

PART 2. SPACECRAFT BUS

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1. INTRODUCTION

The Earth Observing 1 (EO-1) spacecraft has been one of the most trouble-free spacecraft that NASA has launched. A NASA New Millennium Program (NMP) mission dedicated to validating revolutionary technologies that will be used in future government and commercial missions, EO-1 was launched in November 2000 and flies in formation with Landsat 7. As the prime contractor, Swales Aerospace designed and built the spacecraft bus, integrated and tested the EO-1 observatory, and performed launch-site operations.

The EO-1 spacecraft was launched from Vandenberg Air Force Base on November 21, 2000, into a circular, sun-synchronous polar orbit at an altitude of 705 kilometers. It was co-manifested on the Boeing Delta II (7920-10C) launch vehicle with the SAC-C satellite developed by Argentina. The EO-1 orbital inclination (98.2 degrees) and descending nodal crossing time (10:01 a.m.) put it in “formation flight” with Landsat 7 and EOS AM-1 (Terra). With all three satellites following the same ground track, EO-1 flies “behind” Landsat 7 but “ahead of” Terra.

NMP, as a technology-driven mission, accepts a greater level of risk than the level considered acceptable for many other missions. Nevertheless, EO-1’s design sought to minimize risk to the extent possible given cost constraints that dictated a single-string design and which limited redundancy to key areas where risk mitigation was important. These areas of redundancy included mechanisms, e.g., solar array release; the power harness; portions of the command and data handling processor (two separate programmable read-only memories (PROMs) where each contained the full flight code); and an attitude control system safehold mode installed on a separate processor.

Table 1 lists the EO-1 bus salient specifications.

Table 1. EO-1 Specifications

Average Orbit Power	350 W
Spacecraft Bus Dry Mass	410 kg (including WARP and X-band phased array antenna (PAA))
Total Mass	588 kg
Size	1.4 x 1.4 x 2 m high
Payload Attach Fitting	3712
Pointing Stability (Jitter)	0.3 asec/sec
Slew Rate	15 deg/min
ACS	Zero Momentum 3-axis stabilized
GPS	1 receiver
Navigation Accuracy	60 m, each direction (3 sigma)
Science Data Downlink Capacity	105 Mb/s
Science Data Storage Capability	48 Gbits within WARP
C&DH Bus Architecture	Mongoose V, Rad Hard at 12 MHz RISC Architecture
Downlink Formats/Network	CCSDS/STDN, DSN, TDRSS
Downlink Band	S-band (variable to 2 Mbps) X-band (105 Mbps)
Uplink Band	S-band (2 Kbps)
Batteries	Super NiCd/50 Ah
Arrays	3 Panel/Si with GaAs/Articulating/5.25 m
Nominal Voltage	28 V
Structure	Hexagonal; aluminum honeycomb
Propulsion	1 tank/4 thrusters

Propellant Capacity	23 kg
Max delta V	85 m/s
Mission Design Life	1.5 years

The following sections provide details of the EO-1 spacecraft systems design and on-orbit performance. A more detailed description of the EO-1 spacecraft bus can be found in Reference 1.

2. EO-1 SYSTEM DESCRIPTION AND ON-ORBIT PERFORMANCE

Figure 1 shows the EO-1 spacecraft bus configuration and the location of the various technologies that were described in Part 1 - EO-1 Mission of this report.

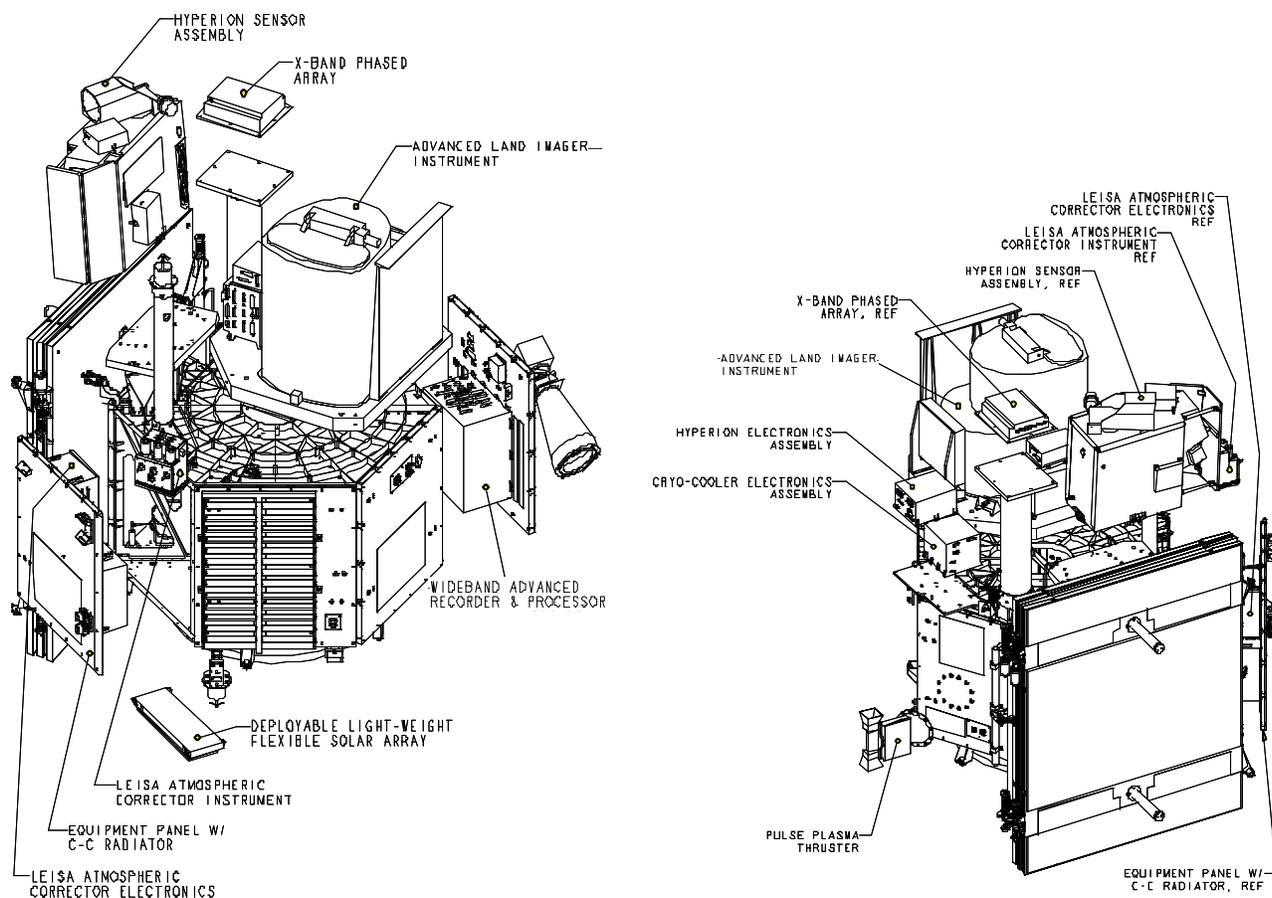


Figure 1. EO-1 Spacecraft

2.1 Structure and Mechanisms

The EO-1 spacecraft is a closed, hexagonal structure consisting of a top, nadir-pointing, deck incorporating the instrument payload interfaces and a bottom zenith-facing deck incorporating a transition adapter interface to the launch vehicle (see Figure 1). The zenith deck is the main interface with the hydrazine

