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ORBITAL REFUELING

A FEASIBILITY STUDY AND DESIGN CONCEPT

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ARMY BALLISTIC MISSILE AGENCY

REDSTONE ARSENAL, ALABAMA

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ORBITAL PROPELLANT TRANSFER

FRONTISPIECE

FOREWORD

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

This presentation is based on preliminary design studies and is respectfully submitted as one workable solution to the varied problems inherently encountered in such an endeavor.

This proposal is predicated on the Saturn C-2 vehicle configuration with a maximum payload of 50,000 lbs. This proposal is preliminary in scope and is not intended as final design criteria.

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I INTRODUCTION

This report is Phase I of a study to determine the practicality of orbital storage and subsequent refueling of cryogenic propellants such as Liquid Hydrogen and Liquid Oxygen.

One method of accomplishing manned space missions is to use a number of Saturn vehicles, as presently envisioned, to put refueling tanks and manned space vehicles into a low geocentric orbit and then transfer the propellants from the refueling tanks to the space vehicle. Since a number of firings will have to be made to place the entire payload into orbit it becomes necessary to maintain orbital storage of the propellants for several months.

The objective of this report is to determine the feasibility of orbital storage and refueling in general and propose a workable concept.

The Phase II supplement to this report will incorporate a more detailed investigation into the propellant transfer scheme proper, certain specific hardware requirements for prelaunch, in-flight and orbital storage, and an investigational outline of the operating procedure in orbit.

II CONCLUSIONS

The feasibility of storing liquid propellants in an earth orbit, for a period of time sufficient to permit use of a number of Saturn vehicles to place the entire payload in orbit, has been shown.

Liquid hydrogen can be stored in a no-loss condition in an earth orbit for at least two months. The maximum expected temperature under the assumed conditions will be only 20.4°K and the corresponding vapor pressure is only 15.3 psia.

Liquid oxygen can be stored in a no-loss condition in an earth orbit for considerably longer than the required nine months. After 10,000 hours the maximum temperature (under the assumed conditions) is only 81.3°K and the corresponding vapor pressure only 5.37 psia.

Storage of hydrazine (N_2H_2) and nitrogen tetroxide (N_2O_4) can be achieved for an indefinite period of time by proper selection of tank material and surface finish.

III SYMBOLS

A = Area (ft^2)

C = Specific heat ($^{BTU}/lb^{o}F$)

D = Diameter (ft)

h = Convective film coefficient ($^{BTU}/ft^2$-^oF-hr)

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

$J_1(ns)$ = Bessel Function of first kind, first order

k = Thermal conductivity ($^{BTU}/hr$-ft-^oF)

L = Length (ft)

M = Mass (lb)

Q = Heat ($^{BTU}/hr$)

r = Distance from center at which temperature is to be determined. (ft)

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

T (t) = Temperature ^oF

U = Overall heat transfer coefficient $^{BTU}/hr$-ft^2-^oF

V = Volume (ft^3)

ΔX = Thickness (inches)

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

$\propto$ = Thermal diffusivity (ft^2/hr)

θ = Temperature excess after time τ

τ = Elapsed time (hrs)

θ_i = Temperature excess at $\tau = 0$

x = Distance from median plane

Φ_v = (ns) = roots obtained from $\frac{ns}{bs}$ = cot (ns)

Ψ_v = (ns) = roots obtained from bs = 1 - cot (ns)

IV PRELIMINARY PAYLOAD STUDIES

The following investigation on the feasibility of orbital refueling, Ref, Figure 1, is based on the Saturn C-2 vehicle configuration having a payload capability of 50,000 lb.

In light of present weight assumptions and the objective of a 10,000 lb. returnable payload, approximately 21,000 lb. of fuel can be ground loaded and carried into obrit with the completely assembled, orbital launched, space vehicle. An additional 297,000 lb. of propellants will then have to be transferred into the space vehicle by a currently undetermined number of refueling missions. Each refueling mission shall not exceed a gross payload of 50,000 lb.

Space vehicle propellant weight breakdown, by stages, is presented as follows:

TABLE I

	1st Stage	2nd Stage	3rd Stage	Total
Fuel (LH_2)	36,000	11,890		47,890
(N_2H_4)			15,420	15,420
Oxidizer LO_2	180,000	59,390		239,390
(N_2O_4)			15,900	15,900
Total				297,000

If the above estimated weights should increase as a result of more detailed vehicle studies, the 21,000 lb of ground loaded propellant will be reduced accordingly.

The above information has been presented in report form by R. Reichert, S&M Lab, ABMA, See Ref. No. 1.

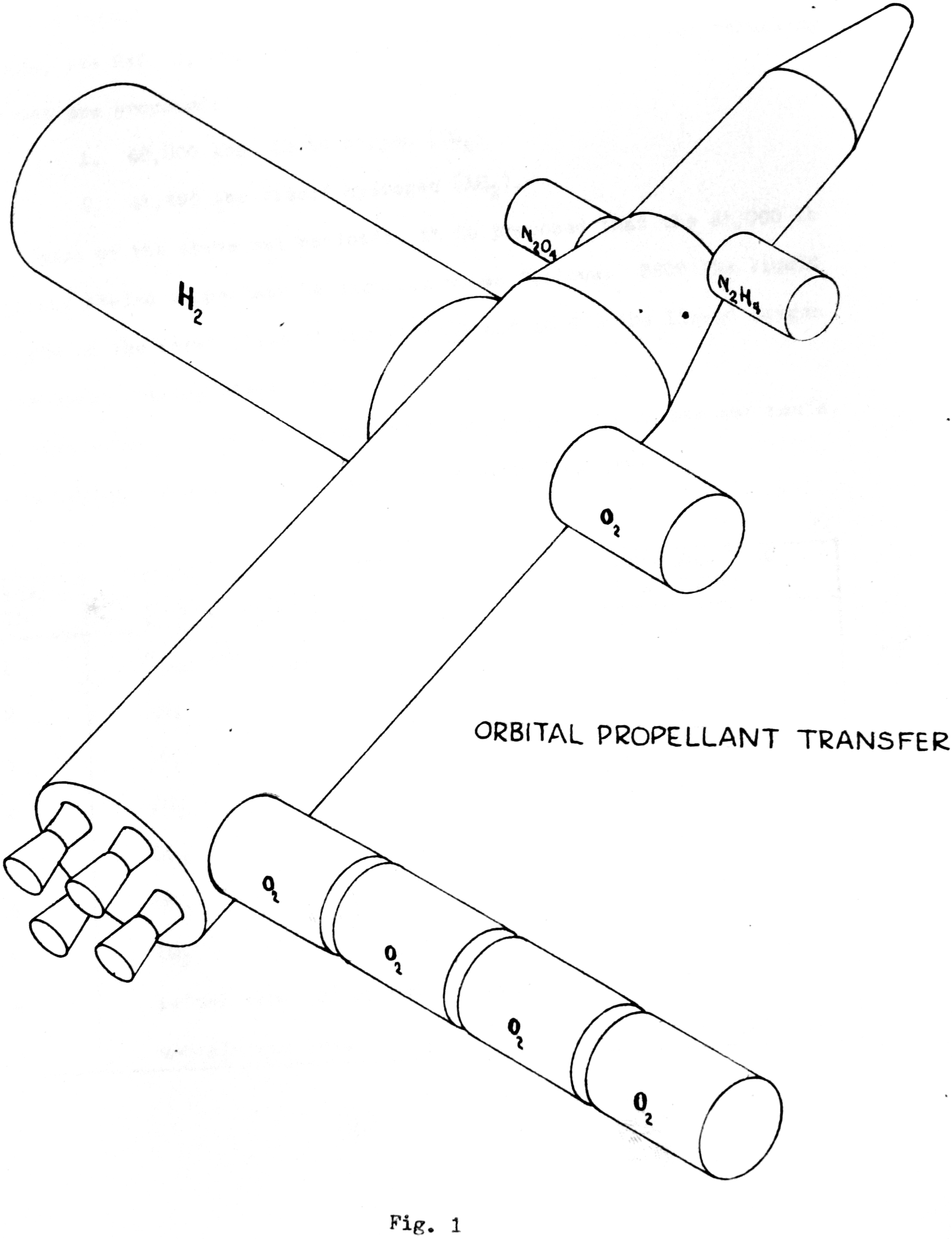


Fig. 1

As a result of the more detailed investigation of the LO_2 and LH_2 systems, see Ref. 2, the following net propellant payloads for refueling missions are proposed:

1. 48,000 lbs liquid oxygen (LO_2)
2. 41,790 lbs liquid hydrogen (LH_2).

Based on the above net payloads, it is proposed that the 21,000 lb of ground loaded propellant be distributed as follows: 8600 lbs liquid hydrogen in the first stage of the vehicle and 12,400 lbs liquid oxygen in the lunar braking stage, Ref. Figure 2.

