Mr. von Tiesenhausen

29 April 1960

Report No. DLM-TN-35-60

ORBITAL REFUELING

A FEASIBILITY STUDY AND DESIGN CONCEPT

PHASE II

FOR INTERNAL USE ONLY

ARMY BALLISTIC MISSILE AGENCY

REDSTONE ARSENAL, ALABAMA

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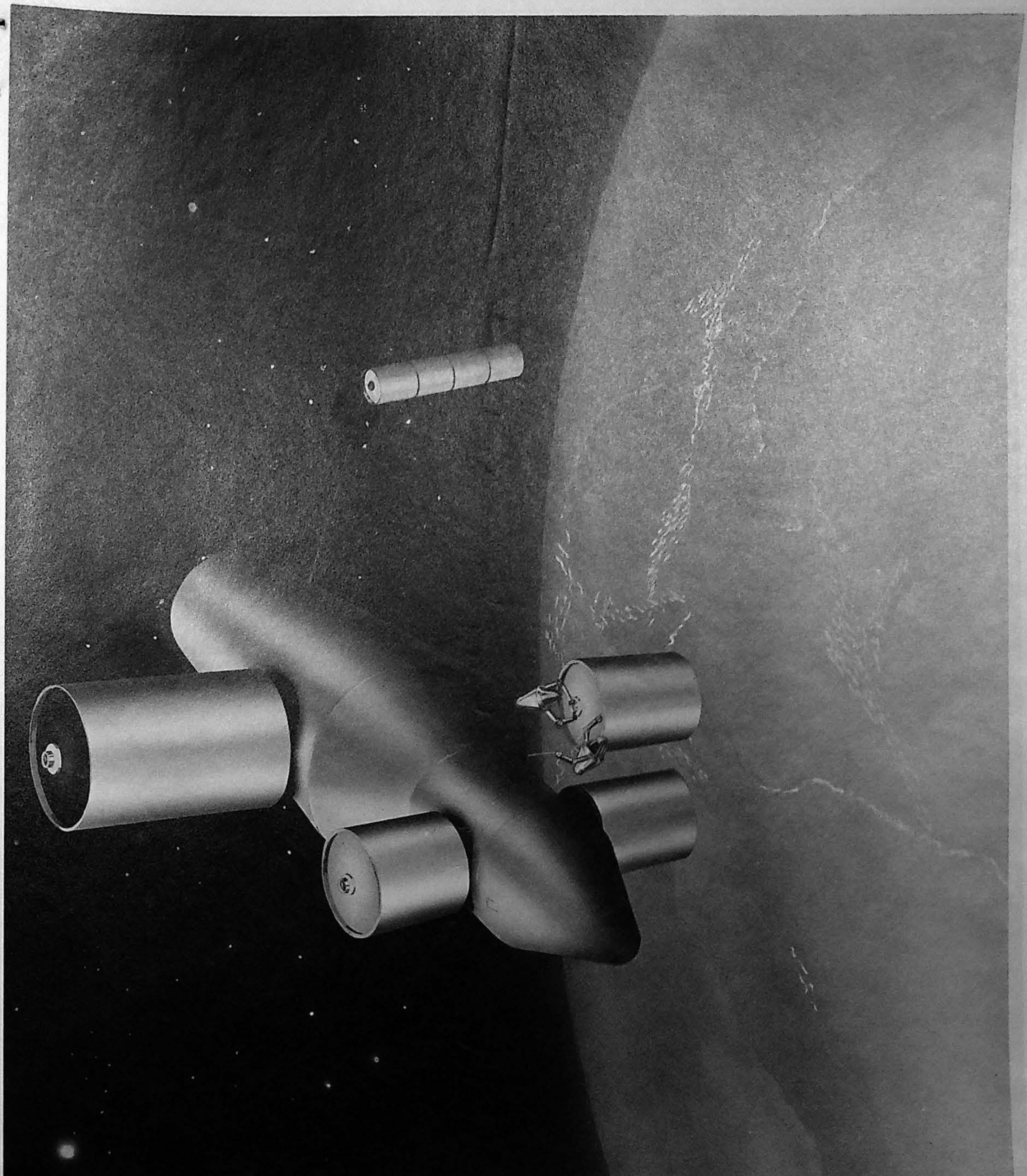
PHASE II

MECHANICAL BRANCH
SYSTEM SUPPORT EQUIPMENT LABORATORY
ARMY BALLISTIC MISSILE AGENCY



ORBITAL REFUELING ASSEMBLY

FRONTISPIECE

ABSTRACT

The objective of this phase of the orbital storage and refueling study is to present a more detailed investigation into the propellant transfer scheme proposed in Phase I, define certain specific hardware requirements for prelaunch, in-flight, and orbital storage, and to present an outline of an operating procedure for the orbital refueling crew.

The feasibility of orbital transfer of the stored propellants has been shown. The hydrazine, nitrogen tetroxide, and liquid oxygen can be expelled in approximately six (6) minutes per tank. The liquid hydrogen requires a greater period of time for transfer and must receive a definite amount of thermal energy from an external source to supplement the helium working fluid.

It is proposed that the above mentioned thermal energy be obtained from solar radiation impingement on the tank end bulkhead. Solar radiation and chemical sources were investigated with the former appearing more advantageous due to design simplicity, reliability, overall system weight and orbital handling problems.

Specific hardware requirements covered by this report indicate readily acceptable solutions are available for the described problem areas.

An orbital operation procedure has been established which is felt to be entirely feasible. The assembly, transfer, and disassembly bar graph presented herein is considered adequate and lends itself to modification without adversely effecting storage and transfer of the LH_2.

Sufficient data has been established which determines that orbital storage of liquid hydrogen is feasible for periods in excess of the 1440 hours presented throughout these studies.

FOREWORD

The information contained herein is intended primarily for in-house use in determining the practicality of refueling a space vehicle in a low geocentric orbit.

This presentation is based on the preliminary design studies presented in Report No. DLM-TN-21-60, Orbital Refueling, a Feasibility Study and Design Concept, Phase I, dated 15 March 1960, see Ref. 1. Subject matter presented herein is a continuance of design parameters previously established by the above report.

This proposal is preliminary in scope and is not intended as final design criteria.

SYMBOLS

A = Area, ft^2

c_p = Specific heat, BTU/lb-°F

d = Distance, ft

h = Convective heat transfer coefficient, BTU/hr-ft^2-°F

k = Thermal conductivity, BTU/hr-ft-°F

Q = heat, BTU/hr-ft^2

Q_r = Reradiated energy, BTU/hr-ft^2

Δt = Temperature difference, °F

w = Weight flow rate, #/hr

a = absorptivity

ε = emissivity

σ = Stefan-Boltzman constant, BTU/ft^2-hr-$°R^4$

θ = Temperature excess after time τ

θ_i = Temperature excess at $\tau = 0$

τ = elapsed time, hrs

α = thermal diffusivity, ft^2/hr

J_o (ns) = Bessel function of first kind, zero order

J_1 (ns) = Bessel function of first kind, first order

X_v = ns = roots obtained from $\frac{ns}{bs} = \frac{J_o (ns)}{J_1 (ns)}$

r = Distance from center at which temperature is to be determined, ft

s = Radius of cylinder or sphere or distance to median plane, ft

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I INTRODUCTION

This report constitutes Phase II of the study to determine the practicality of orbital storage and refueling of cryogenic propellants. Phase I was published under DLM-TN-21-60 on 15 March 1960.

The objective of this phase of the orbital storage and refueling study is to present a more detailed investigation into the propellant transfer scheme proposed in Phase I, define certain specific hardware requirements for prelaunch, in-flight, and orbital storage, and to present an outline of an operating procedure for the orbital refueling crew or manipulators.

The methods proposed are the results of further studies based on the parameters and design concept established in the Phase I Report which is considered requisite reading for maintaining project continuity.

II PROPELLANT TRANSFER FOR LIQUID HYDROGEN

A. General

Two systems for expelling the propellant from the supply tanks were proposed in the Phase I Report. The system using compressed helium as a working fluid is preferred and was investigated in more detail in this study.

In addition to expanding the high pressure (750 psia) helium into the storage tank, a considerable amount of thermal energy must be added to the helium to provide adequate expansion. Solar and chemical energy are two likely sources of thermal energy.

Any energy supply employed should be selected in accordance with the specific requirements of the given system. The following criteria for this transfer system, in order of their relative importance, were established as:

1. Reliability and design simplicity
2. Weight
3. Acceptable propellant transfer rate and control
4. Minimum work maneuvers.

In accordance with the above, solar and chemical energy systems were investigated to determine a suitable thermal energy source.

B. Solar Energy

The method of using solar energy as a heat source requires orientation of the LH_2 tank, after it is connected to the space vehicle, so that the free end is facing the sun. The insulation is then removed from the bulkhead.

Radiation may strike the bulkhead directly or be concentrated on a small area by the use of a concave reflector. The latter was considered but eliminated due to reradiation problems and design complications. Hence, the following discussion is limited to direct solar radiation upon the tank bulkhead.

During expulsion, an effort is made to absorb as much radiation as possible. Bulkhead heat loss by infrared reradiation tends to reduce the net heat input, thus increasing the expulsion time. Therefore, it is important that the bulkhead temperature be kept at a level, about $250^{\circ}K$ or less, which will minimize this reradiated energy. The bulkhead temperature can be kept at a reasonable level if the helium gas is able to readily receive thermal energy from the bulkhead wall. Of the three basic modes of heat transfer, only forced convection appears capable of removing heat at an acceptable rate.

To achieve the high rates of heat transfer possible with forced convection, a thin metal plate is placed parallel to the bulkhead at a distance of one-eighth inch, (see Fig. 1). Helium gas is expelled from small openings in the center tube, where it joins the bulkhead, and flows at high velocity radially out into the space between the bulkhead and the plate. The gas absorbs heat thru forced convection as it flows over the entire bulkhead surface and discharges into the tank upon reaching the cylindrical tank wall.

