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TRW

SYSTEMS GROUP

ONE SPACE PARK • REDONDO BEACH, CALIFORNIA

SUMMARY REPORT

TASK VII

SPACE STORABLE PROPELLANT MODULE

ENVIRONMENTAL CONTROL TECHNOLOGY

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TASK VII SPACE STORABLE PROPELLANT MODULE ENVIRONMENTAL CONTROL TECHNOLOGY

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SUMMARY

The objective of Task VII was to make sufficient design and analytical studies of the candidate thermal control systems for a F_2/N_2H_4 propulsion module (devised during Task VI) to select one system for the F_2/N_2H_4 module. The results of this design and analysis are given in this report.

It has been established that the general structural configuration adopted for the OF_2/B_2H_6 module (two equal size propellant tanks, one pressurant tank located below the propellant tanks, and a tubular frame) should also be used for the F_2/N_2H_4 module. Two inch, closed cell foam should be used as the insulation on the fluorine tank and 10 layers of aluminized Mylar used as the hydrazine tank insulation.

LN_2 circulated through a coil inside the fluorine tank may be used as the groundhold coolant but it may be necessary to allow the fluorine tank to self pressurize to 25 psia. If it is desirable to substantially subcool the fluorine, helium which has been prechilled by liquid hydrogen may be used as the coolant. A helium cooling system, however, would be extremely 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. Whether a radiator or louver assembly is used to moderate the temperature is dependent upon 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:

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

- A multilayer aluminized Mylar radiation barrier must be placed between the two propellant tanks
- The spacecraft surface which views the fluorine tank must be well insulated with multilayer aluminized Mylar.

The manner used to control the helium tank temperature is entirely dependent upon the temperature at which the helium is to be stored. It is relatively easy to thermally connect the helium tank to either the warm hydrazine tank or the cold fluorine tank. General considerations indicate a more reliable system would be had if the helium were stored at the hydrazine temperature level.

Finally, the analyses revealed that there are thermal design problems relative to maintaining the main propellant valves and catalyst bed at prescribed temperatures. These problems are entirely solvable but they can only be solved in conjunction with the detailed engine design. Their solutions depend upon an accurate knowledge of the engine construction.

1. INTRODUCTION

This is the Task VII Summary Report of the Space Storable Propulsion Module Environmental Control Technology Project accomplished under Contract No. NAS 7-750. Task VII had as its objective, the thermal analysis of the F_2/N_2H_4 propulsion module designs specified during Task VI, Reference 1. The analysis of Task VII was to be in sufficient detail as to establish the single design which appears most promising. Section 2.0 of this report describes in detail the propulsion module design which has been established as the most promising based on information available at this time. The structural design analysis is also presented in Section 2.

The thermal analyses were broken into two separate parts:

- Determination of the thermal characteristics of the several designs for both flight and groundhold.
- Determination of the propulsion system operational characteristics under the thermal environment previously established.

Section 3 describes the computer programs used in the analyses together with the supporting equations where appropriate.

Section 4 lists the results of the analyses and lists advantages and disadvantages of the various designs. Section 5 weighs the advantages and disadvantages, and lists conclusions which may be drawn from the analysis. Section 6 gives recommendations as to the areas in which further work is needed.

2. F_2/N_2H_4 MODULE DESCRIPTION AND DESIGN

2.1 PROPULSION SYSTEM

The F_2/N_2H_4 propulsion system contained within this module is a pressure-fed feed system connected to a bi-mode engine. Figure 2-1 is a schematic diagram of the baseline system. Helium, stored at high pressure, flows from the helium tank into the fuel and oxidizer tank ullages by way of a regulator that reduces the pressure to a constant 300 psia. The liquids are stored in the two tanks until released into the feedlines by isolation valves. Flow into the engine is controlled by two propellant valves.

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.

2.1.1 Operating Sequence and Baseline Data

The proposed operational sequence is to use the monopropellant mode for all midcourse trajectory correction maneuvers, and the first and last two seconds of the orbit insertion and orbit inclination change maneuvers. From two seconds after start until two seconds before shutdown during the latter maneuvers, the engine will be operated in the bipropellant mode.

Table 2-1 shows the baseline operating points, performance goals and certain key masses. The propellant masses were calculated using the assumed specific impulse values and an initial spacecraft mass of 4400 lb for the given velocity increments (delta-V's) of:

Midcourse maneuvers—100 m/sec

Orbit insertion maneuver—1460 m/sec

Orbit inclination maneuver—2170 m/sec

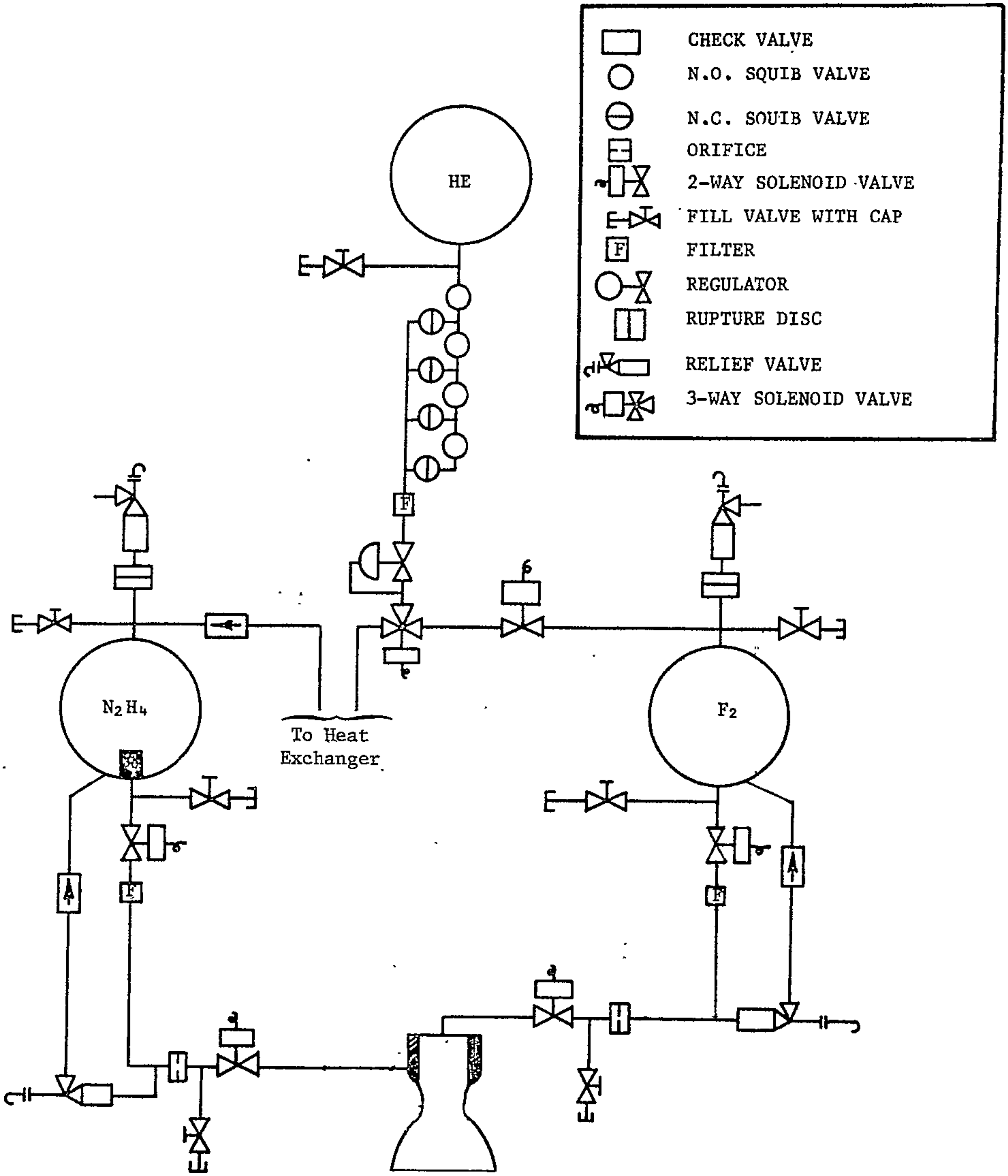


Figure 2-1. Schematic Diagram of F_2/N_2H_4 Spacecraft Propulsion System (Baseline Version with Cold Helium Storage)

Table 2-1. F_2/N_2H_4 Propulsion System Baseline 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 Flowrate	1.7316 lb_m/sec
Fuel Flowrate	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 Flowrate	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 cubic ft.
Regulated Pressure	300 psia*
Oxidizer Mass	1733 lb_m + 75 lb_m residual
Fuel Mass	1058 lb_m + 37 lb_m residual
Propellant Tank Volumes	22.9 cubic feet (each)

*Specified by JPL.

These delta-V's were converted into propellant consumption using the usual relationship:

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

Although the propellant flowrates are implied for the bipropellant mode (i.e., $M_p = F/I_s$), the monopropellant flowrate is an unknown because it depends upon the monopropellant reactor and gas injector design (unspecified). However, if a flowrate of one pound per second is assumed for the monopropellant mode and the delivered average specific impulse during midcourse maneuvers is 240 lb_m-sec/lb_f, the total maximum midcourse burning time is 180.15 seconds. Orbit insertion burn time (total) would be 524.3 seconds and orbit inclination burn time (total) would be 484.6 seconds.

It should be noted that the above calculations assumed a very high efficiency. Should more conservative values of C_F and C* be used, the average I_s would be reduced, the burn times increased, and more hydrazine would be consumed during the midcourse maneuvers. Numerically, the assumed I_s of 240 would drop to 232.5 if C* = 4300 ft/sec and C_F = 1.72.

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 inch. The major external diameter is 32.5 inches and the external height is 20.4 inches. A groundhold cooling coil constructed of 1/2 inch aluminum tubing, 8 feet long is located inside the tank.

Helium is carried from the tank to the valving package via a 1/4-inch 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 the regulator, internally senses outlet pressure and is to be set to 300 psia.

A solenoid operated three-way isolation valve separates the fuel and oxidizer pressurization lines. From this common point, helium flows through a 1/4-inch 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 back-flow of hydrazine vapor which might be heated and decomposed in the exchanger.

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 reseatable relief valve then assumes the valving function. Connection is provided between the ground system and the ullage via 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 percent of the fluorine volume at $180^{\circ}R$. Using identical tanks for both oxidizer and fuel results in a rather large ullage in the fuel tank (24.7 percent); this presents an opportunity to perform the midcourse maneuver firings in a blowdown mode.

Structurally, the tanks are similar to the helium tank, being boron-epoxy ribbon wrapped over a 10-mil thick aluminum liner. Geometrically, they are right cylinders with oblate hemispheroidal ends. The wall thickness is 0.120 inch in the cylindrical portion and 0.050 inch in the ends (including the liner).

Design details of the tank ports and internal equipment have not been determined. It is known, however, that a cooling coil, identical to the coil in the helium tank, must be included inside the fluorine tank and some type of propellant acquisition device will be necessary in the hydrazine tank (but perhaps not in the fluorine tank if two seconds of monopropellant operation is sufficient to settle the fluorine before its withdrawal commences).

A component module (cluster of valves, etc.) is connected into 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 the tank contents from the feedline during all but prefiring 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 connected to each tank outlet port, via a 1/2-inch tube, is a manual fill valve through which fluids are pumped into the propellant tank.

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

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

The drawing included at the end of this section indicates the integration of this hardware.

2.2 STRUCTURAL DESIGN

The basic arrangement of structure and components that was selected as being at or near optimum during Task II has been retained. The reasons for selecting the configuration were discussed in Task II and are still applicable for this 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 modification resulting from this factor was to make the lower platform smaller and to move the truss intersection points closer together since a closer truss arrangement can be used to encompass the tank envelope. The F_2 and N_2H_4 tanks are positioned to place the center of gravity of the propellants along the center line of the spacecraft in order to prevent lateral center of gravity displacement.

Thermal control considerations required the separation 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 attached fittings. The optimum arrangement for the tank is to locate the truss tubes horizontally and attached closely to the tank boss so that loads are transmitted tangentially into the tank shell. This design was used in Task II. However, in this case, the consequence would be to extend the attachment fitting below the spacecraft interface in order to mount the horizontal tubes and this would cause the fitting to be much heavier as well as require more moment carrying structure in the spacecraft. If the tubes were left horizontal but moved up close to the spacecraft in order 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 and this component would then become unduly heavy and a large moment would be imposed on the tank. This 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 but, because of the angle, 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 either of the other alternatives.

Thermal control requirements also dictated a change in the materials used in the tank support platform. Assuming both the F_2 and He tanks are maintained at cryogenic temperatures, the section of the 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 thermal conduction members in the platform between it and the other tank. These members consist of the section of the center beam extending 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 they were for the Task II design. A thermal radiation shield is now located between the F_2 and N_2H_4 tanks and extending down between the N_2H_4 and He tanks. The shield is light weight, being made of aluminized Mylar, but requires a structural edge support to carry environmental loads. This support is not shown on the drawing but an estimated weight is included in the weight summary.

The design conditions and load factor used for structural design are the same as those used in Task II and are restated herein 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 is used to obtain ultimate loads. The spacecraft weight and center of gravity are those of Task II and are 1100 pounds located 13.7 inches below the interface.

The estimated weights for the propulsion module used for design are as follows:

Gas circuit	26.5 lbs
Liquid circuit	16.4
Thrust chamber assembly	60.5
Tankage	184.6
Fluids	2929.0
Miscellaneous	48.0
Meteoroid shield	14.3
Thermal control	23.0
Structure	102.0

The center of gravity of each component was considered in order to distribute its load into the structure.

Utilizing the above information together with the geometric configuration shown on Drawing SK 406961, the maximum load, size, and weight of each structural member was determined. Appendix A presents this summarized information in tabular form together with sketches and the geometry. Fittings have not been designed in detail. However, conservative estimates of the weight have been made and are included in the tables showing a summary of the system weight.

2.3 THERMAL CONTROL SYSTEM

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.

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, the heat pipes which thermally connect the RTG and the N_2H_2 tank, and one is also used as a radiation barrier between the hot and cold tanks. In all cases these blankets are constructed with the Mylar side facing outward to space. In addition, all blankets are composed of 10 layers of Mylar except for that portion of the spacecraft insulation which is on the fluorine tank side of the barrier. That particular section of multilayer insulation 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-pound density foam. Nonmetallic members in contact with the tanks are foam covered 6 to 8 inches from the point of contact. Metallic members are foamed at least 18 inches from the point of contact.

A 4 sq ft louver assembly is used to moderate the N_2H_4 temperature. It is located on the side of the tank in a position such 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 $520^{\circ}R$, 0.72 for fuel temperature above $545^{\circ}F$, and it varies linearly with temperature between $520^{\circ}R$ and $545^{\circ}R$.

All surfaces of the frame, the helium tank and the fluorine tank which may see solar radiation are covered with second surface silvered Teflon. Since this material has a high emissivity and a relatively low solar absorptivity, it will aid in reducing the heating effects of the sun.

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

All of the above mentioned items are called out in drawing SK 406922 included at the end of this section. Not shown on the drawing are the heat pipes needed for conducting heat from the RTG to the N_2H_2 tank. For purposes of this study, it was assumed that the heat pipe system is composed of three 3-ft rigid heat pipes 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 desired configuration as determined by the analysis reported herein. As will be indicated below, other configurations were considered in the analysis.

3. METHOD OF ANALYSIS

As stated in the introduction; the analysis of the module was divided into two parts. First, a thermal analysis of the various designs was made for a variety of mission conditions. These calculations provided module fluid and component temperature characteristics for variations in the module design.

The second step in the overall analysis was to investigate the general operating characteristics of the propulsion system as they are influenced by temperature variations.

3.1 THERMAL ANALYSIS MODELS

The propulsion module temperatures were calculated by representing the physical system by an equivalent electrical network which was solved by the SINDA* computer program on the Univac 1108 computer. For purposes of this analysis, three computer models of the module were formulated: the basic model for analysis of all flight conditions, the revised model for analysis of groundhold condition, and the engine model for analysis of the valves and catalyst bed. Appendix B lists all the nodes together with the controlling conduction paths.

It will be noticed that these models are considerably less detailed than the models used in the analyses of Task IV. However, the decision to use less complicated models was prompted by results from the Task IV. 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 was because the resistors connecting these nodes were either so extremely large as to make the nodes entirely independent of the propellant and propulsion systems or so extremely 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 insure a realistic model. In the present case, this

*TRW Report 11027-6003-R0-00, " Systems Improved Numerical Differencing Analyzer, User's Manual, " by J. D. Gaski, L. C. Fink and T. Ishimoto, dated September 1970.

has been done by dividing the insulation into many nodes and including nodes for the frame to account for heat conduction between the tanks.

One other point should be mentioned. It was decided to formulate a special program to investigate thermal problems in the local region of the engine rather than investigate these problems with the larger programs. This was felt desirable since to obtain the detail necessary in this 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.

In order 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 fluorine cooling coil is given by the equation:

$$h_o = 0.72 \frac{k}{D} \left[\frac{D^3 \rho^2 \beta g \Delta T}{\mu^2} \frac{c\mu}{k} \right]^{0.25}$$

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. There are no problems relative to zero gravity heat transfer.

These are the same assumptions made in the Task IV analysis. For a detailed analysis of why these assumptions can be made, the reader should consult the Task IV summary report, Reference 2.

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:

$$h_i = 0.0243 \frac{k}{D} \left[\frac{VD\rho}{\mu}\right]^{0.8} \left[\frac{c\mu}{k}\right]^{0.4}$$

where V is velocity in consistent units.

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 established to be less than 10 percent for the coolant helium temperature range experienced.

4. RESULTS OF ANALYSIS

4.1 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 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 4-1 shows a definite advantage of increasing the foam thickness to at least two inches. In contrast, it can be seen that increasing the LN_2 coolant flowrate by a factor of five results in only a minor decrease in the fluorine temperature.

From Figure 4-1, it can be seen that a LN_2 coolant flow rate of 215 lbs/hr coupled with 2-inch thick insulation on the tanks will result in a temperature of 149°R. This is only 3° 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 realistically possible to reduce the LN_2 coolant temperature by about 6°R by reducing the pressure on the coolant supply dewar to about 10 psia. For a constant coolant flowrate, 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 necessitate utilizing a pump to force the liquid through the cooling

Note: Coolant Coils: 1/2" Diameter Tubing
8' Long, Formed into 6" diameter coil.