With these net payloads and a maximum firing rate of one per month, the following proposed launch sequence appears feasible:

TABLE II

FIRING ORDER	PAYLOAD DESCRIPTION	STORAGE TIME IN MONTHS
1	N_2O_4 and N_2H_4	8
2	LO_2	7
3	LO_2	6
4	LO_2	5
5	LO_2	4
6	LO_2	3
7	LH_2	2
8	refuel crew (2) or remote manipulator	1
9	vehicle and crew	refill time

If two propellant loads were lost the storage time would increase two months. If payload no. 8 (crew) or no. 9 were lost it is assumed that the project would be indefinitely postponed.

On the basis of the foregoing weight and payload studies, the liquid hydrogen (LH_2) storage time has been tentatively established at two months plus refill time.

ORBITAL REFUELING AND

GROUND LOADING SCHEME

Fig. 2

V LIQUID HYDROGEN REFUELING CONCEPT

A. Tank Size and Payload Determination

It is initially assumed, and later verified by weight estimates, that 15 per cent of the gross payload will represent hardware. This leaves a net of 42,500 lbs of LH_2.

Prelaunch LH_2 and tank temperature is 14^oK. Liquid hydrogen temperature at the end of two months storage will be approximately 21^oK. Tank volume, based on LH_2 density at end of storage will be:

$$LH_2 V_1 = \frac{LH_2 \text{ weight}}{\text{Density at } 21^oK} = \frac{42,500}{4.4} = 9650 \text{ cu ft}$$

therefore, the tank will be designed for a 10,000 cu ft volume. By ground loading 8600 lbs, the liquid hydrogen refueling requirements of the space vehicle can be accomplished in one payload which will include 5% for transfer losses.

B. Insulation

1. Design Features. A cross section through the proposed orbital refueling tank would reveal the following pertinent design features:

a. A stainless steel propellant tank designed to support the entire payload.

b. A two inch thick layer of Linde SI-4 super insulation.

c. An outer aluminum shell, or foil, to provide protection from aerodynamic forces.

d. A helium gas purge system (ground equipment) to remove air from the SI-4 super insulation prior to tanking.

e. A six inch thick layer of styrofoam. This insulation blanket will be removed prior to launch by the umbilical tower swing arms.

2. Prelaunch and In-flight. In accordance with items a. and b. above, the tank shall support the entire payload and be attached to the third stage at approximately six points. Access holes in the permanent insulation will be provided at the support points. Upon separation, spring loaded, insulated doors will cover the openings to reduce the heat transfer.

The permanent insulation will be a two inch thick layer of Linde SI-4 super insulation, see Ref. 8. This insulation is designed primarily to operate in a vacuum (.01 micron Hg or less) such as between the evacuated walls of containers of the Dewar type.

The permanent SI-4 super insulation will be covered with an aluminum shell, or foil, to protect the SI-4 from aerodynamic forces encountered during powered flight. The surface finish of this protective covering should also have the following characteristics:

a. High reflectivity to solar radiation

b. High emmissivity to infrared.

The helium gas purge system will be provided to obviate the problem of liquefying the inherently entrapped air and consequent replacement air due to the low pressure created by condensation. If left uncorrected, this condition of liquefication of air within the insulation proper would destroy the heat transfer properties of the SI-4 super insulation.

A six inch layer of styrofoam, see Ref. 9, will be provided to supplement the Linde SI-4 insulation during prelaunch since SI-4 is extremely inefficient unless in a vacuum. The styrofoam will be employed in conjunction with a launch site helium refrigeration system capable of maintaining the loaded propellant at 14 ^{o}K. Utilization of the styrofoam will allow the capacity of the refrigeration unit to be economically within reason.

The prelaunch heat input calculations are based on six inches of styrofoam. The actual thickness will be determined on the basis of economics involving refrigeration capacity vs. insulation cost. Included in the latter would be the cost of installation and operation for the umbilical equipment required.

C. Heat Transfer Calculations

1. Prelaunch. The following prelaunch heat transfer calculations, Ref. Fig. 3, neglect the temperature drop through the metallic walls.

For the purpose of this initial calculation, and since the tank diameter is quite large, flat wall characteristics are assumed.

The surface area of the tank based on 220 inch diameter and a 37 foot length, is 2700 sq ft.

a. Neglect outside film coefficient, see Ref. 4 and 5 then:

$Q = UA\Delta T$

$Q = UA\,(T_4 - T_1)$

$A = 2700\ ft^2$

$$U = \frac{1}{\frac{\Delta X_S}{K_S} + \frac{\Delta X_L}{K_L}}$$

$\Delta x_s = 6$ inches

$\Delta x_L = 2$ inches

$k_s = .15$ BTU-in/ft^2-hr-^{o}F, based on mean t = -150^{o}F (see Fig. 4)

$k_L = .058$ BTU-ft/ft^2-hr-^{o}F

$$U = \frac{1}{\frac{6}{.15} + \frac{2}{(12)(.058)}} = .0233$$

$$Q = (.0233)(2700)(100 + 435) = 33,700 \frac{BTU}{hr}$$

Mean temperature of styrofoam was assumed to be -150°F.

Check on mean temperature assumption,

$$Q = \frac{k_s}{\Delta X_s} A\Delta T$$

$$T = \frac{Q}{A}\frac{\Delta X_s}{k_s} = \frac{(33,700)(6)}{(2700)(.15)} = 500^{\circ}F$$

$$T_4 - T_2 = 500; \; T_2 = 100 - 500 = -400^{\circ}F$$

$$T_{mean} = \frac{T_4 + T_2}{2} = \frac{100 - 400}{2} = -150^{\circ}F$$

b. Assume outside film results from still air, (for a vertical plate in still air).

$$h = (.3)(\Delta t)^{.25}$$

Assume $\Delta T = 20^{\circ}F$

$$h = (0.3)(20)^{.25} = (.3)(2.12) = .636$$

$Q = UA\Delta T$ Assume mean temperature of styrofoam $-160^{\circ}F$

$$U = \frac{1}{\frac{1}{h} + \frac{\Delta X}{K_s} + \frac{\Delta X}{K_L}} = \frac{1}{\frac{1}{.636} + \frac{6}{.147} + \frac{}{(6)(.058)}} = \frac{1}{(1.58) + 41 + 2.88}$$

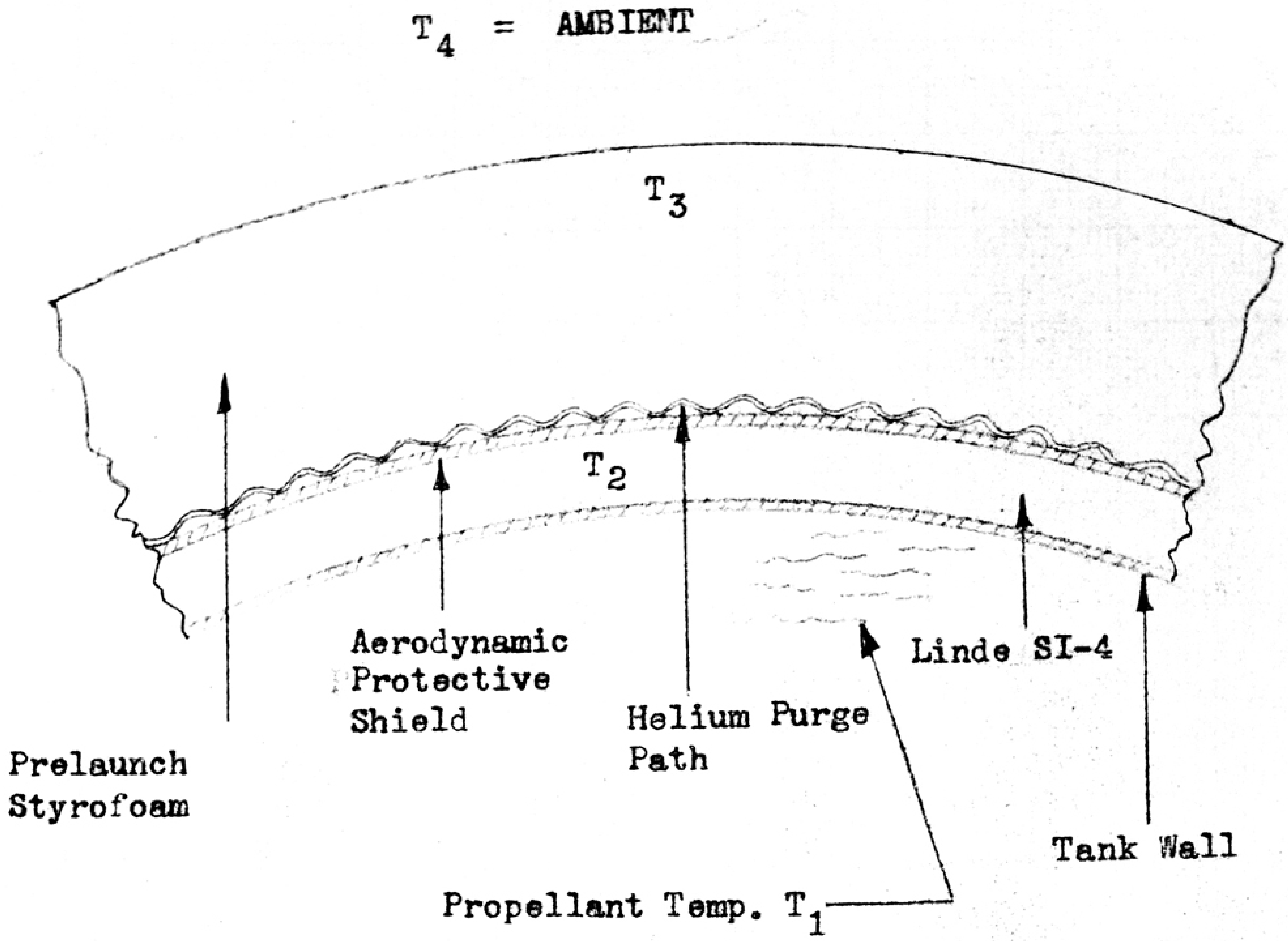
$$U = \frac{1}{45.46} = .022$$

$$Q = (.22)(2700)(535) = 31,700 \frac{BTU}{hr.}$$

Check on ΔT and mean temperature assumption.

