100-HOUR EXAM
AEROSP-582

DOPE -
Distribution of Precipitation Element

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1. Introduction

1.1 Objective

The paper presents a preliminary design draft for a satellite to measure the global distribution of precipitation and its diurnal variability. It discusses the design considerations in terms of top-level requirements, derived functional requirements, identification of system drivers, sub-system design, mission concept, concept of operations, timeline, architecture, technical risk and many more details. This study will be delivered to the "The Red Team" at the close of business on Wednesday, November 13th, 2019.

1.2 Requirements

The top-level requirements were proposed by the stakeholder and are given below verbatim.
The satellite shall:
1) Provide global coverage.
2) Provide spatial resolution of at least 20 km at nadir (lower off nadir).
3) Provide temporal resolution (revisit time) of at least 6 hours to investigate the diurnal variability of precipitation.
4) Fit into a small satellite bus: The spacecraft (including all subsystems) shall not exceed a mass of 50 kg, and a volume of 0.5m x 0.5m x 0.6 m.
5) Have a design lifetime of at least 3 years.

1.2.1 Payload Description

The payload onboard, has a 4 channel passive microwave radiometer capable of making measurements of thermal radiation at 10, 19, 22, and 37 GHz. The payload and antenna specifications are:
- Total Power: 25 W
- Total Mass: 15 Kg
- Electronics Box: 15 cm x 15 cm x 25 cm
- Antenna Panel: 10 cm x 50cmx 3 cm
- Electronics: Heterodyne total power receivers; MIC integrated RF electronics plus digital back-end on PCBS for multiple channels per each imaging element/beam/footprint
- Channels: At 10, 19, 22, and 37 GHz respectively; each channel measures at 40 locations/beams/footprints cross-track
- Data: For each beam/footprint, the radiometer makes measurements at16 bits per channel

1.2.2 Antenna Details

- Type: Rectangular phased array mounted on thin, light-weighted panel
- Construction: 1 mm-thick copper layer on the tip of 2.2 cm honeycomb polystyrene with a 5 mm-thick PCB board between 1mm-thick layers of carbon fiber
- Mounting to S/C: nadir pointing with long dimension oriented cross-track
- Number of beams/footprints: The antenna creates 40 consecutive beams/footprints with no gaps between lined up in a row cross-track in a "push-broom" configuration
- Beam-width per beam/footprint: 2° x 2° (cross-track x along-track per beam)
- Required pointing accuracy: 0.1°

Given these specifications and requirements, a preliminary design is presented in this paper. It is to be noted that these design considerations a part of the first iteration cycle in the design life of the satellite and not to be considered finalized design decisions.

2. Orbit Constellation and Design

The design of the spacecraft is carried forward by considering the scientific objectives of the mission at the highest priority. This involves a careful consideration of the various orbit altitudes, inclinations and constellation configurations, which critically affect the result of meeting these requirements and therefore the success of this mission. Upon careful examination with the assistance of simulation tools like STK, the orbital analysis is hereby presented.

2.1 Data Collection Assessment

a) From the requirements of the stakeholder for a minimum resolution of 20km at nadir pointing and the beam-width of the push-broom imager elements/beams, the calculation given below help us gain insight in the minimum altitude which will make this possible.

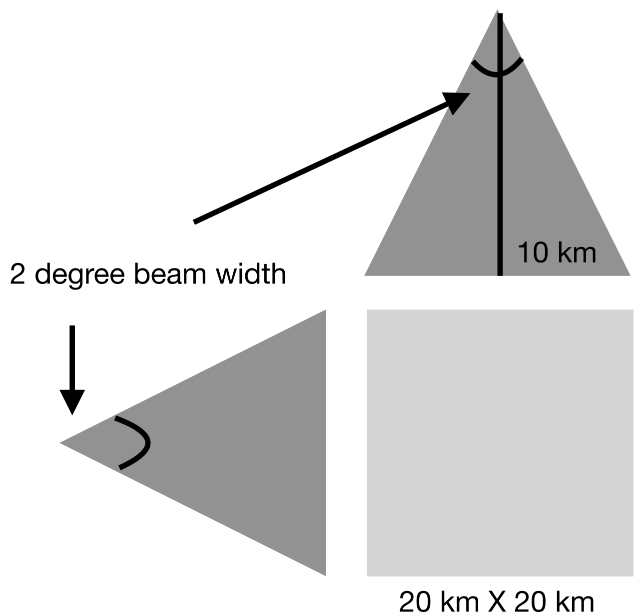


Figure 1: Calculation of the altitude requirements for the resolution of 20 km at 2 degree beam-width.

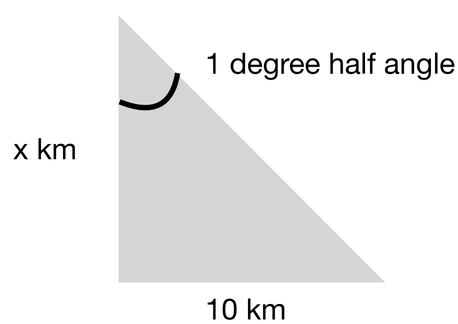


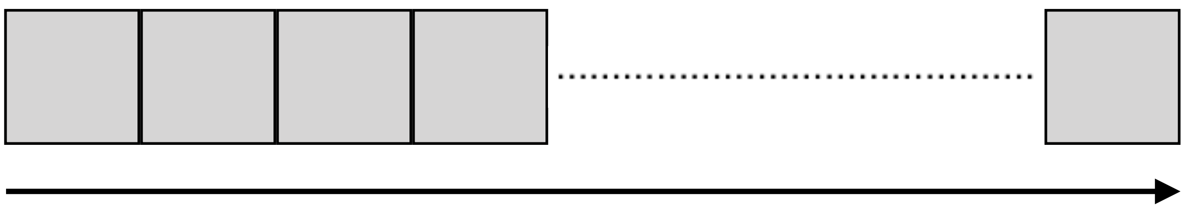
Figure 2: Half triangle for altitude determination through trigonometric calculations.

Considering the right-triangle to compute the altitude from Figure 1, Figure 2, shows the half-triangle which yielded the required altitude.

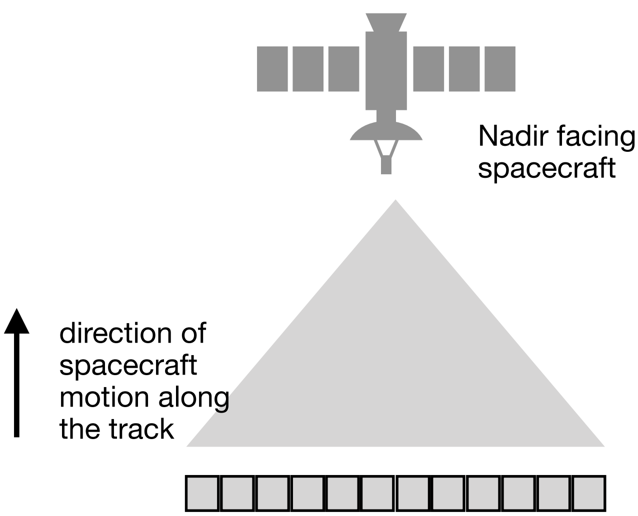
$$tan(\Theta) = \frac{10}{x} \quad \rightarrow \quad x = \frac{10}{tan(\Theta)}$$

$$x = \frac{10}{tan(1)} = 572.9km$$

From this altitude, the swath width can be calculated by using the resolution of 2 degree beam-width for each element in the swath. As per the antenna specification, 40 elements/beams of 2° X 2° degrees are aligned in the horizontal path perpendicular to the direction of the velocity/ ground track of the satellite, as shown in figure 3.



# of elements in swath = 1, 2, 3, 4…….. 40.



full swath width* (not to scale)

Figure 3.1: cross-track swath

Figure 3.2: full swath width with 40 elements per swath and 2° X 2° degree beam angle per element.

As seen in Figure 3.1 and 3.2, the total beam angle the payload projects is 80° angle cross-track and 2° angle along-track. With these projection angles and altitude, we can estimate the size of of the swath projection on the ground. We use the half angle width of 40° cross-track to find the cross-track length. Ther

$$tan(40) = \frac{x}{572.9} \quad \rightarrow \quad x = tan(40) \times 572.9$$

$$x = 480.1km$$

This gives a swath size of about 961.44 km cross-track and 20 km along-track projection on the ground as shown in figure 4. A key assumption here is that the middle of the phased array is pointing at the sub-satellite point and the swath projection distance assumes the earth to be flat since the distance is negligible compared to the curvature of the surface.

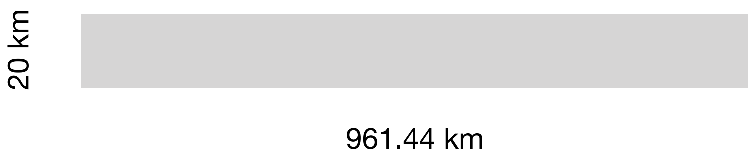


Figure 4: Swath size projection on the ground.

2.2 Satellite Coverage

2.2.1 Orbital Considerations

After obtaining the altitude from the resolution requirements, it was also noted that the spatial requirements and the global coverage were in fact functions of the orbital inclination and have to be strictly met for the mission success. First a study of the impact of various inclinations was conducted to see a comparison of the net coverage with the swath size from above. STK was used to simulate the coverage. Three different inclinations of 0°, 45° and 90° were simulated and starting on September 8th, 17:00 UTC and the results are tabulated below.

inclination = 90°	Ended	% Coverage	RAAN (true anomaly)
3 sats	13 Sept 9:22	100	90, 45, 0
2 sats	13 Sept 9:22	100	90, 0
1 sat	17 Sept 3:12	100	0
inclination = 45°			
2 sats	10 Sept 10:40	77.78	90, 0
1 sat	11 Sept 5:04	77.78	0
inclination = 0°			
1 sat	8 Sept 18:41	5.25	0

Table 1: Comparison of the effect of inclination on global coverage.

As it can be seen that the polar orbits (i = 90^0) are favorable for global coverage as they yield the fastest coverage rates and also has higher number of passes in the a given day which decreases the revisit time as well. Here RAAN abbreviates to right ascension of ascending node. It should be noted that only circular orbits with evenly spaced true anomalies were considered in the analysis to have a homogenous system of nodes and keep the mission simplistic.

b) Now finalizing that high inclinations has better coverage, a single satellites coverage was analyzed in a given day for which the results are shown below.

```
                                                    9 Nov 2019 12:48:07
FOR UNFUNDED EDUCATIONAL USE ONLY
CoverageDefinition-Seed_Coverage


                        Coverage Properties
-------------------------------------------------------------------
Global Coverage
Grid Altitude: 0.0000 (km)
Ground Altitude set from grid altitude reference (On ellipsoid)
Resolution: 6.0000 (deg)
Number of Points: 1148
Assets required for a valid access: At Least 1
Assigned Assets:
    Satellite/Seed_i_90R_0SEED/Sensor/Sensor11:    Active
Access Interval: 8 Nov 2019 17:00:00.000 to 9 Nov 2019 17:00:00.000
Regional Acceleration: Automatic
Light time delay: Ignored
Maximum Sampling Time Step: 360.000 secs
Minimum Sampling Time Step: 0.010 secs
Time Convergence: 5.000e-03 secs
Value Convergence: Relative 1.000e-08 - Absolute 1.000e-14
```

```
                   9 Nov 2019 16:53:00.000         0.00               63.72
                   9 Nov 2019 16:54:00.000         0.00               63.72
                   9 Nov 2019 16:55:00.000         0.00               63.72
                   9 Nov 2019 16:56:00.000         0.00               63.72
                   9 Nov 2019 16:57:00.000         0.00               63.72
                   9 Nov 2019 16:58:00.000         0.00               63.80
                   9 Nov 2019 16:59:00.000         0.00               63.80
                   9 Nov 2019 17:00:00.000         0.00               63.89

Global Statistics
-----------------
Min % Coverage     8 Nov 2019 17:01:30.000         0.00                0.09
Max % Coverage     9 Nov 2019 06:51:00.000         0.19               51.95
Mean % Coverage                                    0.00
```

Figure 5: The result from STK for coverage of a single satellite at 90° inclination. Note: 1st column shows the time, 2nd column shows the individual percentage coverage and the 3rd column shows the cumulative coverage.