Coolant Temperature: $140^{o}R$

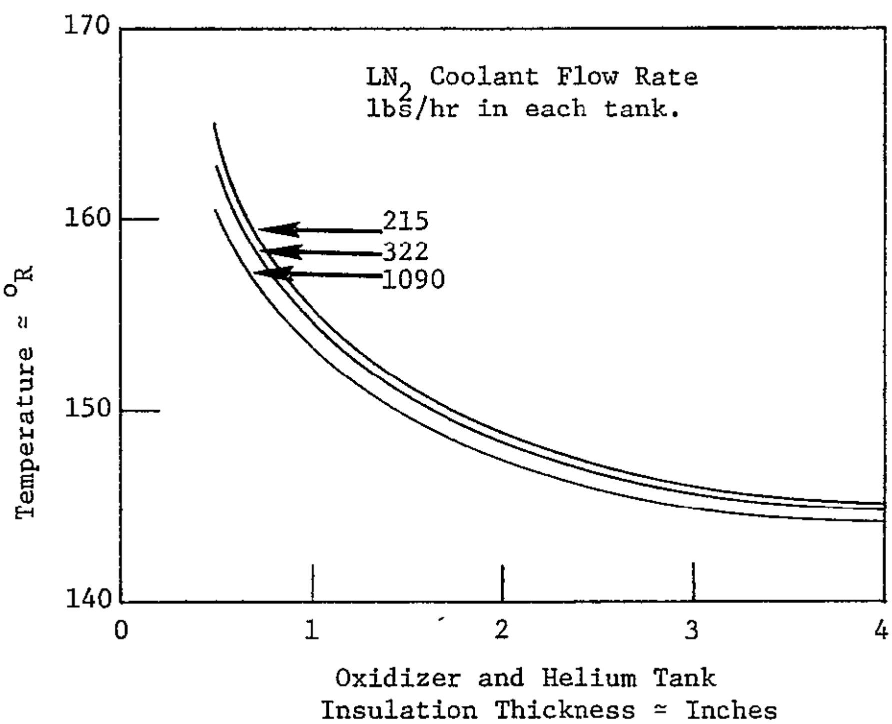


Figure 4-1. Effect of Oxidizer and Helium Tank Insulation on Oxidizer and Helium Tank Temperatures

coils as opposed to utilizing a pressure-feed system.* As a consequence, much of the 6°R temperature drop might disappear due to heat addition by the feed pump and associated plumbing.

As will be indicated later, there are advantages to launching with the fluorine in a substantially sub-cooled condition, 110°R to 130°R. With LN_2 as the ground hold coolant, this is impossible. However, it would be possible to obtain these lower fluorine temperatures by first circulating gaseous helium through liquid hydrogen (37°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 4-2 and 4-3. To understand the limitations of this cooling method, the two curves of Figure 4-2 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 flowrates, it is the finite heat capacitance which limits the cooling capability. It is not until the helium flowrate exceeds 25 lbs/hr (for an initial temperature of 60°R) that the controlling factors are film coefficient and temperature differential.

The steady state fluorine temperatures which should be expected at different helium flowrates and inlet temperatures are given in Figures 4-3. The point to note here is that there is indeed a danger of actually freezing fluorine within the fluorine tank and it certainly 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°R. This would still make it possible to cool the fluorine to about 115°R with a helium flowrate of 15 lbs/hr (based on the use of 2-inch thick foam insulation).

Even though the use of chilled helium as the groundhold coolant makes it possible to substantially reduce the flourine temperature, there are

*It might be possible to avoid a pump-feed system by utilizing 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 $60^{o}R$.

Half-inch Diameter Cooling Coil, 8' long.

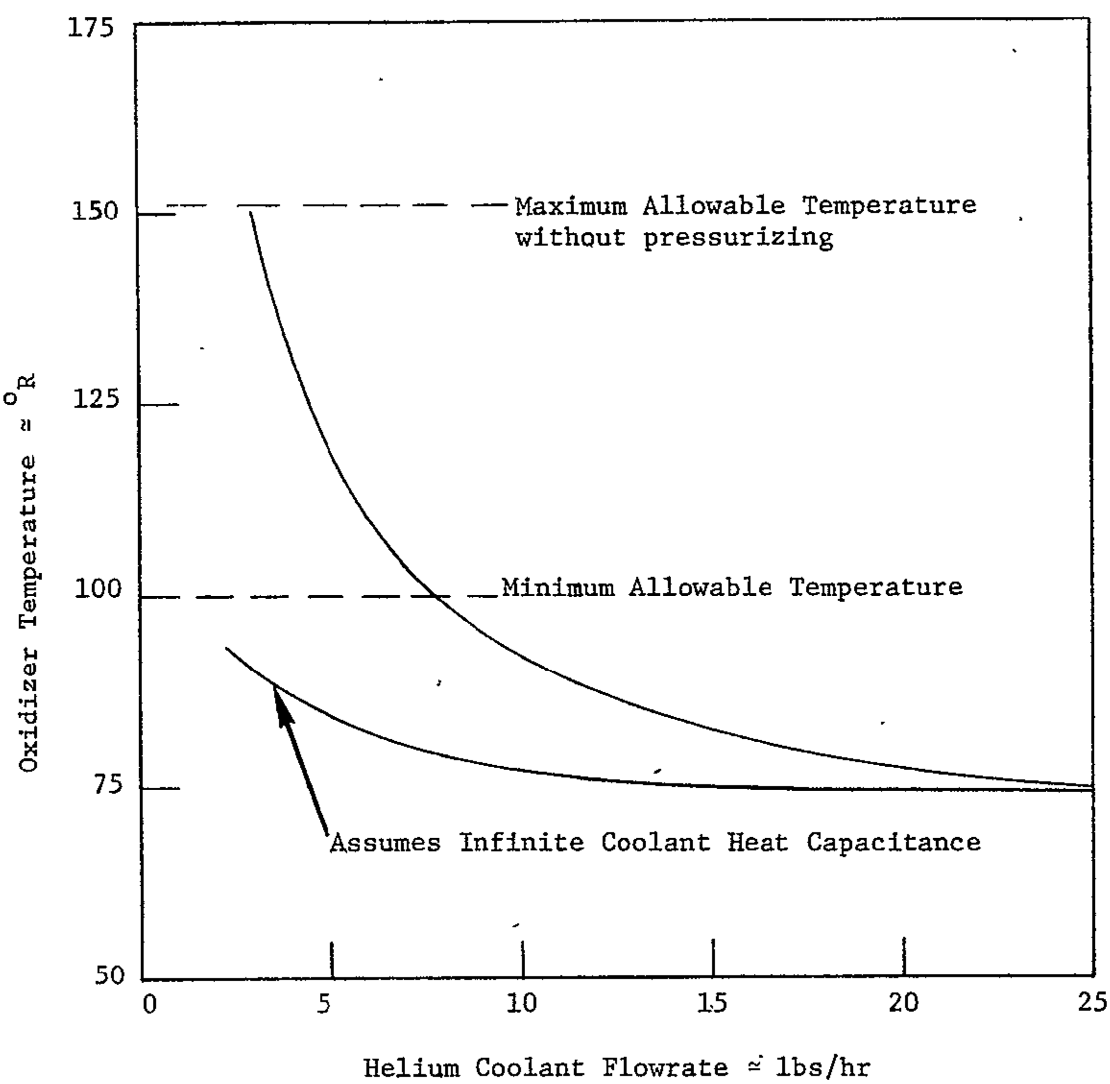


Figure 4-2. Effect of Helium Coolant Flowrate on Steady-State Oxidizer Temperature

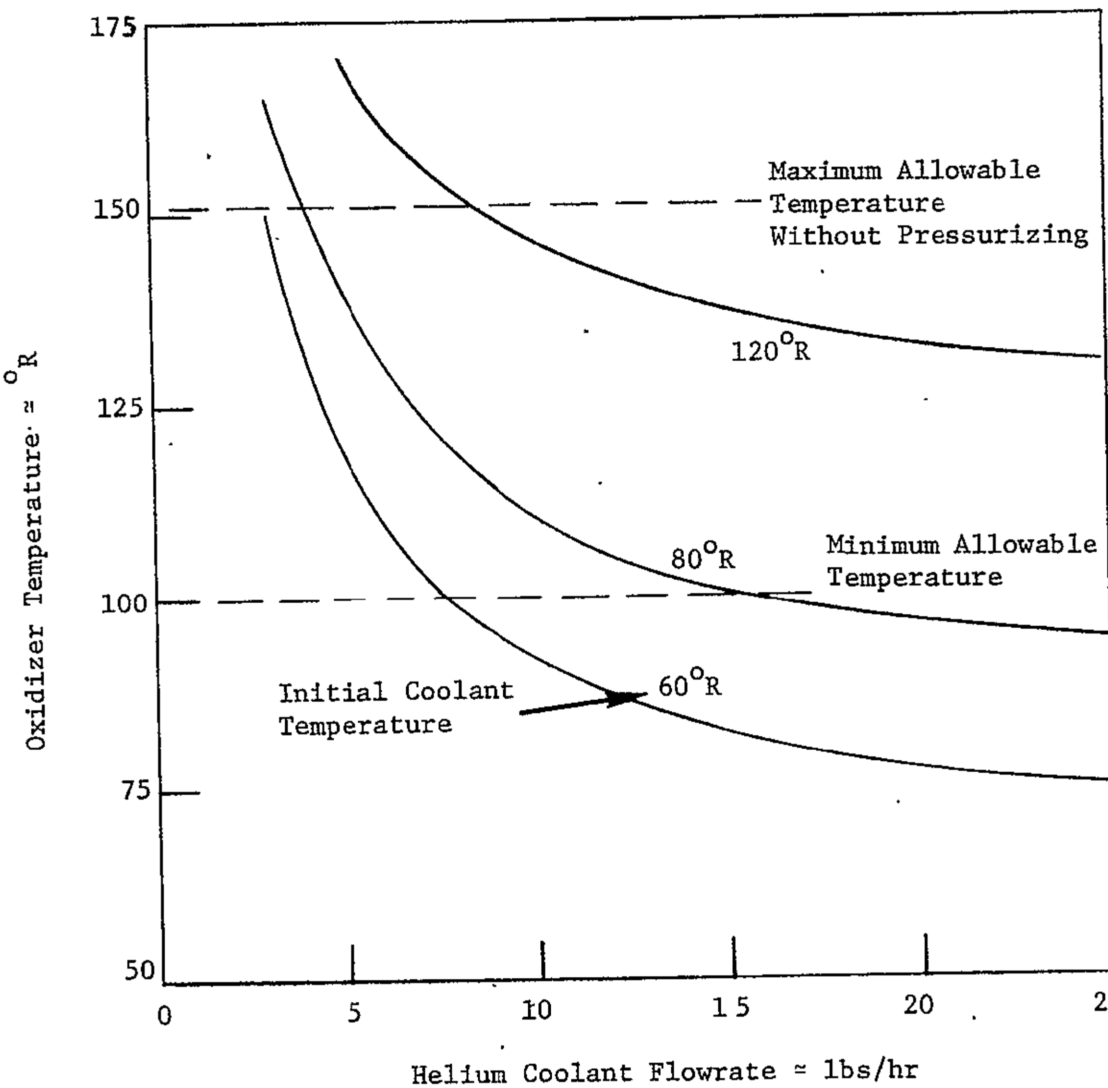


Figure 4-3. Effect of Helium Coolant Flowrate and Temperature on Oxidizer Temperature

serious ground support problems to be solved. The use of helium at a rate of 15 lbs/hrs for weeks is indeed 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 jacketed 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 $162^{\circ}R$, that is, allow the tank to pressurize to 10 psig; 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. Figure 4-4 shows the thermal response of the fluorine which may be expected without any coolant flow and the recovery rate with a high flowrate of LN_2 coolant. For the temperature range of interest, the rate of temperature rise of the fluorine is nearly constant at less than $0.7^{\circ}R$ per hour.

Figure 4-5 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.

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

1. The fluorine and helium can be maintained below $152^{\circ}R$ using LN_2 as the coolant. If appreciable subcooling is desired, chilled gaseous helium will be required.

2. Two inches of foam insulation should be used in order to reduce the groundhold cooling load.

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.

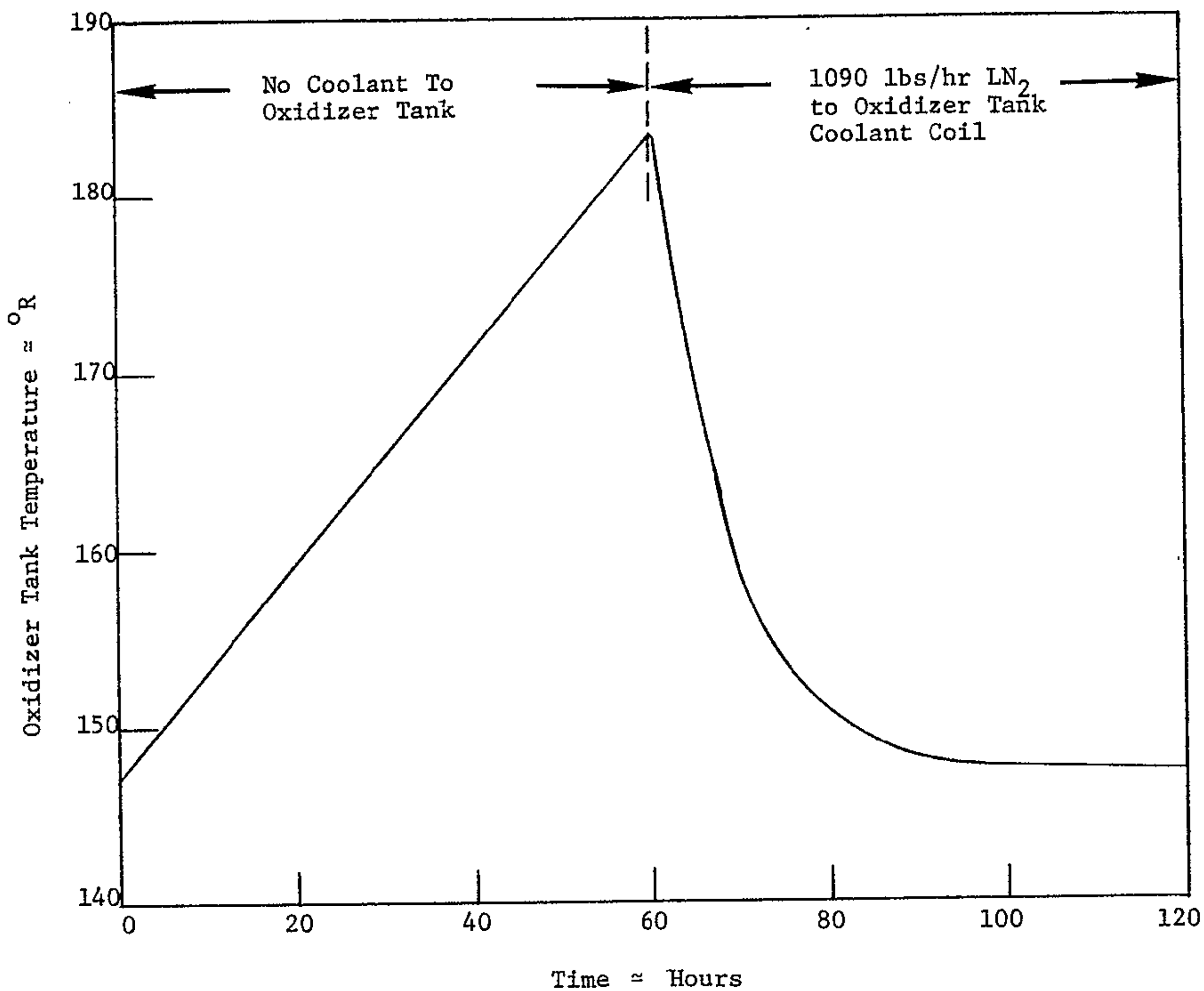


Figure 4-4. Thermal Characteristics of Oxidizer Tank With and Without LN_2 Coolant Flow. Two-Inch Thick Foam Insulation

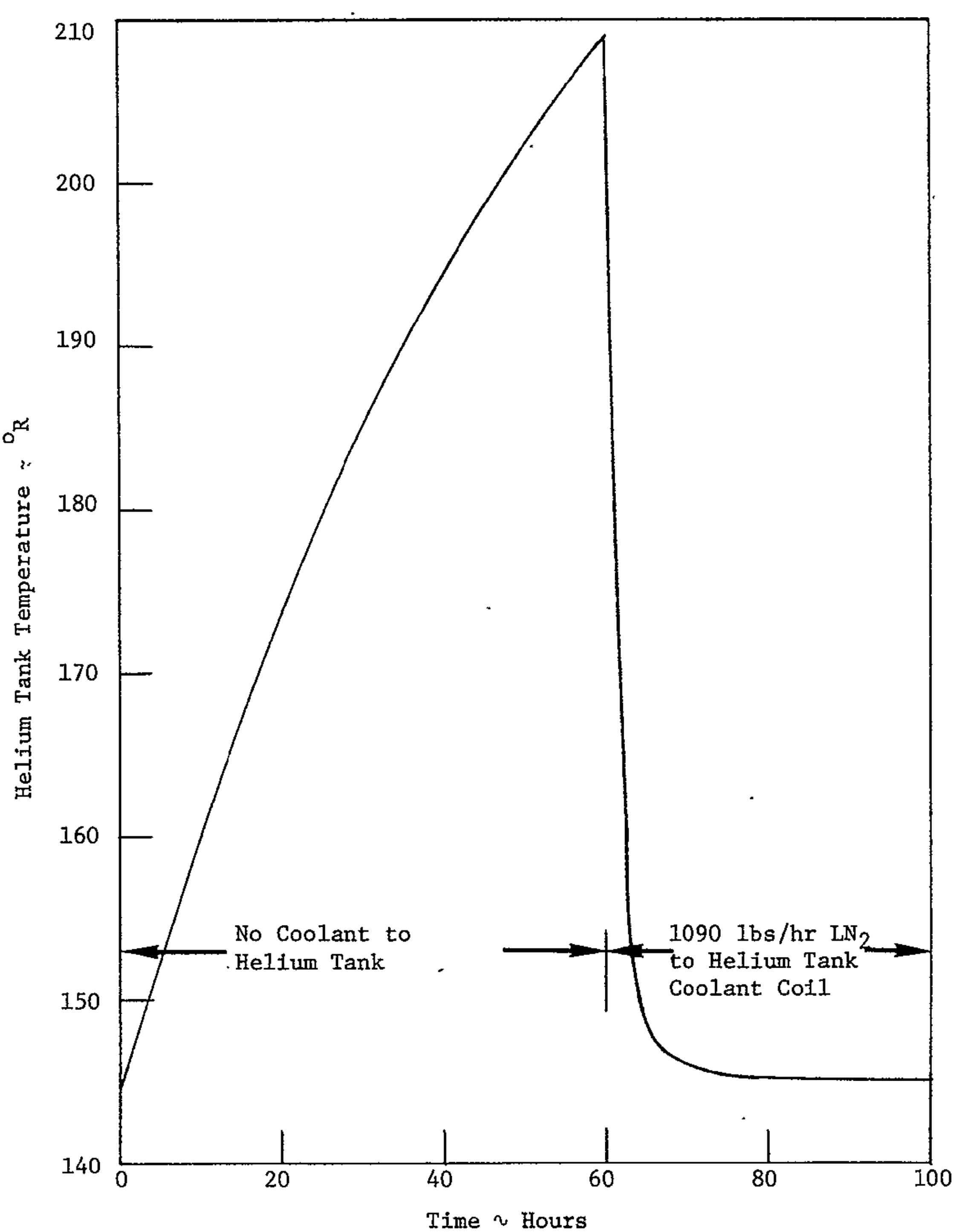


Figure 4-5. Thermal Characteristics of Helium Tank With and Without LN_2 Coolant Flow. Two-Inch Thick Foam Insulation

4.2 FLIGHT THERMAL ANALYSIS

During Task VI, concepts were established for controlling the module temperatures during flight. Those concepts are listed in Table 4-1. Three comments are in order concerning the choices made there as to systems chosen for further study.

