AERO 3841:

Spacecraft Design I

Final Report

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1.0 Introduction

A weather satellite is primarily used to monitor the weather and climate of the Earth. The objective of this report to get the preliminary estimation of satellite's parameter. The report discusses orbit selection, mass, power, and link budget. The expected useful orbit lifetime of the satellite is five years, and the system should be ready in three years.

2.0 Mission Statement

2.1 Primary Mission Objectives

The primary objective of this mission is remote sensing and use as a weather satellite. The payload will contain a camera in the optical spectrum, and a series of sensors to measure atmospheric parameters (temperature, pressure, humidity, and wind speed). The satellite will store the data (image, temperature, pressure, etc.) and it will transmit the data to the master Earth station when it is in view.

2.2 Secondary Mission Objectives

The secondary objective could be use the camera for different purpose. The high-resolution data can be useful for applications such as urban planning and mapping, while the average resolution can be used for vegetation discrimination, land mapping, and natural resources management. The satellite can also help look into debris into LEO orbit as well as help other satellite to let them the danger of debris impact. It could be used as a GPS to serve a strong connection between other satellite communication.

3.0 Orbit Selection

The orbit for the mission was chosen to be LEO Sun Synchronous Orbit. LEO satellites orbit between 2,000 and 200 kilometres above the earth. An important advantage of low earth orbit satellites is that they can detect temperature and moisture structure below the cloud top due to the use of microwave instruments as they are much closer to the Earth. The Sun synchronous orbiting satellites cover the entire globe on a regular basis and provide repetitive coverage on periodic basis [1]. Some advantages of LEO orbit are:

- As it is near to the earth, LEO satellites provides better signal strength. Hence less power (about 1 watt) is needed for transmission.
- It has least propagation delay (about 10ms) compared to other orbits due to closeness to the Earth. Due to lower latency, it can be used for real time critical applications.
- It provides high elevation for polar regions of the Earth. Hence better global coverage can be achieved.

Weather and climate satellites were common to have LEO Sun Synchronous Orbit. [2]

Table 1.1 Orbital elements for the satellite

RAAN	Inclination	Eccentricity	Semi-major axis	True anomaly	Argument of perigee	Period	Period for repeating ground tracks
132.9	98.6	0.00012	7178	264.63	95.5	100.87 min	205.15 min

4.0 Spacecraft Bus

4.1 Mass and Power Budget

Mass Budget

The percent of mass budget of spacecraft was researched by looking at historical data of previous remote sensing satellites launched [2]. Payload would consist various sensors that are crucial for the mission. [3]

Table 1.2 Dry Mass Calculation

Subsystem	**Percent (%)**	**Mass (Kg)**
Payload	33.8	200
Structure and Mechanisms	18.5	109.47
Thermal	4.2	24.85
Power	23.75	140.53
TTC and C&DH	6.75	39.94
AODCS	6.5	38.46
Propulsion	6.5	38.46
Total Dry Mass	100	591.72
Total Dry Mass with Margin	Margin (25%)	739.65

Power Budget

Table 1.3 Power Budget Calculation

Subsystem	**Percent (%)**	**Power (W)**
Payload	48.5	350.00
Structure and Mechanisms	2.75	19.85
Thermal	2.25	16.24
Power	18.5	133.51
TTC and C&DH	15	108.25
AODCS	10	72.16
Propulsion	3	21.65
Total	100	**721.65**
Total with Margin	Margin (10%)	**793.82**

The percent of power budget of spacecraft was researched by looking at historical data of previous satellites launched [4]. It also falls within range given in the lecture notes. Payload taking significant amount of power (48.5%), while propulsion the lowest (3%). After taking 10% margin of total power budget 793.8 W was calculated.

4.2 Configuration

Table 1.4 Spacecraft size estimation

Characteristic	Estimate
Volume (m^3)	3.03
Linear Dimension (m)	4.00
Body Area (m^2)	11.98
Moment of Inertia ($kg - m^2$)	49.02

The payload used for this mission would a cylinder with diameter of 1.5 m and length of 1 m. Taking payload into account i.e., 1.766 m^3, there is 2.234 m^3 vacant in the satellite for other subsystem. To accommodate for subsystems, a cylindrical satellite would be a viable option.