As seen from Figure 5, a single satellite at an inclination of 90° covers about 63.89% of the land mass in a given day. The sensor is defined with a rectangular projection with a vertical half angle of 1° and a horizontal half angle of 40^0 in STK with an altitude of 573 km.

2.3 Constellation Design

2.3.1 Orbital Configurations

From Table 1, it is seen that the 3 satellites at 90° inclinations took about 5 days to cover the entire surface of the earth, which points that there needs to be more number of satellites with quicker coverage in the constellation which are equally phased via the right ascension of the ascending node.

c) We compare two configurations, to conduct some analysis in STK. Looking at heritage, 8 polar satellites were used for GPM (Global Precipitation Measurement)[1] to obtain global coverage with a 3-hour of sampling time. So if we scale our requirements of 6-hour sampling time, and lower altitude compared to GPM, a constellation of 6 satellites is a good approximation.

Considering that, two simple and robust Walker Star constellations were obtained using that,
Constellation 1: Two satellites per orbit for three polar orbits of RAAN separation of 120 deg and a true anomaly of 60 deg.
Constellation 2: Three satellites per orbit for two polar orbits of RAAN separation of 90 deg and a true anomaly of 30 deg. Figure 6 & 7 below show the pictorial representation of both the constellations in a geocentric frame in STK.

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[1] https://link.springer.com/content/pdf/10.1007%2F978-1-4020-5835-6.pdf

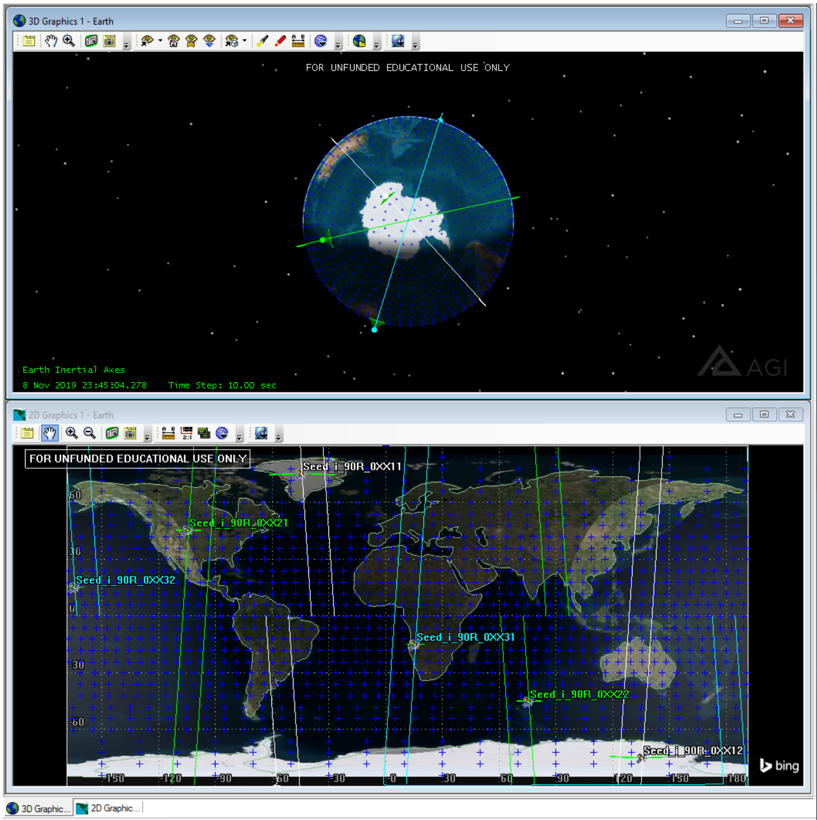

Figure 6: Constellation-1 with two satellites per orbit for three polar orbits of RAAN separation of 120 deg and a true anomaly of 60 deg.

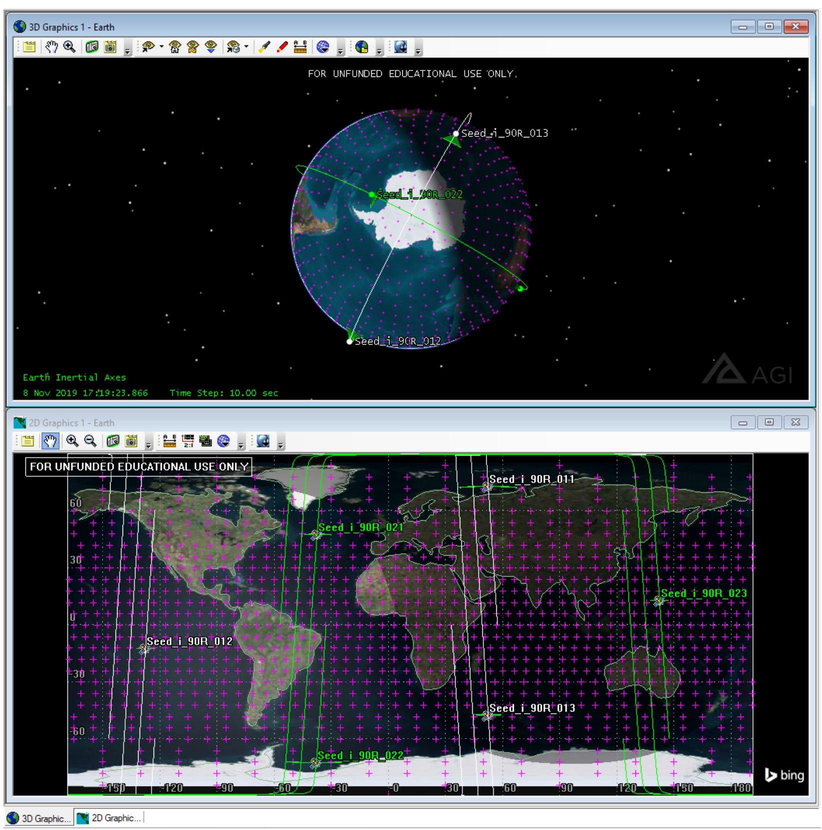

Figure 7: Constellation-1 with three satellites per orbit for two polar orbits of RAAN separation of 90 deg and a true anomaly of 30 deg.

The results are from the STK analysis for both the constellations are shown below:

	Starting 8th Nov 17:00:00 UTC	100% Coverage Time
Constellation 1	9th November 5:09:00 UTC	12 hours and 8 seconds
Constellation 2	8th November 23:08:00 UTC	6 hours and 8 seconds

Table 2: Comparison of potential constellation for coverage and time resolution requirements.

As seen above, Constellation-2, with three satellites per orbit for two polar orbits (inclination = 90^o) of RAAN separation of 90 deg and a true anomaly of 30 deg meets our time resolution requirements of 6 hours with a 100% global coverage. It also should be noted that this constellation also has a simpler orbital insertion since it only has two orbits in its configuration. The results from STK are shown below for Constellation-2.

```
                                                          9 Nov 2019 01:01:18
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CoverageDefinition-X6_Coverage


                        Coverage Properties
-------------------------------------------------------------------
Global Coverage
Grid Altitude: 0.0000 (km)
Ground Altitude set from grid altitude reference (On ellipsoid)
Resolution: 6.0000 (deg)
Number of Points: 1148
Assets required for a valid access: At Least 1
Assigned Assets:
    Constellation/SensorsX6:    Active
        Satellite/Seed_i_90R_011/Sensor/Sensor1:    Active
        Satellite/Seed_i_90R_012/Sensor/Sensor2:    Active
        Satellite/Seed_i_90R_013/Sensor/Sensor3:    Active
        Satellite/Seed_i_90R_021/Sensor/Sensor4:    Active
        Satellite/Seed_i_90R_022/Sensor/Sensor5:    Active
        Satellite/Seed_i_90R_023/Sensor/Sensor6:    Active
Access Interval: 8 Nov 2019 17:00:00.000 to 9 Nov 2019 17:00:00.000
Regional Acceleration: Automatic
Light time delay: Ignored
Maximum Sampling Time Step: 360.000 secs
Minimum Sampling Time Step: 0.010 secs
Time Convergence: 5.000e-03 secs
Value Convergence: Relative 1.000e-08 - Absolute 1.000e-14

8 Nov 2019 23:03:00.000          0.00              99.74
8 Nov 2019 23:04:00.000          0.00              99.74
8 Nov 2019 23:05:00.000          0.00              99.82
8 Nov 2019 23:06:00.000          0.00              99.91
8 Nov 2019 23:07:00.000          0.00              99.91
8 Nov 2019 23:08:00.000          0.00             100.00
8 Nov 2019 23:09:00.000          0.25             100.00
8 Nov 2019 23:10:00.000          0.00             100.00
8 Nov 2019 23:11:00.000          0.00             100.00
8 Nov 2019 23:12:00.000          0.00             100.00
8 Nov 2019 23:13:00.000          0.00             100.00
8 Nov 2019 23:14:00.000          0.00             100.00
```

Figure 8: Coverage results from STK for Constellation-2. Note: 1st column shows the time, 2nd column shows the individual percentage coverage and the 3rd column shows the cumulative coverage.

From the above analysis, one can decide the orbital inclinations, an optimal constellation configurations and the coverage characteristics associated with them backed by STK results. This is indeed one of the most critical assessments and the baseline for the rest of the mission which was kept in mind while considering the potential solutions.

3. Communication System

3.1 Data Rates

i) Based on the payload information, payload data description and orbital velocity of the satellite, the on-orbit data rate is determined to make key decisions regarding the ground station selection and communication sub-system architecture. This calculation was made using the assumption that the subsequent scan lines just barely touch neighboring lines.

The data rates were calculated from the formula below starting from the data rate and breaking it down into its elements.

$$datarate = \frac{bits}{sec} = \frac{bits}{swath} \times \frac{swath}{sec}$$

$$= \frac{bits}{swath} \times \frac{swath}{km} \times \frac{km}{sec}$$

And from the payload antenna data collection regime, the number of bits for a single swath can be obtained using:

$$\frac{bits}{swath} = \frac{40elements}{swath} \times \frac{16bits}{channel} \times \frac{4channels}{element}$$

$$\frac{bits}{swath} = \frac{2560bits}{swath}$$

So, the total data rate can be written as:

$$\frac{bits}{sec} = \frac{2560bits}{swath} \times \frac{1}{20km/swath} \times \frac{7.5725km}{sec}$$

$$= \frac{969.28bits}{sec} \simeq \frac{970bits}{sec}$$

Assuming a housekeeping data rate of 1000 bits/sec from (SMAD pg. 330 10.4.4), the total data rate is known to be approximately 1970 bits/sec. A screenshot of the excel sheet for the calculations has been added to the appendix for further reference.

