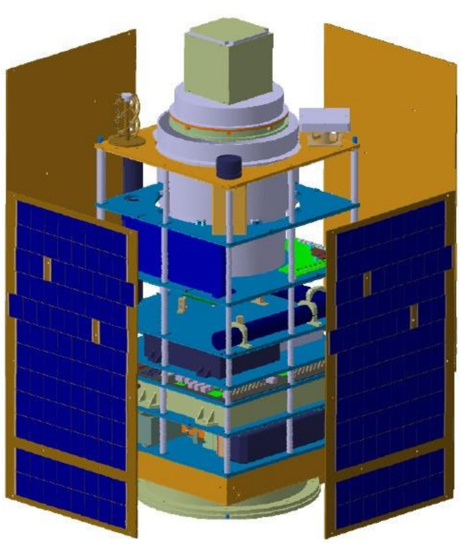
Student Small Satellite (SSS-1) System Specification Analysis (Assignment B-4.2)

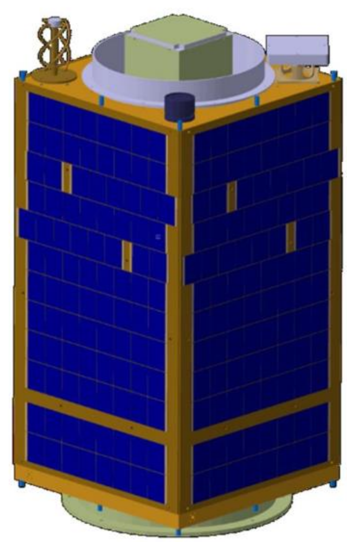
Team Bangladesh

By: MD. Sadman Siraj [1], Sheikh Fariha Hossain [2]

Department of Electrical and Electronic Engineering
University of Dhaka, Bangladesh

Assignment Supervisor: Associate Professor Chen Pei, BUAA





Contents:

Abstract:

Microsatellites, weighing between 10 to 100 kilograms and complementing the concept of "piggybacking", is fast becoming the crucial medium of choice for universities and research institutes for scientific experimentation, remote sensing/imaging and new technology demonstration; as well as being the focal point for students to study and develop space technologies due to the low-costs, lightweight and simple features associated with the platform. In this system specification analysis report, we shall qualitatively and quantitatively discuss the following topics pertaining to the APSCO SSS-1 microsatellite in light of the Feasibility Study report:

- Preliminary analysis of payload camera (in light of the FS report).
- Calculation of optical resolution and imaging integration time.
- Attitude stability requirements (in light of payload camera).
- Requirements for other subsystems (in light of payload requirements).
- Trade off (among different optical resolution configurations).

Background:

Asia-Pacific Space Cooperation Organization (APSCO) has initiated the Student Small Satellite (SSS) project to develop a small satellite constellation containing a microsatellite and two nanosatellites. The aim of the project is to train students and faculties from Member States (MSs) of APSCO on satellite engineering and space engineering via hands-on experience, leading to the development of the flight model through contribution of space education in APSCO MSs.

The SSS project was approved by APSCO council meeting as a basic activity in 2015, with Beihang University as the project leader. The Feasibility Study (FS) report was discussed by representatives of all MSs in the kick-off meeting in December 2016. The SSS-1 contract was signed with Beihang University in January 2017. The Project requirement Review (PRR) meeting was held in May 2017.

According to the requirements of APSCO Council, the SSS project shall involve the cooperation of universities, institutes and companies from all MSs, while enabling the participants to gain insights into:

- Satellite System Design and System Engineering.
- Satellite Payload, Subsystem Design and Development.
- Satellite Assembly, Integration and Launching (AIT).
- Satellite Launch and Operation.
- Satellite Payload Data Processing and Application.