propulsion system. The decks are separated by six radial supports, which transfer payload interface loads to the launch vehicle. The length of these radial supports was governed by the maximum component height with allowance for cable harness bend radii. Both the decks and radial supports are machined from AL (7075-T73). Tubular struts are used to transfer shear loads from the top to bottom deck. The transition adapter is a one-piece machined AL (6061-T651) conical fitting with one flanged end that mounts to the zenith deck and the other end machined to match the Delta 3712C payload attach fitting.

The structure is closed-out along each hexagonal face by an equipment/radiator panel. Each panel is a one-inch-thick honeycomb sandwich panel. Both the facesheets and core of these panels are made from aluminum. The equipment panels are used to mount avionics components and are the heat sinks for components. Bonded aluminum edge inserts were co-cured into the panels and used to secure the panels to the structure. Post-fabrication potted inserts were cured into each panel at the desired component mounting locations. The structure was designed so that these equipment panels could be easily removed during satellite I&T for access to internal system components.

The top-level design requirements were to support the spacecraft systems and the instrument throughout the mission, with the ascent environment driving most of the design.

Three inter-hinged panels with silicon solar cells were deployed to form a single-wing, photovoltaic solar array. The solar panels are attached to the actuator by a 3-piece tubular composite boom. Each tube is manufactured from M40J/934 composite fiber laminate.

The solar array panels are composed of one-inch honeycomb sandwich panels with composite facesheets and aluminum core. The panel cell side is electrical insulated with a 0.002-inch layer of Kapton film. Thermal control is provided for on the non-cell side with a 0.001-inch layer of Tedlar film. Inserts were bonded along the edge of each panel to provide for GSE handling. Inserts for the solar array release system were co-cured into the panel in areas of higher density core.

The EO-1 solar array uses three nearly identical hinges at the panel-to-panel and panel-to-yoke interfaces. A unique main-deployment hinge is used at the yoke-to-actuator interface. Each hinge line uses a viscous damper to dissipate energy and a potentiometer to verify position. A constant force negator spring drove the deployment hinge, and torsion springs were used at the panel hinges. The torque margins on all hinges were at least 5 times the minimum required. The hinge designs are a derivative of the COBE, XTE and TRMM projects at GSFC. The solar array deployment was controlled by a cable and pulley system.

The EO-1 solar array was restrained during launch by a two point semi-kinematic system. One restraint assembly was mounted on the spacecraft nadir deck and the other was mounted to the zenith deck. These assemblies used High Output Paraffin (HOP) actuators. Two HOPs were used in each assembly for redundancy, only one was required to actuate for solar array deployment. Actuation of the HOPs released a spring-loaded restraint rod that traveled through the panels into a containment can located on the outermost panel. Once the rod was released, four deck-mounted “kick-off” springs and the hinge spring torque initiated panel deployment.

2.2 Attitude Control System (ACS)

2.2.1 Introduction

The spacecraft ACS performs slew-and-hold maneuvers to point the body-fixed instrument for sun calibrations on an approximately weekly basis, and performs a complex series of slew-and-scan maneuvers for lunar calibrations every month.

The pointing budget for EO-1 allowed a total ground targeting pointing error of 132 arc second (asec) in roll, 174 asec in pitch and 122 asec in yaw. The budget included an ACS allocation of 54 asec in roll and yaw and 108 asec in pitch for attitude determination, and 30 asec in each axis for attitude control errors. The requirement was to meet these values as a 2σ variance and the goal for imaging was to meet the same values as a 3σ variance. The mass properties growth associated with the Hyperion instrument addition was a challenge to retaining attitude control performance. The launched spacecraft mass was 571 kg and the diagonal elements of the inertia tensor were estimated as [443 179 429] kg-m² with the solar array deployed at the 0 degree position.

2.2.2 ASC Design Summary

The architecture and major components of the ACS are presented in Figure 2. All of the attitude control functions are performed within the Attitude Control and Data System (ACDS). The circuit cards of interest within the ACDS are along the center of Figure 2. The primary attitude control software resides in the Mongoose V main spacecraft computer. The Attitude Control Electronics (ACE) provides electrical interfaces to most ACS components and hosts the Safe Hold Mode controller. Table 2 includes a description of each of the ACS components.

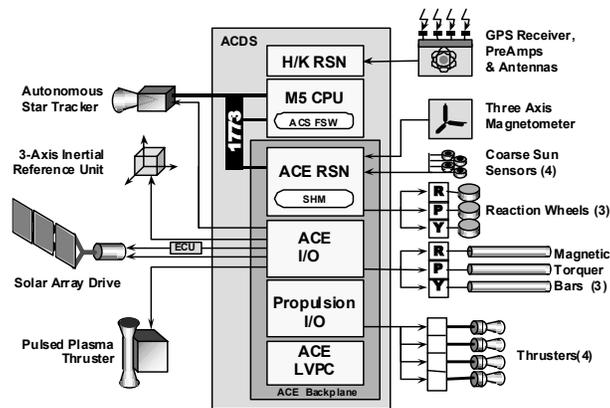


Figure 2. EO-1 ACS Components

Table 2. ACS Component Capabilities

ACS Component	Vendor and Model	Description
Reaction Wheels Assembly	Ithaco Space Systems Type A w/MDE	± 4 N-m-s momentum at 5100 rpm, ± 0.025 Nm maximum torque; three wheels utilized with on-axis orthogonal mounting
Magnetic Torquer Bars	Ithaco Space Systems TR60CFR	± 60 Am linear dipole moment with on-axis orthogonal mounting and linear coil drive capability
Three-Axis Magnetometer	SAIC/Ideas and Nanotesla NT600s	± 100.0 μ Tesla range on each of three axes; resolution of 0.05 μ Tesla
Inertial Reference Unit (IRU)	Litton Guidance & Control SS-SIRU (with 3 HRG)	± 10 degree/sec max rate; bias stability $\leq 0.015^\circ$ /hr over 8 hours; ARW $\leq 0.001^\circ$ /hr ^{1/2} (3σ); resolution of 0.05 asec
Autonomous Star Tracker (AST)	Lockheed Martin / ATC AST-201 with sun shade	8×8 deg field of view (FOV), 1 Hz ECI attitude quaternion output, with 5, 5, 25 asec, 1-sigma accuracy; up to 50 stars in solution
Global Positioning System (GPS) Receiver	Space Systems Loral GPS Tensor with 4 antennas	1 Hz time, position to ± 150 meters, 1-sigma and velocity to ± 0.55 meter/second, 1-sigma

