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Design of a Low Cost Electrical Power Subsystem for a Small Satellite

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Abstract. Students at the University of Colorado at Colorado Springs (UCCS) and the United States Air Force Academy (USAFA) are designing, building, and testing a spacecraft scheduled to fly on a Lockheed Martin Atlas/Centaur in October, 1997. The primary mission objective is to capture Global Positioning System (GPS) data from above the GPS constellation for use in orbit determination.

This paper describes the design and high altitude balloon flight test of the low cost, compact, self-contained electrical power subsystem (EPS) which supports this mission. This EPS design supports the two week mission without on-orbit charging. Commercial off-the-shelf components were used and this subsystem was built and tested in an academic environment.

Design and schedule issues, test procedures, and balloon flight test results are discussed. EPS component designs are discussed in detail. A key design factor was safety as this mission is a secondary payload. An overview of the satellite mission is included.

Mission Overview

The mission, named Falcon Gold, is a joint project between the University of Colorado at Colorado Springs (UCCS) and the United States Air Force Academy (USAFA). It places a small payload in orbit attached to a Lockheed Martin Centaur upper stage booster. After inserting its primary payload, this booster will be in a geosynchronous transfer orbit, providing the Falcon Gold spacecraft the opportunity to collect Global Positioning System (GPS) data from above the GPS constellation. Since GPS satellites direct their energy toward the Earth, Falcon Gold will receive signals past the limb of the Earth from GPS satellites on the opposite side of the constellation.

Falcon Gold will transmit the data back to ground stations located at UCCS and USAFA. The data will be used to investigate feasibility of orbit determination using GPS from above the GPS constellation. An additional goal is to qualify the spacecraft subsystems for use in future missions.

The mission is due to launch in October, 1997. It takes its name from the USAFA mascot, the Peregrine Falcon, and gold from the University of Colorado school colors.

Introduction

The Falcon Gold spacecraft is powered by the Electrical Power Subsystem (EPS). This subsystem takes energy from 28 batteries and provides regulated and unregulated voltages for a GPS receiver, a flight computer, a transmitter, and a terminal node controller (packet modem). Some of these voltages are switched under flight computer control. The EPS also provides a ground support equipment (GSE) interface. The GSE interface is used during testing and pre-launch operations for monitoring and control of spacecraft functions.

The spacecraft is constructed of two metal boxes stacked on a mounting plate. The GPS and transmitting antennae are mounted on top of the upper box. The EPS, including all batteries, occupies the lower box. Connections between the EPS and the rest of the spacecraft pass between the two boxes inside the spacecraft. See Figure 1.

The EPS consists of three main sub-assemblies: batteries, the EPS printed circuit board (PCB), and payload/GSE connections. These main functional blocks are shown in Figure 2.

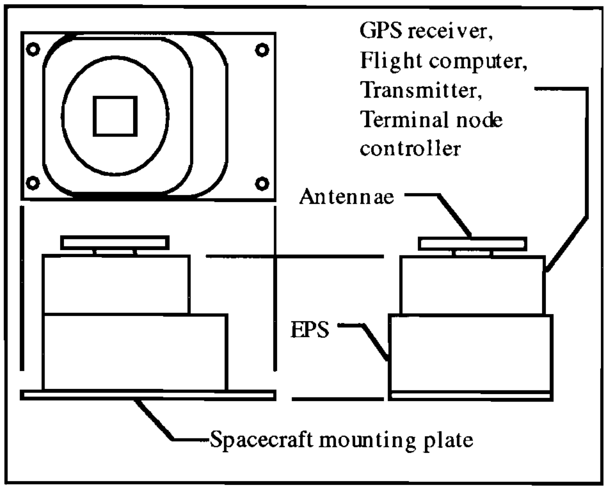

Figure 1 - The Falcon Gold spacecraft.

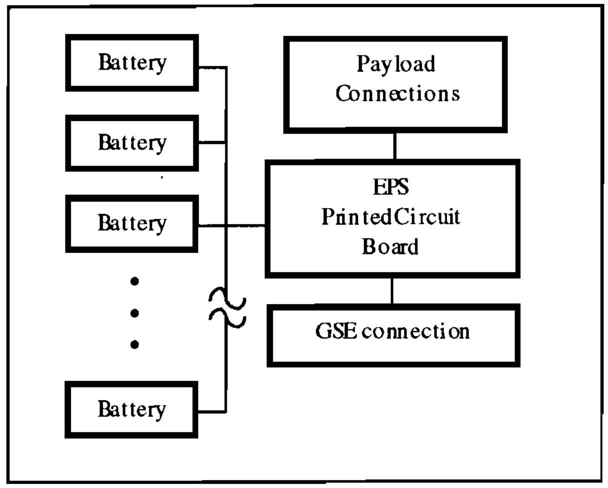

Figure 2 - EPS Block Diagram

The batteries provide a nominal voltage of 12.5VDC with a nominal capacity of 1.5Ah each. There is no on-orbit charging, so the batteries are used in a primary role. Each battery contains integral safety devices and a temperature sensor used to monitor temperature during battery charging on the ground.

