Electrical Power System for Low-Earth-Orbit Spacecraft Applications

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The results of a trade off study to select an electrical power system (EPS) configuration that is best suited for low-Earth-orbit (LEO) spacecraft with various output power capabilities are presented. The selection is based on parameters like weight, volume, and costs of the power system components. Two EPS configurations used for LEO spacecraft, i.e., one using a direct energy transfer (DET) approach, and the other using a peak power transfer (PPT) approach, are described in detail and compared. At a spacecraft power requirement of 800 W or less, it is highly advantageous to choose the PPT approach to the design of an EPS for LEO spacecraft. However, for a spacecraft in higher orbit and/or with higher power requirements, the DET-based power system is the optimum choice.

I. Introduction

VARIOUS electrical power system (EPS) configurations have been used for spacecraft in low Earth orbit (LEO), and they can be broadly divided into two types, i.e., 1) one using a direct energy transfer (DET) approach, and 2) one using a peak power transfer (PPT) approach. All other EPS configurations are variations, derivations, or combinations of these two types. The main purpose of this study is to optimize or select an EPS configuration that is best suited for LEO spacecraft with various output power capability ranges. As an introduction to this analysis and evaluation, spacecraft power systems utilizing the PPT and the DET approaches are briefly summarized.

II. Spacecraft Electrical Power Systems

The important building blocks of typical spacecraft power systems are shown in Fig. 1, i.e., solar arrays, energy storage batteries, and power processing electronics (PPE). These building blocks can be interconnected in different ways to result in different spacecraft power systems. In any type of spacecraft power system, the outputs of the solar cell array and storage battery are to be conditioned so as to match with the requirements of the various subsystems. The battery has to be charged from the solar cell array during the orbital day and discharged to provide power during the orbital night or when the load demand exceeds the solar cell array capability. All of these functions are carried out by means of PPE.

PPE can be classified into two main types of systems on the basis of their working principle, i.e., 1) PPT systems, which extract the maximum power from the solar cell array and hence dissipate very little power internally; and 2) DET systems, which do not extract maximum power from the solar cell array. In the latter case, all of the available power is not utilized by the loads and by the battery and, hence, any unused power is dissipated by employing shunt regulators or voltage limiters.

A. PPT Electrical Power Systems

The power systems shown in Figs. 2 and 3 use the PPT approach. The main advantages of this approach are fewer thermal problems and improved efficiency of the overall power system.

The maximum power available from the solar array is processed by the peak power tracker, and its output is used to charge the batteries as well as to supply power to the loads. When the batteries are fully charged and/or the load demand is less than the solar array output, the peak power tracker electronically moves the operation of the solar array off the maximum power point, toward the open circuit voltage. As the storage batteries are placed across the bus, the bus voltage varies as the batteries are charged or discharged.

Thus, this system, as shown in Fig. 2, is called an unregulated bus power system. There are various techniques for maximum power transfer; detailed descriptions are given in Ref. 1. In the power system depicted in Fig. 3, an additional regulator is added to regulate the bus, resulting in a regulated

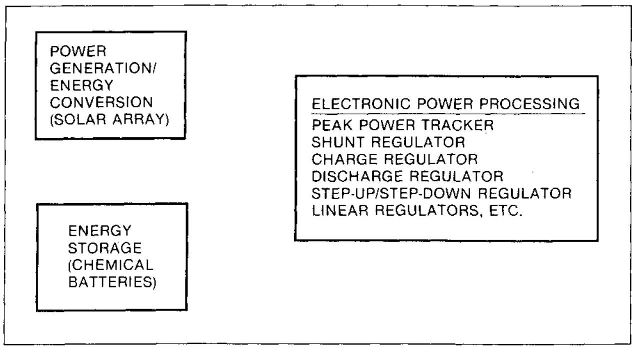

Fig. 1 Building blocks of typical spacecraft electrical power systems.

