Recovery of the Wide-Field Infrared Explorer Spacecraft

David F. Everett
NASA/Goddard Space Flight Center, Code 730, Greenbelt, MD 20771
(301) 286-1596, David.F.Everett.1@gsfc.nasa.gov

Thomas E. Correll
NASA/Goddard Space Flight Center, Code 573, Greenbelt, MD 20771
(301) 286-6047, Thomas.E.Correll.1@gsfc.nasa.gov

Scott Schick
Space Dynamics Laboratory, Logan, UT
(435) 797-4426, sschick@sdl.usu.edu

Kimberly D. Brown
NASA/Goddard Space Flight Center, Code 545, Greenbelt, MD 20771
(301) 286-2627, Kimberly.D.Brown.1@gsfc.nasa.gov

Abstract. The Wide Field Infrared Explorer was developed to perform astronomy using a cryogenically cooled infrared telescope. Shortly after launch, rapid venting of the cryogen, caused by an untimely cover removal, sent the spacecraft into an uncontrollable spin which exceeded 60 revolutions per minute. Over the next week, the WIRE team developed a plan and successfully executed the procedures necessary to de-spin the spacecraft and gain attitude control, but the cryogen for cooling the instrument was depleted. The recovery of the spacecraft enabled a thorough checkout of most of the subsystems, including the validation of several new technologies. Although the primary science mission was lost, WIRE is making breakthrough astroseismology measurements using its star tracker. This paper describes the recovery of the WIRE spacecraft and the performance of its key technologies, including the two-stage solid-hydrogen cryostat, an all-bonded graphite-composite structure with K-1100 radiator panels, composite support struts, a dual-junction gallium arsenide solar array module, a concentrator solar array module, and a 300 Mbyte solid-state recorder.

Introduction

The Wide-Field Infrared Explorer (WIRE) is the fifth Small Explorer (SMEX) mission launched by the National Aeronautics and Space Administration's (NASA) Goddard Space Flight Center (GSFC) in Greenbelt, Maryland. The spacecraft bus was built in-house at Goddard, and the instrument was built at the Space Dynamics Laboratory in Logan, Utah under contract with the Jet Propulsion Laboratory in Pasadena, California.

WIRE features the smallest solid-hydrogen cryostat ever flown in space, part of an infrared telescope instrument which was designed to study the evolution of starburst galaxies. The spacecraft utilized the first fully-bonded graphite-composite structure flown and an arc-second-class, 3-axis stabilized Attitude Control System (ACS). Launch mass was 258.7 kg and orbit average power is 132 W including the instrument.

WIRE was launched from Vandenberg Air Force Base on March 4, 1999, at 6:57 p.m. Pacific Standard Time on a Pegasus XL. Thirty minutes later, contact was established with WIRE over the McMurdo, Antarctica ground station as planned. The pyrotechnic driver electronics box was turned on during the pass to open the secondary vent, and this action led to the eventual loss of the WIRE science mission.

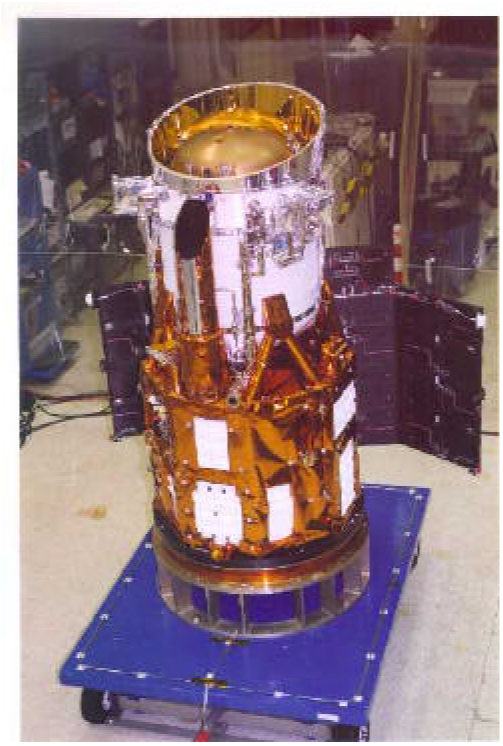

Figure 1: The WIRE spacecraft as seen from the anti-sun side.

Mishap Related Operations

The details of the WIRE mishap are documented in the official mishap report.[1,2] This section contains a brief summary of the mishap followed by a detailed look at the recovery.

Launch

The Orbital Sciences Corporation Pegasus XL is an air-launched rocket designed for relatively small payloads (263 kg to 540 km sun-synchronous orbit). The L-1011 Orbital Carrier Aircraft (OCA) carries the Pegasus launch vehicle to its launch point at 39,000 feet. At the appropriate time and within a designated "drop box" location, the pilot releases the rocket, and it falls for five seconds before the first stage ignites. The "captive carry" phase of the mission from OCA takeoff until drop lasts approximately 1 hour.

Two weeks prior to launch, the cryostat was loaded with hydrogen, and the hydrogen was frozen. From that point on, the cryostat was kept constantly cold with liquid helium to keep the hydrogen below its triple point of 13.8 K and 52 torr.

The "hold time" of the cryostat, the time it took to warm the hydrogen from liquid helium temperatures (around 4 K) to the triple point, was approximately 8 hours. The disconnect, final closeouts, captive carry, and launch all needed to occur within this time, so that the hydrogen was still solid when the vents were opened in orbit. Prior to vent opening, the hydrogen stayed safely sealed inside the cryostat. Burst disks prevented over-pressurization, should the cryostat unexpectedly warm due to a failure. All burst disks were manifolded together through a quick-disconnect (QD) joint, and into a load-isolation system on the OCA. The QD was simply a notched, stainless-steel pipe with a tremendous qualification history--64 units were made and 60 were tested to destruction to guarantee the performance of four flight units (three were spares). Flight data from the launch showed that the quick disconnect worked as designed.

WIRE's first launch attempt was March 1, 1999, but it was aborted due to a launch vehicle problem less than a minute before drop. The OCA returned to

Vandenberg just before the cryostat reached the triple point.

WIRE's second launch attempt on March 4th was successful. An on-board sequence automatically turned on the spacecraft transmitter when the receiver locked on to a strong up-link carrier. This sequence greatly reduced the time required to locate the spacecraft on the first pass--it typically takes another minute or two, out of an 8 minute pass, to bring up the subcarrier modulation and send the transmitter-on command. And we did not have the unnecessary power drain of a transmitter turned on by a timer--if the ground station was not ready for telemetry, it would not bring up its carrier.

The pass plan called for a verification of the spacecraft status and opening of the instrument secondary vent valve. The ground system's pre-programmed sequence sent three commands in rapid succession to power on the pyro box and open the valve. We saw spacecraft body rates increase, but we attributed that to the blow-down at vent opening. It was a month later before detailed telemetry analysis showed that the rates started to increase after the pyro box was turned on but before the command to open the valve. A startup problem in the pyro box caused all pyros to fire, both the secondary valve pyros and the cover pyros. Without the cover, the interior of the cryostat was exposed to earth and sun heat loads 100 times larger than the design load, which caused rapid cryogen venting, which overwhelmed the torque authority of the spacecraft despite a thrust nullifier on the outlet. By the second pass, we knew we had a problem, since the spin rate had increased, but we still didn't realize that the cover was gone.

We continued to take passes once or twice per orbit, but we only left the transmitter on long enough to get a snapshot of the telemetry. Energy in the battery was a precious commodity. We could not do anything about a possibly deployed cover, so we focused on potential Attitude Control System (ACS) problems that could have explained a spacecraft spin-up. We tested for phasing problems and tried swapping polarity of the control, but the spin rate continued to increase.