ACS Component	Vendor and Model	Description
Coarse Sun Sensors	Adcole Corporation Model 29450	Peak output current of 650 micro-amps; four eyes provide full 4π steradian coverage

The fastest solar array motor stepping rate is 25 Hz (0.19°/sec), and the normal daylight rate of 8 Hz can be adjusted by $\pm 1\%$. solar array rotation is positive/forward for “orbit day” and negative/reverse for rewinding during “eclipse.”

2.2.3 ACS Flight Software and Control Modes

The EO-1 ACS flight software architecture was derived from the software used on GSFC’s Tropical Rainfall Measuring Mission (TRMM). The EO-1 ACS software included attitude determination and closed-loop control modes for magnetic de-spin following separation from the Delta II launch vehicle, initial stabilization and sun acquisition, nadir-pointed science data collection and downlink, thruster maneuvers for delta-V, and solar/lunar slew/scan maneuvers for instrument calibrations.

The ACS flight software mode transitions are illustrated in Figure 3. Following separation from the Delta II launch vehicle, the ACS nulled the tip-off rates via a B-dot magnetic control law and stabilized the spacecraft. During initial Sun Acquisition, the spacecraft maintained an inertially fixed, solar pointing attitude with the instruments facing away from the sun. During normal operations in Mission Idle, the body fixed science instruments point toward the earth as the spacecraft maintains a fixed attitude with respect to the orbit frame. Solar calibration requires a slew maneuver to point the instruments toward the sun, followed by an Inertial Hold. A series of transitions between slew maneuvers and holds are used to perform the lunar calibration raster scan that sweeps the moon across each of the instrument detectors. A transition to delta-V is preceded by a slew maneuver to orient the spacecraft for the thruster burn. Transition back to Earth pointing from any attitude is achieved using an Earth Acquisition slew. The ACS Fault Detection and Correction (FDC) system includes actions that can force transitions to either ACS Sun Acquisition or ACE Safe Hold.

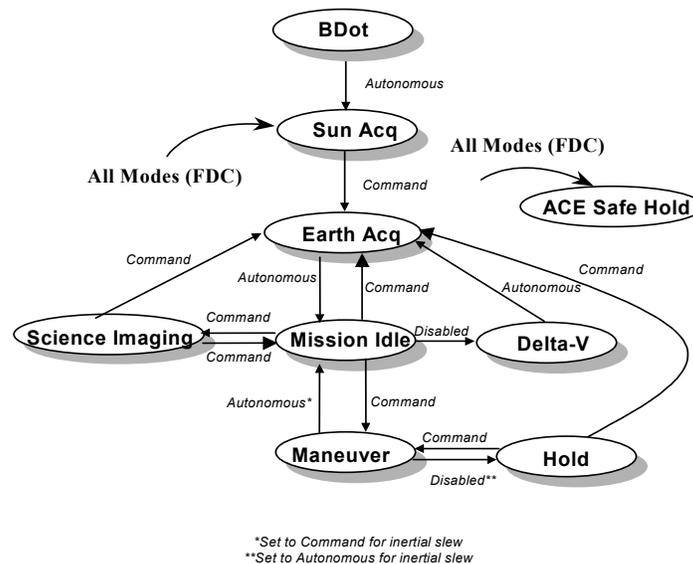


Figure 3. ACS Control Mode Diagram

2.2.4 Safe Hold Mode

The independent Safe Hold algorithm software resides in the ACE remote services node (RSN) processor. If a fault condition is detected, this fully autonomous control capability will drive the solar array to the

0-degree reference position (orbit noon position) and put the EO-1 spacecraft in a thermal and power safe attitude that is inertially fixed relative to the sun. Coarse sun sensors on the solar array and spacecraft main body will provide spacecraft attitude with respect to the sun with the IRU providing rate feedback. The control about the sunline will only be rate damped during orbit day, but all axes will be inertially fixed during eclipse. Continuous magnetic unloading of reaction wheel momentum will be performed using a cross-product control law.

2.2.5 Earth Pointing, Mission Idle

ACS pointing performance has been well within mission requirements, as shown in Table 3.

Table 3. Science Pointing Accuracy Assessment

	Requirement	On-Orbit Performance
Controller Error	[30, 30, 30] asec, 3σ	X and Z consistently under 30 asec Y varies between 0 to 50 asec depending on settling time after image prep slew maneuver
Attitude Knowledge	[54, 108, 54] asec 3σ	All axes consistently under 36 asec, 3σ during normal nadir-pointed operations
Navigation Accuracy	[130m Cross-Track, 100m Along Track] 3σ	Cross-track 45m, Along track 55m, Radial 30m, 3σ
Jitter/Rate Stability		Better than 0.5 asec/sec, 3σ during imaging

Figure 4 demonstrates that the typical attitude control accuracy achieved during an image observation meets the 30 asec, 3σ goal. The position error plot is over a 6-hour period, and the bottom plots zoom in on 30-second image duration.

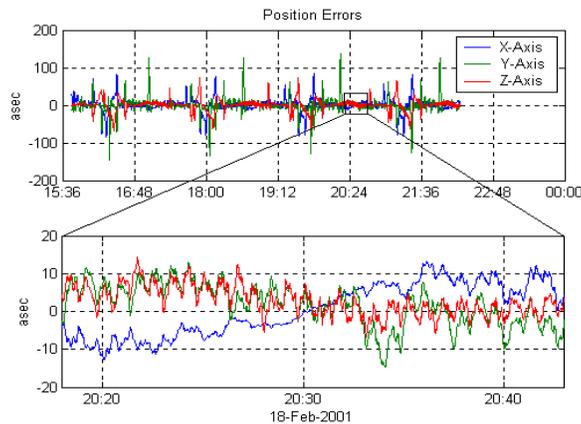


Figure 4. Position Control Errors During Image

Figure 5 shows the 300-asec peak transient in the pitch axis due to opening the Advanced Land Imager (ALI) cover prior to acquiring an image. The opening of the cover is timed such that this transient is zeroed within 2 minutes prior to the start of an image. The thermal snap transients as a result of night-to-day and day-to-night transitions have a 40-to-80-asec peak error per axis with a 2-minute width.

