3.0 FLIGHT HARDWARE SUBSYSTEMS

3.1 INSTRUMENT DESCRIPTIONS

3.1.1 General Description

The WIRE Instrument is a cryogenically cooled, infrared telescope. The telescope is a Ritchey-Chretien Cassegrain design with a 30 cm aperture/primary mirror, 3.3 cm focal range and a 33x33 arc minute field of view.

The WIRE instrument has no moving parts once the vent valves have been opened and the aperture cover is removed on orbit. The entire optical assembly is an integral, independent module that plugs into the cryostat. The Cryostat is a two stage, solid hydrogen design. It will cool the detector arrays to 7.5 K and the optics to about 12 K.

3.1.2 Structural/Mechanical Subsystems

The WIRE instrument is mounted on the forward end of the spacecraft bus structure. The mechanical configuration of the instrument consist of an aluminum housing and deployable aperture cover. The cryostat incorporates “four mounting pads” for mounting to the spacecraft. The instrument is made up of several components: the telescope (which encompasses the optics and two focal plane arrays), a cryostat, an aperture shield, external plumbing and vent system, and the WIRE instrument electronics (WIE).

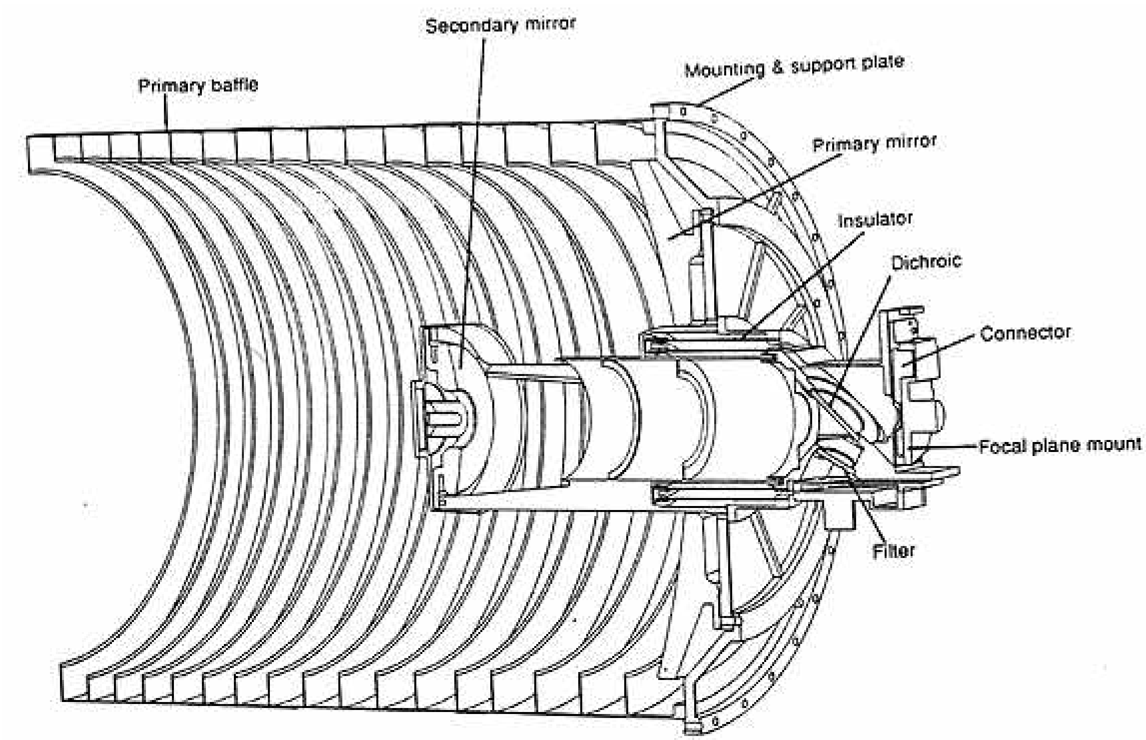
3.1.2.1 Telescope Subsystem

The WIRE telescope gathers the optical energy from the targets of interest and directs it to the focal plane detectors. Figure 3-1 shows a sectional view of the WIRE Telescope Assembly. The telescope design is an on-axis Ritchey-Chretien configuration with no moving parts. The diameter of the entrance aperture is 30 cm, the focal length is 100 cm, and the aperture stop is at the secondary mirror. Weighing roughly 12 kg, the telescope structure including the baffles, is made of aluminum with the exception of some G-10 fiberglass insulating tubes. The telescope is thermally and structurally coupled to and completely contained by the cryostat system. Structural analysis shows positive margins of safety for mechanical loading environments specified in the WIRE Single Design Review. The telescope cavity itself contains no cryogens.

The telescope design imposes a weak hyperbolic shape on the two WIRE mirrors, which are made of 6061-T6 stress-relieved aluminum. The mirrors, as well as their interfaces, are diamond turned and are Denton coated with evaporated gold. G-10 fiberglass insulating rings thermally isolate the mirrors and the post-optics assemblies.

The post-optics assemblies include a silicon dichroic beamsplitter that separates the incident energy into two, broad wavelength bands centered around 12 and 25 µm. These beams illuminate

Figure 3-1 Sectional View of WIRE Telescope Assembly



two arsenic-doped silicon focal plane arrays of the back illuminated blocked impurity band (BIB) design. The format of the arrays is 128 x 128 pixels; their pixel size is 75 μm. The range of the primary spectral band is 21 to 27 μm; the range of the secondary band is 9 to 15 μm. The post optics also include a germanium passband filter that along with the dichroic is held captive in an aluminum cubical housing.

Each detector is mounted to a beryllium thermal post providing good thermal conductivity and rigidity to survive launch loads exceeding 14 G's. The post, in turn, is shrink fitted to a copper cup that is linked by copper braiding to a second beryllium ring. The ring is connected via a shrink fitting to an aluminum hub off the primary tank. See Figure 3-2. This beryllium-to-aluminum interface provides excellent heat transfer. For the electronics, a Kapton cable is mounted to a connector on each focal plane; the cabling passes through the shrink-fit assembly, penetrates the inner and outer vapor shields, and exits through connectors in the vacuum shell to the WIRE electronics box.

3.1.2.2 Cryostat Housing Assembly

The WIRE cryogenic support system is composed of three pressure/vacuum vessels: the primary hydrogen tank, the secondary hydrogen tank, and the outer vacuum shell. The telescope/sensor and the two hydrogen tanks are enclosed in the same vacuum shell. A cross-sectional drawing of the WIRE instrument is shown in Figure 3-3.

Figure 3-2 Focal Plane Beryllium Mount Shrink Fit

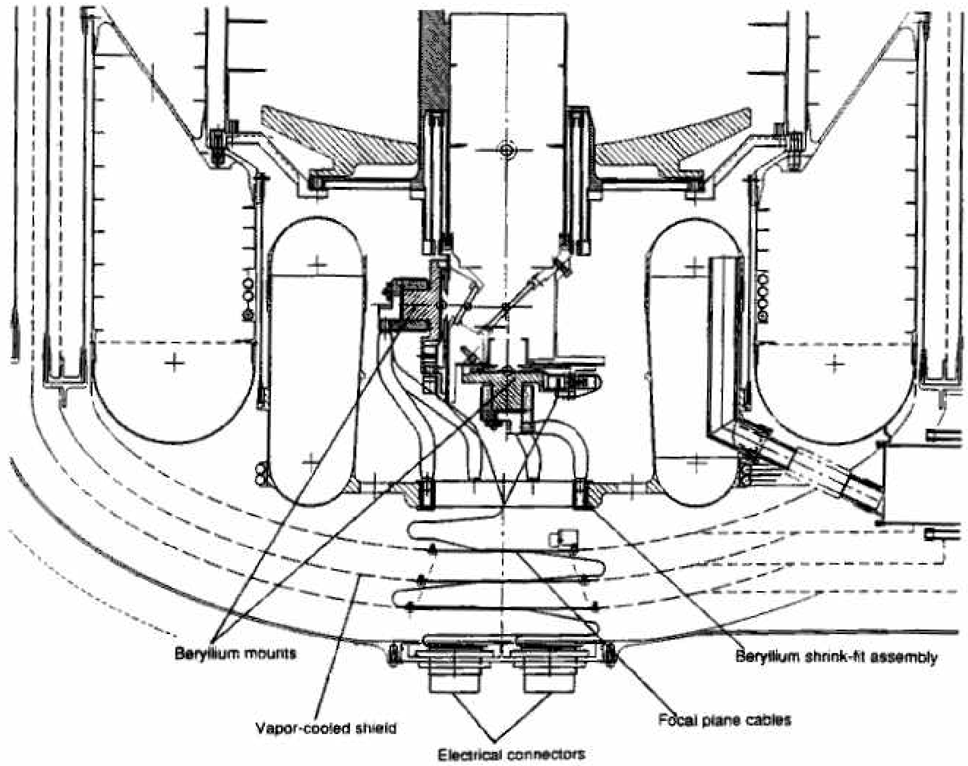
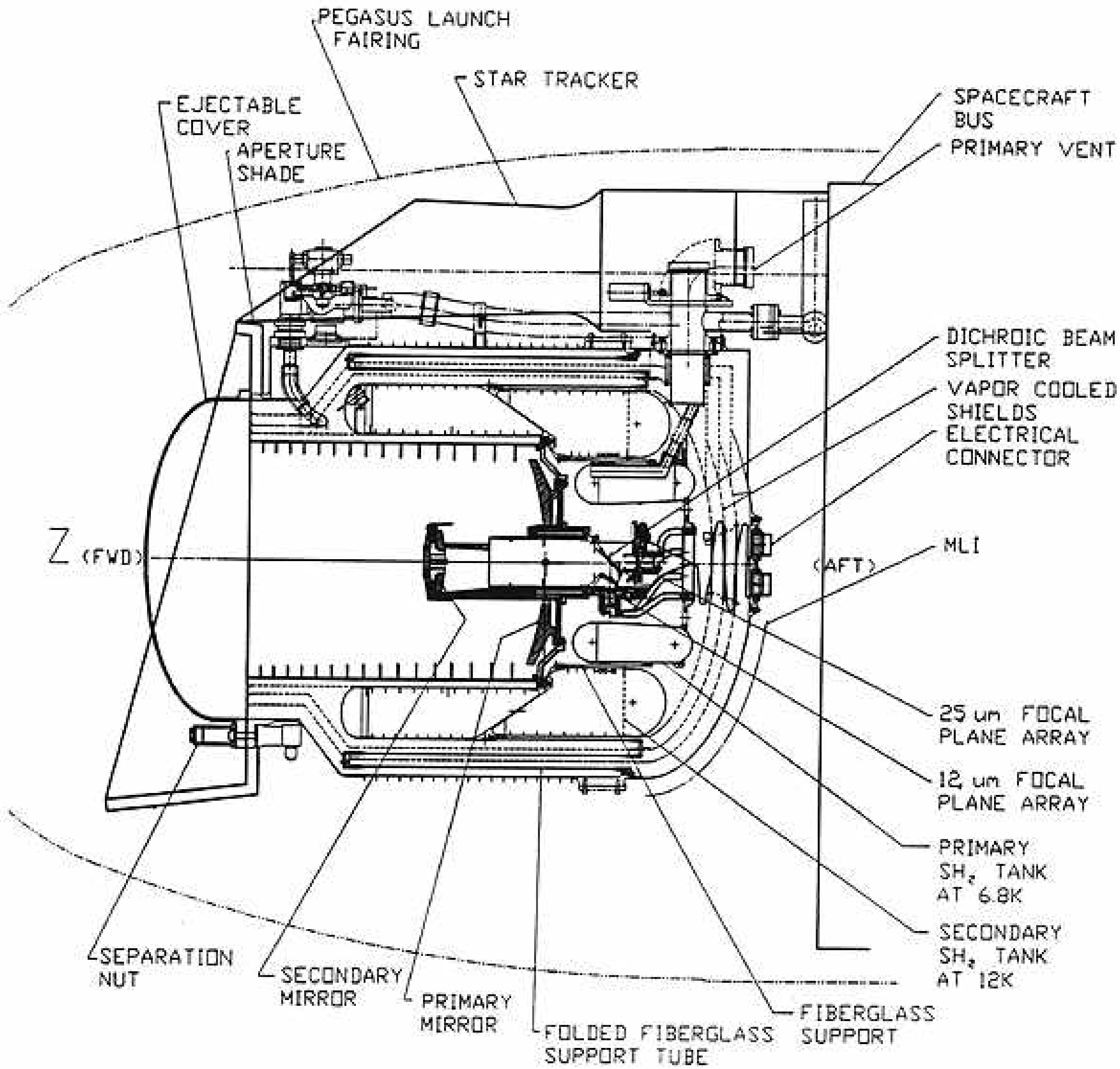
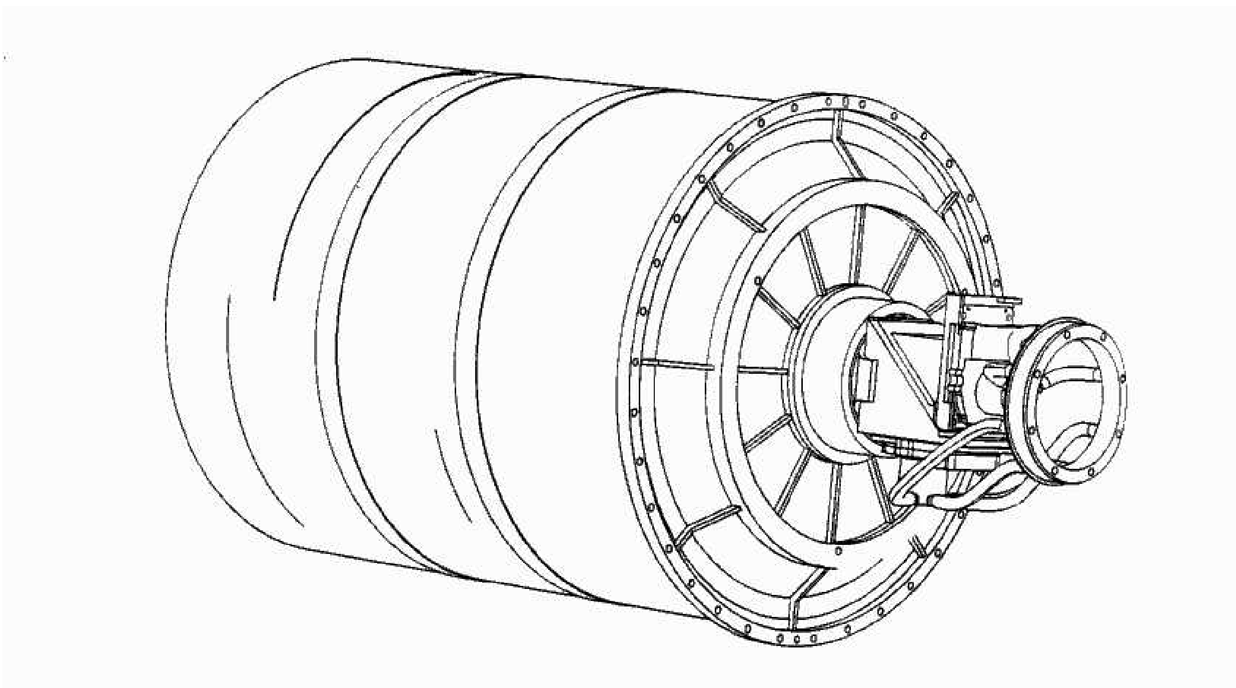


Figure 3-3 Cross Section of WIRE Instrument



The WIRE telescope, the telescope with its mirrors, baffles, and post-optics are contained in an aluminum cylindrical structure. See Figure 3-4. Emerging from the cylinder's aft section is a narrow "neck" containing the focal plane assembly. The two angular shaped hydrogen tanks, encircle the telescope-focal plane housing. The smaller, primary tank surrounds only the narrower “neck”. The secondary tank, which is the larger toroid, surrounds both the primary tank a significant section of the larger enclosure.

Figure 3-4 WIRE Telescope Structure Showing Baffle and Post Optics



The telescope structure and the secondary tank are supported off the vacuum shell by three concentric fiberglass epoxy tubes. The inner tube is connected to the secondary tank by epoxy and screws. All three tubes are connected together by epoxy bonding to aluminum rings that are quasi-match machined based on actual tube dimensions. The outer of the three tubes is bonded to a ring that is bolted to the vacuum shell. The primary tank is supported off the secondary tank by means of a fiberglass epoxy tube. The design type and criteria have been used on all previous Lockheed-Martin Missiles and Space (LMMS) cryostats, including Cryogenic Limb Array Etalon Spectrometer (CLAES) and Space Infrared and Imaging Telescope (SPIRIT III).

Four mounting pads machined into the aft cylinder of the WIRE vacuum shell provide the structural and mechanical interface to the spacecraft. The primary support for the external plumbing and its bracketry is at the forward end of the vacuum shell.

