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STUDY AND DESIGN OF A CRYOGENIC PROPELLANT ACQUISITION SYSTEM

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PREFACE

This the sixth quarterly progress report on the program, "Study and Design of a Cryogenic Propellant Acquisition System." The period covered is 1 October to 30 December 1972. This work is being carried out by McDonnell Douglas Astronautics Company (MDAC) for the National Aeronautics and Space Administration, Marshall Space Flight Center, Huntsville, Alabama, under Contract NAS8-27685. Mr. G. M. Young serves as the principal NASA contracting officer representative. The MDAC technical effort is being conducted under the direction of G. W. Burge, Program Manager, and Dr. J. B. Blackmon, Deputy Program Manager. Major contributions to this report were made by J. N. Castle, B. R. Heckman, and D. W. Kendle.

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Section 1
INTRODUCTION

1.1 OBJECTIVES

The objectives of this project are to investigate, define, and demonstrate, through ground testing, an acquisition system for supplying subcooled LH_2 and LO_2 under in-orbit conditions to satisfy integrated cryogenic feed system requirements for advanced space systems such as a Space Shuttle cryogenic auxiliary propulsion system (APS) and main propulsion for an Advanced Spacecraft Propulsion Module (ASPM). This effort will concentrate on concepts that utilize the favorable surface tension characteristics of fine-mesh screens and will significantly advance cryogenic acquisition technology in general. The anticipated analytical and experimental results will provide a sound technology base for the subsequent design of cryogen supply subsystems for future space vehicles. These objectives will b achieved by a four-phase program covering 27 months.

1.2 PROGRAM SUMMARY

1.2.1 Phase I – Analysis

The objectives of this phase are to: (1) evolve conceptual designs for candidate acquisition systems, (2) formulate the analytical models needed to analyze these systems, and (3) generate parametric data on overall candidate system performance, characteristics, and operational features in sufficient depth to establish critical design problems and criteria to support a sound system design and evaluation.

1.2.1.1 Task A – Design Studies

Candidate surface-tension-type acquisition systems will be conceptually defined, relative to anticipated requirements for candidate applications and studied in detail. This will include not only the acquisition subsystem but also all other subsys s that interact with the acquisition device, such as the propellant storage, pressurization, and vent subsystems. This will be approached by establishing a workable design for a baseline system using the distributed channel acquisition concept; analyzing this system in detail with respect to failure modes, performance, design criteria, and areas of potential and significant improvement; and perturbing or evolving the baseline design in areas where these potential improvements exist and can technically be accomplished. This procedure may thus result in establishing several variations in a system design or several different system designs with individual or specialized characteristics that will ultimately be compared. Analysis and design models and/or procedures will be modified or developed as necessary to support this investigation. The study will include a failure mode analysis for the promising candidates.

1.2.1.2 Parametric Studies

Critical parametric data will be generated for each promising candidate to identify and define critical design factors and criteria for each concept. Design limits and performance parameters such as head retention capability and weight will be evaluated over a range of conditions so that the impact of variation in system design requirements can be assessed for each promising candidate concept.

1.2.2 Phase II – Design

The objective of this phase will be to use the theoretical models and parametric results generated in Phase I to arrive at (1) a selected acquisition concept and resulting preliminary design for a Shuttle-class cryogenic APS and for a representative ASPM cryogen feed system, (2) a test prototype design for a representative acquisition subsystem that will permit meaningful ground testing to verify the design concepts, and (3) a test plan to control the prototype testing to produce maximum usable results.

1.2.2.1 Task A – Preliminary Design/Comparison

Feed system preliminary designs will be produced, based on the candidate acquisition concepts and the general results from Phase I. These designs will be in sufficient detail to permit a valid performance comparison of the potential candidates. This task will be completed with the final selection of the recommended feed system design for a Shuttle-class cryogenic APS and an ASPM cryogen feed system. Selection criteria will stress the ability to satisfy flexible vehicle mission and duty cycle requirements and compatibility with a minimum-cost, high "probability of success" development program.

1.2.2.2 Task B – Bench Testing

Bench testing will be conducted relative to critical problems that must be resolved to realistically complete the preliminary designs. These tests will be conducted in parallel with the design activity.

1.2.2.3 Task C – Prototype Design

The objective of this task is to prepare a detailed design for a large-scale prototype acquisition system test apparatus, suitable to support a ground test program, that is compatible with the systems selected in Task A of Phase II. The prototype will be designed and instrumented to demonstrate the critical operational aspects of the systems and show that practical fabrication is possible. The current plan is to incorporate the acquisition hardware into the MSFC H_2/O_2 APS breadboard.

A test plan defining the installation and the tests to be conducted will be prepared as part of the design activity.

1.2.2.4 Task D – Reporting

Monthly and quarterly reports, and a final and an interim report will be submitted as defined by the program schedule. This effort will also include oral reviews and status reports.

1.2.3 <u>Phase III – Fabrication</u>

During this phase, component parts and subassemblies for the prototype design generated under Task C of Phase II will be fabricated and/or assembled. These will then be shipped to NASA-MSFC for subsequent installation into NASA tankage and feed system breadboard.

1.2.4 <u>Phase IV – Testing</u>

The objective of this task is to coordinate test operations at MSFC to verify the performance of the prototype system and to analyze and evaluate the test results.

1.2.4.1 Task A — Checkout and Ship

Leak tests will be conducted as necessary on the fabricated hardware. After final assembly, the completed test prototype device(s) will be sent to MSFC.

1.2.4.2 Task B — Test Operation

Engineering support will be provided at MSFC to direct and coordinate installation and performance evaluation testing of the prototype system as outlined in the developed test plan.

1.2.4.3 Task C — Analysis and Reporting

The test results will be analyzed to assess the demonstrated performance and characteristics of the prototype feed system and to compare them with anticipated behavior. These results will be documented in the final report, thus concluding the program.

Section 2
SUMMARY

During the sixth quarter of this program, effort concentrated exclusively on Phase II — Design, in accordance with the revised program plan shown in Figure 1. Emphasis during the quarter was on Task A2, ASPM Class Preliminary Design/Comparison and on Task C, Prototype Design. Task A is now essentially completed and as the seventh quarter is initiated only Task C of Phase II remains to be completed.

With respect to the ASPM preliminarv design/comparison effort, the following was achieved.

A. An in-orbit thermal management study was completed and it was concluded that the optimum propellant thermal management system consists of a vented LH_2 tank and an LO_2 tank effectively cooled by the GH_2 vent gases. LH_2 tank venting is accomplished by a cooled shroud thermodynamic vent system (TVS) that also serves to thermally guard the acquisition device. A radiation shroud, conveying GH_2 vent gases, surrounds the LO_2 tank to provide LO_2 tank cooling, or more specifically, heat interception.

B. An in-atmosphere propellant thermal control study was completed and it was concluded that the best system employs a minimum thickness foam substrate with the space-optimized MLI lay-up purged with GN_2 on the ground and helium for the initial reentry phases.

C. Detailed pressurization system calculations were made for the baseline ASPM case.

D. A spectrum of possible ASPM missions was identified and their impact on the start-tank acquisition system sizing was assessed. It was found that all realistic missions and even increases in thrust by a factor of 2 could be handled by the basic start tank design previously evolved and presented in Reference 1.

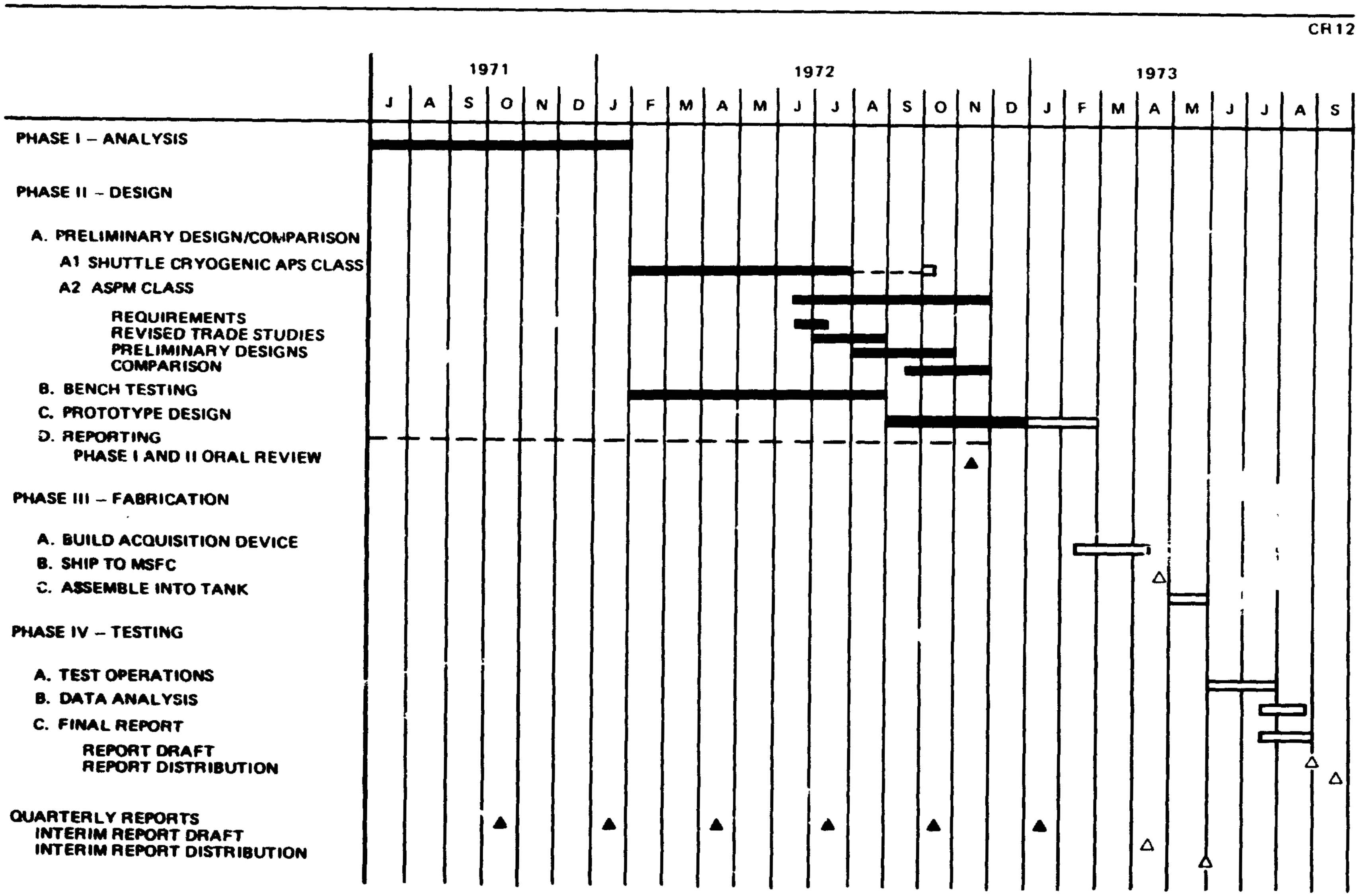

Figure 1. Program Plan (Revised January 1 1973)

E. Detailed studies of the start-tank acquisition system were conducted and the preliminary design to satisfy ASPM requirements was essentially completed. This involves a 0.95 m^3 (33.6 ft^3) start tank integrated into the main tank dome. Access is provided through a 0.97 m (32 in.) manhole. The acquisition screens are thermally protected by using: (1) cold helium pressurization in the start tank, (2) a cooled main tank-wall shroud to restrict wall heating, (3) cooled feed lines, and (4) optional foam insulation on the start-tank common dome.