By 2:30 a.m. Pacific Time, 7-1/2 hours after launch, the battery state of charge was down to 55%, and the ACS lead engineer recommended that we turn off the entire attitude control system to save power, since the spin rate had exceeded its control bandwidth.

We turned the Attitude Control Electronics (ACE) on again about 3 hours later since the battery had recovered to 80% state of charge. We found then that the spacecraft had settled into a stable spin of almost 400° per second about the -x axis, with the x-axis oriented north-south. This spin orientation put the spacecraft y-z plane within about 10° of the sun-line, providing enough power to recharge the battery, as the solar arrays swept nearly normal to the sun once per spin.

The secondary tank, originally designed to last 120 days, was empty in less than 12 hours. Without the protection of the secondary tank, the primary tank began to vent at a high rate, reducing the spacecraft spin rate. Thirty five hours after launch, the primary tank was empty, and the spacecraft spin rate was 315° per second about the -x axis. The battery was completely charged, and the spacecraft was spin stabilized, but the battery was being charged and discharged almost once per second.

De-spin and Recovery

Even before the primary tank had completely emptied, the telemetry clearly showed that the

WIRE spacecraft had settled into a power-positive spin about its major moment of inertia. Clearly, the first step of the recovery process had to be the reduction of the spin rate as quickly as practical without losing the power-positive nature of the spin.

The spin rate of 315° per second was well beyond both the designed 9° per second capacity of the ACS, and the test-determined phase inversion at 60° per second to 90° per second for the acquisition modes. In order to handle this unexpected condition, four software table loads were quickly developed that would provide 0°, 90°, 180°, and 270° phase corrections for the magnetic torquers in spacecraft computer system (SCS) safehold mode. These tables were used according to a simple manual determination--whenever the deceleration fell below about 60% of peak efficiency the next table was loaded (theoretically 70% of peak efficiency could have been maintained with more analysis).

In addition, the ACS design stored a bias momentum in the reaction wheels to stabilize the Y-axis. This bias momentum would have caused loss of power-positive orientation as the system momentum was reduced. Thus, the reaction wheels were turned off during de-spin. Also, only two of the three available magnetic torquers were used, since the third axis would have primarily acted to provide undesirable precession of the spin axis.

Some initial delay was experienced when Earth albedo effects were misinterpreted as precession of the spin axis. Nevertheless, the 4 day de-spin process began in earnest within 48 hours of launch.

We felt relatively comfortable starting the de-spin without much analysis, since we knew that our actions would have very little effect on the direction of the spin axis--the high spin rate made the dynamics very stiff. Our big concern was the final transition from spin stabilization to 3-axis control. How slow could we go without losing the favorable attitude which was keeping the battery charged? How much energy did we really have left in the battery after the beating it had taken?

The ACS team ran models and studied the situation while the whole team monitored the progress of the de-spin. As the spacecraft slowed, the battery state of charge slowly drifted downward--the longer charge times allowed the voltage/temperature (V/T) controller to begin to taper the charge current, reducing the charge efficiency. We adjusted the V/T level, taking care to not allow overcharge of the battery. We noticed variation in one of the potentiometers which indicated the position of the solar array. Was the solar array flapping around? We decided that it was probably a noisy pot, but it was one other factor to consider in all of our decisions.

The spacecraft was designed to acquire the sun with tip-off rates as high as 9° per second using analog acquisition. In order to get into analog acquisition, we needed to turn the ACE box off and then back on again. The flight operations team (FOT) wrote a command sequence to cycle ACE box power. We would load this sequence to the spacecraft and execute it rather than sending each command individually--we did not want to risk turning off the ACE and being unable to get a command in to turn it back on.

As the spin rate came down, the ACS became more efficient at damping the rates, so we had to make sure we didn't slow the spacecraft down to zero before we were ready for the transition to analog acquisition. By March 11th, we had our detailed plans in place. At our mid-afternoon pass, we expected a 5° per second rate--low enough to jump to analog acquisition, though not yet the 1.4° per second

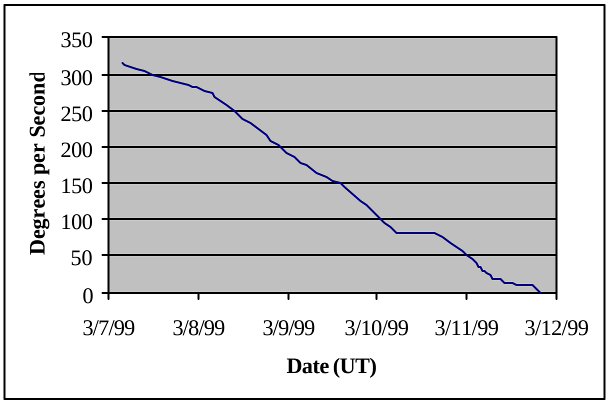

Figure 2: Decrease in Spin Rate During Recovery

optimum at which the analyst consensus recommended such a transition. Actually, the rate had dropped to 0.75° per second, so we immediately executed the sequence which power cycled the ACE box. By the end of that ten minute pass, the spacecraft had nearly acquired the sun. Figure 2 shows the history of the WIRE recovery.

Since the spacecraft bus was now performing nominally under normal conditions, the remainder of the recovery process followed the pre-planned Launch and Early Orbit (L&EO) procedures at a more relaxed pace.

Subsequent analysis has shown that the ACS performance during and after the mishap exceeded its requirements. Science mode pointing accuracy of 1.6 ± 0.9 arcsec was well within both the one arcmin requirement and the two arcsec goal. Slewing and settling times were also less than specified. This analysis of in-flight data has been described in a previous paper.[3]

Performance of Key/New Technologies

The recovery of the WIRE spacecraft has enabled the flight validation of several key technologies and a thorough checkout of its subsystems. Thermal performance, mass and power history, solar array output (including two experiments), data system operation, and the earth sensor performance are described on the following pages. By far, the most significant technological development of the WIRE mission was its solid hydrogen cryostat.

Cryostat

The WIRE instrument was cooled by a two-stage solid hydrogen cryostat built by Lockheed Martin Advanced Technology Center, see Figure 3. Being only the second hydrogen cryostat flown in space for cooling an infrared sensor, it employed a novel concept for cooling infrared detectors below 7 Kelvin. A large solid hydrogen tank, referred to as the secondary tank, provided cooling to below 12 Kelvin for the telescope. It also provided the important function of intercepting the majority of the parasitic heat from the environment. A smaller tank, referred to as the primary tank, resided within the larger tank and also contained solid hydrogen. The primary tank operated below 7 Kelvin to provide cooling for the two long wave infrared detectors and a small portion of the optics. Protecting this primary tank of solid hydrogen from the external parasitic heat loads allowed an extremely low sublimation

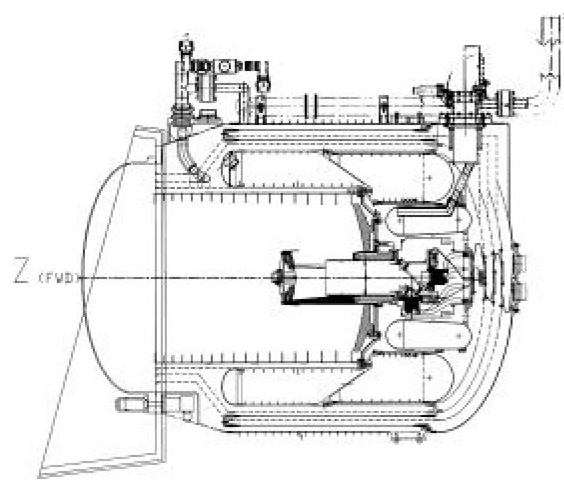

Figure 3: WIRE Instrument Showing Telescope and Two-Stage Cryostat

rate to the vacuum of space. A combination of this low flow rate of hydrogen and a high conductance vent line kept the vapor pressure over the hydrogen extremely low. The cryostat design and development has been described in previous papers.[4,5] This section summarizes the ground test data, ground processing of the cryostat, and on orbit data that supports the use of this cooling technology.