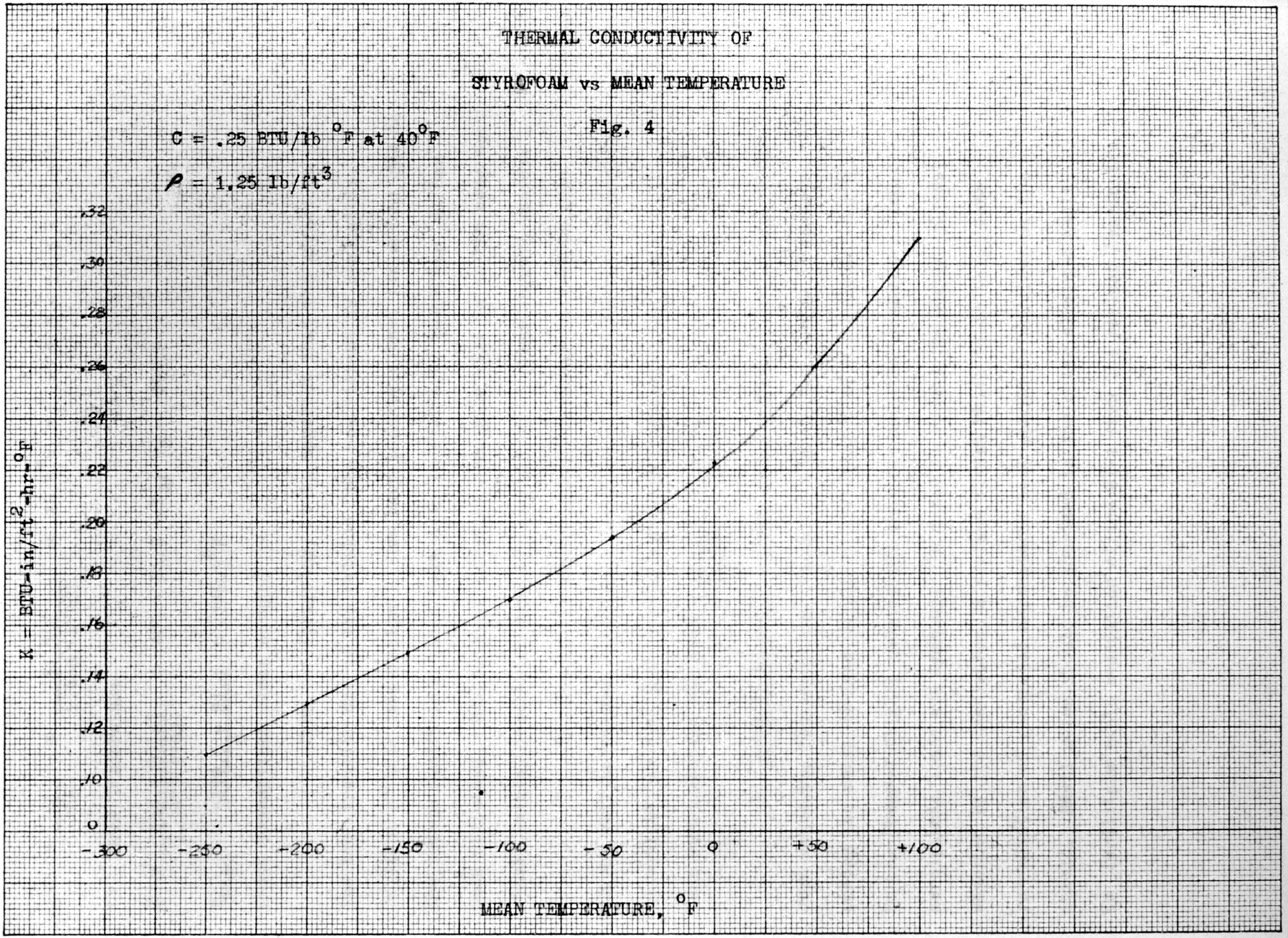
(a) ΔT

$$Q = hA\,\Delta T$$



CROSS SECTION THROUGH STORAGE TANK AT PRELAUNCH

Fig. 3



THERMAL CONDUCTIVITY OF
STYROFOAM vs MEAN TEMPERATURE
Fig. 4
C = .25 BTU/lb °F at 40°F
ρ = 1.25 lb/ft³
K = BTU-in/ft²-hr-°F
.32
.30
.28
.26
.24
.22
.20
.18
.16
.14
.12
.10
0
-300
-250
-200
-150
-100
-50
0
+50
+100
MEAN TEMPERATURE, °F

$$\Delta T = \frac{Q}{hA} = \frac{31,700}{(.636)(2700)} = 18.5^{\circ}F; \; t_3 = 100 - 18.5 = 81.5^{\circ}F$$

(b) Mean temperature

$$T = \frac{(31,700)(6)}{(.147)(2700)} = 480 = 81.5 - T_2; \; t_2 = -398.5$$

$$T_{mean} = \frac{-398.5 + 81.5}{2} = 158.5^{\circ}F$$

(Both assumptions close enough)

From the foregoing, it is apparent that the effect on the heat transfer due to the outside film coefficient is negligible. In light of this comparision it has been theorized that the effect due to the inside coefficient would have even less effect, therefore, it has been neglected. Additional sources of heat gain will be at the nose cone attach points, insulation supports and the third stage support points.

In view of the above, a heat transfer rate of 40,000 BTU/hr will be assumed in order to estimate the refrigeration required to obviate the heat gain during prelaunch preparations.

Tons refrigeration required $= \frac{40,000}{12,000} = 3.34.$

2. Powered Flight. At some predetermined time in the count-down sequence, the styrofoam insulating blanket is removed, thus exposing the outer surface shield of the Linde SI-4 which is at -400°F. During powered flight, this shield must protect the insulation from aerodynamic loads and heating. In order to reduce the amount of heat absorbed by the insulation, a surface finish which tends to promote re-radiation of generated aerodynamic heat should be provided on the shield. Small holes should be provided in the outer aluminum shell to facilitate rapid evacuation of the SI-4 when the payload leaves the

atmosphere. Linde SI-4 with small holes in each sheet of aluminum foil would also aid in rapid evacuation.

The problem of aerodynamic heating and its effect on the orbital refueling proposal is being investigated by the Aerodynamic Heating Section of the S & M Lab, ABMA.

3. Orbital Storage. Storage of liquid hydrogen in a low geocentric orbit for relatively long periods of time may be accomplished as follows:

a. Vented tanks with a net positive heat input.

b. Non-vented tanks with a net positive heat input.

c. Non-vented tanks with a zero heat input.

In evaluating the above methods it is apparent that any concept which depends upon venting for storage is immediately faced with the basic problem of venting only the vapor while retaining the liquid. Since under zero gravity conditions, the position of a liquid surface level can not be predicted, proper location of a vent valve is difficult. Proper vent valve location is possible in the event that a centrifugal force field is created by inducing a spinning motion into the tank. The introduction of such a motion however, would require energy, be hard to control, and possibly present personnel safety probelms.

Storage by maintaining a zero heat input would require some kind of refrigeration equipment and its power supply. For the specific problem being considered it is felt that adequate storage time could be realized without this additional equipment. Future projects requiring much longer storage times may, however, warrent the development of such equipment.

The most practical storage scheme appears to be in the form of non-vented tanks. Very low initial vapor pressures will be employed to keep the final storage pressure as low as practical.

It is proposed that liquid hydrogen be initially placed in the orbital storage tank at 14^{o}K with a vapor pressure of 1.13 psia. Final tank conditions just prior to transfer will be approximately 21^{o}K (18 psia).

Since the tank will not be built to structually withstand a negative internal pressure, it will have to be pre-filled with gaseous helium at atmospheric pressure or above. The tank will have a pressure regulated vent valve to allow the helium to escape as it is displaced by the liquid hydrogen being loaded. After the ground filling operation is completed the total pressure within the tank will consist of a hydrogen partial pressure of 1.13 psia plus at least 13.6 psia helium.

