ACE SPACECRAFT

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Abstract. The Johns Hopkins University Applied Physics Laboratory (JHU/APL) was responsible for the design and fabrication of the ACE spacecraft to accommodate the ACE Mission requirements and for the integration, test, and launch support for the entire ACE Observatory. The primary ACE Mission includes a significant number of science instruments – nine – whose diverse requirements had to be factored into the overall spacecraft bus design. Secondary missions for monitoring space weather and measuring launch vibration environments were also accommodated within the spacecraft design. Substantial coordination and cooperation were required between the spacecraft and instrument engineers, and all requirements were met. Overall, the spacecraft was kept as simple as possible in meeting requirements to achieve a highly reliable and low-cost design.

1. Introduction

The ACE spacecraft accommodates a total of ten instruments; nine scientific instruments for the primary mission and one engineering instrument for a secondary mission.

– Solar Energetic Particle Ionic Charge Analyzer (SEPICA) – University of New Hampshire, Max-Planck Institute for Extraterrestrial Physics.

– Ultra Low Energy Isotope Spectrometer (ULEIS) – University of Maryland, The Johns Hopkins University Applied Physics Laboratory (JHU/APL).

– Solar Wind Ion Mass Spectrometer (SWIMS) – University of Maryland, University of Bern (Switzerland).

– Solar Wind Ion Composition Spectrometer (SWICS) – University of Maryland, University of Bern (Switzerland).

– Solar Wind Electron, Proton and Alpha Monitor (SWEPAM)–Los Alamos National Laboratory (includes SWEPAME, the electron component, and SWEPAMI, the ion component).

– Electron, Proton and Alpha Monitor (EPAM) – JHU/APL.

– Magnetic Field Experiment (MAG) – University of Delaware Bartol Research Institute, National Aeronautics and Space Administration (NASA) Goddard Space Flight Center.

– Solar Isotope Spectrometer (SIS) – California Institute of Technology, Goddard Space Flight Center, Jet Propulsion Laboratory.

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– Cosmic Ray Isotope Spectrometer (CRIS) – California Institute of Technology, Goddard Space Flight Center, Jet Propulsion Laboratory, Washington University (St. Louis).

– Spacecraft Loads and Acoustics Measurements (SLAM) – Goddard Space Flight Center.

Three of these science instruments, SEPICA, SWICS, and SWIMS, share a data processing unit, designated S3DPU. Nine of the instruments (all but SLAM) support the primary ACE science mission. Four of them – MAG, EPAM, SWEPAM, and SIS – also provide data for the Real Time Solar Wind (RTSW) experiment to monitor space weather (e.g., solar flares and geomagnetic storms). The RTSW experiment is a secondary mission on ACE funded by the National Oceanographic and Atmospheric Administration (NOAA). The ACE spacecraft collects and packages the RTSW data from the four instruments for low-rate transmission during the times when the primary mission data are not being transmitted to the NASA Deep Space Network (DSN). It is anticipated that the RTSW data will be transmitted in this manner 21 hours per day. NOAA is responsible for operating the ground system for the collection and distribution of RTSW data for solar weather monitoring. The RTSW experiment on ACE was implemented at very low cost by using only the spacecraft resources needed for the primary mission. The SLAM instrument is another secondary mission incorporated during the ACE spacecraft development. SLAM is an experiment conducted by Goddard Space Flight Center Code 740 to measure the vibration environment within the spacecraft structure during the first 5 min of launch (before fairing jettison). The flight hardware is self-contained relative to other spacecraft resources; it has its own power supply, transmitter, processor, and sensors. Data from this experiment will go into a database to be used in predicting vibration loads for future spacecraft structural designs.

2. Spacecraft System Description

Figure 1 shows a photograph of the completed ACE Observatory during ground testing. The major technical parameters of the ACE spacecraft are provided in Table I.

2.1. Mechanical description

The ACE structure design was driven primarily by the field-of-view requirements for the instruments, antennas, and attitude components and by the need for a spinning spacecraft to be mass balanced. Further complications were presented by the need for individual thermal radiators on the instruments and structure, which needed views to space, and by handling, access, clearance, and harness routing issues.

An orbital view of the spacecraft is shown in Figure 2, and an exploded view is shown in Figure 3. The primary structural components of ACE consist of ten

TABLE I

Technical parameters of the ACE spacecraft

Attitude	*Star tracker*
	– $20° \times 20°$ field of view
	– +0.1 to +4.5 sensitivity range
	– 30 arc sec (1σ) total random error
	– 1 to 5 simultaneous stars
	Sun sensor
	– $\pm 64°$ field of view
	– Sun angle is gray-coded in 0.5° increments
	– $\pm 0.02°$ short-term repeatability of most significant bit
	Observatory orientation
	– Known (after the fact) to $\pm 0.7°$; stable to $\pm 0.5°$
	Pointing at Earth
	– Angle between the observatory Z axis and the Earth – observatory line is within $\pm 3°$, to stay within the required beamwidth of the high-gain antenna
	– Angle between the Sun–Earth line and the observatory – Earth line is $\geq$ 5°, to limit the solar noise contribution to the receiving system noise temperature
Maneuvering capability	*Tanks*
	– Four tanks with total of 195 kg of hydrazine fuel expelled in blow-down (97% efficiency), providing mission average specific impulse of 216–221 s at 10–21 °C
	Thrusters
	– Four axial and six radial thrusters
	Spin rate
	– Maintained at 5 ± 0.1 rpm to meet science requirements
	Maneuvers
	– All maneuvers except onboard autonomy performed under ground control
Communications	*Downlink data rate*
	– 434 bps (low rate and NOAA)
	– 6944 kbps (real-time transmission)
	– 76 384 kbps (recorder playback interleaved with real-time data)
	– Uplink data rate
	– 1000 bps
	RF frequencies
	– 2097.9806 MHz uplink
	– 2278.35 MHz downlink
Data storage	1.073 Gbit solid-state memory per recorder capacity beginning of life

TABLE I
Continued

Ground contact	DSN network 26 m (primary), 34 m (backup)
	Telemetry designed to be compatible with CCSDS
	Contact nominally once/d (requirement is for recorder capacity to support one missed contact)
Spacecraft safing (see safing section)	*Autonomy*
	C&DH autonomy:
	– during launch to switch to redundant shunt regulator in case of an analog shunt short
	– post launch vehicle separation to turn transmitter off in case of a problem
	– spacecraft and instrument health monitoring
	– abort thruster firing in case of maneuver or attitude problems
	– protection against commands which result in an improper spacecraft configuration
	– support for other autonomous spacecraft actions such as switching recorders when one is full
	Power Subsystem autonomy:
	– shunt regulator switched from primary to redundant for bus voltage over or under conditions
	Spacecraft power bus health monitor
	– load shed and switching to redundant shunt regulator, if required
	Watchdog timers
	– RF antenna switching
	– redundant thruster firing timers
	Reset to restore critical parameters
	– C&DH hardware reset and software boot and intitialization C&DH last-resort timer
	– power subsystem processor reset
Observatory power	280285 W nominal; 385 W peak (136 W payload, 249 W spacecraft) Solar Array
	support >425 W (observatory budget) at 28 V for 5 years
	Launch power 59 W (12-Ah 18-cell NiCd battery supplies 200 W-h to loads)
Observatory wet mass	771 756.54 kg (launch vehicle can support 785 kg)
Structure	Decks and Panels: Honeycomb with aluminum alloy facesheet
	Support structure: Longerons and rails aluminum alloy
	Corner Brackets: Machined titanium

TABLE I
Continued

Mechanisms	Four deployable solar panels, each 86.4 × 149.9 cm
	Two deployable magnetometer booms (magnetometer sensors on ends of boom)
Thermal control	Thermostatically controlled heaters, instrument-specific radiators, and
	observatory radiators are used
Orbit	Halo orbit about L_1
	A_Y = 264 071 km
	A_Z = 157 406 km

Note: C&DH = command and data handling, CCSDS = Consultative Committee on Space Data Systems, DSN = Deep Space Network