A heat transfer investigation of the hydrogen tank bulkhead during expulsion of the LH_2 was made to determine what bulkhead temperatures might be expected and if infrared reradiation is appreciable. The investigation was based on the following assumptions:

1. Steady state heat transfer conditions exist.

2. Reradiation will be negligible so that a uniform net heat input exists over the entire bulkhead surface. Reradiation is considered neglibible if it is less than 5 per cent of the solar energy input to the bulkhead.

3. Bulkhead solar absorptivity, a = 0.90.

4. Bulkhead infrared emissivity, $\epsilon = 0.3$.

5. Solar constant = 420 BTU/hr-ft^2.

6. Helium gas stored at 750 psia and 18°K.

7. Pressure of the helium flowing along bulkhead is maintained at 25 psia.

8. Effective expulsion temperature of helium 150°K.

Additional heat (34,300 BTU) must be added to the helium to allow for heat transfer to the tank walls (see Phase I, Table III). To compensate for this loss, it is found that the helium must be heated to 200°K before it leaves the bulkhead surface.

The temperature distribution of the helium along the bulkhead (see Fig. 2) was determined from the relationship

$$\frac{Q}{A} = w\, c_p\, \Delta t$$

where: Q = solar constant x bulkhead absortivity

= 420 x 0.9 = 378 BTU/hr-ft^2

A = Area between adjacent concentric circles 1 ft apart

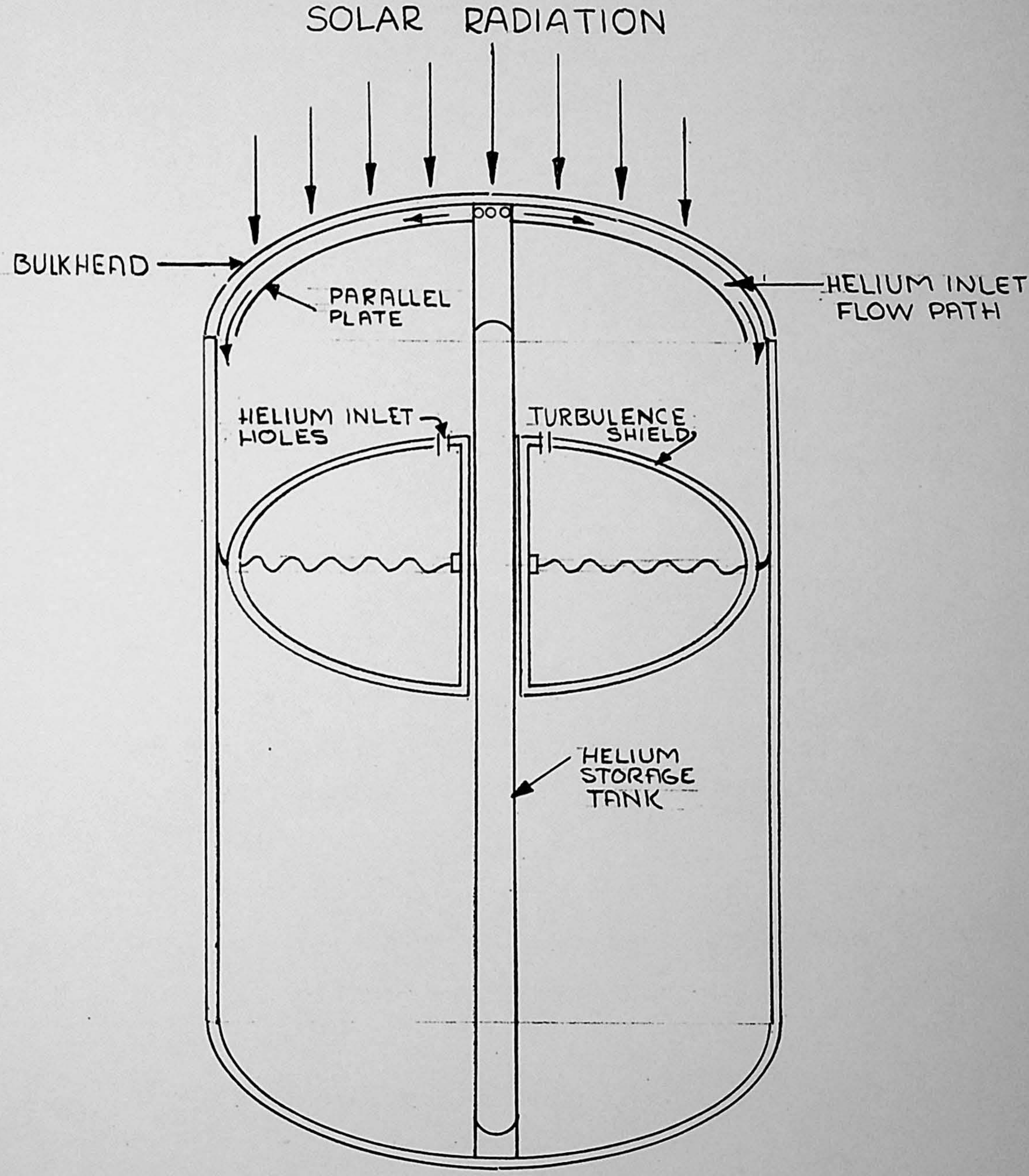
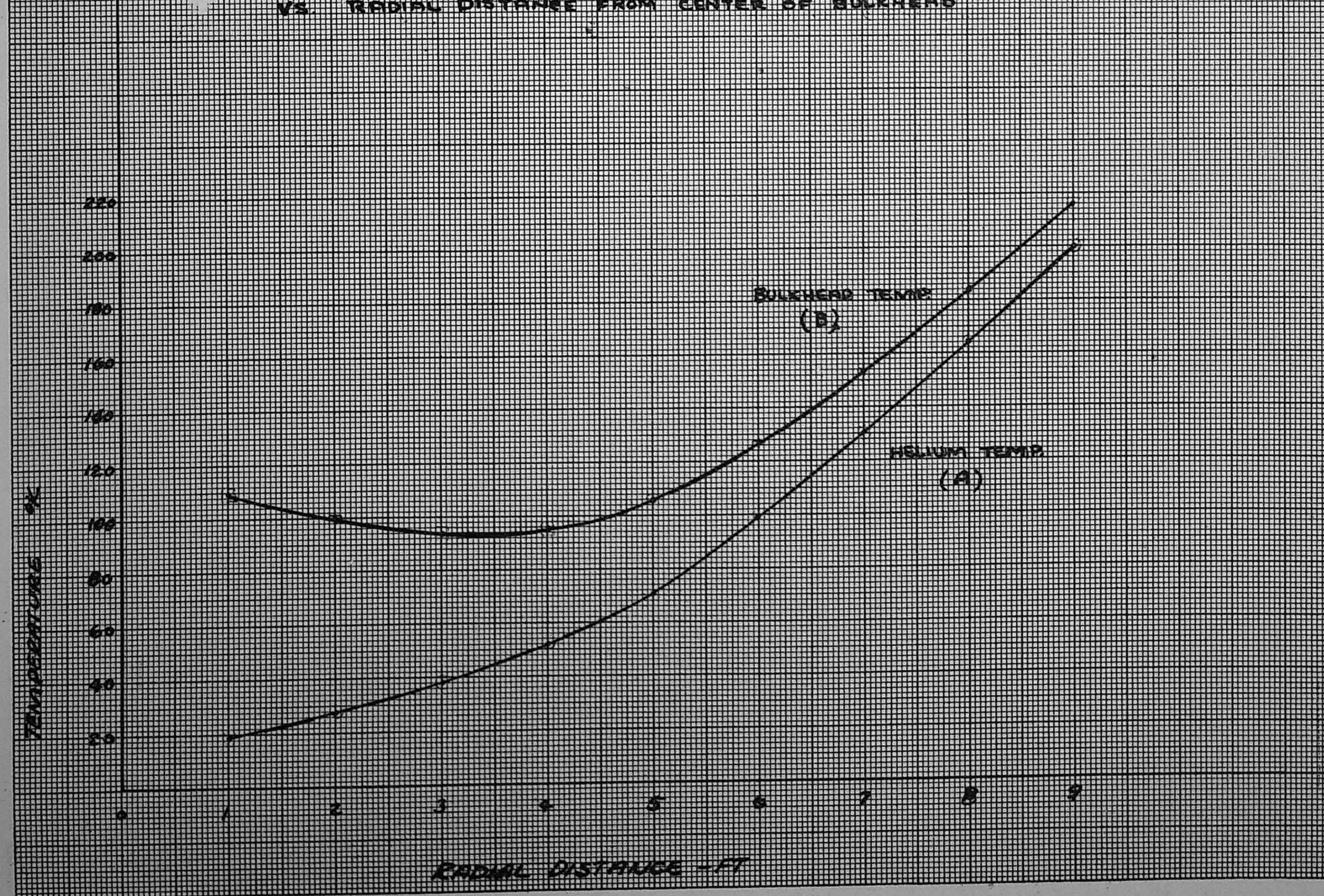