1. Since the completion of Task VI, it has been established that the operating temperature of the RTG may vary in excess of ±50°R. In addition, it will be difficult to predict the RTG operating temperature in space within 10°R. Therefore, the passive control systems proposed for the N_2H_4 tank were reconsidered in this study to establish that indeed a passive system will not provide sufficient control. In addition, the area of study was expanded over that which was recommended in the Task VI report to include louvered systems since the passive systems did show marginal thermal control.

2. It will be seen from Table 4-1 that a polar tank support concept was contemplated for the fluorine tank. This choice was based upon the fact that this structural configuration reduced the heat transfer into the fluorine tank via the frame. During the initial study phases of Task VII it was clearly realized that such an approach has certain disadvantages:

   a. It results in the necessity of fabricating propellants tanks having different dimensions.

   b. The polar support approach is not readily accommodated by a tank constructed of Boron filament.

   c. Two different shaped tanks cannot be efficiently integrated into the module by using just one frame.

   For these reasons, it was decided to reconsider in this task methods of supporting the fluorine tank which would eliminate heat transfer to the tank and yet allow the use of identical propellant tanks supported by a single frame and platform. The resulting configuration has been discussed in Section 2 and as will be shown below, this approach is thermally satisfactory.

3. Finally, the Task VI study was based on the assumption that the helium will be stored in a warm condition. Whether it should be stored in a warm or cold condition is discussed below. However, for purposes of this study, it was decided to assume that the helium will be stored cold, less than 180°R. The consequences of switching the present design to accommodate warm helium will be discussed later.

Table 4-1. Summary of Thermal Control Concepts for Evaluation (From Task VI Summary Report, Reference 1)

Propulsion Module Component	Thermal Control Concept	Evaluation Rating	Recommended for further study
Hydrazine Thermal Control	Uninsulated area/passive rad.	12	•*
	Solid conduction bar/ passive rad.	15	•*
	Heat pipe/passive rad.	16	•*
	Louvered panel/passive rad.	16	•
	Uninsulated area/ louvers to space	21	
	Solid conduction bar/ louvers to space	25	
	Heat pipe/louvers to space	25	
Fluorine Thermal Control	Basic Isolated Tank	21	•
	Isolated Tank + Deployable Rad.	42	
Fluorine Tank Support	Spherical tank truss suppoer (type 1)	8	
	Polar Tank Support (type 2)	0	•
	Cylindrical Tank Support (type 3)	8	
Helium Thermal Control	Solid Conduction Bar/ Passive Rad.	3-18	•
	Heat Pipe/passive rad.	17	•

*Acceptability subject to RTG temperature uncertainty or variance being ±50°R or less.

Turning now to the flight thermal control of the N_2H_4 tank, the first point to consider is the insulation requirements. Figure 4-6 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 above about 0.01 Btu/ft^2-hr, the variation in conductance is not too important. Fortunately, it is not difficult to manufacture multilayer insulation having a conductance less than 0.01.

If a purely passive insulation system having a conductance of 0.01 were to be used, the propellant temperatures would vary as a function of RTG temperature as shown in Figure 4-7. It can be seen that the RTG could vary between 840°R and 980°R without causing the fuel to exceed its limits. It should be understood that this plot is a unique plot 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 4-8.

As a side point, it should be noted that the fluorine is relatively insensitive to the RTG temperature and should the N_2H_4 tank be 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 4-7.

The ideal thermal response characteristic is a small change in fuel temperature for a large change in RTG temperature (small slope). A step in this direction is 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 4-9. 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.

Note: For normal aluminized Mylar insulation blanket construction, the approximate relation between conductance and number of layers is as shown (Ref 3):

k/ℓ	Layers
0.003	25
0.01	10
0.03	4

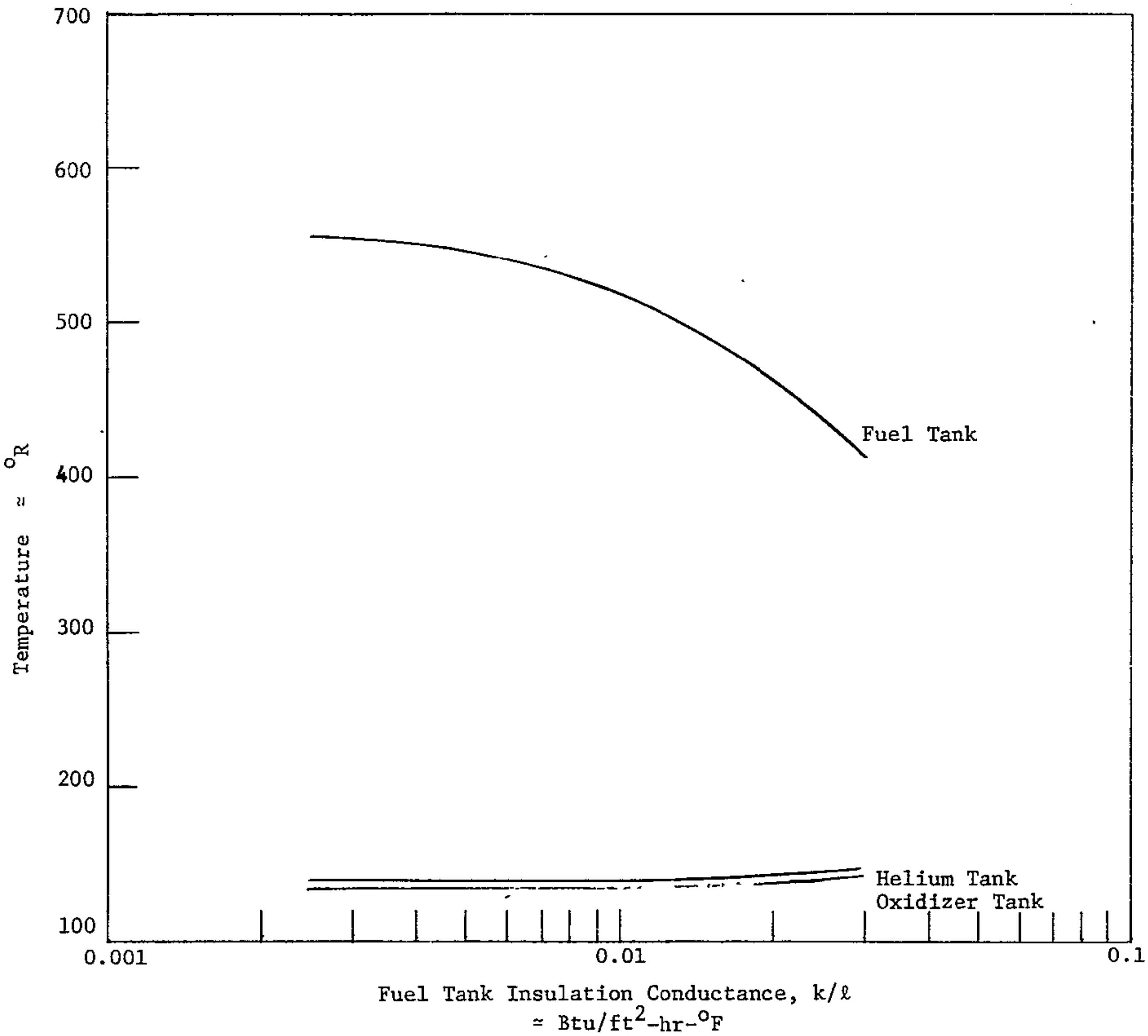


Figure 4-6. Effect of Fuel Tank Insulation Conductivity; RTG = 960°R

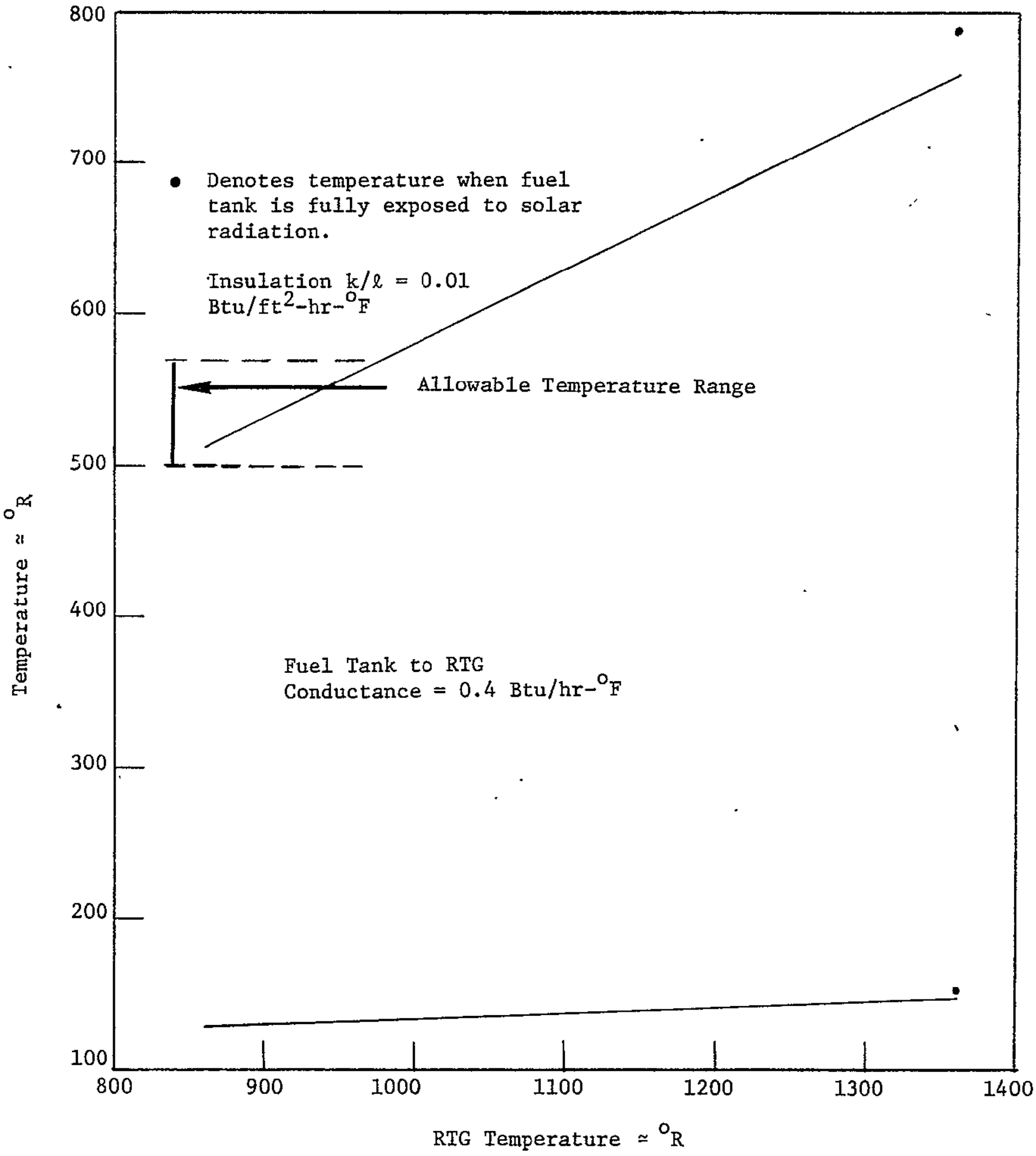


Figure 4-7. Effect of RTG Temperature on Propellants, Fuel Tank Fully Insulated

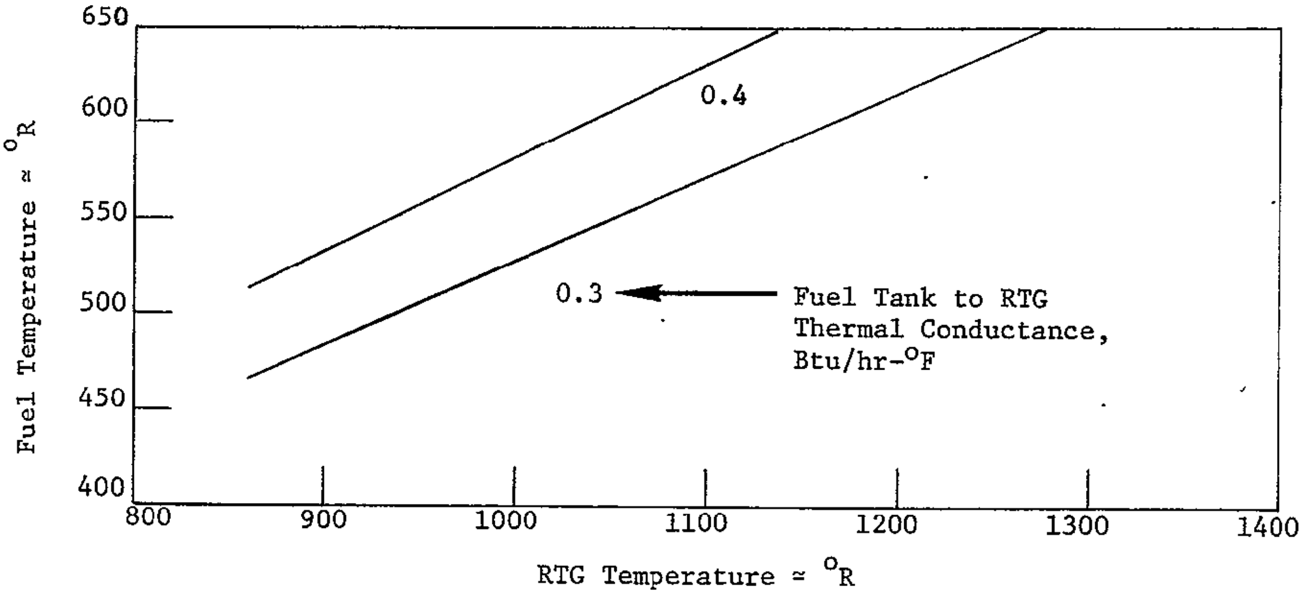


Figure 4-8. Effect of Fuel Tank/RTG Thermal Coupling on Fuel Temperature, Fuel Tank Fully Insulated

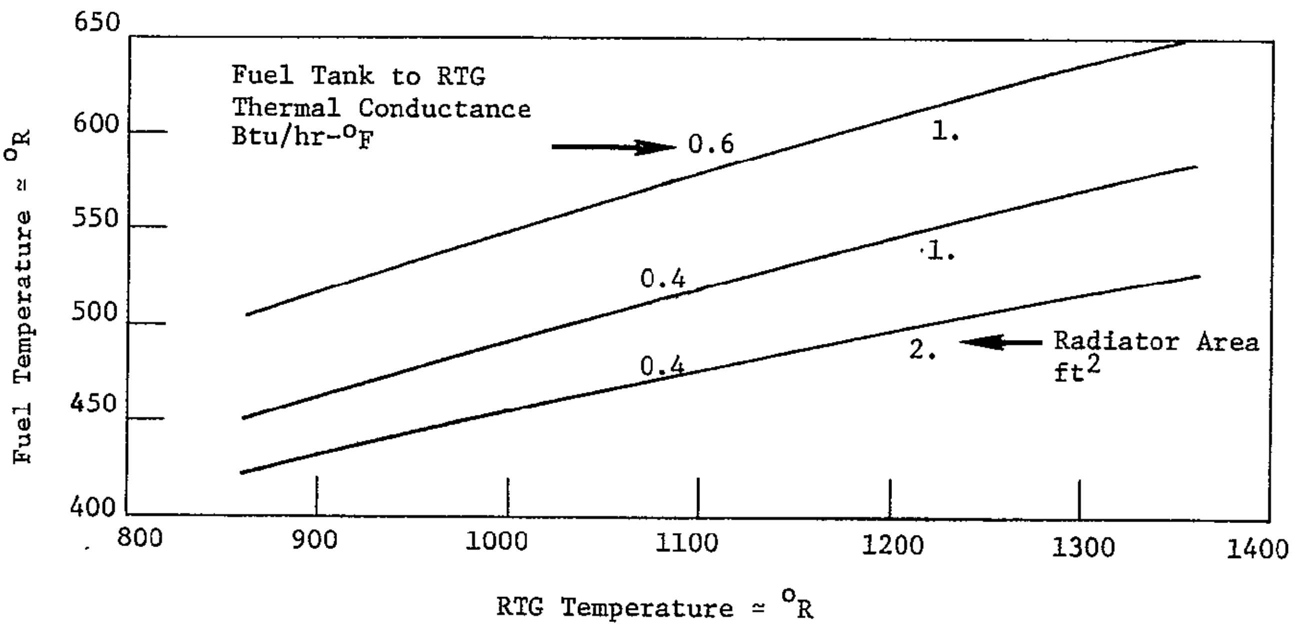


Figure 4-9. Effect of Fuel Tank/RTG Thermal Coupling and Radiator Area on Fuel Temperature; Radiator Control

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 2, the slope of the response curve is further reduced. Figure 4-10 shows the response which can be expected with a 2 square foot 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 can be made to vary radically by varying the louver area and the fuel tank-to-RTG conductance. This is shown in Figure 4-11. Changing the fuel tank-to-RTG conductance to 0.6 Btu/hr-^{o}F results in a shift of the curve only, but also changing the louver area to 4 sq ft stretches the curve. For this particular design the allowable RTG variation is about 440^{o}F. From this it is seen that there is an optimum louver size and fuel tank-to-RTG conductance associated with each RTG operating temperature.

