	AIT shall ensure that the Satellite complies with all requirements imposed by the Launch Provider.	The Launch Provider is TBD until manifested by NASA.	SAT-02	Inspection
AIT-05	AIT shall ensure that the Satellite complies with all requirements imposed by the Deployer Provider.	The Deployer Provider is TBD until manifested by NASA. The deployer is expected to be NanoRacks CubeSat Deployer (NRCSD).	SAT-03	Inspection

Legend: Compliant Compliant by CDR Compliant by TRR Compliant By FRR

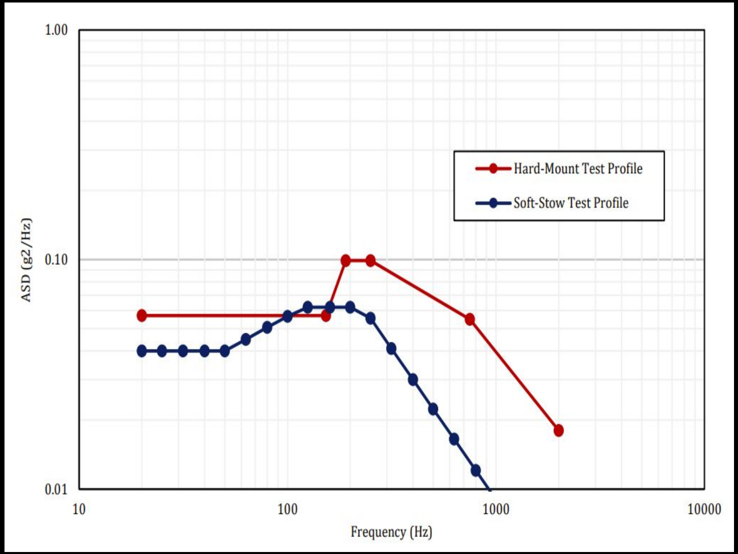
AIT Process Flow



Vibration Testing

Random Vibration:

- Apply broadband random vibration on each of 3 perpendicular axes for 1 minute
- Frequency band spans from 20 - 2000 Hz
- Acceleration spectral density (ASD) spans from 4E-2 to 2.6E-3 g^2/Hz
- The flight acceptance levels should be at least MPE (maximum predicted environment)
- Testing will be undergone in a soft stow configuration unless otherwise indicated by the deployer



Sinusoidal vibration testing (low level sine sweep) may be required if requested by the Deployer Provider and/or Launch Service Provider.

NanoRacks dispenser random vibration profile (for hard mount and soft stow configurations)

Testing requirements are preliminarily derived from NanoRacks CubeSat Deployer (NRCSD) Interface Definition Document (IDD)

Thermal Testing

Thermal Vacuum Cycling:

- ❑ INCLUDES THERMAL BALANCE
- ❑ Vacuum Levels: $10x10^{-4}$ Torr
- ❑ Temperature Range: MPE+/- 5°C, Minimum Range: -9-3/+0°C to +66-0/+3°C
- ❑ Minimum Temperature Rate of Change: 5°C/Minute
- ❑ Number of Cycles: 2
- ❑ Dwell Time: 1 hour minimum @ extreme temp. after thermal stabilization

*MPE will be determined by using thermal desktop simulations (simulation results +11 degrees C)

Testing requirements are derived from NASA Launch Services Program Level Dispenser (LSP-REQ-3.17) and Cubesat Requirements Document (CDS 14.1)

Thermal Testing Continued

Thermal Vacuum Bakeout:

- ❏ Vacuum Levels: 1×10^{-4} Torr
- ❏ Temperature: 70°C (minimum)
- ❏ Minimum Temperature Rate of Change: < 5°C/minute
- ❏ Number Cycles: 1
- ❏ Dwell Time = Minimum of 3 hours after thermal stabilization
- ❏ Equipped with residual gas analyzer to determine outgassing

Electronic Burn-in:

- ❏ Combined duration of all thermal testing (vacuum + bake out + cycling) + additional burn-in shall be at least 100 hours **failure free**
- ❏ May be performed at ambient pressure

Bakeout testing requirements are pulled from NASA Launch Services Program Level Dispenser (LSP-REQ-3.17) and Cubesat Requirements Document (CDS 14.1)
Burn-in testing requirements are pulled from Air Force Space Command SMC Standard SMC-S-016 Test Requirements for Launch Upper-Stage and Space Vehicles

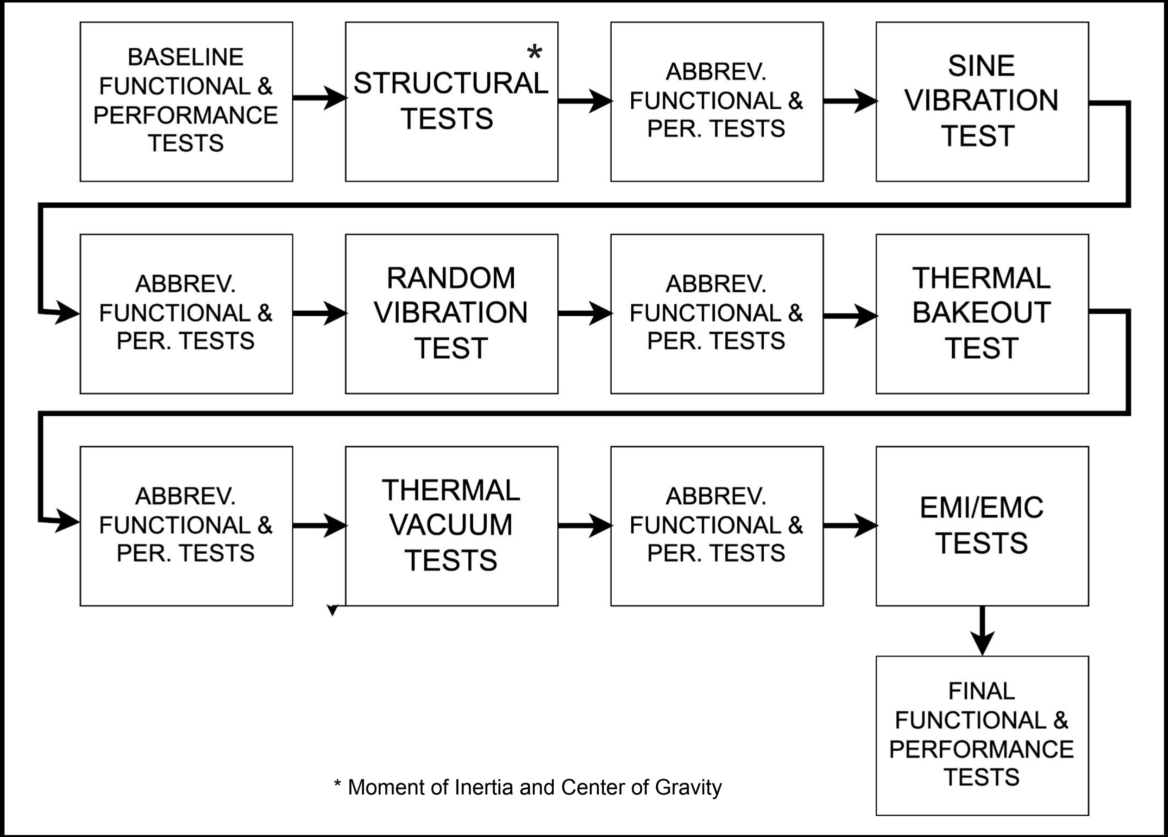
EMC Testing

EMC:

- To be incorporated into test levels
- **Margin:** 6db
- **Time:** 20 minutes at each space vehicle transmitter frequency for radiated susceptibility (these numbers are to be determined; they are somewhere between VHF and UHF)
  - 15ms duration for emission requirements
- **Pressure:** Ambient pressure shall be used except where degradation due to a vacuum environment may be anticipated
  - In this case, **pressure**: 13.3 mPa (10^{-4} Torr) or less

Testing requirements are pulled from Air Force Space Command SMC Standard SMC-S-016 Test Requirements for Launch Upper-Stage and Space Vehicles

Potential Test Sequence



Environmental testing on the integrated Satellite will be done in the same order that the Satellite will experience environmental loads:

1. Vibration
2. Thermal bakeout
3. Other thermal vacuum
4. EMI/EMC

Between each environmental test, **abbreviated functional and performance tests** will be performed to ensure that the tests did not damage the Satellite.

