ADVANCED COMPOSITION EXPLORER (ACE)

LESSONS LEARNED AND FINAL REPORT

JULY 1998

Explorers Project, Code 410
Flight Projects Directorate
NASA/Goddard Space Flight Center
Greenbelt, MD 20771

This document provides lessons learned from the development of the ACE mission and is the final report for the execution phase of the mission. It is hoped that this document will serve as a useful reference for future GSFC missions, as well as a repository of key summary data and information about ACE.

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1.0 INTRODUCTION AND SUMMARY OF FINDINGS

1.1 INTRODUCTION

This document is the final report for the Advanced Composition Explorer (ACE), which was successfully launched August 25, 1997. The purposes of the report are to document the history and uniqueness of the ACE project; to objectively demonstrate how successfully ACE has met its objectives and the NASA expectations of faster, better, cheaper; and to document lessons learned from ACE that may aid future NASA projects.

To assure objectivity, this report has been compiled by an individual not associated with the ACE project. Questionnaires were used to elicit facts and opinions from several key individuals covering all project activities. There were iterations to assure clarity and completeness. The author then integrated and summarized the responses and other materials to prepare this report.

The following section 1.2 summarizes the key findings from this examination of ACE. Section 2.0 provides an overview of the ACE mission. Section 3.0 demonstrates the bottom line success ACE had in terms of on-orbit performance, cost, and schedule. Section 4.0 discusses unique and innovative aspects of ACE. Section 5.0 reviews the risks and ways they were mitigated. Section 6.0 examines the lessons learned from the ACE experience. Section 7.0 provides detailed discussions of the elements of the ACE mission. Section 8.0 presents details of the ACE costs and Civil Service work force.

1.2 SUMMARY OF FINDINGS

1.2.1 Bottom Line Success

The ACE project is a success. ACE is performing very well on orbit, the project was significantly underbudget, and launch was on schedule. Also, ACE did well in comparison to historical cost and schedule trends.

1. On-Orbit Performance

Thus far, ACE is performing satisfactorily on orbit and in a manner to achieve its success criteria.

2. Cost Performance

ACE underran its budget by **$34.3 million for a 24 percent savings**. The project also underran its Civil Service work force budget by **25.2 work years for a 25 percent savings**.

ACE costs are **less than historical expectations by $30.8 million or 22 percent**. Civil Service work years are **39 work years or 34 percent less than historical expectation**.

3. Schedule Performance

ACE **beat its not-to-exceed launch date by four months, and met its planned launch date without deviation**.

The ACE **schedule duration from start to launch is consistent with historical experience and was not any faster**.

1.2.2 Key Reasons for Success

While a large number of things contributed to the success of ACE, there was general agreement among the various ACE personnel on the key reasons:

- People--good, dedicated, hardworking
- Cooperation among all the groups--team effort
- Good up-front planning
- Adequate Phase B study effort
- Adequate funding when needed
- Good management

Elements of good ACE management which are worth emulating in the future include:

- Maintaining full and open communications in all directions
- Being open, honest and frank in discussions with contractors or other supporting groups
- Giving appropriate credit to the doers of good work
- Defining project/instrument descope options early in the program
- Doing what needed to be done and not standing on ceremony
- Sticking to design where "good is good enough" and not allow hardware changes just for the sake of "improvement"
- Placing heavy emphasis on maintaining schedule
- Providing the ability to reallocate funds as needed
- Always remembering who the customer is

Everything mentioned above was key to the success of the ACE project. However, some unique aspects of ACE that were critical are:

- ACE had a long well-funded Phase B. This allowed the resolution of problem areas that normally are worked out during Phase C/D.

- ACE used a Principal Investigator (PI) mode of developing and managing the science payload. Considerable delegation of responsibility was given to the PI institution, and the PI was a full member of project management.

- ACE was the first major NASA mission to adopt a common ground system (i.e., the same core hardware and software) for the I&T ground system, mission operations ground system and science data center ground system. This achieved significant savings.

- ACE had a fixed price and fixed launch date approach, similar to the very successful approach for the X-Ray Timing Explorer (XTE) project. ACE

- management took the fixed price mandate very seriously, and the fixed price concept was used throughout all elements of the project. Schedule was maintained tightly, which provided a control of cost.

Given the success of both XTE and ACE, it appears that the GSFC Explorers Project Office, in conjunction with Headquarters OSS, have constructed a more effective and efficient mode of project management. One that fits well with the NASA concept of faster, better, cheaper.

2.0 OVERVIEW OF ACE MISSION

2.1 SUMMARY

The ACE is designed to study the origin and subsequent evolution of both solar system and galactic material in investigations of fundamental questions in space physics. ACE is in orbit at the Earth-Sun L1 libration point to conduct in situ measurements of particles originating from the solar corona, the interplanetary medium, the local interstellar medium and galactic matter.

The ACE mission concept was proposed in response to a NASA Headquarters Space Science and Applications Notice, dated March 1986, requesting participation in the Explorer Concept Study Program. It was one of four missions selected for a Phase A Concept Study. The ACE Phase A study was conducted from early 1988 to mid-1989, ending with science objectives and a defined payload complement. ACE then entered a Phase B Definition Study in early 1991 for a thorough study of about three years. The Phase B study supported the NASA Non-Advocate Review (NAR) in September 1993, which led to approval to proceed into the execution phase (Phase C/D) at the start of Fiscal Year (FY) 1994 in October 1993. See Figure 2.1-1 for the ACE project timeline. The thoroughness of the Phase B effort allowed an overall and spacecraft Preliminary Design Review (PDR) to be held in the first month of the execution phase. Mission development was managed at Goddard Space Flight Center (GSFC) by the Explorers Project. Payload development was managed by the California Institute of Technology (CIT). The spacecraft was developed and integrated by the Johns Hopkins University's Applied Physics Laboratory (APL). The ground system was developed by GSFC and the ACE Science Center (ASC) by CIT. Following the integration of the instruments and spacecraft, the observatory underwent vibration testing at APL. It was then taken to GSFC for the remainder of the environmental test program and then, after eight of the nine instruments were removed for rework or calibration, shipped to the Kennedy Space Center for final integration, checkout and launch.

On August 25, 1997 ACE was successfully launched from Cape Canaveral Air Force Station on a Delta II 7920 expendable launch vehicle. Then four months of orbit maneuvers took ACE to its mission halo orbit around the L1 libration point in the latter part of December 1997. During this period all instruments were outgassed and then turned on for checkout and calibration during the cruise to L1. It was during this period, on November 4 and 6, 1997, that two large energetic solar particle events occurred. Most of the ACE instruments were able to obtain data from these events and produce outstanding scientific results. Figure 2.1-2 illustrates the transfer trajectory and mission orbit. ACE was declared operational February 1, 1998. During on-orbit operations the spacecraft propulsion maneuvers ACE to maintain communication with the Deep Space Network (DSN). In this configuration the nine scientific instruments measure the isotopic and elemental composition of the solar wind and energetic particles in the interplanetary medium with a mission design lifetime of at least two years and an operational goal of five years. Spacecraft control originates at the GSFC ACE Mission Operations Center (MOC), with commands and telemetry through the DSN. Daily stored data dumps and real-time data are collected by the DSN, transmitted to GSFC, and then delivered to the ASC at CIT. The ASC processes the data to higher levels, analyses and archives it, and distributes it to the Co-Investigators and the scientific community. The ASC also places the data into the public domain.

FIGURE 2.1-l

ACE PROJECT TIMELINE

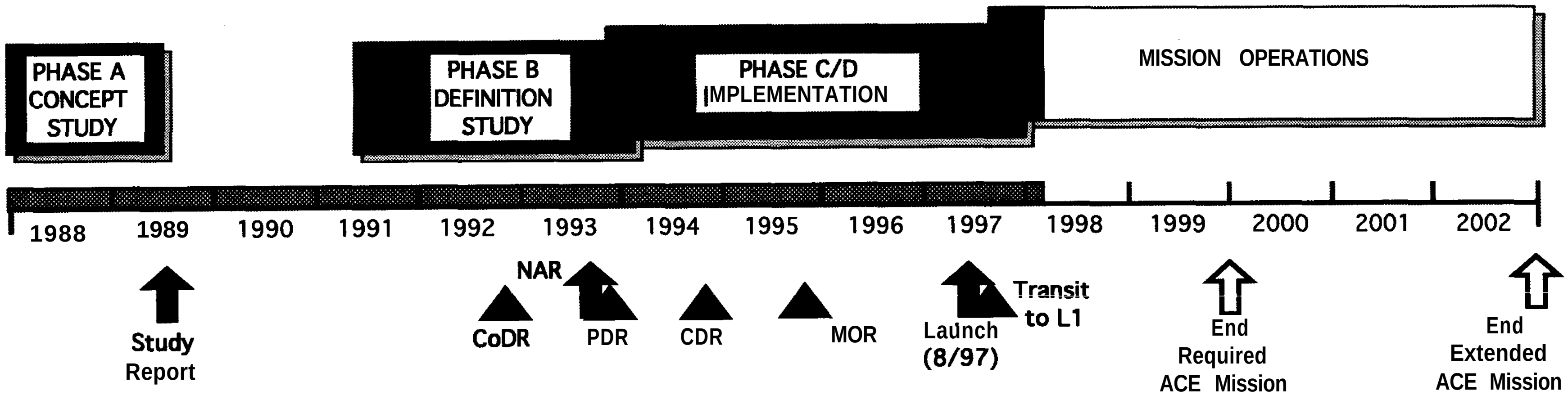
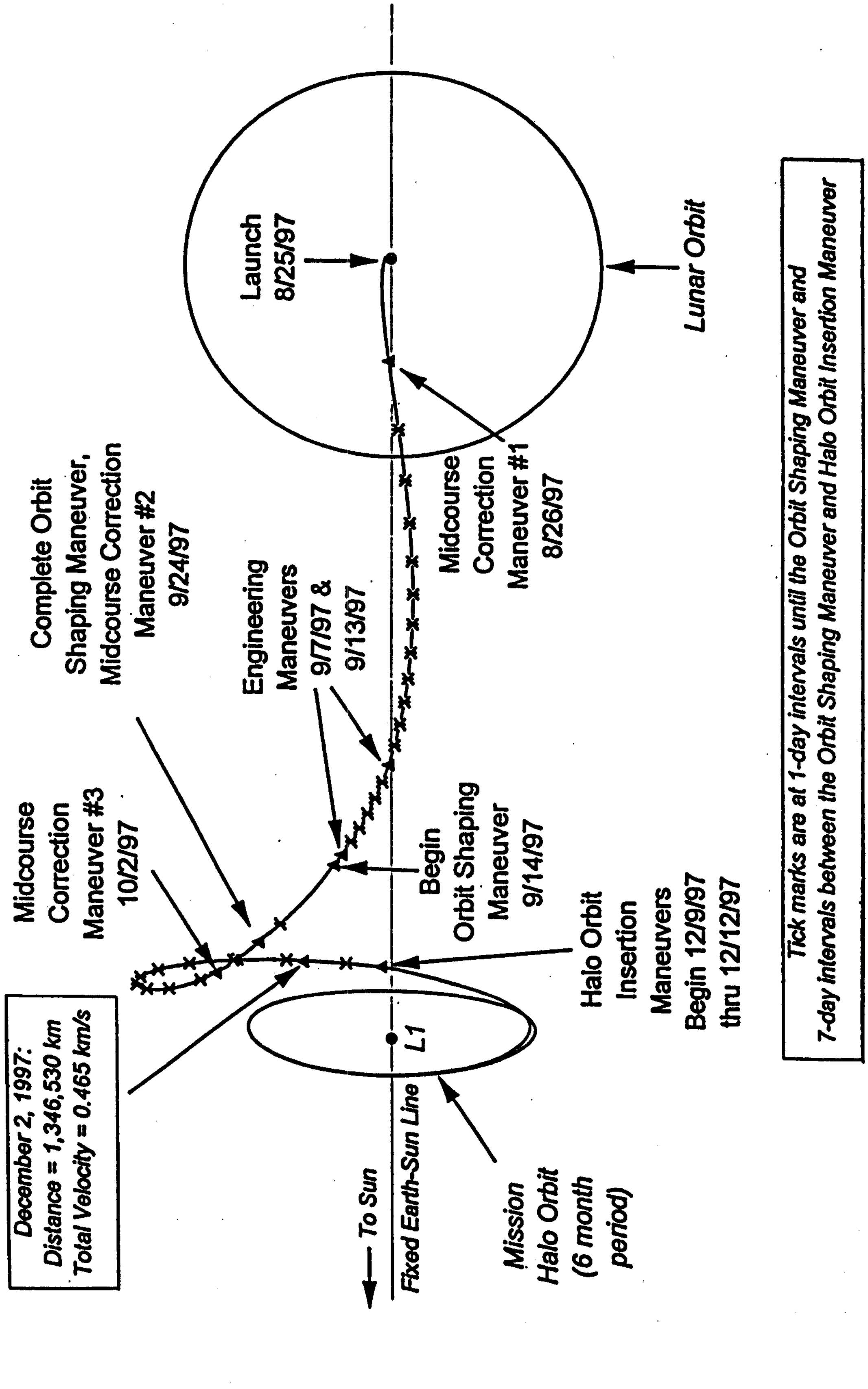


Figure 2.1-2

ACE Transfer Trajectory and Mission Orbit

XY (Ecliptic) Projection in Geocentric Solar Ecliptic Coordinates



2.2 PROJECT ORGANIZATION AND RESPONSIBILITIES

There were a number of organizational interfaces for ACE, although they were relatively straightforward. Figure 2.2-1 shows the project organization and Figure 2.2-2 illustrates the organizational responsibilities.

The Project Office had overall responsibility for the mission. APL was responsible for overall observatory systems engineering. The spacecraft (observatory less instruments) was designed and fabricated by APL. APL prepared the overall system specifications, integration and test plan, handling plan, and other overall system plans. Interface to the launch vehicle contractor was initially through the Orbiting Launch Services Project at GSFC. Although this was the official route, direct communication between APL engineers and Boeing engineers was encouraged and implemented. Direct communication between APL engineers and Instrument Development Teams was also implemented to quickly resolve interface issues.

The CIT Payload Management Office (PMO) initiated subcontracts with the University of Maryland for the Solar Wind Ion Composition Spectrometer (SWICS), the Solar Wind Ion Mass Spectrometer (SWIMS) and half of the Ultra Low Energy Isotope Spectrometer (ULEIS); the University of New Hampshire for Solar Energetic Particle Ionic Charge Analyzer (SEPICA); the University of Delaware for the Magnetic Field Monitor (MAG); and CIT for the Cosmic Ray Isotope Spectrometer (CRIS) and the Solar Isotope Spectrometer (SIS). GSFC implemented contracts with the Los Alamos National Laboratory for the Solar Wind Electron, Proton and Alpha Monitor (SWEPAM) and with APL for the Electron, Proton and Alpha Monitor (EPAM) and half of ULEIS. Instrument development support was provided by JPL and by GSFC (testing, parts procurement) when requested. Most of the observatory test program was conducted at GSFC.

With nine science instruments and more than that number of organizations involved, every interface was treated uniquely. Yet all received their technical direction from the CIT Payload Management Office, from which the majority of documents governing instrument development emanated. These included instrument design and test specifications, the payload Performance Assurance Requirements (PAR), and many other controlling documents. Funding was handled by subcontracts from CIT, except for Co-I groups at U.S. government laboratories who, except for JPL, received their money from the Goddard Project Office. JPL support was provided by way of internal CIT Work Orders.

The interface between each ACE instrument and the spacecraft was carefully defined in interface control documents (ICDs). These documents represented the formal agreement between the instrument supplier, the ACE Project Office (GSFC) and APL, and permitted design efforts to proceed to the final development stage. Changes to the documents required formal Change Board approval. In general, this system worked well and no unusual interface problems were experienced.

Ground system development for operations was done by GSFC, and both APL and the ASC developed their own systems. The core of each system was the same, but unique features were added as appropriate. Significant systems interfaces between these organizations were required.

Figure 2.2-1: ACE Project Organization

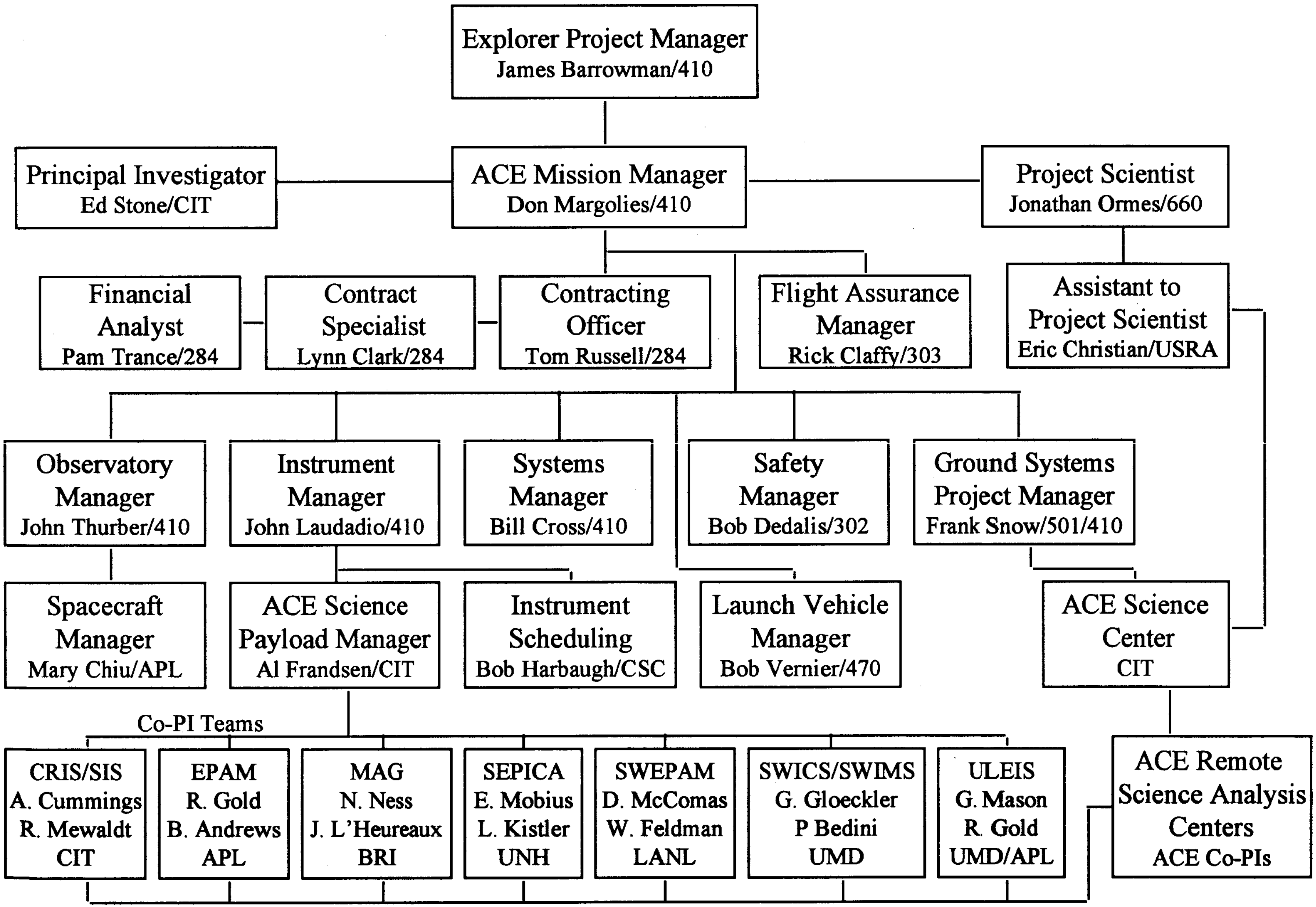
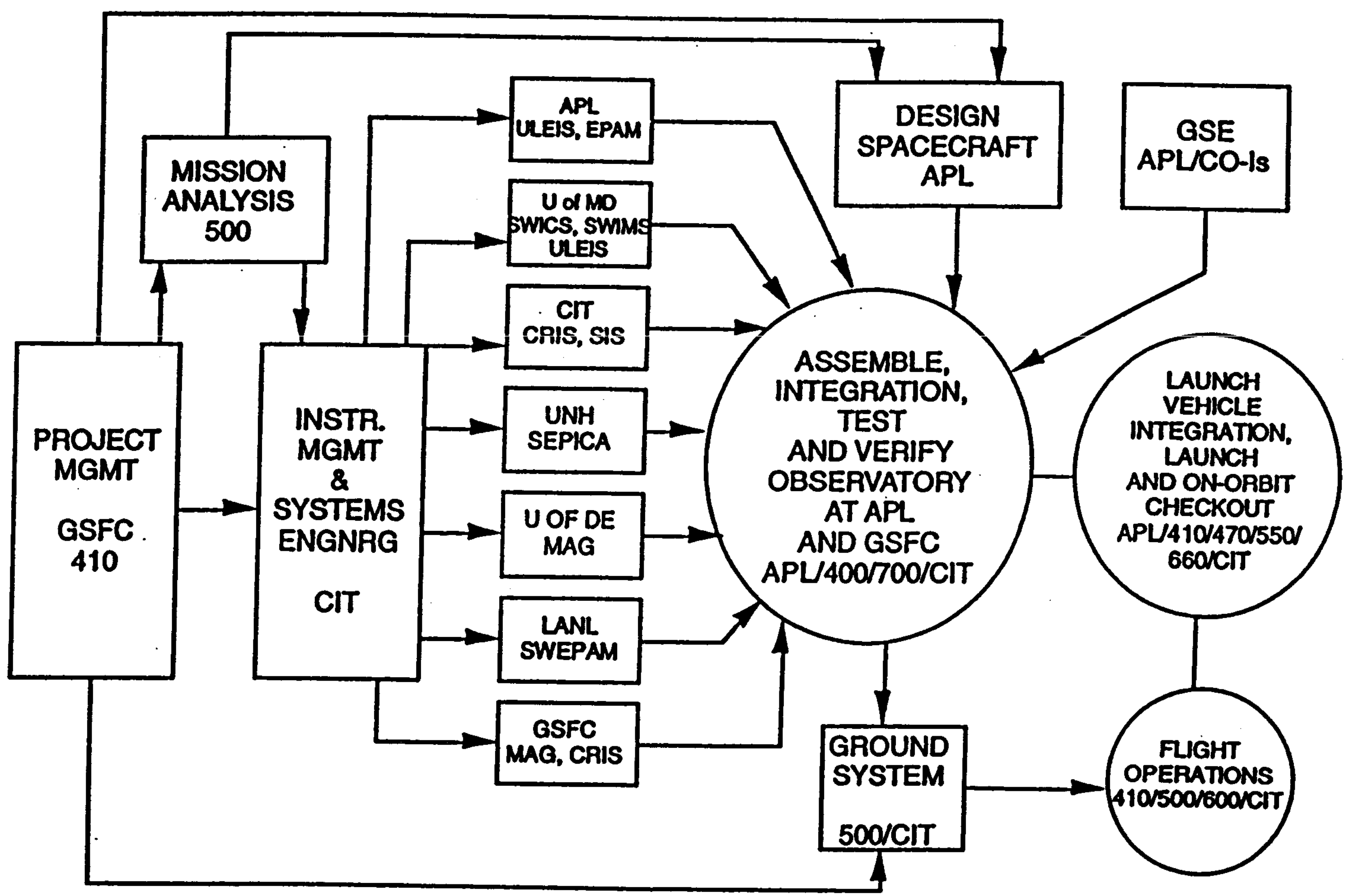


Figure 2.2-2

ACE Project Responsibilities



2.3 OBJECTIVES AND REQUIREMENTS

2.3.1 Program Characteristics

The NASA Explorer Program is characterized by relatively low to moderate cost and small to medium sized satellites capable of being built, tested and launched in a short time compared to the large observatories. The Explorer missions are capable of supporting various scientific disciplines. ACE fits this mold very well and is part of the overall Explorer Program administered by the Astrophysics Division of the Office of Space Science (OSS) at NASA Headquarters.

2.3.2 Programmatic Objectives

The ACE mission development, covering the execution phase (Phase C/D), was undertaken by GSFC and OSS on a not-to-exceed cost basis. They agreed that the total cost of ACE from start of Phase C/D through launch plus 30 days would not exceed $141.1 million (in real year dollars). In addition, GSFC agreed to launch ACE no later than December, 1997. As will be detailed later, the actual cost was considerably less than the limit and the launch date was earlier than the objective.

2.3.3 Mission Objectives

The prime objective of ACE is to determine and compare the elemental and isotopic composition of several distinct samples of matter, including the solar corona, the interplanetary medium, the local interstellar medium, and galactic matter. This objective is to be achieved by performing comprehensive and coordinated determinations of the elemental and isotopic composition of energetic nuclei accelerated on the Sun, in interplanetary space, and from galactic sources. These observations will span five decades in energy, from solar wind to galactic cosmic ray energies, and will cover the element range from hydrogen to zirconium ($Z=1$ to 40). The comparison of these samples of matter will be used to study the origin and subsequent evolution of both solar system and galactic material by isolating the effects of fundamental processes that include nucleosynthesis, charged and neutral-particle separation, bulk plasma acceleration, and the acceleration of suprathermal and high-energy particles. These observations will allow the investigation of a wide range of fundamental questions in the following four major areas of the ACE Level 1 Science Requirements.

2.3.4 Level 1 Science Requirements

1. The Elemental and Isotopic Composition of Matter

A major objective is the accurate and comprehensive determination of the elemental and isotopic composition of the various samples of “source material” from which nuclei are accelerated. ACE measurements will be used to:

- Generate a set of solar isotopic abundances based on direct sampling of solar material.
- Determine the coronal elemental and isotopic composition with greatly improved accuracy.

- Establish the pattern of isotopic differences between galactic cosmic ray and solar system matter.
- Measure the elemental and isotopic abundances of interstellar and interplanetary “pick-up” ions (galactic neutrals ionized by solar ultraviolet radiation or the solar wind).
- Determine the isotopic composition of the “anomalous cosmic ray component”, thought to represent a sample of the local interstellar medium.

2. Origin of the Elements and Subsequent Evolutionary Processing

Isotopic “anomalies” in meteorites indicate that the solar system was not homogenous when formed, while other data suggest that the solar composition continues to evolve. Similarly, the galaxy is neither uniform in space nor constant in time due to continuous stellar nucleosynthesis. ACE measurements will be used to:

- Search for differences between the isotopic composition of solar and meteoritic material.
- Determine the contributions of solar-wind and solar flare nuclei to lunar and meteoritic material, and to planetary atmospheres and magnetospheres.
- Determine the dominant nucleosynthetic processes that contribute to comic ray source material.
- Determine whether cosmic rays are a sample of freshly synthesized material (e.g., from supernovae) or of the contemporary interstellar medium.
- Search for isotopic patterns in solar and galactic material as a test of galactic evolution models.

3. Formation of the Solar Corona and Acceleration of the Solar Wind

Solar energetic particle, solar wind and spectroscopic observations show that the elemental composition of the corona is differentiated from that of the photosphere, although the processes by which this occurs, and by which the solar wind is subsequently accelerated, are poorly understood. The detailed composition and charge-state data provided by ACE will be used to:

- Isolate the dominant coronal formation processes by comparing a broad range of coronal and photospheric abundances.
- Study plasma conditions at the source of solar wind and solar energetic particles by measuring and comparing the charge states of these two populations.
- Study solar wind acceleration processes and any charge or mass-dependent fractionation in various types of solar wind flows.

4. Particle Acceleration and Transport in Nature

Particle acceleration is ubiquitous in nature and is one of the fundamental problems of space plasma astrophysics. The unique data set obtained by ACE measurements will be used to:

- Make direct measurements of charge and/or mass-dependent fractionation during solar flare and interplanetary acceleration.
- Constrain solar flare and interplanetary acceleration models with charge, mass and spectral data spanning up to five decades in energy.
- Test theoretical models for ^{3}He-rich flares and solar gamma ray events.

2.3.5 <u>Minimum Mission Success Criteria</u>

There are ten measurements that constitute the ACE mission success criteria for science. ACE will be declared successful if at least seven of the ten measurements are achieved. Since the mission officially started on February 1, 1998, it is not yet possible to assess how well ACE will do these measurements. Several require at least a one year period of observation and some require two years. The criteria are:

a. Composition of heavy nuclei in both the bulk solar wind and in several high speed streams.

b. Composition of coronal mass ejection events over a one year period.

c. Solar wind pick-up ions over a one year period.

d. Composition of heavy nuclei in co-rotating interaction region events over a one year period.

e. Composition of heavy nuclei in energetic storm particle events over a one year period.

f. Composition of heavy nuclei in ten solar particle events, including three large events.

g. Composition of heavy nuclei in small impulsive solar flares over a one year period.

h. Isotopic composition of anomalous cosmic rays.

I. Abundances of radioactive clock isotopes in galactic cosmic rays.

j. Isotopic composition of the "primary" galactic cosmic ray elements from carbon to zinc.

2.4 SCIENCE PAYLOAD

There are nine scientific instruments on ACE to make the comprehensive and coordinated in situ measurements to accomplish the scientific objectives. These instruments are:

High Resolution Spectrometers

CRIS	Cosmic Ray Isotope Spectrometer
SIS	Solar Isotope Spectrometer
ULEIS	Ultra Low Energy Isotope Spectrometer
SEPICA	Solar Energetic Particle Ionic Charge Analyzer
SWICS	Solar Wind Ion Composition Spectrometer
SWIMS	Solar Wind Ion Mass Spectrometer

Monitoring Instruments

EPAM	Electron, Proton and Alpha Monitor
SWEPAM	Solar Wind Electron, Proton and Alpha Monitor
MAG	Magnetic Field Monitor

Table 2.4-1 shows the relationship of the instruments to the scientific objectives (section 2.3.4), and Table 2.4-2 indicates the relationship of the instruments to the minimum success criteria (section 2.3.5). Section 2.4.1 briefly describes the instruments (a detailed discussion of the ACE science payload is provided in section 7.1). Section 2.4.2 discusses the technology developments in the nine mission instruments.

In addition to the nine mission instruments, there are two secondary payloads which are independent of the ACE mission. They are noted below in section 2.4.3.

2.4.1 Mission Instruments

1. Cosmic Ray Isotope Spectrometer (CRIS)

CRIS measures the elemental and isotopic composition of galactic cosmic rays over the energy range from approximately 100 to 600 MeV/nucleon, with an element range from helium to zinc and with a collecting power more than 50 times greater than previous instruments of its kind. It determines the nuclear charge, mass and kinetic energy of incident cosmic rays. CRIS is a new instrument developed by CIT, GSFC, JPL and Washington University (St. Louis).

2. Solar Isotope Spectrometer (SIS)

SIS measures the elemental and isotropic composition of energetic nuclei from helium to zinc in the energy range of about 10 to 100 MeV/nucleon. The energy range includes transient fluxes of energetic nuclei accelerated in large solar particle events, as well as anomalous cosmic rays and low-energy galactic cosmic rays. It computes the nuclear charge, mass and kinetic energy of incident particle. The collecting power of SIS is about 100 times greater than that of previous solar particle isotope spectrometers. SIS is a new instrument developed by CIT, GSFC and JPL.

