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Feasibility Study for a Cryogenic On-orbit Liquid Depot-Storage, Acquisition and Transfer (COLD-SAT) Satellite

Final Report for Phase A Study

Contract Period: February 1988 - March 1990

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TABLE OF CONTENTS

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TABLE OF CONTENTS (Continued)

DUA0:[CRYO]CLDSATFR.AG9;43.TOC-20-AUG-90

TABLE OF CONTENTS (Continued)

DUA0:[CRYO]CLDSATFR.AG9;43.TOC-20-AUG-90

TABLE OF CONTENTS (Continued)

DUA0:[CRYO]CLDSATFR.AG9;43.TOC-20-AUG-90

TABLE OF CONTENTS (Continued)

ILLUSTRATIONS

TABLE OF CONTENTS (Continued)

ILLUSTRATIONS

TABLE OF CONTENTS (Continued)

DUA0:[CRYO]CLDSATFR.AG9;43.TOC-20-AUG-90

TABLE OF CONTENTS (Continued)

TABLE OF CONTENTS (Continued)

TABLE OF CONTENTS (Continued)

DUAØ:[CRYO]CLDSATFR.AG9;43.TOC-2Ø-AUG-9Ø

TABLE OF CONTENTS (Continued)

FOREWORD

This Final Report summarizes the technical effort performed by Ball Aerospace Group (BASG) in conjunction with its team members McDonnell Douglas Space Systems Company (MDSSC) and Boeing Aerospace and Engineering (BA&E) for NASA Lewis Research Center (LeRC) under contract no. NAS3-25054. The contract was administered by LeRC and the following are the lead personnel involved with the study.

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LIST OF ACRONYMS

ACP	Attitude Control Processor
ACS	Attitude Control Subsystem
A/D	Analog-to-Digital
AFM	Air Force Manual
AMP	Amplifier
ATP	Authorization To Proceed
BAEC	Boeing Aerospace and Electronics Company
BAPS	Biaxial Antenna Pointing System
BASG	Ball Aerospace Systems Group
BIF	Bus Interface Processor Command Decoder Unit (portion of TCP)
BITE	Built In Test Equipment
BPS	Bits Per Second
BSSD	Ball Space Systems Division
CCAFS	Cape Canaveral Air Force Station
CDHF	Central Data Handling Facility
CFMFE	Cryogenic Fluid Management Flight Experiment
CG	Center of Gravity
CM	Center of Mass
CMOP	COLD-SAT Mission Operations Procedures
CMOS	Complimentary Metal-Oxide Semiconductor
COBE	Cosmic Background Explorer (Satellite)
CPOCC	COLD-SAT Project Operations Control Center
CRRES	Combined Release and Radiation Effects Satellite
CRT	Cathode Ray Tube
CSM	Command Storage Memory
CSM	Critical Status Monitor
CTV	Compatibility Test Van
DAC	Data Acquisition and Control
DC	Direct Current
DCF	Data Capture Facility
DSN	Deep Space Network (replaced GSTDN)
DSU	Data Storage Unit
ECP	Experiment Control Processor

LIST OF ACRONYMS (Continued)

EED	Electro-Explosive Device
ELV	Expendable Launch Vehicle
EMC/EMI	Electromagnetic Compatibility/Interference
EOL	End of Life
EPS	Electrical Power Subsystem
ERBS	Earth Radiation Budget Satellite
ERD	Experiment Requirements Document
ERTS	Earth Resource Technology Satellite
ESMCR	Eastern Space and Missile Center Range Safety
ETR	Eastern Test Range
FDF	Flight Dynamic Facility
FOT	Flight Operations Team
FTG	First To Go Power Bus
GFM	Gas Flow Meter
GG	Gravity Gradient
GHe	Gaseous Helium
GH_2	Gaseous Hydrogen
GRT	Germanium Resistance Thermometer
GSE	Ground Support Equipment
GSFC	Goddard Spaceflight Center
GSRD	Ground Segment Requirements Document
GSTP	Ground System Test Plan
HOP	High Output Paraffin Linear Actuator
HSE	Horizon Scanner Electronics
HX	Heat Exchanger
ICD	Interface Control Document
INT	Internal Reference Unit
IPD	Information Processing Division
IRAS	Infrared Astronomical Satellite
IR&D	Internal Research and Development
IRU	Inertial Reference Unit
I&T	Integration and Test
JPL	Jet Propulsion Laboratory

J-T	Joule-Thomson Device
KBPS	Kilo-Bits per Second
KSC	Kennedy Space Center
L/D	Length to Diameter Ratio
L/V	Liquid/Vapor Sensor in Experiment
LAD	Liquid Acquisition Device
LAN	Local Area Network
LeRC	NASA Lewis Research Center
LFM	Liquid Flow Meter
LH_2	Liquid Hydrogen
LTG	Last To Go Power Bus
MA	Multiple Access, TDRSS Mode
MDAC	McDonnell Douglas Astronautics Company
MDC	Mission Directors Center
MILA	Merit Island Tracking Station
MLI	Multiple Layer Insulation
MMC	Martin Marietta Corporation
MOP	COLD-SAT Mission Operations Plan
MSOCC	Multi-Satellite Operations Control Center
MUX	Multiplexer
M/V	Mass-to-Volume Ratio
NASCOM	NASA Communications
NCC	Network Control Center
NFPA	National Fire Protection Association
NGT	NASA Ground Terminal
NMI	Nautical Miles
NSI	NASA Standard Initiator
NVF	No-Vent Fill
OAO	Orbiting Astronomical Observatory
OS	Outer Shell
OSC	Orbit Support Computing
OTV	Orbital Transfer Vehicle
PA	Power Amplifier in TT&C

Section 1
INTRODUCTION

An improved understanding of low-gravity cryogenic fluid behavior is critical for the continued development of space-based systems. Initially, research concentrated on fluid positioning in a low-gravity environment, but as it became clear that future space missions would require cryogenic liquids, test efforts expanded to include self-pressurization due to induced heat loads. The early programs (conducted primarily in drop-towers or on Aerobee sounding rockets) provided some fundamental understanding of zero-gravity cryogenic fluid behavior. However, more extensive flight data is required to design cryogenic liquid storage and transfer systems with confidence for space applications. As NASA's mission concepts evolved, now including Space Station Freedom and the Space Exploration Initiative, the demand for optimized in-space cryogenic systems is increasing.

COLD-SAT is an experiment designed as a free-flying satellite to address the major technological issues associated with on-orbit supply, storage, and transfer of cryogenic liquids. Delivered into orbit by an expendable launch vehicle, COLD-SAT will conduct experiments in passive and active pressure control, tank chilldown and filling, and filling and draining of liquid acquisition devices using liquid hydrogen as the test fluid. These experiments, which require approximately 3 months to complete, will extend the existing low-gravity fluid database by many orders of magnitude and provide future system designers with vital performance data from an on-orbit environment.

1.1 BACKGROUND FOR LOW-G EXPERIMENTS

Low-gravity cryogenic systems can be divided into 4 broad classes: (1) Supercritical systems (such as the PRSA shuttle tanks) which maintain cryogenic fluids above their critical points to facilitate expulsion of single-phase fluid, (2) Superfluid helium systems (such as IRAS or COBE) that operate at reduced pressures and use thermal gradients through a porous plug to collect and transfer fluid, (3) Solid cryogenic coolers which provide cooling by maintaining a solid cryogen in the sublimation region, and (4) Liquid cryogen systems which are stored at pressures less than 350 kPa (50 psia) near their boiling point and require special subsystems for venting (pressure control) and liquid expulsion. Of these four systems, the first three have extensive flight heritage, but liquid (2-phase) cryogenic systems have little or no flight heritage.

Storing cryogens as liquids at low pressures near their normal boiling points is more weight-efficient for most future applications; therefore, development of sub-critical cryogenic storage systems is essential. A flight experiment called Cryogenic Fluid Management Flight Experiment (CFMFE) was developed by NASA Lewis Research Center and the Martin Marietta Company in the early 1980's, but was ultimately rejected as a shuttle payload because of hydrogen safety related issues. The COLD-SAT experiment was then conceived and developed to address the critical fluid technology issues.

1.1.1 Need for Low-Gravity Fluid Technology

Three fundamental problems with low-gravity two-phase liquid systems which must be addressed are: (1) tank pressure control while venting vapor only, (2) vapor-free liquid expulsion from the tank, and (3) chilldown and filling of a warm tank.

All cryogenic storage systems require some form of pressure control. Supercritical systems vent single-phase fluid directly; superfluid helium systems do not stratify and use porous plugs to achieve phase separation and control pressure; solid cryogen coolers vent vapor only by sublimation. In 2-phase (liquid-vapor) systems a unique problem arises in the low-gravity environment: how to control pressure by venting vapor only without discarding valuable liquid. The most promising technique for pressure control uses the thermodynamic vent system (TVS) either in a passive mode or in an active mode with a fluid mixer. This system is designed to vent only vapor, but relies on effective heat transfer to the cryogen inside the tank to accomplish pressure control. It is very likely that all future liquid cryogen systems will use some type of TVS system.

Liquid cryogens must be stored on orbit for many reasons, including propellant supply, instrument cooling, and life support. To capitalize on the advantages of liquid storage, the liquid must be readily available for delivery and distribution from the storage vessel. In a low-gravity environment, surface-tension devices, such as fine mesh screened channels, are the most promising approach for liquid delivery. Although these devices have been used successfully for storable propellants (such as hydrazine), they have not been flown in cryogenic systems.

Many future mission concepts require re-supply of cryogen storage vessels. In these concepts, a cryogen tank is empty and must be chilled and filled in the low-gravity environment. In ground-based systems, these operations are carried out routinely because vapor always vents out the top of the tank. In low-gravity, vented tank fills require fluid settling using induced thrust, which is not desirable for most large systems. An alternative concept, the charge-hold-and-vent chilldown and subsequent no-vent fill appears promising for on-orbit resupply operations.

1.1.2 Previous Work on Low-Gravity Fluid Management

Low-gravity fluid management technology has been under development for over twenty years. Self-pressurization studies were conducted in Aerobee sounding rockets in the mid-1960's to determine the effects of heating on cryogenic storage tanks. Numerous TVS system concepts have been developed by different contractors. However, ground testing of TVS systems can not demonstrate on-orbit performance for controoling tank pressure; therefore, actual flight data is required to validate the TVS concepts.

Surface tension devices have been studied extensively by the McDonnell Douglas Company, the Martin Marietta Company, and more recently, other contractors have initiated development programs for screened-channel liquid acquisition devices (LADs). Flight data for LADs in hydrazine and water tanks has shown that these devices are effective for collection and expulsion of liquid in a low-gravity environment. Cryogenic systems, however, are more sensitive to heating from the external environment or from warm pressurant introduced for liquid expulsion, and their breakdown thresholds are lower than for storable fluids.

Chilldown and filling of warm tanks have not been tested in low-gravity. These processes are not amenable to short tests (such as drop tower tests) due to the time required to conduct a chill or fill operation. However, extensive modeling efforts sponsored by Lewis Research Center have been underway since 1980, and recently, ground testing at the NASA Plumbrook facility has begun.

The fluid dynamics of tank filling and draining was extensively investigated using drop tower experiments and a small tank (non-cryogenic) in the 1960s. These tests demonstrated that fluid rapidly repositions itself in a low-gravity environment and that "normal" methods of filling and draining will not operate satisfactorily. The results of these tests support the need to develop alternative filling and draining procedures using no-vent fill and LADs.

Although a considerable amount of work has been done on low-gravity fluid management, a comprehensive flight experiment is vital to provide engineering data and to validate proposed technology. The existing database is not adequate to provide system designs for missions in the 1990's and beyond. The COLD-SAT flight experiment will be a major milestone in cryogenic fluid technology for low-gravity applications and will provide the necessary technology springboard to begin flying two-phase liquid cryogen systems.

1.2 COLD-SAT SPECIFIC OBJECTIVES

In general, COLD-SAT will acquire the data to validate and refine new and existing models for low-gravity fluid behavior; specifically, the technical objectives of the COLD-SAT mission are:

Pressure Control: To evaluate the ability of passive thermodynamic vent systems and fluid mixing devices coupled with thermodynamic vent systems to control cryogenic tank pressure in low gravity.

Chilldown: To provide engineering data on the influence of fluid motion on chilldown heat transfer rates for a variety of fluid injection methods (i.e., spray patterns) and determine the quantities of cryogen required to chilldown a tank.

No Vent Fill: To investigate the filling of tanks with subcritical cryogens without venting by characterizing the effects of supply pressure and thermodynamic state, inflow rate, and injection technique on tank filling.

LAD Fill: To evaluate the filling and refilling of LADs in a low-gravity environment. To characterize the effects of heat, venting, and subcooling on LADs.

Pressurization: To investigate pressurization and pressurant requirements for cryogenic tankage in low gravity using condensible and noncondensible gases during liquid expulsion.

Low-g Drain: To characterize liquid settling times and to investigate fluid expulsion techniques for tanks without liquid acquisition devices under induced low-level acceleration.

Low-g Fill: To evaluate techniques for the vented fill of cryogenic tanks in low-gravity. To determine the mass and thermodynamic state of the fluid transferred.

LAD Expulsion Performance: To determine the efficiency of screen type LADs for the collection and outflow of fluid from orbital cryogen storage tanks in a microgravity environment.

Line Childown: To develop and characterize methods for childown of the transfer line (including lines, valves, fittings, and instrumentation) between source and receiving tanks which optimize the usage of liquid hydrogen.

Thermodynamic State Control: To evaluate the effectiveness of thermodynamic vent systems and pressurization systems for control of the thermodynamic state of the outflowing fluid.

Fluid dumping: To evaluate the effectiveness of low-gravity fluid dumping techniques which do not involve induced accelerations or liquid acquisition devices.

1.3 COLD-SAT CONCEPTUAL DESIGN SUMMARY

To meet these low-gravity fluid management technical objectives, the Ball Aerospace team has designed a satellite, COLD-SAT shown in Figure 1-1, capable of conducting the necessary experiments to provide both technology demonstration and data for model validation. Its launch weight is 2,963 kg (6,533 lb) which includes 234 kg (516 lb) of liquid hydrogen and 495 kg (1,091 lb) of hydrazine. Overall spacecraft reliability is 0.932, based upon a three month mission and component redundancy where required.

COLD-SAT is designed to be launched on a Delta II into a 926 km circular orbit at a 0.5 radian (28.7 degree) inclination. These orbit parameters were selected to keep the drag and gravity gradient induced accelerations below 10^{-6} g, stay below the Van Allen radiation belts and provide more than 500 years orbital life. The spacecraft flies with its long axis (Z) oriented along the orbit normal and its solar array always sun pointing (quasi-inertial). This attitude minimizes orbit perturbation due to orbit normal thrusting for experiment induced accelerations, eliminates coriolis accelerations and allows for fixed solar arrays.

The COLD-SAT spacecraft consists of seven major subsystems: (1) experiment, (2) structural, (3) thermal control, (4) propulsion, (5) attitude control, (6) tracking, telemetry and control, and (7) electric power.

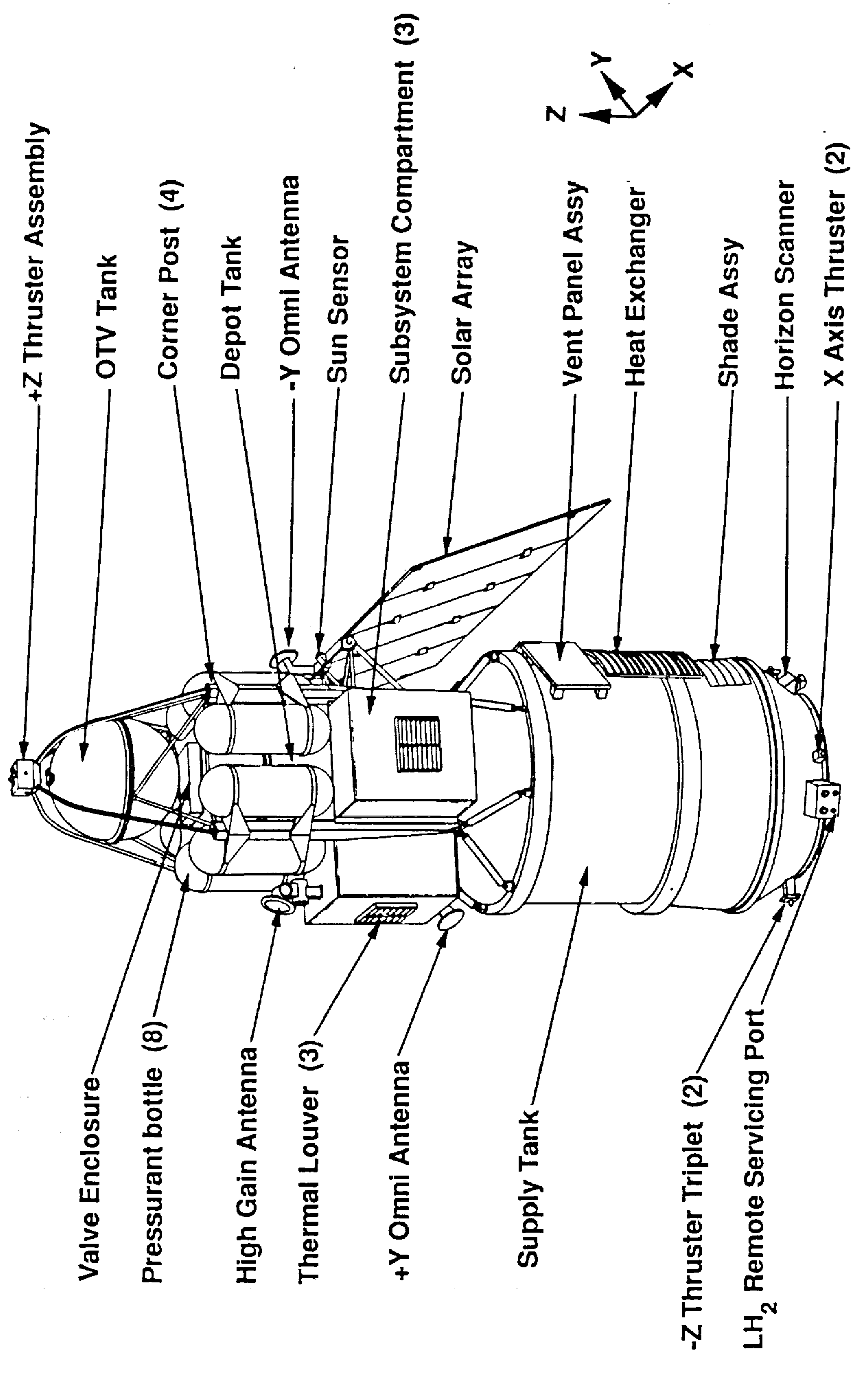

Figure 1-1. COLD-SAT external arrangement.

1.3.1 Experiment Subsystem

The experiment subsystem consists of a LH_2 supply dewar, depot and OTV receiver tanks, their interconnecting plumbing and valving, a tank pressurization system, instrumentation, and an experiment control processor (ECP).

The supply tank has a capacity of 3.498 m^3 (123.5 cu ft), is the principal tank for pressure control experimentation and serves as the primary spacecraft structure. It is a vacuum jacketed dewar, has an aluminum pressure vessel supported by fiberglass-epoxy straps and its insulation system consists of multi-layer insulation (MLI) with a vapor cooled shield. A thermal control shield around the pressure vessel provides a range of heat fluxes for the pressure control experiments.

The two receiver tanks are designed to simulate a space-based storage depot and an orbital transfer vehicle (OTV) fuel tank. The depot tank is used primarily for liquid acquisition device (LAD) testing, the OTV tank for chilldown and no-vent fill testing. Both tanks are insulated with 60 layers of MLI, but neither are vacuum jacketed. The depot contains a LAD and an axial spray; the OTV contains axial, radial, and tangential spray system.

To transfer the LH_2 between tanks, a LH_2 distribution subsystem was designed to accommodate all the experiments defined in the COLD-SAT experiment set. It has built-in redundancy such that no single valve or instrument failure will prevent successful completion of the mission. For safety, the system is a minimum of two fault tolerant. All valves are solenoid actuated and are magnetically latched, open or closed. Turbine flow meters and thermal mass flow meters are positioned to keep track of all fluid vented from the system and the amounts of fluid transferred between tanks. There are three vent paths to space: (1) a 105 kPa (15 psia) back pressure path for the low-g fill, continuous flow chill down and any settled fluid condition where venting may be desirable, (2) a 10.3 kPa (1.5 psia) path for the TVS vent flow and (3) an open path for tank evacuation during chilldown.

Each transfer operation requires pressurizing the tank from which the LH_2 is being transferred. The pressurization system provides both gaseous hydrogen (GH_2) and helium (GHe) for tank pressurization. The GH_2 is stored in seven aluminum graphite composite bottles at a storage pressure of 27.6 MPa (4000 psia). All seven bottles are manifolded together but are isolated by a check valve in the event one bottle develops a leak. The GHe pressurant is stored in a single bottle of identical design and is plumbed into the same distribution system as the GH_2 pressurant. Pressure is reduced from the supply bottles to either 138, 172, or 207 kPa (20, 25 or 30 psia) and routed to the appropriate tank. Redundant mass flow meters monitor and integrate the pressurant flow rate to provide the total pressurant used.

Instrumentation consists of temperature, pressure, flow rate, acceleration, and liquid/vapor detection. The type, number, location, and accuracy of sensors were driven by the experimental requirements. A summary of the different types of sensors and their locations is given in Table 1-1.

Table 1-1
COLD-SAT EXPERIMENT SET
INSTRUMENTATION SUMMARY

	TEMPERATURE SENSOR		LIQUID-VAPOR SENSOR	PRESSURE SENSOR	FLOW METER	ACCELER-OMETER (3-AXIS)
	DIODE	GRT				
Supply tank	78	24	38			
Depot tank	51					
OTV tank	35					
LAD, supply			8			
LAD, depot	40		20	8		
TVS	58			(5)*		
LH_2 subsystem	18		9	23	14**	
Press subsystem	8			12	2	
Miscellaneous	37					1
Totals	325	24	75	43	16	1

* Included in the LH_2 Subsystem
** 5 turbine meters, 9 velocimeters

The experiment control processor (ECP) serves three primary functions: software-driven experiment control, telemetry gathering, and signal conditioning. Experiments are controlled in real-time by an 80C86 microprocessor with preloaded software modules and uplinked tables of test parameters. Valve actuations, heater power, and mixer speed are commanded by the ECP using standard Remote Command and Telemetry (RCT) cards coupled to drive circuits developed for COLD-SAT. The ECP gathers and blocks telemetry data for approximately 500 sensors in the experiment subsystem. Signal conditioning and multiplexing electronics are contained in the ECP box which weighs approximately 25 kg and requires less than 25 W of operating power.

1.3.2 Structural Subsystem

The spacecraft structure uses the supply tank outer shell (vacuum jacket) as a primary load carrying member. The upper girth ring provides for attachment of the four main support posts which support the electrical subsystem boxes, receiver tanks, and the rest of the spacecraft upper structures. Thick honeycomb shear panels stabilize the support posts and provide areas for attaching the electrical subsystem boxes and the solar array support structure. A pair of pressurant bottles are hung from each corner post. A support ring is attached to the lower girth ring of the supply tank and tapers inward to transfer the launch loads to the supply tank. The propulsion subsystem is supported by a honeycomb shelf attached to the lower support ring.

1.3.3 Thermal Control Subsystem (TCS)

The TCS provides adequate temperature margins on all spacecraft components from launch through on-orbit operations and provides flexibility for low temperature protection during contingency operations. The TCS consists of thermal coatings, multi-layer insulation (MLI) blankets, louvers, shades with high output parafin thermal actuators and logic controlled heaters.

Thermal coatings selected maintain acceptable temperature ranges by balancing absorbed and radiated heat while MLI blankets are used to minimize spacecraft heat loss to space. Louvers are used to minimize temperature variation due to transient heat loads. Shades are used for passive control when allowable temperature ranges are narrow and the variation in the solar intensity over time is large. Heaters are used where the above methods are insufficient or for low temperature protection during contingency operations.

Due to the short length of the mission, the supply tank outer shell can be warm (300 K) without causing excessive boil off of the cryogen. This is advantageous because the additional heat conducted into the spacecraft reduces heater power for temperature sensitive elements near or on the supply tank. The supply tank exterior thermal control consists of a two zone thermal coating on the sun side of the tank while the rest of the tank exterior is insulated. The two zone coating minimizes temperature gradients across the exposed surface area of the tank while the tank vacuum jacket conducts heat around the circumference of the tank. Minimum temperature extremes can be mitigated by controlling heat losses through the anti-sun side MLI.

1.3.4 Propulsion Subsystem

The propulsion system consists of one 0.42 m (16.5 inch) nitrogen pressurant tank, six 0.56 m (22 inch) propellant tanks containing monopropellant hydrazine, and twenty-two thrusters with integral dual series control valves. This system is pressure regulated with valving for pressurant isolation and overpressure protection. It employs no propellant isolation valves and uses single string plumbing but does use dual series valving at each thruster to preclude leakage. The system uses four levels of thrusters (nominally 0.51 N, 0.90 N, 1.80 N, and 25.88 N). A nine-thruster cluster on top of the vehicle (+Z end) contains two thrusters for the low and intermediate induced-g acceleration levels and a single large thruster for the high acceleration level. The remaining four +Z thrusters operate in pairs with oppositely directed thrusters on the bottom of the vehicle to provide pure-couple limit cycle torques about the vehicle X and Y axes. The thirteen-thruster arrangement at the bottom of the vehicle includes five induced-g thrusters, two each for the intermediate and high levels of acceleration and a single small thruster for the lowest level of acceleration. The remaining eight ACS thrusters include four thrusters for torques about the X and Y axes and four for pure couple torques about the Z axis.

1.3.5 Attitude Control Subsystem (ACS)

The ACS provides spacecraft attitude control for the various environmental and experiment induced disturbances, and for the vehicle to achieve its preferred attitude from an arbitrary orientation. The ACS employs a redundant 3 axis Inertial Reference Unit (IRU) which is periodically updated by a horizon scanner and sun sensor to measure vehicle pointing and rate errors which are sent to the attitude control processor (ACP) for processing. Using this information the ACP generates timed pulse commands which are sent to the attitude control thrusters thus completing the "bang-bang" attitude control loop.

The use of an RCS for attitude control minimizes the cost of the ACS since an RCS is required to provide the induced gravity environment required by the experiments. This approach avoids the addition of costly system elements such as reaction wheels, torque rods, and the software to drive them which would otherwise be required to maintain vehicle attitude. Disturbances applied to the vehicle by thruster firings occur on the order of every 30 minutes with durations of less than 100 ms, thus providing negligible perturbations to the experiment. The fuel needed to maintain attitude control is a small fraction of the fuel needed to provide the induced gravity environment required by the experiment, hence it does not drive the size of the RCS.

1.3.6 Tracking Telemetry and Control (TT&C) Subsystem

The TT&C subsystem provides the bi-directional communications link between the ground and the spacecraft nominally via TDRSS and in a back-up mode via the Deep Space Network (DSN). The multiple access mode is used to communicate with TDRSS using a high gain antenna mounted on a double gimbal pointing system. Back-up communications with the DSN is accomplished by a number of omni antennas mounted on the vehicle which enable adequate communication regardless of vehicle attitude.

At the heart of the TT&C subsystem is the Telemetry and Command Processor (TCP) which furnishes the overall intelligence to the TT&C subsystem. It provides the telemetry formatting of downlink data, the initial interpretation of uplink data and commands, data/command routing and the intelligence to drive the high gain antenna pointing system such that communications with TDRSS is achieved and maintained. Additionally, the TCP is the master spacecraft processor controlling timing and communication through the local area network (LAN) with the other on-board processors which comprise the COLD-SAT distributed computing architecture.

Finally, the TT&C subsystem via the TCP monitors spacecraft health and will generate autonomous safing commands should it detect out of tolerance situations that threaten the spacecraft. Because of its importance to overall spacecraft performance, the TCP has been made redundant with an automatic switch over feature in case of a malfunction to ensure that the COLD-SAT spacecraft will have a high probability (> 0.92) of successfully performing its mission.

1.3.7 Electrical Power Subsystem (EPS)

Major components which comprise the electrical power subsystem are the solar array, the Power Control Unit (PCU), the two batteries, the power distribution components and the pyrotechnic drive circuits. The solar array is deployed to a fixed cant angle of 0.35 radians (20 degrees) from the major axis of the spacecraft. The PCU is designed such that no single point failure will render the system inoperative. It is modular in construction with 12 circuit boards. Two 20 A-hr capacity nickel-cadmium rechargeable batteries provide energy storage capability. The selected battery cells are qualified to a wider temperature range than that required for the COLD-SAT mission. The distribution system consists of the power wiring harness between all spacecraft and experiment electrical components and 3 relay boxes for bus and load switching and single point grounding. Pyrotechnic devices include redundant actuators for the solar array deployment and valve actuators for emergency closure of cryogenic fluid transfer lines.

The total on-orbit power required to operate the COLD-SAT spacecraft is 305 W, including a 10 percent contingency. Power requirements for the major subsystems are given in **Table 1-2**.

Table 1-2
COLD-SAT POWER REQUIREMENTS

SUBSYSTEM	POWER (WATTS)
Experiment	116.0
Propulsion	20.0
TT&C	53.9
ACS	42.3
Thermal Control	33.0
Electrical Power	12.1
Total	277.3
With 10% Contingency	305.0

Section 2

EXPERIMENT REQUIREMENTS

2.1 EXPERIMENT SET DEFINITION AND REQUIREMENTS

The COLD-SAT experiment set consists of four Class I experiments and six Class II experiments. Of the seven Class I experiments originally under consideration, passive and active pressure control were combined into a single pressure control experiment (1.0), quantity gauging (5.0), slosh dynamics (6.0), long-term thermal performance (7.0), and start baskets (15.0) were dropped from the experiment set. **Table 2-1** lists the current COLD-SAT experiment set.

Table 2-1
COLD-SAT EXPERIMENT SET

Experiment 1.0	Pressure Control	Class I
Experiment 2.0	Tank Chilldown	
Experiment 3.0	No-Vent Fill	
Experiment 4.0	LAD Fill	
Experiment 8.0	Pressurization	Class II
Experiment 9.0	Low-g Fill and Drain	
Experiment 10.0	LAD Expulsion Performance	
Experiment 11.0	Line Chilldown	
Experiment 12.0	Thermodynamic State Control	
Experiment 13.0	Fluid Dumping	

The four Class I experiments are enabling technology for space-based cryogenic applications, and the Class II experiments are considered enhancing technology. Therefore, the basic configuration of the COLD-SAT experiment was driven by requirements for the Class I experiments. Class II experiments were incorporated into the experiment concept but were not major drivers in the design.

The COLD-SAT experiment requires a minimum of three tanks to successfully accomplish the technological objectives of the Class I experiments. The supply tank is the primary tank for pressure control and must have reasonably good thermal performance. One receiver tank serves as the primary tank for LAD testing, and the other receiver tank for chilldown and no-vent fill experiments. Each receiver tank must have a means of pressure control sized to accommodate their parasitic heat leaks.

These top-level requirements and the individual COLD-SAT experiment requirements provide tank sizing, pressurization, valving, interconnecting lines, and vent system requirements. Tanks for the COLD-SAT mission were sized by combining fluid requirements for the entire experiment set with launch vehicle constraints. The supply tank size was restricted to approximately 3.4 cubic meters (120 cubic feet) for weight considerations with an L/D between 1.2 and 1.6 to provide similarity to future on-orbit supply tankers. Experimentally, the only sizing requirements for the Depot and OTV tanks are a variation in L/D and M/V ratio and a minimum length requirement for the depot tank to test a full-size spray nozzle. Since pressure control requires supply tank tests at 95, 75, and 50 percent fill levels, the OTV and depot receivers need to be sized to provide these fill levels in the supply tank after each receiver is filled.

Hardware requirements are summarized by experiment in the following sections. Some requirements are "top" level, but most are derived from top-level requirements. For example, top-level mixing requirements are for regions I and IV mixing (see Reference 2.1); derived requirements are mixer jet size and flow rate. The detailed experiment requirements are given in the Experiment Requirements Document (Reference 2.2) and are not repeated here.

2.1.1 Pressure Control Requirements

The primary tank for pressure control experiments is the supply tank. Pressure control tests are divided into stratification/destratification testing and TVS testing. During stratification, the tank is locked up and allowed to self pressurize from its heat input; destratification is accomplished by turning on an in-tank mixer. For TVS testing; two thermodynamic vent systems are used (with and without the mixer) to reduce tank pressure by venting after the stratification/destratification test is completed. To characterize low-g mixing, mixing tests are conducted at two different flow rates. During these tests, the influence of heat flux, g-level, mixer flow rates, and tank fill level will be investigated. Pressure control experiments are also conducted in the OTV and depot receivers during the 75 percent and 50 percent full tests in the supply tank. A total of 42 pressure control tests will be run at the beginning of the mission, requiring approximately 3 weeks to complete.

The variation in pressure control parameters establishes most key design requirements for the supply tank. Supply tank background heat flux must be less than 0.3 W/m^2 (0.1 $Btu/hr\text{-}ft^2$), it must be equipped with a vapor-cooled shield (VCS) and a liquid-acquisition device (LAD) to provide vapor-free liquid to the TVS systems. A thermal control shield (TCS) is required to provide a uniform heat flux of 0.95 W/m^2 (0.3 $Btu/hr\text{-}ft^2$) and 1.89 W/m^2 (0.6 $Btu/hr\text{-}ft^2$) over the entire tank surface.

Mixing experiments require a mixer located at one end of the supply tank which is capable of providing Region I and Region IV mixing (Reference 2.1). For a 3.4 m^3 (120 ft^3) supply tank, the calculated flow rates are 0.17 and 0.65 ms/min (0.6 and 2.3 ft^3/min). The mixer must have a minimum of two speeds, but a continuously variable speed mixer would benefit the experiment by providing additional flexibility. Jet diameters between 6.4 and 7.6 cm (2.5 and 3.0 inches) are required to provide similarity (D/d ratio) with previous zero-g mixing experiments. Mixer power dissipation within the supply tank should be less than 10 percent of the total tank heat load at minimum induced heat flux (0.95 W/m^2) to avoid excess heating of liquid hydrogen. The mixer must be close-coupled to a compact heat exchanger on the suction side to provide a sub-cooled liquid jet and the heat exchanger inlet should be open to the tank so that liquid, vapor, or a two-phase mixture can flow through the heat exchanger and mixer.

2.1.1.1 TVS Requirements for Pressure Control

Requirements for the TVS systems on COLD-SAT are given in **Table 2-2.** The high-flow (active) supply tank system is sized to reduce tank pressure by 8.6 kPa/hr (1.25 psia/hr) at the maximum heat input from the TCS; the remaining systems are sized based on background heat leaks.

Table 2-2

PRESSURE CONTROL TVS REQUIREMENTS

TVS	HIGH FLOW SUPPLY TANK	LOW FLOW SUPPLY TANK	DEPOT TANK	OTV TANK
Thermal Capacity W (Btu/hr)	171 (585)	7.3 (25*)	13 (44*)	8.8 (30*)
Pressure Reduction, kPa/hr (psi/hr)	-8.6 (-1.25)	-0.2 (-0.03)	-1.9 (-0.28)	-2.3 (-0.33)
Mass Flow Rate kg/min (lb/hr)	24.6 (3.25)	1.1 (0.14)	1.8 (0.24)	1.3 (0.17)
J-T Pressure Drop kPa (psid)	90 (13)	90 (13)	90 (13)	90 (13)
H-X Pressure Drop kPa (psid)	<14 (<2.0)	<3.4 (<0.5)	<3.4 (<0.5)	<3.4 (<0.5)
H-X Location	Internal	Internal	Internal and External	External
H-X Support	Thermally isolated	Thermally isolated	Thermal Contact LAD and Wall	Wall-Mounted

*Capacity requirement Q_{TVS}= 2x(background tank heat input)

CS.649

Mass flow rates were determined assuming that only heat of vaporization of the liquid remaining after J-T expansion was available for cooling (approximately 93 percent of the total flow). Mass flow rates were calculated from:

$$\dot{m} = \frac{\dot{Q}}{\theta\eta}$$

where: $\dot{Q}$ = heat removal rate in watts

θ = specific heat input (449.8 J/g for hydrogen, approximtely the latent heat of vaporization)

η = the fraction of liquid produced downstream of the J-T (0.93)

Pressure drop across the J-T was assumed to be 90 kPa (13 psid).

2.1.2 Tank Chilldown

The chilldown experiment uses a charge-hold-vent cycle in which cryogen is introduced into the warm evacuated tank through a sprayer, held until the system is nearly in thermal equilibrium, and then vented to near-vacuum (approximately 7 kPa; 1 psia). The OTV tank is "bare" of most flow restrictions at the wall (such as baffles and LADs) so that effective flow/swirl patterns can be established inside the tank. Some chilldown tests will also be conducted in the depot tank which should therefore have a different M/V and L/D than the OTV tank.

Since the primary tank for chilldown is the OTV receiver, its requirements are derived from similarity and scaling to full-size OTV hydrogen tanks. It is equipped with radial, tangential and axial sprays and extensive instrumentation to measure fluid and wall temperatures. Each spray system needs to deliver 90.7 kg/hr (200 lb/hr) of LH_2 with a velocity in excess of 9.1 m/s (30 ft/s) and a pressure drop less than 27.6 kPa (4.0 psid). Radial and tangential sprays in the OTV receiver tank should each provide full surface coverage, the axial spray should provide over 50 percent coverage.

The combination of flow rates and spray systems will be systematically explored to develop an optimum chilldown procedure for the OTV tank. Cryogen requirements and thermodynamic efficiency are particularly important for scaling to other systems. A total of 16 tank chill down tests will be conducted in the first 2 months of the COLD-SAT mission.

2.1.3 No-Vent Fill

No-vent fill tests are conducted in both the depot and the OTV tanks. In each test, the tank is prechilled to a "target" temperature using the chilldown procedure. After the last venting is completed for chilldown, the vent is closed and liquid hydrogen is introduced into the tank. As the tank fills, incoming liquid condenses vapor in the tank and the fill level rises. Tank filling to greater than 95 percent is the goal of the no-vent fill tests. Extensive instrumentation in both the depot and OTV tanks will provide data on vapor temperatures, liquid temperatures, wall temperatures, and inflowing fluid temperatures during the experiment.

Tests with variable flow rates 22.7-90.7 kg/hr (50-200 lb/hr) and different fluid injection methods will be conducted in each tank. To simulate the effect of a full-scale spray nozzle in a full-size depot, the depot receiver must be at least 1.5 meters (5 feet) long with an L/D ≈2. The effects of settling the fluid over or away from the spray nozzle will also be assessed. Some tests will evaluate re-filling a partially full tank to determine the effectiveness of vapor bubble collapse. A total of 16 no vent fill tests will be conducted during the first 2 months of the COLD-SAT mission.

2.1.4 LAD Fill

A four channel LAD similar to an orbital depot system LAD will be tested in the depot receiver tank. Two approaches to LAD fill will be evaluated: 1) TVS subcooling of the LAD to condense the entrained vapor, and 2) direct venting of the LAD to space. The LAD will be filled through its screens using fluid in the tank or directly using the transfer line from the supply tank. The success or failure of each LAD fill will be determined by a short test to verify vapor-free liquid at flow shortly after filling is completed.

The depot-tank LAD TVS must be sized for a flow rate of 104 g/hr (0.23 lb/hr), with a J-T pressure drop of 90 kPa (13 psid), to provide cooling necessary for vapor collapse in the channels. The depot tank LAD instrumentation requirements are temperature, pressure, and liquid/vapor sensors for determining the progress and effectiveness of the fill processes. For direct venting, liquid detection is required on the LAD vent line to determine when the vent floods with liquid. A total of 7 LAD tests will be conducted during the first two months of the COLD-SAT mission.

2.1.5 Class II Experiments

2.1.5.1 Pressurization

Pressurization operations must be conducted prior to every liquid transfer. The pressurization experiment is designed to measure the amount of pressurant required under varying conditions of g-level, tank fill level, and diffuser/fluid positioning. The mass flow rate of pressurant will be controlled by the pressure regulator set-point pressure, actual flow rate measurements will be made with flowmeters. The supply tank requires diffusers at each end of the tank, and each receiver tank requires one diffuser. Velocity reduction of approximately 500:1 should minimize gas impingement on the liquid surface. A total of 9 pressurization tests will be conducted in the first 2 months of the COLD-SAT mission.

Quantity of usable pressurant was calculated from the initial density at 27.57 MPa (4,000 psia) and 300 K (540 R) and the "empty" density at 3.45 MPa (500 psia) and 300 K (540 R). Seven hydrogen bottles and one helium bottle, each with a volume of 0.127 m^3 (4.5 ft^3) are required for the COLD-SAT experiment set. Pressurant should be stored at a minimum temperature of 255 K (460 R) to optimize pressurant consumption. A regulated delivery system is required so that tank pressurization to 138, 172, and 207 kPa (20, 25, and 30 psia) is available.

2.1.5.2 Settled Transfer

The OTV tank is the primary tank for settled outflow and vented-fill testing. Since it has no LAD, all outflows from the OTV tank must be conducted as settled outflows. The effects of g-level, flow rate, pressurant, and the use of a screened outflow baffle for vapor pullthrough suppression will be evaluated during the testing. Vapor detection in the outflow line is required for experiment control. Pressurant will be supplied to the OTV tank to maintain the differential required to return the fluid to the supply tank (bypassing the LAD). A 15-W distributed heater is required to vaporize and vent the liquid residuals at completion of each experiment.

In the vented fill process, the OTV tank will be filled directly to determine if filling can be conducted without venting liquid hydrogen. The liquid inflow diffuser must be designed to minimize interfacial disruption during fill (velocity reduction of approximately 250:1). If sensors in the pressure-regulated vent line detect the presence of liquid, the inflow will be shut off. If the tank is nearly full, after a brief hold period, filling will be resumed to determine the maximum fill level obtainable under low-g conditions. If the tank is nearly empty when liquid is detected, a hold period will be scheduled to allow tank cooling prior to resuming the fill. An adjustment in inflow rate may also be required if liquid cryogen is detected in the vent too frequently. Temperature sensors in the OTV tank will be located to provide data on the fill level and liquid-vapor interface configuration within the tank during fill.

Four tests will be conducted to optimize the low-g outflow process during the early part of the COLD-SAT mission, and three vented-fill tests will be conducted after the Class I experiments are completed.

2.1.5.3 LAD Expulsion

Multiple expulsions will be performed with the depot tank LAD at background accelerations and at stressful g-loads (10^{-3} g), opposite to the direction of LH_2 outflow, to determine expulsion efficiencies and LAD screen breakdown characteristics. The effect of flow rate on expulsion efficiency will also be determined. GHe and GH_2 pressurant will be used for expulsion to investigate LAD sensitivity to different pressurant gases. Near the end of the COLD-SAT experimentation, a final expulsion test will be conducted in the supply tank to determine the efficiency of the supply tank LAD. Six expulsion tests will be conducted in conjunction with the Class I LAD fill tests during the COLD-SAT mission.

The depot LAD is instrumented with temperature, pressure, and liquid-vapor sensors to monitor the expulsion process. The sensor locations have been chosen to detect the presence of vapor in the channel as well as temperature distributions. In addition, the temperature sensors in the tank will provide data on the tank fill level, liquid temperature and the liquid positioning within the tank.

The depot LAD must be designed so that breakdown will occur at 10^{-3} g and a flow rate of 200-225 kg/hr (450-500 lb/hr), which requires a coarse screen (10 x 52 mesh) LAD. A 25-W distributed heater will be required to vaporize and vent the liquid residuals.

2.1.5.4 Line Cooldown

Each time fluid is transferred from one tank to another, the interconnecting transfer lines and their components must be chilled down. The COLD-SAT transfer lines are instrumented with temperature and pressure sensors to follow the progress of line chilldown. Two different line lengths will be chilled, depending on which receiver tank is being filled, to provide two discrete data points.

2.1.5.5 Fluid Subcooling

Outflow from the supply tank for a chilldown, no-vent fill, or a LAD fill can be sub-cooled using a high-efficiency heat exchanger located on the supply tank outflow line. A portion of the outflowing liquid is cooled by isenthalpically expanding it through a J-T valve, then back through a subcooler heat exchanger to provide 1.7 K (3 R) subcooling of the primary outflow. The heat exchanger and outflow line are instrumented with temperature and pressure sensors so the thermodynamic state of both the subcooler vent flow and the liquid outflow can be determined. The primary experiment parameters are the subcooler mass flow rates (measured by liquid and vapor flow meters), pressure drop across the expansion valve and liquid outflow rate.

2.1.5.6 Fluid Dumping

Fluid dumping is usually required for an abort scenario but little or no data is available for on-orbit dumping of cryogenic fluids. The COLD-SAT dump experiment will be conducted from the OTV tank after all other tests have been completed and will require opening a dump valve, then determining liquid residuals upon completion of the experiment. Formation of solid hydrogen below 1 psia must be avoided by combining a backpressure regulator with the ejector nozzle. Estimated dumping rates are 45 kg/hr (100 lb/hr) and the dumping cannot rely on either fluid settling (induced thrust) or LADs for fluid collection.

2.2 INTEGRATED EXPERIMENT SET AND REQUIREMENTS

Many requirements for the COLD-SAT mission had to be determined from a combined (integrated) experiment set in which all tests were listed in chronological order. This section describes requirements which were derived from the integrated experiment set and not determined by any one experiment.

2.2.1 Venting Requirements

The COLD-SAT experiment set was analyzed to determine all venting requirements for the mission and the results are summarized in **Table 2-3.**

Table 2-3

SUMMARY OF COLD-SAT VENTING REQUIREMENTS

TYPE OF VENT	EXPERIMENT	BACKPRESSURE		MASS FLOWRATE		DURATION
		(kPa)	(psia)	(g/min)	(lb/hr)	
Supply TVS (active)	P.C./ventdown	10	1.5	24.6	3.25	0.5 - 3.0 hr.
Supply TVS (passive)	Standby	10	1.5	1.1	0.14	10 - 200 hr.
Supply Subcooler	Subcooled NVF	10	1.5	30	4	1 - 2 hr.
Depot Primary	Chilldown	0	0	91-302	12 - 40	1 min.
Depot Primary/LAD	LAD Fill	103	15	0-7.6	0 - 1	1 - 10 min.
Depot TVS (P.C.)	Standby	10	1.5	1.8	0.24	10 - 200 hr.
Depot TVS (LAD)	LAD Fill	10	1.5	1.8	0.24	1 - 2 hr.
OTV Primary	Chilldown	0	0	42 - 91	5.6-12	1 min.
OTV Primary	Vented Fill	103	15	7.6 - 15	1-2	1 hr.
OTV TVS	Standby	10	1.5	1.3	0.17	10 - 200 hr
OTV HX (TVS bypass)	Cont. Cooldown	103	15	7.6 - 30	1 - 4	1 hr.
OTV 2-phase	Fluid Dumping	10	1.5	380 - 760	50-100	1 hr.
Transfer Line	Line Cooldown	103	15	30 - 60	4-8	1 hr.

CS.517

Each pressure control TVS system was sized to reduce or maintain tank operating pressure; the mass flow rate through each vent system was determined thermodynamically (see Section 3.1.1). The depot tank TVS is also used to subcool fluid in the LAD channels for bubble collapse during and after a LAD fill experiment.

Primary vents are used for chilldown, vented fill and for direct vent LAD experiments (Depot tank only). Chilldown venting requirements were determined assuming that 3 vent cycles were allowed to vent the charge mass in 30-second periods. For vented fill, experiment times of 30 minutes to 1 hour require venting rates of 0.5 to 1 kg/hr (1 to 2 lb/hr).

When the transfer lines are cooled to LH_2 temperature, all fluid used for cooling is vented overboard. It is currently estimated that 0.5 to 1 kg (1 to 2 lb) of liquid will be required for cooling, and line cooldown should take 10 to 30 minutes. Required venting rates are therefore 1.8 to 5.4 kg/hr (4 to 12 lb/hr).

The supply tank subcooler provides subcooling of the supply tank outflow for no-vent fill experiments. The required subcooler vent flowrate is 1.8 kg/hr (4 lb/hr) for subcooling 45 kg/hr (100 lb/hr) of LH_2 outflow.

The fluid dumping experiment is designed to dump a 75 percent full OTV tank, 17.7 kg, (39 lb) in approximately 30 minutes. These high rates of 2-phase fluid require a dedicated vent system with flow capability of 22 to 45 kg/hr (50 to 100 lb/hr).

2.2.2 Instrumentation

The measurement requirements in **Table 2-4** represent the most stringent requirements for COLD-SAT. A detailed table of measurement requirements and locations for each experiment is found in the Experiment Requirements Document (ERD).

Table 2-4

EXPERIMENT MEASUREMENT REQUIREMENTS

MEASUREMENT	RANGE	ACCURACY (1)	JUSTIFICATION
Temperature			
Liquid and Vapor	17-340K(30-600 R)	±0.5K (±1.0 R)	NVF analysis
Liquid or Sat'd Vapor	17-28K (30-50 R)	±0.1K (±0.2 R)	NVF and TVS analysis
Liquid or Sat'd Vapor	17-28K (30-50 R)	±0.05K (±0.1 R)	PC analysis
Pressure - absolute	0-340 kPa (0-50 psia)	±3.4 kPa (0.5 psia)	Chilldown analysis
Presure - (delta - P)	0-340 kPa (0-50 psia)	±1.4 kPa (0.2 psia)	PC analysis
Mass Flowrate			
Liquid	9-91 kg/hr (20-200 lb/hr)	±2%	Chilldown analysis
Vapor	0.4-3.8 g/min (0.05 - 0.5 lb/hr)	±2%	TVS analysis
	7.6 - 76 g/min (1-10 lb/hr)	±2%	TVS analysis
	38 - 380 g/min (5-50 lb/hr)	±2%	Chilldown analysis
Liquid/Vapor Detection	100% Liquid 100% Vapor	<5 sec response <5 sec response	LAD and Low-g transfer analysis
Acceleration	10^{-6} g to 10^{-3} g	±10% over any decade	Bond number Error <10% @ $2x10^{-5}$ g

CS.659

(1) Overall system accuracy

Three different temperature measurement requirements were identified for the experiment subsystem. The controlling accuracy requirements were determined by error analysis of the no-vent fill and pressure control experiments, most sensors requiring 0.1 K accuracy for liquid temperature measurements.

Most pressure measurements must be accurate to 3.4 kPa (0.5 psia). Repeated measurements on the same gauge for delta-P determination require greater accuracy (1.4 kPa; 0.2 psia).

Mass flowrate measurements are driven by chilldown and TVS experiments.

Liquid/vapor detection is used as a criterion for experiment control; therefore, their accuracy requirement is expressed as time for detection based on allowable liquid mass losses in flow lines.

Acceleration measurement requirements are driven by Bond number accuracy requirements at the lowest settling thrust ($2x10^{-6}$ g). To keep the Bond number error less than 10 percent, the g-level measurement must be accurate to 1 micro-g.

For each experiment all relevant sensors must be monitored at appropriate measurement frequencies which are determined by the rates of change of the processes taking place. For example, during stratification testing, a sensor sweep is required every 60 seconds, but during no-vent fill, a sweep is required every second. Most experiments require a sensor sweep at the 1 Hz rate.

Required data rates were determined from telemetry formats for each experiment and the required measurement frequency. Experiments requiring liquid transfer to the depot tank use the most sensors and therefore have the highest data rate (3700 bps). Four orbits of data storage at the high rate would require 80 megabytes of RAM.

Several critical top-level requirements were identified for the data acquisition and control (DAC) system which must collect data from all the sensors during experiments. Specifically, the DAC system must scan the sensor set once per second and must be sized to handle 3700 bps of data. In addition to telemetry functions, the DAC system must provide complete software control capabilities, store and use uplinked tables or parameter lists, re-load software in orbit (backup only), and gather telemetry for processing and downlink. Details of the DAC system can be found in Section 4.8.2.

2.2.3 Mission Model

All experiments and other fluid management operations were integrated into a spreadsheet model called CSATTIME. CSATTIME is a spreadsheet program developed using Symphony 2.10, and currently occupies 325 kbytes of disk space. It is a time-sequenced mathematical description of the COLD-SAT experiment mission profile, and includes sections for the supply tank and each receiver tank. Inputs include the key parameters for the experiments such as heat flux, g-level, fluid transfer and vent rates, the test sequence, and other data such as fluid properties, tank sizes, and test duration. Output is available as a summary or as a function of mission time. Numerous graphs are built into the program, including tank pressures and fill levels as a function of mission time.

The overall layout of the spreadsheet is shown in **Figure 2-1**. The top 60 lines (approximately) are used for global parameters, the next 235 lines are used for the mission timeline. Each entry corresponds to a specific operation or experiment, and is assigned a duration time and numerous other characteristics. The model tracks pressures, fill level, heat input, g-level and g-sec of thrust, mass vented, mass transferred, receiver tank fill levels heat fluxes, and some fluid properties (tank density and the energy derivative, phi). A bookkeeping section of the spreadsheet (columns Q - W) tracks fluid losses in 5 different categories.

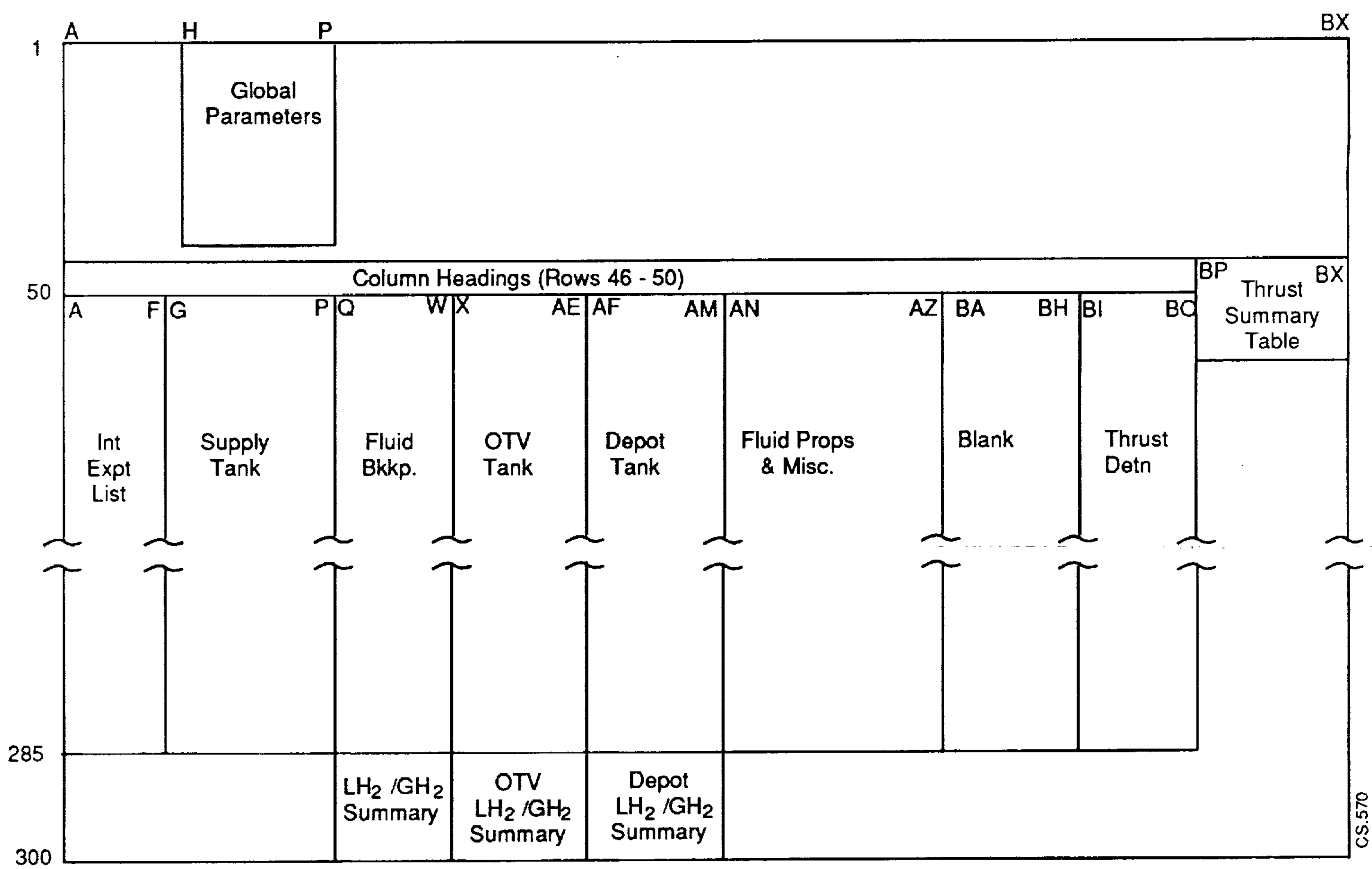

Figure 2-1. Layout of CSATTIME spreadsheet model.

Fluid accounting summaries are located at the bottom of the mission timeline; these summaries are also used for graphing fluid allocation. CSATTIME tracks all liquid hydrogen uses and losses, including transfers to receiver tanks, chilldown losses, residual losses, standby boiloff losses, and pressure reduction losses. Gaseous hydrogen requirements are also included for both the supply and receiver tanks. In the case of the supply tank, pressurant is included in the fluid balance.

In addition to hydrogen requirements, CSATTIME tracks total thrust required in g-sec. A separate location in the spreadsheet (columns BI - BO) is used to sort experiments by g-level and determine how much thrust is required at each level for each experiment.

2.2.3.1 Experiment Scheduling

Figure 2-2 shows the overall experiment timeline for the COLD-SAT mission. The markers on the schedule denote the beginning and ending times for each experiment, but in most cases, the experiments are not running continuously during the time period shown. The Class I experiments are completed in 59 days, and the Class II experiments are finished in approximately 64 days. Pressure control experiments are completed in the first 21 days of the mission and then the remaining fluid transfer experiments are conducted.

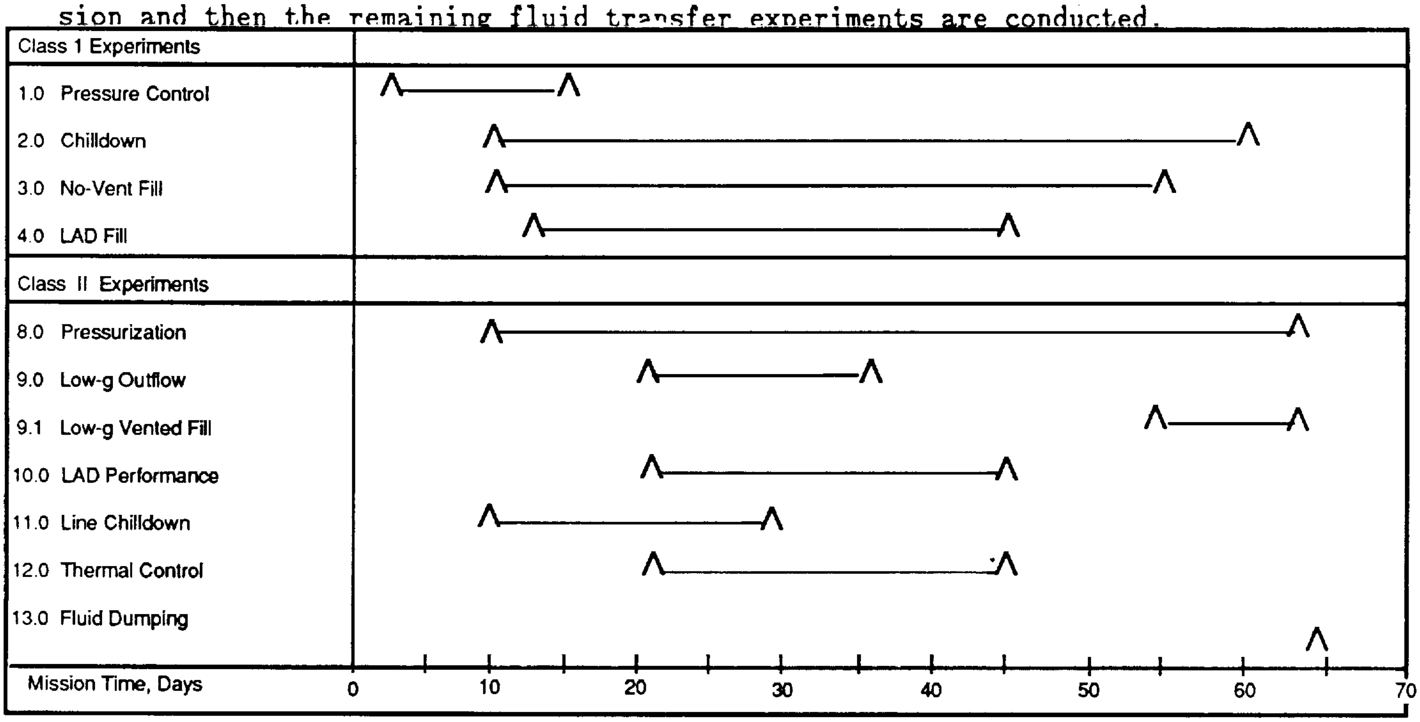

Figure 2-2. COLD-SAT experiment timeline.

CS.262

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After pressure control testing has been completed, a nominal 7-day operating sequence is used for the remaining COLD-SAT experiments. The cycle begins on day 1 with warm, dry receiver tanks and proceeds through supply tank pressurization, two chilldowns, and two no-vent fills. Day 1 has the most activity and requires approximately 6 hours of operations to complete these tasks. On day 2, fluid from the receivers is returned to the supply tank, which requires 4 to 6 hours of operations. Days 3 and 4 repeat days 1 and 2 respectively. The remaining 3 days of the seven-day period are reserved for data analysis or operational repeats if any problems were encountered.

This 7-day cycle begins on day 21 and continues until day 65 when all the class I and II transfer experiments are completed. The schedule provides a reasonable work-week for all science and mission personnel but is success-oriented with respect to available time for recovering from unanticipated developments. Although days 1 and 3 are aggressively scheduled with activities, it is essential to conduct both chilldowns and no-vent fills using the same pressurization cycle to conserve hydrogen. Extending the schedule reduces fluid available for experiments by approximately 1 kg/day (2 lb/day).

Figure 2-3 shows tank fill levels for each COLD-SAT tank during the entire mission. Dips in the supply tank fill level correspond to spikes in the receiver tank fill levels, and they occur each time fluid is transferred. Receiver tank spikes are labelled with the appropriate no-vent fill or LAD fill test number (see Reference 2.2).

In order to have a 7-day cyclic operating schedule after day 21, there are two occasions when fluid remains in a receiver tank for a 3-day hold period. In the depot tank, fluid remains in the tank from day 24 to 27, and in the OTV tank from day 38 to 41.

The final spike on the OTV graph represents fluid dumping, in which fluid is held in that tank for 48 hours prior to dumping.

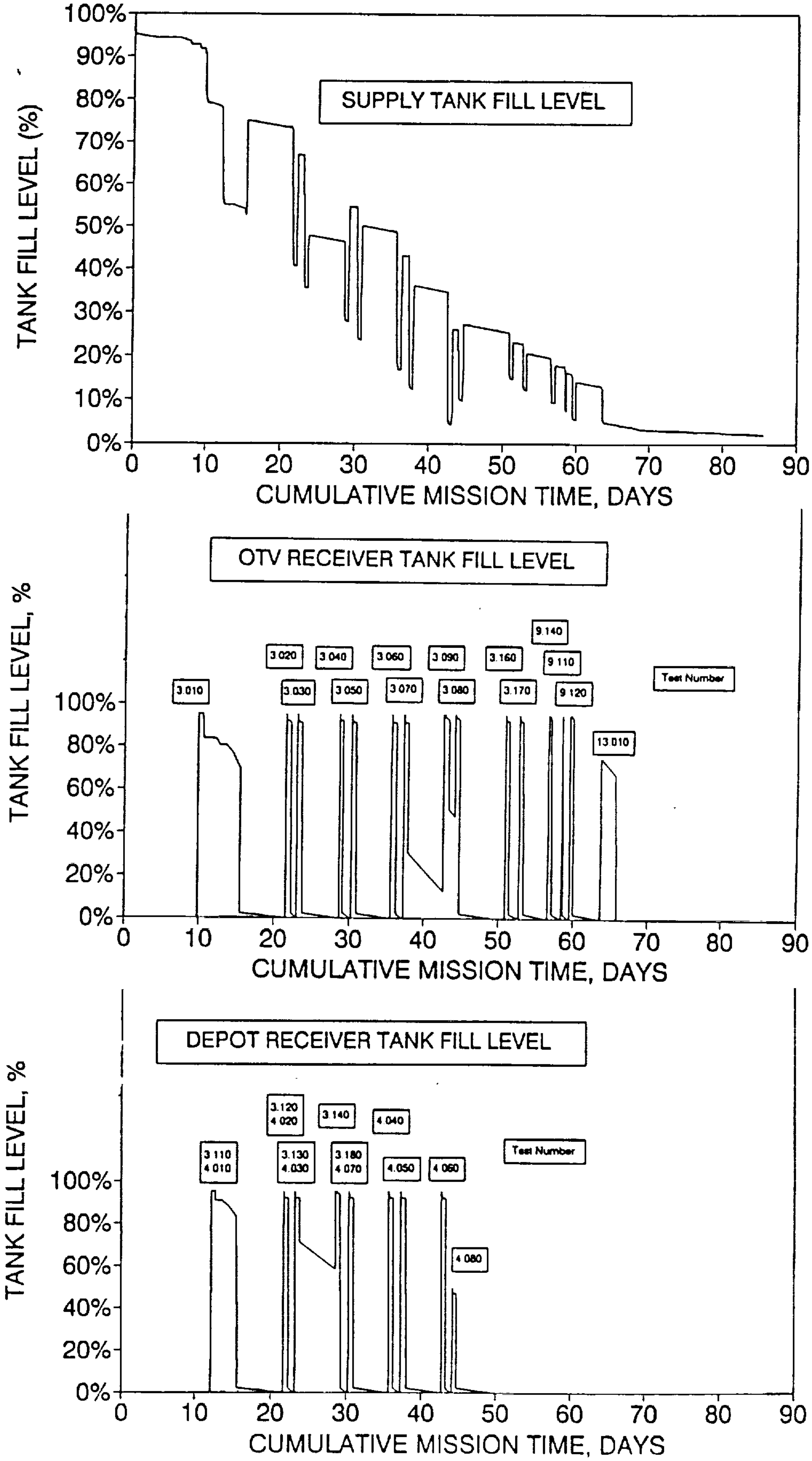

Figure 2-3. Tank fill levels for COLD-SAT experiment set.

2.2.3.2 Consumable Requirements

2.2.3.2.1 Power Requirements

In order to properly establish the experiment subsystem power requirements, the mission model CSATTIME was modified to incorporate sensor status for each experiment. By assigning the appropriate power consumption values to the sensors, tank heater on/off times, component operation and considering the thrust level requirements for each experiment, a detailed power profile was generated.

The average experiment power consumption is 81.6 watts, and the peak period is 114 to 116 W. The power "floor" consists of 18 W for the data acquisition system, 25 W for the vapor flowmeters, 23 W for the pressure sensors (they remain on for housekeeping purposes), and 8 W for the accelerometer. The remaining instruments, heaters, mixer, and thruster valves are switched on and off according to the particular experiment which is being conducted.

The peak period occurs during pressure control where the TCS is fully energized, the mixer is on, and thrusters are on.

Figure 2-4 shows how the maximum experiment power was determined. The appropriate power level for each sensor or group of sensors is shown in the graphs below the total experiment power plot. Note that all pressure gauges, the accelerometer, and the ECP are on 100 percent of the time. Each of the design power loads is listed in **Table 2-5** below.

Table 2-5

EXPERIMENT SUBSYSTEM DESIGN POWER

ECP	18.2 W
Accelerometer	8.4 W
Pressure Gauges	23.4 W
Mixer	1.8 W
Flowmeters	25.0 W
TCS	29.7 W
Thruster valves	9.5 W
Total Power	116.0 W

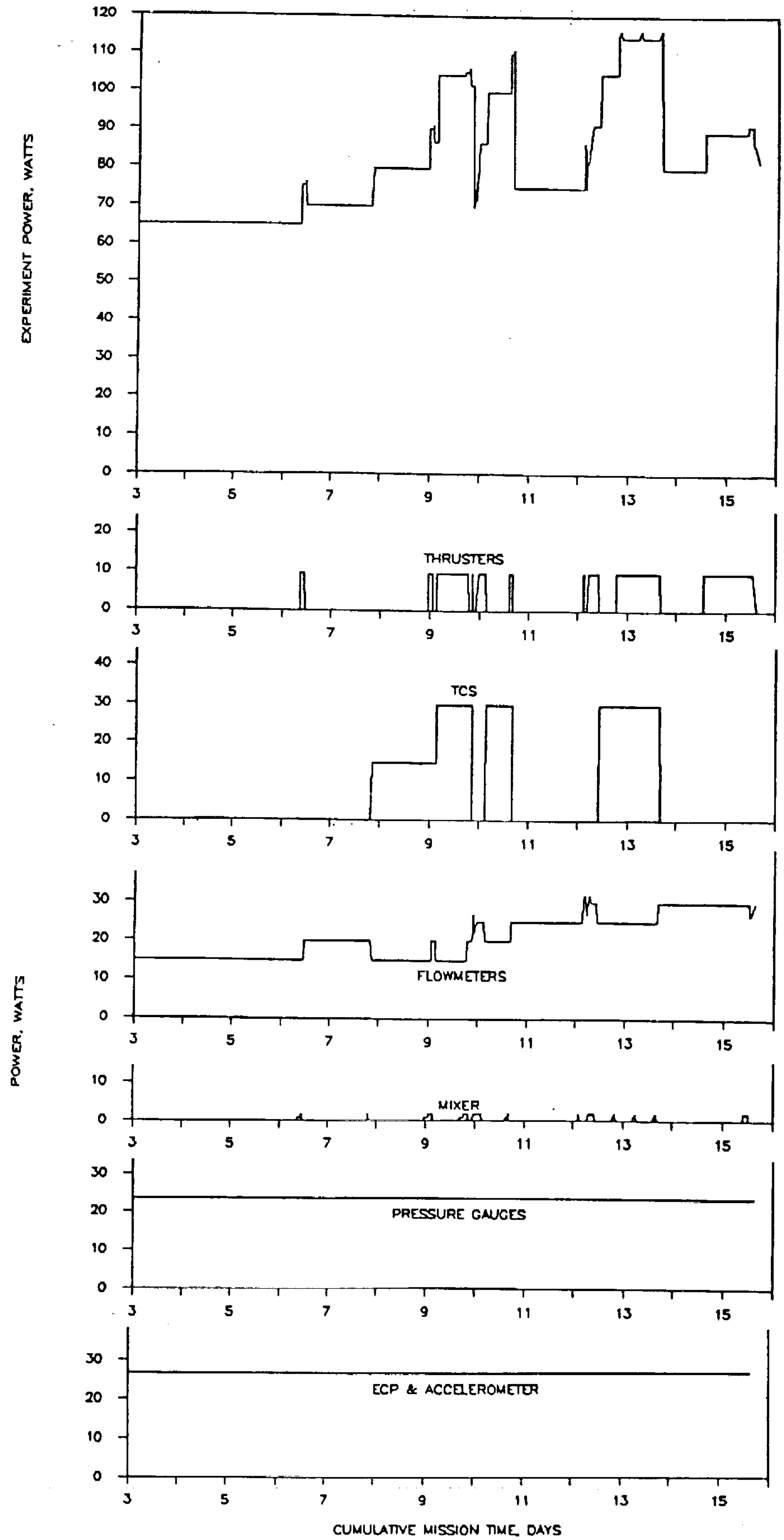

Figure 2-4. Detailed power loads for maximum experiment power.

The valves require approximately 50 W to actuate; however, the actuation time is only 300 milliseconds. Therefore, even 10 valves actuated sequentially add a power pulse of 50 W for 3 seconds. Since the mission time scale is in days, the small pulses for valve actuation do not contribute to the peak experiment power load (except for bus sizing to handle the momentary 2 A current draw).

2.2.3.2.2 Liquid Hydrogen Requirements

CSATTIME tracks and records the usage of LH_2 for different experiments (Reference **Figure 2-3**). The pie chart in **Figure 2-5** is a graphical representation of the overall LH_2 budget. **Table 2-6** lists liquid hydrogen allocations which are plotted in **Figure 2-5**. The supply tank contains 123.5 cubic feet of hydrogen (less 5 percent for ullage space at launch). It is assumed that the tank has a 2 percent residual due to LAD performance and that approximately 17 lb. of gaseous hydrogen pressurant are introduced into the supply tank during the mission. Each portion of the fluid usage pie chart is defined as:

Pressure Control: Fluid used in the pressure control experiments.

Fill/Drain Residual: Fluid left in the receiver tanks after transferring fluid back to the supply tank for reuse.

Childown: Fluid used for chilling tanks and lines.

Pressure Reduction: Fluid lost due to venting the supply tank from high pressure (172 kPa; 25 psia) to normal pressure (103 kPa; 15 psia) after a pressurization, and for venting receiver tanks from 124 kPa (18 psia) to 103 kPa (15 psia) after a no-vent fill. It is a large piece of the pie because almost 4 kg (9 pounds) of liquid are vented for every pound of pressurant introduced to the supply tank.

Standby Boiloff: Represents fluid lost by venting during hold periods in the supply and receiver tanks due to scheduling constraints.

Table 2-6

LIQUID HYDROGEN ALLOCATION FOR COLD-SAT

Category	LH_2 (kg)	LH_2 (lb)
Pressure Control	30.1	66.4
Fill/Drain Residuals	13.7	30.2
Chilldown	40.4	89.1
Fluid Reserves	3.3	7.2
Pressure Reduction	56.9	125.5
Fluid Dumping	17.7	39.0
LH_2 Subcooling	5.0	11.0
Standby Boiloff	69.9	154.2
LAD Residual	4.9	10.9
Total LH_2	241.9	533.5

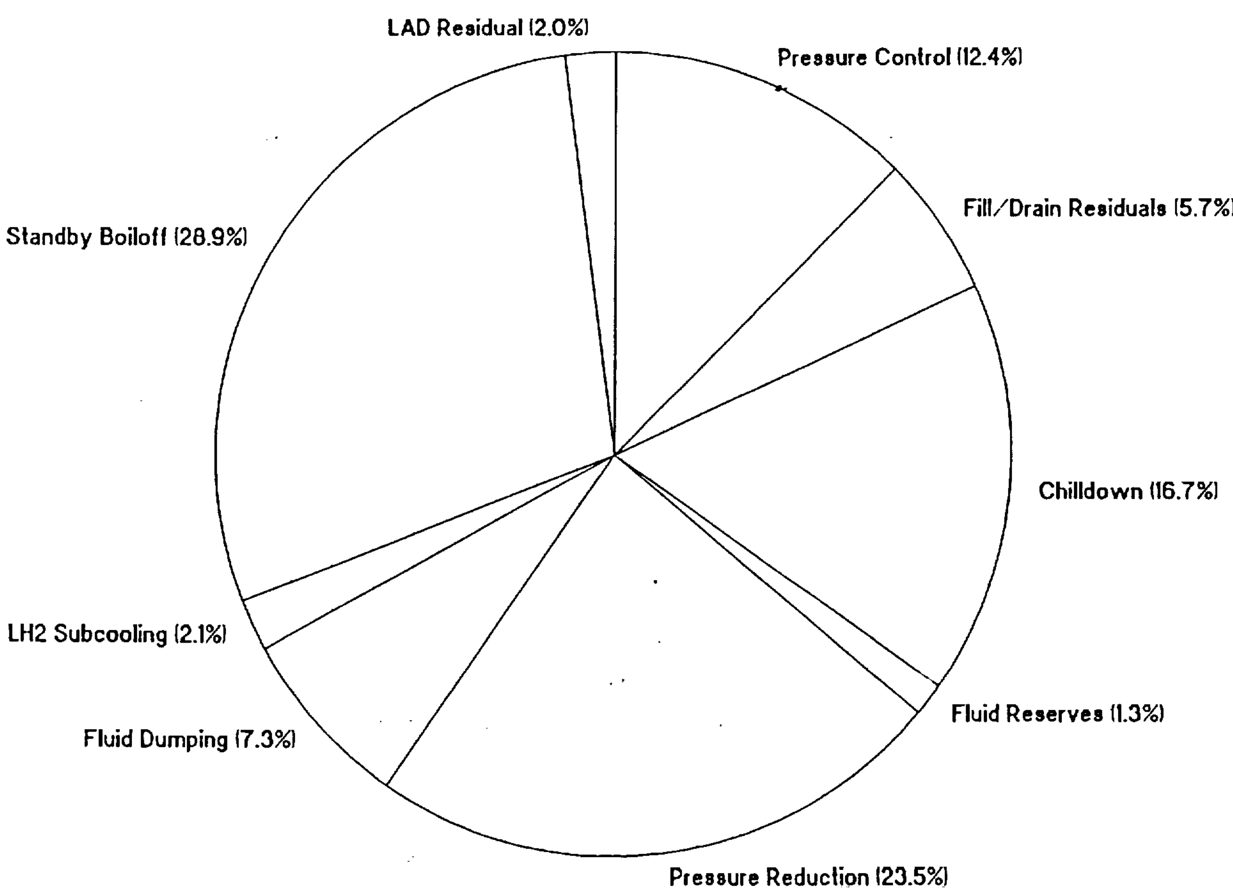

Figure 2-5. Integrated experiment set fluid requirements.

LH_2 Subcooling: That portion of the fluid vented which is used to subcool the supply tank outflow.

Fluid Dumping: Fluid vented overboard during the liquid dump experiment.

LAD Residuals: Fluid left in the supply tank at the end of the mission.

2.2.3.2.3 Pressurant Requirements

Requirements for GH_2 pressurant are listed in **Table 2-7**. Pressurant consumption for supply tank pressurization and outflow were calculated assuming 50 percent approach to equilibrium for ullage compression and 85 percent approach to equilibrium for outflow. Ullage compression in the supply tank represents the largest single usage of pressurant. Pressurant consumption for receiver tank expulsions (low-g drain and LAD expulsion) were calculated using Moore's correlation (Reference 2.3).

Table 2-7

GH_2 PRESSURANT REQUIREMENTS (BY EXPERIMENT)

EXPERIMENT	GH_2 REQUIRED*		GH_2 ALLOCATED	
	kg	lb	kg	lb
Pressure Control	0	0	0	0
Chilldown	--	--	--	--
No-Vent Fill	6.9	15.2	10.3	22.8
LAD Fill	--	--	--	--
Low-g Fill, Drain	1.2	2.6	1.8	3.9
LAD Expulsion*	1.2	2.6	1.8	3.9
Line Chilldown	0	0	0	0
Thermal State Control	0	0	0	0
Fluid Dumping	0.5	1.0	0.6	1.4
Totals	9.8	21.4	14.5	32.0

-- Pressurant for these tests is included under no-vent fill
*4 lb of GHe is also reserved for LAD expulsion and settled outflow

2.2.3.2.4 Thrust Requirements

Four thrust levels to provide constant accelerations along the long axis of the tank are required for COLD-SAT. The lowest level ($2x10^{-5}$ g) provides fluid settling over the mixer in the supply tank at a Bond Number of 4.6. The two intermediate levels ($7x10^{-5}$ and $1.4x10^{-4}$ g) provide fluid settling at either end of the receiver tanks at Bond Numbers of 4.5 and 9 for fluid transfer experiments. The highest level ($1x10^{-3}$ g) is for settling fluid away from the depot LAD outlet for LAD breakdown testing.

Thrust requirements for the COLD-SAT experiment are listed in **Table 2-8**, total thrust time in hours is also given for each thrust level. The thrust in g-seconds was calculated by multiplying the g-level by the total test duration time in seconds. The thruster propellant requirements are directly proportional to the total g-seconds for a fixed spacecraft weight.

Table 2-8

THRUST REQUIREMENTS SUMMARY (GROUPED BY g-LEVEL)

EXPERIMENT		THRUSTER REQUIREMENTS (g-sec)				
		$1x10^{-3}$g	$1.4x10^{-4}$g	$7x10^{-5}$g	$2x10^{-5}$g	TOTAL
1.0	Pressure Control	0.00	0.00	2.53	4.12	6.66
2.0	Chilldown	0.00	0.25	0.38	0.00	0.63
3.0	No-vent Fill	0.00	0.00	1.93	0.00	1.93
4.0	LAD Fill	0.00	0.53	0.00	0.00	0.53
	Quantity Gauging	0.00	0.00	0.13	0.07	0.20
8.0	Pressurization	0.00	0.00	0.33	0.00	0.33
9.0	Vented Fill/Drain	0.00	4.54	3.02	0.00	7.56
10.0	LAD Expulsion	3.60	0.00	1.76	0.00	5.36
	Pressure Reduction	0.00	0.00	0.00	2.57	2.57
	TOTALS (g-sec)	3.60	5.32	10.08	6.76	25.77
	TOTALS (hours)	1.0	10.6	40.0	93.9	

• All experiments accommodated with substantial propellant margin

CS.266

2.3 REFERENCES

2.1 Aydelott, J.C., "Modeling of Space Vehicle Propellant Mixing," NASA TP-2107, January, 1983.

2.2 COLD-SAT Experiment Requirements Document, Revision 3.1, March 23, 1990.

2.3 R.W. Moore et al, "Gas-Pressurized Transfer of Liquid Hydrogen", Adv. Cry. Eng., Vol. 5, 450-459, 1960.

Section 3
EXPERIMENT OPERATIONS

Experiment control is a complex and detail-oriented task. During this study, all primary and secondary control criteria, control sensors and their appropriate limiting values, and valve actuations were defined for each test listed in the detailed experiment test matrices (see the Experiment Requirements Document, Reference 3.1). Control algorithms for "non-experiment" operations (such as supply tank pressure reduction following expulsion tests) were not defined, although most operations are similar to existing experiments.

Each COLD-SAT experiment is controlled by the Experiment Control Processor (ECP) without ground intervention or monitoring. Initially, control algorithms were developed for all experiments. After the experiment behavior was predicted from analytical models, control criteria based on physical properties were established for determining end-points (pressure rise of 5 psia, for example). These control criteria were then incorporated into flow charts which described the control algorithm for a particular experiment. After the flowcharts were formulated and evaluated, transducers were selected within the experiment subsystem which would provide the required information for decision making, and specific action steps (valve, heater, and mixer actuations) were defined for each test.

The experiment control concept was developed using existing instrumentation and passive liquid hydrogen flow control orifices. All valves are actuated sequentially by the ECP, but the exact order of valve actuations within a particular test was not defined. A time-out criteria is required for all tests so that the ECP will always drop out of the control loop if an unanticipated condition arises (not all flowcharts reflect this time-out). Experiment operating procedures are described in greater detail in the ERD.

3.1 CLASS I EXPERIMENT OPERATIONS

3.1.1 Pressure Control

Thermal stratification (self-pressurization), jet-induced fluid mixing (destratification), and thermodynamic vent system (TVS) experiments will be performed in the supply tank at different acceleration, heat flux, and tank liquid fill levels.

Stratification tests begin by closing all vent valves and activating the thermal control shield (TCS) to the desired heat flux level. The tank is "locked up," and pressure will begin to rise at a rate which depends on fill level, heat flux, g-level, and level of liquid stratification. Stratification tests continue until the tank pressure rises 35 kPa (5 psia) or the allocated time elapses.

After the stratification test is completed, the fluid is settled in the tank using induced-g thrusters. The liquid jet mixer is then turned on to provide Region I mixing [approximately 0.017 m^3/min (0.6 ft^3/min)]. The mixing experiment is terminated after a preset time elapses, the pressure-time curve flattens out, or the fluid temperature starts to rise.

After the Region I mixing test is complete, the mixer flow rate is increased to provide Region IV mixing [approximately 0.065 m^3/min (2.3 ft^3/min)]. Region IV mixing continues until $dP/dt = 0$ or a preset time elapses. A flowchart describing the stratification-mixing tests is given in **Figure 3-1**. Primary decision criteria are highlighted by dark shading, and secondary criteria with lighter shading. **Table 3-1** lists control criteria and required action for each stratification and mixing test in the pressure control test matrix. Note that no valve actuations are required for these tests.

The purpose of TVS testing is to evaluate the effectiveness of passive (no mixing) and active (mixing) systems for maintaining or reducing tank pressure. The passive system is sized to maintain constant pressure at background heat flux; the active system is sized to reduce tank pressure by 8.6 kPa/hr (1.25 psia/hr) at maximum heat flux.

Passive TVS tests will be conducted at 95 percent and 50 percent fill levels. Each test begins with a well-mixed (destratified) tank. The passive TVS system is activated by opening the J-T valve to the 10 kPa (1.5 psia) vent. Testing continues until the pressure drops the required amount (7 kPa; 1 psia) or until the test time elapses.

