10. Electrical Power

The GOES N-P electrical power subsystem (EPS) provides conditioned power to all the spacecraft subsystems and instrument payloads from three regulated bus sources during sunlight and eclipse periods. The primary bus voltage of 53.1 V ± 0.25 V is produced by regulating the solar array voltage during sunlight and the battery voltage during eclipse periods. This primary bus provides power to the communications, telemetry and command RF, attitude control (reaction wheels), and thermal subsystems. In addition to providing spacecraft subsystem power, the primary bus provides input power for the generation of two secondary buses. One of the secondary buses, which is regulated at 29.4 V to 30.3 V, provides power to the magnetometers, battery cell voltage monitor, telemetry and command subsystem, and the attitude control subsystem. The other secondary bus, which is regulated at 42.0 V ± 0.5 V, supplies the instrument payload power. The EPS consists of a single panel solar array with additional circuits on the solar array yoke, a single nickel-hydrogen battery, an integrated power controller (IPC), three power distribution units (PDUs), a battery cell voltage monitor (BCVM), and multiple relay load switches.

The EPS's primary function is to generate, store, condition, control, and distribute the required mission mode power. Figure 10-1 shows a top-level block diagram of the EPS. During sunlight operation, primary power is generated by the solar array to support the bus load and charge the battery. The solar array drive (SAD) rotates the solar array so that it tracks the sun throughout the orbit. The solar array features a large main panel and a solar array yoke. Both the main panel and the solar array yoke are populated with dual-junction GaAs solar cells. During eclipse or peak power periods, primary power is generated by the battery. The battery consists of 24 123-AHr nickel-hydrogen cells in series. The battery is packaged into three, 8-cell battery packs in series. A battery cell voltage monitor (BCVM) unit is used to monitor the battery cell voltages. The Integrated Power Controller (IPC) conditions, controls, and regulates the +53 V primary bus, the +42 V secondary bus, and the +30 V secondary bus. The IPC distributes the +42 V bus directly to the instrument loads. However, the +53 V bus and +30 V buses are sent to the power distribution units for distribution to their respective loads. There are two bus power distribution units (BPDUs) and one payload power distribution unit (PPDU). The BPDUs and PPDU fuse and distribute +53 V bus power and +30 V bus power. Load switching and signal turn-on and turn-off is provided by the load switch units.