Ground Testing. After the cryostat had been assembled and vibration tested, a test hydrogen fill was performed to prove fill procedures and the thermal performance. To perform the test safely, the work was done at a Lockheed Martin hydrogen test facility in Santa Cruz, CA. The secondary tank was filled and then frozen using liquid helium. Following the fill of the secondary tank, the smaller, primary tank was filled and frozen. To simulate the vacuum of space, each vent line utilized a vacuum pump. A rough pump was adequate to handle the high flow of the larger tank and a turbo molecular pump was used on the primary tank to achieve extremely low vapor pressures over the primary tank hydrogen. Figure 4 illustrates the vapor pressure of hydrogen.

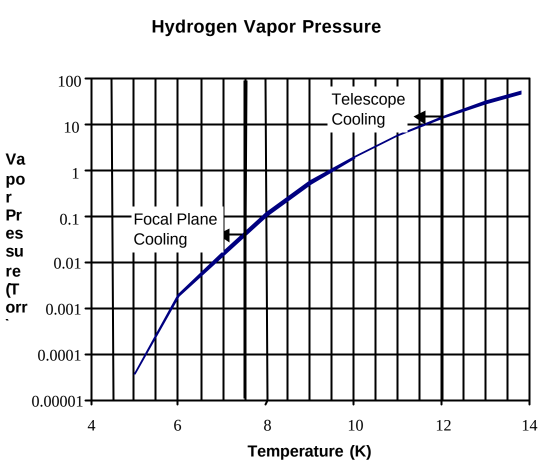

Figure 4: Hydrogen Vapor Pressure

All testing was done using a room temperature vacuum shell. Because of this, the temperature of the secondary tank was slightly warmer than the predicted on-orbit condition. The secondary tank did intercept the majority of the parasitic heat load, and the primary tank was able to cool the operating focal planes to 6.8 Kelvin with a heat load of 10 milliwatts. This performance allowed significant margin below the 7.5 Kelvin requirement.

Launch Site Operations. While on the ground the cryostat required continuous maintenance once it was filled with hydrogen. The WIRE launch vehicle was a Pegasus XL and thus cooling operations were required in the processing facility as well as the flight line on the L-1011.

Following the hydrogen fill of each of the two tanks in the Astrotech payload processing facility, the tanks were sealed up such that no hydrogen ever vented from the system until it was safely in orbit. Parasitic heat entering the <13.8 K tanks while on the ground was handled by a combination of liquid helium coolant and allowing the thermal mass to slow the warming.

Launch site operations required several disconnects of all ground equipment from the system for payload and launch activities. During the periods of time where liquid helium coolant could not be provided, the hydrogen mass would slowly warm. To handle the disconnected time safely the system was designed to take advantage of the low vapor pressure of hydrogen and take into account the changes in density between solid and liquid. The WIRE system was not very large and to allow sufficient operation time without liquid helium coolant for some ground operations the system was designed to allow the hydrogen to warm, reach triple point and melt completely. This allowed hydrogen's large heat of fusion to provide the necessary time for operation as the hydrogen melted. For launch, the hydrogen needed to remain solid so that excess cryogen would not be lost on orbit trying to expend the energy required to cool the mass back down.

Before the hydrogen completely melted the ground crew had to reconnect the liquid helium coolant, refreeze the hydrogen, and cool the solid to approximately 5 K to maintain it in a safe and launchable condition. This connect and disconnect process was time consuming since the procedures had to ensure that air would not enter the coolant line. Air in the coolant line would freeze and plug the coolant lines and eliminate the option for cooling.

The cooling operations occurred almost continually, from the hydrogen fill until the L-1011 took off for the Pegasus launch. Every attempt was made to minimize the number of connects and disconnects to maintain the system in the safest of conditions while personnel worked around the payload. Operations became increasingly difficult as the payload moved onto the flight line and weather became a factor as well. Clean tents were used around the Pegasus fairing openings to keep the instrument clean as the processing crew serviced the cryostat through two small access doors. Equipment was limited within the tents and most equipment was outside and submitted to the weather. Constant monitoring of the instrument continued from the ground crew to the Pegasus launch personnel within the L-1011. A temperature-monitoring unit within the

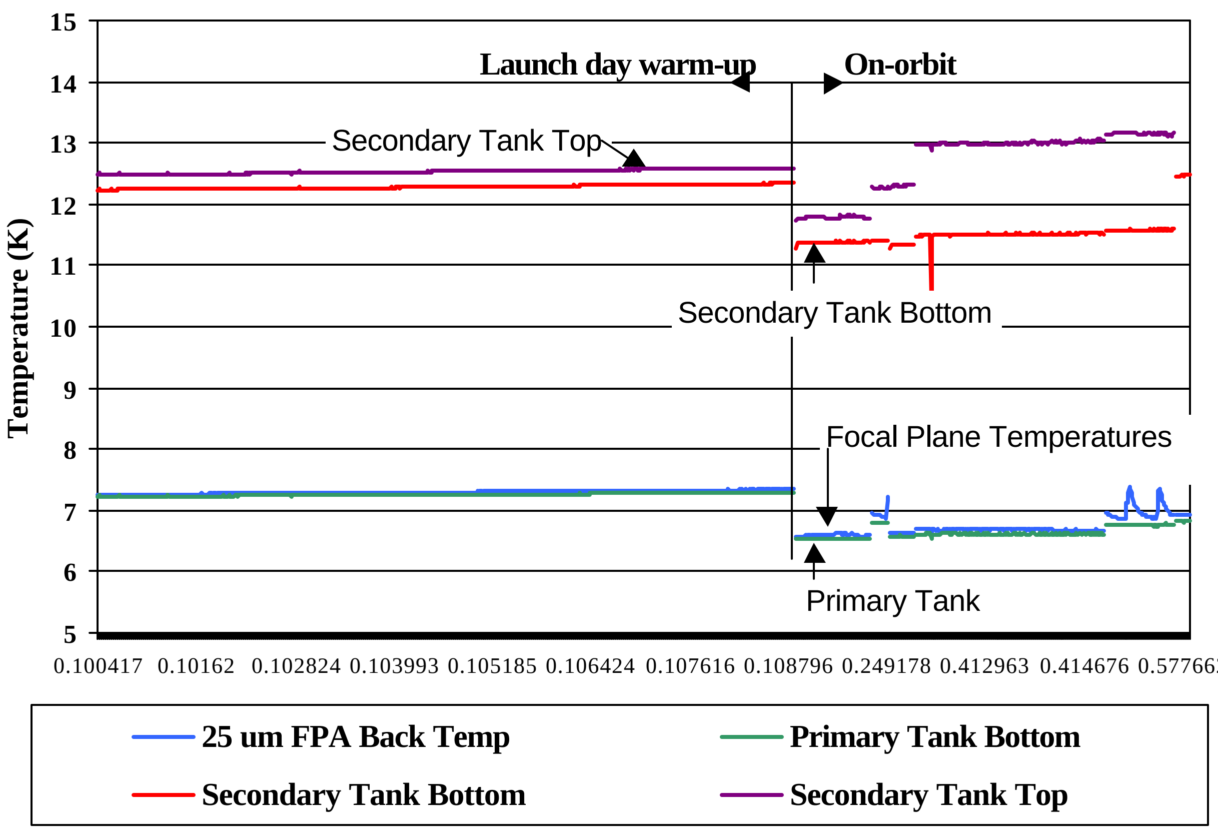

Figure 5: Flight Temperatures Showing Primary and Focal Plane Cooling Below 15 Kelvin

plane allowed the flight crew to monitor the cryostat status continually until the Pegasus was dropped.