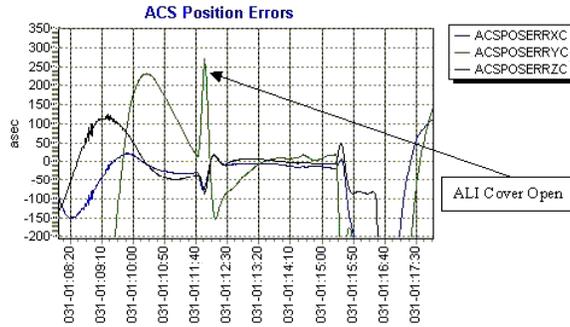


Figure 5. Position Errors During ALI Cover Open

2.2.6 Lockheed Martin AST-201 Star Tracker

Overall, the EO-1 AST performance has been better than expected. All initial AST attitude acquisitions have been successful. The mean AST RMS error has been approximately 30 μ rad, which correlates closely with ground processing estimates. The AST root mean square (RMS) error has varied between 10 and 100 asec on average with spikes as high as 500 asec. The AST has been tracking between 10 to 45 stars, with a mean of 25 stars. The AST Effective Focal Length (EFL) error telemetry indicates a mean of less than 10 μ m of error with no need for updates to the EFL. The AST has shown susceptibility in the South Atlantic Anomaly, with a range of 0 to 4 events per day – either coast frames, reacquisitions, or reboots – all with autonomous recovery to track mode.

2.2.7 Litton G&C Space Inertial Reference Unit

The IRU has performed nominally since launch. There have been no lost packets, packet checksum errors, or invalid packets. The IRU calibration identified negligible scale factor error and the IRU-to-AST on-orbit alignment calibration resulted in less than 0.2-degree (720 asec) change. The gyro drift biases were estimated at (1.02 0.08 1.58) deg/hr and have remained constant.

2.2.8 Loral Space Systems GPS Tensor

The GPS receiver successfully performed two cold start acquisitions in the sun pointing orientation. The receiver remained in track during all of the IRU calibration slews, delta-V slews, and imaging slews. It had been anticipated that the receiver may have difficulty during these off-nadir operations, but the orbit determination Kalman Filter within the GPS Tensor has demonstrated robustness through these events. The GPS position and velocity navigation vector performance has been well within the 100-m specification.

2.2.9 ACS Conclusion

The EO-1 ACS design and implementation had numerous challenges as part of a “faster, better, and cheaper” mission. The on-orbit pointing accuracy, with attitude controller errors and knowledge errors combined via a Root-Sum-Square, was approximately 40 asec, 3-sigma during science observations. This performance enabled the instrument team to perform alignment calibrations using nadir scenes. The lunar raster scan maneuver exceeded expectations and provided a significant radiometric calibration source for instrument detectors. The performance and predictability of the delta-V mode has conserved propellant usage and set the stage for extended mission operations with the Enhanced Formation Flying (EFF) experiment.

2.3 Power System

2.3.1 Description

The EO-1 Power System is a 28-V, unregulated, Direct Energy Transfer (DET) power system with an articulated solar array and one 50-Ah Super nickel-cadmium (SNiCd) battery connected directly to the power bus. (Used in conjunction with NiCd technology, ‘Super’ is a Trademark TM of the Hughes Aerospace Corporation.) It regulated the spacecraft energy balance, providing the required power of 330W (nominal orbit average) while wasting very little energy inside the spacecraft to be dissipated as heat. It accomplished this by pulse width modulating and short-circuiting unwanted solar cell segments, leaving their energy out on the solar array. It distributed 5- and 15-volt regulated power internal to the Power Supply Electronics (PSE) and 28-volt unregulated power to the systems and instruments and provided fault protection.

2.3.2 Power Supply Electronics (PSE)

The PSE is contained in one electronics box consisting of the following six modules, connected to a backplane: 1 Solar Array Module (SAM), 1 Battery Module, 2 Output Modules, 1 RSN Control Module, 1 Low Voltage Power Converter (LVPC).

The PSE interfaces with the solar array, the battery and the loads. Its primary function is to provide power control (including battery charge control). But it also contains the logic and circuitry for fault protection and a telemetry and command system interface. In addition, it contains the solid state, commandable relays for power distribution of the unregulated bus power and serves as the location for the Single Point Ground.

2.3.3 Solar Array

The EO-1 solar array is a single wing, comprised of three panels. The array is canted at 30 degrees and utilizes a single-axis drive assembly to clock the array at an orbital rate, essentially keeping the array normal to the sun, minimizing cosine losses. A cable wrap mechanism is implemented between the rotating array and the spacecraft, which requires the array to “rewind” during eclipse periods, preparing for the next sunlit period in which it moves forward at orbital rate.

Table 4 provides details on the EO-1 solar panel design.

Table 4. Solar Array Summary

Item	Type	Size and/or Comment
Solar Panel Substrate	Aluminum Honeycomb Composite	49.5 X 56.75 in.
Face Sheet	Epoxy-graphite Composite	15-mil thick
Face Sheet Insulator	Kapton	2-mil thick
Cell Bonding Adhesive	CV 2568 Silicone	NA
Thermal Control Coating	Tedlar	1-mil thick
Si Cell(s)	15% BSFR Si	24.611 cm ² 3.896 X 6.317 cm
	By-pass diode protection for 31-cell sub-strings on outboard panel (diodes located on rear of panel).	
DJ GaAs (Cascade) Cell(s)	22% Efficient Dual Junction	24.312 cm ² 3.846 cm X 6.322 cm

Item	Type	Size and/or Comment
	Inherent solar cell by-pass diode protection (located on the back (underside) of each cell).	
Cell Interconnect Material	Invar	Redundant
Cell Connection Method	Solder	NA
Temperature monitoring	2 platinum resistance temperature (PRT) sensors, mounted directly opposite one another on the front and rear surface of the outboard panel.	

The combination of two different cell technologies, as shown in Table 4, resulted from the addition of another payload (Hyperion) after the start of spacecraft I&T. The original solar array design consisted entirely of 15% back surface field and reflector (BSFR) Si 10 ohm-cm cells. The additional power requirements of the Hyperion instrument resulted in the addition of 22% Dual Junction Gallium Arsenide (DJ GaAs) (Cascade) cells on the remaining available solar panel area, resulting in an additional 1.4 amps available at the predicted operating point. These cells were wired in as a “fixed” segment, with no shunt dedicated for their control. The total solar array current is 22.1 amps, BOL.

