Recently, there has been extensive interest in Special Purpose INexpensive SATellites (SPINSAT) that perform cost effective space oriented missions. Reduced reliability and short delivery schedules are two key issues relating to the design of such satellites for minimizing acquisition costs. These issues are important factors in the design of a spacecraft power system which can be considered to be the heart of the entire satellite. The power system constitutes a substantial portion of the overall satellite system cost that must be reduced for SPINSAT missions. Minimum cost or inexpensive satellite implies prudent power system design with minimal redundancy, inexpensive energy source and battery with high cycle life. All of these factors must be considered during the design phase of a small SPINSAT and in particular, of the power system.

Fairchild Space Company was recently awarded a contract to build such a small satellite for the PROFILE (Passive Radio Frequency Interference Location Experiment) program. The power system for the PROFILE program has gone through the considerations mentioned above in terms of cost, reliability, mission life and requirements compliance. Some of the major factors that were addressed during the design phase for PROFILE are the selection of battery type, battery charge control, main bus voltage level, power control and conditioning, dependence on other parts of the satellite, and system redundancy. Various trade-off studies were conducted in order to achieve an optimum design for minimizing the satellite acquisition costs. This paper describes the PROFILE power system design, the rationale for component selection (including system topology and bus voltage), and how the system design satisfies the primary SPINSAT objectives.

INTRODUCTION

In late 1988, the Office of the chief of Naval Research (ONR) awarded a contract to the team of Ardak Corporation and Fairchild Space Company to build the fourth SPINSAT which will be used in an experiment to locate sources of radio frequency interference. This SPINSAT (conceptually illustrated in Figure 1), is called PROFILE for Passive Radio Frequency Interference Location Experiment and will be launched during the first quarter of FY91. PROFILE is being designed for launch either on the Space Transportation System (STS) as a Get Away Special (GAS) or on the Scout
Launch vehicle. However, it will be launched by means of the Scout launch vehicle. It is categorized as a DoD (Handbook 343) Class 'D' spacecraft. The spacecraft will be spin stabilized and placed in a low earth orbit at 300 NM and 72° inclination and requires about 12W of orbital average power. The major system components for PRoFILE are shown in Figure 2 and the load power requirements for the main operating modes are summarized in Table 1. The spacecraft power system is a direct energy transfer (DET) system that consists of a solar array with silicon solar cells, a Nickel-Cadmium battery, and power conditioning circuitry.

**OVERALL POWER SYSTEM DESIGN**

The functional block diagram for the power system is presented in Figure 3. The system topology is based on a sequential full shunt regulator for the solar array voltage control that maintains the bus voltage regulation to ensure proper battery charging. The system includes a 22 cell (2.5 ampere-hour (AH)) Nickel-Cadmium battery and a solar array with 11 strings of 76/1 string of 68 series silicon solar cells to satisfy the load power requirements at 28 ± 6 volts dc.

Bus voltage levels within the range from 7 to 15 volts were initially considered but rejected in favor of 28 volts to 1) Minimize the power conversion losses, 2) take advantage of the existing spacecraft power system designs, 3) maintain industry standard, 4) enhance component selection capability, 5) allow for future growth, and 6) promote adaptability of this design on similar spacecraft in order to minimize the design and development costs (two of the SPINSAT's primary objectives).

Solar array/battery power control and bus voltage regulation by load management utilizing the C&DH processor was considered but the stand alone, independent, power system topology based on a full shunt regulator was selected for design simplicity and to eliminate the software development costs.

