Designing a Small Satellite in LEO for Remote Sensing Application

Hasan Murtaza
Pakistan Space and Upper Atmosphere Research Commission,
Satellite Research and Development Centre, Lahore, Pakistan
Email: hasanmuskaan@gmail.com

Abstract—Small Satellites are becoming increasingly popular due to their low cost, minimized volume and reduced design and development time yet fulfilling all the ne9cessary objectives of a space mission. This paper presents the design of a small satellite using COTS, with 55 Kg of weight in Low earth orbit for remote sensing application. The satellite executes an orbit phase change of 90 degree in four weeks. The suitable COTS components are also identified with the respective manufacturer information as well. At the end, the overall Power, mass and volume budget is calculated satisfying the pre-defined requirements and objectives.

Index Terms— COTS, Remote Sensing, Small Satellites, Space Mission

1 Introduction

Small satellites are playing a very important role in the field of remote sensing, navigation and surveillance. The current design is of a small satellite for remote sensing purposes, transmitting the data to a ground station located at surrey space centre. All of the subsystems are designed with necessary calculations. At the end, the budget of power, mass and dimension is calculated.

2 Satellite Design Process

The satellite design process follows the below mentioned steps;

1. Satellite orbit design
2. Payload design
3. Propulsion system design
4. Telemetry and Telecommand link
5. Video data transmit unit (VTU)
6. On-board computer
7. Thermal control system design
8. AODCS Design
9. Power system designs

The necessary design formulae are taken from Ref.1. The detail of the design of all these systems is explained next.

2.1 Satellite orbit design

Satellite Tool Kit (STK) is used to conduct the satellite orbit analysis. A circular satellite is created at an altitude of 550 Km in the scenario and ground facility is created at Surrey Space Centre (SSC).

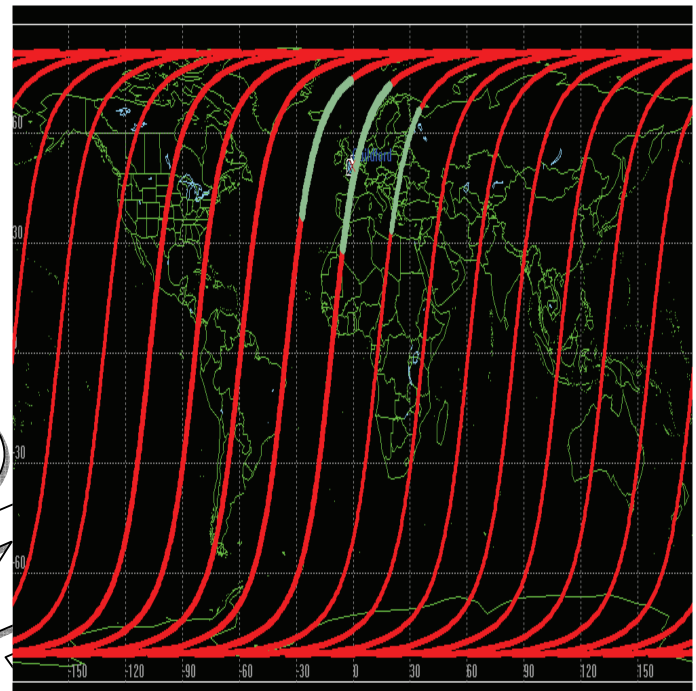

Figure1 - Satellite footprints

The times when the satellite accesses the Ground station at Surrey Space Centre, Guildford, Surrey, are calculated as given in table 1.

2.2 Satellite payload design

The payload system on the satellite is a multispectral imager operating in three wavelength ranges, green, red and NIR.

