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THE SIGNIFICANCE OF PARAMETERS AFFECTING THE HEAT TRANSFER TO THE LIQUID HYDROGEN IN THE SATURN S-IVB STAGE FOR THE LUNAR ORBIT RENDEZVOUS MISSION

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THE SIGNIFICANCE OF PARAMETERS AFFECTING THE HEAT TRANSFER TO THE LIQUID HYDROGEN IN THE SATURN S-IVB STAGE FOR THE LUNAR ORBIT RENDEZVOUS MISSION

By

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ABSTRACT

The Saturn S-IVB stage has a requirement for orbiting around the earth for up to 4.5 hours with approximately 60 percent of its initial propellant remaining at the end of the coast (prior to restart). Extensive analyses must be performed to insure that this requirement is met. Both the maximum and minimum heat transfer rates are important because the maximum rates affect the hydrogen boiloff losses and thus the initial propellant loading requirements. The minimum rates are important because the boiloff gases are used to maintain a minimum axial thrust level by venting the gases continuously through aft facing nozzles. This provides for a settling of the propellant throughout the orbital coast and alleviates the need for periodically venting the tank under zero gravity.

Described are the significance of the following parameters on the heat transfer rates to the liquid hydrogen: thermal conductivity, specific heat, density, surface emissivity and absorptivity, launch date and time, stage orientation and structural "heat leaks." Tank surface radiative property effects are shown where changes in solar absorptivity are six times more significant than equal changes in emissivity.

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CREDIT

All work discussed herein was the result of studies performed as part of the NAS7-101 contract to the NASA Marshall Space Flight Center, Huntsville, Alabama, for the development of the Saturn S-IVB stage.

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LIST OF ILLUSTRATIONS

1.0 INTRODUCTION

The S-IVB stage of the Saturn V space rocket (figure 1) has a requirement for orbiting about the earth (100 nautical miles) for up to 4.5 hours (3 orbits) with approximately 60 percent of its initial propellant (hydrogen and oxygen) remaining. This remaining propellant is required for a second burn of its 200,000 lb thrust engine which is to impart escape velocity to the Apollo spacecraft. (The initial 40 percent of the propellant is consumed during the first burn, resulting in an orbital coast.)

The S-IVB stage employs a unique system called continuous venting. Boiloff gases are vented through aft facing nozzles to provide a continuous "g" level axial thrust. This provides for a continuous settling of the propellant so that liquid is not vented overboard. In order to maintain a minimum specified g-level, a minimum boiloff rate must be maintained. This is unique in orbital propellant heating requirements where usually boiloff losses are minimized so that the affect on payload capability is minimized. This is still important, of course and it would be desirable to have the difference between the maximum and minimum boiloff rates be as small as possible while being slightly above the continuous venting requirement. The purpose of this paper is to discuss the factors that affect maximum and minimum propellant heating rates. Discussed are each of the major contributions to propellant heating from material properties, environmental effects, and stage design effects.

2.0 STAGE DESCRIPTION

A cut-away view of the S-IVB stage is shown in figure 2. The tank is divided into discrete regions for convenience of analysis. The largest region is the

S-IVB-1985

SATURN V SPACE VEHICLE

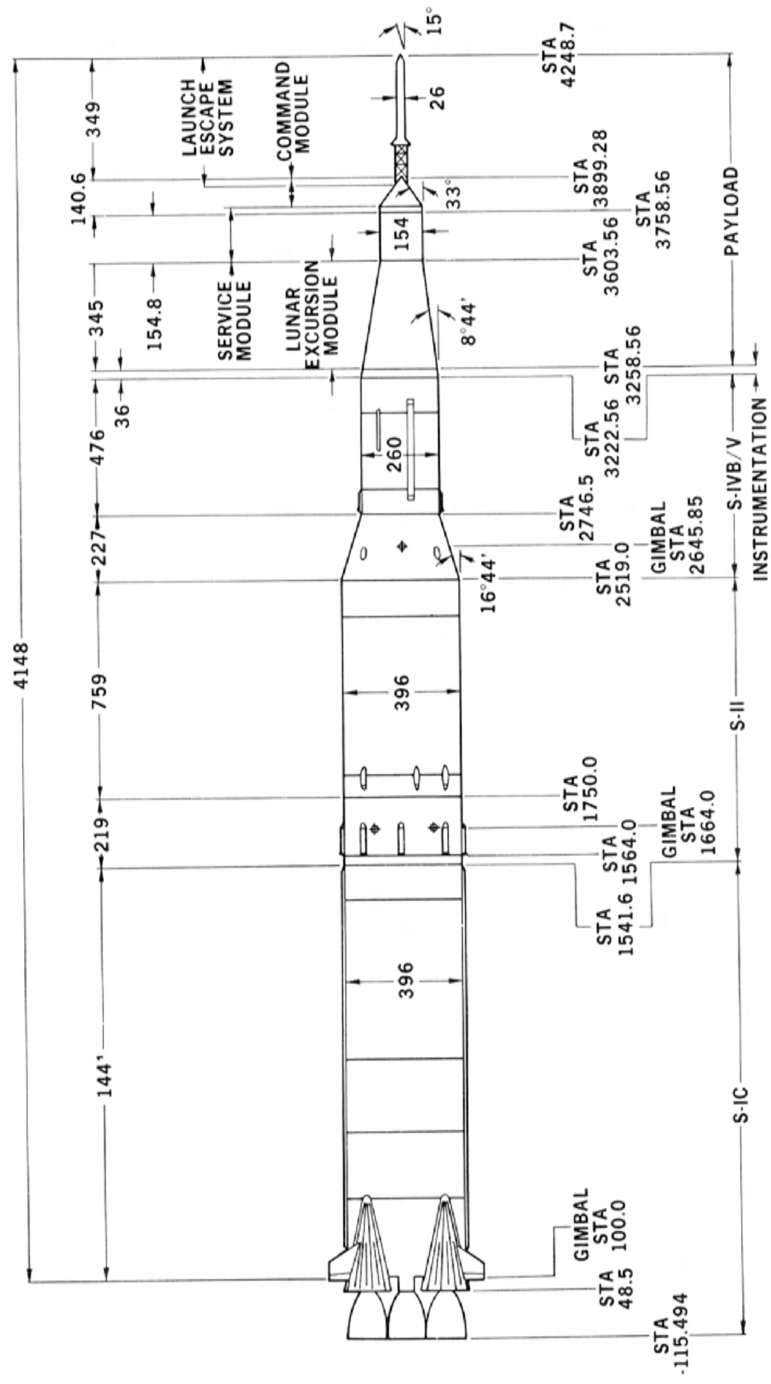
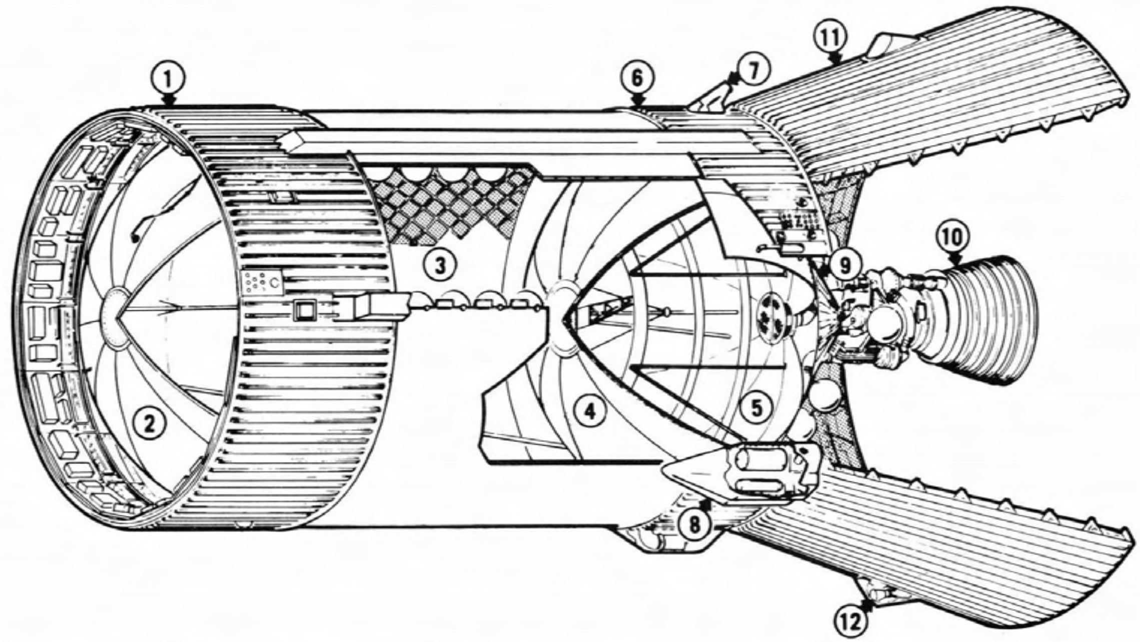


FIGURE 1

SATURN V/S-IVB STAGE
LOCATION OF MAJOR COMPONENTS



1	FORWARD SKIRT ASSEMBLY	7	ULLAGE ROCKET MOTOR
2	FUEL TANK FORWARD DOME	8	APS MODULE
3	FUEL TANK INTERIOR	9	THRUST STRUCTURE ASSEMBLY
4	COMMON BULKHEAD	10	J-2 ENGINE
5	OXIDIZER TANK INTERIOR	11	AFT INTERSTAGE ASSEMBLY
6	AFT SKIRT ASSEMBLY	12	S-II RETRO ROCKET

FIGURE 2

cylindrical section which has an area of about 1500 sq. ft. It is constructed of 2024-T6 aluminum, machine milled to form an internal waffle pattern for stiffening. The ribs of the waffle are approximately 0.75 inch thick and the size of each waffle is approximately 10 inches by 10 inches square. The interior of the cylindrical section is insulated with 1-inch thick reinforced polyurethane foam, cut to fit into each waffle segment. The foam density is a maximum of 5.5 lb/ft^3. The insulation thickness is the minimum practical value which will cover the ribs, thereby preventing direct heat transfer from the tank wall to the liquid hydrogen. A portion of an insulated tank is shown in figure 3. The forward dome is made of chemically milled aluminum gores welded to form the dome. Waffle stiffening is not required so the insulation thickness can be optimized with respect to heat transfer considerations. The common bulkhead is of honeycomb construction as shown in figure 2. The core is an insulating fiberglas honeycomb of 4 lb/ft^3 density. Either face is of aluminum construction, 0.032 inch thick on the hydrogen side and 0.055 inch thick on the liquid oxygen side. The common bulkhead joint between the two propellant tanks is shown in figure 4. An insulating block minimizes conductive heat transfer through the joint. Spheres, twenty-three inches in diameter, are mounted inside the tank. Each contains helium gas at approximately 3000 psia for pressurization of the liquid oxygen tank. These spheres each contribute to the "heat leaks." Also affecting the heat leaks are contributions from the joints of the forward and aft skirt to the hydrogen tank. Here heat is conducted from the relatively warm skin and stringer skirt to the propellant, through the insulation. (The problem of controlling the heat leak is minimized by use of internal insulation since the joints do not come in contact directly with the liquid hydrogen.)