FIG 1 HELIUM EXPULSION SYSTEM

FIG. 2 HELIUM & BULKHEAD TEMPERATURE DISTRIBUTION
VS. RADIAL DISTANCE FROM CENTER OF BULKHEAD



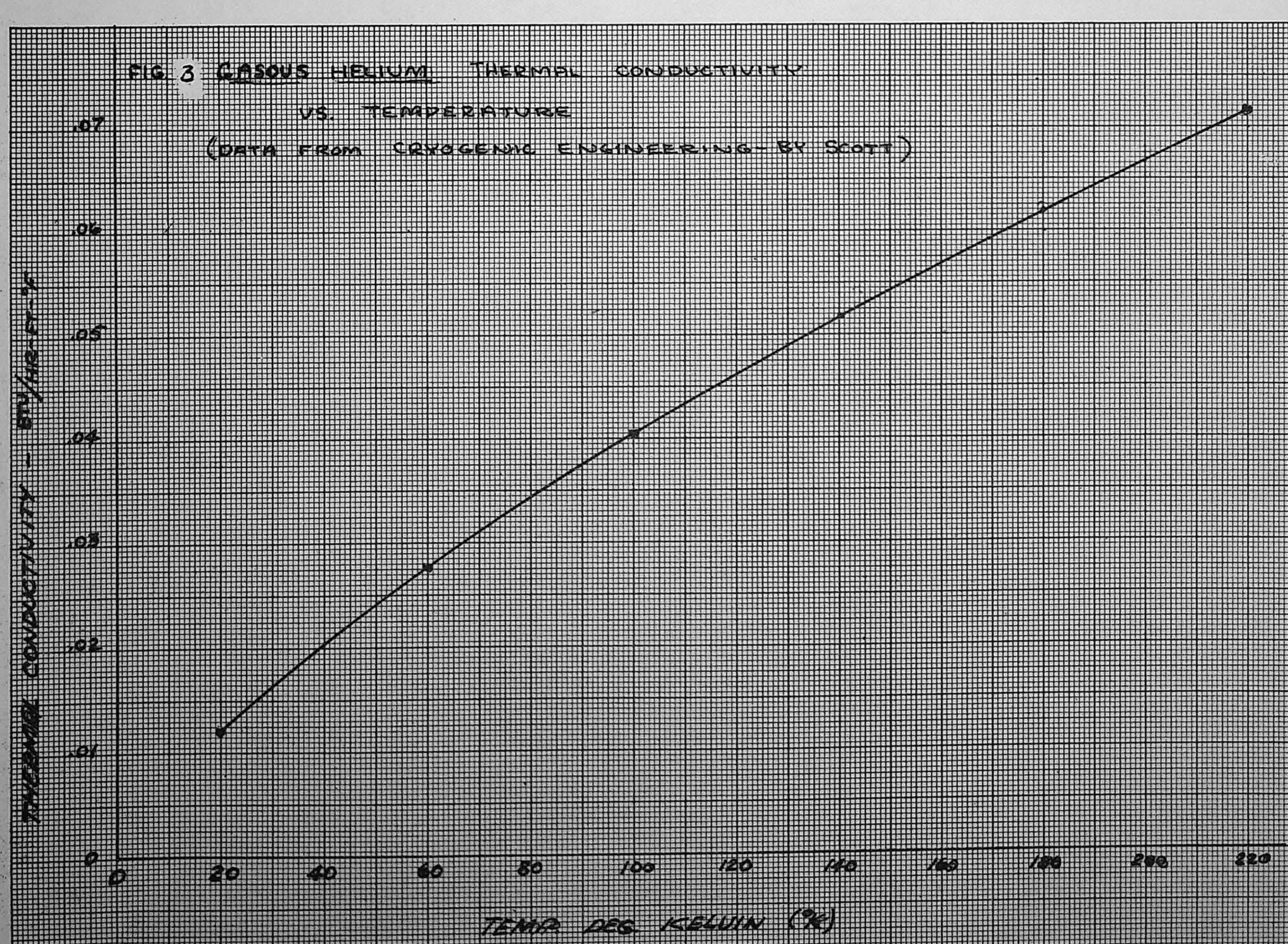


FIG. 3 GASOUS HELIUM THERMAL CONDUCTIVITY
VS. TEMPERATURE
(DATA FROM CRYOGENIC ENGINEERING - BY SCOTT)

w = weight flow rate of helium

= 246 lb/hr

c_p = specific heat of helium

= 1.242 BTU/lb-$^{\circ}$F

Δt = temperature increase for the area under consideration.

The bulkhead temperature distribution was determined by finding the temperature difference (Δt) between the bulkhead and the helium working fluid and adding this increase to the helium temperature. This temperature difference can be obtained from

$$\Delta t = \frac{Q}{hA}$$

Where: Q = heat input to bulkhead = 378 BTU/hr-ft^2

A = area between adjacent concentric circles 1 ft apart

h = convective heat transfer film coefficient

However, "h" must be determined first. For forced convection heat transfer to a fluid flowing between two parallel plates one of which is heated, Jakob, "Heat Transfer", Vol. I, (see Ref 2) gives the following empirical relation:

$$\frac{hs}{k} = 2$$

Where: s = distance between the parallel plates = 0.0104 ft

k = thermal conductivity of the helium.

Since "k" varies with temperature, a curve of values of k vs temperature was plotted (see Fig 3). Data was obtained from "Cryogenic Engineering" by Scott, (see Ref. 3). Hence, using the values of the helium temperature at various radial distances (as obtained from curve A, Fig. 2), the corresponding value of the conductivity can be obtained from Fig. 3 , and values for a curve of the film coefficient "h" vs radial distance were obtained and plotted (see Fig. 4). Using these values and the corresponding increments of area, values of the temperature difference between bulkhead and helium gas can be calculated. From these values curve B, Fig. 2 was obtained.

As can be seen from the curves of Fig. 2 the maximum helium temperature of 200°K occurs at a radius of nine feet and the maximum bulkhead temperature of 216°K also occurs at this point. The maximum local reradiation would therefore occur at the nine foot radius and can be calculated from:

$$Q_R = \sigma \mathcal{E} A T_B^4$$

Where: Q_R = reradiated energy, BTU/hr-ft^2

$\mathcal{E}$ = emissivity of bulkhead = 0.3

A = unit area = 1 ft^2

T_B = Bulkhead temperature = 216°K (389°R)

σ = Stefan-boltzman constant = 0.173×10^{-8}

$$Q_R = (0.3)\ (0.173 \times 10^{-8})\ (1)\ (389)^4$$

$$= 12 \text{ BTU/hr-ft}^2$$

or if expressed as a percentage of the input energy, the per cent reradiation equals 3.2 per cent; therefore, assumption number 2 above is valid.

To determine the total reradiation, the bulkhead was divided into annular areas one foot wide. The reradiation from each area was determined from its average temperature taken from the bulkhead temperature curve of Fig. 2 . The resulting reradiation was found to be

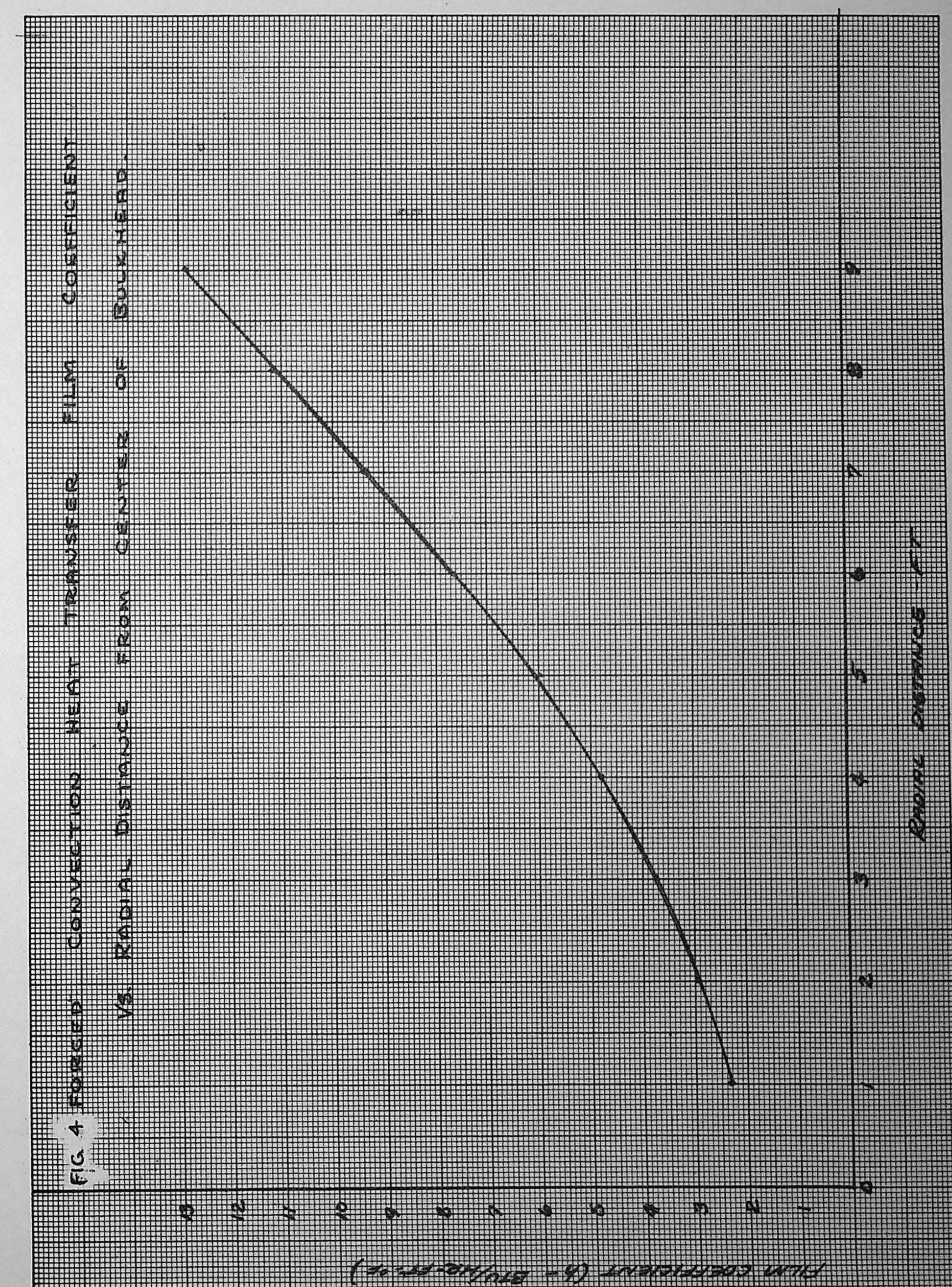


FIG. 4 FORCED CONVECTION HEAT TRANSFER FILM COEFFICIENT VS. RADIAL DISTANCE FROM CENTER OF BULKHEAD.

approximately 625 BTU/hr or about 0.63 per cent of the total energy input (100,000 BTU/hr).