Potential Testing Facilities

UIUC

- ❑ TVAC chamber
- ❑ Vibration table
- ❑ Helmholtz cage
- ❑ Measure moment of inertia / center of gravity

FermiLab

- ❑ TVAC chamber
  - ❑ More convenient location

UChicago

- ❑ Clean room (ISO Class 8 or better required for ADCS)

On-Orbit Deployer: **Nanoracks** (1/2)

- **Rail Design:**
  - Rail has minimum width 6mm (XY face) and radius 0.5 +/- 0.1mm
  - 3U Rail Length = 340. 50 mm, extends at least 2mm beyond z faces
- **Mass Limits:**
  - 3U Mass Limit = 4.8 Kg
  - COM is +/- 2 cm for x and y axis, +/- 6 cm for z-axis
- **Switch Requirements:**
  - At least 3 independent electrical inhibits switches
  - Switch should reset payload to pre-launch state if cycled within first 30 minutes
  - Total Switch Force = max 9N
- **Post Deployment:**
  - CubeSat shall not operate any system for at least 30 minutes where hazard potential exists

On-Orbit Deployer: **Nanoracks** (2/2)

- **Electrical:**
  - Ground charge circuit should not energize satellite systems or flight computer
  - RBF feature to keep satellite in unpowered state during ground handling and integration
  - RBF feature should preclude power from any source operating satellite except for battery charging
  - Maximum 6 inches of wire 26AWG or larger between power source and first inhibit
  - Protection circuitry and safety to prevent internal/external short circuit and over/under voltage conditions
- **Battery:**
  - Battery Energy Density (72.36) < 80 WH/kg
  - Lithium Cells should be restrained at all times
- **Environment:**
  - Withstand random vibration environment with appropriate safety margin (softstow configuration)
  - Withstand force of 1200 N across all load points equally in z direction
  - Withstand pressure extremes airlock depressurization
  - Comply with NASA guidelines for outgassing and hazardous materials

On-Orbit Deployer: **Exo Launch**

- **Rail Design:**
  - 3U Rail Width = at least 6.4 mm
  - 3U Rail Length = 340. 5 +/- 0.5 mm
  - Maximum space between rails = 87.2 mm for x and y axis
  - Rail clamping force = 330 -450 N
- **Mass Limits:**
  - 3U Mass Limit = 6 kg
  - COM is +/- 2 cm for x and y axis, +/- 7 cm for z-axis away from geometric center

Safety Requirements

Safety Data Template: Satellite design information relevant for processing the satellite through the Safety Review Process. This requires CAD renders, block or systems diagrams, engineering drawings, material data sheets, various spec sheets, and other technical information as requested by integrator

Bill of Materials: To be utilized for external outgassing contamination assessment and formation of Materials Identification Usage List (MIUL)

Vibration Test Report: Integrated test report outlining test set-up, as-run accelerometer response plots, and post-vibration functional and inspection results

Battery Test Report: Test report that shows compliance with ISS Battery Test Statement of Work (SoW)

Outgassing: All non-metal vacuum exposed materials must be listed in the Bill of Materials. Each materials should not exceed TML of >1.0% and CVCM of >0.1%. No leeway due to duration of vacuum exposure in ISS vicinity

Toxicity: All materials deemed by toxicologists to be above THL 2 will need to be noted in the Ground Safety Package

Risk Assessment and Mitigation

Risk #	Potential Failure Mode	Potential Causes	Potential Effects	Mitigation Plan
1	**Deployables failure**	Defective nichrome wire, insufficient power to burn the wire, circuit malfunction	Unable to deploy antennas, solar panels, magnetometer	Redundant circuits, burn wire testing, deployable testing, simulation
2	**Battery failure**	Batteries become too cold to function, degrade at a faster rate than expected	Inability to power systems	80/20 charge cycle, battery heaters, additional non-rechargeable battery
3	**ADCS failure**	Failure to detumble, reaction wheel failure due to vibration damage, EMI interference	Inability to point to the GS making PAT impossible	Buy a space-rated COTS ADCS, consider buying a 4 reaction wheel ADCS
4	**GPS failure**	Vibration, thermal, electrical shorts, radiation	Inability to determine telemetry preventing PAT	Predict orbit using previous data
5	**RF Comms failure**	Vibration, thermal, electrical shorts, radiation	Inability to communicate with ground	Daily reset of comms system, buy space grade radio
6	**EPS failure**	PDU malfunction, vibration causes cable disconnection, solar panel degradation	Inability to power systems	Buy space grade COTS components

The subsystems with the **most catastrophic consequences of failure** also have the **lowest risk of failure**.

These risks are mitigated by purchasing **space-grade COTS** components as well as using **redundancy** and performing **workmanship tests**.

Risk #	Potential Failure Mode		Potential Effects	Mitigation Plan
1	**Deployables failure**		Unable to deploy antennas, solar panels, magnetometer	Redundant circuits, burn wire testing, deployable testing, simulation
2	**Battery failure**		Inability to power systems	80/20 charge cycle, battery heaters, additional non-rechargeable battery
3	**ADCS failure**		Inability to point to the GS making PAT impossible	Buy a space-rated COTS ADCS, consider buying a 4 reaction wheel ADCS
4	**GPS failure**		Inability to determine telemetry preventing PAT	Predict orbit using previous data
5	**RF Comms failure**	shorts, radiation	Inability to communicate with ground	Daily reset of comms system, buy space grade radio
6	**EPS failure**	PDU malfunction, vibration causes cable disconnection, solar panel degradation	Inability to power systems	Buy space grade COTS components

Risk #	Potential Failure Mode	Potential Causes	Potential Effects	Mitigation Plan
7	**Software failure**	Radiation corrupting data, bugs	Various	Shield radiation sensitive components, watchdog processor, redundant hardware, update via RF
8	**Real time clock failure**	Radiation, software bugs, power issues, hardware failure	Satellite cannot perform PAT and RF	Use rad-hard model, redundancy (with GPS, OBC etc containing their own clocks)
9	**CAN bus failure**	Radiation, hardware issues, power supply issues	Loss of communication between subsystems, inability to deploy components	Redundant communication lines, error detection and correction
10	**CAN bus overload**	Send too many packets	Loss of communication between subsystems, dropped packets	Redundant communication lines, error detection and correction
11	**Flight computer (OBC) failure**	Software corruption, improper firmware updates, radiation damage, software corrupted by radiation, CAN transceiver damaged	Inability to perform maneuvers, loss of communication	Watchdog processor, potential redundancy with Payload Controller

Risk #	Potential Failure Mode	Potential Causes	Mitigation Plan
7	**Software failure**	Radiation corrupting data, bugs	Shield radiation sensitive components, watchdog processor, redundant hardware, update via RF
8	**Real time clock failure**	Radiation, software bugs, power issues, hardware failure	Use rad-hard model, redundancy (with GPS, OBC etc containing their own clocks)
9	**CAN bus failure**	Radiation, hardware issues, power supply issues	Redundant communication lines, error detection and correction
10	**CAN bus overload**	Send too many packets	Redundant communication lines, error detection and correction
11	**Flight computer (OBC) failure**	Software corruption, improper firmware updates, radiation damage, software corrupted by radiation, CAN transceiver damaged	Watchdog processor, potential redundancy with Payload Controller

Software-related failure is the **most likely** problem that we expect to encounter, especially due to **radiation**.

Fortunately, the consequences of these risks are the **least extreme** and can sometimes be **repaired after launch**.

These risks are less severe because of **redundant hardware** as well as use of a **watchdog processor**.

Risk #	Potential Failure Mode	Potential Causes	Potential Effects	Mitigation Plan
12	**Payload lens misalignment**	Thermal expansion of mountings, vibration damage, heat damage to lenses	Inability to perform PAT	Simulation (Zemax/Thermal Desktop), testing
13	**Quadcell APD failure**	Saturation from pointing directly at the sun, vibration damage, misalignment	Inability to perform PAT	Bandpass and shortpass filters, angle FSM away from quadcell when off, test in solar simulator
14	**FSM failure**	Low temperature, vibration damage	Inability to perform PAT	Testing, buy space-grade COTS for MEMS FSM
15	**Payload Controller failure**	Radiation damage, software corruption, vibration damage, overheating, short	Inability to perform PAT	Watchdog processor
16	**FPGA failure**	Radiation	Inability to transmit data via polarized laser	Regularly reprogram, shield computers from radiation
17	**Modulation failure**	Vibration, thermal or radiation damage	Inability to transmit data via polarized laser	Thermal control, maintain constant electrical signal
18	**Ground Station failure**	Comms, PAT, electrical issues, optics damage	Inability to perform ground operations	On site analysis and debugging

Risk #	Potential Failure Mode	Potential Causes	Potential Effects
12	**Payload lens misalignment**	Thermal expansion of mountings, vibration damage, heat damage to lenses	Inability to perform PAT
13	**Quad-cell failure**	Saturation from pointing directly at the sun, vibration damage, misalignment	Inability to perform PAT
14	**FSM failure**	Low temperature, vibration damage	Inability to perform PAT
15	**Payload Controller fai**	Radiation damage, software corruption, vibration damage,	Inability to perform PAT
16	**FPGA**		Inability to transmit data via polarized laser
17	**Modu fai**	ation	Inability to transmit data via polarized laser
18	**Ground Station failure**	Comms, PAT, electrical issues, optics damage	Inability to perform ground operations

