High Power, High Voltage Electric Power System for Electric Propulsion

Harry Aintablian, Harold Kirkham, Paul Timmerman
Jet Propulsion Laboratory

Abstract

The Modular, Reconfigurable, High Energy (MRHE) spacecraft concept features modules assembled in orbit to meet the demands for routine exploration and cost savings through multiple-unit production. The concept uses advanced solar power generation with direct-drive electric propulsion. The MRHE Power Delivery System (PDS) transfers power from the solar panels to the electric thrusters. In addition, it provides power to a science payload, survival heaters and various subsystems.

This paper provides an overview of the 30 KW, 600 V MRHE power subsystem. Descriptions of the power subsystem elements, the mode of power transfer, and power and mass estimates are presented. A direct-drive architecture for electric propulsion is considered which reduces mass and complexity. Solar arrays with concentrators are used for increased efficiency. Finally, the challenges due to the environment of a hypothetical lunar mission as well as due to the advanced technologies considered are outlined.

1.0 Introduction

The initial phase of the Modular, Reconfigurable, High Energy (MRHE) spacecraft was funded by NASA's office of Exploration System as part of the Human and Robotic Technology tasks. The MRHE technology demonstrator project was to support the development of a cluster of identical satellites assembled in orbit thus offering cost savings. The concept uses identical, solar-powered buses with direct-drive electric propulsion systems. JPL teamed with MSFC to define the project requirements and to undertake the system design.

The Power Delivery System (PDS) would transfer the solar array power to the electric propulsion system. The PDS architecture would be capable of handling high voltages and powers.

2.0 Design Description

Figure 1 shows the one-line diagram of the PDS of a single spacecraft. Figure 2 shows a more detailed block diagram of the system. The high voltage arrays provide power to the electric thrusters during the cruise phase and to the payload and spacecraft loads once the spacecraft enters orbit. The body mounted array and provides power to the spacecraft prior to the high voltage array deployment. The battery powers the payload and the spacecraft loads only. Thus, electric propulsion will not be possible during eclipse payloads.

The MRHE PDS design is unique in that it uses a direct drive architecture for electric thrusters, which saves mass and complexity. In addition, power is transmitted from the solar arrays to the thrusters at a higher than normal voltage level resulting in transmission cable mass savings. The major components of the PDS are described in the following paragraphs.

