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Spacecraft Attitude Impacts on COLD-SAT Non-Vacuum Jacketed LH_2 Supply Tank Thermal Performance

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SPACECRAFT ATTITUDE IMPACTS ON COLD-SAT NON-VACUUM JACKETED LH_2 SUPPLY TANK THERMAL PERFORMANCE

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Abstract

For determining the optimum on-orbit attitude for the COLD-SAT cryogenic experimentation satellite, a comparative analytical study has been performed to determine the thermal impacts of spacecraft attitude on the performance of the COLD-SAT non-vacuum jacketed LH_2 Supply Tank. Tank thermal performance has been quantified by total conductive and radiative heat leakage into the pressure vessel due to the absorbed solar, earth albedo and infrared on-orbit fluxes, and also by the uniformity of the variation of this leakage on the vessel surface area. Geometric and thermal analysis math models have been developed for the spacecraft and the tank as part of this analysis, based on their individual thermal/structural designs. Two quasi-inertial spacecraft attitudes have been investigated and their effects on the tank performance compared. The results of this study are one of the criteria by which the spacecraft orientation in orbit has been selected for the in-house NASA Lewis Research Center design.

Introduction

The Cryogenic On-Orbit Liquid Depot - Storage, Acquisition and Transfer (COLD-SAT) spacecraft has recently completed Phase A feasibility design studies. These studies were conducted by three concurrent contracted efforts through Ball Aerospace, General Dynamics and Martin Marietta, and also by an in-house design group at NASA Lewis Research Center (NASA Lewis). This low-earth orbiting spacecraft will perform fluid management experimentation on the behavior of subcritical liquid hydrogen (LH_2) in the low-gravity environment of space. It will provide test data to validate analytical models for the storage, supply, acquisition and transfer of LH_2 in space. These validated models will then be used to develop design criteria for subcritical cryogenic fluid management (CFM) systems onboard future NASA vehicles employing LH_2 as a propellant. These vehicles are needed for the transfer and transportation functions for the Space Exploration Initiative missions to the moon and Mars.

In the NASA Lewis in-house spacecraft design, which is described in this paper (Fig. 1), the 6600 lb COLD-SAT spacecraft will be launched in 1998 aboard an expendable launch vehicle (ELV) into a 550 NM circular orbit, with an inclination of 18° relative to the equatorial plane. The spacecraft will have a 6-month active lifetime during which 11 prioritized CFM experiments will be performed with LH_2.[1] The higher priority Class I experiments include tank pressure control, no-vent fill of tanks and tank chilldown. At the end of the spacecraft lifetime, all of the LH_2 will be exhausted, due to consumption by the experiments and operations, and by boiloff due to the parasitic heat leaks into the system.

One of the earliest and most critical decisions in the in-house feasibility study was the selection of the spacecraft attitude while in orbit. Attitude selection has a critical impact on the thermal performance of the three LH_2 experiment tanks on COLD-SAT. This impact is due to the absorbtion of incident on-orbit solar, earth albedo and earth infrared thermal radiation fluxes by the tank surfaces. The largest of these tanks is called the Supply Tank, which will be the only one filled with LH_2 upon launch. This tank will provide the necessary fluid for performing experiments not only within itself, but also to the two other, smaller, receiver tanks.

Thermal performance of the COLD-SAT Supply Tank is characterized by two major design goals: (1) a total radiation and conduction heat flux on the pressure vessel (PV) and (2) a flux variation uniformity criteria at any location on the PV surface. Both of these criteria, which are quantified later, are necessary to meet the initial condition for performing the Class I active and passive pressure control experimentation. The first design goal also reduces parasitic boiloff losses from the tank to within 0.32 percent per day. This low boiloff will conserve LH_2 for the 6-month active life of the spacecraft. Uniformity is important because a vacuum-jacket or a vapor-cooled shield has not been included in the design.

This paper describes the results of a comparative analysis on the thermal impacts of two quasi-inertial spacecraft attitudes on Supply Tank performance. An initial and conceptual design of the Tank was used in this analysis. The results then formed one of the criteria for selecting the attitude which has the greatest possibility of meeting the goal for total heat flux and uniformity criteria. Other criteria considered were factors such as orbit perturbations due to experimental thrusting, gravity gradient torques, etc. Results of this study are spacecraft and tank design dependent and may not be applicable to other, similar spacecraft configurations.

Spacecraft Configuration

COLD-SAT is a 3-axis stabilized spacecraft and is configured for a 11 ft diameter medium Atlas-Centaur payload fairing. It has all the usual housekeeping subsystems of structural, thermal control, telemetry, tracking and command, attitude control, power and propulsion. In addition, there is an experiment subsystem that is comprised of LH_2 tankage, two independent pressurization systems, electronics boxes, instrumentation and fluid handling components. The spacecraft is configured in a modular design consisting of five separate modules, with considerations for manufacturing, testing, assembly, accessibility and maintainability in the layout. As shown in Fig. 2, each module is independently supported by its longerons. These longerons also form the primary structure of the spacecraft. The modules are then individually bolted at the interfacing flanges of the longerons to form the complete assembly of the spacecraft. Following is a description of the major components of COLD-SAT, with the Supply Tank module being described in a later section.

Electronics bays. - The Electronics Bay number 1 module weighs 745 lb and consists of mostly electronics boxes and batteries which are mounted on aluminum (Al) honeycomb sandwich panels forming the sidewalls of the electronics bay. It also contains four, 22-in. diameter propellant tanks containing 600 lb of hydrazine and other components of the propulsion subsystem. These tanks are mounted on an Al honeycomb plate located in the center of the bay. On four of the eight sidewalls are mounted propulsion thrusters to provide the required attitude control, operational and experimental torques. In addition, there is a gimballed thruster for performing bi-axial thrusting with a maximum gimbal angle of $\pm 15^\circ$ per axis. Sun sensors and their electronics are also located here. The Electronics Bay number 2 is similar in design to the Electronics Bay number 1, but only contains electronics boxes which are mounted on the bay honeycomb sidewalls. Horizon sensor optics and electronics are located here. This module weighs 262 lb.

The internals of both electronics bays are radiatively coupled together with high emissivity (ϵ) black paint on the boxes, panels and the propulsion tanks to minimize thermal control heaters. On the external space-facing sidewalls of the bays are located low absorbtivity (α) and high ϵ optical solar reflectors (OSRs) to radiate to space the unwanted heat from individual boxes, thereby maintaining their temperatures within specified operating ranges. Multilayer insulation (MLI) blankets protect the bays on all surfaces except where OSRs are located. Since the thermal requirement of these modules dictate warm, room temperature conditions for the electronics and the hydrazine, they are conductively isolated from the LH_2 Supply and Receiver Tank modules by using G-10 fiberglass spacers at the longeron interfacial attach points.

