AIAA -2000-2833

Earth Observing System (EOS) Terra Spacecraft 120 Volt Power Subsystem

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35th Intersociety Energy Conversion Engineering Conference
24–28 July, 2000
Las Vegas, NV

EARTH OBSERVING SYSTEM (EOS) TERRA SPACECRAFT 120 VOLT POWER SUBSYSTEM©

Requirements, Development, and Implementation

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ABSTRACT

Built by the Lockheed-Martin Corporation, the Earth Observing System (EOS) TERRA spacecraft represents the first orbiting application of a 120 Vdc high voltage spacecraft electrical power system implemented by the National Aeronautics and Space Administration (NASA) Goddard Space Flight Center (GSFC). The EOS TERRA spacecraft's launch provided a major contribution to the NASA Mission to Planet Earth program while incorporating many state of the art electrical power system technologies to achieve its mission goals. The EOS TERRA spacecraft was designed around five state-of-the-art scientific instrument packages designed to monitor key parameters associated with the earth's climate.

The development focus of the TERRA electrical power system (EPS) resulted from a need for high power distribution to the EOS TERRA spacecraft subsystems and instruments and minimizing mass and parasitic losses. Also important as a design goal of the EPS was maintaining tight regulation on voltage and achieving low conducted bus noise characteristics.

This paper outlines the major requirements for the EPS as well as the resulting hardware implementation approach adopted to meet the demands of the EOS TERRA low earth orbit mission. The selected orbit, based on scientific needs, to achieve the EOS TERRA mission goals is a sun-synchronous circular 98.2-degree inclination Low Earth Orbit (LEO) with a near circular average altitude of 705 kilometers. The nominal spacecraft orbit is approximately 99 minutes with an average eclipse period of about 34 minutes. The scientific goal of the selected orbit is to maintain a repeated 10:30 a.m. +/- 15 minute descending equatorial crossing which provides a fairly clear view of the earth's surface and relatively low cloud interference for the instrument observation measurements.

The major EOS TERRA EPS design requirements are single fault tolerant, average orbit power delivery of 2,530 watts with a defined minimum lifetime of five years (EOL). To meet these mission requirements, while minimizing mass and parasitic power losses, the EOS TERRA project relies on 36, 096 high efficiency Gallium Arsenide (GaAs) on Germanium solar cells adhered to a deployable flexible solar array designed to provide over 5,000 watts of power at EOL. To meet the eclipse power demands of the spacecraft, EOS TERRA selected an application of two 54-cell series connected Individual Pressure Vessel (IPV) Nickel-Hydrogen (NiH2) 50 Ampere-Hour batteries. All of the spacecraft observatory electrical power is controlled via the TERRA Power Distribution Unit (PDU) which is designed to provide main bus regulation of 120 Vdc +/-4% at all load interfaces through the implementation of majority voter control of both the spacecraft's solar array sequential shunt unit (SSU) and the two battery bi-directional charge and discharge regulators.

This paper will review the major electrical power system requirement drivers for the EOS TERRA mission as well as some of the challenges encountered during the development, testing, and implementation of the power system. In addition, spacecraft test and early on orbit performance results will also be covered.

SYSTEM REQUIREMENTS OVERVIEW

The EOS TERRA spacecraft utilizes a multi-tiered requirement documentation approach. Top level requirements are contained in four basic categories that were used to define a Contract End-Item (CEI) Specification. The documents are; (1) Performance Assurance Requirements, (2) Requirements Document, (3) Unique Instrument Interface Documents, and (4) General Instrument Interface Specification.

The CEI Specification was used to derive the Interface Control Requirements, Verification Requirements, Spacecraft Control Plans, and the General Interface Specification (GIS). All of these requirements were, in turn, flowed down to Major Assembly Specifications, Subsystem Specifications, Flight Software System Specifications, and Instrument Interface Control Documents (ICDs). The focus of this paper will address requirement overview of the EPS specifically with regard to the GIS requirements and the resulting spacecraft implementation architecture.

EPS DRIVING REQUIREMENTS

Major GIS Requirements

The EOS TERRA major power subsystem requirements are defined in a top level General Interface Specification (GIS) and subsequently flow down into an Electrical Power Subsystem (EPS) Specification. In addition to electrical requirements, the GIS also defines the mechanical, thermal, and environmental requirements that are common to most of the spacecraft bus elements. There are two basic power interfaces defined in the GIS, one for the "primary" power users and the other being for "secondary" power users. The major difference between the two is the voltage provided at the user load interface. The primary user voltage is defined as a 120 Volt direct current (Vdc) interface and the secondary user voltage is a 28 Vdc interface. The primary interface requirement for bus regulation is 120 Vdc with a tolerance of +/-4% at the load interface. As a result of this tight voltage regulation, the source impedance requirement is specified as a series combination of resistance less than or equal to 0.8 ohms and an inductance of 2.5 microhenries (with a resulting corner frequency of 51 kHz) over the frequency range of dc to 150 kHz. Under all abnormal conditions (e.g. fault conditions) the power system is required to maintain primary EPS voltages between 0 and 132 Vdc. The primary power system is also required to provide fusing to protect the EPS and spacecraft harnesses from all potential load or harness faults.