This operation of disconnecting and recooling was a constant scheduling difficulty for the program. Exacting schedules had to be worked out to allow spacecraft and Pegasus work to occur. The personnel from GSFC, Orbital, JPL, SDL, and Lockheed-Martin worked continually to ensure that operations occurred timely and safely. It is to their credit that it was shown that a small cryogenic experiment was possible for a Pegasus launch vehicle.

On-Orbit Performance. The flight information was limited because of the untimely cover deployment that allowed an extremely high heat load to enter the secondary tank before the spacecraft was stabilized. An average of >40 watts was entering the secondary tank, but, despite this load, the hydrogen was able to remain solid during its limited life. The high heat loads occurred through the roughly 15-inch, open telescope as the spacecraft tumbled through views of the earth and sun. Even with the high heat loads into the system, the primary tank was still able to cool the focal planes to below 6.8 K while they were operating, see Figure 5.

Cryostat Summary. Cooling to temperatures below 15 Kelvin is necessary for many infrared missions. Solid hydrogen has been used in this application before, but WIRE proved that detector cooling below 7 Kelvin was possible for a space experiment. WIRE was able to maintain 6.8 Kelvin focal planes for ground testing and limited on-orbit data.

Many people at first thought it would be impossible to fly a cryogenic payload utilizing a flammable gas on a semi-manned Pegasus launch system within the budget and schedule of a SMEX mission. The safety concerns, operational complications, extra equipment, and constant servicing would stop the mission long before it ever got off of the ground. But the WIRE team answered each challenge with a solution. We analyzed hazards and worked processes and procedures from the beginning of the program to ensure safe operations for all involved from ground processing through launch. In the end, the glitch that ended the mission had nothing to do with the cryostat. WIRE showed that cryogenic experiments are within the reach of Pegasus-class SMEX missions.

Thermal System

Overview. The thermal control system for WIRE consists of flight heaters, radiators, and multi-layer insulation (MLI). The operational and survival heaters are thermostatically controlled. These heaters are cycled during cold mission phases for operational and survival conditions. The flight heater power predictions are provided in Table 1 for the cold operational and cold survival conditions. The sun side (+Y) of the spacecraft MLI has 0.127 mm (5 mil) silver teflon for the outer layer of the 18 layer blanket. The anti-sun side (–Y) has 0.076 mm (3 mil) kapton as the outer layer of the MLI. The photo of WIRE in Figure 1 shows the layout of the MLI.

Each of the WIRE electronics boxes uses its associated equipment panel as a dedicated radiator. The boxes are mounted to K-1100 composite panels with a sheet of Chotherm for thermal conductivity, and 2.5 cm (1 inch) wide copper tape wrapped around the edge of the Chotherm provides electrical conductivity from the box to the mount panel. The radiator areas were individually sized for each electronic component, and each radiator was painted with A276 white paint. The K-1100 composite, painted radiator panels are used to radiate heat to space from the exposed orthogrid surface. The thermal conductivity values for K-1100 in plane were tested earlier in the WIRE program.[6] The results of the test indicate an in-plane thermal conductivity of 260 W/m-K, which was used in the WIRE system thermal model.

Table 1: Flight Heater Power Predictions

Description	Heater Power (W)	Cold Operational		Cold Survival	
		% Duty Cycle	Predict Power	% Duty Cycle	Predict Power
SPE Operational Survival	 7 7	 Off Off	 0 0	 Off 79%	 0 5.5 W
SCS Operational Survival	 5 5	 Off Off	 0 0	 Off Off	 0 0
Star Tracker Operational Survival	 17 17	 Off Off	 0 0	 37 % Off	 6.3 W 0
Battery Operational Survival	 10 5	 66% Off	 6.6 W 0	 47% 100%	 4.7 W 5.0 W
Gyro Operational Survival	 7 10	 Off Off	 0 0	 Off 100%	 0 10.0 W
WIE Operational Failed Survival	 0 5	 Off	 0	 100%	 5.0 W
ACE Operational Survival	 7 7	 Off Off	 0 0	 Off Off	 0 0
TOTAL			**6.6 W**		**36.5 W**

Gamma alumina struts provide the mechanical interface between the instrument and the spacecraft and are shown blanketed with 18-layer MLI with an outer layer of 0.127 mm (5 mil) silver teflon. The struts were designed to minimize the conductive heat transfer from the spacecraft to the instrument cryostat. The thermal conductivity of the gamma alumina struts was measured with a low conductivity of 0.77±0.19 W/m-K.[7] A thermal skirt, also made of 18-layer MLI with an outer layer of 0.127 mm (5 mil) silver teflon, as shown in Figure 10 (prior to application of the silver teflon outer layer), was designed to reflect and minimize entrapment of solar energy from the cryostat shell.

The interior of the spacecraft is bare composite structure and the electronic boxes are black anodized to provide a high emittance of 0.87 and 0.81, respectively for internal radiation inside the bus structure. The reaction wheels radiate to the inside of the composite structure and black boxes. The battery panel is isolated from the composite frame with a 0.32 cm (1/8 inch) thick G10 fiberglass spacer. The baseline design assumed a panel-to-frame conductance no greater than 0.4 W/°C. However, test data from the system level thermal vacuum (TV) test showed it to be approximately 0.2 W/°C conductance. In addition, the battery was blanketed internally with 18-layer MLI with a 0.076 mm (3mil) kapton outer layer.

The instrument harness, which routed inside the structure from the WIRE Instrument Electronics (WIE) box to the instrument, is blanketed with a 6-layer cable wrap, vapor deposited aluminum (VDA) outer layer. This VDA MLI wrap minimizes heat loads from spacecraft to the harness and to the cryostat.

The star-tracker is mechanically mounted external to the structure on a graphite-composite stand. The M55J of the stand has low thermal conductivity. The tracker is thermally isolated from the bus and instrument to minimize heat transfer across the interface. The star-tracker was built and tested at Ball Aerospace. Ball designed the tracker to radiate from the tracker shade and not from the tracker body to minimize temperature gradients in the body. Ball had requested a radiator area on the shade of 0.05 m^2. We analyzed the design and adjusted the radiator area to 0.035 m^2. We changed Ball's original

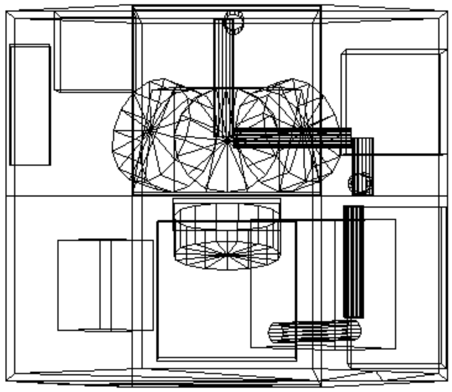

Figure 6: WIRE TSS Internal Geometric Model

design of black anodized radiator to silver teflon to meet the worst-case temperatures during the acquisition phase of the mission (immediately following launch). A coating specialist covered over the black anodized surface with 0.127 (5 mil) silver teflon tape. The GSFC blanket shop also built the 18-layer MLI that covered the tracker body and aperture shade externally. The interior of the aperture was painted black. Our analysis and thermal vacuum (TV) testing showed that most of the star tracker's heat radiated out of the aperture to deep space and only some of its energy radiated from the radiator built on the shade (-Y side).