Ground Station failures are of least severity as they can be reworked on site

Most risks to the **Optical Payload** are caused by **thermal or vibration damage.**

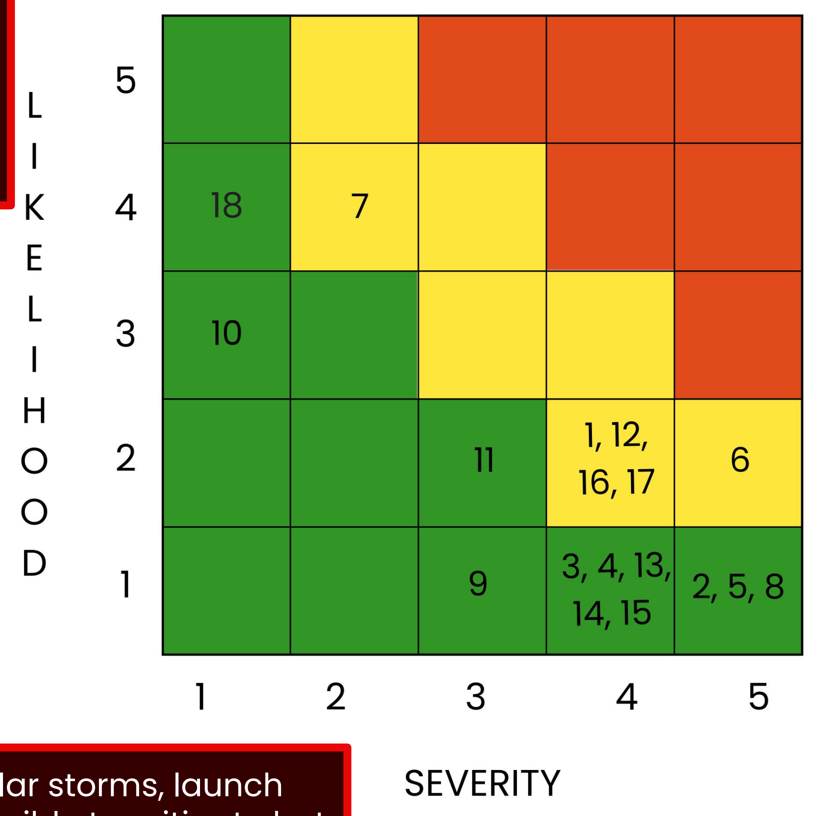
Damage to Payload components can make accomplishing our primary objective of transmitting data via polarized laser **impossible**.

However, damage to Payload components will not completely eliminate the Satellite's ability to communicate with the Ground Station and perform some functions.

Risk Square

Color key:
Green: manageable risk
Yellow: significant risk
Red: unacceptable risk

1. Deployables failure
2. Battery failure
3. ADCS malfunction
4. GPS malfunction
5. RF comms malfunction
6. EPS failure
7. Software failure
8. Real time clock failure
9. CAN bus failure
10. CAN bus overload
11. Flight computer (OBC) failure
12. Misalignment of optical payload lenses
13. Quadcell failure
14. FSM failure
15. Payload Controller failure
16. FPGA failure
17. Polarization switch/modulator failure
18. Ground Station failure



Other potentially mission ending risks include extreme solar storms, launch vehicle failure, and deployer failure. These risks are impossible to mitigate but are extremely unlikely to happen.

Risk Square

1. Deployables failure
2. Battery failure
3. ADCS malfunction
4. GPS malfunction
5. RF comms malfunction
6. EPS failure
7. Software failure
8. Real time clock failure
9. CAN bus failure
10. CAN bus overload
11. Flight computer (OBC) failure
12. Misalignment of optical payload lenses
13. Quadcell APD failure
14. FSM failure
15. Payload Controller failure
16. FPGA failure
17. Polarization switch/modulator failure
18. Ground Station failure

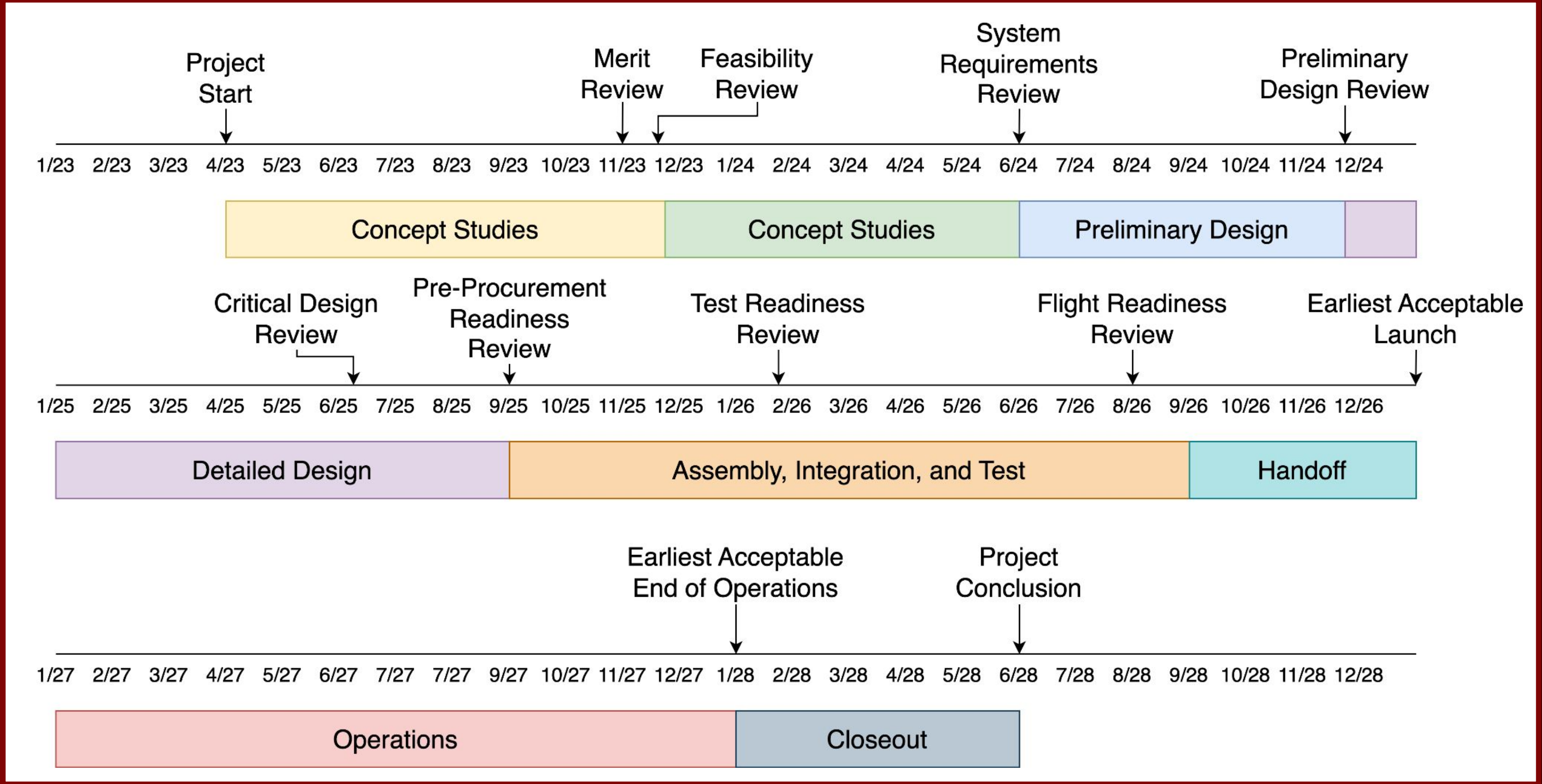
Risk Severity Key:
1. Minor, easily fixable
2. Moderate
3. Significant disruption
4. Unable to achieve mission goal of data transmission via polarized laser
5. Completely unable to function, satellite cannot contact ground

