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Design of Electrical Power Subsystem of a Low Orbit Satellite

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Abstract

An electrical power subsystem of a satellite is the satellite subsystem responsible for the supply of energy to other subsystems of the satellite. The aim of this work is to design the electrical power subsystem for a low earth orbit which will orbit at 650 km altitude above the earth's surface. The average power of the satellite is 800 W and it has a life span of 8 years. The design was done using direct energy transfer method and mathematical computation process to calculate various parameters of the power generation, storage, regulation and distribution units. The result showed that a total of 9647 cells, consisting of 254 cells in parallel and 38 series (string) cells and array mass of the solar array was 65.5 kg will be needed for the power generation unit while the power storage unit is made up of 28 batteries of 1.25 V battery voltage and 53 Ahr capacity. In addition, the charge controller of amperage, 64 A is used for the charge regulation of the batteries while the parameters for the power distribution unit (bulk converter) calculated accordingly.

Keywords

Battery, Bulk Converter, Energy, Orbital Period, Power Regulation Unit, Satellite

1. Introduction

A spacecraft electrical power subsystem is the satellite subsystem that generates power, regulates it, stores it for periods of peak demand or eclipse, and distributes to the entire spacecraft. Electrical power subsystem of a satellite supplies electrical energy to all other subsystems of the satellite. It is made up of four units-power generation (solar array), power storage (batteries), power regulation (charge controller) and power distribution (boost or bulk converter) units. The power distribution unit is designed to supply electrical power in a more sophisticated way but at the same it enables simpler design for subsystems which lessen the concerns about required power for them [1–4].

According to Samina, the design of satellite subsystems should be able to meet the mission objectives of minimum cost, and also considering the weight limits since it is one of the most important and challenging aspects of the design process [5]. Previous studies have shown that the higher the mass (weight) of a satellite, the higher the cost especially in terms of launching the satellite into space, thus in designing an electric power subsystem of a satellite, weight must be a major factor as it is directly proportional to the cost of the satellite.

Solar energy which is the energy from the sin is the primary power source of any satellite. The electrical power subsystem uses solar power directly when this power source is available and it is also responsible for charging and monitoring the change of the spacecraft's batteries and triggering power conservation activities if the level of overall power falls below a preset level [6]. This implies that the solar cells (i.e. the electrical power generation units) of a satellite is very important as it is the primary power source of a satellite.

Various works have been done in this area by many researchers but this work focuses on the design of electrical power subsystem of a low orbit satellite which will orbit at altitude 650km above the surface of the earth. The satellite has an average power of during eclipse and sunlight to be 800W, and will spend 8 years before its life span elapses. The design will adopt direct energy transfer (DET) method. Direct energy transfer is the type of energy transfer in which the solar power is transferred to the loads with no series component in between except the slip rings to provide a rotary joint between the Earth-facing spacecraft body and the sun-facing solar array, and the power distribution unit consisting of load switching relays and fuses to protect the power system from faults in the load circuits [7].

2. Methodology

2.1. Parameters

The parameters required for the design are average power of the satellite (800W), distribution unit currents and voltage (0.1A, 1.6A and 4.0A, and 3.3V, 5.0V, 12V), the altitude of the satellite (650km), number of years of the operation of the satellite in space. Other data needed are parameters of solar cell (Gallium Arsenide) such as current, voltage, percentage degradation, efficiency, and the parameters of the Nickel hydrogen (NiHe) battery such as depth of discharge, voltage, efficiency and specific energy density from the datasheet and related literatures as well as the satellite bus voltage [8].

2.2. Methods

This design makes use of direct energy transfer method with mathematical computations which include various formulas or equations for calculating the various components of all the units of the electrical power subsystem of the satellite.

2.2.1. Design Analysis

a) Calculation of Satellite Orbital Altitude:

Orbital altitude is the sum of radius of the earth and the height of the satellite above the earth's surface.

