Preliminary Power Budget Analysis for Equatorial Low Earth Orbit (LEO) Communication Satellite

Desti Ika Suryanti[1,3], Sri Ramayanti [2], Mohammad Mukhayadi [3]

[1]Satellite Technology Center, National Institute of Aeronautics and Space (LAPAN), Indonesia

e-mail: desti.ika@lapan.go.id

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Abstract

Satellite Technology Center – LAPAN would develop a constellation of 9 communication satellites in a low equatorial orbit. These satellites would perform as data collection platforms for many sensors that spread across the Indonesian territory. The data from the sensors will be downlink to Indonesia's ground stations in real-time. This research aims to analyze the power budget of those satellites to decide how many solar panels and batteries are required to perform their mission. The method in this research began by calculating the power requirements of each mission per orbit period to estimate power consumption and calculate the power generated by the solar panels. The results of these calculations will be implemented to the power system design to find the satellite solar cells/ panels arrangement and battery capacity allocation. To minimize the development time and cost, the solar array design in this study considers the utilization of previous solar panel design of LAPAN-A series satellites as a design constraint. This study shows the configuration of 3 body-mounted solar panels and 2 deployable solar panels that could support the mission operation of communication satellite in the low equatorial orbit. For energy storage, these satellites should be equipped with 28 V Li-ion in the 8Sx3P configuration.

Keywords: *Micro satellite; communication satellite; power budget, solar panel; li-ion battery.*

1. Introduction

Manuscript Satellites have a universal characteristic, with a lot of flexibility in the application, efficient in terms of costs, and able to answer various needs such as data communication or voice wireless, the need for financial transactions, internet connection means, and so on. Communication satellites are essentially relay stations placed on the Earth to receive, strengthen, and transmit the accepted analog signals and convert them into digital signals or radio frequencies. One of the satellite applications is its utilization as a means of communication due to many advantages compared to the terrestrial communication system. Satellite Technology Center (Pusteksat) of LAPAN has a heritage in developing low orbit small satellites such as LAPAN-TUBSAT (Triharjanto et al., 2004), LAPAN-A2 (M.A. Saifudin & Mukhayadi, 2013), LAPAN-A3 (Utama, Saifudin, & Mukhayadi, 2018). LAPAN-TUBSAT have imaging mission using video camera as its payload. Meanwhile the missions of LAPAN-A2 are earth observation, maritime monitoring and supporting disaster mitigation. And for the LAPAN-A3, the mission is similar with LAPAN –A2 with addition scientific research mission for earth magnetic field measurement. Aiming for mastery the satellite technology, Pusteksat develop satellites in low orbit constellation with communication mission. Due to the difference in missions, it is necessary to analyze the power budget of this satellite.

Figure 1-1 shows the conceptual design of a low orbit satellite constellation that carries a communication mission, which is composed of four segments, as follows:

a. *Space segments*. The space segment consists of nine satellites orbiting the equatorial Low Earth orbit (LEO) at an altitude of ± 600 km and a 0° inclination angle to perform a real-time communication for Indonesia and other countries in the equatorial region. The satellite carries several communication payloads:
   1. VHF Data Exchange System (VDES), which serves to collect data from sensors, in particular for disaster mitigation and early warning systems.

2. Voice Repeater (VR) and Automatic Packet Reporting System (APRS) to support the amateur communication as well as disaster/ emergency communication.
3. Automatic Identification System (AIS) receiver that serves for vessel detection.
4. Automatic Dependent Surveillance-Broadcast (ADSB) receiver for aviation surveillance.
5. Experimental communications in S and Ka-band.

The satellite in real-time using the X/S band frequency will retransmit all of the received digital messages from VDES, AIS, and ADSB. For Telemetry Tracking and Command (TTC), the satellite uses the UHF frequency.

b. *User segments*. The user segment consists of (1) the user terminals in the field to support disaster sensors and early warning such as tidal sensors, tsunami buoys, geomagnetic sensors, weather stations, and seismographs; (2) Handheld sound repeater to support disaster communication; and (3) additional missions for aircraft and ship monitoring and supervision.
c. *Ground segment*. The ground segment is responsible for mission control consists of the ground station located in Bukittinggi, Bogor, Parepare, and Biak. This will download all the payload data on the satellite using the S-Band frequency.
d. *Network segment*. Data from satellites will be processed and transferred in real-time via cloud servers to agencies, ministries, or private parties.

The design of the power system on a satellite must pay close attention to the power budget because satellite operations carry out many missions in the space duty cycle. This study aims to know the number of solar panels as well as the battery capacity required to support the mission operations of small satellite carrying communication devices in the low Earth orbit. As the design constraint to minimize the development time and cost, the solar array design in this study should estimate whether the previous solar panel design, i.e. panel design of LAPAN-A4 satellite, still could meet the requirements or not.