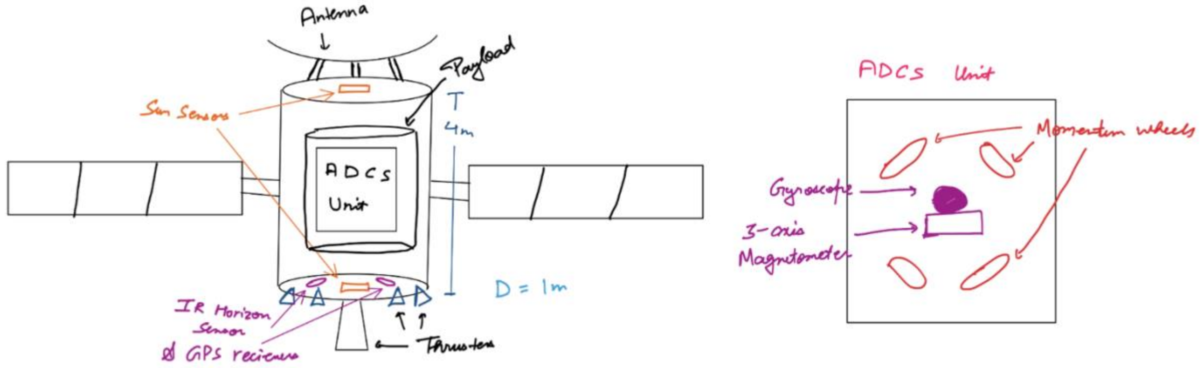

Figure 1: Configuration of satellite with payload and ADCS unit

4.3 Preliminary design of spacecraft subsystems

4.3.1 Communication subsystem

The total access time was calculated by using a satellite toolbox in MATLAB taking orbital elements and Gatineau Satellite Ground Station's latitude and longitude. The total access time was calculated to be 80 minutes. The chosen modulation was QPSK as it has good noise immunity and for the same bit error rate, the bandwidth required by QPSK is reduced to half as compared to BPSK.

The total data rate was calculated to be 496521.83 bps. Bitrate for downlink is taken as 80% of total data rate while 20% for uplink. Assumption such as line, atmospheric and attenuation loss with efficiencies were taken from textbook [5]. Losses such as rain attenuation, polarization loss, etc were not considered. Ground station parameters were taken from Gatineau Satellite Ground Station [6]. Satellite communication takes S- band frequency into account which lie in the range of 2-4 GHz [7]. Pointing error for antennas were assumed to be 1 degrees for satellite and ground station. Atmospheric loss was taken from frequency chart. Losses such as line, ground feeding and attenuation loss were taken from textbook. [5]. Calculation for uplink and downlink can be found in

Appendix. The margin for downlink and uplink was calculated to be 10.4 dB and 28.45 dB respectively.

Table 1.5 Link Budget

Parameters	Downlink	Uplink
Frequency (GHz)	2.2	2
Satellite		
Power Transmitted (W)	90	-
Dish antenna diameter (m)	0.12	0.12
Dish antenna efficiency	0.6	0.6
Noise Temperature (K)	-	300
Antenna Gain	4.56	3.78
Antenna Pointing Error (degrees)	1	1
Ground Station		
Power Transmitted (W)	-	10
Dish antenna diameter (m)	13	13
Dish antenna efficiency	0.7	0.7
Noise Temperature (K)	300	-
Antenna Gain	62785.84	51889.12
Antenna Pointing Error	1.1	1.1
Losses		
Path loss	-166.8 dB	-166 dB
Line loss [dB]	2	2
Atmospheric loss [dB]	1	1
Ground station feeding loss [dB]	2	-
Pointing Loss [dB]	22.4	0.0004
Attenuation loss [dB]	7	9
Modulation and Data		
Modulation	QPSK	QPSK
Bit Rate (bps)	397217.467	99304.367
BER	0.00001	0.00001
Bandwidth (bps)	238330.48	59582.62
Noise Ratio		
Achieved Carrier to Noise ratio [dB]	57.36	53.01
Required Carrier to Noise ratio [dB]	12.56	12.56
Margin	10.4	28.45

4.3.2 Power Subsystem

For the mission, a panel mounted solar array of Si would be used. Satellites in LEO use photovoltaic source for the power, and Si has theoretical efficiency of 20.8%, degradation of 3.75% / year and cheaper compared to other alternatives [5]. The maximum eclipse time for orbit was found to be 35.13 minutes. The bus voltage chosen for this mission is 25 VDC, and bus being regulated and DET controlled. The table below shows all the chosen parameters [5] and calculations required for power subsystem.