The secondary power voltage regulation requirement is 28 Vdc +/-2% at the load interface. The secondary power system has two specifications for impedance requirements depending on the length of the harness to the load interface. One impedance requirement is less than 0.4 ohms at dc, increasing at a 750 microhenry slope up to 300 Hz, then 1.5 ohms from 300 Hz to 100 kHz, then increasing at 0.7 microhenry slope from 100 kHz to 1 MHz, for harness lengths of 10 feet or less. For those users with harness lengths greater than 10 feet, the impedance specification is less than 0.9 ohms at dc increasing at a 1000 microhenry slope from 300 Hz to 100 kHz, then increasing at a 2 microhenry slope from 100 kHz to 1 MHz. Again, under abnormal or fault conditions, the secondary voltage requirement is to always be within 0 to 35 Vdc range.

The EPS is required to provide a single point ground reference for the spacecraft primary power system while all secondary power returns are required to be referenced either at the component chassis reference or at the power source chassis reference. The use of the power source chassis reference was normally the preferred means for secondary grounding. The maximum allowable resistance for all ground reference ties is 2.5 milliohms.

Major EPS Performance Specification Requirements

The GIS requirement flow down forms the basis for development of the major EPS performance requirements. In turn, the EPS performance specification requirements are the basis for development of individual component performance specifications, once the architecture is defined. For this paper, only the major subsystem performance specification requirements are presented.

The overall architecture for the EPS was derived initially on the top-level requirements and was refined through multiple trade studies in order to meet the necessary tolerances imposed. Given the nature of the spacecraft orbit and the power distribution needed, a Direct Energy Transfer (DET) system was the obvious choice. The details of the actual component implementation will be covered later in this paper.

The maximum average spacecraft load power requirement for the EOS TERRA spacecraft was defined at 2,530 Watts, excluding energy storage requirements, at a defined end of life (EOL) scenario of 5 years. The power system is also required to provide up to 3000 watts in peak load power for up to 15 minutes per orbit, all without violating the 2530-Watt orbit average power requirement.

In the event of a non-catastrophic loss of all spacecraft power on orbit, the EPS is also required to be capable of autonomously restarting whenever the bus voltage rises above 50 Vdc.

To meet the maximum average load distribution power requirement as well as the battery charge requirement after five years on orbit, the solar array power requirement was derived to be 5 kW at year 5, EOL worst day conditions. The energy storage requirements for the eclipse portion of the TERRA orbit were derived based on the defined maximum average orbit load power requirements and ultimately resulted in the selection of two 50 Ampere-hour Nickel-Hydrogen batteries to provide the necessary energy capacity.

Recognizing the requirement for 120 Vdc +/-4% at the load interface, the EPS derived the requirement for the main bus regulation to be defined as 120 Vdc +4/-2%, in order to allow for worst case voltage drop conditions in spacecraft harnesses. The voltage ripple requirement imposed on the EPS was 280 millivolts peak-to-peak (100 millivolts rms) in order to assure low noise characteristics for the instrument reference planes. The peak-to-peak ripple requirement represents a value of only 0.23% of the nominal primary bus voltage. Under transient load conditions, the EPS is required to maintain regulation within a maximum of 3 volts overshoot or undershoot and is required to recover to within 10% of it's final value within 10 milliseconds for a primary load step of 10 amperes (1200 Watts).

To meet the requirement for autonomous start whenever the bus voltage rises above 50 Vdc, the EPS derived requirements that it must operate within specification down to 50 Vdc. Also, as a result of the GIS imposed impedance requirement, the EPS derived a requirement for source impedance, as measured across the main bus capacitor, of less than 200 milliohms for frequencies between 0.1 Hz to 100 kHz. This impedance requirement was derived based on subtracting expected maximum impedance characteristics of worst case harness lengths to load interfaces with appropriate margin included.

In addition to the derived performance requirements, conducted and radiated noise characteristics were also imposed on the EPS. Although they will not be covered in this paper, the details of the imposed requirements can be found in the EOS TERRA Electromagnetic Compatibility (EMC) Control Plan.*

Using normal spacecraft program development mass allocation techniques, the EPS was given a mass allocation based, ultimately, on the total lift capability of the baseline launch vehicle. After the Atlas IIAS was selected as the launch vehicle, the total allocated mass for the EOS TERRA EPS was defined at 1315 pounds, or about 15% of the vehicle lift capability. Of primary importance in the design phase was the requirement for the EPS to be single fault tolerant to any failure within the subsystem. The implications of this requirement resulted in all EPS components having at least internal redundancy to any single failure mechanism. Some aspects of the redundancy implementation will also be covered in discussions concerning the components later in this paper.

The last major top level requirements are the maximum orbital eclipse period and expected beta angles. The eclipse is defined as 34.84 minutes, and is consistent with the worst (longest) case for the selected spacecraft orbit altitude and inclination. The beta angle variation is defined as being between 13.5 and 30.8 degrees, while the EPS has a derived requirement to also operate at a beta angle of 0 degrees when in a sun pointing safe hold mode.