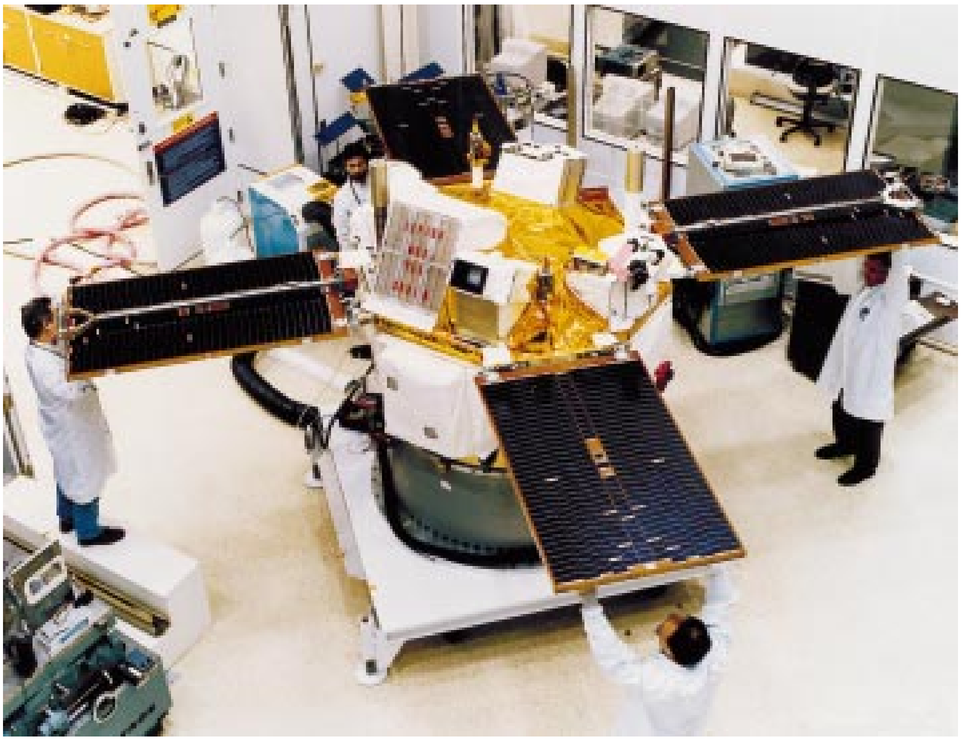


Figure 1. Photograph of the ACE spacecraft during ground testing (solar arrays manually deployed).

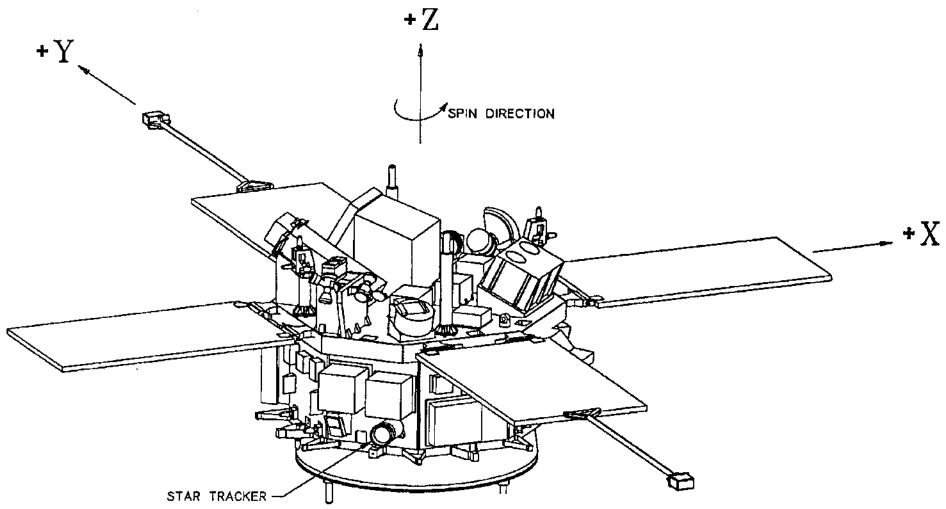


Figure 2. Line drawing of the ACE Observatory in on-orbit configuration.

aluminum honeycomb panels that form a 142.2×76.2 cm closed octagon supported by an internal aluminum and titanium frame. The Observatory Attach Fitting is a 22.9-cm-high aluminum cylinder that attaches the ACE primary structure to a 5624 Delta Payload Fitting with a clampband. The observatory mass, with fuel, is 756.5 kg. Most of the payload instruments (SEPICA, SIS, SWICS, SWEPAME, SWEPAMI, ULEIS, EPAM, and S3DPU) are mounted to the top ($+Z$) deck facing the sun. The $+Z$ deck is isolated from the rest of the structure via buttons (ultem material) and titanium brackets in order to meet instrument temperature requirements. The CRIS and SWIMS instruments and most spacecraft subsystems are mounted to the side decks. The lower ($-Z$) deck houses most of the RF subsystem and the SLAM instrument.

The propulsion subsystem consists primarily of four conospheric titanium tanks and ten thrusters. Thrusters were mounted and located to minimize plume heating effects on the solar arrays and nearby instruments. All propulsion components were integrated and welded together on the $-Z$ deck. This eliminated the need for field joints on the fuel lines and facilitated the final assembly of the system with the rest of the spacecraft primary structure. The tanks were filled with 195 kg of hydrazine fuel and mounted so the primary structural load path was almost directly into the Orbital Attach Fitting. Once the structure assembly was complete, access to the propulsion subsystem was limited. Fill and drain valves and electrical interconnects were all accessible from the outside of the $-X$ side panel.

Four 86.4×149.9 cm deployable aluminum honeycomb solar panels are hinged from the $\pm X$ and $\pm Y$ sides of the $+Z$ deck and are restrained to the $-Z$ deck with pin puller mechanisms during launch. A 152.4-cm titanium boom attaches a magnetometer to the $+Z$ end of each of the $\pm Y$ solar panels. For launch, the $+Z$ end of each boom is restrained to the $\pm Y$ solar panels with a pin puller mechanism.

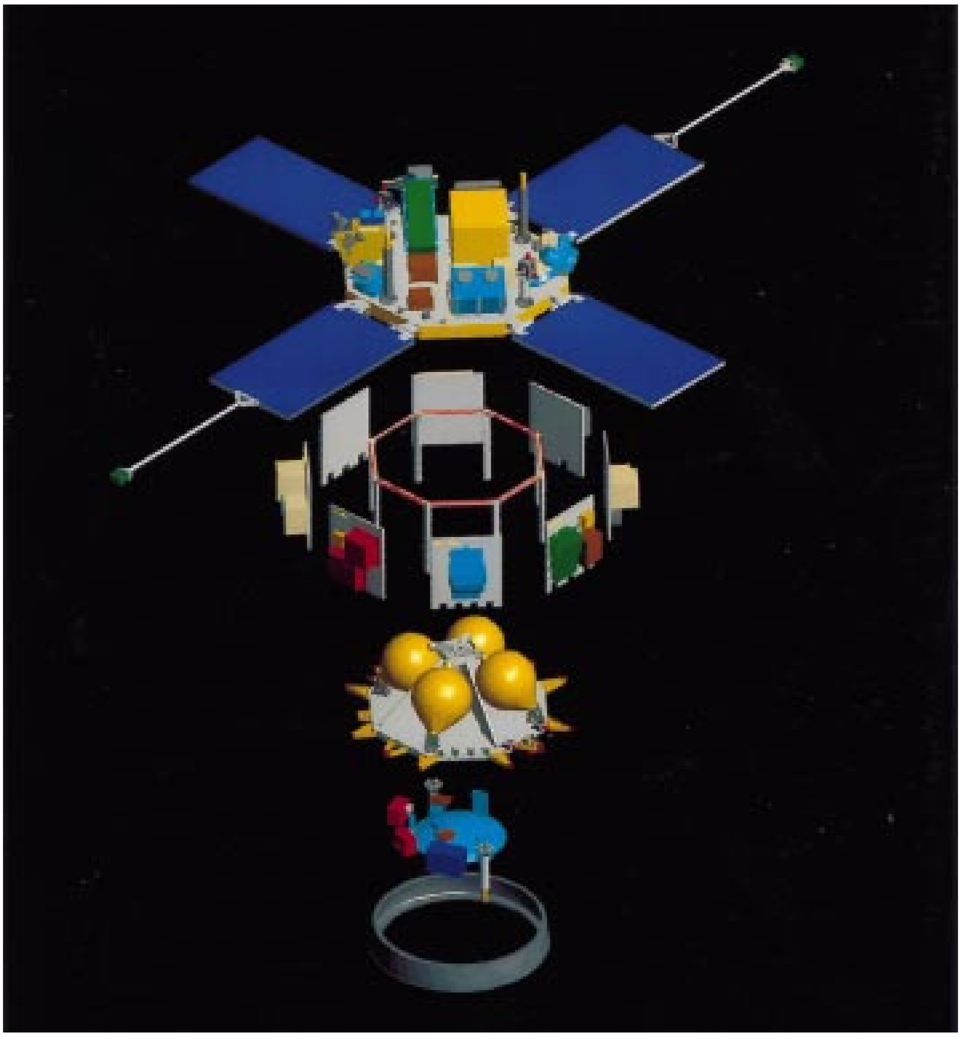


Figure 3. Exploded view of the ACE spacecraft.