LIKELIHOOD	1	2	3	4	5
5					
4	18	7			
3	10				
2			11	1, 12, 16, 17	6
1			9	3, 4, 13, 14, 15	2, 5, 8
	SEVERITY				

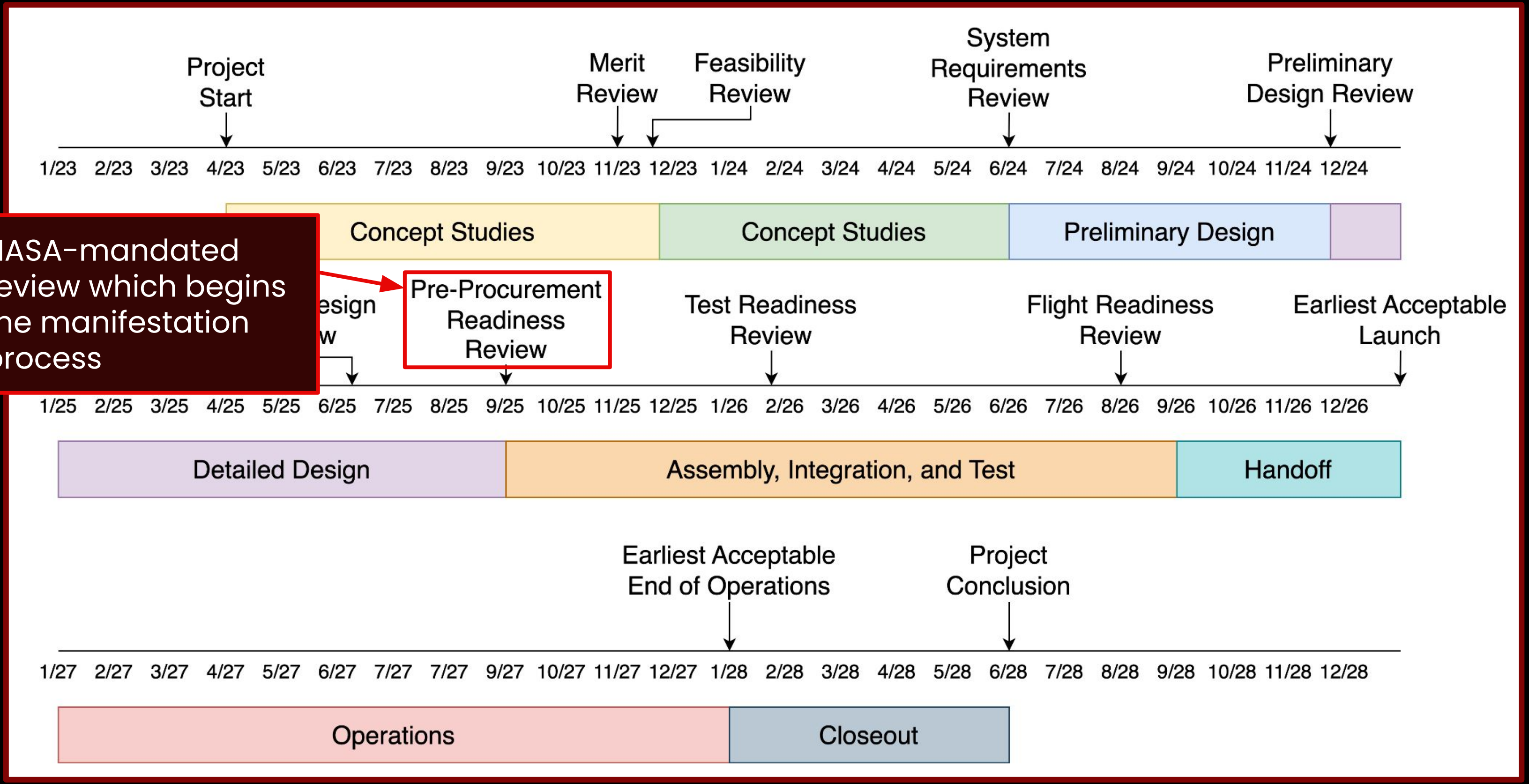
Risk Likelihood Key:
1. Assembly/components purchased COTS (space grade)
2-3. Components purchased COTS but assembled or modified by PULSE-A
4-5. Assembly and components primarily custom designed by PULSE-A

Schedule, Budget, and Funding

Schedule

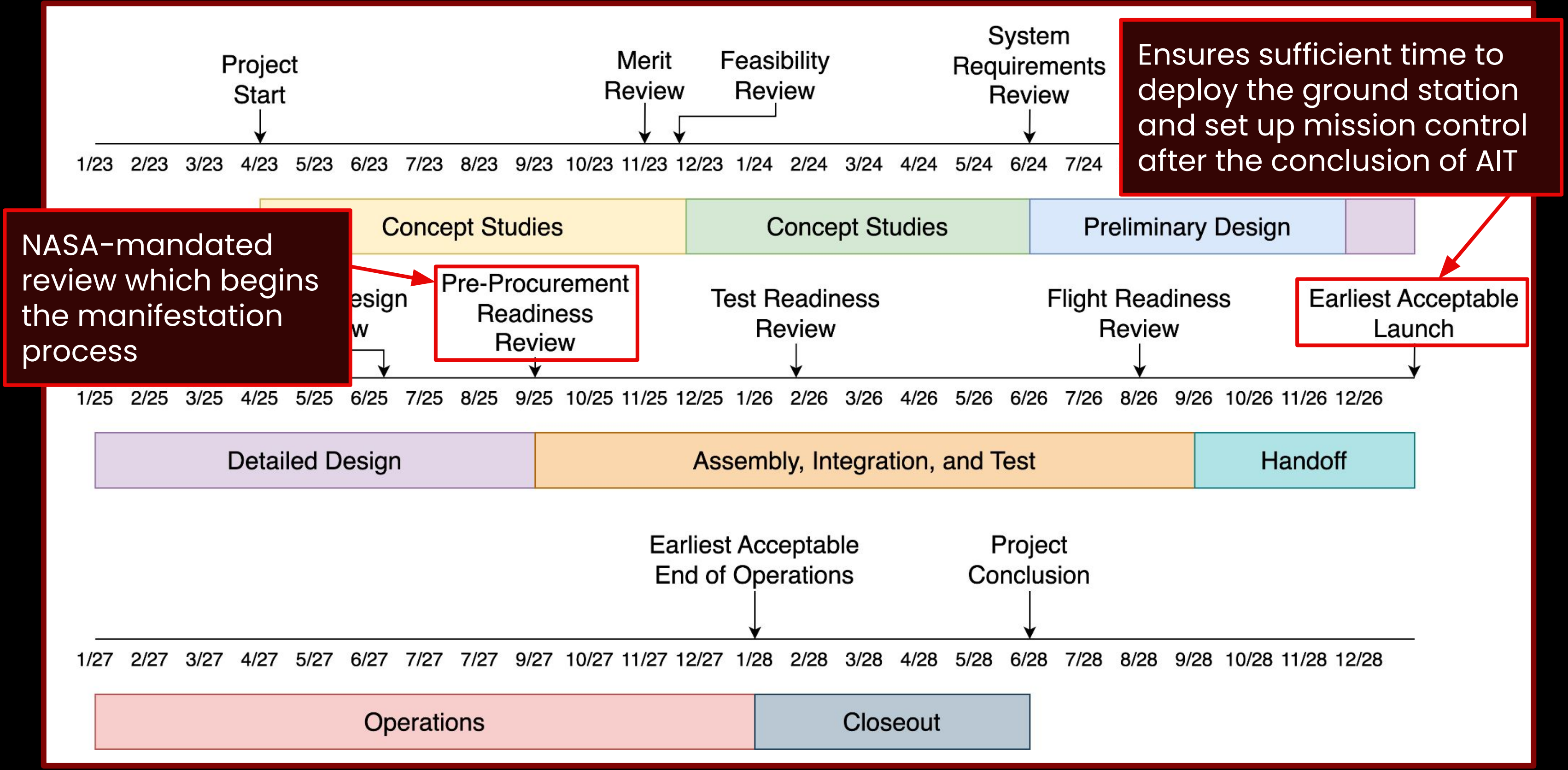

Project Start
Merit Review
Feasibility Review
System Requirements Review
Preliminary Design Review
1/23 2/23 3/23 4/23 5/23 6/23 7/23 8/23 9/23 10/23 11/23 12/23 1/24 2/24 3/24 4/24 5/24 6/24 7/24 8/24 9/24 10/24 11/24 12/24
Concept Studies
Concept Studies
Preliminary Design
Critical Design Review
Pre-Procurement Readiness Review
Test Readiness Review
Flight Readiness Review
Earliest Acceptable Launch
1/25 2/25 3/25 4/25 5/25 6/25 7/25 8/25 9/25 10/25 11/25 12/25 1/26 2/26 3/26 4/26 5/26 6/26 7/26 8/26 9/26 10/26 11/26 12/26
Detailed Design
Assembly, Integration, and Test
Handoff
Earliest Acceptable End of Operations
Project Conclusion
1/27 2/27 3/27 4/27 5/27 6/27 7/27 7/27 9/27 10/27 11/27 12/27 1/28 2/28 3/28 4/28 5/28 6/28 7/28 8/28 9/28 10/28 11/28 12/28
Operations
Closeout

