TEAM 12
PACIFIC PIONEERS

Fire Risk Observation System (FROST) Constellation - Initial Analysis

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TEAM 12 PACIFIC PIONEERS

1 SYSTEM ENGINEERING EXECUTIVE SUMMARY

1.1 Executive Summary (Brown)

The Pacific Pioneers responded to a study contract by the US Government customer. The proposed system is a constellation of 9 small satellites that provide the capability for real-time alerting for risk of fires. Each satellite carries a payload used for surveillance of the Earth's surface. The satellite constellation communicates with the nearest Ground Entry Point, or can transmit alert messages via crosslink until a GEP can be reached. This document includes Concept of Operation, Key Requirements, Key Trades, Preliminary System Budgets, and Subsystem Designs for the proposed conceptual design.

1.2 Mission CONOPS (Demyanek)

Full-sized CONOPS can be viewed in enclosure 1.

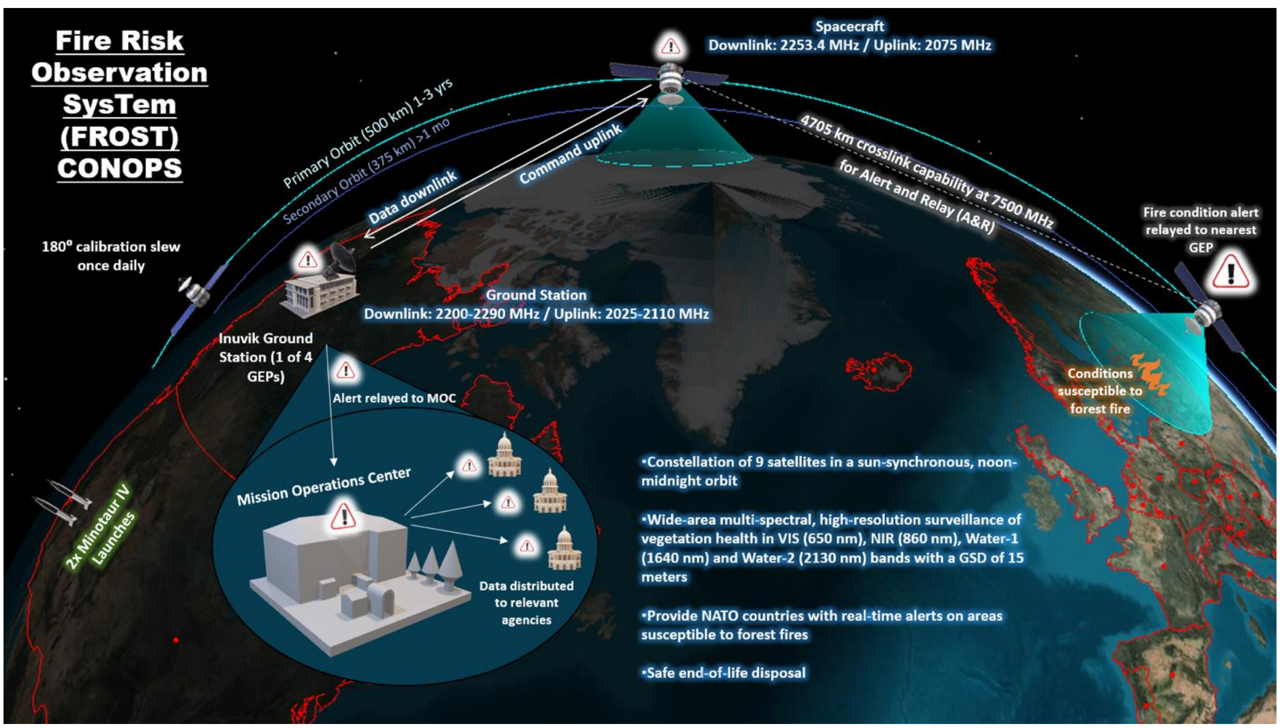

Figure 1 FROST CONOPS

1.3 Key Requirement Drivers (Rosenberg)

The FROST mission has a total of nine Key Requirement Drivers that led to the development of the System Requirements Document (found in the data package).

Driver Number	Description	Reasoning
KRD-01	Payload ground sample distance drove payload dimension limitations	Due to required Ground Sample Distance of 15m, payload exceeds NTE payload volume above 641km.
KRD-02	Primary and Secondary Mission orbit heights	Primary and Secondary Mission orbits have different heights, which necessitates a propulsion system
KRD-03	Near-real times A&R messages	Requires an orbit that communicates with a Ground Entry Point (GEP) no more than every 10 minutes. Requires constellation crosslinks.
KRD-04	Twice daily coverage of target areas	This drove the final selection of orbit height and number of SVs.
KRD-05	Payload pointing requirement of 0.1 degrees	Mandates the use of a 3-Axis Control or Spin Stabilization for spacecraft method of control. Other KRD drove decision to 3-Axis Control.
KRD-06	Payload calibration slew	Required agility drove decision of method of control to 3-Axis Control
KRD-07	Usage of Minotaur IV launch vehicle	Customer specified mission choice between Minotaur I and IV. Minotaur IV was selected as it is more economical to launch 9 satellites with.
KRD-08	ESPA size constraint	The customer has specified that the mission conform to ESPA constraints (mass, volume, etc.)

Table 1 Key Requirements

1.4 Key Trade Studies (Brown)

The trade studies that provided the essential information for conceptual mission and spacecraft design are as follows:

- Orbital altitude
- Processor architecture
- Power regulation
- Actuators
- Launch vehicle selection

1.4.1 Trade Study 1 – Payload: Orbital altitude

The purpose of this payload trade study is to evaluate the primary mission orbit between 500 km and 1000 km.

1.4.2 Trade Study 3 – C&DH: Processor architecture

The purpose of this C&DH trade study is to evaluate the processor architecture between central and distributed.

1.4.3 Trade Study 4 – Power: Power Regulation

The purpose of the power trade is to evaluate the power regulation between Direct Energy Transfer (DET) and Power Point Tracking (PPT).

1.4.4 Trade Study 8 – ADCA: Method of Attitude Control

The purpose of the ADC trade is to determine the most efficient means of controlling system momentum.

1.4.5 Trade Study 10 – Configuration, Structures, Launch: Launch vehicle selection

The purpose of the launch trade is to evaluate the launch vehicle performance between Minotaur I and Minotaur IV.

1.5 Final TOR (Brown)

Reference online blackboard TOR tool.

2 PRELIMINARY SYSTEM DESIGN BUDGETS

2.1 Mass (Brown)

2.1.1 Mass Budget Overview

Full subsystem mass allocations are shown in the table below.

Subsystem	% max SV dry mass	Max Expected Value (kg)	Allocated Margin (%)	Allocated Margin (kg)	Current Best Estimate (kg)
Payload	30.3%	40.00	0%	0.00	40.00
Structure and Mechanisms	27%	35.64	15%	4.65	30.99
Thermal	2%	2.64	15%	0.34	2.30
Power (harness)	21%	27.72	15%	3.62	24.10
TT&C	2%	2.64	15%	0.34	2.30
C&DH	5%	6.60	15%	0.86	5.74
ADCS	6%	7.92	15%	1.03	6.89
Propulsion	3%	3.96	15%	0.52	3.44
Other	3.7%	4.88	15%	0.64	4.24
Total	100.0%	132.00		12.00	120.00

Table 2 Mass Budget

2.2 Power (Rosa)

2.2.1 Power subsystem allocations

Breakdowns of power allocations are displayed in table 3. A 20% margin and a 1.3 contingency factor has been applied to each subsystem's values. Throughout the design lifecycle, margins can be decreased, but not to exceed the limit power values. Margin and contingency were not applied to the payload power.

Subsystem	SMAD Avg. Power Draw	Limit (W)	Margin (20%)	Contingency (1.3)	Midpoint Base (W)
Payload	43%	60	0.00	0.00	60.00
Structure	0%	0	0.00	0.00	0.00
Thermal	5%	7	1.40	1.29	4.29
Power (harness)	10%	14	2.79	2.58	8.59
TT&C	11%	15	3.07	2.83	9.45
Processing	13%	18	3.63	3.35	11.16
ADCS	18%	25	5.02	4.64	15.46
Propulsion	0%	0	0.00	0.00	0.00
Total Power	100%	139.53	15.91	14.68	108.94

Table 3 Power Subsystem Allocations

2.2.2 Operating States

The power budget is based off of three power modes:

1. Payload On
2. Payload Off
3. Payload Calibrate

Since the payload is on over land and off over the ocean, this was the logical way to break out the modes. We also assumed that the payload would only enter calibrate mode when over water so as to not take away from valuable data collection time when over land (20% of overwater time). Total Orbit Average Power (OAP) is calculated as 73.2 W. A summary of the breakdown is depicted in table 4 below, but full details are outlined in enclosure 8.