Table 2.4-1

ACE Science Objectives

	CRIS	SIS	ULEIS	SEPICA	SWIMS	SWICS	EPAM		MAG
Composition of Matter									
Generate table of solar isotopic abundances		P	P	P	P	C	C	C	
Determine coronal composition		P	P	P	P	P	C	P	
Compare cosmic ray and solar isotope pattern	P	P	P	P	P	C	C	C	
Measure interstellar/interplanetary pickup ions				C	C	P		C	C
Determine anomalous cosmic ray composition	P	P	P	C			C		
Origin/Evolution of the Elements									
Indentify solar/meteoritic composition differences		P	P	P	P	P	C	C	
Solar particle contributions to moon/meteorites/planets		P	P	C	P	P	C	C	
Identify cosmic ray nucleosynthesis processes	P	P							
Determine age of cosmic ray source material	P	P							
Search for evidence of galactic evolution	P	P	P	C	P	C			
Coronal Formation/Solar Wind Acceleration									
Isolate coronal formation processes		P	P	P	P	P	C	P	
Study solar plasma conditions				P	P	P		P	C
Study solar wind acceleration/fractionation					P	P		P	P
Particle Acceleration/Transport									
Fractionation in solar flare/interplanetary acceleration		P	P	P	P	P	C	P	C
Constrain particle acceleration models	P	P	P	P	P	P	P	P	P
Test 3He-rich and gamma ray flare models	C	P	P	P			P		C
Measure cosmic ray acceleration/transport time scales	P	P							
Test anomalous cosmic ray origin	P	P	P	P					

P = Primary Measurements
C = Contributing Measurements

Table 2.4-2

Instrument Requirements for Each Measurement to Meet Minimum Success Criteria

Instr ---> Meas	CRIS	SIS	ULEIS	SEPICA	SWIMS	SWICS	EPAM	SWEPAM	MAG
a)					1	1		+	+
b)					1	1		+	+
c)					+	1		+	+
d)		+	2	2	2	+	+	+	+
e)		+	2	2	2	+	+	+	+
f)	+	2	2	2	+	+	+	+	+
g)	+	+	1	1	+	+	+	+	+
h)	+	1	1						
i)	1	1							
J)	1	1							

1 - One of these instruments must work for a successful measurement

2 - Two of these instruments must work for a successful measurement

+ - Significant supporting measurement

3. Ultra Low Energy Isotope Spectrometer (ULEIS)

ULEIS measures the mass and kinetic energy of nuclei from hydrogen to nickel from about 45 keV/nucleon to several MeV/nucleon. Studies of ultra-heavy particles, those heavier than iron, are also performed in a more limited energy range near 0.5 MeV/nucleon. The energy range covered by ULEIS includes solar energetic particles, particles accelerated by interplanetary shocks and low-energy anomalous cosmic rays. ULEIS has a collecting power for solar flare isotopes more than 10 times greater than any previous instrument. It provides more than 1,000 times improvement in detection for the study of co-rotating interaction region events, and is a significant advance in the research of anomalous cosmic ray isotopes. ULEIS is a new instrument developed by the University of Maryland and JHU/APL.

4. Solar Energetic Particle Ionic Charge Analyzer (SEPICA)

SEPICA measures the ionic charge state, elemental composition and energy spectra of energetic solar ions. Measurements of solar energetic particle charge states provide information on the temperature of the source plasma, as well as possible charge-to-mass dependent acceleration processes. SEPICA also measures anomalous cosmic ray charge states. It covers a range from approximately 0.5 to 5 MeV/charge for charge state composition, and up to 10 MeV/nucleon for element analysis. SEPICA achieves improvements of a factor of 3 in charge resolution and a factor of 20 in collecting power over previous instruments. SEPICA is a new instrument developed by the University of New Hampshire and the Max Planck Institute for Extraterrestrial Physics, Germany.

5. Solar Wind Ion Composition Spectrometer (SWICS)

SWICS determines the elemental and ionic charge state composition of all major solar wind ions from hydrogen to iron. It also measures the temperature and speeds of these ions, covering solar wind speeds ranging from 145 km/s (protons) to 1,532 km/s (iron). These data tell scientists not only about the nature of the solar wind, but also of solar flares, energetic storm particles, co-rotating interaction regions and pick-up ions. EPAM was built by the University of Maryland and the University of Bern, Switzerland, and is the same as an instrument fully developed and tested for the Ulysses mission. A flight spare from that mission was used for ACE.

6. Solar Wind Ion Mass Spectrometer (SWIMS)

SWIMS has excellent mass resolution (M/dM > 100) and measures solar wind composition for all solar wind conditions. It determines, every few minutes, the quantities of most of the elements and a wide range of isotopes in the solar wind. The abundances of rare isotopes are determined every few hours, providing information crucial to the understanding of pick-up ions and anomalous cosmic rays. SWIMS will extend knowledge of solar wind composition to a wide range of additional elements and isotopes. SWIMS measures speeds, depending on particle mass, ranging from about 200 to 1,500 km/s for helium and from 200 to 500 km/s for iron. SWIMS was built by the University of Maryland and the University of Bern, Switzerland. It is a copy of portions of the CELIAS experiment from the SOHO mission, modified slightly to optimize it for ACE.

7. Electron, Proton and Alpha Monitor (EPAM)

EPAM characterizes the dynamic behavior of electrons and ions with energies about .03 to 5 MeV/nucleon that are accelerated by impulsive solar flares and by interplanetary shocks associated with coronal mass ejections and co-rotating interaction regions. EPAM measures the composition of elements up through iron. EPAM was built by JHU/APL. It is the flight spare of the HI-SCALE instrument from the Ulysses mission.

8. Solar Wind Electron, Proton and Alpha Monitor (SWEPAM)

SWEPAM measures the solar wind plasma electron and ion fluxes as functions of direction and energy. These data provide detailed knowledge of the solar wind conditions every minute. Electron and ion measurements are made with separate sensors. The ion sensor measures particle energies between about 0.26 to 35 KeV, and the electron sensor's energy range is between 1 and 900 eV. SWEPAM was built by the Los Alamos National Laboratory from the spare solar wind electron and ion analyzers from the Ulysses mission, with selective modifications and improvements.

9. Magnetic Field Monitor (MAG)

Precise examination of the interplanetary and solar magnetic fields and their dynamics provides essential supporting information for the other ACE instruments. MAG consists of two magnetometers that detect and measure the magnetic fields in the vicinity of the spacecraft. MAG measures the strength and direction of the interplanetary magnetic field 30 times per second and can calculate any pattern of variations in it. MAG was built by the University of Delaware/Bartol Research Institute and GSFC, and is a flight spare from the WIND mission.

10. S3DPU

As part of the payload accommodation to the ACE spacecraft, a data processor design was adapted from the Solar and Heliospheric Observatory (SOHO), which then served three of the nine ACE instruments: SEPICA, SWIMS, and SWICS. This unit came to be known as the S3DPU. The other six instruments had data processors or micro-controllers of their own, and interfaced directly with the spacecraft Command and Data Handling system.

2.4.2 Technology Developments

Four large ACE science instruments involved new inventions intended to achieve a giant leap forward in collecting power over previous designs flown on earlier missions. One such invention, the Scintillating Optical Fiber Trajectory (SOFT) hodoscope at the front end of the CRIS, involved adapting a successful technique heretofore used on stratospheric balloon flights, but never before in a spaceflight instrument. The large-area "stack detectors" at the back-end of CRIS, (which are needed for achieving the sought-after increase in collecting power), necessitated the development of custom VLSI chips which were then hybridized to further conserve mass and volume.

Another new ACE instrument, SIS, involved the design and use of a different type of custom (analog) CMOS VLSI circuit to read out the electronic charge deposited by penetrating energetic particles, and then preprocess these signals from the unprecedented number of strips in its front-end matrix detectors. At the back end of SIS, read out of its stack detectors was achieved using the same type of hybridized VLSI circuits developed for CRIS. The Lithium-drifted stack detectors used in CRIS, and the ion-implant stack detectors used in SIS, both overcame development problems which arose in part because of their unusually large collecting area.

Another new instrument, ULEIS, needed a carefully engineered system design along with yet more custom VLSI in order to achieve the necessary time of flight accuracy allowing it to meet its requirement for unprecedented mass resolution between particle species.

A fourth new instrument, SEPICA, required front end collimators that involved fine machining to especially tight boresight requirements. The implementation of these requirements resulted in the development of new technology that is now being commercialized. It also necessitated designing an isobutane gas flow control system operating at near vacuum pressures, but which could be run and tested at one atmosphere.

Even one of the “design-to-print” instruments, SWIMS, turned out to be more difficult than anticipated. This occurred when it was learned that its predecessor instrument on the WIND mission was limited by the Helium background in space. An unplanned development effort ensued and significant redesign resulted before the modified SWIMS instrument was finished.

Compared to SWIMS, the other four heritage instruments on ACE underwent relatively minor internal upgrades, along with the significant interface changes needed to adapt them to the ACE spacecraft.

2.4.3 Secondary Payloads

ACE carries two payloads that are not necessary to accomplishing the project mission. Both were funded outside the ACE budget.

1. Real-Time Solar Wind (RTSW)

Geomagnetic storms are a natural hazard that the National Oceanic and Atmospheric Administration (NOAA) forecasts for the benefit of the public, as it does hurricanes and tornadoes. The location of ACE enables it to provide about one-hour advance warning of impending major geomagnetic activity.

A subset of data is sent from SWEPAM, EPAM, MAG and SIS during the time that ACE is not transmitting its full telemetry. For about 21 of 24 hours per day, ACE broadcasts the real-time solar wind data. The data are received by the ground stations around the world and sent directly to NOAA. The ground system developed by NOAA receives the data broadcast by ACE in real time. This system also includes the Communications Research Laboratory in Japan, the US Air Force Satellite Control Network, and NASA ground stations. During the three hours per day that NASA ground stations are receiving full ACE telemetry, NOAA receives a copy of the data. This gives NOAA 24 hours per day forecasting capability. NOAA processes the data at its Space Environment Center in Boulder, Colorado, which issues alerts of any potential geomagnetic problems.

This is the first real-time solar wind monitor ever flown.

2. Spacecraft Loads and Acoustics Measurement (SLAM)

The SLAM instrumentation gathered payload launch environmental data only during the first five minutes of powered flight until the Delta fairing separated. SLAM was a GSFC in-house project within the Engineering Directorate. The SLAM system was a microprocessor-based system designed to determine the structural accelerations and vibroacoustic levels encountered by the ACE spacecraft during those first five minutes of launch. SLAM was turned on approximately 30 seconds prior to launch, remained active through all launch events, and was turned off prior to spacecraft-launch vehicle separation. SLAM read analog signals produced by transducers, digitized the signals, and passed them to an S-band FM transmitter for downlink. The output from the transmitter was sent by RF coaxial cable to an omni antenna mounted to the launch vehicle fairing.

2.5 SPACECRAFT

This section presents a summary description of the ACE spacecraft. Detail is provided later in section 7.2.

The ACE spacecraft consists of a two-deck irregular octagon, about 1.6 meters across and about 1 meter high. It spins about its axis so that one end always point toward the sun and the other toward the Earth. It has a spin rate of 5 revolutions per minute (rpm). The spacecraft subsystems are: mechanical, thermal control, command and data handling, radio frequency (RF) communications, attitude determination and control, propulsion, and power. Figure 2.5-1 shows the spacecraft's deployed configuration.

To meet observing requirements and to simplify access to the instruments and spacecraft subsystems, all components except the propulsion system are mounted on the external surfaces of the spacecraft body. Six of the instruments are mounted on the top (sunward facing) deck and two are mounted on the sides. The instruments are positioned to keep their various fields of view clear of each other while maintaining weight balance around the spacecraft. The two magnetometer sensors are on booms attached to two of the four solar panels. With the solar panels and magnetometer booms extended, the ACE spacecraft has a wingspan of about 8.3 meters. The four arrays of solar cells provide sufficient power to allow operations to continue for at least five years.

The spacecraft contains redundant equipment for collecting and storing data, and transmitting the data back to Earth. Data are transmitted via a highly directional parabolic dish antenna mounted on the bottom (Earth facing) deck of the spacecraft. Four other broad-beam antennas, capable of transmitting at lower rates, are also available if needed. Twenty-four hours of science and housekeeping data, or about one gigabit, recorded on one of two solid-state recorders are transmitted to Earth in a single three to four hour telemetry pass each day. Spacecraft attitude is provided by a star tracker/scanner and digital sun sensors. Table 2.5-1 gives the weight, power and data rate breakdown for the observatory and the fields of view for the instruments. The fact that ACE had sufficient weight and power margins enabled these resources to be used to solve problems, in lieu of the use of additional time and money.

The ACE spacecraft was developed by APL to meet the mission requirements in the simplest and least costly way possible. It was a high heritage spacecraft, based on the Active Magnetospheric Particle Tracer Explorer (AMPTE) as well as other spacecraft. Most of the equipment in the subsystems was standard and procured from the specializing vendors. APL designed and built the command and data handling processors and the power system controllers. There are two advanced technologies used on ACE: new high capacity solid state recorders and a new star scanner. The recorders have a capacity of greater than one gigabit. Since the recorders are used both by ACE and the Near Earth Asteroid Rendezvous (NEAR) mission, and APL was simultaneously building the ACE and NEAR spacecraft, the development of the recorders was a shared effort by the two missions. The star scanner is similar to the star tracker flown on the X-Ray Timing Explorer, but required design modifications to operate in a star scanning mode.

Figure 2.5-1

ACE OBSERVATORY

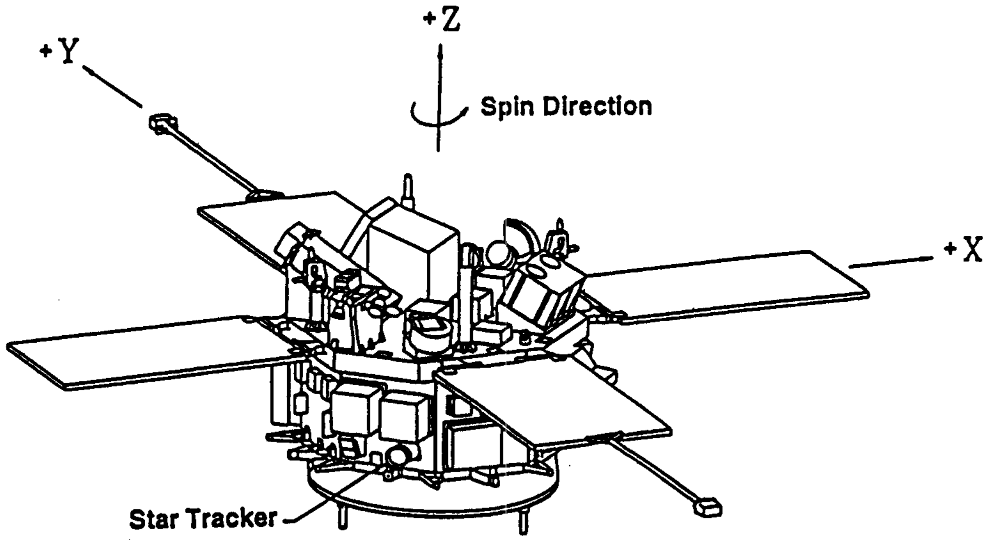


Table 2.5-1: Summary of Weight, Power, Data Rate and Field of View

	Weight (kg.)	Peak Power (watts)	Nominal Power (watts)	Data Rate (bps)	Field of View
Instruments:					
CRIS	31.6	16.6	12.2	464	90 x 90 deg.
SIS	22.4	22.4	17.5	1,992	104 deg. conical
ULEIS	21.9	15.1	14.6	1,000	24 x 20 deg.
SEPICA	38.3	17.5	16.5	608	61.5 x 17.5 deg.
SWICS	6.0	6.1	5.0	504	82 x 10 deg.
SWIMS	8.6	7.8	6.8	512	62 x 62 deg.
EPAM	12.8	4.2	3.8	168	60 deg. conical
SWEPAM	6.8	6.1	5.8	1,000	160 x 30 deg. (E) 80 x 10 deg. (I)
MAG	4.1	2.4	1.8	304	
S3DPU	3.9	3.5	2.5		
Subtotal	156.4	101.7	86.5	6,552	
Spacecraft	586.0	248.0	177.0	392	
Other	14.1	34.3	21.5		
TOTAL	756.5	384.0	285.0	6,944	
Maximum Allowable Weight	785				
End of Life Power		>440			

2.6 GROUND SYSTEM

To increase ground system efficiency and to help the ACE project stay within its cost cap, a common ground system approach was utilized for integration and test, mission operations, and data processing and distribution. The three major subsystems of the ACE ground system are a) the spacecraft I&T ground support system, b) the Mission Operations Center (MOC), and c) the ACE Science Center (ASC). These have many common functions, and the ACE common system architecture facilitates sharing of software, databases and testing procedures. This approach is unique for larger missions.

Each of the three ground subsystems utilize the same computational platforms (HP 715 and HP 748), and as much as possible use a common set of software modules. To do this the performance requirements for instrument checkout and data verification had to be understood far earlier than is usually the case, and procedures had to be developed on a pace that matched the spacecraft development.

The ground support equipment for the integration and test of the spacecraft and the instruments is known as the Integration and Test Operations Control Center (ITOCC). Spacecraft operations and level zero processing of payload data are performed in the MOC at GSFC. The data is then transferred to the ASC at CIT. The ASC is responsible for initial processing of the data and for making the data accessible to the co-investigators and guest investigators located at their home institutions. Instrument teams are located at ACE Science Analysis Remote Sites (ASARS) where they can review and analyze data and evaluate instrument performance, and, as necessary, forward instrument commands to the ACE MOC for transmission to the observatory. In practice, three teams (including contractors) at APL, GSFC and CIT had to work closely together to make this common ground system approach work.

An early Phase B analysis led to the conclusion that the Transportable Payload Operations Control Center (TPOCC), a Unix based system which has been a foundation system for the past several years for most spacecraft control centers at GSFC, could provide much of the needed functionality in the ITOCC and the ASC, as well as the MOC. A generic ACE system was adapted from the XTE TPOCC to include all functionality common to the ACE MOC, ITOCC, and ASC, since many of the XTE unique capabilities were useful for ACE. Separate copies of the generic system were then augmented individually with unique capabilities needed in the three subsystems. Most of the MOC unique capabilities are to be transferred to the ASC, making it possible to extend the duration of the ACE mission by operating it at low cost from the ASC.

A single database was used for spacecraft development and testing, and is being used for spacecraft and instrument operations. Using the TPOCC System Test and Operations Language (TSTOL) for all procedures and displays allowed the instrument teams to use common elements throughout all phases of instrument development, test, checkout and monitoring during integration and testing, and flight operations.

Members of the Flight Operations Team (FOT) were included in the design and development from the early phases of the mission. This enabled a smooth transition between spacecraft design and ground operations and avoided the frequently encountered situation where ground operations are made unnecessarily complicated by spacecraft design decisions. The concept of the integrated team approach across spacecraft design, integration and test, and operations has worked successfully on smaller spacecraft, but ACE is the first time NASA has attempted this approach on a larger program. The approach worked very well and is recommended for future missions.

The multiple uses of the generic ground system has saved millions of dollars. Although the generic TPOCC task required more lines of code than needed for the MOC alone, it was substantially less than the lines of code required for the three separate systems. A similar savings in maintenance will be realized, since enhancement or correction in one system will enhance or correct all three systems. The approach led to the deletion of the software training simulator for the FOT. Instead, the FOT received high fidelity training with the spacecraft at APL, thus reducing risk to mission operations. The one database utilized for all three systems eliminated time consuming and costly translations from one database to another while increasing the reliability of the operations database.

3.0 BOTTOM LINE RESULTS

This section examines how well the ACE project performed in three areas: a) on-orbit performance, the primary measure of success, b) cost performance in terms of dollars and Civil Service work force, and c) schedule performance. In summary, ACE is performing very well on orbit, the project was significantly underbudget, and launch was on schedule. Also, ACE did well in comparison to historical trends.

3.1 ON-ORBIT PERFORMANCE

At this point about four months after ACE was declared operational in its mission orbit, **the satellite is performing satisfactorily and in a manner to meet its mission success criteria**. There are no problems with the spacecraft bus. There is a problem with an isobutane regulator on SEPICA, which reduces its science data.

3.2 COST PERFORMANCE

ACE cost and Civil Service work year performance is analyzed in two ways: a) actual vs. planned and b) actual vs. historical trends.

3.2.1 Actual Cost vs. Planned

The agreement between GSFC and OSS was that the total cost of ACE for Phase C/D through launch plus 30 days of checkout would not exceed $141.1 million. The final project cost was $106.8 million, **a $34.3 million underrun for a 24 percent savings**. Table 3.2-1 provides the cost breakdown for the starting baseline Program Operation Plan (POP) 93-1 and the final POP 98-1. Even after excluding the planned contingency, the final costs of all the program elements were less than planned.

Table 3.2-1 also shows the planned and final direct Civil Service work years, and again there was a significant **underrun of 25.2 work years for a 25 percent savings**.

Table 3.2-1: ACE Cost and Civil Service Work Years
(Planned and Actual; Cost in Real Year Millions)

	Planned POP 93-1 (M$)	**Actual** POP 98-1 (M$)
Project Management	6.8	5.1
Spacecraft	52.1	47.0
Science Payload	53.4	50.2
Ground Systems	2.6	1.5
Performance Assurance	2.2	1.2
Flight Operations	2.2	1.8
Subtotal	119.3	106.8
Contingency	21.8	0
Total	141.1	106.8
C. S. Direct Work Years	100.3	75.1

3.2.2 Comparison to Historical Experience

This section breaks down the costs in a way to compare the planned and actual costs with the GSFC Resource Analysis Office (RAO) cost model estimates done at the same time as the project planned costs. The RAO models are based on a database of completed projects and, thus, represent historical trends. Table 3.2-2 presents these comparisons, including total direct Civil Service work years.

Table 3.2-2: Cost Comparison to Historical Experience
(Cost in Real Year Millions)

	Planned POP 93-1	Actual POP 98-1	Historical (RAO Model) 9/93	Difference (Actual - Historical)	Percent Difference
Instruments			45.3		
Credit for prior H/W			-2.3		
Total Instruments	44.0	42.9	43.0	-0.1	-0.2%
Spacecraft Subsystems	31.3	28.4	26.2	+2.2	+8.4%
Ground Systems					
MOC	2.6	1.5			
ASC	2.8	2.0			
Total Ground Sys.	5.4	3.5	2.5	+1.0	+40.0%
Flight Operations	2.2	1.8	2.1		
MPS (HQ Tax)	3.2	3.1	3.2		
Total w/o Wraparound	86.1	79.7	77.0	+2.7	+3.5%
Wraparound					
Project Mgt. less MPS	3.6	2.0			
Payload Management	6.6	5.3			
S/C Mgt. and I&T	20.8	18.6			
Performance Assur.	2.2	1.2			
Total Wraparound	33.2	27.1	33.1	-6.0	-18.1
Total less Contingency	119.3*	106.8	110.1	-3.3	-3.0%
Contingency	21.8	incl.	27.5		
Total	141.1	106.8	137.6	-30.8	-22.4%
Direct C. S. Work Years	100.3	75.1	114	-38.9	-34.1%

* Contains about $7 million of schedule reserve.

Interesting points emerge from the comparisons:

- The total RAO estimate was $137.6 million. Thus, the actual cost of $106.8 million is **$30.8 million or 22 percent less than the historical expectation**.

- Since the RAO estimate and the project planned estimate are within 3 percent of each other, this result is similar to the comparison above of actual to planned. The project planned estimate reflected historical expectation; ACE results did not.

- Excluding contingency, the basic RAO estimate ($110.1 million) and actual cost are only 3 percent apart. The RAO estimates for individual mission elements (e.g., instruments) and the actuals are very close, with the area of **wraparound costs having the largest difference**. Wraparound costs cover the functional activities of management, systems engineering, performance assurance, and integration and test that bring the people, hardware, software, and other project elements together to achieve an integrated operationally ready space mission. The actual work content for instruments, spacecraft hardware, ground systems, and preparations for flight operations was very close to historical expectation.

- Again, the project planned estimates (excluding contingency) are quite close to historical expectations. In fact, the planned total without contingency ($119.3 million) contained about $7 million of schedule reserve. Excluding this reserve brings the basic planned estimate to a 5 percent difference from the actual cost and a 2 percent difference from the RAO estimate. As with the RAO estimate, the largest area of difference is wraparound.

- **The major difference between the bottom line actual and historically expected costs is the expected use of contingency**. In the end, the project did not use the anticipated amount of contingency and schedule reserve.

- The actual Civil Service total of 75.1 work years is **39 work years or 34 percent less than historical expectation**.

The primary conclusion is that the ACE project performed significantly better than historical expectations. The analysis in this section strongly indicates that the ACE project was managed differently than most historical projects. The management took the fixed price mandate very seriously. First, the contingency and schedule reserve were “banked” early in the project with the desire to manage without them. Second, the fixed price concept was used throughout all elements of the project. Third, schedule was maintained tightly, which provided a control of cost. Similar results and patterns were seen in the XTE project. It appears that the GSFC Explorers Project Office, in conjunction with Headquarters OSS, have constructed a more efficient mode of project management. One that fits well with the NASA concept of faster, better, cheaper.

3.3 SCHEDULE PERFORMANCE

3.3.1 Actual Schedule vs. Planned

The agreement between Headquarters and GSFC stated that the launch would be no later than December 1997. At the very beginning of Phase C/D the Project Office established a launch date for the end of August, 1997. The ACE satellite was launched August 25, 1997. **There was no deviation between the planned and actual launch dates, and the not-to-exceed launch date was beat by four months**.

3.3.2 Actual Schedule vs. Historical Experience

The Authority to Proceed (ATP) was the beginning of FY1994, or October 1, 1993. Launch was August 25, 1997, which gives a project duration of 46.8 months from ATP to launch. This was 3 months longer than the average project duration for 8 spacecraft projects in the RAO database with total satellite dry weights from 800 to 1,500 pounds. ACE has a total dry weight of 1,226 pounds. The range for project duration for these 8 spacecraft is 34 to 54 months, and the ACE project duration is within 1/2 standard deviation of the database average. This indicates that while the ACE figure is above the historical average, it is close enough to conclude that the **ACE schedule is consistent with historical experience and was not any faster**.

4.0 UNIQUE AND INNOVATIVE ASPECTS OF ACE

This section summarizes key overall characteristics of ACE that are unique and innovative. More detailed innovative elements, such as in the individual instruments and spacecraft subsystems, are discussed in section 7.0 below, and some have been mentioned above in section 2.0.

1. Extensive Phase B

Because of Explorer Program funding priority shifts, the planned execution phase start for ACE was postponed. This allowed ACE to have an extended Phase B of around three years. Some Phase B activities continued after Phase C/D started. Phase B also was well funded.

The exceptionally thorough nature of the Phase B effort by an experienced science team and APL spacecraft developer got the mission off to a good start. For example, that effort included demonstration of a new VLSI approach that greatly simplified CRIS and SIS. In addition, Phase B helped to refine the scope of all instruments and the spacecraft.

Despite the demonstrated success and savings (see section 3.0) of the ACE mission, ACE may be the last of the Explorers to enjoy the advantages of such an extensive Phase B study. Most of the currently planned missions will be limited to Phase B studies in the order of a few months.

2. Capped But Adequate Budget

ACE was proposed as a straightforward, low-cost mission in the mid-1980s. When ACE finally was selected to proceed to Phase B, the NASA environment was rapidly changing (1991 time frame). ACE followed the innovative pattern set by XTE by having a cost cap agreement between the OSS and the GSFC Project Office. This allowed a much stronger stand on requirements growth than what was normal in that time period. This played a large role in maintaining the original low-cost nature of ACE.

OSS committed up front to provide the Phase C/D funding as planned for each fiscal year. The establishment up front of adequate resources (including contingency) was crucial to the success of ACE. That provided the flexibility to make development tradeoffs and work-arounds which made it possible to meet a launch date that did not slip.

3. Descope Plan

As part of the agreement between OSS and the Project Office to contain the total project cost, there also was a scope reduction agreement. Provided that the Level 1 Science Requirements were preserved and that full use of contingencies were used, the ACE project, in consultation with the Project Scientist, could pursue scope reduction as a means to control cost. As it turned out, no major descoping was needed. However, having this plan in place provided a useful backup.

4. Principal Investigator (PI) Mode

ACE was the first full-sized Explorer mission to employ the PI mode, a new way of doing business. ACE, despite having nine separate instruments, had only one PI. The responsibility for the instruments was delegated to CIT (the PI institution), with minimal oversight from Goddard. For its time, the amount of responsibility turned over to the mission PI organization (and, incidentally, to the spacecraft contractor) was innovative.

The PI was consulted for all significant decisions that the Project Office had to make regarding the instruments. In fact, the Mission Manager and the PI communicated on a weekly basis from Phase B through the completion of Phase C/D so that the PI was always aware of what was happening and was always in the major decision making loop.

5. Contracting Arrangements

The ACE contracting arrangements were unique. The Project Office had separate contracts with APL, CIT and the GSFC Mission Operations and Data Systems Directorate. APL provided the spacecraft, system integration and test, and launch support. The Mission Operations and Data Systems Directorate provided the ground system, the Mission Operations Center and Flight Dynamics support. CIT provided the science payload and the ACE Science Center.

The contract with CIT provided for the payload (contract authority for six and one half of the instruments), the Payload Management Office (PMO), and the ACE Science Center (ASC). Full authority to subcontract with the investigator institutions and to direct them was implicit. Instruments built by the Los Alamos National Laboratory and APL (EPAM and one half of ULEIS) were funded directly by GSFC; however CIT was delegated total management responsibility and authority. In monitoring the CIT contract the ACE Project Office had an oversight role. In executing the contract, however, the Project Office, at CIT's request, agreed to become a part of the PMO team and supported the CIT efforts for the East Coast instruments. In addition the Project Office provided scheduler/planner support to CIT for all instruments. This effort resulted in increased efficiency in resolving problems, faster responses in replanning, and development of mutual trust between organizations.

6. Common Ground System

ACE was the first major NASA mission to adopt the same core hardware and software system for the I&T ground system, mission operations ground system and science data center ground system. This eliminated much of the effort previously required to make three separate ground systems, made the system far more efficient, saved development dollars, and provided better trained operations personnel.

7. Instrument Integration With Spacecraft

The handling and scheduling of the instrument integration was innovative and successful. The instruments were all integrated prior to thermal vacuum testing and all but one was removed for calibration and/or repair, and shipped to KSC independent of the spacecraft. At the KSC processing facility, the instruments were re-integrated and tested. While this approach was cumbersome, it allowed for late delivery of instruments with minimal impact to the overall schedule.

8. NASA Classification

Classification was Class C for the spacecraft bus, but was relaxed to Class D for the highly redundant instrument suite. This allowed CIT/JPL flexibility to handle the delicate SR&QA tasks in the University environment, while preserving a more normal approach between GSFC and APL for the spacecraft. This arrangement forced a clear delegation of responsibility and a concentration on the higher priorities, rather than trying to tightly control everything from one place. Cooperation resulted, and it worked quite well.

9. Secondary Payloads

The near-real-time reporting of space weather using a subset of data from four scientific instruments is both innovative and unique. ACE carried the first Real Time Solar Wind monitor, ever, for NOAA/SEC. Also, the incorporation of the SLAM package represented an important and unique first step in gathering payload launch environmental data.

10. Management Approaches

The following unique management approaches were used by the ACE Project Office at GSFC:

- Separate contracts with APL and CIT. Both were effectively managed by the respective institutions.
- Use of contractor operating and R&QA systems wherever possible. Not insisting that they do it Goddard's way.
- Different Product Assurance requirements for instruments and spacecraft. (Placed primary responsibility for instrument quality in the hands of CIT and Co-I teams. Provided for less than Class C requirements but preserved overall Class C mission.)
- The GSFC Instrument Manager played two major roles. The first was providing oversight, from a Project point of view, of the CIT effort. The second was providing management support to CIT, as a member of the CIT/PMO management team, in looking after the East Coast instruments. This dual role was unique.
- Full and open communications with all parties minimized friction and led to establishing trust between groups.
- Recognition that people sometimes make mistakes. When they did the problem was fixed without penalizing the individual. This led to a greater willingness for all team members to provide full and factual information (even if it hurt).

Using CIT to manage the development of the instruments and the ASC was a unique and highly successful management approach. What made possible the successful science payload management of ACE by a small PI institution such as CIT was easy access to the much larger technical and managerial resources of the JPL. Without those, CIT would have been hard pressed to provide the staff necessary to manage the development of an ACE science payload. JPL was in effect used as a cafeteria from which technical and management services were selected on an as-needed basis.

The Project Office management treated ACE completely as a team effort, and trusted the science team to complete their tasks without direct oversight more than is commonly done. This helped instill a sense of team spirit. Yet the Project Office was also willing to push the science team to accept help in problem areas as early as these problems were identified. This was effective in keeping the project on time and underbudget.

5.0 RISK AND RISK MITIGATION

The major risks were those things that were schedule driven. There were adequate weight and power margins and these were used to solve problems rather than initiate lengthy and costly redesigns. In general, the real risks were in the development of key items for the four new instruments.

1. Instruments

The major risks were in developing four new instruments of unprecedented collecting power within the available time and resource constraints. An extended and well funded Phase B significantly helped reduce, but not eliminate, the instrument development risks. Defining a set of mission objectives in Phase B which required that just seven out of ten be met in order for the mission to be successful was an important mitigation. Thorough and ongoing reviews of development plans made it possible to avoid overly ambitious undertakings, and to identify descope options, should they become necessary. Such reviews were conducted not only by the CIT Payload Management Office and the Goddard Project Office, but also by independent review boards composed of technical experts paid for by the CIT PMO. Helpful suggestions often arose from members of this independent board whose membership was tailored to the occasion. A core group of standing members provided continuity across the science payload, as well as an ongoing cross-pollination of lessons learned among the various instrument development groups.