Pyrotechnic devices actuate the pin pullers, which release the solar panels and booms. Preloaded torsion springs deploy and center the solar panels and booms in their appropriate positions.

To avoid dynamic coupling of the spacecraft with low-frequency Delta II launch vehicle modes, the primary structure had to be designed with fundamental frequencies above 35 Hz in the thrust (Z) direction and 12 Hz in the lateral (X, Y) directions when mounted to a Delta 5624 Payload Attach Fitting. The spacecraft primary structural modes were measured as about 100 Hz in the thrust direction and 40 Hz in the lateral directions.

The ACE Observatory was designed for launch on a Delta II 7920. The Delta II 7920 flight events produce loads from steady-state and dynamic environments. The steady-state environment produces a maximum thrust acceleration at the end of the first-stage burn or main engine cut-off. The dynamic environments produced by the Delta II are sinusoidal vibration, acoustics, and shock. The sinusoidal environment is a result of liftoff transients, oscillations before main engine cut-off, and engine ignition and shutdown. A spacecraft random vibration environment is generated by launch vehicle acoustics. The ACE Observatory was designed to withstand the loads from these environments. A protoflight test program (flight qualification levels at acceptance-level duration) was used to verify the observatory strength and stiffness. Vibration (sine and random), acoustic, and shock (clampband separation) testing was performed on the fully flight configured observatory.

2.2. Thermal Control Description

The thermal design of the ACE Observatory was created through the joint efforts of the spacecraft and instrument thermal engineers. Early in the program, it was decided that most of the instruments should be thermally isolated from the spacecraft. Uncertainty in the instrument schedules, together with stringent and differing instrument temperature requirements, made thermal isolation the best path to success. Instruments and instrument components whose temperature requirements could be managed by the spacecraft (SWEPAME, SWEPAMI, S3DPU, ULEIS data processing unit, ULEIS analog electronics) were allowed to thermally conduct their heat to the spacecraft.

The observatory uses a combination of multilayer insulation (MLI), thermal radiators, and thermostatically controlled heater circuits to meet its thermal design requirements, without the need for more active (and expensive) methods of moving heat such as louvers and heat pipes. Aluminum doubler plates (additional aluminum material) from 0.15 to 0.32 cm thick were added where necessary to enhance heat conduction away from an area. Where required, instruments were thermally isolated at the mounting interface using a combination of bushings made of insulating material (ultem), titanium mounting hardware, and MLI blankets.

Most of the MLI blankets on the ACE Observatory were fabricated using an outer layer of white Beta Cloth (a type of thermal insulating material) with an embedded 10.2×15.2 cm graphite weave. The Beta Cloth provides durability and the desired optical properties while minimizing specular reflections. Although a conductive spacecraft was not a requirement for the ACE program, the graphite weave was incorporated into the Beta Cloth to provide some protection from electrostatic discharge.

In some areas, special requirements dictated the use of alternative materials for the MLI outer layer. Thruster plume impingement is possible on the $+X$ side panel in the area of the battery and on the Observatory Attach Fitting below the $+X$ and $-X$ aft axial thrusters. The MLI blankets were hardened to increased temperature by replacing the Beta Cloth with five layers of embossed 0.25-mil aluminized Kapton and a 3-mil aluminized Kapton outer layer. Specular reflections are not a problem in these areas. In addition to the special requirements imposed by the spacecraft, the SWIMS and SWEPAME instruments had special requirements that dictated the use of more electrically conductive MLI outer layer materials. ITO-coated aluminized Kapton was used for SWIMS and for the $+X+Y$ side panel below SWEPAME. The alternative material was allowed only on the aperture face of SWEPAME to avoid problems with specular reflections.

The forward ($+Z$) deck of the observatory is covered by thermal insulation to shield it from the sun as much as possible. Heat is rejected from the forward deck via radiators attached to each of the eight deck edges. The silvered Teflon radiators face radially away from the axis of symmetry of the observatory to maximize heat rejection.

The aft ($-Z$) deck of the observatory serves as the mounting platform for the internal components of the propulsion system. In addition, components of the RF subsystem and the SLAM instrument are mounted to the space side of the aft deck. The Observatory Attach Fitting is bolted to the aft deck and is covered with silvered Teflon to serve as a thermal radiator for the side panels and the aft deck. Except for the RF-radiating surfaces of the two low-gain antenna towers and the high-gain antenna dish, the aft deck is protected from a direct view toward space by an MLI blanket.

The spacecraft side panels are used primarily for mounting spacecraft components. However, the $+X-Y$ and $-X+Y$ panels are used for the isolated mounting of the SWIMS and CRIS instruments, respectively. The spacecraft thermal design minimizes heat exchange between the side panels and space by covering the side panels with MLI blankets that extend from the bottom of the forward deck radiators to the Observatory Attach Fitting mounted to the aft deck. For side panels with instruments, the spacecraft blankets provide radiative isolation between the instrument and side panel. Sun sensor and star scanner apertures, thruster nozzles, and umbilical connectors remain free from MLI blockage.

The spacecraft radiators were oversized during design and fabrication. The effective radiator sizes were then tailored using MLI after correlation between the results of the observatory-level thermal vacuum test and the observatory thermal model.

On orbit, the ACE spacecraft thermal environment is expected to show little or no short-term variation. The environment will vary as the sun angle changes from a nominal minimum (4°) to a nominal maximum (20°). However, sun angle variation will occur on a scale of months and will be corrected periodically via the propulsion system. The actual maximum sun angle will be controlled to less than 20° to keep the high-gain antenna pointed at the DSN. The additional control provides greater margin in the spacecraft hot-case thermal analysis, where 20° was used as a worst-case assumption. The spin of the spacecraft (5 rpm) will dampen any short-term azimuthal variations.

The observatory thermal analysis includes worst-case variations in all thermal design parameters, including incident solar flux, sun angle, MLI effectiveness, optical properties of thermal radiators and other external surfaces, and minimum voltage on heaters. Therefore, the analysis results bracket all possible temperature variations over the entire mission life.

The spacecraft thermal control system is robust and versatile. A primary requirement of the thermal control system is to minimize peak heater power while providing adequate support of observatory temperatures. Survival, operational, and interface heaters manage the resources to meet the observatory's thermal requirements.

Survival heater circuits provide control when instruments are off and when most spacecraft components are either off or in low-power states. Operational heater circuits provide control during normal observatory power dissipation states. To minimize peak heater power, interface heaters are used to replace part of the dissipation of a component placed in a nonoperational state while the operational heaters are active. Interface heaters are used for all isolated instruments and for some higher-power spacecraft components. Without interface heaters, larger operational heater circuits would have been needed to accommodate the occasional lower-power modes. The result would have been higher peak operational heater power.

The extensive use of thermal components to distribute heater power about the observatory provides added reliability to the design. In general, heater circuits consist of multiple thermostats, each with multiple small heaters. The failure of an individual heater element is not likely to cause significant problems. Many of the survival thermostats have the same temperature set points as the operational thermostats due to the overriding requirement that hydrazine in the fuel lines not be allowed to freeze. Therefore, in many cases, the survival heaters can provide additional backup capabilities.

2.3. Spacecraft Subsystem Descriptions

The spacecraft subsystems are described in the following paragraphs. A simplified block diagram is provided in Figure 4.