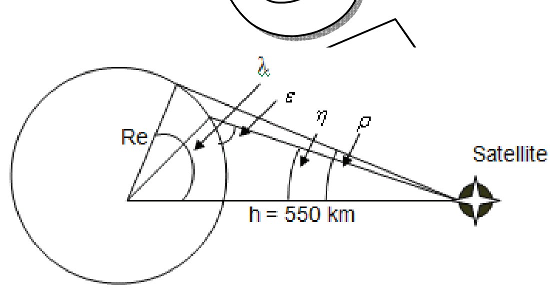

Figure 2: Satellite orbit geometry

Table 1 - Satellite access times

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
1	1 Jun 2009 12:34:35.413	1 Jun 2009 12:46:50.810	735.397
2	1 Jun 2009 14:09:57.721	1 Jun 2009 14:19:51.413	593.692
3	1 Jun 2009 21:51:23.151	1 Jun 2009 22:01:36.614	613.463
4	1 Jun 2009 23:24:38.903	1 Jun 2009 23:36:54.242	735.340
5	2 Jun 2009 01:01:41.437	2 Jun 2009 01:10:51.321	549.884
6	2 Jun 2009 10:57:10.603	2 Jun 2009 11:06:36.206	565.604
7	2 Jun 2009 12:31:12.577	2 Jun 2009 12:43:28.131	735.554
8	2 Jun 2009 14:06:32.581	2 Jun 2009 14:16:36.154	603.573
9	2 Jun 2009 21:48:07.572	2 Jun 2009 21:58:11.599	604.028
10	2 Jun 2009 23:21:15.942	2 Jun 2009 23:33:31.495	735.552
11	3 Jun 2009 00:58:08.336	3 Jun 2009 01:07:33.254	564.918
12	3 Jun 2009 10:53:52.525	3 Jun 2009 11:03:03.128	550.603
13	3 Jun 2009 12:27:49.824	3 Jun 2009 12:40:05.187	735.363
14	3 Jun 2009 14:03:07.552	3 Jun 2009 14:13:20.597	613.045
15	3 Jun 2009 21:44:52.291	3 Jun 2009 21:54:46.471	594.180
16	3 Jun 2009 23:17:53.250	3 Jun 2009 23:30:08.662	735.412
17	4 Jun 2009 00:54:35.818	4 Jun 2009 01:04:14.816	578.998
18	4 Jun 2009 10:50:34.856	4 Jun 2009 10:59:29.424	534.568
19	4 Jun 2009 12:24:27.155	4 Jun 2009 12:36:41.976	734.821
20	4 Jun 2009 13:59:42.630	4 Jun 2009 14:10:04.749	622.119
21	4 Jun 2009 21:41:37.311	4 Jun 2009 21:51:21.226	583.915
22	4 Jun 2009 23:14:30.821	4 Jun 2009 23:26:45.749	734.928
23	5 Jun 2009 00:51:03.847	5 Jun 2009 01:00:56.044	592.196
24	5 Jun 2009 10:47:17.645	5 Jun 2009 10:55:55.042	517.398

The calculations of data rates, orbit and sensor optics is carried out as explained below.

Orbital period, $P = 1.658669 \times 10^{(-4)} \times (6378.14 + h)^{(3/2)}$
$= 95.64$ min $=5738.4$ sec $\rightarrow$ 15 orbits/day

Ground track velocity, $Vg = (2\pi Re)/P = 6.98$ km/s

Node shift, $\Delta L = (P/1436) \times 360 = 23.97$ deg

Incidence angle, IA = 55 deg

Nadir angle, η= arcsin ((cos (ε)) x (sin(ρ))) = 48.94 deg

one way Swath width, λ=90- η – ε = 6.0 deg

Swath width, SW=2x λ = 12 deg

These equations imply that the sensor onboard the satellite will have to swing back and forth through an angle of ±48.94 deg to cover the swath. The swath width on the ground will be 12 deg wide in Earth-central angle, with a maximum distance to the far edge of the swath of 1293 km.

Max. Along track ground sampling distance =50m

Aperture =105 mm

Ground Resolution of band 1=2.44 x h x1000 x λ_1/aperture
= 7.5152m

Ground Resolution of band 2= 2.44x h x1000x λ_2/aperture
= 8.72m

Ground Resolution of band 3= 2.44xhx1000x λ_3/aperture
= 10.73m

Where λ_1, λ_2, λ_3 are the wavelengths of blue, green and NIR respectively.

IFOV = (Y_max/(Rsx1000))x(180/ π)
= 0.0032 deg

Calculating No.of pixels recorded in 1sec

Z= Zc x Za
$= 6.89 \times 10^6$

B=12; 'B' is no. of bits used to encode each pixel (bits)

Data rate, DR = ZxB = 82 Mbps

Diffraction-limited aperture diameter, Diff_D= (2.44 λ_1 x f x Q)/d = 0.0836 m

F-number of optics, F= f/ Diff_D = 6.69

Field of view of optical system, FOV = IFOV x Nm = 11.10^0

Where f is the focal length.