In particular, during Phase B, the major instrument risks were identified as being the detectors for CRIS and SIS; the custom very large scale integrated circuits (VLSI) for the SIS matrix detector readout; the custom VLSI/hybrids developed for CRIS, SIS and ULEIS; the scintillating optical fiber trajectory (SOFT) hodoscope for CRIS; the Time of Flight System for ULEIS; the collimators, gas controller and power supplies for SEPICA. Adequate funding and planning were provided to mitigate these risks. In most cases suitable alternatives were identified even though their use would have resulted in reduced science or a descoped instrument.

This risk mitigation approach paid off. Each of the instruments was able to complete its development properly and all of the high-risk items were flown. For new state-of-the-art instruments keeping instrument development up front and funded is an important part of a successful mission. In the case of ACE, it was the Phase B that allowed the instruments time and money to mature to the point where they could be successfully completed in Phase C/D, although, even so, it was close.

2. Spacecraft

The spacecraft did not have major risks, although some risks were identified:

The primary risk for the AD&C subsystem was a schedule risk, specifically the delivery of the star scanner. The reason for the risk was that the ACE star scanner is the first to fly using the Time Delay Integration technique, and the vendor did not have an adequate number of people assigned to the software development effort. The scanner was delivered nine months behind schedule.

To mitigate any risk in communicating with the Observatory, the design incorporated a redundant RF system and a redundant command and data handling system.

In terms of the flight software, the primary risk was that all of the necessary tools were not available at the start of the program. These tools were developed within APL, and there was limited staff who could do this. This risk was mitigated by the fact that there was another program that had several sub-systems that needed the same tools, so resources were shared.

3. Ground System

Schedule risk was the primary risk concern for the ground system. The ground system team needed to provide a system to support the I&T of the ACE observatory, and I&T was in the critical path to launch. If this commitment to the APL I&T team was not met, the launch would have been delayed. At the same time the ground system team needed to develop a similar, but not identical system to support flight operation.

Risk was reduced by employing the Flight Operations Team (FOT) as I&T test engineers. In this way the needed training time for the FOT was reduced and carried many of the lessons learned from the I&T system to the flight operations system.

6.0 LESSONS LEARNED

6.1 KEY REASONS FOR SUCCESS

While a large number of things contributed to the success of ACE, there was general agreement among the various ACE personnel on the key reasons:

1. People--good, dedicated, hardworking
2. Cooperation among all the groups--team effort
3. Good up-front planning
4. Adequate Phase B study effort
5. Adequate funding when needed
6. Good management

It was a unanimous opinion that the people on the project and their teamwork made the primary difference. The success of ACE is due to having proper project management and a team composed of the right mix of talented, dedicated and experienced personnel who knew how to anticipate and eliminate problem areas before they developed and to quickly react in an appropriate fashion to solve problems that surfaced. In addition, having a realistic ACE funding budget and schedule, based on detailed carefully prepared estimates, contributed greatly to the success. Also, having reasonable freedom to reallocate funds quickly when needed to solve unexpected problems added greatly to the team's overall performance.

An important reason for the success of the ACE payload development was the use by CIT of a standing review board of technical experts. The particular membership for a given review was tailored to the occasion. Its members contributed useful suggestions and sometimes stayed on after a review to participate in work-arounds during the fast-paced days of instrument development. The effectiveness of these outside experts was made possible by the effort spent beforehand in establishing an environment of openness and trust among the Co-Investigator groups, the CIT Payload Management Office, and the Goddard Project Office. After some initial resistance, all groups contributed to this important spirit of cooperation. For the most part, blame was avoided as a first reaction to problems.

6.2 OTHER LESSONS LEARNED

6.2.1 Project Office Perspective

ACE was a true team effort. Things done in ACE which are worth emulating in the future include:

- Maintaining full and open communications in all directions.
- Being open, honest and frank in discussions with contractors or other supporting groups.
- Giving appropriate credit to the doers of good work.
- Defining project/instrument descope options early in the program.
- Doing what needed to be done and not standing on ceremony.
- Always remembering who the customer is.

The ACE approach was to stick to design where “good is good enough” and not allow hardware changes just for the sake of “improvement”. A reasonable schedule with key milestones was established at the start of Phase C/D, and heavy emphasis was placed on maintaining it. Adequate funding was established along with the ability reallocate these funds as needed.

Synergism between ACE and NEAR worked very well. APL utilized many of the same technical people on both projects. Consequently many of the solutions to problems were solved on “the other project” and never became an “ACE problem”. Contractor work force costs were significantly reduced as they were a shared resource.

Although agreements were reached between the GSFC Project Office, APL and CIT regarding Safety, Reliability and Quality Assurance (SR&QA) requirements, arriving at these agreements was not always easy. In fact, the CIT PMO felt that too much time and effort was spent trying to get to the agreements. The PMO felt that in order to establish who is responsible for what aspect of SR&QA, the responsible SR&QA team members needed to informally talk about the dimensions of needed SR&QA activities. Agreement had to be reached on responsibilities and budget. The agreement was documented and accepted by project management and organizational management. In going through this process everyone knew what corners could be cut during the execution phase.

The PMO also felt that as long as roles were clearly defined, and objectives agreed to, the team, acting efficiently and in collaboration with each other, would be able to reduce documentation time. This was demonstrated.

6.2.2 Payload Perspective

Define the job well up front. Decide on its success criteria. Define an achievable scope and schedule. Allocate sufficient resources along with adequate margins. And provide key technical and management support to the investigators as needed.

ACE worked as well as it did because good people were involved. But as a hand-off of responsibility to the PI institution, it went only part way. The management arrangement was an innovative approach for its time, but resulted in a 3-way nexus of centers, (Goddard, CIT and APL). Each had enough power and independence that when things went wrong, a natural temptation arose for two to point a finger at the third. Fortunately, with good people at the helm, appropriate, reasoned solutions were always worked out. In the future however, handing off more overall development responsibility to the mission PI organization, as is now being done on Small Explorers and MIDEXs, should be even more effective.

In the course of managing the instrument payload, the CIT PMO used the following ABC's of project management:

a) Anticipate what might occur and take preventative steps to avoid problems,
b) Build a good team and endeavor to sustain their interest, and
c) Communicate both upwards, downwards and sideways with project peers.

Specific observations from the perspective of the CIT PMO related to the way in which ACE was managed are as follows:

1. The effort spent in fostering good relations and team work among all participants is well worth it. Good relations and team work are essential to any successful effort where members must trust and depend upon one another, especially a faster, better, cheaper project. To the extent it can be achieved, co-location in conducive facilities helps, as does an adequate travel budget to support frequent fact finding visits and problem solving trips.

2. Despite the PMO's best intentions, the Instrument Functional Requirements Documents (IFRDs) were for the most part not completed until after the hardware was built and evaluated. Co-Is building new instruments were not sure beforehand what performance could be achieved within their operating constraints, and therefore resisted committing to a set of performance specifications until finished. To their credit, Co-I developments ended when "good-enough" instrument performance was achieved.

3. In order to tell the spacecraft integrator how their instrument should be tested at the system level, instrument developers first needed to evaluate how the design turned out, just what performance could be achieved, and how their new invention behaved when operated. Meanwhile, in order to develop test procedures, the system integrator wanted to know the answers well before the developers knew themselves. Unless a better way is found for early scoping of system test procedures, and provisions and resources made available for many last minute changes, this dilemma will only exacerbate schedule conflicts late in the program on the upcoming faster, better, cheaper missions.

4. Managers should be given adequate reserves, and use them early and wisely while there is still time to do some good. Drawing a tight constraints box around a developer while withholding the wherewithal to help him solve his problem will only cause him to dissipate excessive resources working against one brick wall (e.g., mass reduction), while sacrificing another (e.g., schedule). In the end, both the developer and the project lose out.

5. Managers should develop a hierarchical strategy regarding milestones. Decide which are the important ones, and put extra effort into meeting those. Use intermediate milestones to hold people's feet to the fire. But recognize that for understandable reasons some will slip. Such slips necessitate frequent replanning of work flow schedules in order to preserve the important milestones. Throwing project resources at a given problem can often help, provided the developer is persuaded that this help is to his benefit and will not make his job even more difficult, and that he will not be unduly punished for accepting help. The management team needs to continually work at fostering the idea that they are there to help, and not to punish those do not meet every expectation. Openness is needed between participants, not an attitude of pull up the draw bridges and batten down the hatches so that no manager will ever find out what is really going on in the development.

6. In some cases, the rework efforts on new instruments became so extensive that a whole new build was initiated. Therefore, a number of partial engineering models resulted. These turned out to be very useful in helping to achieve the necessary

performance of flight units. The capability to begin a whole new build and continue testing in parallel on engineering models increased the likelihood of overall success. Having the schedule flexibility and adequate financial reserves to do so, made this approach possible and in the end worthwhile.

7. The inevitability of schedule replanning can lead to a non-monotonic "percent complete" estimate. This complicates earned value estimates which means that its absolute value should not be taken too literally. Nonetheless, an adverse trend in an earned value plot may serve as a good indicator of impending trouble. But such a trend needs to be understood in the context of the recent problems that caused it, as well as a realistic assessment of efforts and prospects for overcoming those problems. For example, if all the right things are being done to solve a problem, and no new ones have surfaced, it may be possible to safely discount an adverse trend in earned value.

8. Although a Work Breakdown Structure (WBS) and a companion WBS dictionary was required by the Project Office to establish credible cost estimates, the effort spent on creating these for the science payload was not useful to the PMO.

Other payload lessons learned:

Be careful about specifying acoustic and vibration specifications. Even though vibration and acoustic levels came from the best data available for the launch vehicle and from past GSFC experience, the vibration specification for SIS was much too large (i.e., by a factor of 10 or more), and that caused six months of difficulties and extra costs. The estimate for the specification was way too conservative, and this was a serious problem for SIS.

Development of a database for successful techniques, vendors, guidelines, etc. that can be easily disseminated via the web for review by prospective mission contributors would help avoid unnecessary duplication of effort. Some “wheels” do not need to be re-invented for each mission.

Many work hours could have been saved if the NASA Project Office had someone assigned to help with logistics issues at the beginning of the project. This individual should be well versed in the Department of Transportation requirements for transportation of radioactive sources, hazardous materials, such as isobutane. The SEPICA team had to learn most of the protocols by stumbling over them for months. The efforts put forth for SEPICA were late in the program.

Also, many work hours were spent rewriting safety and environmental test procedures to satisfy GSFC safety and environmental test facility personnel (mostly safety personnel). Skeleton procedures should be developed by GSFC to help incoming “customers” using the facilities to insure that at least minimum facility and GSFC safety requirements are met. Many iterations were submitted and painfully scrutinized before approval was granted to proceed with testing. Much of this could have been done beforehand if these skeletal procedures were given at the beginning of the project to instrumentors planning to perform environmental tests at GSFC.

6.2.3 Spacecraft Perspective

During the extended Phase B, APL purposefully kept costs low while contributing positive effort during this phase by careful planning, focusing on top level requirements and maintaining communications with the instrumentors. ACE provided excellent science at relatively low cost. This was accomplished by laying out the mission requirements early and then sticking to them. The ACE team demonstrated success in designing to cost.

There were several organizations involved in ACE, and all had managers/leads, some more than others. There was a tendency in the beginning of the program for some managers/leads to cross organizational lines and cause confusion. The confusion resulted from situations in which a manager/lead from one organization would try to direct individual engineering staff in another organization, without the involvement of the manager/lead in the engineer's organization. This is always counterproductive. This was straightened out with time, but led to significant inefficiencies in the beginning.

In ACE there was a unique difference between the spacecraft and the instruments in the flight "rules" for documentation, review and testing. The spacecraft was Class C, while instruments were Class D. It is appropriate to have the spacecraft come under tighter scrutiny for design and fabrication, because of the fact that it serves the entire instrument payload. But for ACE, the Class D designation for instruments resulted in problems when it came time for integration of the instruments with the spacecraft. Documentation and/or records of interfaces and functional, environmental and performance testing were not always available, and it was difficult to incorporate these instruments into a configured spacecraft and still satisfy the requirements levied by GSFC at the Observatory level. There needs to be some documented requirements for the integration of Class C and D components that reflects the reality of the situation.

Some subsystem and I&T specific lessons learned follow:

A more standard interface for the star scanner (i.e., a 1553 interface instead of the custom serial-digital interface) would have greatly simplified the manufacturing of the star scanner by the vendor. Significant time was lost, resulting in a late delivery, because of difficulties with finding bugs in the timing synchronization of the star scanner's custom interface.

More oversight of the solar cell production might have uncovered earlier a crack in the manufacturer's calibration standard, which made performance look slightly better than actual. Testing should have included a hot flash to verify no cracks. Another NASA program had solar cell trouble, causing the concern for cracks, but too late to risk the test on ACE panels.

Also in the power area, it was learned that It is possible to fly battery cells which have been in dry storage for many years.

In the integration area, several features of the ITOCC system helped the ACE program attain its cost and schedule goals. These included using spacecraft redundancy and flexible ground systems to streamline spacecraft testing, using a scaleable ground system architecture that was easy to expand and shrink as needed, the on-line command GUI, and providing early access to the spacecraft by the mission operations team.

The software used for I&T was TSTOL, which the mission operation center is using in post launch operations. If the software, which had to be modified for ACE, had presented itself earlier in the integration scheduled time period, the bugs could have been coped with better. The development of I&T software should have been given precedence over flight operations software much earlier than was done.

6.2.4 Ground System Perspective

The successful implementation of a common ground system for ACE integration and test, and mission and science operations proves that this is a viable concept for future missions. Currently several NASA missions, including the Transition Region and Coronal Explorer (TRACE), the Wide-Field Infrared Explorer (WIRE), the Microwave Anisotropy Probe (MAP), and the Earth Orbiter-1 (EO-1) are developing common ground systems for spacecraft integration and test and mission operations. While these missions have chosen different control center architectures, they are following many of the same principles as ACE. The FOT has been staffed early as mission design decisions are being made. The FOT is actively involved in developing the command and telemetry database and in developing and executing spacecraft integration and test procedures. The common ground system allows the command and telemetry database, displays and test procedures used in integration and test to be reused with minor modifications for on-orbit operations. The experience the FOT gains with both the spacecraft and ground system allows a smaller team to handle on-orbit operations, thereby reducing sustaining mission operations cost.

The attempt to reuse functionality from the XTE's TPOCC proved to be a mixed blessing. While the reuse of the XTE software reduced costs, the lack of control over the XTE software development schedule increased ACE schedule risk. The XTE system was less mature than expected and its use became a source of errors in the ACE Generic. Moreover, the ACE Generic staffing profile was not adequate to fix these errors and develop all the new capabilities as scheduled. Therefore, constant tradeoffs were made to prioritize the XTE discrepancy fixes versus the new capabilities requested. Similarly, GTAS, planned for trend analysis in the ITOCC, was dropped because of the lack of maturity. It was concluded that to be the first user of government off the shelf (GOTS) or commercial off the shelf (COTS) systems may not be worth the additional schedule risk, especially when having hard deadlines that cannot accommodate extensive system debugging.

The assumption that a common ground system could support both integration and test and flight operations spacecraft control requirement proved largely correct. However, adapting a system originally designed for flight operations to service integration and test requirements presented some unique challenges. A description of each of these challenges is provided below, and a more detailed discussion appears in section 7.4.

1. A considerable amount of effort was expended up front trying to document all the requirements for the MOC, ITOCC and ASC for ACE Generic. However all the ITOCC

requirements could not be defined in advance because the spacecraft was not yet built. The development team accommodated the growth in requirements by stretching out the ACE Generic delivery schedule and delaying the implementation of the less mission-critical ACE MOC specific requirements.

2. The original delivery schedule of providing ACE Generic releases 3 or 4 months apart did not meet the ITOCC requirements for delivery of new capabilities or fixing high priority system discrepancies. Thus the development team abandoned the system release approach and created a "function drop" delivery mechanism. Function drops were provided to the ITOCC on a weekly basis. This new delivery approach resulted in better responsiveness to ITOCC needs and allowed new requirements to be more easily accommodated within the development schedule.

3. Development of a common ground system must take into account the resources required to support the different objectives and delivery timing of spacecraft I&T and operations. The ACE development effort ran into significant resource problems starting approximately two years prior to launch when the I&T effort started to ramp up and the initial MOC systems were due for initial release to support early FOT testing with the spacecraft. On future common system development efforts the development team resources must be ramped-up significantly at approximately launch minus two years through the completion of the I&T effort. This will ensure that the needs of both groups are met in a timely manner.

4. Another challenge was the different approach for managing schedule reserve. MOC development at GSFC has tended to vest schedule reserve in the ability to accommodate a schedule slip or failure of a function by postponing work to future releases. However, delays of this scale cannot be accommodated when building an I&T system. Schedule reserve for unforeseen work must be included in the basic schedule for an I&T system just as it is for any spacecraft component or instrument.

5. Spacecraft integration and test stresses the command and control system in a different fashion than flight operations. Spacecraft integration and test sends a much larger number of real-time commands, interfaces with the spacecraft for longer periods of time, and has a much greater dependency on real-time telemetry for diagnosing problems and confirming spacecraft health. The FOT relies on recorded data for trending and determining spacecraft anomalies. To accommodate these integration and test requirements, changes had to be made to the size and handling of data files.

6. Similarly, another "cultural" difference pertained to procedures. I&T procedures are based on complete component and spacecraft functionality, whereas FOT procedures are based on nominal and contingency operational scenarios. Thus, FOT procedures tend to be more segmented than those used for I&T. This required the FOT to develop many procedures from scratch, using the I&T procedures only as a baseline. This procedure conversion process also led to a greater configuration and control effort than originally planned.

7. During development of a common I&T/operations ground system, the ground system developer must ensure that maximum commonality is maintained between the systems to ensure that the only variations between the systems are those that are absolutely necessary for each group to perform unique functions. On ACE, the development

teams working on the I&T system performed a large number of system modifications that “improved” I&T operations but were not critical to I&T functional objectives. Since these changes were done after the baseline system was delivered to I&T, these changes were not reflected back in the MOC system developed for operations. Future common system development efforts should include a development team Control Board that reviews and approves all non-common changes submitted by either the I&T or operations teams.

8. A large number of the high priority discrepancies with the I&T team dealt with inconsistencies with the user interface. While a FOT member will become highly trained on the intricacies of the MOC user interface, many I&T users (such as system engineers) have little to no familiarity with the ITOCC. A consistent user interface allows such users to quickly come up to speed using the ITOCC with minimal training while they trouble shoot problems. The development team had to become sensitized to this issue and give higher priority to fixing user interface problems when they arose.

There is a very large overlap between functions performed for spacecraft I&T and the functions performed in operating a spacecraft. However, the needs, cultures, and approaches of the Integration and Test team are different from that of the Flight Operations Team. Coping with these differences presented the greatest challenge to the ACE Mission Operations Team. These challenges were overcome through extensive coordination among the various team members. Many meetings were needed between the I&T team, FOT, and the software developers to review the status of, assign priorities to, and establish the schedule for system capabilities and discrepancies. Communications was the key to avoiding and resolving problems.

A better understanding of the requirements for the APL I&T system would have reduced the schedule risk to the flight operations. Employing some software developers more familiar with spacecraft I&T, would have enhanced understanding by the flight operation team of the needs of the I&T team.

7.0 DISCUSSION OF MISSION ELEMENTS

This section provides a more detailed discussion of the science payload, spacecraft, integration and ground systems. Where there is a discussion on complexity rating, it is based on judgments of the relevant ACE team members using the following scale of 1 to 5 to rate the level of complexity or difficulty in developing and building the particular mission element:

1 Easy	= Low Complexity
2	= Medium-Low Complexity
3 Average	= Medium Complexity
4	= Medium-High Complexity
5 Hard	= High Complexity

7.1 SCIENCE PAYLOAD

Table 7.1-1 provides the performance requirements for the ACE instruments.

7.1.1 Cosmic Ray Isotope Spectrometer (CRIS)

7.1.1.1 Description

California Institute of Technology, Pasadena, California was the developing institution. Co-Investigators are: Alan Cummings, Mark Wiedenbeck, Robert Binns, Richard Mewaldt, and Tycho von Rosenvinge.

The following description of CRIS was extracted from E. C. Stone et al., The Cosmic Ray Isotope Spectrometer for the Advanced Composition Explorer, accepted for publication in *Space Science Reviews*, 1998. A source for obtaining this paper and the full detail about CRIS is Alan C. Cummings, CIT.

CRIS is designed to cover the highest decade of the ACE's interval, from ~ 50 to ~ 500 MeV/nucleon, with isotopic resolution for elements up to Z = 30. The nuclei detected in this energy interval are predominantly cosmic rays originating in our Galaxy. This sample of galactic matter can be used to investigate the nucleosynthesis of the parent material, as well as fractionation, acceleration, and transport processes that these particles undergo in the Galaxy.

Charge and mass identification with CRIS is based on multiple measurements of dE/dx and total energy in stacks of silicon detectors, and trajectory measurements in a scintillating optical fiber trajectory (SOFT) hodoscope. The instrument has a geometrical factor of ~ 250 cm^2 ster for isotope measurements, and should accumulate ~ 5 x 10^6 stopping heavy nuclei (Z>2) in two years of data collection under solar minimum conditions.

Table 7.1.1-1 provides a summary of CRIS characteristics and Table 7.1.1-2 gives CRIS design requirements.

Figure 7.1.1-1 displays the CRIS instrument cross section, top view and side view. The side view shows the fiber hodoscope which consists of three hodoscope planes (H1,

Table 7.1-1

Performance Requirements for ACE Science Instruments

Particle Spectrometers

Instrument	Required Geometry Factor (cm2.sr)	Required Species Coverage	Energy Range for Element Composition (MeV/nuc) C (Z=6)	Fe (Z=26)	Energy Range for Isototopes(1)/Charge States (2) (MeV/nuc) C (Z=6)	Fe (Z=26)
CRIS	>200	$3 \le Z \le 28$	70 to 200	100 to 500	70 to 200	100 to 500
SIS	>20	$2 \le Z \le 28$	10 to 100	15 to 200	10 to 70	20 to 100
ULEIS	>0.5	$2 \le Z \le 28$	0.1 to 10	0.3 to 2	0.2 to 10	0.3 to 2 (3)
SEPICA	>0.2	$2 \le Z \le 28$	0.3 to 10	0.1 to 20	0.3 to 2	0.1 to 2
SWIMS		$2 \le Z \le 28$	.0001 - .005	.0001 - .001	.0001 - .005	.0001 - .001
SWICS		$2 \le Z \le 28$	.0001 - .01	.0001 - .01	.0001 - .01	.0001 - .01

Energetic Particle and Solar Wind Monitoring Instruments

Instrument	Required Geometry Factor (cm2.sr)	Required Species Coverage	Electron Energy Range (keV)	Proton Energy Range (Mev)	Helium Energy Range (MeV)	Time Resolution (min)
EPAM	0.5	Electrons, H, He	50 to 200	0.1 - 2	0.5 - 3	2
SWEPAM		Electrons, H, He	.001 - 0.5	.0001 - .01	.0001 - .01	2

Magnetometer

Dynamic Range:	+/- 4 nT to +/- 60,000 nT
Precision:	better than +/- 0.5%
Accuracy:	< 0.1 nT
Sampling Time:	< 0.35 sec

Notes:

(1) To resolve individual isotopes requires an rms mass resolution of < 0.3 amu; to resolve $\Delta M = 2$ isotopes requires an rms mass resolution of < 0.6 amu.

(2) For SEPICA and SWICS, the energy range for resolving charge states is given.

(3) For ULEIS the Fe energy range corresponds to < 0.6 amu rather than < 0.3 amu

H2, H3) and one trigger plane (T) which are located above the four stacks of silicon detectors. The top view shows the fiber readout which consists of two image intensified CCDs at either end of the fibers and the four stacks of silicon detectors.

Figure 7.1.1-2 is a photograph of the fully assembled instrument, which consists of two boxes bolted together. The upper box contains the SOFT system, while the lower box contains the Si(Li) detector stacks with their pulse-height analysis electronics, as well as the main CRIS control electronics. The large window on the top of the SOFT box is the CRIS entrance aperture. The two smaller patches are thermal radiators for cooling the two CCD cameras.

Figure 7.1.1-3 is a photograph showing a top view of the SOFT detector before integration with the rest of the CRIS instrument. The fiber plane which is visible is the trigger plane. The hodoscope planes are below that and are not visible. The fiber outputs are routed to two image intensifer/CCD assemblies, one at the upper right and one at the lower left of the photograph. Thermal radiators are attached to the tops of the assemblies in flight (not shown in this photograph) to cool the image intensifiers and CCD arrays. The high voltage power supplies for the image intensifiers are located to the upper left of the fiber plane. The readout electronics for each camera are in the black boxes in the upper left of the photograph.

Figure 7.1.1-4 provides the CRIS electronics block diagram.

7.1.1.2 Challenges and Solutions

The development challenge was in developing custom VLSI and hybrid chips. These were also difficult to manufacture and there were problems with loose wires for the custom hybrids. The extended Phase B period was very helpful in developing the custom VLSI.

The major risk for CRIS was the new trajectory system. The scintillating optical fiber hodoscope had never flown before on a spacecraft.

The electronics and structural/mechanical system were all new. The modular mechanical design of CRIS, made possible in part by the generous mass allocations available on ACE, wherein detectors were part of boards that plugged into a card cage, made working on the instrument far simpler than in past instruments of this type. The use of zero-insertion force connectors greatly helped in this regard. One engineering model and one protoflight instrument were built. As a fall-back position, some descoping could have been done if necessary: at the cost of collecting power, some back detectors or half the detectors could have been eliminated.

7.1.1.3 Complexity Rating

Ratings ranged from 4 to 5 for a 4.5 average, essentially categorizing CRIS as a high complexity instrument. The reasons for this high rating are a) the new scintillating optical fiber technology (SOFT) front-end hodoscope, b) new large-area lithium-drifted stack detectors, and c) new custom hybridized stack read out VLSI.

Table 7.1.1-1: Summary of CRIS Characteristics

Measurement Objective	Elemental and isotopic composition of galactic cosmic rays
Measurement Technique	Multiple-ΔE vs. residual energy plus trajectory
Sensor System	
Energy Loss Measurements	Si(Li) detectors (3 mm thick × 10 cm diameter, some with active guard rings) in 4 stacks of 15 detectors each
Trajectory Measurements	Scintillating Optical Fiber Trajectory (SOFT) hodoscope, position resolution $< 100\,\mu$m rms, 7.2 cm lever arm
Charge Interval	
Primary Interval	$4 \leq Z \leq 28$
Extended Interval	$1 \leq Z \leq 30$
Energy Interval for Mass Analysis	
O	60–280 MeV/nucleon
Si	85–400 MeV/nucleon
Fe	115–570 MeV/nucleon
Field of View	
Primary FOV	45° half angle
Full FOV	70° half angle
Geometrical Factor (in primary FOV)	$\sim 250\,\mathrm{cm}^2$ ster
Event Yields (solar minimum; $\theta < 45°$)	
Be	3.0×10^4/yr
O	8.9×10^5/yr
Si	1.7×10^5/yr
Fe	1.4×10^5/yr
Mass Resolution (rms)	
O	$\lesssim$0.15 amu
Fe	$\lesssim$0.25 amu
Resource Utilization	
Dimensions ($l \times w \times h$)	54.4 cm × 43.9 cm × 23.6 cm
Mass	29.2 kg
Instrument Power	11 W
Bit Rate	464 bits/sec

Table 7.1.1-2: CRIS Design Requirements

Parameter	Requirement
Trajectory System	
Zenith Angle Resolution	$< 0.1°$
Relative Positon Resolution	< 0.13 mm rms
Absolute Position Resolution	< 1 mm rms
Detection Efficiency	
$Z = 4$ (Be)	$> 50\%$
$Z \geq 8$	$> 90\%$
Silicon Detectors	
Active Thickness	3.0 ± 0.1 mm
Thickness Variation	$< 60\,\mu m$
Dead Layer Thickness	$\lesssim 50\,\mu m$
Depletion Voltage	< 250V
Leakage Current (20°C)	$< 15\,\mu A$
Pulse Height Analyzer Electronics	
Dynamic Range	$> 700:1$
Quantization Resolution	≥ 12 bits
Linearity	$< 0.05\%$
Gain Temperature Coefficient	< 200 ppm/°C
Offset Temperature Coefficient	< 200 ppm/°C
Ballistic Deficit	$\lesssim 0.2\%$
Noise (rms for guards)	< 50 keV
Power (per pha)	100 mW
Mechanical Construction	
Lateral Position Tolerance	
Trajectory System	0.25 mm plane to plane 1 mm with respect to Si detectors
Silicon Detectors	1 mm detector to detector
Orientation co-planarity	
Trajectory System	0.1°
Silicon Detectors	0.1°
Detector azimuthal orientation	
Trajectory System	0.1° plane-to-plane 1° with respect to Si detectors
Silicon Detectors	1° detector-to-detector