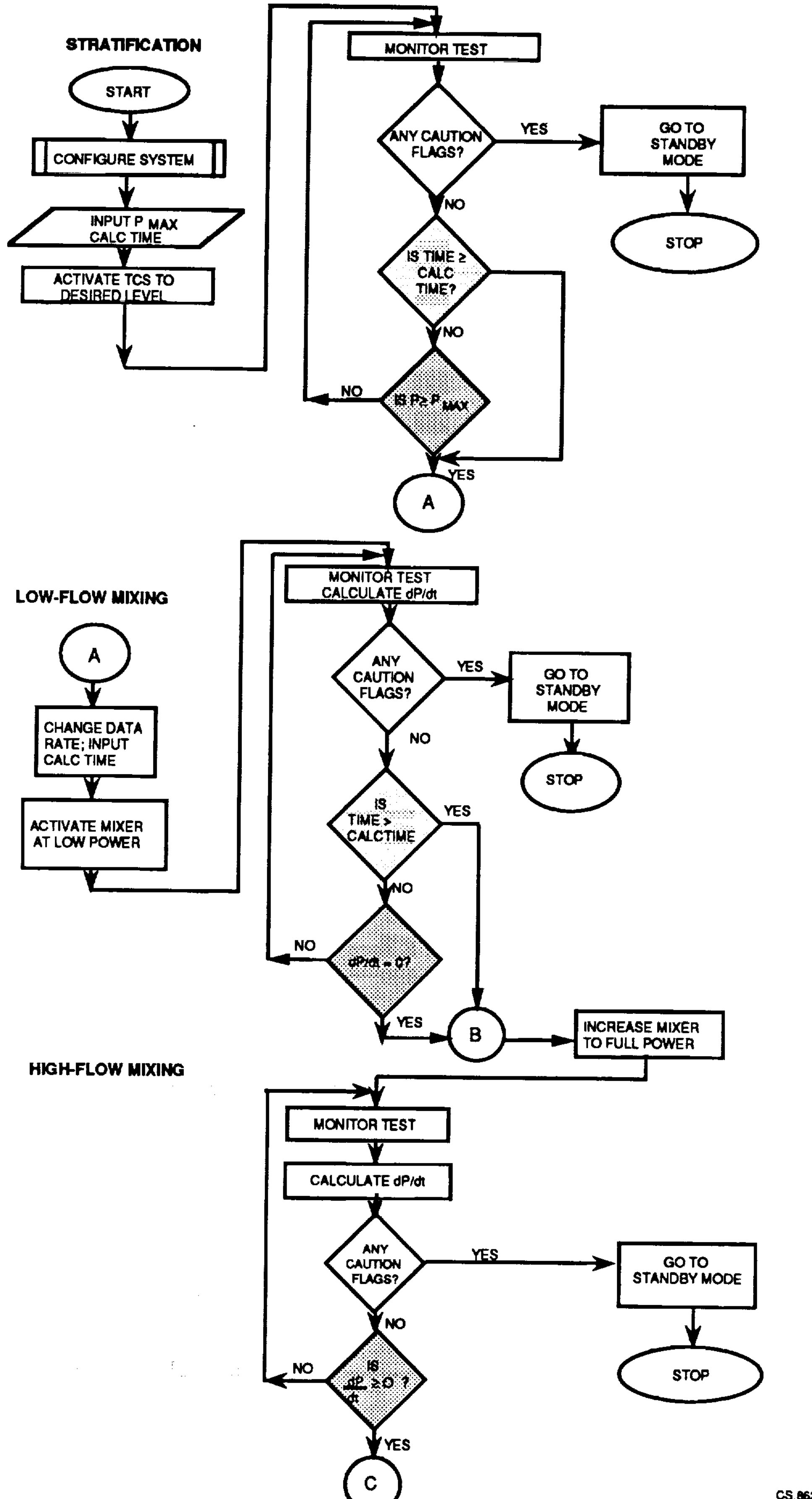

Figure 3-1. Flowchart for stratification and mixing experiments.

Table 3-1

OPERATIONS TABLE FOR STRATIFICATION AND MIXING TESTS

DESCR	TEST #	MIXER TCS	MIXER CFM	g-LEVEL	VALVES OPEN	CRITERIA #1	CRITERIA #2	ACTION
STRAT	1.010	OFF	OFF	BKGD	NONE	DP=5 PSI	TIME	MIXER ON LOW; THRUST @ 2.0E-5
DESTRAT-LO	1.011	OFF	0.6	2.0E-05	NONE	dP/dt =0	TIME	MIXER ON HIGH
DESTRAT-HI	1.012	OFF	2.4	2.0E-05	NONE	dP/dt =0	TIME	MIXER OFF; TVS 1.020; THRUST OFF
STRAT	1.040	0.3	OFF	BKGD	NONE	DP=5 PSI	TIME	MIXER ON LOW
DESTRAT-LO	1.041	0.3	0.6	BKGD	NONE	dP/dt =0	TIME	MIXER ON HIGH
DESTRAT-HI	1.042	0.3	2.4	BKGD	NONE	dP/dt =0	TIME	ACT TVS 1.050
STRAT	1.060	0.6	OFF	BKGD	NONE	DP=5 PSI	TIME	MIXER ON LOW; THRUST @ 2.0E-5
DESTRAT-LO	1.061	0.6	0.6	2.0E-05	NONE	dP/dt =0	TIME	MIXER ON HIGH
DESTRAT-HI	1.062	0.6	2.4	2.0E-05	NONE	dP/dt =0	TIME	ACT TVS 1.070
STRAT	1.080	0.6	OFF	BKGD	NONE	DP=5 PSI	TIME	MIXER ON LOW; THRUST @ 2.0E-5
DESTRAT-LO	1.081	0.6	0.6	2.0E-05	NONE	dP/dt =0	TIME	MIXER ON HIGH
DESTRAT-HI	1.082	0.6	2.4	2.0E-05	NONE	dP/dt =0	TIME	MIXER, TCS, THRUST OFF; TVS 1.090
STRAT	1.110	0.6	OFF	BKGD	NONE	DP=5 PSI	TIME	MIXER ON LOW; THRUST @ 2.0E-5
DESTRAT-LO	1.111	0.6	0.6	2.0E-05	NONE	dP/dt =0	TIME	MIXER ON HIGH
DESTRAT-HI	1.112	0.6	2.4	2.0E-05	NONE	dP/dt =0	TIME	MIXER OFF
STRAT	1.120	0.6	OFF	2.0E-05	NONE	DP=5 PSI	TIME	MIXER ON LOW
DESTRAT-LO	1.121	0.6	0.6	2.0E-05	NONE	dP/dt =0	TIME	MIXER ON HIGH
DESTRAT-HI	1.122	0.6	2.4	2.0E-05	NONE	dP/dt =0	TIME	MIXER OFF; THRUST @ 7.0E-5
STRAT	1.130	0.6	OFF	7.0E-05	NONE	DP=5 PSI	TIME	MIXER ON LOW
DESTRAT-LO	1.131	0.6	0.6	7.0E-05	NONE	dP/dt =0	TIME	MIXER ON HIGH
DESTRAT-HI	1.132	0.6	2.4	7.0E-05	NONE	dP/dt =0	TIME	MIXER OFF

CONTROL SENSORS FOR ALL TESTS ARE P19 AND P20

Active TVS tests are designed to lower tank pressure to a pre-determined level (usually 103 kPa; 15 psia). Tests begin by opening the high-flow TVS valve to the 10 kPa (1.5 psia) vent. The tank mixer is then activated at the desired power level, and testing continues until tank pressure drops to the set point or until time elapses. Flowcharts describing passive and active TVS testing are shown in **Figure 3-2**. Control of passive and active TVS tests will be based on pressure readings from the supply tank pressure transducers. **Table 3-2** lists required valve actuations and control criteria for TVS testing.

3.1.2 Chilldown

The primary tank for chilldown is the OTV receiver. This tank is equipped with radial, tangential, and axial sprays and extensive temperature sensing to measure fluid and wall temperatures. The chilldown experiment uses a charge-hold-vent cycle in which cryogen is introduced into the warm tank through a sprayer, held until the system is nearly in equilibrium, and then vented to space.

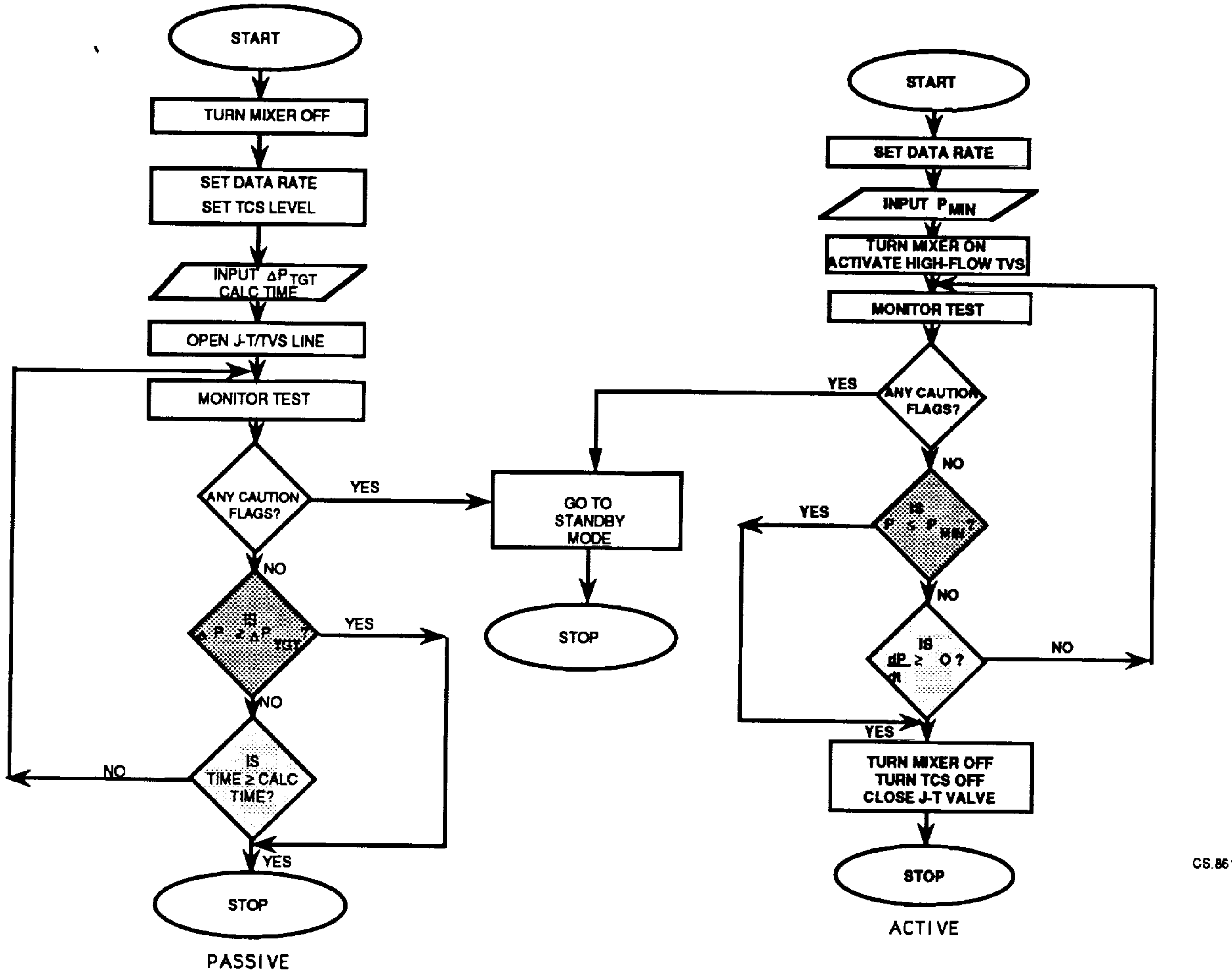

Figure 3-2. Flowcharts for passive and active TVS experiments.

Table 3-2

OPERATIONS TABLE FOR PASSIVE AND ACTIVE TVS TESTS

DESCR	TEST #	MIXER TCS	MIXER CFM	g-LEVEL	VALVES OPEN	CRITERIA #1	CRITERIA #2	ACTION
PASSIVE	1.020	OFF	OFF	BKGD	5,38	TIME	DP=-1 PSI	CLOSE 5; OPEN 4; MIXER ON
ACTIVE	1.030	OFF	2.4	BKGD	4,38	P=15 PSI	dP/dt=0	CLOSE 4; MIXER OFF
ACTIVE	1.050	0.3	2.4	BKGD	4,38	P=15 PSI	dP/dt=0	CLOSE 4; MIXER OFF
ACTIVE	1.070	0.6	2.4	BKGD	4,38	P=15 PSI	dP/dt=0	CLOSE 4; MIXER OFF
PASSIVE	1.090	OFF	OFF	BKGD	5,38	TIME	DP=-1 PSI	CLOSE 5; OPEN 4; MIXER ON; THRUST @ 2E-5
ACTIVE	1.100	OFF	2.4	2.0E-05	4,38	P=15 PSI	dP/dt=0	CLOSE 4; MIXER OFF; THRUST OFF
PASSIVE	1.140	OFF	OFF	BKGD	5,38	TIME	DP=-1 PSI	THRUST @ 2.0E-5
PASSIVE	1.150	OFF	OFF	2.0E-05	5,38	TIME	DP=-1 PSI	CLOSE 5; OPEN 4; MIXER ON
ACTIVE	1.160	OFF	2.4	2.0E-05	4,38	P=15 PSI	dP/dt=0	CLOSE 4; MIXER OFF; THRUST OFF

CONTROL SENSORS FOR ALL TESTS ARE P19 AND P20

With the tank evacuated and the vents closed, a small metered amount of liquid is injected through the spray nozzles. The liquid vaporizes in part due to flashing, and eventually, in total due to heat transfer with the tank walls or with its own vapor. The duration of the charge period is typically less than one minute, and fluid metering can be accomplished by integrating the liquid flow meter signal or by using timed valve actuations (timing determined from ground testing).

Following liquid inflow, the resultant cold vapor is held in the tank for several minutes to allow adequate time for heat transfer from the warmer tank walls and internal hardware to the colder vapor. Venting will begin when the heat transfer rate becomes small (as measured by the rate of change of fluid or wall temperature), or when the vapor temperature is within 95 percent of the wall temperature. The charge and hold processes are depicted in the flow chart in **Figure 3-3**.

Since each test uses different combinations of valve actuations during the charge cycle, a detailed operations table (**Table 3-3**) was developed to describe the required actuations and control criteria for the charge cycle. Valve operations for continuous flow cooldown are also listed in **Table 3-3**. Venting occurs in stages down to specified target pressures [currently 140 and 70 kPa (20 and 10 psia)] to allow further heat transfer to occur between the isentropically cooled vapor and the tank wall. Each vent cycle begins when heat transfer between vapor and wall becomes small. Ground testing will be required to determine the appropriate heat transfer limits for experiment control. Venting continues until the tank pressure is less than 7 kPa (1 psia). Each tank chilldown requires multiple charge-hold-vent cycles to reach the desired target temperature. The venting process is diagrammed in the flow chart in **Figure 3-4**. Note that there are several branch points depending on which venting cycle is occurring and whether or not the tank has reached the prechill temperature. Valve actuations for hold and vent cycles (not listed in **Table 3-3**) are more straightforward because each tank has only one primary vent system.

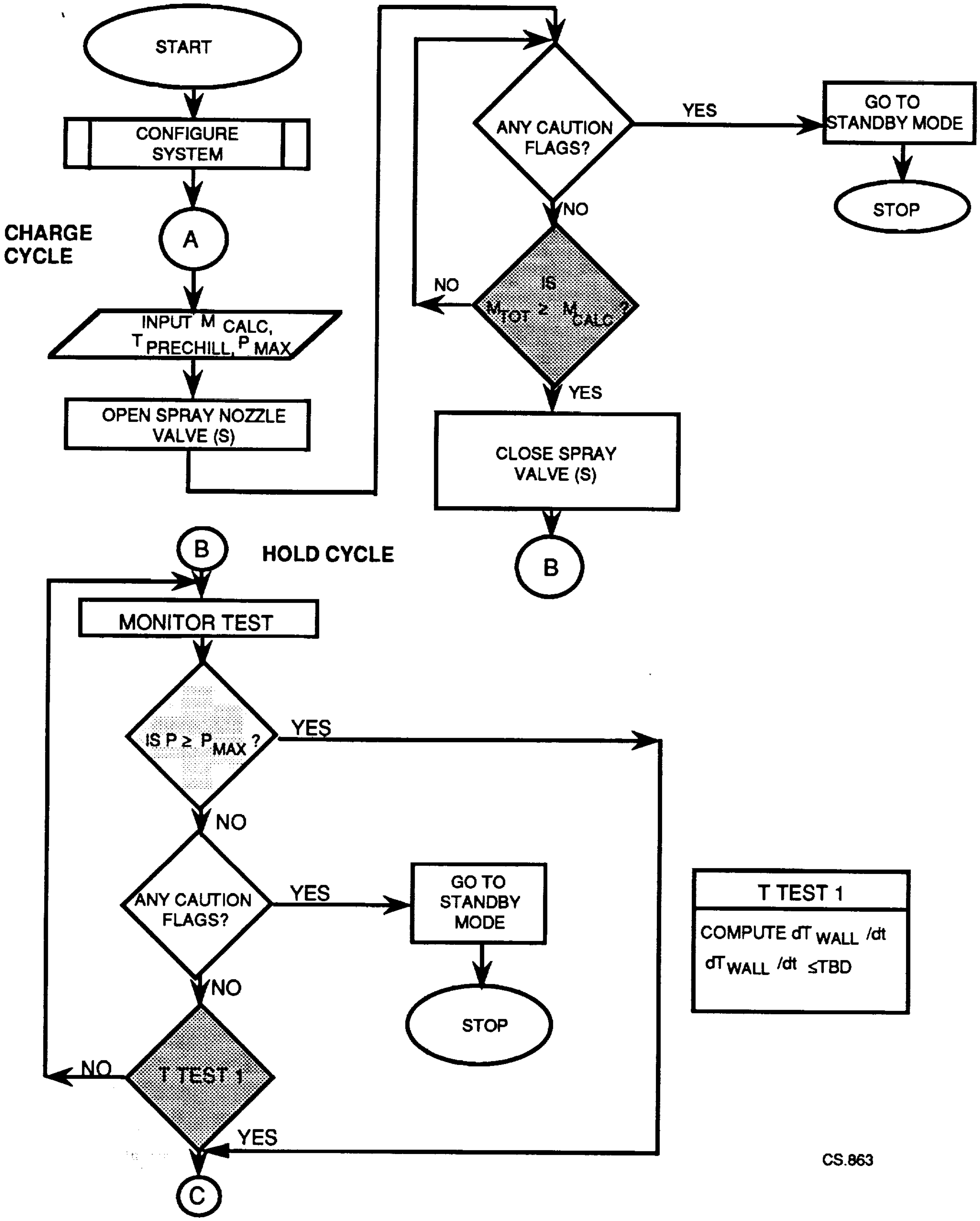

Figure 3-3. Flowchart for chilldown experiment charge and hold cycles.

Table 3-3

OPERATIONS TABLE FOR CHILLDOWN CHARGE CYCLES

DESCR	TANK	TEST#	VALVES OPEN	SENSORS	CNTRL CRITERIA	ACTION
TANGENTIAL	OTV	2.010	19,33,24	LFM1	M=MCHG	CLOSE 24
RADIAL	OTV	2.020	19,33,27	LFM1,4	M=MCHG	CLOSE 27
RADIAL	OTV	2.030	19,32,27	LFM1,4	M=MCHG	CLOSE 27
TANG/RAD	OTV	2.040	19,33,24,27	LFM1,4	M=MCHG	CLOSE 33,24,27
TANG/RAD	OTV	2.050	19,33,24,26	LFM1,4	M=MCHG	CLOSE 33,24,26
TANG/RAD	OTV	2.060	19,33,24,26,27	LFM1,4	M=MCHG	CLOSE 33,24,26,27
T/R SEQ	OTV	2.070	19,33,24;19,33,27	LFM1,4	M=MCHG1,2	CLOSE 24 OR 27
G-LEVEL1	OTV	2.090	19,33,27	LFM1,4	M=MCHG	CLOSE 27
G-LEVEL2	OTV	2.100	19,33,27	LFM1,4	M=MCHG	CLOSE 27
BKGD	OTV	2.110	19,33,27	LFM1,4	M=MCHG	CLOSE 27
LAD ONLY	DEPOT	2.120	19,33,34	LFM1,3	M=MCHG	CLOSE 34
LAD/AXIAL	DEPOT	2.130	19,33,34,22	LFM1,2,3	M=MCHG	CLOSE 33,34,22
AXIAL/BKGD	DEPOT	2.140	19,33,22	LFM1,2	M=MCHG	CLOSE 22
AXIAL/GLVL	DEPOT	2.150	19,33,22	LFM1,2	M=MCHG	CLOSE 22
CONT FLOW	OTV	2.160	19,31,24,15,16,13,37	TBD	TW≤TPRCHL	CLOSE 24,15
CONT FLOW	OTV	2.170	19,31,24,26,15,16,13,37	TBD	TW≤TPRCHL	CLOSE 31,24,26,15

NOTES: 1. Each charge cycle requires that the supply tank be pressurized to 20 psia by opening valves 51, 53, and 58. These valves should be closed after each cycle to conserve pressurant.
2. Test 2.080 was dropped from the matrix due to schedule limitations.

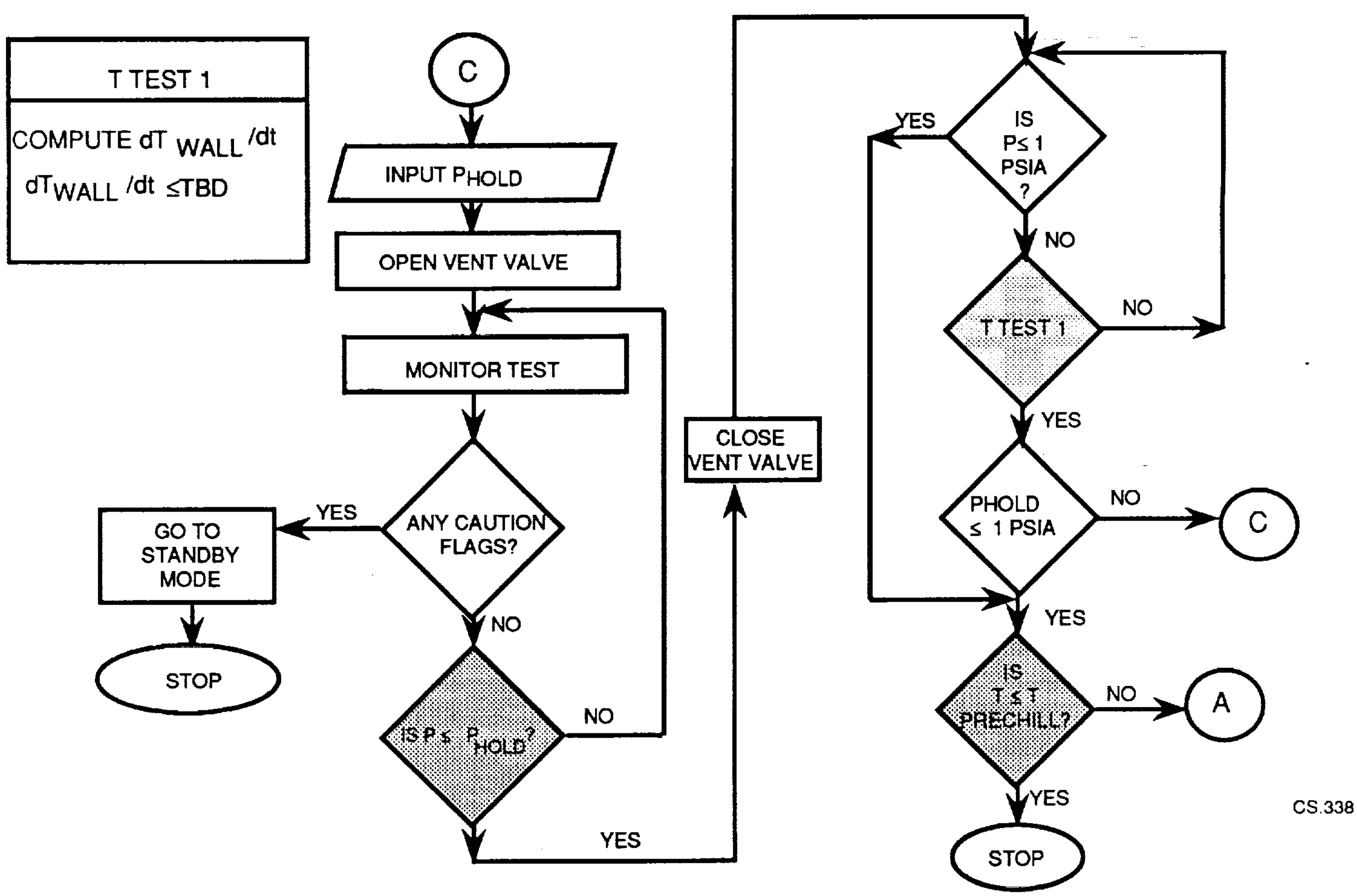

Figure 3-4. Flowchart for chilldown experiment vent cycle.

An alternative experiment called continuous flow cooldown will also be investigated in the OTV tank. The experiment begins with an empty, warm receiver tank (temperature above 250 K) at 7 kPa (1 psia) or less. The supply tank is pressurized to approximately 138 kPa (20 psia), flow valves are opened to the regulated vent line and between the supply and receiver tank, and liquid hydrogen flow begins through the tangential spray nozzle. Initially, the pressure will rise rapidly as hydrogen is vaporized in the warm tank. At 105 kPa (15 psia), the back-pressure regulator will allow vapor flow to begin through the wall-mounted heat exchanger. The process continues until the chilldown target temperature is reached. The vent is then closed and the no-vent fill test is started.

3.1.3 No-Vent Fill

No-vent fill will be investigated in two receiver tanks with different mass-to-volume ratios. The receiver is initially chilled down to a target temperature (based on ground test results) and vented to near space vacuum (less than 10 kPa). After closing the vent valves, liquid hydrogen is introduced to the receiver tank via a spray system to promote fluid mixing. As the fill proceeds, the pressure will rise and the liquid transfer rate will decrease accordingly. In some tests, it may be necessary to re-direct flow through an alternate spray system to collapse vapor in a central bubble.

The fill process will terminate when the receiver tank is 95 percent full or when the pressure reaches a pre-determined level. If the receiver tank exceeds the maximum pressure allowed during the test, a hold period will be initiated to allow the system to come to thermal equilibration, followed by an attempt at additional filling. Following each fill, the TVS is activated to reduce pressure from the final state [approximately 130 kPa (18.5 psia)] to the standby state [103 kPa (15 psia)].

Control of no-vent fill tests is based on receiver tank pressure sensors and totalized output of mass flow meters. The no-vent fill experiment flowchart is given in **Figure 3-5**. The operations required for each test in the no-vent fill matrix are listed in **Table 3-4**.

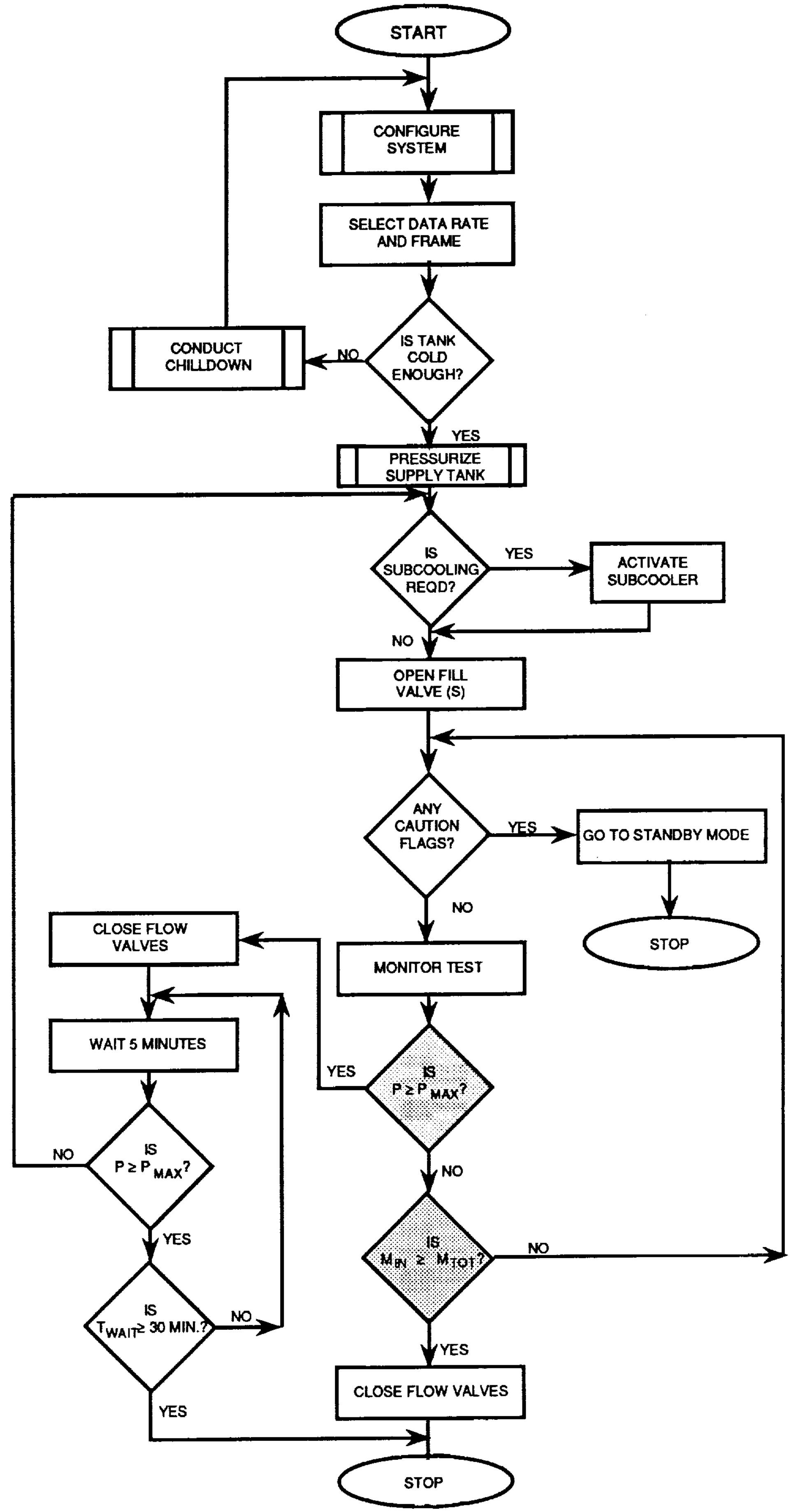

Figure 3-5. Flowchart for no-vent fill experiment. CS.170

Table 3-4

OPERATIONS TABLE FOR NO-VENT FILL TESTS

DESCR	TANK	TEST#	g-VECTOR	VALVES OPEN	CNTRL SENSORS	CRITERIA	ACTION
AXIAL	OTV	3.010	+Z	19,31,25	LFM1	M>48 LB	CLOSE 25
AXIAL	OTV	3.020	+Z	19,32,25	LFM1	M>48 LB	CLOSE 25
AXIAL	OTV	3.030	+Z	19,33,25	LFM1	M>48 LB	CLOSE 25
AXIAL	OTV	3.040	+Z	TBD	LFM1	M>48 LB	CLOSE 25
AXIAL	OTV	3.050	+Z	19,TBD,25	LFM1	M>48 LB	CLOSE 25
AXIAL	OTV	3.060	BKGD	19,TBD,25	LFM1	M>48 LB	CLOSE 25
AXIAL	OTV	3.070	-Z	19,TBD,25	LFM1	M>48 LB	CLOSE 25
AXIAL	OTV	3.080	-Z	19,TBD,25	LFM1	M>29 LB	CLOSE 25
AXIAL	OTV	3.090	-Z	18,TBD,25,3,38	LFM1	M>11 LB	CLOSE 25,3
AXIAL	DEPOT	3.110	+Z	19,31,22	LFM1,2	M>90 LB	CLOSE 22
AXIAL	DEPOT	3.120	+Z	18,32,22,3,38	LFM1,2	M>90 LB	CLOSE 22,3
AXIAL	DEPOT	3.130	+Z	18,33,22,3,38	LFM1,2	M>90 LB	CLOSE 22,3
AXIAL	DEPOT	3.140	-Z	18,33,22,3,38	LFM1,2	M>24 LB	CLOSE 22,3
RAD/TANG	OTV	3.160	BKGD	19,31,24,26	LFM1,4	M>48 LB	CLOSE 31,24,26
RAD/TANG	OTV	3.170	BKGD	19,33,24,26	LFM1,4	M>48 LB	CLOSE 33,24,26
LAD	DEPOT	3.180	BKGD	19,33,23	LFM1,3	M>90 LB	CLOSE 23

NOTES: ALL NO-VENT FILL TESTS REQUIRE SUPPLY TANK PRESSURE OF 25 PSIA MAINTAINED BY OPENING VALVES 51, 54, AND 58 OR 59. ALL NO-VENT FILL TESTS HAVE A SECONDARY CONTROL CRITERIA OF P>PMAX BASED ON SENSORS P17 AND P23 (OTV) AND P21 AND P22 (DEPOT).

3.1.4 LAD Fill

In the LAD fill experiment, the LAD channels are initially warm and free of liquid. The channels are cooled to 20 K (36 R) by liquid hydrogen flowing into the depot tank, but the channels will not fill without either being vented or subcooled to collapse the trapped hydrogen vapor. Most tests in the matrix concentrate on TVS cooling of the LAD channel; one test will assess the effectiveness of LAD filling by direct venting.

To initiate filling the channel using the TVS, the line connected to the J-T valve is opened to the 10 kPa (1.5 psia) back pressure regulator. Hydrogen enters the J-T valve at saturation conditions and expands to a cooler temperature (from 20 K to 17 K). The TVS line is thermally coupled to the LAD channel and chills the LAD and its contents, which condenses vapor trapped in the channel. Condensation decreases pressure within the channel which draws liquid in from the tank. TVS fill tests are conducted for two hours to insure that the entire channel is filled with liquid.

Although time is the primary criteria for controlling TVS fill tests, liquid detection in the TVS vent (indicating inadequate heat transfer), periodic checks with liquid-vapor sensors inside the LAD channel, and an outflow sequence following the fill will all indicate the effectiveness of the fill process. A flowchart describing TVS filling of LAD channels is given in Figure 3-6 and a table of required operations is given in Table 3-5.

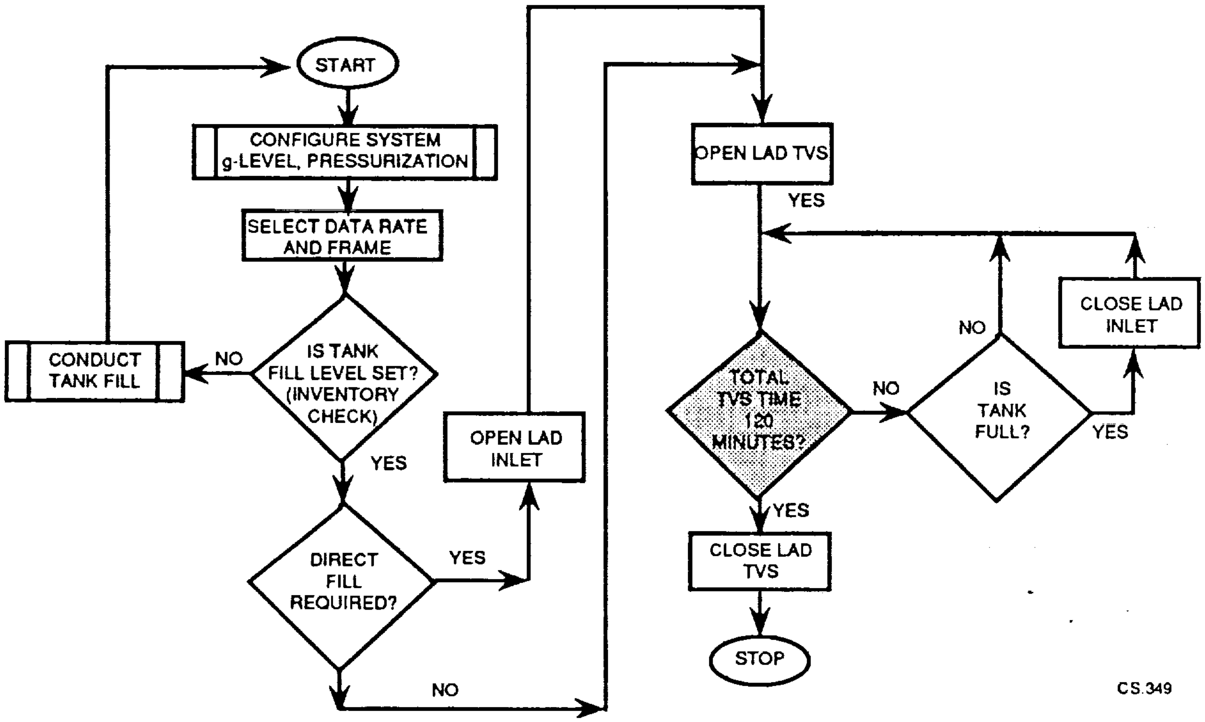

Figure 3-6. Flowchart for TVS fill of LAD channels.

Table 3-5

OPERATIONS TABLE FOR LAD FILL

DESCR	TEST #	VALVES OPEN	CONTROL SENSORS	CRITERIA	ACTION
Tank Fill	4.010	12*,38	N/A	Time	Close 12
Tank Fill	4.020	12,38	N/A	Time	Close 12
LAD Fill	4.030	19,32,23, 12*,38	LV56-75	Liquid	Close 23,12
LAD Fill	4.040	19,32,23,	LV56-75	Liquid	Close 23,12
Tank Fill	4.050	12*,38	N/A	Time	Close 12
LAD Fill	4.060	19,32,23, 12*,38	LV56-75	Liquid	Close 23,12
LAD Fill	4.070	19,32,23, 12*,38	LV56-75	Liquid	Close 23,12
Vent	4.080	19,32,23, 11,37	LV4-6	Liquid	Close 11,23

*50% valve duty cycle
LAD fill tests require supply tank pressurization at 172 kPa (25 psia) using valves 51, 54, and 58

3.2 CLASS II EXPERIMENT OPERATIONS

Control of Class II experiments is more straightforward than Class I experiment control, and therefore, Class II control flowcharts are not included in this report; however, they are included in the ERD.

3.2.1 Pressurization

Since pressurization is a supporting operation for all fluid transfers, its control loop is secondary to the primary fluid transfer control loop. The only software control criteria is over-pressure shut-off of the pressurization valve. Constant pressure control is managed by the mechanical regulator. Pressure is selected by opening one of three low pressure regulator valves. Control at a pressure other than one of the three regulated levels could be incorporated by modulating the regulator valve at a certain rate, or by incorporating a simple feedback loop based on tank pressure with a reasonable dead band.

3.2.2 Settled Transfer

Settled transfer experiments simulate normal-gravity fluid operations using an open vent during filling and an outflow baffle during draining. Primary control for both experiments requires vapor/liquid detection in the transfer lines, which can be accomplished using a bank of 3 carbon resistors. When liquid floods the vent line during a fill operation, the sensors will detect the change from vapor to liquid and the ECP will close the fill valve. A secondary control criteria for filling is a fill level greater than 95 percent (as determined by the liquid flow meter), but it is doubtful that the secondary criteria will be important in most settled fill tests. When vapor pulls through the baffle during outflow, the sensors will detect the change from liquid to vapor and will close the outflow line. After the primary transfer valve is closed, other valves will be closed in the proper order to allow pressure relief in the transfer lines.

3.2.3 LAD Expulsion

Similar to the settled transfer experiments, a LAD expulsion test ends when vapor is detected in the outflow line by a bank of 3 carbon resistors. Although the tests require variations in the induced g-level, mass flowrate, and type of pressurant, the primary control criterion remains the same. Mass flowrate variation is accomplished by varying the internal pressure of the depot tank during outflow.

3.2.4 Line Chilldown

On-orbit line chilldown will follow procedures developed and tested during ground testing. Their primary purpose will be to assess any differences in the effects of zero-g heat transfer and 2-phase flow on line chilldown. The control algorithm is straightforward; the liquid supply valve is cycled at a rate determined by ground testing until the end point of the line reaches 20 K (36 R). Once the line is cold, a fluid transfer test begins.

3.2.5 Fluid Subcooling

The fluid subcooling tests are designed to support no-vent fill tests by providing subcooled liquid from the supply tank LAD. The subcooler valve is opened first to allow the heat exchanger to cool down. Once the heat exchanger is cold, outflow begins and continues until the no-vent fill test is completed. The subcooler valve is then closed, and the heat exchanger performance is evaluated from the temperature and pressure data.

3.2.6 Fluid Dumping

Control of the fluid dumping experiment is limited because it is not currently feasible to measure the rate of 2-phase flow out of the receiver tank. Consequently, the dump valve is opened and the tank dumps fluid until the tank pressure drops to just below 10 kPa (1.5 psia). The dump valve is closed, and residuals are measured by venting the tank contents through the TVS system. An additional repressurization and dump cycle can be added to the dump experiment if desired.

3.3 REFERENCES

3.1 COLD-SAT Experiment Requirements Document, Revision 3.1, March 23, 1990.

Section 4
EXPERIMENT SUBSYSTEM DESIGN

4.1 EXPERIMENT SUBSYSTEM CONFIGURATION

The experiment subsystem consists of a 3.483 m^3 (123 ft^3) supply tank, a 0.708 m^3 (25 ft^3) cylindrical receiver tank, a 0.334 m^3 (11.8 ft^3) spherical receiver tank, 8 composite pressurant bottles 0.127 m^3 (4.5 ft^3) the LH_2 distribution and pressurization subsystems with their interconnecting lines, and the experiment control processor (ECP). The locations of these major components on the COLD-SAT spacecraft are shown in **Figure 4-1**.

Total lift-off weight for the experiment subsystem is 1516.4 kg (3,344 lbs); this assumes a 95 percent full supply tank. The weight, C.G. coordinates, and moments of inertia are shown in **Table 4-1** for each of the subsystem hardware components.

Table 4-1

EXPERIMENT SUBSYSTEM WEIGHT SUMMARY

			WEIGHT (KILOGRAM)	C. G. COORDINATE (CENTIMETER)			MOMENTS OF INERTIA (KG-METERS SQUARED)		
CODE	ITEM	DWG	WT	X	Y	STA.	IXX	IYY	IZZ
SCIENCE									
10.20.01E	OTV TANK ASSY		12.700	0.00	0.00	555.00	3.33	3.33	3.33
10.20.02E	DEPOT TANK ASSY		39.800	0.00	0.00	414.00	8.00	8.00	4.80
10.20.03E	SUPPLY TANK ASSY		759.000	0.00	0.00	190.50	550.00	550.00	440.00
10.20.04C	PRESSRNT BOTL PR +X		64.000	66.00	0.00	460.00	19.60	14.60	8.60
10.20.05C	PRESSRNT BOTL PR +Y		64.000	0.00	66.00	460.00	14.60	19.60	8.60
10.20.06C	PRESSRNT BOTL PR -X		32.000	-66.00	25.00	460.00	9.80	7.30	4.30
10.20.07C	PRESSRNT BOTL PR -Y		64.000	0.00	-66.00	460.00	14.60	19.60	8.60
60.20.08S	PRESSURANT +X		5.400	66.00	0.00	460.00	0.00	0.00	0.00
60.20.09S	PRESSURANT +Y		5.400	0.00	66.00	460.00	0.00	0.00	0.00
60.20.10S	PRESSURANT -X		2.700	-66.00	25.00	460.00	0.00	0.00	0.00
60.20.11S	PRESSURANT -Y		5.400	0.00	-66.00	460.00	0.00	0.00	0.00
60.20.12S	OTV TANK LIQUID		0.000	0.00	0.00	555.00	0.00	0.00	0.00
60.20.13S	DEPOT TANK LIQUID		0.000	0.00	0.00	414.30	0.00	0.00	0.00
60.20.14S	SUPPLY TANK FLUID		234.300	0.00	0.00	190.50	110.00	110.00	84.00
10.20.15E	LOWER MANIFOLD		35.000	0.00	0.00	327.00	40.00	40.00	16.00
10.20.15E	UPPER MANIFOLD		40.000	0.00	0.00	505.00	40.00	40.00	16.00
10.20.15E	LH2 SERVICING MNFLD		14.000	0.00	0.00	160.00	20.00	20.00	8.00
10.20.15E	PLUMBING,LWR		10.000	0.00	0.00	327.00	0.00	0.00	0.00
10.20.15E	PLUMBING,UPR		10.000	0.00	0.00	505.00	0.00	0.00	0.00
10.20.15E	PLUMBING,LH2 SERVICE		20.000	0.00	0.00	160.00	0.00	0.00	0.00
40.20.16E	ELECT PROCESSOR #1		24.600	-96.50	-20.00	337.00	0.00	0.00	0.00
40.20.16E	ELECT PROCESSOR #2		24.600	-96.50	20.00	337.00	0.00	0.00	0.00
60.20.17E	HELIUM TANK		32.000	-66.00	-25.00	460.00	0.00	0.00	0.00
60.20.18S	HELIUM		5.000	-66.00	-25.00	460.00	0.00	0.00	0.00
60.20.19S	VENT H-X ASSY		9.500	0.00	94.00	127.00	0.00	0.00	0.00
60.20.20E	ACCELEROMETER		2.500	-101.60	-21.20	374.60	0.00	0.00	0.00
60.20.21E	MIXER POWER SUPPLY		0.500	-99.00	-21.20	388.60	0.00	0.00	0.00
SUM OF SCIENCE			1516.400	-3.431	0.509	267.625	3105.228	3140.893	786.195

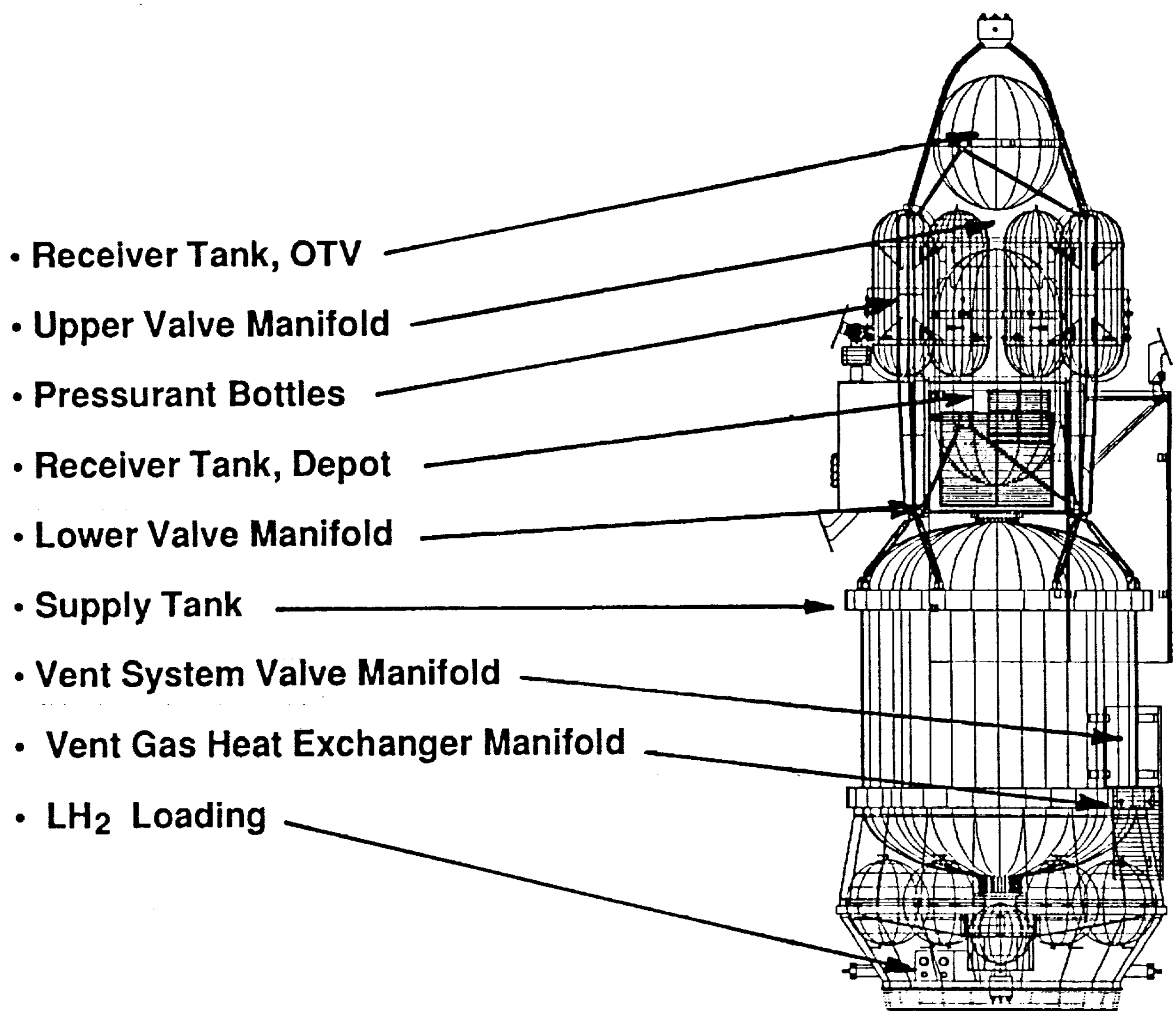

Figure 4-1. Experiment subsystem topology.

4.2 LH_2 DISTRIBUTION SUBSYSTEM

The LH_2 distribution subsystem is designed to accommodate all the experiments defined in the COLD-SAT integrated experiment set. It has built-in redundancy such that a single valve or instrument failure will not prevent successful completion of the mission. For safety, the system is a minimum of two fault tolerant. A schematic of this system is shown in **Figure 4-2**.

All transfer lines are 1.3 cm x 0.071 cm wall (1/2 inch x 0.028 inch) SS, insulated with 20 layers of MLI, and are thermally isolated from the warm spacecraft structure. Line routing between the primary valve assemblies is along the four main support posts and the supply tank outer shell. The only vacuum jacketed lines are those used for ground filling and venting of the supply dewar. Tank overpressure protection is provided by a burst disc and relief valve combination; line overpressure protection uses only relief valves. The supply tank has an additional burst disc to provide relief in the event of loss of guard vacuum during ground operations.

All solenoid valves are magnetically latched, open or closed, and each has a position indicator to verify valve status. Flow control is accomplished using fixed orifices on separate parallel paths. Turbine flow meters are used in the LH_2 lines; thermal mass flow meters will be used on all vent lines. Flow meters are positioned to keep track of vapor vented from the system and liquid transferred between tanks. Vent lines are routed to a sun facing radiator to warm the flow, which decreases the operating temperature range of the mass flow meters (increasing their accuracy) and permits the use of non-cryogenic back pressure regulators. There are three vent paths to space: (1) a 103 kPa (15 psia) back pressure vent for line chill down, low-g fill, continuous flow tank chill down, and settled fluid conditions where venting is required, (2) a 10 kPa (1.5 psia) vent for TVS vent flows, and (3) an open (0 kPa) vent for tank and transfer line evacuation.

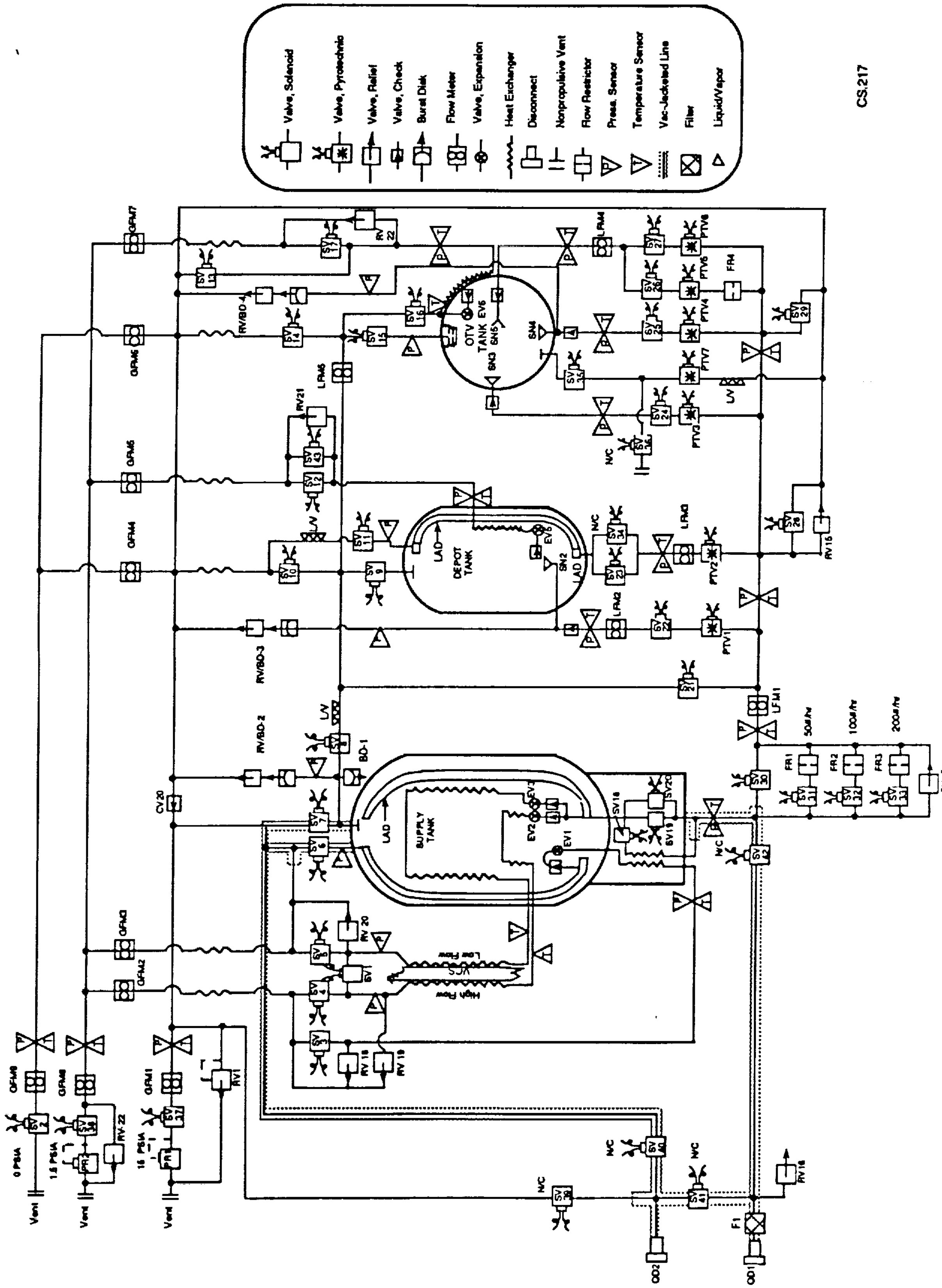

Figure 4-2. LH_2 subsystem schematic.

Figure 4-3 identifies the primary locations for the LH_2 and pressurization subsystem components: (1) upper diaphragm panel, (2) lower diaphragm panel, (3) vent manifold/heat exchanger, and (4) the LH_2 servicing panel. There are also some cold valves and pressurization valves attached directly to or near each tank. Generally, the upper diaphragm panel contains the plumbing for the OTV tank and pressurization subsystem, the lower diaphragm panel contains plumbing associated with the supply and depot tank, and the vent panel contains all the vent flow plumbing and gas flowmeters. Located on the LH_2 servicing assembly panel are the male halves of the cryogenic disconnects, the fill and vent line shut off valves, and a cross-over valve used to purge the lines after tank filling is complete. This panel is attached to the supply tank support ring and aligned to the fairing access door. **Figure 4-4** shows layouts of the LH_2 distribution subsystem valve assemblies on the upper and lower diaphragm panels and the vent assembly.

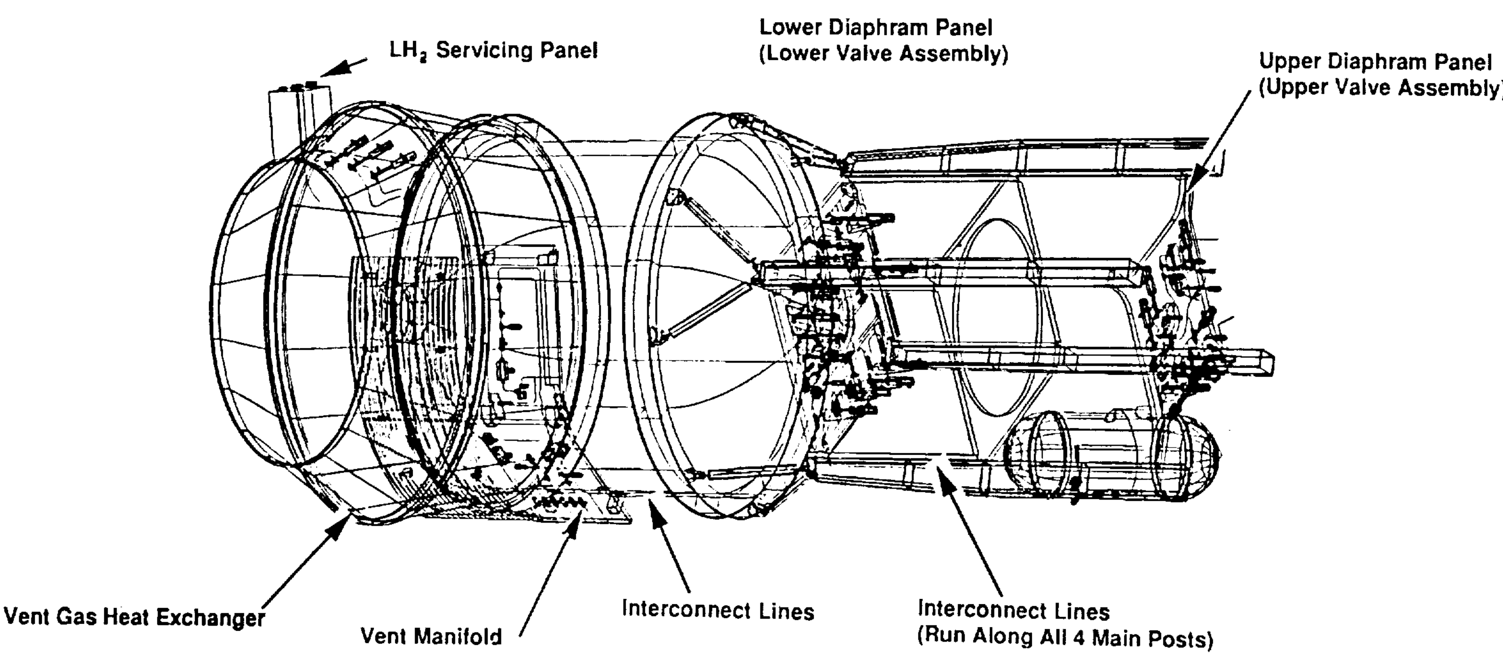

Figure 4-3. COLD-SAT plumbing general arrangement.

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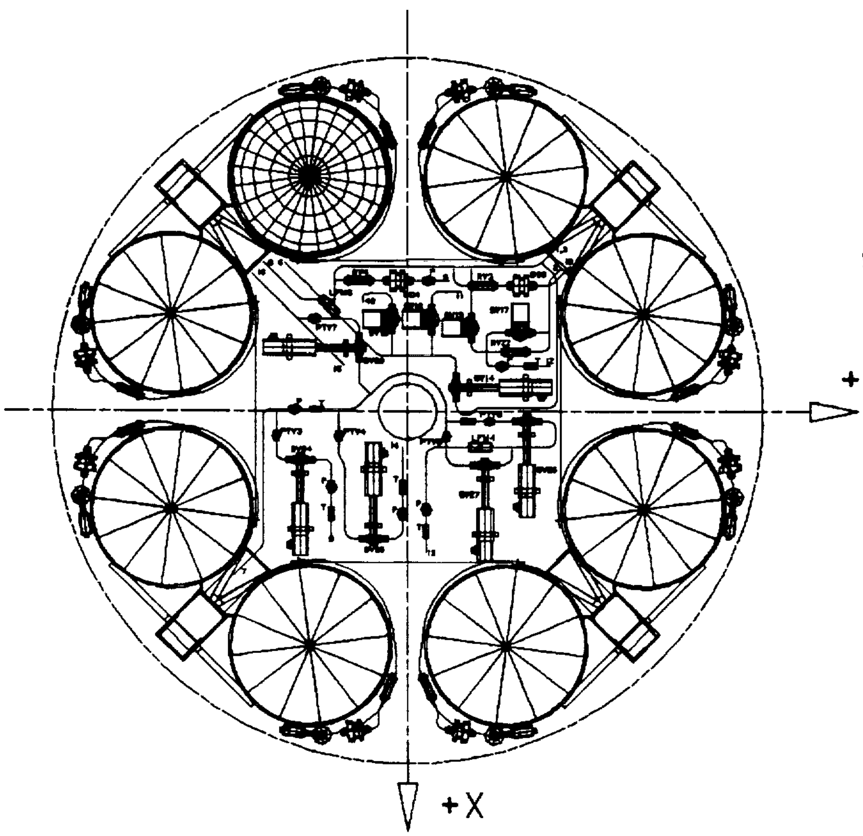
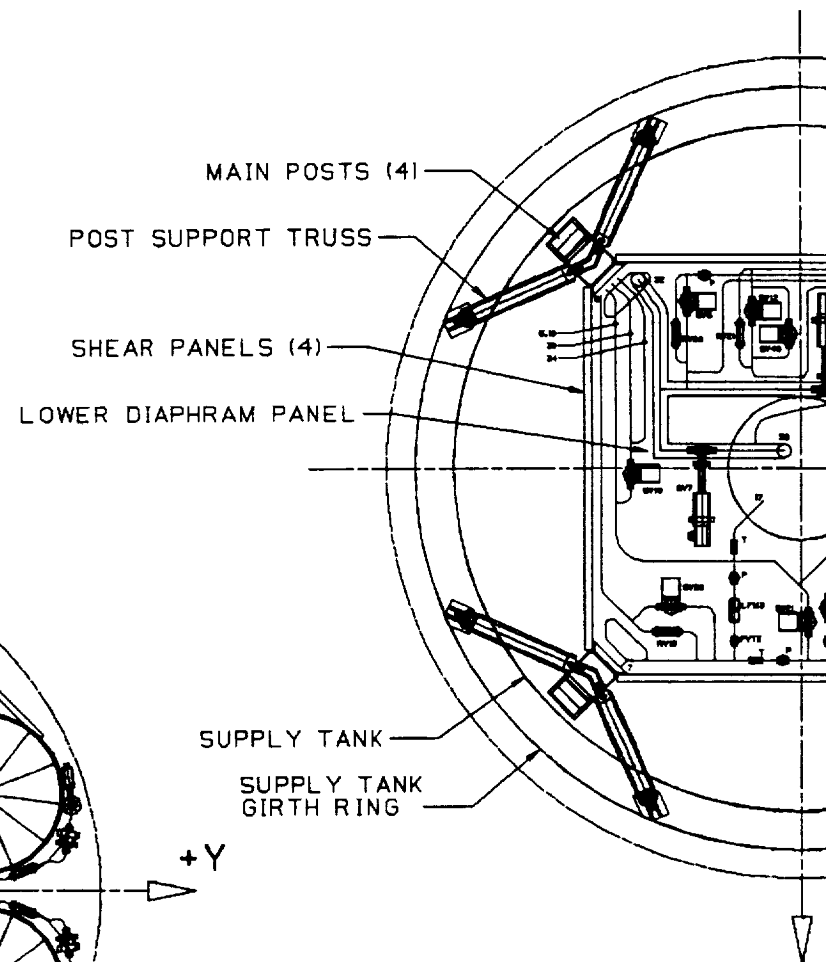


• LH_2 Distribution Subsystem

• Upper Diaphram Panel, Top Side

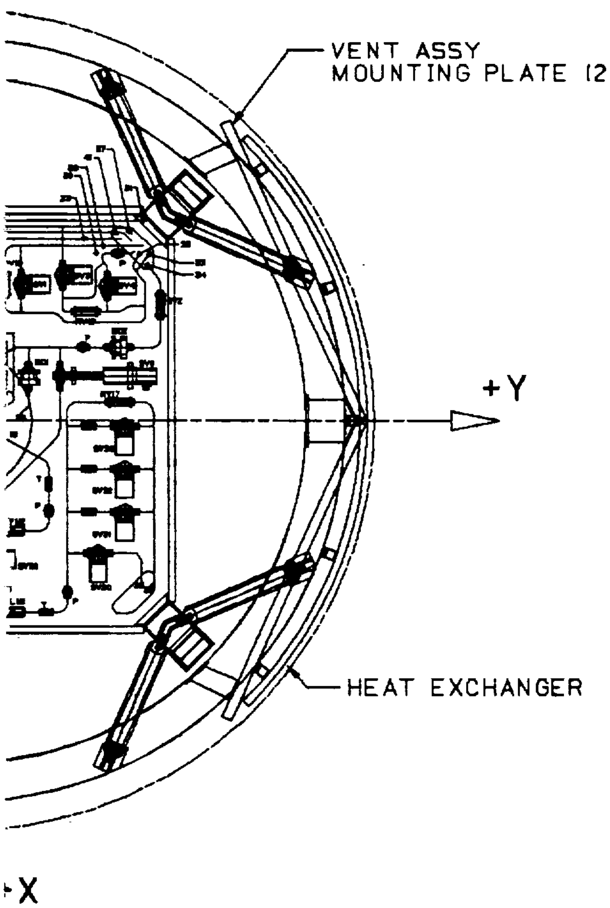
Figure 4-4a. Component Installation, Upper Valve Assembly.

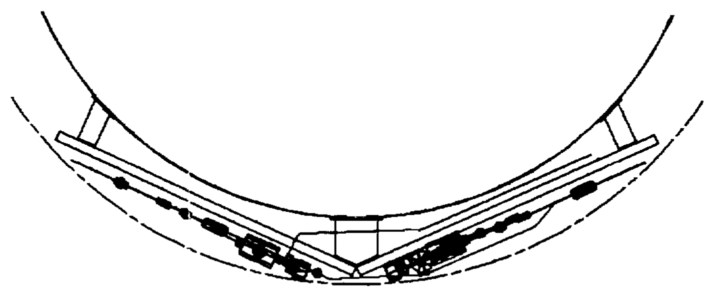


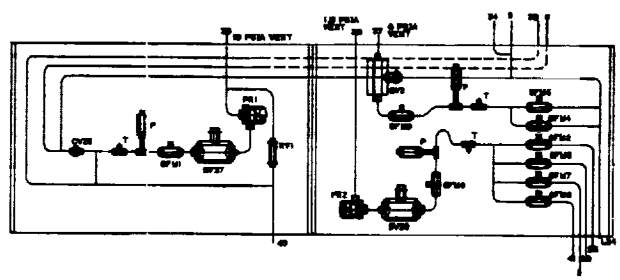
• LH_2 Distribu

• Lower Diapl

Figure 4-4b. Compor Low







ion Subsyetem

am Panel

ent Installation,
Valve Assembly.

Figure 4-4c. Component Installation,
Vent Valve Assembly.

Pressure losses were calculated for the LH_2 supply line and its components for maximum flow path length and maximum flowrate. The analysis assumed:

1. The maximum flowrate of 91 kg/hr (200 lb/hr)

2. Saturated liquid at 103 kPa (15 psia) is in the supply tank before pressurization to 172 kPa (25 psia)

3. Desired fluid condition at the receiver tank is subcooled LH_2 at 117 kPa (17 psia), giving a design pressure drop of 55.2 kPa (8 psid)

Total pressure drop through the system, exclusive of the flow control orifice is 40.4 kPa (5.86 psid). The difference between the design ΔP and the calculated ΔP is 14.8 kPa (2.15 psi), which is used to size the flow control orifice. The diameter of a square edged orifice which gives a 14.8 kPa pressure drop at 91 kg/hr (200 lb/hr) flowrate is 0.591 cm (0.23 inch). Two additional flow restrictions in parallel provide flow control at mass flowrates of 45 and 22.7 kg/hr (100 and 50 lbs/hr). A summary of the system pressure drop and flow control orifice parameters is given in **Table 4-2.**

Table 4-2

EXPERIMENT SUBSYSTEM FLOW CONTROL ORIFICE SIZING

Maximum Flowrate		System ΔP		Orifice ΔP		Orifice Diam.	
(kg/hr)	(lb/hr)	(kPa)	(psi)	(kPa)	(psi)	(cm)	(in)
90.7	200.0	40.4	(5.86)	14.8	(2.15)	0.591	(0.23)
45.4	100.0	10.1	(1.47)	45.1	(6.54)	0.325	(0.15)
22.7	50.0	2.6	(0.38)	52.6	(7.63)	0.221	(0.09)

4.3 PRESSURIZATION SUBSYSTEM

The pressurization system provides both warm hydrogen and helium gas for tank pressurization. Gaseous hydrogen (GH_2) is stored in seven 0.127 m^3 (4.5 ft^3)

bottles at a pressure of 27.6 MPa (4000 psia). All seven bottles are manifolded together but are isolated by check valves in the event one bottle develops a leak. Gaseous helium (GHe) pressurant is stored in a single bottle of identical design and is plumbed into the same distribution system as the GH_2 pressurant. The helium system is isolated from the hydrogen system by parallel redundant isolation valves and check valves. A pair of valves (SV60 and SV61) are used to vent the supply tank pressurization lines to vacuum for purging them clean of any GHe pressurant. A schematic of the pressurization subsystem is shown in **Figure 4-5.**

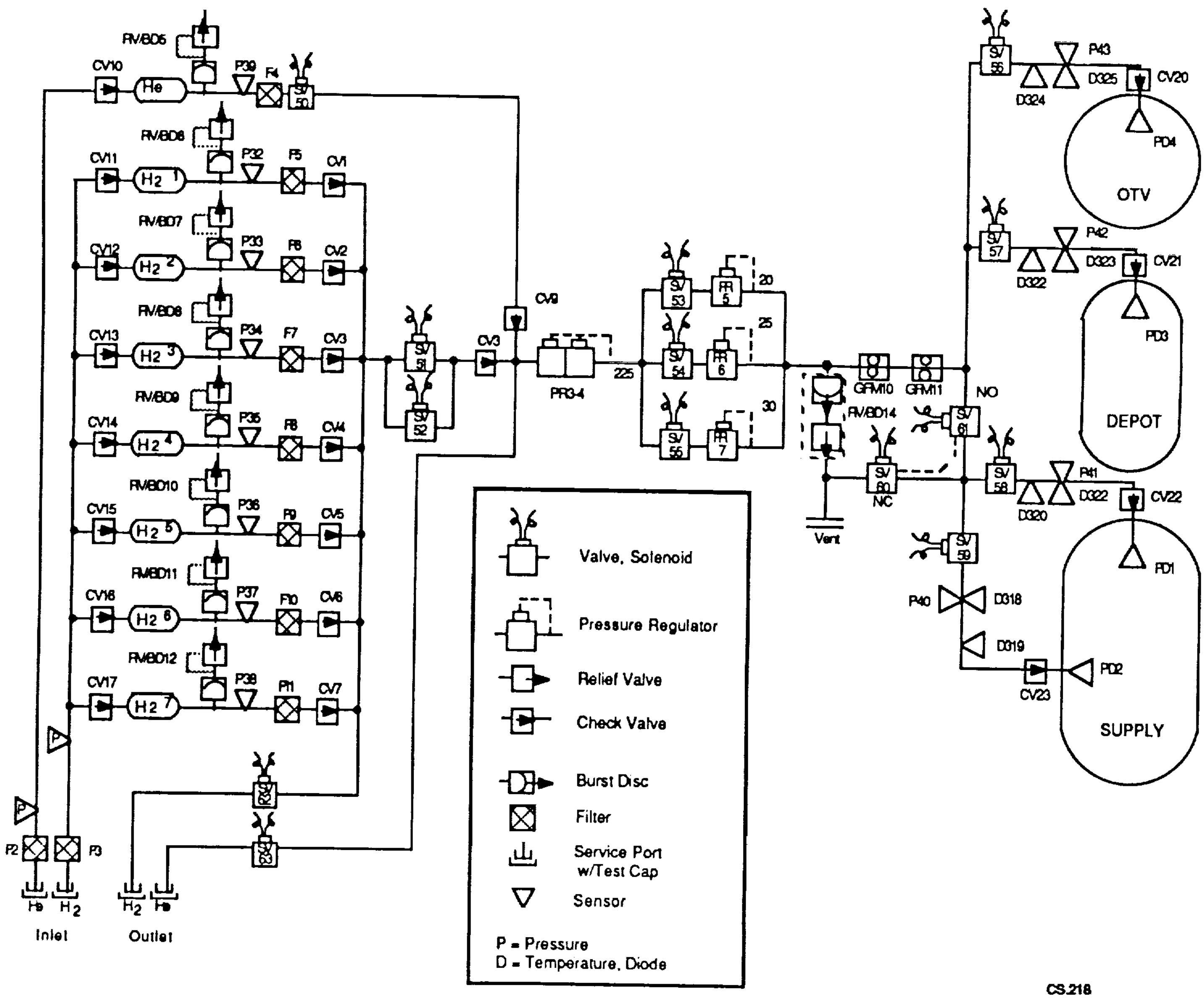
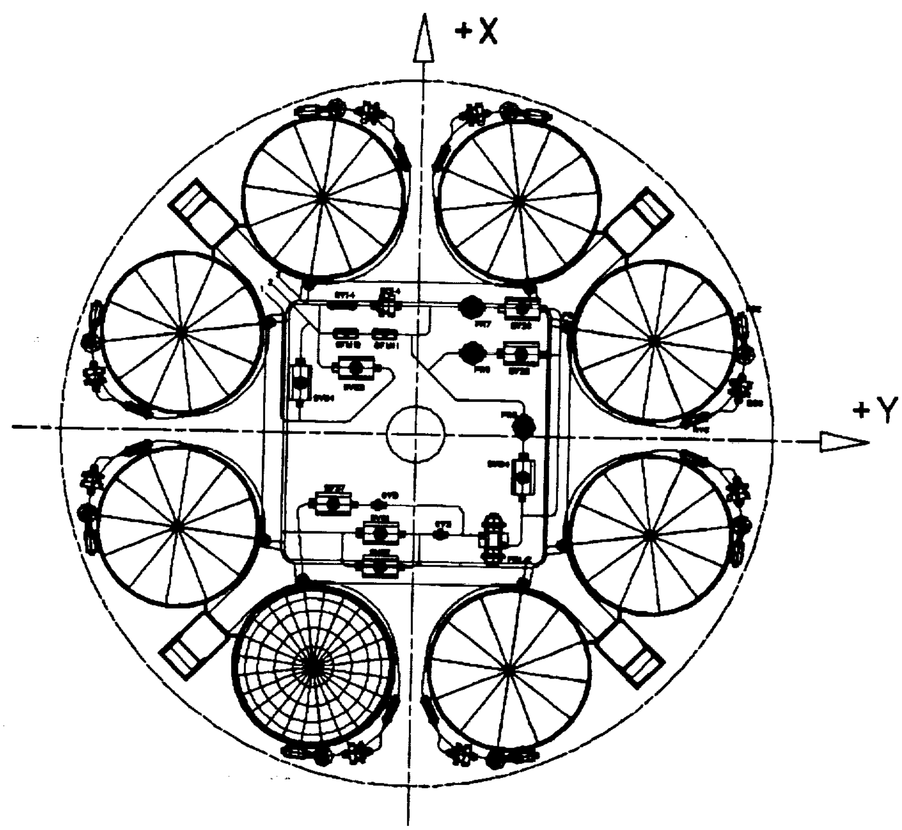


Figure 4-5. Pressurization subsystem schematic.

Pressure from the pressurant bottles is reduced to 1.55 MPa (225 psia) by a series redundant regulator. The pressure is then further reduced to either 138, 172, or 207 kPa (20, 25 or 30 psia) and routed to the appropriate tanks. Redundant mass flow meters monitor and integrate the pressurant flow rate to provide the total pressurant used. A secondary method which will be used for determining pressurant usage is monitoring pressure and temperature of the pressurant in the pressurant tanks. Temperature and pressure are also monitored near the inlet to the diffusers so the thermodynamic state of the pressurant is known. A layout of the pressurization subsystem components is shown in Figure 4-6.



• Upper Diaphram Panel, Lower Side

Figure 4-6. Pressurization subsystem component installation.

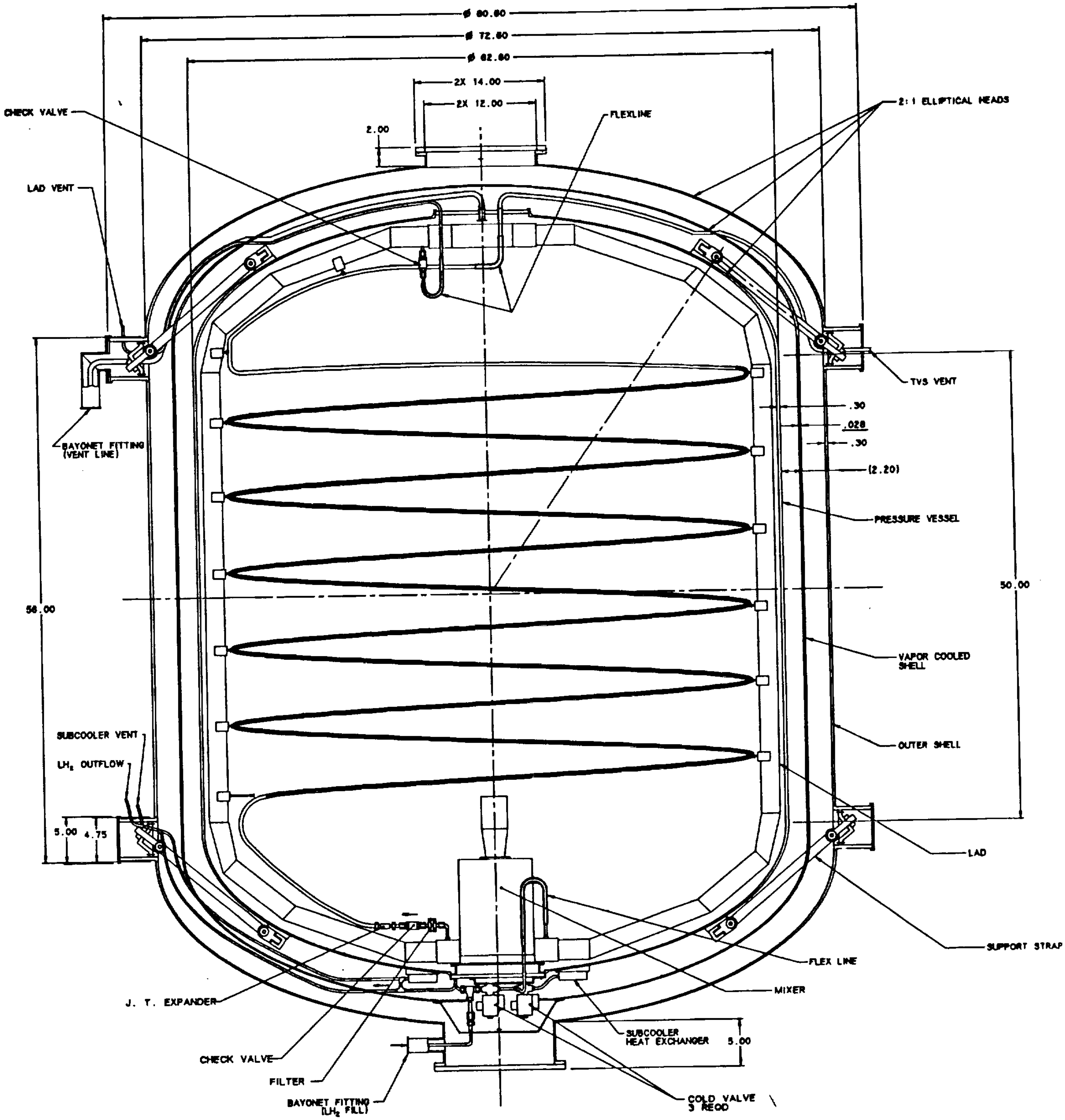
The pressurant bottles are a composite graphite fiber wrapped over a seamless aluminum liner. This is a standard Structural Composites Industries (SCI) design for which similar pressure bottles have been space qualified. The aluminum liner was selected for hydrogen compatibility, the graphite epoxy overwrap for high strength and low weight. Our current design has a working pressure of 27.6 MPa (4000 psi), a 2.0 safety factor and meets the leak-before-burst criteria. The bottles are attached to the spacecraft bus with metal collars. A vacuum tight fitting is welded on each tank to insure leak-tight connection of the pressurization lines.

4.4 SUPPLY TANK

The supply tank (**Figure 4-7**) was designed to hold sufficient liquid hydrogen for all the COLD-SAT experiments, to be the principal tank for pressure control experiments, and to serve as the primary spacecraft structure. It has an aluminum pressure vessel (PV) supported by fiberglass-epoxy straps, a vapor cooled shield (VCS), multilayer insulation (MLI), and a thermal control shield (TCS).

Heads for both the PV and outer shell (OS) are elliptical; the cylindrical sections are forged and machined with the girth rings machined into the outer shell forging. "Manholes" are located at each end of the outer shell to facilitate access to the fluid lines, cold valves and electrical feed-throughs. For ease of construction, assembly and repair (if necessary) the mixer, cold valves, all plumbing and wiring feed throughs are mounted to the dewar neck plugs.

The instrumentation tree assembly (see **Figure 4-8**) is designed to provide temperature sensor mounting locations distributed throughout the bulk tank fluid. It is a lightweight framework constructed of G-10 fiberglass to minimize its thermal mass. Its concentric ring sections have a 2.54 x 1.27 mm (0.100 x 0.050 inch) cross-section, the spokes connecting the inner and outer rings are 4.76 dia x 1.02 mm (3/16 dia. x 0.040 inch) tubes. The small cross-section of its members minimizes interference with liquid or vapor motion and the 30.5 cm (12 inch) diameter center ring provides an unobstructed path for the mixer jet.



- Cylindrical w/elliptical heads
- Volume: 3483L (123 ft^3)
- Length: 251 cm (99 in)
- Diameter: 206 cm (81 in)
- Vapor cooled shield
- Thermal control shield
- 60 layers MLI
- S-glass epoxy tension bands

Figure 4-7. Supply tank design.