Table 1.6 Power Subsystem with solar panel and battery sizing

Efficiencies	X_e (Eclipse)	0.65
	X_d (Daylight)	0.85
$P_e = P_d$ (Eclipse and Daylight Power)	721.65 W	
P_{sa} (Solar Array Power)	1440.729 W	
Summer Solstice (Worst Case)		
Po	Si	195.508 W/m^2
	GaAs	244.385 W/m^2
	Multijunction	290.62 W/m^2
Inherent Degradation (I_d)	0.8	
Worst case angle (Deployable)	0^O	
P_{BOL}	156.41 W/m^2	
Fractional degradation/ year	3.75 %	
P_{EOL}	129.2 W/m^2	
Degradation for 5-year life (L_d)	0.826	
Array Size	12.27 m^2	
Mass of solar array (assuming 25 W/kg)	63.39 kg	
Number of Solar Panels		
Battery Sizing		
DOD	20%	
Battery Capacity (C_{batt})	2575.12 Wh	
Number of batteries (N)	1	
Transmission Effectiveness (n)	1.1	
Battery Mass	73.57	
P_{BOL}	1918.54 W	
P_{EOL}	1584.8 W	
Bus Voltage	25 VDC	
Mass of Harness (10% of power subsystem)	14.05 kg	
Power lost in Harness	23 W	
Total Mass	151.01 kg	

Total mass of power subsystem calculated (151.01 kg) is slightly higher compared to mass estimation (140.53 kg). If we take a margin of 10% margin for mass of power subsystem, this issue could be resolved (154.58 kg).

4.3.3 Altitude Determination and Control Subsystem

Three axis stabilization to control the torques about the axes of 3-axis systems (roll, pitch, and yaw) with the combination of reaction wheels, control moment gyroscopes, thrusters and magnetic torquers. Zero momentum strategy would be used i.e., reaction wheel on each axis for stabilization and control. It is adapted to determine from attitude signals and attitude set point signals attitude correction signals for the momentum wheels to apply to the satellite primary attitude correction torques [8].

Fine attitude control is invariably produced by reaction wheels—heavy spinning gyroscopes—grouped in packs of four to control all three axes, with a fourth spare wheel mounted skew to the others should any wheel fail. Torque impulses are produced by electric motors that increase or retard the rotation rate. [9]. Four thrusters would be on the sides of the satellite and a control moment gyroscope for stabilization. The sensors that would be used for the satellite are two axis coarse sun sensor, two IR horizon scanners, GPS receivers and three axis magnetometers. Sun sensors would be placed on both rear and front end of the satellite, IR sensors and GPS receiver at the end of satellite orienting towards earth and three axis magnetometer at the centre for stabilization and control [5].

For Nadir pointing, satellite will rotate at the same angular rate as the orbit for the satellite to always point down do earth. This would be achieved by roll-yaw coupling where satellite has inertially fixed momentum vector to orbit plane. The coupling is due to relative motion of Earth, can be used to control roll and yaw over a quarter orbit using a roll sensor [5].

4.3.4 Guidance and Navigation Subsystem

LEO orbit experiences orbital perturbations namely atmospheric drag, solar radiation pressure and earth oblateness. These perturbations affect the altitude and lead to orbit decay overtime [5]. To overcome these perturbations, orbit control is required to get to the desired orbit.

There are two general orbital control methods against the orbital perturbations, orbit maintenance and station keeping. Orbit maintenance can be done by altitude maintenance in the orbit. Navigation can be done by using actuators and sensors discussed in ADCS.

For actuators, control moment gyroscopes, thrusters and reaction wheels and GPS receivers and IR sensors would be used for sensors for the satellite orbit to maintain 5 km accuracy.

For autonomous control, communication network among all participating spacecraft allows for transferring and receiving information on position and pointing directions and to coordinate the appointment of the lead spacecraft. [9] Ground stations are surface-based facilities which are designed to provide real-time communication with satellites. The crew at these stations send radio signals to the satellite (uplink), receive data transmissions from the satellite (downlink), and in some cases, serve as command-and-control centres for the satellite network [10]. The main advantage of autonomous control is that it is safe, economical, and more efficient than ground control for satellite guidance control [5].

