SUMMARY REPORT

TASK IV SPACE STORABLE PROPELLANT MODULE

ENVIRONMENTAL CONTROL TECHNOLOGY

November 16, 1970

Report No. 14051-6005-T0-00

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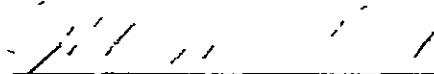
Contract No 7-750

Report No. 14051-6005-TO-00

by

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SUMMARY

The objective of Task IV was to make a detailed analysis of the B_2H_6/OF_2 propulsion module established by the work of Tasks I, II, and III The results of this analysis are given in this report

The analysis considered all phases of the mission, including ground-hold   The approach used was to first establish by analysis the thermal characteristics and temperature histories of the propulsion module during its life   This was followed by an analysis of the propulsion system which accounted for the thermal environment as established by the thermal analysis

This investigation shows that the thermal control system, as designed, maintains the required thermal environment throughout the mission   It also shows that considerable mission variations can be accommodated without ill effects   Where it was originally believed that exposure to the sun for only one or two hours could be accepted, as much as 20° off-pointing may be acceptable during the first days of the mission even if no special shielding is provided   After day 100 of the mission, continuous 90° off-pointing is acceptable.

The investigation shows that, though louvers aid in controlling the module temperature, they are not very effective   For this reason, unless a remotely controlled louver mounted on the RTG is acceptable, it is recommended that a totally passive thermal control system (replace the louvers with radiator plates) be utilized

Finally, it is concluded that, in the light of the accuracy of spacecraft thermal analyses, an experimental program should be intiated   The test program objectives should be to 1) prove the validity of the analysis during groundhold and flight, and 2) obtain sufficient information concerning the module thermal characteristics that the thermal control system may be "trimmed" for particular missions.

1 0 INTRODUCTION

This is the Task IV Summary Report of the Space Storable Propulsion Module Environmental Control Technology Project accomplished under Contract No NAS 7-750 Task IV had as its objective the thermal analysis of a propulsion module designed during Task III, Reference 1 Section 2 of this report describes in detail the propulsion module and Section 3 describes the mission profile and constraints.

The analyses were broken into two separate parts

- o The temperature history and thermal characteristics of the module were determined

- o The effects of the module thermal environment on the operation of the propulsion system were determined

Section 4 describes the computer programs used in this work and the basis of these programs Section 5 lists the results of the analysis and Section 6 gives the conclusions which can be drawn from the investigation.

2 0 MODULE DESCRIPTION

The basic module design is as shown in Drawing SK 406876, sheets 1 and 2, and the schematic for the propulsion system as given in Figure 2-1.

The propulsion system is comprised of three types of equipment tanks, engine and plumbing Two propellant tanks, one for each propellant, are used. Each tank is constructed of boron filament and lined with 0 010 inch aluminum. Each tank is 36 inches in diameter and utilizes hemispherical ends having an eccentricity of 0 784 The support attachments for each tank provide both axial and shear restraint at the bottom while only shear restraint at the top. To avoid the transmission of any moments into the tanks, universal joints are used at both ends

Internal to each propellant tank is a capillary-type propellant acquisition device (exact details to be established by JPL) and a ground-hold heat exchanger. The heat exchanger consists of a 1/2-inch diameter aluminum tube, 8 feet long, formed into a coil having a diameter of approximately 5 inches. Through this coil will pass LN_2 during ground-hold for propellant cooling purposes

The helium tank is also constructed of boron filament This tank is suspended at the top by an aluminum cross beam and is laterally stabilized at the bottom by boron filament tubes running from the tank to the engine support frame. A cooling coil, similar to those in the propellant tanks is installed in the helium tank

Other than a weight penalty, there is no technical reason why the boron filament tanks cannot be replaced with metal tanks. The thermal characteristics of the module would be essentially the same since the thermal conductivity of the walls of the tanks is very large compared to the insulation conductivity

The engine, as indicated in the drawing, is supported below the helium tank by a small triangular frame which is, in turn, suspended from the main frame by boron filament tubular members. For purposes of analysis, it was assumed that the engine is held in place by a classic gimbal, comprised of an outer ring fixed with respect to the spacecraft, a floating

inside ring and appropriate crossed axes which are perpendicular to each other and the engine centerline. The gimbal assembly is held by a thrust assembly as shown in Figure 2-2

At the direction of JPL, a film cooled columbium engine, having a 40·1 expansion ratio, has been assumed. Other engine parameters assumed are:

Mixture Ratio	3 0
Chamber Pressure	100 psia
I_{sp}	400 sec
Thrust	1000 lbs

Components are clustered in order to facilitate access and the fuel and oxidizer connections are separated for safety reasons   All gas circuitry is 1/4-inch except for the vent and relief lines which are 1/2-inch   All propellant circuitry is 3/4-inch except for the fill lines which are 1/2-inch   All tubing is assumed to be 300 series stainless steel with welded or brazed connections, except where bolted or flanged joints are necessary for assembly and/or test purposes

Where flexibility in the lines is necessary, short bellows or corrugated inconel hoses are used   All propellant lines have been routed to facilitate passivation and drainage   In addition, the following design provisions are shown on the drawings

o   A separate helium filter upstream of the regulator is utilized

o   The propellant valve solenoid pilot valve is made a part of the gimballing portion of the engine to improve response

o   The feedline isolation and relief return valving is positioned as close to the tank as is possible in order to minimize the length of liquid filled line

o   Injector purge solenoid and check valves are included

o   Gas supply lines to the pilot and purge valves are looped to provide flexibility.

There are three main control panels, one propellant tank pressurization control panel for each propellant and one helium control panel The helium control panel contains the helium squib valves, regulator,

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filter and fill valve   All of the fluid line disconnect fittings have
been mounted directly on one of these three panels   This places them all
approximately 30 inches inside the shroud line   If it becomes desirable
for reasons of shroud design or groundhold procedures   to mount these
fittings nearer the shroud, special mounting brackets will be required

A space truss structure is utilized to support the propulsion hard-
ware and also the spacecraft   The reasons for this choice are given in
Reference 2   The entire module and spacecraft are supported, when attached
to the boost vehicle, by 16 boron filament tubular struts having aluminum
fittings.   These struts remain with the boost vehicle upon separation
The separation fittings (pyrotechnic devices) are located on the main
platform

Both propellant tanks, the helium tank, and all propellant-filled
lines are insulated with 3/4-inch of two-pound density, closed-cell poly-
urethane foam [1]   This foam is extended to cover all metallic hardware
(frame, gas circuitry, etc) which contact the tanks to a distance of one
foot from the point of contact   Non-metallic members are similarly insu-
lated for a distance of 4 inches   In addition, the aluminum beam which
supports the helium tank is entirely insulated with foam

Aluminized Mylar blanket insulation is also utilized in several
places   A 10-layer blanket, attached to the lower surface of the electron-
ics package (spacecraft) which extends down on to the spacecraft support
struts, is required   Three other blankets of insulation are shown in
Drawing 406876, one around each of the propellant tank helium vent panels
and one around the helium control panel   The purpose of these blankets
is to maintain the temperature of the enclosed equipment near propellant
temperatures during flight and also at or near ambient temperature during
the groundhold phase   These blankets have flap-type doors to allow easy
access to the enclosed equipment

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[1] As specified by North American Rockwell Specification MBO 130-077, "Spec-
ification for Two-Pound Density Polyurethane Spray Foam "

In order to maintain most of the fluid circuitry lines at the pre-
scribed temperatures, the tubes in question are routed along the surface
of adjacent propellant or pressurant tank insulation and then overlayed
with 20 layers of aluminized Mylar. Where no such adjacent tank exists,
the tube is insulated around its entirety with the aluminized Mylar A
sketch of this arrangement for one of the propellant lines is given in
Figure 2-3. Since in flight, conductivity of the foam is much higher
than the conductivity of the aluminized Mylar, the tube adopts the temp-
erature of the tank in the region where the tube is adjacent to the tank.

In all cases, the aluminized Mylar is installed with the Mylar side
out In addition, a 10-layer blanket of aluminized Kapton attached to
the lower meteoroid shield on the helium tank side, is necessary Kapton
must be used in place of Mylar since, during engine operation, the allow-
able temperature for Mylar is exceeded

Two openings are left in the foam insulation on each propellant tank
to accommodate either radiator plates or louver assemblies If louvers
are used, they will be secured to the tanks by bonding the flanges of the
assemblies frame to the tank wall.

Louver assemblies consist of a frame within which is mounted bi-
metallic springs. A radiator plate covered with second-surface silvered
mirrors having an emittance of 0.8 is attached to the outside of the frame
Thus, when the insulation discussed above is removed, the mirrored surface
"sees" space and the aluminum louvers see the back of the radiator plate
on one side and the tank surface on the other.

Louver assemblies are designed such that the actuator springs sense
the tank surface temperature and thus actuate (rotate) the louvers open
or closed, depending on the tank temperature The effective emissivity
of the assemblies as a function of tank temperature are assumed to be as
given in Figure 2-4. This assumption is based on previous TRW spacecraft
designs (Pioneer, OGO).

If only radiator plates are used, they will merely consist of a
section of tank wall to which is bonded second-surface silvered mirrors

To prevent frost formation within and on the louvers after propellant
loading, a sophisticated insulation cover, which is removable after launch,

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is provided. This cover is shown on Drawing SK 406876, sheet 2 and is described in Reference 1   As mentioned in Reference 1, the major problem in providing a cover for louvers is to provide a way of allowing the louvers to "breathe" through the cover without allowing frost to accumulate on the louvers. If a radiator plate is used, the necessity of providing for the breathing dissapears and the cover then becomes a relatively simple removable section of insulation.

For purposes of this analysis, it has been assumed that the spacecraft, or else a platform mounted between the module and the spacecraft, will partially shield the module from solar heating   In a portion of the analysis, this shield was considered to be 10 feet in diameter (shroud diameter is 11 feet)   The effect of reducing this shield diameter is also considered.

The total weight of the propulsion module is approximately 3348 pounds. Approximately 27 pounds is attributible to the thermal control system A detailed weight breakdown of the module is given in Table 2-1

One general comment is appropriate at this point. Every effort was made to avoid peculiar design requirements or exotic materials   The objective was to produce the most simple design commensurate with the mission requirements   Thus, such things as special coatings were avoided where possible   It will become obvious through the discussion, that there are places (for example, the frame) where minor changes could be made to reduce temperatures. When mission requirements or module limitations are further defined, it may become necessary to institute some changes, but within the mission constraints as they are now specified, these changes are not necessary

Legend:

Symbol	Description
⊟←	Check Valve
○	N. O. Squib Valve
⊖	N. C. Squib Valve
⊡	Orifice
⊸⊳◁	Solenoid Valve
⊢⋈	Fill Valve with Cap
⋈	Regulator
○⋈	Pneu. Actuated Valve
⊓	Rupture Disc
⊳	Relief Valve

HE

B₂H₆

OF₂

← Liquid Filled Line

← Gas Filled Line
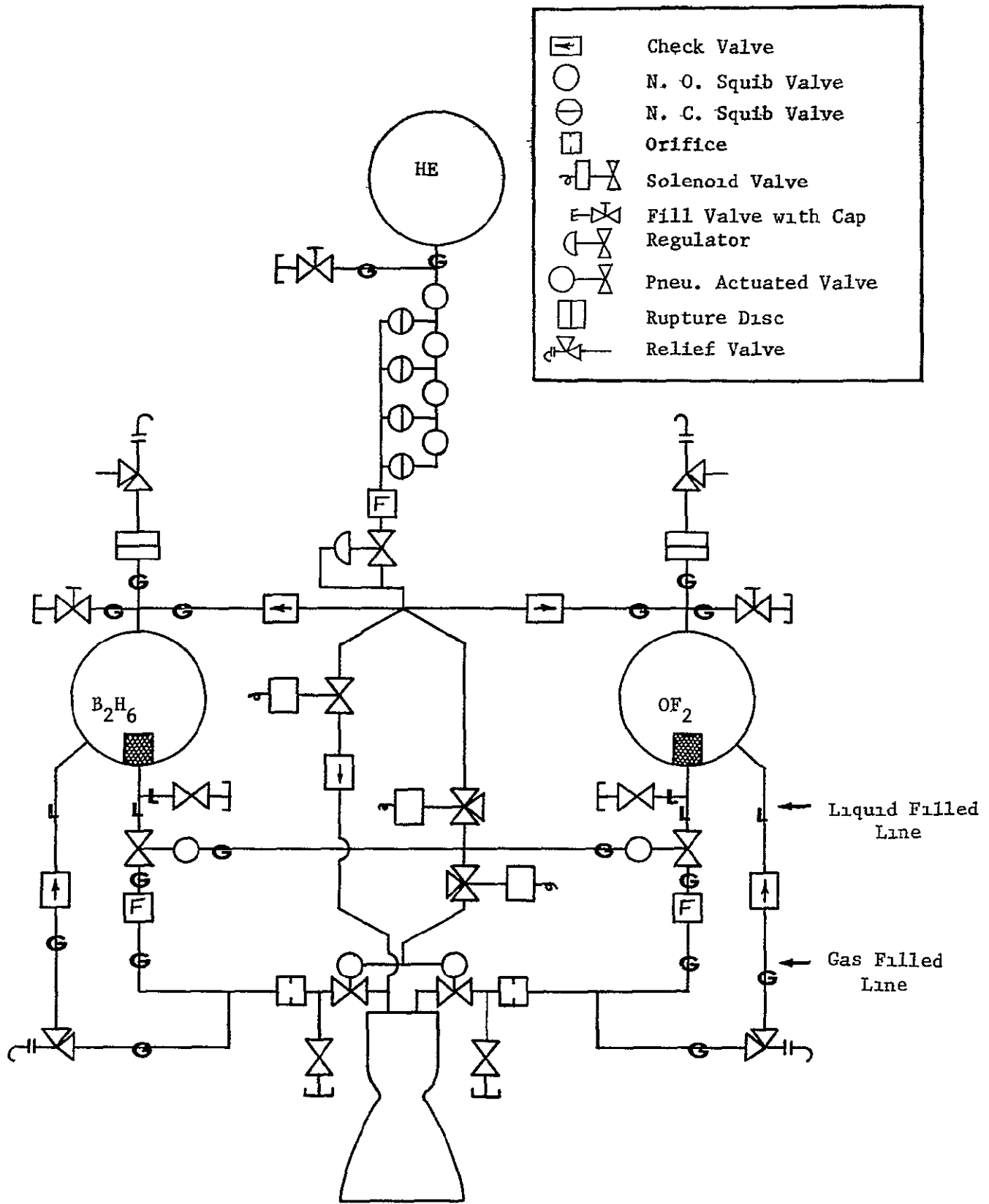
Figure 2-1   Schematic Diagram of OF_2/B_2H_6 System

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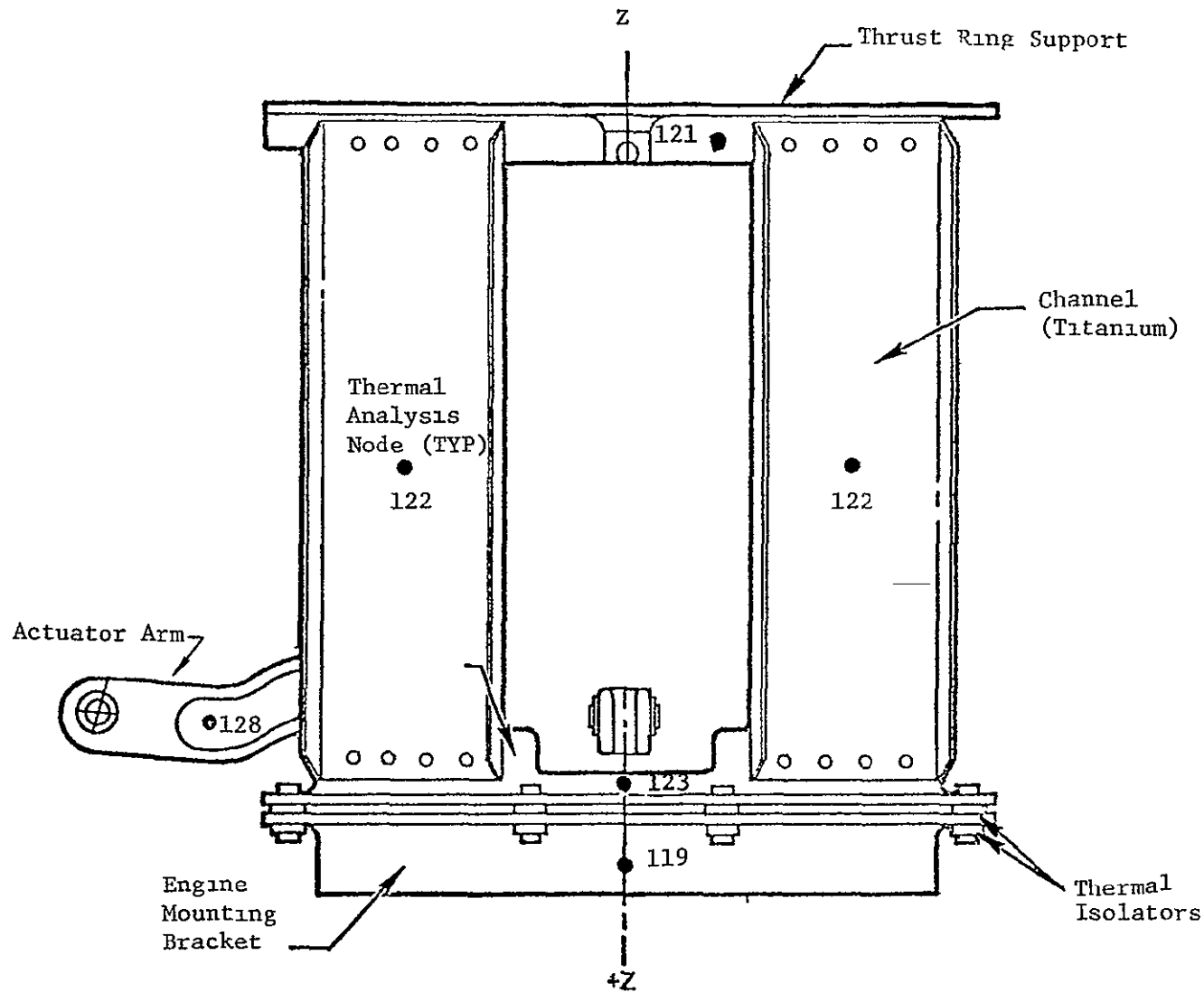
Figure 2-2   Engine Thrust Support Design
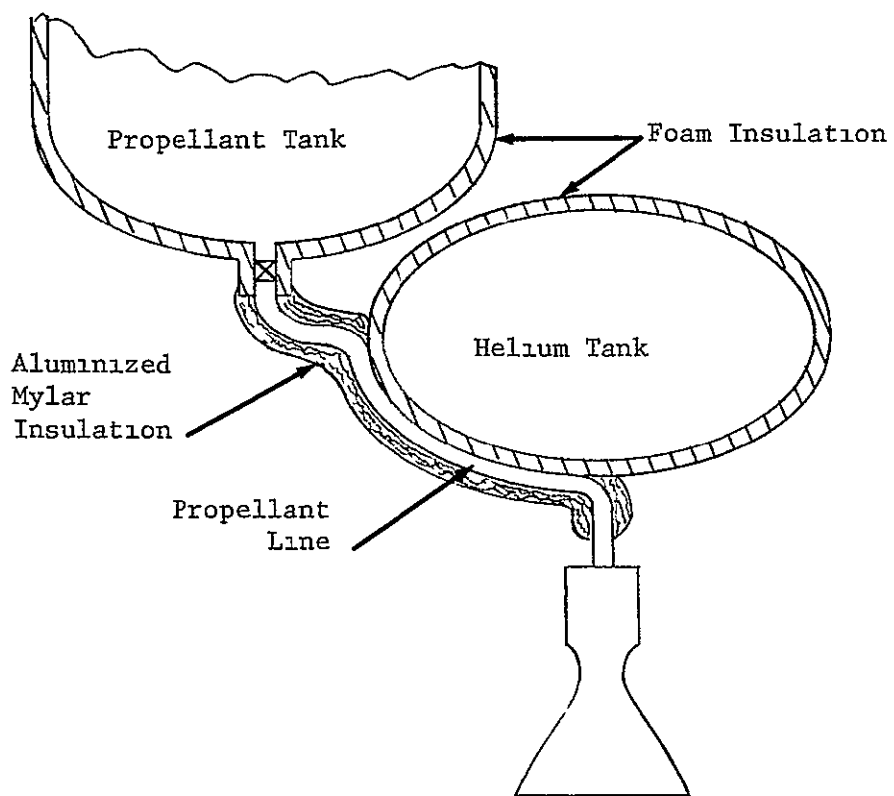
Figure 2-3 Schematic of Propellant Line Insulation

+X Louver Assemblies

Internal Effective Emissivity

Actuator Temperature °R

-X Louver Assemblies

Internal Effective Emissivity

Actuator Temperature °R
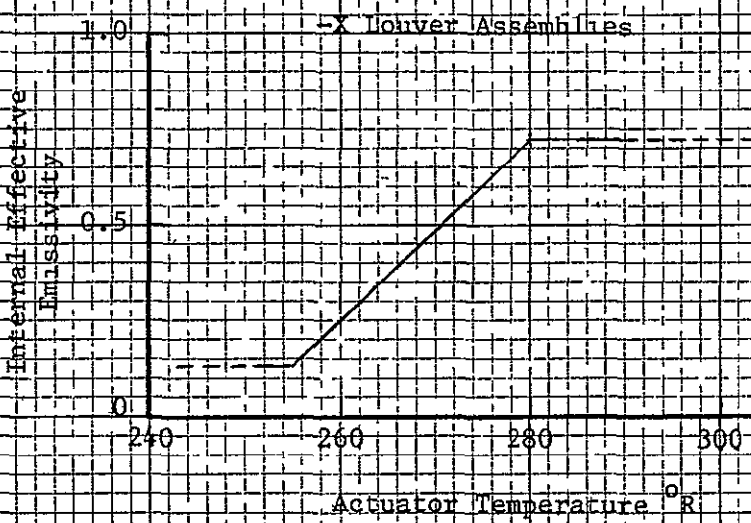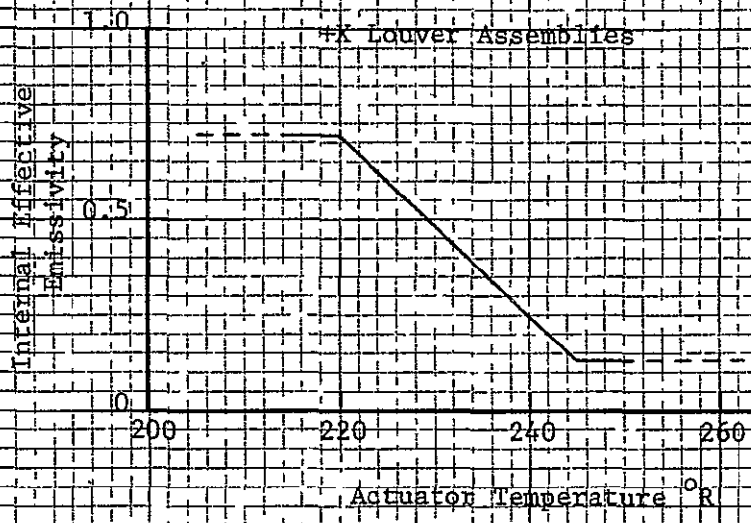
Figure 2-4 Louver Internal Effective Emissivity

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Table 2-1

Summary of Estimated Subsystem Weight

Tankage

1 - helium tank at 90 lb each		90 0 lb
2 - propellant tanks at 59 6 lb. each		119 2 lb
2 - propellant surface tension screens at 2 lb each		4 0 lb
	Total	213 2 lb

Liquid Circuits

2 - passivation valves at 1 lb each		2 0 lb
2 - fill valves at 1 lb each		2 0 lb
2 - isolation valves at 2 lb each		4 0 lb
2 - filters at 1 lb each		2 0 lb
2 - relief modules at 1 2 lb each		2 4 lb
2 - check valves at 1 lb each		2 0 lb
	Total	14 4 lb

Gas Circuit

1 - fill valve at 1 lb each		1 0 lb
4 pr - explosive valves at 3 lb each		12 0 lb
1 - filter at 1 lb each		1 0 lb
1 - regulator at 1 lb each		2 0 lb
2 - check valves at 0 5 lb each		1 0 lb
2 - relief modules (disc plus valve) at 1 lb each		2 0 lb
2 - pre-pressurization and vent valves at 1 lb each		2.0 lb
2 - solenoid valves at 2 lb each		4 0 lb
	Total	25 0 lb

Thrust Chamber Assembly

1 - thrust chamber w/gimbal mount at 45 5 lb		45 5 lb
2 - gimbal actuators at 2 25 lb each		4 5 lb
1 - propellant valve w/pilot solenoid valve at 7 lb each		7 0 lb
1 - purge check valve at 0 5 lb each		0 5 lb
2 - mixture ratio trim orifices and flanges at 0 5 lb each		1 0 lb
	Total	58.5 lb

Fluids

Oxidizer (OF_2)		2107 0 lb
Fuel (B_2H_6)		702 0 lb
Helium (He)		31 7 lb
	Total	2840 7 lb

Structure - Above Separation Plane

Upper truss members	20 67 lb
Tank upper support members	1 44 lb
Spacecraft attachment fittings	4 25 lb
Platform members	8 37 lb
Platform fittings	5 25 lb
Engine support truss members	1 68 lb
Engine support platform	2 87 lb
Tank end fittings	2 70 lb

Table 2-1   (Continued)

Valve assembly brackets			6.80 lb
Meteoroid shield			18.23 lb
	Total		72 26 lb

Structure - Below Separation Plane

Truss members			44 22 lb
Fittings (separation)			2 50 lb
Stabilizing frame			1 00 lb
	Total		47 72 lb

Miscellaneous

Lines and fittings			20 00 lb
Instrumentation			4 00 lb
Command and squib harness			8 00 lb
Contingency			16 00 lb
	Total		48 00

Thermal Control

Foam - 2 propellant tanks			16 20 lb
Foam - pressurant tank			2 66 lb
Cooling coils			1 00 lb
Louvers			6 0 lb
Mylar			2 0 lb
	Total		27 86 lb

GRAND TOTAL		3347 6

3.0 MISSION PROFILE

The general system requirements on thermal control is to maintain the propulsion module components within design temperature limits during all phases of the mission. For purposes of thermal analysis, the mission can be considered as composed of four distinct phases

1. Groundhold
2. Launch and Parking Orbit
3. Jupiter Transfer Phase
4. Jupiter Orbit

3 1 Groundhold

Groundhold is that period from initiation of passivation to launch which may last as much as one month   In addition to the passivation process which must be followed (Reference 6 specifies the recommended passivation procedure and equipment), the stage must be maintained at specified temperature limits during this phase. It is also necessary to prevent frost or water accumulation on any flight hardware. The three fluid control panels and the thrust chamber with its related valves and lines may remain at ambient temperature during the groundhold phase

3 2 Launch and Parking Orbit

The vehicle will be launched into a 100 nautical mile parking orbit by a Titan/Centaur/Kick stage. It is assumed that there are no restrictions as to the time of launch and that the protective shroud will be jettisoned at approximately 225,000 feet altitude. Maximum coast time in the parking orbit will be one hour. Sun exposure of the propulsion module during the launch and parking orbit is assumed to be random

3 3 Jupiter Transfer Phase

During this phase, the spacecraft will be oriented with the propulsion module nearly shaded except for one-hour reorientation maneuvers during each of the mid-course correction firings   It is expected that up to three such mid-course corrections firings may be required with an aggregate

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firing time of 44 seconds. For purposes of this thermal analysis, however, it is assumed that only one firing occuring on the seventh day which accomplishes a 100 meter/sec trajectory correction, will be required.

A single Jupiter orbit insertion firing of 533 seconds is assumed to occur 716 days after launch Prior to this firing, the spacecraft orientation for a normal mission is assumed as given in Figure 3-1. During this transit period, the solar intensity at the vehicle will be as given in Figure 3-2.

3 4 Jupiter Orbit

A final engine firing will occur at some arbitrary time after Jupiter encounter plus 23 days (773 days after launch) The purpose of this correction is to change the Jupiter orbit inclination angle This will require about 528 seconds of engine operation The initial Jupiter orbit will be 4 x 98 8 R inclined to the Jupiter equator at an angle of less than one degree The orbit period will be approximately 45 4 days

3 5 Module Operating Requirements

As indicated above, during groundhold it is only necessary to hold the propellant and pressurant temperatures within specified limits These limits are $250 \, {}^{+30}_{-40} \, {}^{\circ}R$ for the propellants and nominally $280 \, {}^{+0}_{-100} \, {}^{\circ}R$ for the helium It is not necessary to maintain any of the component temperatures at any given value solely for the sake of the component Only to the extent that such components temperatures affect the fluid temperatures within the tanks will such component temperatures be held near $250^{\circ}R$

During flight, it is also necessary to maintain the propellant temperatures within the specified limits. However, as shown in Reference 2, it is also necessary that, at engine operation, the two propellants be at nearly the same temperature and that various components be within specified limits Note carefully that these additional requirements exist only during and and just prior to engine firing. Table 3-1 lists the various component temperature requirements.

A comment is in order concerning the helium temperature requirements. Upon helium tanking, it is necessary to keep the temperature of the helium at or below $280^{\circ}R$ or else a severe over-pressure will be experienced.

However, there are distinct advantages if the helium temperature is maintained above propellant temperatures. First, it allows a more efficient utilization of the pressurant and second, it reduces the chances of over-pressurization because the warmer helium is subsequently cooled by the propellant when it enters the propellant tank. Thus, there is no apparent reason why the helium could not be allowed to exceed $280^{\circ}R$, provided maximum helium tank pressure is not exceeded. This means that the helium temperature after the first firing can exceed $280^{\circ}R$.
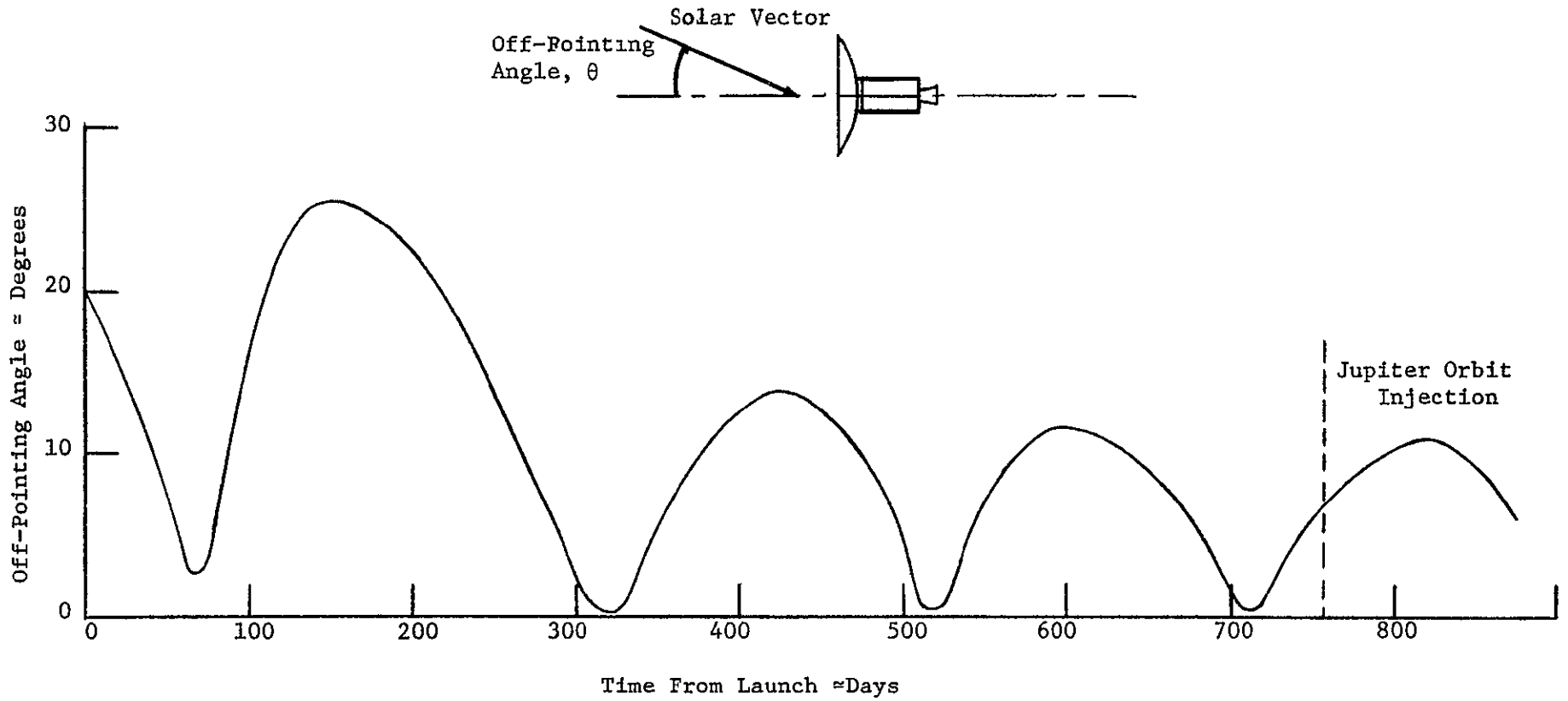
Figure 3-1 Off-Pointing Angle During Mission
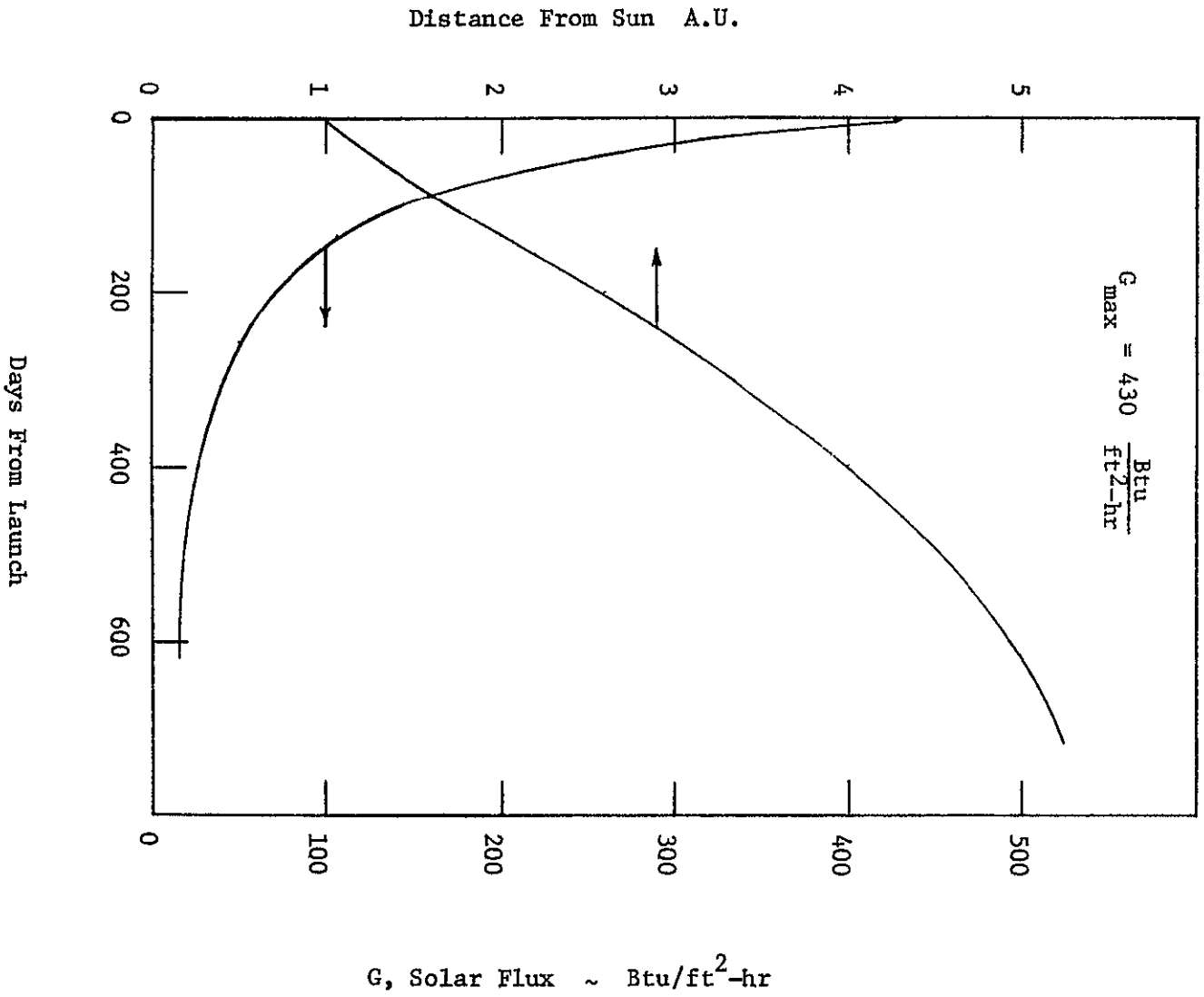
Figure 3-2 Solar Flux at Spacecraft

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Table 3-1 Recommended Component Temperature Ranges

Component	Operating Temperature (a)		Probable Effect of Operation at Temperatures above Recommended Maximum	Initiation of Operation of Temperatures above Maximum (b)
	Minimum, $^\circ$R	Maximum, $^\circ$R		
Helium Fill Valve	150	560	Essentially none	ok
Helium Explosive Valves	200	560	Essentially none	ok
Helium Filter	200	560	Negligible change in pore rating	ok
Regulator	200	300	Regulated pressure will be 5 to 10% low	avoid
Helium Check Valves	200	300	Seating pressure will be 5 to 10% low	ok
Tank Relief Modules	200	300	Cracking pressure will be 5 to 10% low	ok
Propellant Fill and Vent Valves	150	560	Essentially none	ok
Propellant Isolation Valves	210	280	Propellant heating and vaporization	avoid
Propellant Filters	200	290	Negligible change in pore rating, propellant heating	avoid
Passivation Valves	200	290	Propellant heating and vaporization	avoid
Mixture Ratio Trim Orifices	200	290	Negligible change in area, propellant heating	avoid
Engine Propellant Valve	200	290	Propellant heating and vaporization slower response	avoid
Injector	200	290	Propellant heating and vaporization	avoid
Feedline Relief Modules	200	290	Cracking pressure will be 5 to 10% low, propellant heating	avoid
Feedline Relief Check Valves	200	280	Propellant heating and vaporization	avoid
Actuation Pressure Valve	200	300	Slowed response – probably not critical	ok
Purge Actuation Valve	200	300	Slowed response – probably not critical	ok
Purge Check Valve	200	300	Reduced seating pressure (5 to 10%)	ok

(a) "Operating" implies preparation for and execution of a firing

(b) "Initiation" implies these temperatures will quickly decrease to the recommended maximum, or lower, immediately upon firing

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4.0 METHOD OF ANALYSIS

The analysis of the module is divided into two parts. First, a thermal analysis of the complete module is made for the various phases of the mission   These investigations provide module fluid and component temperatures at any time during the mission and they consider such variables as spacecraft position relative to the sun, RTG operation, insulation properties, weight of propellants on board , etc.