Schedule

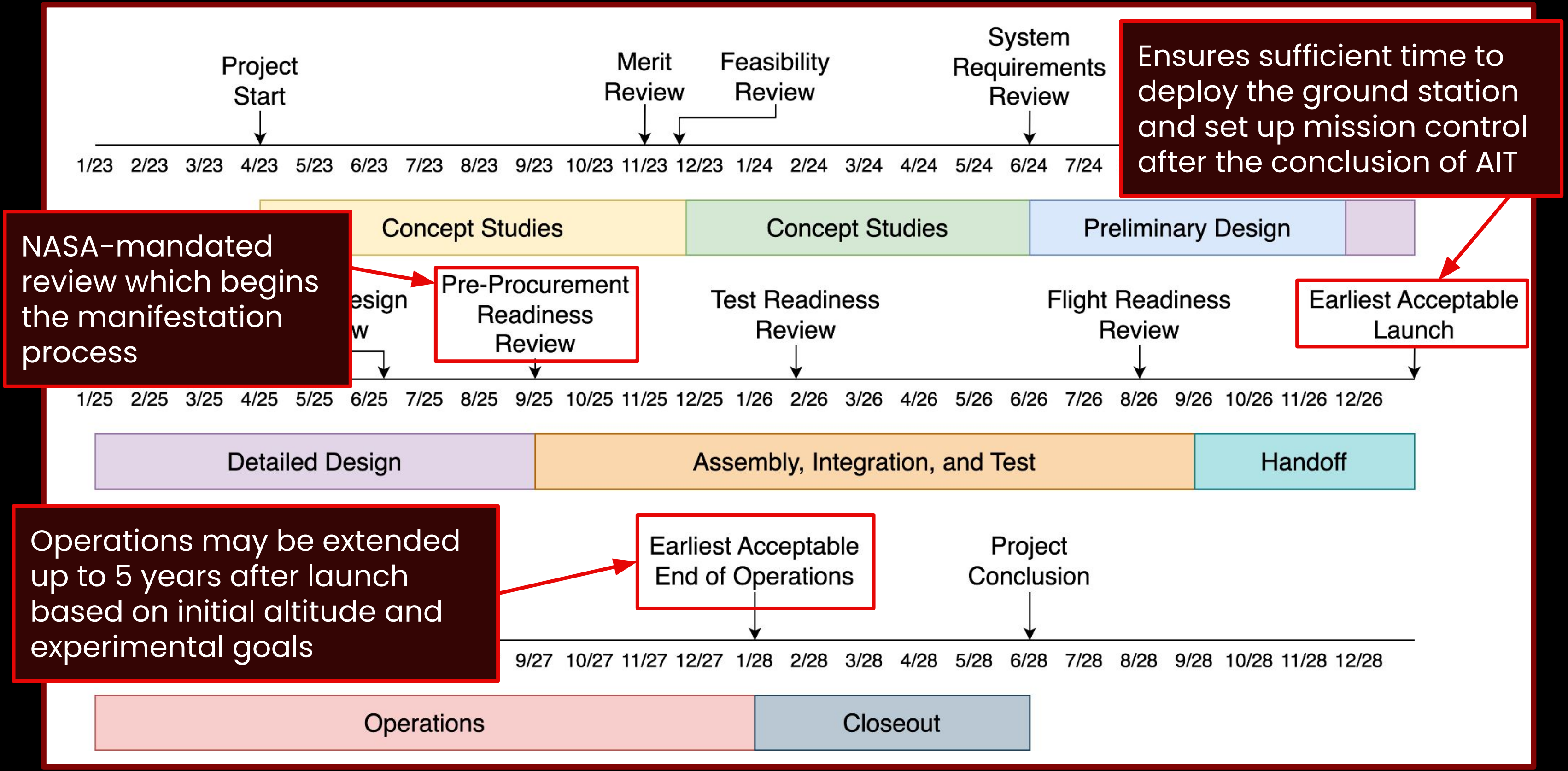


NASA-mandated review which begins the manifestation process

Schedule


Project Start
Merit Review
Feasibility Review
System Requirements Review
Ensures sufficient time to deploy the ground station and set up mission control after the conclusion of AIT
1/23 2/23 3/23 4/23 5/23 6/23 7/23 8/23 9/23 10/23 11/23 12/23 1/24 2/24 3/24 4/24 5/24 6/24 7/24
Concept Studies
Concept Studies
Preliminary Design
NASA-mandated review which begins the manifestation process
Pre-Procurement Readiness Review
Test Readiness Review
Flight Readiness Review
Earliest Acceptable Launch
1/25 2/25 3/25 4/25 5/25 6/25 7/25 8/25 9/25 10/25 11/25 12/25 1/26 2/26 3/26 4/26 5/26 6/26 7/26 8/26 9/26 10/26 11/26 12/26
Detailed Design
Assembly, Integration, and Test
Handoff
Earliest Acceptable End of Operations
Project Conclusion
1/27 2/27 3/27 4/27 5/27 6/27 7/27 7/27 9/27 10/27 11/27 12/27 1/28 2/28 3/28 4/28 5/28 6/28 7/28 8/28 9/28 10/28 11/28 12/28
Operations
Closeout

Schedule


Project Start
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Earliest Acceptable Launch
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Detailed Design
Assembly, Integration, and Test
Handoff
Operations may be extended up to 5 years after launch based on initial altitude and experimental goals
Earliest Acceptable End of Operations
Project Conclusion
9/27 10/27 11/27 12/27 1/28 2/28 3/28 4/28 5/28 6/28 7/28 8/28 9/28 10/28 11/28 12/28
Operations
Closeout

Financial Budget Summary

Subsystem	**Development Budget ($)**
Payload	53,769.96
Structure	5,000.00
Power	1,000.00
CDH	825.00
Communications	9,074.36
Thermal	1,650.00
ADCS	68,296.05
RFGS	6,721.77
OGS	28,599.29

Subsystem	**Development Budget ($)**
AIT and Operations	15,000.00
Total Development Cost	189,936.43
20% Development Contingency (Excludes Launch)	37,987.29
Launch Cost	300,000.00

Total Pre-Contingency Cost	489,936.43
Total Cost	**527,923.72**

Funding

Funds Acquired:

- ❏ NASA CSLI Grant to cover up to $300k in launch costs
- ❏ $25,500 – Private donations, including $25,000 from one generous private donor
- ❏ $25,000 – Academic departments at the University of Chicago
- ❏ $8,000 – University of Chicago's Registered Student Organization (RSO) office
- ❏ $7,500 – SEDS-USA grant
- ❏ $5,000 – Independent fundraising

Current Fundraising Initiatives:

We are focused heavily on alumni outreach, and are currently partnering with the university for promotion during bi-annual giving days. We are waiting to hear back from the Illinois Space Grant Consortium, and are preparing to apply to internal and NSF grants. We anticipate the budget will be closed by the Pre-Procurement Readiness Review.

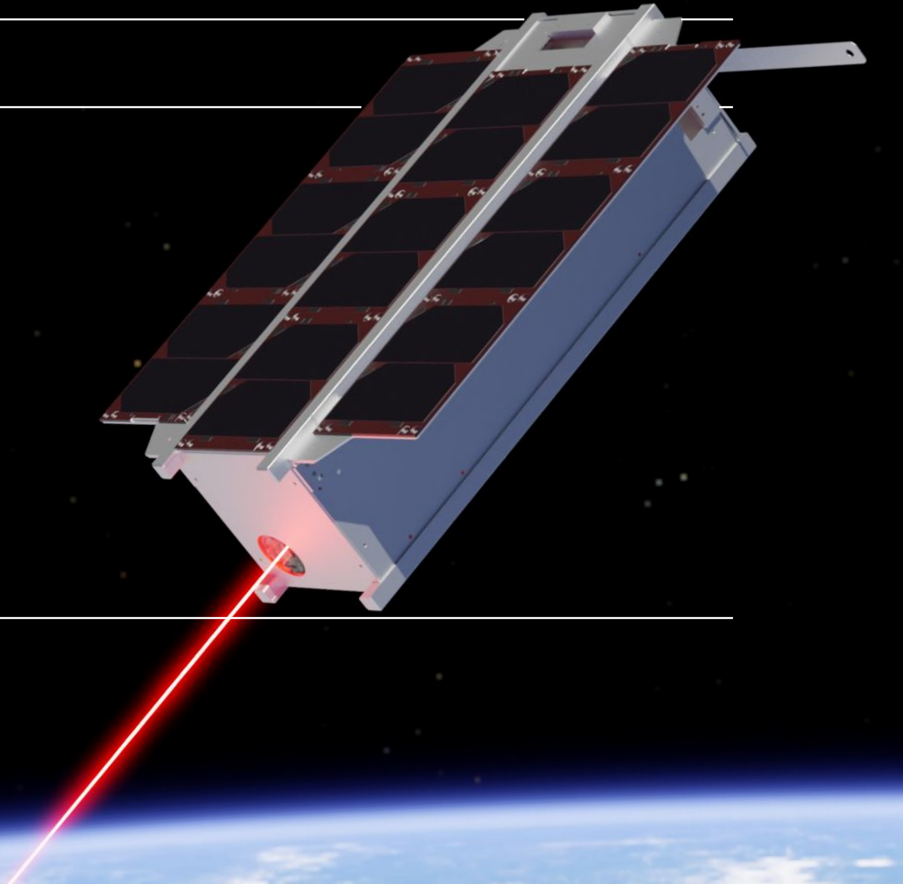
Conclusion

PDR Exit Criteria

- ❑ The top-level requirements including mission success criteria and TPMs are agreed upon, finalized, stated clearly, and consistent with the preliminary design.
- ❑ 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 mission goals and objectives.
- ❑ The preliminary design is expected to meet the requirements at an acceptable level of risk.
- ❑ Definition of the technical interfaces is consistent with the overall technical maturity and provides an acceptable level of risk.
- ❑ Adequate technical interfaces are consistent with the overall technical maturity and provide an acceptable level of risk.
- ❑ Adequate technical margins exist with respect to TPMs.
- ❑ The project risks are understood and have been credibly assessed. A plan, a process, and resources exist to effectively manage them.
- ❑ The operational concept is technically sound, includes (where appropriate) human factors, and is derived from the flow down of requirements for its execution.