Receiver tanks. - The Large Receiver Tank module weighs 263 lb and consists of a 21 ft^3 non-vacuum jacketed AL 5083 insulated tank. The tank is cylindrical with ellipsoidal heads, having a diameter and length of 32.5 and 51.5 in., respectively. The Small Receiver Tank module weighs 200 lb and is similar to the previous module and supports a cylindrical, 13.5 ft^3 tank with ellipsoidal heads. Made from AL 5083, the tank is 31.5 in. long and 36 in. in diameter. Both modules have an 0.5 in. thick Al honeycomb "can" with cylindrical, truncated cone heads which surrounds the PV. Each "can" supports two, 1/2-in. thick double aluminized Kapton (DAK) MLI blankets. The tanks are each supported by 10, S-2 glass/epoxy struts from the module longerons. Individual valve panels on the domes provide a convenient location for mounting cold valves close to the LH_2. In order to reduce parasitic losses into the tank through the MLI, a second surface silverized Teflon (SSST) outer layer is prescribed for minimizing the outer surface temperature of the MLI blanket.

Fluid system. - There are three fluid system legs for providing transfer flows at the required rates of 50, 100 and 200 lb/hr from the Supply Tank to the receiver tanks. Due to the complexity of the fluid system and its inherent safety and redundancy considerations, an enormous number of fluid components have to be accommodated. These include cryogenic magnetically latching valves, pressurant gas control valves, active pressure control mixer pumps, thermodynamic vent system (TVS) heat exchanger (HX) and Joule-Thompson (J-T) valves, relief and check valves, and flight vent and tank chilldown spray nozzle control components and fill, drain and purge system components.

To comply with the modular philosophy in design, all of the fluid components required for a particular module are mounted on panels which are bolted to its longerons, but conductively isolated by fiberglass spacers. The plumbing that is needed for transferring fluid, both LH_2 and pressurant, between individual tanks and bottles are located on a plumbing tray that traverses the entire length from the Supply Tank module aft end to the middle of the Small Receiver Tank module. This tray also carries the wiring harnesses that are located between modules.

The longerons for all modules are fabricated from AL 6061 and are MLI covered. Attached to the Supply Tank module longerons are two fixed solar arrays, canted at an optimum 13° angle to reduce the cosine Beta (β) angle penalty of solar flux on power generation. (Beta angle is the angle of the sun line to the orbit plane). The arrays have a 2050 W beginning of life capability with a 190 ft^2 total area. A high gain antenna is also mounted off the longerons on a boom, through which TDRSS communication is established 13 min per orbit for downlink of data.

Spacecraft Attitude Considerations

The 6600 lb COLD-SAT spacecraft can be launched from Cape Canaveral in Florida to achieve a 550 NM, 18° inclination circular orbit utilizing the lift capability of the Atlas-Centaur ELV. At this inclination, the sun makes a β angle of ±41° relative to the orbit plane. The orbit has a period of 105.46 min, nodal regression rate of 5.66 deg/day and eclipse times ranging from 28 to 35 min. Spacecraft geometric blockage of the TDRSS communication link is less than 2 percent. A sun synchronous orbit was not selected due to the higher inclination of approximately 80° that would be needed and is not possible on an ELV from the east coast of the United States. The altitude was selected to achieve a design orbital lifetime of 500 years, thereby precluding the possibility of major portions of the spacecraft re-entering the atmosphere. In addition, drag acceleration at this altitude is less than 10^{-6} g, which is an experiment requirement for background acceleration. No launch window constraints were imposed, although, minimum orbital perturbations due to experimental thrusting, gravity gradient torques and minimization of solar flux on the cryogenic system were the major considerations for the selection of the final attitude.

An initial ground-rule that was imposed on the spacecraft design was that only fixed solar arrays would be used. This ground-rule precluded the acceleration disturbances that would be created in the LH_2 tankage due to articulating arrays and their drives. This decision also eliminated the cost, integration and reliability implications of including array drives. As a result, only sun-tracking, "quasi-inertial" attitudes were possible. In these attitudes, the spacecraft is oriented to have the fixed arrays constantly face the sun while rotating about one of its axes to track the sun at 1 revolution per year. The spacecraft will also rotate clockwise about the earth's polar axis at approximately 5.7 revolutions per year due to nodal regression.

These attitudes resulted in large portions of the spacecraft becoming incident to the solar flux, thereby creating a "hot" side of the spacecraft. Considerably higher temperature excursions occur on this "hot" side as compared to the other anti-sun side. The anti-sun side, being exposed to radiation to space and the comparatively lower earth albedo and infrared fluxes, acts as a "cold" side and thus performs in lower temperature ranges.

It has been calculated that in a 6-month period, given the COLD-SAT orbital parameters and for any launch window, the sun crosses the orbital plane a maximum of five times. To maintain the "hot"/"cold" side restriction on the spacecraft periphery, the spacecraft will have to perform a 180° roll maneuver around its long axis whenever the sun crosses the orbit plane. This is necessary to ensure that the same side of the spacecraft remains in view of the sun at β = ±41°, and also to keep the arrays generally sunpointed.

This "hot" and "cold" side bifurcation of the spacecraft is a welcome opportunity to be exploited in the design of the experiment subsystem. Since this subsystem is comprised of cryogenic components, an important consideration here is to reduce parasitic heat leakage into the tanks. Initial studies had indicated that, among the PV penetrations, the 576 manganin and Al wires, ranging in size from 22 AWG (American Wire Gage) to 26 AWG and 8, SS 304 plumbing lines ranging from 1/4 to 1 in. in diameter, are the major contributors. Therefore, it was decided to route all wiring and plumbing lines on the plumbing tray that will be located on the spacecraft "cold" side. This tray will be conductively isolated from the spacecraft structure, and will be radiatively cooled to space to achieve as low a temperature as possible. The support struts, however, have to be distributed around the tank to achieve a uniform support configuration which precludes the possibility of connecting the struts to the longerons only on the "cold" side. As explained later, the thermal control of the longerons is designed to minimize their temperature, thereby reducing strut leakage. A similar approach is used for the purge diaphragm outer layer of the tank MLI itself.

With these quasi-inertial attitudes as the optimum thermal orientation for the cryogenic tankage, two specific attitudes were considered, as indicated in Fig. 3. In one of the attitudes under consideration, named Attitude "A", the long axis (x-axis) of the spacecraft is coplanar with the orbit plane. The spacecraft aft end is aligned with the projection of the sun in that plane. The fixed solar arrays are canted at an optimized 13° to orbit normal. When the angle of the sun-line to the orbit plane (defined as Beta angle, β) is 0°, the spacecraft Electronics Bay number 1 module shields the Supply Tank and other tanks from solar flux. This provides the coldest tank thermal condition for this attitude. As the sun travels to the maximum of $\beta = 41°$ (for the selected orbital inclination of 18°), the projected area of the Supply Tank to the sun-line is impacted from solar flux. This creates the warmest environment for this attitude.

The other attitude investigated herein is referred to as Attitude "B", in which the spacecraft long axis is normal to the orbit plane. The fixed solar arrays are canted as before. By rotating the spacecraft around its z-axis, as shown in Fig. 3, arrays remain sun-pointed for $\beta = \pm 41°$. For $\beta = 0°$, the solar flux impacts the projected Supply Tank area broadside. This produces the warmest environment for the Supply Tank. For the maximum $\beta = \pm 41°$, the least projected area is available for the solar flux, hence the coldest conditions for the tank. A third attitude in which the same side of the spacecraft continuously faces the earth was discounted. This was due to the necessity of including articulating arrays along with their inherent concerns as described earlier.

