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SPACE STORABLE PROPELLANT MODULE THERMAL CONTROL TECHNOLOGY

SUMMARY REPORT

VOLUME II

F_2/N_2H_4 PROPULSION MODULE

CONTRACT NAS7-750

REPORT NO. 14051-6009-R0-00

15 MARCH 1971

Submitted to
JET PROPULSION LABORATORY
PASADENA, CALIFORNIA

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

JET PROPULSION LABORATORY

Under Contract 7-750

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CONTENTS

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O. Haroldsen
Project Manager

SUMMARY

Thermal control of the F_2/N_2H_4 propulsion module is difficult but can be reliably accomplished. Liquid fluorine cannot radiate appreciable heat to space because of its low temperature. Thus, the fluorine must be isolated from all sources of heat. In contrast, the N_2H_4 must obtain heat from the RTG to prevent freezing.

This study has established a general design which meets the divergent requirements of the propellants. It uses the same general structural configuration established for the OF_2/B_2H_6 module (Volume I), except that non-metallic structural members are used in most places to prevent conduction of heat to the fluorine.

A single tank for each propellant and the pressurant is provided. The fluorine tank (and helium tank if the helium is stored cold) is insulated with 2-in. thick non-porous foam, and the N_2H_4 tank is insulated with multilayer, aluminized Mylar. Also structural members and propulsion components are insulated as necessary to maintain their required temperatures.

During groundhold, fluorine tank (and possibly helium tank) conditioning is achieved by circulating LN_2 or chilled helium through cooling coils mounted inside the tank.

During flight, fluorine temperature control is effected by allowing the tank to radiate heat to space. However, to keep the required radiated heat rate low enough so that excessive fluorine temperatures are not required, a radiation barrier between the F_2 tank and the N_2H_4 tank is used. Because of the heat capacitance of the fluorine, it may be exposed to direct solar radiation for limited durations. To increase the allowable duration of solar exposure, second surface silvered Teflon covers those areas which may be exposed to the sun. Still, the mission must be tailored to keep the fluorine tank shaded by the spacecraft from continuous solar heating until the 350th day after launch.

Control of the N_2H_4 tank temperature during flight is effected by obtaining heat from the RTG and radiating excess heat to space by means of louvers mounted on the tank surface.

The study has also established that effective thermal control over the engine and its related components may be maintained by providing the proper conductive and radiative heat transfer paths between the equipment itself and between the equipment and the RTG.

Even though this investigation has established that a reliable thermal control system for a F_2/N_2H_4 propulsion module may be designed and fabricated using state-of-the-art concepts and equipment, additional in-depth studies must be made before the complete thermal control system may be specified. The required areas of study are as follows:

- Groundhold Coolant System. The quantity and type of coolant (LN_2 or gaseous He) needed cannot be specifically established until more information is available concerning the manner in which the module is to be handled during groundhold and the required F_2 launch temperature. When this information becomes

available, a study must be initiated which considers the coolant requirements in the light of module handling requirements during ground transit, module-booster mating, and launch sequence.

- N_2H_4 – RTG Conductor. The manner in which heat should be obtained from the RTG has not been definitely established. Initially, this study should consider the possibility of locating the RTG sufficiently close to the module that the necessary heat may be obtained by radiation. If this proves unwise because of hard radiation effects upon the module, the study should be directed to establishing the relative merits of a heat pipe versus a solid conductor as a means of obtaining the required heat. Preliminary studies discussed herein indicate that a heat pipe may be used with a substantial advantage. However, additional study is needed to establish the required design for such a heat pipe and the relative advantages and disadvantages of a heat pipe when compared with a solid bar conductor (or possibly a fluid loop) in the area of weight reliability and operational characteristics.

- Thermal Control of Engine Components. Before a complete thermal control system for a F_2/N_2H_4 module can be specified, the optimum helium storage temperature and the engine component (valves, regulators, etc.) temperatures must be established. It may be stated now that the helium may be stored warm or cold (or warm and cold by providing two helium storage tanks), and the engine component temperatures can be adjusted to meet propulsion requirements. Once sufficient propulsion design criteria is available in this area, a study should be initiated, preferably in concert with the propulsion system design, which has as its objective the detail design of the thermal control system in the local vicinity of the engine.

- RTG Interface Requirements. It was not within the scope of the present study to consider possible thermal problems associated with the RTG. During the study however, it became clear that even though the RTG serves as a convenient heat source, care must be exercised so that a RTG malfunction is not caused by localized RTG temperature variations. Such temperature variations could be caused by the manner in which the N_2H_4 – RTG conductor is attached to the RTG. Also, the variations in the radiated heat from the module could conceivably cause RTG problems, particularly if the RTG is located close (2 or 3 ft) to the module. Therefore, a study of the RTG which considers the RTG operational characteristics in conjunction with the operation of the thermal control system should be initiated.

NOMENCLATURE

A	Area, ft^2
c, c_p	Specific heat, Btu/lb-^{o}R
D	Diameter, ft
F	Thrust, lb
G	Solar constant, 430 Btu/ft^2-hr
g	Gravitational acceleration constant, ft/hr^2
h_i	Internal film coefficient, Btu/ft^2-^{o}R-hr
h_o	External film coefficient, Btu/ft^2-^{o}R-hr
I_s	Specific impulse, 1/sec
k	Thermal conductivity, Btu/ft^2-hr-^{o}R/ft
ℓ	Thickness or length, ft
M	Mass of propellant or helium, lb
q	Heat transfer, Btu
R	Thermal resistance, hr-^{o}R/ft
ΔT	Temperature, ^{o}R
ΔV	Velocity, ft/hr
w	Mass, lb
T	Temperature difference, ^{o}R
V	Velocity change, ft/hr
α_s	Solar absorptivity
β	Coefficient of volumetric expansion, 1/^{o}R
ε	Emissivity
ρ	Density, lb/ft^3
μ	Viscosity, lb/ft-hr
θ	Off-pointing angle, degrees

1. INTRODUCTION

Volume I of the final report presents the results of a study of thermal control systems for a Jupiter spacecraft propulsion module which uses OF_2/B_2H_6 as the propellants. This volume presents similar information except for a propulsion module which uses F_2 and N_2H_4 as the propellants. The work was divided into Task VI and Task VII (the first five tasks were concerned with the OF_2/B_2H_6 module):

Task VI Perform sufficient conceptual design analysis to establish at least three thermal control schemes applicable to F_2/N_2H_4 propellant module.

Task VII Perform sufficient design and analytical studies of the concepts (from Task VI) to select the most promising thermal control design concept.

The manner in which this study was conducted was identical to that of the OF_2/B_2H_6 study. Within each task, the work was broken into four areas (Figure 1-1). Each area was independent of the other three in requirements and objectives. The conclusions from each area were combined and a composite system evaluation made to establish advantages and disadvantages of each aspect of the study. The final thermal control design concepts were chosen by a reiteration of this process as the tasks were performed.

The F_2/N_2H_4 thermal control study was initiated after most of the OF_2/B_2H_6 work had been completed. This was done so that the experience and knowledge gained from the OF_2/B_2H_6 study might be applied to the F_2/N_2H_4 study and thereby conserve funds. In this respect, the approach was highly successful for, as it will be noted later, much of the F_2/B_2H_6 work in the area of module structural design and groundhold thermal control was directly applicable.

Within the level of funding, it was not intended that the F_2/N_2H_4 study would go to the depth of the OF_2/B_2H_6 propulsion module study. Rather the most promising thermal control concepts were to be established if possible and, just as important, problem areas requiring further study were to be established.

Task reports have been issued summarizing the work of each task. This volume reports the work of both Tasks VI and VII. To give a complete picture, Section 2 describes the various phases of the Jupiter mission and the restraints imposed by the mission. Section 3 lists the study design criteria for the F_2/N_2H_4 module.

Section 4 describes in detail the module design which appears best at this time. This section also presents the rationale used to establish the design. Section 5 discusses the thermal analysis in detail, listing all areas of investigation covered during the tasks. Section 6 discusses the propulsion system operation to the extent that it is understood at this time. Unfortunately, time and funding did not allow a total investigation of this phase of the problem.

Section 7 summarizes all the work relative to the F_2/N_2H_4 propulsion module and lists areas which require further study in order to completely specify the module design.

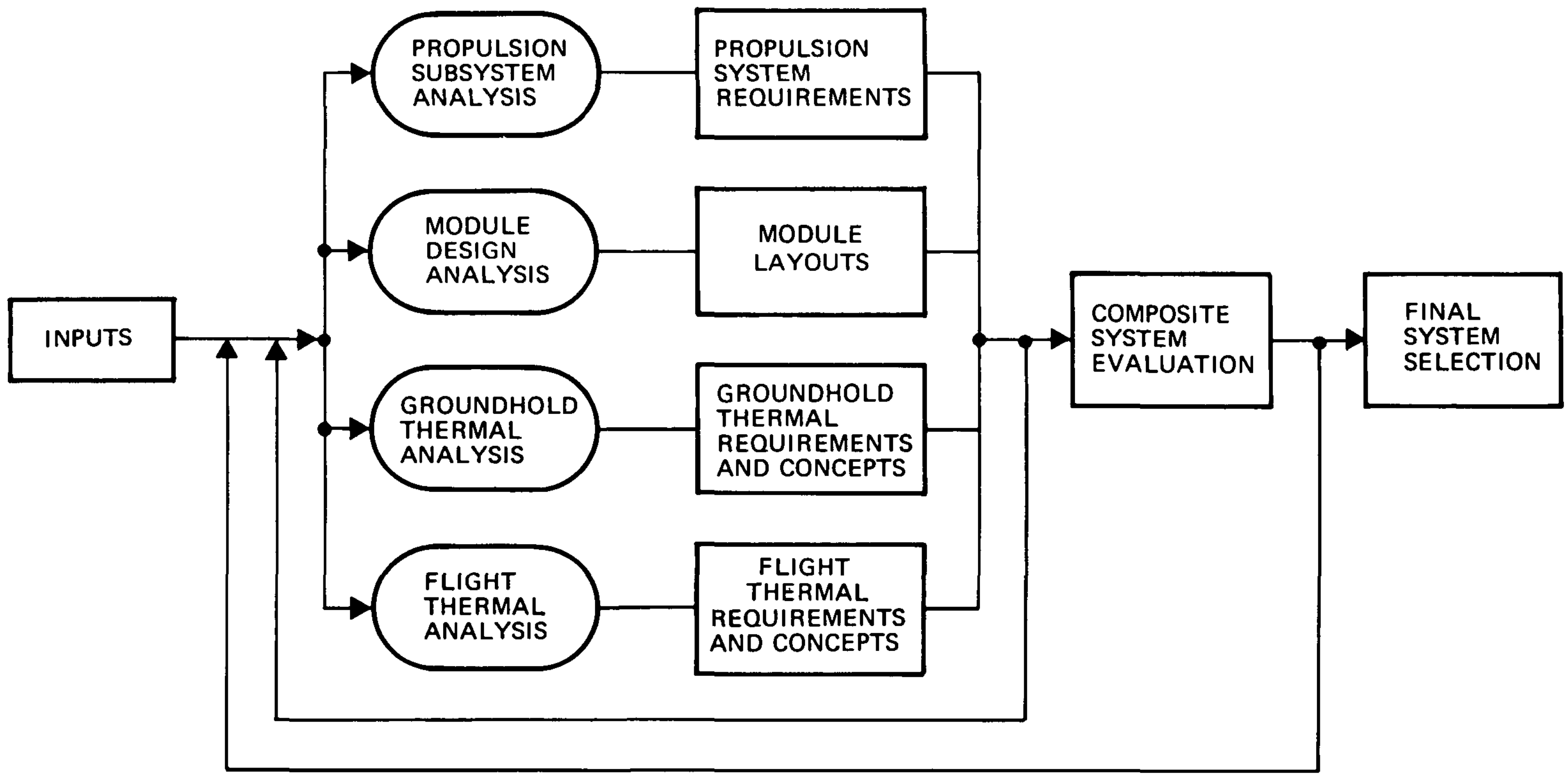


Figure 1-1. Information Flow Diagram

2. MODULE MISSION

The module mission consists of four phases: groundhold, launch and parking orbit, Jupiter transfer, and Jupiter orbit. During these phases, the propulsion module must be maintained within design temperature limits.

2.1 GROUNDHOLD

Groundhold, the period from initiation of passivation to lift-off, may last up to one month. During this phase, stage conditioning (holding propellant temperatures within allowable limits) must be maintained at all times. Proper propellant conditioning eliminates the need to vent or top the propellant systems. Additionally, frost accumulation on any flight hardware must be prevented. Module components not in contact with cold fluids, however, do not require conditioning during groundhold.

For the purposes of this study, it was assumed that the propellants are tanked prior to mating the module with the boost vehicle on stand. It was also assumed that, after tanking, the shroud may or may not be in place. The module will be mated to a high-energy Titan-Centaur kick stage, (fueled with liquid hydrogen any time after 12 hours before launch). It was assumed that a diaphragm positioned between the kick stage and the propulsion module will prevent convective heat transfer.

2.2 LAUNCH AND PARKING ORBIT

The vehicle will be launched into a 100-nautical-mile parking orbit by the Centaur stage. Maximum coast time in parking orbit will be one hour. For this study it was assumed that there are no restrictions as to the time of launch, that the protective shroud is to be jettisoned at an altitude of approximately 225,000 feet, and the sun exposure of the propulsion module during the launch and parking orbit is random.

2.3 JUPITER TRANSFER

During the transfer phase, the period (756 days) between ejection from the earth parking orbit to injection into the Jupiter orbit, three mid-course correction firings, with an aggregate firing time of about 40 seconds, are permitted. For this study, however, one firing on the seventh day for a 100 m/sec (meters per second) trajectory correction was assumed.

The amount of solar heating of the module during transfer is determined by spacecraft geometry construction, orientation and distance from the sun. Figure 2.3-1 is a graph of expected solar flux as a function of mission time; Figure 2.3-2 shows off-pointing angle (orientation) of the module during the mission. ("Off-pointing" refers to the angle between the module's Z-axis and the solar vector, 0° being the normal case of the module totally shaded by the spacecraft.)

A single Jupiter orbit insertion diring for a 1460 m/sec trajectory change was also assumed in this study.

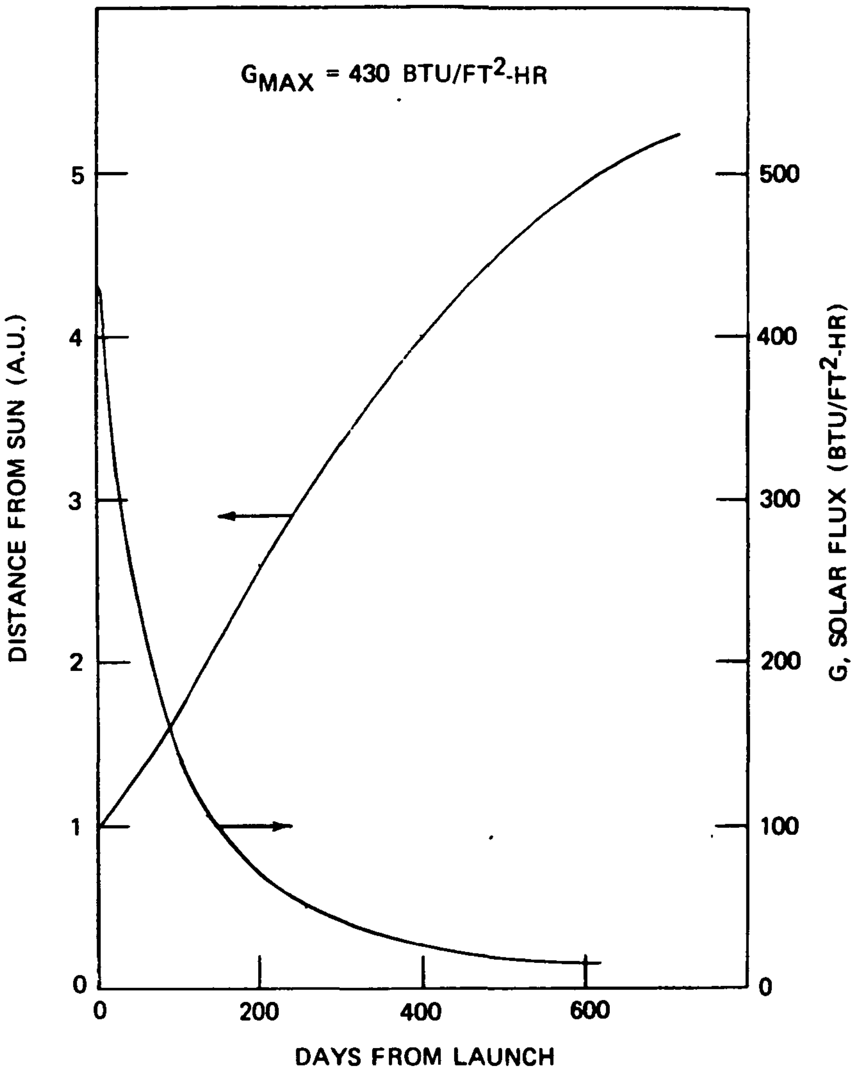


Figure 2.3-1. Solar Flux at Spacecraft

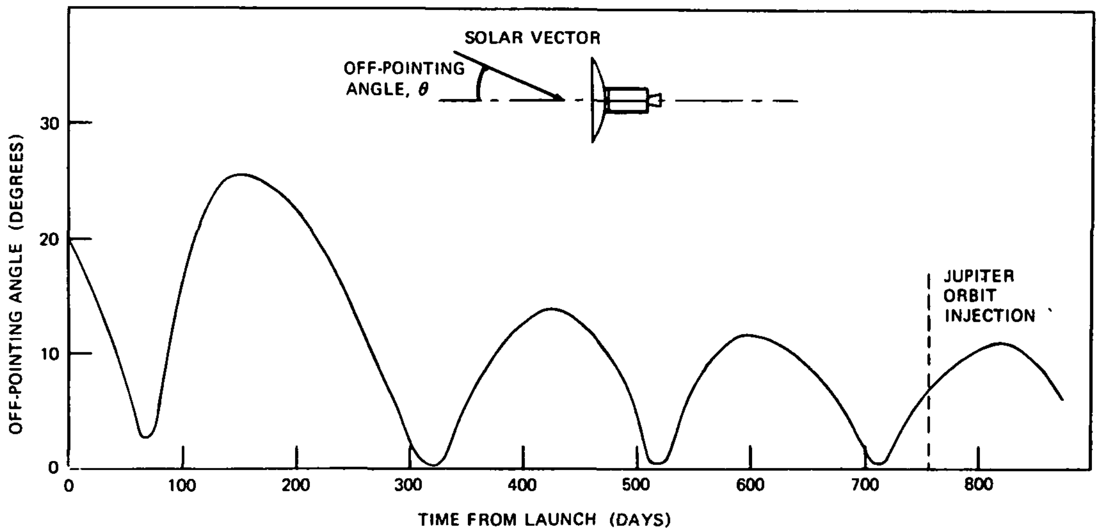


Figure 2.3-2. Off-pointing Angle During Mission

2.4 JUPITER ORBIT

A final firing to correct the Jupiter orbit inclination angle will occur at an unspecified date 23 days after Jupiter encounter (773 days after launch). Engine operation time must be sufficient for a trajectory correction of 2320 m/sec. The initial Jupiter orbit (a period of approximately 45.4 days) will be 4 x 98.8 R inclined to Jupiter's equator at an angle of less than one degree.

3. FLUORINE/HYDRAZINE (F_2/N_2H_4) PROPULSION MODULE DESIGN GUIDELINES

The basic design guidelines for the F_2/N_2H_4 propulsion module were the same as for the OF_2/B_2H_6 propulsion module. The mission is as described in Section 2 and the gross lift-off weight is 4400 lb for both module and spacecraft.

The propulsion system contained within the module was to be a pressure-fed system connected to a bi-mode engine. The schematic diagram of the propulsion system is given in Figure 3-1 and Table 3-1 lists the initial physical data for the system.

The bi-mode engine can be operated either in a monopropellant (hydrazine) mode or in a bipropellant mode. In the former case, the hydrazine flow is decomposed in an auxiliary chamber packed with Shell 405 catalyst. The resultant hot, gaseous products flow into the main combustion chamber where they are mixed with fluorine during bipropellant operation.

The proposed operational sequence uses the monopropellant mode for all mid-course trajectory correction maneuvers, and for the first and last two seconds of the orbit insertion and orbit inclination change maneuvers. From two seconds after start until two seconds before shut-down during the latter maneuvers, the engine will be operated in the bipropellant mode.

The initial guidelines assumed the helium to be stored cold (approximately $180^{\circ}R$). The plumbing schematic reflects this approach by showing a heat exchanger in the fuel line. As will be discussed later, this concept is questionable.

Table 3-1. Propulsion System Design Guidelines

Mixture ratio	2.0
Chamber pressure	100 psia
I_s	385
Thrust	1000 lb_f
Propellant temperature limits	
fuel	$530 \pm {}^{20}_{30}\ {}^{\circ}R$
oxidizer	$155 \pm {}^{25}_{55}\ {}^{\circ}R$
Pressurant	Helium
Pressurant initial pressure	4000 psia @ $180^{\circ}R$
Propellant and helium tanks	Boron filament wound with 0.010 aluminum liner
Propellant tank volumes (equal volumes)	1.1 x propellant volume
Propellant tank pressure	300 psia
Oxidizer mass	1735 + 73 (residuals) lb_m
Fuel mass	984 + 37 (residuals) lb_m
Pressurant mass	36 lb_m

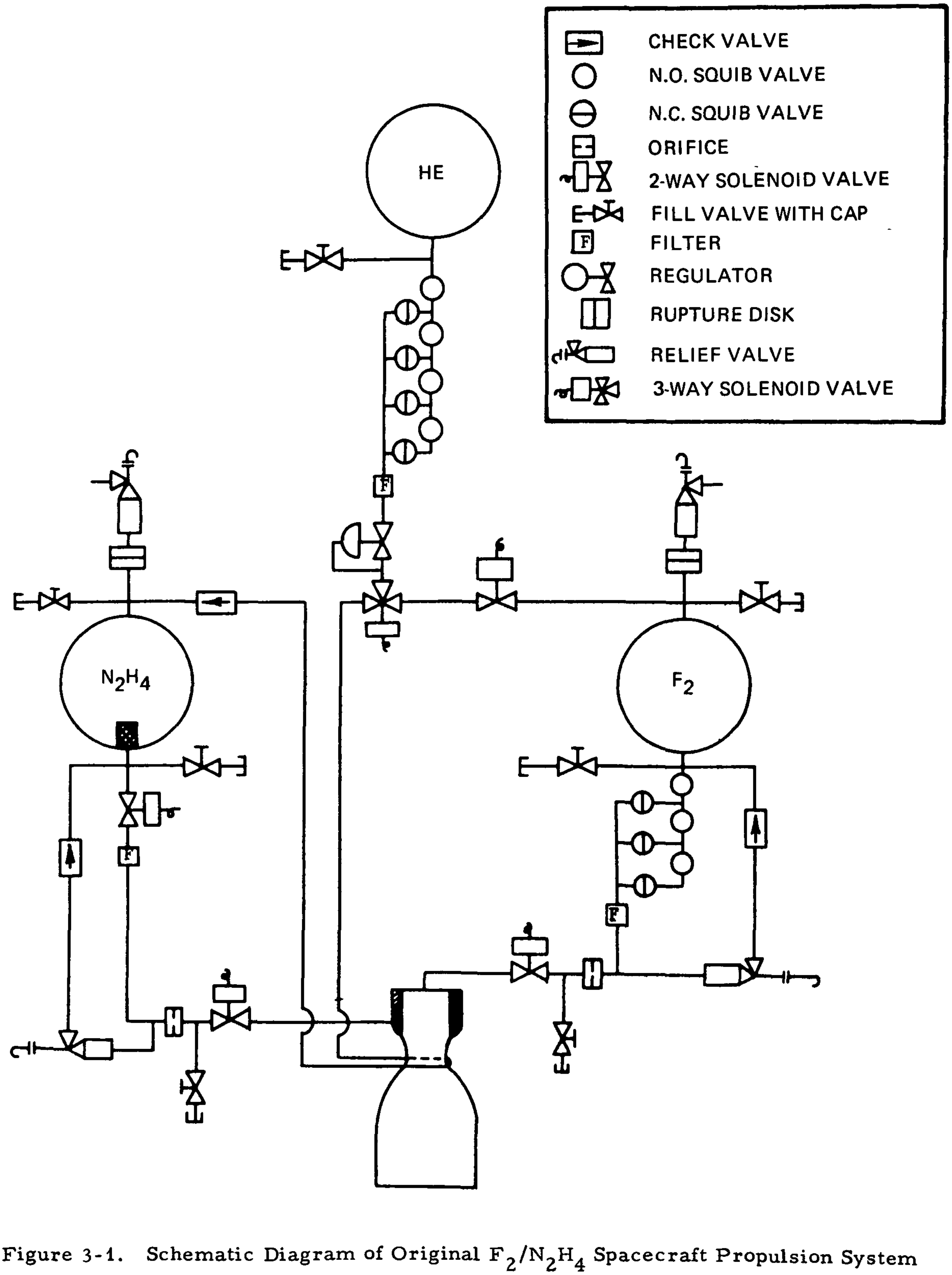


Figure 3-1. Schematic Diagram of Original F_2/N_2H_4 Spacecraft Propulsion System

4. F_2/N_2H_4 PROPULSION MODULE FINAL SYSTEM DESIGN

4.1 STRUCTURAL LAYOUT

The module is an inert-gas-pressurized bi-mode propulsion system which depends upon constancy of tank pressure, as established by a gas pressure regulator and calibrated flow resistances, to achieve the desired flow rates. Initiation and termination of all operations are by electric signals to pyrotechnic or solenoid valves which provide on-off control of the gas flow. Helium gas, stored at high pressure, is used to pressurize the propellants.

The propulsion system is comprised of three tanks (two contain propellants, the third helium), an engine, and plumbing (see drawing SK 406922). The propellant tanks, constructed of boron filament and lined with 0.010-in. aluminum, are 32.8 in. in diameter and have hemispherical ends with an eccentricity of 0.784. Supporting attachments provide axial restraint at the bottom and shear restraint at both top and bottom. Universal joints are used at both ends of each tank.

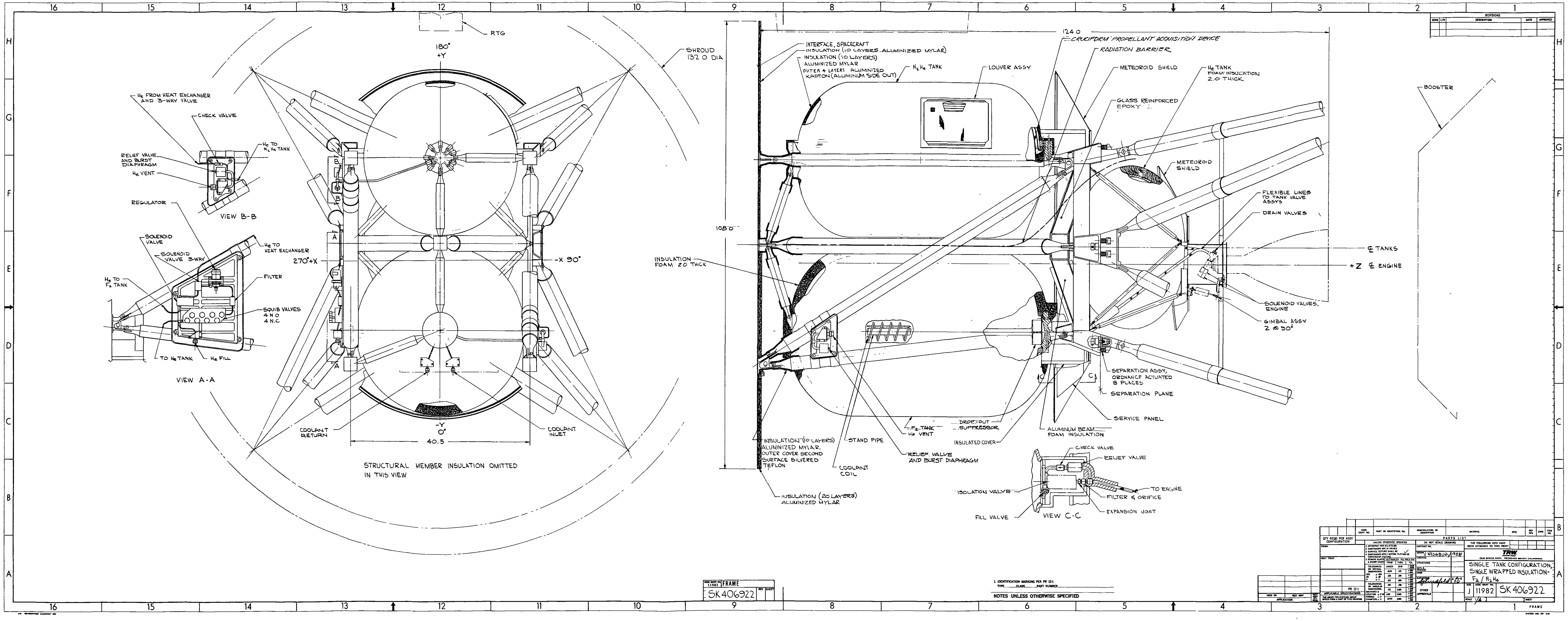
The fluorine tank is fitted with a groundhold heat exchanger. The heat exchanger is an aluminum tube, 8 ft long and 1/2 in. in diameter, formed into a coil approximately 5 in. in diameter. For propellant cooling purposes, LN_2 or chilled helium will pass through the coil during groundhold.

The helium tank, also constructed of boron filament, is suspended at the top by an aluminum cross beam and is laterally stabilized at the bottom by boron filament tubes that extend from the tank to the engine support frame. The helium tank has a cooling coil similar to that installed in the fluorine tanks.

Except for weight penalty, no technical reason prohibits replacement of the boron filament tanks with metal tanks. The thermal characteristics of the module would be the same.

A space truss structure, predominately of non-metallic members supports the propulsion hardware and also the spacecraft. The entire module and spacecraft when attached to the boost vehicle are supported by 16 boron-filament tubular struts with aluminum fittings. These struts remain with the boost vehicle upon separation. The separation fittings (pyrotechnic devices) are located on the main platform.

This general configuration is the same as that chosen for the OF_2/B_2H_6 propulsion module. The reasons for selecting this configuration were discussed in Volume I, Sections 4 and 5, and are still applicable for the F_2/N_2H_4 system, even though the details have been modified to some extent to accommodate the difference in basic requirements. One of the factors that dictated a small change in truss geometry is the smaller tank size permitted by the decreased volume of propellant. The modifications resulting from this factor were to make the lower platform smaller and to move the


RTG
SHROUD
132.0 DIA
180°
+Y
270° +X
-X 90°
-Y
0°
40.5
COOLANT RETURN
COOLANT INLET
STRUCTURAL MEMBER INSULATION OMITTED IN THIS VIEW
He FROM HEAT EXCHANGER AND 3-WAY VALVE
CHECK VALVE
He TO N2H4 TANK
RELIEF VALVE AND BURST DIAPHRAGM
He VENT
VIEW B-B
REGULATOR
SOLENOID VALVE
SOLENOID VALVE 3-WAY
He TO HEAT EXCHANGER
He TO F2 TANK
FILTER
SQUIB VALVES 4 N.O. 4 N.C.
TO He TANK
He FILL
VIEW A-A
INTERFACE, SPACECRAFT
INSULATION (10 LAYERS, ALUMINIZED MYLAR)
INSULATION (10 LAYERS) ALUMINIZED MYLAR OUTER 4 LAYERS ALUMINIZED KAPTON (ALUMINUM SIDE OUT)
N2H4 TANK
LOUVER ASSY
124.0
CRUCIFORM PROPELLANT ACQUISITION DEVICE
RADIATION BARRIER
METEOROID SHIELD
He TANK FOAM INSULATION 2.0 THICK
GLASS REINFORCED EPOXY
METEOROID SHIELD
BOOSTER
108.0
INSULATION FOAM 2.0 THICK
FLEXIBLE LINES TO TANK VALVE ASSYS
DRAIN VALVES
℄ TANKS
+Z ℄ ENGINE
SOLENOID VALVES, ENGINE
GIMBAL ASSY Z ± 30°
SEPARATION ASSY, ORDNANCE ACTUATED 8 PLACES
SEPARATION PLANE
SERVICE PANEL
ALUMINUM BEAM FOAM INSULATION
INSULATED COVER
DROPOUT SUPPRESSOR
F2 TANK He VENT
STAND PIPE
RELIEF VALVE AND BURST DIAPHRAGM
COOLANT COIL
INSULATION (10 LAYERS) ALUMINIZED MYLAR. OUTER COVER SECOND SURFACE SILVERED TEFLON
INSULATION (20 LAYERS) ALUMINIZED MYLAR
CHECK VALVE
RELIEF VALVE
TO ENGINE
ISOLATION VALVE
FILTER & ORIFICE
EXPANSION JOINT
FILL VALVE
VIEW C-C
1. IDENTIFICATION MARKING PER PR 12-1
NOTES UNLESS OTHERWISE SPECIFIED
PARTS LIST
DO NOT SCALE DRAWING
DRAWN J WEDGBURY 18 MAY 70
TRW
ONE SPACE PARK REDONDO BEACH, CALIFORNIA
SINGLE TANK CONFIGURATION, SINGLE WRAPPED INSULATION- F2 / N2H4
SIZE J
CODE IDENT NO. 11982
SK406922
SCALE 1/4
FRAME

truss intersection points closer together, since a closer truss arrangement can be used to encompass the tank envelope. To prevent lateral c. g. displacement, the F_2 and N_2H_4 tanks are positioned so their c. g. is along the center line of the spacecraft.