	POWER MODE 1			POWER MODE 2			POWER MODE 3		
Component	**Peak Power (W)**	**Duty Cycle**	**Avg Power (W)**	**Peak Power (W)**	**Duty Cycle**	**Avg Power (W)**	**Peak Power (W)**	**Duty Cycle**	**Avg Power (W)**
C&DH	11	100%	11	11	100%	11.2	11.2	100%	11.2
ADCS	15	100%	15	15	100%	15.5	15.5	100%	15.5
TT&C	9	100%	9	9	100%	9.4	9.4	100%	9.4
Thermal	4	100%	4	4	100%	4.3	4.3	100%	4.3
Power	9	100%	9	9	100%	8.6	8.6	100%	8.6
Payload	60	100%	60						
Payload				0	100%	0			
Payload							30	100%	30
	Peak		**Avg**	**Peak**		**Avg**	**Peak**		**Avg**
TOTAL	108.9		**108.9**	48.9		**48.9**	78.9		**78.9**
DUTY CYCLE		32%			47%			21%	
OAP			**34.5**			**23.1**			**16.7**
Total OAP									**74.3**

Table 4 Power Budget

2.3 Delta-V (Rosenberg)

2.3.1 Delta-V Budget Overview

The purpose of the mission Delta-V budget is to tabulate the propellant required per space vehicle to perform all propulsive activities such as initial constellation deployment, payload calibration slewing, constellation station keeping, changing orbital height, and more. All propulsive activities are listed in table 5. As the I_{sp} of the intended monopropellant-based propulsion system is 210 seconds per the customer specifications, the total Delta-V can be taken and converted to the amount of propellant required. This budget will drive thruster sizing and propellant tank sizing.

Type	Identifier	Description	Mission Phase
Maneuver	MAN-01	Constellation deployment	Deployment
Maneuver	MAN-02	Keep SV in constellation formation	All
Maneuver	MAN-03	Lower orbital height from Primary Mission to Secondary Mission	Secondary Mission
Attitude Change	ATT-01	Slew SV to point payload to and from deep space for calibration	Primary Mission, Secondary Mission

Table 5 Propulsion Maneuvers

2.4 RF Link Analysis (Rosa)

The details below describe specific aspects of the link budget and link analysis results. The full link budget with all calculated parameters is outlined in enclosure 10.

2.4.1 Spacecraft to GEP Data Rate

Pacific Pioneers has decided to use 4 GEPs. With an elevation angle constraint of 10 degrees, each satellite will be overhead the GEP for an average of approximately 5.5 minutes as outlined in the mission design summary in section 4.1. Each GEP can support a maximum data rate of 5Mbps. Based on the amount of data that needs to be downlinked as outlined in section 2.5, the spacecraft will need to use the max of 5Mbps downlink rate, uncoded. Each GEP does support coding, but the spacecraft was not constrained by power output and so the decision was made to use an uncoded downlink. Uncoded downlink was also chosen to keep the total data size down. With the desired requirement to store and downlink all payload data, this max data rate will not be sufficient to downlink all payload data. Pacific Pioneers recommends using the s-band antenna for housekeeping and alert messages only, and adding a dedicated medium gain antenna dedicated only to payload data (using a frequency that allow for a larger bandwidth, such as x-band or Ka band). This recommendation is further explained in section 4.1.4.

2.4.2 Downlink Parameters

With a 2.7 in diameter antenna and outlined parameters in table 6 below, an antenna gain of 1.71 dBi and beamwidth of 134 degrees results in a required transmitter power of .377 watts to close the link.

SPACECRAFT DOWNLINK		
Downlink Frequency	2253.4	MHz
Downlink Wavelength	0.1330	m
Transmitter Power	0.377	W
Transmitter Power	-4.23	dBW
SC Passive Loss	3.0	dB
SC Ant Tx Power	-7.2	dBW
SC Ant Beamwidth	134	deg
Bandwidth Factor κ	70	
SC Antenna d/λ	0.52	
SC Antenna Directivity	2.70	
Antenna Efficiency	55%	
SC Antenna Gain	1.71	dBi
Downlink EIRP	-5.52	dBWi
Link Eb/No	15.8	dB
Antenna Diameter	2.7	in
Downlink Data Rate	5000	Mbps

Table 6 Downlink Parameters

The spacecraft is limited to communications with a GEP only when it is within 10 degrees of the horizon. At 10 degrees, the link margin is 3dB and increases to 13.6 dB at nadir, as outlined in figure 2 below.

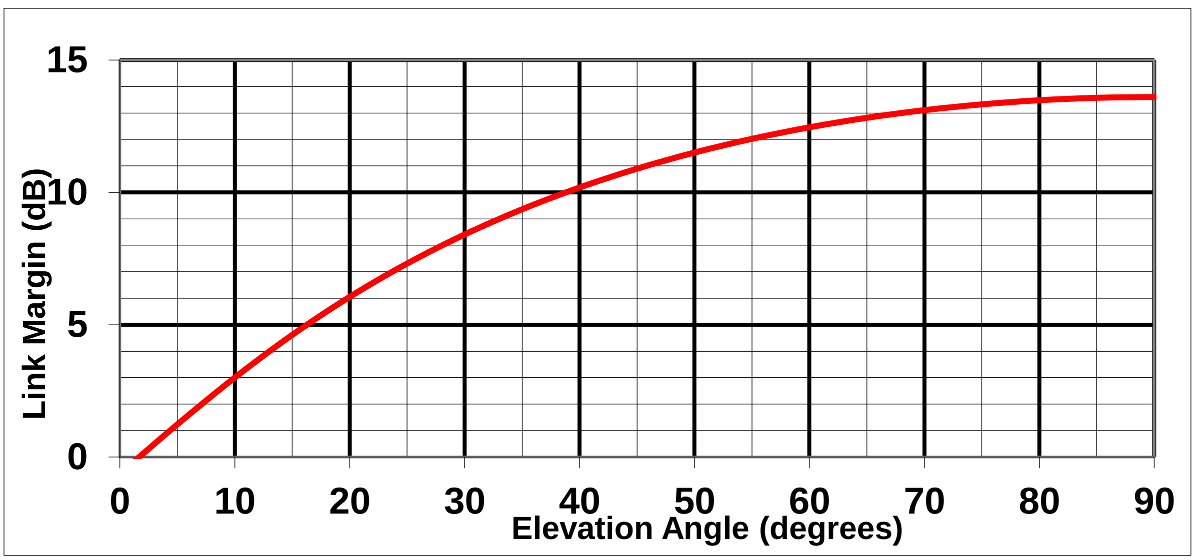

Figure 2 Link Margin

2.4.3 Crosslink Data Rate

The crosslink Alert and Relay (A&R) message is a 1064 byte message every 60 seconds. This translates into an 8512 bit message. For worst-case traffic, we can assume that 8 satellites have generated an alert message and those 8 alert messages are being sent to the 9th satellite for downlink to the GEP. This calculates to a total data traffic size of 68 kbits bits. With 100% margin, this translates to a traffic size of 137 kbits. A data rate of 150 kbps is sufficient to account for this traffic size, and when coded, allows for even lower power requirements. Using 0.5 FEC and antenna diameter of 0.2m and gain of 21.33 dBi, a crosslink power output of 2.66W is required. This power output required could be reduced to around 1W by increasing the antenna diameter to .25 meters. This analysis could be conducted in a future trade.

Link Margin	**3**	**dB**
Implementation Loss	**4**	**dB**
.5 FEC Coding Gain	**3.5**	**dB**
Reqd 1E-6 BER Eb/No	**10.8**	**dB**
Threshold Eb/No	**14.3**	**dB**
Data Rate	**150**	**kbps**
C/No	**66.06**	**dB**
Receiver NF	**2**	**dB**
Rx Noise Temp	**170**	**K**
Input Attenuation (Ai)	**1**	**dB**
Equiv Input Noise Temp	**289**	**K**
Clear Sky Temperature	**30**	**K**
Clear Sky System Temp	**319**	**K**
Antenna Beamwidth	**14**	**deg**
Crosslink Antenna d/λ	**5**	
Antenna Directivity	**246.74**	
Antenna Efficiency	**0.55**	
Antenna Gain	**21.33**	**dBi**
Terminal G/T	**-3.71**	**dB-K**
Frequency	**7500**	**MHz**
Wavelength	**0.04**	**m**
Antenna Diameter	**0.20**	**m**
Rx Isotropic Power	**-158.83**	**dBW**
Slant Range (9 sat ring)	**4705**	**km**
Space Loss	**183.40**	**dB**
EIRP	**24.57**	**dBW**
Antenna Gain	**21.33**	**dBi**
Antenna Input Power	**3.25**	**dBW**
Passive Loss	**1**	**dB**
Transmitter Power	**4.2**	**dBW**
Transmitter Power	**2.66**	**W**

Table 7 Crosslink Parameters

2.5 Data (Teste)

The data budget is an analysis examining (1) how the various spacecraft subsystem requirements impact data budget, (2) determine data constraints, and (3) identify solutions if compromises are necessary.