Figure 7.1.1-1: CRIS Instrument Cross Section

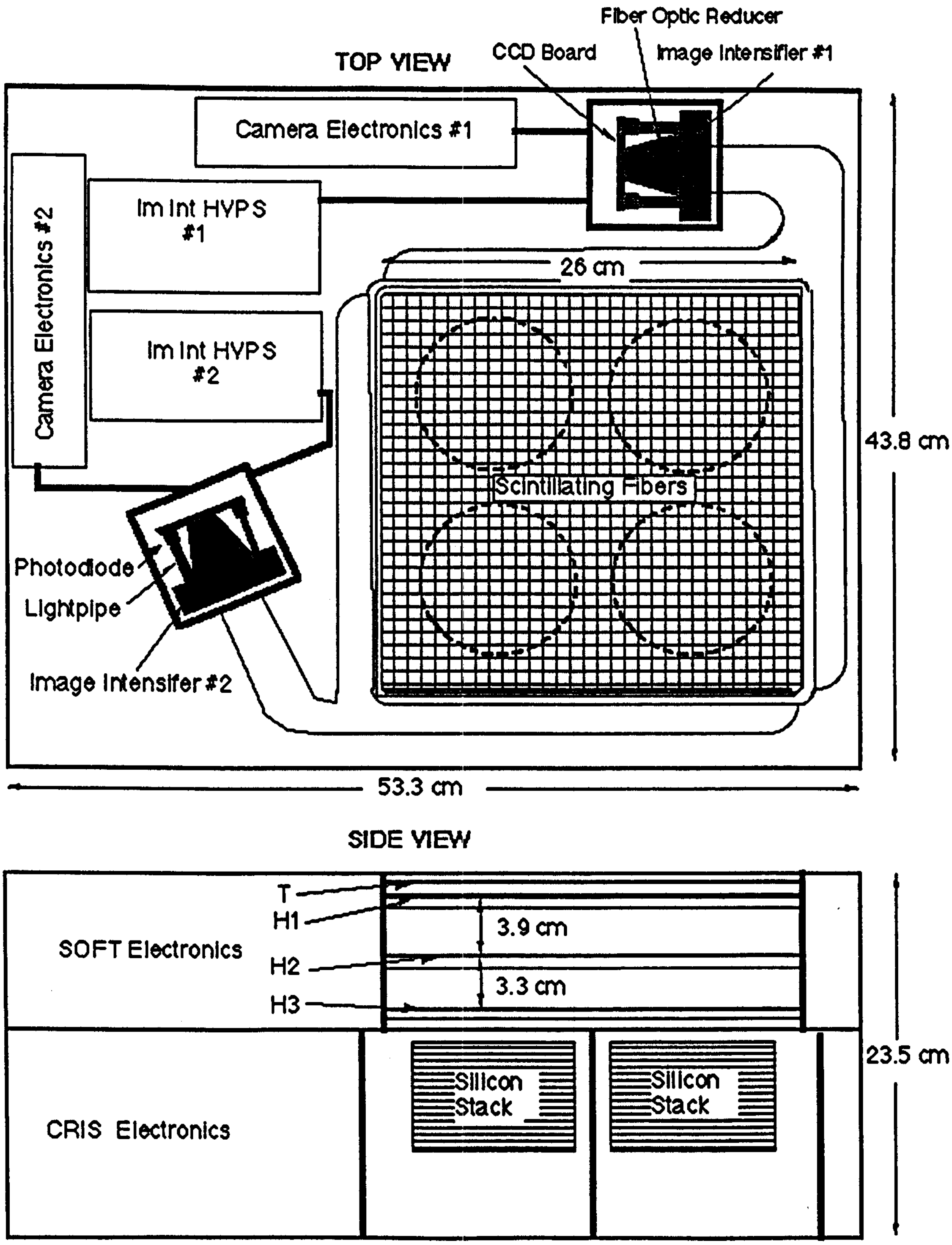


Figure 7.1.1-2: CRIS Instrument

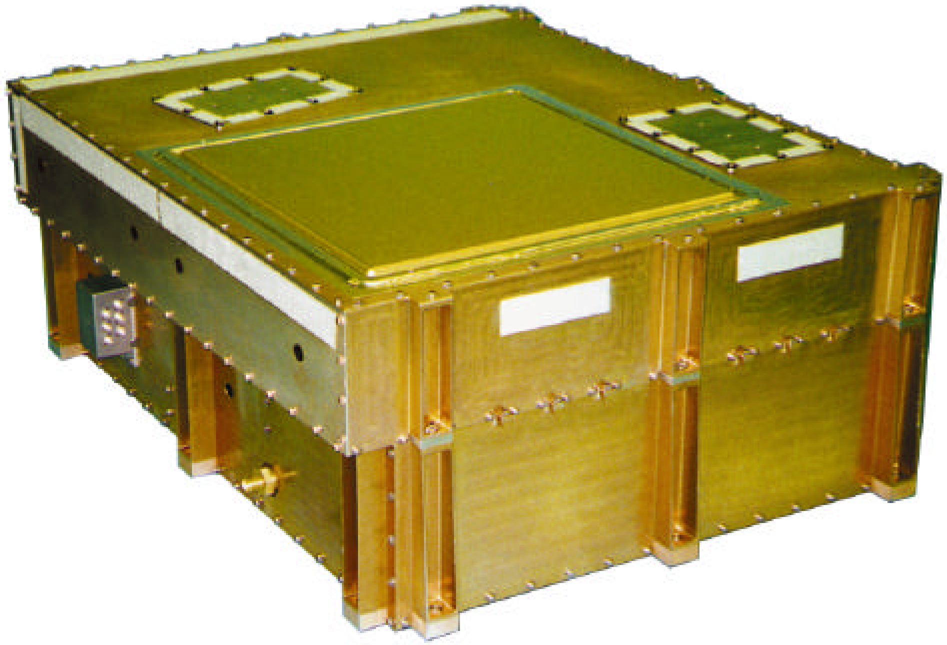


Figure 7.1.1-3: CRIS – Top View of SOFT

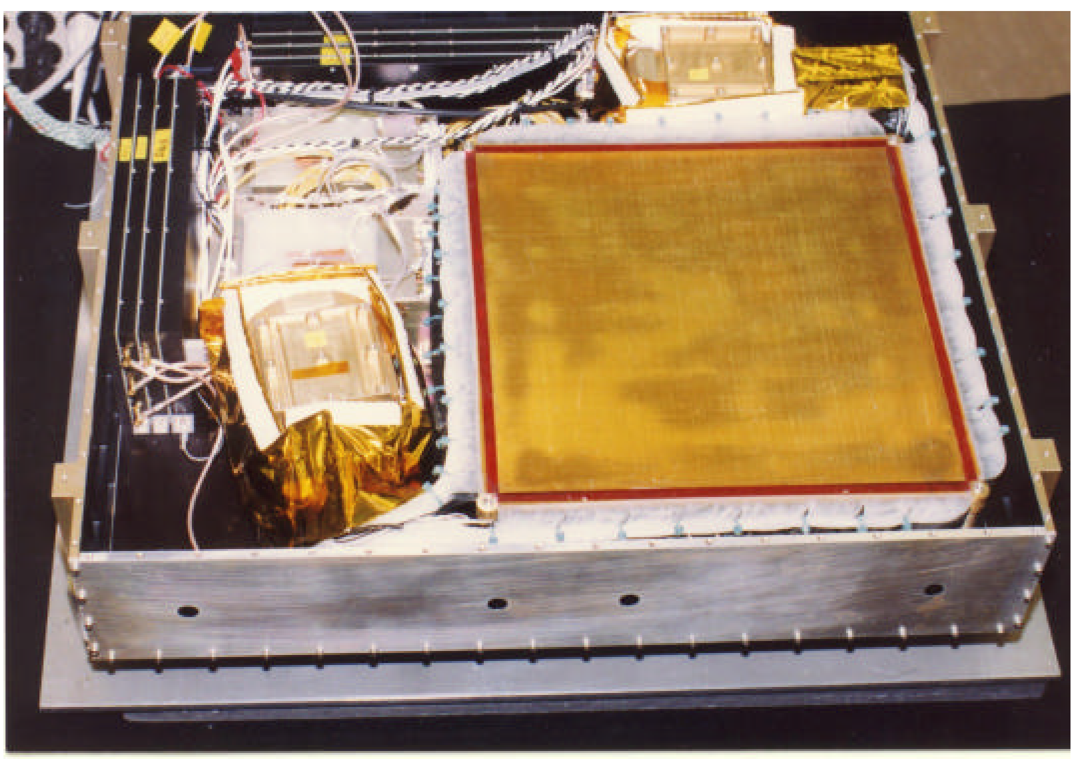
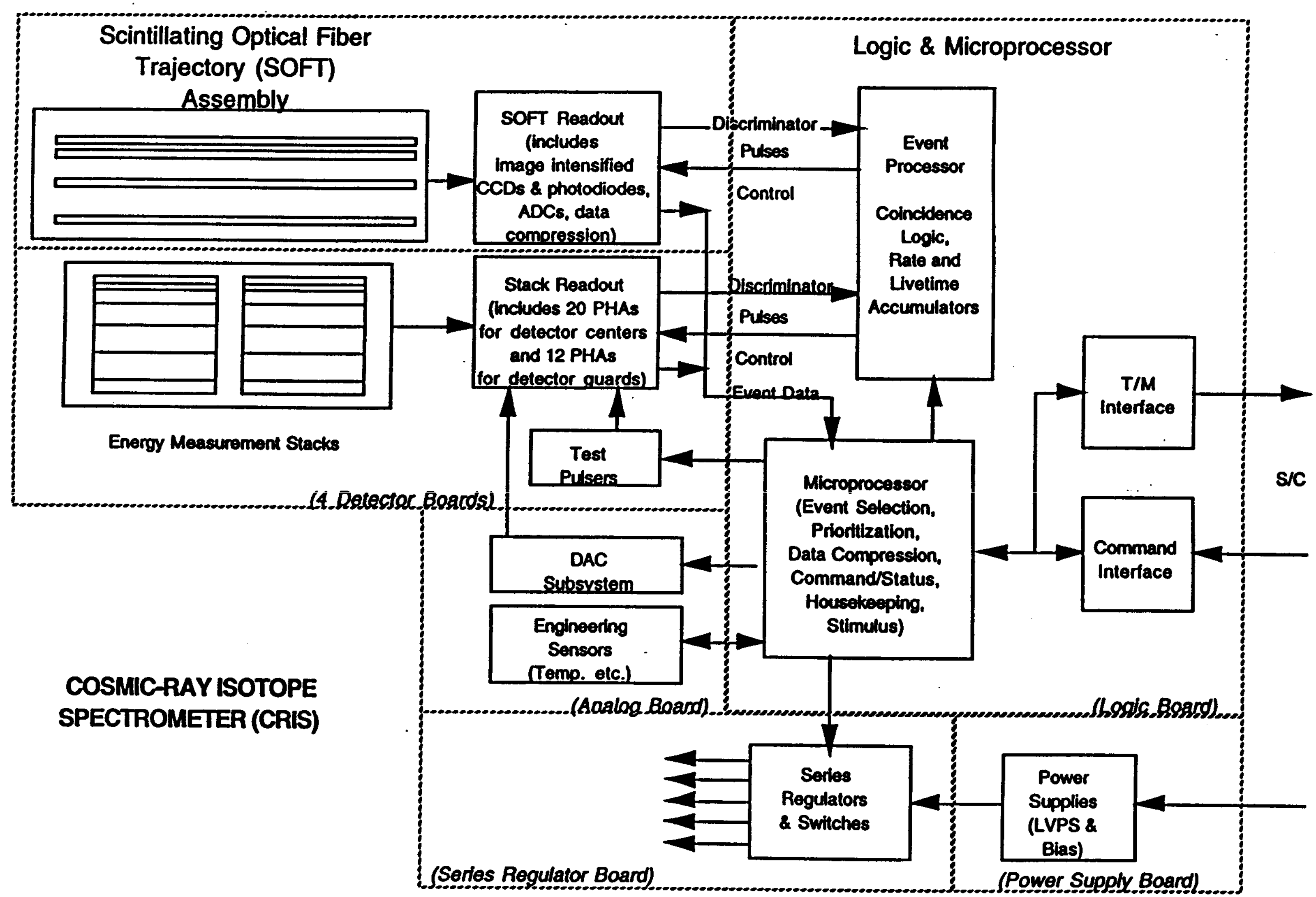


Figure 7.1.1-4: CRIS Electronics Block Diagram



7.1.2 Solar Isotope Spectrometer (SIS)

7.1.2.1 Description

California Institute of Technology, Pasadena, California was the developing institution. Co-Investigators are: Alan Cummings, Mark Wiedenbeck, Richard Mewaldt, and Tycho von Rosenvinge.

The following description of SIS was extracted from E. C. Stone et al., The Solar Isotope Spectrometer for the Advanced Composition Explorer, accepted for publication in *Space Science Reviews*, 1998. A source for obtaining this paper and the full detail about CRIS is Alan C. Cummings, CIT.

SIS is designed to provide high resolution measurements of the isotopic composition of energetic nuclei from He to Zn (Z = 2 to 30) over the energy range from ~10 to ~100 MeV/nucleon. During large solar events SIS measures the isotopic abundances of solar energetic particles to determine directly the composition of the solar corona and to study particle acceleration processes. During solar quiet times SIS measures the isotopes of low-energy cosmic rays from the Galaxy, and of the anomalous cosmic ray component, which originates in the nearby interstellar medium. SIS has two telescopes composed of silicon solid-state detectors that provide measurements of the charge, mass and kinetic energy of incident nuclei. Within each telescope particle trajectories are measured with a pair of two-dimensional silicon strip detectors instrumented with custom very-large-scale integrated (VLSI) electronics to provide both position and energy-loss measurements. SIS was especially designed to achieve excellent mass resolution under the extreme, high flux conditions encountered in large solar particle events. It provides a geometry factor of ~40 cm^2 sr, significantly greater than earlier solar particle isotope spectrometers. A microprocessor control the instrument operation, sorts events into prioritized buffers on the basis of their charge, range, angle of incidence, and quality of trajectory determination, and formats data for readout by the spacecraft.

The electronics and packaging design follows closely that of CRIS. Figure 7.1.2-1 provides the SIS electronics block diagram.

Figure 7.1.2-2 shows a photograph of the SIS stack detector mounted in its board. Figure 7.1.2-3 displays the SIS mechanical design, and Figure 7.1.2-4 provides a photograph of the SIS instrument.

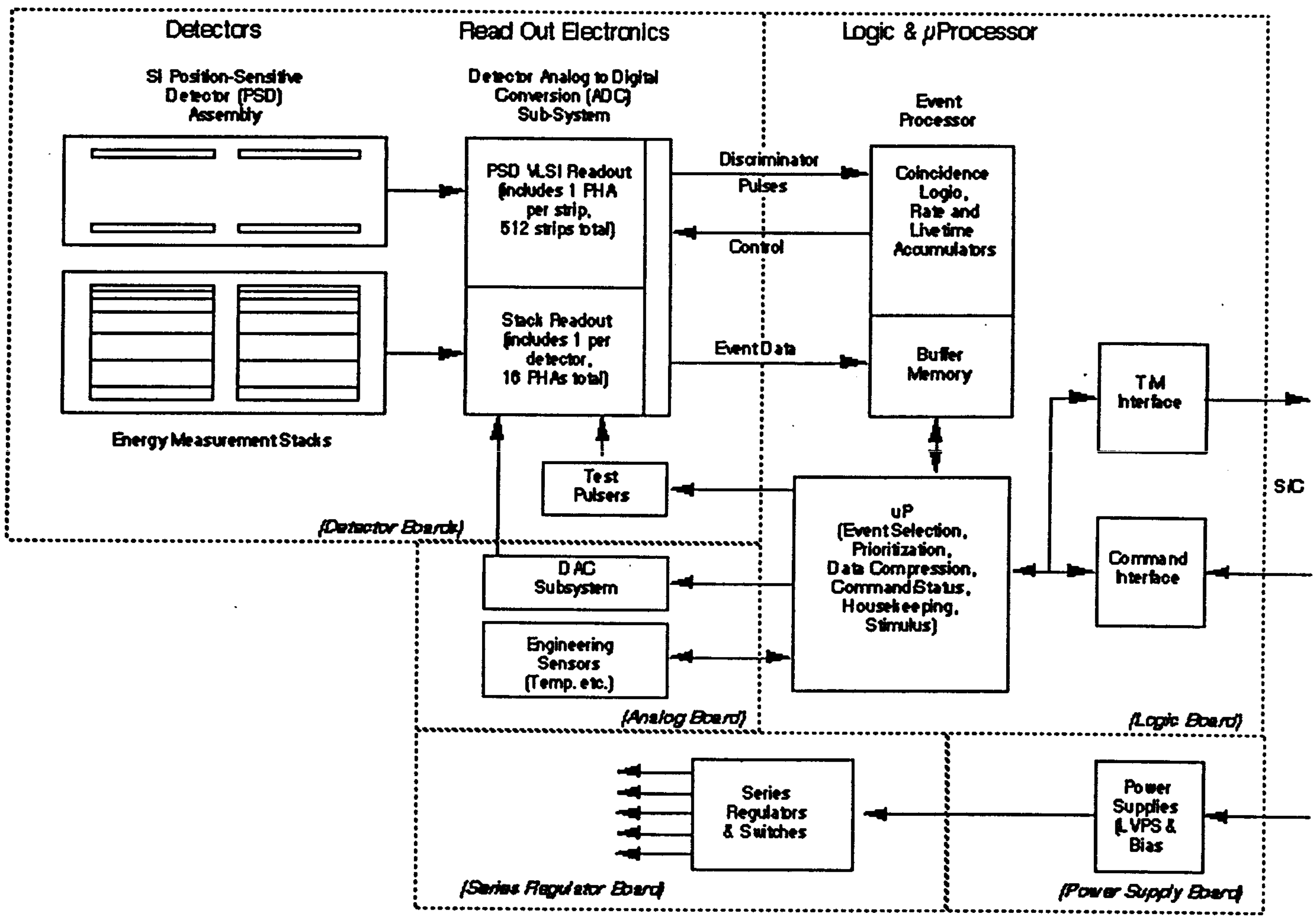
7.1.2.2 Challenges and Solutions

The tough development areas were the matrix detectors and, as with CRIS, the custom VLSI and hybrid chips. As with CRIS, the chips were also difficult to manufacture and there were problems with loose wires for the custom hybrids. The extended Phase B period was very helpful in developing the custom VLSI.

The major risk for SIS was getting enough matrix detectors, since it is very difficult to make large area, thin devices.

Figure 7.1.2-1

SOLAR ISOTOPE SPECTROMETER (SIS) Block Diagram



A major problem for SIS was the large vibration specification set by GSFC. It was far too conservative (i.e., too large by a factor of 10 or more), and this took extra time and funds by the instrument developers.

The electronics and structural/mechanical system were all new. As with CRIS, detectors being part of boards made mechanical construction simpler than on other instruments of this kind. Units built were:

- one engineering model for electronics testing
- one calibration model (duplicate of flight unit)
- one protoflight instrument

As a fall-back position, some descoping could have been done if necessary. Half the detectors could have been cut, but this would have reduced the collecting power by a factor of two.

7.1.2.3 Complexity Rating

Most ratings were 5's, categorizing SIS as a high complexity instrument. The reasons for this high rating are a) the new large-area, high resolution, front-end matrix detectors, b) new large-area ion-implanted stack detectors, c) custom CMOS matrix readout VLSI, and d) the need to add covers to protect from the launch acoustic environment.

Figure 7.1.2-2: SIS Stack Detector

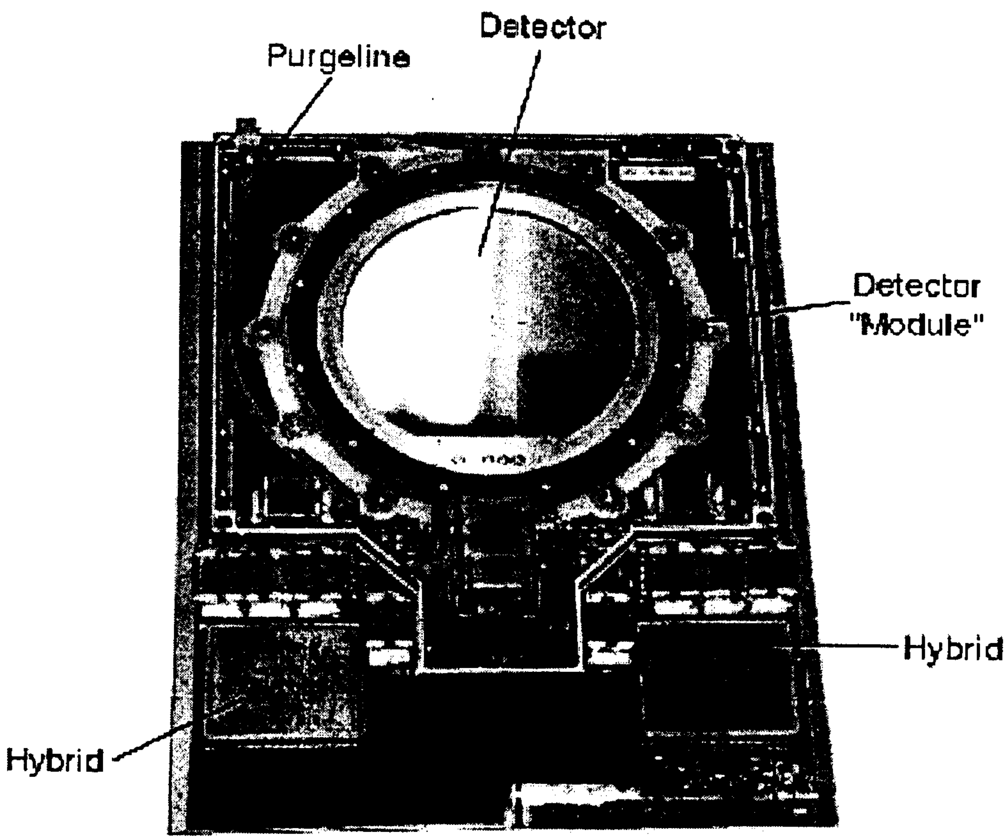


Figure 7.1.2-3: SIS Mechanical Design

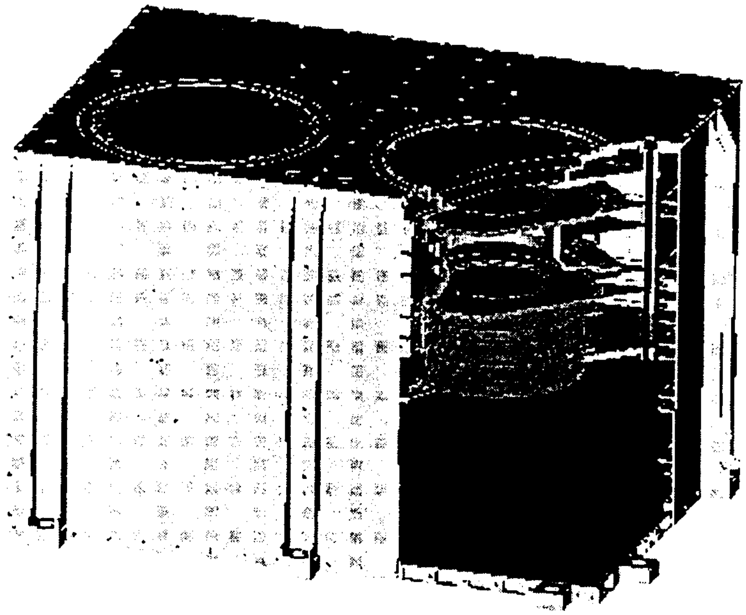


Figure 7.1.2-4: SIS Instrument

7.1.3 Ultra Low Energy Isotope Spectrometer (ULEIS)

7.1.3.1 Description

ULEIS was developed by the University of Maryland and the Johns Hopkins University's Applied Physics Laboratory. Glenn Mason, UMD, is Co-Investigator and Robert Gold, APL, is Lead Investigator.

ULEIS measures ion fluxes over the charge range from He through Ni from about 20 keV/nucleon to 10 MeV/nucleon. Exploratory measurements of ultra-heavy species (i.e., mass range above Ni) are performed in a more limited energy range near 0.5 MeV/nucleon.

The ULEIS instrument is composed of three items: the time-of-flight (TOF) telescope, the analog electronics box, and the digital system box (or data processing unit). The telescope is mounted on the sunward side of the spacecraft and points at a 30° angle to the spacecraft spin axis. The analog electronics box is located nearby to minimize detector lead lengths, and the digital electronics box is located within 100 cm of the analog box. Figure 7.1.3-1 gives the interconnection scheme for the three subsystems of ULEIS. Figure 7.1.3-2 provides the functional block diagram for the telescope and analog electronics. Figure 7.1.3-3 shows the prototype instrument.

The ULEIS sensor telescope is a time-of-flight mass spectrometer which identifies ions by measuring the TOF and residual kinetic energy of particles that enter the telescope cone and stop in one of the arrays of silicon solid-state detectors. The TOF is determined by START and STOP pulses from microchannel plate (MCP) assemblies that detect secondary electrons which are emitted from the entrance foil and other foils within the telescope when the ion passes through them.

Solar UV would cause secondary electron emission from the telescope foil, thus increasing background. The ULEIS design avoids this by pointing the telescope at 30° to the spin axis, which is approximately the solar direction, and including a sunshade that prevents sunlight from striking the foil directly.

To prevent electronic pile-up during intense flux periods, a mechanical shutter (or iris) slides partially closed under on-board control depending on the START MCP singles counting rate. This iris can decrease the geometrical factor by a range of about 100:1, thereby making it possible for ULEIS to operate in the most intense events. The iris also serves as an acoustic cover during launch.

All ions triggering the system have their digitized TOF and Si detector energy deposit measurements processed on-board by a microprocessor. Because of the high event rates it is not possible to transmit all particle information within available telemetry allocations. Two types of information are thus sent: first, detailed pulse-height-analysis (PHA) data at a rate of about 5 events/second, and second, "species rates", which are sectored counting rate data derived from the PHA matrices. The species rates cover He, O, Si, and Fe groups in 15 energy intervals that cover the energy range 20 keV/nucleon to 9 MeV/nucleon. These rates are telemetered approximately once per minute.

ULEIS operates in its normal mode continuously except for about one hour/week when an on-board calibrator is activated by ground command. In addition, the iris is exercised every 4-6 weeks to ensure its operational performance. If necessary, there is a provision to upload new look-up tables for the species rates to take account of possible instrumental drifts or inflight performance. The ability to adjust MCP bias, dual START pulse systems, and multiple redundancy in the solid-state detector array helps protect against a wide range of possible drifts and malfunctions.

The data processing unit is a single processor system, running on an RTX2010 processor and using the FORTH software language.

7.1.3.2 Challenges and Solutions

Initially a protoflight approach was selected to most easily meet cost and schedule. As the project progressed, it was found that in more and more areas (e.g. foil design) a second set of hardware was required for development. The main design continued with its set of hardware, and by the end of the project two complete instruments had been built. So, in fact a prototype approach was used. It would have been better to have started the project with the prototype approach.

The development challenges were:

1. achieving required time-of-flight (TOF) performance
2. designing control logic which at the same time a) provided necessary signals to control CIT hybrids, b) allowed energy and position systems to operate independently and at high rates, and c) collected time, energy and position data from legitimate time-energy coincidences
3. achieving a foil design both sufficiently thin for science data and strong to survive launch.

The fabrication of the thin foils was the primary manufacturing area of difficulty.

To mitigate the risk of failing to achieve the TOF performance, a second flight unit was built, as noted earlier, to allow the selection of the better of the two units. Also particle calibrations were performed to determine optimum operating configuration for the instrument.

To reduce the risk of having a broken foil, the foil design was tested thoroughly and individual foils were screened carefully. Double foil was used in the most critical front position.

To mitigate the risk of suffering a short from CIT hybrids, x-ray inspections were performed, with GSFC support, on all CIT hybrids used on flight boards. One bad unit was found and replaced.

The electronics and structural/mechanical system were both 90 percent new.

As a fall-back position, some descoping could have been done if necessary, such as removal of the middle wedge, HVPS and second TOF, but at considerable loss in instrument performance.

7.1.3.3 Complexity Rating

Ratings ranged from 3.5 to 5, with an average of slightly over 4, categorizing ULEIS as a medium-high complexity instrument. The reasons for a rating of 5 or a high rating are the difficulties in a) achieving 300 picosecond particle time-of-flight timing resolution over a wide dynamic range of energies and fluxes, and b) achieving a suitable low energy threshold using extremely thin entrance foils that were still strong enough to withstand the difficult ACE launch acoustic environment. The thinking behind a rating of 3.5 is that while virtually all of the electronics were new and presented some challenges, the sensor concepts had been proved on previous missions and the science is relatively straightforward. In addition, things were kept as simple as possible consistent with getting the science measurements needed.

Figure 7.1.3-1

ULEIS Interconnection Scheme

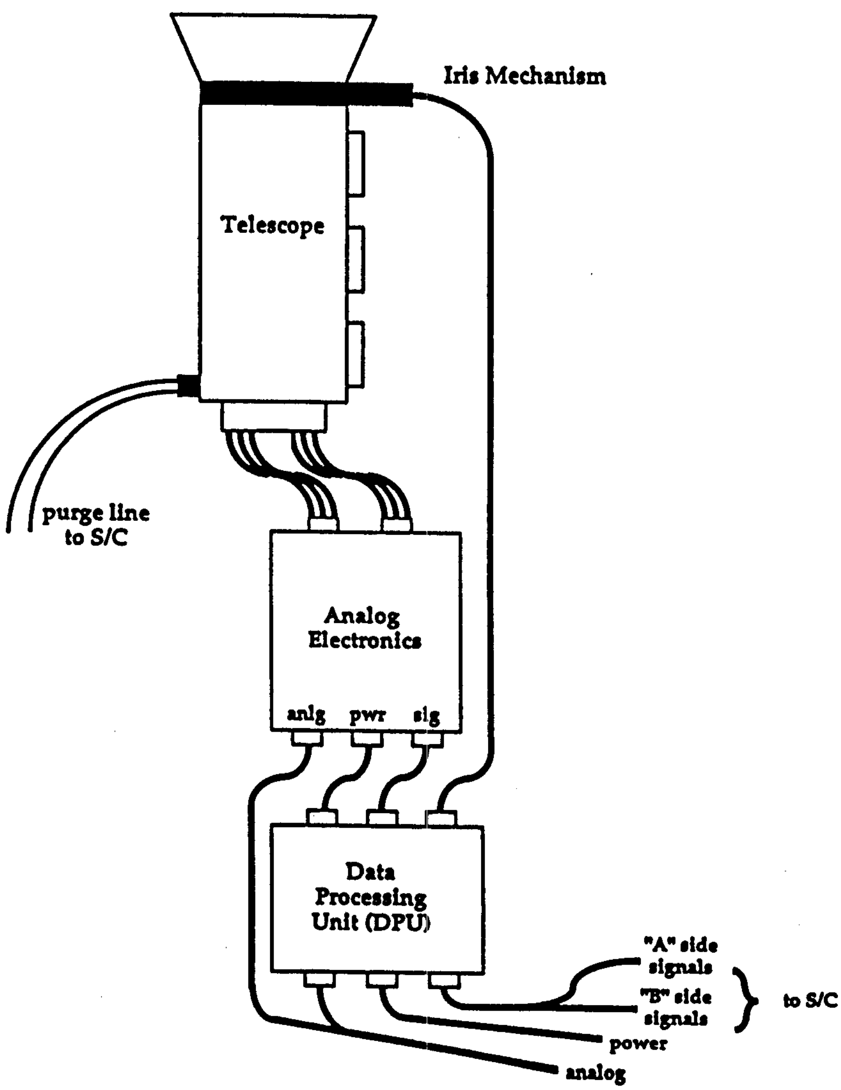


Figure 7.1.3-2: ULEIS Functional Block Diagram

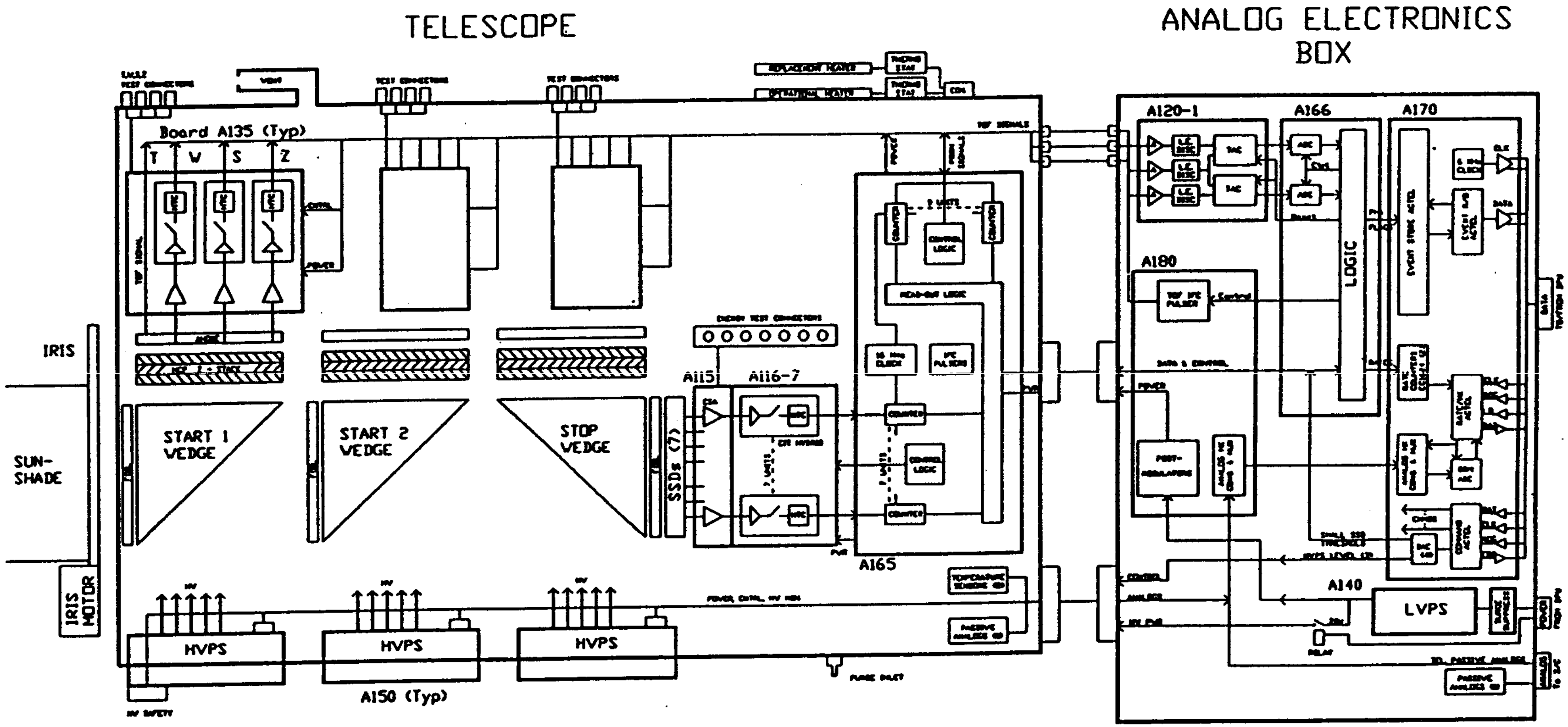
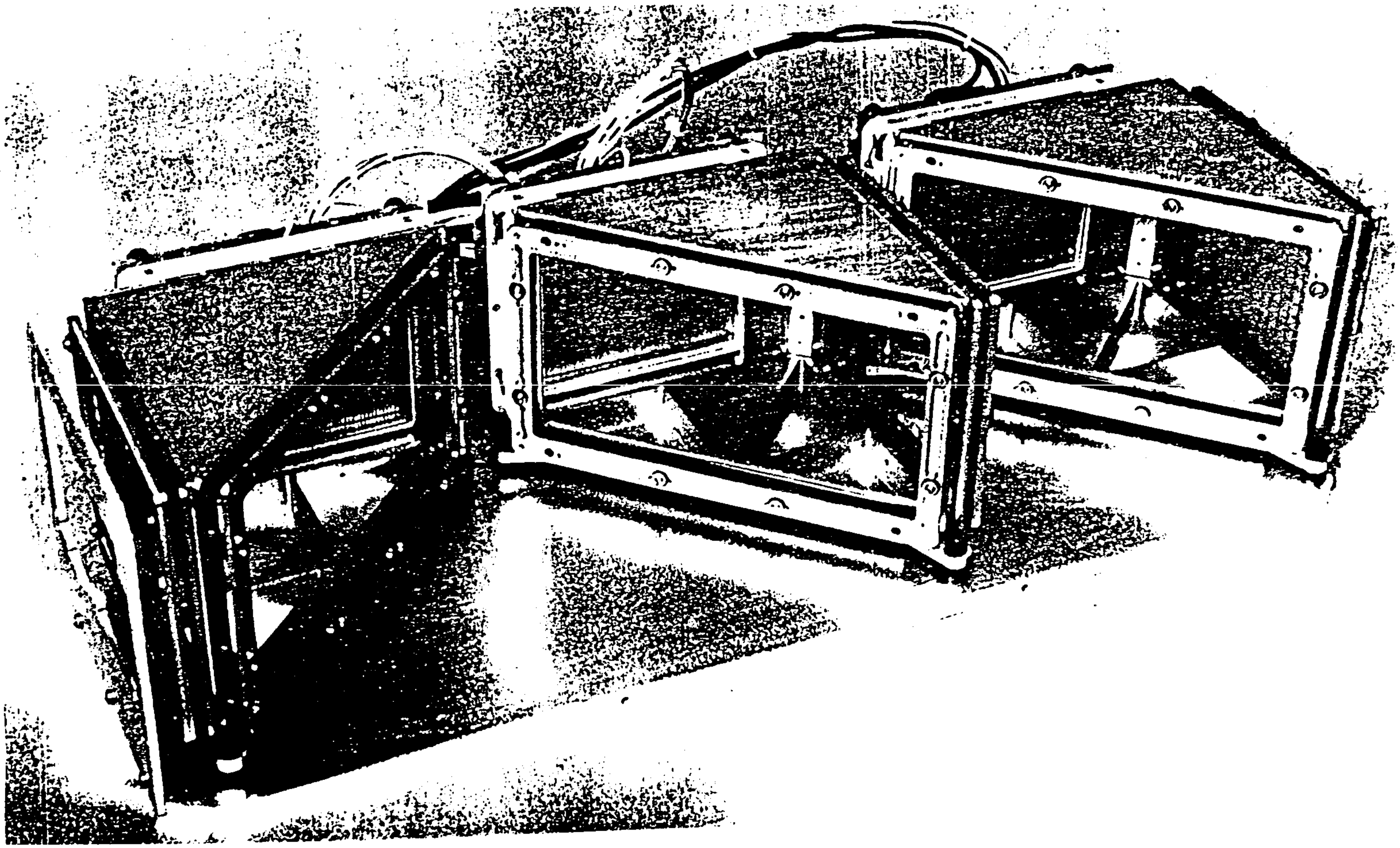


Figure 7.1.3-3: ULEIS Three Wedge Prototype Instrument



7.1.4 Solar Energetic Particle Ionic Charge Analyzer (SEPICA)

7.1.4.1 Description

Co-Investigators are Eberhard Möbius and Lynn Kistler, University of New Hampshire, and Berndt Klecker, Max Planck Institute for Extraterrestrial Physics, Germany. SEPICA was a new instrument with some heritage: SAMPEX/HILT for isobutane gas system, MPE designed barometrically actuated valve, and general Isobutane system issues. ROSAT/P.C. Camera for proportional counter, wire frame concepts and fabrication techniques. ISEE/ULECA & AMPTE/CHEM for basic collimator principles.