The EPS PCB provides spacecraft activation, regulation, and power distribution. This PCB is the heart of the EPS.

The payload and ground support equipment connections provide power and telemetry connections to the rest of the spacecraft, and monitoring and control points to the GSE connector.

The only control on the spacecraft is a master switch on the EPS box.

The Falcon Gold spacecraft was designed, the prototype built and successfully flown, on a high altitude balloon in approximately two and a half months. This ambitious schedule required shortcuts to be taken in the EPS design. In general, these shortcuts contributed to over-design and decreased efficiency primarily because components had to be chosen quickly. Thus, components were selected based on availability and were chosen with conservative design margins. Once a component passed the balloon flight test, the design of that component was frozen since there was neither time nor resources to re-test changes. In effect, the design was frozen while requirements were still solidifying.

Batteries

The Falcon Gold batteries were designed to provide maximum, safe power to the spacecraft. Nickel-metal hydride (NiMH) batteries were used in order to qualify them for use in future missions. Unlike Falcon Gold, these missions will have on-orbit charging capability. The batteries were constructed from commercial off-the-shelf cells.

The Falcon Gold batteries incorporate a fuse for over current protection and a temperature cutoff device to prohibit current flow at high temperatures. The fuse protects battery wiring, connectors, and printed circuit board traces. The temperature cutoff protects against charging or discharging the battery if its temperature is too high. Current flow at sufficient levels, such as high rate charging or fault currents during discharging, can cause an increase in battery internal temperature, leading to an increase in internal cell pressure. If the battery is already at an elevated temperature, this combination of conditions can cause the battery to out-gas.

Construction

A simple, lightweight design was chosen wherein the cells were bonded together using silicon adhesive. Intra-cell connections were made using tabs spot-welded between the cell ends. See Figure 3. The fuse served as the cell-to-cell connection at one end of the battery, making it accessible for replacement. The temperature cutoff was connected in series with the battery as one of the cell-to-cell connections, also. This provides a battery with ten series NiMH cells for a nominal voltage of 12.5VDC with integral over-current and over-temperature protection.

In addition, a 3-terminal integrated circuit temperature sensor was bonded to one of the center cells. This

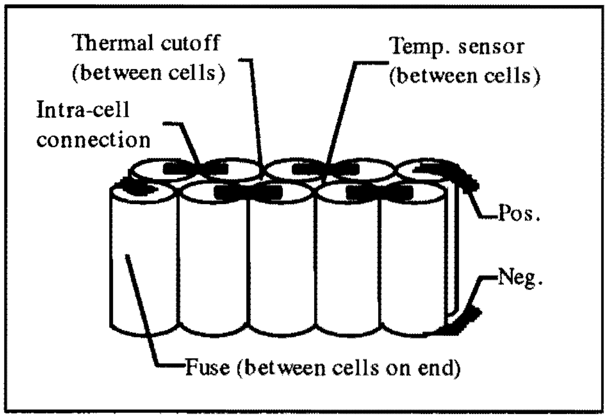

Figure 3 - Battery construction details.

sensor measures battery temperature for determining charge termination. This type of sensor was chosen for its simple implementation and direct output of 10mV/°C. This enables a voltmeter set to read millivolts to read temperature directly (except for the decimal point which is easily shifted to the left one digit mentally). This makes it easy to monitor temperature while charging or discharging batteries.

The fuse chosen was a 3A, fast acting type. This current rating is more than 200% of the maximum expected spacecraft load, therefore, each battery is capable of supporting the entire spacecraft. This gives adequate margin for space environment effects on the fuse while protecting the wiring, connectors, and printed circuit board traces under fault conditions.

Mounting

Inside the EPS box, the batteries are sandwiched between Teflon battery mounting plates in two layers. The plates provide both insulation and restraint. With through-bolts and lid brackets holding the sandwich to the spacecraft mounting plate, the battery mass is connected nearly directly to the spacecraft mounting plate. This provides a direct force path to the spacecraft mounting plate for the high mass batteries. See Figure 4.