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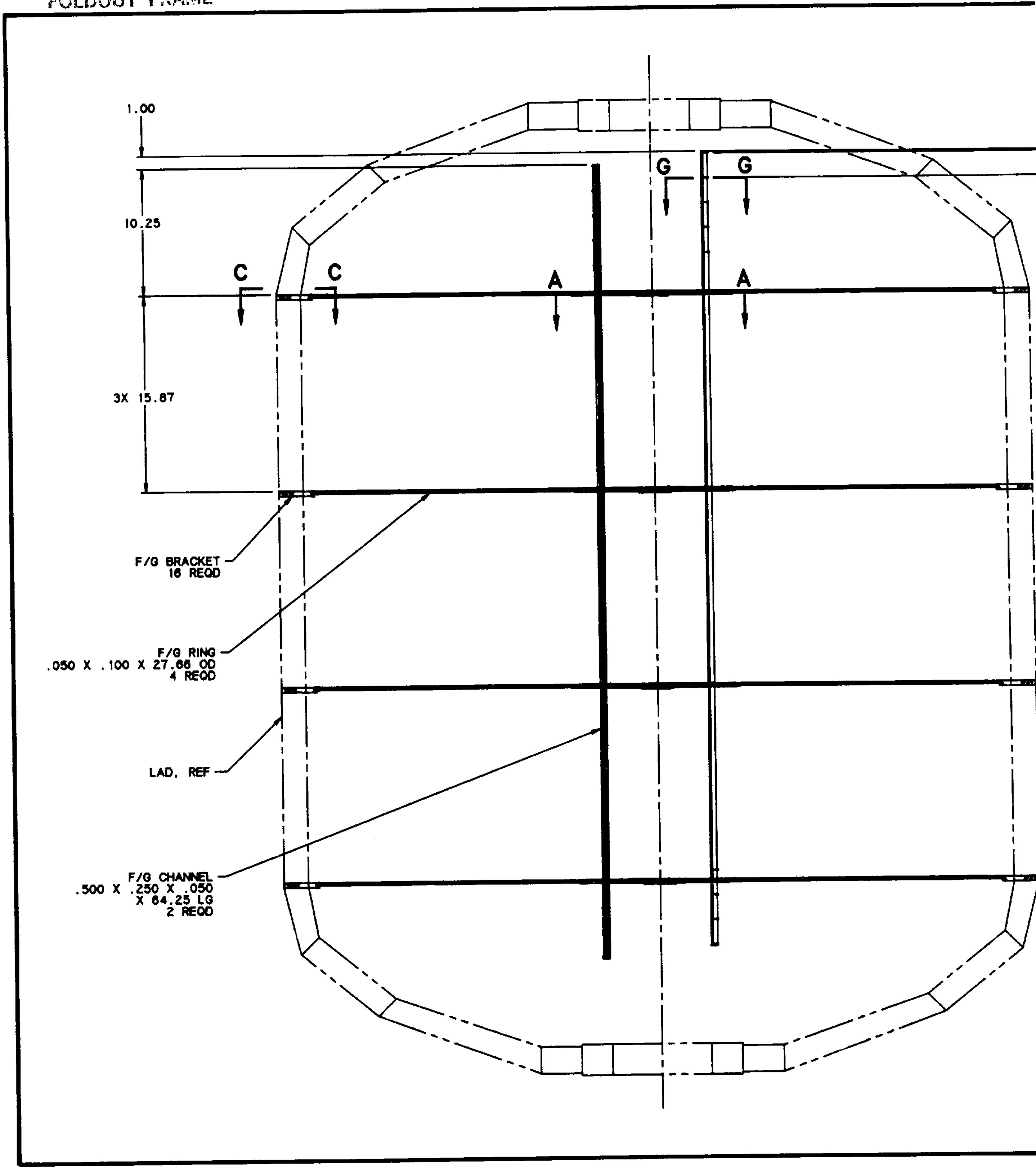

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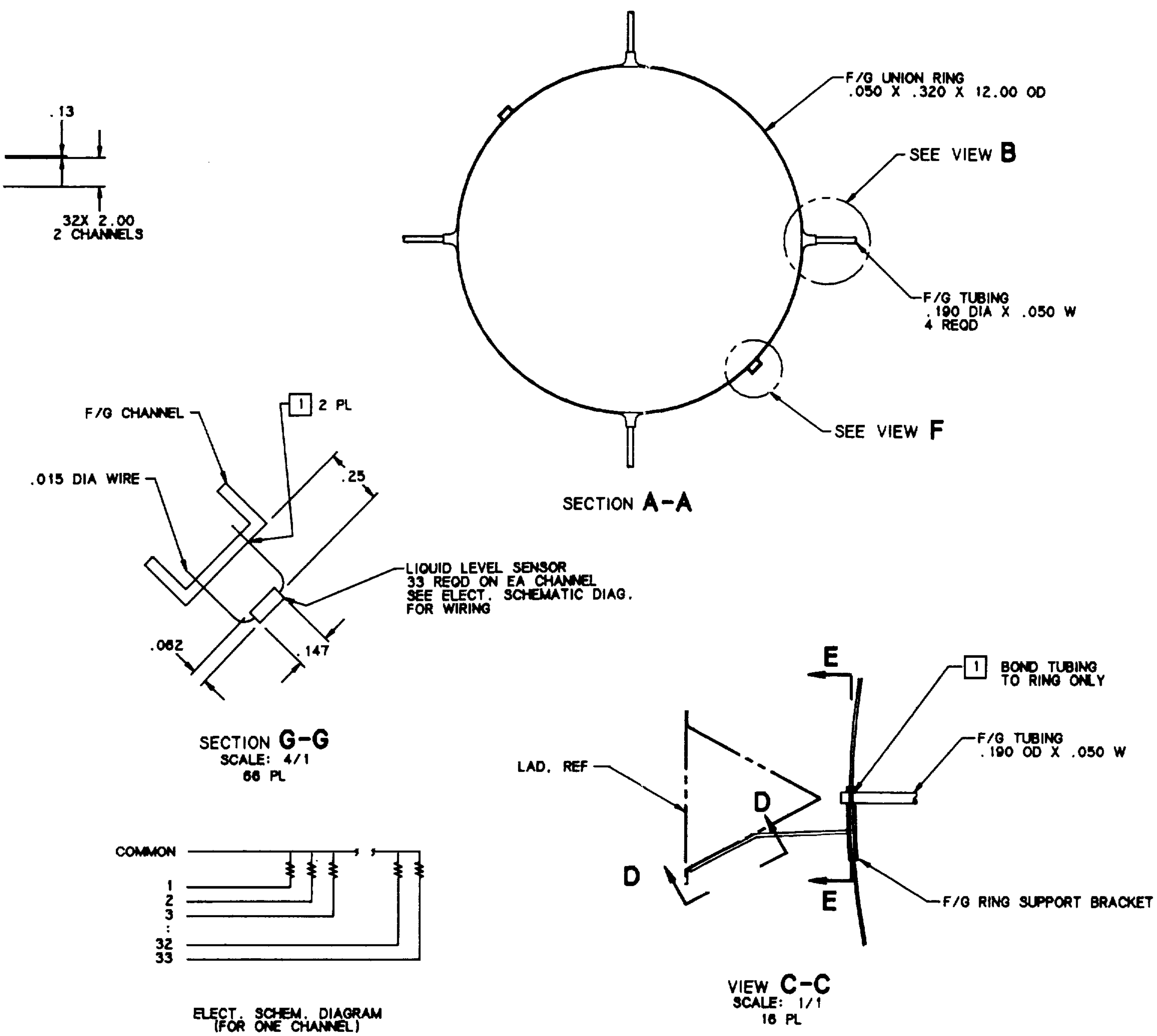

Figure 4-8. Supply tank instrumentation tree assembly.

The temperature sensors and their lead wires are bonded directly to the fiberglass structure. Because of their spacing and the low thermal conductivity of G-10, it is expected that the temperature sensors will accurately reflect the surrounding fluid temperature. Two depth probes the length of the supply tank are mounted to the center rings and instrumented with liquid vapor sensors. Thirty-three sensors, spaced 5.08 cm (2 inches) apart, are located on each depth probe. The two depth probes are staggered by 2.54 cm (1 inch) to measure liquid depth within 2.54 cm (1 inch).

4.5 RECEIVER TANKS

There are two receiver tanks on COLD-SAT; the depot tank, and the OTV tank. The depot tank is designed to simulate an on-orbit cryogenic refueling depot, the OTV tank is designed to simulate the fuel tank for an Orbital Transfer Vehicle (OTV). The depot tank is cylindrical and has a liquid acquisition device, while the OTV tank is designed to have a minimum M/V with thin, bare walls and spherical geometry. The characteristics of the two receiver tanks are given in **Table 4-3.**

Table 4-3

COMPARISON OF DEPOT AND OTV RECEIVER TANKS

CHARACTERISTIC	DEPOT TANK	OTV TANK
Geometry	Cylindrical	Near Spherical
Internal Hardware	LAD Axial Spray Only Instrumentation Tree Pressurant Diffuser	Vapor Pull-Through Suppression Baffle 3 Spray Systems: Axial, Radial & Tangential Instrumentation Tree Pressurant Diffuser
TVS	Passive Only	Passive Only
L/D	1.88	1.08
M/V (lb/ft^3) (kg/m^3)	3.0 48.1	1.4 22.4
Thermal Performance	2.0 percent/day	3.6 percent/day

4.5.1 Depot Tank

The depot tank (**Figure 4-9**) is a bare PV insulated with a 60 layer MLI blanket. The tank diameter is 84 cm (33 in) and its volume is 0.708 m^3 (25.0 ft^3). It is constructed from two hemispherical spun heads, a rolled and machined center section, and two girth rings machined from 2-inch aluminum plate. The wall thickness is 1.27 mm (0.050 in) and is fabricated from 5083 Al, used in its annealed condition. The primary purpose of this tank is no-vent fill and LAD testing. Therefore, mass to volume (M/V) ratio and LAD accommodation are primary design drivers. The M/V is approximately 48 kg/m^3 3.0 lb/ft^3), which includes a LAD weight of 10 kg (22 lbs). The main attach points for the LAD are located at the girth rings to prevent excessive loads in the PV skin. Also, thickened bosses at each end of the tank provide additional support for the LAD and load paths for plumbing feedthroughs and cold valve supports. The tank is supported by tubular G-10 fiberglass struts attached to the girth ring at one end and to the spacecraft at the other end. The MLI blankets attached to the tank have a 4 mil Kapton outer layer for rigidity and debris-meteoriod protection.

At the lower end of the tank a pressurant diffuser and axial spray are positioned inside the toroidal LAD manifold. The TVS line and the LAD vent exit the tank through the upper girth ring; the LAD outflow/inflow line exits through the lower girth ring. A low-g vent is located at the tank upper end.

The same general design requirements and constraints were used for the depot tank instrumentation tree as were used for the supply tank instrumentation tree. Since the depot tank does not have a mixer, a 6.35 mm dia x 1.02 mm wall (1/4 x 0.040 inch) fiberglass tube, to which the instrumentation tree spokes are attached, runs axially along the centerline of the tank. The depot tank instrumentation tree is attached to the LAD prior to installation into the tank.

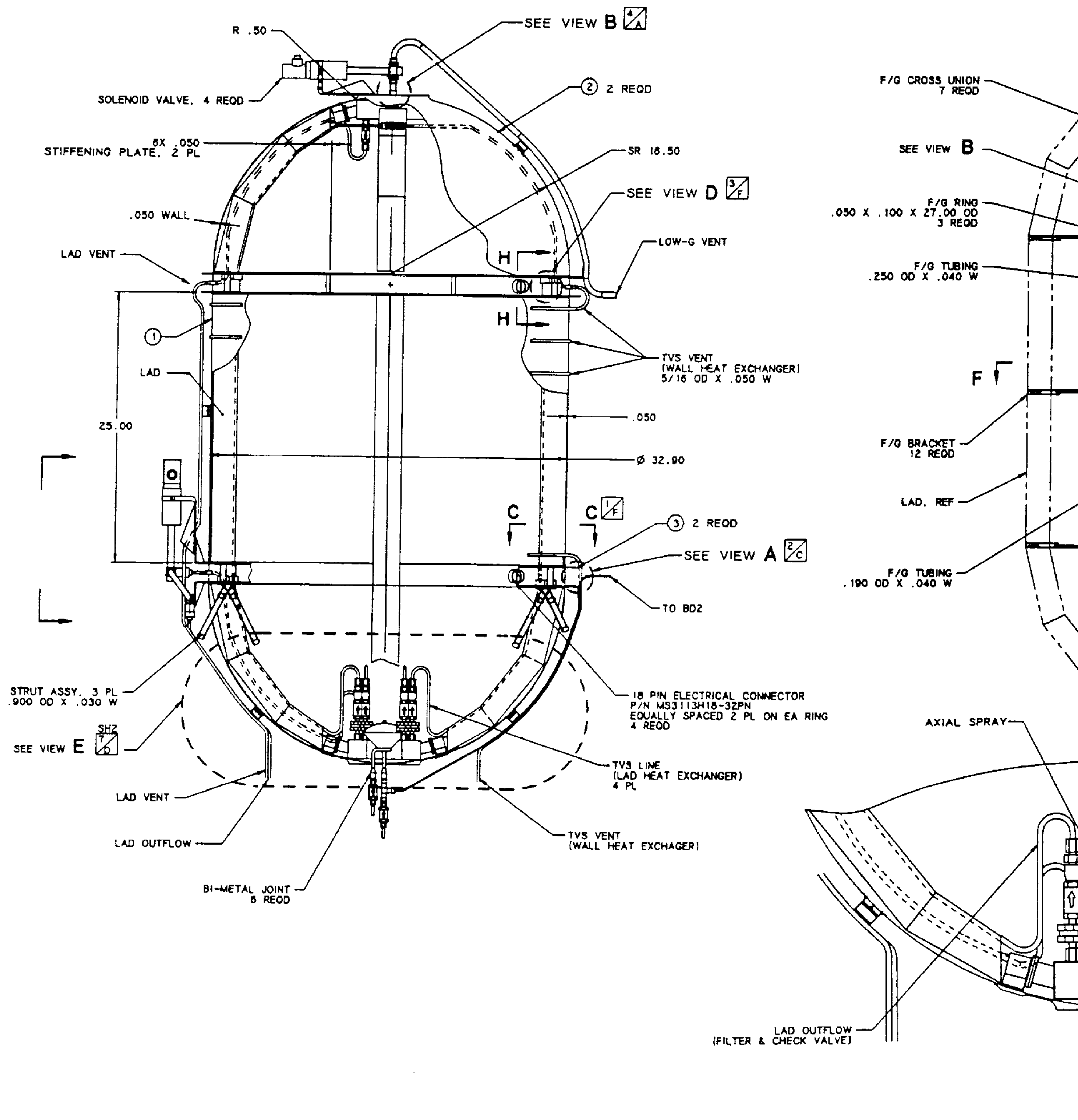

Figure 4-9. Depot tank design.

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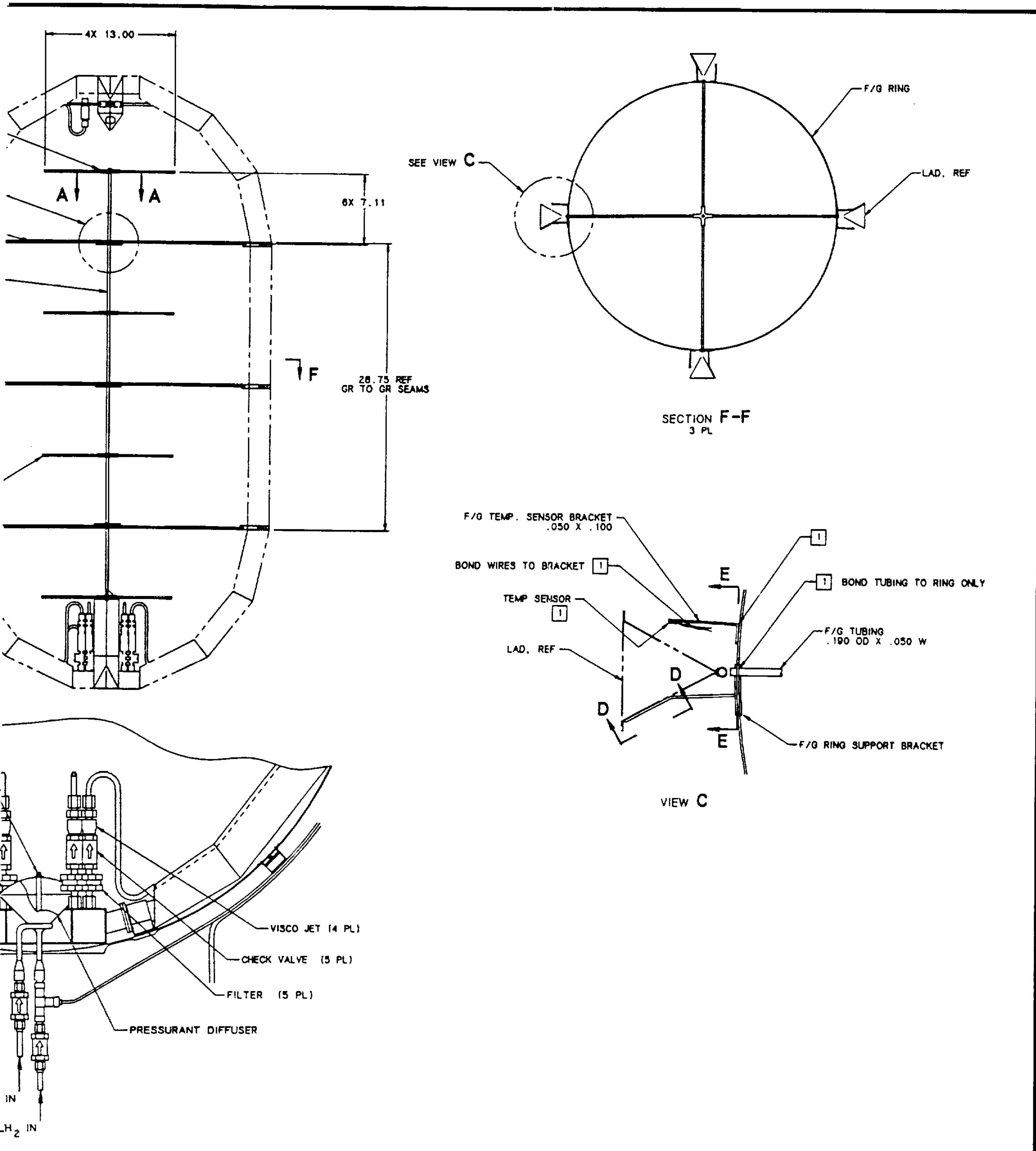


4.5.1.1 Depot Tank Structural Analysis

The McDonnell Douglas document MDC H3224, Delta II Commercial Spacecraft User's Manual (Reference 4.1), was used as a guide in deriving structural performance requirements for the experiment subsystem (Table 4-4). The experiment modal frequencies should be at least $\sqrt{2}$ above the minimum spacecraft frequencies of 35 Hz axial and 15 Hz lateral suggested in the reference.

Table 4-4

STRUCTURAL PERFORMANCE REQUIREMENTS, EXPERIMENT SUBSYSTEM

• Modal Frequency:	Lateral Axial	25 Hz 50 Hz
• Quasistatic Loads:	Primary Struc.	9g Axial 4.5 g Lateral
	Secondary Struc.	30 g Axial 15 g Lateral
• Safety Factors:	Press. Vessels	1.2 x Fty, 1.5 x Ftu } Supply 1.2 x Fty, 1.5 x Ftu } Receiver
	Primary/Secondary Struc.	1.4 x Fty 1.8 x Ftu
	Plumbing (Press.)	4 x Ftu

CS.749

The design quasi-static loads for primary structure (which includes support struts and girth rings of the receiver tanks) comes directly from the reference table of spacecraft cg limit load factors and the accompanying text which states that loads at the upper part of the spacecraft can be up to 1.5 times the values in the table. Secondary lightweight structure typically experiences considerably greater loads. The values listed here are based on experience gained in design and analysis of similar BASG spacecraft (RME, CRRES, etc.).

The safety factors of 1.4 and 1.8 on yield and ultimate, respectively, are consistent with industry standards for flight hardware. Lower safety factors of 1.2 and 1.5 are used for the receiver tanks as these are designed to be lightweight tanks and are launched empty.

The structural analysis of the Depot tank used BOSOR4, COSMOS/M and hand analysis methods; the results are summarized in **Table 4-5**. Internal pressure was the governing design criteria for the pressure vessel and has the lowest margin of safety of all failure modes. BOSOR4 was used to determine shell vibration modes, shell stability in the high-g launch environment, shell stress loads due to supporting the cold valves from the bosses at the top and bottom of the tank and stress in the girth ring.

Table 4-5

DEPOT TANK STRUCTURAL ANALYSIS SUMMARY

COMPONENT	LOAD CONDITION	MARGIN OF SAFETY	PREDICTED FREQUENCY (Hz)
Pressure Vessel	Internal Pressure Shell Load: Valve Shell Load: Launch	+0.01 YLD +0.63 YLD +0.41 Buck	172
Girth Ring	Launch Loads	+3.43 YLD	
External Plumbing	Launch	+0.22 to 1.27 YLD	87 to 300
Instrumentation Tree	Launch	+0.24 ULT	8.4
Support Struts	Launch	+0.06 Buck	47 Lateral 83 Axial
Valve Bracket	Launch	+5.11 ULT	114

CS.750

Finite element models of the external plumbing, instrumentation tree, support struts and the valve bracket were used to predict their modal frequencies and stress levels. The low modal frequency for the instrumentation tree is not considered significant because its very small mass (about 225 gms) will not significantly effect the spacecraft modal frequencies. Should induced loads be greater than the 30-g/15-g vertical/lateral, positive margins on stress can easily be attained by local reinforcing of the fiberglass instrumentation tree structure at support points.

4.5.1.2 Depot Tank Thermal Analysis

Thermal analysis for the depot tank is summarized in **Table 4-6**. Integrated thermal conductivities of 300 series stainless steel, G-10 fiberglass, and manganin were used to calculate conduction heat transfer rates. Assumptions made in calculating the conductive heat leaks were:

1. Warm and cold boundary temperatures are 300 K and 20 K (540 R and 36 R, respectively.

2. All plumbing lines consist of 0.635 mm (0.025 inch) wall SS tubing.

3. 24-gage manganin wires are used for instrumentation.

4. The plumbing lines and instrumentation wiring have a thermal length equivalent to 1/2 the circumference of a hemispherical head.

5. The solenoid valve thermal length is 10.2 cm (4 inches) (the valve stem and its cylindrical housing section).

6. The support strut thermal length is the entire length of the G-10 fiberglass tube (each strut is wrapped with a five layer MLI blanket).

7. An average cross-sectional area for the valve supports were used since its cross-section is not uniform.

Classical radiation theory was used to predict heat transfer through the MLI using a 300 K warm boundary temperature and a 20 K pressure vessel temperature. A performance degradation factor of 2.5 was used to account for blanket lay-up compression effects and penetrations. An effective blanket emissivity was calculated using a 0.042 surface emissivity for the doubled aluminized Mylar. In **Table 4-6**, the average tank heat flux represents the total tank heat load distributed evenly over the tank surface.

Table 4-6

DEPOT TANK HEAT LEAK SUMMARY

COMPONENT	HEAT LOAD WATTS	(BTU/hr)	PERCENT OF TOTAL
Valves (3)	3.71	(12.66)	56
Valve supports (3)	0.45	(1.54)	7
Plumbing lines	0.41	(1.40)	6
Instrument wiring	0.14	(0.48)	2
Support struts	0.13	(0.44)	2
MLI	1.74	(5.94)	27
	6.58	(22.46)	100

CS.833

Tank surface area 4.14 m^2 (44.6 ft^2)

Average heat flux 1.58 W/m^2 (0.50 Btu/hr-ft^2)

4.5.2 OTV Tank

The OTV is a bare, 83.8 cm (33-inch) diameter, near-spherical PV insulated with MLI. Its volume is 0.334 m^3 (11.8 ft^3) and it is designed to simulate the fuel tanks of a typical orbital transfer vehicle (OTV). The OTV tank, shown in Figure 4-9, consists of two hemispherical heads attached to a girth ring. The primary purpose for this tank is chilldown and no-vent fill testing; therefore, its primary design parameter is minimum mass-to-volume (M/V) ratio. Our current design has an M/V of about 224 kg/m^3 (1.4 lb/FT^3) which is ach.eved by chem-milling the two heads to an average thickness of approximately 0.635 mm (0.025 inches). Because of the thin walls, the heads will be fabricated from 5083 aluminum and used in the annealed condition. Thickened bosses will be located at each end to provide support and load

FOLDOUT FRAME /.

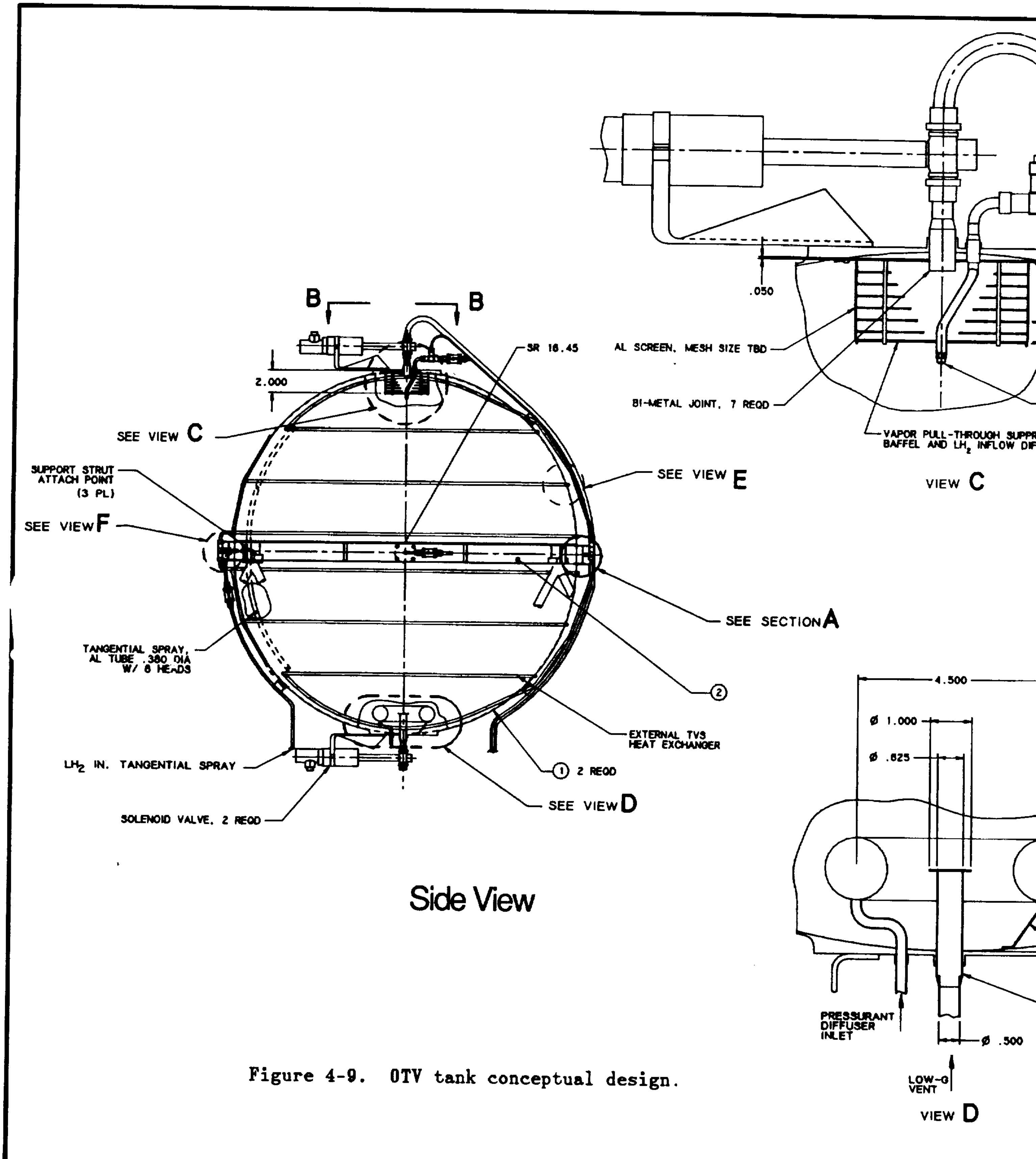
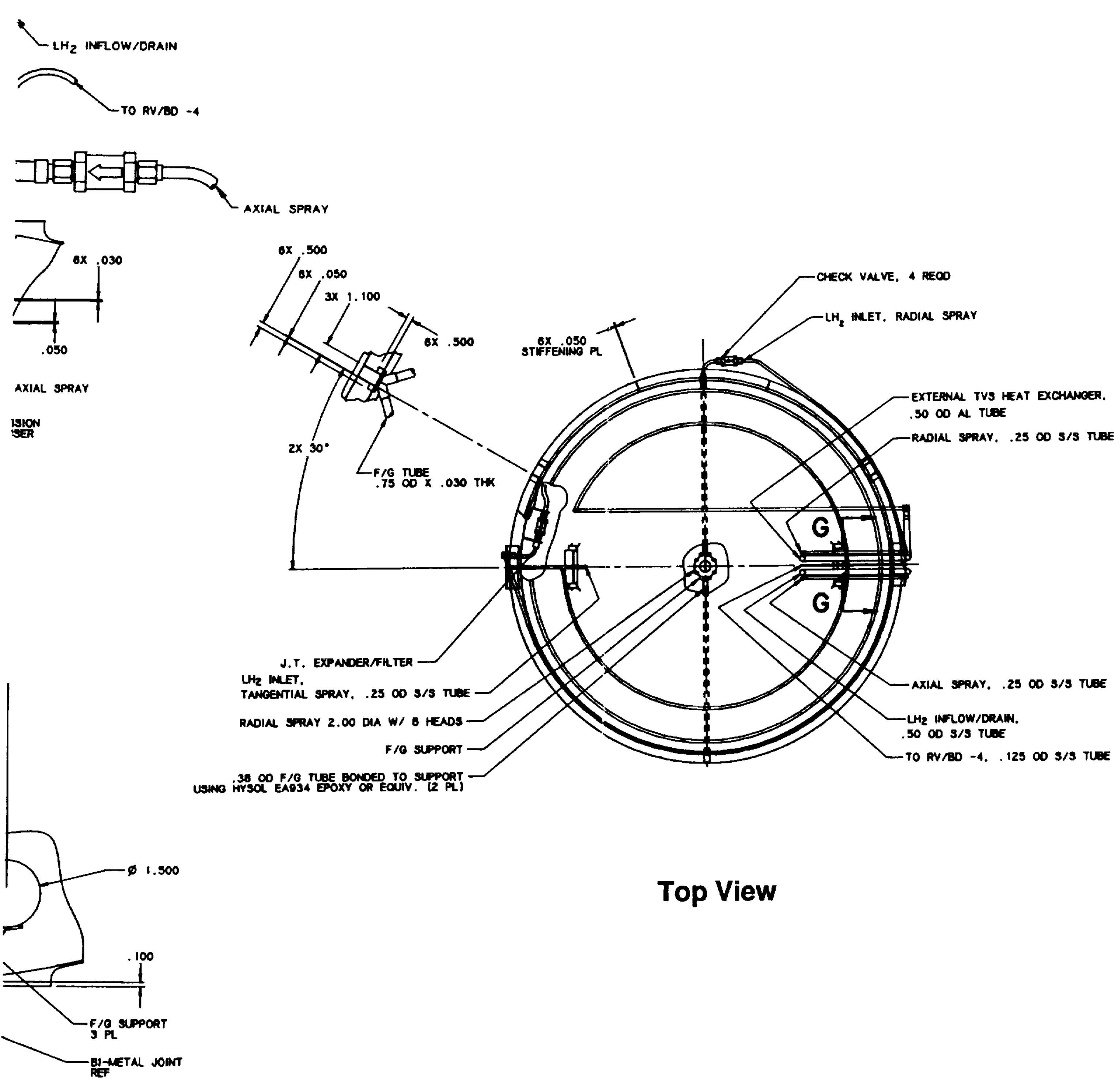


Figure 4-9. OTV tank conceptual design.



Top View

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paths for the plumbing feed throughs and cold valve supports. Its insulation system consists of a 3.8 cm (1.5 inch) 60 layer blanket of MLI and a TVS with an external wall mounted heat exchanger, consisting of 13.9 m (45.5 ft) of aluminum tube attached to the pressure vessel hemispheres. The girth ring and hemispheres are identical to those of the depot tank, and the support struts are similar. At the top of the tank is located a combination vapor pull-through suppression baffle and inflow diffuser for low-g LH_2 drain and fill (view C). The axial spray is attached to this assembly and oriented in the -Z direction. Located at the opposite end of the tank is a low-g vent and a pressurant diffuser (view D).

The inflow diffuser consists of six stacked circular plates with 0.25" gaps. Holes of decreasing size are centered in each plate to encourage equal flow of liquid through the sides. The top plate (furthest from the inlet) is solid to prevent direct flow of liquid toward the vent port. The stacked plate concept has been demonstrated successfully in drop-tower testing at LeRC.

Vapor pull-through suppression is obtained by wrapping the cylindrical section of the inflow diffuser with a pleated screen, which can have a relatively high bubble point without inducing significant flowing pressure drop. Since the top plate is solid, outflow will occur along the tank wall until the screen breaks down.

The axial spray nozzle is attached to the upper plate of the vapor pull-through suppression baffle and is fed by a liquid line which runs inside the open area of the inflow diffuser. Spray angle coverage is wide enough to contact the girth-ring area which represents the most massive section of the tank.

The toroidal design of the pressurant diffuser was selected to accommodate the low-g vent. As in the supply and depot tanks, a minimum flow velocity ratio of 500:1 is specified.

The low-g vent is a 7.9 mm (5/16 inch) diameter tube which penetrates approximately 2.5 cm (1 inch) into the tank. It has a 2.5 cm (1 inch) diameter rim at its inlet to prevent ingestion of liquid which will tend to flow up the tank walls.

The same general design requirements and constraints were used for the OTV tank instrumentation tree as were used for the supply tank instrumentation tree. Its attachment and primary support is the fiberglass tube which supports the radial spray.

4.5.2.1 OTV Tank Structural Analysis

Structural requirements and analysis methods for the OTV tank were the same as those for the depot tank. The OTV tank structural analysis is summarized in **Table 4-7**. Unlike the depot tank, the OTV tank has internal plumbing, tangential and radial sprayers. Their analysis predicted positive margins but low modal frequencies, which are not a concern since their low masses will not significantly effect the spacecraft modal frequencies.

Table 4-7
OTV TANK STRUCTURAL ANALYSIS SUMMARY

COMPONENT	LOAD CONDITION	MARGIN OF SAFETY	PREDICTED FREQUENCY (Hz)
Pressure Vessel	Internal Pressure Shell Load: Valve Shell Load: Launch	+0.01 YLD +0.63 YLD +3.08 Buck	700
Girth Ring	Launch Loads	+7.75 YLD	N/A
Internal Plumbing			
Tan Spray	Launch	+1.70 YLD	74
Rad Spray	Launch	+0.05 ULT	15
External Plumbing	Launch	+0.22 to +1.27 YLD	87 to 300
Instrumentation Tree	Launch	+0.04 ULT	7.3
Support Struts	Launch	+0.24 Buck	53 Lateral 120 Axial
Valve Bracket	Launch	+5.11 ULT	114

CS.751

4.5.2.2 OTV Tank Thermal Analysis

Thermal analysis for the OTV tank is summarized in **Table 4-8**. Assumptions made in calculating thermal area-to-length ratios, OTV heat leak, and average heat flux were the same as these for the depot tank (see Section 4.5.2.1).

Table 4-8
OTV TANK HEAT LEAK SUMMARY

COMPONENT	HEAT LOAD WATTS	(BTU/hr)	PERCENT OF TOTAL
Valves (2)	2.47	(8.43)	57
Valve supports (2)	0.30	(1.02)	7
Plumbing lines	0.41	(1.40)	9
Instrument wiring	0.10	(0.34)	2
Support struts	0.13	(0.44)	3
MLI	0.98	(3.34)	22
	4.39	(14.97)	100

CS.834

Tank surface area $2.34\ m^2$ ($25.2\ ft^2$)
Average heat flux $1.86\ W/m^2$ ($0.59\ BTU/hr\ ft^2$)

4.6 LAD DESIGN AND ANALYSIS

4.6.1 LAD Design

The liquid acquisition device (LAD) is designed to guarantee vapor-free liquid delivery from the tank. For COLD-SAT, a screen channel LAD was chosen for the supply and the depot tanks, and their design requirements and features are given in **Table 4-9**.

Table 4-9

LAD DESIGN REQUIREMENTS AND FEATURES

REQUIREMENTS	FEATURES
Supply LAD	
• Reliable, high-performance delivery of LH_2 at 0-500 lbm/hour, 10^{-3} - 10^{-6} g, and with warm pressurant environment	• All aluminum design; 200 x 1400 weave aluminum screen; triangular channels close to walls
• Fill remotely and remain full during launch (6 g)	• Vent; no more than 3 in. exposed screen during launch
Depot LAD	
• Reliable, high-performance delivery (<2% residuals) at less than 450 lbm/hour and with warm pressurant environment	• Aluminum channels; finer than 10 x 52 weave aluminum screen; triangular channels close to walls; (use of GHe and GH_2 pressurant)
• Intermediate expulsion efficiency at 450-500 lbm/hour and 10^{-3} g	• 10 x 52 weave screen
• Allow for evaluation of fill techiques - Direct fill/tank fill - LAD TVS fill - LAD Vent fill	• TVS on LAD; vented channels; instrumentation inside channels

The supply LAD is required to support all COLD-SAT supply requirements with an expulsion efficiency greater than 97 percent. It must be filled prior to and remain full through launch, which requires a vent at the top of the LAD for initial filling and a fill level resulting in no greater than 3 in. exposed screen during launch (6 g's).

The depot LAD is used for fill and expulsion experiments. It must provide for reliable performance at nominal conditions and for intentional breakdown at maximum out-flowrates and an adverse acceleration of 10^{-3} g. Fill experiments require a LAD vent and a TVS thermally coupled to the LAD along the length of its channels. Instrumentation includes temperature sensors inside and on the LAD channels, liquid-vapor detectors and pressure sensors.

The channel screen surface through which bulk fluid is acquired is adjacent to the wall and is supported by a beaded perforated plate. During assembly, the channel side walls of a single segment are first formed with two pieces of aluminum welded together. For the depot LAD, all the segments of each channel leg are then connected and the TVS tube welded along the apex. The beaded plate is welded to the side walls along the length of the channel. The screen bottom will be made of a series of screen patch assemblies approximately 13 cm (5 inches) long each. Each screen patch will be lined on its edges with a ribbon of 0.25 mm (0.01 inch) aluminum which extends approximately 6.35 mm (10.25 inch) outside its perimeter. This ribbon allows for welding of the patch to the side walls and to each other. The patches are attached to the channel one at a time.

4.6.1.1 Supply Tank LAD Design

The supply tank LAD design, shown in **Figure 4-10**, is made of aluminum to eliminate the effects of differential contraction. It consists of four channels along the length of the tank joined with a toroidal manifold at the bottom and support structure at the top. The toroidal manifold has a 20.3 cm (8.0-inch) internal diameter in order to accommodate the supply tank mixer. All joints are mitered and welded; flanged joints were eliminated as they represented dead space for the trapping of vapor and complicated flow paths. Support brackets are located at the girth rings for each channel and at the bottom of the manifold. The screen, an aluminum 200 x 1400 twilled double dutch weave (TDDW), was selected to preclude premature breakdown during the mission. This configuration allows the LAD to be built and tested as a separate subsystem prior to integration with the tank and other systems.

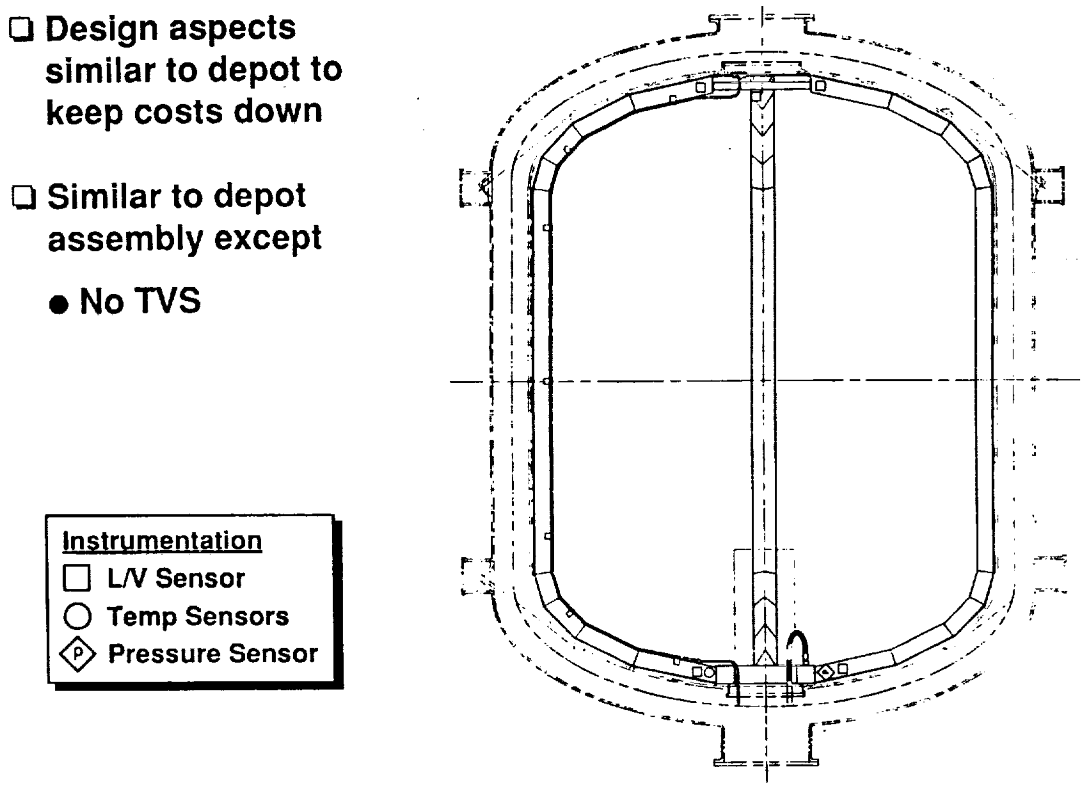
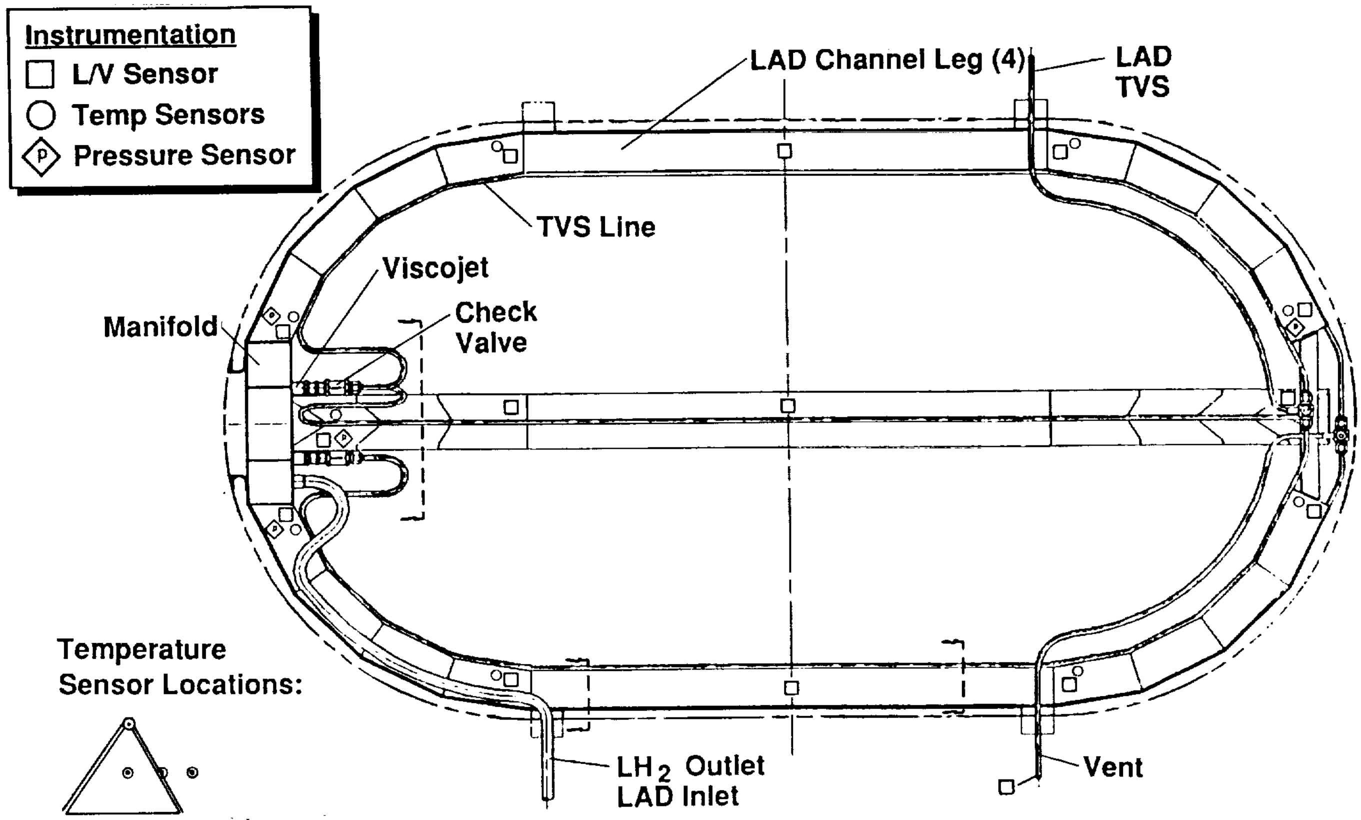


Figure 4-10. Supply tank LAD design.

4.6.1.2 Depot Tank LAD Design

The depot tank LAD, shown in **Figure 4-11**, is similar to the supply tank LAD. A TVS line is attached to the apex of each channel leg. At the top, the TVS lines are manifolded together before exiting the tank. Vent lines tap into the top of each of the legs, then join together into a single line which exits the tank and vents to space. Support brackets are located on the girth rings and manifold for each leg (they are rotated away from the channel and manifold to allow for assembly).



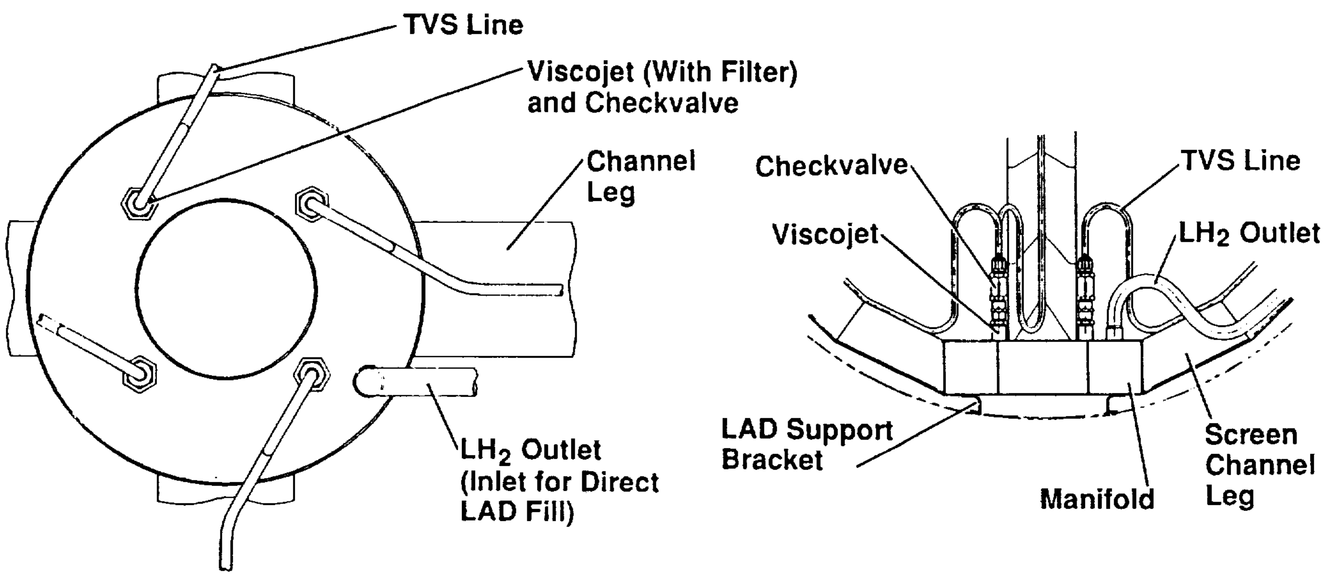


Figure 4-11. Depot LAD configuration and instrumentation.

Instrumentation includes 8 differential pressure sensors (2 per leg), 80 temperature sensors (20 per leg with 4 at each of the five locations indicated), and 16 liquid-vapor sensors (4 per leg).

The depot LAD manifolding is shown at the bottom of **Figure 4-11**. The toroidal manifold has a square cross-section to facilitate fabrication and a 10.2 cm (4.0 inch) internal diameter to accommodate the pressurant diffuser and axial spray. The TVS lines for each channel leg are fed from individual J-T expanders (Lee Viscojet) connected to the toroidal manifold. The viscojet includes an upstream filter and is followed by a checkvalve to preclude backflow of vapor produced by warming of the TVS lines.

The LAD channel cross-section and sensor mounting concept are shown in **Figure 4-12**. The internal sensor probes are mounted through a port in the channel side wall. The sensor support and wiring are supported by a 6.4 mm (10.25 inch) dia. by 12.7 mm (0.5 inch) long tube which is part of the sensor mounting assembly. The sensor mounting assembly is attached to the sidewall of the LAD channel with screws and sealed with an indium o-ring.

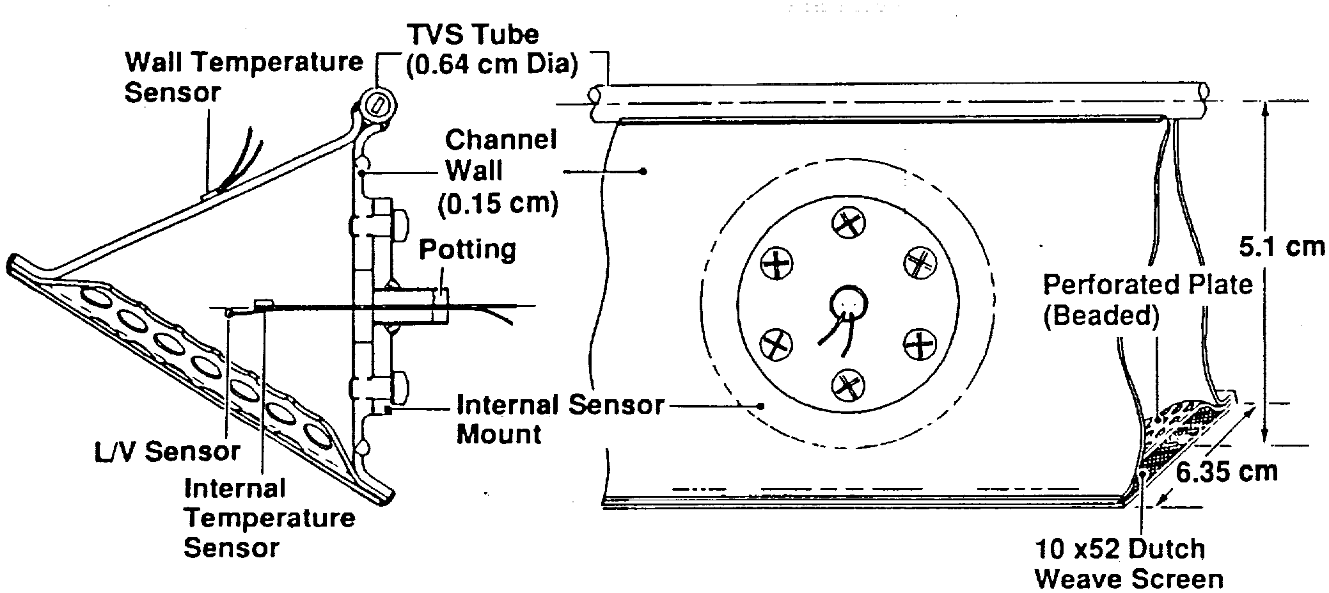


Figure 4-12. Depot LAD channel design and L/V sensor mount.

4.6.2 LAD Analysis

LAD fill testing is to demonstrate and evaluate the effectiveness of the fill processes under expected operational environments (<10-4 g's). Two approaches will be evaluated: direct venting of the LAD and condensation of LAD contents using a TVS heat exchanger. Two sources of LH_2 will be used during LAD fill: (1) LH_2 introduced directly into the LAD through the LAD outflow line (direct fill), and (2) LH_2 contained in the tank fills the LAD channels by flowing through the screen (tank fill).

The objective of expulsion experiments is to demonstrate the expulsion efficiency performance of LADs and to stress a LAD under high outflow and g conditions to characterize its breakdown.

4.6.2.1 LAD Filling

In a direct fill, LH_2 flows directly into the LAD channels after the LAD vent is opened. LH_2 will tend to flow through the screen into the tank and capillary forces will encourage filling of the gap between the channel and the tank wall. Because gap dimensions are smaller than channel dimensions, the gap will tend to fill faster than the channel. The channel-to-wall gap and the channel liquid columns will tend to grow according to the capillary pressures defined by their geometry and the acceleration level. The net difference between the channel and the gap for the depot tank LAD design is 14.4 Pa (0.3 lb_f/ft^2). This translates to a 2.1 m (6.8 ft) static head at 10^{-4} g's; i.e., a liquid column in the gap which can be supported above the channel liquid level. Consequently, most of the liquid entering the channel goes toward filling of the gap and wickover occurs with a channel fill level of approximately 20 percent. The same considerations render refill (of a partially full tank) by LAD venting unproductive as the channel is likely to be wicked over at the start of the refill.

For the TVS LAD fill, the tank is full of liquid including the gap between the channel screen and tank wall. The LAD channels are full of vapor, and maybe some liquid. The TVS fill process, shown in **Figure 4-13**, starts at Time 1 when the TVS flow is initiated cooling the LAD channels. The cooling results in condensation of vapor inside the channel, as indicated by the cross-section shown at Time 2. This condensation results in a drop in pressure inside the channel causing the outside liquid to flow through the screen and into the channel. TVS cooling continues until the channel is filled with liquid, although as the process progresses, conduction through the liquid layer tends to slow the process.

A nodal model of heat transfer in the LAD channel was constructed to evaluate the channel design and the TVS fill concept. **Figure 4-14** summarizes fill times for different TVS flowrates. The design flowrate corresponding to the total tank TVS flowrate prescribed by pressure control experiments is indicated in the figure.

4.6.2.2 LAD Expulsion

The ability of screen channels to function as liquid acquisition devices is based on their ability to remain filled (liquid retention) even when not completely submerged in the bulk fluid of the tank. Upon channel exposure to the ullage, a liquid-vapor interface is established at the screen due to surface tension. This interface has the capability to resist the passage of vapor into the channel (i.e., withstand a pressure drop from the ullage to the inside of the channel). The pressure capability of the interface is defined by the bubble point (denoted as a pressure drop), which is characterized by the liquid surface tension and the screen pore size. When the pressure difference across the LAD exceeds the bubble point, the liquid-vapor interface "breaks down" allowing vapor to pass into the LAD which generally terminates liquid acquisition.

The supply tank screen is designed not to break down throughout the mission over the range of outflow; however, the depot tank LAD has a very low bubble point screen [(10x52 Plain Dutch Weave aluminum screen with bubble point of

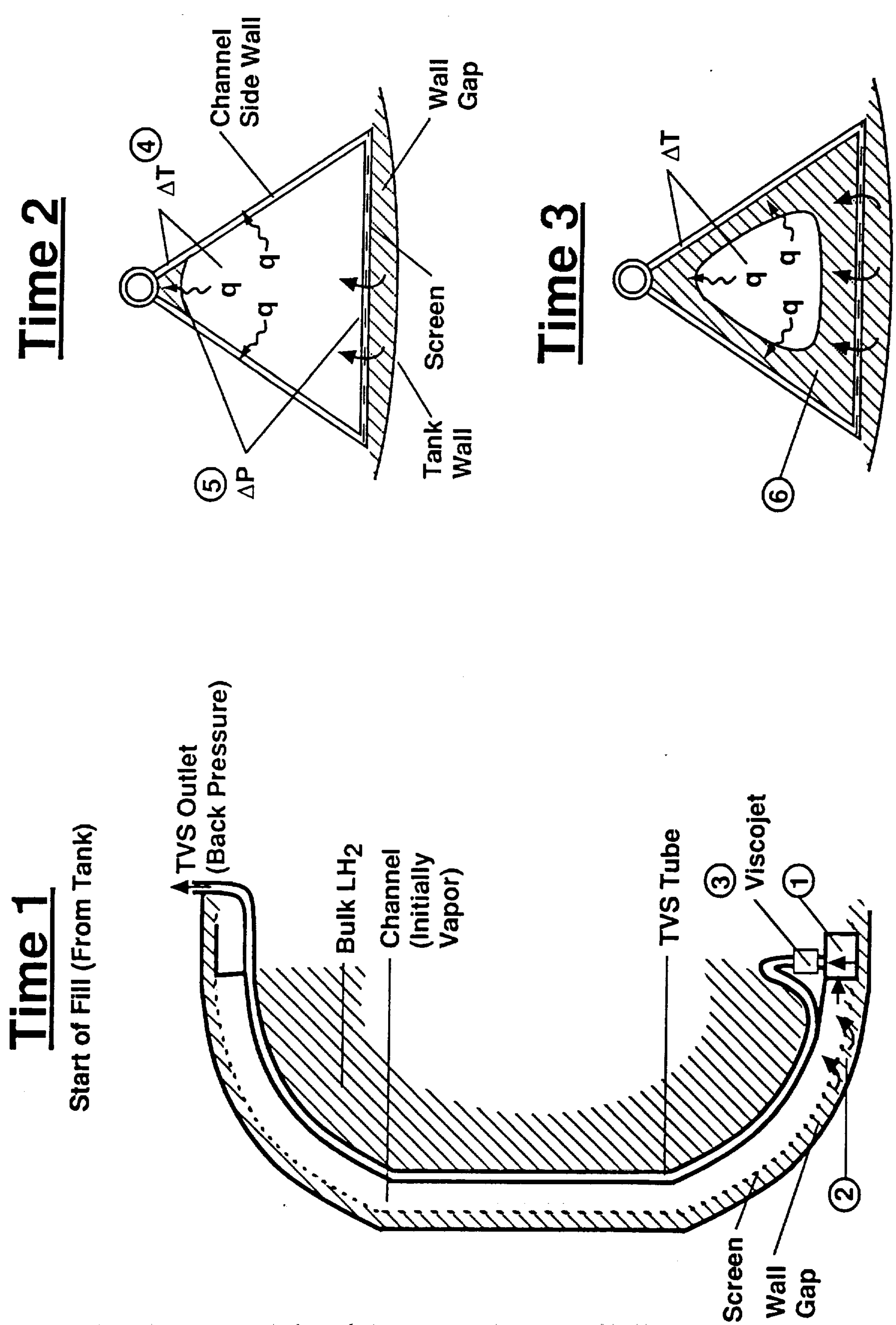


Figure 4-13. LAD fill processes, LAD TVS fill.

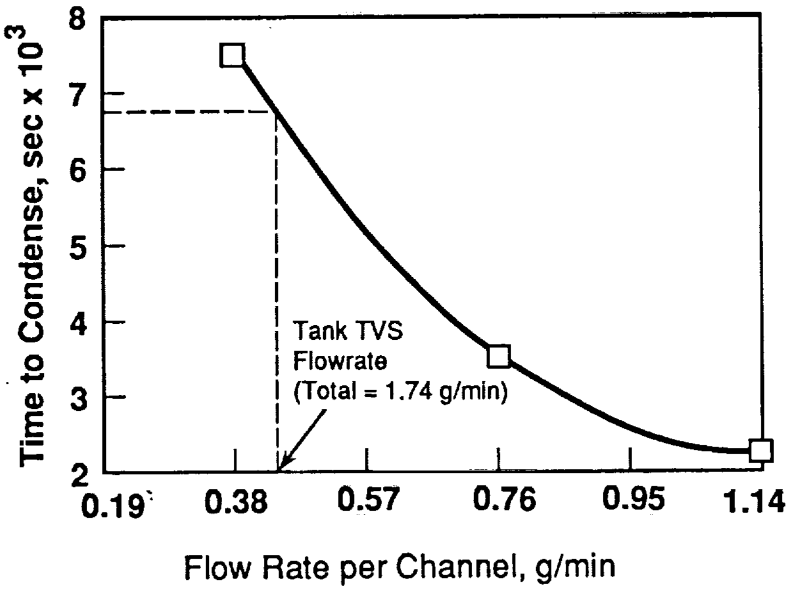
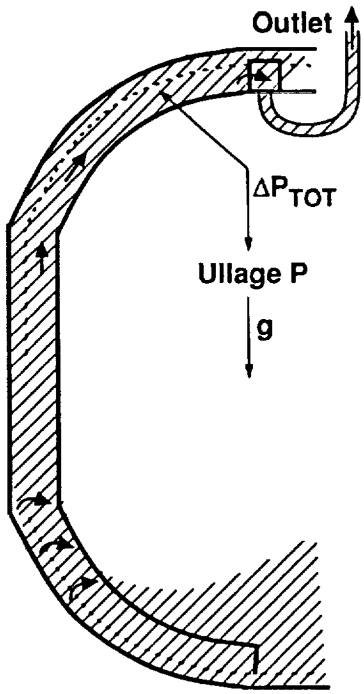


Figure 4-14. LAD fill time vs. TVS flow rate.

approximately 14 Pa, (0.3 lbf/ft^2)] to allow it to break down within experimental conditions. Its design limits operation below 227 kg/hr (500 lbm/hour) mass flow rate and 10^{-3} g's. Dynamic losses are a constant throughout the expulsion and frictional losses are negligible; but, hydrostatic and screen flow-through losses begin to grow at approximately 20 percent expulsion. As more channel is exposed, the screen flow-through area decreases, increasing in screen losses. The screen flow-through losses rise significantly toward the end of the expulsion, eventually exceeding the bubble point, which results in LAD breakdown, the subsequent detection of vapor in the channel (or the tank outlet), and expulsion termination. **Figure 4-15** summarizes the depot LAD expulsion analysis; the top figure identifies the governing relationships, the lower graph shows the different contributions to LAD pressure drop during expulsion.

4.7 TVS DESIGN AND ANALYSIS

A thermodynamic vent system (TVS) is an open-loop refrigerator which provides a means of controlling tank pressure at any g-level while venting vapor only. It extracts liquid from the tank, expands the liquid isenthalpically to a lower pressure and temperature, and routes the resulting two-phase flow



- LAD functions as long as $\Delta P_{TOT} < \Delta P_{BP}$; at breakdown $\Delta P_{TOT} \geq \Delta P_{BP}$
- $\Delta P_{TOT} = \Delta P_d + \Delta P_h + \Delta P_s + \Delta P_f + \Delta P_c$

  ΔP_c → turning
  ΔP_f → friction
  ΔP_s → screen flow through
  ΔP_h → hydrostatic
  ΔP_d → dynamic

$$\Delta P_d = \frac{\rho\, v^2}{2\, g_c}$$

$$\Delta P_h = \frac{\rho\, g h}{g_c}$$

$$\Delta P_s = \rho\left(A\,V + B\,V^2\right)$$

A, B → empirical constants

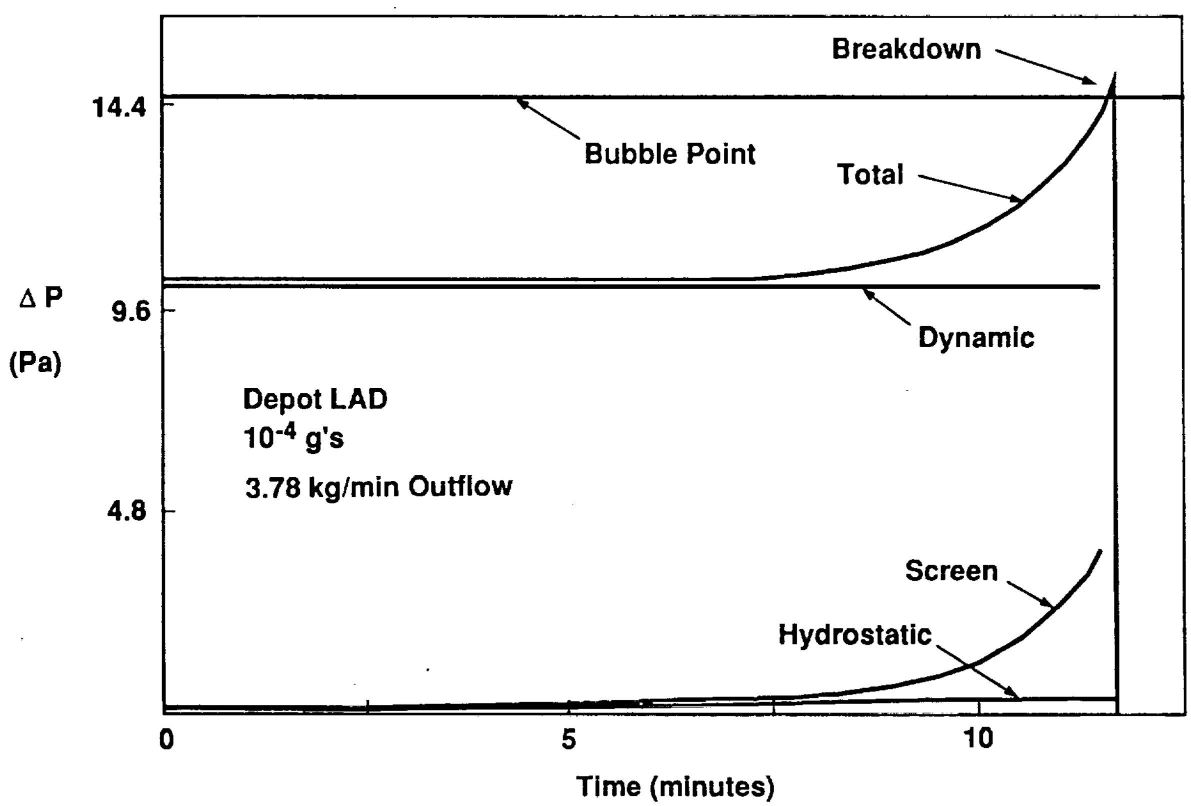


Figure 4-15. LAD expulsion analysis.

through a heat exchanger in contact with the bulk tank fluid. Heat transferred from the tank fluid to the TVS vaporizes the TVS fluid. The heat exchanger is sized to completely vaporize the TVS fluid, thus removing heat from the tank and reducing tank pressure while insuring that only vapor is vented.

Requirements and operating characteristics for the COLD-SAT TVS's are summarized in **Table 4-10**. The supply tank has two systems, one active and one passive; the depot and OTV tanks each have a passive system.

Table 4-10

TVS DESIGN REQUIREMENTS AND OPERATING CHARACTERISTICS

ITEM	REQUIREMENT	OPERATING CHARACTERISTICS	COMMENT
1. Supply tank active TVS heat exchanger	• dP/dt = - 8.62 kPa/hr at 22 W heat load • Compact heat exchanger on mixer suction side	• 171 W capacity • 24.6 g/min vent flow • Counterflow, helical paths	HX area 100% over-sized to insure no liquid is vented
2. Supply tank active TVS mixer	• region I and IV mixing • D tank/D jet ≅ 25 • Maximum power = 2 W	• Variable flow rate: 0 to 0.11 m^3/min • Design point 0.065 m^3/min with Δp = 0.2 kPa	Permanent magnet brushless DC motor
3. Supply tank passive TVS	• 7.3 W capacity • Distributed heat exchanger	• 1.06 g/min vent flow • 26.5 m of 1.27 cm diam finned tube	HX area 100% over-sized
4. Depot tank TVS	• 12.9 W capacity • Heat exchanger integral with LAD channel	• 1.81 g/min vent flow • 5.8 m of 0.635 cm tube on LAD channels • 21 m of 0.794 cm tube external wall mounted	Internal channel plus external wall wrap (internal flow split four ways) HX area 100% over-sized
5. OTV tank TVS	• 8.8 W capacity	• 1.29 g/min vent flow • 14 m of 0.794 cm tube, external wall mounted	HX area 50% over-sized

The supply tank active TVS consists of a compact heat exchanger and mixer. It is designed to reduce tank pressure at a rate of 8.6 kPa/hr (1.25 psia/hr) under maximum supply tank heat flux. The passive TVS consists of a distributed, finned tube heat exchanger. It is required to remove 7.3 W (25 BTU/hr), twice the heat load derived from a background heat flux of 0.3 W/m^2 (0.1 $BTU/hr\text{-}ft^2$).

The depot tank TVS consists of four internal tubes, one tube attached to each LAD channel, feeding into a single external heat exchanger tube wrapped around the tank wall. It is required to remove 12.9 W (44 BTU/hr), twice the background heat load. It must also cool the LAD channel, as well as the bulk tank fluid, to condense any vapor during the LAD fill operations.

The OTV tank TVS consists of a heat exchanger tube wrapped around the external tank wall. It is required to remove 8.8 W (30 BTU/hr), twice the background heat load.

4.7.1 Supply Tank Active TVS Concept

The supply tank active TVS consists of a compact counterflow heat exchanger and an axial jet mixer. It is mounted on a support at one end of the tank and is thermally isolated from the tank and LAD structure. The design requirement for the heat exchanger is to provide a pressure reduction rate of 8.6 kPa/hr (1.25 psi/hr) with a maximum thermal control shield heat load of 22 W (75 BTU/hr). The requirement for the mixer, which pulls the tank bulk fluid through the heat exchanger warm-side flow path, is to provide region I and region IV mixing (see the ERD for details). The ratio of tank diameter to jet exit diameter is approximately $D_{tank}/D_{jet} = 25$ to provide similitude with past experiments.

Temperature sensors are located at the inlet and outlet of each heat exchanger flow path to measure heat exchanger effectiveness. Pressure sensors are located upstream of the J-T valve and at the vent outlet to monitor the performance of the J-T valve and determine the pressure drop through the cold-side (vent) flow path. Mixer electronics will monitor RPM, current and power. Mixer RPM is related directly to flow rate by preflight calibration, so a separate flow meter is not required. Mixer current will indicate whether the mixer is immersed in liquid or vapor, and mixer power indicates how much heat is being introduced into the tank by the mixer.

4.7.1.1 Heat Exchanger Design

The compact counterflow heat exchanger (Figure 4-16) requires cold side (vent) flow rates of 24.6 g/min (3.25 lb/hr) and axial jet flows of 0.065 m^3/minute. The common wall area is formed into a cylinder with the flow paths wrapped around it in a helical geometry. This cylindrical shape results in a compact configuration while the helical flow paths provide an artificial g-field to insure that the walls of each flow path remain wetted (the heat transfer coefficient correlations assume wetted walls).

- Flow rates:
  - Cool-side = 24.6 g/min (3.25 lb/hr)
  - Warm-side = 0.065 m^3/min (2.3 cfm)
- Geometry: cylindrical common wall separating helical flow paths
- Dimensions:
  - External diameter = 20.3 cm (8 in)
  - Common wall length ~ 25.4 cm (includes 100% margin)
  - Cool-side flow path: 0.66 cm x 0.66 cm (0.26 in x 0.26 in)
  - Warm-side flow path: 3.05 cm x 2.5 cm (1.2 in x 1.0 in)
- Heat transfer coefficients:
  - Cool-side: h_i = 1.59 kW/m^2-K (280 BTU/hr-ft^2-F)
  - Warm-side: h_o = 2.55 kW/m^2-K (450 BTU/hr-ft^2-F)
  - Overall: U = 0.96 kW/m^2-K (170 BTU/hr-ft^2-F)

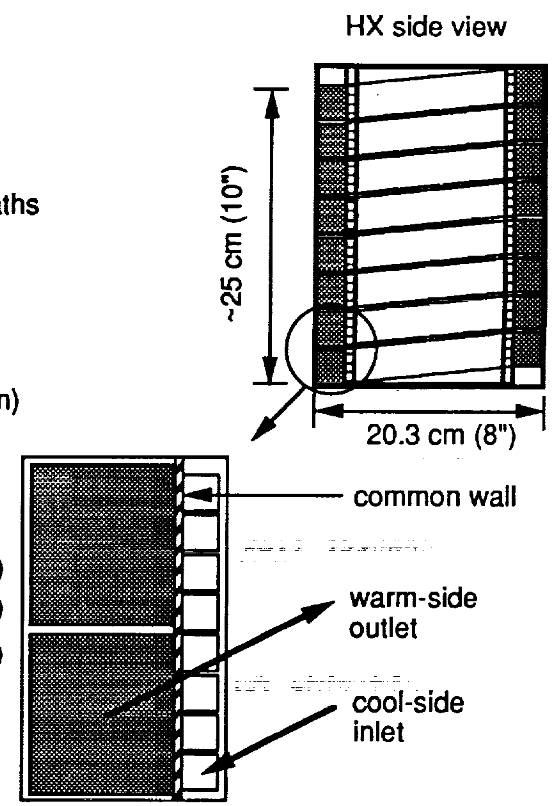


Figure 4-16. Supply tank active TVS heat exchanger design.

The heat exchanger is 25.4 cm (10 inches) long with a common wall diameter of 14.2 cm (5.6 inches) and an exterior diameter of 20.3 cm (8 inches). The cold-side flow path, located on the inner surface of the common wall, has a square cross-section 0.66 cm (0.26 inches) on a side. The warm-side flow path, on the outer surface of the common wall, has a rectangular cross-section of 3.0 cm by 2.5 cm (1.2 inches by 1.0 inches). It is sized for steady-state operation at 171 W (585 BTU/hr) with the tank fluid saturated at 103 kPa (14.7 psia). The analysis assumed the J-T valve dropped the pressure by 90 kPa (13 psia) to 34.5 kPa (5.0 psia), producing a two-phase fluid with a quality (vapor mass fraction) of about 7 percent. Once sized, the heat exchanger common wall area was doubled to provide margin for uncertainties in the heat transfer (to insure no liquid would be vented). The pressure drops in flow path were then checked to make sure they remained below their allowable limits.

4.7.1.2 Supply Tank Mixer

Based upon prior mixing experiments in one-g and low-g, a submerged axial flow jet, located at one end of the tank and directed along the centerline toward the opposite end of the tank, has proven to be an effective geometry for mixing tank fluid. A mixer whose outlet diameter is approximately 1/25th of the tank diameter provides a good balance between mixing efficiency (i.e., the total energy required to mix), mixing time and mixer weight. During active TVS operation, region IV mixing is desired at a single flow rate for each of the three fill levels to be investigated. In addition, it is also desired to investigate region I mixing at each fill level.

The mixer is an axial flow pump, driven by a variable-speed, brushless DC motor with electronic commutation. This configuration provides high efficiency in a compact design, along with the flexibility of software selectable speed control. Prior work in the late 1960's by AiResearch demonstrated this type of mixer operation in liquid hydrogen, with a flow rate of about 0.14 m^3/min (5 cfm), provides a reasonable baseline for COLD-SAT mixer sizing. **Figure 4-17** shows the supply tank mixer concept and lists some of its features.

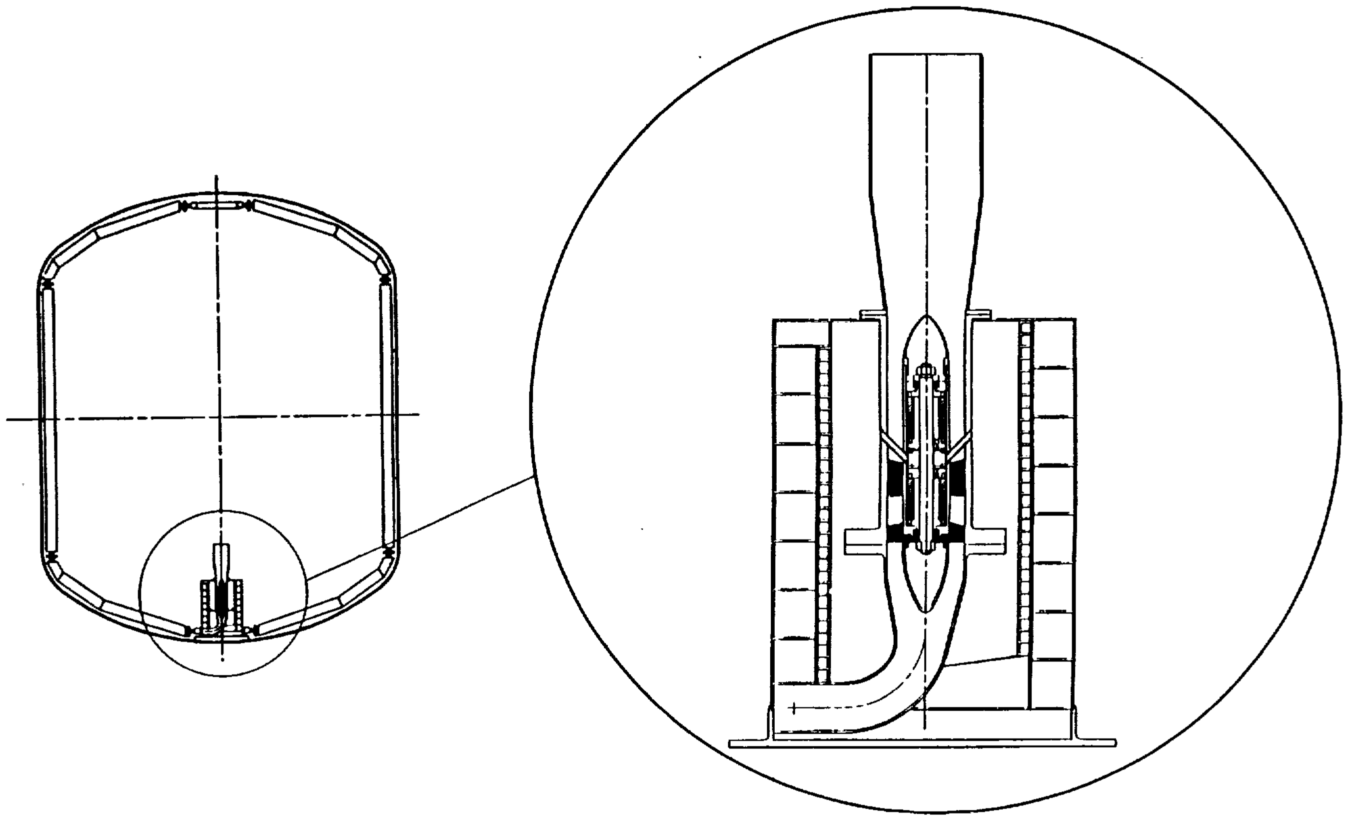


Figure 4-17. Supply tank active TVS concept.

Characteristics

- Axial flow jet along tank centerline
- D tank/D jet $\cong$ 25
- Region IV mixing at all fill levels
- Region I mixing capability at each fill level
- Permanent magnet brushless DC motor
- Maximum pressure rise = 0.5 kPa (0.07) psi

Instrumentation

- Hall sensors for shaft position and RPM
- Motor current and power

Mounting

- Downstream of HX

The mixer motor uses electronic commutation with Hall sensors mounted on the motor shaft to determine shaft position and RPM. Calibration done prior to flight relates RPM to flow rate, hence a separate flow meter is not required. The motor controller output is monitored to determine the current and power delivered to the motor.

The mixer is located at the outlet of the heat exchanger to provide subcooled liquid to the mixer inlet, which reduces the risk of cavitation. During destratification testing the mixer is operated without the TVS and the risk of cavitation increases, but the need for cavitation-free mixing during these tests is less critical.

4.7.1.3 Supply Tank Passive TVS Concept

The supply tank passive TVS (Figure 4-18) is a low-flow TVS designed to intercept the background heat flux. It has a capacity of 7.3 W (25 BTU/hr), twice the heat load derived from the background heat flux of 0.3 W/m^2. This capacity includes a margin of 100 percent to account for uncertainties in the free convection heat transfer correlations used to predict its performance. This heat exchanger is distributed along the cylindrical section of the tank using a helical wrap which sets up an artificial g-field within the tube to help keep the internal walls wetted. The TVS mounts to the LAD using insulating supports so that the effective area for heat transfer is known.

The heat exchanger is a finned tube made of 6063 Aluminum and was sized based upon a vent flow rate of 1.06 g/min (0.14 lb/ hr). The length of the heat exchanger is 26.5 m (87 feet); consisting of 6 wraps with a coil spacing of approximately 20.3 cm (8 inches). Diode temperature sensors are located every 3.14 radians (180 degrees), two per wrap, for a total of 12. These temperature sensors will detect the transition between two-phase flow and superheated vapor, thus measuring the effective heat exchanger length to vaporize the given vent flow. To monitor the performance of the J-T valve and determine the pressure drop through the passive TVS, pressure sensors will be located upstream of the J-T valve and at the vent outlet.

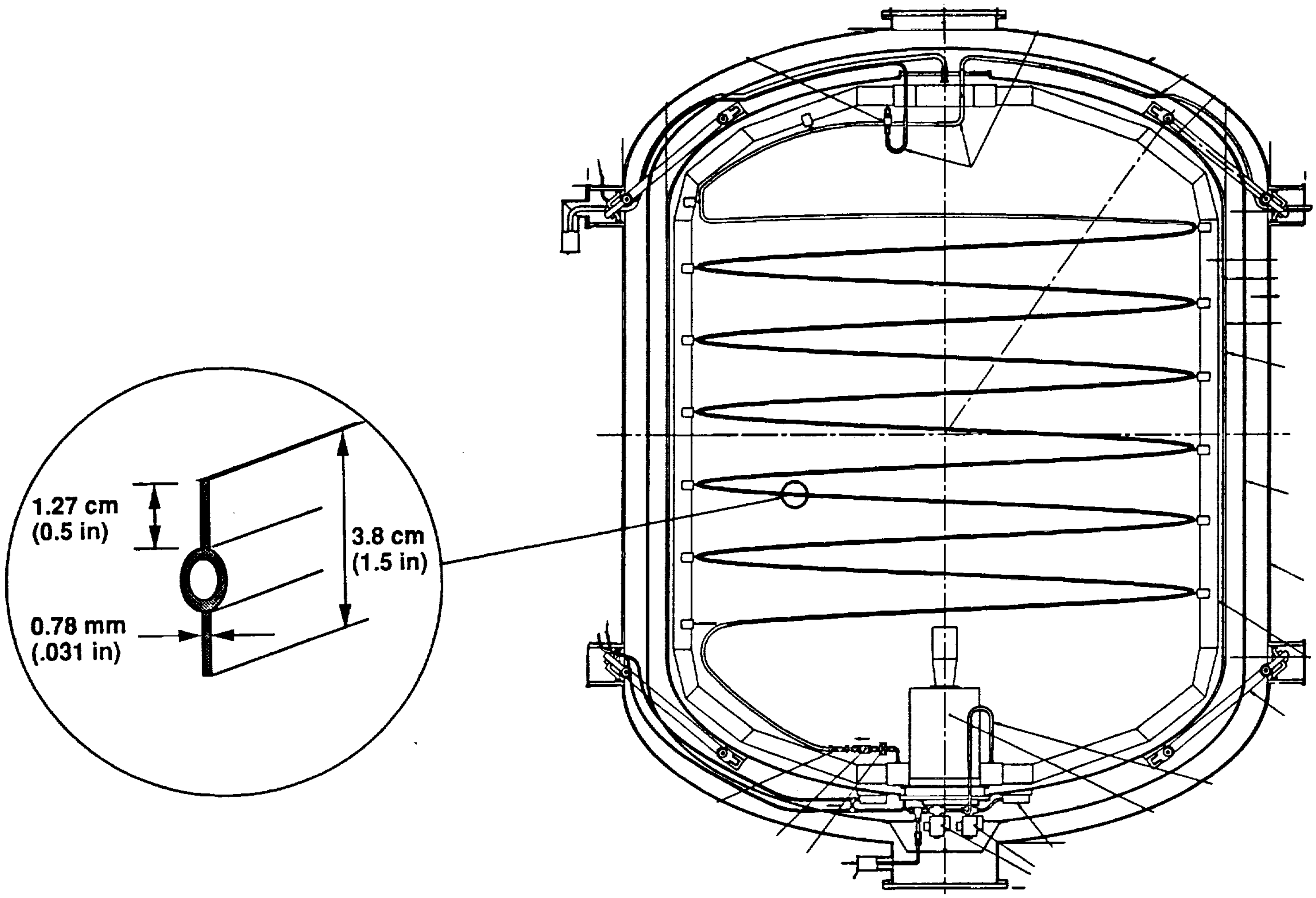


Figure 4-18. Supply tank passive TVS design concept.

4.7.1.4 Depot Tank TVS Concept

The depot tank TVS (**Figure 4-19**) is a low flow passive TVS designed to remove 12.9 W (44 BTU/hr), twice the heat load imposed by the background heat flux of 1.6 W/m² (0.5 BTU/hr-ft²). It serves two purposes: (1) to cool the tank bulk liquid, and (2) to condense vapor in the LAD channels. Therefore, the TVS is an integral part of each LAD channel and is limited to 19 feet of length inside the tank. To provide redundancy and to avoid parallel flow instabilities, each LAD/TVS heat exchanger is independently supplied by its own J-T valve. The TVS, a 0.635 cm (0.25 inch) O.D. tube attached to the apex of the LAD channel, is limited to the LAD length internally; therefore, additional heat transfer area required to meet the heat load requirement is provided by a wall mounted heat exchanger. This heat exchanger is made by joining the internal TVS tubes into one 0.794 cm (5/16 inch) O.D. tube which wraps around the cylindrical portion of the tank.

Characteristics:

- Capacity = 2 x heat load = 12.9 W (44 BTU/hr)
- Four independent LAD/TVS channels inside tank (5.8 m total)
  - Tube size = 6.35 mm O.D. x 0.71 mm wall (0.25 in. O.D. x 0.028 in. wall)
- External wall mounted TVS as needed for remaining heat load
- Material = 6063 aluminum

Instrumentation:

- Pressure sensors upstream of J-T valve and at vent outlet
- Temperature diodes distributed along each channel and at hx inlet and outlet

Mounting:

- Internal portion: integral part of LAD channel
- External portion: wall-mounted on cylindrical section of tank

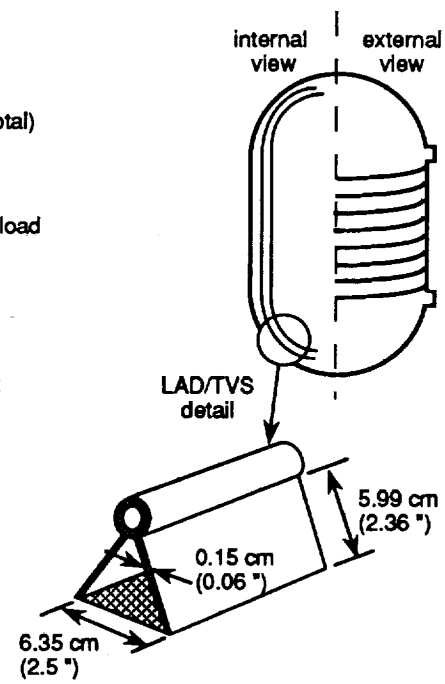


Figure 4-19. Depot tank TVS concept.

Based on sizing analyses with the given LAD/TVS geometry and a heat transfer coefficient of 2.2 W/m^2-K (0.4 BTU/hr-ft^2-F), the four legs of the TVS inside the tank will remove a total of 5.9 W (20 BTU/hr). With an external heat exchanger length of 21 m (69 feet), 8 wraps, spaced 7.6 cm apart) wrapped around the cylindrical section of the tank wall, an additional 7.0 W (24 BTU/hr) of heat is removed from the tank fluid.

4.7.1.5 OTV WALL TVS

The OTV thermodynamic vent system (TVS) is sized for 8.8 W (30 Btu/hr), twice the background OTV heat leak. The inlet side of the J-T is located next to the pressure vessel wall to promote liquid delivery to the TVS. Fluid conditions just downstream of the J-T are 34.5 kPa (5 psia) saturated at a quality of 7 percent. The flow exits the tank through a penetration in the girth ring and splits with half of the total flow routed to the upper and half to the lower hemisphere wall mounted heat exchangers. Each heat exchanger half consists of 6.9 m (22.8 feet) of 7.94 mm x 1.27 mm wall (5/16 x 0.050 inch) aluminum tube welded to bosses machined into the hemispheres. The two flow lines are then combined and are routed to the 10 kPa (1.5 psia) vent system. The OTV TVS was analyzed (using 2-phase flow heat transfer correlations) to verify that the wall mounted heat exchanger design vaporizes all of the vent flow.