The sensor employs electrostatic deflection of incoming ions in a collimator-analyzer assembly, and then measures their impact position in the detector plane. dE/dx - E are determined using a proportional counter and solid state detector combination. Potential background from penetrating radiation is suppressed by the use of an anti-coincidence detector.

To simultaneously determine the energy E, nuclear charge Z and ionic charge Q of the incoming particles several methods are combined:

- Energetic particles entering the multi-slit collimator, which focuses the particles on a line in the detector plane, are electrostatically deflected between a set of electrode plates which are supplied with a high voltage up to 30 kV (to be set by telecommand).

- The deflection, which is inversely proportional to energy per charge, E/Q, is determined in a multi-wire thin-window proportional counter.

- The thin-window proportional counter is also used to measure the specific energy loss dE/dx, which depends on the energy E and the nuclear charge Z of the particle.

- Finally, the residual energy of the particle, E_{Res} , is directly determined in the solid-state detector.

- An anti-coincidence system, which consists of a CsI scintillator and silicon photodiodes, is used to suppress background signals from penetrating high energy particles. This is of particular importance for the study of low fluxes in weak solar events and during quiet times.

Using the relations

$$dE/dx \sim Z^2 (E/M)^a$$

$$Q = (Q/E)*E, \quad \text{and}$$

$$M \sim 2*Z$$

the ionic charge, Q, the initial energy, E, and the atomic number, Z, as well as the atomic mass number M (for low mass ions, e.g., He) can be separately derived for individual particles. The coefficient a is close to 1/2 for energies greater than ≈ 5 MeV/Nucleon. At lower energies the energy loss is reduced due to incomplete ionization of the particles. The correct relation can be taken from tables and will be calibrated with an ion accelerator for the flight instrument.

Figure 7.1.4-1 displays the SEPICA principles of operation. Figure 7.1.4-2 shows the functional and electrical block diagram, and Figure 7.1.4-3 provides a front view of SEPICA with all three apertures. Figure 7.1.4-4 gives the SEPICA proportional counter assembly, as seen from the front side. The gas in- and outlet are on the left. In the fan assembly the center bar of the window support is located exactly behind the HV deflection plate of the electrostatic analyzer. Figure 7.1.4-5 shows the complete SEPICA sensor from the rear side.

Figure 7.1.4-1

SEPICA Principles of Operations

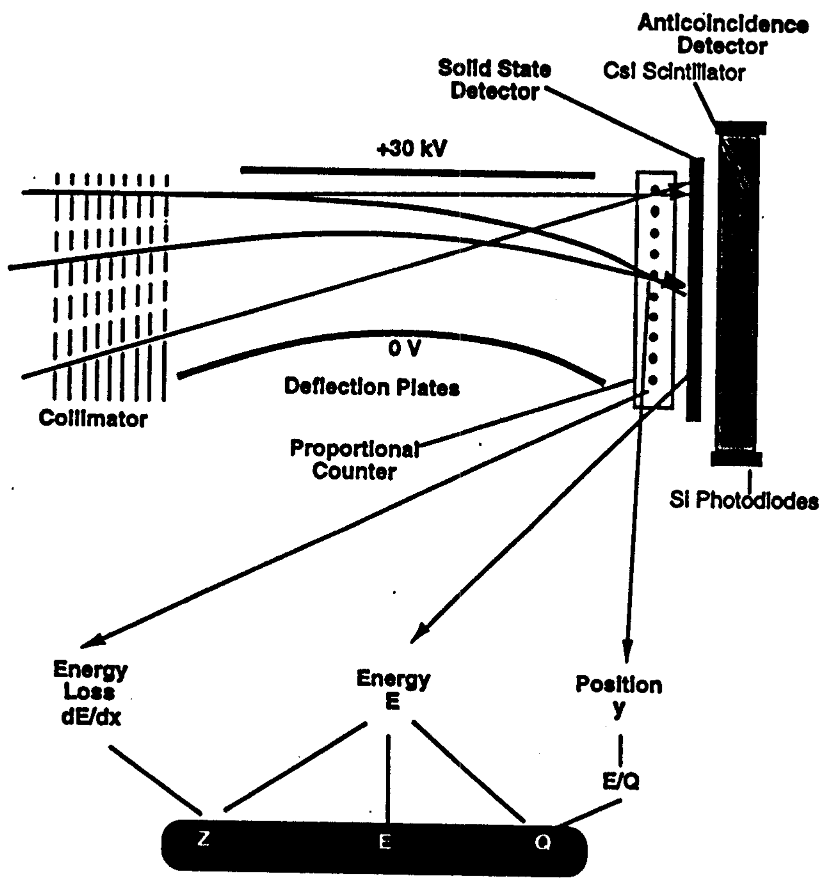


Figure 7.1.4-2: SEPICA Functional and Electrical Block Diagram

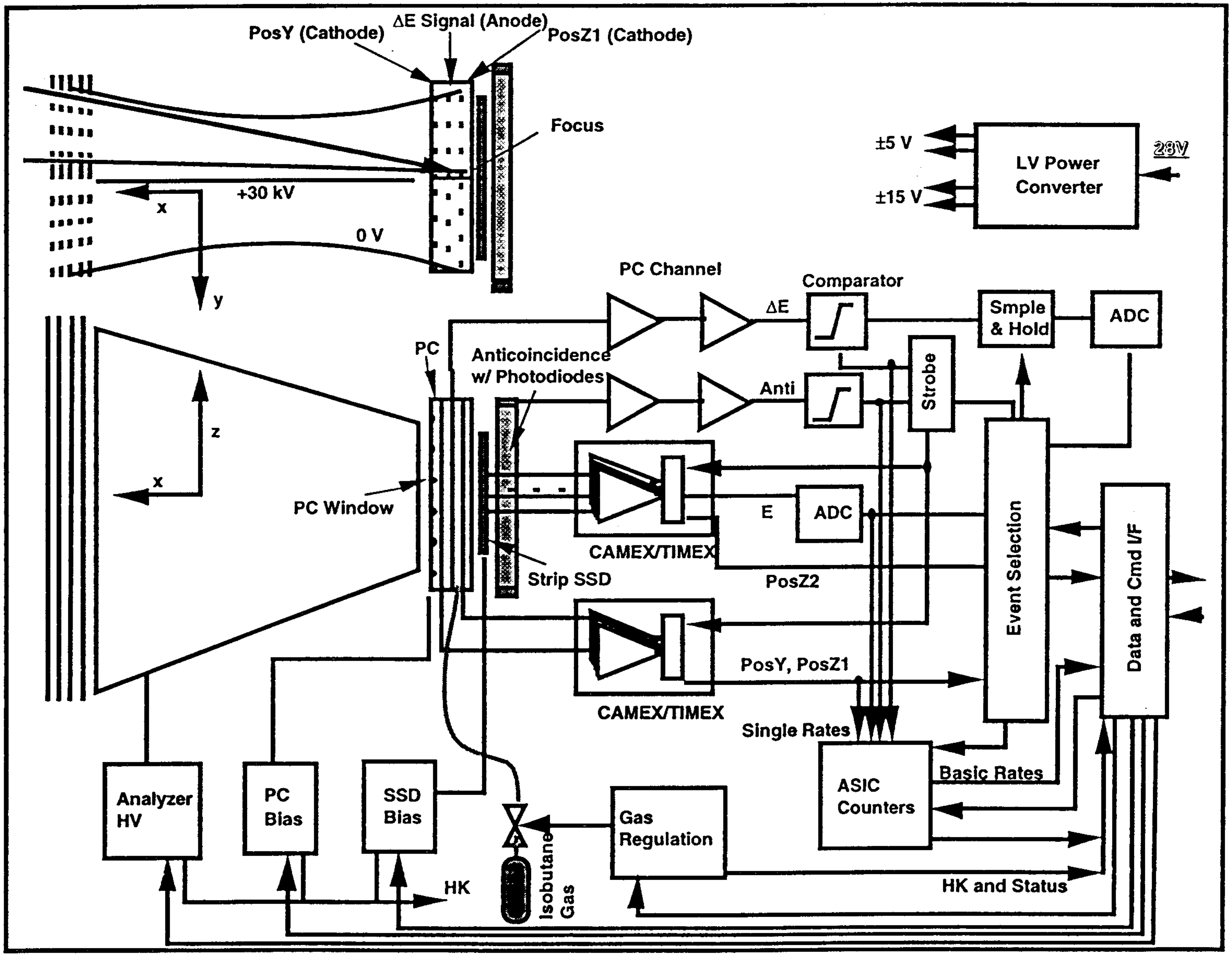


Figure 7.1.4-3: SEPICA Front View

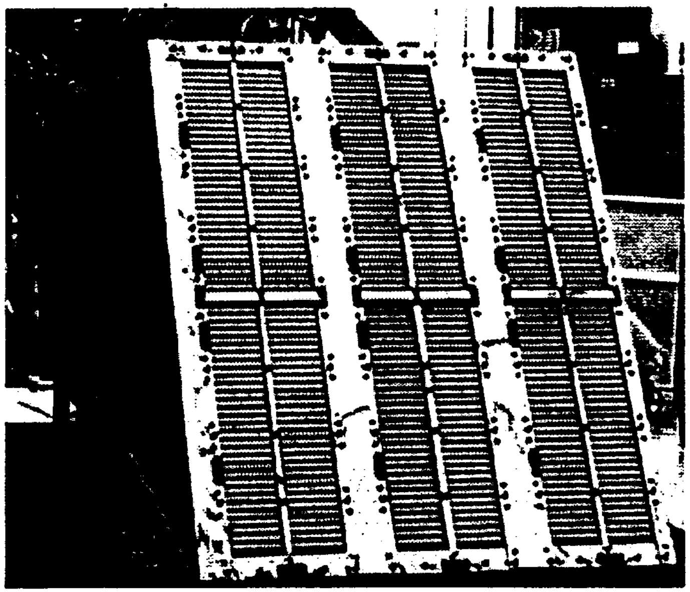


Figure 7.1.4-4: SEPICA Proportional Counter Assembly

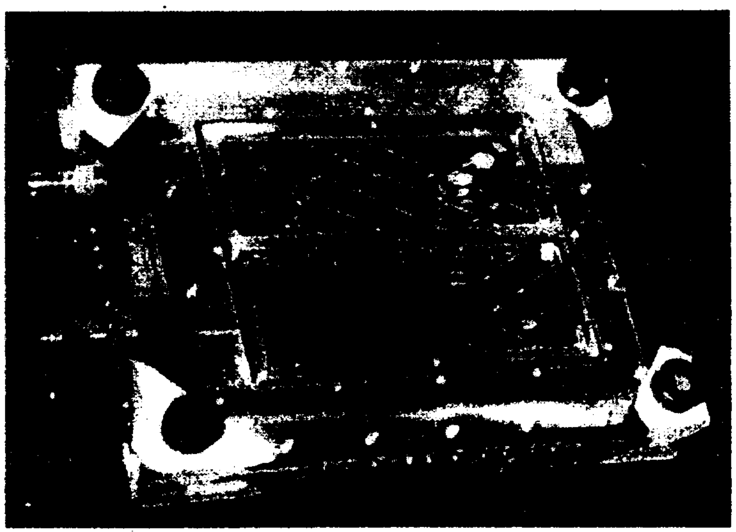
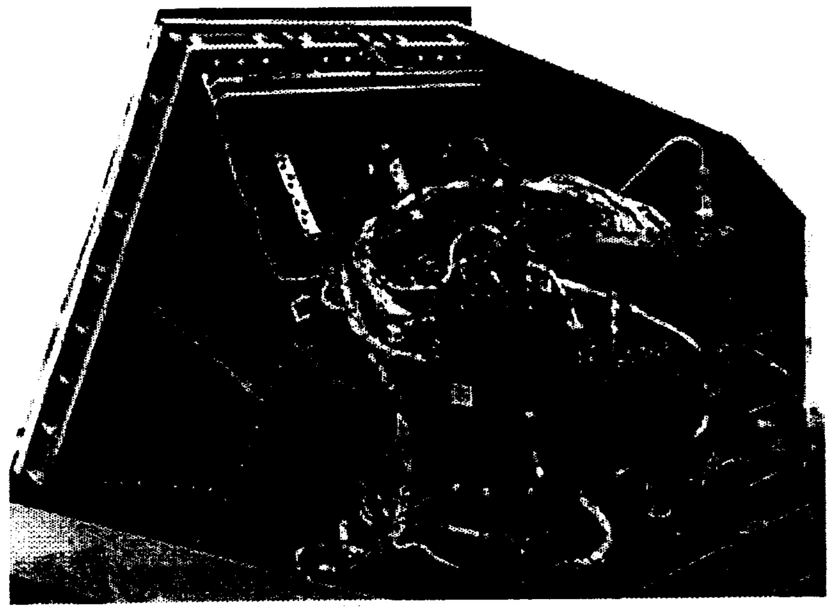


Figure 7.1.4-5: SEPICA Instrument Rear View



7.1.4.2 Challenges and Solutions

1. Development and manufacturing challenges:

a) Fabrication of the optical collimator was a true challenge, particularly in the area precision wet etching for slit size and location tolerances. The development team worked closely with Brush-Wellman, Thermal Electron/Tecomet, Northeast EDM, American Electroplating, and the UNH-SSC Machine shop to ensure all aspects of the near zero alignment collimator assembly would result in adequate plate alignment. The team selected a mill hardened, 0.004" thick, Beryllium Copper sheet, alloy XHMS 190, to ensure the plates would hold their shape after etching; developed etch masks directly from UNH supplied AutoCad graphics files to avoid interpretation errors; milled and wire EDM'd directly from AutoCad graphics files; worked with the shop to select and screen the best alignment pins, and developed methods and machining sequences which minimized machine induced tolerance stack-ups.

b) Previous gas filled counters put the anode and cathode plates within a bulky/structural pressure vessel. With the mass and envelope constraints put on SEPICA, the only way to package the counters was to put the pressure envelope <u>within</u> the counter anode and cathode plates and develop a unique "surface trace electrical feedthru" in the surface of the Macor ceramic plates. O-rings sealed against these flush ground traces allowing the integrity of the pressure envelope to be maintained while passing the signals to the flight electronics. The inlet and outlet of the pressure envelope was sealed with 0.5 micron aluminized Mylar film windows, whose support frames also sealed against the smooth Macor plates with custom o-rings. Critical in the success of this counter design were the following vendors: CMS, Inc., Melrose, MA for their contributions to the fabrication of the "surface trace electrical feedthrus," and Luxel Corp., Friday Harbor, WA for the window assemblies.

c) The development of a compact, low power, reliable, autonomous and <u>sensitive</u> gas metering system, capable of measuring very low absolute pressures and making constant corrections to the proportional counter pressures was key to the success of this instrument. Previous heritage instruments used systems requiring too much power and bulk, and with 3 counters present, were not practical for SEPICA. The identification, and eventual integration of micro silicon transducers and valves fabricated by IC Sensors, Milpitas, CA was a tremendous step forward in precision gas metering. The design of the control electronics by UNH and consulting engineers allowed the feedback loop to work under all flight and test conditions.

d) Development of a homespun high voltage supply that could actually work at 30 kV without discharging was a painful design and fabrication challenge. Once the supply cascade was found capable of achieving 30 kV, distributing it without discharge damage was the key to success. The use of high impedance suppression resistors in the distribution cables became a remarkably successful fix.

2. In terms of Observatory integration, SEPICA was more cumbersome than most due to it's relative bulk, high weight of 82.3 pounds, and the presence of the Isobutane gas supply tank. This instrument presented a hazard to personnel, facility and spacecraft safety, requiring extra-special handling measures and leak check monitoring. The high

weight required the use of hoisting equipment, which in turn added to concern about this instrument.

3. There are operating challenges:

Gradual increase in the operating voltage, and avoidance of discharges, common to all HV systems, were required with SEPICA. Management of the “autonomous” gas regulation system has proven to be a bit more hands on than expected. Apparently the silicon micro-valves are either sticking in a slightly open position, or their zero current state is closely matching the exchange rate of the counters. Close contact with spacecraft operating personnel was critical to insure setting of pressure alarms to protect the sensitive P/C windows. The interactive nature of the gas system is an aspect of the instrument that makes it more complicated and unique than others.

4. The primary risk with SEPICA was the presence of the Isobutane gas which presented a unique risk to personnel, spacecraft and facility safety. However the implementation of safety procedures and general high level of awareness ensured SEPICA was always handled and tested in a safe manner. One incident that almost jeopardized the instrument and facility safety was the handling of the Isobutane within the thermal/vacuum chamber testing. Inadvertently, the Teflon GSE lines handling the Isobutane exhaust were left exposed to the chamber LN2 shroud, resulting in cryo trapping (freezing) of the Isobutane during the first such test at GSFC. Gradual warming of the shroud and bleed off of the gas ensured that no damage was done, however it was very stressful for all involved. In subsequent tests, provisions were made to ensure the gas lines were safely blanketed from the shroud with thermocouples installed to ensure that constant monitoring of the lines was maintained. The lines were kept above -40c at all times. As an added measure of safety, heaters and thermostats were installed in case an inadvertently cold condition were to exist.

5. SEPICA did not require descoping. If it had been necessary there could have been a reduction in the number of detector fans and respective electronics, but at a serious compromising of performance.

7.1.4.3 Complexity Rating

SEPICA had a difficulty rating of 5, equating to high complexity. This high complexity resulted from:

1) Achieving the extremely tight machining tolerances needed for fabrication of entrance collimators. SEPICA pushed the envelope of technology on collimator aperture tolerances.

2) Achieving the needed tight regulation of low pressure isobutane gas flow. This self-contained Isobutane gas system increased its handling constraints.

3) Achieving satisfactory performance from the newly designed dE/dx proportional counter at the heart of the instrument.

4) SEPICA was a heavy instrument with some intricate assemblies, e.g., SSDs, multiwire frames. A lot of vibration effects analysis was required. Moreover, it contained two independent high voltage systems.

7.1.5 Solar Wind Ion Composition Spectrometer (SWICS) and Solar Wind Ion Mass Spectrometer (SWIMS)

SWICS and SWIMS are discussed together because they were developed simultaneously by the same Co-Investigator and development team. The Co-Investigator is George Gloeckler, who is affiliated both with the University of Maryland, College Park, and the University of Michigan, Ann Arbor.

7.1.5.1 Description

SWICS uses 4 components to study the mass and isotopic composition of incoming ions: a) E/q filtering by electrostatic deflection, b) post-acceleration of the filtered ions by up to 30 kV, c) TOF measurement, and d) E measurement using ion-implant solid-state detectors (SSDs). Knowing the E/q, E, and TOF, one knows the mass (M) and the mass per charge (M/q) since $E = (M/2)*v^2$. SWICS was the flight spare of the Ulysses GLG experiment, launched in October 1990. Figure 7.1.5-1 pictures the SWICS instrument.

SWIMS uses E/q selection by electrostatic deflection, along with TOF measurement. The nature of the electrostatic field in SWIMS allows mass measurement without the need for direct E measurement. SWIMS is similar to the MTOF sensor of the CELIAS experiment on the SOHO spacecraft, launched in December of 1995. Figure 7.1.5-2 shows the SWIMS instrument.

7.1.5.2 Challenges and Solutions

The technology advances for SWICS were accomplished as the Ulysses/GLG experiment. All that remained for ACE that was the least bit troubling was to develop a thermal design that kept the sensitive SSDs below 15C at all times.

The SWICS flight spare unit was inherited from the Ulysses project containing a surface coating problem that caused the instrument to produce output data long after an input test beam was turned off. Significant early work was devoted to solving this problem. For reasons of enhancing SWICS's science return, a high voltage transformer was redesigned to extend the instrument's upper operating voltage from 6 kilovolts to 10 kilovolts.

For SWIMS, by far the biggest challenge was creating a hyperbolic electrode that would maintain a uniform equipotential surface while, at the same time, be as transparent as possible in order to let the overwhelming number of neutral ions pass through it. Neutrals that do not pass through are liable to reflect onto the Stop microchannel plate (MCP) detector and cause obscuring background counts.

A problem with the Helium background encountered early in the SOHO mission led to significant redesign of SWIMS's entrance aperture. Also, the SWIMS redesign added redundant Start MCPs, and associated power supplies, for co-incidence.

The only operational complication was not a surprise at all, and is the fact that the high-voltages of both SWICS and SWIMS must be turned down for spacecraft maneuvers, since the hydrazine fuel used is extremely corrosive to the MCPs used in both sensors.

Also, since SWICS potentially has an electric field gradient of 30,000V over 6mm, locally increased pressure imposes a risk of discharge.

Throughout the development of SWICS and SWIMS there were a number of risks inherent to the fragile nature of the components, but probably the biggest risk was possible damage to the MCPs or SSDs. The SSDs in SWICS had to be kept below 25C at all times, and preferably colder. This posed the problem, for example, of airline transport, where the item could not be left in a customs warehouse which is not air-conditioned. Since the MCPs are also very sensitive to humidity and ACE launched in August at the Cape, it was important to maintain continuous N2 purge until launch.

No descoping was done at all. If necessary there could have been descoping in the early stages, for example, by not upgrading the SWICS E/q power supply, compromising very little performance. For SWIMS, the redundancy in the power supplies for the MCP assemblies could have been removed.

7.1.5.3 Complexity Rating

Although SWICS was an inherited flight spare, the modifications were significant enough to warrant difficulty ratings from 2 to 3, for a complexity between medium and medium-low.

Ratings for SWIMS ranged from 3 to 4, or medium to medium-high complexity.

FIGURE 7.1.5-1: SWICS Instrument

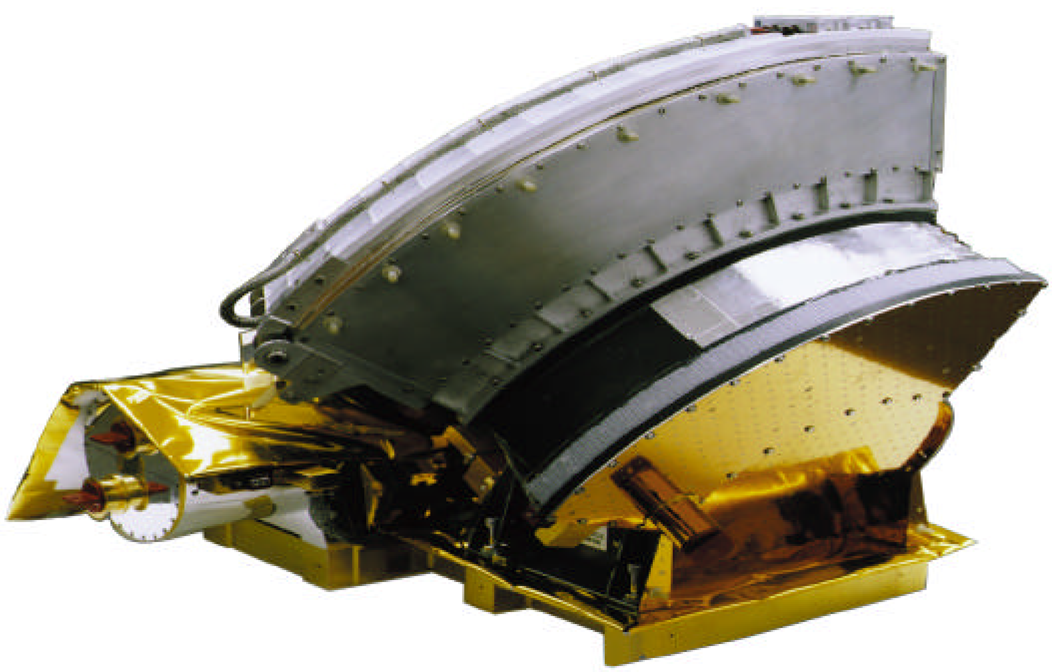
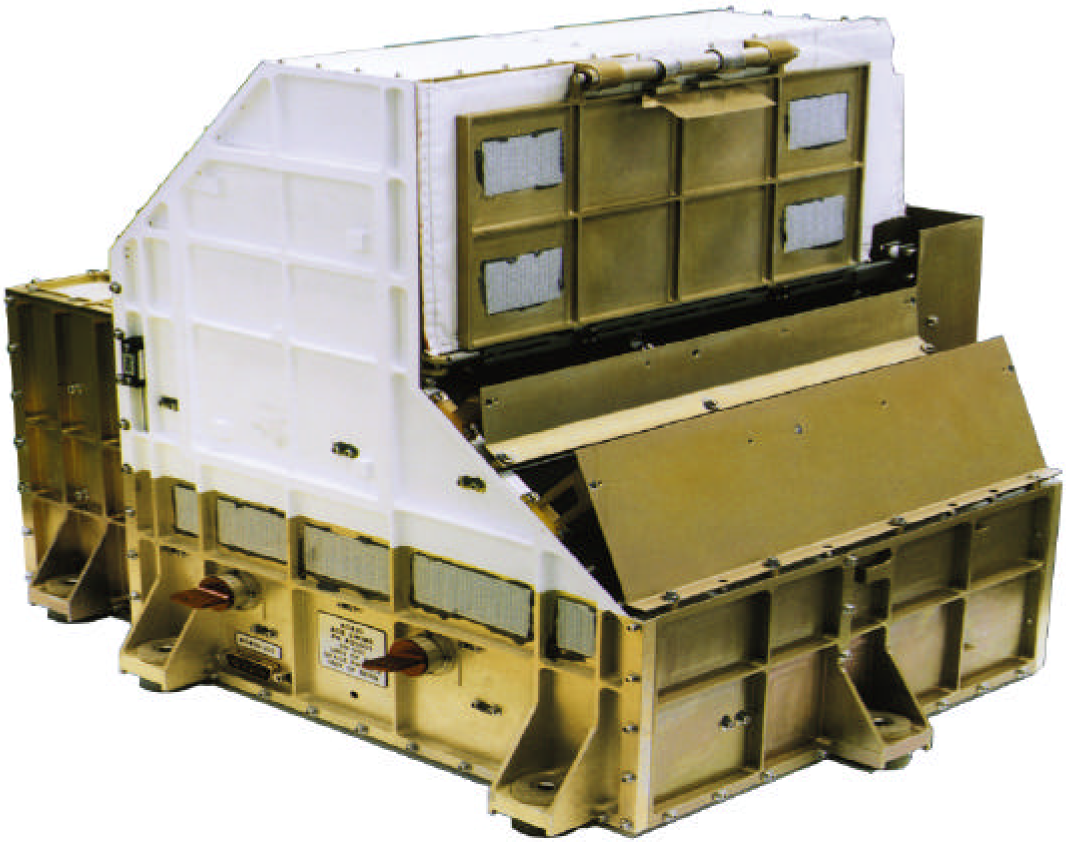


FIGURE 7.1.5-2: SWIMS Instrument



7.1.6 Electron, Proton and Alpha Monitor (EPAM)

7.1.6.1 Description

EPAM was built by the APL. Robert Gold is the Co-Investigator. EPAM is the flight spare of the HI-SCALE instrument (also known as LAN) from the Ulysses spacecraft.

EPAM uses low energy solar particle fluxes as probes of the morphological changes of coronal and large-scale interplanetary magnetic field structures. EPAM also investigates solar flare processes using non-relativistic and relativistic electrons. Also, it provides insight into particle energization in interplanetary space.

The EPAM development effort consisted of reconstituting EPAM from a fully qualified and recently checked flight spare instrument, moving the sensors from the HI-SCALE electronics box to a separate pedestal, designing a reasonably straightforward spacecraft command and telemetry interface adapter, and revising the GSE to meet ACE interfaces.

EPAM consists of five apertures in two telescope assemblies mounted by means of two stub arms and a box containing all of the instrument electronics. The instrument is shown in Figure 7.1.6-1. It measures ions ($E_i \geq 50$ keV) and electrons ($E_e \geq 30$ keV) with essentially complete pitch angle coverage from the spinning ACE spacecraft. It also has an ion elemental abundance aperture using the ΔE vs. E technique in a three-element telescope. The detectors form three distinct silicon solid-state detector systems. These are Low Energy Magnetic Spectrometers (LEMS), Low Energy Foil Spectrometers (LEFS) and the Composition Aperture (CA). The LEMS/LEFS systems provide pulse-height-analyzed single-detector measurements with active anticoincidence. The CA provides elemental composition in an energy range similar to those of LEMS/LEFS, and provides ^3He/ ^4He isotope resolution for identifying ^3He-rich events.

The five separate detector systems are contained within two mechanical structures, as shown in Figure 7.1.6-2. The individual telescopes are referred to as LEFS 60, LEFS 150, LEMS 30, LEMS 120, and CA 60, where the number indicates the inclination of the telescope axis with respect to the spacecraft spin axis. The MF, M'F' and BC detector pairs are identical for ease of replacement. Each consists of two 200 μm silicon surface barrier detectors. The D detector is a thin (5 μm) silicon detector of the epitaxial type.