Absolute station keeping is best implemented autonomously because it uses many small thruster burns, rather than the small number of larger burns used in more traditional station keeping. Results of long-term simulations show that in most cases position can be controlled indefinitely to within less than 0.2 sec (1s), corresponding to an in-track position error of less than 1.5 km [11].

Autonomous control system will be used for the satellite. Therefore, evasive maneuvers would be implemented against debris impact. This is performed by having small ΔV maneuvers using actuators and sensors. To simply calculation, ΔV is approximated as 65 m/s for altitude and orbit control. The propellent mass is added in the propulsion section of the report.

To get the ΔV for deorbiting the satellite to a circular orbit of 350 km altitude, Hohmann transfer maneuverer should be carried out. The lower LEO orbit is suitable for deorbiting because within a year, the satellite would decay due to drag and other effects.

$$\Delta V_1 = \sqrt{\frac{\mu}{6378+350}} - \sqrt{\mu\left(\frac{2}{6378+808} - \frac{1}{7178}\right)} \quad = 0.253\,\frac{km}{s} = 253\,\frac{m}{s}$$

4.3.5 Thermal Subsystem

The upper and lower allowable temperatures for the mission is $40^{o}C$ and $5^{o}C$, respectively. If we take a margin of $5^{o}C$, maximum and minimum temperature would be $35^{o}C$ and $10^{o}C$.

Using a spherical spacecraft approach, taking area of $11.98\ m^2$ from spacecraft configuration.

Multi-layer insulation of ½ mil aluminium Kapton coating would be used on the spacecraft. The absorptivity and emissivity for BOL and EOL are assumed same for the coating used i.e., 0.53 and 0.67 [5].

For worst case hot and cold with all the variables are given in the table below.

Table 1.7 Input and Output for Thermal Subsystem

Input

Sphere	
Altitude (km)	800
Spacecraft surface area (m^2)	11.98
Power dissipation (W)	793.1
Emissivity	0.67
Absorptivity	0.53
Solar Flux (W/m^2)	1421
Max. Earth IR energy flux (W/m^2)	258
Min. Earth IR energy flux (W/m^2)	216
Albedo	0.35

Output

Earth angular radius (deg)	62.69
Earth albedo reflection factor	0.1496
Solar energy absorbed (W)	2255.6
Max. IR radiation absorbed (W)	442.41
Min. IR radiation absorbed (W)	370.39
Heat absorbed from Albedo (W)	674.64
Maximum equilibrium temperature (C)	34.28
Minimum equilibrium temperature (C)	-51.71
Heater Power Required (kW)	4.82

Limits without margin	
Upper temperature limit (C)	35.00
Lower temperature limit (C)	10.00

The maximum equilibrium temperature (T_H) was calculated to be $34.28^{O}C$ while minimum equilibrium temperature (T_c) to be $-51.71^{O}C$. The upper limit for the mission is satisfied while same was not true for lower limit. This is because, the spherical approach normally overestimated the heater power requirements. To satisfy this condition, a heater with a power of 4.82 kW could be added to the system.

For payload to be maintained at $20^{O}C$ with a margin of $1^{O}C$. Thermoelectric cooler would be added with payload to attain higher temperatures [12]. This suggests there would be hybrid thermal system implemented to the satellite's thermal subsystem.

4.3.6 Propulsion Subsystem

Spacecraft would include a liquid propulsion system of bipropellant (hydrazine, and an oxidiser). A network of smaller thrusters would be used to apply fine adjustment impulses and are distributed about the spacecraft to allow all combinations of forces and torques to be applied to any face or about any axis. A larger single axis thruster could be included to allow gross changes in spacecraft translational position.

PSLV and Falcon 9 are known to place satellites into LEO sun-synchronous orbit.

The mass of propellant by using rocket equation, taking bipropellant hydrazine ($I_{sp} = 290\ s$).

$$m_{prop} = m_{dry}\left[1 - \frac{1}{e^{\frac{\Delta V}{g I_{sp}}}}\right] \qquad (1)$$

From above equation with ΔV calculated earlier, we get,

$m_{prop\_deorbit} = 50.44\ kg$ and $m_{prop\_orbit\_control} = 13.36\ kg$

Table 1.8 Calculated Propellant Mass

$m_{prop\_deorbit}$	50.44 kg
$m_{prop\_orbit\_control}$	13.36 kg
ADCS propellant (5%)	2.52 kg
Margin (12.5%)	6.31 kg
Residual (2%)	1.00 kg
Total propellant mass	73.64 kg

4.3.7 Structures and Mechanism

The proposed configuration for the satellite, it would be a cylinder of volume 3.03 m^3 and area of 11.98 m^2. The payload would also be a cylinder with diameter of 1.5 m and length of 1 m.