4.8 INSTRUMENTATION

Table 4-11 lists the transducers chosen for each type of measurement on COLD-SAT along with the required range and accuracy, vendor, and part number. Diodes were chosen as the primary temperature sensors for COLD-SAT because they provide temperatures over the entire range of interest (20 to 300 K), they are lightweight, and they dissipate very little power. Pressure sensors are distributed throughout the liquid hydrogen subsystem (see **Figure 4-20**) so that each tank has a minimum of two and each line segment has at least one. Velocimeters are used to measure vent flow (vapor), and turbine meters are used to measure liquid flow rates.

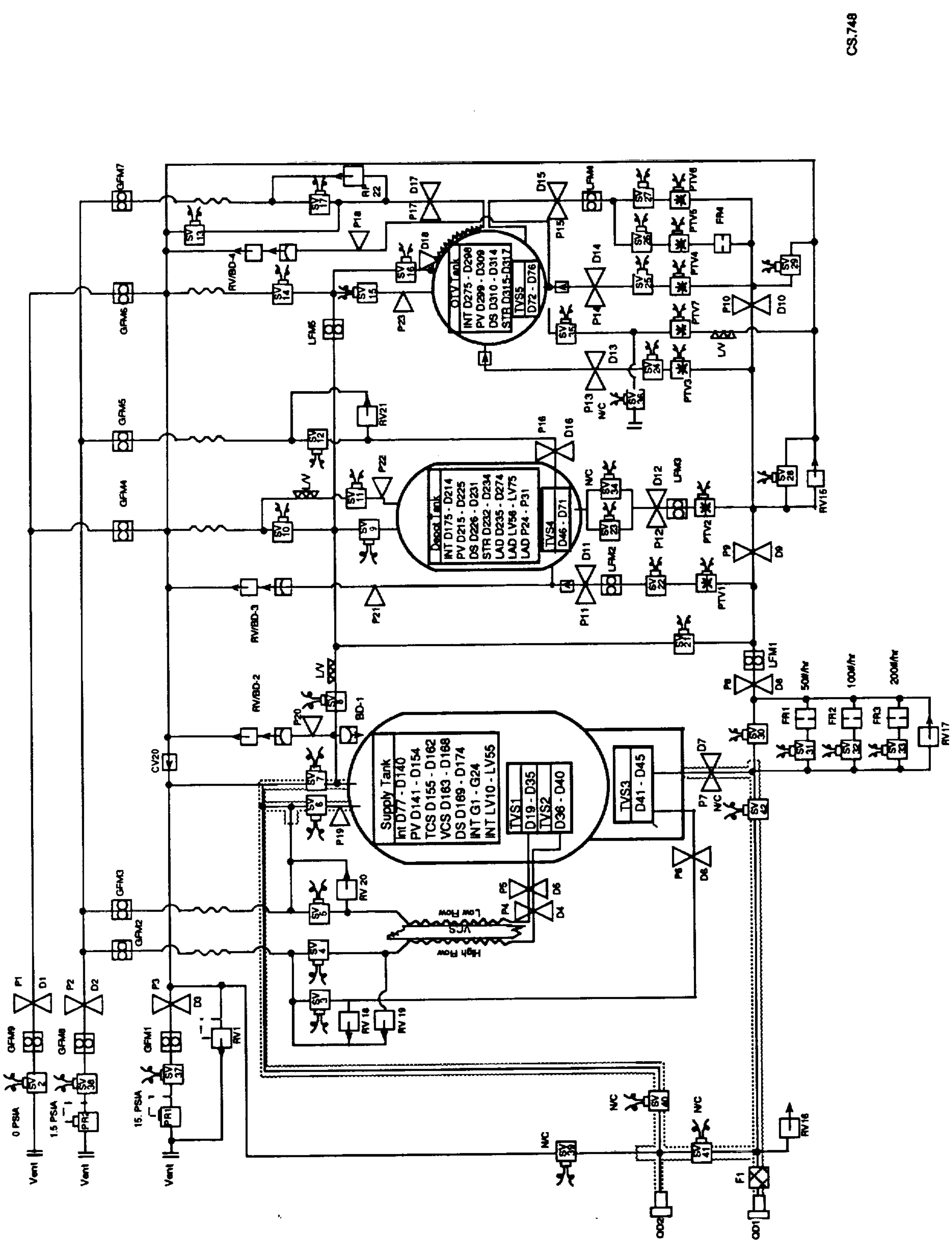


Figure 4-20.   COLD-SAT experiment instrumentation schematic.
(See Figures 4-2 and 4-5 for symbol definition)

Table 4-11
COLD-SAT TRANSDUCER SUMMARY

SENSOR	TYPE	RANGE	DEVICE ACCURACY	OVERALL ACCURACY	VENDOR	PART NO./MODEL
Temperature	Diode	15-30K (27-54R)	±0.05K (±0.1 R)	±0.1K (±0.2 R)	Lakeshore	DT-470-SD-11
	Germanium	15-30K (27-54R)	±.01K (±.02 R)	±0.05K (±0.1 R)	Lakeshore	GR-200A-2500-4B
Pressure	Strain Gauge	0-28 MPa (0-4,000 psia)	±0.5%	±1.0%	Teledyne-Taber	Model 2403
	Strain Gauge	0-350 kPa (0-50 psia)	±0.3 kPa (±0.05 psia)	±0.7 kPa (±0.1 psia)	Teledyne-Taber	Model 2215 LT
Flow	Velocimeter	0-5 gm/sec (0-40 lb/hr)	±1.0%	±2% FS	Fluid Components Inc.	Model 009460
	Turbine	2.5 - 25 gm/sec (20-200 lb/hr)	±1.0%	±2% FS	EG&G Flow Technology	Model FT 8-8
Accelerometer	Proof-Mass	1-1000 µg	±1 micro-g	±1 µg	Bell Aerospace	"Mesa" Model
Liquid-Vapor	Carbon Resistor	Liquid - Vapor	N/A	NA	TBD	TBD

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The experiment instrumentation concept was developed by combining measurement requirements,* system monitoring requirements, and redundancy requirements with the physical system configuration (tank shape and size, etc.). The COLD-SAT experiments require extensive temperature and pressure instrumentation in tanks and lines in order to accurately determine thermal gradients which drive zero-g heat transfer and fluid behavior. **Figure 4-20** is a flow schematic for the liquid hydrogen subsystem which has been modified to show the location and number of all sensors. Quantities of sensors used in the COLD-SAT experiment are summarized in **Table 4-12**.

4.8.1 Sensor Locations

Sixty-four Si-diode temperature sensors are distributed throughout the supply tank interior to determine bulk fluid temperature. Adjacent to approximately half of the temperature sensors is a liquid-vapor sensor. The locations (see **Figure 4-21**) were selected for overall coverage and ease of installation.

* Measurement requirements, including number of sensors, location, range, and accuracy are given for each experiment in the COLD-SAT Experiment Requirements Documents (ERD).

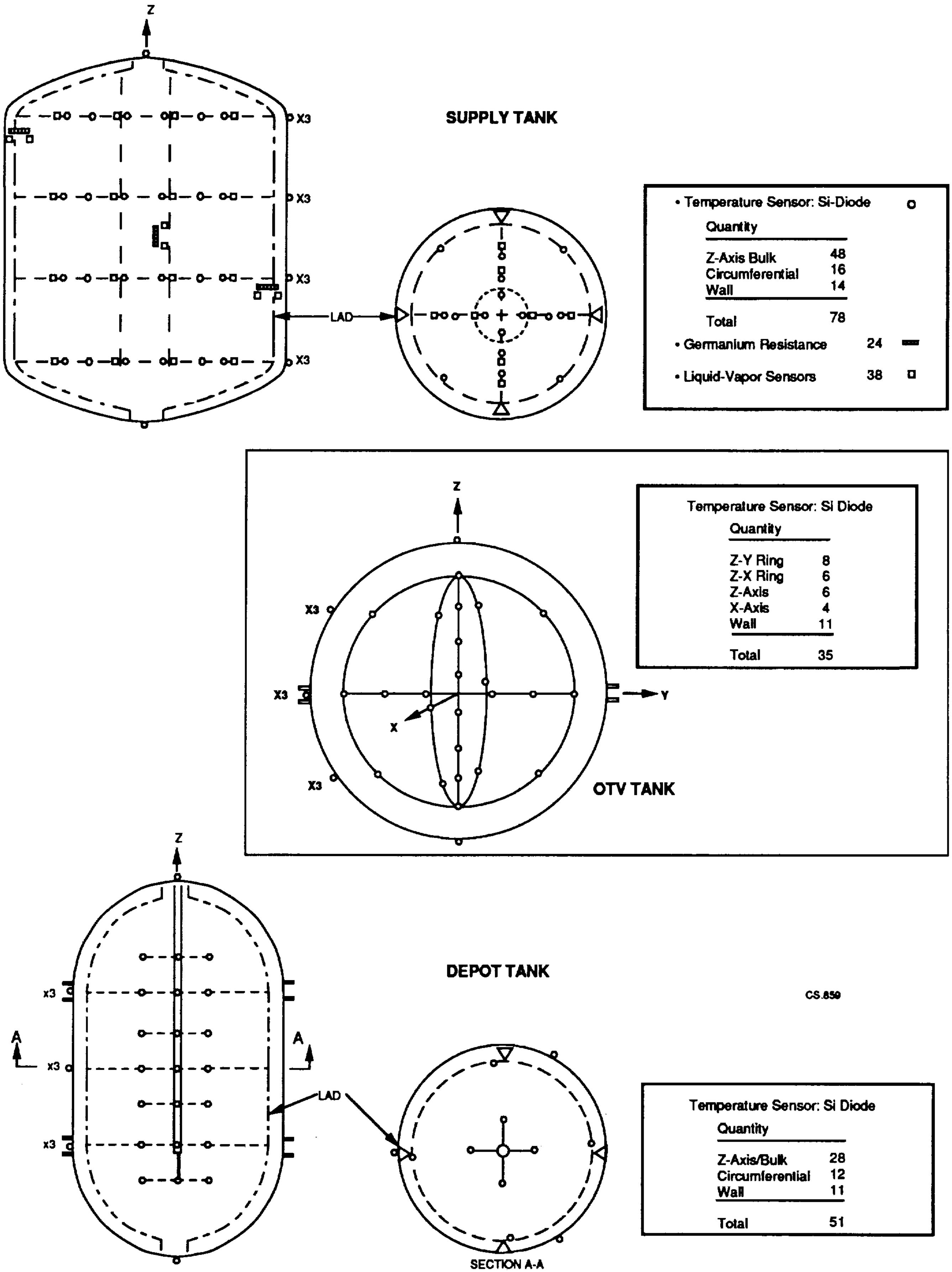

Figure 4-21. Temperature and liquid-vapor sensors in COLD-SAT tanks.

Table 4-12

COLD-SAT EXPERIMENT SET

INSTRUMENTATION SUMMARY

	TEMPERATURE SENSOR		LIQUID-VAPOR SENSOR	PRESSURE SENSOR	FLOW METER	ACCELER-OMETER (3-AXIS)
	DIODE	GRT				
Supply tank	78	24	38			
Depot tank	51					
OTV tank	35					
LAD, supply			8			
LAD, depot	40		20	8		
TVS	58			(5)*		
LH_2 subsystem	18		9	23	14**	
Press subsystem	8			12	2	
Miscellaneous	37					1
Totals	325	24	75	43	16	1

* Included in the LH_2 Subsystem
** 5 turbine meters, 9 velocimeters

The sensors will be bonded to an instrumentation tree made of thin, low-thermal conductivity fiberglass bands which use the four channels of the LAD as their primary support structure. With this approach, the temperature and liquid-vapor sensor "tree" can be assembled onto the LAD and checked out before the LAD is attached to the tank.

There are three linear arrays of eight germanium resistance thermometers also located within the supply tank. One array is located at approximately the 50 percent fill level and two arrays are located adjacent to the wall to assess liquid stratification in those locations. At either end of these arrays is a liquid-vapor sensor to determine if the arrays are in liquid, vapor, or at the liquid-vapor interface.

To determine tank wall temperatures, fourteen sensors have been located on the pressure vessel exterior wall. There is one sensor at each end where the major plumbing penetrations are located, and twelve sensors located circumferentially around the tank, several of which are located in the vicinity of the support strap attach points.

Two fiberglass channels run the length of the supply tank with 33 liquid-vapor sensors each mounted at 2-inch intervals. The channels are offset from each other by one inch providing liquid-vapor sensing at 1 inch intervals. These sensors will be used as a depth probe when fluid is settled in the supply tank.

There are forty Si-diode temperature sensors distributed throughout the bulk fluid region of the depot tank (see **Figure 4-21**). Since the primary purpose of this tank is LAD testing, liquid-vapor sensors are not included in the bulk fluid region. The circumferential array of sensors is mounted to the LAD while the Z-axis sensors are mounted on a central rod. This configuration was selected for ease of assembly and for minimum interference with the chilldown and no-vent fill spray operations. They are located to provide sufficient information to determine the overall liquid or vapor temperature distribution throughout the tank. As in the supply tank, the eleven exterior wall temperature sensors are located to provide a representative temperature distribution of the pressure vessel wall. The circumferential sensors are located just off the LAD channel to provide a local fluid temperature measurement during LAD TVS operations.

As in the depot tank, the OTV uses only Si-diode sensors. Their locations (see **Figure 4-21**) provide bulk fluid or vapor temperature distributions with minimal interference to the sprays. The support frame on which the sensors are located is attached to the girth ring through low thermal conductivity fiberglass tubes and bands. The eleven wall-mounted temperature sensors are located to provide a representative wall temperature distribution.

Table 4-13 summarizes diode locations for TVS instrumentation. In 4 of the 5 TVS systems, 2 Si-diodes are located at the entrance and exit of each heat exchanger to provide inlet and outlet temperatures. Since 2 of the TVS systems are designed to measure heat exchanger performance in more detail, they have additional sensors located internally.

Table 4-13
TVS INSTRUMENTATION SUMMARY

SYSTEM	LOCATION	NUMBER OF SI-DIODES				
		UPSTRM J-T	DNSTRM J-T	HX EXIT	HX INTERNAL	TOTAL
TVS 1	Supply Passive	1	2	2	12	17
TVS 2	Supply Active	1	2	2	0	5
TVS 3	Supply Subcooler	1	2	2	0	5
TVS 4	Depot LAD	0	4x1	2	4x5	26
TVS 5	OTV External	1	2	2	0*	5

* 11 sensors on PV wall will reflect TVS 5 performance

A minimal instrumentation configuration will be used to monitor the expansion valve and heat exchanger performance for three of the five TVS systems. Temperature and pressure sensors located upstream and downstream of the expansion valve completely determine the fluid states, and therefore the performance of the expansion valve. Pressure sensors are remotely located; the upstream sensor pressure tap is at the LAD outlet, and the downstream pressure tap is at the vapor cooled shield outlet. The LAD pressure sensor will accurately predict the upstream pressure because the LH_2 flow rates are so small that the pressure measured is essentially a static pressure. The downstream flow will have a small pressure drop that can be calculated during ground testing to accurately determine pressure just downstream of the expansion valve.

The heat exchangers have redundant temperature sensors located on the inlet and outlet sections. Pressure and pressure drop through the heat exchanger will be inferred in the manner discussed above. From the inlet and outlet temperatures, pressure and bulk liquid temperature, the heat exchanger performance can be characterized.

For the passive TVS instrumentation in the supply tank, twelve temperature sensors are located throughout the heat exchanger tube. Two sensors are located at the inlet and two at the outlet from the supply tank. The sensor spacing will enable determination of internal wall dryout to within 7 feet (8 percent of total area).

The depot tank TVS is heavily instrumented to support pressure control testing and to evaluate heat transfer during LAD vapor bubble collapse. Each leg of the TVS heat exchanger has 5 diodes distributed over its length. Diodes located immediately downstream of the J-T valve and at the vent exit will characterize overall heat exchanger performance.

In order to fully characterize heat transfer during vapor bubble collapse, each leg of the depot LAD is instrumented at 5 stations with 3 temperature sensors and a liquid vapor sensor (See **Figure 4-22**). The sensors are located to characterize heat transfer from the channel interior through the LAD wall, and into the TVS tube and surrounding bulk fluid. Bulk fluid temperatures are measured at 3 stations using the tank sensors mounted to fiberglass bands. The mounting concept for LAD channel instrumentation is shown in more detail in **Figure 4-12**.

4.8.2 Data Acquisition System

The block diagram in **Figure 4-23** shows how the sensors and experiment control devices are connected into the spacecraft electronics. The large dashed box labeled "Experiment Control Processor" (ECP) was developed under a Ball IR&D program. Each component in the experiment processor is a standard piece of hardware (except the SFI cards). The SFI cards (Special Function Interface) are used

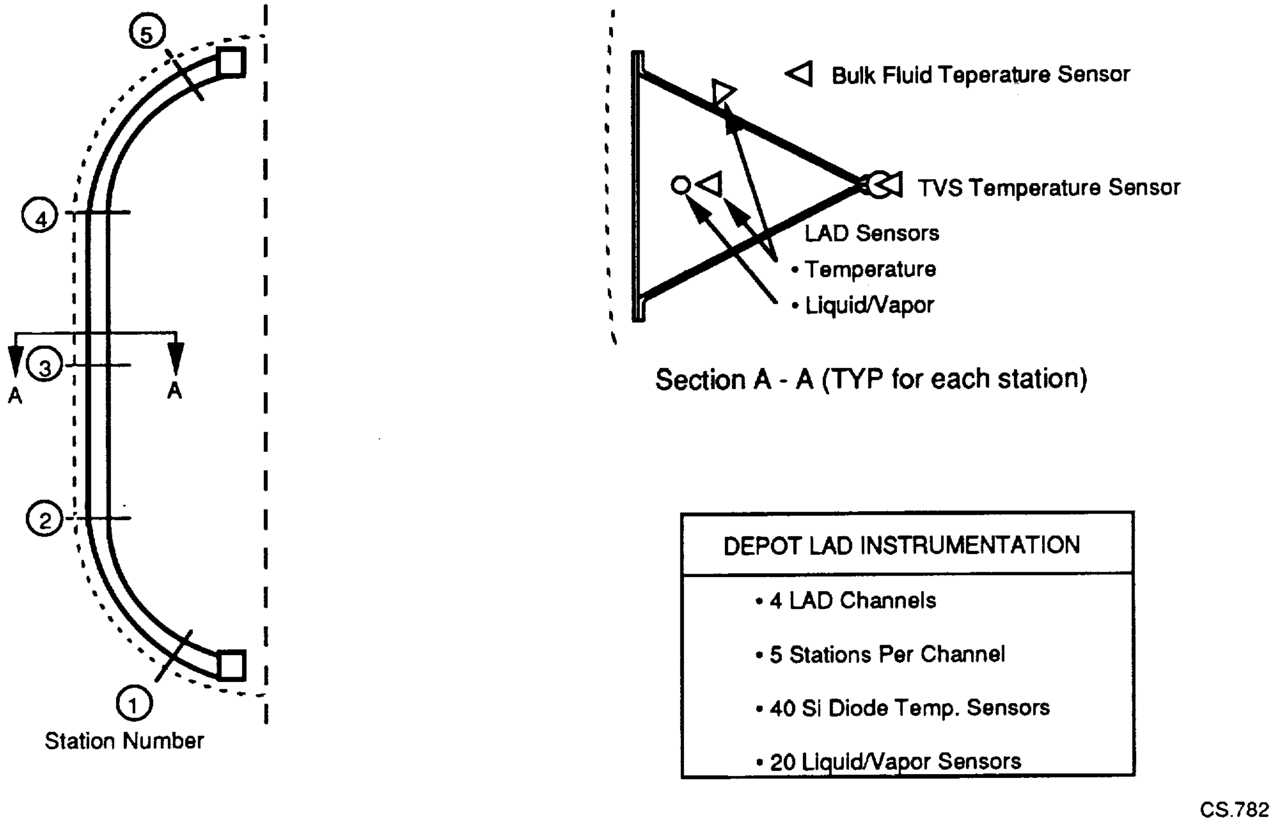

Figure 4-22. Depot-tank LAD instrumentation.

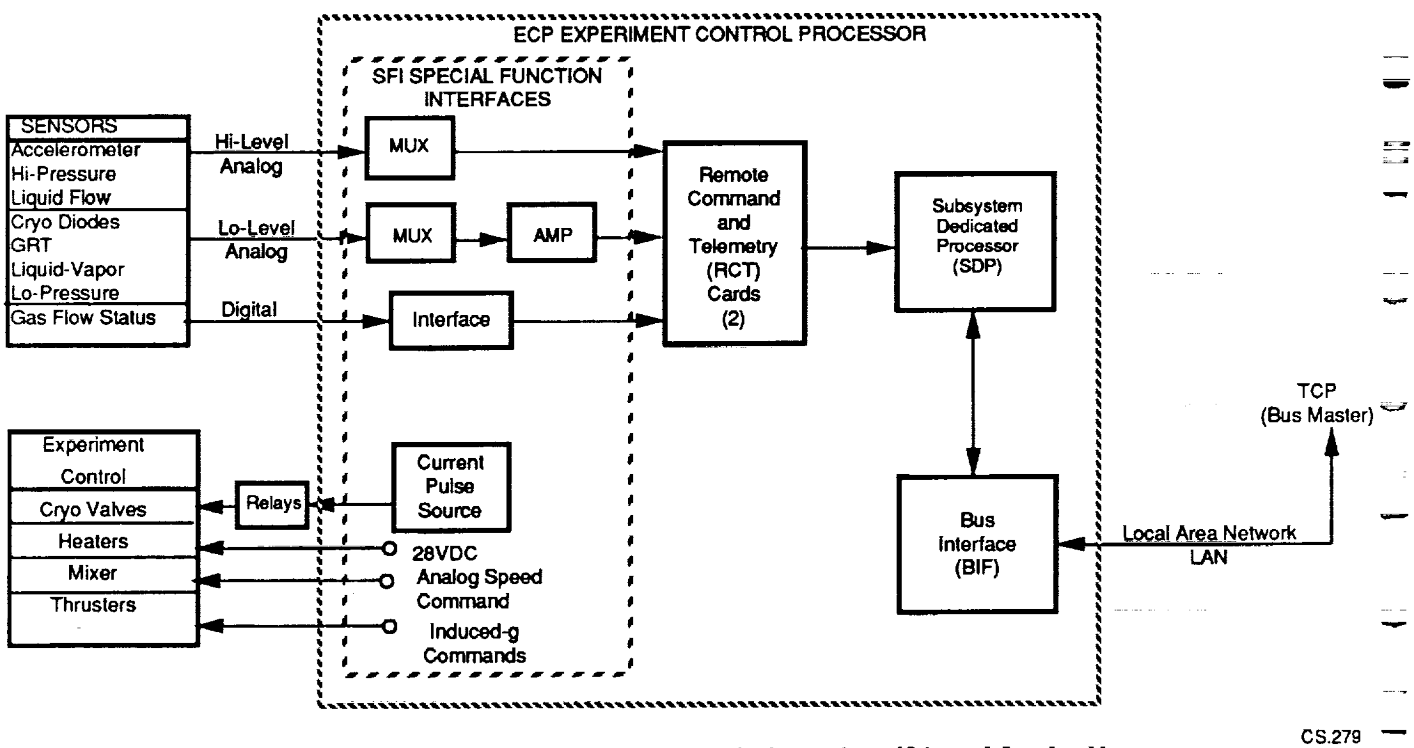

Figure 4-23. Experiment subsystem command and data handling block diagram.

to accommodate special needs of the sensors (such as multiplexing or signal amplification). Although they are not off-the-shelf items, they use existing circuitry and standard components.

The role of each of the components is described in more detail in paragraph 6.4 of this report. The bus interface serves as a communication link between the experiment subsystem and the spacecraft LAN. The subsystem dedicated processor is an 80C86 microprocessor which is programmed with experiment control and data handling routines. The remote command and telemetry cards have multiple channels for analog inputs/outputs and built-in A/D converters.

Operationally, the ECP reads all sensors into RAM once per second. Sensors are sampled at constant time intervals and at precisely determined times to improve data processing accuracy. The sensor reading sequence is invariant throughout the mission. A sub-set of sensors is extracted from memory and blocked according to a pre-determined format for transmission to the ground.

Four telemetry formats were created to allocate sensors into blocks which reduces overall data rate requirements. Telemetry formats are matrices which assign slots to individual sensors for data transmission. The four formats are summarized in **Table 4-14**.

Table 4-14

TELEMETRY FORMATS

FORMAT #	NAME	TOTAL # OF SENSORS	TOTAL DATA BITS
1	Supply tank pressure control	304	2732
2	Receiver tank pressure control	309	2960
3	Depot tank transfer	374	3676
4	OTV tank transfer	285	2652

Each format contains a subset of sensors which are used for a similar purpose. Although formats overlap substantially, no format contains all of the sensors. This reduces the total data rate required in the data acquisition system and simplifies data handling on the ground. Detailed tables and use of the formats for individual experiments are given in the Experiment Requirements Document (ERD).

All formats include experimental data and enough housekeeping data (pressures, valve status, critical temperatures, etc.) so that a complete status of the experiment subsystem is always available. The experiment data is added to spacecraft housekeeping data to create the total telemetry format.

The design capacity for handling sensor inputs is summarized in **Table 4-15**. All sensors are multiplexed to optimize usage of analog input channels. The number of spare channels can be expanded by adding additional SFI circuitry and using some of the 22 spare analog channels. In the table, the abbreviation OFS represents one channel which is set aside for offset correction.

End-to-end error analysis for diodes, GRT's, and pressure transducers is given in **Table 4-16**. Although all error contributions are added to give the worst-case error, it is unlikely that all errors will occur simultaneously at their peak magnitude and in the same direction. Therefore, actual errors will probably be less than the worst case error. Extending the temperature range of the diode to higher values will not significantly change the total error because each diode is individually calibrated and the circuit error contributions are not strong functions of sensor temperature.

Driver circuits for silicon diodes and GRT's are shown in **Figure 4-24**. Note that current sources are shared by sets of diodes and GRT's, and multiplexers switch the current source to the next sensor after approximately 62 milliseconds.

4.9 REFERENCES

4.1 DELTA II Commerical Spacecraft Users Manual, McDonnell Douglas Document MDC H3224.

Table 4-15

ECP ANALOG TELEMETRY CHANNELS

BITS	TELEMETRY	③ SFI MUX	RCT CHANNELS	SUB MUX's USED CHAN	SUB MUX's SPARE CHAN ③	SAMPLE RATE
12	Silicon Cryo Diodes	15+OFS:1	24	325	35	1 Hz
12	Germanium Resistance Thermometers GRT	15+OFS:1	2	24	6	1 Hz
12	31 Pressure Transducers 0-15, 0-50 PSIA	7+OFS:1	5	31	4	1 Hz
12	3 Accelerometer Axes	16:1	3	35	14	1 Hz
12	3 Accelerometer Peak over 1 Second					
8	1 Accelerometer Temperature					
8	4 Pressure Transducers 0-100 PSIA					
8	8 Pressure Transducers 0-5000 PSIA					
8	5 Liquid Flow Meters (LFM)					
8	11 Gas Flow Meters (GFM)					
8	Liquid/Vapor Sense Resistors, Tanks	16:1	5	66	14	1 Hz
8	Liquid/Vapor Sense Resistors, Lines	4:1	3	9	3	4 Hz
	Spare (Two RCT Cards)		22 / 64			

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Table 4-16

EXPERIMENT SENSOR ERRORS

ERRORS ECP Circuit Temperature = 0°C to 30°C	DIODE (17K TO 23K) ±mv @ADC	DIODE (17K TO 23K) ±mK @19K	GRT (17K to 27K) ±mv @ADC	GRT (17K to 27K) ±mK @27K	PRESSURE (0-50 psia) ±mv @ADC	PRESSURE (0-50 psia) %
Offset Channel Correction (±1/2 LSB)	0.62	1.3	0.62	5.0	0.62	0.01
Quantization Error (±1/2 LSB)	0.62	1.3	0.62	5.0	0.62	0.01
Instr. Ampl. & Filter Gain Tempco	0.76	1.5	0.76	6.1	0.76	0.01
ADC Linearity (±1 LSB)	1.25	2.5	1.25	10.1	1.25	0.02
Gain Calibration	2.60	5.3	2.60	21.0	2.60	0.05
Current Leakages (Mux & Ampl.)	2.95	6.0	0.38	3.0	0.00	0.00
Thermal Barrier Tempco	0.00	0.0	0.00	0.0	—	—
Sensor Self Heating	<9.88	20.0	0.62	5.0	—	—
Current Source (±0.1%)	3.60	7.3	4.64	37.5	—	—
Voltage Source	—	—	—	—	5	0.05
Circuit Worst Case Total	22.3	45.1	11.5	92.7	10.85	0.15
Sensor	—	50	—	15	—	0.18
Combined Worst Case Total	—	95	—	108	—	0.33

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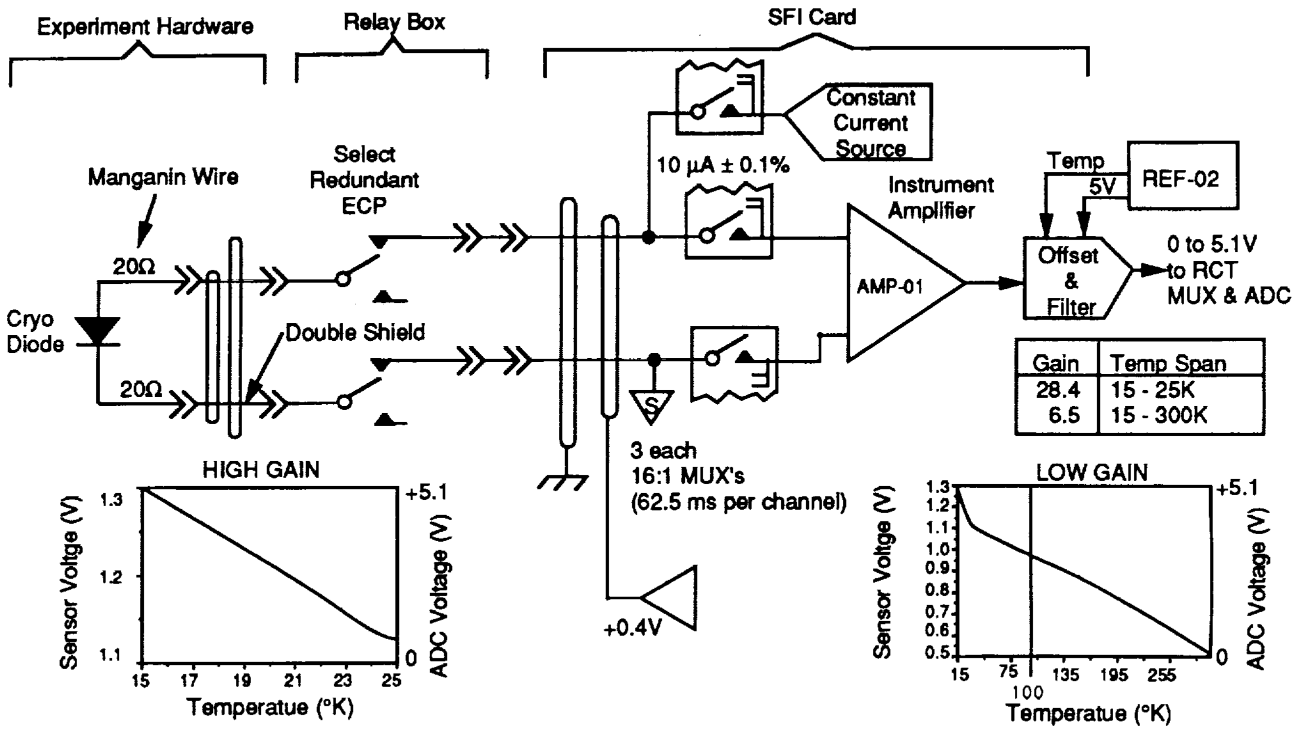
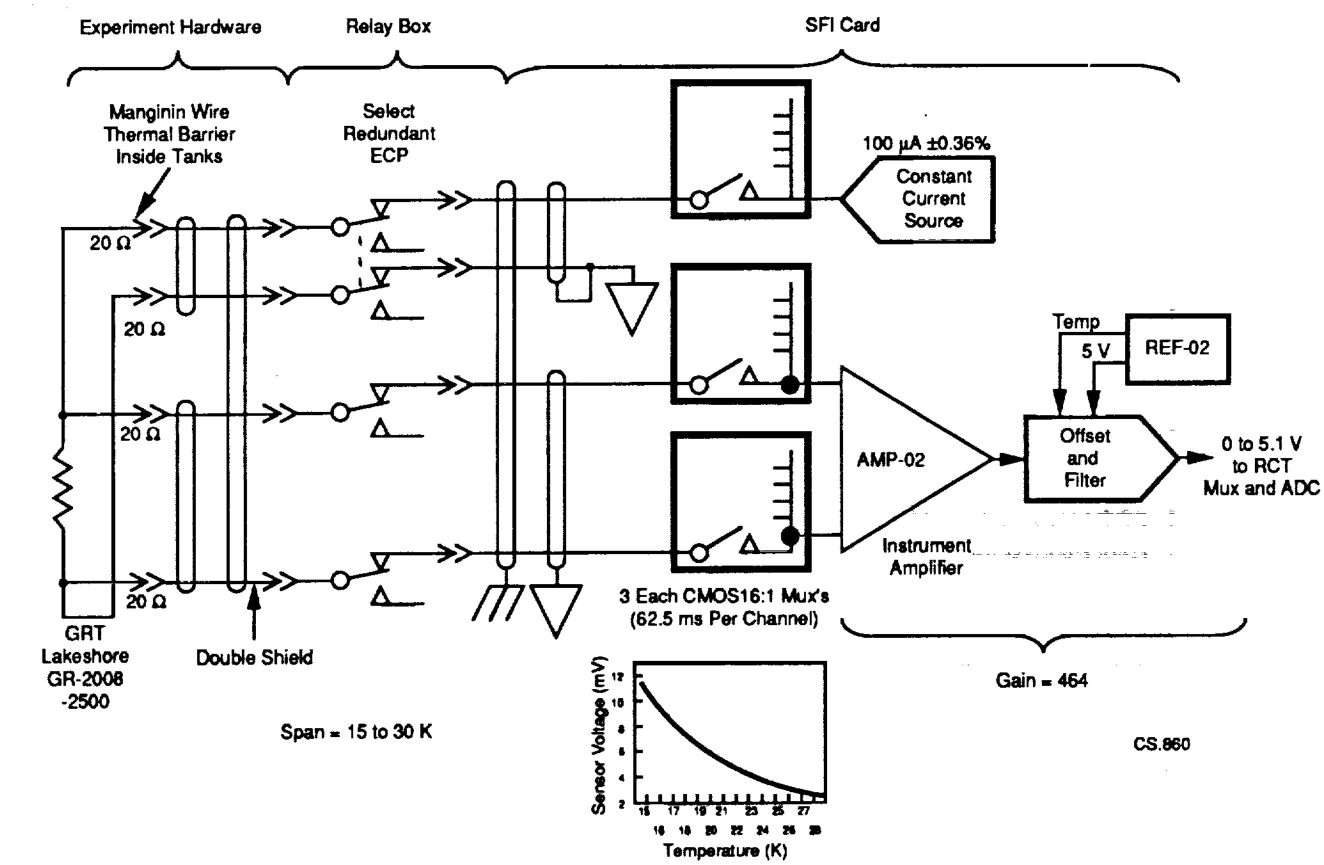


Gain	Temp Span
28.4	15 - 25K
6.5	15 - 300K

(a) Si-Diode Circuit



(b) GRT Circuit

Figure 4-24. Si-diode and GRT circuit diagrams.

Section 5
MISSION DESIGN

This section summarizes the top level COLD-SAT mission requirements, baselines the orbit and attitude combination which we believe is most appropriate for meeting these requirements, and analyzes launch vehicle performance and the various other factors which led us to these selections.

5.1 MISSION REQUIREMENTS

Table 5-1 shows the requirements that size or drive the space and ground segment designs. Accommodating the tanks drives the configuration when coupled with the fairing envelope. The induced acceleration further drives the envelope because of the volume occupied by the fuel.

Table 5-1
MISSION REQUIREMENTS

ITEM	REQUIREMENT
Science Accommodation	
LH_2 Supply Tank	3.4-3.5 m^3 (120-125 ft.)
Depot Tank	0.70-0.85 m^3 (25-30 ft.)
Receiver Tank	0.28-0.57 m^3 (10-20 ft.)
GH_2 Pressurant @ 4000 psi	0.89 m^3 (31.5 ft.)
GHe Pressurant @ 4000 psi	0.13 m^3 (4.5 ft.)
Power (design)	116 W
Data Rate	>3.7 Kbps
Induced Acceleration (nominal)	$2x10^{-5}$, $7x10^{-5}$, $1.4x10^{-4}$, $1x10^{-3}$ g
Background Acceleration	$<10^{-6}$ g
Launch Vehicle	Delta II
Max Spacecraft Wt (w/o contingency)	See Section 6.4
Orbit (circular)	926 km at 28.7 deg
Communication	TDRSS MA
Contact Duration	10 min per orbit
Control Center	GSFC Multisat
Science Center	At LeRC
LH_2 Loading	L-3 hr
Wearout Life	>1 yr
Orbit Life	>500 yr

TDRSS multiple access (MA), the real time telemetry rate and the contact time size the transmitter and antenna. Gas and cryogenic loading at L-3 hours drives the pad modification. Other launch requirements are met without facility modification.

Control of the space segment is accomplished using the TDRSS and GSFC multi-sat facilities and support functions. A Science Operations Center (SOC) will be set up at LeRC. This will require some project funded equipment. Software will be created to provide displays of experiment status and performance.

5.2 ORBIT CHARACTERISTICS

The chief parameters which influence orbit selection consist of 1) the quiescent background acceleration field to which the experiments will be subjected, 2) the susceptibility of the electronics to background radiation, and 3) the long term reentry probability of the vehicle.

Background accelerations derive from the separate effects of drag and the earth's inverse square central gravity field. The former produces spatially uniform accelerations which are directed parallel to the instantaneous velocity. The latter produces non-uniform accelerations from point to point within the vehicle because of the instantaneous differential gravity vector at each locale relative to the gravity vector experienced at the CM. Additionally, if a spacecraft is rotating however slowly, as in the case of earth-oriented vehicles, this "gravity gradient" (GG) effect is further compounded by GG-like centrifugal accelerations. This overall GG effect is not a function of inclination but is a weak function of altitude, decreasing as $1/r^3$. To enhance data correlation it will be desirable to keep the background acceleration envelope not only low, but also more or less constant over the experiment set. This implies use of a circular orbit.

Orbit selection relative to radiation effects generally involves avoidance of the Van Allen Belts. This can be generally accomplished by staying below about 1000 km altitude.

The long term lifetime of greater than 500 years will be met through the use of the mandated 926 km (500 NMI) circular initial orbit together with an induced-g thrusting scheme which will produce only very minor perturbations to this orbit. As will be shown in Section 5.4, a Delta II class ELV is quite capable of injecting COLD-SAT into such an orbit with good mass margin.

5.2.1 Background Accelerations and Lifetime

A 926 km (500 nmi) circular orbit was selected to eliminate concerns related to early reentry and impact area uncertainty. Using characteristic COLD-SAT ballistic coefficients, **Figure 5-1** displays drag acceleration and resulting passive lifetime as a function of starting circular altitude. At 926 km, and even for a very conservative exospheric temperature of 1200 °K, lifetime is greater than 500 years and the drag acceleration is less than 10^{-9} g, making gravity gradient effects the dominant source of the background acceleration.

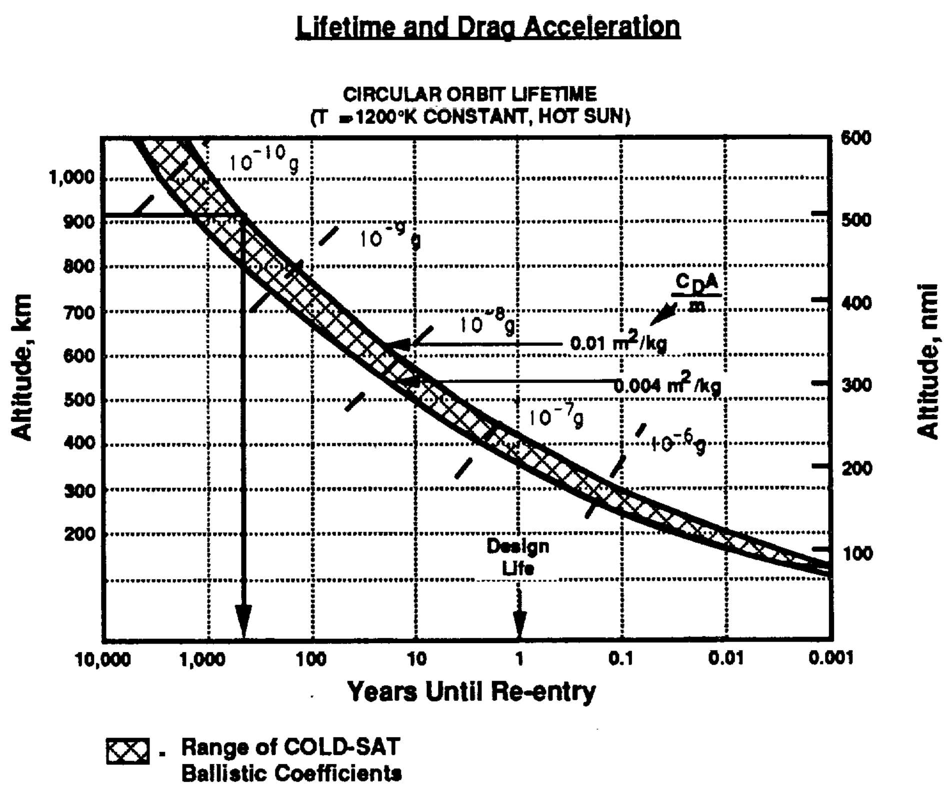


Figure 5-1 Circular orbit drag accelerations and lifetimes.

5.2.2 Disturbance Torques

Gravity gradient and drag forces also cause disturbance torques which generally diminish with altitude at the same overall rate as was previously described for the background accelerations. Such torques together with those induced by solar pressure and residual magnetic dipole moment (the latter also a function of altitude) are estimated for 926 km and are shown in Table 5-2. Simulations show that the compensation for the total of such disturbance torques will require less than 13 kg fuel.

Table 5-2
NON-THRUSTING DISTURBANCE TORQUES

SOURCE	N-m (Peak)
Aero-drag (10 cm offset)	9.4×10^{-7}
Solar	1.4×10^{-5}
Magnetic (1.5A m^2 each axis)	6×10^{-5}
Gravity Gradient*	
About long axis	5×10^{-5}
About each cross axis	3×10^{-4}

*For ±1° deadband

These disturbance torques are actually small compared to those imposed by the science. During induced-g phases, for example, torques are produced because the thrust vector will not go through the CM on any real design. Also venting over 250 kg of H_2 cannot be done torque free. The dumping experiment produces relatively large torque disturbances. These torques size the thrusters and are discussed in the sections dealing with design of the attitude control and propulsion systems.

5.2.3 Background Radiation

Given that the above considerations establish general lower altitude bounds, criteria associated with an upper bound must also be addressed. The two most important such criteria are those connected with launch vehicle capability and radiation exposure. Presently, the wet mass margin is 1,237 kg with respect to the design altitude of 926 km. The expected radiation dose versus a typical design threshold is shown in Figure 5-2. For the majority of most electronic components to be non-hardened, a dose of less than 1000 rads over the active portion of a mission is considered typical. For a mission lasting up to a year, but with the concentration of experiments heavily weighted to occur within the first three months, a nonhardened spacecraft will be fully adequate.

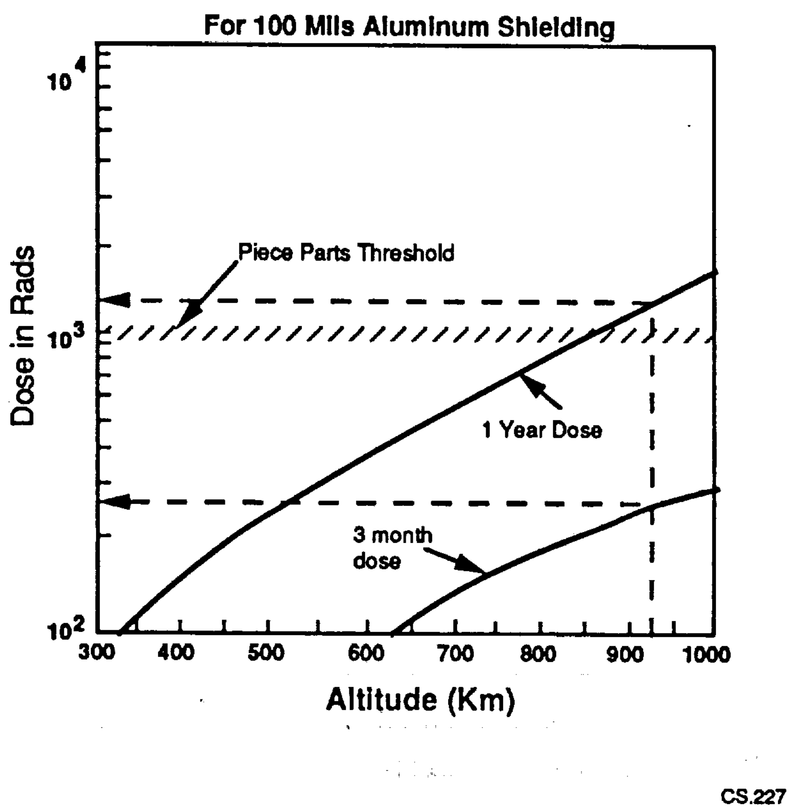


Figure 5-2 Radiation dose.

5.2.4 Baseline Orbit

The baseline orbit parameters are as shown in **Table 5-3**. The altitude of 926 km circular represents a good compromise relative to drag, gravity gradient, radiation, and long term survivability considerations. The launch vehicle is capable of putting considerably more mass into a 28.7 deg inclined orbit than is required and, other than a standard daily nodal preference to optimize initial sun angle, there should be no launch window restrictions.

5.3 ATTITUDE CHARACTERISTICS

The selected attitude should support the low-g background field during all free-flight phases as well as facilitate increasing the g-field to the required discrete uniform levels. Additionally, orbit perturbations should be minimized as a result of such g-field augmentation.

Table 5-3
BASELINE ORBIT

ELEMENT	VALUE	REMARKS
Altitude	926 km (500 NMI),	- Non-hardened Electronics - Survivability >500 years - Drag $<10^{-9}$ g
Inclination	28.7° Due East ETR	- Affords best mass margin
Initial Ascending Node	Keyed to Sun of Date	- 40 Minute Standard Launch Window - Initial Y axis to Sun <5°

During both low-g and augmented -g phases it will also be highly desirable if experimental results are not corrupted by the possible masking effects of induced Coriolis accelerations. Finally, the selected attitude should support the use of simplified attitude control and propulsion techniques and configurations as well as permit the use of simplified solar array and communications subsystems designs.

5.3.1 Orbit Distortion Susceptibility During Induced-g Thrusting

During constant thrusting along a fixed body axis the magnitude of the sensed acceleration is completely independent of the thrust direction measured relative to orbit geometry. On the other hand, the manner in which the orbit may change in shape, orientation, and by what rate, is intimately related to thrust direction relative to the orbit.

Basic thrust directions strategies versus resulting orbit changes are illustrated in **Table 5-4** with the understanding that combinations of these strategies could be used to produce many other possibilities. The goal is to identify the thrusting program which is consistent with other requirements, is convenient, and yields the smallest cumulative change in orbit altitude.

Table 5-4

ORBIT PERTURBATION VERSUS THRUST DIRECTION SUMMARY

LONG AXIS THRUST DIRECTION	FIRST ORDER PERTURBATION OVER CUMULATIVE INDUCED-g (25.8 g-seconds)
Parallel to velocity	300 Km in altitude
Parallel to nadir/zenith	19 Km (bounded) in altitude
Inertial in orbit plane	225 Km in altitude
Parallel to orbit normal (baseline selection)	0 Km in altitude 0.08 deg (bounded) in orientation

The first method, in which the thrust is parallel to the instantaneous velocity, adds (subtracts) both energy and angular momentum to the orbit, causing the vehicle to spiral upwards (downwards) depending upon thrust direction.

At the low thrust levels associated with COLD-SAT, the second method yields, in contrast, a small bounded altitude variation over each orbit revolution.

The boundedness results from the thrust which, being directed towards (away) from the force center, cannot alter the angular momentum of the orbiting vehicle.

The third method results in a secular translation of the orbit at right angles to the direction of the applied thrust. Here the dimensions of the original orbit are not greatly altered. However, a reduction in angular momentum occurs which is manifested as an increase in eccentricity, together with steady increases and decreases in apogee and perigee altitude, respectively.

The last method, which employs continuous thrusting along the orbit normal is the recommended thrusting approach and attitudes which support this thrusting program are preferred. This method leaves the altitude of the original circular orbit intact to the first order. There will be small secondary variations plus a small but mostly inconsequential change in the orientation of the orbit in space.

5.3.2 Advantages of Inertial Attitudes

Spin stabilized attitude control is obviously incompatible with experiment objectives. Candidate three-axis-stabilized attitudes include those which are earth pointing and those which are inertial. For COLD-SAT the latter are preferred over the former because the background gravity gradient (GG) field is generally about two-thirds that of the former and the Coriolis field is zero with respect to any fluid in motion. Additionally, an inertial attitude is particularly favorable with respect to pointing a fixed solar array continuously towards the sun. If the vehicle is inertially oriented, with the vehicle long axis directed along the orbit normal as well, then as will be described in the next section, the maximum gg acceleration can be held to less than 4x10-7 g.

5.3.3 Gravity Gradient Characteristics for Baseline Design

The GG field within an inertially oriented vehicle with its long axis parallel to the orbit normal (our baseline attitude) is characterized in the next four figures. As depicted in **Figure 5-3** the perceived acceleration at a general point not located on the vehicle z-axis is a vector which roughly cones about an axis parallel to the z-axis, but at a rate which is twice orbit rate. (Depicted instantaneous acceleration directions correspond to the starting earth-nadir direction as shown.) **Figure 5-3** also shows qualitatively the linear relationship between GG magnitude and the overall distance of points from the center of mass (CM). Note that at no time is the resultant gravity gradient acceleration ever zero except at the CM itself.

Figure 5-4 summarizes the resultant GG accelerations. The upper dashed ellipses of **Figure 5-4** show the GG acceleration for points passing through the local vertical plane (position of maximum GG acceleration) while the lower solid circles show the same for points passing through the local horizontal plane (the position of minimum GG acceleration).

The two sets of contours have been separated for clarity. They do not exist simultaneously in the vehicle at the upper and lower body frame positions shown. Rather one set or the other may be used independently to estimate maximum (upper set) or minimum (lower set) field intensities in accordance with the local vertical plane and rotating vector description from the preceding figure.

The contours of **Figure 5-4** are referenced to the CM position when approximately half the cryogens and propellants have been consumed. As implied, the field would be displaced slightly to the left or to the right for beginning of life or end of life, respectively.

By way of a final GG summary using **Figure 5-4**, it is observed that the acceleration, relative to the baseline inertial attitude, should not exceed 4×10^{-7} g anywhere in the tanks over the life of the mission.

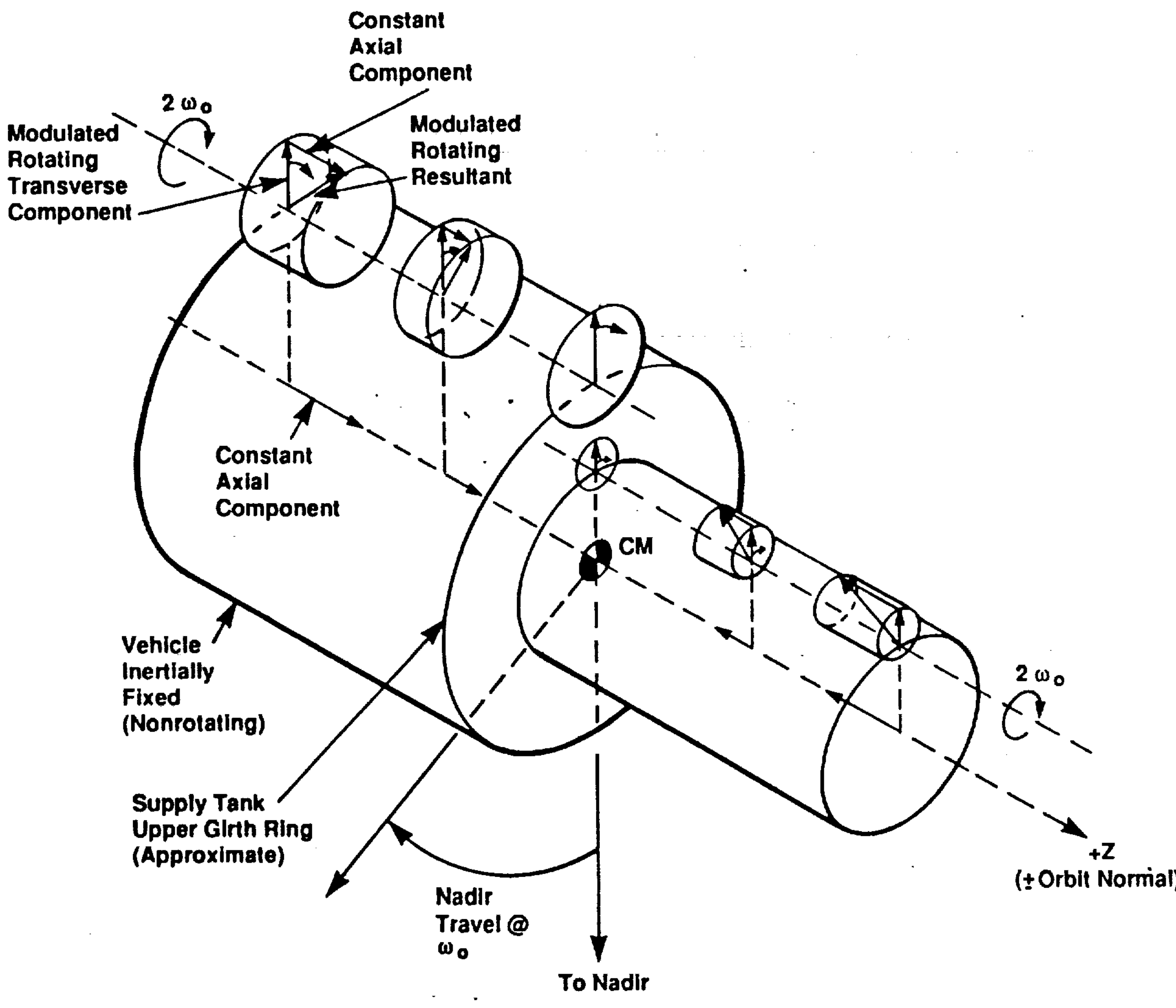


Figure 5-3 Gravity gradient characteristics for baseline attitude.

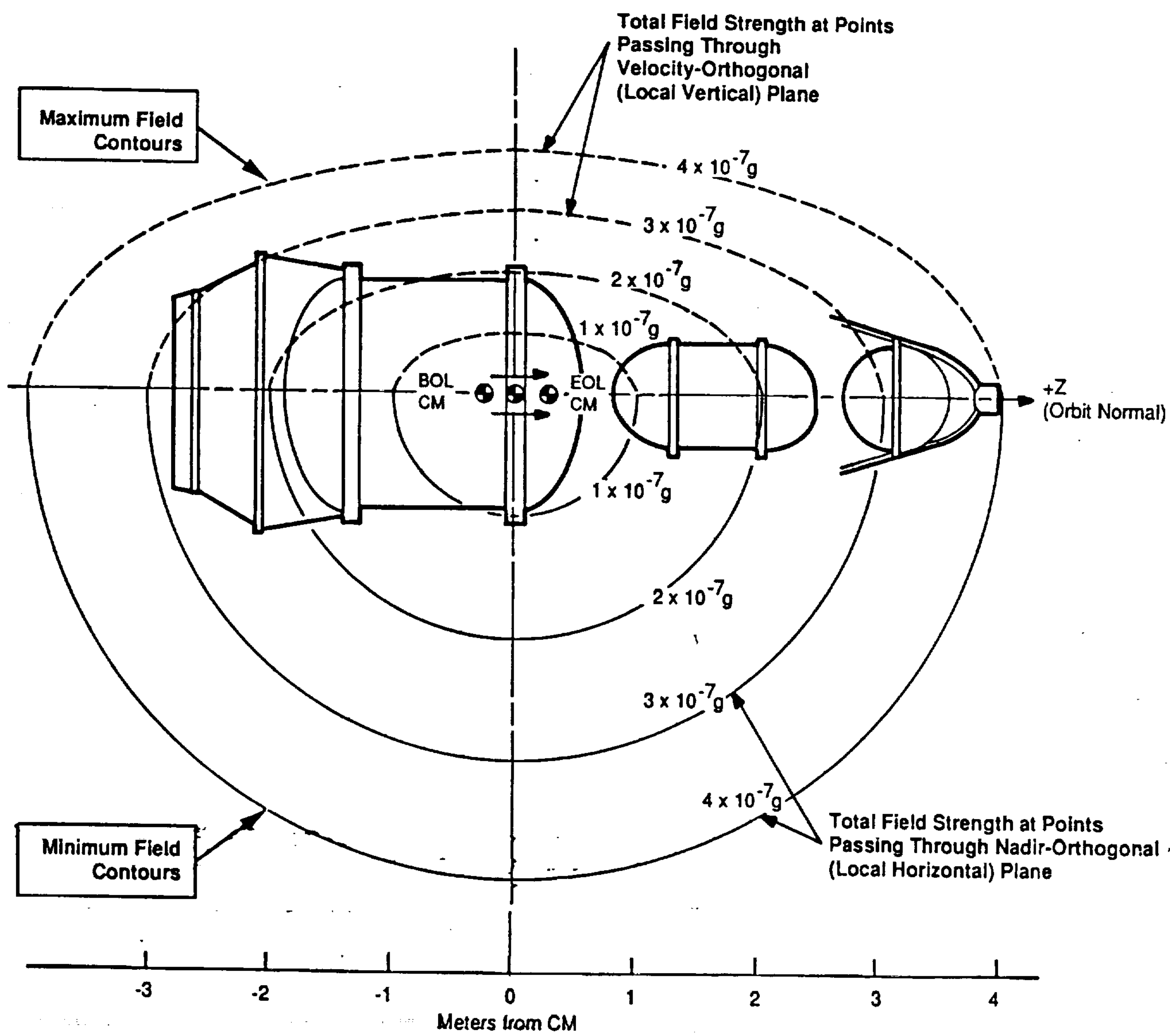


Figure 5-4 Maximum gravity gradient acceleration is less than 4x10^{-7} G.

5.3.4 Baseline Attitude

Our baseline attitude is quasi-inertial with the vehicle long axis directed along the orbit normal. The term quasi is used because to maximize power production, the pitch axis is continuously rotated very slowly (<1 deg per day) to keep the solar array normal within the plane defined by the sun and orbit normal directions. Furthermore, the vehicle long axis will be made to continuously track the regressing orbit normal during both the quiescent and induced-g phases of the mission. For 926 km altitude and 28.7 deg of inclination, the orbit node regression rate is -5.6 deg per day. Finally, 180 deg yaw turns will be needed about every four weeks. This derives from the fixed solar array cant versus the 6.6 deg relative motion of the sun (5.6 deg/day for orbit plus 1.0 deg/day for sun). This will be covered more fully in the satellite power subsystem description.

5.4 LAUNCH VEHICLE REQUIREMENTS AND CONSTRAINTS

Figure 5-5 and 5-6 illustrate Delta II Class Expendable Launch Vehicle capabilities with respect to allowable spacecraft CM location and allowable launch mass, respectively. Per Figure 5-5 our best estimate wet launch mass of 2,963 kg (6,533 lbm) centered at 2.29 m (90.2 inches) above the spacecraft separation plane lies well below both the 3,410 kg (7,500 lbm) mandated weight limit contour and also to the left of the regions in which supplemental rivets would be required.

Per Figure 5-6 this same launch mass will yield a 1,237 kg (2,727 lbm) margin with respect to the Delta's capability of inserting 4,200 kg (9,240 lbm) into the baseline 926 km (500 nmi) circular, 28.7 deg inclined orbit.

The spacecraft was designed to the launch vehicle environments in Section 3.0 of The Delta II Commercial Spacecraft Users manual. The structural design and mass properties discussed in Section 7.1 are consistent with these constraints.

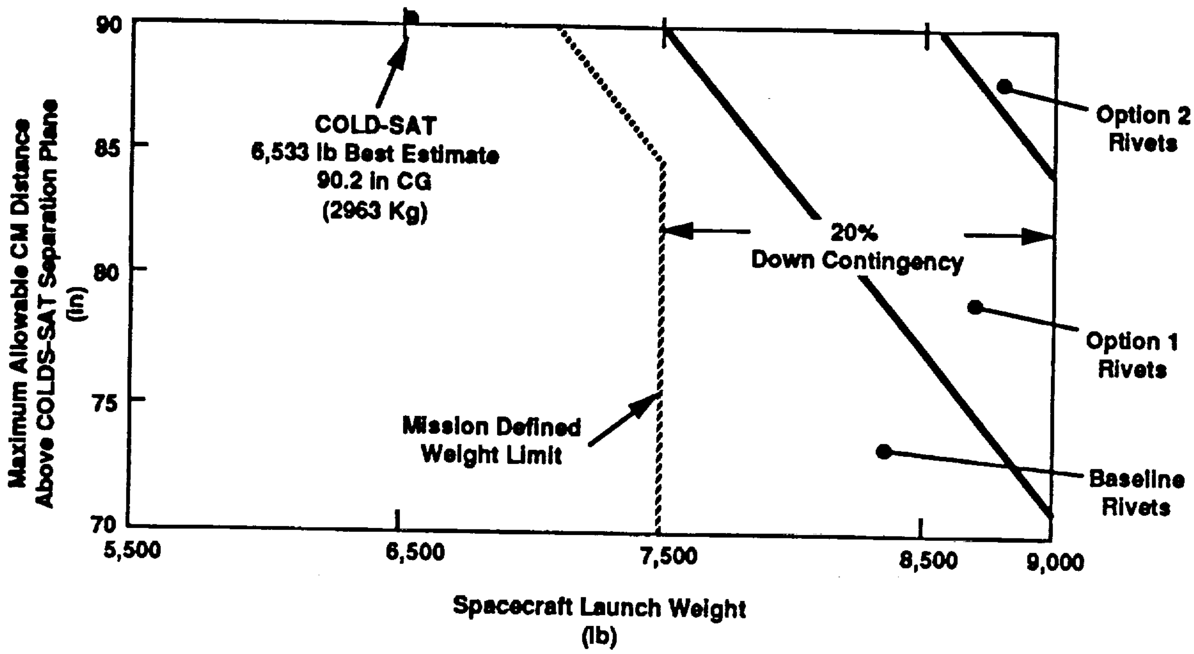


Figure 5-5 Center of mass constraint.

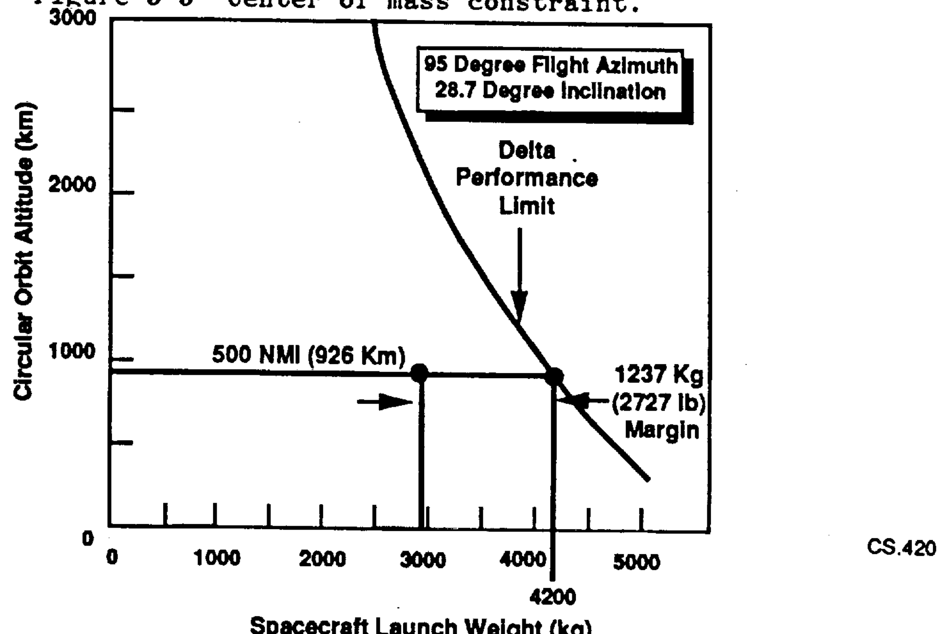


Figure 5-6 Insertion mass constraint.

5.5 LAUNCH SEQUENCE

The launch sequence described by **Figure 5-7** and **Table 5-5** will insert the spacecraft into orbit within ±5 deg of the desired initial operational attitude and will require a 40 minute launch window. The 40 minute launch window can easily be accommodated by Delta. The design allows a wider launch window at the expense of a little additional operational complexity.

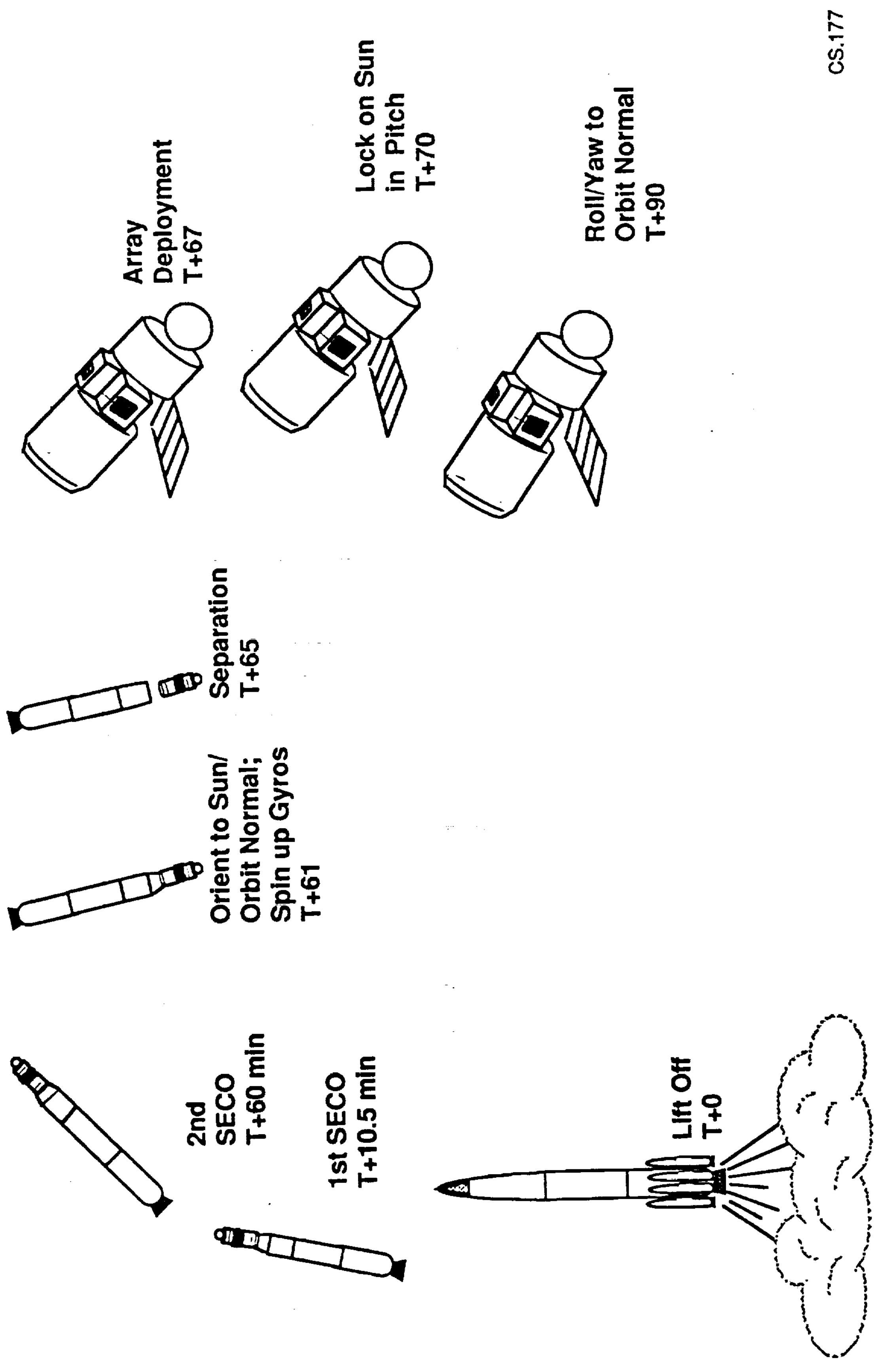


Figure 5-7 Launch sequence.

Table 5-5

LAUNCH SEQUENCE DETAIL

T -3 day	S/C tower tests, cool supply tank to 60 K, pressurize GH_2 and GHe tanks to 13.8 MPa (2,000 psi)
T -3 day	Top GH_2 and GHe tanks to 27.6 MPa (4,000 psi)
T -8 hour	Final S/C checkout, all boxes off except ECP and TCP
T -3 hour	Load LH_2
T -30 minute	Loading complete, vent LH_2 tank through tower stack, remove battery charge; configure for launch, switch to undervoltage state, freeze clock at zero, power tank pressure sensor from GSE.
T -0	Liftoff, pull vent and loading lines and electrical umbilical.
T +5 minute	Fairing separation, start S/C stored command clock.
T +10.5 minute	1st SECO (transfer orbit perigee)
T +60 minute	2nd SECO (injection into first orbit)
T +61 minute	Launch vehicle starts maneuver to sun and orbit normal and stored command switches to normal voltage state, starts gyro spin-up. Note: gyro integrator held at zero and thrusters inhibited by separation connector shorts.
T +65 minute	Launch vehicle separation at local noon, gyro integrator released and thrusters armed by pulling separation connector. (RCS control starts.)
T +67 minute	Deploy array using stored commands. Deployment inhibited if rates large.
T +70 minute	Sun sensor has updated pitch gyro integrator (1 minute time constant).
T +90 minute	Roll and yaw information from horizon sensors has updated gyro integrators to align Z axis with the orbit normal.

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The spacecraft +Y axis is aligned to the -yaw axis (uprange) of the Delta launch vehicle (solar array west when on the launch pad). This puts the S/C in a near normal attitude referenced to the sun during coast. The vehicle sequence proceeds as follows (all times are given in the time at the earth subpoint):

- Launch at 2000 hr ±30 minutes local time (beyond dusk terminator)
- First SECO (perigee) occurs at 2240 hr local time (0120 hr before descending node)
- Perform a near Hohmann transfer to the injection point (180 deg or 1040 hr local time)
- Restart the 2nd stage to circularize the orbit
- Make a 30 deg pitch down and a 90 deg yaw maneuver to put the nose along the anti-orbit normal (south)

This sequence places the spacecraft +Z axis along the antiorbit normal and the array aligned to the sun. The solar β angle will be near zero. The spacecraft then performs maneuvers of less than 5 deg to obtain the operational attitude. The orbit precesses, relative to the sun, at 6.6 deg per day in a direction that increases the initial β angle toward the anti-orbit normal. Thus, a yaw maneuver will not be needed for about a month.

Section 6
BUS DESIGN

The following subsection describes the design of the various subsystems that comprise the COLD-SAT bus which include the mechanical/structural, thermal control, electric power, tracking telemetry and command, attitude control and propulsion subsystems. The requirements for each of the subsystems are given, followed by a discussion of the design and a comparison between the design capabilities and the original requirements. At the end of each subsection is a component list, including heritage, that comprises the subsystem being described. In addition, a description of the bus software is included in the last subsection.

6.1 CONFIGURATION/STRUCTURAL SUBSYSTEM

The on-orbit arrangement of COLD-SAT is shown in **Figure 6-1**. The major elements of the spacecraft are also called out in this figure.

6.1.1 Requirements

The configuration/structural requirements for the S/C are summarized in **Table 6-1**. There were two primary drivers that shaped the configuration; the Delta 2.9 m (114 inch) fairing that was baselined, and the accommodation of the tanks required by the science. The envelope of the bulbous fairing was taken advantage of as apparent by the fit of the S/C within the fairing shown in **Figure 6-2**. The size of the solar array dictated by the power system was easily accommodated with a 50 percent growth margin available by stretching each panel in length to the limits of the fairing and/or adding additional leaves. The requirement for a remote LH_2 fill and disconnect was provided for by placing fittings just above the S/C separation plane. A NASTRAN model determined that the S/C structure meets the Delta stiffness requirements for launch.

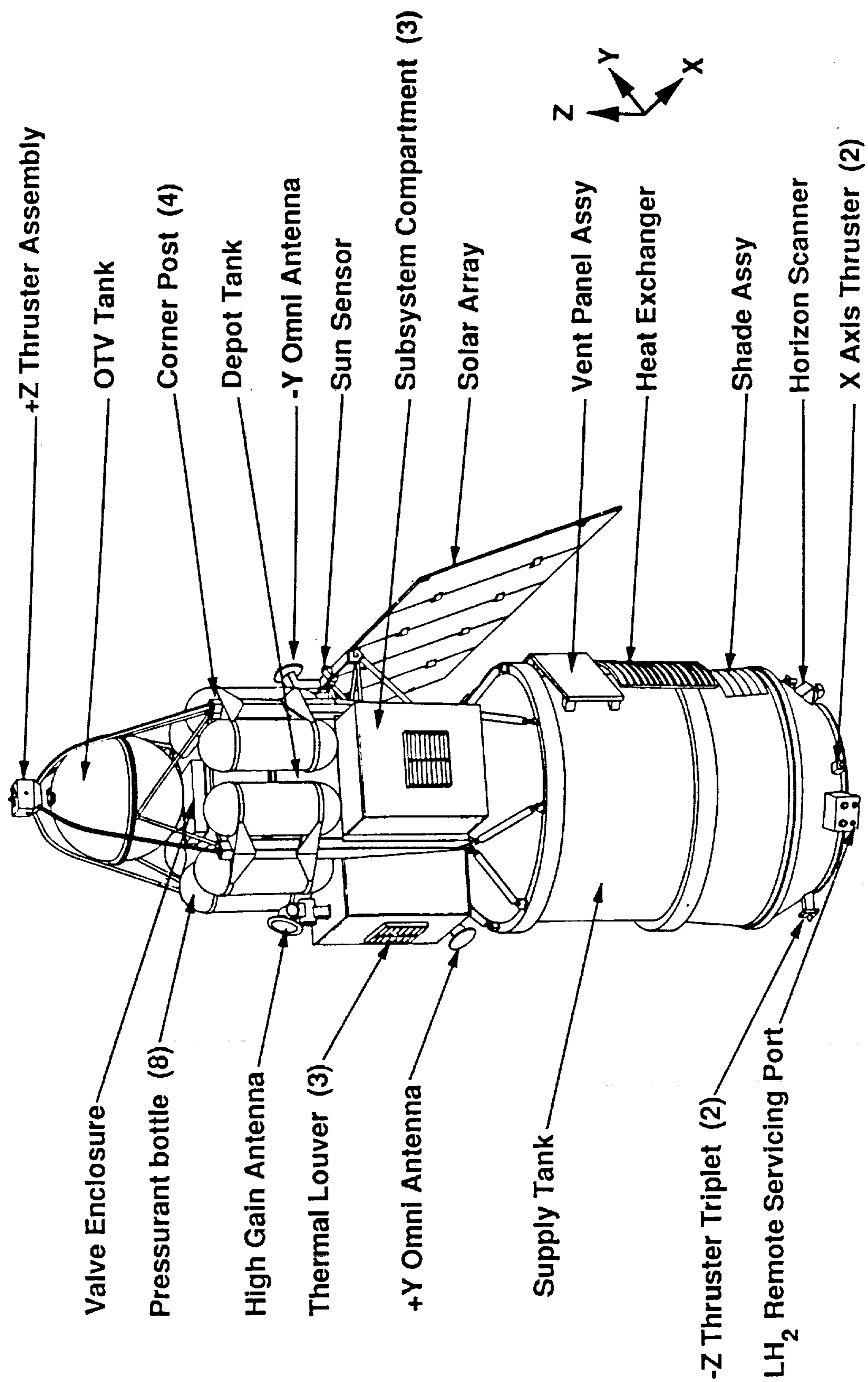

Figure 6-1. COLD-SAT on-orbit configuration.

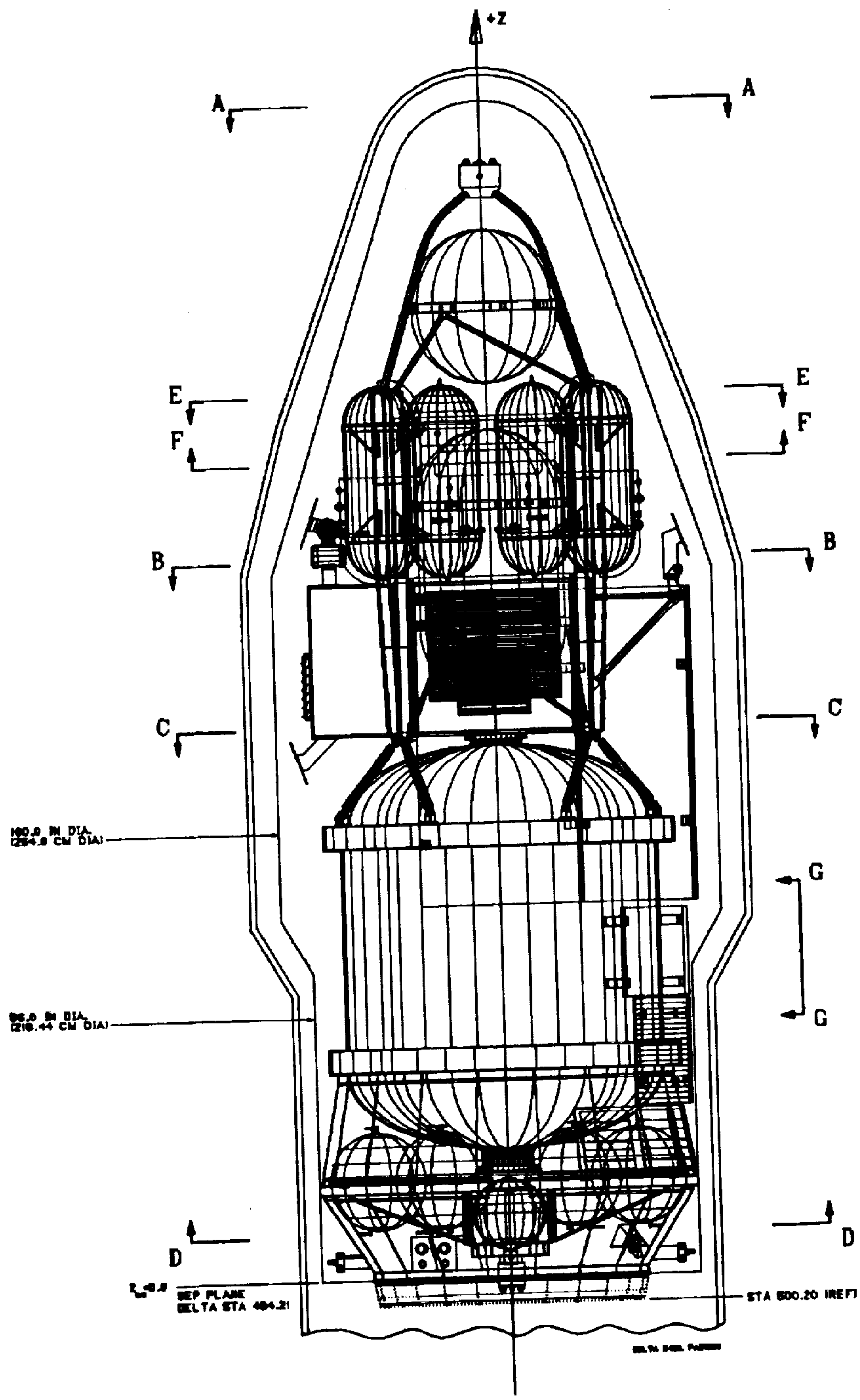

Figure 6-2. COLD-SAT within the Delta fairing (views A-G are given in Figure 6-3).

Table 6-1

CONFIGURATION/STRUCTURAL REQUIREMENTS

ITEM	SOURCE	REQUIREMENT	PERFORMANCE	COMMENT
S/C Envelope Size	Delta Manual	Delta 2.9-m (114-in) fairing	Complies	
S/C CG Position	Delta Manual	Moment <1.05×10^5 N-m Ref. Sta.518	8.1×10^4 N-m	Baseline rivets acceptable using best est. wts.
CG Offset	Delta Manual	X $\leq \pm 5.0$ cm, Y $\leq \pm 5.0$ cm	complies	
S/C Delta Interface Adapter	McDac	Use standard payload attach fitting	Delta 6306	
Cryo Tank	Exp. Subsys.	Liquid hydrogen tanks – Supply tank 3.48 m^3, 2.06 m, dia x 2.51 m – OTV tank 0.33 m^3, 86 m dia x 0.91 m – Depot Tank 0.71 m^3, 0.84 m dia x 1.58 m	complies	
Solar Array	Pwr Budget	>7.6m^2 solar array; 20 deg cant to Z axis	7.6 m^2	10.2 m^2 available
Hydrazine	Accel. Budget	>495 Kg	495 Kg full tanks	(Launched full)
Pressurant Bottles	Cryo Subsys.	2.4 m^3	2.4 m^3	
LH_2 Servicing		Accommodate LH_2 fill, vent lines to mate with umbilical	complies	
Structural Frequency	Delta Manual	15 Hz lateral, 35 Hz axial	18 Hz lateral 35 Hz axial	S/C 1st modes

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6.1.2 Design

6.1.2.1 Configuation Design

Figures 6-3 and **6-4** are detailed configuration views taken from the section cuts on **Figure 6-2**. The S/C configuration was driven by the Delta fairing volume and the mission requirements for payload tank sizing and arrangement, hydrazine, and solar array sizing. The hydrazine tanks were located as low as practical to maintain a low CG for launch load consideration. The Delta 6306 payload attach fitting (PAF) is baselined and uses a marmon type band clamp for S/C separation. These features are shown on the sectioned front view, **Figure 6-2**. The S/C launch support ring lower conical fitting interfaces the S/C to the Delta PAF. The six 22 in. hydrazine tanks and single pressurant tank mount on a reinforced aluminum honeycomb panel to form the hydrazine tank shelf. (View D-D, **Figure 6-4**.) The supply tank support ring

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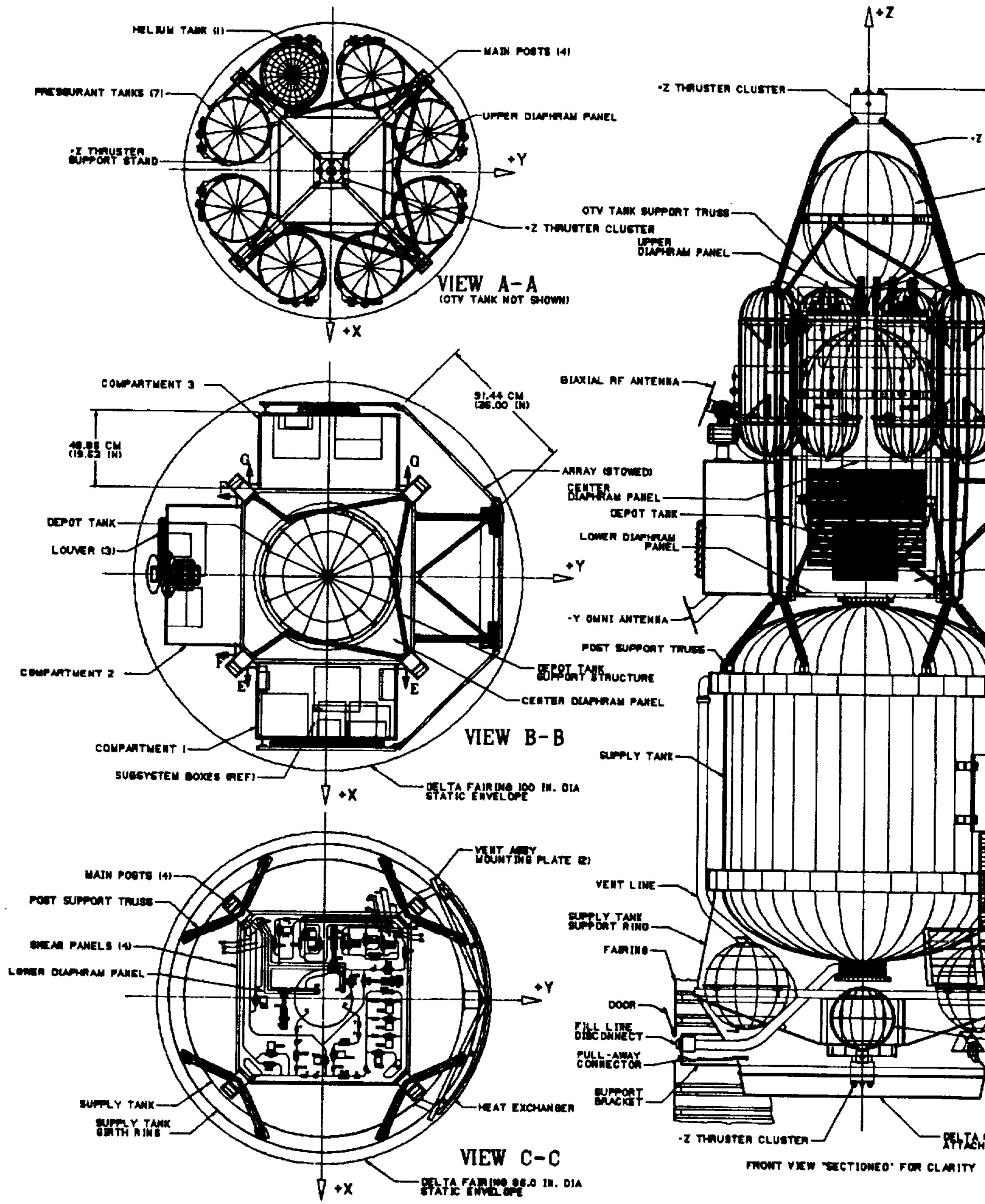

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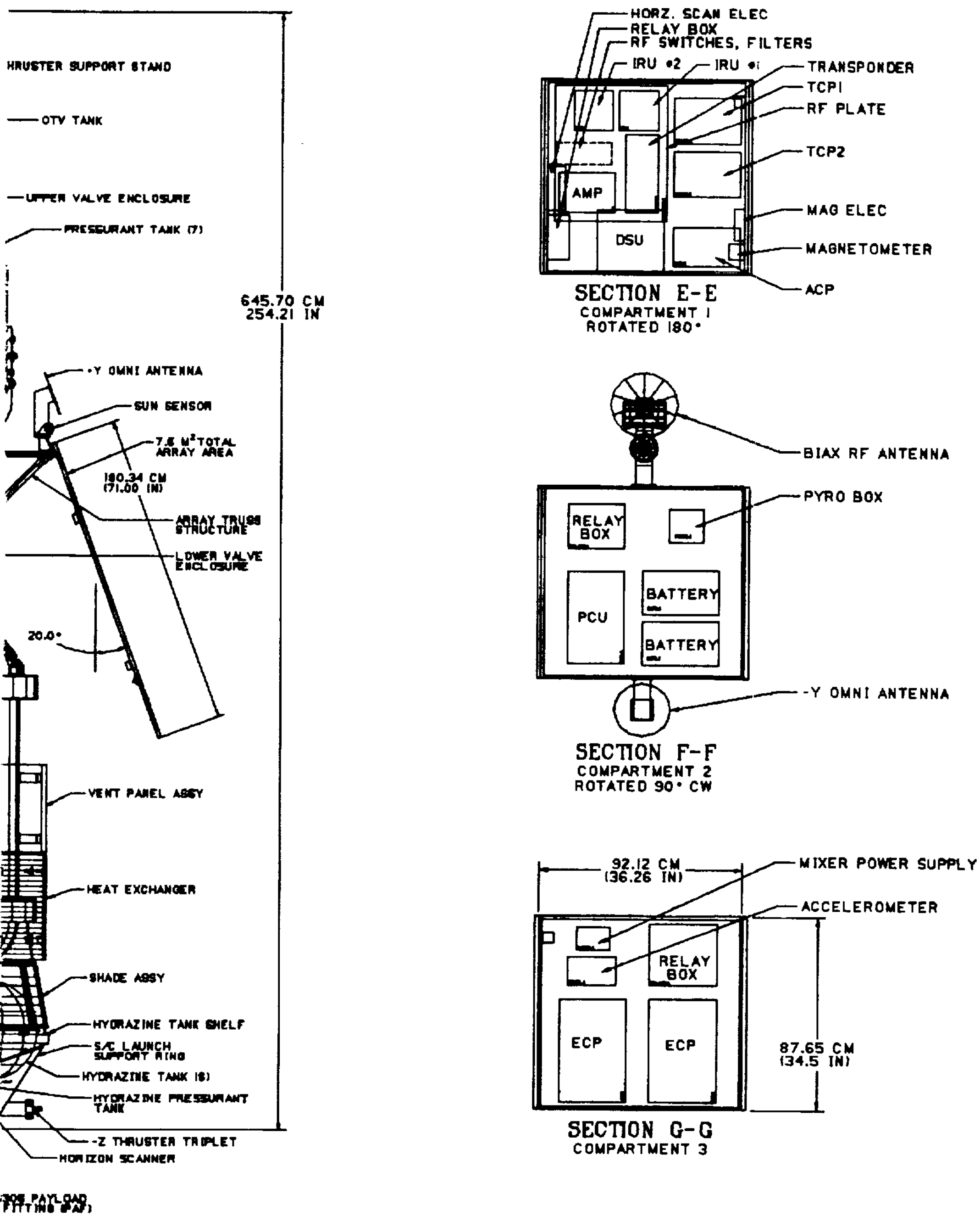

gure 6-3. Configuration details - spacecraft.

