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DESIGN AND EARLY IN FLIGHT PERFORMANCE OF THE TROPICAL RAINFALL MEASURING MISSION (TRMM) POWER SUBSYSTEM

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ABSTRACT

The Tropical Rainfall Measuring Mission (TRMM) is a joint endeavor of the United States and Japan. The National Aeronautical and Space Administration (NASA)'s Goddard Space Flight Center (GSFC) in Greenbelt, Maryland built the spacecraft in-house with four U.S. instruments and one Japanese instrument, the first space flown Precipitation Radar (PR). The TRMM Observatory was successfully launched from Tanegashima Space Center in Japan on an H-II Expendable Launch Vehicle on November 27, 1997. This paper presents an overview of the TRMM Power System including its design, testing, and in flight performance for the first 70 days. Finally, key lessons learned are presented.

The TRMM power system consists of an 18.1 square meter deployed solar array fabricated by TRW with Tecstar GaAs/Ge cells, two (2) Hughes 50 Ampere-Hour (Ah) Super NiCd™ batteries, each with 22 Eagle-Picher cells, and three (3) electronics boxes designed to provide power regulation, battery charge control, and command and telemetry interface.

INTRODUCTION

TRMM is the first mission dedicated to measuring tropical and subtropical rainfall through microwave and visible infrared sensors. The National Space Development Agency of Japan (NASDA) provided the PR and the H-II rocket. GSFC, together with many industry contractors, provided the Observatory and four instruments.

Tropical rainfall comprises more than two-thirds of global rainfall and is the primary distributor of heat energy which drives atmospheric circulation. Understanding

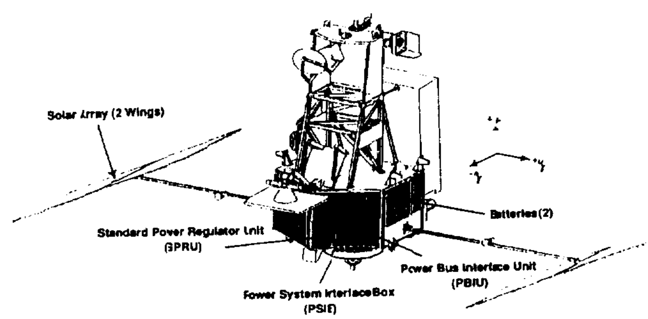

Figure 1 TRMM Observatory

rainfall and its variability is crucial to understanding and predicting global climate change. Knowledge of tropical rainfall is poor because of ground measurement restrictions, especially over the oceans. Use of a low-altitude (350km) circular orbit, at a 35° inclination, allows TRMM to fly over each position on the Earth's surface at a different local time each day providing the first map of rain variation over a 24 hour period. The observatory, shown in Figure 1, weighed 7,751 lbs (3,523 kg) at launch and is about 17 feet tall (5m) by 12 feet (3.6m) in diameter.

SYSTEM LEVEL REQUIREMENTS

GSFC sized the power subsystem to provide 1,100 watts of load capability to the observatory for a three-year mission with up to six additional months for in-orbit check-out. Since a regulated bus was not required for

TRMM, the power subsystem was designed to provide power at 28±7 volts to the instruments. The 350 km altitude produced a 91.5 minute orbit and required that the batteries and solar arrays be designed to perform for over 20,000 low earth orbit (LEO) cycles. The 35° inclination produces a varying beta angle, defined here as the angle between the sunline and the orbit plane, from 0° to ±58.5°. The eclipse varies from 26 minutes, at Beta=58.5°, to 36 minutes, at Beta=0°. A three-axis stabilized attitude control system was required to keep the instruments nadir pointed to within 0.2 degrees. One of the instruments, the Visible Infrared Scanner (VIRS), required an anti-sun side, chosen to be the +Y side, therefore requiring that the observatory perform a 180° yaw maneuver whenever the sun crosses the orbit plane (i.e. the beta angle crosses 0°).

The low altitude requirement was a major design driver for the TRMM power system. Initially, the science team required an altitude of 300 km. However, 350 km was ultimately selected to insure a minimum 3 year mission. The TRMM Observatory carried the maximum capacity (890kg) of Hydrazine required to perform orbit altitude boost burns to overcome atmospheric drag and maintain the altitude within ±1.25km. All Power System designs were therefore completed with a goal towards reducing drag.