Supply Tank Module Description

The Supply Tank module (Fig. 4) weighs 1696 lb and consists of the following: (1) a LH_2 PV which supplies fluid to other subsystem components, (2) an MLI assembly to shield the tank from heat gains and provide a more uniform heat flux to the tank, (3) a GHe purge system for ground purging of the PV surface and the MLI, (4) a radiator plumbing tray to support plumbing and wiring harnesses and to act as a heat sink for the penetrations before entering the tank, (5) a gaseous hydrogen (GH_2) system for vaporizing LH_2 to be used for pressurized transfers, (6) a GHe pressurant system as an additional source for the Supply and receiver tanks, (7) a vent panel and HX which allows venting gas to warm before being expelled from the system, (8) and a ground interface panel for Supply Tank fill/drain and GHe purge operations.

Pressure vessel. - The 144 ft^3 AL 5083 PV is cylindrical with ellipsoidal dome ends, and when filled to a 92 percent fill level, has 565 lb of LH_2 at 20 psia. The vessel surface area is 143 ft^3 with a 0.08 in. thick shell reinforced by two girth rings located fore and aft on the cylindrical barrel section. It is supported by eight fiberglass struts, four fore and four aft, which attach to the girth rings symmetrically around the circumference from the module longerons. The tank contains instrumentation rakes for monitoring temperatures and pressures and determining fluid levels, and a LAD for withdrawing liquid from the two-phase fluid inside the PV under low-gravity. On the external wall of the PV are six cold valve panels mounted directly to the tank, four on the forward dome and two on the aft. The following are the forward dome plumbing penetrations: a 1 in. ground/ascent vent line which penetrates the PV wall and extends down to the 92 percent fill level; a 1 in. relief line for overpressure protection; a 1/2 in. pressurization line which provides high pressure GH_2 or GHe for pressurized expulsions and transfers, and a 5/8 in. refill line for return of cryogen to the Supply Tank from any of the two receiver tanks. The following are the aft dome plumbing penetrations: a 5/8 in. ground fill/drain and on-orbit transfer line and three, 1/4 in. TVS lines that provide subcooled liquid obtained from a J-T expansion valve to a passive HX on the tank wall, a HX on the LAD and a integrated pump/mixer active TVS HX, respectively. These HXs are required for operational and experimental requirements for tank pressure control.

For experimental data gathering in the Supply Tank module, there are 646, 26 AWG manganin wires for temperature and pressure sensors and liquid/vapor detectors that proceed from the plumbing tray to the honeycomb MLI "can". Five hundred and fourteen of these go on to penetrate the PV in 11 harnesses of 47 wires each. Four of these harnesses penetrate the fore and aft domes, respectively, while three penetrate the barrel section. The difference is for the temperature measurements of the MLI blankets and "can", and do not actually enter the PV wall. There are 54, 24 AWG Al wires in 2 harnesses of 22 and 32 wires each, that contact the fore and aft domes, respectively, for providing current to the cold valves and active TVS mixer/pump that is mounted on the PV wall. In addition, there is 1 harness of 8, 22 AWG Al wires proceeding to the barrel section for the heaters that are required to provide the desired heat fluxes for the pressure control experiments, as described later. These heaters consist of etched metal foil resistive elements laminated between layers of Kapton insulation and will be applied to the PV wall using an adhesive.

<u>MLI assembly</u>. - The PV is encapsulated by an Al honeycomb sandwich "can" which supports a MLI blanket and thermally isolates the PV from direct MLI radiation and conduction. The "can" consists of a 3/8 in. AL 5056 core bonded to 0.02 in. Al 2024 facesheets with an epoxy-polyamide resin. The cell size is 1/4 in. and the foil thickness is 0.001 in. The "can" is supported off the tank support struts by thermally conducting, bonded sleeves at a distance of 2 in. from the PV. MLI blankets cover the "can" and are held in place by Velcro fasteners and nylon pins.

The MLI "can" has two vent doors located fore and aft on the spacecraft "cold" side and are spring loaded closed, but are held open at launch by space qualified pyrotechnic pin pullers. Approximately, 14 hr after launch, the doors, which are covered with MLI blankets, will be closed to reduce thermal radiation. Door stops will keep the doors from closing completely to allow a gap for residual gases in the MLI to migrate out.

<u>GHe purge system</u>. - External to the MLI layers is a purge diaphragm which contains the GHe used to purge the volume between the purge diaphragm and the PV (72 ft^3), including the MLI. Purging will be performed prior to filling the PV with LH_2, to prevent the liquefaction of air on the tank and related components. This system is similar to the one designed for Shuttle/Centaur.[2] It consists of a purge diaphragm, two relief valves, and two ascent vent doors. A positive purge pressure will be maintained during pre-launch activities to prevent ingress of the payload fairing atmosphere. Sections of the diaphragm are sewn and taped to minimize leakage. The assembled pieces are fastened to two support rings of 0.03 in. fiberglass by Al retaining rings. The support rings are hung off the cylindrical portion of the MLI "can." The relief valves are attached to the aft support rings.

The purge diaphragm also has two ascent vent doors which are offset from the doors in the MLI "can" to reduce radiation losses, are supported off the fiberglass support ring and are spring loaded open. They will be held closed during ground purging operations by the same type of pyrotechnic pin pullers used on the MLI "can" vent doors. Seconds after liftoff, the pin pullers will be activated and the door will open for GHe ascent venting. Each door is sized for the full flow of the venting GHe in case either pin puller should fail. Once open, the doors remain open for the duration of the flight which allows continuous venting and outgassing of the MLI layers.

GHe will be introduced near the top or forward end of the purge diaphragm for ground purging. A positive pressure of between 0.1 and 0.3 psid will be maintained. Burst pressure of the diaphragm is estimated to be 0.5 psid. Two relief valves located near the bottom or aft end will be set at 0.4 psid to protect the diaphragm from rupture.

<u>Plumbing tray</u>. - The plumbing tray is located just outside the spacecraft structure on the "cold" side until it approaches the Bay number 2, after which it runs inside the spacecraft structure on its way to the receiver tanks. The AL 6 061 tray contains the plumbing and wiring harnesses with sufficient space for contraction and expansion bends. Its purpose is to allow the penetrations to get as cold as possible before they enter the MLI and tank surface. Six support brackets are welded to crossmembers of the spacecraft structure. The tray is attached to the brackets with SS bolts and G-10 fiberglass spacers in an attempt to thermally isolate the tray from the spacecraft.

Pressurization systems. - Also in the module are two, 2000 psia, stainless steel (SS) 304 hydrogen vaporizer bottles with a storage capacity of 3.5 lb of LH_2 each. These vaporizers produce gas from LH_2 withdrawn from the Supply Tank for autogenous pressurization prior to transfer. In addition to the vaporizers, two 3000 psia, filament wound, metal lined composite bottles store 8.5 lb of GHe to provide another source of pressurant.

Panels. - In keeping with the modular concept, individual components are mounted on several panels based on functional requirements. These panels are namely: a vent panel, a vaporizer panel, two connector panels, a helium panel, and a T-4 fill/drain disconnect panel. The connector panels provide interface points for the instrumentation, control and heater wiring for this module. The ground interface T-4 panel will be located on the "cold" side of the spacecraft to allow plumbing and wiring harnesses and disconnects to be cooled. Remaining panels are located on the spacecraft "hot" side to take advantage of warmer temperatures and higher heat fluxes, based on their requirements. Plumbing from all panels proceed to the plumbing tray before penetrating the PV. The "hot" side panels provide solar flux blockage to the Supply Tank.