Thermal control considerations required the distance between the tanks and the electronics compartment to be increased. Because of this increased vertical distance between the top of the tanks and the attachment fittings of the truss to the spacecraft, a problem arose as to the best arrangement of the truss tubes used to transmit lateral tank loads to these attach fittings. The optimum arrangement for the tank is to locate the truss tubes horizontally and attach them close to the tank boss so that loads are transmitted tangentially into the tank shell. This design was used previously (Volume I, drawing 406876). However, in this case, the consequence would be to extend the attachment fitting below the spacecraft interface in order to mount the horizontal tubes; this would cause the fitting to be much heavier and would require more moment carrying structure in the spacecraft. If the tubes remained horizontal, but were moved closer to the spacecraft to reduce attachment fitting weight and spacecraft moment requirements, the boss on the tank would have to be extended upward for attachment to the truss tubes. The boss would then become unduly heavy and a large moment would be imposed on the tank. This situation is undesirable in a shell structure. The other alternative, and the one selected, is to run the truss tubes at an angle so that one end is attached close to the spacecraft interface and the other end close to the tank contour. In this way, moments at each end are minimized. Because of the angle, however, lateral tank loads induce axial forces on the tank. Although this is undesirable, the additional axial forces can be carried more efficiently than the moments that arise from other alternatives.

Thermal control requirements also dictated a change in the materials used for the tank support platform. Assuming both the F_2 and He tanks are maintained at cryogenic temperatures, the section of center beam between them is made of aluminum alloy to provide good thermal conductivity. The N_2H_4 tank, being at a much higher temperature, requires the utilization of low conductivity members between it and the cold tanks. These members consist of the section of the center beam which extends to the He tank and the edge members of the platform parallel to the center beam. They are now glass-reinforced plastic. Diagonal members of the platform are boron-filament tubes as in the previous design. A thermal radiation shield is now located between the F_2 and N_2H_4 tanks and extends down between the N_2H_4 and He tanks. The shield is light weight, aluminized Mylar, but requires a structural edge support to carry environmental loads. This support is not shown in the drawing, but is included in the weight estimate.

The design conditions and load factors used for structural design are the same as those used in the OF_2/B_2H_6 module design and are restated here for convenient reference.

Condition 1	6 g axial-tension	
2	8 g axial-compression	
3	± 3 g lateral	
	4 g axial-compression	
or	3 g axial-tension	

A factor of safety of 1.25 was used to obtain ultimate loads.

Based on these load factors and estimates of module component weights, the maximum load, size, and weight of each structural member was determined, Reference 1. That work is reproduced in Appendix A. Table 4.1-1 lists the structure weight along with other module component weights.

4.2 PROPULSION SYSTEM LAYOUT

Table 4.2-1 lists the physical data for the propulsion system in its final configuration. The propellant masses were calculated using the assumed specific impulse values and an initial injected mass of 4400 lb for the incremental velocity changes listed in Section 2. These velocity changes were converted to propellant consumption using the relationship:

$$\Delta M_p = M_o \left[1 - \frac{1}{\ell n^{-1} \frac{\Delta V}{g I_s}} \right]$$

One point should be clearly noted. In order to achieve the total ΔV performance specified with the I_s and total injected weight specified, 2903 lb of propellants are required. But for the module wieght indicated to hold within a total weight of 4400 lb with a 1200 lb payload necessitates off-loading the propellant tanks (10%) and accepting a reduction in capability (approximately 20%) during Jupiter orbit inclination firing.

Although the propellant flow rates are implied for the bipropellant mode (i.e., $\dot{M}_p = F/I_s$), the monopropellant flow rate is an unknown because it depends upon the monopropellant reactor and gas injector design (unspecified). However, if a flow rate of 1 lb/sec is assumed for the monopropellant mode and the delivered average specific impulse during mid-course maneuvers is 240 lb_f-sec/lb_m, the total maximum mid-course burning time is 180.15 sec. Orbit insertion burn time (total) would be 524.3 sec. and orbit inclination burn time (total) would be 484.6 sec.

The pressurization system is conventional in most respects. Helium filling is accomplished through a manual shutoff valve. The storage tank is an oblate spheroid constructed of aluminum and boron-epoxy composite (ribbon-wound). Nominal wall thickness is 0.400 in. The major external diameter is 32.5 in. and the external height is 20.4 in.

Table 4.1-1. Summary of Estimated Subsystem Weights

	Pounds (lb)
Tankage*	
1-Helium tank at 74.3 lb ea.	74.30
2-Propellant tanks at 54.13 lb ea.	108.26
1-Propellant acquisition device	2.00
	184.56
Liquid Circuit	
2-Fill valves at 1 lb ea.	2.0
2-Isolation valves at 4 lb ea.	8.0
2-Filters at 1 lb ea.	2.0
2-Relief modules at 1.2 lb ea.	2.4
2-Check valves at 1.0 lb ea.	2.0
	16.4
Gas Circuit	
1-Fill valve at 1 lb ea.	1.0
4-Pr. explosive valves at 3 lb ea.	12.0
1-Filter at 1 lb ea.	1.0
1-Regulator at 2 lb ea.	2.0
1-Check valve at 0.5 lb ea.	0.5
2-Relief modules (disc plus valve) at 1 lb ea.	2.0
2-Pressurization and vent valves at 2 lb ea.	4.0
2-Solenoid valves at 2 lb ea.	4.0
	26.5
Thrust Chamber Assembly	
2-Propellant valves at 5.0 lb ea.	10.0
2-Orifice assys, w/flanges at 0.5 lb ea.	1.0
2-Bleed valves at 1 lb ea.	2.0
1-Thrust chamber w/gimbal mounts	43.0
2-Gimbal actuators at 2.25 lb ea.	4.5
	60.5
Fluids	
Oxidizer (F_2)	1673.0
Fuel (N_2H_4)	1004.6
Helium (He)	36.0
	2713.6
Structure-Above Separation Plane*	
Upper truss members	22.02
Tank upper support member	1.37
Spacecraft attachment fittings	4.25
Platform members (frame)	8.63
Platform fittings	5.25
Engine support truss members	1.57
Engine support platform	2.75
Tank end fittings	2.70
Valve assembly brackets	6.76
Meteoroid shields	14.32
	69.61

*Weights listed assume boron filament construction. Structurally and thermally, the tanks could be titanium or aluminum and all struts could be made of fiberglass. The only consequence would be increased weight.

Table 4.1-1. Summary of Estimated Subsystem Weights (Continued)

Structure-Below Separation Plane*	
Truss members	45.96
Fittings (separation)	2.50
Stabilizing frame	1.00
	49.46
Miscellaneous	
Lines and fittings	20.0
Instrumentation	4.0
Command and squib harness	8.0
Contingency	16.0
	48.0
Insulation	
Aluminized Mylar (N_2H_4 Tank)	0.91
Foam (F_2 Tank)	15.40
Foam (He tank)	6.72
Foam (beams)	1.64
Aluminized Mylar and support (radiation barrier)	2.00
Louvers (N_2H_4 Tank)	2.25
Cooling coil assembly (F_2 and helium tanks)	1.50
	30.42
Total	
Stage weight	3149.6
Total weight (including structure below separation plane)	3199.0

*Weights listed assume boron filament construction. Structurally and thermally, the tanks could be titanium or aluminum and all struts could be made of fiberglasss. The only consequence would be increased weight.

Helium is carried from the tank to the valving package by a 1/4-in. tube. This tube connects to a series/parallel arrangement of normally-open and normally-closed, explosively-actuated valves. Downstream of this valve cluster is a filter which protects downstream components from particulate contamination. The regulator is set to control the outlet pressure at 300 psia.

A solenoid-operated, three-way isolation valve separates the fuel and oxidizer pressurization lines. From this common point, helium destined for the fuel tank flows through a 1/4-in. line to a heat exchanger, if the helium is stored cold. The location and source of heat for this heat exchanger is discussed below. A check valve in the fuel pressurization line downstream of the heat exchanger blocks any potential backflow of hydrazine vapor which might be heated and decomposed in the exchanger. Helium flow to the oxidizer tank is controlled by a two-way solenoid valve downstream of the common three-way valve.

Table 4.2-1. F_2/N_2H_4 Propulsion System Data

Engine - In bipropellant mode	
Mixture ratio	2 to 1 (O/F)*
Chamber pressure	100 psia*
Estimated specific impulse	385 lb_f-sec/lb_m
Thrust	1000 lb_f*
Oxidizer flow rate	1.7316 lb_m/sec
Fuel flow rate	0.8658 lb_m/sec
Engine - in monopropellant mode	
Estimated average specific impulse	240 lb_f-sec/lb_m
Estimated specific impulse at start	220 lb_f-sec/lb_m
Estimated thrust	240 lb_f
Estimated fuel flow rate	1 lb_m/sec
System	
Pressurant	Helium gas*
Pressurant mass	36 lb_m*
Initial pressurant storage pressure	4000 psia at 180°R*
Pressurant storage tank volume	5.95 cu ft.
Regulated pressure	300 psia*
Propellant tank volumes	22.9 cu ft (each)

*Specified by JPL.

Connected to the pressurant inlet port of each propellant tank is a relief module and manual pressurization valve. The former includes a burst disc and a relief valve in series with each other. The disc is included to assure a leak-tight seal unless venting is mandatory, in which case the resealable relief valve then assumes the valving function. Connection is provided between the ground system and the ullage by the manual pressurization valves so that inert gas purges, passivation fluids, propellant vapor, and pressurant gases may be pumped into and out of the propellant tanks.

The two propellant tanks are sized to 110% of the fluorine volume at 180°R. Using identical tanks for both oxidizer and fuel results in a rather large ullage in the fuel tank (24.7%). This presents an opportunity to perform the mid-course maneuver firings in a blowdown mode.

Structurally, the propellant tanks are similar to the helium tank—boron-epoxy, ribbon-wrapped over a 10-mil thick aluminum liner. They may be constructed entirely of metal, however. Geometrically, they are right cylinders with oblate hemispheroidal ends. The wall thickness is 0.120 in. in the cylindrical portion and 0.050 in. in the ends (including the liner).

Design details of the tank ports and internal equipment have not been determined.

A component module (cluster of valves, etc.) is connected to the feedline near the outlet port of each propellant tank. Within each module is an isolation (shutoff) valve, a filter, a return relief valve, and a return check valve. The isolation valve seals off tank contents from the feedline during all but pre-firing and engine firing periods. The return relief valve and check valve constitute a one-way, by-pass circuit around the isolation valve, so that propellant trapped between the isolation valve and the engine propellant valve after any firing can be vented back to the propellant tank in the event that heat, soaking back into the trapped propellant from the engine, causes a pressure rise due to bulk expansion. Also, a manual fill valve through which fluids are pumped into the propellant tank is connected to each tank outlet port by a 1/2-in. tube.

The 3/4-in. diameter feedlines to the engine are constructed of corrugated Inconel alloy hose and covered with a woven metal braid. These lengths of flexible hose provide the freedom of motion required when the gimbal-mounted engine is moved to alter the direction of thrust.

Calibrated orifices are necessary in each feedline to assure delivery of the rated mass flows of propellants. In the diagram, these orifices are shown at the connecting fittings between the hoses and the propellant valve inlets. Also indicated near these points are manual bleed valves which are necessary to permit flow through the feedlines down to the propellant valves during system passivation.

The propulsion system follows the schematic of Figure 3-1.

In order to keep the fluorine and helium cold during groundhold, each of these tanks is equipped with an internal cooling coil. Each coil is composed of an 8-ft section of 1/2-in. aluminum tubing formed into a coil 6-in. in diameter. To provide structural rigidity for the coil, a support member to secure the coil is required in each tank.

All of the above-mentioned items are called out in drawing SK 406922. Not shown on the drawing are the conductors needed for conducting heat from the RTG to the N_2H_2 tank. If heat pipes are used, it was assumed that the heat pipe system is composed of three, 3-ft rigid heat pipes 3/4-in. diameter, which are thermally connected in series and attached to the RTG at one end and the N_2H_4 tank at the other. The final analysis indicated the heat pipe system must transport about 175 Btu/hr.

It should be understood that the design concepts depicted in drawing SK 406922 represent the probable desired configuration as determined by the analysis reported here. As will be indicated below, other configurations were considered in the analysis.

4.3 THERMAL CONTROL LAYOUT

The thermal control system, as finally devised for the module, consists of multilayer, aluminized Mylar insulation blankets, foam insulation, a louver assembly, second-surface silvered Teflon, cooling coils, and heat pipes. As will be indicated later, additional analysis is needed concerning the heat pipes.

Aluminized Mylar blankets are used to insulate the N_2H_4 tank, portions of the hardware which attach to the N_2H_2 tank, the bottom surface of the spacecraft, and the heat pipes which thermally connect the RTG and the N_2H_2 tank. Another blanket serves as a radiation barrier between the hot and cold tanks. In all cases, these blankets are constructed with the Mylar side facing out to space. All blankets are composed of 10 layers of Mylar except for that portion of the spacecraft insulation on the fluorine tank side of the barrier which is composed of 20 layers of Mylar.

The fluorine tank, helium tank, and all hardware which contacts these tanks are covered with two inches of closed-cell, 2-lb density foam (Reference 2). Non-metallic members in contact with the tanks are foam covered 6 to 8 in. from the point of contact. Metallic members are foamed at least 18 in. from the point of contact.

A 4-sq ft louver assembly is used to moderate the N_2H_2 temperature. It is located on the side of the tank, positioned so that its view of the RTG is negligible (<0.01), and its view of space is excellent (>0.8). The effective emissivity of the louver assembly is 0.13 for fuel temperatures below 520°R, 0.72 for fuel temperatures above 545°R; emissivity varies linearly with temperature between 520°R and 545°R.

All surfaces of the frame, the helium tank, and the fluorine tank which may receive solar radiation are covered with second-surface silvered Teflon. Since this material has a high emissivity and a relatively low solar absorptivity, it helps reduce the sun's heating effects.

5. F_2/N_2H_4 PROPULSION MODULE THERMAL ANALYSIS

5.1 COMPUTER PROGRAMS FOR THERMAL ANALYSIS

The F_2/N_2H_4 propulsion module thermal characteristics were determined in the same general manner as for the OF_2/B_2H_6 module, Volume I Section 5. Preliminary investigations were made with simple, on-line computer programs to establish general trends and to eliminate unpromising thermal control concepts, followed by in-depth studies with detailed programs of the remaining concepts. In this case, three main computer programs were written in SINDA for solution on the Univac 1108 computer. The programs simulated (1) the module during groundhold, (2) the module during flight, and (3) the engine and propellant valves of the module. The last model was used for the analysis of the engine, valves, and catalyst bed during flight. It was desirable to formulate a separate program for this part of the study since the detail necessary in the engine region would have meant long computer run times had the information been obtained on the longer programs. It was possible to do this since, from the standpoint of the engine, the propulsion module appears essentially as three constant temperature boundary nodes.

The first two programs listed, the programs for analyzing the module's thermal behavior during groundhold and flight, represented the module by 38 nodes. The locations and description of the nodes appear in Tables 5.1-1, 5.1-2 and drawing 406922 of this section. This is considerably less detailed than the models used in the analyses of the OF_2/B_2H_6 module. However, the decision to use less complicated models was prompted by results from that work. It was observed there that many of the nodes, particularly those associated with the frame, did not influence the thermal characteristics of the module. This is because the resistors connecting these nodes are either so large as to make the nodes entirely independent of the propellant and propulsion systems or so small as to make the nodes essentially dependent upon the propellant and propulsion temperatures.

It is, of course, necessary that a thermal model retain enough nodes in critical places to ensure a realistic model. In the present case, this has been done by dividing the insulation into many nodes and by including nodes for the frame to account for heat conduction between the tanks.

To simplify the models, the following assumptions were made:

1) It was assumed that the external film coefficient used in determining the atmospheric convection heat transfer during groundhold is independent of temperature.

2) It was assumed that the tank walls are at the same temperature as the internal fluid which is in contact with the wall.

3) The film coefficient on the outside of the flourine and helium cooling coils is given by the equation:

$$h_o = 0.72 \frac{k}{D} \left[\frac{D^3 \rho^2 \beta g \, \Delta T}{\mu^2} \frac{c\mu}{k} \right]^{0.25} \qquad (5\text{-}1)$$

where

h_o = external coefficient of heat transfer, Btu/hr-ft^2-^{o}R

D = tube diameter, ft

k = propellant thermal conductivity, Btu-ft/hr-ft^2-^{o}R

ρ = propellant density, lb/ft^3

β = propellant coefficient of volumetric expansion, 1/^{o}R

ΔT = temperature difference between tube and propellant (T_B-T_t)

g = constant, 4.17×10^8 ft/hr^2

μ = propellant viscosity, lb/ft-hr

c = propellant specific heat at constant pressure, Btu/lb-^{o}R

4) The film coefficient on the inside of the flourine and helium coiling coils (if LN_2 is the coolant) is given by:

$$h_i = 0.029 \frac{k}{D} \left[\frac{VD\rho}{\mu} \right]^{0.8} \left[\frac{c\mu}{k} \right]^{0.4} \qquad (5\text{-}2)$$

5) There are no problems relative to zero-gravity heat transfer.

These are the same assumptions made in the groundhold thermal analysis of the OF_2/B_2H_6 module analysis, Volume I, Section 5. For a detailed analysis of these assumptions, consult that section. In addition, it was assumed in this analysis that if gaseous helium is used as the coolant in the fluorine tank during groundhold, the internal film coefficient of the coil is given by the equation:

$$h_i = 0.0243 \frac{k}{D} \left[\frac{VD\rho}{\mu} \right]^{0.8} \left[\frac{c\mu}{k} \right]^{0.4} \qquad (5\text{-}3)$$

In calculating h_i, it was further assumed that all temperature varying properties may be treated as constants evaluated at the coolant mean temperature. A hand calculation was made to establish the probable magnitude of error introduced by this assumption. The error was less than 10% for the coolant helium temperature range experienced.

5.2 GROUNDHOLD THERMAL ANALYSIS

The problem of maintaining the fluorine below its boiling point during groundhold is similar to the groundhold conditioning problem studied in the OF_2/B_2H_6 systems except with fluorine the problem is heightened because of the lower fluid temperature. The temperature must be kept below 152°R if tank self-pressurization is to be avoided. However, the work statement specifies 180°R as the allowable maximum temperature.

To handle this problem, two approaches may be taken. The hardware to be kept cold may be well insulated and/or the capacity of the cooling system may be increased. With respect to the advantages of increasing the insulation around the fluorine and helium tanks, Figure 5.2-1 shows a definite advantage in increasing the foam thickness to at least two inches. In contrast, it can be seen that increasing the LN_2 coolant flow rate by a factor of five results in only a minor decrease in the fluorine temperature.

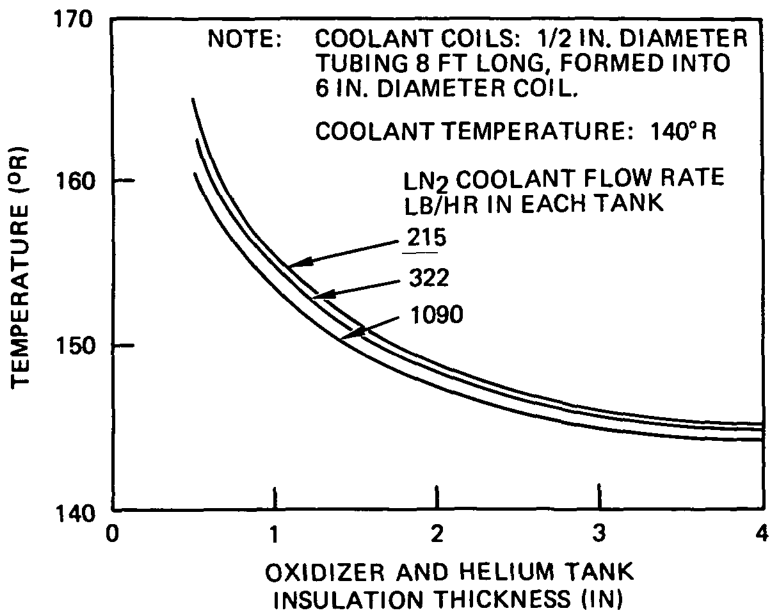


Figure 5.2-1. Effects of Insulation on Oxidizer and Helium Temperatures

From Figure 5.2-1, it can be seen that a LN_2 coolant flow rate of 215 lb/hr coupled with 2-in. thick insulation on the tanks will result in a temperature of 149°R. This is only 3°R below the boiling point of fluorine at one atmosphere. The real problem here is that the LN_2 coolant lacks cooling capability because of its 140°R boiling temperature. It is possible to reduce the LN_2 coolant temperature about 6°R by reducing the pressure on the coolant supply dewar to about 10 psia. For a constant coolant flow rate, a 6°R drop in coolant temperature results in a 6°R drop (to 143°R) in the fluorine temperature.

Such an approach, i.e., lowering the LN_2 temperature by lowering its pressure, is not without serious disadvantages. It would require that a working pump be kept on the storage dewar much of the time. In addition, it would require a pump to force the liquid through the cooling coils as opposed to using a pressure-feed system.* As a consequence, much of the $6^{\circ}R$ temperature drop might disappear due to increased heat addition to the LN_2 through the additional plumbing.

As will be indicated later, there are advantages to launching with the fluorine in a substantially subcooled condition, $110^{\circ}R$ to $130^{\circ}R$. With LN_2 as the groundhold coolant, this is impossible. However, it would be possible to obtain these lower fluorine temperatures by first circulating gaseous helium through liquid hydrogen ($37^{\circ}R$) to chill it and then passing the chilled helium through the module cooling coils. The fluorine temperatures which could be obtained in this manner are shown in Figures 5.2-2 and 5.2-3. To understand the limitations of this cooling method, the two curves of Figure 5.2-2 should be noted. The lower curve, which shows the fluorine temperature that would result if the helium had an infinite heat capacitance, demonstrates that it is not a question of getting the heat into the helium coolant; rather, at low helium flow rates, it is the finite heat capacitance which limits the cooling capability. It is not until the helium flow rate exceeds 25 lb/hr (for an initial temperature of $60^{\circ}R$) that the controlling factors are film coefficient and temperature differential.

The steady-state fluorine temperatures which should be expected at different helium flow rates and inlet temperatures are given in Figures 5.2-3. There is danger of actually freezing fluorine within the fluorine tank and it would not be difficult to cool the fluorine below the established minimum. To avoid these problems, it would be wise to limit the initial helium coolant to a minimum temperature of about $90^{\circ}R$. This would still make it possible to cool the fluorine to about $115^{\circ}R$ with a helium flow rate of 15 lb/hr (based on the use of 2-in. thick foam insulation).

Even though the use of chilled helium as the groundhold coolant makes possible substantial reduction of the flourine temperature, there are serious ground support problems. The use of helium at a rate of 15 lb/hr for weeks is a high usage rate. However, a closed-loop system could be used. The main problem would be the insulation requirements on the helium transfer lines. In all cases, vacuum jacket lines would be mandatory.

It is impossible to definitely establish at this time the better coolant since it depends to a large extent upon the desirability of launching with the fluorine in a substantially subcooled condition. That, in turn, depends upon spacecraft guidance and control requirements during the initial days of the mission flight. At this time, it would appear best to use LN_2 cooling during most of the groundhold phase, and allow the fluorine temperature to rise to $180^{\circ}R$, that is, allow the tank to pressurize to

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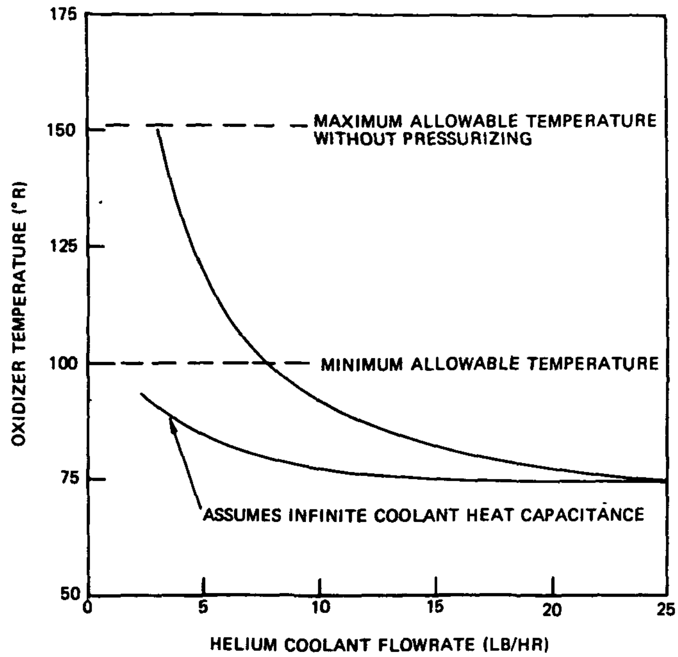
*It might be possible to avoid a pump-feed system by using two interchangeable storage vessels. One would supply subcooled LN_2 by pressurizing while the other was being pumped down to 10 psia. Then, when the first vessel either ran out of LN_2 or became too warm, it would be replaced by the second vessel.



NOTE: INITIAL COOLANT TEMPERATURE EQUALS 60° R
1/2-IN. DIAMETER COOLING COIL, 8 FT LONG.

Figure 5.2-2. Effect of Helium Coolant Flow Rate on Steady-State Oxidizer Temperature

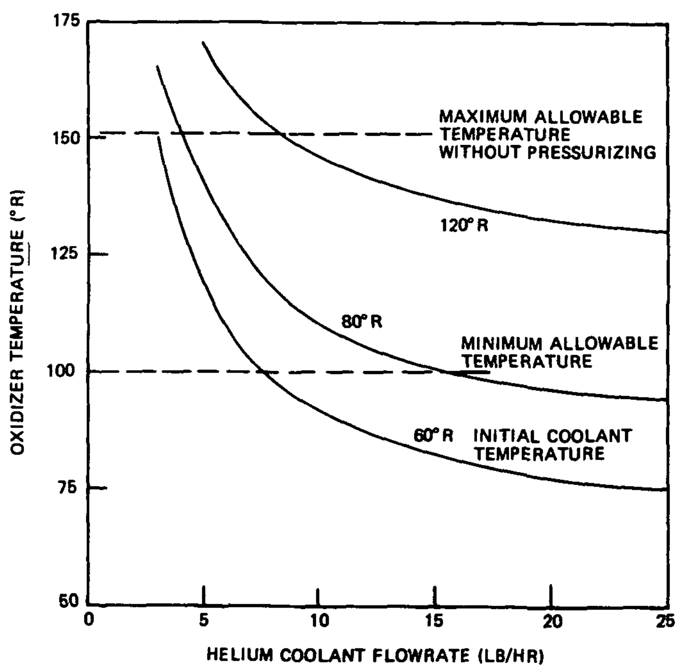


Figure 5.2-3. Effect of Helium Coolant Flow Rate and Temperature on Oxidizer Temperature

60 psia; then, if subcooling is mandatory at launch, circulate chilled helium by means of an open-loop, pressure-fed system for three or four days prior to launch.

Regardless of the coolant used, there may be periods when the coolant will have to be stopped. For example, it may be necessary to discontinue coolant flow during transit to the launch pad, during the spacecraft mating sequence, or during launch count-down. Figure 5.2-4 shows the thermal response of the fluorine which may be expected without any coolant flow and the recovery rate with a high flow rate of LN_2 coolant. For the temperature range of interest, the rate of temperature rise of the fluorine is nearly constant at less than 0.7°R per hour. This shows that the coolant could be stopped for appreciable periods. However, the exact manner of controlling the cooling must be coordinated with the ground handling procedures and requirements.

Figure 5.2-5 is a similar curve for the helium tank. As would be expected, the helium tank reacts much more rapidly. Still, the cooling could be stopped for about one day.

The results showed that there is no problem relative to the hydrazine tank. At all times it stays very near ambient temperature (within 5°R). As in the case of the OF_2/B_2H_6 module, all structural members and plumbing in contact with the cold tanks must also be insulated with foam insulation. Non-metallic members must be insulated 6 to 8 in. from the point of contact and metallic members must be insulated for 18 in. from the point of contact.

If the helium is stored warm, no groundhold problems arise. The frame construction would have to be changed so that the helium tank is thermally connected to the hydrazine tank and not the fluorine tank as it is now. The helium tank would stay warm just as the hydrazine tank does now.

Summarizing the groundhold thermal control analyses results, the following points should be noted:

1) The fluorine and helium can be maintained below 152°R using LN_2 as the coolant. If appreciable subcooling is desired, chilled gaseous helium or subcooled LN_2 will be required.

2) Two inches of foam insulation should be used in order to reduce the groundhold cooling load (based on k/ℓ = 0.038 Btu/ft^2-hr-°R).

3) Groundhold cooling of the fluorine can be stopped for 5 to 50 hours, depending on the initial temperature and allowable tank pressure. Cooling of the helium may be stopped for 3 to 20 hours.

5.3 FLIGHT THERMAL CONTROL ANALYSIS

An F_2/N_2H_4 propulsion module presents somewhat more difficult flight thermal

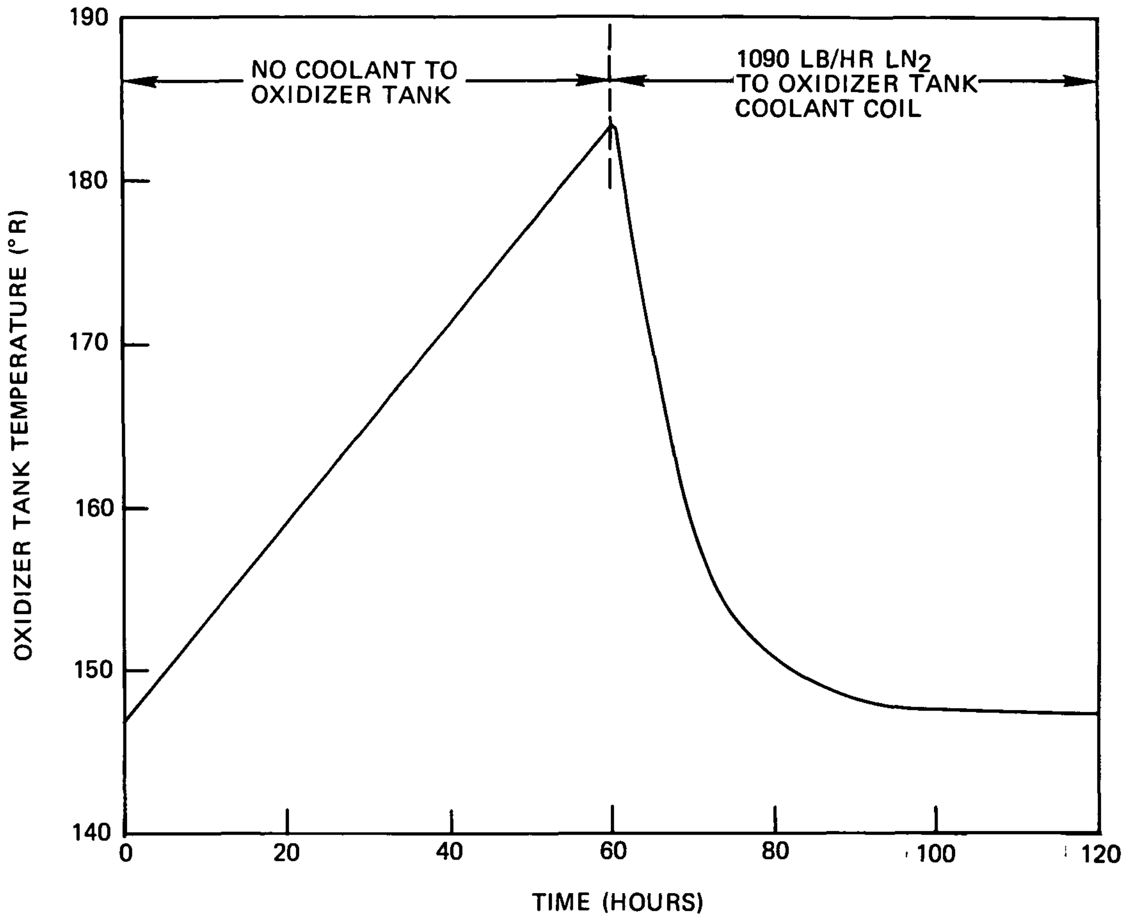


Figure 5.2-4. Thermal Characteristics of Oxidizer Tank With and Without LN_2 Coolant Flow. (Two-Inch Thick Foam Insulation)

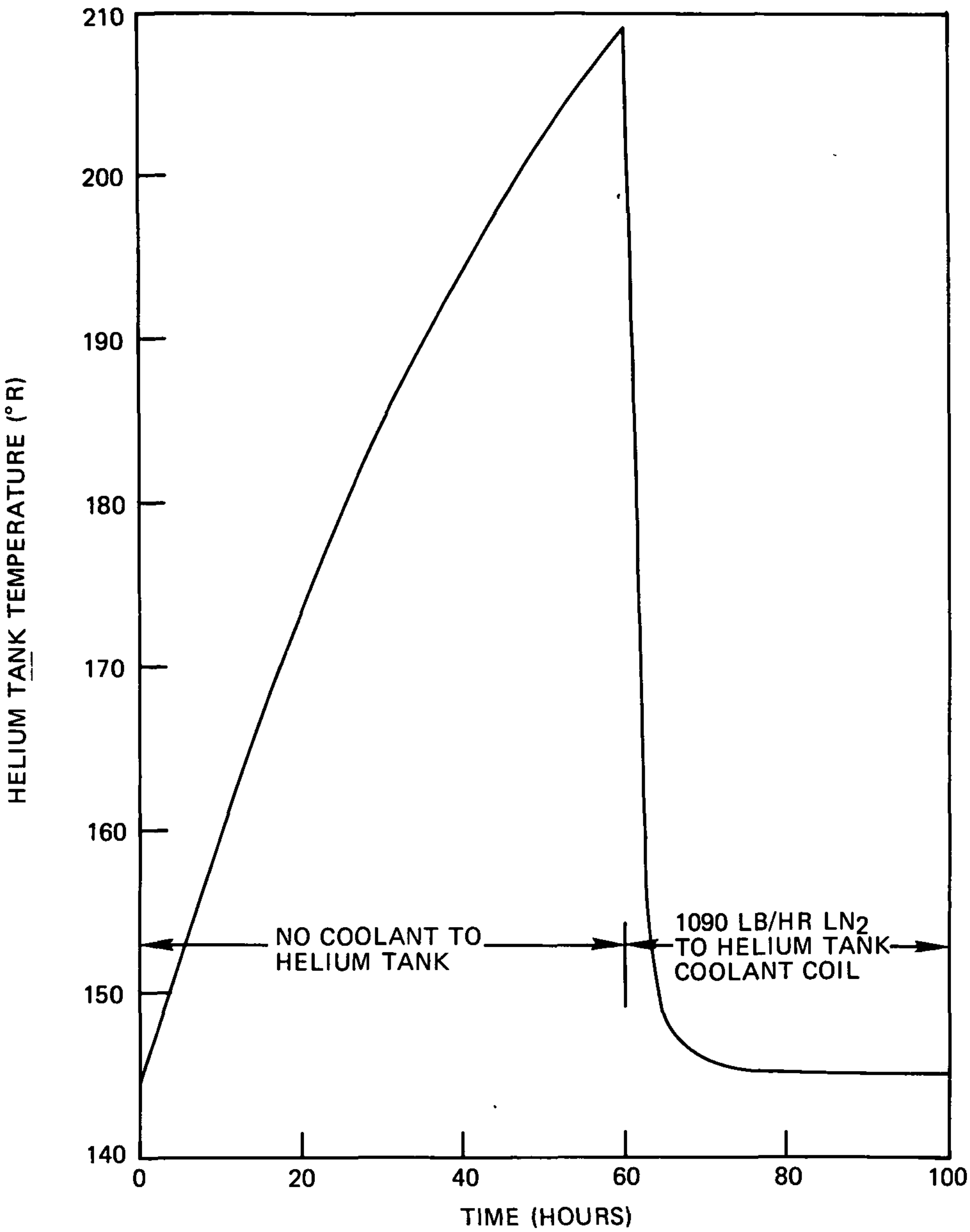


Figure 5.2-5. Thermal Characteristics of Helium Tank With and Without LN_2 Coolant Flow. (Two-Inch Thick Foam Insulation.)

control problems than an OF_2/B_2H_6 module because fluorine must be stored at a colder temperature (<200°R *) and the hydrazine at a warmer temperature (490-560°R). The wide difference in temperature makes thermal isolation of the fluorine tank (and possibly the helium tank) from the hydrazine tank, the RTG, and the sun extremely critical. The temperature of the hydrazine tank can be controlled quite easily by balancing heat input from the RTG with heat rejection to space. But rejection of absorbed heat from the fluorine tank to space is much more difficult because of fluorine's low storage temperature. For example, an uninsulated B_2H_6 tank can radiate to space 10 times more heat than an uninsulated fluorine tank.