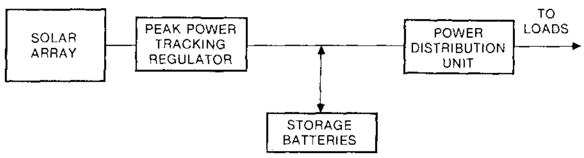

Fig. 2 Unregulated bus power system—PPT approach.

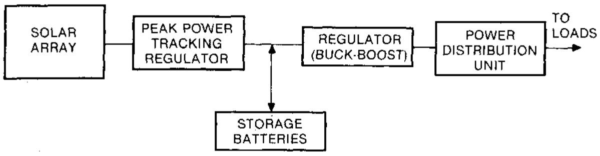

Fig. 3 Regulated bus power system—PPT approach.

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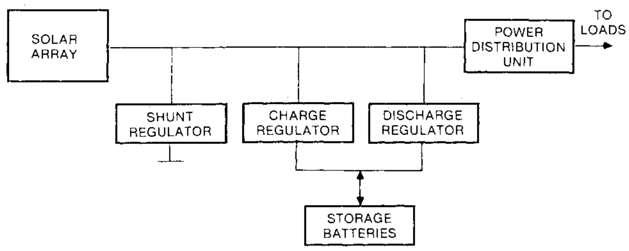

Fig. 4 Regulated bus power system—DET approach.

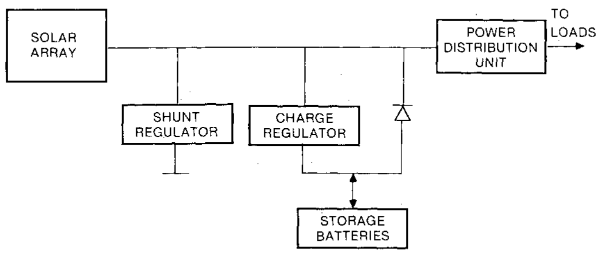

Fig. 5 Unregulated bus power system—DET approach.

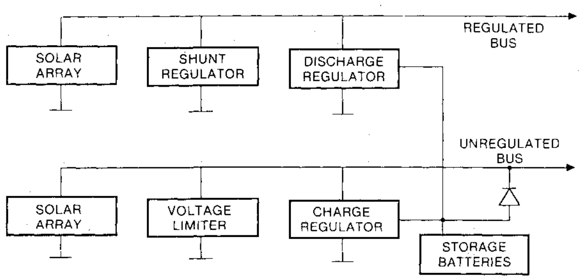

Fig. 6 Partially unregulated bus power system.

Table 1 Advantages and disadvantages of regulated bus power systems

Advantages	Disadvantages
Permits lighter load regulator/converter units	Buffering of units from bus is limited
Some loads may run directly from bus	Dissipation of excess solar array power in shunt regulator
Low bus impedance	Three types of regulators are required
Solar array working point is fixed	Interface problems more severe. More difficult to avoid single point failures.

Table 2 Advantages and disadvantages of unregulated bus power systems

Advantages	Disadvantages
Simple interface	Complex load regulator/ converter units
Units are buffered from noise on bus	Significant weight penalty, especially with input filter if loads must work over a wide bus voltage range
Only two types of regulators are required	Unit switch on surge currents may prevent operation of solar array at maximum power point
Large currents available to blow clear faulty units	
Easier to avoid single-point failure	
Regulation can be tailored to each load separately	

bus power system. Regulated and unregulated bus power systems each have unique advantages and disadvantages, which will be addressed. The NASA Multimission Modular Spacecraft (MMS) Modular Power Subsystem (MPS) uses the PPT approach and is currently flying on three satellites, i.e., SMM, LANDSAT-4, and LANDSAT-5. In addition, it is planned for use on the GRO, UARS, AXAF, TOPEX, and a Department of Defense satellite.