Under Task C, conceptual prototype designs were evolved. These included a screen ring-type device representative of that used in the Shuttle APS start tank and a baffled start tank with removable cylindrical screen elements representative of the ASPM recommended design. Detailed designs for these devices are now underway.

Details of this work are reported in Sectio[illegible]

Section 3
TECHNICAL DISCUSSION

During the sixth quarter of the program, work proceeded on Phase II - Design according to the revised program plan. Effort concentrated on completing the preliminary design of the ASPM class acquisition system, (Task A2) and in initiating the design of the prototype acquisition hardware (Task C).

3.1 PHASE II, TASK 2A, PRELIMINARY DESIGN/COMPARISON ASPM APPLICATION

The general design features of the acquisition system to satisfy baseline ASPM requirements are shown in Figure 23 of the last progress report (Reference 1). Detailed studies have been conducted relative to this concept during the sixth quarter.

3.1.1 Propellant Thermal Management Study

A detailed analysis of the basic in-orbit propellant thermal management concepts for the ASPM has been completed. Vehicle performance values in terms of stage gross weight were computed, using the MDAC multi-start space propulsion system sizing and optimization program (H109), for three options: (1) both nonvented LH_2 and LO_2 tanks, (2) both vented (to 1 `2 x 13^3 N/m^2 (14.7 psi)) LH_2 and LO_2 tanks, and (3) a vented LH_2 tank and a nonvented LO_2 tank cooled by using GH_2 vent gases to intercept incoming heat. All calculations were made for the baseline duty cycle (Table 21, Reference 1) and are summarized in Figure 2. (Figure 2 is an updated version of Figure 17 from Reference 1.) In the case of vented or cooled tanks, vapor-cooled shields were assumed and for locked-up nonvented tanks, internal mixers that would prevent temperature stratification were assumed.

Figure 2 shows that, for the ASPM application, a significant reduction in gross weight results by using vented and/or cooled tanks rather than a nonvented system. The optimum amount of insulation also shifts significantly

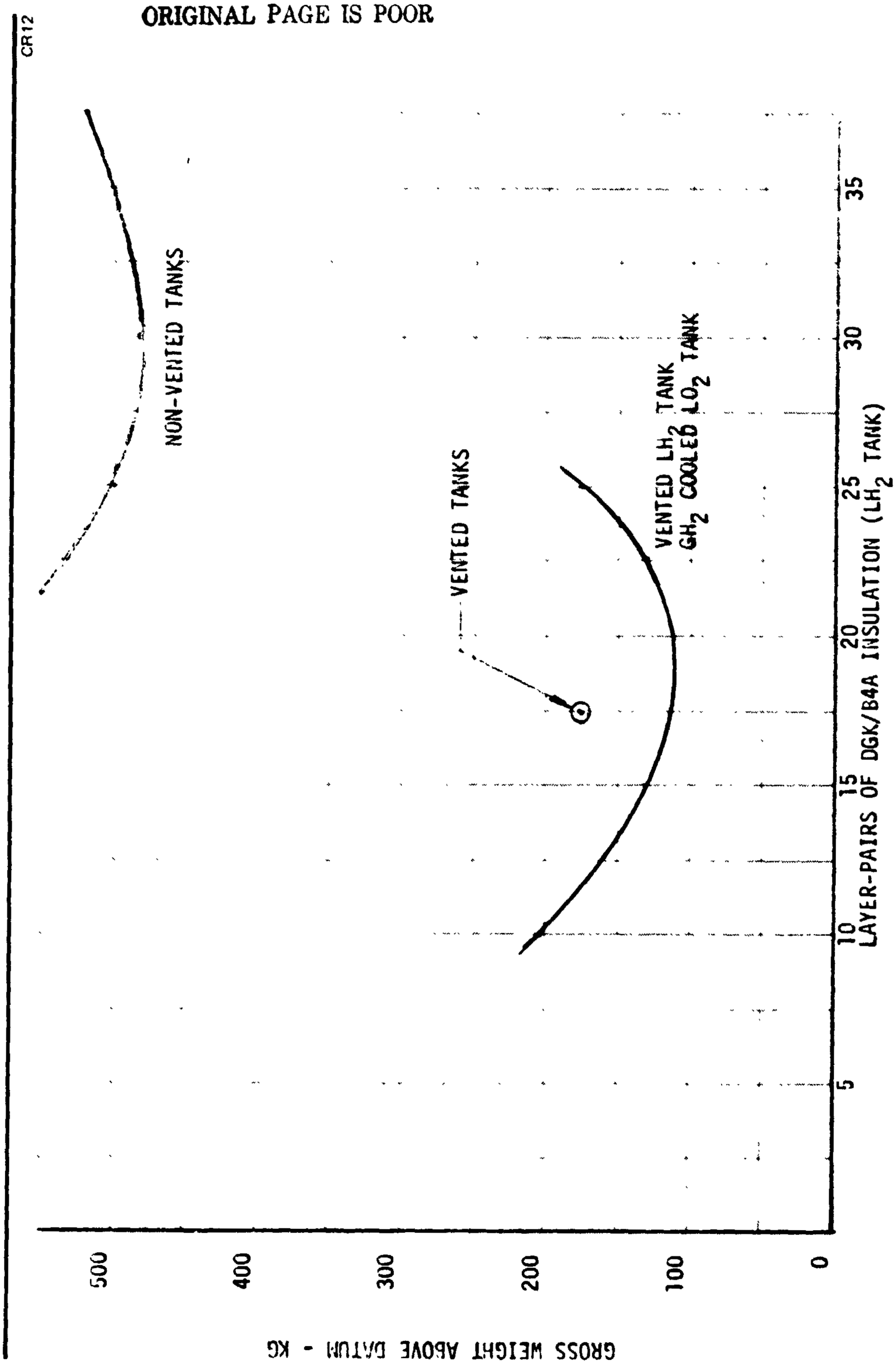

Figure 2. Effects of Insulation and Propellant Management Concept on ASPM Gross Weight

for the various thermal management approaches. For nonvented tanks, the optimum LH_2 insulation consists of 30 pairs of DGK/B4A MLI and results in a gross weight above datum of 480 kg. With a vented LH_2 tank and a cooled LO_2 tank, the optimum LH_2 tank insulation is 18 layer-pairs of MLI and the gross weight above datum is 130 kg. Venting of the LO_2 tank rather than cooling results in a gross weight increase to 260 kg. Table 1, which lists the weights of the thermal management concept affected items, provides insight into how this gross-weight difference comes about. As shown in the table, the vented system results in a lighter burnout weight (by about 76 kg) but requires venting 80 kg of LH_2. However, the burnout weight has a stronger influence on stage performance (produced velocity change) than the boiloff loss which does not have to be accelerated for the full stage ΔV. Thus, less propellant is required to produce the needed ΔV with the vented tank approach. The 350 kg gross weight change is primarily a difference in required propellant mass. This strong impact of small inert weight changes on gross weight and the distinction between true inert and consummable weight, illustrates the need for careful overall system weight comparisons. This type of performance sensitivity, which is common in high-energy propulsive stages, was not evident in the Shuttle APS application because the APS weights had very little effect on the weight of the relatively massive Shuttle orbiter vehicle.

Based on the above results, the selected propellant thermal management concept consists of a vented LH_2 tank and a LO_2 tank cooled by the vent gases from the LH_2 tank. A cooled shroud or an internal tank mixer/heat exchanger TVS could be used to effect the LH_2 venting. The cooled shroud would be about 20 kg heavier than the mixer case but is desirable because of its passive nature, g independence, and continuous venting provides a convenient coolant supply. As noted in subsection 3.1.4, the cooled shroud also serves to provide the thermal protection essential for the acquisition screen device. Therefore, the cooled shroud TVS was selected as the best choice for this application.

3.1.2 In-Atmosphere Propellant Thermal Protection

The study discussed in the preceding section dealt only with in-orbit propellant storage. Provisions are also required to limit propellant losses during ground hold and launch, and reentry and landing under certain conditions. This was

Table 1

THERMAL MANAGEMENT CONCEPT WEIGHT (kg) COMPARISON – ASPM (LH_2 TANK)

	Vented (kg)	Nonvented (kg)
Purge Bag	28.4	28.4
Face Sheets	27.6	27.6
MLI Layers (1)	20.6	34.4
Attachments	3.0	3.0
Tank Basic Structure	173.0	222.0
Cooling Shroud	25.5	0.0
Internal Mixers	0.0	5.0
Final Ullage Gases (GH_2) (3)	78.0	112.0
Tank System Burnout Weight	356.1	432.4
In-Orbit Boiloff	80.0(2)	0.0
Total	436.1	432.4

[1] 18 and 30 layers for vented and nonvented, respectively.

[2] 117-hr coast for baseline mission.

[3] Tank Pressure = 147×10^3 N/m^2 (21.3 psia) vented.
= 200×10^3 N/m^2 (29 psia) nonvented.

investigated for the Space Shuttle APS application (see References 2 and 3) and this work has been extended to this application.

Figure 3 shows the heat load associated with each operational regime as a function of MLI thickness, assuming a simple helium-purged insulation concept. These values agree with the data presented in Figure 3 of Reference 2, but have been extended to smaller MLI layups by using the MDAC transient thermal analysis program.

The weight penalty for in-atmosphere propellant thermal storage for a simple helium-purged MLI system concept is shown in Figure 4 for three operational requirements. The weight of added MLI and the summation of the added MLI and the resulting hydrogen boiloff are shown. These results indicate that increasing MLI thickness does not result in a weight saving

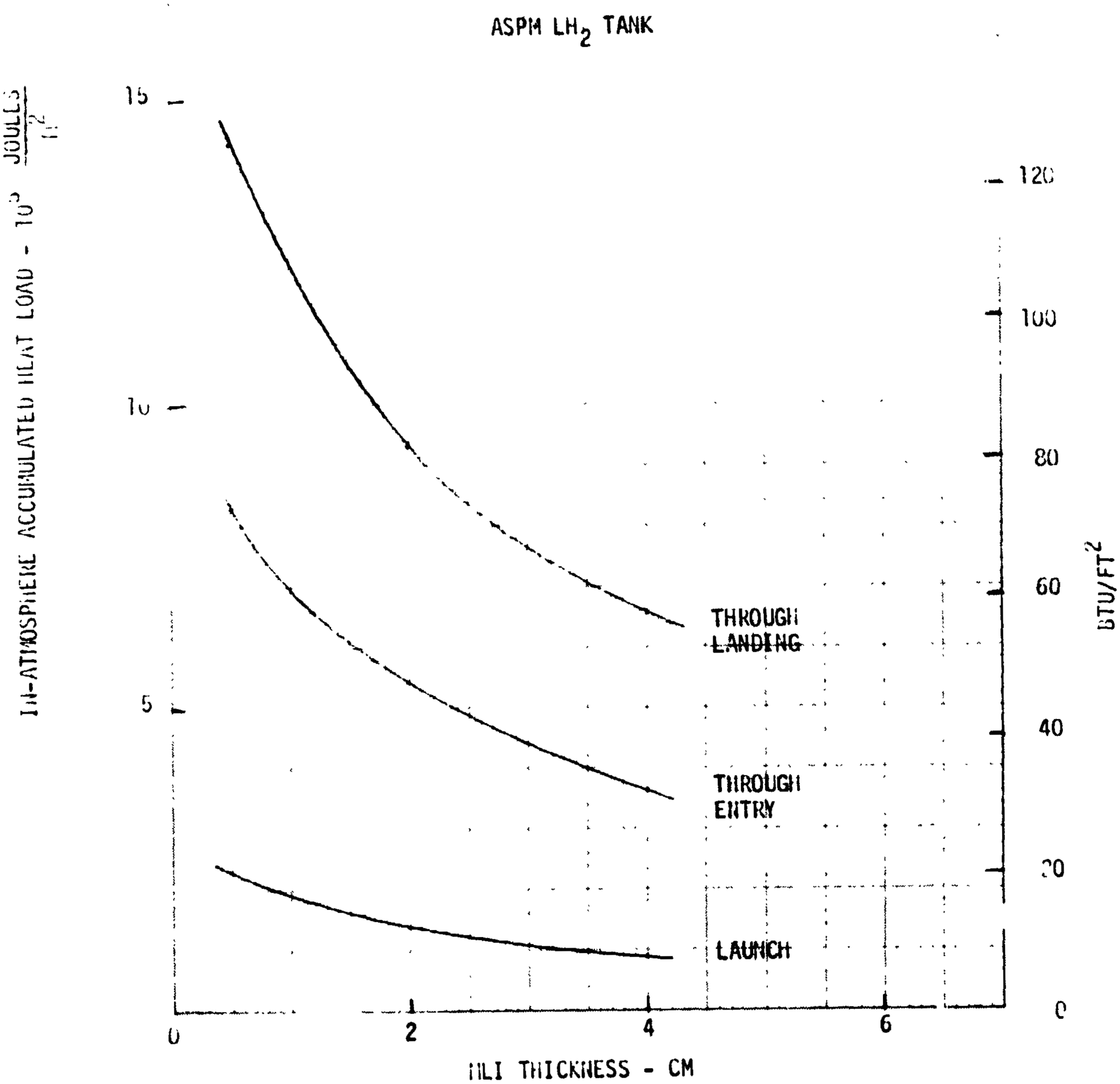

Figure 3. In-Atmosphere Accumulated Heat Load (Helium Purged MLI Concept)