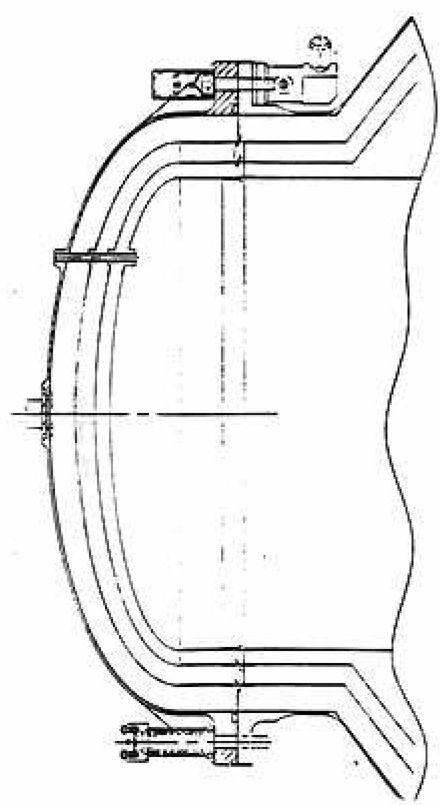
Before and during ground and prelaunch operations, the telescope will be protected by an aperture cover. In about 2-4 days after launch, the cover will be ejected when it is established that the spacecraft is in orbit and the surrounding environment will not contaminate the telescope optics. All components of the telescope assembly are completely enclosed within the cryostat vacuum shell, and it will not be opened until the door is ejected.

After the door has been deployed, the telescope aperture is exposed to environmental heat loads associated with solar, albedo, and earthshine. To reduce the impact of these external heat sources on the aperture, an aperture shade has been provided. This shade is a two-stage aluminum structure supported off the WIRE external vacuum shell. The inner stage is radiatively cooled to ~ 100 K and is supported from the outer stage using three fiberglass tubes. The geometry is that of a truncated half-right fustrum positioned to shade the aperture from direct solar heat, albedo, and earthshine.

3.1.2.3 Aperture Door Assembly

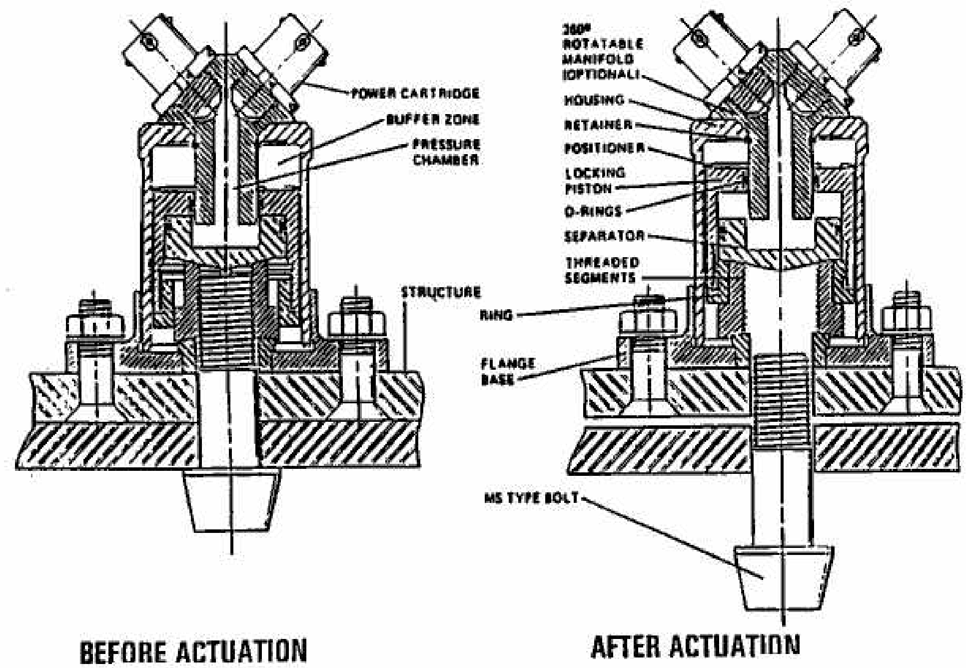
The WIRE instrument employs a fully ejectable door to cover the telescope aperture. Figure 3-5 shows the aperture door. This door provides the vacuum close-out about the telescope aperture while on the ground and is ejected free from the vacuum shell/instrument once in space.

Figure 3-5 Aperture Door



After launch, the aperture door will be deployed leaving a clear field of view for the telescope. Simultaneous initiation of the three pairs of pyrotechnic initiators releases the fasteners for the door as shown in Figure 3-6. The expanding gas from the initiator pushes a plunger, expanding the threads holding the bolt in place. During the travel, the plunger contacts the bolt, ejecting it from the nut release mechanism.

Figure 3-6 WIRE Door Nut Release Mechanism

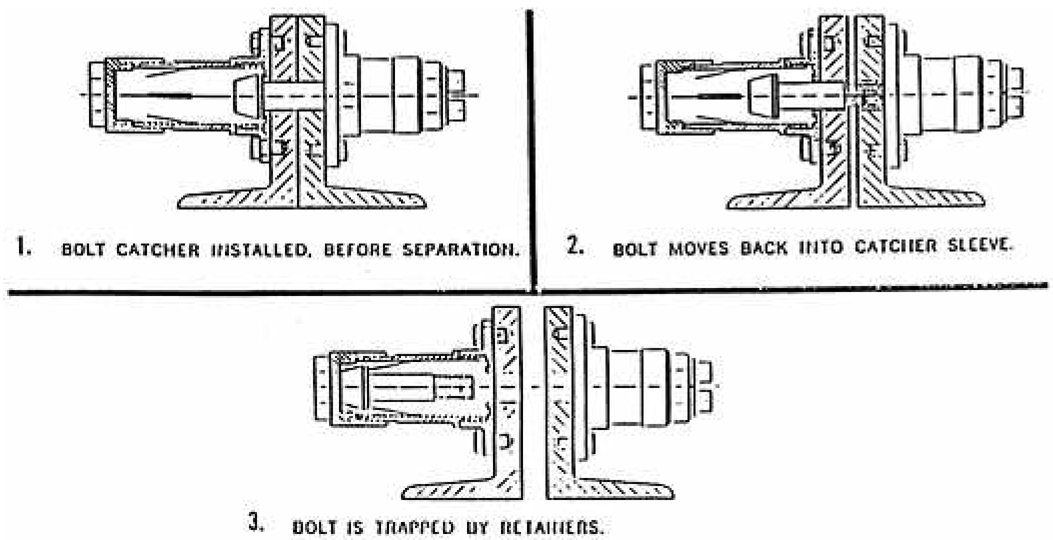


The major components of the door ejection mechanism include six pyrotechnic initiators, three nut release mechanisms, three bolt catchers and six spring plunger assemblies. The mass of the ejected system is approximately 4.2 kg.

The expelled bolt is trapped by the bolt catcher, shown in Figure 3-7. The bolt catcher is a cup with steel tabs facing aft and extending into the interior. The expelled bolt collapses the tabs as it passes into the cup and is caught in place when the tabs spring back into position. Both the bolt release mechanism and the bolt catcher are made by OEA Mfg., Inc. The door is pushed away from the cryostat by six, 23.9-lb nominal, springs that remain captive in the door flange. The calculated steady-state velocity of the ejected door is approximately one meter per second relative to the spacecraft.

If the bolt release mechanism is inadvertently actuated, the outcome would vary depending upon the number of bolts released and the situation in which they released. If all three of the nut release mechanisms were actuated on the ground, no hazard is expected since the pressure differential between the internal vacuum of the cryostat and the ambient air pressure would hold the door in place. During captive carry, above 45,000 feet altitude, however, the door could be expected to release if all the bolts were actuated inadvertently because the ambient air pressure would no longer provide enough differential to hold the door in place. Once released, the 4.18-kg door would be loose inside the payload fairing and could present a hazard to the spacecraft bus. If only one or two of the bolts released during captive carry, a worst-case analysis indicates the vacuum seal may be compromised and the WIRE experiment lost because of contamination; however, warming of the hydrogen (H_2) will result and gas will be vented through the safety vent. Finally, if the bolt release mechanism does not activate on orbit when commanded, the WIRE telescope cannot function.

Figure 3-7 WIRE Ejectable Door Mechanisms Bolt Catcher



3.1.2.4 Materials

Material selection was dictated by the functional parameters involved with H_2. Materials are limited to those that will maintain adequate mechanical properties in the extreme cold and will not be susceptible to H_2 embrittlement. Thermal insulation properties are important for maintaining temperatures and maximizing the system lifetime. Low outgassing materials are necessary to comply with the contamination constraints imposed by the instrument and the requirement to maintain a hard vacuum around the SH_2 tanks. The WIRE materials selection combines these unique requirements with standard requirements for materials compatibility, flammability, and potential for static discharge.

The H_2 tanks and the telescope are made of 6061-T6 aluminum alloy. In general, aluminum alloys are not subject to attack by H_2 or loss of ductility at cryogenic temperatures. Specifically, 6061-T6 aluminum has been shown to be compatible with, and is highly recommended for use with, cryogenic H_2. The external plumbing system is primarily 6061-T6 aluminum, 304 stainless steel, which meet ASTM A268 SS or MIL-S-25042.

The focal plane detectors are connected to the primary H_2 tank by means of beryllium/flexible copper braid thermal links. Beryllium is used for the shrink-fit thermal connection to the flight sensor. The braid is a non-structural component and not exposed to H_2.

Other construction materials include fiberglass, epoxy, and fluorinated ethylene propylene (FEP Teflon) tubing. These materials are compatible with H_2 and have been used extensively by LMMS in other cryogenic cooler applications.

Fiberglass is primarily used for tank support and thermal isolation. Structural support for the WIRE tubes is Owens Corning S-glass S-901/1543 HTS mixed with Fiberite 954-3 epoxy resin. The tube supports are not in contact with H_2, but are subjected to cryogenic temperatures. G-10 Fiberglass tubes are also used for thermal isolation in the primary vent bonded around the tube to form a hermetic seal.

Epibond 1210A/9615, is the epoxy that was used to bond the cryogen's components. The epoxy has been used successfully on other cryogenic cooler programs and has undergone considerable developmental testing to evaluate and understand the effects of the bond line thickness, surface preparation, thermal cycling, temperature limitations, and aging. During the SPIRIT III rebuild, additional quality assurance steps were added to the Teflon bonding process that were then added to the WIRE fabrication.

At cryogenic temperatures to - 452° F, the epoxy shows strengths greater than 2000 psi in lab shear tests. For calculating design margins, a value of 1000 psi was used. The epoxy used to bond the plumbing lines and the H_2 tank shells is sampled each time a new mixture is made. The sample is checked for hardness, indication of the epoxy mix ratio, and quality of the materials. The bond thickness are controlled during manufacturing to maintain the spacing between the components being bonded together. The epoxy bonds were thermal cycled from 300 K to less than or equal to 100 K for a minimum of three times and then leak checked. (Cooldowns do not count as thermal cycling.

Short sections of corrugated FEP Teflon are used in the cooling loop and all tank plumbing for thermal isolation except the primary orbit vent. FEP has been used in this capacity on many cryostats built by LMMS.

Approximately 80 layers of silk net and double-aluminum Mylar multi-layer insulation (MLI) are to be placed in the space between the cryostat housing shell and secondary tank to minimize the parasitic heat leak into the sensor.

3.1.2.5 Structural Analysis and Testing

Table 3-1 gives the design limit loads together with a 1.25 factor of uncertainty.

Table 3-1 DESIGN LIMIT LOADS X 1.25 UNCERTAINTY FACTOR

Event	T_x	T_y	T_z	R_x	R_y	R_z
Drop Transient	1.25	6.1	1.25	60.38	8.5	1.75
Stage Burn	1.5	1.5	12.9			
Aerodynamic Pull-up	1.5	4.17	5.9			
Taxi, Abort, Captive Carry	0.88	4.5	1.25			

T = translational acceleration, units in g for axis in subscript
R = rotational acceleration, units in rad/sec^2 applied at *spacecraft* center mass

For the support tube analysis, the design accelerations resolved into the following axial and lateral loads:

Drop transient: 1.25 G (axial), 9.1G (lateral)
Staging: 12.9 G (axial), 2.1 G (lateral)

The WIRE systems requirements also specify factors of safety to be applied against stressed and buckling load factors. Table 3-2 lists these specified factors of safety.

Table 3-2 REQUIRED FACTORS OF SAFETY

	F.S. (yield)	F.S. (ultimate)
Drop Transient	1.25	1.4
Stage Burn	1.25	1.4
Aerodynamic Pull-up	1.25	1.4
Taxi, Abort, Captive Carry	1.5	1.75

Tubes will be sized for buckling (ultimate failure).

1 σ at ultimate
Pretested to ultimate load condition
Probability of tube failure at limit load = 0

In addition to these loads, design factors of safety applied to the pressure systems include 2.0 for external (vacuum) pressure and 4.0 for internal pressure. In addition to the 2.0 factor of safety used for external pressure conditions, a 3 σ knockdown factor has been used to account statistically for instability failures due to buckling.

All structural components will be analyzed by LMMS. The analysis will rely on analytical tools verified by tests on other LMMS cooler programs. These tools include:

- PANDA (Panel Design Analysis) - for optimization of stiffened cylinders and plates,

- BOSOR (Buckling of Shells of Revolution) - for static and dynamic analysis of rotationally symmetrical shells, and
- STAGS (Structural Analysis of Shells) - for the static and dynamic analysis of general structures.

During the manufacturing process, only the metallic components of the tanks, their outer walls, plumbing, and structural components were dye-penetrate inspected or pressurized to verify that flaws capable of propagating into a failure were not present. Because all the pressure systems demonstrate leak-before-burst design of the tanks, prior to any operation involving a WIRE pressure system, the system integrity is verified by vacuum leak checking. The vacuum leak check ascertains whether a crack has developed within the system. In addition, all non-structural hardware is visually inspected with a 3x-5x magnifier for cracks.

Because of the low pressures involved, the aluminum structure can be shown to have greater than ten lifetimes, as required by NTST 1700.B, section 208 4a, "Pressure Vessels". The exception to this is the H_2 tanks where the ends are sealed to the tank with epoxy. Fatigue data for the epoxy are not available; therefore, the epoxy bond lines cannot be certified by analysis to have ten design lifetimes. Lack of this analysis does not compromise the safety of the system, however, because the loads on the H_2 tanks are small (see LMMS Document 95-0027, *WIRE Cryostat Structural Report*, December 21, 1995, which provides buckling and stress analysis data on the primary and secondary tanks). The pressure limits for the system are controlled by the plumbing with dual burst disks at the primary and secondary tanks but not the tank structures. Even if either tank were to burst, the rupture would be hydrostatic and contained within the vacuum shell, which is also equipped with a burst disk.

Because of the nature of the cryostat, the completed assembly cannot easily be repaired if a component fails proof pressure during the full-up assembly testing. Testing will therefore be performed throughout the subassembly process at the component or assembly level. Each vacuum joint (epoxy bond, welds) in the tank and tank plumbing will undergo thermal cycling and vacuum leak checks both prior to and after proof-pressure tests. Leak testing was performed using a He mass spectrometer leak detector. The maximum allowable leak rate is 1×10^{-7} std cc He/sec. Proof pressure tests at the component level were performed at 1.5 times the Maximum Differential Pressure (MDP). As a completed assembly, the WIRE tanks and vacuum system have undergone a proof pressure test at 1.5 times the burst disk setting as required by MIL-STD-1522A.

In the case of the fiberglass support tubes, the analysis was followed by a static loads test. During testing, the fiberglass tubes are loaded to their ultimate load, verifying they will not buckle.

As an assembly, the ejectable aperture door underwent vibration testing with the cryostat. Door ejection tests were also performed to test the release mechanisms. During this test, the door was mounted on a flat plate, and identical separation nuts (each cold, gas actuated) were used during the test firings. In the first test series, three nuts were fired simultaneously; in the second series, one separation nut was fired, followed shortly by two fired simultaneously.

A pressure test, He leak test, and a deployment test were performed on the primary orbit vent door mechanism subassembly and seal to assure their proper operation.

3.1.2.5.1 Vibration Testing

For the vibration test, the vacuum shell and tanks will be evacuated. The assembled cryostat will then be subcooled to <100K using LN_2. Vibration testing at these low temperatures is important because the strength of the fiberglass increases with lower temperatures. In addition, each tube has relatively high temperature stresses at the tube ends that should be properly simulated to reflect the launch temperature. The aperture shade, as well as an engineering model telescope and flight support struts, will be attached to the cryostat during the vibration test. Vibration testing should verify that the assembled cryostat will survive launch loads, plus it will provide a model signature on three axes that will be models.

Several components will be vibration tested individually, including the normally closed pyrotechnically actuated valves, the burst disk, the check valve in the safety vent line, and the primary vent valve.

3.1.2.5.2 H_2 Test Fills

Two test H_2 fills have been performed on the WIRE cryostat. Their purpose was to verify the H_2 fill and servicing procedures, ground-hold times, the thermal model, and the primary tank temperature. The ground-hold results of the first fill, performed in August 1997, were affected by the test configuration of the WIRE cryostat and did not yield accurate results. The first test did, however, demonstrate that all temperature requirements would be met. During the second test fill, conducted in early 1998, the operation was performed with the cryostat in its flight configuration and verified a ground-hold time of approximately ten hours to triple point. Both fills were performed successfully using the developed procedures, and all red lines have been incorporated into the payload processing facility (PPF) hydrogen fill procedure (WIRE-02, "PPF Cryogenic Operations").