C. Chemical Energy

Due to the large number of chemical energy sources available, a detailed investigation of the subject was not undertaken. An attempt has been made, however, to determine some of the basic advantages and disadvantages which might be offered by a chemical energy supply. Preliminary investigations were centered around the decomposition of hydrazine and hydrogen peroxide as monopropellants, burning hydrogen in oxygen and the use of solid propellants.

Monopropellants such as hydrazine (N_2H_4) and hydrogen peroxide (H_2O_2) would provide a relatively simple system but approximately 300 pounds of propellant would be required. This propellant weight is based on the thermodynamic properties of the decomposition products remaining at 300°F after passing thru a heat exchanger and giving up heat to the helium. The development of a heat exchanger to operate with the decomposition products at a temperature of approximately 1000-1200°F on one side and helium gas at a temperature of -420°F on the other would pose severe design problems. The possibility of mixing the decomposition products directly with the helium gas would theoretically reduce the weight of propellant required to less than half. However, a unique mixing system would have to be developed which would prevent hot gases (above -100°F) from coming in contact with the tank walls and diaphragm. If thorough mixing was not achieved, an increased heat transfer to the liquid hydrogen being transferred would occur. Consequently, a greater vaporization loss and additional transfer problems due to mixed gas and liquid would result.

A bi-propellant system such as hydrogen plus oxygen would require approximately 35 lbs of propellant. A total system weight of 100 lbs, including hardware, appears reasonable if the propellants were stored cryogenically. Cryogenic storage of the liquid oxygen in or attached to the LH_2 storage tank may present more problems than the LH_2 storage. It may be kept in one of the LO_2 storage tanks until it is needed, but this approach might merely substitute handling problems for storage problems. In addition, the same disadvantage of a heat exchanger or controlled mixing as in the monopropellant system is still present.

A solid propellant system would be a simple and reliable source of chemical energy. This system, however, is very inflexible in that it offers little on-off control of the propellant transfer process.

In summary, therefore, the chemical energy sources investigated appear to possess the following basic characteristics.

1. Advantages.

a. Their operation is not interrupted by the earth's shadow or any other characteristic of the particular orbit selected for transfer.

b. The energy source is independant of storage tank orientation.

c. Energy may be supplied at any desired rate resulting in a completely adequate transfer rate potential.

2. Disadvantages.

a. Some degree of additional complexity is added over that of the solar radiation energy source, with a resultant inherent decrease in reliability.

b. Additional weight is added due to:

(1) The propellant itself plus storage tank and insulation.

(2) Combustion or decomposition chamber.

(3) Heat exchanger or controlled mixing chamber.

D. Proposed Energy System

In light of the above investigative studies, it is herewith proposed that the solar energy system be employed to supplement the helium working fluid. This system is considered to have more merit due to the following:

1. Reliability and design simplicity
2. Weight
3. Acceptable propellant transfer rate and control
4. Minimum work maneuvers.

A more detailed investigation into the various chemical energy sources was beyond the scope of this report due to submittal date limitations. Further studies into chemical systems could conceively result in the development of a compact energy source adaptable to the orbital expulsion system.

III SPECIFIC HARDWARE REQUIREMENTS

The following hardware must be incorporated in the propellant tank design. These items are presented to indicate the requirement and illustrate a workable solution and are not necessarily suggested as final design hardware, Ref. Fig. 5.

A. Propellant Valves and Couplings

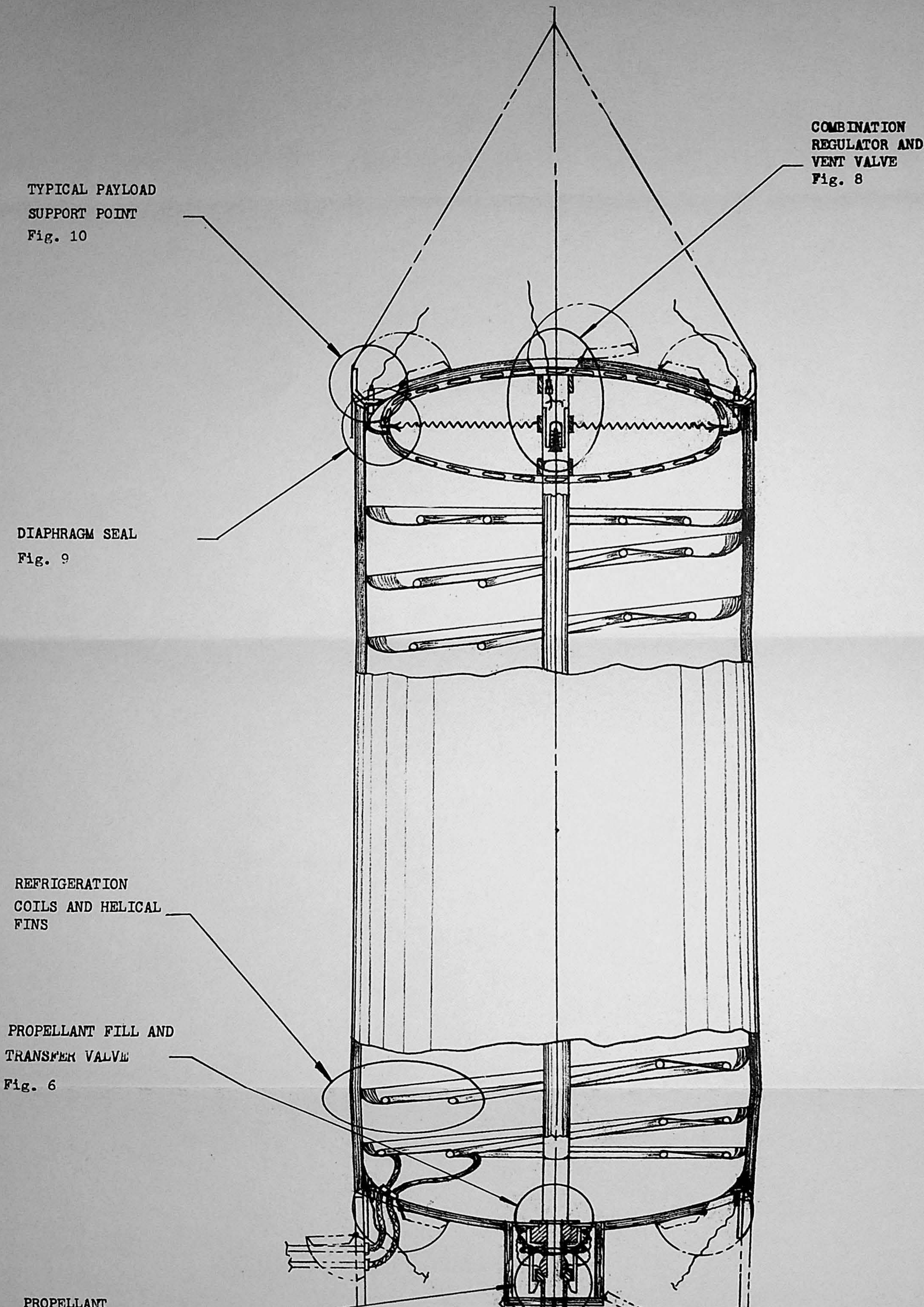
1. The propellant valve illustrated in Fig. 6 is employed for ground filling and propellant transfer in orbit. This valve was designed specifically for the four series connected LOX tanks. Since propellant transfer is from the outermost tank to the space vehicle, the flow through the central transfer tube must be unrestricted.

2. The tank-to-tank or tank-to-vehicle coupling shown in Fig. 7 was designed to accomodate ease of connecting and disconnecting the tanks. Ample angular misalignment can be accomodated.

B. Vent Valve

The propellant tanks will not be structurally strong enough to withstand a negative internal pressure. The low vapor pressure of the LH_2 payload must be supplemented to maintain a total internal tank pressure equal to or above atmospheric pressure. For purposes of discussion it will be assumed that an internal pressure of 5 psig must be maintained during prelaunch and during the major portion of powered flight. It is proposed that this be accomplished by utilizing the combination regulator and vent valve shown in Fig. 8 During prelaunch, valve A is closed, valve B is open and valve C is closed.