The second step of the overall analysis is to use the results of the thermal investigation, i.e., fluid and component temperatures, to determine the performance of the propulsion system during operation

4.1 Thermal Analysis Models

The propulsion module temperatures are calculated by representing the physical system by an equivalent electrical network which is solved by the CINDA computer program on the Univac 1108 computer. The equivalent electrical network is referred to as an analytic, mathematical, or computer model.

For purposes of this analysis, two computer models of the module were formulated· the basic model for analysis of all flight conditions and the revised model for analysis of groundhold conditions.

The basic computer model assumes the propulsion module and its environment may be represented by 163 uniform temperature elements, or nodes, connected by appropriate radiation and conduction resistances. Appendix A gives a detailed description of these nodes and the conduction and radiation heat transfer conductances between the various nodes. Thermal characteristics and property values of the various components were obtained from published data or from developmental tests conducted on previous programs at TRW Systems.

The revised model was obtained by making changes to the basic model which account for groundhold conditions. The specific changes were as follows:

  o Atmospheric convection heat transfer to all external nodes was provided.

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o   Background radiation level was changed from $0^{\circ}R$ to $525^{\circ}R$.

o   Nodes were added to account for the cooling coils inside the fluid tanks.

o   RTG in a stowed position.

o   Louver nodes were deleted to account for insulation over the top of the louvers.

o   All separation assembly interfaces were assumed to be held at $530^{\circ}R$.

As developed, the revised model had 168 nodes   In order to simplify the revised model, three assumptions were made   First, it was assumed that the external film coefficient used in determining the atmospheric convection heat transfer was independent of temperature. Ordinarily, this coefficient is porportional to $(\Delta T)^{0.25}$ in natural convection where ΔT is the temperature difference between the surface being heated or cooled and the ambient air   However, in this case, the effect of temperature variation upon the film coefficient will be small because the temperature difference between the air and module surfaces will remain small, 5° to $15^{\circ}F$, and fairly constant. More to the point, however, the coefficient is more dependent upon the velocity of the ambient air   For quiescent natural convection, the film coefficient may be as low as 0 25 Btu/hr-ft-$^{\circ}F$ But for high velocity gas, i.e., 35 fps, the coefficient may reach 10 Btu/hr-ft^{2}-$^{\circ}F$. Even such a wide variation in film coefficient has little effect upon the overall operation of the module during groundhold since the film coefficient presents very little resistance to heat flow compared to the foam insulation   Thus, for sake of convenience, a constant value of 1 Btu/hr-ft^{2}-$^{\circ}F$ was used in the computer program for most parts of the analysis. However, as will be seen, the effect of larger film coefficients was considered.

The second assumption made was that the tank walls are at the same temperature as the internal fluid which is in contact with the tank wall. Within the accuracy of calculations, this is sufficiently correct For normal conditions, the heat transfer rate through the walls will be shown to be of the order of 50 Btu/hr-ft^{2}-$^{\circ}F$ A hand calculation will show that, for the propellants under study, such a heat transfer rate

will result in less than a 3°F temperature drop across the film at the wall. This assumption is also the most conservative because it results in answers indicating higher heat transfer rates into the tank than will actually occur

This is not the same assumption that is made for the analysis of the operation of the propulsion system during engine operation. As will be shown below, during engine operation a thermal gradient at the wall is assumed.

The third assumption has to do with the film coefficients at the surfaces of the cooling coils As was pointed out in Reference 1, it has been found that the principles of physical similitude and the scaling laws are valid for cryogenic application, and that the applicable empirical equation for the external film coefficient is

$$h_o = 0.72 \frac{k}{D} \left[\frac{D^3 \rho^2 \beta g \Delta T}{\mu^2} \frac{c\mu}{k} \right]^{.25}$$

where h_o = external coefficient of heat transfer, $Btu/hr\text{-}ft^2\text{-}^{\circ}R$

    D = tube diameter, ft

    k = propellant thermal conductivity, $\dfrac{Btu\text{-}ft}{hr\text{-}ft^2\text{-}^{\circ}R}$

    ρ = propellant density, lb/ft^3

    β = propellant coefficient of volumetric expansion, $1/^{\circ}R$

    ΔT = temperature difference between tube and propellant $(T_B\text{-}T_t)$

    g = constant, 4.17×10^8 ft/hr^2

    μ = propellant viscosity, $lb/ft\text{-}hr$

    c = propellant specific heat at constant pressure, $Btu/lb\text{-}^{\circ}R$

Theoretically, several of the variables of this equation are temperature dependent and should be evaluated at some temperature between the propellant bulk temperature and the tube surface temperature A hand calculation will show that the maximum error induced by assuming temperature independent properties is 12% Since the properties of the propellants are not known to a high degree of accuracy, it is hard to justify accounting for temperature variations present in the analysis Even if the properties of the propellants were known accurately as a function of temperature, it is not logical to try to improve the accuracy of analysis by

including temperature dependent variables in the analysis because the greatest uncertainty in the analysis stems from unknowns about the convection currents within the propellants.

The film coefficient on the outside of the cooling coil is highly dependent upon the motion of the natural convection currents which will be set up in the tank However, these currents are influenced not only by the propellant properties but also by tank shape, tank size, and, in this case, most importantly by the shape and size of the propellant acquisition device. The equation listed for h_o is a generalized empirical equation obtained by correlating experimental results from tests in which cooling coils were immersed in containers of fluids. Though the equation is the best available, it obviously will not predict accurately ($\pm$ 20%) the film coefficient on the outside of the coils since it is taken from generalized data. It is also obvious that attempts to make it more accurate by using temperature dependent properties in its solution is not logical To obtain a more accurate knowledge of the heat transfer properties in the region of the coils, it is mandatory to conduct experimental investigations using the tank configuration of the flight module.

For these reasons, the approach taken in this analysis has been to use the equation listed and then show from the results that the design is sufficiently conservative to accommodate any errors introduced by the equation. Thus, h_o is made a function of only ΔT in the revised computer program The actual values of h_o as a function of ΔT were hand calculated and are inserted in the program as a table.

The equation chosen in Reference 1 for the coil internal coefficient is

$$h_i = 0.029 \frac{k}{D} \left[\frac{DV\rho}{\mu}\right]^{0\,8} \left[\frac{c\mu}{k}\right]^{0.4}$$

where h_i = internal coefficient of heat transfer, Btu/hr-ft^2-^{o}R

D = tube diameter, ft

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

V = liquid coolant velocity, ft/hr

ρ = liquid coolant density, lbs/ft^3

μ = liquid coolant viscosity, lbs/hr-ft

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

Although this equation contains variables which are influenced by a temperature change, no consideration needs to be given to this problem for the following reasons. As reported in Reference 3, the modified Colburn equation

$$\frac{hD}{k} = 0.023 \left[\frac{VD\rho}{\mu}\right]^{0\ 8} \left[\frac{c\mu}{k}\right]^{0.4}$$

correlates within 10% the data obtained from tests of heat transfer to a turbulent fluid flowing inside a tube provided no boiling takes place and the viscosity of the fluid is not greater than twice that of water This correlation was made with all properties evaluated at the initial bulk temperature except viscosity, which was evaluated throughout the cooling tube at the film temperature, T_f

$$T_f = T + 0\ 5\ (T_s - T)$$

where T_s is the local saturation temperature and T is the local bulk temperature However, data for the case of saturated liquids being boiled showed that for runs in which the Reynolds numbers exceed 65,000, the true mean coefficient, h_m, for the _entire_ _tube_ based on the mean temperature (saturation temperature minus bulk temperature) throughout the tube length, averaged 1.26 times the value predicted by the modified Colburn equation Thus, for boiling liquids inside tubes, the coefficient in the Colburn equation is changed to 0.029.

In the present investigation, the Reynolds Number of the LN_2 will be in excess of 65,000 except for the very low flows If, in addition, it is assumed that the LN_2 enters the tube at its saturation temperature, the mean temperature (the temperature at which all the properties should be evaluated) is identical to the bulk temperature and it is, for all practical purposes, constant throughout the tube

If the major method of heat transfer ceases to be heat transfer to a turbulent vapor, not only does the constant temperature assumption fail but the coefficient, 0.029, reverts back to 0 023. The data used in Reference 3 shows that the coefficient of 0 029 holds within 10% for a coolant quality at the tube exit of 7% or less vapor by weight provided the Reynolds number is in excess of 65,000. If the quality increases to 50% vapor at the exit, the accuracy of the coefficient drops to $\pm$ 25% As

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will be shown later, for all normal situations, the quality of the coolant in the case under study will be less than 30%, and in all probability will be less than 10%

Again, it should be clearly noted that it is not wise to attempt to infuse additional accuracy into the analysis by either using more accurate propellant property information or performing additional analytical work Experience in this field of heat transfer has shown that the additional accuracy can only be achieved by testing.

With these assumptions (constant temperature and sufficient flow), h_1 is reduced to a function of liquid velocity (coolant flow rate) In the program, h_1 is treated as a constant, the value of the constant being chosen according to the following hand calculated table

LN_2 (Coolant) Flow Rate, lb/hr	h_1 Btu/hr-ft^2-$^{\circ}$R
0	0
123	123
247	214
1317	2470

Due to a lack of adequate theory and, more particularly, due to the wide variety of conditions which may exist, both programs are undoubtedly somewhat in error in their representation of temperature gradients within the three fluid tanks Both computer models assume that the fluid and walls of each tank may be divided into three discrete nodes In the flight model, it is assumed that there is no convective heat transfer between the nodes (no convective fluid currents), only conductive heat transfer This assumption is the most conservative because it results in answers which indicate large thermal gradients During zero-g flight, it is generally assumed that only conduction heat transfer occurs, but it seems logical that if any spacecraft maneuvering occurs, the fluid will be "stirred" up It follows that the temperature of a given propellant entering the engine upon engine operation will be near the average of the three propellant nodes

There is the added problem of accounting for the position of the propellants within the tanks during zero-g flight A detailed study of this problem is beyond the scope of this present investigation, but the following

generalized comments are pertinent to this investigation.

The location and condition of the propellants within the tank during zero-g is dependent upon the propellant properties, tank shape and size (including propellant acquisition device), heat transfer rate, and locality of heat transfer It is conceptually possible for the vapor and liquid to be interspersed into a homogenous mixture completely filling the tank It is highly unlikely this will be the situation Rather, the vapor and liquid will usually be separated such that the liquid will form a continuous media with a pocket (or pockets) of gas Since the configuration is highly susceptable to change when the location of heat transfer to the tank changes, it is difficult to predict the fluid configuration with a high degree of accuracy. This is part of the problem here since, for a normal mission, the location and level of heat transfer varies appreciably with time.

It was, therefore, assumed for purposes of the flight thermal analysis that the fluid is evenly distributed around the surface of its tank and that all the ullage gas is contained in a single pocket of gas centrally located inside the liquid. This is the most conservative assumption for two reasons.

1. It results in the lowest calculated temperatures when no external heating exists and in the highest calculated temperatures when solar heating exists This is because no thermal resistance between the tank wall and propellant is considered

2. It results in the greatest calculated thermal gradient within the propellant This is because higher heat transfer rates occur when no internal vapor film exists adjacent to the wall and because the assumed configuration results in the maximum length and minimum area heat transfer path between the several propellant nodes

As was indicated above, during groundhold operations, convective currents within the propellants will be established but the exact mode of the currents is difficult if not impossible to predict But the fact that the currents do exist makes the assumption used in the flight analysis of only conductive heat transfer within a propellant tank invalid To directly account for convective currents inside a tank in a computer program is essentially impossible. Such a program would be correct only to the

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extent the convection currents are known Not only would the current paths have to be known, but the current velocities would have to be known. In addition, the correct nodal arrangement would be totally dependent upon the convection current paths and velocities. Thus, for even minor changes in the heat transfer pattern (for example, a shift in the external air currents), a change in nodal arrangement would have to be made

It is possible to indirectly account for the convective heat transfer by increasing the thermal conductivity of the propellant by an amount such that the apparent heat transfer rate indicated by conduction only would be the same as that which actually occurs with both conduction and convection. This approach results in a distorted picture of the temperature profiles within a tank. But, from an overall point of view, the error is negligible and therefore, this approach was adopted References 3 and 4 were used to estimate the apparent conductivity used in this analysis

Recounting the approximations and assumptions made in formulating the thermal computer models, it will be seen that errors introduced into the computations can be grouped into three classes

1. Errors due to the nodal configuration used to simulate the module

2   Errors due to unknowns in material properties and hardware construction (conductivities, reflectivities, interface thermal resistance, etc.)

3. Errors due to lack of information concerning the dynamic and heat transfer characteristics of fluids in zero gravity fields

Errors of the first type can be reduced to any desired level by using a sufficiently small nodal grid In reality, this reduces to a trade-off between the degree of accuracy necessary and the resulting complexity in the computer program which results Judgement derived from other programs (Pioneer, OGO, MSS) indicated that little would be gained by increasing the model complexity over that which exists in the formulated programs Though a detailed error analysis of this particular aspect of the programs was not made, analysis from other programs would indicated the error due to this source is about $\pm 5^{\circ}F$.

An analysis of the second type of errors was made for this program This was done by first estimating the extreme limits which could logically

exist for each of the conductances of the programs, and then determining the resulting temperature shift which would result if all these extremes were to occur simultaneously This analysis showed that if all the extremes "lined up" to shift the module temperature in one direction, the average module temperature would shift some 21°F If the possible variations in the conductances occurred in a 3σ random pattern, which is much more likely to be the case, the temperature error due to this source will be no more than $\pm 7^{\circ}$ during groundhold and $\pm 11^{\circ}$ during flight. This is comparable with experience from past TRW thermal control projects To obtain a better prediction of actual performance, actual tests must be run

Errors of the third type, as discussed above, are difficult to assess due to a lack of both theory and experimental data Without an indepth analysis, it is impossible to assign a value to this error. Based upon theory which is available and the particular nature of the module, we believe that temperature variations caused by these unknowns will not be major, less than 5°F for the heat transfer rates existing in this case

4 2 <u>Propulsion Analysis Models</u>

Two digital computer programs were written and used to calculate system pressures, flow rates, temperatures, etc , as a function of initial propellant and gas temperatures. The first program simulates firing conditions It is comprised of a set of equations in which time is one of the variables so that "time-advancing" calculations can be made of the progressive changes in temperatures and masses which will occur with time during firings. The second program simulates cruise periods It is a set of equations, which are unrelated to time, that calculate the equilibrium propellant tank pressure for any combination of physically compatible values of temperature, propellant mass, and helium mass.

It was assumed that the propellant and gas circuitry would be properly calibrated to deliver nominal performance at 250°R (i.e., mixture ratio would be 3 0 and chamber pressure would be 100 psia). Calculations were made to determine the mixture ratio and chamber pressure excursions

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if the propellants were conditioned to temperatures other than the nominal design point temperature by the thermal control system Also, calculations were made of the ullage gas temperature, helium sphere temperature, and pressure histories during the nominal temperature (250°R) mission firings and post-firing cruise periods

Firing durations were calculated for the nominal mission based on a delivered specific impulse of 400 lb_f-sec/lb_m and a spacecraft mass of 4400 lbs at launch. Velocity increments (ΔV's) of 100 m/s (meters per second), 1460 m/s and 2320 m/s were used for the midcourse, orbit insertion, and orbit inclination maneuvers, respectively The midcourse ΔV represents the maximum total for three firings but these were lumped together since one long firing is a "worst case" compared to three shorter firings Burn times for these firings were calculated according to the equation

$$\theta_b = \frac{M_o}{\dot{M}_p} \left[1 - \frac{1}{\ln^{-1} \frac{\Delta V}{gI_s}} \right]$$

where θ_b = burning time, seconds

M_o = spacecraft mass at start of firing, lb_m

ΔV = velocity increment, ft/sec

g = gravitational constant, 32 174 ft/sec^2

I_s = specific impluse, lb_f-sec/lb_m

$\dot{M}_p$ = propellant consuption rate, lb_m/sec

A nominal propellant consumption of 2 5 pounds per second yields firing durations of 44, 533, and 528 seconds for the three maneuvers. These conditions require total propellant masses which are larger than those given in the work statement; however, the propellant loads given in the work statement (and the resultant tank sizes) were used in the calculations since the differences are small.

4 2 1 Discussion of Method of Analysis

The entire sequence of calculations utilized in the propulsion system analysis is included in Appendix B. However, certain comments concerning the method of analysis are in order at this point   As a matter of introduction, it should be recognized that a simpler or more complex method of analysis could have been chosen   The method chosen represents a judgement as to the optimum between useful results and complexity.

4 2.1.1 Helium Supply

Calculations of helium consumption rates are only as accurate as the solubilities, heat transfer rate models and helium properties used   In this case, solubility equations were obtained by curve fitting to the empirical data contained in Reference 7   No attempt was made to predict the rates at which the helium would go into or come out of solution, but instead it was judged that these rates were relatively slow so that no significant changes would occur during firing periods.

For the calculations made in this study, it was assumed that the propellants contain no dissolved helium until after the midcourse firing (seven days). After each firing, the helium in the ullages partially dissolves in the liquid propellant until an equilibrium concentration is reached   Prior to each firing, the ullages are brought up to regulated pressure in sufficient time for the helium tank and ullage gases to return to equilibrium temperatures prior to start, yet soon enough before the firing that no change occurs in the amount of helium dissolved in the liquids   Should pressurization occur only until immediately prior to the firings, a greater consumption of helium would result. The sudden withdrawal of the amounts needed to both raise the tanks to operating level and to expel the propellant causes the temperature in the helium tank to drop to a lower level than in the case with an interim warm-up period

During the two longer firings, the temperatures of the ullage gases drop below equilibrium temperature. This is because the helium in the helium tank becomes colder as a firing proceeds due to the fact that the withdrawal part of the helium allows the remaining helium to expand

Therefore, helium drawn from the tank is progressively colder and so chills the ullage gases as it mixes with them, reducing the average temperatures of these mixed ullage gases to below the initial temperature by the end of a long firing   (It was assumed, however, that the liquid propellant temperatures remained unchanged during firings). Therefore, there is a post-firing warming to equilibrium temperature which results in a pressure rise above regulated pressure unless a substantial fraction of the helium is dissolved in the liquid propellant After the orbit insertion burn, the propellants already contain an appreciable fraction of the equilibrium concentration of helium so that little of the newly added helium goes into solution   After the orbit inclination burn, the nearly saturated propellant absorbs very little helium and, therefore, the warming to equilibrium temperature will cause the pressure to exceed regulated pressure in some cases by 9 psi

The amount of helium which is required to expell the propellants is influenced by the amount of heat transferred to the gas within the helium tank, hence, the importance of the convective heat transfer coefficient Despite the widespread use of stored, high-pressure gas supplies, no proven, generalized analytical method of predicting the rate of heat transfer from a vessel wall to a diminshing gas supply has been developed [*] For the present program, a greatly simplified free convection model was adopted to simulate the gas film coefficient together with a simple, three-slab representation of the tank wall   It was assumed that no significant amount of heat was transferred to the tank from outside during the firing period (i.e , the only external energy available to the gas during firings was that stored in the tank wall)[**] Constant values of conductivity and heat capacity were used in all cases   The errors introduced by this simplification are probably smaller than the errors in the presently available physical constant values for the tank wall material   Small errors are also introduced by the use of constant

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[*] Apparently, Reynolds and Kays worked only with low pressures (Reference 8), and Keith covered very rapid blowdown rates (Reference 9). See also Reference 10.

[**] Originally, the planned program also was to account for heat extracted from the helium system components but this was not included due to the exigencies of time available   Were this energy available, the ullage gas temperature might by slightly higher than now calculated.

"average" values for the thermal properties of helium.

4.2.1 2   Ullage Conditions

No attempt was made to account for less-than-perfect mixing of the ullage gases   Analytical models of diffusion and convective mixing are rather complex and of mediocre accuracy so it was not deemed worthwhile to attempt using them   A point of interest is that the incoming helium will be more dense than the saturated propellant vapors in most cases. This means that the acceleration produced by the engine will tend to promote convective mixing of the ullage gases. Therefore, during such periods, substantial mixing may be expected. Nevertheless, it must also be expected that temperature, density and concentration gradients will be present in real tanks to some extent.

The equations were derived on the assumption that the ullage gases are always uniformly mixed and that the vapor pressures and densities are the saturation values at the average ullage temperature. This relation is essentially true when ullage temperatures fall below the liquid surface temperature. An initial difficulty did develop because originally the enthalpy balance, used to calculate ullage temperature change, was based on the assumption that the vapor flux entering the ullage is at the average ullage temperature (i.e., that the liquid surface is at the average ullage temperature, an assumption consistent with the dependency of vapor pressure upon ullage temperature). The mutual dependency of the vapor flux enthalpy and ullage temperature upon each other caused solution instability. By assuming the vapor flux to be at the liquid bulk temperature, the ullage temperature becomes dominated by the influence of the increasingly colder helium temperature. This is borne out by experience.

A remaining shortcoming is the lack of any accounting for the latent heat of condensation within the enthalpy balance equations   This is considered to be of minor importance.

4.2.1.3   Liquid Propellant Flow

Liquid temperature is important because it affects the flow rates and therefore thrust and mixture ratio. Liquid temperature was assumed constant during each firing period. As will be seen in Section 5, this

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assumption is correct as far as external heat transferred to the bulk liquids is concerned. The initial flow to the engine will be heated or cooled somewhat if the lines, valves, filters and injectors are not at the same temperatures as the liquids. But these parts have a relatively limited heat capacity and the turbulent flow conditions will promote rapid adjustment of their temperatures to that of the liquids

The major source for significant liquid temperature changes are the thermal interactions of the liquid-to-ullage interfaces Vaporization and heat transfer are the two important mechanisms. For the higher temperature cases, the vapor mass evolved during the orbit insertion firing becomes sizable, enough in fact to change the _average_ bulk temperature of the fuel by more than 2 degrees if the heat of vaporization were to be uniformly drawn from all the liquid Contriwise, if convective currents are real, thermal stratification could develop zones of relatively colder and denser liquid at the interface A sudden initiation of convection could conceivably carry this denser propellant to the tank outlet port The likelihood of this is uncertain. More probably, there would be a gradual commencement of circulation within the liquid before substantial gradients have developed. This tendency is less pronounced in the oxidizer tank. During the orbit inclination firing, when the interfaces drop close to the outlet ports and much of the colder propellant would be consumed, the tendency for such action to occur appears greater. However, as will be shown later, the consequences of this occurring are small The thrust and mixture ratio could shift one or two percent, but no harm to the engine would results.

Heat transfer between the ullage gases and the liquids will also tend to chill the liquid surface towards the end of the larger firings The magnitude of this effect was not investigated.

Resistance to propellant flow to the engine depends upon the design geometry. Several kinds of resistance relations are likely. Some components have discharge coefficients which are essentially constant over the range of Reynolds Numbers involved. For the present calculations, it was assumed that the injector was designed with this characteristic. The injector nominal design point chosen for these calculations was a fuel velocity of 140 ft/sec, a total oxidizer orifice

area of 1.4383 times the fuel orifice area, and an orifice C_D of 0.75 on both sides. (These values were derived from data supplied by JPL.)

Flow resistances in pipes and many fluid components are commonly approximated by a term which includes the product of a "friction factor" term (which is a non-linear function of Reynolds Number) and an equivalent pipe length-to-diameter ratio. All of the feedline losses, including the valving losses, were assumed to be characterized by a relation of this form. In the calculations, total resistances, inversely proportional to the squares of the flow rates, were selected which would give the rated flows at $250^{\circ}R$ with nominal tank pressure (300 psia) and chamber pressure (100 psia) The injector resistances were subtracted from these to leave the total resistances presented by the lines, valves, filters, etc. Calculated resistances for typical valve, filter and line designs yielded total resistances less than these differences, therefore, the remainders can be attributed to the trimming orifices.

The foregoing simplifications contain departures from reality For example, the corrugated metal hoses which comprise much of the feedline lengths would have resistances following totally different relations to Reynolds Number than do smooth-bore pipes Friction factors for such hoses are constant up to transition ranges of Reynolds Number Above the transition range, the friction factors increase to new constant values, Reference 11, Calculations show that the oxidizer hose will operate in a Reynolds Number range over which the friction factor is constant, but that the fuel hose's friction factor will vary by as much as 60%. The absolute magnitudes of the hose losses are fairly low, however, so the errors introduced by assuming "pipe flow" are small Filters typically impose pressure losses which are directly proportional to the flow rate rather than following the square law. And the trimming orifices may operate in the constant C_D range, whereas they were treated as equivalent L/D ratios in the present calculations.

4 2.1.4 <u>Engine</u>

Very small errors were introduced by assuming that the specific impulse is a function of mixture ratio alone, in practice, specific impulse will be slightly sensitive to chamber pressure as well

5.0 RESULTS OF ANALYSIS

5.1 Groundhold Thermal Control

All of the results from the analysis show that the thermal control system functions very well during the groundhold phase. Those module components which must be kept cold can indeed be maintained at the required low temperatures. Also, such components as the frame, insulation outside surface, helium control panel, and bipropellant valve act as relatively constant temperature components by remaining within $10^{\circ}F$ of the ambient temperature. The valve and filter located at the bottom of each propellant tank will, of course, remain within $20^{\circ}F$ of the tank temperature. The actual thermal characteristics of those components which must be kept cold after propellant loading are best demonstrated by the curves of Figures 5-1 through 5-4.

Figure 5-1 is a plot of the equilibrium temperature of the three tanks as a function of LN_2 coolant flow rate in each tank for an outside film coefficient of 1.0 Btu/hr-ft^2-$^{\circ}F$ (the film coefficient for still air would be about 0.35 Btu/ft^2-hr-$^{\circ}F$). It can be seen that it is no problem to keep the fluid temperature within limits In fact, for the design established, that is, an 8-foot coil of 1/2-inch tubing 8 feet long, the problem may be one of excess cooling to the point that freezing of the B_2H_6 may occur. With a pressure drop across the coil of only 25 psia, the coolant flow rate capability is considerably in excess of 1000 lbs/hr Yet, it can be seen from Figure 5-1 that a continuous flow rate of only 40 lbs/hr will result in a coil surface temperature which may be low enough to cause freezing on the surface of the coil ($190^{\circ}R$) Resolution of this problem, through coolant flow control, will be discussed later

The temperature of any particular tank is relatively independent of the temperature of the other two tanks. For example, a variation in the oxidizer tank temperature of $50^{\circ}F$ will result in a shift in the fuel temperature of less than $3^{\circ}F$.

If the oxidizer and helium tanks are maintained at $250^{\circ}R$ and all coolant to the fuel tank is eliminated, the resulting equilibrium fuel tank temperature would be near $480^{\circ}R$. The dependency of the helium tank temperature upon the combined effects of the other tank temperature is

slightly more pronounced. No coolant to the helium tank will result in an equilibrium helium tank temperature in excess of $440^O R$, depending on the ambient wind conditions. These temperatures apparently preclude an ambient access to a particular tank while the others are conditioned.

Also, this limited interdependence results in a certain characteristic which should be noted If the propellants are loaded, but the helium is not, there will be a tendency for the helium tank pressure to drop since the helium tank temperature will fall in response to the propellant tanks. Therefore, attention must be given to preventing the collapse of the helium tank. Unless the helium tank is designed to withstand negative pressures, additional helium must be added during the tanking of the propellants

The extent to which all coolant flow may be eliminated is indicated in Figure 5-2 As should be expected, the rate of temperature rise of each tank is closely related to the mass and specific heat of that tank Thus, the temperature response of the helium tank is most pronounced because of its low heat capacitance. Obviously, the time during which cooling may be eliminated is also related to the initial temperature. Assuming an initial helium temperature of $225^O R$, coolant to the helium may remain off only 9 hours before it's maximum temperature occurs In comparison, it will take some 40 hours for the OF_2 or B_2H_6 temperature to rise $40^O F$

The characteristics described above make it possible to utilize a simple on-off technique for controlling the propellant temperatures which will, at the same time, solve ground support equipment problems At most launch sites, the LN_2 supply is some distance away and for purposes such as proposed here, the lines are generally insulated with a mineral-type insulation such as Armaflex. Lengthy transport lines for the LN_2 will require high flow to obtain high quality liquid coolant at the tanks without necessitating vacuum jacketed lines As noted above, high flow rates in the fuel tank coolant coil could cause local freezing of the fuel. Therefore, in the present design, the coolant to the fuel tank will have to be operated on an intermittant basis with two different sensors controlling A temperature sensor attached to the coil, stops flow to prevent freezing and a sensor immersed in the fluid near its surface initiates coolant flow.

If groundhold thermal control is accomplished by flowing large quantities of LN_2 for relatively short periods of time, cyclic thermal response similar to that shown in Figure 5-3 may be expected   Graph A is for the particular case of LN_2 flowing at 1000 lbs/hr in each coil   Graph B is for the case of LN_2 flowing at 35 and 200 lb/hr in the fuel tank coil.

In this analysis, it was assumed that the LN_2 flow in the helium and oxidizer tanks is controlled by thermocouples immersed in the fluid and that LN_2 coolant flow would be initiated when the fluid temperatures exceed $270^O R$ and is stopped when the fluid temperatures drop below $220^O R$ It was also assumed that the coolant flow in the fuel tank is initiated when the bulk temperature exceeds $270^O R$ but that the coolant flow is stopped when the coolant coil temperature drops below $190^O R$, the temperature at which freezing on the coil could conceivably start

The effect of controlling the fuel temperature in this manner is clearly demonstrated in Figure 5-3   At a flow rate of 1000 lbs/sec, the minimum bulk temperature of the fuel is about $252^O R$ This is because at the high coolant flow rate, the coil is substantially colder than the surrounding liquid   In comparison, if the flow rate is 35 lbs/hr and the lower control temperature is kept at $190^O R$ at the coil, the fuel bulk temperature will drop to approximately $203^O R$ This clearly points out three facts

1. The cooling response rate and rate of temperature drop is not materially affected by the LN_2 flow rate because the outside film coefficient is the controlling parameter, not the inside coil coefficient

2. There is a maximum LN_2 flow rate which may be accommodated, approximately 1500 lbs/hr. A higher flow rate will result in the low temperature sensor (coil temperature) giving a signal to stop LN_2 flow in order to prevent possible local freezing when the bulk temperature is still in excess of $270^O R$ The actual result would be that the LN_2 flow control valve would turn on and off fairly rapidly (possibly several times each hours) without any effective cooling of the fuel resulting

3   There is also a minimum allowable LN_2 flow rate which is required to maintain the bulk fuel temperature above $210^O R$ if a constant control temperature of $190^O R$ is maintained. This rate is about 175 lbs/hr.