INTERNAL INSULATION IN THE SATURN S-IVB
LIQUID HYDROGEN TANK

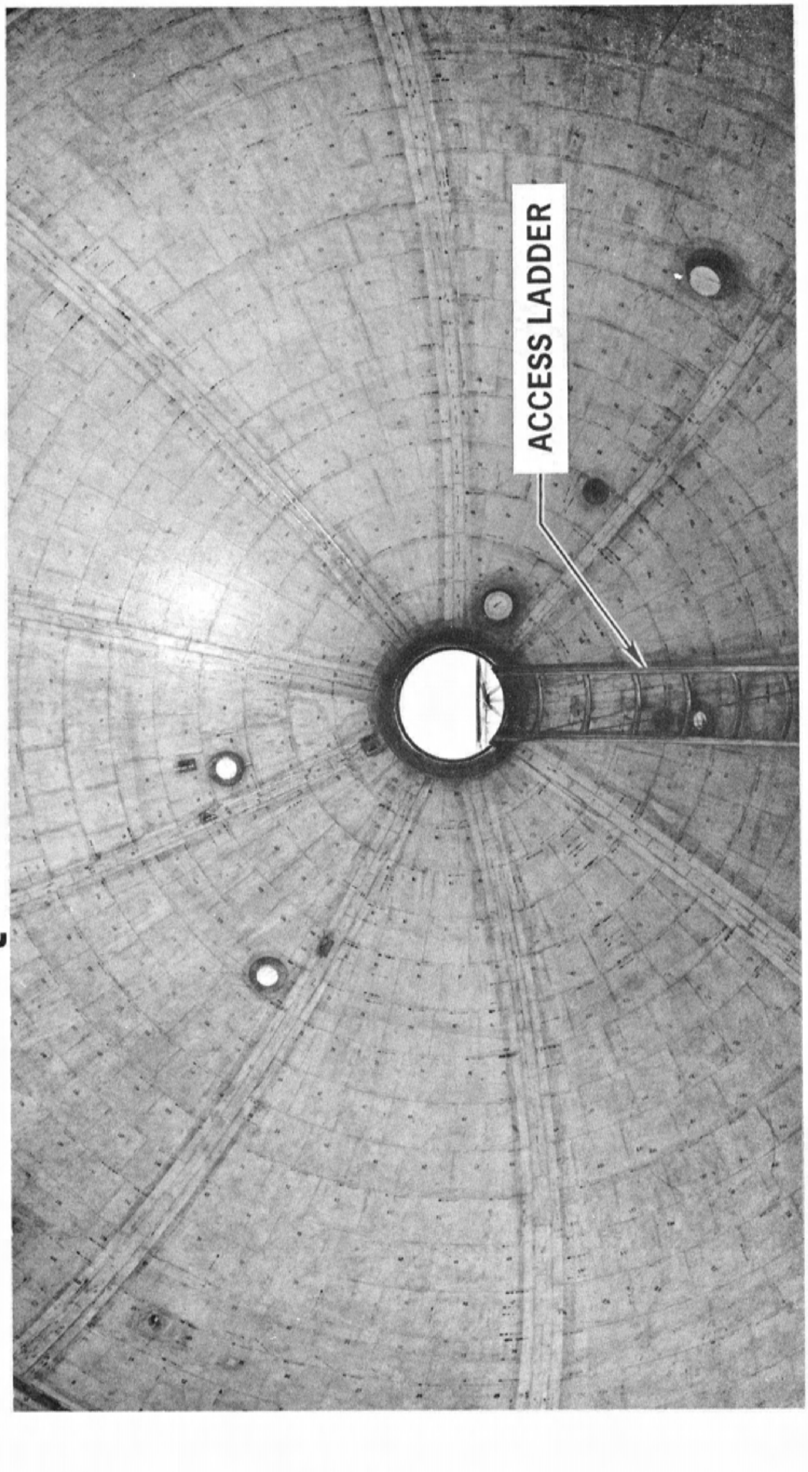


FIGURE 3

S-IVB STAGE COMMON BULKHEAD JOINT

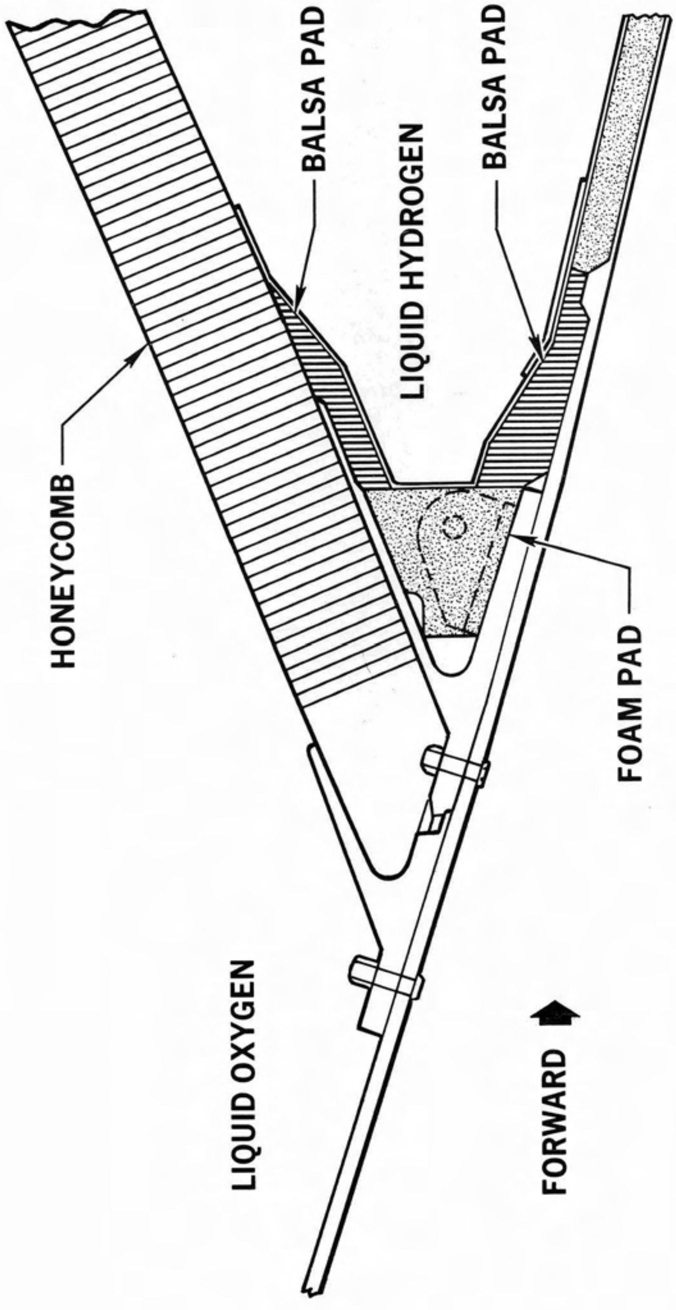


FIGURE 4

3.0 EFFECTS OF MATERIAL PROPERTIES

3.1 THERMAL CONDUCTIVITY

3.1.1 Internal Insulation

The thermal conductivity of the internal insulation is one of the most significant parameters affecting propellant heating. The maximum wetted area of the hydrogen tank is large (2500 sq. ft.) and thus, the area for heat transfer is large. The thermal conductivity is greatly affected by the direct exposure to liquid hydrogen. Numerous tests have been conducted (References a and b) to quantitatively measure this parameter; however, the spread in data is significant. This necessitated evaluating the possible effects. Figure 5 shows integrated propellant heating curves for the cylinder for a wide range of insulation conductivity. A maximum value for the conductivity is believed to be 0.035 Btu/hr-ft-$^{\circ}$F while the minimum value is believed to be 0.025 Btu/hr-ft-$^{\circ}$F based on test data (Reference b). (A nominal value of 0.03 Btu/hr-ft-$^{\circ}$F has been frequently used in arriving at data presented in this paper.) This range in conductivity results in a possible variation in integrated heating of from 620,000 to 540,000 Btu. Figure 6 shows heating rate curves for the cylinder for the maximum and minimum conductivities. The quantities from figures 5 and 6 form the principal factors for total propellant heating as indicated in figures 17 and 18.