These problems were analyzed in three distinct steps:

1) The thermal control concepts of a fluorine tank were investigated.

2) The thermal control concepts of a hydrazine tank were investigated.

3) The thermal characteristics of an integrated module were investigated.

5.3.1 Fluorine Tank Thermal Control Concepts

There are two basic approaches in thermal control of cryogens such as fluorine in space.

1) Thermally isolate the cryogen as much as possible from the remainder of the spacecraft, and rely upon the heat capacity of the cryogen and the tank to absorb all heat input which may occur during the mission. This approach requires a well insulated system particularly if a mission of long duration is contemplated.

2) Thermally isolate the cryogen from the remainder of the system and protect it from all stray radiation, such as solar radiation and then rely upon the system to radiate excess heat to space. This approach requires that the cryogen tank be uninsulated so that it may radiate to space as effectively as possible.

A combination of the above two concepts involves thermal isolation of the cryogen from heat sources and reliance on the cryogen's heat capacitance to absorb most short term excess heating. With this approach, the tank is allowed to return to its normal temperature by radiating to space over a long time period the excess heat absorbed. The approach adopted depends on the heat capacitance of the system (type of cryogen), the steady-state potential heat leaks to the cryogen (vehicle construction), and the mission contemplated. For the reasons discussed below, the last approach appears superior for the present system.

*Liquid, subcooled, 20°R below saturation temperature at 300 psia.

5.3.1.1 Fluorine Heat Capacitance

The estimated in-flight heat capacity is based on the following assumptions:

1) The fluorine and its tank are initially at a temperature just above the freezing temperature of fluorine ($100^{\circ}R$).

2) Eighteen hundred pounds of fluorine are to be stored for 1793 days.

3) No fluorine is vented and none is used for mid-course maneuvers.

4) Temperature of the fluorine and its tank can be allowed to rise to $200^{\circ}R$.

These are optimistic assumptions since not all 1800 lb of the fluorine is on board during the full mission and the temperature limits assumed are the extreme limits. Total heat that can be absorbed by the fluorine then is

$$q_{fluorine} = w_F c_{p_F} \ (T_f - T_i) = (1800)(.363)(200-100) \quad (5\text{-}4)$$

$$q_{fluorine} = 67,300 \text{ Btu} \quad (5\text{-}5)$$

Total heat that can be absorbed by the fluorine tank is

$$q_{tank} = w_T c_{p_t} \ (T_f - T_i) = (59.6)(.25)(200-100) \quad (5\text{-}6)$$

$$q_{tank} = 1530 \text{ Btu} \quad (5\text{-}7)$$

The total net heat leakage than can be absorbed by heat capacitance is

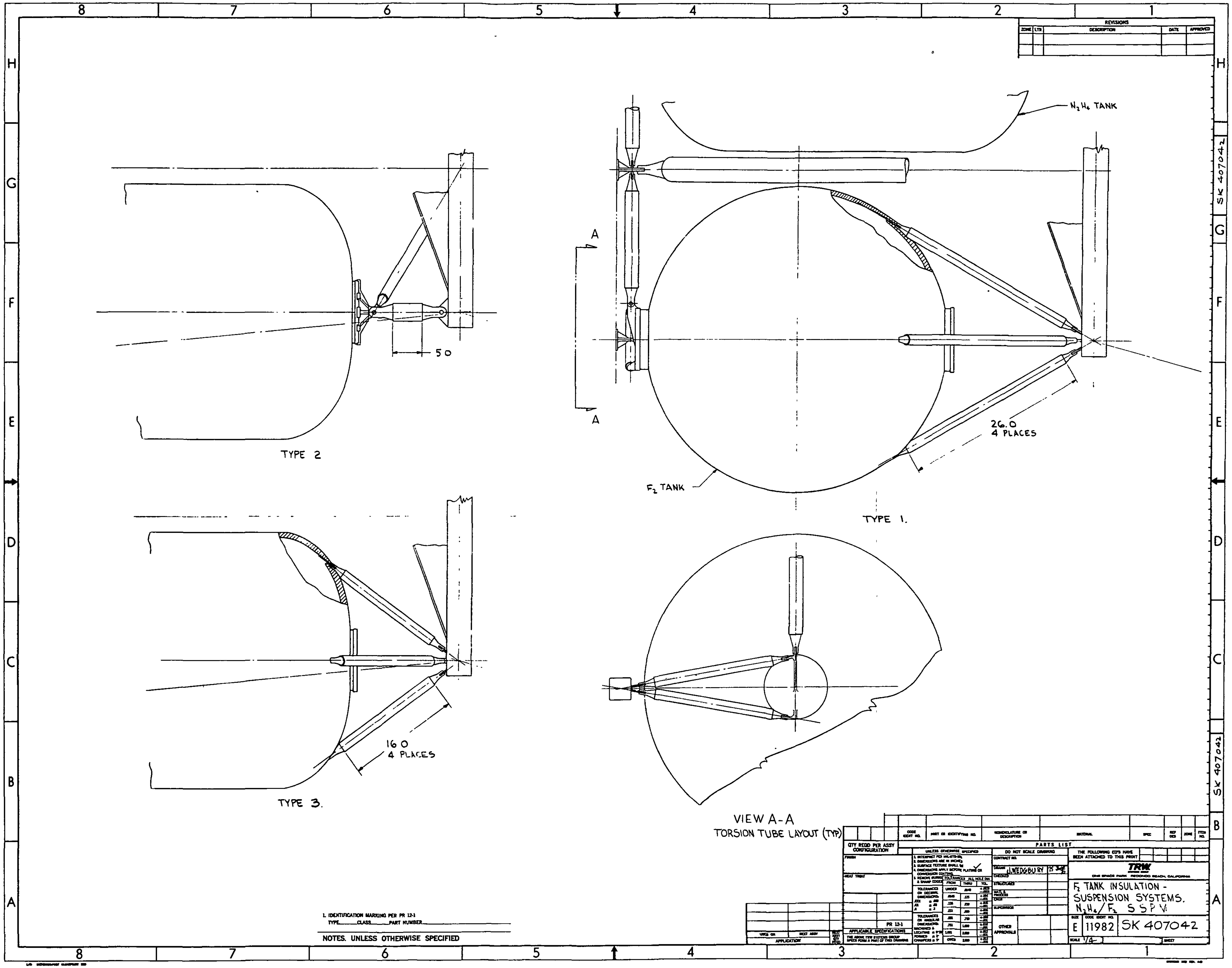
$$q_{total} = 67,300 + 1530 = 68,830 \text{ Btu} \quad (5\text{-}8)$$

This would indicate that an average heat leak into the tank of 1.6 Btu/hr could be tolerated.

5.3.1.2 Potential Heat Leaks to Fluorine Tanks

Expected heat leaks cannot be estimated accurately without a rather detailed thermal model of the entire fluorine tank, its supporting structure, and the surrounding boundary conditions. Two potential heat leak sources are the support struts and plumbing lines, since these components are subject to strength and fluid flow requirements which limit the amount of thinning and lengthening that can be done in the interest of reducing thermal conductance.

Drawing SK 407042 shows three configurations that have been considered for supporting the 1800 lb of fluorine. The thermal conductance for each type of support system has been estimated as shown below in Table 5.3-1. All support struts are six-ply, resin-impregnated fiberglass (0.030 in. total wall thickness, $k \approx 0.15$ Btu/ft-hr-$^{\circ}R$).



N_2H_4 TANK
50
TYPE 2
A
A
26.0
4 PLACES
F_2 TANK
TYPE 1.
16.0
4 PLACES
TYPE 3.
VIEW A-A
TORSION TUBE LAYOUT (TYP)
1. IDENTIFICATION MARKING PER PR 12-1
TYPE____CLASS____PART NUMBER____
NOTES, UNLESS OTHERWISE SPECIFIED
PARTS LIST
DO NOT SCALE DRAWING
DRAWN J.WEDGBURY
TRW
ONE SPACE PARK REDONDO BEACH, CALIFORNIA
F_2 TANK INSULATION -
SUSPENSION SYSTEMS.
N_2H_4/F_2 SSPV
E 11982 SK 407042
SCALE 1/4
PR 12-1
APPLICABLE SPECIFICATIONS
SK 407042

Table 5.3-1 shows that the Type 1 support system has the lowest thermal conductance. With that system, an end-to-end temperature difference of, for example, 100°R on the struts, will result in a heat leak of 0.613 Btu/hr. However, this support concept has the decided disadvantage of requiring a different tank design for each tank. The Type 2 support system which is more practical, has approximately twice the thermal conductance, and this concept is not readily accommodated by the tank constructed of boron filament.

All of these support concepts have a disadvantage in either fabrication or weight (the Type 3 support system is approximately 16 lb heavier than the support system shown in SK 406922). In any event, if a 200°R differential temperature exists, the transfer rate from the support alone is excessive.

Table 5.3-1. Thermal Conductance of Various Fluorine Tank Support Systems

Support System	Specific Heat Paths	Tube Diameter (inches)	Tube Length (inches)	Conductance* (Btu/hr-$^{\circ}$R)	Total Conductance (Btu/hr-$^{\circ}$R)
Type 1	4 comp. tubes	1.5	26.0	.00240	.00613
	2 torsion tubes	1.5	17.5	.00202	
	1 brace tube	2.0	13.875	.00171	
Type 2	1 comp. tube	2.5	5.0	.00590	.01132
	1 stabilizing tube	1.5	23.0	.00077	
	2 torsion tubes	1.5	14.25	.00248	
	1 brace tube	2.0	10.875	.00217	
Type 3	4 comp. tubes	1.5	16.0	.00440	.00905
	2 torsion tubes	1.5	14.25	.00284	
	1 brace tube	2.0	10.875	.00217	
Final Design	2 frame struts	-	-	.011	.0127
	2 brace tubes	1.25	27.8	.0017	

*Crinkled aluminized Mylar placed within the tubes will make lengthwise radiation negligible.

Plumbing to the fluorine tank consists (at present) of three stainless steel lines. One of these is a 0.75 in. (nominal) inner diameter, 1.14 in. (nominal) outer diameter, convoluted stainless steel flex-hose with a wall thickness of 0.013 in. for transporting propellant approximately 38 in. to the rocket engine. Another is a 1/4-in. (nominal) O.D., 0.016-in. wall thickness pressurant line running approximately 40 in. to the pressurant valve panel. The third is a 1/2-in. (nominal) O.D., 0.16-in. wall thickness line running approximately 28 in. to the pressure relief valve. Estimated end-to-end thermal conductances for these three potential heat flow paths are given below in Table 5.3-2. Cooling coil lines and instrumentation wires were also considered, but deemed negligible compared to those represented in Table 5.3-2.

Table 5.3-2. Thermal Conductances of the Plumbing Lines into the Fluorine Tank

Line	Nominal Diameter (inches)	Wall Thickness (inches)	Effective Length (inches)	Conductance* (Btu/hr-°R)
Propellant	.95	.013	76	.000425
Pressurant	.25	.016	40	.000240
Relief	.50	.016	28	.000750

*Includes a slight amount of conductance due to gaseous helium conduction (k = .07 Btu/hr-ft-°R) within each tube.

If the lines are well-insulated along their length, they will not present any major heat leak problem even if the valve blocks at the far end are several hundred degrees warmer than the fluorine tank because of the low conductances.

Another potential heat leak is thermal radiation from the hydrazine tank. An eight-node analytical model as shown in Figure 5.3-1 was used in a rough evaluation of this heat path. The Figure 5.3-1 network has been solved for various fixed values of fluorine tank temperature (T_1) and various thicknesses of foam insulation (k = .00625 Btu/ft-hr-°R). The resulting net heat rate ($q = q_{6-1}$) into the fluorine tank is presented in Table 5.3-3. Table 5.3-4 gives the corresponding results with the radiation shield removed. Negative values of q indicate a net heat loss (to space) by the fluorine tank.

A comparison of Tables 5.3-3 and 5.3-4 indicates that insertion of a flat plate radiation shield between the tanks is an effective means of reducing heat input to the fluorine tank, especially at lower fluorine tank temperatures. Even with the shield in place, the fluorine tank might absorb appreciable quantities of heat by radiation from the hydrazine tank.

The final potential source of heat input to the fluorine tank considered in this portion of the analysis is direct or reflected solar impingement. An analysis was done by applying various amounts of heating to node 6 (outer surface of the foam insulation) on the existing eight-node analytical model and re-evaluating the net heat rate into node 1 (the fluorine tank). The results are plotted in Figure 5.3-2. Increasing the thickness of the foam insulation will significantly reduce the effect of solar impingement. Even with three inches of foam, however, solar impingement must be avoided or restricted to short periods. For example, suppose one solar constant ($G = 430$ Btu/ft^2-hr) impinges at right angle ($\theta = 90^\circ$). The projected area (A_p) of the fluorine tank for a side-looking sun is approximately 12 ft^2. If the outer surface of the insulation were

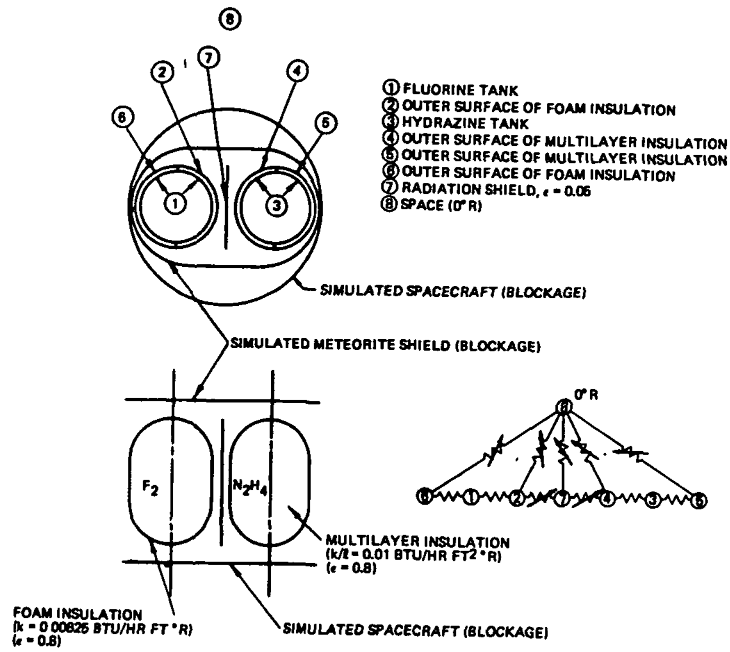


Figure 5.3-1. Eight-Node Analytical Model for Estimating Radiation Interchange Between Propellant Tanks

Table 5.3-3. Net Heat Rate into Fluorine Tank due to Thermal Radiation*
(Inter-tank Radiation Shield in Place)

Insulation Thickness (inches)	0.75	1.5	3.0
Fluorine tank temperature, (T_1) (°R)	q (Btu/hr)	q (Btu/hr)	q (Btu/hr)
200	-37.43	-31.35	-24.45
100	0	+ 0.03	+ 0.11
0	+ 2.74	+ 2.74	+ 2.74

*Hydrazine Tank Temperature = 530°R, incident solar flux = 0.

Table 5.3-4. Net Heat Rate into Fluorine Tank due to Thermal Radiation*
(Inter-tank Radiation Shield Removed)

Insulation Thickness (inches)	0.75	1.5	3.0
Fluorine tank temperature, (T_1) (°R)	q (Btu/hr)	q (Btu/hr)	q (Btu/hr)
200	-19.25	-15.90	-12.33
100	+39.50	+18.45	+12.70
0	+43.10	+43.20	+42.40

*Hydrazine Tank Temperature = 530°R, incident solar flux = 0.

covered with one layer of second-surface, silvered Teflon (for minimum and stable α_s/ϵ_H), the surface would absorb heat at the following rate:

$$GA\alpha_s \cos\theta = (430)(12)(.1)(1) = 515 \text{ Btu/hr} \quad (5\text{-}9)$$

Figure 5.3-2 shows that with three inches of foam insulation, this surface heat rate would result in a net heat input rate of about 70 Btu/hr to the fluorine tank. An insulation thickness of 3/4 in. would increase the net heat input rate to about 220 Btu/hr. Based on the previously estimated heat capacity, the latter rate could be sustained for approximately ten days before the upper temperature limit for firing would be exceeded.

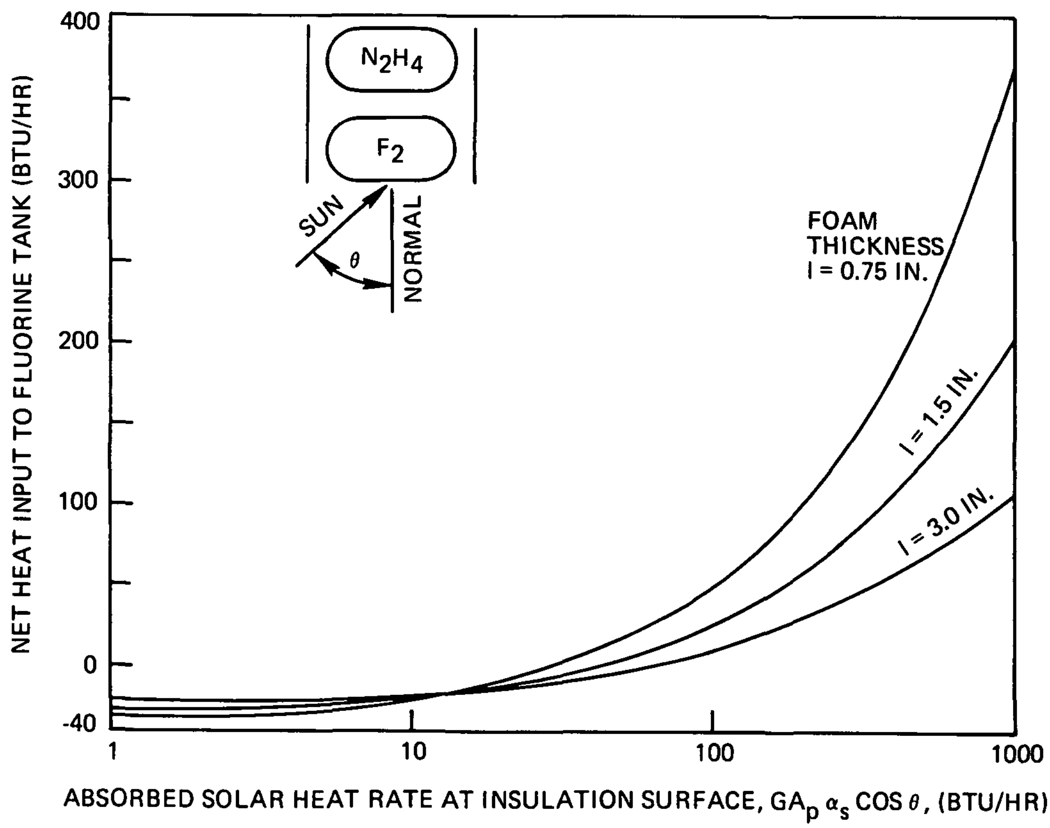


Figure 5.3-2. Net Heat Input Rate to Fluorine Tank Due to Direct or Reflected Solar Impingement (Thermal Conductivity, k, of the foam = 0.00625 Btu/ft-hr-°R)

5.3.1.3 Preliminary Analysis Conclusions (Fluorine Tank Thermal Control

Based upon thermal control state-of-the-art, it is unwise to rely solely on the cryogen to prevent excessive temperatures. In theory, the fluorine tank with the proper support and shielding could absorb all the heat if solar heating were totally eliminated. However, fabrication techniques are not sufficiently precise to control the heat transfer rate with the required accuracy at these low temperatures. The fluorine tank must be allowed to radiate to space during the long periods that it does not have to view the sun.

Of course, every reasonable effort should be made to reduce the heat transfer rate to the fluorine tank. However, since the predominate mode of heat transfer is by radiation, unusual support methods for the fluorine tank are questionable. For this reason, the tank support concept from the OF_2/B_2H_6 study should be kept and heat conduction into the tank minimized by non-metallic members in the lower support frame. These conclusions are reflected in drawing SK 406922.

The analysis also indicated that radiation heat transfer to the fluorine tank must be minimal. Therefore, the following conditions must be met:

1) The RTG must not "see" the fluorine tank.

2) A radiation shield is required between the fluorine and hydrazine tanks.

3) Radiation from the spacecraft to the fluorine tank must be reduced by insulating the bottom surface of the spacecraft which "sees" the fluorine tank.

4) The fluorine tank must be well shielded from solar radiation.

All information concerning the temperature control of the fluorine tank applies directly to the helium tank if the helium is stored cold. The merits of warm and cold helium storage will be discussed later.

5.3.2 Hydrazine Tank Thermal Control Concepts

Thermal control of the hydrazine tank is accomplished by shielding the tank as much as possible from the varying solar flux (430 Btu/ft^2-hr at Earth, 16 Btu/ft^2-hr at Jupiter) and then balancing heat input from the RTG with thermal radiation to space. This is the same basic approach that was used for the OF_2/B_2H_6 module. However, due to the higher storage temperature of the hydrazine, several alternate methods of transporting heat had to be reconsidered.

Within the passive or semi-passive device constraint, Figure 5.3-3 shows the most promising passive and semi-passive concepts for transporting heat from the RTG to the hydrazine tank, while Figure 5.3-4 shows three passive and semi-passive methods of rejecting heat to space from the hydrazine tank. While it is impossible to establish precisely how each of these concepts will function with simple calculations, it is possible to establish general characteristics by simple calculations.

5.3.2.1 Passive System Characteristics

Based on a simple "voltage divider" network, a purely passive means of thermal control requires that the RTG temperature vary no more than about twice the allowable

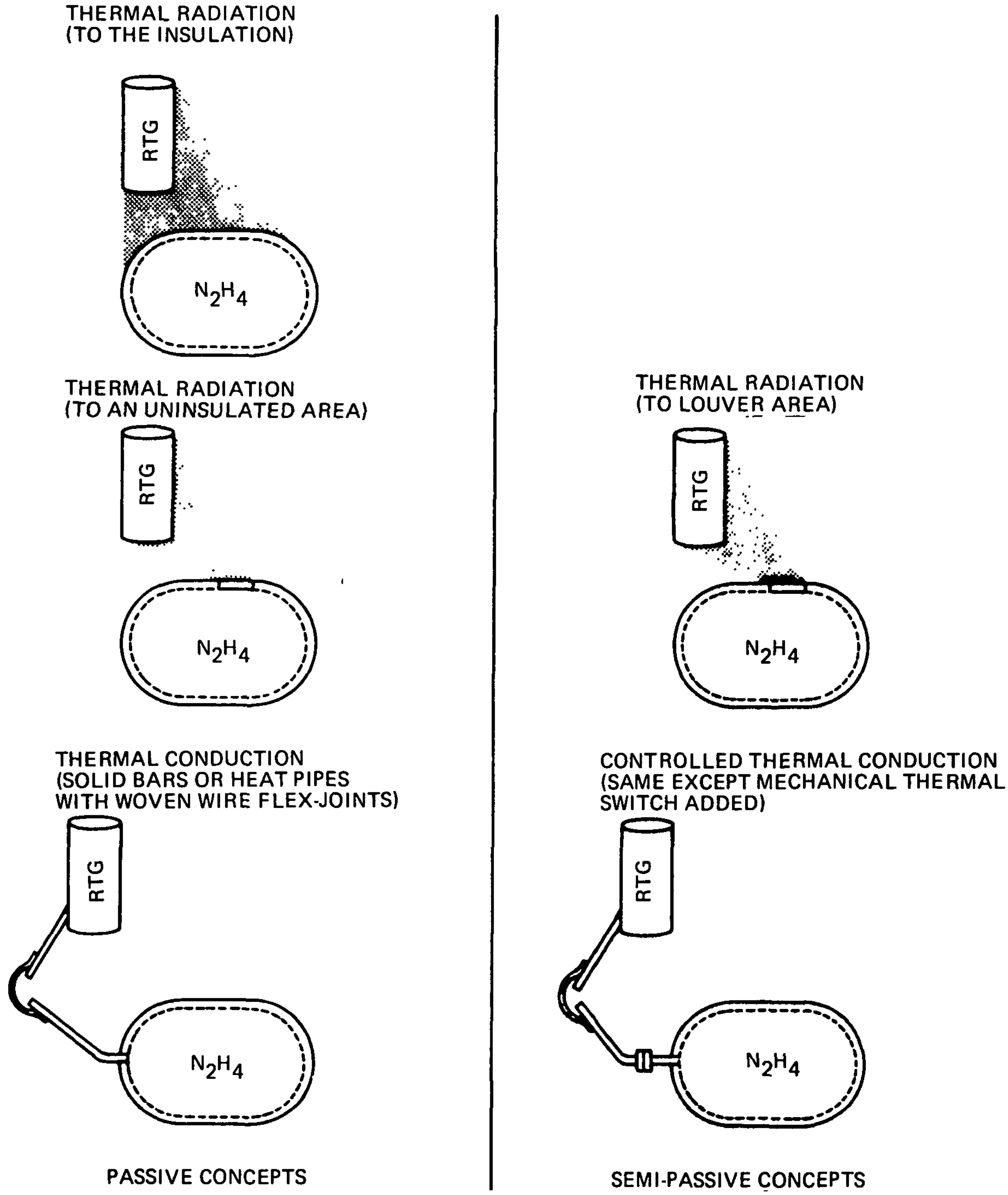


Figure 5.3-3. Concepts for Transporting Heat From the RTG to the Hydrazine Tank

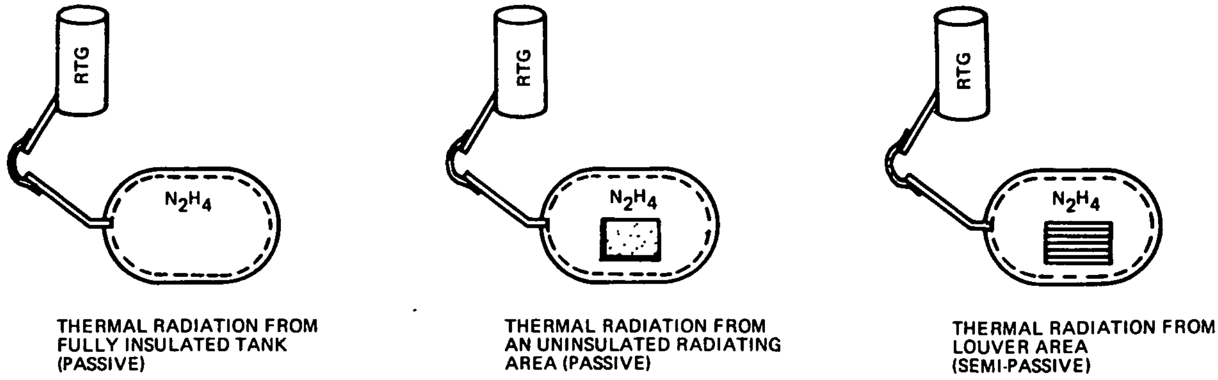
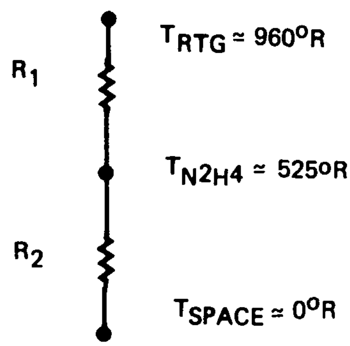


Figure 5.3-4. Hydrazine Tank to Space Thermal Coupling Concept

variance in hydrazine temperature. Consider the following diagram with nominal temperatures indicated.



where:

R_1 = overall equivalent thermal resistance between the RTG and the hydrazine tank

R_2 = overall equivalent thermal resistance between the hydrazine tank and space.

The resistances R_1 and R_2 can be considered linear over small ranges of temperature change so that by proportion:

$$T_{N_2H_4} = \frac{R_2}{R_1 + R_2} T_{RTG} \tag{5-10}$$

Differentiating both sides gives:

$$dT_{N_2H_4} = \frac{R_2}{R_1 + R_2} dT_{RTG} \tag{5-11}$$

By solving equation 5-10 for $R_2/(R_1+R_2)$ and substituting the result into equation 5-11, one obtains:

$$dT_{N_2H_4} = \frac{T_{N_2H_4}}{T_{RTG}} dT_{RTG} \tag{5-12}$$

or numerically:

$$\Delta T_{N_2H_4} \simeq \frac{1}{2} \Delta T_{RTG} \tag{5-13}$$

Thus, if the hydrazine tank must be held to $525^{\circ}R \pm 25^{\circ}R$, a purely passive system cannot be used unless the RTG temperature is approximately $960^{\circ}R \pm 50^{\circ}R$ under all conditions.

A purely passive system using thermal radiation, rather than conduction, as the coupling agent between the RTG and the hydrazine tank was briefly examined using the eight-node analytical model shown in Figure 5.3-5. The model was run repeatedly, each time either the uninsulated tank area (A_4) or the view factor (F_{41}) from the uninsulated area to the RTG was varied to obtain the results plotted in Figure 5.3-6. For convenience, it was arbitrarily assumed that the view factor F_{21} from the insulated area (node 2) to the RTG was equal to F_{41}. This is not a necessary condition, but it could be easily achieved if desired. It can be seen in Figure 5.3-6 that, for a given view factor, varying the amount of uninsulated area has little effect on the equilibrium temperature of the hydrazine unless the exposed area is quite small. The reason for this is that the total heat rate radiated to space by the tank is dominated by the uninsulated area, if that area is large. Thus, doubling the exposed area under this condition doubles not only the heat input rate from the RTG, but also the heat rejection rate to space. With a small exposed area, radiation from the large insulated area accounts for a large part of the total heat rejection rate to space so that a change in exposed area still has a significant effect on the heat input rate, but not on the heat rejection rate. The important conclusion that can be drawn from Figure 5.3-6 then is that in the absence of solar impingement on the uninsulated area and with a stable $960^{\circ}R$ RTG temperature, the hydrazine equilibrium temperature can be passively maintained within the required limits ($\approx 500^{\circ}R$) by providing approximately 4 sq ft of uninsulated area with this area having a view factor to the RTG of approximately 0.1.