The approach of sizing the current payload is that by comparing it with an existing similar system. For this purpose, a Micro satellite Imager with the dimensions of 600 mm x 600 mm x 710 mm, 119 Kg mass, 170 W power and 400 mm aperture is selected.

First computing aperture ratio (R);

R= 0.105/0.400 = 0.262

Size = (600 x 600 x 710) x 0.262 = 157 mm x 157 mm x 186 mm

Weight = 2 x 119 x (0.262) ^3 = 4.28 Kg

Power = 2 x 170 x (0.262) ^3 = 6.11 W

All of the parameters calculated above are summarized in the table 2.

According to the requirement, the Mass memory unit is required to store 10 square images.

This results in size of Memory unit Bytes = 11.14×10^{10} bits /8=13.92 GB,

Now for storing 13.92 GB of data with the data rate of 82 Mbps, SSTL product High speed data recorder is used with 16GB Mass storage.

Table 2 - Satellite payload design parameters

Parameter	Value	Parameter	Value
Altitude, h	550 Km	Size	157mmx 157mmx186 mm
Inclination, i	97.59	Weight, W	4.28 Kg
Swath width	12 deg	Power, P	6.11 W
Nadir angle range, η	48.94 deg	Data rate, DR	82 Mbps
Min. Elevation angle, ε	20	Diff. limited Diameter, D	0.0836 m
Instrument	Visible & NIR	IFOV	0.0032 deg
Ground resolution, Visible	7.5m	F No.	6.69
Ground resolution, NIR	10.73m	FOV	11.10 deg
Aperture	105 mm		

2.3 Satellite Propulsion system design

The total propellant utilization for this satellite is for producing;

1) ΔV for phase change is ΔV_1 and propellant mass for this ΔV_1 is Mp_1

2) ΔV for balancing the effect of drag force is ΔV_2 and propellant mass required is Mp_2

3) Margin, Mp_3

These parameters are calculated as;

The ΔV_1 required for phasing is calculated as;

ΔV_1= (4 x r x $\Delta\Phi$)/ (3 x t) = 12.0579 m/sec (Ref. 2)

$Mp_1 = Mf\,[e^{(\Delta V_1/Ispxg)} - 1] = 0.9$ Kg

$\Delta V_{rev} = (\pi \times Cd \times A \times \rho \times a \times v)/m = 16.45$ m/sec

$Mp_2 = Mf\,[e^{(\Delta V_2 / Ispxg)} - 1] = 1.33$ Kg/5 years

Total propellant, $Mpt = Mp_1 + Mp_2 = 0.9 + 1.33 = 2.23$ Kg

Taking the margin, $Mp_3 = 15\%\ (Mpt) = 0.15 \times 2.23 = 0.34$ Kg

All of the parameters calculated above are summarized in the table 3.

Table 3 - Propulsion system design parameters

Parameters	values	Parameters	values
Mp for phasing, Kg	0.9	Tank mass, Kg	0.264
Mp for Drag force effect avoidance, Kg/5 years	1.74	Total propulsion system mass, Kg	2.90
Mp for margin, Kg	0.345	Tank material	Titanium
Total propellant required, Kg	2.64		

2.4 Telemetry and Telecommand link

The TT&C subsystem comprises of onboard VHF Receiver and UHF Transmitter for TM/ TC. A VHF receiver is used to receive Telecommand signal from ground station. Block diagram of VHF receiver is shown in Figure 3. A monopole antenna with 0dB gain receives signal from ground station. Signal is then passed through amplifier, down converter and filter stage. Finally, FSK Demodulator demodulates the Telecommand signal and generates bit stream. That bit stream is then given to command decoder for further processing. The link budget formulae are taken from Ref. 5.