With OF_2, the cooling capacity is insufficient to cause freezing and there is no freezing problem with helium

It should be noted that it is not clear whether local freezing on the coil is objectionable   On refrigeration coils operating in the air, local freezing of atmospheric vapor is undesirable since the frost formed may be fluffy and act as an insulator   Whether freezing of B_2H_6 would be objectionable for the same reason depends on the magnitude of the thermal conduction of the "ice" in comparison to the film coefficient on the outside of the "ice"

In addition to the problem of ice reducing the efficiency of the fuel cooling coil, there is the possible problem of loose ice being injected into the propulsion system upon engine operation   This is not a serious problem   If the LN_2 is turned off while the bulk temperature is still above $210^{\circ}R$, any ice formed will melt within hours Only if the bulk fuel temperature is very near the freezing point and/or a very large quantity (tens of pounds) of ice has been formed will more than a few hours be required to melt the ice   This characteristic, however, may be an important consideration when the time of first engine firing is scheduled.

This problem of freezing could be overcome by reducing the size of the cooling coil   However, such a size reduction also reduces the capability of accommodating unusually large heating loads which might occur in an emergency such as insulation failure   As will be indicated below, the present design is well suited to handle such emergencies and considering the chemical characteristics of the propellants, it is considered wise to retain this emergency capability

There is one point which must be given serious thought and planning   As will be shown later, there are distinct advantages to launching with all fluids at their minimum temperatures   But, from Figure 5 3, it can be seen that using a purely automatic system will

result in all the fluid tanks thermally cycling at different frequencies
It would thus be difficult to assure that all temperatures are at their
minimum at the time of launch   This problem may be overcome in either
of two ways, but in either case, a manual override in the thermal control
system would be required   The most obvious way is to include in the
launch count-down a step for initiating a "last cooling" sequence for
each tank. The coolant to each tank would be turned on at a time such
that given the cooling characteristics of that tank, it will just reach
its lowest temperature at the time of launch. A second way to handle
the problem is to lower the upper control temperature limits to very
near the minimum allowable temperatures. This approach would be wasteful
of LN_2 but could be done 48 hours prior to launch to reduce the waste

All of the discussion heretofore is applicable reagrdless of
the location of the coils within the tanks provided the coils are
submerged. But, if it becomes necessary to locate the coils near
the bottom of the propellant tanks in order to accommodate require-
ments established by the propellant acquisition devices, the average
temperature of an entire tank will be somewhat higher   This is
because cooling occurring near the tank top aids in setting up nat-
ural convection currents, whereas cooling at the bottom leads to
thermal stratification   An engineering estimate, based on the char-
acteristics of the propellants, indicates that the average propellant
tank temperature will be raised some $10^{\circ}F$ if the coil is relocated
from the top to the bottom

The capacity of the system to accommodate variations in the
heat transfer rates to the tanks is excellent. Such an increase
would be caused by the following.

o   Increase in outside film coefficient

o   Degradation of the foam insulation

o   Failure of the insulation by separation from the tank(s)

The first cause is of little concern since its relative contribution to the overall resistance to heat flow is minor compared to the resistance of the foam For example, a 500% increase in h_o results in only a 1^oR increase in the propellant temperature. In comparison, a 50% increase in the insulation conductivity will result in approximately an 18^oF temperature rise if the coolant flow rates are maintained constant Either of these variations is readily counteracted by slight changes in the coolant flow rate.

The extent to which increased heat transfer rates may be overridden by increased coolant flow is indicated by Figure 5-4. This figure gives the tank temperatures which would result for varying coolant flow rates if an additional 2000 Btu/hr were added to each tank. Recalling that the coolant coils can readily pass 1000 lbs/hr of LN_2 it can be seen that a very sizable heat transfer rate increase can be accommodated.

To experience an additional heat load of 2000 Btu/hr into a given tank, a major insulation failure would have to occur. Reference 5 reported that the overall conductance of an uninsulated, thin walled (0 05 inches) metal tank containing LN_2 varied from 0 81 to 3.53 Btu/ft^2-hr-^oF as the outside wind velocity varied from 0 to 20 fps These experimental data were apparently obtained after some frost had collected on the outside surface If it is assumed that the conductance is doubled in the absence of frost and that the heat transfer properties of LN_2 and the propellants under study are similar then a heat addition of 2000 Btu/hr could be experienced if approximately 1 ft^2 of insulation were removed

It should be realized that the results of Figure 5-4 assumes that no propellant vapor is formed or, if it is formed, it immediately condenses back as a part of the liquid bulk As such, the results cannot be used if boiling without recondensing occurs. The extent to which severe local boiling could not be handled can only be established by test

5.2 Flight Thermal Control

The general approach used in the flight thermal analysis was to investigate discrete sections of the mission with an eye not only to determining the operation of the particular vehicle under study but also to establishing general principles which may be applicable to other vehicles

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having similar propellants and missions. Thus, the results and discussion below consider both normal operation and the effects of changes in design and mission.

There are several major factors which bear upon the module temperatures They are

- o Off-pointing angle, θ (see Figure 3-1)
- o Solar radiation intensity, G (see Figure 3-2)
- o Transient factors including module temperatures at launch, mass of fluids on board, trajectory, and engine operation
- o Extent of sun shielding by spacecraft (size and shape of sun shield)
- o RTG temperature

In order to fully understand the influence of these factors, it is necessary to consider them somewhat independent of each other and then combine them in building block fashion to show their combined effects

5 2 1 Solar Radiation and Off-Pointing Angle Effects

The effect of the first two items can be seen from Figure 5-5   In this plot, the effects of constant, but different, solar intensities and/ or off-pointing angles are accounted for in the abscissa where G is the solar radiation intensity at any time during the mission and G_{max} is the solar constant at 1 A.U. (430 Btu/ft^2-hr). For example, at a distance of 2 A U. where the solar intensity is roughly 100 Btu/ft^2-hr, the average fuel temperature will be approximately $250^{\circ}R$ if the off-pointing angle is held constant at 30° Note carefully that this plot applies only for a design which utilizes an abbreviated sun shield as indicated in Figure 5-5 and a normal RTG temperature of $960^{\circ}R$ [*]

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[*] The curves of Figure 5-5 are applicable only to the extent that the solar heating is a continuous function of the off-pointing angle. If a shield were present, such that a given component were shielded from the sun for a quantum of off-pointing and then became exposed to the sun for increased off-pointing angles, the effects of both G and θ could not be combined into the single function $\frac{G \sin \theta}{G_{max}}$ For the case in which an abbreviated shield is assumed, this criteria is effectively met.

Three points should be noted from these curves

1. Steady state operation in the shade, either 0° off-pointing
    or in the shade of Jupiter, will result in major component
    temperatures of approximately 225^OR.

2. For radiation from the +X (or -X) side, the helium and pro-
    pellant temperature differentials remain small.

3. The propellant valve responds to solar radiation much more
    readily than do the propellants and helium

The reason that the valve responds differently is that it is strongly
coupled with the engine bell and thus, its temperature is very strongly
controlled by the engine bell temperature

Because of these characteristics, it is apparent that the orientation
which could be accommodated with an abbreviated shield, that is, the
amount of solar radiation that is allowable, is determined by the prop-
ellants, particularly the OF_2. However, the tendency of the valve to run
hot would dictate that certain maneuvers be made prior to engine start in
order to drop the valve temperature If, during the initial stages of
the mission when the sun intensity is high, the craft were to be oriented
such that the relative solar intensity were about 0 2, the propellant
temperatures would be within limits, but the valve temperature would be
too high. Therefore, if an engine start were attempted at such a time,
either the off-pointing angle would have to be reduced long enough to
allow the valve to cool or a "hot" start would have to be made. Of
course, the valve temperature will be near the propellant temperature
within a short time after propellant flow is initiated

Similar data exists for the propellant feed lines, the insulation
valves, and the helium control panel. Except for those portions of the
two propellant feed lines adjacent to the bipropellant valve, the temp-
erature of these two lines will follow within 5^OF of the respective
propellant tank temperature. Obviously, the ends of the lines which
attach to the main valve will follow the valve temperature.

As indicated above, the isolation valves located below each tank may
be as much as 20^OF above the temperature of the tank to which it is
attached during groundhold During flight, however, the heat transfer
rates to these valves are much smaller and consequently the isolation

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valve temperatures follow the propellant tank temperatures within $4^\circ F$

The helium control panel temperature follows substantially the temperature of the OF_2 tank but because of its position and attachment method it does fluctuate somewhat. If it receives no solar radiation, its temperature will be approximately $11^\circ F$ below the OF_2 tank temperature   But if the module were to be oriented such that the -X side is exposed to solar radiation at a relative intensity of 0 3, the helium panel will exceed the OF_2 temperature by approximately $22^\circ F$

5.2.2 Transient Factors

By itself, Figure 5-5 does not give a totally clear picture of the module temperature characteristics when an abbreviated shield is used since it gives only quasi-steady-state temperatures, that is, it does not account for the heat capacitance of the module or the varying nature of the solar heating

The varying nature of the solar heating is indicated in Figure 5-6 The solid curve gives the relative solar radiation intensity on a black surface located in a Y-Z plane if it were to have the mission parameters given in Figures 3-1 and 3-2 (the normal mission for this study)   This curve shows that solar heating decreases rapidly and is highly dependent upon the off-pointing angle   If no shading were provided, not even the abbreviated shield, and the surface were constantly perpendicular to the solar rays, the relative intensity would be as given by the dashed curve of Figure 5-6   A comparison of these two curves indicates the increase in solar heating which is caused by off-pointing only when no side shielding is provided

Using hand calculated data of solar heating (similar to that of Figure 5-6) for the various components which can receive solar heating during a normal mission when only the abbreviated shield is used, a series of computer runs were made to ascertain the module temperature characteristics during various phases of the mission   These runs established one point which is extremely important to an understanding of the thermal analysis   That is, except for those situations in which sudden changes take place, the results given in Figure 5-5 are correct   Stated differently, the environmental conditions surrounding the craft change

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slowly enough such that the module may be considered in a thermally quasi-steady-state condition during all phases of the mission except for the following four conditions.

1. Approximately the first 20 days after launch when the module is adjusting to the flight environment.

2. Immediately after and during a major module orientation maneuver which shifts its position relative to the sun.

3. Upon entering or leaving the shadow of Jupiter

4 During and after an engine firing

The typical thermal response which can be expected immediately after launch is given in Figures 5-7 and 5-8. Note that these results do not consider the effect of ascent heating (or cooling), engine operation, or earth radiation and albedo The ascent heating or cooling will have no effect upon the fluid temperatures since, in the extreme, this phase lasts only a few minutes and, as shown in the groundhold analysis, the module will not respond even to severe environmental changes in such a short interval. Also, for a normal mission, earth effect are small and last for only one or two hours. Only if the mission includes an extended period of time in a near earth orbit, will earth effects be appreciable.

If the mission did include an extended period of time in earth orbit, the effect would be highly dependent upon the orientation of the craft The worst condition would occur if the module were non-spinning and oriented in a 90° off-pointing angle. Since earth emission is approximately 68 Btu/ft^2-hr, and albedo is approximately 168 Btu/ft^2-hr, it is obvious that the module temperatures will rise towards totally unacceptable levels, > $325^\circ R$. The best orientation would be in a 0° off-pointing angle. Here, the heating of the tanks would be predominately due to earth emission since the tanks would be shadowed by the aft shield from the albedo during those periods that the view of albedo is largest. And of course, albedo heating occurs for only about half the orbit time for low altitude orbits and it is small for high altitude orbits. These characteristics are shown in Figure 5-9.

A very conservative approximation of the equilibrium temperature of

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the tanks during a 0° off-pointing earth orbit can be had by assuming that the tanks are exposed to a constant heating load of half the albedo plus half the earth emission, or 118 Btu/ft^2-hr   This is a relative solar intensity of 0.275 (118/430)   From Figure 5-5, the equilibrium temperature of the OF_2 tank is indicated as 283°R, 3°R over the allowable limit.   A more rigorous solution would show a maximum temperature somewhat lower.   In addition, when the effects of the shadowing produced by the standard 10-foot diameter shield are considered, the equilibrium temperature would be even lower.

Note, however, that even a 0° off-pointing angle will result in an excessive bipropellant valve temperature if a low earth orbit is maintained because the engine will receive a substantial amount of heat from the earth.

Regardless of whether an extended time period is spent in an earth orbit, several important points are demonstrated by the curves of Figures 5-7 and 5-8.   First, the assertion made previously that the curves of Figure 5-5 can be used in determining module temperatures during most of the mission because the environment is only slowly changing is well demonstrated   Figure 5-6 shows that the relative intensity at 20 days as 0 22.   Using this value to enter Figure 5-5 to obtain the quasi-steady-state temperatures results in temperatures very close to those given in Figure 5-7 at 20 days.   The oxidizer temperature at 20 days is approximately 5°F below quasi-steady-state and from the temperature trend shown it appears it will not reach equilibrium until about day 22

It is logical that the oxidizer should take longer to come to quasi-equilibrium since it has a higher heat capacity, wc_p.   In comparison, the bipropellant valve is in quasi-equilibrium by day 6.   The reason it does not come to quasi-equilibrium sooner is that its quasi-equilibrium is directly dependent upon the engine.

Figure 5-8 shows that the very low heat capacity components will be in quasi-equilibrium within one day.

Looking again at Figure 5-7, it will be noticed that the thermal histories of the helium and propellants during the initial days after

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launch are highly dependent upon the launch temperature It is this characteristic which makes it possible to keep the propellants within the specified limits during the initial days of the normal mission if only an abbreviated shield is used. The reason for this is as follows. If the module were launched with propellant and helium temperatures around $270^{\circ}R$ to $280^{\circ}R$, the quasi-equilibrium temperatures should be reached within four days. But at four days, the relative sun intensity is sufficiently high to cause overheating of the OF_2. The point to be gained from this is that to avoid propellant overheating, quasi-equilibrium must be delayed until about day 14 or sun shielding must be provided The necessary delay can be accomplished either by launching with cold propellants or reducing the off-pointing angle during the first 13 days. But as will be shown later the standard design provides sufficient shielding to eliminate this problem

Figure 5-7 and Figures 5-5 and 5-6 also show that with only the abbreviated shielding, an engine firing may not be initiated within the first 33 days without first orienting in such a manner as to cool the bipropellant valve unless firing with a hot valve is permissible. The requirements of such a pre-firing maneuver is dependent upon the time of firing and the required orientation at time of firing.

Figure 5-10 shows the typical thermal response of the module if it were to move from a 0° off-pointing angle to a 90° off-pointing angle in 15 minutes, stay at 90° for 34 hours and then move back to a 0° angle in 5 minutes. As should be expected, the valve readily responds to the solar radiation whereas the fuel responds very slowly The oxidizer, having a higher heat capacitance, responds more slowly than does the fuel.

Applying this data, it is possible to see that if an engine firing were to be made on the 7th day when the valve is $307^{\circ}R$, the craft would first have to be maneuvered to 0° off-pointing and held in that position for approximately 11 or 12 hours in order to reduce the valve temperature to $280^{\circ}R$, the maximum temperature limit for the oxidizer. Even if the craft were launched into a 0° off-pointing position and held there, Figure 5-11 shows that it would be approximately two days before the valve would reach $280^{\circ}R$.

There may be one possible way to decrease the valve temperature when the engine is exposed to solar radiation and that would be to coat the engine surfaces which see the sun with a material having a low α/ϵ. However, it would be necessary that the material burn off after the first burn. The disadvantage of such an approach is that if the off-pointing which occurs prior to the first burn is not sufficient the valve would then be too cold at the time of the first firing

Figure 5-10 is for the specific case of 90° off-pointing near earth, and full tanks If the tanks should happen to be only partially filled, the response rate at any given temperature will be inversely proportional to the mass of the tanks. If the reorientation should happen to occur at a later date in the mission when the solar intensity is reduced, the response rate will be reduced in porportion to the intensity reduction

The ability to continuously accommodate solar intensities ranging from 0 to that which exists near earth necessarily includes variations which will be encountered while in orbit around Jupiter. For the Jupiter orbit given (for the normal mission), the effects of Jupiter emission and albedo are negligible since the view factor of Jupiter is less than 0.05 and the Jupiter emission and albedo are only 2 Btu/fr^2-hr and 8 Btu/fr^2-hr respectively. If, however, the orbit diameter around Jupiter were to be reduced so that the view factor of that planet increased to an appreciable value, say 0.3, the module temperatures could rise significantly Particularly, if the module were oriented with the -X side always towards Jupiter, the temperature rise would be large since the module would be "boxed in" with the warm RTG on the +X side and the relatively warm Jupiter on the -X side

The remaining normal mission transient condition to be discussed is the time during engine firing.

Data supplied to TRW Systems indicates that during steady-state engine operation, the outside temperatures on the engine will be approximately as given in Figure 5-12. However, for purposes of analysis, the entire bell (from the throat to the exit) was assumed to be at 2600°R and the remainder of the engine was assumed to be at 1000°R.

It was assumed that during the firing, the bipropellant valve would be maintained at the propellant temperature because of the cooling effects of the propellant flowing through it.

As will be shown later in the propulsion system analysis, during a firing the helium will drop in temperature about as indicated in Figure 5-13. Therefore, in the thermal analysis of the engine firing, the helium temperature was constrained to follow the curve of Figure 5-13. It may be argued that this approach is not adequate in that during the firing, radiated and conducted heat from the engine may prevent the helium from dropping as far as indicated by Figure 5-13. To establish that, at least during the firing, the helium temperature is independent of the engine temperature, a computer run was made in which all parameters were as they are for a quasi-steady state analysis except the engine was assumed to be at 3000°R. That run showed that the helium temperature did not noticeably respond to this perturbation for over an hour. It was therefore concluded that the approach described above was justified.

The results of this portion of the analysis are shown in Figure 5-14. As was expected, the firing of the engine has only a small effect upon the propellant temperatures, and its effect upon the helium temperature, other than through the helium consumation process, is not severely large A hand calculation reveals this characteristic more clearly. For example, if it is grossly assumed that all the thermal energy contained in the hot engine at shut down is transferred directly into a half full B_2H_6 tank, the resulting temperature rise in the fuel tank is only 17°F.

There are two areas which are affected, however. First, it should be noticed that the aft shield reached 824°R. This is sufficiently warm to dictate that the multilayer insulation blanket which rests against the shield be made of aluminized Kapton insteady of aluminized Mylar

The second affected area is the propellant feed lines. The valve temperature rises after shut down to approximately 596°R because of the heat soak back from the engine. The propellant feed lines will, in turn, rise in temperature. Consequently, propellant held in the feed lines at shut down will be boiled to a certain extent. A return relief line which

dumps the propellant vapor so formed back into the tank is provided, but
the extent of this boiling process is unknown. The computer program does
not consider this problem at all since to do so would require a detailed
knowledge of transient boiling heat transfer in zero gravity. A hand
calculation which assumed that all the heat which caused the temperature
rise in the lines at shut down was instead used to vaporize propellants
indicated that approximately 0.05 lbs of B_2H_6 and 0 11 lbs of OF_2 would
be vaporized. How the system would function if these quantities of vapor
were vented back into the tanks is not known but it appears it would be
recondensed immediately unless the entire propellant bulk were at the
saturation temperature.

5.2.3 Effects of Sun Shielding

Up to this point, it has been necessary to eliminate the effects of
solar shielding as much as possible in order to discern the effects of
the other variables. Shielding effects will now be considered.

In general, the presence of a surface near the module has two dis-
tinct effects. It may prevent the module from being exposed to solar
radiation (this depends on the module orientation relative to the sun)
and it does reduce the module's view factor of space. For the case of an
abbreviated amount of shielding at the top of the module as considered
in the previous discussion, these effects are minimized If an extended
shield is incorporated into the module design, the effects can be very
substantial. Such is the situation with the present module design in
which a 10-foot diameter flat shield is assumed to exist at the space-
craft/module interface. Where any off-pointing resulted in all the
tanks, the engine and the frame receiving solar radiation when the abbrev-
iated shield was assumed, no solar radiation is received by any of the
module for off-pointing angles of up to 22° when the standard shield is
considered. In addition, an off-pointing angle of 36° results in only
the frame, helium tank and engine receiving solar radiation when the
normal shield is used. The general effect of this shield is to cause
the module to function as if it were the shade for off-pointing angels
up to 22° From 22° to 90°, the various components receive varying amounts
of solar radiation depending on their location.

Unfortunately, it is impossible to graphically represent the steady-state thermal characteristics of the shielded module by a single set of curves similar to those presented in Figure 5-5. Instead, several families of curves similar to the family for B_2H_6, shown in Figure 5-15, are required. The value of G/G_{max} in Figure 5-15 is as given by the dashed line of Figure 5-6

The point to observe from Figure 5-15 is that regardless of G/G_{max}, the quasi-steady-state fuel temperature is constant because the fuel is always shielded from the sun. Since this is the case, it is readily apparent that the RTG should be adjusted to cause the fuel to run at about $250^{\circ}R$ This can be accomplished by increasing the RTG temperature to approximately $1100^{\circ}R$ or moving the RTG about 1 foot closer to the module.

· 5 2.4 Mission Operation

Utilizing all of the information and techniques which have now been discussed, it is possible to construct the temperature histories of the module components for an entire mission. In order to fully demonstrate the thermal characteristics of the module, this has been done for both a module which has only an abbreviated shield and one which has the normal shield. The results are presented in Figures 5-16 and 5-17 The major differences between the two configurations are two

1. The spacecraft with the normal 10-foot shield does not need to be reoriented prior to the first engine operation since the shielding aids the valve to attain operational temperature limits

2. The module temperatures will remain constant once the initial launch transient is overcome.

The 10-foot diameter shield does add a substantial safety factor to the thermal control system design Figure 5-16 shows that the module operates near its maximum temperatures immediately after launch and at its minimum temperatures when the off-pointing is zero if an abbreviated shield is used. But with the larger shield, module temperatures remain constant once the initial transient is overcome. In fact, if the LN_2 coolant flow rate during groundhold is properly adjusted, the propellant temperatures will remain constant during the entire mission. Thus, it is possible to

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accommodate, within limits, additional unforeseen variations such as shifts in RTG temperature. However, for large off-pointing angles, >50°, for long periods of time, the large shield would actually be a slight detriment because it reduces the module's view of space.

Curves similar to those of Figures 5-16 and 5-17 can be constructed for almost any conceivable mission to Jupiter (except ones which included extended orbit time near earth or low altitude orbits around Jupiter) by using just the curves presented to this time. Such results would necessarily be less reliable but they would serve as excellent first order approximations. For example, suppose the mission were to be altered to the extent that instead of the module being pitched to allow solar heating on the +X side, it was pitched to allow the off-pointing solar heating on the -Y side. That is, only the fuel tank would receive solar heating If the lift-off temperatures were as the case of Figure 5-7, the results would be approximately as follows:

Module with Abbreviated Shield

o   The fuel temperature history would approximate the temperature history during a normal mission, Figure 5-16, since solar heating would be about the same.

o   The oxidizer temperature would remain at about 225°R since it will effectively operate as if it were continuously in the shade.

o   The helium tank will substantially act as it does during a normal mission since it will receive about the same amount of solar heating and the "hot" fuel and "cold" oxidizer will counterbalance each other relative to the heat conduction through the aluminum support structures.

o   The bipropellant valve will function as it will during a normal mission since the engine bell receives about the same amount of solar heating regardless of the direction of the sun, and it is predominately the bell which controls the valve temperature.

Module with Regular Shield

o   The temperatures would be as indicated by Figure 5-17 since the module will still remain shaded

The reason such an approximation of module operation can be made is that the various parts of the module are, to a limited degree, thermally

independent   The errors involved in making such an approximation stem
from two sources. First, the extent of shadowing is not entirely in-
dependent of the direction of the sun even though the off-pointing is
held constant. Second, and more important, whether a -X louver or a
+X louver, or no louver at all, receives solar heating affects the tank
temperature   If, for example, the -X side were to be exposed to the sun,
it would be through that louver that a significant amount of heat would
be added to the module tank.

5 2.5 Effects of RTG and Louver Changes

To this point, the discussion has considered variations in the
mission and variations in the foam insulation qualities. However,
no consideration has been given to the consequences of varying the RTG
temperature or replacing the louvers with totally passive radiators
These two variations will now be considered

The sensitivity of the module to variations in the RTG temperature
depends upon the amount and location of solar heating which is occuring
The module is most sensitive to RTG changes for the condition of 0° off-
pointing, that is, no solar heating   Figure 5-18 is a plot showing the
sensitivity at this condition. This plot indicates that the RTG temper-
ature could be increased to approximately 1135^OR before the maximum al-
lowable propellant temperature of 280^OR is exceeded. This is indeed true
for 0° off-pointing, but with any appreciable solar heating, the maximum
allowable temperatures would then be exceeded

At the other extreme, the curves show that the RTG temperature must
not be less than approximately 800^OR if the fluid temperatures are to be
maintained at all times above 220^OR. Of course, the RTG temperature can
drop considerably without harm if solar heating occurs.

The module was designed to function properly when the RTG is at
about 960^OR. It could have been designed to operate with a different
nominal RTG temperature. It should also be noted that it could have been
designed to maintain the propellants at different temperatures   This
could prove to be a decided advantage should a clearer definition of the
mission may indicate that one propellant tank will receive more solar heating

than the other   Such changes in the design are readily handled by moving and/or rotating the RTG only minor amounts relative to the module

In line with this last comment, it should be noted that the design established is not necessarily the optimum   This design, however, does permit sufficient temperature control for the normal mission specified, but it may prove wise to vary the mission, or RTG temperature or even a combination of several variables in order to accommodate a specific objective of the overall project   To arrive at the optimum design, it would be necessary to make an exhaustive parametric study which would consider the following variables

      o   Off-pointing angle during the first 40 days

      o   Module orientation during the first 40 days

      o   Time of first firing

      o   RTG temperature

      o   RTG temperature variation

      o   Louver area

As for replacing the louvers with radiator plates, the effects are not as striking as might be expected   A series of computer runs were made to establish how a totally passive system with an abbreviated shield would function when exposed to different intensities of solar radiation The results are given in Figure 5-19   This plot is similar to Figure 5-5 and may be compared directly to establish the effectivity of louvers as opposed to radiator plates   It will be seen that replacing the louvers with radiators of equal area reduces the allowable solar intensity to the propellants by about 25%.   It should also be noted that the equilibrium temperature for no solar heating is slightly higher, 5°R   By decreasing the +X radiator area slightly (the radiator which sees the RTG), the curves could be made to shift down slightly.   As it turns out, radiator plates having the same area as the louvers are not the optimum size.   If the radiator area were changed to the optimum size, the allowable solar intensity to the propellants would be only 10% less than it is when louvers are used.   Thus, where a semi-passive system can withstand a

relative solar intensity of 0 34, the passive system can withstand an intensity of 0.31.

Of course, the louvers also make it possible to accommodate a larger fluctuation in RTG temperature   However, it can be seen that the louvers do not increase the flexibility of the thermal control system as much as might be expected   The reason the louvers have such a minimal effect stems from their locations.

The -X louvers, those which do not see the RTG, transfer less than 15 Btu/hr when fully open because of their low radiating temperature   The +X louvers, when fully open, receive less than 35 Btu/hr from the RTG because of the view factor   In comparison, the propellants may loose as much as 110 Btu/hr to space through the insulation and they can gain as much as 120 Btu/hr from the RTG through the insulation   In essence, the thermal control exerted by the louvers is only a partial control

The choice of putting the louvers on the tanks was dictated by the ground rule that no active thermal control can be utilized   Obviously, the ideal place to locate the louvers is on the RTG with a sensor attached to the tanks so that the louvers, though located on the RTG, would function according to propellant temperatures   With this arrangement, heat loss to space through the insulation would be about 100 Btu/hr but the heat gain from the RTG could be made to vary from essentially 0 to 200 Btu/hr   It must be clearly understood that the louvers so located would have to operate from tank sensors since the louvers would have no way to compensate for varying solar heating, or, in the alternative, 100% sun shading would have to be provided

There is, of course, the possibility of eliminating the radiator plates also and having only insulation on the tanks   This can be done theoretically, but the thermal control system would have very little capability to accommodate normal variations in fabrication (normal material variations) let alone mission variations   Where the louvers give a control range of approximately 44 Btu (variation in heat transfer via the louvers) and the radiator plates give a control range of about 32 Btu, the insulation has a control range of less than

10 Btu The reason for this becomes apparent if the last two cases are compared to the simple circuit given below
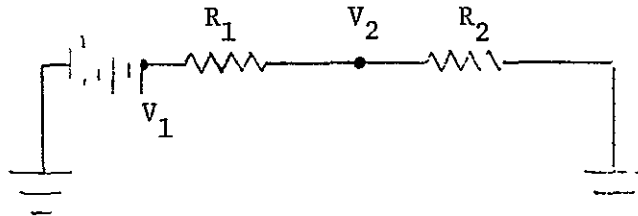
For the case of insulation, the resistence of the insulation, R_1 is constant but the radiation resistence, R_2, is somewhat smaller and is inversely proportional to V_2^4. Thus, when R_1 is very large compared to R_2 the network acts as a simple linear network in which the current is directly porportional to the voltage (heat transfer is directly proportional to the temperature). For the case of a radiator plate, however, R_1 is zero and the current is proportional to V_2^4 (heat transfer is porportional to temperature to the fourth power)

Considering the nature of this propulsion system, an engineering judgement would dictate the use of radiator plates since they are less complicated than the louvers and yet they provide sufficient control which only insulation does not

5.3 Propulsion System Results

Before describing the results of the propulsion analysis, it should be recalled from Section 4 that the following assumptions are made

1. The propulsion system is calibrated to deliver 100 psia chamber pressure at a mixture ratio of 3 0 when both propellants are at $250^{\circ}R$ and pressurized to 300 psia

2   The propellants are not saturated with helium prior to launch.

3. The propellant tank ullages are not pressurized for the first time until shortly before the midcourse firing

4   For all firings, the ullages are not suddenly pressurized
     just prior to firing. Rather, they are pressurized soon
     enough before the firing that both ullages and the helium
     tank may return to thermal equilibrium prior to the firing

These assumptions were made on the belief that they represent the
best compromise between standard pre-launch handling requirements and
helium conservation during flight. The practicality of the sequence
of Item 4 will be discussed later.

As shown above, the predicted propellant and helium equilibrium
temperatures are very nearly constant during the normal fully-shaded
mission. A set of computations were made to reveal the system per-
formance and operating points during this type of mission. Fuel and
oxidizer temperatures were assumed to be a constant $250^{\circ}R$ and equili-
brium temperature was set at $256^{\circ}R$.

With the propellant equilibrium temperatures held constant through-
out the period from before the midcourse firing to the end of the orbit
inclination maneuver, the mixture ratio and chamber pressure will be
the same during each firing for constant tank pressures (assuming no
degradation in propellant or hardware)   At the end of a single 44-
second midcourse manuever, the oxidizer ullage temperature is calculated
to be 252 5$^{\circ}R$ and the fuel ullage temperature is 251.7$^{\circ}R$. A total of
27.18 pounds of helium at 251.5$^{\circ}R$ is left in the helium tank.

During the following cruise period, the propellant tank ullage gases
will return to $250^{\circ}R$. Since the liquid propellants were free of helium
until the firing, a portion of the helium in the ullages will dissolve
into the liquids. When equilibrium helium concentration at $250^{\circ}R$ is
attained, the total pressures (vapor pressure plus partial pressure of
helium) will be 291.8 psia in the fuel tank and 248.9 psia in the
oxidizer tank   Prior to the orbit insertion firing, these ullages
must again be pre-pressurized to 300 psia.

At the conclusion of the 533-second orbit insertion firing, the
oxidizer ullage is at 245.4$^{\circ}R$ and the fuel ullage is at 242.3$^{\circ}R$. At
this point, the temperature of the 17 07 pounds of residual helium
is 215.8$^{\circ}R$

The post-firing cruise period allows the propellant tank ullage gases to return to equilibrium temperatures. This causes the pressures to rise above operating level since the propellants have become saturated during the previous cruise period Hence, the propellants absorb very little additional helium at the new equilibrium conditions Oxidizer tank pressure at equilibrium cruise condition is 208.3 psia, the fuel tank pressure is 308.4 psia.

Starting the orbit inclination firing at these elevated pressures generates a momentary surge in chamber pressure to 102 1 psia and the mixture ratio drops to 2 976 to 1. In less than 20 seconds, both of the tank pressures decline to regulated levels (300 psia) and engine performance is again nominal At the end of the 528-second firing, the oxidizer ullage is at $241.2^{\circ}R$ and the fuel is at $240 5^{\circ}R$. The final residual helium mass is 7.14 pounds at $205.6^{\circ}R$, this corresponds to a final pressure of 821 psia. These results are given in Figure 5-17.

Of course, it is possible to analyze any given mission and obtain a picture similar to that given for the normal mission.

In order to obtain a generalized picture of what may occur relative to engine performance should the fuel and oxidizer temperatures be other than $250^{\circ}R$, computations were done for a matrix of cases The results are given in Figures 5-20 and 5-21. These plots show the effect of propellant temperatures on mixture ratio and chamber pressures It will be seen that mixture ratio never exceeds 3 16 nor falls below 2 88 The extreme values of chamber pressure are 104.3 and 96.5 psia These values occur when the oxidizer temperature falls to $200^{\circ}R$ or rises to $290^{\circ}R$, with corresponding fuel temperatures of 210 and $270^{\circ}R$. If neither propellant exceeds the 210 or $270^{\circ}R$ limitations, then the mixture ratio range is approximately 3.13 to 2 93, and the chamber pressure range is 103 6 to 98.2 psia

No generalized data is presented which indicates the helium consumption for various arbitrary missions simply because the range of variables is nearly limitless Not only is helium consumption dependent upon propellant temperatures, but also upon mission profile.

From the comments and data presented, it can be seen that the
module propulsion system performance is relatively insensitive to prop-
ellant temperatures   Neither mixture ratio nor chamber pressure
changes more than 1% from the nominal design points if the propellants
are kept at the same temperatuie and within 12°R of the nominal (250°R)
temperature.   Under the worst conditions, that is, either propellant
at the extreme limits and the other propellant temperature differing
from it by a full 20°R, the mixture ratio and chamber pressure will be
within 5% of the nominal values   (All of these conclusions relate to
firings with both propellant tanks at 300 psia)

It should be recalled that the basic module design results in an
off-set between the module centerline and the spacecraft centerline of
approximately 10.62 inches   One concern has been the possibility of
a shift in spacecraft C.G. should the mixture ratio not remain nominal
Calculations show that for the normal mission, the C.G shift after the
last burn will be approximatley 0 25 inches towards the fuel tank.   Only
a minor shift should be expected for the normal mission since the engine
operation is very nearly constant.   If a peculiar mission were attempted,
however, the C G. shift could become considerably larger.