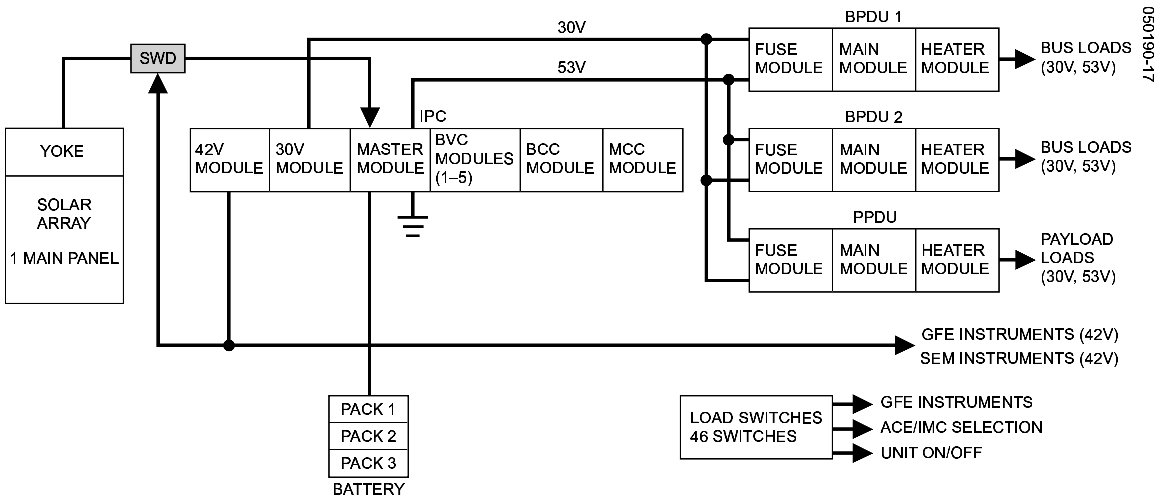

Figure 10-1. Electrical Power Subsystem Architecture

All EPS functions are on-line and operating during all mission phases. Prelaunch regulated power is supplied through an umbilical connection at the battery interface to the IPC. After preflight disconnection, battery power is available on the launch pad to perform any final electrical tests. During transfer orbit the solar array panel is stowed against the spacecraft body, ensuring a benign thermal environment. The solar panel alternates from being illuminated to being shadowed during the spin-stabilized portion of transfer orbit. When the solar array is illuminated by the sun during the spin-stabilized phase of the transfer orbit, the solar array generates sufficient power to support the load and charge the battery.

Solar Array

Primary power is supplied by a single solar array (Figure 10-2) composed of a main solar panel and a yoke panel populated with solar circuits. Both the main solar panel and the yoke panel are populated with dual-junction GaAs solar cells. Each circuit delivers a minimum EOL voltage of 54.8 V. Diode isolation is provided on each solar circuit. The solar array back-wiring retraces the circuit path to reduce out-of-plane magnetic effects and lower the induced magnetic dipole moment. Solar array primary power is delivered through solar array power slip rings to the IPC. Each individual solar array group is connected to a solar array drive (SAD) power slip ring. The solar array is attached to a single-axis, sun-tracking solar array drive (SAD) that rotates about the spacecraft pitch axis. The solar array remains stowed and attached to the spacecraft until orbit raising is completed. Pyrotechnic devices release the array and allow it to be deployed its on-orbit position. The solar array can produce an end-of-life, (EOL) summer solstice power of 1900 watts and an EOL autumnal equinox power of 2084 watts assuming no failures.