3.2 Ground Systems Selection

3.2.1 Selection Considerations

ii) While considering a ground station network for this type of mission a comparison between a dedicated ground systems network was done with a decentralized/alternative commercial on-demand network. Current ground stations networks such as The Air Force Satellite Control Network (AFSCN) are great for a mission which requires frequent data downloading and accesses for monitoring and can provide redundancy in the time of emergencies. Further given the mission requirements, it was not deemed worthy of a dedicated ground station network given its limited mission life of only 3 years and the mission low data rates. Some of the other considerations and supporting reasons for a decentralized and commercial ground station network are:

- Since our mission is based in polar orbit, not worth the investment to build a dedicated system since existing stations already exist near the north and south poles.
- Alternative systems usually save a lot of money and have a defined and predictable cost schedule.
- High predictable, reliability and availability. Although not necessarily designed for mission specific needs, most are highly reliable because they have many dispersed assets on the ground, making them more risk averse.
- With the polar setup, it is important to have geographically symmetric ground control stations for achieving the latency of 1 hour, which results in a more favorable configuration of a hybrid of alternative and dedicated ground control centers.

These reasons given us ample leverage to design for existing commercial ground station networks. So upon further research and based on our latency requirements, it is logically sound to look for ground stations at the poles for serving our polar orbit satellites as this would allow for low latency and would require less on-board data storage. It was found from the NASA[2] website that KSAT (Kongsberg Satellite Services) was an ideal choice for our mission which promised a latency of about 50 minutes or less for polar orbiting spacecraft. Figure 8[3] shows a numerous ground stations as a part of the network.

Figure 9: KSAT ground station network.

It should also be noted that the KSAT network has ample redundancy for its singular polar ground stations on both the poles. There are multiple other stations close to the polar regions which serve as backup stations which makes this network very useful for our mission. Finally, two stations for our mission were selected namely Svalbard Svalsat Station on the North Pole and Antartica Trollsat Station on the South Pole. These stations were modeled in STK to calculate the access times for our polar satellites to determine the on orbit data storage. The ground station access frequency is given below from STK analysis in Figure 10 and the access times are given in Table 3.

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[2] https://sst-soa.arc.nasa.gov/11-ground-data-systems-and-mission-operations

[3] https://www.ksat.no/services/ground-station-services/#

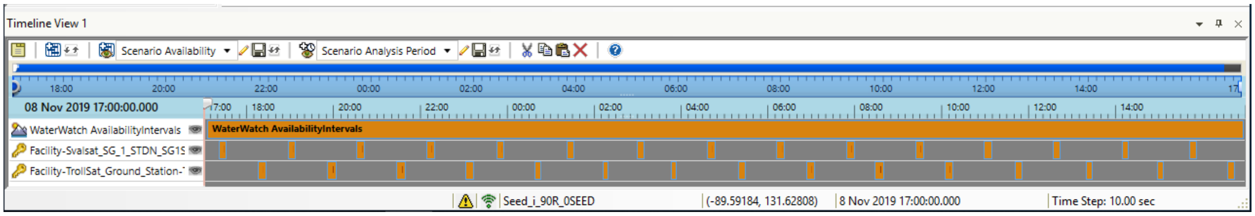


Figure 10: Access frequency for selected ground stations

	Mean Access Time / Orbit	No. of Acesses per Orbit
Svalbard Svalsat	689 seconds	1 access/orbit
Antartica Trollsat	644 seconds	1 access/orbit
Total Access	1342 secs OR 22 minutes	2 accesses/orbit

Table 3: Access times for selected ground stations

3.2.2 Downlink Data Rates and Data Volume

Based on this information, the required data storage can be calculated as a function of the data rate and the period of the orbit of the spacecraft which is calculated accordingly:

$$datastorage = \frac{bits}{sec} \times \frac{sec}{orbit}$$

$$= \frac{1970 bits}{sec} \times \frac{5767 sec}{orbit}$$

$$= \frac{11356858.58 bits}{orbit} = \frac{11.35 Megabits}{orbit}$$

For the downlink rate for the above on orbit data collection, the downlink rate can be given by:

$$= \frac{totalstoreddata}{dumptime} = \frac{11356858.58 bits}{1333.73 seconds} = \frac{8515.17 bits}{sec}$$

From the ground specifications given by KSAT[4], the maximum data rate provided by the ground station is about 2 Mbits/sec which would only require an access time of 165 seconds

[4] https://sbir.gsfc.nasa.gov/sites/default/files/453-NENUG%20R2.pdf

for our spacecraft to downlink all the data. As it can be seen this access time is significantly less then our required access time which allows us to comfortably downlink all the on-orbit stored data.

3.3 Link Budget

Assuming X-band link to the satellites with typical values of all the other parameters a link budget was developed with a 3dB link margin along with the data rate from part-ii). Two antennas from RUAG and AntDev were analyzed and compared for our link budget since specifications sheets with abundant data were available.

The following methodology and underlying assumptions were recognized for this link budget.

- The frequency for the x-band antenna was chosen to be 8.025 GHz as a majority of the COTS (Commercial off-the-shelf) parts in the market provide this frequency.
- The amplifier efficiency is based on the type of power amplifier (TWTA or SSPA). For our mission SSPA (Soli-states power amplifier) is used since out output power requirements are low and the amplifiers are light and more reliable. We assume an efficiency of 50% for SSPA.
- The line loss can be a value between -1dB and -3dB. In this case the average is taken to be -2dB.
- Transmitter antenna beam-width is based on antenna type and can be found from Table 13-14 SMAD. For this application helix antennas were chosen as they often have a lighter mass and are easier to mount on a satellite structure (SMAD pg. 571 13.4).
- Peak transmit antenna gain was found from Eq 13-18b SMAD. This value is often given by the manufacturer of the component as well.
- Transmitter antenna efficiency is assumed to 0.55 from Table 13-14 SMAD.
- The transmitter pointing error is approximated to be 10% of the beam-width assuming if tracking is used.
- The point loss can be computed from the above entities using Eq 13-21 SMAD.
- Transmit antenna net gain can be calculated from the sum of peak transmit antenna gain and antenna pointing loss.
- Equiv. isotropic radiated power can be computed from the sum of transmitter power, transmitter line loss and transmit antenna net gain.
- Propagation path length is just the distance between the transmitter and receiver antennas.
- Space loss can be computed from Eq 13-23 SMAD. This is dependent from the ground station position and satellite orbit. This is assumed to be worst case viewing angle/distance.
- Propagation and polarization is -0.5 dB since the frequency will moderately get affected. This can be seen from Fig 13-11 SMAD.
- Receive antenna diameter, efficiency, peak receiver antenna gain and receiver antenna beam-width can be found using the specifications sheet from KSAT Ground Stations.[5]
- Receive antenna pointing error is assumed to be 10% of beam width.
- Receiver antenna pointing loss can be computed by Eq 13-21 SMAD, which consider the losses due to pointing inaccuracies.
- Receiver antenna net gain can be derived from the sum of receiver antenna gain and receiver pointing losses.
- System noise temperature can be found using Table 13-10 SMAD.
- Data rate is used from part-ii) calculations as 8515 bits/second.
- Modulation rate is noted from Fig 13-9 as a function of Eb/No and here we use BPSK modulation scheme with error correction as that is consistent with the ground station.
- Computer implementation efficiency is assumed to be 90% or 0.9.
- Eb/No and noise to density ratio can be found by Eq 13-13 and 13-15a SMAD.
- Required Eb/No can be found using 13-9 SMAD.

[5] https://sbir.gsfc.nasa.gov/sites/default/files/453-NENUG%20R2.pdf

- The margin is calculated by using Eb/No, Required Eb/No, implementation loss and rain attenuation.

Note: The calculations and equations can be found in the attached Excel Sheet and the referenced equations, tables and figures are from SMAD 3rd Edition.

After all the values were in place the transmitter power (DC) was adjusted to give a link margin of 3dB. The transmitter power was found to be 0.0081 Watts for RUAG and 0.0102 Watts for AntDev antennas. This can also be verified by following Figure 11.

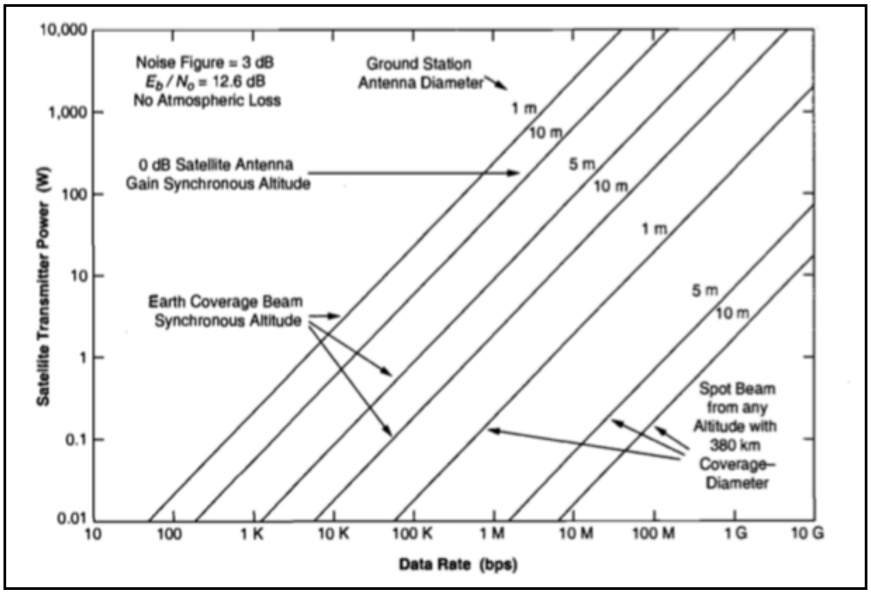


Figure 11: Transmitter Power as a function of data rate and receiver antenna diameter. (SMAD Fig. 13-5)

As it can been seen, for a data rate of about 8Kbps and approximately 10m ground station antenna diameter, the transmitter power is on the scale of 0.01 Watts which is in the same range as values found through the link budget.

Since both the antenna yielded similar results, Antdev Helix Antenna was chosen as it was documented on the NASA website as a popular smallest antenna and more data was available. The link budget for further reference.

The components for the communication sub-system are listed in the table below with their mass, power and vendor information link.

Table 4: Communications sub-systems component selection

Components	Mass		Power		Links
ANTdev Antenna	0.25	kg	0.0102	Watts	Spec Sheet
X Link Transciever	0.2	kg	15	Watts	Spec Sheet

4. Attitude Determination and Control Systems

4.1 Control Method Selection

i) From our given critical requirements of pointing accuracy of 0.1 deg and a nadir facing payload, a Zero Momentum (3-wheels) would be ideal type of Attitude Control method. This is because this control method offers no constraints for pointing and meets the pointing requirement of 0.1 degree. It should be also noted that this is not the only system capable of meeting our requirements but it gives the most controllability for pointing. SMAD table 11-4 for information about this. A magnetorquer is chosen for momentum dumping as the satellite is in a polar orbit where the magnetic fields are strong which assist with this and further, the magnetroquers are efficient in LEO and are relatively small. With this control method, the momentum wheel will be used for pitch and magnetorquers for momentum dumping and roll/ yaw.