In order to minimize the solar array size, the battery size, and the system costs, the power system design is configured to minimize the load power that is required in the standby mode. This is achieved by supplying power to only those components which are required to be ON continuously. For example, power to the communications command and data handling (C&DH) system processor is supplied continuously, while power to the modem, communication security network and the RF section is supplied only when the satellite is communicating with the ground station. Similarly, power to the payload's controller and synthesizer is supplied continuously while the RF section power is applied only when the payload is operational. Further, the power system supplies conditioned power to all other subsystems with two dc-dc converters that provide the required voltage levels. Converter 1 supplies both switched and unswitched power while power from converter 2 is switched on and off in conjunction with the payload. This on/off duty cycle conserves energy and minimizes the converters contribution to the standby power mode.
To ensure proper battery charging for increased battery cycle life, battery voltage-temperature (V-T) control and ampere-hour integration (AHI) control were evaluated for charge control. Also for AHI based charge control, battery charge/discharge (C/D) ampere-hour ratio computation by discrete electronics or by the C&DH processor software was considered. However, for the final power system design only the battery V-T limited charge control was selected for PROFILE. Active current control based on AHI was eliminated because of the higher circuit complexity hence, higher cost. Furthermore, since the battery depth-of-discharge (DOD) can be kept very low (less than 10%) during cycling, battery V-T limited charge control should be more than adequate to satisfy the battery cycle life requirements.

The power system design includes a sufficient number of telemetry monitors for performance verification and to provide information that could be applied to future designs.

SOLAR ARRAY

As shown in Figure 1, the spacecraft body is a twelve faceted cylindrical structure. The entire solar array is mounted on the cylindrical surface and is configured so that each facet includes a string of 76 cells in series except for the facet with the umbilical connectors which has 68 cells in series.

The string of 76 cells on each facet is divided into two panel assemblies. The panel-based rather than direct body mounted solar cell design was selected for ease of handling and replacement. The solar cell panels are attached to the spacecraft cylindrical facets by means of screws and thermal adhesive. Each panel assembly includes a 62 mil thick aluminum substrate, mica ply insulation, and N on P; high efficiency; back surface field, back surface reflector; 1.89x2.6x0.025 cm; silicon solar cells with 6 mil thick coverglass and interconnecting circuitry. Gallium Arsenide solar cells were considered for their higher efficiency and lower radiation loss, but eliminated due to their higher cost when compared to space qualified silicon solar cells.

Each string of 76 solar cells includes a blocking diode and all of the twelve strings are connected in parallel. There are three blocking diode assemblies which correspond to the three wall sections, each with four facets. The series/parallel interconnections are made in the blocking diode assemblies as shown in Figure 4. The diode assemblies are located directly behind the solar cell panels on each of the three side walls inside the spacecraft.

The solar array is capable of supplying up to 21 watts peak at the end of life that is based on a temperature of 35°C for a sun angle of 70 or 110 degrees and includes the effects of sun intensity variation, radiation, UV, thermal cycling and operating voltage. The solar array panels are expected to be within 20 degrees of being normal to the sun on the launch date.
The PROFILE power system includes a 22 cell, Nickel-Cadmium (Ni-Cd), battery that utilizes 2.5AH cells which are supplied by Eagle-Picher. The battery cells are spares that are available from the Space Shuttle's Payload Assist Module (PAM) program.

Key issues in the selection of a battery for any space application include cycle life, cost, previous flight experience, charge control electronics design, handling requirements, and the launch environment. Both sealed lead-acid and Nickel-cadmium batteries were evaluated for the PROFILE application. Key factors in selection were the overall battery cost, cycle life and the launch environment. The sealed lead-acid battery, though least cost intensive, was eliminated due to insufficient cycle life capability and minimal test/flight history. The commercial Ni-Cd battery was found to be unsuitable for the same reasons and because of the low reliability. Considering the battery capacity, size, and the above mentioned requirements, the most suitable choice would be space qualified Ni-Cd batteries. However, these are expensive and far removed from the SPINSAT's low cost objective. Therefore, an attempt was made to locate a sealed, space qualified Ni-Cd type battery with minimum test and documentation requirements which would maintain the SPINSAT low cost objective while also satisfying the extent possible the battery cycle life and reliability requirements. Eagle-Picher's PAM Ni-Cd cells were found to be in this category and were considered to be the prime candidate for the PROFILE power system battery.