One last point should be noted.   The present helium tank volume is
adequate for normal mission temperatures, but lower temperatures
(approximately 245°R) probably would cause sufficeint helium depletion
that the final moments of the orbit inclination firing would occur in
a blowdown mode within the propellant tanks   This should not be viewed
with great alarm since it is rather easy to adjust the thermal design
slightly and obtain a higher average helium temperature
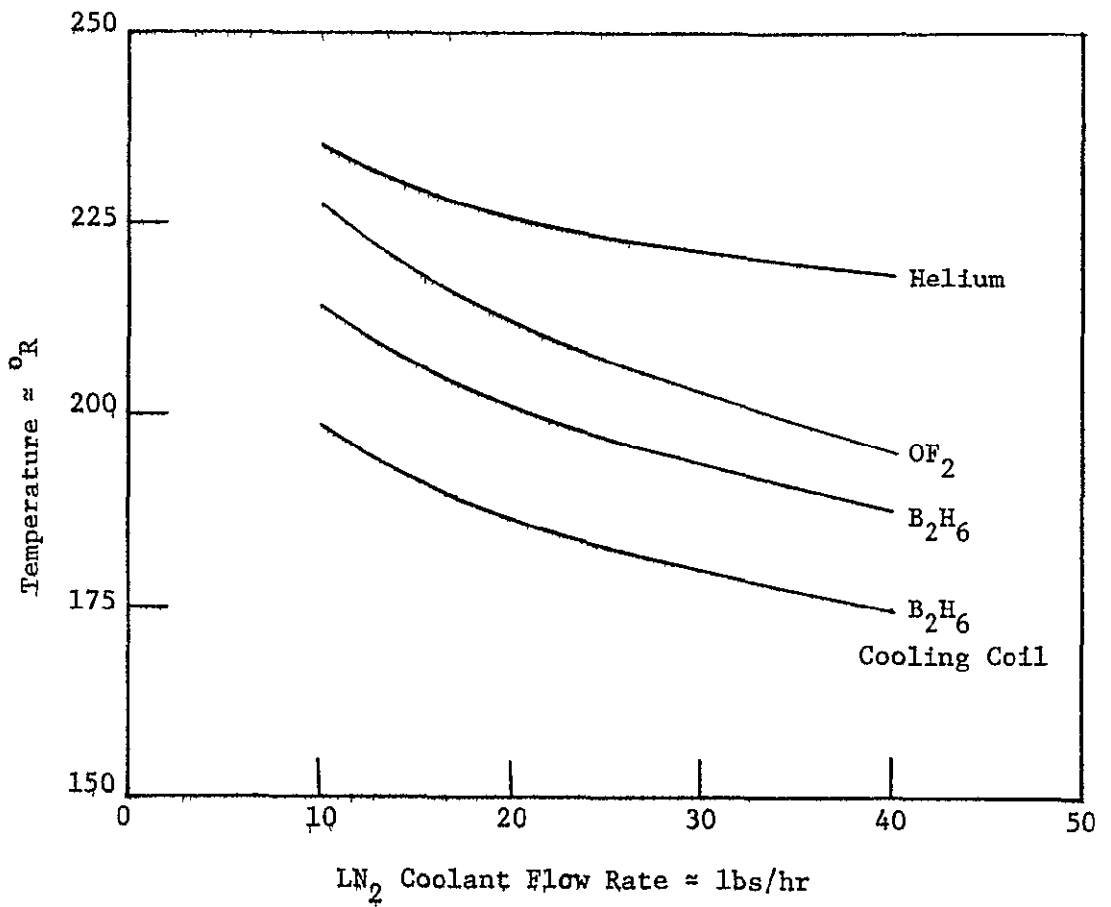
Figure 5-1 Equilibrium Temperature as a Function of LN_2 Coolant Flow Rate. Outside Film Coefficient = $1.0 \frac{Btu}{ft^2-hr-^oF}$
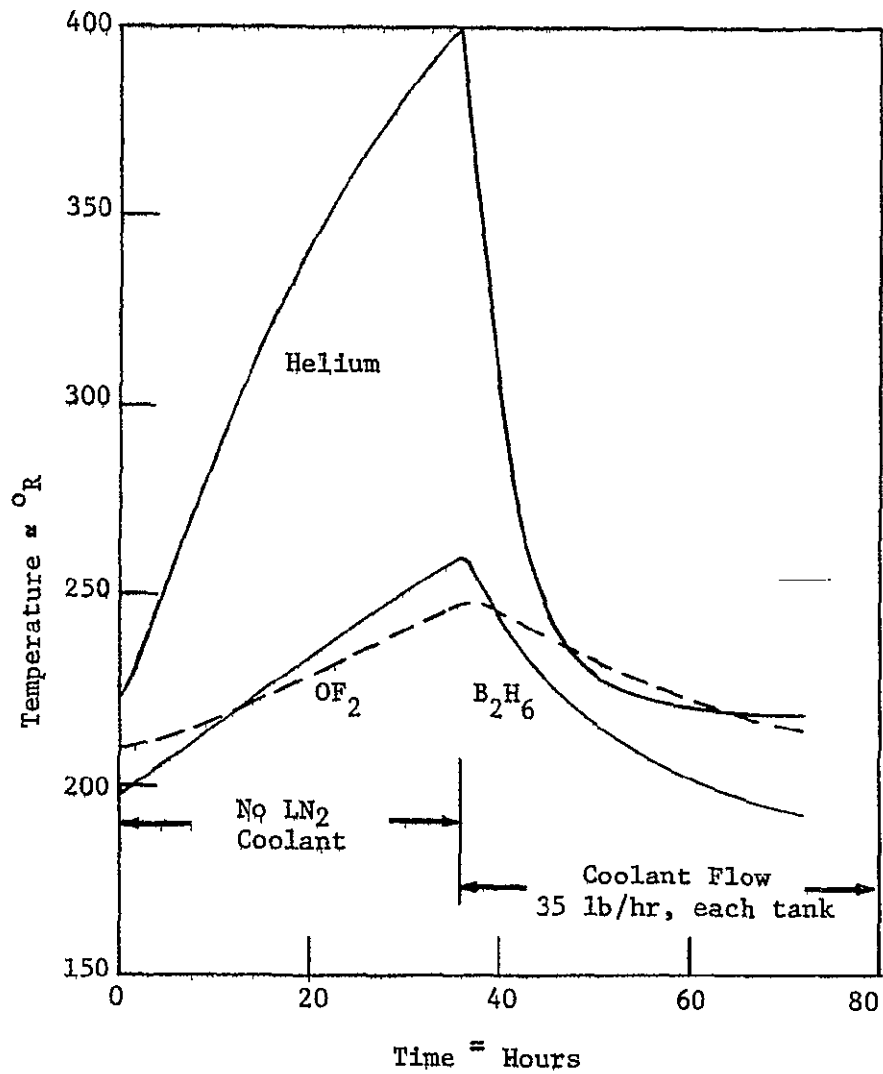
Figure 5-2   Transient Warm-Up and Cooldown
During Groundhold   Outside Film Coefficient $= 1.0 \dfrac{Btu}{ft^{2}-hr-°F}$

Notes: Helium and OF_2 Temperatures Controlled Between 220°R and 270°R by Immersion Temperature Sensors

B_2H_6 Maximum Temperature Controlled by Immersion Sensor Set at 270°R, and Low Temperature controlled by Sensor Attached to the Cooling Coil Set at 190°R
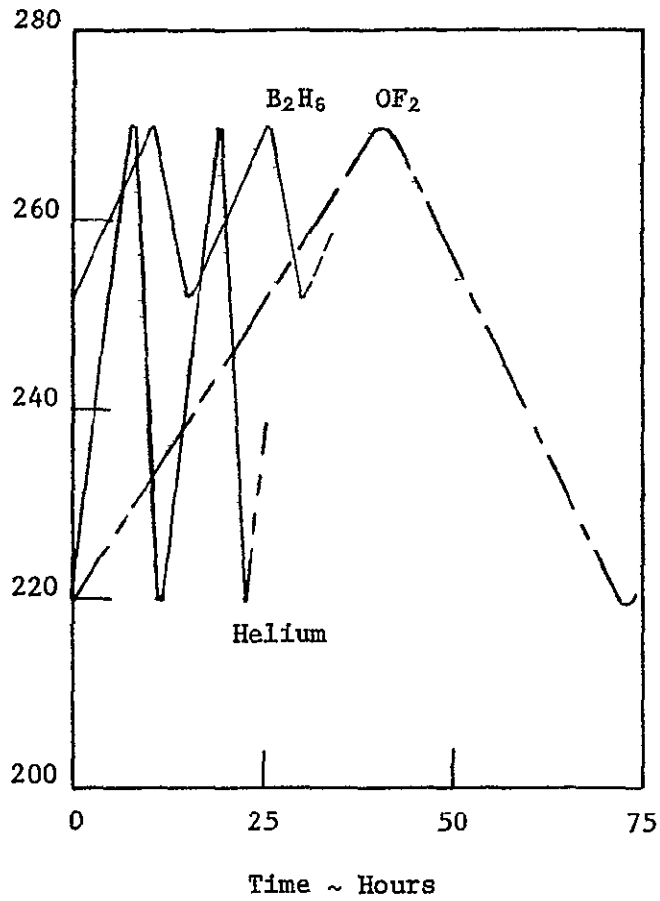
Graph "A"   Coolant Flow Rate of 1000 lbs/hr
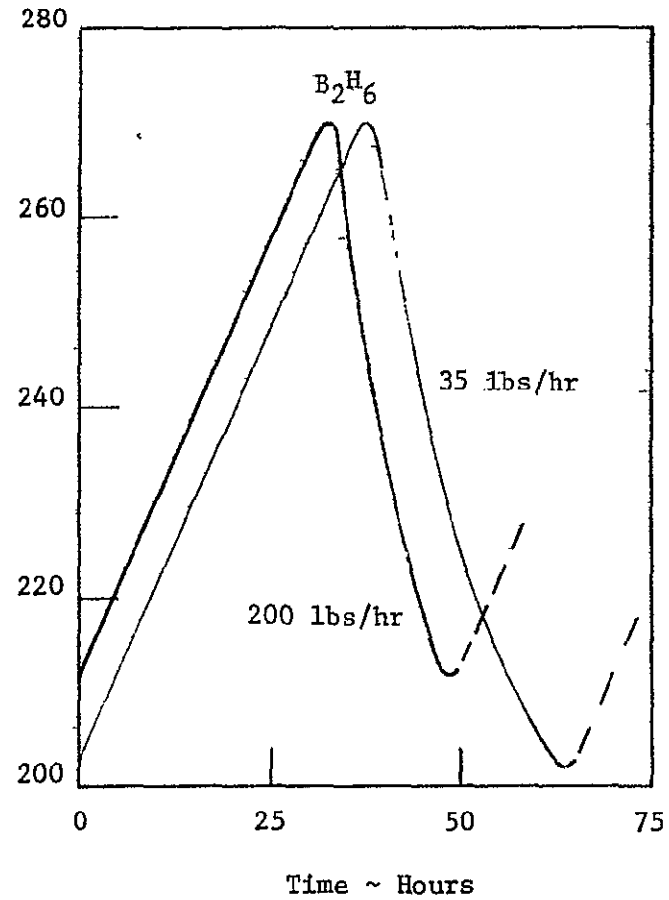
Graph "B"   Coolant Flow Rate of 35 and 200 lbs/hr

Figure 5-3   Fluid Temperature Characteristics with Cyclic Control
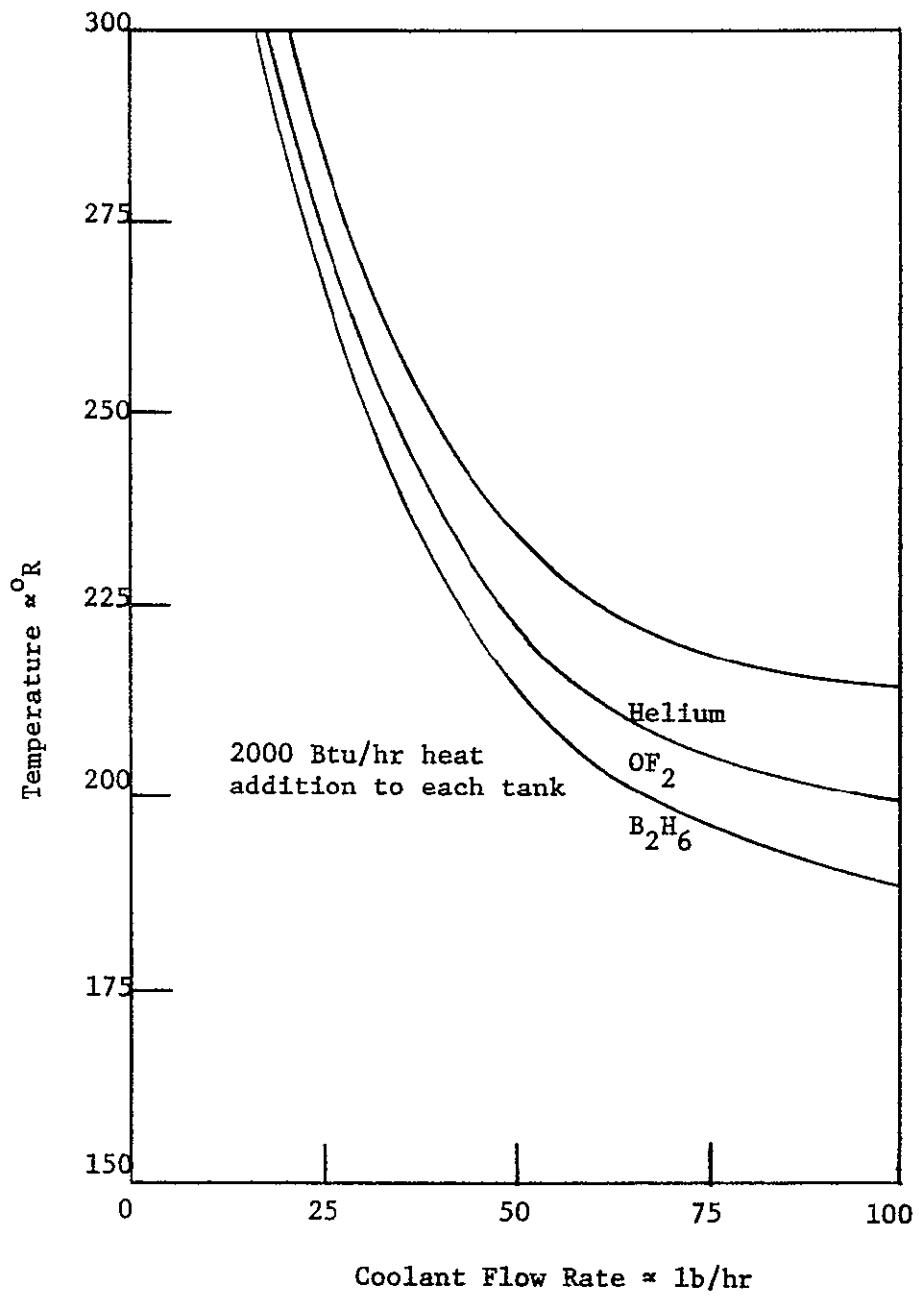
Figure 5-4 Tank Temperature With 2000 Btu/hr Heat Addition
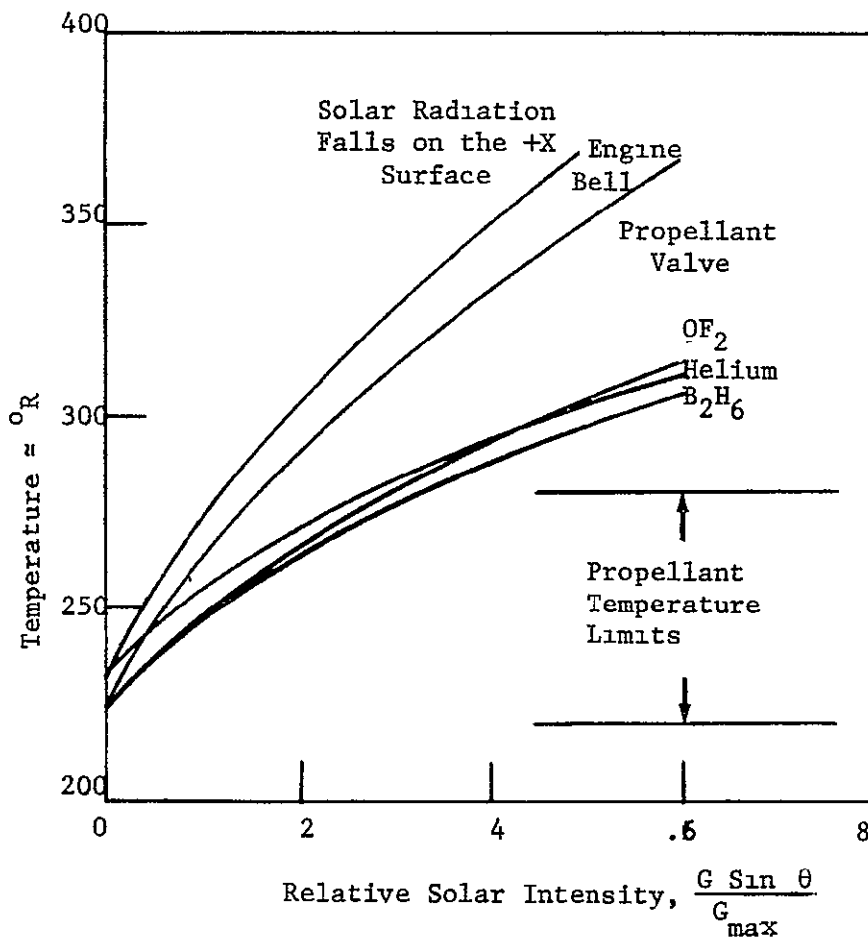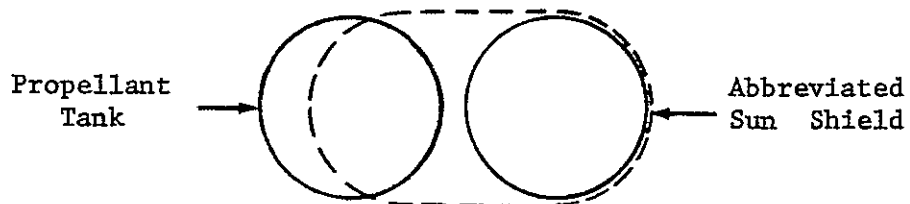
Figure 5-5 Steady State Temperatures
for Varying Solar Radiation, 960°R RTG, Abbreviated Shield
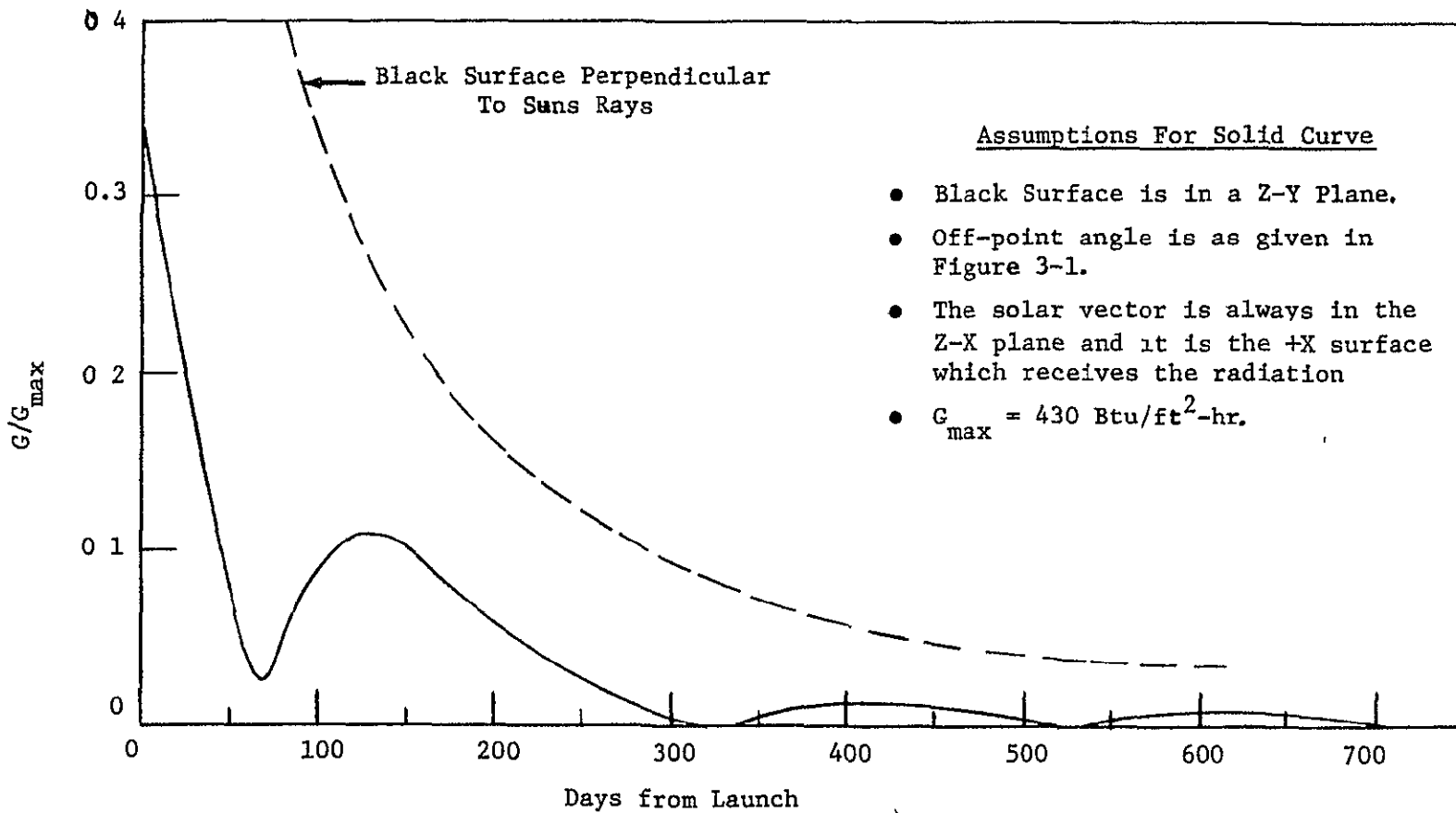
Figure 5-6   Relative Solar Heating Rate On a Black Surface During Normal Mission
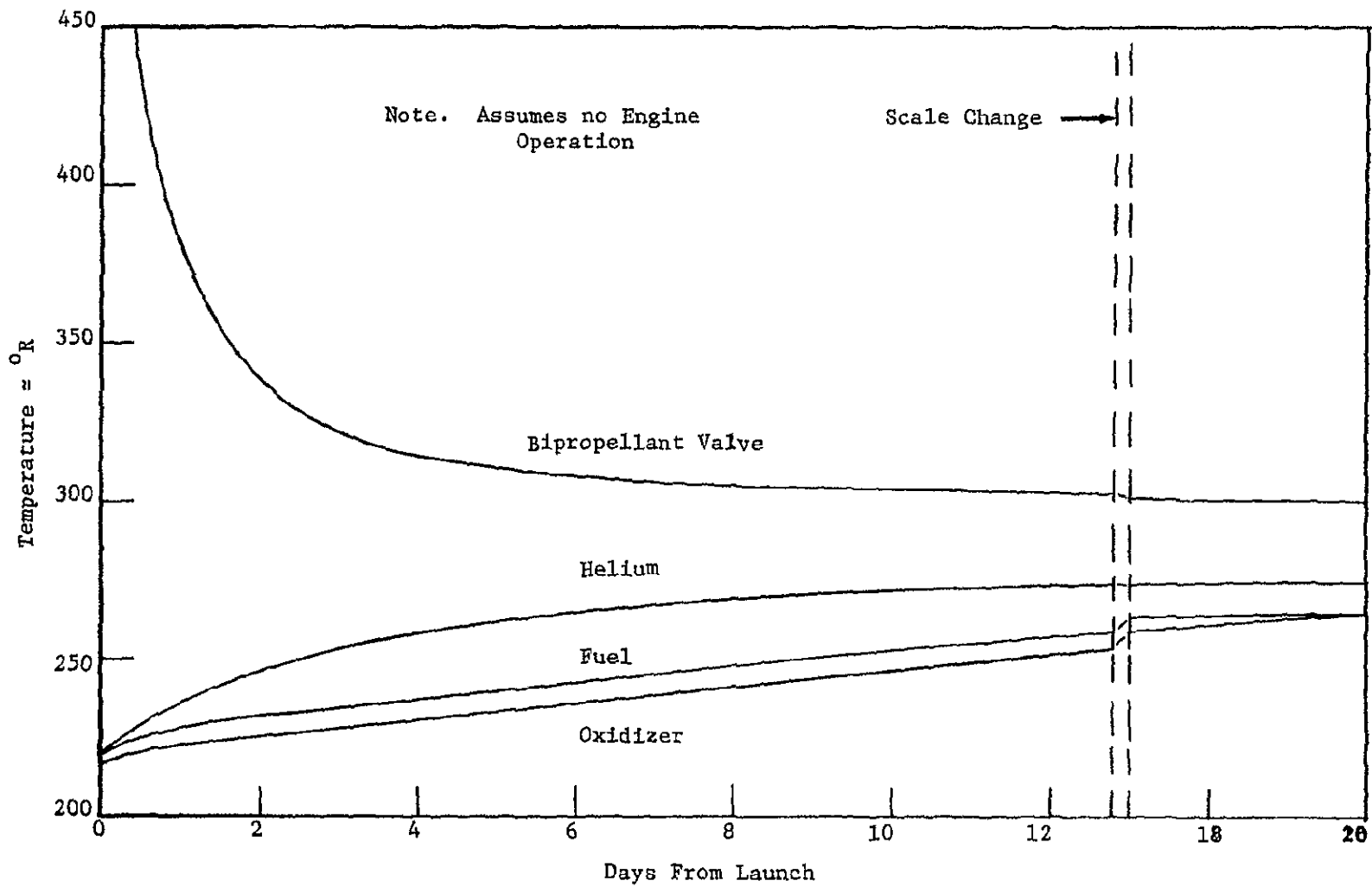
Figure 5-7 Module Temperatures After Launch on Normal Mission; 960°R RTG, Abbreviated Sun Shield

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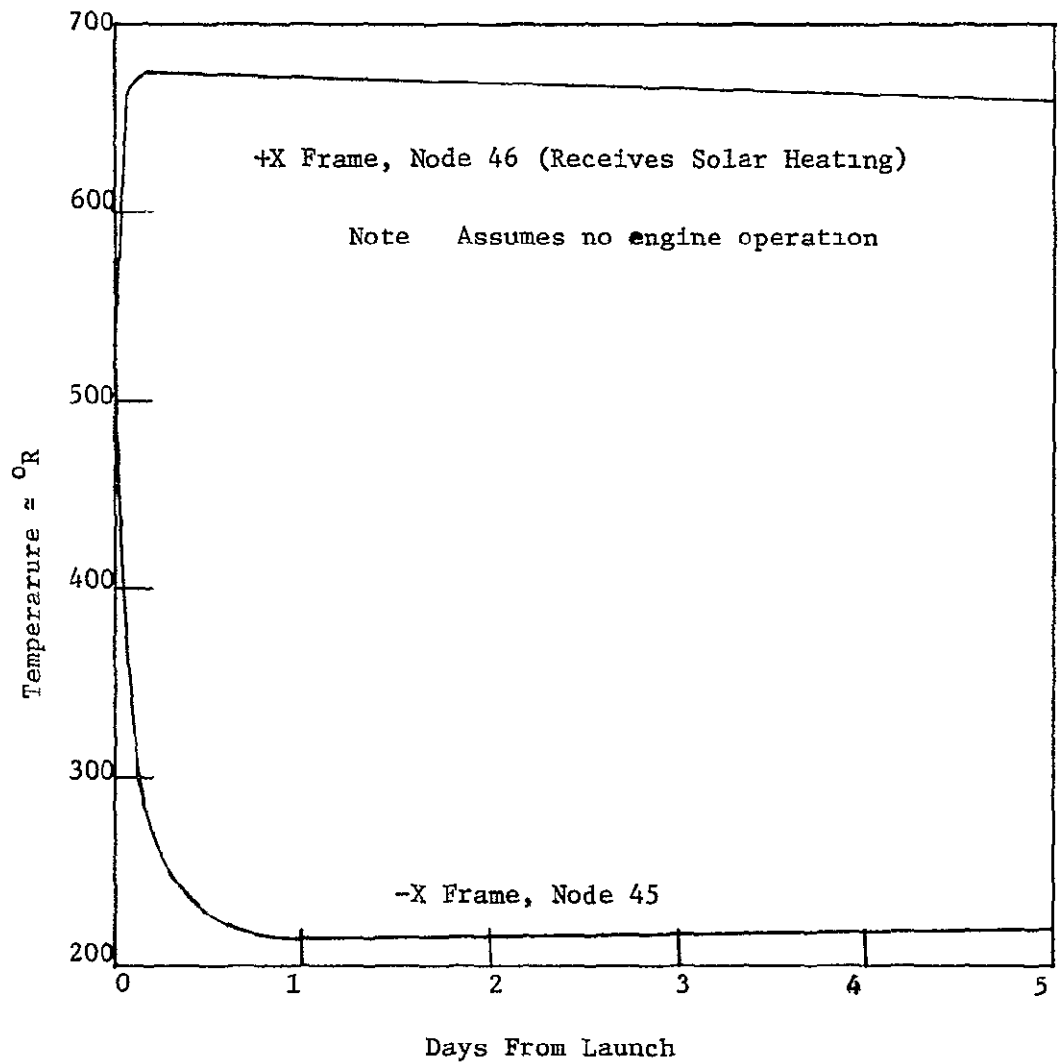
Figure 5- 8 Module Frame Temperatures After Launch on
Normal Mission; 960°R RTG,
Abbreviated Sun Shield

1. Solar heating ≈ 0
   (Shaded by spacecraft)
2. Earth Radiation Heating ≈ 0
   (Shaded by aft meteoroid shield)
3. Earth albedo heating ≈ 0
   (Shaded by aft meteoroid shield)

Earth

1. Solar heating ≈ 0
2. Earth radiation heating
   at maximum
3. Earth albedo heating
   past maximum (decreasing)

1. Solar heating = 0
   (shaded by earth).
2. Earth radiation heating ≈ 0
   (shaded by spacecraft).
3. Earth albedo heating = 0
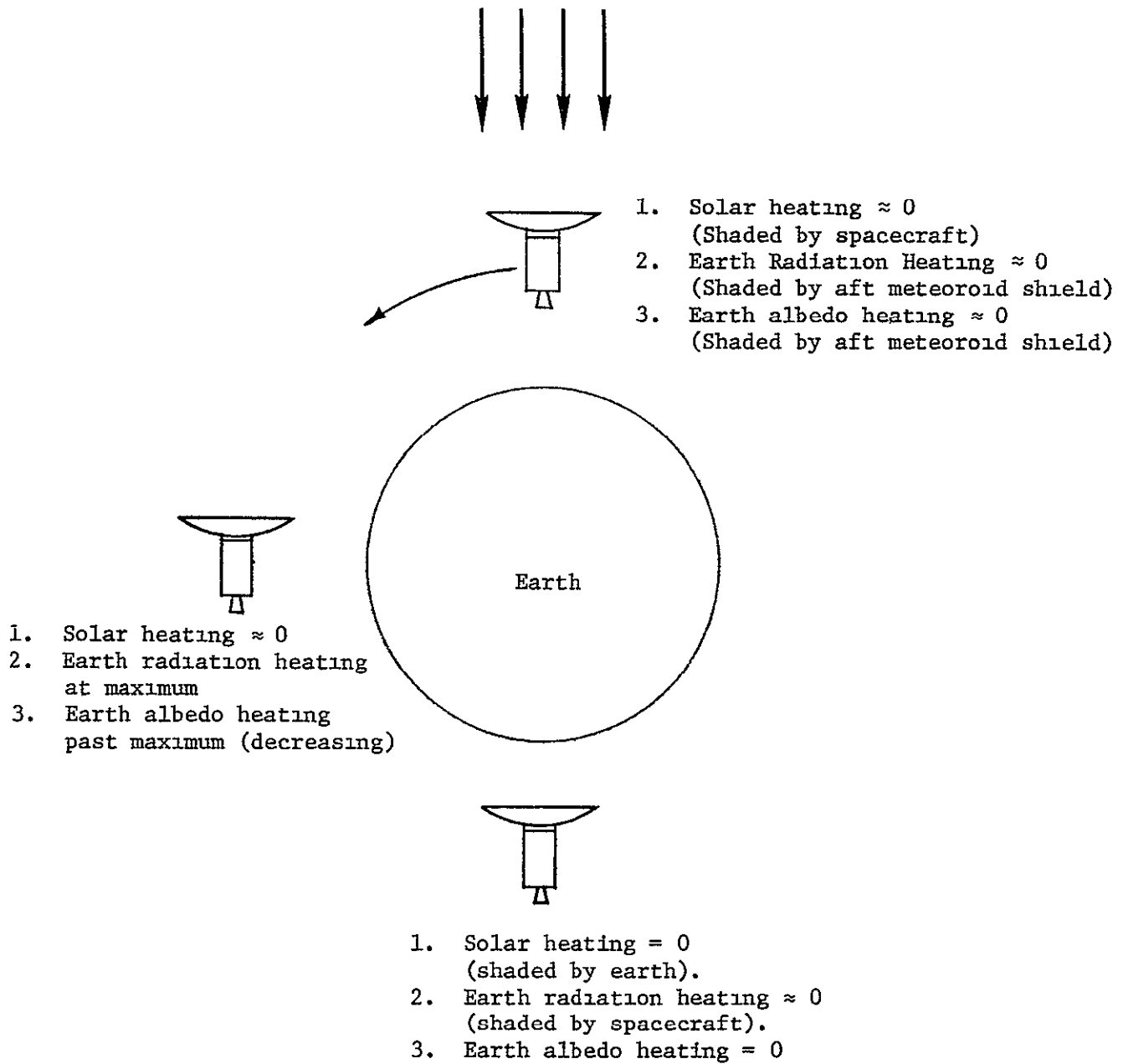
Figure 5-9   Sources of Module Heating
While in Earth Orbit