B. DET Electrical Power Systems

The power systems represented in Figs. 4 and 5 use the DET approach. In this approach, the power from the solar array is directly transferred to the load without the use of any series connected regulator or converter, hence the name direct energy transfer. However, to limit or maintain the bus at a predetermined voltage level, a shunt limiter or regulator is employed. Storage batteries are charged using charge regulators. If the bus is of regulated type, then a battery discharge (boost) regulator is also employed to regulate the bus when the batteries are supplying power.

Figure 4 shows the block schematic of a DET regulated bus power system. The shunt regulator maintains the bus voltage at a fixed value while the charge regulator charges the batteries at a constant current during the orbital day. During the orbital night, or when the batteries are used to supplement solar array power, the discharge regulator boosts the voltage of the storage batteries to the bus voltage level and maintains it at this level. Thus, regulated bus power is available continuously. Figure 5 presents the block schematic of a DET unregulated bus power system. The voltage limiter limits the bus voltage at a fixed value while the charge regulator charges the batteries during the orbital day. The storage batteries are connected across the bus via a diode; thus, the bus voltage varies as the batteries charge and discharge. During the orbital night, the storage batteries directly supply power to the loads. Thus unregulated bus power is available continuously.

C. Other Power System Configurations

Some spacecraft power systems use a hybrid approach. As an example, Fig. 6 shows a partially unregulated bus power system. Many variations and derivations of the previously described two approaches and types of power systems have been used.

D. Regulated vs Unregulated Bus Power Systems

The bus voltage can be regulated or unregulated, irrespective of whether the PPE uses the PPT or DET approach. A comparison of these two types of power systems has been performed.[2,3] Table 1 gives the advantages and disadvantages of regulated bus power systems, and Table 2 presents those of the unregulated bus power systems.

E. Power Distribution Using Centralized and Decentralized Regulation Approaches

Whether the spacecraft power bus is regulated or unregulated, the spacecraft subsystems require different positive and negative voltages with varying regulation requirements. Therefore, the bus voltage is further regulated, leveled up, leveled down, and/or inverted using regulators and dc-dc converters. If this process of further regulation, etc., is carried out at each load end separately, then such an approach is termed as a decentralized regulation approach (DRA). In this case, obviously, there is no need to use a regulated main bus, and, hence, a spacecraft using DRA for power distribution usually employs an unregulated main bus. Alternately, if this process of further regulation is carried out in the main power system for all the loads, then such an approach is termed a centralized regulation approach (CRA). Table 3 presents a qualitative comparison of both approaches.

From the study, it appears that CRA is advantageous for spacecraft with average power requirement of about 100 W, while at higher power levels, DRA is advantageous.

III. Analysis

Comparative analyses of EPSs using the DET and PPT approaches with respect to their power extraction capability, weight, volume, cost, etc., are presented in the following sections. For these analyses, it is assumed that the average spacecraft load is constant over the entire duration of the mission and is approximately constant during orbital night and day. As the average load of the spacecraft is constant, the design shall take care of worst-case eclipse duration. Thus, only the maximum eclipse duration is considered for all calculations and analyses. Once the design takes care of maximum eclipse duration, it can then meet the spacecraft load requirement throughout its mission. Thus, the analysis does not consider any other case where the eclipse period can be lower than worst case maximum, and even it can be zero under certain altitude and inclinations during certain periods in a year.

Table 3 Comparison of CRA vs DRA to power distribution

Parameter	Centralized regulation approach	Decentralized regulation approach
Regulation requirements	Compromise	No compromise
Isolation	Needs independent filters at the input of filters	Filters are with the individual converters
Protection	Needs current limiters on each line	Protection is designed into each converter
Preferable	At low power levels and for smaller aircraft	At high power levels and for bigger spacecraft

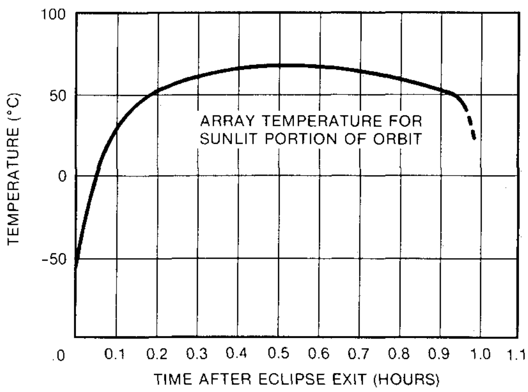

Fig. 7 Solar array temperature profile.