Orbital altitude is given as;

$$r = R_e + h \tag{1}$$

where h = 650km and R_e = 6370km, then, r = 7020km when values are substituted into (1).

b) Calculation of Orbital Period:

Orbital period is the total time spent by the satellite to make one complete revolution or movement round its orbit. It is calculated using the formula given from as [9]:

$$T = \sqrt{\frac{4\Pi^2 r^3}{GM}} \tag{2}$$

where r = 7020km, G = 6.67×10^{-11}Nm^2/kg^2 (Gravitational constant of the earth), and $M_{earth} = 5.98 \times 10^{24}$kg and when substituted gives

$$T\ orbit = 5852.32 \text{ seconds} = 97.54 \text{ minutes} = 1.63\text{hrs}.$$

Therefore, the total time taken by the satellite to cover its orbit is approximately 98 minutes.

c) Calculation of Angle of View of the Satellite, α:

$$Sin\alpha = \frac{\text{Re}}{\text{Re}+h} \tag{3}$$

when the values are substituted, it gives;

$$\alpha = \sin^{-1}(0.9074) = 65.15^o$$

Now, for the time spent by the satellite under eclipse, *Te*, it is gotten using the formula [9];

$$Te = \frac{2\alpha}{2\Pi} \text{ x } Torbit \tag{4}$$

$$Te = \frac{2 \text{ x } 65.15}{360} \text{ x } 98 \text{ minutes} = 35.5 \text{ minutes}$$

Then, for the time spent by the satellite during the sunlight, it is the difference between the orbital period and time under eclipse. Therefore,

$$T_{sun} = T_{orbit} - T_{Eclipse} \tag{5}$$

$$T_{sun} = (98 - 35.5) \text{ minutes} = 62.5 \text{ minutes.}$$

d) Power Generation Design:

The power generation is done by solar cells onboard the satellite. The solar cell used in this work is Gallium Arsenide solar cell.

2.2.2. Gallium Arsenide Cell Sizing

A solar array must be designed so that the spacecraft's power needs during daylight and eclipse phases are met [9]. The parameters of Gallium Arsenide cell used here are: Efficiency = 19%, Cell voltage = 0.85V, Cell current = 0.2A, Fractional degradation per year = 2.75% [10]. The bus voltage for the design is 28V. This is the total voltage supplied by the power source before it is distributed to the appropriate components using it. The procedures used for designing the solar array are as follows:

a) Procedure 1: Determination of Solar Array Power:

The solar array power is calculated using the formula [9];

$$P_{array} = \left(\left(\frac{P_{sun} \text{ x } T_{sun}}{X_{sun}} + \frac{P_e \text{ x } T_e}{X_e} \right) \div T_{sun} \right) \tag{6}$$

where T_{sun} = Time in which the satellite is under sunlight (or day time), T_e = Time in which the satellite is under eclipse, P_{sun} = Power needed during sunlight, P_e = Power needed during eclipse, T_{sun} = Time spent in sunlight, T_e = Time spent under eclipse, X_{sun} = Path efficiency under sunlight or during daytime and is taken for this design as 0.85, X_e = Path efficiency under eclipse and is taken for this design to be 0.65. These are the efficiencies for the direct energy transfer.

And power under sun = power under eclipse, P_{sun} = P_e = 800 W.

Thus;

$$P_{array} = \left(\left(\frac{800 \text{ x } 62.5}{0.85} + \frac{800 \text{ x } 35.5}{0.65}\right) \div 62.5\right) = 1640\text{W}.$$

b) Procedure 2: Calculation of Ideal Power Output of the Solar Array:

Ideal power output of the solar array is the product of the efficiency of the cell and the solar constant. Solar constant, *S* ranges from 1310W/m^2 to 1400W/m^2 [9]:

$$P_{Ideal} = \eta S \tag{7}$$

where S = solar radiation and is taken to 1358W/m^2 in this design. This is to allow the ideal power to be in between the lowest and the highest acceptable value for the satellite:

$$P_{Ideal} = 0.19 \text{ x } 1358 = 258.02\ W / m^2$$

c) Procedure 3: Determining the Beginning of Life Power (P_{BOL}) Production Capability:

Beginning of life power production capability is the total amount of power the cell can produce at the first stage or at the beginning of its launch. It is calculated as follows:

(i) Obtain the inherent degradation: Inherent degradation (I_d) is defined as the nominal power loss as a result of inherent inefficiencies in solar array power system. It is taken here to be 0.77 [9].