1st summer camp of the APSCO SSS Project

Preliminary Analysis of the Payload Camera:

Table I outlines the Technical Specifications of Payload Camera:

Image Parameters	
Product Model	SIIS - 1
Sensor Type	CMOS
Shutter	Global Shutter
Pixel Size	5.5um X 5.5um
Noise	13e-
DR	55dB
SNR	45dB
Data Interface	Local Bus
Video Format	1080p 360fps
Color	Mono/Color
Lens Parameters	
Focus	75mm
Aperture	F2.0
Lens Size	1”
Image Processing	
Compressing Format	H264
Compressing Rate	1:4 – 1:10
ISP	Auto Exposure
Image Record	32MB
Image Output Interface	CAN 9600bps
Control Interface	CAN 9600bps
Mechanical/Environmental Parameters	
Weight	500g
Size	
Voltage	12VDC
Power	5W
Working Temperature	-40°C ~ +65°C
Storage Temperature	-50°C ~ +90°C
Humidity Environment	90% @ +60°C

Optical Resolution and Other Calculations:

Angular Resolution:

$$\Theta = 1.22\frac{\lambda}{D}$$

where Θ = angular resolution

λ = average wavelength of light = 555 x 10^{-9}m

D = diameter of the lens = F2.0 = Focal Length/2 = 0.075/2m

Ground Resolution:

$$d = \frac{\mathrm{WD}}{f}Pixel\_size$$

where d = ground resolution

WD = work distance = 500000m

Pixel_size = 0.0000055m = 5.5um

Ground Sampling Distance (GSD):

$$GSD = \frac{H}{f}d$$

where GSD = ground sampling distance

H = orbit altitude f = focal

length of the lens

d = pixel size of the charge-coupled device (CCD)

Ground Resolving Distance (GRD):

$$GRD = 1.22 \frac{H * c}{D * \nu}$$

where GSD = ground resolving distance
H = orbit altitude = 500000m

D = diameter of the lens = F2.0 = Focal Length/2 = 0.075/2m

c = speed of light = 3 x 10^8 ms^{-1}

ν = average frequency of light wave = 5.045 x10^{14} Hz

Field of View (FOV):

$$FOV = 2arctan\left(\frac{W}{2H}\right) = 2arctan\left(\frac{n * d}{2f}\right)$$

where FOV = field of vision W

= swath width

H = orbit altitude n

= number of pixels

d = pixel size of the picture

f = focal length of the lens

Imaging Integration Time:

Signal to Noise Ratio (SNR):

$$SNR = \frac{N_s}{N_n}$$

where N_s = Number of electrons collected N_n

= Noise value = 1620e^-/pixel

Now, the standard value of SNR ≥ 39 and N_n = 1620. Therefore, Ns = 39 X 1620 = 63180.

Extraterrestrial Spectral Irradiance:

$$Es = \frac{2 * \rho * E_{s0} * r^2 * f(\beta)}{3d^2}$$

where ρ = diffuse reflectance = 0.7

$E_{3,}$ = Extraterrestrial Solar Spectral Irradiance = 890.26W/m^2 r

= Target area equivalent sectional radius = GSD x (4000/2)m

$f(\beta) = (\pi - \beta)\cos(\beta) + \sin(\beta)$; β = 5degree = π/36rad ; $f(\beta) = 3.12986$

d = observational distance = orbit radius = 500000m However,

irradiance varies with date too. Hence,

$$Es' = \frac{Es}{\left[1 + 0.034\cos\left\{\frac{2\pi(d_n - 3)}{365}\right\}\right]}$$

d_n = date number in a year for example: January 1 is 1

Integration Time:

$$t_{int} = \frac{Ns * N_{pixel} * h * \nu}{E'_s * S_{aperture} * Q_e}$$

N_{pixel} = Target Resolution = Number of pixels of picture = 4000 X 4000

h = Planck's constant = 6.626 X 10^{-34} Js

ν = Average Frequency of light wave = 5.045 X 10^{14} Hz

E'_s = Extraterrestrial Spectral Irradiance from target area

$S_{aperture}$ = aperture area of the lens = (π x D)^2/16; D = Diameter of lens

Q_e = Quantum efficiency (Range: 26%-45%, Ideal: 35%; 0.35)

t_{int} = Imaging Integration Time

Simulink Model:

The Simulink model to calculate Optical Resolution & Integration Time:

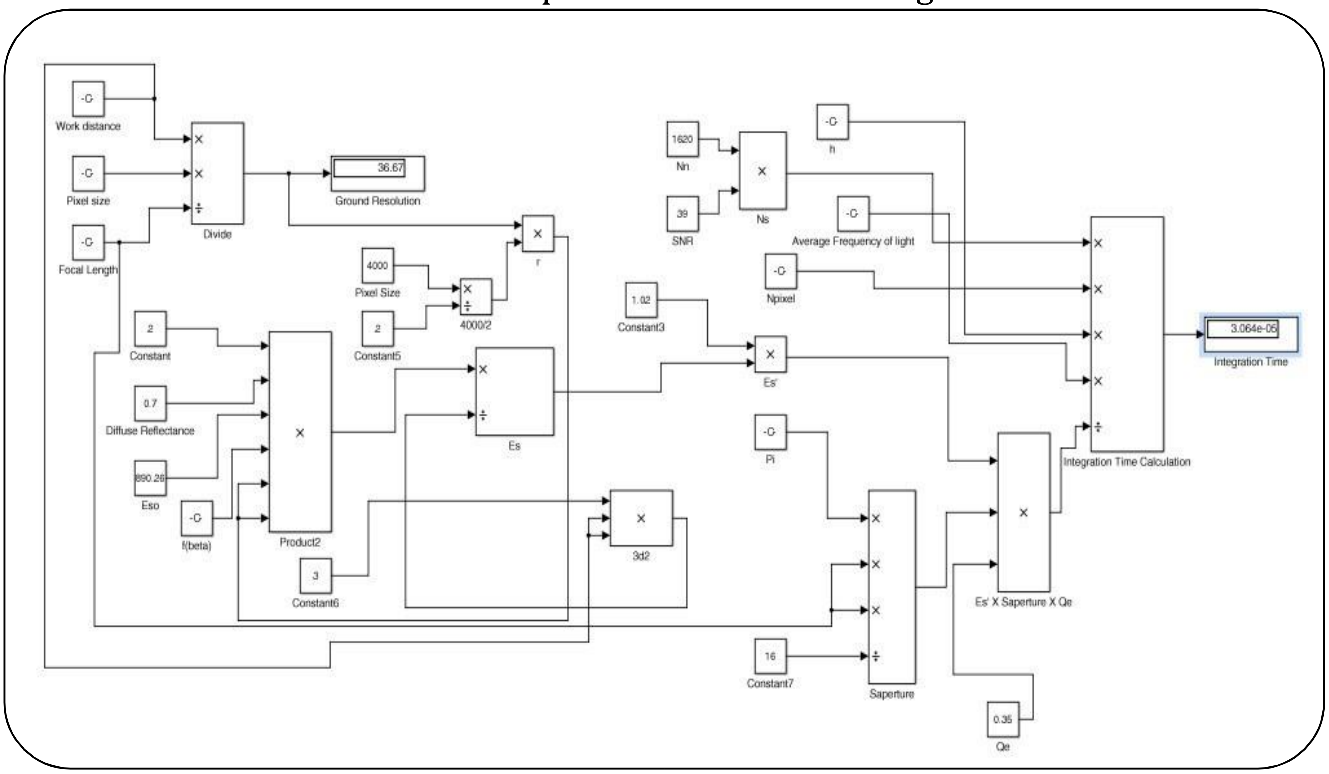


Figure 1: Simulink model to determine optical resolution and imaging integration tine.