2.3.4 Battery

EO-1 used a single 22-cell, 50-Ah, 28-V SNiCd battery. The voltage range was originally specified as 22 to 36 volts (21 volts minimum at the load), consistent with the conventional NiCd technology with one battery cell failure. The actual voltage range of this battery is 24 to 34 volts (23 to 32.5 if one cell were to short, later in life).

2.3.5 Power System Redundancy and Risk

EO-1 solar array cell interconnects are triple redundant, and all wiring on the solar array is dual redundant. This dual redundant wiring is carried from the solar array back to the solar array inputs at the PSE. Within the PSE, each segment has diode protection, eliminating the possibility of a short on a single string or segment from causing mission failure. Overall, the solar array implementation allows for graceful degradation and resistance to individual failures. The loss of a segment to either short or open circuit has the net effect of reducing the total power available from the solar array, allowing the spacecraft to continue operations in a potentially degraded mode.

The PSE is single string, but has features to help mitigate risk with respect to internal failures. An over-voltage protection (OVP) circuit provides protection against bus voltages over 34.5 VDC by sequentially shunting array segments to maintain control. This is implemented in hardware, and has priority over the normal PSE software control loop. Also, the PSE has internal fault detection and correction that increase the likelihood of surviving system failures or upsets.

2.3.6 Power System Conclusion

The power system of the EO-1 spacecraft successfully implements single string design for a low-Earth-orbit application. The initial prediction for solar array current was 21.8 amps, but initial telemetry revealed a maximum array current of 24 amps. Evaluation of system performance at mission day 200 showed the power system to be still performing within mission requirements and the maximum solar array current was just over 22 amps. After adjusting the model for initial conditions, maximum orbit average power (OAP) predictions can be seen in Figure 6. Radiation and ultraviolet (UV) degradation of the array have not been as severe as predicted. Figure 6 reveals that meeting the spacecraft’s OAP need of 330W is achievable with significant margin through mission day 550 and beyond. Therefore, the entire system continues to exceed predictions on overall performance (orbit average power support), and is capable of operations at full capacity well beyond the initial design lifetime.

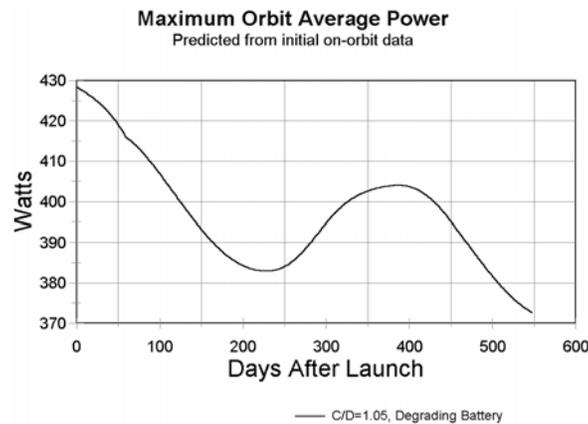


Figure 6. Maximum Orbit Average Power

2.4 RF Communication

EO-1 has both S-band and X-band communication systems. The S-band system provides command uplink, housekeeping data downlink, and backup science data downlink, while the X-band system is the primary downlink for the science data. The S-band system uses two omni antennas, one nadir and one zenith pointing, to provide near-spherical coverage. These antennas are driven by a Lockheed Martin CSX-600B transponder, which is controlled by the Communication RSN. The X-band system uses the NMP phased-array antenna, which is steered electronically for each transmission pass over a ground station. All data formats for uplink and downlink streams conform to the CCSDS recommendations. A block diagram of the communications system is provided below in Figure 7.

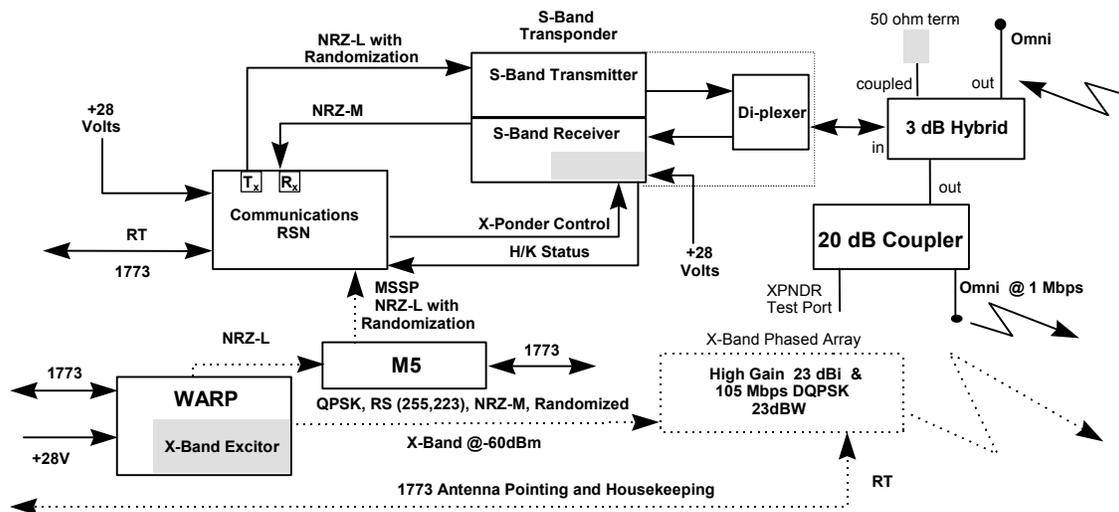


Figure 7. Spacecraft RF Communications Systems

The primary ground station for uplink and downlink was the Spitzbergen Ground Station located at Svalbard, Norway. During Launch and Early Orbit, TDRSS was used to provide real-time housekeeping telemetry. The spacecraft required at least two ground station passes per day for a “stay-alive” mode, but typically used 6 to 8 passes per day for operations.

2.4.1 Command Processing

Commands originating at the EO-1 MOC are encoded into the NRZ-M format at the ground station at the rate of 2 kbps. The formatted command is used to phase shift key (PSK) modulate a 16-kHz subcarrier. A 2-kHz clock is also provided by dividing the 16-kHz clock in the ground processor by 8. The modulated subcarrier will phase modulate (PM) the uplink transmitter at the ground network (GN) ground stations, which operate at a frequency of 2039.645833 MHz.