These requirements represent the major top level imposed requirements from which the TERRA EPS design was generated with the hardware implementation being covered next.

HARDWARE IMPLEMENTATION

Architecture Selection

Given the LEO orbit selected for the EOS TERRA spacecraft and the relatively high power requirements necessary to support the instrument packages, the project initially performed a trade study to evaluate high voltage power distribution versus the more traditional (heritage) 28 Vdc bus architecture. Of particular importance for TERRA was the mass trade for spacecraft harnesses and the resulting parasitic I^2R losses. The high power distribution around a large spacecraft structure (TERRA spacecraft dimensions are diameter of 3.5 m, length of 6.8 m, and weight of 5190 kg), especially when considering the single fault tolerance requirement, becomes very unwieldy at 28 Vdc. For just the distributed load power requirements, the 28 Vdc architecture would be required to distribute approximately 90 amperes of current and when considering the expected maximum

* Electromagnetic Compatibility (EMC) Control Plan for the EOS-AM Spacecraft (SEP-106), PN20005869B, July 7, 1994, Lockheed-Martin Corporation.

current flow from the solar array, the total current flow becomes almost 180 amperes.

Selection of higher distribution voltages result in proportionately lower overall electrical current flow and, in this case, at 120 Vdc the current flow becomes a much more manageable 42 amperes from the solar array. Also, given the initial association of TERRA with the International Space Station program, where 120 Vdc was ultimately selected as the baseline for power distribution, the options considered for TERRA were ultimately limited to 28 Vdc and 120 Vdc in hopes of utilizing space qualified parts that were either already available or in development for 120 Vdc. One additional benefit perceived by NASA/GSFC in this trade study evaluation is the development of a second "standard" voltage (in addition to 28 Vdc) for any large future high-powered spacecraft power systems developed at or for NASA/GSFC.

NASA GSFC engineers worked closely with Lockheed-Martin Corporation and Virginia Polytechnic Institute (specifically the Virginia Power Electronics Center or VPEC) to develop a viable spacecraft EPS architecture utilizing 120 Vdc. A significant portion of the development took place while the EOS TERRA project was associated with the International Space Station program. The results of the efforts at VPEC culminated in a 120 Vdc test bed development and verification.[†] Based on these successful results and the need to maintain highly regulated power distribution characteristics, minimize I^2R losses, minimize harness mass, and lower overall launch cost considerations, the base line selection of 120 Vdc as the preferred power system distribution architecture for TERRA was made.

Major Subsystem Trade Studies

During the preliminary design review process of the EOS TERRA power subsystem, there were many trades studies that were undertaken to optimize cost, schedule, and mass. The primary trade study, as discussed earlier, focused on 120 Vdc versus 28 Vdc distribution of power with 120 Vdc ultimately being selected on the basis that it would save a large amount in both mass and distribution losses. The estimated mass savings for the EOS TERRA spacecraft using 120 Vdc versus 28 Vdc was determined to be approximately 900 pounds.[‡]

Another early trade study involved whether to use fuses for circuit protection or resettable type circuit breakers. Again, the trade focused mainly on mass and ultimately was decided in favor of the reliable and lighter solid state fuses made by Mepcopal, Inc.

From a power generation perspective, the solar array was required to provide 5+ kW power to the spacecraft at EOL and obviously could be derived using flexible or rigid panels with either Silicon or Gallium Arsenide (GaAs) on Germanium solar cells. From a launch vehicle perspective, the ability to package the array within the launch vehicle fairing dynamic envelope leaned heavily in favor of a flexible array that could be stowed in a relatively small volume. TRW, Inc. was nearing completion of a Jet Propulsion Laboratories (JPL) funded research program to develop a lightweight flexible array concept named Advanced Photovoltaic Solar Array Program (APSA) that held promise for increasing the power to mass ratio for larger arrays. The results of the development effort are contained in the APSA Final Technical Report[§] and ultimately also heavily influenced a decision in favor of selecting the TRW APSA based design for TERRA using higher efficiency GaAs solar cells.

As with all power system selection processes, the fundamental options for the power system architecture focused on Peak Power Tracker versus Direct Energy Transfer systems. Given the high power requirements and the need for tight regulation on the power system bus, the DET architecture was the clear favorite in the trade, as can be expected for the given requirements.

Energy storage requirements were also driven by a desire to maximize energy storage capability while minimizing mass. To this end, the primary options available for TERRA were Nickel Cadmium (NiCd) versus the more recently developed Nickel Hydrogen (NiH2) battery cells. Even though the cost was somewhat higher for the NiH2 cells, the overall ability to maximize the energy storage density for a given weight led the decision process to select NiH2 cells manufactured by Eagle-Picher Technologies.

Since the desire was to tightly regulate the bus, the batteries were going to be isolated from the bus via some form of charge/discharge regulation and therefore it was not imperative that the number of cells closely mimic the subsystem operating voltage. Based on a variety of factors during the evaluation process, as well as the availability of NiH2 IPV ampere-hour ratings at the time of the study, two batteries, each consisting of 54 50-Ampere-hour Individual Pressure Vessel (IPV) cells connected in series were selected for TERRA. The selection of this combination

[†] EOS Satellite Power System Testbed, Dr. Dan Sable, VPEC, Virginia Polytechnic Institute, Blacksburg, VA presented at the High Voltage Spacecraft Power Technology Workshop, May 4-5, 1993.