2.3.1. *Command and Data Handling Subsystem*

The ACE command and data handling (C&DH) subsystem provides capabilities for ground and stored commanding; onboard autonomy and safing; collection, formatting, and storage of science and engineering data; switching of spacecraft power and energizing of pyrotechnic devices; thruster firing control; and generation and distribution of sun pulses and spin clocks. The C&DH subsystem consists of redundant C&DH components, two 1-Gbit solid-state recorders (SSRs), a power switching component, and an ordnance fire component. Special design features are included to avoid any single-point mission failures and to allow continuous

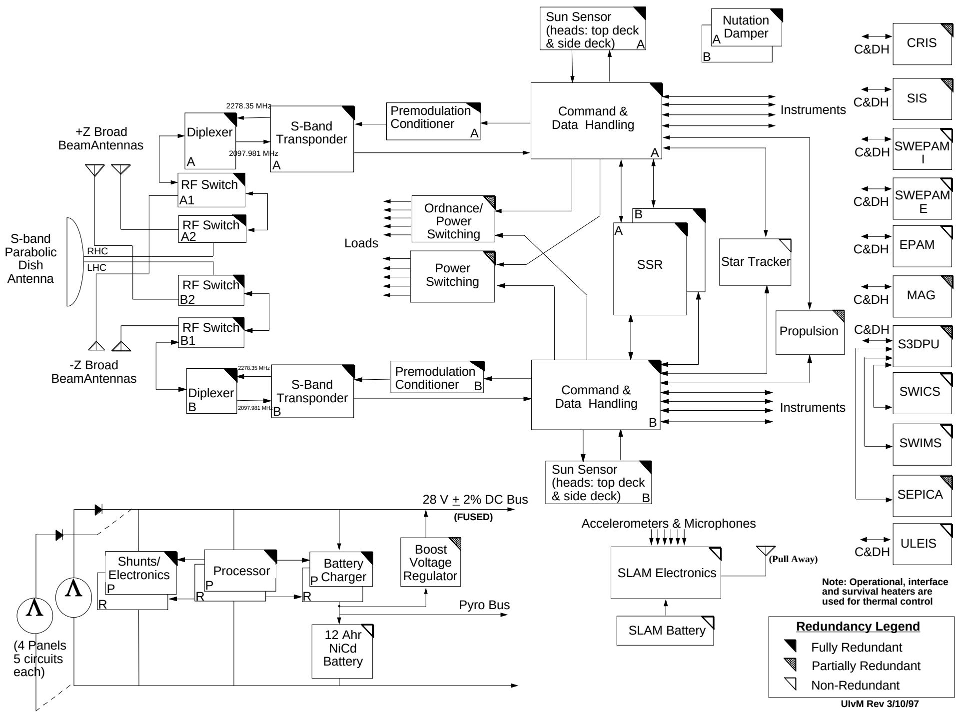


Figure 4. System block diagram of the ACE Observatory.

operation through solar flares. Major requirements of the C&DH subsystem are shown in Table II.

Each C&DH component processes and executes commands, formats downlink and stored telemetry frames, controls thrusters, and generates sun pulses and spin clocks. Uplinked commands are received in Consultative Committee Space Data Systems (CCSDS) – compatible telecommand transfer frames at a rate of 1000 bits per second (bps). Transfer frames are accepted and ground notification is performed using a modified CCSDS Command Operation Procedure-1 command protocol. Commands can be executed immediately, stored and executed at a specific spacecraft time, executed in response to an out-of-limit spacecraft health parameter, or stored as a command macro. Blocks of stored commands are executed in response to discrete fault indications from the power subsystem and also when the spacecraft has separated from the launch vehicle. Antennas are automatically reconfigured if no valid commands are received in a period that can be set from 0.5 to 96 hours by ground command. All command outputs include design features to prevent single failures from causing false commands to be sent. Data commands transfer 1 to 512 bytes of user data to other onboard subsystems at a rate of 1200 bps. Logic pulse commands provide mode switching pulses to other subsystems. Both data and logic pulse command signal outputs are edge-

TABLE II
Major requirements of the C&DH subsystem

Major requirement	Impact
No single point mission failures	Redundant C&DH components; include numerous watchdogs and safety circuits to prevent anomalous operation; redundant SSRs; redundant control of relays
Continuous operation through solar flares	Hardened logic; triple voting FPGAs; EDAC on memory
Interface with heritage instruments and fixed data rate to each instrument	Point-to-point discrete interfaces for science data collection
CCSDS-compatible downlink	Package minor frames in CCSDS Packet-Virtual Channel Transfer Frame; Reed Solomon and convolutional encoding
Time-tag data to allow correlation to 0.1 s with Universal Time	Use of Temperature-Compensated Crystal Oscillator
Perform thruster firing, meet safety requirements	Design tolerates two or more failures before improper thruster activation
Store 50 hours of science data (missed ground contact)	Redundant 1-Gbit SSRs
Downlink real-time science with SSR playback	Interleave Virtual Channel 1 real-time Science frames with Virtual Channel 2 SSR playback frames containing Virtual Channel 4 recorded frames
Respond to uplink and stored commands	Execute real-time commands and store commands in time-tagged and block bins
Reconfigure antennas if no ground contact C&DH component includes RF watchdog timer to periodically reconfigure antennas in absence of valid real-time commands	
Respond to out-of-limit telemetry and fault conditions	Evaluate autonomy rule bins once per second; respond to discrete fault indication signals from power subsystem
Generate and distribute sun pulse and spin clock signals to instruments	Use sun angle information to generate signals

Note: C&DH = command and data handling, CCSDS = Consultative Committee on Space Data Systems, EDAC = error detection and correction, FPGA = field-programmable gate array; SSR = solid-state recorder.

rate controlled to prevent electromagnetic interference problems. Table III shows the number and type of each command interface.

Each C&DH component can operate in Full Collection mode (collect all data and generate sun pulses and spin clocks), Reduced Collection mode (collect all data except science), or Disabled Collection mode (collect no data). One C&DH

TABLE III
C&DH subsystem user commands

Command type	Number provided	Comment
Data command	16	1 to 512 bytes of user data
Logic pulse	16	40-ms pulse
Relay	101	2-A and 10-A latching relays; 2-A and 5-A nonlatching relays

Note: C&DH = command and data handling.

TABLE IV
C&DH subsystem telemetry channels

Telemetry channel type	Number provided	Comment
Serial digital	16	Includes major and minor frame pulses, clock, read-out gate
0–5 V single-ended analog	62	
0–50 mV differential analog	62	Used primarily for measuring currents
AD590 temperature transducer	62	Measures temperatures in range of $-60\,^\circ$ to $+100\,^\circ$C
PT103 temperature transducer	62	Measures temperatures in range of $-100\,^\circ$ to $+150\,^\circ$C
Digital telltale-logic	32	Used primarily for RF transponder telemetry
Digital telltale-switch	16	Used primarily to detect the state of switches

Note: C&DH = command and data handling.

component is designated the active C&DH and operates in Full Collection mode. The other C&DH component is designated the inactive C&DH and operates in Reduced Collection mode. Digital and analog telemetry data are collected with six types of telemetry channels (Table IV) from onboard subsystems at a single rate and format. Data are collected in a major/minor frame structure at a rate of 1 minor frame per second, with 16 minor frames per major frame. The collected data are stored in the Data Collection Buffer resident in C&DH component memory. The Data Collection Buffer is the source of data in all telemetry formats and for autonomy (i.e., checking out-of-limit values of spacecraft telemetry). Both C&DH components can simultaneously limit-check spacecraft telemetry data.

Three requirements – the need to be compatible with several existing instruments, the need to simplify the C&DH – instrument interface by allocating each instrument a continuous fixed science data rate, and the need to have a CCSDS-compatible downlink – led to the development of a hybrid telemetry format scheme that combines a traditional major/minor frame structure with a modern CCSDS packet/transfer frame structure. Each telemetry frame consists of a Virtual Channel Transfer Frame containing one telemetry packet. Each packet contains a single telemetry minor frame in its data field. Real-time data can be formatted into one of ten minor frame formats for downlinking or recording (Table V). The Downlink and Record formats are independently selected. Data can be downlinked at 434 bps (low-rate engineering data or real-time solar wind data), 6944 bps (science and engineering data), or 76 384 bps (real-time and SSR playback data) while another format is being recorded at 6944 bps. During normal operations, data are recorded in the science and engineering format at 6944 bps while real-time solar wind data are downlinked to NOAA ground stations at 434 bps for 21 hours per day. For 3 hours per day, real-time data are interleaved with SSR playback data at a composite rate of 76 384 bps and transmitted to the NASA DSN. Eleven frames are downlinked per second: one real-time frame and ten SSR playback frames. Downlink frames are encoded with the CCSDS standard Reed Solomon (Interleave = 4) and convolutional coding.

