Power subsystem microsatellite

This subsystem is responsible for supplying power to the entire satellite, converting solar cell energy to on-board battery energy, and distributing power to various other subsystems.

The power subsystem of the microsatellite is designed for a remote sensing mission to carry out on sun-synchronous orbits at 700 km altitude at an inclination of 98. 19 degrees. The payload of the microsatellite includes a multispectral remote sensing camera which takes picture of polar region in a visible spectrum and a surrey GPS receiver especially designed for low earth orbit. Microsatellite payload weighs 5 kg and with a mean power consumption of 9W. Sub-system power budget is estimated according to the payload power requirement with 15 percent margin. Total estimated power requirement for the microsatellite is 70W.

Microsatellite subsystem Power Allocation : -		
End Of Life Estimated Microsatellite Power –		70 W

Subsystem	% of Operating Power	Power (W)
Payload	15	10.5
LSTS Bus		
Propulsion	0	0
Thermal Control	10	7
Attitude Control	15	10.5
Power	15	10.5

Communications	20	14
C & D Handling	10	7
Structure	0	0
Margin	15	10.5
Total	100	70

The power subsystem of the microsatellite is designed for Low Earth Orbit for five years period. The power estimated for subsystem has a 15% contingency margin. Primary power source for the satellite is the solar array that is body mounted on the microsatellite. The satellite is in near polar sun-synchronous orbit at an altitude of 700 km, total orbital period of the satellite is 98. 77 min. The microsatellite experience eclipse for about 35. 29 min. Solar array for the microsatellite is designed according to the mission requirement. Batteries are secondary power source during the eclipse when no sun light is available. The selection of the solar cell and batteries are made according to power required end of life of the satellite and trade study between different solar cell and batteries but decision is made to satisfy the estimated mass size and power budget of the satellite. As the satellite is a cube shaped and spins stabilized body mounted solar panels to places on all the four sides of the satellite for a uninterrupted supply of power to the subsystems.

Altitude	700.00	km
Earth's Radius	6,378.00	km
Total Power Requirement (const. day and night)	70.00	Watts
Earth's Gravitational Constant	3,98,600.00	km^2/s^2

Power transfer efficiencies:-		
Xd	0. 85	
Xe	0. 75	
Inherent Degradation Id	0. 80	
Worst Case(θ deg)	23 . 00	(de g)
Mission Life (yrs)	5. 00	(Yr s)
Life time Degradation (Ld)	0. 98	
Angle α (rad)	1.	(ra

	12	d)
Angle α (deg)	64 . 30	(de g)
Orbital Peroid (P) (sec)	5, 92 6. 21	(se c)
Maximum Eclipse Peroid (Tn) (sec)	2, 11 7. 08	(se c)
Minimum Power Sunlight (Td) (sec)	3, 80 9. 12	(Se c)
Average Solar array power (Psa) (W)	13 4. 23	W

Multijunction Solar (GainP/GaAs) Po	301.00	W/m^2
BOL Power (Pbol)	221.66	W/m^2
EOL Power Requirement (Peol)	216.17	W/m^2
Solar array Area (m^2)	0.62	m^2
Mass of Solar Array (kg)	3.36	kg
Solar array weight (body mounted so Msa x 4)	13.42	kg

The primary power source of the microsatellite is chosen to be Multijunction Solar cells (GainP/GaAs). These solar cells have an efficiency of 23 percent and most advanced for their category. The required solar panel area of the microsatellite to sufficiently support the power requirement of the microsatellite subsystem is 0. 62 m^2 but for a body mounted microsatellite, all the four faces of the cube shaped satellite will have the following area. The estimated weight of the solar panels is 3. 4 kg and the total weight of all the panels on the satellite is 13. 5 kg. The main advantage of the body mounted solar panels is such that they have more life expectancy as they are not exposed to radiation for a long time, but it is compensated with the additional weight of the solar panels. The primary power source should be able to generate 135 Watts of power to sustain the power requirement of the subsystems as well as enough to charge the batteries as they are the secondary power source of the mission.