Tank Thermal Performance Criteria

The thermal environment of an on-orbit cryogen storage tank includes radiation from the sun, earth albedo and infrared, radiation to deep space, radiation from other spacecraft surfaces, and on-board conduction paths such as struts, plumbing, insulation, and wiring for instrumentation, power and heaters. Heat addition to a tank containing cryogen can result in pressure rise and temperature stratification in the fluid. A reasonable degree of fluid mixing can reduce temperature stratification and reduce the rate of pressure rise. However, with a given rate of heat input, any cryogenic tank will require venting to prevent overpressurization, even if perfect mixing occurs. The COLD-SAT tank pressure control experiment is designed to address the key issues related to on-orbit pressure control in cryogenic systems.

One of the key requirements of the pressure control experiment is to quantify the heat input into the contained LH_2 in the Supply Tank, and to have the capability of selecting the heat input at three known, uniform preset levels of 0.1, 0.3 and 0.6 Btu/hr-ft^2. These levels are representative of heating rates for an on-orbit long term cryogen storage depot, a space based transfer vehicle and earth-to-orbit resupply tanker, respectively. Since it is required that the heat flux be variable and accurately known over the range described above, only known thermal control coatings and surface finishes can be applied to the inner facesheet of the MLI "can" and the PV surface. By doing so, emissivities will be accurately known for the calculation of the radiant heat input to the tank. The "can" temperature, along with the temperature gradients of the necessary conductive paths to the tank (plumbing lines, struts and wiring), will allow an accurate prediction of the heat input into the LH_2 tank.

To be able to accurately maintain the lowest required flux level of 0.1 Btu/hr-ft^2, it was decided to have a tank thermal design goal of less than 0.1 Btu/hour-ft^2, so that additional heat may be provided by heaters to bring up the flux to the required level. As a goal, the conductive component was required to be less than 10 percent of the total radiant heating of the PV for the lowest heat flux imposed. Although uniformity of flux variation around the surface of the PV was not quantified, it was accepted as a design goal to achieve a less than 20 percent variation in flux, as related to the total flux.

Consequently, thermal performance of the COLD-SAT Supply Tank is characterized by: (1) a total background radiation and conduction heat flux of less than 0.1 Btu/hr-ft^2 on the PV surface, (2) a flux variation uniformity within 20 percent of the total at any location on the surface, and, additionally, (3) conductive component is supposed to be less than 10 percent of the radiant heat flux.

Module Thermal Design

The honeycomb "can", described above, supports the MLI blankets and a purge diaphragm system and acts as a radiation shield to which all the penetrations are thermally shorted to minimize conductive parasitic heat leaks into the PV. Radiative and conductive heat leaks through the MLI also are intercepted by the "can", which being made from highly conductive Al, rapidly diffuses the heat to achieve a nearly (± 3 °F) uniform temperature all around. This provides a uniform radiative heat transfer to the PV, which is a requirement of the pressure control experiment.

The MLI "can" is made up of cylindrical and truncated cone sections which are mechanically fastened together. This simple geometry, having no doubly curved surfaces, allows the MLI blankets to be made from flat patterns (no gore sections). Reducing the seam lengths decreases radiation tunnelling which improves the blanket performance. Consequently, a relatively low degradation factor of 1.5 has been used for the MLI in the thermal analysis. The MLI "can" is supported by sleeves attached to the PV support struts. These sleeves are designed to distribute the load of the "can", the MLI and purge diaphragm over the 8 struts. The sleeve also thermally grounds the struts to the MLI "can", thereby shunting heat into the "can" instead of the PV.

The inner facesheet of the "can" is considered to have an optimum thermal ϵ of 0.05, as does the PV surface.[3] It was determined[4] by parametric analysis that having a lower value of ϵ, which tends to radiatively decouple the "can" inner facesheet from the PV surface, decreases the total value of the heat flux on the PV.

The MLI for the tank consists of two blankets, each approximately 1/2 in. in thickness, for a total of 60 layers for the two. The outer and innermost layers consist of a laminate of Nomex scrim sandwiched between two layers of Kapton. This material was selected for its low weight and rip-resistant features. The other layers are Kapton with vacuum-deposited Al on both sides and are separated by Dacron net spacers. The blankets overlap at all seam locations and are held in place by nylon positioning pins and grommets. A five layer MLI blanket is also used to cover the positioning pins and seams. In addition, Velcro fasteners are used between the MLI "can" and blankets and to hold seams together. The MLI is electrically grounded to the spacecraft.

The purge diaphragm, which is outside the MLI, consists of two Kevlar-cloth reinforced shields, separated by an embossed Kapton shield. The outer shield has a layer of SSST ($\alpha/\epsilon = 0.09/0.75$). The high-strength reinforced shields are required to withstand a purge system design pressure of 0.5 psid. These shields are actually laminates with the high-strength Kevlar cloth sandwiched between two layers of Kapton. All shield surfaces have a vacuum-deposited layer of Al applied to achieve ϵ of 0.05 or less.

In the plumbing tray, fairleads secure the plumbing and harnesses to the tray. The portion of a fairlead attached to the inboard side of the tray will be made of insulating fiberglass while the portion of the fairlead attached to the radiating surface will be made of conductive Al. The radiating surface of the tray is covered in an MLI blanket with an outer layer of SSST ($\alpha/\epsilon = 0.09/0.75$).

In an effort to minimize conductive leaks to the PV, SS 304 plumbing is used because of its lower thermal conductivity relative to Al. All wiring harnesses and plumbing coming from the radiator tray is shunted to the MLI "can" and penetrates the "can" barrel section, before entering the PV. Twelve inches of plumbing and wiring runs are baselined from the tray to the "can". Lengths inside the MLI have been maximized to 24 in. to reduce the area to length (A/L) ratio for conduction. Penetrations are attached to the tank wall with fiberglass fairleads to limit conduction while providing structural support. To improve the uniformity of heat gain to the tank, each dome mounted AL 5083 cold valve panel has six legs, with each leg having a 6 in.^{2} of contact area. Struts are made of low conductivity, filament wound S-2 glass/epoxy with SS end fittings, each 13 in. long, with a conductive length of 10 in. outside the "can" and 3 in. from the facesheet to the PV. The inner tube of the struts is assumed filled with pieces of Mylar to reduce radiation tunneling.

The module longerons and the struts, plumbing and wiring harnesses from the tray to the PV are all covered with a 12 layer DAK MLI blanket with an outer layer of $\alpha/\epsilon = 0.09/0.75$ to keep them as cold as possible. The spacecraft is configured to maximize the field-of-view of the purge diaphragm outer layer to space for radiative cooling.

Modeling and Analysis

In order to investigate the orbital effects of the two spacecraft attitudes "A" and "B" on the Supply Tank, detailed geometric and thermal math models were developed. The Thermal Radiation Analysis System (TRASYS-II) code was used to develop a 480-surface model of the external surfaces of the spacecraft, including the Supply Tank purge diaphragm. The model uses thermo-optical properties of the spacecraft thermal control surfaces. It accounts for shading from the structural longerons and the various experiment subsystem panels on the "hot" side of the spacecraft, along with the "cold" side plumbing tray. The model was used to obtain absorbed orbital thermal environments of solar, earth albedo and infrared fluxes incident on the Supply Tank by direct and also reflected radiation from other spacecraft surfaces such as the solar arrays. Radiation conductors (RADK's) between the tank and spacecraft surfaces and space were obtained.