(ii) Calculation of worst case incident angle. It is the angle at which least output of power generation from the solar cell is expected. In this design, the angle value θ = 23° is used. It varies between 20°-40° [9].

(iii) Calculation of Beginning of Life power (P_{BOL}) at the incident angle of θ°. P_{BOL} is calculated using the formula given from [9] as:

$$P_{BOL} = P_{ideal} \text{ x } I_d \times \cos\theta \tag{8}$$

By recalling that P_{ideal} = 258.02W/m^2, I_d = 0.77 and θ = 23°.

$$P_{BOL} = 258.08 \times 0.77 \cos 23 = 182.88167\text{W/m}^2.$$

d) Procedure 4: Determining of end of End of Life Power (P_{EOL}) Production Capability:

This is the total amount power the solar array is expected to produce at the end of its life or before its life span elapse. The life degradation (L_d) of the solar array is calculated using the formula given as;

(i) Life degradation,

$$L_d = \left(\left[1 - \left[Percentage \text{ degradation } per\ year\right]\right]\right)^{satellite\ life\ span} \tag{9}$$

The percentage degradation per year of the solar cell is 2.75%. The satellite life span is 8 years, thus;

$$L_d = (1 - 2.75\%)^8 = 0.8$$

(ii) Calculation of End of Life Power (PEOL). End of Life Power is the product of the beginning of life power (P_{BOL}) and life degradation (L_d) [9]:

$$P_{EOL23} = P_{BOL\,23} \times L_d \tag{10}$$

(23° is the worst case angle (θ = 23°))

$$P_{EOL23} = 182.88167 \times 0.8 = 146.3143\text{W/m}^2.$$

e) Procedure 5: Calculation of Solar Size:

It is the calculation of the area of the solar cell. It is as [9]:

$$A_{array} = \frac{P_{array}}{P_{EOL}} \tag{11}$$

where P_{array} = 1640W and P_{EOL} = 146.3143W/m^2.

$$\therefore\ A_{array} = \frac{1640W}{146.3143W / m^2} = 11.21 \approx 12m^2$$

f) Procedure 6: Calculation of Number of Cells Needed for the Design:

It is the product of the number of cells in series (*n)* and the number cells in parallel (*m*). It is calculated from one cell power and solar array power [9]. Mathematically;

$$P_{cell} = I_{cell} \text{ x } V_{cell} \tag{12}$$

where I_{cell} = 0.2A and V_{cell} = 0.85V, Thus, $P_{cell} = 0.2 \text{ x } 0.85 = 0.17\ W$

Then for the number cells, n x m, it is calculated from the ratio of P_{array} to the power of a cell. That is;

$$\text{Total number of cells (n x m)} = \frac{P_{array}}{P_{cell}} \tag{13}$$

$$\text{n} \times \text{m} = \frac{1640}{0.17} = 9647\ cells$$

g) Procedure 7: Calculation of Number of Cells Per String:

This is the number of cells to be connected in series. The number of series cells (n) is calculated as [9]:

$$n = \frac{V_{Bus} + \Delta v}{V_{cell}} \tag{14}$$

where Δv has a standard value of ±4 which either added or subtracted from the bus voltage during the design. In this design, it is added to the bus voltage so that losses due to heat as the current flows through wires is taken care of. Thus,

$$\therefore \quad n = \frac{28+4}{0.85} = \frac{32}{0.85} = 38$$

h) Procedure 8: Calculation of Number of Strings, m:

This is the total number of series connected cells that are connected in parallel.

i.e. $m = \dfrac{n \text{ x } m}{\text{n}} = \dfrac{9647\ cells}{38} = 253.86 \approx 254$

i) Procedure 9: Calculation of Solar Array Mass, M_{array}. It is calculated by using the formula;

$$M_{array} = 0.04 \times P_{array} \tag{15}$$

$$M_{array} = 0.04 \times 1640 = 65.6\text{kg}$$

j) Procedure 10: Calculation of Power to be dissipated on board the spacecraft (s/c), P_{dis}:

The power to be dissipated on board the spacecraft is gotten as the difference between the power produced at the BOL, 0° (that is zero degree) incident and the EOL, 23 (that is 23 degree) incident angles. Mathematically,

$$P_{dis} = P_{BOL,0} - P_{EOL,23} \tag{16}$$

$$\text{But } P_{BOL,0} = P_{ideal}\, I_d \cos\theta \tag{17}$$

where θ here is 0, and I_d and P_{Ideal} are 0.77 and 258.02 W/m^2 respectively.