An overall schematic block diagram of the EPAM electronics system is shown in Figure 7.1.6-3. The detector signals are fed to charge-sensitive preamplifiers, linear shaping amplifiers, and then to discriminators. The analog electronics are largely of hybrid construction. In addition to the preamplifier, amplifier and discriminator circuits, these hybrids include logarithmic amplifiers, accumulators, one-shots, peak detectors, power controllers, and modules required for the pulse-height-analyzer system.

EPAM has a dedicated processor. The instrument is under the overall control of a radiation-hardened microprocessor and its associated peripheral devices. The system uses 6K ROM and 3K RAM memory. Provision is made for recovery after a partial failure by loading a block of program steps into RAM and then setting this block into the normal program execution sequence. The science data are collected over a cycle of ten spacecraft spins (~120 s) and output to the spacecraft, time synchronized with the major and minor spacecraft frame counters. Since some of the EPAM interface signals were unique to the Ulysses spacecraft, a small adapter box was added to ensure compatibility with the ACE command and data handling system formats.

7.1.6.2 Challenges and Solutions

Although the development effort consisted of modifications to an existing flight unit, there were some challenges:

1) Relocation of this Ulysses flight spare instrument's sensors to meet the ACE science requirements, while avoiding the need for new preamplifiers, was a challenge. Eventually it was met by combining the sensor brackets with those used for mounting the spacecraft digital sun sensors,

2) A special command and data interface box had to be built for accommodation with the spacecraft, and

3) One of the old Ulysses flight detectors failed during testing and had to be replaced.

7.1.6.3 Complexity Rating

The ratings for difficulty ranged from 1 to 3, or from easy to average. This gives an average of medium-low complexity.

FIG. 7.1.6-1: EPAM Instrument

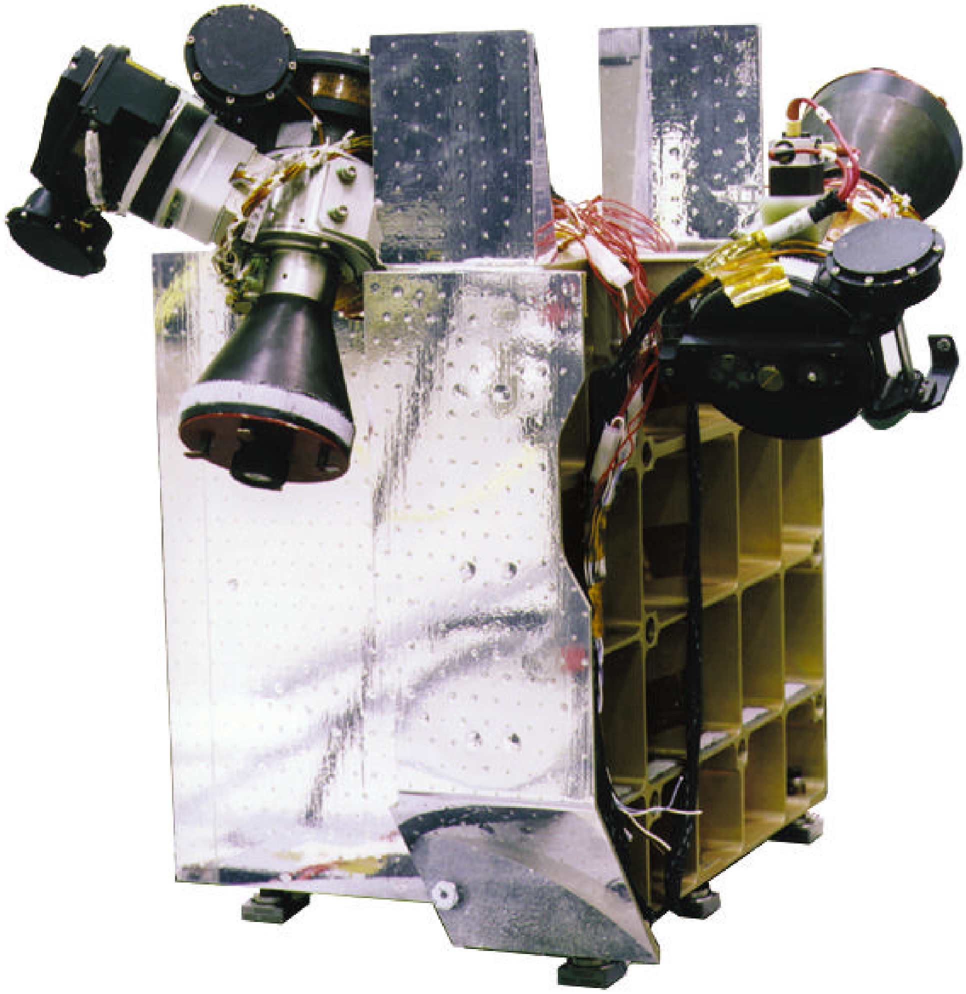
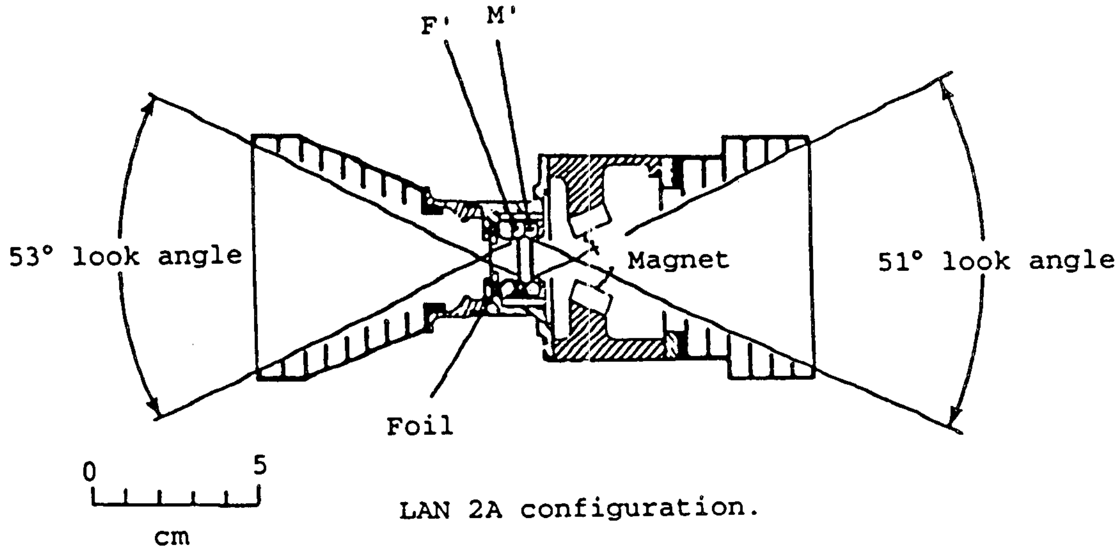
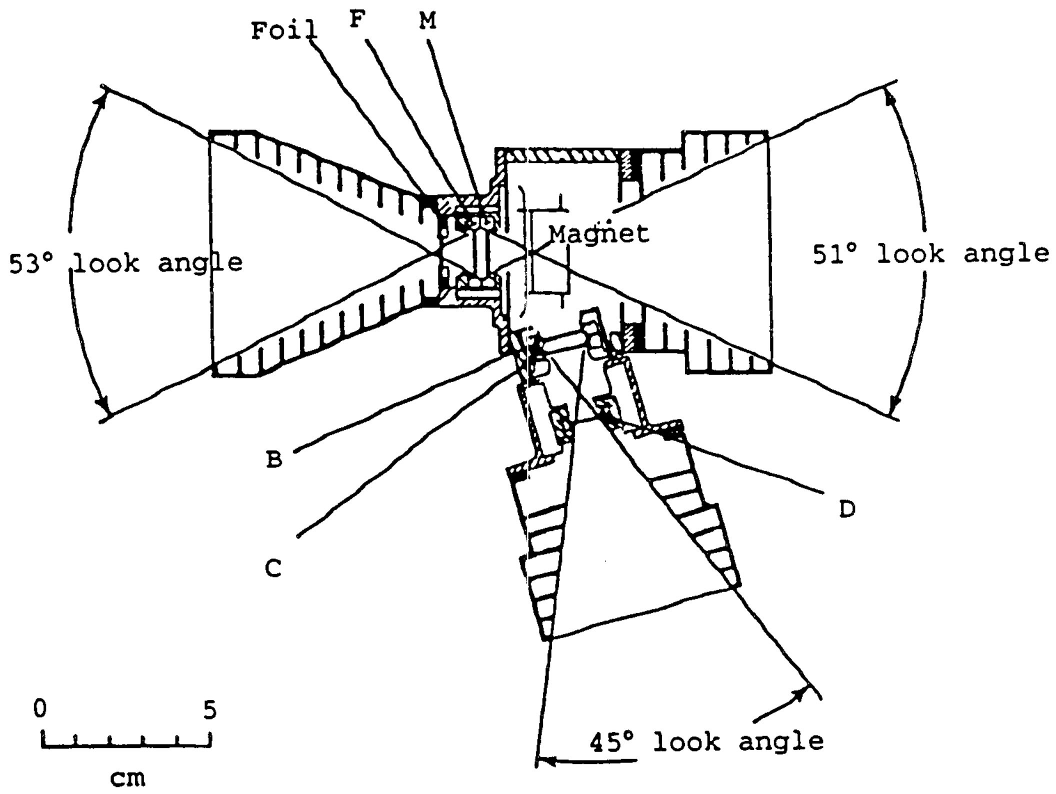


Figure 7.1.6-2: EPAM

Schematic Outline of Detector Telescope Configurations in the Two Separate Mechanical Mounts



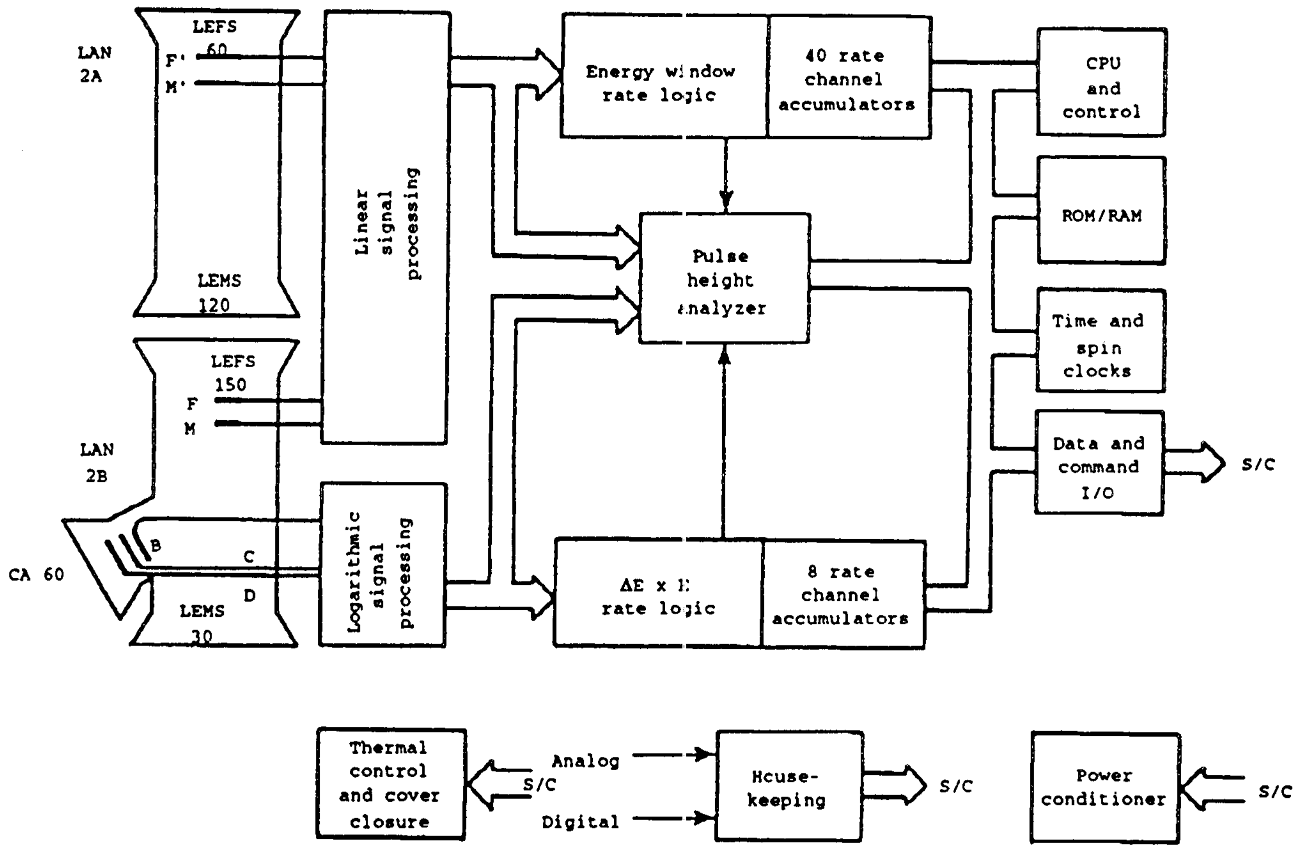
LAN 2A configuration.



LAN 2B configuration.

Figure 7.1.6-3

Block Diagram of EPAM Electronics System



7.1.7 Solar Wind Electron, Proton and Alpha Monitor (SWEPAM)

7.1.7.1 Description

The SWEPAM Co-Investigators are D. J. McComas and W. C. Feldman of the Los Alamos National Laboratory.

SWEPAM will provide detailed knowledge of solar wind conditions. It provides high quality measurements of electron and ion fluxes in the low-energy solar wind range (electrons: ~1 to ~1450 eV; ions: ~0.26 to 35 kV). SWEPAM is a plasma analyzer and consists of two instruments, SWEPAM-I for ion measurements and SWEPAM-E for electron measurements. A curved plate electrostatic analyzer (ESA) is followed by channel electron multiplier (CEM) detectors. The multiple CEMs provide spatial sampling in elevation while the spacecraft spin provides spatial sampling in azimuth. A microprocessor controls ESA high voltage supply, data acquisition, and command and data handling.

The two SWEPAM instruments are the Ulysses BAM-I and BAM-E flight spare instruments, adapted for ACE requirements: 1 Astronomical Unit (AU) instead of the Ulysses 5 AUs, foregoing the Ulysses data processing unit, separating the two sensor heads into separate autonomous units, and adding smart interfaces to each. The breadboards, prototypes and spare parts were transferred along with the flight spare instruments from the Ulysses project to the ACE project. Figure 7.1.7-1 provides mostly a front view of SWEPAM, and Figure 7.1.7-2 gives a side view.

7.1.7.2 Challenges and Solutions

Although the technology developments had been accomplished for Ulysses, there were areas new to the SWEPAM development team:

1.) Space electronics design implementation in Field Programmable Gate Arrays (FPGAs) using VHDL.

2.) Force limited vibration testing to avoid excessive loads on the heritage SWEPAM hardware.

In terms of manufacturing areas of difficulty, retrofitting the instruments with minimum impacts to the heritage hardware, having few flight spare parts, and with a tightly constrained budget presented challenges.

The ACE project permitted printed wiring board fabrication to the MIL-P-55110D requirements with product acceptance contingent upon an acceptance inspection by GSFC. The differences between MIL-P-55110D acceptance inspection processes and the GSFC acceptance inspection processes resulted in conflicts. The instrument team believes MIL-P-55110D to be good manufacturing specifications for ACE-like missions, but in the future this potential area of conflict needs to be addressed in a way to minimize impacts to schedules, costs and supplier relations.

In terms of Observatory integration, the most difficult interface task was identifying and procuring an electrically conductive multi-layer insulation (MLI) blanket which satisfied the instrument requirements for spacecraft charging control while at the same time meeting Observatory thermal, contamination and magnetic control requirements. The resolution was an Indium Tin Oxide (ITO) coated beta cloth. The final flight blanket was installed late in the integration to preclude degradation in the coating integrity by handling of the blanket during Observatory I&T.

To minimize impacts to the heritage hardware, work-arounds had to be found for a few standard interface control and performance assurance issues in order to control costs while maintaining acceptable risk exposure. The lack of flight spares for many of the flight critical items was an additional driver. In general, all parties on ACE worked these kind of issues with good professional attitudes. These issues include:

- EMC waivers for conducted emissions and turn-on transients.
- Reduction of test exposure levels to minimum safe levels.
- High voltage operation and contamination control constraints to preclude damage to both sensors and supplies for which there were no spares.
- Use of a thermally coupled interface for which the heritage hardware had been designed when the ICD preference was for isolated interfaces.

There were problems with the project-provided spacecraft simulators. Nevertheless, the ACE project's decision to provide each instrument team with a standard spacecraft simulator was a good one. Part of the problem on ACE was that the use of the standard was optional, which diluted user review of the simulator design and may have caused follow-up support to suffer.

In terms of descoping possibilities, if needed, the only descoping that could have taken place was to forgo the retrofit of the processing electronics. Besides complicating the adaptation of the instrument to the ACE spacecraft interface, this would have severely compromised the instrument performance.

7.1.7.3 Complexity Rating

Although the SWEPAM instruments started as inherited flight spares, the extensive modifications kept the level of difficulty at the 3 rating, or medium complexity.

Figure 7.1.7-1: SWEPAM -- Mostly Front View

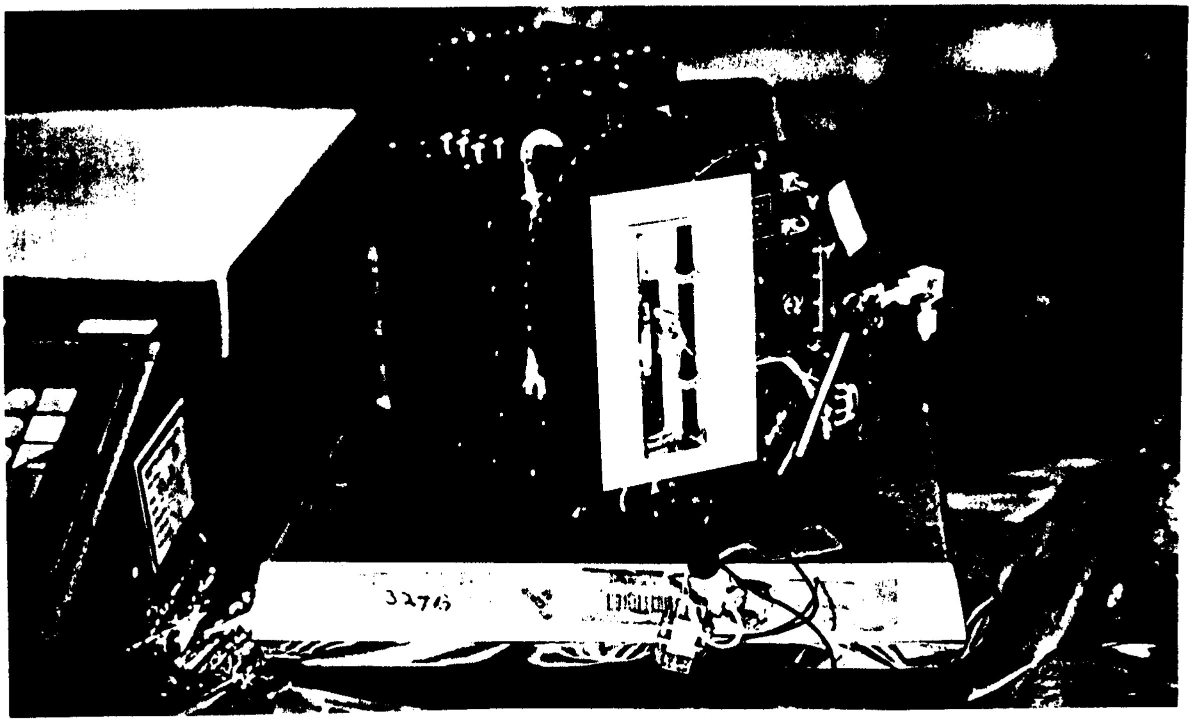
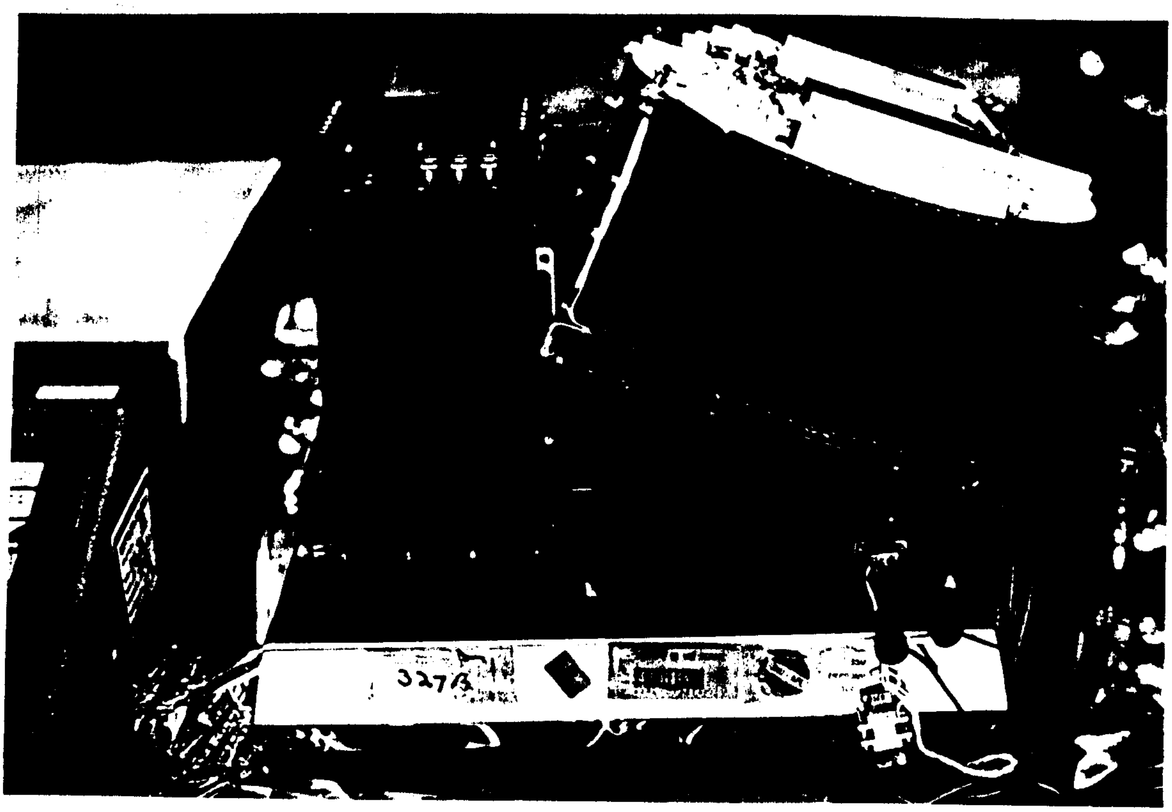


Figure 7.1.7-2: SWEPAM -- Side View



7.1.8 Magnetic Field Monitor (MAG)

7.1.8.1 Description

MAG was developed by Bartol Research Institute, the University of Delaware and GSFC. The Co-Investigator is Norman Ness. MAG is the reconditioned flight spare of the WIND/MFI experiment.

The instrument consists of a pair of twin, wide-range (+/- 0.001 to +/- 65,536 nT) triaxial fluxgate magnetometers mounted on two deployable titanium booms. Both sensors are deployed 165 inches (or 4.19 meters) from the center of the spacecraft (along the +/-Y axes) on opposing solar panels. The electronics and digital processing unit are mounted on the top deck of the spacecraft. The two triaxial sensors provide a balanced, fully redundant vector instrument and permit some enhanced assessment of the spacecraft's magnetic field. The instrument provides data for Browse and high-level products with 3 to 6 vector s^{-1} resolution for continuous coverage of the interplanetary magnetic field. Two high-resolution snapshot buffers each hold 297 seconds of 24 vector s^{-1} data while on-board Fast Fourier Transforms extend the continuous data to 12 Hz resolution. Real-time observations with one second resolution are provided continuously to the Space Environmental Center of the NOAA for near-instantaneous, world-wide dissemination in service to weather studies. High instrument reliability is obtained by the use of fully redundant systems and conservative designs.

Table 7.1.8-1 summarizes the principal characteristics of MAG, and Figure 7.1.8-1 provides a block diagram. Each sensor assembly consists of an orthogonal triaxial arrangement of ring-core fluxgate sensors plus additional elements required for autonomous thermal control. The fluxgate sensors are the latest in a series developed for weak magnetic field measurements. Figure 7.1.8-1 provides a schematic of standard fluxgate operation.

7.1.8.2 Challenges and Solutions

Through clever modification of interface circuitry internal to the WIND flight spare magnetometer, it became possible to adapt this unit to the ACE spacecraft without using a separate adapter box at first envisaged. However, cramming more circuitry into the MAG electronic box led to a vibration-induced problem caused by transformer insulation wearing through and shorting. An appropriate remedy was identified and implemented,

A misunderstanding of the sensor mechanical mounting interfaces caused the polarity of two axes to be inverted on both sensor assemblies. Changing the polarity of these two axes within the instrument rectified the situation.

Since most of the hardware had been previously tested for the WIND instrument, several environmental tests could not be performed at the ACE stated level or had to be performed at a lower to prevent over-stressing of the hardware. This led to several waivers. No failures occurred.

A manufacturing area of difficulty was the backwiring of solar arrays to reduce magnetic noise.

The sensor pointing requirements imposed strict requirements on the boom orientation after deployment.

MAG did not require descoping, but if it had been necessary, the onboard FFT and buffers could have been removed. Also, only one sensor could have flown, replacing the missing sensor with dead weight to balance the spacecraft.

7.1.8.3 Complexity Rating

MAG had an average difficulty rating of 2, which makes it a medium-low complexity.

Table 7.1.8-1: Summary of MAG Characteristics

Instrument type:	Twin, triaxial fluxgate magnetometers (boom mounted)
Dynamic ranges (8):	±4 nT (Range 0); ±16 nT (Range 1); ±64 nT (Range 2); ±256 nT (Range 3); ±1024 nT (Range 4); ±4096 nT (Range 5); ±16,384 nT (Range 6); ±65,536 nT (Range 7)
Digital Resolution (12-bit):	±0.001 nT (Range 0); ±0.004 nT (Range 1); ±0.016 nT (Range 2); ±0.0625 nT (Range 3); ±0.25 nT (Range 4); ±1.0 nT (Range 5); ±4.0 nT (Range 6); ±16.0 nT (Range 7)
Bandwidth:	12 Hz
Sensor noise level:	< 0.006 nT RMS, 0-10 Hz
Sampling rate:	24 vector samples/s in snapshot memory and 3, 4 or 6 vector samples/s continuous data stream
Signal Processing:	FFT Processor, 32 logarithmically spaced channels, 0 to 12 Hz. Full spectral matrices generated every 80 seconds for four time series (B_x, B_y, B_z, $\|B\|$)
FFT Windows/Filters:	Full despin of spin plane components, 10% cosine taper, Hanning window, first difference filter
FFT Dynamic range:	72 dB, μ-Law log-compressed, 13-bit normalized to 7-bit with sign
Sensitivity threshold:	$\sim 0.5 \times 10^{-3}$ nT/Hz in Range 0
Snapshot memory capacity:	256 Kbits
Trigger modes (3):	Overall Magnitude Ratio, Directional max.-min. peak-to-peak change, Spectral increase across frequency band (RMS)
Telemetry Modes:	Three, selectable by command
Mass:	Sensors (2): 450 g. total Electronics (redundant): 2100 g. total
Power Consumption:	2.4 watts, electronics - regulated 28 Volts ± 2% 1.0 watts, heaters - unregulated 28 Volts

Figure 7.1.8-1: MAG Block Diagram

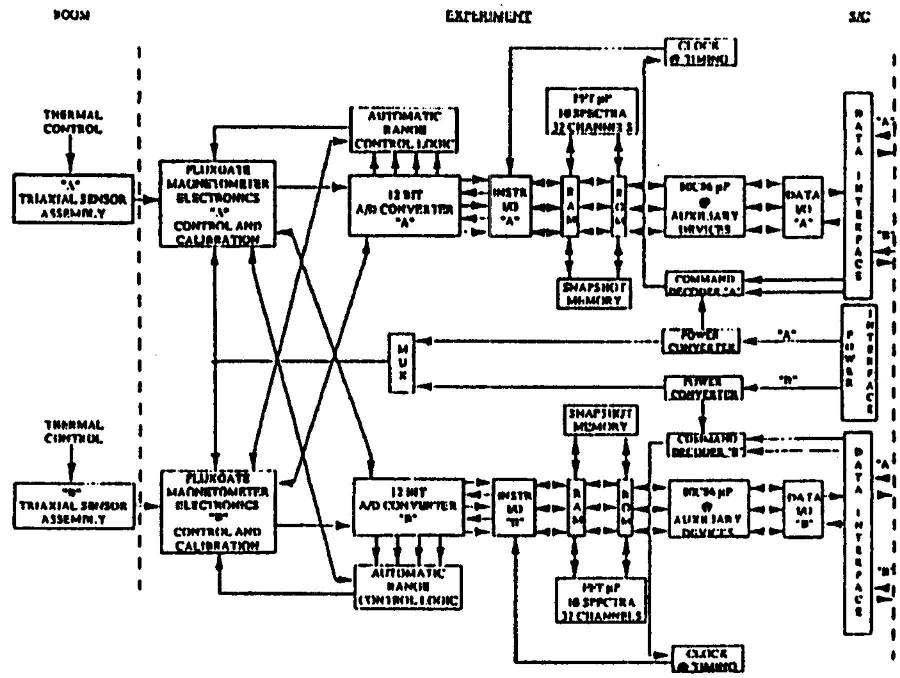
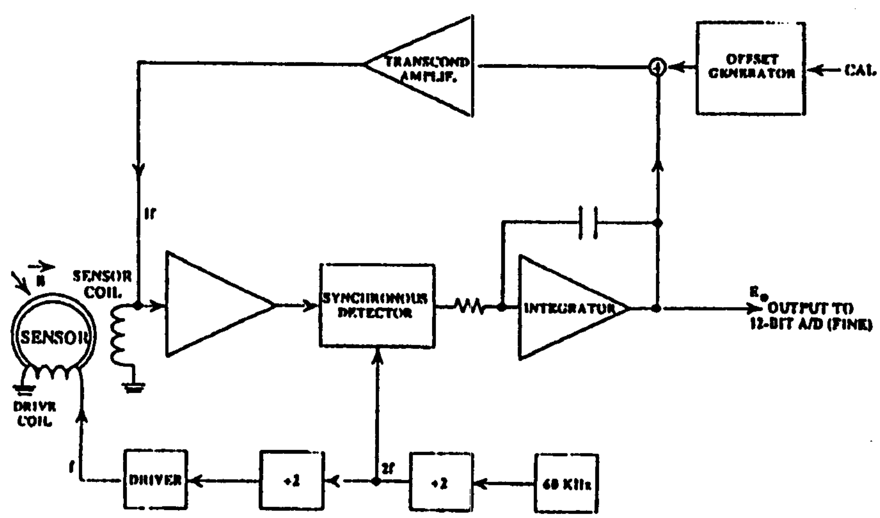