2.5.1 Data Budget Overview

During operation, the spacecraft generates 0.1908 Kbps (see Figure 9). Each 94.6-minute orbit thus makes 16.46 Mb of housekeeping data. This assumes that all subsystems are continually in use and thus outputting maximum data as well as a ~15% margin for data collected. On the other hand, the payload data generated is 14.46 Mbps. Each orbit has an average duty cycle of 31.7% thus resulting in 26.02 Gb of payload data per orbit. This assumes data collected has a 15% margin and that all data obtained is stored and kept, resulting in 26.02 Gb of storage capacity per orbit. With a contingency factor of 1.3, an extra 7.8 Gb of storage is recommended. This brings the total storage capacity to 33.83Gb, per orbit without downlinking the information. Full details are outlined in enclosure 11.

HK Data Rate	0.1908	Kbps
HK Duty Cycle	100	%
HK Data per orbit	1.083	Mb
HK Downlink Time	3.30	s
HK CBE Data Per Day	16.49	Mb

Payload Data Rate	14.46	Mbps
Payload Duty Cycle	31.7	%
Payload Data per orbit	2602.8	Mb
Payload Downlink Time	5204	s
Payload CBE Data Per Day	396.1	Gb

Total Data Rate	14.46	Mbps
Total Downlink Time	5207.5	sec
Total CBE Data Per Day	396.1	Gb
Total data/orbit	26.02	Gb

Table 8 Data Budget

2.5.2 Data constraints

The payload data generated per orbit is 26.02 Gb. At the nominal maximum downlink rate, this would take 5204 seconds to fully downlink. However, the maximum contact duration with a GEP is merely 304 seconds. It is clear to see the incompatibility of these two constraints; it would take 18 contacts of this type to completely downlink a single orbit's payload data. Further expounding on this, if the data were to be cross-linked to other satellites to downlinked from there, this would delay data transfer for other members of the constellation.

2.5.3 Data solutions

As the data generated by the payload vastly outpaces the downlink capacity per orbit, several options are available to address this issue. See section 4.1.4 for details.

3 PROPOSED PAYLOAD CONCEPTUAL DESIGN (DEMYANEK)

3.1 Telescope Design Specs and Earth Coverage Trade Study

A final orbital altitude of 500 km was decided upon for this proposal. Table 9 details the associated payload parameters at this altitude. The analysis traded payload specifications at orbital altitudes from 500 to 1000 km in intervals of 100 km. At altitudes at and above 700 km (641 km was found to be the absolute maximum), the 50 cm not-to-exceed value for telescope length is exceeded (This is calculated using a rule-of-thumb that the physical length is 65% of the effective focal length). Orbital altitudes of 500 km and 600 km are both suitable for payload volume. A 500 km orbit maximizes the Area Coverage Rate (ACR), or surface area of the earth that can be analyzed for fire susceptible vegetation per unit time. Additionally, a 500 km orbit minimizes the physical length required of the telescope, resulting in the lowest total payload volume and therefore lowest total payload mass. An earth coverage analysis was conducted in STK using these payload values and can be found in section 4.1.4. The tabular data from this trade study can be found in enclosure 12.

Ground Track Velocity	7058.75 m/s
Instantaneous Field of View (IFOV)	30 microradians
Effective Focal Length (EFL)	60 cm
Physical Telescope Length	39 cm
Cross-Track and Along-Track Full Field of View (FFOV)	1.76 degrees x 1.76 degrees
Cross-Track and Along-Track Ground Swath	15.36 km x 15.36 km
Exposure Time	0.00106 seconds
Area Coverage Rate	222049.06 km^2/s

Table 9 Telescope Design Specifications at 500 km

3.2 Q-Value Analysis

An analysis was conducted for f/#’s between 0 and 6 in increments of 0.5 for the payload at an orbital altitude of 500 km. At f/#’s of 2.5 and below, the mechanical diameter exceeds 50 cm. At f/#’s of 3.5 and above, the minimum

SNR of 110 is not met. An f/# of 3 meets all payload requirements and does not exceed any payload constraints. An f/# of 3 indicates that the proposed system is a relatively high light collecting, faster system. A Q-value of 0.355 indicates that the core diameter of the point spread function will be just larger than a single pixel so the system will produce high quality, reliable images. An EPD of 20 cm is a wide enough aperture to satisfy the imaging requirements. The mechanical diameter, D, of 43 cm is narrow enough to fit within the physical constraint of 50 cm (The mechanical diameter is determined for this study using the rule of thumb that D is 2.15 times the EPD for TMA systems). A signal-to-noise ratio of approximately 120 for the dimmest target means that we can have high confidence that the content of the images being produced is accurate and not corrupted by noise. The proposed payload will produce quality images for measuring Normalized Difference Vegetation Index (NDVI) and Normalized Multiband Difference Index (NMDI). The tabular data from this trade study can be found in enclosure 13.

Entrance Pupil Diameter (EPD)	20 cm
Mechanical Diameter (D)	43 cm
Q-value	0.355
Signal-to-noise Ratio (SNR)	119.82

Table 10 Q-Value Analysis

3.3 Electrical Design Spec

The Teledyne H1RG HgCdTe detector will capture the high-quality images needed for DNVI and NMDI measurements in four distinct bands that are described in section 3.4. The electronic design specifications of this detector are detailed in table 11. The H1RG focal plane array interfaces directly with the SIDECAR application-specific integrated circuit (ASIC). The SIDECAR ASIC significantly reduces the power, size and weight required by the focal plane electronics. It is fully programmable. The 10 MHz, 12-bit analog-to-digital converter (ADC) option is being considered (as opposed to the 500 kHz 16-bit ADC) to decrease payload data rate (enclosure 14). An analysis should be conducted to account for the increased noise (about 40 electrons CDS) associated with this high-speed readout mode.

Pixel Pitch	18 microns
Number of Pixels (Cross-Track and Along-Track)	1024 x 1024
Pixel Full Well	80,000 e-
Average Quantum Efficiency for each band	70%
Read Noise	15.5 e-
Dark Current	0.05 e-/pixel/s

Table 11 Electrical Payload Design Specifications

3.4 Optical Design Spec

The proposed payload will make wide-area multi-spectral, high-resolution observations of vegetation in the visible (VIS), near-infrared (NIR) and short-wave infrared (Water-1/2) bands. The central wavelength at each band is listed in table 12 and is key for the NDVI and NMDI measurements being taken. Four filter strips will be laid in the along-track dimension to image the area of interest at each wavelength.

Center Wavelength for each band	VIS: 650 nm NIR: 860 nm Water-1: 1640 nm Water-2: 2130 nm
Average Filter Transmission	90%
Full Field of View (Cross-Track and Along-Track)	1.76 x 1.76 degrees
Entrance Pupil Diameter (EPD)	20 cm
Effective Focal Length (EFL)	60 cm
f/#	3

Table 12 Optical Payload Design Specifications

3.5 Source Design Spec

The photon radiance from the dimmest target in "Table 2: Target Photon Radiance for the Payload Instrument" is Vegetation: Wet in the Water-2 band with a radiance of 6.94x10^17 photons/second/m^2/sr.

3.6 System Design Spec

The following table contains the final system design specifications for the proposed payload.

Ground Sample Distance (GSD)	15 m
Platform Jitter	5 µrad RMS at > 1000 Hz
Platform Drift	0.5
Orbital Altitude	500 km
Ground Track Velocity	7.059 km/s
Instantaneous Field of View (IFOV)	30 microradians
Ground Swath (Cross and Along-Track)	15.36 x 15.36 km
Exposure Time	0.00106 seconds
Area Coverage Rate (ACR)	222,049.06 km^2/s
Q-Value	0.355
SNR for Dimmest Target	119.82
Mechanical Diameter (D)	43 cm
Mechanical Length	39 cm
Payload Volume	0.018 m^2

Table 13 System Payload Design Specifications

3.7 Signal Saturation

This section discusses signals received from the brightest and dimmest target in each band, and how to avoid saturation of the optical system.

	Category	Radiance (Lp)	Signal (e-)
VIS brightest	Soil: Dry	8.23E+18	173070.78 **
VIS dimmest	Vegetation: Wet	2.30E+18	48367.29
NIR brightest	Vegetation: Dry	1.70E+19	357497.36 **
NIR dimmest	Soil: Wet	5.10E+18	107249.21 **
Water-1 brightest	Vegetation: Dry	6.66E+18	140054.85 **
Water-1 dimmest	Soil: Wet	2.56E+18	53834.90
Water-2 brightest	Soil: Dry	2.89E+18	60774.55
Water-2 dimmest	Vegetation: Wet	6.94E+17	14594.30

Table 14 Payload Signal Saturation Analysis

- ** Indicates a value that exceeds the pixel full well capacity.