For Given Ni H cell –		
Assuming Data for 700 km altitude		
Energy Density	100. 00	W. h/k

		g
DOD	1. 60 – 0. 27 log10 [cycles]	
Power during Eclipse	70. 00	W
Altitude	700. 00	km
Battery Voltage	28. 00	Vol ts
Xb-l	0. 90	
No . of eclipes per day	15. 00	
	5	

	year Mission	
Orbital Peroid (P) (sec)	5, 926. 21	Sec
Time of Night (Tn) (sec)	2, 117. 08	Sec
Eb (energy supplied during eclipse) (W. h)	45. 74	W. h
Cycles	26, 607. 25	
Depth Of Discharge	0. 41	

(DOD)		
1a)Ebcap (energy battery capacity required) (W. h)	112. 87	W. h
1b) Battery Capacity (A. h)(assuming voltage is 28 v)	4. 03	A. h
2. Total Battery Mass (kg)	1. 13	Kg

The secondary power source is required to generate power during eclipse in the orbit to sustain microsatellite subsystems. The secondary power source for the mission is chosen to be NiH batteries as they are good for long cycle life and they have advantage of mass and volume over most of the current batteries available. They have good specific energy density of 50 W. hr/kg. The main advantage of these batteries is such that they are widely used in space mission and constantly updated with new technologies. They have depth of discharge of 40% that is good for this kind of mission. Total secondary power source weight is 2. 3 kg.

(((((((((References SMAD and System Integration Aegis))))))))

Communication subsystem

The communications subsystem is the lead for the interface between the satellites and the ground stations. The communications subsystem helps in demodulating the received uplink signals and transmitting downlink signals . The subsystem also helps us to maintain a track over the satellite by transmitting received range tones and by acting as logic between receive and transmitted signals.

Data Rate

The remote sensing microsatellite is designed for a Low Earth orbit at an altitude of 700 km. The payload of the satellite is a multispectral camera that takes picture of the poles in visible spectrum. The 20 degrees minimum elevation angle and a resolution of 50 is assumed for the satellite and the data rate is calculated for the satellite.

Altitude (km)	700. 00
Radius of Earth (km)	6378. 14
Orbit Peroid	98. 77

(mins)	
Ground Velocity (km/s)	6. 76
Node Shift (ΔL = S) (deg)	24. 76
ε (deg)	20. 00
η (deg)	57. 86
Zc	27818. 52
Za	133. 06
Z	3701467. 63
DR (Visible) (bps)	37014676. 33
Maximum Time in	6.

View (min)	66

The data rate calculated is 37Mbps adding 10 percent margin data required to send back to ground station is estimated to 40Mbps.

Band Link Technology

For the current microsatellite mission an S-Band telecommunication system is researched, analyzed, and chosen as the best system for establishing communication between satellite and the ground station.

Application	Specifications
Downlink Rate Max	2. 5Mbps
Power RF Output	. 4W
Power Consumption	3. 4W
Weight	420g

Volume	190X135X2 2 mm3

The table above shows the specification if the Surrey Satellite S band communication system transmitter details. This has an advantage of low mass, power and data rate which completely satisfy the mission constrains. The above transmitter system also has a S-Band antenna for this transmitter which has specifications as follows.(((((((((((((((((((memo com2 // surrey satellite))))

	Specifications
Number of Antennas Needed	4
3dB Beamwidth	± 35°
Weight	80g
Volume	82X82X20 mm3

Link Budget

Link budget for the system S band communication system is designed considering the factor to transmitting the data rate of 40Mps within 6. 5mins or 400 sec.

The link budget is a process of accounting all the possible gains and losses during transmitting and receiving the signals from transmitter to receiver.

The equations below are used to determine link budget:

Total spacecraft received power (uplink budget)

Uplink Signal to Noise ratio (Will help determine probability of bit error)

Total Ground Station received Power (downlink budget)

Downlink Signal to Noise ratio (Will help determine probability of bit error)

2. 4. 1 Slant Range

The Slant range was calculated as follows for a 5 degree elevation angle.