3.1.3 Pressure Subsystems

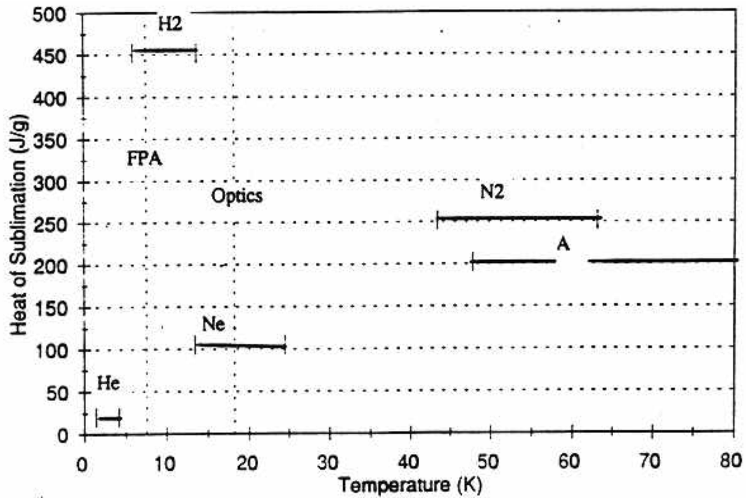
The Wire Cryostat has two pressurized subsystems that consist of a primary tank and a secondary tank. Both tanks are filled with H_2. See Table 3-3 for information on the pressurized subsystems. The H_2 is used to maintain the temperatures of the optics at less than 18 K, and the focal plane detectors at less than 7.5 K. The cryogen is expected to last approximately 4 months.

Figure 3-8 shows the heat of sublimation (for solids) and heat of vaporization (for He) for candidate cryogens to cool the WIRE instrument. Superimposed are the sensor temperature limits for the four temperature zones. Figure 3-8 provides evidence suggesting that in order to

Table 3-3 CRYOSTAT PRESSURIZED SUBSYSTEMS

	Primary Tank	**Secondary Tank**
Volume Foam fraction Fill fraction	7.75 liters 0.08 liters 0.75 liters	56.5 liters 0.015 liters 0.825 liters
Operating temperature Heat load On-orbit lifetime Ground time to start of melt Time until must pump vacuum	6.8 K 15 mW 5.1 months (at 12 K secondary tank temperature)	12 K 184 mW 4.1 months at 230 K shell temperature 10 hours 49 hours
Dry mass Hydrogen mass	3.0 kg 0.47 kg	11.0 kg 4.02 kg
Orbit vent line Emergency vent line Burst disks	0.5"o.d. x 6" long + 2.0" i.d. 6" long (located on aft dome) 0.375" o.d. x TBD long (located on aft dome) one each on orbit vent line and emergency vent line	0.625" o.d. x 18" long, located forward 0.50" o.d.x 18" long, located forward one each on orbit vent line & emergency vent line
Working pressure Design external pressure Design internal pressure Test pressure	1.3 Pa 1 atm with safety factor of 2 30 psi with safety factor of 4 37.5 psi	1.8kPA 1 atm with safety factor of 2 44 psi with safety factor of 4 55 psi

Figure 3-8 Heat of Sublimation



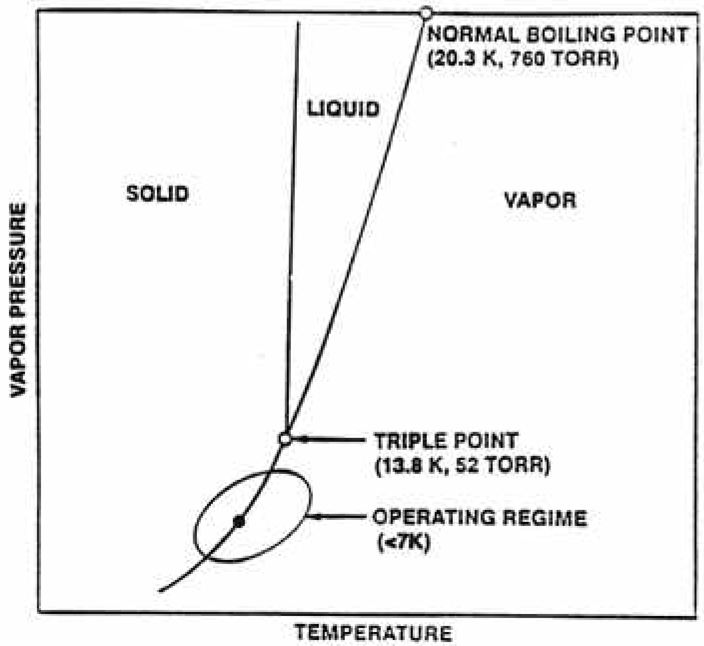
meet the focal plane cooling requirements, only supercritical or superfluid He and SH_2 are viable candidates.

Of the two candidate cryogens, H_2 is at least five to six times more efficient (by weight) than He, making H_2 a better choice for providing reliable, long-term, instrument cooling. In fact, for the instrument heat load, operating duty cycle, required system lifetime, and weight and volume constraints, SH_2 is really the only candidate cooling system for WIRE.

As shown by Figure 3-9, the triple point for H_2 is 13.8 K at 52 torr. During space operation, the H_2 is pumped by space to maintain it as a solid at an operating temperature less than 7.0 K.

Whether para or ortho H_2, the safety properties of H_2 are the same. H_2 liquid and vapor are colorless, the vapor has no odor and is classified as a simple asphyxiant with no threshold limit value (TLV) per the ACGIH 1996. H_2 is flammable from 4%-95% in air, and listed as a Class 1, Division 2, Group B location per the National Electric Code (NEC). The Environmental Protection Agency lists H_2 as a fire and sudden release-of-pressure hazard. There are acute cryogenic hazards associated with LH_2 such as severe cold burns, frostbite, and oxygen (O_2) displacement due to the expansion of the liquid to gas in the ambient area.

Figure 3-9 Hydrogen Triple Point



3.1.3.1 Hydrogen Tanks

The cryostat tanks are filled first with gaseous LH_2 that is then solidified using LHe. The primary tank has a total "usable" volume (tank volume - 8 percent aluminum foam volume) of 6.91 liters and contains 430 grams of solid H_2 (volume at STP = 4,783 liters; mass = 430 grams). At a temperature of 13.8 K, this amount of SH_2 leaves 28 percent ullage. The primary tank is filled with eight percent dense aluminum foam heat exchanger. The foam prevents movement of the solid H_2 within the tank during spacecraft slew operations and acts to greatly reduce temperature differentials in the tanks as H_2 sublimes away from the tank wall.

The secondary tank has a total "usable" volume (tank volume - 1.8 percent aluminum foam volume) of 55.51 liters and contains 3,980 grams of SH_2 (volume at STP = 44,271 liters; mass = 3,980 grams). At a temperature of 13.8 K, this amount of SH_2 leaves 17.1 percent ullage. The secondary tank is filled with a 1.8 percent dense aluminum foam heat exchanger.

Fill levels are important in maintaining a correct amount of ullage in the tank. Mass flow meters will be used to measure the quantity of H_2 passing into each tank as it is filled. The fill technique using the flow meters will be verified during a test H_2 fill prior to operations at VAFB. Manufacturer specifications indicate the flow meters are accurate to one percent, though to be conservative, we are assuming accuracy to only three percent. Tests run on the meters prior to each fill indicate the three percent number is satisfactory. Below the H_2 fill, the flow meters will be checked by flowing H_2 gas from a bottle and comparing the meters to the amount of weight the bottle loses. During the operations at VAFB, redundant mass flow meters will be in place to compare readings and to maintain a constant back-up in case one instrument fails.

During the times when the LHe is not being flowed around the H_2 tanks, the H_2 will slowly warm from its LHe-cooled state (~5 K) up to the triple point. The H_2 begins to melt until the entire mass is liquefied. This period when the H_2 is not being kept cold by flowing LHe is call the "ground hold". The ground hold times for the primary and secondary tanks are listed in Figure 3-10.

Hold times from the start of disconnect to triple point and then through triple point were 10 hours and 49 hours. As operational constraints it takes three hours to disconnect the LHe cooling equipment from the cryostat, the time from start of disconnect to when the valves have to be open on orbit is less than 10 hours. During ground operations it is allowable to have the H_2 go into triple point and melt for up to 24 hours. This extends the working time from the beginning of the disconnect to 34 hours at which time the LHe colling has to be started. All of the measured times should allow sufficient time to prepare for launch and launch the solid H_2 cryostat.

During ground hold, the pressure in the tanks will vary between 1.3×10^{-5} Torr, after sub-cooling to 5K, to 52 Torr (~1 psia), when triple point is reached. The MDP for both the primary tank and secondary tank is 40 psi. During normal operations, a maximum pressure differential will be reached only if a procedural error occurs whereby the H_2 is allowed to warm past the triple point and continue to warm to the point (19.4 ± 2 K) where the available ullage in the tank is zero. Further melting will cause a hydrostatic pressure to develop, rupturing one or both tank external burst disks. The tanks are protected by redundant burst disk assemblies that are designed to release at 15 psi and will experience a maximum of 30 psi differential, respectively, at the time the burst disk ruptures. Table 3-4 lists the pressure parameters of the H_2 tanks.

Figure 3-10 Ground-Hold Times for the Primary and Secondary Tanks

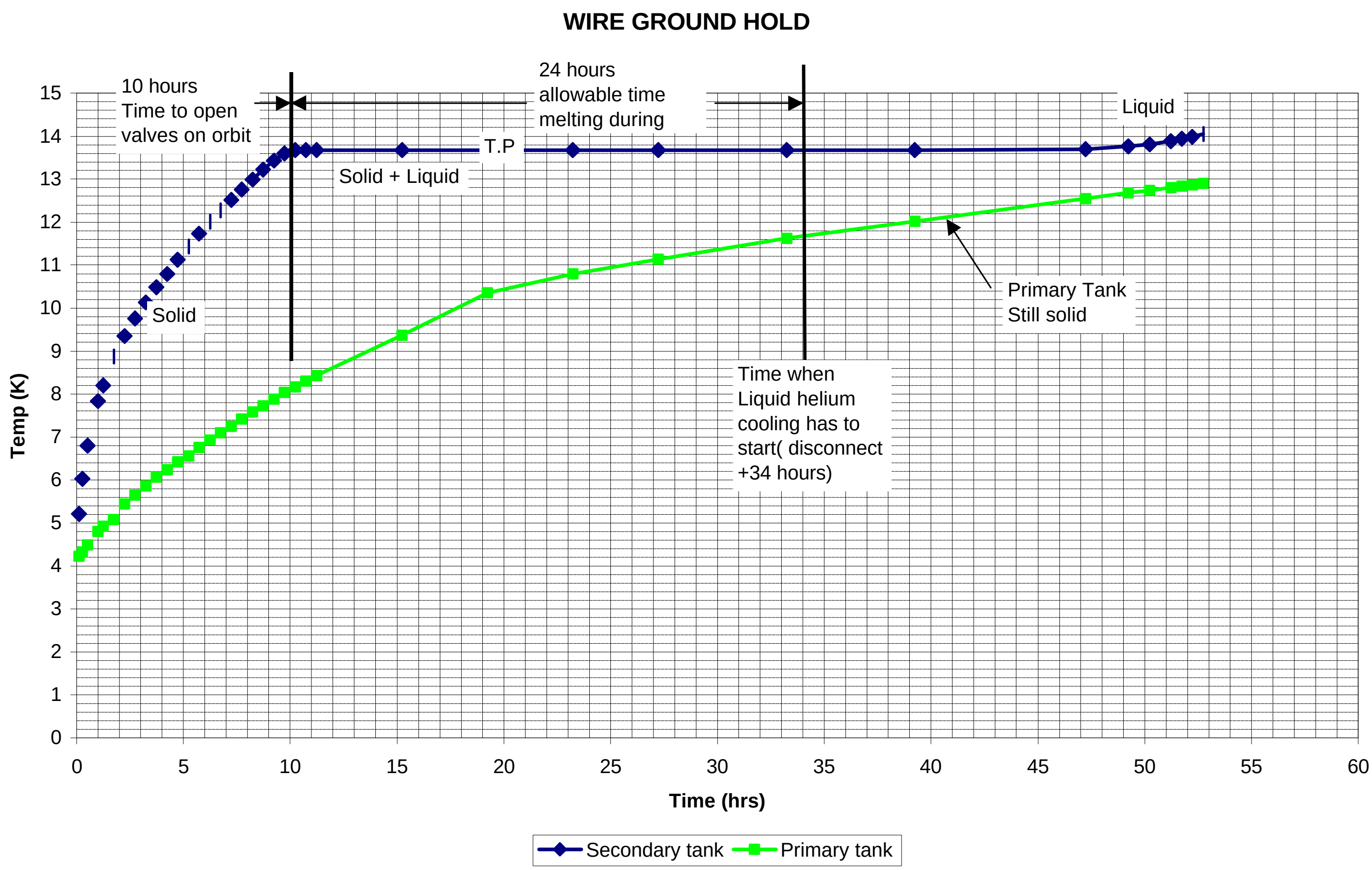


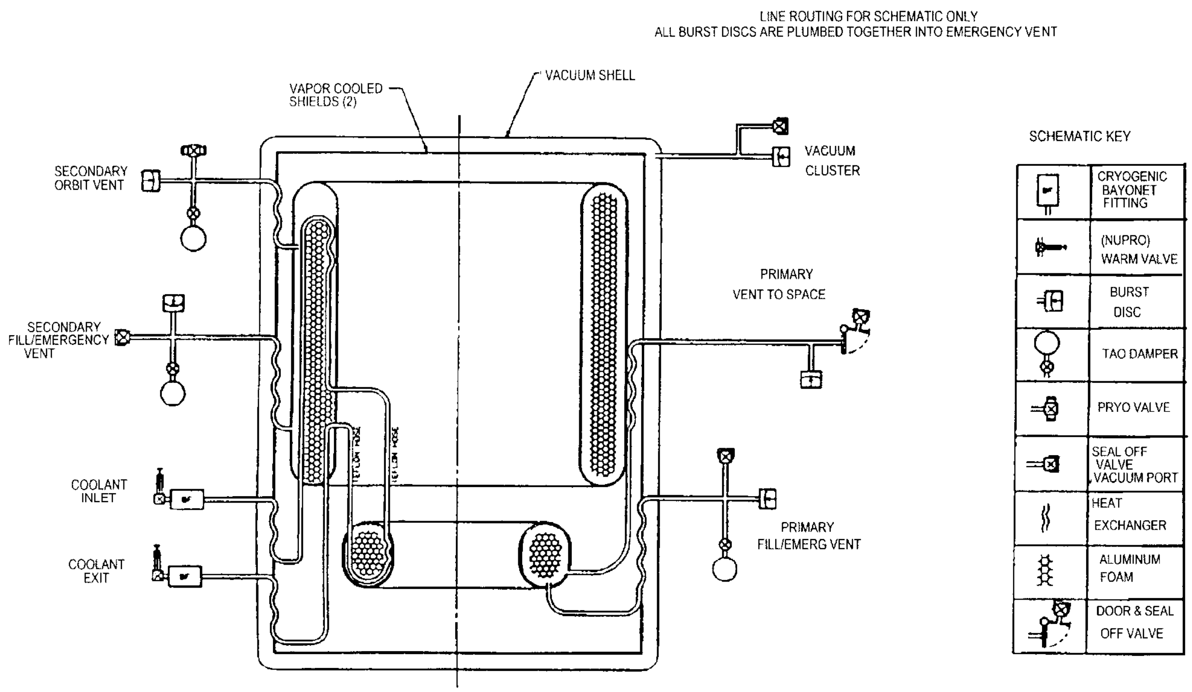
Table 3-4   HYDROGEN TANK DATA

Item			Comments
Primary Tank	MDP 40 psi	Design pressure 160 psi (4 x F.O.S.) Proof pressure 60 psi External pressure 30 psi	Based on transient pressure during failure of the vacuum shell insulation space
Secondary Tank	MDP 40 psi	Design pressure 160 psi (4 x F.O.S.) Proof pressure 60 psi External pressure 30 psi	Based on transient pressure during failure of the vacuum shell insulation space
Burst Disk Levels	15 ± psi		All 4 tank burst disks

3.1.3.2. Plumbing and Vents

Figure 3-11shows the plumbing schematic for the WIRE cryogenic system. Two bayonet fittings are provided to circulate LHe coolant through the cooling coils within the tanks. These lines are used for the initial cooldown of the H_2 tanks and telescope from ambient; they are then used to fill and solidify the H_2 and re-cool it in preparation for ground hold.

Figure 3-11 Schematic of WIRE Plumbing

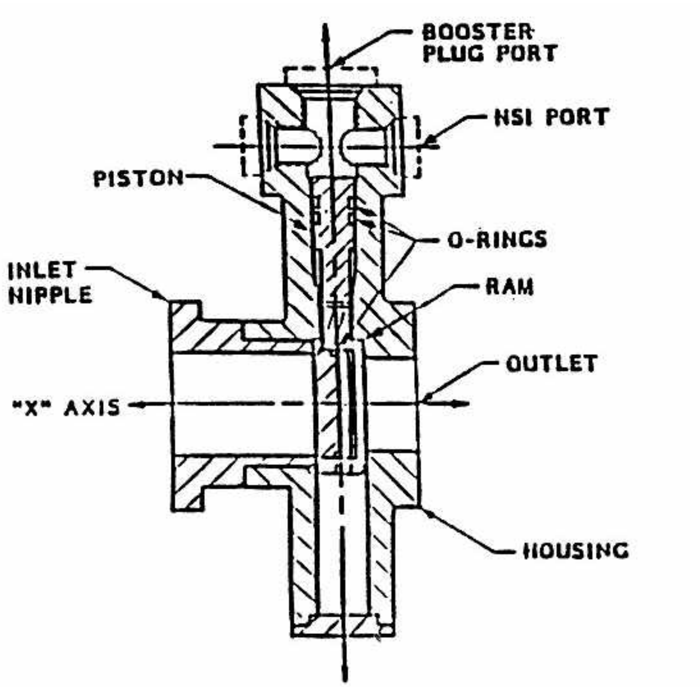


In space, there is a requirement to "pump" on the H_2, maintaining a vacuum on the ullage space in the tank, in order to maintain the cryogen below the triple point temperature. By opening the primary and secondary H_2 tanks to the ambient environment of space vacuum, temperature control of the H_2 is easily maintained. One, half-inch, normally closed, pyrotechnically actuated vent valve is provided on the secondary tank's external plumbing cluster of the orbit vent line; and a single two-inch spring-retracted valve is provided on the primary tank's external plumbing cluster of the orbit vent line to perform this function.