This construction allows a minimum amount of structural strength in each individual battery yet still provides a system robust enough to survive the launch environment. The stack consists of an insulating bottom plate, a layer of batteries, a middle plate separating the two layers of batteries (and separating the batteries from each other), a top layer of batteries, and a top plate. The top plate is held down around its perimeter by lid mounting brackets attached to the EPS box lid. This compresses the top plate edges when the lid is installed. Through-bolts are used to compress the

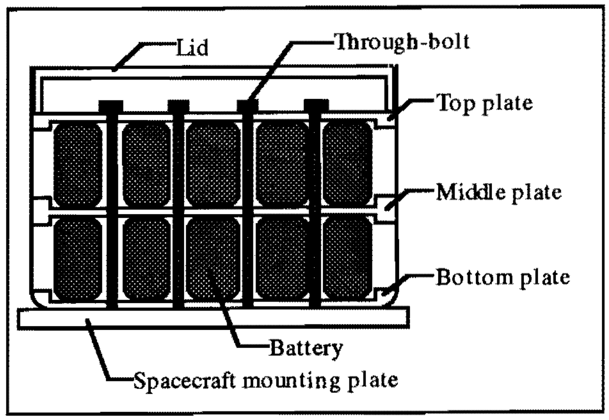

Figure 4 - Cross section of battery mounting structure.

middle of the stack and hold it directly to the spacecraft mounting plate.

Use of a soft material such as Teflon for these plates is risky from a structural viewpoint. This apparent weakness is mitigated in this design by restraining the plates along nearly their entire perimeter, and then using a sufficient number of through-bolts through the middle span of the plates to spread forces.

Testing

There were two groups of tests performed on the batteries: safety and capacity. Safety testing is designed to determine that battery safeguards function properly and that the mechanical design of the batteries and battery mounting is sound. Capacity testing verifies the capacity of the batteries and improves the prediction of mission lifetime.

Four tests comprise the safety tests for these batteries: mechanical, over-current protection, over-temperature protection, and out-gassing.

Mechanical testing determines if the battery construction and mounting are strong enough to survive launch. This test consists of mounting the batteries in the spacecraft and performing system level shock and vibration tests that exceed the launch environment. This test is performed first so that the rest of the tests include possible effects from shock and vibration.

Over-current protection testing consists of applying a constant current load to the battery under test and determining the hold time and trip time of the fuse. Hold time is measured as the time the fuse holds 100% of the fuse rating until the battery under test is completely discharged. Trip time is measured from the time the load is applied until the fuse blows. The fuses will be tested at the levels specified in Table 1.

Table 1 - Battery fuse test levels.

% Rated Load	Test Current	Battery Rate	Hold or Blow time
100%	3A	2C	Battery discharged
200%	6A	4C	60s, max

Over-temperature protection tests verify that the thermal breaker operates properly. This test consists of heating the batteries and measuring the temperature at which the thermal breaker opens. In this case, the trip point of the breaker was chosen at 50°C. This temperature allows for battery operation even under anomalous conditions, yet prevents current flow under conditions that might cause out-gassing.

The out-gassing test consists of heating the batteries to 50°C, placing the cells in a vacuum, and determining if venting takes place. The problem with venting is that NiMH batteries can vent an explosive mixture of hydrogen and oxygen gasses. This test is performed in conjunction with the over-temperature tests to take advantage of the elevated temperatures required during that test.

It should be noted that both the over-temperature and out-gassing tests could be destructive to the battery. Heating these batteries beyond 40°C can damage their ability to accept and hold charge.

Capacity testing involves fully charging the batteries, then measuring their ability to deliver current over time. This is performed by timing how long the battery can deliver a constant current until it reaches its end of discharge (EOD) voltage. In this case, EOD occurs at 10.0VDC. One benefit of this method of capacity measurement is that the batteries are put through several charge/discharge cycles. NiMH batteries must be cycled to reach 100% capacity after long periods of storage. Usually, 3 or 4 cycles will bring the batteries to 100% capacity.

Ideally, prior to building batteries, the cells used are matched by individual cell capacity. This guarantees maximum available capacity from each battery as each cell dies at nearly the same time. Due to schedule constraints, this cell matching was not accomplished for the Falcon Gold batteries. This could impact mission lifetime, but the degree of impact is likely to be small given the brief mission lifetime of two weeks. An additional benefit of matching the cells is that each cell is cycled several times and the poorest performing cells are culled before constructing the batteries.

EPS Printed Circuit Board

The EPS PCB is the core of the EPS and provides spacecraft activation, voltage regulation, and power distribution to the rest of the spacecraft. It provides connections for all batteries, the GSE, and power connections to the GPS receiver, flight computer, transmitter, and terminal node controller. There are also EPS telemetry connections to the flight computer for temperature and main bus voltage.

Spacecraft Activation

Among the design criteria were requirements to prevent spacecraft activation until the primary payload was safely away from the booster. These requirements led to a 3-step activation process. First, the master switch must be closed. This is done on the ground during final launch preparation. The master switch is a key switch on the outside of the spacecraft. It is the only control on the exterior of the spacecraft. This switch applies power to two series-connected pressure switches. During ascent, as the rocket passes approximately 45,000 feet, the pressure switches close. Two switches are used in series to provide tolerance against a single switch failing closed. With the pressure switches closed, power is applied to a fail-safe timer. Once the timer times out, it closes a latching relay. This applies power to the main spacecraft bus. See Figure 5.