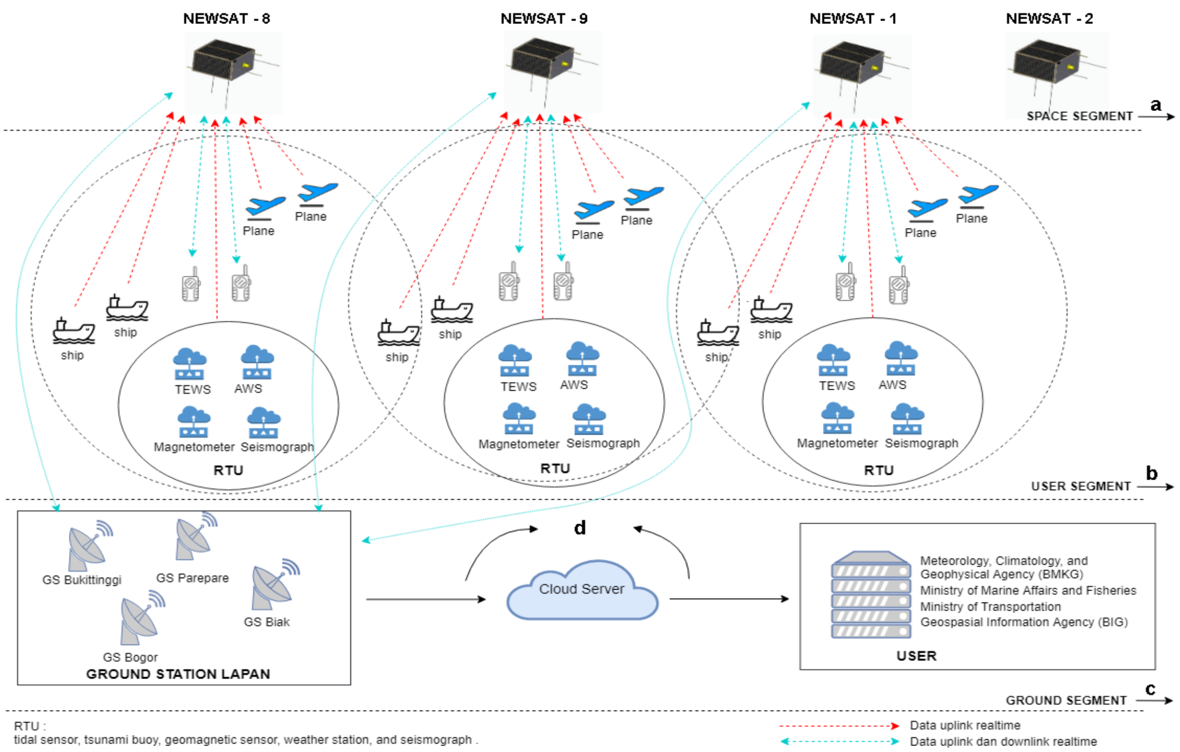

Figure 1-1: Low Orbit Communication Satellite Mission Architecture.

2. Methodology

The power budget analysis in this study is conducted by the following steps:

2.1. Identifying the amount of power consumption required per device

The first step in this study is to identify the devices that will be installed in the satellites, the number of each device to be carried, as well as the operating current and the voltage of each device. In general, the power requirements of a satellite calculated are divided into two groups, namely the power requirements of the bus system and the power requirements of the payload system. The bus system requires power for operations of Power Control & Data Handling (PCDH), reaction wheels, gyro, magnetic torques, star sensors, Global Positioning System (GPS), and TTC. Meanwhile, the payload system requires power for the magnetometer operation, AIS receiver, VDES, ADSB, VR, APRS, S-band for Mobile Satellite Service (MSS), X/S Band Payload Transmitter, Ka-band communications, and payload data handling systems (PDHS).

2.2. Establishing the mission operation of low orbit communication satellites.

To establish the mission operation of low orbit communication satellite, roughly the satellite operations split into two modes, as follows:

a. Acquisition mode, where all the components of the bus and some satellite payloads ON continuously to collect data from sensors on Earth.
b. Active communication mode, where some communication devices are active while the satellites are in the Ground station coverage. Active communication mode is divided into 5 modes, namely:
    1. TTC active mode, in which the operator performs TTC activity for satellite's housekeeping or attitude maintenance through a series of command and telemetry.
    2. VR mode, in which the VR device is activated to repeat any voice signal from users.
    3. Data downlink mode, when the satellite sends the recorded data to the Earth station through X/S Band frequency.
    4. MSS mode, when the satellite delivers a telecommunication service to the mobile users.
    5. Ka-Band communications mode, which is used for experimental communication using a microwave.