3.1.3.2.1 Secondary tank pyro-actuated valve

The pyrotechnically actuated valve on the secondary tank's orbit vent cluster, with highlights of the mechanism, is shown in Figure 3-12. The valve is designed so that hot gases from the ordnance initiation are not released into the fluid passageways. The half-inch valve uses dual O-rings to seal the firing chamber; like all O-rings in the system, the seals will not be exposed to temperatures less than -15°C on the ground.

Figure 3-12 Pryo Valve Assembly for Secondary Tank Venting



The circuitry and initiators for activating the valve mechanism are discussed in Section 3.1.5. In general, two independent firing circuits will be provided, either of which may be commanded to fire either of the two redundant initiators (two initiators for the single, half-inch valve).

The pyrotechnic initiation generates the actuation pressure necessary to drive an internal tube cutter into a parent metal plug, machined into the gas passageway. The tube cutter strikes the plug with sufficient force to shear it off, opening up the gas passageway and into a wedge shaped area where it is permanently captured. The valve is actuated as shown in Figure 3-13

3.1.3.2.2 Primary tank wax-actuated valve

Figure 3-14 shows the activated valve for the primary tank vent. The valve is operated by a Starsys Research (EP-5025) paraffin actuator. Activation is caused by the direct expansion of the paraffin, which in turn is caused by the application of heat. This results in the retraction of the shaft, a form of hydraulic mechanical work. The heating is done by two 78 Ω independent heating elements in parallel. Reliability of the actuator is greater than 0.9999 under worst-case conditions.

Figure 3-13 Pryo Mechanisms for Secondary Tank Venting

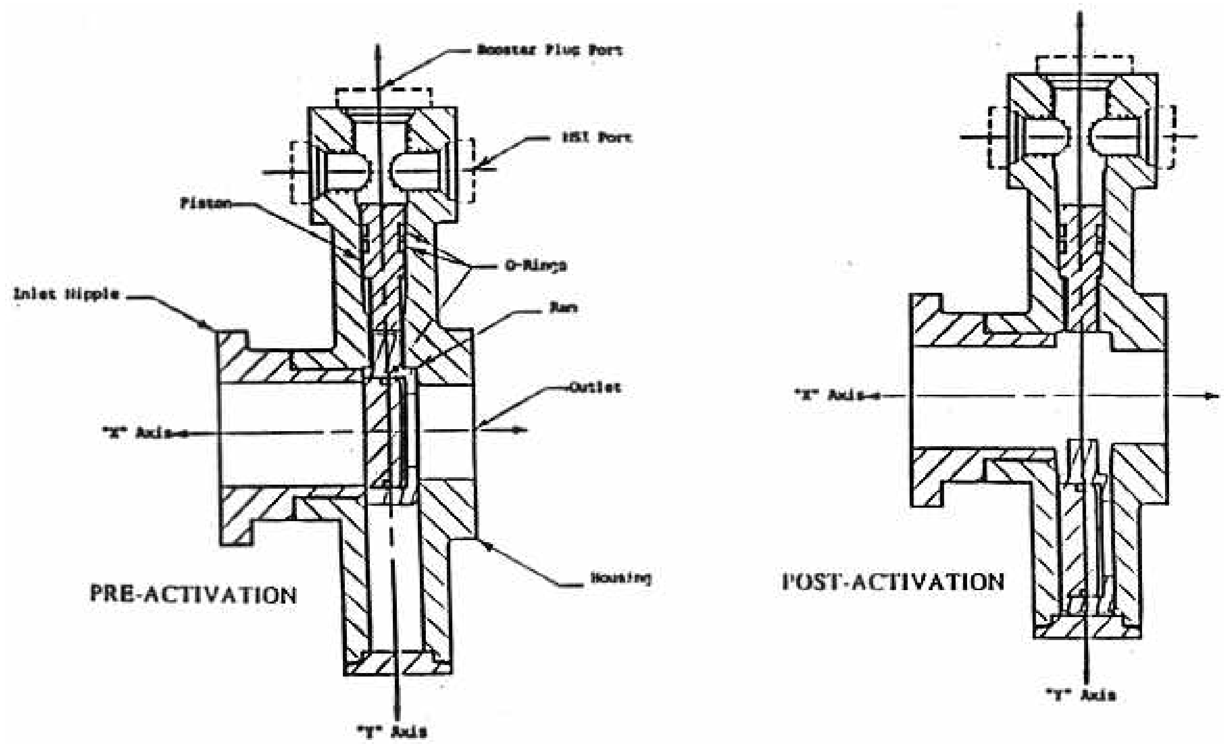
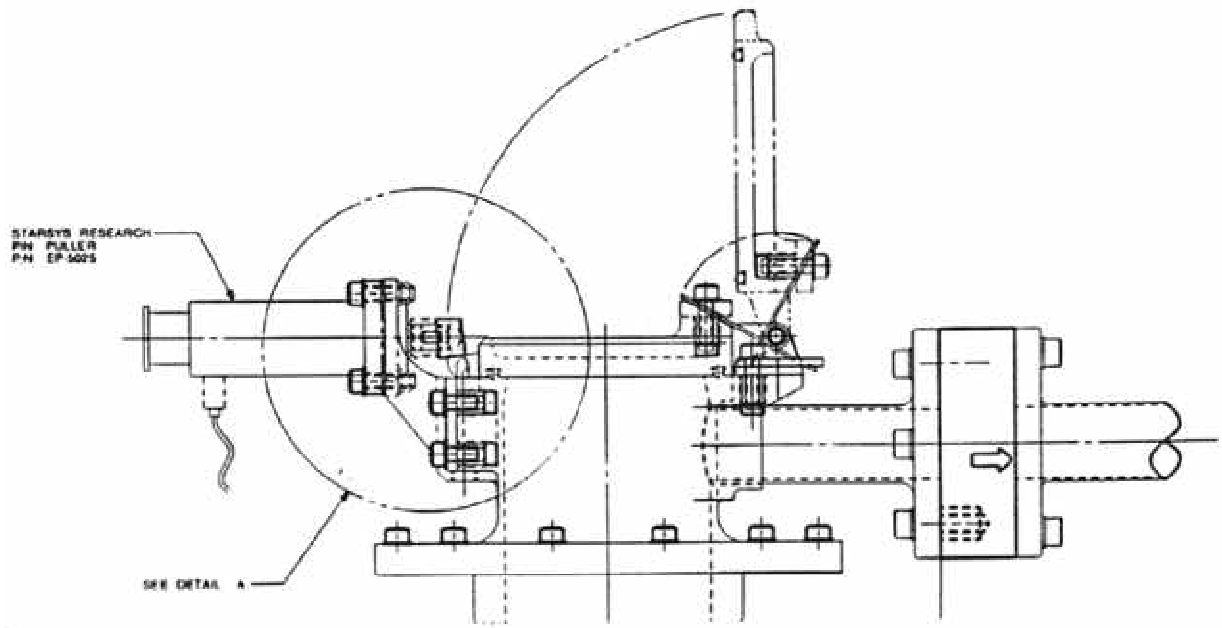


Figure 3-14 Primary Tank Wax-Actuated Valve

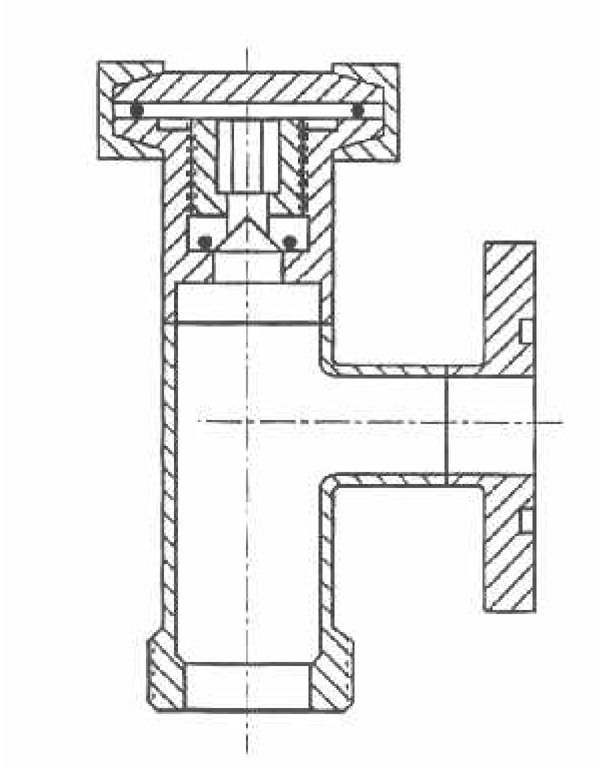


If either of the H_2 orbit vent valves inadvertently fire on the ground, an ice plug will form in the orbit vent line well upstream of the tank. After some period of time the tanks will most likely warm up. The WIRE instrument cannot function at operating temperatures with such a plug and each tank will rupture its single remaining burst disk. To recover the instrument, the H_2 would have to be warmed and pumped out through the ground fill and vent line, the vent valve and burst disks replaced, and the tank recooled and refilled.

If either the pyro- or wax-actuated valves failed to open on orbit, the SH_2 in the tanks would warm to triple point, turn to a liquid, hydraulically rupture the burst disks, and vent through the safety vent line. The H_2 would be unable to be recooled below the triple point because of the check valve in the on-orbit safety vent, and the sensor would remain cold until all the H_2 vented through the burst disk assembly.

There are two lines going to each tank, one fill and one vent. Each tank's orbit vent line is capped by the pryo- or wax-actuated valve as discussed above. The fill line for each tank is capped by closing the vacuum-type operator. See Figure 3-15. In addition to the valve closure, a secondary vacuum seal is placed over the inlet to the valve operator. Both the cap and valve on the assembly would have to open to have communication with a tank. When in use, the valve and cap are sequentially opened according to a written procedure.

Figure 3-15 Vacuum Operator for Primary and Secondary Tank Fill

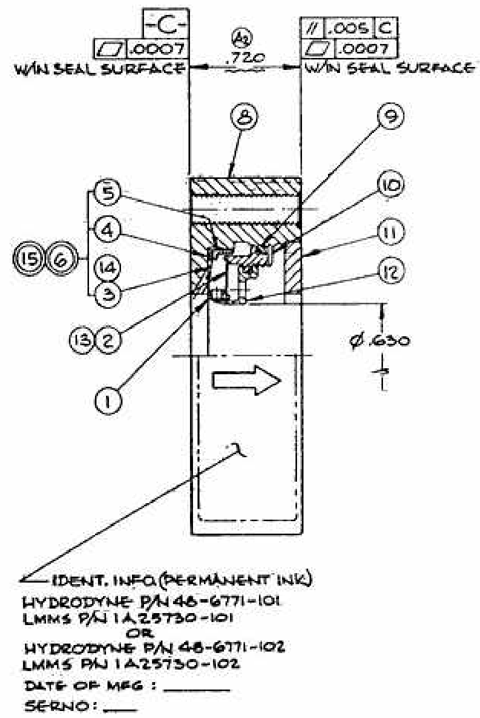


Burst disk assemblies are provided on the H_2 fill and vent lines for safety venting tank relief. The burst disk valve assembly is shown in Figure 3-16. The burst pressure for the tank burst disks is set at 15 ± 1 psi, allowing for safe venting of H_2 within the internal tank design pressure limits. A single check valve in the safety vent prevents the back flow of air into the H_2 tank after a burst disk rupture.

Figure 3-15 Burst Disk Valve Assembly

FIND NO	QTY REQD	QTY REQD	FSCM NO	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL	MATERIAL SPECIFICATION
15	1	—		46-6771-14	DIAPHRAGM ASSY	———	———
14	1	—		-13	DIAPHRAGM	316L S.S.	MIL-S-5059
13	1	—		-12	BELLEVILLE	17-4 PH S.S.	AMS-5643
12	1	1		-11	PUNCH	17-4 PH S.S.	AMS-5643
11	1	1		-10	PLATE	304L S.S.	VAC. ARC OR ESR QQ-S-763
10	1	1		-09	RETAINER	300 SERIES S.S.	QQ-S-763
9	4	4		-08	FRICTION PLUG	KEL-F	AMS-3650
8	1	1		46-6771-07	BODY	304L S.S.	VAC. ARC OR ESR QQ-S-763
7	—	—		———	———	———	———
6	—	1		46-6771-06	DIAPHRAGM ASSY	———	———
5	1	1		-05	GUIDE	304L S.S.	QQ-S-763
4	1	1		-04	DOUBLER	304L S.S.	QQ-S-763
3	—	1		-03	DIAPHRAGM	316L S.S.	MIL-S-5059
2	—	1		-02	BELLEVILLE	17-4 PH S.S.	AMS-5643
1	1	1		46-6771-01	SLEEVE	300 SERIES S.S.	QQ-S-763
—	-102	-101		46-6771	ASSEMBLY	———	———

PARTS LIST



The safety vent line provides connection to a remote facility vent in the event that the tank burst disks ruptures. A 1.5-inch diameter manifold connects to the exit of the burst disk assemblies. This manifold connects to a facility safety vent line to vent the gas safely outside the facility. Ice blockage is not viewed as a credible occurrence in the vent line because of the warm ambient temperature and the large (greater than or equal to 1.5-inches) diameter of the manifold and vent lines.

During the L-1011 phase of the launch, Orbital will be responsible for the vent system from the L-1011 to the single-point connection of the safety vent. Orbital plans a continuous purge of He in the airborne vent system that will begin during mating of the Pegasus launcher with the L-1011 and continue through captive carry. The cryostat vent design requires less than 5 psid pressure drop downstream at the instrument to the Orbital interface.

Operations associated with the cryostat at Vandenberg Air Force Base (VAFB) will include:

- A connection vent test of all GSE/instrument mechanical interfaces.
- Evacuation of the cryostat insulation space.
- Evacuation of the H_2 tanks.
- Cooling of the cryostat through cooling coils.
- Filling the H_2 tanks.
- Solidification and sub-cooling the H_2 using LHe (repeated).
- Periodic re-cooling the SH_2 using LHe.

Additionally, contingent operations include:

- H_2 pumping.
- Emptying of tanks (H_2).
- Rapid emptying of tanks (H_2).

The ground operations associated with H_2 fill and He cooling processes are given in detail in section 6.0.

3.1.4 Electrical and Electronic Subsystems

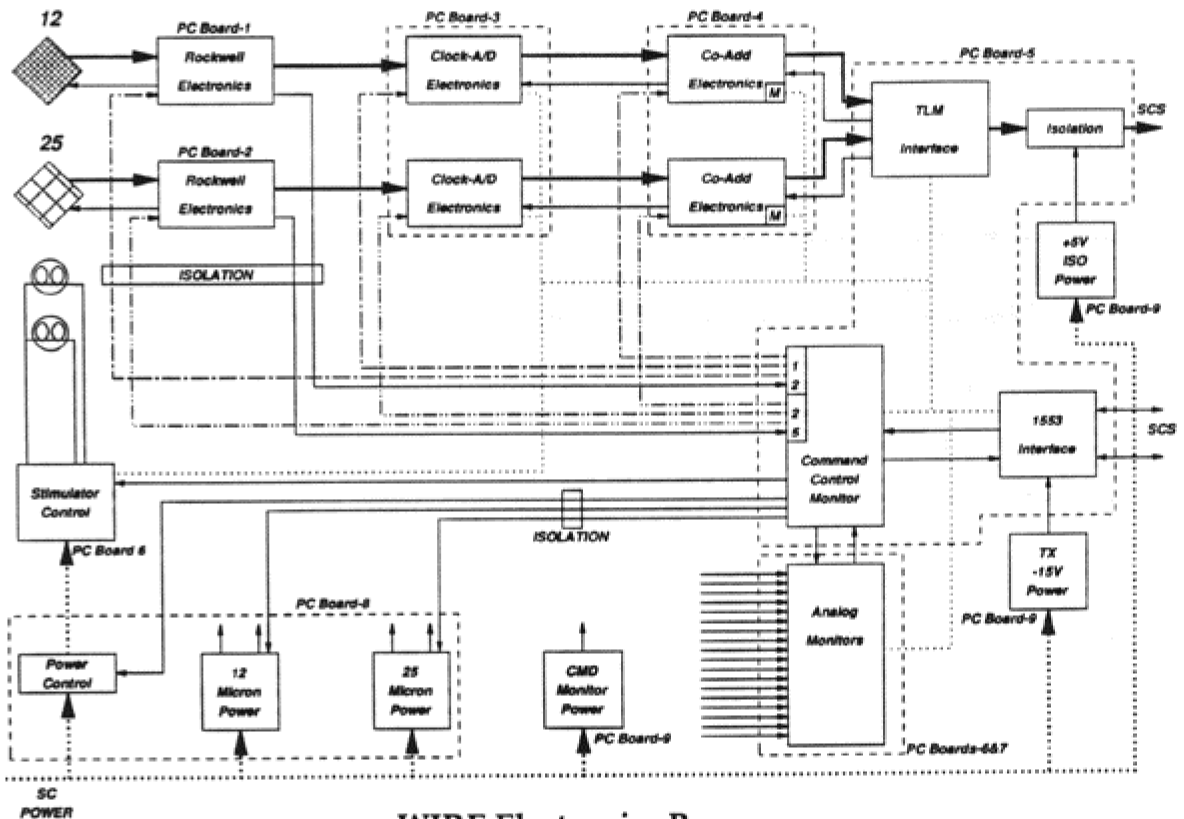
3.1.4.1 WIRE Electronics Box

The WIRE electronics (WIE) box contains all the required electronic assemblies to interface to the WIRE cryostat and the spacecraft's computer system. The box is mounted at +X on the equipment panel. It has seven interface connections: two 1553 connectors, one power connector, one telmetry connector, and four cryostat connectors. There are no identified hazards for the WIE box. All wiring is sized per the standard derating guidelines in the GSFC Preferred Parts List (PPL-21), appendix B, table 16, "Wire and Cabling Derating Criteria", March 1995. The spacecraft power system will shut down the WIE box if the box draws excessive current. External surfaces will be at spacecraft ground potential. There are no batteries used in this system. While some flammable materials may be used in the electronics, there are no available ignition sources. Wiring is TEFZEL, derated as noted. Figure 3-17 is a top-level data flow diagram of the WIE box.