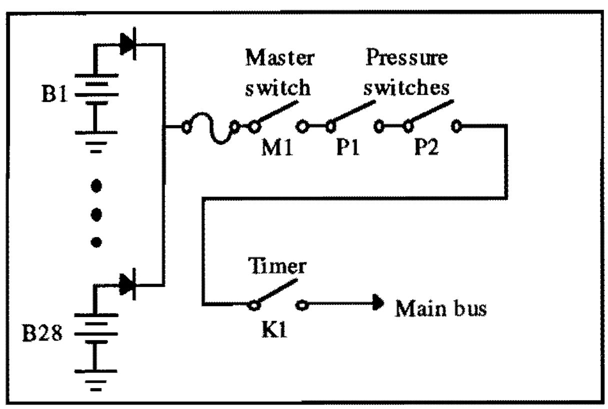

Figure 5 - Battery connection and activation schematic diagram.

The timer is set for 4000 seconds. This is the estimated amount of time from lift off until the primary payload is safely away from the booster, plus a safety factor. This timer protects against pressure switch failure during liftoff. Even if both pressure switches fail due to launch vibrations or shock, the payload will not activate for another 1 hour, 6 minutes, and 40 seconds.

Table 2 - Power sources required.

Device	Voltage (VDC)	Current (mA)	Switched
Timer	5.0	5	No
Flight Computer	8.0	150	No
TNC†	12.0	45	Yes
GPS Receiver	8.0	120	No
Transmitter	13.6	900	Yes

Voltage Regulation

Several power sources are required for the payload devices and the EPS PCB itself. These are all generated on the EPS PCB and are listed in Table 2. The requirements shown in the table are the maximum power required. In the case of the timer, this power is only required while the timer is operating, the first 4000s of the mission. Once the timer times out, the timer current draw drops to less than 1μA. In the case of the GPS receiver, it only draws the maximum while acquiring and transferring data. The flight computer also has similar "sleep" characteristics. The transmitter and terminal node controller have power switched to them under flight computer control. This minimizes unnecessary power draw when the devices are not needed.

There is no on-orbit charging, so this power system operates in only one mode, discharge.

The spacecraft acquires, processes, and transmits data at 5-minute intervals. Once every 5 minutes, the flight computer wakes up, activates the transmitter and terminal node controller, wakes the GPS receiver and directs it to acquire data, transmits the data, turns off the transmitter and terminal node controller, then goes back to sleep. The GPS receiver goes to sleep on its own. The on and off times and the current and voltage requirements for each device are shown in Table 3.

Table 3 - Voltage, current, operating time, and average power consumption of each payload device.

Device	V_{nom}	I_{min}	I_{max}	t_{on} (s)	t_{off} (s)	P_{ave} (mW)
Computer	8.0	150μA	150mA	15.0	285.0	61
GPS Receiver	8.0	900μA	120mA	12.0	288.0	45
TNC	12	0.0	45mA	9.95	290.05	18
Transmitter	13.6	0.0	900mA	9.95	290.05	293

Based on specifications for each device, regulators were needed for 5V, 8V, and 13.6V (the terminal node controller has a wide enough operating voltage range to run directly from unregulated battery voltage). The current requirements for the 5V regulator are very small in that it only provides power for the timer circuit, so a linear regulator was chosen. The 13.6V regulated supply automatically requires a switching regulator since the output voltage is above the nominal battery voltage. The 8V regulator could be either switched or linear given the nominal battery voltage. Calculating the required power at 8V, the input power required for a linear and switching regulator with a given efficiency can be determined. Then, with the input power determined for each case, the losses can be evaluated. This calculation is summarized in Table .

The difference in power consumption, or power lost, between the linear and switching regulators is 21mW. This is on the order of 14 hours lost out of a 23 to 25 day estimated mission lifetime or approximately a 2.5% loss. Given the additional complexity and parts count of a switching regulator, a linear regulator was chosen for the 8V supply.

The 13.6V regulator is a boost topology switching regulator. It converts unregulated battery voltage to 13.6VDC. Since it is a switching regulator, its efficiency is a function of the input and output voltages and currents. The output voltage is constant as is the current drawn by the transmitter, therefore regulator power output is constant. The input voltage varies over the range of battery voltages from fully charged (13.5VDC) to dead (10.0VDC).

The efficiency of the regulator over the input range

Table 4 - 8V power consumption.

$P_{req'd}$	106mW	106mW
Regulator	**Switching**	**Linear**
Efficiency	0.77	0.67
P_{in}	138mW	159mW

† Terminal Node Controller

described above was measured. This was performed using a constant current load set to 900mA. The input and output voltages and currents were measured and the efficiency calculated. The results are shown in Figure 6. The nominal efficiency is approximately 87.5%.