[‡] 120 Vdc and 28 Vdc Power Distribution Trade Off White Paper, R. Stone and B. Beaman, NASA GSFC Internal Memorandum, February 12, 1992.

[§] Advanced Photovoltaic Solar Array Program, Final Technical Report (CDRL 012), November 15, 1994, TRW Space & Technology Division, TRW Report No. 51760-6006-UT-00 resulted in a nominal on-orbit useable battery voltage range of approximately 60 to 90 Vdc.

Major EPS Components

With the experience of testing performed at VPEC, the block diagram architecture took shape in the form shown in Figure 1. The basic major components comprising the EPS were; Sequential Shunt Unit (SSU), Power Distribution Unit (PDU), Battery Power Conditioners (BPC), Two Nickel-Hydrogen Batteries, and One Solar Array utilizing GaAs solar cells. For primary to secondary power conversion, Electronic Power Conditioners (EPC) were proposed and adopted which allowed conversion to 28 Vdc at, or very near, the load interface.

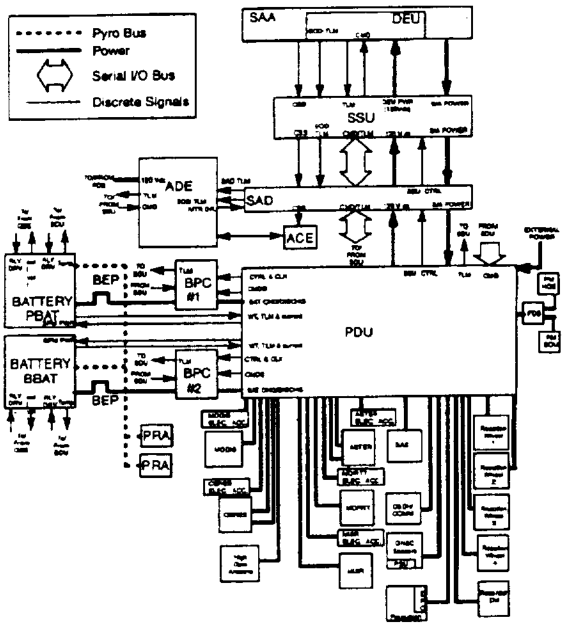

Figure 1 EOS TERRA EPS Block Diagram

Deployable Flexible GaAs Solar Array

All of the power needed to operate the EOS TERRA spacecraft is generated by the solar array. The TERRA solar array is a flexible blanket design utilizing GaAs on Germanium solar cells attached to a Germanium coated Kapton substrate with graphite fiber reinforced plastic (GFRP) reinforcement. The entire substrate was vapor deposited with a 1000-Angstrom thick Germanium coating to provide ESD charge dissipation capability.

The solar array contains 36,096 solar cells, each cell measuring 2.4 cm x 4 cm x .014 cm. The solar cells are inter-connected together to form a total of 192 solar cell strings, each consisting of 188 series connected cells. Eight strings are wired in parallel to form 24 circuits. All 24 circuits are brought to the SSU via a slip ring assembly within the solar array drive assembly. The beginning-of-life (BOL) design characteristics attempted to achieve a total power generation of approximately 8 kW of power at standard temperature conditions (STC). Completed array flash test results at TRW verified this design capability.

A picture of the fully deployed TERRA solar array is shown in Figure 2. Full solar array deployments were only performed at the manufacturer's facility (TRW) on a specially made deployment fixture.

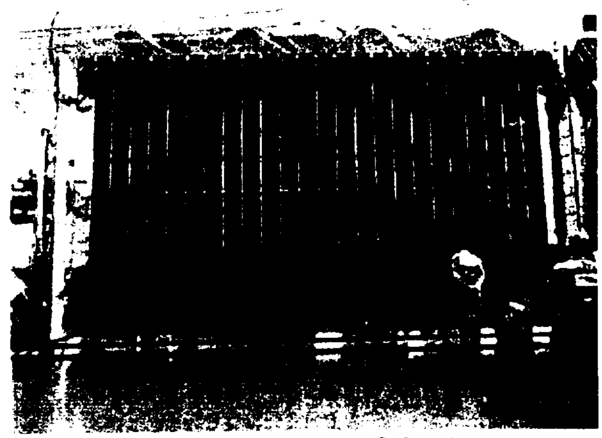

Figure 2. EOS TERRA Solar Array

Power Distribution Unit (PDU)

The PDU is the main bus and contains the electronics used to maintain the 120 Vdc bus regulation. The PDU regulates the bus voltage via current control of both the SSU and the BPCs, depending on whether the spacecraft is in sunlight or orbital eclipse. Control is achieved using a majority voter control logic with 3 voter circuits. The PDU also houses all of the main bus switches for non-essential loads (such as instruments and High Gain Antenna) and can be used in the event load shedding is necessary or if there is a desire to switch from A to B-side operation.