except possibly for the case where LH_2 must be stored completely through landing. For the ASPM, storage through landing is a requirement only for abort operation where thermal performance is not critical. Therefore, one candidate approach to providing in-atmosphere storage capability would be to purge the MLI insulation, sized for orbital storage, with helium. This would result in the weight penalties indicated in Figure 4 at "zero" increased MLI thickness.

Preliminary calculations have been made to substantiate that condensation of the GN_2 purge used in the Space Shuttle payload bay during ground hold will not be a significant problem, even with helium purged MLI layups as low as 18 layer-pairs.

An alternate concept was analyzed and compared with simple helium purging. This alternate concept involves using a minimum layer of external foam between the tank wall and the MLI with the MLI being purged with GN_2 rather than helium during ground hold. A foam thickness of 0.4 cm was used, which results in a MLI/foam insulation temperature well above the nitrogen and/or air condensation points. This foam insulation is discussed in Reference 2 and would weigh 17.8 kg as installed on the entire ASPM LH_2 tank. The corresponding LH_2 boiloff for ground hold and boost operation is 12.3 kg resulting in a total weight penalty of 30.1 kg. Table 2 shows the weight penalties for

Table 2

COMPARISON OF SIMPLE HELIUM-PURGED MLI AND A GN_2-PURGED MLI/FOAM SUBSTRATE

Operation	Weight Penalty* (kg) Foam Substrate	Weight Penalty* (kg) Simple Helium Purge
Launch Only	30.1	35
Complete Mission Through Entry	84	130
Complete Mission Through Landing	138	225

* Foam insulation weight + in-atmosphere boiloff weight

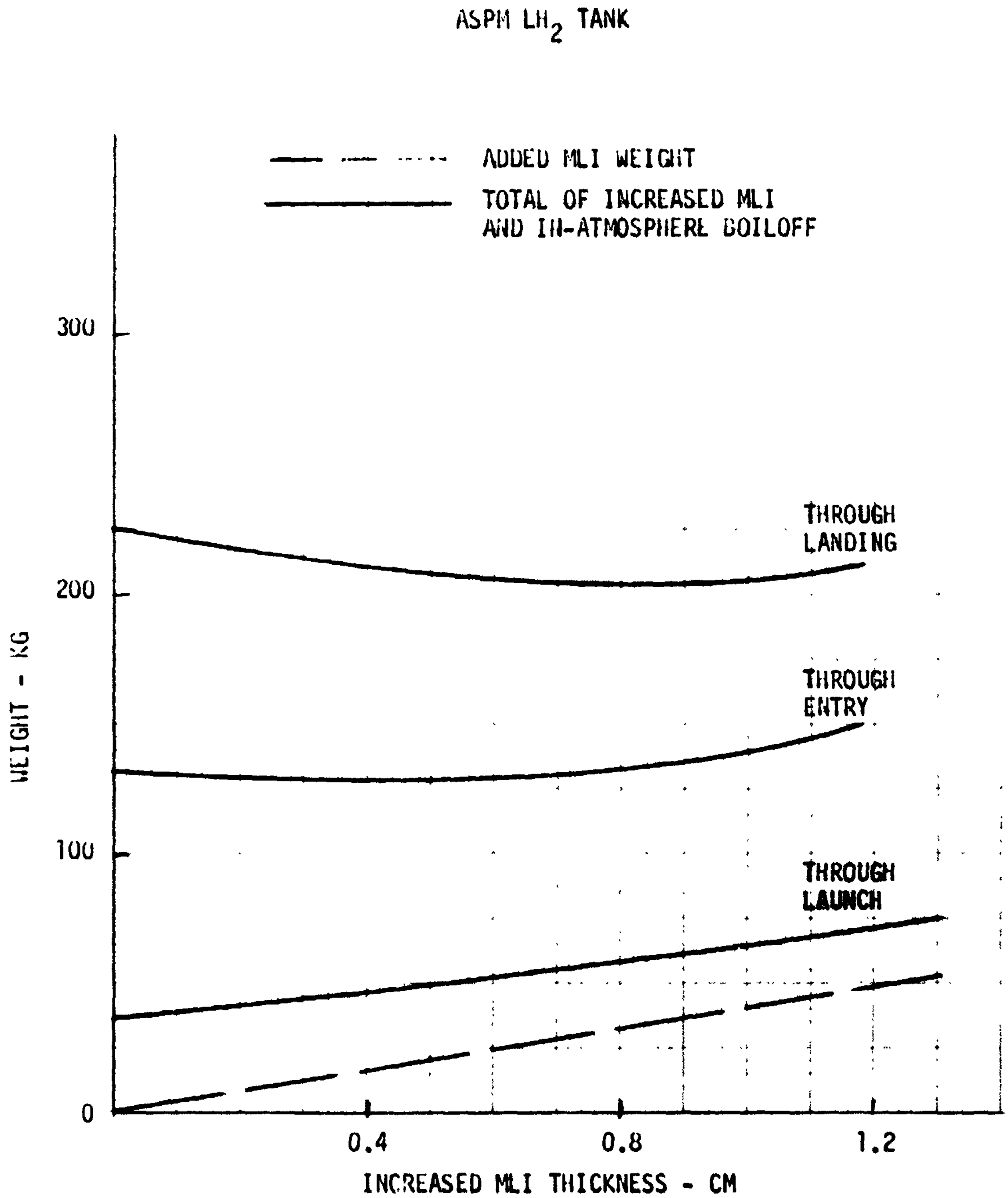

Figure 4. Weight Penalty — In-Atmosphere Propellant Storage (Simple Helium Purged MLI Concept)

all operational regimes although the reentry storage requirement does not require optimum storage since it is needed only during an abort situation. As can be seen, the foam substrate always results in some weight savings. The foam system also permits a simpler GN_2 ground purge system.

3.1.3 Pressurization System

As discussed in subsection 3.1.4, a basic refillable pressure-isolated start-tank acquisition system has been selected for the ASPM application. A probable LH_2 start tank size of 0.95 m^3 (33.6 ft^3) was also defined. With this size and the baseline duty cycle, the overall LH_2 tank pressurization system was analyzed. This included the pressurization of the start tank with cold gaseous helium as well as pressurization of the main tank with 200°R GH_2. The various weight elements are summarized in Table 3. Main-tank penalties include the effects of increased tank volume to accommodate the pressurants and increase in tank-wall thickness over minimum gage to withstand maximum tank pressure, 147 x 10^3 N/m^2 (22.3 psia). A 0.52 m (1.69 ft) diameter helium bottle will be required for the LH_2 side.

The LO_2 tank pressurization system was also analyzed assuming an LO_2 start tank volume of 0.244 m^3 (8.6 ft^3) (based on values presented in subsection 3.1.4). Main LO_2 tank pressurization calculations are summarized in Table 4 for a range of helium inlet temperatures. This table shows that pressurant weights are only slightly affected by the inlet temperature for the baseline ASPM duty cycle. However, the maximum tank pressure at low inlet temperatures does tend to increase over the minimum gage pressure level of 158 x 10^3 N/m^2 (23 psi). Therefore, it would be desirable to use an inlet temperature of about 222°K (400°R) that could probably be provided by a simple passive structural heat sink-type heat exchanger. The resulting weights are summarized in Table 5. If the LO_2 tank helium were stored in a separate high pressure bottle within the LH_2 tank, a volume of 0.43 m^3 (1.53 ft^3) with a diameter of 0.44 m (1.44 ft) would be needed. In practice, the LH_2 and LO_2 tank helium supplies would be incorporated into a single high-pressure bottle. This would be 0.111 m^3 (3.93 ft^3) in volume with a diameter of about 0.61 m (2 ft) which is quite reasonable. This combined bottle would weigh about 22.1 kg including support provisions.

Table 3

ASPM LH_2 TANK PRESSURIZATION SYSTEM WEIGHT ESTIMATES
0.95 m^3 START TANK VOLUME,
34.5 x 10^3 N/m^2 TRUE NPSP CONTROL

		(kg)
Start Tank Usable Helium		8.4
Helium Bottle and Supports		13.6
Main Tank GH_2		76.0
Main Tank Penalties		14.0
	TOTAL	112.0

Table 4

INFLUENCE OF INLET TEMPERATURE
ON ASPM LO_2 TANK PRESSURIZATION

(0.244 m^3 START TANK VOLUME)
(34.5 x 10^3 N/m^2 (5 psi) TRUE NPSP CONTROL)

Inlet Temperature (^oK)	Ullage Mass - (kg)	Helium Mass - (kg)	Maximum Tank Pressure $10^3N/m^2$ (psia)
111	73	6.1	210 (30.4)
222	74	4.9	166 (24)
333	76	4.4	153 (22)

Table 5

ASPM LO_2 TANK PRESSURIZATION SYSTEM WEIGHT ESTIMATES
(0.244 m^3 START TANK VOLUME, 34.5 X 10^3 N/m^2 TRUE NPSP CONTROL,
222^{o}K INLET TEMPERATURE)

	(kg)
Start Tank Usable Helium	0.4
Main Tank Helium	4.9
Helium Bottle and Supports	8.5
Main LO_2 Tank GO_2	69.1
Main Tank Penalities	0.4
	8. 3

A GO_2 pressurization system was also analyzed. In this case the optimum inlet temperature was about 333°K (600°F) and the total system weight was estimated at 116.4 kg. This is about 33 kg heavier than the helium system.

3.1.4 Acquisition Subsystem

In Reference 1, subsection 3.2.4, the relative merits of distributed and localized screen acquisition devices are discussed for the ASPM missions and it was concluded that only localized devices could meet the requirements of high acceleration, high flow rate, and variable propellant off-loading volume. In view of the significant weight savings achieved with the start-tank design in the cryogenic shuttle APS case, this concept was considered for the ASPM application. The resulting preliminary ASPM start-tank design met the specified mission requirements, provided high retention safety factors (>2) for all mission environments, and was shown to have no technological uncertainties. Because of the high accelerations in the ASPM (>1g), the worst-case condition can be tested and all operational aspects (refill, expulsion, vent, etc.) confirmed by ground testing the ASPM prototype in the NASA-MSFC APS breadboard (see subsection 2.2.3, Reference 1).