$P_{BOL,0} = P_{Ideal}\, I_d \text{ Cos } 0 = 258.02 \times 0.77 \times 1$, $P_{BOL,0} = 198.6754$.

$$P_{BOL,0} - P_{EOL,23} = 198.6754 - 146.31 = 52.3654\text{W/m}^2.$$

k) Procedure 11: Calculation of System Voltage and Current:

System voltage,

$$\text{Vsa} = \text{Number of series cells x Voltage of each cell} \tag{18}$$

$$\text{Vsa} = 38 \times 0.85 = 32.3\text{V}$$

System current, Isa = Number of cells in parallel x Current of each cell

$$\text{Isa} = 254 \times 0.2 = 50.8 \text{ A}$$

2.2.3. Electrical Power Storage (Battery)

The battery used in this design is Nickel Hydrogen (Individual vessel pressure design) battery. For Nickel hydrogen battery, depth of discharge (DOD) is from 40 to 56% but 40% is used in this design. It has cell voltage of 1.25V, specific energy density = 30Whr/kg and efficiency of 80% is selected for use here [9].

a) Procedure 1: Calculation of Electrical Energy:

Calculation of electrical energy that the battery is to deliver in order to power the components when the need arise. The electrical energy is the product of the electrical power and the time in which the power is expended. The battery electrical energy, E_{bat} is calculated by using the formula [9]:

$$E_{bat} = \frac{P_e \text{ x } t_e}{X_e} \tag{19}$$

where P_e is the power required by the satellite to operate during eclipse, t_e is the time during of the satellite when under eclipse and X_e is the path efficiency of the battery, taken as 80% in this design. Since P_e = 800W, and t_e = 35.5min, E_e will now be gotten as below;

$$E_{bat} = \frac{800\ W \text{ x } (35.5 \text{ min}/60) \text{ hr}}{0.80},$$

and E_e = 592W-hr.

b) Procedure 2: Calculation of Total Number of Eclipse (N) for the Life Span:

Knowing the total number of eclipse that a satellite will experience will help the designer to select the battery with a depth of discharge. The number of maximum eclipse a satellite will experience can be calculated using any of the formula as [9]:

$$N = \frac{\text{Mission } life \text{ span x } No.\ of\ \ days\ in\ one\ year \text{ x } No\ of\ \ hrs\ in\ a\ day \text{ x } No\ of \text{ min}\, s.\ in\ an\ hr}{\text{T}_{\text{Orbital(in minutes)}}} \tag{20}$$

The satellite life span is 8 years and the orbital period is in minutes, i.e., 98 minutes, thus,

$$N = \frac{\text{8yrs x 365 days x 24hrs x 60 mins}}{\text{98 minutes}}$$
$$= 42906.12, \approx 42{,}907 \text{ eclipse}$$

It is rounded up because any little eclipse or absence of sunlight that is not considered during the design can be disastrous. Thus, the number of charge/discharge ratio per year is;

$$\text{Charge/discharge ratio per year} = \text{N/satellite life span years} \tag{21}$$

$$= 42{,}907/8 = 5363.37$$

which is approximately = 5,364.

c) Procedure 3: Calculation of Depth of Discharge (DOD):

The depth of discharge of nickel hydrogen battery, is given as 40-56% because the cycle is greater than or equal to 4000 but less than 10,000 [9]. But in this design, 40% is used. This means that the state of discharge is now 60%.

Therefore, DOD = 40%

d) Procedure 4: Calculation of Battery Efficiency:

This can be calculated and can also be gotten from the battery's data sheet. The efficiency of Nickel hydrogen (NiH_2) Battery here is 80%.