2.3. Calculating the orbital period of Eclipse and Daylight.

In the power budget analysis, the duration of the eclipse and the daylight is crucial in determining the satellite power requirements during its operating cycle. Figure 2-1 gives the geometrical relation between spacecraft orbit and the Earth to find maximum eclipse duration. The eclipse will occur in every cycle of the orbital period. When the satellite enters the eclipse period, the solar panels on the satellite will not get sunlight so there is no power generated. Thus, satellites use the battery as a resource to keep running all the electronic devices. The orbital period can be found by Equation (2-1), while the duration of the eclipse, that illustrated in Figure 2-1, is obtained using Equation (2-2) (Apgar et al., 2005).

$$P = 1.658\times10^{-4}\times(6378.14 + h)^{3/2} \tag{2-1}$$

Where P is the orbit period (minutes), h is satellite altitude (km).

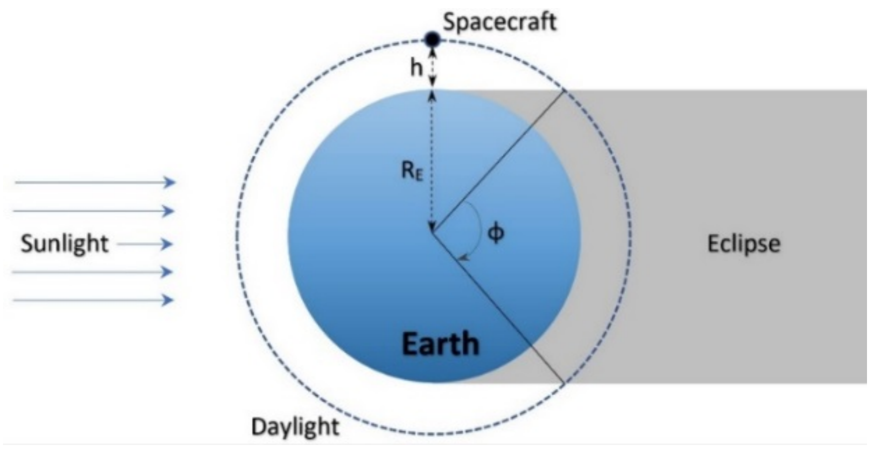

Figure 2-1: Illustration of duration eclipse time.

$$T_E = P(\Phi/360\,deg) \quad (2\text{-}2)$$

With Φ follow:

$$sin\left(\frac{\Phi}{2}\right) = [\frac{R_E}{R_E+h}]$$

Where R_E is the radius of Earth (km); h is satellite altitude (km); Φ is eclipse cover (degree); and T_E is eclipse period (minutes).

2.4. Set the concept design of the satellite power system

The design concept of satellite power systems for low orbit communications on this study adopted the power system design of the LAPAN-A4 satellite as shown in Figure 2-2. The main power resource comes from the solar panels that are mounted on the spacecraft body or assembled in the deploy mechanism. The solar panel consists of a solar cell compiled into 4 strings, and each string consists of 16 cells that produce a nominal voltage of 28 V with an operating voltage range of 24 V to 33 V. The electrical power generated by the solar panels is then stored in the Lithium-Ion batteries. The battery pack serves to provide the power to be used during the mission. The charging and discharging process of the battery is controlled by the Battery Charge Regulator (BCR) on PCDH (Muhammad Arif Saifudin & Karim, 2016). PCDH serves to detect battery under voltage and control charging and discharging processes.