Note· These curves are for full helium and propellant tanks   For only partially filled
tanks, the response rate of the helium and propellants would be higher but the
propellant valve curve would remain unaltered
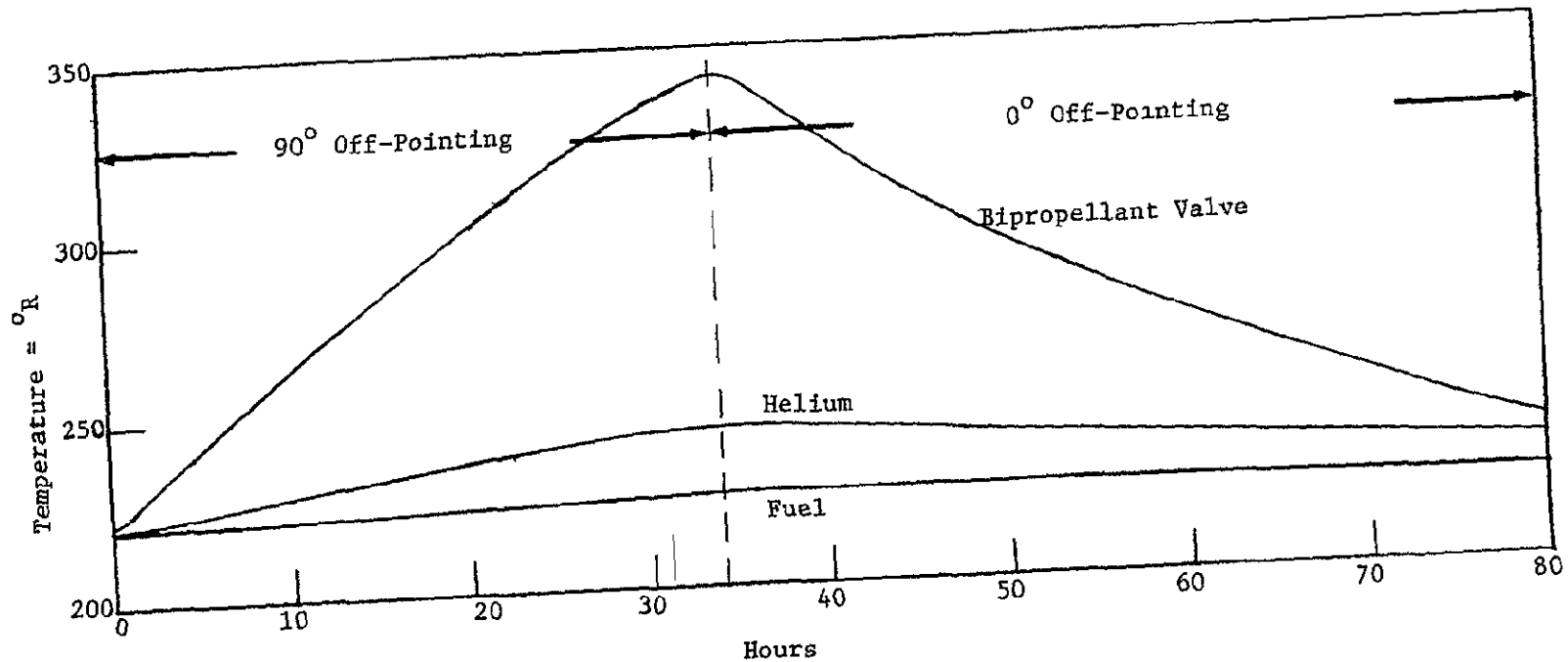
Figure 5-10 Effect of Orienting from 0° Off-Pointing to 90°
Off-Pointing While Near Earth
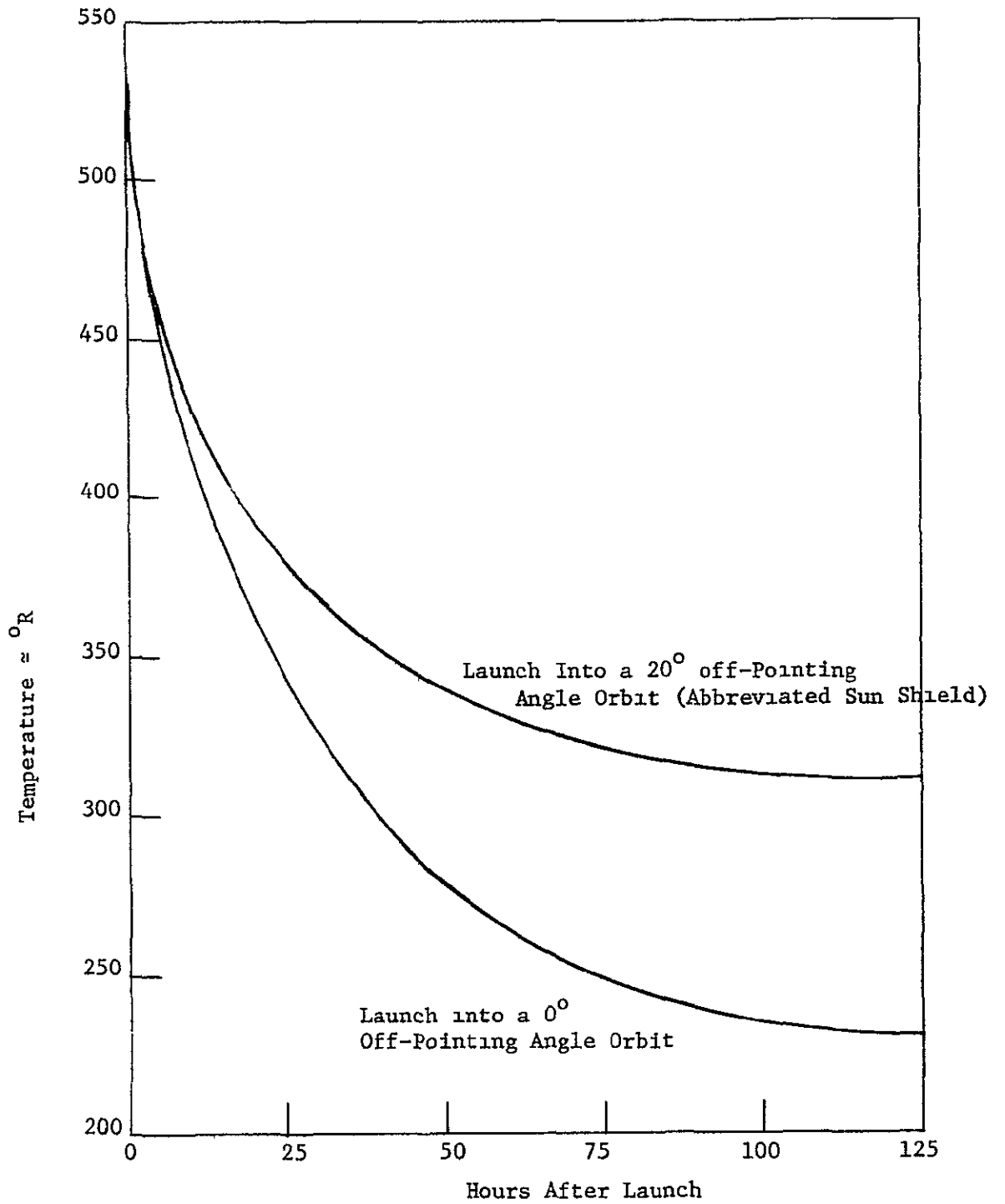
Figure 5-11 Temperature History of Bipropellant
Valve Immediately After Launch
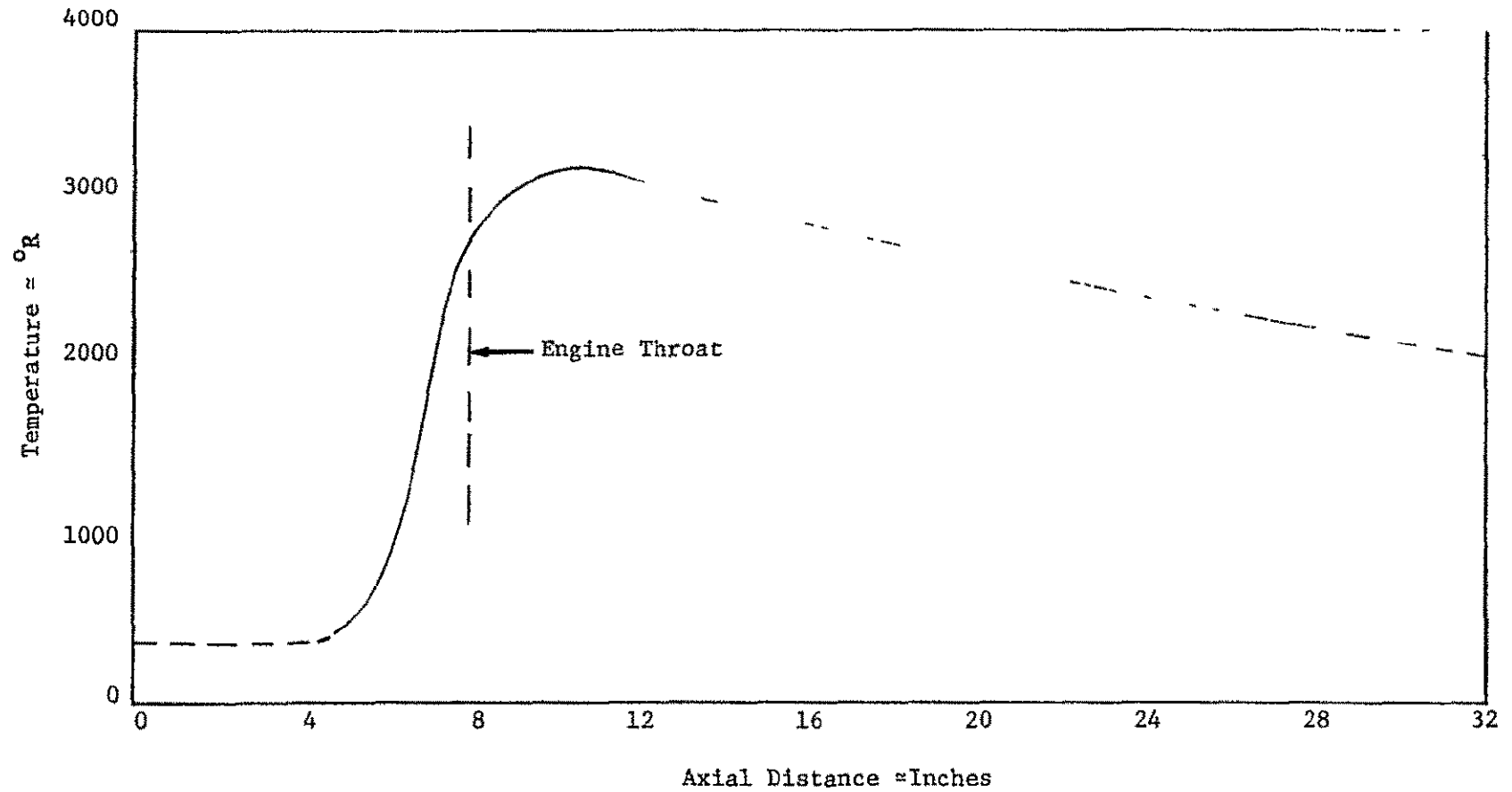
Figure 5-12 Outside Surface Temperature of Engine During Operation
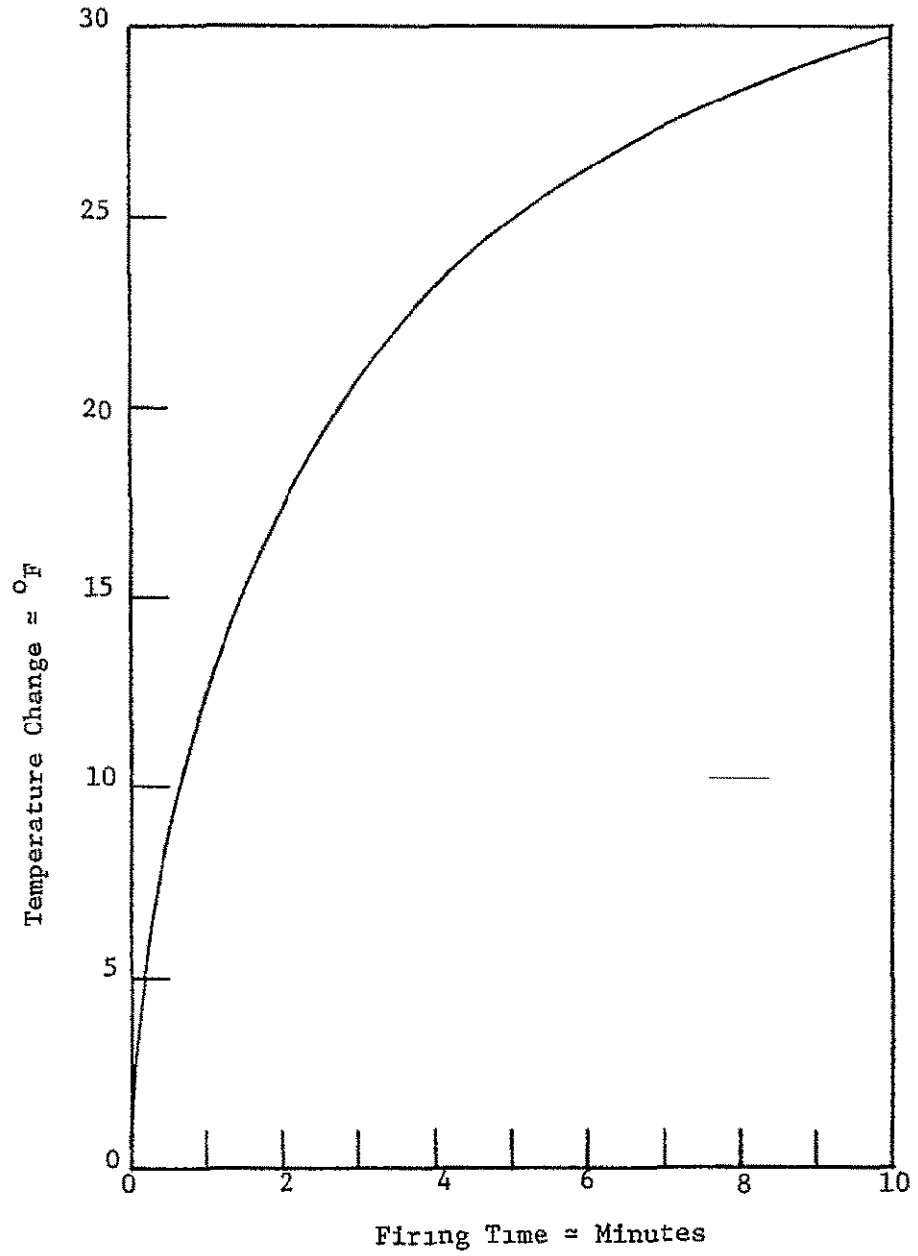
Figure 5-13   Helium Temperature
Drop During Engine Firing
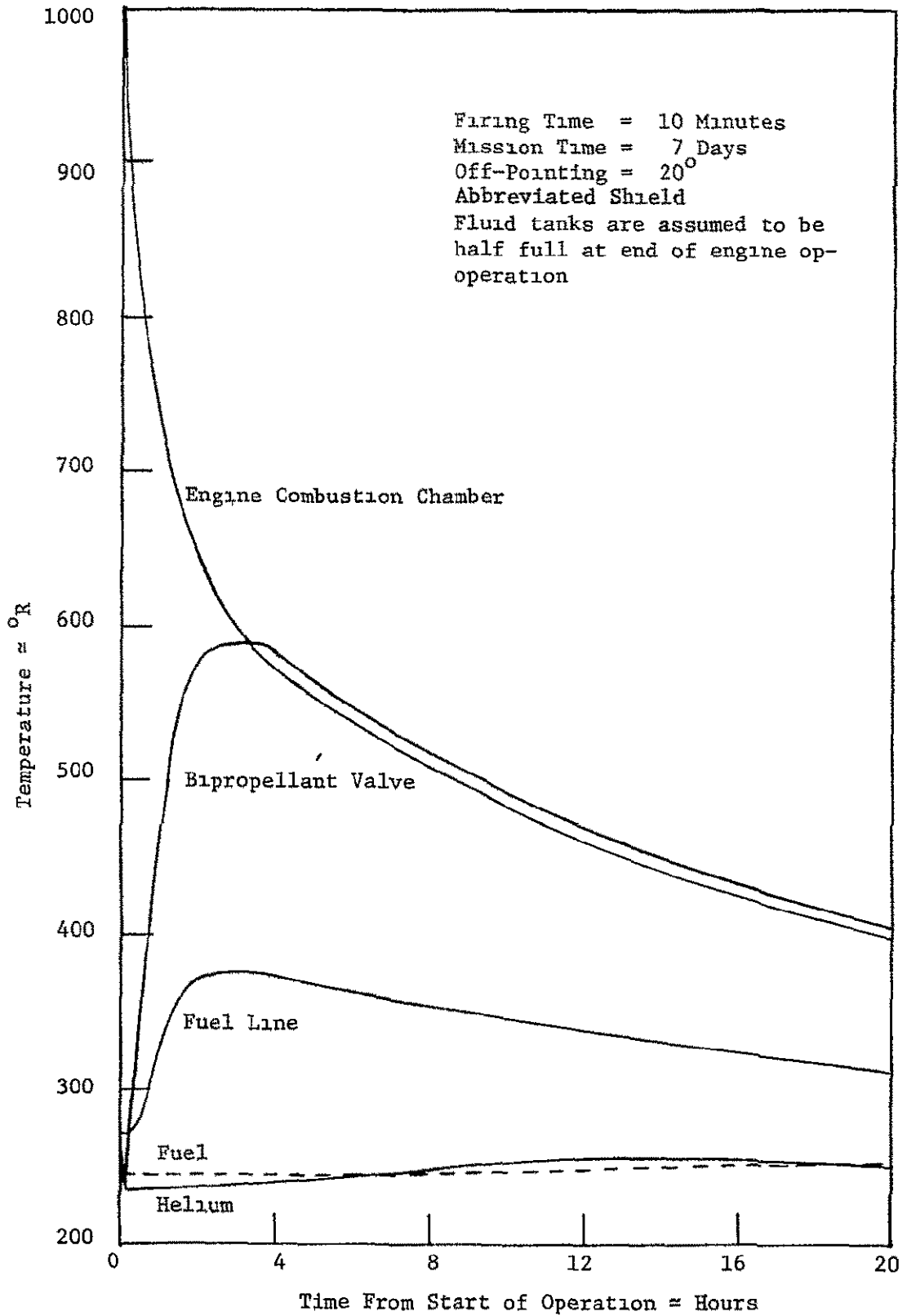
Figure 5-14 Temperature Histories During and
After Engine Operation

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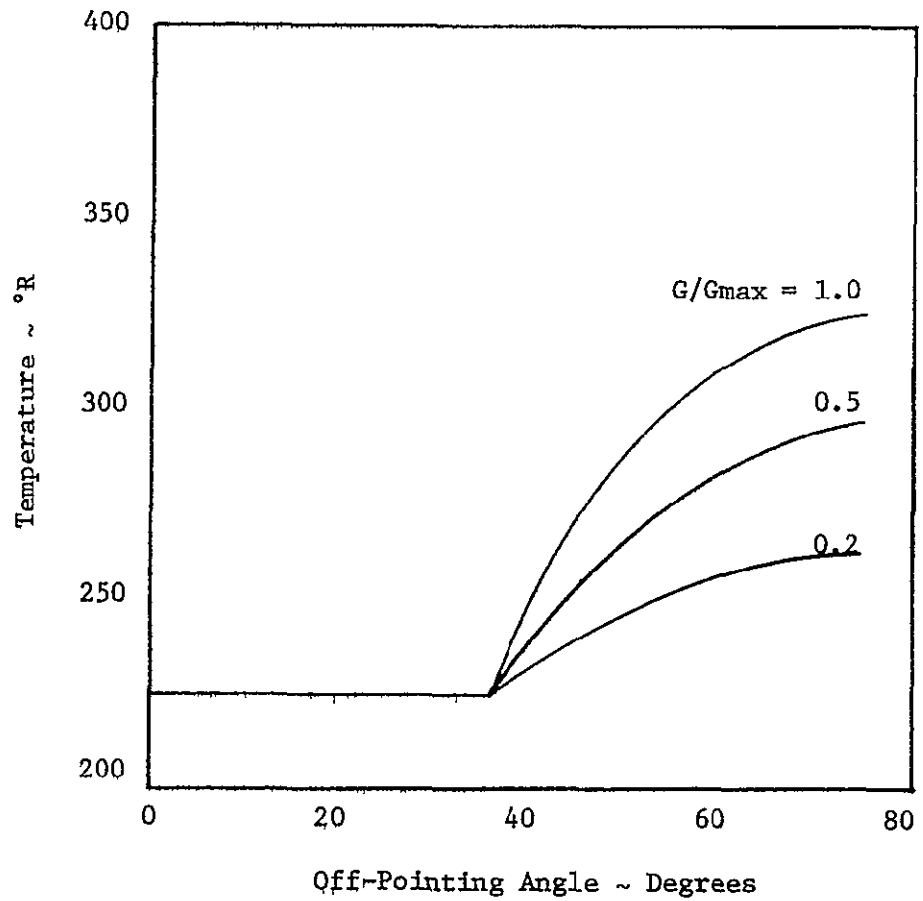
Figure 5-15 Steady-State Fuel Temperatures
for Varying Off-Pointing Angles,
Standard 10 ft dia. Shield, 960°R RTG
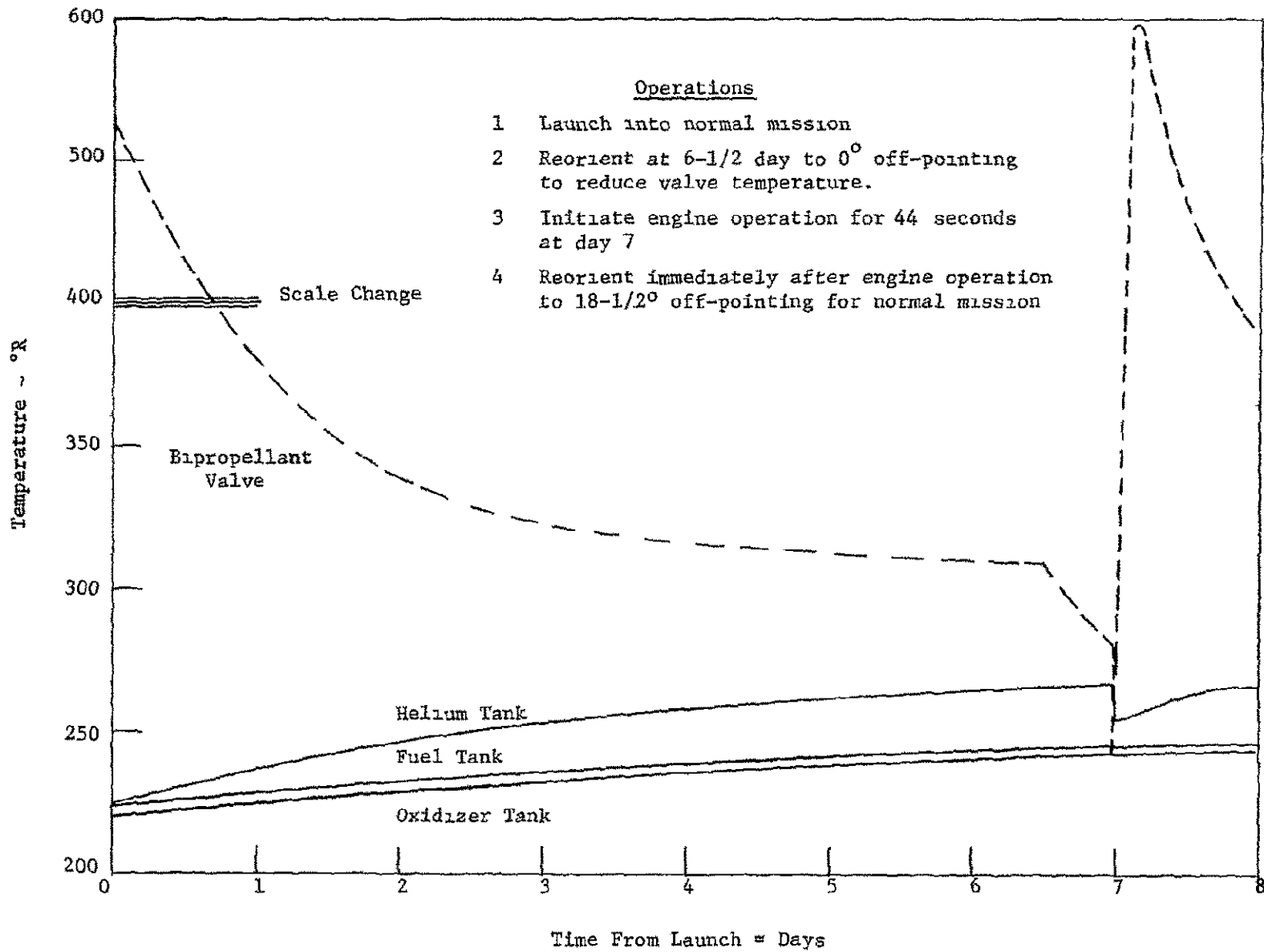
Figure 5-16A   Thermal History of Major Components During Normal Mission;   Module
with Abbreviated Shield
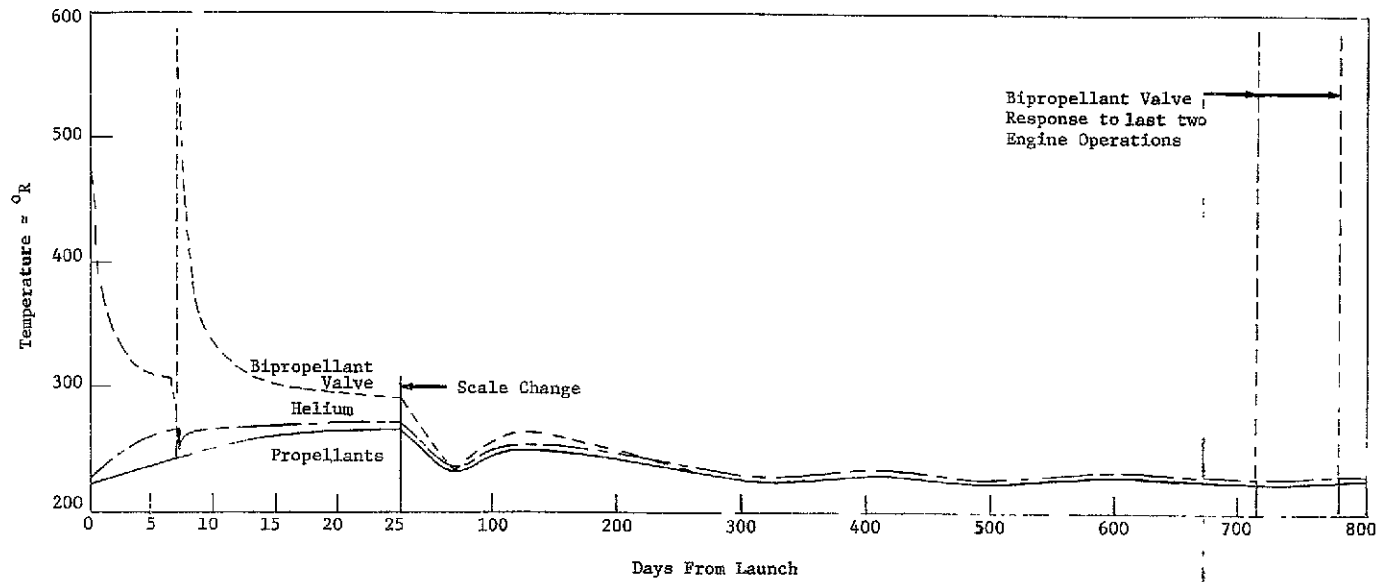
Figure 5-16B Thermal History of Major Components During Normal Mission; Module
with Abbreviated Shield
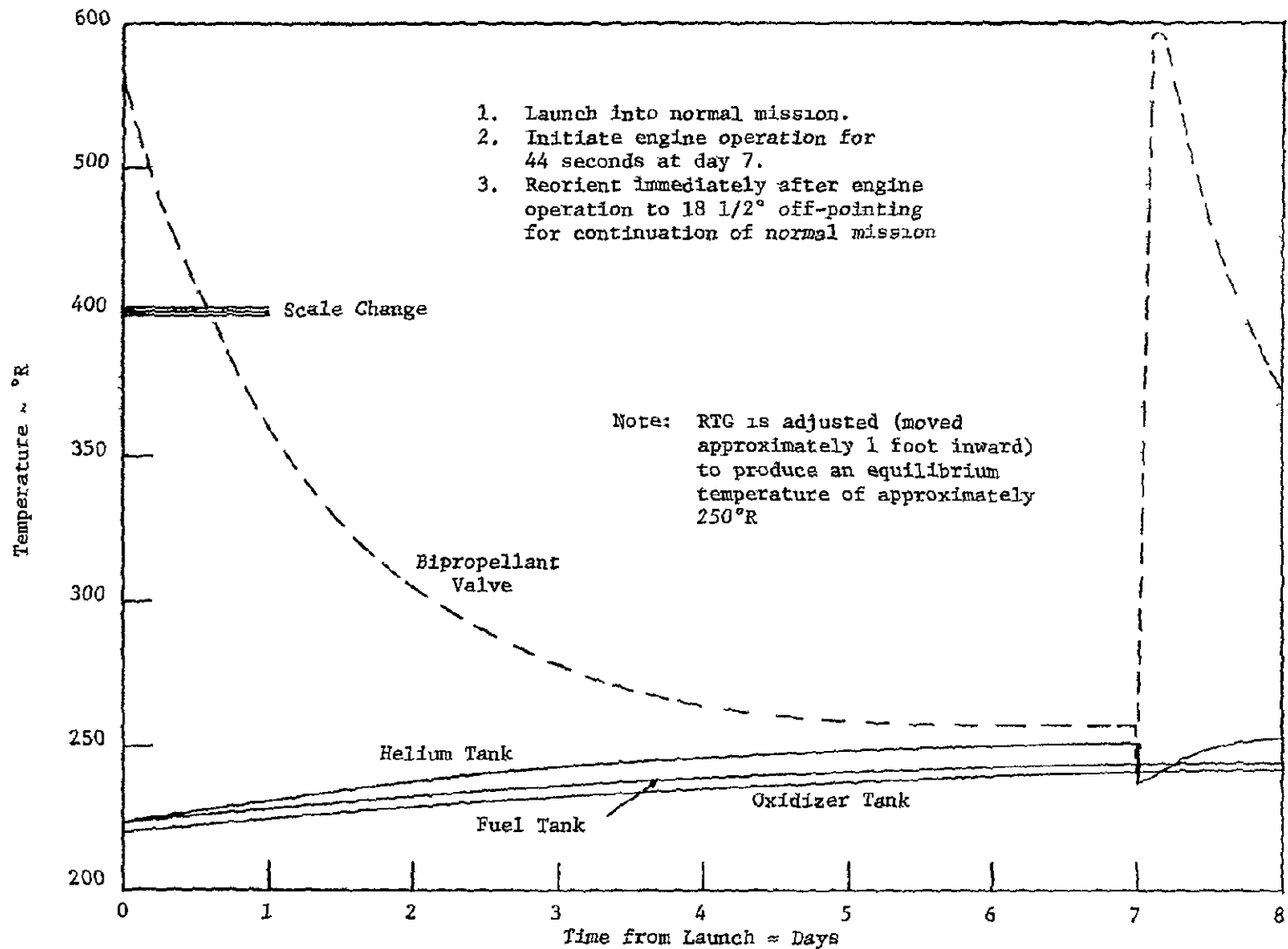
Figure 5-17A   Thermal Histories of Major Components
During Normal Mission;   Standard Shield
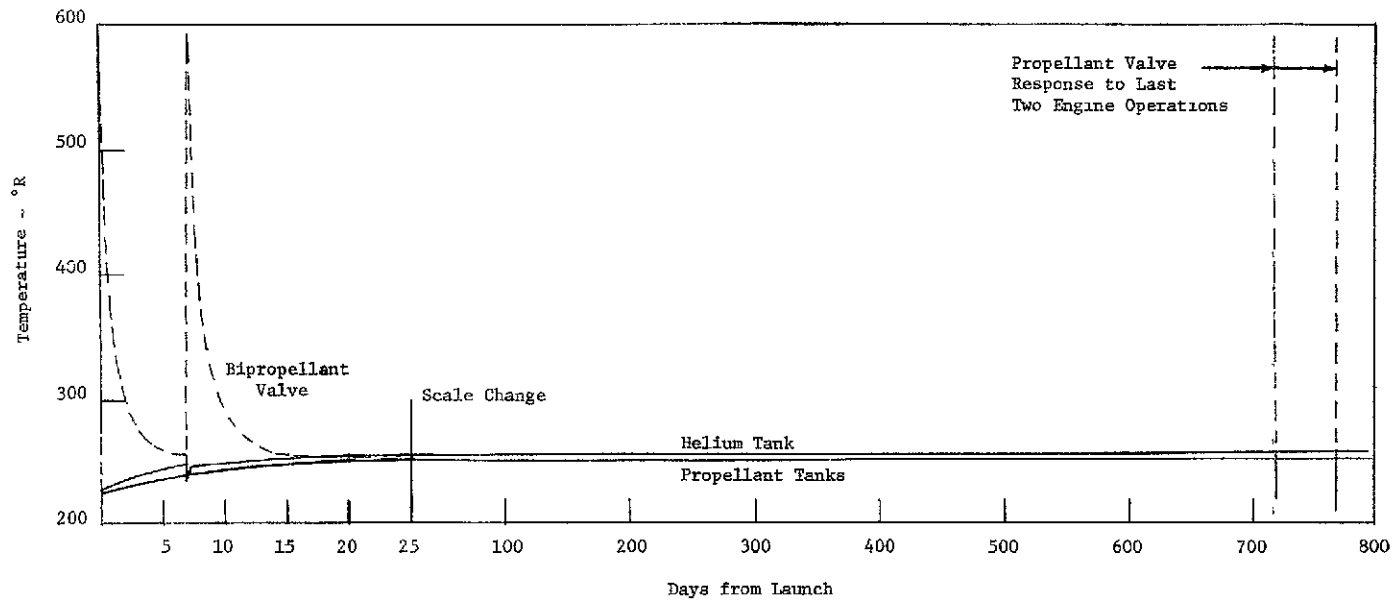
Figure 5-17B Thermal Histories of Major Components
During Normal Mission; Standard Shield
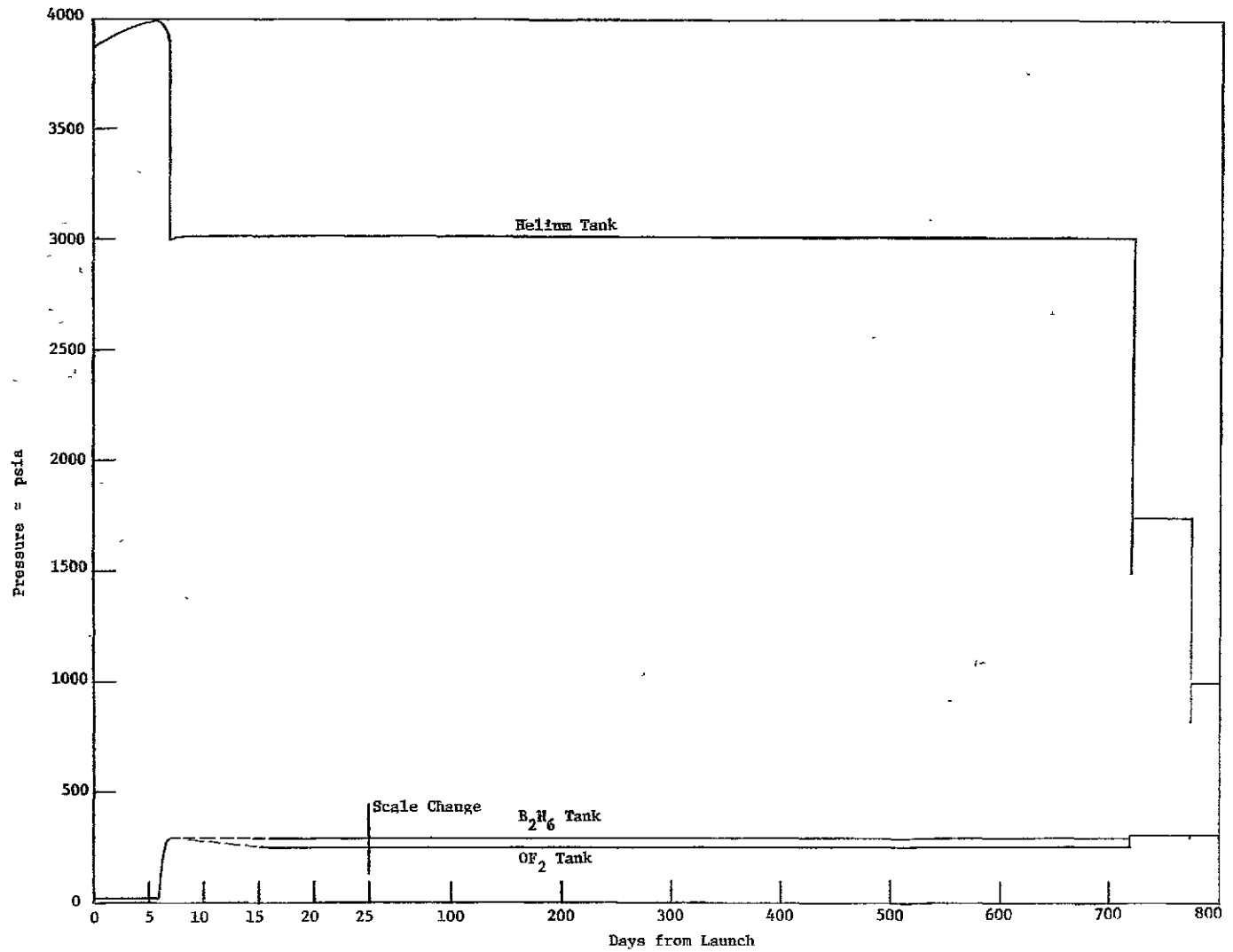
Figure 5-17C Pressure Histories of
Major Components During Normal Mission: Standard Shield
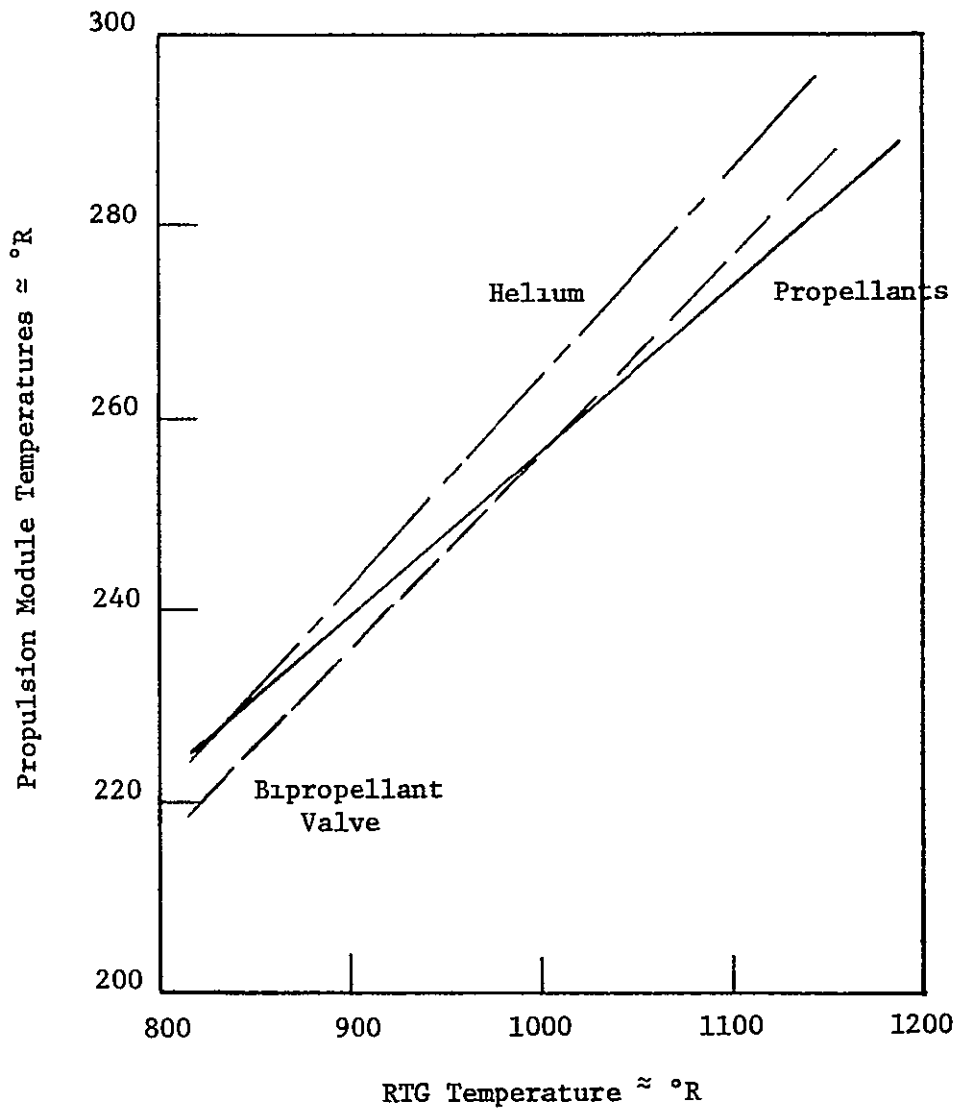
Figure 5-18   Effect of Varying RTG Temperature on
Quasi-Steady State Module Temperature,   Standard   Shield
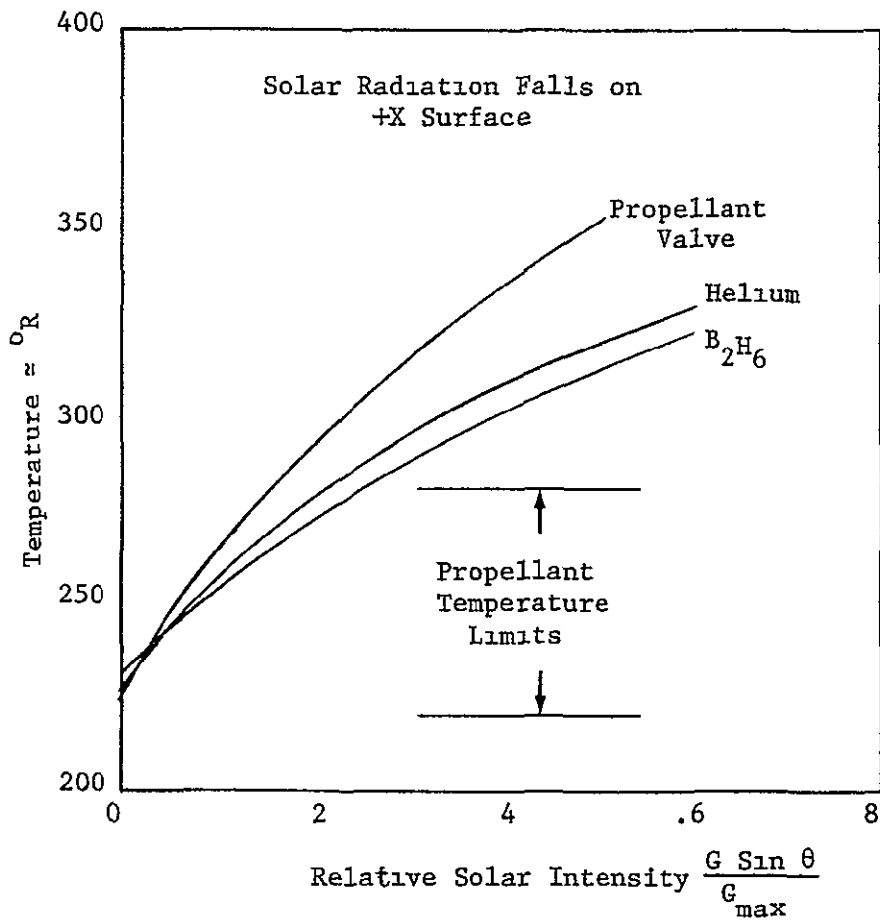
Figure 5-19   Steadystate Temperatures For
Varying Solar Radiation, 960°R RTG, Louvers Replaced by Radiator Plates
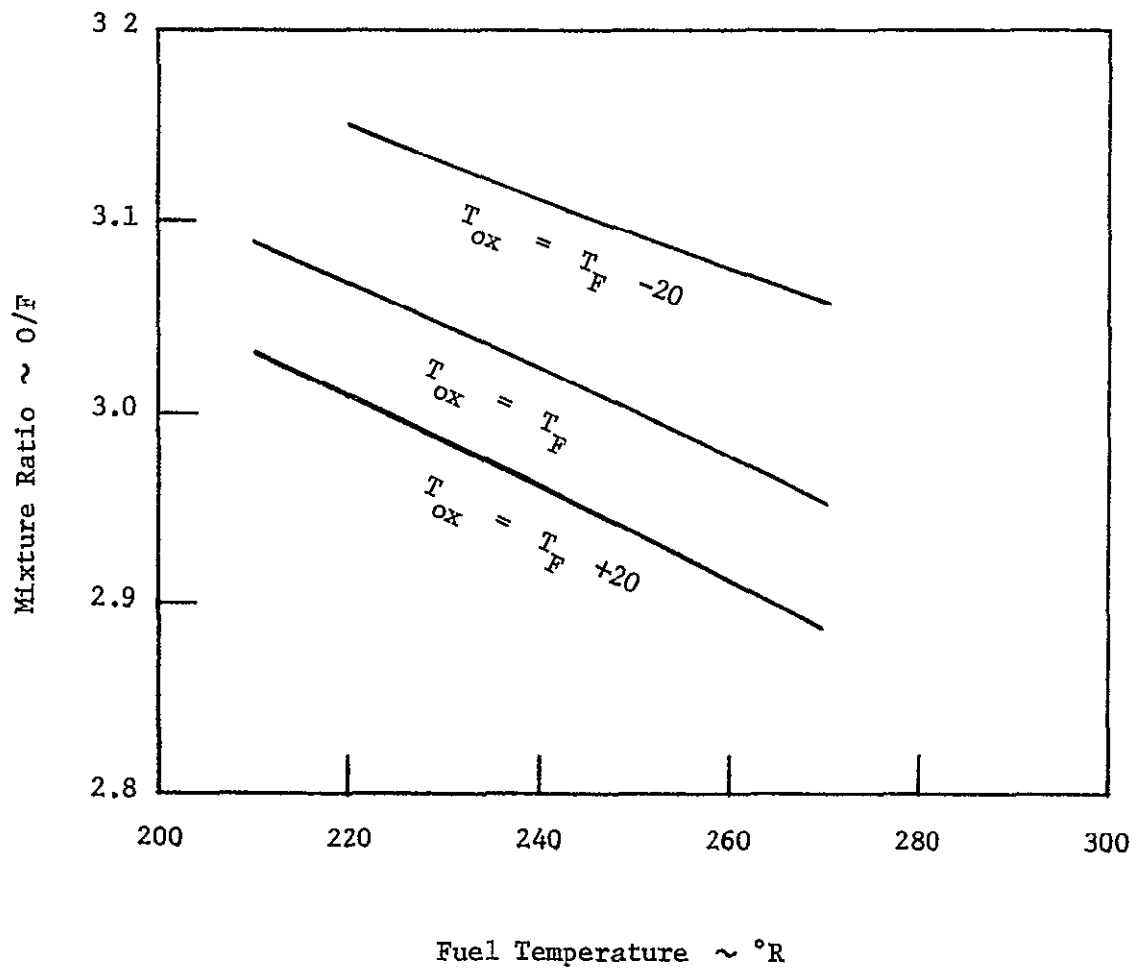
Figure 5-20 Mixture Ratio vs.
Propellant Temperatures
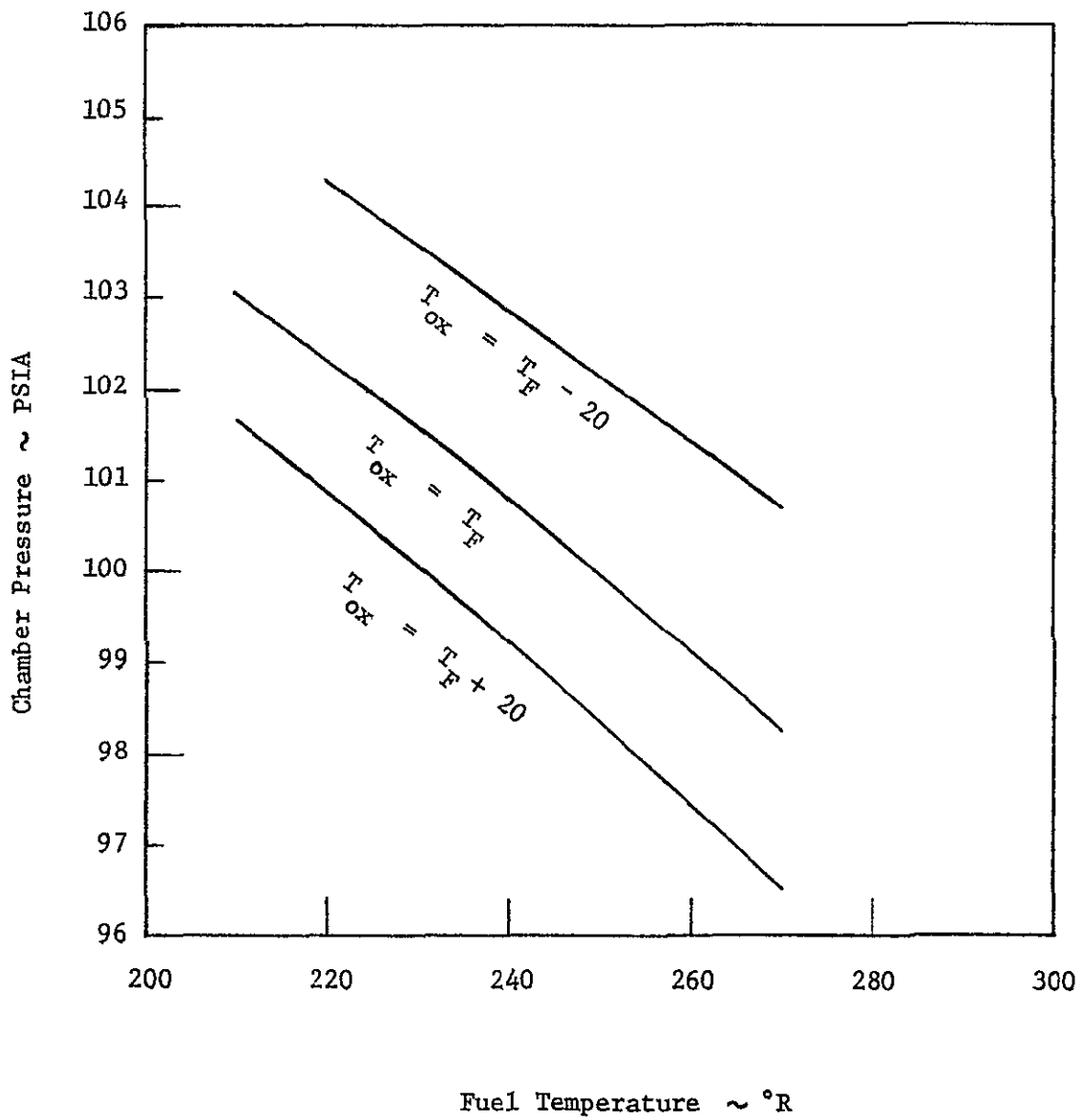
Figure 5-21   Chamber Pressure vs.
Propellant Temperatures