The preliminary design was based on a baseline mission as specified in Reference 1. Additional missions have been analyzed, and as a result the ASPM start-tank design has been refined. The following subsections document the ASPM start-tank preliminary design, including system weights.

Mission Requirements

An ASPM start-tank design compatible with the baseline mission (Table 21, Reference 1) was presented in Reference 1 (Figure 23). Through consultation with personnel in the MDAC Space Tug/OOS system design group, three other missions were identified and critical information was then generated for each of these. The new missions included a deployment, a retrieval, and an interplanetary launch. These complemented the baseline deployment/retrieval roundtrip mission. Pertinent data for these mission duty cycles are summarized in Tables 6, 7, and 8. These missions are similar to one another and to the baseline in their general sequence of events and arrangement and magnitudes of the various burns. All three new mission time periods are

Table 6

BURN SEQUENCE (DIRECT DEPLOYMENT MISSION)
(PAYLOAD WEIGHT = 4, !00 kg)

Activity	Time (Hr From Launch)	ΔV (Thruster)* (m/sec)	Propellant Consumed (kg)	Acceleration Level (g)
Separate from Shuttle	5	3 (RCS)	36	0.0009
Phasing Initiation	6.1	3 (RCS)	30	0.001
Plane Change	13.5	337 (PP)	1,940	0.157 - 0.168
Apogee Burn No. 1	15.3	2,190 (PP) (RCS)	10,100 20	0.168 - 0.27
Apogee Burn No. 2	20.7	1,795 (PP) (RCS)	5,330 20	0.27 - 0.395
Rendezvous	20.9	21 (RCS)	40	0.0024
Deployment	26.8	3 (RCS)	10	0.0024
Transfer	35.6	1,790 (PP) (RCS)	2,360 20	0.6 - 0.9
Lower Apogee Burn	40.9	1,615 (PP) (RCS)	1,500 10	0.9 - 1.28
Mid Course	43.1	11 (RCS)	15	0.005
Circularization	43.2	880 (PP)	617	1.28 - 1.55
Terminal Rendezvous	44.2	22 (RCS)	30	0.006

* (RCS) Reaction Control System
(PP) Primary Propulsion
(TPP) Throttle (20 percent) Primary Propulsion

Table 7

BURN SEQUENCE (DIRECT RETRIEVAL MISSION)
(PAYLOAD WEIGHT = 2,050 kg)

Activity	Time (Hr From Launch)	ΔV (Thruster)* (m/sec)	Propellant Consumed (kg)	Acceleration Level (g)
Separate from Shuttle	5	3 (RCS)	36	0.0001
Phasing Initiation	6.1	3 (RCS)	30	0.0001
Plane Change	13.5	337 (PP)	1,820	0.166 - 0.178
Apogee Burn No. 1	15.3	2,190 (PP) (RCS)	9,650 20	0.178 - 0.283
Apogee Burn No. 2	20.7	1,795 (PP) (RCS)	5,140 20	0.283 - 0.415
Rendezvous	20.9	21 (RCS)	40	0.0053
Docking and Pickup	26.8	3 (RCS)	10	0.0031
Plane Change	35.6	1,790 (PP) (RCS)	4,130 20	0.35 - 0.51
Lower Apogee Burn	40.9	1,615 (PP) (RCS)	2,630 10	0.51 - 0.73
Mid Course	43.1	11 (RCS)	15	0.0044
Circularization	43.2	380 (PP)	1,090	0.73 - 0.89
Terminal Rendezvous	44.2	22 (RCS)	30	0.0053

* (RCS) Reaction Control System
(PP) Primary Propulsion
(TPP) Throttle (20 percent) Primary Propulsion

Table 8

BURN SEQUENCE (INTERPLANETARY VENUS MISSION)
(PAYLOAD WEIGHT = 1,060 kg)

Activity	Time (Hr From Launch)	ΔV (Thruster)* (m/sec)	Propellant Consumed (kg)	Acceleration Level (g)
Separate from Shuttle	5	3 (RCS)	36	0.0009
Perigee Burn No. 1	8	2,100 (PP) (RCS)	14,050 20	0.16 - 0.316
Apogee Burn No. 2	12	2,100 (PP) (RCS)	4,560 15	0.31 - 0.505
Payload Deployment	12.2	3 (RCS)	10	0.003
Rotate Stage	12.3	1 (RCS)	5	0.003
Retro-Burn	12.4	1,800 (PP) (RCS)	2,550 10	0.58 - 0.856
Deorbit	60	1,610 (PP) (RCS)	1,875 10	0.86 - 1.22
EOS Base Rendezvous	61	1,340 (PP) (RCS)	956 10	1.22 - 1.64
Docking	62	22 (RCS)	30	0.01

* (RCS) Reaction Control System
(PP) Primary Propulsion
(TPP) Throttle (20 percent) Primary Propulsion

shorter than the baseline period. Because of the manner in which payload is handled, the maximum acceleration varies for the different missions, going as high as 1.64 g in the case of the interplanetary mission.

These duty cycles were used in conjunction with the MDAC start-tank sizing program to generate start-tank size and other design values for each mission. The size results are summarized in Table 9. The first three sets of values for start-tank sizes apply to the baseline roundtrip deployment/retrieval mission. The first set of values is that directly generated by the computer program and is essentially based on volume usage demands with no influence of screen retention limits. The second set of sizing numbers shows the adjusted baseline values with the start-tank size increased to prevent the liquid level from dropping to the point where screen retention breakdown might be possible during the burn sequence. The third set of values is the baseline size adjusted so that a fixed settling time can be used which would permit the use of a simple fixed start-up control logic. The resulting volumes are 0.952 m^3 (33.6 ft^3) for LH_2 and 1244 m^3 (8.6 ft^3) for LO_2. The last three sets of sizing numbers, which apply to the three new duty cycles, are the volumes directly computed from the sizing program. The results from Table 9 and supporting retention-head analyses show that the baseline roundtrip mission results in the largest volume start-tank, and its size is therefore compatible for the other missions considered.

Table 9

ASPM START-TANK SIZES

	Start-Tank Volume m^3 (ft^3)	
	Fuel	Oxidizer
1. Baseline (Round Trip)	0.518 (18.3)	0.119 (4.2)
2. Baseline (Adjusted Size)	0.762 (26.9)	0.173 (6.1)
3. Baseline (Fixed Settling Time)	0.952 (33.6)	0.244 (8.6)
4. Direct Deployment	0.504 (17.8)	0.105 (3.7)
5. Direct Retrieval	0.49 (17.3)	0.105 (3.7)
6. Interplanetary	0.241 (8.5)	0.065 (2.3)

ASPM Start-Tank Configuration and Operation

The operation of the ASPM start tank is discussed in subsection 3.2.4 of Reference 1 for the preliminary design. This concept has been modified to meet the requirement of the additional mission, but the basic configuration and operation are the same as discussed previously.

The ASPM LH_2 start-tank design, as shown in Figure 5, is divided into three regions: a top region, a primary trap region, and a secondary trap region. The screen tubes in the top region communicate propellant to the primary region under all propellant configurations encountered in the low-g or high-g level conditions. The primary screen tubes are designed to retain liquid under the worst-case conditions which correspond to a high positive g level (e.g., 1.6 g) under maximum propellant outflow.

Since it is assumed that the 134 N (30 lb) thrust RCS maneuvers result in the acceleration being applied in an arbitrary direction, the effects of lateral accelerations are also considered in the design. However, even with the relatively large effective height (i.e., 1 meter) associated with lateral acceleration, the hydrostatic band imposed across the set of channels is much smaller than that imposed along the vehicle axis during the primary propulsion system burns.

Consideration of Figure 5 reveals that all of the screen devices can be installed by one man with access through the manhole. The screen tubes in the top and primary trap regions are placed in the tank and bolted into place. A compatible seal, such as Indium tin, would be used. The flat screen element, perhaps temporarily supported by flat stock, is then placed into position and bolted into place, and its temporary support removed. The manhole cover, containing the four additional screen tubes connected directly to the sump, is then brought into position and sealed. The feed line can then be joined to the tank with a Marman flange.

A typical start-tank pressurization system is shown in Figure 5 which consists of a helium bottle, a 500-psi regulator, two solenoid valves, a high flow rate orifice and a low flow rate orifice (e.g., VISCO jets). Two separate pressurization rates are used because of the low RCS flow rates and the high flow rates associated with the main engine operation.