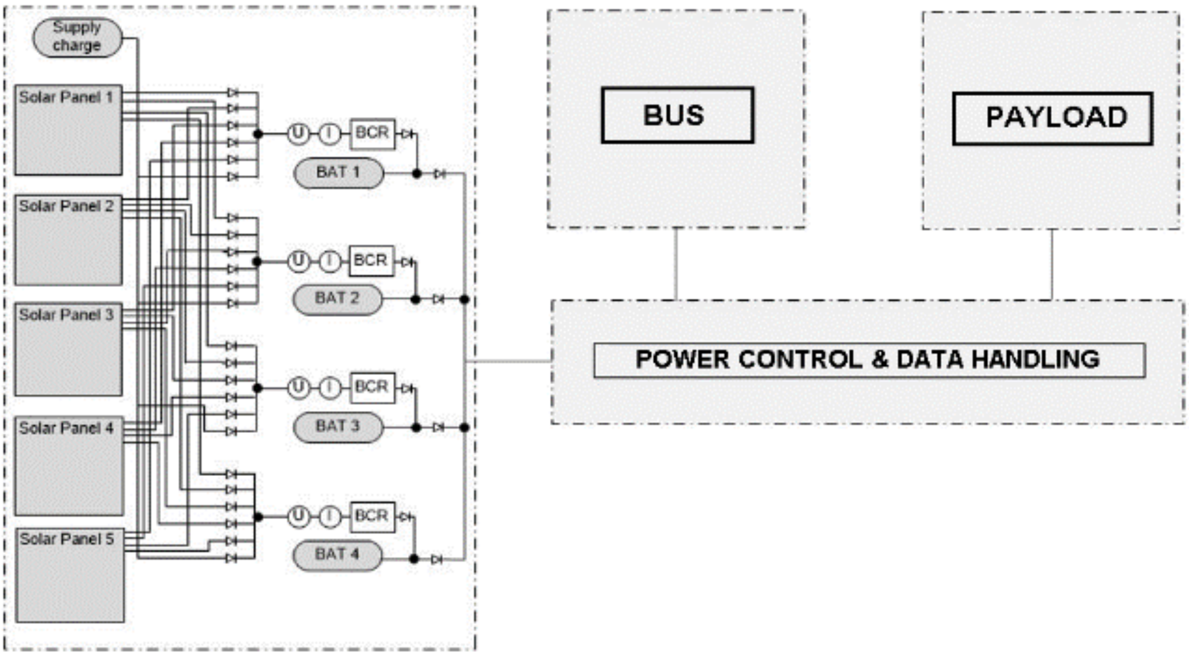

Figure 2-2: Low Orbit Communication Satellite Power System.

2.5. Calculating the power requirement

To determine the number of power needs on this satellite, calculation of the amount the power that should be produced using Equation (2-3).

$$P_{sa} = \frac{(\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d})}{T_d} \quad (2\text{-}3)$$

Where Psa is power that must be generated by solar panels (W); Pe is spacecraft power requirement during eclipse (W); P_d is spacecraft power requirement during daylight (W); T_e is lengths of eclipse period (minutes); T_d is lengths of daylight period (minutes); X_e is path efficiency value for the eclipse; X_d is path efficiency value for daylight.

2.6. Calculating the power production of solar panel systems and the needs of energy storage systems

After calculating the amount of power to be produced, then the next step is to select the type of solar cell and estimate the power output, P_o, with the Sun normal to the surface of the cells by using Equation (2-4).

$$Po = 1367W/m^2 \times ŋ \tag{2-4}$$

Where P_o is the power output (W/m2); here P_o is the power output (W/m2) and ŋ is the efficiency of the solar cell; 1367 is the efficiency of the solar cell; 1367 W /m^2; ŋ is the solar constant in the vicinity of the Earth. By knowing the power output of the solar cells, then the beginning-of-life (BOL) power production capability per unit area of the array can be determined by using Equation (2-5).

$$P_{BOL} = P_o I_d \cos\theta \tag{2-5}$$

Where P_{BOL} is power production capability at beginning of life (W/m^2); Po is the power output (W/m^2); $\boldsymbol{Id}$ is inherent degradation; θ is sun incidence angle (degree).

The solar array must be able to supply power for operation and recharging the battery until the end of life (EOL). To determine the EOL power production capability, P_{EOL} (W/m^2), the life degradation of the solar array (L_d) should be determined. Factors that influence that L_d of solar array include thermal cycling in and out of eclipses, micrometeoroid strikes, plume impingement from the thrusters, and material out gassing for the duration of the mission (Apgar et al., 2005). Generally, the degradation of gallium arsenide and multi-junction in LEO is 2.75% per year and 0.5% per year respectively. Therefore, the actual lifetime degradation can be estimated using Equation (2-6).

$$L_d = (1 - degradation/yr)^{satellite\ life} \tag{2-6}$$

Therefore, the estimation of the array performance per unit area at EOL is obtained using Equation (2-7).

$$P_{EOL} = P_{BOL} L_d \tag{2-7}$$

Knowing the power requirement and the power EOL would enable to estimate the solar array area, A_{sa} (m^2) required for a spacecraft by using Equation (2-8).