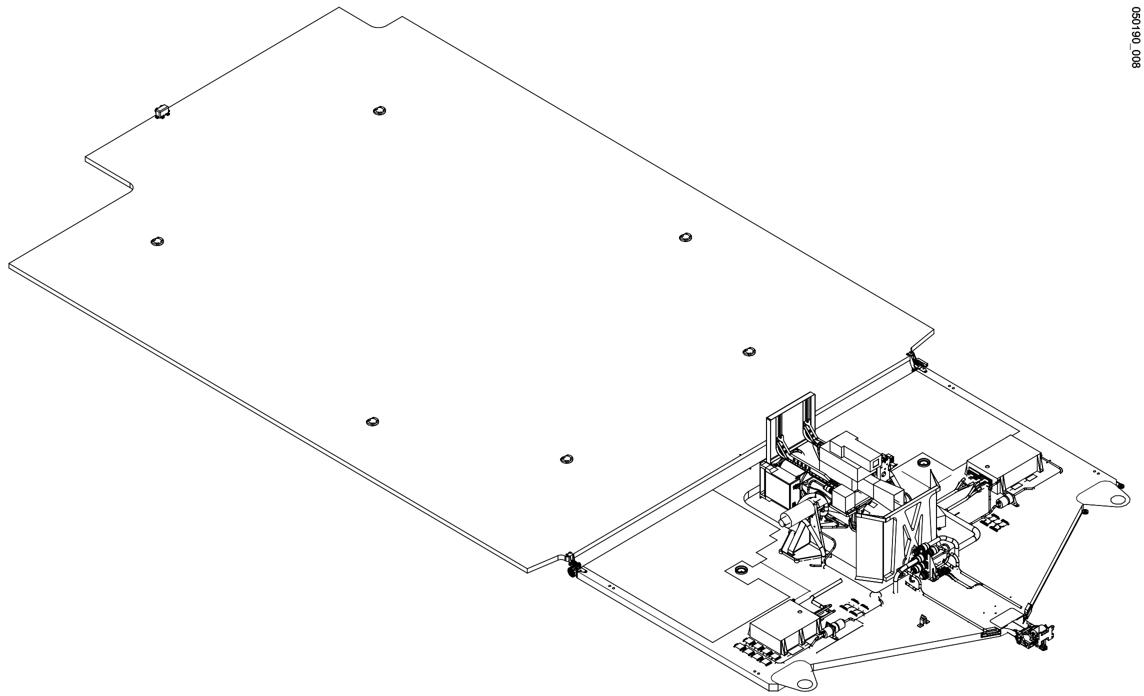

Figure 10-2. Solar Array

Battery

The single Ni-H_2 battery (Figure 10-3) is the secondary power source on the spacecraft. Its primary function is to store energy and provide power to the spacecraft when the solar array produces less than the required power to support the spacecraft load. The battery provides power during launch, orbit raising eclipses, on-orbit eclipses, and periods of peak power demand. The battery consists of 24 123-AHr nickel-hydrogen cells connected in series. The 24 cells of the battery are packaged into three battery packs. Each pack has eight cells connected in series. Harnessing between the battery packs connects the battery packs into a 24-cell 123-AHr battery. Each battery pack has magnetic cancellation harnessing to reduce the battery's magnetic dipole moment. To prevent the loss of the battery due to an open cell, each cell is equipped with bypass diodes in the charge path and bypass relay circuitry in the discharge path. Two strain gauge amplifier circuits mounted to two different cells within each battery pack measure cell pressure to establish the battery state of charge (SOC) for battery charge management. Three thermistors within each battery pack are used for monitoring the battery pack temperatures. Two of the three thermistors are used for on-orbit battery temperature monitoring and the third is used for battery temperature monitoring through the umbilical prior to launch. Thermostats within each battery pack are used to inhibit battery charging if the battery temperature exceeds 35 °C. Each battery pack is equipped with primary and redundant heaters which are controlled by the spacecraft processor to keep the battery packs above –10 °C. The battery heaters are located on each battery pack chassis and