The above discussion has been centered around the idea that the objective is to establish that control system which is most capable of accommodating wide variations in the RTG temperature. Actually, the objective is to establish that system which is most capable of accommodating any type of variation in the 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 is capable of allowing the widest RTG 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 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 simple approach of using a fully insulated system should be used. If the RTG might vary by more than 200^{o}R there is no option but to utilize louver control. This should not be viewed as an appreciable penalty. Louvers

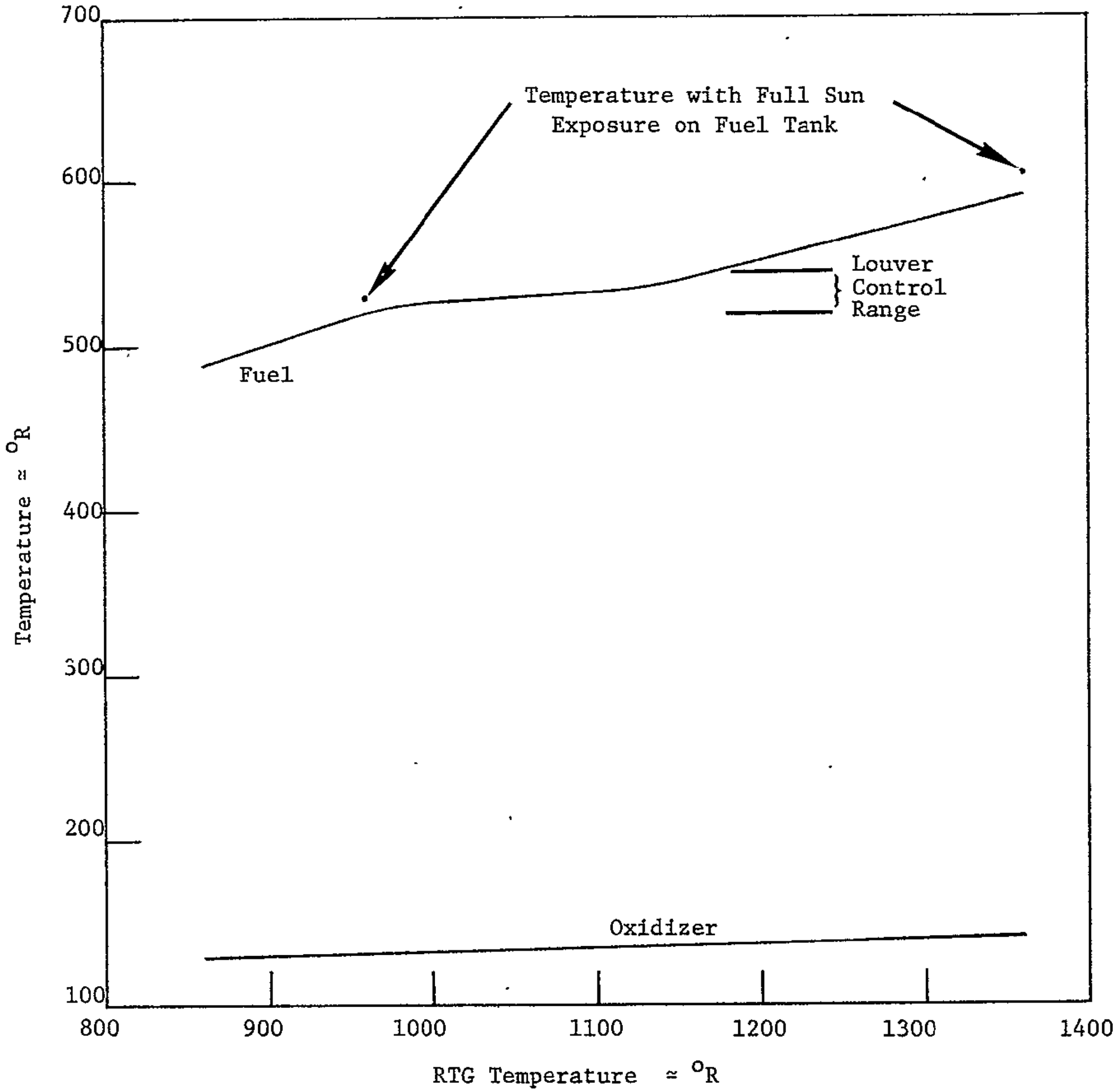


Figure 4-10. Effect of RTG Temperature on Propellants; Louver Control

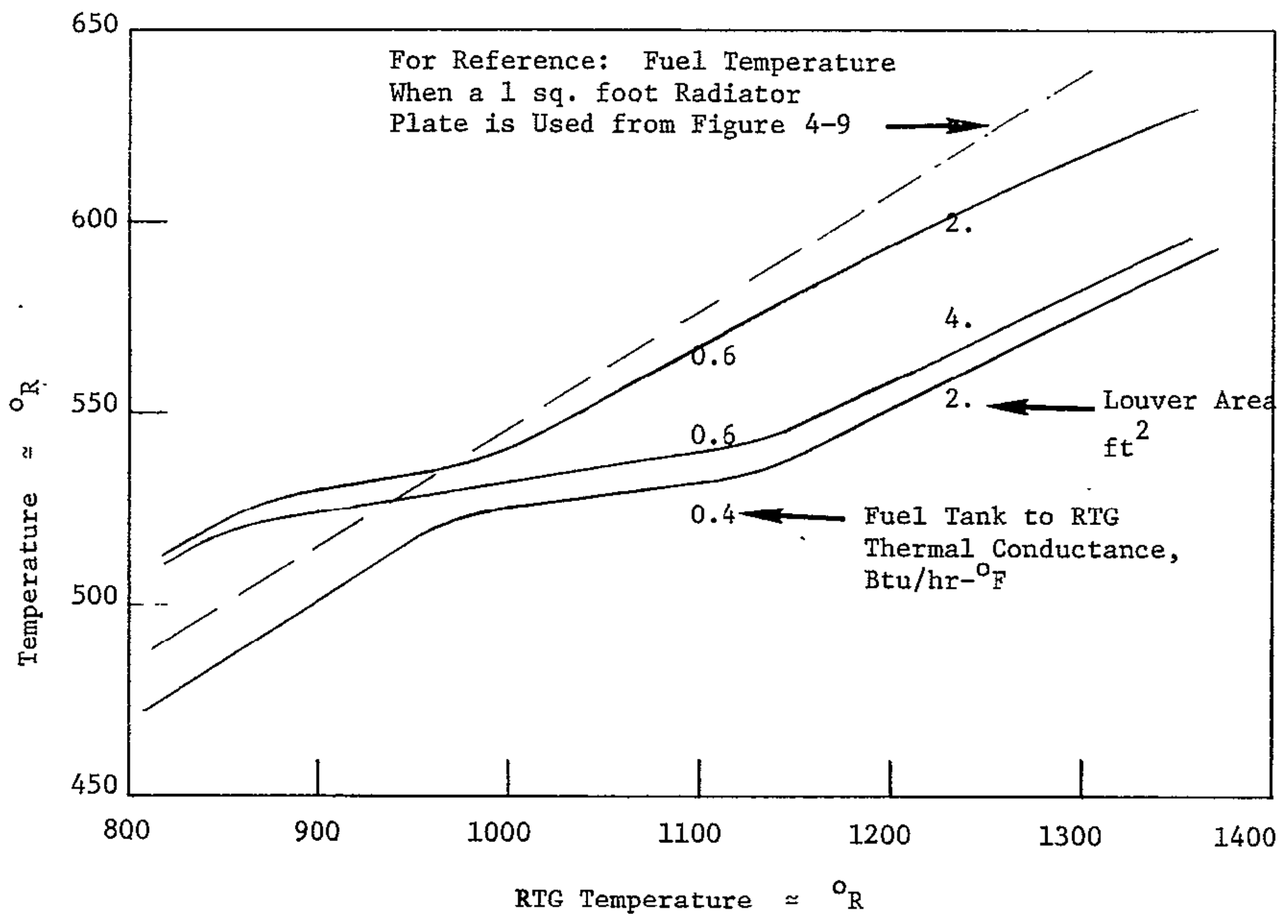


Figure 4-11. Effect of Fuel Tank/RTG Thermal Coupling and Louver Area on Fuel Temperature; Louver Control

have shown remarkable reliability in actual use and if properly designed are fairly light (approximately 0.8 lbs/ft^2). Considering the fact that the operating temperature and characteristics of the RTG are not known and may not be known until after the thermal control system design must be frozen, it may be wise to arbitrarily decide to use louvers. The penalty in weight and reliability is actually minor.

To this point, the discussion has been directed towards the manner in which the N_2H_4 tank is to lose heat to space. There is also the problem of obtaining heat. Basically, there are three ways in which this heat may be obtained.

1. Internal Heat Source Radioisotope heaters could be used. These heaters weighing about 0.2 ounces per 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. The disadvantage of this approach is that the use of radioactive elements within the propulsion module may cause a radiation field which interferes with the experimental packages or communication equipment.

2. External RTG/Louver. It is possible to gain sufficient heat from the RTG via a louver which views the RTG provided the RTG is placed sufficiently close to the louver, (approximately 2 feet for a 960°R RTG). If the louvers which see only space are retained, this approach would show an unusually large capability of accommodating variations in the thermal environment. The louvers which see only space would produce the thermal response indicated in Figure 4-11. The addition of an RTG louver would more than double the allowable variation of the RTG since that louver could effectively cut off heat from the RTG should the RTG temperature rise too high. By eliminating the louvers which see only space, the weight would be held to a minimum and yet the thermal response curve of the system would be similar to that which is shown by the center curve of Figure 4-11.

This approach has two disadvantages. First, the necessity of moving the RTG close to the module may set up unacceptable radiation fields. Second, the RTG 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 in the blades. Thus, the closing of the louvers will result in the RTG temperature shifting upwards appreciably. The amount of shift is dependent 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.

3. External RTG/Conductor. This approach would allow heat to be transferred directly from the RTG to the N_2H_4 tank via some type of conductor. The discussion above indicated that the conductance between the RTG and the N_2H_4 tank would have to be of the order of 0.3 to 0.8 Btu/hr-°R for a 960 °R RTG. Translated into heat transfer this means about 175 Btu/hr. Assuming the RTG is 6 feet away from the tank, a 2-inch diameter solid conductor of aluminum would be required. This is obviously quite heavy, greater than 20 pounds. However, a 3/4-inch ammonia heat pipe would be capable of passing approximately 1500 Btu/hr and its weight would be about 3 pounds.

   The disadvantages of a heat pipe are two, both of which are minor. First, a heat pipe is susceptible to corrosion over a long period of time. However, ammonia 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 pipes 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.

Considering all three of these approaches, the first is by far the superior from a thermal standpoint. If it is unacceptable because of field interference, the use of a heat pipe running from the RTG to the N_2H_4 tank appears best. It is light and has ample heat transfer capacity. However, care would still have to be exercised in preventing it from causing unacceptable gradients in the RTG.

Turning now to the thermal control of the fluorine and helium tanks, it should first be noted that the approach adopted is to reduce the heat transfer to these tanks to a minimum and then allow the tanks to lose heat to space. Only in the case of relatively short durations is the heat capacitance of these tanks relied upon to prevent excess temperatures.

The first aspect to consider is the insulation requirements. Figure 4-12 shows that it is relatively unimportant to flight thermal control how much foam is utilized on 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 2 inches of foam be used on the tanks is entirely compatible with flight thermal control.

There are insulation requirements in two areas which must be met in order to keep the heat transfer to the cold tanks to a minimum. First, a radiation barrier 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. As indicated in Reference 1, such a shield prevents a heat gain by the fluorine tank of 6 to 8 Btu/hr.

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 4-13. This plot applies to the particular situation in which the distance between the spacecraft insulation and the fluorine tank insulation is approximately 9 inches. This is a critical factor, in fact 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 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 percent comes by radiation from the spacecraft to the top of the fluorine tank.

Limited consideration was given to placing the helium tank above the propellant tanks as shown in Figure 4-14. The objective was to give the bottom of the spacecraft and the top of the fluorine tank a better view of

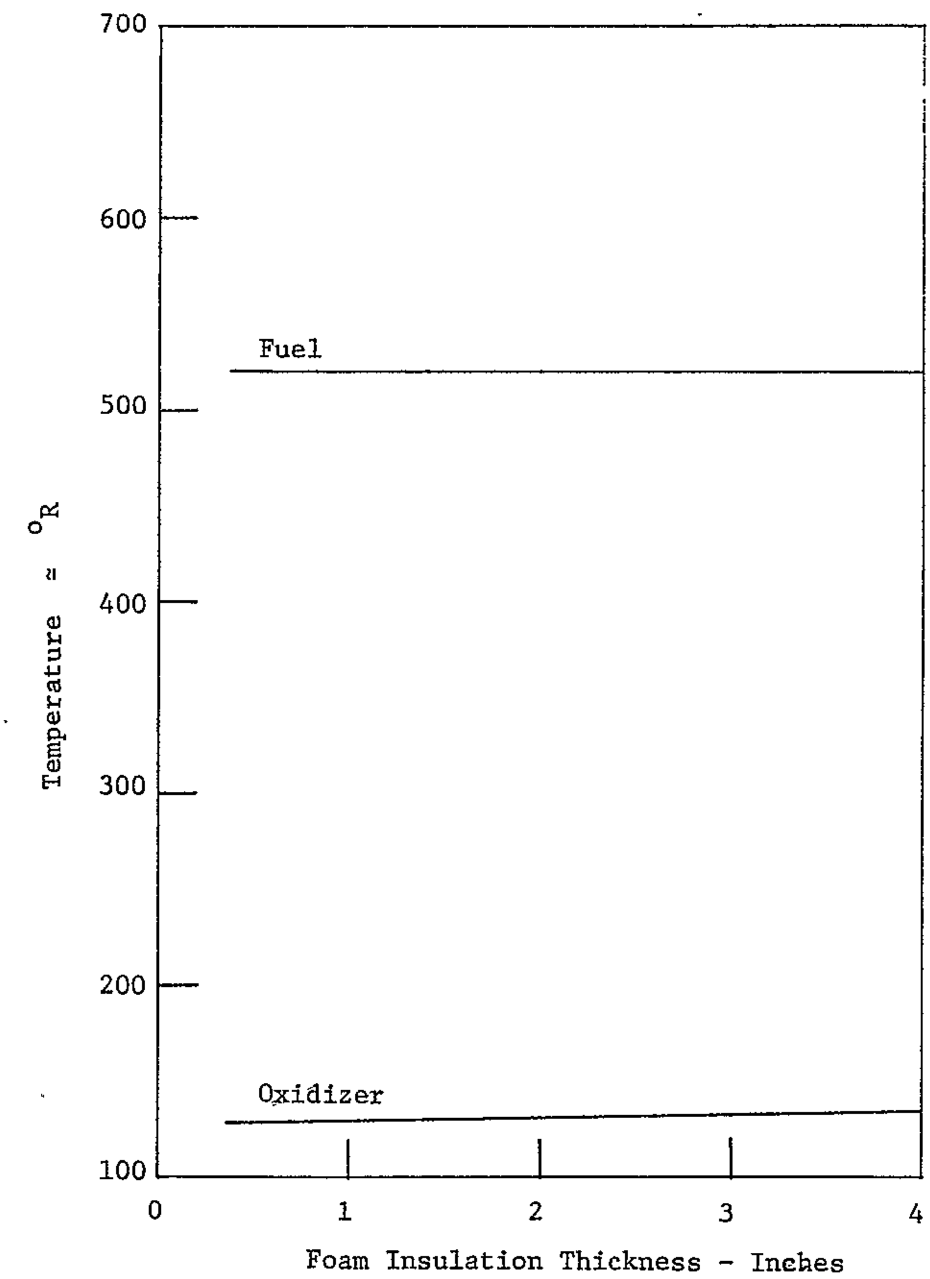


Figure 4-12. Effect on Propellant Temperatures of Thickness of Oxidizer Tank Foam Insulation; Fully Shaded Condition

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

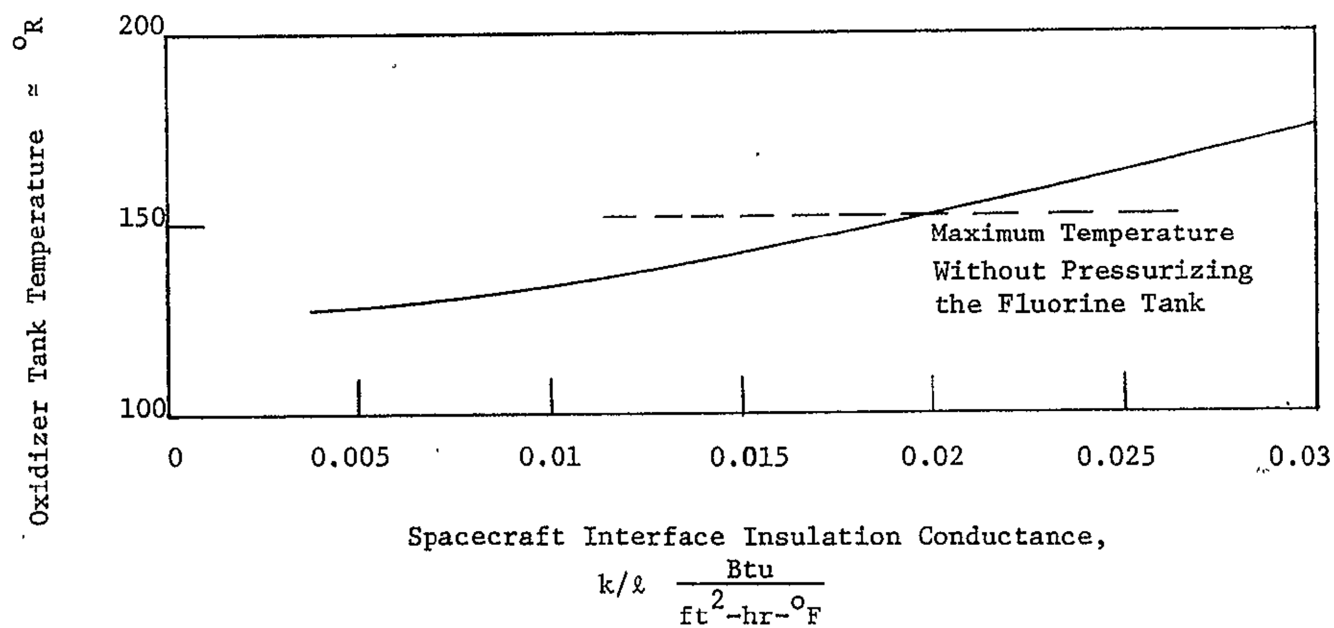


Figure 4-13. Effect of Conductance of Spacecraft Interface Insulation Conductance on Oxidizer Tank Temperature

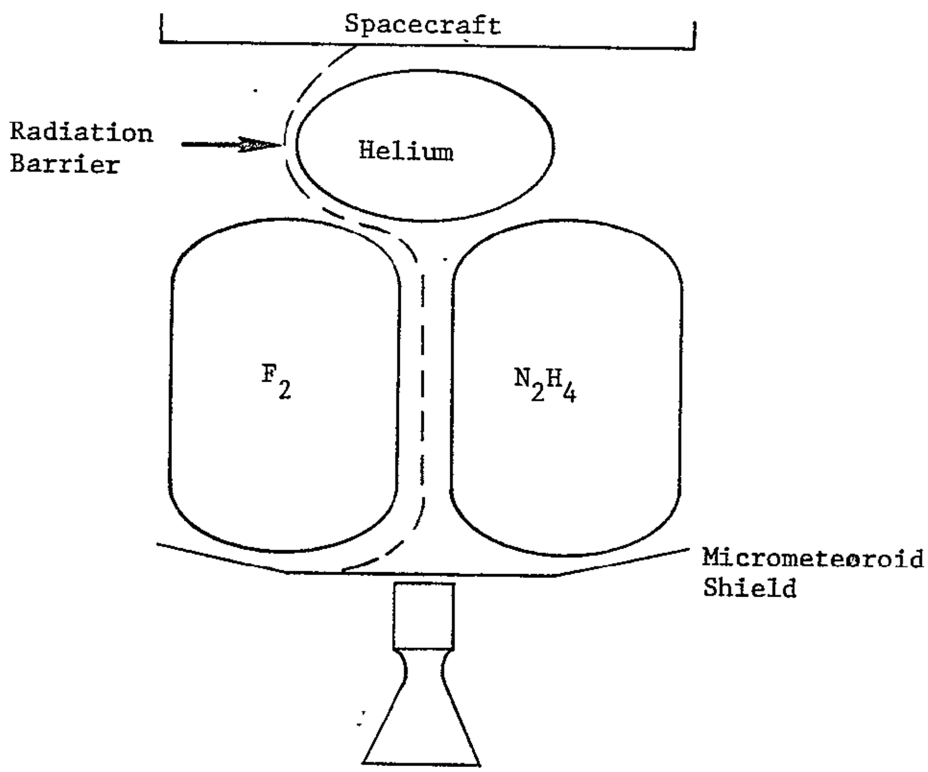


Figure 4-14. Alternate Configuration

space and thereby reduce the quantity of heat emanating from the payload which gets to the fluorine, and increase 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 located and near the bottom of the spacecraft results in it gaining nearly as much additional heat as the incremental amount which the fluorine tank 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, being 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 barrier
- The helium tank receives more heat from the spacecraft in this position than did the fluorine tank because it is centrally located under the spacecraft.