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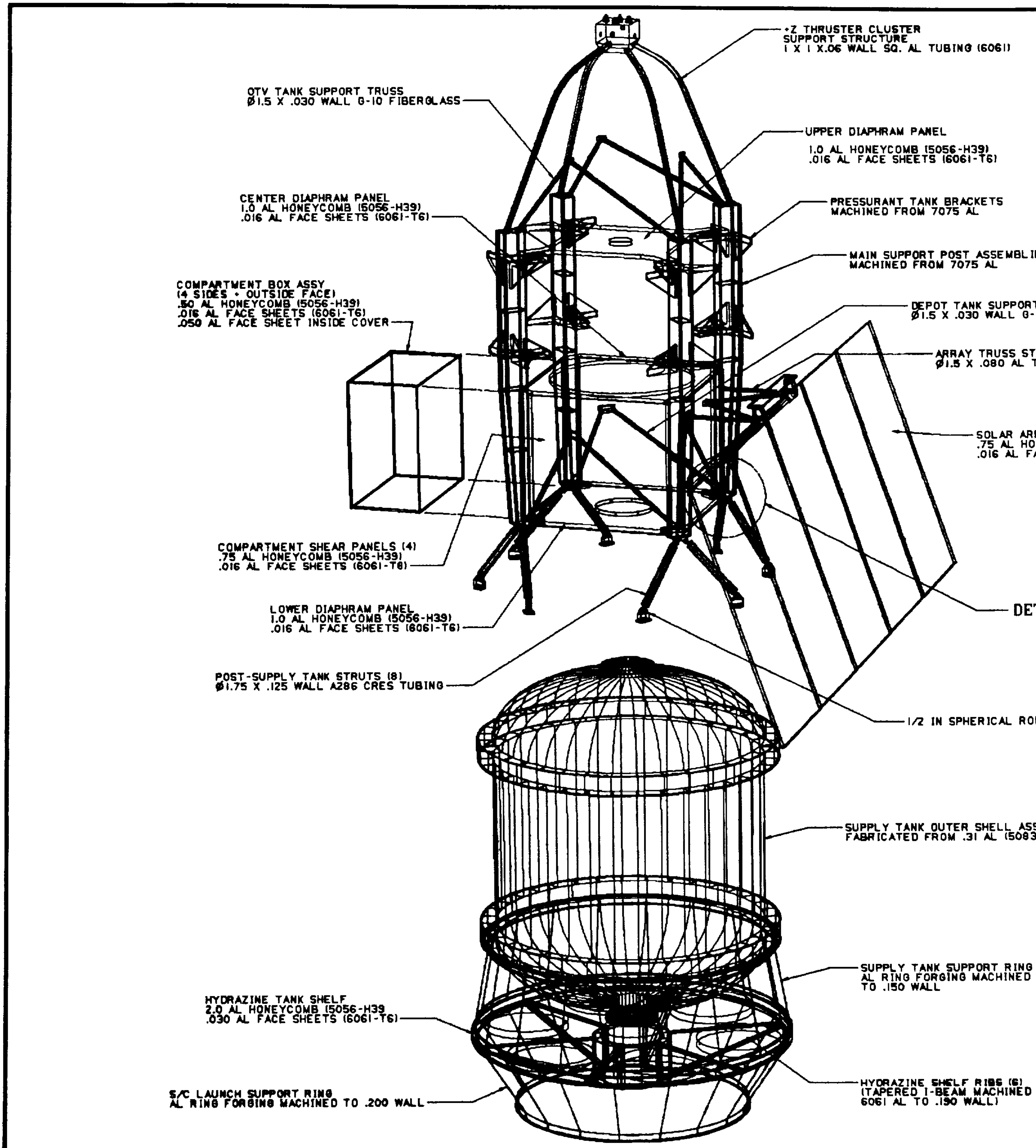

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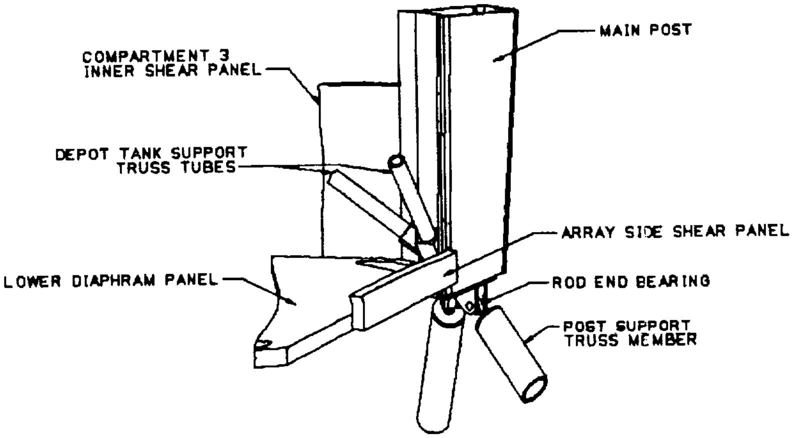

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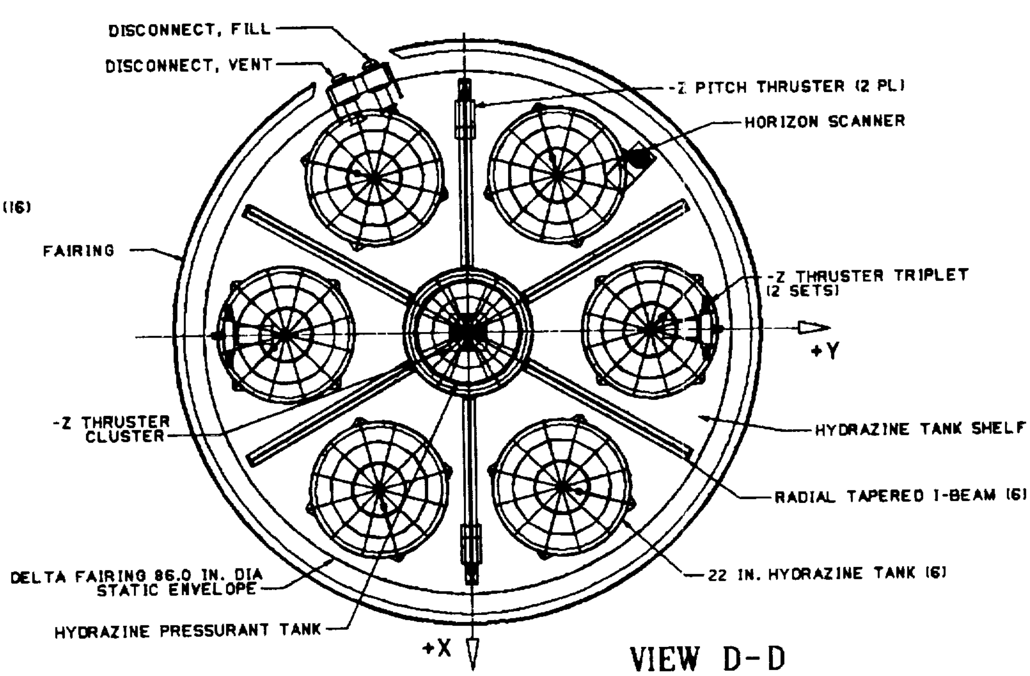

VIEW D-D
(S/C LAUNCH SUPPORT RING
REMOVED FOR CLARITY)

Figure 6-4. Configuration details - structure.

interfaces the hydrazine shelf to the supply tank by attaching to the supply tank lower girth ring. The subsystem boxes are located in three compartments located just above the supply tank (View B-B, **Figure 6-3**. The biaxial RF antenna is located on the top of compartment 2. The sun sensor is mounted to the solar array truss structure. The horizon scanner is located on the outside of the launch support ring to accommodate its field of view requirements (view D-D, **Figure 6-4**). The three subsystem compartments, the arrays, receiver tanks, and pressurant bottles are mounted onto a box structure whose vertical edges are formed by four corner posts attached to the supply tank through an eight member truss structure. The two receiver tanks are independently supported by a fiberglass truss-work mounted to the corner posts.

View A-A of **Figure 6-3** shows the 8 pressurant tanks located above the subsystem compartments. Note that each main post individually supports one pair of pressurant tanks. The OTV tank is removed from this view but its four point-to-three point support truss is shown. The fairing clearance envelope shown is taken at the closest proximity to the pressurant tanks.

View B-B illustrates the box structure formed by the main posts acting as the corners of the box and the depot tank and its support truss inside of the box. The three subsystem compartments hang separately off of the sides of the box. The solar array support truss uses the forth "open" side to mount to. Note the 5 stowed array panels and launch locks mounted on compartments 1 and 3. Eight launch latches, mounted in the outer corners of the array, secure the panels during launch and release the array using redundant pin pullers once on orbit. Eight panel hinges carry integral torsion springs to provide the energy for deploying the array. A damped torsion spring mechanism also provides the 20 deg angular deployment of the array after activation of the pin pullers.

All of the subsystem boxes are located in three identically sized compartments. Sections E-E, F-F, and G-G shown in **Figure 6-3** are views into compartments 1, 2, and 3 respectively. The subsystem compartments are fabricated from 1/2-in. aluminum honeycomb and are hinged from the shear panels to provide easy access. The subsystem boxes are generally located on the outer face of each enclosure for thermal reasons. A passive louver is located on the opposite face of each of the outer panels to maintain the boxes within their temperature limits. Compartment 1 houses the attitude control and TT&C boxes, compartment 2 houses the power subsystem components, and compartment 3 houses the experiment processors. Note compartment 2 is also used to locate the Biaxial RF antenna and the -Y omni antenna.

6.1.2.2 Structural Design

The COLD-SAT structure is based on using the supply tank outer shell (vacuum jacket) as a primary load carrying member. The shell upper girth ring provides for attachment of struts which support the subsystem boxes, receiver tanks, and accompanying structures (see structured arrangement view, **Figure 6-3**). The main support posts are machined from aluminum and together create the framework for supporting these elements. The four 3/4-in. thick honeycomb compartment shear panels stabilize the lower half of the posts and provide areas for attaching the subsystem compartments and the solar array support structure. The box structure formed by the lower posts and compartment shear panels is closed out with three diaphragm panels. The upper diaphragm panel is notched for clearance for the pressurant bottles. A pair of pressurant bottles is hung from each corner post by a set of machined aluminum brackets. The center diaphragm panel has a large hole for clearance around the depot tank. The supply tank support ring and s/c launch support ring are machined from 1 piece aluminum ring forgings. The supply tank support ring tapers inward to transfer the launch loads to the inboard edge of the supply tank lower girth ring. All of the primary structure will be fabricated from aluminum except stainless steel will be used for the supply tank struts, and fiberglass for the depot and OTV tank truss members for improved thermal isolation.

6.1.2.3 Mass Properties

Table 6-2 summarizes the COLD-SAT mass property data. The subsystem weights are best estimates, i.e., without any contingency factor added. The Delta 6306 Payload attach fitting (PAF), although a payload chargeable item, is

Table 6-2

MASS PROPERTY DATA

SUBSYSTEM	WEIGHT kg	WEIGHT lb
Structure	517.0	1140.0
Science (dry)	1258.2	2774.3
Power	164.3	362.3
Propulsion (dry)	88.4	194.9
Thermal	47.1	103.8
ADCS	19.0	41.9
TT&C	71.7	158.1
Ballast, Balance	36.0	79.4
Dry S/C	2201.7	4854.7
Hydrazine	495.0	1091.5
Propulsion Pressurant	8.5	18.8
Liquid Hydrogen	234.3	516.6
Gsaeon Hydrogen	18.9	41.7
Gaseous Helium	5.0	11.0
Wet S/C	2963.4	6534.3
Delta 6306 PAF Maron	0	0
Total	2963.4	6534.3
Vehicle Allowable	4200.0	9261.0
Launch Margin	1236.6	2726.7

shown as 0 weight because its mass is already factored into the vehicle allowable weight of 4,200 Kg. Setting its weight at 0 therefore provides the actual launch margin (difference between the allowable S/C weight to achieve its intended orbit and the actual S/C weight). The vehicle allowable weight is determined from the Delta Performance curve shown in Figure 5-6, Section 5.4.

The principal axis inertias are summarized in **Table 6-3**. Note that these values are calculated using best estimate weights. The difference between Beginning of Life and Launch is to account for the array deployment. The end of life condition reflects all liquids (fuel, pressurant, and science LH_2) being set to 0. Also shown is the S/C center of gravity position at launch.

The Z position is the dimension above the S/C separation plane (Z = 0.0). The X and Y values are toleranced to indicate the level of accuracy the S/C will be balanced to about these axes.

Table 6-3
S/C INERTIAS & CG

S/C Launch CG Position
Z = 229.1 cm (90.2 in)
X = 0.0 ± 1.0 cm
Y = 0.0 ± 1.0 cm

	SEQUENCED INERTIAS $Kg\text{-}m^2$		
CONDITION	I^*_{xx}	I^*_{yy}	I^*_{zz}
Launch	7910	7956	1716
Beginning of Life	7976	7998	1819
End of Life	5608	5630	1445

* Calculated without contingency

6.1.2.4 Mechanisms

The only moving hardware on the S/C is the solar array (1 time deployment to a locked position) and the biax RF antenna and the horizon scanner which are self contained units driven for the life of the mission and not considered mechanisms per se. The array is configured such that two movements are necessary to fully deploy the array from its stowed position; the unwrapping motion and the 20° tilting motion. These operations are allowed to happen simultaneously since sequencing them provide no reliability or operational benefits and would only add to hardware complexity (i.e., cost). The array is secured for launch by using a pair of redundant pin pullers on the outboard corners of each array except for the center panel (1 pr. per each of 4 panels = 8 pin pullers). The center panel is restrained by the outboard panels and the torsion drive hinges on its upper edge. The 8 passive hinge mechanisms between the array panels include integral torsion springs, dampers, and position stops/locks to unwrap the panels from the S/C once the pin pullers are activated. The pair of hinges on the center panel upper edge also include integral torsion springs, dampers, and position stops/locks on a larger scale. These passive hinges are used to rotate the array to its 20° cant position and subsequently provide the structural interface between array and spacecraft while on orbit. Note that there are no motors involved in the deployment, only electrical signals to the pin pullers are required.

6.1.3 Structural Analysis

A 413 node, 575 element NASTRAN model was created to verify the structural integrity of the design and to show that the design meets the Delta stiffness requirements. Loads were calculated by superimposing the sinusoidal and random vibrations over the steady state accelerations and multiplying the factors by 1.25 to obtain the ultimate loads. Most margins of safety proved to be high, indicating the structural sizing is more stiffness driven than strength driven (except for the receiver tanks support trusses). The first lateral mode occurs at 18 Hz with bending of the tops of the main posts. The first axial mode occurs at 35 Hz with a vertical diaphragm action of the hydrazine shelf. These frequencies are within the Delta requirements. Below is a tabulation of the primary structural elements of interest their margin of safety and the first significant modes associated with them.

Margins of Safety*

Post Support Truss	=	+3.7
Depot Tank Truss	=	+0.3
Main Posts	=	+10.3

$$\text{*Margin} = \frac{\text{Yield Load}}{\text{Worst case (ultimate) loading x f.o.s}} - 1$$

Stiffness

18 Hz - Lateral bending of posts above shear panels (1st lateral mode)
26 Hz - Torsional bending of posts above shear panels

28 Hz - Lateral movement of Depot tank
35 Hz - Vertical oscillation of hydrazine shelf (1st axial mode)
35 Hz - Supply tank pressure vessel

Delta requirement >15 Hz lateral, 35 Hz axial

6.1.4 Components

Table 6-4 summarizes the structure subsystem components with mechanisms broken out separately.

Table 6-4
STRUCTURE SUBSYSTEM COMPONENTS

COMPONENT	(kg)*	(watts)**	VENDOR	COMMENT/HERITAGE
Primary Structure	421.1	--	BASG	
Secondary Structure	84.8	--	BASG	
Array Deployment, hinge mechanism				
set #1	1.5	0.0	BASG	ERBS
set #2	1.5	0.0	BASG	ERBS
Array Deployment, Panel Drive	5.1	0.0	BASG	RADARSAT
Array Deployment, Launch Locks				
Qty 8 0.37x8	= 3.0	150		ERBS
TOTAL	517.0	150 W		

* best estimate
** per unit when active

6.2 THERMAL CONTROL SUBSYSTEM

The thermal control subsystem (TCS) provides adequate temperature margins on all spacecraft components from launch through on-orbit operations in addition to providing flexibility for low temperature protection during contingency operations. A 3.14 radian (180 deg) yaw reversal maneuver limits the sun angle between 0 and 0.91 radians (52 deg) with a maximum sun angle of ±1.08 radians (±62 deg) during contingency operations. Thermal control of COLD-SAT is achieved through the use of louvers, thermal control surfaces, multi-layer insulation (MLI) and temperature controlled heaters. The thermal control subsystem is completely autonomous.

6.2.1 Thermal Control Requirements

TCS temperature limits for components and structure, shown in **Table 6-5**, are derived requirements based on performance criteria set by the various subsystem designs. Expected performance meets all requirements.

Table 6-5

THERMAL SUBSYSTEM REQUIREMENTS

	REQUIREMENT	SOURCE	PERFORMANCE	MARGIN*	COMMENTS
ACS Subystem					
ACP	-10° to 50°C	ACS	21° to 28°C	22	
IRU	-10° to 50°C	ACS	17° to 26°C	24	
Horizon Sensor Elec.	-10° to 50°C	ACS	25° to 30°C	20	
Magnetometer Elec.	-10° to 50°C	ACS	18° to 26°C	24	
TT&C Subsystem					
TCP #1	-10° to 50°C	TT&C	19° to 25°C	25	
TCP #2	-10° to 50°C	TT&C	17° to 23°C	27	
Transponder	-10° to 50°C	TT&C	19° to 26°C	24	
RF Power Amp	-10° to 50°C	TT&C	14° to 36°C	14	
DSU	-10° to 50°C	TT&C	23° to 31°C	19	
EPD Subsystem					
Battery #1	0° to 20°C	EPS	10° to 17°C	3	
Battery #2	0° to 20°C	EPS	10° to 17°C	3	
PCU	-10° to 50°C	EPS	12° to 23°C	22	
Pyro Box	-10° to 50°C	EPS	10° to 21°C	20	
Solar Array	-100° to 70°C	EPS	-60° to 66°C	4	
Misc					
Relay Box #1	-10° to 50°C		22° to 27°C	23	
Relay Box #2	-10° to 50°C		10° to 21°C	20	
Relay Box #3	-10° to 50°C		16° to 23°C	26	
Experiment					
ECP #1	0° to 30°C	Exp.	18° to 27°C	3	
ECP #2	0° to 30°C	Exp.	14° to 20°C	10	
Accelerometer	-10° to 50°C	Exp.	20° to 28°C	22	
Mixer Power Supply	-10° to 50°C	Exp.	16° to 24°C	26	
OTV Tank Blanket	-60° to 50°C	Exp.	-44° to 10°C	16	
Pressurant Tanks	0° to 35°C	Exp.	4° to 11°C	4	Heater Controlled
Depot Tank Blanket	-60° to 50°C	Exp.	11° to 14°C	36	
Suppy Tank Skin	-40° to 50°C	Exp.	-34° to 34°C	6	
Upper Valve Box	-40° to 50°C	Exp.	6° to 7°C	43	
Lower Valve Box	-40° to 50°C	Exp.	16° to 23°C	27	
Propulsion					
Hydrazine Tanks	10° to 50°C	Prop.	11° to 37°C	1	Heater Controlled
Pressurant Tank	10° to 50°C	Prop.	13° to 31°C	3	

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*Worst Cast Margin (°C)

Typical baseplate temperature limits of -10 to 50 °C provide for a wide operational range while minimizing required thermal control and heater power. Selecting a low battery temperature range provides for high reliability and cell longevity while limiting solar array maximum temperatures reduces solar array size. Maintaining narrow temperature limits for the ECP protects the accuracy of the experiment temperature probes and limiting the experiment component minimum temperatures assures regulator and flow meter capability is not exceeded. The propulsion subsystem temperature range provides for protection against freezing and excessive tank pressures.

6.2.2 Thermal Control Design

Key features of the TCS are summarized in **Figure 6-5**. Selected thermal finishes maintain acceptable temperature ranges by balancing absorbed and radiated heat while MLI blankets minimize heat loss to space. Louvers are used to minimize temperature variation due to transient heat loads and changing environment. Shades are used where the sun is relied upon for passive control, temperature ranges are narrow and variation in the sun angle over time is large. Heaters are used where the above methods are insufficient or for low temperature protection during contingency operations.

Each equipment compartment rejects heat through a Fairchild louver sized for the expected peak heat load. Louver blade operation open to closed is 10 °C to minimize orbital temperature extremes and heater power. MLI covers all other external compartment surfaces. Most of the power dissipating boxes are mounted directly on the radiator panel. Internal surfaces with power dissipating boxes are painted to enhance heat transfer to the radiator panel and minimize internal gradients. For high power dissipating boxes, RTV is used in the baseplate/panel interface to enhance conduction to the radiator. The RF power amp is mounted on a 0.635 cm (0.25 inch) aluminum plate to mitigate temperature transients.

The experiment tanks are enclosed in MLI which is secured in place by blanket anchors attached to vertical guide wires. The equipment compartments provide a heat source for the pressurant tanks by radiating heat through the painted

1.

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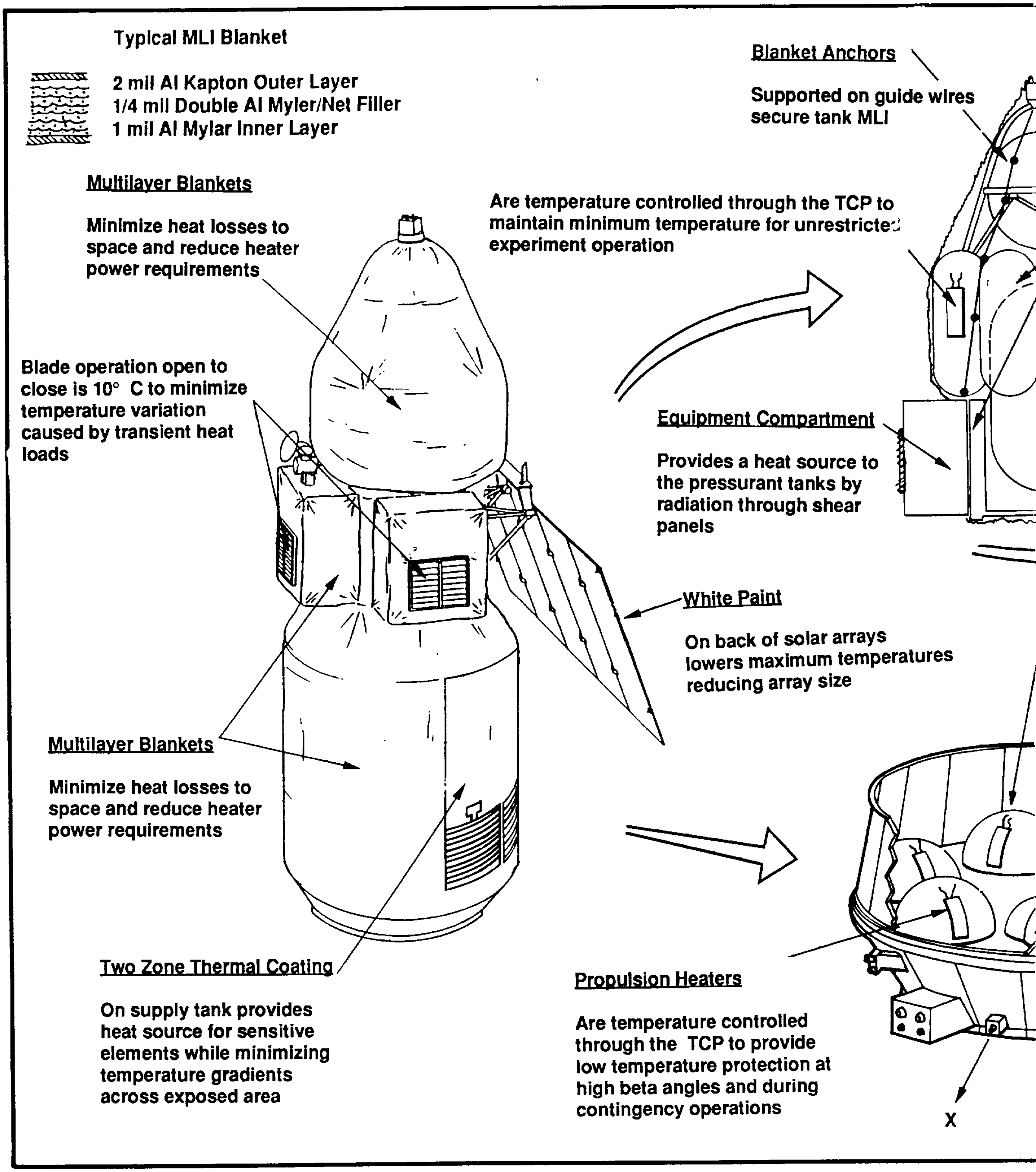

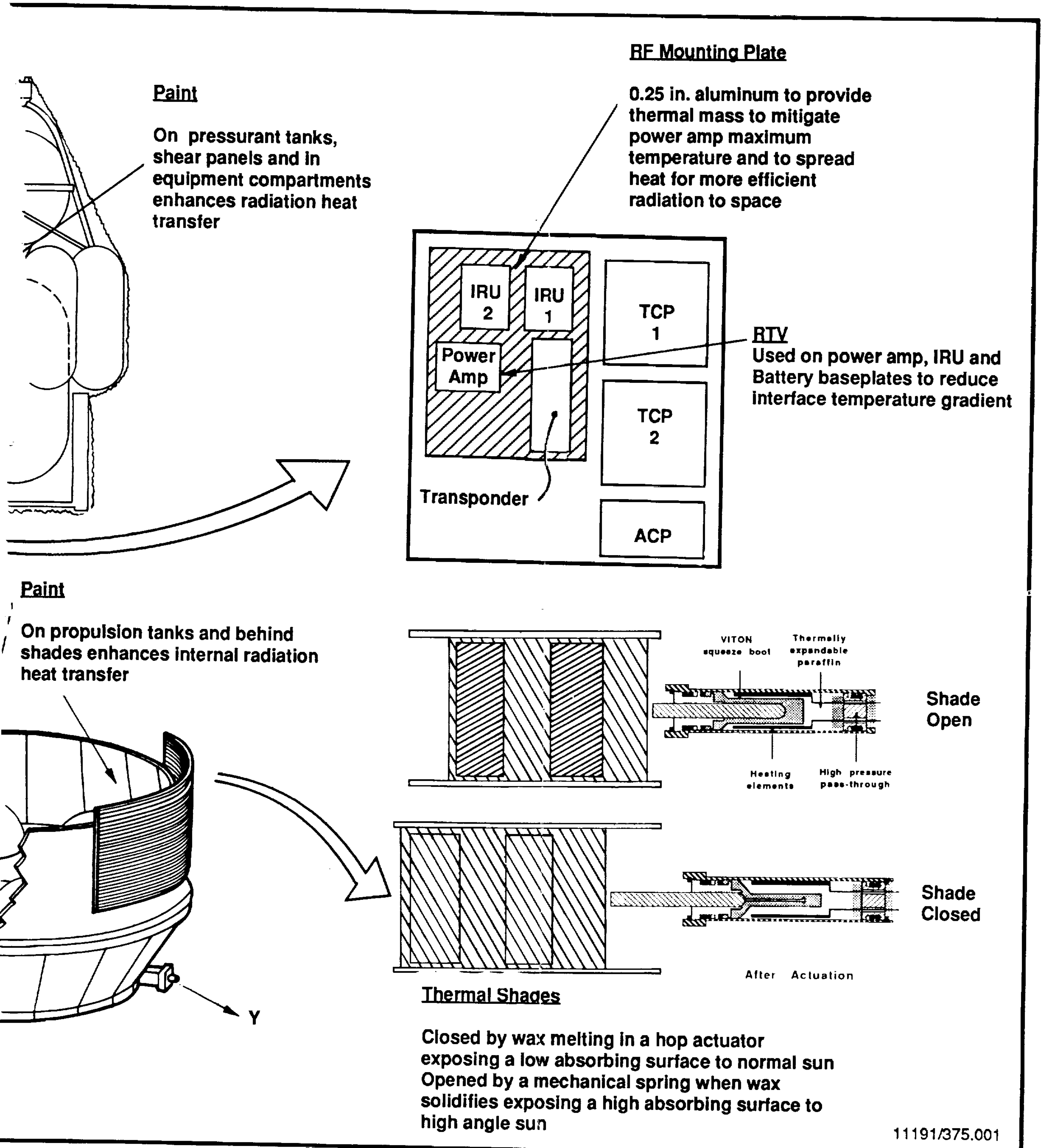

Figure 6-5. Thermal control design key features.

shear panel into the tank enclosure. All surfaces are low emissivity except the pressurant tanks which are painted to enhance heat transfer. Two watt heaters on the pressurant tanks allow unrestricted experiment operation.

The sun is used as a heat source for warming temperature sensitive elements near or on the supply tank. This is feasible due to the short length of the mission where the supply tank outer shell can be warmed without causing significant boil off of the cryogen. The supply tank exterior thermal control has a two zone thermal coating on the sun side to minimize temperature gradients across the exposed surface area while the remaining exterior is insulated to mitigate minimum temperature extremes. The zone normal to the sun has a surface coating with an α/ϵ=0.25 while the zone at an angle to the sun has an α/ϵ=0.50.

The sun is also used as a heat source for the propulsion subsystem. A support ring mounted thermal shade allows for surface property variation providing for narrow band temperature control and heater power minimization. The shade is closed by a high output paraffin (HOP) thermal actuator driven by the expansion of melting wax and opened by a mechanical spring when the wax solidifies. The shade can be manually closed with a ground commandable heater. The support ring interior under the shades and the hydrazine tanks are painted to enhance heat transfer. Propulsion compartment parasitic heat leaks are minimized by using internal and external MLI elsewhere on the support ring and by using an aft closeout blanket. Propulsion lines are located toward the sun side of the spacecraft and spiral wrapped with MLI to minimize heater power. Cat bed heaters and MLI dog houses maintain thruster cluster temperatures.

6.2.3 Thermal Control Analysis

6.2.3.1 Thermal Model and Case Parameters

Transient thermal analysis was performed with a 101 node spacecraft model and the thermal programs described in **Figure 6-6**. Several subsystem detail models were developed to provide additional information in thermally critical

SUPVIEW	Computes geometric view factors for radiation heat transfer models
ALBEDO	Computes incident orbital fluxes on surfaces
REFLECT	Applies diffuse surface finishes to radiation model and computes radiation interchange factors and absorbed orbital fluxes
TAK II	General purpose finite differencing thermal analyzer similar to Sinda-85. Can actively model louvers and heaters in transient and steady state analysis

Spacecraft Model
- 101 nodes
  20 for boxes
  3 for louvers
  26 for structure
  51 for MLI
  1 boundary
- 248 conductors

Detailed subsystem models were developed for thermally critical areas
- Experiment pressurant tanks
- Supply Tank
- Propulsion Platform

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Figure 6-6. Thermal analysis model

areas. Nominal and failure cases were examined. Hot and cold case temperatures were determined using stacked worst cases, shown in **Figure 6-7**, so no additional modelling error is included in the results.

To allow for component maturation, the thermal design accommodates power dissipation uncertainty factors. Transient powers (Transponder and RF Amp) and the additional ECP power (6.4 watts) are used only in the hot case analysis. Battery power dissipation assumes an 85 percent power conversion efficiency during eclipse and a 3 percent overcharge. Component power dissipation for the failure mode is a minimum power configuration with batteries cycling nominally.

PARAMETER	NOMINAL CASE	HOT CASE	COLD CASE	FAILURE CASE
Orbit angle (Beta)	36 deg	0 deg	52 deg	52 deg
Sun angle	36 deg	0 deg	52 deg	62 deg
Solar constant ($^{W}/m^3$)	1354	1420	1288	1288
Earth shine ($^{W}/m^3$)	237	244	230	230
Albedo constant	0.30	0.32	0.28	0.28
Length of Eclipse (minutes)	30	34	21	21
MLI ε*	0.015	0.005	0.025	0.025
Power dissipation (watts)	Nominal	Maximum	Minimum	Survival
Compartment # 1				
TCP # 1	8.4	9.7	7.1	2.7 (standby)
TCP # 2 (standby)	3.2	3.7	2.7	2.7
Transponder	12.0	12.0 + (18.0)	12.0	12.0
RF amp	0.0	(172)	0.0	0.0
ACP	7.1	8.2	6.0	0.0
IRU # 1	24.0	24.0	24.0	0.0
DSU	6.5	7.5	5.5	0.0
Horizon sensor electronics	7.0	7.0	7.0	0.0
Magnetometer electronics	1.1	1.2	1.0	0.0
Compartment # 2				
Battery 1	20/4	26.2/5	15/3	3.7/0.7
Battery 2	20/4	26.2.5	15/3	3.7/0.7
PCU	7.8	8.2	7.4	7.4
Compartment # 3				
ECP # 1	15.8	18.2 + 6.4	13.4	0.0
Accelerometer	8.0	8.4	7.6	0.0
Mixer power supply	0.5	0.6	0.4	0.0

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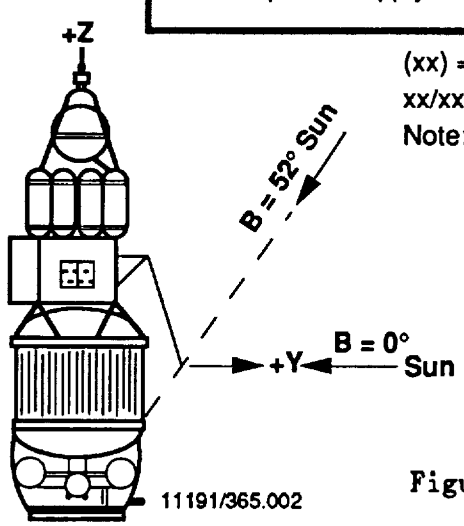
(xx) = 10 minute duty cycle, hot case only
xx/xx = discharge/overcharge power dissipation
Note: RF power amp is 172 + 28 RF = 200W



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With 180 deg yaw reversal maneuver, sun angle is between 0 and 52 deg. In failure mode, maximum sun angle is ±62 deg

Figure 6-7. Thermal design cases.

6.2.3.2 Analysis Results

Transient component power was the driving factor in the increased temperature swings seen in the hot case. With stable powers, orbital temperature variation is small, generally less than 2 °C with the exception of the OTV tank blanket and the solar arrays. Overall temperature variation due to changing beta angle was generally less than 5 °C again with the exception of the OTV tank blanket and the solar arrays. Temperatures are summarized in **Table 6-6**.

Power Amp temperatures in the hot case increase 18 °C in 10 minutes when the amp is cycled. The IRU shows a temperature rise in the hot case because of its proximity to the power amp in compartment #1. Battery temperatures show stability despite orbital charging/discharging and varying spacecraft power requirements. ECP temperature variation between hot and cold case, including 6.5 watt of transient power in the hot case, is well within the allowable temperature range.

Receiver tank temperatures are constant with the exception of the OTV tank blanket which is decoupled from spacecraft heat sources. MLI nodes are modelled as arithmetic nodes (i.e., no mass) where thermal response is instantaneous. This is the primary reason for the large OTV tank blanket temperature changes during eclipse. Propulsion temperatures show little environmental effect due to the large mass. Solar array temperature response is quick due to low structural weight.

During the failure mode, the spacecraft is spun about the +Y axis, components are set to a minimum power configuration and the sun angle may vary ±10 deg from nominal. The analysis assumed a non-spinning spacecraft with a sun angle of 62 deg toward the +Z axis which provided a worst case for the propulsion compartment. Minimum margins occur in the battery compartment due to the lower spacecraft power demand and in compartment 3 where all experiment boxes are off.

Table 6-6

MIN/MAX TEMPERATURE PREDICTIONS

TEMPERATURE PREDICTIONS (°C)	HOT CASE		NOMINAL CASE		COLD CASE		SURVIVAL CASE	
	MIN	MAX	MIN	MAX	MIN	MAX	MIN	MAX
ACS Subsystem								
ACP	28	28	25	25	21	21	5	5
IRU	19	26	18	25	17	19	5	6
Horizon Sensor Elec.	30	30	27	28	25	26	5	6
Magnetometer Elec.	26	26	22	23	18	19	3	4
TT&C Subsystem								
TCP #1	24	25	22	22	19	19	6	6
TCP #2	22	23	20	21	17	18	6	6
Transponder	22	26	21	24	19	20	10	10
RF Power Amp	17	36	15	34	14	16	5	6
DSU	31	31	27	27	23	23	4	4
EPD Subsystem								
Battery #1	15	17	12	14	10	11	3	4
Battery #2	15	17	12	14	10	11	3	4
PCU	23	23	17	17	12	13	3	3
Pyro Box	21	21	15	15	10	10	0	1
Solar Array	-60	63			-53	50		
Misc								
Relay Box #1	27	27	24	25	22	23	5	6
Relay Box #2	21	21	15	15	10	10	0	1
Relay Box #3	23	23	18	18	16	16	-4	-4
Experiment								
ECP #1	26	27	20	21	18	18	-4	-4
ECP #2	20	20	16	16	14	14	-4	-4
Accelerometer	28	28	23	23	20	21	-4	-4
Mixer Power Supply	23	24	18	19	16	17	-4	-3
Tanks & Valves								
OTV Tank Blanket	-38	0	-50	7	-54(-44)	5(10)	-60	-2
Pressurant Tanks	6	9	-6	0	-15(4)	-7(11)	-25	-19
Depot Tank Blanket	14	14	7	7	2(11)	2(11)	-11	-11
Supply Tank Skin	9	31			-37	-20		
Upper Valve Box	6	6	-4	-4	-10(7)	-10(7)	-21	-21
Lower Valve Box	23	23	20	20	16	16	2	2
Propulsion								
Hydrazine Tanks	25	34	12/27	19/38	5(11)	14(21)	-8(10)	0(14)
Pressurant Tank	29	29	14/31	14/31	8(13)	8(15)	-6(11)	-6(12)

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Note: Temperatures shown are without heater power
(xx) = temperatures with heaters
A/B = A, shades closed B, shades open

Representative transient temperature curves are shown in **Figure 6-8** with time = 0 representing where the spacecraft crosses the dawn terminator. Hot case eclipse (i.e., when spacecraft enters Earth's shadow) occurs from 1.01 to 1.58 and cold case eclipse occurs from 1.12 to 1.47.

Average heater power was determined for hot and cold nominal cases, shown in **Figure 6-9**. The hot case requires only propulsion line heater power. At high beta angles, minimal propulsion heater power is required and for low power configurations, pressurant tank heat power is required. Pressurant tank temperatures are critical only when an experiment is run so only propulsion heater power is required during a failure mode.

6.2.4 Thermal Control Components

A thermal control component summary is shown in **Table 6-7**. All components have been flown on BASG spacecraft and meet failure rate criteria.

6.3 ELECTRICAL POWER SUBSYSTEM

6.3.1 Requirements and Constraints

The basic requirements of the Electrical Power System (EPS) are to satisfy the electrical energy demands of the COLD-SAT during all phases of mission activity. Electrical loads were analyzed to determine the worst case minimum and maximum power conditions and the solar panels and batteries sized accordingly. The major requirements are summarized in **Table 6-8**. The EPS performance is based on a one year mission, a 926 km orbit with a period of approximately 103.5 minutes and solar panel illumination varying from 69.5 to 82.5 minutes.

6.3.2 Functional Description

A functional diagram of the EPS is shown in **Figure 6-10**. The batteries are connected in parallel with the solar panel outputs and spacecraft loads. Solar panel power and/or battery discharge power is connected directly to the

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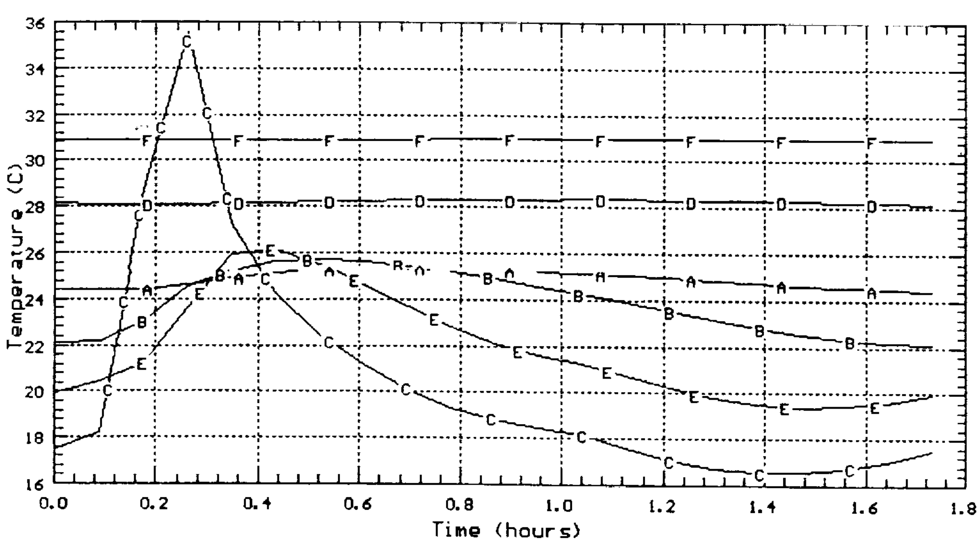
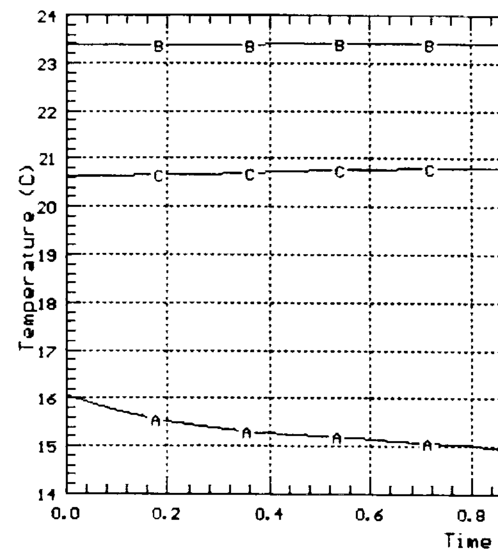
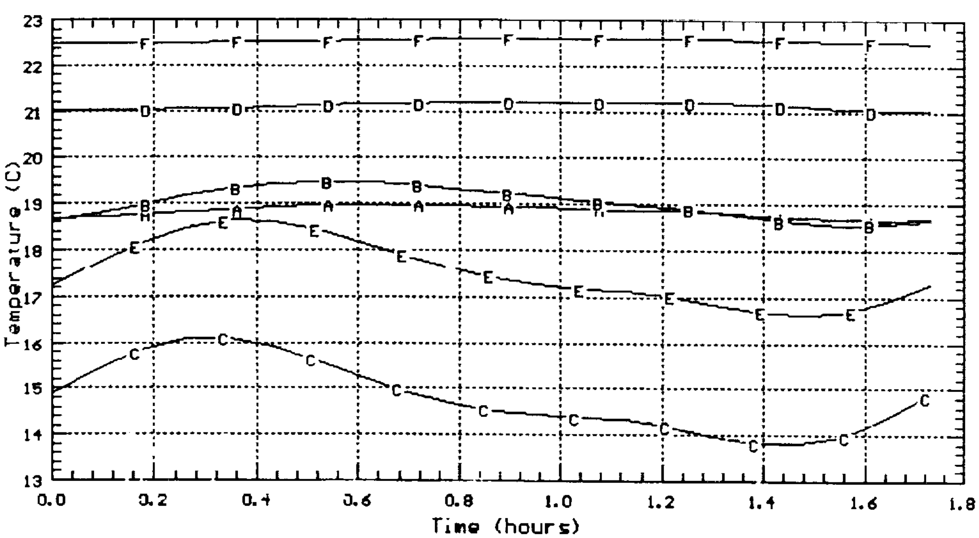
Compartment No. 1 Worst Case Hot



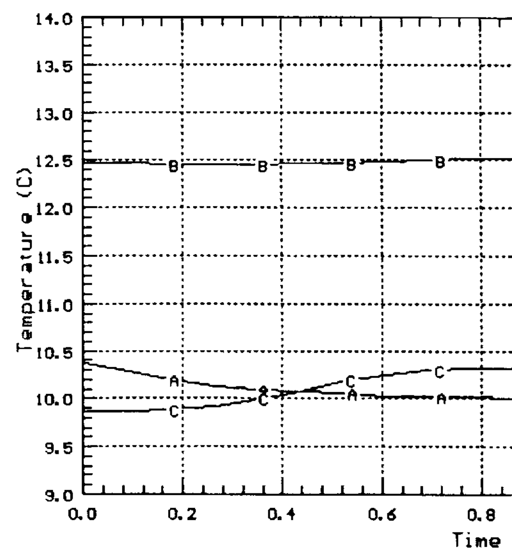
Compartment No.



Compartment No. 1 Worst Case Cold



Compartment N



CURVE SUMMARY	TEMPERATURE (°C)	
	MINIMUM	MAXIMUM
A = TCP # 1	19	25
B = Transponder	19	26
C = RF AMP	14	36
D = ACP	21	28
E = IRU	17	26
F = DSU	23	31

CURVE SUMMARY	M
A = Batteries	
B = PCU	
C = Pryo box	

Figure 6-8. Transient tempera

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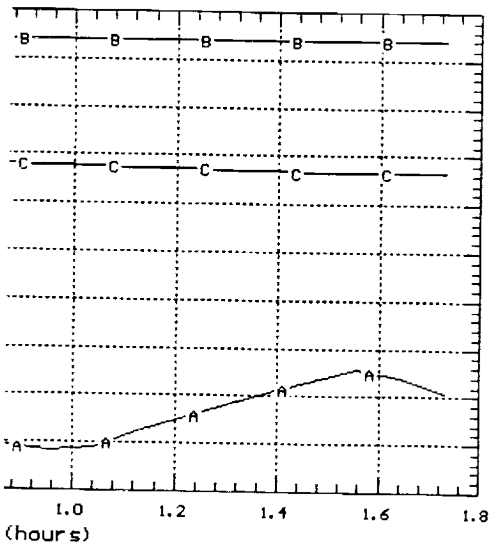
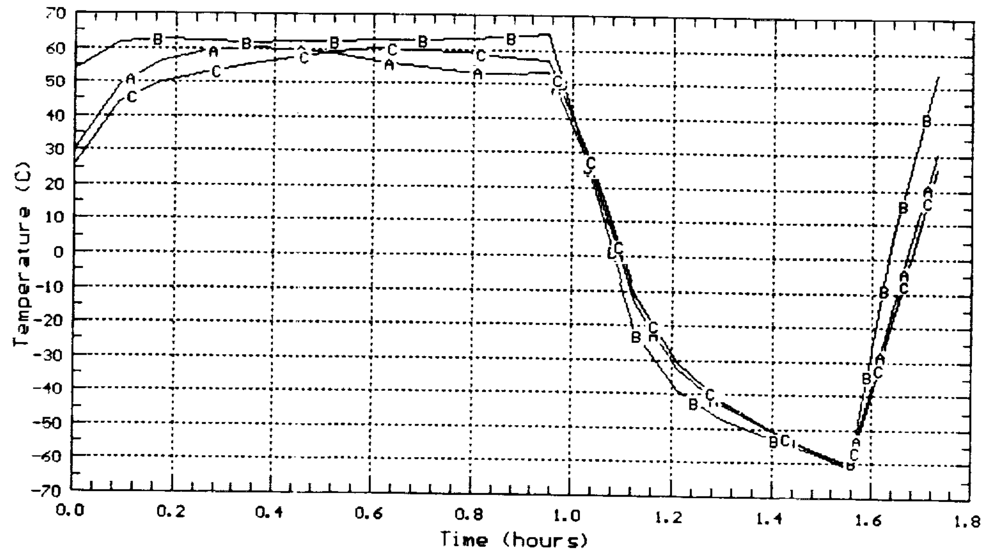
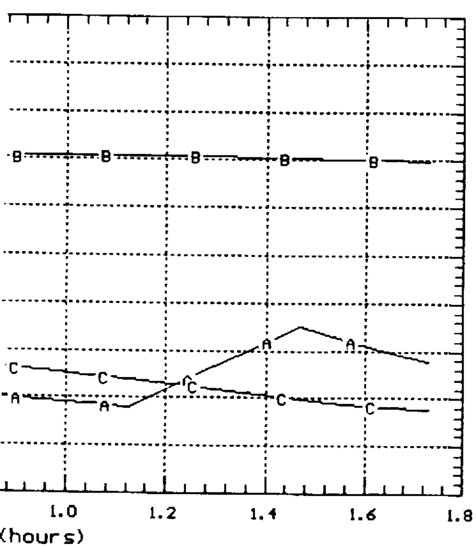
2 Worst Case Hot



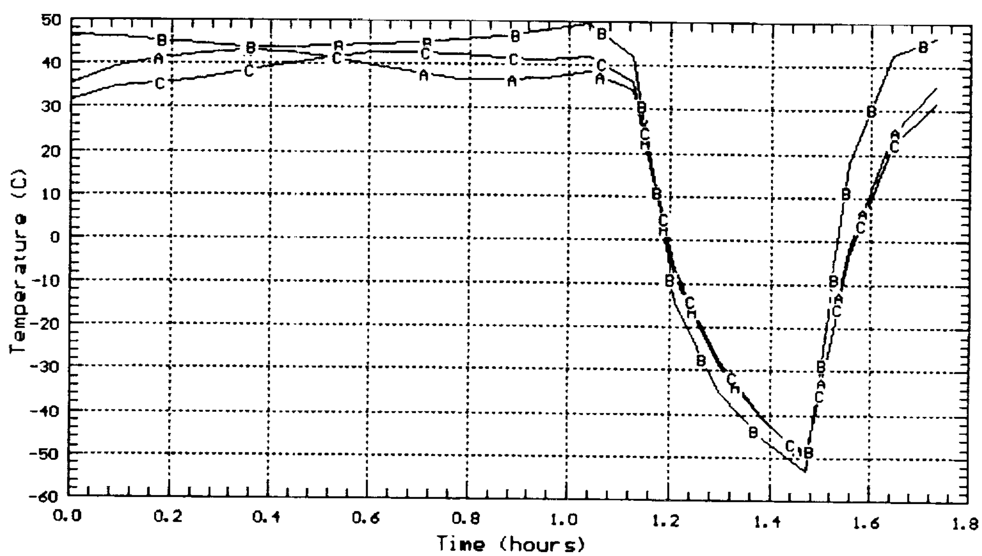
Solar Array Worst Case Hot



). 2 Worst Case Cold



Solar Array Worst Case Cold



TEMPERATURE (°C)	
INIMUM	MAXIMUM
10	17
12	23
10	21

CURVE SUMMARY	TEMPERATURE (°C)	
	MINIMUM	MAXIMUM
A = +X Panel	-60	60
B = Center Panel	-60	66
C = -X Panel	-60	60

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-e cover for representative components.

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	HEATER SIZE	POWER REQUIRED (W)		
	PEAK PWR @ MIN V	HOT CASE	COLD CASE	FAILURE
Cryo Pressurant Tanks 2 watts x 8 tanks set points 5°/8°C	16W	0W	11W	N/A
Battery 10 watts x 2°/5°C	20W	0W	0W	12W (2)
Hydrazine Tanks 10 watts x 6 tanks set points 11°/14°C	60W	0W	12W	42W (1)
Hydrazine Lines*	20W	5W	10W	15W (1)
Shades 5 watts x 3 units	15W	0W	0W	15W (3)
Totals	131W	5W	33W	

*Estimates

Failure Modes

(1) Spacecraft spun about +Y axis, sun angle = 62°
(2) Battery charging (endothermic) for more than two hours
(3) Used when +Y axis is anti-sun

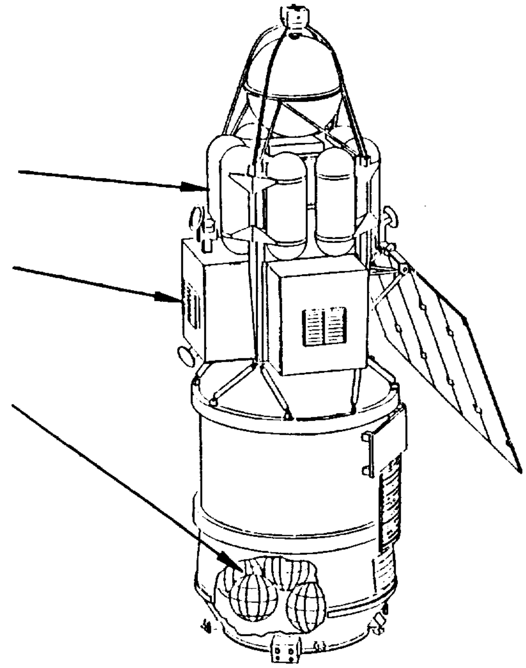

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Figure 6-9. Heater power requirements.

Table 6-7

THERMAL COMPONENT LIST

COMPONENT	WEIGHT (kg)	VENDOR	HERITAGE	FAILURE RATE x 10^{-6}
MLI	28.4	Sheldahl	ERBS, IRAS	N/A
Louvers (3)	4.4	Fairchild	ERBS	0.25
Heaters	1.2	Minco	ERBS	0.74
Controller	In TCP	BSSD	ERBS	0.21
Shades (3)	9.6	BSSD, Maus	P-188	0.25
Paint	3.5	Chemglaze	ERBS	N/A
Total	47.1			

Table 6-8

POWER SUBSYSTEM REQUIREMENTS

ITEM	SOURCE	REQUIREMENT	PERFORMANCE	COMMENT
Average Power Delivered	S/C derived	>277 W	327 W	545 W available on substrate area
Unregulated Bus Voltage	S/C derived	24V to 32V	27V to 31V	
Battery Charge Control	Internal to EPS S/S	Allow full recharge without excessive overcharge	1.03 recharge ratio	w/10°C Battery
Undervoltage/Over-current Protection	Same	Automatic	Comply	
Monitors	Same	Solar array current, bus current, battery current, battery voltage, battery temperature	Comply	
Energy Storage Capacity	S/C and orbit derived	>246.9 W-hr	560 W-hr	50% maximum DOD 1120W-hr nameplate capacity
Power Margin	S/C derived	10% over uncertainties	10%	
Pyrotechnics	Structure Experiment	Solar array deployment, cyro transfer line closure	NSI	

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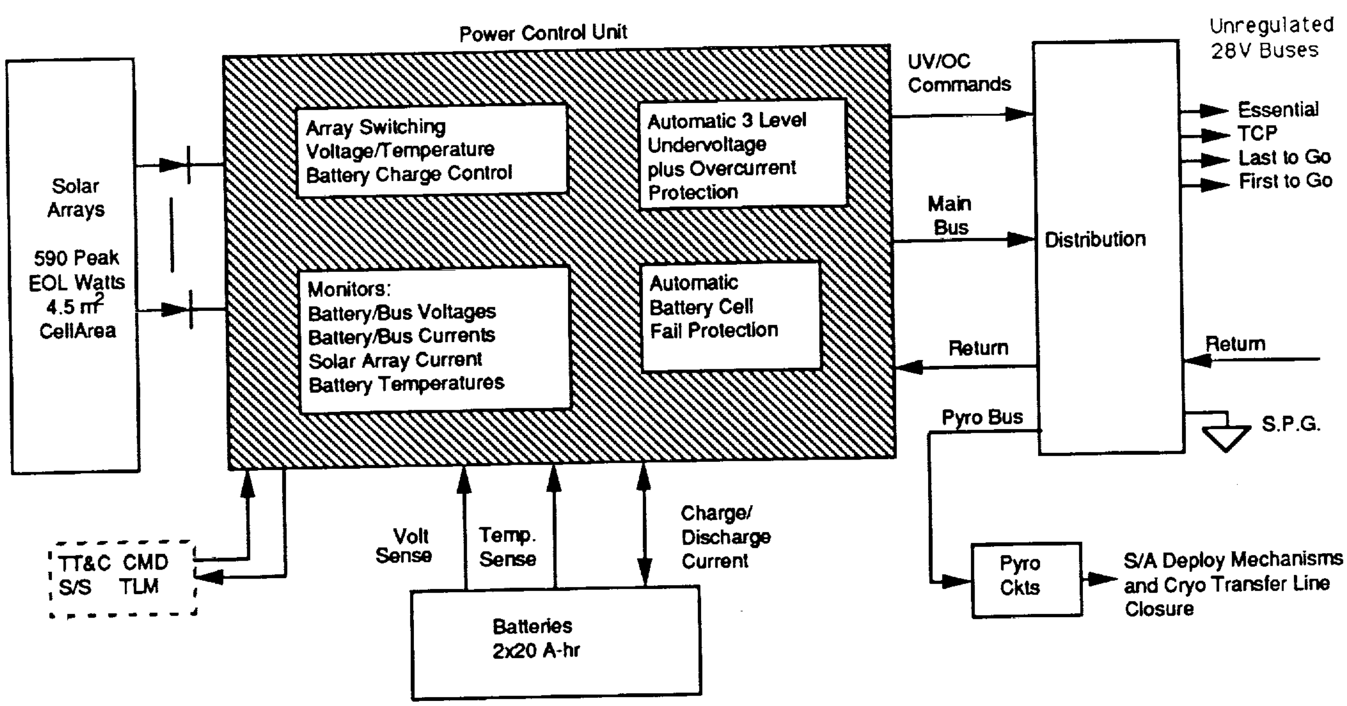

Figure 6-10. EPS block diagram.

spacecraft loads, achieving a direct energy transfer system. This provides a low impedance bus and allows minimum battery and solar panel sizing.

The 4.5 m^2 solar array normal is canted 0.35 radians (20 deg) to the orbit plane. The array produces 327 W averaged over an orbit at Beta = 0.91 radians (52 deg). The array panels are not fully covered with cells. The array can be expanded to support a 545 W load, end of life.

Electrical energy from the solar panel is stored in two 22-cell, 20 amp-hour batteries connected to the 28 Vdc bus through individual relays in the Power Control Unit (PCU). This results in the lowest practical bus impedance with the best transient response characteristics for load changes. Matched cells having plates for two batteries selected from the same plate lot, in conjunction with a mounting surface providing a temperature gradient <3 °C, maximize the ability of the two batteries to share current equally during charge and discharge.

The average depth of discharge is 22 percent. The depth of discharge with one battery would be 44 percent which would still give adequate life for a one year mission.

Battery temperature is always a design concern. All the batteries should operate at the same temperature to assure load sharing, and temperature excursions must be limited to ensure operation between 0 ° and 20 °C. The thermal analysis indicates that temperature can be maintained between these limits. Proper location and application of thermal coatings should keep temperatures within the accepted upper bound, but a small heater may be required to prevent excessively low temperatures.

Overcharge protection of the batteries is accomplished by sensing the temperature of each battery with redundant thermistors and by monitoring the bus (battery) voltage. The PCU derives a temperature-compensated voltage limit control signal which is dependent on the sensed battery temperature and the selected V/T level.

As the battery voltage approaches the maximum allowable value, the control signal causes array circuits to be switched off limiting the bus voltage. This "constant voltage" charge produces a "tapered" charge current which controls pressure and heat buildup in the cells. This technique has proved to be an excellent in-flight overcharge control for long-life missions such as NIMBUS, ERTS, OAO and P78-1. Eight different "V/T Curve" levels may be selected by command to assure a charge level consistent with possible long term temperature changes or battery charge trends.

Cell failure detectors monitor the voltage balance between the "top" and "bottom" groups of cells in each battery and will automatically (if enabled by ground command) disconnect the charge capability from an excessively unbalanced battery. This protective feature ensures avoidance of the potentially catastrophic failure associated with reversal of a low capacity cell during battery discharge or excessive gas generation during overcharge. The battery can be reconnected for diagonostic purposes by command. Only one battery at a time can be disconnected by the automatic disconnect feature to preclude automatic removal of both batteries from charge capability.

Battery arming is provided by an arming plug located on the outer surface of the SC and in series with the negative side of the batteries. Arming will occur several days before launch and the batteries will be kept charged by the GSE through the launch vehicle umbilical.

The batteries are protected from an inadvertent deep discharge or overload by an undervoltage switch which disconnects all nonessential loads insuring against cell reversal. The three undervoltage detection circuit levels are set at 25.1 V, 24.1 V, and 23.5 V with the condition that undervoltage state must be present for at least 50 msec. Ground commands for manual enable and disable of the undervoltage switch are also provided. The undervoltage relay may be bypassed with a relay requiring an enable command plus the bypass command. An overcurrent monitor senses the non-essential bus current and disconnects the non-essential bus, by activating the undervoltage relay, should the current exceed a command selectable level for >50 msec.

Pyrotechnics are used for solar panel deployment and pyrotechnic valves. The safety features of the circuitry meet the requirements of ESMCR 127-1. An arm plug is installed before flight to provide power to the ordnance bus arm/safe relay. The ordnance bus arm/safe relay arms both the hydrazine electrical circuit and the pyrotechnic bus. Both the hydrazine electrical circuit and the pyrotechnic bus have downstream enabling relays that must be closed to power the pyrotechnic firing relays and the hydrazine valves.

The pyrotechnic initiators will be NSI-1 types. A typical initiator firing circuit is shown in **Figure 6-11**. The primary initiator in each panel release mechanisms are fired by one command with the backups being fired by a redundant command.

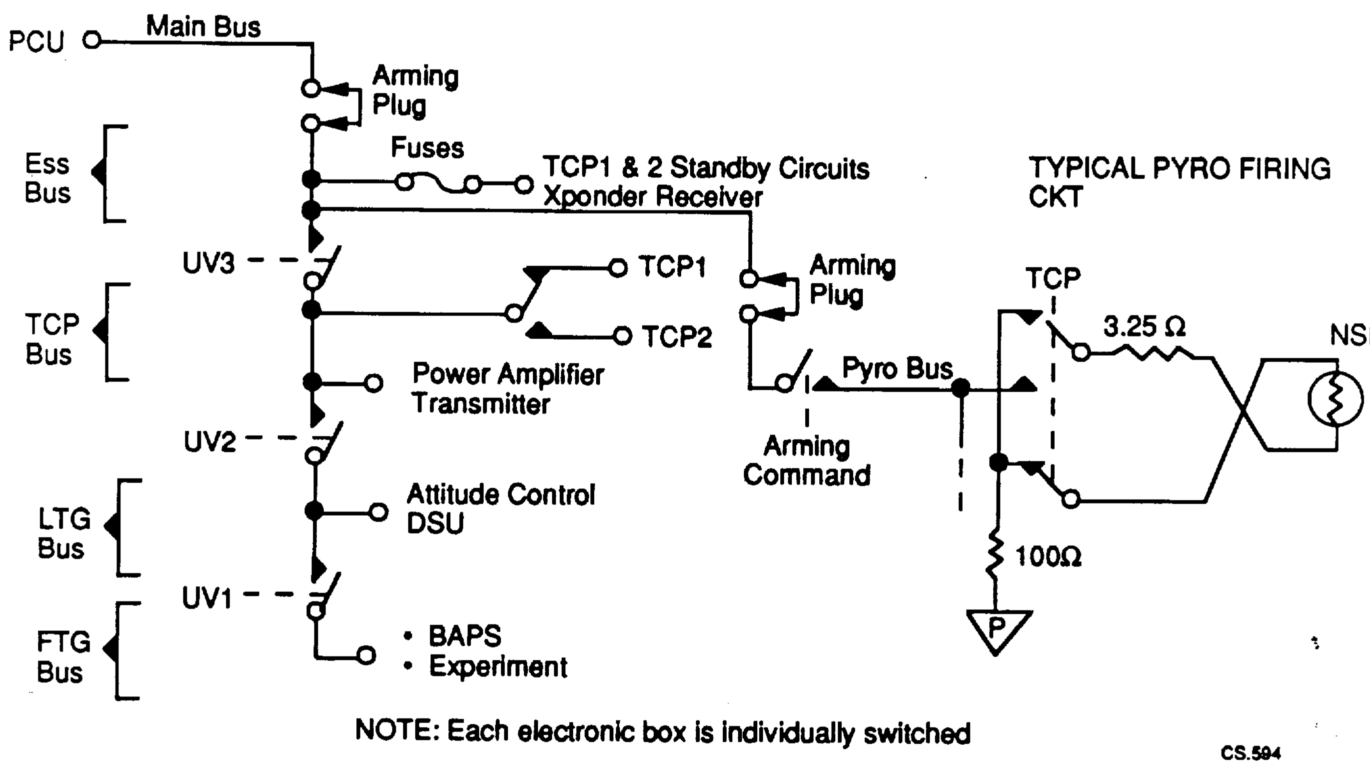

Figure 6-11. Power distribution details.

The following fusing philosophy will be implemented:

- Power to nonredundant essential loads is not fused. Essential loads are classified as all equipment that is necessary for survival of the spacecraft to support the primary mission requirements.

- Active redundant essential loads have both power inputs fused at 4 times expected load current or 1A, which ever is larger.

- Power to nonessential loads is not fused. Fault protection is provided by the overcurrent/undervoltage relay and the load switches.

6.3.3 EPS Analysis

6.3.3.1 Sizing of EPS Components

The power system components were sized considering a variety of launch and orbital conditions. The design case used was:

- Panel temperature: 70 °C
- Solar incidence angle: 32 °
- Solar intensity factor: 0.975
- Degradation: 3 percent
- Load: 305 W
- Eclipse: 35.1 minutes

Figure 6-12 shows the calculations for battery and solar array sizing. The load power is summarized in **Table 6-9**. **Table 6-10** provides detail load analysis for each component. The table lists all the components in the spacecraft giving the best estimate of power required, adding an uncertainty based on heritage, obtaining a duty cycle from the operation time line, and finally taking the product to obtain the load power averaged over an orbit. The power used in sizing the array is 305 W which has a 10 percent contingency for any loads added as the mission becomes better defined.

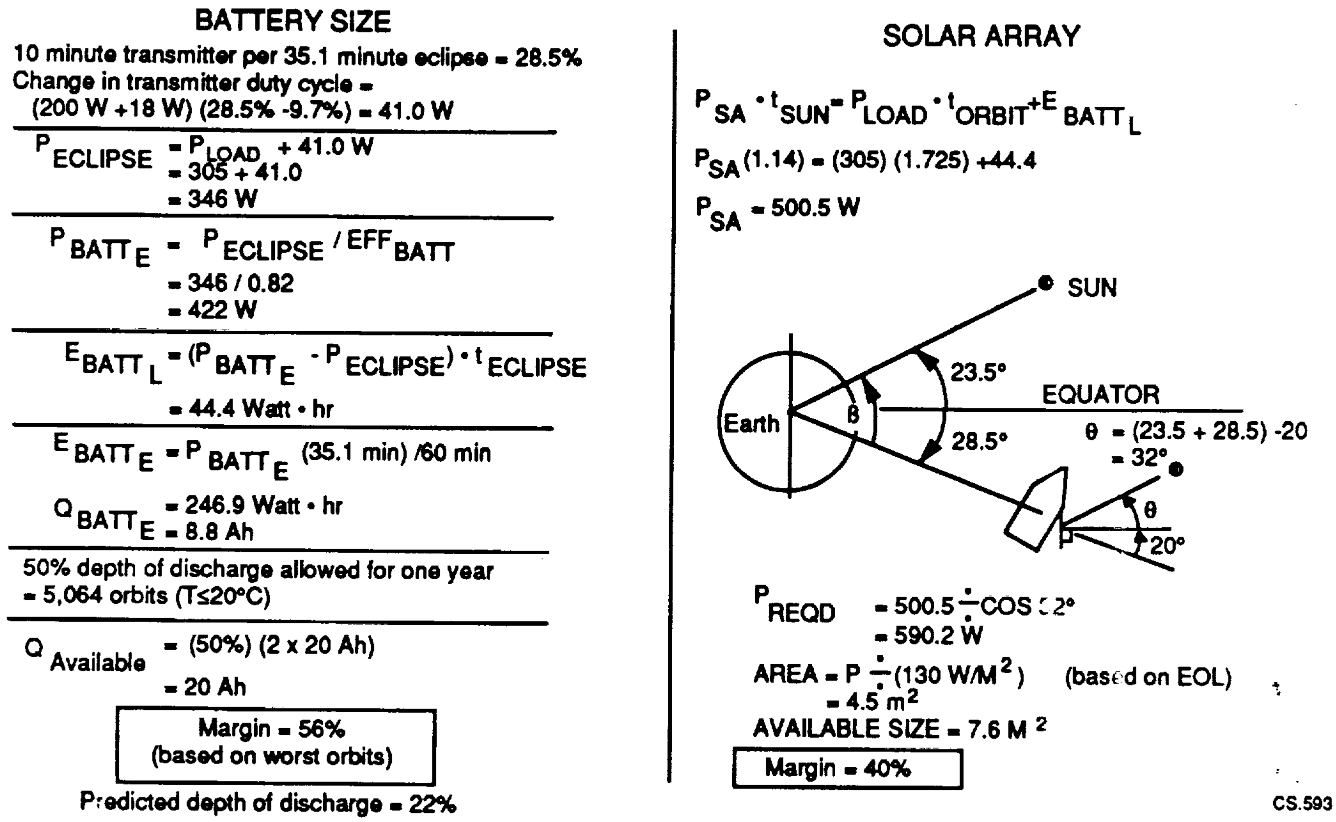

Figure 6-12. EPS component margins.

Table 6-9

ESTIMATED POWER REQUIREMENTS

PHASE → / SUBSYSTEM	PRELAUNCH PEAK POWER (W)	LAUNCH ①		ORBIT	
		PEAK (W)	TOTAL (Ah)	SA SIZING (W)	UNDER-VOLTAGE (W) ②
Experiment	139.0	0	0	116.0	0
PS	20.0	18.4	0	20.0	0
TT&C	250.9	32.9	0.39	53.9	19.4
ACS	43.4	42.3	0.13	42.3	0
Thermal Control SS	0	0	0	33.0	5[illegible]
EPS	33.2	25.9	0.33	12.1	8.[illegible]
SUMS	486.5	119.5	0.85	277.3	84
WITH 10% CONTINGENCY	535.2	131.5	0.94	305.0	93 ③

CS.592

① T-30 seconds to T + 65 minutes

② 70% depth of discharge takes (40 Ah)(70%)/(93.1/24V) = 7.2 hr (no sun)

③ Worst sun angle to charge in U.V.: $\theta = \cos^{-1}[(93.1W/590.2W) \bullet (1.73\ hr/1.1\ hr)]$
$\theta = 75.6°$

Table 6-10
ORBIT POWER BUDGET

COMPONENT	EST. POWER (1) W	UNCER-TAINTY (%)	"ON" POWER (W)	DUTY CYCLE (%)	QTY "ON"	SA SIZING PWR (W)	COMMENTS
EXPERIMENT	175					116.0	Experiment Subtotal
ECP	15.8	15	18.2	100	1	18.2	OTV = 15 W Depot = 25 W
Receiver Heaters	40	5	42.1	0	–	0	
Induced-G Thruster	9	5	9.5	100	1	9.5	
TCS Heater	22.2 + 5.6	5, 15	29.7	100	1	29.7	
Mixer	1.5	15	1.8	100	1	1.8	
Accelerometer	8.0	5	8.4	100	1	8.4	
Liquid Flow Meters	0.44 + 0.12	15	0.6 ea.	0	–	0	
Gas Flow Meters	5.0 ea.	0	5.0 ea.	100	5	25	
28V Press. Xducers	1.0 ea.	5	1.05 ea.	100	12	12.6	
10V Press. Xducers	0.26 ea.	5	0.3 ea.	100	36	10.8	
Cryo Valve	56.0	0	56.0	0	–	0	
PROPULSION SS	80.0					20.0	Propulsion Subtotal
ACS Thrusters (12)	9	5	9.5	<0.1	12	0	
Thruster Valve Htrs	0.25	5	0.3	100	22	5.8	
Induced - g Thrusters							Induced-g accounted in Experiment
Low (1)	9	5	9.5	–	1	0	
Medium 1 (4)	9	5	9.5	–	4	0	
Medium 2 (4)	9	5	9.5	–	4	0	
High (1)	28	5	29.4	–	1	0	
ACS Cat Bed Htrs	1	5	1.1	100	12	12.6	Exp Htr. on for 15 min before induced-g
Press. Xducers (3)	0.5	5	0.5	100	3	1.6	
GN_2 Valves	15	15	17.3	<0.1	2	0	
					Total	136	

CS.595

(1) A + B in estimated power column means A = load, B = associated circuit in ECP

Table 6-10
ORBIT POWER BUDGET (Concluded)

COMPONENT	EST. POWER	UNCER-TAINTY (%)	"ON" POWER (W)	DUTY CYCLE (%)	SA SIZING PWR (W)	COMMENTS
TT&C SS	263.1				53.9	TT&C Subtotal
TCP (active)	8.4	15	9.7	100	9.7	Redundant TCP; 2nd one
TCP (standby)	3.2	15	3.7	100	3.7	always on standby
DSU	6.5	15	7.5	100	7.5	
Xpndr (Xmit)	18	0	18	9.7	1.7	10 minute transmitter per 103.5 minute orbit
(RCV)	12	0	12	100	12	
PA (Xmit)	200	0	200	9.7	19.3	10 minute transmitter per 103.5 minute orbit
BAPS	15	15	7.3	<0.1	0	
ACS	42.3				42.3	ACS Subtotal
ACP	7.1	15	8.2	100	8.2	
IRU	24.0	0	24.0	100	24.0	
Horizon Sensor	10.0	0	10.0	100	10.0	
Sun Sensor	0.1	0	0.1	100	0.1	
Magnetometer	1.1	0	1.1	0	0	Rarely used (Acquisition Mode)
THERMAL CONT. SS	131				33	Thermal Subtotal
Exp. Press. Bottle Htr	16.0	5	16.8	66	11	Temperature controlled (best estimate)
HOP	15.0	5	15.8	0	0	Failure contingency only
PS Tank Htr.	60	5	31.5	38	12	Temperature controlled (best estimate)
PS Line Htr.	20.0	5	21.0	48	10	Temperature controlled (best estimate)
Batt Htr (both)	20.0	5	21.0	0	0	
EPS	11.2				12.1	EPS Subtotal
PCU	7.8	5	8.2	100	8.2	
SC Wiring	3.4	15	3.9	100	3.9	18 awg: (0.006 Ω/ft) (40 ft) (14 ckts) (1 amp) 2
				Total	141.3	Sum = 277.3 W
				Previous Page	136.0	

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DUA0: [CRYO]CLDSATFR.AB4;43.S6-13-AUG-90

Similar tables for undervoltage and launch were developed to assure launch loads did not drive battery sizing and assure there is a net charge in the contingency Y spin mode.

6.3.4 EMC/EMI Design and Implementation

The COLD-SAT EMC/EMI design and implementation requirements plan must include the specifications and guidelines used by BSSD and imposed on subcontractors. The requirements in this plan will be reflected in the spacecraft design, development, production, and implementation to assure the development of an electro-magnetically-compatible Space Segment. The design and test methods of MIL-STD-461A and MIL-STD-462 will be compiled within the EMC program. A single point ground approach will be used with careful use of twisted return wires in high frequency circuits to keep the enclosed area small.

EMC control must also include criteria for bonding, grounding, shielding, circuit decoupling and isolation, and cabling. These criteria are stated below:

- The bond methods and techniques employed in making the electrical connections (using MIL-B-5087B, Class R as a guide) will ensure a minimum DC resistance and RF impedance among the above items. The electrical bonding technique employed will be designed to provide a maximum dc bonding resistance of 2.5 milliohms.
- Cables will be separated into noisy and quiet bundles isolated by metal channels or physical separation where possible. Wrapped shields around chosen cables will also be employed if required. EED cabling will also be separated and use twisted-shielded pair cabling.
- All cabling will be routed in proximity to the spacecraft structure.
- All short rise/fall signals will have sources and loads matched to the cable (75 Ω for coax; 250 Ω for twisted pairs).
- Electro-explosive ordnance circuits will use twisted-shielded pair cabling. Additional wrapped shielding will be used to meet a 40 db analytical margin.

- Shields on ordnance, digital, and RF wires will be terminated to the chassis at each end. Shield termination lengths and unshielded wire lengths will be minimized to maximize shielding effectiveness.
- Magnetic fields will be controlled by (a) use of non-magnetic material, (b) twisting leads carrying AC or DC power, and (c) by measuring the magnetic dipole of components and compensating each assembly to a minimum dipole in each axis.
- In-rush currents will not result in voltage transients exceeding those specified in MIL-STD-1541.

6.3.5 Electrical Power Subsystem Components

6.3.5.1 Component Characteristics

EPS component characteristics are summarized in **Table 6-11.** It should be noted that all components comprising the EPS have been used on past BASG flight programs.

Table 6-11
EPS COMPONENT LIST

COMPONENT	WEIGHT* (kg)	POWER* (watts)**	VENDOR	COMMENT/HERITAGE	FAILURE RATE $x10^{-6}$ hr
Solar Cells	8	NA	Spectrolab or Solarex	ERBS, CRRES, RME	0.13
PCU	24.5	7.8	BSSD	CRRES	3.28
Batteries (2)	31.8	0	BSSD, Ford or McDac	ERBS, CRRES, RME	0.30
Relay Box (3)	12	0	BSSD	ERBS, CRRES	0.94
Pyro Box	4	0	BSSD	ERBS, CRRES	0.24
Wiring	84	3.4		Wt. based on ERBS	
Total	164.3	11.2			

* Best Estimates
**Each When On

6.4 TT&C INTRODUCTION

The Tracking, Telemetry, and Command Subsystem provides the bi-directional communications link to the ground, nominally via TDRSS. This subsystem implements the final formatting and storage of telemetry data, performs the initial interpretation of all commands uplinked to the other subsystems, and also generates its own autonomous responses when necessary for spacecraft survival. This section describes the operation of the TT&C and a list of the software modules employed in the TT&C is also given.

A detailed description of the TT&C Subsystem software is located in Section 6.7.

6.4.1 TT&C Engineering

6.4.1.1 TT&C Requirements

Command and telemetry data handling requirements are summarized in **Table 6-12**. Command requirements for the experiment include the ability to operate from a stored list of at least 300 commands with an execution resolution of 2 seconds. Execution of immediate commands is also required. The command structure must be compatible with DSN requirements, but exploit the use of on-board computers where advantageous to the operation of the mission or reliability of the spacecraft. To provide enhanced reliability through flexibility, provision to uplink parameters and software changes is also required.

Telemetry rate requirement for the experiment is 3808 BPS including about a 3 percent contingency. At least 400 BPS are required for engineering data from the rest of the spacecraft, for a total telemetry rate of 4208 BPS. The emergency mode telemetry rate requirement (if primary RF link fails) is 1000 BPS. An additional experiment requirement is that a minimum of 95 percent of the real-time data be recoverable.

Operations requirements for telemetry include Payload Operations Control Center (POCC) compatibility, formats compatible with NASA's Standard Test Operations Language (STOL), and on board storage of 5 orbits of data.

Table 6-12

TT&C SUBSYSTEM REQUIREMENTS

ITEM	SOURCE	REQUIREMENT	PERFORMANCE	COMMENT
Uplink Format	STDN 101.2	60 bit commands	1000 bps nominal 125 bps backup	125 to 1000 bps
Stored Commands	Exp. Reqt.	300 commands	>300	Exp. and S/C combined
Stored Command Execution Resolution	Exp. Reqt.	≤2 sec	0.125 sec	up to 36 hours delays
Experiment Data Rate	Exp. Reqt.	up to 3808 bps	4352 bps realtime telemetry	1kbps backup
Engineering Data Rate	Derived	up to 480 bps		
Realtime Data Recovery	Expt. Reqt.	95% of mission	no designed in loss	DSU saves 5 orbits
Data Storage	Operations	5 orbits	5 orbits	4.32Kbps x 5 orbits = 134.2M bits
Formats	Operations	Compatible NASA's STOL	4 experiment, 1 Eng.	Eng. has custom section for diagnostics
Contact Time Allowed	TD	10 minutes per orbit	P/B at 50 kbps	Transponder Q chnl. for P/B I chan for R/T
Realtime TDRSS coverage	Analysis	50% nominal 98% backup	Gimballed SME Antenna gives 2.5 dB margin Dual Hemi Ant-Backup	TDRSS MA
Tracking	TDRSS	±5 km	±5 km	TDRSS does ranging
Frequency	TDRSS	up 2106.14 mHz down 2287.5 mHz	Complies	MA

CS.208

TDRSS multi-access (MA) mode will be the primary communications and tracking link. The design shall be compatible with an average of 10 contact minutes per orbit. Two to 8 db margin is required for TDRSS-MA and the link frequencies are shown in the requirements table.

6.4.2 TT&C System Design

The TT&C subsystem block diagram is shown in **Figure 6-13.** The principal component in the TT&C Subsystem is the Telemetry and Command Processor (TCP), which acts as the master controller. This component is redundant in accordance with the reliability/cost trade-off analysis described in Section 12. The subsystem also contains a Data Storage Unit (DSU), a Transponder, and a Power Amplifier (PA). The Transponder and PA are mounted on a plate along with the other RF components - switches, diplexer, coupler, and filter. This RF plate provides the rigid platform to facilitate RF cabling, and also acts as a heat sink for the PA. The primary antenna system is a high gain antenna for bi-directional link to the TDRSS. This antenna is mounted on a Biaxial Antenna Pointing System (BAPS). Two opposite pointing omni-directional (hemispheric) antennas provide an emergency link via TDRSS-SSA, and back-up links via DSN.