At a predetermined time during powered flight, the combination regulator and vent valve will be fully opened by opening valve A and



PROPELLANT
TANK HARDWARE
Fig. 5

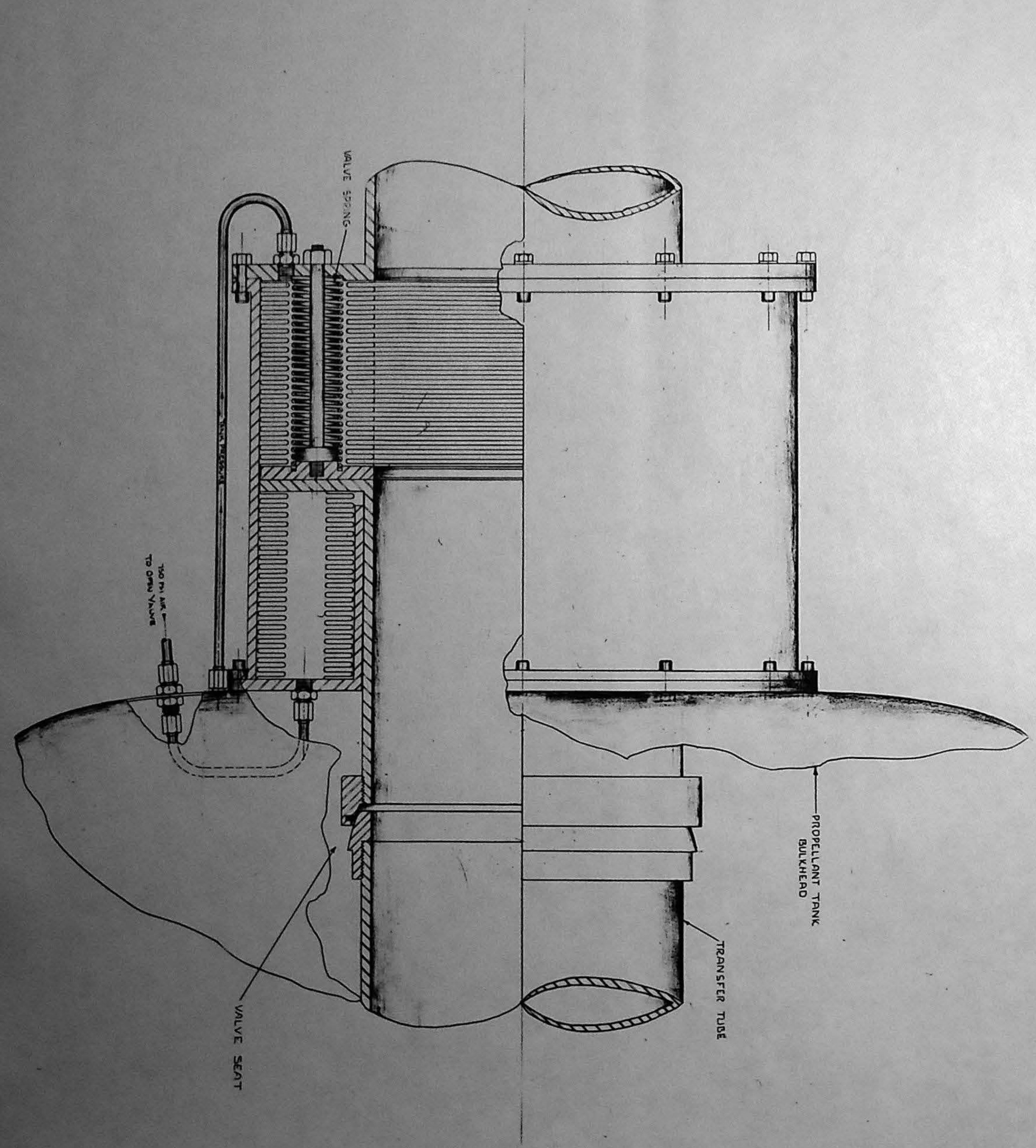


FIG. 6 FILL & TRANSFER VALVE ORBITAL STORAGE PROPELLANT TANKS

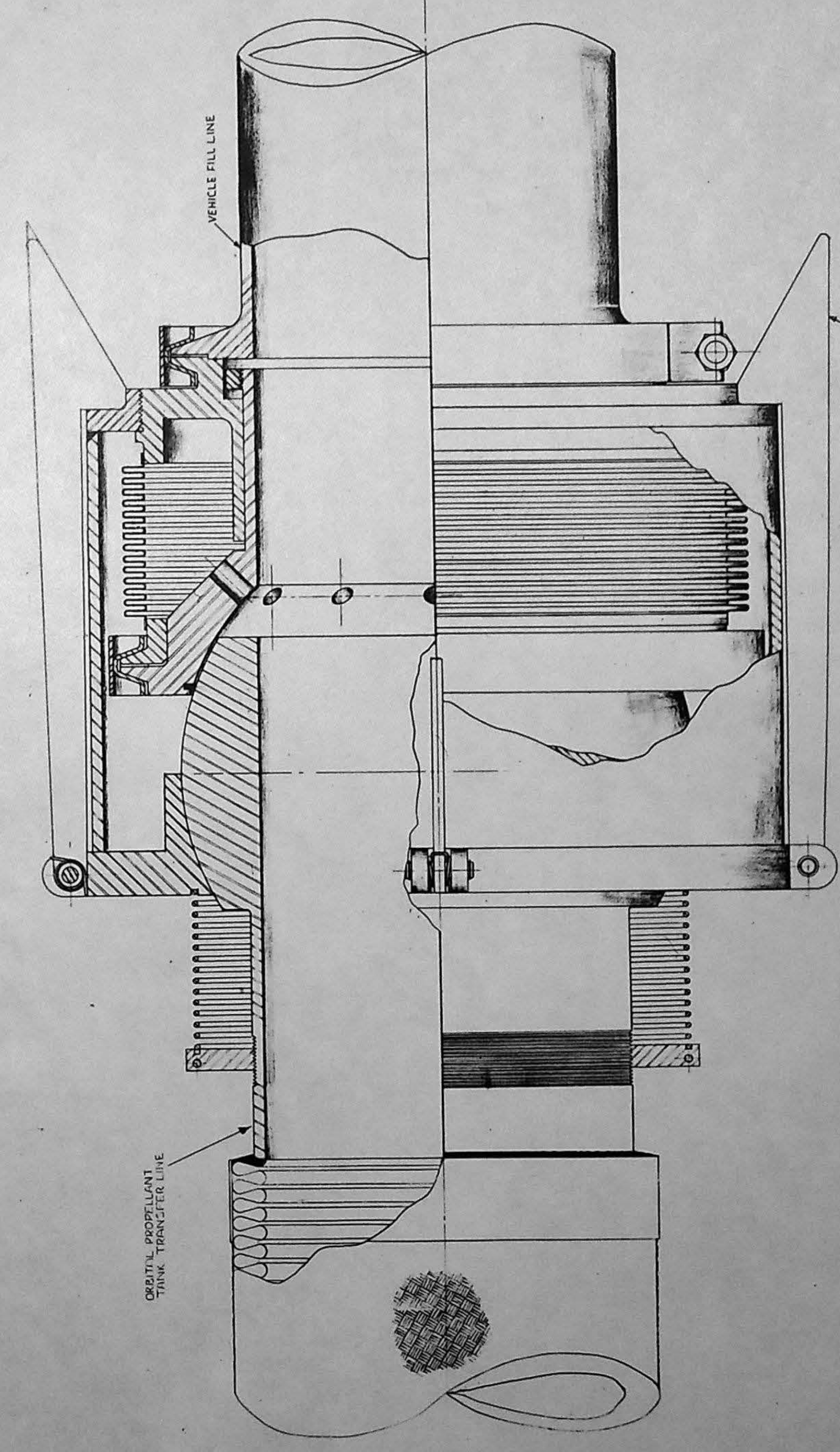
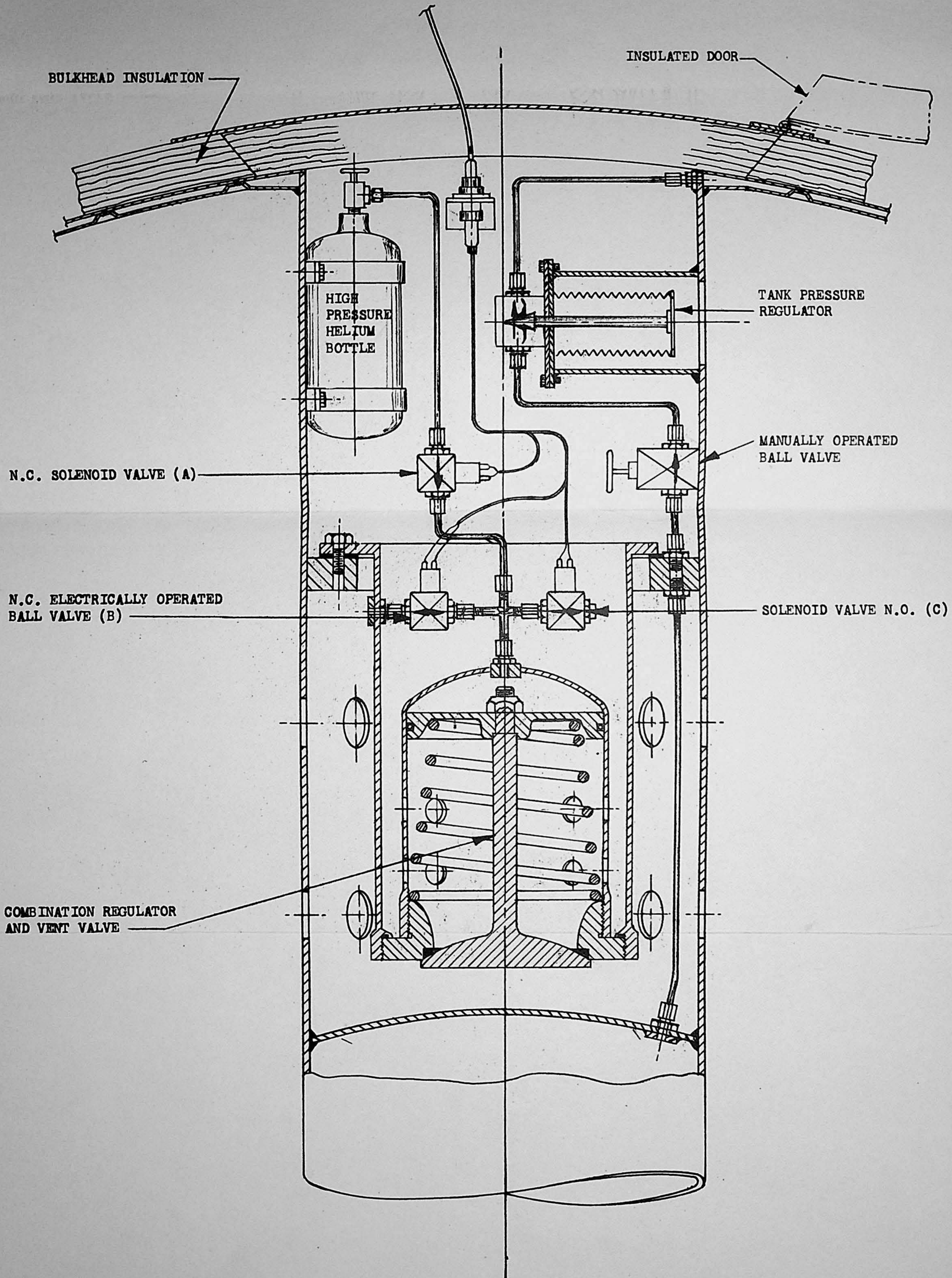


FIG. 7 PROPELLANT TRANSFER COUPLING



COMBINATION REGULATOR AND VENT VALVE

Fig. 8

closing valve B. This operation is necessary to flush the remaining helium gas which was used to supplement the LH_2 vapor pressure (1.13 psia). This venting process must take place under powered flight conditions, i.e., while a definitely located liquid level is maintained. However, since the tank design may depend upon internal pressure for strength, the venting process is delayed until aerodynamic loads are minimized.