This model was exercised for each of the attitudes for the Beta angle environments of 0° and 41°, respectively. Heat absorbed versus time array data for a 360° orbit was obtained for each surface. This data is used for transient thermal analysis using a interpolation subroutine which calls for orbit time and the area for each node. Orbital average absorbed heat data was also obtained for performing steady-state thermal analysis, and contains the time-integrated average heating rates and area for each node.

To calculate radiation conductors for the internal surfaces of the tank, another 26-surface TRASYS-II model was developed for the MLI "can" inner facesheet and the PV surface. Radiation from the tank penetrations was neglected since each internal component will be MLI covered, and hence not involved in minimal radiative exchange.

Outputs from the internal and external TRASYS-II models were incorporated into a spacecraft and Supply Tank, 460-node, three-dimensional finite difference SINDA'85 thermal model to yield temperature-time histories and flux data. Steady-state analyses was performed using the orbital average absorbed heat data to determine initial conditions for the transient analyses. A simplified thermal network is shown in Fig. 5 to represent the detailed tank thermal model.

In the SINDA'85 model, the PV surface area was divided into 12 nodes, 4 each for the domes and barrel. The purge diaphragm, MLI, the 2 honeycomb facesheets and the core were all divided into 14 nodes each, respectively. For each plumbing line, wiring harness and strut, there was a node representing the length from the plumbing tray or longeron to the outer facesheet of the "can". Similarly, there were another node for the length from the outer facesheet to the location of the actual penetration on the PV, for each penetration. This resulted in a total of 16 plumbing nodes, 30 harness nodes and 16 strut nodes. All plumbing, harnesses and struts were assumed to make a perfect thermal short with the outer facesheet before proceeding to the PV. Table 1 gives the A/L ratios from the "can" outer facesheet to the PV, as modeled, and the locations of the penetrations. Lateral conduction for the purge diaphragm and the MLI blankets were neglected, but lateral conduction in the "can" facesheets and core have been included.

Transverse conduction from the outer facesheet to the core and from the core to the inner facesheet, and intra-facesheet radiation were modelled from Ref. 5, where heat exchange across the "can" is given by:

$$Q = \frac{k\Delta A}{\ell A_1}(T_1 - T_2) + \frac{k_A}{\ell}(T_1 - T_2)$$
$$+ \left[0.664\ (\lambda + 0.3)^{-0.69}\epsilon^{1.63(\lambda+1)^{-0.89}}\sigma\left(T_1^4 - T_2^4\right)\right]$$

where Q is the rate of heat transfer per unit area, k is thermal conductivity, ΔA is the core material cross-sectional area for conduction, $\Delta A/A_1$ is solidity, ℓ is the core height, T_1 and T_2 are the temperature of facesheets 1 and 2, λ is the ratio of core height to cell diameter, ϵ is emissivity, σ is the Stefan-Boltzmann constant, and subscript A is for air. The effect of conduction through the air in the cells can be neglected for orbital conditions. This equation was easily modelled as radiation and conduction conductors in the SINDA'85 conductor data block. The MLI was modeled using the blanket thermal conductance equation developed and verified from the IRAS and COBE dewars[6]:

$$k = 0.045\sigma\left(T_h^2 + T_c^2\right)(T_h + T_c)/(2n + 1) + 1.5\times10^{-6}\ \text{mW/cm-K}$$

where T_h and T_c are the boundary temperatures and n is the number of layers. A factor of 1.5 was applied to account for degradation due to compression, penetration gaps and edge effects, using a 60 layer, 1 in. blanket. Cross sectional areas for all plumbing lines were calculated using the ANSI B31.3 code. Cross sectional area for the struts was obtained from Ref. 7 for a S-901 glass/Epon 828 strut for a 175 ft^3 LH_2 tank. Conductivity of the strut was reported by Lockheed[7] to fit the curve:

$$k = 0.050 + 6.35\times10^{-4}\ T$$

where T is the temperature of the strut. Temperature dependent properties for all materials were modeled. Transient runs using implicit forward backward finite differencing for each of the two attitudes "A" and "B", with the Beta angles of 0° and 41° respectively, were performed for 100 orbits in order to determine temperatures and flux results after equilibrium conditions had been reached.

Results and Discussion

For establishing spacecraft attitude dependent heat leaks into the tank, it is important to determine the hot boundary condition for each of the sources of parasitic heat leaks: purge diaphragm for the MLI, longerons for the struts and the plumbing tray for the wiring and plumbing. Analysis results for the cases of interest are presented in Fig. 6 for the purge diaphragm where the surface temperature is plotted versus orbital time. Since the diaphragm was modeled as 14 nodes for achieving enough accuracy in the thermal model, for presentation purposes, a nodal area-weighted average surface temperature was calculated as shown. It can be seen that the attitude "A" for a $\beta = 0°$ has the lowest temperature due to the blocking of the solar flux by the aft end of the spacecraft. This temperature increases for the $\beta = 41°$ case and goes to the extreme warmest for the attitude "B", $\beta = 0°$ and 41° cases, since the sun is now impinging more broadside on the Supply Tank. The temperature profile for the attitude "B", $\beta = 41°$ is slightly higher at its peak than the $\beta = 0°$ case. It can be surmised that this may be due to the forward end of the purge diaphragm being heated through the central cavity of the Electronics Bay number 2 at this high Beta angle. It may also be due to higher radiation contribution from the front MLI of the Bay number 1.

The profiles for the average temperature of the eight longerons and also the plumbing tray was found to be constant. These values are tabulated in Table 2. For the attitude "A", $\beta = 0°$, due to the lack of solar flux, both the longerons and tray were identical at -110 °F. However, as the sun angle goes to $\beta = 41°$, the longeron average temperature rises to -44 °F, but since the tray is on the anti-sun or "cold" side, it remains at a stable -106 °F. Temperatures for both components increase tremendously for the attitude "B".

It was found that due to the high conduction of the honeycomb material (Al), temperature of the inner and outer facesheets and the core are identical. Nodal area-weighted average honeycomb "can" temperatures are presented in Fig. 7, where the initial temperature for each transient was obtained from a steady-state analysis, based on orbital average fluxes. After 50 hr, the transients reach dynamic equilibrium and have a spread from -350 °F to -335 °F, progressing from attitude "A", $\beta = 0°$ to attitude "B", $\beta = 0°$.

Total transient heat fluxes on the PV are presented in Fig. 8, where after transient analyses of 50 hr, equilibrium conditions are achieved. It is evident that for the thermal design used in this analysis, the attitude "A" best case of $\beta = 0°$ produces a 48 percent lower heat flux (0.0559 Btu/Hr-Ft^2) relative to the $\beta = 41°$ case for attitude "B" (0.0826). Comparably, the $\beta = 41°$ worst case for the attitude "A" (0.0728) produces a 22 percent lower heat flux than $\beta = 0°$ for attitude "B"(0.0891).