6 0 CONCLUSIONS AND RECOMMENDATIONS

From the data that has been presented, it is possible to draw several important conclusions

1) Liquid nitrogen circulated through coils, submerged in the fluids inside the tanks, has ample capacity to maintain the fluids within their prescribed temperature limits during groundhold even if a sizeable insulation failure occurs.

2) Freezing of the fuel on the surface of the cooling coil is possible if the average fuel temperature is sufficiently near the fuel freezing temperature and the velocity of the liquid nitrogen in the coil is too high.

3) To avoid fuel freezing, it will be necessary to design the liquid nitrogen supply system so that it will supply the nitrogen in liquid form at a rate not exceeding 1500 lbs/hr For better temperature control, the liquid nitrogen flow rate should be variable with a minimum flow rate of about 175 lbs/hr and a manual override

4) To avoid freezing of the fuel, it will also be necessary to control the on/off flow of the liquid nitrogen by temperature sensors The sensor for controlling when flow is initiated should be immersed in the fuel just below the surface. The sensor for controlling flow stoppage should be attached to the coil at or near its lowest point

5) To avoid frost or moisture collection on portions of the frame, foam insulation will be required on all members and lines which contact the three tanks. Metal parts must be insulated for a distance of about one foot from the tanks and non-metal members insulated for about four inches.

6) The helium and propellant temperatures during the first six or seven days of flight are, to a large extent, determined by the temperatures which exist at launch because of the fluid heat capacitance

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7) The allowable steady-state, off-pointing angle which may be tolerated for the first 100 days is dependent on the amount of sun shielding provided For the module as designed, the allowable off-pointing is approximately 20° at launch, increasing to 90° about day 100

8) The module will be kept within the desired temperature limits if the louvers are replaced by radiator plates However, the margin of safety will be slightly less.

9) The design of the pin joints at the bottom of each tank is not critical from the standpoint of thermal control Since it is necessary to insulate the attaching frame as indicated above in (5), a joint having a high thermal conductance does not result in excessive heat transfer via the frame because of the frame insulation

10) Unless it is allowable to initiate engine operation when the valve temperature is above the allowable fuel temperature, the first engine firing may not be made prior to day 3 because it takes that long for the valve to cool down from the ambient temperature of groundhold In addition, if the module is launched into an orbit which causes the engine bell to be exposed to solar radiation, it may be necessary to orient the craft in a 0° off-pointing position for several hours prior to engine firing in order to drop the valve temperature.

11) Heat soak-back from the engine after engine firing is of minor consequency except for the bipropellant valve and connecting propellant lines The valve will rise to approximately $600^o R$ and small amounts of propellants will be vaporized and return to the tank to avoid high pressure.

12) The propellant lines and other control components will stay within the required temperature limits.

13) For the normal mission, the propulsion system functions properly with sufficient helium for the mission

In the light of the conclusions and also the basic conclusions reached in the previous task reports, References 1 and 2, the following recommendations are given.

1. Radiator plates should be used and not louvers   This choice i based upon four items

    a)   The amount of sun shielding provided.

    b)   The degree of off-pointing

    c)   The complexity of the removable louver insulation design

    d)   The expected variation in RTG temperature

The added capability to accommodate mission and/or design varia-tions which is furnished by the louvers is small   This marginal increase might be justified if a smaller shield were to be used But with the large shield and the mission as now contemplated, the inclusion of louvers adds little to the overall module capa-bility.   In addition, when louvers are used, a rather complicated mechanism for removing the insulation on top of the louvers must be provided and this is a decided disadvantage   It is true that the insulation will still have to be removed if a radiator is used, but the mechanism can be substanially less complicated

It should also be pointed out that it it becomes desirable to exercise greater control over the module temperature, it will a also become necessary to consider using louvers which are located near the RTG but which still sense and operate according to the propellant temperature

2   The standard sun shield (10-foot diameter) should be used   Al-though a much smaller shield is acceptable and lighter, the larger shield allows considerably more latitude in mission op-eration   More important, however, is that with a larger shield the thermal control system has a much larger safety factor

3   To establish that a reliable flight model has actually been designed, it is recommended that a series of module thermal

control tests be made. These tests should have the following objectives·

o Establish operational capability for the hottest portion of the mission

o Establish operational capability for the coldest portion of the mission.

o Establish the simplest manner of "trimming" the craft for a particular mission

o Establish operational characteristics during ground-hold

In conjunction with this last item, it is earnestly suggested that a simple groundhold test be made which utilizes only a single insulated container   This test should investigate thermal gradients in the fluids during groundhold and methods of LN_2 coolant control   If properly run, this small initial test will insure the success of the module thermal test and actually save funds and time

4. A study program should be initiated which investigates boiling of the propellants in the feed lines upon engine shutdown Side effects such as vapor venting back into the tanks should be considered. However, this study should not be initiated until the bipropellant valve design is started

5. The engine thrust structure should be designed to stabilize the bipropellant valve temperature Ordinarily, the gimbal brackets and the gimbal mounting supports are designed to prevent heat transfer to the structure from the engine during firing From the standpoint of thermal control of the module, this is of little concern since the entire thrust structure is effectively isolated from the rest of the module by the long, thin boron filament tubes which support the thrust sturcture It is impossible to state explicit design guidelines at this time, since they are highly dependent upon the design of the valve and the manner in which it is secured to the engine

6　Attention must be given to the helium control system　In Sections
4 2 and 5.3, it was pointed out that pressurization, sufficiently
in advance of firing to allow thermal equilibrium to be achieved
prior to the firing, may conserve helium　However, to accomplish
this, the helium valves must be reusable (not squib type) and
must be leak tight.　This may be difficult to reliably achieve
Obviously, unless the quantity of helium which can be saved is
"appreciable" there is no advantage to early pressurization　The
quantity of helium which would constitute an "appreciable" amount
can be determined only be considering all facets of the module
including its cost to development and manufacture　From the
calculations made, it would seem that the helium savings would
at most, be one to three pounds　With this amount of saving in
helium consumption, it is hard to justify going to the effort to
save it　But, in some cases, it could be critical　It is there-
fore recommended that as soon as the mission parameters are firm,
a study in the area of helium conservation and required components
be made.

7.　A study should be initiated which will have the objective of es-
tablishing a reliable means of remotely controlling louvers.　If
such a mechanism existed, louvers could be placed at their most
effective position (near the RTG) and still fluid temperatures
could control their operation　This would be a tremendous ad-
vantage in that very little restriction by thermal control, if
any, would be placed on the module design (shield requirements)
and module orientation relative to the sun

REFERENCES

1. TRW Report No. 14051-6004-TO-00, "Summary Report, Task III Space Storable Propellant Module Environmental Control Technology," R. E. DeLand, O. O. Haroldsen, and R. N. Porter, dated 1 June 1970.

2. TRW Report No. 14051-6002-RO-00, "Summary Report, Task II Space Storable Propellant Module Environmental Control Technology," R. E. DeLand, O. O. Haroldsen, and R. N. Porter, dated 1 April 1970

3. Heat Transmission, W. H. McAdams, McGraw-Hill Book Co., 1942.

4. "Stratification in a Pressurized Container with Sidewall Heating," Joseph M. Ruder, AIAA Journal, Vol. 2, No. 1, January 1964

5. Convair, San Diego Report No. 7-9516 Model 7, "Heat Transmission in Liquid Oxygen," H W Wigner and T H Buckley, dated 1955.

6. TRW Report No. 14051-6006-TO-00, "Propulsion Module Thermal Test Plan," O O. Haroldsen, to be issued

7. "Pressurization Systems Design Guide, Volume III," McDonnel Douglas Report No. DAC-60510-F1, Contract NAS 7-548, July 1968

8. "An Analytical Investigation of the Blowdown and Charging Processes in a Single Gas Receiver, Including Effects of Heat Transfer," Technical Report No T-1, Stanford University, Contract NF-065-104, dated 1 October 1955.

9. "Calculation of Real-Gas Effects in the Depressurization of Air Storage Cylinders," Journal of Spacecraft and Rockets, Volume 7, No. 7, T. G. Keith, Jr., dated July 1970

10. "Design Guide for Pressurized Gas Systems, Anonymous, Contract NAS 7-388, IIT Research Institute, dated March 1966.

11. "Determining Pressure Drop in Flexible Metal Hose," Machine Design, C. M. Daniels and J. R. Cleveland, dated 25 November 1965.

APPENDIX A   THERMAL COMPUTER MODEL DETAILS

1.0   NODAL ARRANGEMENT

The thermal computer model is a nodal model of the propulsion module shown in Drawing SK 406876 included at the end of this Appendix. A general description of the module is given in Section 2 of this report   A description of all nodes is given in Table A-1

The computer model is complete from the standpoint of a thermal analysis. It is not complete from a structural viewpoint. For example, representative nodes for the small boron filament struts which support the engine thrust frame from the main platform are not provided   The reason for this is that thermally, they are of no consequence since they are small and have extremely low thermal conductivities   The components located at the base of the oxidizer tank (isolation valve, check valve, filter, etc.) are represented by a node as is the oxidizer line leading to the engine propellant valve   However the comparable equipment for the fuel is not nodally represented since such a deletion does not materially alter the thermal characteristics of the module and the temperature of the "unrepresented" fuel components will be very near the temperatures established for the oxidizer components. The same reasoning was used in justifying the decision to delete nodes representing the helium pressurization line going to the fuel tank.

There are several places where "perfect" insulation is assumed in the model, that is, it is assumed that no insulation exists on the hardware surface and the hardware losses no heat to the surroundings by convection or radiation   This approach is taken in regards to the insulation around nodes 9, and the ends of nodes 14 through 19.

2 0   MATERIAL PROPERTIES

Material properties utilized in the computer model are based on published data where it is available   Where such data is not available, approximations have been made   Table A-2 is a table of the general properties used in the analysis.

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3 0 RADIATION VIEW FACTORS

Radiation view factors to a large extent were obtained by readings taken with a form factometer on quarter and third scale models of the module. Where possible, these values were also checked against analytically determined view factors To assure any errors in these values would be eliminated, a summation of all F_aA (view factor times area) values for each node was made and compared with the area of the node If these values did not compare favorably, a recheck of all F_aA values was made.

4 0 COMPUTER PROGRAM FORMAT

The format of the computer model is the standard CINDA format A typical output for the flight analysis computer model is included at the end of this Appendix A review of this output will reveal that there are seven basic sections of input information

A Node Data

This consists of a listing of all nodes together with their initial fahrenheit temperatures and their heat capacitances, wc_p Nodes do not need to be arranged in numerical order They do need to be arranged in three groups. first, regular nodes, second, zero-capacitance nodes (node which have $wc_p = 0$), third, constant temperature boundary nodes. Zero-capacitance nodes are denoted by a -1 0 in the wc_p column and constant temperature nodes are denoted by a minus node number.

B Conductor Data

This consists of a listing of all conduction and radiation conductors together with the nodes connected by the conductor and the value of the conductor For conduction, the conductor value is kA/X For radiation, the conductor is "σ" A, and it is denoted as a radiation conductor by a minus in front of the conductor number Conductors -300 to -303 are the variable radiation conductors of the four louvers. The program obtains the value of these conductors by first looking in Table A-2 or A-1 for a value as a function of the temperature of the first listed node and then multiplying that value by 0.136×10^{-8}. The minus in the listed constant denotes that the first

listed node is the independent variable of the look-up array table and the constant multiplyer accounts for the size of the louver, the view factor between the two listed nodes, and the Stefan-Boltzmann constant. The dependent variable of the table lists the varying emissivity of the louvers as a function of tank temperature

C   Constants Data

This block of input data includes convergence criteria requirements, damping criteria and constant input heating in Btu/hr.

D   Array Data

This block lists all array tables to be used In the printout there are listed four such arrays. Arrays 1 and 2 are for variable emissivity for the louvers, referred to as A1 and A2 in conductors -300 to -303 Array A1 merely states that the emissivity is 0 72 at $-240^{\circ}F$, 0.13 at $-215^{\circ}F$ and that it varies linearly between these two points. The third array is an array of multiplying factors as a function of time in hours The fourth array varies the damping factors

E   Execution

This block establishes the specific problems to be solved and the order of solution. First, the instruction is given to multiply all (420) radiation conductors starting with -305 by the Stefan-Boltzmann constant CINDSL then instructs it to make a steady-state run To do this, it must refer to the VARIABLES 1 and VARIABLES 2 blocks. These blocks are described below.

Upon completion of the steady-state solution, that is the CINDSL instruction, the program proceeds by resetting the several constants as listed and then making a transient analysis as commanded by the instruction CNFRDL ITEST and LTEST values are required for instruction purposes in the VARIABLES BLOCKS "LINECT-100" instructs the machine to skip a page on the printout at this point TIMEND sets the time duration of the transient analysis and OUTPUT specifies the time interval for printout

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F. VARIABLE 1

Upon a command to perform a computational sequence the program scans through the various statements and subroutines listed in VARIABLES 1 and responds according to the information contained there The first two listed subroutines command the program to use variable damping factors as a function of the reiterations performed (LOOPCT) according to the array A99. The next two Fortran "IF" statements give instructions as to skipping various subroutines according to whether ITEST or LTEST are equal to 1. Thus, if both ITEST and LTEST are NOT equal to 1, as is the case when the CINDSL function is is being performed, the first block of 6 STFSEP subroutines will be performed in accomplishing CINDSL but the second set of D1D1WM subroutines will be skipped by jumping to statement 20. The STFSEP (K12,Q55) is an instruction to take the 12th constant in the CONSTANTS DATA, 88., and apply it as a constant heat addition to node 55 Note that all 17 of the constants are not used. It is through this manner that constant solar heating of any node is simulated.

If ITEST has been set to 1 0 in the EXECUTION block, the program will skip the STFSEP subroutines but will pick up the D1D1WM subroutines. This occured in the example case after computing the steady-state CINDSL case in preparation for computing the transient CNFRDL case. In this particular transient case, the subroutine D1D1WM (TIMEN, A3, K1, Q24) instructs to use the actual time as the independent variable in array 3, multiply the first constant in the CONSTANTS block by the corresponding dependent variable of array 3 and use the result as a varying heat input to Node 24

It should be observed that by utilizing the EXECUTION block and the VARIABLES 1 block, it is possible to compute several steady-state and/or transient cases successively, each having a different input However, unless other provisions are made, the node temperatures obtained at the end of any particular case are used as the initial node temperatures for the next case.

G. Output Calls

This block is used to specify the printed output. In the example, temperatures are printed (TPRINT) only once for the CINDSL However, by use of the several "IF" statements, the JTEST function, and the KTEST function, the program is made to print out the temperatures at the time specified by OUTPUT = X.

There are many other subroutines which may be called in order to accomplish a desired result To fully utilize the capacity of the CINDA program, the reader must consult a CINDA manual

A sensitivity analysis was made to determine which nodes and associated conductors are controlling Table A-3 lists these major conductors The nodes associated with each listed conductor are of necessity the controlling nodes. Table A-3 also lists the basis for the values of each conductor

Table 1   Nodal Arrangement

NODE	DESCRIPTION
1	B_2H_6 Tank $-X$
2	B_2H_6 Tank Center
3	B_2H_6 Tank $+X$
4	OF_2 Tank $-X$
5	OF_2 Tank Center
6	OF_2 Tank $+X$
7	B_2H_6 Top Fitting
8	OF_2 Top Fitting
(9)	Spacecraft Support (5)
10	Bottom Vertical Strut $-X$, $-Y$
11	
12	
13	
14	B_2H_6 Side Strut
15	B_2H_6 Center Strut
16	OF_2 Center Strut
17	OF_2 Side Strut
18	Center, Vertical Strut $-X$
19	Center, Vertical Strut $+X$
20	Top, Diagonal Strut $+X$
21	Top, Vertical Strut $+X$, $+Y$
22	Upper, Diagonal Strut $+X$
23	Upper, Vertical Strut $+X$, $+Y$

Table 1   (Continued)

NODE	DESCRIPTION
24	Center, Diagonal Strut +X
25	Center, Vertical Strut +X, +Y
26	Top, Vertical Strut +X, -Y
27	Top, Vertical Strut -X, -Y
28	Top, Diagonal Strut -X
29	Upper, Vertical Strut -X, -Y
30	Upper, Diagonal Strut -X
31	Center, Vertical Strut -X, -Y
32	Center, Diagonal Strut -X
33	Center, Vertical Strut -X, ±Y
34	Lower, Diagonal Strut -X
35	Lower, Vertical Strut -X, +Y
36	Bottom, Diagonal Strut -X
37	Bottom, Vertical Strut -X, +Y
38	Frame -X, +Y
39	Bottom, Vertical Strut +X, -Y
40	Bottom, Diagonal Strut +X
41	Frame, +X, -Y
42	Bottom, Vertical Strut +X, +Y
43	Frame +X, +Y
44	Frame -X, -Y

Table 1   (Continued)

NODE	DESCRIPTION
45	Frame -X
46	Frame +X
47	Frame +Y
48	Frame -Y
49	Helium Control Panel
50	OF_2 Control Panel
51	B_2H_6 Control Panel
52	Insulation Exterior, Node 1
53	Insulation Exterior, Node 2, -Y
54	Insulation Exterior, Node 2, +Y
55	Insulation Exterior, Node 3
56	Insulation Exterior, Node 4
57	Insulation Exterior, Node 5, -Y
58	Insulation Exterior, Node 5, +Y
59	Insulation Exterior, Node 6
60	Insulation Exterior, Helium Panel
61	Mating Foot +X, +Y
62	Insulation Exterior, B_2H_6 Panel
63	Mating Foot +X
64	OF_2 Bottom Fitting Insulation
65	Insulation Exterior, OF_2 Panel
66	Louver B_2H_6 +X

Table 1   (Continued)

NODE	DESCRIPTION
67	Louver B_2H_6 -X
68	Louver OF_2 +X
69	Louver OF_2 -X
70	Mylar Insulation Interior, OF_2 Panel
71	Mylar Insulation Exterior, OF_2 Panel
72	Mylar Insulation Interior, B_2H_6 Panel
73	Mylar Insulation Exterior, B_2H_6 Panel
74	Mylar Insulation Interior, Helium Panel
75	Mylar Insulation Extreior, Helium Panel
76	Propellant Valves
77	Propellant Valves, Insulation Interior
78	Propellant Valves, Insulation Exterior
79	OF_2 Feed Line, Upper
80	OF_2 Feed Line, Middle
81	OF_2 Feed Line, Lower
82	Bipropellant Valve
83	OF_2 Feed Line Insulation, Lower
84	OF_2 Feed Line Insulation, Middle
85	OF_2 Feed Line Insulation, Upper
86	OF_2 Bottom Fitting
87	B_2H_6 Bottom Fitting
88	Mating Foot +X, -Y
89	Mating Foot -Y
90	Mating Foot -X, -Y

Table 1 (Continued)

NODE	DESCRIPTION
91	Mating Foot -X
92	Mating Foot -X, +Y
93	Mating Foot +Y
94	Aluminum Beam
95	Aluminum Beam
96	Aluminum Beam
97	Aluminum Beam
98	Aluminum Beam
99	Mating Foot Insulation, Node 61
100	Beam Insulation, Node 94
101	Beam Insulation, Node 95
102	Beam Insulation, Node 96
103	
104	Beam Insulation, Node 97
105	Beam Insulation, Node 98
106	Mating Foot Insulation, Node 88
107	B_2H_6 Bottom Fitting Insulation
108	Cross Beam +X
109	Cross Beam -X
110	
111	
112	Helium Tank, Upper
113	Helium Tank, Center

Table 1 (Continued)

NODE	DESCRIPTION
114	Helium Tank, Lower
115	Helium Tank Insulation, Upper
116	Helium Tank Insulation, Center
117	Helium Tank Insulation, Lower
118	Helium Tank Lower Fitting
119	Thrust Frame +X
120	Thrust Frame –X
121	Upper Thrust Ring
122	Thrust Cylinder
123	Lower Thrust Ring
124	Combustion Chamber
125	Thrust Cone
126	Engine Purge Line, Upper
127	Engine Purge Line, Lower
128	Actuator, +Y
129	Actuator, +X
130	OF_2 Vent Line, Upper
131	OF_2 Vent Line, Middle
132	OF_2 Vent Line, Lower
133	OF_2 Vent Line Insulation, Upper
134	OF_2 Vent Line Insulation, Center
135	OF_2 Vent Line Insulation, Lower
136	Engine Purge Line Insulation, Upper, Node 126

Table 1 (Continued)

137	Engine Purge Line Insulation, Lower, Node 127
138	Engine Valve Line, Upper
139	Engine Valve Line, Lower
140	Engine Valve Line Insulation, Upper, Node 138
141	Engine Valve Line Insulation, Lower, Node 139
142	Shield -Y
143	Shield -X
144	Shield +Y
145	Shield +X
146	Aft Shield
147	——
148	
149	
150	
151	
152	
153	Spacecraft Insulation Exterior -Y
154	Spacecraft Insulation Exterior -X
155	Spacecraft Insulation Exterior +Y
156	Spacecraft Insulation Exterior +X
157	RTG
158	Space
159	Helium Line, Upper

Table 1 (Continued)

NODE	DESCRIPTION
160	Helium Line, Middle
161	Helium Line, Bottom
162	Helium Line Insulation, Upper
163	Helium Line Insulation, Middle
164	Helium Line Insulation, Lower
165	
166	Insulation Exterior, Top, Node 1
167	Insulation Exterior, Top, Node 2
168	Insulation Exterior, Top, Node 3
169	Insulation Exterior, Bottom, Node 1
170	Insulation Exterior, Bottom, Node 2
171	Insulation Exterior, Bottom, Node 3
172	Insulation Exterior, Top, Node 4
173	Insulation Exterior, Top, Node 5
174	Insulation Exterior, Top, Node 6
175	Insulation Exterior, Bottom, Node 4
176	Insulation Exterior, Bottom, Node 5
177	Insulation Exterior, Bottom, Node 6
178	
179	
180	LN_2 Coolant
181	B_2H_6 Coolant Coil

Table 1 (Continued)

NODE   DESCRIPTION

182   OF_2 Coolant Coil

183   He Coolant Coil

184   Ambient Air

185   Air, OF_2 Control Panel Box

186   Air, $B_2 H_6$ Control Panel Box

187   Air, He Control Panel Box

188   Air, Propellant Valve Box

NOTES

1. ( ) Denotes constant temperature node for flight thermal analysis

2. [ ] Denotes constant temperature node for groundhold analysis.