A. Power Output

The output of the solar array varies as a function of temperature and other parameters, i.e., radiation, illumination, angle of sun incidence, aging, etc. A spacecraft in LEO goes through a large number of eclipses compared to a spacecraft in geosynchronous orbit. Whenever the spacecraft comes out of eclipse, the solar array is at very low temperature, and it generates maximum power. As the solar array is heated by the sun, its output decreases. This happens in every orbit. Changes in other parameters that will affect the solar array output are relatively slow. Figure 7 shows a typical temperature profile of a solar array in LEO. The shape of this curve depends on various factors of the panel, including the absorption coefficient (absorbtivity/emissivity), thickness, rigidity, etc. To study the effects of temperature on the solar array, the solar array I-V and P-V characteristics are developed at 10°C steps between the extremes of minimum and maximum temperature limits.

Figure 8 presents the solar array I-V characteristics at three temperatures: −60, 0, and 70°C. As the spacecraft comes out of eclipse, the solar array temperature is very low and, hence, produces higher power. As the temperature of the solar array increases, it produces lower power. Thus, the power available at 0°C is lower than that available at −60°C and the power available at 70°C is lower than that available at 0°C.

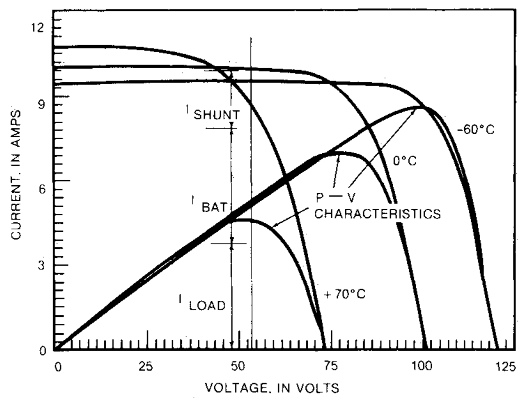

Fig. 8 Solar array I-V and P-V characteristics.

Table 4 Power availability at various operating points at different temperatures (500 km)

Temp, °C	Temp, K	I_{SC}	V_{OC}	V_{MPP}	P_{MPP}	P (51.2)	P (54.4)	P (57.6)
−70	203	9.26	125	105.6	920	475	504	534
−60	213	9.37	121	99.2	889	480	510	540
−50	223	9.47	117	96.0	859	485	515	546
−40	233	9.58	114	92.8	828	490	521	551
−30	243	9.68	110	89.6	796	495	526	557
−20	253	9.78	106	83.2	762	501	532	563
−10	263	9.89	102	80.0	730	505	537	567
0	273	9.99	99	76.8	698	510	541	571
10	283	10.09	95	73.6	664	513	544	572
20	293	10.19	91	70.4	629	516	545	571
30	303	10.30	87	65.6	594	515	542	565
40	313	10.40	83	60.8	560	512	534	551
50	323	10.50	79	57.6	526	502	518	526
60	333	10.60	75	54.4	491	485	491	486
70	343	10.70	72	51.2	456	456	449	426
80	353	10.80	68	48.0	422	414	389	345
90	363	10.90	64	44.8	387	353	308	237

I_{SC} is the solar array short circuit current in amperes
V_{OC} is the solar array open circuit voltage in volts
V_{MPP} is the solar array voltage at maximum power point, in volts
P_{MPP} is the solar array power at maximum power point, in watts
P (51.4) is the solar array power at 51.4 V in watts
P (54.4) is the solar array power at 54.4 V in watts
P (57.6) is the solar array power at 57.6 V in watts

Table 5 Orbital power and energy calculations (500 km)