Therefore, Efficiency = 80%

e) Procedure 5: Calculation of Battery Capacity:

This is the total amount of energy that can be stored by the batteries under certain conditions. The battery capacity C, is calculated by using the formula given as [9]:

$$C = \frac{E_{bat}}{\eta_{bat} \text{ x } DOD} \tag{22}$$

$$\therefore \quad C = \frac{592}{0.8 \text{ x } 0.4} = 1850 \ Whr$$

That is, only 40% of 1850W-hr will be used during the discharge part of the cycle.

In order to calculate the battery capacity in Ampere-hour, the battery voltage, V_{bat} will be calculated using the formula given as;

$$V_{bat} = \frac{V_{bus}}{\eta_{bat}} \tag{23}$$

and $V_{bat} = \frac{28}{0.8} = 35 \ V$

Furthermore, the battery capacity in Ampere-hour is calculated using the relation;

$$C_{amp-hr} = \frac{C_{W-hr}}{V_{bat}} \tag{24}$$

so that $C_{amp-hr} = \frac{1850}{35} = 52.85 \approx 53 Ahr$

f) Procedure 6: Calculation of Number of Batteries Required:

This is the total number of batteries that will be connected in series in order to supply the adequate energy to the satellite subsystems. It is the ratio of the battery voltage to the voltage of the battery chosen for the design. The voltage of the battery chosen for this design is 1.25V. Thus, mathematically, it is represented as;

$$N_{bat} = \frac{V_{bat}}{V_{bat-selected}} \tag{25}$$

and $N_{bat} = \frac{35}{1.25} = 28$.

Therefore, 28 batteries of capacity 53A-hr will be connected in series. Furthermore, 30 batteries can be used to leave out boost converters.

g) Procedure 7: Calculation of Battery Mass:

The mass of the battery is calculated from the ratio of the battery capacity in watt-hour to the specific energy of the battery. Specific energy capacity of the battery can be gotten from the data sheet or labeled on the battery. The specific capacity used here is 30 W-hr. It is expressed mathematically as [9]:

$$M_{bat} = \frac{C_{w-hr}}{Sp - energy_{bat}} \tag{26}$$

Thus, $M_{bat} = \frac{1850}{30} = 61.676 \ kg \approx 62 \ kg$

h) Procedure 8: Calculation of Charge Rate:

It is the amount of energy that "goes" into the battery during charging. It is calculated as;

$$C_{rate} = \frac{E_{bat}}{\eta_{bat} V_{bat}} \tag{27}$$

$$= \frac{592 \ Whr}{35 \text{x } 0.8 \ \ V} = 21.14 \ \ Ahr$$

i) Procedure 9: Calculation of Charge Current:

It is the current that is required for charging the batteries. Charge current is calculated using the formula [7]:

$$I_{charge} = \frac{Charge \ rate}{T_{sun}} \tag{28}$$

where T_{sun} = 62.5min,

Thus, $I_{charge} = \frac{21.14 \ Ahr}{(62.5 \min \div 60) \ hr} = 20.29 \ \ Amp$

j) Procedure 10: Calculation of Discharge Current

Discharge current is the ratio of the power during the eclipse to the battery voltage and battery path efficiency (ie between the battery and loads). It is given as;

$$I_{discharge} = \frac{P_e}{V_{bat} \text{ x } \eta_{path-bat}} \tag{29}$$

and $I_{discharge} = \frac{800}{35 \text{ x } 0.8} = 28.57 Amp$

2.2.4. Charge Controller Sizing

The charge controller prevents the battery of overcharging and over discharging. The type of charge controller used in this design will not be maximum power point tracking (MPPT) type for simplicity since system voltage in all the cells will be greater than the bus voltage, considering the tolerance value that will be added as margin. The amperage of the charge controller is the only factor for its selection and is calculated using the formula given as [10, 11]:

$$I_{charger} = \frac{P_{array}}{V_s} \tag{30}$$

For gallium Arsenide solar cells, P_{array} = 1640w, and V_s = 32.3V.

$$\therefore \quad I_{charger} = \frac{1640}{32.3} \text{ x } 1.25 \ = 64\text{A}.$$

2.2.5. Power Distribution Unit Sizing

In this work, only bulk converter will be used to step down the generated voltage to the operating voltages of the loads or components. The components for the design of the converter such as the resistor, inductor, capacitor and the switching device such metal oxide field effect transistor (MOSFET) are gotten from careful calculation based on the components operating current and voltage as well as the voltage input to the converter which the system voltage of the designed system. Bulk converter is a device that is used reducing one dc voltage to higher dc voltage.