It was therefore concluded best to retain the present configuration.

As might be expected, it takes a very small steady state heat transfer rate to cause excessive temperature in both the oxidizer and helium tanks. The curves of Figure 4-15 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°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 is dependent upon the intensity of the sun (time since launch), module orientation, and the solar absorptivity of the insulation. Figure 4-16 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 material advantage of launching in a substantially subcooled condition. For each degree of subcooling approximately 2 hours of solar exposure may be

Notes: Direct solar exposure on sides of oxidizer and helium tanks. All surfaces exposed to solar heating are covered with second surface-silvered Teflon.

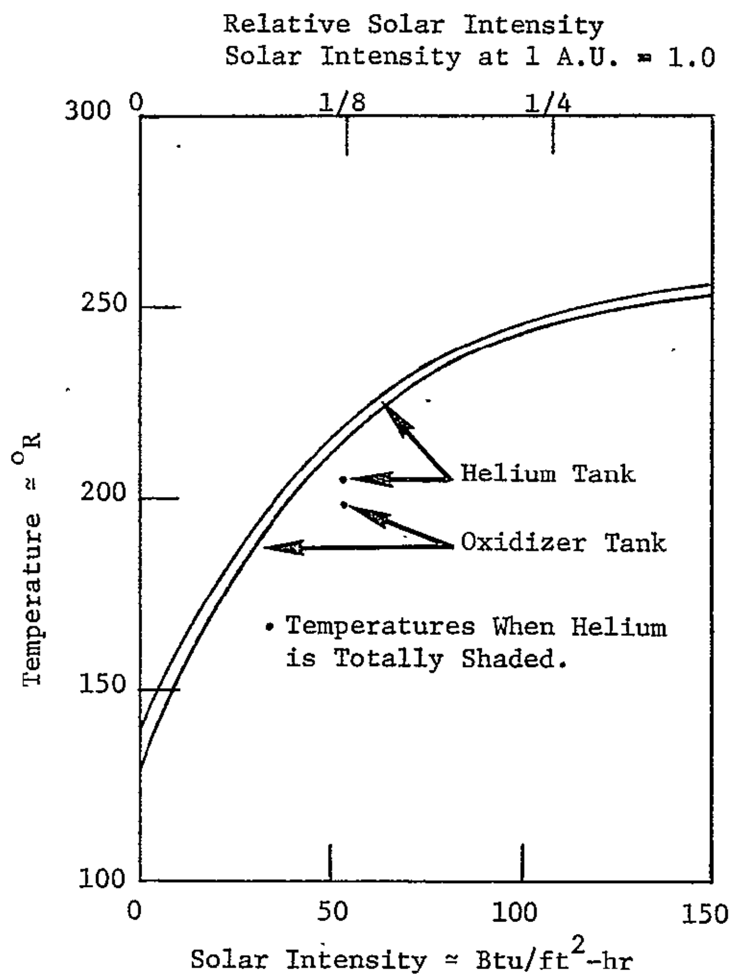


Figure 4-15. Effect of Solar Radiation on Steady-State Oxidizer and Helium Tank Temperatures

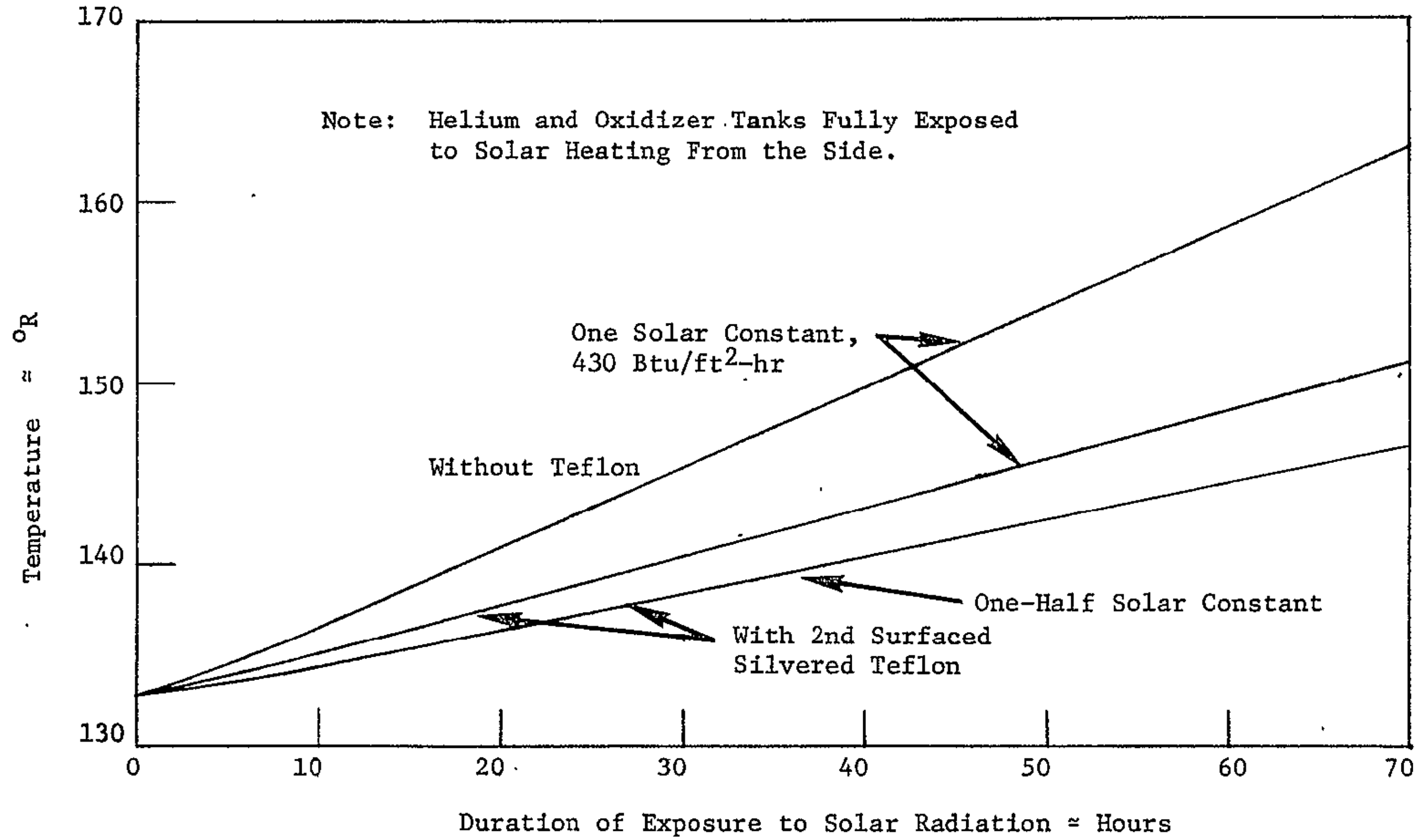


Figure 4-16. Effect of Exposure to Solar Radiation on Oxidizer Tank Temperature

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 6 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 being dependent upon the sun intensity and initial temperature.

As will be indicated later, there is the possibility that propulsion considerations will dictate that the helium be stored in a warm condition, approximately 520°R. During the Task VI analysis it was assumed that the helium is warm, and in the Task VI report, Reference 1, it was stated that keeping the helium warm presented no problem. It was also stated that warm helium presented no problem to keeping the fluorine cold. Although a detailed analysis of that particular situation was not pursued during Task VII, there is no reason to believe that the helium may not be stored warm. By using a nonmetallic frame the conduction of heat from the helium to the fluorine can be effectively eliminated, and the radiation barrier could 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 but on the other hand it must be noted that 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.

The last area to consider in the 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 4-17. This is not an accurate picture simply because the engine configuration is not yet known; but it does furnish a sufficient configuration to allow the determination of general thermal characteristics.

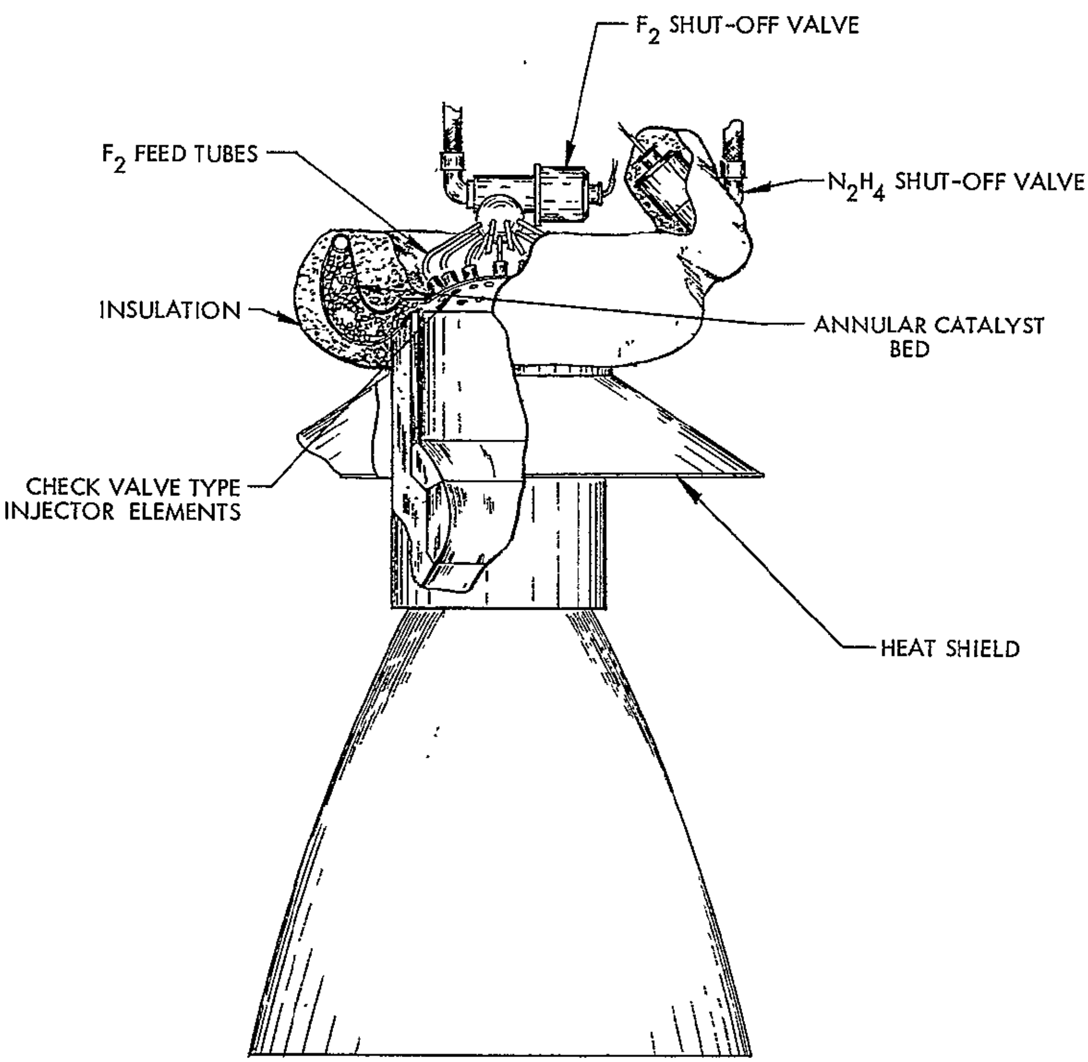


Figure 4-17. Pictorial Conception of Bi-Mode Engine

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.

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. As such, the engine temperature may be held at any desirable level between 110°R and 400°R by controlling its view factor of the RTG.
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 were to be maintained at the same temperature such an approach might be feasible. Second, some auxiliary energy source must be available in order to bring the catalyst bed and N_2H_4 shutoff valve up to 490°R 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 4-18, 4-19, and 4-20. From Figure 4-18, it can be

Note: Constant Heat Addition to Bed and Fuel Valve of 10 Btu/hr each.

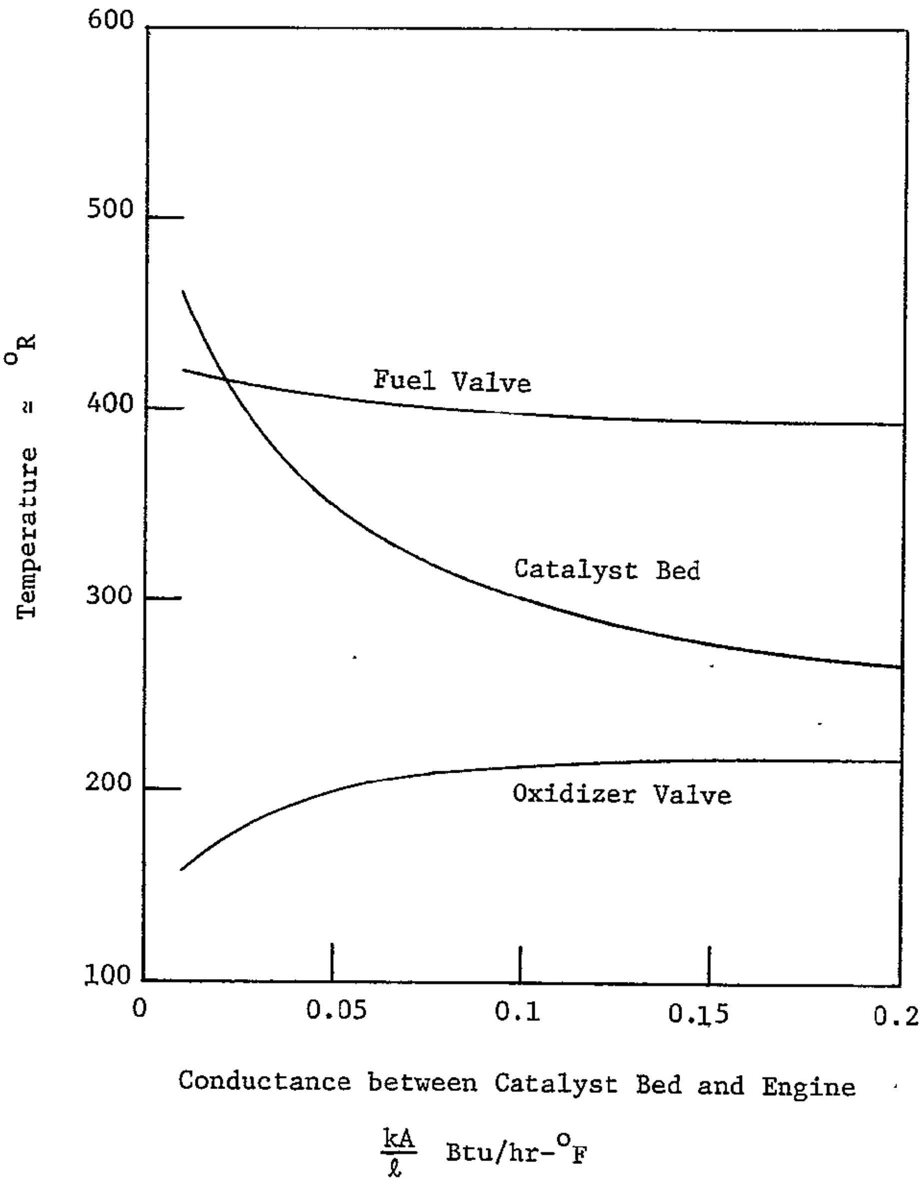


Figure 4-18. Effect of Conductance Between Catalyst Bed and Engine on Valve Temperatures. 20 But/hr Heat Addition, Non-Operative Engine

Note: Constant Heat Addition to Bed and Fuel Valve of 20 Btu/hr each.

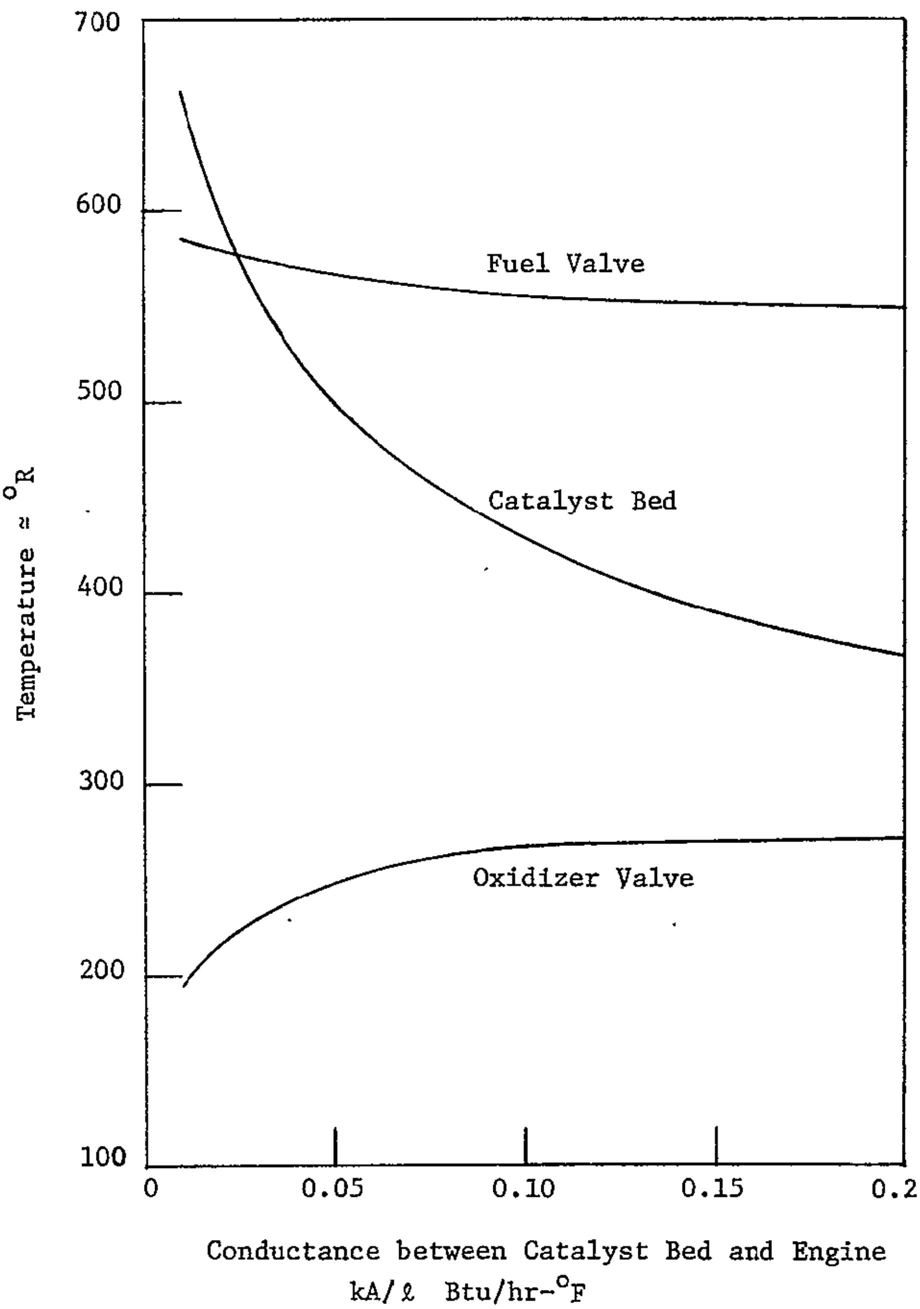


Figure 4-19. Effect of Conductance between Catalyst Bed and Engine on Valve Temperatures. 40 Btu/hr Heat Addition, Non-Operative Engine