POWER SUBSYSTEM DESIGN

The power system consists of three major components: the solar array, the batteries, and the Power System Electronics (PSE)--the Standard Power Regulator Unit (SPRU), the Power Bus Interface Unit (PBIU), and the Power System Interface Box (PSIB).

The TRMM Power System is shown in Figure 2. The controlling unit is the SPRU which performs peak power tracking of the solar array, battery charge control, and provides bus over-voltage protection. A peak power tracking system was considered advantageous for TRMM to minimize the solar array area required and therefore the drag.

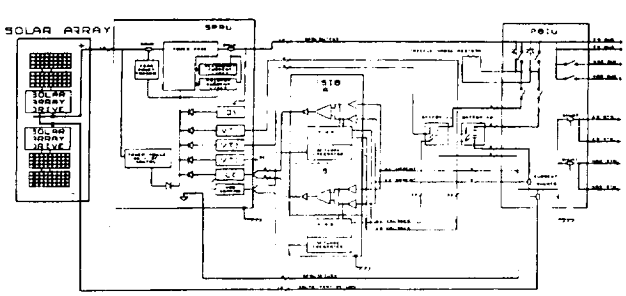

Figure 2 TRMM Power Subsystem Block Diagram

Two 50Ah nickel-cadmium (NiCd) batteries were required to keep the Depth-Of-Discharge (DOD) under 30%. Under normal operation, the two batteries are charged and discharged in parallel. The PSE supplies power to the observatory via four redundant busses: the unswitched, essential bus, powered continuously for launch and mission, the switched essential bus, the switched non-essential bus, and the pyrotechnic bus.

SOLAR ARRAY DESIGN

The solar array sizing, configuration, and substrates were designed by GSFC. TRW performed the detailed component design and solar cell layout fabricated the substrates and solar cell arrays. The solar array consists of two wings. Each wing has two panels canted 26.5° to minimize the effect of the variation of beta angle. The solar panels on -Y side of the observatory do not get shadowed; however, the +Y panels have significant shadowing especially at the high beta angles. The solar array is driven about the Observatory Y-axis to track the sun.

The average power required from the solar array under worst case conditions during the sunlight period as a function of beta angle is shown in Figure 3. The solar array was required to provide 3914W at 58.9V at Beginning of Life (BOL) and 28°C with 1 sun normal to the panel and no shadowing measured at the individual panel output connectors.

Each solar panel is 2.13 x 2.13 m, providing a total solar panel area of 18.1 square meters. Tecstar GaAs/Ge solar cells, with a BOL bare cell efficiency of 18.1%, were used. The use of GaAs instead of silicon cells reduced the array area by 30% and simplified deployment by reducing the number of panels from six to four. This improved deployment reliability. Each panel has 34 strings of 68 cells connected in series. The cells are 4.0 x 4.4 x .02 cm.

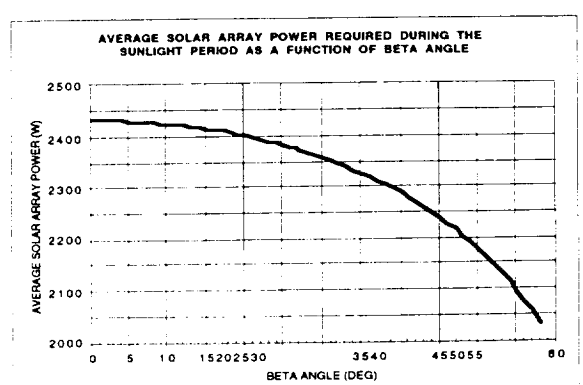

Figure 3 Solar Array Sizing

Each cell is covered with .015 cm thick Optical Coating Laboratory, Inc. (OCLI) ceria doped borosilicate coverglass and is interconnected (8 solder joints between series connected cells) with a TRW designed solder plated Kovar interconnect formed in a U-shape with an out-of-plane expansion loop. The cells are mounted on a substrate comprised 0.15mm thick aluminum (Al) alloy facesheets over 19.6mm thick perforated 5056 Al Core. The facesheets were covered with 0.12mm thick fiberglass epoxy insulator. Each cell is adhered to the substrate with 3 to 5 mil thick CV1-1142.

Each string has redundant positive and negative harness connections. Every two strings on a panel are connected in parallel via the harness before connection to the anodes of two diodes located on the back side of the inboard panel. The cathodes of the diodes are connected to a common solar array bus that is brought into the Observatory via a harness extending across the boom.