Calculation procedure:

(i) Determination of Load Resistance: The load resistance is calculated using the formula given as [12]:

$$R_L = \frac{V_{out}}{I_{out}} \tag{31}$$

Where V_{out} = Load or satellite onboard subsystems operating voltage, I_{out} = Load or satellite onboard subsystems operating current.

(ii) Determination of Inductor Size: The inductor is used for storing energy. The size of the inductor is calculated from the duty cycle of the system. Duty cycle of a bulk converter is the ratio of the voltage output of the converter (i.e. load operating voltage) to the voltage input of the converter (i.e. system voltage). Duty cycle of a bulk converter is calculated as [13]:

$$D = \frac{V_{out}}{V_{in}} \tag{32}$$

The inductor size is gotten using the formula given as:

$$L = 10 \text{ x } L_{\min} \tag{33}$$

where $L_{\min}$ is the minimum inductance of the inductor, which is also calculated using the formula as [14]:

$$L = \frac{(1-D)R_L}{2f} \tag{34}$$

where f is the switching frequency of the switching device. The switching frequency is always preferred to be larger in order to have smaller inductors and capacitors for the design.

(iii) Determination of Capacitor Size: The converter capacitor is calculated using the formula given as [15]:

$$C = \frac{(1-D)}{8 \text{ x } L \text{ x } \Delta V_{out} \text{ x} f^2} \tag{35}$$

where ΔV_{out} is the ripple factor voltage output or load operating voltage. It is calculated using the equation given as [16]:

$$\left.\begin{array}{r} \Delta V_{out} = \textit{Ripple factor} \text{ of } \textit{about } 0.85\% \ (\textit{always less than } 1\%) \\ \times \textit{ output voltage or load voltage } (V_{out}) \end{array}\right\} \tag{36}$$

The 0.85% used in this design is to reduce the ripple factor voltage output thereby increasing the capacitance of the capacitor to enable it store more energy but still mindful of the size which does not allow us use lesser value than 0.85% due to size constrain on the satellite.

$$\therefore \ \Delta V_{out} = \frac{0.85 \text{ x } \textit{load voltage } (V_{out})}{100} \tag{37}$$

In this design, the loads will operate on three “lines” or voltage path as well as three currents. Line one load will operate at 3.3V dc voltage and 0.1A dc current, Line two load will operate at 5V dc voltage and 1.6A dc current and Line three load will operate at 12V dc voltage and 4A dc current. Also, the switching frequency used in this design work is 25kHz. The higher value of the switching frequency is to allow the use of smaller size capacitor and inductor since size is an important factor in the design of any satellite. Ripple factor is always less than 1%, thus 0.85% is used in this design. The parameters are substituted accordingly and the result in Table 3 was gotten.

3. Summary of Results and Discussions

The electrical power subsystem of a low earth orbit satellite which will orbit at the altitude of 650km above the earth’s surface and with an average power of 800W has been designed using direct energy transfer method with mathematical computations. The satellite will spend 8 operational years in space using extendable solar array. The satellite orbital period is 1.63hrs and will spend 35.5 minutes and 62.5 minutes under eclipse and sunlight respectively. Table 1 shows the calculated parameters for the power generation unit (Gallium Arsenide cell). The result showed that the total array power of 1,640W will be required from the solar array. Based on this array power, it was observed that the total number of cells needed is 9647, consisting of 254 cells in parallel and 38 series (string) cells. The total mass of the solar array was 65.5kg, while the solar array will have the area of 12m^2. Also, from the result, the end of life power of the satellite was146.3143W/m^2 while the Beginning of Life power was182.88167W/m^2 for the 8 years. The reduction of the power was as a result of the yearly percentage degradation of 2.75% that the solar cell experienced for the period of operation. In order to reduce the effect of degradation, proper insulation of the solar cells is required before launching of the satellite.