Notes: 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)

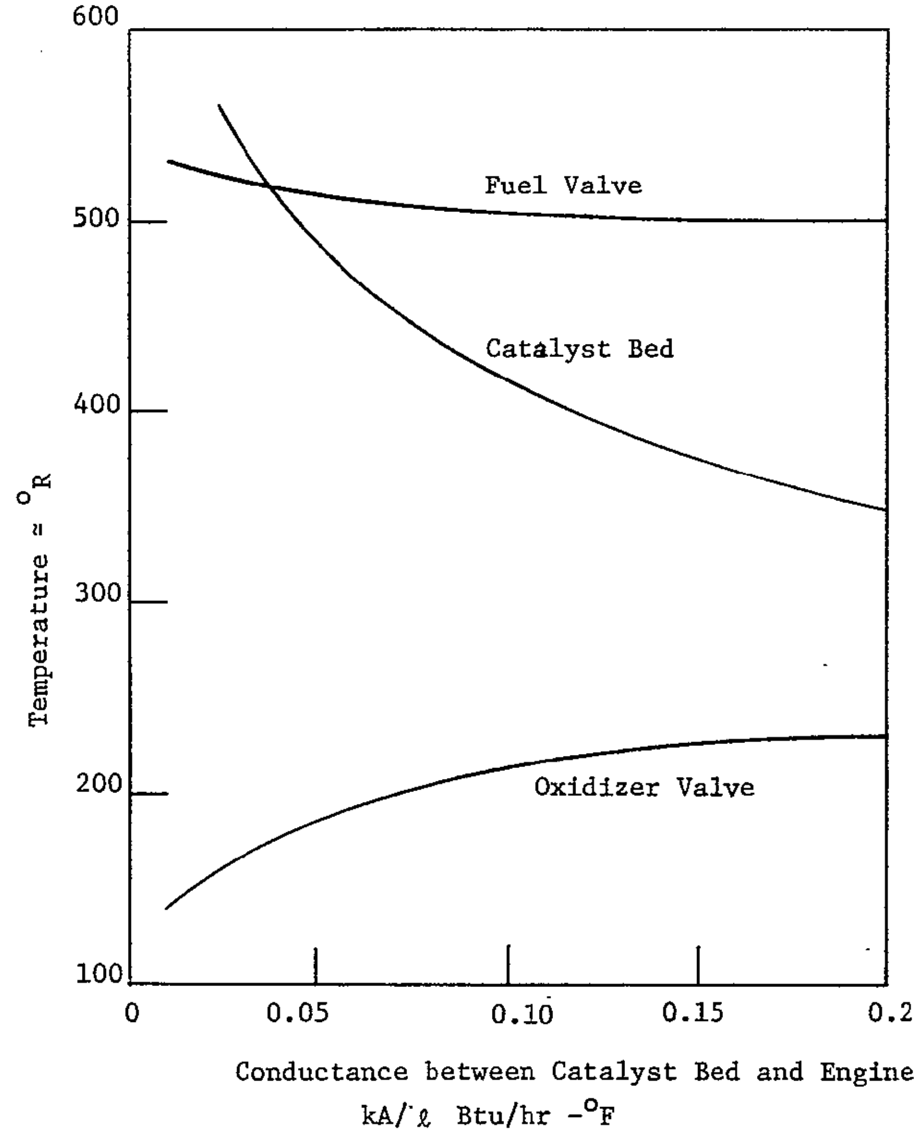


Figure 4-20. Thermal Characteristics of Adjusted System, Non-Operative Engine

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 4-19), but it also drives the oxidizer valve to excessively high temperatures. There is the possibility that fluorine valve temperatures in the neighborhood of $200^{\circ}R$ to $250^{\circ}R$ 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 4-20 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-$^{\circ}F$ 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 either, provided the valve is shielded from radiation from the engine.

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, 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 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 4-17. Although the valves are maintained at their proper temperature level during engine operation, the insulation will get too hot without radiation protection.

After shutdown it appears that all components except the fluorine valve will drop in temperature if the auxiliary heating is stopped. Figure 4-21 indicates the typical thermal response at shutdown. It shows the fluorine valve rising some 150^{o}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, let it be noted that it is not clear that conditioning is necessary. The various aspects of this requirement are discussed below. 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. It appears that the propellant which is pressurized by the conditioned helium can be used to condition the helium. If this approach were used, the conditioning process would raise the propellant temperature (or lower the propellant temperature depending on the type of conditioning) some 2^{o} to 6^{o}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 so that local freezing of N_2H_4 is avoided.

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

1. The N_2H_4 tank can be kept within the required temperature limits by obtaining heat from either small radioisotope heaters located inside the insulation or the external RTG.

2. If the RTG is relied upon as the source of heat, a small heat pipe should be utilized to transfer the heat to the N_2H_4 tank.

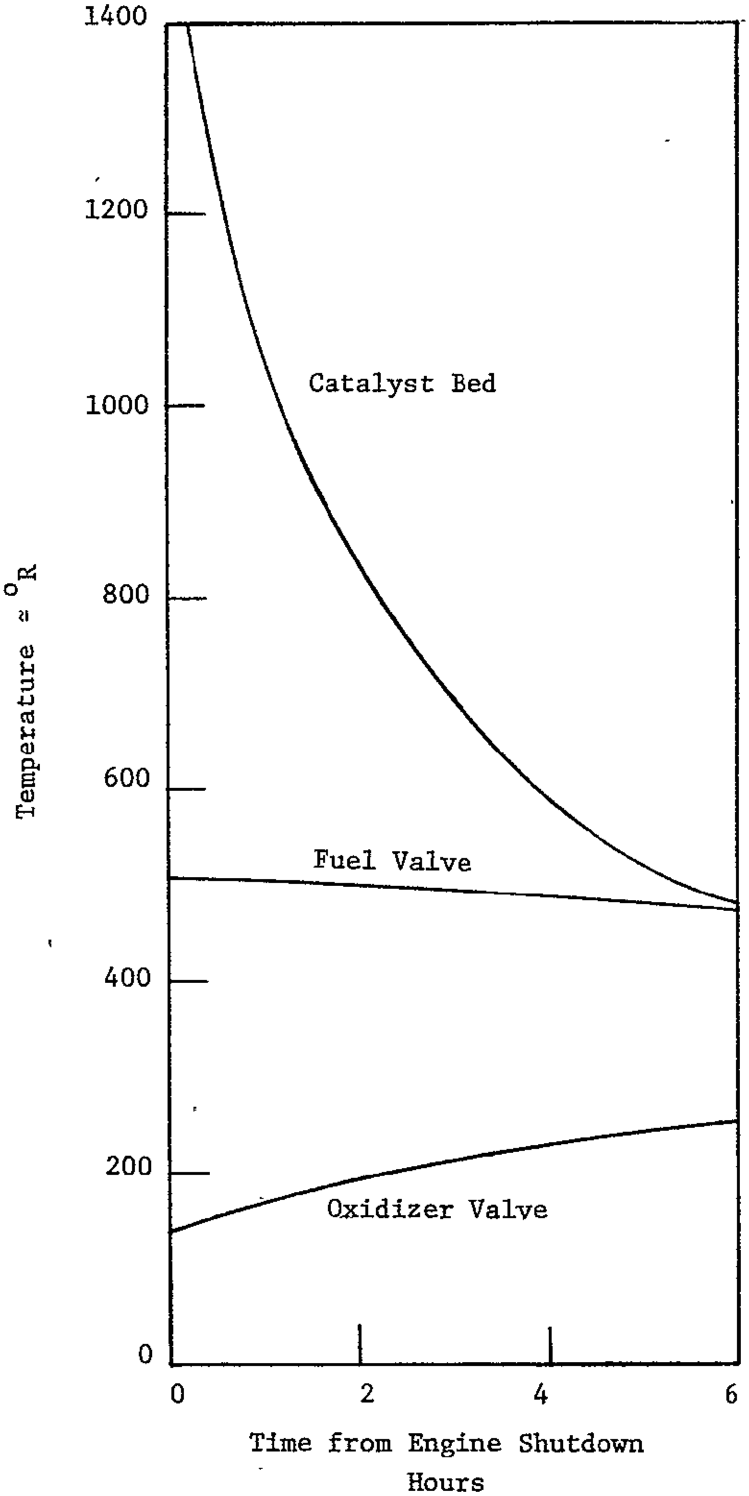


Figure 4-21. Thermal Transient After Engine Operation Non-Operative Engine

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

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

5. The fluorine and helium tanks may be kept below their maximum allowable temperature limit provided continued exposure to solar radiation is avoided, the spacecraft/fluorine tank interface is well insulated, a radiation barrier between these tanks and the N_2H_4 is provided, and a properly designed nonconductive frame is used.

6. Depending on the particular 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.

7. The steady state nonoperative temperature of the engine support components (valves, catalyst bed, etc.) is highly dependent upon the system design.

8. Any necessary conditioning of the helium can probably be accomplished by a heat exchanger which utilizes the propellant which is to be pressurized.

4.3 PROPULSION ANALYSIS

Early in Task VI, a preliminary estimate was made of the thermal constraints which should be imposed upon the F_2/H_2H_4 SSPM due to various propulsion system considerations. It is worthwhile to reproduce the results of that effort together with pertinent comments.

a) Minimum hydrazine temperature - The JPL-established minimum temperature of 40°F is reasonable since it allows more than 5°F margin above the freezing point.

b) Maximum hydrazine temperature - The JPL-established maximum temperature of 90°F is a conservative limit, chosen to minimize both the spontaneous and catalytic decomposition rates. No other problems are associated with this temperature.

c) Minimum fluorine temperature - 100°R is a rather low temperature for a fluorine system. Ordinarily a lower temperature of 120°R or so 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) makes the thermodynamic freezing problems disappear.

d) Maximum fluorine temperature - Operation of 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).

e) 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 as follows. Parts below the isolation valve may be allowed to rise as high as 120°F. If possible, post-firing heat soak-back should not raise the trapped liquid above 660°F. 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 be likely to detonate.

f) 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 as follows. Parts below the isolation valve may be allowed to rise to 200°R. Post-firing heat soak-back should not raise the trapped fluorine to any higher temperature than necessary since the reaction potential is somewhat increased as temperature increases.

The work performed during Task VII revealed nothing which would indicate the comments (A) through (E) are not applicable. However, the Task VII effort has indicated that it may be wise to qualify the comments of (F) in relationship to the allowable thermal transients upon engine ignition. Also during Task VI, preliminary considerations indicated that there were valid arguments for storing the helium cold as well as warm. This point was reconsidered during Task VII in an effort to clarify the issue, and again, certain qualifying restrictions become apparent. These two areas will now be discussed.

4.3.1 Optimum Helium Storage Temperature

In establishing the best temperature range for storing the helium, many aspects must be considered. Ultimately, these aspects can be reduced to three points:

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 in order to result in 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 the approach designwise.

As for the propulsion system requirements, the real problem here is probably one of reliability. The use of warm or cold helium should not directly affect engine performance. Only to the extent that (a) how much helium is required or, (b) how it must be conditioned 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 engine auxiliary equipment.

The establishment of the minimum weight pressurant system is an extremely complicated problem, one which is not adequately understood or for which there is sufficient experimental data available upon which responsible decisions may be made.

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 or 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; i.e., 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 quite 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 now the ullage gas temperature is unknown.

It is possible to approach these problems by what is essentially an empirical approach which depends upon experimental data. This approach is described in Reference 4. In order to establish at least an indication of the comparative weights of a warm and cold helium system, it was decided to proceed along this path.

First, the final pressure remaining in the helium tank was calculated for the case where the storage temperature was the same as the fluorine bulk temperature (i.e., 150°R) and 36 pounds 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 (i.e., $w/\dot{w} = 1$). Midcourse 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 Appendix C.

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

A similar claculation 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 (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 indicated that the helium would be exhausted before completion of the orbit inclination burn. Translated into weight, the warm system contains only 17.8 pounds 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.,

$$M_{T} + HE = 6.22\ M_{HE} \text{ at } 550^{o}R$$

$$M_{T} + HE = 3.07\ M_{HE} \text{ at } 180^{o}R$$

Furthermore, the collapse factor in the fluorine tank ullage works to the disadvantage of the warm system. A collapse factor of 1.82 for $500^{o}R$ fluid means that 1.82 times as much helium would be required as would be required if the incoming gas were not chilled once it was in the ullage. Therefore, the effective mean temperature is not $500^{o}R$ but $275^{o}R$. 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.

Though the above comments do indicate clear trends, they must be considered with caution since the collapse factor correlation taken from Reference 4 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 four feet in diameter compared to the present 2.7 foot 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 $500^{o}R$ and $550^{o}R$ 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's must be added to the helium during each midcourse maneuver, 830 Btu's must be added during the orbit insertion maneuver, and 770 Btu's added 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 prepressurize 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's 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 dimishing 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 40°F 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 to be eliminated 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 the in-flight deployment of the RTG requiring two flexible sections in the tubing runs. A much easier design problem to solve is the use of a heat exchanger in the hydrazine feedline which would extract heat from the liquid flow. During steadystate operation, this process would chill the hydrazine by less than $3^{o}R$. A special advantage of this scheme 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.

4.3.2 Engine Related Thermal Problems

In addition to the heat exchanger problem, it appears that 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 $480^{o}R$ might be specified, although higher temperatures (say $500^{o}R$) 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 $+1900^{o}F$. 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 Soakback. 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 be assured that each of these problems is under control, it is necessary to devise a detailed accurate thermal model of the engine and its support equipment with realistic engine heat load inputs and to analyze the magnitudes of each troublesome mechanism.

4.3.3 Propulsion Analysis Summary

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 engine must be considered. The same may be said of the thermal problems of the engine support equipment (valves, catalyst bed, etc.). It is possible to make the following generalized comments:

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

2. A warm helium storage system presents the least design problems.

3. Several thermally-caused problems may arise in and near the engine.

5. EVALUATION AND RECOMMENDATION

The objective of Task VII was to

1) Investigate the fluorine/hydrazine module thermal control concepts proposed for further study as a result of the Task VI work

2) Establish by considering their relative merits the best thermal control system

3) Establish areas which should be further studied.

In the past, the various systems were weighed as to their relative merits according to six separate subjective standards (Reference 5):

Weight
Reliability
Effectiveness
Adaptability
Testability
Cost

In addition, for a system to be considered worthy of further attention, it was mandatory that it meet three specific requirements:

1) It had to show a weight savings relative to a propulsion module which uses earth storable propellants.

2) It must not collect any frost or water during the groundhold phase.

3) It had to maintain the module components, and propellants within the specified temperature limits at all times.

Unfortunately, it is not entirely possible to perform such a complete evaluation in the present case because the module design cannot be specified with sufficient finality as to allow such an evaluation. Unlike the OF_2/B_2H_6 system studied in the first five tasks, the fluorine/hydrazine system is critically determined in certain respects by detail engine design, RTG operating characteristics, and specific mission parameters.

However, the evaluation will be made to the extent possible. First, it is wise to review the general results of previous sections when viewed as a unit.

5.1 STRUCTURAL CONFIGURATION

From a purely structural point of view the basic arrangement of two propellant tanks, a single pressurant tank located below the propellant tanks, and a lower support frame is superior. The use of equal size propellant tanks does result in a substantially oversize fuel tank, but adjusting the supporting structure to accommodate different size tanks will, in the final analysis, result in considerably more complexity and just as much weight. The thermal requirement that the fluorine and helium tanks be thermally isolated can be adequately met without resorting to a spherical fluorine tank by utilizing non-metallic members in the supporting frame. Also, the requirement to separate the fluorine tank and the bottom surface of the spacecraft by at least nine inches can be met without severe consequences.

The use of foam insulation on the helium and fluorine tanks and multilayer insulation on the hydrazine tank presents no problems. In fact, the use of the multilayer insulation on the hydrazine tank materially reduces the problems of integrating a louver assembly or a radiator plate into the tank insulation.

One general requirement must be noted. The fluorine tank and helium tank (if helium is stored cold) must be shaded from continuous solar radiation up to the 350th day of flight. Therefore, either the spacecraft must provide the necessary shade or a shield must be provided. It is possible to allow the cold tanks to see the sun for limited periods. The maximum duration of such periods depends upon the mission time and the initial fluorine and helium temperature.

5.2 GROUNDHOLD THERMAL CONTROL

The same general approach used for the OF_2/B_2H_6 module groundhold thermal control may be used here. It is desirable to increase the thickness of the foam insulation because of the lower temperatures involved. Liquid nitrogen may be used as the coolant but it may be necessary to allow the tank pressure to rise slightly above atmospheric pressure. Helium which has been prechilled with liquid hydrogen may also be used as the coolant and would be capable of holding the fluorine at lower temperatures, but the supporting groundhold equipment would have to be very complicated.

5.3 FLIGHT THERMAL CONTROL

The temperature limits imposed by propulsion considerations can be met, but certain restrictions may be necessary. No form of heat venting apparatus such as a radiator plate or louver assembly need be provided if the operating temperature range of the RTG is sufficiently constrained. Should the RTG temperature be large (>50^{o}R), either a radiator or a louver must be provided.

The necessary heat for keeping the hydrazine tank warm may be obtained either from small radioisotope heaters attached directly to the tank or from the external RTG. If the external RTG is relied on for this purpose, the heat may be obtained via radiation through a louver provided the RTG is sufficiently close to the module. It may also be obtained by conduction through a solid conductor or by means of a heat pipe. The solid conductor has the disadvantage of high weight while the heat pipe is susceptible to failure due to long term corrosion.

Auxiliary equipment may be properly conditioned but the exact manner in which this should be accomplished is dependent upon the engine design and the temperature at which the helium is to be stored.

From the standpoint of thermal design and structural design, it is not difficult to store the helium at either the fluorine or hydrazine temperature. Preliminary indications are that a weight savings is realized if the helium is stored cold. Experience would indicate that it would be less difficult to store it warm.

5.4 EVALUATION OF THERMAL CONTROL CONCEPTS

In the light of these general comments the evaluation of the various concepts will now be made. Table 5-1 summarizes the evaluation results.

5.4.1 Groundhold Cooling

Both liquid nitrogen and prechilled helium fulfill the mandatory requirements. Both can maintain the fluid temperatures at their prescribed levels and neither will cause frost or water formation.

Weight. The liquid nitrogen and prechilled helium coolant systems will have identical flight weight.