The combination regulator and vent valve will be permanently closed by opening valve C after the internal tank pressure has reached the LH_2 payload vapor pressure (1.13 psia). Valve C is a normally open valve selected to prevent pressure build-up on the combination regulator and vent valve piston during orbital storage. The closing operation might be triggered by a pressure sensing switch or a time delay. If the operation was based on elapsed time, a possible basis for operation would be third stage cut-off.

C. Flexible Expulsion Diaphragm Seal

One important design problem connected with the use of the flexible expulsion diaphragm, as mentioned in the Phase I Report, is the design of a satisfactory seal between the diaphragm and tank wall.

The helium working fluid will be at a slightly greater pressure than the LH_2 due to sliding friction between the diaphragm and tank wall. Since the cross sectional area of the diaphragm is so large (38,100 sq in.) the excess helium pressure required should be extremely small. Any leakage permitted by the diaphragm seal would thus be helium gas moving to the LH_2 side of the diaphragm.

In dealing with a diaphragm as large as 18 feet in diameter, it is felt that a minimum radius clearance of one inch between the diaphragm and tank wall is in order. The decrease in tank radius due to thermal contraction as the tank is cooled from 80^{o}F to -420^{o}F is .337 inch and a reasonable fabrication tolerance for the tank and diaphragm based on the radius, is estimated as one-quarter inch.

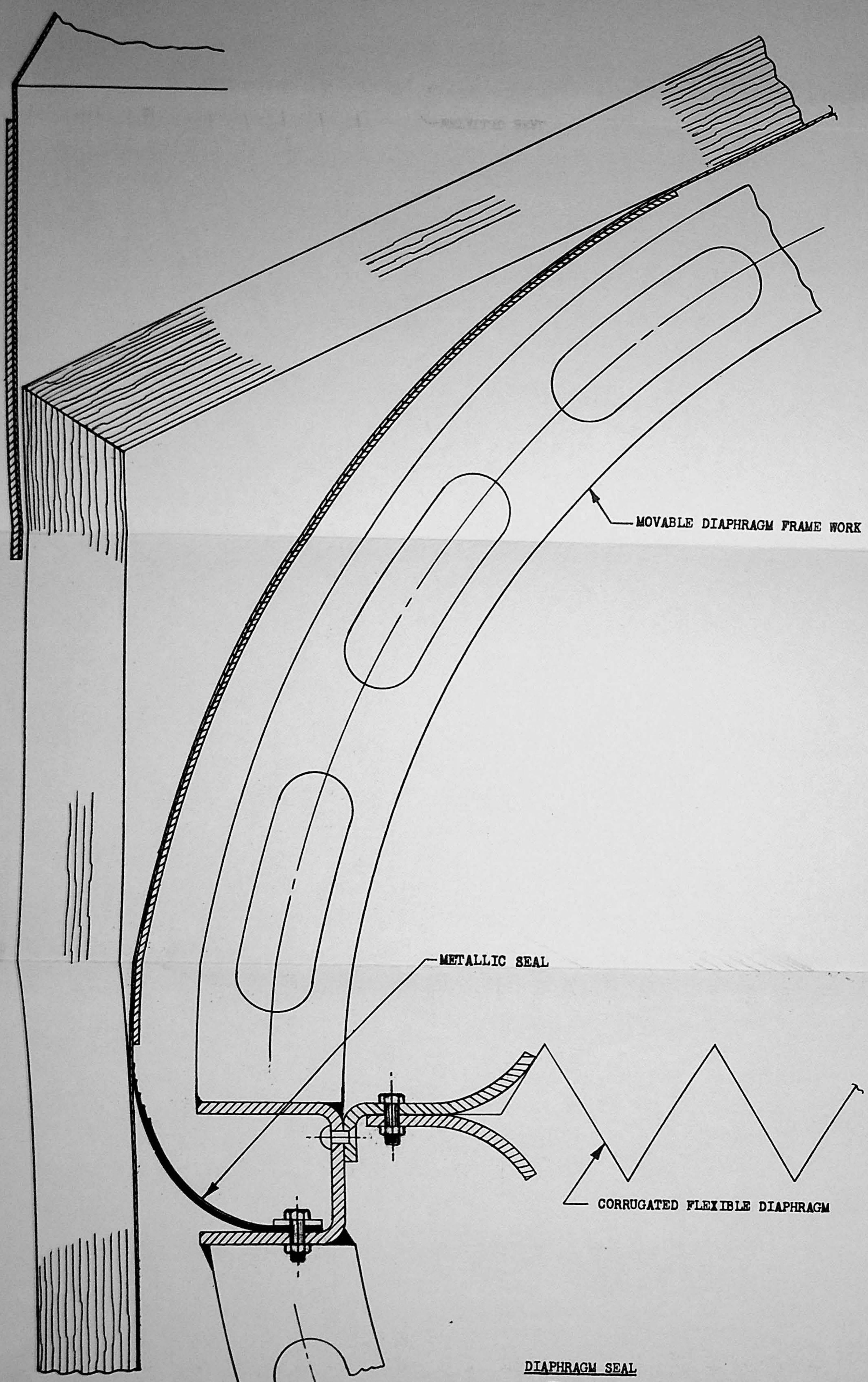
A thin metal lip seal is proposed for use between the diaphragm and tank wall, Ref. Fig. 9. Plastic materials appear inferior due to their high thermal expansion characteristic and poor low temperature properties. A metal with good "spring" characteristics should be selected so that the seal will follow any change in tank wall to diaphragm clearance.

To prevent the lip seal from buckling under compression, radial "V" shaped slots should be cut in the seal at intervals so that the remaining circumference equals the minimum circumference of the tank. To prevent leakage from these slots, a second lip seal can be mated with the first but with its slots off set. To facilitate fabrication these seals can be made up in short circular segments which will form the complete seal when bolted side by side around the diaphragm.

It is realized that a seal of this type would permit some leakage. However, the small pressure differential across the diaphragm will allow leakage to be kept within acceptable limits.

D. Support Points

As specified in the Phase I Report, the tank shall support the entire payload and be attached to the third stage at approximately six



DIAPHRAGM SEAL

Fig. 9

points. Access holes in the permanent insulation will be provided at the support openings to reduce the heat transfer. A typical support point is illustrated in Fig. 10.

E. Installation of SI-4 Insulation

A super-insulation developed by the Linde Company known as "Linde SI-4", has been proposed as the insulating media for the storage tanks while in orbit. This insulation, consisting of 40-80 layers per inch of sub-micron glass-fiber paper and aluminum foil, has been in service use on thousands of cryogenic vessels over the past 4 years, (see Ref. 4).

Their service record indicates that the problems associated with its use have been resolved. These problems include applying the insulation to geometrically irregular contours, accommodating structural supports and piping through the insulation and optimizing insulation density consistent with mechanical stability under the condition of shock and vibration encountered in the handling and transporting of the vessel.

Although the vibration and acceleration forces acting on the insulation attached to an orbital storage tank are different than those acting on a transportation vessel, the background and experience gained in solving the latter problems should at least indicate the proper direction for a successful solution to the former.

An additional complication exists in the case of the orbital storage tanks in that an outer vacuum jacket is not used. However, a thin covering is proposed to take care of aerodynamic heating and aerodynamic forces during powered flight and conceivably this could be used to help maintain the insulation in place. Other problems that will have to be investigated concern the recovery of the insulation to its original shape after compaction due to acceleration forces and possibly to the method of holding the insulation to the tank and means for bleeding off the helium gas used as a purge during ground launch preparations.