The initial estimates for atomic oxygen (AO) fluence were on the order of 8.9×10^{22} atoms per square centimeter. Because AO erodes teflon wire insulation, several steps were taken to protect the solar array harness insulation. The interface to the connectors is covered with Tedlar blankets which have a proprietary OCLI coating that resists AO erosion. The harness extending between the panels and from the solar array wings into the Observatory was covered with Beta cloth. The front side and backside wiring were coated with CV1-1142 silicon adhesive.

The solar array weighs 106.1kg. The substrate weighs 72.2kg. The cell stacks, panel harnessing, connectors, diode boards, etc. mounted to the panels weigh 33.94kg.

BATTERIES

Two (2) Hughes Aircraft Company Super NiCd™ batteries each made up of 22, 50Ah Eagle-Picher prismatic cells connected in series were flown on TRMM. Each battery weighs 60.1kg and is 48 x 30 x 25 cm. Super NiCd batteries were selected for TRMM because of design features that allow for greater LEO cycling life at temperatures up to 30°C. These features include a non-nylon separator, larger electrolyte volume, greater electrode spacing, and electrochemically impregnated positive and negative plates. These features gave confidence, at the time of battery chemistry selection, that two batteries with a maximum of 25% DoD would meet mission requirements for 21,000 LEO cycles at high temperature.

POWER SYSTEM ELECTRONICS

The TRMM PSE was designed to support a maximum load of 1200 watts and up to three batteries. The PBIU and PSIB, however, were fabricated for the two battery configuration of TRMM. With TRMM considered a Class-B spacecraft, the PSE was designed to be single fault tolerant with built-in redundancies. A detailed description of each electronics box follows.

SPRU

The Standard Power Regulator Unit (SPRU), built by Engineered Magnetics, contains electronics to control the operating point on the solar array current-voltage (I-V) curve to maximize available array power to recharge, in parallel, one to three 22-cell Nickel-Cadmium batteries, to provide power to observatory loads and to provide bus overvoltage protection. The SPRU operates in one of four modes: peak power tracking mode, voltage limit mode, current limit mode,.and standby mode. The SPRU peak power tracks whenever solar array power is available and the total load on the SPRU exceeds the maximum power available from the solar array. The total load is the instantaneous sum of the observatory load currents and the battery charge currents at the current bus voltage. The SPRU dynamically adjusts the current drawn from the solar array to meet the total load to within 5% of the maximum power available from the solar array. Whenever the observatory load exceeds the power available from the solar array, the batteries discharge to supply the difference. Peak Power Tracking is terminated when the first battery reaches the temperature compensated voltage limit selected by ground command. The SPRU has eight selectable linear voltage vs. temperature (V/T) levels. The highest, V/T 8, is 33.44±.33V at 0°C. The slope of each voltage vs. temperature curve is -51.26 ± 4.4mV/°C and the separation between levels is 0.44 ± 0.044V. Each battery has its own temperature senser. The warmest battery will reach the V/T limit first, and the SPRU will decrease the current drawn from the solar array reducing SPRU input power and output current. A taper charge current results for both batteries. Current limit mode can be selected by command so that the total battery charge is controlled to one of three constant currents: .75A, 1.5A, and 3.0A. The output signal for the over-voltage mode, V/T mode, and Current Limit mode circuits are or'ed together such that the mode demanding the least amount of output current from the SPRU will control. The standby mode occurs during eclipse. The only circuits which are powered are the command logic. These circuits are powered by the battery via two series bypass diodes from the battery to the SPRU input across the power modules. The SPRU output capability is specified to be 108A or more.

R. Kichak[1] reported details of the SPRU design. The SPRU, designed in the 1970s, has had a very successful flight heritage in the Modular Power Subsystem (MPS) programs: Solar Maximum Mission (SMM), Landsat, Gamma Ray Observatory (GRO), Upper Atmosphere Research Satellite (UARS), and TOPEX. The SPRU was selected for use on TRMM on the basis of its flight success and its cost efficiency over a new design.

Two modifications to the design discussed in reference 1 were made for TRMM. A resistive current shunt was placed in the return path of each of the two batteries. They replaced magnetic current sensors. The signal conditioning for each battery current is done by the PSIB. The SPRU design uses a input voltage signal, proportional to the total battery current, in a feedback loop to control the total battery current to one of three constant current charge levels. The TRMM PSIB adds a programmable gain setting for the total current signal that is fed back to the SPRU to allow for 7 constant current charge levels. The original application of the SPRU used constant current mode to provide a trickle charge current to the batteries when they were fully charged. The TRMM application uses those trickle charge levels and, in addition, uses higher constant current levels (i.e. 6.0A, 12.0A, 24.0A, and 48.0A) to limit the charge rate to the batteries at the beginning of day when the solar array is cold.