3.1.2 Common Bulkhead

Shown in figure 2 is a cutaway of the S-IVB stage common bulkhead. This derives its name because it is the singular separation between the liquid hydrogen and liquid oxygen. The fiberglas honeycomb core provides sufficient

SATURN V/S-IVB CYLINDRICAL TANK CONTRIBUTION TO PROPELLANT HEATING SHOWING VARIATION WITH INSULATION CONDUCTIVITY

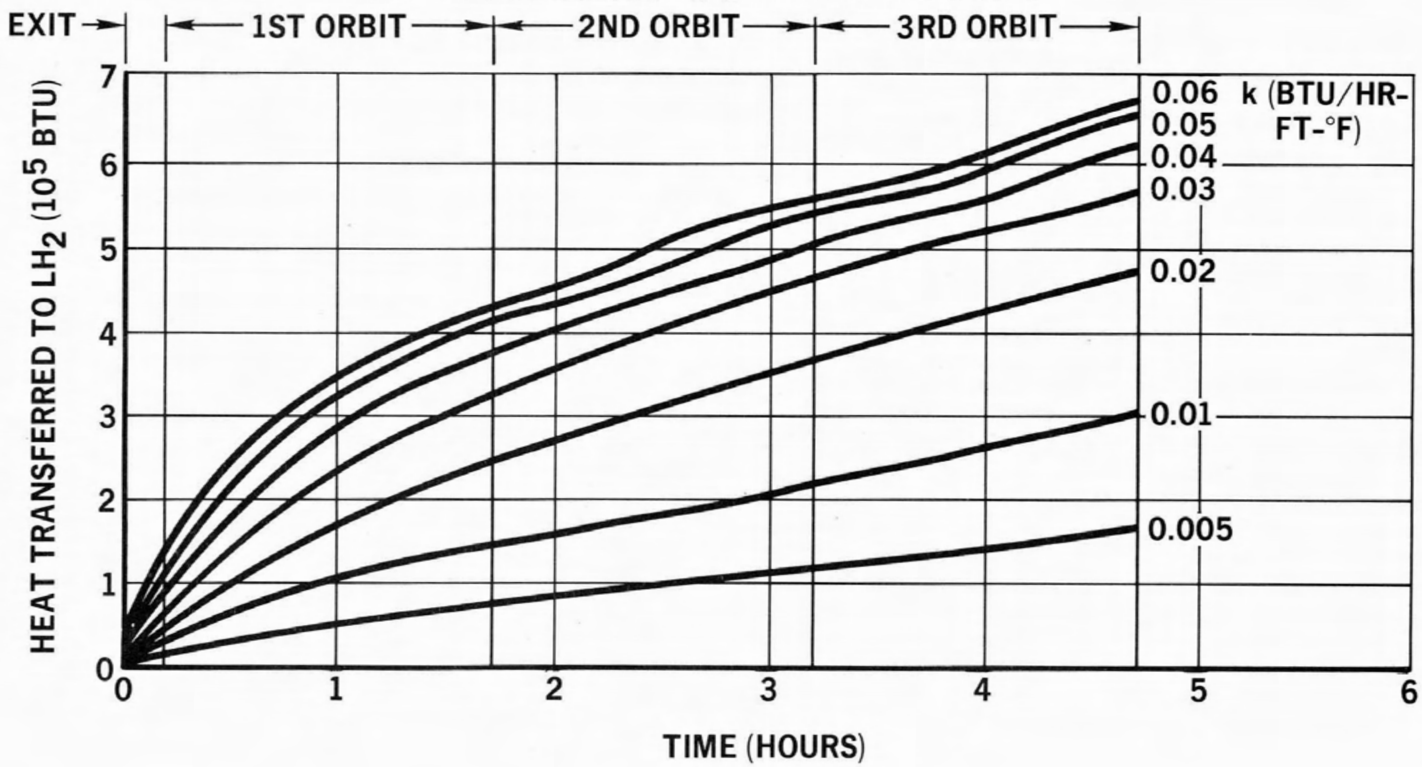


FIGURE 5

MAXIMUM & MINIMUM PROPELLANT HEATING RATES THROUGH THE LH_2 TANK CYLINDRICAL SECTION

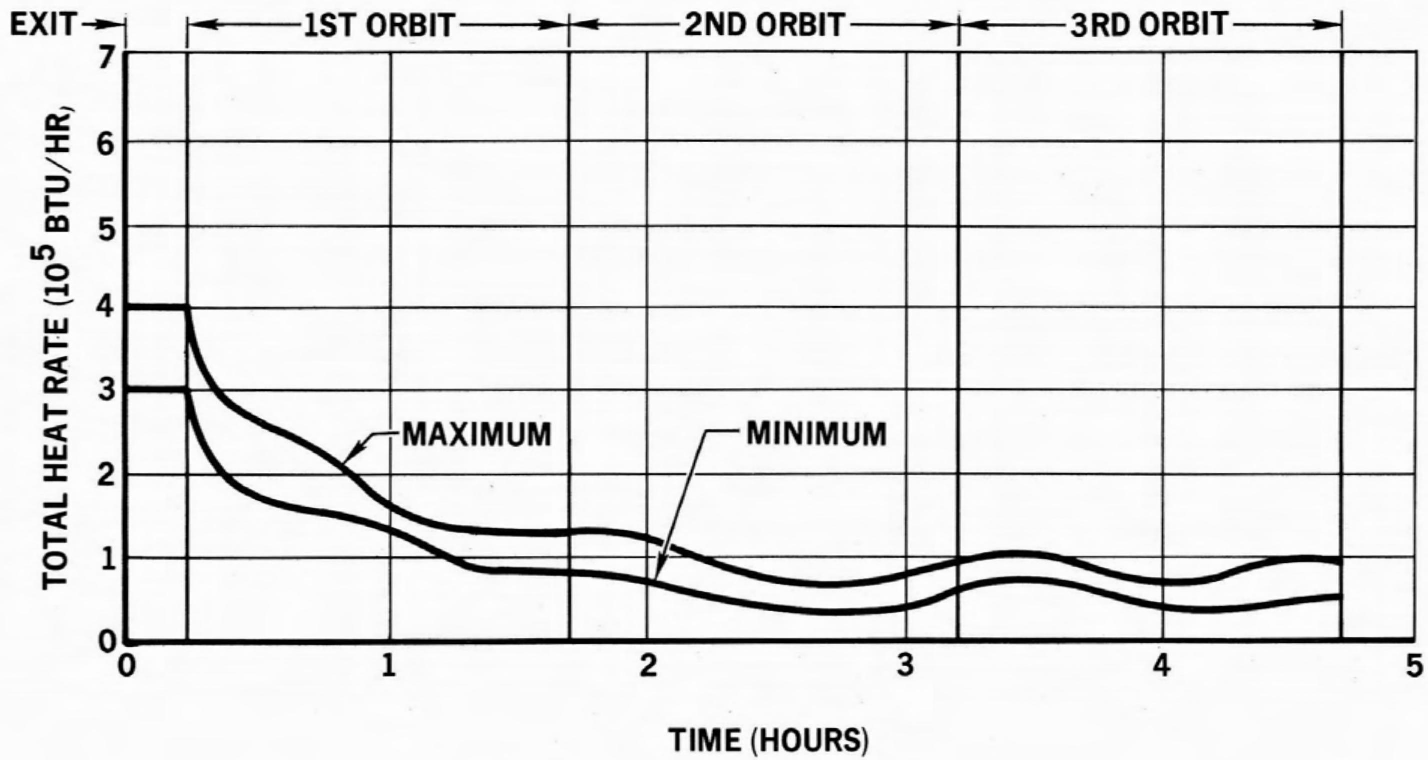


FIGURE 6

insulating qualities while exposed to a constant temperature differential of approximately 125°F (70°K). The equivalent conductivity is about 0.004 Btu/hr-ft-°F. This value includes the radiation between bulkhead faces, conduction through the cell walls and conduction through residual air which may be present in the cells. It is expected that such air would be highly cryopumped, but even if present as a gas it will have an extremely low conductivity in the temperature range of liquid hydrogen and liquid oxygen. It is also noted that even if gaseous air is present there should be no significant convection because of the near zero-g environment during orbital coast.

The common bulkhead heat transfer model incorporated the honeycomb core, the attach joint and the lower (non-cylindrical) portion of the hydrogen tank. This was done to attain greater accuracy in determining heat transfer in this region. Both the heat transfer rate and the total accumulated heat are shown in figure 7.

3.1.3 Aluminum Tank

The thermal conductivity of this structure varies somewhat with temperature, but has little effect on propellant heating since this conductivity is far greater than that of the insulation.

3.2 DENSITY-SPECIFIC HEAT PRODUCT

These two parameters are treated together for convenience because of their interrelationship in heat transfer calculations.