2. 4. 2 Attenuation of the Signal

The biggest contributor to the attenuation of the signal is free space loss. There are many other losses such as cable loss, polarization loss, cloud, rain, etc.

The frequency used for the S-Band calculation is 2. 2GHz.

Atmospheric loss is caused by absorption due to such factors as oxygen and water vapor in the atmosphere. Atmospheric, rain, clouds and ionosphere scintillation were assumed to be 0. 5dB for 2. 2GHz. Further investigation into these effects needs to be completed next semester. With X-Band the total loss due to these factors was calculated to be 0. 76dB. S-Band is expected to have a much lower loss.

Polarization loss was estimated from [9]

2. 4. 3 Calculating EIRP

There will be passive losses in the equipment such as losses in the coax cables. This number was used from the previous year.

Power transmitted was obtained from the specification on the Surrey transmitter as 0. 4 Watts.

Looking at the Co-Polar gain on Figure 2 it is seen that there is a gain of at least 0dB for angles between +/- 70°.

2. 4. 4 Ground Station Antenna Gain

Using an antenna that is 4. 5m in diameter with efficiency of 0. 55 the gain is calculated as follows:

2. 4. 5 Signal to Noise Calculation

The signal to noise ratio will determine the Bit Error Rate (BER), as determined from the following graph [8].

From this graph it can be seen that to obtain a Bit Error Rate of 10^{-5} which is typical of space missions, a signal to noise ratio of 4. 4 dB is needed.

The Link Budget calculations will determine if the system will meet the 4. 4 dB of signal to noise ratio at the ground station.

System Noise is a function of temperature and was determined from table 13-25 [2].

4. 8dB is above the minimum 4. 4dB theoretical signal to noise ratio required. This leaves only a 0. 4dB margin which needs to be approved upon. The output RF power could easily be increased from 0. 4Watts by using an amplifier, but would be at the expense of the satellite power budget. The Surrey Satellite equipment is a viable solution.

Thermal Subsystem

The thermal control subsystem is the integral part of the satellite design. It helps out all the components that are exposed to thermal environment are not affected badly. Thermal control subsystem accomplish safe working of all the satellite subsystems and their components by constituting a thermal model. The following process includes inputs from different subsystem of the satellite by identifying the thermal loads that will acting on them during the mission lifetime as well as their operating tempertature for the smooth running of the mission.

Thermal Loads

The satellite experience or exposed to thermal enviroment during gound testing, transportation, launch , orbit transfer and operational orbits. The thermal environment concerned is during its operation in space. There are four main loads acts on the satellite during its mission.(smad)

- Direct Solar Radiation: The main source of direct solar radiation is the Sun. It is major source of environmental heating on the satellite, it is a stable energy source and it constant to the fraction of second. The intensity of the sunlight on the earth's mean distance of 1 Astronomical unit (AU) is 1367 W/m^{2}.
- Earth's Albedo: Albedo is the reflected sunlight reflected from earth . It is highly as it is the fraction of incident sunlight that is refected back to space. Refletivity increases over land rather than in oceans. Reflectitivy increases with decreasing local solar -elevation angle.
- Earth's Infrared Energy: It is also refereed as blackbody radiation, all incident sunlight do not reflected back as abledo rather earth absorbs it and re-emit it as IR (infrared Energy) or blackbody radiation.
- Free Molecular heating: This load is result of the bombardment of the individual molecules present in outer reaches of the atmosphere. It affects during the launch ascent of the satellite.

The thermal control susbsystem is designed for a sun synchronous Low Earth Orbit at an altitutde of 700km and at an inclination of 98. 19 degrees. The main aspect in designing the thermal control system is to first define the worst case hot (maximum loads) and worst case cold (minimum loads) acting on the satellite in the orbit and the opertonal temperature operational and survival temperature of each component installted