Power Distribution

The EPS PCB is the connection point for all power input and output. There are 30 battery input connectors (28 for the mission plus 2 spares). The batteries are connected with 5-pin locking connectors. There is one connection each for the battery positive and negative, and three connections for the temperature sensor. Power for the payload devices is distributed using pigtails directly from the EPS board to 5-pin locking connectors for each device. Each wire is strain-relieved where it leaves the PCB by a small amount of silicon adhesive placed around the connection. Power wires are laced into a cable toward the center of the PCB where it exits the EPS box and enters the payload box.

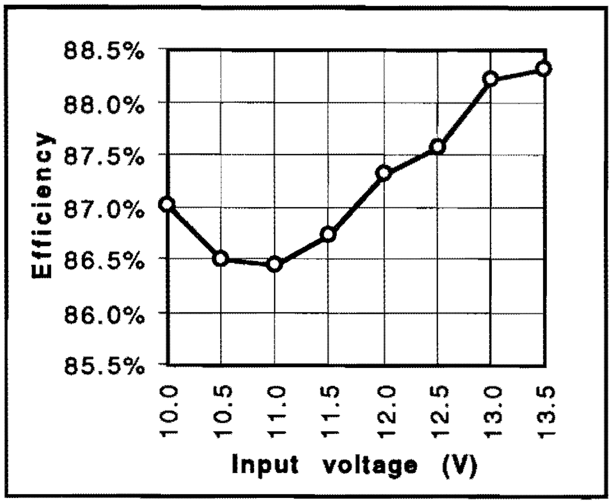

Figure 6 - 13.6V regulator efficiency.

GSE Connections

The GSE connection provides a way to monitor and control the spacecraft while on the ground. Through the GSE connector, the batteries can be charged or discharged. The spacecraft can be activated and voltages can be monitored. The fail-safe timer can also be set or reset.

Telemetry

There are two telemetry points in the EPS: main bus voltage and EPS temperature.

The main bus voltage is tapped off the output of the timer and converted to a useful range for the flight computer A/D converter. The conversion is accomplished using a simple combination of a precision Zener diode in series with a potentiometer. See Figure 7. The potentiometer, R_{lim}, limits Zener current to the proper range and provides scaling of the resultant voltage. The potentiometer also provides a method of calibration.

This conversion scheme has the advantage of simplicity. While it does not make maximum use of the A/D input range, it does provide approximately 1.3mV/count resolution while drawing a nominal current of less than 180μA while activated (the circuit is switched on by the flight computer only when measurements are being taken).

With the values of Zener voltage and potentiometer used, the voltage telemetry output is quite linear in the range of interest of bus voltage. The range of interest of main bus voltage is from 13.5V (freshly charged batteries) to 10.0V (dead batteries). Theoretical and measured outputs of this circuit are shown in Figure 8. As can be seen from the graph, the telemetry voltage tracks well with the expected voltage (in the range of interest).

Since the relationship between the bus voltage and telemetry voltage is linear in the range of interest, a simple equation is used to reconstruct actual bus voltage from voltage telemetry readings.

The A/D converter is a 12 bit (4096 count), 4.096V maximum input converter. It is the count value that is returned as the telemetry value. From Figure 8, the

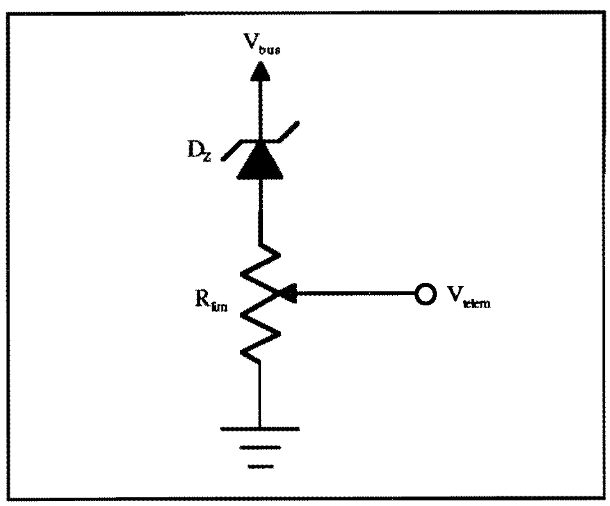

Figure 7 - Voltage telemetry circuit.