Figure 7.1.8-2: MAG -- Schematic of Standard Fluxgate Operation



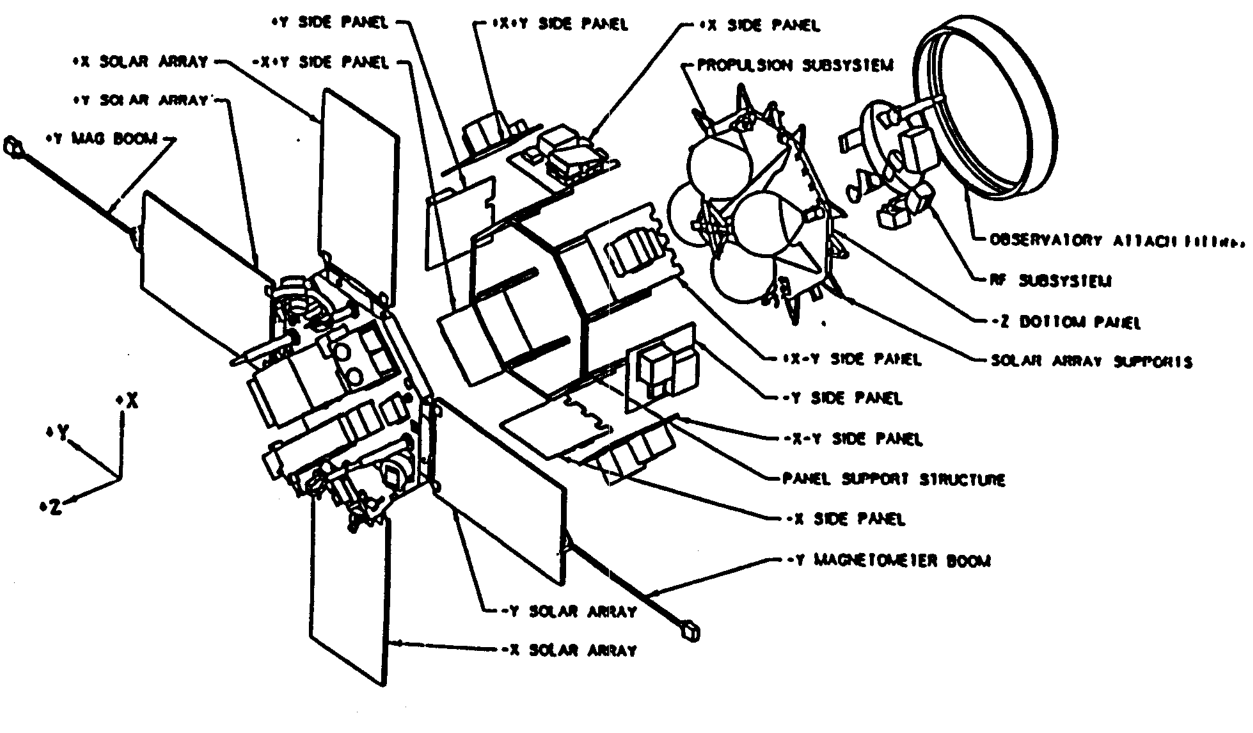
7.2 SPACECRAFT

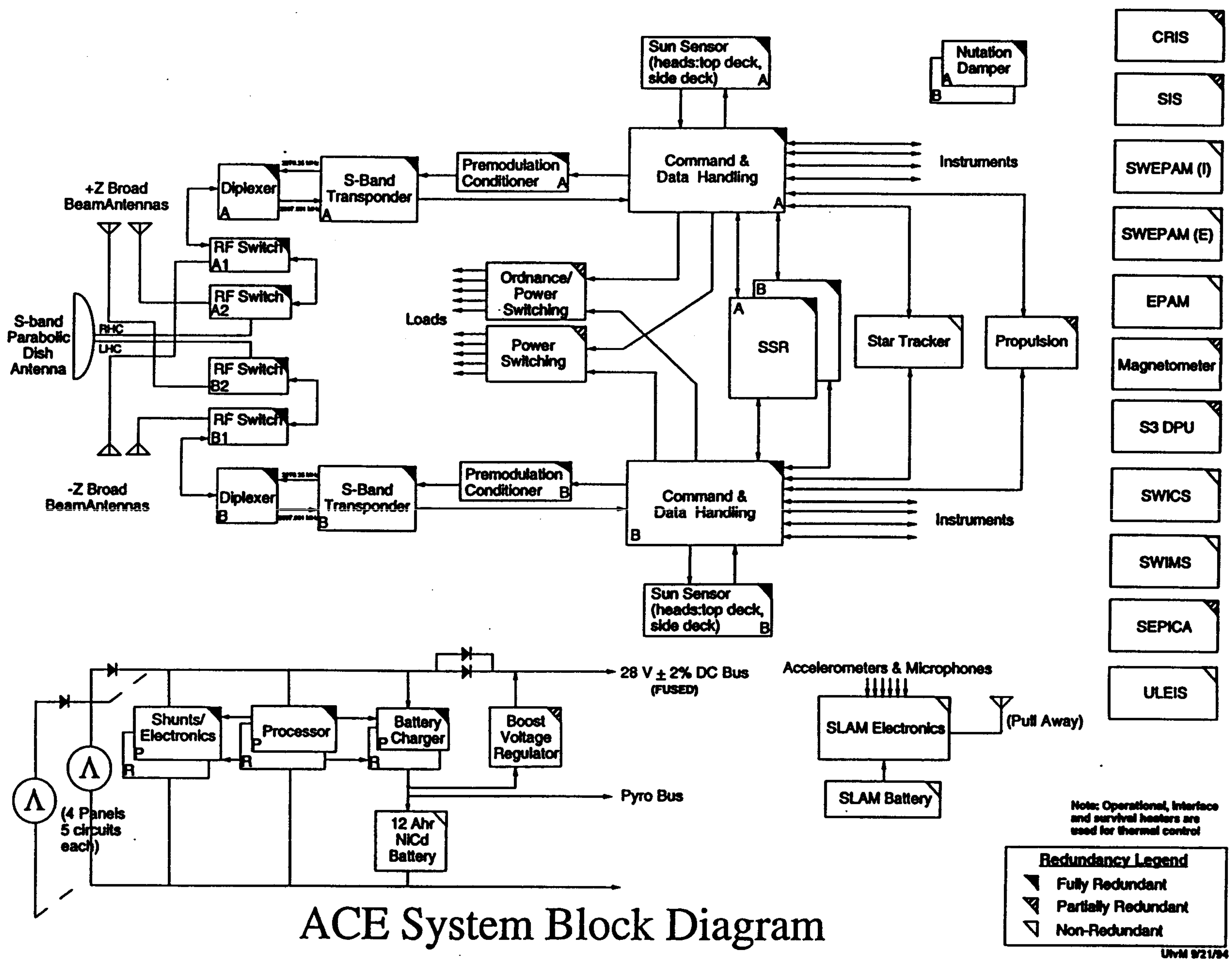
This section provides additional descriptions and discussions of the ACE spacecraft. Some of the material was extracted from a paper to be published: M. C. Chiu <u>et al</u>, <u>ACE Spacecraft</u>.

The Johns Hopkins University Applied Physics Laboratory (APL) was responsible for the design and fabrication of the ACE spacecraft and for the integration, test and launch support for the entire ACE Observatory. The primary ACE mission includes a significant number of science instruments whose diverse requirements had to be factored into the overall spacecraft bus design. Secondary payloads were also accommodated within the spacecraft design. Substantial coordination and cooperation were required among 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.

Figure 7.2-1 provides an exploded view of the spacecraft, and Figure 7.2-2 gives the ACE system block design. Table 7.2-1 lists technical parameters of the ACE spacecraft.

Figure 7.2-1: ACE Spacecraft Exploded View





ACE System Block Diagram

Figure 7.2-2

Table 7.2-1: Technical Paramaters of ACE Spacecraft

Attitude	***Star tracker*** • 20° × 20° field of view • +0.1 to +4.5 sensitivity range • 30 arcsec (1σ) total random error • 1 to 5 simultaneous stars ***Sun sensor*** • ±64° field of view • Sun angle is gray-coded in 0.5° increments • ±0.02° short-term repeatability of most significant bit ***Observatory orientation*** • Known (after the fact) to ±0.7°; stable to ±0.5° ***Pointing at Earth*** • Angle between the observatory Z axis and the Earth–observatory line is within ±3°, to stay within the required beamwidth of the high-gain antenna • Angle between the Sun–Earth line and the observatory–Earth line is ≥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 blowdown (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
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)

Table 7.2-1 Continued

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 • power subsystem processor reset
Observatory power	280W 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	756.54kg (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
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)

7.2.1 Mechanical

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 resulted from the need for individual thermal radiators on the instruments and structure, which required views to space, and by handling, access, clearance and harness routing issues.

The primary structural components consist of ten aluminum honeycomb panels that form a 142.2 x 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 primary structure to a 5624 Delta payload fitting with a clampband. Most of the instruments (SEPICA, SIS, SWICS, SWEPAM, 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 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 payload.

Propulsion 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 with the rest of the spacecraft primary structure. The tanks were 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.

Four 86.4 x 149.9 cm deployable aluminum honeycomb solar panels are hinged from the +/-X and +/-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 +/-Y solar panels. For launch, the +Z end of each boom was restrained to the +/-Y solar panels with a pin puller mechanism. Pyrotechnic devices actuate the pin pullers, which release the solar panels and booms. Preloaded torsion springs deployed and centered 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 the Delta payload attach fitting. The spacecraft primary structural modes were measured at about 100 Hz in the thrust direction and 40 Hz in the lateral directions.

The Delta II flight events produce loads from steady-state and dynamic environments. 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 was used to verify the observatory strength and stiffness. Vibration (sine and random), acoustic, and shock (clampband separation) testing were performed on the fully flight configured observatory, using a flight like payload attach fitting. This was the first time that Boeing allowed its attach fitting hardware to be used in this manner.

In terms of complexity, the structure and mechanisms were rated as 3 and 4, for an average of 3.5. This is a shade less than a medium-high complexity. Although mechanisms and structure for ACE were based on previous spaceflight designs, there was significant tailoring. With so many instruments on board, the field of views, radiators, access considerations, and balancing for a spinning spacecraft provided significant challenges.

7.2.2 Thermal Control

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, along with stringent and differing instrument temperature requirements, made thermal isolation the best approach. Instruments and instrument components whose temperature requirements could be managed by the spacecraft (SWEPAM, 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 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 ACE MLI blankets have an outer layer of white Beta Cloth to provide durability and the desired optical properties while minimizing specular reflections. There also is an embedded graphite weave to provide some protection from electrostatic discharge, although a conductive spacecraft was not a requirement for ACE.

In some areas, special requirements dictated the use of alternative materials instead of the MLI outer layer. Possible thruster plume impingement led to the use of MLI blankets with aluminized Kapton layers to provide hardening to increased temperatures. Also the SWIMS and SWEPAM(E) instruments had special requirements that dictated the use of more electrically conductive MLI outer layer materials. ITO-coated aluminized Kapton was used.

The forward (+Z) deck is covered by thermal insulation to shield it from the Sun as much as possible. Heat is rejected from the forward deck via silvered teflon radiators attached to each of the eight deck edges.

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 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 thermal design minimizes heat exchange between the side panels and space by covering the side panels with MLI blankets. For side panels with instruments, the blankets provide radiative isolation between the instrument and side panel.

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.

The observatory thermal analysis included worst-case variations in all thermal design parameters. Therefore, the analysis bracketed all possible temperature variations over the entire mission life. On orbit, the spacecraft thermal environment shows little short-term variation. The sun angle variation occurs on a scale of months, and will be corrected periodically via the propulsion system.

The thermal control system is robust and versatile. A primary requirement of the 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 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. This would have resulted in higher peak operational heater power.

In terms of complexity, thermal control was rated as 3 and 4, for an average of 3.5. This is a shade less than a medium-high complexity. Although the spacecraft has a steady environment for operations, with so many instruments on board having differing requirements, designing for all possible situations provided a challenge for a passive low-cost implementation.

7.2.3 Command and Data Handling (C&DH)

The 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 one-Gbit solid state recorders, a Power Switching component, and an Ordnance Fire component. These components were designed and fabricated by APL, except for the solid state recorders (SSRs) which were designed and fabricated by SEAKR Corporation. Special design features are included to avoid any single-point mission failures and to allow continuous operation through solar flares. Table 7.2.3-1 provides major requirements of the C&DH subsystem. Table 7.2.3-2 covers C&DH user commands, Table 7.2.3-3 gives telemetry channels and Table 7.2.3-4 has the telemetry formats. Figure 7.2.3-1 gives a block diagram showing the major components of the C&DH subsystem and the other systems with which it interfaces.

The C&DH components comply with the Consultative Committee for Space Data Systems (CCSDS) uplink and downlink standards. They can execute commands in real time from the ground, store commands for execution based on time and out-of-limit conditions, and store commands as part of a block of commands. Command outputs include serial data commands, logic pulse commands, and relay commands. The C&DH components collect data using a series of standard interfaces from instruments and other subsystems. Special purpose interfaces are used to collect data from sun sensors. Sun pulses and spin clocks are generated and distributed to instruments. The C&DH components also control thruster firings.

The C&DH component was a new design that utilizes a Harris RTX2010 processor executing the FORTH language. The RTX2010 is fabricated in a CMOS/SOS process that is exceptionally hardened to single-event upsets, which makes it suitable for operation through solar flares. To be hardened the C&DH component design included EDAC on memory, hardened logic and triple voting modules in Actel field-programmable gate arrays (FPGAs). Spot tantalum shielding was used on a few parts.

The SSRs 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 anticipated for the ACE mission. Data is stored using IBM 16-Mbit dynamic random access memories (DRAMs) for storage and a Harris 80C85 microprocessor for control. Operationally data are recorded at an average rate of 6,944 bps and played back at an average rate of 68,640 bps. The SSRs are capable of operating much faster than this. The SSR design was loosely based on those for the Clementine spacecraft.

The Power Switching and Ordnance Fire components were 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.

Some additional lessons learned are:

- In terms of development, a difficulty was gaining sufficient confidence to fly DRAMs in plastic packages. It was difficult to find a test house that could properly test and screen the parts.

- In terms of fabrication, the chassis design of the C&DH component reduced costs compared to earlier programs by milling the chassis out of a single block of metal, instead of individually milling out each side and assembling the sides together. This minimized the number of drawings required and minimized concerns about tolerances and error buildup in assembly.

- The design of the NEAR spacecraft was concurrent with ACE. The two C&DH subsytems were designed with as much commonality as possible. This allowed the design costs of many boards to be split between the two programs.

- A significant effort was put into testing C&DH software. This uncovered most software bugs early and minimized problems during spacecraft integration and test.

In terms of difficulty, the C&DH rated as a 3, categorizing it average difficulty or medium complexity.

Table 7.2.3-1: Major Requirements of 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 & 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 & 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)

Table 7.2.3-2: 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 7.2.3-3: C&DH Subsystem Telemetry Channels

Telemetry Channel Type	Number Provided	Comment
Serial digital	16	Includes major & 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° to +100°C
PT103 temperature transducer	62	Measures temperatures in range of –100° to +150°C
Digital telltale-logic	32	Used primarily for RF transponder telemetry
Digital telltale-switch	16	Used primarily to detect the state of switches

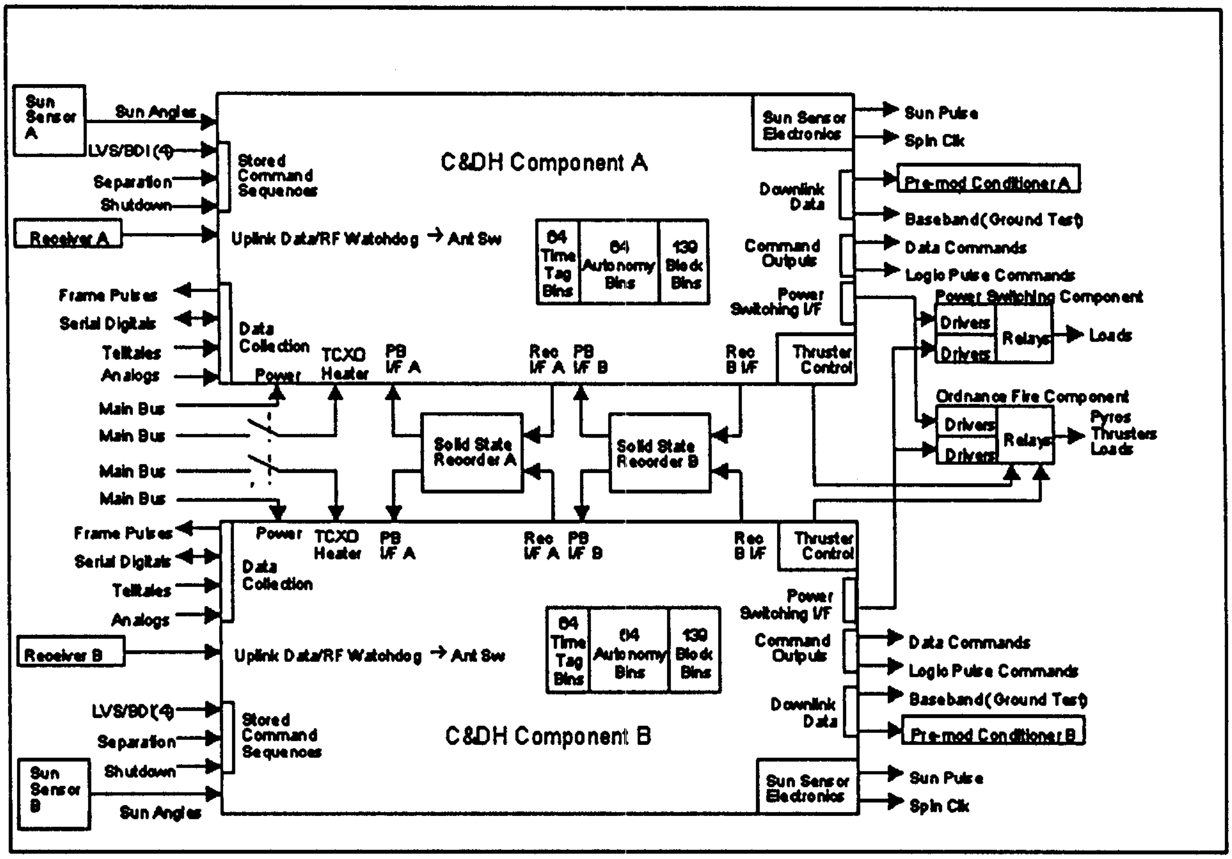
(Note: C&DH = command and data handling)

Table 7.2.3-4: C&DH Telemetry Formats

Downlink and Record Telemetry Formats	Rate (bps)	Description
Attitude determination and control	6944	Contains all housekeeping and AD&C 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 AD&C data and most housekeeping data
Real-time solar wind	434	Includes science data for real-time evaluation of solar wind

(Note: AD&C = attitude determination and control, C&DH = command and data handling, SSR = solid-state recorder)

Figure 7.2.3-1: C&DH Block Diagram



7.2.4 RF Communications

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 bit/s, to Earth stations supporting the NOAA Real-time Solar Wind (RTSW) 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 broadbeam 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) broadbeam antenna or to the aft high-gain parabolic antenna. Watchdog timers, implemented in software within the C&DH subsystem, are designed to switch the broadbeam 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.

The communications subsystem concept was similar to those in the AMPTE Charge Composition Explorer (CCE) and the Midcourse Space Experiment (MSX) spacecraft. Similar Loral-Conic S-band transponders have been flown previously on TIROS, Clementine and other missions. However, 75 percent of the electronics design was new.

There was some descoping. The RF power radiated by the spacecraft was cut 50 percent early in the program to eliminate the cost of an RF power amplifier. The size and gain of the high gain antenna was increased to partially compensate for the power loss. Since the beamwidth of the antenna decreases as the size increases, there was a reduction in the angular coverage of the antenna. The result is that the attitude of the spacecraft must be perturbed more frequently to keep the earth within the antenna beam. These perturbations have some impact on the collection of data by the science instrumentation.

In the original concept, the RTSW experiment had an independent X-band communications system that could operate continuously with relatively small earth terminals. The final RTSW system takes advantage of the on board S-band telecommunications system to transmit RTSW data about 21 hours per day. Ground stations require large antennas (up to 10 meters depending on sensitivity) to support this system. Ground station support is provided by NOAA, the Air Force (AFSCN) and foreign ground stations, but none of this support is continuous. As a result there are periods when the solar flare activity is not reported.

The RF Communications subsystem was designed to be straightforward, which resulted in an average difficulty rating of 3, or medium complexity.

7.2.5 Attitude Determination and Control (AD&C)

The AD&C 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 AD&C 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 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°/s, which is two orders of magnitude greater than nominal star tracker angular rates, with no significant impact on the error budget (one sigma error of 30 arcsec). 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 ±0.7° after the fact, with a goal of ±0.5° for the magnetometer. The spacecraft components were assigned budgets for attitude errors, which are given in Table 7.2.5-1.

Table 7.2.5-1
Observatory Instantaneous Attitude Error Budget

Parameter	Angular Error (degrees)
Star scanner accuracy	0.025 (3 sigma)
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

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°/day 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.

Although the AD&C subsystem is very simple, reliable and straightforward, it had innovative applications. The Ball-built star tracker/scanner is the first application of a solid-state star tracker in a high-rate scanning application. The time delay integration technique used in the ACE tracker is believed to be the first use of this in flight.

Also, the innovative use of two-axis digital sun sensors on the spinning spacecraft solved a potential problem with sun sensor accuracy that would have existed had a traditional sun sensor been used. With a traditional spinning sun sensor and ACE's attitude orientation, the accuracy of the sensor would have been greatly degraded due to the sun being very near the edge of the field of view (FOV). Putting a two-axis digital sensor on the sunward facing side of the spacecraft provided continuous sun angle information in the center of the FOV of the sensor, rather than at the edge of the FOV.

The design of the sun sensors by Adcole was the same as used on the MSX and NEAR spacecraft. The design of the star scanner was the first of a new design by Ball based on their existing line of star trackers. Software changes were required to provide the time delay integration function. The star tracker is similar to the one flown on the NEAR spacecraft. The nutation dampers were a new design based on an ISEE-3 approach.

A difficulty in integrating the AD&C subsystem with the satellite was that there was no way to test the star scanner in its operational scanning mode because of the problem in providing an accurate moving star source. The star scanner was tested on a rate table at Ball, and at APL it was tested outside of the spacecraft with a rotating mirror. However, once the star scanner was integrated, the scanning function could not be verified until after launch.

Despite the complexity raised by the star scanner, the overall level of difficulty of the AD&C subsystem was rated at a 2, or a medium-low complexity. This is because ACE used a reliable, flight proven design strategy to produce a fairly simple subsystem. The open-loop ground-commanded design places the burden on the flight controllers to maneuver the spacecraft and determine its attitude.

7.2.6 Propulsion

The propulsion subsystem corrected launch vehicle dispersion errors, injected the spacecraft into the L1 halo orbit, adjusted the orbit, adjusts spin axis pointing, and maintains a 5 rpm spin rate. The subsystem is a hydrazine blowndown unit that uses nitrogen gas as the pressurant and comprises 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 7.2.6-1 presents a schematic of the subsystem and illustrates the cross-strapping between the A and B side tanks, service valves, and thrusters. Figure 7.2.6-2 shows the location and function of the ten thrusters.

The four propellant tanks are 65.1-cubic meter titanium cone spheres with a total capacity of 195 kg, which provides a mission lifetime of 5 years, with considerable margin. The tanks were initially pressurized to 305 pounds per square inch absolute (psia) at 21 degrees 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 used to isolate a leaking thruster of 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 over the 0 degree 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 Aerospace) and shipped to APL for integration with the spacecraft after thermal blanket installation. It was environmentally tested at the protoflight level using water to simulate propellant and using mass simulators for the rest of the spacecraft subsystems.

Figure 7.2.6-1: Propulsion Subsystem Schematic

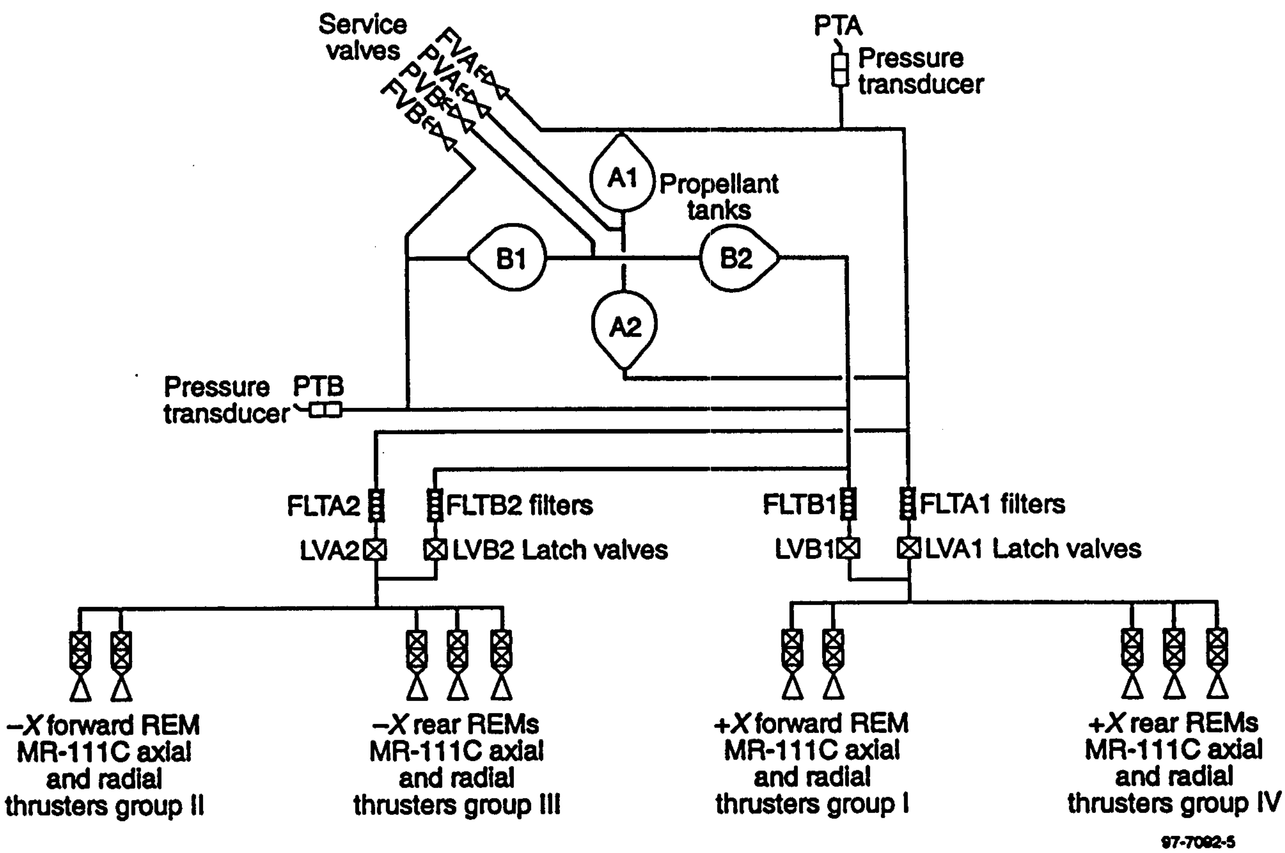
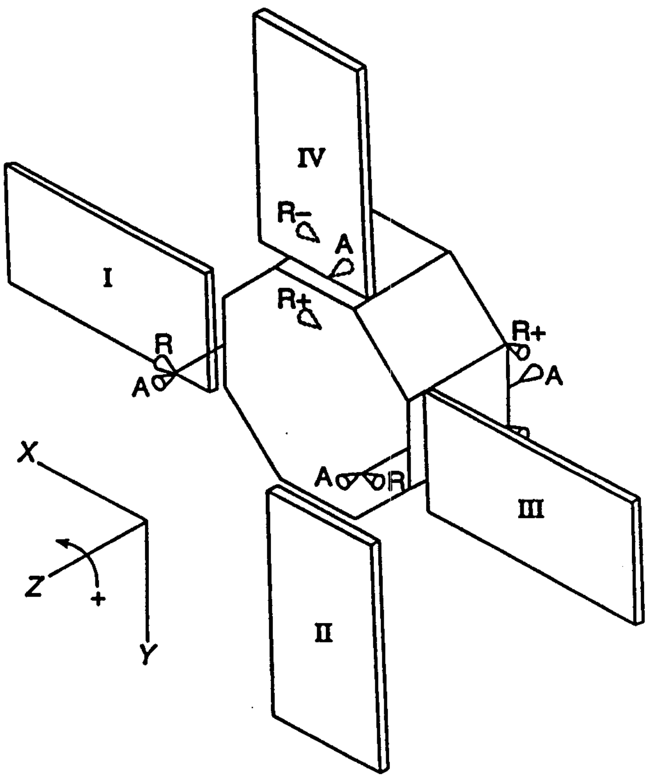


Figure 7.2.6-2: Propusion -- Location and Function of Thrusters



Thrusters	Velocity Control +Z	Velocity Control −Z	Velocity Control X–Y plane		Spin Axis Pointing				Spin Rate Increase	Spin Rate 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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7.2.7 Power

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 ±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.

1. 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 × 149.9 × 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 antireflective 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.

2. 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 × 3.0 × 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).

3. 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 percent 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 semirigid 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 contains 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 (±2%), battery charge limits (0 to 1.5 A), and temperature-compensated battery voltage limits.

Figure 7.2.7-1 displays the electrical power system functional block diagram, and Table 7.2.7-1 gives weights and dimensions of the power subsystem components.

The level of difficulty for the power subsystem was rated as a 3, or a medium complexity level. The subsystem was designed to be straightforward, but did have to provide a regulated bus voltage and be as magnetically clean as possible without a major cost impact.

Figure 7.2.7-1

Advanced Composition Explorer
Electrical Power System
Functional Block Diagram

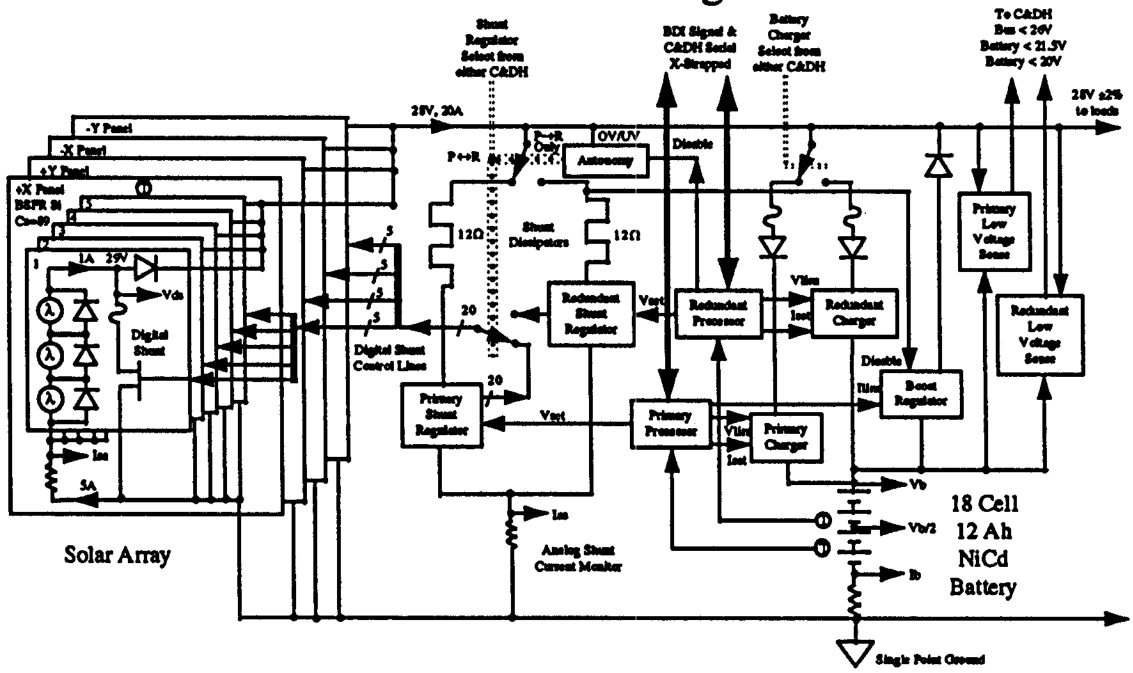


Table 7.2.7-1: Power – Weights and Dimensions

ACE Power Electronics, Power Switching, and Terminal Board Weights and Dimensions		
Item	Weight	Dimensions
	lbs (kgs)	inches w/o feet
Terminal Boards & Shunt Plate	12.16 (5.51)	16.5 x 9.0 x 5.5
Digital Shunts (1/4)	1.21 (0.55)	4.25 x 4.5 x 2.2
PSDE	8.94 (4.05)	12.0 x 10.0 x 3.625
PSCE	10.75 (4.88)	9.0 x 9.0 x 6.538
Power Switching	12.16 (5.51)	9 x 9 x 10
Ordnance	13.90 (6.30)	9 x 9 x 10
	Kgs	Cm
Solar Panels (1/4)	8.5	86.4 x 149.9 x 3.2
Hinges	1.9	
Battery	14.2	34.3 x 26.7 x 17.1

7.2.8 Environmental Design Drivers

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.