The operating input voltage range of the SPRU is 45 to 125V. The solar array was designed to provide a peak power voltage on the low end of this range to increase the operating efficiency of the SPRU buck regulators while providing enough margin to insure that at high temperatures and End of Life (EOL), the solar array voltage would not go below 45V. In addition, lowering the voltage of the array allowed for more solar cell strings which reduces the shadowing losses.

The SPRU weighs 17.2kg and is approximately 70 x 23.5 x 14.4 cm.

PBIU

The Power Bus Interface Unit (PBIU), designed by GSFC and fabricated by Jackson and Tull Inc. (J&T) and Litton Amecom, contains the components for directing power to the essential, non-essential, and pyro power busses and for disconnecting the batteries from the power busses when commanded by the PSIB or spacecraft ground support equipment. It also contains the PSE bus capacitors, the non-essential bus relays, the current sense shunts, the battery reconditioning circuits and resistors, and ground power interface. A block diagram is shown in Figure 4. The PBIU is a NASA/GSFC design for the X-Ray Timing Experiment (XTE) Program launched in December 1995.

The PBIU weighs 20.4 kg and is 31 x 42 x 16 cm.

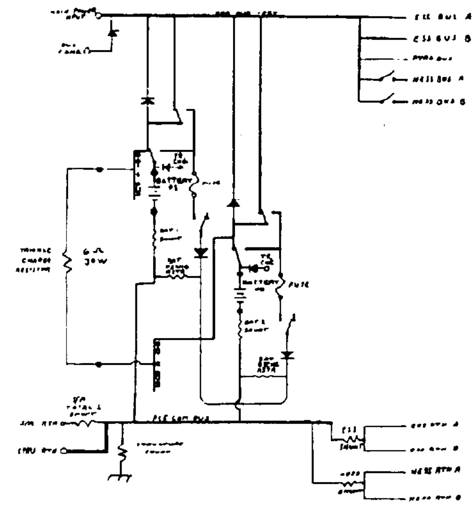

Figure 4 PBIU Block Diagram

PSIB

The Power System Interface Box (PSIB), designed by GSFC and fabricated by J&T and Litton Amecom, provides the command and telemetry interface between the power system and the TRMM observatory and performs all internal command and status monitoring functions for the power system. The PSIB provides 253 telemetry words and fast sampling of 1 telemetry point up to 1 kHz. The TRMM observatory design was primarily a modular, distributed architecture, using a MIL-STD-1773 fiber optic data bus as the main interconnect between the various independent spacecraft subsystems. The PSIB controlled and monitored all discrete data functions within the power subsystem and handled all command and telemetry interactions with the TRMM Command & Data Handling (C&DH) subsystem over the fiber optic bus. It also performed a variety of autonomous internal power system functions.

The PSIB consists of a processor card and various discrete I/O cards, communicating over an internal backplane. Two sets of cards were included in a single enclosure to provide complete block redundancy. The processor card included a 32-bit rad-hard R3000 type microprocessor (Mongoose-I) with SRAM, EEPROM, an RS-422 serial interface and a 1773 fiber optic bus interface. I/O cards for relay control/monitoring, thermistor/PRT monitoring, analog telemetry/commanding and digital telemetry/commanding completed the configuration. All of the PSIB cards were built using surface mount components to reduce the PSIB box size and weight.

Some of the autonomous internal power system functions that the PSIB performs are:

1. Ampere-hour (Ah) Integration

The PSIB samples the battery current at a rate of 250 times a second and increments or decrements the Battery 1 and Battery 2 State-Of-Charge (SOC) counters accordingly. The Battery SOC counters provide the percentage SOC that the battery currently has assuming 50 Ah of capacity at 100% SOC. In charge, the PSIB divides the Ampere-hours into the battery by a ground commandable Charge-to-Discharge (C/D) ratio. The C/D ratio is designed to compensate for battery inefficiencies.

2. Autonomous Commanding Of The SPRU

The PSIB issues a ground selectable command to the SPRU every time the first battery goes below a ground selectable threshold for SOC and will issue a second ground selectable command to the SPRU every time both batteries hit 100% SOC. This function can be disabled by ground command.