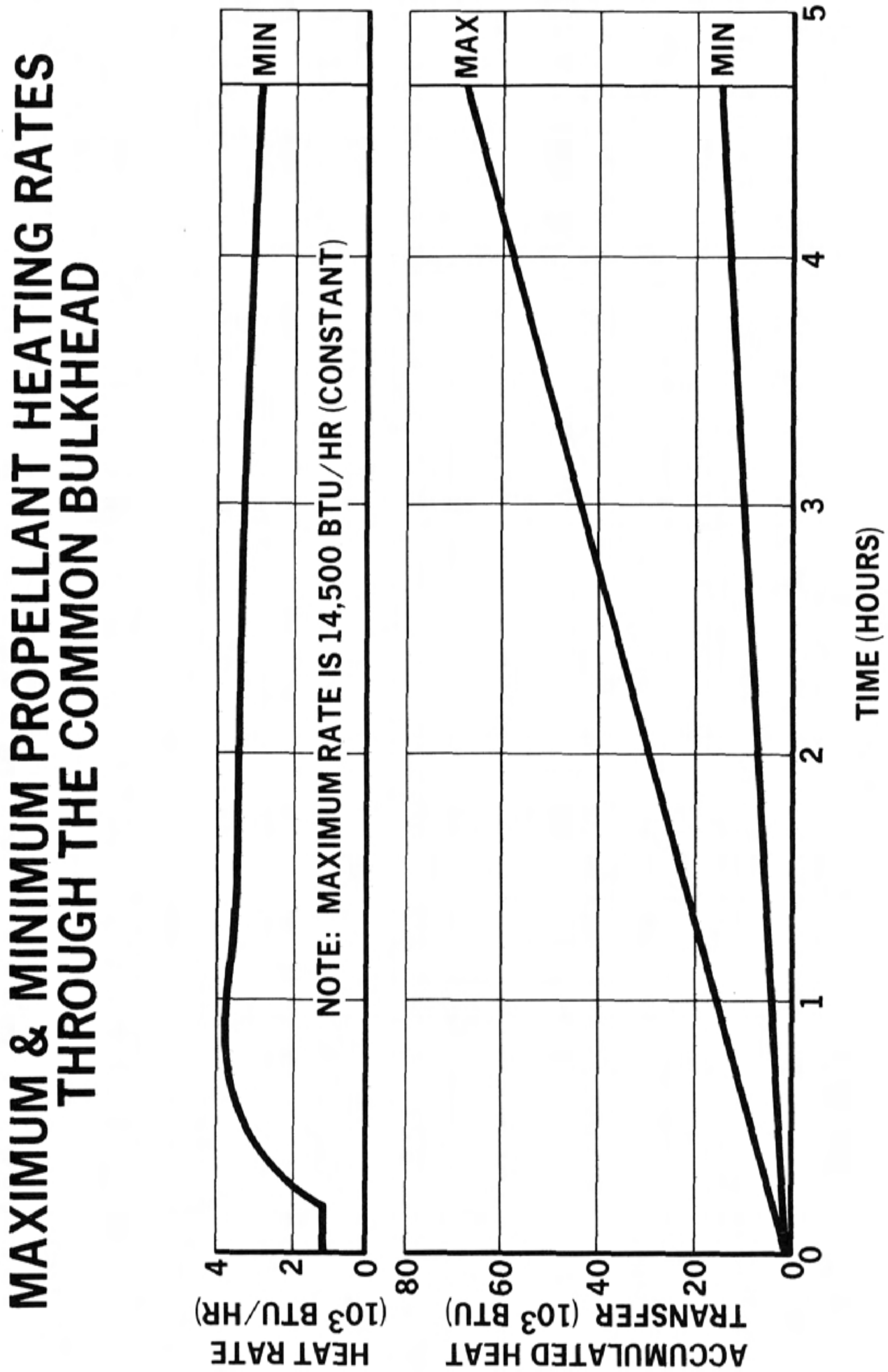

MAXIMUM & MINIMUM PROPELLANT HEATING RATES
THROUGH THE COMMON BULKHEAD
HEAT RATE
(10³ BTU/HR)
ACCUMULATED HEAT
TRANSFER (10³ BTU)
NOTE: MAXIMUM RATE IS 14,500 BTU/HR (CONSTANT)
MIN
MAX
MIN
TIME (HOURS)
0
1
2
3
4
5
0
2
4
0
20
40
60
80

FIGURE 7

3.2.1 Internal Insulation

The density of the insulation varies little during production (5.2 ±0.3 lb/ft^3). The specific heat has been measured and the combination of expected variations in these parameters will have less than 1 percent change in propellant heating. This is principally due to the small heat storage capacity and temperature change of the insulation.

3.2.2 Common Bulkhead

The fiberglas honeycomb has a low density (4.0 lb/ft^3) and specific heat (0.20 Btu/lb-$^\circ$F). Variations in these parameters and the small change in temperature due to constant contact with liquid hydrogen and oxygen on either side results in an insignificant effect on propellant heating (less than 1 percent).

3.2.3 Aluminum Tank Wall

The heat stored in the aluminum tank wall during ground hold plus the additional heat which is added during ascent is partially radiated to space and partially transferred internally to the liquid hydrogen. Since the tank wall maximum temperature is less than 150°F (339°K), radiation is greatly limited. In fact, the amount of heat that is transferred internally is approximately two to three times greater than that radiated. Arbitrary deviations in the density and specific heat product of ±10 percent and ±20 percent were analyzed. Figure 8 shows the results of the analysis on total accumulated heat.

SATURN V/S-IVB CYLINDRICAL TANK CONTRIBUTION TO PROPELLANT HEATING SHOWING VARIATION WITH ρC OF ALUMINUM TANK WALL

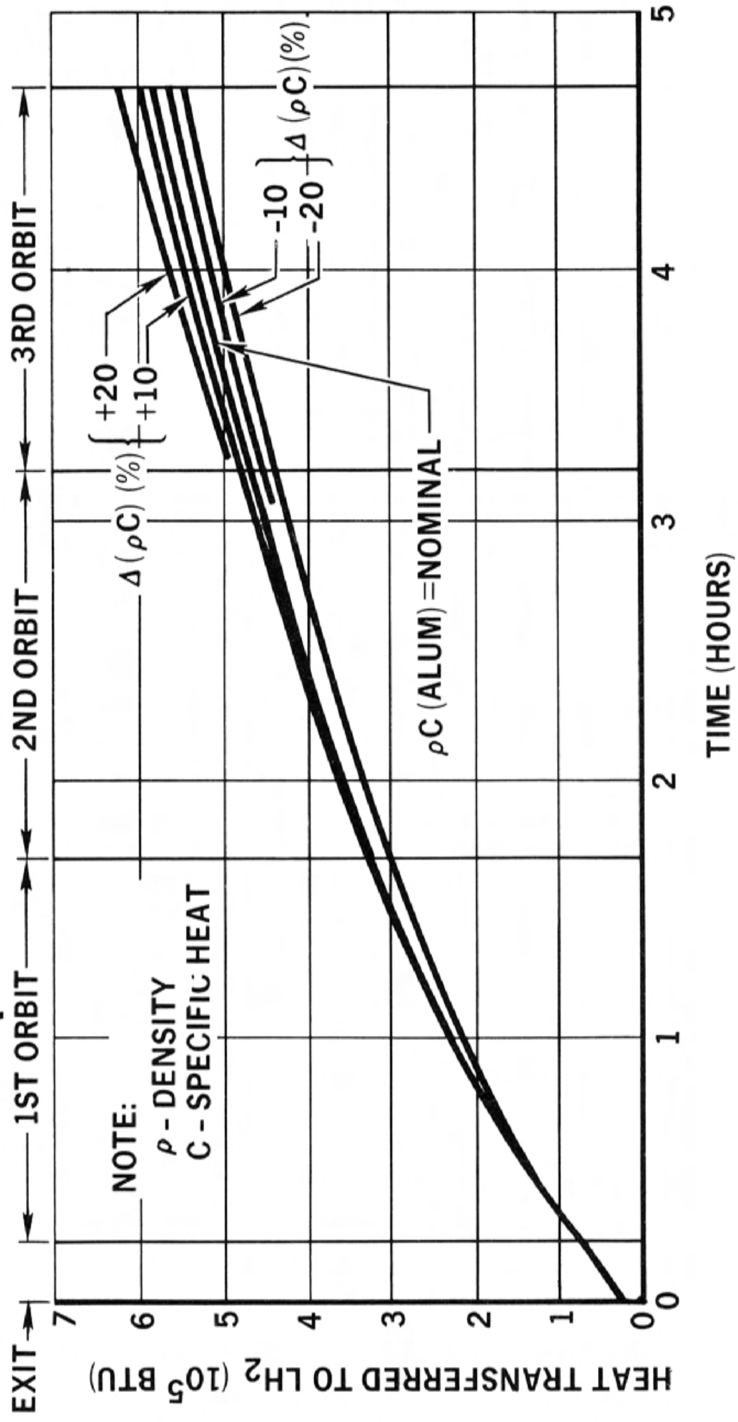


FIGURE 8

3.3 TANK SURFACE EMISSIVITY

3.3.1 Cylindrical Section

Figure 9 shows that increasing values of emissivity (ϵ) slightly reduces the heating rate in the early portions of the mission (first orbit) however, as the mission progresses, the lower values of emissivity provide reduced heating. (These studies were done to establish the significance regarding the coating selected for the outer surface. While special paints (aluminum silicone having α and ϵ values both of about 0.25) could have been selected, the minimum heating rates would be less than that required for the continuous vent system. For the white paint currently in use, a value of 0.9 for the emissivity (and 0.3 for the absorptivity) is assumed. Test data now indicate that the actual emissivity value ranges from 0.80 to 0.90 which is equivalent to a slight (considerably less than 1 percent) reduction in propellant heating.

3.3.2 Forward Dome

The forward dome is a unique situation compared to the cylindrical section because the basic dome is a monocoque aluminum structure compared to the waffle pattern construction of the cylinder. This allowed for the insulation thickness to be optimized whereas the cylinder required a minimum thickness of 1.0 inch to cover the ribs of the waffle by 0.25 inch. Since the principal source of heat transfer to the forward dome is radiation from within the structure (in the infrared wavelength band) the emissivity of the outer dome surface was predominant in controlling the heat transfer to the propellant in this region. Figure 10 shows the effect of emissivity on propellent heating.