2.4.2 S-Band Telemetry Processing

Two independent paths transfer data from the C&DH to the Communication RSN: a Medium Speed Serial Port (MSSP, RS422) link, and the 1773 bus. Both paths feed a single physical downlink path to the transponder. The Communication RSN provides hardware based encoding of the data and determines the downlink bit rate. The MSSP link feeds high rate data from the WARP directly through hardware to the transponder. The transponder is responsible for the radio frequency transmission of the digital data stream received from the Communication RSN. Fill data is automatically generated when no data is available.

The EO-1 spacecraft S-band transponder always operates in STDN receive mode during normal on-orbit operation. The uplink signal is received at the spacecraft (via either omni antenna) by the S-band transponder, which detects and phase locks to the signal swept at rates between 5 and 35 kHz/sec. The recovered command data and clock signal, along with a lock indicator, are sent to the Communication RSN for command processing and execution.

During S-band communications, the GN generates two-way Doppler (i.e., range rate) tracking data that is derived from the spacecraft to GN S-band RF carrier, which is a coherent turnaround of the GN RF carrier. In the coherent mode, the spacecraft's transmit carrier frequency is related to the received carrier frequency by the ratio 240/221. The GN will also generate one-way Doppler tracking data that is derived from the spacecraft to GN S-band RF carrier. The GN provides angles data for all passes. TDRSS will generate one-way Doppler tracking data using the return link S-band RF carrier from the EO-1 spacecraft.

Tone ranging is supported by this transponder, but this feature was not used.

2.4.3 RF On-Orbit Performance

Overall, the EO-1 communications margins were larger than expected, which is typical since link margin calculations tend to be conservative. The S-band system operated perfectly beginning with the initial TDRSS acquisition 6.5 minutes after the EO-1 launch. The X-band flight system has performed perfectly but the ground station system initially had difficulty dealing with tracking the beam. This corrected issue is discussed in the contained X-band validation report. All of the occasional problems encountered with the ground communication have been associated with ground systems and not the flight hardware.

2.5 Propulsion

2.5.1 Description

The EO-1 spacecraft requires a reaction control system (RCS) for orbit adjust and precision orbit maintenance. The EO-1 RCS is a monopropellant hydrazine system operating in blow-down mode. The entire RCS is located on the lower deck of the spacecraft opposite to the payload attach fitting. The system's four MR-103G thrusters are mounted such that they fire through holes in the lower deck, and are canted at a 15-degree angle to provide coupling for spacecraft yaw control. The thrusters incorporate dual coil dual seat valves in series to mitigate the risk of valve leakage. A latch valve can be used to isolate the thrusters from the propellant tank. The RCS propellant tank is a Pressure Systems Inc. spherical titanium tank identical to the TRW STEP IV propellant tank (P/N 80225-1). This tank has a propellant capacity of 1375in³ and a Mean Effective Operating Pressure (MEOP) of 320 psig. An AF-E-332 elastomeric

diaphragm inside the tank provides positive propellant expulsion. Nitrogen was used to pressurize the system prior to launch. The RCS is instrumented with a pressure sensor, latch valve position indicator and several temperature sensors.

A schematic of the EO-1 RCS is shown in Figure 8. The physical arrangement of propulsion system components is shown in Figure 9. Following propellant loading and pressurization, the latch valve was closed for transportation to the launch pad. The latch valve was opened prior to launch.

Temperature control for the tank, lines, and isolation/latch valve and thruster valves is done by thermostat switches that autonomously connect +28 VDC power to resistive heater loads. The thermostat switches close when cooled to some setpoint temperature and then open when warmed to some higher temperature. There are two redundant separately controlled heater system circuits. Each is powered by a separate +28V switched service from the J2 connector on the PSE LVPC. Each service is capable of providing up to about 28 W on a continuous basis.

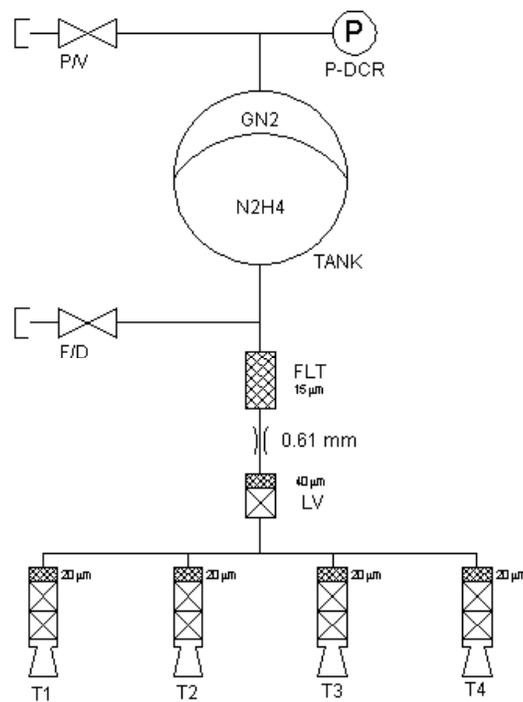


Figure 8. EO-1 Reaction Control System Schematic

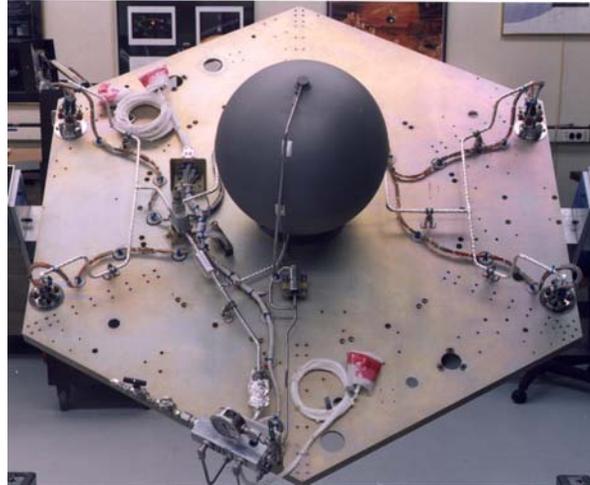


Figure 9. EO-1 Reaction Control System

2.5.2 Propulsion On-Orbit Performance

The orbital maneuver campaign to position the EO-1 satellite in orbit with respect to the Landsat 7 satellite was initiated on Day 4 of the mission. A 60-second calibration burn was performed to evaluate thruster operation and confirm the expected attitude control trajectories with respect to pre-launch simulations. Since the orbital separation between EO-1 and Landsat 7 continued to drift on a daily basis, the maneuver campaign required performing the first four burns relatively close to one another. The delivered delta-V accuracy for the calibration burn was within 5% of the prediction and all subsequent burns have been performed within 0 to 2% error. The fuel mass usage over time is shown in Figure 10.