A big thruster would be used to deploy the satellite into the orbit. The satellite would be in launcher, that would be maximum size for undeployed satellite. Once the launcher detaches the satellite and throw it in the orbit, thruster would help calibrating it into the final orbit [13].

The structure of the spacecraft must support instruments and propellant tanks and accommodate other spacecraft systems over the life cycle of the spacecraft. It must be strong enough to survive the high gravity, wide-band vibration of launch, interfacing securely with the launcher, yet still be light enough to conserve spacecraft mass. Traditionally, combinations of aluminium alloy honeycomb, carbon fibre and titanium are used to produce a stiff spacecraft skeleton [14]. Shielding will be included to protect the spacecraft from micrometeoroid and space junk impact.

The mechanism required for the satellite would be deployable solar panels, reaction wheels for ADCS, antenna for communication and thrusters for autonomous control against debris impact. Reaction wheels, thrusters and solar panels are most likely to fail during the course of the mission. To avoid this risk, redundancy with actuators and sensors can be implemented into the satellite. A extra layer of body mounted solar panel with a combination of momentum wheels with magnetic torquers. Two sets of small back up thrusters could be added to the sides for orbit control as well as debris impact. Health of each part in the satellite could be checked regularly to ensure proper functioning of the satellite.

5.0 Launcher Selection Trade Study

The launcher used to place satellites in LEO orbits would be Falcon 9 or Polar Satellite Launch Vehicle (PSLV) by Indian Space Research Organisation (ISRO) and SpaceX. These launchers are commonly used for LEO and GEO sun synchronous orbit.

The Pro and Cons for these launchers are discussed below:

Table 1.9 Pro and Cons for selected launchers

Launcher	Pro	Cons
PSLV	Cost is competitively low for satellite launch	Payload capacity of 1750 Kg at an altitude of 600 km above earth
	52 successful launches with 94% success rate	Satellites need to be carried to India for launch
Falcon 9	Cost is low due to reusable rockets	Expensive for dedicated launch
	Payload of 22,800 kg can be carried to LEO orbit	Need permission for launch i.e., 2 years

A weighted trade study can be done for the chosen launcher with 1 as worst and 3 being the best.

Table 1.10 Weighted Trade Study for selected launchers

	PSLV	Falcon 9
Cost	3	2
Performance	2	3
Launch location	2	3
Reliability	3	3
Availability	2	3
Total	12	14

As seen, Falcon 9 won the weighted trade study. It is cheap compared to its other competitors and has a higher success rate and only downside being permission to launch need to take 2 years prior. This can be accommodated as satellite for this mission should be ready in three years.

In the context of this report, dedicated launches for small satellites are those that use launch vehicles which are generally meant to be used to launch satellites with a mass less than 180 kg. The primary orbit for this type of launch is low-Earth orbit. Dedicated launch controls the mission requirements in whole- need, orbit selection, and location of launch. They generally have a readiness or stop call on launch day in case something goes wrong with their satellite pre-launch. They can also request special launch accommodations, such as a nitrogen purge or late battery charge, that are generally not available to a shared launch [15].

Until recently, there were only a few launchers that allowed shared launch. The majority of small spacecraft are carried to orbit as secondary spacecraft, using the excess launch capability of larger rockets. Standard shared launch consists of a primary mission with surplus mass, volume, and performance margins which are used by another spacecraft. On the other hand, there are downsides of shared launch. The launch date and trajectory are determined by the primary spacecraft. Generally, the secondary spacecraft are given permission to be deployed once the primary spacecraft successfully separates from the launch vehicle, but there are instances where the shared spacecraft separate prior to the primary satellite [16]. For this mission, shared launch would be feasible as it is more cost efficient, can launch multiple payloads with window for the mission is open throughout year for LEO satellites.