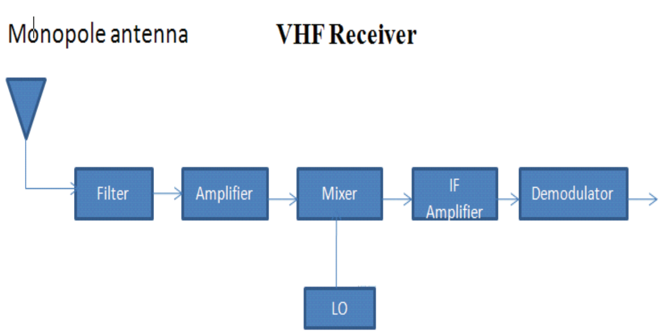

Figure 3 – VHF receiver

A UHF transmitter is used to transmit telemetry data to ground station. Block diagram of UHF transmitter is shown in Figure 4. FSK modulator at 19 kHz modulates the data and then after frequency mixing, amplification and filtering, signal is transmitted to ground by using a monopole antenna with 0 dB gain.

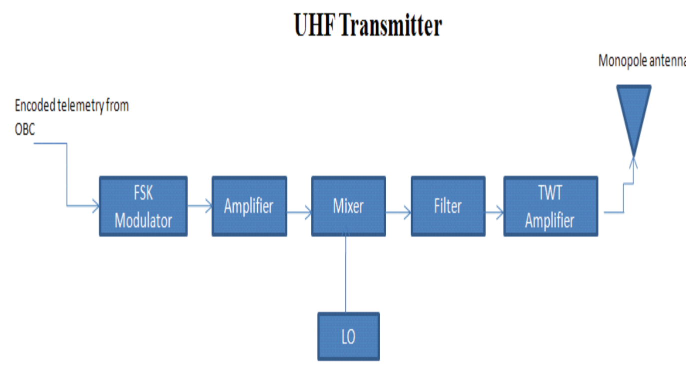

Figure 4 – UHF Transmitter

The link budgets for Telecommand uplink and telemetry downlink are performed using the formulae in ref.5 and listed in tables 4 and 5 respectively.

Table 4 - Link budget for Telecommand uplink

Parameter	Value	Parameter	Value
Frequency, f	142 MHz	Receive antenna gain, Gr	0 dBi
BER	10 -6	Data Rate	9600 bps
Eb/No (required)	14.5 dB	(Eb /No) calculated	34.25 dB
Transmit Antenna gain, Gt	11.2 dBi	Link margin	19.75 dB
Power transmitted, Pt	9 dBW	Transmitter feeder loss, Lft	1 dB

Table 5 - Link budget for telemetry downlink

Parameter	Value	Parameter	Value
F	435 MHz	Receive antenna gain, Gr	12.1 dBi
BER	10 -6	Data Rate	19kbps
Eb/No (required)	14.5 dB	(Eb /No) Total	29.5 dB
Transmit Antenna gain, Gt	0 dBi	Link margin	15 dB
Power transmitted, Pt	2.5 dBW	Transmitter feeder loss, Lft	1 dB

2.5 Designing Imaging data transmitting unit

S-band transmitter transmits video payload data to ground station. Block diagram of S-band transmitter is shown in Figure 5. Data captured and stored by payload section is transmitted to ground station during eclipse period. S-band transmitter consists of QPSK modulator, limiter, mixer, frequency oscillator, filters and final stage amplifier. Helix antenna is finally used for transmission of data to ground station. Wave shaping is usually done in an onboard FPGA. Output power level of transmitter is determined by performing link budget analysis.

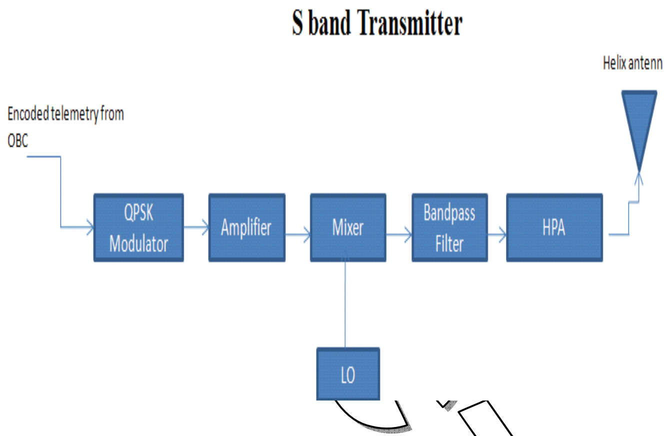

Figure 5 - S band transmitter

2.6 Designing AODCS subsystem

The AODCS is designed for the following requirements;