At a predetermined time during powered flight, the vent valve should be locked open until an altitude is reached where the atmospheric pressure is equal to the vapor pressure of LH_2 at 14^{o}K which is 1.13 psia.

This procedure is necessary to eject the helium, leaving nearly pure hydrogen vapor in the unfilled portion of the tank.

A. <u>Preliminary Heat Transfer Study</u>. To determine if orbital storage for a period of two months (1440 hrs) is possible with a non-vented tank, we will first assume the liquid hydrogen possesses infinite thermal conductivity. This allows the whole LH_2 mass to uniformly absorb heat. Assuming two inches of Linde SI-4 insulation ($k = 2.5 \times 10^{-5}$ $^{BTU-ft}/ft^2$-hr-^{o}F) with an outside temperature of -9^{o}F.

Heat input, $Q = kA \dfrac{\Delta T}{\Delta X}$

ΔT = outside skin temperature - temperature of LH_2

LH_2 temperature = $-435^{o}F$

$$Q = \frac{2.5 \times 10^{-5}\ 2700 \times (-9 + 435)}{1/6} = 173.00 \frac{BTU}{hr}$$

$$\text{Total heat gain} = 173 \frac{BTU}{hr} \times 1440 \text{ hrs} = 249{,}000 \text{ BTU}$$

Amount of temperature rise:

$Q = MC\Delta T$

$\Delta T = Q/CM$

$C = 2.0\ ^{BTU}/lb\text{-}^{o}F$

$M = 41{,}790$ lb

$$\Delta T = \frac{249{,}000}{(41{,}740)(2)} = 2.97^{o}F$$

The storage time of two months is possible with a LH_2 final temperature of 15.6 ^{o}K. Since a maximum temperature of $21^{o}K$ is permissable, quite a safety factor is provided.

In reality, the LH_2 possesses a finite thermal conductivity and the heat absorbing process is not quite so efficient. The LH_2 near the tank surface will experience a much faster temperature rise than the LH_2 near the tank center. Due to the absence of convective forces, localized "hot spots" must be avoided. An important factor in storage time is the temperature gradient through the cylinder. Since the LH_2 near the tank wall will reach the upper temperature limit first, its' temperature history during storage is of major importance. At any time during storage, the tank pressure will be equal to the vapor pressure corresponding to the maximum fluid temperature within the tank.

From a heat transfer standpoint, a cylindrical tank should have hemispherical ends to avoid localized temperature build up. From a stress standpoint the best design consists of spherical bulkheads with a radius equal to the cylinder diameter. The transition between bulkhead and cylinder is made by a radius equal to one-tenth the diameter.

b. <u>Transient Heat Transfer Study</u>. Under the previously described launch conditions it can be assumed that the tank of liquid hydrogen would reach its orbit in the subcooled condition ($-434^{o}F$) and that an equilibrium radiation temperature of $32^{o}F$ would exist on the outer skin. Linde SI-4 insulation has a conductivity of $k = 2.5 \times 10^{-5}$ $\text{BTU-ft}/\text{hr-ft}^2\text{-}^{o}F$ in a vacuum (.01 micron or less) but due to possible effects of construction and compacting during powered flight a value of 5.0×10^{-5} $\text{BTU-ft}/\text{hr-ft}^2\text{-}^{o}F$ was assumed. Also, since no heat capacity factor for the Linde SI-4 insulation was available, the transient heat transfer thru the insulation was neglected and a steady state heat transfer coefficient of $h = 30 \times 10^{-5}$ $\text{BTU}/\text{hr-ft}^2\text{-}^{o}F$ exists at time 0. These assumptions, in all probability, will provide an additional small factor of safety.

In order to further simplify the heat transfer calculations, it was assumed that the equilibrium radiation temperature existed over the entire surface and throughout the storage period. These assumptions will give a very pessimistic temperature rise but thus provide an additional margin of safety.

In "Heat Transfer", by Jakob, Volume I, see Ref. 3, are derived the equations for an infinitely long cylinder and for a sphere of radius "s" for a sudden temperature change of the environment.

While neither of these conditions fits the present actual problem the solution will be between them if the radius of the sphere is taken equal to that of the cylinder.

For the infinitely long cylinder subjected to a sudden temperature change of its environment:

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

Where:

θ = Temperature excess after time τ

= $t - t_o = t - 32$

θ_i = temperature excess at time $\tau = 0$

= $t_i - t_o = -434 - 32 = -466^oF$

s = radius of cylinder = 9'

r = distance from center at which temperature is to be determine = s = 9'

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

τ = elapsed time = 1440 hours

k = conductivity of LH_2 = 655×10^{-4} BTU-ft/hr-ft^2-^oF

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

$bs = \frac{hr}{k} = \frac{30 \times 10^{-5} \times 9}{655 \times 10^{-4}} = 0.04122$

$J_o(ns)$ = Bessel function of first kind, zero order, see Ref. 6 and 7

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

Solving:

$0.04122 = ns \frac{J_1(ns)}{J_o(ns)}$ for the first 3 roots gives;

$ns_1 = 0.2857 = X_1$

$ns_2 = 3.8345 = X_2$

$ns_3 = 7.0215 = X_3$

substituting these values for the first 3 terms of the summation gives,

$\frac{\theta}{\theta_i} = 0.98029$ or 0.9803 to four places.

and therefore, $\frac{t - 32}{-466} = 0.9803$

$t = -456.8 + 32 = -424.8^{\circ}F$ ($19.2^{\circ}K$)

For the sphere subjected to a sudden temperature change of its environment:

$$\frac{\Theta}{\Theta_i} = \sum_{V=1}^{V=\infty} 2 \, \frac{\sin \psi_V - \psi_V \cos \psi_V}{\psi_V - \sin \psi_V \cos \psi_V} \, e^{-\psi_V \frac{\alpha \tau}{S^2}} \, \frac{\sin\left(\psi_V \frac{r}{S}\right)}{\psi_V \frac{r}{S}}$$

Where:

$\psi_V =$ roots obtained from $bs = 1 - \cot(ns)$ and other quantities as previously defined.

Solving:

$0.04122 = 1 - \cot(ns)$ for the first 2 roots

gives:

$\psi_1 = 0.3544$ and $\psi_2 = 4.503$,

substituting these values for the first 2 terms of the summation gives:

$\frac{\theta}{\theta_i} = 0.9763$

and:

$t = 0.9763 (-466) + 32$

$= -455 + 32 = -423^{\circ}F$ ($20.4^{\circ}K$)

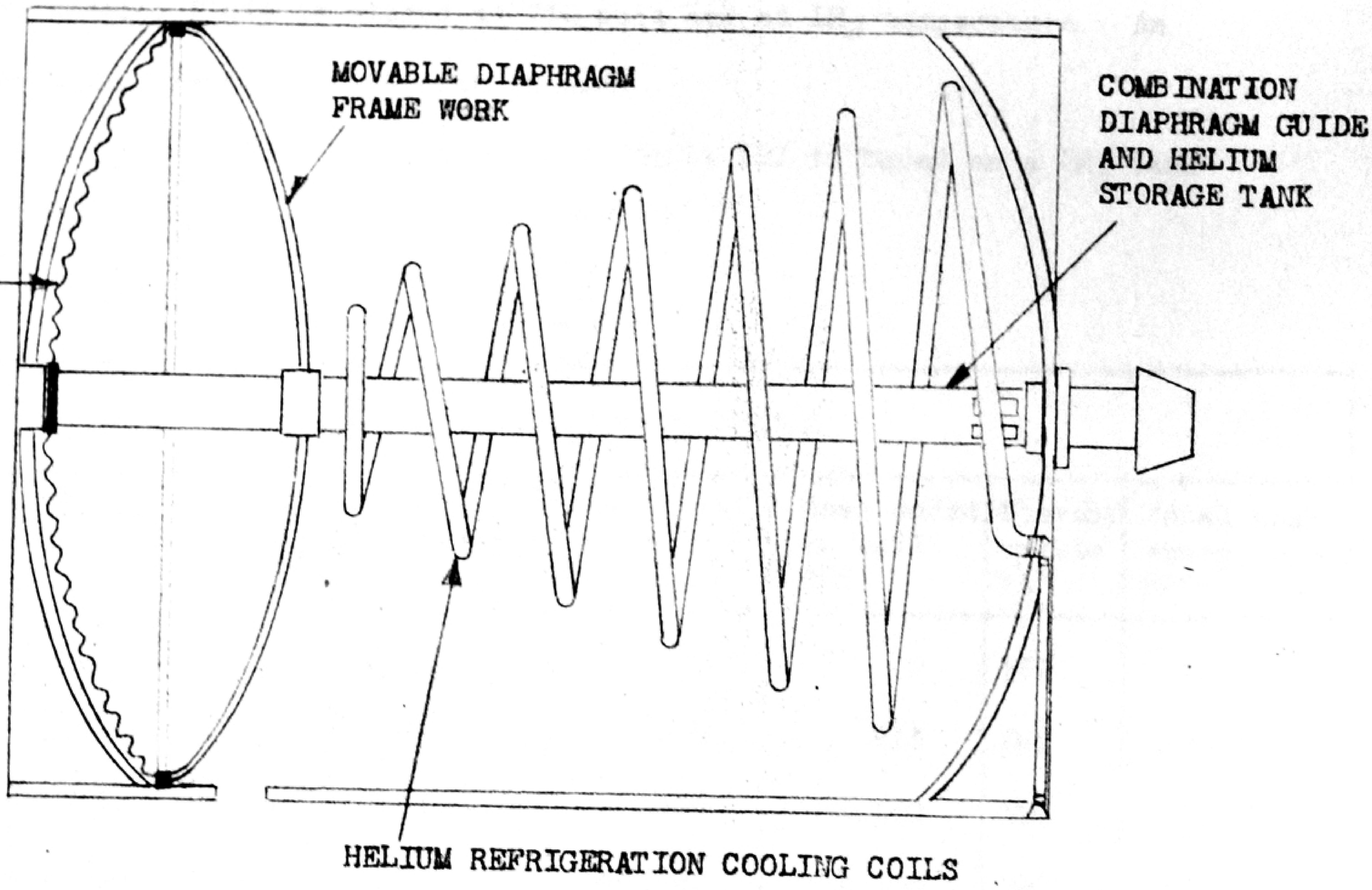
The actual tank, with L/D of approximately 2 and curved bulkheads, will have a liquid temperature near the outer wall and at the midpoint of the cylindrical portion approximately equal to that of the infinite cylinder. As one approaches either end of the cylinder the temperature will rise and approach that of the sphere. It will not reach the temperature of the sphere however, because the ends are joined to the cylindrical portion and heat will be conducted from the ends into the cylinder, thus raising the temperature of the cylindrical part while decreasing the temperature of the curved ends.