Four of the signals shown in table 14 saturate the optical system by exceeding the pixel full well capacity of 80,000 electrons. One method that could be employed to avoid pixel saturation is to reduce the exposure time. For example, the largest signal is NIR band Vegetation: Dry at 357,498 electrons. If the exposure time is reduced from 0.001 to 0.0002 seconds, the signal drops to 67,293 electrons. This is below the 80,000 e- pixel full well capacity.

4 PRELIMINARY SUBSYSTEM ASSESSMENT, ANALYSIS, DESIGN RESULTS

4.1 Mission Design Summary (Rosenberg)

The Mission Design Summary is a detailed analysis of four different studies: (1) a trade of three different constellation orbital heights, each with three different numbers of space vehicles, (2) a detailed analysis of the selected constellation architecture with respect to country accesses, (3) the calculation of time over ocean to support ADCS, Power, and CD&H, and (4) a detailed study of the required number and location of Ground Entry Points (GEPs) required to allow Alert and Relay (A&R) messages to be quickly downlinked, as well as support the downlinking of payload data and space vehicle state of health telemetry.

4.1.1 Orbit Altitude Selection Trade

The Orbit Altitude Selection Trade consisted of an analysis between different constellation architectures and orbital heights. These were then compared to one another by weighing NATO country primary area coverage during daytime, number of SVs, and height of orbit. Only orbits above the customer specified 500 km limit and below the maximum orbit height of 641 km imposed by the payload were considered. Higher coverage, lower numbers of SVs, and lower orbital height were preferred. Due to the number of simulations required, an analysis period of one week was used. The analysis period occurred during winter solstice to ensure that the coverage requirement of greater than 80% coverage is met during worst case. The results of this study can be seen in table 15. All constellations were analyzed with Noon/Midnight Sun-synchronous orbits.

Constellation Name	Altitude (km)	Number of SVs	Number of Areas with One Contact	Number of Areas with Zero Contacts	Coverage (>2 daylight passes per day)
8SV_500km	500	8	3	1	85.7%
9SV_500km	500	9	3	1	85.7%
10SV_500km	500	10	3	1	85.7%
8SV_600km	600	8	6	1	75.0%
9SV_600km	600	9	4	0	85.7%
10SV_600km	600	10	3	0	89.29%
8SV_641km	641	8	4	1	82.1%
9SV_641km	641	9	2	2	85.7%
10SV_641km	641	10	3	1	85.7%

Table 15 Orbit Altitude Trade

The outcome of the trade was the selection of 9SV_500km. This was selected as it met the customer specified coverage requirement of greater than 80%, used a minimal amount of space vehicles (see section 4.9 for more information), and is at an orbital altitude where two Minotaur IVs could be used to launch the entire constellation.

8SV_500km and 8SV_641km were also considered, but ultimately rejected. While 8SV_500km has the same coverage as 9SV_500km, the minimum number of SVs required to support crosslinks at a 500km altitude is nine due to atmospheric interference. 8SV_641km was ultimately rejected as more than two Minotaur launch vehicles would be required.

4.1.2 Detailed Country Access for 9SV_500km

Once a constellation architecture was selected, a higher fidelity analysis of 9SV_500km's NATO Primary Areas accesses was done to obtain a country by country breakdown. A time period of one month over winter solstice was selected to balance higher fidelity and available simulation computation resources. The results of this trade broken down by country can be seen in figure

17. More detailed information of country coverage and results by SV can be found in the Mission Design Summary spreadsheet, enclosure 3.

Access Durations	Min (seconds)	Mean (seconds)	Max (seconds)
NATO (All Areas)	0.604	46.93514286	372.707
Albania	3.908	28.173	47.957
Belgium	3.965	20.985	34.296
Bulgaria	5.246	32.013	47.997
Canada	2.68	190.313	358.336
Croatia	1.801	17.617	39.742
Czech Republic	5.284	28.631	39.384
Denmark	0.604	22.489	33.957
Estonia	2.883	24.332	37.296
France	3.561	67.94	129.381
Germany	2.612	66.337	124.268
Greece	2.114	25.547	76.681
Hungary	2.642	27.842	38.713
Iceland	2.649	28.708	51.522
Italy	4.245	37.395	94.761
Latvia	5.564	25.544	41.747
Lithuania	3.337	29.486	43.194
Luxembourg	2.74	11.014	14.006
Netherlands	3.352	19.192	38.347
Norway	3.491	32.385	76.529
Poland	2.781	67.24	93.201
Portugal	3.427	35.189	78.43
Romania	4.003	51.422	75.196
Slovakia	3.582	19.577	27.395
Slovenia	2.638	14.406	21.15
Spain	2.781	59.698	116.205
Turkey	3.155	67.708	99.632
United Kingdom	3.741	37.015	130.086
United States	0.733	225.986	372.707

Table 16 NATO Access Durations

4.1.3 Ocean Access for 9SV_500km

To support the development of the Electrical and Power Subsystem, an analysis of payload field of view over water was done. Per the customer specifications, the payload is to be powered off while over water. This period directly influences the sizing of SV solar arrays and batteries. The results are found below in table 17. More information, including durations by SV and overland statistics can be found in the Mission Design Summary spreadsheet in the data package.

Min Duration Over Ocean per Orbit	0.412 (seconds)
Average Duration Over Ocean per Orbit	685.896 (seconds)
Max Duration Over Ocean per Orbit	2997.091 (seconds)
Mission Total Ocean per Orbit Min percent	41.80%
Mission Total Ocean per Orbit Avg percent	60.73%
Mission Total Ocean per Orbit Max percent	86.53%

Table 17 Ocean Access

4.1.4 Ground Entry Point Access for 9SV_500km

A critical deliverable for the Mission Design Summary is the identification of Ground Entry Points (GEPs) to support downlinking of mission data, state of health information, and Alert and Relay (A&R) messages; as well as the uplinking of commands. To accomplish this, the minimum, mean, and maximum access durations were developed for each Swedish Space Corporation (SSC) Primary GEP. Data relating to GEP access times can be found in the Mission Design Summary spreadsheet. The use of SSC's Ground Station Network was specified by the customer. After an analysis of the constellation to ensure that the largest duration between GEP accesses for the constellation as a whole was about or less than 10 minutes per the customer, the following GEPs were selected:

- Clewiston, Florida, United States
- Esrange, Sweden
- Inuvik, Northwest Territory, Canada
- Punta Arenas, Magallanes and Antartica Chilena, Chile

The access durations tabulated by GEP can be found in table 18 below.

	Min GEP Accessed (seconds)	Mean GEP Accessed (seconds)	Max GEP Accessed (seconds)	GEP Total Accessed per Orbit (seconds)	GEP Total Accessed per Day (seconds)
Clewiston	17.16	348.25	441.80	560.61	8532.18
Esrange	2.85	332.12	450.88	1604.68	24422.30
Inuvik	15.73	309.19	451.29	1728.18	26301.84
Punta Arenas	8.80	349.52	447.96	870.39	13246.80

Table 18 GEP Accesses

As shown above, in table 18, the shortest GEP access is 2.85 seconds for the Esrange GEP. This is shorter than the required 1.67 seconds to downlink an A&R message. The minimum, mean, and max durations between GEP accesses can be found in table 19.

Min Gap Duration	**0.054 (s)**
Average Gap Duration	390.63 (s)
Max Gap Duration	645.11 (s)

Table 19 GAP Duration Values

While performing downlink performance calculations it was identified that the mission cannot transmit all generated payload data due to the use of a low bandwidth S-Band antenna. Pertinent information is listed in table 20 below.

PLD Data Rate (kbit/s)	**Time Over Land per Day (s)**	**Mission Total GEP Accessed Per Day (s)**
15000	218979.1087	72503.1403
TTC Data Rate (kbit/s)	**Data Collected Over Land per Day (kbit)**	**Data Downlinked per Day (kbit)**
5000	3284686631	362515701.5
	Percent Data Downlinked per Day w/ 2x Compression	**11.04%**

Table 20 Mission Data Information

While the TTC radio can transmit a maximum of 5000 kbit/s during GEP access, the payload generates 15000 kbit/s while over land. Since the payload is over land for a period three times greater than the TTC radio is communicating with a GEP, only 11.04% of payload data can be downlinked. Therefore, the system as currently designed cannot transmit all generated payload data. If all available SSC GEPs are used, 20% of payload data can be downloaded per day. As such, four solutions have been developed.