1) In-track pointing of the butane thrusters during burns
2) Pointing accuracy to targets during imaging < 0.1 deg.
3) Maximum off-track roll-off during imaging +-30 deg.
4) Keeping sun normal to the solar panel
5) External torque rejection
6) Pointing the imager towards nadir in eclipse times

The given data is,
Satellite's mass properties:
Mass of Satellite = M = 55 Kg, Dimensions, x=0.6 m, y= 0.6 m, z=0.4 m
Mass Moment of Inertia = MOI along x-axis = M $(y_2 + z_2)/12$ = 2.38 kgm^2
Mass Moment of Inertia = MOI along y-axis = M $(x_2 + z_2)/12$ = 2.38 kgm^2
Mass Moment of Inertia = MOI along z-axis = M $(y_2 + x_2)/12$ = 3.30 kgm^2

Misalignment of thrust vector results in acting of torque on satellite as offset of vector and CoM works as moment arm of F (thrust). This torque acts as a disturbance torque and a counter torque must be applied on the satellite to keep the thruster aligned.
Maximum misaligned thrust vector, d = 0.05 m
F thrust= 0.1 N
Torque due to thruster misalignment, T = F x d= 5×10^{-3} Nm = 5 m Nm
To counter for the effect of this disturbing torque, AODCS should provide opposite amount of torque for appointing the thrusters accurately.

Momentum, H=T. tb
Tb is the burn time, assuming it 1sec, H=5 mNm x 1=5 mNm.

AODCS is responsible to keep solar panel normal to Sun when imaging is inactive. As well be explained in power section, the incidence angle of the sun to the satellite is taken as 20 degrees at its worst case scenario. So the main purpose of the AODCS will be to point the satellite solar panels normal to the sun, i.e. to maneuver the incidence angle to zero degree. Taking the slew rate equal to 20deg/1 min, the torque is calculated as;
T = 9.29 x 10^-4 Nm, H = 0.056 Nms
Same amount of torque can be used to reorient the satellite towards Nadir when it enters eclipse.

Satellite is required to perform 30^0 roll slew in 60 seconds.
$T = 4\times\theta\ Ix/t^2$
Torque required for 30^0 slew in 60 sec = $1.4\times10^{-3}\ Nm$
H = T×t = 0.08 Nms
Where H is momentum stored in the Reaction wheel during slew.

The external disturbing torques are calculated as;

1) Gravity Gradient is calculated to be T gravity, off-track = 1.4×10^{-6} Nm
2) Solar Radiation Torque, Tsp=2.4×10^{-7} Nm
3) Magnetic Torque, Tmagnetic = 4.8×10^{-5} Nm
4) Aerodynamic Disturbance, Ta = 5.7×10^{-7} Nm

From all the four disturbance torques, Maximum disturbance torque is of Magnetic torque, $T_{magnetic} = 4.8\times10^{-5}$ Nm.
Maximum momentum storage for disturbance can be estimated by using the following formula if maximum disturbance torque is experienced due to magnetic field;
$H = T_{magnetic}$ x Orbital period x 0.707/4 = 0.05 Nms

Table 6 - Design parameters of the Transmitting unit

Parameter	Value	Parameter	Value
Transmit frequency, f	2250MHz	Receive antenna gain, Gr	38.2 dBi
BER	10 -6	Data Rate, R	60 Mbps
Eb/No (required)	11 dB	(Eb /No) Total	19 dB
Transmit Antenna gain, Gt	13 dBi	Link margin	8 dB
Power transmitted, Pt	3.1 dBW	Transmitter feeder loss, Lft	1 dB
Transmit antenna Diameter, Dt	30 cm	Receive antenna Diameter, Dr	4.5m

3-axis stabilization and control is a suitable technique for such a satellite. 3-axis technique with zero momentum bias is used to control the attitude.

Three orthogonal Reaction Wheels are used as actuators. Each reaction wheel controls single axis attitude of satellite. Reaction wheel can be sized with two parameters, Torque generation and Momentum storage. In above calculations, it is obvious that maximum torque is required for In-track pointing of thrusters and that is $T = 5\times10^{-3}$ Nm. Moreover, maximum storage capacity required is H = 0.08 Nms.