Reliability. Unquestionably, the liquid nitrogen coolant system would be the most reliable. The prechilled helium system would require vacuum jacketed lines, pumps, hydrogen vent equipment and possibly even a liquid nitrogen back-up system. Experience has shown that operating at $100^{o}R$ or less is considerably more difficult than operating at $140^{o}R$.

Effectiveness. Because of the great cooling capacity of the pre-chilled helium system, that system is capable of compensating for errors of design within the system. It must be concluded that the liquid nitrogen system may be basically less effective in maintaining the necessary temperatures.

Adaptability. The prechilled helium system shows superior ability to adapt to variations in the thermal environment or temperature require-ments. It would be able to subcool the fluorine and helium so that cooling could be eliminated for longer durations or so that extended sun exposure could be tolerated after launch. The LN_2 system would not be able to show any such adaptability.

Testability. Because of the simplicity of the LN_2 system, as com-pared to the complexity of the prechilled helium system, the LN_2 system would be considerably easier to test. The testing of the prechilled helium system would be too difficult, however.

Cost. Obviously, the cost of a prechilled helium cooling system would be high compared to a LN_2 cooling system. The LN_2 system would cost approximately $2000 to $3000 (assuming a storage vessel is already available) whereas the prechilled helium system would probably cost in excess of $80,000.

5.4.2 Hydrazine Tank Heat Source

There are acutally four possible ways of obtaining heat for the hydrazine tank.

1) Small radioisotope heaters attached directly to the tank
2) Via radiation from the RTG
3) Via solid conductor from the RTG
4) Via a heat pipe from the RTG.

All of these concepts fulfill the mandatory thermal requirements. However, assuming that there are restrictions as to the level of hard radiation which may be tolerated near the module, only concepts 3 and 4 are acceptable. But note carefully, if such restrictions do not exist, concept 1 is superior in all aspects. Considering concepts 3 and 4, the following evaluation may be made.

Weight. The use of a heat pipe as opposed to a solid conductor will result in about a 17 pound weight savings.

Reliability. The solid conductor must be rated somewhat more reliable since it has no "working" parts. Ammonia heat pipes have been fabricated but to date no great backlog of data is available to show high reliability. The modes of failure have been established and basically they all reduce to material compatibility problems. In this respect, large quantities of data are available which show the long term compatiblity of the various materials involved, and the manner of fabrication required to assure no failure.

Effectiveness. Both the solid conductor and heat pipe will demonstrate the same capability of adapting to variations in the RTG temperature provided the heat pipe is properly designed. As long as the heat transfer capacity of the pipe is not exceeded, its thermal response curve (curve of heat transfer versus ΔT) will look like the thermal response curve of a solid conductor. If the heat pipe's capacity is exceeded, however, the heat transfer rate does not just drop slightly; rather it drops nearly to zero. Therefore, the heat pipe would have to be "overdesigned" to assure its ability to adapt to wide variation in the RTG temperature.

Testability. Both the heat pipe and solid conductor would be readily testable.

Cost. The heat pipe(s) required for transfering the heat to the hydrazine tank would cost less than $10,000 to design, manufacture, and qualify. The cost of a solid conductor would be negligible.

5.4.3 Hydrazine Tank Heat Venting

There are essentially three ways in which heat may be vented to space:

1. Through the insulation, that is, no special heat venting provision.

2. Radiator plate

3. Louver assembly.

All three concepts will satisfy the mandatory thermal requirements provided the RTG temperature variation is not large. For evaluation purposes, it will be assumed that the RTG temperature may vary $\pm 100^{o}R$ as indicated in the work statement. Under such an assumption concept 1 does not meet the mandatory requirement of maintaining the propellant temperature within limits. Therefore, the evaluation which follows considers concepts 2 and 3 only. But if the $\pm 100^{o}R$ RTG variation proves to be incorrect the various concepts must be reconsidered.

Weight. The louver assemblies would be substantially heavier than radiator plates. As a rule, louver assemblies will weight about 0.8 to 1.0 lb/sq ft. For the present module, radiator plates would show about a 3.9 lb weight savings.

Reliability. Radiator plates are essentially failure free since they are totally passive and have no working parts. On the assumption that second surface mirrors would be used on the radiator plate, the only possible area of failure is the separation of mirrors from the plate. In contrast, louver assemblies do have working parts which are susceptible to breakage. However, experience has shown that louver assemblies are highly reliable. To date, TRW has had no louver failures.

Effectiveness. Considering design and manufacturing errors which might normally occur, the radiator plate would have to be classified as marginal in its effectiveness in keeping the hydrazine within temperature limits for the allowable RTG temperature range. The louvers show no such limitation.

Adaptability. The radiator plate is incapable of adapting to off-normal operation of the RTG or minor variations in the planned mission. The louvers, because of their semi-passive nature, can respond to wide fluctuations in the RTG temperature or other thermal environments.

Testability. A radiator controlled thermal system or a louver controlled thermal system can be easily tested in a thermal/vacuum chamber. It is somewhat more difficult to analyze the data from a louver controlled thermal system test because of the varying nature of the louver. However, the adequacy of either concept can readily be demonstrated by test.

Cost. The cost of louvers, percentage-wise, is considerably higher than radiators, however, the cost of louvers is not prohibitive to their use.

5.4.4 Helium Tank Thermal Control

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 provisos may be stated.

1. If the helium is to be 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 not wise since very little experience is available concerning cryogenic heat pipes. The weight penalty might be large (5 to 10 pounds) but that would be offset by the high reliability of a solid conductor.

2. If the helium is to be stored at the hydrazine temperature, a heat pipe between the hydrazine tank and helium tank should 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.

5.5 RECOMMENDED SYSTEM

Based upon the evaluation as summarized in Table 5-1, it is recommended that:

1. A LN_2 groundhold coolant system be used and that the fluorine tank be allowed to pressurize itself to 25 psia.

2. A heat pipe(s) should be used to conduct heat from the RTG to the hydrazine tank.

3. Louvers should be used to vent heat to space when necessary.

Repeating again, these recommendations are based upon the assumptions that the RTG temperature variation might be $\pm 100^{\circ}R$ and that hard radiation must be kept away from the module. Should either of these assumptions prove incorrect, a re-evaluation must be made.

Table 5-1. Relative Evaluation Factors for Thermal Control Concepts

	Weight	Reliability	Effectiveness	Adaptability	Testability	Cost	Total Rating	Recommended Concept
LN_2 Coolant System	0	2	5	5	1	0	13	X
Prechilled Helium Coolant System	0	10	1	1	5	5	22	
Small Radioisotope Heaters	Unacceptable							
Radiation From RTG	Unacceptable							
Solid Conductor from RTG	15	0	0	0	0	0	15	
Heat Pipe from RTG	2	6	0	1	0	2	11	X
Hydrazine Tank Control - Insulation	Unacceptable							
Hydrazine Tank Control - Radiator	1	0	10	10	0	0	21	
Hydrazine Tank Control - Louvers	3	5	1	1	2	5	17	X

6. AREAS OF FURTHER STUDY

The results of the work reported in this report have shown that within certain limitations, a thermal control system may be devised which will provide adequate control of the module temperatures during the entire Jupiter mission. However, before a complete thermal control system can be specified, it is necessary that additional effort be concentrated in three areas:

1. A study should be made to establish the optimum helium storage temperature (warm or cold).
2. A study of the engine component temperature control should be made which considers the detail design of the engine.
3. A study of zero gravity heat transfer phenomena inside the fluorine tank during times of solar heating should be made.

6.1 HELIUM STORAGE TEMPERATURE

As indicated in the previous sections, it is possible to store the helium in either a warm or cold state. Therefore, this 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 would indicate that the 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.

This study should have as its specific objectives the determination of 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.

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. For this reason it is suggested that all testing be carefully planned with a clear understanding of this limitation in mind. It is also suggested that cryogenic propellant/helium pressurant data from past flight programs (Atlas, Centaur, Saturn) be carefully screened for applicable information.

6.2 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. But this requirement 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 study program should be initiated which has as its specific objectives the determination of the above listed items. It should be clearly noted that for this study to be worthwhile, the specific hardware configuration of the engine must be used in the study. The greatest return from this study would be realized if it were carried on concurrently with the engine design. Not using the design of the engine in this analysis will probably result in unreal and unreliable answers. Performing this work after the engine has been designed will probably result in the discovery of thermal constraints which will necessitate a redesign of portions of the engine.

These problems are by no means insurmountable but they are ones which must be given attention. It would be wise to give the attention now.

6.3 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 specific objective of the study should be to establish if the zero-gravity heat transfer problems which have been postulated do indeed exist and, if so, how they may be circumvented.

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

6.4 RTG EFFECTS

Another area which should be studied, though not listed above because it is not directly a part of the thermal control system, is the effects of draining heat from the RTG. For the most part, the heat drain from the RTG will be constant (±10 percent), but it may be necessary to carefully design the manner in which this energy is drained from the RTG or it might cause unacceptable thermal gradients. Of course, if the RTG is not used as the source of heat for the hydrazine tank, this problem disappears.

REFERENCES

1. R. E. DeLand, et al., "Summary Report, Task VI Space Storable Propellant Module Environmental Control Technology," TRW Report 14051-6007-T0-00, 17 July 1970.

2. O. O. Haroldsen and R. N. Porter, "Summary Report, Task IV Space Storable Propellant Module Environmental Control Technology," TRW Report 14051-6005-T0-00, 16 November 1970.

3. P. E. Glaser, et al., "Thermal Insulation Systems," NASA Report SP-5027, 1967.

4. M. Epstein, "Prediction of Liquid Hydrogen and Oxygen Pressurant Requirements," Advances in Cryogenic Engineering, vol. 10, Plenum, New York, 1964.

5. O. O. Haroldsen, "Propulsion Module Thermal Control System Design Evaluation," TRW IOC 69-8526.13g-48, 5 December 1969.

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.

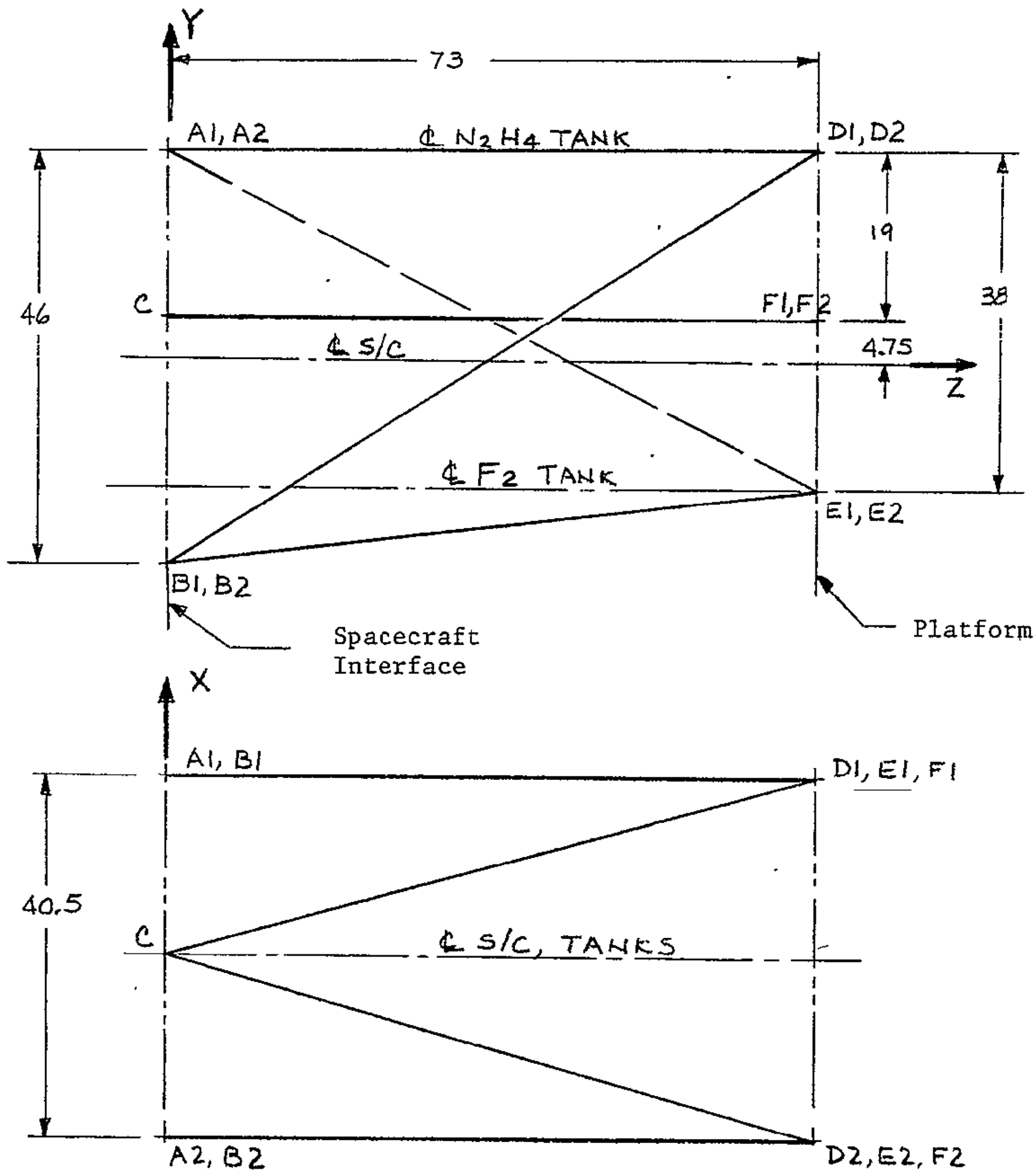


Figure A-1. Upper Truss Structure

Table A-1. Upper Truss Structure

Ultimate Loads

MEMBER	CONDITION 1	2	3a	3b	3c	3d	4a	4b	4c	4d
	$n_z = -7.5$	$n_z = 10$	$n_x = 3.75$		$n_x = -3.75$		$n_y = 3.75$		$n_y = -3.75$	
			$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$
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

Member Sizes and Weights

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.

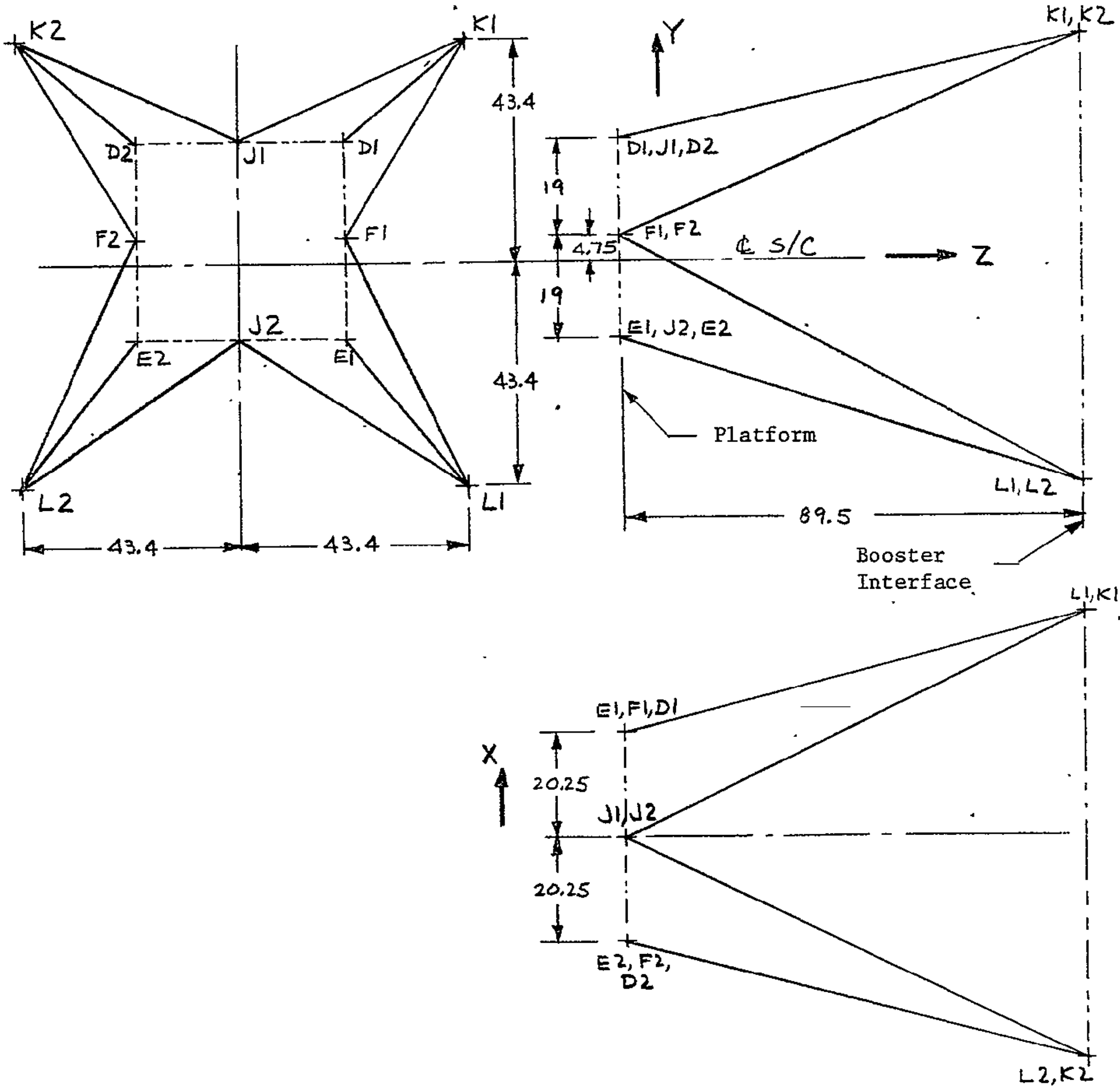


Figure A-2. Lower Truss Structure

Table A-2. Lower Truss Structure

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 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