Each C&DH component contains a temperature-compensated crystal oscillator that is the source of all C&DH telemetry timing. A 32-bit time count is provided in each minor frame. The precision of the oscillator permits correlation of onboard data with Universal Time (UT) to at least 100 ms accuracy. Each C&DH component utilizes sun angle data from the attitude determination and control (ADandC) subsystem to generate sun pulse and spin clock signals that are distributed to instruments. For each revolution of the spacecraft, 16 384 spin clock pulses are generated. All science data can be correlated to telemetry synchronizing signals (major and minor frame pulses) that are generated at known times or to the sun pulse/spin clock signals. Each sun pulse is tagged with the current spacecraft time.

The two SSRs each provide 1 Gbit of storage. They are designed to operate with a less than 10^{-7} bit error rate after 26 hours of continuous solar flare. (This radiation level significantly exceeds the worst-case scenarios ever anticipated during the ACE mission.) In operation, data are recorded at an average rate of 6944 bps and reproduced at an average rate of 68 640 bps. Data are reproduced in forward order. Under normal operation, the observatory will be under ground control once every 24 hours. However, the observatory is designed for one missed contact, with autonomous collection and storage of data for up to 48 hours. If an SSR fills up with data, the active C&DH component will use the autonomy to start recording data on the second SSR.

Each C&DH component has interfaces to the Power Switching and ordnance fire components to allow the selection and activation of relays. Each includes latching relays for providing switched power to spacecraft subsystems and nonlatching relays for providing pulses. The latching relays include dual coils for redundant control. Redundant nonlatching relays are used. The ordnance fire component also includes interfaces to each C&DH component for firing thrusters.

TABLE V
C&DH telemetry formats

Downlink and record telemetry formats	Rate (bps)	Description
Attitude determination and control	6944	Contains all housekeeping and ADandC data; limited science data; repeats every second
Science	6944	Contains all science data; housekeeping data repeats every 16 s
C&DH memory dump	6944	Replaces housekeeping data in science format with C&DH memory dump data
C&DH bin dump	6944	Replaces housekeeping data in science format with C&DH bin dump data
Recorder test pattern	6944	Data field contains pseudorandom pattern to load SSR with known data
Low-rate housekeeping	434	Contains all housekeeping data, repeats every 16 s
Low-rate C&DH memory dump	434	Identical to low-rate housekeeping except includes C&DH memory dump data
Low-rate C&DH bin dump	434	Identical to low-rate housekeeping except includes C&DH bin dump data
Low-rate attitude determination and control	434	Contains all ADandC data and most housekeeping data
Real-time solar wind	434	Includes science data for real-time evaluation of solar wind

Note: ADandC = attitude determination and control, C&DH = command and data handling, SSR = solid-state recorder.

Each C&DH component controls the firing of thrusters in the propulsion subsystem. Thrusters can be fired for a single specific time duration (axial firing) or they can be pulsed during multiple spacecraft spins (sector-based firing). Thrusters are organized into Top Deck Thruster Fire and Bottom Deck Thruster Fire groups. Each C&DH component controls Thruster Select relays in the ordnance fire component. The Thruster Select relays allow thrusters in each group to be selected for firing; Thruster Arm relays enable firing to take place. When thruster firing is to begin, a C&DH component activates a Top Deck or a Bottom Deck Thruster Fire signal to the ordnance fire component for the duration of the firing. The thruster control circuitry is designed so that at least two failures would have to occur before any thruster would inadvertently fire. Independent watchdog timers are used to detect and halt thruster firing that is not halted by the primary control mechanism. Three commands must be successfully executed in order to begin thruster firing.

The C&DH component was designed and fabricated at JHU/APL. It utilizes a Harris RTX2010 processor executing the FORTH language. The RTX2010 is fabricated in a CMOS/SOS process that is exceptionally hard to single-event upsets (SEUs), making it suitable for operation through solar flares. Code is stored in electrically erasable/programmable read-only memory (EEPROM) and downloaded into random-access memory (RAM) for execution. EEPROM can be reloaded on the ground, and the RAM can be patched in flight. Both RAM and EEPROM utilize error detection and correction (EDAC) circuitry to correct single errors and detect double errors. Digital logic is implemented with radiation-hardened Harris HCS series logic, radiation-tolerant National FACT series logic, and Actel field-programmable gate arrays (FPGAs). The FPGAs are designed with triple voting cells to minimize the probability of SEUs. The RTX2010 is programmed in FORTH using a FORTH kernel and cross-compiler developed at JHU/APL and previously used on the Freja and Near Earth Asteroid Rendezvous (NEAR) spacecraft.

The SSRs were designed and fabricated at SEAKR Corporation. They utilize IBM 16-Mbit dynamic random access memories (DRAMs) for storage and a Harris 80C85 microprocessor for control. Error detection and correction is done on 16-bit words and allows for correction of single errors and detection of double errors. Failed memory segments are automatically mapped out.

The power switching and ordnance fire components were designed and fabricated at JHU/APL. They are implemented with a modular design having redundant relay coil driver cards and application-specific relay cards. They operate directly off the spacecraft power bus. The design was previously used on the Ballistic Missile Defense Organization's Midcourse Space Experiment spacecraft and on the Near Earth Asteroid Rendezvous spacecraft.

2.3.2. *RF Communications Subsystem*

The primary function of the RF communications subsystem is to serve as the observatory terminus for radio communications between the observatory and the NASA Deep Space Network of Earth stations. A secondary function is to transmit downlink data in real time, at 434 bps, to Earth stations supporting the NOAA Real Time Solar Wind project. The system is designed to receive uplink commands and transmit downlink telemetry data concurrently with coherent ranging. The system operates at 2097.9806 MHz for the uplink and 2278.35 MHz for the downlink.

The system consists of two identical (redundant) and independent communications subsystems and a single high-gain, dual-polarized parabolic reflector antenna. Each communications subsystem consists of a transponder (transmitter and receiver), diplexer, coaxial switching network, and two broad-beam antennas. There is no cross strapping between RF subsystems. The coaxial switching network is used to connect a given transponder to an aft $(-Z)$ or forward $(+Z)$ broad-beam antenna or to the aft high-gain parabolic antenna. Watchdog timers, implemented in software within the C&DH subsystem, are designed to switch the broad-beam antennas if no uplink spacecraft commands are received within a preset time. The timers provide a means to recover spacecraft communications in the event of a communications system or attitude anomaly. Both receivers are operated continuously, but only one transmitter is to be powered and one antenna energized at any given time.

Antennas

The purpose of the broad-beam antennas is to provide hemispherical uplink coverage forward and aft of the spacecraft. The broad-beam antennas also provide sufficient gain in the region $\pm 34°$ off boresight for the transmission of 434 bps data to a DSN 26-m ground station. The broad-beam antennas provide right circularly polarized coverage (gain) over a hemisphere. The gain of the broad-beam antennas over the hemisphere is greater than -7 dBic at uplink frequencies, -8 dBic at downlink frequencies. By configuring the RF system so that one receiver is connected to a forward broad-beam antenna and the other to an aft broad-beam antenna, omnidirectional coverage of the spacecraft can be achieved. At angles less than 34° off boresight, the broad-beam antenna gain is -1 dBic at uplink frequencies, 0 dBic at downlink frequencies.

The purpose of the high-gain antenna is to support DSN uplink commands and the transmission of downlink data at 76 384 or 6944 kilobits per second (kbps). The high-gain antenna is also used to transmit 434-bps data to NOAA Real Time Solar Wind sites. The specified gain of the high-gain antenna at 4.25° off boresight is greater than 18 dBic at the uplink frequency, 19 dBic at the downlink frequency. The high-gain antenna is pointed by ground commands to the spacecraft attitude subsystem. The antenna must be pointed within $\pm 3°$ of the spacecraft–Earth line in order to have sufficient gain to achieve the signal-to-noise ratios required for the data links. On orbit, the attitude of the observatory must be adjusted periodically to keep the Earth within the $\pm 3°$ beam width of the antenna.

Transponder

The transponder consists of a receiver and transmitter that are diplexed onto a single line feeding the antenna. The receiver is a phase-locked, dual conversion, superheterodyne type with a detector and command demodulator. The transmitter is a phase-shift-keyed type with an RF output power of 5 W minimum. The output spectrum consists of a residual carrier with data sidebands.

Communications modes

The transponder receiver and transmitter may be operated independently (noncoherently) or coherently, where the downlink frequency is derived from the uplink signal frequency. The coherent mode permits the measurement of two-way Doppler. The functional modes of operation are command reception, telemetry transmission, and ranging.

Commands reception
The uplink command data rate is 1000 bps. The pulse code modulation is NRZ-L. The command data are phase-shift-keyed onto a 16-kHz subcarrier and then phase modulated onto the 2097.9806-MHz RF carrier.