3.1.4.2 WIRE Pryo Electronics Box

The WIRE pryo electronics (WPE) box contains all the electronics to interface the WIRE aperture cover pyros, secondary vent pyros, and primary vent wax actuator to the spacecraft's computer system. Mounting is the + x and + y lower equipment panel. The WPE box has a total of three interface connections: one power connector, one command input connector, and one output pryo connector, which uses 26-gauge twisted shielded pair wire. All shields are tied to the connector backshell which is EMI protected. Though safety critical, the WPE box has no identifiable hazards because all activation commands originate with the spacecraft command electronics. Figure 3-18 shows the pryo command electronics. In addition, the spacecraft

Figure 3-17 WIRE Electronics Box



WIRE Electronics Box

housekeeping electronics check the status of the WPE’s current, voltages, firing pulses, etc. All WPE box wiring is sized per the standard derating guidelines in the GSFC Preferred Parts List (PPL-21), appendix B, table 16, "Wire and Cabling Derating Criteria", March 1995. External surfaces shall be at spacecraft ground potential. There are no batteries used in this system. While some flammable materials may be used in the electronics, there are no available ignition sources. Wiring is TEFZEL, derated as noted. A top-level flow diagram of the WPE box is shown in Figure 3-19.

During the week of November 17, 1997, SDL discovered a large glitch on the output of the WPE box when the arm command was sent. This glitch was due to the field effect transistors (FET's) momentarily turning on as 28 V was applied by the closing of the normally-open arm relay. The glitch was about 1 A for about 1 µs.

To correct this problem, SDL added a bleed resistor (15 kΩ) across the arming relay. This resistor provides enough current to charge the stray capacitance of the FET's so that closure of the relay does not cause a sudden increase in voltage. The glitch is now about 10 mA for about 1 µs. The only drawback to this resistor is that it provides leakage current at 125°C and 80 V drain to source.

The bottom line being that less than 250 µA will flow through each one-amp-no-fire pyrotechnic device when the first inhibit is removed. A 1 µs pulse of 10 mA will flow through each device when the second inhibit is removed. These currents are small enough to preclude the possibility of an inadvertent firing of any device until the third and final inhibit is removed. No inhibits will be removed prior to spacecraft insertion in orbit.

Figure 3-18 Flow Diagram of Spacecraft Pyro Command Electronics

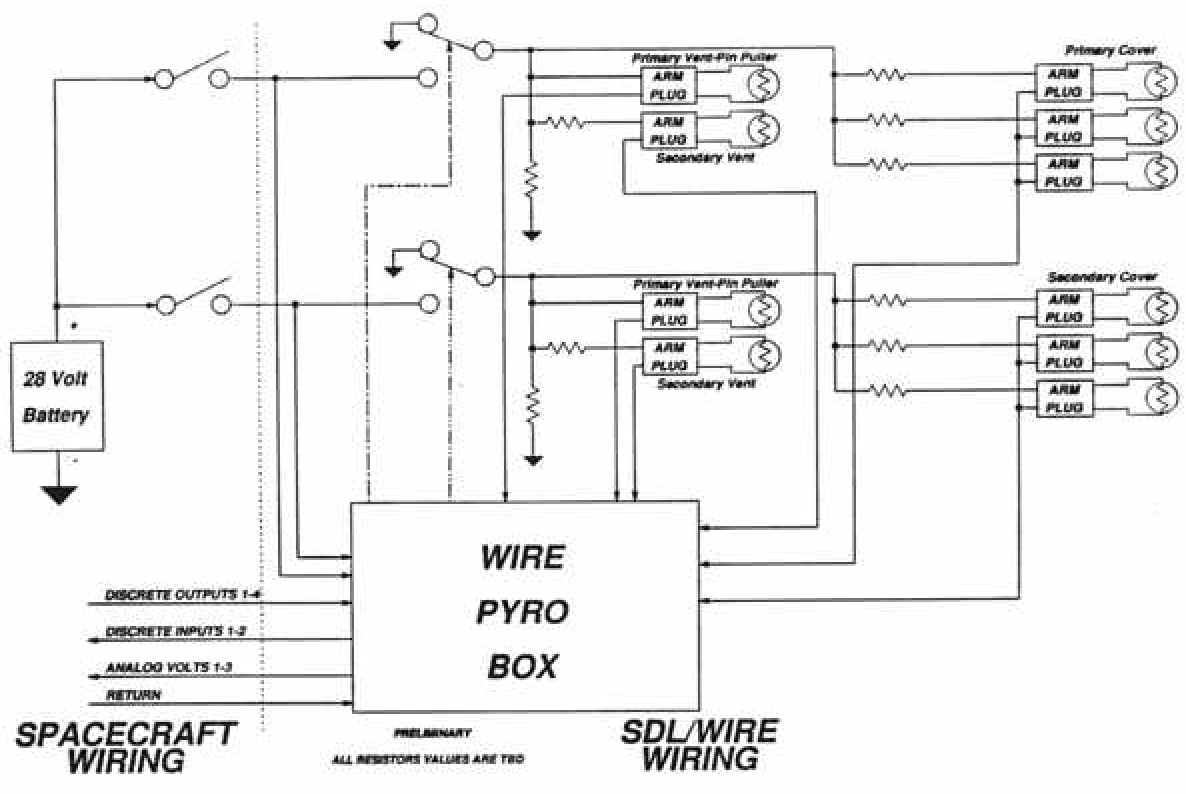
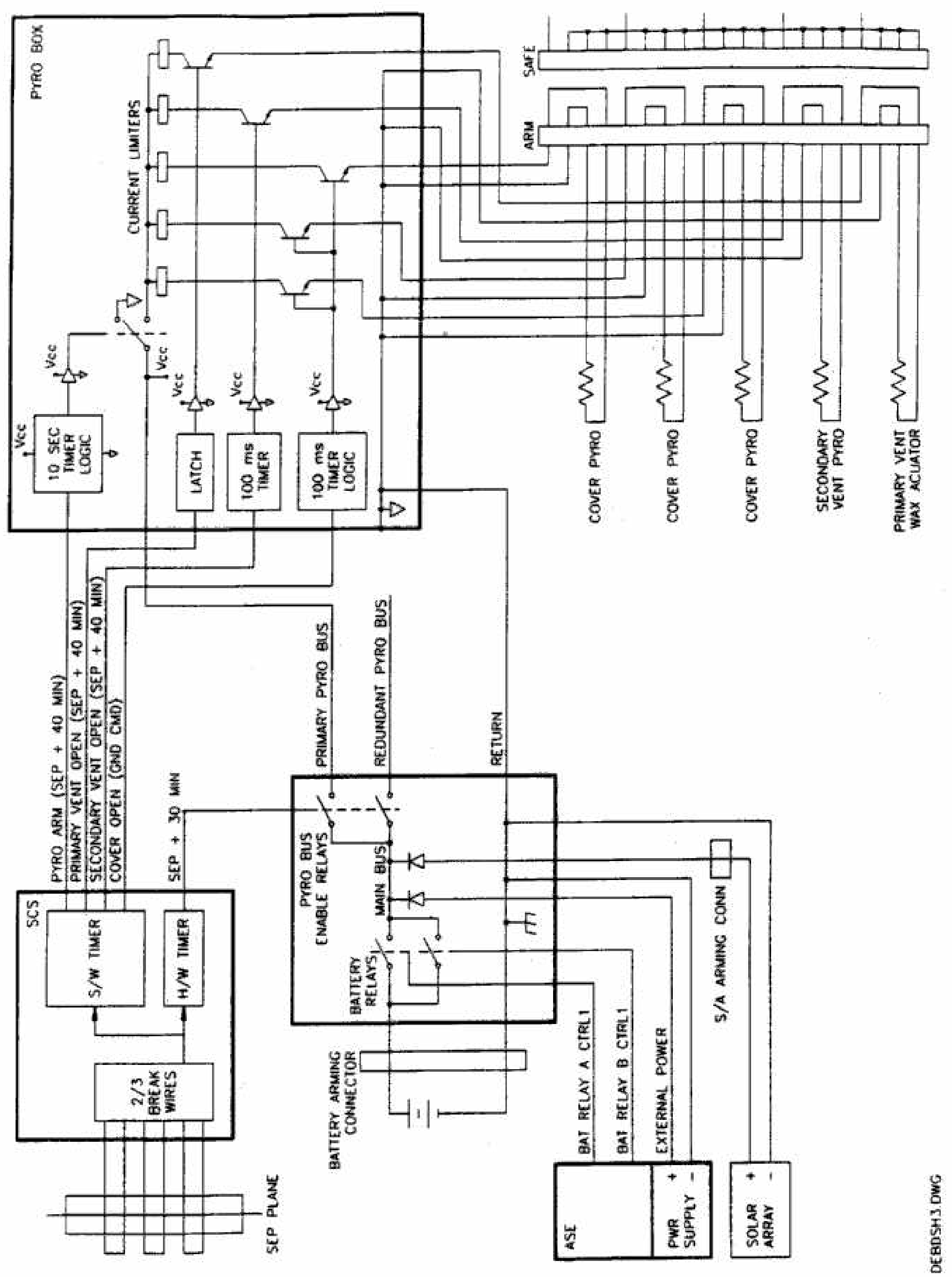


Figure 3-19 WIRE Pyro Wiring Diagram



3.1.5 Ordnance Subsystem

The ordnance subsystems consist of pyrotechnic devices that, when electrically initiated after orbital insertion, will open valves and eject covers.

The electroexplosive devices for all the pyrotechnic devices will be procured by LMMS. The electroexplosive devices are the NASA Standard Initiator (NSI). A NSI is shown in Figure 3-20. In addition to the NSI, a pressure cartridge is used in the pyro-actuated WIRE cryogen valve to boost the output from 750 to 1200 psi. The characteristic of the electroexplosive device are provided in Table 3-5.

Figure 3-20 NASA Standard Initiator

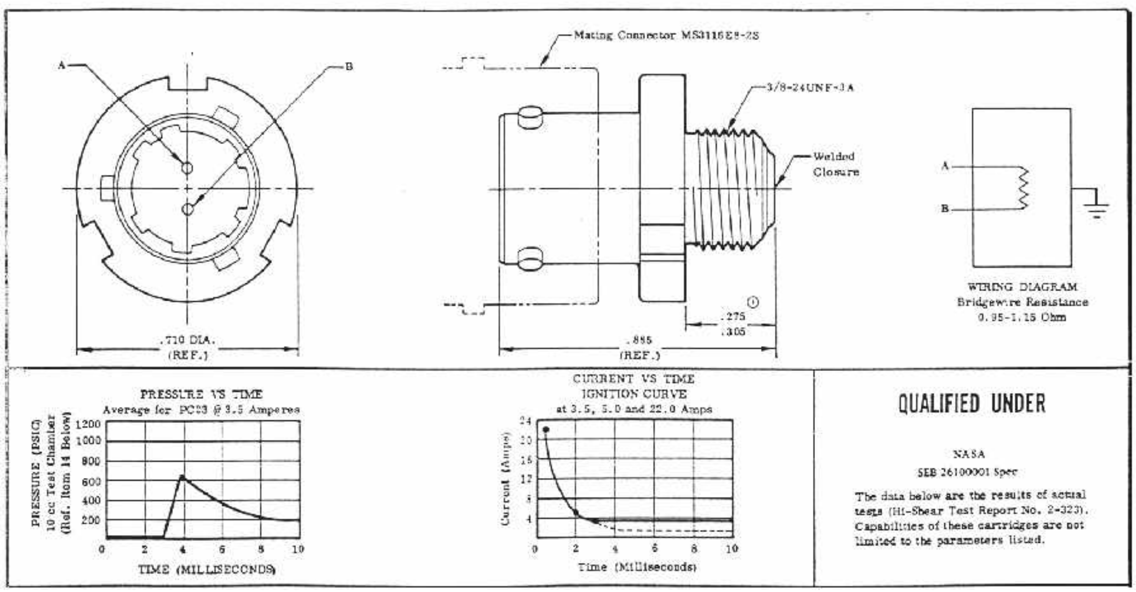


TABLE 3-5 Pyrotechnic Safety Data

	NASA Standard Initiator
Part Number	PC 23-29
Manufacturer	HI-SHEAR
Explosive composition	GMS zirconium/postassium perchlorate/Vitron B graphite formulation
Net explosive weight	114 mg
Range safety category	A
Bridge resistance	0.95-1.15 Ω
Maximum safe no fire	1 Amp-1 W
Minimum fire current	3.5 Amps
Temperature and humidity	0-105 EF, 95% RH
Special data	10-yr life

Qualification testing and lot acceptance for the NASA Standard Initiator will performed in accordance with NASA Specification SKB 2600053. The safety of the NASA Standard Initiator has been demonstrated over its 24-year history involving literally thousands of initiators in space systems.

All pyrotechnic devices are classified as Range Category A, considering their functional application on the WIRE instrument. When transported and stored in their output guards, however, they can be classified as Range Category B.

3.1.5.1 Inhibits

The first pyro inhibit is the pyro bus enable relay; one each for the primary and redundant pyro bus. These latching relays are open until 30 minutes after separation. When the SCS detects two of the three spacecraft to Pegasus breakwires open and an associated hardware timer starts. The pyro bus enable relays are energized 30 minutes later. The WPE remains unpowered until one of these relays is closed.

The second inhibit is the non-latching pyro arm relay; one each for the primary and redundant pyro bus. A software timer in the SCS commands arming 40 minutes after two of the three "WIRE to Pegasus" breakwires have been opened. The relays are closed only for ten seconds, governed by the ten second timer logic in the pyro box, to allow for firing of the secondary vent (or aperture cover). For the primary vent open, the software timer will start a loop which resends the arm command every five seconds so that the wax actuator will stay enabled for five minutes. If the software freezes for any reason, the 10 second timer will time-out and the arm relay will open.

The third inhibit is the fire transistor switch. The transistor will be latched with logic devices. (It is not a latching relay.) The secondary vent, a 100 millisecond pulse is sent to the current limited firing circuit during the ten seconds the pyro arm relays are closed so that the EED will fire. If the pulse is sent to the ordnance device other then during the ten second arming period nothing can occur. The primary vent, the transistor switch remains closed as long as the SCS holds the open command active since a sustained current for several minutes is needed to open the wax actuator. The aperture cover ejection is a ground commanded event but also relies on the ten second timer logic arm and the 100 millisecond fire command from the SCS.

3.1.5.2 Pyro Bus Monitoring

The Airborne Support Equipment (ASE) telemetry allows monitoring the pyro bus enable relays in flight in the L-1011. The breakwires can be monitored by way of decommutated Pegasus telemetry received on the ground. The SCS hardware and software timers can also be monitored in this way. In the event that one or more inhibits were lost in the ordnance system, the ASE can be used to safe the spacecraft by opening the battery relays. (The battery relay can only be opened by the ASE, but it can be closed by ground command if the ASE is providing external power to the spacecraft or from the ASE.)

3.1.5.3 Safing Plugs

Pyro safing plugs are installed throughout processing up to launch day. Final bridge-wire resistance checks will be performed prior to the H_2 load. These checks use a calibrated piece of test equipment which is approved for bridge-wire checks. The resistance will be checked at the pyro arming connector using an approved procedure.

The other ordnance test involves attaching a piece of test equipment to the pyro arming connector and capturing the pulses as they are sent from the pyro box. This test will only be performed prior to the H_2 fill. No ordnance testing will be after H_2 is loaded in the cryostat. Arm plugs are not installed until hours before L-1011 take-off.

A separate command is required to remove each of the inhibits. The commands are marked "hazardous" in the ground system database. Each command displays a warning message that the command is a pyro command which must not be sent on the ground when H_2 is loaded, and a separte "allow" action is required by the operator for each of the commands which remove an inhibit.

Inhibits can only be removed, through a deliberate act. All procedures which run on the spacecraft after the H_2 is loaded will be the same procedures that are executed before H_2 is loaded. The procedures are written in Spacecraft Operations and Test Language (STOL) and run on ground support equipment under test conductor control. STOL procedures that enable, arm, or fire pyros will not be run after H_2 is loaded. Even if the procedures were run, the operator would be required to "allow" each command manually.