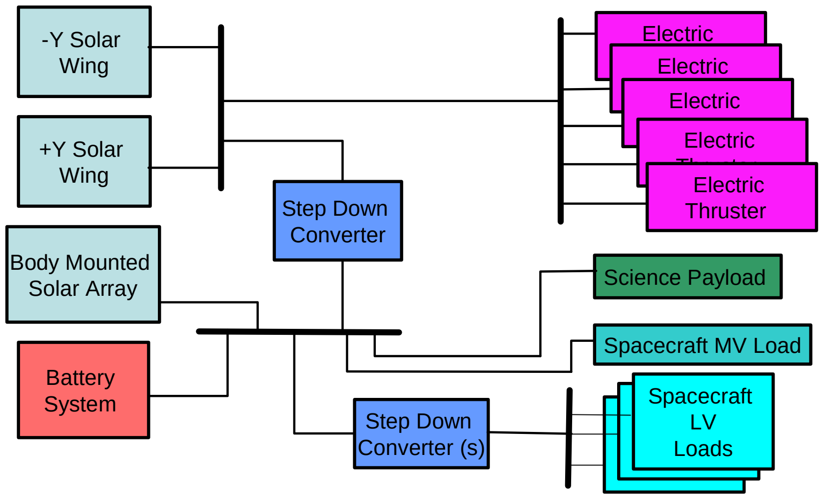

Figure 1: Power Delivery System One-Line Diagram

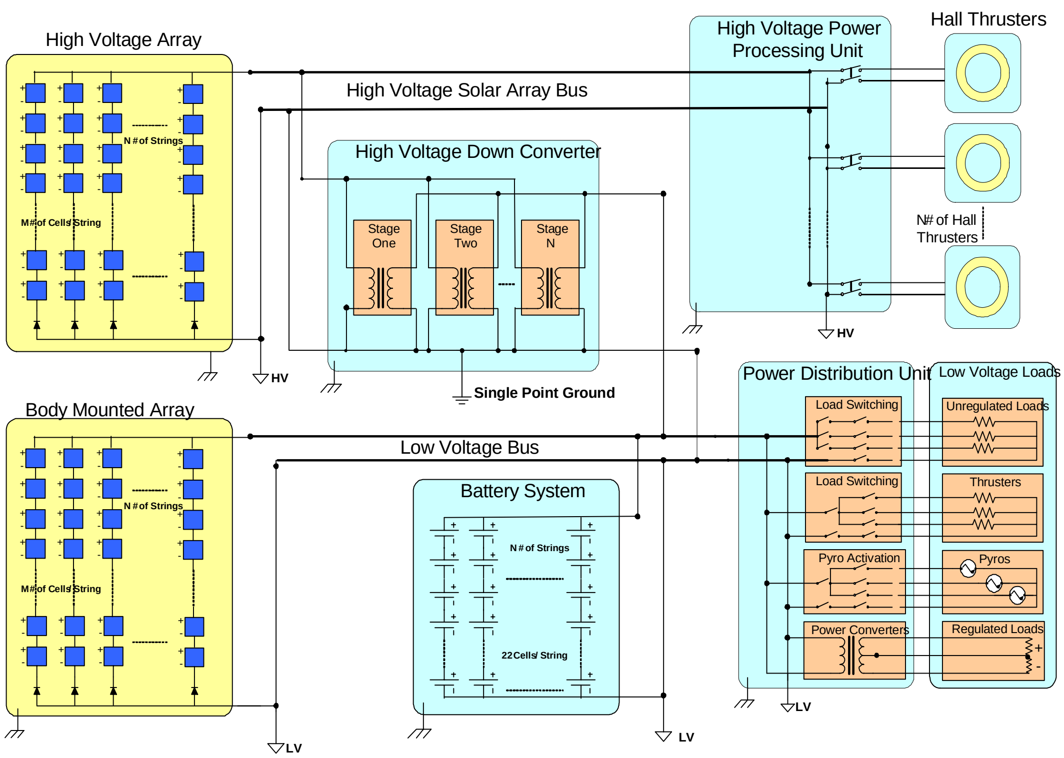

Figure 2: Power Delivery System Block Diagram

Solar Arrays

The Stretched Lens Array (SLA), developed by Entech Corp., was selected by MSFC. The SLAs are capable of generating up to 50 kW of power at an output voltage of 600 V. The SLA technology inherently provides a modular architecture. The concentrating technology results in higher solar efficiencies, and the ability to radiate waste heat effectively. This may reduce the need for both a peak power tracker system, and a shunt regulator. The high voltage photovoltaic circuits must, of course, be insulated to prevent corona failure of dielectrics, particularly in the radiation environment encountered during spiral out from Earth.

Electric Thrusters

Five BPT-4000 Hall Thrusters (built by Aerojet Corp.) were selected for use in the demonstration of electric propulsion. These thrusters represent the present state of the art. Each thruster requires 4.5 kW of power. The BPT-4000 thruster is qualified for space applications for up to 400 V. However, it is capable of operating at voltages of up to 600 V. The high voltage operation (600 V) allows a high I_{sp} (2500 s) and therefore a lower propellant consumption.

Direct Drive PDS

A direct-drive architecture was chosen, which means that there is no power conversion in the main power path to the thrusters. The direct drive approach therefore saves mass and complexity. However, direct-drive removes the isolation between the thrusters and the solar arrays. This complicates arc mitigation within thrusters. It also directly links the solar arrays to the plasma environment created by the thrusters. Neither problem is deemed insoluble.