The maximum temperature of the LH_2 will not be more than 20.4°K, and the vapor pressure at this temperature is only 15.3 psi. Hence, under the conditions set forth, it is feasible to store LH_2 in an earth orbit for 2 months without loss.

In addition it can also be seen that raising the equilibrium radiation temperature does not greatly effect the end result. If the outer temperature of the sphere is raised from 32°F to 70°F the maximum liquid temperature would be increased to only -422°F (21.0°K) a change of only 1/2°F.

D. <u>Propellant Transfer in Orbit</u>

1. <u>Proposed Transfer System</u>. The most satisfactory solution found was the use of a flexible diaphragm mounted on a movable framework, Ref. figure 6. This design will provide for complete LH_2 expulsion. The flexible diaphragm would not be subjected to severe bending as would a bladder. In order to eliminate high order stresses, the diaphragm will be designed with concentric corrugations. The data of Tables III and IV is preliminary and will be supplemented by further heat transfer studies currently being conducted.



DIAPHRAGM TRANSFER SYSTEM FOR LIQUID HYDROGEN

Fig. 6

2. Expulsion Methods. Two systems for expelling the propellant were investigated and are presented on the following pages. The one using compressed helium is preferred. The second method employs LH_2.

a. Compressed Helium. Storage of a compressed gas at LH_2 temperature restricts the choice of a suitable working fluid to Helium. The working fluid is stored at 750 psia and at LH_2 temperature. An expulsion pressure of 25 psia is used.

The following data in Table III is based on a LH_2 tank expulsion pressure of 25 psia.

TABLE III

COMPRESSED HELIUM WORKING FLUID							
Gas Temp. ^oK	Density lb/ft^3	Enthalpy change BTU/lb	Working fluid needed lb	Heat added to fluid BTU	Heat added to wall BTU*	Storage volume ft^3	Total heat energy req. BTU
18	.25	2	2500	5,000	0	360	5,000
33	.156	37.8	1560	59,000	318	225	59,318
55	.104	86.5	1040	90,000	2,140	150	92,140
75	.0642	131.5	624	82,000	5,340	90	87,340
100	.0480	189.0	480	90,600	12,500	69.4	103,100
150	.0312	298.0	312	93,000	34,300	45	127,300

* The heat added to the tank wall was computed with C_v vs ^oK, Ref. Fig. 5.

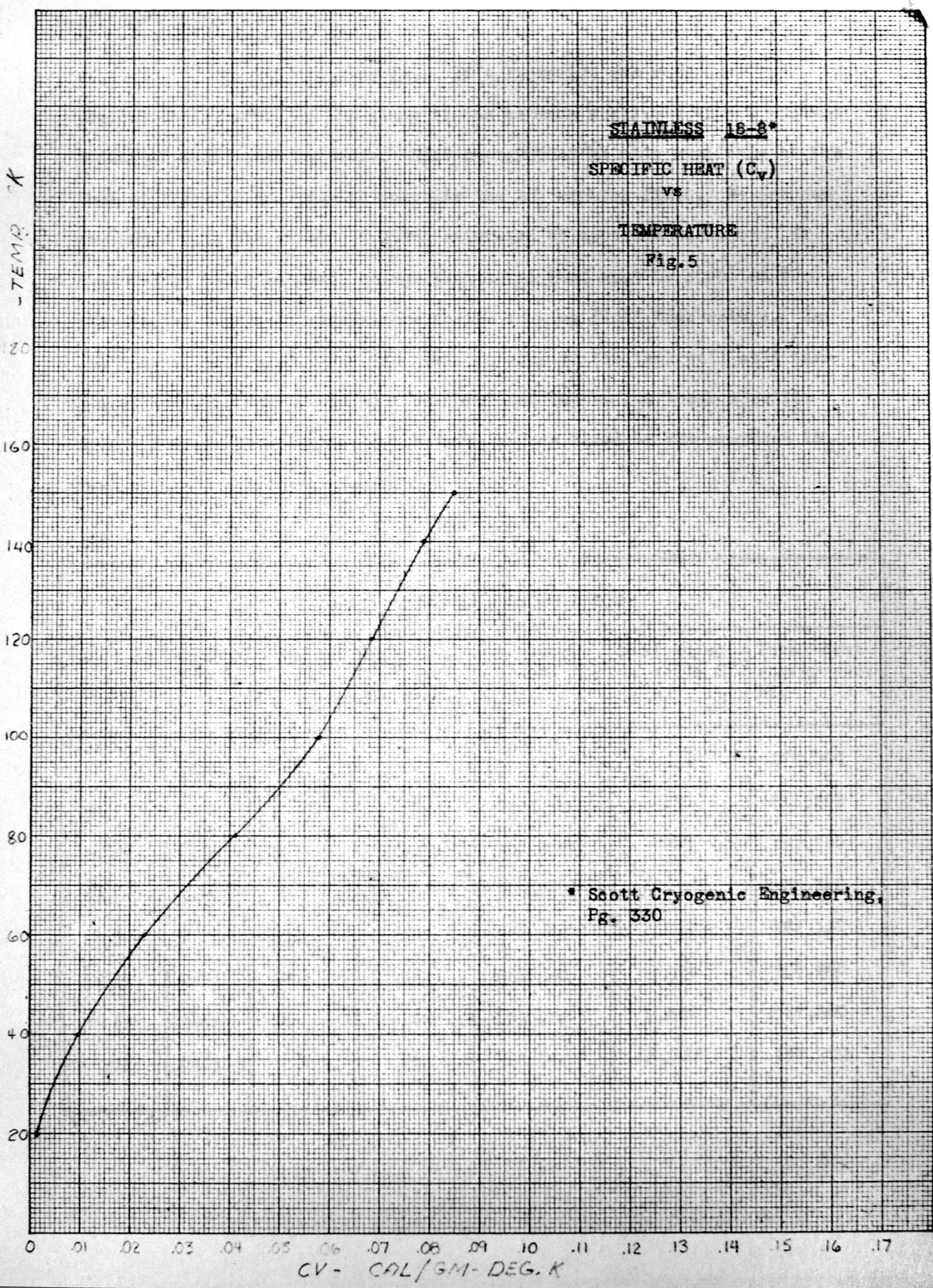
Approximately 45 cu ft of compressed helium at 750 psi may be stored in the center tube with little weight penalty. The volume limitation makes an operational gas temperature of 150 ^oK mandatory. This corresponds to a heat input requirement of 127,300 BTU in 50 minutes to maintain a flow of 1500 GPM. This input is equivalent to 44.6 KW. The energy available from the sun striking one end of the tank, with insulation removed and assuming 0.9 absorptivity, would be 100,000 BTU per hour. If a slower fill rate could be tolerated, a fill time of 1.27 hrs could be effected using solar energy alone.

The expulsion system weight and heat requirements are two very important characteristics which must be considered when choosing a system. The data in Tables III and IV show that the system weight decreases as the working fluid is superheated. With a sacrifice in weight, the heat required may be reduced in the helium system by reducing the degree of super heat. This feature is not offered in the LH_2 system.