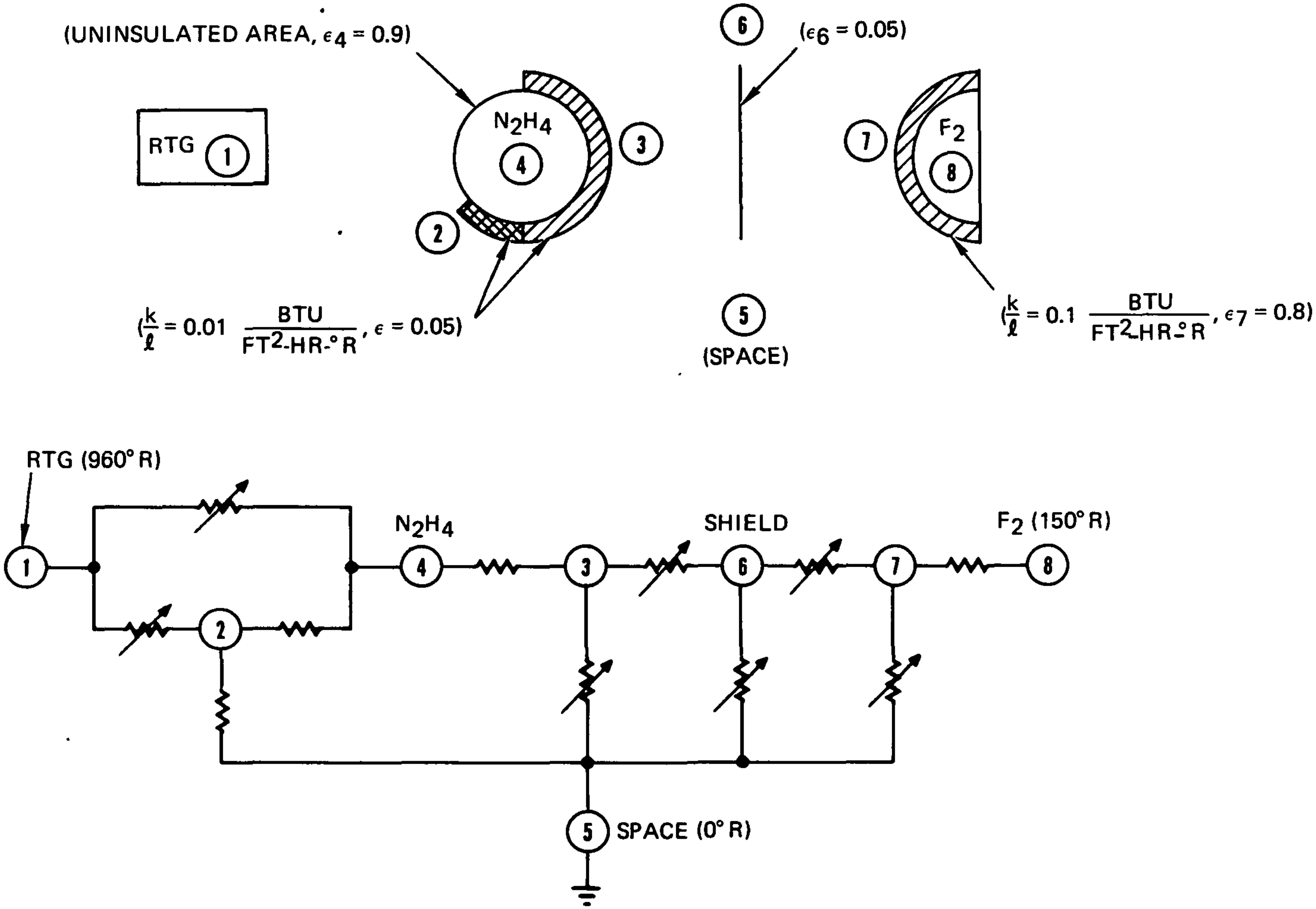


Figure 5.3-5. Eight-Node Thermal Resistance Network for Analyzing the RTG to Hydrazine Tank Thermal Radiation Coupling

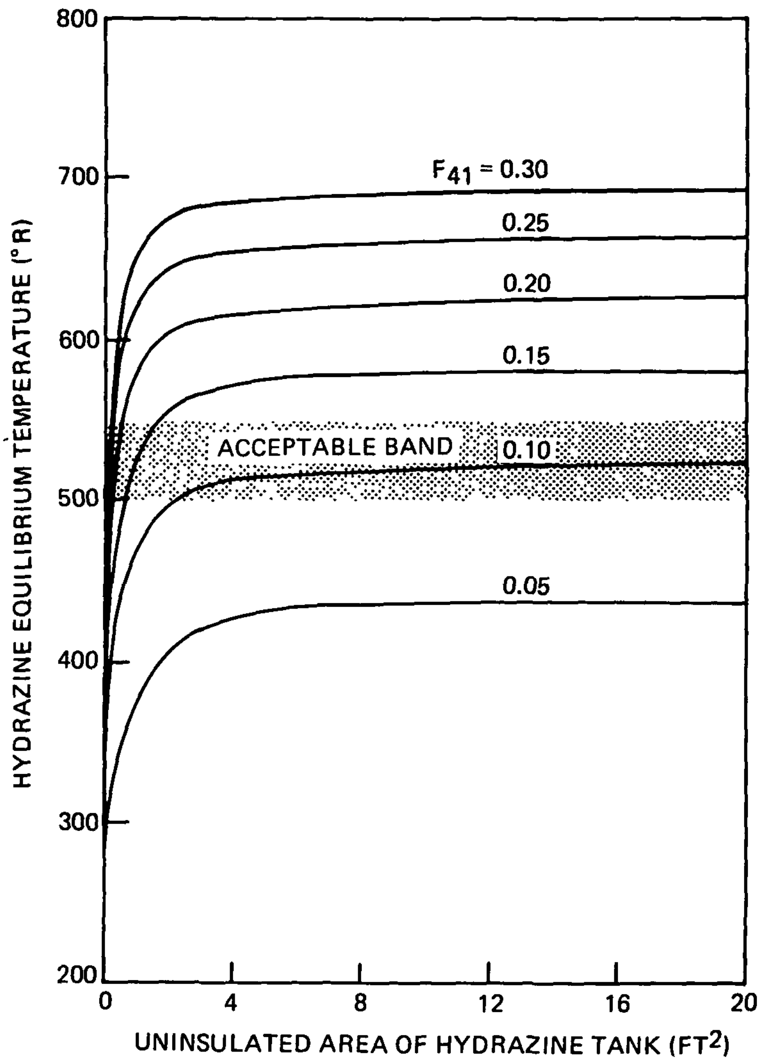


Figure 5.3-6. Equilibrium Temperature of Hydrazine Tank with Thermal Radiation Coupling to the RTG. Parameter F_{41} Is the View Factor from the Uninsulated Tank Area to the RTG

Since a fully insulated tank offers maximum protection against inadvertent solar impingement, some of the data (for A = 0) from Figure 5.3-6 has been cross-plotted in Figure 5.3-7 to show more clearly how the hydrazine equilibrium temperature is related to the view factor (F_{21}) for a fully insulated tank. It can be seen from Figure 5.3-7 that a view factor of approximately 0.6 would be required in order to maintain the nominal hydrazine equilibrium temperature with only thermal radiation coupling and a fully insulated tank. If the RTG and the hydrazine tank are characterized as parallel cylinders of infinite length and zero separation (external tangential contact) with respective diameters of 10 and 34 in., the maximum possible view factor F_{21} from half the tank (node 2) to the RTG is approximately 0.15. Radiation coupling between the RTG and the hydrazine tank is, therefore, an unworkable concept if the hydrazine tank is fully insulated.

Even if the hydrazine tank is not fully insulated, there may be other reasons that radiation coupling between the RTG and hydrazine may not be acceptable as a means of passing heat to the fuel tank. To maintain the fuel temperature within limits, the view factor of the RTG by the fuel tank must be about 0.1 (Figure 5.3-6). This requires that the RTG be placed fairly close to the module, approximately 2 to 3 feet away depending on the orientation. This may result in unacceptable levels of hard radiation near the spacecraft or propulsion module. If the RTG must be displaced a certain amount to keep the radiation level down, heat pipes or conduction bars between the RTG and fuel tank will be mandatory. If radiation is not an issue, serious consideration should be given to using small radioisotope heaters. These heaters, weighing about 0.2 oz/Btu/hr output, could be bonded directly to the tank walls under the insulation. This approach has two advantages. First, it is the lightest; second, by using several small units distributed over the tank surface, the problem of local hot spots within the fluid, which some authorities believe might occur in zero-gravity flight, is materially reduced if not actually eliminated. In fact, it may be possible to use these heaters to help control the location of the liquid within the tank during periods of zero gravity. For purposes of this analysis it is assumed that hard radiation from the RTG is detrimental to the module and, therefore, the RTG must be separated from the module and small radioisotope heaters may not be used. It should also be noted that even if hard radiation from the RTG to the module is not a problem, a closely linked RTG presents serious deployment mechanism problems.

5.3.2.2 Semi-passive System Characteristics

There are only three possible semi-passive systems (Figures 5.3-3 and 5.3-4). However, other considerations limit the choice to louvers mounted on the tank for the purpose of increasing the amount of heat radiated to space when the tank gets too hot. As noted above, a louver mounted on the tank to accept heat from the RTG is precluded because of hard radiation. In addition, if the RTG is mounted near a louver it will be exposed to a highly varying thermal field. For example, with open louvers the RTG will "see" a relatively cold surface, but with closed louvers the RTG will "see" a considerably warmer surface and will probably be able to "see" itself mirrored in the blades.

Thus, closing the louvers will result in an appreciable upwards shift of the RTG temperature. The amount of shift depends on the shape and nominal temperature of the RTG. It does appear as if this effect upon the RTG would be of a sufficient magnitude to be unacceptable. However, such an approach would show very good module thermal control over a large range of environmental conditions.

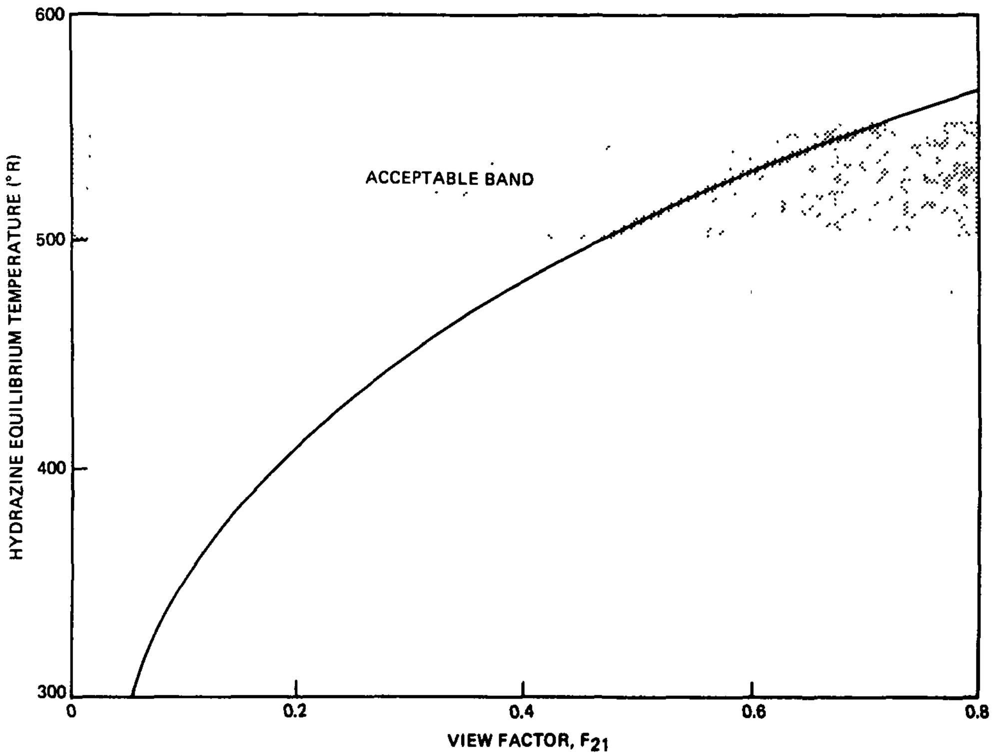


Figure 5.3-7. Hydrazine Temperature with a Fully Insulated Tank and Thermal Radiation Coupling to the RTG

A thermal switch in the conduction path between the tank and RTG must be eliminated since thermal switches have not been perfected for the heat transfer rates involved in this case. The characteristics of the one remaining system will be considered in the next section.

5.3.2.3 Preliminary Hydrazine Tank Thermal Control Analysis Conclusions

The preliminary calculations concerning the hydrazine tank thermal control indicated two main points. First, if a purely passive control system is used, the heat supplied by the RTG must be conducted, not radiated to the hydrazine tank. Secondly, if a semi-passive system is used, it must consist of tank-mounted louvers which radiate heat to space.

5.3.3 Thermal Characteristics of Integrated Module

Based on the previous discussion, the following flight thermal control concepts can, in theory, suffice for the F_2/N_2H_4 propulsion module during the Jupiter mission.

Hydrazine Tank Heating

Heat from RTG via heat pipe

Heat from RTG via solid conductor

Hydrazine Tank Cooling

Radiation to space from surface of insulation

Radiation to space from radiating surface (radiator)

Radiation to space from louver assembly

Fluorine Tank Cooling

Radiation to space from surface of insulation

Radiation to space from radiating surface

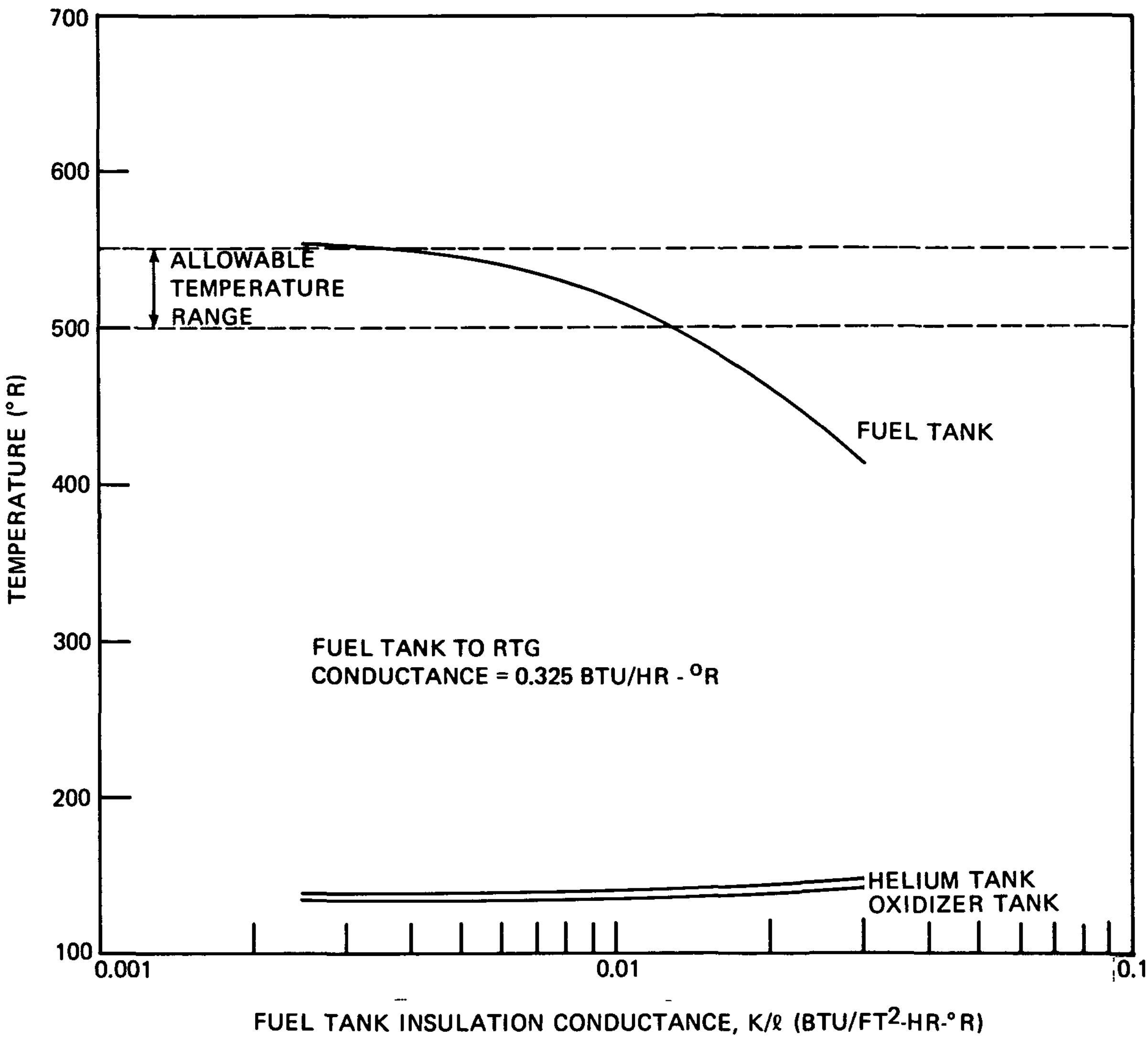
The purpose of this portion of the analysis was to investigate in detail the relationship of these concepts to the module as a whole. The flight thermal control computer model described in Section 5.1 was used in this effort. The general module configuration shown in drawing SK 406922 was assumed, but local variations were made for the several computer runs in order to isolate specific thermal characteristics. For purposes of this portion of the study, it was assumed that the helium is stored at less than 180°R. This is consistent with the assumptions made in the groundhold thermal analysis and the description of the system given in Section 4. The consequences of switching to a warm helium storage system will be indicated however.

5.3.3.1 Flight Thermal Control of N_2H_4 Tank

The effect of the tank insulation was considered first. Figure 5.3-8 shows the variation in N_2H_4 steady-state temperature as a function of the insulation conductance. Although this curve is for the particular case of a 960°R RTG and louvers on the tank, it does show that as long as the insulation conductance is below about 0.01 Btu/ft^2-hr, the variation in conductance is not too important. Fortunately, it is not difficult to manufacture multilayer insulation with a conductance less than 0.01.

If a purely passive insulation system with a conductance of 0.01 were to be used, the propellant temperatures would vary as a function of RTG temperature as shown in Figure 5.3-9. The RTG could vary between 840°R and 980°R without causing the fuel to exceed its limits. This is as predicted in Section 5.3.2. This plot is unique in that the system could have been made to function properly at a RTG temperature of 1400°R by merely reducing the fuel tank-to-RTG conductance by the ratio of 900/1400 or decreasing the multilayer insulation conductance by approximately that amount. In any event, the allowable variation in RTG temperature would still be about 140°R. This characteristic is demonstrated by the curves of Figure 5.3-10.

The fluorine is relatively insensitive to the RTG temperature and, if the N_2H_4 tank is exposed to full solar radiation at 1 A.U., the fluorine tank temperature variation is small. This is shown by the two dots of Figure 5.3-9.



NOTE: FOR NORMAL ALUMINIZED MYLAR INSULATION BLANKET CONSTRUCTION, THE APPROXIMATE RELATION BETWEEN CONDUCTANCE AND NUMBER OF LAYERS IS AS SHOWN (REFERENCE 3):

K/ℓ	LAYERS
0.003	25
0.01	10
0.03	4

Figure 5.3-8. Effect of Fuel Tank Insulation Conductivity, RTG = 960°R

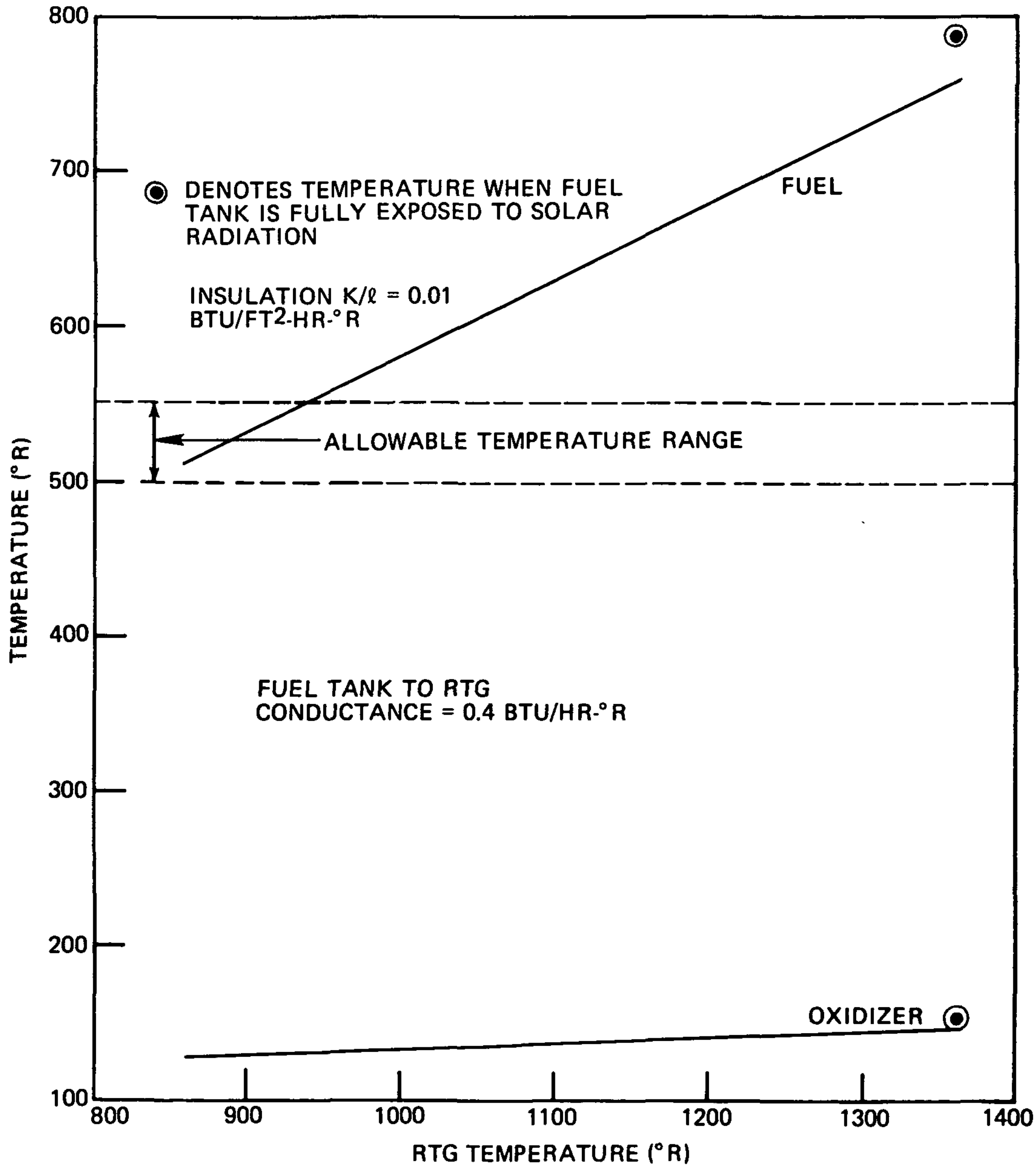


Figure 5.3-9. Effect of RTG Temperature on Propellants, Fuel Tank Fully Insulated

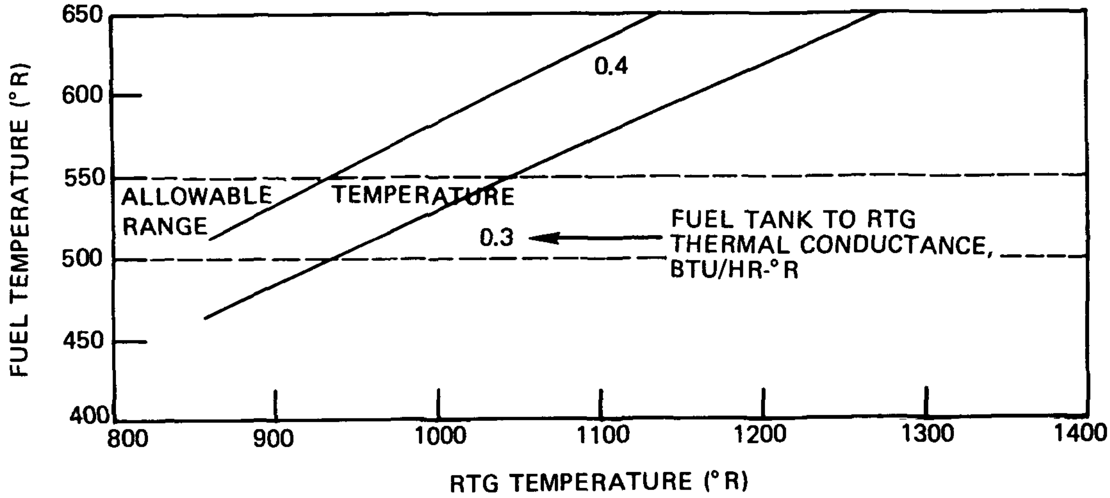


Figure 5.3-10. Effect of Fuel Tank/RTG Thermal Coupling on Fuel Temperature, Fuel Tank Fully Insulated

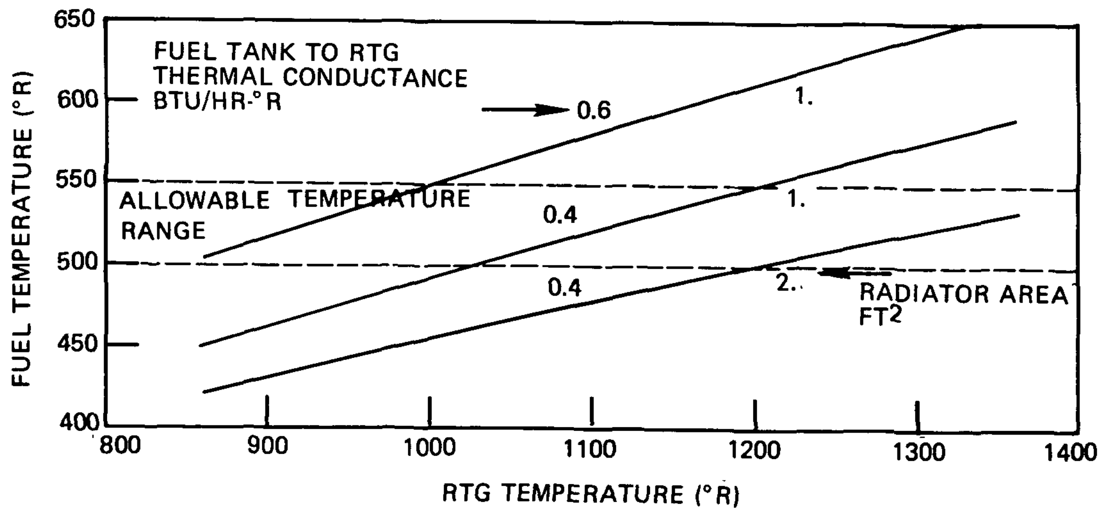


Figure 5.3-11. Effect of Fuel Tank/RTG Thermal Coupling and Radiator Area on Fuel Temperature (Radiator Control)

The ideal thermal response characteristic is a small change in fuel temperature for a large change in RTG temperature (small slope). This is partially realized if a section of the tank insulation (insulation which has a full view of space, but which does not "see" the RTG) is eliminated. The exposed tank surface acts as a radiator which aids in moderating the fuel temperature. This characteristic is shown by the curves of Figure 5.3-11. The open hole in the insulation does drop the overall fuel temperature, but it also reduces the slope of the temperature response curve. To compensate for the temperature drop, the insulation conductance could be reduced. It is more realistic to increase the conductance between the RTG and the hydrazine tank, however.

But again, the slope of the response curves are nearly constant regardless of the radiator and/or fuel tank-to-RTG conductance. In this case, the RTG may vary about 210^{o}R without causing the N_2H_4 to exceed its limits.

If the radiator is replaced by a louver assembly as described in Section 4, the slope of the response curve is further reduced. Figure 5.3-12 shows the response which can be expected with a 2 sq ft louver. In this case, the allowable RTG variation is about 370^{o}R. In contrast to the characteristics of a radiator controlled system, the shape of the response curve changes radically with variations in the louver area and the fuel tank-to-RTG conductance. This is shown in Figure 5.3-13. Changing the fuel tank-to-RTG conductance to 0.6 Btu/hr-^{o}R results in a shift of the curve only; changing the louver area to 4 sq ft stretches the curve. For this particular design the allowable RTG variation is about 440^{o}R. There is an optimum radiator size and fuel tank-to-RTG conductance associated with each RTG operating temperature.

The above discussion emphasized determination of the control system most capable of accommodating wide RTG temperature variations. The true objective is to establish the system most capable of accommodating any type of variation in thermal environment. However, plotting the temperature response of any given system as a function of RTG temperature is a convenient way of displaying the relative thermal control merits of that system. The system which allows the widest RTG temperature variation is also capable of allowing the widest variation in other thermal environments, i.e., external thermal sources.

Of the three N_2H_4 control approaches discussed (full insulation, radiator, or louver), it is impossible to state specifically which is best in this case. If it is certain that the RTG temperature variation will not exceed approximately 100^{o}R and that there are no power dissipating units within the propulsion module which have wide variations in output, then the fully insulated system should be used. If the RTG temperature varies more than 200^{o}R, louver control is the only option. This should not be viewed as an appreciable penalty. Louvers have shown remarkable reliability in actual use and, if properly designed, are fairly light (approximately 0.8 lb/ft^2). Calculated reliability is in excess of 0.999, Reference 4, and to date TRW has had no louver failures on any

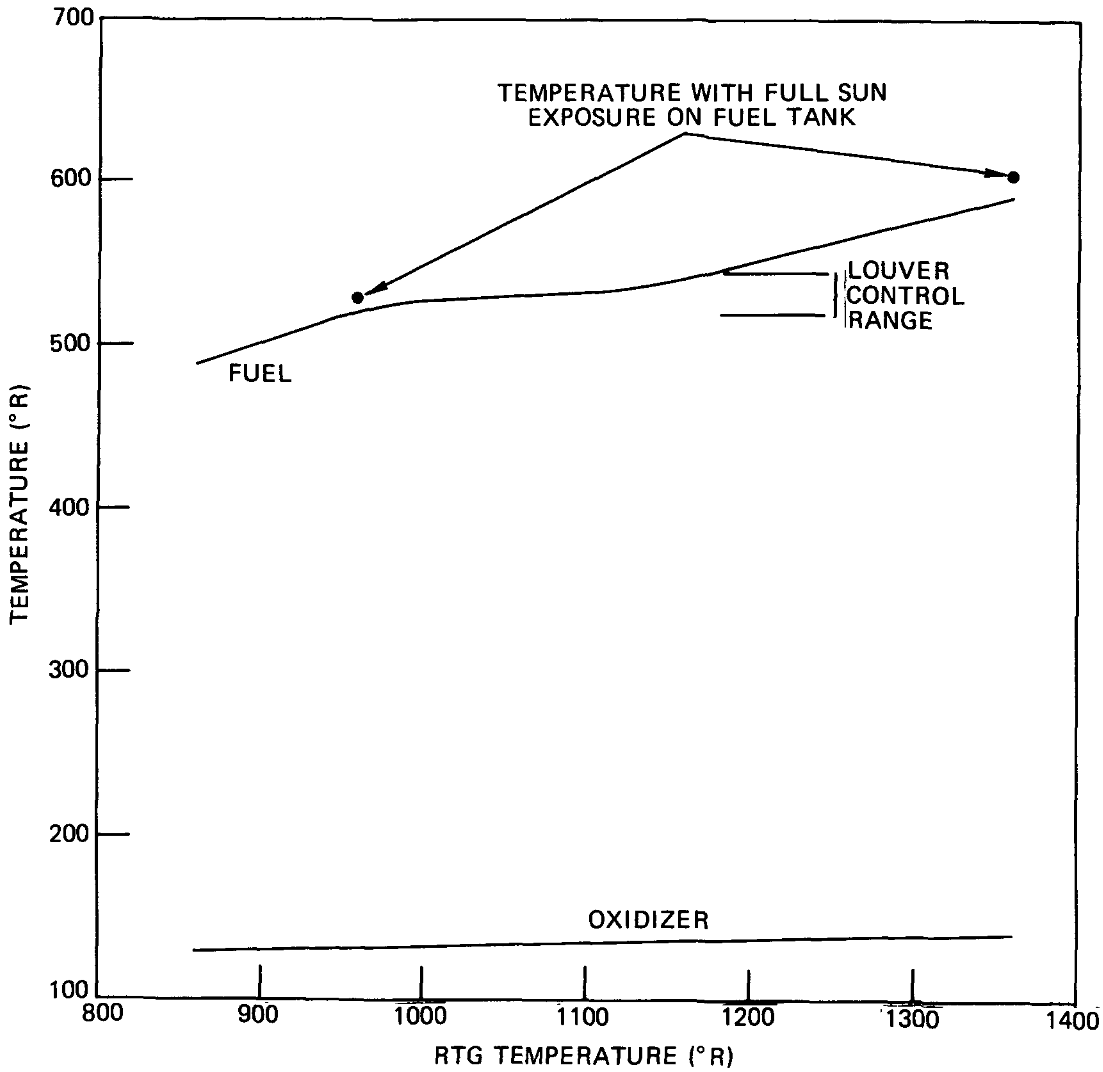


Figure 5.3-12. Effect of RTG Temperature on Propellants (Louver Control)

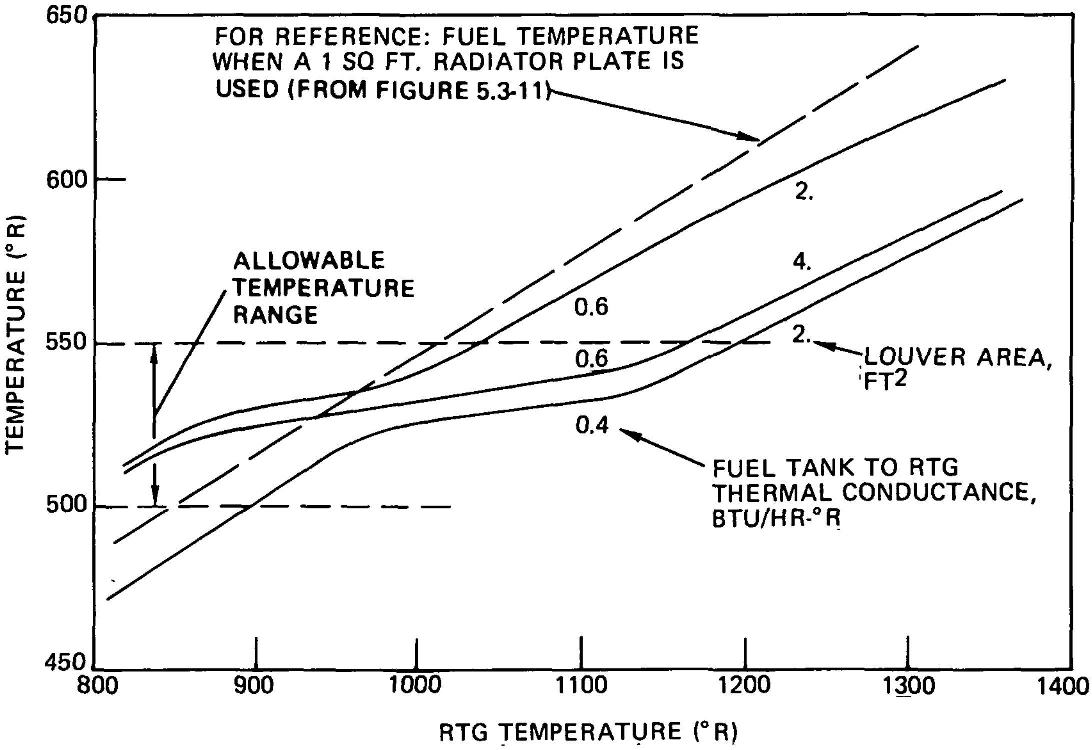


Figure 5.3-13. Effect of Fuel Tank/RTG Thermal Coupling and Louver Area on Fuel Temperature (Louver Control)

vehicle. Considering that the operating temperature and characteristics of the RTG are unknown and may not be known until after the thermal control system design is frozen, it may be wise to arbitrarily decide to use louvers. The penalty in weight and reliability is actually minor.