A picture of the flight PDU while in test is shown in Figure 3.

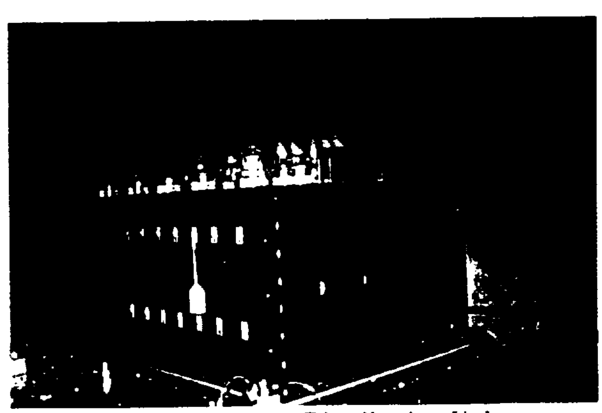

Figure 3. Power Distribution Unit

Sequential Shunt Unit (SSU)

The SSU actively controls the amount of current supplied to the TERRA PDU from the 24 available solar array circuits. The current provided by each circuit is controlled via pulse width modulation (PWM) and can be up to approximately 2.5 amperes of current at about 130 Volts for each circuit. All unneeded bus current generated by the solar array is shunted back to the array via the SSU. The input feeds from the solar array to the PDU are each protected by a 10-amp fuse that primarily protects the bus from a solar array to PDU harness short.

A picture of one of the 6-shunt modules that make up the SSU is shown in Figure 4. Each SSU shunt module contains 4 shunt circuits.

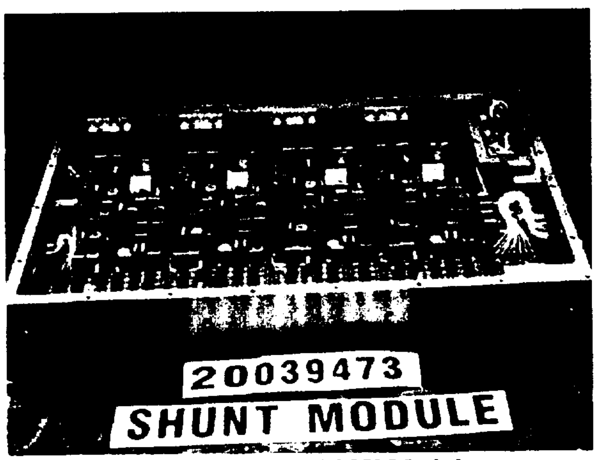

Figure 4. One of 6 SSU Modules

Battery Power Conditioners (BPC)

The two BPCs provide the means to buck regulate charge current to the spacecraft batteries during the sun lit portion of the orbit as well as providing a boost regulation capability from the batteries to the main bus during the eclipse portion of the orbit. Again, the BPCs utilize a PWM boost regulator design approach which will vary the electrical current provided to the main bus, while in orbit eclipse, proportionally based on bus load demand. The discharge current control signal is provided from the PDU via a MIL STD-1553 data bus.

The BPCs are each designed for four-channel operation, in parallel, with a 4 for 3 redundant architecture. Each channel of the BPC can provide up to 5.5 amperes charge current at battery voltage and provide up to 30 amperes output under fuse clearing conditions (limited ultimately by battery fuse sizing).

The PDU, BPCs and one of the two spacecraft batteries are contained within a Power Equipment Module (PEM). The equipment module approach was implemented throughout the TERRA spacecraft in an effort to minimize the integration complexity.

A picture of one of the flight BPCs as installed in the Power Equipment Module is shown in Figure 5.

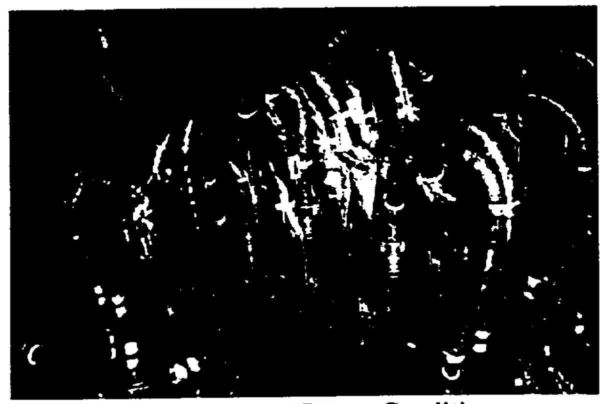

Figure 5. Battery Power Conditioner

Nickel-Hydrogen (NiH2) Batteries

The two NiH2 batteries were sized to provide all of the energy storage capability for the TERRA spacecraft. Each battery is comprised of 54 series connected 50 amp-hour NiH2 battery cells. The requirement for the TERRA mission was to meet all of the power demands of the spacecraft during the eclipse of the orbit and keep the maximum normal orbit depth of discharge (DOD) of the batteries to 30%, or less, of nameplate capacity. The DOD requirement assumes maximum spacecraft design load conditions and the loss of one battery cell in each battery.