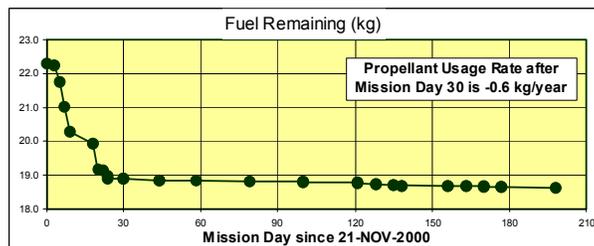


Figure 10. Fuel Mass Usage

2.5.3 End-of-Mission Orbit Lowering

Following completion of all mission objectives, all propellant will be expended. Initial study indicates that the de-orbit operation will require over 40 burns of 16-minute duration due to the blowdown curve. Circular orbital altitude will be lowered from 705 km to 590 km, with uncontrolled reentry to occur in approximately 10 years.

2.6 Command and Data Handling (C&DH)

2.6.1 Description

The C&DH and attitude control system (ACS) software reside in the Attitude Control and Data System (ACDS), which contains a 12-MHz Mongoose V processor with 1.8 Gbits of storage. Most systems

contain a Remote Services Node (RSN), which is a R000 processor for both 1773 bus interface and system control.

Since the launch of EO-1, there have been no hardware or software anomalies associated with the C&DH system. No single event upsets have occurred, and no processor resets have been required. The only events of note are expected periodic bit errors in the C&DH random access memory, which are detected and corrected via the on-board error detection and correction (EDAC) memory. The flight operations team (FOT) regularly performs table uploads into the processor memory, utilized for stored command sequences and changes to operational parameters.

The following information highlights the components of the C&DH software with respect to their on-orbit performance:

2.6.2 Health and Safety Monitoring

- All 15 real-time tasks (including ACS and EFF), in the M5 processor have performed as designed.
- No event messages or warm restarts caused by either critical or non-critical tasks failing to “report in” to health and safety on a timely basis.
- No restarts have occurred in the main onboard computer or any of the other processors on the EO-1 spacecraft since launch.

2.6.3 Telemetry Output

- All housekeeping data is being sent to the ground in real-time when in contact with the spacecraft as designed.
- Telemetry is consistently being routed for storage in the solid-state recorder for ground playback.
- All of the ground passes at the various telemetry downlink rates has been performed nominally.
- To date there has not been any unexplained anomalous behavior in the C&DH telemetry output string.

2.6.4 Time Code Management

- Maintains and distribute time with sufficient accuracy to support all aspects of the mission, including attitude determination and control and all technologies.
- Provides time keeping that has enabled precise execution of the onboard Stored command sequences.
- Provides time management for autonomously configuring the spacecraft for ground contacts.

2.6.5 Command Processing

- Provides consistent and reliable commanding to support validation of all technologies and systems.
- Absolute and relative time stored command software performance has provided autonomous spacecraft operations.
- The stored command sequences loaded on a daily basis as an integral part of mission planning for autonomous operations.

2.6.6 Memory Management

- ACS has used table load capability for optimizing the attitude control parameters of the EO-1 spacecraft.

- Table loads have been used to load a new medium rate data storage filter table and to load new TSMs for failure detection and correction.
- Partial Table loads have been used to modify existing TSMs.
- The table load features are used only a daily basis to load the absolute and relative time stored command sequences for mission planning.
- To date there has been no problems identified with the memory management s/w during the on orbit checkout.
- EFF has used memory load capability to load new software algorithms.

2.6.7 Data Storage

- All engineering data, GPS data, and, spacecraft's events are being stored in the solid-state recorder as designed.
- The FOT is consistently able to play back and dump the stored data to the ground during the scheduled ground passes.
- The command and control of the data storage features has worked as designed without any unexplained anomalous behavior.

2.6.8 Anomaly Management

- All single-bit errors have been corrected when detected in the dynamic random access memory (DRAM) and no multi-bit errors have been detected.
- No Memory Checksum failures have been detected in any of the processors.
- All events that are being generated are being stored in the recorder memory and are being sent to the ground as real time telemetry when the spacecraft is in ground contact
- Telemetry and statistics monitor (TSM) task has responded to occurrence of spacecraft anomalous conditions as designed.

2.6.9 Enhanced Formation Flying

- Attitude determination and control in formation with Landsat 7 has been successfully demonstrated.
- EFF flight software has been successfully activated and capabilities demonstrated.
- M5 central processing unit (CPU) utilization peaks at over 100% CPU (as expected) utilization while EFF processes its data with no anomalous behavior onboard the spacecraft.

2.6.10 1773 Bus Communications Control

- 1773 Bus Control software is consistently synchronizing bus transactions for all of the unique processors in the system acting as remote terminals (RT)
- Routing telemetry and commands to and from all of the RTs has been performed flawlessly with no commands or telemetry being dropped.
- There has been no unexplained bus errors or retries in any of the RTs that are part of the EO-1 spacecraft.

2.6.11 Housekeeping (HK) RSN

- Successfully detected Launch Vehicle Separation and initiated the solar arrays deployment sequence precisely as designed.
- Provides control and monitoring of the Light Weight Flexible Solar Array (LFSA) as required for the technology validation.
- Collection of spacecraft housekeeping thermistor data that includes Structural temperatures, Carbon-Carbon Radiator temperature, and other thermistors throughout the spacecraft.

- Providing consistent management and control of the GPS receiver to acquire GPS State Vectors, GPS time acquisition and distribution, and data collection on request for technology validation.
- One significant issue in the HK GPS data collection software, that caused the HK RSN to erroneously mark valid packets as being invalid and persisted long enough for the ACS to quit using GPS for attitude control, was identified. An operational work-around was implemented in the ACS software that negates the problem.

2.6.12 Communication RSN

- Telemetry downlink of science and housekeeping data from the spacecraft has been successfully performed at all programmable rates.
- The ability to downlink WARP data through the S-band antenna has been successfully demonstrated as the downlink source for the first EO-1 images to the ground.
- No unexplained anomalous behavior has been identified in the performance of the software.

The EO-1 C&DH system has performed almost flawlessly since launch. The only projected maintenance was standard flight operations tasks, and adjustment to TSMs as other system performance changed with spacecraft life.

2.7 Thermal Design

2.7.1 Description

The EO-1 system is designed with a cold-biased passive thermal control system (TCS). The thermal hardware includes: thermostatically controlled heaters, multi-layer insulating (MLI) blankets, thermistors, thermal louvers, and optical coatings. The majority of the spacecraft exterior is covered with MLI to minimize heat loss to space and to reduce the effects of incident solar and earth flux. Thermal louvers are used on the battery equipment panel to reduce heater power and help maintain the desired battery temperature.