<u>WIRE Thermal Model.</u> The geometric model of WIRE was built by GSFC using TSS (Thermal Synthesizer System) and consisted of an external model and an interior model of the spacecraft bus. The TSS geometric model of the interior of the bus is shown in Figure 6, which details the instrument harness, reaction wheels, and electronic boxes. The external geometric model, as shown in Figure 7, was used to calculate the environmental heat loads with TSS heat rate program. The external geometric model of the Ball star tracker and the instrument cryostat model from SDL were incorporated into the all-up system level

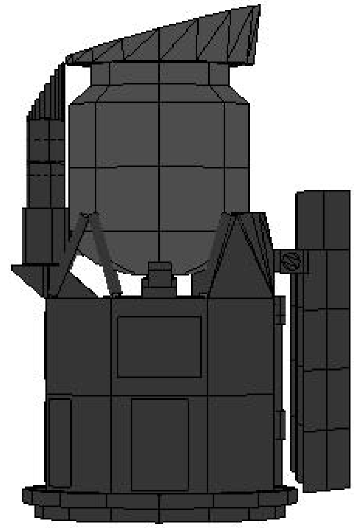

Figure 7: WIRE TSS External Geometric Model

model. GSFC adjusted the as-built radiator for the aperture shade. The WIRE TSS model was also used to generate the radiation couplings to space and to other spacecraft surfaces using the RADK program. The orbital parameters were defined in the ORBIT program of TSS and the animation file was viewed. The fluxes and radiation couplings calculated by TSS were included in a Systems Improved Numerical Differencing Analyzer (SINDA) thermal model of the WIRE Observatory. This SINDA model also included all conduction couplings, component power dissipations and heater logic for cold cases. The SINDA model consisted of approximately 600 nodes which represented the WIRE Observatory. The thermal model was used to predict on orbit flight temperatures for the various mission phases. The thermal model was correlated with test data from the system level TV test which was conducted in April/May 1998.

In the all-up TV system-level tests, the following test objectives were met: power dissipations were measured, MLI blanket effective emittance verified, heater duty cycles and heater performance verified, conductive heat paths were confirmed, and interfaces verified between the instrument and spacecraft and the tracker and spacecraft. From the system-level test, the tracker body and aperture shade are decoupled. Thermal balance conditions were simulated in the TV tests for three on-orbit conditions as shown in Table 2 to verify the thermal design.

Table 2: Simulated On-Orbit Conditions

Hot Operational	β=+90°, -30°	Tilt Away From Sun
Cold Operational	β=+90°, +15°	Tilt Towards the Sun
Cold Survival	β=+90°, +15°	Tilt towards the sun

Flight Validation. After the launch and recovery effort for WIRE, the thermal model was then correlated with flight data in support of the Mishap Board investigation efforts. The orbital parameters assumed a sun synchronous orbit with an altitude of 505 km and attitude of β=90° and boresite tilt angle of 10° towards sun. The data collected and correlated was for Orbit Day 82. The model assumed the environmental constants: solar constant of 1353 W/m^2, earth IR of 237 W/m^2, and solar albedo of 0.30 (unitless). The model also used Beginning of Life (BOL) optical properties for the MLI and radiators. The model correlation is provided in Table 3 and compares the actual and predicted temperatures. The thermal model correlates well with flight thermistor data within 0-2 degrees for all major components. The model, in general, predicts a few degrees higher for some of the components. The low cryostat shell temperature demonstrates the thermal isolation provided by the gamma alumina instrument-support struts.

Table 3: Flight Model Correlation

List of Components	Flight Temperature (In Celsius)		
	BOL Model Predicts	On Orbit Day 82	Temp Diff Δ
SCS	9	8	1
WIE	3	2	1
ACE	9	8	1
BATTERY	5	4	1
SPE	7	6	1
SHUNT	7	9	2
TRANSPONDER	20	20	0
GYRO	14	15	1
REACTION WHEEL Y	21	20	1
REACTION WHL A-C	16	14	2
DSS HEAD	27	27	0
DSSE	21	21	0
EARTH SENSOR +X	20	20	0
EARTH SENSOR –X	16	16	0
MAGNETOMETER HEAD	9	10	1
STAR TRACKER HOUSING	-3	-4	1
SOLAR ARRAY +X INNER	69	68	1
SOLAR ARRAY +X OUTER	74	75	1
SOLAR ARRAY –X INNER	75	74	1
SOLAR ARRAY –X OUTER	74	73	1
CRYOSTAT SHELL TOP (K)	191	191	0
CRYOSTAT SHELL BOTTOM (K)	194	194	0

The power dissipations used in the thermal model for all the electronic components are provided in Table 4. The bar chart in Figure 8 shows the qualification limits (white), predicted temperature range (dark blue), and actual temperature range (light blue) for the major WIRE components.

Table 4: Flight Power Dissipations for Orbit Day 82 (In Watts)

Components	Power Dissipation (In Watts)
SCS	19.5
WIE	7.9
ACE	33.6
Battery	2.9
SPE	8.0
Shunt	10.0
Xponder	8.9
Gyro	6.0
Reaction Wheel Y	3.0
Reaction Wheel A-C	3.0 each
DSS Head	1.0
DSSE	0.4
Earth Sensor +X	0.8
Earth Sensor –X	0.8
WAES	0.7
Pyro	Off
Magnetometer Head	0.1
X Torquer Rod	0.1
Y Torquer Rod	0.1
Z Torquer Rod	0.1
Star Tracker	6.9

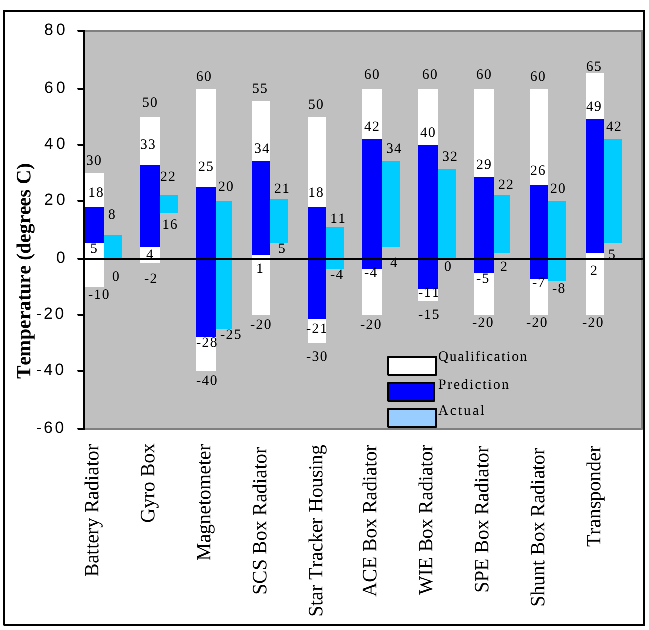

Figure 8: Qualification, Prediction, and Actual Temperature Extremes for WIRE Components

<u>Thermal System Summary.</u> During and after the recovery effort, the thermal subsystem was completely checked out with on-orbit flight temperature data. The thermal system is performing as expected and flight temperatures have been nominal since launch. High-quality thermal balance testing on the ground has paid off with excellent correlation between the thermal model and the flight data. We have a valuable analytical tool, a thermal model of the WIRE system, used to predict on-orbit temperatures during the life of the mission. This model correlation can enhance future modeling techniques used on other flight programs using composite structures.

Composite Structure

The primary reason WIRE used a composite structure was to save mass. Early designs allocated 28% to 32% of WIRE's total mass to the instrument, but the composite structure enabled WIRE to carry an instrument and its related hardware at 41% of the total spacecraft mass. The final structural mass was only 11% of the total--about half the weight of a conventional aluminum structure. The tremendous weight savings was made possible by the fully-bonded graphite composite structure built by Composite Optics, Incorporated (COI).[8]

Table 5 lists the mass of the spacecraft components. The instrument total includes the cryostat cover and the hydrogen. The thermal system includes heaters, thermostats, and blankets. The 0.13 mm silver-teflon outer layer added 3 kg to the spacecraft bus blankets, a late surprise for all of us. Table 6 shows the inertia matrix for the spacecraft in its current configuration, without the 6.2 kg instrument cover or 4.6 kg of hydrogen.