The battery package design includes in addition to the power system interface connector, a test connector for cell/battery prelaunch conditioning. The package is designed to maintain the normal battery operating temperature within -5°C to +25°C with the spacecraft temperatures at similar levels. Two temperature sensors are included for the V-T limit charge control and battery temperature measurement by the C&DH system processor. Eight ground selectable V-T levels for proper battery charge control, are available through three parallel binary bit commands. The C&DH processor provides battery under- and over-temperature protection by selecting preset V/T levels when the battery temperature falls below -10°C or rises above 35°C. The C&DH processor also monitors the battery under-voltage (over-discharge) condition and inhibits payload operation.

A battery/main bus disconnect relay is included in the power system for pre-launch battery conditioning and trickle charging through the battery test connector or the umbilical interface.

The battery size, that has been based on the readily available PAM cells, was determined to maintain a low battery DOD (less than 10% for an average orbit) which should extend its cycle life and improve reliability. Another reason for maintaining the low battery DOD is that the selected battery will not be as comprehensively tested and conditioned as a fully space qualified battery would, in order to keep the battery cost down.
POWERSYSTEM ENERG Y BALANCE ANALYSIS

The power system energy balance analysis was performed to ensure that sufficient solar cell array energy is available at the end of life (EOL) to both support the satellite loads and to fully recharge the battery during each orbit. Energy balance analysis results for ten EOL orbits are shown in Figures 5 through 7. Figure 5 shows the spacecraft typical load profile that was used as an input for the energy balance analysis. The solar array EOL power output is shown in Figure 6 and the variation in the battery terminal voltage due to charge/discharge cycling as a function of orbit time is shown in Figure 7. Figure 6 and 7 verify that there is sufficient power available from the solar array to fully recharge the battery. The drop in battery terminal voltage during the first orbit (corresponding to a battery DOD of about 11 percent) is due to spacecraft receive and transmit load which are in addition to standby and search loads that occur during the eclipse. The battery fully recharges within a couple of orbits following this high discharge. This is confirmed as shown in Figure 7 by the battery charge-over-discharge (C/D) ampere-hour ratio which is in excess of 1.12 - typically required to ensure 100 percent battery state-of-charge. In subsequent orbits when the power system is supplying the standby and search mode power the battery DOD never exceeds 7 percent and there is sufficient array power available to fully recharge the battery during each orbit. Since the average orbit battery DOD is limited to less than 10%, it is expected that the battery will more than meet the end of mission cycle life requirement which is approximately 5800 charge/discharge cycles for a one year mission.

SUMMARY

The PROFILE power system is a low cost design. Every attempt has been made to conserve energy in order to reduce the solar array and battery size. As a single string non-redundant system, it is designed for the expected EOL conditions and not for a combination of all possible worst case conditions. Therefore, to improve reliability, the design was kept simple, independent and utilizes designs and components that have previous flight history or have gone through sufficient testing to provide the necessary confidence.

ACKNOWLEDGEMENT

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Figure 1. PRoFILE Spacecraft

Figure 2. PRoFILE Major Subsystem Components
Figure 3. PRoFILE Power System Functional Block Diagram

Figure 4. Solar Cell Panel Interconnections
Table 1. PROFILE Load Power Requirements and Operating Modes Summary

<table>
<thead>
<tr>
<th>Operating Mode *</th>
<th>Load Power Required at 28V</th>
</tr>
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<tbody>
<tr>
<td>I. Standby</td>
<td>8W (continuous)</td>
</tr>
<tr>
<td>II. Search</td>
<td>9W</td>
</tr>
<tr>
<td>III. Receive</td>
<td>16W</td>
</tr>
<tr>
<td>IV. Transmit</td>
<td>70W</td>
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</tbody>
</table>

* Modes II, III and IV are mutually exclusive (see Figure 5)

Figure 5. Spacecraft Load Profile
Figure 6. Minimum End-of-Life Solar Cell Array Output

Figure 7. Battery/Bus Voltage and C/D Ratio