Extracting Best Reliable Scheme for Electrical Power Subsystem (EPS) of Satellite

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Abstract— **This paper describes the reliability comparison of different schemes connection for Electrical Power Subsystem (EPS) of satellite. EPS has to be able to provide sufficient power to the satellite subsystems under all possible satellite attitudes. There are eight schemes for EPS, which could be used for power supplying. In these schemes, two main system groups are Peak Power Tracking (PPT) and Direct Energy Transfer (DET) systems. Also in each system, we have four different connections which are Unregulated Bus Using Parallel Batteries, Unregulated Bus Using Linear Charge Current Control Recharge Control, Quasi-Regulated Bus with Constant Current Chargers and Systems Using a Fully Regulated Bus. In this paper we will compare the reliability of different schemes and will introduce the best reliable connection in these two systems for ESP.**

Keywords-reliability; peak power trackimg; direct energy transfer; unregulated bus; quasi-regulated bus

I. INTRODUCTION

The power subsystem generates power, conditions and regulates it, stores it for periods of peak demand or eclipse operation and distributes it throughout the satellite. The power subsystem may also need to convert and regulate voltage levels or supply multiple voltage levels. It frequently switches equipment on or off therefore to increase the reliability, we should protect it against short circuits and isolates faults. This subsystem design is also influenced by space radiation, which degrades the performance of solar cells. Finally, battery life often limits the satellite's lifetime.

As we know, satellite will not receive any power from outside during solar eclipse and it will charge the batteries every time it receives enough power from the solar cells. Therefore EPS should be more reliable. The principle of EPS is provision of electrical energy and characteristic of it is power and voltage stability.

A satellite can cease due to failure or because it has reached to the end of its lifetime. In this respect it is necessary to distinguish between the reliability and the life time of a satellite. Reliability is a measure of the probability of failures and depends on the reliability of the equipment and any schemes to provide redundancy. Effect of failures can be reduced and reliability raised by changing the design scheme, selecting more reliable hardware or adding redundant hardware and software to the system.

Reliability is a parameter under the designer's control. We should consider its potential effect on the EPS design by examining failures from wear out and random causes. In other words, we should identify the schemes for EPS, in which to eliminate or limit failures to acceptable levels. Propellant supply and battery cycle life are examples of these components which should be more reliable in the EPS.

Searching for and identifying the ways in which equipment can fail is a basic part of design for reliability. This process, called Failure Modes Effects and Criticality Analysis (FMECA) assumes that we can identify the ways in which equipment can fail and analyze the effect. The key to this process is identifying and eliminating single point failure modes failures that by themselves can shut down the EPS and kill satellite mission. If we cannot eliminate them, we must minimize their probability of occurrence during the satellite mission.

As shown in Fig. 1, the electrical power subsystem (EPS) would breakdown to four blocks of provision, storage, distribution and regulation & control satellite electrical power.

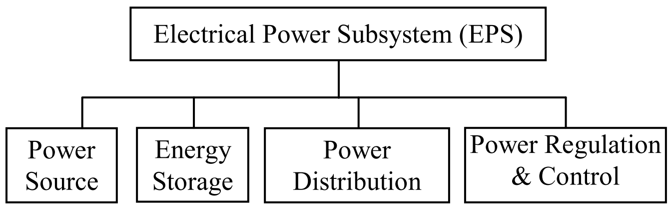

Figure 1. Functional Block Diagram of Power Subsystem.

The most important requirements are the supply of average and peak electrical power. First of all, we should identify the electrical power loads of satellite mission at Beginning-Of-Life (BOL) and End-Of-Life (EOL). In many missions, the EOL power demands must be reduced because solar array performance will degrade at EOL. Usually to obtain peak power requirements for attitude control, payload, thermal and EPS (when charging the batteries), we multiply average power

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by 2 or 3. Fortunately, all the subsystems of satellite do not require peak power at the same time during the mission.