4.1.4.1 Software Solution – Compression Ratio

It may be possible to increase the payload data compression ratio of the CD&H subsystem. This would allow larger amounts of data to be transmitted over a given time. This has been evaluated as a medium cost, medium risk solution due to the required development of a twenty to one compression ratio. The ratio is twenty to one due to the current system only being able to downlink 5.52% of payload data without compression.

4.1.4.2 Software Solution – Filter Data

The use of a filter that only stores payload data of fire-prone areas could be implemented. This system is deemed low cost, low risk due to the already planned development of an onboard algorithm to identify fire-prone areas. However, it must be noted that a large amount of payload data will be discarded with this solution.

4.1.4.3 Mission Operations Solution – Command Downlinking

The Mission Operations team could command the downlink of a specified time of interest. This would require an increase of the onboard data recorder size. This solution has been deemed low cost, low risk as the design of the mission's SVs would not change except for the increase of data storage. However, all data not in the specified period would be lost.

4.1.4.4 RF Solution – Mission Data Link Radio

The implementation of a second, high bandwidth radio could be implemented into the SV design. This radio would only be used to downlink payload data while over a GEP and would serve no other purpose. This solution has been deemed low risk, high cost due to the flight heritage of this architecture and availability of commercial off the shelf (COTS) high bandwidth small sat antennas. This is the preferred solution.

4.2 C&DH (Teste)

4.2.1 Safe Mode Overview and Descriptions

Safe Mode immediately reduces all operations to the minimum. ADCS will keep the spacecraft antennae-oriented nadir (via the support of the avionics). Communications will wait for and implement instructions from the ground station. Payload operations halt and the system is turned off. Solar panels will be set to a specific position to optimize power received while minimizing spacecraft actions necessary (TBD). At a reduced state, demands on the thermal subsystem will also be reduced, further decreasing spacecraft activity. Lastly, TT&C will be reduced to detailed housekeeping data as well as payload information (upon ground request). Together, the objective is to reduce spacecraft demands to a level that crippled subsystem can manage. The spacecraft only exits safe mode after receiving ground input. Safe mode descriptions:

Prelaunch – Spacecraft 1st turns on on the launch pad, performs final checks of the spacecraft.

Launch – Spacecraft mode during transport to orbit.

Deployment – At desired orbit, spacecraft separates from launch vehicle. Unfolds from its compacted shape.

Nominal Operations – During orbit, spacecraft performs the task(s) it was designed to do. This includes the ADCS, avionics, communications, payload, power, thermal, and TT&C subsytems.

Maneuver – This mode is used to rectify or adjust the spacecraft's orbit and the attitude.

Standby – Mode that places all non-essential systems on hold until further instructions are received. Minimal autonomy -such as keeping communication relays oriented towards earth (nadir)- functional.

Safe Mode – When the spacecraft has determined a critical problem and is unable to resolve it itself (or has received a command to do so) or spacecraft boots up for the first time, this mode minimizes spacecraft operations until further notice. Mode allows ground control to directly edit, change, monitor, and control all spacecraft systems.

End of Life – The final mode the spacecraft enters before self-destruction. Separate from maneuvers to prevent accidents.

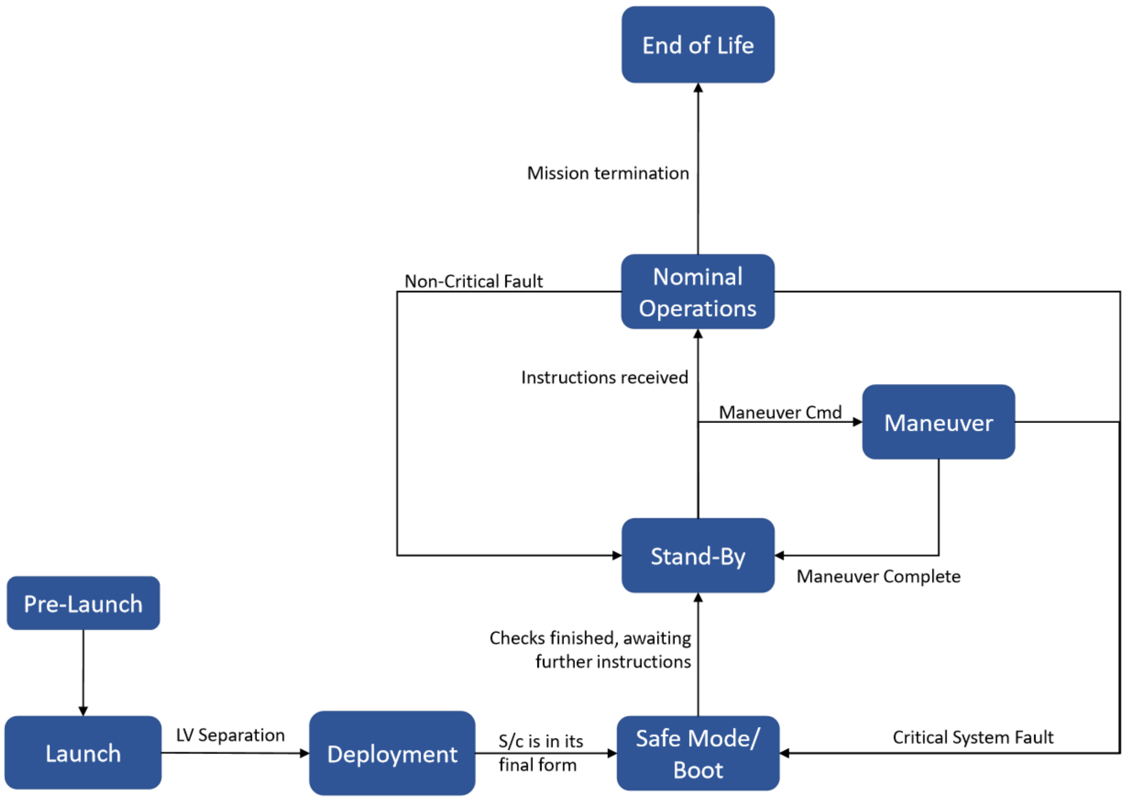

Figure 3 Safe Mode Block Diagram

4.2.2 Potential faults and solutions for safe mode

<u>Insufficient Power</u> – Determine precise location of error, then compare/contrast data to expected values. Reset sensors and then s/c. Decrease power loads within s/c, orient solar panels to pre-determine alignment, and await further input from ground.

<u>Critical Payload Error</u> – Collect detailed information from payload error and transmit information to ground. Turn payload off until further notice (from ground).

<u>Loss of attitude control</u> – Determine whether sensors in disagreement (see invalid ephemeris) or if spacecraft experiencing loss of control. If loss of control, decrease power loads and record s/c attitudes. Continually transmit to ground until further instructions.

<u>Critical propulsion error</u> – Immediately halt propulsion maneuver (if in one), isolate root cause, and register data from propulsion leading to this fault. Reset sensors. Await further instructions from ground.

<u>Command-induced safe mode</u> – Await further instructions from ground.

Critical thermal error – Determine location of error (subsystem-wise or unexpected value). Decrease power loading which reduces thermal loads, and orient the s/c in a pre-determined alignment. Reset sensors. Collect data leading up to failure and transmit to ground.

Invalid orbital maneuver – Immediately halt any propulsion maneuvers. Reset sensors. If sensors in agreement and the remaining de-orbit margin is sufficient, request orbital boost-up from ground. Await ground commands.

Critical memory error – Register memory corruption and reboot processor. See if problem persists. Boot to previous saved (and uncorrupted) state if it does and transmit to ground.

Invalid ephemeris – Isolate axis (axes) of uncertainty. Locate sun, earth, and another designated point of interest. Reset sensors. If problem persists, transmit to ground and await further instructions.

Timekeeping error – Attempt to correct error via a previous state and/or crosslinking with another satellite. Transmit error to ground and await confirmation or commands.

Command execution failure – Isolate the failure point and register commands/data leading to this fault. Reset the processor and/or the s/c. If problem persists, boot to previous saved (and uncorrupted) state. Transmit to ground and await further instructions.

4.2.3 Space Vehicle Block Diagram (Rosa)

Preliminary spacecraft block diagram is depicted in figure 22 below. A larger image is included in enclosure 4.