For determining the uniformity in variation of flux, Table 3 shows the breakdown of the nodal heat fluxes on the forward and aft domes (nodal area = 7.967 ft^2) and the barrel section (nodal area = 19.739 ft^2), as calculated and normalized by the total flux on the PV for each attitude. It can be seen that all nodes have a flux variation within 20 percent of the total heat flux. The forward dome node 1 shows the greatest continuing nonuniformity for all attitudes and β angles. This is due to a 1-in. vent line and an Al heater wire harness penetration at this node. The highest nonuniformity of 19.5 percent is in the aft dome node 1. This node has, among other penetrations, a high heat leak Al wire harness of 32, 24 AWG wires. A breakdown of the conductive and radiative fluxes for each attitude is presented in Table 4. Here, it can be seen that for all attitudes, the total conduction is less than 10 percent of the total radiation, thereby meeting the thermal performance requirements.

Conclusions

For the Supply Tank, the purge diaphragm area-weighted average temperatures in eclipse are similar for all attitudes (-135 °F). However, in full sun attitude "B", $\beta = 41°$ best case (-5 °F) is higher than the attitude "A", $\beta = 41°$ worst case (-30 °F). The attitude "A" best case $\beta = 0°$ (-80 °F) is far superior to attitude "B" best case $\beta = 41°$ (-5 °F). These temperature variations are reflected in the total heat leaks discussed before. Based on these background heat fluxes, boiloff rates will range from 1 lb/day (0.18 percent/day) to 1.6 lb/day (0.28 percent/day). This provides a boiloff rate low enough to conserve LH_2 for the 6-month period, for all attitudes.

All attitude cases meet the thermal performance criteria of total heat flux, uniformity and total conduction. In attitude "A", additional benefits can be obtained by using excess ELV capability to achieve a lower orbit inclination, which will lower the flux even further. For attitude "B", improvements can only be obtained by adding a sun shield. Consequently, maintaining as low a heat flux as possible will significantly help in meeting the heat flux uniformity and controlled low-level heat flux requirements, a conclusion that favors the attitude "A". Thermal considerations, similar to those for the Supply Tank, also apply to the receiver tanks.

Additional spacecraft thermal surfaces are also effected by the two attitudes. Attitude "A" always has the sun on the aft end of the spacecraft which eases the temperature control of the propulsion system hydrazine components located at that end. For attitude "B", this surface always faces space. Additionally, reduced solar flux on the electronics bays will ease thermal design for the attitude "A", where all avionics have a 100 percent duty cycle requiring heat dissipation to space.

The advantage of attitude "B" is that there are essentially no spacecraft orbital perturbations due to experimental thrusting. In attitude "A", experimental thrusting reduces perigee altitude, and therefore, orbital life. These effects can, however, be compensated by increasing the initial altitude, as was performed here by increasing the altitude from 500 NM to 550 NM. This altitude is well within the ELV capability. Also, an additional 50 lb of hydrazine can be carried on-board to correct these perturbations.

For minimizing effects of gravity gradient torques and the disturbances associated with an all-thruster attitude control system, as is the case here, attitude "B" is superior to "A". Both attitudes have low average gravity gradient torques, however, peak torque for "A" is more than 2.5 times greater than "B".

The final conclusion is that either attitude will be successful for the COLD-SAT mission. However, attitude "A" has lower risk, is preferable from a thermal standpoint and has, therefore, been selected for the COLD-SAT in-house design. Final attitude selection is a compromise between minimizing gravity gradient disturbance torques, minimizing orbital perturbations due to experimental thrusting and minimizing incident solar flux on the experiment tankage.

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TABLE 1. - DESCRIPTION OF PRESSURE VESSEL PENETRATIONS AND LOCATIONS

	Penetrations	A/L,[a] ft	Forward dome	Barrel	Aft dome
Struts	8, S-2 fiberglass epoxy struts	7.696×10^{-3} each strut	------------	X	------------
Wiring	514, temperature and liquid vapor instrumentation, 26 G manganin wires, grouped in 11 harnesses of 47 wires each	6.489×10^{-5} each harness	X (4 harnesses)	X (3 harnesses)	X (4 harnesses)
	54, valve and mixer motor power, 24 G aluminum wires, grouped in 2 harnesses of 22 and 32 wires, respectively	4.843×10^{-5} and 7.044×10^{-5}	X (1 harness 22 wires)	------------	X (1 harness 32 wires)
	8, pressure vessel heater, 22 G aluminum wires, grouped in 1 harness	2.806×10^{-5}	------------	X (1 harness)	------------
Plumbing	1 in., SS304 vent line	2.569×10^{-4}	X	------------	------------
	1 in., SS304 relief line	2.569×10^{-4}	X	------------	------------
	1/2 in., SS304 pressurization line	1.042×10^{-4}	X	------------	------------
	5/8 in., SS304 refill line	1.319×10^{-4}	X	------------	------------
	5/8 in., SS304 fill/drain/ transfer line	1.319×10^{-4}	------------	------------	X
	3, 1/4 in., SS304 thermo-dynamic vent system lines	5.555×10^{-5} each line	------------	------------	X

[a]From honeycomb MLI can outer facesheet to pressure vessel.

TABLE 2. - SUPPLY TANK LONGERON AND PLUMBING TRAY TEMPERATURES, °F

Component	Attitude A, $\beta = 0°$	Attitude A, $\beta = 41°$	Attitude B, $\beta = 41°$	Attitude B, $\beta = 0°$
Longerons (average of 8)	-110	-44	-28	-8
Plumbing tray	-110	-106	-87	-71

TABLE 3. - NODAL HEAT FLUXES ON PRESSURE VESSEL WALL, Btu/hr-ft^2

Location	Node	Attitude A, β = 0°		Attitude A, β = 41°		Attitude B, β = 41°		Attitude B, β = 0°	
		Actual	Normalized[a]	Actual	Normalized	Actual	Normalized	Actual	Normalized
Forward dome	1	0.0641	1.146	0.0821	1.127	0.0923	1.117	0.1004	1.126
	2	.0595	1.064	.0767	1.053	.0868	1.050	.0945	1.060
	3	.0565	1.010	.0751	1.031	.0855	1.035	.0921	1.033
	4	.0587	1.050	.0776	1.065	.0882	1.067	.0954	1.070
Barrel	5	0.0528	0.994	0.0673	0.924	0.0759	0.918	0.0824	0.924
	6	.0537	.960	.0684	.939	.0775	.938	.0839	.941
	7	.0515	.921	.0687	.943	.0786	.951	.0835	.937
	8	.0558	.998	.0735	1.009	.0838	1.014	.0896	1.005
Aft dome	9	0.0610	1.091	0.0870	1.195	0.0924	1.118	0.0988	1.108
	10	.0567	1.014	.0730	1.002	.0828	1.002	.0900	1.010
	11	.0579	1.035	.0761	1.045	.0868	1.050	.0936	1.050
	12	.0560	1.001	.0740	1.016	.0845	1.023	.0908	1.019
Total pressure vessel	Total of 12 nodes	0.0559	1.000	0.0728	1.000	0.0826	1.000	0.0891	1.000

[a]Data normalized by total flux on pressure vessel for each attitude.