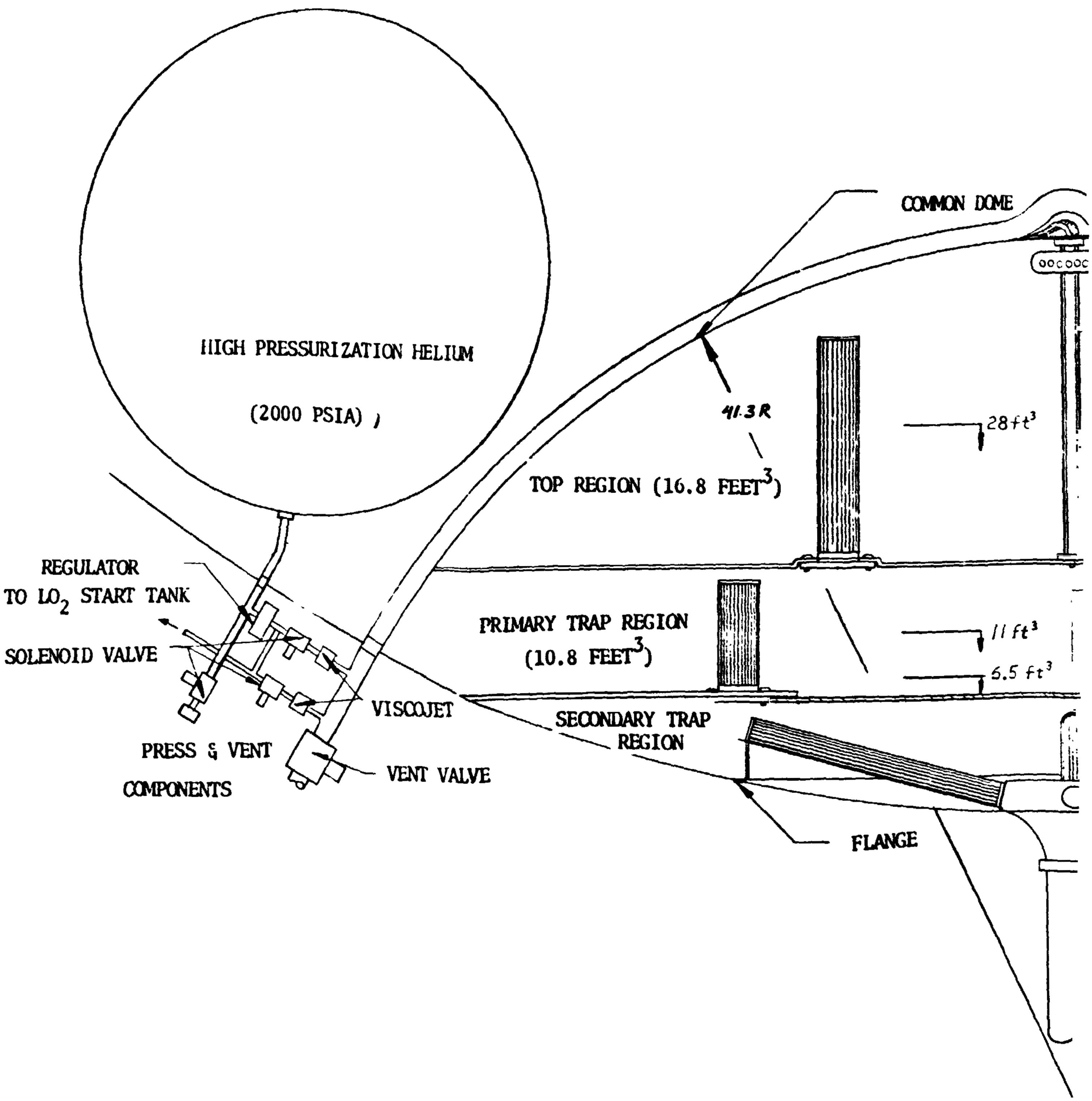

Figure 5. ASPM LH_2 Start Tank Acquisition System Preliminary Design

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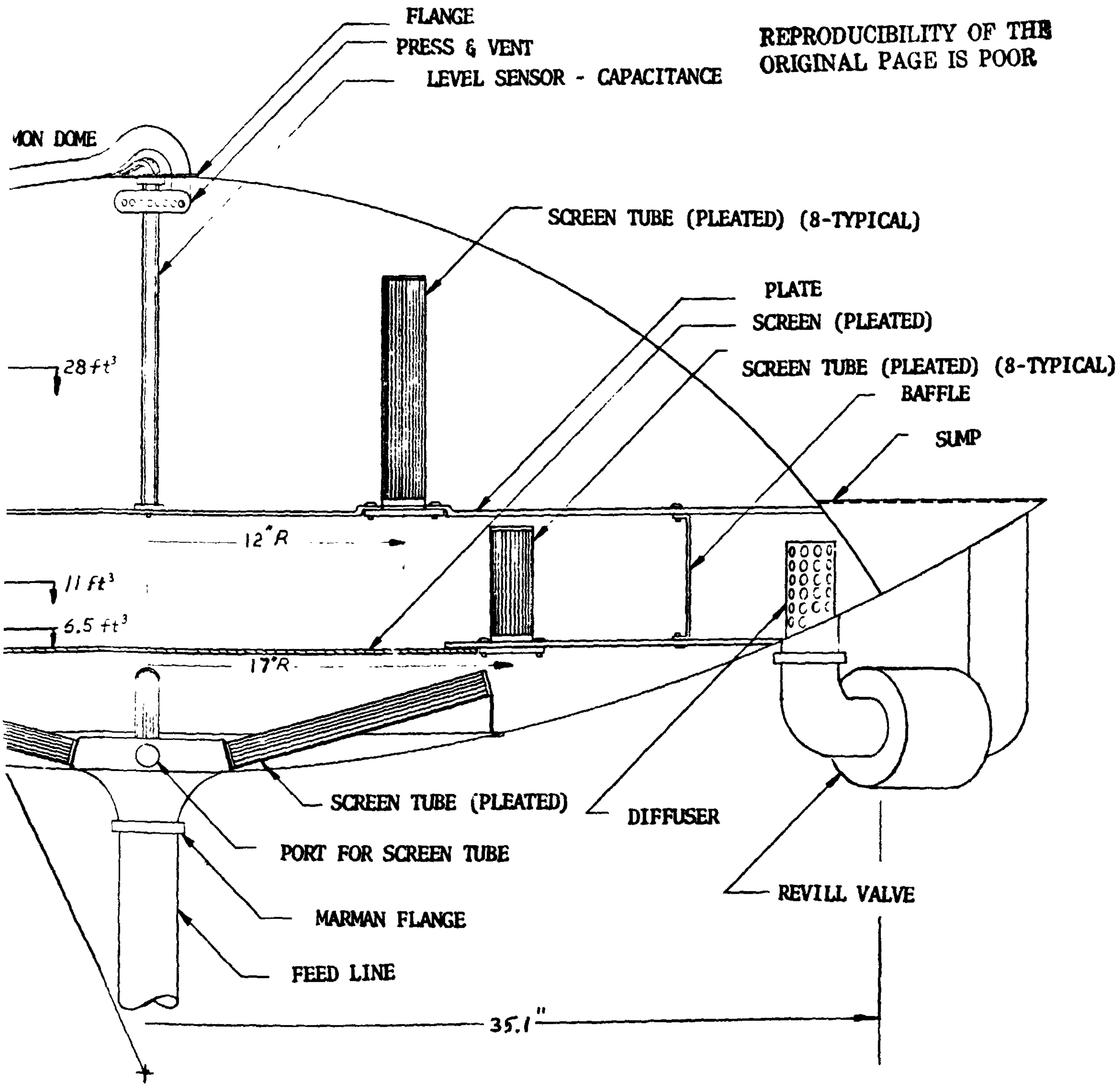

FOLDOUT FRAME 2

A capacitance probe liquid-level sensor is included as a backup determination of the propellant quantity. The probe would be used in conjunction with a timed circuit to control the start-tank refill. Details of the control logic have not been formalized, but this system is similar to that fabricated for the Interface Demonstration Unit (NAS8-27571), which is scheduled for testing within the next quarter.

Screen Acquisition Design

With the conservative LH_2 start-tank volume associated with a fixed settling time, the appropriate sizing and placement of the screen tubes was determined, which meets the requirements of a retention safety factor (RSF) of at least 2.0. The start-tank configuration has also been modified from that reported in Reference 1, to minimize the dome weight. This configuration change affects the placement, length, and mesh size of the screen tubes. Figure 6 shows the generalized geometrical relationships for the start-tank design based on a start-tank volume of 0.95 m^3 (33.6 ft^3). A preliminary study was conducted to determine the start-tank structural-weight penalty including the common dome (isogrid) weight and the main-tank dome weight increase to accommodate load distribution at the bulkheadjuncture. This is shown in Figure 7, which illustrates that the weight penalty decreases as the total start-tank height increases. Thus, it is desirable to use a smaller radius on the common dome than on the main-tank dome (maintaining constant volume). However, going beyond a total height of 0.7 m results in only a minor weight savings. (This corresponds to a start-tank diameter of 1.8 m.) Also, increasing height either limits retention-head capability or requires a finer screen mesh. The change from 0.56 to 0.7 m necessitates only a slight change in mesh size and does not have a significant impact on the acquisition device design or its overall capabilities.

The baseline system was originally designed to use relatively coarse mesh screens (200 x 600 or 165 x 800). Such screens were found to be adequate for the accelerations encountered in the baseline mission with a main-engine thrust of 44,500 Newtons (10,000 lb) (see Reference 1). However, thrust levels of 66,700 Newtons (15,000 lb) or even 89,000 Newtons (20,000 lb) have been discussed in independent studies (Reference 4). Increases in acceleration level above those previously reported have resulted with the additional missions

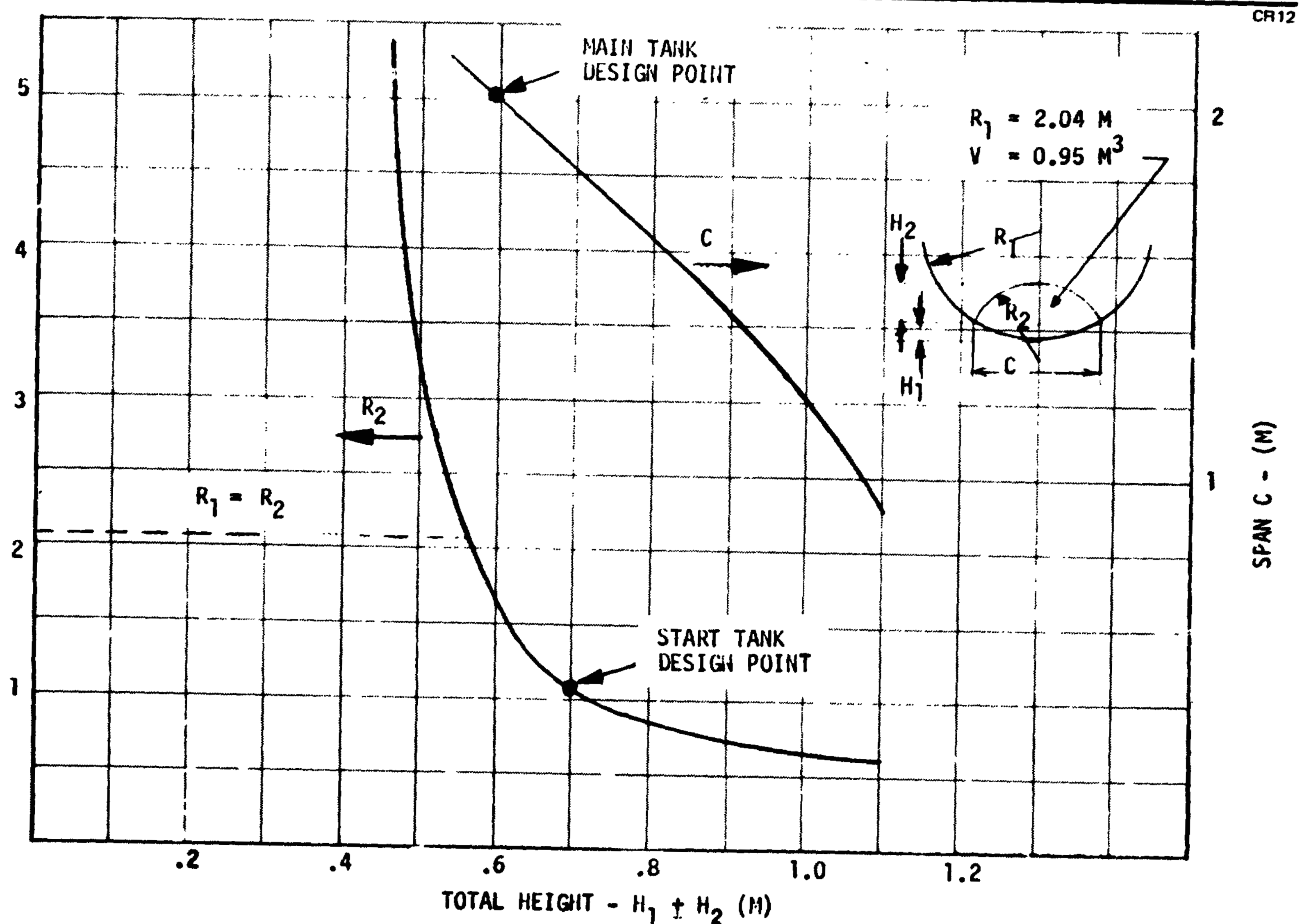

Figure 6. Start Tank Geometr[illegible] l Relationships (ASPM LH_2 Tank)