Attitude determination sensor suite applicable for nadir pointing requirements, and 3-axis stabilization, horizon sensor for local vertical reference can be used for controlling pitch and roll (Table 11-6 SMAD). Either type in that category:- fixed head (static) for tighter pointing requirements and scanner piper for LEO applications can be used. Along with this, a sun or star sensor for third-axis reference and altitude determinations can be used. This sensor-suite will help us determine our attitude in the 3-dimensional space.

4.2 Control Actuator Sizing

4.2.1 Disturbance Torques

ii) Our disturbance torques are sized based upon the following parameters. Values like C_g offset, solar reflectivity, magnetic dipole are assumed to be designed for. Moment of inertia calculations are done assuming that there is a uniform mass distribution throughout the satellite.

Known Parameters			
Mission Lifetime	3	years	requirements
Mass	50	kg	constraint
Spacecraft Size	0.5 X 0.5 X 0.6	m X m X m	constraint
Alititude	573	km	requirements
Pointing Accuracy	0.1	degrees	requirements
Area	0.3	m^2	derived parameter
c_g offset	0.1	m	controlled design
solar reflectivity	0.95	-	controlled design
drag coefficient	3	-	scaled from similar size
magnetic dipole	0.1	A m^2	controlled design
moments of inertia	2.541	kg m^2	$I_x = I_z$ SMAD Fig. 11-49

Known Parameters			
moments of inertia	2.08	kg m^2	I_y SMAD Fig. 11-49

Table 5: Known parameters for calculations of disturbance torques.

There are four major types of torques which a spacecraft can undergo while in orbit namely: gravity-gradient based, solar radiation, magnetic field and aerodynamic drag. They are calculated as follows by using table 11-9A SMAD.

The torque due to gravity was found by using the following equation. The following values were used:-
R = 6851000 m and theta = 0.00174532 radians.
The calculated value was found to:- T_g = 2.86E-09 N m.

$$T_g = \frac{3\mu}{2R^3}\left|I_z - I_y\right|\sin(2\theta)$$

where T_g is the max gravity torque; μ is the Earth's gravity constant (3.986×10^{14} m^3/s^2); R is orbit radius (m), θ is the maximum deviation of the Z-axis from local vertical in radians, and I_z and I_y are moments of inertia about z and y (or x, if smaller) axes in kg·m^2.

Solar radiation pressure, T_{sp}, is highly dependent on the type of surface being illuminated. A surface is either transparent, absorbent, or a reflector, but most surfaces are a combination of the three. Reflectors are classed as diffuse or specular. In general, solar arrays are absorbers and the spacecraft body is a reflector. The worst case solar radiation torque is

$$T_{sp} = F(c_{ps} - cg)$$

where $F = \frac{F_s}{c} A_s (1+q)\cos i$

and F_s is the solar constant, 1,367 W/m^2, c is the speed of light, 3×10^8 m/s, A_s is the surface area, c_{ps} is the location of the center of solar pressure, cg is the center of gravity, q is the reflectance factor (ranging from 0 to 1, we use 0.6), and i is the angle of incidence of the Sun.

The torque due to solar radiation was found by using the following equation.

The following values were used:- A_s = 0.3 m^2; q = 0.95; i = 0^o; F = 2.77E-06 N and (c_{ps} - c_g) = 0.1 m.

The calculated value was found to be:-
T_{sp} = 2.77E-07 N m.

The torque due to magnetic field was found by the using the following equation. The following values were used:- D = 0.1 A m^2 and R as above.

The calculated value was found to:- T_m = 4.74E-06 N m.

$$T_m = DB$$

where T_m is the magnetic torque on the spacecraft; D is the residual dipole of the vehicle in amp·turn·m^2 (A·m^2), and B is the Earth's magnetic field in tesla. B can be approximated as $2M/R^3$ for a polar orbit to half that at the equator. M is the magnetic moment of the Earth, 7.96×10^{15} tesla·m^3, and R is the radius from dipole (Earth) center to spacecraft in m.

Atmospheric density for low orbits varies significantly with solar activity.

$$T_a = F(c_{pa} - cg) = FL$$

where $F = 0.5\,[\rho\, C_d A V^2]$; F being the force; C_d the drag coefficient (usually between 2 and 2.5); ρ the atmospheric density; A, the surface area; V, the spacecraft velocity; c_{pa} the center of aerodynamic pressure; and cg the center of gravity.

The calculation for torque due to aerodynamic density was done by the using the following equation. The following values were used:- (c_{ps} - cg) = 0.1 m; rho = 1.8E-12; C_d = 3; A = 0.3 m^2; V = 7572.59 m/s from circular orbit velocity formula.

The calculated value was found to be:- 4.64E-06 N m.

As seen from the above calculations, the worst-case disturbance torque is caused due to the atmospheric drag and magnetic field which can be stipulated since the altitude is quite low and the orbit is selected as a polar orbit where the magnetic forces are the most significant.

Now, that we have an estimate of the worst-case torques and disturbances, we can design for them using out momentum storage in the momentum wheels. The momentum stored in the wheels is given by the following formula.

Momentum Storage in Momentum Wheel	Roll and yaw accuracy depend on the wheel's momentum and the external disturbance torque. A simplified expression for the required momentum storage is: $$T \times \frac{P}{4} = h\theta_a$$ T = torque P = orbit period h = angular momentum θ_a = allowable motion

The equation (Table 11-12 SMAD) shows how 'h', angular momentum is a function of 'T' maximum external disturbance torque, 'P' is the period, theta is the allowable motion for pointing. Here, a slight modification is made and 'T' is calculated to be the sum of all the torque disturbances as this is critical for real-world applications. Another important consideration here is that we design for 'P/4', which is the quarter-orbit momentum dumping strategy so that the need for a bigger momentum wheel is eliminated.

The following values were used for calculations: T_d = 9.66E-06 N m from the total torque disturbance from above, P = 5767.42 s, which can be calculated by dividing the circumference of the orbit by the orbital velocity V; and theta is 0.001745 radians. The calculated angular momentum storage required is found to be:- h = 7.98 Nms.

For our momentum dumping via a magnetorquer, we use the formula on the left to estimate the capacity of the magnetorquer. Here 'T' represents the worst-case disturbance torque which in this case is from magnetic field, where the choice of including a margin of 30% is made for accounting for the lack of complete directional control and also higher than usual disturbances due to stronger magnetic field in the polar orbits. The following values of T = 6.16E-06 N m s (including the margin) was used and B = 4.5E-05 Tesla was used as a worst-case Earth field. The value for the magnetorquer was found to be D = 1.37E-01 A m^2.

Magnetic torquers use the Earth's magnetic field, *B*, and electrical current through the torquer to create a magnetic dipole (*D*) that results in torque (*T*) on the vehicle:

$$D = \frac{T}{B}$$

Magnets used for momentum dumping must equal the peak disturbance + margin to compensate for the lack of complete directional control.

Based on our calculations and sensor/actuator design choices, the following list of components were selected.

Components	Mass		Power		Product Links
RW8 Momentum Wheel	4.1	kg	10	Watts	Spec Sheet
2 X Magnetorquer Rods	0.06	kg	0.4	Watts	Spec Sheet
Sun Sensor	0.375	kg	0.25	Watts	Spec Sheet
2 Horizon Sensors	0.264	kg	0.066	Watts	Spec Sheet
Magnetometer	0.22	kg	1.5	Watts	Spec Sheet

Table 6: Component Selection for Attitude Determination and Control Systems

5. Command and Data Handling

5.1 C&DH System Breakdown

The C&DH sizing process started with identifying the functions to be performed by the system. These functions include command processing with an onboard computer. The system shall process telemetry data. it should be designed to process at least 1000bit/sec of housekeeping data and 1000bit/sec of payload data. The system shall be keep mission time. The constraints around which this system is designed around are as follows:- the system shall fit within the CubeSat bus of 0.5 X 0.5 X0.6 meters. The reliability of the system should be at least 98% with a design life of more than 3 years. System complexity definition is given in the Table 7 below:-

Requirement/ Constraint	System Complexity	Justification
Command Processing	Simple: 50 cmds/sec	System shall be capable of simple commands
Telemetry Processing	Simple: 500-4kbps	System shall be capable of simple telemetry handling
Mission Time Clock	No	Not required/critical to the mission
Bus Contraints	Single Unit	System shall be a single unit system which fits in the satellite bus
Radiation Environment	< 2 krads	System shall be able to radiation to survive for at least 3 years

5.2 Component Selection & Sizing

Upon the definition of system functions, C&DH can be sized and components are selected. From Table 11-29 typical C&DH systems which are simple complexity harboring telemetry and command functionalities range between 2500-6000 cm^3. Researching COTS components, RAD750 was found to be meeting the requirements with exceeding performance and reduced mass and power costs. The table below describes the specifications of the C&DH system.

Components	Mass		Power		Product Links
RAD750 BAE Systems	0.549	kg	10.8	Watts	Spec Sheet

Table 9: Component Selection for Command and Data Handling

6. Radiation Analysis

6.1 Orbit Radiation Exposure

i) Radiation can be of the most dangerous “forces" in the space environment which can severely damage the spacecraft and can cause irreversible damage to its components. Therefore it is critical to design for radiation for the entire length of the mission lifetime, to select parts which are immune to radiation doses, and have payload and instruments shielded from it. This directly, affects the mass of the spacecraft by the amount of shielding required as well as the cost of mission as the radiation treated electronics are much more expensive. The following section deals with the radiation doses the spacecraft is destined to see and the design decision that are taken as a result. Using SPENVIS, an analytical tool to measure radiation at a given time on a spacecraft in a given orbit, Figure 12 show the orbit averaged flux during the mission timeline. One key consideration while setting up the simulation was to account for the upcoming solar maximum in the next 5 years and design for the worst-case radiation scenario for robustness.

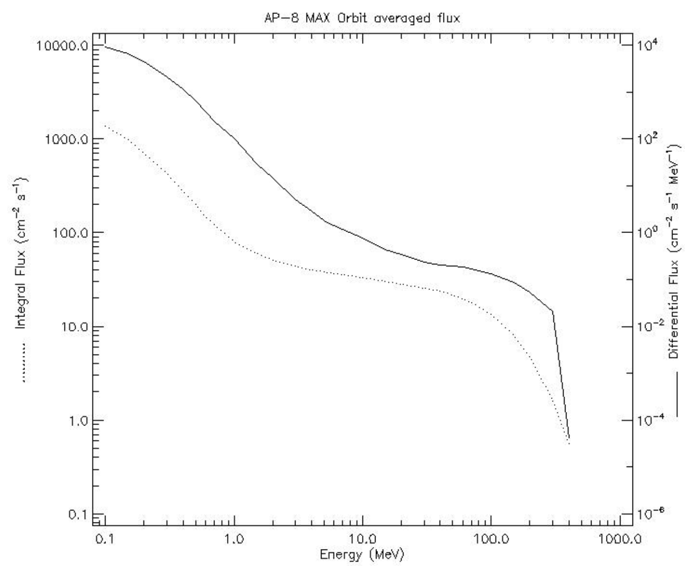


Figure 12: Orbital average flux for spacecraft during the mission timeline.