SATURN V/S-IVB CYLINDRICAL TANK CONTRIBUTION TO PROPELLANT HEATING SHOWING VARIATION WITH TANK SURFACE EMISSIVITY

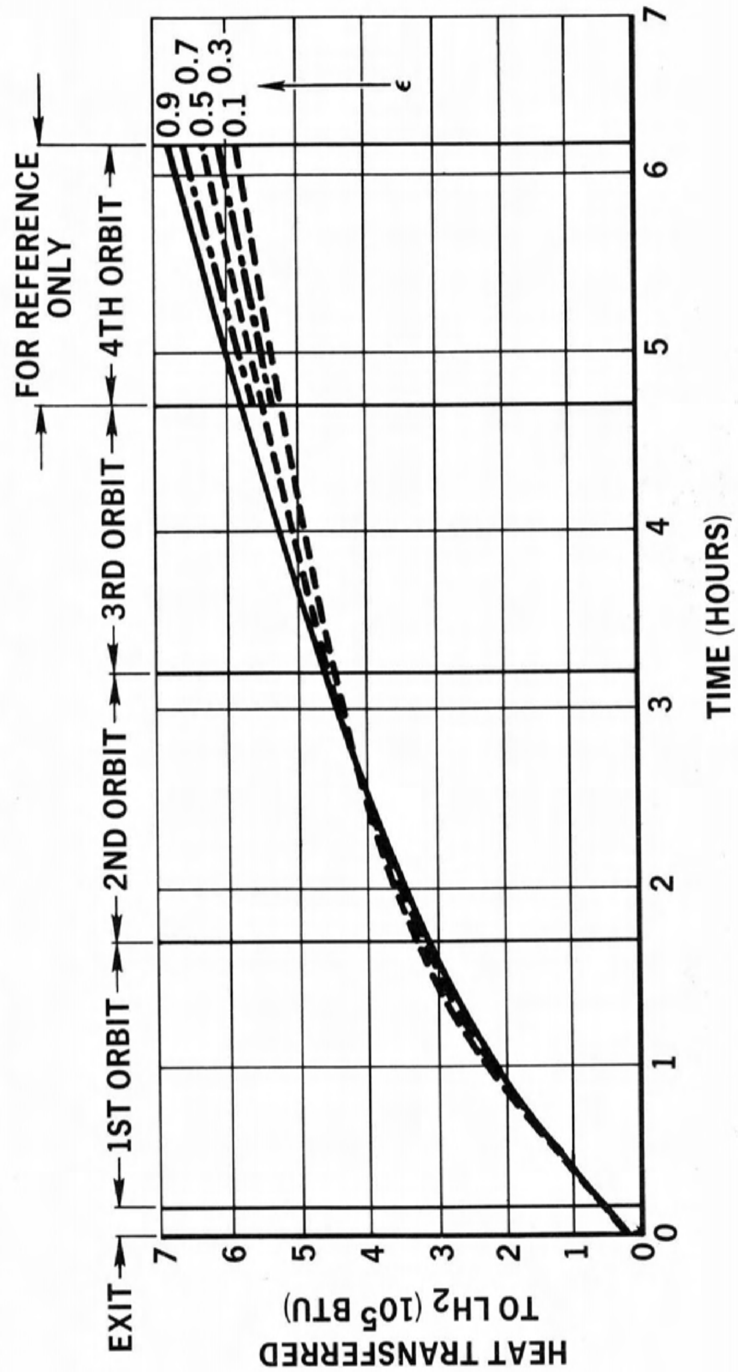


FIGURE 9

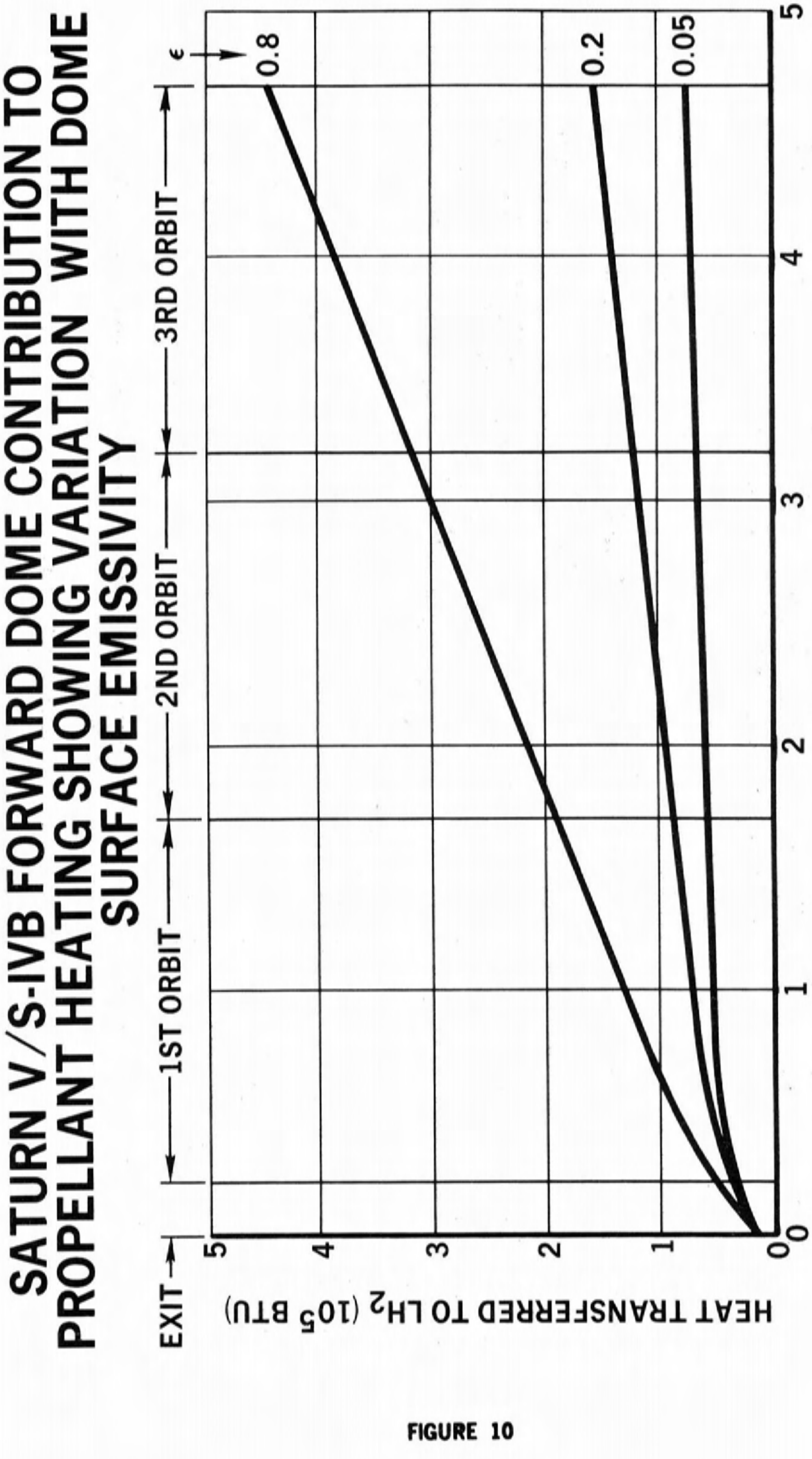

SATURN V/S-IVB FORWARD DOME CONTRIBUTION TO PROPELLANT HEATING SHOWING VARIATION WITH DOME SURFACE EMISSIVITY
EXIT
1ST ORBIT
2ND ORBIT
3RD ORBIT
ε
0.8
0.2
0.05
HEAT TRANSFERRED TO LH_2 (10^5 BTU)
0
1
2
3
4
5
0
1
2
3
4
5

FIGURE 10

The design value of 0.05 is achieved by attaching a single layer of aluminized mylar to the outer surface of the dome. It is noted that the initial steeper slope is due mainly to the energy stored in the dome structure at launch transferring to the propellant shortly after launch. The significance of this selection on payload weight is shown in figure 11. The term "equivalent weight" reflects both structural weight (which is equivalent to payload weight) and propellant boiloff (which becomes equivalent by multiplying by a factor of 0.43 to account for the fact that the propellant which is vented in orbit does not have to be accelerated to escape velocity). The initial design consisted of 1 inch thick insulation and a standard high emissivity paint on the outer surface. The first reduction was to consider the use of a low emissivity coating (in the range of 0.02 to 0.2). While this provided a substantial payload gain, a further gain was attained by reducing the insulation thickness.

Variations in propellant heating from the nominal design values are expected to be small since deviations from design value emissivity are likely to be small. The assumption up to this point has been that the interior of the forward dome is constantly or recurrently wetted by propellant, thereby absorbing nearly all of the heat stored in the aluminum structure. It is possible, however, that the hemispherical tank segment will be virtually free of liquid because of the venting thrust. Should this occur, the orbital heating (see figure 10) would be reduced to near-zero.

3.4 SOLAR ABSORPTIVITY

The cylindrical section of the hydrogen tank is the only portion of the tank that is exposed to solar heating. This includes both direct solar incident

SATURN V/S-IVB EFFECT OF INSULATION THICKNESS ON THE SYSTEM EQUIVALENT WEIGHT

FORWARD DOME

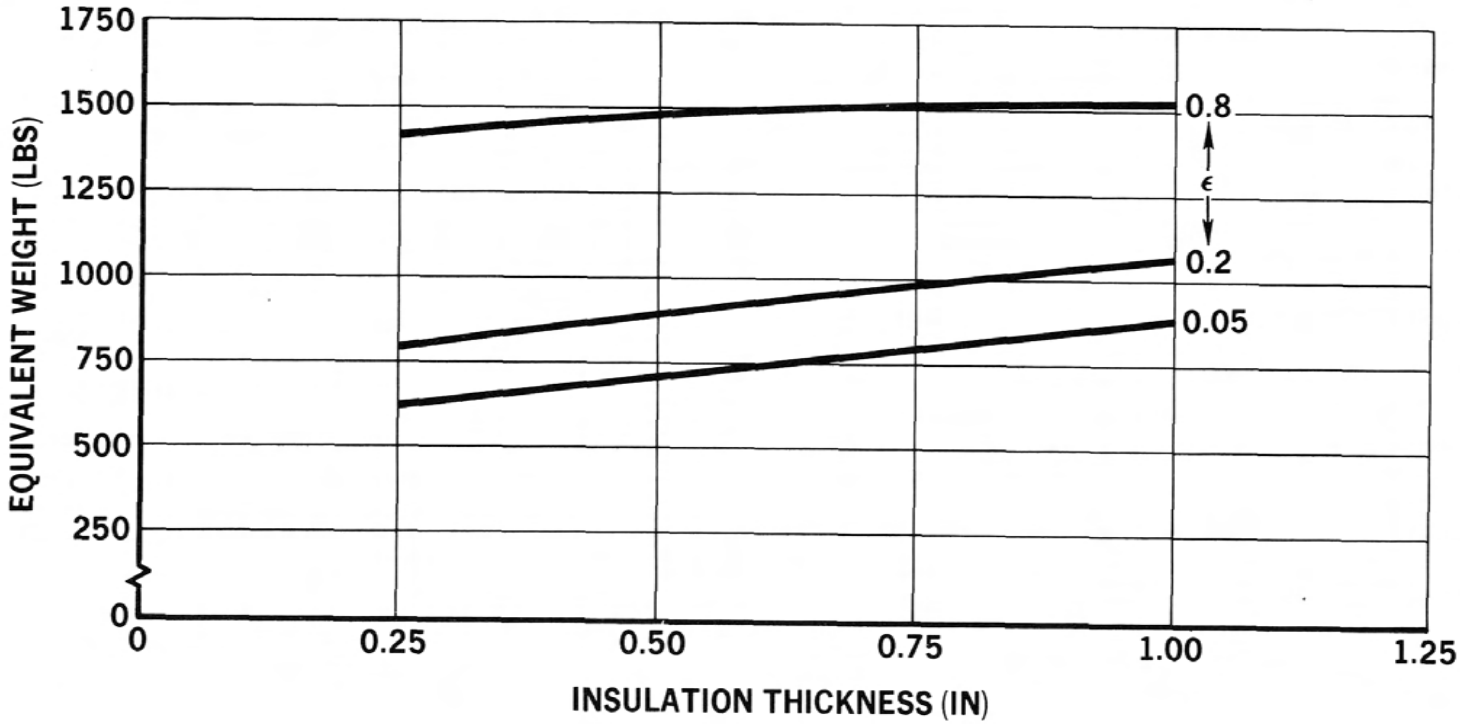


FIGURE 11

energy as well as the albedo (earth reflected solar radiation) which was assumed to be 0.4. As was the case noted in paragraph 3.3.1, studies were initially conducted to determine possible surface coatings that could result in minimum heating rates. However, too great a reduction could result in minimum heating rates less than the minimum allowable for the continuous vent system. Figure 12 shows the effect of absorptivity changes for a constant emissivity of 0.9. Figure 13 shows the combined effects of emissivity and absorptivity. It is noted that the absorptivity is six times more significant than equal changes in emissivity.