Time	Temp (Ave), °C	Time duration, h	P (MPP) P (ave), W	P (MPP) E (ave), W-h	P (40 V) P (ave), W	P (40 V) E (ave), W-h	P (48 V) P (ave), W	P (48 V) E (ave), W-h	P (57.6 V) P (ave), W	P (57.6 V) E (ave), W-h
0.01	−55	0.01	874	8.74	483	4.83	513	5.13	543	5.43
0.02	−40	0.01	828	8.28	490	4.90	521	5.21	551	5.51
0.04	−20	0.02	762	15.24	501	10.02	532	10.64	563	11.26
0.06	−2	0.02	704	14.08	509	10.18	540	10.80	570	11.40
0.10	+19	0.04	633	25.30	516	20.64	545	21.80	571	22.84
0.18	+40	0.08	560	44.80	512	40.96	534	42.72	551	44.08
0.26	+55	0.08	509	40.68	494	39.48	505	40.36	506	40.48
0.80	+67	0.54	467	251.91	465	250.94	462	249.26	444	239.76
0.94	+55	0.14	509	71.19	494	69.09	505	70.63	506	70.84
Orbital average energy, W-h				480.22		451.03		456.54		451.60
Orbital average power, W			510.87		479.82		485.68		480.42	

P (ave) is the average power, in Watts
P (MPP) is the power at maximum power point, in watts
P (40 V) is the power at 40 V, in W
P (48 V) is the power at 48 V in W
P (57.6 V) is the power at 57.6 Volts in watts
E (ave) is the average energy, in Watt hours

Also, the open circuit voltage of a solar array decreases with increasing temperature, and the short circuit current increases with increasing temperature.

Table 4 presents the power availability at various operating points on the solar array at every 10°C interval starting from −70 to +90°C. Table 5 presents the temperature, time duration, and the solar array power output at various operating points over an orbit at 500 km altitude. At the bottom of the table is the orbital average solar array output power at different operating points, including the maximum power point.

1. PPT Approach

In general, in any peak power tracker, the peak power point is followed continuously as the temperature of the array changes, thereby extracting all of the power from the solar array. From Table 5, the orbital average peak power is about 511 W.

2. DET Approach

In this approach to the power system design, the solar array is designed such that the end-of-life (EOL) solar array maximum power point at highest temperature occurs at the bus voltage. From the solar array I-V and P-V characteristics of Fig. 8, it is clear that the available current at the bus voltage varies as the temperature varies. Also, the shunt regulator needs a minimum amount of power for its proper operation to maintain the bus voltage at a fixed point. Thus, the useful power available is the power at the highest temperature at EOL, minus the minimum power required by the shunt regulator for its operation.

The operating point voltage on the solar array can be moved to a lower voltage level by reducing the number of series connected cells and increasing the parallel cells in the solar array. The output of the solar array at the optimal operating point is about 450 W.

The advantage or gain provided by the PPT approach is 61 W (about 14%) compared to the DET approach. However, depending upon the hardware technology and efficiencies accompanying each system, the useful power and the apparent relative advantage might be different.

B. Weight and Volume

Assuming that storage batteries and power distribution electronics will be same for the PPT- and DET-based EPS's, the remaining units are considered for weight and volume estimations. The weight and volume of the power system is approximately proportional to the power dissipated in the system. This is true because the higher the power dissipation the system needs, the higher the thermal heat sinking capability, and, hence, the higher the weight. Also, the higher the power dissipation the system needs, the higher the heat sink surface area and, thus, the higher the volume. Thus, the higher the weight and volume, the higher the cost. Power dissipations in power systems using the PPT and DET approaches are computed in the following sections.

1. Assumptions

Assumptions made for these analyses are presented in Table 6.