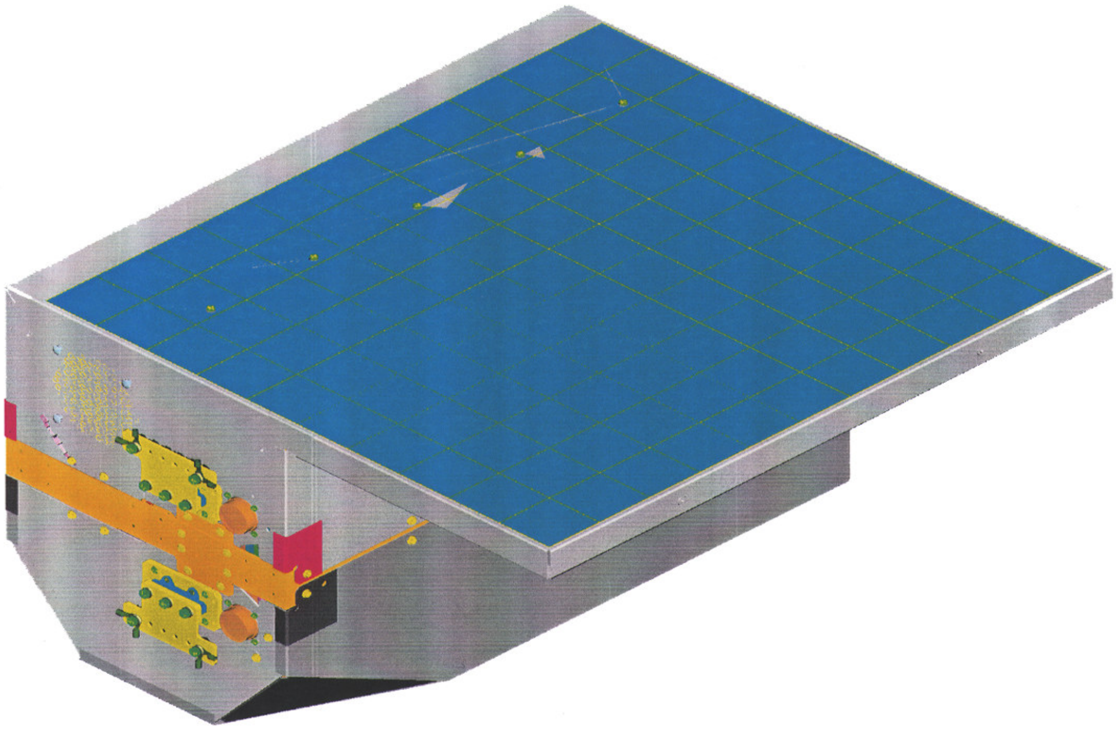

Figure 10-3. Battery

maintain the battery pack at software-selectable temperature settings. The default settings for the battery heaters are –8 °C and –10 °C.

Integrated Power Controller

The IPC is the source of all regulated power for the spacecraft. It conditions, controls, and regulates the +53 V primary bus and the +42 V and +30 V secondary buses. As indicated in Figure 10-4, the IPC consists of several modular units integrated together. The master module is the central control module of the IPC. The master module performs the following redundant functions: primary bus voltage and battery charge control, primary bus overvoltage clamping, serial telemetering and commanding, solar array current sensing, battery charge current and discharge current sensing, primary bus undervoltage detect and latch, housekeeping power generation, and driving of row and column relays. Other master module functions include primary bus voltage sensing, battery charge controller (BCC) input current sensing, and primary bus fault clearing through battery bypass diodes. The IPC's control functions are accomplished with pulse commanding and serial commanding through the IPC's master module. Pulse commanding to the IPC is used for primary and redundant bus and charge control selection, primary and redundant serial command decoder selection, connecting and