The nominal design values are 0.3 for solar absorptivity and 0.9 for emissivity. A realistic range of absorptivity values for white paints is approximately 0.20 to 0.40. The emissivity is in the 0.80 to 0.90 range. It can be seen (figure 13) that a significant deviation from the nominal propellant heating may occur depending on which white paint is selected for the cylinder.

4.0 ENVIRONMENTAL FACTORS

4.1 AERODYNAMIC HEATING

The extent of aerodynamic heating of the liquid hydrogen, although substantially less than the effect of initial energy stored in the tank wall at launch is significant. Trajectory dispersions during ascent were considered with the results showing that total dispersion effects on propellant heating can be neglected so that only the nominal ascent trajectory need be considered.

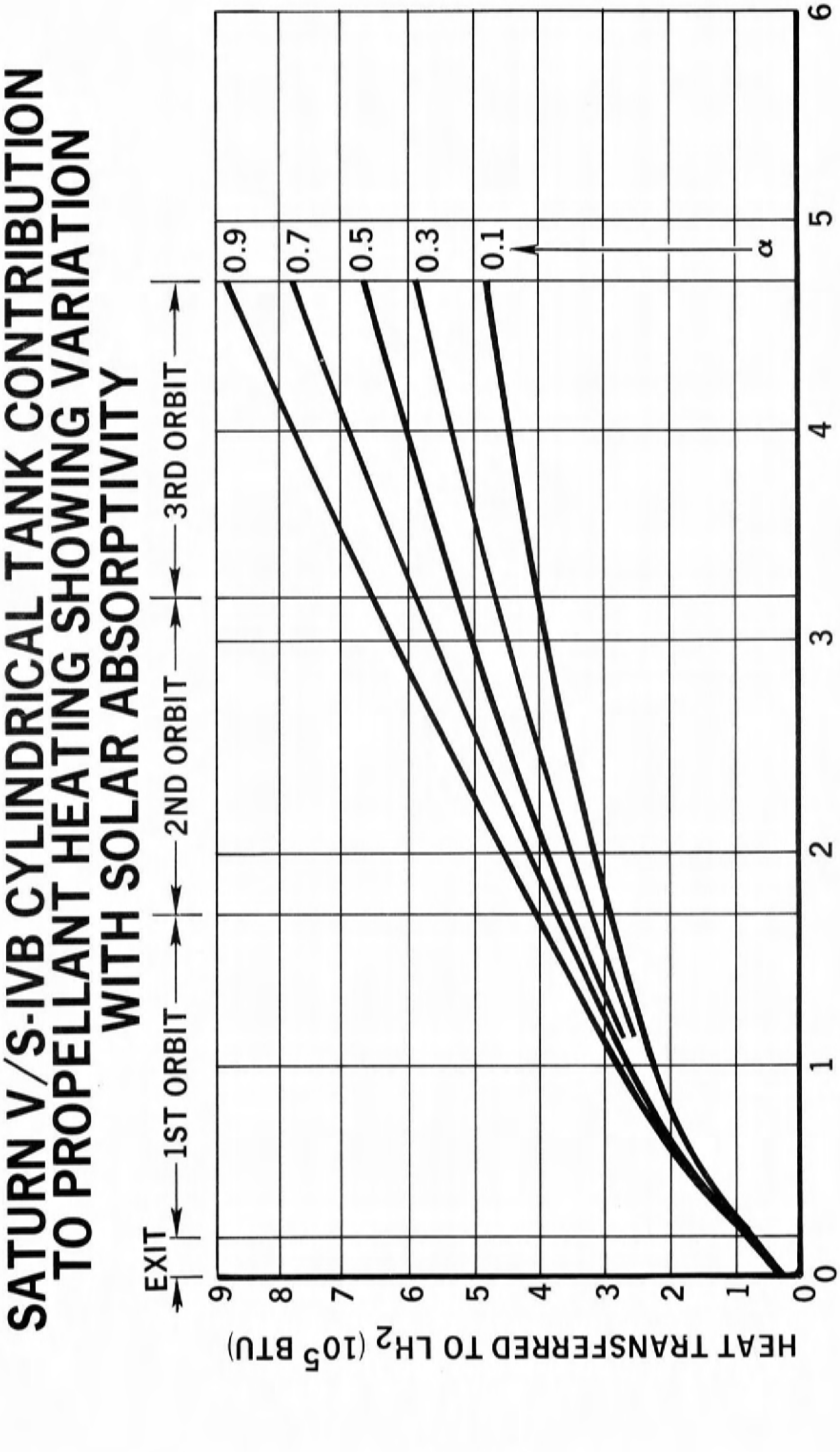

SATURN V/S-IVB CYLINDRICAL TANK CONTRIBUTION TO PROPELLANT HEATING SHOWING VARIATION WITH SOLAR ABSORPTIVITY
EXIT
1ST ORBIT
2ND ORBIT
3RD ORBIT
0.9
0.7
0.5
0.3
0.1
α
HEAT TRANSFERRED TO LH2 (10^5 BTU)
TIME (HOURS)
0
1
2
3
4
5
6
7
8
9

FIGURE 12

SATURN V/S-IVB CYLINDRICAL TANK CONTRIBUTION TO PROPELLANT HEATING SHOWING ABSORPTIVITY-EMISSIVITY DEPENDENCE

TOTAL MISSION DURATION- 4.7 HOURS

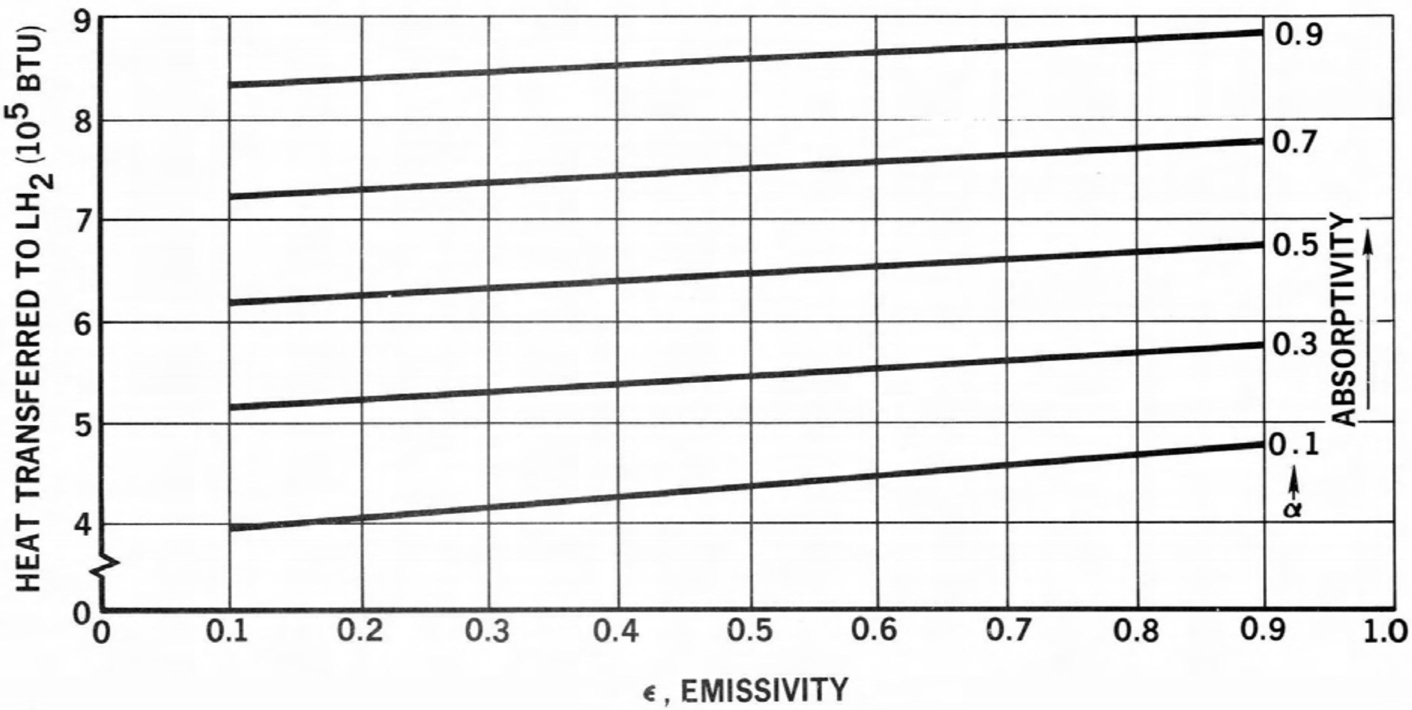


FIGURE 13

4.2 STAGE ORIENTATION

The S-IVB stage is nominally oriented in orbit such that the longitudinal axis is parallel to the velocity vector. Also, the same "side" of the stage continuously faces the earth except for brief maneuvers. These conditions form the reference point for this study. Other possible orientations were considered in an effort to determine where possible reductions in propellant heating could be made. The most obvious reduction was to align the longitudinal axis parallel to the solar rays (inertial orientation) so that direct solar incident radiation was minimized. This effect, combined with other orientations are shown in figure 14 and Table I. It is evident that significant reductions in propellant heating are possible by selection of orientation. Such a selection, however, imposes flight constraints which might complicate the maneuvers which are required at intervals during the mission. Nevertheless, this technique is worthy of further consideration in order to determine the nature and magnitude of the associated problems.