The plot below shows the radiation dosage in Aluminum as a function of the its thickness which helps us design spacecraft exterior and key components with walls of necessary thickened to limit the radiation dosage.

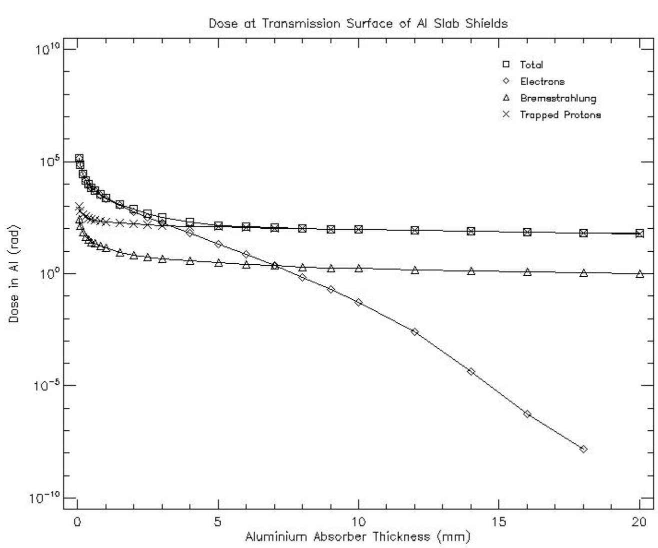


Figure 13: Radiation dosage as a function of Aluminum wall thickness.

It is also important to see the characteristics of charged oxygen and hydrogen particles and their penetration ranges as a function of the particle energies. The figure below was obtained using SRIM, an analytical tool which is a Monte-Carlo program. It shows the range of O and H particles' range for energies from 1-100 Mev/nuc.

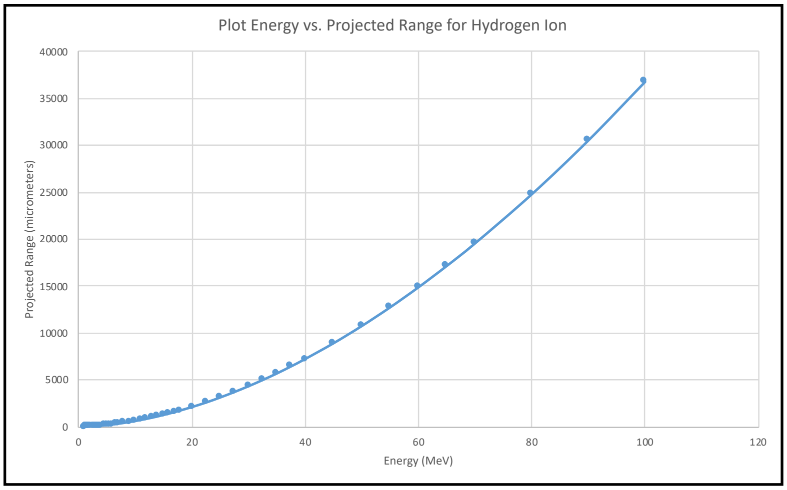


Figure 14: Range of Hydrogen ions for Energies of 1-100 Mev/nuc.

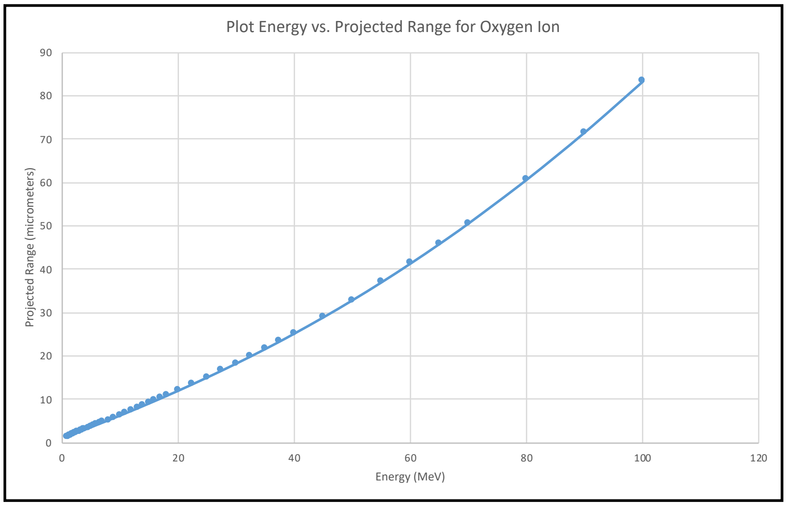


Figure 15: Range of Oxygen ions for Energies of 1-100 Mev/nuc.

To determine the thickness and the mass of the Al shielding needed to eliminate 90% of all O+ at 10Mev/nuc, an iterative approach was used. Initially a relatively thick Al sheet was used to see the average depth of penetration for the particles. As shown below, a thickness of 10 um was used for the first trial.

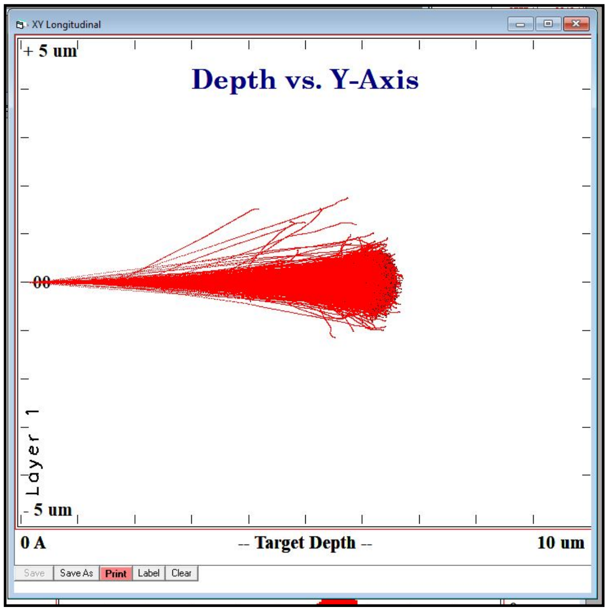


Figure 16.1: Iteration 1 for finding thickness of Aluminum to eliminate 90% of all O+ at 10Mev/nuc.

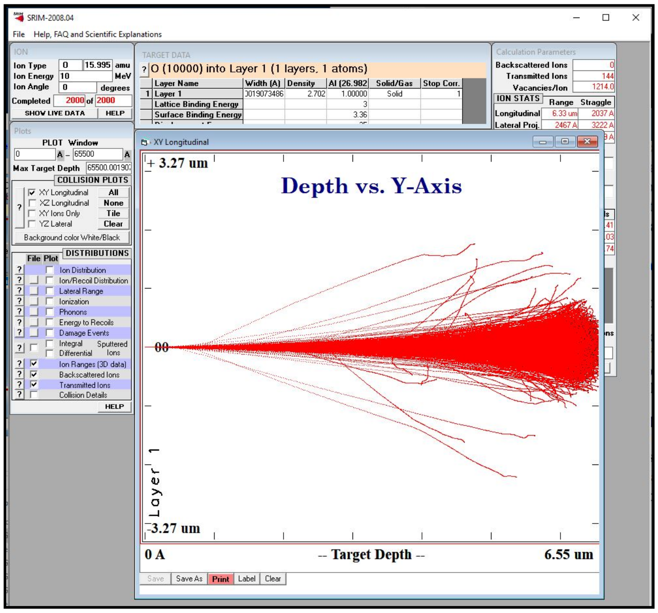
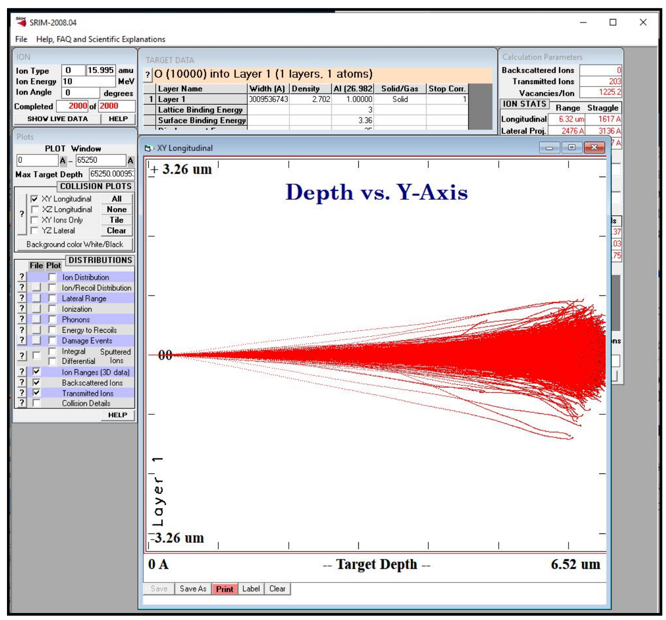


Figure16.2: Iteration 2 for finding thickness of Aluminum to eliminate 90% of all O+ at 10Mev/nuc.

Using the previous plot, the thickness of about 6.55 um was used. The plot below shows the result. The upper right hand corner in the image below, shows the number of ions which passed through. It is seen that 144 ions passed through which is a little less permeable than the required transmission of 90%.

Following on to this trial and error method, a final value of 6.525 um was obtained for which the results are shown below. This allows only 10% transmission through the Al layer. The input and output are shown below.

Figure16.3: Final iteration for finding thickness of Aluminum to eliminate 90% of all O+ at 10Mev/nuc.



Considering all the radiation effects in the space environment for the satellite it would be naive to design an aluminum wall for a 6.25 nanometer which is why a thickness of 2 mm is chosen. This will limit the dose of radiation fo 10^3 rad (Figure-13) to protect all the electronics. COTS equipment in the recent generation products usually offers a protection/normal functions under 10^3 rads so this margin should be sufficient to shield the electronics.

7. Power System

7.1 Power Architecture

7.1.1 Component Power Consumption

As sub systems are defined and components selected, power budgeting becomes critical and expensive which can lead to sizing issues and tight tolerances in power management. The table shows a list of components selected for each sub-system and the power requirements associated to them.

	Components	Power (Watts)
Payload	Payload	25
	Antenna Panel	
	Redundant Antenna	
TT&C	ANTdev Antenna	0.0102
	X Link Transciever	15
	Redundant Antenna	0.0102
	Redundant Transciever	15
ADCS	RW8 Momentum Wheel	10
	2 Magnetorquer Rods	0.4
	Sun Sensor	0.25
	2 Horizon Sensors	0.066
	Magnetometer	1.5
C&DH	RAD750	10.8

Table 9: Components and associated power consumption.

As seen above, the largest and the most critical power requirement is from the payload, followed by ADCS and TT&C. It is very important to determine operating power requirements during various phases in orbit: eclipse and daylight. To carry out all necessary operations, the power system is designed for a net positive power for all of the time during operations. A healthy margin of 40% was considered when designing for the solar array sizing and the battery requirements. This is to account for the thermal heating requirements, anomalies in power draws, solar panel malfunction etc so spacecraft can still function at minimal functions.