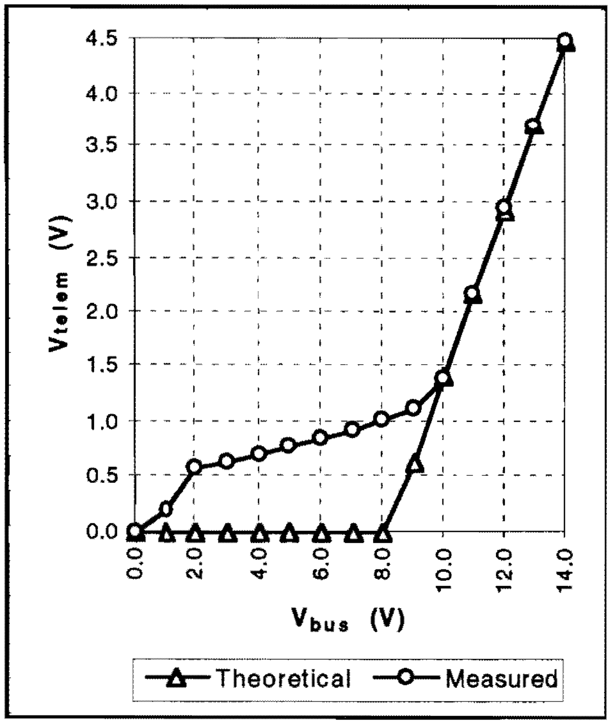

Figure 8 - Theoretical and measured outputs of the voltage telemetry circuit.

range of telemetry values expected is from approximately 1400 to 4000 counts.

These correspond to bus voltages of approximately 10V to 13.5V. Since the correspondence between bus voltage and telemetry reading is a straight line, the equation for calculating bus voltage from telemetry readings is a simple linear equation:

$$V_{bus}(counts) = \frac{V_{\max} - V_{\min}}{count_{\max} - count_{\min}} counts + V_{zener}$$

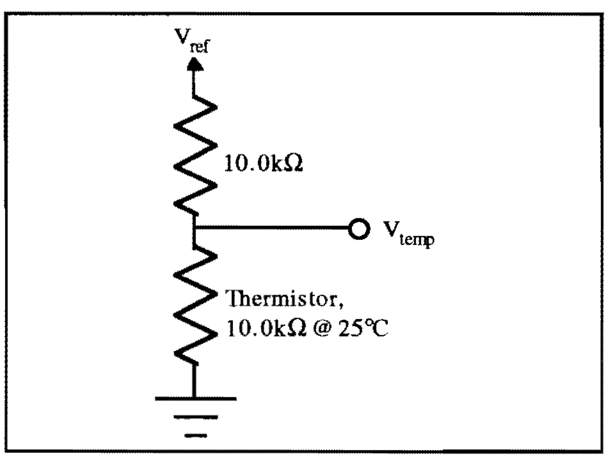

Figure 9 - Thermistor circuit.

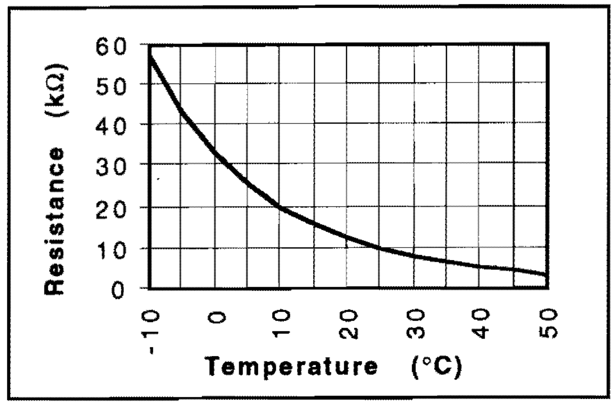

Figure 10 - Thermistor resistance vs. temperature.

where *counts* is the independent variable, V_{max} and V_{min} are the maximum and minimum voltages corresponding to $count_{max}$ and $count_{min}$, and V_{zener} is the voltage of D_z in Figure 7.

Substituting the known values for the variables in the previous equation, it becomes

$$V_{bus}(counts) = 0.00135 counts + 8.192$$

Using the estimates from Figure 8 of 1400 and 4000 counts, the voltages returned by the equation are 10.1V and 13.6V respectively, verifying the equation does return the predicted voltage.

Temperature of the EPS is measured using a thermistor on the EPS PCB. This thermistor is connected in a voltage divider as shown in Figure 9. A reference voltage is applied to the top of the voltage divider when the flight computer takes a temperature reading. The resultant voltage is read via the flight computer A/D converter.

Thermistor resistance changes exponentially with temperature as shown in Figure 10. The voltage divider has a "1/x" shape to its transfer function. An interesting result of choosing a divider resistor equal to the nominal thermistor resistance, 10kΩ in this case, is that the voltage output is nearly linear as plotted in Figure 11.

The function is non-linear enough, however, to require compensation for maximum accuracy. A look-up table is used to convert the voltage reading into temperature, and the non-linearity is compensated for in that table.