Table 5: Measured Mass Distribution in Launch Configuration

Subsystem	Actual Mass (kg)	% of Total
Instrument and support	107	41%
Launch vehicle hardware	4.0	2%
Structure	27.9	11%
Mechanisms	3.6	1%
Power Electronics	9.6	4%
Battery	11.7	5%
Solar Array	9.7	4%
ACS	44.8	17%
Data system	8.0	3%
RF system	4.3	2%
Thermal system	11.0	4%
Electrical harness	17.4	7%
Total	259	100%

Table 6: WIRE Inertia Matrix, On-Orbit Configuration with Tanks Empty ($k\text{-}m^2$)

I	X	Y	Z
X	79.23	-0.39	0.86
Y	-0.39	75.81	-6.12
Z	0.86	-6.12	33.78

Figure 9 shows the history of WIRE's estimated mass. The increase from proposal to definition-phase baseline reflects the change from the standard Pegasus to the XL, a baselining of the Submillimeter Wave Astronomy Satellite (SWAS) spacecraft bus, and an increase in instrument aperture from 25 to 28 cm. The large spike at the beginning of the implementation phase reflects a 50% mass growth of the cryostat as the instrument grew to 30 cm and the cryostat engineering team took a more detailed look at the design. Also at this time, we selected a more realistic orbit which reflected the large (20 km x 90 km) Pegasus dispersions and the approach of solar maximum. We began careful tracking of the spacecraft mass and soon discovered another 9 kg of instrument-support hardware and electronics which were not included in the total estimates. We switched to a composite structure, and the resulting margin accommodated additional mass growth of other components and a small increase of the orbit altitude. The system design review and the start of spacecraft integration and test (I&T) are marked on the chart.

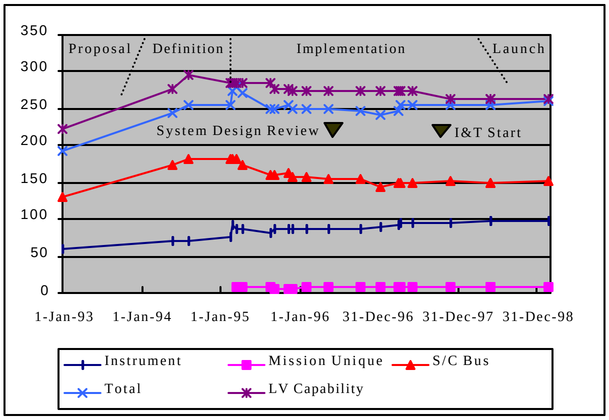

Figure 9: Mass History

The structure was built with weight savings in mind, so we could not afford copper clad decks for grounding. Since most of the electronics boxes on the spacecraft communicated via differential signals such as the MIL-STD-1553 bus, a low-impedance spacecraft ground was unnecessary. It was only necessary to provide enough connection between electronics to dissipate and distribute charge buildup. By adding nickel spheres to the epoxy for bond-line control (instead of the usual glass spheres), we assembled a composite structure with good conductivity between any two points (< 50 ohms). For the one case where we did have single-ended signals between the attitude control electronics and the gyro package, we added a copper ground strap to ensure a good ground reference. Throughout all spacecraft operations, we have had no adverse effects due to noise. During electronic integration, we specifically looked for high noise levels and found none. WIRE proved that grounding on a composite structure does not need to be expensive.

Power System

The spacecraft uses a direct energy transfer (DET) power system where the gallium arsenide solar arrays are diode-ORed directly onto the main power bus. The battery is also directly across the bus, providing the voltage reference for the system. All electronics operating off the main spacecraft power must handle 28 ± 7 V. We also require survival of 0 to 40 V indefinitely without damage to protect against mistakes during ground tests.

The battery charge control circuitry shorts half strings of the solar array as the battery reaches a pre-set voltage, tapering the battery current. This voltage varies automatically with temperature. When the amp-hour integrator (AHI) circuit determines that the battery is fully charged, a current controller takes over and maintains a constant 90 mA battery trickle charge. Both the voltage/temperature (V/T) controller and the AHI are analog circuits, providing battery charge control independent of the spacecraft computer.[9]

WIRE has two deployed solar panels, each made from nine solar array modules bonded to a composite frame. This modular design allowed early procurement of the individual solar array modules in an easy-to-handle format, with later sizing and assembly of the composite frame, eliminating the solar arrays as a schedule driver in WIRE's development. This modular design had minimal impact on the mass of the arrays, and very little surface area was lost (see Figure 10). The 20.9 cm x 43.6 cm modules require 0.4 mm of epoxy around the perimeter, a 1.0 mm between modules, and 1.0 mm of composite around the perimeter of the panel. Only 1.6% of WIRE's 1.67 m^2 surface area is lost to the mounting of the modules on the frame. The composite modular arrays on WIRE achieved 5.8 kg/m^2 as compared with 4.4 kg/m^2 for honeycomb arrays--a bit high, but WIRE's solar array has a 45-degree bend which would add some mass to a honeycomb panel. Also, WIRE incorporated antenna mounts in the panel, another item that would add more mass in an aluminum honeycomb implementation.

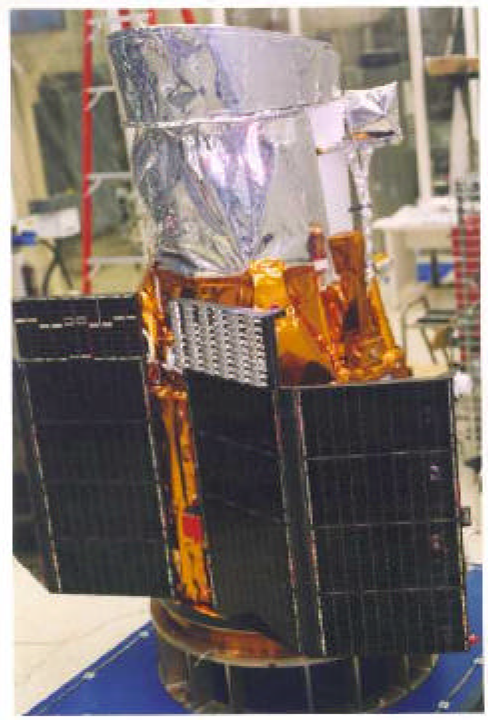

Figure 10: Sun-side View of WIRE Prior to the Installation of Silver-Teflon Outer Blanket Layer

WIRE required only sixteen modules to meet its power needs, so the other two modules were devoted to flight experiments. The +x panel carries a dual-junction gallium arsenide module, and the -x panel carries a concentrator module. Each test module has a thermistor mounted on its back to measure temperature and a series resistor to measure current. Each panel has one other thermistor, and a series resistor provides a current reading for the entire panel, including the test module.

Plots for each of the panels vs. sun angle are shown in Figure 11. Note how the concentrator module causes the -x panel to deviate from the cosine law. Effects from cover glass reflections are visible in the +x data.