The pyro safing connector will be in place until about five hours before launch. No hazardous condition can occur with this inhibit in place. After this point, the launch-panel operator (LPO) on the L-1011 will constantly monitor the status of the pyro bus enable relays. The LPO will immediately remove power from the spacecraft if either of these relays close for any reason. The ground personnel monitoring spacecraft telemetry will verify all telemetry and instruct the LPO to remove power from the spacecraft if necessary for any other reason.

3.1.6 Non-Ionizing Radiation

The WIRE instrument has no sources of non-ionizing radiation.

3.1.7 Ionizing Radiation

The WIRE instrument has no sources of ionizing radiation

3.1.8 Acoustical Subsystems

The WIRE instrument does not generate any high noise sources.

3.1.9 Thermal Control Subsystems

Section 3.2.9 discusses thermal issues for the instrument as well as spacecraft.

3.1.10 Hazardous Materials

Hazardous materials used include helium, hydrogen, and cleaning fluids. See Table 3-6. A material Safety Data Sheet (MSDS) for each hazardous material is provided in Attachment D.

Table 3-6 INSTRUMENT HAZARDOUS CHEMICALS

Chemical Name	Quantity	Use	Hazard
Helium	Unlimited	Cryostat-liquid	Asphyxiant
Hydrogen	45 liters	Cryostat-gas-solid	Asphyxiant, explosive

Helium (He) is a colorless liquid. It is non-toxic but acts as an asphyxiant in high concentrations. He is a cryogen.

Hydrogen (H_2) is a colorless, tasteless, and odorless gas. It is non-toxic but acts as an asphyxiant in high concentrations. H_2 has an extremely wide flammability range and low ignition temperature. A H_2/air mixture will burn when ignited and under certain conditions, can explode. All inginition souces must be eliminated. H_2 is a cryogen.

Hazardous materials which are part of the instrument include arsenic (As) and beryllium (Be). Minute quantities of solid arsenic are enclosed within the telescope assembly. Arsenic is bonded to the silicon and cannot be released unless vaporized. The total mass of arsenic in both focal plane arrays is less than 0.4 micrograms. Each detector is mounted to a beryllium thermal post. The beryllium is in solid form. All beryllium materials are contained inside the cryogenic vessel. The arsenic and beryllium materials are not considered to be hazardous since both materials are in solid form and contained. Only materials in powdered form are considered to be hazardous.

3.1.11 Computing Systems Data

All critical commands are discussed in Section 3.2.11.

3.2 SPACECRAFT DESCRIPTION

3.2.1 General Description

The spacecraft bus shown in Figure 3-21 supports the WIRE instrument and spacecraft hardware. The instrument is attached to the top of the spacecraft. The bottom of the spacecraft connects to the Pegasus XL launch vehicle.

3.2.2 Structural/Mechanical

The WIRE spacecraft structural components are shown in Figure 3-21. The primary structure is comprised of three decks (supported by a truss structure), eight equipment panels, and eight access panels. There are 13 major components and/or electronic boxes mounted internally in the spacecraft structure:

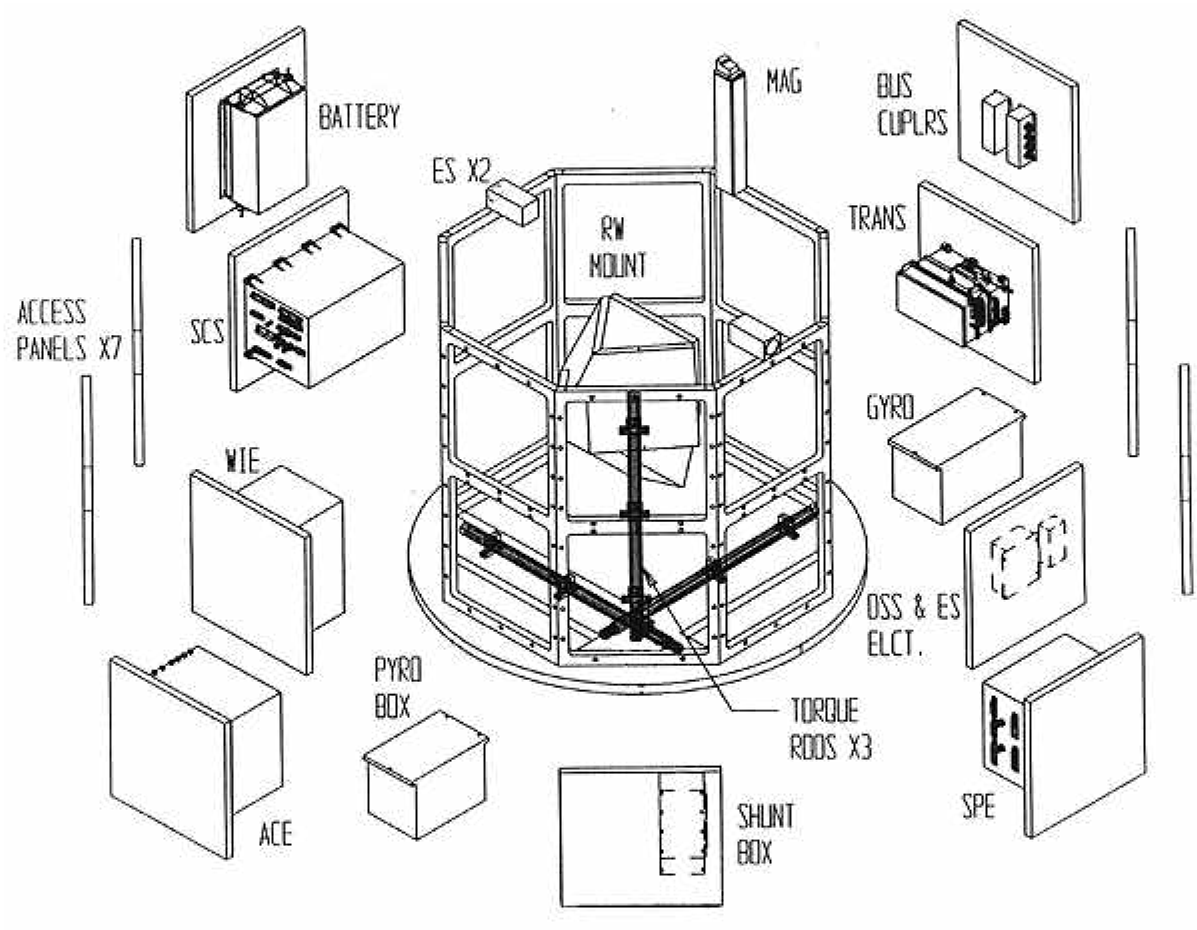
1. WIRE Instrument Electronics (WIE)
2. Spacecraft Computer System (SCS)
3. Attitude Control Electronics (ACE)
4. Gyro Assembly
5. Transponder
6. System Battery
7. Spacecraft Power Electronics (SPE)
8. Reaction Wheels
9. Shunt Drier Box
10. Magnetic Torquer Rods
11. 1553 Bus Couplers
12. Digital Sun Sensor Electronics
13. Earth Sensor Electronics.

Additionally, the following must be mounted externally on the spacecraft structure:

1. Magnetometer (MAG) Head
2. Digital Sun Sensor Head
3. Coarse Sun Sensor Head
4. Star Tracker
5. Earth Sensor Heads (2)
6. Communication Antennas
7. Test Connector Panel.

The structure must also accommodate the harness between the boxes. The high heat generating boxes will be placed on the anti-sun side of the spacecraft, to minimize the thermal load on the spacecraft and instrument.

Figure 3-21 Wire Spacecraft Structural Components



The WIRE Spacecraft is being launched from the Pegasus-XL that has a static envelope diameter of 44 inches height of 83 inches, with the upper portion being a curved ogive section. The payload I/F is a 38 inch diameter aluminum ring with a 60 x ¼ inch bolt hole pattern.

3.2.3 Pressure, Propellant, and Propulsion Systems

The WIRE spacecraft has no pressure, propellant, or propulsion systems.

3.2.4 Electrical and Electronics Subsystem

This section describes elements of WIRE electrical and electronics subsystem. The Spacecraft Computer System (SCS), the Power Subsystem, the Attitude Control Electronics (ACE) Box, Transponder, and WIE comprise the spacecraft electrical and electronics systems.

3.2.4.5 Spacecraft Computer System (SCS)

The SCS is an on-board flight computer programmed to operate the spacecraft and perform mission unique functions as required when the spacecraft is not in contact with the ground station. The SCS controls the spacecraft attitude and provides primary command and control of the WIRE instrument and spacecraft subsystem, and interfaces with the spacecraft communications subsystem. See Data Flow Block Diagram Figure 3-22.

The SCS consist of: the Command and Data Handling (C&DH)-386 Board, the Bulk Memory Board, the Up/Down Board, and the Input/Output (I/O) Board. The C&DH-386 Board contains an Intel 80386 CPU with an Intel 80387 math co-processor and is responsible for C&DH, general spacecraft control and software control of all non-safehold ACS functions. The Bulk Memory Board is responsible for storage of science data and spacecraft housekeeping. The Up/Down Board interfaces to the transponder. The I/O Board interfaces to the SCS to the Pegasus and the spacecraft thermal subsystem. It controls miscellaneous I/O functions and supports the spacecraft time maintenance.

For general spacecraft control, the CPU within the SCS monitors data to determine the general status of the spacecraft, the WIRE instrument, and data system. The SCS can check critical parameters to determine if limit conditions have been exceeded. If so the SCS can execute predetermined stored command sequences, working in conjunction with a safehold mechanism. The SCS periodically performs a variety of self tests and initiates corrective action.

The SCS also collects data from the different subsystems and the WIRE instrument, stores the data, processes the data, and sends the data to the ground using Consultative Committee for Space Data System (CCSDS) packet telemetry standards.

The SCS receives commands from the ground via the transponder using CCSDS packet telemetry standards. It has real-time ground, relative time sequencing, and absolute time sequencing command capabilities. The SCS sends commands to the subsystems and the WIRE instrument over 1553 busses.

3.2.4.6 Power Subsystem

The WIRE power system supplies ±6 volt power to the spacecraft via two busses: the essential bus and the non-essential bus. Figure 3-23 contains the Power Flow Diagram. The essential bus provides continuous power to the essential spacecraft functions. The non-essential bus provides power to the WIRE instrument and the star tracker. The busses are routed to the SPE for distribution switching and fusing (where appropriate). The power subsystem includes an isolation relay to disconnect the non-essential bus in the event that the spacecraft is drawing excessive current, the battery is in an undervoltage condition, the battery has reached a low state of charge, four or the spacecraft has gone into safehold. The essential and pyro busses are unprotected in the power subsystem.

Figure 3-22 Data Flow Block Diagram

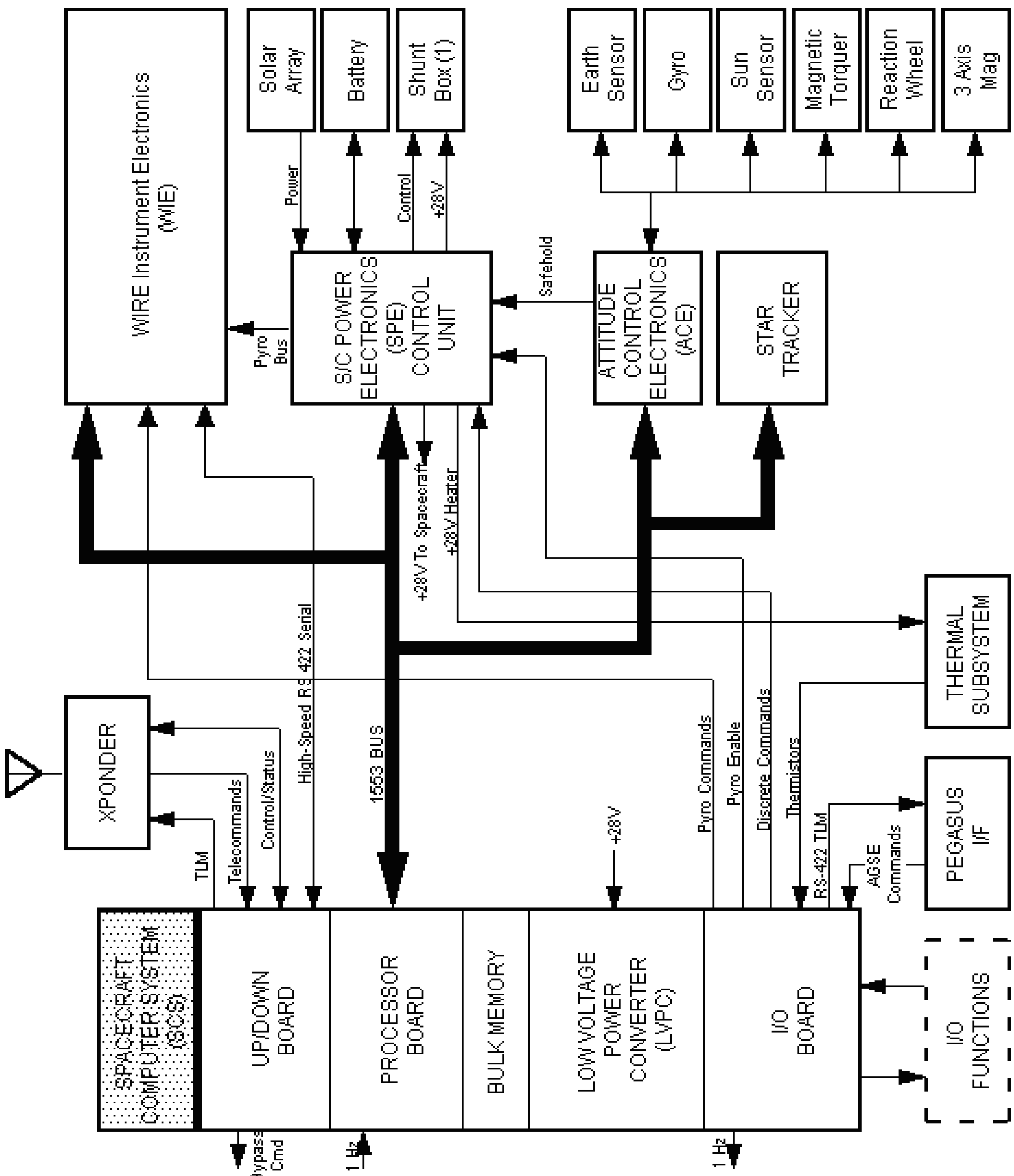
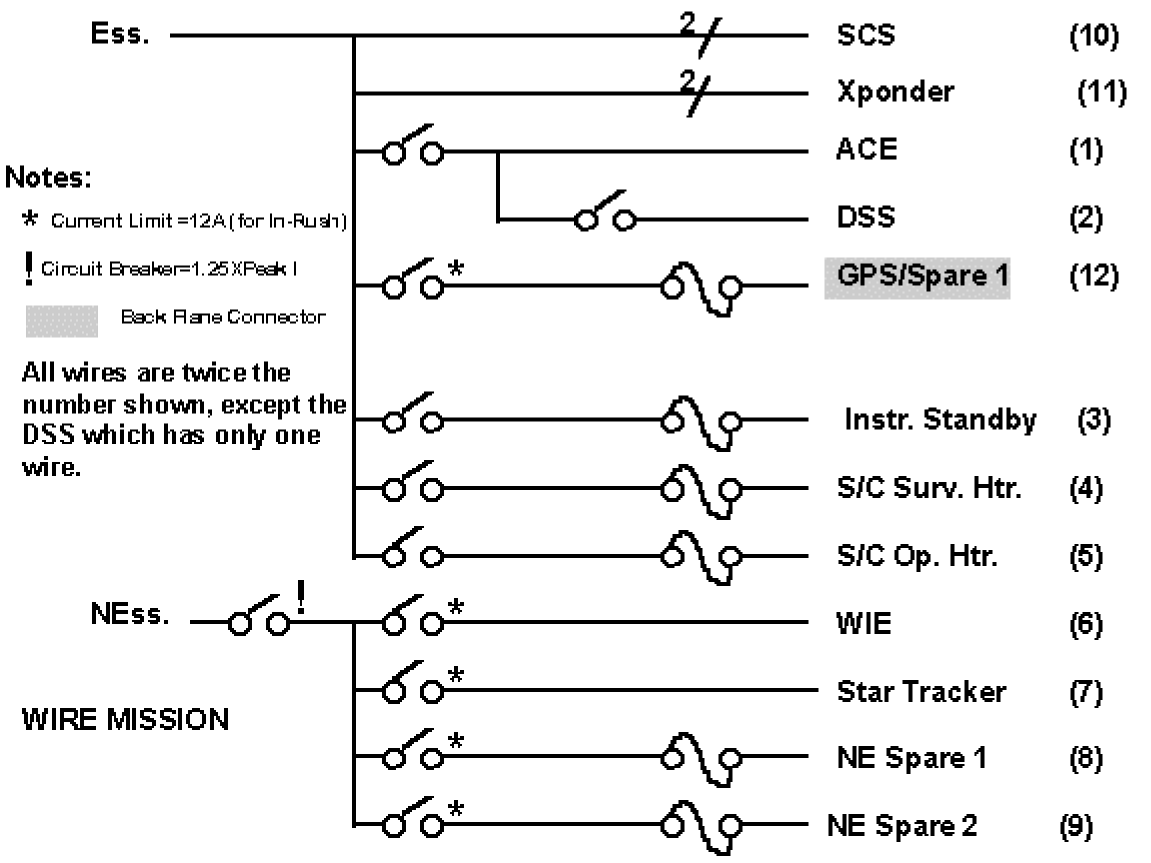


Figure 3-23 Power Flow Diagram



Power generated by the solar arrays is supplied directly to the observatory loads through the unregulated busses. Each subsystem provides its own DC to DC converter and power conditioning. Shunt regulators are used to dissipate any excess array power, and a Super Nickel-Cadmium (NiCd) battery supplies energy when spacecraft power requirements exceed array capability and during eclipse periods. The SPE controls battery charging and dissipation of excess energy through shunt regulators.