SSTL's small satellite RW (Microwheel) can provide up to 10×10^{-3} Nm torque and momentum storage of 0.42 Nms. Therefore, it is a suitable choice for this satellite. Margins provided by this reaction wheel are;

Torque Margin = 5 mNm

Momentum Storage Margin = 0.34 Nms

Dumping of momentum stored in Reaction wheel can be carried out through magnetic torquers.

Maximum stored momentum = H = 0.08 Nms. assuming 1 minutes for dumping the extra momentum. of RW, the area required for the reaction wheel can be calculated as below.

D = 29.2 Am^2

SSTL Magnetorquer (30 Am^2) can be used to serve the purpose.

The sensors are;

1. Three Gyros as external torque generators
2. Three Sun sensors
3. One Star sensor

3-axis attitude knowledge better than 0.1^0 is required in this satellite. For inertial pointing, sun and star sensors are the most suitable choice because these sensors can locate the sun in external environment. Star sensor helps to meet the pointing accuracy of $< 0.1^0$.

2.7 Power system Design

The power system is using a single body mounted GaAs solar panel as its primary power source on the largest body facet of the satellite. The satellite will be sun tracking between imaging maneuvers to maximize the power.

The battery is of 3.6 Volt, 1.5Ah Li-ion cells (size AA). The nominal bus voltage is 14.4Volt.

Average power required during daylight and eclipse is 52W, the eclipse duration is calculated to be 35.61 minutes then the total amount of power produced by the solar arrays is calculated to be 116.4024W.

The power of each solar panel is 58.2012W.

For GaAs solar cell, the ideal solar cell output performance per unit area with the sun normal to the surface of cells P_O is 252.8950W / m^2. Inherent degradation of GaAs solar panel is 0.77 and the degradation per year is 2.75%, the sun incident angle is taken to 15 degrees in worse case. With these values, the power at BOL is calculated to be P_BOL = 185.1984W / m^2 and at end of life is P_EOL = 182.6659W / m^2. Total solar array area then is 0.6372m^2.

Area of each solar panel is 0.3186m^2 with the mass of solar array = Ma = (0.04) x (Psa) = 4.66 kg.

The Li-Ion Secondary battery is used to offer the power during eclipse length of 15 eclipses per day, the DOD of this battery is 20%, the nominal bus voltage = Vbus = 14.4Volt. The total battery capacity is calculated to be Cr = 42.8639W - hr or 2.9767A – hr, the mass of battery is calculated to be 0.262 Kg=262g

2.8 OBC Design

The on-board computer (OBC) is also called command and data handling unit (C&DU). It receives, validates, decodes and distributes commands to other systems and gathers, processes and formats housekeeping and mission data for downlink. According to the given requirements, integrated system architecture is implemented in the OBC. An integrated OBC integrates the command, housekeeping and attitude control system. This type of system architecture provides a reduced hardware requirement and cost due to the increased capability provided by the processor.

The On-board computer is based on a **Hitachi Super-H4 RISC** processor with **8 M byte** of **FLASH** and **8 M byte** of **EDAC** protected SRAM. The average power for the OBC is 3 watt and it weighs 0.5 kg.

For redundancy, it has dual controller area network (CAN) bus to communicate with all the subsystems for TT & TC, AODCS sensors and actuators at 1 Mbps. A module can be commanded to change to the secondary bus or it will automatically switch buses in case the primary bus fails and the module does not receive any messages for a specific period.