The majority of thermal energy is transferred from electronic boxes to the spacecraft equipment panels by conduction at the box/structure interface. Cho-Therm™ is used at the box/structure interface to enhance the heat transfer at the spacecraft interface. In addition, the boxes that contain heat-dissipating components within the spacecraft have high emittance (>0.8) coatings on the outside surfaces to help maximize internal radiation heat transfer and reduce thermal gradients. The equipment panels perform as radiators by placing a specific amount of silver Teflon on the space viewing side of the panels and radiating the excess energy to space.

For all the radiators, the radiating area is determined by using hot-case assumptions and allowing the radiator to dissipate the internal energy and absorbed environmental flux while maintaining the mounting surface below 40°C. Depending on the amount of sun exposure and the thermal-dissipation of the mounted components, between 10 and 60 percent of each panel is dedicated to radiating heat. The radiator temperature is maintained above 0°C using thermostatically controlled heaters and 5 mil silver Teflon as the radiator thermal coating. The radiator sizes were verified during the spacecraft level thermal balance test. The battery radiator panel consists of thermal isolation using G10 spacers and includes a louver for battery temperature control because of special requirements (Table 5).

Table 5. Component Thermal Requirements

Description	Operational	Survival
Propulsion	+10°C to +40°C	+5°C to +50°C
Battery	+0°C to +20°C	-20°C to +35°C
Star Tracker	+13°C to +23°C	-10°C to +35°C

2.7.2 Thermal Design Requirements

The spacecraft thermal design was based on meeting the temperature requirements in Table 6.

Table 6. Nominal Thermal Requirements

Description/Limits	Hot	Cold
Survival	50°C	-10°C
Qualification/Operational	40°C	5°C
Acceptance	35°C	5°C
Design/Predictions	30°C	10°C

Some components have specific temperature control requirements. The EO-1 NiCd battery, the hydrazine propulsion system, and the AST had identified specific temperature requirements. The thermal designs for the components listed in Table 5 were tailored so they meet the thermal requirements for all phases of the mission.

2.7.3 Thermal Control System On-Orbit Performance

The EO-1 TCS has been nominal since launch. While on the launch pad the EO-1 battery was kept cool by receiving controlled fairing air. During launch/ascent the battery TCS performed as predicted, increasing less than a 1°C at the completion of spacecraft ascent. Achieving the predicted temperatures on the HOPS actuators assisted in a successful solar array deployment. In fact, the performance of the EO-1 TCS has been excellent, and no changes have been made to the nominal TCS configuration since launch, and none are expected. The EO-1 instruments, ALI, Hyperion and Linear Etalon Imaging Spectral Array/ Atmospheric Corrector (LEISA/AC) have all reported that the interface temperatures were as predicted and the individual thermal performance of each instrument was excellent. All NMP technologies also reported nominal thermal conditions.

The success of the EO-1 TCS can be attributed to keeping the TCS simple: radiators, thermal coatings, heaters, and louvers. Since the thermal analysis/design is only as good as the information provided by the other systems, obtaining accurate power dissipations, controlling the thermal interfaces and measuring the material thermal properties were essential in achieving an accurate thermal model and thermal design.

3. CONCLUSION

The on-orbit performance of the EO-1 spacecraft bus has been nearly flawless, meeting or exceeding all functional and performance requirements. This was accomplished despite severe programmatic constraints and the challenges of a technology-driven NMP mission.

4. REFERENCES

[1] Mark E. Perry Ph.D., Peter Alea, Michael J. Cully, Michael McCullough, Paul Sanneman, Nicholas Teti and Bruce Zink, "Earth Observing-1 Spacecraft Bus," 15th Annual/USU Conference on Small Satellites, SSC01-V-6, 2001.

ACRONYMS AND ABBREVIATIONS

ACDS	Attitude Control and Data System	MEOP	Mean Effective Operating Pressure
ACE	Attitude Control Electronics	MHz	Megahertz
ACS	Attitude Control System	MLI	Multi-Layer Insulating
Ah	Ampere-hour(s)	MOC	Mission Operations Center
Al	Aluminum	MSSP	Medium Speed Serial Port
ALI	Advanced Land Imager	Nm	Newton-meter(s)
Am	Ampere-meter(s)	NMP	New Millennium Program
asec	arc second	NRZ	Non-Return to Zero
AST	Autonomous Star Tracker	OAP	Orbit Average Power
BSFR	Back Surface Field and Reflector	OVP	Over Voltage Protection
C&DH	Command and Data Handling	PAA	Phased Array Antenna
CCSDS	Consultative Committee for Space Data Systems	PM	Phase Modulate
COBE	Cosmic Background Explorer	PROM	Programmable Read Only Memory
CPU	Central Processing Unit	PRT	Platinum Resistance Temperature
DET	Direct Energy Transfer	PSE	Power Supply Electronics
DJ GaAs	Dual Junction Gallium Arsenide	PSK	Phase Shift Key
DRAM	Dynamic Random Access Memory	RCS	Reaction Control System
DSN	Deep Space Network	RF	Radio Frequency
ECU	Electronic Control Unit	RMS	Root Mean Square
EDAC	Error Detection and Correction	rpm	revolutions per minute
EFF	Enhanced Formation Flying	RSN	Remote Services Node
EFL	Effective Focal Length	RT	Remote Terminal
EO	Earth Observing	SAC	Satelite de Aplicaciones Cientificas
EOS	Earth Observing System	Si	Silicon
FDC	Fault Detection and Correction	SNiCd	Super Nickel Cadmium
FOT	Flight Operations Team	STDN	Space Tracking and Data Network
FOV	Field of View	TCS	Thermal Control System
Gbit	Gigabit	TDRSS	Tracking and Data Relay Satellite System
GN	Ground Network	TRMM	Tropical Rainfall Measuring Mission
GPS	Global Positioning System	TSM	Telemetry and Statistics Monitor
GSE	Ground Support Equipment	UV	Ultraviolet
GSFC	Goddard Space Flight Center	VDC	Volts Direct Current
Hz	Hertz	WARP	Wideband Advanced Recorder Processor
HK	Housekeeping	XTE	X-ray Timing Explorer
HOP	High Output Paraffin	µm	micrometer
I&T	Integration and Test		
IRU	Inertial Reference Unit		
LEISA/AC	Linear Etalon Imaging Spectral Array/Atmospheric Corrector		
LFSA	Lightweight Flexible Solar Array		