Since little is known, at the present time, about the mechanical properties of SI-4 the following suggested solution is merely an indication of one possible method and is subject to change as more detailed information becomes available.*

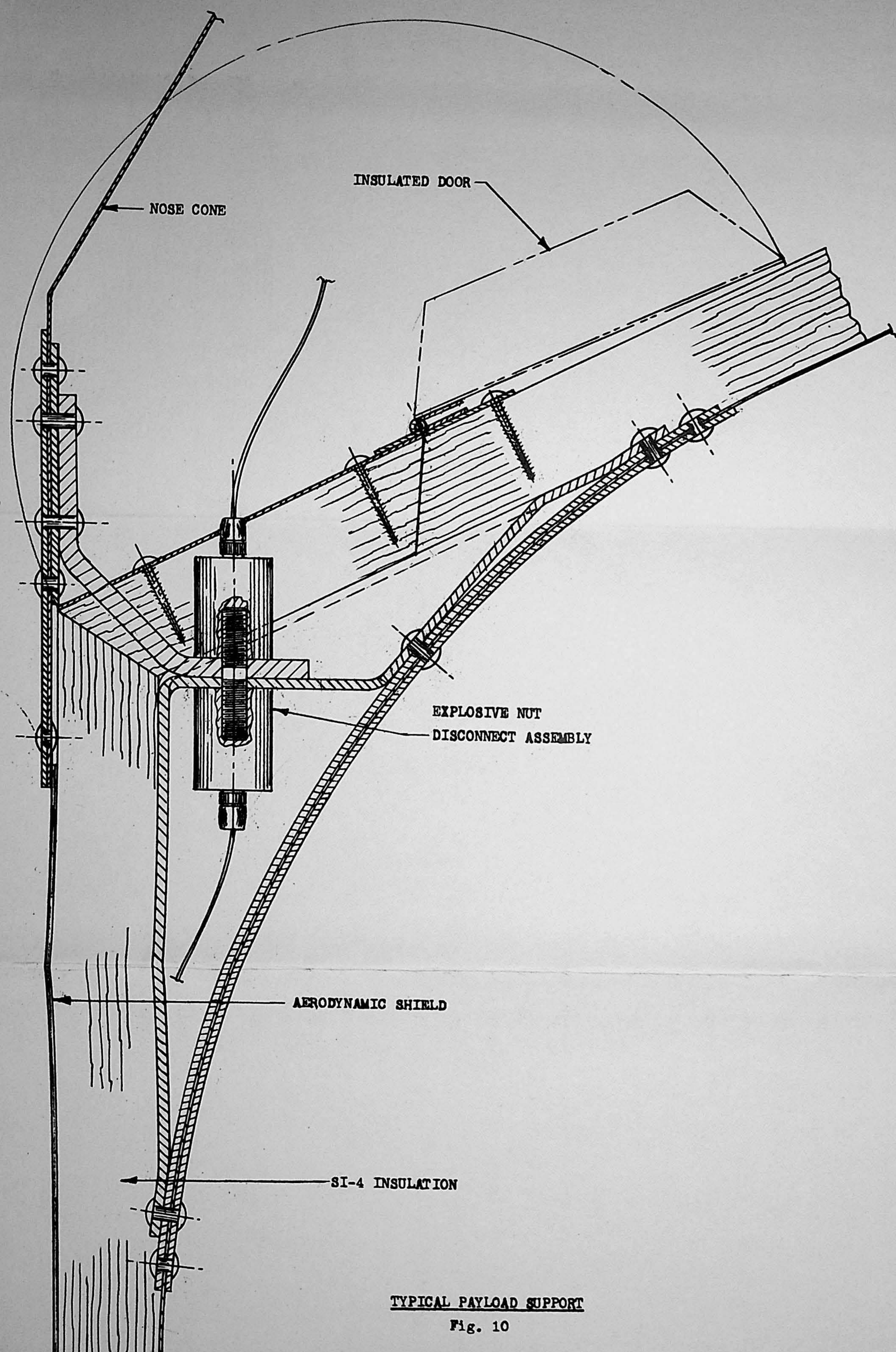
The thin aluminum covering used to protect the insulation from aerodynamic forces and heating could be made in sections and held tightly against the insulation with several clamping bands thus compressing the insulation against the liquid tank. When the band connections are severed the sections will peel off and allow the insulation to expand to its original size. To aid in this expansion and also to bleed off the helium gas used as a purge, a series of randomly spaced minute holes would be made in each layer of aluminum foil as the insulation is fabricated.

* The following information was obtained in a meeting on 20 April 60, with Messrs L. L. Kelley and M. Holloway, Jr. of The Linde Company:

a. A feasible method of supporting SI-4 insulation on a cylindrical tank wall against a 7-g acceleration force has been developed. Only a small increase in thermal conductivity was noted.

b. Recovery tests of SI-4 insulatinn after compaction have indicated very good recovery characteristics.

c. Aluminum at cryogenic temperatures does not show any effects



TYPICAL PAYLOAD SUPPORT

Fig. 10

from radiation.

This information plus details will be available in report form during the latter part of May 1960.

IV ORBITAL OPERATING PROCEDURE

A. Background

It was established by the Phase I Report that the propellant tanks could be stored in a low geocentric orbit for the required time.

Orbital storage of liquid hydrogen in a no loss condition presented the most critical problems. For this payload it was found that after two months storage (1440 hours) the temperature near the tank wall was approaching $20.4^{\circ}K$ (maximum allowable is $22^{\circ}K$) although the mean bulk temperature was only $17.3^{\circ}K$.

With the required payloads clustered in orbit and properly rendezvoused with the space vehicle, it becomes apparent that an operating procedure must be established which will most expeditiously accomplish the refueling task.

B. Refueling Schedule

The process of connecting and disconnecting the refueling tanks to and from the space vehicle, as well as starting and stopping propellant transfer, may be accomplished by: (1) direct action of refueling crew members in a suitable protective environment, (2) manipulators remotely controlled by the refueling crew while in orbit with the tanks and space vehicle, or (3) manipulators remotely controlled from ground stations. Each of these methods has merit and the final choice will depend on the outcome of more detailed studies and investigation.

The following assumptions, subject to modification as more detailed concepts and designs appear, are utilized to establish the time schedule illustrated in Fig. 11.

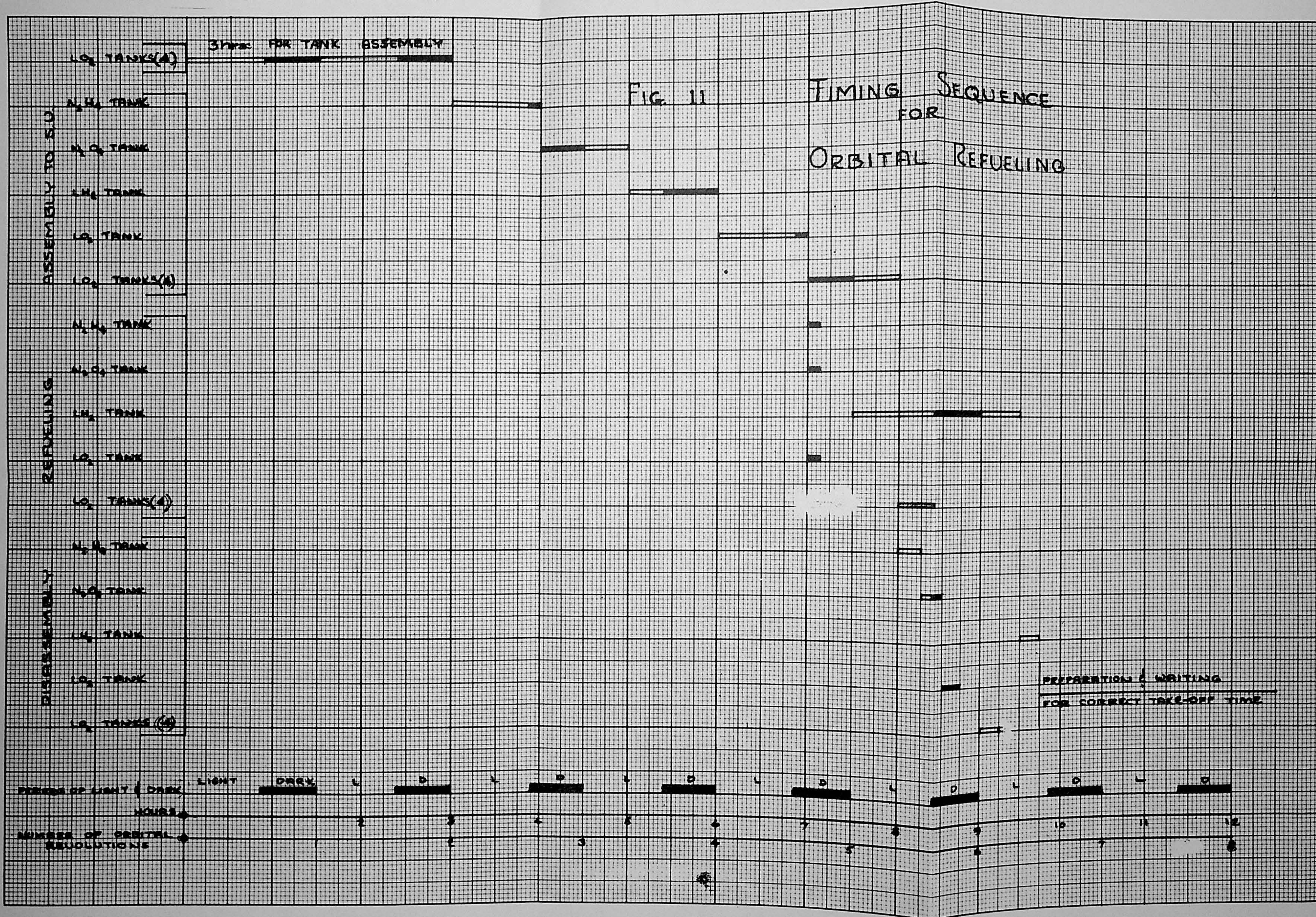


FIG. 11 TIMING SEQUENCE FOR ORBITAL REFUELING

1. Assembly of each tank to the space vehicle requires one hour. This also applies to assembly of one LO_2 tank to another in the case of the four LO_2 tanks for refueling the main oxidizer stage.

2. Propellant transfer from the LH_2 tank requires 84 minutes of exposure to solar radiation.

3. Maximum propellant transfer time for all other tanks is six (6) minutes.

4. Disassembly time per tank is fifteen (15) minutes.

5. Work tasks can be performed in the shadow of the earth.

To establish periods of light and darkness, it was logically assumed that during 60 per cent of a 90 minute orbital transit time the tanks would be in the sunlight. As can be seen from Figure 11 the minimum time required for assembly, refueling, and disassembly is 9 3/4 hours. Allowing an additional 5 hours orbital parking time, the overall storage time for LH_2 is only increased one per cent and the additional temperature rise is negligible. Further considerations of heat transferred to the liquid hydrogen can now be made.