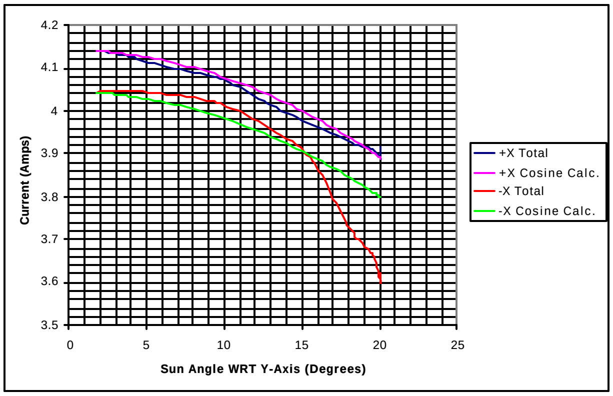

Figure 11: Plot of solar array current vs. sun angle at 31 V, on day 159.[10]

Table 7 lists the average power output of the solar array as measured during four different passes over 16 months of on-orbit operations. Temperatures across the panels were 68°C to 72°C. The passes were selected because of their identical attitude and mode configurations. Little variation in output was seen over the 16 months. The table lists power delivered to the bus, including diode drops. Current was measured with a series resistor, and voltage was measured across the bus. The dual junction module produces 14.8 W at 31.2 V, 104% of the 14.2 W produced by the 8 single-junction modules on the +x panel. The concentrator module produces 10.6 W, 75% of the single-junction modules, using only 33% of the solar cell area, reducing the production costs.[11] The impact of the direct energy transfer (DET) power system is obvious in the power per area values for the WIRE arrays. Even accounting for the 22.5 degree tilt of the panels, the GaAs modules only produce 166 W/m^2. They are being operated at the bus voltage, 31.2 V, which is far below the peak power point of nearly 40 V. The modules were designed for a minimum open-circuit voltage of >35 V at 100°C at the end-of-life to support a variety of missions. The DET system was selected for its simplicity, reliability, and lower cost. A peak-power tracker would deliver > 30% more power from the arrays.

Table 7: Solar Array On-Orbit Performance

	Power (W)	Power per Area (W/m^2)	% of Single-Junction
+X Panel	129	154	99.0%
+X w/o Experiment	114	154	98.6%
-X Panel	124	149	95.5%
-X w/o Experiment	114	153	98.2%
Dual-Junction Module	14.8	162	104.0%
Concentrator Module	10.6	116	74.7%
Single-Junction Module	14.2	156	100.0%

Table 8 lists the power consumption of the WIRE spacecraft. These values are averaged over two different passes with the same attitude and ACS mode. Harness losses are included with each component, since current and voltage were measured in the spacecraft power electronics box. The orbit average transmitter power assumes an 11% duty cycle (one pass per orbit). The ACS power will peak substantially higher during a slew, but the average is not impacted much (<1%).

Table 8: Measured Power Consumption

	Xmitter On	Orbit Average
ACS	54.7	54.7
Star Tracker	7.3	7.3
Power Electronics	13.0	13.0
Battery Trickle	1.2	1.2
Spacecraft Computer	19.9	19.9
Receiver	5.6	5.6
Transmitter	32.4	3.6
Heaters	0.0	0.0
Bus Total	**134**	**105**
Instrument	26.8	26.8
Spacecraft Total	**161**	**132**

Figure 12 shows the history of WIRE's estimated power consumption and production. The available power assumes a 15-degree tilt of the y-axis from the sun line, and it neglects power from the experimental modules. The margin remained high throughout the implementation phase, enabling the flight of the experimental test modules. The estimated available power increased during I&T when we measured the flight solar array output. At about the same time, we dropped the estimate for heater power based on the spacecraft thermal balance testing.

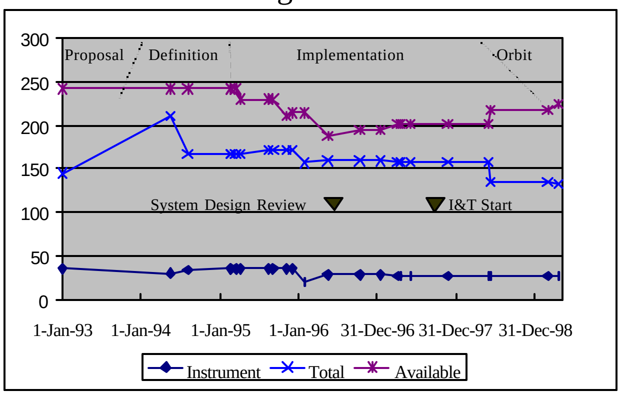

Figure 12: Power History

Earth Sensor

The WIRE spacecraft flew a new Wide Angle Earth Sensor (WAES) designed and manufactured by Servo Corporation. This sensor was based upon two dual-element infra-red scanners to achieve a nearly linear measurement of Earth angle over a 120^o field-of-view with a low-production-cost sensor.

Unfortunately, this sensor failed during the cryogen release phase of the mishap, leaving little in-flight data available to verify its performance. Ability to determine performance during the short post-launch phase in which this sensor was operational is also limited because the fine attitude sensors were not yet operational. However, the data indicates that the sensor was probably functioning correctly until well into the mishap.

Data System

The WIRE data system uses a radiation-hard 80386 processor and a 80387 math co-processor running at 16 MHz. The 300 Mbyte solid-state recorder resides on a single card. Dynamic random access memory (DRAM) circuits are refreshed by hardware on the card. The card has built-in error detection and correction (EDAC) circuitry for the inevitable bit errors caused by the orbital radiation environment. The EDAC uses 20% of the memory, so 240 Mbytes was available for storage of mission data. DRAM was selected to reduce the cost and increase the memory of the TRACE and WIRE data systems. Three cards were produced for TRACE and WIRE (the two flight units plus a spare) at a cost of $90,000 per card plus one man-year of labor supporting all three.[12]

A background task in the processor "scrubs" the memory by reading from each location and re-writing, with the EDAC correcting single-bit errors. All single-bit errors are logged, and multi-bit errors create an event message for ground controllers. WIRE experienced 10,100 single-bit errors in its first year of operation, an average of over 27 per day, concentrated in the South Atlantic Anomaly and the polar regions. During the same time, there were only 3 multi-bit errors. The processor has experienced no restarts due to watchdog timeouts, radiation hits, or software errors.

Ground System

The WIRE ground system uses the same software that controlled and monitored the spacecraft during integration and test (I&T). The Integrated Test and Operations System (ITOS) was originally developed as the I&T

system for all of the Small Explorer (SMEX) spacecraft at Goddard. As each mission was developed, the ITOS team added necessary features while maintaining compatibility with previous missions. The result is a system which boasts over 10,000 hours of ground-test time with flight spacecraft, a system which is now operating five SMEX missions at low-cost with high reliability.

The high reliability of ITOS has enabled the flight operations team to add additional autonomous capability to the flight operations environment. The Spacecraft Emergency Response System (SERS) automatically sends text messages detailing critical spacecraft events to a prioritized list of spacecraft operators. Two-way paging ensures that the page has been received and acted upon.[13]

The automated system normally handles both WIRE passes per day, although a person will supervise special command loads or spacecraft experiments. The SERS paging system has worked well, enabling the rapid identification of dropped telemetry (usually due to ground station problems in the field) and anomalous spacecraft conditions. The contacted operator has the information necessary to immediately decide whether an emergency or contingency pass must be scheduled, and whether an operator must actually be present at the console for that pass. The automated ground system has enabled the FOT to operate four spacecraft with a full-time staff of thirteen people (including managers, secretary, and system support personnel) working a single eight-hour shift five days per week, while still recovering over 99% of the data collected in orbit.

Science and Experiments

The biggest payoff of WIRE's recovery has been the science and engineering experiments. In May 1999, we began astroseismology measurements with WIRE. Since that time, we have expanded WIRE operations to include other test bed activities.

Astroseismology

Astroseismology is the study of oscillations in stars. Just like seismologists study the interior structure of the earth, scientists use astroseismology measurements to determine the interior structure of stars by studying the propagation of seismic waves. Many different modes of oscillation have been observed in the sun, and high amplitude oscillations have been detected in other stars, but no multi-mode oscillations had been unambiguously detected in any cool stars other than the sun.