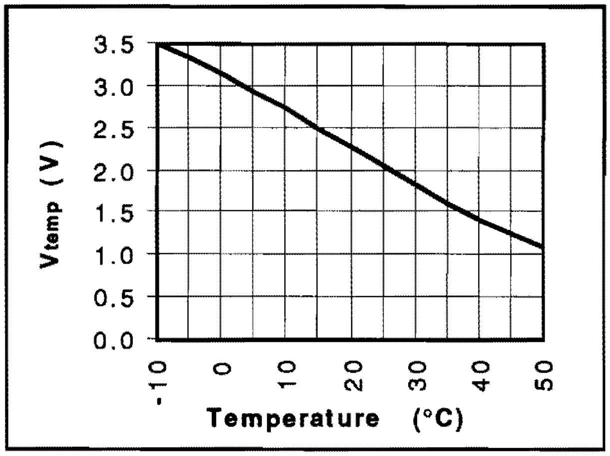

Figure 11 - Temperature telemetry voltage.

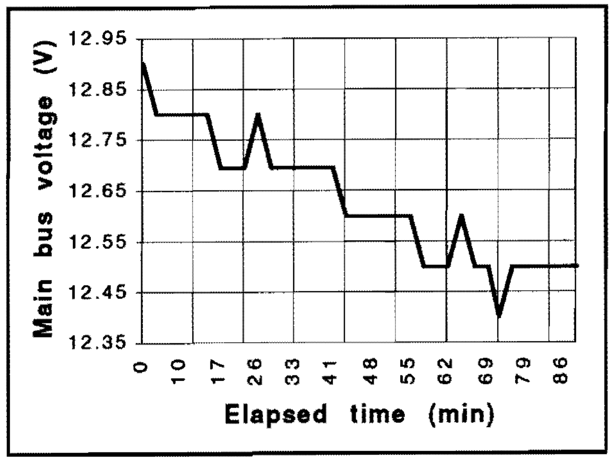

Figure 12 - Main bus voltage vs. time.

Balloon Test Flight

Prototypes of all the components of the EPS were constructed and flown on a high altitude balloon. This tested all the components in a near-flight configuration.

There were differences between the prototype balloon flight version of the EPS PCB and the final flight version. The prototype had connections for only 4 batteries. This was more than sufficient to power the spacecraft for the estimated 4 hour maximum flight duration. The fail-safe timer was set to the latched state, therefore there was no start-up delay once the pressure switches closed. The timer was allowed to time out prior to launch, verifying that it was functional. The EPS PCB itself was milled on a PCB milling machine, thus, there were no plated-through holes. This complicated PCB assembly as component leads had to be soldered on both sides of the board. This was a problem with components such as a 14 pin socket for the timer chip and the latching relay. These components were left standing up away from the board to allow soldering of the component side pads.

Another difference in balloon flight components was that the fuse was bypassed on two of the 4 batteries flown. This decision was driven by the fact that the fuses had blown several times prior to the balloon flight. The reason the fuses were blowing was eventually attributed to mishandling as opposed to a design flaw. The batteries did not have a complete covering, such as heat shrink tubing, which is commonly used to cover a battery. This left the top and bottom connections of each battery exposed, permitting shorting of the battery and blowing of the fuse. No covering was used because of difficulty in obtaining a material that was flight-approved by Lockheed Martin. Therefore, even momentarily setting a battery on a conductive surface, or handling it with a ring on a finger, could easily blow the fuse. Once the batteries were in the mounting plates, there were no more problems with fuses blowing. This was also true of the qualification testing batteries. Several fuses were blown before the batteries were mounted in the EPS box in their mounting plates. Once mounted, there were no more blown fuses.

Despite these challenges, the Falcon Gold prototype spacecraft functioned successfully during the balloon flight. The spacecraft flew to approximately 105,000 feet. This provided at least a small measure of space environment testing of the spacecraft and its subsystems.

The payload activated at the expected altitude and transmitted data until the spacecraft descended below the pressure switch threshold altitude. A plot of main bus voltage vs. elapsed time is shown in Figure 12. The elapsed time is the time the spacecraft was activated. This duration was limited by the pressure switches to the time spent above 45,000 feet. A plot of temperature and altitude data is presented in Figure 13. The trace labeled "Top temp" is the temperature inside the payload box. The "Bottom temp" trace is the EPS temperature profile. The EPS box was better insulated than the top (payload) box. This may explain the smaller temperature excursion of the EPS box. The balloon flight proved that the EPS design was sound. The team acquired valuable information on construction during the assembly of the balloon flight prototype.