Multi Bus Design

There are three bus voltages in the PDS design, with high, medium and low voltage levels. The high voltage (600 V) bus results in reduced currents, reduced conductor sizes and reduced losses in the high power circuits to the thrusters. Lower PDS losses result in reduced solar array and total spacecraft masses. The medium voltage (120 V) bus, which is connected to the batteries, supplies power to the 2 kW payload, and the thermal and attitude control subsystems. The commonly used 28 V (low voltage) bus supplies power to other subsystems such as the robotics and avionics subsystems.

Other Design Items

A battery system is used during eclipse, and supplies power to the medium voltage bus and the low voltage loads only. Li-Ion batteries are chosen for their low specific masses. A body mounted solar array is required for power during deployment.

3.0 Power and Mass Estimates

The PDS component power estimates are shown in Table1. Efficiency and uncertainty factors are applied to the power estimates. A high efficiency (98%) value is applied to the direct drive Power Processing Unit (PPU). A 94% efficiency is applied to the high to medium and to the medium to low voltage converters. These converters are not thought to be available “off-the-shelf.”, but have been prototyped at JPL and shown to achieve high efficiencies (> 90%). The spacecraft subsystems power budget has a 30% margin to account for added uncertainties. The total power estimate for one spacecraft module is approximately 28.5 kW.

The PDS mass estimates are shown in Table 3.2.4-2. A 3.21 kW-hr Li-Ion battery with a 50% DOD is considered. The body-mounted array (1 kW) is sized assuming a 23% cell efficiency and using an orbit period of 1.5 hours and eclipse duration of 0.5 hours. The total PDS mass of one spacecraft module is estimated at approximately 171 kgs. This number does not include the masses of the main solar arrays, the cables and the battery control electronics. A 20% mass margin is added to the total PDS mass at the system level.

Table 1: Power Delivery System Power Estimates

Power Subsystem					28.47 KW
Component	Description	Power (CBE)	Efficiency (percent)	Uncertainty (percent)	Power (CBE + Losses + U/C)
Electric Thrusters (5)	5 * 4.5 KW = 22.5 KW with Direct Drive	22.50 KW	98%	10%	25.26 KW
Payload (science)	2 KWs	2.00 KW	94%	10%	2.34 KW
Survival Heaters	300 W (for PPUs, Payload, Subsystem electronics)	0.30 KW	94%	10%	0.35 KW
Subsystems	Communication (104 W), Thermal Control (70W), RAS (50 W)	0.38 KW	94%	30%	0.52 KW
	Total of Payload + Heaters + Subsystems (no thrusters)				3.21 KW

Table 2: Power Delivery System Mass Estimates

Power Subsystem			170.8 Kg
Component	Rating	Description	Mass
Power Processing Unit	25.26 KW	Direct Drive PPU, mass based on parametric estimates by Kerslake NASA TM-2003-212304 (0.55 KW/kg)	45.9 Kg
High Voltage Down Converter	3.21 KW	Based on advanced HVDC using Dawn heritage, JPL Estimate (0.12 KW/Kg)	26.8 Kg
Low Voltage Power Distribution	0.52 KW	Based on light weight version PDU with 300 Krad radiation rating behind 100mils @ RDM of 2, JPL Estimate	20.0 Kg
Battery	3.21 KW-hr	Single String Lithium Ion Battery sized for 3.21 KW watts for 1 hours at 50% DOD, JPL Estimate (125 WHr/Kg)	51.4 Kg
Bus Mounted Solar Array	1.87 KW	This array is based on a 90 min orbit, 1 KW load, rigid body mounted design, including rigid structure mass, and 70 W/Kg FOM	26.7 Kg

4.0 Issues and Challenges

The voltage level chosen for the MRHE PDS is higher than typical. There are several phenomena that occur in the presence of high voltage, some of which are listed below.