The TCP is a Standard Subsystem Processor (SSP). It contains the standard complement of circuits (modules) - SDP, BIF, PWR, and one RCT. Two Special Function Interface (SFI) modules are also used. Refer to **Figure 6-14,** TCP Block Diagram.

The following software modules are standard SSP modules (some modified for use on COLD-SAT):

- Initialize Hardware and Memory (reset recovery)
- Executive Control Loop
- Telemetry Formatter (feeds real-time data to DSU)
- Uplink and Stored Command Interpreter
- Hardware In/Out

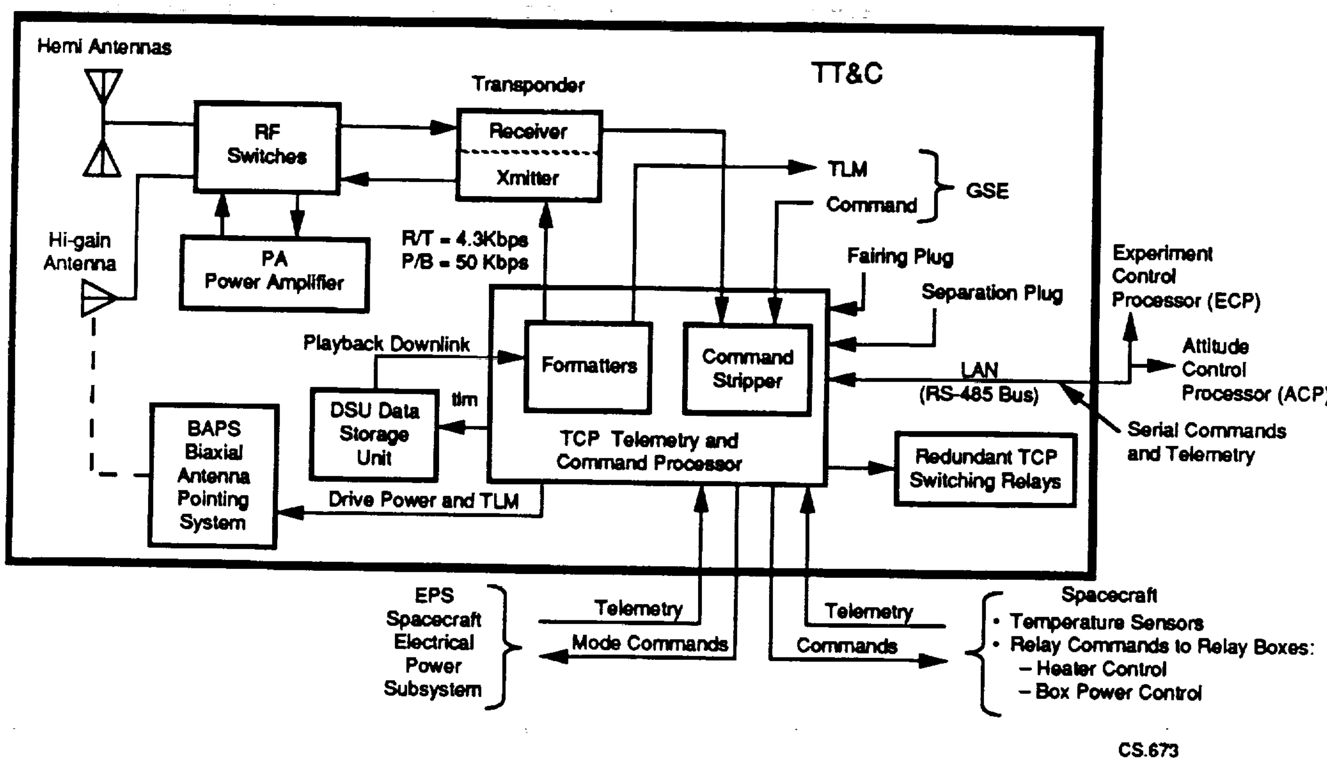

Figure 6-13. TT&C subsystem block diagram.

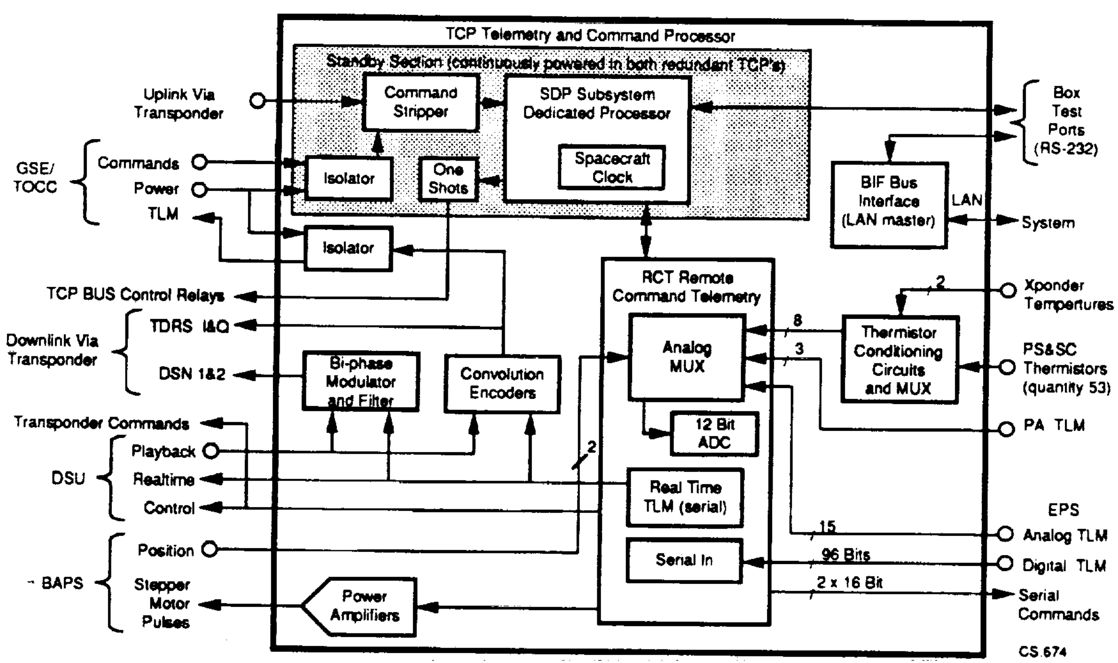

Figure 6-14. TCP block diagram.

The following software modules are generic SSP modules located in the TCP because it is the local area network (LAN) bus master:

- Spacecraft clock
- Uplink Response

The following TCP software processes are custom designed for COLD-SAT:

- Position BAPS Per Ephemeris
- Temperature Control
- Autonomous Anomaly Response
- Critical Status Monitor

The design philosophy for the TT&C Subsystem emphasizes cost effectiveness and reliability. New design is minimized by use of off the shelf components and software that have previous space-flight heritage. A microprocessor for subsystem control permits low cost enhancement of reliability by including a critical status monitor to respond autonomously to spacecraft anomalies. Autonomous response, however, is limited to only those cases where immediate action is required to preserve the spacecraft and experiment in the event of loss of power or attitude control. Those actions which can be taken from the ground without increased risk are not autonomous. Wherever cost effective, software replaces hardware, for example in closed loop temperature control of spacecraft and propulsion system heaters (instead of thermostats). As an added benefit, parameters like temperature limits can be adjusted during the mission to provide added work-around capability. The standard BSSD SSP component has been designed for versatility so that much of the TT&C interfacing is directly off a standard interface card. The use of standard hardware and software also reduce testing costs.

6.4.3 TT&C Functional Operation

This subsection describes how the components listed in the above subsection work together to perform the TT&C Subsystem functions. Software operation is included in the functional description that follows. Additional software information is contained in Section 6.8.

6.4.3.1 TCP Command Response

There are two types of uplink commands - RCT and Subsystem. The RCT commands all act directly on RCT card outputs and function much like traditional command decoders. These commands can set, clear, or pulse any discrete command line in any of the SSPs on the LAN bus. Pulses can be up to 65.5 seconds duration with 1 ms resolution. Any serial digital port can also be updated with up to 32 bits of data via RCT command uplink. RCT commands are used for all normal operations such as power and mode control.

The second type of command is called subsystem command. These commands upload flags, data tables, or program code to any SSP processor memory space in the spacecraft. They can also command any SSP processor to begin or end a process by loading that process jump address.

Both types of commands can be tagged for delayed execution (stored command) in blocks of up to 256 commands for up to 24.27 days with 1/8th second resolution. Neither type of commands require programming skills to use. The operator selects desired actions from menus on a CRT to develop a command list. The command uplink rate is 1000 commands per minute, and data tables or code can be uplinked at a rate of 25,000 words (16 bit) in 10 minutes. **Table 6-13** summarizes the command uplinks.

Table 6-13
UPLINK COMMANDS SUMMARY

	TT&C		EXP.		ACS	
COMMAND CATEGORY	RCT	SS	RCT	SS	RCT	SS
Power load on/off	112		41			
Valve pulses				108	10	
Data or program load		3		4		4
Process or mode control		4		11		9
RCT total = 163	112		41		10	
Subsystem total = 143		(7)		123		13
RCT: controls output with no other activity						
Subsystem: executes program in PROM or RAM memory						

6.4.3.2 TCP Telemetry Handling

The TCP acts as the telemetry collector for the entire spacecraft. This subsection describes how data is collected from other subsystems and folded together with TT&C telemetry, and how the total real-time stream is routed via the DSU to the downlink.

Each SSP in the spacecraft collects telemetry data with a generic software module and following a format list contained in that processor's memory. Five standard sets of lists will be flown and new lists can be created and uplinked after launch if needed. List size for each subsystem is variable, so list selection is made spacecraft wide. The total spacecraft telemetry rate always equals 4208 BPS. When combined with overhead bits, the real-time rate becomes 4352 BPS. Data collection from hardware is performed by software modules separate from formatting so data sampling timing and hardware design are completely decoupled from the telemetry formatting process. This means that all hardware is sampled all the time in an invariant sequence, regardless of what subset is collected for telemetry. This collected data is also accessed in its memory image by other software modules for process or subsystem control and critical status monitoring in each of the spacecraft SSPs.

The experiment and ACS subsystems send their formatted telemetry blocks over the LAN bus to the TCP. The TCP also collects unformatted data from the TT&C components and from the PCU in the Electrical Power Subsystem. The TCP SDP collects all data according to its selected format list, and routes the single 4352 BPS formatted stream to the Data Storage Unit (DSU).

Normally, TDRSS contact occurs for 10 minutes per 103.5 minute orbit. At pre-programmed times the SDP executes the BAPS positioning software to point the high gain antenna toward the anticipated TDRSS location. When the SDP detects a valid uplink stream, it begins interpreting commands. DSU playback (50 KBPS) from the desired starting point may be initiated by these uplinked commands, otherwise the TT&C will playback the last 103.5 minutes of stored telemetry data. Real-time data is simultaneously sent by the TCP to the transponder. All data is convolution encoded for TDRSS or bi-phase encoded for DSN by hardware on a TCP SFI card. The transponder modulates the RF carrier with real-time in phase and playback in quadrature for simultaneous transmission.

6.4.3.3 DSU Operation

The DSU is a solid state memory using CMOS 131 static 128Kx8 RAMs. To reduce the effect of single event upsets, Hamming code hardware is employed as in SSP memories. A 23 bit counter can be preloaded by the TCP to address any 16 bit word location in the memory. Address counter advance and loading are controlled by the TCP. Data is received and transmitted in serial form by shift registers in the DSU.

6.4.3.4 TT&C LAN Bus Operation

The local area network connecting all SSPs in the spacecraft is a standard BSSD design employing the RS-485 hardware interface standard. Protocol and format are fixed by the standard design. All data transfers are synchronous as shown in **Figure 6-15**, Local Area Network Format. Experiment data is transmitted as Hi-rate, and ACS data is transmitted as lo-rate data. The TCP is the bus master, and all LAN communication is handled by auxiliary processors (BIFs) in each SSP. The maximum data rate requirement for COLD-SAT is 60 bits per 15.6 ms LAN frame. LAN throughput capacity is much greater. Commands and parameters from the TCP and data from the slaves is deposited in respective SDP memory space (2 port RAM) by the BIF processors.

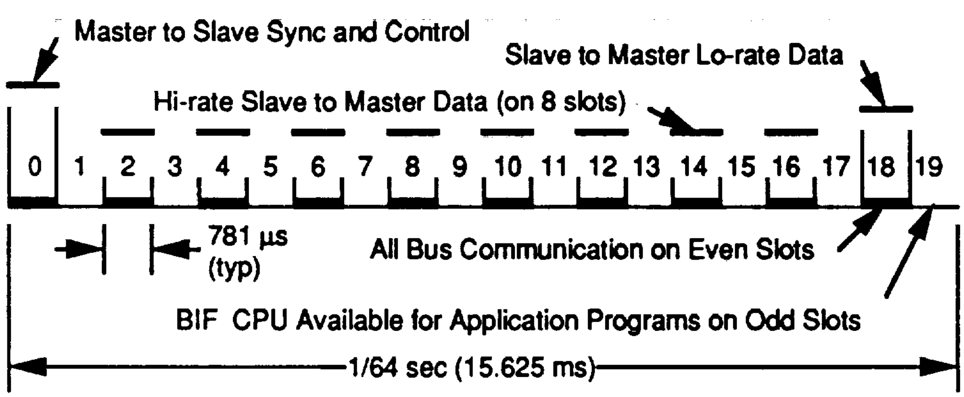

Figure 6-15. Local area network format.

6.4.3.5 TT&C EPS Control

The Electrical Power Subsystem has its own built in intelligent control in the Power Control Unit (PCU). Two serial digital commands from the TCP can command the PCU to change control mode or update parameters used for EPS operation. Mode control uplink commands permit periodic battery reconditioning. Parameter uplink commands adjust the V/T curve used by the PCU for battery charge control and adjust the trip voltages for the first to go (FTG) and last to go (LTG) busses. The TT&C Subsystem passes along uplink commands, but does not autonomously command the PCU.

Bus power control and individual box power switching relays are controlled by the TCP. The TCP requires control of power loads so autonomous and uplinked procedures can deal with power loss, recovery and other anomalies. The TCP also can control bypass relays, for each bus relay, to work around PCU problems.

The TCP controls all pyro-circuits, the experiment pyro-valves and the pyro-bus arming relay. Pyro firing commands for solar array deployment are part of the launch sequence and is described in Section 5.5.

6.4.3.6 TT&C Redundancy Operation

The TCP is the only redundant component in the TT&C Subsystem as determined by the redundancy/cost analysis. Since the TCP is the command decoder, a portion of the idle TCP must remain powered on (standby) to receive commands. The portion constantly powered has been held to a minimum by careful design as shown in **Figure 6-16.**

The standby portion includes the SDP, the transponder receiver interface circuit, a minimum of power relay control circuits, and an autonomous 5 V power supply. Since the spacecraft clock resides in the SDP, both the active and idle clocks run at all times, though the idle one is only used when and if its TCP becomes active.

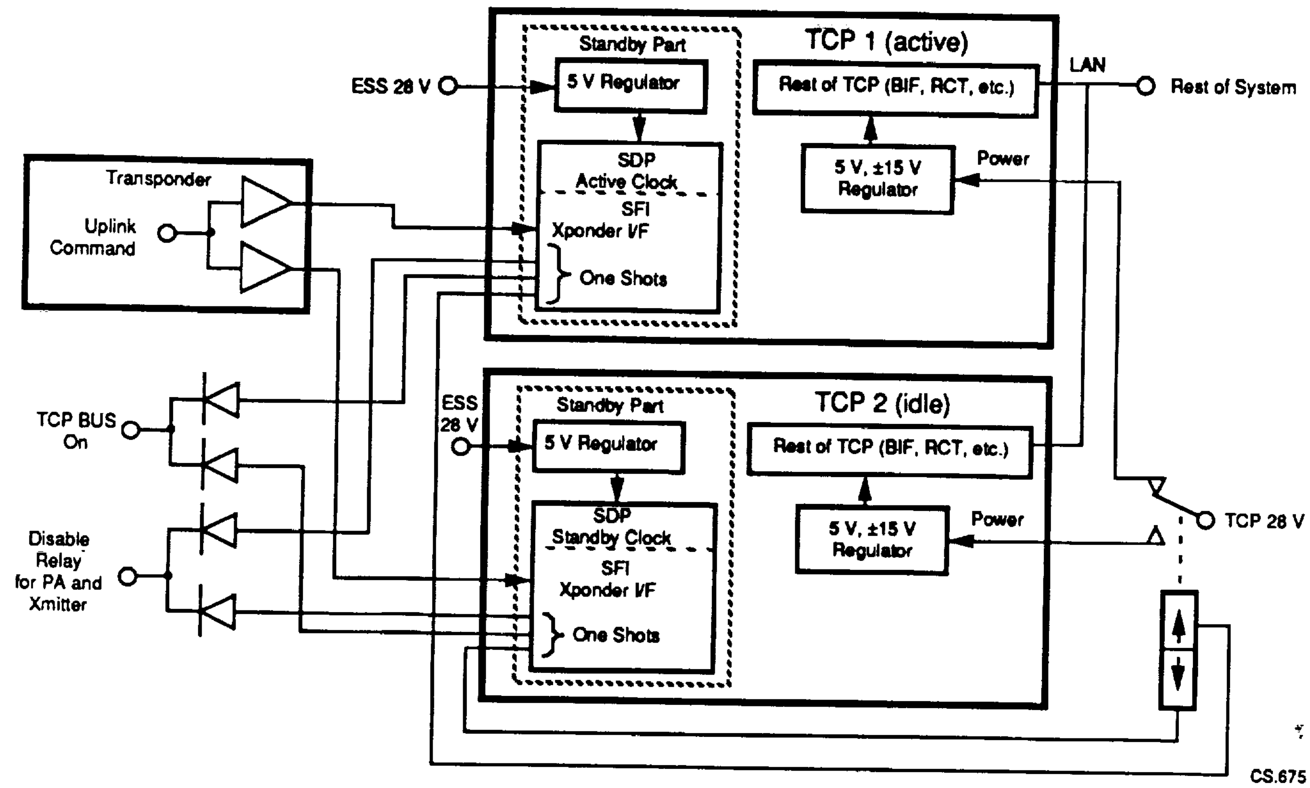

Figure 6-16. TCP redundancy

Uplinked commands are individually addressed to TCP 1 or 2. Note that the transponder has two separate buffered outputs so that failure in one TCP cannot prevent access to the other. An idle TCP can be commanded to take over the active role. Its response is to throw the TCP Select latching relay so that power is directed away from the other TCP and into the commanded TCP. Bus activity is interrupted by this process, but the start up sequence of the newly activated TCP will reestablish synchronous links to the other SSPs. No autonomous switching of TCPs takes place except as a part of power loss recovery described below.

6.4.3.7 TT&C SC Critical Status Autonomous Response

SDP and BIF processors employ watchdog timers and automatically reset and recover when the processor has failed to retrigger its timer.

Each SSP contains a critical status monitor (CSM) software module to perform self health checks and to take any necessary autonomous action when required for preservation of the spacecraft or mission. The TCP CSM reinitializes the LAN bus when reception errors are detected. Additionally, the TCP CSM checks health of the Experiment and ACS by monitoring slave status bytes from the LAN bus. These bytes convey mode status and also toggle one bit continually. These checks are limited to gross performance of each subsystem, since subsystems have their own CSM software.

A bit toggle failure on the status byte from the ECP SDP is interpreted as an ECP failure. The TCP autonomously commands the ECP to standby. If the ECP does not indicate return to standby (all valves in their safe positions), then the TCP switches power to the idle ECP. Power up of any ECP results in pulsing all valves to their safe positions.

The TCP CSM tests ACS telemetry for an excessive position error in the closed loop mode. This error results in reset commands autonomously sent to the ACP from the TCP. Closed loop is again commanded, and a time limit is placed on reaching the dead-band. Failure to meet the time limit results in powering down the attitude control system and initiating y spin-up, described next. Like the experiment, bit toggle failure on the ACS status byte from the ACP SDP also initiates the above autonomous action by the TCP.

Y spin-up is the process of inducing a rotation of 25 mrad/sec about the y axis with attitude control turned off. This maneuver is generated by a timed pulse directly from the TCP via a relay to the y axis thrusters. The y axis is nominally within 0 to +52 degrees of the sun, so the solar array normal will be no more than 72 degrees from the sun as a result of these anomaly responses. The experiment is also commanded to standby, and then powered down to conserve power. The TCP places the power load relays into a status equivalent to the undervoltage condition - all unnecessary loads off. Analysis has shown that battery charge is maintained in this state of low power and y axis spin. The y axis has the maximum moment of inertia so that the spin is stable.

Power loss can occur due to PCU response to an undervoltage or overcurrent condition. When this happens, the spacecraft must take autonomous action to save the mission. The experiment must be placed into the standby mode to prevent loss of consumables, and the spacecraft must either maintain attitude control or spin up about the y axis as described above.

The TCP standby circuits monitor relay status of all three busses - TCP, LTG, and FTG - and preserve a hardware flag bit to indicate active/standby status. The flag bit survives for 1 second during overcurrent transients so the SDP can read it after reset recovery. When a TCP standby circuit detects that the TCP bus has opened, it autonomously toggles the TCP select latching relay to make itself the active TCP. The standby circuit then commands other loads off the TCP bus and enables the TCP bus thus powering up the rest of the TCP circuit. The regular load switching circuits are then used to unload the LTG and FTG busses and then enable those busses. The standby ECP is powered up so that an automatic entry into standby mode is executed in the experiment. In this way, the spacecraft is set up to choose the alternate ECP unless that ECP was previously determined to be unusable, thus reducing chances of causing a repeat of whatever caused the original power problem. After the ECP has had sufficient time to place all valves into their safe condition, the TCP commands the ECP off. This state is maintained until ground controllers troubleshoot the spacecraft and take corrective action.

6.4.3.8 TT&C Thermal and Propulsion Subsystems Heater Control

The TCP reads thermistors from the spacecraft and propulsion system. **Table 6-14** summarizes their location and quantities. Those heaters which are closed loop temperature controlled are either cycled by an autonomous heater software process or by overriding uplinked commands. On and off temperature levels are parameters that can be uplinked. Telemetry can include any thermistor and any heater relay on/off status.

Table 6-14

SPACECRAFT AND PROPULSION SYSTEM THERMISTORS

QTY	SC LOCATION	QTY	PS LOCATION
2	Electronic Compartment 1	6	Hydrazine Tanks
2	Electronic Compartment 2	1	Pressurant Tank
2	Electronic Compartment 3	6	PS Paltform
4	Experiment Pressure Bottles	2	+2 Thruster Lines
1	Helium Tank	22	Thrusters
4	Solar Array		
1	TCP		
2	Transponder		
1	DSU		
19	Total Spacecraft	37	Total PS

6.4.3.9 TT&C Ground and Launch Operations

During ground operations, the TT&C Subsystem is wired through the umbilical to the GSE or TOCC. These connections are optically isolated and provide command and telemetry interfaces at the juncture between the transponder and the TCP. Signals simulate normal TDRSS communication. Link via antennas is also possible on the ground for RF testing.

Launch sequence events are detected in the TCP via fairing and separation plugs. The spacecraft electronics is placed in a power mode analogous to undervoltage with only the TCP standby sections, PCU, and receiver powered. This state is entered 30 seconds before launch using blockhouse lines. At fairing separation, the spacecraft clock begins counting. At T+61 minutes, the IRU is enabled to spin up and stabilize the gyros. At separation, the ACS is powered up and closed loop attitude control is initiated. Two minutes after separation is detected, (T+65 minutes), the solar arrays are deployed by TCP commands. The entire sequence is controlled autonomously by a program stored in the TCP SDP. Further details of the launch sequence are provided in Section 5.5.

6.4.4 TT&C Analysis

This section presents analysis to verify the performance of the TT&C Subsystem.

6.4.4.1 TT&C Command and Telemetry Throughput Rates

Commands are easily handled at the maximum uplink rate of 1 KBPS. Each command is 60 bits as received by the SC so the TCP must read and execute a command every 60 ms (16.7 commands per second). The TCP Main Loop software accesses the Command Response routine every 3.67 ms permitting a much higher response than required. The LAN command bus throughput rate is 32 commands per second.

Telemetry real time rate is 4352 BPS, or 272 words (16 bits) per second. The Main Loop software accesses the Telemetry Formatter software once per 3.67 ms, so one 16 bit word must be collected and sent at each access. All telemetry collected by this routine comes from memory locations in the SDP. TT&C and EPS telemetry is collected by another routine, Hardware In/Out, in the TCP. Other subsystem telemetry is delivered to the SDP two port RAM via the TCP BIF and the LAN. The LAN telemetry bus throughput rate is 266,240 BPS (16.64 K words per second).

6.4.4.2 TT&C RF Link Margins

A summary of link margins for all paths of communication to the spacecraft is shown in **Table 6-15.**

6.4.4.3 Available TDRSS Contact Time

TDRSS contact is required to be nominally 10 minutes per 103.5 min. orbit. **Figure 6-17,** TDRSS Visibility, shows that this is easily achieved.

Table 6-15

LINK MARGIN SUMMARY

LINK	F1	F1R	F2	R1		R2		R3	
Function	Normal Command	Normal Command with Ranging	Emergency Command	Normal Telemetry		Emergency Telemetry		Backup Telemetry	
Path	TDRSS - MA	TDRSS - MA	TDRSS - SSA	TDRSS - MA		TDRSS - SSA		GSTDN	
COLD-SAT Ant.	High-gain	High-gain	Low-gain	High-gain (power ampl.)		Low-gain (power ampl.)		Low-gain (no power ampl.)	
Data Rate	1 Kbps	1 Kbps	0.125 Kbps	Q 50 Kbps	I 4.352 Kbps	Q 1 Kbps	I 1 Kbps	1 128 Kbps	2 12.8 Kbps
Margin (db) No RFI)	4.2	3.8	7.4	3.0	7.7	2.0	-4.0	16.8	20.3
Margin (db) (with RFI)	4.2	3.8	See Chart	2.5	7.2	See Chart	Not Used	-	-

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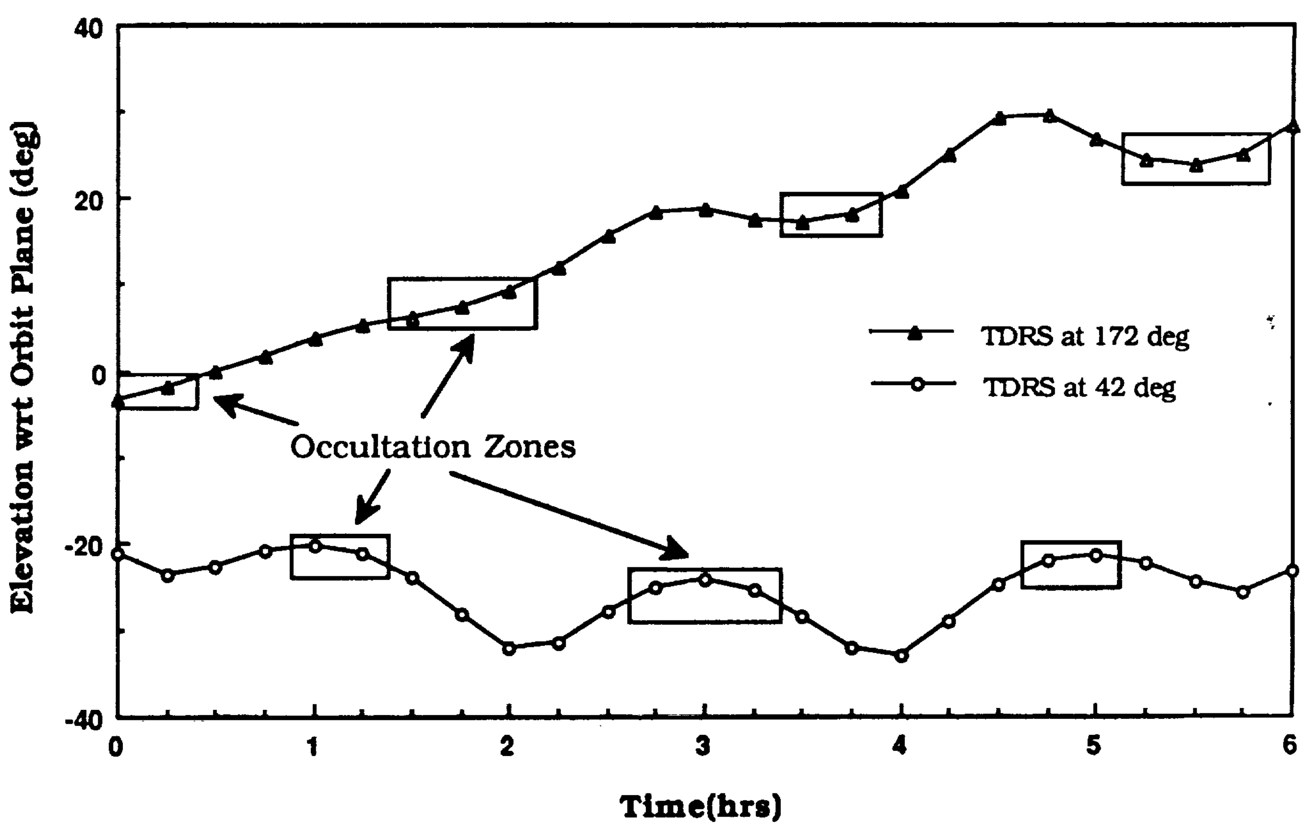

Figure 6-17. TDRS visability.

C-3

6.4.5 TT&C Components

The sources of the components that make up the TT&C Subsystem are described in **Table 6-16.** It should be noted that all components employed in the TT&C subsystem have flight heritage and were used on previous BASG flight programs.

Table 6-16
TT&C COMPONENT LIST

COMPONENT	WEIGHT* (kg)	POWER* (watts)	VENDOR	HERITAGE
SSP (TCP) (2)	31.6	8.4 + 3.2	BSSD	RME
DSU	12.3	6.5	BSSD	
Power Amplifier (RF)	3.4	200	Loral	ERBS
Transponder (1)	6.8	30	Motorola	ERBS
Misc. RF Switches, Filters and Cables	3.3	-	Transco, Sage	ERBS, CRRES, RME
Omni Antenna (2)	4.6	-	BSSD	SME, ERBS, CRRES
Telemetry Antenna with BAPS	9.7	15	BSSD	SME, NIMBUS
TOTAL	71.7	263.1		

*Best Estimate

6.5 ACS INTRODUCTION

The Attitude Control Subsystem (ACS), provides pulse commands to the Propulsion Subsystem (PS) thrusters to maintain spacecraft attitude (bang-bang control) during the low and induced gravity mission modes. It also provides the ability of acquisition/reacquisition from an arbitrary orientation and implements the contingency mode (Y spin) orientation.

A detailed description of the ACS software is located in Section 7.7 along with the other spacecraft software. This section lists the software modules employed in the ACS.

6.5.1 ACS Engineering

6.5.1.1 ACS Requirements

This subsection describes the requirements that dictate the design of the ACS. Design implementations to meet these requirements are presented in the design subsection below. Requirements are derived from the experiment, mission, launch, propulsion, and other spacecraft subsystems and all requirements are met. The basic ACS requirements are summarized in **Table 6-17.**

The payload does not require any stringent pointing accuracy, so 5 degrees is specified. Actual system dead-band will be determined for minimum fuel usage as described below.

Fluids are expelled from the spacecraft as a part of the payload experiments, resulting in torque disturbance. A ninety percent torque reduction is assumed for these dumping torque disturbances through the use of opposing vents. Internal momentum transfer is generated by fill experiments; liquid in the filled tank is assumed to be rotating at 9.1 m/sec (30 ft/sec) peripheral velocity. Momentum transfers to the liquid as transfer begins and, is absorbed by the SC as the liquid slows in the receiving tank.

The ELV separation attitude includes allowance for a 40 minute launch window. Additional disturbance torques due to environment and payload operations are specified in **Table 6-18.** The assumptions used in the derivation of these torques are listed in **Table 6-19.**

Some experiments require small induced accelerations for extended periods of time. These are always normal to the orbit plane. Thruster misalignments during these operations result in torques shown in **Table 6-18.** The orbit is 926 km (500 naut. miles) circular, with a period of 103.5 min.

Table 6-17

ACS SUBSYSTEM REQUIREMENTS

ITEM	SOURCE	REQUIREMENT	PERFORMANCE PREDICTION	COMMENT
1. Pointing Accuracy	EXP	±5°	±1°	
2. Orientation	Mission and EPS	Major axis (Z) normal to orbit with Y axis toward sun	complies	
3. Fuel Consumption	PS	<60 kg	56.6 kg	
4. Torque Authority - Venting Disturbance - Dumping Disturbance	EXP	 $>7 \times 10^{-3}$ N·m >0.57 N·m	2.0 N·m (Z axis)	
5. Internal Momentum Transfer	EXP	>1.3 N·m	2.0 Nm	
6. Separation Rate	ELV	>2 deg/sec	6 deg/sec	gyro limit
7. Separation Attitude	ELV	±5 deg from operational	any	acq. any att.
8. Pulse Width Accuracy	ACS	±10% on 65 to 114 ms pulses	±4% (±2ms)	quantized by S/W
9. Minimum Pulse	PS	50 ms	55 ms	

Table 6-18
ACS DISTURBANCE TORQUES

SOURCE		MAGNITUDE (Nm-peak)	
Aerodynamic Drag (10 cm offset)		9.4×10^{-7}	
Solar Pressure		1.4×10^{-5}	
Magnetic Field 1.5 A·m (each axis)		6.0×10^{-5}	
Gravity Gradient	About Long Axis	5×10^{-5}	(Nominal attitude ±1 deg control)
	About Velocity Axis	3×10^{-4}	
Thruster Misalignment		2.0	(During high level z-axis acceleration 1°)
Venting		7×10^{-3}	
Cryo Dump		0.57	(Assumes all gas)

Table 6-19
DISTURBANCE TORQUE ASSUMPTIONS

The torques disturbing the ACS were calculated using the following assumptions:

Center of pressure - center of mass offset	10 cm
Atmospheric density	1.4×10^{-12} kg/m^3
Coefficient of drag	2.2
Velocity	7387 m/s
Projected frontal area	10 m^2
Reflection factor	1.5
Residual dipole moment (total)	2.6 amp-turns-m^2
Magnetic flux density	30×10^{-6} Wb/m^2
Crosscouple during induced acceleration	1° off cg
Ix	8330 Kg-m^2
Iy	8359 Kg-m^2
Iz	1833 Kg-m^2

These parameters and disturbance torques were used to estimate the fuel budget shown in the propulsion section.

Operations in addition to the normal closed loop mode are occasionally required. To optimize sun exposure on the fixed solar array, the spacecraft is periodically rotated 180 degrees to point to the opposite orbit normal. This maneuver is required about once per month as the sun passes above or below the orbit plane due to orbit plane regression. An acquisition mode is also required as a contingency to re-orient the spacecraft should its attitude vary more than 20 degrees from nominal.

6.5.1.2 ACS System Design

The components of the ACS consist of the attitude control processor, ACP, and a variety of attitude sensing instruments as shown in **Figure 6-18**. Two redundant inertial reference units, IRU, are employed to sense inertial angular rates. An earth horizon sensor and a sun sensor are employed to correct long term IRU drift. A three axis magnetometer is included only for use in the acquisition mode. All these components are mounted in electrical compartment 1 except for the sun sensor which is mounted on the solar array hinge frame.

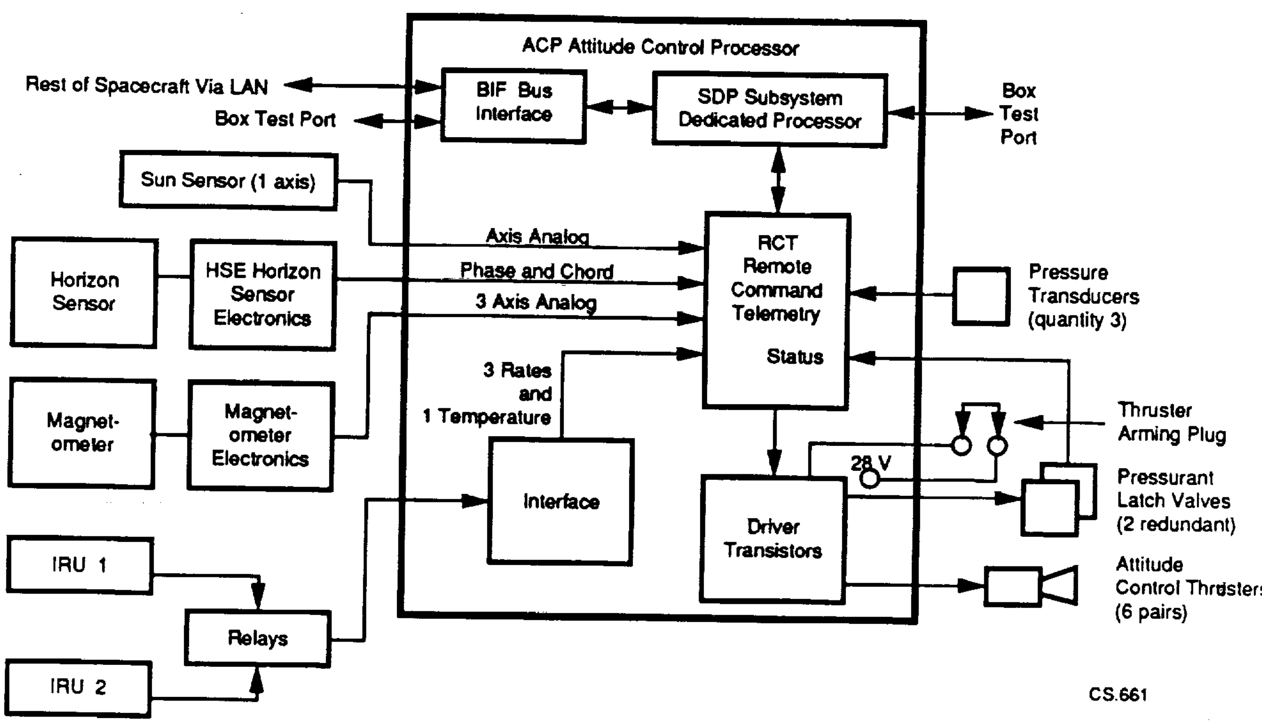

Figure 6-18. ACS hardware block diagram.

The ACP is a Standard Subsystem Processor (SSP). The generic design of this BSSD standard component is described in Section 6.7. The ACP contains the standard complement of circuits (modules) - SDP, BIF, PWR, and one RCT. One Special Function Interface (SFI), module is also used.

The following software modules are standard SSP modules (some modified for use on COLD-SAT):

- Initialize Hardware and Memory (reset recovery)
- Executive Control Loop
- Telemetry Formatter (feeds telemetry to LAN bus per format table)
- Hardware In/Out

The following ACP software processes are custom designed for COLD-SAT:

- Command Response
- Attitude Control
- Acquisition
- Critical Status Monitor

The design philosophy for the ACS emphasizes cost effectiveness and reliability. New design is minimized by use of off the shelf components and software that have previous space-flight heritage. A microprocessor for subsystem control permits low cost enhancement of reliability by including a critical status monitor to respond autonomously to detected attitude control errors. Autonomous response, however, is limited to only those cases where immediate action is required. Those actions which can be taken by ground control without increased risk are not autonomous. The standard BSSD SSP component has been designed for versatility so that much of the ACP interfacing employs a standard interface card. The use of standard hardware and software also reduce testing costs.

6.5.2 ACS Functional Operation

COLD-SAT will maintain a quasi-inertial attitude during orbital operations such that the Z axis of the vehicle is aligned with an orbit normal and the spacecraft Y-Z plane contains the sun-line. This attitude allows thrusting along the spacecraft Z axis to induce the G levels needed for payload experiments while the orbit remains essentially distortion free. Maintaining the sun-line in the spacecraft Y-Z plane allows use of a fixed solar array. The normal to the solar array is in the Y-Z plane and canted 20 degrees to the Y axis.

The functional cperation of the ACS is described in the following sequence:

- IRU Closed Loop
- IRU Update Via Optical Sensors
- 180 Degree Yaw Turn
- Acquisition Mode
- Command Response

6.5.2.1 ACS IRU Closed Loop

Upon initial power up of the ACP, or when commanded to enter the open loop mode, thruster pulses are inhibited. In this open loop condition, the ACP can be used to provide diagnostic telemetry information. Normally the ACP is commanded to a closed loop mode so that attitude control is maintained by generating pulse commands in the ACP to actuate thrusters in the Propulsion Subsystem. **Figure 6-19** shows the main software portion of the closed loop inside double lines. Three axis inertial rate information is received from the IRU. Rate and software derived position information are summed with dead-band feedback to derive thruster pulses. When the rate plus position threshold is exceeded, a pulse duration is computed to null the incoming rate plus provide a quiescent outgoing rate (35 μrad/sec). A third term increases the quiescent rate during acquisition if large position errors exist. The return rate will be 10 times normal for a 20° error.

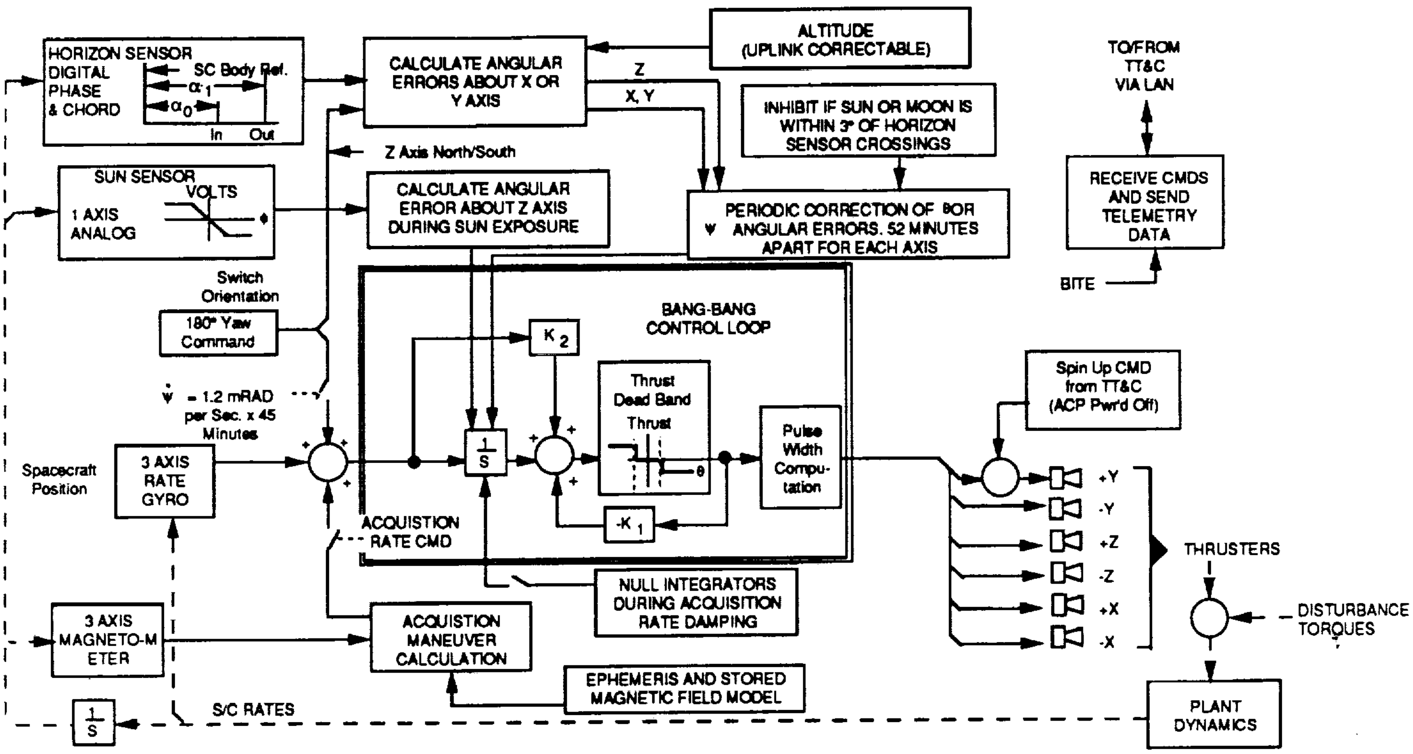

Figure 6-19. ACS functional block diagram.

6.5.2.2 ACS IRU Update Via Optic Sensors

The IRU derived position must be updated periodically to correct for long term drift. Rotation error about the Z axis is corrected by a single axis sun sensor during sun exposure. The maximum eclipse is 35.1 minutes. The sun sensor null plane is parallel to the Y-Z plane.

A Horizon scanning sensor corrects X and Y axis rotation errors. This sensor contains a rotating head that senses Earth interception with its cone surface field of view as shown in **Figure 6-20**. The output of this sensor is two digital words representing phase and chord as shown in the figure. When the spacecraft X axis is along nadir, the chord length is used to determine errors about the Y axis for Y integrator correction. The proper chord length is a function of altitude and is an uplinked parameter. The two points in the orbit when the X axis is aligned to nadir is autonomously determined by the phase output of the horizon sensor. A body fixed pulse is initiated each

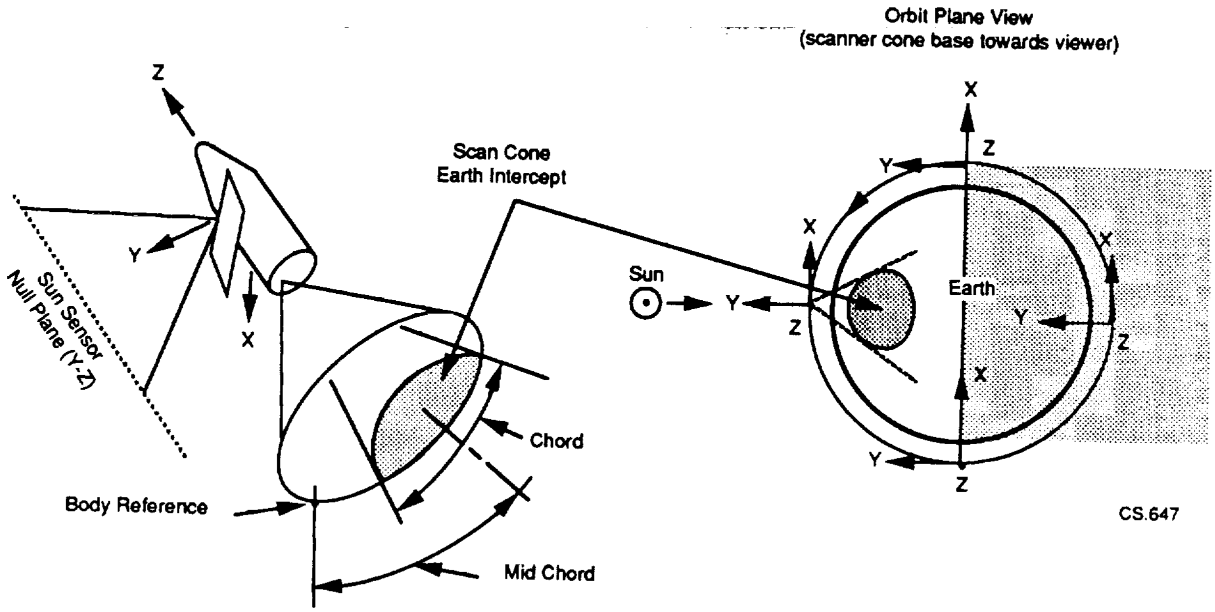

Figure 6-20. ACS optic sensors orientation.

time the horizon scanner goes through the "XZ" vehicle plane. The angle between this pulse and the center of the chord signal is the phase of the horizon scanner output. Hence, the y axis integrated gyro output is updated when the phase angle is 0 deg or 180 deg (i.e., the x axis is along nadir), and x axis integrated gyro output is updated when the phase angle is 90 deg or 270 deg (i.e., the y axis is along nadir).

When the sun or moon is within a few degrees of the anticipated earth intercept points of the horizon sensor, the position updates will be skipped to avoid errors. The occurrence of these events are calculated on the ground and skip commands are uplinked and stored in the ACP. It may also be necessary to skip one of the four updates per orbit during induced-G thrusting by the payload, due to optical interference from the thruster exhaust.

6.5.2.3 ACS 180 Degree Yaw Turn

When the sun-line passes through the orbital plane, the spacecraft must be rotated 180 degrees about the Y axis to maintain proper illumination of the solar arrays. This maneuver will be required about every 30 days. The mission time of the first maneuver will be 30 days after launch since the launch window puts the sun in the orbit plane. In the ACS, the maneuver is accomplished by adding an artificial rate input into the Y integrator. This input is commanded via uplinked stored commands. A rate of 1.2 mrad/sec is used to reduce fuel consumption so that the maneuver takes 1/2 orbit. The start and stop thruster pulses will each be 1.7 seconds. One position integrator update has to be skipped during this maneuver.

6.5.2.4 ACS Acquisition Mode

As a contingency operation, the ACS can perform an acquisition procedure using the three axis magnetometer to regain proper orientation. This procedure is only initiated from uplink command, never autonomously. Recovery can be made from any starting orientation. Figure 6-21, outlines the sequence used for wide angle acquisition.

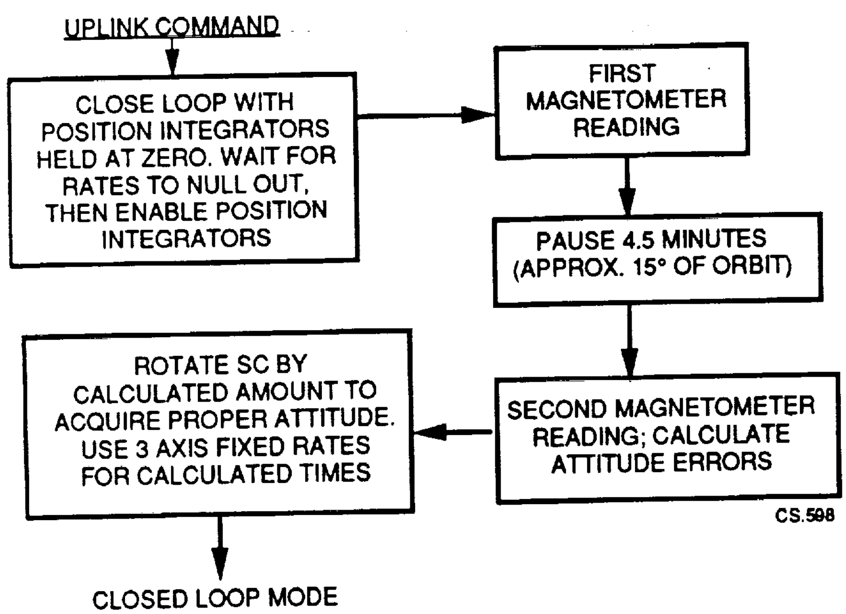

Figure 6-21. ACS acquisition mode.

When an uplink command for acquisition is forwarded to the ACP by the TCP, the attitude control loop is initially closed, but with the position integrators held at zero so that all three rates are damped to zero. The position integrators are then enabled so that the (arbitrary) attitude is maintained. The first of two magnetometer readings is taken. The TCP powered up the magnetometer at the time that it forwarded the acquisition command to the ACP. The ACP next pauses for a fixed time interval of about 4.5 min. to permit the spacecraft to pass through 15 degrees of its orbit. A second magnetometer reading is then taken.

The ACP then autonomously calculates attitude error based on an on-board ephemeris and magnetic field model. The ACP calculates the required spacecraft rotations needed to reach the desired attitude. Timed rate bias commands are fed to their respective integrators to accomplish the required rotations. When this maneuver is complete, with errors less than 20°, the ACP enters normal closed loop control.

6.5.2.5 ACS Command Response

ACS uplink command types are described in Section 6.4. All ACP outputs can be individually stimulated from the ground via uplinked (time tagged if desired) RCT commands. Additionally, the ACP can be commanded for mode selection, telemetry format selection (one of three), or an arbitrary jump from either TT&C processor for forced reset or execution of uplinked code. Uplink data includes:

- custom telemetry format
- stand by control law
- ephemeris
- altitude (earth chord)
- dead-band and other control parameters.

6.5.3 ACS Control System Simulation

A three degree of freedom simulation of the COLD-SAT bang-bang control system was performed to predict and minimize fuel usage and determine frequency and size of thruster pulses. Assumptions used in the simulation are shown in **Table 6-20**. Gyro drift and the optical sensors used for low rate updates

Table 6-20

SIMULATION ASSUMPTIONS

1.	The spacecraft is a perfectly rigid body.
2.	Liquid sloshing effects are zero.
3.	Sources of disturbance torques acting on the spacecraft are limited to Earth gravity gradient and magnetic field.
4.	The Earth magnetic field may be modeled by a tilted, but not offset, dipole.
5.	The spacecraft magnetic character may be modeled by a simple magnetic dipole.
6.	The Earth is a perfect sphere.
7.	The main thruster is off.
8.	Control jets on/off time constants are negligible. (This assumption is reasonable, because the quiescent rate inside the dead-band was chosen, such that the minimum jet on-times are greater than 50 millisec.)
9.	The inertia tensor cross coupling terms are zero.
10.	The attitude control is a continuous (as opposed to a sampled data) system.
11.	Gyro noise is zero.

were not included in the simulation. The nominal parameter values used are listed in **Table 6-21**. Fuel consumption was shown to be relatively insensitive to dead-band size, and minimal in the range of 1.5 degrees to 4 degrees (7 to 10 kg fuel per year). Pulses ranged from about 60 to 100 ms in duration, and occur about every 20 to 30 minutes per axis. **Figure 6-22** is a plot of fuel consumption vs. dead band where a second order polynominal was used as a fit to the actual data points obtained from the simulation. **Figure 6-23** shows the time respones for rate and angular position about the three vehicle axes.

Table 6-21

NOMINAL VALUES OF SYSTEM PARAMETERS

Orbit Inclination	0.49 radians
Orbit Altitude	924 km
Orbit Time	103.5 minutes
Moments of Inertia	X = 8330 Y = 8359 Z = 1833 kgm^2
Spacecraft Mass	3094 kg
Jets Force	1.023 N
Jets Lever Arm	l_{BF} = 2.20 m, l_{TF} = 4.18 m, d_F = 1.00 m
Quiescent Rate	35 μrad/sec
Switching Line Slope	-0.02 rad/sec per rad
Initial Angular Rates	25, -50, and 75 μrad/sec for pitch, yaw, and roll, respectively
Magnetic Dipole	1.5 Am^2 each axis
Specific Impulse	1079 Nsec per kg

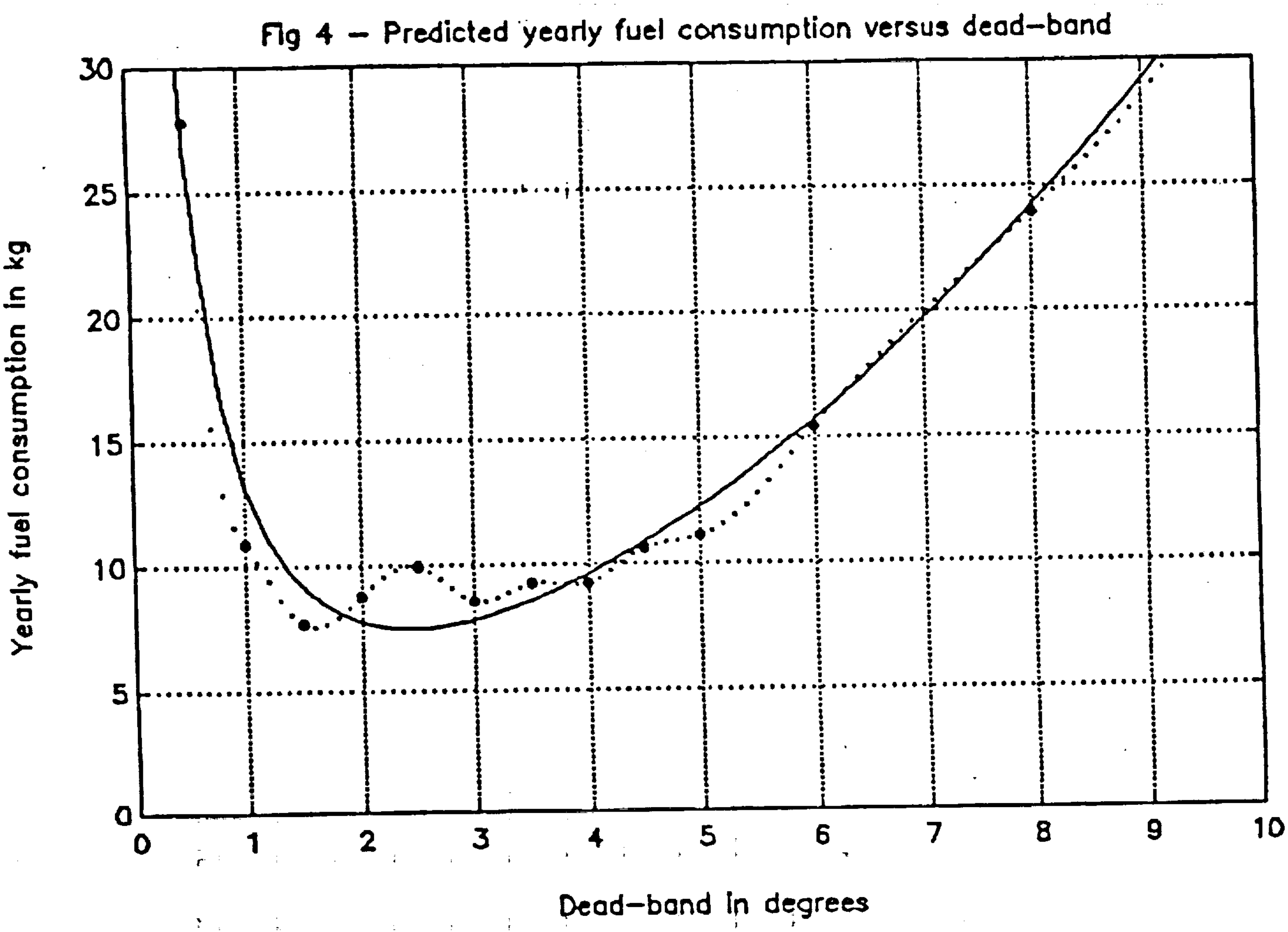

Figure 6-22. ACS fuel use vs. dead-band.

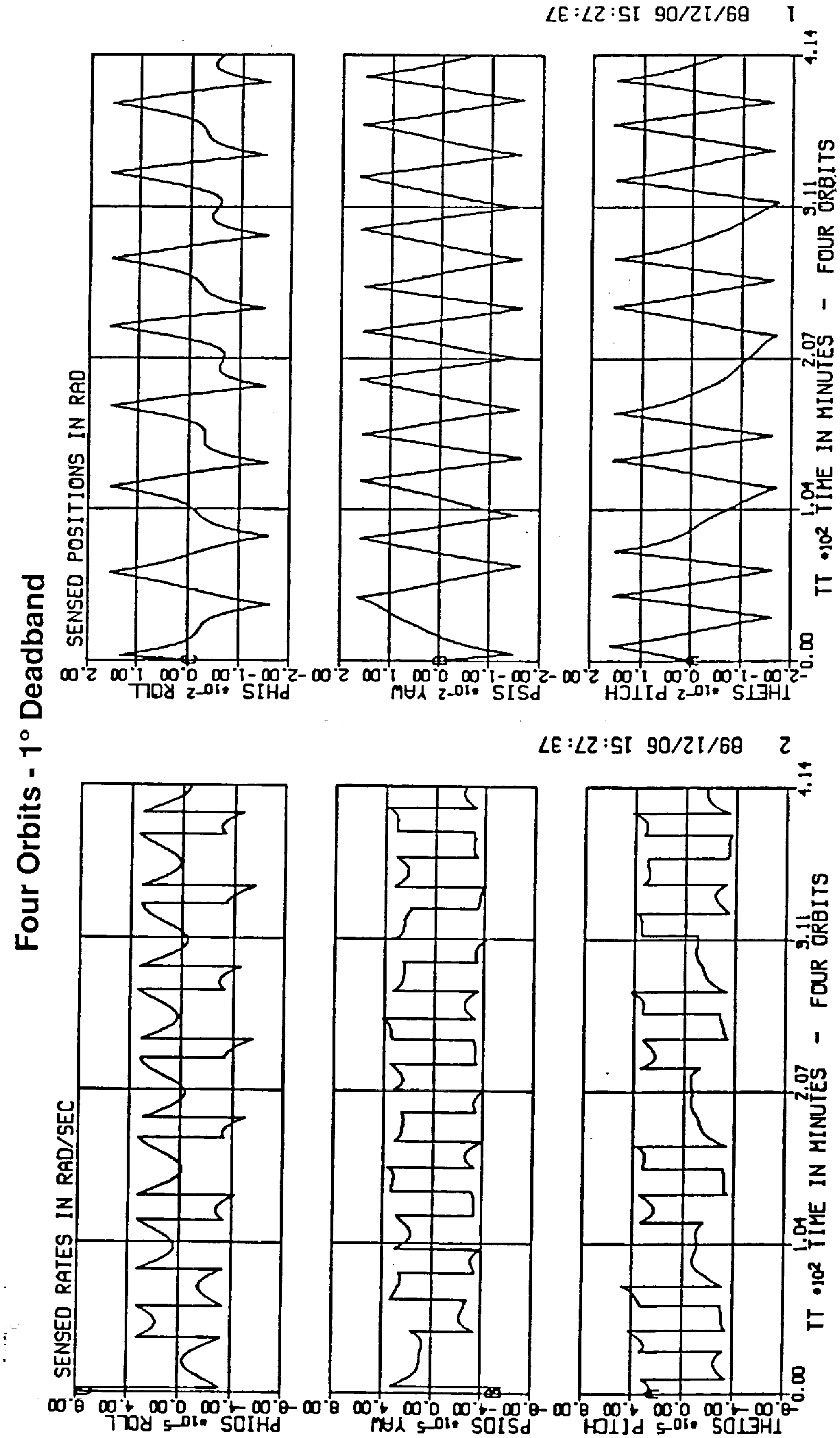

Figure 6-23. ACS time response.

6.5.4 ACS Components

The components that make up the ACS are summarized in **Table 6-22.** As seen from the table, all components have been used on past flight programs.

Table 6-22

ACS SUBSYSTEM COMPONENTS

COMPONENT	WEIGHT* (kg)	POWER* (watts)	VENDOR	HERITAGE	FAILURE RATE x 10^{-6}
Subsystem Processor	8.8	7.1	BSSD	RME	5.2
IRU	2 x 2.6	24.0	Northrop	ERBS	46.7
Horizon Sensor	1.4	3.0	Barnes	RME	7.0
HSE Elect.	2.5	7.0	Barnes	RME	10.0
Sun Sensor	0.1	0.1	Adcole	GOES	0.05
Magnetometer	0.3	0.004	Schoenstad	ERBS	1.7
Mag. Elect.	0.7	1.1			--
Total	19.0	42.3			

*Best Estimate

6.6 PROPULSION SUBSYSTEM

6.6.1 Propulsion System Requirements

The COLD-SAT propulsion subsystem provides four force levels for sustained induced-g axial accelerations, plus three-axis limit cycle control torques during both the induced-g and quiescent mission phases. The principal requirements and corresponding subsystem performance relative to these functions are summarized in **Table 6-23.** Making allowances for attitude control functions, the total available impulse for induced-g in units of g-seconds is approximately 34.5 versus 25.8 required. This is based on a starting best

Table 6-23

PROPULSION SUBSYSTEM REQUIREMENTS

ITEM	SOURCE	REQUIREMENT	PERFORMANCE	COMMENT
Z-Axis Acceleration (induced-g direction)				
Total g-Second Impulse	Experiments	25.8 g-seconds	34.5 g - seconds	33.8% Margin
High Level (-Z)	Experiments	$1 \times 10^{-3} g \pm 25\%$	$1 \times 10^{-3} g \pm 11.6\%$*	One Thruster
Medium Level-2 (±Z)	Experiments	$1.4 \times 10^{-4} g \pm 25\%$	$1.4 \times 10^{-4} g \pm 11.6\%$*	Sum of 2 Thrusters
Medium Level-1 (±Z)	Experiments	$7.0 \times 10^{-5} g \pm 25\%$	$7.0 \times 10^{-5} g \pm 11.6\%$*	Sum of 2 Thrusters
Low Level (+Z)	Experiments	$2 \times 10^{-5} g \pm 25\%$	$2 \times 10^{-5} g \pm 11.6\%$*	One Thruster
Transverse Axis Cross Coupling	Derived	<±5% of Axial	±1.7%	CG Pointing Error
Short-Term Instability (Roughness)	Experiments	<±8% of Nominal	±6%	
Three-Axis Attitude Control	Experiments	Inertial for ≥1 Years	>1 Year	
Tangential Acceleration (Limit Cycle)	Experiments	$<1 \times 10^{-6} g$ 99% of Time	$<4 \times 10^{-7} g$ ≥99.975% of Time	Max gg Background
Control Torque	ACS	$<5 \times 10^{-4} g$ 1% of Time	$4.5 \times 10^{-4} g$ Peak ≤0.025% of Time	Deadband Turn-around
X-Y Z		>1.8 Nm >0.6 NM	5.3 Nm 1.8 Nm	5.8 lbf Max Offset Liquid Dump

*Variation over Mission

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estimate (no contingency) wet mass of 2963.4 kg. The total starting mass of hydrazine is 495 kg, for which an induced-g specific impulse of 2108 N-sec/kg (215 seconds) has been assumed.

Nominal induced-g acceleration levels are as shown. Fuel consumption rates from highest to lowest g-levels are roughly 44 kg/hr, 6 kg/hr, 3 kg/hr, and 1 kg/hr. The transverse axis cross-coupling performance is based upon 1 degree overall effective misalignments of each induced-g thruster with respect to the vehicle CM during thrusting. This, together with vehicle pointing errors of up to about 3-1/2 degrees would produce an altitude variation of no more than 20 km over the thrusting portion of the mission.

Using the same size thrusters for attitude control as used for the Medium Level-1 induced-g acceleration, simulations indicate that the three axis attitude control system will produce quiescent mode limit cycle torques averaging about 100 ms duration at intervals of about 7 minutes (all axes taken together). The indicated tangential accelerations at less than 0.025 percent duty cycle reflect the effects of such pulsing. The tangential acceleration during the great majority of time is simply that attributable to the background gravity gradient acceleration. The greatest X and Y axis disturbance torques are based upon the large induced-g thruster being effectively misaligned by one degree overall (misalignments plus CM shift). These considerations yield comfortable thruster to disturbance torque authority ratios of 3:1 using the aforementioned size of thruster for three-axis attitude control.

6.6.2 Propulsion Subsystem Design

A summary of derived thruster size requirements based upon application of the previous strategies is presented in **Figure 6-24.** A system of induced-g thrusters consisting of two modules of five thrusters each, mounted on each end of the vehicle's body-z axis was the chosen. In this arrangement two single thrusters, one on each end marked A and D, supply the axial forces needed to induce the smallest and largest values, respectively, of acceleration. The remaining thrusters, marked B and C, act in pairs to supply the two intermediate axial accelerations. These can induce accelerations in either the plus or minus directions as shown. These thrusters are also canted very slightly so that each thrust vector passes through the nominal mid-mission CM location on the z-axis. The thrusters can thus be used in various other non-standard combinations, with some increased ACS fuel consumption, to induce levels of acceleration other than those required if desired.

With regard to attitude control, total movement of the CM was computed to be 50 cm along the z-axis as fuel and cryogen is consumed, and up to 1.5 cm laterally due to shifting fluids. Induced-g Thruster mounting and pointing errors of up to 1.0 cm and 0.5 degrees, respectively, when combined with the above yielded an effective misalignment of 1.0 deg overall for each thrust level. As mentioned, this analysis yielded a torque authority of greater than 3:1 for the thrusters used for attitude control.

(Note: All but highest and lowest force thrusters operate in pairs)

Thruster Type	A	B [1]	C	D
Acceleration Level	2e-5g	7e-5g	1.4e-4g	1e-3g
Newtons per thruster	0.51	0.90	1.80	25.68
Pound force per thruster	0.12	0.20	0.40	5.77

(1) Also used for 12 ACS thrusters (Torque authority >3:1)

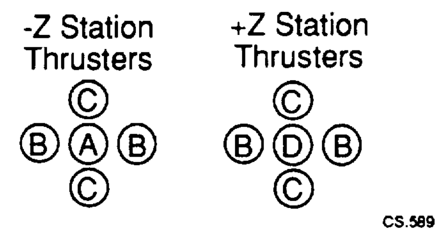

Figure 6-24. Induced-g thruster complement.

As illustrated in **Figure 6-25** the baseline propulsion system consists of one 41.9 cm (16.5-inch) pressurized nitrogen supply tank, six 55.9 cm (22-inch) bladder propellant tanks containing monopropellant hydrazine, and twenty-two thrusters with integral dual series control valves and filters. The system is of a pressure regulated type controlled by mechanical regulators, together with suitable valving for pressurant isolation and overpressure protection.

As mentioned, the system employs four levels of thrusters (nominally sized at 0.51 N (0.12-lbf), 0.90 N (0.20-lbf), 1.80 N (0.40-lbf), and 25.88 N (5.77-lbf) for optimum mid-mission performance) whose general arrangement is sketched as shown. The nine-thruster cluster on top of the vehicle (+Z end) is thermally isolated from the vehicle and contains two thrusters for each intermediate induced-g acceleration level and a single large thruster for the high acceleration level. The remaining four +z thrusters operate in pairs with oppositely directed thrusters on the bottom of the vehicle to provide pure-couple limit cycle torques about the vehicle X and Y axes.

The thirteen-thruster arrangement at the bottom of the vehicle includes five central induced-g thrusters for the two intermediate levels of acceleration including a single small thruster for the lowest level of acceleration. The

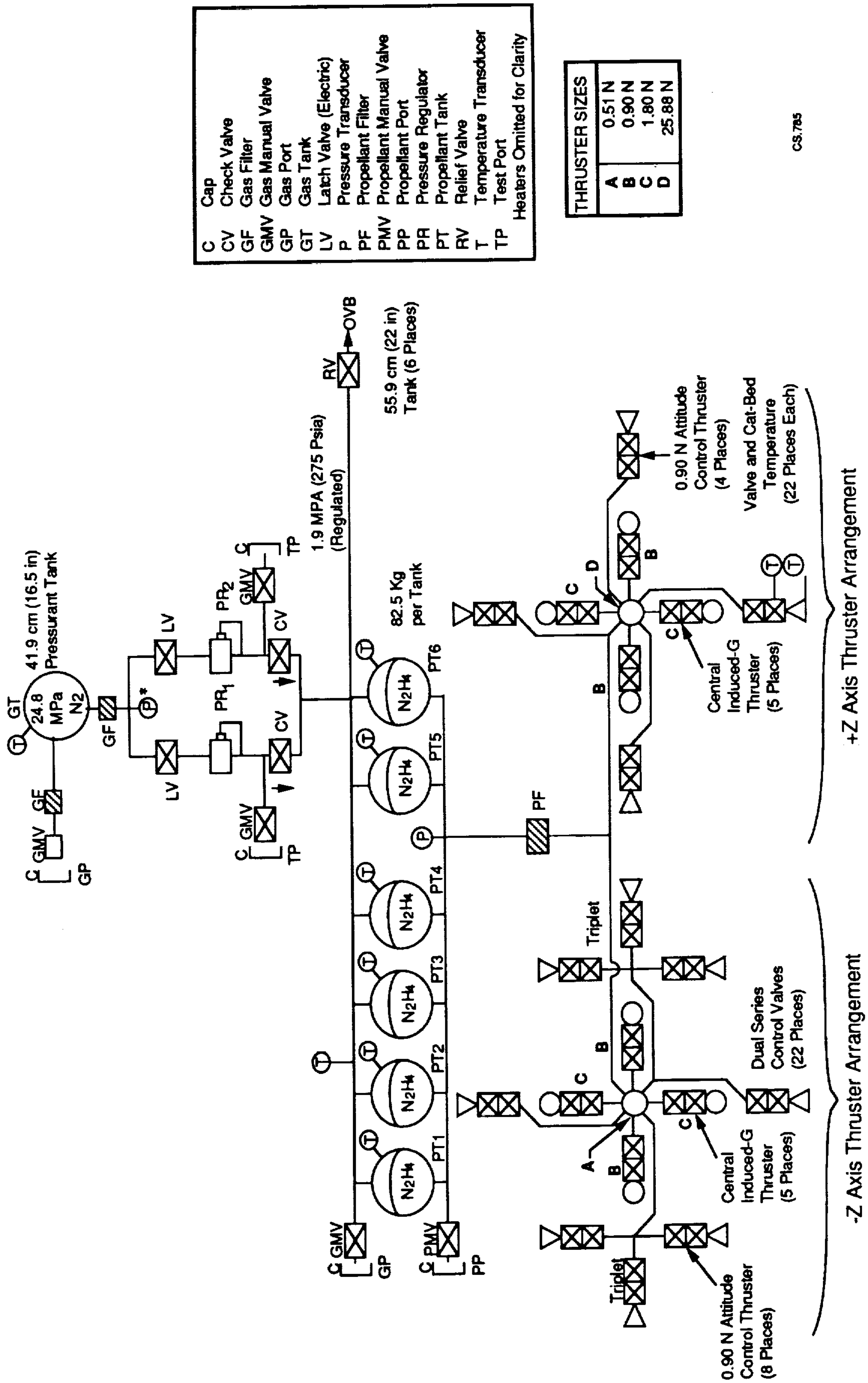

Figure 6-25. Propulsion subsystem.

remaining eight ACS thrusters are mounted outboard to avoid any plume interactions with the vehicle. These include four thrusters for torques about the X and Y axes as previously described and four others used for pure couple torques about the Z axis.

The system employs no propellant isolation valves and uses single string plumbing to reduce costs, but does use dual series valving at each thruster to preclude leakage. The thruster/control valve assemblies are purchased and tested as off-the-shelf components.

The locations of the six propellant tanks and single pressurant tank plus arrangement of the twenty-two thrusters are shown in **Figure 6-26.** The thruster to thruster lever arms for the attitude control torques are nominally six meters for the X and Y-axis and two meters for the Z-axis.

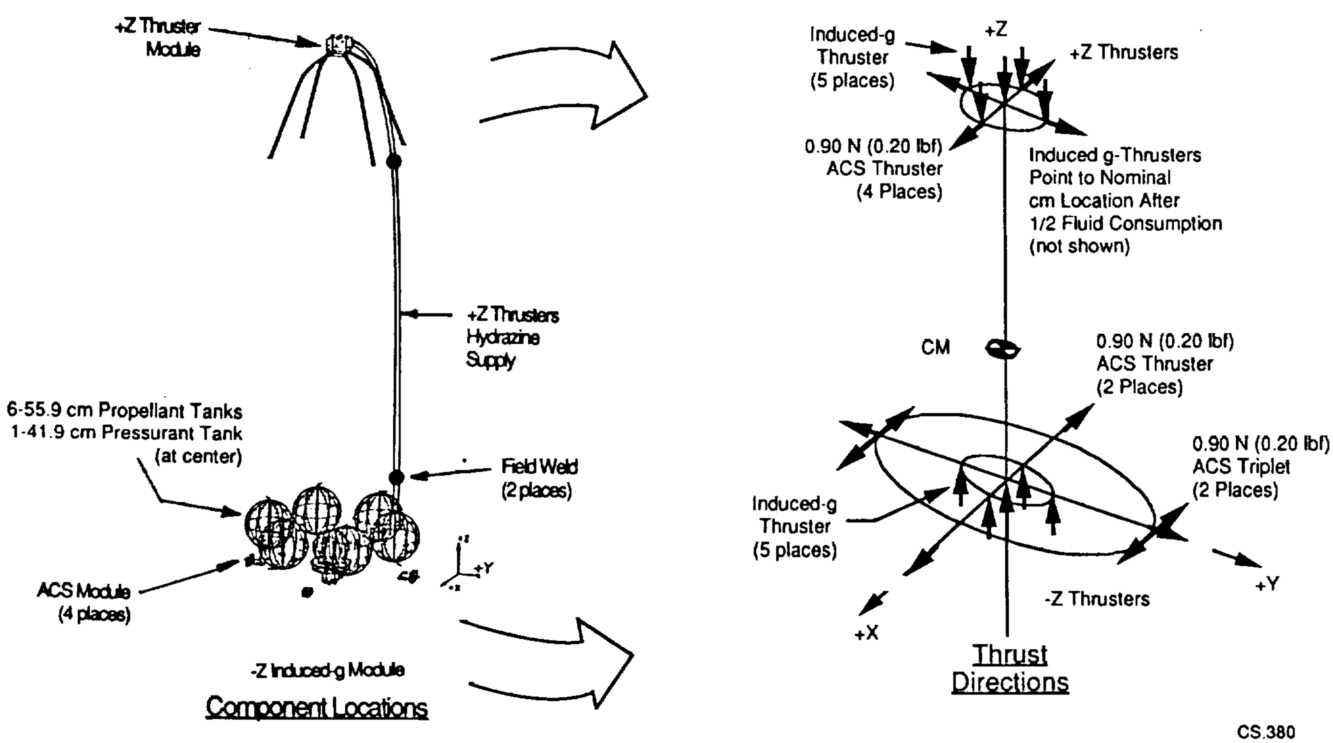

Figure 6-26. Propulsion subsystem arrangement.

Buildup and testing of the top and bottom components can be handled as two separate modules at the propulsion system vendor's facility. BSSD would supply the vendor with the top and bottom flight modules for such buildup. Two field welds, to connect a single tube between the top and bottom assemblies, would be accomplished at BSSD during the satellite integration phase.

6.6.3 Propulsion Subsystem Analysis

The next series of figures relates to computation of the COLD-SAT fuel budget. **Figure 6-27** shows the induced-g-only fuel component needed to supply the required nominal value of 25.8 g-seconds based on the best estimate initial wet mass. Note that by employing the impulse requirement in units of g-seconds, the program of thrusting relative to the size and duration of specific thruster usage is irrelevant in the computation of required fuel. The average inertial mass, in particular the rate at which cryogen mass is vented and ACS fuel is consumed during the early part of the mission, has an effect on the amount of propellant needed to yield the required g-seconds. We have based fuel requirements on an effective cryogen mass equal to half the initial manifest and half the ACS fuel requirement as shown. The ordinate values of the graph include only the fuel needed to supply the required impulse. The ACS fuel needed during the first three months of operation is shown separately in the caption at 41.8 kg (which includes ACS fuel consumed during induced-g thrusting).

The curves of **Figure 6-28** supplement the 6-tank pressure regulated performance indicated by **Figure 6-27.** Given a g-second requirement, there are many ways to select individual thruster thrust level, all of which would meet that requirement. In fact, one may regard a g-second requirement by itself as being equivalent to a cumulative delta velocity requirement. The overall fuel requirement is identical, the only variable is the thrust level which then governs the amount of time required to achieve the overall delta velocity. If the use of small thrusters (and lower accelerations) predominate, more time is needed and vice versa.

In our case, a scheme in which acceleration level with respect to the nominal requirement for each level, is balanced over the first three months between beginning and final values was judged to be the best approach and the one taken. The significant variables are the rates at which cryogen and ACS fuel

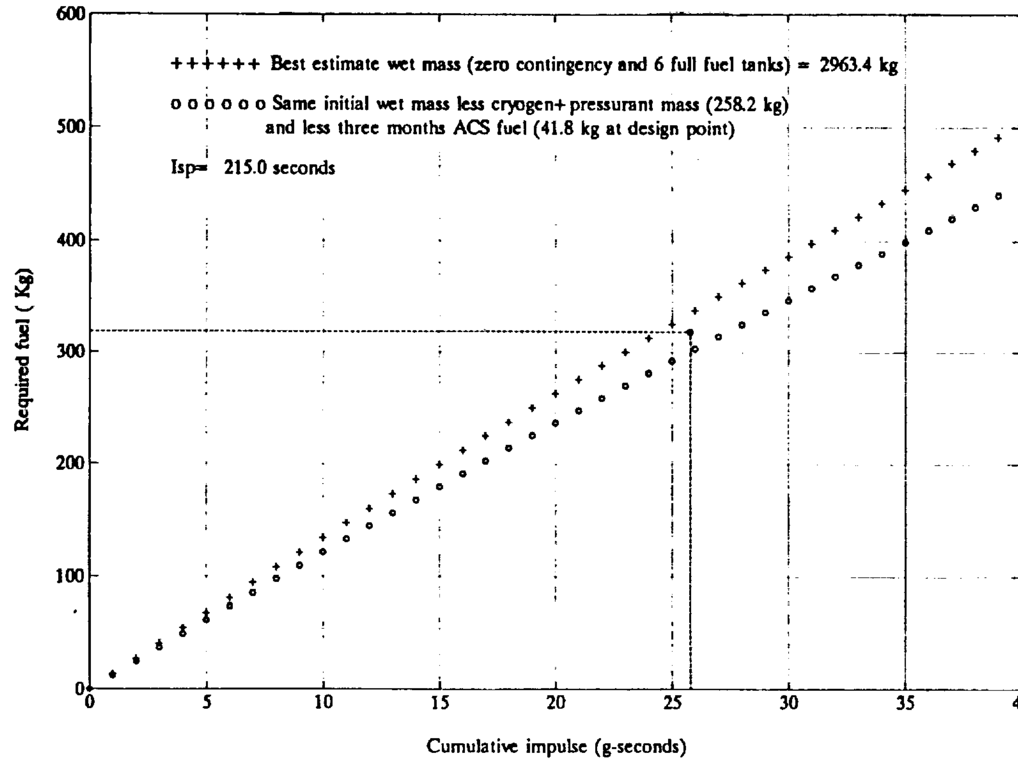

Figure 6-27. 318.2 Kg of non-ACS propellant is needed for 25.8 g-seconds of induced gravity.

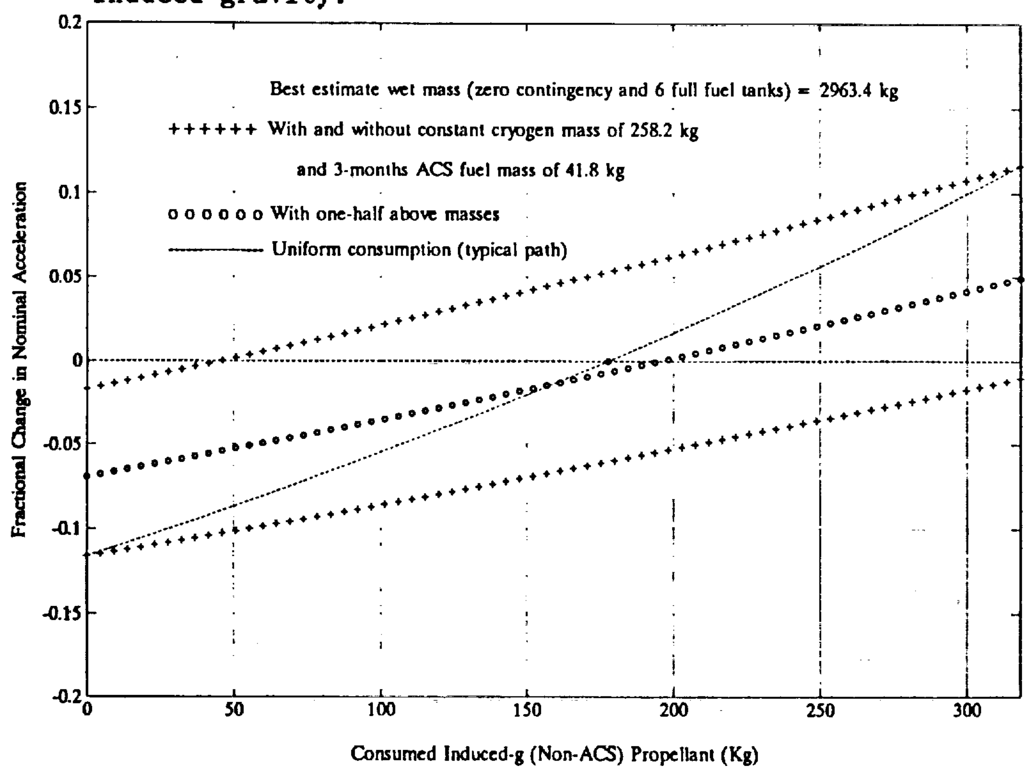

Figure 6-28. Max acceleration variation over 25.8 g-second (3-mos) due to depleting cryo plus fuel = ±11.6 percent.

are consumed as this effects the inertial mass undergoing acceleration. Here we assumed uniform consumptions with time (see dashed path in figure) starting with a full load of cryogen at the start of the mission decreasing to no remaining cryogen mass plus simultaneous depletion of nominal ACS and induced-g fuel allocations. This yielded a very acceptable ±11.6 percent acceleration variation over three months with nominals occurring roughly at the midpoint. At the end of this interval (or at any time for that matter) the pressurant supply can be isolated from the propellant tank ullages which effectively changes the propulsion system configuration into a blowdown system. This changes the acceleration profile into one that is now gently and monotonically decreasing at a rate dependent upon the amount of remaining cryogen and fuel.