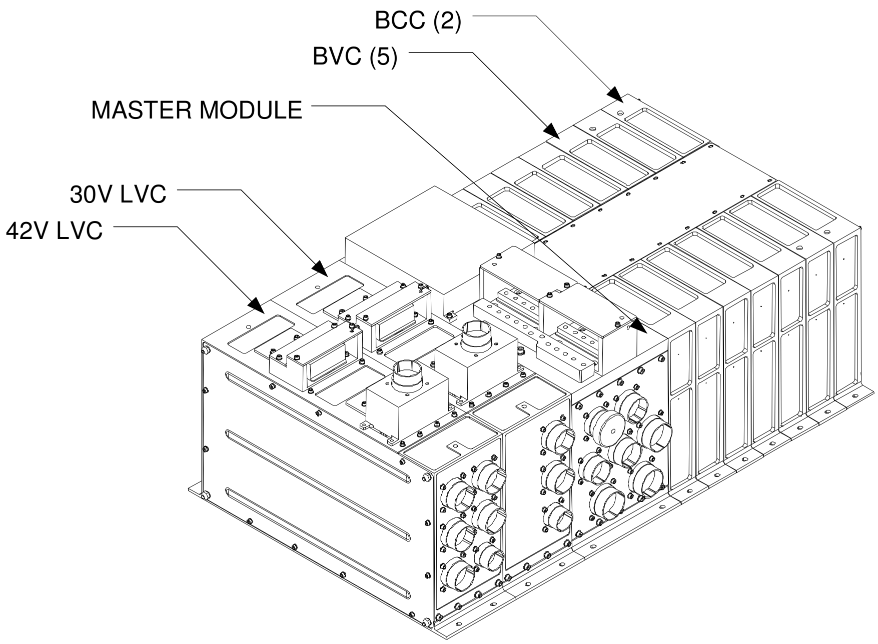

Figure 10-4. Integrated Power Controller

disconnecting the battery to the IPC during test, and enabling and disabling the +30 V and +42 V low voltage controller circuits. The IPC employs serial commanding for enabling and disabling the bus voltage limiting (BVL) and battery discharge controller (BDC) circuits of the IPC, enabling and disabling the BCCs, enabling and disabling battery over temperature shutdown, selecting charge rates, and resetting primary bus under voltage latches. Five bus voltage controllers (BVCs) provide primary bus regulation. Two BCCs charge the battery at the commanded charge rate. The master module accomplishes bus voltage control through an error voltage amplifier, which provides a central control signal. During sunlight operation, the bus voltage limiting (BVL) circuits of the BVC regulate the solar array voltage to produce the +53 V spacecraft bus voltage. Based on the master module's central control signal, the BVL circuits will shunt solar array current, bypass solar array current, or actively control solar array power to match the load demand to produce the +53 V bus. Similarly, the central control signal determines whether the BCCs will charge at the commanded charge rate or linearly reduce the charge rate to meet the spacecraft load demand. During eclipse operation, the central control signal will cause the BDC circuits of the BVCs to boost the battery voltage to produce the +53 V spacecraft bus voltage. The +53 V primary bus feeds low voltage controller modules to produce the secondary bus voltages. A +30 V low voltage controller (LVC) is used to produce the +30 V secondary bus, which is regulated at the IPC at +29.4 V to +30.3 V. A +42 V LVC is used to produce the +42 V secondary bus, which is regulated at the IPC at +41.5 to +42.5. The LVC converters can be enabled and disabled individually. Each converter has overvoltage protection and overcurrent protection to shutdown failing circuits. To meet stringent EMI/EMC requirements, common mode filtering and differential mode filtering are implemented on both the LVC input and output. The LVC input return is isolated from the LVC output return and the LVC chassis. The LVCs provide output voltage and input current telemetry to the spacecraft. Unfused power is delivered from the +30 V LVC and fused and unfused power is delivered from the +42 V LVC. The +42 V LVC delivers its power directly to the instrument loads. Regulated power from the +53 V bus and the +30 V bus is delivered to the power distribution units.

Power Distribution Units

The primary function of the power distribution units is to fuse and distribute +53 V bus and +30 V bus power to their respective loads. The BPDUs fuse and distribute +53 V bus power to the reaction wheels, propulsion transducers, squib driver unit, battery strain gauges, battery bypass reset, and load switch units. The PPDU fuses and distributes +53 V bus power to the telemetry and command RF and communications subsystem components. The BPDUs also fuse and distribute +30 V bus power to the attitude control electronics (ACE 1 & ACE 2), central telemetry and command unit (CTCU 1 & 2), star trackers, magnetometers, and hemispherical inertial reference units (HIRUs). The PPDU fuses and distributes +30 V bus power to the instrument remote telemetry and command unit (IRTCU) and angular displacement sensors (ADSs). Fuses are derated to 50% of their current rating for single fused outputs. For parallel fused outputs, the fuses have been derated to 25% of their current rating. Other key functions of the BPDUs and PPDU are sensing +53 V spacecraft bus current, sensing +53 V heater bus current, and distributing +53 V spacecraft heater power. The BPDUs and PPDU provide fused and switched +53 V heater power to spacecraft bus components. Thirty-six fused and switched heater outputs are provided within each PDU. In each PDU, 30 of the heater outputs are 1-A heater outputs and six are 2-A heater outputs. Primary and redundant serial commanding within each PDU enables individual heater turn-on and turn-off or simultaneous turn-off of all heaters. The PDUs also provide two 1-A relay switched outputs and a battery bypass reset function which are controlled through serial commanding. The BPDU is shown in Figure 10-5.

Battery Cell Voltage Monitor

The BCVM provides cell voltage telemetry from each individual battery cell to the telemetry and command subsystem for incorporation into the flight telemetry. The cell voltages are sampled each telemetry frame. The BCVM is shown in Figure 10-6.

Figure 10-5. Bus Power Distribution Unit

Figure 10-6. Battery Cell Voltage Monitor

Relay Switches

The EPS includes relay switches that enable unit turn-on/turn-off and the reconfiguration of instrument functions. There are 48 such switches whose magnetic latching relays provide connect and disconnect functions. Each relay provides status telemetry to the telemetry and command subsystem for incorporation into the flight telemetry.