Several other parameters for varying altitude (160-2000 km) are shown below:

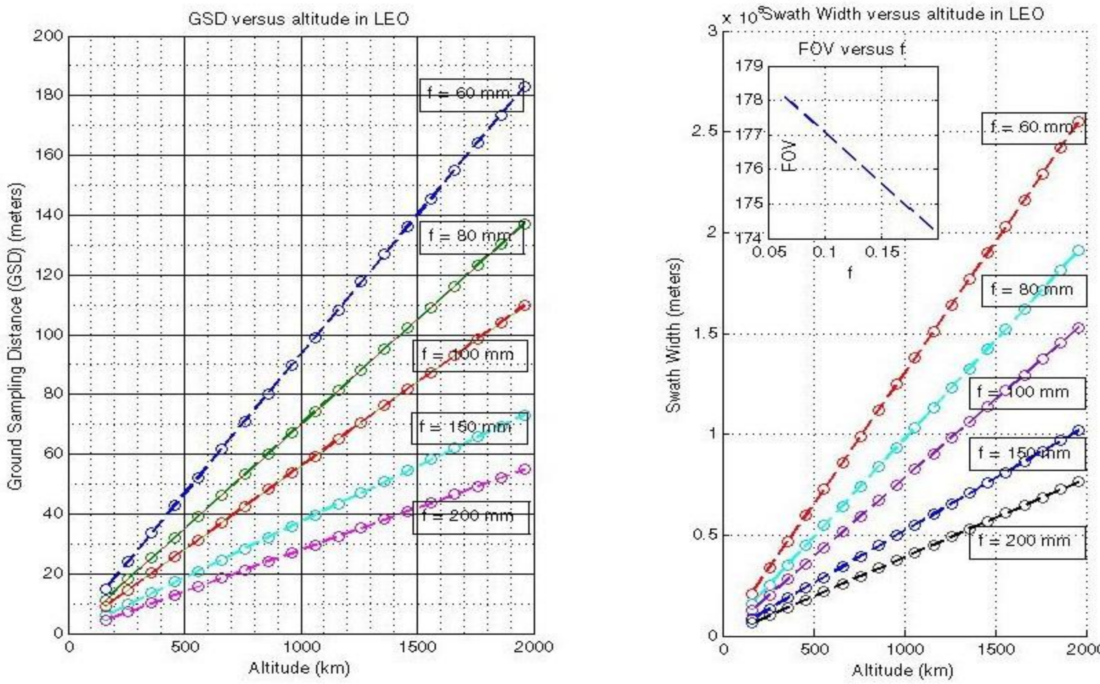


Figure 2: (Left) GSD versus Altitude for SSS-1. The focal length of the lens have been kept within 60-200 mm. (Right) Swath Width versus Altitude for SSS-1. (Inset) Field of View of the camera versus focal length.