3   Delete Nodes 66, 67, 68, 69 and all associated resistances for groundhold analysis.

4   Assume RTG in stowed position for groundhold analysis

5. Background radiation changed from $0^O R$ for flight analysis to $525^O R$ for groundhold analysis

Table A-2 Material Properties

Material	$k \dfrac{Btu}{ft^2-fr\ {}^oF/ft}$	$C_p \dfrac{Btu}{lb-{}^oF}$	Solar Adsorbtivity	Emissivity
Boron Filament	1.17 (Parallel to Fiber)	0.28	0.80	0 80
Foam Insulation	0 1*	0.16	0 80	0.90
Silvered Teflon	–	–	0.20	0 90
Second Surface Mirrors	–	–	0 10	0 80
Aluminized Mylar Blanket	0 01	–	0 24	0.80
Titanium	20 00	0.095	0.30	0 10
Columbium	35.00	0.06	0.40	0.30
Aluminum	90.00	0 20	0 40	0 05
Stainless Steel	0.10	0.10	–	–
Fiberglass	0.12	0 15	0 80	0.80
LN_2	0.08	0 49	–	–
Helium	0 07	1.30	–	–
B_2H_6	0 72	0.65	–	–
OF_2	0.175	0 35	–	–
RTG	–	–	–	0.90

*There is a good indication that this value may be as small as 0.02.

Table A-3 Controlling Conductors and Nodes

Conductor No.	Node-to-Node	Conductor Value, kA/1
1	1-2	75 x 11.3/.8
2	1-52	0.00625 x 5.76/0 0625
4	2-3	same as No 1
6	2-53	0.00625 x 3/0 0625
7	2-54	same as No. 6
9	3-55	0.00625 x 6.1/0.0625
10	4-5	0.175 x 11 3/.75
11	4-56	0 00625 x 4.75/0.0625
13	5-6	same as No. 10
15	5-57	same as No. 6
16	5-58	same as No. 6
20	6-59	same as No 2
116	94-95	90 x 0.0037/0.75
118	95-96	same as No. 116
120	96-97	same as No. 116
124	96-112	90 x 0.016/0.125
126	97-98	90 x 0.0037/0.7
132	112-115	0.00625 x 8.05/0.0625
134	113-116	0.00625 x 3.1/0.0625
136	114-117	same as No. 132

Conductor No.	Node-to-Node	Radiation Conductor, (A) $\varepsilon_1 \varepsilon_2$
-455	52-158	4.92 x 0 9 x 1
-460	53-158	3 84 x 0 9 x 1.
-471	54-158	2.44 x 0 9 x 1
-483	55-157	0.062 x 0.9 x 0.9
-484	55-158	4.90 x 0 9 x 1.
-494	56-158	4.0 x 0.9 x 1.
-504	57-158	1.8 x 0 9 x 1.
-509	58-158	4.13 x 0 9 x 1.

Conductor No.	Node-to-Node	Radiation Conductor, $(A)\epsilon_1\epsilon_2$
-522	59-157	0.059 x 0.9 x 0 9
-523	59-158	3.94 x 0.9 x 1
-622	115-158	1.14 x 0 9 x 1
-630	116-157	0 0303 x 0 9 x 0 9
-631	116-158	3.22 x 0.9 x 1
-637	117-157	0 015 x 0 9 x 0 9
-638	117-158	0 5 x 0 9 x 1
-665	125-158	4.56 x 0.3 x 1.
		2 18 x 0 8 x 1
-697	166-158	1 45 x 0 9 x 1
-707	168-157	0 0086 x 0 9 x 0 9
-708	168-158	1.29 x 0.9 x 1.
-722	174-157	0 0049 x 0 9 x 0.9

```
BCD  3THERMAL LPCS
BCD  9TASK 4   STEADY NEAR EARTH
END
BCD  3NODE DATA
```

Node ———▶
Initial Temperature ——▶
Heat Capacitance ◀———

Node	Initial Temperature	Heat Capacitance
1,	-210.,	147.
2,	-210.,	175.
3,	-210.,	147.
4,	-210.,	243.
5,	-210.,	269.
6,	-210.,	243.
10,	-250.,	.1
14,	-200.,	.034
16,	-200.,	.036
17,	-200.,	.053
18,	-300.,	.56
19,	-100.,	.56
20,	10.,	.39
21,	10.,	.07
22,	-200.,	.12
23,	-200.,	.081
24,	-50.,	.435
25,	-50.,	.296
26,	-50.,	.14
27,	-10.,	.06
28,	-20.,	.05
29,	-200.,	.001
30,	-200.,	.078
31,	-250.,	.308
32,	-250.,	.126
33,	-250.,	.105
34,	-200.,	.132
35,	-200.,	.059
36,	-250.,	.06
38,	-250.,	.076
39,	-50.,	.11
40,	-50.,	.11
41,	-50.,	.077
42,	-50.,	.09
43,	1.,	.055
44,	-100.,	.065
45,	-100.,	.076
46,	1.,	.102
47,	-200.,	.086
48,	-200.,	.065
49,	-200.,	2.75
50,	-200.,	.44
51,	-200.,	.44
52,	-300.,	.19
53,	-200.,	.092

```
54,    -200•,      •132
55,    -100•,      •206
56,    -300•,      •16
57,    -200•,      •132
58,    -200•,      •092
59,    -100•,      •142
60,    -200•,      •046
66,    -200•,      •1
67,    -200•,      •1
68,    -200•,      •1
69,    -200•,      •1
76,    -200•,      1•3
82,    -200•,      1•5
94,    -200•,      •09
95,    -200•,      •09
97,    -200•,      •09
98,    -200•,      •09
112,   -200•,     18•6
113,   -200•,     26•2
114,   -200•,     18•6
115,   -200•,      •16
116,   -200•,      •126
117,   -200•,      •16
119,   -200•,      •038
120,   -200•,      •038
121,   -200•,      •171
123,   -300•,      •3
124,   -200•,      2•08
125,   -200•,      •96
128,   -300•,      •33
129,   -300•,      •33
142,   -300•,      •9
143,   -300•,      •9
144,   -300•,      •9
145,   -300•,      •9
169,   -200•,      •032
170,   -200•,      •019
171,   -200•,      •032
172,   -200•,      •032
173,   -200•,      •019
174,   -200•,      •032
175,   -200•,      •032
176,   -200•,      •019
177,   -200•,      •032
  7,   -210•,     -1•      ← Zero
  8,   -210•,     -1•        Capacitance
 15,   -200•,     -1•
 37,   -200•,     -1•
 61,   -300•,     -1•
```

108

```
62,    -200',   -1.
63,    -300',   -1.
64,    -200',   -1.
65,    -200',   -1.
70,    -200',   -1.
71,    -60',    -1.
72,    -200',   -1.
73,    -300',   -1.
74,    -200',   -1.
75,    -300',   -1.
77,    -200',   -1.
78,    -200',   -1.
79,    -200',   -1.
80,    -200',   -1.
81,    -200',   -1.
83,    -200',   -1.
84,    -200',   -1.
85,    -200',   -1.
86,    -200',       -1.
87,    -200',       -1.
88,    -300',   -1.
89,    -300',   -1.
90,    -300',   -1.
91,    -300',   -1.
92,    -300',   -1.
93,    -300',   -1.
96,    -200',       -1.
99,    -300',   -1.
100,   -200',   -1.
101,   -200',   -1.
102,   -200',   -1.
104,   -200',   -1.
105,   -200',   -1.
106,   -200',   -1.
107,   -200',   -1.
108,   -200',   -1.
109,   -200',   -1.
122,   -200',   -1.
126,   -200',   -1.
127,   -200',   -1.
130,   -200',   -1.
131,   -200',   -1.
132,   -200',   -1.
133,   -200',   -1.
134,   -200',   -1.
135,   -200',   -1.
136,   -200',   -1.
137,   -200',   -1.
138,   -200',   -1.
139,   -200',   -1.
```

```
140,    -200.,    -1.
141,    -200.,    -1.
146,    -300.,      -1.
153,    -200.,    -1.
154,    -200.,    -1.
155,    -200.,    -1.
156,    -200.,    -1.
159,    -200.,    -1.
160,    -200.,    -1.
161,    -200.,    -1.
162,    -200.,    -1.
163,    -200.,    -1.
164,    -200.,    -1.
166,       1.,      -1.
167,       1.,      -1.
168,     -30.,      -1.
```

Constant Temperature Node ⟶

```
  -9,      65.,    3.0
-157,     500.,    0.0
-158,    -460.,    0.0
```

Conductor Number ⟶   Connected Nodes ⟶   ⟵ Conductance

```
3CONDUCTOR DATA
 1,    1,      2,    10.15
 2,    1,     52,    .576
 3,    1,     62,    .1
 4,    2,      3,    10.15
 5,    2,      7,    64.8
 6,    2,     53,    .304
 7,    2,     54,    .440
 8,    2,     87,    81.5
 9,    3,     55,    .610
10,    4,      5,    3.0
11,    4,     56,    .475
12,    4,     60,    .275
13,    5,      6,    3.0
14,    5,      8,    64.8
15,    5,     57,    .440
16,    5,     58,    .307
17,    5,     76,    .08
18,    5,     86,    81.5
19,    5,    130,    .0033
20,    6,     59,    .555
21,    6,     65,    .1
```

22,	6,	131,	.0033
23,	7,	14,	.0005
24,	7,	14,	.0005
25,	8,	16,	.0005
26,	8,	17,	.0005
27,	8,	130,	.001
28,	9,	14,	.0005
29,	9,	15,	.0005
30,	9,	16,	.0006
31,	9,	17,	.0009
32,	9,	18,	.0009
33,	9,	19,	.0009
34,	9,	20,	.064
35,	9,	21,	.064
36,	9,	26,	.064
37,	9,	27,	.064
38,	9,	28,	.064
39,	9,	33,	.0009
40,	9,	153,	.056
41,	9,	154,	.056
42,	9,	155,	.056
43,	9,	156,	.056
44,	10,	31,	.004
45,	10,	44,	.064
46,	18,	45,	.0009
47,	19,	46,	.0009
48,	20,	22,	.0036
49,	21,	23,	.003
50,	22,	24,	.001
51,	22,	50,	.004
52,	23,	25,	.001
53,	23,	50,	.004
54,	24,	40,	.001
55,	25,	42,	.001
56,	27,	29,	.003
57,	28,	30,	.0036
58,	29,	31,	.001
59,	29,	51,	.004
60,	30,	32,	.0017
61,	30,	51,	.004
62,	32,	34,	.0017
63,	33,	35,	.0004
64,	34,	36,	.002
65,	34,	49,	.006
66,	35,	37,	.0011
67,	35,	49,	.006
68,	36,	38,	.064
69,	37,	38,	.064
70,	38,	45,	.013
71,	38,	47,	.019

111

72,	38,	92,	.029
73,	39,	41,	.064
74,	40,	41,	.064
75,	41,	46,	.023
76,	41,	48,	.012
77,	41,	88,	.029
78,	42,	43,	.064
79,	43,	46,	.023
80,	43,	47,	.019
81,	43,	61,	.029
82,	44,	45,	.013
83,	44,	48,	.012
84,	44,	90,	.029
85,	45,	91,	.029
86,	46,	63,	.029
87,	47,	86,	.318
88,	47,	93,	.029
89,	47,	94,	.45
90,	47,	99,	.006
91,	48,	87,	.318
92,	48,	89,	.029
93,	48,	98,	.45
94,	48,	106,	.006
95,	49,	126,	.0006
96,	49,	138,	.0006
97,	49,	159,	.0006
98,	50,	132,	.001
99,	64,	86,	.0033
100, 70,	71,	.005	
101, 72,	73,	.005	
102, 74,	75,	.01	
103,	76,	79,	.025
104,	77,	78,	.167
105,	79,	80,	.025
106,	79,	85,	.015
107,	80,	81,	.025
108,	80,	84,	.0002
109,	80,	113,	.0025
110,	81,	82,	.025
111,	81,	83,	.015
112,	82,	124,	2.0
113,	82,	127,	.001
114,	82,	139,	.001
115,	87,	107,	.0033
116,	94,	95,	.45
117,	94,	100,	.008
118,	95,	96,	.45
119,	95,	101,	.008
120,	96,	97,	.45
121,	96,	102,	.008

122,	96,	108,	.02
123,	96,	109,	.02
124,	96,	112,	11.5
125,	96,	161,	.001
126,	97,	98,	.48
127,	97,	104,	.008
128,	98,	105,	.008
129,	108,	46,	.02
130,	109,	45,	.02
131,	112,	113,	.84
132,	112,	115,	.536
133,	113,	114,	.84
134,	113,	116,	.207
136,	114,	117,	.53
137,	114,	119,	.0001
138,	114,	120,	.0001
139,	119,	121,	.039
140,	120,	121,	.039
141,	121,	122,	.26
142,	121,	128,	.043
144,	121,	129,	.043
145,	122,	123,	.26
146,	123,	124,	.026
147,	124,	125,	.2
148,	124,	128,	.035
149,	124,	129,	.035
150,	126,	127,	.001
151,	126,	136,	.0006
152,	127,	137,	.0006
153,	130,	131,	.001
154,	130,	133,	.00017
155,	131,	132,	.001
156,	131,	134,	.00017
157,	132,	135,	.0006
158,	138,	139,	.001
159,	138,	140,	.0006
160,	139,	141,	.0006
161,	159,	160,	.001
162,	159,	162,	.0006
163,	160,	161,	.001
164,	160,	163,	.0006
165,	161,	164,	.0006
166,	166,	1,	.128
167,	167,	2,	.067
168,	168,	3,	.144
169,	169,	1,	.128
170,	170,	2,	.067
171,	171,	3,	.145
172,	172,	4,	.119
173,	173,	5,	.067

113

	174,	174,	6,	.140
Radiation	175,	175,	4,	.119
Conductor	176,	176,	5,	.067
	177,	177,	6,	.146
CGS	-300,	1,	67,	A2, -.395E-8
CGS	-301,	3,	66,	A1, -.316E-8
CGS	-302,	4,	69,	A2, -.316E-8
CGS	-303,	6,	68,	A1, -.316E-8
	-305,	14,	168,	.022
	-306,	14,	153,	.036
	-307,	14,	156,	.005
	-308,	14,	157,	.00025
	-309,	14,	158,	.051
	-310,	15,	166,	.0026
	-311,	15,	54,	.0095
	-312,	15,	168,	.0026
	-313,	15,	57,	.0052
	-314,	15,	153,	.027
	-315,	15,	154,	.0031
	-316,	15,	156,	.0031
	-317,	15,	157,	.0003
	-318,	15,	158,	.032
	-319,	16,	54,	.0052
	-320,	16,	172,	.0026
	-321,	16,	57,	.0095
	-322,	16,	174,	.0026
	-323,	16,	154,	.0031
	-324,	16,	155,	.027
	-325,	16,	156,	.0031
	-326,	16,	157,	.0003
	-327,	16,	158,	.032
	-328,	17,	172,	.0275
	-329,	17,	173,	.0119
	-330,	17,	154,	.037
	-331,	17,	155,	.037
	-332,	17,	158,	.084
	-333,	18,	19,	.172
	-334,	18,	52,	.202
	-335,	18,	54,	.121
	-336,	18,	56,	.202
	-337,	18,	57,	.121
	-338,	18,	115,	.081
	-339,	18,	154,	.115
	-340,	18,	158,	1.08
	-341,	19,	54,	.121
	-342,	19,	55,	.201
	-343,	19,	57,	.121
	-344,	19,	59,	.202
	-345,	19,	115,	.081

Array Table

Multiplier

-346,	19,	156,	.115
-347,	19,	157,	.0125
-348,	19,	158,	.94
-349,	22,	23,	.0027
-350,	22,	50,	.013
-351,	22,	65,	.019
-352,	22,	70,	.0012
-353,	23,	50,	.013
-354,	23,	65,	.019
-355,	23,	70,	.0012
-356,	24,	55,	.40
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115

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121

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-696,	169,	158,	.51
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-706,	168,	154,	.28
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-708,	168,	158,	1.16
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-712,	175,	143,	.29
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-716,	175,	158,	.51
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END

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BCD 3CONSTANTS DATA
   NLOOP,1000
   DRLXCA,,01,  ARLXCA,,01
   DAMPD,,1,  DAMPA,,1
   1,340.,  2,250.,,  3,86.,  4,473.,  5,75.,  6,75,
   7,145.,  8,125.,  9,172.,  10,64.,  11,430,
   12,88.,  13,15.,  14,15.,  15,29.,  16,17.,  17,26.
END
BCD 3ARRAY DATA
   1,-240.,,72,-216.,,13,END ◄─────────── Louver Emissivity Table
   2,-205.,,13,-180.,,72,END
   3   S Q TABLE AS A FUNCTION OF TIME
      0.0,      0.0
      .042,     .13
      .083,     .26
      .167,     .5
      .25,      .71
      .33,      .86
      .5,      1.
      96.0,    1.0
   END
   99, 0.,,1,,1000.,,9, END  S DAMPING FACTORS AS A FUNCTION OF LOOPCT
END
BCD 3EXECUTION                                            | Fortran ───────► F
DIMENSION X(2000)                                         | Statement      F
NTH=0                                                                      F
NDIM=2000                                                                  F
   ARYMPY(920,G305,,1714E-8,G305)                                   Comment ────► C
***  CASE 1   S/S                                                   Statement
      CINDSL                                                                C
***  CASE 7   TRANS                                                         C
   NLOOP=100                                                                F
   ARLXCA=.05                                                               F
   LTEST=1                                                                  F
   JTEST=1                                                                  F
   LINECT=100                                                               F
   TIMEND=96.                                                               F
   OUTPUT=.167                                                              F
      CNFROL
END
BCD 3VARIABLES 1
   RTEST=LOOPCT                                                             F
      D1DEG1(RTEST,A99,DAMPD)
      STFSEP(DAMPD,DAMPA)
   IF (LTEST.EQ.1) GO TO 10                                                 F
   IF (LTEST.EQ.1) GO TO 20                                                 F
      STFSEP(K12,Q55)
      STFSEP(K13,Q41)
      STFSEP(K14,Q43)
      STFSEP(K15,Q46)
      STFSEP(K16,Q115)
      STFSEP(K17,Q125)
   GO TO 20                                                                 F
10  CONTINUE                                                                F
      D1D1WM (TIMEM,A3,K1,Q24)
      D1D1WM (TIMEM,A3,K2,Q25)
      D1D1WM (TIMEM,A3,K3,Q66)
      D1D1WM (TIMEM,A3,K4,Q55)
      D1D1WM (TIMEM,A3,K5,Q41)
      D1D1WM (TIMEM,A3,K6,Q43)
      D1D1WM (TIMEM,A3,K7,Q46)
      D1D1WM (TIMEM,A3,K8,Q116)
      D1D1WM (TIMEM,A3,K9,Q125)
      D1D1WM (TIMEM,A3,K10,Q71)
      D1D1WM (TIMEM,A3,K11,Q69)
20  CONTINUE                                                                F
END
BCD 3VARIABLES 2
END
BCD 3OUTPUT CALLS
   TPRINT
   IF (JTEST.EQ.0) GO TO 10                                                 F
   KTEST=KTEST+1                                                            F
   IF (KTEST.EQ.2) OUTPUT=.333                                             F
   IF (KTEST.EQ.3) OUTPUT=.5                                               F
   IF (KTEST.EQ.4) OUTPUT=1.                                               F
   IF (KTEST.EQ.5) OUTPUT=2.                                               F
   IF (KTEST.EQ.6) OUTPUT=4.                                               F
   IF (KTEST.EQ.8) OUTPUT=12.                                              F
10  CONTINUE                                                                F
END
```

Computer Output, Initial Node Temperatures

TIME= 0.00000   DTIMEU= 0.00000   CSGMIN(0)= 0.00000   TEMPCC(0)= 0.00000   RELXCC(0)= 0.00000

```
T     1= -2.10000+02   T     2= -2.10000+02   T     3= -2.10000+02   T     4= -2.10000+02   T     5= -2.10000+02   T     6= -2.10000+02
T    10= -2.50000+02    T    14= -2.30000+02   T    16= -2.00000+02   T    17= -2.00000+02   T    18= -3.00000+02   T    19= -1.00000+02
T    20=  1.00000+01    T    21=  1.00000+01   T    22= -2.00000+02   T    23= -2.00000+02   T    24= -5.00000+01   T    25= -5.00000+01
T    26= -5.00000+01    T    27= -1.00000+01   T    28= -2.80000+01   T    29= -2.00000+02   T    30= -2.00000+02   T    31= -2.50000+02
T    32= -2.50000+02    T    33= -2.50000+02   T    34= -2.00000+02   T    35= -2.00000+02   T    36= -2.50000+02   T    38= -2.50000+02
T    39= -5.00000+01    T    40= -5.00000+01   T    41= -5.00000+01   T    42= -5.00000+01   T    43=  1.00000+00   T    44= -1.00000+02
T    45= -1.00000+02    T    46=  1.00000+00   T    47= -2.00000+02   T    48= -2.00000+02   T    49= -2.00000+02   T    50= -2.00000+02
T    51= -2.00000+02    T    52= -3.00000+02   T    53= -2.00000+02   T    54= -2.00000+02   T    55= -1.00000+02   T    66= -3.00000+02
T    57= -2.00000+02    T    58= -2.00000+02   T    59= -1.00000+02   T    60= -2.00000+02   T    66= -2.00000+02   T    67= -2.00000+02
T    68= -2.00000+02    T    69= -2.00000+02   T    76= -2.00000+02   T    82= -2.00000+02   T    94= -2.00000+02   T    95= -2.00000+02
T    97= -2.00000+02    T    98= -2.00000+02   T   112= -2.00000+02   T   113= -2.00000+02   T   114= -2.00000+02   T   115= -2.00000+02
T   116= -2.00000+02    T   117= -2.00000+02   T   119= -2.00000+02   T   120= -2.00000+02   T   121= -2.00000+02   T   123= -3.00000+02
T   124= -2.00000+02    T   125= -2.00000+02   T   128= -3.00000+02   T   129= -3.00000+02   T   142= -3.00000+02   T   143= -3.00000+02
T   144= -3.00000+02    T   145= -3.00000+02   T   169= -2.00000+02   T   170= -2.00000+02   T   171= -2.00000+02   T   172= -2.00000+02
T   173= -2.00000+02    T   174= -2.00000+02   T   175= -2.00000+02   T   176= -2.00000+02   T   177= -2.00000+02   T     7= -2.10000+02
T     8= -2.10000+02    T    15= -2.00000+02   T    37= -2.00000+02   T    61= -3.00000+02   T    62= -2.00000+02   T    63= -3.00000+02
T    64= -2.00000+02    T    65= -2.00000+02   T    70= -2.00000+02   T    71= -6.00000+01   T    72= -2.00000+02   T    73= -3.00000+02
T    74= -2.00000+02    T    75= -3.00000+02   T    77= -2.00000+02   T    78= -2.00000+02   T    79= -2.00000+02   T    80= -2.00000+02
T    81= -2.00000+02    T    83= -2.00000+02   T    84= -2.00000+02   T    85= -2.00000+02   T    86= -2.00000+02   T    87= -2.00000+02
T    88= -3.00000+02    T    89= -3.00000+02   T    90= -3.00000+02   T    91= -3.00000+02   T    92= -3.00000+02   T    93= -3.00000+02
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T   134= -2.00000+02    T   135= -2.00000+02   T   136= -2.00000+02   T   137= -2.00000+02   T   138= -2.00000+02   T   139= -2.00000+02
T   140= -2.00000+02    T   141= -2.00000+02   T   146= -3.00000+02   T   153= -2.00000+02   T   154= -2.00000+02   T   155= -2.00000+02
T   156= -2.00000+02    T   159= -2.00000+02   T   160= -2.00000+02   T   161= -2.00000+02   T   162= -2.00000+02   T   163= -2.00000+02
T   164= -2.00000+02    T   166=  1.00000+00   T   167=  1.00000+00   T   168= -3.00000+01   T     9=  6.50000+01   T   157=  5.00000+02
T   158= -4.60000+02
```

Computer Output, Steady-State Temperatures

```
**********
TIME=  0.00000     DTIMEU=  0.00000     CSGMIN(   0)=  0.00000     TEMPCC(   0)=  0.00000     RELXCC(   4)=  9.24301-03