To design for reliability, we can analyze the failure modes of our equipment in several ways. For example, the all-part method simply analyzes each of the satellite’s parts to determine the effect of its failure. On a large satellite this method is a lot of work but is straightforward and easy to do. The all-part method requires us to analyze shorts and opens systematically searching for wires or printed traces on circuit boards that can cause failure if opened or shorted together. We can also use scenarios to find potential failure modes. To do so, we simulate the spacecrafts launch, deployment, and operation to ensure that telemetry can detect failures and that the command system can correct them. This simulation normally occurs when operational procedures are being prepared, but it can more effectively detect design errors if done earlier.

Another way to identify failure modes is the jury method. In many cases new designs do not have a lot of experience behind them, but people have had experience with similar equipment. We can poll them and use their experience to identify likely failure modes and probable effects [2].

As a background for reliability electrical power of satellite, in [3] the reliability analysis has been carried out for power subsystem of L.E.O. Satellite only for Direct Energy Transfer (DET) systems. Also [4] optimized the power subsystem design which may not be more reliable. In [5] a new for a satellite power distribution and control subsystem has been presented which as it is clear, power distribution has a more participation at unreliability of power subsystem which should consider. The paper is organized as follows. Section II describes the EPS of satellite according to Fig. 1. In section III Reliability for EPS will review and discuss. Section IV will search the reliable scheme for EPS. Finally we will conclude the paper.

II. DESCRIPTION OF EPS

EPS contains four blocks as Fig. 1 which will describe in this section.

A. Power Source

The power source generates electrical power within the satellite. At the primary, batteries use as the power source for electrical loads, but batteries alone are too massive for missions that take from weeks to years. Therefore missions need a source that can generate electrical power over many orbital cycles to support electrical loads and recharge the batteries.

Generally, we use four different types of power sources for satellite. Photovoltaic solar cells are the most common power source for satellite which converts solar radiation to electrical energy. Static power sources (nuclear reactor) use a heat source for electrical power provision, typically plutonium-238 or uranium-235. Dynamic power sources also use a heat source to produce electrical power using the Brayton, Stirling, or Rankine cycles, typically concentrated solar radiation, plutonium-238, or enriched uranium. The last power source is fuel cells that used on manned space missions such as Gemini, Apollo, SkyLab and the Space Shuttle. The most common static power source for satellite is the thermoelectric which uses the temperature gradient between the p-n junction of individual thermoelectric output. Power efficiencies for a thermionic power conversion are typically 10-20%.

In contrast to static sources, dynamic power sources use a heat source and exchanger to drive an engine in a thermodynamic power cycle. The heat source may be concentrated solar energy, radioisotopes or a controlled nuclear-fission reaction. As stated, dynamic power sources use one of three methods to generate electrical power, Stirling cycle, Rankine cycle or Brayton cycle. Power-conversion efficiencies for Stirling engines are 25-30%, Rankine cycle l5-20% and Brayton cycle 20-35%. Fuel cells convert the chemical energy of an oxidation reaction to electricity. The energy conversion efficiency of fuel cells can run as high as 80% for low current and 50-60% for high current draws. However in comparison to other power sources, fuel cell efficiencies are high [1].

Although photovoltaic sources are not attractive for interplanetary missions because solar radiation decreases, they often use for low-power missions (less than 15 kW).

B. Energy Storage

Energy storage is an integral part of the satellite's electrical-power subsystem to provide all the power of mission. Any satellite that uses photovoltaics or solar thermal dynamics as a power source should have a system to store energy for peak power demands and eclipse periods. Energy storage normally occurs in a battery. A battery consists of individual cells that are connected in series. The number of cells will determine by the bus-voltage. As we know, batteries can be connected in series to increase the voltage or in parallel to increase this current output.

We should provide a stable voltage for all operating conditions during the mission life, that difference in energy-storage voltage between end of charge and end of discharge often determines the range of bus voltage. To design energy storage capability some issues should be considered such as : size, weight, configuration, operating position, static and dynamic environments, voltage, current loading, duty cycles, number of duty cycles, activation time, storage time and limits on depth of discharge cost, shelf and cycle life, mission, reliability, maintainability and produceability.