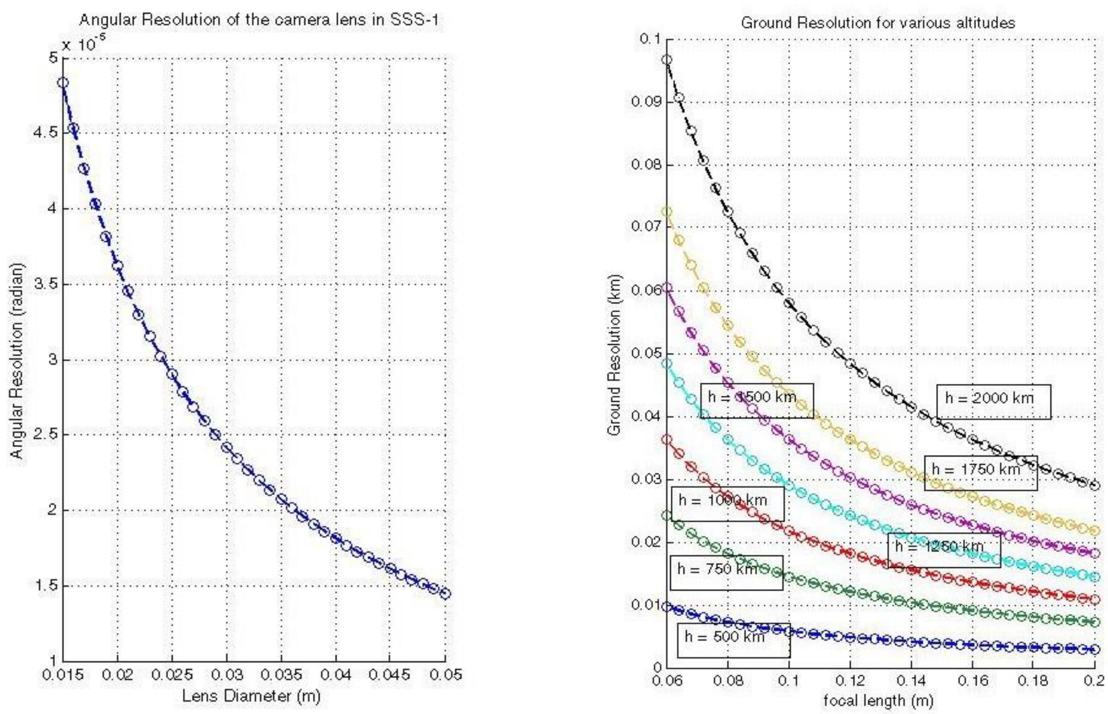


Figure 3: (Left) Angular Resolution versus Lens Diameter for SSS-1. (Right) Ground Resolution of the lens of SSS-1.

Attitude Stability Requirements:

The Attitude Determination & Control Subsystem (ADCS) configurations basing on pointing stability of the satellite is given below:

Pointing Stability (°/s)	Control Mode	Power Consumption	Cost
Greater than 5	Gravity Gradient	No	Lowest
1 ~ 5	Self-Spin	No	Low
0.1 ~ 1	Dual Spin Tri-Axis	Mid	Mid
0.01 ~ 0.1	Tri-Axis	Mid	High
Less than 0.01	Tri-Axis	High	Highest

- The ADCS must be able to keep the satellite three-axis stabilized.
- Nominal pointing attitude attained by the ADCS: Nadir pointing.
- The ADCS must provide pointing control accuracy of 0.1-1 degrees and pointing stability of better than 0.01 degrees/second.
- The ADCS must ensure sun-avoidance of payload camera.
- The ADCS must provide sufficient information to the OBDH and GS to define orbital position, attitude information, reconstruction of attitude and determination of attitude control loop characteristics.
- The ADCS shall supply sufficient data on health and operation through telemetry.
- Mass of the ADCS should be within 5 kilograms.
- Power Budget of the ADCS should be within:
    - 5W (long-term)
    - 8W (short term)
- The ADCS should be compatible with the inertia matrices shown below:

$$A = \begin{bmatrix} 1.147 & -0.0010 & 0.0010 \\ -0.0010 & 1.1360 & -0.0070 \\ 0.0010 & -0.0070 & 0.5330 \end{bmatrix} kg.m^2$$

$$B = \begin{bmatrix} 7.6590 & -0.0020 & 0.0380 \\ -0.0020 & 7.6490 & -0.0170 \\ 0.0380 & -0.0170 & 0.5330 \end{bmatrix} kg.m^2$$

Matrix B exists only when the coilable mast has been deployed.

- ADCS must be able to run in the following modes:
    - Three axes stabilized
    - Attitude Determination Accuracy; Angular error: 0.5 degrees (3**σ**) with an angular velocity error of 0.001 degrees/sec (3**σ**).
    - Attitude Control Accuracy; Angular error: 1 degrees (3**σ**) with an angular velocity error of 0.01 degrees/sec (3**σ**).

- Technical Specifications of the ADCS to meet design requirements:

Equipment	Notes
Sun Sensor	• 2 axis • FOV: ±60 degrees • Accuracy in FOV: 0.3 degrees • Accuracy on zero degrees: 0.05 degrees • P_{avg}: 36 mW • Supply Voltage: 5V • Dimensions: 30 * 30 * 30 mm • Mass: 25 grams
3-axis MEMS gyro	• Type: ADIS16355 • Gyro Measurement Range: ±75, ±150, ±300 degrees/sec (14-bit resolution) • Accelerometer Measurement Range: ±10g (14-bit resolution) (350 Hz bandwidth)
3 axis magnetometer	• Range: ± 2 Gauss in three axis • Resolution: 67 microGauss • Sampling Speed; 10-157 /sec • BCD, binary or ASCII coded output @ 9600 or 19200 bps
BD/GPS Receiver and Antenna	• 2D accuracy: < 7m • Precision: <10m • Speed precision: 0.01 m/s • Operating Range: -45 to +85 degrees Celsius • Supply Voltage: 5V • Dimensions: 96 * 96 * 825 mm • Mass: 100 grams
3 axis magnetorquer	• Magnetic Moment; $12Am^2$ • Linearity: ±5% • Residual Moment: <0.001 Am^2 • P_{avg} = 4W from 9-12.5V supply • Random vibration: 15 g rms • Overall diameter: < 30 mm • Length: 25 cm • Mass: 3 kg • Operating Range: -35 to +75 degrees Celsius
Momentum Wheel	• Angular Momentum: 0.34 Nms @ 7800 rpm • Nominal rotation speed: 6000 rpm • Maximum rotation speed: 7800 rpm • Nominal torque: 0.015 Nm