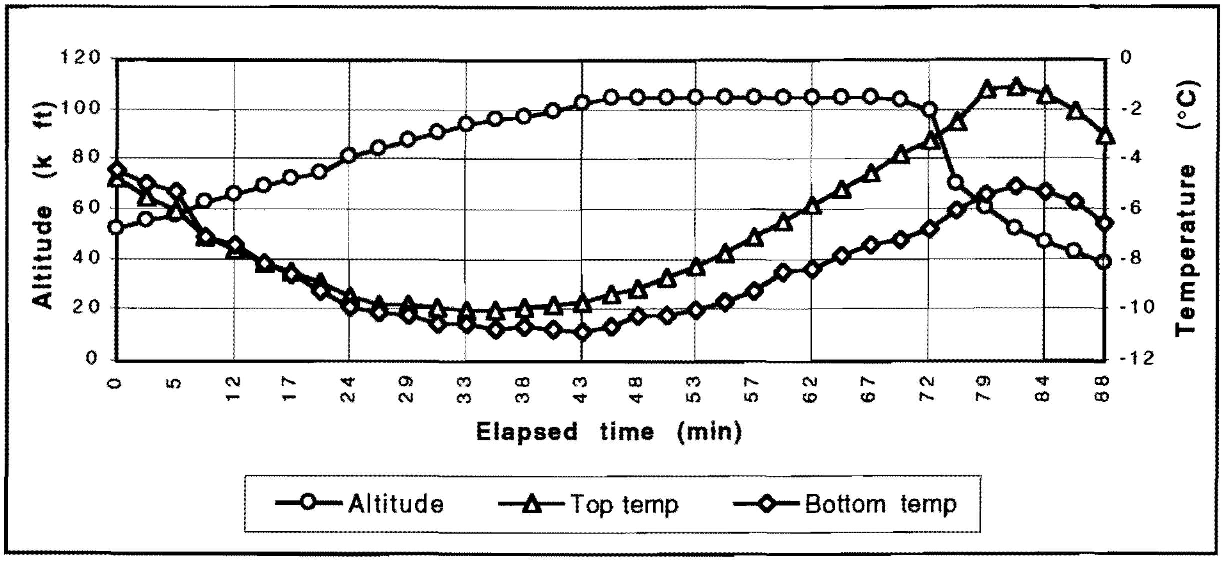

Figure 13 - Altitude and temperature profiles.

Conclusion

A low cost power system using primarily commercial off-the-shelf parts is feasible. One caution, depending on mission requirements, is to balance the low cost of the off-the-shelf components with the cost of additional testing required to prove those components flight worthy. In the case of Falcon Gold, requirements were limited to proving the spacecraft did not pose a threat to the primary payload.

The Falcon Gold hardware had an overall lack of redundancy, leading to many possible single points of failure. The choice here, however, was one of building and flying the spacecraft with relatively low reliability or not flying at all.

Falcon Gold's aggressive schedule did not allow time to evaluate alternatives. Therefore, obvious, readily-available, design solutions were chosen. More extensive design trade-offs are certainly warranted to improve regulator efficiency and reduce quiescent power draw of all circuits. More time to evaluate alternatives would allow an improved battery capacity model for better prediction of EPS performance.

A possible problem using secondary batteries without on-orbit charging is that battery self-discharge could limit mission lifetime. Due to the pre-launch schedule, the spacecraft will not be available for battery charging just prior to lift-off. Depending on the length of time spent on the launch pad and the temperature of the batteries, self-discharge could sap up to several days of mission life. To mitigate this possibility, a mixture of primary and secondary batteries could be used. This guarantees a minimum mission lifetime and still accomplishes flight qualification of the secondary batteries.

Problems not anticipated using a large number of batteries were interconnection and heat build-up during charging. Twenty-eight battery wire bundles consisting of two 22 gauge and three 26 gauge wires had to be bundled, routed, and connected. The large number of batteries also drove the use of a 100-pin GSE connector since access to each battery positive and temperature connection was required for battery charging. The charging process of NiMH batteries is exothermic which limited the number of batteries that could be charged simultaneously.

The final step for this project is a successful launch and return of data from the Falcon Gold spacecraft in its mission orbit.

Acknowledgments

The Falcon Gold spacecraft, including the electrical power subsystem, was funded and built jointly by the United States Air Force Academy and the University of Colorado at Colorado Springs.

Excellent cooperation between University of Colorado and Air Force Academy students, cadets, and faculty was paramount in successfully designing, building, and testing this small satellite in less than a year.

Author's Biography

Mr. Sipple is a candidate for Master's of Engineering in Space Operations with emphasis in Aerospace Vehicle Control at the University of Colorado at Colorado Springs. He received a B.S. degree in Electrical Engineering from the University of New Mexico in May 1985. He is a member of the American Institute of Aeronautics and Astronautics and the Institute of Electrical and Electronics Engineers.

He previously worked on implementation of the attitude determination and control subsystem on the Space Test Experiment Platform satellites, missions 0 and 1.

His interests are spacecraft subsystem and sensor design in general, and electrical power and attitude control subsystems in particular.