Depth Of Discharge (DOD) will define as the percent of total battery capacity will empty during a discharge period. Higher percentages imply shorter cycle life. If we know the number of cycles and the average depth of discharge, we can determine the total capacity of the batteries.

C. Power Distribution

A satellite's power distribution system includes of cabling, fault protection and switching gear to turn power on and off to the satellite loads. It also has the command decoders to command specific load relays on or off. Power distribution will design for various power systems depend on the source characteristics, load requirements and subsystem functions. In selecting a type of power distribution, we should focus on the minimum power losses and mass while maximum reliability, survivability and power quality.

Power switches are usually mechanical relays because of their proven flight history, reliability, and low power dissipation. Solid-state relays, which are made by Metal-Oxide Semiconductor Field Effect Transistors (MOSFET), will be used.

The load profile of a satellite is a key determining factor in the design specifications of a power distribution subsystem. Spacecraft loads such as radar, communication system, motors and computers may require dc, single phase ac or three-phase ac which all should get from power dc bus. Because the regulation for these loads varies, the bus voltage may need further regulation. Therefore transient behavior within a load may produce noise that the distribution system send it to other loads and will be harmful for other components.

The converters typically should isolate the load from the noise on the bus and regulate the power provided to the load against disturbances from the load and the bus. They also should keep load failures from damaging the power distribution system and provide on/off control to desired loads.

Systems for distributing power on satellites have dc power because satellite generates direct current power. Conversion to ac would require more electronics, which would add mass to the EPS. Power distribution systems are either centralized or decentralized, depending on the location of the converters. The decentralized approach places the converters at each load separately, whereas the centralized approach regulates power to all spacecraft loads within the main bus. The decentralized approach implies an unregulated bus because distributed converters regulate power. A regulated power bus typically has some power converters at the load interface because electronics may require different voltages (+5, ±12 Vdc). An advantage of the centralized system is that we do not have to modify design the EPS for different applications. Usually larger satellite with high power levels uses the decentralized distribution systems, with an unregulated bus.

Fault protection within the EPS focuses on detection, isolation and correction of faults. Its main purpose is to isolate a failed load that could eventually cause loss of the mission or the spacecraft. A failed load typically implies a short circuit, which will draw excessive power. If this condition continues, the failed load may stress cables and drain the energy storage reserve. Typically, we would isolate these faults from the EPS bus with fuses. Most satellite power loads have some sort of fuse in series with the power bus to isolate faults. Of course, if the mission requires us to know where load faults occur, we can add fault detection circuits.

D. Power Regulation and Control

The energy source determines how we regulate a satellite's power. Here we will examine power regulation emphasizing solar panels. Power regulation divides into three main categories: controlling the solar array, regulating bus voltage and charging the battery.

We must control electrical power generated at the array to prevent battery overcharging and undesired satellite heating. The two main power control techniques, illustrated in Fig. 2, are a Peak Power Tracker (PPT) and Direct Energy Transfer (DET) systems. A PPT is a non-dissipative system because it extracts the exact power that satellite requires up to the array's peak power. The DET system is a dissipative system because it dissipates power not used by the loads. However a DET subsystem can dissipate this power at the array or through external banks of shunt resistors to avoid internal power dissipation. DET systems commonly use shunt regulation to maintain the bus voltage at a predetermined level. Fig. 2 illustrates the main functional differences between varying PPT and shunt-regulated DET sub-systems.

A PPT is a dc-dc converter operating in series with the solar array (SA). Thus, it dynamically changes the operating point of the solar array source to the voltage side of the array and tracks the peak power point when energy demand exceeds the peak power. It allows the array voltage to swing up to its maximum power point and then the converter transforms the input power to an equivalent output power, but at a different voltage and current. Solar source characteristics permit us to extract large amounts of power when the array is cold (post eclipse) and at the beginning of life. A PPT replaces the shunt regulation function by backing off the peak power point of the arrays toward the end of the battery's charging period. Because the PPT is in series with the array, it uses 4-7% of the total power. A PPT has advantages for missions less than 5 years that require more power at BOL than at EOL.