2. Power System Output

The relationships to calculate the power system outputs are defined as follows:

PPT approach

$$P = P(\text{day}) + P(\text{eclipse}) \left[\left(\frac{T(e)}{T(d)}\right) \left(\frac{\text{BCV}}{\text{BDV}}\right) (\text{n RCH}) \right]$$

DET approach

Charge regulator output:

$$P(\text{ch}) = P(\text{eclipse}) \left[\left(\frac{T(e)}{T(d)}\right) \left(\frac{\text{BCV}}{\text{BDV}}\right) (\text{n RCH}) \right]$$

Shunt regulator output:

$$P(\text{sh}) = P(\text{eclipse}) + \frac{P(\text{ch})}{(n\ \text{ch})} - P(\text{day min})$$

Table 6 Assumptions for analyses

1) Life—3 years
2) Orbit altitude—500 to 2000 km; inclination is 40 deg
3) Battery end of charge voltage (BCV)—1.48 V
4) Battery end of discharge voltage (BDV)—1.15 V
5) Battery recharge factor (n RCH)—1.10
6) Minimum average operating temperature of the solar array for more than 5 min—0°C
7) Peak power tracker efficiency (PPT approach)—90%
8) Charge regulator efficiency—90%
9) Analog portion of the shunt regulator efficiency—90%

where P is the power system output, P(day) the power required by the load during day, P(eclipse) the power required by the load during eclipse, $T(e)$ the eclipse duration in minutes, $T(d)$ the duration of orbital day in minutes, P(ch) the charge regulator output, P(sh) the shunt regulator output, P(day min) the minimum load power during the day, and n ch the charge regulator efficiency.

The dissipation in shunt regulator is computed as follows. The solar array is assumed to be divided into five sections/1000 W of capability. Thus, a 3000 W solar array will be divided into 15 sections. The number of stages in the digital shunt regulator (DSR)[4] will be equal to the number of solar array sections minus one, since one solar array section is connected permanently across the bus. The analog shunt portion of the shunt regulator is rated slightly greater (about 120%) than one solar array section output capability. On the average, each switch is assumed to dissipate about 5 W. Thus, a five-section solar array will have a four-stage DSR and there will be four switches, each dissipating about 5 W. The power dissipation in the shunt regulator is then calculated using the following relationship:

$$
\begin{aligned}
P(\text{dissipation}) &= P(\text{SA})[1/5 + 20/1000] \text{ for } P(\text{SA})\ 0.00 - 1000 \text{ W}\\
&= P(\text{SA})[1/10 + 45/2000] \text{ for } P(\text{SA})\ 1001 - 2000 \text{ W}\\
&= P(\text{SA})[1/15 + 70/3000] \text{ for } P(\text{SA})\ 2001 - 3000 \text{ W}\\
&= P(\text{SA})[1/20 + 95/4000] \text{ for } P(\text{SA})\ 3001 - 4000 \text{ W}\\
&= P(\text{SA})[1/25 + 120/5000] \text{ for } P(\text{SA})\ 4001 - 5000 \text{ W}\\
&= P(\text{SA})[1/30 + 145/6000] \text{ for } P(\text{SA})\ 5001 - 6000 \text{ W}\\
&= P(\text{SA})[1/35 + 170/7000] \text{ for } P(\text{SA})\ 6001 - 7000 \text{ W}\\
&= P(\text{SA})[1/40 + 195/8000] \text{ for } P(\text{SA})\ 7001 - 8000 \text{ W}
\end{aligned}
$$

The minimum load is assumed to be about 10% of the average load.

3. BOL Maximum/EOL Minimum Power Ratios

Assuming an orbital altitude of 500 km and inclination of 40 deg for a 3-yr mission, the total EOL loss factor for the solar array from Table 7 is 0.6363. The BOL/EOL power ratios are then given as follows:

PPT approach: BOL maximum power factors are a diode loss of 0.94, a temperature (0°C) of 1.1176, a maximum sun intensity of 1.035, and an overall loss factor of 1.0873 and the ratio between BOL max/EOL min power is 1.0873/0.6363 = 1.7089.