The batteries were designed to be capable of monitoring individual cell voltages as well as individual cell temperatures through a Heater Control Electronics (HCE) component. These parameters were considered crucial during the design phase for such large series connected battery cells. The active heater control in the HCE utilized heater groups, each composed of 9 battery cells. The heater group temperatures are averaged in the spacecraft control computer and that value is used to provide the HCE with the temperature control signal. The HCE design temperature range for each battery is between -1 and -5 degrees C. The active thermal control pulse width modulated heaters will operate at 0% duty cycle at -1 C and at 100% duty cycle at -5 C.

An example of one of the two spacecraft flight batteries is shown in Figure 6 without its battery cover.

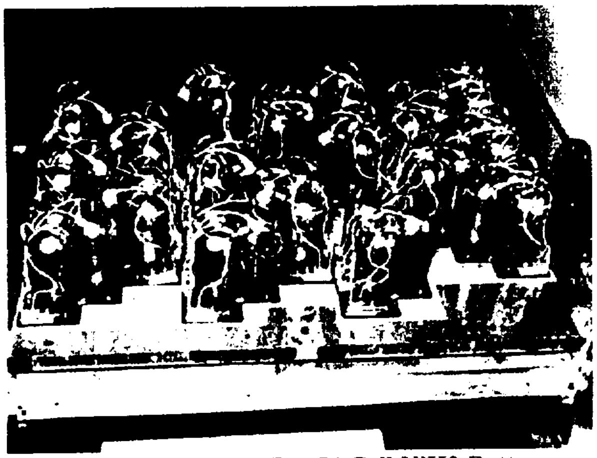

Figure 6. One TERRA 54 Cell NiH2 Battery

Electronic Power Conditioners (EPC)

The EPCs were developed in conjunction with the 120 Vdc bus architecture as a means to provide various low voltage regulated power to all spacecraft subsystems that required them. The EPCs were originally envisioned to be a common component capable of various load demands with standard secondary voltage levels (+/-5, +/-10, +/-15, +18, +28 Vdc). After taking stock of the wide power requirements of the various subsystem needs, the single common EPC power supply design approach was eliminated to more appropriately tailor the converters to their specific load requirements. This modified approach resulted in ten basic similar designs for EPCs with a total numerical count of 42 EPC converters needed for the various spacecraft subsystems.

From an EPS perspective, only one of the spacecraft's subsystem EPCs is considered as an integral part of the EPS and that is the Propulsion System EPCs. The Propulsion System EPCs were solely 120/28 V converters and were used to adapt heritage based propulsion components into the EOS TERRA spacecraft design.

A picture of a typical EPC in an open configuration is shown in Figure 7.

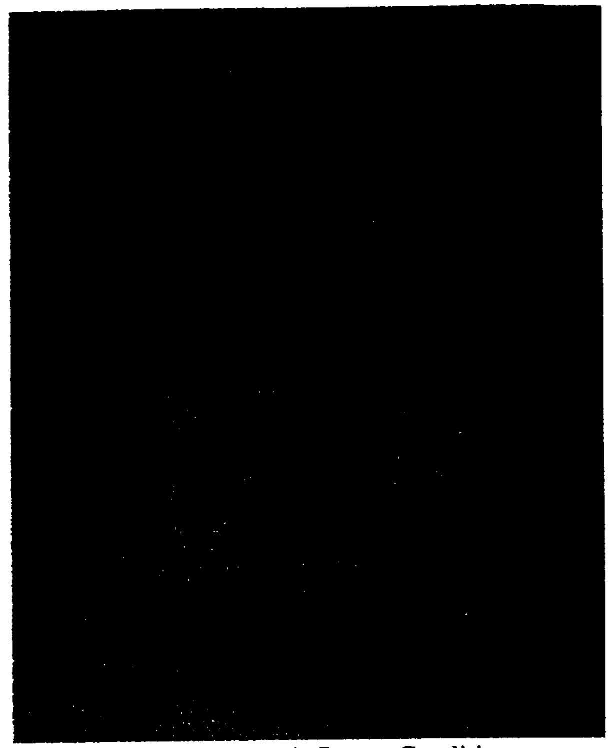

Figure 7. Electronic Power Conditioner

TEST PERFORMANCE RESULTS

The EOS TERRA spacecraft 120 Vdc power system operation concept was initially verified through the development and test effort at VPEC, and once those breadboard versions of the power system components were considered mature enough, a development effort using engineering models was undertaken by Lockheed-Martin Corporation. After comprehensive component level testing was completed the engineering models were placed in a test bed and operated over the required limits imposed on the EPS. Some minor differences were expected between the test bed and the actual flight system and were needed in order to conveniently, and cost effectively, perform the system level testing. The deviations from planned flight configuration were primarily the use of lead acid batteries in place of the NiH2 batteries and the use of only one half of the control electronics in the PDU (equivalent to using only one side of the redundant control electronics).