In the above analysis, the amount of heat obtained from the RTG by the fuel tank was correlated in terms of fuel tank-to-RTG conductance. This correlation is applicable to both a heat pipe and a solid conductor in this area. The above discussion indicated that the conductance between the RTG and the N_2H_4 tank would have to be about 0.3 to 0.8 Btu/hr-^oR for a 960^oR RTG temperature. Translated into heat transfer, this means about 175 Btu/hr. Assuming the RTG is 6 ft away from the tank, a 2-in. diameter solid conductor of aluminum would be required. This is obviously quite heavy, greater than 20 lb. However, a heat pipe would be capable of passing approximately 1500 Btu/hr, and its weight would be about 3 lb. A heat pipe may have four disadvantages. First, a heat pipe is susceptible to corrosion over a long period of time. However, heat transfer fluid loops have been in operation for years and sufficient data is available to establish reliable materials and fabrication methods. Second, the heat pipe would have to be attached to the RTG support boom. Thus, the heat pipe would have to be capable of bending as the RTG is deployed. Flexible heat pipes have been made and it appears that such an approach could be taken here. The simplest approach would probably be to make two or three short rigid heat pipe "legs" which are connected in series by flexible "battery strap" type conductors. These joints would line up with the hinges of the RTG support beam and, therefore, allow realignment as deployment takes place.

The third disadvantage with a heat pipe is that it cannot be turned off, i.e., it is always operating. The fourth disadvantage is that it cannot operate in adverse acceleration fields. The reasons for these two characteristics and their implications in this particular case are discussed in Appendix B. However, preliminary analysis indicates that attaching the heat pipes above the level of the deployed RTG would circumvent both problems. At this time, the use of a heat pipe instead of a solid conductor appears best but this is not certain. It is lightest, has ample heat capacity, and probably would present fewer problems during groundhold. However, it is necessary that additional effort be directed to the area. Specifically, it is necessary to establish the pipe materials, the geometric location, the method of attaching it to the N_2H_4 tank and RTG, and its operational characteristics. In addition, care would have to be exercised to prevent unacceptable thermal gradients in the RTG. Once this information is obtained, it must be compared with similar data for a solid conductor in order to establish for certain that a pipe is superior.

In summary of the flight thermal control of the N_2H_4 tank, the following hardware is recommended (assuming a $960^{\circ}R$ RTG):

- Ten-layer aluminized Mylar insulation blanket
- Four sq. ft. louver for emitting heat to space
- Heat pipe in two or three sections to transport heat from RTG. This recommendation is conditional upon affirmative results from additional heat pipe studies.

5.3.3.2 Flight Thermal Control of F_2 Tank

The first aspects to consider are the insulation requirements. Figure 5.3-14 shows that it is unimportant to flight thermal control how much foam, it any, is applied to the fluorine and helium tanks. This is because the thermal resistance of the foam is small compared to the low potential ability to radiate heat to space at the low temperature of liquid fluorine. Therefore, the groundhold requirement that two inches of foam be used on the tanks is entirely compatible with flight thermal control.

As pointed out earlier, there are insulation requirements in two areas which must be met for minimal heat transfer to the cold tanks. First, a radiation shield must be placed such that it prevents the N_2H_4 tank from viewing the two cold tanks even though the N_2H_4 tank is already insulated with multilayer insulation. Such a shield prevents a heat gain by the fluorine tank of 6 to 8 Btu/hr when at its normal temperature. The second area which must be well insulated is the bottom surface of the spacecraft which "sees" the top of the fluorine tank. The influence of the thermal conductance of this insulation upon the fluorine temperature is given in Figure 5.3-15. This plot applies to the particular situation in which the distance between the spacecraft insulation and the fluorine tank insulation is approximately nine inches. This is a critical factor, nearly as important as the spacecraft insulation conductance. If the spacing is too small, the major portion of the heat which does escape from the spacecraft through the insulation radiates directly to the fluorine tank insulation. In addition, the fluorine

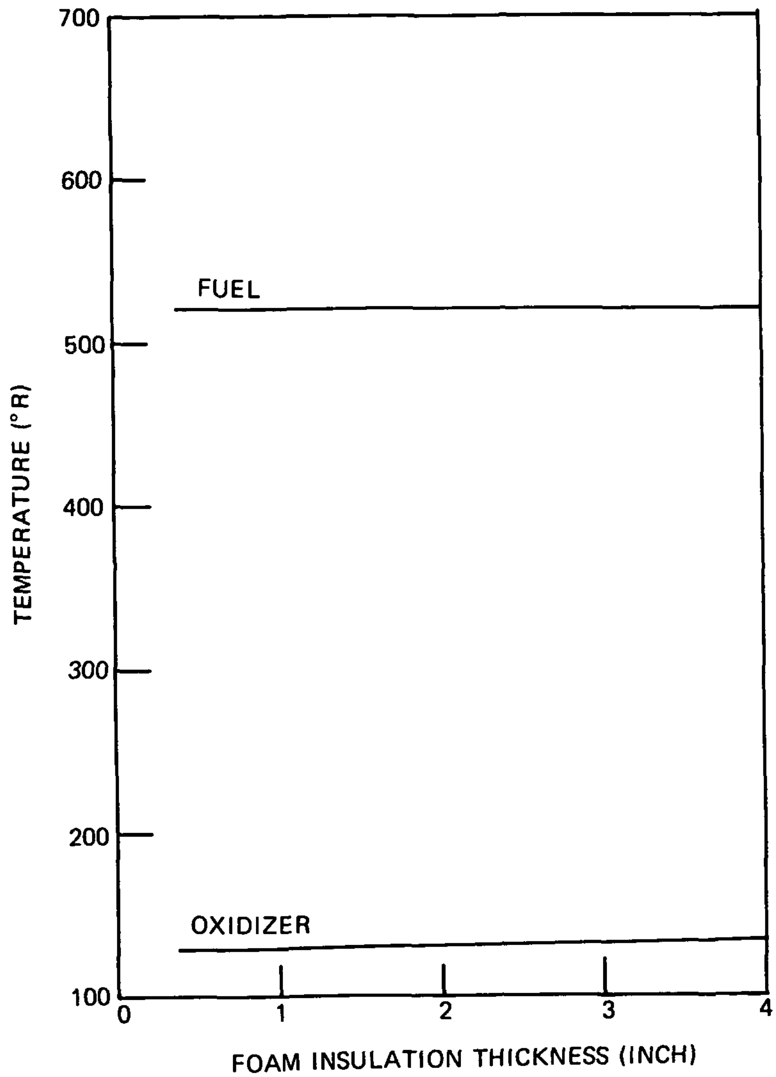
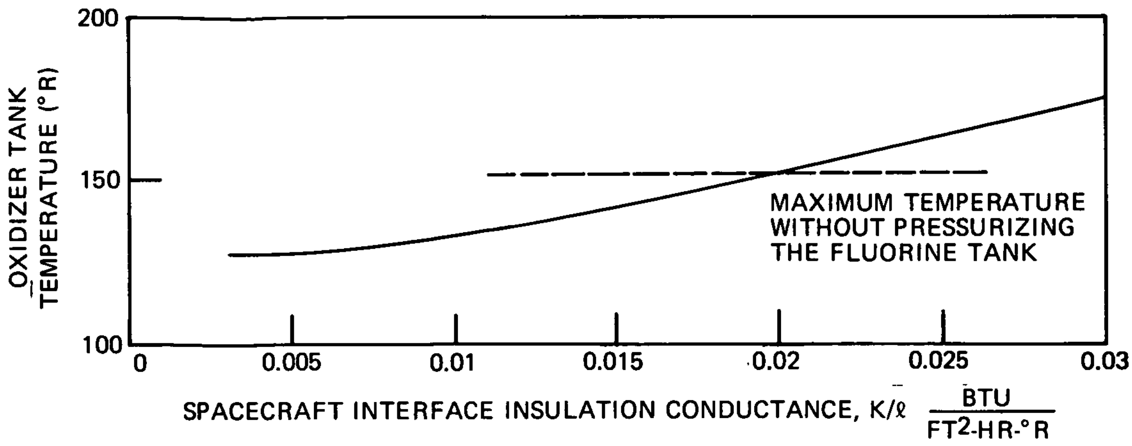


Figure 5.3-14. Effect of Thickness of Oxidizer Tank Foam Insulation on Propellant Temperatures (Fully Shaded Condition)

tank has a smaller view of space and cannot radiate as much energy to space. For the configuration shown in drawing SK 406922, the total heat transfer to the fluorine tank is approximately 10 Btu/hr. Of this, approximately 50% is radiated from the spacecraft to the top of the fluorine tank.



NOTE: FOR NORMAL ALUMINIZED MYLAR INSULATION BLANKET CONSTRUCTION, THE APPROXIMATE RELATION BETWEEN CONDUCTANCE AND NUMBER OF LAYERS IS AS SHOWN:

K/ℓ	LAYERS
0.003	25
0.01	10
0.03	4

Figure 5.3-15. Effect of Spacecraft Interface Insulation Conductance on Oxidizer Tank Temperature

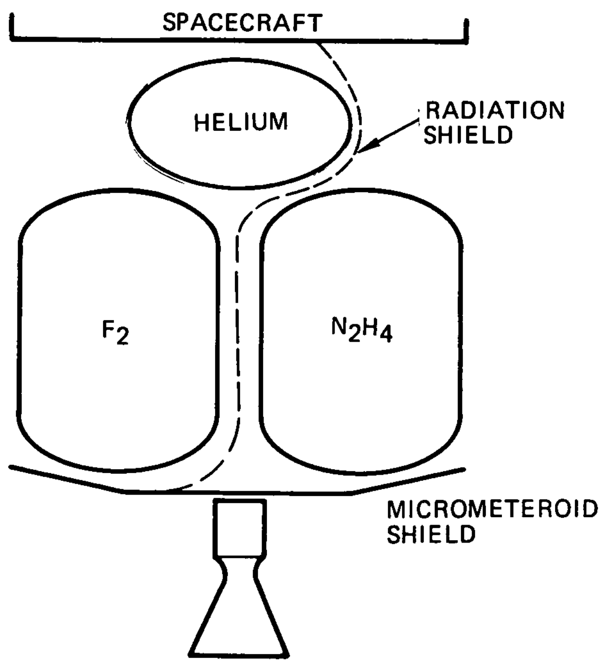


Figure 5.3-16. Alternate Configuration

Limited consideration was given to placing the helium tank above the propellant tanks as shown in Figure 5.3-16. The objective was to give the bottom of the spacecraft and the top of the fluorine tank a better view of space thereby reducing the quantity of heat emanating from the payload to the fluorine and increasing the magnitude of heat transfer from the fluorine tank to space. As it turns out, this arrangement has little or no advantage thermally. Placing the helium tank centrally, near the bottom of the spacecraft, results in the fluorine tank gaining nearly as much additional heat as the incremental amount which it is capable of radiating. The main reasons for this are as follows:

- Additional structure is required which partially blocks the fluorine tank's view of space.
- The helium tank, nestled down between the two propellant tanks, adds blockage to the fluorine tank's view of space.
- The radiation barrier in this configuration encloses more of the spacecraft interface surface on the cold side of the shield.
- The helium tank receives more heat from the spacecraft in this position than the fluorine tank did when the helium tank was at the bottom because it is centrally located under the spacecraft.

It was decided, therefore, to retain the present configuration.

As might be expected, it takes a very small steady-state heat transfer rate to cause excessive temperatures in both the oxidizer and helium tanks. The curves of Figure 5.3-17 show that if both of these tanks are exposed to direct solar radiation at an intensity in excess of 30 Btu/ft^2-hr, the tanks will exceed the 180^{o}R maximum limit. This means that prior to the 350th day from launch, the cold tanks must not be exposed to the sun for any extended period of time.

It is possible to expose the cold tanks to the sun for limited periods of time without exceeding the maximum temperature limits. The length of this duration depends on the intensity of the sun (time since launch), module orientation, and the solar absorptivity of the insulation. Figure 5.3-18 gives the thermal response of the fluorine tanks (with and without second-surface, silvered Teflon) when exposed to solar radiation. This figure shows that the second-surface, silvered Teflon aids materially in increasing the allowable time of exposure to the sun. It also points out the advantage of launching in a substantially subcooled condition. For each degree of

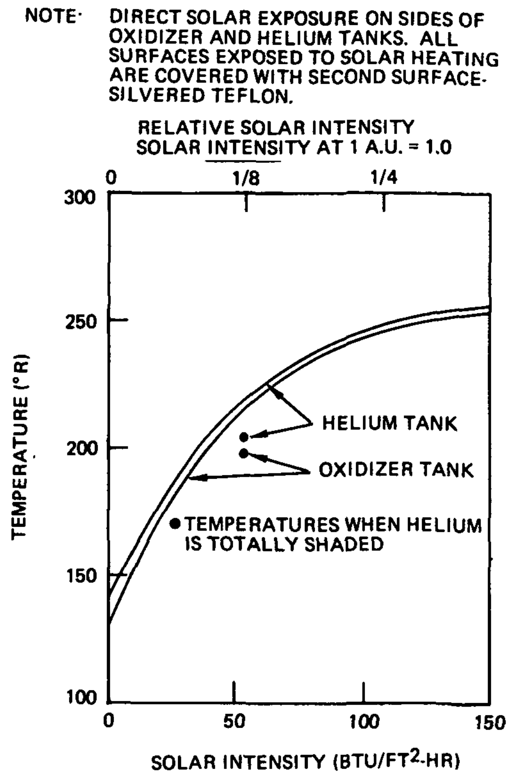


Figure 5.3-17. Effect of Solar Radiation on Steady-State Oxidizer and Helium Tank Temperatures

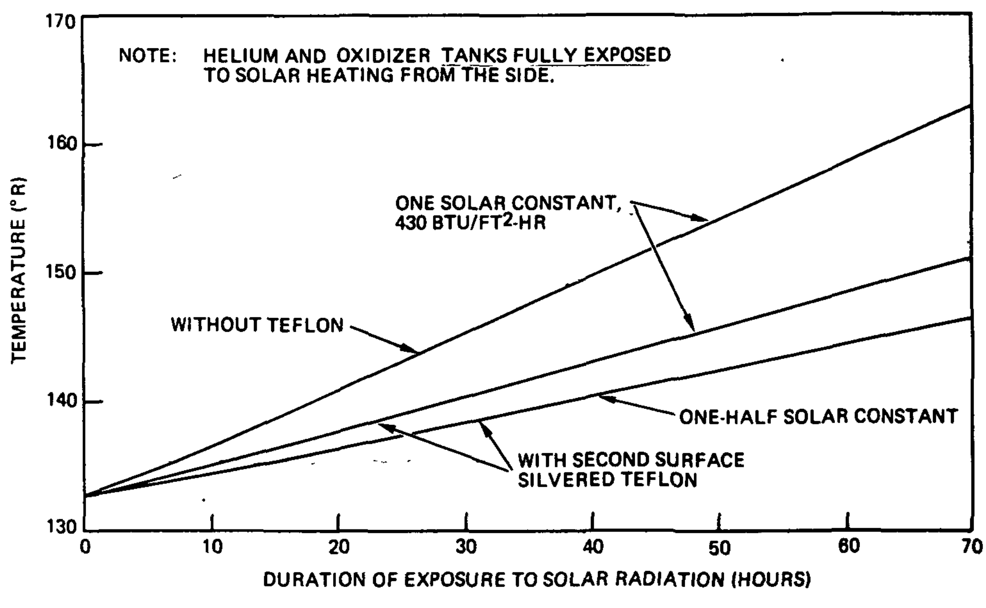


Figure 5.3-18. Effect of Exposure to Solar Radiation on Oxidizer Tank Temperature

subcooling approximately two hours of solar exposure may be tolerated (assuming no local boiling due to zero-gravity heat transfer phenomena). Thus, if LN_2 is used as the groundhold coolant and fluorine tank pressurization is not allowed, the allowable period of full sun exposure would be about six hours. In contrast, subcooling to 133°R would allow at least 40 hours of solar exposure.

From the above discussion, it may be concluded that the fluorine and helium tanks may be kept sufficiently cold provided prolonged exposure to the sun does not occur prior to the 350th day after Jupiter transfer orbit injection. It is also possible to tolerate a limited amount of solar radiation (the amount depends on the solar intensity and initial temperature).

As will be indicated later, there is the possibility that propulsion considerations will dictate warm helium storage, approximately 520°R. Although a detailed analysis of this particular situation was not pursued, there is no reason to believe that the helium may not be stored warm. By rearranging the non-metallic frame members, the conduction of heat from the helium to the fluorine can be effectively eliminated, and the radiation shield can be placed between the helium and fluorine tanks to prevent radiation interchange. It is true that the fluorine tank's view of space would be slightly curtailed. On the other hand, the second largest source of heat to the fluorine tank in the present scheme is the cold helium. To effectively moderate the helium temperature at the higher level might require the installation of a small heat pipe running between the helium and N_2H_4 tanks. To reiterate, it is fairly easy to accommodate a warm helium tank, if warm helium is desirable.

5.3.3.3 Flight Thermal Control of Engine Hardware

The last area to consider in flight thermal analysis is the engine and its related equipment and plumbing. The pictorial conception of the engine which was used as the basis of this phase of the analysis is given in Figure 5.3-19. This is not an accurate picture because the engine configuration is unknown; but it does allow determination of general thermal characteristics.

For purposes of analysis, it was assumed that:

- The N_2H_4 shutoff valve must be above 490°R just prior to engine operation and during all periods that N_2H_4 is in contact with it.
- The catalyst bed must be above 490°R just prior to engine operation and its average temperature at engine shutdown is approximately 1900°R.
- The F_2 shutoff valve must be below 180°R just prior to engine operation and during all periods that F_2 is in contact with the valve.
- At engine shutdown, the outside surface temperature in the region of the combustion chamber dome, throat, and expansion bell will be 1700°R, 3500°R, and 2000°R, respectively.

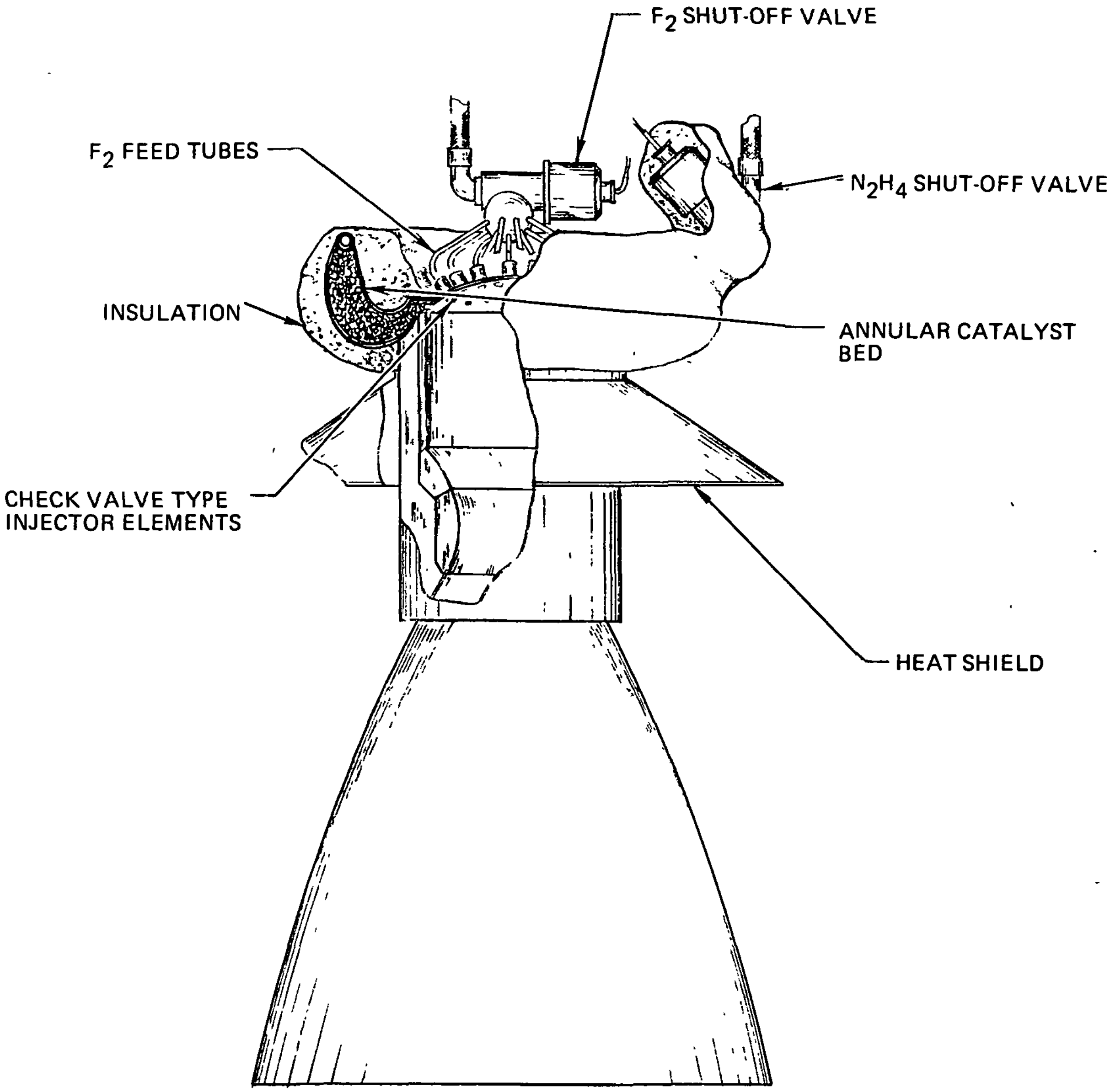


Figure 5.3-19. Pictorial Conception of Bi-mode Engine

The analysis confirmed the findings of previous investigations in the following general aspects:

1) The entire engine assembly temperature during non-operative periods is controlled by its view of the RTG and space. As such, the engine temperature may be held at any desirable level between 110°R and 400°R by controlling these factors.

2) The temperatures of auxiliary equipment, such as valves, will follow the engine temperature fairly closely unless special steps are taken.

3) The temperatures of auxiliary equipment may be made to deviate from the engine temperature by increasing the thermal resistance between the engine and equipment and/or applying heat locally to the equipment.

From these three items, two important consequences emerge. First, it is impossible to control both valve temperatures by coupling them to the engine and then controlling the engine temperature. If both valves are maintained at the same temperature, such an approach is feasible. Secondly, some auxiliary energy source must be available to bring the catalyst bed and N_2H_4 shutoff valve up to 490^oR prior to engine firing.

To establish the general thermal characteristics of equipment in this area, a series of computer runs were made in which the controlling conductances and auxiliary heating were varied. Typical results are given in Figures 5.3-20, 5.3-21, and 5.3-22. From Figure 5.3-20, it can be seen that 10 Btu/hr addition to both the catalyst bed and fuel valve is insufficient and yet the oxidizer valve could be too warm. The addition of another 10 Btu/hr to both the bed and fuel valve does drive the temperatures of these components up to acceptable levels (see Figure 5.3-21), but it also drives the oxidizer valve to excessively high temperatures. There is the possibility that fluorine valve temperatures in the neighborhood of 200^oR to 250^oR just prior to firing might be acceptable since the valve temperature will rapidly (seconds) descend to the liquid fluorine temperature once fluorine starts to flow.

If the fluorine valve must be near the fluorine temperature prior to firing, the equipment would have to be arranged such that the fluorine valve has a view factor of space of about 0.25. By adjusting the system properly, i.e., proper bed-to-engine conductance and oxidizer valve view factor, the temperatures indicated in Figure 5.3-22 may be attained. In this case, all equipment would be within the specified limits provided a bed-to-engine conductance of 0.05 Btu/hr-^oR were provided.

Another computer run was made in which the F_2 shutoff valve was assumed to be mounted on the side of the engine. It was found that this presented no problem, provided the valve is shielded from radiation from the engine.

It can be stated that the soak-back problem is not particularly serious. It will probably be necessary to provide a highly reflecting heat shield as indicated in Figure 5.3-19. Although the valves are maintained at their proper temperature level during engine operation, the insulation will get too hot without radiation protection.

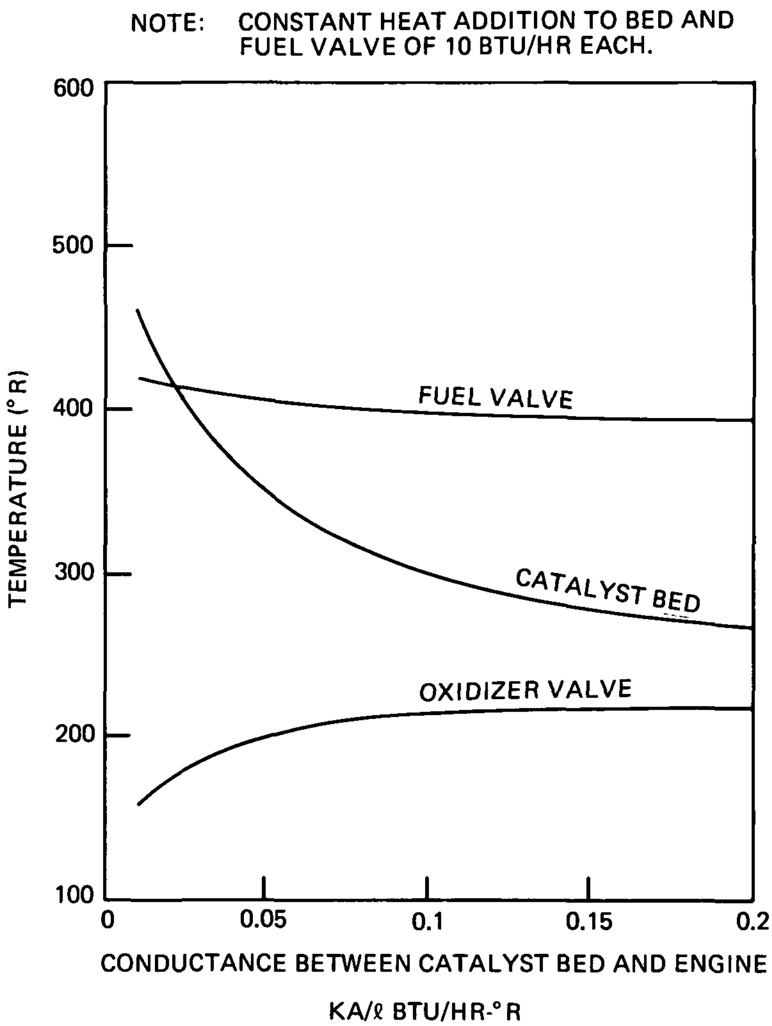
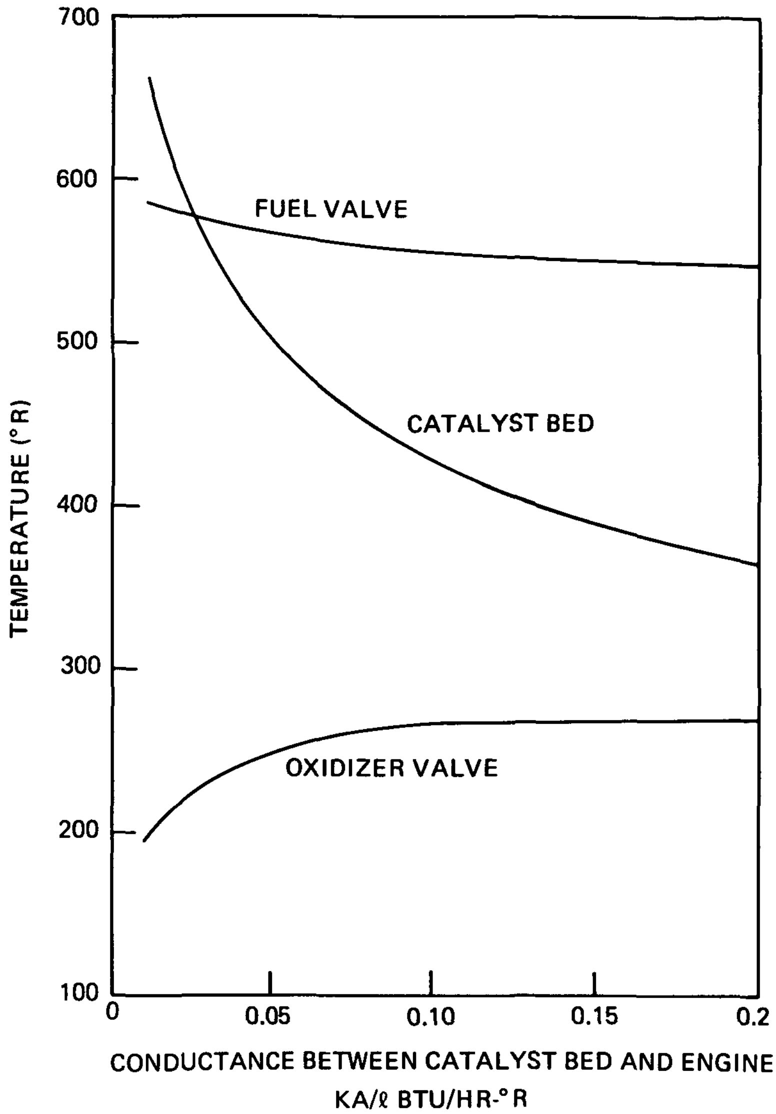
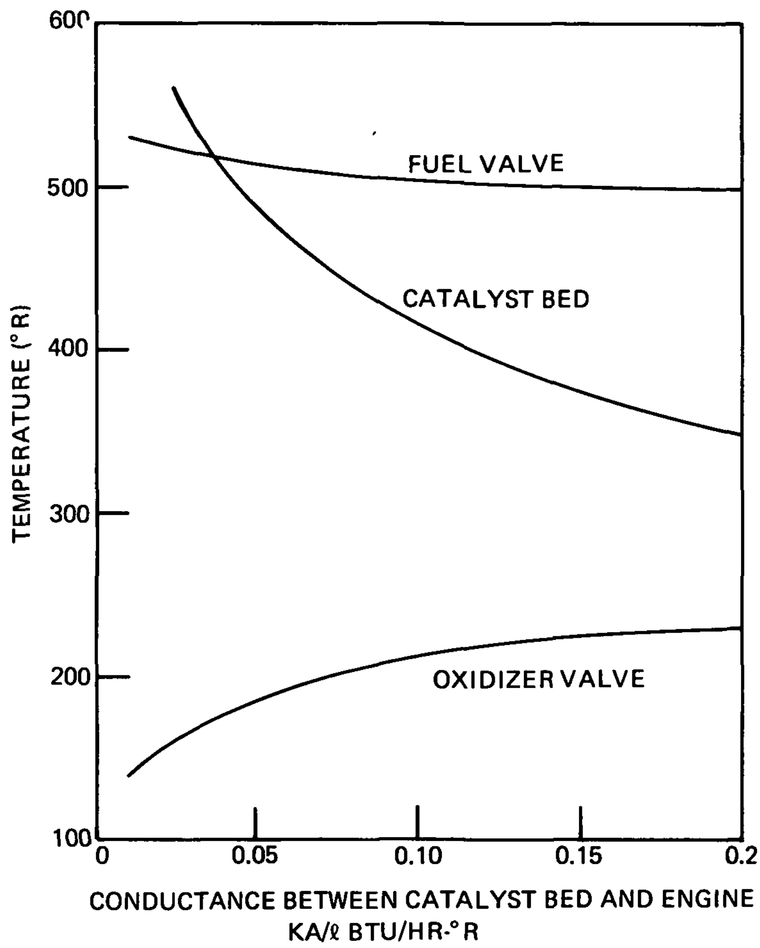


Figure 5.3-20. Effect of Conductance Between Catalyst Bed and Engine on Valve Temperatures (20 Btu Heat Addition, Non-operative Engine)



NOTE: CONSTANT HEAT ADDITION TO BED AND FUEL VALVE OF 20 BTU/HR EACH.

Figure 5.3-21. Effect of Conductance Between Catalyst Bed and Engine on Valve Temperature (40 Btu/hr Heat Addition, Non-operative Engine)



NOTE: CONSTANT HEAT ADDITION OF 16 BTU/HR TO THE FUEL VALVE AND 20 BTU/HR TO THE CATALYST BED.

OXIDIZER VALVE MOVED TO INCREASE VIEW OF SPACE (0.25).

Figure 5.3-22. Thermal Characteristics of Adjusted System (Non-operative Engine)

After shutdown it appears that all components except the fluorine valve will drop in temperature if the auxiliary heating is stopped. Figure 5.3-23 indicates the typical thermal response at shutdown. It shows the fluorine valve rising some 150°R at shutdown. It should be understood that this plot is only a tentative indication of thermal transients at shutdown. An accurate analysis can be made only when the engine system design is specified.

There is also the problem of conditioning the helium before it is used by the propellant tanks for pressurizing. First, it is not clear that conditioning is necessary. If conditioning is necessary, a heat exchanger must be provided. Again, the nature and design of such an exchanger cannot be specified until the helium storage temperature and engine design are determined. However, limit-case hand calculations indicate that providing for helium conditioning should not be a difficult problem. The propellant which is pressurized by the conditioned helium can be used to condition the helium. If this approach is used, the conditioning process will raise the propellant temperature (or lower the propellant temperature depending on the type of conditioning) some 2° to 6°R. A heat exchanger for this purpose could be inside the propellant tank or possibly on the propellant supply line leading to the shutoff valve. In the case of heating cold helium with N_2H_4, care would have to be exercised to avoid local freezing of N_2H_4.

The main lesson to be gained from this information is that it is possible to design the system to hold the components at various temperature levels (approximately 100°R to 400°R), by adjusting the engine-RTG view factor, but such designing must be done in conjunction with the engine design. There are numerous ways of thermally isolating or coupling the various parts of equipment, but it cannot be done without considering the propellant system requirements. The major variables which must be considered are:

1) Allowable propellant line run lengths.
2) Outside surface temperatures of the engine.
3) Shape of catalyst bed.
4) Availability of auxiliary heating power.
5) Allowable soak-back temperature.

It must be remembered that the analysis contained in this Section (5.3.3.3) is based on an assumed engine design and, consequently, indicates only trends and design approaches. When additional propulsion system design criteria become available, it will be necessary to reconsider thermal control system design and engine design.