Telemetry transmission
There are three selectable downlink data rates: 434 bps, 6944 bps, and 76 384 bps. The pulse code modulation is bi-phase-L. The downlink data streams are encoded by the C&DH subsystem before they are routed to the transmitter. The coding consists of a convolutional code (7, $\frac{1}{2}$) concatenated with a (248 217) Reed-Solomon code. The data stream is directly phase modulated onto the 2278.35-MHz downlink carrier, producing a residual carrier and data sidebands. The modulation index is selectable (high/low) by ground command. The low modulation index is required for transmitting the 434-bps data, and the high modulation index is used for the 6944-bps and 76 384-bps data rates.

Ranging
The system is capable of simultaneous command reception, telemetry transmission, and ranging. However, a noninterfering signal structure, such as that produced by the DSN Sequential Ranging Assembly (SRA), is required. The ranging clock frequency is approximately 512 kHz; the lower frequency codes are expected to be component numbers 6 through 17.

2.3.3. *Attitude Determination and Control Subsystem*
The ADandC subsystem was designed to minimize spacecraft cost and complexity while maximizing reliability and mission success. It utilizes the inherent gyroscopic stability of a spinning spacecraft for attitude control coupled with telemetered sun sensor and star scanner data for determining attitude on the ground. The ADandC subsystem consists of a solid-state star scanner, a redundant sun sensor system (which acts as an on-orbit backup to the star scanner), two fluid-filled ring nutation dampers, the ten thrusters of the propulsion system, and the command capability of the C&DH subsystem. It is an extremely simple system that has proven itself on a variety of other missions.

Each redundant sun sensor system from the Adcole Corporation consists of two sun angle sensors and an associated electronics box. Each sensor digitally encodes to an 8-bit value, the sun angle in nominal 0.5° increments over a field of view of ±64° in each of two orthogonal axes. One sun angle sensor of a pair is located on the $+Z$ deck of the spacecraft with its normal parallel to the $+Z$ axis; the other is mounted on the side deck with its normal canted 125° away from the spacecraft $+Z$ axis. Unless there is an attitude anomaly, the sun will always shine on the top-deck sun angle sensor. The sun sensor electronics forwards to the C&DH subsystem the two encoded 8-bit sun angles from the illuminated sun angle sensor and two identification bits indicating which of the two sensors is providing

TABLE VI
Observatory instantaneous attitude error budget

Parameter	Angular error (deg)
Star scanner accuracy	0.025 (3σ)
Star scanner mechanical mounting (total)	0.023
Side panel thermal distortion	negligible
Principal axis misalignment (max)	0.20
Residual nutation (max)	0.25
Total (RSS)	0.498

the data. The C&DH subsystem records the sun angle data for inclusion in the downlinked telemetry and both generates a sun-crossing pulse and initializes a sector clock based on the transition of the most significant bit of one of the two sun angle values from the illuminated sensor.

The observatory attitude is determined on the ground by combining the telemetered sun angle data with high-accuracy data provided by a Ball Aerospace Systems Division CT-632 solid-state star scanner. The CT-632 star scanner is a star tracker modified to operate at the ACE spin rate of $30^\circ\ s^{-1}$, which is two orders of magnitude greater than nominal star tracker angular rates, with no significant impact on the error budget (1σ error of 30 arc sec) (Radovich 1995). Time delay integration is used to accumulate the star image signal on the charge-coupled device (CCD) so that standard star tracker image processing algorithms can be used to determine star centroids and magnitudes. The data from the star scanner are collected by the C&DH subsystem for telemetering to the ground, where they are combined with the sun angle data to determine the attitude of the observatory.

The requirement for attitude knowledge is $\pm 0.7^\circ$ after the fact, with a goal of $\pm 0.5^\circ$ for the magnetometer. The spacecraft components were assigned budgets for attitude errors, which are given in Table VI.

Attitude control is achieved by the inherent passive, gyroscopic stability of a major-axis spinning spacecraft. Two 0.46-m-diameter hoops filled with an ethylene-glycol solution provide purposeful energy dissipation to damp nutational motion. Open-loop, ground commanded firings of the hydrazine thrusters are used to precess the observatory spin axis to follow the nominal $1^\circ\ day^{-1}$ apparent motion of the Earth and Sun and to adjust the spin rate as needed. Operationally the spin axis is precessed once every 5 or 6 days, whereas spin-rate adjustments will be rare.

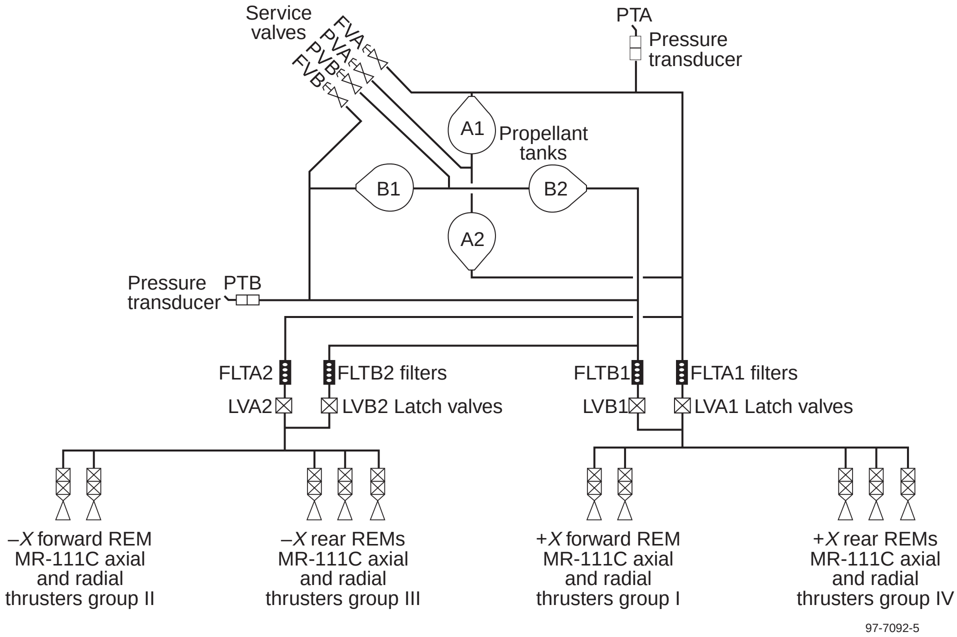


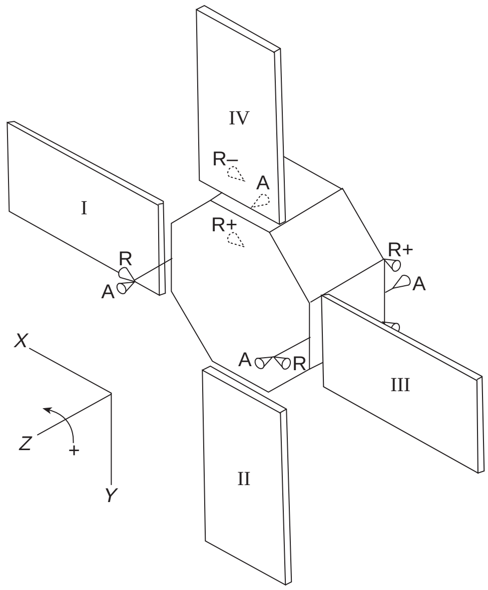
Figure 5. Schematic of the propulsion subsystem, illustrating the cross strapping between the A and B side tanks, service valves, and thrusters (PV = pressurant valve; FV = fill valve; REM = reaction engine module).

2.3.4. *Propulsion Subsystem*

The ACE propulsion subsystem will correct launch vehicle dispersion errors, inject the spacecraft into the L_1 halo orbit, adjust the orbit, adjust spin axis pointing, and maintain a 5 rpm spin rate. The subsystem is a hydrazine blowdown unit that uses nitrogen gas as the pressurant and is made up of four fuel tanks, four axial thrusters for velocity control along the spin axis, and six radial thrusters for spin plane velocity control and spin rate control, as well as filters, pressure transducers, latch valves, service valves, heaters, plumbing, and structure. Figure 5 presents a schematic of the subsystem and illustrates the cross strapping between the A and B side tanks, service valves, and thrusters. Figure 6 shows the location and function of the ten thrusters.