TABLE 4. - PRESSURE VESSEL[a] HEAT LEAK SOURCE SUMMARY, Btu/hr

Spacecraft attitude	Struts	Plumbing	Wiring	Radiation from honeycomb	Total conduction	Total radiation	Total
A, β = 0°	0.218	0.099	0.212	7.460	0.529	7.460	7.989
A, β = 41°	.246	.116	.246	9.788	.608	9.788	10.396
B, β = 41°	.275	.125	.263	11.145	.664	11.145	11.809
B, β = 0°	.284	.130	.274	12.018	.688	12.018	12.706

[a]Pressure vessel surface area = 142.9 ft^2.

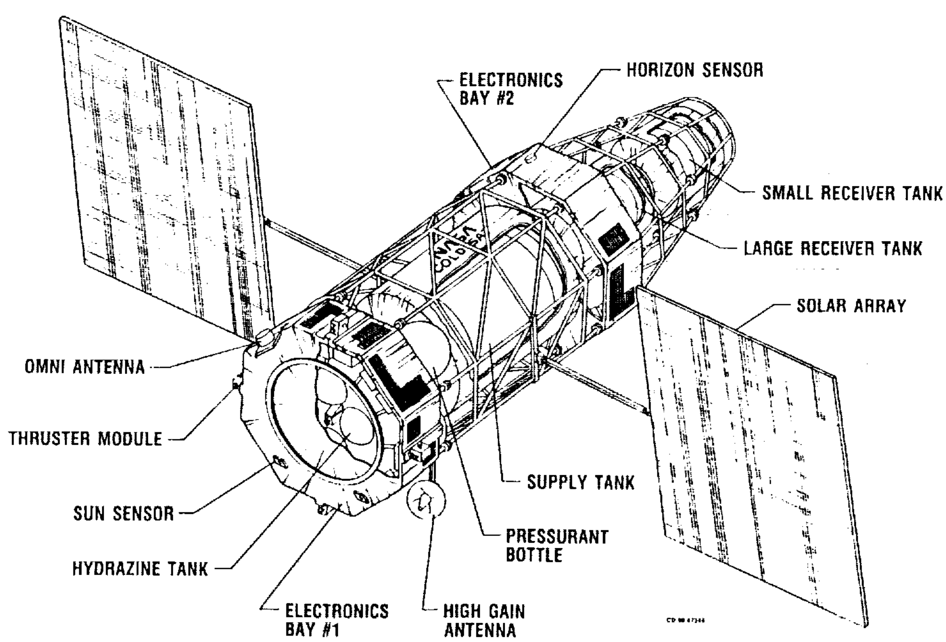


FIGURE 1. - CRYOGENIC ON-ORBIT LIQUID DEPOT-STORAGE, ACQUISITION, TRANSFER (COLD-SAT) SPACECRAFT.

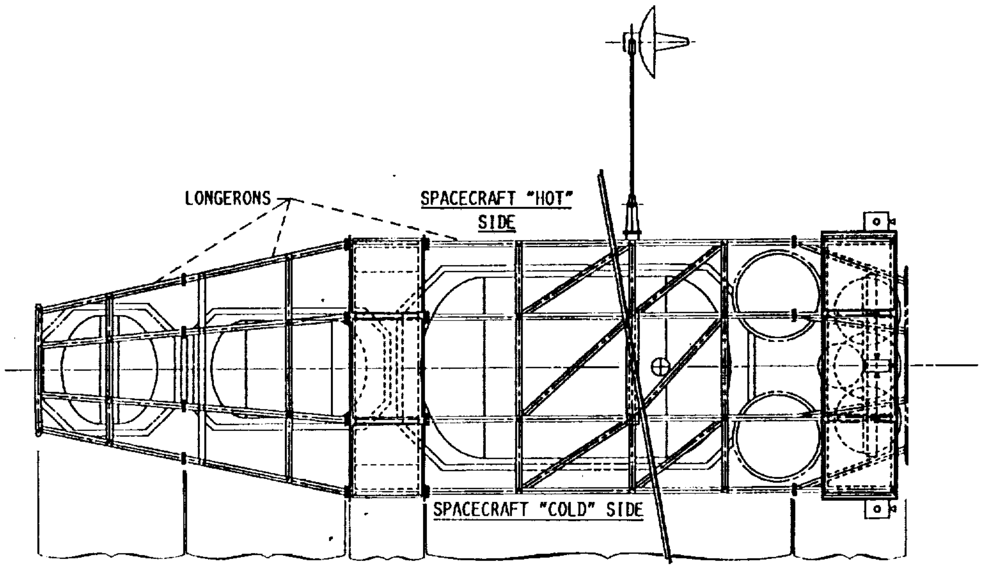
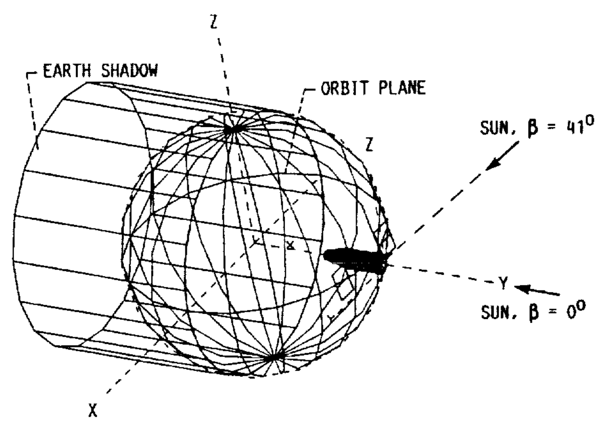
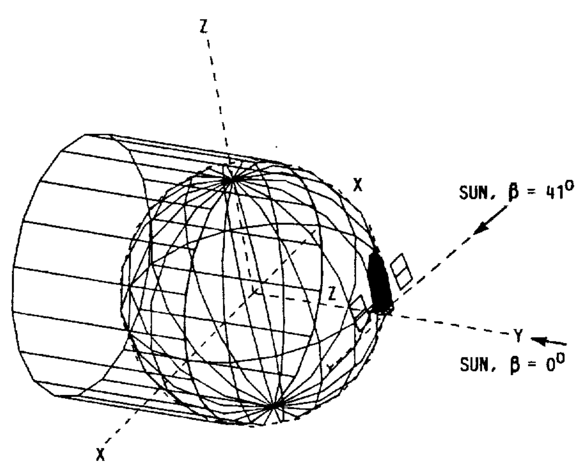


FIGURE 2. - COLD-SAT SPACECRAFT MODULES.



SPACECRAFT ATTITUDE "A"



SPACECRAFT ATTITUDE "B"

FIGURE 3. - COLD-SAT SPACECRAFT ATTITUDES CONSIDERED IN CIRCULAR LOW EARTH ORBIT.