3. Low Power Detection

The essential bus and all 44 battery cells (or any subset thereof determined by ground command) can be monitored for low power. Low power is defined by ground commandable thresholds for the essential bus voltage and the battery cell voltages. If any telemetry point drops below its threshold, low power is detected and causes the non-essential loads to be shed. This function can be disabled by ground command.

The PSIB weighs 14kg and is approximately 36 x 27 x 18 cm.

POWER SYSTEM FAULT DETECTION & CORRECTION

The Battery SOC is used for fault detection and autonomous safing of the observatory. The spacecraft processor monitors the Batteries' SOC. If either Battery goes below 70% SOC, the spacecraft processor reconfigures the SPRU into peak power tracking mode, sets the V/T limit to the V/T 5 curve, and powers off the non-essential bus loads.

The Power System does not generate any command to send the observatory into safehold. It does, however, respond to a system safehold command by disabling the non-essential bus loads. Upon safehold detection, the TRMM observatory slews so that the Observatory +x axis is inertially fixed on the sunline providing maximum array output without the solar array drive.

INTEGRATION AND TESTING (I&T)

The Power Subsystem flight components went through a very extensive test regime at the component, subsystem, and Observatory level outlined in the Table below.

TABLE I: Power System Testing

Test	Solar Array	Batteries	SPRU	PBIU	PSIB
Random Vibration		V	V	GC	GC
Sine Burst		V	V	GC	GC
Sine Sweep	O	V,O*	V,O	GC,O	GC,O
Mechanical Shock	O	O*	O	O	O
Mass Properties	V	V	V	GC	GC
EMI/EMC		O*	V,GS,O	GS,O	GS,O
Thermal Vacuum	V	V,O*	V,GS,O	GS,O	GS,O
Thermal Balance		O*	O	O	O
Bake Out	V	V,O*	V,GS,O	GS,O	GS,O
Acoustics	V,O	O*	O	O	O
Magnetics	A	A,O*	O	O	O
Leak		V,GC			
Life Test	V*	V*			

V=test done at Vendor at Component Level

GC=test done at GSFC at Component Level

GS=test done at GSFC at Subsystem Level

O=test done at GSFC at Observatory Level

A=analysis

*=Test Performed With Qualification Unit Not Flight Unit

A=analysis

To minimize the risk of schedule impact to the Observatory level Integration and Test (I&T), Engineering Test Units (ETUs) of the PBIU and PSIB were built. They, together with a GRO flight spare SPRU, were integrated and functionally tested at the subsystem level ahead of the flight units to flush out problems. They also became I&T spare units in the event a flight unit would need to be deintegrated from the Observatory for independent testing and/or rework.

A distinct advantage of the TRMM PSE for I&T was the separation of the umbilical power interface to the Observatory and the Observatory Power Bus from the rest of the PSE in the PBIU. It enabled us to deliver the ETU PBIU to the Observatory for an early integration of the C&DH and enabled I&T to continue when a relay driver inside the flight PSIB failed requiring removal of the flight unit for repair.

In addition to the traditional complement of tests outlined above, the flight solar array went through extensive visual inspections and ambient pressure thermal cycling tests after delivery to GSFC to determine if solar cell cracking would cause power loss in flight.

TRW personnel inspected both wings after integration, observatory level mechanical testing and two thermal cycles for the -Y wing and one thermal cycle for the +Y wing. The -Y wing had 56 (1.2%) of its cells cracked and the +Y wing had 51 (1.1%) of its cells cracked. Six additional thermal cycles were done on the +Y wing to determine if additional cells would crack. Thirty-five cells cracked after two thermal cycles; 22 cells cracked after four thermal cycles; and 13 cells cracked after six thermal cycles for a total of 121 cells cracked (2.6%). There was no power reduction or anomalous I-V curves at elevated temperature for any of the panels. Some of the cells with cracks which could open a string were replaced and the -Y wing was launched with 0.99% cracked cells and the +Y wing was launch with 1.99% of its cells cracked.

Although the crack rates were higher than expected and the ambient pressure thermal cycles produced additional cracks, according to the visual inspections, the TRMM panels were launched based on the nominal elevated temperature I-V curves and the decreasing crack rate with additional cycles.