The four propellant tanks are 65.1-m^3 titanium cone spheres with a total capacity of 195 kg, which provides for a mission lifetime of 5 years with considerable margin. The tanks are initially pressurized to 305 pounds per square inch absolute (psia) at 21°C, and they blow down to 91 psia when the propellant is fully expended. The ten thrusters each provide 4.4 N nominal thrust and, combined with the mission duty cycle, provide a mission average specific impulse of 216 s minimum.

The pressure transducers have a 0–500 psia range and are individually powered through relays. The latch valves are flown in the open position and are normally used only for post-loading shipment at Kennedy Space Center. They could be



Thrusters	Velocity Control				Spin Axis Pointing				Spin Rate	
	+*Z*	–*Z*	*X*–*Y* plane						Increase	Decrease
IA		x			x					
IR			x				x			
IVA	x					x				
IVR+			x					x	x	
IVR–			x					x		x
IIA		x				x				
IIR				x				x		
IIIA	x				x					
IIIR+				x			x		x	
IIIR–				x			x			x

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Figure 6. Location and function of the ten thrusters.

used to isolate a leaking thruster or propellant tank if such a failure were detected during the mission. Tank, line, and thruster valve heaters are thermostatically controlled and maintain the propellant components comfortably above the 0°C hydrazine freezing point. Thruster catalyst bed heaters are relay controlled and preheat the bed prior to thruster firings to prevent cold-start degradation. The subsystem plumbing is all welded and sized to minimize any differential flow pressure drop that might cause a spin imbalance. Orifices are used in the latch valves and at the entrance to the pressure transducers to minimize pressure surge and water hammer effects. Filters upstream of each latch valve and at the inlet of each thruster remove any potential contaminating particulate before it reaches the component.

The propulsion subsystem was designed and fabricated by Primex Corporation (formerly Olin Aeorospace) and shipped to JHU/APL for integration with the spacecraft after thermal blanket installation. It was environmentally tested at protoflight levels using water to simulate propellant and using mass simulators for the rest of the spacecraft subsystems.

2.3.5. *Power Subsystem*

The ACE electrical power subsystem provides 444 W at spacecraft end of life (EOL), with EOL defined as 5 years. The electrical power load budget is 425 W peak. The low-magnetic-emission, fixed planar silicon solar array is connected directly to the 28 V $\pm 2\%$ shunt-regulated bus. The electrical power subsystem also contains an 18-cell 12 A-h NiCd battery booster capable of supporting a 165-W launch load. This subsystem is one-fault tolerant with its autonomously controlled redundant shunt regulator and cross-strapped redundant battery chargers.

Solar array

TECSTAR was contracted to fabricate and lay down five strings, each 89 solar cells long, on each of four aluminum honeycomb substrates. The $86.4 \times 149.9 \times 3.2$ cm substrates, insulated with 3-mil Kapton, were provided by Applied Aerospace Structures Corporation. The 3.9×6.3 cm, 15.1% efficient silicon cells have a diffused boron back surface field/reflector and are covered with 12-mil, ceria-doped microsheet. The cells and covers have anti-reflective coatings, and the covers also have ultraviolet reflective coatings. Strings are back-wired for magnetic emissions below 0.1 nT at the magnetometer, which is housed on a boom extending 150 cm from the panel edge. Silver interconnects, which are nonstandard, were used to achieve low magnetic emissions. TECSTAR provided a coupon (i.e., a sample) and a qualification panel with accelerated thermal cycle testing to validate the silver interconnects.

The solar arrays were designed for several worst-case operating conditions. Maximum temperature expected is 58°C while under thruster plume impingement of the ultrapure hydrazine propellant. Conservative degradation estimates project enough EOL power to tolerate one string failure with no effect on spacecraft performance. A solar cell patch located on each panel supports an experiment to measure the degradation over mission life. Strings also have bypass diodes to protect against shadows cast by spacecraft instruments, antennas, and thrusters. Pyrotechnic-actuated, spring-loaded hinges deploy the panels after spacecraft separation from the Delta II.

Battery

JHU/APL designed and built the 18-cell stack with redundant thermostatic controlled heaters and current, temperature, and full- and half-stack differential voltage monitors. The thermal design will keep the battery between 0° and 25°C. The cells for the flight battery are a SAFT Gates Aerospace Batteries (GAB, Gainsville, Florida) 12AB28 standard profile design ($4.6 \times 3.0 \times 0.9$ cm, 536 g). Although aged for 8 years in dry storage at Gainsville, cells successfully activated in France perform well (typically 15 A-h).

Electronics

The electronics consists of six boxes and a shunt resistor dissipater assembly. Each solar panel has a digital shunt box nearby containing five metal oxide semiconductor field-effect transistors (MOSFETs) and blocking diodes for low-dissipation shorting of each string. The panel current and the voltage of each string are monitored. Fuses, which help filter the regulated bus, protect the bus from a shorting failure of either blocking diodes or capacitors.

The power subsystem dissipater electronics contains redundant linear transistors, the booster, and redundant battery chargers. The 90% efficient booster can be configured in flight to provide partial battery reconditioning. The battery chargers provide closed-loop current and voltage limiting. All boxes are constructed with a combination of semi-rigid flex printed circuit boards and heat sink subassemblies. The traces are sized for 70-A fuse blowing capability and 15-A continuous power to the bus with less than a 2-V drop from the solar panel connector to the subsystem dissipater electronics output.

The power subsystem control electronics contain redundant hybrid power converters from Interpoint, shunt regulation electronics, and a circuit that will switch from the primary to the redundant side in response to bus under- or overvoltage. It also contains redundant processors and low battery voltage and low bus voltage sensors. The 80C85RH-based processors digitize all electrical power subsystem telemetry for transmission via cross-strapped serial digital links to the redundant spacecraft telemetry systems. The processors also decode commands cross-strapped from the redundant spacecraft command systems. Digital-to-analog converters allow adjustment of the bus voltage set point ($\pm 2\%$), battery charge limits (0 to 1.5 A), and temperature-compensated battery voltage limits.

3. Environmental Design Drivers

As a result of specific ACE science and mission concerns, three environmental factors were given significant attention during the design and fabrication of the ACE Observatory. By addressing these factors from the beginning of the design process, goals and requirements were achieved with relatively little additional cost.

3.1. Magnetic Cleanliness

The observatory has two magnetometer sensors, one at the end of each magnetometer boom. Although ACE had no magnetic requirement, there were magnetic goals. Magnetic cleanliness precautions were undertaken to the extent practical (i.e., without driving the cost), to limit the magnetic fields from the spacecraft components that could radiate to the sensors. The goal was to achieve a spacecraft static magnetic field at the magnetometer sensors of less than 0.1 nT. The goal for AC interference at the sensor locations was less than 0.001 nT over a frequency range of 0 to 10 Hz and the specific frequencies of 15 kHz ($\pm$200 Hz), 30 kHz ($\pm$200 Hz), and 60 kHz ($\pm$200 Hz). These goals were achieved as follows:

– The spacecraft battery was wired to reduce the magnetic field and to minimize magnetic loops, and the battery was demagnetized and located as far as practical from the magnetometer sensors on the $+X$ side panel (sensors are at $\pm Y$ axis).

– Solar panels were back-wired for compensation. Magnetic loops at the end of the panels were minimized.

– Twisted pair wire was used for power lines; a single-point ground system was used; and grounding was designed to minimize ground loops, all to the extent possible.

– Magnetic materials used in spacecraft components were reviewed on a global basis, and a random selection of typical parts was tested.

– Boom material is titanium; hinges and materials close to sensors are nonmagnetic; and de-permed tools were used.

– The magnetometer team used ‘sniffing’ devices to map the magnetic emissions of selected suspect spacecraft components and parts (e.g., analog shunt resistive strips and latch valves). As a result, propulsion subsystem latch valves and latch valves for the SEPICA instrument were shielded with Met-glass.

– The ordnance fire component was compensated with external magnets after pre-integration tests.

– Nonmagnetic connectors were used for spacecraft components as much as possible.

– The magnetic field generated by the shaker tables used for spacecraft components and the observatory was measured (both with and without the degauss coils), and, where required, degauss coils were used to reduce the large fields generated by the exciter.

– Instruments and spacecraft subsystems were screened prior to integration to reduce the possibility that compensation of the entire spacecraft would be required.

– No observatory-level compensation was performed after preflight static field tests.

Two preliminary results of the static test prior to launch indicated that if the source locations are neglected, and if the conservative assumption is made that the source of the field is at the center of the spacecraft, a worst-case field of 2.3 nT at each flight sensor is predicted. If the sources have been correctly identified, based in part on the static test and on pre-integration screening results, then the in-flight field components are only $\frac{1}{10}$ this value. The preliminary results of the AC test revealed no AC signals at a level that could significantly affect the magnetic field measurements.