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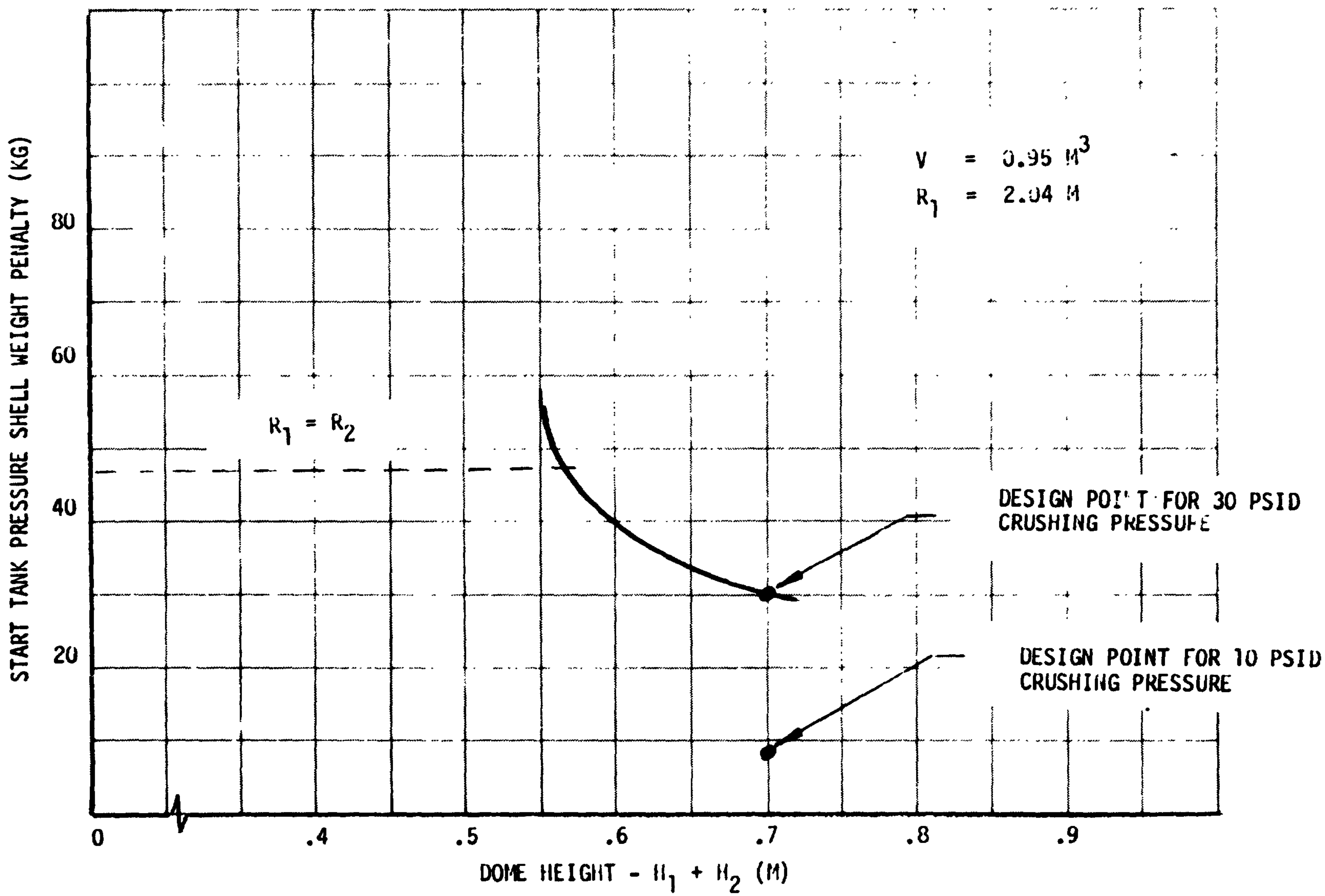

Figure 7. ASPM Start Tank Structural Weight Penalty

considered. The maximum acceleration is 1.64 g, assuming a 44,500 Newtons (10,000 lb) thrust engine for the Interplanetary Venus Mission. The start-tank size and configuration selected above is compatible with all of these missions.

The appropriate screen mesh has been increased to 200 x 1,400, instead of 200 x 600 or 165 x 800, to meet the new mission requirements with the higher accelerations (1.64 g).

Considering the overall design shown in Figure 5, the primary screen tube and flat screen between the primary and secondary trap regions is seen to be the critical region for flow-loss calculations. The flow losses associated with the primary trap region are composed of hydrostatic head and pressure drop through the screen, with the propellant flowing through the screen at 1.64 g. The LH_2 volume flow rate is 0.0213 m^3/sec (0.75 ft^3/sec). The flat screen is assumed to be pleated with a factor of three in area; the screen diameter is 0.71 m (28 in.). Eight pleated screen tubes with a pleating factor of three, two inches in diameter and five inches long, are used. The hydrostatic head is based on a submerged depth of 6.35 cm (2.5 in.), which is conservative compared to the operating conditions of the start-tank. The pressure drop due to flow loss through the screen tubes and the flat screen is 46.7 N/m^2 (0.975 lb/ft^2). The hydrostatic head is 93.1 N/m^2 (0.95 lb/ft^2). The total pressure loss is 140 N/m^2 (2.925 lb/ft^2). With 200 x 1,400 mesh screens, having a bubble point of 335 N/m^2 (7.0 lb/ft^2), the retention safety factor is 2.4. This design is flexible in several ways in terms of providing increased capability. The pressure drop through the screen can be decreased by increasing the pleating factor to 4, increasing the number of screen tubes, and increasing the flow area of the flat screen. An increased hydrostatic head requirement due to an increase in the vehicle acceleration from 1.6 to 3.2 g could be met by use of a finer mesh screen. Mission requirements leading to the need for increases in the storage capacity of the start-tank can also be met without significant changes in the overall configuration. The communication screen tubes in the top region pose no problems.

Based on the hydrostatic head associated with the low-g levels of the RCS system, and the negligibly small pressure drops due to flow, very small mesh count screens (30 x 30) would be workable. However, extraneous impact acceleration may be present as a result of docking, etc., which implies the need for finer mesh screens. One such impact acceleration has been estimated based on a typical ASPM docking mechanism.

The Space Tug docking mechanism described in subsection 4.2.2.5 of the NASA-MSFC Baseline TUG Definition Document, Rev A, June 26, 1971, consists of a square frame supported by eight pneumatic/hydraulic shock absorbers/actuators. During docking, the frame moves from its deployed position to its retracted position, a distance of 1.18 m (46.3 in.), absorbing the docking impact energy. The maximum approach velocity prior to docking is 0.305 m/sec (1.0 ft/sec). Thus, during docking the acceleration on the vehicle could be of the order of 0.004 g. This allows for nonlinear deceleration during docking, and additional effects of misalignment and lateral motion which would increase the maximum acceleration imposed on the screen devices. Therefore, even though a 30 x 30 mesh screen is adequate to maintain retention during the 0.005-g acceleration, a finer mesh screen is recommended to avoid any chance of breakdown due to docking. There are no adverse affects in using a finer mesh screen, such as 50 x 250 or even 165 x 800, and the retention capability is increased by a factor of 10 or more over the 30 x 30. Thus, a screen such as the 50 x 250 or 165 x 800 will be used.

ASPM LH_2 Start-Tank Weight

The LH_2 start-tank weight has been determined, including components, screens, the common dome, and an access manhole.

The LH_2 start-tank dome weight has been estimated, assuming a spherical isogrid start-tank shell welded to the main-tank dome. The weight of the isogrid has been determined, based on the curves of Reference 5 for a crushing pressure of 69×10^3 and 207×10^3 N/m^2 (10 and 30 psid). The welded joint weight has also been estimated for these two cases; results are shown in Table 10. To use the lower weights associated with the lower pressure, a relief valve must be used to assure that the start-tank pressure does not fall more than 69×10^3 N/m^2 (10 psi) below the main-tank pressure. Under normal operating conditions, the start-tank pressure would be less than 13.8×10^3 N/m^2 (2 psi) below main-tank pressure. Since the vacuum vent/refill mode of refill is not used for this design, a high crushing pressure differential would occur only in the event of a structural failure or serious malfunction. Since failure modes of this type are considered in the selection of components and safety factors so as to meet the specified reliability factors for mission success, there is no requirement to penalize the start-tank dome weight by unnecessarily high pressure loads. The 69×10^3 N/m^2 (10 psid) is thus considered to be conservative.

Table 10

LH_2 START-TANK DOME WEIGHT

Item	Weight (kg)	Weight (lb)
Spherical Isogrid Dome (10 psid)	5.2	(11.5)
Welded Ring (10 psid)	4.0	(8.8)
Total	9.2	(20.3)
Spherical Isogrid Dome (30 psid)	10.5	(23)
Weld Ring (30 psid)	20.0	(44)
Total	30.5	(67)

Accessibility provisions have also been under study. Using a conservative manhole weight penalty of 23 kg/m of diameter, the weight of 0.81 m (32 in.) diameter manhole would be 18 kg. This would provide good access to the start-tank interior. Integration of the tank manhole flange buildup and the beef-up zone on the main-tank bottom is probably not advisable since undesirable seal loadings may be encountered. If the complete start-tank were to be removed, the manhole would have to be placed above the dome intersection at a diameter of about 2.13 m (7 ft). This would weigh approximately 49 kg, which is 31 kg heavier than the smaller access hatch in the main-tank bottom. Thus, the common dome will be welded into the main tank and a tank-bottom manhole will be used for start-tank access.

The weight of the screen tubes has been determined, assuming that each is composed of a support screen, (e.g., 10 x 10 mesh, 0.028 inch wire) fittings (as shown in Figure 5) and the appropriate mesh screen (200 x 1,400) for primary screen tubes and 165 x 800 for top and secondary trap region screen tubes) a pleating factor of 3.0 is used. The flat screen element is formed from pleated 200 x 1,400 stainless steel screen 0.782 kg/m^2 (0.16 lb/ft^2) with a bolted flange-type fitting 2.54 cm wide and 0.25 cm thick. The top and bottom plates are ribbed aluminum with nominal thickness of 0.0635 cm (0.025 in.). The flat screen is pleated with an area factor of 3.0.

These weights are summarized in Table 11. The LH_2 start-tank component weights are based on the item included in Figure 5, and are summarized in Table 12.

Table 11
START-TANK ACQUISITION SYSTEM WEIGHTS (LIQUID HYDROGEN)

Item	Weight (kg)	Weight (lb)
Pressure Shell (Common Dome, Spherical Isogrid, Crushing ΔP = 10 psid)	9.2	(20.3)
Access Manhole Penalty (32-in. ID, Plus Bolts)	18.0	(40.0)
Top Region Stainless Steel Screen Tubes (8)	2.2	(4.7)
Primary Region Stainless Steel Screen Tubes (8)	1.6	(3.5)
Secondary Region Stainless Steel Screen Tubes (4)	0.6	(1.3)
Stainless Steel Flat Screen	2.6	(5.8)
Aluminum Top Plate (Nominal 0.025-in. Thick)	3.7	(8.0)
Aluminum Bottom Plate (Nominal 0.025-in. Thick)	1.2	(2.8)
	39.1	(86.4)

3.2 PHASE II – TASK B – BENCH TESTING

A summary of the pressure decay induced screen breakdown tests involving the "milk carton" screen device was given in the 5th quarterly report. Since that time, the motion picture film has been processed and reviewed. Lighting was sufficient to observe failure of the screen device in all cases. However, all but the last test showed screen failure immediately upon raising the screen device into the ullage region, even for conditions with the ullage only 15°R warmer than the liquid. In the last test, the ullage was

Table 12

ASPM LH_2 START-TANK SYSTEM COMPONENT WEIGHTS*

Quantity	Component	Weight (kg)	Weight (lb)
1	Refill Valve (2-in. Ball)	6.38	(14.0)
1	Main Feed Line Valve (2-in. Ball)	6.38	(14.0)
1	Vent Valve (1-in.)	0.91	(2.0)
2	Viscojets	0.11	(0.25)
1	Regulator (Helium Bottle)	2.25	(5.0)
4	Solenoid Valves (1/2-in.)	1.46	(3.2)
1	Quick Disconnect 1/2-in. Helium Bottle Fill)	0.45	(1.0)
1	High Pressure Relief Valve (Helium Bottle)	0.91	(2.0)
1	Low Pressure Relief Valve	0.22	(0.5)
1	Pressure Controller (Split With LO_2 Tank)	2.25	(5.0)
1	Pressurization Diffusers	0.45	(1.0)
1	Capacitance Probe	0.91	(2.0)
1	Pressure Sensor	0.32	(0.7)
2	Temperature Sensor	0.22	(0.5)
1	Marman Flange	1.36	(3.0)
	Total	24.18	(54.15)

* Principal components associated with main-tank pressurization, expulsion, and fill and drain are not included.

approximately 12° R warmer than the liquid, and no failure was observed. Although more tests of this type would be of interest, the available results clearly indicate that even a quiescent GH_2 ullage having a temperature of 52° R or warmer can dry out an LH_2 screen device and cause loss of retention. The implications of these tests appear to impose a serious constraint on the use of fine mesh screens with an autogenous GH_2 pressurization system. Since the screen device was purposely designed with virtually no solid surfaces exposed to the warm ullage, potential boiling in the screen device from this solid surface was alleviated. Still, rapid screen dry out was observed. At this point, it must be concluded that screen devices cannot be expected to retain LH_2 if exposed to a warm GH_2 ullage while under destabilizing accelerations.