Battery cell testing was performed at COMSAT Laboratories to determine the expected loss in capacity that occurs when Super NiCd cells are stored 30 days in the open-circuit condition. These tests were performed at 0°, 10°, and 28°C. The cell manufacturer provided a guideline not to store the batteries open-circuit discharged for more than 14 days. The TRMM project wanted to install the batteries prior to Observatory shipment to the launch site in Japan. Fourteen days, within the guideline, was not enough time to accomplish this.

At Comsat, six Super NiCd cells were initial capacity tested and stored for two 30 day periods with capacity checks at 0°C, 10°C, and 20°C after each. The results are summarized in the Table below. Although all the cells lost capacity after the first 30 day storage, the cells stored at cold temperatures (0° and 10°C) exceeded their initial 0°C and 10°C capacities after the second storage period. One of the cells stored at 28°C did not regain the initial 0°C and 10°C capacities after the second storage but it was only 1.3% lower. All cells lost 20°C capacity after both storage periods. Based on the 10°C flight predictions for the TRMM batteries, storage with the batteries open-circuit and discharged at 28°C on the Observatory for shipment was approved.

TABLE II: Results Of 30 Day, Open-Circuit, Discharged Battery Storage Tests

	Cell 1	Cell 2	Cell 3	Cell 4	Cell 5	Cell 6
Storage Temperature	0°C	0°C	10°C	10°C	28°C	28°C
Initial Test						
20°C	59.8	59.2	58.9	58.9	59.3	59.3
10°C	61.7	61.3	60.8	61.0	61.1	61.3
0°C	62.6	62.1	61.6	61.9	61.9	62.1
After 30 Days Storage						
20°C	57.8	57.7	57.4	57.5	56.8	56.7
10°C	59.6	59.4	58.9	58.7	56.8	57.0
0°C	60.9	60.5	59.9	59.6	57.7	58.2
After Second 30 Days Storage						
20°C	57.0	56.7	56.8	56.8	55.9	55.9
10°C	63.6	62.9	62.5	63.1	61.2	60.5
0°C	65.0	64.2	63.7	63.9	62.1	61.3

IN FLIGHT PERFORMANCE

The PSE has operated nominally in flight. A typical charge profile for the battery from Launch through early February 1998 is shown in Figure 5. The SPRU provided full peak power from the solar array, charging the batteries at a maximum of 46A, until V/T level 5 was achieved. The V/T limit was reached early in the charge cycle, after 5 to 10 minutes. The batteries were then charged with a tapering current, to maintain the V/T limit, until the C/D ratio of 1.05 was reached where the SOC for both batteries was 100%. The taper charge lasted 16-20 minutes. The trickle charge setting was 0.75 Amps (0.375 Amps per battery). The trickle charge period was approximately 33 minutes. The PSIB commanded the SPRU back to peak power tracking mode whenever the Battery SOC went below 98% (indicating eclipse).

The observed power level for beta angles ranging from -58.5° to +16° was 700W (average). This is considerably lower than the 1100W design requirement.

During the first 60 days of the mission, the essential bus voltage varied from about 32.4 Volts (0.7 Volts higher than the batteries) to 27.5 Volts (0.2 Volts lower than the batteries).

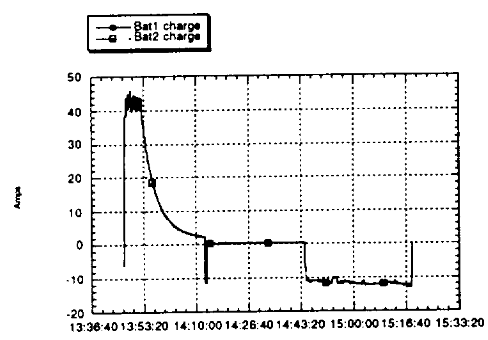

Figure 5 Typical Early Mission Charge Cycle (Orbit #625 on Day 006)

Battery temperatures have been stable. Battery-1 has operated between 5.8°C and 10.8°C and Battery-2 has operated between 8.8°C and 12.2°C. The temperature varies with the beta angle with the highest battery temperatures occurring near beta angle 0° (Figure 6). This is well below the specified maximum of 20°C.

The maximum DOD has been between 14%, at Beta=-58.5°, and 16%, at Beta=0°. These are well below the specified maximum of 25%.