A solar array simulator was manufactured by ELGAR Corporation that accurately simulated up to 28 solar array circuits and, utilizing sophisticated software programming capability, precisely matched the actual GaAs cell and string characteristics. A photo of the EPS engineering test model (ETM) test bed configuration is shown in Figure 8. The operating performance of the ETM test bed confirmed wellbehaved stable performance as well as low conducted noise characteristics.[¶]

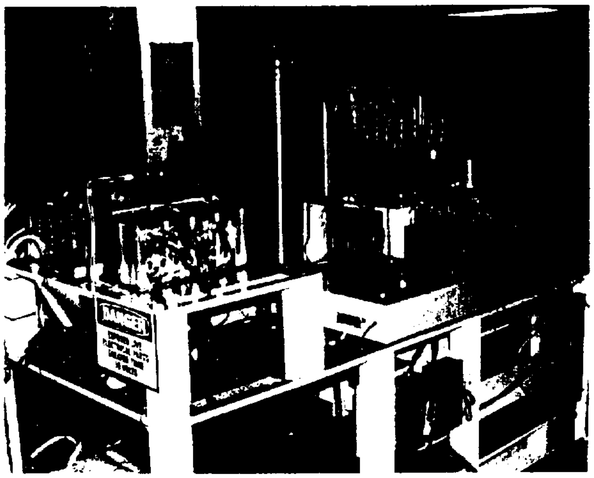

Figure 8. EOS TERRA ETM Test Bed

Problems Encountered During Development/Fabrication

During the course of developing the flight EPS hardware, the project was faced with many challenges, from a technical perspective, associated with a new power system architecture, new solar array and battery technologies, as well as development and test schedules. Early in the acceptance testing of some of the flight EPCs, the TERRA project found evidence of via failures on EPC printed wiring boards (open circuit conditions as a result of trace to barrel fractures), which were ultimately traced to the board manufacturing processes. In all cases, the flight printed wire board coupons were evaluated for acceptance and all of the identified failed boards initially appeared to have acceptable coupons. A second round of coupon analysis was performed at which time several board lots were found to be suspect. Unfortunately, at this time at least two of the suspect flight boards were already fabricated and tested and were scheduled to be integrated with their next higher assemblies. To mitigate this potential problem, and minimize a potentially large schedule impact needed in order to re-fabricate the boards, a series of thermal cycle tests were performed on the completed boards that were specifically designed to surface the potential problem. In the case of the two suspect boards, there was no evidence of this problem during the testing and they were ultimately cleared for flight use.

A second challenge became apparent when the stability characteristics of some of the EPCs were found to be marginal at low load conditions, primarily as a result of the predicted design loads on the converters being significantly more than the actual loads. The approach taken to alleviate the performance problems at low load conditions was to implement pre-loads on those EPCs that were being significantly under utilized. While this provided somewhat of a quandary as to effectively lowering the converter efficiency of the EPCs, the trade off was, again, a significant slip in schedule while the designs were modified to accommodate the revised loads. In most cases, the pre-loads were not substantially altering the full load efficiency, although the largest power EPC (Propulsion EPC) ultimately required an externally switchable 28-Watt load resistor bank to meet the necessary performance desired at low load conditions.

A third problem that surfaced during fabrication and testing of the EPCs was a series of failures of Vishay (precision surface mount) resistors. This particular problem was ultimately traced to improper handling of the resistors during the parts kit process and once tighter control over the handling was implemented, the failure rate dropped to near zero.

One of the most challenging problems encountered during the TERRA EPS development was the identification of electrostatic discharge (ESD) induced solar array arcing as a potentially destructive phenomenon to the TERRA solar array design. This particular problem was identified as a result of multiple orbiting spacecraft solar array degradation failures. While the mechanism is not absolutely conclusive, the on-orbit array degradation failures were most likely linked to cover glass ESD surface charging which, in turn, resulted in sporadic discharge arcing that ultimately provided a condition that supported high to low voltage cell-to-cell string short circuit failures.[#] This particular phenomenon was identified as a potentially high risk item after the on-orbit anomalies because the EOS TERRA array was designed similar to those on orbit with the highest voltage solar cell in a string located directly next to the lowest voltage cell in that string. Substantial testing of the EOS solar array design was conducted in a specially designed vacuum chamber at NASA Glenn Research Center. The results of the testing indicated that the EOS TERRA solar array was most definitely susceptible to this form of potentially destructive

[¶] EOS-AM EPS ETM S/S Test Report, Design Note EOS-DN-EPS-090, Lockheed-Martin Corporation, August 1, 1995.

[#] Katz, Davis, and Snyder, "Mechanism for Spacecraft Charging Initiated Destruction of Solar Arrays in GEO", AIAA, 1998 arcing on orbit. As a result, several modifications** were made to the EOS solar array, most notably the addition of diodes in every solar array string intended to severely limit the available current for supporting the cell-to-cell arcing in the event the arcing phenomenon was indeed present on orbit.