Requirements of Other Subsystems:

The **general requirements** of the SSS-1 satellite as outlined in the FS report are:

- System Lifetime: 6 months in space 1 year on-ground shelf life.
- Orbit: Sun-synchronous orbit @ 500 km.
- Dimensions: 350 X 350 X 650 mm.
  Mass: 30 kg max.
- Average power generated from solar panels: 18W. Payload Power: 30% of P_{avg}.
- Attitude Control: Tri-axis stabilized with Nadir Pointing Accuracy of 0.1-1 degrees and Point Stability better than 0.1 degrees/second.
- Communication: Telemetry: UHF/VHF, Payload Data: S-band @ > 500 kbps.
- Primary Payload: Visible Band Video Imaging Payload Camera with GSD better than 100 meters @ 500 km.
- Experimental Payload: ADS-B receiver, deployable coilable mast, Communication Experiment (Inter-Satellite Link component) and Scientific Experiment (radiation dosimeter).

From the general requirements and interested expressed by all MSs and approved in the FS report, the SSS Project covers the following four areas of space technology and application:

- Remote sensing.
- In-orbit new technology demonstration.
- Space scientific experiments.

Ground station network for operating a satellite constellation for a joint mission.

Structure Subsystem:

- Must be a plate-beam structure
- Total Mass: < 8.4 kg
- Lateral Natural Frequency > 25 Hz
- Longitudinal Natural Frequency > 45 Hz
- Response of Linear Static Analysis < 2 mm
- Fulfill the strength and stiffness of other subsystems or equipment
- The structure shall provide attachment for and support all other satellite systems on ground, during launch and in orbit under all natural and induced environments.
- The structure shall support the alignment of payload, satellite sensors, actuators and antennae.
- The structure shall be designed to withstand the static and dynamic loads induced by the launch vehicle.
- Mounting interfaces shall allow for easy maintenance, mounting and dismounting.
- The layout of the structure shall provide sufficient accessibility to facilitate integration, removal and maintenance activities.

- Handling and transportation interfaces shall permit ground handling of the integrated satellite and its units.
- The minimum main mode frequencies of the satellite hard-mounted at the launcher interface shall be higher than the required frequencies by the launch service provider.
- Spacecraft design shall ensure the survival of the structure under the worst feasible combination of mechanical and thermal loads for the complete mission life of satellite.
- Thermal mismatch between structural members shall be minimized such that stresses generated in the specified temperature range for the item are acceptable.
- External surfaces of the spacecraft shall have conductive grounding elements.
- Finite Element Methods (FEM) and physical tests must be performed to ensure the thermal and structural integrity.
- Technical specifications of the structure to meet design requirements:

<table>
<tr><th colspan="2">Parameter</th><th>Values</th></tr>
<tr><td colspan="2">Mass</td><td>5.0 kg</td></tr>
<tr><td colspan="2">Dimension</td><td>320 mm X 320 mm X 320 mm</td></tr>
<tr><td rowspan="3">Natural Frequency</td><td>1st Lateral Frequency</td><td>27.259 Hz</td></tr>
<tr><td>2nd Lateral Frequency</td><td>27.837 Hz</td></tr>
<tr><td>1st Longitudinal Frequency</td><td>70.487 Hz</td></tr>
</table>