DET approach: Loss factors, for DET regulated bus operation are a temperature (70°C) of 0.8236, a correction due to 0°C (from solar array curves, see Fig. 8) of 1.129, a diode loss factor of 0.94, a maximum sun intensity of 1.035, and an overall loss factor = 0.9046.

The ratio between BOL max/EOL min power is 0.9046/0.6363 = 1.4217.

Table 7 Solar array degradation and loss factors

Factors affecting solar array EOL power	
Initial losses	
Cell mismatch	0.9900
Measurement errors	0.9900
Wiring and diode losses	0.9400
Temperature (70°C)	0.8236
End of third year	
UV degradation	0.9800
Micrometeorite damage	0.9900
Random failures	0.9900
Summer solstice losses	0.9650
Radiation fluence effects	0.9050
Total loss factor	0.6363

4. Power Dissipation Ratings

Power dissipation ratings for the PPT and DET approaches are computed using the previously given assumptions, data, and relationships. Figure 9 presents the relative power dissipation rating vs the load power for PPT and DET approaches.

At low-load power levels, the power dissipation rating of the PPT-based power system is lower than the DET-based power system. As the load power increases, the power dissipation rating of the PPT-based power system becomes equal to and then exceeds that for the DET-based system.

These calculations are iterated for orbital altitudes of 1000 and 2000 km. Figures 10 and 11 present the relative power dissipation rating vs the load power for PPT and DET approaches for 1000 and 2000 km, respectively.

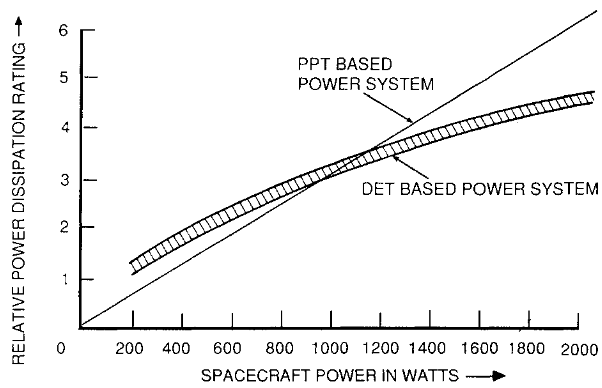

Fig. 9 Power dissipation vs spacecraft power for PPT- and DET-based power systems (500-km altitude).

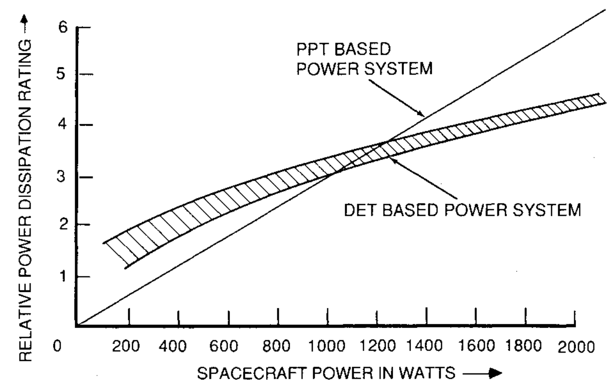

Fig. 10 Power dissipation vs spacecraft power for PPT- and DET-based power systems (1000-km altitude).

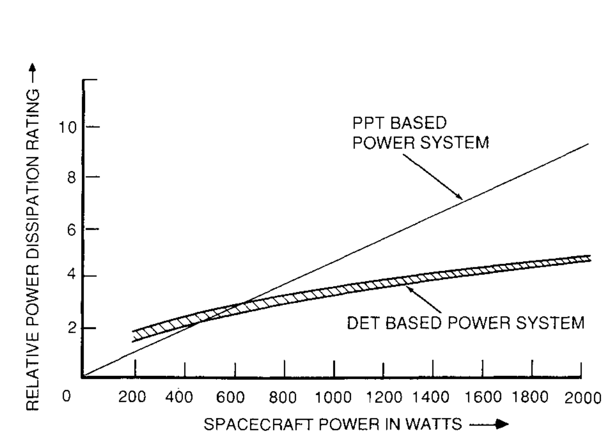

Fig. 11 Power dissipation vs spacecraft power for PPT- and DET-based power systems (2000-km altitude).