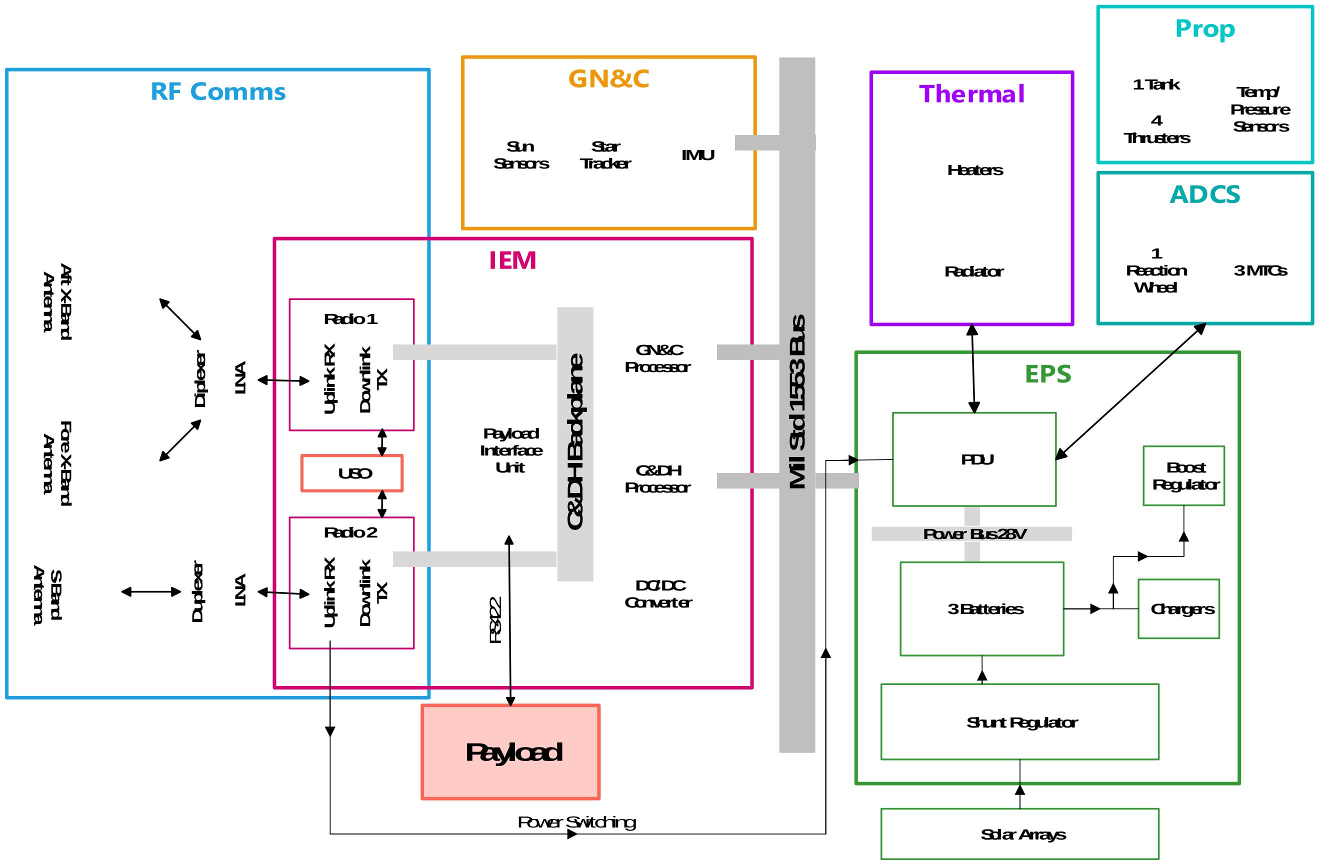

Figure 4 Spacecraft Block Diagram

4.3 Power (Rosa)

4.3.1 Power Regulation Trade

Direct Energy Transfer (DET) and Peak Power Tracking (PPT) were considered in the power regulation architecture. Both types of power regulation have pros and cons. DET transfers power directly from the solar arrays to the power bus which means the solar arrays must operate at the same voltage as the power bus. With PPT, the solar array and bus can operate at different voltages.

Several parameters were analyzed in this trade to include efficiency at EOL, mass, complexity, power loss, and cost. A value of 1 indicates good (meets requirement), 0 indicates ok, (marginally meets requirement) and -1 indicates bad (does not meet requirement). The values were multiplied by the weight to get the score for each criterion. Those are then added to generate the total score for each option as outlined in table 21.

	Efficiency	Complexity	Power Loss	Mass	Cost	Score
Weight	5	4	3	2	1	
DET	Good	Good	Bad	OK	Good	7
	5	4	-3	0	1	
PPT	OK	OK	Good	OK	OK	3
	0	0	3	0	0	

Table 21 DET vs PPT Trade Analysis

DET scores higher than PPT based off the assigned weights in the preliminary informal trade. A more detailed analysis will be performed after initial feedback from the customer on assigned weights. Another factor to consider is required regulation of spacecraft loads. The payload is the largest draw on the power system (60W max power) and is required to constantly take images of NATO countries overland. While the payload is allowed to remain in the off state over water, if it previously identified vegetation at risk of fire but is not in contact with a Ground Entry Point (GEP), then the payload will send an alert message every 60 minutes until the spacecraft receives confirmation that the message is received. This requires tighter load regulation and DET can offer that capability. Based off this informal trade, the recommendation is to use DET for the power system. Specifically, a DET system with a fully regulated architecture as outlined in figure 5.

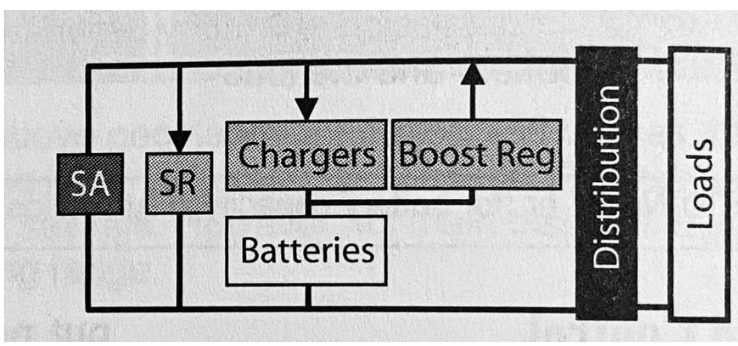

Figure 5 DET with Fully Regulated Load

4.3.2 EPS Component Sizing

Initial sizing has been completed for the solar arrays and batteries using a DET power regulation architecture. For the solar arrays, input parameters include an OAP of 74.3W, eclipse time (Te) of 35.4 min, sunlit time of 59.2 min, an altitude of 500km, and a 3-year spacecraft lifetime. Total solar array power:

$$Psa = \frac{\left(\frac{74.3 * 35.4}{.65}\right) + \left(\frac{74.3 * 59.2}{.85}\right)}{59.2} = 155.8\,W$$

The power output of triple junction GaAs solar panels is 253 W/m^2. Using this value, a worst-case solar angle of 23.5 degrees, and an inherent degradation (Id) of .77, the Power at Beginning of Life (BOL) per unit area is calculated as 178.7 W/m^2. Power at End of Life (EOL) is calculated as 153.68 W/m^2. This leads to the final calculation of the total array size required to produce the necessary power.

$$Asa = \frac{155.8}{153.68} = 1.01\ m^2$$

For comparison, silicon cells result in a required solar array area of 1.22m^2. Further trade analysis will need to be conducted to determine whether silicon or triple junction GaAs is the better choice for the solar array.

For the overall battery capacity calculation, OAP, eclipse time (.59 hours), worst case DOD (20%), battery quantity, and efficiency (90%) were factored in. For a 28V DC bus capacity, a total quantity of 3 batteries was chosen for redundancy. This battery capacity below leads to a battery mass of around .48kg. Battery capacity can be calculated as:

$$C = \frac{74.3 * .59}{.2 * 3 * .9} = 81.2\ W\text{-}hr$$

4.4 TT&C (Rosa)

4.4.1 Crosslink Antenna Cant Angle

A 9 satellite constellation at 500 km altitude results in a 4705km crosslink range and a crosslink antenna cant angle of 70 degrees as depicted in figure 6. Future analysis will be conducted to determine the trade-off between a fixed and tracking antenna.