1. Electron Collection: At voltages above 300 V the solar array will collect electrons from induced (thruster plume) plasma environments. This results in parasitic current loss that reduces solar array string current.
2. Arcing (1): In a plasma environment (due to thrusters or the radiation environment), arc discharges are generated when the array is biased negative with respect to the plasma. Sustained arcing between solar cells can damage the cells and thus significantly shorten the lifetime of the array. This problem can be mitigated by active charge control – neutralizer/plasma contactor.

3. Arcing (2): At high electric fields (due to high voltage), corona discharges occur inside electronic devices. Corona discharges progressively damage insulation, even inside electronic parts, leading to premature failure.

The PDS design and the technologies presented in this report are feasible. Yet, there are challenges associated with such a system that need to be addressed. The two main challenges to the proposed direct drive architecture are the need for isolation and the proper grounding between the solar arrays and the electric thrusters. In the technology area, the development of high voltage cables, connectors, switches and power converter parts are most challenging. For the high voltage arrays, both the topology and parts availabilities of the control electronics present real challenges.

5.0 Verification

The PDS design was demonstrated via the MRHE Simulation Platform (MSP). The PDS model within the MSP (Figure 3) predicts the power available under various orbital conditions. It receives various inputs, such as array position and subsystem power demands, and computes power, voltage and current levels, and battery state of charge as outputs. The solar array model current-voltage curve is based on data provided by the manufacturer. The battery model determines current, voltage and power during both charge and discharge cycles. The PDS model accounts for power losses of the converters. Table 3 lists the major test cases that were demonstrated via the MSP.

Table 3: MRHE PDS demonstration test cases

Test Case 1	Demonstrate the ability of the PDS to provide adequate power to the Electric Propulsion System and spacecraft loads during sunlight
Test Case 2	Demonstrate the ability of the PDS to provide adequate power to spacecraft loads during eclipse
Test Case 3	Demonstrate the ability of the energy storage system to receive adequate recharge power during sunlight