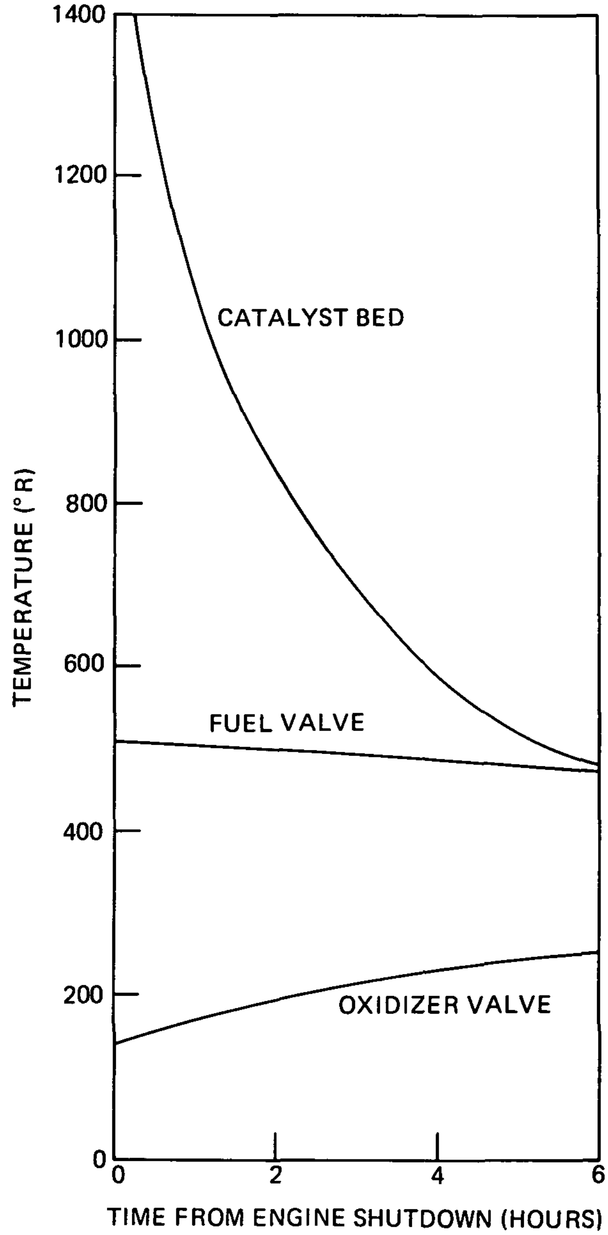


Figure 5.3-23. Thermal Transient After Engine Operation (Non-operative Engine

5.3.4 Summary of Flight Thermal Control Analysis

Summarizing the flight thermal control results, the following major points may be noted:

1) The N_2H_4 tank (and helium tank if stored warm) may be kept within the prescribed temperature limits most effectively by obtaining heat from the RTG (probably by a heat pipe) and radiating excess heat to space via a louver.

2) If the thermal environment does not fluctuate too widely, i.e., RTG temperature variation is no more than 100°R, no form of thermal control need be exercised other than restriction in craft orientation.

3) The use of a louver on the N_2H_4 tank permits the RTG temperature variation to be approximately 440°R. Louvers also make it possible to continuously expose the fuel tank to the sun.

4) Within the constraints of state-of-the-art technology and passive or semipassive control, the fluorine (and helium if stored cold) may be kept below its maximum allowable temperature limit provided continued exposure to solar radiation is avoided, the spacecraft/fluorine tank interface is well insulated, a radiation shield between the tank and the N_2H_4 is provided, and a properly designed nonconductive frame is used. Should the constraints of passive or semi-passive control be removed, different approaches may be considered. Appendix C gives a brief survey of other control schemes which could be considered.

5) Depending on the situation, the fluorine and helium tanks may be exposed to solar radiation for up to 40 hours. Covering the portions of the tanks which receive solar heating with second-surface, silvered Teflon aids materially in lengthening the allowable time of sun exposure.

6) The steady-state, non-operative temperature of the engine support components (valves, catalyst bed, etc.) depends upon the propulsion system design. It is impossible to specify the thermal control system in this area until more definitive information is available concerning propulsion system design criteria. When such information becomes available, it will be necessary to design the thermal control system concurrently with the design of the propulsion system.

7) Any necessary conditioning of the helium can probably be accomplished by a heat exchanger which utilizes the propellant to be pressurized. The thermal design in this area must also await additional information concerning the engine and pressurization system requirements.

The system described in Section 4 is entirely compatible with these conclusions (assuming a widely fluctuating RTG temperature). Stated differently, the optimum flight thermal control concept is compatible with structural, propulsion, and ground-hold thermal control requirements.

6. PROPULSION SYSTEM ANALYSIS FOR THE F_2/N_2H_4 PROPULSION MODULE

The F_2/N_2H_4 propulsion system analysis was limited to defining the F_2/N_2H_4 propulsion system and isolating the problem areas. This analysis did not consider the system performance as a function of mission time, as in the OF_2/B_2H_6 study.

A preliminary estimate of the thermal constraints for the F_2/N_2H_4 module was made early in the program. The results of the estimate based on propulsion system considerations are presented below:

1) Minimum hydrazine temperature - The JPL-established minimum temperature of 500°R is reasonable since it allows a margin of more than 5°R above the freezing point.

2) Maximum hydrazine temperature - The JPL-established maximum temperature of 550°R is a conservative limit, chosen to minimize both spontaneous and catalytic decomposition rates. No other problems are associated with this temperature.

3) Minimum fluorine temperature - 100°R is a rather low temperature for a fluorine system. Ordinarily a higher temperature (approximately 120°R) would be chosen. However, no problems due to increased viscosity are expected, and the unusual character of the injector (i.e., mixing the injected F_2 with hot gas) eliminates thermodynamic freezing problems.

4) Maximum fluorine temperature - Operation at 180°R should create no problem in the fluorine circuit, except possibly that of tank pressure between firings (due to vapor pressure and liquid expansion).

5) Hydrazine circuit - None of the parts of the hydrazine feed system or the injector/catalyst bed should be allowed to fall below 500°R or rise above 550°R when in contact with hydrazine, except for parts located below the isolation valve which may be allowed to rise as high as 580°R. If possible, post-firing heat soak-back should not raise the trapped liquid temperature above 660°R. Local temperatures of 660 to 810°R may cause greatly accelerated decomposition. At temperatures of 810°R or above, depending upon conditions, the hydrazine may detonate.

6) Fluorine circuit - None of the parts of the fluorine feed system or the injector should be allowed to drop below 100°R or rise above 180°R, whenever contacted by fluorine except for parts located below the isolation valve which may be allowed to rise to 200°R. Post-firing heat soak-back should not raise the trapped fluorine temperature any higher than is necessary, since reaction potential somewhat increases as temperature increases.

The subsequent work performed revealed nothing which would invalidate (1) through (5) above. However, later considerations indicated it may be wise to qualify the comments of (6) relative to allowable thermal transients in engine components upon engine ignition. Also, considerations indicate that there are valid arguments for storing the helium warm as well as cold. These two problems, engine-related thermal problems and helium storage temperature, will now be discussed.

6.1 ENGINE-RELATED THERMAL PROBLEMS

Several severe thermal problems will exist in and near the engine unless the engine design is "thermally engineered" in concert with the thermal engineering of the remainder of the module. Specifically, the probable trouble spots are:

1) Temperature control of the catalyst bed. The catalyst bed must be warm enough prior to any start to avoid "flooding." The exact temperature is a characteristic of the starting flow transient, the injector and bed design, and the condition (reactivity) of the catalyst. Typically, a temperature of not less than 480^oR might be specified, although higher temperatures (about 500^oR) provide more safety margin. After start, the problem is to avoid temperatures which are too high. Excess temperature will degrade the catalyst by increasing the rate of evaporation of the surface active material. A reasonable upper limit would be about 2360^oR. As indicated above in the thermal analysis discussion, it is necessary to insulate the surface of the catalyst bed in order to maintain the minimum temperature upon engine ignition. Yet this insulation may cause the excess temperatures during operation which must be avoided.

2) Fluorine injector and valve temperature. Monopropellant operation generates heat which may be conducted, convected, or radiated to the liquid fluorine injector and valve during periods when these parts are not cooled by the flow of fluorine. Two possible problems are distortion, or other physical deterioration, due to overheating and excessive heat addition from these hot parts to the initial fluorine flow. Development tests may not reveal the true in-flight heat loads on these parts unless the thermal environment (conduction paths, radiation view factors, etc.) are the same as in the module.

3) Heat Soak-back. Flow of heat from the engine, either during or after firing, upstream into the propellant feedlines can cause several problems: liquid bulk expansion, propellant vaporization, hydrazine decomposition, increased fluorine attack, distortion of parts etc. To assure that each of these problems is controlled, it is necessary to devise a detailed thermal model of the engine and its support equipment with realistic engine heat load inputs and to analyze the magnitude of each troublesome mechanism.

6.2 OPTIMUM HELIUM STORAGE TEMPERATURE

In establishing the best temperature range for storing the helium, three points must be considered:

1) Design Requirements. At what temperature should the helium be stored in order to facilitate design objectives, particularly in relation to thermal control and safety?

2) Propulsion System Requirements. At what temperature should the helium be stored for the most efficient and reliable propulsion system?

3) Weight. At what temperature should the helium be stored in order to realize the minimum weight pressurization system?

As indicated in the thermal control discussion above, there are no apparent reasons, from a thermal point of view, for preferring either a cold helium storage system or a warm helium storage system. Either approach may be taken and it is entirely possible to accommodate either approach in the design.

As for the propulsion system requirements, the real problem here is reliability. The use of warm or cold helium should not directly affect engine performance. Only in relation to (a) helium requirements and (b) helium conditioning prior to its use in pressurizing the propellants does the helium temperature affect the propulsion system. As will be shown below, this bears upon the design of the auxiliary engine equipment.

The establishment of the minimum weight pressurant system is an extremely complicated problem which is not adequately understood. Experimental data in this area are insufficient.

Two significant masses vary with storage temperature: the helium tank and the helium gas. First, consider the helium necessary to pressurize the two propellant tanks. The total mass of helium in the ullages at the end of the last firing is a function only of the helium partial pressures and the ullage gas temperature, since tank total pressures and final ullage volumes are fixed. For the case of warm helium, the partial pressure of helium in the fuel tank is essentially the same as regulated pressure since the partial pressure of the N_2H_4 is less than 1 psi. The partial pressure of helium in the oxidizer tank is unknown because the liquid propellant surface temperature and the propellant vapor density variation within the ullage are not presently calculable using available analytical methods. Grossly simplified models must be introduced to obtain any numerical answers; a possible choice is to assume that the liquid surface temperature is the same as the liquid bulk temperature (i. e., it is neither warmed by the ullage gases nor chilled by removal of the latent heat of vaporization), and that saturated vapor corresponding to that surface temperature exists uniformly throughout the ullage. Then, once the F_2 vapor partial pressure is determined, the helium partial mass is simply a function of the only remaining variable--final average ullage gas temperature. In any event, the results are only as good as the assumptions made to simplify the problem.

If the reverse situation exists, that is, if helium at the fluorine temperature is used as the pressurant, the analytical problem is not as severe. The solution of the heat and mass transfer problem within the F_2 tank to a substantial degree disappears because of the small temperature gradients within that tank. However, there is a problem within the N_2H_4 tank, because the ullage gas temperature is now unknown.

It is possible to consider these problems by what is essentially an empirical approach (Reference 5) which depends upon experimental data to determine tentative comparative weights of a warm and cold helium system.

First, the final pressure remaining in the helium tank was calculated for the case where the storage temperature equals fluorine bulk temperature (i. e., $150^{\circ}R$) and 36 lb of gas was stored. It was assumed that a heat exchanger provided the necessary energy input to make the mean inlet temperature for the fuel tank equal to 530°R. No collapse factor* was used for either tank. Mid-course firings were neglected, and no external heat inputs except for that through the heat exchanger were considered. Between the orbit insertion and orbit inclination firings, all gases in the ullages were assumed to come to equilibrium with the liquid bulk temperatures. Absorption of helium into the propellants was neglected. The exact manner in which these calculations were carried out is given in Reference 1.

The results indicate that about 1050 psia will remain in the helium tank after the last firing when the storage temperature is held at 150°R.

A similar calculation was then made for an equal mass system (helium plus tankage) stored at 530°R. In this case the calculated mass of helium added to the fluorine tank was multiplied by a collapse factor of 1.82. No heat exchanger was assumed; therefore, the mean inlet temperature for the fuel ullage was lower than for the cold gas storage system. This calculation indicates that the helium will be exhausted before completion of the orbit inclination burn. Translated into weight, the warm system contains only 17.8 lb of helium which is insufficient for the mission.

The lower density of helium when stored at higher temperatures means that the ratio of helium to tank mass is far less advantageous for warm storage than for cold storage. For example, if the comparison is between helium stored at 550°R versus 180°R, the tank plus helium masses are approximately 6.22 and 3.07 times the total mass of helium, respectively; i.e.,

$$M_T + M_{HE} = 6.22\ M_{HE} \text{ at } 550^{\circ}R$$

$$M_T + M_{HE} = 3.07\ M_{HE} \text{ at } 180^{\circ}R$$

Furthermore, the collapse factor in the fluorine tank ullage works to the disadvantage of the warm system, because the effective mean temperature is not 500°R, but about 275°R. In the cold storage case, there would be no collapse; therefore, the calculated mean inlet temperature is a truer measure of the effectiveness of the helium in pressurizing the oxidizer. Yet another effect is the collapse factor suffered by the helium in the ullage during the coast between firings. This initial mass of helium will be cooled to essentially the liquid bulk temperature before the orbit inclination firing is made, so its final collapse factor will be more like 500/150 = 3.33. That is, for the final firing, only the helium pumped into the fluorine ullage during the firing period will benefit from the higher storage temperatures. All the helium stored in the ullage during the coast period will be no more effective than if its mean inlet temperature were the same as the liquid fluorine temperature.

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*Collapse factor is the ratio of warm gas actually needed to pressurize a cold liquid to the amount needed if the warm gas is not chilled down by the colder liquid.

Though the above comments indicate definite trends, they must be considered with caution since the collapse factor correlation taken from Reference 5 was not made for the propellant, pressures, or size of tank involved in the present case (i.e., correlated up to 100 psia instead of to 300 psia, and in tanks greater than 4 ft in diameter compared to the present 2.7 ft diameter). A brief survey of the literature disclosed no similar set of empirical correlation coefficients for fluorine.

Even though the cold helium system appears to present a considerable weight advantage, one other point must be considered. The material weight advantage is realized only if the helium used for pressurizing the N_2H_4 tank is heated prior to its use in the N_2H_4 tank. It must be heated to between 500°R and 550°R if excessive chilling or heating of the hydrazine surface is to be avoided.

Two problems are immediately evident in designing such a helium heat exchanger: (1) a nearly constant temperature source of heat is required, and (2) the heat exchanger may introduce a pressure loss which will result in a lower-than-regulated pressure level within the fuel tank.

It is estimated that 110 Btu must be added to the helium during each mid-course maneuver, 830 Btu during the orbit insertion maneuver, and 770 Btu during the orbit inclination maneuver. (These amounts are for the actual propellant expulsion period only, and do not include additional amounts of heat needed to warm up the gas used to pre-pressurize the fuel tank ullage to operating level prior to firing the engine.)

Removing these quantities of energy from a passive heat source seems impractical unless that heat source is the hydrazine itself. For example, the tank shell would drop nearly 55°R if 830 Btu were extracted from it alone. Using heat from the hydrazine poses the problem of excessive chilling at the end of the last firing because of the diminishing mass of hydrazine. An approximate calculation shows a final temperature of about 6°R colder than initial bulk temperature for this case. If this were the case, hydrazine initially at 500°R before the orbit inclination maneuver probably would freeze on the heat exchanger at the end of the firing period. From a practical standpoint, this type of heat exchanger is undesirable in that it would complicate the fuel tank design or fabrication.

An "active" source of heat would appear to be more attractive provided it was maintained at a nearly constant temperature during firings so that the helium outlet temperature would remain within the narrow range specified. The use of a heat exchanger on either the thrust chamber or catalyst bed seems undesirable because of the high temperatures attained; two possible exceptions would be a regeneratively-cooled chamber or a location near the liquid fuel injector. At the present time, it is not possible to evaluate either of these alternatives since the variables involved will be very sensitive to the detailed engine design. A parametric study is feasible, but was not conducted because it is beyond the scope of the presently funded tasks.

Two alternative "active" heat sources are available. First, the RTG. By a suitable arrangement, it probably is possible to obtain sufficient heat by conduction. The problems involved here include (1) the remoteness of the RTG requiring long tubing runs to carry the gas to and from the heat exchanger, and (2) the in-flight deployment of the RTG requiring two flexible section in the tubing runs. A much easier design solution is the use of a heat exchanger in the hydrazine feedline which would extract heat from the liquid flow. During steady-state operation, this process would chill the hydrazine by less than $3^{o}R$. A special advantage is that the helium outlet temperature could never become excessively high. The liquid side pressure loss should be small.

Imposition of a pressure loss on the gas side of the heat exchanger is a more serious problem, for it can result in mixture ratio shifts unless held nearly constant.

Of course, one way to avoid all the problems associated with the helium temperature is to use two helium tanks, one maintained at the hydrazine temperature and the other at the fluorine temperature. This possibility has not been investigated, but it would appear that the weight penality would not be severe and such a pressurant system would probably be highly reliable.

6.3 SUMMARY OF F_2/N_2H_4 PROPULSION SYSTEM ANALYSIS

Again, it should be noted that the problems discussed above have not been completely solved. To adequately solve the helium temperature problem, considerably more effort is required and the particular design and operating characteristics of the actual engine must be considered. The same may be said of the thermal problems of the engine support equipment (valves, catalyst bed, etc). But it is possible to make the following generalized comments:

1) From a propulsion system standpoint, there is no apparent reason propulsion system considerations would dictate a system different from that described in Section 4. The thermal control requirements are quite compatible with the propulsion system requirements.

2) Based on a rough hand-calculation, it appears that a cold helium storage system presents a weight advantage.

3) A warm helium storage system presents the least amount of design problems.

4) Several thermally-caused problems may arise in and near the engine which must be solved in conjunction with engine design.

7. CONCLUSIONS AND RECOMMENDATIONS CONCERNING THE F_2/N_2H_4 PROPULSION MODULE STUDY

The general structural configuration adopted for the OF_2/B_2H_6 module (two propellant tanks of equal size, one pressurant tank located below the propellant tanks, and a tubular frame) should also be used for the F_2/N_2H_4 module. The use of same size propellant tanks results in a substantially oversize fuel tank; but adjusting the supporting structure to accommodate different tank sizes results in more complexities and more weight. Closed-cell foam, probably 2-in. thick, should be used to insulate the fluorine tank and 10 layers of aluminized Mylar should be used for the hydrazine tank.

LN_2 circulated through a coil inside the fluorine tank may be used as the ground-hold coolant, but it may be necessary to allow the fluorine tank to self-pressurize to 60 psia. If it is desirable to substantially subcool the fluorine, helium prechilled by liquid hydrogen or subcooled LN_2 may be used as the coolant. A helium cooling system, however, would be complicated and susceptible to malfunction.

Flight thermal control of the hydrazine tank may be effected by supplying heat to the tank and moderating the tank temperature by either a radiator or louver assembly. A small heat pipe running from the RTG can supply the heat. The choice of a radiator or louver assembly to moderate the temperature depends on the stability of the RTG temperature. If the variation in the RTG temperature is greater than $\pm 100^{\circ}R$, a louver assembly should be used.

To control the temperature of the fluorine tank during flight, four requirements must be met:

1) The fluorine tank must be shielded from continuous solar radiation for the first 350 days of the mission. Some intermittent solar heating may be accommodated, however.

2) The main support frame must be partially constructed of fiberglass members to eliminate heat conduction between the two propellant tanks.

3) A multilayer, aluminized Mylar radiation barrier must be placed between the two propellant tanks.

4) The spacecraft surface which views the fluorine tanks must be well insulated with multilayer, aluminized Mylar.

Though the exact manner in which the helium tank temperature is to be controlled cannot be established because it is dependent upon the storage temperature, the following provisions may be stated.

1) If the helium is stored at the fluorine temperature, it should be thermally connected to the fluorine tank by a solid conductor. A heat pipe for this purpose is undesirable since very little experience is available concerning cryogenic heat pipes. The weight penalty might be large (5 to 10 lb), but that would be offset by the high reliability of a solid conductor.

2) If the helium is stored at the hydrazine temperature, a heat pipe between the hydrazine tank and helium tank could be used in order to realize a weight savings, provided it is determined that a heat pipe between the RTG and hydrazine tank is to be used. Otherwise, a solid conductor should be used.

Before a complete thermal control system for an F_2/N_2H_4 module can be specified, additional areas must be studied to determine:

1) Optimum helium storage temperature (warm or cold).

2) Engine component temperature control for a detailed engine design.

3) The feasibility of using small radioisotope heaters bonded directly to the hydrazine tank as the heat source.

4) The feasibility of using a heat pipe to obtain heat from the RTG.

5) The required fluorine temperature at launch (and consequently the groundhold coolant requirements).

6) Zero-gravity heat transfer phenomena inside the fluorine tank during times of solar heating.

Helium Storage Temperature

As previously indicated, helium can be stored in either a warm or cold state (or warm and cold if two tanks are provided). Therefore, a helium storage temperature study would be primarily a propulsion-stage design study. Such a study cannot be independent of thermal constraint considerations. For example, if the propulsion analysis indicates that helium should be stored cold, but conditioned prior to its use in the hydrazine tank, thermal considerations may dictate the routing of the helium lines, the routing of the main hydrazine supply line, the sequence of operation prior to engine ignition, the location of the helium heat exchanger, etc.

A helium storage temperature study should determine the following:

- The storage temperature of the helium.
- The desirability of helium conditioning prior to its use.
- The heat exchanger requirements if helium conditioning is required.
- The transient temperature histories of the propellant ullages.
- The transient temperature history of the helium tank.
- The weight and reliability differentials associated with helium storage at different temperatures.

Any assumptions and/or analyses made in this area should be verified by tests. However, it should be recognized that only limited applicable data may be obtained by tests, because the effects of a zero-gravity field would materially change the results.

All testing should be carefully planned with a clear understanding of this limitation. Cryogenic propellant/helium pressurant data from past flight programs (Atlas, Centaur, Saturn) should be carefully screened for applicable information.

Engine Component Temperatures

The fuel components of the engine (catalyst bed, valve, lines, etc.) must be warm at all times that fuel may come into contact with them. This requirement, however, cannot be considered apart from the following items:

- Fuel component insulation requirements.
- Effect of subsequent engine operation on insulation and component temperature.
- Oxidizer valve temperature requirement.
- Auxiliary power requirements.
- Heat soak-back after engine operation.

Therefore, a second program should be initiated to study the above items. The specific hardware configuration of the engine must be used in the study and, for best results, the study should be concurrent with engine design. Ignoring engine design in this study will probably result in unreal and unreliable answers. Performing this work after the engine has been designed may result in the discovery of additional thermal constraints necessitating re-design of portions of the engine.

In addition, this particular study must account for the radiative thermal coupling between the RTG and the engine nozzle. Since the heat radiated from the RTG determines the non-operative engine temperature, and engine temperature in turn determines to a large extent the temperature of attached components, any attempt to design the hardware in this region without considering the RTG (its temperature, size, and location) is futile.

Aside from the fact that the F_2 valve design and location influences the N_2H_4 valve design and location, there are problems with the F_2 valve itself. First, it must be determined if temporary flashing in the valve is permissible and, secondly, the valve must be designed to withstand the temperature rise which will occur at engine shutdown.

Radioisotope Heaters

The use of radioisotope heaters as the heat source for the hydrazine tank has real advantages. A careful study of the limitations imposed by the spacecraft on hard radiation in the vicinity of the module is recommended. If no such limitations exist, use of small heaters as heat sources should be considered.

If it is found that hard radiation is not a problem but that radioisotope heaters are not applicable, additional consideration must be given to placing the RTG close to the module (less than 3 feet away) and obtaining the needed heat by radiation to a second set of louvers which directly view the RTG. This study would have to consider a tradeoff between louvers, heat pipes, solid conductors, and structure in the areas of weight, reliability, and mechanical design.

Heat Pipe Application

Associated with the study listed above is the necessity of establishing with certainty that a heat pipe can be used to obtain heat from the RTG. The object of this area of study would be to establish:

- Heat pipe materials and fluid.
- Method of construction.
- Method of construction.
- Location and method of attaching to N_2H_4 tank and RTG.
- Location and orientation during groundhold and powered flight.
- Consequences should the heat pipe temporarily stop functioning due to adverse acceleration field.
- Consequences arising from continuous operation of heat pipe.
- Method of testing assembled RTG/heat pipe/N_2H_4 system.

After the design and thermal characteristics of the heat pipe are established, the tradeoff study indicated above, i. e. , weight and reliability, must be conducted. This is not a difficult area of study but unless it is considered, it must be assumed that only a solid conductor may be used.

Groundhold Cooling of Fluorine

It is mandatory that the required fluorine temperature at launch be established. This would be established by considering the mission profile during the first 30 days. Should it be found that the fluorine must be colder than that which can be achieved with LN_2, this study will have to be extended to include an investigation of the alternate methods of effecting groundhold cooling: subcooled liquid nitrogen and chilled helium.

The investigation would have to consider:

- Existing groundhold facilities, i. e. , availability of LN_2 dewars, LH_2 dewars, gaseous helium, vacuum pumping equipment, and vacuum-jacketed transfer lines.
- Required temperature range of the F_2 at launch.
- Time required to hold fluorine at low temperature.
- Reliability.

- Safety.
- Cost.

Along with this study, logistics problems during the entire groundhold phase may have to be considered. It may be necessary to discontinue cooling for substantial lengths of time (during ground transit or mating sequences). To accomplish this, substantial subcooling prior to stopping the cooling process may be necessary. It may even be necessary to provide a portable coolant supply system. These problems can be answered only by considering the cooling requirements in the light of ground handling equipment capability.

RTG Effects

Another area which should be studied (although not directly a part of the thermal control system) is the effect of draining heat from the RTG. For the most part, the heat drain from the RTG will be constant ($\pm 10\%$), but it may be necessary to carefully control the manner in which this energy is drained from the RTG to avoid unacceptable thermal gradients. This is no problem, of course, if the RTG is not used as the heat source for the hydrazine tank.

Zero-Gravity Heat Transfer

There will be no problems relative to local boiling (and the resulting tank pressurization) if the fluorine tank is shielded from solar radiation. If it is necessary to expose the fluorine tank to the sun for more than one or two hours, this problem must be investigated. The study should establish the existence of those zero-gravity heat transfer problems which have been postulated and how they may be circumvented.

In this case, it is strongly suggested that data from past flight programs be searched for applicable information.

All the above study areas must be considered in order to establish the design of a F_2/N_2H_4 propulsion module. Some of the problems which will be encountered will be difficult (particularly the helium storage temperature problem). However, all the problems may be solved with reasonable effort.

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2. "Spray Application of Two Pound Density Polyurethane Foam Material," Specification MA 0-606-050; "Specification for Two Pound Density Polyurethane Spray Foam," Specification MBO 130-077; "Specification for Two Pound Density Polyurethane Pour Foam," Specification MBO 130-046; North American Rockwell, Huntington Beach, California.

3. P. E. Glaser, et al., "Thermal Insulation Systems," Report No. SP-5027, National Aeronautics and Space Administration, 1967.

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APPENDIX A

WEIGHTS AND STRESS TABLES

The following figures and tables present in detail the geometrical arrangement of the structural elements, ultimate loads on each member and the size and weight of each truss element. Where items have not been designed, conservative estimates of the weights have been included in the weight summary.

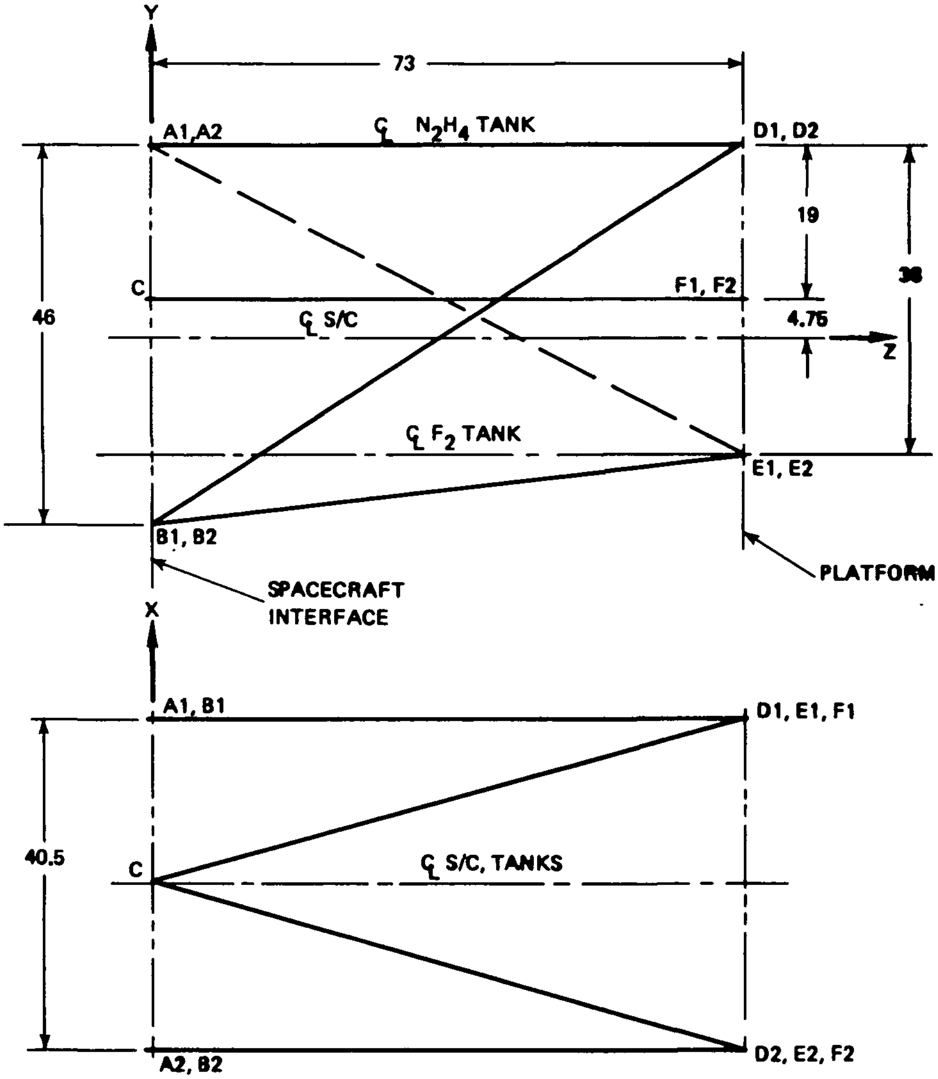


Figure A-1. Upper Truss Structure

Table A-1. Upper Truss Structure

<u>Ultimate Loads</u>

MEMBER	CONDITION 1	2	3a	3b	3c	3d	4a	4b	4c	4d
	$n_z = -7.5$	$n_z = 10$	$n_x = 3.75$		$n_x = -3.75$		$n_y = 3.75$		$n_y = -3.75$	
			$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$
A1 D1	2070	-2760	158	2570	-2920	-500	-500	1910	-2260	160
A2 D2	1605	-2140	580	2450	-2720	-850	-9820	-7950	7680	9550
B1 E1	2680	-3580	1060	4190	-4640	-1510	7210	10340	-10790	-7660
B2 E2	2210	-2950	-3540	-955	585	3170	-2130	453	-823	1760
B1 D1	-547	730	-2400	-3030	3130	2490	-10960	-11600	11690	11050
A2 E2	525	-700	-3350	-2740	2650	3260	10480	11100	-11180	-10570
C F1	0	0	-18850	-18850	18850	18850	0	0	0	0
C F2	0	0	18850	18850	-18850	-18850	0	0	0	0

<u>Member Sizes and Weights</u>

MEMBER	MAXIMUM COMPRESSION	L	t	DIAMETER	WEIGHT (LBS) TUBE	* END-FITTINGS	TOTAL
A1 D1	2920	73	.0104	2.40	.83	.31	1.14
A2 D2	9820	73	.0208	2.85	1.50	1.04	2.54
B1 E1	10790	73.4	.0208	2.96	1.56	1.14	2.70
B2 E2	3540	73.4	.0104	2.56	.89	.37	1.26
B1 D1	11600	86.3	.0208	3.37	2.09	1.23	3.32
A2 E2	11180	82.3	.0208	3.22	1.91	1.19	3.10
C F1	18850	75.7	.0208	3.63	1.98	2.00	3.98
C F2	18850	75.7	.0208	3.63	1.98	2.00	3.98
							22.02

* Estimated from weight of typical end fittings.