Electrical Power Subsystem (EPS)

- Should store, regulate and distributes electrical power at load requirements to the payload/instruments at all mission phases.
- Must be capable of providing any transient or peak power including fault clearing/isolation and transient loads.
- Capable of autonomous revival on sunlight availability if the whole system shuts down; work in emergency mode.
- Solar panels should be the main source of power. They should be body-mounted and attachable to the external faces of the satellite.
- EOL Margin of panels should be 10% of predicted power.
- Solar Panels: power generation. EPS: Power Management and Transmission. Power Storage: batteries.
- Secondary power source must be a Li-Ion Battery.
- The battery must have a DOD of less than 30% to ensure sufficient power is supplied during worst case eclipse load and operational load.
- Incorporate charging mechanisms on the EPS board through umbilical connectors.
- Power distribution should be made in accordance to static and dynamic load/interface requirements as well as characteristics of power/energy sources.

- Power bus overvoltage condition must be limited as much as possible to avoid permanent damage of any components or the satellite.
- Critical systems shall remain operational and perform normally during overvoltage conditions.
- Critical units shall remain operational and shall meet performance requirements under overvoltage conditions.

- The non-critical loads shall automatically shut down as per the defined shutdown sequence when the bus voltage decreases below the normal operational range. Such under-voltage switch-off shall ensure that these loads do not operate under conditions which would damage or permanently degrade their performance, reliability or life.
- Critical units shall remain connected to the electrical bus. Such equipment shall be designed to provide full performance capability of essential functions down to a bus voltage at least 85 percent below the nominal bus voltage and shall not be damaged or stressed by any continuously applied voltage down to zero (0) volts.
- The structure shall not be used as intentional DC or AC current path. It shall only serve as a ground reference and to provide shielding against emitted electromagnetic fields and against fields externally generated.
- Prevent overcharge and over-discharge; act as central hub for control, regulation and distribution.
- Electrical or magnetic interference, including switching transients, shall not affect the performance of the power distribution subsystem or any other satellite subsystem throughout.
- Maximum Power Generation: >= 18W, Least: 7W.
- The EPS shall accept supply from external sources during ground operations.
- Technical specifications of the EPS to meet design requirements:

Equipment	Quantity	Notes	Model Used
Solar Panels	4	• Triple Junction GaAs solar cells • Mass: <= 1.85 kg • Power: 36 (EOL) to 48 (BOL) W • Optimal Voltage: 9 – 12.5V • Installation: Body Mounted • Efficiency >= 28%	FM
PCU	1	• Mass: <= 2.5 kg • Operating Temperature: -10 to 45 degrees Celsius • Power Consumption: <= 2W • Bus Voltages: Output: 5.2V @ <= 150 mV P-P ripple.	EM, FM

Batteries	1	• Type: Li-Ion • Operating Temperature: 10 to 30 degrees Celsius • Nominal Capacity: 9-12.5V • Nominal Capacity: 10 Ah • Maximum DOD: 30%	EM, FM
Kill Switches	3	• Mass: <= 0.15 kg • Load Capacity: 3A • Operating Temperatures: <55-110 degrees Celsius	EM, FM
Harness	1 set	• Mass <= 0.15 kg • Lod Capacity: 3A	EM, PM

On-board Data Handling Subsystem (OBDH): • Be the central processing unit (command/control) in the following situations:

- ○ Receive, execute and distribute uplink commands, including parameters setting, code upload, TLE/time update, power supply control and mission control.
- ○ Telemetry and payload data collection, packing and downlink management. ○ Thermal Control via temperature sensors and heat film interfaces. ○ Payload mission control and data storage. ○ Housekeeping tasks such as time keeping, calibration and OBE management.
- ○ Monitoring heath state and handling system faults.

- Pass the reliability screening test.
- Implement component degrading design for lifetime and reliability improvement.
- Implement self-testing.