The heat required for expulsion may be obtained from chemical energy or, as has already been indicated, from solar energy. Heat from chemical energy such as the combustion of H_2 and O_2 or the decomposition of hydrazine or hydrogen peroxide, may be supplied at any desired rate. Additional hardware is required if these systems are used.

Solar energy is readily available and may prove useful, especially if flow rates in the order of 900-1000 GPM can be tolerated. More solar energy may be captured with the use of folding reflectors if higher flow rates are necessary.

Another important consideration in system choice is pressure control. Compressed helium would probably be superior to LH_2 since a pressure regulated valve would control the flow from the storage tube, whereas controlling the flow of the LH_2 working fluid would present many problems under zero gravity conditions.


STAINLESS 18-8*
SPECIFIC HEAT (C_v)
vs
TEMPERATURE
Fig. 5
* Scott Cryogenic Engineering, Pg. 330
-TEMP. K
20
40
60
80
100
120
140
160
180
0
.01
.02
.03
.04
.05
.06
.07
.08
.09
.10
.11
.12
.13
.14
.15
.16
.17
CV - CAL/GM-DEG. K

b. Liquid Hydrogen. If a liquefied gas were used the logical choice would be LH_2 since it can be made available with a minimum amount of additional hardware. Heat energy must be made available at the time of transfer in order to vaporize the LH_2 working fluid. We are now presented with the choice of using saturated or superheated vapor. If superheated vapor is used, enough additional heat must be supplied to elevate the tank wall temperature to the working fluid temperature. In the case of saturated vapor, no heat transfer to the tank wall will take place, however, more fluid must be vaporized due to its lower specific volume. Table IV will illustrate this, and is based on the proposed LH_2 tank size and operating pressure.

TABLE IV

LIQUID HYDROGEN WORKING FLUID						
Gas Temp ^{o}K	Density lb/ft^3	Enthalpy change BTU/lb	Working fluid needed lb	Heat added to fluid BTU	Heat added to walls BTU	Total heat energy req. BTU
Saturated (22.5)	.1460	185	1460	270,000	0	270,000
40.0	.0785	274	785	215,000	673	215,673
50.0	.0505	319	505	161,000	990	161,990
60.0	.0449	364	449	163,000	2,570	165,570
70.0	.0393	410	393	161,000	4,130	165,130
80.0	.0337	457	337	154,000	6,260	160,260
90.0	.0303	504	303	152,000	9,300	161,300
100.0	.0281	550	281	154,500	12,500	167,000

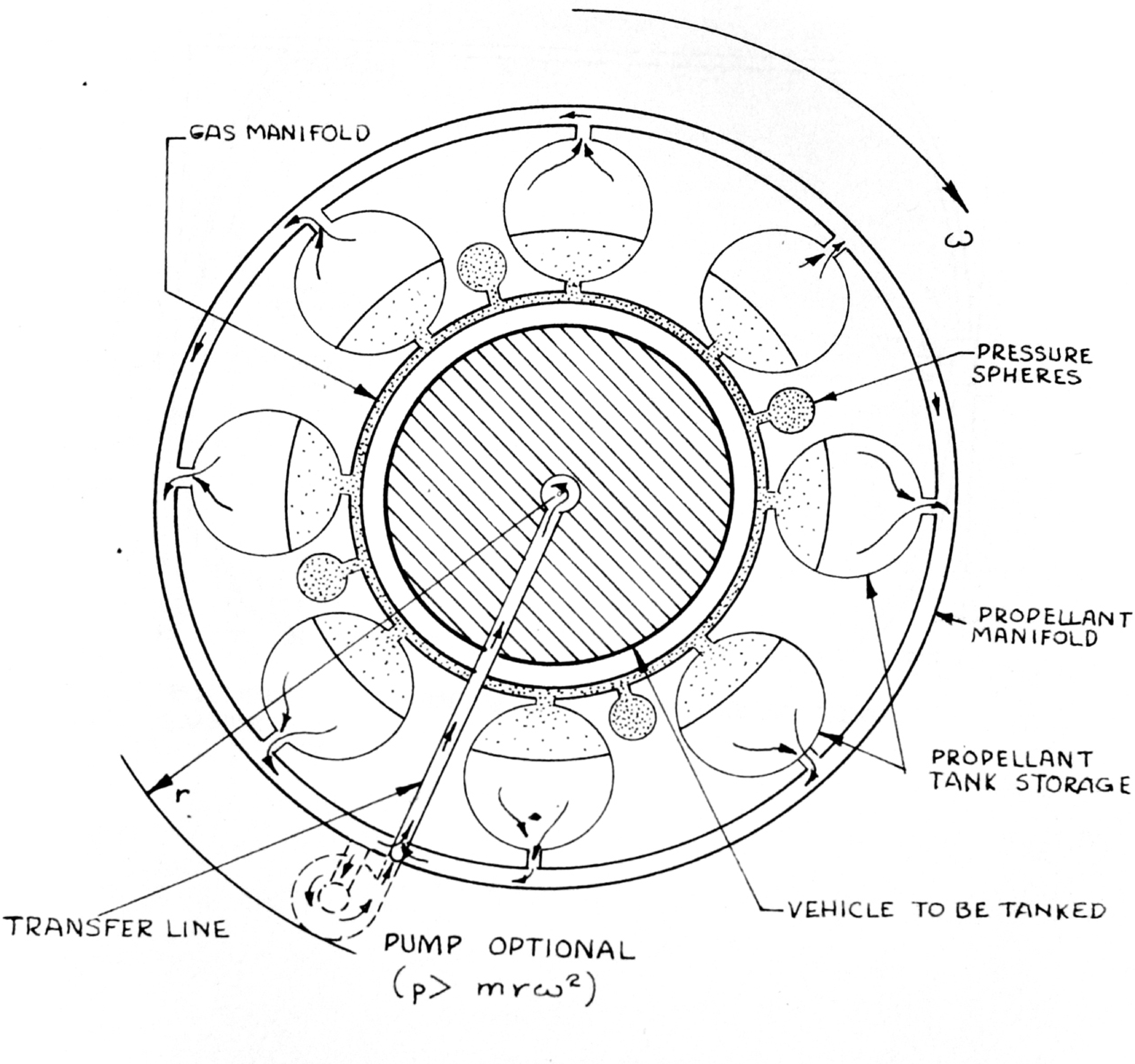
Based on an orbital fill rate of 1500 GPM (220 ft^3/min.) 50 minutes would be required to empty the tank. If the working fluid is superheated to 100^oK the rate of heat input to maintain 1500 GPM would be 167,000 BTU in 50 minutes or 58.6 KW. Solar heat supply would empty the tank in 1.67 hours assuming infinite thermal conductivity of the working fluid.

3. Other Transfer Systems. Some other transfer systems investigated included the following:

a. Introducing a spin motion into the tank, Ref. Fig. 7 This would establish a predictable fluid level but requires an energy input, manual plumbing and assembly job of considerable magnitude and would probably be hazardous from a personnel standpoint, both on and off board.