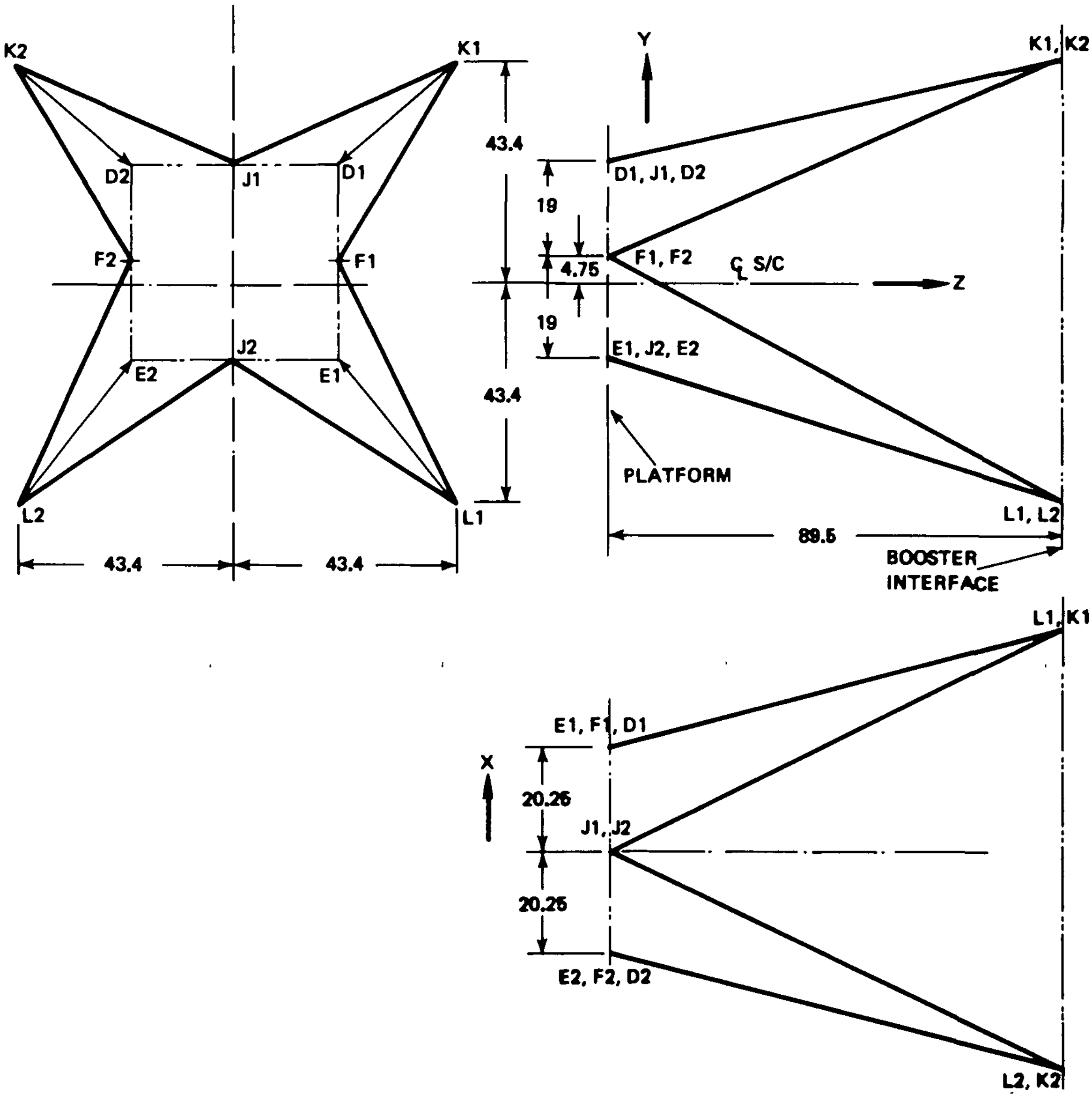


Figure A-4. Lower Truss Structure

Table A-2. Lower Truss Structure

Ultimate Loads

MEMBER	CONDITION									
	1	2	3a	3b	3c	3d	4a	4b	4c	4d
			$n_x = 3.75$		$n_x = -3.75$		$n_y = 3.75$		$n_y = -3.75$	
	$n_z = -7.5$	$n_z = 10$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$
D1 K1	2020	-2700	-2730	-373	33	2390	-10350	-7990	7650	10010
D2 K2	1910	-2550	965	3200	-3520	-1285	-10280	-8050	7730	9960
E1 L1	3210	-4280	920	4670	-5200	-1455	7380	11130	-11660	-7920
E2 L2	3320	-4430	-7310	-3430	2880	6750	7310	11180	-11740	-7860
J1 K1	5120	-6840	-6000	-20	-840	5140	-3530	2450	-3310	2670
J1 K2	5200	-6930	-1870	4190	-5060	1010	-3580	2490	-3360	2710
J2 L1	8460	-11290	-12720	-2840	1425	11300	-5800	4080	-5500	4380
J2 L2	8400	-11210	3650	13450	-14860	-5050	-5760	4050	-5460	4350
F1 K1	2260	-3010	-11860	-9220	8850	11480	-5880	-3240	2870	5500
F1 L1	-2250	3000	-9000	-11630	12000	9380	6040	3420	-3040	-5670
F2 K2	2320	-3100	8850	11560	-11950	-9240	-5950	-3240	2850	5560
F2 L2	-2320	3090	12020	9310	-8930	-11630	6130	3420	-3040	-5740

Member Sizes and Weights

MEMBER	MAXIMUM COMPRESSION	L	t	DIAMETER	WEIGHT (LBS)		
					TUBE	* END-FITTINGS	TOTAL
D1 K1	10350	94.5	.0208	3.44	2.34	1.10	3.44
D2 K2	10280	94.5	.0208	3.44	2.34	1.10	3.44
E1 L1	11660	97	.0208	3.66	2.55	1.24	3.79
E2 L2	11740	97	.0208	3.66	2.55	1.24	3.79
J1 K1	6840	101.4	.0208	3.16	2.30	.73	3.03
J1 K2	6930	101.4	.0208	3.16	2.30	.73	3.03
J2 L1	12720	103.6	.0208	4.14	3.08	1.57	4.65
J2 L2	14860	103.6	.0208	4.14	3.08	1.57	4.65
F1 K1	11860	100.2	.0208	3.76	2.70	1.27	3.97
F1 L1	11630	104.2	.0208	3.82	2.86	1.24	4.10
F2 K2	11950	100.2	.0208	3.76	2.70	1.27	3.97
F2 L2	11630	104.2	.0208	3.82	2.86	1.24	4.10
							45.96

* Estimated from weight of typical end fittings

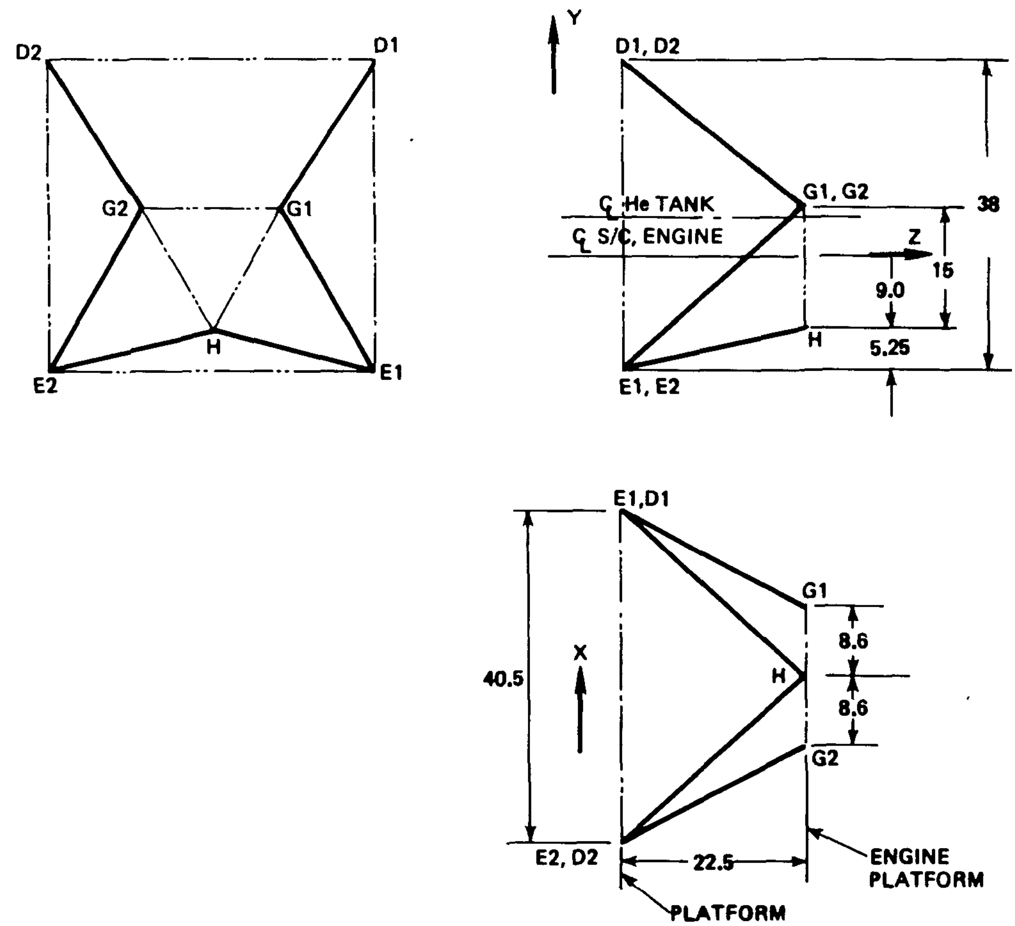


Figure A-3. Engine Support Truss

Table A-3. Engine Support Truss

Ultimate Loads

MEMBER	CONDITION										
	1	2	3a	3b	3c	3d	4a	4b	4c	4d	THRUST
	$n_z = -7.5$	$n_z = 10$	$n_x = 3.75$		$n_x = -3.75$		$n_y = 3.75$		$n_y = -3.75$		ASSUME 2000 LBS ULTIMATE
			$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	
D1 G1	-167	223	317	121	-93	-289	-145	-341	369	173	-517
E1 G1	-105	140	-315	-438	455	332	241	118	-101	-224	-324
E1 H	-177	236	-200	-407	436	229	212	5	24	-183	-547
D2 G2	-167	223	-93	-289	317	121	-145	-341	369	173	-517
E2 G2	-105	140	455	332	-315	-438	241	118	-101	-224	-324
E2 H	-177	236	436	229	-200	-407	212	5	24	-183	-547

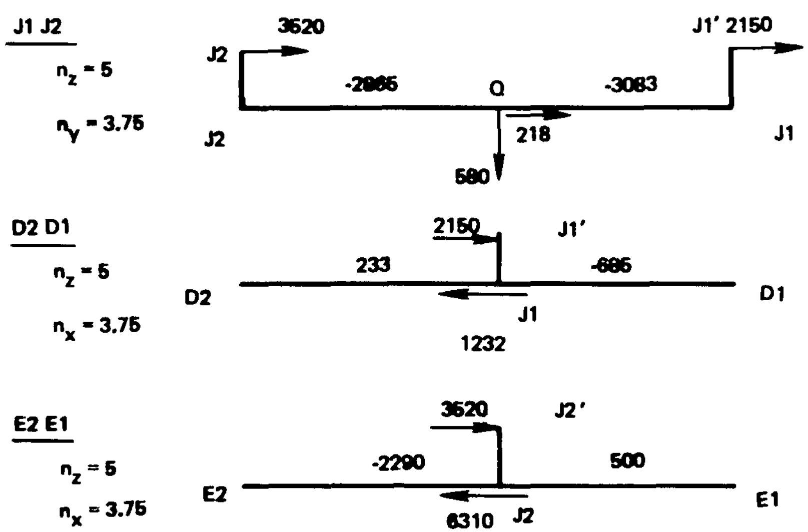
Member Sizes and Weights

MEMBER	MAXIMUM COMPRESSION	L	t	DIAMETER	WEIGHT (LBS)		
					TUBE	END FITTINGS	TOTAL
D1 G1	517	30.9	.0104	.76	.12	.14	.26
E1 G1	438	32.4	.0104	.74	.12	.14	.26
E1 H	547	30.7	.0104	.77	.12	.14	.26
D2 G2	517	30.9	.0104	.76	.12	.14	.26
E2 G2	438	32.4	.0104	.74	.12	.14	.26
E2 H	547	30.7	.0104	.77	.12	.14	.26
							1.56

Table A-4. Platform Members

Ultimate Loads

MEMBER	CONDITION									
	1	2	3a	3b	3c	3d	4a	4b	4c	4d
	$n_z = -7.5$	$n_z = 10$	$n_x = 3.75$		$n_x = -3.75$		$n_y = 3.75$		$n_y = -3.75$	
			$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$
D1 F1	810	-1080	535	1480	-1615	-670	3870	4820	-4950	-4010
F1 E1	-1350	1805	2430	850	-623	-2200	4570	3000	-2770	-4350
D2 F2	495	-660	248	826	-908	-330	-2020	-1440	1360	1940
F2 E2	1100	-1460	-1386	-106	-74	1206	-3180	-1900	1720	3000
F1 Q	0	0	-218	-218	218	218	0	0	0	0
J1 F1	-980	1310	773	-372	537	-608	2860	1710	-1550	-2690
J2 F1	1010	-1350	35	1215	-1385	-205	-2880	-1700	1530	2710
J2 F2	1010	-1350	-641	539	-709	471	-2880	-1700	1530	2710
J1 F2	-980	1310	95	-1050	1215	70	2860	1710	-1550	-2690



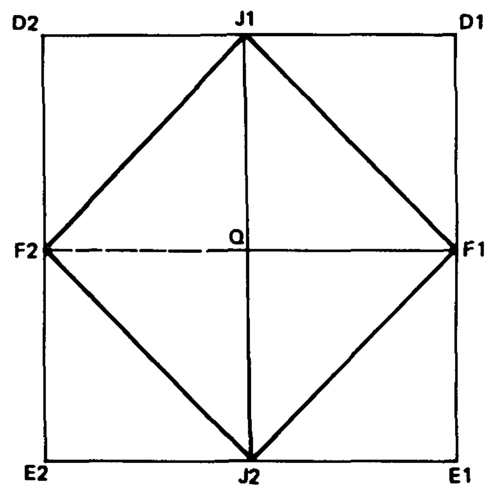


Table A-5. Platform - Member Sizes and Weights

Tubular Members (Diagonals)

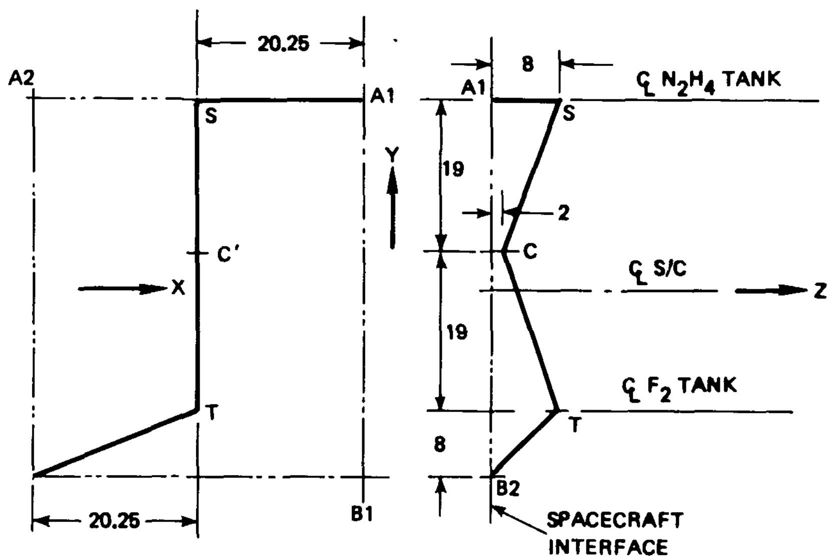
MEMBER	MAXIMUM COMPRESSION	L	t	DIAMETER	WEIGHT (LBS)		
					TUBE	END FITTINGS	TOTAL
J1 F1	2690	27.8	.0104	1.25	.17	.17	.34
J2 F1	2880	27.8	.0104	1.25	.17	.17	.34
J2 F2	2880	27.8	.0104	1.25	.17	.17	.34
J1 F2	2690	27.8	.0104	1.25	.17	.17	.34
							1.36

Edge and Cross Members

MEMBER	MAXIMUM MOMENT	MAXIMUM COMPRESSION	TYPE	DEPTH	WIDTH	t	MATERIAL	WEIGHT **
E1 E2	10560	2290	⊏	4	1	.024	Ti	1.03
D1 D2	6450	685	⊏	4	1	.020	Ti	.86
J1 Q	12900	3083	I	4	2	.09	GRP	1.02
J2 Q	21100	2865	I	4	2	.053	Al	.92
D1 F1*	0	4950	⊏	4	1	.10	GRP	.74
F1 E1*	0	4350	⊏	4	1	.10	GRP	.74
F2 E2*	0	3180	⊏	4	1	.09	GRP	.67
D2 F2*	0	2020	⊏	4	1	.08	GRP	.67
F1 Q *	0	218	⊏	4	1	.04	GRP	.31
F2 Q *	-	-	⊏	4	1	.04	GRP	.31
								7.27

* Channel section provides stability for edge and corner fittings.
** Does not include local fittings.

Table A-6. Tank Upper Support Members



Load, Size, Weight

MEMBER	MAXIMUM COMPRESSION	L	t	DIAMETER	WEIGHT (LBS) TUBE	END FITTINGS	TOTAL
B2 T	4160	23.2	.0104	1.26	.14	.30	.44
T C	3820	19.9	.0104	1.10	.10	.28	.38
A1 S	2380	21.8	.0104	1.00	.10	.18	.28
S C	2320	19.9	.0104	.94	.09	.18	.27
							1.37

APPENDIX B

SPACECRAFT APPLICATION OF HEAT PIPES

Section 5.3.3.1 of this report examines various techniques for thermal coupling of the RTG and fuel tank so that excess RTG energy can be used to prevent fuel tank freezing. It is concluded from this examination that heat pipes, potentially, represent a most effective solution to this thermal transport problem. Heat pipes are simple devices capable of transporting large quantities of heat long distances with small temperature differences. Heat pipes can be very light and reliable.

On the other hand, there are a number of problems associated with the space application of heat pipe systems. With careful consideration of each of these potential problems, a heat pipe can enhance thermal control subsystem performance. Heat pipes are not miraculous heat transfer devices with an answer to all spacecraft thermal problems. Rather, they are important thermal design tools which can be extremely effective under certain conditions.

This Appendix discusses the more important aspects of the use of heat pipes, and is directed to the particular application described in Section 5.3.3.1.

1. INTRODUCTION

Heat pipe technology has come a long way from the "cut-and-try" approach characteristic of five years ago. TRW Systems, for example, has developed a number of computer codes for designing heat pipe systems and for predicting their performance for a wide variety of operating conditions. A partial list of these programs include:

- Systems Heat Pipe Analysis Re-iterative Program (SHARP). Design program to evaluate wide variety of wick configurations and fluid combinations. Contains a catalog of empirical wick performance data.
- Estimated Heat Pipe Systems Performance (ESP). Program for detailed evaluation of a specific heat pipe system design.
- Vapor-Gas Front Analysis Program (VAP). Program for calculation of temperature and mass distributions in gas loaded heat pipe.

A substantial amount of test data has been collected to verify the accuracy of these programs.

In the case of conventional heat pipe systems using, for example, selected layers of screen wire against the wall of a tube, the hydrodynamic and thermodynamic principles are well understood. The above programs have consistently demonstrated conservatism when comparing predictions with test data. TRW Systems has for several years conducted a program for the experimental determination of wick properties (porosity, permeability) to provide necessary empirical data in support of the analytical effort, and has amassed a sizable catalog of information. A preliminary evaluation of the particular application under consideration indicates that a conventional wick design will easily meet the requirements.

Analytical predictions of the hydrodynamic and thermodynamic performance of more complex arterial wick designs is also well advanced. Potentially, these designs offer considerably higher capacity and pumping length for the same conditions. However, extensive testing of arterial designs at TRW Systems has uncovered a number of failure modes not totally understood or analytically predictable. Consequently, more development testing is required to achieve a reliable design in a particular application than would be necessary with a conventional design.

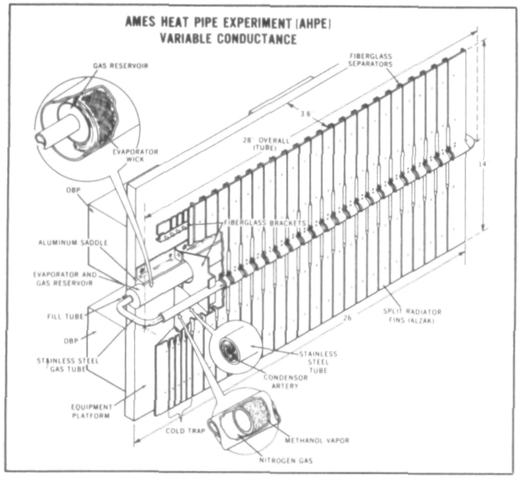
Gas-loaded heat pipes capable of nearly constant temperature operation for wide variations in heat transport and environment are also under analytical and experimental study by TRW Systems. Figure B-1 shows a particular system scheduled for flight on the AOA-C spacecraft in 1972. While not necessary for this specific application, the so-called "constant temperature system" appears to be a major area of heat pipe application. Simple (conventional) heat pipe designs thus appear to be satisfactory for this application. Good analytical understanding, design capability, and experimental experience exist for these configurations.

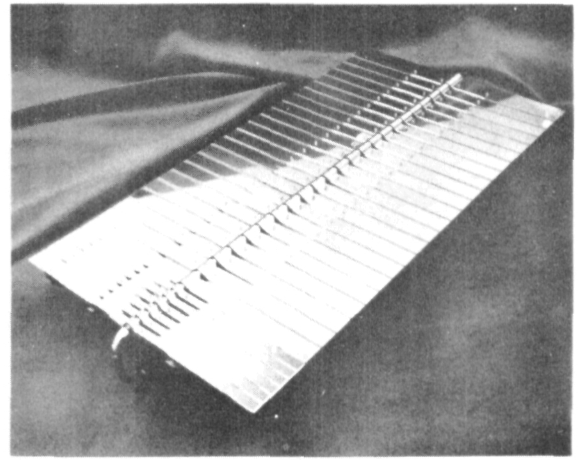
2. FLUID AND MATERIAL SELECTION

Figure B-2 presents a plot of the liquid transport factor as a function of temperature for a number of fluids of potential interest. This factor is a measure of the hydrodynamic capability of each particular fluid; the higher the factor the higher the pumping capacity of the fluid. Water is clearly the "best" fluid above its freezing point ($32^{\circ}F$); ammonia is better at lower temperatures. All other potential fluids are a poor third. Sodium is attractive at temperatures above $250^{\circ}F$ although use of a liquid metal presents obvious containment problems.

Water, although attractive hydrodynamically, is not compatible with an aluminum heat pipe/wick system. A number of proven material choices do exist for water, among them stainless steel, copper, and titanium. Ammonia, on the other hand, is compatible with aluminum and stainless steel; considerable life testing data is available to justify this combination. Ammonia does have relatively high vapor pressures (200 psia at $100^{\circ}F$), and the container must be structurally designed to withstand these pressures. It is clear that a detailed design tradeoff study would be required to select the best fluid for this application; different fluids in each "leg" of the thermal path may be appropriate.

No conventional fluid is operative at temperatures beyond $500^{\circ}F$. Since a liquid metal "leg" is probably not acceptable, it is necessary to introduce a ΔT (thermal resistance) between the RTG and the first pipe. The first "leg" would then be operating around $250^{\circ}F$ (the saturation pressure for water at this temperature is 400 psi). This pipe would have to be structurally designed to handle much higher pressures in case the thermal resistance between the RTG and the first pipe failed and the pipe temperature approaches the RTG temperature.





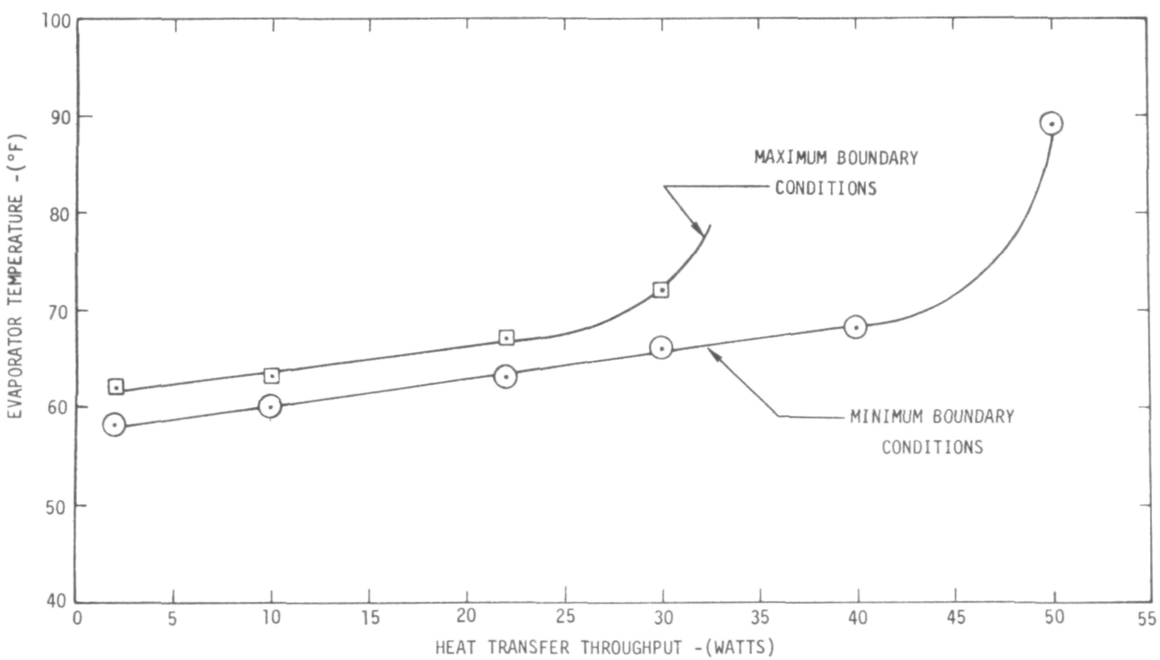


Figure B-1. NASA Ames Heat Pipe Experiment

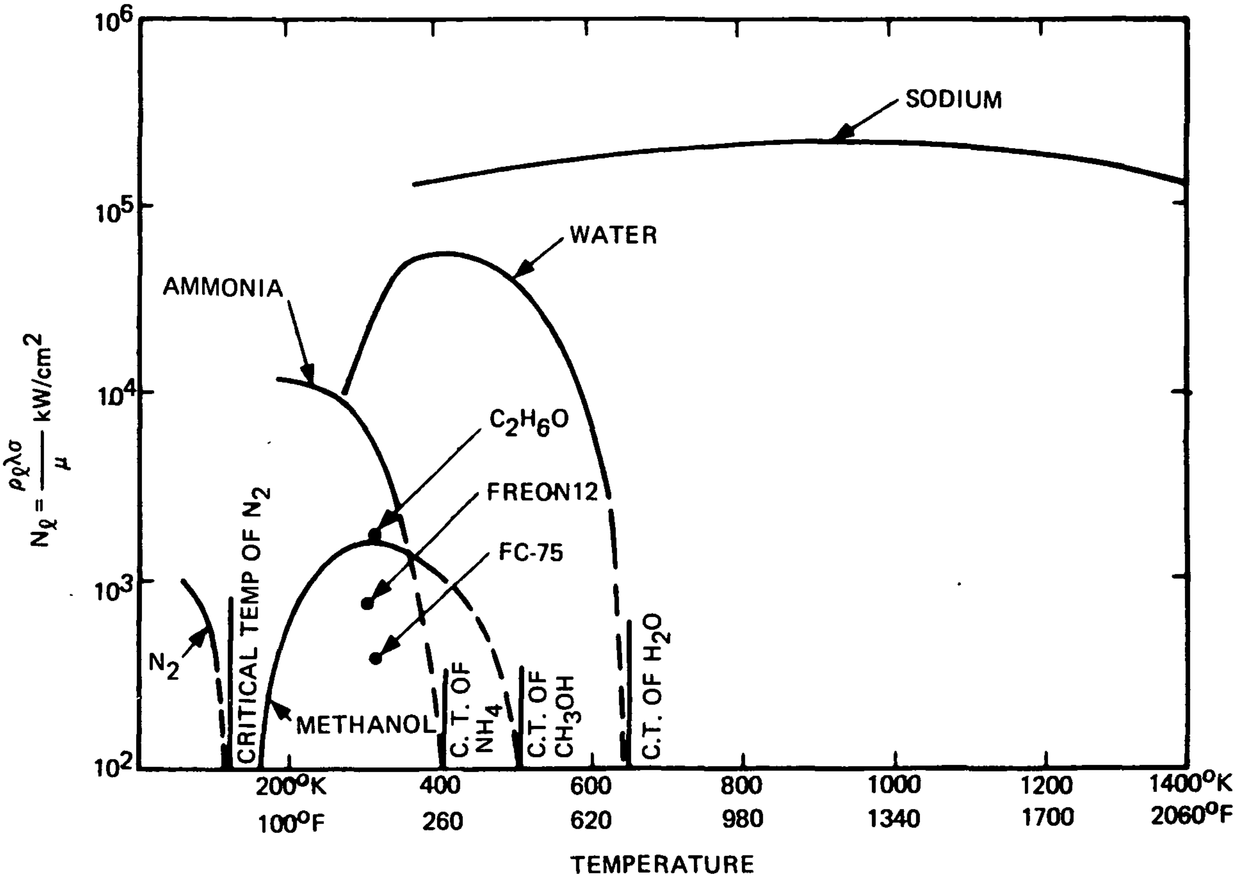


Figure B-2. Liquid Transport Factors for Numerous Heat Pipe Working Fluids

This brings up one area which requires careful design. During groundhold (and during the powered flight modes), the heat pipes would not be capable of pumping the condensed fluids against adverse gravitational gradients. Therefore, if the pipes were oriented such that the condensed liquid falls against the cold ends of the pipes, the pipes will fail to operate, and the temperature of the pipe near the RTG will approach the temperature of the RTG. On the other hand, by placing the RTG below the hydrazine tank, the condensed liquid will collect in the hot ends of the pipes. Under this condition, the heat pipes will actually operate much more efficiently than when in a zero-gravity field, and heat pipe temperatures would remain low.

At the lower temperature of 40°F, water is still an acceptable choice, although this level is close to its freezing point. However, it should be acceptable for the last "leg" since the hydrazine must never drop below 40°F.

TRW Systems has, for the last four years, conducted a considerable amount of life testing with a variety of fluid/container combinations, both in real time and under accelerated conditions. Numerous combinations demonstrating no measurable gas generation over extended mission periods (>10 years) have been tested.

3. GROUND TESTING

The capillary pressure head in conventional heat pipe systems is quite small (about 1 lb/sq ft), and as stated above, in a 1 "g" environment, these systems are not capable of pumping fluids to significant adverse heights. Meaningful ground testing, therefore, requires orientation in a level mode. When this is not possible because of spacecraft geometry, extensive element testing prior to installation on the spacecraft is required.

4. HEAT PIPE SPACECRAFT EXPERIENCE

Heat pipe technology has progressed to a point of practical application to the thermal problems of spacecraft. This is amply demonstrated by a number of flight experiments (GEOS-B, ATS-E, OAO, OSO-PAK) and by upcoming plans for numerous other flight systems (OAO-C, Apollo/Lunar Surface Experiment, ATS F&G, etc.) TRW Systems has, over the last seven years, been concerned with space application of heat pipes concerning:

- Development of analysis capability (conventional, artery, and constant temperature heat pipes).
- Acquisition of experimental data for a wide variety of configurations.
- Generation of extensive life data.
- Development of manufacturing/production techniques.
- Investigation of heat pipe/spacecraft integration.

Hardware programs undertaken by TRW, under contract, include:

- Fifty-foot circumferential Heat Pipe System for NASA/MSC.
- OAO-C Variable Conductance, Constant Temperature Heat Pipe Experiment for NASA/Ames (due to fly early 1972).
- Heat Pipe Thermal Control System for 1/2 kw Long-Life, High Reliability Battery, Wright-Patterson AFB.
- Heat Pipe for Lunar Surface Magnetometer for NASA/Ames.

Although not of direct interest here, there has also been a considerable amount of work done on liquid metal heat pipe systems.

SUMMARY

Heat pipe systems have achieved status and should receive careful attention as a powerful thermal control tool. They do not solve all problems; however, properly applied, heat pipes can provide substantial advantages over other thermal control processes and techniques. Simple heat pipe units with high reliability appear to meet all the requirements of the particular application identified in this report, provided the RTG is below the point at which the heat pipe attaches to the hydrazine tank.

APPENDIX C
AUXILIARY COOLING OF FLUORINE TANK

During the initial stage of the investigation of the fluorine flight thermal control, it became abvious that severe consequences would result should the heat transfer rate to the fluorine be appreciably larger than expected. Four methods were considered for increasing the heat rejection rate from the fluorine tank in the event that inadvertent heat leaks from the sun, the RTG, the hydrazine tank, the helium tank, the rocket engine, or the electronics package should cause the fluorine temperature to rise above the maximum desired temperature ($200^{o}R$). These methods are discussed separately in the following subsections to provide a basis of comparison. It will be observed that the restrictions against active thermal control were not observed in this study.

1. SELECTIVE INSULATION REMOVAL

During groundhold and the first few hours of flight, the entire surface of the fluorine tank must remain thermally insulated to prevent frost and planetary heating, respectively. Once the vehicle leaves the vicinity of Earth, however, heat removal from the fluorine tank could be increased by removing insulation from those areas of the tank surface that are shaded from the sun, but have a substantial view of space. This could include perhaps as much as one half the total surface area of the tank or about 22 sq ft. The 8-node analytical model, Section 5. 2, was used to estimate how much benefit could be derived by removing the outboard half (Node 6) of the foam insulation. Table C-1 shows the resulting heat rejection rate increase due to total removal of three different initial thicknesses of insulation.

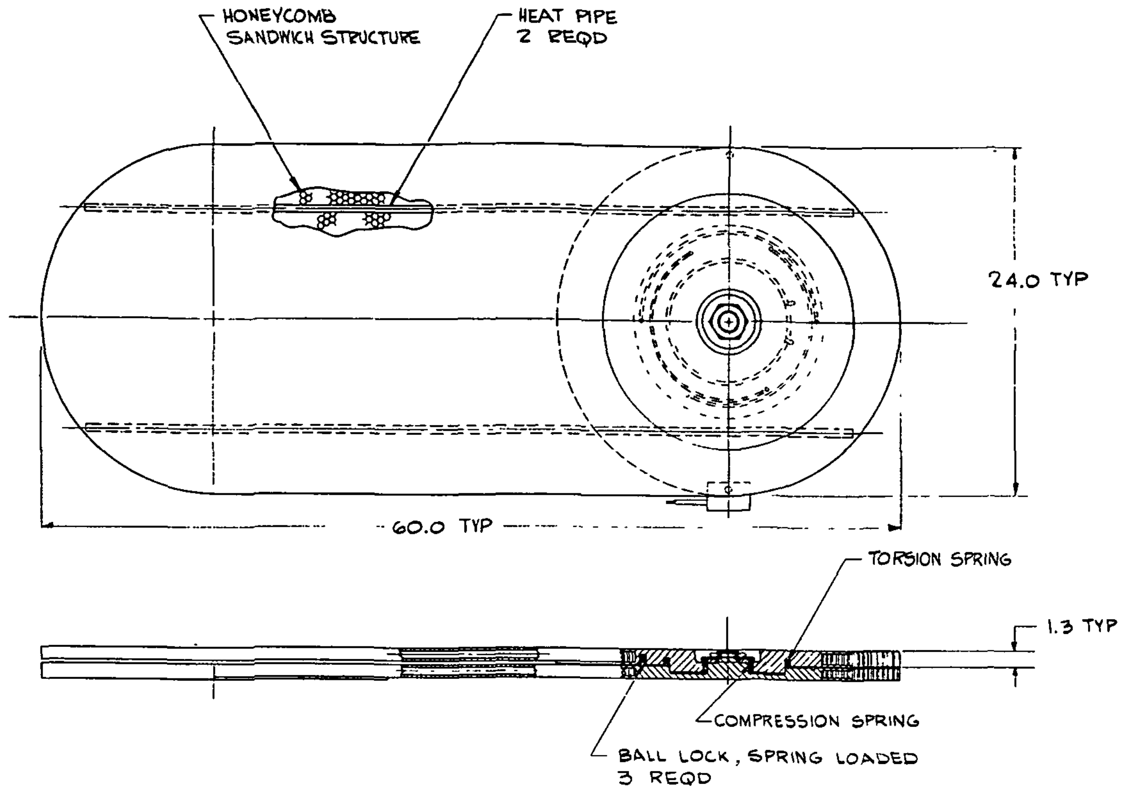
It is quite clear from Table C-1 that removing insulation from the tank does not buy a great amount of heat rejection capability at the low temperatures required. In addition, removing the insulation makes avoidance of solar impingement all the more critical.