$$A_{sa} = \frac{P_{sa}}{P_{EOL}} \tag{2-8}$$

Knowing the required solar array area is the starting point to design a configuration of solar panels. The other key reason to design the solar array configuration is an illumination of the Sun. Solar arrays illumination intensity depends on orbital parameters such as the Sun incidence angles, eclipse periods, solar distance, and concentration of solar energy. Aiming the maximum illumination, the solar panel needs to be always pointing to the Sun. However, the power production of solar panels depends on the relative motion of the Sun against the normal direction of solar panels. Especially for equatorial orbit, the Sun illumination is seasonal, in which the illumination is varying between equinox and solstice. In a one-year cycle, the equinox occurs in March and September, while the solstice took place in June and December. Equinox is a condition of the Earth when day and night has the same duration, while the solstice conditions make the area on the Earth experience the longest or shortest daylight. In the equatorial orbit, the power produced by the solar panels in the solstice could be reduced due to the inclination of the Sun would be up to 23.50° (Zahran, 2006). The comparison of Sun's illumination between the equinox and solstice condition of the solar panel for nadir pointing satellites in the equatorial orbit can be seen in Figure 2-3. The illumination value (I_l), calculated using the following Equation (2-9).

$$I_l = \cos\alpha . \cos\ (lat_{sun}) \tag{2-9}$$

Where α = ω. t in which ω is satellite's orbit angular velocity (0.062 deg/sec); lat_{sun} is the sun latitude position.

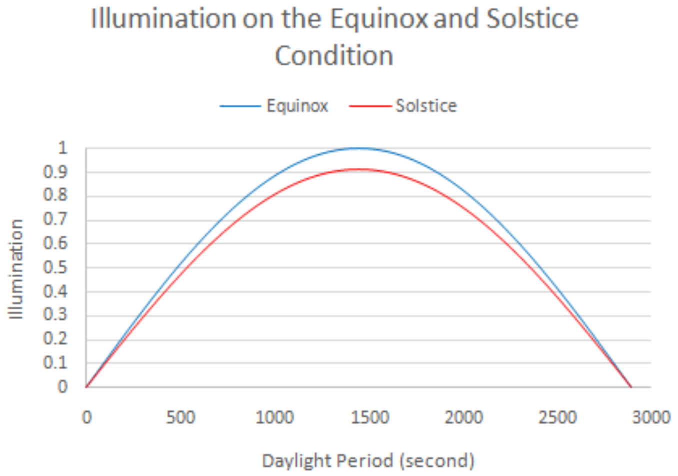

Figure 2-3: Illuminating solar panel of equatorial LEO satellite at 600 km altitude.

Included in the power system, batteries are a key component of the satellite that provides power during the eclipse. Indeed, in the daylight periods, the satellite is powered by the solar panels that in parallel recharge the battery; but in the shadow periods, the battery takes the relay and delivers power to the bus and payload (Borthomieu, 2014). Currently, Lithium-Ion type of batteries has been vastly used in the space mission. Table 2-1 shows the parameters for batteries to be used in LEO missions.

Table 2-1: Parameters for batteries (Borthomieu, 2014)

Parameter	Value
Life duration	2-15 years
DOD	10-40%
Charge rate	C/3
Discharge rate	C/2 to C/1.5
Temperature range	0°C to +40°C
Compliance standard	ESA and NASA standards

Two battery configurations by Li-ion satellite, namely S-P topology and P-S topology. This study uses S-P topologies that are characterized by serial strings of cells assembled in parallel. Usually, Equations (2-10) can be used to calculate the number of series cells

$$N_s = \frac{V}{V_c} \tag{2-10}$$

Where Ns is number of series cells; V is bus voltage (V); Vc is one cell voltage (V). Then, the calculation of battery capacity uses Equations (2-11).

$$C = \frac{P_e \times T_E}{N_s \times \tilde{V} cell_{EOL} \times DoD} \tag{2-11}$$

Where C is battery capacity (Ah); Pe is eclipse power (W); T_E is eclipse period (hour); $\tilde{V} cell_{EOL}$ is average cell voltage at EOL (V); DoD is depth of discharge. To calculate how many numbers of parallel cells, Equation (2-12) is used.

$$N_p = \frac{C}{C_c} \tag{2-12}$$

Where N_p is number of parallel cells; C is battery capacity (Ah); C_c is capacity of one cell (Ah).

3. Result and Analysis

3.1. Power Consumption of Devices

Satellites carry devices that consume certain power and duration to support the mission operation as shown in Table 2-2.