For direct energy transfer systems a shunt regulator (SR) operates in parallel to the array and shunts the array current away from the subsystem when the loads or battery charging do not need power. Power subsystems with shunt regulation are extremely efficient. They dissipate little energy by simply shunting excess power at the array or through shunt resistor banks. A shunt regulated subsystem has advantages such as fewer parts, lower mass and higher total efficiency at EOL.

Techniques for controlling bus voltage on electrical power subsystems fall into three categories as unregulated, quasi-regulated or fully regulated. Fig. 2 illustrates the main differences between these techniques. An unregulated subsystem has a load bus voltage that varies significantly. The bus voltage regulation derives from battery regulation which varies about 20% from charge to discharge. In an unregulated subsystem, the load bus voltage is the voltage of batteries.

Quasi-regulated subsystems regulate the bus voltage during battery charge but not during battery discharge. A battery charger is in series with each battery or group of parallel batteries. During charge the bus voltage fixes at a potential several volts above the batteries. As the batteries reach full charge, the drop across the chargers decreases but the bus voltage is still constantly regulated. The bus becomes unregulated during discharge when the voltage is about a diode drop lower than the batteries and decreases as the batteries further discharge. A quasi-regulated power subsystem has low efficiency and high electromagnetic interference if used with a PPT.

The fully regulated power subsystem is inefficient, but it will work on a satellite that requires low power and a highly regulated bus. This subsystem employs charge and discharge regulators. We can design the regulators so the charge regulator uses linear technology and the discharge regulator is a

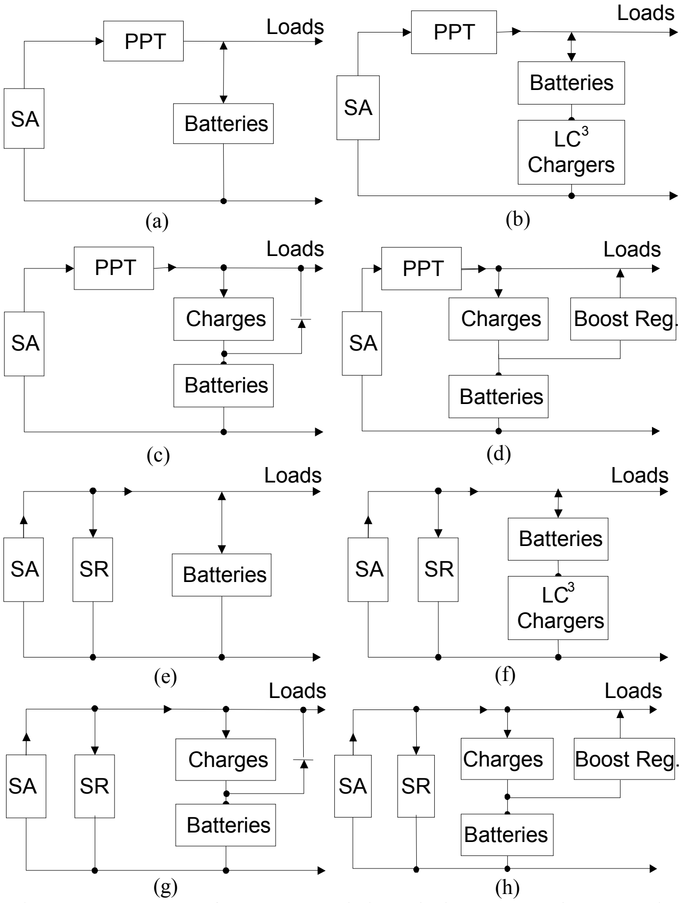


Figure 2. Tecnniques for Power Regulation. The basic approaches are Peak Power Tracking (PPT), which places a regulator in series with the solar arrays and the load, and Direct EnergyTransfer (DET), which uses a regulator in parallel with the solararrays and load.

switching converter, but for best efficiency both should be converters. The advantage of this type of power subsystem is that when we connect it to the loads, the system behaves like a low impedance power supply, making design integration a simple task. But it is the most complex type of power subsystem with an inherent low efficiency and high electromagnetic interference when used with a PPT or boost converter.