Using the WIRE star tracker, the astroseismology team discovered several oscillations of Alpha Ursae Majoris with amplitudes of 100-400 μmagnitude and 1.82 μHz fundamental frequency.[14] This discovery was the first of its kind, since ground-based observations cannot detect such small variations due to the turbulence of the earth's atmosphere.

WIRE Test Bed

"The WIRE test bed provides an affordable and accessible on-orbit spacecraft to enable science observations, accelerate technology readiness and infusion, and promote educational outreach."[15] An experimenter interested in using the WIRE spacecraft submits a proposal. The WIRE team evaluates the proposal for feasibility and helps the proposer get a sponsor and estimate costs. Five different experiments have been executed on WIRE, eight are currently being worked, and seven more are currently being studied. Code S at NASA Headquarters has received at least ten new proposals for science observations.

In May 2000, the geostationary operational environmental satellites (GOES) project at Goddard and Ball Aerospace conducted an experiment on the WIRE test bed. The WIRE ACS uses a Ball CT-601 star tracker for fine pointing. Several other low-Earth orbit (LEO) satellites also use this tracker, and all of these satellites have experienced brief loss of track associated with the South Atlantic Anomaly and the polar regions. As part of the WIRE test bed program, Ball Aerospace analyzed WIRE tracker data and uplinked a software patch to correct the problem. In-flight testing on WIRE has demonstrated that the patch is effective, reducing the risk for the future GOES missions which plan to use a modified version of the CT-601 in the geostationary environment, where solar protons could have a much greater impact on flight operations.

Summary

The Wide-Field Infrared Explorer did not take a single infrared exposure. But WIRE's robust attitude control system enabled the team to recover the satellite after the tragic mishap. Subsequent operations successfully demonstrated the superior performance of nearly all of WIRE's subsystems. The mission clearly demonstrated the viability of a hydrogen cryostat on a Pegasus vehicle. Now, in addition to advancing space flight technology, WIRE is advancing science through novel use of its star tracker. The positive results from WIRE will have a lasting impact on space science.

Acknowledgements

Martin Taylor, Wendy Jones, and the rest of the WIRE flight operations team provided excellent support to the authors by gathering the flight data necessary for the completion of this paper. The entire WIRE team[16] devoted exceptional effort to the design, build, launch, and recovery of the spacecraft.

References

[1] Branscome, Darrell R. (chairman of mishap board) and others, "WIRE Mishap Investigation Board Report," http://rk.gsfc.nasa.gov/richcontent/Reports/wiremishap.htm, June 8, 1999.

[2] Everett, David F., "WIRE Anomaly Information," WIRE Web Site, http://sunland.gsfc.nasa.gov/smex/wire/mission/anomaly.html, February 2000.

[3] Laher, Russ, and others, "Attitude Control System And Star Tracker Performance Of The Wide-Field Infrared Explorer Spacecraft," *AAS/AIAA Space Flight Mechanics Meeting*, Clearwater, Florida, January 23-26, 2000, Paper AAS 00-145.

[4] Costanzo, Brenda J., P. A. Menteur, Scott Schick, and W. G. Foster, "Design and Performance Analysis of the Wide-Field Infrared Explorer H2/H2 Cryostat," *Proceedings of SPIE*, Vol. 2814, p. 147-153, Cryogenic Optical Systems and Instruments VII, 1996.

[5] Murray, David, J. Clair Batty, and Scott Schick, "Cryogenic System for the Wide-Field Infrared Explorer," *Proceedings of SPIE*, Vol. 2227, p. 190-198, Cryogenic Optical Systems and Instruments VI, 1994.

[6] Parrish, Keith, "The Use of High Thermal Conductivity Composites in the Satellite Bus Structure of the Wide-Field Infrared Explorer", *Proceedings of the Space Technology & Applications International Forum (STAIF-97)*, Albuquerque, NM, October 1996.

[7] Powers, C. of NASA's GSFC in a technical memo, January 9, 1997.

[8] Rosanova, Giulio G., "Composite Bus Structure for the SMEX/WIRE Satellite," *Proceedings of the 12th Annual AIAA/USU Small Satellite Conference*, SSC98-IV-4, Logan, Utah, 1998.

[9] Everett, D. F. and L. Sparr, "Wide-Field Infrared Explorer Spacecraft System Design," *Proceedings of the 1996 IEEE Aerospace Applications Conference*, Snowmass at Aspen, Colorado, February 3-10, 1996, Volume 2, pp. 145-158.

[10] Lyons, John, memo on solar array performance, August 16, 1999.

[11] T. Stern and J. Lyons, "Flight Test of a Technology Transparent Light Concentrating Panel on SMEX/WIRE," *Proceedings of the 35th Intersociety Energy Conversion Engineering Conference*, Las Vegas, Nevada, July 24-28, 2000.

[12] Voyton, Mark at NASA/GSFC, June 13, 2000.

[13] Prior, Mike, Keith Walyus, and Richard Saylor, "Autonomous Command Operations of the WIRE Spacecraft," *Proceedings of the Second International Symposium On Spacecraft Ground Control And Data Systems (SCD II)*, Iguassu Falls, Brazil, February 8-12, 1999.

[14] Buzasi, Derek and others, "The Detection Of Multimodal Oscillations On α Ursae Majoris," *The Astrophysical Journal Letters*, Volume 532, April 1, 2000, pp. L133–L136.

[15] Crouse, Patrick, WIRE Test Bed Web Site, http://wiretestbed.nascom.nasa.gov/.

[16] Everett, David F., "The WIRE Team," WIRE Web Site, http://sunland.gsfc.nasa.gov/smex/wire/mission/credits.html .

Biographies

Dave Everett is the system engineer for the WIRE mission. After working on radar receivers for 5 years at Westinghouse, he joined NASA/Goddard Space Flight Center in 1991. He was a test conductor on the SAMPEX mission (launched in July 1992), and he was electrical system engineer on FAST (launched in August 1996), before taking the lead technical role on WIRE in 1994. He has a BSEE from Virginia Polytechnic Institute and State University (1986) and an MSEE from the University of Maryland (1989). Mr. Everett is currently the senior system engineer at Goddard's Integrated Mission Design Center.

Thomas Correll is the Lead Engineer for the WIRE Attitude Control System. He has worked on Attitude Control Systems at NASA/Goddard Space Flight Center since 1984, including those for the SAMPEX, SWAS, TRACE, and WIRE missions. He has a BS (1986) and an ME (1987), both in Electronics Engineering from Rensselaer Polytechnic Institute. Mr. Correll is currently working on Attitude Control System testing for the Triana mission.

Scott Schick is the lead cryogenic engineer on the WIRE program and is a senior engineer at Space Dynamics Laboratory. He has a masters degree in mechanical engineering from Utah State University (1990). He worked on SPIRIT III, which was the first solid hydrogen experiment flown in space. Mr. Schick is experienced in cryogenics, thermal management, and integration for infrared instruments used in space.

Kimberly D. Brown is the lead thermal engineer for the SMEX/WIRE mission. She was the lead spacecraft thermal analyst on COBE from 1985 through launch in 1989. She then led the instrument thermal engineering effort for the TRMM Mission from 1991 through launch in 1997. Mrs. Brown received her M.S. in Chemical Engineering from the University of Virginia (1985) and B.S. in Engineering/Physics and B.A. in Chemistry from West Virginia Wesleyan (1982). She is currently the lead thermal engineer of the Microwave Anisotropy Probe (MAP) Observatory and a group leader of the Spacecraft and Instrument Design Section in the Thermal Engineering Branch, Code 545.

Reference No: SSC00-V-1
Session: V (Lessons Learned in Success & Failure)