3.2.4.7 Spacecraft Power Electronics (SPE)

The SPE controls Spacecraft power distribution and battery charge control. It monitors and detects power system safehold trigger conditions and collects all power system temperature telemetry.

The SPE provides the following functions:

- Battery charge control
- Power Distribution-three busses (the essential bus and the non-essential bus). The essential bus which carries all primary load required to support the core spacecraft. The non-essential bus carries all primary loads not required for safehold; the pyro bus (pyro commands are initiated by the SCS but controlled by the SPE and WPE).
- Capability to accept external simulated solar array power.
- Generate a safehold request for the SCS-this is generated when triggered by SPE internal hardware safehold monitors.
- Automatic disconnect of the non-essential bus upon detection of a battery UV/OL condition, very low battery state of charge, or the receipt of the safehold discrete from the Attitude Control Electronics (ACE).
- Monitor and detect power safehold trigger conditions.
- Provide power to the solar array release actuators (wax actuators) via the command timers and ground command.
- Limit bus voltage to never exceed 35.0 volts.
- Enable and disable pyro relays via the commands timers and ground command.

The SPE provides the following functions through the External Umbilical:

- Disconnect the battery by GSE command. The SPE does not have the capability to disconnect the battery by ground command,
- Deliver 28 volt external spacecraft power, and
- Battery trickle charging.
- Provide critical analog telemetry including bus voltage, battery voltage, bus current, battery current, battery relay status, and pyro bus relay status.

The only fused elements are those which, are not essential to mission success. The heaters circuits require flight fusing protection on the 28 volt bus level. Spacecraft power sources are externally fused during testing. Fusing and wiring sizes are described in Appendix C.

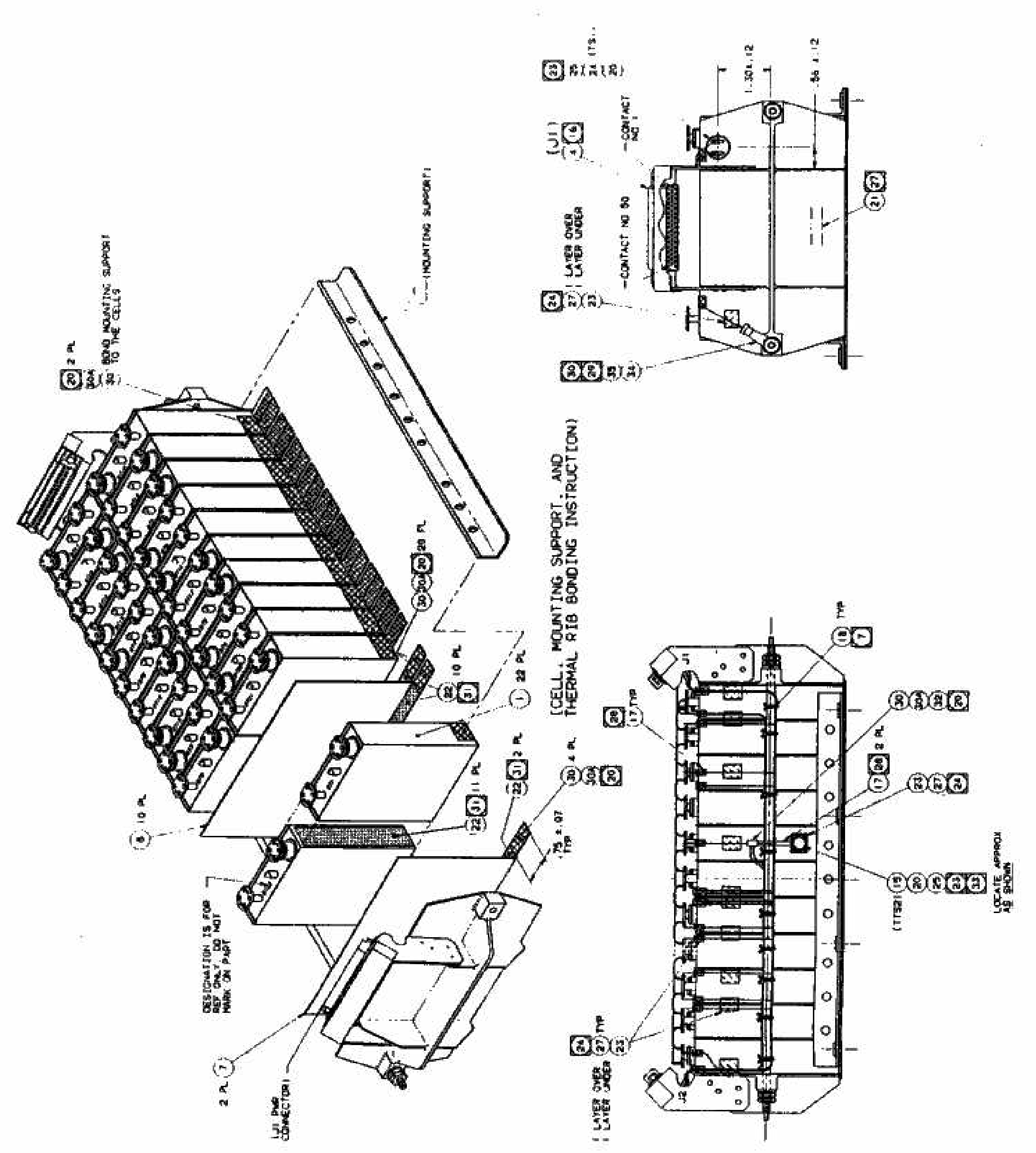
The battery charge control is maintained by a voltage/temperature (V/T) controller and a current controller. The V/T controller has 16 selectable V/T levels. The current controller is regulated by an Amp Hour Integrator (AHI). These electronics provide for V/T battery charge control and overcurrent/overvoltage control.

The shunt drivers are physically separated from the SPE so as to provide for a more efficient spacecraft thermal system design. The 8 stage sequential shunt drivers are housed in a separate box. They are sized to dissipate peak at beginning of life (BOL) power.

3.2.4.8 Battery and Charging

A 9 Ampere Hour Super Nickel-Cadmium (9 AH Super NiCd) battery stack, with 22 series connected cells, provides 22-34 volts. Figure 3-24 shows the battery pack I/F hardware. The cells contain a 31percent potassium hydroxide (KOH) electrolyte concentration. The battery

Figure 3-24 Battery Diagram


(CELL, MOUNTING SUPPORT, AND THERMAL RIB BONDING INSTRUCTION)
DESIGNATION IS FOR REF ONLY, DO NOT MARK ON PART
MOUNTING SUPPORT
LOCATE APPROX AS SHOWN

supplies energy when spacecraft power requirements exceed array capability. The battery is trickle charged from the solar arrays and operates in the range of 22 to 34 volts, with 26.4 nominal discharge voltage.

The battery can withstand a minimum of five thousand charge/discharge cycles, with the maximum depth of discharge at 25 percent. Two charge control temperature sensors are mounted on the top center of the battery.

The temperature range for the battery is from -5 to +25ºC. The battery is equipped with five temperature sensors mounted on top of the battery, and an over temperature thermostatic switch to terminate charging at +35° ± 2.3° C.

The C&DH subsystem monitors the battery state of charge. If the state of charge drops below a ground variable limit (default=85%) the science pointing timeline will be aborted and the spacecraft placed in zenith sunpoint.

The SPE controls battery charging and dissipation of excess energy through the shunt regulators. WIRE utilizes the Transition Region and Coronal Explorer (TRACE) heritage Direct Energy Transfer (DET) system. The SPE controls battery charge, monitors power distribution, operates the flip mirror, and collects power system temperature telemetry.

The SPE detects battery under voltage conditions and “very low battery state of charge”. The “very low battery state of charge” limit is variable by ground command. If the battery goes into an under voltage condition, has a very low battery state of charge, or receives a safe-hold discrete from the ACE, the SPE disconnects the non-essential bus.

The battery and SPE are mounted so that interconnecting harnesses are less than three feet in length. The battery is mounted to allow for easy removal and installation at the launch site. There are two keyed electrical connectors. Removal and installation do not require the removal of or disassembly of any other box or instrument.

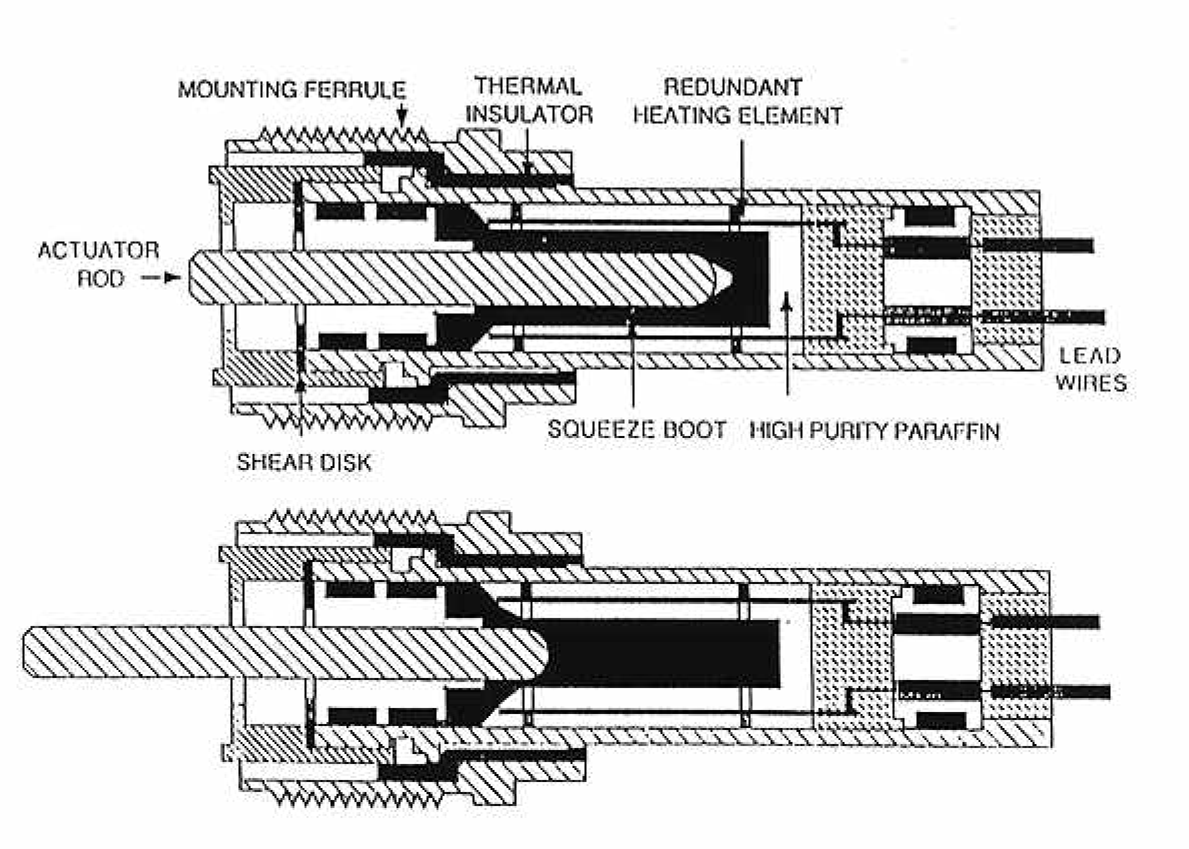
The battery weighs 11.7 Kg. The cell burst pressure, based on Hughes pneumatic burst test, is greater then 500 psia. The maximum expected operational pressure is 34 psig (49 psia), based on similarity to Solar Anomalous and Magnetosheric Particle Explorer (SAMPEX). During acceptance test, cells are pressurized to 150 percent of maximum expected operating pressure (MEOP) for leak check. Thus the safety factor is at least 10:1.

3.2.4.9 Solar Arrays

The solar arrays consists of two deployable panels. The arrays use GaAs/GE solar cells with a 10 mil Ceria-Doped Microsheet (CMX) coverglass to protect solar cells from particulate radiation, especially low-energy protons. The arrays are required to support a 160 Watt orbit average load at the end of a four month mission.

The Solar Arrays are deployed when heat from the resistive heaters are applied to the wax actuators at 300-400mA. WIRE uses the Starsys RL-50-C Launch latch. Redundant internal heaters melt a paraffin charge inside the actuators. The paraffin expands during melting and that expansion is transformed to gentle, high force shaft extension in approximately 5-10 minutes. When the actuator is energized, it acts upon an internal latch pin, causing it to slide relative to the mating, T-shaped release bar. Each solar array is released separately. There is a latch on both of the +X and -X sides of the solar arrays. The release of the solar arrays is not considered hazardous due to their slow release and use of non-hazardous wax actuators. See Figure 3-25.

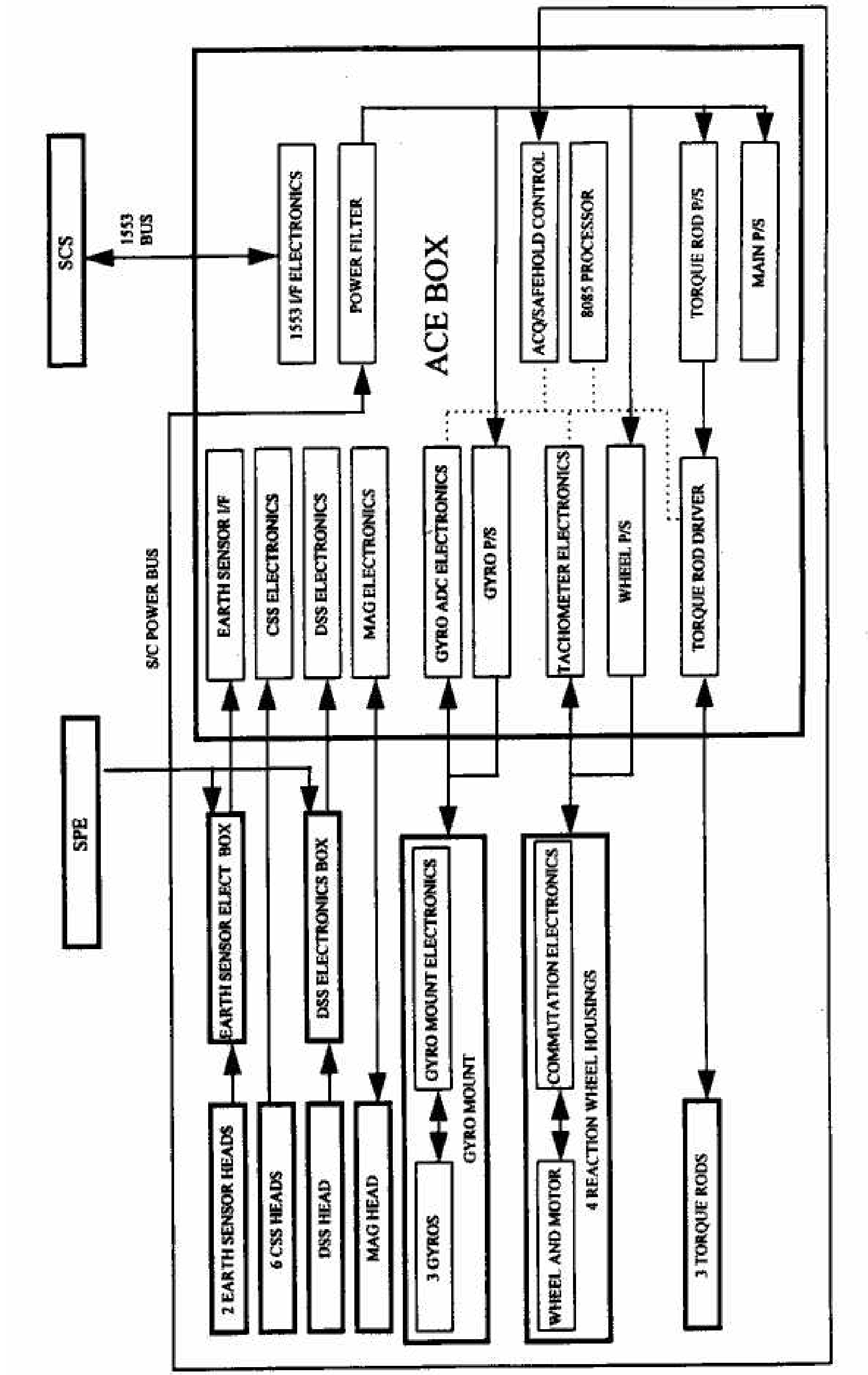
Figure 3-25 Wax Actuator



3.2.4.10 Attitude Control Electronics

The Attitude Control Electronics includes the signal conditioning electronics, wheel drivers and independent analog safehold circuitry. The ACE box provides the I/F for all of the ACS sensors and actuators. The electronics which are active in safehold operation or triggering are designed for maximum reliability, i.e., the component deratings are conservative, devices are Single Event Upset/Single Event Latchup (SEU/SEL) immune (Lethal Exposure Threshold (LET) threshold > MeV • cm^2/mg of Si), and are capable of tolerating three times the mission total dose radiation requirement. The requirement also applies to the sensors and actuators used for safehold. See Figure 3-26 ACE Functional Block Diagram.