Table 2-2: Power Consumption of Equatorial LEO Communication Satellite

Device	No	Voltage (V)	Current (mA)	Duration (hour)	Power (Watt)	Energy (Wh)
BUS SYSTEM						
PCDH	1	28.0	98.15	1.6	2.75	4.40
TTC Active	2	28.0	685.71	0.3	7.20	311.52
TTC Idle	2	28.0	58.97	1.3	2.68	4.29
S Band TTC Uplink	1	28.0	58.97	1.6	1.65	2.64
S Band TTC Downlink	1	28.0	560.82	0.3	2.94	4.71
Reaction Wheel X	1	28.0	10.15	0.3	0.05	0.09
Reaction Wheel Y	1	28.0	114.73	1.6	3.21	5.14
Reaction Wheel Z	1	28.0	10.15	0.3	0.05	0.09
Gyro X	1	5.0	108.0	0.3	0.10	0.16
Gyro Y	1	5.0	108.0	1.6	0.54	0.86
Gyro Z	1	5.0	108.0	0.3	0.10	0.16
Coil	3	28.0	30.00	1.6	2.52	4.03
Star Sensor	2	28.0	67.33	0.3	0.71	1.13
GPS	1	28.0	99.62	1.6	2.79	4.46
TOTAL POWER CONSUMPTION OF BUS SYSTEM					27.31	43.69
PAYLOAD SYSTEM						
Magnetometer	1	28.0	32.80	1.6	0.92	1.47
AIS receiver	1	28.0	27.39	1.6	0.77	1.23
ADSB	1	3.3	400.0	1.6	1.36	2.18
VDES	1	28.0	27.39	1.6	0.77	1.23
VR	1	28.0	685.71	0.4	4.80	7.68
APRS	1	28.0	716.56	1.6	20.06	32.10
S Band MSS	1	28.0	560.82	0.3	2.94	4.71
X/S Band Payload Transmitter	1	28.0	2455.48	0.4	17.19	27.5
Ka Band Voice Communication	1	28.0	2455.48	0.3	12.89	20.63
PDHS	1	28.0	404.46	1.6	11.33	18.12
TOTAL POWER CONSUMPTION OF PAYLOAD SYSTEM					73.02	116.84
TOTAL					100.33	160.53

Each device has certain functions and tasks, some of them are working all the time, which are indicated by 1.6 hours duration equal to the orbital period. Generally, 70% of total power consumption is spent in the payload system. PCDH is the main computer on the spacecraft, which consists of a combination of the Power Control Unit (PCU) and Onboard Data Handling (OBDH) units (Karim, 2015). The reaction wheel is the main actuator for spacecraft attitude control that produces angular torque by rotating the wheel. Gyro is a sensor that measures angular motion. Magnetic torque (Coil) is a device made from coil wire that interacts with the Earth's magnetic field to produce control torque. It can compensate for magnetic residue on the spacecraft as well as dumping the angular momentum when the reaction wheel is saturated (Dear, 2010). Star Sensor is a reference sensor that provides star position data to navigate the spacecraft's orientation (Asadnezhad, Eslamimajd, & Hajghassem, 2018). GPS is used to provide position and time synchronization for the spacecraft. The TTC is a device that provides a connection between the satellite and the facilities on the ground. A magnetometer is a sensor for measuring the Earth's magnetic field. AIS receiver is a satellite payload that receives messages from ships in certain formats and frequency (Judianto & Agung, 2014). VDES is a radio communication system that enhanced the loading of the VHF Data Link (VDL) of the AIS and supports new digital services. ADSB is a system to detect aircraft that transmits the information of altitude, position, speed, direction, and other information (Nurhayati, 2014). VR is a device that receives a voice signal and retransmits it. APRS is a communications protocol and device that receives and retransmits digital broadcast messages (Hardhienata, Triharjanto, & Mukhayadi, 2011). S-Band and Ka-Band communication are a payload is used to support experimental communication between users on the ground. X/S Band payload transmitter is a system that transmits the payload data to the ground station. PDHS is a device that handles all data management processes ranging from mission data collection, onboard data recording/ retrieval, and data protection (Suhermanto, 2017).

3.2. Solar Array Design

The constellation of communication satellites in this study will be placed into low Earth orbit within 600 km altitude and an inclination angle 0 degrees for 5 years' lifetime. By putting the altitude to the Equation (2-1), it found the orbital period of the satellite is 97 minutes while the eclipse duration from Equation (2-2) gives values of 35 minutes. According to Table 2-2, the average power consumption is defined as 100.33 Watt either during daylight (Pd) or during the eclipse (Pe). The efficiency of the satellite power subsystem in the daylight and eclipse are roughly Xd = 0.995 and Xe = 0.956 (Mukund, 2005). Nowadays, the efficiency of GaAs solar cells could achieve 29.5%, the inherent degradation ($\boldsymbol{Id}$) is 0.981, and the degradation per year is 2.034 % (Company, 2016). For the prediction of solar cell degradation, this study uses the spectral irradiance reference of air mass zero (AM0) which is the extraterrestrial solar irradiance at a distance of one astronomical unit from the Sun (*ASTM E490 Standard Solar Constant and Zero Air Mass Solar Spectral Irradiance Tables*, 2006). The calculation result of the power requirement is summarized in Table 2-3.