6.0 Conclusion

The primary objective of this mission is remote sensing and use as a weather satellite. The orbit chosen for remote sensing satellite was LEO Sun Synchronous. Total dry mass was calculated to be 739.65 kg, propellant mass being 73.64 kg. For power, total power was taken as 793.82 with 10% margin W, with payload taking 48.5% of total power. The chosen shape for satellite is cylindrical with volume and moment inertia of 3.03 m^3 and 49.02 $kg\ m^2$. Taking frequency as 2.2 and 2 GHz for downlink and uplink, margin of 10.4 dB and 28.45 dB was calculated for communication subsystem. ADCS and G&N system use actuators and sensors needed for satellite's stability, control, and orbit maintenance. Roll-yaw coupling would be used for Nadir pointing and autonomous control system would be used for evasive maneuvers against debris impact. Spherical approach used to determine thermal subsystem, multi-layer insulation of ½ mil aluminium Kapton coating was used for solar panel. The maximum equilibrium temperature (T_H) was calculated to be $34.28^O C$ while minimum equilibrium temperature (T_c) to be $-51.71^O C$, suggesting implementation of hybrid thermal system n to satisfy payload temperature margin. Liquid propulsion system of bipropellant would be used as propellant. The Pro and Cons of different launchers with weighted trade study was done and Falcon 9 was chosen. Project schedule and cost estimate are discussed in the Appendix.

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Appendix 1: Project Schedule

Project Schedule is based on NASA's Program/Project Life Cycle [17]

Design Process	**Duration**
Concept and Requirement	165 days
Preliminary design	250 days
Final Design	145 days
Fabrication	100 days
Integration Testing	100 days
Preparation for Launch	700 days
Launch and Deployment	31 days

Appendix 2: Cost Estimate

The launcher selected i.e., Falcon 9 would take about $6 million for shared launch. The cost of insurance would be about $350 million. For payload with actuators and sensors with high-definition camera, to would be around $2-3 million. The cost of the mission would be roughly around $400 million. The data was taken from already launched satellites [2] and spaceflight.com.

Appendix 3: Details of Calculations

MATLAB Code for total access time:

```
startTime = datetime(2023, 02,27 , 00, 30, 0);
stopTime = startTime + days(1);
sampleTime = 60;
sc = satelliteScenario(startTime, stopTime, sampleTime);
lat = 45.58521000466262;
lon = -75.80870895972373;
gs = groundStation(sc, lat, lon);

semiMajorAxis = 7178*1000;
eccentricity = 0.00012;
inclination = 98.6;
rightAscensionOfAscendingNode = 132.9;
argumentOfPeriapsis = 95.5;
trueAnomaly = 264.63;
sat = satellite(sc, semiMajorAxis, eccentricity, inclination, ...
    rightAscensionOfAscendingNode, argumentOfPeriapsis, trueAnomaly);

ac = access(sat, gs);
intvls = accessIntervals(ac)

play(sc)
```

```
>> Aero_Sat

intvls =

  7×8 table
```

Source	Target	IntervalNumber	StartTime	EndTime	Duration	StartOrbit	EndOrbit
"Satellite 2"	"Ground station 1"	1	27-Feb-2023 00:41:00	27-Feb-2023 00:50:00	540	1	1
"Satellite 2"	"Ground station 1"	2	27-Feb-2023 02:17:00	27-Feb-2023 02:31:00	840	2	2
"Satellite 2"	"Ground station 1"	3	27-Feb-2023 03:58:00	27-Feb-2023 04:11:00	780	3	3
"Satellite 2"	"Ground station 1"	4	27-Feb-2023 14:29:00	27-Feb-2023 14:41:00	720	9	9
"Satellite 2"	"Ground station 1"	5	27-Feb-2023 16:09:00	27-Feb-2023 16:23:00	840	10	10
"Satellite 2"	"Ground station 1"	6	27-Feb-2023 17:49:00	27-Feb-2023 18:00:00	660	11	11
"Satellite 2"	"Ground station 1"	7	28-Feb-2023 00:16:00	28-Feb-2023 00:23:00	420	15	15

Calculation of Period of repeating ground tracks:

$$sin\ \rho = \frac{R_E}{R_E + h} = \frac{6378}{6378 + 808} = 0.89$$

$cos\ \delta = \cos 10^o = 0.984$

By using $sin\ \rho \cdot cos\ \delta = \sin \eta$ to find η

$\eta = 61.2^o = 1.07\ rad$

By geometry, $\lambda - 180^o - 90^o - 10^o - 61.2^o = 18.8^o = 0.328$

Sample access time can be found:

$$Access\ time_1 = \frac{0.328}{\pi}(102) = 10.6\ min$$

According to Satellite toolbox, there are 7 revolutions within 1 sideral day. Therefore, period would be:

$P = 1 \cdot \frac{1436.07}{7} = 205.15\ min$

Calculation for total data rate:

Downlink data rate = $\frac{No.of\ bits}{Total\ access\ time\ (sec)}\ 24 * 60$

= 397217.4667 bps

Taking ratio of downlink as 80% and uplink as 20%,

Uplink Data rate = $\frac{0.2 \cdot (397217.4667)}{0.8}$ = 99304.36667 bps

Total Data Rate = Downlink + Uplink Data Rate = 496521.8333 bps

Calculation for Margin and Losses for Downlink:

Worst case distance for uplink/downlink:

$$d = R_E\left(\sqrt{\left(\frac{R+h}{R_E}\right)^2 - (\cos\delta)^2} - sin\delta\right)$$

$$d = 6378\left(\sqrt{\left(\frac{6378+808}{6378}\right)^2 - (\cos 10)^2} - sin\ 10\right) = 2382.7\ km$$

Bandwidth for Downlink

$B(QPSK) = 0.6R$

= 0.6 (39717.47) = 238330.48

Taking BER as 10^{-5},

$$\frac{E_b}{N} = \frac{1}{B}\ln\left(\frac{1}{2BER}\right)$$

$$= \frac{1}{238330.48}\ln\left(\frac{1}{2 * 10^{-5}}\right) = 45.4 \cdot 10^{-6}$$

$$\left(\frac{C}{N}\right)_{req} = R\ \left(\frac{E_b}{N}\right) = \frac{R}{B}\ln\left(\frac{1}{2BER}\right)$$

$$\left(\frac{C}{N}\right)_{req} = 397217.467 \cdot 45.4 \cdot 10^{-6} = 18.033$$

$$\left[\frac{C}{N}\right]_{achieved} = 10\log(18.033)\ dB = 12.56\ dB$$

Antenna Gains:

$$G_t = \eta\left(\frac{\pi D_t}{\lambda}\right)^2 = 0.6\left(\frac{\pi\ 0.12}{0.14}\right)^2 = 4.58$$

$$G_r = \eta\left(\frac{\pi D_r}{\lambda}\right)^2 = 0.7\left(\frac{\pi\ 13}{0.14}\right)^2 = 62785.84$$

Achieved carrier to noise ratio:

$$\frac{P_r}{N} = \left(\frac{C}{N}\right)_{achieved} = \frac{P_t G_t G_r}{kTB}\left(\frac{\lambda}{4\pi d}\right)^2$$

$$= \frac{90\ (4.58)(62785.84)}{1.38*10^{-23}(300)(238330.48)}\left(\frac{0.14}{4\ (3.14)(2383.7*10^3)}\right)^2 = 544770.66$$

$$\left[\frac{C}{N}\right]_{achieved} = 10\log(544770.66)\ dB = 57.36\ dB$$

<u>Calculation for losses and margin:</u>

$$Path\ loss\ (dB) = 32.5 + 20(\ \log(d) + \log(f)\ )$$

$$= 32.5 + 20(\ \log(2383.74) + \log(2200)\ = -166.83\ dB$$

Antenna Pointing loss:

$$\theta_{3dB} = \frac{70^o \lambda}{D}$$

$$= \frac{70^o(0.14)}{13} = 19.03^o$$

$$[L_{point}] = 12\left(\frac{\varepsilon}{\theta_{3dB}}\right)^2$$

$$= 12\left(\frac{1.1}{19.03}\right)^2 = 27dB$$

$$Margin = \left[\frac{C}{N}\right]_{achieved} - \left[\frac{C}{N}\right]_{req} - losses$$

$$= 57.36 - 12.56 - 22.4 - (2 + 1 - 2 + 7)dB = 10.4\ dB$$

<u>For thermal subsystem</u>, spherical spacecraft approach was used:

$$\mathrm{G_s}\ \left(\frac{A_s}{4}\right)\alpha_s + \left(\dot{q_E \sin^2\rho}\right)A_{sc}\left(\frac{1-\cos\rho}{2}\right)\varepsilon_{IR} + G_s A_{sc}\ a\ \alpha_s\ \left(\frac{1-\cos\rho}{2}\right)\sin^2\rho + \dot{Q_w} = \varepsilon_{IR} A_{sc}\sigma T^4$$

Parameters used are in the Table 1.7