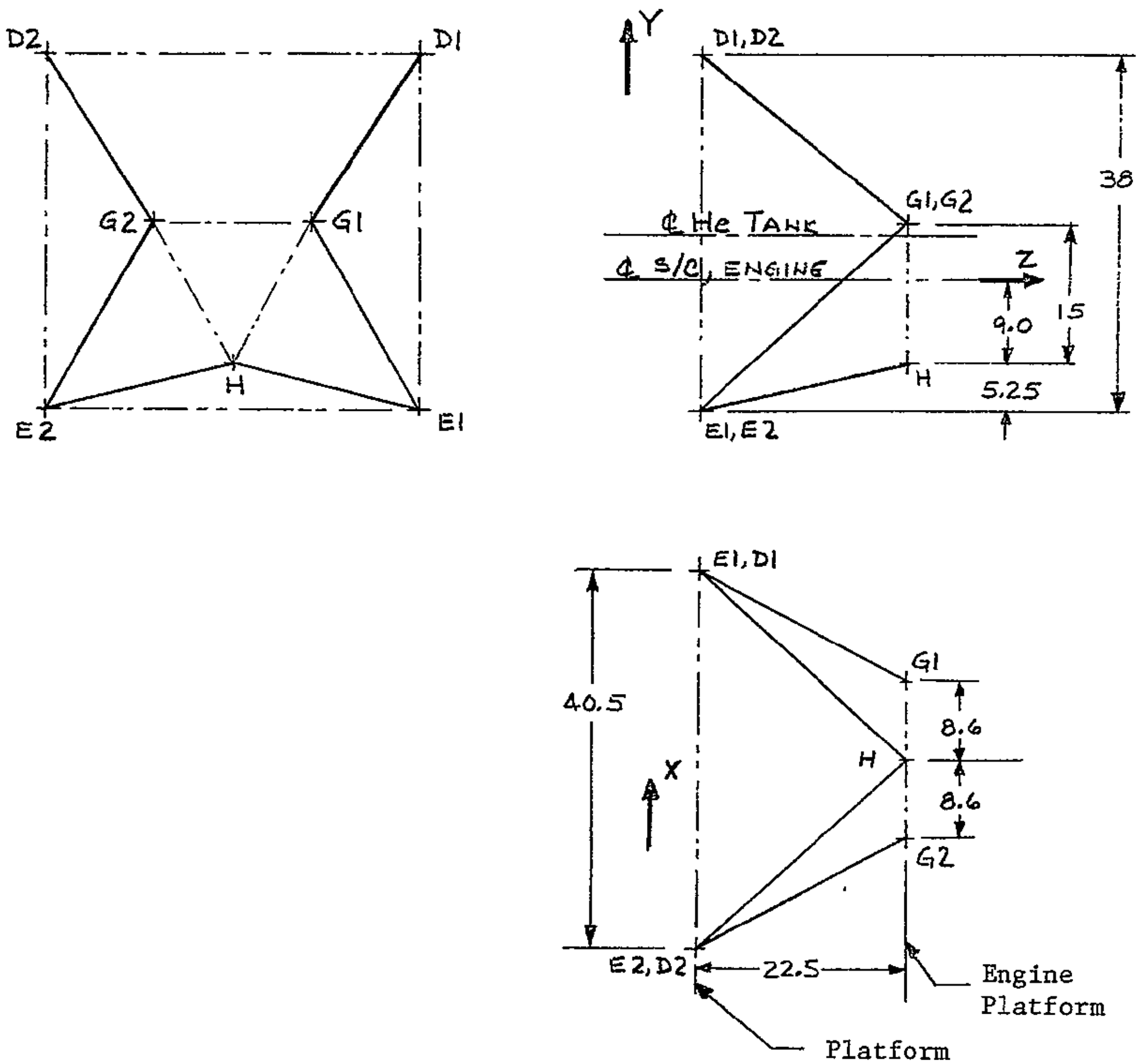


Figure A-3. Engine Support Truss

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

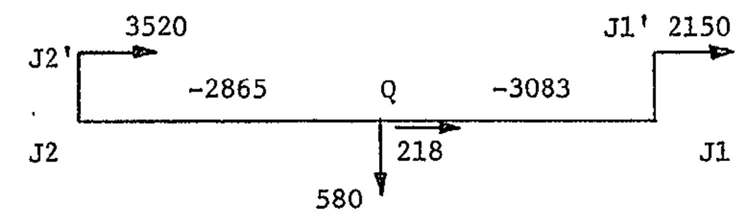
Ultimate Loads

MEMBER	CONDITION 1	2	3a	3b	3c	3d	4a	4b	4c	4d
			$n_x = 3.75$		$n_x = -3.75$		$n_y = 3.75$		$n_y = -3.75$	
	$n_z = -7.5$	$n_z = 10$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$	$n_z = 5$	$n_z = -3.75$
D1 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

J1 J2

$n_z = 5$

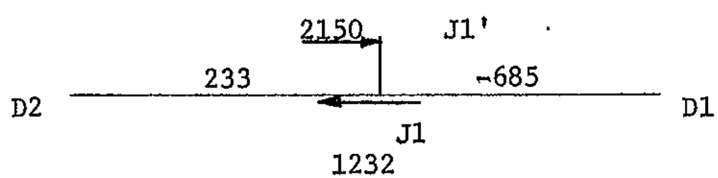
$n_y = 3.75$



D2 D1

$n_z = 5$

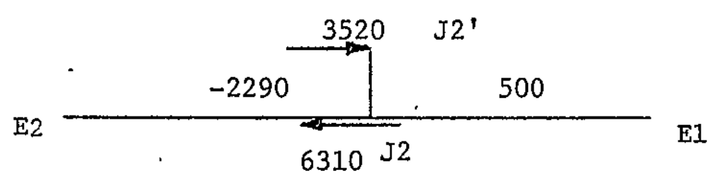
$n_x = 3.75$



E2 E1

$n_z = 5$

$n_x = 3.75$



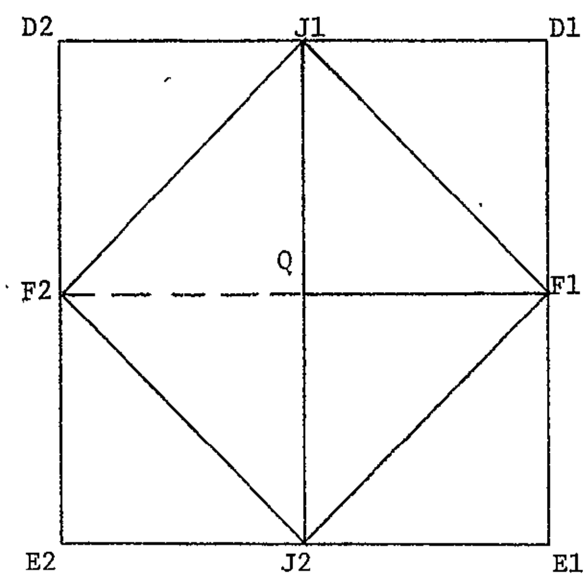


Table A-5. Platform - Member Sizes and Weights

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

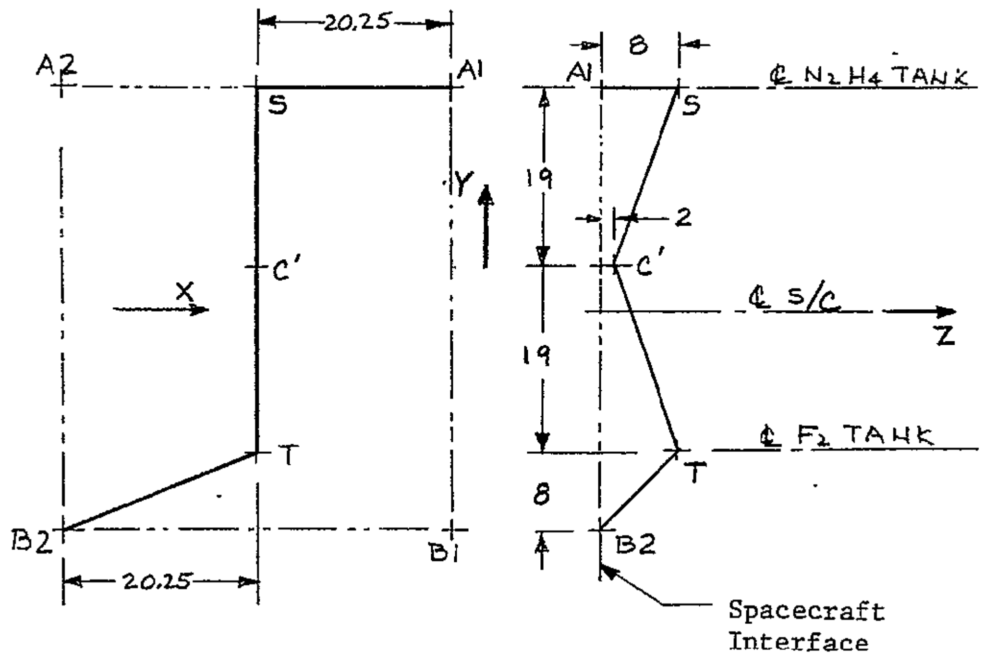
Edge and Cross Members

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

Configuration



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

Table A-7. Summary of Estimated Subsystem Weights

Tankage	1-Helium Tank @ 74.3 lb. ea.	74.30 lb
	2-Propellant Tanks @ 54.13 lb. ea.	108.26 lb
	1-Propellant Acquisition Device	2.0 lb
		184.56 lb
Liquid Circuit		
	2-Fill Valves @ 1 lb. ea.	2.0 lb
	2-Isolation Valves @ 4 lb. ea.	8.0 lb
	2-Filters @ 1 lb. ea.	2.0 lb
	2-Relief Modules @ 1.2 lb. ea.	2.4 lb
	2-Check Valves @ 1.0 lb ea.	2.0 lb
		16.4 lb
Gas Circuit		
	1-Fill Valve @ 1 lb. ea.	1.0 lb
	4-PR. Explosive Valves @ 3 lb. ea.	12.0 lb
	1-Filter @ 1 lb. ea.	1.0 lb
	1-Regulator @ 2 lb. ea.	2.0 lb
	1-Check Valve @ 0.5 lb ea.	0.5 lb
	2-Relief Modules (Disc plus valve) @ 1 lb. ea.	2.0 lb
	2-Pressurization & Vent Valves @ 2 lb. ea.	4.0 lb
	2-Solenoid Valves @ 2 lb. ea.	4.0 lb
		26.5 lb
Thrust Chamber Assembly		
	2-Propellant Valves @ 5.0 lb. ea.	10.0 lb
	2-Orifice Assys, W/Flanges @ 0.5 lb. ea.	1.0 lb
	2-Bleed Valves @ 1 lb. ea.	2.0 lb
	1-Thrust Chamber W/Gimbal Mounts	43.0 lb
	2-Gimbal Actuators @ 2.25 lb. ea.	4.5 lb
		60.5 lb
Fluids		
	Oxidizer (F_2)	1808.0 lb
	Fuel (N_2H_4)	1085.0 lb
	Helium (He)	36.0 lb
		2929.0 lb

Table A-7. Summary of Estimated Subsystem Weights (Continued)

Structure-Above Separation Plane		
	Upper Truss Members	22.02 lb
	Tank Upper Support Member	1.37 lb
	Spacecraft Attachment Fittings	4.25 lb
	Platform Members (Frame)	8.63 lb
	Platform Fittings	5.25 lb
	Engine Support Truss Members	1.56 lb
	Engine Support Platform	2.75 lb
	Tank End Fittings	2.70 lb
	Valve Assembly Brackets	6.76 lb
	Meteoroid Shields	14.32 lb
		69.61 lb
Structure-Below Separation Plane		
	Truss Members	45.96 lb
	Fittings (Separation)	2.50 lb
	Stabilizing Frame	1.00 lb
		49.46 lb
Miscellaneous		
	Lines and Fittings	20.0 lb
	Instrumentation	4.0 lb
	Command and Squib Harness	8.0 lb
	Contingency	16.0 lb
		48.0 lb
Insulation		
	Aluminized Mylar (N_2H_4 Tank)	.91 lb
	Foam (F_2 Tank)	15.40 lb
	Foam (He Tank)	6.72 lb
	Foam (Beams)	1.64 lb
	Aluminized Mylar & Support (Radiation Barrier)	2.00 lb
	Louvers (N_2H_4 Tank)	2.25 lb
	Cooling Coil Assembly (F_2 Tank)	1.50 lb
		30.42 lb
Total		
	Stage weight	3365.0 lb
	Total weight (including structure below separation plane)	3414.5 lb

APPENDIX B

THERMAL COMPUTER PROGRAMS

Three computer programs were formulated during Task VII. Two were for the module as shown in drawing SK 406961 given in Section 2. The only difference in these two programs was that one included conductors to account for atmospheric conviction and groundhold coolant flow during the groundhold phase. The third model was for the engine as shown in Figure 4-17.

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

All computer programs were in the standard SINDA format.

Tables B-1 through B-4 lists the nodes of these models and the major conductances and radiation conductors. It should be noted that in some cases the various conductors were varied during the analysis since they were actually the subject of the investigation. The values listed in Tables B-2 and B-4 are merely the nominal values of such conductors.

Table B-1. Nodal Arrangement of Module Computer Model

Node Number	Description
1	F_2 Tank
2	N_2H_4 Tank
3	Helium Tank
4	Thrust Cone
5	Combustion Chamber
6	Propellant Valves
7	F_2 Tank Insulation, top (-Z)
8	F_2 Tank Insulation, side (+Y)
9	F_2 Tank Insulation, size (+X)
10	F_2 Tank Insulation, side (-Y)
11	F_2 Tank Insulation, side (-X)
12	F_2 Tank Insulation, bottom (+Z)
13	F_2 Tank Insulation, removable
14	N_2H_4 Tank Insulation, top (-Z)
15	N_2H_4 Tank Insulation, side (+Y)
16	N_2H_4 Tank Insulation, side (+X)
17	N_2H_4 Tank Insulation, side (+Y)
18	N_2H_4 Tank Insulation, side (-X)
19	N_2H_4 Tank Insulation, bottom (+Z)
20	N_2H_4 Tank Louver

Table B-1. Nodal Arrangement of Module Computer Model (Continued)

Node Number	Description
21	FWD Meteoroid Shield, +Y
22	FWD Meteoroid Shield, -Y
23	Aft Meteoroid Shield
24	N_2H_4 Thermal Shield, N_2H_4 side
25	Helium Thermal Shield
26	Helium Insulation, -Z
27	Helium Insulation, +Z
28	Frame, +Y
29	Frame, -Y
30	Spacecraft Insulation, -Y
31	Spacecraft Insulation, +Y
32	RTG
33	Space
34	LN_2 (or Helium) Coolant
35	Spacecraft
36	N_2H_4 Thermal Shield, F_2 side
37	F_2 Tank Cooling Coil
38	Helium Tank Cooling Coil

Table B-2. Controlling Conductors and Paths of Module Model

Conductor Values (Btu/hr-°F)	Node-To-Node
0.003	1-2
0..003	2-3
0.11	1-7
0.22	1-8
0.45	1-9
0.45	1-10
0.45	1-11
0.45	1-12
0.09	2-14
0.07	2-15
0.09	2-16
0.09	2-17
0.09	2-18
0.09	0-19
0.4	2-32
0.06	31-35
0.16	30-35
Radiation Values ($\mathcal{F}A\epsilon_1\epsilon_2$)	**Node-To-Node**
Variable	2-20
3.11	4-33
2.0	7-31
1.9	7-33
6.0	9-33
7.0	10-33
6.0	11-33
3.2	14-31
0.004	14-32
1.9	14-33
0.07	15-32
4.8	15-33

Table B-2. Controlling Conductors and Paths of Module Model (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

Table B-4. Controlling Conductors and Baths of Engine Model

Conductor Value (Btu/hr-^{o}F)	Node-To-Node
0.25	1-2
6.0	2-3
0.23	3-9
0.032	4-5
0.032	4-6
0.15	4-9
0.035	7-9
0.01	4-8
0.1	3-13

Radiation Value ($\mathcal{F} A \epsilon_1 \epsilon_2$)	Node-To-Node
3.0	1-12
0.04	2-16
2.3	2-12
1.1	3-12
0.14	3-16
0.003	5-7
0.003	5-9
0.036	5-11
0.25	6-12
0.045	6-16
0.009	7-9
0.003	7-11
0.03	7-12
0.07	9-11
0.19	13-11
0.4	13-12
0.24	14-12
0.16	12-16

APPENDIX C

PROPULSION SYSTEM ANALYSIS

The calculation involved in sizing the helium tank is a trial-and-error process. The simplified approach used to obtain the results mentioned above proceeds as follows. First, the ullage volumes to be filled with gas are calculated for firing.

$$V_{u_{(n)}} = V_T - \frac{M_{p_{(n)}}}{\rho_p}$$

Where M_p is the mass of propellant which remains in the tank at the end of the particular firing (n).

Next, a collapse factor (C) is calculated (Reference 4) for the oxidizer ullage; this requires an estimate to be made of the mean inlet temperature ($\bar{T}$). An initial mass of helium (M_o) is estimated. Then, for the first firing (n = 1), with no helium in the ullage, the required masses of helium to pressurize a tank are:

$$\Delta M_{HE} = \frac{(P_R - P_V)\ V_U C}{Z\ R\ \bar{T}} \approx \frac{(P_R - P_V)\ V_U C}{Z\ R\ (\frac{T_o + T_f}{2})}$$

There is a ΔM_{HE} for each tank, of course.

Assuming adiabatic expansion in the helium tank,

$$T_f = T_o \left[\frac{P_f}{P_o}\right]^{\frac{\gamma - 1}{\gamma}}$$

and substituting for P_f,

$$P_f = P_o \left(\frac{M_f}{M_o}\right) \left(\frac{Z_f T_f}{Z_o T_o}\right) = P_o \left[M_o - \frac{(\Delta M_{HEOX} + \Delta M_{HEF})}{M_o}\right] \left(\frac{Z_f T}{Z_o T_o}\right)$$

a final expression is obtained, in terms of "givens," an estimated Z_f, which is solved by trial and error for T_f.

$$T_f = T_o \left\{ \frac{\left[M_o - \overbrace{\frac{(P_R - P_V)\, V_U C}{Z\, R \frac{(T_o + T_f)}{2}}}^{\text{For Ox.}} - \overbrace{\frac{(P_R - P_V)\, V_U C}{Z\, R \frac{(T_o + T_f)}{2}}}^{\text{For Fuel}} \right] Z_f T_f}{Z_o M_o T_o} \right\}^{\frac{\gamma - 1}{\gamma}}$$

After solving for the ΔM_{HE} values for the first firing, the helium consumption for succeeding firings are calculated in the same manner except that the initial helium mass is updated ($M_{O(n)} = M_o - \sum_{0}^{n-1} \Delta M_{HE}$) and the helium already in each ullage at the beginning of the firing must be accounted for as a pressure, P_{HE}.

$$P_{HE} = \frac{(M_{HEU})\, Z\, R\, T_O}{V_{U_n}}$$

Where T is the liquid equilibrium bulk temperature and M_{HEU} is the accumulated total of all helium in the ullage.

Then, $$\Delta M_{HE_n} = \frac{(P_R - P_V - P_{HE})\, V_{U_n} C}{Z\, R\, T}$$

Note that if some helium dissolves into the propellants, it is subtracted from M_{HEU} before P_{HE} is calculated.

If the final helium residual pressure in the helium tank after the last firing is not within the allowable range, then a new calculation is made with a new estimate for M_o.

The final pressure is approximately,

$$P_F = P_o \left(\frac{T_f}{T_o}\right) \left(\frac{M_f}{M_o}\right) \left(\frac{Z_f}{Z_o}\right).$$

where all zero subscripts refer to concurrent conditions sometime before the first firing.

Symbols

C = Collapse factor (w/w^o) or ratio of mass of pressurant actually required to mass required if the ullage temperature were equal to the mean pressurant inlet temperature, dimensionless.

M = Mass, lb_m

P = Pressure, lb_f/ft^2

R = Gas constant (1545/mol. wt.), $ft\text{-}lb_f/lb_m/^oR$

T = ^oR

V = Volume, ft^3

Z = Compressibility factor, dimensionless

ρ = Density, lb_m/ft^3

Subscripts

F = Fuel

f = Final condition

HE = Helium

(n) = Firing number (n = current firing, n-1 = previous firing)

o = Initial condition

ox = Oxidizer

P = Propellant, fuel or oxidizer

R = Regulated (pressure level)

T = Tank

U = Ullage

V = Vapor (pressure level)

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27 January 1971

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