We can charge batteries individually or in parallel. A parallel charging system is simpler and has the lower cost, but does not allow flexibility in vehicle integration. It can also stress batteries that they degrade faster. When batteries are charged in parallel, the voltage is the same but the current and temperature are not. Parallel batteries eventually end up balancing out; therefore we could use them for missions under five years. To ensure a battery life greater than five years, we should seriously consider independent chargers, such as the Linear Charge Current Control (LC^3) design in Fig. 2.

III. Reliability of EPS

When a bulb in our desk lamp burns out, it is easily replaced. When the switch that controls the bulb fails, the replacement is not quite as simple but still within the capabilities of most people. We expect a higher reliability of the switch than of the lamp because it requires more effort to repair a failure. When a satellite EPS fails on orbit what we can do? The EPS has to be much more reliable than the light bulb or the switch on the desk lamp. Therefore EPS need a very high reliability in all parts.

The elementary expression for the reliability of a single item, not subject to wear out failures is

$$R = e^{-\lambda t} \tag{1}$$

where λ is the failure rate and *t* is the time. Here *R* is the probability that the item will operate without failure for time *t* or success probability. Therefore the probability of failure, *F*, is given by:

$$F=1-R \tag{2}$$

For the EPS which made up of non redundant elements, all equally essential for correct EPS operation, the system (or series) reliability, Rs, or success probability is computed as

$$R_s = \prod_1^n R_i = e^{-\Sigma \lambda_i t} \tag{3}$$

where *Ri* (*i = 1 . . .n*) is the reliability and λ_i the failure rate of the individual elements.

For failure probabilities (λ_t) less than 0.1 or reliability greater than 0.9, the following approximation is frequently used

$$e^{-\lambda t} \approx 1 - \lambda t \tag{4}$$

Most reliability computations, particularly prior to detailed design, use failure probabilities (which can be summed) rather than reliability values (that must be multiplied).

Where a system consists of *n* elements in parallel and each of these elements can by itself satisfy the requirements, the parallel reliability, *Rp*, is given by

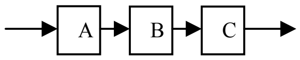
$$R_p = 1 - \prod_1^n (1 - R_i) \tag{5}$$

If we assume the reliability of the parallel elements is equal R_a, the above equation simplifies to

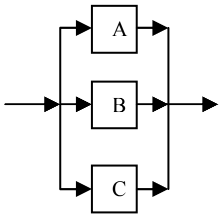
$$R_P = 1 - (1 - R_a)^n \tag{6}$$

Examples of series and parallel structures are shown in Fig. 3.

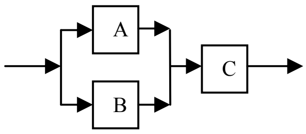
In global applications it is customary to distinguish between active and inactive failure rates, the latter being about one tenth of the active rates. This reduction accounts for the absence of electrical stress when a component is not energized. However the high reliability requirements of the space environment cause components to be safe so that the failure probability due to electrical stresses even in the active mode is quite small. The distinction between active and inactive failure rates is therefore much less important for satellite.



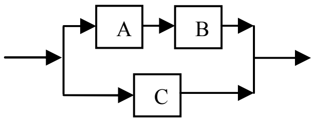
$R_S = R_A.R_B.R_C$
CASE 1
Series Reliability



$R_S=1-(1-R_A).(1-R_B).(1-R_C)$
CASE 2
Parallel Reliability= Full Redundancy



$RS=R_C.[1-(1-R_A).(1-R_B)]$
CASE 3
Partial Redundancy



$RS=1-(1-R_A.R_B).(1-RC)$
CASE 4
Non-identical, Full Redundancy

Figure 3. Series and Parallel Reliability Models. R_S is the system reliability. R_A, R_B and R_C denote the reliability of the A, B and C components respectively.

IV. Finding Best Reliable Scheme

Failure Mode and Effects Analysis (FMEA) or FMECA is an analysis technique which facilitates the identification of potential problems in the design or process by examining the effects of lower level failures.