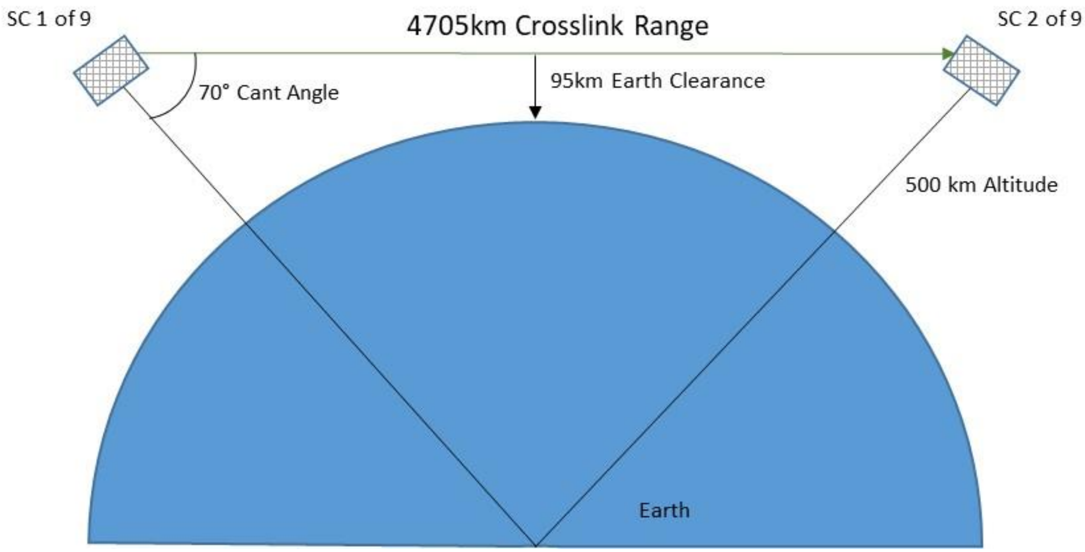

Figure 6 Crosslink Cant Angle (not to scale)

4.5 Thermal (Teste)

The thermal control system (TCS) is designed to continually keep spacecraft components within their respective functional temperature ranges. Heat-sensitive components are to be isolated in various thermal zones; these being separated by multi-layered insulation (MLI). For components that generate significant heat or are subject to intense radiative heating from the environment, a radiator shall be used to ensure the temperature does not surpass a specific threshold. On the other hand, certain components may require a minimum temperature for nominal operation when the spacecraft traverses the umbra. This needs to be regulated through heaters with thermostats to ensure the components remain in their optimal temperature ranges.

Overall, the mass budget allocates 2.64 kg of mass for the TCS and the power budget allocates 4.3W of power. Heaters and thermostats require little mass but are a significant source of power consumption.

Radiators require significant mass but less power. Thus, it is reasonable to presume that a single radiator shall be placed atop the components generating the most excess heat while the heater and thermostat are utilized to maintain critical spacecraft subsystems such as payload, propellant, or solar panels.

To be investigated in-depth at a later stage in the spacecraft's development.

4.5.1 Thermo-optical trade study

For best radiator efficiency, a trade study to minimize radiator size was conducted by comparing two finalist thermo-optical materials: white paint (A-276) and 5-mil silver Teflon. Furthermore, several different radiator control temperatures and electrical heat dissipation scenarios were considered. The spacecraft is assumed to be well insulated from radiative heat transfer and thus the single point of exchange is at the radiator. A 10% margin was applied to all environmental loads. Full details are outlined in enclosure 17.

Nadir Pointing S/C		Flux (W/m^2)
	Orb Avg Solar	36.5
	Orb Avg Albedo	109.4
	Orb Avg Planet	200.6

	BOL	
	A-276 White Paint	5 mil Silver teflon
Absorptivity	0.26	0.08
Emissivity	0.83	0.77

Table 22 Thermal Paint Trade

Note: Only BOL effects are considered in this trade study.

Electrical Heat Dissipation	Radiator Control Temp.	Thermo-Optical Materials: A-276	Silver Teflon	Units
25W	0degC	0.6746	0.4149	m^2
	-15degC	-1.5713	2.2499	m^2
	-30degC	-0.4139	-0.8287	m^2
1W	0degC	0.0270	0.0166	m^2
	-15degC	-0.0629	0.0900	m^2
	-30degC	-0.0166	-0.0331	m^2
0W*	0degC	0	0	m^2
	-15degC	0	0	m^2
	-30degC	0	0	m^2

Table 23 Radiator Area per Case

*In the 0Watt case, there is no electrical heat dissipation. However, the 0W case can be treated as prior or it can be interpreted as if the radiator does not have an area (which can be seen from the results above). Thus, if there is no radiative area, the spacecraft is assumed to be well insulated everywhere else, and there is no electrical heat dissipating, then there is no heat transfer occurring in that model.

Examining the results above, it becomes evident that in the cases that the resulting radiator area is negative are unfeasible and any areas greater than 3 m^2 are equally so. Overall, it is recommended to apply silver Teflon as the radiator's thermo-optical material with a minimum control temperature of -15 degrees Celsius.

4.6 Attitude Determination and Control Subsystem (Rosenberg)

The Attitude Determination and Control Subsystem (ADCS) was designed in response to the maximum size, maximum mass, and expected operational modes of the mission. The ADCS subsystem has been developed with two studies:

- Selection of Attitude Control Method
- System Momentum Management

4.6.1 Method of Attitude Control Trade Study

To determine the mission's method of attitude control, a trade was conducted between four different control architectures:

- Spin Stabilization
- Gravity Gradient
- 3-Axis without Propulsion
- 3-Axis with Propulsion

The results of this study can be found below in table 22.

Method of control	Pointing Accuracy	Comments	Baseline
Spin Stabilization	1-5 Degrees	Payload FOV shape not compatible. SV attitude changes constantly	No
Gravity Gradient	>5 Degrees	Does not meet pointing requirement or slew requirements	No
3-Axis without Propulsion	<0.1 Degrees	Good pointing. Unable to meet slew requirements	No
3-Axis with Propulsion	<0.1 Degrees	Good pointing. Able to meet slew requirements	Yes

Table 24 ADCS Trade Analysis

4.6.1.1 Spin Stabilization

The first method of attitude control investigated was spin stabilization. This requires spinning the entire spacecraft to impart active stability in axis direction and passive stability in the two other axes. This method of attitude control was not selected as the spacecraft would have to be repeatedly spun and despun whenever a payload calibration slew is performed. Another issue with spin stabilization is the rectangular shape of the payload field of view. While a rotor-stator system could be implemented to keep the payload static, this would add complexity and cost to the mission.

4.6.1.2 Gravity Gradient

Gravity Gradient attitude control requires the mass distributions of the mission space vehicles be formatted in a way that causes Earth's gravitational field to orient the spacecraft in the proper direction. This form of control was not selected as it does not provide the pointing control required by the mission and does not allow for payload calibration slew maneuverability.

4.6.1.3 3-Axis

3-Axis control requires the use of torque actuators to add and subtract from the system's angular momentum. Reaction wheels (RWAs), thrusters, Control Moment Gyros (CMGs), and Magnetic Torque Coils (MTCs) are examples of 3-Axis control actuators. These inputs result in a change in the system attitude. This method was selected as it provides the required pointing accuracy and can perform payload calibration slews.

Once 3-Axis was selected as the method of control, a study was performed to determine which 3-Axis control architecture suited the mission best. The results of the trade can be seen below in table 23. A trade that determined MTCs are more applicable de-torquers than thrusters can be found in Section 4.6.2.

Actuator Name	Mass (kg)	Power (W)	Torque Min (N*m)	Momentum Storage (N*m*s)	Accuracy Min (deg)	Comments	Baseline?
Zero Momentum RWA with 3x MTCs	6	30	0.03	1.2	<0.1	Most common. Greatest pointing and agility.	No
Momentum Bias Wheel with 3x MTCs	2	10	0.01	0.4	2.0 - 0.1	Only needs one RWA	Yes

Table 25 3-Axis Trade Analysis

4.6.1.3.1 *Zero Momentum RWA with 3x MTCs*

A Zero Momentum RWA 3-Axis control architecture consists of a minimum of three RWAs that absorb system momentum. The result of this is that the overall system momentum is constantly at zero. MTCs would then be used to remove momentum from the system. This method of attitude control is extremely precise, but was not selected due to its high mass, power usage, and cost.

4.6.1.3.2 *Momentum Bias Wheel with 3x MTCs*

A Momentum Bias Wheel 3-Axis control architecture utilizes a single RWA configured to actively control the Z axis of the spacecraft. The spacecraft structure is then configured so the RWA provides passive stability in the X and Y axes. The MTCs that are already being used to remove momentum from the system can also be used to control the X and Y axes. This solution was chosen as it has a low subsystem size, mass, power, and cost while providing the requisite pointing accuracy.

4.6.2 System Momentum Trade Study

An important action of the ADCS subsystem is the desaturation of reaction wheel momentum. As the RWAs operate, they absorb momentum that would otherwise be imparted by the space vehicle. To remove this accumulated momentum from the system, actuators that impart a torque on the external environment are used. Using the MATLAB script provided in the data package, a required system momentum storage (with 50% margin) was calculated to be 0.297 N*m*s. This study analyzed the use of thrusters and Magnetic Torque Coils for the FROST mission. The results of this trade are summarized in figure 28.