1. Magnetic Cleanliness

The Observatory has two magnetometer sensors, one at each boom. Although ACE had no magnetic requirement, there were magnetic goals. Magnetic cleanliness precautions were undertaken, to the extent that costs would not be significantly affected, 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 o 15 kHz (+/-200 Hz), 30 kHz (+/-200 Hz), and 60 kHz (+/-200 Hz). To achieve these goals a number of steps were taken, including the following:

- Battery was wired to reduce magnetic field and magnetic loops
- Solar panels were back-wired for compensation
- Twisted pair wire was used for power lines; a single-point ground system was used
- Magnetic materials used in spacecraft components were reviewed
- Boom material is titanium; hinges and materials near sensors are nonmagnetic
- Mapped magnetic emissions of selected suspect spacecraft components and part

Preliminary results of the static test prior to launch indicated a worst-case field of 2.3 nT at each flight sensor, and if the sources had been correctly identified the in-flight field components are only 1/10 of this value. Preliminary results of the AC test revealed no AC signals at a level that could significantly affect the magnetic field measurements. The actual field will never be known, but from data for the early phase of the mission, it is estimated to be <1 nT at the sensors.

2. Radiation Environment

Spacecraft components had to be designed to meet the following radiation requirements. Parts were required to survive a total ionizing dose of 15 krad (Si) without part failure (with spot shielding required). The 15 krad (Si) requirement was based on a five year mission goal. Since ACE can operate for four years at solar maximum, the components have to withstand a total proton fluence of about 2^{11} particle/cm^2 for protons with energies above 4 MeV. Components had to be immune to latchup. Parts susceptible to single-event upset (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 of 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, the amount of data lost would not cause violation of the component specification requirements,.

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 (i.e., this level requires only the use of white coats). Chemicals and compounds were carefully screened. Nitrogen purge at a temperature of 15 to 25 degrees C was supplied to the instruments. Each instrument used a restrictor to obtain its required flow rate. The purge remained connected until launch. The sun sensor and star scanner optical portions were covered unless ground support equipment was attached.

7.2.9 Spacecraft Safing

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

The C&DH autonomy rules are widely used on the spacecraft to put an instrument, a spacecraft component, or the whole Observatory 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 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. Autonomy is used on the ACE spacecraft for all phases of the mission: launch, post-launch vehicle separation, and on-station operation.

7.3 INTEGRATION

The initial integration activities started with the flight harness and the two main terminal boards. These boards housed main bus power, current sensing resistors for all the components, fuses in series for all component power that needed to be switched on or off, current limiting resistors for all pyro devices, and return ground bus bars to establish single point ground. All of the wiring that left and returned to these main distribution points formed the basic flight harness wiring to every component on the ACE Observatory. The input and output wiring was distributed and secured on the flight structure, and continuity and isolation measurements were made to verify correct wiring to every component. External power and signal sources were used where applicable.

The satellite bus components were then integrated to the bus. At this stage of integration it is vital to have the power relay components for integration first to achieve successful scheduling and progress. The integration team did indeed receive the Power Switching and Ordnance components in a timely fashion for a successful integration of the components distributed via the flight harness to all interfaces that needed power distribution by switching components.

The Command and Data Handling components were delivered, and integration of these components was accomplished using automated software procedures (PROCS).

The RF Communication components were integrated utilizing a ground support equipment RF power rack to communicate to the antenna receiving and transmitting transponders. Performance procedures were written in software to accomplish all performance evaluation testing of the RF components.

The Propulsion System was delivered for integration already mounted to the -Z bottom deck structure. The tanks were partially filled with dry nitrogen for integration and testing purposes. Testing was accomplished with commands sent manually before a performance procedure verified all aspects of this system.

The Attitude components were integrated with simulators and stimulators attached to the components. Ground support systems were used to excite these components while performance software testing was performed.

Power components were integrated utilizing a solar array simulator to produce the solar array power to provide the power to continue Observatory testing with the real power system. Prior testing of all components was accomplished via a ground support equipment power rack with an all purpose test connector and a pull away umbilical connector.

Each instrument had its own ground support equipment that interfaced with the instrument during electrical testing, which was done by automated computer software procedures (PROCS). A comprehensive electrical baseline test was run as a flyable Observatory, and the results served as a comparable baseline database for further testing. Electrical functional and performance testing of every component was repeated at various stages throughout the environmental exposures. The Observatory was exposed to and tested for vibration, mass properties, acoustics, shock/separation/deployables, electro-magnetic compatibility, static magnetics, wet spin balance, dry spin balance, and thermal vacuum balance and performance.

In addition, Deep Space Network testing was accomplished with the GSFC Mission Operation Center (MOC). Several mission tests were run with the mission operators stationed at the MOC. These tests gave the mission operators valuable experience for post launch operations.

The ACE Integration and Test Operational Control Center (ITOCC) was the system used to control the ACE spacecraft during integration, environmental test, and launch site operations. The ITOCC system has several unique and innovative aspects. The core system software was developed for use in both the ITOCC and the MOC. This provided cost efficiencies, cross training synergy, and software debugging that made the MOC better and more stable at launch than in the typical development process.

The integration team used an innovative technique to run tests on multiple spacecraft systems simultaneously. This saved cost and schedule by shortening the time required to complete a full spacecraft test. This scaleable, distributed architecture of the ITOCC system and the redundant command systems on the spacecraft allowed the simultaneous testing to be accomplished. Two instruments were tested simultaneously by sending commands through separate command systems, but monitoring a single stream of spacecraft telemetry.

The integration team worked more closely with the mission operations team than on previous programs of this type. One way this was done was to have a mission operations team member be a full time integration engineer. In addition, the mission operations team was provided early access to the spacecraft to test their system and train team members. They were also allowed more opportunities to operate the actual spacecraft prior to launch than on a typical spacecraft project.

Due to the frequent and early interaction between the integration and test (I&T) and mission operations teams, several spacecraft software functions were added or changed to improve or ease operations of the spacecraft. A key item in this area involved downlinking real time status of the solid state recorder dump process.

There were difficulties in the Observatory integration. The instrument deliveries were late. Given the large number of instruments on board, this made integration on a schedule very difficult. This was coupled with the fact that some of the instruments had not been thoroughly checked out (i.e., primarily for interfaces and software, but sometimes for performance) prior to delivery for integration. These instrument problems were because ACE instrument deliveries were on such a tight schedule that developers had barely begun to learn the behavior of their flight instrument when the spacecraft developer was already asking for its “operating manual” to write system test procedures, and yet the operational details were not known even to the instrument builders. This situation slowed the integration process, but no relief was given in schedule by the GSFC project management. The solution was many long nights and weekends, not always a reliable fallback. APL software people worked closely with the lead instrument engineers to produce automated procedures for electrical and performance testing, so that their data could be received and dumped to their ground support equipment for their serial instrument data. Each instrumentor had his own work station to retrieve his individual serial data through the C&DH components and transmitted via selected transponders. This unique setup for each instrument, after some involved software difficulties, seemed to work well. These same work stations are being used on an every day basis at the Mission Operations Center at GSFC. Another unique solution was to allow instrument removal after thermal/vacuum testing for further work by the instrumentors, and then re-integration at KSC for unit tests and end-to-end tests.

Also the software for the ITOCC was late. ACE was the first GSFC project to use a common ground system for the integration and test function, mission operations, and later, the data analysis center. The core/generic software was to be expanded to perform each of these functions with the understanding that the I&T function was needed first. The function of I&T and what was needed to expand the software was significantly under-estimated in the amount of effort needed. This software was being written by a contractor to GSFC, and it was the first time that it was to be used for the I&T function. A primary workaround was to add APL software engineers to modify the software for some needed I&T functions, and then communicate these changes to the contractor. Then, at a later time when the contractor could incorporate these changes into the larger common system, they would be sent back to APL via an official software update. The official update was not always exactly the same as the original APL "patch" which led to some operational and configuration problems. ACE was a learning experience in this area, and although it was difficult during the program, this should be extremely useful in future programs for reducing costs.

The integration of the bus subsystems was rated as a 3, or average difficulty, which is a medium complexity. The integration of the bus and instruments had an average rating somewhat greater than 4, or at least a medium-high complexity.

7.4 GROUND SYSTEM

Much of what is covered in this section was extracted from Frank Snow et al., Implementation of the ACE Generic System for the Multiple Uses of Spacecraft Integration and Test and Mission and Science Operations, 1997. Contact Frank Snow at GSFC for more information.

7.4.1 Ground Elements

APL was responsible for integration and testing (I&T) of the spacecraft and its instruments and providing the I&T Ground Support Equipment (GSE) system whose chief component is the Integration and Test Operations Control Center (ITOCC).

Spacecraft Operations are performed at the GSFC in the ACE Mission Operations Center (MOC), which was built by the Mission Operations and Data Systems Directorate at GSFC. Level-0 processing of payload data is performed in the ACE MOC. The data is then transmitted to the ACE Science Center (ASC) located at and built by the California Institute of Technology (CIT). The ASC is responsible for scientific processing of the data, making the data products available to the ACE Science Analysis Remote Sites (ASARS), and transmitting instrument commands to the ACE MOC for transmission to the on-board ACE Command & Data Handling (C&DH) Data Processing Unit (DPU). The commands are then transmitted to the desired instrument's DPU.

The basic component for the ACE MOC is the Transportable Payload Operations Control Center (TPOCC), a Unix based system, which has been a foundation system for most spacecraft control centers at the GSFC for the past several years. Over time, GSFC has found itself increasingly tied to obsolete computers hosting unique systems supporting aging spacecraft. A primary goal of the TPOCC, which is computer independent, is to enable a control center to upgrade its computer equipment and rehost the software without having to completely rebuild the software. TPOCC provides the standard control center functions. Mission unique software components are developed and then interfaced with TPOCC. A secondary goal of TPOCC usage is to enable replacement of TPOCC itself with upgraded versions.

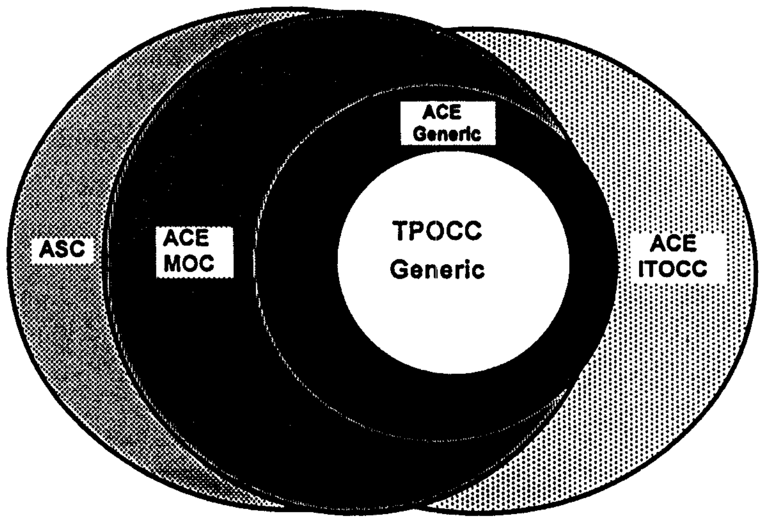
7.4.2 Ground System Strategy

Given the NASA mandate to reduce budgets and conform to standards, the ACE Ground Project decided to build a common system, based upon TPOCC, to support the three ground elements, the APL ITOCC, the CIT ASC, and the ACE MOC. To encourage the ASC and APL participation, the common software was funded through the GSFC Mission Operations and Data Analysis (MO&DA) budget. Under the new single-project management approach, operation funds from MO&DA (institutional support/MOC) and Code S (mission unique support/ASC, APL) was vested in one person, the Ground System Project Manager (GSPM). This allowed the GSPM to base decisions on the overall NASA ground system costs, which include the ACE MOC, the ASC, and the ITOCC.

An early Phase B analysis led to the conclusion that the TPOCC could provide much of the needed functionality in the ITOCC and the ASC, as well as the ACE MOC. The

Ground Project decided that a generic ACE version of TPOCC, which has subsequently come to be called just ACE Generic, would be developed to include all functionality common to the ACE MOC, the ITOCC, and the ASC. The ACE Generic was adapted from the X-ray Timing Explorer's (XTE) TPOCC since many of the XTE unique capabilities were useful for ACE. Separate copies of the ACE Generic were then augmented individually with various capabilities needed in the ITOCC, the ACE MOC or the ASC. Most of the MOC unique capabilities are also slated for use in the ASC. A simple Venn diagram depicting this is shown in Figure 7.4-1.

Figure 7.4-1. ACE Ground Systems Common Functionality



A natural extension to the use of ACE Generic's common core of software was to use a common database. Previously, it has been accepted procedure to develop multiple databases. This would include one or more for use in developing the spacecraft and integration of the spacecraft and its instruments, another for operations personnel, and yet another or even several more for science operations. This invariably was a source of errors and time lost making corrections and modifying the databases to accommodate data being transferred from one database to another. The ACE Ground Project decided there would be a single database for spacecraft development and testing, for operations, and for science operations. Similarly, the ACE Ground System utilized the TPOCC System Test and Operations Language (TSTOL) for all procedures and display development. This enabled reuse of procedures and displays among the MOC, ITOCC, and the ASC. Once developed for spacecraft I&T, procedures and displays were adapted for use in operations or the ASC.

7.4.3 <u>Ground System Design And Implementation</u>

Members of the Flight Operations Team (FOT) were included in the Ground Support team from the early phases of the mission. This enabled a smooth transition between spacecraft design and ground operations, and avoided the frequent situation of ground operations being made unnecessarily complicated by spacecraft design decisions. Early FOT support facilitated the multiple use of TPOCC. During development of the C&DH and its software, members of the FOT worked directly at APL to participate in developing and executing test procedures for the C&DH. The FOT also participated in the planned usage and design of the ASC. Through these activities, the FOT developed a comprehensive, valuable knowledge of the three areas (ASC, Spacecraft I&T, and Mission Operations) that carried into operations. The concept of the Integrated Team approach working across spacecraft design, integration and test, and operations has worked successfully on smaller spacecraft, but this is the first time NASA has attempted this approach on a program of this magnitude. The ability to bring this integrated team into being was greatly aided by vesting the MO&DA and Code S ground budgets under a single GSPM.

The development of ACE Generic was contracted to Computer Sciences Corporation (CSC) as a task under the MO&DA task order contract. This task also included development of the MOC specific enhancements to ACE Generic. APL let a separate contract to CSC for development of the ITOCC specific enhancements. Many personnel at CSC worked on both the ITOCC and MOC contracts, providing a common contractor base for both unique developments. CIT detailed a single person to learn about TPOCC and provide ASC input into the design and development of ACE Generic. This effort required close cooperation between all users and the developers. Questions and decisions concerning design and development of ACE Generic were made at regular Joint Action Development (JAD) sessions augmented by frequent teleconferences. APL subsequently enlarged their development team to include personnel supporting the development of the ACE Generic.

For the ACE MOC, some of the ACE Generic enhancements were:

- Level zero processing,
- Mission planning and scheduling,
- Attitude determination,
- Addition of Generalized Systems Analysis Assistant (GENSAA)
- Addition of Generalized Trend Analysis System (GTAS)

The ASC uses essentially the same software and hardware systems as the MOC, with the addition of the science data analysis software.

For ITOCC, some of the ACE Generic enhancements were:

- Command Echo - compares an echoed command with what was transmitted,
- Raw Telemetry Socket Server - sends a telemetry stream via Ethernet to any user,
- GSE Command-enable - ITOCC control of the GSE and collection of its telemetry,
- Spacecraft Subsystem Run Time Processing - ensures that certain spacecraft components are run requisite minimum amounts of time and that relays with lifetime limits on the number of state transitions are not over exercised,
- Spacecraft Housekeeping Trending - developed in GTAS, a TPOCC adjunct system.

The core computer hosts for the ITOCC, the ACE MOC, and the ASC are HP 748 hosts with multiple processors. These are augmented with UNIX workstations, HP 715s in the MOC and higher-powered HP 735s with more memory in the ITOCC. The MOC HP 748 has an Internet Protocol with its workstations connected to an Ethernet-based local area network. The HP 748 supports mass data storage, as well as additional NASA communications Internet Protocol interface software for receiving spacecraft telemetry data and for sending commands and data to the spacecraft. Figure 7.4-2 shows the ACE MOC and ASC and their relationships with ACE and other entities. Figure 7.4-3 depicts the ITOCC and its relation to the spacecraft and ACE.

7.4.4 Benefits

7.4.4.1 Cost Reductions

1. System Development

Significant benefits have accrued from multiple use of the ACE Generic system, saving NASA millions of dollars. Although the ACE Generic/MOC task required more lines of code than expected for the MOC, it was substantially less than the lines of code that would have been needed for the three separate systems, the ASC, I&T, and the MOC. Also, the common systems between the MOC and spacecraft I&T led to elimination of the training simulator for the FOT. Instead, the FOT trained with the spacecraft at APL. Similar savings in maintenance will be realized; enhancement or correction in one system will enhance or correct all three.

Only one database is utilized for all three systems. This eliminates time consuming, costly translations from one database to another. This also led to collaboration between the FOT and I&T personnel in developing the command and telemetry database and STOL procedures. This led to a much greater understanding by the FOT of the structure and contents of the database. This also allowed the FOT to develop and test the subsystem activation and test procedures that will actually be used in on-orbit operations. This also enhances the overall reliability of the database content, and TSTOL procedures developed for I&T were adapted for use in the MOC and the ASC.

2. Reduced FOT

Since the FOT participated in early mission design and spacecraft I&T, the spacecraft functions that affect ground operations (e.g., telemetry, commanding, star tracker, sun sensors, onboard commands, safing routines) have been implemented to simplify ground operations with no adverse impact on cost or schedule. This allows post-launch cost savings through a smaller FOT staff. There is only one shift, seven days per week. In several instances the Ground Project performed trade studies resulting in retention of critical requirements while enabling relaxation of minor requirements. This was done efficiently because the user, the FOT, was involved. In past missions where the user was not involved, when cost and schedule demands impacted requirements critical ground requirements would frequently be relaxed, necessitating development of operational work arounds and additional software. Such decisions often increased the staff size of the FOT.

Mission cost savings are in part due to the savings by using MO&DA software in ITOCC and ASC development. The ACE Ground Project was able to provide the common ground system to the ITOCC and the ASC without exceeding its original budget.

Figure 7.4-2: ACE MOC and Interfaces With ASC and Other Support Systems

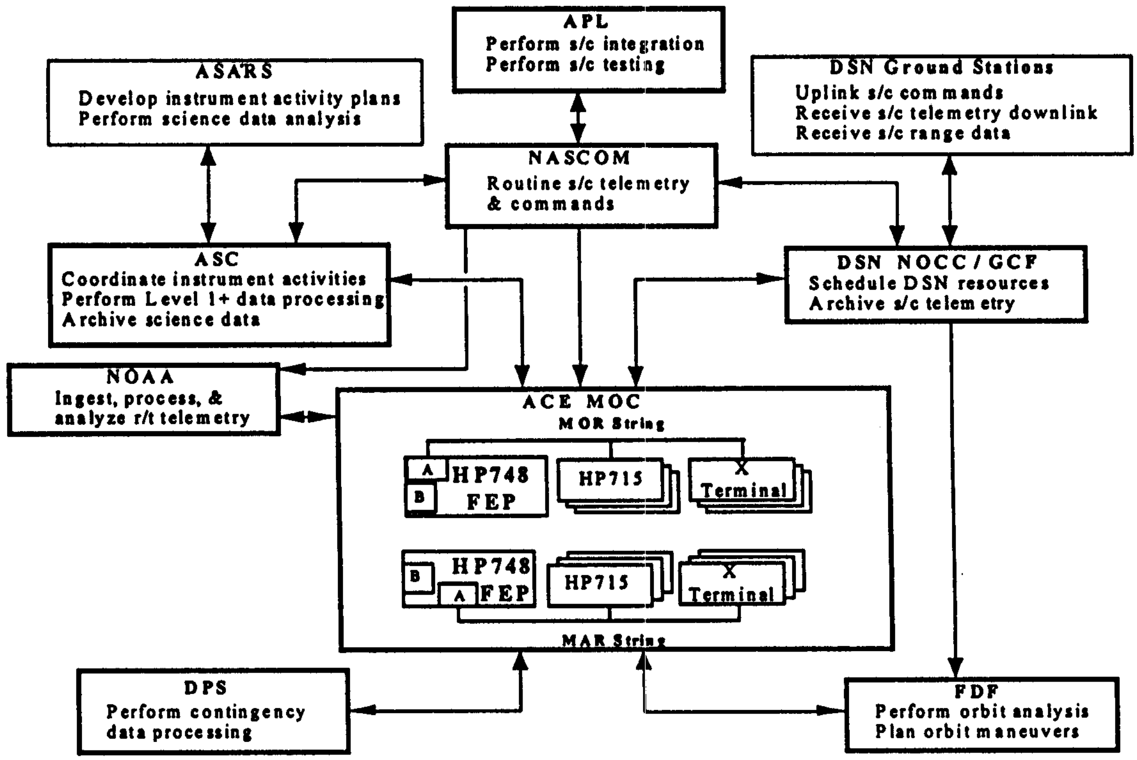
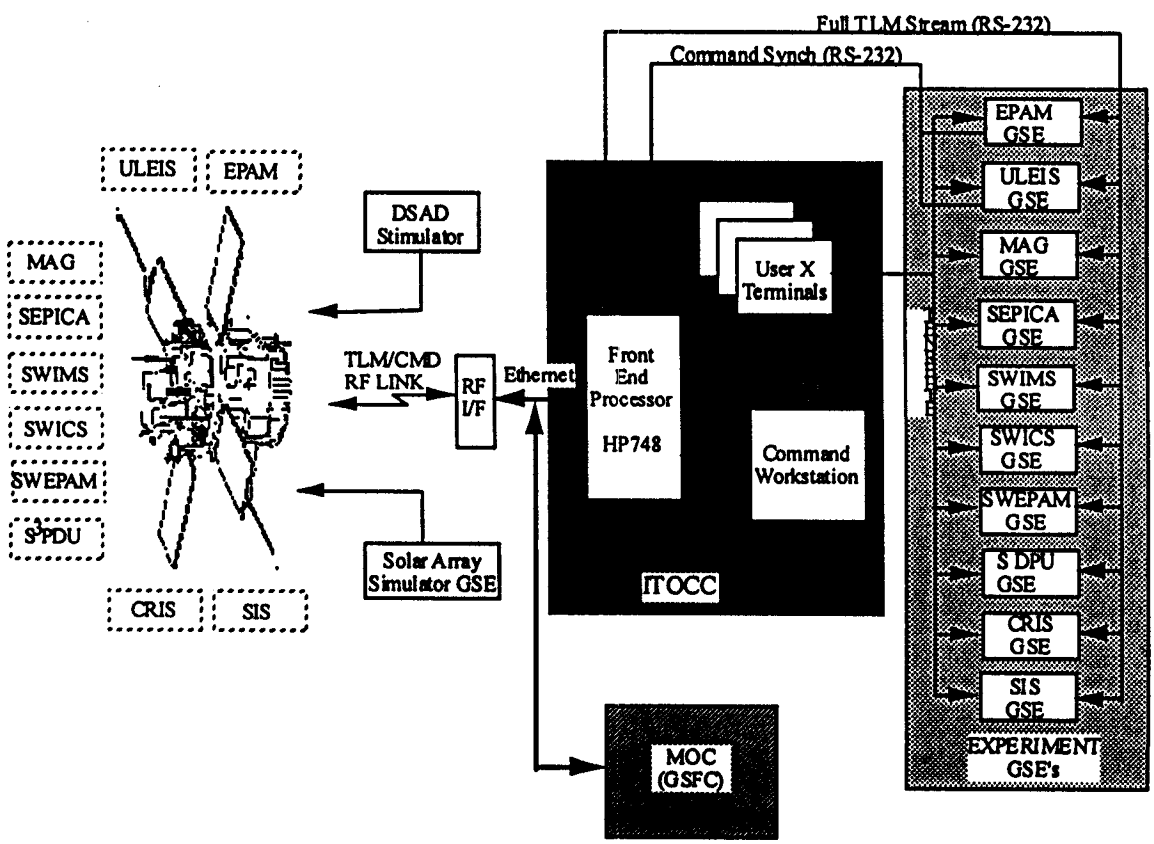


Figure 7.4-3: ACE ITOCC



7.4.4.2 Training

Significant benefits were accrued in training, because the FOT was trained using the ITOCC system and the spacecraft. This training was far more comprehensive than a FOT would normally receive. The FOT not only learned how to use procedures, but actually designed, implemented, and tested the procedures. This gave the FOT more insight into the procedures and enables the FOT to respond more quickly to spacecraft anomalies during operations.

When the spacecraft engineers and scientists came to GSFC for Launch and Early Operations little additional training was required since they already had been working with the TPOCC system during spacecraft I&T. This reduced the need for multiple simulations and enabled a smoother transition from spacecraft I&T to operations.

The Mission Operations Team came together early in Phase B. Therefore, the Mission Operations Team which includes the spacecraft I&T team, the ASC, and the MOC personnel had been involved early in decisions on mission requirements and mission goals. The team had been collaborating for over two years and developed confidence in each other's abilities and judgment. With the synergy of the combined team, even though key personnel had left the project, the corporate knowledge within the FOT and the I&T team helped train and support new personnel.

7.4.4.3 IGSE Interface

The Instrument Ground Support Equipment (IGSE) uses the same interface to the MOC as to the ITOCC. Pre-flight interface testing between the MOC and the IGSE was greatly simplified, only requiring the configuration of the appropriate network connections. Moreover, the instruments were able to reuse many of the same activation and checkout procedures that were developed during Observatory integration of their instruments. This helped in the often difficult process of converting instrument procedures to the operational system during the last few months prior to launch.

7.4.5 Common System Challenges and Lessons Learned

The assumption that a common ground system could support both integration and test and flight operations spacecraft control requirements proved largely correct. However, adapting a system originally designed for flight operations to service integration and test requirements presented some unique challenges. These and related lessons learned were discussed briefly in section 6.2.4, and are discussed more fully here.

- A considerable amount of effort was expended up front trying to document all the requirements for the MOC, ITOCC and ASC for ACE Generic. However, all the ITOCC requirements could not be defined in advance because the spacecraft was not yet built. Also, as the ITOCC personnel gained experience with early deliveries of ACE Generic they discovered that they could not use some of the designed features as planned. Changes in ITOCC requirements resulted in an approximately 25% increase in ACE Generic requirements. The development team accommodated the growth in requirements by stretching out the ACE Generic delivery schedule and delaying the implementation of the less mission-critical ACE MOC specific requirements. Any development approach for a common ground system should be able to accommodate this expansion in requirements and have the capability to change release schedules.

- The original delivery schedule of providing ACE Generic releases 3 or 4 months apart did not meet the ITOCC requirements for delivery of new capabilities or fixing of high priority system discrepancies. The integration and test team preferred to get delivered capabilities as soon as they were ready even if all testing of the new capability was not completed. This resulted in the development team abandoning the system release approach and creating a "function drop" delivery mechanism. Each function drop comprised a single new capability or discrepancy fix, system integration instructions and a "function" test plan. Function drops were provided to the ITOCC on a weekly basis. ITOCC personnel served as system integrators and testers for each function drop. This new delivery approach resulted in better responsiveness to ITOCC needs and allowed new requirements to be more easily accommodated within the development schedule.

- Development of a common ground system must take into account the resources required to support the different objectives and delivery timing of spacecraft I&T and operations. The ACE development effort ran into significant resource problems starting at approximately two years prior to launch when the I&T effort started to ramp up and the initial MOC systems were due for initial release to support early FOT testing with the spacecraft. At this time the I&T team found a variety of ground system problems that could have potentially delayed the I&T schedule if they were not fixed in a timely manner. At the same time, the FOT was ramping up for their initial spacecraft compatibility testing and required an initial release of the MOC ground system. This caused significant resource problems for the spacecraft developers and resulted in resources being moved from the MOC development effort to assist the troubleshooting of the I&T system. On future common system development efforts the development team resources must be ramped-up significantly at approximately launch minus two years through the completion of the I&T effort. This will ensure that the needs of both groups are met in a timely manner. It will also ensure that resources will not be drained away from the MOC development effort to focus on near-term, I&T needs, with the long-term impact of not enough resources being applied to the MOC development effort prior to launch.

- Another challenge was the different approach for managing schedule reserve. MOC development at GSFC has tended to vest schedule reserve in the ability to accommodate a schedule slip or failure of a function by postponing work to future releases. However, delays of this scale cannot be accommodated when building an I&T system. Schedule reserve for unforeseen work must be included in the basic schedule for an I&T system just as it is for any spacecraft component or instrument.

- Spacecraft integration and test stresses the command and control system in a different fashion than flight operations. Spacecraft integration and test sends a large number of real-time commands, interfaces with the spacecraft for fairly long periods of time (sometimes greater than 24 hours) and has a much greater dependency on real-time telemetry for diagnosing problems and confirming spacecraft health. Whereas the flight operations team would rarely send more than one hundred commands during a real-time contact (a typical ACE pass will last 3 1/2 hours and never more than 10 hours), the integration and test team may send thousands. Further, the FOT will rely on recorded data for trending and determining spacecraft anomalies. To accommodate these integration and test requirements, the maximum size of the command buffer had to be expanded significantly. Disk space had to be reallocated to accommodate large history files that record all real-time and playback telemetry, which are recorded during a session with the spacecraft. Also the capability to close history files and open a new one without losing data had to be added. This allowed the integration and test team to trend real-time data while continuing with the integration and test session.

- Similarly, another "cultural" difference pertained to procedures. I&T procedures are based on complete component and spacecraft functionality whereas FOT procedures are based on nominal and contingency operational scenarios. Thus, FOT procedures tend to be more segmented than those used for I&T. This required the FOT to develop many procedures from scratch, using the I&T procedures only as a baseline. This procedure conversion process also led to a greater configuration and control effort than originally planned.

- During development of a common I&T/operations ground system the ground system developer must ensure that maximum commonality is maintained between the systems to ensure that the only variations between the systems are those that are absolutely necessary for each group to perform unique functions. On ACE, the development teams working on the I&T system performed a large number of system modifications that “improved” I&T operations but were not critical to I&T functional objectives. Since these changes were done after the baseline system was delivered to I&T these changes were not reflected back in the MOC system developed for operations. Some of these changes affected how ground system procedures were run. As a result, the FOT was not able to use many of the procedures developed for I&T on the MOC ground system. These procedures required extensive editing to remove the I&T specific items, replacing each with a MOC equivalent to try to functionally maintain the procedures integrity. Future common system development efforts should include a development team Control Board that reviews and approves all non-common changes submitted by either the I&T or operations teams. This board would address the implications of each change prior to allowing the implementation to proceed.

- A large number of the high priority discrepancies with the Integration and Test team dealt with inconsistencies with the user interface. While a FOT member will become highly trained on the intricacies of the MOC user interface, many I&T users (such as system engineers) have little to no familiarity with the ITOCC. A consistent user interface allows these novice users to quickly come up to speed using the ITOCC with minimal training while they trouble shoot problems. The development team had to become sensitized to this issue and give higher priority to fixing user interface problems when they arose.

7.4.6 Complexity Rating

Because of the difficulty of integrating a system for three functions, the ground system was rated a 4, or medium-high complexity.

8.0 RESOURCES

8.1 COSTS

Table 8-1 contains the final ACE cost breakdown by fiscal year (POP 98-1).

Table 8-1: ACE Costs, POP 98-1
(Real Year Thousands)

	FY94	FY95	FY96	FY97	FY98	Total
Project Management	877	1,092	1,103	1,488	580	5,140
Spacecraft	12,156	17,031	9,285	8,091	415	46,978
Science Payload	7,073	15,347	15,161	9,777	2,835	50,193
Ground System	174	536	282	377	108	1,477
Performance Assurance	68	362	379	373	11	1,193
Flight Operations	181	325	419	660	213	1,798
Total	20,529	34,693	26,629	20,766	4,162	106,779

8.2 CIVIL SERVICE WORK FORCE

Table 8-2 provides the fiscal year end Civil Service work years for ACE by GSFC Directorate.

Table 8-2: ACE Civil Service Work Years by Directorate
(Fiscal Year End Actuals)

Directorate	FY94	FY95	FY96	FY97	FY98	Total
200: Management Ops.	.1	.7	.8	.4	.2	2.2
300: Flight Assurance	.1	1.7	1.9	2.3	.4	6.4
400: Flight Projects	4.8	7.0	7.0	7.5	4.3	30.6
500: Mission Ops/Data Sys.	2.1	5.7	5.4	3.1	2.1	18.4
600: Space Sciences	0	2.5	2.2	1.7	1.9	8.3
700: Engineering	.8	2.1	3.4	2.4	.5	9.2
Total	7.9	19.7	20.7	17.4	9.4	75.1

POSTSCRIPT

This report was prepared by Donald H. Strope of COST, Inc., Laurel, Maryland under Purchase Order Number S-10557-G.