Recommended actions or compensating provisions are made to reduce the likelihood of the problem occurring and mitigate the risk, if in fact, it does occur.

The FMEA determines, by failure mode analysis, the effect of each failure and identifies single failure points that are critical. It may also rank each failure according to the criticality of a failure effect and its probability of occurring. The FMECA is the result of two steps:

- Failure Mode and Effect Analysis (FMEA)
- Criticality Analysis (CA)

A. Criticality Analysis

The purpose of the Criticality Analysis is to rank each failure mode as identified in the FMEA, according to each failure mode's severity classification and its probability of occurrence. MIL-STD-1629 is an excellent data source for the implementation of a Criticality Analysis. The result of the Criticality Analysis will leads itself to the development of a Criticality Matrix. The failure mode criticality number for each specific failure mode (C_m) is calculated as follows:

$$C_m = \beta.\alpha.\lambda_p.t \tag{7}$$

where C_m is failure mode critically number, β is conditional probability of failure effect, α is failure mode ratio, λ_p is part failure rate per million hours and t is duration of the relevant mission phase (operation) e.g. 20 hours.

The criticality number of each assembly (or system) is calculated per each severity category. This criticality number is the sum of the specific failure mode criticality numbers related to the particular severity category:

$$C_r = \sum_{n=1}^{j}\left(\beta.\alpha.\lambda_p.t\right)_n \tag{8}$$

where n is the current failure mode of the item being analyzed and j is the number of failure modes for the item being analyzed. The resulting FMECA analysis will enable a criticality matrix to be constructed. The criticality matrix displays the distribution of all the failure mode criticality numbers according to the severity category and referring to the criticality scale. According to MIL-STD-1629 the scale is divided into five levels:

- Level A – Frequent, is defined as a probability which is equal or bigger than 0.2 of the overall system probability of failure during the defined mission period.
- Level B - Reasonable probable, is defined as probability which is more than 0.1 but less than 0.2 of the overall system probability of failure during the defined mission period.
- Level C - Occasional probability, is defined as a probability, which is more than 0.01 but less than 0.1 of the overall system probability of failure during the defined mission period.
- Level D - Remote probability, is defined as a probability, which is more than 0.001 but less than 0.01 of the overall system probability of failure during the defined mission period.
- Level E - Extremely unlikely probability, is defined as probability which is less than 0.001 of the overall system probability of failure during the defined mission period.

B. Severity Classification

A severity classification category assigned to each failure mode depending upon its effects of an equipment and/or system operation. The severity classification are consistent between MIL-STD-1629 and MIL-STD-882, and are listed below

- Category I - Catastrophic - A failure which may cause death or weapon system loss (i.e. aircraft, tank, etc.)
- Category II - Critical - A failure which may cause severe injury, major property damage, or major system damage which will result in a mission loss.
- Category III - Marginal - A failure which may cause minor injury, minor property damage, and minor system damage which will result in a delay or loss of availability or mission degradation.
- Category IV - Minor - A failure not serious enough to cause injury, property damage or system damage, but which will result in unscheduled maintenance or repair.

C. Applying FMECA to EPS Schemes

According to Fig. 2 we assume our power subsystem may implement with 2 main categories PPT and DET, which each category have four different types.

According to the Table I, we calculated failure rate, failure mode, failure mode ratio, severity, failure effect probability and C_m of different scheme of EPS to decide the best reliable scheme. For our calculations, we have used the (1) to (8) and inserted results into the Table I.

TABLE I. DEFINITION OF PARAMETERS FAILURE RATE, FAILURE MODE, FAILURE MODE RATIO, SEVERITY, FAILURE EFFECT PROBABILITY AND C_M OF DIFFERENT SCHEME OF EPS.