3.3 PHASE II – TASK C – PROTOTYPE DESIGN

Philosophically, the test prototype should provide the capabilities to investigate all critical design, operational, and fabrication aspects of the recommended acquisition system concepts for both the Space Shuttle APS and ASPM applications. Although some compromises may be required because of overall test operations, available hardware, and cost constraints, the above statement should clearly be the design goal. At present, it would appear that two separate devices will be required: (1) a screen ring channel installed within the 105-in. tank representing the Shuttle APS application, and (2) a smaller screen device probably located within a small pressure vessel within the 105-in. tank representing the ASPM application. Ideally, these should be compatible with simultaneous installation within the test tank.

The 105-in. tank has a volume very close to that of the Shuttle start tank (450 ft^3). Thus, it would appear that a circular all-screen ring channel, incorporating all major system design features, could be installed into this tank to represent the Shuttle APS system. This installation would represent the same kind of practical installation problem posed by the actual system. Limiting flow tests could be run, even at 1-g, to establish fluid performance capability.

Figure 8 shows the flight vehicle start-tank acquisition channel design for the Shuttle APS application. This has a 17.8 cm (7 in.) diameter duct channel using 165 x 800 mesh screen with installation details essentially as illustrated in Figure 7 of Reference 6. This uses a minimum of 8 joint/couplings which divides the rings into segments that can be reasonably handled by two mechanics working in the tank.

Figure 9 shows the general layout of the prototype system representing the Space Shuttle APS case. This is about full scale for the Shuttle APS start-tank

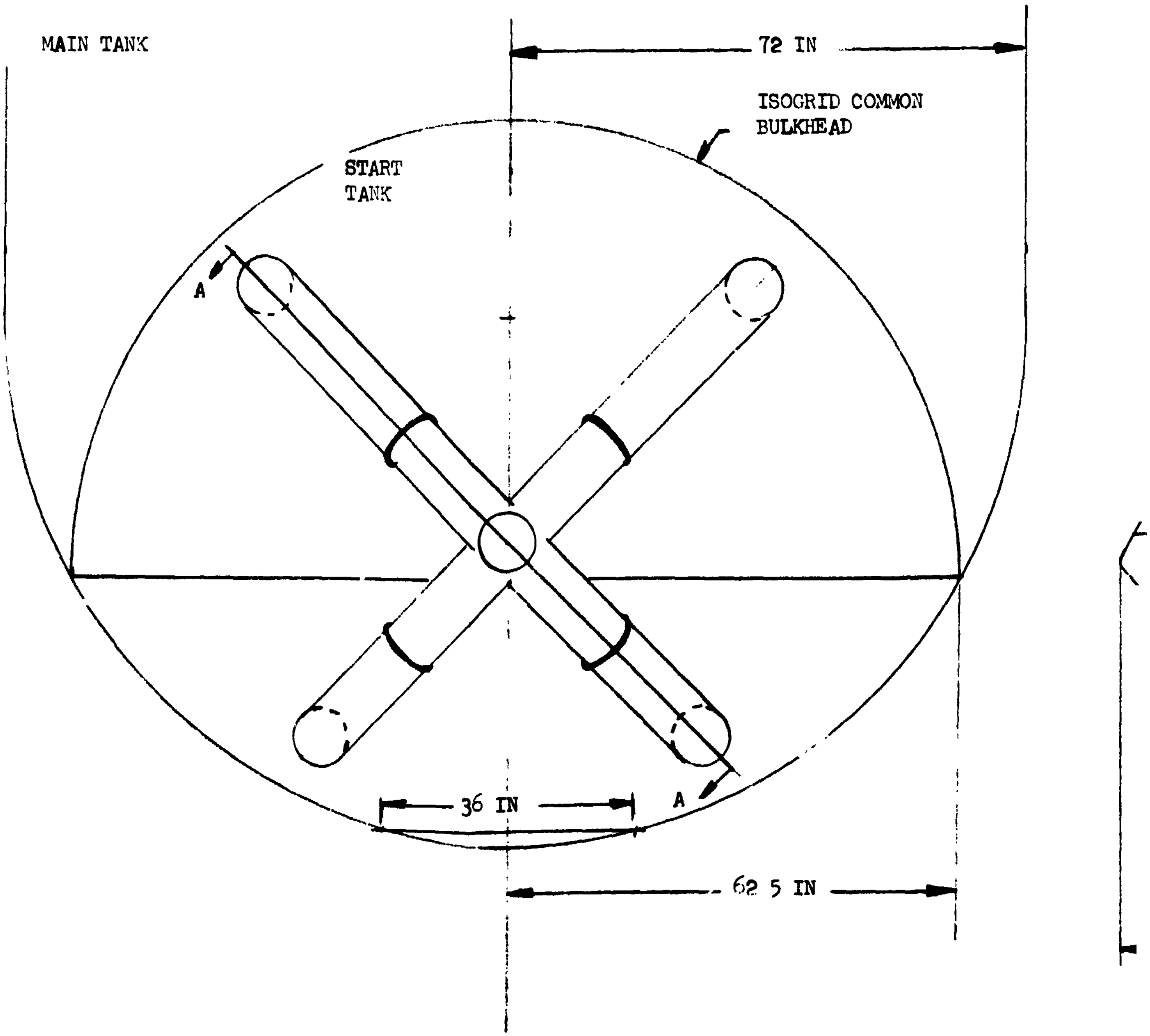

Figure 8. Shuttle APS Flight Vehicle Acquisition Device Design

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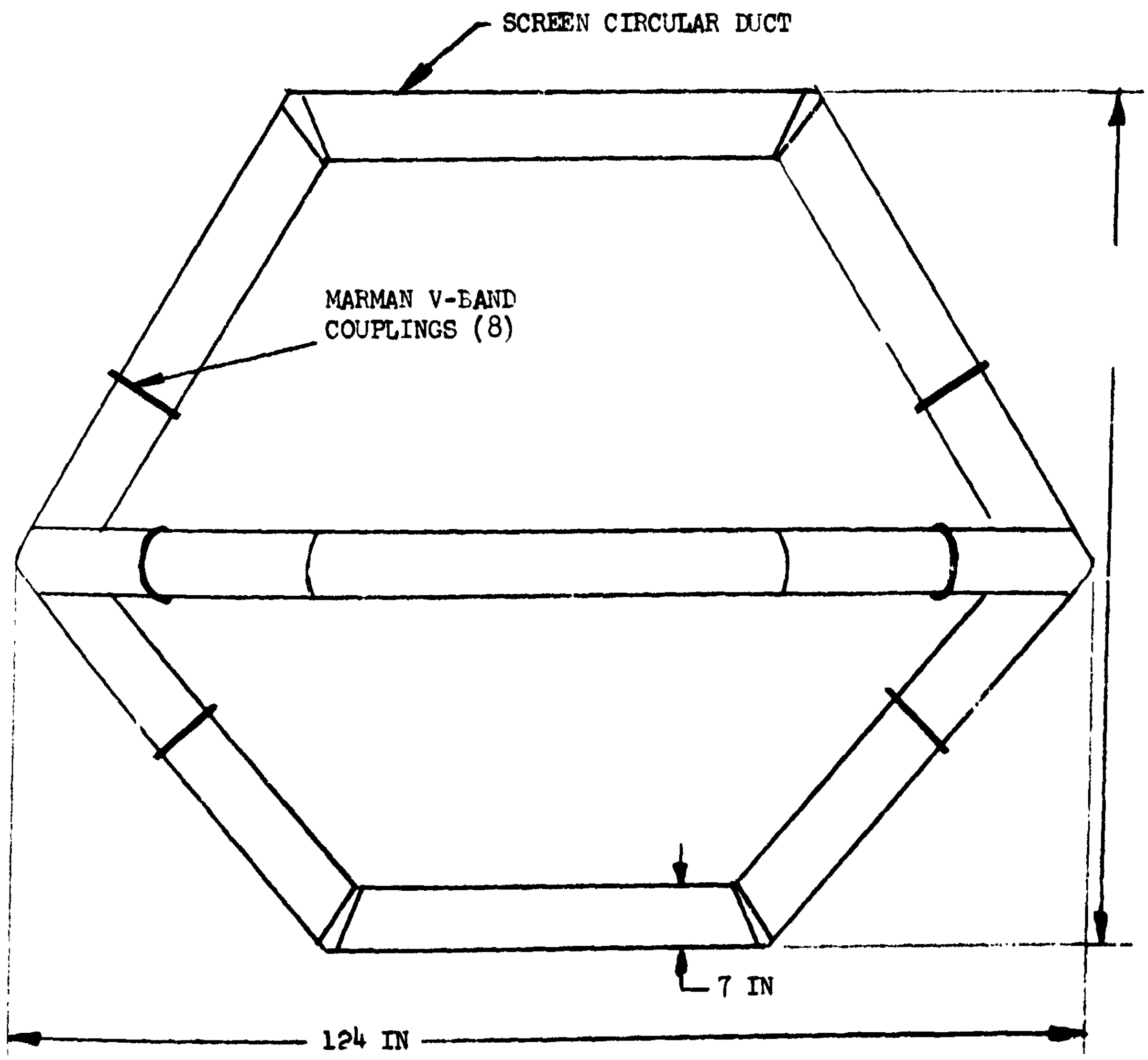

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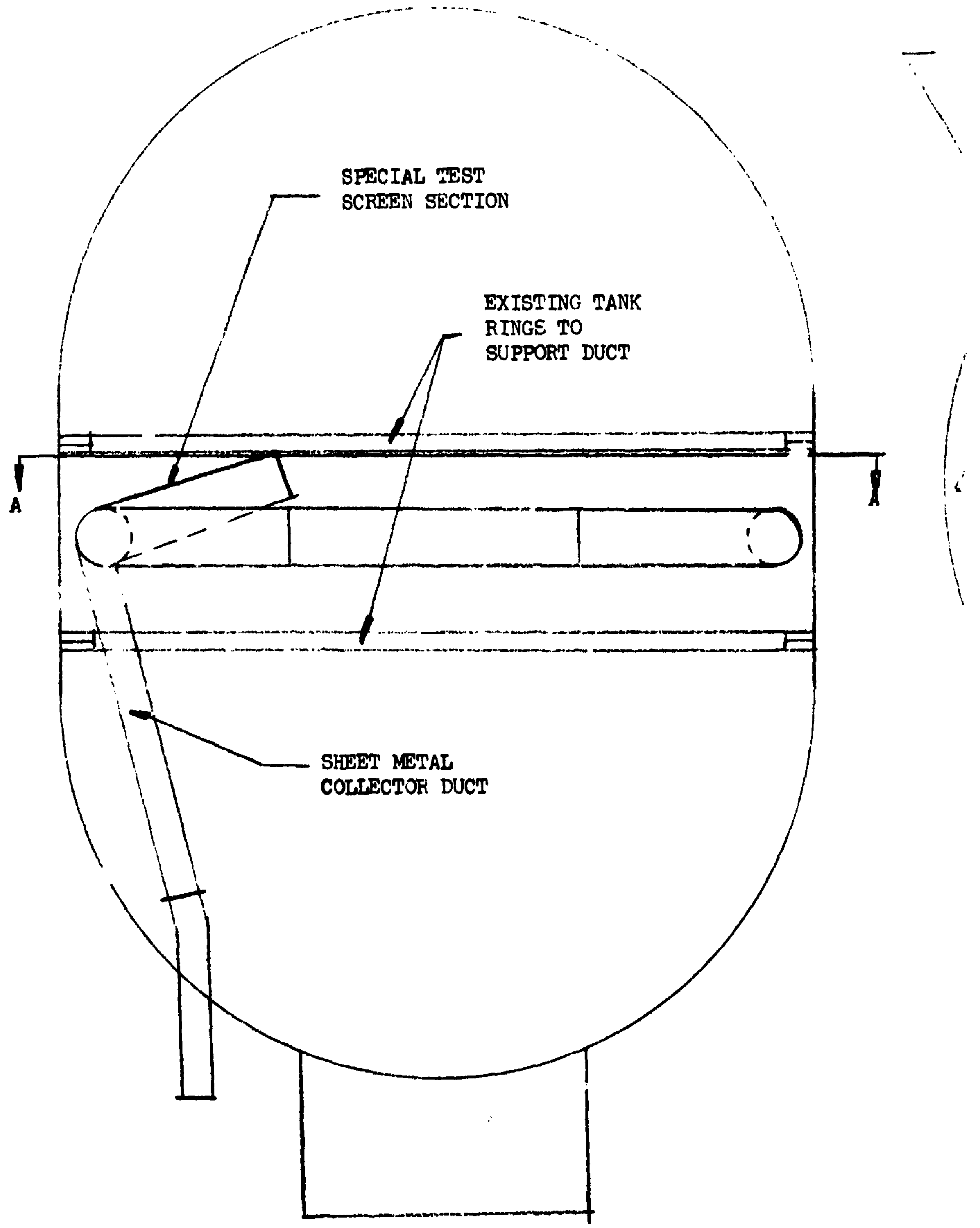
FOLDOUT FRAME 2