The calculation for the solar array sizing was conducted with the following parameters:

System Power	80	Watts	Total power
P_e & P_d	112.05	Watts	40% margin
X_e and X_d	0.6 & 0.8 resp.	60% & 80%	assuming peak power tracking
T_e	30.7	minutes	eclipse duration (STK)
T_d	65.39	minutes	daylight duration (STK)

Table 10: Parameters for solar array sizing

The total power that must be produced by solar arrays is given by Eq 11-5 from SMAD:

$$P_{sa} = \frac{\left(\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}\right)}{T_d}$$

From the parameters, P_{sa} = 227.74 Watts. Based on a Germanium ITJ solar cell by spectrolab, the efficiency is about 0.225. The nominal power output with Sun normal to the surface is given by:- P_o = 1367 W/m^2 X (efficiency), which yields about 307.575 W/m^2 of power output.

The beginning of life power output is given by the following formula where the cosine term is the power production loss due to the sun ray incident angle at theta = 35^o. I_d is assumed to be 0.77 which is defined as inherent degradation caused by defects in the design, temperature, manufacturing and assembly of solar panels.

$$P_{BOL} = P_o I_d \cos \theta$$

The P_{BOL} is computed to be about 217.19 W/m^2. Further to account for the radiation damage to the solar arrays, the life degradation and power at end-of-life is given by:

$$L_d = (1 - degradation/yr)^{satellite\ life}$$

$$P_{EOL} = P_{BOL}\, L_d$$

It should be noted that the solar array is sized primarily based on the degradation of the solar cells at the end of life so that the solar array has the ability to generate power even after degrading. Degradation usually occurs because of thermal cycling in and out of eclipses, plume impingement from thrusters and material outgassing for the duration of the mission. The P_{EOL} value was found to be: 199.7 Watts with an L_d = 0.919 for a degradation of 2.75% per year for a GaAs solar cell. Finally the required area of the solar cells can be found by the following formula which is computed to be approximately: 1.14 m^2.

$$A_{sa} = P_{sa} / P_{EOL}$$

By using the unit area of a single cell and the weight per cell from the specifications sheet, the total weight of the solar cell can be computed to be 0.95 kg. As a result the battery capacity can be computed by:

$$C_r = \frac{P_e T_e}{(DOD)Nn} \text{ W-hr}$$

where DOD is depth-of-discharge which is the battery draining level from the full capacity and it is a function of cycles, N is the no of batteries and n is the battery efficiency. For computing the battery capacity, DOD is taken from Fig 11-11 S,AD as function of 16400 cycles for a lifetime of 3 years in orbit at 573 km altitude, N=2 for redundancy, n =0.9 as a typical battery efficiency which gives a required battery capacity of C_r = 76.44 W-hr per battery. Hence, the battery selected which fulfills this requirement is a Lithium Ion battery by EaglePicher Technologies.[6] With the battery weighing about 2.02kg and with a capacity of 121 W-hr/kg, one battery is more than sufficient but two are considered for redundancy. With our batteries and solar panels sized, the components selected which meet our requirements are listed in the table below.

Components	Mass		Power		Product Links
Solar Panels ITJ	0.958	kg	—	—	Spec Sheet
SLC-16050 Battery X 2	4.04	kg	—	—	Spec Sheet
Solar Deployment Mechanism	1	kg	2	Watts	Spec Sheet
Harness/Cabling	1.19	kg	—	—	assume 20% of power mass
Power Control Unit	1.6	kg	—	—	2% of total power (SMAD pg. 423)
Regulators/Converters	2	kg	—	—	2.5% of total power (SMAD pg. 334)

Table 11: Component Selection for Power System

8. Thermal Systems

8.1 Radiator & Heater Sizing

Temperature fluctuations in space environment is extremely volatile and it can cause some major temperature gradients from the coldest temperature to the hottest ones. Many electrical components are major victims of the temperature variations which make it essential for thermal management in the spacecraft so that instruments can operate in their nominal temperature ranges. For this spacecraft, the battery is assumed to be the system driver as it has to operate in a nominal temperature range. The table below summarizes the requirements of the battery from the spec sheet and some other values which are used in the thermal calculations.

	Operating Range			Non-operating Temperature		
Battery	30	10	C	-5	C	from battery spec sheet - system driver
	303	283	K	268	K	from battery spec sheet - system driver
Waste Heat	22.4	16.8	Watts			20% - 15% of total power converted

Table 12: Battery operating & non-operating temperatures as a system driver for thermal systems

[6] https://www.eaglepicher.com/sites/default/files/EP_SLC_16050_DATA_SHEET.pdf

It should be noted that, the waste heat generated by the spacecraft is assumed to come from the power dissipation and waste heat generated by electrical components and is approximately 20% of the power converted by the voltage regulator. (SMAD pg. 334). To size the radiators, we first need to conduct an energy-balance check to see if the spacecraft needs heaters or radiators during it maximum heating/cooling. So we model two cases: hot and cold. The tale below holds the parameters for both the cases.

Parameter	Hot Case		Cold Case		
k	1420	W/m^2	1360	W/m^2	solar constant
F	1	—	1	—	worst case
MLI EOL	0.1	—	0.05	—	MLI blanket for heat transfer
$A_{s/c}$	0.3	m^2	0.3	m^2	area of spacecraft face

Table 13: Parameters for calculating solar heat loads

It is assumed that the radiator emissivity of 0.8, BOL alpha of 0.05, and EOL alpha of 0.15. Also, its assumed that the Earth facing radiator and no backloads. The absolute worst-case scenario for heating is presumed to make the necessary calculations. We start with the following equations for energy balance:

From a generalized heat balance equation:

$$Q_{in} = Q_{out} \tag{11-12}$$

$$Q_{external} + Q_{internal} = Q_{radiator} + Q_{MLI} \tag{11-13}$$

And $q_{external}$ can be broken down into:-

$$q_{external} = q_{solar} + q_{albedo} + q_{EarthIR} + q_{backload} \tag{11-16}$$

From assumptions $q_{backload} = 0$. The following formula gives the q_{solar}:-

$$I_{avg} = A\,K\,F \tag{5-11}$$

A is the area of the face and K = 1420 W/m^2 is the solar constant in the vicinity of the Earth for hot case (Table 11-48A). F is the time average fraction of the surface area projected in the direction of the Sun and must lie between 0 and 1. F is chosen to be 1 for worst-case and MLI EOL = 0.1. Plugging in these values we get q_{solar} = 42.6 W/m^2 for the hot case.

The term q_{MLI} is computed for the 4 sides of the spacecraft which can be broken down into 3 sides facing the space environment/cold side and 1 side facing the sun/hot side. So Q_{MLI} can be computed with MLI values of 0.01 for cold side and of 0.03 for hot side during the hot case from Table 11-48A in SMAD. q_{MLI} = 4.18 W/m_2 and 12.5 W/m^2 for the cold side and the sun side respectively. Q_{MLI} is found by multiplying the areas of 3 cold faces (2 side+1 top) and 1 hot face(1 side) with the respective cold and hot side values for q_{MLI} and we get Q_{MLI} = 7.31W/m^2.

From Table 11-48B, the values for q_{albedo} and q_{IR} are obtained to be 21 W/m^2 and 161 W/m^2.

Further, $q_{external}$ was found to be 182 W/m^2; waste heat was 22.4 W; operating temperature was 30 C considering a 10 C margin; with an emissivity of 0.8. Combining all the individual terms:-

A_{rad} = (22.4 + $A_{s/c}$*q_{solar} - Q_{MLI})/(q_{rad} - $q_{external}$), where q_{rad} and external are terms without the radiator A_{rad}.

This yielded an area of radiator: A = 0.38 m^2.

From Table 11-49, the density for the radiator was found to be 3.3 kg/m^2. The mass of the radiator was obtained to be m = 1.25 kgs.

Now we solve for the cold case where: from Table 11-48B, the values for q_{albedo} and q_{IR} are obtained to be 0 W/m^2 and 148 W/m^2. Q_{solar} can be computed by using Eq 5-11, to be 20.4 W/m^2. Solving for the left side of the energy equations to find:

$Q_{in} = Q_{external} + Q_{internal}$

$= 16.8 + Q_{solar} + A_{rad}*(q_{albedo} + q_{IR}) = T^4*(q_{MLI} + q_{rad})$

T = 265.4 K without margin and 255.4 with margin.

As it can been noted, T is lower than our operating temperature so we would need a heater. But we do not stop at this stage to find the heater power required as there is an even colder case when there are minimal electronics running and least amount of heat is being produced to keep the temperatures high. So we design the heater for the non-operating case.

We assume that the power at the non-operating case only has the communications and ADCS systems functional and 15% of power is dissipated as waste heat which is equal to 4.08 Watts. So again solving the left hand side equations:

$Q_{in} = Q_{external} + Q_{internal}$

$= 4.08 + Q_{solar} + A_{rad}*(q_{albedo} + q_{IR}) = 80.54$ Watts

$T^4 = 80.54/(q_{MLI} + q_{IR})$

T = 255.84 K

This temperature is certainly lesser than our minimum operating temperature, so we need a heater, as expected. Considering $T_{non-operating}$ = -5 C, with a margin of +10 C, we have T = 278K.

Using that temperature and solve for $Q_{heater} = Q_{MLI} + Q_{rad} - Q_{solar} - q_{external}*A_{rad}$, and we get a heat of 35.82 Watts required for heating for the spacecraft.

9. Overall Subsystems
9.1 Satellite mass and volume budget

Components	Length (m)	Width (m)	Height (m)	Volume (m³)	Mass (kg)	Power (watts)	Data (bits/s)
Envelope	**0.5**	**0.5**	**0.6**	**0.15**	**50**		
Payload	0.1500	0.1500	0.2500	5.63E-03	15.000	25.00	970
Antenna Panel	0.1000	0.5000	0.0300	1.50E-03	1.1400	—	
Redundant Antenna	0.1000	0.5000	0.0300	1.50E-03	1.1400	—	
ANTdev Antenna	0.1016	0.1016	0.0762	7.87E-04	0.2500	0.0102	
X Link Transciever	0.0200	0.0200	0.0200	8.00E-06	0.2000	15.000	
Redundant Antenna	0.1016	0.1016	0.0762	7.87E-04	0.2500	0.0102	
Redundant Transciever	0.0200	0.0200	0.0200	8.00E-06	0.2000	15.000	1000
RW8 Momentum Wheel	0.1900	0.1900	0.0900	3.25E-03	4.1000	10.000	
2 Magnetorquer Rods	0.0200	0.0200	0.1400	5.60E-05	0.0600	0.400	
Sun Sensor	0.1080	0.1080	0.0520	6.07E-04	0.3750	0.2500	
2 Horizon Sensors	0.0860	0.0636	0.0636	3.48E-04	0.2640	0.0660	
Magnetometer	0.1000	0.0500	0.0400	2.00E-04	0.2200	1.5000	
RAD750	0.1000	0.1600	0.0500	8.00E-04	0.5490	10.800	
Solar Panels ITJ	1.140		0.0055	6.27E-03	0.9500	—	
Solar Deployment Mech.					0.9900	2.0000	
SLC-16050 Battery X 2	0.1730	0.081	0.0569	7.97E-04	2.0200	—	
	0.1730	0.081	0.0569	7.97E-04	2.0200	—	
Harness & Cabling					1.1900	—	
Power Control Unit	—	—	—	—	1.6000	—	
Regulators/Converters	—	—	—	—	2.0000	—	
Satellite Walls	0.0020	0.0020	0.0020	0.0017	4.7421	—	
Radiator	0.38 m²		—	—	1.25	—	
Heater	—	—	—	—	—	35.82	
Total				0.0250	40.52	115.85	1970