Item	λp	Failure Mode	α	β	System Effect	Severity	C_m
PPT	0.02	Open	0.20	0.30	Life time >1 week	II	0.001
		Open	0.20	0.70	Life time < 1 week	I	0.002
		Short	0.80	0.90	Life time > 1 year	IV	0.014
		Short	0.80	0.10	Life time < 1 year	III	0.001
SA	0.40	Open	0.30	0.30	Life time >1 week	II	0.036
		Open	0.30	0.70	Life time < 1 week	I	0.084
		Short	0.70	1	Unable to work	I	0.28
		Short	0.70	0	---	---	---
SR	0.15	Open	0.20	1	No effect	IV	0.03
		Open	0.80	0	----	---	---
		Short	0.40	1	Unable to work	I	0.06
		Short	0.60	0	---	---	---
Batteries	0.30	Open	0.15	0.25	Life time < 1 month	II	0.011
		Open	0.15	0.75	Life time > 1 month	III	0.033
		Short	0.85	1	Unable to work	I	0.255
		Short	0.85	0	---	---	---
LC^3	0.03	Open	0.40	0.25	Life time < 1 month	II	0.003
		Open	0.40	0.75	Life time > 1 month	III	0.009
		Short	0.60	1	No effect	IV	0.018
		Short	0.60	0	----	---	---
Charges	0.20	Open	0.35	0.20	Life time > 1 year	IV	0.014
		Open	0.35	0.80	Life time < 1 year	III	0.056
		Short	0.65	1	No effect	IV	0.13
		Short	0.65	0	---	---	---
Boost Reg.	0.05	Open	0.10	1	No effect	IV	0.005
		Open	0.10	0	---	---	---
		Short	0.90	0.20	Life time > 1 year	IV	0.009
		Short	0.90	0.80	Life time < 1 year	III	0.036

In Table I, we assume two conditions for each component which is open circuit or short circuit, although they may be in the middle state, but generally we have these two conditions.

Table II, shows the criticality number C_r for each component of Table I.

TABLE II. CALCULATION OF CRITICALITY NUMBER OF PPT, SA, SR, BATTERIES, LC^3, CHARGES AND BOOST REGULATOR.

Item	C_r
PPT	0.02
SA	0.40
SR	0.09
Batteries	0.30
LC^3	0.03
Charges	0.20
Boost Regulator	0.05

According to some relation that mentioned in Fig. 3, we will calculate the final reliability schemes to find the best of them. In Table III the rank number of each diagram have been shown.

TABLE III. BEST RELIABLE SCHEMES ACCORDING TO FIG. 2.

Item	Reliability Criteria	Rank No.
(a)	0.8764	5
(b)	0.8677	7
(c)	0.8187	8
(d)	0.8735	6
(e)	0.9892	2
(f)	0.9884	3
(g)	0.9842	4
(h)	0.9925	1

As we can see in Table III, the diagram (h) in Fig. 2 has a best reliability between eight different diagrams named DET using a fully regulated bus.

V. CONCLUSION

Best reliable EPS configuration of satellite is introduced in this paper. In this study, to find the reliable EPS configuration, for two main PPT and DET configuration with four different connections, failure mode critically number and then with respect to critically number, reliability criteria is calculated.

According to the results, we found that the DET diagrams are more reliable than PPT diagrams and between DET diagrams, the fully regulated bus has the best reliability.

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REFERENCES

[1] J. Wertz and W. Larson, Space Mission Analysis and Design, Boston, Kluwer Academic Publishers, 1991.

[2] B. W. Johnson, Design and Analysis of Fault-Tolerant Digital Systems, Addison-Wesley Publishing Company, 1989.

[3] M. Zahran, S. Tawfik and G. Dyakov, “L.E.O. Satellite Power Subsystem Reliability Analysis,” in Journal of Power Electronics, vol. 6, no. 2, pp.104-113, 2006.

[4] M. Kordon, G. Klimeck, D. Hanks and H. Hua, “Evolutionary computing for spacecraft power subsystem design search and optimization,” IEEE Aerospace Conference Proceedings, Big Sky, MT., March 2004.

[5] H. I. Mohammed H. N. Ahmed A. E. Eliwa W. Sabry R. Mostafa, “Innovative Design of a Cube Satellite Power Distribution and Control Subsystem,” The Online Journal on Electronics and Electrical Engineering (OJEEE), vol. 1, no. 1, Reference Number JO-0002.