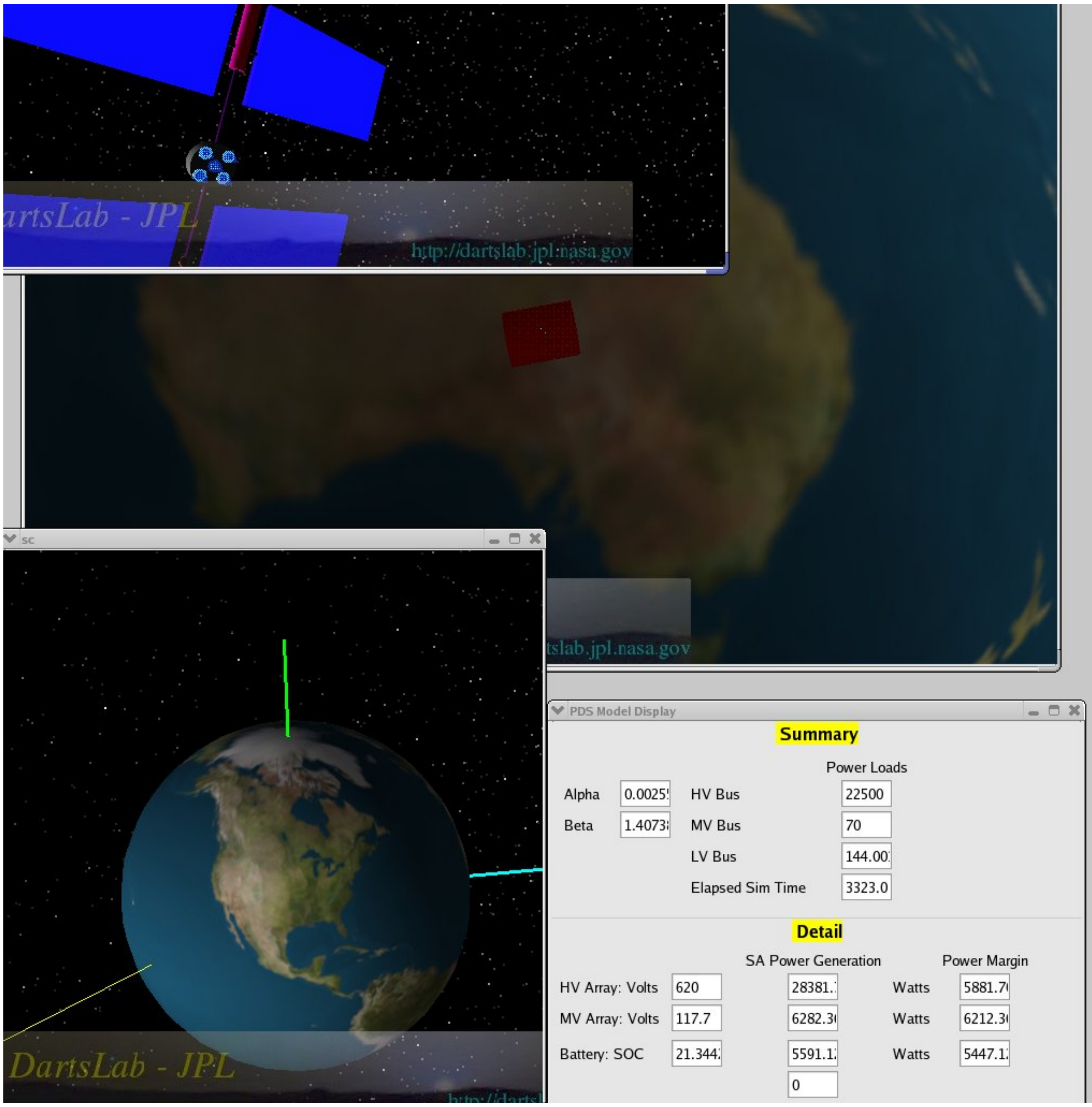

Figure 3: MRHE simulation platform PDS sample display

6.0 Future Work

Any power delivery system is susceptible to various instabilities. In MRHE the issue is complicated by both the large solar arrays and the different modes of array operation: transitions between sunlight and eclipse, large temperature variations etc. The impact of the unique Stretched Lens Array (SLA) operation on the output impedance and the VI characteristics is not well understood.

We propose to further extend the PDS efforts in several ways. The first involves laboratory testing of prototype SLA cells in order to characterize the cells for electrical performance. Voltage and current measurements as well as impedance measurements will be made at the output of the prototype array under varying light and temperature conditions. The results of these tests will be extrapolated to reflect the electrical behavior of a complete array (with series and parallel branches) such as that proposed for the MRHE spacecraft.

The next step involves using the test results to build an electrical model (using a tool such as PSpice) of the SLA. This array model will be integrated into a system level model with a representative PDS electrical model that includes power converters, cables, switchgear and loads. The integrated power system model will be utilized to assess system stability, and the response to load transients and faults. Simulation cases will include varying the SLA temperature, light intensity and electrical loads.

The results of the simulations will highlight some critical operating conditions of the MRHE power system. They will provide valuable insight in choosing the method of SLA control i.e. peak-power tracker, direct-energy transfer or other. The simulations will also shed light on issues related to the direct drive architecture such as proper grounding and isolation between the solar arrays and the thrusters. The results of these efforts will have lessons learned for the power systems of future missions such as lunar missions.

7.0 Conclusions

The goal of the work on the MRHE power Delivery System was to develop a power architecture capable of handling high voltages and powers. The power delivery system design presented in this paper incorporates many advanced technologies, such as the direct drive architecture and the high voltage bus (600 V). The direct drive approach results in significant mass savings within the power delivery system. The high voltage operation of the thrusters results in significant propellant mass savings.

We propose to extend the MRHE PDS work through selected tests for electrical model development, modeling simulations to assess system stability and to help mitigate some of the problems associated with a high-voltage, high-power direct-drive system.