SPECIAL TEST
SCREEN SECTION
EXISTING TANK
RINGS TO
SUPPORT DUCT
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COLLECTOR DUCT

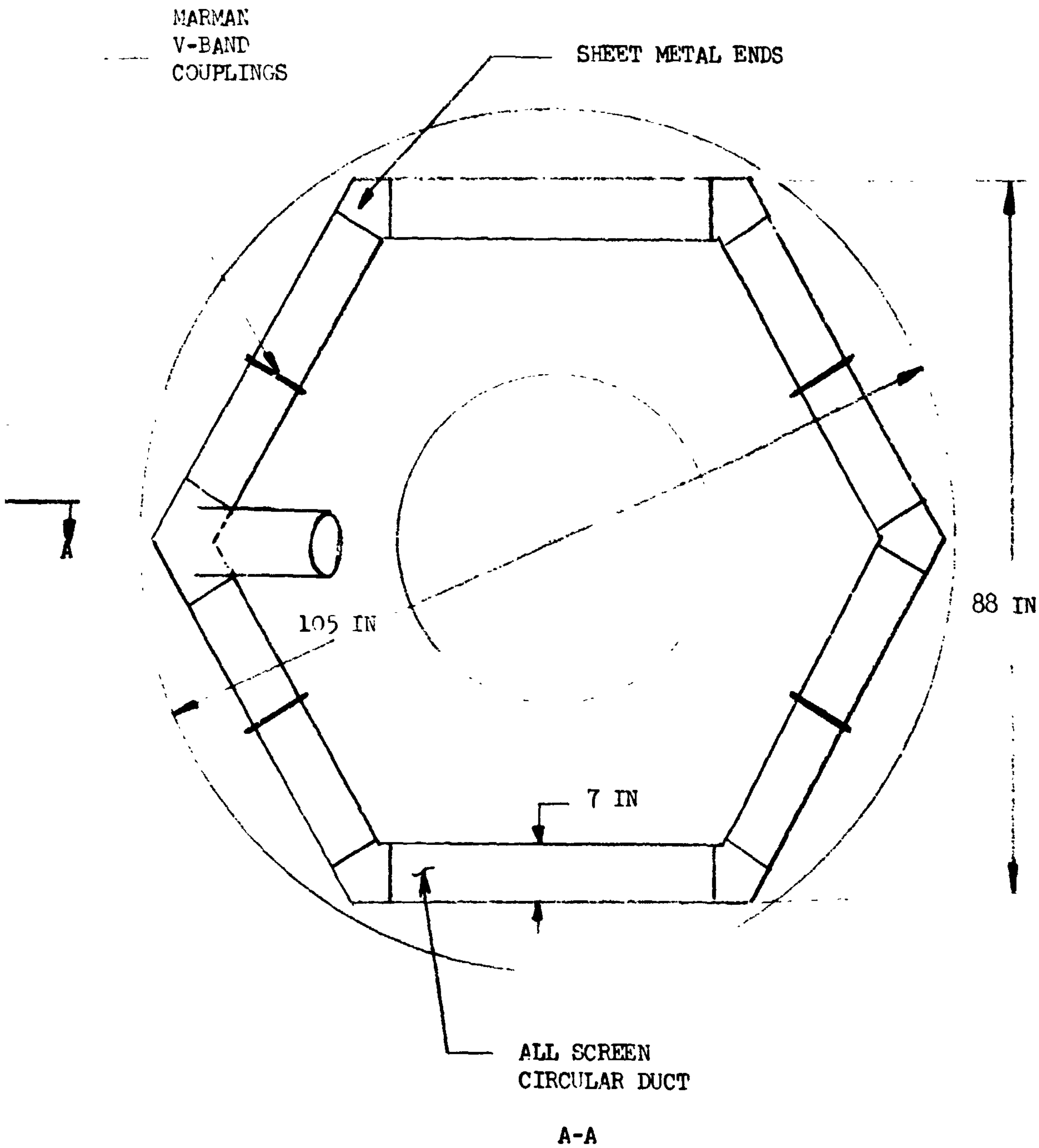

Figure 9. Conceptual Prototype Design – Shuttle APS Application

and the acquisition device itself, including one of the two required channel-rings. The ring would be designed, fabricated, and installed in a manner very similar to that required for the flight vehicle. A sloping element has been added to permit the experimental assessment of head capability for the basic device. The channel will also use the flight vehicle screen mesh size and flight weight hardware wherever possible.

The number and general locations of the joints is the same in the flight vehicle and prototype designs which will thus pose a similar installation and hookup problem. In the prototype, the ring device will be supported by some form of mechanical devices from the existing internal rings in the 105-in. tank. Detailed designs are currently being prepared for this system. Because of the long lead time for joint/coupling hardware, these were selected and ordered in late December. These include fema flanges (Aeroquip Part 4560-700-5), O-Ring flanges (Aeroquip Part 4570-700-5), couplings (Aeroquip Part 4583-700) and a variety of seal elements.

The start-tank for the ASPM application is much smaller (about 10 percent of that for the Shuttle APS) and is made up of more distinct individual elements. It now appears that a portion of the IDU hardware being developed by MDAC under contract NAS 8-27571 could be used (with no physical modifications of parts) to form an ASPM prototype incorporating most of the important design features. This would appear to permit simultaneous incorporation of acquisition hardware for both applications into the 105-in. LH_2 tank, thus leading to a highly cost-effective program.

Figure 10 illustrates a modification to the IDU to represent the ASPM start-tank system as shown in Figure 5. This would result in a device about 1/2 the volume of that required for the actual ASPM vehicle. In reality, the only part of the IDU that would be used is the bottom plate and all valves. A completely new cover and all internal parts would be built, thus leaving the original IDU hardware intact. The device shown in Figure 10 will use the same screen mesh sizes and approximately the same screen element overall dimensions as in the flight vehicle (see Figure 5). The general flow schematic is also identical for the prototype and flight vehicle.

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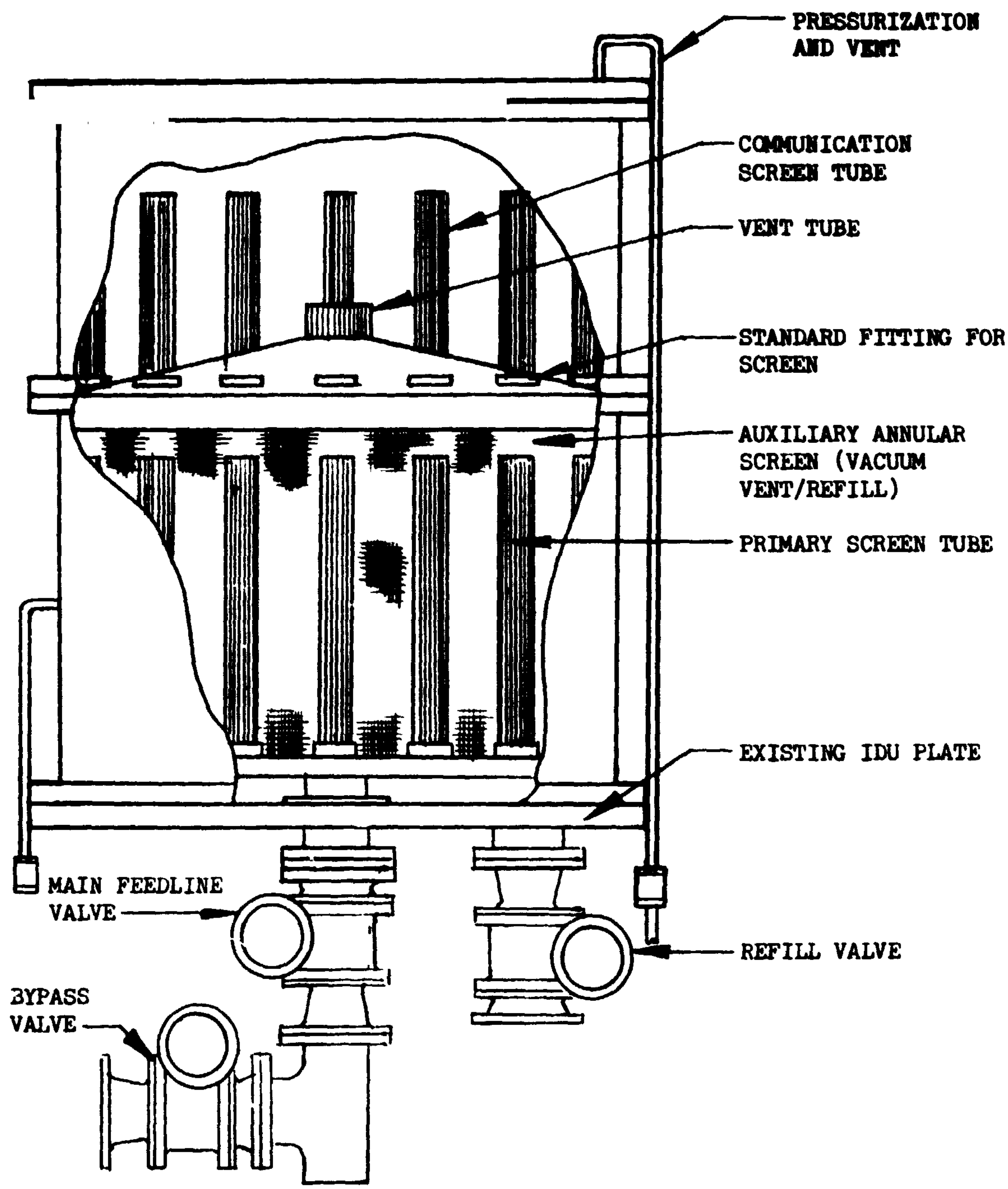

Figure 10. Modified LH_2 Interface Demonstration Unit (IDU) for ASPM Simulation

These general concepts were reviewed with the NASA project officers and it was agreed to proceed along this line. Detail design studies are now underway.

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