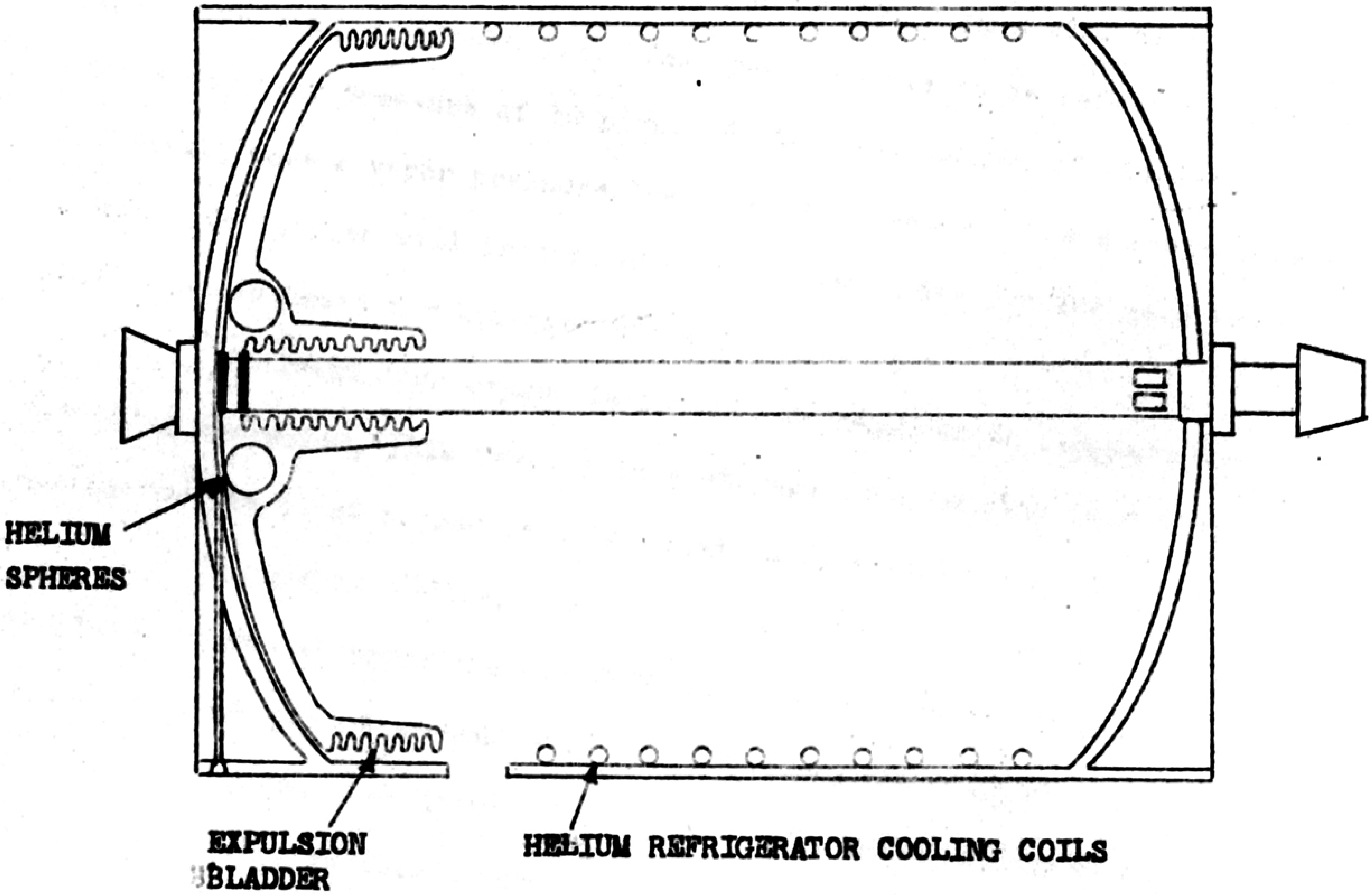
b. Introducing a linear acceleration to the tank would successfully establish a fluid level, however, this method has the inherent disadvantage of requiring a high form of energy and would also greatly effect vehicle location and orbit.

c. The use of a flexible bladder to separate the LH_2 from the working fluid, Ref. Fig. 8. This system would work at reasonable temperatures, but material problems would be presented at liquid hydrogen temperature if severe flexing were demanded.



SPIN CLUSTER TRANSFER SYSTEM

Fig. 7



BLADDER TRANSFER SYSTEM

Fig. 8

VI LIQUID OXYGEN REFUELING CONCEPT

A. General

The maximum storage time required for oxygen will be approximately nine months, (Table II). The space vehicle to be refueled operates with a LO_2 tank pressure of 30 psia. At the termination of storage the LO_2 should have a vapor pressure less than this value. The storage scheme proposed for oxygen will incorporate non-vented tanks for the same reasons which are discussed for hydrogen storage.

At least five oxygen tanks will be required to complete the refueling operation. This results in a storage time ranging from nine months for the first oxygen tank launched to five months for the last.

The ground filling operation (Fig. 2) will accomodate 12,400 lbs of liquid oxygen at approximately $70^{o}K$ and 0.9 psia vapor pressure. Orbital storage may be accomplished until the LO_2 temperature reaches $95^{o}K$ and 23,7 psia. The proposed pressure limit was established by a storage tank designed for 30 psia.

The number of LO_2 payloads was based on a net payload of 48,000 lb. A preliminary weight investigation will be presented later to verify this assumption. The expulsion system is the same as proposed for hydrogen storage except the helium working fluid is stored in four spheres located between the end bulkhead and the flexible diaphragm and no heat will be required to accomplish transfer at 1500 GPM.

The tanks will be connected in series during the refueling operation as shown in figure 1. Twenty-one cubic feet of helium, stored at 750 psia and liquid oxygen temperature, will provide enough energy to completely expel the LO_2 from one tank. Each tank will contain its own helium supply. The center tube is used both as a diaphragm guide and as an internal manifold through which LO_2 from connected tanks will flow to the space vehicle during transfer.

B. Prelaunch Heat Transfer Calculations

Values for thermal conductivity of styrofoam and Linde SI-4 vary greatly with temperature, (Fig. 4). The conductivity of the Linde SI-4 is taken as that of helium gas. Conductivity values will be based on mean temperature assumptions for both types of insulation. A trial and error method of solution is used to determine heat input.

Total heat transfer is through the bulkheads and cylindrical portion of the tank. These two surfaces will be handled separately.

The hydrogen tank calculations assumed flat wall conditions due to the large tank diameter. For the oxygen tank claculations this assumption will not be made, Ref. Fig. 3. Flat surfaces are assumed for bulkheads only.

Q_T = Q (cylinder) + Q (bulkheads)

Final mean temperature assumptions and k values.

Linde SI-4 (helium) at $100^{\circ}K$ and $k = 0.0405 \dfrac{BTU - ft}{ft^2\ hr\ ^{\circ}F}$

Styrofoam at $-100^{\circ}F$ and $k = 0.014 \dfrac{BTU-ft}{ft^2\ hr\ ^{\circ}F} = 0.17 \dfrac{BTU - in}{ft^2\ hr\ ^{\circ}F}$

$$Q_{cyl} = \frac{2\ (t_1 - t_3)\ L}{\dfrac{\ln D_2/D_1}{k_{12}} + \dfrac{\ln D_3/D_2}{k_{23}}}$$

$$Q_{cyl} = \frac{(6.28)(9.1)(443)}{\dfrac{\ln\left(\dfrac{9.43}{9.1}\right)}{0.0405} + \dfrac{\ln\left(\dfrac{10.43}{9.43}\right)}{0.014}} = \frac{25,300}{\dfrac{0.0363}{0.0405} + \dfrac{0.1016}{0.014}}$$

$$Q_{cyl} = \frac{25,300}{0.897 + 7.25} = \frac{25,300}{8.147} = 3100 \text{ BTU}$$

$$Q_{bulkheads} = UA\,(t_1 - t_3)$$

$$U = \frac{1}{\dfrac{\Delta x_s}{k_s} + \dfrac{\Delta x_L}{k_L}} = \frac{1}{\dfrac{6}{0.17} + \dfrac{1}{(6)(0.405)}} = 0.0253$$

(k_s at -100°F AND k_L at 100°K)(Assumptions)

$$Q_B = (.0253)(140)(443) = 1570 \text{ BTU}$$

$$Q_T = 3100 + 1570 = 4670^{\circ} \text{ BTU/hr}$$

Launch site refrigeration necessary to counteract heat gain through insulation.

$$\text{Refrigeration tons} = \frac{4670}{12,000} = .39 \text{ tons}$$

Weight of six inches of styrofoam

$$
\begin{aligned}
Wt &= \text{Wt cylinder} + \text{wt bulkhead} \\
&= A \times L \times \text{density} + A\ t\ \text{density} \\
&= (9.1)(.0785)(D_3^{\,2} - D_2^{\,2})(1.25) = (70)(.5)(1.25) \\
&= \left[7.14\,(\overline{1043}^{\,2} - \overline{9.43)}^{\,2} + (35)\right] 1.25 \\
&= (143.0 + 35)\ 1.25 = 179 + 43 \\
&= 222.0 \text{ lb}
\end{aligned}
$$

The top bulkhead styrofoam will fly with the payload. The bottom needs no insulation because the third stage contains LH_2. These conditions result in a net weight of 179 lb of styrofoam to be removed prior to lift-off.

C. Transient Heat Transfer Study

Using the same assumptions and conditions for orbital storage of the liquid oxygen as were used for liquid hydrogen, the temperature of the outermost layer of the liquid oxygen was calculated. Since LO_2 is much more easily stored, an outer skin equilibrium temperature of $70^{o}F$ was used and the storage time taken as 10,000 hours. Four cases were investigated, the sphere, the infinitely long cylinder, the infinitely wide plane plate, and combination of the plane plate and cylinder.

Using the corresponding equations from "Heat Transfer" by Jakob, Volume I, the following temperatures were obtained.

1. For a sphere subjected to a sudden temperature change of environment, $\frac{\theta}{\theta_i} = 0.9369$ and $t = -307.6^{o}F$ $(84.5^{o}K)$.

2. For the infinitely long cylinder subjeced to a sudden temperature change of environment, $\frac{\theta}{\theta_i} = 0.9762$ and $t = -323.4^{o}F$ $(75.8^{o}K)$.

3. For the infinitely wide plane plate subjected to a sudden temperature change of environment.

$$\frac{\theta}{\theta_i} \sum_{v=1}^{v=\infty} \frac{2 \sin(\phi_v)}{\phi_v + \sin\phi_v \cos\phi_v} e^{-\phi_v^2 \frac{\alpha\tau}{S^2}} \cos(\phi_v \frac{x}{S})$$

Where: $\theta = t - t_o = t - 70$

$\theta_i = t_i - t_o = -333 - 70 = -403^{o}F$

$\alpha = 0.00408\ ft^2/hr$

$X = S = 4.55\ ft$

$\tau = 10{,}000$ hours

$k = 0.121\ BTU\text{-}ft/hr\text{-}ft^2\text{-}^{o}F$

$\frac{\alpha\tau}{S^2} = 1.9708$

ϕ_v = roots obtained from $\frac{ns}{bs} = \cot(ns)$

Where: $bs = \frac{hx}{k} = 0.011281$

Hence the first two roots are $= (ns)_1 = 0.105884$ and $= (ns)_2 = 3.1452$.