UCHICAGO SPACE PROGRAM

THANK YOU!

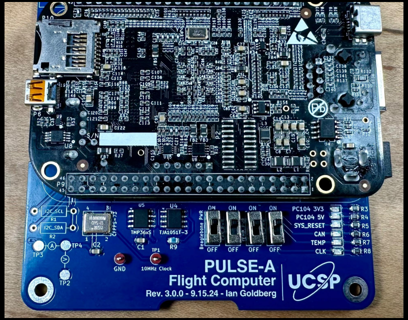


NOVEMBER 23 2024

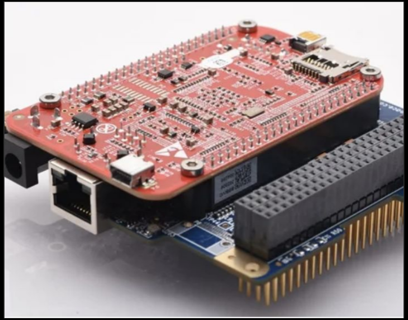
Appendix

C&DH: Comparison to Commercial Option

Name	UCSP On Board Computer (OBC)	Pumpkin Space Motherboard Module 2
Developed by	Beagleboard / PULSE-A	Beagleboard / Pumpkin Space
Cost	<$1000 (including development costs)	$7,225 - $9,800 (per unit)
Description	Beaglebone Black mounted to a PC-104 cape	Beaglebone Black mounted to a PC-104 cape
Watchdog	Radiation resistant External TI MSP430 Maskable Watchdog Processor with 15 KB FRAM	Watchdog timer compatible (No onboard WDT).
Real Time Clock	RTC with battery backup	RTC with battery backup
Extra Storage	SPI-interface 1 Gigabit external NAND memory	SPI-interface SD Card
Readiness	TRL 6 (by launch)	TRL 9



^ PULSE-A OBC



^ Pumpkin Space Motherboard Module 2

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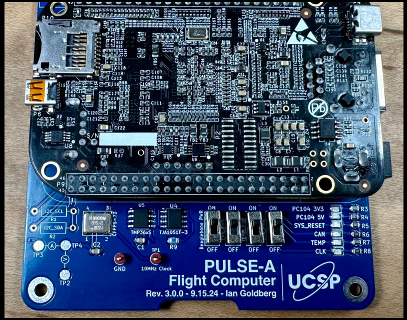
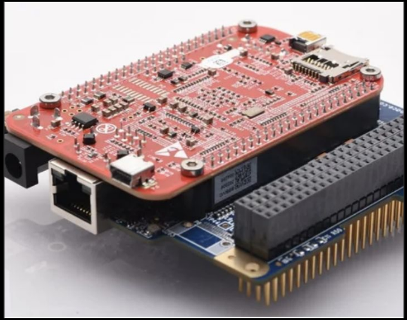
Extra Storage	SPI-interface 1 Gigabit external NAND memory	SPI-interface SD Card
Readiness	TRL 6 (by launch)	TRL 9

Custom-designed OBC selected in favor of commercial option despite low TRL due to cost, capabilities, and knowledge gain for PULSE-A team.



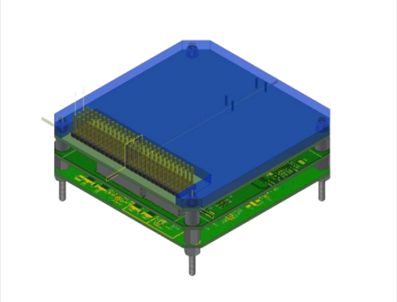
^ PULSE-A OBC



^ Pumpkin Space Motherboard Module 2

Communications Subsystem: Trade Study Result

Radio	**AmSat Linear Transponder**	**GOMspace NanoCom AX100-U**
Directionality	Full duplex	Half-duplex
Frequency	UHF Uplink, VHF Downlink • Two monopole antennas for Rx and Tx	UHF Uplink/Downlink • One monopole antenna
Data Protocol	CAN or I2C	CAN, I2C, or UART
Data Rate	1.2 kbps	0.1 to 38.4 kbps
Data configuration	Telemetry is downlinked as frames of no more than 665 bytes, a 4-byte CRC check, and 3 32-byte Reed Solomon (255,223)	RF parameters are fully configurable on-orbit. E.g. carrier frequency, filter bandwidths, baud rate, framing, etc
Encoding	BPSK	GMSK



^ AmSat LTM 1

^ GomSpace AX100

Communications Subsystem: Trade Study Result

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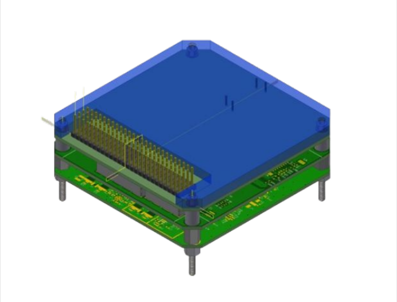
Encoding	BPSK	GMSK

Frame specifications are defined by AMSAT and limit packet transport time flexibility

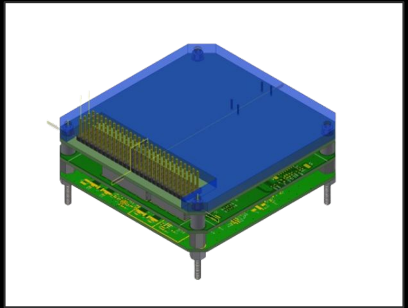


^ AmSat LTM 1

^ GomSpace AX100

Communications Subsystem: Trade Study Result

Radio	**AmSat Linear Transponder**	**GOMspace NanoCom AX100-U**
Price	Free given provided partnership with AmSat on amateur radio applications	[redacted]
Dimensions	92 mm x 92 mm x 20.8 mm	65 mm x 40 mm x 6.5 mm
Global Telemetry	Yes	No



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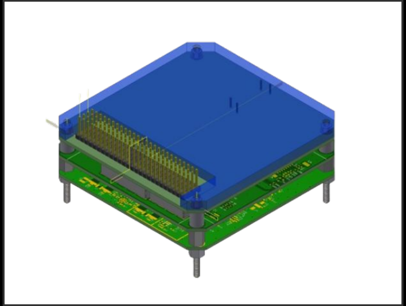
LTM-1 size is a significant concern for volume budget



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Communications Subsystem: Trade Study Result

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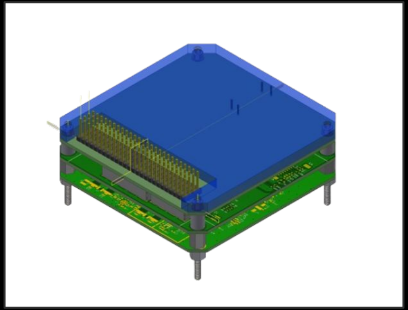
Dimensions	92 mm x 92 mm x 20.8 mm	65 mm x 40 mm x 6.5 mm
Global Telemetry	Yes	No

LTM-1 size is a significant concern for volume budget

TRADE STUDY CONCLUSION: GomSpace AX100 selected for preliminary design based on data rates, volume budget, and lack of flexibility in software usage.