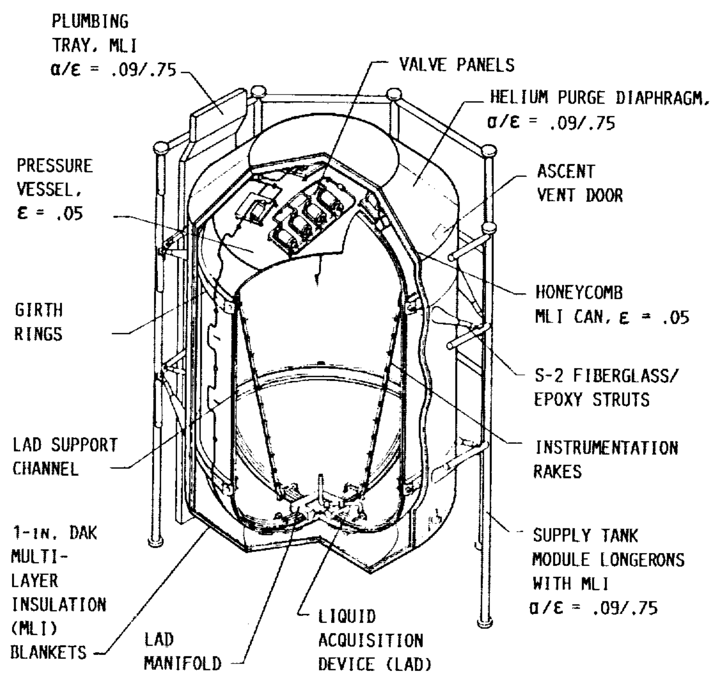


FIGURE 4. - COLD-SAT LIQUID HYDROGEN SUPPLY TANK MODULE.

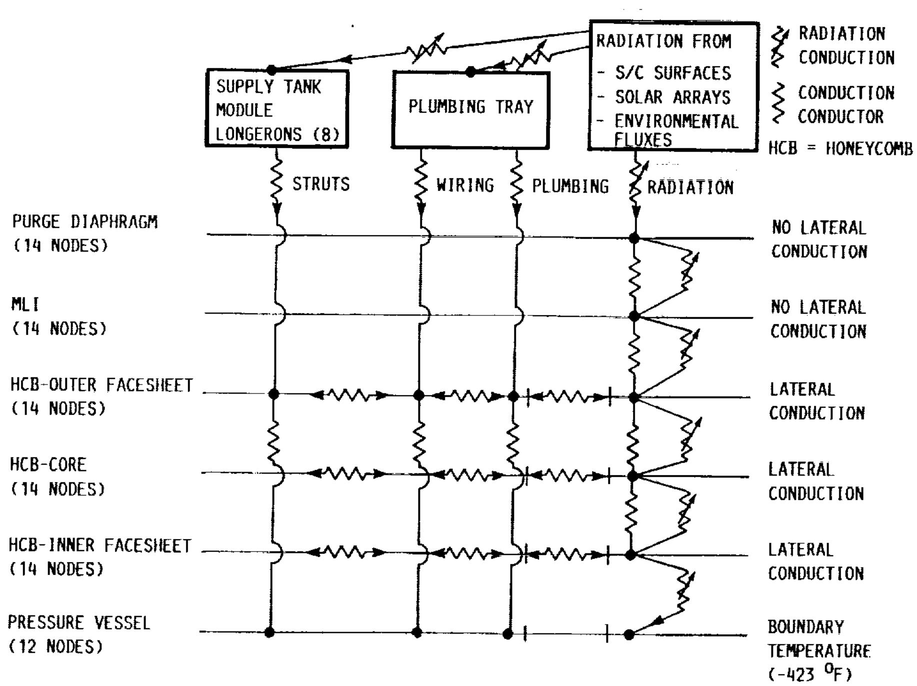


FIGURE 5. 3-D THERMAL NETWORK FOR SUPPLY TANK ANALYSIS.

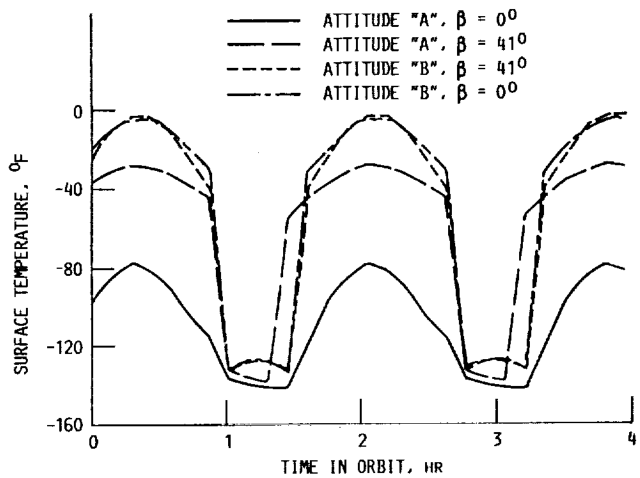


FIGURE 6. - NODAL AREA-WEIGHTED SUPPLY TANK PURGE DIAPHRAGM AVERAGE SURFACE TEMPERATURE.

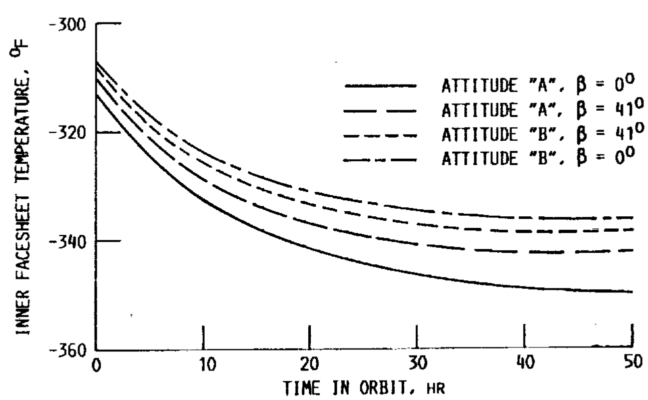


FIGURE 7. - HONEYCOMB NODAL AREA-WEIGHTED AVERAGE TRANSIENT TEMPERATURE INITIATING FROM STEADY-STATE.

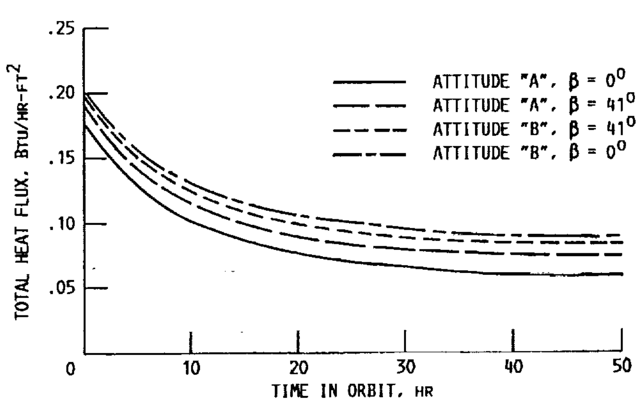


FIGURE 8. - TRANSIENT TOTAL HEAT FLUX ON PRESSURE VESSEL INITIATING FROM STEADY-STATE.

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16. Abstract

For determining the optimum on-orbit attitude for the COLD-SAT cryogenic experimentation satellite, a comparative analytical study ~~has been~~ performed to determine the thermal impacts of spacecraft attitude on the performance of the COLD-SAT non-vacuum jacketed LH_2 Supply Tank. Tank thermal performance ~~has been~~ quantified by total conductive and radiative heat leakage into the pressure vessel due to the absorbed solar, earth albedo and infra-red on-orbit fluxes, and also by the uniformity of the variation of this leakage on the vessel surface area. Geometric and thermal analysis math models ~~have been~~ developed for the spacecraft and the tank as part of this analysis, based on their individual thermal/structural designs. Two quasi-inertial spacecraft attitudes ~~have been~~ investigated and their effects on the tank performance compared. The results ~~of this study~~ are one of the criteria by which the spacecraft orientation in orbit ~~has been~~ selected for the in-house NASA Lewis Research Center design.

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