4.3 LAUNCH DATE AND TIME

It was initially believed that differences in the solar flux resulting from the many possible launch times would be significant. This would be true for high altitude orbits where the solar flux is predominant, but at low altitudes (100 n.mi.) albedo and earth radiation are as important as solar radiation as shown in Table I for $\gamma = 90^\circ$ and $\phi = 0^\circ$.

The change in heating rate for various launch dates is not large, therefore, maximum conditions (winter noon launch) are used. The possible extremes are shown in figure 15 for launch conditions resulting in maximum and minimum

SATURN V/S-IVB CYLINDRICAL TANK CONTRIBUTION TO PROPELLANT HEATING SHOWING VARIATION WITH ORBITAL VEHICLE ORIENTATIONS

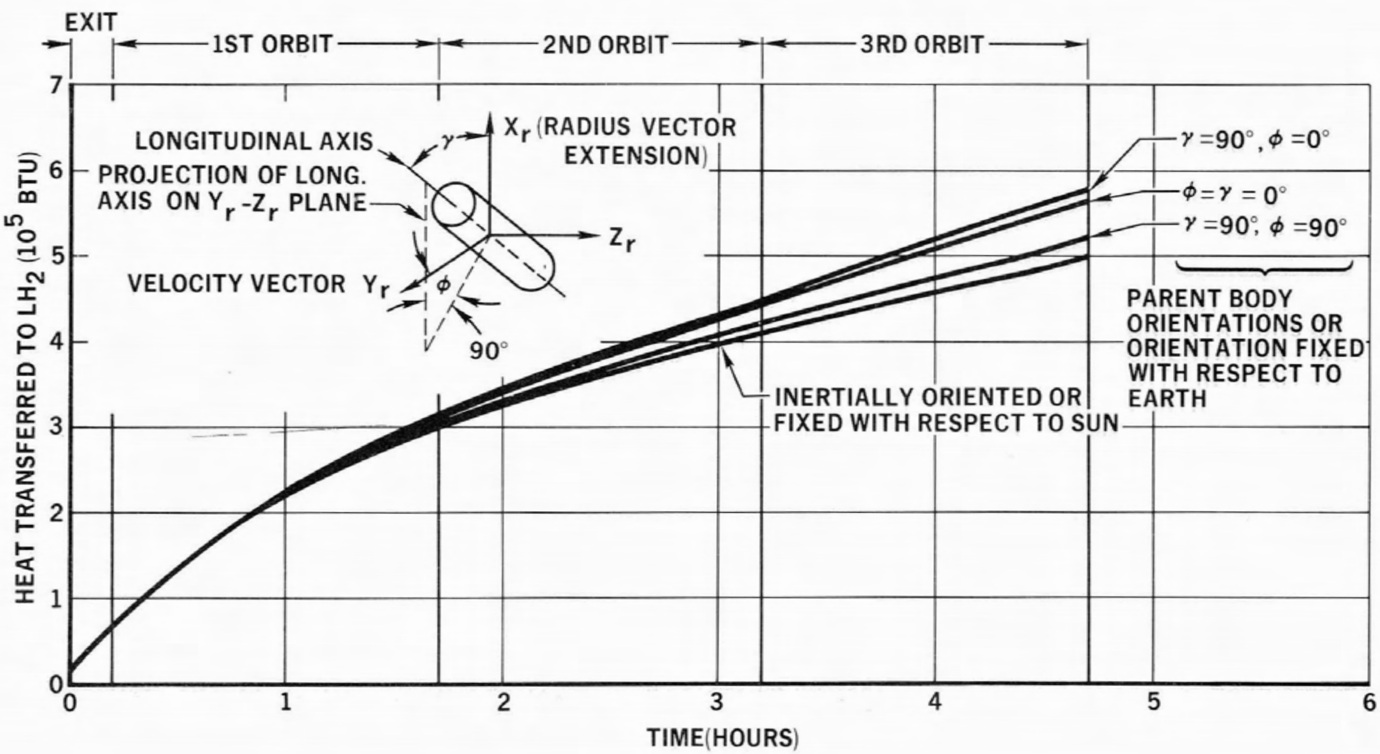
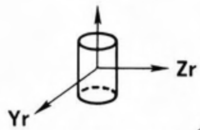
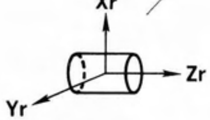
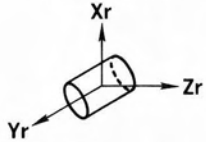


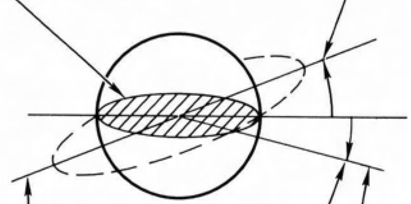
FIGURE 14

ABSORBED TOTAL ENERGY FOR ONE ORBIT PERIOD-88 MINUTES

ORIENTATION	SOLAR DIRECT BTU	SOLAR REFL. (ALBEDO) BTU	PLANETARY IR BTU
$\gamma=0, \phi=0$ 	6,000	8,000	46,695
$\gamma=90=\phi$ 	33,646	8,464	53,453
$\gamma=90,\ \phi=0$ 	54,077	8,478	51,885
INERTIALLY ORIENTED: VEHICLE LONGITUDINAL AXIS PARALLEL TO SOLAR RAY.	0	8,478	51,885

WINTER NOON CONDITION

$\alpha_s = 0.3\ \epsilon = 0.9$



NOTE: Xr-Yr IS ORBITAL PLANE.
γ AND ϕ AS DEFINED ON FIGURE 14

TABLE I

SATURN V/S-IVB CYLINDRICAL TANK CONTRIBUTION TO PROPELLANT HEATING SHOWING VARIATION WITH LAUNCH DATE & TIME

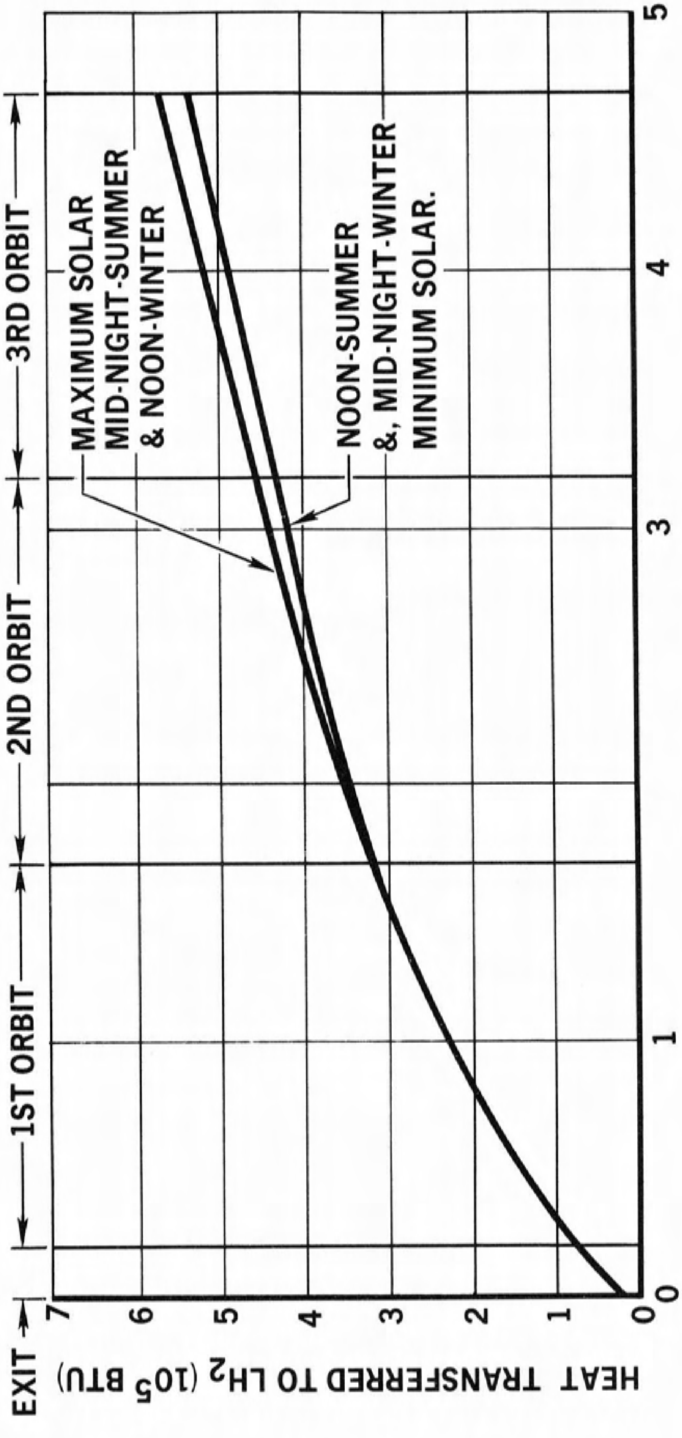


FIGURE 15

solar exposure (for the fixed inclination angle). Propellant heating should not be considered in the selection of launch date and time.

5.0 STAGE DESIGN FACTORS – HEAT LEAKS

Estimates of the heat transfer rates through the various heat leak paths have been made, based on a best estimate of the various factors. Therefore, only a single solution has been obtained, not maximums and minimums. Most of the factors tend towards the maximum, which is unconservative for the minimum condition. However, since all heat leaks comprise less than 10 percent of the total, it is believed that the effect on minimum rates is within a few percent.

5.1 FORWARD JOINT

As mentioned in paragraph 2.0, the forward joint contributes to the propellant heating by conduction from the relatively warm structure. In orbit, the forward skirt temperature cycles between $150^{\circ}F$ ($339^{\circ}K$) and $-230^{\circ}F$ ($128^{\circ}K$). The temperatures of course vary, depending on whether or not the side of the tank is exposed to or facing away from the earth and sun. Minimum and maximum conditions are both considered and the results form the inputs to a three-dimensional heat transfer computer program which calculates the temperature gradients along the structure, through the bolted joint, and into the insulation, thus resulting in a heating rate to the liquid.