The overall fuel budget for the first year of operation is shown in **Table 6-24.** Because of the precise Delta orbit insertion capability, the low atmospheric drag due to 926 km orbital altitude and because perturbations to the orbit during induced-g thrusting is expected to be so small, no fuel need be allocated to correct orbit parameters. The ACS fuel is needed for four categories of disturbances. One degree effective induced-g thruster misalignments accounts for the largest fuel portion as shown. The environmental category needs were derived from simulations. ACS fuel budgeted for H_2 venting and fluid dumping is based on ejection through T-nozzles. In each case it is assumed that the unspoiled residual impulse subsequent to passing through the T equals 10 percent of the theoretical impulse assuming 100 percent gas passing through an equivalent single nozzle. The somewhat generous fuel margin of 33.8 percent is viewed as appropriate, in relation to our zero contingency wet mass basis.

6.6.4 Propulsion System Components

The primary hardware elements of the selected baseline system are outlined in **Table 6-25.** There is considerable flight heritage for each component. Having successfully built and flown the ERBS Spacecraft, Ball is especially familiar with the test and flight performance of the ERBS spacecraft thruster

Table 6-24

FUEL BUDGET

USAGE	KG
Injection Error Fix	0.0
Induced-g Thrusting	318.2
ACS for Induced-g Misalignments	22.2
ACS Environmental	13.0
ACS H2 Venting (10% Unspoiled)	16.3
ACS Fluid Dump (10% Unspoiled)	0.2
Total	369.9
Available (6x82.5 Kg)	495.0
Margin (Kg)	125.1
Margin (Percent)	33.8

Total Impulse = 25.8 g-sec
Induced-g Isp = 215 secs
ACS Isp = 110 secs
Duration = 1 year
Best Estimate Initial Wet Mass = 2963.4 KG

which has been selected to provide both the two intermediate levels of induced-g acceleration as well as all the ACS functions. This one thruster accounts for 20 out of the total complement of 22 COLD-SAT thrusters.

6.7 FLIGHT SOFTWARE INTRODUCTION

The flight software description is collected in this one chapter to aid in understanding the overall plan and to facilitate assessment of the size of the total COLD-SAT software generation task. Subsystem software has a great deal of commonality due to the use of the same SSP hardware design in the TT&C Subsystem, ACS, and Experiment Subsystem. Much of this software is of standard BSSD design.

Table 6-25

PROPULSION SUBSYSTEM COMPONENTS

COMPONENT	WEIGHT* (kg)	POWER (1) (watts)	VENDOR	HERITAGE	FAILURES/10^{6} HRS
Pressurant Tank-16.5 in. (1)	16.0	-	PSI	Classified Prog. - 2 lots	0.04
Latch Valve (Electric) (2)	0.7x2=1.4	15	Moog, Inc	HS 350 (Hughes)	0.35
Pressure Regulator (2)	0.7x2=1.4	-	Consolidated Controls	INSAT,GOES,Viking	0.55
Check, Relief Valves (2,1)	0.1x3=.3	-	VACCO	INTELSAT VI, MMB	0.35, 0.24
Gas, Liquid Filters (1,1)	0.1x2=.2	-	Puroflow	ERBS, Space Shuttle	0.12, 0.21
Pressure Transducer (3)	0.1x3=.3	0.5	Kulite	HAS, ERBS	1.39
Gas, Liquid Valves (Manual) (4,1)	0.3x5=1.5	-	Pyronetics	ERBS, MJS, HEAO	0.7, 0.7
Propellant Tanks (6)	7.6x6=45.6	-	PSI	FLTSATCOM, Centaur	0.35
Induced g-Thrusters 0.51 N (0.14-lbf) Thruster/valve (1)	0.3x1 = 0.3	9	Rocket Research	SPACENET, SATCOM	1.84
0.90 N (0.25-lbf) Thrusters/valves(4)	0.3x4 = 1.2	9	Rocket Research	ERBS	1.84
1.80 N (0.49-lbf) Thrusters/valves (4)	0.3x4 = 1.2	9	Rocket Research	ERBS	1.84
25.88 N (7.0-lbf) Thruster/valve (1)	0.7x1 = 0.7	28	Rocket Research	Viking	1.84
ACS Thrusters 0.25-lbf Thruster/valve (12)	0.3x12 = 3.6	9	Rocket Research	ERBS	1.84
Plumbing, Brackets	14.7	N/A	N/A	N/A	N/A
Thruster CAT BED Heaters (22)	N/A	1.0	Tayco	INTELSAT, GPS,ERBS	N/A
Thruster Valve Heaters (22)	N/A	0.25	Tayco	INTELSAT, ERBS	N/A
TOTAL	88.4	80.75			

*Best Estimate
(1) On Power per Unit

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6.7.1 Flight Software Requirements

This subsection describes the requirements that dictate the design of flight software. Design implementations to meet these requirements are presented in the subsections below.

The payload experiment imposes requirements on COLD-SAT that impact the software design. A total of five experiment telemetry formats must be accommodated. Multiple processes must be operated in parallel within each SSP--for example, experiment and critical status monitoring. Payload instrumentation and control items are summarized in **Table 6-26**, Experiment Interface Requirements.

Table 6-26

EXPERIMENT INTERFACE REQUIREMENTS

ECP INTERFACED SENSOR/ACTUATOR	LOCATION									TLM CHANNELS	
	LH_2 SS	TVS	LAD Supply	LAD Depot	Supply Tank	Depot Tank	OTV Tank	Press SS	Misc	Total Used	Spare Ckt*
Diode Temp. Sensor	18	58		40	78	51	35	8		325	35
Press. Sensor	23			8				12		43	4
Solenoid Valve	5	15	26	3	3	2				54	-
Pyrotechnic Valve						2	5			7	-
Heater					2	1	1			4	0
Liquid Flow Meter, LFM	5									5	-
Gas Flow Meter, GFM	9							2		11	-
Germanium Temp. Sensor					24					24	6
Liquid/Vapor Sensor	9		8	20	38					75	18
Mixer					1					1	-
Mixer Power Supply (Comp 3)									1	1	-
Accelerometer (Comp 3)									1	1	-

*All SFI muxes have one channel for offset correction

The maximum read rate of all experiment instrumentation is 1 Hz, except that line liquid/vapor (L/V) detectors must be read at 4 Hz. These rates and quantities lead to a requirement to read two analog words every 1.9 ms (including contingency). Reading of some instrumentation items must be in close time proximity; this is accomplished by a combination of multiplexer wiring and maintaining an invariant read sequence in the software. Cryo-valve actuation is a nontrivial process necessitated by the hardware requirement for a shared current source and the fact that multiple requests can be generated from each of three sources--experiment process, critical status monitor, and uplink commands.

The collection of data must be time-invariant with respect to the hardware (sample timing errors are equivalent to amplitude errors in changing data). This is especially important in the experiment since the amount of data items to be read necessitates the use of multiplexing of signals. Therefore, even constant signals become step functions after multiplexing.

In the interests of minimizing cost, several objectives are imposed on COLD-SAT software. The design shall exploit existing software as much as practical to reduce development, test, documentation, and maintenance costs. Substantial BSSD software, developed outside of the COLD-SAT program, is used to achieve this goal. High level language shall be used whenever practical. Software shall be so structured as to ease the tasks of initial design and modification of the experiment control processes. Software design shall also facilitate ground test as well as post launch diagnostics and modification.

The above characteristics also tend to enhance reliability. In the further interests of reliability, the software design shall provide for uplinked override of failed portions of the spacecraft or experiment code where such design provides overall enhancement of reliability. Duplication of software throughout subsystems shall be exploited wherever it leads to reduced total size of the development task. The design shall be so structured that it can be analytically proven that no combination of events can bring about a failure or degradation of required performance of the spacecraft. This latter requirement is crucial in providing the level of reliability required for flight software.

6.7.2 Software Design

The additional software for COLD-SAT will be written in a higher-level language (probably "C"). The interface to the physical hardware is accomplished by calls to operating system routines developed at BSSD. Figure 6-29 provides an example of the essential structure of these subroutines.

In this figure, the rc_analog_in routine is shown. To use this routine, the applications program only need to include a single line of the form "rc_analog_in(cn,y-field)". cn is the channel number of the analog channel to be read, and y-field is the variable in which the value read at the analog channel is to be stored.

Each of the other subroutines controlling standard I/O channels on the RCT cards is equally simple to use. They are listed in this same figure. The

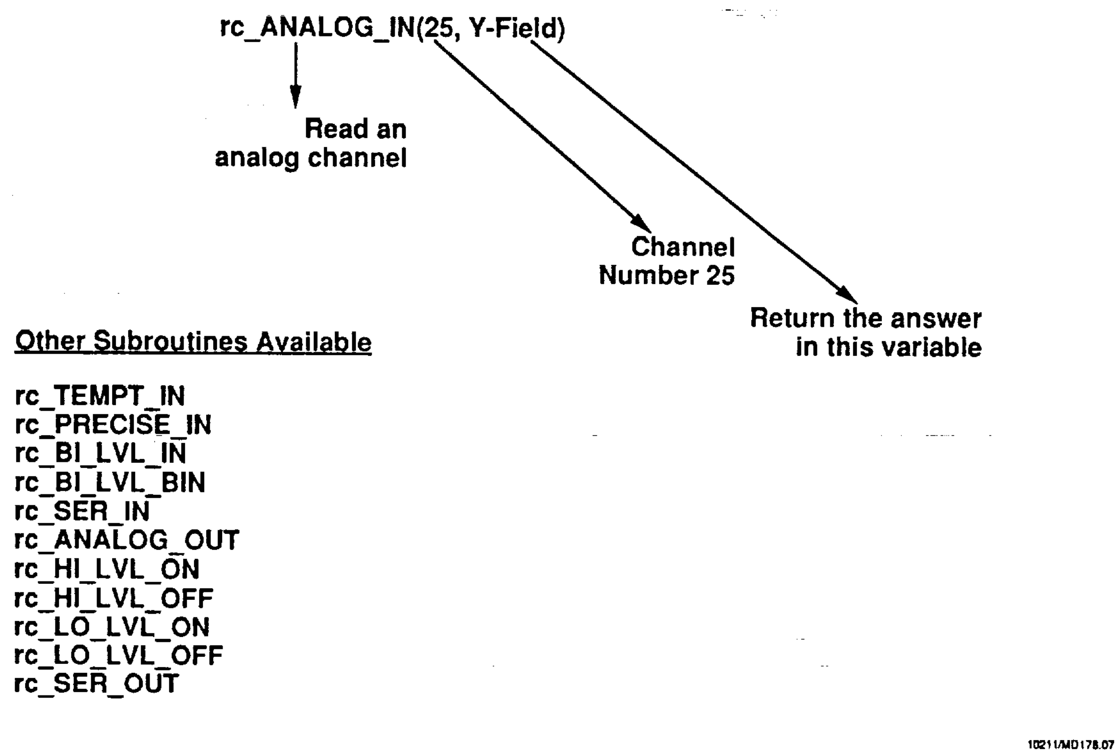

Figure 6-29. Operating system greatly simplifies software tasks.

applications program need only specify the type of channel to read (or write), the channel number, and either values to output or variables in which to return values read. The engineer creating the applications software need not become intimate with the detailed functioning of the interface hardware.

To appreciate the level of effort which is conserved by this modular approach to software design, consider **Figure 6-30.** This figure illustrates the process required to read an analog input channel. This process need not be implemented, or even understood by the individual software engineer developing an application for the subsystem processor. He only needs to invoke the operating system routine with a single line of code. The detailed interaction has already been designed and tested.

The memory structure utilized by the experiment control software is conceptually illustrated in **Figure 6-31.** The experiment executive will be contained in RAM. This executive will consist solely of broad commands which may be

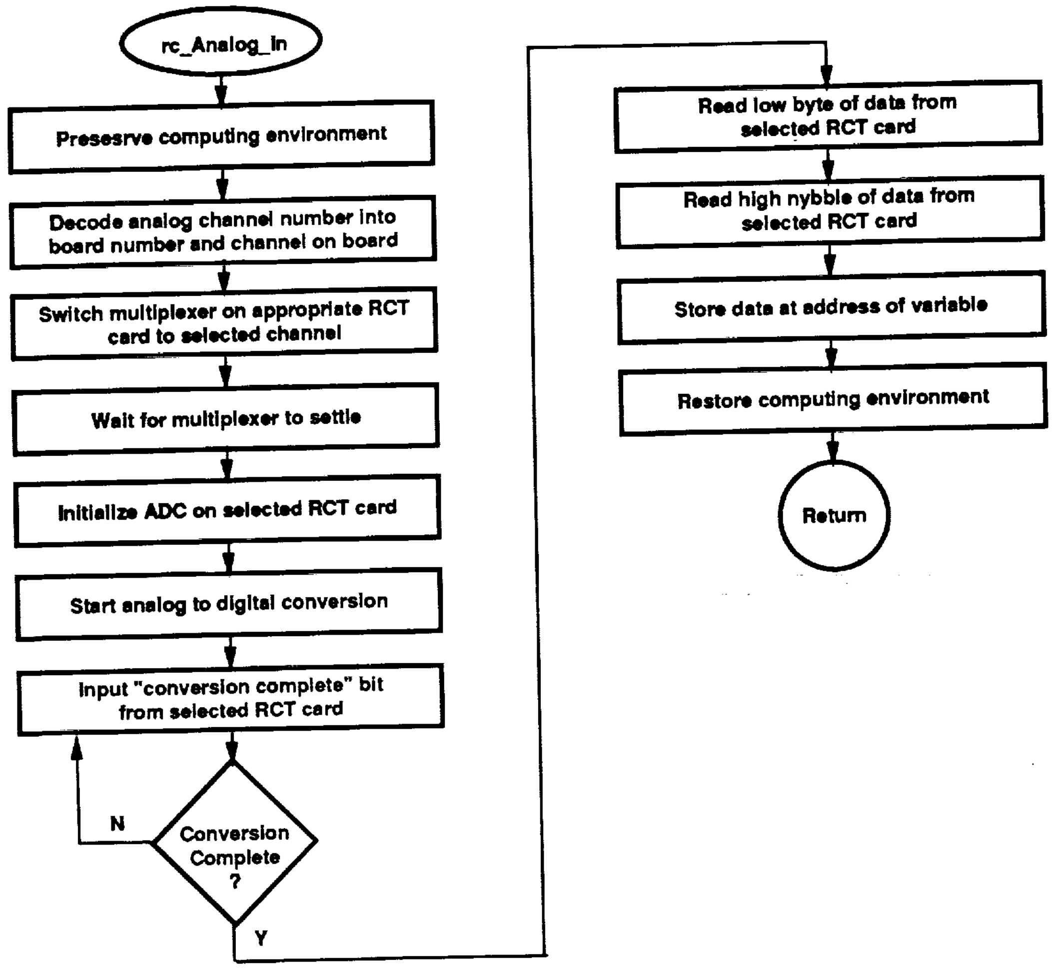

Figure 6-30. Functional flowchart of rc_ANALOG_IN.

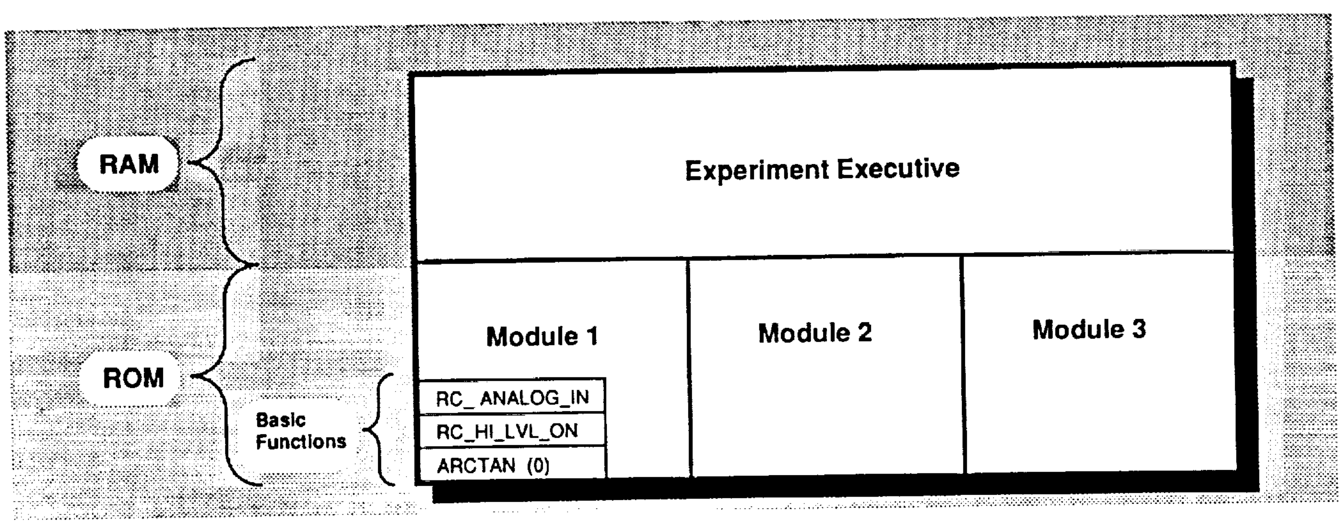

Figure 6-31. Conceptual representation of program structure.

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used to describe any of the possible experiments and the order in which these commands are to be executed. The modules which actually perform the broad commands are contained in processor ROM (Module 1, Module 2, etc.). These large modules, in turn, contain simple "C" code, and calls to even lower-level functions, which are also contained in ROM. This structure allows us to minimize the amount of program stored in RAM. Thus, the amount of station contact time to make program changes is minimized.

Figure 6-32 further illustrates this concept, by detailing the result of a processor reset. First, the processor performs some simple diagnostics to verify its own health. Next, the processor copies a default experiment executive (the "best guess" available at launch) from ROM to RAM. The processor then jumps into the RAM program. Periodically, the processor checks to see if a new experiment executive is being uploaded from the ground. If one is, the operation of the experiment executive is suspended until the uplink is complete. Once the new program has been uplinked, the processor once again jumps to the start of the RAM program, and begins to execute the new experiment executive.

Figure 6-33 illustrates the subsystem processor software development environment. This figure implies the following timeline.

The early development of the application program is done on the development station, and IBM compatible PC. Because the subsystem processor is based on the 80C86, it is possible to perform a substantial portion of the initial development under MS-DOS.

When the applications software is sufficiently mature, it is transferred to SDP RAM via an RS-232 link (a standard COM1 serial port on any MS-DOS machine). The software is then tested on what is substantially actual flight hardware.

The next step is to connect a compliment of RCT cards and any additional sensors and actuators required for further software testing. This is the sub-

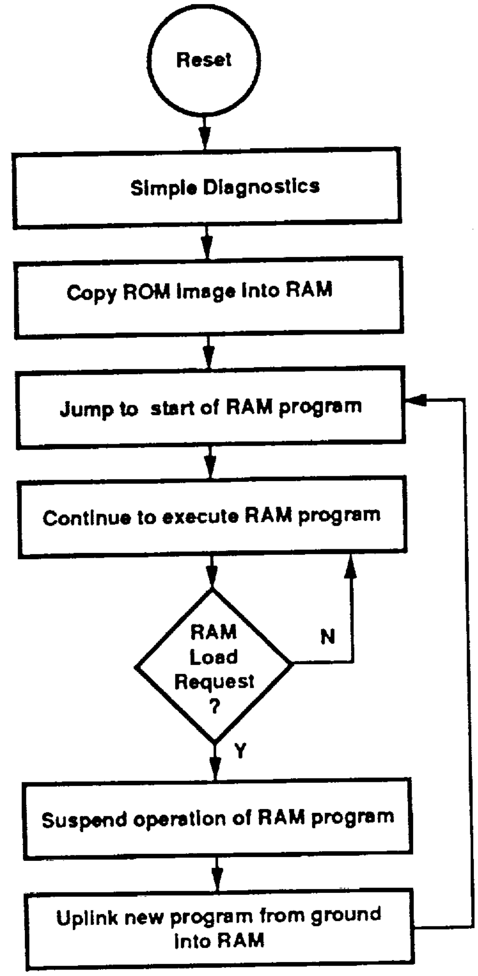

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Figure 6-32. Conceptual flowchart of RAM program management.

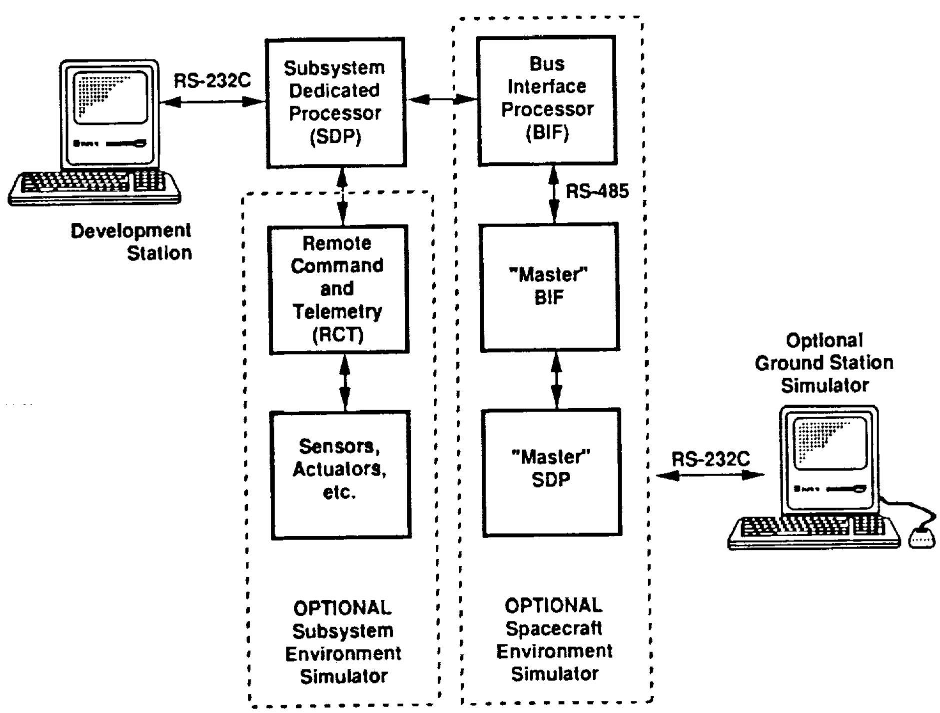

Figure 6-33. Subsystem processor software development environment.

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system environment simulator in our illustration. Note that only the sensors associated with this subsystem need to be supplied or simulated in this setup.

The interface between the applications program and the entire remainder of the spacecraft can be tested with the spacecraft environment simulator in our figure. This simulator consists of three readily available cards: 2 BIFs and a single SDP. Because the interface of each subsystem to the spacecraft at large is the LAN bus, these cards are all that are needed to simulate the entire spacecraft. The master SDP simulator must contain the same telemetry and command tables as will be used on the actual SDP bus master.

6.7.3 Software Modules

Most software for the experiment subsystem must be created as custom code, but a few modules can be modified from existing routines developed for the Ball subsystem processor. The modules in **Table 6-27** include all necessary features of the experiment software. Sizing estimates are based on comparison of these modules with previously developed modules of similar complexity. Details of the experiment control module sizing are given on **Table 6-28.** RAM estimates were made assuming that each line of C required 5 assembly level commands, and each command required 5 bytes. The ECP contains 312 K of RAM, so there is a large memory margin available for storing data tables and telemetry values.

Bus

Table 6-29 and **6-30** show the estimated complexity for COLD-SAT bus software which reside in the TT&C and ACS subsystem processors. The estimate, in lines of "C," were made by comparing modules for COLD-SAT with modules on other BASG programs. RAM size was estimated assuming 25 bytes of memory is needed for each line of "C" and adding memory for the data base and temporary storage needs.

Table 6-27

LINE OF "C" CODE FOR EXPERIMENT CONTROL SOFTWARE MODULES

MODULE	CURRENT ESTIMATE	CONTINGENCY	SEGMENTING AND FLAG CHECKS	TOTAL	ITERATION RATE
Master	150	150	195	495	10/sec
Pressure control	138	138	180	456	1/sec
Tank chilldown	150	150	195	495	10/sec
No-vent fill	96	96	125	317	10/sec
LAD fill-TVS	108	108	140	356	1/sec
LAD fill- vent	108	108	140	356	4/sec
Depth gauging	64	64	84	212	1/sec
Low-g fill	96	96	125	317	4/sec
Low-g drain	96	96	125	317	4/sec
LAD expulsion	96	96	125	317	4/sec
Line chilldown	96	96	125	317	10/sec
LH_2 subcooling	50	50	65	165	1/sec
Fluid dumping	96	96	125	317	1/sec
Totals	1,344	1,344	1,749	4,437	

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Table 6-28

EXPERIMENT SOFTWARE SIZING

MODULE	DEVOPMENT STATUS	LINES OF C	RAM (k bytes)	ITERATION RATE
Initialization	MS	60	1.5	1/sec
Main loop	S	100	2.5	16/sec
Hardware in/out	MS	150	3.75	16/sec
Counter updates	C	20	0.5	16/sec
Command interpreter	MS	150	3.75	16/sec
Telemetry formatting	S	150	3.75	1/sec
Cryo valve driver	C	100	2.5	4/sec
L/V sensor driver	C	100	2.5	4/sec
Critical status monitor	C	400	10.0	1/sec
Experiment control (13)	C	4,437	111.0	N/A
Experiment subroutines	C	700	17.5	10/sec
Totals		6,367	159.25	

S = Standard SSP code
MS = Modified Standard SSP Code
C= COLD-SAT Custom Code

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Table 6-29

SPACECRAFT TT&C MODULES

SOFTWARE MODULE	LINES OF "C" CODE	RAM REQUIREMENTS	ITERATION RATE
Antenna Pointing			1/10 min
Compute ephemeris	500 (M)	6.2	
Generate stepping cmds using gimbal readouts	150 (N)	1.9	
Command Processing			16/sec
Interpret uplink signals	100 (M)	1.3	
Execute stored and real time commands	160 (M)	2.0	
Telemetry Processing			1 frame/sec
Gather telemetry data	370 (M)	4.6	
Format data & create serial stream	130 (M)	1.7	
Data Storage Unit			1 dump/50 min (cycle through total memory)
Route record data	170 (E)	2.1	
Address playback data	150 (E)	1.9	
Bad section blockout	70 (M)	0.9	
Decode DSU commands	40 (M)	0.5	
Contingency Operation			1/sec (monitor)
Monitor selected Tlm.	210 (N)	2.6	
Execute contingency command list	70 (N)	0.9	
Thermal Control			1/5 sec
Read telemetry	30 (N)	0.4	
Provide heater on/off control	70 (N)	0.9	
Data Tables	N/A	2.0	N/A
TT&C Total	2220	29.9 Kbytes	

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(N) = New software
(M) = Modified software
(E) = Existing software

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Table 6-30

SPACECRAFT CONTROL SYSTEM MODULES

SOFTWARE MODULE	LINES OF "C" CODE	RAM REQUIREMENTS	ITERATION RATE
Normal Mode			1/5 sec
Integrate rates	100 (N)	1.3	
Sum rate & position sig.	50 (N)	0.6	
Make limit check	150 (N)	1.9	
Calculate "on time"	130 (N)	1.7	
Bias Z integrator based on sun position	70 (N)	0.9	
Correct X + Y integrator based on horizon sensor	200 (N)	2.5	
Software control	300 (N)	3.7	
Acquisition Mode			1/10 min.
Generate ephemeris	500 (N)	6.3	
Calculate desired mag field in S/C coordinates	300 (N)	3.7	
Find angular difference in desired & actual fields	150 (N)	1.9	
Compute thruster firings from 2 different measurements	80 (N)	1.0	
Total Control System	2030	25.5 Kbytes	
Battery charge control	320 (M)	4.0	1/sec
Load shedding	80 (M)	1.0	20/sec
Total Power System	400	5.0 Kbytes	
S/C TT&C (From Table 6-29)	2220	29.9 Kbytes	
Total Bus	4650	60.4 Kbytes	

CS.805

(N) = New software
(M) = Modified software
(E) = Existing software

Section 7
INTEGRATION AND TEST

This section describes the integration and test of COLD-SAT from the delivery of components (black boxes) for integration to the shipment of a fully assembled and tested spacecraft to the launch site.

7.1 INTEGRATION AND TEST

To facilitate overall COLD-SAT integration and test, system checkout/integration for the bus and experiment subsystems will initially proceed in parallel. **Figure 7-1** shows the bus/spacecraft integration and test flow while **Figure 7-2** shows the integration and test flow of the experiment subsystem up to its delivery for overall spacecraft integration. To enable this parallel approach mockups of the spacecraft structure need to be fabricated to mount compartments 1 and 2 containing the bus subsystem components since the experiment subsystem provides the primary structure for COLD-SAT.

Figure 7-3 shows the buildup of the experiment subsystem/prime spacecraft structure. The experiment LAN (RS 485 interface) is routed to a computer (that simulates the spacecraft and ground segment) and is used for testing the subsystem. Before assembly with the bus, the experiment subsystem will have been proof tested, cycled using hydrogen, tested with hydrogen in a thermal-vacuum chamber and the supply tank will have been vibrated to protoflight levels. Boil-off rates will be measured to estimate heat leaks and internal/external gas leaks measured. This will assure all systems are operating satisfactorily including the electronics and software. Final tests will be run using LH_2 and running some typical flight experiments from the SOC to identify and fix any problems identified with the total experiment subsystem. After completion of these tests the experimental subsystem is delivered for spacecraft integration.

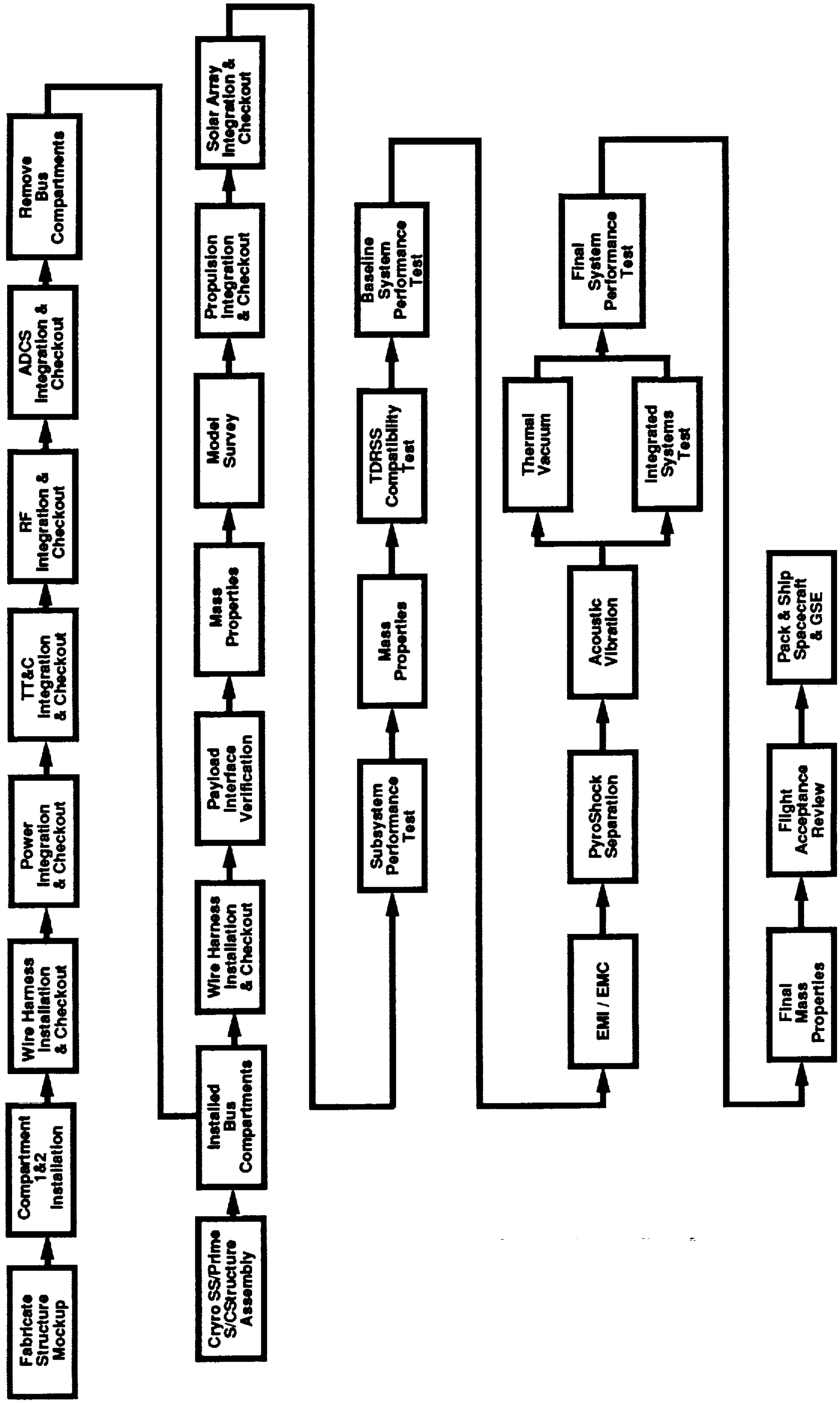

Figure 7-1. Bus/spacecraft integration and test flow.

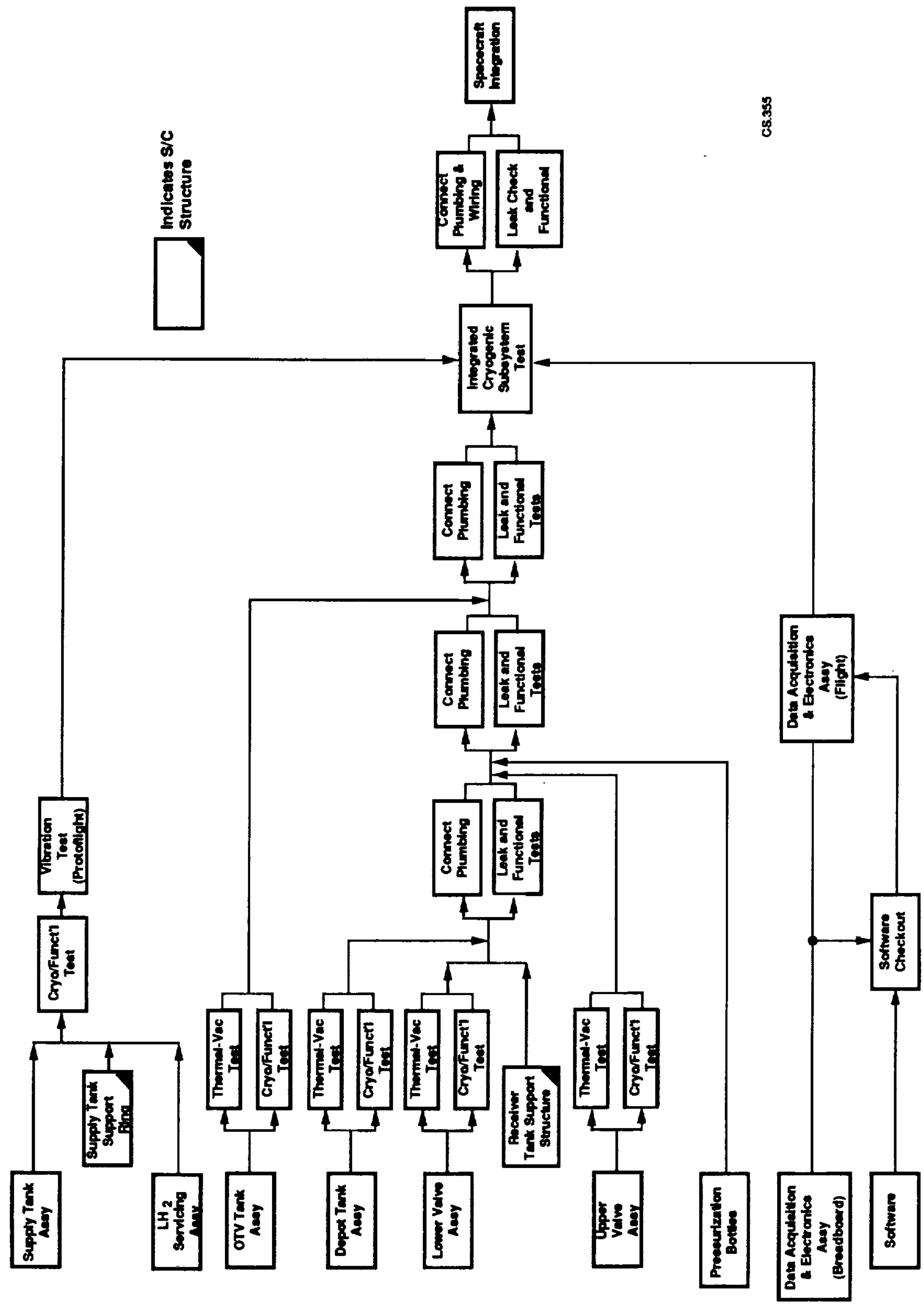

Figure 7-2. Integration and test flow, Experiment Subsystem.

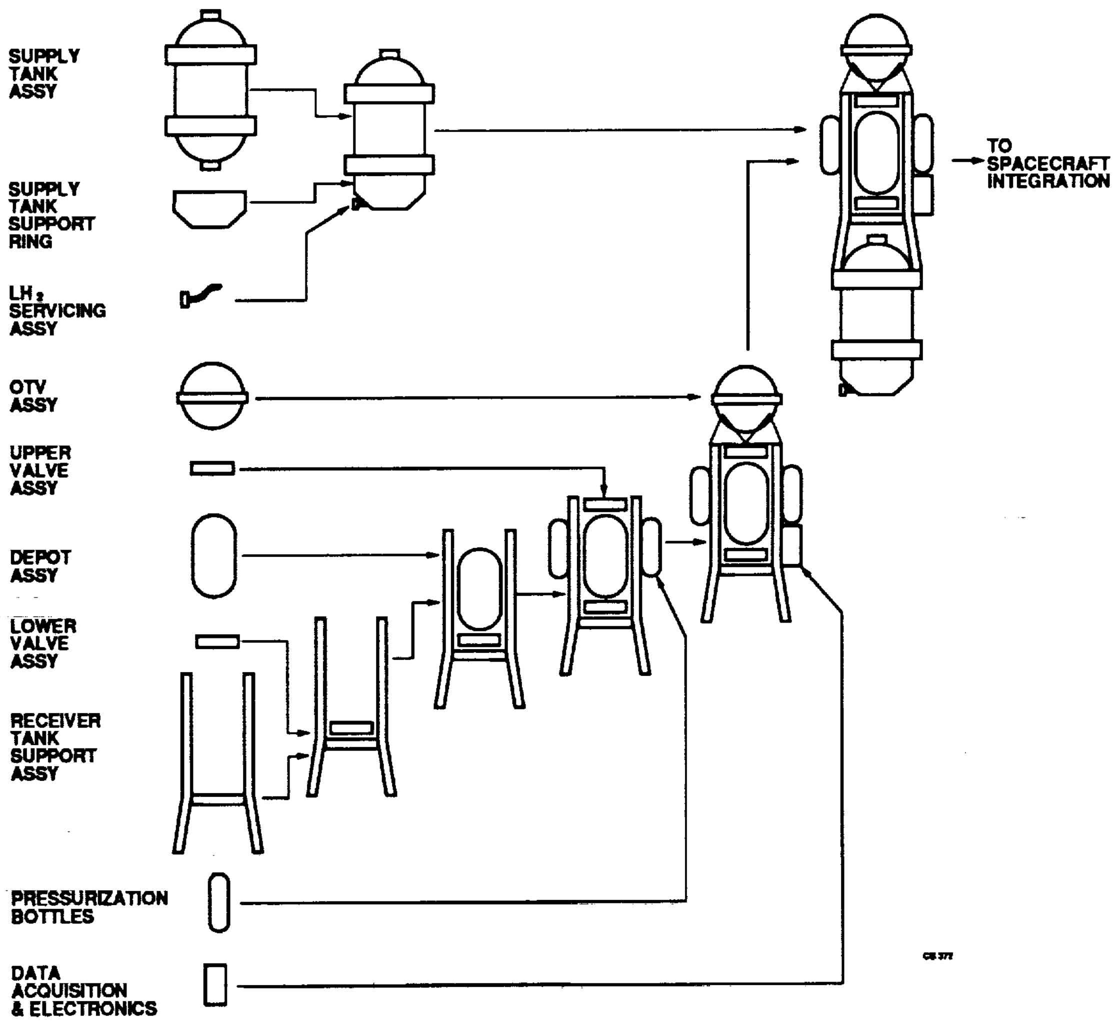

Figure 7-3. Assembly sequence, Experiment Subsystem/Prime S/C structure.

When the experiment subsystem is received the bus compartments are mated along with the propulsion module structure (simulated propulsion components) and simulated arrays. Mass properties are measured and a modal survey run to verify the analytical model of the structure.

Assembly is completed by installing the solar array and propulsion components. Functional testing is performed under room and thermal vacuum environments. During thermal vacuum, tests are run from the POCC and SOC to verify compatibility. Performance tests are run before and after acoustic exposure.

After final performance tests to assure spacecraft health and final mass property measurement, the spacecraft is packed and shipped to CCAFS.

Section 8

GROUND SEGMENT DESIGN

The COLD-SAT ground segment design can be broken down into two different support elements, the facilities and equipment needed for launch site operations and the facilities and equipment needed for post launch or flight operations. The ground segment design will utilize the standard KSC and GSFC facilities with hardware modifications to these facilities kept to an absolute minimum.

8.1 LAUNCH SITE DESIGN

The following paragraphs define the launch site ground segment design.

8.1.1 Launch Site Description

Figure 8-1 shows the launch site communication and data system flow, with each facility performing a specific task as defined below.

- MILA/MILA Relay - performs RF compatibility and serves as the gateway for communications to and from the NASCOM facility at GSFC

- Network Control Center (NCC) - not shown in figure, provides scheduling interface for all TDRSS elements, KSC and DSN

- Building AE - provides the following support at KSC
  - Houses the Mission Directors Center (MDC)
  - Launch Operations Communications Center
  - NASCOM interface (via building X/Y)
  - Coordination with the NCC for launch support
  - Monitoring of launch operations and overall mission activities

- Building AO - provides the following support at KSC
  - Receive, inspection and checkout of COLD-SAT (non-hazardous)

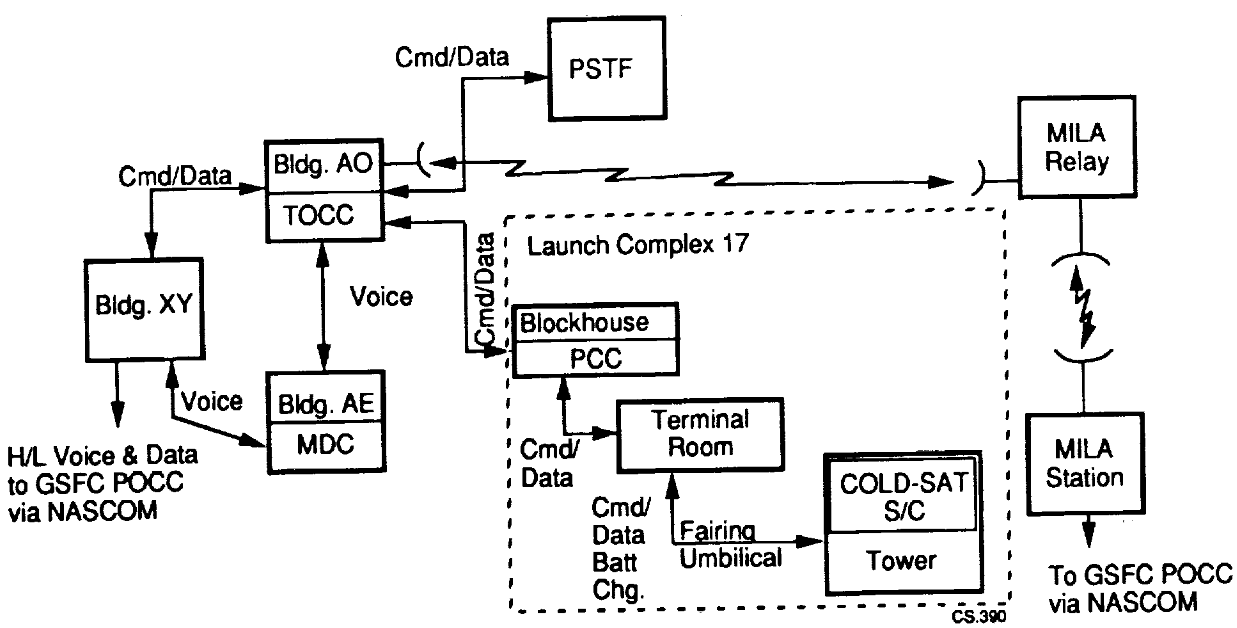

Figure 8-1. Launch site communcations and data system.

- Location of the Test Operations Control Center; test and operations personnel
- Building AM serves as an alternate facility

- Payload Spin Test Facility (PSTF) - provides the following
  - Install ordnance (hazardous)
  - Load hydrazine propellant
  - Pressurize propellant system

- Launch Complex 17 - provides the following
  - Interfaces the COLD-SAT and the TOCC
  - Clean environment for COLD-SAT on the launch vehicle
  - Accommodation for the H_2 loading systems (not currently available)

8.1.2 Launch Site Requirements

The following paragraphs define the requirements for the launch site facility.

Building AO

The processing facility requirements that are required at building AO are as follows.

• Clean room floor space	30 x 40 ft
• Cleanliness level	100,000
• Electrical power level	1 Kw
• Equipment floor space	15 x 20 ft
• Door size	20 ft
• Hook height	20 ft
• GSE storage	10 x 15 ft
• Office space	20 x 30 ft
• NASCOM interface	Modem, 9.6 Kbps
• TOCC/cleanroom interface	3 Tw. shield pair

Payload Spin Test Facility (PSTF)

Once the spacecraft has completed its integration and test activities within the non-hazardous processing facility, building A0, the spacecraft will be transported to the hazardous facility, PSTF, for processing and continued checkout. The facility requirements are defined as follows.

• Floor space	15 x 25 ft
• Cleanliness	100,000
• Electrical power	3 Kw, 110 & 220 volts
• Door size	15 x 17 ft wide
• Hook height	20 ft
• GSE	$N_2 H_4$ loading cart

Launch Complex 17

Prechilling is accomplished by flowing cold helium through the supply dewar. The helium is supplied by a LHe tanker located adjacent to the LH_2 storage, the layout and positioning of the various cryogenic tanks are shown in Figure 8-2. The helium is routed to the supply tank through the LH_2 transfer line, prechilling the line along with the supply tank. The GSE will support the 2-stage pressurization of the COLD-SAT pressurant bottles. The bottles will be pressurized to their mission requirement during the final launch countdown sequence.

8.1.3 Facilities Modifications

The COLD-SAT spacecraft and supporting elements will make use of the existing facilities without modifications or specialized hardware or software to the greatest degree as possible. The following elements require no mission unique modifications to support COLD-SAT.

- MILA and MILA Relay
- Building A0
- Building X/Y
- Mission Directors Center (MDC)
- Payload Spin Test Facility (PSTF)
- NASA Communications (NASCOM)

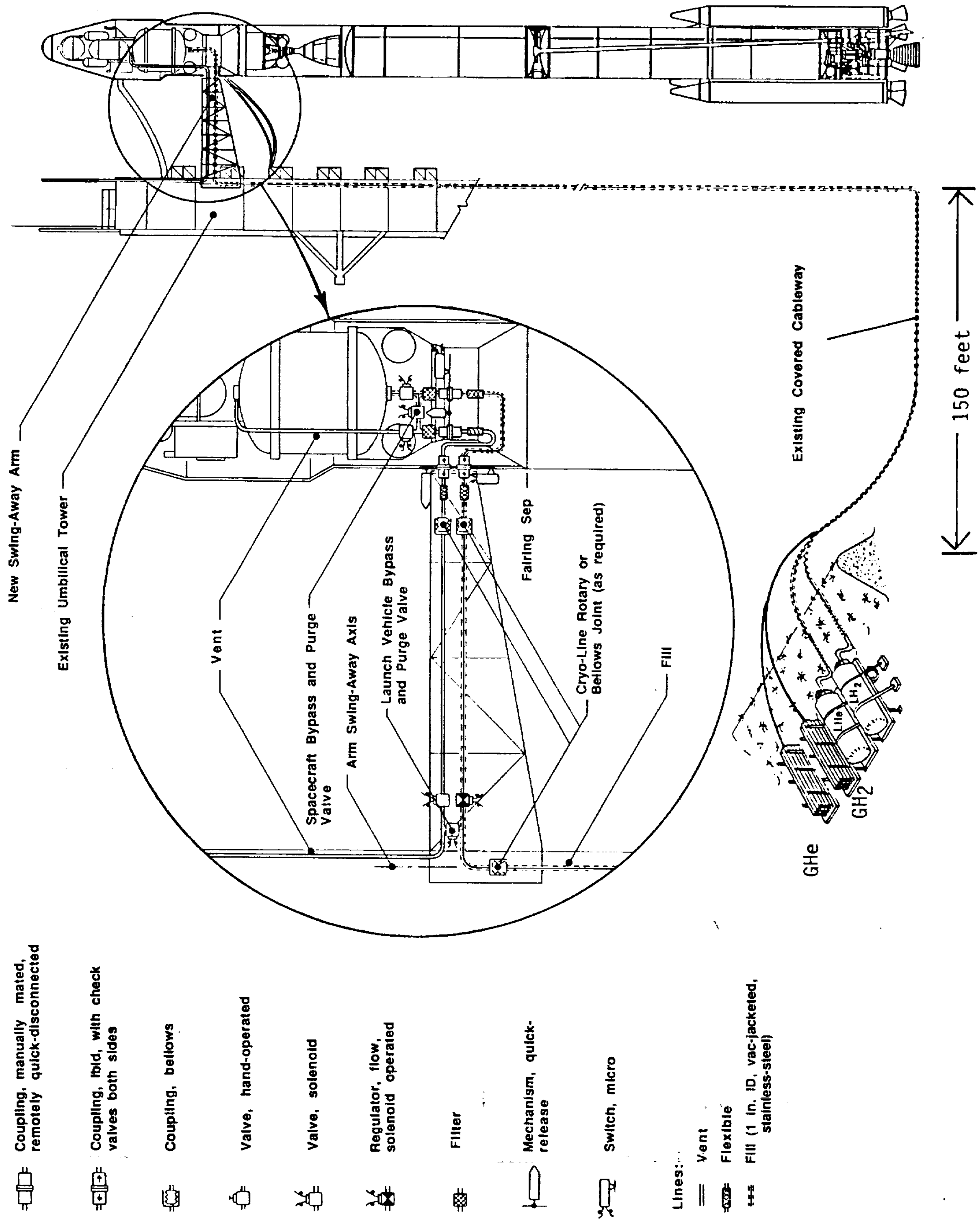

Figure 8-2. LH_2 loading concept.

The following facilities will require minor changes to their software systems or data bases to incorporate the mission specific COLD-SAT support parameters (i.e., spacecraft formats, data rates and communication routing) and scheduling requirements.

- TDRSS data acquisition and communications
- Network Control Center (NCC)
- Central Data Handling Facility (CDHF)
- Deep Space Network (DSN)

The launch facility will require modifications to complex 17 in order to support the COLD-SAT mission, these modifications are defined as:

- An automatic disconnect system used in loading LH_2 and pressurant. This system will automatically disconnect and swing away at liftoff.

- The umbilical tower will be modified to provide a vent/burn stack for hydrogen.

- Modifications to supply vacuum jacketed lines and a pad for the loading system GSE.

Quantitative requirements as defined as:

ITEM	PHYSICAL	ELECTRICAL
• Experiment		
- Exp. monitor lines (powered from chg line)	N/A	4 - TS pair
- Cryo loading control console (blockhouse)	24" in 17" rack	300 W
- Control lines to loading system	N/A	10 #20
• Bus		
- Power control console (Payload level)	16 sq ft	1 Kw

ITEM	PHYSICAL	ELECTRICAL
- Battery charge panel (blockhouse)	24" in 17" rack	200 W
- Charge lines, blockhouse to pad	N/A	2 #12 & 2 #20
- Telemetry lines, Payload level to blockhouse	N/A	TS pair
- Command lines, blockhouse to Payload level	N/A	TS pair
- Cmd & Tlm: hanger to Payload level	N/A	2, 12.6 Kbits
- Voice: Payload, blockhouse, hanger	N/A	1 circuit

8.1.4 Ground Support Equipment

While at KSC the COLD-SAT spacecraft will require support from the Ground Support Equipment (GSE), most of which will be the same equipment utilized at BSSD during the Integration and Test (I&T) phase. The loading systems will be similar to that used at BSSD except the source tanks will be larger. The GSE required at the launch site is as follows.

- GH_2 loading system
- LH_2 loading system
- LHe pre-cooling system
- GHe loading system
- Vacuum pump
- N_2H_4 loading cart
- Blockhouse rack
  - COLD-SAT control
  - H_2 loading
- Test Operations Control Center (TOCC)

The TOCC data handling hardware and software system design is based upon a low cost micro-processor design developed at BSSD, the same configuration will be used for the development of the SOC system design.

8.2 GROUND SEGMENT DESIGN FOR FLIGHT OPERATIONS

The post-launch data systems required for support to the COLD-SAT mission are the standard support services given to other GSFC TDRSS missions. The GSFC flight support system is designed to give mission operations support to a variety of different missions with a minimal of hardware or software changes. Once the COLD-SAT mission requirements have been defined they will be incorporated into the GSFC ground system design for flight support. **Figure 8-3** shows the software changes required to support the COLD-SAT mission. Hardware changes require the installation of operational communications data circuits (9.6 Kbps) between the CPOCC and SOC at LeRC. **Figure 8-4** shows the end-to-end COLD-SAT ground segment design. The ground segment used during the flight operations is largely transparent to the CPOCC and FOT. It provides the command and telemetry interfaces to the COLD-SAT spacecraft via TDRSS, NASCOM and MSOCC while the remaining facilities, except the SOC, provide operational data products in pre-defined formats and methods. The facilities listed below provide the following support.

- Tracking & Data Relay Satellite Systems (TDRSS) provides
  - MA telemetry, tracking and command forward/return services
  - Relays COLD-SAT data to/from the CPOCC
  - SA back-up as required
  - White Sands Ground Terminal interface to the NGT

- Deep Space Network (DSN) 26 meter subnet provides
  - emergency backup support for TDRSS

- NASA Ground Terminal (NGT) provides
  - Interface between the White Sands Ground Terminal and CPOCC
  - TDRSS forward and return link monitoring
  - COLD-SAT data recording and playback

- Network Control Center (NCC) provides
  - Monitoring and control of the TDRSS
  - Schedule support elements based upon inputs from the CPOCC
  - Coordinate back-up and contingency support to COLD-SAT

Project Operations Control Center (POCC)

Functions:
- Realtime satellite command, control, health, and status
- Realtime experiment data and command interface to LeRC SOC
- Command management system (experiment inputs from LeRC SOC)
- Realtime attitude data stripped and sent to Flight Dynamics
- Coordination with NCC for TDRSS (or DSN backup) mission support

New Mission Unique Hardware:
- POCC modem/data line interface to LeRC for experiment data and command

Mission unique Software Modifications (std. GSFC service):

	Estimated Effort	Program Risk
- Cold-Sat POCC data base	6 Man-months	Low
- Multisat Applications Executive	24 Man-months	Low
- Stripping Attitude Data for FDF	6 Man-months	Low
- Realtime Experiment Data Interface to SOC	6 Man-months	Low
- Command Management System	18 Man-months	Low
- Realtime SOC Cmd throughput	9 Man-months	Low
Total =	69 Man-months	Low

Flight Dynamics & Orbit Support Computing (FDF/OSC)

Functions:
- Process tracking data and prepare orbit predicts & vectors
- Prepare yaw maneuvers and monitor attitude/orbit maneuver performance

New Mission Unique Hardware: NONE

Mission Unique Software Modifications:

	Estimated Effort	Program Risk
- Develop yaw maneuver commands and make backup attitude computations from magnetometer data	24 Man-Months	Low

Data Capture Facility

Functions:
- Receives & stores incoming telemetry & schedules data
- Distributes data quality information
- Strips & forwards attitude data to Flight Dynmics Facility

New Mission Unique Hardware: NONE

Mission Unique Software Modifications:

	Estimated Effort	Program Risk
- Software required to strip attitude data for FDF (playback data)	6 Man-Months	Low

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Figure 8-3. Facilities hardware and software; modified.

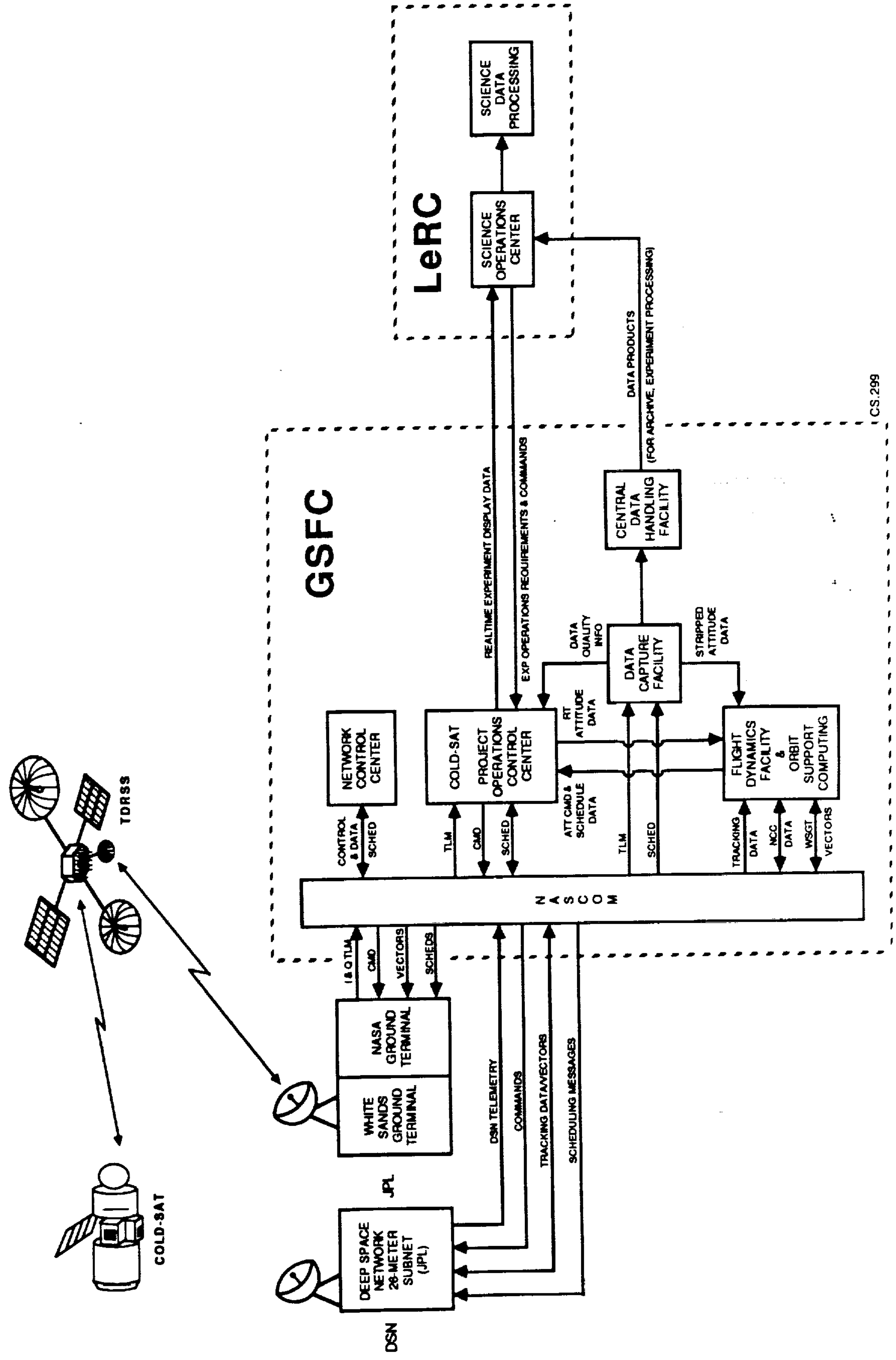

Figure 8-4. Post launch ground data communications system.

- NASA Communications (NASCOM) provides
  - Communication interfaces between all GSFC support element
  - Communications circuits 9.6 Kbps between the CPOCC and SOC
  - Data quality monitoring for COLD-SAT communications
  - Back-up circuits to all support elements

- MultiSat Operations Control Center (MSOCC) provides
  - Space for CPOCC and operations personnel
  - Computer hardware and associated support equipment
  - Software data bases to drive the resident spacecraft OCC's
  - NASCOM interfaces for data communications

- COLD-SAT Project Operations Control Center (CPOCC) provides
  - Realtime satellite command, control, health and status
  - Command management system (experiment inputs from LeRC)
  - Realtime attitude data stripped and sent to Flight Dynamics
  - Coordination with the NCC for TDRSS and DSN support
  - Coordination with the NCC for GSFC facility support, FDS, DCF, etc.
  - Interfaces directly with the SOC for realtime Cmd and Tlm support

- Flight Dynamics and Orbit Support Computing Facility
  - Process tracking data and prepare orbit predicts and vectors
  - Prepares for yaw maneuvers and monitor maneuver performance

- Data Capture Facility
  - Receive and store incoming telemetry data
  - Distribute data quality information
  - Strip and forward attitude data to Flight Dynamics for processing

- Central Data Handling Facility
  - Receive, process and archive mission data
  - Ship mission data to LeRC

- Science Operations Center (SOC) at LeRC
  - Assemble experiment command sequences
  - Analyze experiment data
  - Input into CPOCC scheduling

The Science data center at LeRC ground system design is based upon a 80386 micro-processor based workstation that has been utilized at BSSD for a number of DoD and commercial system projects and has been designed to support the GSFC utilized STOL operational system. This design incorporates both low cost and system flexibility that are required for COLD-SAT.The facility requirements at LeRC to support the SOC workstations are:

• Workstation areas	3, 4x8 foot desks
• Office area	20x20 foot area
• Electrical	110 volts
• Storage/filing cabinets	2
• Book cases	3
• Communications	1 voice, 1 data 9.6 Kbps
• Public telephone	1 voice, 1 FAX line

The SOC will have 3 workstations which can be configured identically and interchanged without any hardware modifications. Two of the workstations will be utilized for COLD-SAT command and telemetry support and the other unit will be used for SOC/LeRC mission planning and command management. The SOC will receive the COLD-SAT telemetry data from the CPOCC via the NASCOM provided circuits, once the data is received at the SOC the data will be recorded on a 96 Mbyte hard disk and post pass dumped to 8 mm data cartridges for historical purposes. The telemetry data will be also be displayed in realtime on the system monitors for operational support, the data will be converted to engineering units for display, the out of limit conditions will be displayed and alarmed to provide operator alerts. The out of bounds command data will be echoed back to the hardware system for validation prior to being transmitted to the CPOCC. The SOC software operational system will be a window menu driven system that will allow the operator to select the telemetry display parameters and also select the appropriate STOL command sequences that will be transmitted to the CPOCC. The CPOCC data base will be developed to allow specific COLD-SAT commands directly from the SOC, hazardous or critical command sequences originating from the SOC will require a release from the CPOCC software system. Although these data and communications system are tested during the pre-launch phase of operations their prime support phase is during the post-launch phase or mission operations. The same design for the SOC will be utilized for the TOCC during the I&T phase of operations. **Figures 8-5 and 8-6** depict the system layout and hardware/software system architecture.

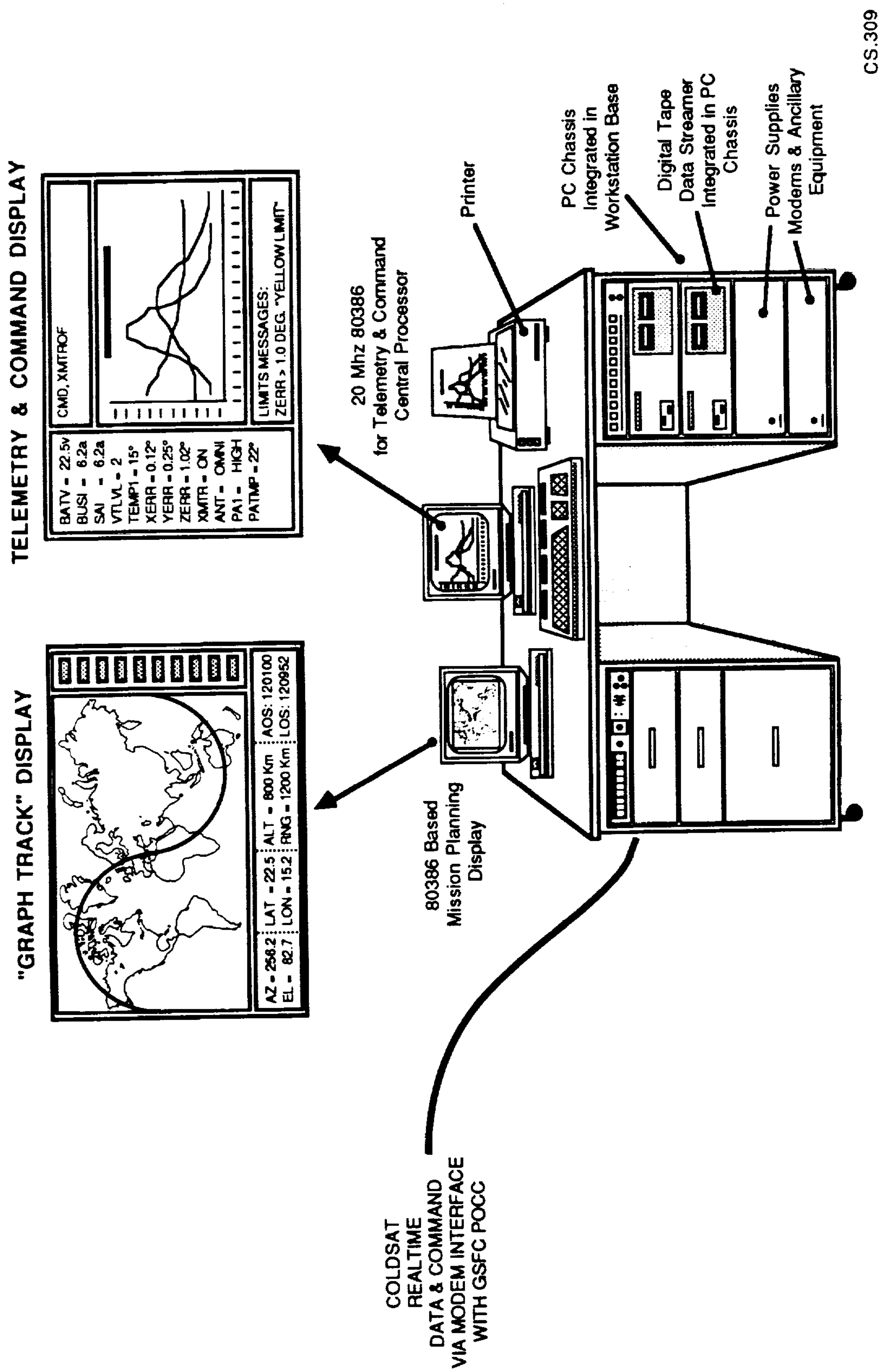

Figure 8-5. Science operations center design is workstation based.

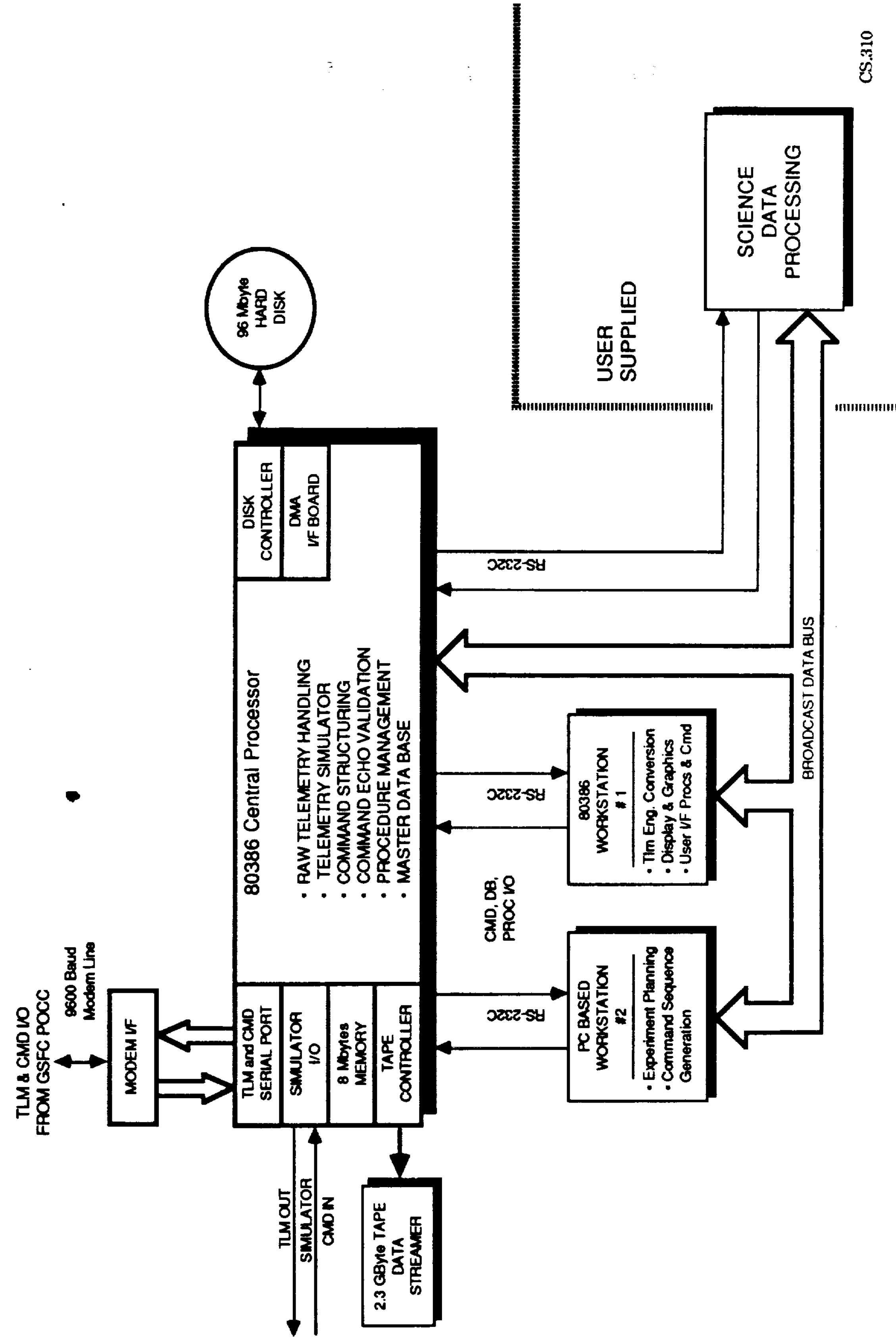

Figure 8-6. SOC architecture replicates TOCC used during I&T.

Section 9
COLD-SAT OPERATIONS

The COLD-SAT operations scenarios for both the launch and flight operations are based upon the successful methods utilized at BSSD in support of other Goddard Space Flight Center (GSFC) missions such as Earth Radiation Budget Satellite (ERBS) and Solar Mesosphere Explorer (SME). In these scenarios COLD-SAT shall take full advantage of the standard GSFC support services provided through the Tracking and Data Relay Satellite System (TDRSS), Multi-Satellite Operations Control Center (MSOCC), NASA Communications (NASCOM) and the other GSFC resident support facilities. In addition to the normal TDRSS communications support given to the COLD-SAT mission, backup support will be provided by the Deep Space Network (DSN) 26 Meter Subnet utilizing the three remaining ground network tracking stations. Utilizing the standard services and avoiding any unique mission hardware and software configurations allows for a cost effective and efficient mode of operation. The COLD-SAT operations can be broken down into two phases of operations, first, Pre-launch operations that commence 30 to 36 months prior to launch and conclude with the second phase of operations or on-orbit activities that will continue until successful mission termination. Section 10.1 details the launch operations and Section 10.2 covers the on-orbit operations. **Figure 9-1** shows a typical mission operations flow chart required for COLD-SAT operations. These activities are preformed by the COLD-SAT Flight Operations Team (FOT) and begin at the mission Preliminary Design Review (PDR) to insure the mission requirements are well defined and incorporated into the over all COLD-SAT mission design.

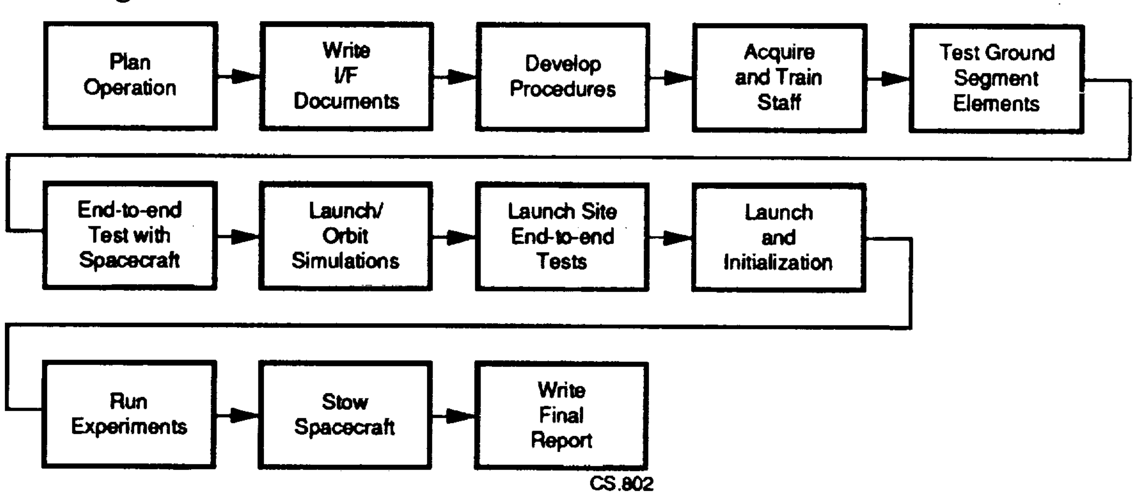

Figure 9-1. Operations flow chart.

9.1 LAUNCH SITE OPERATIONS

The KSC pre-launch activities will commence with the arrival of the GSE and COLD-SAT spacecraft at the cape. After the payload has been unpacked and inspected a comprehensive series of tests, identical to those tests run after the COLD-SAT environmental tests, will be run and the data compared to the previous tests. The final end-to-end test with the ground segment (MILA/CPOCC/NASCOM/SOC) will be run to assure all segments, space and ground are ready for launch. **Figure 9-2** shows the launch site processing flow, while **Figure 9-3** shows the documentation requirements and schedule. The CPOCC will monitor the launch site operations via NASCOM and the Test and Operations Control Center (TOCC) GSE. The COLD-SAT spacecraft and associated ground support equipment will arrive at KSC for receiving, inspection and testing. Building AO, or AM as the alternate, will house the spacecraft for the first 4 weeks at KSC, while the TOCC and test personnel will remain within AO until the COLD-SAT spacecraft is initialized on orbit. While in Building AO the spacecraft will undergo identical testing that was preformed at the BSSD facility to verify the spacecraft performance and experiment integration. The TOCC and test personnel will test the spacecraft in accordance with detailed spacecraft and launch test plans and procedures. These plans and procedures will define the electrical, mechanical and communication link verification step required prior to moving the spacecraft to the PSTF. Upon completion of the checkout at building AO the spacecraft will be moved to the PSTF for the hazardous processing. While within the PSTF, COLD-SAT will undergo Reaction Control System (RCS) and cryogenic leak checks and hydrazine loading.

After the spacecraft has completed its processing and hydrazine loading and pressurization at the PSTF the spacecraft will be moved to the launch complex 17 and mated to the launch vehicle. Command and telemetry signals will be routed to the TOCC through the 2nd stage umbilical. The CPOCC at GSFC will continue to monitor the on going launch pad operations via the MILA and NASCOM interfaces. The umbilical interface will also allow for the monitoring and battery changing within the complex 17 blockhouse. At launch minus 5 days the composite pressure bottles will be pressurized to 13.8 MPa (2000 psia), this will allow for safe pad access, even though the bottles have been

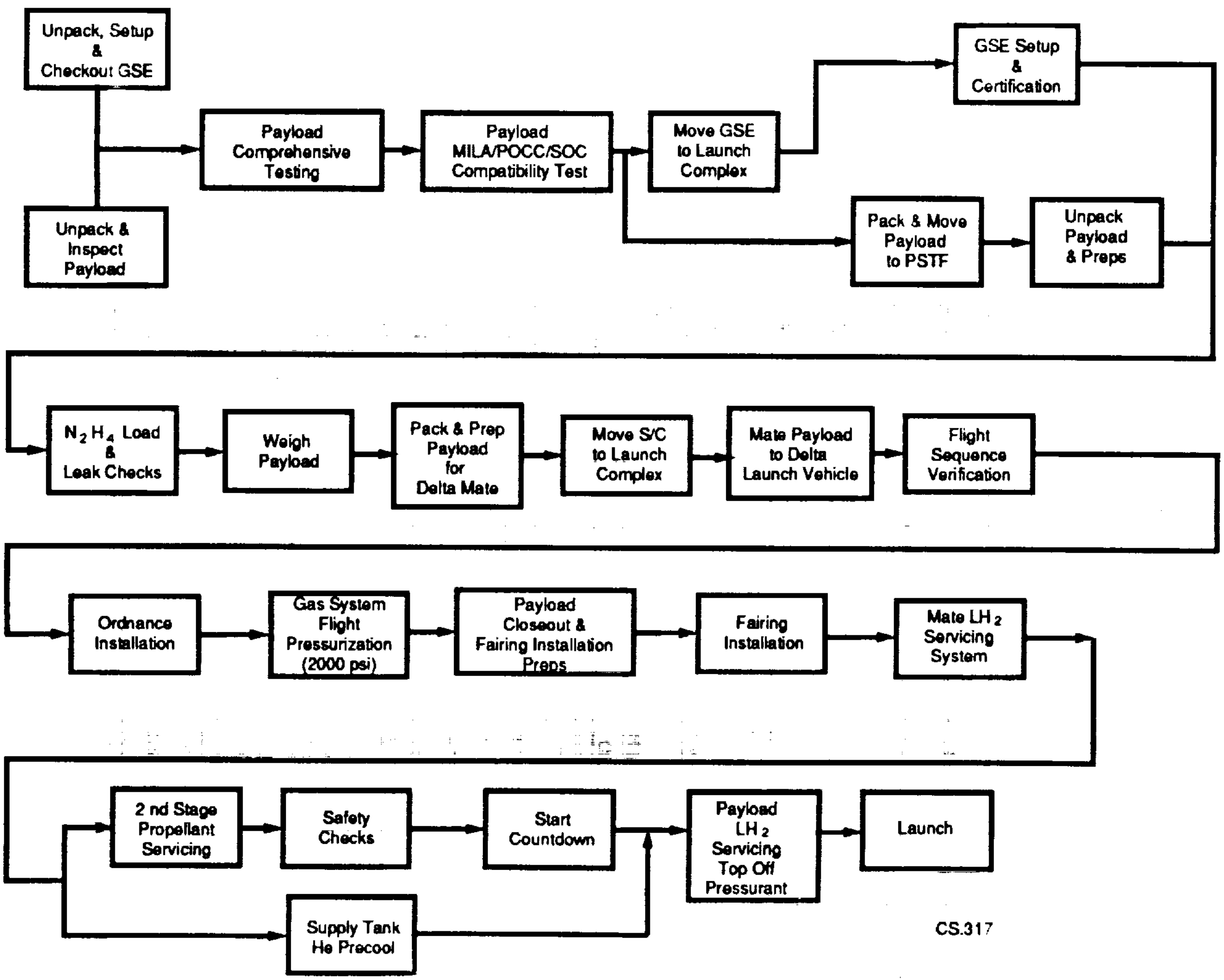

Figure 9-2. Launch site processing flow.

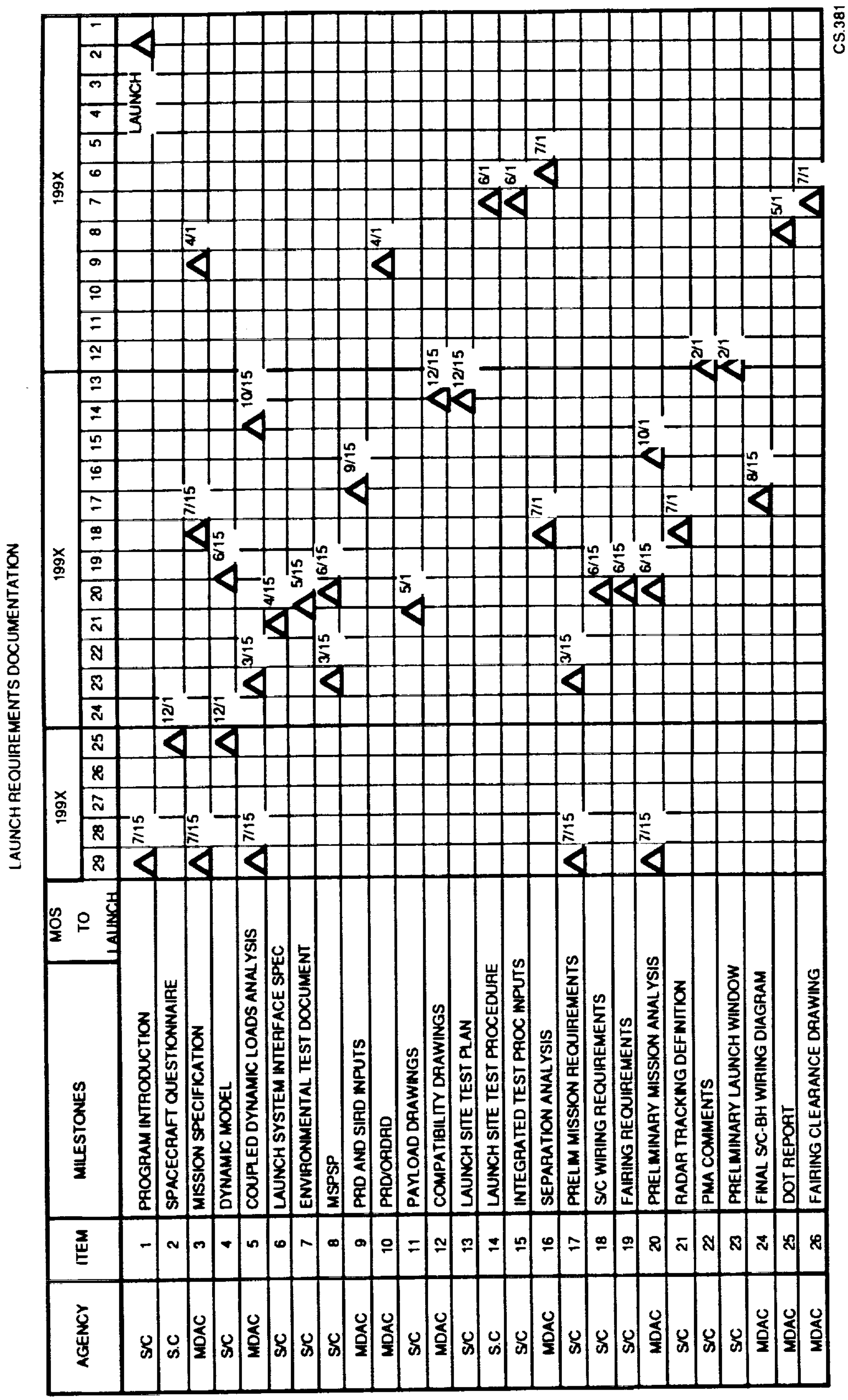

Figure 9-3. Documentation requirements.

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designed for a safety factor of two based upon a 27.6 MPa (4000 psia) rating. A prechill will start several days prior to the LH_2 loading sequence to allow thermal stabilization of the supply dewar and insulation system. The supply tank will be pre-chilled to less than 60 K thus removing 98 percent of the thermal energy of the supply tank cooldown mass. At launch minus 3 hours the LH_2 loading will commence; filling of the supply tank will take approximately two hours. After the supply tank has been filled the tank will be locked up and the transfer lines warmed with a helium gas purge. The LH_2 loading at L-3 hours will have to be closely monitored so as not to interfere with the Delta terminal countdown operations. In addition to the LH_2 loading, GH_2 and GHe will be used to fill the COLD-SAT pressurant bottles. As stated earlier at launch minus 5 days the bottles will be pressurized to 13.8 MPa (2000 psia), at launch minus 6 hours the bottles will be pressurized to their mission requirements, 27.6 MPa (4000 psia). This is just prior to tower pull-back and the disconnection will be made manually.

The culmination of the COLD-SAT launch operations will be the launch from complex 17. The CPOCC at GSFC will monitor the launch vehicle terminal count and monitor the health and status of the spacecraft systems via the telemetry communications links. The mission operations plans and procedures will define the state of the various subsystems on board the spacecraft. The control of the mission will be under the direction of the KSC and the mission directors center from launch until COLD-SAT separation from the launch vehicle third stage. Once separated from the vehicle the direction of the COLD-SAT mission will pass on to the CPOCC at GSFC.

9.2 FLIGHT OPERATIONS

The first step in properly planning and execution of the mission requirements is to insure that the requirements are well defined and documented within the existing documentational systems. The top level COLD-SAT mission operations requirements are defined within the Ground Segment Requirements Document (GSRD), these requirements then flow down to the other GSFC support documents as shown in **Figure 9-4**. The following are top level documents that drive the GSFC system.