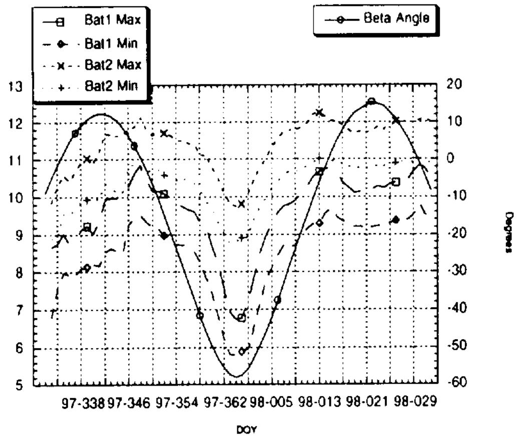

Figure 6 Battery temperatures and beta angle

Excessive voltage has developed on Cell 1 of Battery 2 during charge near the point where the V/T Limit is reached. A plot of the Battery 2 Cell 1 daily maximum voltage is shown in Figure 7 along with a few other representative cells from the battery. The cause of the high cell voltage is unknown. The following corrective actions were taken in early February to avoid further increases in the voltage:

1. Lowered the C/D ratio from 1.05 to 1.03 (1.01 temporarily)
2. Lowered the V/T level from V/T 5 to V/T 4 temporarily
3. Lowered the charge rate from 49A to 12A per battery

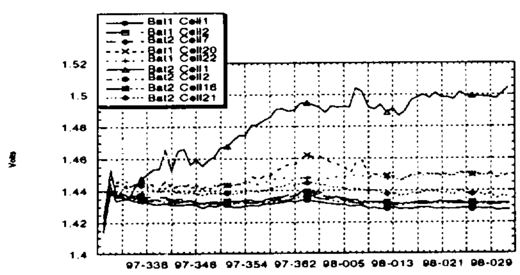

Figure 7 Battery Cell Voltage Peaks

As of late April, no further increase was detected. The batteries are being monitored continuously in flight for changes.

The average solar array power output, while peak power tracking, is approximately 3833W which is significantly greater than that which is required to provide energy balance for a 700W load. An analysis of the flight data is ongoing to determine if any effects of solar cell cracking can be detected.

The current versus voltage (I-V) curve was measured on the ground for each of the four solar panels at the panel output connectors. The individual panel's I vs. V curves were corrected for diode and harness drop and then added to produce a ground test "baseline" I-V curve for the total solar array. The solar array in-flight performance for the first 60 days of the TRMM mission was analyzed. For each orbit, the total solar array current, array bus voltage, and individual panel temperature telemetry were collected from sunrise, when the SPRU was peak power tracking, until after the V/T limit was reached and the solar array current is reduced. Each orbit's flight telemetry was adjusted from flight conditions to ground conditions for comparison to the ground test "baseline" I-V curve.

The normalization of the flight data consists of adjustment of each flight current and voltage telemetry points from their value under flight conditions to the ground test conditions (i.e. 28°C, 1 sun, normal sun, no shadowing). The table below provides the flight conditions for each day analyzed. Each flight current and voltage data point corresponded to a different temperature. The average of the four flight panel temperatures was used to adjust the flight panel current and voltage to 28°C using the coefficients -1.93mV/cell/°C and 17.1uA/cm^2/°C. Each point was then plotted against the ground test "baseline" I vs. V curve.

Day	Beta Angle	Cosine Angle	# Shadowed Strings	Intensity
97-332	-8.85	-18	3	1.028
97-351	-10.67	-16	3	1.030
97-356	-32.7	62	10	1.030
97-362	-55.3	28.8	11 to6	1.030
98-010	-19.2	-7.3	4	1.033
98-016	4.96	-22	3	1.030
98-026	11	-16	3	1.030
98-036	-25.2	-1.3	4	1.030

The result of 8 orbits is plotted below.

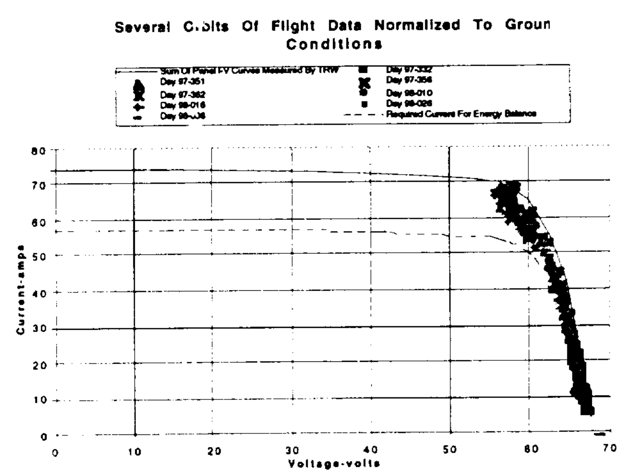

Figure 8 TRMM Solar Array Flight Performance Normalized To Ground Conditions

The normalized flight data should lie on the "measured" curve. The normalized flight data curves have the following characteristics: large data scatter; lower than expected voltage; an apparent second knee in the I-V curve.