Using these values: $\frac{\theta}{\theta_i} = 0.97452$ and $t = -322.7^{o}F$ $(76.1^{o}K)$.

4. It has also been shown (Ref. 5) that the solution of transient heat conduction problems for finite bodies, such as rectangular parallelepiped or short cylinders, is the product of the solutions for the semi-infinite cases. Thus for a short cylinder with flat ends

$$\left(\frac{\theta}{\theta_i}\right)_{sc} = \left(\frac{\theta}{\theta i}\right)_c \left(\frac{\theta}{\theta_i}\right)_{FP} = 0.9762 \times 0.97452 - 0.95128 \text{ and } t = -313.4^{o}F\ (81.3^{o}K).$$

Of the above, item 4 approaches the actual tank configuration more closely than do any of the other cases but still represents a pessimistic view since the ends are not flat but curved. Hence it is safe to say that the maximum temperature of LO_2 will not exceed -313.4^{o}F (81.3^{o}K). The vapor pressure at this temperature is only 5.37 psia.

VII LAUNCH SITE REFRIGERATION

Liquid hydrogen and liquid oxygen are commonly available at atmospheric pressure or above, see Ref. 10, 11, and 12. The feasibility of "no loss" orbital storage depends on obtaining sub-cooled liquids at, or very close to, their freezing point. A helium refrigeration system will, therefore, be employed to lower the propellants to the required temperature and maintain this condition until launch.

The one month period between firings will be used to accomplish the desired propellant sub-cooling, thus reducing the required capacity of the refrigeration system. This will take place in refrigerated storage tanks located near the launch site. Sufficient information is not available at this time, on the heat transfer characteristics of the storage tanks, to estimate the amount of refrigeration necessary to accomplish initial cool down.

The sub-cooled propellant is transferred to the orbital storage tank during prelaunch and this sub-cooled condition is maintained by a helium refrigeration unit located at the launch site, possibly on the umbilical tower. (The refrigerator cooling coils are located inside of the orbital storage tank and will fly with the vehicle, Ref. Fig. 6. A quick disconnect coupling will be employed to separate the cooling coils from the refrigeration unit. During orbital transfer these coils will be collapsed by the expulsion diaphragm.) Preliminary heat transfer investigations indicate that approximately 3.5 tons of refrigeration are necessary to maintain the LH_2 payload at $14^{o}K$ during prelaunch. However, due to unavoidable heat gained during the filling process, additional refrigeration capacity will have to be added to reduce the temperature to its original value. Line length, line insulation and filling time will determine the heat gained, and the length of prelaunch time available to remove this heat will determine the additional tons of refrigeration necessary. Approximately 0.4 tons will be necessary for LO_2.

VIII HYDRAZINE AND NITROGEN TETROXIDE

These two propellants may be combined into one payload since their combined net weight is only 32,850 lb and their relatively high density results in a small total volume. The freezing and boiling points of hydrazine (N_2H_4) are 35°F and 236°F respectively at atmospheric pressure. The freezing and boiling points of nitrogen tetroxide (N_2O_4) are 11°F and 70°F respectively at atmospheric pressure. The upper temperature limit may be increased if the vapor pressure during storage were allowed to rise above atmospheric. (Vapor pressure of N_2O_4 at 100°F is 30.7 psia, see Ref. 13.) Storage under these conditions may be achieved for indefinite periods of time with no insulation other than choosing the proper tank surface thermal radiation properties. Due to their storage ease, these propellants should be the first payload to be launched.

The expulsion system may be either in the form of a bladder or mechanical diaphragm. A bladder may be the lightest of the two. Compressed helium will be used as the working fluid.

IX PRELIMINARY DESIGN SUMMARY

A. Liquid Hydrogen Payload

1. Hydrogen Tank

Volume	10,000 ft^3
Outside diameter	220 in.
Length	42 ft
Wall thickness	.03 in.
Material, stainless 301	
Construction	Monocoque

2. Expulsion System

Type	Diaphragm
Working fluid	Helium gas
Helium storage pressure	750 psi
Helium storage temp.	LH_2 storage temperature
Helium storage volume	45 ft^3
Stored in center tube	

3. Hardware Weight

Tank		3400 lb
Prelaunch insulation (removable)		1783 lb
Permanent insulation (Linde SI-4)		2110 lb
Aerodynamic protection (Aluminum cover .02 in. thick)		755 lb
Expulsion system		
Helium	213 lb	
Center tube	383 lb	
Total		695 lb

Miscellaneous hardware

Ground fill valve	50 lb	
Orbital fill valve	50 lb	
Tank to vehicle coupling	100 lb	
Attitude control	150 lb	
Tank to 3rd stage hardware	100 lb	
Expulsion diaphragm	800 lb	
Total misc. hardware		1250 lb

Total hardware	8210 lb

Net payload

Gross payload	50,000 lb
Hardware	- 8,210 lb
Net payload	41,790 lb

B. Liquid Oxygen Payload

1. Oxygen Tank

Net payload	48,000 lb
Tank volume	678 ft^3
Diameter and length	9.1 ft
Material stainless 301	
Wall Thickness	.03 in.
Construction	Monocoque

2. Expulsion System

(Diaphragm or bladder)

Working fluid	helium gas
Helium storage conditions	750 psia and 80^{o}K

Helium expulsion conditions	30 psia and 80°K
Storage volume	25.1 ft^3
Storage sphere diameter, 4 re'd	2.3 ft
Sphere material, stainless 301, (y.s. = 116,000 psi)	
Safety factor	1.3

3. Hardware Weight

Tank		500 lb
Center tube (8" O.D. x 11.5 ft long)		124 lb
Expulsion system		
Helium	42.4 lb	
Spheres	162.0 lb	
Misc. Hardware	20.0 lb	
Total		224 lb
Aerodynamic cover (Al - .02 in. thick)		115 lb
Misc. hardware		
Expulsion diaphragm	243 lb	
Attitude control	150 lb	
Valves & flex. line	200 lb	
3rd stage connections	100 lb	
Total		693 lb

TOTAL HARDWARE WEIGHT 1,979 lb

Net Payload

Gross payload	50,000 lb
Hardware	1,979 lb
Net payload	48,021 lb

C. Hydrazine (N_2H_4) Partial Payload

1. Storage tank data for N_2H_2

N_2H_4 required	15,450 lb
Add for transfer losses	750 lb
Net payload	16,200 lb

$$\text{Volume of } N_2H_4 = \frac{\text{Net payload}}{\text{density}}$$

$$= \frac{16{,}200 \text{ lb}}{63.0 \text{ lb/ft}^3}$$

$$= 257 \text{ ft}^3$$

Make tank volume 275 ft^3 to allow for expansion.

Assume bulkheads are dished with a radius equal to the tank diameter.

Let length of cylinder equal diameter of cylinder = 6.72 ft.

2. N_2H_4 Hardware Weight

Tank		273 lb
Expulsion system		
"Mylar" bladder	4.7 lb	
Helium	0.2 lb	
Spheres	331.0	
Total		335.9 lb
Misc. hardware		
Valves and hose	150.0 lb	
3rd stage connection	50.0 lb	
Total		200.0 lb
TOTAL HARDWARE		808.9 lb

3. Gross N_2H_4 Component of Payload

N_2H_4 net	16,200 lb
Total hardware	809 lb
Gross N_2H_4 weight	17,009 lb

D. Nitrogen Tetroxide (N_2H_4) Partial Payload

1. Storage Tank

N_2O_4 required	15,900
Add for transfer losses	750
Net payload	16,650

$$\text{Volume of } N_2O_4 = \frac{\text{Net payload}}{\text{Density}}$$

$$= \frac{16,650}{92.8}$$

$$= 180 \text{ ft}^3$$

Make tank volume 190 ft^3 to allow for expansion.

Assume tank diameter 6.72 ft.

Length = 4.43 ft.

2. N_2O_4 Hardware Weight

Tank		212 lb
Expulsion system		
"Mylar" bladder	3.6 lb	
Spheres	230.0	
Total		233.6 lb
Misc. hardware		
Valves and hose	150 lb	
N_2O_4 tank connections	50 lb	
Total		200.0 lb
Total Hardware		645.6 lb

3. <u>Gross N_2O_4 Component of Payload</u>

N_2O_4 net	16,650 lb
Hardware	645
Gross N_2O_4 weight	17,295 lb

4. <u>Total Gross Payload</u>

Hydrazine partial payload	17,009 lb
Nitrogen tetroxide partial payload	17,295 lb
Total payload	34,304 lb

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