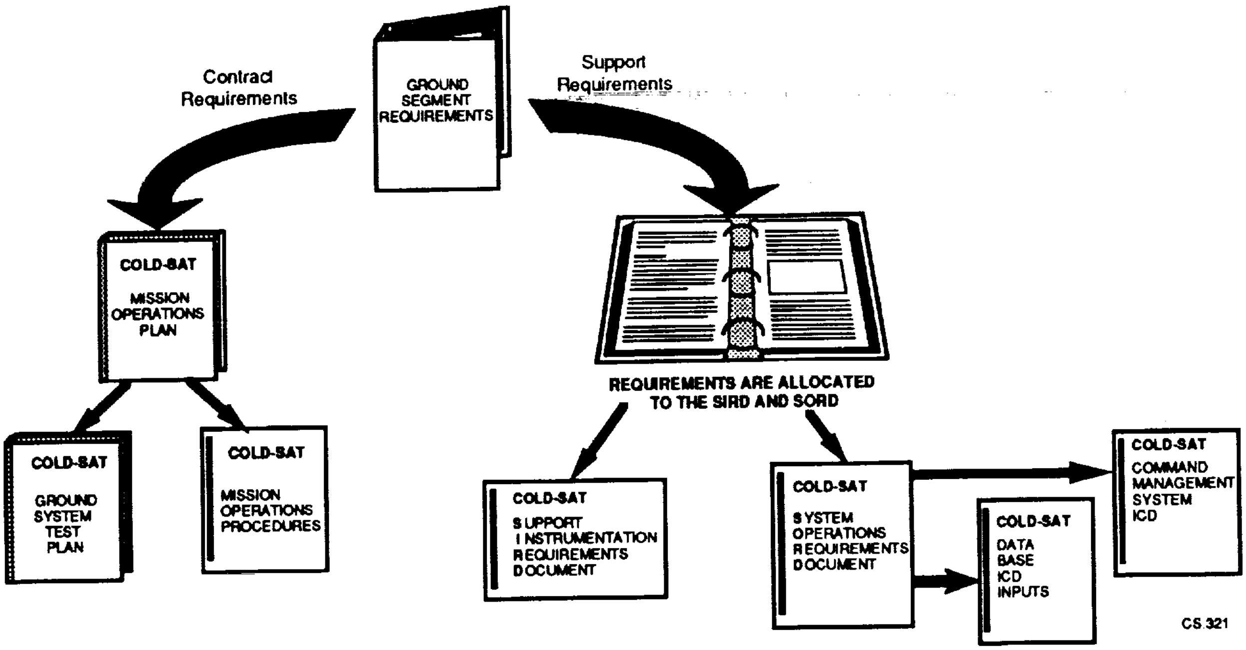


Figure 9-4. Requirements documentation.

Support Instrumentation Requirements Document (SIRD), defines the GSFC facility level requirements for the mission and secures the initial NASA commitment for GSFC resources to the mission. This document is signed-off and approved by the all supporting elements, i.e., Goddard Space Flight Center (GSFC), Lewis Research Center (LeRC), Jet Propulsion Laboratory (JPL) and NASA Headquarters. This document is typically developed beginning at launch minus 30 months and signed-off by launch minus 18 months, thus giving all elements the needed time to respond to its requirements.

System Operations Requirements Document (SORD), defines the detailed support requirements, H/W, S/W, Ops, I/F's etc., for each supporting element.

Interface Control Documents (ICD's), defines and agrees to specific support products, formats and interfaces between the project and the each supporting facility.

The above documents define the high level requirements that are levied upon various facilities or elements that will support the COLD-SAT mission. These requirements then flow down into mission unique plans and procedures for each facility, this same process occurs within the COLD-SAT Project Operations Control Center (CPOCC).

COLD-SAT Mission Operations Plan (MOP), defines the overall COLD-SAT mission operations for both the space and ground systems including nominal, contingency and emergency operations.

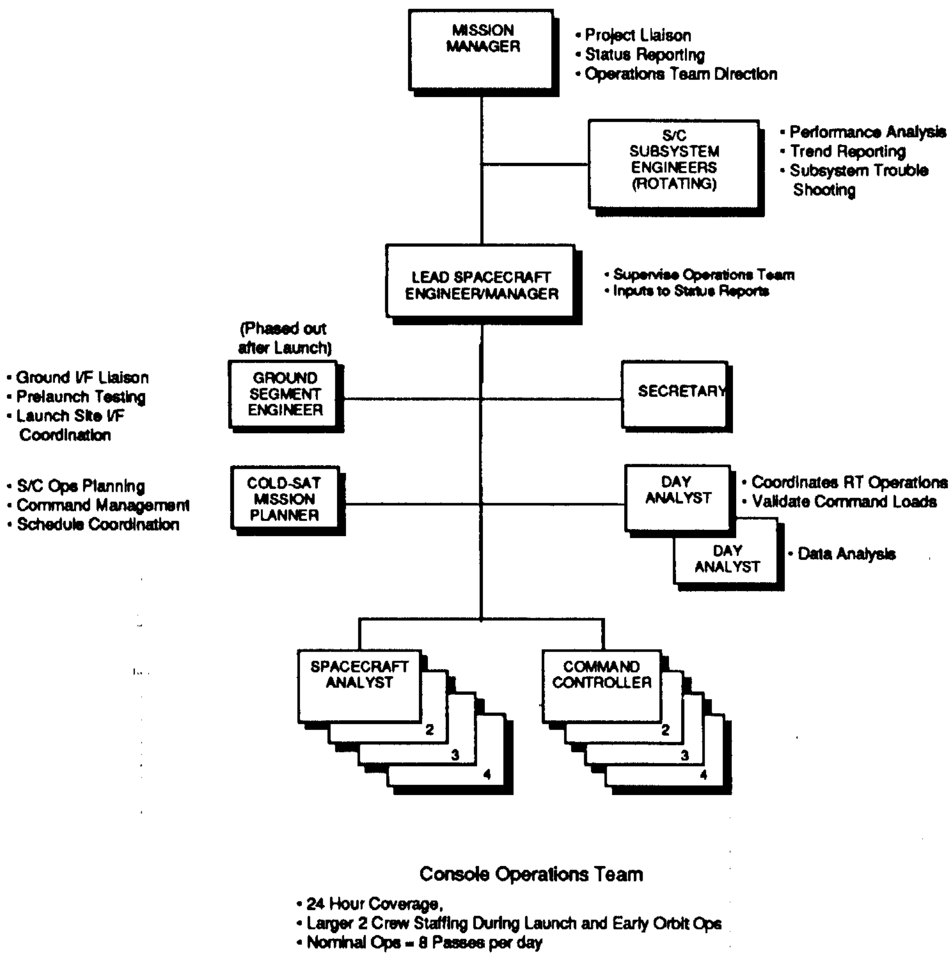
COLD-SAT Ground Systems Test Plan (GSTP), defines the tests to be conducted between all supporting facilities, interfaces, operations and products required to support the mission.

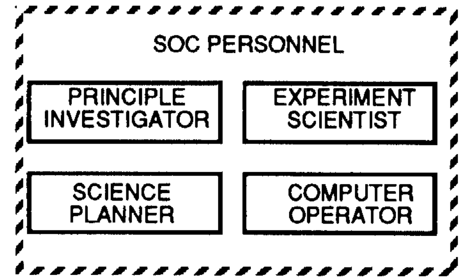
COLD-SAT Mission Operations Procedures (CMOP), defined the detailed and comprehensive command, control and operations procedures for each experiment, spacecraft and ground system.

These documents are developed, reviewed and tested by the COLD-SAT mission key members of the FOT prior to being used and validated in mission operations simulations. The entire FOT staff is brought on board no later than launch minus six months for testing, training, simulations and mission operations. The operations staff, and coverages at each facility is shown in **Figure 9-5**.

The mission FOT personnel will, as part of the MOP and CMOP, follow the COLD-SAT mission master schedule and insure all mission requirements are being met within the mission time table. **Figure 9-6** shows the nominal COLD-SAT mission schedule and activities required for mission success. The master schedule is used as a planning tool to evaluate the on going experiment operations over the entire life of the mission. The schedule will be updated as required as each experiment operation is completed or re-scheduled as needed.

The activities described above define the mission operations flow, documentation, FOT personnel and the COLD-SAT master operation schedule. They are all





STAFFING SUMMARY

- 4 persons in SOC
- Crew of 16 persons in POCC consisting of:
- 1 mission manager
- Rotating subsystem engineer (1 equivalent person)
- 1 lead engineer
- 1 ground system engineer (phased out post launch)
- 1 mission planner
- 2 day analysts (1 RT coordinator, 1 offline analysis)
- 4 shift crews (8 people) cover 24 hrs/day, 7 days/wk
- 1 secretary

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Figure 9-5. Mission operations staff.

DAYS OF MISSION

		0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39
	CLASS 1 EXPERIMENTS																																								
1	Pressure Control				X	X	X	X	X	X	X	X		X	X	X	X	X																							
2	Tank Chilldown											X		X								XX		XX					X		XX					X		X			
3	No Vent Fill & Refill											X		X								XX		XX					XX		XX					X		X			
4	Liquid Acq. Device Fill													X								X		X												X		X			
5	Quantity Gauging											X		X								X		X					X		X					X		X			
	CLASS 2 EXPERIMENTS																																								
8	Pressurization of Tankage											X		X			X												X		X					X		X	X		
9	Low G Outflow & Vented Fill																						X		X					X		X					X				
10	On Orbit LAD Performance																	X					X							X							X		X		
11	Transfer Line Chilldown											X		X								X		X					X		X										
12	Fluid Thermal State Control																																			X					
13	Fluid Dumping																																								
14	Evaluate Instrumentation			X	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-

DAYS OF MISSION

		40	41	42	43	44	45	46	47	48	49	50	51	52	53	54	55	56	57	58	59	60	61	62	63	64	65	66	67	68	69	70	71	72	73	74	75	76	77	78	79
	CLASS 1 EXPERIMENTS																																								
1	Pressure Control																															X	X	X	X	X	X	X	X	X	X
2	Tank Chilldown											X					X		X					X																	
3	No Vent Fill & Refill		X		X					X		X					X		X																						
4	Liquid Acq. Device Fill		X		X					X																															
5	Quantity Gauging		X		X					X		X					X		X					X														X			
	CLASS 2 EXPERIMENTS																																								
8	Pressurization of Tankage																							X																	
9	Low G Outflow & Vented Fill																							X		X	X				X	X									
10	On Orbit LAD Performance			X							X																														
11	Transfer Line Chilldown																																								
12	Fluid Thermal State Control																														X										
13	Fluid Dumping																																								X
14	Evaluate Instrumentation	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-

Figure 9-6. Master schedule.

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designed to take full advantage of nominal operations conducted at GSFC. The COLD-SAT detailed pre- and post-launch operations are described in the following sections.

9.2.1 Prelaunch Activities

Pre-launch activities prepare the COLD-SAT mission and its various supporting elements for operational support. One of the prime pre-launch activities is the development and production of COLD-SAT mission documentation which flows down requirements to the various support organizations.

In addition to the operations documents required by COLD-SAT for on orbit support, the FOT shall also support tests to assure compatibility between the ground and space segment. Pre-mission testing of the ground segment shall be undertaken in a step by step process to insure all ground systems (hardware and software) are properly tested and validated. Mission operations personnel also undergo training, test and simulation verification programs to insure mission readiness. The personnel and equipment exercises are not limited to the COLD-SAT FOT but shall include personnel and facilities from all supporting elements within the GSFC, the remote SOC, KSC, DSN and any other applicable supporting elements. Figure 9-7 shows a typical testing and simulation plan involving the various support elements within the COLD-SAT mission operations scenario.

In order to insure that the COLD-SAT space segment is compatible with the ground support elements the, COLD-SAT spacecraft will participate in compatibility tests with the TDRSS and other ground support elements. To accomplish these tests COLD-SAT will make use of the GSFC Compatibility Test Van (CTV), the CTV would be scheduled to support these tests at the BSSD Boulder facility between launch minus 12 to 3 months. GSFC will produce a compatibility test report in accordance with their standard policies to document the results of the CTV and TDRSS COLD-SAT tests. The CTV will also preform a DSN compatibility test with the COLD-SAT spacecraft. All results will be sent to LeRC.

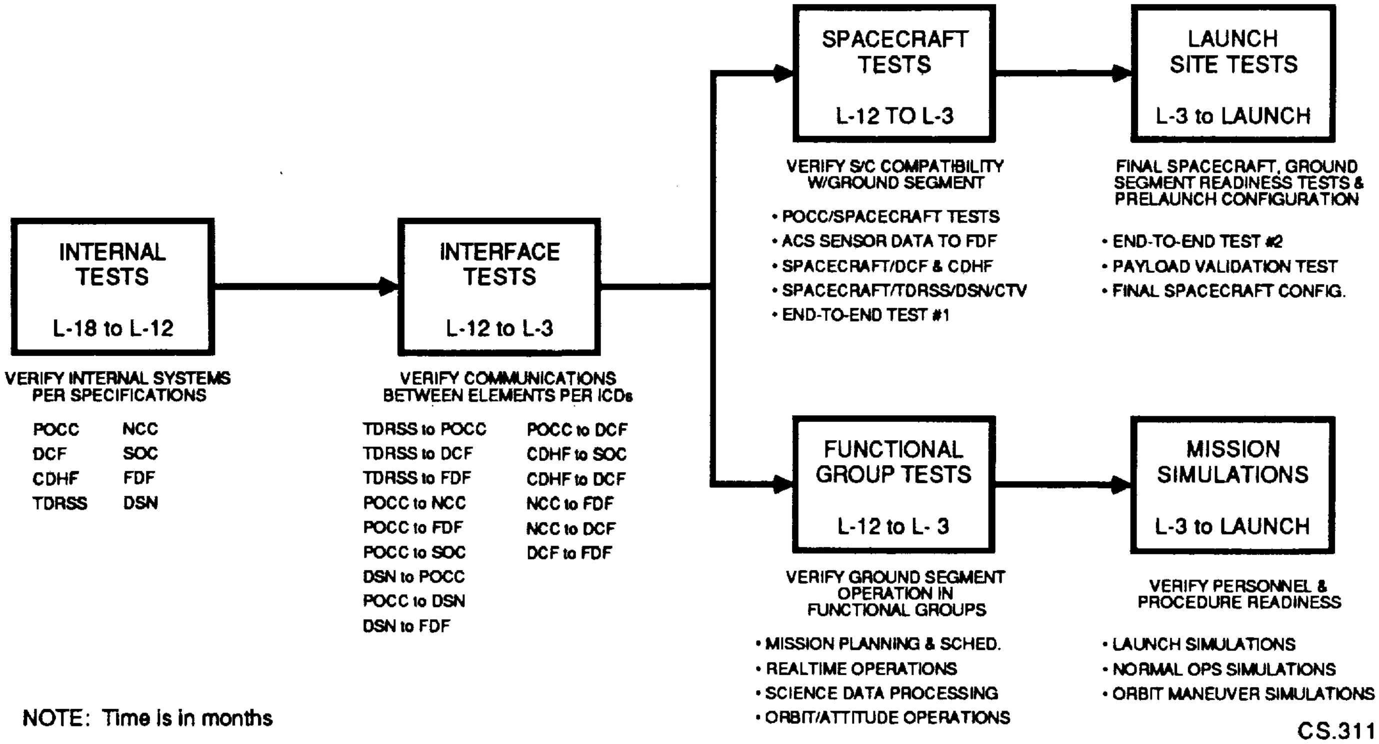

Figure 9-7. COLD-SAT ground segment test plan.

This approach is designed carry as much pre-launch experience as possible into the on-orbit phase and to provide for the early validation of the COLD-SAT mission operations plans, procedures, hardware, software, personnel and facility interfaces. Figure 9-8 defines the nominal COLD-SAT ground system operations test flow. This test flow shall be applicable to both the inter and intra testing as well as the final end-to-end tests. The final pre-launch simulations will encompass all flight elements required to support the COLD-SAT mission, once these simulations have been successfully completed, the COLD-SAT network configuration will be frozen for on-orbit support. All operations up to this point have been in preparation of the actual mission operations or on-orbit support.

9.2.2 On-Orbit Operations

The on-orbit operations commence with the lift-off from KSC and the separation of the COLD-SAT spacecraft from the launch vehicle. The launch vehicle will put the COLD-SAT in nearly the correct attitude, the spacecraft will automatically correct the spacecraft's attitude. The CPOCC will monitor the

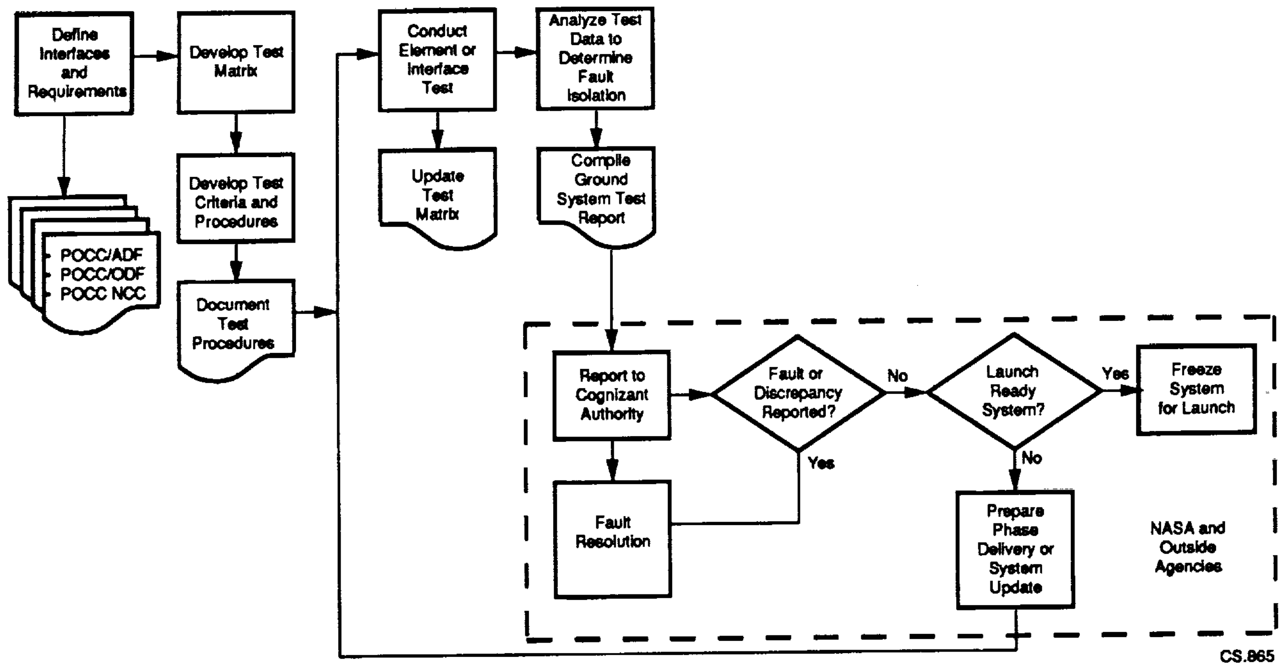

Figure 9-8. COLD-SAT ground system test flow.

corrections via the TDRSS Multiple Access return link (MA) to assure these corrections are made; the last correction will be made 50 minutes after spacecraft injection. After nominal operation and attitude is confirmed by the FOT, initialization of the other spacecraft subsystems and experiments will begin. **Table 9-1** shows the support requirements for each COLD-SAT facility.

After the initial operations phase has been completed, the FOT will begin the mission planning process which will determine the required support for each days mission support activities. The nominal COLD-SAT master support schedule as shown in **Figure 9-6** will now be modified based upon this planning process. The planning process, as seen in **Figure 9-9**, is a 5 step process is defined as follows.

- Step 1: Assesses the TDRSS and ground segment scheduling requirements as defined in **Table 9-1**.

- Step 2: Assess the experiment profile for the entire mission, as defined in the COLD-SAT master schedule, **Figure 9-6**.

Table 9-1

REQUIREMENTS FOR POST-LAUNCH GSFC SUPPORT

SYSTEM OR FACILITY REQUIREMENTS	IMPACT ASSESSMENT
O&M REQUIREMENTS (MSOCC-I): • Computer operations requirements • Data operations control requirements • scheduling requirements	• Fifteen 10-minute pass supports per day; • No special requirements • MSOCC-I/NCC/TDRSS interface; no special requirements
TRACKING DATA RELAY SATELLITE SYSTEM (TDRSS): • While Sands Ground Terminal interface requirements • White Sands NASA Ground Terminal interface requirements • Telemetry and command requirements • Tracking data requirements	• Standard support services; no special requirements • Standard support services; no special requirements SMA supports per day; integral total 150 minutes command support • Nominal four to six ranging passes per day - 5 minutes each
DEEP SPACE NETWORK (DSN) • Compatibility requirements for telemetry, command, and tracking data	• Backup and emergency support only
LEWIS RESEARCH CENTER (LeRC) • Science data requirements • Data evaluation requirements • Instrument operation requirements	• Production data requirements for IPD • Definition of POCC science data and instrument evaluation and criteria • Definition of experiments desired operation modes for each instrument
PROJECT OPERATIONS CONTROL CENTER (POCC): • Data base requirements • Standard software requirements • COLD-SAT applications software requirements • Hardware requirements • Simulator requirements • Processing requirements • Personnel requirements	• MSOCC-I • MSOCC-I • MSOCC-I • Nominal MSOCC-I; no special requirement • Nominal MSOCC-I; no special requirement • Passes per day/ 150 minutes per day offline support
ATTITUDE DETERMINATION FACILITY (ADF): • Data formats and transfer mediums • Frequency and quantity of attitude data processing • Attitude-related command generation • Definitive attitude calculations for final data production	• Defined for POCC software pre-launch • Nominal three-pass evaluations per week • One-time only bias commands to compensate for residual magnetic dipoles • As required by SOC (Le RC)
FLIGHT DYNAMICS SYSTEM: • Maneuver computations and commands	• Standard support services; no special requirements
ORBIT DETERMINATION FACILITY (ODF): • Tracking data requirements • Scheduling assistance data (scheduling matrix) • Attitude-related data	• Nominal for to six ranging passes per day • Generation of normal GSFC mission scheduling aids • As required to provide definitive attitude data to experimenters
NASA COMMUNICATIONS (NASCOM): • Data capture requirements • Data criteria • Data accountability requirements	• Both nominal GSFC support - 15 passes per day
INFORMATION PROCESSING DIVISION (IPD)/TELEMETRY ON-LINE PROCESSING SYSTEM (TELOPS): • Data capture requirements • Data criteria • Data processing requirements • Data accountability requirements	• Fifteen 50 kbps tape dumps per day • 95% of all mandatory science data • Production data per experimenter requirements • TBD
MISSION PLANNING SYSTEM (MPS): • Scheduling requirements	• Generation of weekly COLD-SAT TDRSS support schedules from POCC generic requirements
NETWORK CONTROL CENTER (MCC): • Telemetry and command service scheduling requirements • Data quality requirements and reports • Tracking data services	• Nominal GSFC support services; no special requirements • Nominal GSFC support services; no special requirements • Nominal GSFC support services; no special requirements

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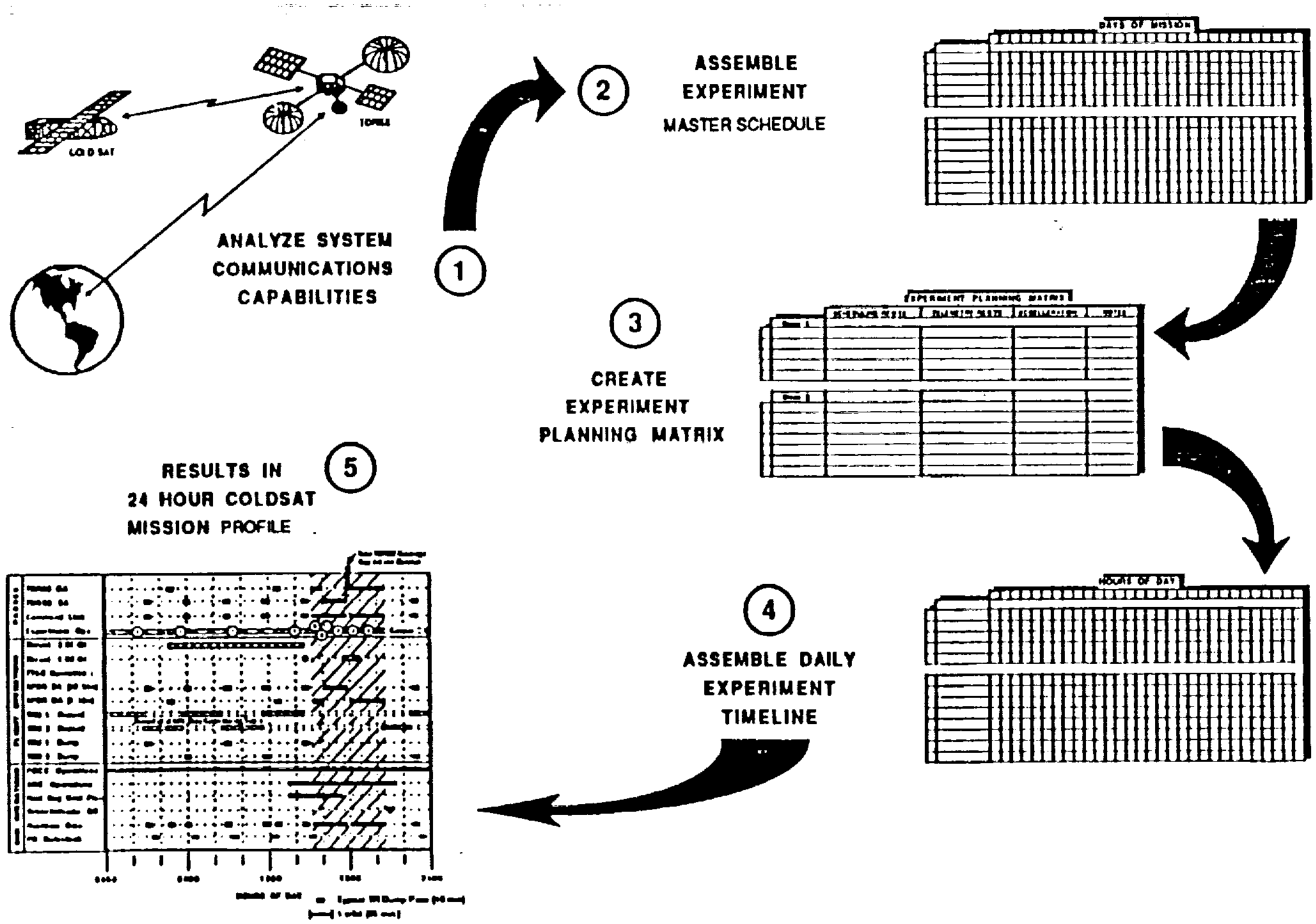

Figure 9-9. Mission planning process analyzes flight and ground segment resources.

- Step 3: Create an experiment planning matrix, consolidate the operational requirements for each class of experiments.

- Step 4: Develop a detailed experiment timeline from the master schedule.

- Step 5: Produce a 24 hour mission profile that is derived from the planning matrix and experiment timeline.

The 5 step process culminates with the 24 hour operations profile. An example of this process is shown in **Figure 9-10,** were the aggregate requirements for all elements are formulated into a 24 hour mission profile. From this profile, detailed plans, mission activities, command loads, realtime and playback activities via TDRSS and associated support elements can be effectively set.

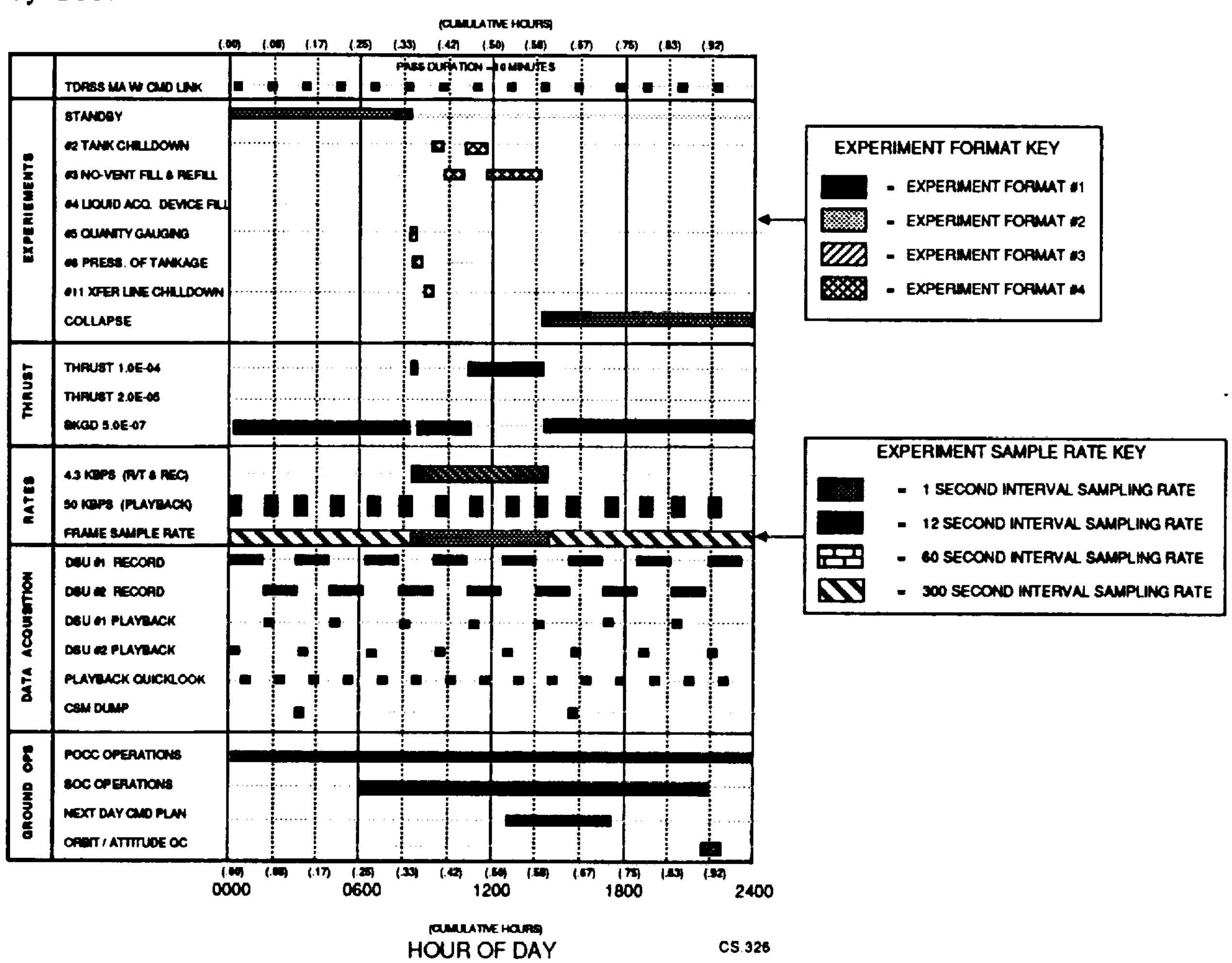

Figure 9-10. 24 Hour COLD-SAT mission profile - day 20.

The final phase of operations that will be addressed is mission contingency operations. During the pre-launch time frame the FOT will develop and incorporate into the mission plans and procedures specific contingency operations. An example of these pre-planned contingency operations are defined in **Table 9-2**. While these plans define specific cases other non-nominal performances by any other space or ground system will be handled in realtime within the CPOCC with corrective action recommended by the FOT.

Table 9-2

CONTINGENCY PLAN

EVENT	FAILURE DETECTION	AUTONOMOUS ACTION	POCC ACTION
Excessive LV Tip-off Attitude/Rate	Horizon sensor reads <20°	Go to spin mode, UV & inhibit array dep.	Commands reacquisition
ACS Gyro Failure	Valve on Time >1 Seconds (TCP)	Go to Spin Mode and Experiment Standby	Command Reacquisition Use Redundant Gyro
Horizon Sensor Failure	Valve on Time >1 Seconds (TCP)	Go to Spin Mode and Experiment Standby	Command Reacquisition Reprogram for 2 Axis Sun Sensor Plus Mag
Sun Sensor Failure	Valve on Time >1 Seconds (TCP)	Go to Spin Mode and Experiment Standby	Command Reacquire Reprogram for Horizon Sensor Plus Mag
ACP Failure	Bad data detected (TCP)	Go to Spin Mode and Experiment Standby	Mission Over Open loop to GG mode?
Propulsion RCS Thruster Failure (off)	Accelerometer readings (ground analysis)	None	None (increase in LH 2 disturbance)
Induced g Thruster Failure (off)	Accelerometer readings (ground analysis)	None	None (Loose 1 level of acceleration)
Leaky Pressure Regulator	Pressure monitor (ground analysis)	Leak overboard	Switch to redundant unit
TT&C TCP Failure	No telemetry (ground analysis)	None	Switch to redundant unit
Prime Forward Link Failure	No telemetry (ground analysis)	None	Switch to BU link (GSTDN)
Transponder Failure (receiver of transmitter)	No telemetry (ground analysis)	None	Mission over
Data Storage Unit Failure	No playback data (grond analysis)	None	Switch out bad sections of memory
EPS Array Deployment Failure	Low power (ground analysis)	None	Extend mission (74% normal power)
Battery Failure	Low end of night voltage (ground analysis)	None	Use one battery
Thermal Paraffin Actuator Faiulure	Low N_2H_2 temperature (ground analysis)	None	Turn on heaters
Propulsion Heater Failure	Low valve or line temp- erature (ground analysis)	None	Use redundant valve heaters; lines above freezing
Experiment Subsystem ECP Failure	Failure detection system (TCP)	Command BU ECP and execute standby list (ROM)	Trouble shoot primary ECP computer

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Section 10
SAFETY

This section discusses the safety issues for COLD-SAT. Many of the issues are commonly dealt with on spacecraft (high pressure tanks and hydrazine) but cryogenic fluids and hydrogen are unusual for spacecraft. This section deals mostly with the unique issues of COLD-SAT.

The COLD-SAT hardware and procedures will be designed to meet the requirements of:

- ESMCR 127-1, Range Safety Manual
- AFM 161-30 Volume II, Chapter 12, Liquid Propellants
- NFPA 50A and 50B, National Fire Protection Association Standard

Loading of high pressure tanks will be done in two stages to reduce exposure: they will be loaded to 13.8 Mpa (2000 psia) (factor of 4 to burst) at day T-5 then brought to 27.6 Mpa (4000 psia) at L-6 hours. The pressurization will be done remotely with personnel only needed to connect and disconnect lines before and after pressurization.

Liquid hydrogen loading will also be done in two stages. The supply tank will be cooled using liquid helium starting at day T-5 to stabilize the insulation system. A remote loading system will be used at T-3 hours to remove any residual helium and load the tank with liquid hydrogen. A vent stack out the top of the umbilical tower will assure no buildup of hydrogen gas inside the payload room. At lift-off, the loading system will automatically disconnect. This late disconnect approach allows the tank to be vented or even unloaded remotely should a launch hold be required.

In addition to the safeguards built into the system discussed above, remote TV and hydrogen sniffers will be located at strategic points to detect any problems.

Table 10-1 shows the hazards associated with the experiment subsystem along with controls to reduce the risk. Any release of hydrogen into the payload room would cause a serious explosion hazard. Leak testing with an inert gas and monitoring are critical to achieving an acceptable risk.

Liquid hydrogen boils at 20 K thus can instantly freeze skin. Even lines or connectors are a frostbite hazard. Protective equipment and good training are essential.

Both helium and hydrogen are colorless, odorless gases. Breathing either gas in excessive concentrations can quickly cause unconsciousness. Personnel must be trained to appreciate the dangers and employ suitable monitoring equipment. A complete list of COLD-SAT hazards, their effects and controls are given in **Table 10-2**.

Materials compatibility with hydrogen is somewhat unusual in spacecraft design and **Table 10-3** identifies the major issues. Some materials become brittle when cold and should not be used, particularly for structure. Materials subject to hydrogen embrittlement become brittle with time and should be avoided. Embrittlement might not be evident during testing but failures could arise later in the mission. Only the solar array, horizon sensor and sun sensor are susceptible to condensables. Normal spacecraft design practices will provide adequate contamination safeguards for this short mission.

Table 10-1

PRELIMINARY EXPERIMENT SUBSYSTEM HAZARDS ANALYSIS

HAZARD: FIRE AND/OR EXPLOSION

- PHASE: Test, Pre-launch, and Launch Operations
- CAUSE: Leak or uncontrolled release of LH_2 or GH_2 into surrounding atmosphere
- CONTROLS: System designed and tested to prevent leaks
  System and facilities designed to safely vent H_2 boiloff
  Procedural controls on operations with LH_2
  Monitor air for presence of H_2

  REMARK: Handling of LH_2 requires facilities specifically designed for such operations

HAZARD: HIGH PRESSURE SYSTEM POSE EXPLOSION HAZARD
GH_2 STORAGE TANKS, GSE AND PRESSURIZATION OPERATIONS
LH_2 STORAGE DEWAR

- PHASE: System and component test, pre-launch
- CAUSE: Excessive pressure build-up, inadequate system design, operational error
- CONTROLS: System design with appropriate safety factors
  System test to verify design
  Passive pressure relief designed into systems to prevent excessive pressures
  Operations controlled by detailed procedures

HAZARD: CRYOGENIC OPERATIONS POSE PERSONNEL INJURY HAZARD

- PHASE: Test, Pre-launch Operations
- CAUSE: Contact with cryogenic materials or system components cause frostbite
- CONTROLS: Protective equipment used during cryogenic operations
  All surfaces exposed to the cryogen will be insulated or shielded against human contact
  Procedural controls including training of operators established for cryogenic operations

HAZARD: MATERIALS POSE PERSONNEL HAZARDS:

GHe - Asphyxiant
GH_2 - Asphyxiant
Solvents and Cleaning Agents - Irritants, Toxic, Flammable

- PHASE: Fabrications, Test, Pre-launch Operations
- CAUSE: Uncontrolled exposure to hazardous materials
- CONTROLS: Use of personal protective equipment
  Operations controlled by detailed procedures
  Limitation on quantities of materials used
  Environmental monitoring to detect hazards

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Table 10-2

PRELIMINARY COLD-SAT HAZARDS ANALYSIS

ITEM	OPERATIONAL PHASE	HAZARD	HAZARD CAUSE(S)	HAZARD CONTROLS	REMARKS
1	Test, pre-launch, & launch operations	Fire and/or explosion	Leak or release of N_2H_4 into surrounding atmosphere	System design to prevent leaks. System leak tests. System design to prevent inadvertent release Procedural controls on operations with N_2H_4	Design driver on flight hardware, ground support equipment, facilities, and operation. Procedural controls include detailed training of operators working with N_2H_4
2	Test, pre-launch, & launch operations	Fire and/or explosion	Leak or uncontrolled release of LH_2 or GH_2 into surrounding atmosphere	System design to prevent leaks. System leak tests System and facilities designed to safely vent H_2 boiloff Procedural controls on operations with LH_2 Monitor air for presence of H_2	Handling of LH_2 requires facilities specifically designed for such operations Remote load at L-3 hours
3	Test, pre-launch operations	Pressurized system explodes releasing shrapnel or hazardous materials	Inadequate design for worst case operating conditions Rapid pressure buildup due to thermal or other conditions	System design for maximum expected pressures with positive safety margins Systems designed with pressure relief capability to control pressure buildup	GHe tanks designed with 4:1 factor of safety
4	Test, pre-launch operations	Battery explodes releasing shrapnel or hazardous materials	Excessive pressure due to excessive charge or discharge rate	Battery charge controls designed to prevent excessive charge rate Battery circuits protected against excessive discharge rates	
5	Test, pre-launch operations	Electrical circuits cause fire	Electrical circuit faults cause arcs, sparks, or high temperatures which ignite flammable materials	Electrical circuits designed with fault protection to prevent ignition of surrounding materials Flammable materials separated from potential ignition sources	Requires special consideration for electrical equipment used during operations with N_2H_4 and LH_2

Table 10-2

PRELIMINARY COLD-SAT HAZARDS ANALYSIS (continued)

ITEM	OPERATIONAL PHASE	HAZARD	HAZARD CAUSE(S)	HAZARD CONTROLS	REMARKS
6	Test, pre-launch operations	Cryogenic operations pose personnel injury hazard	Contact with cryogenic materials or system components cause frostbite	Personal protective equipment used during cryogenic operations Procedural controls including training of operators established for cryogenic operations	
7	Fabrication, Test, pre-launch operations	Materials pose personnel hazards: N_2H_4 - Toxic GN_2 - Asphyxiant GH_2 - Asphyxiant GHe - Asphyxiant LH_2 - Frostbite Solvents & cleaning agents -irritants, toxic, flammable	Uncontrolled exposure to hazardous materials	Use of personal protective equipment Operations controlled by detailed procedures Limitation on quantities of materials used Environmental monitoring to detect hazards	Special personal protective equipment, procedure, and training required for N_2H_4 operations Special facilities, procedures, and training required for LH_2 and GH_2 operations
8	Test, pre-launch operations	Electrical equipment poses personnel shock hazard	Exposed electrical circuits or potential differences between equipment result in personnel contact	Equipment designed to prevent personnel contact with hazardous voltages External surfaces of electrical equipment tied to common ground potential	
9	Test, pre-launch operations	Premature operation of ordnance causes injury or damage or initiates hazardous events	Inadvertent operation caused by operator error or circuit failure Electrostatic or electro-magnetic energy initiates ordnance Stray voltages	Operating systems and procedures prevent premature operation Firing circuits designed with multiple inhibits System design incorporates shielding, grounding, and ESD sensitive initiators System connection procedures require test for stray voltages prior to ordnance connection	

Table 10-2
PRELIMINARY COLD-SAT HAZARDS ANALYSIS (concluded)

ITEM	OPERATIONAL PHASE	HAZARD	HAZARD CAUSE(S)	HAZARD CONTROLS	REMARKS
10	Fabrication, test, pre-launch operations	Lifting and handling of equipment causes injury or damage to equipment or facilities	Improper or inadequate equipment used for lifting and handling Procedural errors cause mishandling	Lifting and handling equipment designed to provide adequate margins of safety Detailed procedures developed for handling operations	
11	Test, pre-launch operations	Personnel exposed to hazardous levels of non-ionizing radiation	RF radiation in excess of permissible limits during system test	Hazardous radiation area identified and exposure barriers established	
12	Test, launch operations	Structural failure of flight equipment causes injury or equipment damage	Structural design inadequate for load conditions during test or launch	Structural design provides adequate margins of safety for all load conditions	

Table 10-3

MATERIAL COMPATIBILITY REQUIREMENTS

COMPATIBILITY ISSUES	REQUIREMENTS/COMMENTS
Cryogenic Temperature	Avoid materials which undergo significant martensitic transformation at low temperatures (aluminum, most 200 & 300 series SS are acceptable)
Flammability	H_2 will not react with most structural materials. Some plastics act as oxidizers
Hydrogen Embrittlement	Use aluminum alloys, copper, austenitic stainless steels: A-286, 304, 310, 316
Outgassing/Contamination	Total mass loss (TML) <1% and volatile condensable materials (VCM) <0.1% per JSC SP-R-0022A (this is a nominal baseline, real requirements depend upon the functional requirements of the system)

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Section 11
RELIABILITY

11.1 REQUIREMENTS

The reliability requirements for COLD-SAT are to complete the Class I experiments with a probability of 0.92 assuming the launch vehicle, ground segment and TDRSS have a reliability of 1.0. The addition of redundancy was primarily governed by this requirement, however, single point failures were eliminated where the cost was minimal and the experiment data was enhanced. This covers additional temperature sensors, valves and flow meters so the loss of some of these components would not appreciably degrade the mission. A second battery was also added to the baseline requirement for single point failure protection.

11.2 RELIABILITY ANALYSIS

The requirement for a reliability of 0.92 drove the amount of redundancy in the COLD-SAT design. The Boeing reliability optimization program was used to find the minimum cost way to reach the reliability requirement. Inputs were component reliability, component cost and definition of the single thread spacecraft. The component cost included component manufacturing and test costs, added integration costs (both analytical and physical) and added spacecraft test costs. The program then produced the curve shown in **Figure 11-1**. The curve shows the differential reliability improvement and cost for each component. The program selects the highest differential reliability improvement per unit cost. The first such improvement identified for COLD-SAT was the redundant heaters and controllers for the hydrazine tanks and lines. Next was the IRU (in spite of its high cost) because of the high reliability improvement. The addition of a redundant TT&C and Experiment processor provided sufficient redundancy to meet the reliability requirement. Adding a redundant transponder and ACS processor would be the lowest cost way to improve reliability further.

Figure 11-2 summarizes the spacecraft reliability analysis. The reliability of the actual design is slightly higher than that shown in **Figure 11-1** because a detailed analysis of the experiment plumbing with assumed work-

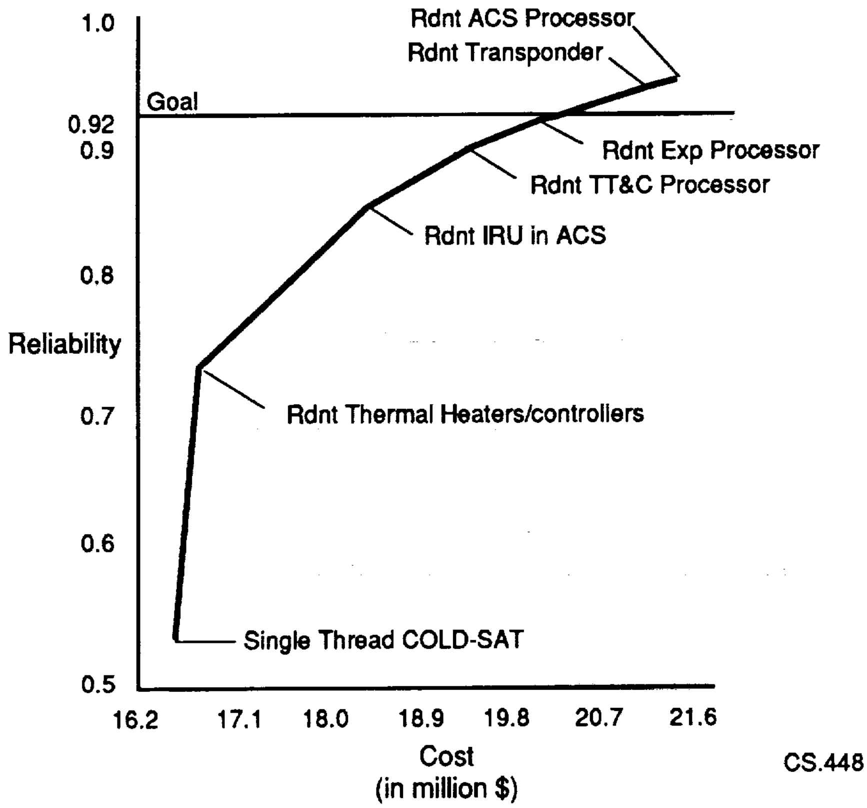
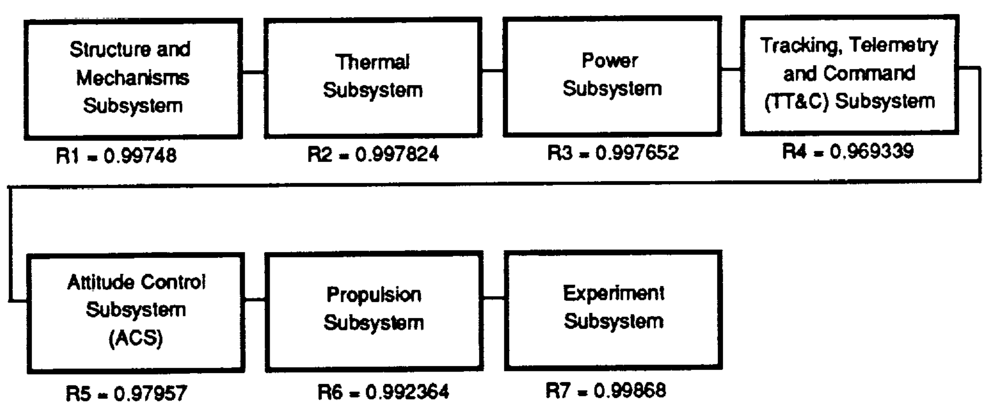


Figure 11-1. COLD-SAT reliability achieved through minimum cost redundancy.



$$(R_1)(R_2)(R_3)(R_4)(R_5)(R_6)(R_7) = 0.93443$$

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Figure 11-2. System reliability diagram.

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arounds, resulted in a higher reliability than assumed in the optimization program.

The diagrams of **11-3** through **11-8** derive the reliability of each bus subsystem. When a number of identical units are needed for mission success such as the array lock/release mechanism of **Figure 11-3**, the number of series elements are indicated above the break in the line. The reliability of the element is raised to a power equal to the number of units to get the combined reliability of the function.

Each box indicates the failure rate (λ(ML) = rate during launch and λ(SF) = rate during orbit operation) and the failure rate source reference "(X)" found in **Table 11-1**. As shown for the power control unit in **Figure 11-5**, 2 of 3 units of the function must work. The reliability of this combination is computed from the binominal theorem i.e., the reliability is the sum of the probability of 3 working and 2 working. The failure rates assume a grade II parts program.

Experiment subsystem reliability analysis was determined by defining required valve actuations for each Class I experiment. An example experiment reliability diagram for tank chilldown is shown in **Figure 11-9**. Each experiment was modeled schematically showing primary (indicated by arrows) and alternate hydrogen flow paths. The valve failure rate (5×10^{-5}) was calculated by inverting the cycle life (1/20,000). Reliability for Experiment 2.0 (see **Figure 11-9**) was calculated for each segment:

R_1: parallel supply-tank cold valves (1 of 2)

R_2: flow control orifice paths (2 of 4)

R_3: radial, axial, tangential sprays (2 of 4)

R_4: vent path

Individual valve segment reliabilities were determined from the appropriate binominal expressions (1 of 2 or 2 of 4) and the required number of cycles.

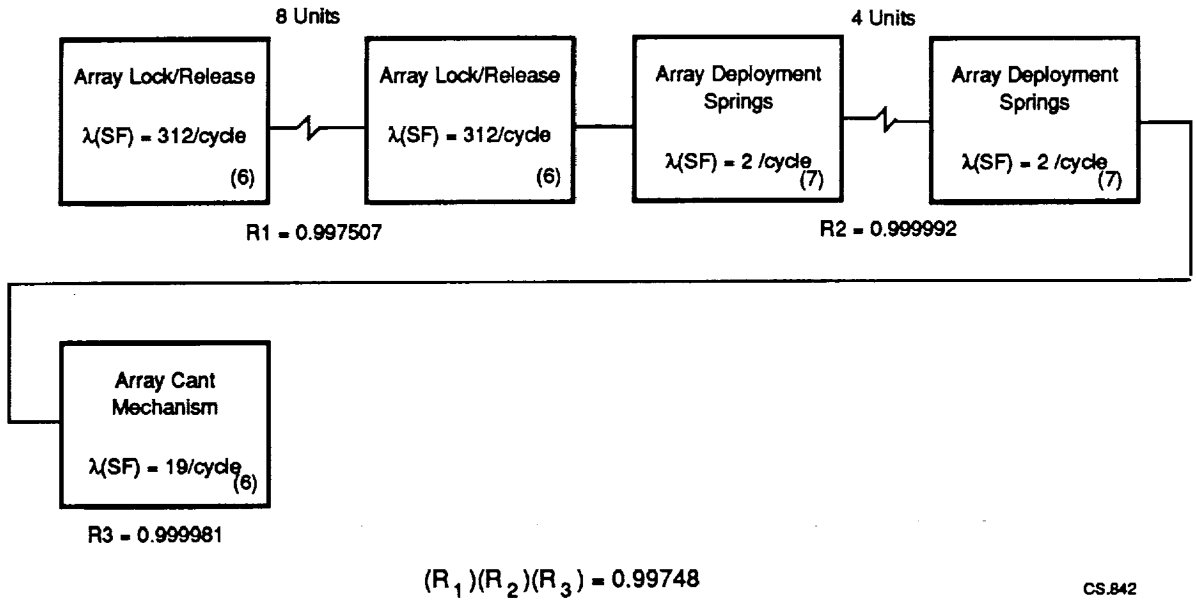

Figure 11-3. Structures and mechanisms subsystem.

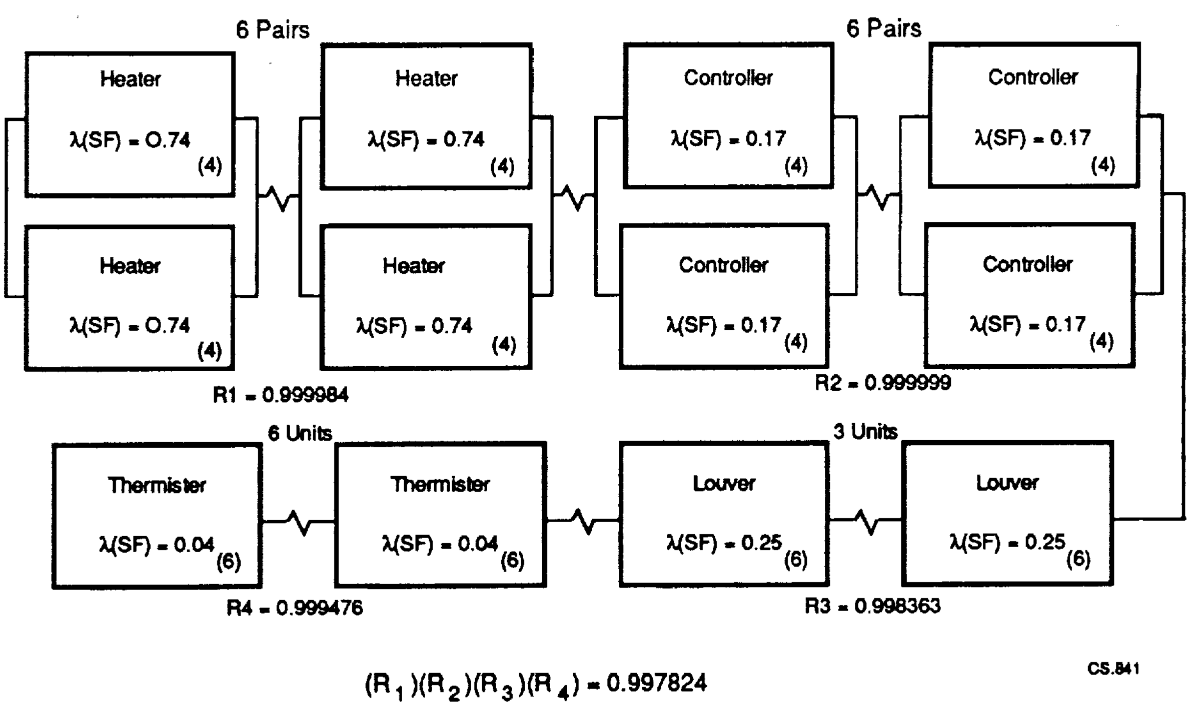

Figure 11-4. Thermal subsystem.

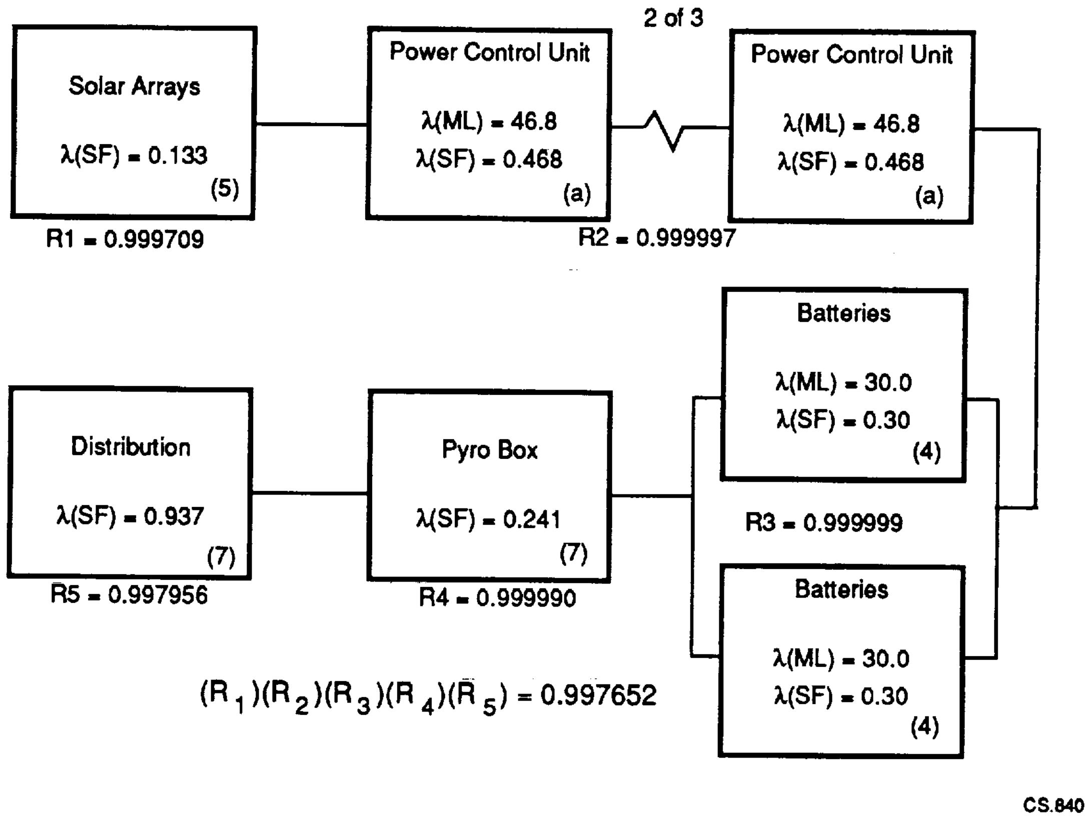

Figure 11-5. Power subsystem.

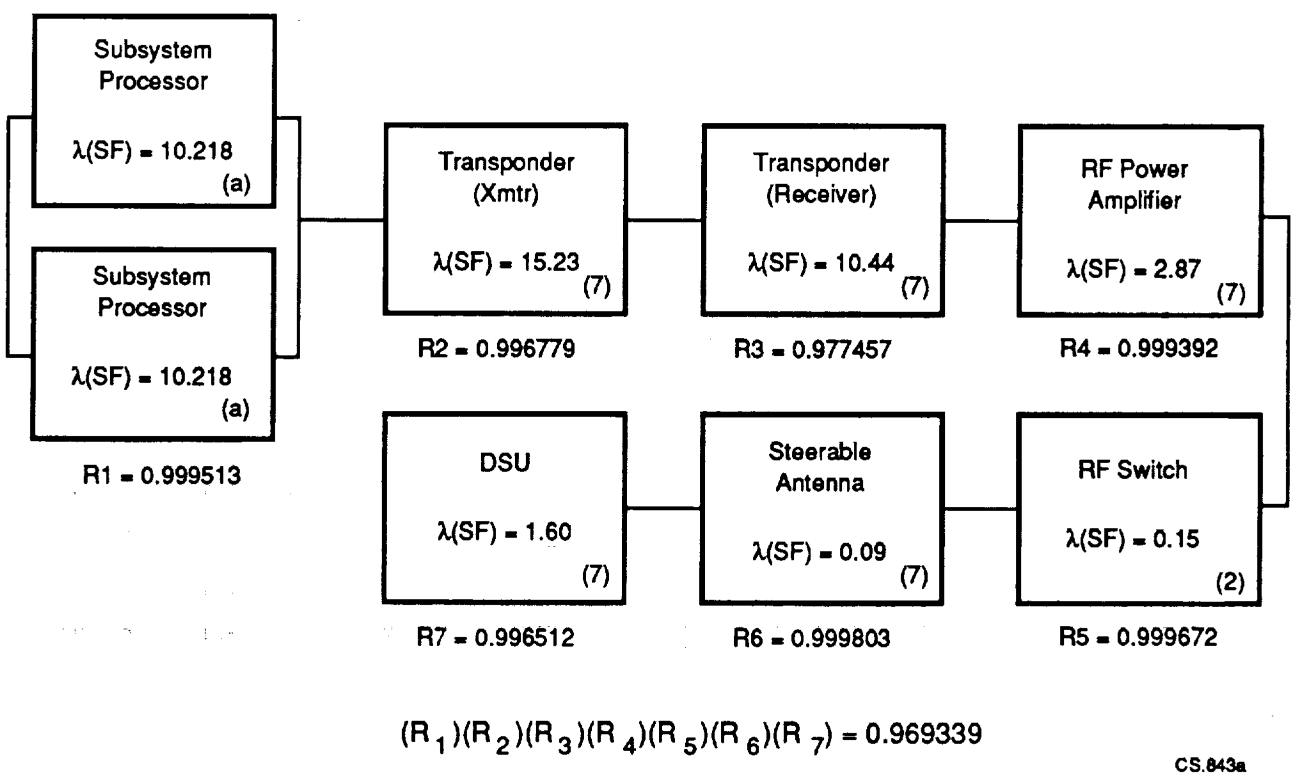

Figure 11-6. TT&C subsystem.

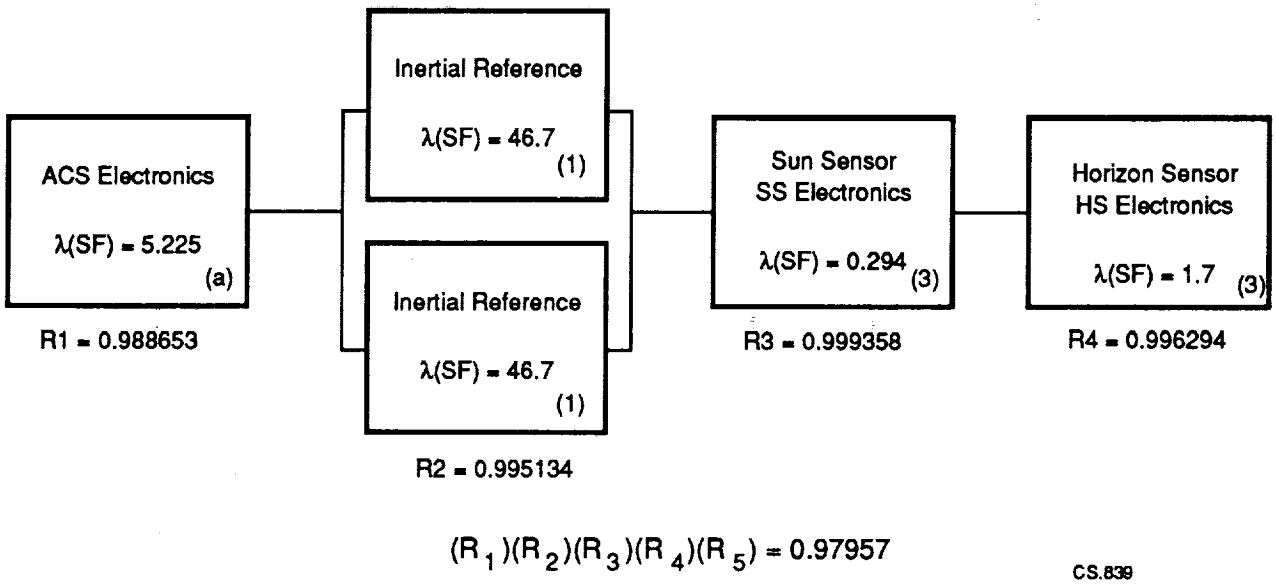

Figure 11-7. Attitude control subsystem (ACS).

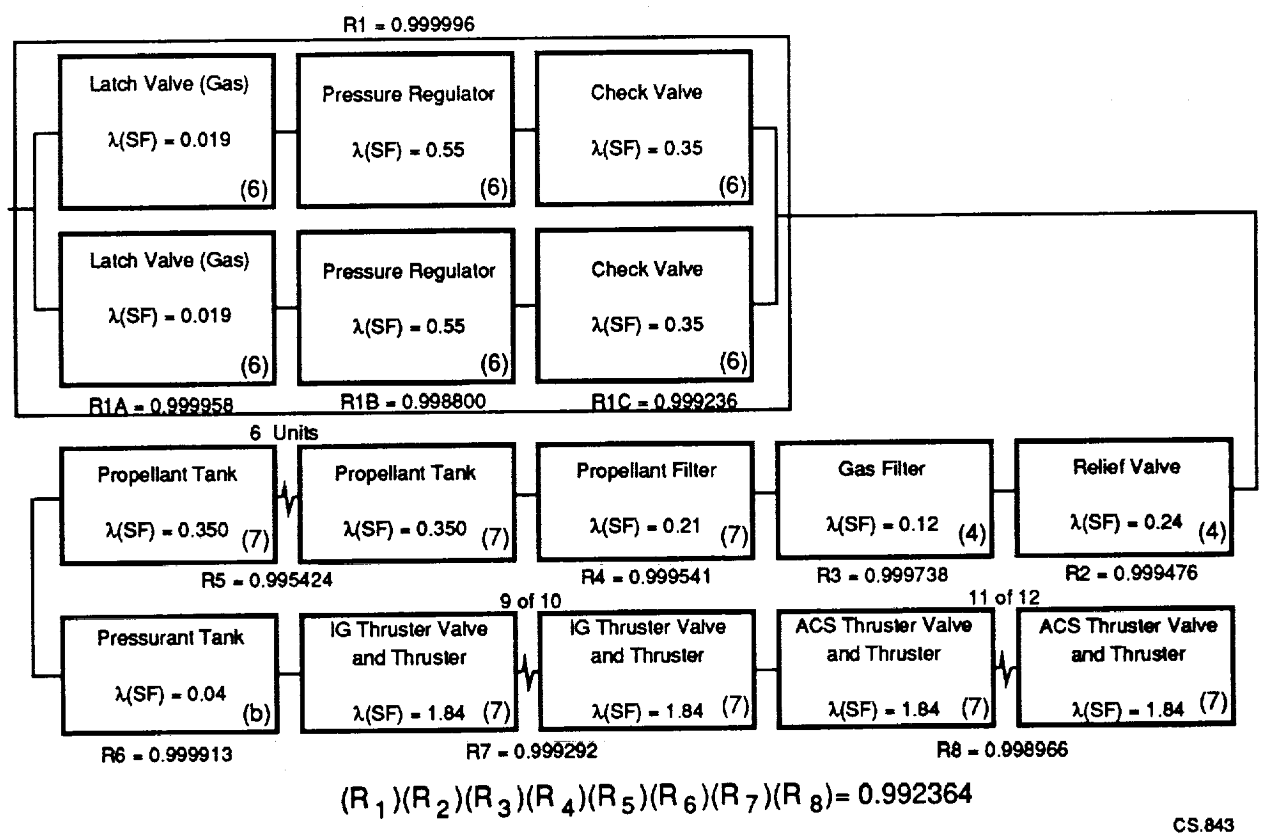

Figure 11-8. Propulsion subsystem.

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Table 11-1

FAILURE RATE REFERENCES SOURCES

[a] Synthesized from failure rates for an Inertial Upper Stage (IUS) card [7]

[1] Ball/Northrop Engineering Data/Analysis: 3-Axis Gyro Package for ERBS 1979/82/89 (Personal communication from W. Follett, Ball Aerospace Division, 01/27/89)

[2] MIL-HDBK-217E, Reliability Prediction of Electronic Equipment, Department of Defense, 1986

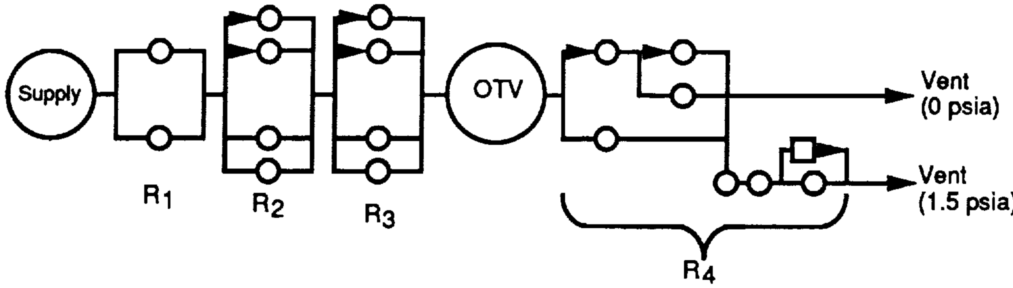
[3] GIDEP Access #F063-0210, "SLD 5-Eye Sun Sensor Stress Analysis, Reliability Prediction, and Transistor Worst Case Gain Analysis," 1986

[4] NPRD-3, Nonelectronic Parts Reliability Data, Reliability Analysis Center, RADC, Griffiss AFB, New York 1985

[5] GIDEP Access #F038-1421, "Electrical Power and Distribution (EP&D) Subsystem Reliability, "General Electric Company, 1982

[6] YVAE-80-005, System Effectiveness Requirements Document for Space Transportation System: Inertial Upper Stage (IUS), USAF/Space Division, 1981

[7] Document D290-10404-1, Reliability & Maintainability Allocations, Assessments and Analysis Report - IUS System, CDRL #050A2, Boeing Company/Aerospace Division, Seattle, WA 1979



- $R_1 = [1 - (FR)^2]^N$ where: Failure rate, $FR = \dfrac{1}{\text{cycle life}} = \dfrac{1}{20{,}000}$

  $R_1 = 0.9999998$ $N = (\text{No. cycles/chilldown})(\text{No. of chilldowns}) = (7)(12) = 84$

- Reliability OTV, $R_{OTV} = 0.999867$

- Reliability Depot, $R_{DEP} = 0.999952$

$$\boxed{\text{Experiment 2.0 Reliability, } R2 = (R_{OTV})(R_{depot}) = 0.999819}$$

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Figure 11-9. Reliability analysis, Exp. 2.0 Childown.

Reliability for the chilldown experiment is the product of R_1, R_2, R_3, and R_4. Similar procedures were used to calculate all Class I experiment reliabilities.

Overall reliability for the experiment subsystem was calculated by multiplying experiment reliabilities, instrumentation reliability, and reliability of the control processor (see Figure 11-10). To conduct the Class I experiments in 90 days, the experiment subsystem reliability was 0.99868.

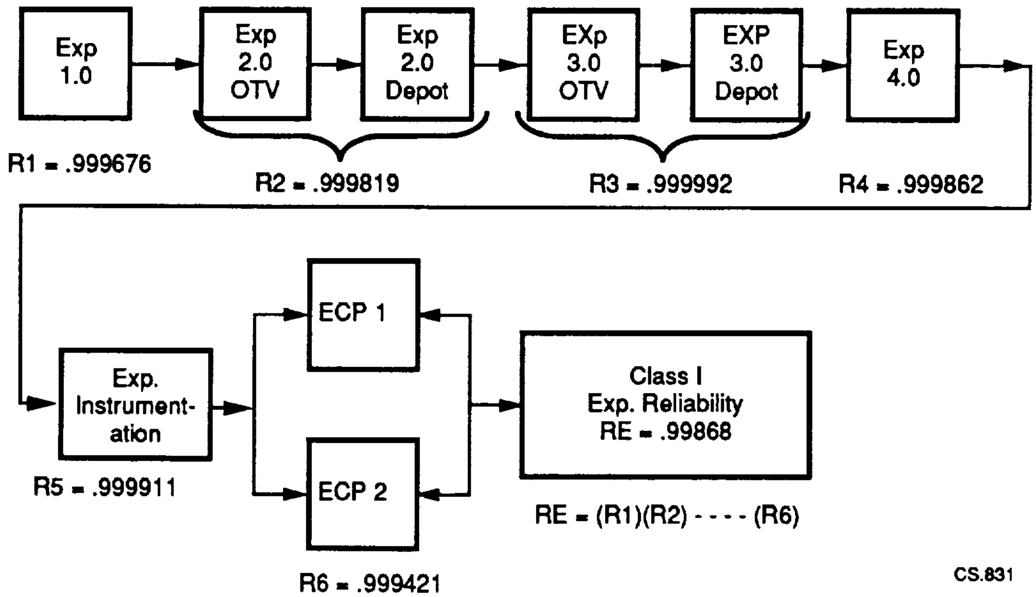

Figure 11-10. Experiment subsystem reliability.

Section 12
PROJECT PLANNING

This section addresses the programmatic issues associated with the implementation of the COLD-SAT program. These issues include the technological risks involved with project implementation, unique personnel and facility resources that may be required, system testing philosophy and overall project schedule.

12.1 TECHNOLOGICAL RISK

An assessment of the state of technology for the various elements of our COLD-SAT design was made and assigned risk categories A through D:

A. Those elements for which existing hardware or qualified designs may be used.

B. Those elements which require new designs but for which existing, proven design techniques are available.

C. Those elements which require new designs and are at or near the state-of-the-art for the technical discipline involved.

D. Those elements which require new designs which are beyond the current state-of-the-art.

A numerical risk factor between 0 and 10 was also assigned. Zero indicates risk free, ten indicates the highest degree of risk, requiring major breakthrough for accomplishment. Correlation between the numerical risk factor and risk category are as follows:

Category A:	0, 1
Category B:	2, 3, 4
Category C:	5, 6, 7
Category D:	8, 9, 10

Table 12-1 outlines the technological risks associated with the Experiment Subsystem. As indicated in the table, there are no components identified as Category C or D. Note that the liquid vapor sensor is a Category A as its design and operation are well developed; however, its performance under COLD-SAT conditions is unknown.

The pressurization subsystem operates in the ambient temperature range and uses components similar to many pressurization systems which have flown; therefore, it is generally a Category A subsystem.

The LH_2 distribution subsystem is a Category B; there exists extensive LH_2 technology and a number of cryogenic systems have flown, but none similar to COLD-SAT.

Table 12-2 outlines the technological risk associated with the spacecraft bus. Most of the spacecraft bus components have existing and qualified hardware designs, significant flight history and have been used in the past and on-going BASG spacecraft programs. Those components that do not have existing qualified hardware designs have proven design techniques used repeatedly in past programs, thus all bus components fall within categories A and B. The bus structure is unique to COLD-SAT and is therefore classified as category B. Similarly, the power subsystem is a category B. The batteries, although an off-the-shelf item, have occasionally been a problem on past programs due to cells not meeting specified performance requirements. As with the experiment subsystem, our bus design approach minimizes technological and schedule risk resulting in a low cost and low risk program.

12.2 PERSONNEL RESOURCES

Although the type of talents (i.e., cryogenic engineers, spacecraft engineers, etc.) needed to implement the COLD-SAT program is generally available in the aerospace community, it is another matter to obtain management and engineers experienced in high performance low cost cryogenic flight hardware design and low cost spacecraft implementation. BASG is one of the few aerospace companies which have access to all of these talents, crucial to a low

Table 12-1

EXPERIMENT SUBSYSTEM TECHNOLOGICAL RISK

ITEM	CATEGORY				RISK
	A	B	C	D	
Tank, Supply		x			3
Tank, Depot		x			3
Tank, OTV		x			2
LADS		x			4
Mixer, Pressure Control		x			3
Sensor Temperature	x				0
Sensor, Pressure	x				0
Liquid-Vapor Sensor	x				1
Liquid Level Sensor		x			2
Flow-meter, Turbine		x			4
Flow-meter Thermal Mass Flow		x			3
Accelerometer	x				1
LH_2/GH_2 Disconnect	x				0
LH_2 Dist. Subsystem					
Valve, LH_2		x			4
Valve, Relief (Cryogenic)		x			3
Valve, Pyrotechnic	x				0
Valve, Check (Cryogenic)		x			3
Back Pressure Regulator		x			3
J-T Expanders		x			2
Burst Disc	x				0
Pressurization Subsystem					
Hi Pressure Bottle		x			2
Valve, Hi-Pressure	x				1
Valve, Relief (Hi pressure)	x				0
Valve, Check (High Pressure)	x				0
Presure Regulator (Hi-Pressure)	x				1
Pressure Regulator (Low Pressure)	x				1
Burst Disc	x				0

CS.440

Table 12-2

SPACECRAFT BUS SUBSYSTEMS TECHNOLOGICAL RISK

ITEM	CATEGORY A	B	C	D	RISK
CONFIGURATION/STRUCTURAL SUBSYSTEM:					
Prime		X			2
Secondary		X			2
Mechanisms		X			4
THERMAL SUBSYSTEM:					
MLI		X			2
Louvers	X				0
Heaters	X				0
Radiators		X			2
Paint	X				1
Shade Assembly		X			3
POWER SUBSYSTEM:					
Solar Array		X			3
PCU		X			2
Batteries	X				1
Distribution		X			2
ACS SUBSYSTEM:					
Attitude Control Electronics		X			3
IRU	X				2
Horizon Sensors	X				1
Sun Sensor	X				0
Magnetometer	X				0
TT&C SUBSYSTEM:					
Subsystem Processor		X			3
Data Storage Unit		X			2
Transponder	X				1
Misc, RF Switches, Filters and Cables	X				0
Omni Antenna	X				1
Gimbal		X			2
Hi Gain Antenna (μstrip)	X				1
PROPULSION SUBSYSTEM:					
Pressurant Tank-16.5 in.	X				1
Gas Valves (Electric)	X				0
Pressure Regulator	X				1
Check, Relief Valves	X				0
Gas, Liquid Filters	X				0
Pressure Transducer	X				0
Gas, Liquid Valves (Manual)	X				1
Propellant Tanks	X				1
Dual Series Control Valves	X				1
Thrusters (0.14, 0.25, 0.49, 7.0)	X				1
Thruster Heaters & Temperature Sensors	X				0

CS.469

risk, low cost COLD-SAT program, under one roof. Mission specific requirements, analysis and specialty items such as liquid acquisition devices and thermodynamic vent systems which are outside our area of expertise will be provided by our team members MDAC and BAEC.

12.3 KEY FACILITIES REQUIRED

Preliminary analyses of the facilities required to implement COLD-SAT reveal that most are readily available in the aerospace community with three key exceptions:

- A large thermal vacuum chamber capable of accommodating the fully assembled COLD-SAT spacecraft

- A relatively large cryogenic thermal vacuum chamber that is compatible with LH_2

- A large acoustic chamber capable of accommodating the fully assembled COLD-SAT spacecraft

BASG has a large shuttle class thermal vacuum chamber (BRUTUS) that has a working space of 18 ft. dia x 24 ft high which can easily accommodate the COLD-SAT spacecraft. In addition, BASG has a liquid hydrogen test facility with a 5 ft dia x 9 ft high vacuum chamber. This facility will be used to conduct COLD-SAT component testing and the integrated cryogenic subsystem test to verify performance with LH_2.

The large acoustic chamber needed to verify COLD-SAT structural integrity is located at Martin Marietta Corporation (MMC) approximately 40 miles from BASG, and is readily available. This acoustic chamber has been used on past BASG programs and all of the interfacing and contractual infrastructure between the two companies is in place.

12.4 MAJOR SYSTEM TESTING

Although component qualification testing will be performed wherever necessary, a protoflight qualification/testing philosophy has been adopted for the COLD-SAT spacecraft assemblies. However, since the depot and OTV tanks are new light weight designs they will undergo full qualification testing at the assembly level. The experiment subsystem will be built up in parallel with the bus and will undergo thermal vacuum testing, cryogenic functional testing with LH_2, leak check, software checkout and an integrated cryogenic subsystem test prior to delivery for integration with the bus. The cryogenic functional test for the OTV and depot tanks will occur in conjunction with their thermal vacuum test since they do not have vacuum jackets. In addition, the supply tank will undergo prototype vibration and qualified to levels specified in MIL-STD-1540B.

Since the experiment and spacecraft bus are being built and functionally tested in parallel a bus structural mock-up is fabricated for mounting the component boxes and installing a test harness. The spacecraft subsystems are individually installed and tested prior to integration with the cryogenic subsystem/prime spacecraft structure. After the interface is verified and the mass properties determined, a modal survey is performed. The propulsion subsystem and the solar arrays are integrated and tested next. The mass properties are again determined and COLD-SAT compatibility with TDRSS is verified. Prior to any environmental testing a COLD-SAT system performance test will be run. This test is used as a baseline to compare with the system performance after environmental verification. Integrated systems testing, which includes abbreviated subsystem functionals and on-orbit simulations, will be conducted during thermal vacuum testing. Once the COLD-SAT spacecraft is environmentally verified, the final mass properties are ascertained. A flight acceptance review will be held prior to shipping the spacecraft. A more detailed description of COLD-SAT test and integration is given in Section 8.0.

12.5 PROGRAM SCHEDULE

The schedule for the COLD-SAT project is shown in **Figure 12-1**. The schedule indicates a four year program from authorization to proceed (ATP) to delivery

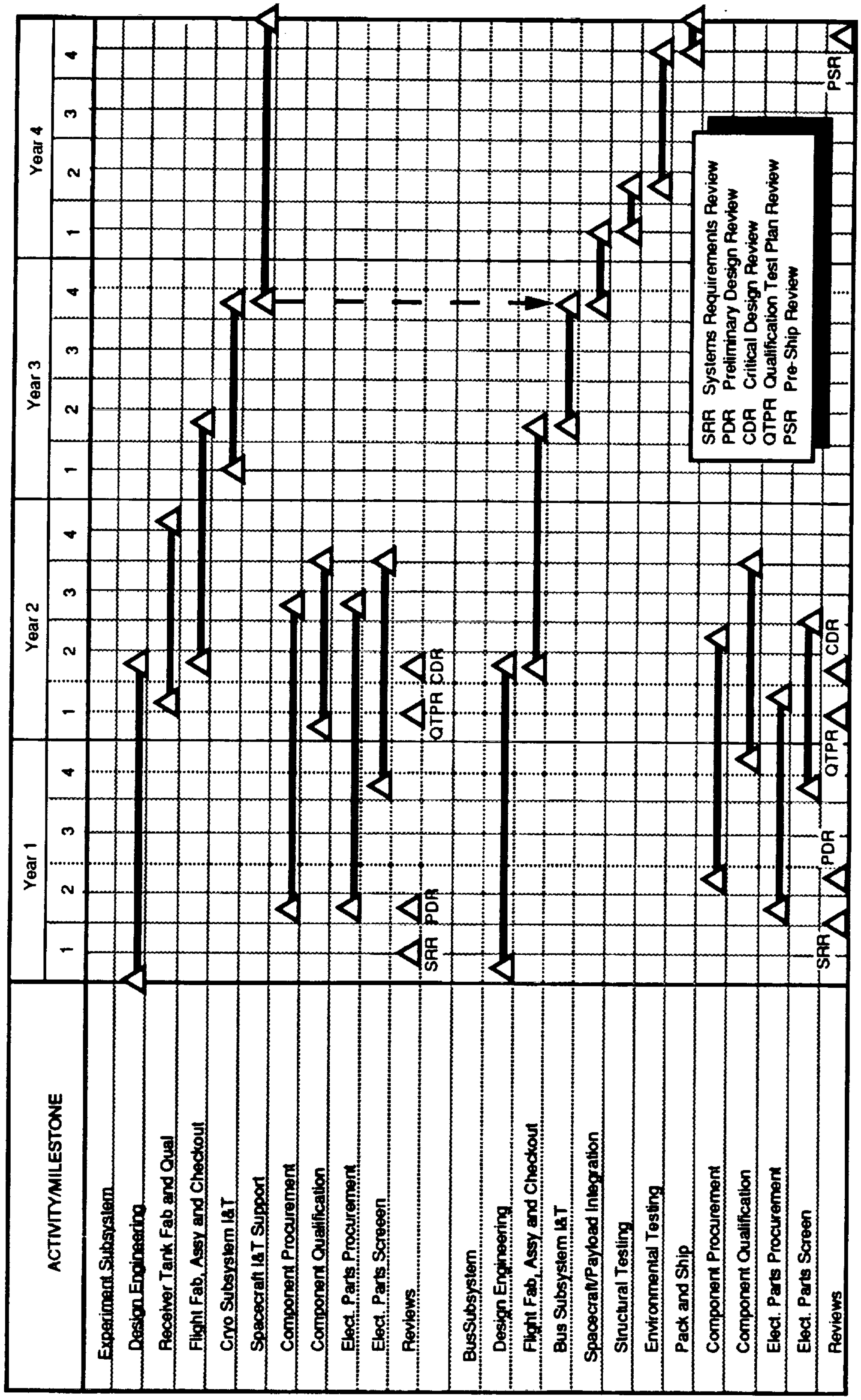

Figure 12-1. COLD-SAT program schedule

of the system to the launch site for integration with the launch vehicle, and is consistent with past BASG projects of similar complexity and scope. However, the schedule is predicted on the availability and use of the bus and experiment subsystem requirements and designs that result from the COLD-SAT Phase B studies. It is this body of data that will enable the experiment subsystem SRR and PDR to occur two and four months respectively after ATP, and for the spacecraft bus SRR and PDR to occur three and five months respectively after ATP. In addition, the Phase B data will enable the initiation of long lead item component procurement approximately four months into the program.

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16. Abstract

This feasibility study presents the conceptual design of a spacecraft for performing a series of cryogenic fluid management flight experiments. This spacecraft, the Cryogenic On-Orbit Liquid Depot-Storage, Acquisition, and Transfer (COLD-SAT) satellite, will use liquid hydrogen as the test fluid, be launched on a Delta II expendable launch vehicle and conduct a series of experiments over a two to three month period. These experiments will investigate the physics of subcritical cryogens in the low-gravity space environment to characterize their behavior and to correlate the data with analytical and numerical models of in-space cryogenic fluid management systems. Primary technologies addressed by COLD-SAT are: (1) pressure control, (2) chilldown, (3) no-vent fill, (4) liquid acquisition device fill, (5) pressurization, (6) low-g fill and drain, (7) liquid acquisition device expulsion, (8) line chilldown, (9) thermodynamic state control, and (10) fluid dumping.

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