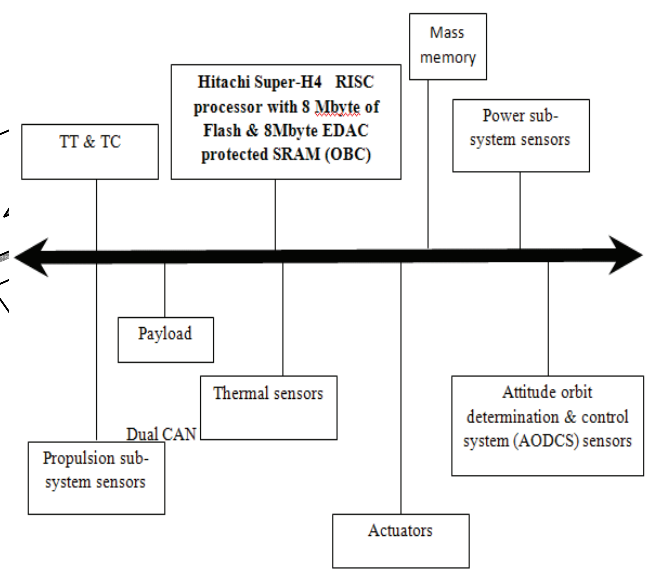

Figure 6 - OBC Architecture

The OBC controls all the devices in the satellite, as it would direct the command not addressed to it to the corresponding devices or executes the command addressed to it, such as switching, telemetry, and store/read data.

As can be seen from the figure 6, different subsystems are connected with the on-board computer. The bus architecture (distributed) is chosen among the other available architectures such as centralized architecture and ring architecture. The bus architecture is more flexible and can add new components later on in the design. This architecture is usually a parallel network of wires. The bus architecture is highly reliable because multiple processing units can be used to execute software as needed. This system architecture provides a high level of redundancy. Because it supports multiple processors in case if one processor fails the other processor will take its place and do the rest of the operation.

2.9 Thermal control system design

The task of the thermal design process is to ensure that all components operate within appropriate temperature bounds given all the possible thermal environments that the satellite may encounter. Some of the most demanding are the payload, the fuel (hydrazine) and the processor electronics. The ideal internal operating temperature range for the batteries and all the electronics is 0 degree C to 30 degree C.
In thermal design there are two main thermal control techniques active and passive. Keeping in view the requirements a passive thermal control technique is selected. The selected thermal control technique is the most efficient solution with respect to cost, power, weight and reliability. Thermal control hardware includes thermal blankets, Multi Layer Insulation (MLI), second surface mirrors (SSM) radiators and heat pipes. Interface fillers, coatings, and insulating washers are also employed at unit interfaces.
The main body of the satellite is insulated from the space environment with multi-layer insulation (MLI) and provides radiator areas with low solar absorptance and high infrared emittance to reject unwanted heat. Using the radiators that have a low solar absorpitivity (alpha), high infrared emissivity (e) to reject waste heat. Using the MLI to prevent heat transfer into and out of the satellite, and the subsequent large variation in internal temperature.

3 CONCLUSION

The current design of the small satellite is very useful for remote sensing application. Mass, Power and volume calculated here satisfies all of the requirements set for the design. The margin can be added at the development stage of the satellite.

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Mass, Power and Volume Budget

	Mass (Kg)	Power (W)	Volume (mm)
Attitude control system	**9.78**	**21.5**	
Reaction Wheels	2.88	10.5	(100 x 100 x 90) x 3
Magnetic Torquers	3.6	3	(378 x 74 x 49) x 3
Star sensor	1.66	8.5	190 x 130 x 45 74 x 95 x 105 150 x 150 x 185
Gyros	1.2	5	(500x100x 100) x 3
Sun sensors	0.9	0.5	(90 x 107 x 35) x 3
OBC	**0.5**	**3**	
Communication	**5.3**	**32.05**	
S-band Transmitter	0.420	6.25	190 x135 x220
S-band HPA	1.0	24	350x130 x190
VHF receiver	0.5	0.85	190 x130 x 220
UHF transmitter	0.5	0.85	190 x130 x 220
Helix Antenna	1.8		L = 250, D = 300
Monopole Antenna	0.5 x 2=1		
Payload	**5.28**	**21.11**	
Camera	4.28	6.11	157 x 157 x 186
MMU	1	15	300x150x30
Propulsion subsystem	**9**	**20**	440 x 440 x 140
Total propellant required	2.64		
Tank mass	4.5		
Heater	2	10	
Power	**5.71**		
Solar panels	4.66	**116.40**	
Battery	1.05	42.8639W - hr	
Harness, connectors etc	**4.0**		
Balance mass	**1.5**		
Interface satellite/Launcher	**4.0**		
Margin	**6**		
Total Mass	**51.07**		

Total Power demand= 97.6W
Solar panel generation= 116 (in worst case)

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