T    1= -2.08920+02   T    2= -2.06429+02   T    3= -2.01797+02   T    4= -2.18493+02   T    5= -2.12541+02   T    6= -2.04915+02
T   10= -2.47544+02   T   14= -2.03932+02   T   16= -1.84528+02   T   17= -2.38920+02   T   18= -2.50687+02   T   19= -1.56739+02
T   20=  5.49407+01   T   21=  5.62437+01   T   22= -1.23900+02   T   23= -1.30567+02   T   24= -1.76110+02   T   25= -1.91070+02
T   26=  6.50000+01   T   27=  5.57324+01   T   28=  6.41222+01   T   29= -1.41975+02   T   30= -1.39260+02   T   31= -2.89674+02
T   32= -2.66615+02   T   33= -2.92353+02   T   34= -2.36965+02   T   35= -2.34441+02   T   36= -2.56067+02   T   38= -2.48762+02
T   39=  1.21893+01   T   40=  9.29235+00   T   41=  1.21895+01   T   42=  4.98446-01   T   43=  3.49018+00   T   44= -2.44911+02
T   45=  2.38926+02   T   46=  1.20998+02   T   47= -2.02672+02   T   48= -2.00570+02   T   49= -2.29490+02   T   50= -1.70964+02
T   51= -1.79296+02   T   52= -2.40289+02   T   53= -2.45040+02   T   54= -2.22994+02   T   55= -1.29330+02   T   56= -2.45147+02
T   57= -2.20149+02   T   58= -2.46275+02   T   59= -1.71475+02   T   60= -2.20985+02   T   66= -1.43571+02   T   67= -2.97616+02
T   68= -1.40558+02   T   69= -2.97787+02   T   76= -2.15259+02   T   82= -1.84275+02   T   94= -2.00704+02   T   95= -1.98569+02
T   97= -1.97555+02   T   98= -1.99110+02   T  112= -1.96246+02   T  113= -1.98721+02   T  114= -2.04226+02   T  115= -1.91663+02
T  116= -1.86411+02   T  117= -2.12955+02   T  119= -1.97808+02   T  120= -1.97808+02   T  121= -1.95978+02   T  123= -1.96630+02
T  124= -1.84015+02   T  125= -1.75218+02   T  128= -1.91234+02   T  129= -1.91234+02   T  142= -2.22637+02   T  143= -2.12056+02
T  144= -2.32147+02   T  145= -1.76118+02   T  169= -2.21920+02   T  170= -2.12057+02   T  171= -1.96863+02   T  172= -2.39316+02
T  173= -2.12931+02   T  174= -2.06363+02   T  175= -2.26979+02   T  176= -2.12240+02   T  177= -1.98418+02   T    7= -2.06431+02
T    8= -2.12524+02   T   15= -1.86579+02   T   37= -2.54008+02   T   61= -2.77142+01   T   62= -2.01719+02   T   63=  5.68436+01
T   64= -2.12464+02   T   65= -1.94721+02   T   70=  1.93038+02   T   71= -2.00050+02   T   72= -3.08622+02   T   73= -3.49504+02
T   74= -2.97783+02   T   75= -3.52572+02   T   77= -2.69229+02   T   78= -2.71533+02   T   79= -2.08664+02   T   80= -1.01185+01
T   81= -1.94826+02   T   83= -2.01821+02   T   84= -9.35250+01   T   85= -2.10154+02   T   86= -2.12486+02   T   87= -2.06408+02
T   88= -2.08643+01   T   89= -2.03705+02   T   90= -2.46025+02   T   91= -2.40254+02   T   92= -2.49706+02   T   93= -2.05675+02
T   96= -1.96282+02   T   99= -2.53696+02   T  100= -2.11243+02   T  101= -2.07265+02   T  102= -1.75834+02   T  104= -1.75930+02
T  105= -2.10214+02   T  106= -2.49406+02   T  107= -2.06413+02   T  108= -1.19656+02   T  109= -2.21577+02   T  122= -1.96629+02
T  126= -2.25112+02   T  127= -2.15490+02   T  130= -2.12223+02   T  131= -2.08087+02   T  132= -2.08739+02   T  133= -2.29132+02
T  134= -2.41801+02   T  135= -2.72833+02   T  136= -2.36779+02   T  137= -2.51482+02   T  138= -2.25240+02   T  139= -2.15812+02
T  140= -2.36787+02   T  141= -2.52630+02   T  146= -2.60279+02   T  153= -2.37553+02   T  154= -2.36068+02   T  155= -2.31064+02
T  156= -2.15711+02   T  159= -2.55360+02   T  160= -2.34237+02   T  161= -2.14526+02   T  162= -2.04146+02   T  163= -2.31899+02
T  164= -2.12084+02   T  166= -2.44264+02   T  167= -2.50395+02   T  168= -2.03020+02   T    9=  6.50000+01   T  157=  5.00000+02
T  158= -4.60000+02
```

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APPENDIX B PROPULSION SYSTEM ANALYSIS EQUATIONS

1.0 FIRING SIMULATION PROGRAM

This is a "time-advancing" program which computes propellant flow rates, the resultant changes in the ullage volumes, the gas flow rates into the ullages, the temperatures, and the pressures that result. That is to say, the result of time is simulated by repeating the entire sequence of calculations for as many increments of time as necessary to cover a total-ized time interval equal to the predicted burning time, calculated values are carried forward as necessary and cumulative totals maintained. The equations used in the program are listed here in a logical order but it should be recognized that they could be solved in any other mathematically logical order without significantly changing the answers. Furthermore, this listing is not the actual computer program per se, but only the engineering equations, the computer program itself must of necessity, include certain instructions, checks, etc., which have been omitted there for the sake of clarity.

Table B-1 lists the symbols, their meaning, and the units used in these computations. Values of "given" and constant quantities are shown in parenthesis

Propellant flow rates to the engine are calculated by a trial-and-error process using these equations

(1)
$$\dot{m}_{ox} = \sqrt{\frac{g\rho_{lox}\,(P_{tox} - P_c)}{R_{ox}}}$$

(2)
$$\dot{m}_f = \sqrt{\frac{g\rho_{lf}\,(P_{tf} - P_c)}{R_f}}$$

(3)
$$r = \frac{\dot{m}_{ox}}{\dot{m}_f}$$

(4)
$$P_c = 14.4\,I_s\,(\dot{m}_{ox} + \dot{m}_f)$$

$$(5) \quad N_{ox} = \frac{1.273 \, \dot{m}_{ox}}{D_{ox} \mu_{ox}}$$

$$(6) \quad N_f = \frac{1.273 \, \dot{m}_f}{D_f \, \mu_f}$$

$$(7) \quad R_{ox} = 0.8105 \left[\frac{f_{ox} L_{ox}}{D_{ox}^5} + \frac{1}{C_{ox}^2 D_{ox}^4} \right]$$

$$(8) \quad R_f = 0.8105 \left[\frac{f_f L_f}{D_f^5} + \frac{1}{C_f^2 D_f^4} \right]$$

To solve these equations, the propellant densities (ρ_{lox} and ρ_{lf}) and
viscosities (μ_{ox} and μ_f) are calculated from equations or look up in
tables as functions of temperature. The trial-and-error process is started
by assuming values for chamber pressure (P_c) and the resistance coefficients
(R_{ox} and R_f), which allows equations (1) and (2) to be solved to get first
estimates of oxidizer and fuel flow rates ($\dot{m}_{ox}$ and $\dot{m}_f$). After mixture
ratio (r) is computed (equation 3), specific impulse (I_s) is looked up in
a table as a function of mixture ratio (r)   Next, the first iteration on
chamber pressure (P_c) is obtained from equation (4). If this value differs
from the initial estimate by more than the allowed error, iteration is nec-
essary. A new estimate for chamber pressure is then calculated (a method
which causes very fast convergence on the correct value is to let the next
estimate of P_c equal 0.2 of the previous estimate plus 0 8 of the calculated
P_c)   Before a recalculation is made, however, the resistance coefficients
should be reestimated by solving equations (5) and (6) for the Reynolds
Numbers (N_{ox} and N_f), looking up the friction factors (f_{ox} and f_f) in a
table as a function of those Reynolds Numbers, and then solving equations
(7) and (8) for the new estimates of the resistance coefficients (R_{ox} and
R_f)   (In passing, it might be noted that these resistance coefficients
have been formulated to account for both constant and variable discharge
coefficients.)

If it is assumed that the propellant temperatures are constant throughout the particular firing being simulated, then the above calculation of chamber pressure, mixture ratio, and propellant flow rates needs to be made only once.

Next, the remaining masses of liquid propellants in the tanks and the ullage volumes are calculated using these simple relations

$$(9) \qquad dm_{lox} = \dot{m}_{ox} d\theta + dm_{vox}$$

$$(10) \qquad dm_{lf} = m_f d\theta + dm_{vox}$$

$$(11) \qquad m_{lox} = m_{lox}(0) - \int_0^\theta dm_{lox}$$

$$(12) \qquad m_{lf} = m_{lf}(0) - \int_0^\theta dm_{lf}$$

$$(13) \qquad V_{uox} = V_{tox} - \frac{m_{lox}}{\rho_{lox}}$$

$$(14) \qquad V_{uf} = V_{tf} - \frac{m_{lf}}{\rho_{lf}}$$

The value of $d\theta$ is equal to the length of the time step used in the calculation (e.g , if the computation is made for each successive second of burning time, $d\theta = 1$ sec) An estimate of the rate of propellant vapor evolution is necessary for the first time step computation only in order to get dm_{vox} and dm_{vf}, but for small time steps the errors resulting from setting these equal to zero for the first time step is very small Values of the liquid propellant masses at the beginning of the firing ($m_{lox}(0)$ and $m_{lf(0)}$) are, of course, basic inputs as are the propellant tank volumes (V_{tox} and V_{tf}).

Next, the partial masses of helium in the propellant tank ullages and the helium flow rates to the ullages are calculated. If the total

pressure of a propellant tank exceeds regulated pressure (P_r), then the existing pressure is used in the calculations, otherwise, it is set to 300 psia on the assumption that the pressure regulator will control the pressure by admitting just enough gas to keep total pressure in the tank at the regulated level (i.e , total pressure equals the vapor pressure plus a partial pressure of helium sufficient to make the sum exactly 300 psia)

Propellant tank total pressures are calculated as follows. First, propellant vapor pressures are computed from equations or looked up in tables as functions of the ullage temperatures (see discussion above on the validity of this assumption). Approximate equations for OF_2 and B_2H_6 vapor pressure were obtained by curve fitting to selected empirical data.

$$P_{vox} = 2\ 7847 \times 10^{7.3474 - \frac{1029\ 15}{T_{uox}}}$$

$$P_{vf} = e^{17\ 5151 - \frac{3217\ 09}{T_{uf}}}$$

Next, the helium partial pressures are computed using the gas law For the first time step, the initial or estimated values of the partial helium masses and ullage temperatures are used For later time steps, the values from the previous time step are used In both cases, the current ullage volumes, calculated as mentioned above, are used Errors in the order of 2% will result unless the compressibility factors are correct These factors are most simply obtained from a table as a function of helium pressure and temperature. For the first time step, it is necessary to provide an estimated ullage temperature and helium partial pressure, or an estimate of the compressibility factor, thereafter, it is sufficiently accurate to use the helium partial pressure and ullage temperature cal- culated during the previous time step. The partial pressures of helium, then are

(15)
$$P_{heox} = \frac{M_{heox}\ Z_{ox}\ R\ T_{uox}}{V_{uox}}$$

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$$(16) \qquad P_{hef} = \frac{M_{hef} Z_f R T_{uf}}{V_{uf}}$$

Propellant tank total pressures are assumed to be the simple sums of
the vapor pressures of the propellants and the respective partial pressures
of helium,

$$(17) \qquad P_{tox} = P_{vox} + P_{heox}$$

$$(18) \qquad P_{tf} = P_{vox} + P_{hef}$$

Whenever the calculated total pressure is less than regulated pressure
(P_r = 300 psia - 43200 psfa), it is discarded and replaced with the reg-
ulated pressure valve   In that case, it is necessary to calculate the
corresponding partial masses of helium in the ullage volume

$$(19) \qquad M_{heox} = \frac{(P_{tox} - P_{vox}) V_{uox}}{Z_{ox} R T_{uox}}$$

$$(20) \qquad M_{hef} = \frac{(P_{tf} - P_{vf}) V_{uf}}{Z_f R T_{uf}}$$

By subtracting the partial masses of helium for the previous time
step from these current masses, a difference is obtained which is the mass
change during this time step. The sum of the changes in the partial masses
of helium in the oxidizer and fuel tanks is the change in the helium mass
within the helium tank (by the conservation of mass law)   (Note that none
of these differential masses can be less than zero in a real system)

$$(21) \qquad dm_{heox} = m_{heox_\theta} - m_{heox_{\theta-1}}$$

$$(22) \qquad dm_{hef} = m_{hef_\theta} - m_{hef_{\theta-1}}$$

(23)
$$dm_{he} = dm_{heox} + dm_{hef}$$

The current mass of helium in the helium tank is simply the initial mass minus the sum of all the changes from time 0 to θ

(24)
$$m_{he} = m_{he(0)} - \int_0^\theta dm_{he}$$

In order to calculate the ullage temperatures, it is necessary to know the temperature of the incoming helium This helium temperature can be estimated by assuming it is at the current average temperature of the helium in the helium tank (T_{he}) plus any changes in temperature incurred as the gas flows through the connecting plumbing, such as heating due to the Joule-Thomson effect within the regulator and convective heat transfer. The average temperature of the helium in the helium tank is assumed to be the initial temperature (time 0) plus the sum of all the incremental changes in temperature to time θ. These incremental changes are due to the net sum of the energy changes caused by expansion of the residual gas as helium flows from the tank, and to the transfer of heat from the tank wall to the gas. In order to estimate the heat transfer rate, a simplified free-convection model was adopted. This model requires a value for the convection coefficient, h_c, so an expression was set up using the average properties of helium

(25)
$$h_c = 1 39 \times 10^{-4} \left[G \left(\frac{P_{he}}{T_{he}} \right)^2 \left(\frac{T_w}{T_{he}} - 1 \right) \right]^{0 4}$$

where "G" is the acceleration of the spacecraft in "g's"

(26)
$$G = \frac{F}{m_{sc}} = \frac{F}{m_{sc(0)} - \int_0^\theta dm_{ox} - \int_0^\theta dm_f}$$

This convection coefficient changes very slowly during a firing so the values used for the variables can be those calculated during the previous time step, the coefficient can be zero at start without incurring any error.

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The incremental change in temperature of the helium due to the convective heat transfer is then obtained by rearrangement of the conventional equation

$$q = \frac{dQ}{d\theta} = \frac{m_{he}\, C_{v_{he}}\, \partial T_{he}}{d\theta}$$

$$\partial T_{he} = \frac{d\theta}{m_{he} C_{v_{he}}}$$

but

$$q = h_c A (T_w - T_{he}) \qquad \text{(convection heat transfer)}$$

so

$$\partial T_{he} = \frac{d\theta}{m_{he} C_{v_{he}}} \left[h_c S (T_w - T_{he}) \right]$$

The change in energy within the helium tank due to withdrawal of dm_{he} pounds of helium is

$$dE = \frac{Z_{he}\, R\, T_{he}\, dm_{he}}{J}$$

And the resultant incremental temperature change is

$$\alpha T_{he} = - \frac{1}{m_{he} C_{v_{he}}} \left[\frac{Z_{he}\, R\, T_{he}\, dm_{he}}{J} \right]$$

Adding these two increments of temperature change together yields the total change affected during each time step $d\theta$.

(27)
$$dT_{he} = \frac{\left[h_c A (T_w - T_{he}) \right] d\theta - \frac{1}{J} (Z_{he}\, R\, T_{he}) d\, m_{he}}{m_{he}\, C_{v_{he}}}$$

The compressibility factor (Z_{he}) may be looked up in a table as a function of T_{he} and P_{he}; omitting this step (i.e., letting $Z = 1$) leads to errors in excess of 20 percent when tank pressures are highest.

Current helium temperature (at time θ) is the initial temperature plus the algebraic sum of all the changes calculated for the individual time steps

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(28)
$$T_{he} = T_{he}(0) + \int_0^\theta dT_{he}$$

The current helium tank pressure is calculated using the gas law

(29)
$$P_{he} = \frac{M_{he} \, Z_{he} \, R \, T_{he}}{V_{he}}$$

The above convective heat transfer calculation also included a value for the tank wall temperature, T_w In the presently outlined order of computation, a new wall temperature is calculated next so it is available as an input for the following time step calculation. To obtain T_w, a simple mathematical model of the helium tank thermal balance was set up Figure B-1 illustrates the nomenclature.

ᴠ Imagining the wall to be composed of three discrete layers within which the temperatures are uniform and from which the heat flows uniformly, the model of the wall reduces to three heat capacitances (C1, C2 and C3) separated by three thermal resistances (R1W, R21 and R32) All parts of the tank and gas are assumed to be at the same temperature at start (time = 0). As gas is withdrawn, the remaining gas expands and is thus cooled to a lower temperature. This creates a temperature difference between the gas and the wall which drives heat from the wall to the gas. As the wall cools, it receives heat across resistance R1W from layer 1 (represented by capacitance C1). Heat extracted from C1 lowers its temperature so it in turn receives heat from C2, and so forth. The set of equations representing this model, and with which the temperature changes and temperatures are calculated, are based on the fundamental conduction relation,

$$q = kA\Delta T/x$$

and the thermal capacity definition,

$$dT = \frac{dQ}{mC_p}$$

Introducing the thermal resistance, R_t, equal to $\frac{X}{KA}$, and the thermal capacitance, C, equal to mc_p, gives the equations

$$q = \frac{\Delta T}{R_t}$$

and $$dT = \frac{dQ}{C}$$

These two relations may be combined in an expression for the conservation of energy law, which in this case says that the net heat flow into any thermal capacitance must be equal to the rate of change in its energy content

$$q_{in} - q_{out} = \frac{dE}{d\theta} = \frac{dQ}{d\theta} = \frac{CdT}{d\theta}$$

or $$dT = \frac{(q_{in} - q_{out})}{C} \, d\theta$$

For any layer, n, in the tank,

$$(30) \quad dT_n = \left[\left(\frac{\Delta T}{R_t}\right)_{in} - \left(\frac{\Delta T}{R_t}\right)_{out} \right] \frac{d\theta}{C_n} = \left(\frac{T_{n+1} - T_n}{R_{t(n+1 \to n)}} - \frac{T_n - T_{n-1}}{R_{t(n \to n-1)}} \right) \frac{d\theta}{C_n}$$

For the outermost layer, which receives no heat, the heat flow out results in a change of.

$$(31) \quad dT_m = -\left(\frac{T_m - T_{m-1}}{R_{t(m \to m-1)}} \right) \frac{d\theta}{C_m}$$

and the current temperature (at time θ) is

$$(32) \quad T_n = T_{n(0)} \int_0^\theta dT_n$$

After having solved these equations for all three layers in the tank, the inside wall temperature (T_w) can be calculated on the basis that the rate at which heat arrives at a surface is the same as the rate at which

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heat leaves a surface (i e , a surface has no thermal capacity and therefore cannot store energy)[*]

$$q_{in} = q_{out}$$

$$q_{in} = \frac{T_1 - T_w}{R_{t\ 1W}}$$

$$q_{out} = h_c A (T_w - T_{he})$$

Rearranging and solving for T_w gives,

(33)
$$T_w = \frac{T_1 + T_{he} h_c A R_{t\ 1W}}{1 + h_c A R_{t\ 1W}}$$

Having the temperature of the gas leaving the helium tank, T_{he} (equation 28), it is now possible to estimate the temperature of the gas entering the propellant tank ullages (T_g). Heat transferring to this flow of gas from the gas circuitry hardware (lines, valves, filters, etc) was ignored in the present model although the effect of this extra heat could be appreciable during the initial moments of flow An attempt was made, however, to approximate the Joule-Thomson effect by constructing a table of temperature rise through the regulator (ΔT_{jt}) as a function of regulator inlet pressure and temperature (P_{he} and T_{he}) The values of temperature rise for the table were obtained from a temperature-entropy diagram by following constant enthalpy lines from their point of intersection with the chosen inlet pressure and temperature conditions to their intersection with the 300 psia line, reading the temperature value there and subtracting from it the inlet temperature Outlet temperature from the regulator (and inlet temperature for the propellant tank ullages) then is

(34)
$$T_g = T_{he} + \Delta T_{jt}$$

_____

[*]Thermophysical properties such as surface energy is ignored in this model

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Propellant vapor densities, rates of vapor evolution, and total
vapor masses are required to calculate the average ullage temperature
Vapor densities can be looked up in tables as a function of ullage
temperature or calculated from equations. An equation for OF_2 vapor
density was obtained by curve fitting selected empirical data

$$\rho_{vox} = e^{8\ 2\ -\ \frac{2180}{T_{uox}}}$$

The rates of vapor evolution or condensation are the differentials
of the mass equation (expressed here for a single component for clarity,
but must be solved for both propellants)

(35) $\qquad\qquad\qquad\qquad m_v = \rho_v V_u$

(36) therefore, $\qquad\qquad dm_v = \rho_v dV_u + V_u d\rho_v$

Initial estimates of ullage volume (V_u) and vapor density (ρ_v) at time
0 must be supplied for the first time step calculations of dV_u and $d\rho_v$
but therafter, the previous time step values are used (i.e , $dV_u =$
$V_{u_\theta} - V_{u_{\theta-1}}$).

Changes in the ullage temperatures can be calculated from the en-
ergy balances. A number of simplifications and assumptions were accepted
in order to make the calculation reasonably easy. These were discussed
above The differential quantities of energy brought into the ullage
by the incoming helium and evolving vapor are

$$dH_{he} = dm_{he}\ C_{p_{he}}\ T_g$$

$$dH_v = dm_v\ C_{p_v}\ T_1$$

The differential quantities of internal energy possessed by these
fluids after they come to equilibrium with the ullage gases are

$$dU_{he} = dm_{he}\ C_{v_{he}}\ T_u$$

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$$dU_v = dm_v \, C_{V_v} T_u$$

The differential quantity of work done by the ullage gas in expelling the differential volume of propellant is

$$dW = \frac{P_t}{J} \, dV_u$$

The energy balance equates the changes in energy of the incoming helium and the propellant vapor being evolved minus the expulsion work done, to the change in internal energy of the ullage gases

$$(dH_{he} - dU_{he}) + (dH_v - dU_v) - JW = dU_{he + v} = (m_{he}C_{V_{he}} + m_v C_{V_v})dT_u$$

Rearranging the terms gives an explicit expression for the change in ullage temperature.

$$dT_u \frac{(dH_{he} - dU_{he}) + (dH_v - dU_v) - dW}{m_{he}C_{V_{he}} + m_v C_{V_v}}$$

$$(37) \qquad dT_u = \frac{dm_{he}(C_{P_{he}} T_g - C_{V_{he}} T_g) + dm_v(C_{P_v} T_1 - C_{V_v} T_u) - \frac{P_t}{J} dV_u}{m_{he}C_{V_{he}} + m_v C_{V_v}}$$

Finally, the new ullage temperatures are calculated by adding the sums of the differential changes (dT_u) to the initial temperatures (at time = 0).

$$(38) \qquad T_{uox} = T_{uox}(0) + \int_0^\theta dT_{uox}$$

$$(39) \qquad T_{uf} = T_{uf}(0) + \int_0^\theta dT_{uf}$$

This completes the entire calculation for one time step (value of θ). The process is, of course, repeated as often as necessary to reach the required total burn time by carrying forward all the values calculated

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2.0 CRUISE-MODE PROGRAM

Equations which determine the equilibrium pressures and masses in the ullages were programmed for a digital computer so that the effects of temperature changes and helium solubility in the propellants could be analyzed. This analysis is specifically designed to determine the magnitude of problems which may arise because equilibrium temperatures may change with time during a mission, or because at the end of a firing, the ullage gas temperatures and helium partial pressures may be substantially different than the equilibrium values. Either of these situations may cause excessive tank pressures to develop Therefore, it is necessary to investigate (1) the effect of reaching thermal equilibrium in the propallant tanks, and (2) the helium partial pressure which will remain after reaching the equilibrium concentration of dissolved helium in the liquids

The approach outlined below uses mass balance, gas law and solubility relations. These are steady-state equations (time is not a variable) Identical calculations are made for fuel and oxidizer. In the description below, the input values (those constants or variables which are known for one temperature condition such as at the end of a firing) are labeled condition (1). The desired values (unknowns being solved for) are labeled condition (2),

First, the propellant liquid and vapor densities are looked up in tables or calculated as functions of the condition (2) liquid and ullage gas temperatures. (Presumably, for equilibrium conditions, the liquid and ullage gas temperatures are identical) Ullage volumes are then computed on the basis of the conservation of mass which means that the total mass of a propellant (liquid plus vapor) remains unchanged although the amounts existing as liquid and vapor change with temperature

At condition (1) $\qquad m_1 = m_{L1} + m_{v1}$

At condition (2) $\qquad m_2 = m_{L2} + m_{v2}$

$$m_1 = m_2 \text{ (from the conservation of mass)}$$

$$\therefore \ m_{L1} + m_{v1} = m_{L2} + m_{v2}$$

$$V_{u2} = V_t - V_{L2}$$

$$V_{L2} = \frac{m_{L2}}{\rho_{L2}}$$

$$m_{L2} = m_{L1} + m_{v1} - m_{v2}$$

$$m_{v2} = V_{u2}\rho_{v2}$$

$$V_{u2} = V_t - \frac{m_{L1} + m_{v1} - V_{u2}\rho_{v2}}{\rho_{L2}}$$

Solving for V_{u_2}

(40)
$$V_{u2} = \frac{V_t\rho_{L2} - m_{L1} - m_{v1}}{\rho_{L2} - \rho_{v2}}$$

Mass of the liquid at condition (2) is

$$m_{L2} = m_{L1} + m_{v1} - m_{v2}$$

(41)
$$m_{L2} = m_{L1} + m_{v1} - V_{u2}\rho_{v2}$$

In order to determine the equilibrium partial pressure of helium, which is a function of helium solubility (s_2) or concentration (lb/lb) it is necessary to establish the solubility divided by the partial mass of helium in the ullage

$$ss_2 = \frac{s_2}{m_{hu2}}$$

But the solubility itself is a function of the partial pressure as well as temperature

$$S_2 = (C_1 + C_2 \, T_{L2} + C_3 \, T^2_{L2}) \ P_{hu2}$$

Data contained in Reference 7 were used to obtain the values of the constants, C_1, C_2. and C_3 by curve fitting   They are.

	C_1	C_2	C_3
OF_2	$+6.9361 \times 10^{-9}$	$-9\ 8294 \times 10^{-11}$	$+3.698 \times 10^{-12}$
B_2H_6	$+1\ 9382 \times 10^{-8}$	$-1\ 8383 \times 10^{-10}$	$+4\ 5944 \times 10^{-13}$

when the units are pounds, pounds per square foot, and rankine.

However, since
$$P_{hu2} = \frac{m_{hu2}\ Z_2\ R\ T_{u2}}{V_{u2}} \quad ,$$

$$s_2 = (C_1 + C_2\ T_{L2} + C_3 T^2_{L2})\ \frac{m_{hu2} Z_2 R\ T_{u2}}{V_{u2}}$$

Therefore,

(42)
$$ss_2 = (C_1 + C_2\ T_{L2} + C_3 T^2_{L2})\ \frac{Z_2 R\ T_{u2}}{V_{u2}}$$

During the cruse periods, no additional helium is added to the propellant tanks, so by the conservation of mass

$$m_{h1} = m_{h2}$$

Each of these total masses consists of two parts, the partial mass of helium in the ullage, and the mass of helium dissolved in the liquid

$$m_h = m_{hu} + m_{hs}$$

The amounts dissolved in the liquids are the solubilities times the liquid masses·

$$m_{hs} = sm_L$$

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$$\therefore \ m_{hu1} + s_1 m_{L1} = m_{hu2} + s_2 m_{L2}$$

However, s is unknown, so set

$$s_2 = ss_2 \ m_{hu2}$$

then $\qquad m_{hu1} + s_1 m_{L1} = m_{hu2} + ss_2 m_{hu2} m_{L1} = m_{hu2} \ (1 + ss_2 m_{L2})$

Solving for m_{hu2}

(43) $\qquad\qquad m_{hu2} = \dfrac{m_{hu1} + s_1 m_{L1}}{1 + ss_2 m_{L2}}$

   With the partial mass of helium in the ullage at condition (2) known, it is now possible to calculate the helium partial pressure by the gas law·

(44) $\qquad\qquad P_{hu2} = \dfrac{m_{hu2} Z_2 \ R \ T_{u2}}{V_{u2}}$

   This pressure plus the propellant vapor pressure (looked up in a table or calculated as a function of T_{u2}) is the total pressure in the propellant tank at equilibrium condition (2)

(45) $\qquad\qquad P_{t2} = P_{v2} + P_{hu2}$

   Finally, the actual solubility can be computed now that the helium partial mass is known

(46) $\qquad\qquad s_2 = ss_2 m_{hu2}$

The mass of helium dissolved in the liquid, and therefore lost as far as useful work is concerned, is

(47) $\qquad\qquad m_{hs2} = s_2 m_{L2}$

Table B-1 Nomenclature

A = inside surface area of helium tank (18 96), sq ft.

C = thermal capacity, $Btu/^\circ R$

$C1$ = thermal capacity of layer 1 of helium tank (7 67), $Btu/^\circ R$

$C2$ = thermal capacity of layer 2 of helium tank (7 27), $Btu/^\circ R$

$C3$ = thermal capacity of layer 3 of helium tank (7 27), $Btu/^\circ R$

C_f = flow resistance coefficient of fuel injector

C_{ox} = flow resistance coefficient of oxidizer injector

C_p = specific heat at constant pressure, $Btu/lb-^\circ R$

$C_{p_{he}}$ = specific heat at constant pressure of helium (1.255), $Btu/lb-^\circ R$

$C_{p_{vf}}$ = specific heat at constant pressure of fuel vapor (0 3), $Btu/lb-^\circ R$

$C_{p_{vox}}$ = specific heat at constant volume of oxidizer vapor (0 1333), $Btu/lb-^\circ R$

$C_{v_{he}}$ = specific heat at constant volume of helium (0.753), $Btu/lb-^\circ R$

$C_{v_{vf}}$ = specific heat at constant volume of fuel vapor (0 2), $Btu/lb-^\circ R$

$C_{v_{vox}}$ = specific heat at constant volume of oxidizer vapor (0 1333), $Btu/lb-^\circ R$

dE = differential amount of energy, Btu

D_f = equivalent fuel feedline diameter (0.5667), ft

dH = differential amount of enthalpy, Btu

dm_f = differential mass of liquid fuel, lb

dm_{he} = differential mass of helium, lb

dm_{ox} = differential mass of liquid oxidizer, lb

dm_v = differential mass of vapor, lb

dm_{vf} = differential mass of fuel vapor, lb

dm_{vox} = differential mass of oxidizer vapor, lb

D_{ox} = equivalent oxidizer feedline diameter (0.05667), ft

dQ   = differential amount of heat, Btu

dT   = differential change in temperature, ^{o}R

dT1   = differential change in temperature of layer 1 of helium tank, ^{o}R

dT2   = differential change in temperature of layer 2 of helium tank, ^{o}R

dT3   = differential change in temperature of layer 3 of helium tank, ^{o}R

dT_{he} = differential change in average temperature of helium in helium tank, ^{o}R

dT_{uf} = differential change in average temperature of ullage gases in fuel tank, ^{o}R

dT_{uox} = differential change in average temperature of ullage gases in oxidizer tank, ^{o}R

dU   = differential amount of internal energy, Btu

dV_{uf} = differential change in volume of ullage in fuel tank, cu ft

dV_{uox} = differential change in volume of ullage in oxidizer tank, cu ft

dW   = differential amount of work, Btu

$d\theta$ = time interval between calculations, seconds

e   = 2 71828

F   = thrust, lb

f_{f} = friction factor in fuel feed system, dimensionless

f_{ox} = friction factor in oxidizer feed system, dimenstionless

g   = gravitational constant (32 174), ft/sec^{2}

G   = acceleration, g's

h_{c} = convective heat transfer coefficient, $Btu/^{o}R-ft^{2}-sec$

I_{s} = specific impulse, $lb_{f}-sec/lb_{m}$

J   = mechanical equivalent of heat (778 16), ft-lb/Btu

k   = thermal conductivity, $Btu/ft-^{o}R-sec$

L_{f} = number of equivalent L/D's of fuel feedline and components, dimensionless

L_{ox} = number of equivalent L/D's of oxidizer feedline and components, dimensionless

m_f = mass flow rate of liquid fuel, lb/sec

$\dot{m}_{ox}$ = mass flow rate of liquid oxidizer, lb/sec

m_{he} = mass of helium in helium tank, lb

m_{hef} = mass of helium in ullage of fuel tank, lb

m_{heox} = mass of helium in ullage of oxidizer tank, lb

m_{lf} = mass of liquid fuel, lb

m_{lox} = mass of liquid oxidizer, lb

m_{sc} = mass of dry spacecraft, lb

m_{vf} = mass of fuel vapor, lb

m_{vox} = mass of oxidizer vapor, lb

N_f = Reynolds Number in fuel feedlines, dimensionless

N_{ox} = Reynolds Number in oxidizer feedline, dimensionless

P_c = chamber pressure, lb/ft^2 (absolute)

P_r = regulated pressure (43200), lb/ft^2 (absolute)

P_{tf} = total pressure in fuel tank, lb/ft^2 (absolute)

P_{tox} = total pressure in oxidizer tank, lb/ft^2 (absolute)

P_{vf} = partial pressure of fuel vapor, lb/ft^2 (absolute)

P_{vox} = partial pressure of oxidizer vapor, lb/ft^2 (absolute)

q = heat flux, Btu/sec

r = mixture ratio, o/f, dimensionless

R = gas constant for helium (386), ft-lb/lb-^{o}R

R_f = flow resistance in fuel circuit, ft^{-4}

R_{ox} = flow resistance in oxidizer circuit- ft^{-4}

R_t = thermal resistance, ^{o}R-sec/Btu

R_{t1w} = thermal resistance between layer 1 center-of-mass and wall surface in helium tank (2 536), $^\circ$R-sec/Btu

R_{t21} = thermal resistance between the centers-of-mass of layers 2 and 1 of the helium tank (5 072), $^\circ$R-sec/Btu

R_{t32} = thermal resistance between the centers-of-mass of layers 3 and 2 of the helium tank (5.072), $^\circ$R-sec/Btu

T1 = average temperature of layer 1 of helium tank wall, $^\circ$R

T2 = average temperature of layer 2 of helium tank wall, $^\circ$R

T3 = average temperature of layer 3 of helium tank wall, $^\circ$R

T_g = temperature of helium entering propellant tank ullages, $^\circ$R

T_{he} = average temperature of helium in helium tank, $^\circ$R

T_1 = temperature of liquid propellant, $^\circ$R

T_{1f} = average temperature of liquid fuel, $^\circ$R

T_{1ox} = average temperature of liquid oxidizer, $^\circ$R

T_{uf} = average temperature of gases in fuel tank ullage, $^\circ$R

T_{uox} = average temperature of gases in oxidizer tank ullages, $^\circ$R

T_w = average temperature of inside wall surface of helium tank, $^\circ$R

V_{he} = volume of helium tank (7.36), cu ft

V_{tf} = volume of fuel tank (27.37), cu ft

V_{tox} = volume of oxidizer tank (27 37), cu ft

V_{uf} = ullage volume in fuel tank, cu ft

V_{uox} = ullage volume in oxidizer tank, cu ft

X = length of thermal path, ft

Z_f = compressibility factor of helium in fuel tank ullage, dimensionless

Z_{he} = compressibility factor of helium in helium tank, dimensionless

Z_{ox} = compressibility factor of helium in oxidizer tank ullage, dimensionless

ΔT = temperature difference across a thermal resistance, ^{O}R

ΔT_{jt} = temperature change due to Joule-Thomson effect, ^{O}R

ΔV = velocity change during spacecraft maneuver, ft/sec

θ = time when computation made, seconds

μ_f = absolute viscosity of fuel, lb/sec-ft

μ_{ox} = absolute viscosity of oxidizer, lb/sec-ft

ρ_{lf} = density of liquid fuel, lb/cu ft

ρ_{lox} = density of liquid oxidizer, lb/cu ft

ρ_{vf} = density of fuel vapor, lb/cu ft

ρ_{vox} = density of oxidizer vapor, lb/cu ft

Symbols used in Cruise simulation equations, not given above

C1, C2, C3 = coefficients in solubility equation, dimensionless

m_h = total mass of helium in propellant (oxidizer or fuel) tank, lb

m_{h1} = total mass of helium in propellant (oxidizer or fuel) tank at condition 1, lb

m_{h2} = total mass of helium in propellant (oxidizer or fuel) tank at condition 2, lb

m_{hu} = mass of helium in propellant (oxidizer or fuel) ullage, lb

m_{hu1} = mass of helium in propellant (oxidizer or fuel) ullage at condition 1, lb

m_{hu2} = mass of helium in propellant (oxidizer or fuel) ullage at condition 2, lb

m_{hs} = mass of helium dissolved in propellant (oxidizer or fuel), lb

m_{hs1} = mass of helium dissolved in propellant (oxidizer or fuel), lb

m_{hs2} = mass of helium dissolved in propellant (oxidizer or fuel), lb

m_{L1} = mass of propellant (oxidizer or fuel) as liquid condition 1, lb

m_{L2} = mass of propellant (oxidizer or fuel) as liquid at condition 2, lb

m_{v1} = mass of propellant (oxidizer or fuel) vapor at condition 1, lb

m_{v2} = mass of propellant (oxidizer or fuel) vapor at condition 2, lb

P_{hu2} = partial pressure of helium in ullage at condition 2, lb/ft^2

s = solubility of helium in propellant (oxidizer or fuel), lb/lb

s_1 = solubility of helium in propellant (oxidizer or fuel) at condition 1, lb/lb

s_2 = solubility of helium in propellant (oxidizer or fuel)

ss_1 = solubility of helium in propellant (oxidizer or fuel) divided by mass of helium in ullage at condition 1, lb/lb/lb

ss_2 = solubility of helium in propellant (oxidizer or fuel) divided by mass of helium in ullage at condition 2, lb/lb/lb

T_{L2} = temperature of liquid propellant (oxidizer or fuel) at condition 2, ^{o}R

T_{u2} = temperature of ullage gases at condition 2, ^{o}R

V_{L1} = volume of liquid propellant (oxidizer or fuel) at condition 1, ft^3

V_{L2} = volume of liquid propellant (oxidizer or fuel) at condition 2, ft^3

V_t = volume of oxidizer or fuel tank, ft^3

V_{u1} = volume of ullage (oxidizer or fuel) at condition 1, ft^3

V_{u2} = volume of ullage (oxidizer or fuel) at condition 2, ft^3

Z_2 = compressibility factor of helium at condition 2, dimensionless

ρ_{L2} = density of liquid propellant (oxidizer or fuel) at condition 2, lb_m/ft^3

ρ_{v2} = density of propellant (oxidizer or fuel) vapor at condition 2, lb_m/ft^3

Thermal Resistance R32

Thermal Resistance R21

Thermal Resistance R1W

Boundary layer with connective film coefficient "h_c"

Helium at temperature "T_{he}" and paressure "P_{he}"
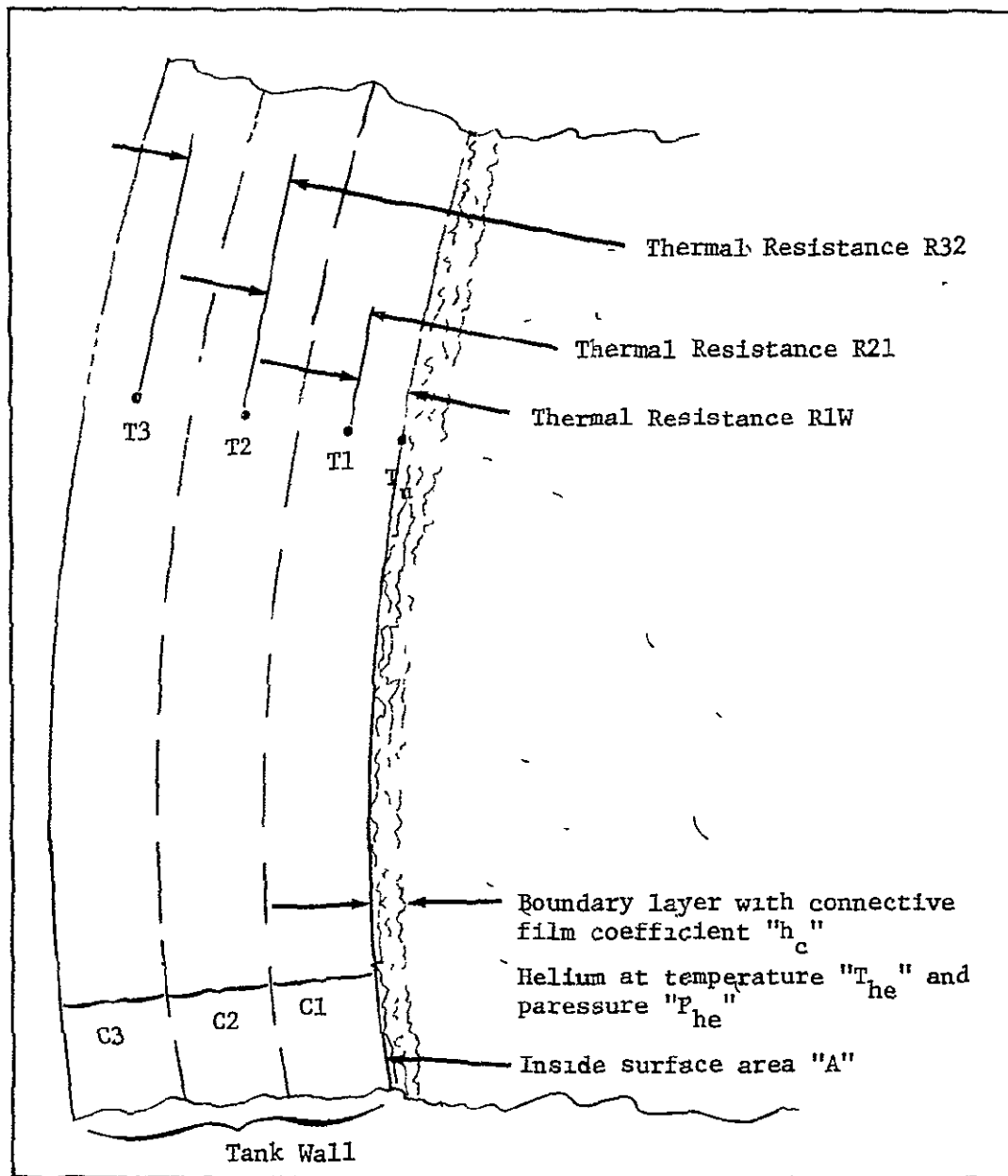
Inside surface area "A"

Tank Wall

Figure B-1 Model Used in Calculating
Heat Transfer from Tank Wall to Helium

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