Table 14: Volume, Mass, Data and Power Budget for Overall System

Based on the subsystems described above, the satellite mass, volume, power and data budget have been tabulated below in Table 14.
As seen from above, this is a preliminary budget which allows us to gain initial insight into the major system drivers and encapsulate an overall idea of the spacecraft systems. The formatting of the sub-systems is driven by the payloads requirements and the envelope will be based around the payload. The nadir earth facing side of the satellite will be covered by the payload and communications antennas, one of the sides will be covered by the radiator, solar panels will go on the sides, heater and other critical components will go in the centre of the spacecraft along with the ADCS actuators. The ADCS sensors will get priority near the outer sides to enable them to function. MLI blankets will cover the rest of the sides.

ii) For link budget, many uncertainties can arise due to inefficiencies in communications, line of sight issues, uneven power supply, etc. This is why there have been redundant antenna, transceiver and payload antenna extras in the system to account for any failures. The ADCS system may have some discrepancies between sensors, and the C&DH will have to be functional enough to take care of that. The power budget still needs to be refined, and many of the elements of the spacecraft are still not included so it may have some discrepancies. Heating and radiator sizing are very basic approximation which have been calculated from estimation of heat dissipations from electronics which need to be re-evaluated in the next design iteration. Overall, the system needs substantial refinement, but this paper is the first step towards identifying these shortcomings and tackle them in the next iteration.

10. Mission Recommendations

10.1 Mission Concept

Data delivery:- The mission data will be collected by 1 sensor namely: the 4-channel passive microwave radiometer capable of making thermal radiation at 10,19, 22 and 37 GHz. The data will be collected at a temporal resolution of less than 6 hours and spatial resolution of at least 20 km which will be stored and downlinked at north and south pole-based ground stations of the orbit. The RAD750 onboard will be responsible for obtaining housekeeping data and transmitting it as well as providing instrument power, functions and executing commands.

Communications architecture:- All instruments will be acquiring mission data constantly and autonomously. The satellite will be using x-band to communicate with and to the ground station. RAD750 will be responsible for data relay, interactions & commanding. The satellite's data (11.3 Mbits/orbit), including science, instrument and s/c housekeeping data) is encoded, stored and downlinked twice every orbit (96 minutes/orbit).

Tasking, Scheduling & Control:- Data is downlinked to the ground system operated by KSAT which is relayed directly to TDRSS Ground Terminal at White Sands, New Mexico. After processing, the telemetry data at MOC, the data are sent to the NASA Goddard's Precipitation Processing System (PPS) which handles the data distribution and archiving. Finally, the data is uploaded on NASA's website and can be accessed by the scientific community.

Mission Timeline:- The proposed program duration from Phase A - F, takes 5 years which includes the 3 year mission duration and 1 year mission extension.

DOPE - Distribution of Precipitation Element mission aims to investigate, monitor and model the global distribution of precipitation and its diurnal variability. The mission will include measurements with highest-to-date time resolutions for turbulent solar winds of approx. 6 hours and resolutions of approx. 20 km. The data will lead to a comprehensive understanding of properties of Earth's water cycle, and the look into the mechanisms by which precipitation patterns emerge.

10.2 Mission Timeline

Phase	Duration (month)	Justification	Proposed Review
Pre-Phase A Advanced Studies	10	Needs analysis/concept exploration	MCR: Mission Concept Review
A Preliminary Analysis	5	Develop baseline mission concept,	Mission Definition Review
B Definition	24	Operations based CONOPS, preliminary design	PDR: Preliminary Design Review
C Design	12	Finalize Design and Fabrication	CDR: Critical Design Review
D Development	12	System integration, assembly and test	ORR: Operational Readiness Review
E Operations	36	Conduct mission & meet mission objectives	PLAR - Post-launch assessment review

Table 15: Mission timeline and planned reviews

Based on this timeline, the proposed launch date is January 2025. Based on a heritage mission GPM - Global Precipitation Measurement, which is successfully ongoing, it is noted that the mission timeline during the Phase B, C, and D are a bit accelerated given the level of heritage from GPM and the expertise and existing research already in place, it should take this mission relatively less time.

10.3 Critical Requirements & System Drivers

Critical requirements are factors which dominate the space mission's overall design and, therefore, most strongly affect performance and cost. The critical requirements for this mission are listed as follow:-

CR1 - The system shall provide global coverage.
This requirement affects mission's most important parameters like altitude, number of satellites, altitude, inclinations, communication architecture, payload field of view, scheduling, staffing requirements.

CR2 - The system shall provide spatial resolution of at least 20 km at nadir.
Again, this requirement affects spacecraft altitude, communications architecture, payload field of view.

CR3 - The system shall provide temporal resolution (revisit time) of at least 6 hours to investigate the diurnal variability of precipitation.
This requirement influences the number of satellites, orbital inclinations communications architecture, ground system architecture, scheduling and staffing requirements.

CR4 - The payload sensor shall have a pointing accuracy of 0.1 degrees.

This requirement strongly affects the entire ADCS system to be designed around a strict pointing accuracy which will linearly map to greater costs and complexity in design.

CR4 - The system shall have a design lifetime of at least 3 years.
The survivability requirement affects the sub system design as it needs to have heavy redundancy, higher radiation dosage tolerant parts, ground stations assignments and support crew for the mission etc.

System drivers are the principle mission parameters or characteristics which influence performance, cost, risk, or schedule and which the use or designer can control.

SD-1 The size of the satellite
Needs complexity in functions to increase the size, requires more performance, increasing risk

SD-2 On orbit weight of the satellite
Needs more structural strength, adds to launch costs, needs more thermal protection

SD-3 Number of the satellites
Increases complexity in design, more nodes of failure, less reliability, increased risk

SD-4 Constellation configurations
Needs more complex maneuvers, orbit keeping is more difficult, increases operational costs

The four system drivers highly affect the performance, cost, risk and schedule to the missions development.

10.4 Architecture

DOPE is a NASA mission designed to monitor, analyze and collect data related to precipitation patterns globally from constellation of satellites by using a specialized 4-channel passive microwave radiometer capable of making measurements of thermal radiation at 10, 19, 22, and 37 GHz. The constellation will function as mesh-system of precipitation data collection at a time resolution of less than 6 hours from any point on the Earth. DOPE will provide, spatial resolutions of 20 km and will cover the Earth every six hours. This data will be transferred down to polar stations on the North and South polls which will be relayed to MOC in TDRSS Ground Terminal at White Sands, New Mexico. After processing, the telemetry data at MOC, the data will be sent to the NASA Goddard's Precipitation Processing System (PPS) for analysis and distribution. DOPE is scheduled to launch in January 2025 with a mission life of 3 years.

The technical risk items for this development are prioritized from the highest order to the lowest order of priority.

1) The phased array antenna which has been invented by the company has never been flown before which presents many challenges associated with risk. Technology readiness level is extremely low for a new technology. A lot of testing and has to be done which might not be favorable for the mission timeline.
2) A radiometer which is capable of making measurements of thermal radiation is being used for precipitation measurements which might not be technically sound application for the mentioned technology.

10.5 Launch Vehicle Selection

Based upon the mission requirements analysis, 6 satellites would be required for the temporal resolution of less than 6 hours. Considering this, the total payload weight for this mission would be about 300 kgs basing from 50 kg per satellite. Upon conducting research, the Polar Launch Satellite Vehicle PSLV - CA by the Indian Space Research Organization seems to be a goof fit for our requirements. This variant of their sun-synchronous 662 km altitude launch vehicle has a payload capacity of 1100 kg. Which allow us to comfortably fit our satellite constellation in the fairing. The PSLV-CA has also had a tremendous track record of 14 successes out of 14 launches without a single failure. But even though this meets our requirements, some of the design considerations would need to be reevaluated at since no launch providers provide launch inclinations of 90 degrees, rather they provide orbit insertion at 98.5 degree sun-synchronous orbit. It would be cost effective and fuel effective if the mission can be carried out in a sun-synchronous orbit without having to changing the inclinations imd-orbit which is very expensive. So to fit with a launch provider we would have to change some inclinations but this should not affect our mission drastically as the un-synchronous orbit is almost a polar orbit.

11. Bonus

11.1. De-orbiting

This equation can be used to find the orbit lifetime, here H is essential to the calculation and is not given in SMAD when I check the back of the book.

For near circular orbits, we can use the above equations to derive the much simpler expressions:

$$\Delta a_{rev} = -2\pi \, (C_D A/m)\rho a^2 \quad (6\text{-}24)$$

$$\Delta P_{rev} = -6\pi^2 (C_D A/m)\rho a^2 / V \quad (6\text{-}25)$$

$$\Delta V_{rev} = \pi \, (C_D A/m)\rho a \, V \quad (6\text{-}26)$$

$$\Delta e_{rev} = 0 \quad (6\text{-}27)$$

where P is orbital period and V is satellite velocity.

A rough estimate of the satellite's lifetime, L, due to drag is

$$L \approx -H / \Delta a_{rev} \quad (6\text{-}28)$$

where, as above, H is atmospheric density scale height given in column 25 of the Earth Satellite Parameter tables in the back of this book. We can obtain a substantially more accurate estimate (although still very approximate) by integrating Eq. (6-24), taking into account the changes in atmospheric density with both altitude and solar activity level. We did this for representative values of the ballistic coefficient in Fig. 8-4 in Sec. 8.1.

11.2. EOL Altitude

No satellite would not fulfill requirements at EOL altitude since the satellite will slow down due to perturbations and as a result, the altitude will increase which will reduce the resolution for the same viewing angle. Three things can be done:-

1. The viewing angle of the payload could be modifiable and decreased to get more resolution out of higher altitudes.
2. Design for the resolution at EOL so that when time=EOL the resolution requirement is still met. At BOL, the satellite will just have a better resolution then EOL.
3. On board propellant can be stored, so it can be used for station keeping and altitude maintenance.

11.2. Overheating Issue

Step-1: In case of any anomaly, heating issue etc, it should be the first priority to ensure that power is preserved. To shut of non-critical components to conserve energy. Orient the solar array towards the sun.

Step-2: Manage critical thermal conditions. Makes sure, the satellite has enough power to keep the heaters on so it can keep the instrumentation at an operating temperature range.

Step-3:Download data required for diagnosis. Obtain telemetry, on-board computer memory etc to see what is happening.

Step-4: Do not try to instantly fix the problem, let the problem play out and then try to manage it. Also do not instantly switch to safe mode, because theres a lot of risk involved with it.

12. Conclusion

DOPE - Distribution of Precipitation Element mission is a step forward in the learning about Earth on a global scale. Cross-cultural collaboration, scientific inquiry and sharing of knowledge are fundamental elements to the success of this mission and with great certainty, I look forward to its birth as a NASA mission from a seedling of an idea.