Figure 3-26 ACE Functional Block Diagram



3.2.5 Ordnance Subsystem

WIRE contains a primary and redundant pyro bus (from the SPE) and is capable of receiving pyro commands (initiated by the SCS but controlled by the SPE and WPE). The pyro bus is capable of being switched off. See section 3.1.5 Instrument Ordnance Subsystem for more details.

3.2.6 Non-Ionizing Radiation

Hazards associated with RF radiation include inadvertent transponder turn-on (including improper test/integration procedure).

3.2.6.1 Transponder

The WIRE spacecraft contains one S-Band transponder operating in full duplex mode. It provides the reception for up-linked commands, transmission of telemetry data and supports the Doppler tracking by the designated ground station. A summary of the transponder is provided in Table 3-7.

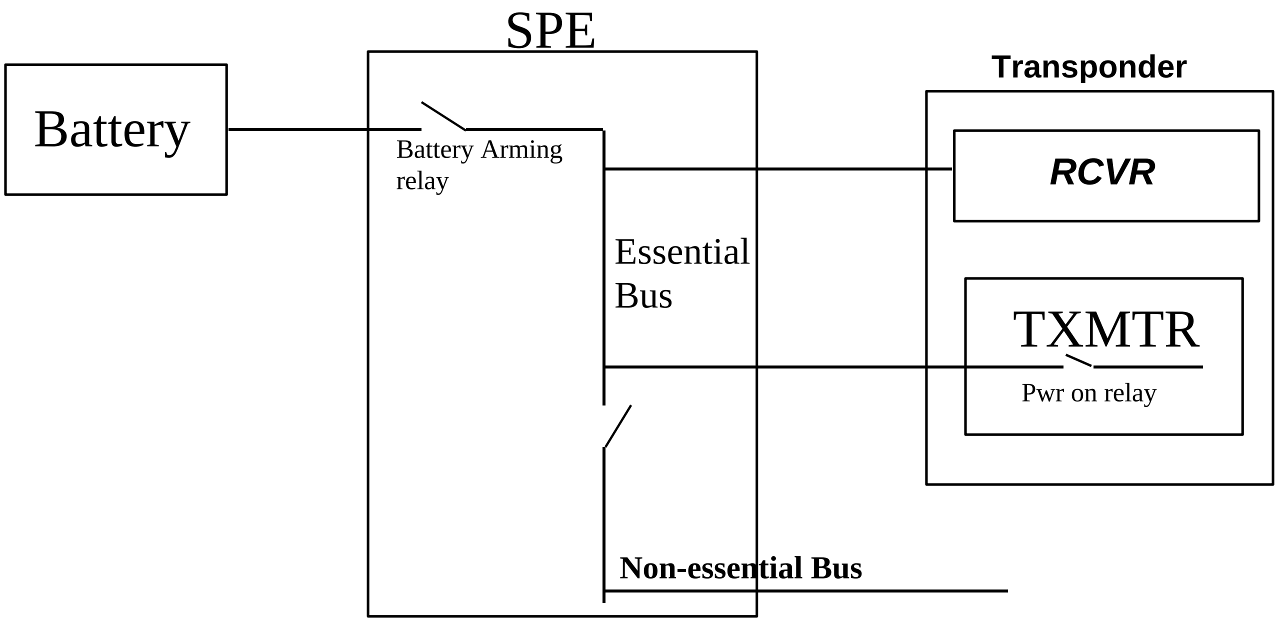
The transponder is mounted on the essential bus and is powered on by a relay in the transponder's transmitter power section. A single command from the ground station or the Integration and Test

Table 3-7 TRANSPONDER CHARACTERISTICS

TRANSMITTER	
Bandwidth	7 Mhz/70 Mhz
Transmitted Power	5 Watts
Type of Modulation	PM/Biphase 1/2 Convoluted PM/PSK
Frequency	2215.0 Mhz
Spurious Rejection	- 60 dBc
Frequency Stability	± 20 ppm or better
Maximum Bit	5 Msysmbols/second
Weight	7.1 pounds
Manufacturer	Loral Conic
RECEIVER	
Bandwidth	12.25 Mhz/120 Mhz
Sensitivity	- 137 dBm
Frequency Stability	± 3 Khz
Tuning	Fixed, Crystal Controlled
Type of Modulation	Dual Conversion Super Heterodyne
Image Rejection	75 dBc
Spurious Response Rejection	94 dB above receiver threshold
Weight	7.1 pounds
Manufacturer	Loral Conic

GSE is all that is required to turn on the five watt transmitter. The essential bus is powered when the spacecraft is powered. Figure 3-27 provides a visual overview.

Figure 3-27 Transponder Block Diagram



During ground processing, procedural controls assure the transmitter is not commanded on. While the transponder functional procedures are taking place, all personnel are required to follow a "safe distance" requirement of ____feet away from the antennas. Arming connectors will be used until final arming of the spacecraft until before launch.

Turning on the transmitter during ground processing through captive carry requires the closure of a relay in the transponder. The spacecraft computer system controls the closure of this relay through the use of Relative Time Sequencing (RTS). To prevent an accidental turn on the transponder during captive carry the flight software monitors the separation loop backs and turns off the transponder if WIRE has not separated. This check is performed once every five seconds. If the transmitter were turned on during assent vibration or through the sending of a command, it would only allow the transmitter to be on for five seconds.

A command to disable the RTS and a command to power on the transmitter would both be needed in order for the transmitter to radiate during captive carry. There is no procedure requiring the use of either these commands during captive carry.

3.2.6.2 Antennas

The WIRE spacecraft has two left-hand circularly polarized quadrifiliar helical fixed antennas mounted on the upper outside corner of each solar array. Theses were the same antennas that will be used on SWAS. Figure 2-1 shows the antenna locations on the solar array.

RF radiation hazard analysis has been performed. The safe minimum radius from the antenna radiation center for human exposure was calculated to be 5.6 inches based on the following:

- 5 m W/cm^2 maximum allowed field strength
- 38 dBm maximum RF output power
- +4 dBi maximum antenna gain (Gt)

Personnel will be restricted to prevent any bodily exposure within 5.6 inches of either antenna while the transponder is activated.

3.2.7 Ionizing Radiation

The WIRE spacecraft does not contain an Ionizing Radiation Producing Subsystem.

3.2.8 Acoustical Subsystems

Currently no equipment, procedures, or operations associated with or related to WIRE have been identified that exceed 84 dBA.

3.2.9 Thermal Subsystems

WIRE uses a classic passive thermal design approach supplemented with heater control. Heat is conducted from each of the individual subsystems to the spacecraft and then radiated within, into and out of the spacecraft. The passive design is optimized by the use of specific thermal control coatings on the exterior and interior of the spacecraft. The appropriate radiative area and performance is controlled by the extensive use of multi-layer insulation (MLI) blanketing, on the equipment panels.

The instrument is thermally isolated from the spacecraft bus, (low conductivity supports, MLI, harness design). The battery is thermally isolated from the rest of the spacecraft due to narrow temperature requirements (0 to 20° C vs -10 to 40° C for the rest of spacecraft). See Figure 3-28 and 3-29 for thermal control design.

The WIRE thermal control system (TCS) will utilize two heater circuits. An operational circuit will supplement the passive control during cold operational periods. A survival circuit will provide needed heat to the spacecraft during launch, safehold, and contingency situations to keep components above their survival/turn on operational limits. Heaters will be selected and sized based on location and bus voltage requirements. Redundant thermostats will control each heater.

The spacecraft computer will control a set of thermistors to monitor spacecraft temperatures. Each subsystem component will have internal thermistors mounted in key locations. Other thermistors will be located throughout the spacecraft to monitor key structural and I/F locations. These thermistors will be individually wired back to the computer since they are located external to any specific component. The spacecraft computer will monitor key instrument locations using thermistors and germanium thermistors.

Figure 3-28 Thermal Control Design (Front View)

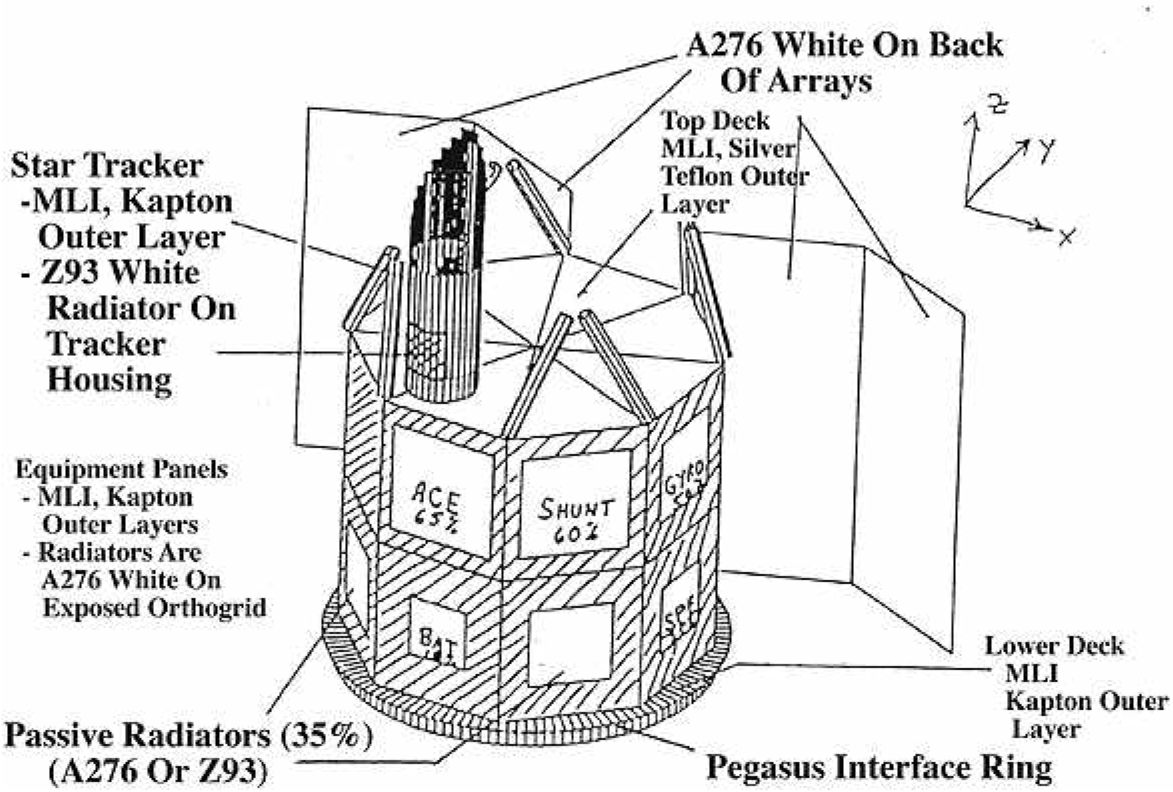
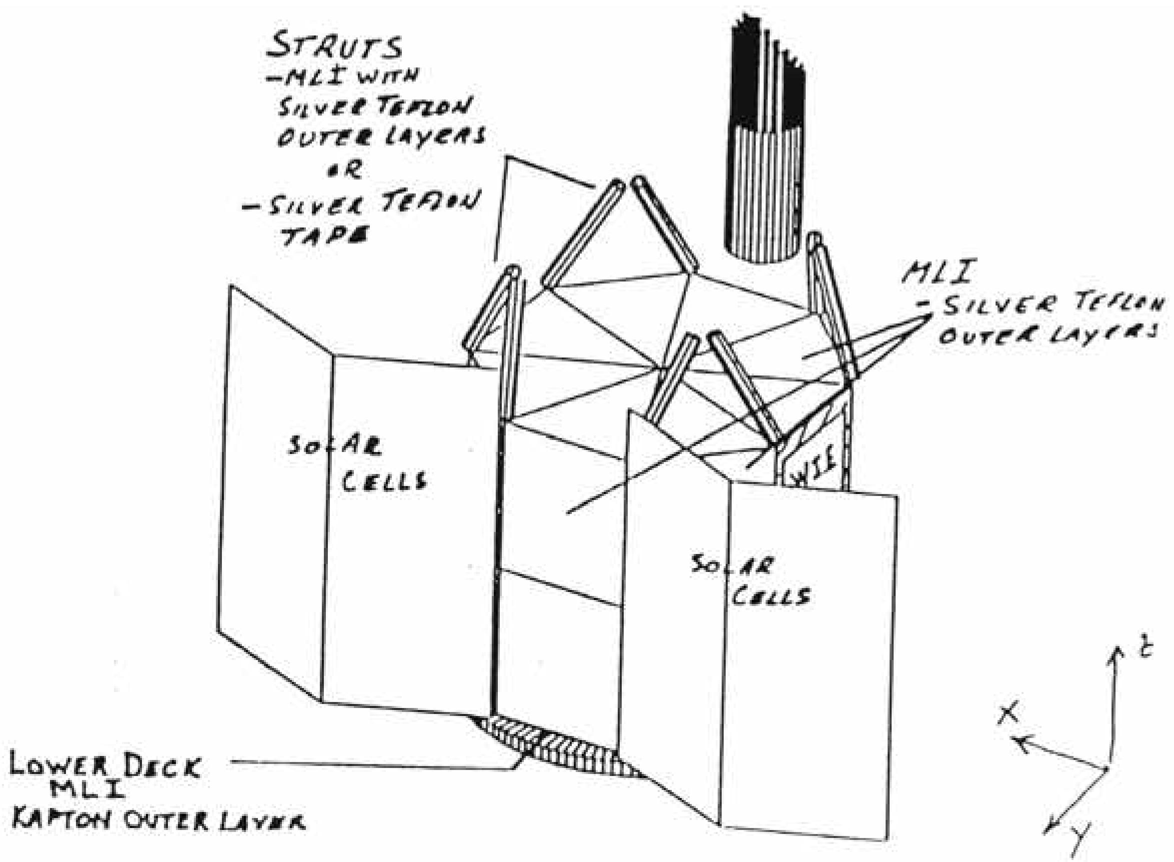


Figure 3-29 Thermal Control Design (Back View)



Five thermistors will be mounted externally on the WIRE cryostat in the following locations, and tied to the SCS analog electronics for temperature monitoring while the WIE box is off.

The locations are:

Thermistor 1: Aperture shade outer cone (+Y)
Thermistor 2: Aperture shade outer cone (-Y)
Thermistor 3: Vacuum shell near door separation plane
Thermistor 4: Vacuum shell at rear dome
Thermistor 5: On primary vent wax housing

Additionally, four critical cryostat temperatures (two on each tank) will be monitored via the spacecraft umbilical. These temperatures will be available to the launch panel operator on the L-1011 during captive carry.

The cryo-diode locations and pin connections are:

Primary tank top GND (-Y) (P)
Primary tank top GND (-Y) (R)
Primary tank bottom GND (+Y) (P)
Primary tank bottom GND (+Y) (R)
Secondary tank top GND (-Y) (P)
Secondary tank top GND (-Y) (R)
Secondary tank bottom GND (+Y) (P)
Secondary tank bottom GND (+Y) (R)

3.2.10 Hazardous Materials

The WIRE spacecraft uses a nine ampere hour battery. This super Ni-Cd battery uses 21 cells. These cells contain electrolyte of concentration 31 percent KOH, with a proprietary additive. The cells are vacuum filled and subjected to a series of pressure stabilization cycles.

The electrochemistry of a super Ni-Cd battery is such that oxygen or hydrogen is generated internal to the battery cells, under certain conditions. The cell casing is hermetically sealed and rated at an internal pressure not to exceed 100 psig. If the casing were to rupture due to external puncture and/or internal overpressurization, the oxygen and/or hydrogen could be released. Released hydrogen and oxygen are flammable. The amount of hydrogen ($\cong$ 3600 cm^2) or oxygen released from a ruptured cell is not sufficient for generating an explosive/flammable atmosphere. Potassium hydroxide (KOH) is a caustic electrolyte which can cause severe burns. If the electrolyte gets onto the skin or eyes, the area must be flushed with copious amounts of water for fifteen minutes. Medical assistance must be obtained. A sample MSDS for nickel cadmium battery is located in Appendix B.

3.2.11 Computing Systems Data

Inadvertant critical commands are precluded by requirement that an operator verify that the command should be issued. The operator then has a chance to cancel the command, or send it to the spacecraft. Critical Commands are listed Table 3-8.

TABLE 3-8 Critical Commands List

SUBSYSTEM	COMMANDS	DESCRIPTION
SPE	/PSPYROA (ON or OFF)	Turns on (or off) pyro bus A
SPE	/PSPYROB (ON or OFF)	Turns on (or off) pyro bus B
SCS	/IPRYO (ARM or RESET)	Arms the pyro bus for 10 seconds
SCS	/IPRIVENT (DEPLOY or RESET)	Enables (or disables) power to the primary vent wax actuator
SCS	/ISCVENT (DEPLOY or RESET)	Enables (or disables) power to the secondary vent pyros
SCS	/IACOVER (DEPLOY or RESET	Enables (or disables) power to the aperture cover pyros
SCS	/IIBILVLOUT	Generic command, sends bilevels out which control the pyro box