<u>Both of the above two steps must be done in accordance with the Chinese Aerospace Standards.</u>

- Apply error detection and correction (EDAC) methods to mitigate single event upset (SEU) effects on SRAM.
- Use overcurrent protection circuits to ameliorate the effects of Single Event Latch-Up (SEL).
- Technical Specifications of the OBDH to meet design requirements:

Parameter	Value
Computer Organization	Two On-Board Computers (OBC) running in parallel and hardwired via an arbitration unit. One OBC is a backup for the other, and vice versa.
Processor Architecture	32 Bit ARM Cortex 7 RISC CPU @ 40 MHz
Operating System	Micro-COS-II

Memory Architecture	2 MB SRAM with EDAC, 2 MB NOR Flash (Program instructions storage), 2 GB SD Card (Data storage)
Power Supply Voltage	• 5V±0.3V @ 0.5A for minimal operation • 9-12V @ 1A for driving peripherals
Power Consumption	< 2W
Interfacing	• Two independent Control Area Network (CAN) bus • I2C interface • RS422 interface • SPI interface • 24 channel 12 bit ADC (Reference Voltage: 4.5V) • 3.3V digital outputs (40) • 3.3V digital inputs (16) • 32 channel 10 bit ADC (temperature measurement) • 5 channel heater power supply (9-12V @ 0.3A)
Timer	RTC Module calibrated by PPS of GPS receiver
Mass	< 1 kg
Dimensions (mm)	150 * 150 *30
Operating Temperature	-20 to 50 degrees Celsius
Lifetime	> 12 months @ 600 km
Component Assurance	Industrial Components and flight tested

Thermal Control Subsystem:

- Provide necessary thermal environment for optimal operation of all systems and instruments.
- Passive TC.
- Use sufficient sensors for temperature monitoring.
- Thermal flux minimization in all areas of the satellite.
- Operational and design temperature margin should be within 10 degrees Celsius uncertainty (5 degrees Celsius for battery).
- Total mass of the thermal control subsystem should be less than 3% of the satellite.
- Maximum Power Consumption: 7W, supplied by 9 – 12.5V bus.
- Primary Control: Passive, Secondary Control: Active.
- Needs to be flight qualified.
- Must carry out:
  - Isothermal Design
  - Enhanced Thermal Analysis
- Easily replaceable and repairable in the AIT stage,
- Application of thermal control coating, 20-layer MLI and Electronic Film Heaters.

- Technical Specifications of the Thermal Control Subsystem to meet design requirements:

Parameter	**Value/Notes**
Control Methods	3595 (63,3%)
Thermal Components	Aluminized Mylar and Nylon Multi-Layer Insulation (MLI), Electronic Film Heater, Temperature Sensors (11 MF501 Sensors); TS mounted on solar panels (4), battery (2), EPS (1), SIIS (2), S-band and OBDH (1 each onboard)
Mass	2067
Power Consumption	3W
Power Supply Voltage	9-12.5V

Trade Off:

The table for imaging specifications for different optical resolutions is given below:

Resolution Type	Ground Resolution	Angular Resolution	Lens Aperture	Focal Length
High	3m	6.771×10^{-6} rad	100mm	0.442m
Mid	10m	11.28×10^{-6} rad	60mm	0.177m
Mid	10m	16.9×10^{-6} rad	40mm	0.176m
Low	30m	5.1×10^{-5} rad	13.25mm	0.0585m

***Here Satellite Altitude = 500km and Wavelength = 555nm I all cases.

The table for trade off between different configurations of optical resolution is given below:

Resolution Type	Ground Resolution	Integration Time	Signal to Noise Ratio (SNR)	ADCS Stability	Detector Required	Conclusion
Low	30m	0.0001s	30.07	0.8	CMOS	Main Mode
Mid	10m	0.00056s	38.37	0.05	TDI/CMOS	Test Mode
High	3m	0.01s	49.12	0.02	TDI	Cancel

Comments & Suggestions:

- Resolution of the optical system is preferably 10m-15m.
- Baseline can be 30m resolution.
- Phase B: Preliminary Design can start with 15m resolution.
- The challenge however remains with Data Transfer & Storage.
- If GRD > GSD – the image will be blur.
- If GRD < GSD – the image will be sharp.