The large data scatter may be attributed to aliasing of the data. The current and voltage are not sampled at the same point in time. In addition, the SPRU dithers the current at 70Hz and the amplitude of the dither is as much as 7.5A peak to peak. The voltage follows the current along the I vs. V curve at 70Hz. The maximum change in voltage is approximately 3V peak to peak. In addition, the PSIB samples the current and voltage at 10Hz. It sends down the current at 1Hz and the voltage at .03Hz. The voltage is averaged over 2 sec before it is sent down. To determine what the true normalized I-V curve is for the flight panels in orbit, a large data sampling is being compiled for statistical averaging.

The lower voltage may be accounted for by additional voltage drop through the system that we did not estimate. The voltage is higher than required.

The apparent second knee in the curve is the area of most concern, because TRMM has a peak power tracker which starts at open circuit voltage and moves up the I-V curve until it reaches a knee. In the worst case the SPRU could peak power track around the lower knee reducing the amount of power available from the solar array. The TRMM peak power tracker is overcoming this first knee and going up to the full peak power of the array. This indicates that the first knee is not rounded enough to cause the electronics to dither there. The apparent second knee in the curve is still under investigation. Once the statistically averaged normalized I-V curve for the flight panels is determined, the true magnitude and shape of the knee will also be determined. From the limited data set presented here, the power at the second knee is approximately 10% higher than the required power at BOL and 5% higher than the required power at EOL. Because of the large data scatter and the limited number of orbits looked at, absolute conclusions about the apparent knee can not be drawn at this time.

A second knee in the I vs. V curve could be caused by the solar cell cracking or shadowing. Shadowing does not seem a viable cause. The number of strings shadowed on Day 98-016, Beta=4.96° is only 3 out of 136 strings. This should produce a 2.2% reduction in power and we are seeing a knee with an approximate 10% reduction in power. In addition, other days which have more severe shadowing, day 362 for example, do not have a worse second knee.

Cracked cells seems the most likely physical cause for a second knee. We launched with 1.5% of cells cracked and with 6 strings containing cracks which, if they propagated entirely through the cell, could open a string. Although the string level I-V curves on the ground were normal, the cracks could have propagated. Additional work is in progress to obtain a cracked cell scenario that would fit the result.

LESSONS LEARNED

1. Power, thermal, and structural subsystem designers should design the battery location to assure mechanical access to the batteries for easier changeout if needed. The TRMM batteries were placed, for thermal reasons on the +Y equipment panel (anti-sun side). Unfortunately, there was no access to the batteries when the +Y solar array wing was stowed. Had the batteries needed to be changed out at the launch site, deintegrating (both mechanically and electrically) the +Y solar array wing and the batteries, re-integrating the batteries and solar array wing and requalifying the wing mechanical integration via deployment test would have taken a minimum of two weeks.
2. Store and telemeter the end-of-eclipse battery voltage for quicker assessments of energy balance.
3. Recently, electrical performance testing at elevated temperature and ambient pressure has been performed on GSFC solar arrays. Testing at increased temperature has the advantage of allowing the array to be fully characterized while under thermal stress, at a much reduced cost and in a much shorter time than the same test in vacuum. The test, therefore, can be repeated at several stages throughout the integration and testing of the array, if desired. An elevated temperature flash test was performed on the TRMM array with no measurable power loss or anomalous I-V curve shape. According to inspection results, however, ambient pressure thermal cycling did induce further cracking of the cells on the TRMM array.
4. Although the exact cause of the TRMM solar cell cracking post lay-up was not determined, solar arrays utilizing GaAs/Ge solar cell technology should be built on substrates which have a coefficient of thermal expansion which more closely matches that of Germanium.
5. The solar array voltage and current should have been sampled by the telemetry system at the same point in time to allow for the reproduction of an I-V curve for the flight array.

REFERENCES

1. R. A. Kichak, "Standard Power Regulator For The Multi-Mission Modular Spacecraft," Proceedings of the 14th Intersociety Energy Conversion Engineering Conference, August 1979.