Table 1. *Gallium Arsenide solar array parameters.*

Parameter	Value
Solar array required power, P_{array}	1,640W

Parameter	Value
Solar array ideal power, P_{ideal}	258.02W
Percentage degradation per year	2.75%
Beginning of life power, P_{BOL}	182.88167W/m^2
Life degradation, L_d	0.8
End of life power, P_{EOL}	146.3143W/m^2
Array area	12m^2
Total number of cells (m × n)	9647 cells
Number of cells per string, n	38 cells
Number of strings for the array, m	254 cells
Solar array mass, M_{array}	65.5kg
System voltage, V_{sa}	32.3V
System current, I_{sa}	50.8A
Dissipated power	15.79373W/m^2

The battery used for this work is Nickel hydrogen battery, and the calculated parameters areas shown in Table 2. The design was done with the 80% cell efficiency and the depth of discharge of 40%. The designed result showed that the storage unit (batteries) is to deliver a total energy of 592W-hr to the load when the solar array energy is no longer available for the operation of the electrical and electronic subsystems of the satellite. From the Table 2, it can be observed that 28 batteries of 1.25V battery voltage and 53A-hr capacity will be needed for the power storage. Also, it was shown that the mass of the storage unit of the electrical power subsystem of the satellite is 62kg. The charge and discharge currents of the backup batteries are 20.29A and 28.57A respectively.

Table 2. *Nickel hydrogen battery parameters.*

Type of battery	Nickel hydrogen (NiH_2)
Energy to be delivered by the battery, E_{bat}	592 Whr
Depth of discharge (DOD)	40%
Battery efficiency, η_{bat}	80%
Selected battery voltage	1.25V
Battery Capacity, C	53Ahr
No of batteries	28
Specific energy	30Whr/kg
Mass of battery	62kg
Charge current rate	21.14Ahr
Charge current, I_{ch}	20.29A
Discharge current, I_{dis}	28.57A

In addition, the charge controller of amperage, 64A, will be needed for the batteries charging regulation. Table 3 shows the calculated parameters of the electrical power distribution unit. The electrical power distribution unit was designed using bulk converter. The satellite components will operate on three different currents of 0.1A, 1.6A and 4.0A, and the voltages of 3.3V, 5.0V, 12V respectively. With these, the inductor and capacitor to be used with the current and voltage of 0.1A and 3.3V are 5.926×10^{-3}H and 1.080×10^{-6}F, current and voltage of 1.6A and 5.0V are 5.28×10^{-4}H and 7.529×10^{-6}F and current and voltage of 4.0A and 12.0V are 3.33×10^{-4}H and 3.268×10^{-6}F. This implies that three different power distribution systems will be used, each across the three different distribution lines.

Table 3. *Bulk converter (Power distribution) components parameters for Gallium Arsenide solar cell.*

V_{out} (V)	I_{out} (A)	R_L (Ohm)	V_{in} (v)	D	f (kHz)	L (H)	C (F)
3.3	0.1	33.000	32.3	0.102167	25	0.005926	1.080 x10^{-6}
5	1.6	3.125	32.3	0.154799	25	0.000528	7.529 x10^{-6}
12	4.0	3.000	32.3	0.371517	25	0.000377	3.268 x10^{-6}

4. Conclusion

The design of electrical power subsystem of a low earth orbit satellite which will orbit at 650km altitude have been done using direct energy transfer method. The power generation unit, storage unit, regulation unit and power distribution unit have been designed using mathematical method. The result of the sizing of the solar cells, batteries, charger controller and bulk converter is shown in Tables 1-3. From the results, it can be concluded that the sizing of electrical power subsystem of a satellite especially the generation, storage and regulation units depends on the satellite life span in orbit and the average electrical power of the satellite.

As a future direction, while this work was done using Gallium Arsenide solar cell for power generation unit, and Nickel hydrogen battery for power storage unit, other solar cells and batteries can be used for the same satellite average power, orbital period, and same operational years in orbit for future work.

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