Table 2-3: Power Requirement Calculation of Solar Panel

Parameter	Value
Power that must be generated by Solar Panel (W)	P_{sa} = 161.66
Power Output (W/m²)	Po = 403.27
Power BOL (W/m²)	P_{BOL} = 395.7038
Life Degradation	L_d = 0.90
Power EOL (W/m²)	P_{EOL} = 357.0667
Solar Array Area (m²)	A_{sa} = 0.45

Photovoltaic cells can be wired together to make solar panels for larger size loads (Shaikh, Waghmare, Labade, Fuke, & Tekale, 2018) (Id et al., 2019), (El-Ghonemy, 2012). The panels can be collected together to create a solar array for large-scale power generation (Heo et al., 2013). The placement of solar panels should orient their normal direction to the Sun so they will generate adequate power throughout the mission (Santoni et al., 2014). Typically, in the previous LAPAN-A satellite series, the solar panels were mounted on the spacecraft body. To meet the power requirements, the spacecraft could implement a deployable mechanism for solar panels. Sometimes, the deployable mechanism is equipped with a sun tracker rotator to maximize the illumination. However, implementing sun tracker rotator in the small satellite is not desired due to the increased complexity of the satellite design. Figure 2-4 illustrated the nominal flight configuration and solar panel placement in the defined axis of the equatorial satellite.

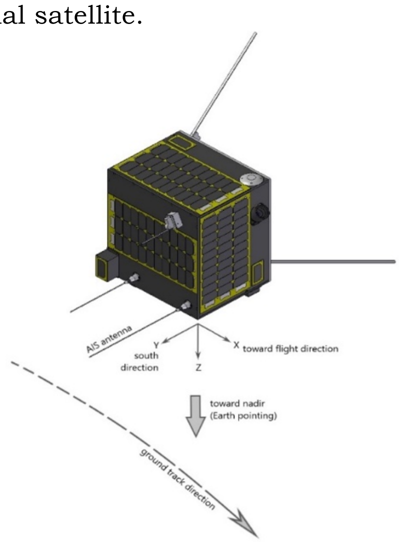

Figure 2-4: Nominal flight configuration and solar panel placement of the equatorial satellite (Source: Hasbi, Kamirul, Mukhayadi, & Renner, 2019).

The power production is equivalent to the Sun illumination on the solar panels (Amajama, 2016). For comparing the power production, there are four configurations of solar panel placement, from the simplest to the more complex one. Refer to the spacecraft axis is shown in Figure 2-5, Configuration A consist of one solar panel installed on -Z side; Configuration B contains three panels installed on +X, -X, and – Z sides; Configuration C contains five panels installed on +X, -X, +Y, -Y, and – Z sides; And Configuration D contains five panels which are three solar panels installed on +X, -X, and -Z sides, while the rest two panels placed on -Z side using the deployable mechanism.

According to Table 2-4, the required solar panel area is 0.45 m^2 to meet the power requirement. While the solar panel design adopts the panels of the LAPAN-A4 satellite that is consisted of four strings, every string contains 16 cells, and each solar cell area is 30.18 cm^2, then a one-piece of a solar panel has an effective area of 0.193 m^2. It means the satellite requires 2.33 solar panels that should face the Sun continuously. During the nadir pointing mission, in which one side of the spacecraft (usually defined as +Z side) always points to the Earth, the Sun illumination on each side is varying follow Equation (2-9) instead of tracking the Sun. The total accumulation of the illumination value acquired by each panel is defined as illuminated panels and presented in Table 2-4. It shows that only Configuration D would meet the power requirement in both equinoxes and solstice. The solar panel design and the configuration of placement are illustrated in Figure 2-5.

Table 2-4: Comparison of Solar Panels Power Production (Suryanti, Ramayanti, & Mukhayadi, 2019)

Parameter	Configuration A		Configuration B		Configuration C		Configuration D	
	1 body-mounted panel		3 body-mounted panels		5 body-mounted panels		3 body-mounted panels + 2 deployable panels	
	Equinox	Solstice	Equinox	Solstice	Equinox	Solstice	Equinox	Solstice
Illuminated Panels	0.64	0.58	1.27	1.17	1.27	1.57	2.55	2.34
Equivalent Area (m^2)	0.12	0.11	0.25	0.23	0.25	0.30	0.49	0.45