^ AmSat LTM 1

^ GomSpace AX100

Financial Budget (1/3)

Component	Subsystem	Unit Cost ($)	Quantity	Net Cost ($)
638 nm Laser	Payload	$2,132.43	2	$4,264.86
1550 nm Laser	Payload	$500.00	2	$1,000.00
Flight Model EDFA (Space Qualified)	Payload	$13,000.00	1	$13,000.00
Prototype EDFA (Not Space Qualified)	Payload	$5,000.00	1	$5,000.00
Phase Modulator	Payload	$2,000.00	2	$4,000.00
RF Amplifier	Payload	$120.61	2	$241.22
Signal Generator (FPGA)	Payload	$1,249.00	2	$2,498.00
Transmission Collimator	Payload	$376.69	4	$1,506.76
1064 nm Bandpass Filter	Payload	$208.00	2	$416.00
1180 nm Shortpass Dichroic	Payload	$303.98	2	$607.96
Fast Steering Mirror kit	Payload	$7,500.00	2	$15,000.00
Beacon Collimator	Payload	$100.00	2	$200.00
Lens Assembly	Payload	$151.33	2	$302.66
Quarter Waveplate	Payload	$507.25	2	$1,014.50
Quad Cell	Payload	$209.00	2	$418.00
Payload Control Board	Payload	$150.00	2	$300.00
Machined Optical Mounting Hardware	Payload	$4,000.00	1	$4,000.00

Component	Subsystem	Unit Cost ($)	Quantity	Net Cost ($)
3U Frame	Structure	$3,100.00	1	$3,100.00
Prototyping Hardware	Structure	$1,900.00	1	$1,900.00
Battery Assembly	Power	$30.00	5	$150.00
PDU Board	Power	$150.00	5	$750.00
Battery Board	Power	$20.00	5	$100.00
Flight Computer Board	CDH	$165.00	5	$825.00
Antenna Board	Communications	$75.00	5	$375.00
Antenna	Communications	$50.00	3	$150.00
Zenith Active Antenna	Communications	$1,149.36	1	$1,149.36
GOMSpace Radio	Communications		1	
Thermal Sensors	Thermal	$68.00	10	$680.00
Projected Other Thermal Hardware	Thermal	$500.00	1	$500.00
Thermoelectric Heaters	Thermal	$94.00	5	$470.00
CubeSpace Gen2 ADCS Unit	ADCS	$50,000.00	1	$50,000.00
CubeSpace Gen2 Development Model	ADCS	$5,220.00	1	$5,220.00
CubeSpace D2S2 Software License	ADCS	$2,800.00	2	$5,600.00
Celeste GPS	ADCS	$7,476.05	1	$7,476.05
Icom 9700	RFGS	$1,750.00	1	$1,750.00

Financial Budget (2/3)

Component	Subsystem	Unit Cost ($)	Quantity	Net Cost ($)
SDR(Ettus USRP B205mini-i: 1x1, 70MHz-6GHz SDR/Cognitive Radio)	RFGS	$1,500.00	1	$1,500.00
Omnidirectional antenna by M2	RFGS	$350.00	1	$350.00
Camera, PanTiltZoom	RFGS	$300.00	1	$300.00
Yaesu 5500 rotator + controller	RFGS	$760.00	1	$760.00
Rohn JRM23810 non-penetrating roof mount 5' x 5' base, 10' tall	RFGS	$500.00	1	$500.00
Cross Yagi UHF	RFGS	$400.00	1	$400.00
1/2" Hardline, 50 ft	RFGS	$113.00	1	$113.00
LMR 400 Ultraflex, 10 ft	RFGS	$40.00	1	$40.00
SSB Electronic SP70	RFGS	$400.00	1	$400.00
Self-contained Antenna Tracker	RFGS	$275.00	1	$275.00
DCW 2004 B - Power coupler	RFGS	$333.77	1	$333.77
Celestron Telescope	OGS	$4,005.99	1	$4,005.99
1064 nm Laser	OGS	$8,000.00	1	$8,000.00
QWP	OGS	$507.25	1	$507.25

Component	Subsystem	Unit Cost ($)	Quantity	Net Cost ($)
PBS	OGS	$438.33	1	$438.33
Dichroic	OGS	$236.40	1	$236.40
1550nm bandpass filter	OGS	$164.67	1	$164.67
638nm bandpass filter	OGS	$164.67	1	$164.67
Tracking Camera	OGS	$1,280.00	1	$1,280.00
Silver Mirror	OGS	$56.26	1	$56.26
FSM	OGS	$3,000.00	1	$3,000.00
APD	OGS	$2,411.88	2	$4,823.76
Voltage amplifier	OGS	$730.00	4	$2,920.00
I/O - FMC Board	OGS	$159.00	1	$159.00
FPGA	OGS	$589.00	1	$589.00
Fiber	OGS	$35.00	2	$70.00
Fiber Collimator	OGS	$294.53	2	$589.06
Plano-Convex Focusing Lens (f=20mm)	OGS	$118.20	1	$118.20
Plano-Convex Focusing Lens (f=150.5mm)	OGS	$238.78	1	$238.78
Plano-Convex Focusing Lens (f=40.1mm)	OGS	$117.92	1	$117.92
Ballast Mass	OGS	$100.00	1	$100.00

Financial Budget (3/3)

Component	Subsystem	Unit Cost ($)	Quantity	Net Cost ($)
Comparator circuit	OGS	$20.00	1	$20.00
Mounting	OGS	$1,000.00	1	$1,000.00
Transportation	AIT and Operations	$5,000.00	1	$5,000.00
Testing Facilities	AIT and Operations	$10,000.00	1	$10,000.00
Launch Service	Launch	$300,000.00	1	$300,000.00
20% of Development Cost	Contingency	$37,987.29	1	$37,987.29

The total cost of PULSE-A is $527,923.72.

Financial Budget (3/3)

Component	Subsystem	Unit Cost ($)	Quantity	Net Cost ($)
Comparator circuit	OGS	$20.00	1	$20.00
Mounting	OGS	$1,000.00	1	$1,000.00
Transportation	AIT and Operations	$5,000.00	1	$5,000.00
Testing Facilities	AIT and Operations	$10,000.00	1	$10,000.00
Launch Service	Launch	$300,000.00	1	$300,000.00
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Launch service is subsidized by NASA up to $300,000 through the CubeSat Launch Initiative.

The total cost of PULSE-A is $527,923.72.

Financial Budget (3/3)

Component	Subsystem	Unit Cost ($)	Quantity	Net Cost ($)
Comparator circuit	OGS	$20.00	1	$20.00
Mounting	OGS	$1,000.00	1	$1,000.00
Transportation	AIT and Operations	$5,000.00	1	$5,000.00
Testing Facilities	AIT and Operations	$10,000.00	1	$10,000.00
Launch Service	Launch	$300,000.00	1	$300,000.00
20% of Development Cost	Contingency	$37,987.29	1	$37,987.29

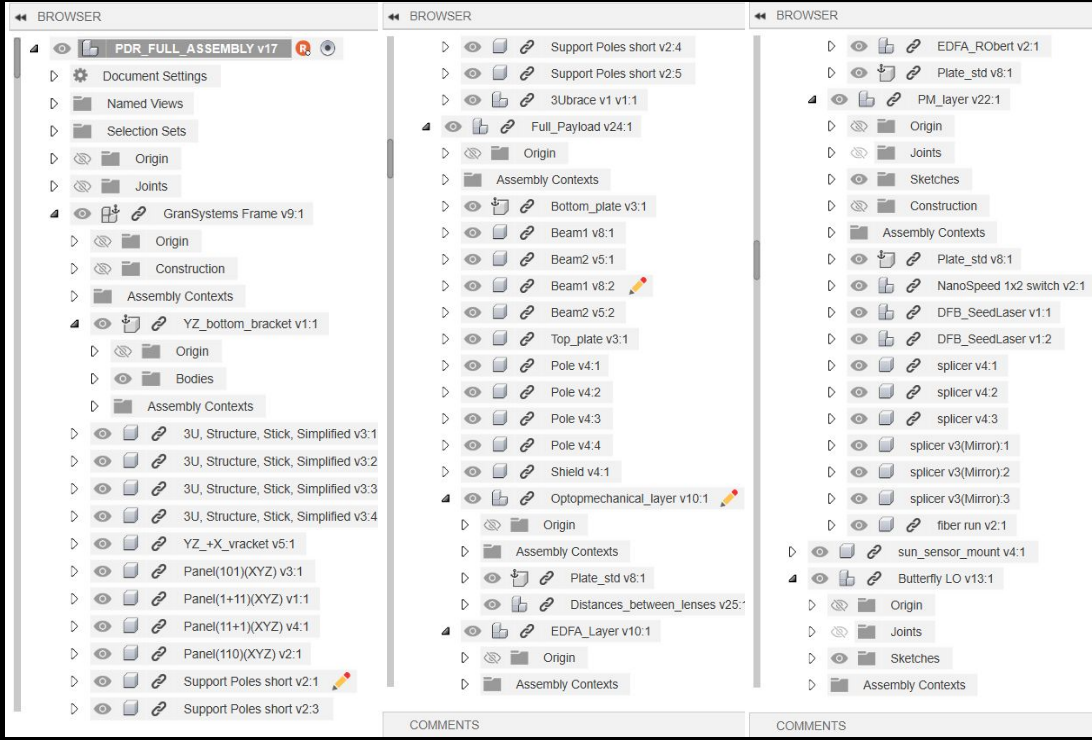
Launch service is subsidized by NASA up to $300,000 through the CubeSat Launch Initiative.