Table 8 Comparison of PPT appraoch vs DET approach for electrical power system

Parameter	PPT approach	DET approach
Thermal and mass constraints	Heat dissipation is limited to a low and nearly constant value	In periods of excess power, heat is usually dissipated in shunt regulator which therefore requires heavy sinks
Radiation damage	More power is available at the BOL to operate additional equipment or increase its operating time	Additional power at the BOL is not properly utilized
Solar array design	Solar array voltage can be selected independently and the design is simple	The choice of solar array voltage depends on electronics, battery, thermal design, etc.; array design freeze is delayed
Solar array loading efficiency	Improved array loading efficiency, especially at the beginning of sunlight period when battery voltage is low and array voltage is high	Low
Degradation	Graceful degradation beyond design life	Rapid degradation beyond design life

C. Performance Parameters

Performance parameters for the PPT and DET approaches are compared and presented in Table 8. The solar array for the DET power system is estimated to cost about 15% more than the PPT-based power system. The PPE for the DET-type power system is estimated to cost about 25% more than the PPT-type power system, as the DET system contains shunt and charge regulators in place of a peak power tracker in the PPT power system.

IV. Conclusions

At low-load power levels, the power dissipation rating of the PPT-based power system is lower. As the load power increases, the power dissipation ratings of both power systems increase, but the DET-based system does so at a lower rate. Hence, the power dissipation rating of the PPT-based power system becomes equal to that of the DET-based power system and exceeds it at higher power levels.

In the PPT approach (Fig. 2), the required units for power processing (unregulated bus) are 1) a peak power tracker, and 2) a power distribution unit, whereas in the DET approach (Fig. 5), the required units (unregulated bus) are 1) a shunt regulator, 2) a charge regulator, and 3) a power distribution unit. In a 500-km altitude orbit, the power dissipation rating of the PPT-based power system equals that of the DET-based power system at about a 800-W spacecraft load. However, the weight and volume of the shunt and charge regulators in a DET system will weigh and occupy more than a single PPT-based system dissipating a power equal to the sum of the shunt and charge regulators. Hence, the weight/volume crossover point might move to perhaps 1200 W. Considering the cost aspects of fabrication, testing, etc., for one big unit vs two small units and the smaller solar array for the PPT system, the overall performance (including weight, volume, and cost), the crossover point may move to about a 1400-1500 W spacecraft load range.

The foregoing conclusions are true only if the orbit is in the neighborhood of 500 km and contains eclipses. At higher altitudes, the crossover points move toward lower spacecraft load range. However, the qualitative results are the same in that the PPT-based power system is optimum for spacecraft in 500 km orbit with power requirement levels of about 800 W. As the power or the orbital altitude increases, gradually the DET-based power system performance exceeds the performance of the PPT-based power system.

The DET-based power system performance can be further improved by designing the charge regulator such that the charge regulator itself maintains[5] the bus at a fixed voltage (shunt regulator is off now) and charges the batteries at a variable current utilizing all of the power available from the solar array minus the load. When the batteries are completely charged, the shunt regulator is turned on, and the charge regulator has to handle more power in this mode of operation.

The results of the analyses presented in this paper assume that the average spacecraft load is constant over the entire duration of the mission, and, hence, the design considered worst-case eclipse duration. Thus, for all other eclipse durations that are less than the worst case considered, the load capability of the system is higher than for the worst-case eclipse. However, loads and instruments shall have the capability for programming so that they can use this extra capability.

Sensitivity of the data presented in the Figs. 9-11 directly depends upon the assumptions made in the analysis. As the assumptions can change for each mission and hardware technology, etc., the reader shall treat and use the results of this paper in the qualitative sense and not in the quantitative sense.

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