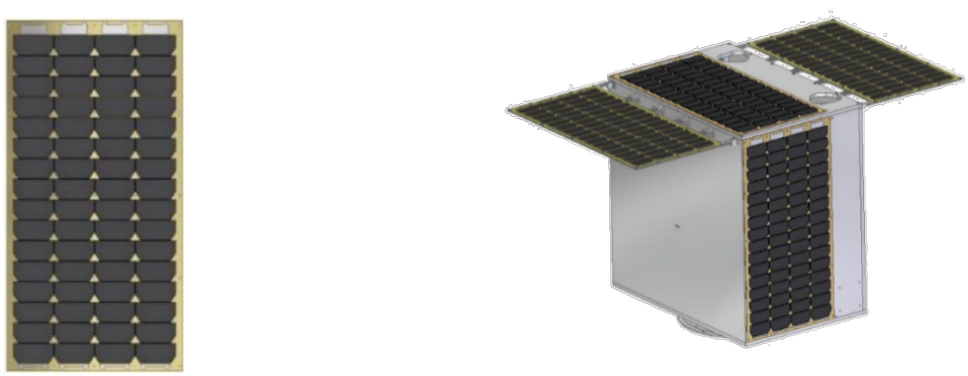

Figure 2-5: Solar Panel Design and the Proposed Placement of Equatorial LEO Satellite

3.3. Energy Storage Design

During the daylight period, spacecraft uses solar arrays to generate power and to charge the batteries. While in the eclipse period, the batteries use their energy storage to perform the role of the solar panels and providing the satellite subsystem with their required power (Abdi, Alimardani, Ghasemi, & Mirtalaei, 2013). In this study, S-P topology is chosen since it is the most used in LEO satellites that using a regulated 28 V bus (Borthomieu, 2014). The bus voltage required to run electronic devices on the satellite is 28 V that has been implemented also on the LAPAN-A4 power system design.

A small Li-ion battery has the end of the discharge cutoff voltage of 2.7 V, average discharge voltages of 3.5 V, and an end-of-charge voltage of 4.2 V (Mukund, 2005). According to the study by Yannick Borthomieu, Satellite Lithium-Ion Batteries, a 28 V of 8S-1P battery with 15 Ah capacities was tested in a cycle within 20% DOD in LEO. After 40,355 cycles, the voltage of each cell in the test pack has been measured that results in the maximum deviation in voltages for the (eight) cells were 23 mV. Therefore, based on the previous operation of Li-ion battery in LEO, the design of the energy storage system in this study takes the value of 3.75 V of average battery cell voltage, 20% of DOD, 3.73 V of average battery cell voltage at EOL after 7 years (more than 40,000 cycles), and 6.5 Ah of Cell Capacity. Based on the aforementioned specification, the design of energy storage in low orbit communication satellites is presented in Table 2-5.

Table 2-5: Design Parameter of Energy Storage for Low Orbit Communication Satellite

Parameter	Value
Bus Voltage (V)	$V = 28$
Cell Voltage (Vc)	$V_c = 3.75$
Number of Series Cells (NS)	$N_s = 8$
Eclipse Power (W);	$P_e = 100.33$
Eclipse Period (hour)	$T_e = 0.6$
Average Cell Voltage at EOL (V);	$\tilde{V}cell_{EOL} = 3.73$
Depth of Discharge (%)	20
Battery Capacity (Ah)	$C = 9.95$
Cell Capacity (Ah)	$C_c = 6.5$
Number of Parallel Cells	$N_p = 2$

Table 2-5 also indicates that the battery configuration of this satellite is 8S x 2P meaning the battery consist of two strings and each string consists of eight cells of the battery-compiled series. In S-P configuration, if the cell fails the entire string is lost, along with the capacity that should have been provided. However, the voltage remains the same and therefore the satellite system can continue to operate the mission without the other effects of a small reduction in battery capacity (Borthomieu, 2014). However, to secure the mission operation, adding one more string could anticipate if one string is failed. Hence, the recommended configuration is 8S x 3P.

4. Conclusions

The design of the satellite power system, including the solar panels as well as the energy storage system, requires comprehensive power budget analysis. In the equatorial low orbit communications satellite, the solar panel design of the LAPAN-A4 satellite still could meet the power requirements. By implementing, the configuration that contains three body-mounted and two deployable panels, the solar panels can produce enough energy to run all operating modes of the satellite that required power of 100.33 Watt. Furthermore, this low orbit communications satellite has to carry 9.95 Ah of 28 V Li-ion batteries at a minimum in the 8S x 3P configuration to support the mission operation during the eclipse.

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Contributorship Statement

DIS, SR and MM together designed the method, analyzed the results and prepared the script. In this paper DIS is a major contributor.

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