Contingency does not account for launch costs.

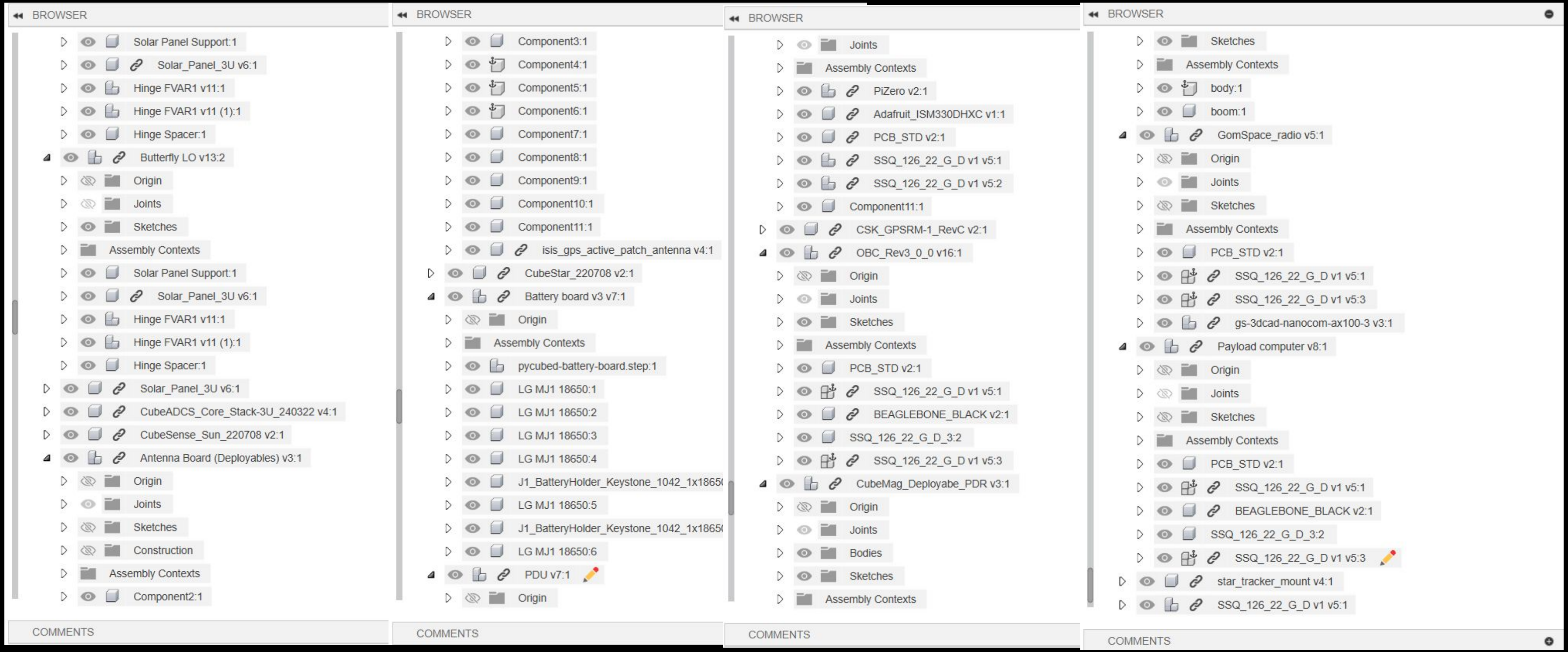
The total cost of PULSE-A is $527,923.72.

Engineering Drawing Tree

- Files from manufacturers imported in Fusion 360 compatible formats
- Assemblies expanded to show components and subcomponents
- Body-level not shown for readability
- Continues on the next slide



Engineering Drawing Tree



Radio Frequency Link Budget Summary (Amsat)

Downlink budget

Parameter	Value
Data Rate	1200 bps
Transmit power with LTM	500 mW low power max, 1W High Power Max
Transmit Power	100 mW
Frequency	435 – 438 MHz
Modulation	1200 BPSK
Channel	30 kHz
Link Margin	**10.1 dB**

Uplink budget

Parameter	Value
Data Rate	1200 bps
Transmit power	15W
Frequency	144 – 146 MHz
Modulation	1200 AFSK-FM
Channel	30 kHz
Link Margin	**17.7 dB**

Power Budget (Nominal Orbit, No Laser Transmit)

Component	Subsystem	Current (A)	Voltage (V)	Quantity	Power (W)	Margin (%)	Design Power (W)	Duty Cycle (%)	Consumption (W)
On Board Computer	C&DH	0.300 A	5.0 V	1	1.500 W	20%	1.800 W	100%	1.800 W
Watchdog Processor	C&DH	0.008 A	3.3 V	2	0.053 W	5%	0.055 W	100%	0.055 W
Payload Controller	C&DH	0.300 A	5.0 V	1	1.500 W	20%	1.800 W	50%	0.900 W
Power Distribution Unit	Power	0.015 A	7.2 V	1	0.108 W	20%	0.130 W	100%	0.130 W
Battery Heater	Power	0.300 A	5.0 V	2	3.000 W	20%	3.600 W	5%	0.180 W
Magnetorquers	ADCS	0.051 A	7.2 V	3	1.110 W	5%	1.166 W	10%	0.117 W
CubeMag	ADCS	0.007 A	7.2 V	1	0.050 W	5%	0.053 W	100%	0.053 W
Reaction Wheels (Peak)	ADCS	0.118 A	7.2 V	3	2.550 W	5%	2.678 W	5%	0.134 W
Reaction Wheels (Nominal)	ADCS	0.025 A	7.2 V	3	0.540 W	5%	0.567 W	95%	0.539 W
ADCS Core	ADCS	0.127 A	7.2 V	1	0.918 W	5%	0.964 W	100%	0.964 W
Sun Sensor (Nominal)	ADCS	0.030 A	3.3 V	1	0.100 W	5%	0.105 W	98%	0.103 W
Sun Sensor (Peak)	ADCS	0.053 A	3.3 V	1	0.175 W	5%	0.184 W	2%	0.004 W
Star Tracker (Nominal)	ADCS	0.050 A	3.3 V	1	0.165 W	5%	0.173 W	98%	0.170 W
Star Tracker (Peak)	ADCS	0.082 A	3.3 V	1	0.271 W	5%	0.284 W	2%	0.006 W
GPS	ADCS	0.020 A	5.0 V	1	0.100 W	5%	0.105 W	100%	0.105 W
Active Antenna	Comms	0.010 A	5.0 V	1	0.050 W	5%	0.053 W	100%	0.053 W
Radio Rx	Comms	0.055 A	3.3 V	1	0.182 W	5%	0.191 W	100%	0.191 W
Radio Tx	Comms	0.800 A	3.3 V	1	2.640 W	5%	2.772 W	16%	0.444 W
Thermal Sensors	Thermal	0.0001 A	5.0 V	8	0.002 W	5%	0.002 W	100%	0.002 W
								OAP (W)	**5.15 W**
								Peak (W)	**15.83 W**
								Energy (Wh)	**7.73 Wh**
								Energy in Eclipse (Wh)	**2.99 Wh**

Energy = OAP × Orbital Period →

Eclipse Energy = OAP × Time in Eclipse (~0.57 h) →

PAT Sequence Breakdown

- Previous Pass
  1. Pass Verification
     a. Verify orbital and safety parameters before PAT pass
- PAT Pass
  1. Verification
     a. Receive authorization from human on the ground to begin optical activities
  2. RF Connect
     a. Ensure RF connection before optical connection
     b. Begin exchanging telemetry and gps information
  3. Beacon Lasers Enabled
     a. Enable OGS and Satellite beacons to begin tracking
  4. FSM Search
     a. Assume accurate body pointing and begin FSM search
     b. If FSM search succeeds at any point skip to Transmit Laser Enabled
  5. Body Search
     a. If OGS fails to detect satellite beacon laser after a timeout, satellite body will begin a discrete spiral search
     b. FSM search paused until uplinked data verifies OGS detects satellite beacon laser

PAT Sequence Breakdown

- PAT Pass
  6. Ground Beacon Laser Alignment
     a. FSM alignment allows for quadcelll to detect OGS beacon laser
     b. Quadcell data used to adjust FSM to ensure OGS beacon laser is aligned to the center of the quadcell
  7. Transmit Laser Enabled
     a. Transmist laser enabled using quadcell OGS beacon data to ensure alignment with ground station
  8. Data Transmit
     a. FEC encoded data is transmitted through the transmit laser
     b. FEC encoded data detected and decoded by the OGS
  9. Laser Shutdown
     a. At defined angle transmission laser and beacon lasers are shutdown
     b. Telemetry verifies laser shutdown state of satellite before end of pass
- Next Pass
  1. Log File Transmission
     a. Detailed timestamped log files are sent down from the satellite to allow for analysis and debugging