Due to practical considerations, insulation removal would probably have to be an irreversible process. Thus, it would not be initiated unless the fluorine temperature were approaching its upper limit. Even then, removal might best be done progressively (perhaps one-third the area at a time) so that the terminal temperature at time of engine firing would not be too cold.

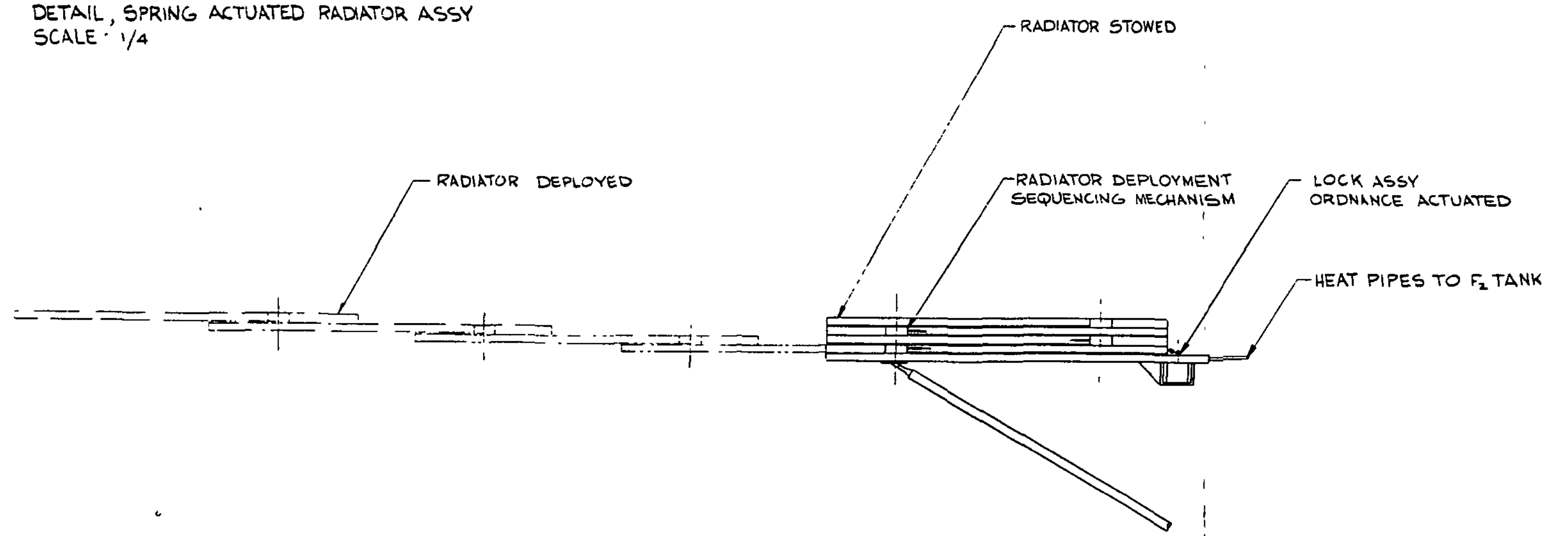
Admittedly, there are some practical problems involved in designing removable foam insulation. There is little doubt, however, that it could be done with negator springs using pyrotechnic or electromechanical release. The entire spring and release assembly would have to be beneath the insulation to avoid local heat leakage.

2. DEPLOYABLE RADIATOR

A deployable radiator could be used as either an alternative or supplement to the removable insulation approach. Drawing SK 407046 is a conceptual drawing of one type of deployable radiator that has been considered. This particular design consists of several overlapping radiator panels that are spring-loaded to unfold by side



DETAIL, SPRING ACTUATED RADIATOR ASSY
SCALE · 1/4



SPRING ACTUATED RADIATOR ASSY.
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1 IDENTIFICATION MARKING PER PR 12-1
TYPE____ CLASS____ PART NUMBER____

NOTES UNLESS OTHERWISE SPECIFIED

PARTS LIST

CODE IDENT NO.	PART OR IDENTIFYING NO.	NOMENCLATURE OR DESCRIPTION	MATERIAL	SPEC	REF DES	ZONE	ITEM NO.

QTY REQD PER ASSY CONFIGURATION

FINISH
HEAT TREAT
PR 12-1
APPLICABLE SPECIFICATIONS
THE ABOVE TRW SYSTEMS GROUP SPECS FORM A PART OF THIS DRAWING

USED ON | NEXT ASSY | NEXT ASSY QTY REQD
APPLICATION

UNLESS OTHERWISE SPECIFIED
1. INTERPRET PER MIL-STD-100
2. DIMENSIONS ARE IN INCHES
3. SURFACE TEXTURE SHALL BE ✓
4. DIMENSIONS APPLY BEFORE PLATING OR CONVERSION COATING
5. REMOVE BURRS & SHARP EDGES

TOLERANCES ON DECIMAL DIMENSIONS: .XXX ± .030, .XX ± .03, .X ± .1
TOLERANCES ON ANGULAR DIMENSIONS: MACHINED & LOCATING ± 0°30', FORMED ± 2°, CHAMFERS ± 5°

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ONE SPACE PARK REDONDO BEACH, CALIFORNIA

DEPLOYABLE RADIATOR CONCEPT, F_2 TANK - SSPM

SIZE E | CODE IDENT NO. 11982 | SK 407046
SCALE NOTED | SHEET

1173-15

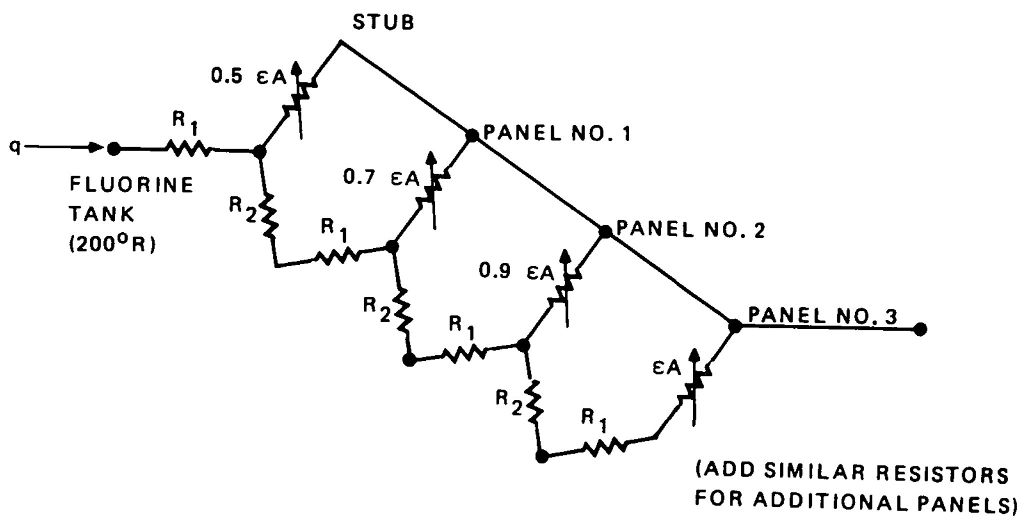
Table C-1. Incremental Heat Rejection Capability Gained by Totally Removing Foam Insulation From the Outboard Half of the Fluorine Tank Surface*

Insulation Thickness (Inches) / Fluorine Tank Temp. (T) (°R)	0.75 Δq (Btu/hr	1.5 Δq (Btu/hr	3.0 Δq (Btu/hr
200	9.83	15.03	20 63
100	0.09	0.17	0.30

*Bare tank surface is assumed to have the same emittance as the foam insulation (ϵ_H = 0.8).

rotation (like a carpenter's folding rule) to form a long rectangular radiator. Each joint is designed to provide easy rotation during deployment, but to lock solidly for low thermal resistance after deployment. Because of the need for low thermal resistance and weight, cryogenic heat pipes are embedded within honeycomb panels to form the individual radiator sections.

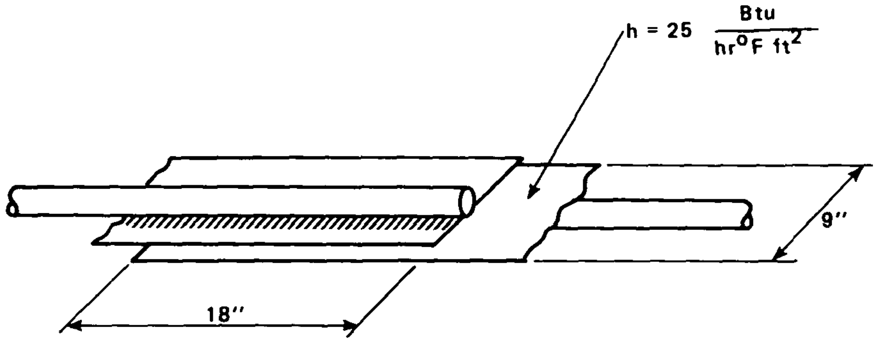
The thermal heat paths for a deployable radiator of this type can be drawn schematically as follows.



where:

R_1	=	heat pipe resistance per section
R_2	=	joint resistance per section
q	=	rate of heat removal from fluorine tank
ϵ_H	=	hemispherical emittance of the radiator (0.9)
A	=	radiator area (single side) per section (6 ft^2)

Preliminary calculations indicate that 1/2-in. diameter heat pipe with a 0.010-in. thick saturated wick, an 18-in long evaporation section, and an 18-in long condensing section would impose a thermal resistance (R_1) of about 0.03°F/hr/Btu. Two such pipes each attached to separate 25 mil face sheets and overlapping each other co-linearly by 18 in. as shown below would produce a joint resistance (R_2) of approximately 0.25°F/hr/Btu.



Thus, with only one heat pipe per radiator section, the total resistance ($R_1 = R_2$) per section would be 0.28°F/hr/Btu. With two parallel heat pipes per section, total resistance ($R_1 + R_2$) would drop to 0.14°F/hr/Btu.

An important problem with the deployable radiator is that the radiating surface(s) must be protected (by shades or insulation) from solar irradiance. Ideally, the deployable radiator should be positioned in the spacecraft shadow and edgewise to the sun and should be allowed to radiate from both sides as shown schematically in Figure C-1. In this position, however, regular off-pointing angle variations and random ±5 degree pointing angle uncertainty (parallel to the plane of the paper) would result in exposing the radiator surface to shallow angle solar irradiance at least part of the time. Even if the radiating surface(s) were covered with a low α_s/ϵ_H coating, such as second surface silvered Teflon, the resulting absorbed solar flux would be unacceptably large (46.2 Btu/hr per panel for only 5 degrees misalignment). To avoid this fate, overhanging edge shades could be added to each panel, but this introduces serious mechanical problems during deployment and reduces the panel view factor to space. A better alternative might therefore be to pitch the deployable radiator 5 to 10° so that the sun never impinges on one side and insulate the sunward side. Figure C-2 shows the heat rejection rate that can be achieved with typical resistances for a one-sided deployable radiator of this type as a function of the number of radiator panels. Each panel would weigh approximately 6.5 lb based on 25 mil aluminum face sheets on both sides. It can be seen that for the particular design considered, the point of diminishing returns is reached at approximately the fourth to sixth panel.

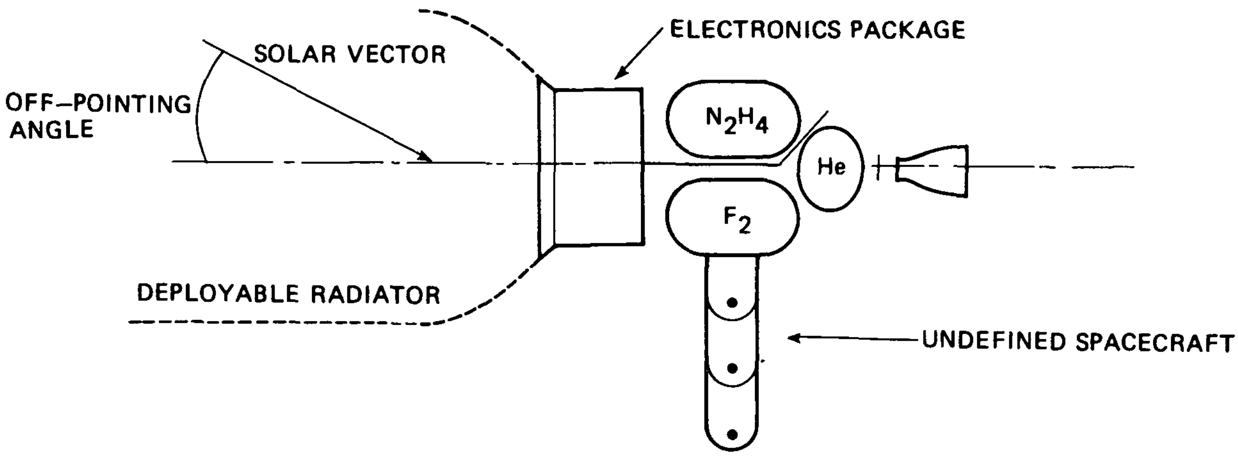


Figure C-1. Expected Angle of Solar Incidence

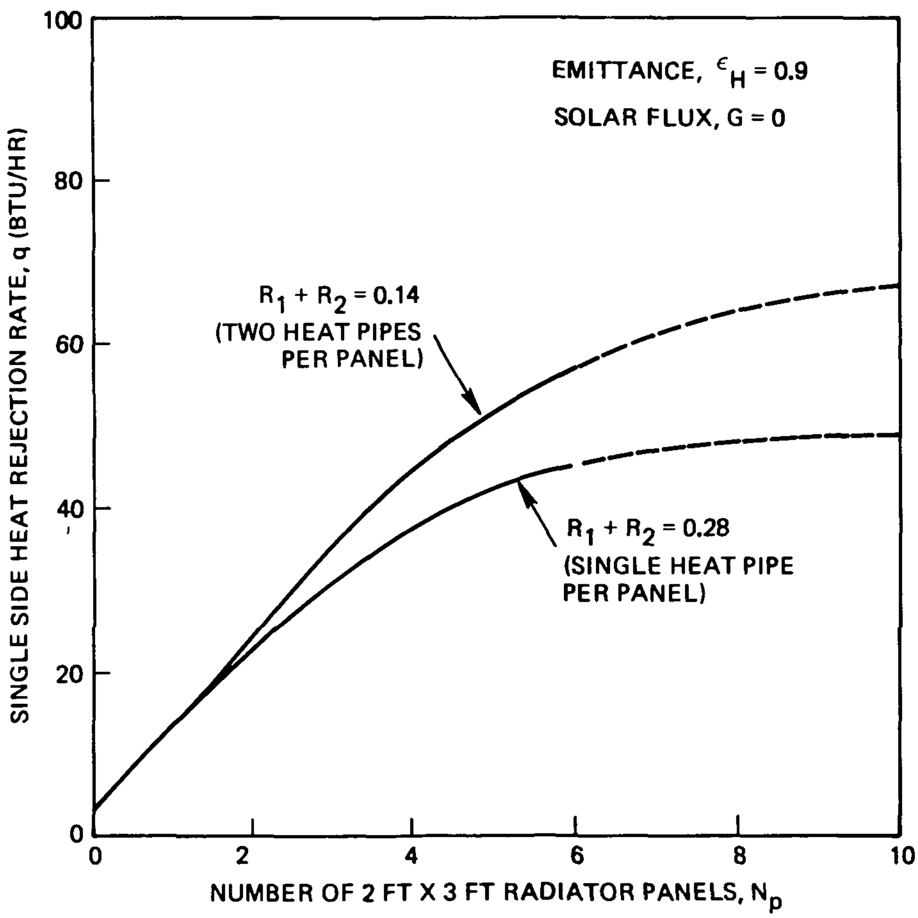


Figure C-2. Heat Rejection Rate for a Deployable Radiator

The main point to be made here is that at least one method of deploying a space radiator appears to be practical and to have some merit as a means of cooling the fluorine tank. Other deployment schemes such as a flexible roll-out or a laterally hinged fold-out or a telescopic slide-out radiator might prove to be superior to the model discussed here.

3. HEAT PUMPS

Four types of heat pumps have been considered for possible use in moving heat from the fluorine tank to a warmer radiator from which it could then be more easily rejected to space. With the possible exception of the Vuilleumier cycle, none appears suitable. A brief discussion of the four types and the reasons they are unsuitable follows.

Vapor Compression Cycle. Vapor compression cycles require mechanical work to turn the compressor and such work is not available in the present application. In addition, it is doubtful whether the required low temperatures could be achieved by vapor compression even with a cascade system.

Absorption System. Absorption systems use heat as the driving force rather than mechanical work. A refrigerant is alternately absorbed and then liberated by the absorbant. The RTG as presently designed operates at 960^{o}R and radiates approximately 10,000 watts of heat to space. This heat in principle could be used to drive an absorption type refrigerator or heat pump. Unfortunately, all of the presently known absorption systems require gravity for operation, and most of them use either water-ammonia or lithium bromide-water as the absorbant-refrigerant combination. Thus for the present application, a wicking system would have to be developed so as to replace hydrostatic pressure due to gravity with capillary pressure. An absorbant-refrigerant combination would have to be found that would allow the cycle to work at the desired low temperature.

Solid State Cooling. Thermoelectric elements are at present limited by practical considerations to temperatures above 230^{o}R. In addition, they require a prohibitive amount of electrical power (200 watts to achieve less than 1 watt of refrigeration at 234^{o}R.)*

Vuilleumier Cycle. The Vuilleumier Cycle is the most promising of the heat pump methods investigated. It is a heat driven refrigeration cycle that is independent of gravity. An experimental model has delivered 5 watts of refrigeration at 135^{o}R.**

---

*B. Shelpuk, M.S. Crouthanel, A. Smith, and M. Yim "Low Temperature Solid-State Cooling Technology," Technical Report AFFDL-TR-68-128, Wright Patterson Air Force Base, Ohio, 1968.

**F.N. Magee and R.D. Doering, "Vuilleumier Cycle Cryogenic Refrigerator Development," Technical Report AFFDL-TR-68-67, Wright Patterson Air Force Base, Ohio, 1968.

That particular model weighed only 18 pounds and required approximately 480 watts of heat from a $1460^{\circ}R$ source. The refrigerator consists of two different sized displacers (pistons) operating at 90 degrees to each other on a common crankshaft pin as shown in Figure C-3. While pressure differentials are small and rotational speed is low, the fact that moving parts are involved would probably make this system unsuitable for full time use because of the long life requirement. It could perhaps be used intermittently (say, once every six months) to compensate for unexpected heat leakages into the fluorine tanks.

AiResearch Manufacturing Company (Torrance, California) is presently developing a Vuilleumier engine which does not have any moving parts. However, the efficiency of that engine is considerably reduced, and the development work is approximately three years from completion.

4. EXPENDABLE FRIGERANT

This method involves storing another cryogenic fluid for venting through a heat exchanger within the fluorine tank. Required properties for the frigerant are:

1) Non-corrosive to spacecraft materials so that venting can be tolerated
2) High heat of vaporization
3) High weight density
4) Boiling temperature near the maximum fluorine storage temperature.

Methane (CH_4) appears to be a relatively attractive candidate. It boils (atmospheric pressure) at $201.4^{\circ}R$, absorbs 219.2 Btu/lb as it boils, and weighs approximately 26.46 lb/ft^3 (as a liquid). It can easily be seen that a large weight penalty must be accepted in order to achieve any significant amount of cooling by this method. For example, to simply match the 68,830 Btu that can be absorbed by the fluorine and its tank would require:

$$W_{methane} = \frac{68,830}{219.2} = 314 \text{ lb}$$

Some additional cooling could be achieved by subcooling the methane to the fluorine freezing point ($97^{\circ}R$) before launch. This would place the methane approximately $66.5^{\circ}F$ below its own freezing point so that the heat of fusion (25.2 Btu/lb.) could be

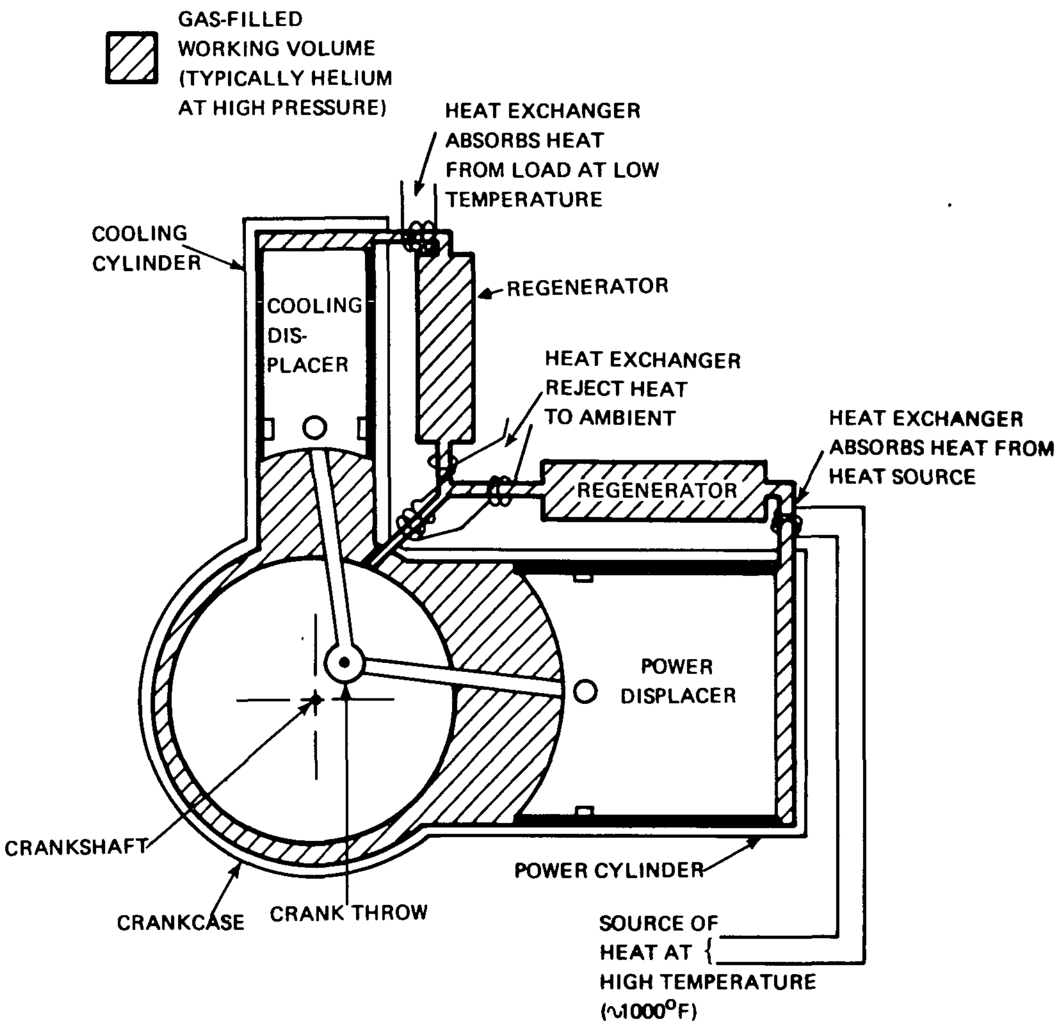


Figure C-3. Schematic of Basic Vuilleumier Cryogenic Refrigerator utilized along with the normal heat capacity associated with temperature rise first as a solid and then as a liquid. The 314 pounds of methane could absorb approximately another 12,000 Btu in this way. Dividing the total heat that can be absorbed by 314 pounds of methane by the total storage time (42,600 hours) gives the average heat leakage rate that could be accommodated by the expendable frigerant.

$$Q = \frac{68,830 + 12,000}{42,600} = 1.9 \text{ Btu/hr}$$

This appears to be totally unattractive as a means of rejecting heat to space. In addition, the reliability of circulating methane through fluorine is highly questionable.

APPENDIX D

COMPUTER PROGRAMS FOR THERMAL ANALYSES

Three computer programs were formulated during Task VII. Two were for the module as shown in drawing SK 406922 given in this appendix. The only difference in these two programs was that one included conductors to account for atmospheric convection and groundhold coolant flow during the groundhold phase.

For the most part, the conductances and radiation view factors were obtained from computer programs formulated during other tasks of this project. In some cases the view factors were determined by constructing a model of the hardware and measuring the view factor with a form factometer.

All computer programs were in the standard SINDA format.

Tables D-1 and D-2 list the nodes of these two models and the major conductances and radiation conductors for the flight configuration. It should be remembered that in many cases the various conductors were varied during the analysis since they were actually the subject of the investigation. The values listed in Table D-2 are merely the nominal values of such conductors.

Table D-1. Nodal Arrangement of Module Computer Model

Node Number	Description	Area (ft^2)	ε
1	F_2 tank	-	-
2	N_2H_4 tank	-	-
3	Helium tank	-	-
4	Thrust cone	4.56+2.18	0.3, 0.8
5	Combustion chamber	1.3	0.05 - 0.8
6	Propellant valves	0.5	0.1 - 0.7
7	F_2 tank insulation, top (-Z)	7.2	0.8
8	F_2 tank insulation, side (+Y)	7.2	0.8
9	F_2 tank insulation, side (+X)	7.2	0.8
10	F_2 tank insulation, side (-Y)	7.2	0.8
11	F_2 tank insulation, side (-X)	7.2	0.8
12	F_2 tank insulation, bottom (+Z)	7.2	0.8
13	F_2 tank insulation, removable	(part of node 10)	
14	N_2H_4 tank insulation, top (-Z)	7.2	0.8
15	N_2H_4 tank insulation, side (+Y)	5.6	0.8
16	N_2H_4 tank insulation, side (+X)	7.2	0.8
17	N_2H_4 tank insulation, side (-Y)	7.2	0.8
18	N_2H_4 tank insulation, side (-X)	7.2	0.8
19	N_2H_4 tank insulation, bottom (+Z)	7.2	0.8
20	N_2H_4 tank louver	1.6	-
21	Fwd meteroid shield, +Y	2.8	0.8
22	Fwd meteroid shield, -Y	2.8	0.8
23	Aft meteroid shield	9.3	0.8
24	N_2H_4 thermal shield, N_2H_4 side	10.5	0.1 - 0.8
25	Helium thermal shield	3.5	0.1 - 0.8
26	Helium insulation, -Z	14.2	0.8
27	Helium insulation, +Z	7.1	0.8
28	Frame, +Y	4-in x 1-in x 0.24 in	-
29	Frame, -Y	4-in x 1-in x 0.20 in	-
30	Spacecraft insulation, -Y	32.0	0.8
31	Spacecraft insulation, +Y	32.0	0.8
32	RTG	15-in dia x 54-in	0.9
33	Space	-	-
34	LN_2 (or Helium) coolant	-	-
35	Spacecraft	-	-
36	N_2H_4 thermal shield, F_2 side	10.5	0.1 - 0.8
37	F_2 tank cooling coil	1.0	-
38	Helium tank cooling coil	1.0	-

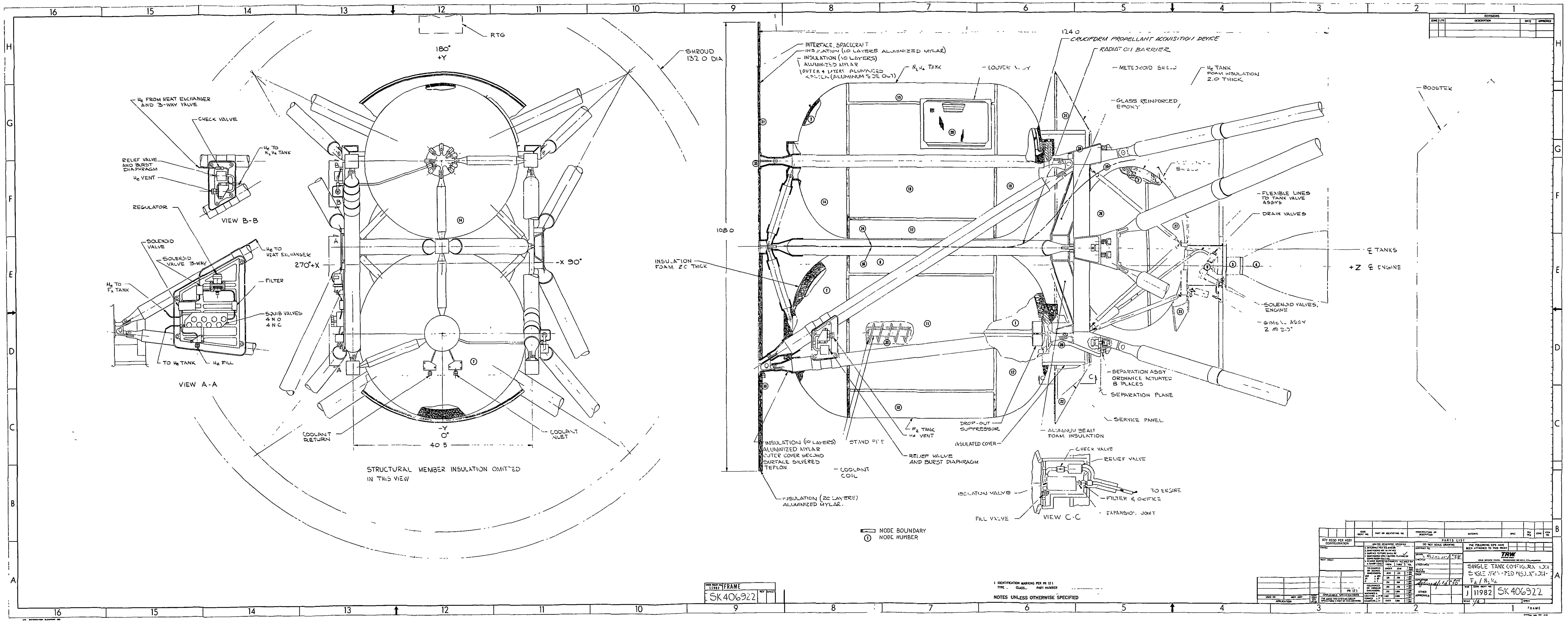
Table D-2. Controlling Conductors and Paths of Module Model During Flight

Conductor Values (Btu/hr-^{o}F)	Node-To-Node
0.003	1-2
0.45	1-7
0.45	1-8
0.45	1-9
0.45	1-10
0.45	1-11
0.45	1-12
0.003	2-3
0.09	2-14
0.07	2-15
0.09	2-16
0.09	2-17
0.09	2-18
0.09	2-19
0.4	2-32
0.06	24-36
0.06	31-35
0.16	30-35

Radiation Values ($\mathcal{F}A\epsilon_1\epsilon_2$)	Node-To-Node
Variable	2-20
0.004	4-32
3.11	4-33
2.0	7-31
1.9	7-33
2.2	8-33
6.0	9-33
7.0	10-33
6.0	11-33
2.6	12-22
2.6	12-23
3.2	13-33
3.2	14-31
0.004	14-32
1.9	14-33
0.07	15-32
4.8	15-33

Table D-2. Controlling Conductors and Paths of Module Model During Flight (Continued)

Radiation Value	Node-To-Node
6.4	16-33
0.54	17-24
2.2	17-33
6.4	18-33
2.6	19-21
0.004	19-32
2.6	19-33
1.5	20-33
1.3	21-33
0.8	25-26
3.5	26-33
1.2	30-33
3.2	31-33
3.0	33-36



RTG
180°
+Y
SHROUD
132.0 DIA
H_e FROM HEAT EXCHANGER AND 3-WAY VALVE
CHECK VALVE
RELIEF VALVE AND BURST DIAPHRAGM
H_e VENT
H_e TO N_2H_4 TANK
VIEW B-B
REGULATOR
SOLENOID VALVE
SOLENOID VALVE 3-WAY
H_e TO HEAT EXCHANGER
FILTER
H_e TO F_2 TANK
SQUIB VALVES 4 N.O. 4 N.C.
TO H_e TANK
H_e FILL
VIEW A-A
270° +X
-X 90°
COOLANT RETURN
COOLANT INLET
-Y
0°
40.5
STRUCTURAL MEMBER INSULATION OMITTED IN THIS VIEW
INTERFACE, SPACECRAFT
INSULATION (10 LAYERS ALUMINIZED MYLAR)
INSULATION (10 LAYERS) ALUMINIZED MYLAR
N_2H_4 TANK
LOUVER ASSY
124.0
CRUCIFORM PROPELLANT ACQUISITION DEVICE
RADIATION BARRIER
METEOROID SHIELD
H_e TANK FOAM INSULATION 2.0 THICK
GLASS REINFORCED EPOXY
BOOSTER
FLEXIBLE LINES TO TANK VALVE ASSYS
DRAIN VALVES
108.0
INSULATION FOAM 2.0 THICK
℄ TANKS
+Z ℄ ENGINE
SOLENOID VALVES, ENGINE
GIMBAL ASSY 2 @ 5.0°
SEPARATION ASSY ORDNANCE ACTUATED 8 PLACES
SEPARATION PLANE
SERVICE PANEL
INSULATION (10 LAYERS) ALUMINIZED MYLAR OUTER COVER SECOND SURFACE SILVERED TEFLON
STAND PIPE
F_2 TANK
H_e VENT
DROP-OUT SUPPRESSOR
ALUMINUM BEAM FOAM INSULATION
INSULATED COVER
RELIEF VALVE AND BURST DIAPHRAGM
COOLANT COIL
INSULATION (20 LAYERS) ALUMINIZED MYLAR
CHECK VALVE
RELIEF VALVE
ISOLATION VALVE
TO ENGINE
FILTER & ORIFICE
EXPANSION JOINT
FILL VALVE
VIEW C-C
NODE BOUNDARY
NODE NUMBER
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