5.2 HELIUM BOTTLES

The eight helium bottles form a direct heat short by virtue of conduction through the hydrogen tank. The one inch of insulation thickness extends up to the neck of the bottle and partially restricts the flow of heat.

5.3 FEED AND CHILL LINES

These two lines are connected directly to the hydrogen tank, the feed line serving the engine and the chill line providing hydrogen for childown.

Although the lines are vacuum jacketed, the connection forms a heat path into the hydrogen.

6.0 TOTAL STAGE RESULTS

Combining together all the factors that contribute to uncertainties in the maximum and minimum propellant heating result in figure 16 and 17. Figure 16 shows that the heating rate history varies approximately 70,000 Btu/hr from maximum to minimum and that this results in an approximate 300,000 Btu extreme (figure 17) between maximum and minimum conditions at the end of the 4.5 hour coast. If under stagnant fluid conditions all of this heat were to vaporize the liquid hydrogen, the uncertainty in the quantity of fluid lost would be 1500 lb. While this is an extreme condition, it does give the reader a "feel" for the problem.

7.0 CONCLUSIONS

The effect of the heat transfer on propellant boiloff rates, tank pressure, ullage thrust, etc. could be the subject of an entire paper. In approximate terms, the maximum heating rates result in approximately 3,000 lb of vented hydrogen or 7 percent of the initial loading. The minimum heating rates result in a predicted thrust level of 2×10^{-5} g which is slightly above the design requirements.

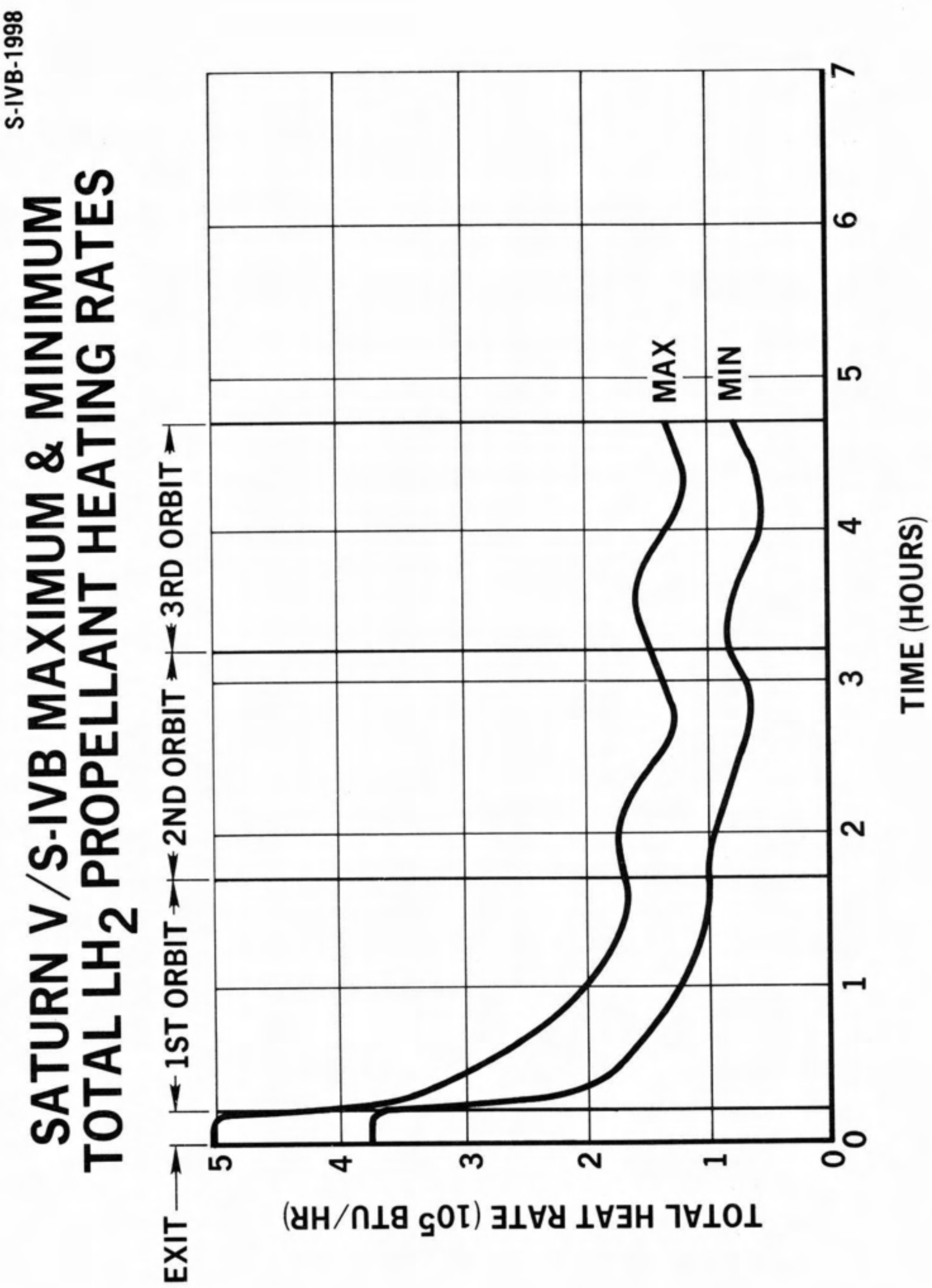

S-IVB-1998
SATURN V/S-IVB MAXIMUM & MINIMUM
TOTAL LH_2 PROPELLANT HEATING RATES
EXIT
1ST ORBIT
2ND ORBIT
3RD ORBIT
TOTAL HEAT RATE (10^5 BTU/HR)
MAX
MIN
TIME (HOURS)
0
1
2
3
4
5
6
7

FIGURE 16

SATURN V/S-IVB MAXIMUM & MINIMUM TOTAL LH_2 PROPELLANT HEATING

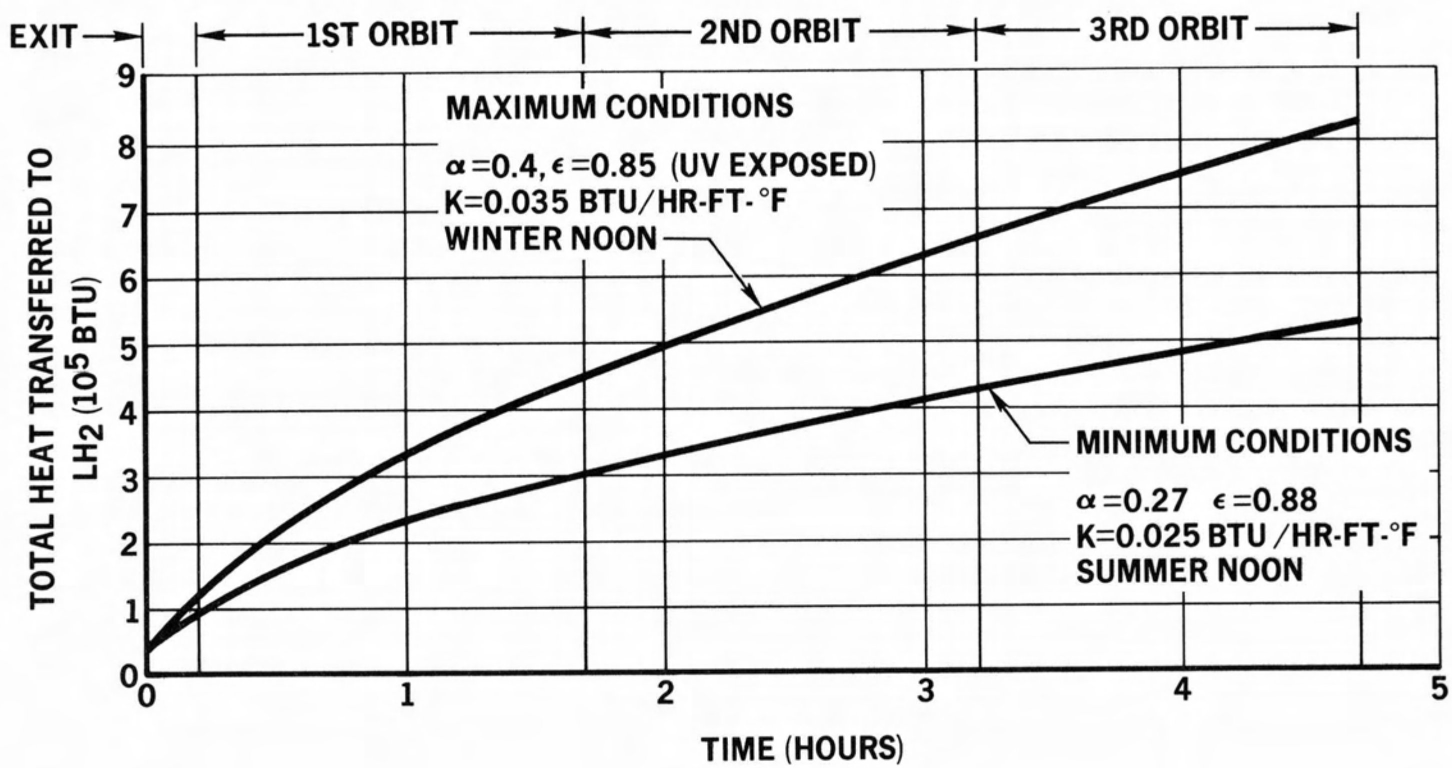


FIGURE 17

The S-IVB stages are well instrumented to obtain data for the verification of the analyses associated with this paper. It is expected that the data will help to narrow the band of propellant heating predictions so that boiloff losses are minimized while yet maintaining the minimum propellant heating rates within requirements.

8.0 REFERENCES

(a) Douglas Report No. SM-42545, Evaluation of the Saturn S-IV Internal Insulation in the Eight-Foot Scale Tank, November, 1962.

(b) Douglas Report No. SM-47195, Improvement Program for the Saturn S-IVB LH_2 Tank Internal Insulation (to be released).