The magnetometer sensors are flown in ‘twin design,’ with one sensor on each boom, as opposed to ‘dual design,’ with two sensors on a long boom on the same side of the spacecraft, so that the actual field on orbit is folded into the magnetometer sensor and electronics calibration. The actual field will never be known but, from data for the early phase of the mission, is estimated to be <1 nT at the sensors.

3.2. Radiation Environment

Spacecraft components had to be designed to meet the radiation requirements as follows. Parts were required to survive a total ionizing dose of 15 krad (Si) without part failure (with spot shielding as required). The 15 krad (Si) requirement was based on a 5-year mission goal using the projected radiation exposure for the mission with a 95% level of confidence and assuming a nominal shielding of 75-mil aluminum. Since ACE could operate for 4 years at solar maximum, the components had to withstand a total proton fluence of approximately 2^{11} particles cm^{-2} for protons with energies above 4 MeV. Components had to be immune to latch-up; parts susceptible to single-event latch-up with linear energy transfer threshold less than 120 MeV cm^2 mg^{-1} could not be used in spacecraft components without latch-up mitigation techniques. Parts susceptible to SEU could not be used in critical components if these could cause mission-critical failures. In parts of noncritical components, SEU could not compromise spacecraft health or mission performance. System-level SEU effects were considered, so that upsets did not cause uncorrectable errors that would affect system performance. SEU-susceptible parts used for data storage memory had to use appropriate error detection and correction techniques so that during exposure to maximum particle flux (as experienced during a worst-case solar particle event), the amount of data lost would not cause a violation of the component specification requirements.

3.3. Contamination Control

ACE instruments are susceptible to hydrocarbons, fluoro-hydrocarbons, particles, and humidity. To satisfy instrument requirements and permit a comfortable working environment, the instruments were purged and the environment kept at class 100 000 clean room level (this level requires only the use of white coats). Chemicals and compounds used were carefully screened. Nitrogen purge with 99.995% or better purity at a temperature of 15° to 25°C was supplied to the instruments through a manifold at a pressure of 3.0 pounds per square inch (psi) nominal (5.0 psi maximum) with a flow rate of 5.9 cm^3 min^{-1} total. Each instrument used a restrictor to obtain its required flow rate. The purge remained connected (with short interruptions of minutes’ duration) until launch. The sun sensor and star scanner optical portions were covered unless ground support equipment (stimulators or sun simulator) was attached.

3.4. Spacecraft Safing

Spacecraft features are used to 'safe' the spacecraft when problems are recognized onboard or when spacecraft actions, independent of ground intervention, are required. These features are implemented with C&DH 'autonomy rules,' built-in power subsystem autonomy, dedicated lines between the power subsystem and the C&DH for power bus fault indications, C&DH RF watchdog timers, and C&DH and power subsystem resets.

The C&DH autonomy rules (called 'autonomy' here) are widely used on the spacecraft to put an instrument, a spacecraft component, or the whole observatory (spacecraft and instruments) in a 'safe' configuration when certain criteria are met. The criteria used to determine the need for action, and the actions are predefined but can be changed. A component parameter such as a current, which requires monitoring, is collected by the C&DH. The value of the parameter is compared ($=$, $>$, $<$, etc.) to a preselected threshold, for example, the maximum expected current for that component. Based on the result, the C&DH autonomy may issue commands such as unpowering the component if the current is too high. Below are examples where autonomy is used on the spacecraft for all phases of the mission: launch, postlaunch vehicle separation, and on-station operation.

- Launch
  - Autonomy switches from the prime to the redundant power subsystem shunt regulator in the unlikely case of an analog shunt short.
- Postlaunch vehicle separation activities
  - After the solar panels are automatically deployed, autonomy controls the safe configuration of the spacecraft. For example, if the solar panels are supplying power (indicating successful panel deployment), the transmitter is commanded to radiate, thereby facilitating first ground contact for the ground operators. Otherwise, the transmitter is commanded to standby, thereby conserving power until ground intervention can determine the cause of the problem.
- Health monitoring
  - When instrument and spacecraft health indicators, such as currents, exceed predefined limits, autonomy takes action, such as unpowering a component, until the cause of the problem can be determined by mission operations.
  - For redundant thermostats / heaters, autonomy removes a failed thermostat / heater of a redundant set but keeps the other powered.
  - During battery reconditioning, autonomy stops battery reconditioning if battery health indicators exceeds limits.
- Maneuver or attitude problems
  - Autonomy aborts thruster firing and closes the ULEIS instrument shutter in case of a spacecraft attitude problem. This problem is indicated by the Sun angle exceeding a predefined limit or by the side panel Sun sensors sensing the Sun.

  - Thruster firing abort commands are issued by autonomy after a maneuver if, because of a problem, the maneuver is not completed with the normal process.
- Protection against commands resulting in improper spacecraft configurations
  - The autonomy feature can be used to protect against possible command errors. Two examples follow. Should both transmit/receive chains be erroneously connected to the top-deck broad-beam antennas, autonomy is used to configured the active side to the bottom-deck antenna (which is oriented toward Earth). If both transmitters are configured to radiate, autonomy is used to configure the inactive side transmitter to standby.
- Support for other autonomous spacecraft actions
  - After an RF watchdog timer time-out, which indicates abnormal operation, the C&DH commands the active transmitter to stand by (prior to reconfiguring the broad-beam antennas). Autonomy is used to command the transmitter to radiate and to select the 'attitude determination and control' format for the downlink, thereby minimizing the time required for the ground station to acquire the spacecraft and to reconfigure the downlink.
  - Under certain conditions, e.g., power shedding, both transmitters may be turned off by the C&DH. Autonomy is used to power the interface heater thermostat to keep the bottom deck at an acceptable temperature for the other components on the deck.
  - When one recorder is filled (for example, when a ground contact where the recorder would be dumped is missed), autonomy is used to start recording on the other recorder so data loss is minimized.

The power subsystem autonomy feature is used to detect bus regulation failures indicated by overvoltage (30 V) or undervoltage (26 V) conditions. For either condition, the power subsystem autonomously switches control from the primary to the redundant shunt regulator in <10 ms. Note that in the load shed sequence described in the next paragraph, the C&DH autonomy takes 90.0 to 181.8 ms to switch to the redundant shunt regulator.

The power subsystem monitors the health of the spacecraft power bus and detects instrument or spacecraft hardware fault conditions that cause the bus to fall below specified limits. Dedicated lines between the power subsystem and the C&DH are used to indicate faults to the C&DH. The latter autonomously executes a predetermined sequence of commands to shed loads. The default sequence, loaded at launch, sheds all noncritical loads and switches control from the primary to the redundant power subsystem shunt regulator. Graduated sequences, which shed loads depending on the severity of the failure, will be used later in the mission. The graduated sequences attempt to preserve telemetry data history and minimize changes to the spacecraft configuration as long as possible to help the ground operators determine the cause of the fault. Note that the C&DH load shed and the power subsystem autonomy can be individually disabled, if necessary

The C&DH RF watchdog timer is used to place the spacecraft in a communications safe mode when a valid command has not been received for a specified time. When the specified time has expired and the RF watchdog timer in the active C&DH has timed out, the active C&DH sets the RF antenna switches in its chain to connect the transmit/receive chain on its side to the bottom-deck broad-beam antenna. When the RF watchdog timer in the inactive C&DH times out, it sets the RF antenna switches in its chain to connect the transmit/receive chain on its side to the top-deck broad-beam antenna, thereby providing hemispheric uplink coverage to facilitate reception of a command. At the next time-out, this configuration is reversed, with the active side connected to the top-deck antenna and the inactive side connected to the bottom-deck antenna. The C&DH toggles between these two configurations for each time-out until a valid command is received, which resets the timer.

Each C&DH has a two independent timers used to halt thruster firing. If one timer fails during a maneuver, the other will still time out and stop the firing.

The C&DH has four sources that can generate a reset: last resort timer, software watchdog timer, error detection and correction double bit error, and C&DH circuit bus time-out error. These result in C&DH hardware reset and software boot and initialization. Critical parameters are restored either to their pre-reset value (if two of the three backup copies of the parameter match) or their default value (if these do not match). The power subsystem processor resets whenever the power processor watchdog timer is not reset within 2 s. Again, majority voting is used to restore critical parameters to their pre-reset value or default value.

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