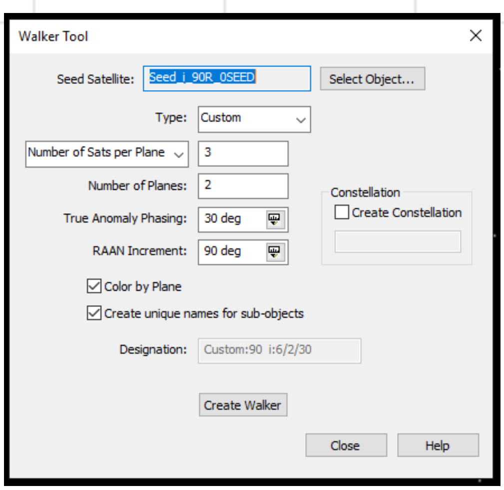
I am grateful for this opportunity and forever humbled by this mission's might. Looking forward to collaborating in the future.

13. Appendix

Components	Length	Width	Height (m)	Volume (m^3)	Mass (kgs)	Power (W)		
Envelope	0.5	0.5	0.6	0.15	50			
							% Mass	% Power(w/o margin)
Payload	0.15	0.15	0.25	5.63E-03	15	25	Payload	
Antenna Panel	0.1	0.5	0.03	1.50E-03	1.14		43.99	22.31
Redundant Antenna	0.1	0.5	0.03	1.50E-03	1.14			
ANTdev Antenna	0.1016	0.1016	0.0762	7.87E-04	0.25	0.0102	Communications	
X Link Transciever	0.02	0.02	0.02	8.00E-06	0.2	15	2.29	26.79
Redundant Antenna	0.1016	0.1016	0.0762	7.87E-04	0.25	0.0102		
Redundant Transciever	0.02	0.02	0.02	8.00E-06	0.2	15		
RW8 Momentum Wheel	0.19	0.19	0.09	3.25E-03	4.1	10	ADCS	
2 Magnetorquer Rods	0.02	0.02	0.14	5.60E-05	0.06	0.4	12.78	10.90
Sun Sensor	0.108	0.108	0.052	6.07E-04	0.375	0.25		
2 Horizon Sensors	0.086	0.0636	0.0636	3.48E-04	0.264	0.066		
Magnetometer	0.1	0.05	0.04	2.00E-04	0.22	1.5		
RAD750	0.1	0.16	0.05	8.00E-04	0.549	10.8	1.397740866	9.638471638
Solar Panels ITJ		1.140077742	0.0055	6.27E-03	0.958272		Power Systems	
Solar Deployment Mech.					0.99	2		
SLC-16050 Battery X 2	0.173	0.081	0.0569	7.97E-04	2.02		18.30	
	0.173	0.081	0.0569	7.97E-04	2.02			
Harness & Cabling	assumpt. 20% of power system mass				1.19765		SMAD pg 423	
Power Control Unit					1.600728		SMAD pg 334	
Regulators/Converters					2.00091			
Satellite Walls	0.002	0.002	0.002	0.001694	4.742102		Structures	
Material thickness 2mm			net volume				12.07	

Mass of Antenna				
Component	Thickness (m)	Volume (m^3)	Density (kg/m^3)	Mass (kg)
1 mm Copper	0.001	0.00005	8960	0.448
2.2 cm polystyrene	0.022	0.0011	45	0.0495
5 mm-thick PCB board	0.005	0.00025	1850	0.4625
1mm X 2 carbon fibre	0.002	0.0001	1800	0.18
Total Mass				1.14

Structures	pg 459



A	B	C	D	E	F	G	H
Given Parameters							
mission lifetim	3	years					
mass	50	kg					
spacecraft siz	0.5 x 0.5 x 0.6	m^3					
altitude	573	km					
	90	deg circular					
pointing accur	0.1	deg					
area	0.3	m^2					
Cg offset	0.1	m					
solar reflectivit	0.95						
drag coeff	3		assumed from table 8-3 pg 207				
magnetic dipo	0.1	A m^2					
Moments of In	2.541	kg m^2	Ix = Iz			SMAD fig 11-39 pg 482	
	2.08	kg m^2	Iy				
Gravity Gradient							
Tg	2.86E-09	N.m					
mu	398600000000(	m^3/s^2					
R	6951000	m					Moment
theta_rad	0.00174532925	rad					
							COTS - HI
Solar Radiation			table 11.9B				
Tsp	2.77E-07	N m	pg 367				
F	2.77E-06	N					PG 362
Fs	1420	w/m^2			worst case solar flux		
c	3.00E+08	m/s^2			speed of light		redesigni
cps-cg	0.1	m					
q	0.95						moment
As	0.3	m^2					
							RW8
Assumptions	sun ray at 0 deg	incidence					
Magnetic Field							
Tm	4.74E-06	N m					
D	0.1	A m^2					
B	4.74E-05						
M	7.96E+15	tesla m^3					
R^3	3.35847E+20	m^3					
Aerodynamic							
Ta	4.64E-06	N m					
F	4.64E-05	N					
rho	1.80E-12	assumed	SMAD back of book				
cd	3						
A	0.3	m^2					
V	7572.599753	m/s					
T_total	9.66E-06	N m	total external torque				
P	5767.427634	s	table 11-12				
theta_rad	0.00174532925	rad	pg 370				
h	3.19E+01	N m s	momentum	reaction wheel			
			storage				
Using Magnetorquers for momentum dumping			cause polar				
D	1.37E-01	A*m^2					
Td	4.74E-06	N.m	looks like aerodynamic drag is the biggest one along with ma				
Td	6.16E-06	N.m	with margin of 30%				
B	4.50E-05	Tesla					
sensors must have magnetometers if using magnetroquers							
I_mag	5.491186029	kg m^2					

						page number		
	System Power	80.0364	W			from power of comms, payload, ADCS		
	P_e & P_d	112.05096	W		40	% margin of total power		
	Xe	0.6		60%		413	assume peak	
	Xd	0.8		80%		413	power tracking	
	Te	30.7				umbra+penumbra		
	Td	65.39				STK		
	Psa = (PeTe/Xe + PdTd/Xd	227.7418483	W			11-5 413		
	efficiency @ EOL	0.225	nu_eol					
	mean solar flux	1367	W/m^2					
	Po	307.575	W/m^2			SMAD pg 413		
								Jus
	Id	0.77				414		Re
	cos(theta)	0.9170600744				417		
	P_BOL = Po * Id* cos(the	217.1898593						
								Lif
	degrad.	0.0275				417		mi
Lifetime degradation	Ld = (1-degrad./year)^sat	0.9197479531				417		ou
								th
	P_EOL = P_BOL * Ld	199.7599286				11-6 417		ye
Area Required Asa	Psa/P_EOL	1.140077742	m2			11-9 417		
Area/Cell	unit are of cell	31	cm2			data sheet		
# of Cells	Asa/unit are of cell	368	cells					
Weight/Cell	unit weight per cm2 x are	2604	mg			data sheet		
Weight of Cells	weight/cell X # of cells	958272	mg					
Weight of Cells		0.958272	kg					

	cycles	Lifetime/period	3yrs/period	16403.84692		cycles	
	Period	5767.427634					
	DOD	about	25	%		421 fig 11-11	
	Time per eclipse	Te	30.7	minutes			
	Number of Batteries	N	2	redundancy			
	Battery efficiency	n	0.9	assume		Table 11-40	
	Power_eclipse	Pe	112.05096	Watts			
						STK	
req.	Battery Capacity	Pe * Te/(DOD)*N*n	76.44365493	W-hr		per battery	
	SLC-16050	Lithium-Ion Secondary Cell					
	Dimensions	0.173	0.081	0.0569	m	2.02	kg
	Specific Energy	121.3	Whr/kg				

		Operating Range			Non-operating Range			
Battery Information (Thermal		30	10	C	-5	C		
System Design Driver)		303	283	K	268	K		
Internal Waste Heat		22.410192	16.807644	Watts	0	Watts		
		about 20%	about 15%					
		of total power produced						
		Hot Case	Cold Case					
	q_solar	42.6	20.4	w/m^2		pg 109		
	k	1420	1360			is the solar constant in the vicinity of the Earth		
	F	1	1	worst case		is the time average fraction of the surface area proje		
	MLI EOL	0.1	0.05	worst case		5-mil thick silver teflon surface finish commonly used		
	As/c	0.3	0.3	m^2				
	q_albedo	21	0	w/m^2	Eq 11-18			
							eta	5.67E-08
	q_earthIR	161	148	w/m^2	table 11-48B			
	q_external	182	148					
	rad emissivity	0.8	0.8					
	q_radiator	3.34E+02					q	em*eta*T
							hot	cold
	cold side	4.18E+00				MLI	em	
q_MLI	sun side	1.25E+01				cold side	0.01	0.03
						sun side	0.03	0.01
	Q_MLI(4 faces)	7.31E+00	3 side + 1 sky					
								table11-48A
radiator area	A	0.38	m^2			Side faces	0.3	
unit mass		3.3	kg/m^2	table 11-49		Earth face	0.25	
radiator mass	m	1.25	kg			sky face	0.25	
						earth facing radiator		

COLD				
q_rad	1.72E-08			
q_mli	1.62E-09			
Q_int	93.26216077			
T_cold	265.399934	K	-7.600066024	C w/o margin
	255.399934	K	-17.60006602	C w margin
since above is lower then min operating, heater is needed.				
now checking for lower non-operating temperature				
Power at non-operating		4.08393	W	@ 15% of comms and adcs systems
Qin	80.54			Qin= Qinternal+ As/c*qsolar+A* (qalbedo+qEarthIR+qbackload)
T_cold for non-operating				
	255.8440107	K		
	-17.15598931	C w/o margin		
	-27.15598931	C w margin	< -5 C	
heater is needed ofcourse				
T_margin	5	(-5+10)		
	278	K		
Q_heater	35.82	Watts		

A	B	C	D	E
h	573000	m		
IA	80	deg		
Y_max	20000	m		
Rs	573000	m		
IFOV	0.0006091899658	deg		
X	20000	m		
Y	20000	m		
nu	40			
Zc	40	# of cross-track pixels		
Vg	7572.5282	m/s		orbital velocity
P	5767	sec		orbital period
Za	0.37862641	swath along track/sec		
Z	15.1450564	pixels/sec		
bits/location	16			
no. of channnels	4			
locations/pixel	1			
B	64	bits/pixel		
DR	969.2836096	bits/sec		
housekeeping data rate				
	1000	bits/sec	pg 330	
total data rate	1969.28361	bits/sec		
groud station network				
svalbard	689.18	sec	per orbit	
trollsat	644.54	sec	per orbit	
access/orbit	1333.72	secs	two stations	
Total data storage	11356858.58	bps/orbit	during full orbit	
	11.35685858	Mb/orbit		
downlink data rate	8515.174532	bps		

D = R(FTmax-Tinitiate)/M						
R	data rate	2000000	bps	from svalbard data rate limits		
D	total data					
F		0.8		average from SMAD		
Tinitiate		120	sec	average from SMAD	time to establish comms	
M		2	approx	margin for accounting for missed passes, station downtime		
solving for T max..						
((M*D/R) + Tinitiate)/F		164.1960732	secs			