	Thrusters	MTCs
Pros	• Provide very short de-torquing timelines. • Use low levels of power when compared to MTCs. • Already used by mission to change orbit	• Does not use propellant • Can be used whenever payload is pointed at ocean • High reliability, no moving parts
Cons	• Uses propellant • Requires more thrusters	• Long De-Torquing timelines • Low power, but over a long period of time

Table 26 System Momentum Trade

4.6.2.1 System Momentum Control with Thrusters

To develop a model upon which a thruster-based detorque action could be based, some assumptions were made. (1) The space vehicle shall have four 1 N thrusters placed on the –Z panel, with one at each corner. (2) One momentum dump shall be performed per day. (3) Each burn firing of the propulsion system will require firing all four thrusters. Thrusters were ultimately not selected, as the size, mass, and cost of a larger propellant tank to accommodate system momentum control. The results of this trade can be found in table 27.

Moment Arm (m) (TBR)	0.3
Thrust (N)	4.0
Torque (N*m)	1.2
Required Power (W)	18.0
Time to De-Torque (s)	0.248
Propellant Used per De-Torque (kg)	0.00194
Propellant Used per Year (kg)	0.71
Power Used per De-Torque (W)	4.46

Table 27 Control with Thrusters Trade

4.6.2.2 System Momentum Control with Magnetic Torque Coils

MTCs are a reliable and low size, weight, and power solution to momentum control. Magnetic Torque Coils act by changing the magnetic dipole of the space vehicle system. This dipole interacts with the Earth's magnetic field and in doing so, imparts an exchange of momentum out of the SV system and into the Earth's. MTCs were selected to perform FROST's system momentum control as they do not use propellant, have no moving parts, and impart a low load onto the spacecraft power system. A preliminary trade was conducted to determine a potential torque coil sizing for a suite of three MTCs. The result of this trade was the selection of 1 A*m design described below in table 28.

Number of MTCs	3
MTC Dipole	1
Momentum to Manage per Day	0.297
B_Earth @ 500 km (T)	4.89E-05
MTC Torque (x3)	1.47E-04
Time to De-Torque (sec)	2023.4
Power (W) (x3)	1.8
Required Power per De-Torque(W)	3642.086331
Watt/Hours	1.011690647
Amp/hours (assuming 32V Bus)	0.031615333

Table 28 Control with Magnetic Torque Coils Trade

4.7 Configuration (Brown)

4.7.1 Final configuration models of stowed SV(s) inside LV(s) for <u>each option</u> considered

The Minotaur I and Minotaur IV launch vehicle configurations of the stowed SVs inside the LV is as follows:

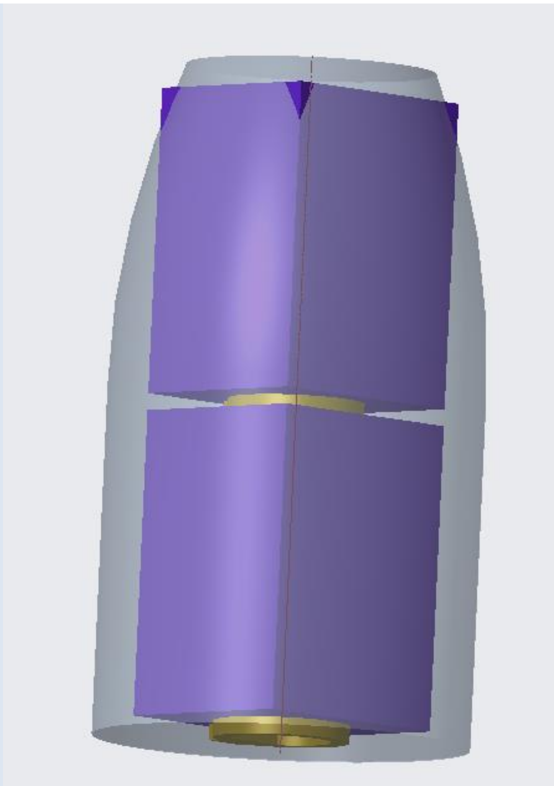

Figure 7. Stowed SVs in Minotaur I.

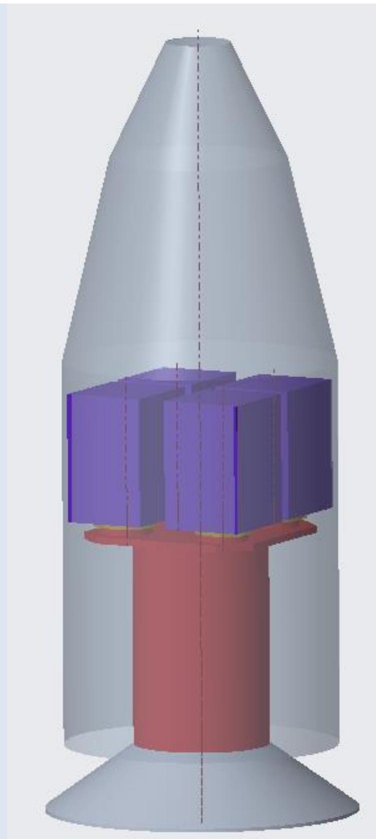

Figure 8. Stowed SVs in Minotaur IV

The Minotaur I can accommodate two SVs, while the Minotaur IV can accommodate four SVs. The Minotaur I configuration has two ESPA class SVs and two Lightband separation systems per LV fairing volume. The Minotaur IV configuration has four ESPA class SVs, four Lightband separation systems, and Multi-Payload Adapter Plate per LV fairing volume.

4.8 Structures (Brown)

4.8.1 Structural requirements

Per Minotaur IV User's Guide, the axial and lateral frequency requirements are:

$$f_{axial} > 35\ Hz$$

$$f_{lateral} > 15\ Hz$$

The spacecraft lateral frequency is:

$$f_{lateral}(Hz) = \frac{1}{2\pi}\sqrt{\frac{k_\theta}{I_m}} = \frac{1}{2\pi}\sqrt{\frac{1.05 * 10^7 \frac{m-N}{rad} * \frac{1\ kg - m/s^2}{1\ N}}{55.08\ kg - m^2}} = 69.3\ Hz$$

Per the Minotaur IV User's Guide, the spacecraft lateral frequency shall be greater than 15 Hz
$f_{lateral} = 69.3\ Hz > 15\ Hz$

For a total space vehicle mass of 180kg*4 SV/LV=720 kg/LV, the approximate lateral acceleration is 6.0 Gs and the approximate axial acceleration is 10.5 Gs.

$f_{payload\ lateral} = 2 * f_{SV\ lateral} = 2 * 69.3\ Hz = 138.6\ Hz$

For a 40kg max payload mass, the corresponding lateral and axial load factor is:

$G = G_{lateral} = G_{axial} = 66.4 * m^{-0.38} = 66.4 * 40^{-0.38} = 16.4\ Gs$

4.9 Launch (Brown)

4.9.1 SV/LV trade study

The team project description limited the type of launch vehicles to two options: Minotaur I and Minotaur IV, both provided by Orbital Sciences Corporation. The important criteria used to evaluate these two candidates include the following:

- Payload capability vs altitude performance
- Launch site support
- Secondary payload adapters
- Reliability (number of launches)

- Cost per launch vehicle

These criteria were provided a weighting factor ranging from 1 to 4 and the two candidate LVs were given a score of 0, 1, or 2 against each criteria. The results of this informal trade study are as follows:

		Candidates			
Criteria	Weight	Minotaur I		Minotaur IV	
Payload capability	4	1	1x4=4	2	2x4=8
Launch site support	2	1	1x2=2	1	1x2=2
Payload adapters	1	2	2x1=2	2	2x1=2
Reliability	3	2	2x3=6	1	1x3=3
Total score		14		15	
Cost per launch		$25M		$55M	
Number of launches		5		2	
Total cost		$125M		$110M	
Total score/Total cost		0.112		0.136	

Table 29 Mass Budget

The Minotaur I and Minotaur IV scores are very close, with the Minotaur I performing better in reliability (longer launch history) and the Minotaur IV performing better in payload capability (more payload mass to 500 km orbit). The factor that made the Minotaur IV the clear preference was the number of launches, cost per launch, and total cost. The Minotaur I option requires five launches to deliver all nine SVs to 500km, while the Minotaur IV option requires only two launches to the same altitude. Comparing the total score/total cost reveals that the Minotaur IV launch vehicle is the clear choice.

For more information, see Trade Study #10 (TS10): Launch Vehicle Selection

4.9.2 Summary of the Preliminary LV Payload Questionnaire

Full Name: Fire Risk Observation SysTem (FROST)

Space Craft and Mission Description: Constellation of ESPA class small satellites that provide real-time alerting for at-risk fire areas.

Altitude: 500 km

Inclination: 98.7 degrees

Nominal Launch Date: 1 January 2022

Launch Site: Vandenberg Air Force Base (VAFB), California

5 CONCURRENT ENGINEERING PARAMETER DATABASE

All parameters associated with this spacecraft, subsystems, and the mission, can be found in the CEPD enclosure 5.