Another particularly disturbing problem that required attention late in the TERRA test schedule was significant capacity fading noted on the workhorse NiH2 batteries. The original project plan was to install of the TERRA flight batteries during spacecraft environmental testing (just prior to spacecraft thermal vacuum testing) and leave the flight batteries installed on the spacecraft until TERRA launched. What appeared at the time to be a result of ambient handling, partial charge and discharge cycles, and potentially improper battery letdown procedures, a significant capacity fading was found to have occurred on the workhorse batteries. Further evaluation of available trend data, prior to shipping the spacecraft to the launch site, indicated this degradation was also beginning to appear on the flight batteries as well. Destructive physical analysis (DPA) performed on sample battery cells were not conclusive as to the cause of capacity fading, but significant blistering was noted on at least one cell. Ultimately, the battery letdown equipment being used was found to promote battery cell reversal while not accurately displaying the reversal effect during letdown. Once discovered, the procedures for battery conditioning and letdown were altered to minimize the possibility of cell reversal in the future and a battery test program was initiated to evaluate the long term effects of the degradation in on-orbit conditions.

During environmental testing, specifically in spacecraft thermal vacuum testing, another problem was identified regarding significant noise on the battery cell voltage telemetry. While never confirmed through disassembly of the flight batteries, the cause of this noise was ultimately believed to be a lack of shielding of the battery cell voltage monitoring wiring within the battery itself and appeared to only be a problem when the heater circuitry (pulse width modulated) was active in PWM mode. The noise was not normally present on telemetry signals of cell voltage in ambient temperature because the thermostat that controls the circuit operation was normally open (thermostat is designed to close when the temperature is approximately 7 degrees C).

In addition to the cell voltage noise, there was also a low level of noise observed on the temperature sense circuits of all of the battery cells. This temperature noise is not believed to be caused by the same phenomenon as the voltage noise but, is instead believed to be a result of noise induced within the heater control electronics itself. The noise associated with the temperature monitoring has been of more concern than the voltage monitoring because the temperature measurements are used by the flight computer in a control manner to ultimately determine the heater duty cycle. If the noise were to get significantly worse, the heater pulse width operation could be adversely affected.

The ultimate resolution of this problem was to fly the hardware "as is" and to monitor the noise levels and plan to turn off the active control if the temperature noise were ever to adversely affect the heater operation. When disabled, the method of maintaining thermal control on the batteries would then be equivalent to the spacecraft safe hold operation, which utilizes thermostatic control of 100% duty cycle heater power whenever the battery temperature groups reach about −10 degrees C. The upper thermostat will open and disable the heater power when the temperature reaches about 0 degrees C.

This particular operating mode was tested in spacecraft thermal vacuum testing and ultimately did provide an adequate backup means to alleviate the problem. Another means that was determined to be an acceptable back up was the reprogramming of the flight computer to operate in a full-on/full-off mode of 100% pulse heater power while disabling the software controlled PWM mode of heater operation. Basically, armed with these alternative means for resolving this potential thermal control problem was rationale enough to proceed with the hardware "as is."

Launch Performance

Launch of EOS TERRA occurred on December 20, 1999 with near flawless performance of all systems. A photo of the complete spacecraft during launch preparations at Vandenberg Air Force Base is shown in Figure 9. Soon after orbit was achieved, the deployment of the flexible solar array was initiated. The onboard software in the flight computer controlled the deployment and all was nominal until one of two micro switches failed a telemetry check to indicate separation of the solar array blanket box.

** Davis, Stillwell, Andiario, and Snyder, "EOS-AM Solar Array Arc Mitigation Design", IECEC, 1999.

Figure 9. EOS TERRA Spacecraft at VAFB

This particular failure mode (along with a long potential list of other failure scenarios) was handled automatically on board using back up deployment procedures. In this particular case, the on-board software was programmed to automatically switch to the back up side of the deployment electronics after a prescribed wait time. After the switchover the remainder of the deployment went as expected.

Although there was an investigation into this deployment failure, it is still unclear as to whether the micro switch itself failed to indicate separation or if the failure was in the deployment electronics telemetry.

On-Orbit Performance

For the roughly 5 months that EOS TERRA has been on orbit, there have been no significant anomalies experienced with the EPS. One EPS related anomaly identified during early TERRA orbit correction maneuvers was a significant plume impingement of the thrusters onto the solar array. When one of the first correction maneuvers was performed, the spacecraft began to roll unexpectedly and subsequently put the spacecraft into a safe hold. Analysis performed after the problem was encountered finally reached the conclusion that the plume field of the thrusters was impacting the solar array and, for that particular maneuver, the solar array happened to be in the ideal position for significant impingement. This analysis was subsequently verified through measured thruster firings with the array in prescribed positions and the resulting momentum changes predicted and verified.

Since there were a number of Delta-V burns yet scheduled to achieve final orbit position, the spacecraft operations team defined a method of stopping the solar array rotation in a position least impacted by the thruster burns with a resumption of array rotation immediately following the correction burns.

From a performance perspective, all EPS subsystem operations are performing as expected with only minor operational anomalies experienced to date.

SUMMARY

EOS TERRA represents the first successfully developed and launched high voltage 120 Vdc based spacecraft power system for NASA. Through significant development efforts and attention to many details associated with high voltage operation in vacuum, EOS TERRA stands as a model for future high power and high voltage power systems. The resulting savings in both parasitic power losses and harness weight coupled with experience on-orbit provides a proven basis for consideration of high voltage dc power systems in the future.