C. Heat Transfer Study

The first consideration will occur in the expulsion process which was described in detail previously in this report. During the expulsion, helium gas at $150^{\circ}K$ is in contact with the diaphragm on one side while LH_2 at $20.4^{\circ}K$ is on the other side. Since the thin stainless steel diaphragm has a much greater conductivity than the liquid hydrogen or the gaseous helium, it will offer negligible resistance to the transfer of heat and hence can be neglected. Due method of admitting the helium gas (see Fig. 1) a thick t layer of helium will be formed next to the diaphragm. If assumed that the total temperature difference extists in this , the heat transferred to the LH_2 can be calculated. In Jakob, Transfer, Vol. I, the following formula has been derived for total heat energy entering a flat plate if the surface temperature suddenly changed and remains constant.

$$Q = 2k \quad A\theta_i \sqrt{\frac{\tau}{\pi\alpha}}$$

Where: k = thermal conductivity of helium at 25 psia and average temperature of $85^{o}K$ = 0.0666 BTU/hr-ft-^{o}K

A = Surface area = 264 ft^2

θ_i = temperature difference = $130^{o}K$

τ = 2.0 hours

α = thermal diffusivity = 1.107 ft^2/hr

Hence: $Q = 2 \times 0.0666 \times 264 \times 130 \sqrt{\frac{2.00}{\pi 1.107}}$

$= 4,650 \sqrt{.575}$

$= 3,452$ BTU

The total time τ = 2 hours is obtained from Fig. 11 and represents the total refueling time including the period when the tank is in darkness and no refueling transfer is taking place.

The heat of vaporization of LH_2 is 193 BTU/lb. Hence th heat absorbed will vaporize $\frac{3,452}{193}$ = 17.89 lb of LH_2.

Since considerable excess LH_2 has been provided for lo this amount is negligible.

D. Space Vehicle LH_2 Tank Consideration

The next logical consideration concerns the hydrogen tanks of the space vehicle, i.e., the main stage hydrogen tank and the lunar braking stage hydrogen tank. It is not within the scope of this report to specify the kind and quantity of insulation to be used on these tanks, but to indicate what the requirements are and suggest a possible type of insulation.

1. Main Stage. The main stage LH_2 tank will still contain approximately 8600 lbs of LH_2, when it arrives in orbit, If a method is used similar to that proposed for "Centaur", wherein the tank is insulated at ground lift-off and then after absorbing the aerodynamic heat energy the insulation is jettisoned, the tank will contain LH_2 at a temperature corresponding to its vapor pressure. These condit might very possible be $T = 21^{o}K$ and $p = 18$ psia, and must be maint with very little change in temperature and pressure during the ne 7 1/2 hours. This time represents orbital parking time before re fueling can begin (see Fig. 11). Since refueling expulsion pres is only 25 psia, careful consideration must be given to a perma type insulation on this tank to prevent the pressure from incre to 25 psia.

2. Lunar Braking Stage. In considering the LH_2 ta of the lunar braking stage, different conditions apply. Eno LH_2 is contained in the refueling tank to completely effect with the required 11,900 lbs of LH_2. Before it is used, ho it must be stored for a few hours in the parking orbit plu required to reach the moon. This time interval represent

approximately 60 hours. During this time the tank should be insulated to prevent vaporization of the LH_2 and consequently the excessive increase in pressure. To obtain an idea of what the initial conditions of the LH_2 should be so that the final pressure does not exceed 25 psia (liquid temperature approximately $22^{\circ}K$) the following assumptions were made:

a. Cylindrical tank surface (r = 9') exposed to sun radiation for 60 hours.

b. One inch of Linde SI-4 insulation - effective heat transfer coefficient $h = 60 \times 10^{-5}$ BTU/hr-ft^2-$^{\circ}$F, and

c. An equilibrium outer skin temperature of $100^{\circ}F$.

Using the formula derived in Jakob, Heat Transfer, Volume I for a sudden temperature change of the environment of a cylinder, the temperature rise and hence initial temperature can be calculated.

$$\frac{\theta}{\theta_i} = \sum_{\nu=1}^{\nu=\infty} \frac{2 J_1(x_\nu)}{x_\nu\left[J_0^2(x_\nu) + J_1^2(x_\nu)\right]} e^{-x_\nu^2 \frac{\alpha \tau}{s^2}} J_0\left(x_\nu \frac{r}{s}\right)$$

Where: θ = temperature excess after 60 hours = $t - t_o$

θ_i = temperature excess at start = $t_i - t_o$

α = thermal diffusivity of LH_2 = 0.00721 ft^2/hr

r = s = 9 ft

τ = elapsed time = 60 hours

k = conductivity of LH_2 = 655×10^{-4} BTU-ft/hr-ft^2-$^{\circ}$F

x_v = (ns) = roots obtained from $\frac{ns}{bs} = \frac{J_o\,(ns)}{J_1\,(ns)}$

$$bs = \frac{h\,r}{k} = \frac{60 \times 10^{-5} \times 9}{655 \times 10^{-4}} = 0.08244$$

J_o = Bessel function of first kind and zero order, see Ref. 5.

J_1 = Bessel function of first kind and first order.

By trial and error the first 2 roots are found to be $X_1 = (ns)_1 = 0.402$ and $X_2 = (ns)_2 = 3.853$. Substituting the proper values in the above equation gives $\frac{\theta}{\theta_i} = 0.9891$ or $\frac{t - t_o}{t_i - t_o} = \frac{-420 - 100}{t_i - 100} = .9891$ and $t_i = -425.7^{\circ}F$ (or $19.0^{\circ}K$). Therefore, under the assumed conditions, the LH_2 temperature in the space vehicle tank should not exceed $19.0^{\circ}K$ after refueling. Since the bulk temperature of the 11,900 lbs of LH_2 transferred is only $17.3^{\circ}K$, it is capable of absorbing considerable heat. The turbulence created by the velocity of the entering fluid will probably be sufficient to assure a uniform bulk temperature as the liquid cools the tank wall. The amount of heat which would be absorbed by 11,900 lb of LH_2 if its temperature is changed from $17.3^{\circ}K$ to $19^{\circ}K$ would be:

$$Q = w\,c_p\,\Delta t = 11{,}900 \times 2.0 \times 3.06 = 72{,}700 \text{ BTU}$$

If the lunar braking stage LH_2 tank is precooled on the launch pad. A small amount of LH_2 should be left in the vehicle tank in order to maintain precooling during powered flight and parking time. Just prior to fill this residual may be at a vapor pressure too high to accomplish propellant transfer. This condition may be alleviated by venting this tank residual to vacuum. The refrigeration effect created by this venting process should reduce the tank temperature to a satisfactory level.

V. FUTURE DESIGN CONSIDERATIONS

The considerations given to the no-loss storage time of the liquid hydrogen propellant tank throughout the orbital refueling investigations have been based on 1440 hours.

If future concepts require longer periods of storage, preliminary investigations to date indicate that this could be achieved by:

a. Increasing the thickness of the SI-4 superinsulation.

b. Internal mechanism for agitating the liquid hydrogen in the propellant tank to approach uniform bulk temperature throughout storage.

c. Externally rotate the propellant tank about one of the principal axes to create an articifical force field and induce free convection of the fluid.

d. Spin the tank about its longitudinal axis to create an artificial force field establishing a definite liquid level so that venting could be employed.

e. Provide helical fins as shown in Fig. 5. These fins will rapidly transfer heat from the warm outer fluid layers to the cold inner core.

f. In addition to the fins illustrated in Fig. 5 the tank could be spun about its longitudinal axis. This combination would provide the best conditions for obtaining an approximately uniform bulk temperature.

Item (e) is proposed since it can be effected with a minimum weight increase and does not require energy for implementation.

The above is submitted to stress the fact that orbital storage of liquid hydrogen is feasible for periods in excess of the 1440 hours presented throughout these studies.

VI CONCLUSIONS

The feasibility of orbital transfer of the stored propellants has been shown. The hydrazine, nitrogen tetroxide, and liquid oxygen can be expelled in approximately six (6) minutes per tank. The liquid hydrogen requires a greater period of time to successfully effect transfer and must receive a definite amount of thermal energy from an external source to supplement the helium working fluid.

It is herein proposed that the above mentioned thermal energy be obtained from solar radiation impingement on the tank end bulkhead. Solar radiation and chemical sources were investigated with the formed appearing more advantageous due to design simplicity, reliability, overall system weight and orbital handling problems.

Specific hardware requirements covered by this report indicate readily acceptable solutions are available for the described problem areas.

An orbital operating procedure has been established which is felt to be entirely feasible. The assembly, transfer and disassembly bar graph presented herein is considered adequate and lends itself to modification without adversely effecting storage and transfer of the LH_2.

Sufficient data has been established which determines that orbital storage of liquid hydrogen is feasible for periods in excess of the 1440 hours presented throughout these studies.

VII REFERENCES

1. Orbital Refueling, A Feasibility Study and Design Concept, Report No. DLM-TN-21-6C, 15 March 1960.

2. Heat Transfer, by Jakob, Vol. I, John Wiley and Sons,Inc., 1956.

3. Cryogenic Engineering, Scott, D. van Nostrand Co. Inc., 1959.

4. Characteristics and Application of Some Super Insulations, Riede and Wang, The Linde Company, 1959.

5. Bessel Functions, Cambi, Dover Publications, Inc., 1948.

6. Other general references not cited in text of this report:

   Elements of Heat Transfer and Insulation, Jakob and Hawkins, John Wiley and Sons, Inc., 1954.

   Heat and Mass Transfer, 2nd Edition, Eckert and Drake, McGraw-Hill 1959.

   Higher Mathematics for Engineers and Physicists, I.S & E.S. Sokolnikoff, McGraw-Hill Book Company, 1941.

   A Study of the Hazards in Storage and Handling of Liquid Hydrogen, by Cassutt, Maddocks, Sawyer of Arthur D. Little, Inc.

   Handling of Liquefied Gases and Technical Data, The Linde Co.

   Linde Cryogenic Literature, The Linde Company

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